National Aeronautics and Space Administration

NASA's Exploration Systems Architecture Study

Final Report

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Preface

The National Aeronautics and Space Administration's (NASA's) Exploration Systems Architecture Study (ESAS) Final Report documents the analyses and findings of the 90-day Agencywide study. Work on this study began in May 2005 and was completed in July 2005. The purpose of the study was to:

- Assess the top-level Crew Exploration Vehicle (CEV) requirements and plans that will enable the CEV to provide crew transport to the International Space Station (ISS) and will accelerate the development of the CEV and crew launch system to reduce the gap between Shuttle retirement and CEV Initial Operational Capability (IOC);
- Define the top-level requirements and configurations for crew and cargo launch systems to support the lunar and Mars exploration programs;
- Develop a reference exploration architecture concept to support sustained human and robotic lunar exploration operations; and
- Identify key technologies required to enable and significantly enhance these reference exploration systems and a reprioritization of near-term and far-term technology investments.

The ESAS Final Report presents analysis and recommendations concerning technologies and potential approaches related to NASA's implementation of the Vision for Space Exploration. Project and contract requirements will likely be derived, in part, from the ESAS analysis and recommendations. However, the analysis and recommendations contained herein do not represent a set of project or contract requirements and are not binding on the U.S. Government unless and until they are formally and expressly adopted as such.

Details of any recommendations offered by the ESAS Final Report will be translated into implementation requirements. Moreover, the report represents the assessments and projections of the report's authors at the time it was prepared. It is anticipated that the concepts in this report will be analyzed further and refined. By the time some of the activities addressed in this report are implemented, certain assumptions on which the report's conclusions are based will likely evolve based on this new analysis. Accordingly, NASA, and any entity under contract with NASA, should not use the information in this report as final project direction.

The ESAS Final Report is separated into two segments. The first segment, which is the main body of the report, includes the Executive Summary. This segment is intended for public distribution.

The second segment is a collection of appendices. Access to the appendices is restricted due to the sensitive nature of the data they contain.

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1. Executive Summary

1.1 Introduction

1.1.1 Background

In January 2004, President George W. Bush announced a new Vision for Space Exploration for the National Aeronautics and Space Administration (NASA) that would return humans to the Moon by 2020 in preparation for human exploration of Mars. As part of this vision, NASA would retire the Space Shuttle in 2010 and build and fly a new Crew Exploration Vehicle (CEV) no later than 2014. Initially, since no plans were made for this CEV to service the International Space Station (ISS), international partner assets would be required to ferry U.S. crew and cargo to the ISS after 2010—creating a significant gap in domestic space access for U.S. astronauts. NASA gradually reorganized to better implement the President's vision and established the Exploration Systems Mission Directorate (ESMD) to lead the development of a new exploration "system-of-systems" to accomplish these tasks. Over the course of the next year, ESMD defined preliminary requirements and funded system-of-system definition studies by Government and industry. More than $1 billion in technology tasks were immediately funded in a wide variety of areas. Plans were established to spend more than $2 billion per year in exploration systems, human, and nuclear-related technologies. Plans were established to fund two CEV contractors through Preliminary Design Review (PDR) and first flight of a subscale test demonstration in 2008, after which selection of a final CEV contractor would be made. In March 2004, a CEV Request for Proposals (RFP) was released to industry despite the lack of a firm set of requirements or a preferred architecture approach for returning humans to the Moon. A wide variety of architecture options was still under consideration at that time—with none considered feasible within established budgets. Preferred architecture options relied on as many as nine launches for a single lunar mission and on modified versions of the United States Air Force (USAF) Evolved Expendable Launch Vehicles (EELVs) for launch of crew and cargo.

Dr. Michael Griffin was named the new NASA Administrator in April 2005. With concurrence from Congress, he immediately set out to restructure NASA's Exploration Program by making its priority to accelerate the development of the CEV to reduce or eliminate the planned gap in U.S. human access to space. He established a goal for the CEV to begin operation in 2011 and to be capable of ferrying crew and cargo to and from the ISS. To make room for these priorities in the budget, Dr. Griffin decided to downselect to a single CEV contractor as quickly as possible and cancel the planned 2008 subscale test demonstration. He also decided to significantly reduce the planned technology expenditures and focus on existing technology and proven approaches for exploration systems development. In order to reduce the number of required launches and ease the transition after Space Shuttle retirement in 2010, Dr. Griffin also directed the Agency to carefully examine the cost and benefits of developing a Shuttle-derived Heavy-Lift Launch Vehicle (HLLV) to be used in lunar and Mars exploration. To determine the best exploration architecture and strategy to implement these many changes, the Exploration Systems Architecture Study (ESAS) team was established at NASA Headquarters (HQ) as discussed in **Section 1.1.2**, **Charter**, and **Section 1.1.3**, **Approach**.

1.1.2 Charter

The ESAS began on May 2, 2005, at the request of the NASA Administrator. The study was commissioned in a letter dated April 29, 2005, which is provided in **Appendix 2A**, **Charter for the Exploration Systems Architecture Study (ESAS)**, from the NASA Administrator to all NASA Center Directors and Associate Administrators. The study was initiated to perform four specific tasks by July 29, 2005, as outlined in the letter and identified below.

- Complete assessment of the top-level CEV requirements and plans to enable the CEV to provide crew transport to the ISS and to accelerate the development of the CEV and crew launch system to reduce the gap between Shuttle retirement and CEV Initial Operational Capability (IOC).
- Provide definition of top-level requirements and configurations for crew and cargo launch systems to support the lunar and Mars exploration programs.
- Develop a reference lunar exploration architecture concept to support sustained human and robotic lunar exploration operations.
- Identify key technologies required to enable and significantly enhance these reference exploration systems and reprioritize near-term and far-term technology investments.

More than 20 core team members were collocated at NASA HQ for the 3-month duration. Over the course of the ESAS effort, hundreds of employees from NASA HQ and the field centers were involved in design, analysis, planning, and costing activities.

1.1.3 Approach

The ESAS effort was organized around each of the four major points of the charter: CEV definition, Launch Vehicle (LV) definition, lunar architecture definition, and technology plan definition. Additional key analysis support areas included cost, requirements, ground operations, mission operations, human systems, reliability, and safety.

The ESAS team took on the task of developing new CEV requirements and a preferred configuration to meet those requirements. The CEV requirements developed by the ESAS team are contained in **Appendix 2B**, **ESAS CEV Requirements**. A wide variety of trade studies was addressed by the team. Different CEV shapes were examined, including blunt-body, slender-body, and lifting shapes. The required amount of habitable volume and number of crew were determined for each mission based on a crew task analysis. Economic-based trades were performed to examine the benefits of reusability and system commonality. The effects of a CEV mission to the ISS were examined in detail, including docking and berthing approaches and the use of the CEV as a cargo transport and return vehicle. The requirements for Extra-Vehicular Activity (EVA) were examined, and different airlock approaches were investigated. Additional trades included: landing mode, propellant type, number of engines, level of engine-out capability, and abort approaches. A phased development approach was defined that uses block upgrades of the CEV system for ISS crew, ISS cargo, lunar, and Mars missions with the same shape and size system.

The ESAS team examined hundreds of different combinations of launch elements to perform the various Design Reference Missions (DRMs). Different sizes of LVs and numbers of launches required to meet the DRMs were traded. The team's major trade study was a detailed examination of the costs, schedule, reliability, safety, and risk of using EELV- and Shuttle-derived launchers for crew and cargo missions. Other trade studies included: stage propellant type, numbers of engines per stage, level of stage commonality, and number of stages.

The ESAS team was tasked to develop new architecture-level requirements and an overall architecture approach to meet those requirements. The architecture requirements developed by the ESAS team are contained in **Appendix 2C**, **ESAS Architecture Requirements**. An initial reference architecture was established and configuration control was maintained by the team. Trade studies were then conducted from this initial baseline. In order to determine the crew and cargo transportation requirements, the team examined and traded a number of different lunar surface missions and systems and different approaches to constructing a lunar outpost. A team of nationally recognized lunar science experts was consulted to determine science content and preferred locations for sortie and outpost missions. The use of in-situ resources for propellant and power was examined, and nuclear and solar power sources were traded. The major trade study conducted by the team was an examination of various mission modes for transporting crew and cargo to the Moon, including: Lunar Orbit Rendezvous (LOR), Earth Orbit Rendezvous (EOR), and direct return from the lunar surface. The number and type of elements required to perform the Trans-Lunar Injection (TLI), Lunar-Orbit Insertion (LOI), and Trans-Earth Injection (TEI) burns associated with these missions were also traded. In addition, a number of different configurations were examined for the lunar lander, or Lunar Surface Access Module (LSAM). Trade studies for the LSAM included: number of stages, stage propellant and engine type, level of engine-out capability, airlock approaches, cargo capacity, and abort options.

The ESAS team was also tasked to determine the architecture technology requirements and to reprioritize existing technology plans to provide mature technologies prior to the PDR of each major element. The team used a disciplined, proven process to prioritize technology investments against architecture-level Figures of Merit (FOMs) for each mission. New technology investments were recommended only when required to enable a particular system, and investments were planned to begin only as required based on the need date.

The various trade studies conducted by the ESAS team used a common set of FOMs for evaluation. Each option was quantitatively or qualitatively assessed against the FOMs shown in **Figure 1-1**. FOMs are included in the areas of: safety and mission success, effectiveness and performance, extensibility and flexibility, programmatic risk, and affordability. FOMs were selected to be as mutually independent and measurable as possible. Definitions of each of these FOMs are provided in **Appendix 2D**, **ESAS FOM Definitions**, together with a list of measurable proxy variables and drivers used to evaluate the impacts of trade study options against the individual FOMs.

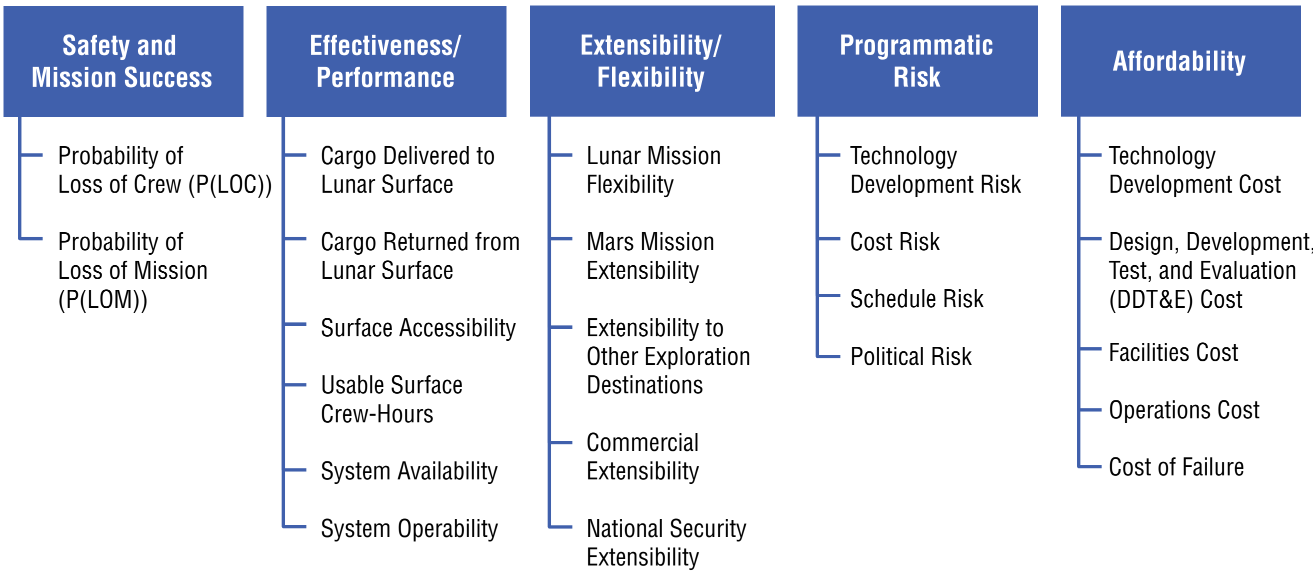


Figure 1-1. ESAS FOMs

1.1.4 Design Reference Missions

A series of DRMs was established to facilitate the derivation of requirements and the allocation of functionality between the major architecture elements. Three of the DRMs were for ISS-related missions: transportation of crew to and from the ISS, transportation of pressurized cargo to and from the ISS, and transportation of unpressurized cargo to the ISS. Three of the DRMs were for lunar missions: transportation of crew and cargo to and from anywhere on the lunar surface in support of 7-day "sortie" missions, transportation of crew and cargo to and from an outpost at the lunar south pole, and one-way transportation of cargo to anywhere on the lunar surface. A DRM was also established for transporting crew and cargo to and from the surface of Mars for a 18-month stay.

1.1.4.1 DRM Description: Crew Transport To and From ISS

The primary purpose of this mission is to transport three ISS crew members, and up to three additional temporary crew members, to the ISS for a 6-month stay and return them safely to Earth at any time during the mission. The architecture elements that satisfy the mission consist of a CEV and a Crew Launch Vehicle (CLV). **Figure 1-2** illustrates the mission. The CEV, consisting of a Crew Module (CM) and a Service Module (SM), is launched by the CLV into a 56- x 296-km insertion orbit at 51.6-deg inclination with a crew of three to six destined for a 6-month ISS expedition. The CEV performs orbit-raising burns per a pre-mission-defined rendezvous phasing profile to close on the ISS. These burns will be a combination of ground-targeted and onboard-targeted burns, the latter performed once rendezvous navigation sensors acquire the ISS. The CEV crew conducts a standard approach to the ISS, docking to one of two available CEV-compatible docking ports. The CEV crew pressurizes the vestibule between the two docked vehicles and performs a leak check. The ISS crew then equalizes pressure with the CEV vestibule and hatches are opened. Once ingress activities are complete, the CEV is configured to a quiescent state and assumes a "rescue vehicle" role for the duration of the crew increment. Periodic systems health checks and monitoring are performed by Mission Control throughout the increment. Upon completion of up to a 180-day increment on the ISS, the crew stows any return manifest items in the CEV crew cabin, performs a pre-undock health check of all entry critical systems, closes hatches and performs leak

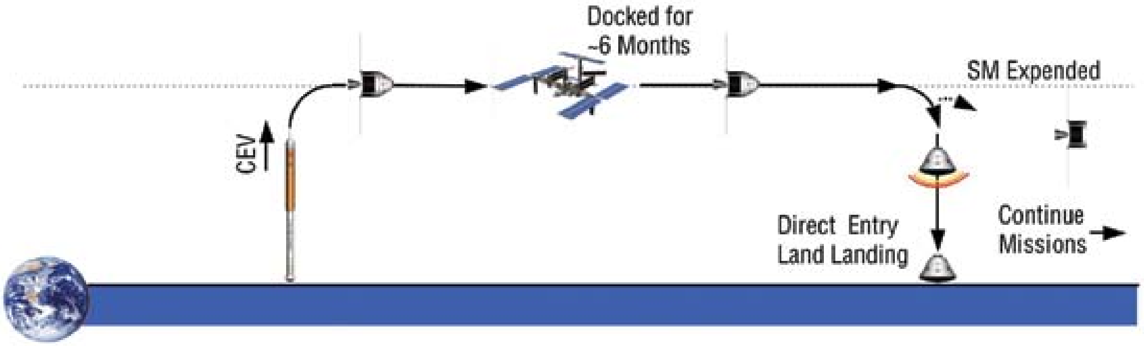


Figure 1-2. Crew Transport to and from ISS DRM

checks, and undocks from the station. The CEV departs the vicinity of the ISS and conducts an onboard-targeted (ground-validated) deorbit burn. After burn completion, the CEV SM is discarded, and the return component is maneuvered to the proper entry interface attitude for a guided entry to the landing site. The CEV performs a nominal landing at the primary land-based landing site.

1.1.4.2 DRM Description: Unpressurized Cargo Transport to ISS

The primary purpose of this mission is to transport unpressurized cargo to the ISS and de-orbit to perform a destructive reentry after 30 days at the ISS. The architecture elements that satisfy this mission consist of a Cargo Delivery Vehicle (CDV) and a CLV. **Figure 1-3** illustrates the mission. The CDV is launched by the CLV into a 56- x 296-km insertion orbit at 51.6-deg inclination with an unpressurized carrier in place of the CEV CM loaded with up to 6,000 kg gross mass of external ISS logistics. The CDV performs orbit-raising burns per a pre-mission-defined rendezvous phasing profile to close on the ISS. These burns will be a combination of ground-targeted and onboard-targeted burns, the latter to be performed once rendezvous navigation sensors acquire the ISS. The CDV performs a standard approach to a safe stationkeeping point in the vicinity of the ISS. Upon validation of readiness to proceed by Mission Control, the CDV is commanded to proceed with approach and conducts a standard onboard-guided approach to the ISS, achieving a stationkeeping point within reach of the Space Station Remote Manipulator System (SSRMS). The ISS crew grapples the CDV and berths it to the Node 2 nadir Common Berthing Mechanism (CBM) port. Once berthing activities are complete, the CDV systems are configured to a quiescent state. The ISS crew performs logistics transfer and systems maintenance EVAs to offload the CDV unpressurized pallet of new Orbital Replacement Units (ORUs) and to load old ORUs for disposal. Periodic systems health checks and monitoring are performed by Mission Control throughout the increment. Upon completion of up to a 30-day mated phase on the ISS, Mission Control performs a pre-undock health check of all entry critical systems. Then, the ISS crew grapples the CDV, unberths it from the CBM, and maneuvers it to its departure point and releases it. The CDV departs the vicinity of the ISS and conducts an onboard-targeted (ground-validated) deorbit burn for disposal.

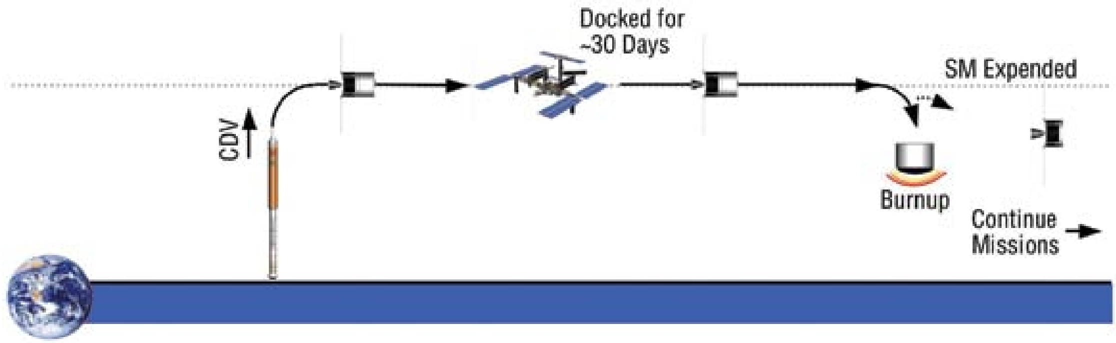


Figure 1-3. Unpressurized Cargo Transport to ISS DRM

1.1.4.3 DRM Description: Pressurized Cargo Transport To and From ISS

The primary purpose of this mission is to transport pressurized cargo to the ISS and deorbit to perform a reentry and safe return of pressurized cargo to Earth after 90 days at the ISS. **Figure 1-4** illustrates the mission. The architecture elements that satisfy this mission consist of a cargo version of the CEV and a CLV. A cargo version of the CEV is launched by the CLV into a 56- x 296-km insertion orbit at 51.6-deg inclination with the pressurized module filled with up to 3,500 kg gross mass of pressurized logistics for delivery to the ISS. The CEV performs orbit-raising burns per a pre-mission-defined rendezvous phasing profile to close on the ISS. These burns will be a combination of ground-targeted and onboard-targeted burns, the latter performed once rendezvous navigation sensors acquire the ISS. The uncrewed CEV performs a standard approach to a safe stationkeeping point in the vicinity of the ISS. Upon validation of readiness to proceed by Mission Control, the CEV is commanded to proceed with approach and conducts a standard onboard-guided approach to the ISS, docking to one of two available CEV-compatible docking ports. Mission Control pressurizes the vestibule between the two docked vehicles and performs a leak check. The ISS crew then equalizes with the CEV and hatches are opened. Once ingress activities are complete, the CEV systems are configured to a quiescent state, and the CEV cargo is offloaded. Periodic systems health checks and monitoring are performed by Mission Control throughout the increment. Upon completion of up to a 90-day docked phase on the ISS, the crew stows any return manifest items in the CEV pressurized cabin, Mission Control performs a pre-undock health check of all entry critical systems, the ISS crew closes hatches and performs leak checks, and Mission Control commands the CEV to undock from the station. The CEV departs the vicinity of the ISS and conducts an onboard-targeted (ground-validated) deorbit burn. After burn completion, unnecessary CEV elements are discarded, and the return element is maneuvered to the proper entry interface attitude for a guided entry to the landing site. The CEV performs a nominal landing at the primary land-based landing site.

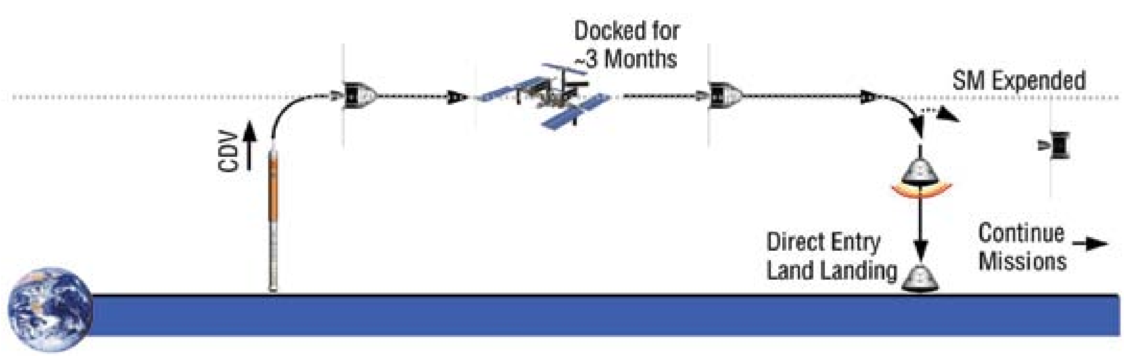


Figure 1-4. Pressurized Cargo Transport to and from ISS DRM

1.1.4.4 DRM Description: Lunar Sortie Crew with Cargo

The architecture provides the capability for up to four crew members to explore any site on the Moon (i.e., global access) for up to 7 days. These missions, referred to as lunar sorties, are analogous to the Apollo surface missions and demonstrate the capability of the architecture to land humans on the Moon, operate for a limited period on the surface, and safely return them to Earth. Sortie missions also allow for exploration of high-interest science sites or scouting of future lunar outpost locations. Such a mission is assumed not to require the aid of pre-positioned lunar surface infrastructure, such as habitats or power stations, to perform the mission. During a sortie, the crew has the capability to perform daily EVAs with all crew members egressing from the vehicle through an airlock. Performing EVAs in pairs with all four crew members on the surface every day maximizes the scientific and operational value of the mission.

Figure 1-5 illustrates the lunar sortie crew and cargo mission. The following architecture elements are required to perform the mission: a CLV, a Cargo Launch Vehicle (CaLV) capable of delivering at least 125 mT to Low Earth Orbit (LEO), a CEV, an LSAM, and an Earth Departure Stage (EDS). The assumed mission mode for the lunar sortie mission is a combination EOR–LOR approach. The LSAM and EDS are predeployed in a single CaLV launch to LEO, and the CLV delivers the CEV and crew in Earth orbit, where the two vehicles initially rendezvous and dock. The EDS performs the TLI burn and is discarded. The LSAM then performs the LOI for both the CEV and LSAM. The entire crew then transfers to the LSAM, undocks from the CEV, and performs a descent to the lunar surface in the LSAM. After up to 7 days on the lunar surface, the LSAM returns the crew to lunar orbit where the LSAM and CEV dock, and the crew transfers back to the CEV. The CEV then returns the crew to Earth with a direct entry and land touchdown, while the LSAM is disposed of via impact on the lunar surface.

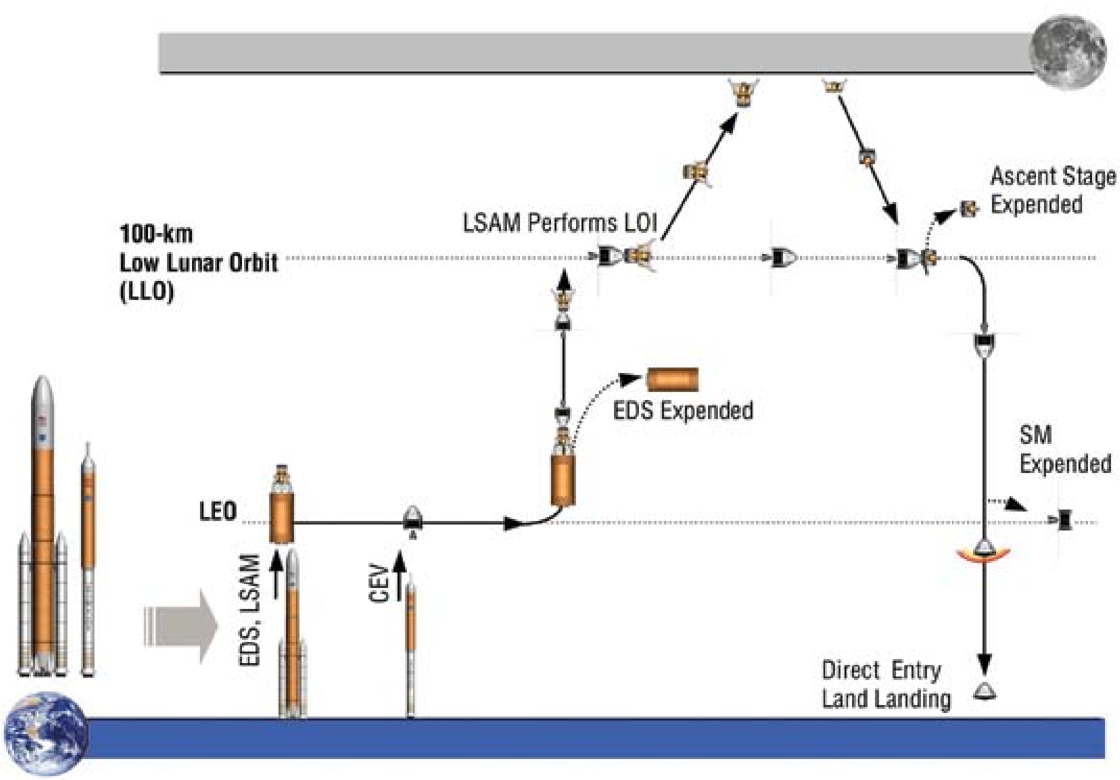


Figure 1-5. Lunar Sortie Crew with Cargo DRM

1.1.4.5 DRM Description: Lunar Outpost Cargo Delivery

The architecture provides the capability to deliver 20 mT of cargo to the lunar surface in a single mission using the elements of the human lunar transportation system. This capability is used to deliver surface infrastructure needed for lunar outpost buildup (habitats, power systems, communications, mobility, In-Situ Resource Utilization (ISRU) pilot plants, etc.), as well as periodic logistics resupply packages to support a continuous human presence.

Figure 1-6 illustrates the lunar outpost cargo delivery mission. The following architecture elements are required to perform the mission: the same CaLV and EDS as the sortie mission and a cargo variant of the LSAM to land the large cargo elements near the lunar outpost site. The cargo variant of the LSAM replaces the habitation module with a cargo pallet and logistics carriers. The LSAM and EDS are launched to LEO on a single CaLV. The EDS performs the TLI burn and is discarded. The LSAM then performs the LOI and a descent to the lunar surface. The cargo is then offloaded from the LSAM autonomously or by the outpost crew.

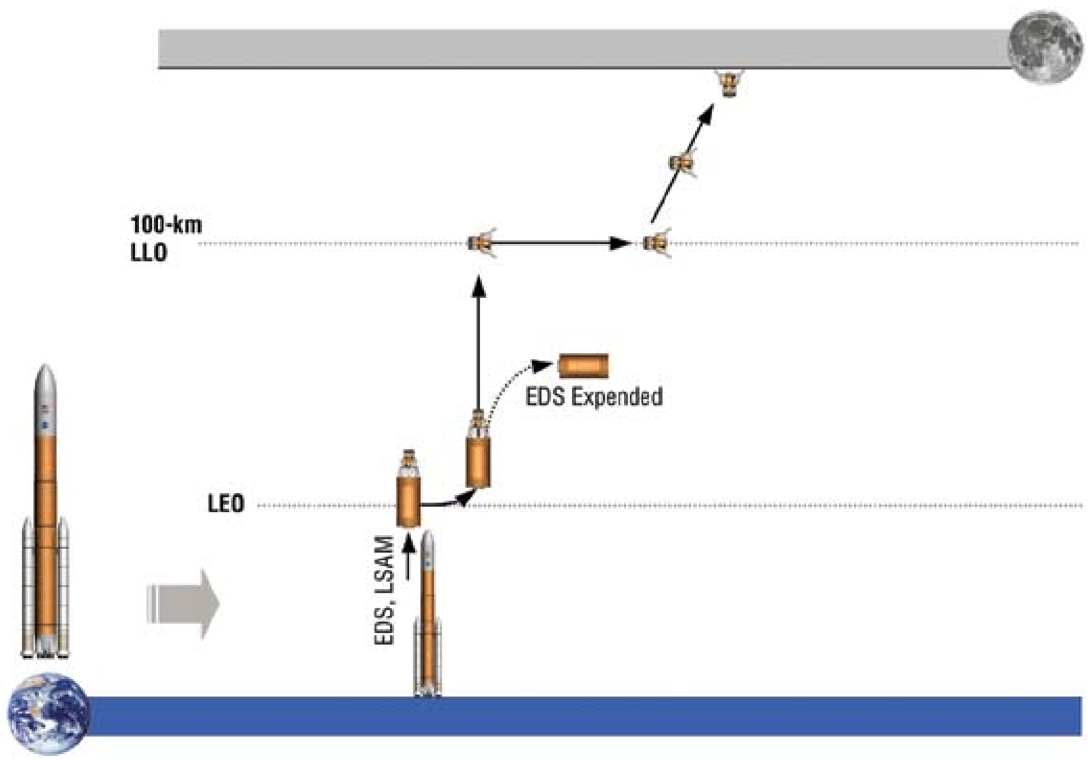


Figure 1-6. Lunar Outpost Cargo Delivery DRM

1.1.4.6 DRM Description: Lunar Outpost Crew with Cargo

A primary objective of the lunar architecture is to establish a continuous human presence on the lunar surface to accomplish exploration and science goals. This capability will be established as quickly as possible following the return of humans to the Moon. To best accomplish science and ISRU goals, the outpost is expected to be located at the lunar south pole. The primary purpose of the mission is to transfer up to four crew members and supplies in a single mission to the outpost site for expeditions lasting up to 6 months. Every 6 months, a new crew will arrive at the outpost, and the crew already stationed there will return to Earth. **Figure 1-7** illustrates this mission.

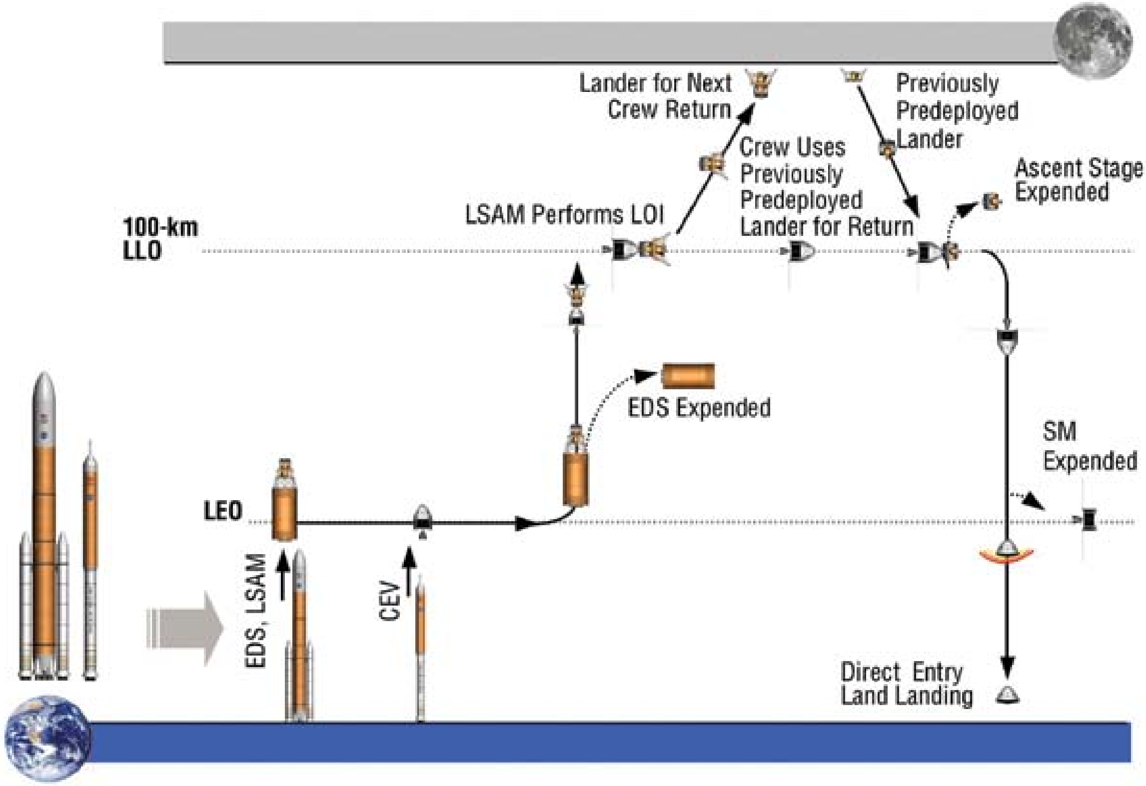


Figure 1-7. Lunar Outpost Crew with Cargo DRM

The entire suite of vehicles developed to support lunar sortie exploration is also required for lunar outpost missions, in addition to a surface habitat, power/communications systems, and other infrastructure elements still to be defined. The following architecture elements are required to perform the mission: a CLV, a CaLV capable of delivering at least 125 mT to LEO, a CEV, an LSAM, and an EDS. The assumed mission mode for the lunar sortie mission is a combination EOR–LOR approach. The LSAM and EDS are predeployed in a single CaLV launch to LEO, and the CLV delivers the CEV and crew in Earth orbit, where the two vehicles initially rendezvous and dock. The EDS performs the TLI burn and is discarded. The LSAM then performs the LOI for both the CEV and LSAM. The entire crew then transfers to the LSAM, undocks from the CEV, and performs a descent to the lunar surface near the outpost in the LSAM. After a surface stay of up to 6 months, the LSAM returns the crew to lunar orbit where the LSAM and CEV dock, and the crew transfers back to the CEV. The CEV then returns the crew to Earth with a direct entry and land touchdown, while the LSAM is disposed of via impact on the lunar surface.

1.1.4.7 DRM Description: Mars Exploration

The Mars Exploration DRM employs conjunction-class missions, often referred to as long-stay missions, to minimize the exposure of the crew to the deep-space radiation and zero-gravity environment while, at the same time, maximizing the scientific return from the mission. This is accomplished by taking advantage of optimum alignment of Earth and Mars for both the outbound and return trajectories by varying the stay time on Mars, rather than forcing the mission through non-optimal trajectories, as in the case of the short-stay missions. This approach allows the crew to transfer to and from Mars on relatively fast trajectories, on the order of 6 months, while allowing them to stay on the surface of Mars for a majority of the mission, on the order of 18 months. **Figure 1-8** provides an overview of the mission approach.

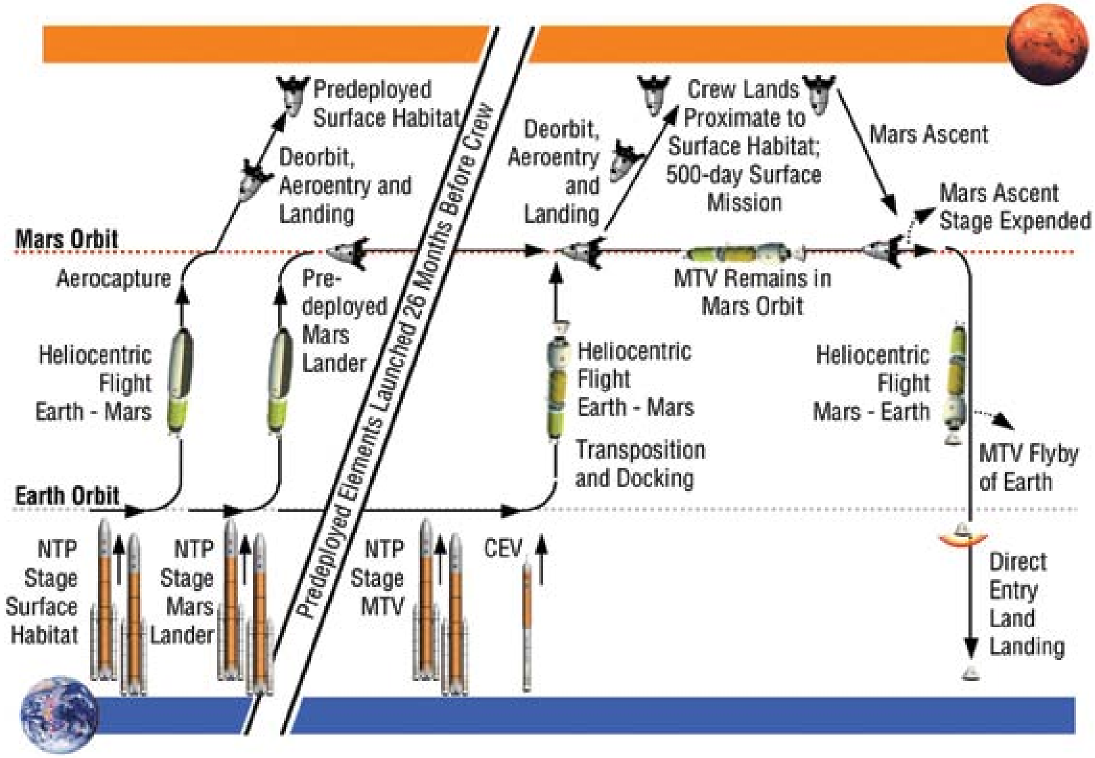


Figure 1-8. Mars Exploration DRM

The surface exploration capability is implemented through a split mission concept in which cargo is transported in manageable units to the surface, or Mars orbit, and checked out in advance of committing the crews to their mission. The split mission approach also allows the crew to be transported on faster, more energetic trajectories, minimizing their exposure to the deep-space environment, while the vast majority of the material sent to Mars is sent on minimum energy trajectories. As can be seen in **Figure 1-8**, each human mission to Mars is comprised of three vehicle sets, two cargo vehicles, and one round-trip piloted vehicle.

The scope of the ESAS was only to address the transportation of the crew to a Mars Transfer Vehicle (MTV) in LEO or reentering from the MTV at the conclusion of the Mars mission, and to provide the design of a CaLV with an LEO cargo capacity of 125 mT.

This DRM utilizes the CEV to transfer a crew of six to and from an MTV as part of a Mars mission architecture. The CEV is launched by the CLV into an orbit matching the inclination of the MTV. The CEV spends up to 2 days performing orbit-raising maneuvers to close on the MTV. The CEV crew conducts a standard approach to the MTV and docks. The CEV crew performs a leak check, equalizes pressure with the MTV, and opens hatches. Once crew and cargo transfer activities are complete, the CEV is configured to a quiescent state. Periodic systems health checks and monitoring are performed by Mission Control throughout the Mars transfer mission.

As the MTV approaches Earth upon completion of the 2.5-year mission, the crew performs a pre-undock health check of all entry critical systems, transfers to the CEV, closes hatches, performs leak checks, and undocks from the MTV. The CEV departs the MTV 24 hours prior to Earth entry and conducts an onboard-targeted (ground-validated) deorbit burn. As entry approaches, the CEV maneuvers to the proper entry interface attitude for a direct guided entry to the landing site. The CEV performs a nominal landing at the primary land-based landing site.

1.2 Ground Rules and Assumptions

At the beginning of the ESAS, a number of Ground Rules and Assumptions (GR&As) were established based on management guidance, internal and external constraints, design practices, and existing requirements.

1.2.1 Safety and Mission Assurance (S&MA) GR&As

The S&MA GR&As are listed below.

- NASA Procedural Requirements (NPR) 8705.2, Human-Rating Requirements for Space Systems, will be used as a guideline for all architecture design activities. Required deviations from NPR 8705.2 will be noted in the applicable requirements documentation.
- Abort opportunities will be provided throughout all mission phases to the maximum extent possible.
- In the event of an abort from the lunar surface, return of crew to the Earth's surface will take no longer than 5 days—independent of orbital alignment.

1.2.2 Operations GR&As

The Operations GR&As are listed below.

- The CEV will deliver crew to and from the ISS through ISS end-of-life in 2016.
- The CEV will deliver and return cargo to the ISS through ISS end-of-life in 2016.
- The architecture will separate crew and large cargo to the maximum extent practical.
- The architecture will support ISS up/down mass needs and other ISS requirements, as required, after Shuttle retirement.
- CEV operations will be performed at the Kennedy Space Center (KSC) through clearing of the launch pad structure.
- On-orbit flight operations and in-flight operations for crewed missions will be performed at NASA JSC.
- Crew and cargo recovery operations from the crew and cargo launches will be managed by KSC with assistance from other NASA and non-NASA personnel and assets as required.
- Architectures will enable extensibility of lunar mission systems to human Mars exploration missions.
- The study will utilize the Mars DRM known as DRM 3.0, "Reference Mission Version 3.0 Addendum to the Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team EX13-98-036, June 1998."
- The architecture will support lunar global access.
- The architecture will support a permanent human presence on the Moon.
- In-space EVA assembly will not be required.
- In-space EVA will only be performed as a contingency operation.
- Human-rated EELV-derived LVs will require new dedicated launch pads.

1.2.3 Technical GR&As

The Technical GR&As are listed below.

- The CEV will be designed for up to a crew of six for ISS missions.
- The CEV will be designed for up to a crew of four for lunar missions.
- The CEV will be designed for up to a crew of six for Mars missions.
- The CEV to support the lunar and Mars exploration missions and the ISS missions will use a single Outer Mold Line (OML) for the entry vehicle.
- Architectures will be designed for the lunar and Mars exploration missions and modified as required to support ISS missions.
- No more than four launches will be used to accomplish a single human lunar mission. This does not include infrastructure launches or supporting logistics.
- The following inert weight contingencies will be used:
  - Zero percent (0%) for existing LV elements with no planned specification change and no anticipated modifications (e.g., Space Shuttle Main Engine (SSME), RS–68, RD–180);
  - Five percent (5%) on existing LV elements requiring minimal modifications (e.g., External Tank (ET), Orbiter aft structure, EELV boosters, upper stages, and shrouds);
  - Ten percent (10%) on new Expendable Launch Vehicle (ELV) elements with direct Shuttle or EELV heritage;
  - Fifteen percent (15%) on new ELV elements with no heritage; and
  - Twenty percent (20%) on new in-space elements with no heritage (e.g., CEV, LSAM).
- Additional margins and factors of safety include the following:
  - Thirty percent (30%) margin for average power;
  - Two percent (2%) margin for reserves and residuals mass;
  - Two percent (2%) propellant tank ullage fractions for LV stages;
  - Fuel bias of nominal mixture ratio * 0.000246 * usable propellant weight;
  - A 2.0 factor of safety for crew cabins;
  - A 1.5 factor of safety on burst pressure for fluid pressure vessels;
  - A 1.4 ultimate factor of safety on all new or redesigned structures;
  - A 1.25 factor of safety on proof pressure for fluid pressure vessels;
  - Ten percent (10%) margin for rendezvous delta-Vs;
  - One percent (1%) ascent delta-V margin on LVs to account for dispersions;
  - Ten percent (10%) payload margin on all LV payload delivery predictions; and
  - Five percent (5%) additional payload margin on CaLV delivery predictions to account for Airborne Support Equipment (ASE).
- Technologies will be Technology Readiness Level-Six (TRL-6) or better by PDR.

1.2.4 Cost GR&As

The Cost GR&As are listed below.

- There will be only one CEV contractor after Calendar Year 2005 (CY05).
- There will be no 2008 CEV flight demonstration as originally planned.
- All Life Cycle Cost (LCC) estimates will include best-effort estimates of "full-cost" impacts (including corporate General and Administrative (G&A) at 5%, Center G&A, Center Civil Service salaries, travel, overhead, and Center service pool costs).
- Cost estimates will use 20 percent reserves for development.
- Cost estimates will use 10 percent reserves for operations.
- Cost estimates will use the April 2005 NASA New Start Inflation Index.

1.2.5 Schedule GR&As

The Schedule GR&As are listed below.

- There is a goal of 2011 for the first CEV human flight to ISS.
- There is a goal of performing the next human lunar landing by 2020—or as soon as practical.

1.2.6 Testing GR&As

The Testing GR&As are listed below.

- Ground Element Qualification
  - Elements will have ground qualification tests to demonstrate readiness for manned flight.
  - Multi-element integrated tests will be performed to demonstrate readiness for manned flight.
- Element Flight Qualification
  - Qualification of the CEV requires a minimum of one flight demonstrating full functionality prior to crewed flights.
  - Qualification of the LSAM requires a minimum of one flight demonstrating full functionality prior to lunar landing.
  - Qualification of any crewed LV requires three flight tests for human certification prior to crewed flight.
  - Qualification of any CaLV requires one flight test prior to flight of high-value cargo.
- Integrated System Qualification
  - Qualification of the EDS for firing while mated to a crewed element requires a minimum of two flights to demonstrate full functionality prior to crewed flight.
  - Lunar mission rehearsal in-space with appropriate architecture elements and crew is required prior to attempting a lunar landing.

1.2.7 Foreign Assets GR&As

- Foreign assets utilized in LV configurations in this study will be assumed to be licensed and produced in the United States.

1.3 Lunar Architecture

1.3.1 Introduction

As defined by this study, the lunar architecture is a combination of the lunar transportation "mission mode," the assignment of functionality to flight elements to perform the crewed lunar missions, and the definition of the activities to be performed on the lunar surface. The trade space for the lunar "mission mode," or approach to performing the crewed lunar missions, was limited to the cislunar space and Earth-orbital staging locations, the lunar surface activities duration and location, and the lunar abort/return strategies.

The mission mode analysis was built around a matrix of lunar- and Earth-staging nodes. Lunar-staging locations initially considered included the Earth-Moon L1 libration point, Low Lunar Orbit (LLO), and the lunar surface. Earth-orbital staging locations considered included due-east LEOs, higher-inclination ISS orbits, and raised apogee High Earth Orbits (HEOs). Cases that lack staging nodes (i.e., "direct" missions) in space and at Earth were also considered.

This study addressed lunar surface duration and location variables (including latitude, longitude, and surface stay-time) and made an effort to preserve the option for full global landing site access. Abort strategies were also considered from the lunar vicinity. "Anytime return" from the lunar surface is a desirable option that was analyzed along with options for orbital and surface loiter.

Definition of surface activities was equal in weight to the mission mode study. The duration, location, and centralization of lunar surface activities were analyzed by first determining the content of the science, resource utilization, Mars-forward technology demonstrations, and operational tests that could be performed during the lunar missions. The study looked at high-priority landing sites and chose a reference site in order to further investigate the operations at a permanent outpost. With the scientific and engineering activities defined, concept-level approaches for the deployment and buildup of the outpost were created. A comprehensive definition of lunar surface elements and infrastructure was not performed because development activities for lunar surface elements are still years in the future. Therefore, the ESAS team concentrated its recommendations on those elements that had the greatest impact on near-term decisions.

Additional details on the lunar architecture trade studies and analysis results are contained in **Section 4**, **Lunar Architecture**, of this report.

1.3.2 Lunar Mission Mode Analysis

1.3.2.1 Option Analysis Approach

The lunar mission mode option space considered the location of "nodes" in both cislunar space and the vicinity of Earth. The study originally considered cislunar nodes at the Earth-Moon L1 libration point, in LLO, and on the lunar surface. Respectively, these translate to Libration Point Rendezvous (LPR), LOR, and Lunar Surface Rendezvous (LSR) mission modes. The study also considered Earth-orbital staging locations in LEO, higher-inclination ISS orbits, and raised-apogee HEOs. In all three cases, elements brought together in any type of Earth orbit were generically termed an EOR mission mode. In the case of both cislunar and Earth orbital nodes, a mission type that bypassed a node completely was termed a "direct" mission or the term for the bypassed node was omitted altogether. Therefore, the Apollo missions were "direct" injection from Earth to the Moon, due to there being no EOR activities, and they were LOR at the Moon, owing to the rendezvous of the Command Module and Lunar Module (LM) following the surface mission. The Apollo mission mode was therefore popularly referred to as LOR.

LPR was eliminated early from the mission mode trade space. Recent studies performed by NASA mission designers concluded that equivalent landing site access and "anytime abort" conditions could be met by rendezvous missions in LLO with less propulsive delta-V and lower overall Initial Mass in Low Earth Orbit (IMLEO). If used only as a node for lunar missions, the L1 Earth-Moon LPR is inferior to the LOR mission mode.

With LPR eliminated, the mission mode question could be illustrated in a simple 2x2 matrix with the axes indicating the existence (or not) of an Earth orbital and lunar orbital node. The mission mode taxonomy could then be associated with each cell in this matrix—a mission that required EOR as well as rendezvous in lunar orbit was termed "EOR–LOR." A mission that injected directly to the Moon (bypassing Earth orbital operations) and returned directly from the surface of the Moon (bypassing lunar orbital operations) was termed "direct-direct." **Figure 1-9** illustrates the lunar mission mode matrix.

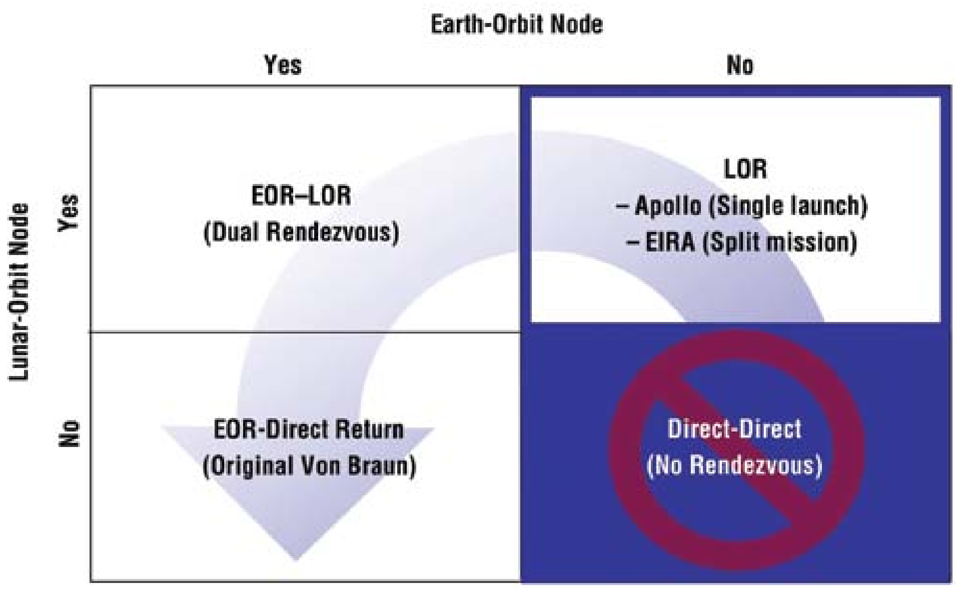


Figure 1-9. Lunar Mission Mode Taxonomy

This matrix becomes clearer when additional descriptions and certain historical lunar missions are added to the respective quadrants. The EOR-direct return mission (lower left-hand quadrant) was the mode favored by Wernher Von Braun early in the Apollo Program, while LOR (upper right-hand quadrant) was the mode eventually chosen. It became clear early in the ESAS analysis that the direct-direct mode (lower right-hand quadrant) would only be possible if the single LV it required had performance upwards of 200 mT to LEO. Because no LVs of this size were contemplated for this study due to cost and ground operations constraints, direct-direct was eliminated as a mission mode. The three remaining mission modes (LOR, EOR–LOR, and EOR-direct return) were analyzed in significant detail.

The EOR-direct return mission mode was examined for several analysis cycles but was eliminated from further consideration prior to the end of the study. In the direct return mode, the CEV must operate in, and transition among, 1-g prelaunch and post-landing, hyper-gravity launch, zero-gravity orbital and cruise, powered planetary landing and ascent, and 1/6-g lunar surface environments. This added significant complexity to a vehicle that must already perform a diverse set of functions in a diverse number of acceleration environments. Additionally, commonality of the SM between lunar and ISS configurations is further reduced in this case. The direct return lunar SM provides lunar ascent and TEI delta-V in excess of 2,400 m/s, the LOR SM is of the order of 1,850 m/s, and the ISS mission requires only 330 m/s. The direct return CEV also requires no docking mechanism since the CEV is the lone crew cabin for the round-trip mission. Conversely, this reduced the commonality from the ISS to the lunar CEV. Ultimately, the ESAS team concluded that the direct return mode entails the greatest number of operability issues and uncertainties, most notably to the configuration of the CEV, and that the complexities of a CEV designed for a surface-direct mission will increase the cost and schedule risks for delivering an ISS-compatible vehicle in the 2011–2012 time frame. Thus, the study team eliminated direct return on the basis of CEV complexity, poor margins, greatest number of operability issues and uncertainties, and highest sensitivity to mass growth.

1.3.2.2 Preferred Mission Mode Options

Mission mode analysis was performed in multiple cycles, with each cycle resulting in performance, cost, reliability, safety, and other FOMs with which to compare the mission options. At the end of each analysis cycle, decisions were made to eliminate certain mission modes or to perform additional studies to further drive out the differences among the options. A baseline was first chosen against which all design options could be compared. The baseline chosen by the ESAS team was a 2-launch LOR split mission termed the ESAS Initial Reference Architecture (EIRA).

For the final analysis cycle, the EIRA mission architecture was compared to a 2-launch EOR–LOR approach, which used two launches of a 100-mT LEO payload vehicle, and a "1.5-launch" EOR–LOR approach, which used a launch of a 125-mT LEO cargo vehicle and a smaller CLV. They were also compared for three different levels of propulsion technology. The baseline option used pressure-fed Liquid Oxygen (LOX)/methane engines on the CEV SM and the lander ascent and descent stages to maximize commonality. A second option substituted a pump-fed LOX/hydrogen system on the lander descent stage to improve performance. The third option also used LOX/hydrogen for the lander descent stage and substituted a pump-fed LOX/methane system for the ascent stage propulsion system.

The three final mission mode candidates are described in the following paragraphs.

1.3.2.2.1 EIRA 2-launch LOR Split Mission Architecture

The assumed mission mode for the EIRA is a 2-launch "split" architecture with LOR, wherein the LSAM is predeployed in a single launch to LLO and a second launch of the same vehicle delivers the CEV and crew in lunar orbit where the two vehicles initially rendezvous and dock. The entire crew then transfers to the LSAM, undocks from the CEV, and performs descent to the surface. The CEV CM and SM are left unoccupied in LLO. After a lunar stay of up to 7 days, the LSAM returns the crew to lunar orbit and docks with the CEV, and the crew transfers back to the CEV. The CEV then returns the crew to Earth with a direct-entry-and-land touchdown, while the LSAM is disposed of on the lunar surface. This mission mode is illustrated in **Figure 1-10**.

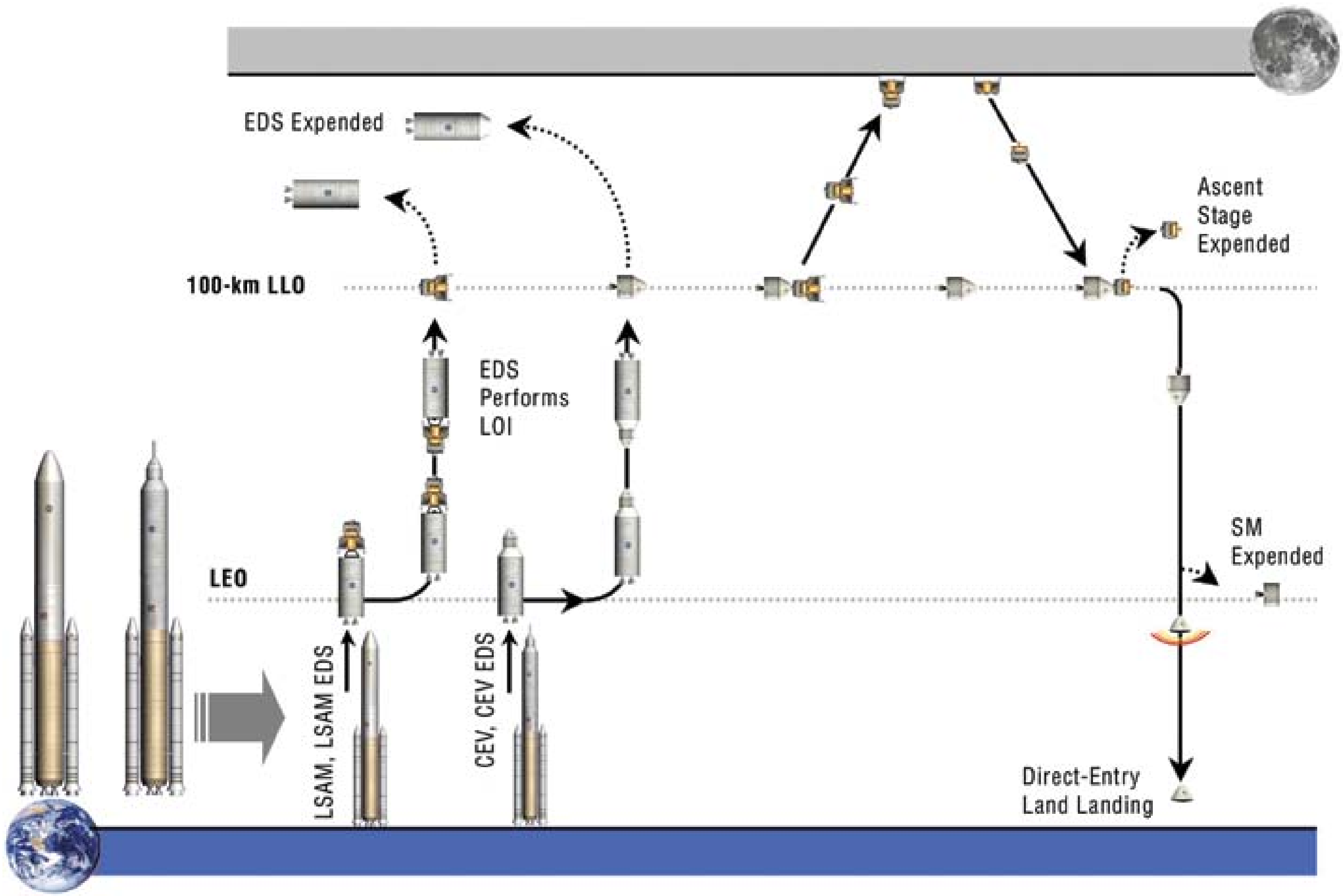


Figure 1-10. EIRA 2-launch LOR Split Mission Architecture

1.3.2.2.2 2-launch EOR–LOR Mission Architecture

The EOR–LOR architecture (**Figure 1-11**) is functionally similar to the EIRA, with the primary difference being that the initial CEV–LSAM docking occurs in LEO rather than LLO. Whereas the EIRA incorporated two smaller EDSs in two launches to deliver the CEV and LSAM to the Moon, the EOR–LOR architecture divides its launches into one launch for a single, large EDS and the second launch for the CEV, crew, and LSAM. The combined CEV and LSAM dock with the EDS in Earth orbit, and the EDS performs TLI. Another difference between the EIRA and EOR–LOR architectures is that, for the baseline pressure-fed LOX/methane propulsion system, the EDS performs LOI for the EIRA. Due to launch performance limitations of the single EDS with EOR–LOR, LOI is instead executed by the CEV for optimum performance. Once the CEV and LSAM reach LLO, this mission mode is identical to the EIRA.

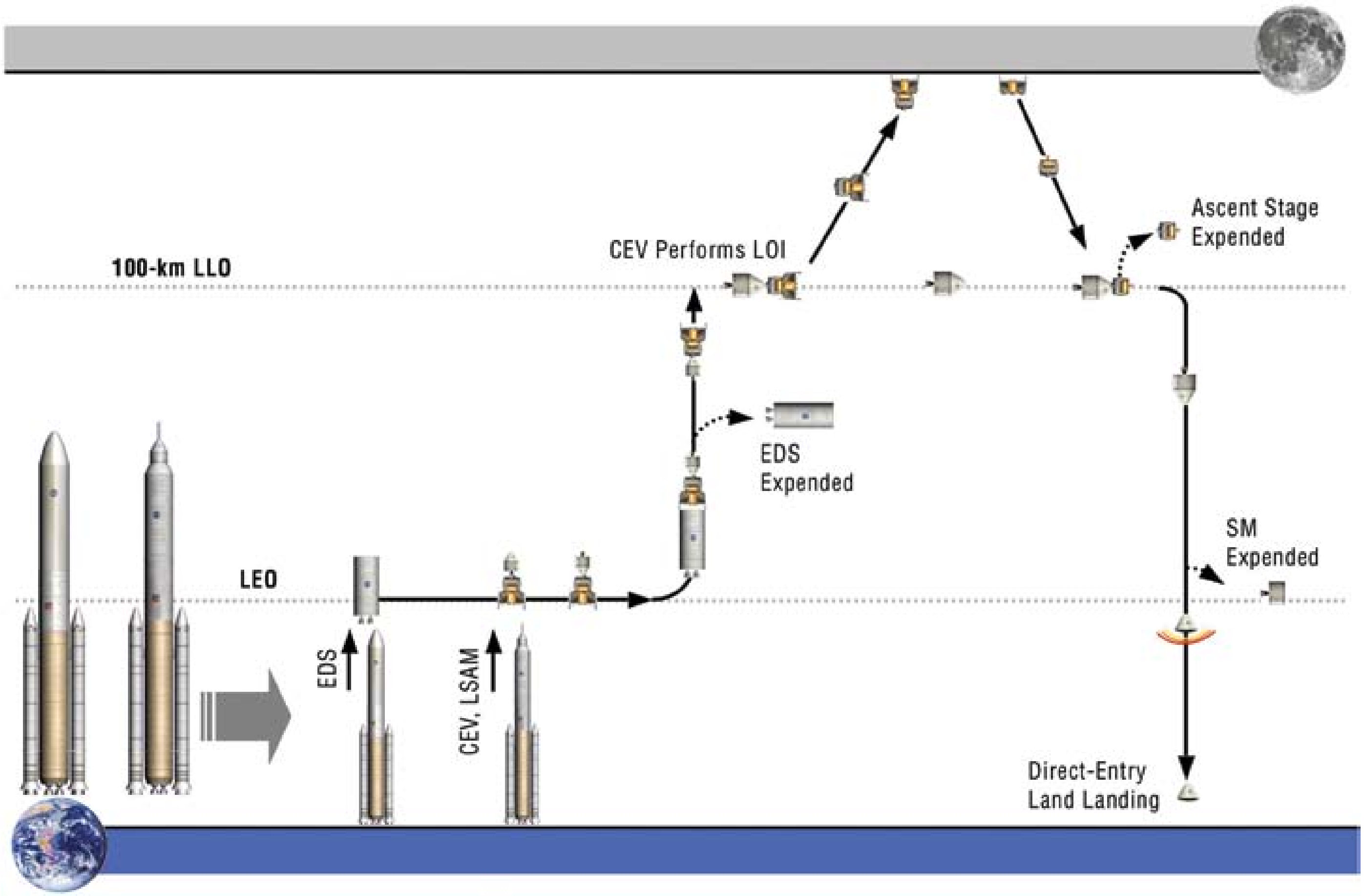


Figure 1-11. 2-launch EOR–LOR Mission Architecture

1.3.2.2.3 1.5-Launch EOR–LOR Mission Architecture

The use of LOX/hydrogen propulsion on the lander reduces the architecture masses sufficiently to enable a second EOR–LOR option. This variant, known as 1.5-launch EOR–LOR, is so named due to the large difference in size and capability of the LVs used in the architecture. Whereas the previous architectures have used one heavy-lift CaLV to launch cargo elements and another heavy-lift CLV to launch the CEV and crew, this architecture divides its launches between one large and one relatively small LV. The 1.5-launch EOR–LOR mission is an EOR–LOR architecture with the LSAM and EDS predeployed in a single launch to LEO with the heavy-lift CaLV. A second launch of a 25-mT-class CLV delivers the CEV and crew to orbit, where the two vehicles initially rendezvous and dock. The EDS then performs the TLI burn for the LSAM and CEV and is discarded. Upon reaching the Moon, the LSAM performs the LOI for the two mated elements, and the entire crew transfers to the LSAM, undocks from the CEV, and performs descent to the surface. The CEV is left unoccupied in LLO. After a lunar stay of up to 7 days, the LSAM returns the crew to lunar orbit, where the LSAM and CEV dock and the crew transfers back to the CEV. The CEV then returns the crew to Earth with a direct- or skip-entry-and-land touchdown, while the LSAM is disposed of via impact on the lunar surface. The 1.5-launch EOR–LOR architecture is illustrated in **Figure 1-12**.

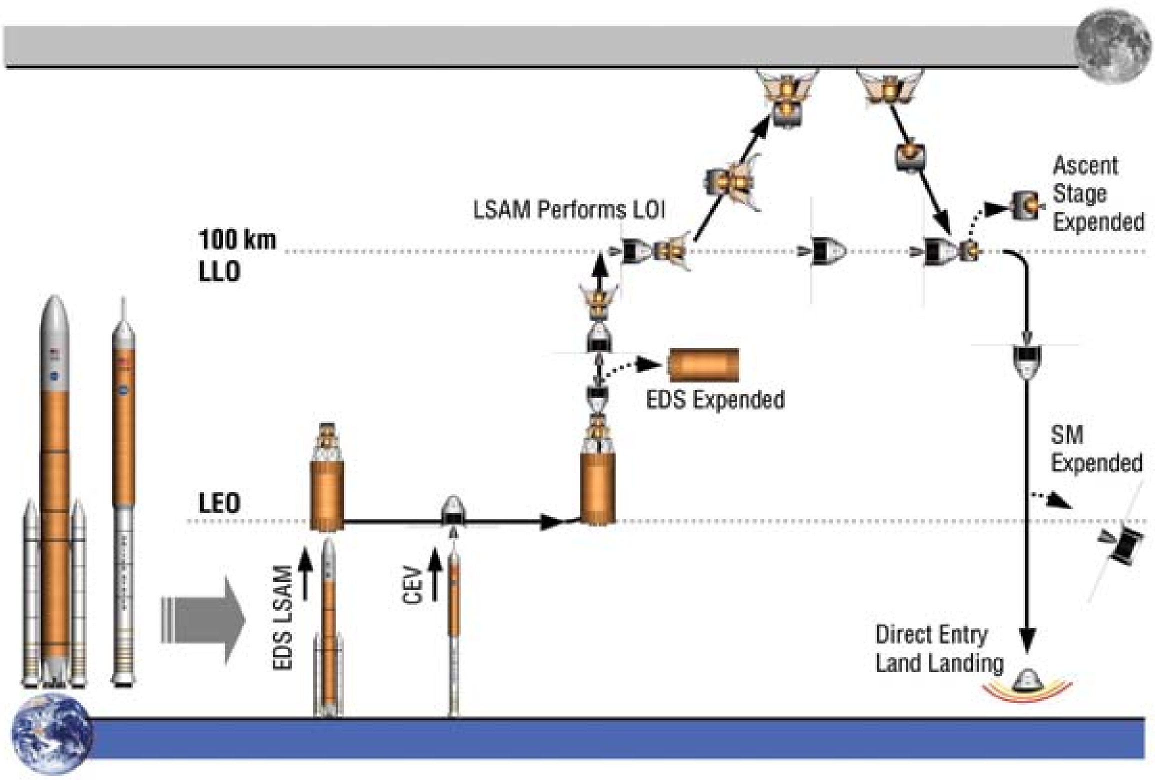


Figure 1-12. 1.5-Launch EOR–LOR Mission Architecture

1.3.2.3 Mission Mode Analysis Results

The team generated mission performance analysis for each option (IMLEO, number of launches required, and launch margins), integrated program costs through 2025, safety and reliability estimates (Probability of Loss of Crew (P(LOC)), and Probability of Loss of Mission (P(LOM)), and other discriminating FOMs.

1.3.2.3.1 Safety and Reliability

Three mission modes were analyzed, with three different propulsion technologies applied. In addition to the LOR, EOR–LOR "2-launch," and EOR–LOR "1.5-launch" modes, analysis was also performed on a single-launch mission that launched both the CEV and lander atop a single heavy-lift CaLV (the same used for the 1.5-launch solution), much like the Apollo/Saturn V configuration. However, the limited lift capability provided by this approach limited its landing site capabilities to the same equatorial band explored by Apollo, in addition to the lunar poles. For each of the mission modes, end-to-end single-mission probabilities of LOC and LOM were calculated for (1) a baseline propulsive case using all pressure-fed LOX/methane engines, (2) a case where a LOX/hydrogen pump-fed engine was substituted on the lander descent stage, and (3) a third case where the lander ascent stage engine was changed to pump-fed LOX/methane. **Figures 1-13** and **1-14** illustrate the P(LOC) and P(LOM) for each of these cases.

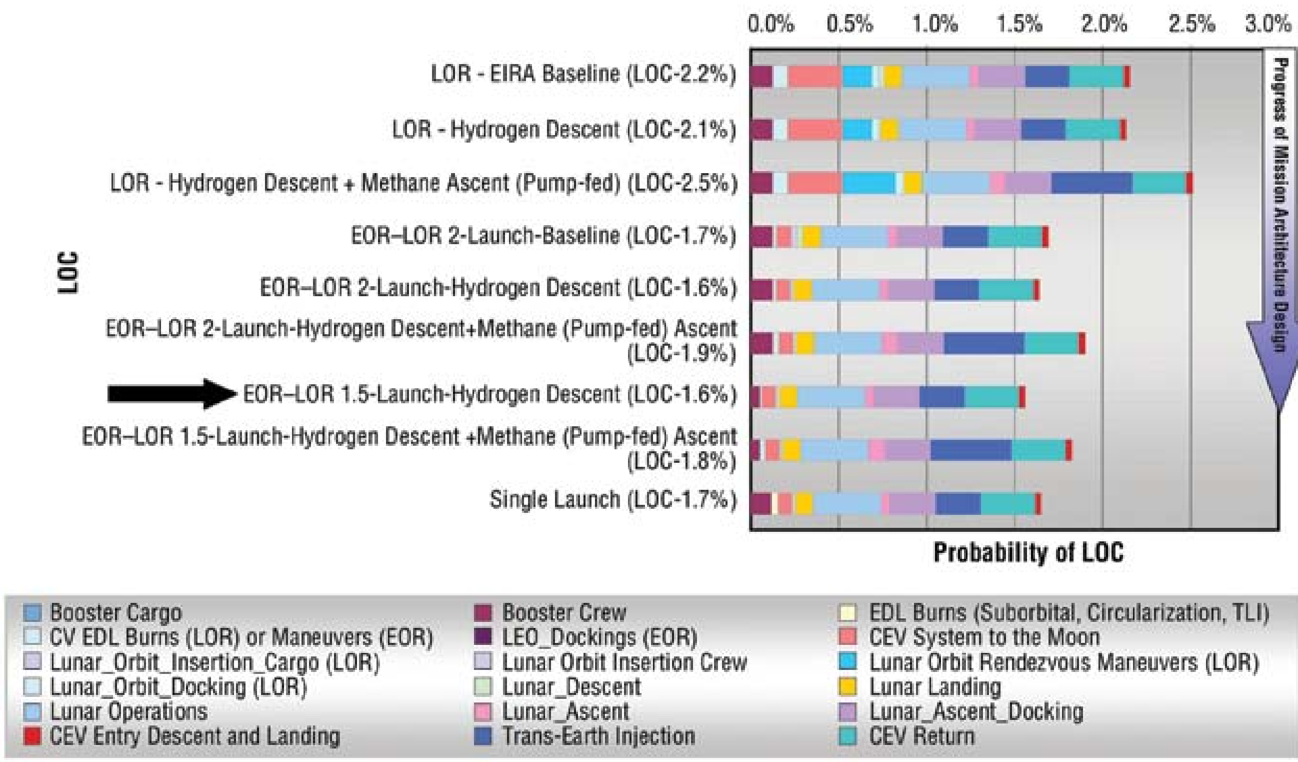


Figure 1-13. LOC Comparison

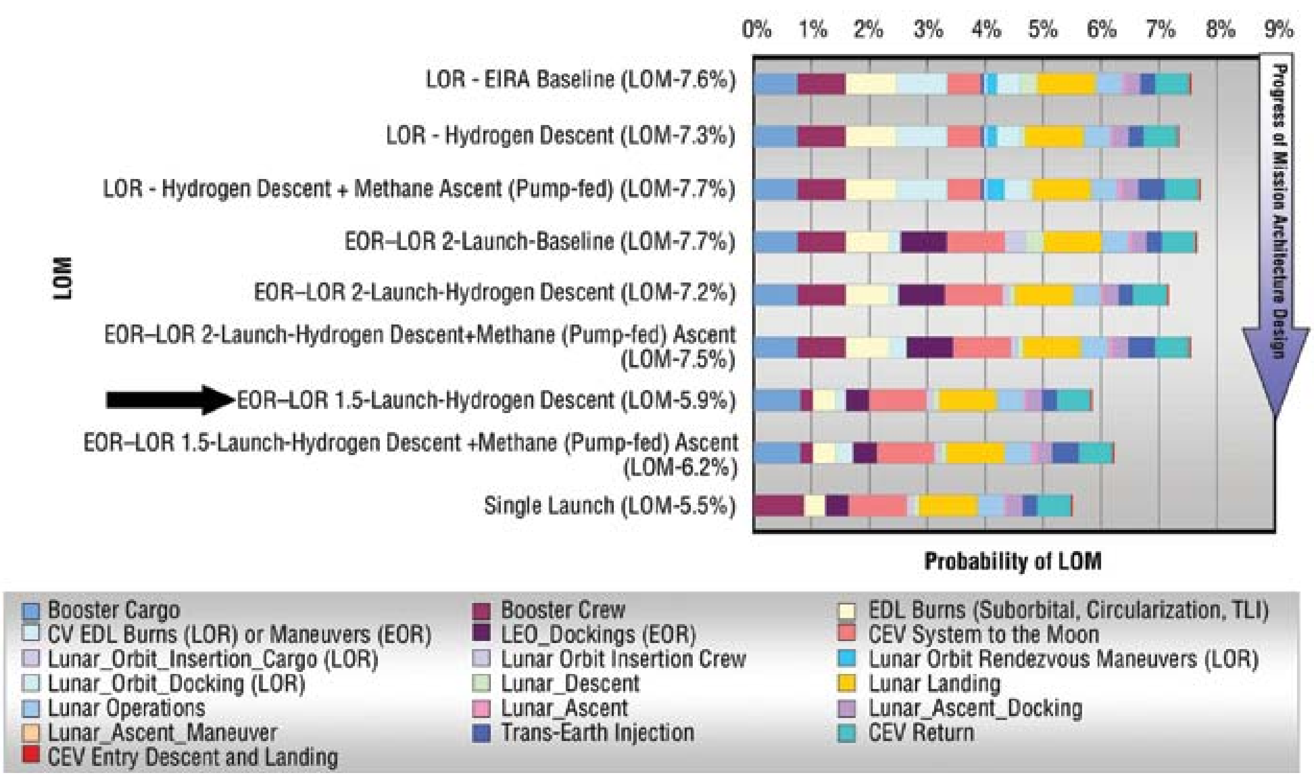


Figure 1-14. LOM Comparison

P(LOC) was dominated by propulsive events and vehicle operating lifetimes. As shown in **Figure 1-13**, LVs varied only slightly between the 2-launch (crew launched on a heavy-lift booster) and 1.5-launch (crew launched on a single Solid Rocket Booster (SRB) CLV) options. The LOR options had added risk due to the lander being sent to the lunar orbit separately from the CEV, and thus not having a backup crew volume during transit to handle "Apollo 13"-like contingencies. The LOR mission also required the CEV SM to perform an LOI maneuver. Generally, each time a pump-fed engine technology was introduced to replace a pressure-fed system, risk increased, although the LOX/hydrogen engine modeled for the lander descent stage had a high degree of heritage from existing RL–10 engine technology.

When all the mission event probabilities were summed, all mission options fell within a relatively narrow range (1.6 to 2.5 percent), but the difference between the highest- and lowest-risk options approached a factor of two. Missions using the LOR mission mode were the highest risk options, while EOR–LOR "1.5-launch" options were the lowest. Missions that utilized a higher-performing LOX/hydrogen lander descent stage scored approximately the same as the baseline option that used pressure-fed LOX/methane, but a change to a pump-fed LOX/methane ascent stage resulted in an appreciable increase in risk. The lowest P(LOC) option was the 1.5-launch EOR–LOR mission using a pump-fed LOX/hydrogen lander descent stage and pressure-fed LOX/methane engines for both the lander ascent stage and CEV SM.

P(LOM) generally followed the same trends as P(LOC). **Figure 1-14** illustrates the reliability benefits of launching crew on the single-SRB CLV, the reduced risk of having a single EDS, and the penalties associated with pump-fed engines. LOR and EOR–LOR 2-launch options exhibited the greatest P(LOM), in a range between 7 and 8 percent per mission. The substitution of a LOX/hydrogen lander descent stage engine actually increased mission reliability by adding engine-out performance to the LOI and lunar landing phases of the mission, but further pushing LOX/methane engine technology toward a pump-fed system lowered reliability by eliminating commonality with the CEV SM engine and adding complexity.

The single-launch mission option scored the highest reliability overall, owing mainly to it requiring only a single launch. Of the missions that provide the full lunar landing site access and return capabilities, EOR–LOR 1.5-launch modes were nearly equal to the single-launch option. Specifically, the EOR–LOR 1.5-launch option using the LOX/hydrogen lander descent stage engines scored the lowest P(LOM) among the full-up mission options. Interestingly, this same mission mode and propulsion technology combination scored the lowest P(LOC) as well.

1.3.2.3.2 Mission Mode Cost Comparison

Figure 1-15 summarizes the Life Cycle Cost (LCC) analysis results. To enable a fair comparison among the options, the complete LCC, including Design, Development, Test, and Evaluation (DDT&E), flight units, operations, technology development, robotic precursors, and facilities, were all included in this analysis. Generally, the choice of mission mode had only a small effect on the LCC of the exploration program. Of the options modeled, the 1.5-launch EOR–LOR mission using a LOX/hydrogen lander descent stage propulsion system exhibited an LCC that was in the same range as the other options.

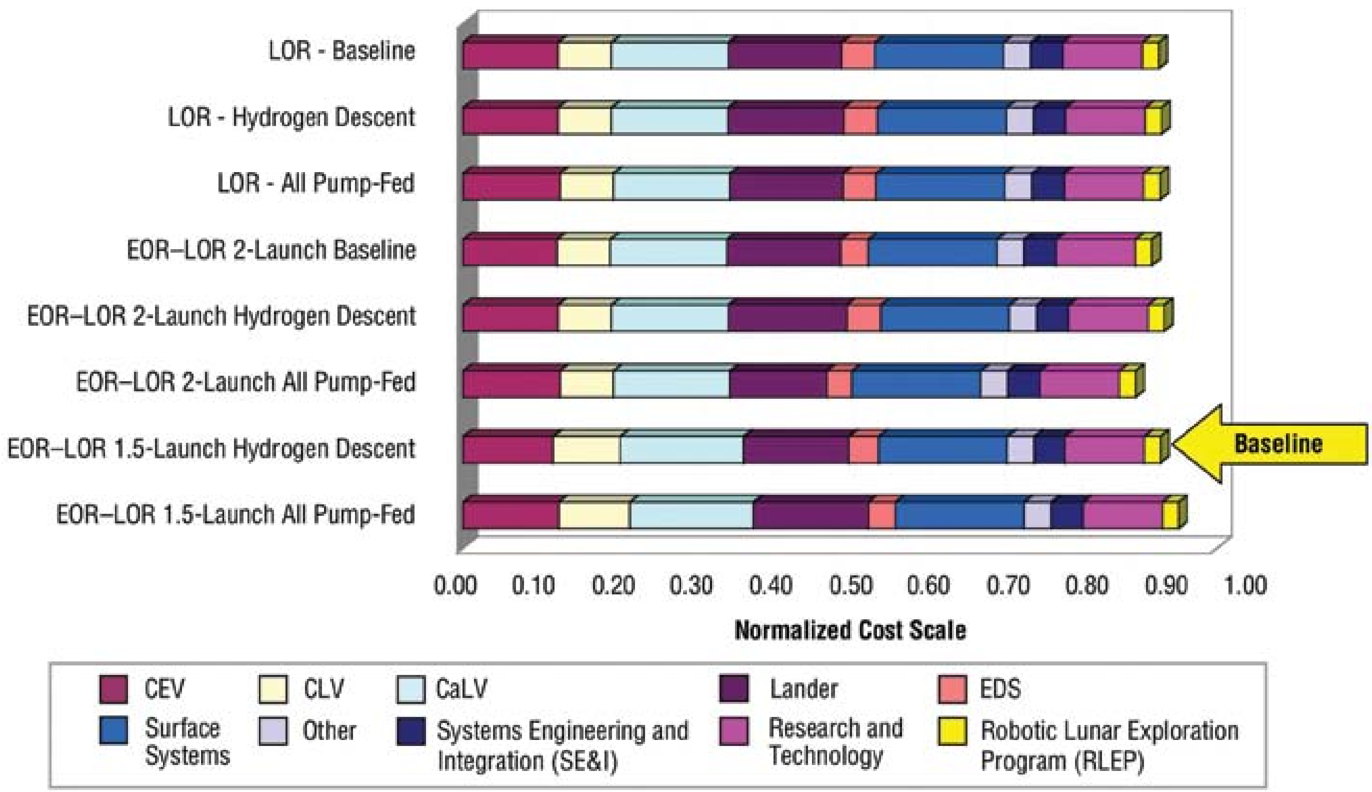


Figure 1-15. Mission Mode LCC Through 2025

Of the full-performance options studied, the 1.5-launch EOR–LOR mode yielded both the lowest P(LOM) and the lowest P(LOC) when flown with a LOX/hydrogen lander descent stage and common pressure-fed LOX/methane propulsion system for both the lander ascent stage and CEV SM. Cost analysis was less definitive, but also showed this same EOR–LOR 1.5-launch option being among the lowest cost of all the alternatives studied. Based on the convergence of robust technical performance, low P(LOC), low P(LOM), and low LCC, the 1.5-launch EOR–LOR option using LOX/hydrogen lander descent stage propulsion was selected as the mission mode to return crews to the Moon.

1.3.3 LSAM Reference Design

The ESAS team examined the unique architecture of the lunar lander, or LSAM. Other architecture element designs and trade studies were also accomplished by the team. The reference LSAM concept, shown in **Figure 1-16**, for the ESAS 1.5-launch EOR–LOR architecture is a two-stage, single-cabin lander similar in form and function to the Apollo LM.

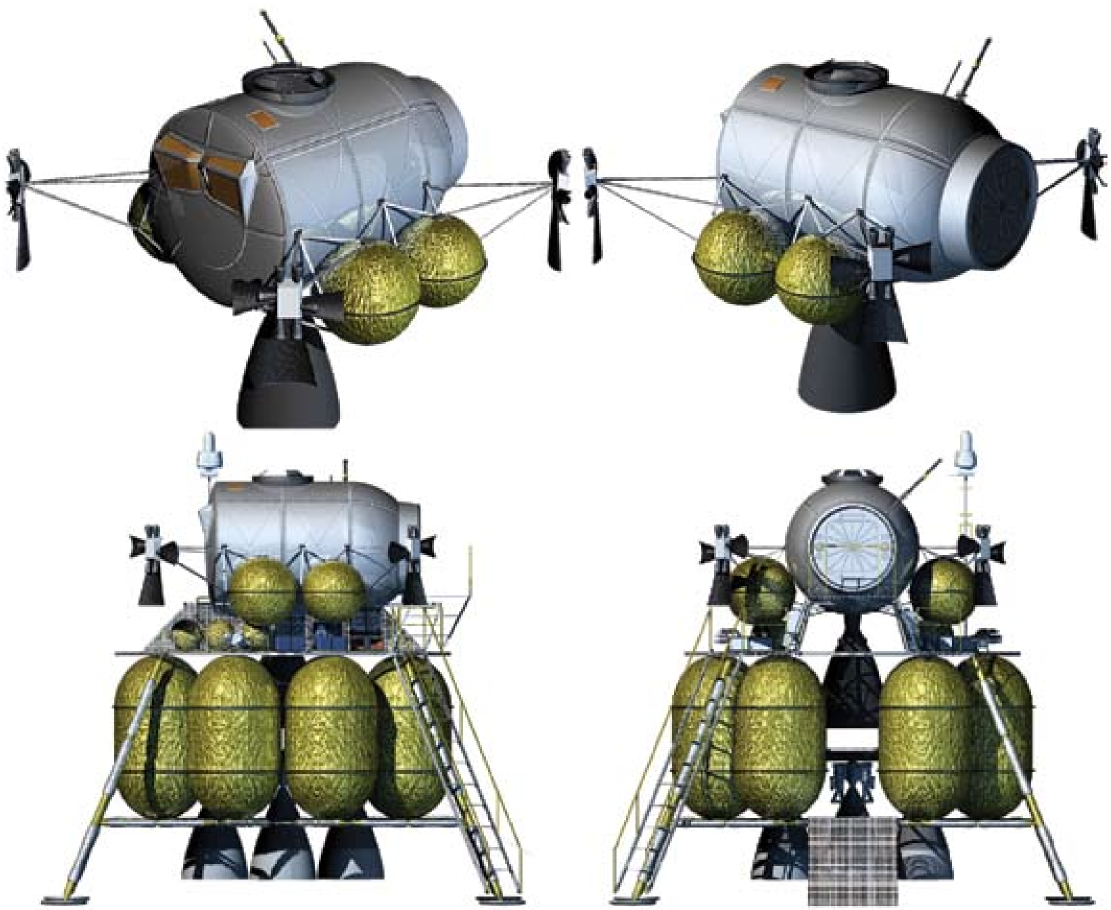


Figure 1-16. LSAM Configuration

The LSAM ascent stage, in conjunction with the descent stage, is capable of supporting four crew members for 7 days on the lunar surface and transporting the crew from the surface to lunar orbit. The ascent stage utilizes an integrated pressure-fed LOX/methane propulsion system, similar to the CEV SM, to perform coplanar ascent to a 100-km circular lunar orbit, rendezvous and docking with the CEV, and self-disposal following separation from the CEV. A single 44.5-kN (10,000-lbf) ascent propulsion system engine and sixteen 445-N (100-lbf) Reaction Control System (RCS) thrusters are used for vehicle maneuvering and attitude control. Spherical ascent stage propellant tanks are sized to perform up to 1,866 m/s of main engine and 22 m/s of RCS delta-V.

The LSAM pressure vessel is a horizontal short cylinder 3.0 m in diameter and 5.0 m long to provide 31.8 m^3 of pressurized volume for the crew during lunar operations. A nominal internal atmospheric pressure for the ascent stage of 65.5 kPa (9.5 psia) with a 30 percent oxygen composition has been assumed. The LSAM's EVA strategy while on the lunar surface is daily

EVA with all four crew members egressing the vehicle simultaneously. For missions lasting beyond 4 days, a rest day between EVAs may be required. Unlike the Apollo LM, the LSAM ascent stage crew cabin includes a bulkhead to partition a section of the pressurized volume, which can serve as an internal airlock. Thus, crew members don their surface EVA suits in the airlock, depressurize the airlock, and egress the vehicle. Ascent stage power generation capabilities include rechargeable batteries for the 3 hours from liftoff to docking with the CEV. Power generation for all other LSAM operations prior to liftoff is provided by the descent stage.

The LSAM descent stage is used in crewed lunar exploration missions to insert the CEV into LLO, land the ascent stage and cargo on the surface, and provide the vehicle's life support and power generation capabilities during an assumed 7-day lunar surface stay. The descent stage uses a pump-fed LOX/hydrogen descent propulsion system to perform LOI and coplanar descent from a 100-km circular lunar orbit. Four 66.7-kN (15,000-lbf) descent propulsion system derived from the RL–10 engine family are used for vehicle maneuvering, while the ascent stage RCS is used for combined-vehicle attitude control. The descent propulsion system engines are arranged symmetrically around the vehicle centerline at the base of the descent stage.

Six cylindrical hydrogen and two cylindrical oxygen descent stage tanks are included on the LSAM to store the propellant needed to perform up to 1,100 m/s of LOI delta-V with the CEV and ascent stage attached, and 1,900 m/s of descent delta-V with only the ascent stage attached. The eight LSAM propellant tanks are mounted around the descent stage in a ring arrangement, leaving two open bays on opposite sides of the stage exterior for surface access and cargo stowage, and a circular opening along the vehicle centerline for housing the single ascent stage engine nozzle. In addition to supporting its own propulsion system, the descent stage structure also serves as a support system and launch platform for the ascent stage, provides attachment for a four-leg landing gear system, provides for crew access to the surface, and serves as the attachment point to the EDS.

Three Proton Exchange Membrane (PEM) fuel cells on the descent stage provide LSAM power generation from Earth launch to lunar ascent. Oxygen reactant for the fuel cells is stored in the oxygen propellant tanks, while hydrogen reactant is stored in the hydrogen propellant tanks. The descent stage also contains the gaseous nitrogen, potable water, and water storage systems needed for the mission up to lunar ascent. These systems were included on the descent stage rather than the ascent stage to avoid the penalty of lifting unnecessary mass back to lunar orbit. Finally, the descent stage provides the mounting location for the Active Thermal Control System (ATCS) radiators. LSAM heat rejection following liftoff from the lunar surface is accomplished using a fluid evaporator system.

1.3.4 Lunar Architecture Recommendations

The lunar architecture defined by the ESAS integrates mission mode analysis, flight element functionality, and the activities to be performed on the lunar surface. An integrated analysis of mission performance, safety, reliability, and cost led to the selection of a preferred mission mode, the definition of functional and performance requirements for the vehicle set, and the definition of lunar surface operations. Additionally, the analysis looked back to examine how the CEV and CLV would be used to transport crew and cargo to the ISS, and forward to define the systems that will carry explorers to Mars and beyond, in order to identify evolutionary functional, technological, and operational threads that bind these destinations together.

1.3.4.1 Mission Mode

The ESAS team recommends a combination of EOR–LOR as the preferred lunar mission mode. The mission mode is the fundamental lunar architecture decision that defines where space flight elements come together and what functions each of these elements perform. The EOR–LOR mode is executed with a combination of the launch of separate crew and cargo vehicles, and by utilizing separate CEV and lander vehicles that rendezvous in lunar orbit. This mission mode combined superior performance with low LCC and high crew safety and mission reliability.

The lunar mission mode study initially considered a wide variety of locations of transportation "nodes" in both cislunar space and the vicinity of Earth. Initial analyses eliminated libration point staging and direct return mission options, leaving the mission mode analysis to investigate a matrix of low-lunar (LOR) and Earth-orbital (EOR) staging nodes.

1.3.4.2 Mission Sequence

The ESAS team recommends a mission sequence that uses a single launch of the CaLV to place the lunar lander and EDS in Earth orbit. The launch of a CLV will follow and place the CEV and crew in Earth orbit, where the CEV and lander/EDS will rendezvous. The combination of the large cargo launch and the CLV is termed a "1.5-launch EOR–LOR" mission. Following rendezvous and checkout in LEO, the EDS will then inject the stack on a trans-lunar trajectory and be expended. The lander and CEV are captured into lunar orbit by the descent stage of the two-stage lander, and all four crew members transfer to the lander and descend to the surface, leaving the CEV operating autonomously in orbit. The lander features an airlock and the capability to support up to a 7-day surface sortie. Following the lunar surface mission, the lander's ascent stage returns the crew to lunar orbit and docks with the orbiting CEV. The crew transfers back to the CEV and departs the Moon using the CEV SM propulsion system. The CEV then performs a direct-Earth-entry and parachutes to a land landing on the west coast of the United States.

1.3.4.3 Lunar Surface Activities

Recommended lunar surface activities consist of a balance of lunar science, resource utilization, and "Mars-forward" technology and operational demonstrations. The architecture will initially enable sortie-class missions of up to 7 days duration with the entire crew of four residing in and performing EVAs from the lunar lander.

The ESAS team recommends the deployment of a lunar outpost using the "incremental build" approach. Along with the crew, the lander can deliver 500 kg of payload to the surface, and up to 2,200 kg of additional payload if the maximum landed capacity is utilized. This capability opens the possibility of deploying an outpost incrementally by accumulating components delivered by sortie missions to a common location. This approach is more demanding than one that delivers larger cargo elements. In particular, the habitat, power system, pressurized rovers, and some resource utilization equipment will be challenging to divide and deploy in component pieces. The alternative to this incremental approach is to develop a dedicated cargo lander that can deliver large payloads of up to 21 mT.

The study team defined high-priority landing sites that were used to establish sortie mission performance. Of those sites, a south polar location was chosen as a reference outpost site in order to further investigate the operations at a permanent outpost. A photovoltaic power system was chosen as the baseline power system for the outpost.

1.3.4.4 Propulsion Choices and Delta-V Assignment

The ESAS team examined a wide variety of propulsion system types and potential delta-V allocations for each architecture element. It is recommended that the CaLV's upper stage will serve as the EDS for lunar missions and will perform the TLI propulsive maneuver. The descent stage of the lunar lander was selected to perform LOI and up to 200 m/s of plane change using LOX/hydrogen propulsion. The lunar lander descent stage will perform a coplanar descent to the surface using the same engine that performed LOI, and the crew will perform the surface mission while the CEV orbits autonomously. The lunar lander ascent stage will perform a coplanar ascent using LOX/methane propulsion that is common with the CEV SM propulsion system. The SM will perform up to a 90-deg plane change and TEI with full co-azimuth control (1,450 m/s total delta-V).

Pump-fed LOX/hydrogen propulsion was selected for the lunar descent stage because of the great performance, cost, and risk leverage that was found when the lunar lander descent stage propulsion efficiency was increased by the use of a LOX/hydrogen system. To achieve a high-reliability lunar ascent propulsion system, and to establish the linkage to in-situ propellant use, common pressure-fed LOX/methane engines were chosen for the CEV SM and lunar ascent stage propulsion systems.

1.3.4.5 Global Access

It is recommended that the lunar architecture preserve the option for full global landing site access for sortie or outpost missions. Landing at any site on the Moon sizes the magnitude of the LOI maneuver. A nominal 900-m/s LOI burn enables access to the equator and poles, and a maximum of 1,313 m/s is required for immediate access to any site on the lunar globe. The architecture uses a combination of orbital loiter and delta-V to access any landing in order to balance additional propulsive requirements on the lander descent stage and additional orbital lifetime of the CEV systems. The lander descent stage was sized for a 900-m/s LOI plus a 200-m/s maximum nodal plane change, for a total of 1,100 m/s in addition to lunar descent propulsion. This value allows the crew to immediately access 84 percent of the lunar surface and to have full global access with no more than 3 days loiter in lunar orbit.

1.3.4.6 Anytime Return

It is recommended that the architecture provide the capability to return to Earth in 5 days or less for sortie missions at any site on the lunar globe. The requirement to return anytime from the surface of the Moon to Earth was the design driver of the SM propulsion system. The lunar mission requires a total of 1,450 m/s of delta-V, combining a 900-m/s TEI maneuver, a worst-case 90-deg nodal plane change, and Earth entry azimuth control. This capability enables "anytime return" if the lander is able to perform a coplanar ascent to the CEV. For sortie duration missions of 7 days or less, the CEV's orbital inclination and node will be chosen to enable a coplanar ascent. Outpost missions will also have "anytime return" capability if the outpost is located at a polar or equatorial site. For other sites, loitering on the surface at the outpost may be required to enable ascent to the orbiting CEV.

1.3.4.7 Lunar Lander

The recommended lunar lander provides the capability to capture itself and the CEV into lunar orbit, to perform a plane change prior to descent, and to descend to the lunar surface with all four crew members using a throttleable LOX/hydrogen propulsion system. On the lunar surface, the lander serves as the crew's habitat for the duration of the surface stay and provides full airlock capability for EVA. Additionally, the lander carries a nominal payload of 500 kg and has the capability to deliver an additional 2,200 kg to the lunar surface. The lander's ascent stage uses LOX/methane propulsion to carry the crew back into lunar orbit to rendezvous with the waiting CEV. The lander's propulsion system is chosen to make it compatible with ISRU-produced propellants and common with the CEV SM propulsion system.

1.3.4.8 ISS-Moon-Mars Connections

Evolutionary paths were established within the architecture to link near-term ISS crew and cargo delivery missions, human missions to the lunar surface, and farther-term human missions to Mars and beyond. The key paths that enable the architecture to evolve over time are the design of the CEV, the choice of CLV and CaLV, the selection of technologies (particularly propulsion technologies), and the operational procedures and systems that extend across the destinations. The CEV is sized to accommodate crew sizes up to the Mars complement of six. The CLV was chosen to be a reliable crew launch system that would be the starting point of a crew's journey to the ISS, Moon, or Mars; and the CaLV was chosen, in part, to deliver 100-mT-class human Mars mission payloads to LEO. Propulsion choices were made to link propulsive elements for the purpose of risk reduction, and to enable the use of future ISRU-produced propellants. These propellant choices are further linked to the ISRU technology experiments to be performed on the planetary surfaces. Finally, EVA systems and mission operations will be developed to share common attributes across all ISS, lunar, and Mars destinations.

1.4 Crew Exploration Vehicle (CEV)

1.4.1 Overview

One of the key requirements to enable a successful human space exploration program is the development and implementation of a vehicle capable of transporting and housing crew on LEO, lunar, and Mars missions. A major portion of the ESAS effort focused on the design and development of the CEV, the means by which NASA plans to accomplish these mission objectives. The CEV reference design includes a pressurized CM to support the Earth launch and return of a crew of up to six, a Launch Abort System (LAS), and an unpressurized SM to provide propulsion, power, and other supporting capabilities to meet the CEV's in-space mission needs.

In response to the ESAS charter, the first crewed flight of the CEV system to the ISS was assumed to occur in 2011. The CEV design requirements were, however, to be focused on exploration needs beyond LEO. Therefore, the team started with the existing ESMD Revision E Crew Transportation System (CTS) requirements and assessed these against ISS needs for areas of concern where CEV may fall short of ISS expectations. Any such shortcomings were then examined on a case-by-case basis to determine whether they were critical to performing the ISS support function. If they were found not to be critical, such shortcomings were considered as guidelines and not requirements on the CEV.

While the CEV design was sized for lunar missions carrying a crew of four, the vehicle was designed to also be reconfigurable to accommodate up to six crew for ISS and future Mars mission scenarios. The CEV can transfer and return crew and cargo to the ISS and stay for 6 months in a quiescent state. The lunar CEV design has direct applications to ISS missions without significant changes in the vehicle design. The lunar and ISS configurations share the same SM, but the ISS mission has much lower delta-V requirements. Hence, the SM propellant tanks can be loaded with additional propellant for ISS missions to provide benefits in launch aborts, on-orbit phasing, and ISS re-boost. Other vehicle block derivatives can deliver pressurized and unpressurized cargo to the ISS.

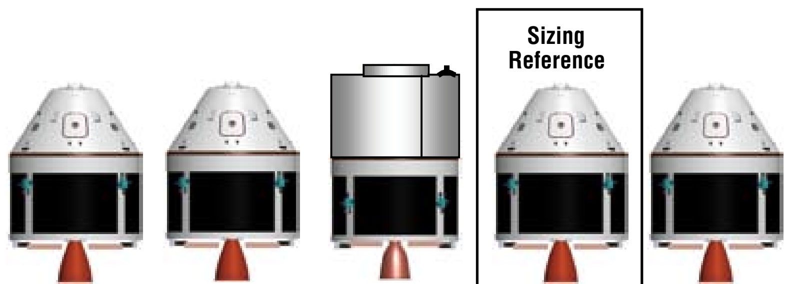
Vehicle size, layout, and mass were of central importance in this study, as each factors into vital aspects of mission planning considerations. Detailed subsystem definitions were developed and vehicle layouts were completed for a four-crew-member lunar DRM and a six-crew-member Mars DRM. The lunar mission was a design driver because it had the most active days with the crew inside. The Mars DRM, which was a short duration mission of only 1 to 2 days to and from an orbiting MTV, drove the design to accommodate a crew of six. Ultimately, the CEV CM was sized to be configurable for accommodating six crew members even for an early mission to the ISS.

Additional details on the CEV trade studies and analysis results are contained in **Section 5**, **Crew Exploration Vehicle**, of this report.

1.4.2 CEV Modular Design Approach

The different CEV configurations were each assigned a block number to distinguish their unique functionality. The Block 1 vehicles support the ISS with transfer of crew and cargo. The Block 1A vehicle transfers crew to and from the ISS. This vehicle can stay at the ISS for 6 months. Varying complements of crew and pressurized cargo can be transported in the Block 1A CM. The Block 1B CM transports pressurized cargo to and from the ISS. The crew accommodations are removed and replaced with a secondary structure to support the cargo complement. The relationship between the Block 1A and Block 1B CMs is similar to that of the Russian Soyuz and Progress vehicles. Unpressurized cargo can be transported to the ISS via the CDV. The CDV replaces the CM with a structural "strong back" that supports the cargo being transferred. The CDV uses the same SM as the other blocks and also requires a suite of avionics to perform this mission. The CDV is expended after its delivery mission. The Block 2 CEV is the reference platform sized to transfer crew to the lunar vicinity and back. Detailed sizing was performed for this configuration and the other blocks were derived from its design. The Block 3 configuration is envisioned as a crewed transfer vehicle to and from an MTV in Earth orbit. The crew complement for this configuration is six. No detailed design requirements were established for this block and detailed mass estimates were never derived.

Design details for each block configuration are discussed in **Section 5**, **Crew Exploration Vehicle**. A mass summary for each block is shown in **Figure 1-17**. Detailed mass statements were derived for each block and are provided in **Appendix 5A**, **CEV Detailed Mass Breakdowns**.



	Block 1A ISS Crew	Block 1B ISS Press Cargo	CDV ISS Unpress Cargo	Block 2 Lunar Crew	Block 3 Mars Crew
Crew Size	3	0	0	4	6
LAS Required	4,218	None	None	4,218	4,218
Cargo Capability (kg)[1]	400	3,500	6,000	Minimal	Minimal
CM (kg)	9,342	11,381	12,200	9,506	TBD
SM (kg)	13,558	11,519	6,912	13,647	TBD
Service Propulsion System delta-V (m/s)	1,544[2]	1,098[2]	330	1,724	TBD
EOR–LOR 5.5-m Total Mass (kg)	22,900	22,900	19,112	23,153	TBD

Note 1: Cargo capability is the total cargo capability of the vehicle including Flight Support Equipment (FSE) and support structure.
Note 2: A packaging factor of 1.29 was assumed for the pressurized cargo and 2.0 for unpressurized cargo.
Extra Block 1A and 1B service propulsion system delta-V used for late ascent abort coverage.

Figure 1-17. Block Mass Summaries

1.4.2.1 Block 2 Lunar CEV

The lunar CEV CM, in conjunction with the SM and LV/EDS, is used to transport four crew members from Earth to lunar orbit and return them to Earth. The CM provides habitable volume for the crew, life support, docking and pressurized crew transfer to the LSAM, and atmospheric entry and landing capabilities. Upon return, a combination of parachutes and airbags provide for a nominal land touchdown with water flotation systems included for water landings following an aborted mission. Three main parachutes slow the CEV CM to a steady-state sink rate of 7.3 m/s (24 ft/s), and, prior to touchdown, the ablative aft heat shield is jettisoned and four Kevlar airbags are deployed for soft landing. After recovery, the CEV is refurbished and reflown with a lifetime up to 10 missions.

A scaled Apollo Command Module shape with a base diameter of 5.5 m and sidewall angle of 32.5 deg was selected for the OML of the CEV CM. This configuration provides 29.4 m^3 of pressurized volume and 12 to 15 m^3 of habitable volume for the crew during transits between Earth and the Moon. The CEV CM operates at a nominal internal pressure of 65.5 kPa (9.5 psia) with 30 percent oxygen composition for lunar missions, although the pressure vessel structure is designed for a maximum pressure of 101.3 kPa (14.7 psia). Operating at this higher pressure allows the CEV to transport crew to the ISS without the use of an intermediate airlock. For the lunar missions, the CM launches with a sea-level atmospheric pressure (101.3 kPa), and the cabin is depressurized to 65.5 kPa prior to docking with the LSAM.

The lunar CEV CM propulsion system provides vehicle attitude control for atmospheric entry following separation from the SM and range error corrections during the exoatmospheric portion of a lunar skip-entry return trajectory. A gaseous oxygen/ethanol bipropellant system is assumed with a total delta-V of 50 m/s.

Illustrations of the reference lunar CEV CM are shown in **Figure 1-18**.

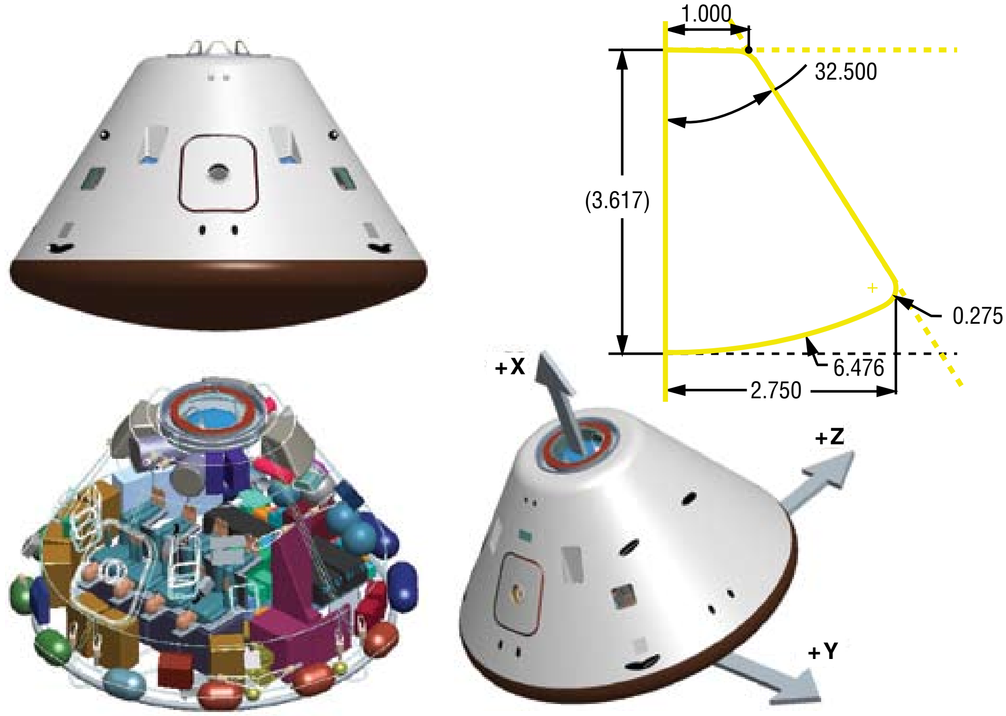


Figure 1-18. Reference Lunar CEV CM

1.4.2.2 Block 2 Lunar CEV SM

The lunar CEV SM is included in the ESAS exploration architecture to provide major translational maneuvering capability, power generation, and heat rejection for the CEV CM. The SM assumes an integrated pressure-fed oxygen/methane service propulsion system and RCS to perform rendezvous and docking with the LSAM in Earth orbit, any contingency plane changes needed prior to lunar ascent, TEI, and self-disposal following separation from the CM. One 66.7-kN (15,000-lbf) service propulsion system and twenty-four 445-N (100-lbf) RCS thrusters, engines common to both the SM and the LSAM ascent stage, are used for on-orbit maneuvering. The SM propellant tanks are sized to perform up to 1,724 m/s of service propulsion system and 50 m/s of RCS delta-V with the CEV CM attached and 15 m/s of RCS delta-V after separation. In the event of a late ascent abort off the CLV, the SM service propulsion system may also be used for separating from the LV and either aborting to near-coastline water landings or aborting to orbit.

Two deployable, single-axis gimbaling solar arrays are also included to generate the necessary CEV power from Earth-Orbit Insertion (EOI) to CM–SM separation prior to entry. For long-duration outpost missions to the lunar surface, lasting up to 180 days, the CEV remains unoccupied in lunar orbit. Solar arrays were selected instead of fuel cells or other similar power generation options because the reactant mass requirements associated with providing keep-alive power during the long dormant period for fuel cells became significantly higher than the mass of a nonconsumable system such as solar arrays. The solar arrays use state-of-the-art, three-junction, photovoltaic cells. Finally, the SM composite primary structure also provides a mounting location for four radiator panels. These panels provide heat rejection capability for the CEV fluid loop heat acquisition system.

Illustrations of the reference lunar CEV SM are shown in **Figure 1-19**.

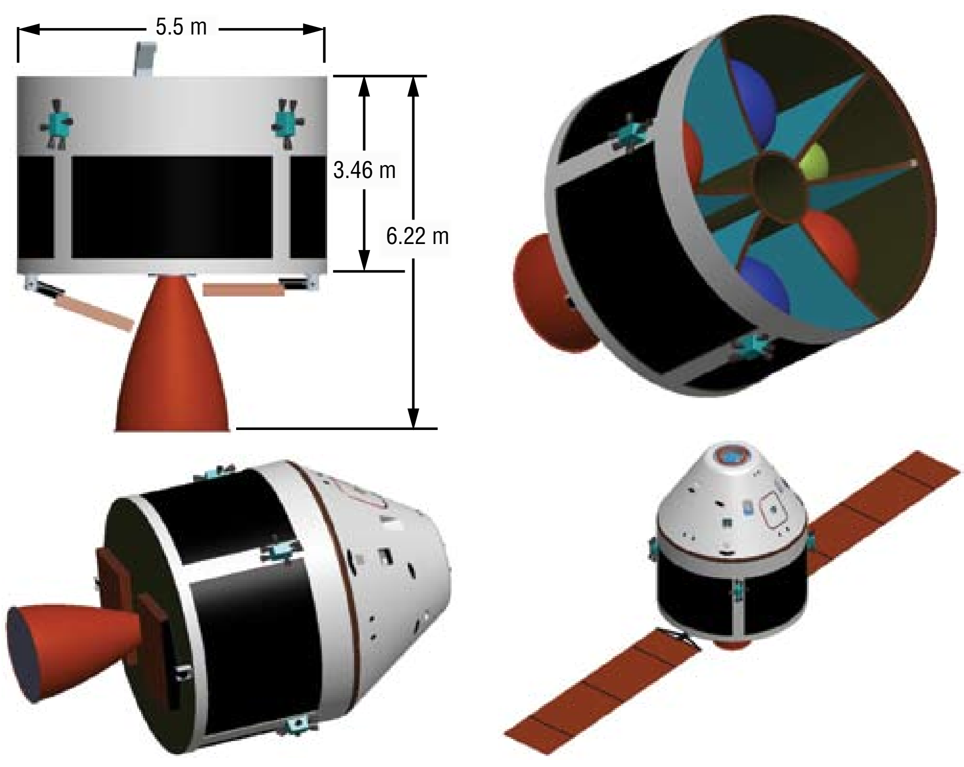


Figure 1-19. Reference Lunar CEV SM

1.4.2.3 Block 2 LAS

The LAS was sized to pull the CEV CM away from a thrusting LV at 10 g's acceleration. The LAS sizing concept is similar to the Apollo Launch Escape System (LES) in that it is a tractor system that is mounted ahead of the CM. The main difference is that the exhaust nozzles are located near the top of the motor, which will reduce the impingement loads on the CM.

The LAS features an active trajectory control system based on solid propellant, a solid rocket escape motor, forward recessed exhaust nozzles, and a CM adaptor. The motor measures 76 cm in diameter and 5.5 m in length, while eight canted thrusters aid in eliminating plume impingement on the CM. A star fuel grain minimizes motor size and redundant igniters are intended to guarantee the system's start.

The LAS provides abort from the launch pad and throughout powered flight of the booster first stage. The LAS is jettisoned approximately 20 to 30 sec after second-stage ignition. Further analyses are required to determine the optimum point in the trajectory for LAS jettison. After the LAS is jettisoned, launch aborts for the crew are provided by the SM propulsion system.

The mass for a 10-g LAS for a 21.4-mT CM is 4.2 mT. **Figure 1-20** depicts the LAS on top of the CM.

Figure 1-20. CEV with LAS

1.4.2.4 Block 1A ISS CEV CM and SM

The ISS CEV CM in the ESAS architecture is the Block 1 variant of the lunar CM designed to rotate three to six crew members and cargo to the ISS. The ISS CM is designed largely to support lunar exploration requirements, with a minimal set of modifications made to support ISS crew rotation. Initial mass for the three-crew ISS CM variant is 162 kg less than the lunar CM mass.

The ISS SM is identical to the SM designed for lunar exploration, except that propellant is off-loaded to reflect the lower delta-V requirements of ISS crew rotation compared to LOR. Propellant requirements for the ISS SM are estimated based on using the largest vehicle the SM may deliver to the ISS and subsequently deorbit, which is currently the unpressurized CDV. Other potential ISS payloads for the SM are the crewed CEV CM and pressurized cargo CEV; however, these have total masses less than the unpressurized CDV. The CDV has a total mass of 12,200 kg, compared to 9,342 kg for the three-crew CEV, 9,551 kg for the six-crew CEV, and 11,381 kg for the pressurized cargo delivery CEV.

1.4.2.5 Block 1B ISS Pressurized Cargo CM Variant

The ESAS architecture also includes a variant of the ISS CEV CM that may be used to deliver several tons of pressurized cargo to the ISS without crew on board and return an equivalent mass of cargo to a safe Earth landing. This spacecraft is nearly identical to the ISS crew rotation variant, with the exception that the personnel and most components associated with providing crew accommodations are removed and replaced with cargo. Initial mass for the uncrewed ISS CM variant is 2,039 kg greater than the three-crew ISS crew rotation CM.

1.4.2.6 ISS Unpressurized CDV

The ISS CDV was sized to deliver unpressurized cargo to the ISS. The CDV is mainly a structural “strong back” with a CBM for attachment to the ISS. The CDV utilizes the same SM as the other block configurations for transfer from the LV injection orbit to the ISS. Because the avionics for the other CEV variants are located within the CM, an avionics pallet is required for the CDV. This pallet would support the avionics and provide the connection to the ATCS on the SM.

The CDV was sized to transport two 1,500-kg unpressurized ORUs for the ISS. Examples of ORUs include Control Moment Gyroscopes (CMGs) and pump packages. The packaging factor for these ORUs was assumed to be 100 percent; therefore, the trays and secondary support structure for the cargo is estimated to be 3,000 kg, for a total cargo complement of 6,000 kg. The total estimate for the CDV without the SM is 12,200 kg.

Operationally, the CDV would perform automated rendezvous and proximity operations with the ISS and would then be grappled by the SSRMS and berthed to an available port. Two releasable cargo pallets are used to provide structural attachment for the ORUs. The cargo pallets can be grappled by the SSRMS and relocated to the ISS truss as required. Once the cargo has been relocated on the ISS, the CDV would depart from the ISS and perform an automated deorbit burn for burnup and disposal in the ocean.

Illustrations of the reference CDV are shown in **Figures 1-21** and **1-22**.

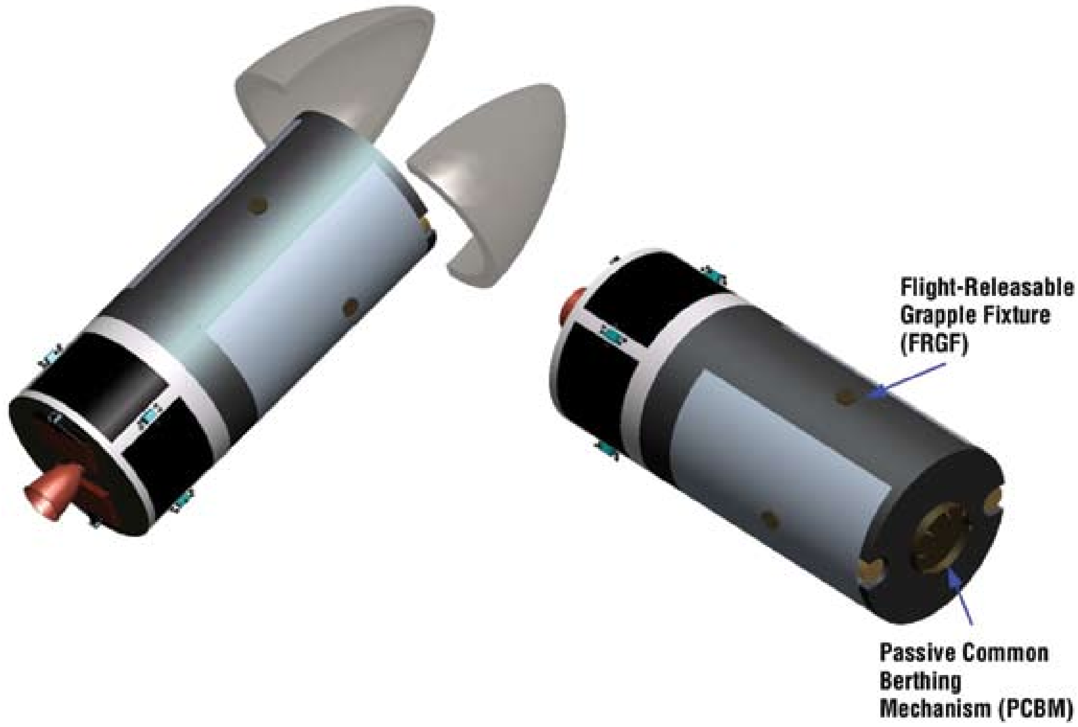


Figure 1-21. CDV

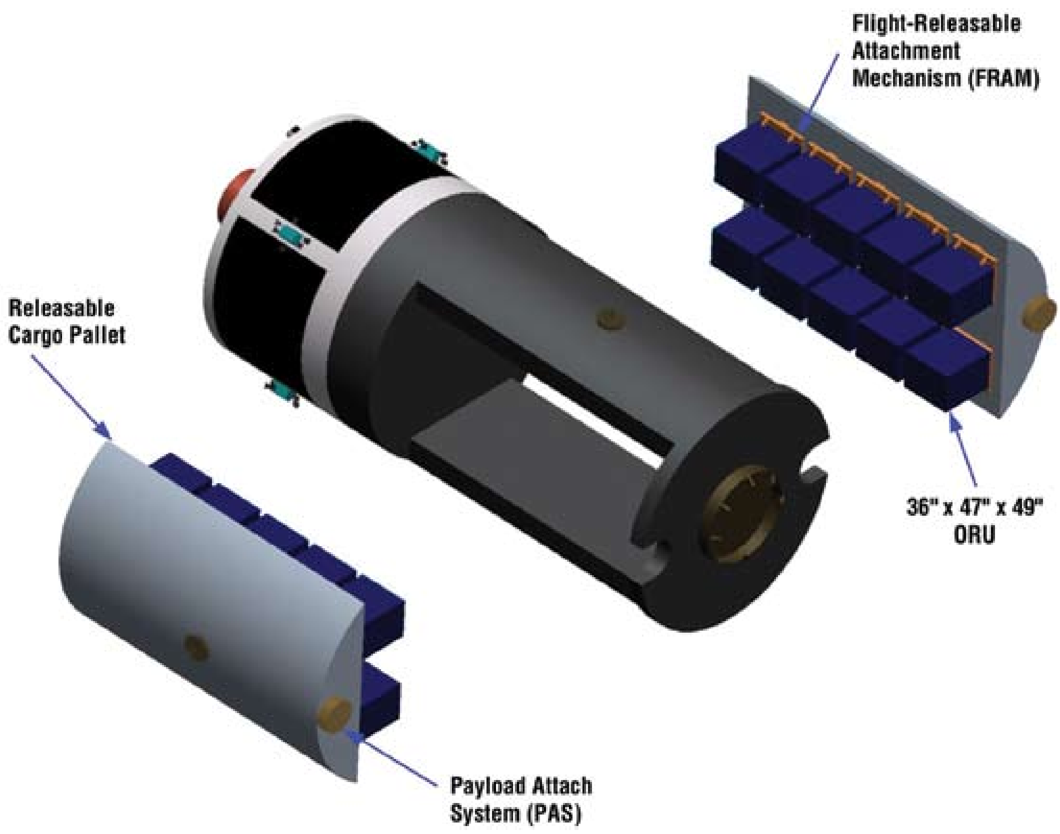


Figure 1-22. CDV Cargo Pallets

1.4.2.7 Block 3 Mars CEV Variant

The ESAS reference Mars mission utilizes a Block 3 CEV to transfer a crew of six between Earth and an MTV at the beginning and end of the Mars exploration mission. A Block 3 CEV CM and SM is launched by the CLV into an orbit matching the inclination of the awaiting MTV. The CEV is first injected into a 55- x 296-km altitude orbit while the MTV loiters in a circular orbit of 800- to 1,200-km altitude. It then takes the CEV up to 2 days to perform orbit-raising maneuvers to close on the MTV, conducting a standard ISS-type rendezvous and docking approach to the MTV. After docking, the CEV crew performs a leak check, equalizes pressure with the MTV, and opens hatches. Once crew and cargo transfer activities are complete, the CEV is configured to a quiescent state and remains docked to the MTV for the trip to Mars and back. Periodic systems health checks and monitoring are performed by the ground and flight crew throughout the mission.

As the MTV approaches Earth upon completion of the 1.5- to 2.5-year round-trip mission, the crew performs a pre-undock health check of all entry critical systems, transfers to the CEV, closes hatches, performs leak checks, and undocks from the MTV. The CEV departs 24 to 48 hours prior to Earth entry, and the MTV then either performs a diversion maneuver to fly by Earth or recaptures into Earth orbit. After undocking, the CEV conducts an onboard-targeted, ground-validated burn to target for the proper entry corridor, and, as entry approaches, the CEV CM maneuvers to the proper Entry Interface (EI) attitude for a direct-guided entry to the landing site. Earth entry speeds from a nominal Mars return trajectory may be as high as 14 km/s, compared to 11 km/s for the Block 2 CEV. The CEV performs a nominal landing at the primary land-based landing site and the crew and vehicle are recovered.

Figure 1-23 shows the Block 3 CEV CM configured to carry six crew members to the MTV.

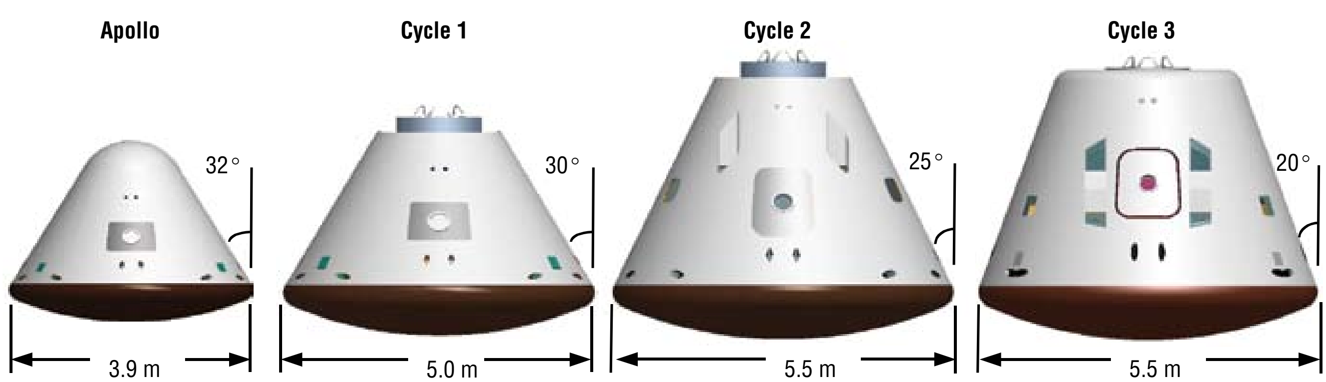
Figure 1-23. Block 3 CEV CM

1.4.3 CEV Design Evolution

The design and shape of the CEV CM evolved in four design cycles throughout the study, beginning with an Apollo-derivative configuration 5 m in diameter and a sidewall angle of 30 deg. This configuration provided an OML volume of 36.5 m^3 and a pressurized volume of 22.3 m^3. The CM also included 5 g/cm^2 of supplemental radiation protection on the cabin walls for the crew's protection. Layouts for a crew of six and the associated equipment and stowage were very constrained and left very little habitable volume for the crew.

A larger CEV was considered in Cycle 2 which grew the outer diameter to 5.5 m and reduced the sidewall angles to 25 deg. Both of these changes substantially increased the internal volume. The pressurized volume increased by 75 percent to 39.0 m^3 and the net habitable volume increased by over 50 percent to 19.4 m^3. The desire in this design cycle was to provide enough interior volume for the crew to be able to stand up in and don/doff lunar EVA suits for the surface direct mission. Most of the system design parameters stayed the same for this cycle including the 5 g/cm^2 of supplemental radiation protection.

Cycle 3 reduced the sidewall angles even further to 20 deg in an effort to achieve monostability on Earth entry. The sidewall angle increased the volume further. Because the increases in volume were also increasing the vehicle mass, the height of the vehicle was reduced by 17 inches, reducing the height-to-width aspect ratio. This configuration showed the most promise in the quest for monostability, but the proper Center of Gravity (CG) was still not achieved. Analysis in this design cycle showed that the supplemental radiation protection could be reduced to 2 g/cm^2. **Figure 1-24** illustrates the progression of the configurations through Cycle 3 of the study as compared to Apollo and the attached table details the changes in diameter, sidewall angle, and volume. Data for Cycle 4 is also shown and is described in the following paragraphs.



Configuration	Diameter (m)	Sidewall Angle (deg)	OML Volume (m^3)	Pressurized Volume (m^3)
Apollo	3.9	32.5	15.8	10.4
Cycle 1 (EIRA)	5.0	30.0	36.5	22.3
Cycle 2	5.5	25.0	56.7	39.0
Cycle 3	5.5	20.0	63.6	39.5
Cycle 4	5.5	32.5	45.9	30.6

Figure 1-24. CEV CM Sizing Progression

Cycle 4 was the final CEV design cycle and began after the decision was made to no longer consider the lunar surface direct mission. The design implications to the CEV (i.e., difficulty including an airlock and complex operations), and the low mass margins surrounding the lunar surface direct mission mode were the primary reasons for taking the mode out of consideration. The Cycle 4 CEV was sized for a 2-launch EOR–LOR mission mode where the CEV performs a rendezvous with the EDS and LSAM in LEO, stays in lunar orbit while the LSAM descends to the lunar surface, and performs another rendezvous with the LSAM in lunar orbit. No supplemental radiation protection was included in the mass estimates for this design analysis due to results from a radiation study reported in **Section 4**, **Lunar Architecture**.

The resulting Cycle 4 CM shape is a geometric scaling of the Apollo Command Module. The vehicle is 5.5 m in diameter and the CM has a sidewall angle of 32.5 deg. The resulting CM pressurized volume is approximately 25 percent less than the Cycle 3 volume, but has almost three times the internal volume as compared to the Apollo Command Module. The CEV was ultimately designed for the EOR–LOR "1.5-launch solution" and volume reduction helps to reduce mass to that required for the mission. **Figure 1-25** depicts how vehicle sidewall angle and diameter affect pressurized volume and the resulting design point for each cycle.

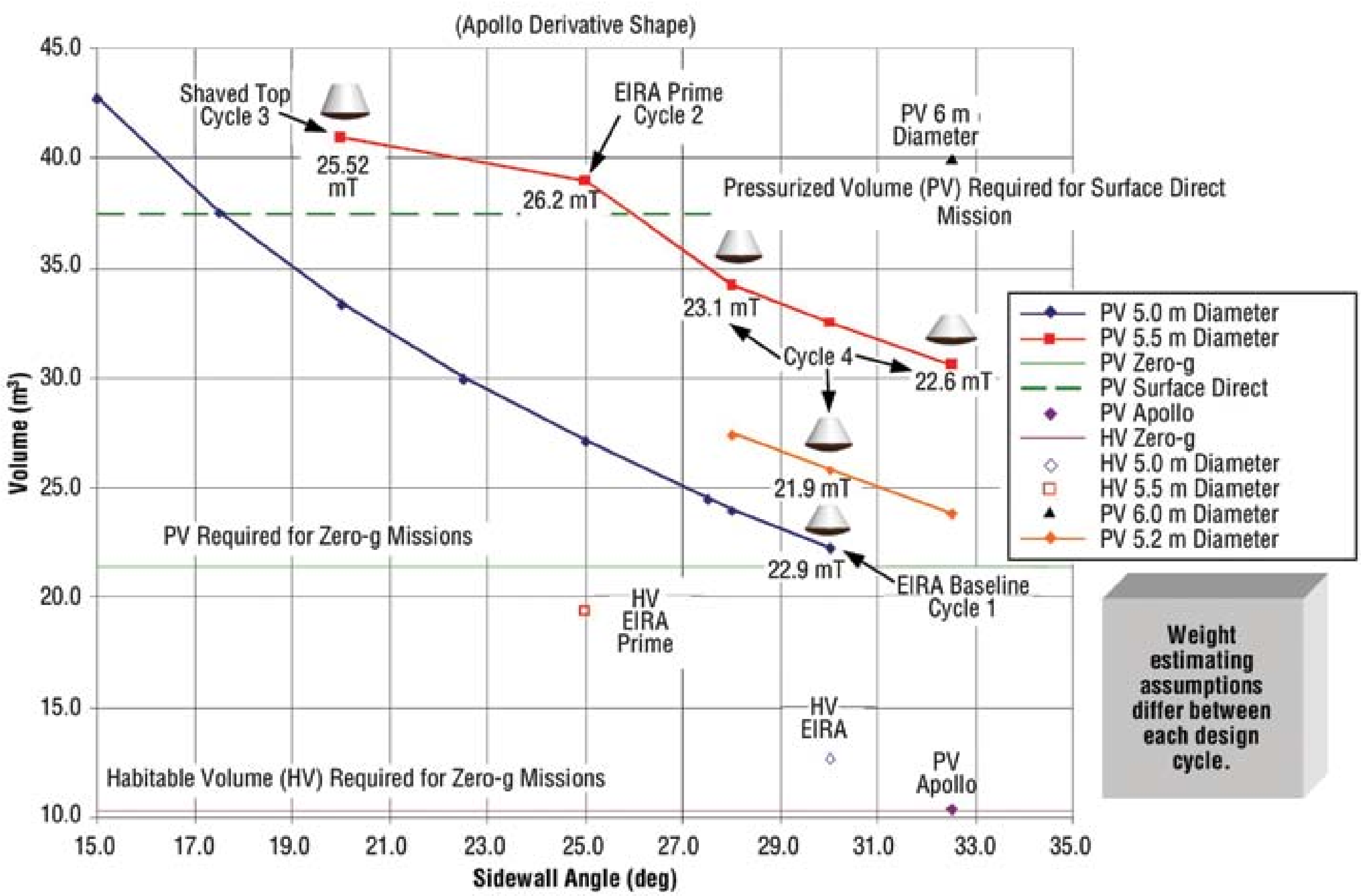


Figure 1-25. CEV Volume Relationships

1.4.4 CEV Recommendations

It is recommended that the CEV incorporate a separate CM, SM, and LAS arrangement similar to that of Apollo, and that these modules be capable of multiple functions to save costs. The CEV design was sized for lunar missions carrying a crew of four. Also, the vehicle was designed to be reconfigurable to accommodate up to six crew for ISS and future Mars mission scenarios. The CEV can transfer and return crew and cargo to the ISS and stay for 6 months in a quiescent state for nominal end of mission, or return of crew at any time in the event of an emergency. The lunar CEV design has direct applications to ISS missions without significant changes in the vehicle design. The lunar and ISS configurations share the same SM, but the ISS mission has much lower delta-V requirements. Hence, the SM propellant tanks can be loaded with additional propellant for ISS missions to provide benefits in launch aborts, on-orbit phasing, and ISS reboost. Other vehicle block derivatives can deliver pressurized and unpressurized cargo to the ISS.

The ESAS team's next recommendation addresses the vehicle shape. Using an improved blunt-body capsule for the CM was found to be the least costly, fastest, and safest approach for bringing ISS and lunar missions to reality. The key benefits for a blunt-body configuration were found to be lower weight, a more familiar aerodynamic design from human and robotic heritage (resulting in less design time and cost), acceptable ascent and entry abort load levels, crew seating orientation ideal for all loading events, easier LV integration, and improved entry controllability during off-nominal conditions. Improvements on the Apollo shape will offer better operational attributes, especially by increasing the Lift-to-Drag (L/D) ratio, improving CG placement, creating a more stable configuration, and employing a lower angle of attack for reduced sidewall heating.

A CM measuring 5.5 m in diameter was chosen to support the layout of six crew without stacking the crew members above or below each other. A crew tasking analysis also confirmed the feasibility of the selected vehicle volume. The recommended pressurized volume for the CM is approximately three times that of the Apollo Command Module. The available internal volume provides flexibility for future missions without the need for developing an expendable mission module. The vehicle scaling also considered the performance of the proposed CLV, which is a four-segment SRB with a single Space Shuttle Main Engine (SSME) upper stage. The CEV was scaled to maximize vehicle size while maintaining adequate performance margins on the CLV.

It is recommended that the CEV utilize an androgynous Low-Impact Docking System (LIDS) to mate with other exploration elements and the ISS. This requires the CEV-to-ISS docking adapters to be LIDS-compatible. It is proposed to develop small adapters to convert ISS interfaces to LIDS. The exact implementation is the source of further study.

An integrated pressure-fed LOX/methane service propulsion system/RCS propulsion system is recommended for the SM. Selection of this propellant combination was based on performance and commonality with the ascent propulsion system on the LSAM. The risk associated with this type of propulsion for a lunar mission can be substantially reduced by developing the system early and flying it to the ISS. There is high risk in developing a LOX/methane propulsion system by 2011, but development schedules for this type of propulsion system have been studied and are in the range of hypergolic systems.

Studies were performed on the levels of radiation protection required for the CEV CM. Based on an aluminum cabin surrounded by bulk insulation and composite skin panels with a Thermal Protection System (TPS), no supplemental radiation protection is recommended.

Solar arrays combined with rechargeable batteries were recommended for the SM due to the long mission durations dictated by some of the DRMs. The ISS crew transfer mission and long-stay lunar outpost mission require the CEV to be on orbit for 6 to 9 months, which is problematic for fuel cell reactants.

The choice of a primary land landing mode was primarily driven by a desire for land landing in the continental United States (CONUS) for ease and minimal cost of recovery, post-landing safety, and reusability of the spacecraft. However, it is recommended that the design of the CEV CM should incorporate both a water- and land-landing capability. Ascent aborts will require the ability to land in water, while other off-nominal conditions could lead the spacecraft to a land landing, even if not the primary intended mode. However, a vehicle designed for a primary land-landing mode can more easily be made into a primary water lander than the reverse situation. For these reasons, the study attempted to create a CONUS land-landing design from the outset, with the intention that a primary water lander would be a design off-ramp if the risk or development cost became too high.

In order for CEV entry trajectories from LEO and lunar return to use the same landing sites, it is recommended that NASA utilize skip-entry guidance on the lunar return trajectories. The skip-entry lunar return technique provides an approach for returning crew to a single CONUS landing site anytime during a lunar month. The Apollo-style direct-entry technique requires water or land recovery over a wide range of latitudes. The skip-entry includes an exoatmospheric correction maneuver at the apogee of the skip maneuver to remove dispersions accumulated during the skip maneuver. The flight profile is also standardized for all lunar return entry flights. Standardizing the entry flights permits targeting the same range-to-landing site trajectory for all return scenarios so that the crew and vehicle experience the same heating and loads during each flight. This does not include SM disposal considerations, which must be assessed on a case-by-case basis.

For emergencies, it is recommended that the CEV also include an LAS that will pull the CM away from the LV on the pad or during ascent. The LAS concept utilizes a 10-g tractor rocket attached to the front of the CM. The LAS is jettisoned from the launch stack shortly after second stage ignition. Launch aborts after LAS jettison are performed by using the SM propulsion system. Launch abort study results indicate a fairly robust abort capability for the CEV/CLV and a 51.6-deg-inclination ISS mission, given 1,200 m/s of delta-V and a Thrust-to-Weight (T/W) ratio of at least 0.25.

1.5 Launch Vehicles and Earth Departure Stages

1.5.1 Overview

A safe, reliable means of human access to space is required after the Space Shuttle is retired in 2010. As early as the mid-2010s, a heavy-lift cargo capability will be required, in addition to the crew launch capability to support manned lunar missions and follow-on missions to Mars. It is anticipated that robotic exploration beyond Earth orbit will have an annual manifest of five to eight spacecraft.

The ESAS team was chartered to develop and assess viable launch system configurations for a CLV and a CaLV to support lunar and Mars exploration and provide access to the ISS.

The ESAS team developed candidate LV concepts, assessed them against the ESAS FOMs (e.g., cost, reliability, safety, extensibility), identified and assessed vehicle subsystems and their allocated requirements, and developed viable development plans and supporting schedules to minimize the gap between Shuttle retirement and CEV IOC. The team developed LV concepts derived from elements of the existing EELV fleet and the Space Shuttle. A principal goal was to provide an LV capability to enable a CEV IOC in 2011.

Additional details on the LV trade studies and analysis results are contained in **Section 6, Launch Vehicles and Earth Departure Stages**.

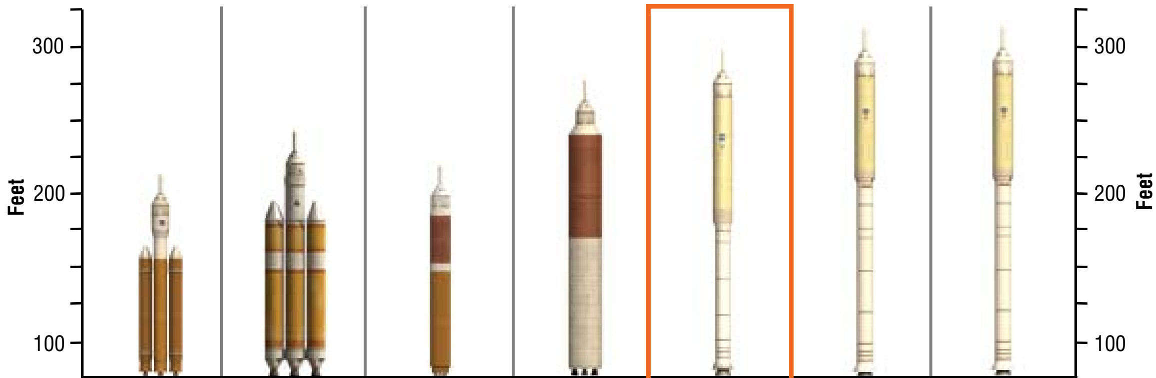
1.5.2 Crew Launch Vehicle (CLV)

1.5.2.1 Results of CLV Trade Studies

A summary of the most promising CLV candidates assessed and key parameters (cost is normalized to the selected option) is shown in **Figure 1-26**.

The EELV options examined for suitability for crew transport were those derived from the Delta IV and Atlas V families. The study focused on the heavy-lift versions of both Delta and Atlas families, as it became clear early in the study that none of the medium versions of either vehicle had the capability to accommodate CEV lift requirements. Augmentation of the medium-lift class systems with solid strap-on boosters does not provide adequate capability and poses an issue for crew safety regarding small strap-on Solid Rocket Motor (SRM) reliability, as determined by the Orbital Space Plane-ELV (OSP–ELV) Flight Safety Certification Study report, dated March 2004. Both vehicles were assessed to require modification for human-rating, particularly in the areas of avionics, telemetry, structures, and propulsion systems.

Both Atlas and Delta derived systems required new upper stages to meet the lift and human-rating requirements. Both Atlas and Delta single-engine upper stages fly highly lofted trajectories, which can produce high deceleration loads on the crew during an abort and, in some cases, can exceed crew load limits as defined by NASA STD 3000, Section 5. Depressing the trajectories flown by these vehicles will require additional stage thrust to bring peak altitudes down to levels that reduce crew loads enough to have sufficient margins for off-nominal conditions. Neither Atlas V or Delta IV with their existing upper stages possess the performance capability to support CEV missions to ISS, with shortfalls of 5 mT and 2.6 mT, respectively.



	Human-Rated Atlas V/New US	Human-Rated Delta IV/New US	Atlas Phase 2 (5.4-m Core)	Atlas Phase X (8-m Core)	4 Segment RSRB with 1 SSME	5 Segment RSRB with 1 J–2S	5 Segment RSRB with 4 LR–85
Payload (28.5°)	30 mT	28 mT	26 mT	70 mT	25 mT	26 mT	27 mT
Payload (51.6°)	27 mT	23 mT	25 mT	67 mT	23 mT	24 mT	25 mT
DDT&E*	1.18**	1.03	1.73**	2.36	1.00	1.3	1.39
Facilities Cost	.92	.92	.92	.92	1.00	1.00	1.00
Average Cost/Flight*	1.00	1.11	1.32	1.71	1.00	.96	.96
LOM (mean)	1 in 149	1 in 172	1 in 134	1 in 79	1 in 460	1 in 433	1 in 182
LOC (mean)	1 in 957	1 in 1,100	1 in 939	1 in 614	1 in 2,021	1 in 1,918	1 in 1,429

LOM: Loss of Mission   LOC: Loss of Crew   US: Upper Stage   RSRB: Reusable Solid Rocket Booster

* *All cost estimates include reserves (20% for DDT&E, 10% for Operations), Government oversight/full cost; Average cost/flight based on 6 launches per year.*

** *Assumes NASA has to fund the Americanization of the RD–180. Lockheed Martin is currently required to provide a co-production capability by the USAF.*

Figure 1-26. Comparison of Crew LEO Launch Systems

Another factor in both vehicles is the very low T/W ratio at liftoff, which limits the additional mass that can be added to improve performance. The RD–180 first-stage engine of the Atlas HLV will require modification to be certified for human-rating. This work will, by necessity, have to be performed by the Russians. The RS–68 engine powering the Delta IV HLV first stage will require modification to eliminate the buildup of hydrogen at the base of the vehicle immediately prior to launch. Assessments of new core stages to improve performance as an alternative to modifying and certifying the current core stages for human-rating revealed that any new core vehicle would be too expensive and exhibit an unacceptable development risk to meet the goal of the 2011 IOC for the CEV. Note the EELV costs shown in **Figure 1-26** do not include costs for terminating Shuttle propulsion elements/environmental cleanup. Finally, both the EELV options were deamed high risk for a 2011 IOC.

CLV options derived from Shuttle elements focused on the configurations that used a Reusable Solid Rocket Booster (RSRB), either as a four-segment version nearly identical to the RSRB flown today or a higher-performance five-segment version of the RSRB using Hydroxyl Terminated Polybutadiene (HTPB) as the solid fuel. New core vehicles with ET-derived first stages (without SRBs) similar to the new core options for EELV were briefly considered, but were judged to have the same limitations and risks and, therefore, were not pursued. To meet the CEV lift requirement, the team initially focused on five-segment RSRB-based solutions. Three classes of upper stage engine were assessed—SSME, a single J–2S+, and a four-engine cluster of a new expander cycle engine in the 85,000-lbf vacuum thrust class. However, the five-segment development added significant near-term cost and risk and the J–2S+/expander engine could not meet the 2011 schedule target. Therefore, the team sought to develop options that could meet the lift requirement using a four-segment RSRB. To achieve this, a 500,000-lbf vacuum thrust class propulsion system is required. Two types of upper stage engine were assessed—a two-engine J–2S cluster and a single SSME. The J–2S option could not meet the 2011 target (whereas the SSME could) and had 6 percent less performance than the SSME-based option (LV 13.1). The SSME option offered the added advantages of an extensive and successful flight history and direct extensibility to the CaLV with no gap between the current Shuttle program and exploration launch. Past studies have shown that the SSME can be air-started, with an appropriate development and test program.

The 13.1 configuration was selected due to its lower cost, higher safety/reliability, its ability to utilize existing human-rated systems and infrastructure and the fact that it gave the most straightforward path to a CaLV.

Figure 1-27. ESAS CLV Concept

1.5.2.2 Preferred CLV Configuration

The recommended CLV concept, shown in **Figure 1-27**, is derived from elements of the existing Space Shuttle system and designated as ESAS LV 13.1. It is a two-stage, series-burn configuration with the CEV positioned on the nose of the vehicle, capped by an LAS that weighs 9,300 lbm. The vehicle stands approximately 290 ft tall and weighs approximately 1.78M lbm at launch. LV 13.1 is capable of injecting a 24.5-mT payload into a 30-x-160 nmi orbit inclined 28.5 deg and injecting 22.9 mT into the same orbit inclined 51.6 deg.

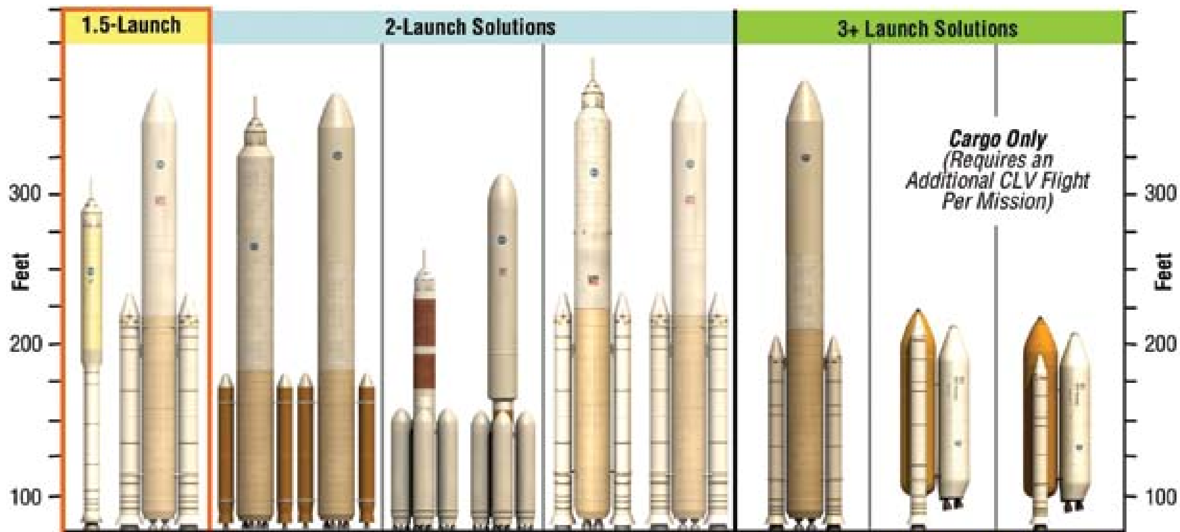
Stage 1 is derived from the Reusable Solid Rocket Motor (RSRM) and is composed of four field-assembled segments, an aft skirt containing the Thrust Vector Control (TVC) hydraulic system, accompanying Auxiliary Power Units (APUs), and Booster Separation Motors (BSMs). The aft skirt provides the structural attachment to the Mobile Launch Platform (MLP) through four attach points and explosive bolts. The single exhaust nozzle is semi-embedded and is movable by the TVC system to provide pitch and yaw control during first-stage ascent. The Space Transportation System (STS) forward skirt, frustrum, and nose cap are replaced by a stage adapter that houses the RSRB recovery system elements and a roll control system. Stage 1 is approximately 133 ft long and burns for 128 sec. After separation from the second stage, Stage 1 coasts upward in a ballistic arc to an altitude of approximately 250,000 ft, subsequently reentering the atmosphere and landing by parachute in the Atlantic Ocean for retrieval and reuse similar to the current Shuttle RSRB.

Stage 2 is approximately 105 ft long, 16.4 ft in diameter, and burns LOX and Liquid Hydrogen (LH2). It is composed of an interstage, single RS–25 engine, thrust structure, propellant tankage, and a forward skirt. Near the conclusion of the ESAS, the CEV concept increased in diameter from 5 m to 5.5 m. Subsequent to the ESAS, LV 13.1 adopted a 5.5-m diameter, 100-ft long upper stage to accomodate the CEV. The interstage provides the structural connection between the Stage 1 adapter and Stage 2, while providing clearance for the RS–25 exhaust nozzle. The RS–25 is an expendable version of the current SSME, modified to start at altitude. The thrust structure provides the framework to support the RS–25, the Stage 2 TVC system (for primary pitch and yaw during ascent), and an Auxiliary Propulsion System (APS) which provides three-axis attitude control (roll during ascent and roll, pitch, and yaw for CEV separation), along with posigrade thrust for propellant settling. The propellant tanks are cylindrical, composed of Aluminum-Lithium (AL-Li) with ellipsoid domes, and are configured with the LOX tank aft, separated by an intertank. The LH2 main feedline exits the OML of the intertank and follows the outer skin of the LOX tank, entering the thrust structure aft of the LOX tank. The forward skirt is connected to the LH2 tank at the cylinder/dome interface and acts as a payload adapter for the CEV. It is of sufficient length to house the forward LH2 dome, avionics, and the CEV SM engine exhaust nozzle. Stage 2 burns for approximately 332 sec placing the CEV in a 30- x 160-nmi orbit. After separation from the CEV, Stage 2 coasts approximately a three-quarter orbit and reenters, with debris falling in the Pacific Ocean.

1.5.3 Cargo Launch Vehicle

1.5.3.1 Results of CaLV Trade Studies

A summary of the most promising CaLV candidates and key parameters is shown in **Figure 1-28**. (Note: Cost is normalized to the selected option.) The requirement for four or less launches per mission results in a minimum payload lift class of 70 mT. To enable a 2- or 1.5-launch solution, a 100- or 125-mT class system, respectively, is required.



	5 Segment RSRB In-Line with 5 SSME Core - Cargo	Atlas Phase X (8-m Core)	Atlas Phase 3A (5.4-m Core)	5 Segment RSRB In-Line with 4 SSME Core	4 Segment RSRB In-Line with 3 SSME Core	5 Segment RSRB Side-mount with 3 SSME	4 Segment RSRB Side-mount with 3 SSME
Payload to 28.5	106 mT (125 mT w/ upperstage)	95 mT	94 mT	97 mT	74 mT	80 mT	67 mT
Lunar LV DDT&E*	1.00	1.29	.59	.96	.73	.80	.75
CLV+Lunar Crew/Cargo DDT&E (family op)*	1.00 (4 Seg RSRB w/1 SSME)	1.26 (Atlas V)	1.02 (Phase 2)	.98 (4 Seg RSRB w/1 SSME)	.83 (4 Seg RSRB w/1 SSME)	1.03 (5 Seg RSRB w/J2S-LR85)	.85 (4 Seg RSRB w/1 SSME)
Lunar LV Facility DDTE	1.00	1.33	2.25	1.00			
CLV+Lunar Facilities Cost	1.00	1.12	1.56	1.00			
Lunar LV Average Cost/Flight*	1.00	1.08	1.19	.87	.78	1.13**	1.13**
LOM – Cargo (mean)	1 in 124	1 in 71	1 in 88	1 in 133	1 in 176	1 in 172	1 in 173
LOC (mean)	1 in 2,021	1 in 536	1 in 612	1 in 915	1 in 1,170	N/A	N/A

* All cost estimates include reserves (20% for DDT&E, 10% for Operations), Government oversight/full cost; Average cost/flight based on 6 launches per year.
**Production costs are higher than in-line due to production of separate side-mount cargo carrier

Figure 1-28. Lunar Cargo Launch Comparison

EELV-derived options for the CaLV included those powered by RD–180 and RS–68 engines, with core vehicle diameters of 5.4 and 8 m. No RS–68-powered variant of an EELV-derived heavy-lift cargo vehicle demonstrated the capability to meet the lunar lift requirements without a new upperstage and either new large liquid strap-on boosters or Shuttle RSRBs. The considerable additional cost, complexity, and development risk were judged to be unfavorable, eliminating RS–68-powered CaLVs. Hydrocarbon cores powered by the RD–180 with RD–180 strap-on boosters proved to be more effective in delivering the desired LEO payload. Vehicles based on both a 5.4-m diameter core stage and an 8-m diameter core were analyzed. A limitation exhibited by the EELV-Derived Vehicles (EDVs) was the low liftoff T/W ratios for optimized cases. While the EELV-derived CaLVs were able to meet LEO payload requirements, the low liftoff T/W ratio restricted the size of EDS in the suborbital burn cases. As a result, the Earth-escape performance of the EELV options was restricted. The 5.4-m core CaLV had an advantage in DDT&E costs, mainly due to the use of a single diameter core derived from the CLV which was also used as a strap-on booster. However, the CLV costs for this option were unacceptably high. (See **Section 1.5.2.1**, **Results of the CLV Trade Studies**.) In addition, there would be a large impact to the launch infrastructure due to the configuration of the four strap-on boosters (added accommodations for the two additional boosters in the flame trench and launch pad). Also, no EELV-derived concept was determined to have the performance capability approaching that required for a lunar 1.5-launch solution. Finally, to meet performance requirements, all EELV-derived CaLV options required a dedicated LOX/LH2 upper stage in addition to the EDS—increasing cost and decreasing safety/reliability.

The Shuttle-derived options considered were of two configurations: (1) a vehicle configured much like today's Shuttle, with the Orbiter replaced by a side-mounted expendable cargo carrier, and (2) an in-line configuration using an ET-diameter core stage with a reconfigured thrust structure on the aft end of the core and a payload shroud on the forward end. The ogive-shaped ET LOX tank is replaced by a conventional cylindrical tank with ellipsoidal domes, forward of which the payload shroud is attached. In both configurations, three SSMEs were initially baselined. Several variants of these vehicles were examined. Four- and five-segment RSRBs were evaluated on both configurations, and the side-mounted version was evaluated with two RS–68 engines in place of the SSMEs. The J–2S+ was not considered for use in the CaLV core due to its low relative thrust and the inability of the J–2S+ to use the extended nozzle at sea level, reducing its Specific Impulse (Isp) performance below the level required. No variant of the side-mount Shuttle-Derived Vehicle (SDV) was found to meet the lunar lift requirements with less than four launches. The side-mount configuration would also most likely prove to be very difficult to human-rate, with the placement of the CEV in close proximity to the main propellant tankage, coupled with a restricted CEV abort path as compared to an in-line configuration. The proximity to the ET also exposes the CEV to ET debris during ascent, with the possibility of contact with the leeward side TPS, boost protective cover, and the LAS. The DDT&E costs are lower than the in-line configurations, but per-flight costs are higher—resulting in a higher per-mission cost. The side-mount configuration was judged to be unsuitable for upgrading to a Mars mission LEO capability (100 to 125 mT). The in-line configuration in its basic form (four-segment RSRB/three-SSME) demonstrated the performance required for a three-launch lunar mission at a lower DDT&E and per-flight costs. Upgrading the configuration with five-segment RSRBs and four SSMEs in a stretched core with approximately one-third more propellant enables a 2-launch solution for lunar missions, greatly improving mission reliability. A final variation of the Shuttle-derived in-line CaLV was considered. This concept added a fifth SSME to the LV core, increasing its T/W ratio at

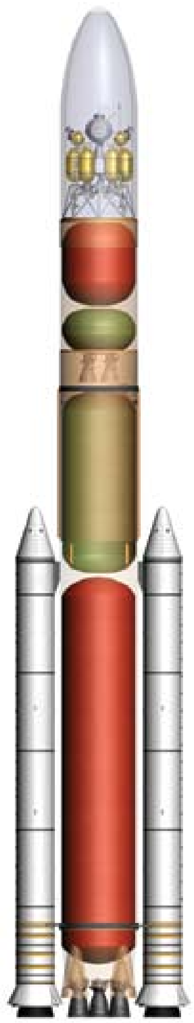


Figure 1-29. ESAS CaLV Concept

liftoff, thus increasing its ability to carry large, suborbitally ignited EDSs. LV 27.3 demonstrated an increased lift performance to enable a 1.5-launch solution for lunar missions, launching the CEV on the CLV and the LSAM and EDS on the larger CaLV. This approach allows the crew to ride to orbit on the safer CLV with similar LCCs and was selected as the reference. This configuration proved to have the highest LEO performance and lowest LV family non-recurring costs. When coupled with the four-segment RSRB/SSME-derived CLV (13.1), LOM and LOC probabilities are lower than its EELV-derived counterparts.

1.5.3.2 Preferred CaLV Configuration

The ESAS LV 27.3 heavy-lift CaLV, shown in **Figure 1-29**, is recommended to provide the cargo lift capability for lunar missions. It is approximately 357.5 ft tall and is configured as a stage-and-a-half vehicle composed of two five-segment RSRMs and a large central LOX-/LH2-powered core vehicle utilizing five RS–25 SSMEs. It has a gross liftoff mass of approximately 6.4M lbm and is capable of delivering 54.6 mT to TLI (one launch) or 124.6 mT to 30- x 160-nmi orbit inclined 28.5 deg.

Each five-segment RSRB is approximately 210 ft in length and contains approximately 1.43M lbm of HTPB propellant. It is configured similarly to the current RSRB, with the addition of a center segment. The operation of the five-segment RSRBs is much the same as the STS RSRBs. They are ignited at launch, with the five RS–25s on the core stage. The five-segment RSRBs burn for 132.5 sec, then separate from the core vehicle and coast to an apogee of approximately 240,000 ft. They are recovered by parachute and retrieved from the Atlantic Ocean for reuse.

The core stage carries 2.2M lbm of LOX and LH2, approximately 38 percent more propellant than the current Shuttle ET, and has the same 27.5-ft diameter as the ET. It is composed of an aft-mounted boattail which houses a thrust structure with five RS–25 engines and their associated TVC systems. The RS–25 engines are arranged with a center engine and four circumferentially mounted engines positioned 45 deg from the vertical and horizontal axes of the core to provide sufficient clearance for the RSRBs. The propellant tankage is configured with the LOX tank forward. Both the LOX and LH2 tanks are composed of Aluminum-Lithium (AL-Li) and are cylindrical, with ellipsoidal domes. The tanks are separated by an intertank structure, and an interstage connects the EDS with the LH2 tank. The core is ignited at liftoff and burns for approximately 408 sec, placing the EDS and LSAM into a suborbital trajectory. A shroud covers the LSAM during the RSRB and core stage phases of flight and is jettisoned when the core stage separates. After separation from the EDS, the core stage continues on a ballistic suborbital trajectory and reenters the atmosphere, with debris falling in the South Pacific Ocean.

1.5.4 Preferred EDS Configuration

The recommended configuration for the EDS, shown in **Figure 1-30**, is the ESAS S2B3 concept, which is 27.5 ft in diameter, 74.6 ft long, and weighs approximately 501,000 lbm at launch. The EDS provides the final impulse into LEO, circularizes itself and the LSAM into the 160-nmi assembly orbit, and provides the impulse to accelerate the CEV and LSAM to escape velocity. It is a conventional stage structure, containing two J–2S+ engines, a thrust structure/boattail housing the engines, TVC, auxiliary propulsion system, and other stage subsystems. It is configured with an aft LOX tank, which is comprised primarily of forward and aft domes. The LH2 tank is 27.5 ft in diameter, cylindrical with forward and aft ellipsoidal domes, and is connected to the LOX tank by an intertank structure. Both tanks are composed of AL-Li. A forward skirt on the LH2 tank provides the attach structure for the LSAM and payload shroud. The EDS is ignited suborbitally, after core stage separation and burns for 218 sec to place the EDS/LSAM into a 30- x 160-nmi orbit inclined 28.5 deg. It circularizes the orbit to 160 nmi, where the CEV docks with the LSAM. The EDS then reignites for 154 sec in a TLI to propel the CEV and LSAM on a trans-lunar trajectory. After separation of the CEV/LSAM, the EDS is placed in a disposal solar orbit by the APS.

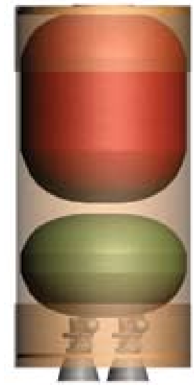


Figure 1-30. ESAS EDS Concept

1.5.5 LV and EDS Recommendations

1.5.5.1 Recommendation 1

Adopt and pursue a Shuttle-derived architecture as the next-generation launch system for crewed flights into LEO and for 125-mT-class cargo flights for exploration beyond Earth orbit. After thorough analysis of multiple EELV- and Shuttle-derived options for crew and cargo transportation, Shuttle-derived options were found to have significant advantages with respect to cost, schedule, safety, and reliability. Overall, the Shuttle-derived option was found to be the most affordable by leveraging proven vehicle and infrastructure elements and using those common elements in the heavy-lift CaLV as well as the CLV. Using elements that have a human-rated heritage, the CaLV can enable unprecedented mission flexibility and options by allowing a crew to potentially fly either on the CLV or CaLV for 1.5-launch or 2-launch lunar missions that allow for heavier masses to the lunar surface. The Shuttle-derived CLV provides lift capability with sufficient margin to accommodate CEV crew and cargo variant flights to ISS and potentially provides added services, such as station reboost.

The extensive flight and test databases of the RSRB and SSME give a solid foundation of well-understood main propulsion elements on which to anchor next-generation vehicle development and operation. The Shuttle-derived option allows the Nation to leverage extensive ground infrastructure investments and maintains access to solid propellant at current levels. Furthermore, the Shuttle-derived option displayed more versatile and straightforward growth paths to higher lift capability with fewer vehicle elements than other options.

The following specific recommendations are offered for LV development and utilization.

1.5.5.2 Recommendation 2

Initiate immediate development of a CLV utilizing a single four-segment RSRB first stage and a new upper stage using a single SSME. The reference configuration, designated LV 13.1 in this study, provides the payload capability to deliver a lunar CEV to low-inclination Earth orbits required by the exploration architectures and to deliver CEVs configured for crew and cargo transfer missions to the ISS. The existence and extensive operational history of human-rated Shuttle-derived elements reduce safety, risk, and programmatic and technical risk to enable the most credible development path to meet the goal of providing crewed access to space by 2011. The series-burn configuration of LV 13.1 provides the crew with an unobstructed escape path from the vehicle using an LAS in the event of a contingency event from launch through EOI. Finally, if required, a derivative cargo-only version of the CLV, designated in this report as LV 13.1S, can enable autonomous, reliable delivery of unpressurized cargo to ISS of the same payload class that the Shuttle presently provides.

1.5.5.3 Recommendation 3

To meet lunar and Mars exploration cargo requirements, begin development as soon as practical of an in-line Shuttle-derived CaLV configuration consisting of two five-segment RSRBs and a core vehicle with five aft-mounted SSMEs derived from the present ET and reconfigured to fly payload within a large forward-mounted aerodynamic shroud. The specific configuration is designated LV 27.3 in this report. This configuration provides superior performance to any side-mount Shuttle-derived concept and enables varied configuration options as the need arises. A crewed version is also potentially viable because of the extensive use of human-rated elements and in-line configuration. The five-engine core and two-engine EDS provides sufficient capability to enable the “1.5-launch solution,” which requires one CLV and one CaLV flight per lunar mission—thus reducing the cost and increasing the safety/reliability of each mission. The added lift capability of the five-SSME core allows the use of a variety of upper stage configurations, with 125 mT of lift capability to LEO. LV 27.3 will require design, development, and certification of a five-segment RSRB and new core vehicle, but such efforts are facilitated by their historical heritage in flight-proven and well-characterized hardware. Full-scale design and development should begin as soon as possible synchronized with CLV development to facilitate the first crewed lunar exploration missions in the middle of the next decade.

1.5.5.4 Recommendation 4

To enable the 1.5-launch solution and potential vehicle growth paths as previously discussed, NASA should undertake development of an EDS based on the same tank diameter as the cargo vehicle core. The specific configuration should be a suitable variant of the EDS concepts designated in this study as EDS S2x, depending on the further definition of the CEV and LSAM. Using common manufacturing facilities with the Shuttle-derived CaLV core stage will enable lower costs. The recommended EDS thrust requirements will require development of the J–2S+, which is a derivative of the J–2 upper stage engine used in the Apollo/Saturn program, or another in-space high performance engine/cluster as future trades indicate. As with the Shuttle-derived elements, the design heritage of previously flight-proven hardware will be used to advantage with the J–2S+. The TLI capability of the EDS S2x is approximately 65 mT, when used in the 1.5-launch solution mode, and enables many of the CEV/LSAM concepts under consideration. In a single-launch mode, the S2B3 variant can deliver 54.6 mT to TLI, which slightly exceeds the TLI mass of Apollo 17, the last crewed mission to the Moon in 1972.

1.5.5.5 Recommendation 5

Continue to rely on the EELV fleet for scientific and ISS cargo missions in the 5- to 20-mT lift range.

1.6 Technology Assessment

1.6.1 Overview

The Vision for Space Exploration set forth by President Bush cannot be realized without a significant investment in a wide range of technologies. Thus, a primary objective of the ESAS was to identify key technologies required to enable and significantly enhance the reference exploration systems and to prioritize near- and far-term technology investments. The product of this technology assessment is a revised ESMD technology investment plan that is traceable to the ESAS architecture and was developed by a rigorous and objective analytical process. The investment recommendations include budget, schedule, and Center/program allocations to develop the technologies required for the exploration architecture.

The three major technology assessment tasks were: (1) to identify what technologies are truly needed and when they need to be available to support the development projects; (2) to develop and implement a rigorous and objective technology prioritization/planning process; and (3) to develop ESMD Research and Technology (R&T) investment recommendations about which existing projects should continue and which new projects should be established.

Additional details on the technology trade studies and analysis results are contained in **Section 9**, **Technology Assessment**, of this report.

1.6.2 Technology Assessment Process

The baseline ESAS technology program was developed through a rigorous and objective process consisting of the following: (1) the identification of architecture functional needs; (2) the collection, synthesis, integration, and mapping of technology data; and (3) an objective decision analysis resulting in a detailed technology development investment plan. The investment recommendations include budget, schedule, and Center/program allocations to develop the technologies required for the exploration architecture, as well as the identification of other investment opportunities to maximize performance and flexibility while minimizing cost and risk. More details of this process are provided in **Appendix 9A**, **Process**.

The ESAS team's technology assessment included an Agency-wide Expert Assessment Panel (EAP). The team was responsible for assessing functional needs based on the ESAS architecture, assembling technology data sheets for technology project(s) that could meet these needs, and providing an initial prioritization of each technology project's contribution to meeting a functional need. This involved key personnel working full-time on ESAS as well as contractor support and consultation with technology specialists across NASA, as needed.

The EAP was a carefully balanced panel of senior technology and systems experts from eight NASA Centers. They examined the functional needs and technology data sheets for missing or incorrect entries, constructed new technology development strategies, and performed technology development prioritization assessment using the ESAS FOMs for each need at the architecture level. They provided internal checks and balances to ensure evenhanded treatment of sensitive issues.

All results were then entered into spreadsheet tools for use by the ESAS team in analyzing technology investment portfolio options. During the final step of the process, the ESAS team also worked with ESMD and the NASA Administrator's office to try to minimize Center workforce imbalance.

1.6.3 Architecture R&T Needs

This assessment was performed in parallel with the architecture development, requiring the ESAS team to coordinate closely to ensure that the technology assessment captured the latest architecture functional needs. The functional needs were traced element-by-element, for each mission, in an extensive spreadsheet tool. These needs were the basis for the creation of the technology development plans used in the assessment. Thus, all technology development recommendations were directly traceable to the architecture. This analysis indicated that R&T development projects are needed in the following areas:

- Structures and Materials,
- Protection,
- Propulsion,
- Power,
- Thermal Controls,
- Avionics and Software,
- Environmental Control and Life Support (ECLS),
- Crew Support and Accommodations,
- Mechanisms,
- ISRU,
- Analysis and Integration, and
- Operations.

These areas are described in additional detail in **Section 9**, **Technology Assessment**, of this report. Each area's section contains the description of its functional needs, the gaps between state-of-the-art and the needs, and the recommended developments. There is a more detailed write-up for each recommended technology development project listed in **Appendix 9B**, **Technology Development Activity Summaries**.

1.6.4 Recommendations

As a result of the technology assessment, it is recommended that the overall funding of ESMD for R&T be reduced by approximately 50 percent to provide sufficient funds to accelerate the development of the CEV to reduce the gap in U.S. human spaceflight after Shuttle retirement. This can be achieved by focusing the technology program only on those technologies required to enable the architecture elements as they are needed and because the recommended ESAS architecture does not require a significant level of technology development to accomplish the required near-term missions. Prior to the ESAS, the technology development funding profile for ESMD was as shown in **Figure 1-31**. The ESAS recommendations for revised, architecture-driven technology development is shown in **Figure 1-32**.

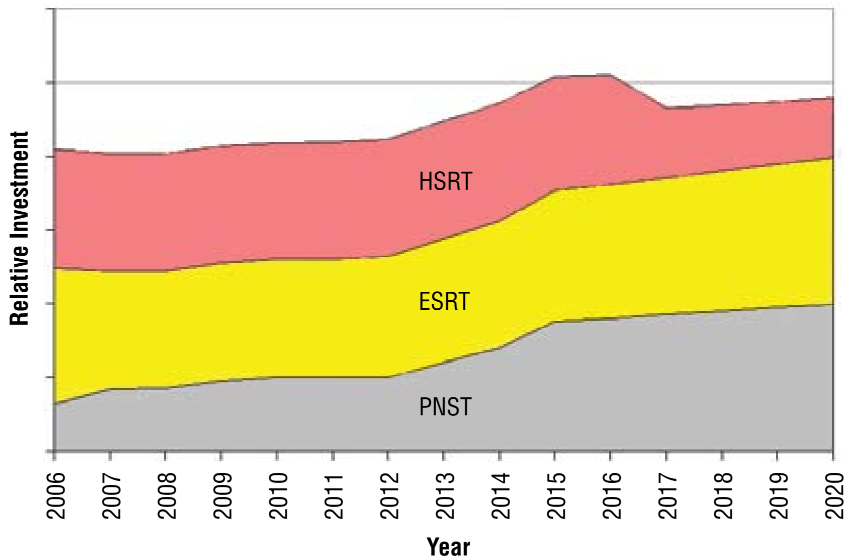


Figure 1-31. FY06–FY19 Original Funding Profile

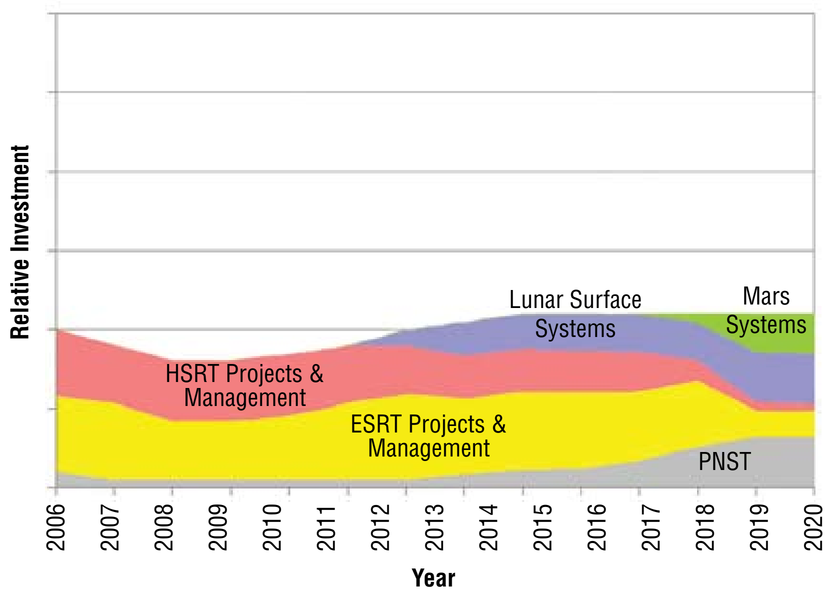


Figure 1-32. FY06–FY19 ESAS-Recommended Funding Profile

Figures 1-33 through **1-35** show, respectively, the overall recommended R&T budget broken out by program with liens, functional need category, and mission. “Protected” programs include those protected from cuts due to statutory requirements or previous commitments.

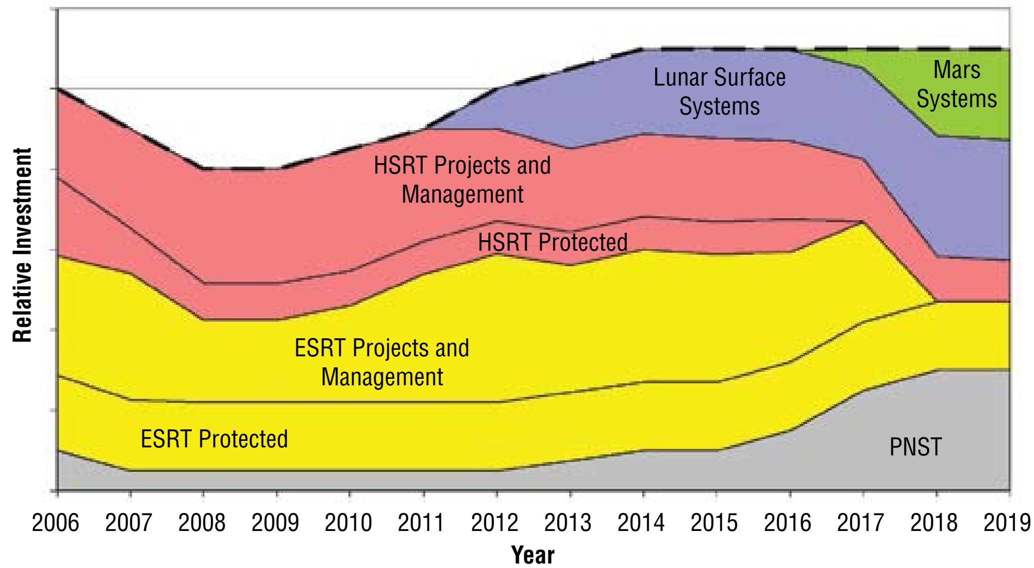


Figure 1-33. Overall Recommended R&T Budget Broken Out by Program with Liens

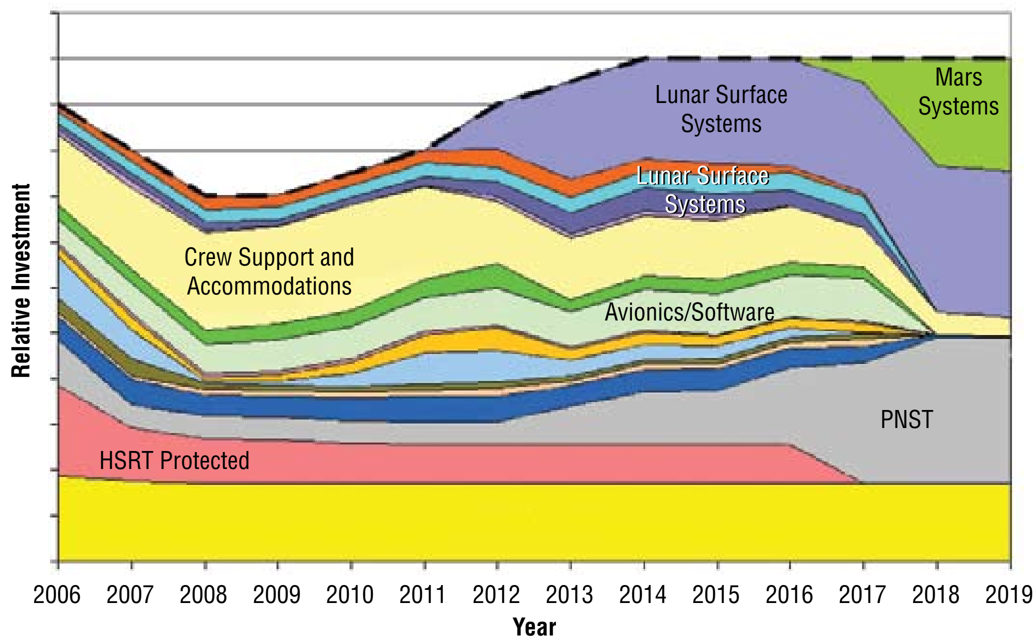


Figure 1-34. Overall Recommended R&T Budget Broken Out by Functional Need Category

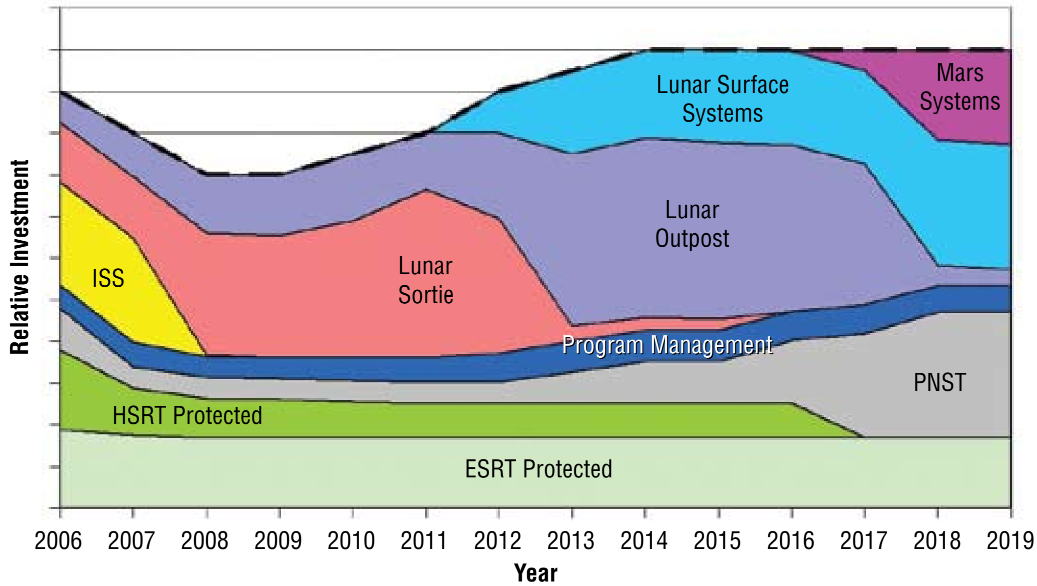


Figure 1-35. Overall Recommended R&T Budget Broken Out by Mission

The funding profile includes 10 percent management funds and approximately 30 percent liens due to prior agency agreements (e.g., Multi-User System and Support (MUSS), the Combustion Integrated Rack (CIR), and the Fluids Integrated Rack (FIR)) and legislated requirements (e.g., Small Business Innovation Research (SBIR), Small Business Technology Transfer (STTR)).

The final recommended technology funding profile was developed in coordination with the ESAS cost estimators using the results of the technology assessment.

Table 1-1.
Technology Project Recommendations

Number	ESAS Control Number	Program	Category	New Projects
1	1A	ESRT	Structures	Lightweight structures, pressure vessel, and insulation.
2	2A	ESRT	Protection	Detachable, human-rated, ablative environmentally compliant TPS.
3	2C	HSRT	Protection	Lightweight radiation protection for vehicle.
4	2E	HSRT	Protection	Dust and contaminant mitigation.
5	3A	ESRT	Propulsion	Human-rated, 5–20 klbf class in-space engine and propulsion system (SM for ISS orbital operations, lunar ascent and TEI, pressure-fed, LOX/CH4, with LADS). Work also covers 50–100 lbs nontoxic (LOX/CH4) RCS thrusters for SM.
6	3B	ESRT	Propulsion	Human-rated deep throttleable 5–20 klbf engine (lunar descent, pump-fed LOX/LH2).
7	3C	ESRT	Propulsion	Human-rated, pump-fed LOX/CH4 5–20 klbf thrust class engines for upgraded lunar LSAM ascent engine.
8	3D	ESRT	Propulsion	Human-rated, stable, nontoxic, monoprop, 50–100 lbf thrust class RCS thrusters (CM and lunar descent).
9	3F	ESRT	Propulsion	Manufacturing and production to facilitate expendable, reduced-cost, high production-rate SSMEs.
10	3G	ESRT	Propulsion	Long-term, cryogenic, storage and management (for CEV).
11	3H	ESRT	Propulsion	Long-term, cryogenic, storage, management, and transfer (for LSAM).
12	3K	ESRT	Propulsion	Human-rated, nontoxic 900-lbf Thrust Class RCS thrusters (for CLV and heavy-lift upper stage).
13	4B	ESRT	Power	Fuel cells (surface systems).
14	4E	ESRT	Power	Space-rated Li-ion batteries.
15	4F	ESRT	Power	Surface solar power (high-efficiency arrays and deployment strategy).
16	4I	ESRT	Power	Surface power management and distribution (e.g., efficient, low mass, autonomous).
17	4J	ESRT	Power	LV power for thrust vector and engine actuation (nontoxic APU).
18	5A	HSRT	Thermal Control	Human-rated, nontoxic active thermal control system fluid.
19	5B	ESRT	Thermal Control	Surface heat rejection.
20	6A	ESRT	Avionics and Software	Radiation hardened/tolerant electronics and processors.
21	6D	ESRT	Avionics and Software	Integrated System Health Management (ISHM) (CLV, LAS, EDS, CEV, lunar ascent/descent, habitat/Iso new hydrogen sensor for on-pad operations).
22	6E	ESRT	Avionics and Software	Spacecraft autonomy (vehicles & habitat).
23	6F	ESRT	Avionics and Software	Automated Rendezvous and Docking (AR&D) (cargo mission).
24	6G	ESRT	Avionics and Software	Reliable software/flight control algorithms.
25	6H	ESRT	Avionics and Software	Detector and instrument technology.
26	6I	ESRT	Avionics and Software	Software/digital defined radio.

Table 1-1. (continued)
Technology Project
Recommendations

Number	ESAS Control Number	Program	Category	New Projects
27	6J	ESRT	Avionics and Software	Autonomous precision landing and GN&C (Lunar & Mars).
28	6K	ESRT	Avionics and Software	Lunar return entry guidance systems (skip entry capability).
29	6L	ESRT	Avionics and Software	Low temperature electronics and systems (permanent shadow region ops).
30	7A	HSRT	ECLS	Atmospheric management - CMRS (CO2, Contaminants and Moisture Removal System).
31	7B	HSRT	ECLS	Advanced environmental monitoring and control.
32	7C	HSRT	ECLS	Advanced air and water recovery systems.
33	8B	HSRT	Crew Support and Accommodations	EVA Suit (including portable life suppport system).
34	8E	HSRT	Crew Support and Accommodations	Crew healthcare systems (medical tools and techniques, countermeasures, exposure limits).
35	8F	HSRT	Crew Support and Accommodations	Habitability systems (waste management, hygiene).
36	9C	ESRT	Mechanisms	Autonomous/teleoperated assembly and construction (and deployment) for lunar outpost.
37	9D	ESRT	Mechanisms	Low temperature mechanisms (lunar permanent shadow region ops).
38	9E	ESRT	Mechanisms	Human-rated airbag or alternative Earth landing system for CEV.
39	9F	ESRT	Mechanisms	Human-rated chute system with wind accommodation.
40	10A	ESRT	ISRU	Demonstration of regolith excavation and material handling for resource processing.
41	10B	ESRT	ISRU	Demonstration of oxygen production from regolith.
42	10C	ESRT	ISRU	Demonstration of polar volatile collection and separation.
43	10D	ESRT	ISRU	Large-scale regolith excavation, manipulation and transport (i.e., including radiation shielding construction).
44	10E	ESRT	ISRU	Lunar surface oxygen production for human systems or propellant.
45	10F	ESRT	ISRU	Extraction of water/hydrogen from lunar polar craters.
46	10H	ESRT	ISRU	In-situ production of electrical power generation (lunar outpost solar array fabrication).
47	11A	ESRT	Analysis and Integration	Tool development for architecture/mission/technology analysis/design, modeling and simulation.
48	11B	ESRT	Analysis and Integration	Technology investment portfolio assessment and systems engineering and integration.
49	12A	ESRT	Operations	Supportability (commonality, interoperability, maintainability, logistics, and in-situ fab.)
50	12B	ESRT	Operations	Human-system interaction (including robotics).
51	12C	ESRT	Operations	Surface handling, transportation, and operations equipment (Lunar or Mars).
52	12E	ESRT	Operations	Surface mobility.

1.7 Architecture Roadmap

As outlined in this executive summary, the ESAS team developed a time-phased, evolutionary architecture approach to return humans to the Moon, to service the ISS after Space Shuttle retirement, and to eventually transport humans to Mars. The individual elements were integrated into overall Integrated Master Schedules (IMSs) and detailed, multi-year integrated LCCs and budgets. These detailed results are provided in **Section 11**, **Integrated Master Schedule**, and **Section 12**, **Cost**, of this report. A top-level roadmap for ESAS architecture implementation is provided in **Figure 1-36**.

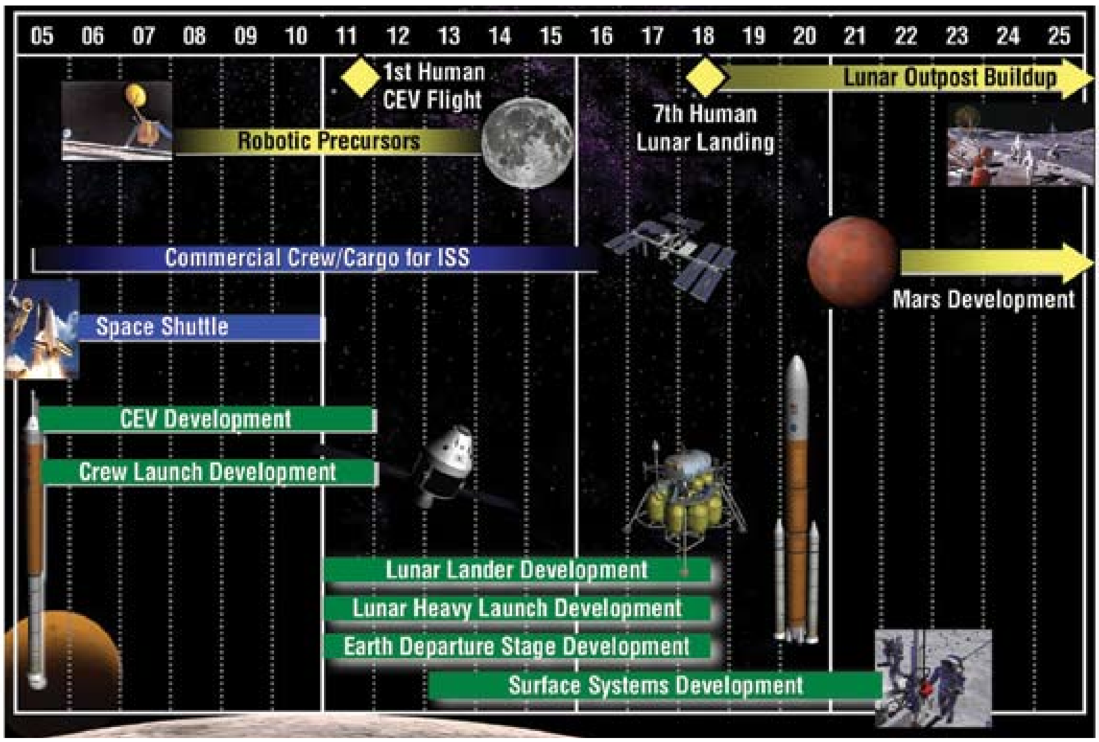


Figure 1-36. ESAS Architecture Implementation Roadmap

In this implementation, the Space Shuttle would be retired in 2010, using its remaining flights to deploy the ISS and, perhaps, service the Hubble Space Telescope (HST). CEV and CLV development would begin immediately, leading to the first crewed CEV flight to the ISS in 2011. Options for transporting cargo to and from the ISS would be pursued in cooperation with industry, with a goal of purchasing transportation services commercially. Lunar robotic precursor missions would begin immediately with the development and launch of the Lunar Reconnaissance Orbiter mission and continue with a series of landing and orbiting probes to prepare for extended human lunar exploration. In 2011, development would begin of the major elements required to return humans to the Moon—the LSAM, CaLV, and EDS. These elements would be developed and tested in an integrated fashion, leading to a human lunar landing in 2018. Starting in 2018, a series of short-duration lunar sortie missions would be accomplished, leading up to the deployment and permanent habitation of a lunar outpost. The surface systems (e.g., rovers, habitats, power systems) would be developed as required. Lunar missions would demonstrate the systems and technologies needed for eventual human missions to Mars.

1.8 Architecture Advantages

The ESAS team examined a wide variety of architecture element configurations, functionality, subsystems, technologies, and implementation approaches. Alternatives were systematically and objectively evaluated against a set of FOMs. The results of these many trade studies are summarized in each major section of this report and in the recommendations in **Section 13**, **Summary and Recommendations**.

Although many of the key features of the architecture are similar to systems and approaches used in the Apollo Program, the selected ESAS architecture offers a number of advantages over that of Apollo, including:

- Double the number of crew to the lunar surface;
- Four times the number of lunar surface crew-hours for sortie missions;
- A CM with three times the volume of the Apollo Command Module;
- Global lunar surface access with anytime return to the Earth;
- Enabling a permanent human presence at a lunar outpost;
- Demonstrating systems and technologies for human Mars missions;
- Making use of in-situ lunar resources; and
- Providing significantly higher human safety and mission reliability.

In addition to these advantages over the Apollo architecture, the ESAS-selected architecture offers a number of other advantages and features, including:

- The Shuttle-derived launch options were found to be more affordable, safe, and reliable than EELV options;
- The Shuttle-derived approach provides a relatively smooth transition of existing facilities and workforce to ensure lower schedule, cost, and programmatic risks;
- Minimizing the number of launches through development of a heavy-lift CaLV improves mission reliability and safety and provides a launcher for future human Mars missions;
- Use of an RSRB-based CLV with a top-mounted CEV and LAS provides an order-of-magnitude improvement in ascent crew safety over the Space Shuttle;
- Use of an Apollo-style blunt-body capsule was found to be the safest, most affordable, and fastest approach to CEV development;
- Use of the same modular CEV CM and SM for multiple mission applications improves affordability;
- Selection of a land-landing, reusable CEV improves affordability;
- Use of pressure-fed LOX/methane propulsion on the CEV SM and LSAM ascent stage enables ISRU for lunar and Mars applications and improves the safety of the LSAM; and
- Selection of the “1.5-launch” EOR–LOR lunar mission mode offers the safest and most affordable option for returning humans to the Moon.

2. Introduction

2.1 Background

In January 2004, President George W. Bush announced a new Vision for Space Exploration for NASA that would return humans to the Moon by 2020 in preparation for human exploration of Mars. As part of this vision, NASA would retire the Space Shuttle in 2010 and build and fly a new Crew Exploration Vehicle (CEV) no later than 2014. Initially, since no plans were made for this CEV to service the International Space Station (ISS), international partner assets would be required to ferry U.S. crew and cargo to the ISS after 2010—creating a significant gap in domestic space access for U.S. astronauts. NASA gradually reorganized to better implement the President's vision and established the Exploration Systems Mission Directorate (ESMD) to lead the development of a new exploration "system-of-systems" to accomplish these tasks. Over the course of the next year, ESMD defined preliminary requirements and funded system-of-system definition studies by Government and industry. More than $1 billion in technology tasks were immediately funded in a wide variety of areas. Plans were established to spend more than $2 billion per year in exploration systems, human, and nuclear-related technologies. Plans were established to fund two CEV contractors through Preliminary Design Review (PDR) and first flight of a subscale test demonstration in 2008, after which selection of a final CEV contractor would be made. In March 2004, a CEV Request for Proposals (RFP) was released to industry despite the lack of a firm set of requirements or a preferred architecture approach for returning humans to the Moon. A wide variety of architecture options was still under consideration at that time—with none considered feasible within established budgets. Preferred architecture options relied on as many as nine launches for a single lunar mission and relied on modified versions of the United States Air Force (USAF) Evolved Expendable Launch Vehicles (EELVs) for launch of crew and cargo.

Dr. Michael Griffin was named the new NASA Administrator in April 2005. With concurrence from Congress, he immediately set out to restructure NASA's Exploration Program by making it a priority to accelerate the development of the CEV to reduce or eliminate the planned gap in U.S. human access to space. He established a goal for the CEV to begin operation in 2011 and to be capable of ferrying crew and cargo to and from the ISS. To make room for these priorities in the budget, Dr. Griffin decided to downselect to a single CEV contractor as quickly as possible and cancel the planned 2008 subscale test demonstration. He also decided to significantly reduce the planned technology expenditures and focus on existing technology and proven approaches for exploration systems development. In order to reduce the number of required launches and ease the transition after Space Shuttle retirement in 2010, Dr. Griffin also directed the Agency to carefully examine the cost and benefits of developing a Shuttle-derived Heavy-Lift Launch Vehicle (HLLV) to be used in lunar and Mars exploration. To determine the best exploration architecture and strategy to implement these many changes, the Exploration Systems Architecture Study (ESAS) team was established at NASA Headquarters (HQ) as discussed in **Section 2.2**, **Charter**, and **Section 2.3**, **Approach**.

2.2 Charter

The ESAS began on May 2, 2005, at the request of the NASA Administrator. The study was commissioned in a letter dated April 29, 2005, provided in **Appendix 2A**, **Charter for the Exploration Systems Architecture Study (ESAS)**, from the NASA Administrator to all NASA Center Directors and Associate Administrators. The study was initiated to perform four specific tasks by July 29, 2005, as outlined in the letter and identified below.

- Complete assessment of the top-level CEV requirements and plans to enable the CEV to provide crew transport to the ISS and to accelerate the development of the CEV and crew launch system to reduce the gap between Shuttle retirement and CEV Initial Operational Capability (IOC).
- Provide definition of top-level requirements and configurations for crew and cargo launch systems to support the lunar and Mars exploration programs.
- Develop a reference lunar exploration architecture concept to support sustained human and robotic lunar exploration operations.
- Identify key technologies required to enable and significantly enhance these reference exploration systems and reprioritize near-term and far-term technology investments.

More than 20 core team members were collocated at NASA HQ for the 3-month duration. Over the course of the ESAS effort, hundreds of employees from NASA HQ and the field centers were involved in design, analysis, planning, and costing activities.

2.3 Approach

The ESAS effort was organized around each of the four major points of the charter: CEV definition, Launch Vehicle (LV) definition, lunar architecture definition, and technology plan definition. Additional key analysis support areas included cost, requirements, ground operations, mission operations, human systems, reliability, and safety.

The ESAS team took on the task of developing new CEV requirements and a preferred configuration to meet those requirements. The CEV requirements developed by the ESAS team are contained in **Appendix 2B**, **ESAS CEV Requirements**. A wide variety of trade studies was addressed by the team. Different CEV shapes were examined, including blunt-body, slender-body, and lifting shapes. The required amount of habitable volume and number of crew were determined for each mission based on a crew task analysis. Economic-based trades were performed to examine the benefits of reusability and system commonality. The effects of a CEV mission to the ISS were examined in detail, including docking and berthing approaches and the use of the CEV as a cargo transport and return vehicle. The requirements for Extra-Vehicular Activity (EVA) were examined and different airlock approaches were investigated. Additional trades included: landing mode, propellant type, number of engines, level of engine-out capability, and abort approaches. A phased development approach was defined that uses block upgrades of the CEV system for ISS crew, ISS cargo, lunar, and Mars missions with the same shape and size system.

The ESAS team examined hundreds of different combinations of launch elements to perform the various Design Reference Missions (DRMs). Different sizes of LVs and numbers of launches required to meet the DRMs were traded. The team's major trade study was a detailed examination of the costs, schedule, reliability, safety, and risk of using EELV-derived and Shuttle-derived launchers for crew and cargo missions. Other trade studies included: stage propellant type, numbers of engines per stage, level of stage commonality, and number of stages.

The ESAS team was tasked to develop new architecture-level requirements and an overall architecture approach to meet those requirements. The architecture requirements developed by the team are contained in **Appendix 2C**, **ESAS Architecture Requirements**. An initial reference architecture was established and configuration control was maintained. Trade studies were then conducted from this initial baseline. In order to determine the crew and cargo transportation requirements, the team examined and traded a number of different lunar surface missions and systems and different approaches to constructing a lunar outpost. A team of nationally recognized lunar science experts was consulted to determine science content and preferred locations for sortie and outpost missions. The use of in-situ resources for propellant and power was examined, and nuclear and solar power sources were traded. The major trade study conducted by the team was an examination of various mission modes for transporting crew and cargo to the Moon, including: Lunar Orbit Rendezvous (LOR), Earth Orbit Rendezvous (EOR), and direct return from the lunar surface. The number and type of elements required to perform the Trans-Lunar Injection (TLI), Lunar-Orbit Insertion (LOI), and Trans-Earth Injection (TEI) burns associated with these missions were also traded. In addition, a number of different configurations were examined for the lunar lander, or Lunar Surface Access Module (LSAM). Trade studies for the LSAM included: number of stages, stage propellant and engine type, level of engine-out capability, airlock approaches, cargo capacity, and abort options.

The ESAS team was also tasked to determine the architecture technology requirements and reprioritize existing technology plans to provide mature technologies prior to the PDR of each major element. The team used a disciplined, proven process to prioritize technology investments against architecture-level Figures of Merit (FOMs) for each mission. New technology investments were recommended only when required to enable a particular system, and investments were planned to begin only as required based on the need date.

The various trade studies conducted by the ESAS team used a common set of FOMs for evaluation. Each option was quantitatively or qualitatively assessed against the FOMs shown in **Figure 2-1**. FOMs are included in the areas of: safety and mission success, effectiveness and performance, extensibility and flexibility, programmatic risk, and affordability. FOMs were selected to be as mutually independent and measurable as possible. Definitions of each of these FOMs are provided in **Appendix 2D**, **ESAS FOM Definitions**, together with a list of measurable proxy variables and drivers used to evaluate the impacts of trade study options against the individual FOMs.

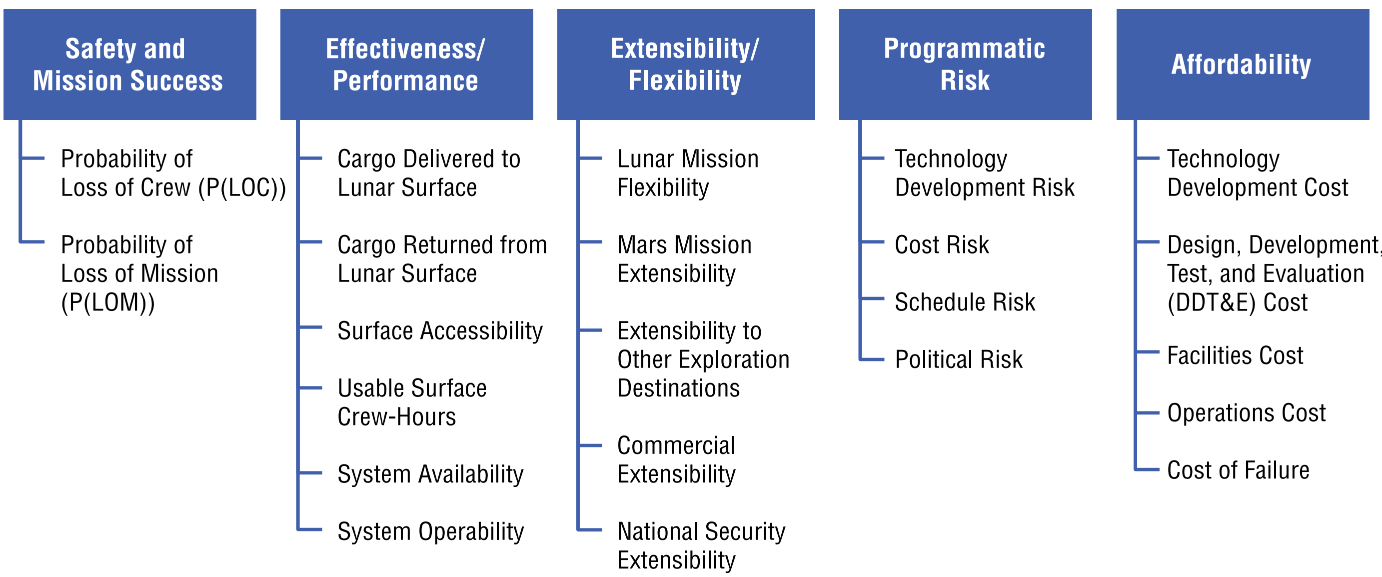


Figure 2-1. ESAS FOMs

2.4 Design Reference Missions

A series of DRMs was established to facilitate the derivation of requirements and the allocation of functionality between the major architecture elements. Three of the DRMs were for ISS-related missions: transportation of crew to and from the ISS, transportation of pressurized cargo to and from the ISS, and transportation of unpressurized cargo to the ISS. Three of the DRMs were for lunar missions: transportation of crew and cargo to and from anywhere on the lunar surface in support of 7-day "sortie" missions, transportation of crew and cargo to and from an outpost at the lunar south pole, and one-way transportation of cargo to anywhere on the lunar surface. A DRM was also established for transporting crew and cargo to and from the surface of Mars for a 18-month stay.

2.4.1 DRM Description: Crew Transport To and From ISS

The primary purpose of this mission is to transport three ISS crew members, and up to three additional temporary crew members, to the ISS for a 6-month stay and return them safely to Earth at any time during the mission. The architecture elements that satisfy the mission consist of a CEV and a Crew Launch Vehicle (CLV). **Figure 2-2** illustrates the mission. The CEV, consisting of a Crew Module (CM) and a Service Module (SM), is launched by the CLV into a 56- x 296-km insertion orbit at 51.6-deg inclination with a crew of three to six destined for a 6-month ISS expedition. The CEV performs orbit-raising burns per a pre-mission-defined rendezvous phasing profile to close on the ISS. These burns will be a combination of ground-targeted and onboard-targeted burns, the latter to be performed once rendezvous navigation sensors acquire the ISS. The CEV crew conducts a standard approach to the ISS, docking to one of two available CEV-compatible docking ports. The CEV crew pressurizes the vestibule between the two docked vehicles and performs a leak check. The ISS crew then equalizes pressure with the CEV vestibule and hatches are opened. Once ingress activities are complete, the CEV is configured to a quiescent state and assumes a "rescue vehicle" role for the duration of the crew increment. Periodic systems health checks and monitoring are performed by Mission Control throughout the increment. Upon completion of up to a 180-day increment on the ISS, the crew stows any return manifest items in the CEV crew cabin, performs a pre-undock health check of all entry critical systems, closes hatches and performs leak checks, and undocks from the station. The CEV departs the vicinity of the ISS and conducts an onboard-targeted (ground-validated) deorbit burn. After burn completion, the CEV SM is discarded, and the return component is maneuvered to the proper entry interface attitude for a guided entry to the landing site. The CEV performs a nominal landing at the primary land-based landing site.

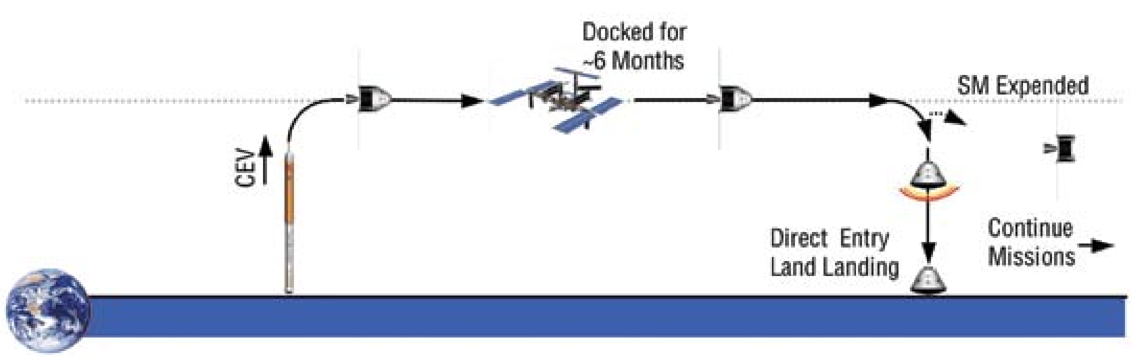


Figure 2-2. Crew Transport to and from ISS DRM

2.4.2 DRM Description: Unpressurized Cargo Transport to ISS

The primary purpose of this mission is to transport unpressurized cargo to the ISS and deorbit to perform a destructive reentry after 30 days at the ISS. The architecture elements that satisfy this mission consist of a Cargo Delivery Vehicle (CDV) and a CLV. **Figure 2-3** illustrates the mission. The CDV is launched by the CLV into a 56- x 296-km insertion orbit at 51.6-deg inclination with an unpressurized carrier in place of the CEV CM loaded with up to 6,000 kg gross mass of external ISS logistics. The CDV performs orbit-raising burns per a pre-mission-defined rendezvous phasing profile to close on the ISS. These burns will be a combination of ground-targeted and onboard-targeted burns, the latter performed once rendezvous navigation sensors acquire the ISS. The CDV performs a standard approach to a safe stationkeeping point in the vicinity of the ISS. Upon validation of readiness to proceed by Mission Control, the CDV is commanded to proceed with approach and conducts a standard onboard-guided approach to the ISS, achieving a stationkeeping point within reach of the Space Station Remote Manipulator System (SSRMS). The ISS crew grapples the CDV and berths it to the Node 2 nadir Common Berthing Mechanism (CBM) port. Once berthing activities are complete, the CDV systems are configured to a quiescent state. The ISS crew performs logistics transfer and systems maintenance EVAs to offload the CDV unpressurized pallet of new Orbital Replacement Units (ORUs) and load old ORUs for disposal. Periodic systems health checks and monitoring are performed by Mission Control throughout the increment. Upon completion of up to a 30-day mated phase on the ISS, Mission Control performs a pre-undock health check of all entry critical systems. Then, the ISS crew grapples the CDV, unberths it from the CBM, and maneuvers it to its departure point and releases it. The CDV departs the vicinity of the ISS and conducts an onboard-targeted (ground-validated) deorbit burn for disposal.

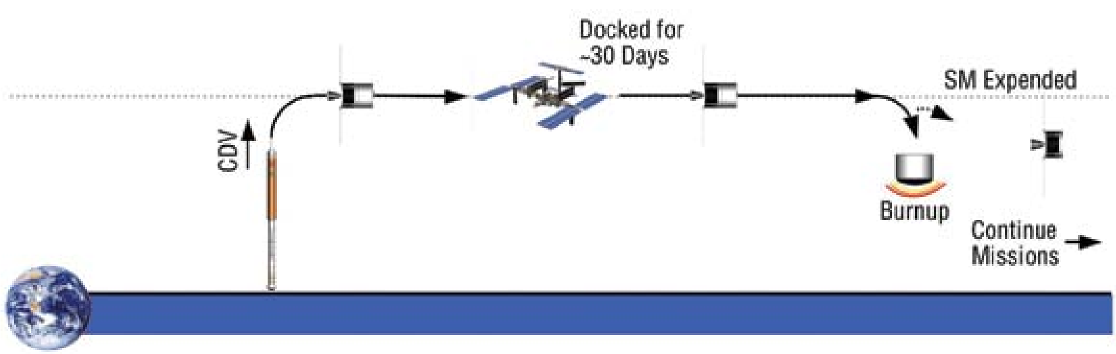


Figure 2-3. Unpressurized Cargo Transport to ISS DRM

2.4.3 DRM Description: Pressurized Cargo Transport To and From ISS

The primary purpose of this mission is to transport pressurized cargo to the ISS and deorbit to perform a reentry and safe return of pressurized cargo to Earth after 90 days at the ISS. **Figure 2-4** illustrates the mission. The architecture elements that satisfy this mission consist of a cargo version of the CEV and a CLV. A cargo version of the CEV is launched by the CLV into a 56- x 296-km insertion orbit at 51.6-deg inclination with the pressurized module filled with up to 3,500 kg gross mass of pressurized logistics for delivery to the ISS. The CEV performs orbit-raising burns per a pre-mission-defined rendezvous phasing profile to close on the ISS. These burns will be a combination of ground-targeted and onboard-targeted burns, the latter to be performed once rendezvous navigation sensors acquire the ISS. The uncrewed CEV performs a standard approach to a safe stationkeeping point in the vicinity of the ISS. Upon validation of readiness to proceed by Mission Control, the CEV is commanded to proceed with approach and conducts a standard onboard-guided approach to the ISS, docking to one of two available CEV-compatible docking ports. Mission Control pressurizes the vestibule between the two docked vehicles and performs a leak check. The ISS crew then equalizes with the CEV and hatches are opened. Once ingress activities are complete, the CEV systems are configured to a quiescent state and the CEV cargo is off-loaded. Periodic systems health checks and monitoring are performed by Mission Control throughout the increment. Upon completion of up to a 90-day docked phase on the ISS, the crew stows any return manifest items in the CEV pressurized cabin, Mission Control performs a pre-undock health check of all entry critical systems, the ISS crew closes hatches and performs leak checks, and Mission Control commands the CEV to undock from the station. The CEV departs the vicinity of the ISS and conducts an onboard-targeted (ground-validated) deorbit burn. After burn completion, unnecessary CEV elements are discarded, and the return element is maneuvered to the proper entry interface attitude for a guided entry to the landing site. The CEV performs a nominal landing at the primary land-based landing site.

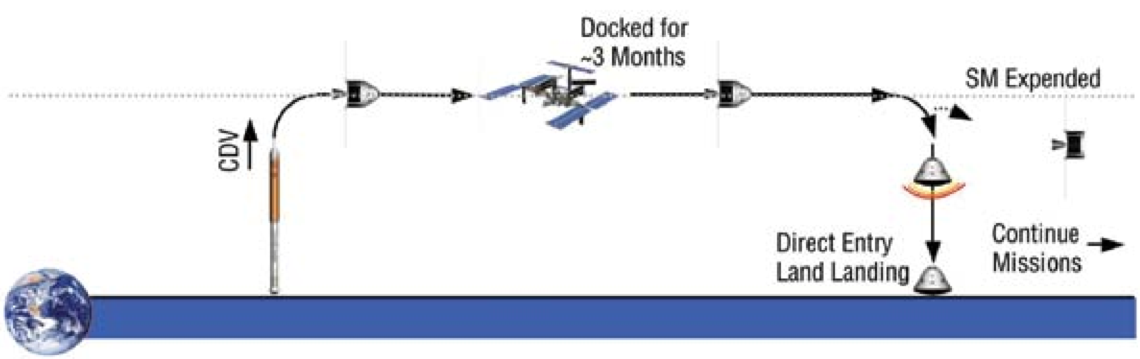


Figure 2-4. Pressurized Cargo Transport to and from ISS DRM

2.4.4 DRM Description: Lunar Sortie Crew with Cargo

The architecture provides the capability for up to four crew members to explore any site on the Moon (i.e., global access) for up to 7 days. These missions, referred to as lunar sorties, are analogous to the Apollo surface missions and demonstrate the capability of the architecture to land humans on the Moon, operate for a limited period on the surface, and safely return humans to Earth. Sortie missions also allow for exploration of high-interest science sites or scouting of future lunar outpost locations. Such a mission is assumed not to require the aid of pre-positioned lunar surface infrastructure such as habitats or power stations to perform the mission. During a sortie, the crew has the capability to perform daily EVAs with all crew members egressing from the vehicle through an airlock. Performing EVAs in pairs with all four crew members on the surface every day maximizes the scientific and operational value of the mission.

Figure 2-5 illustrates the lunar sortie crew and cargo mission. The following architecture elements are required to perform the mission: a CLV, a Cargo Launch Vehicle (CaLV) capable of delivering at least 125 mT to Low Earth Orbit (LEO), a CEV, an LSAM, and an Earth Departure Stage (EDS). The assumed mission mode for the lunar sortie mission is a combination EOR–LOR approach. The LSAM and EDS are predeployed in a single CaLV launch to LEO, and the CLV delivers the CEV and crew in Earth orbit, where the two vehicles initially rendezvous and dock. The EDS performs the TLI burn and is discarded. The LSAM then performs the LOI for both the CEV and LSAM. The entire crew then transfers to the LSAM, undocks from the CEV, and performs a descent to the lunar surface in the LSAM. After a lunar surface stay of up to 7 days, the LSAM returns the crew to lunar orbit where the LSAM and CEV dock, and the crew transfers back to the CEV. The CEV then returns the crew to Earth with a direct entry and land touchdown, while the LSAM is disposed of via impact on the lunar surface.

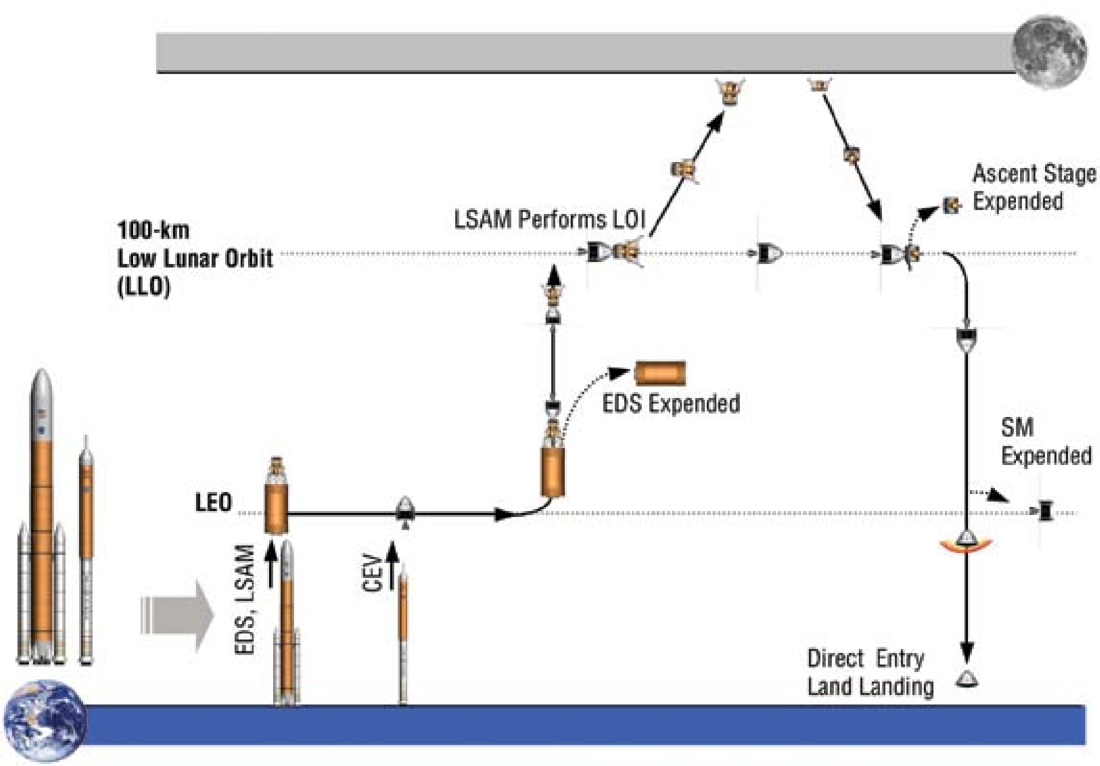


Figure 2-5. Lunar Sortie Crew with Cargo DRM

2.4.5 DRM Description: Lunar Outpost Cargo Delivery

The architecture provides the capability to deliver 20 mT of cargo to the lunar surface in a single mission using the elements of the human lunar transportation system. This capability is used to deliver surface infrastructure needed for lunar outpost buildup (habitats, power systems, communications, mobility, In-Situ Resource Utilization (ISRU) pilot plants, etc.), as well as periodic logistics resupply packages to support a continuous human presence.

Figure 2-6 illustrates the lunar outpost cargo delivery mission. The following architecture elements are required to perform the mission: the same CaLV and EDS as the sortie mission and a cargo variant of the LSAM to land the large cargo elements near the lunar outpost site. The cargo variant of the LSAM replaces the habitation module with a cargo pallet and logistics carriers. The LSAM and EDS are launched to LEO on a single CaLV. The EDS performs the TLI burn and is discarded. The LSAM then performs the LOI and a descent to the lunar surface. The cargo is then off-loaded from the LSAM autonomously or by the outpost crew.

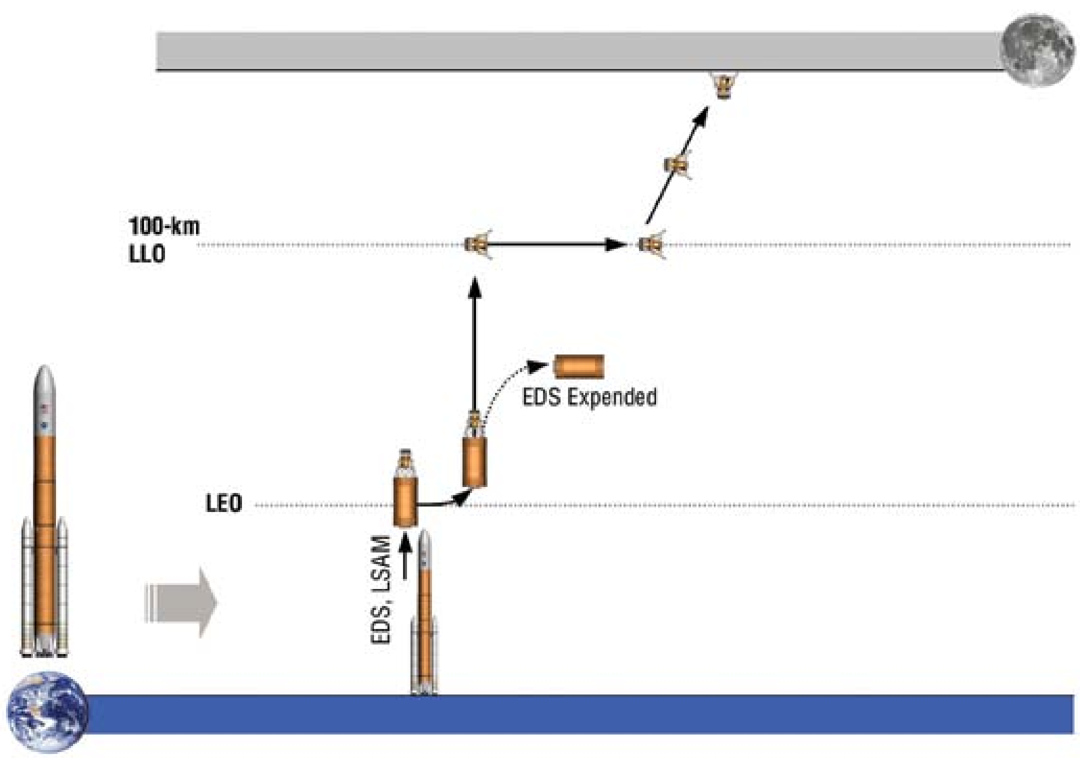


Figure 2-6. Lunar Outpost Cargo Delivery DRM

2.4.6 DRM Description: Lunar Outpost Crew with Cargo

A primary objective of the lunar architecture is to establish a continuous human presence on the lunar surface to accomplish exploration and science goals. This capability will be established as quickly as possible following the return of humans to the Moon. To best accomplish science and ISRU goals, the outpost is expected to be located at the lunar south pole. The primary purpose of the mission is to transfer up to four crew members and supplies in a single mission to the outpost site for expeditions lasting up to 6 months. Every 6 months, a new crew will arrive at the outpost, and the crew already stationed there will return to Earth. **Figure 2-7** illustrates this mission.

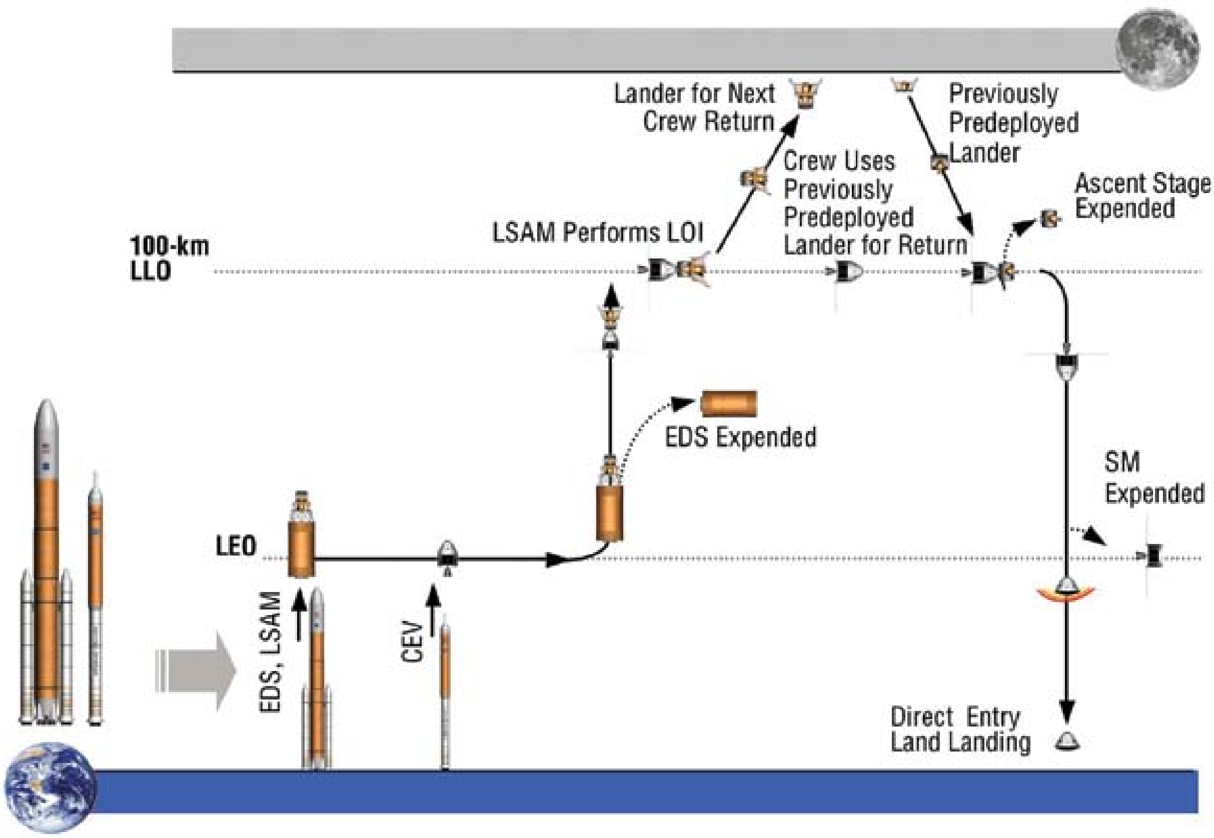


Figure 2-7. Lunar Outpost Crew with Cargo DRM

The entire suite of vehicles developed to support lunar sortie exploration is also required for lunar outpost missions, in addition to a surface habitat, power/communications systems, and other infrastructure elements still to be defined. The following architecture elements are required to perform the mission: a CLV, a CaLV capable of delivering at least 125 mT to LEO, a CEV, an LSAM, and an EDS. The assumed mission mode for the lunar sortie mission is a combination EOR–LOR approach. The LSAM and EDS are predeployed in a single CaLV launch to LEO, and the CLV delivers the CEV and crew in Earth orbit, where the two vehicles initially rendezvous and dock. The EDS performs the TLI burn and is discarded. The LSAM then performs the LOI for both the CEV and LSAM. The entire crew then transfers to the LSAM, undocks from the CEV, and performs a descent to the lunar surface near the outpost in the LSAM. After a surface stay of up to 6 months, the LSAM returns the crew to lunar orbit where the LSAM and CEV dock, and the crew transfers back to the CEV. The CEV then returns the crew to Earth with a direct entry and land touchdown, while the LSAM is disposed of via impact on the lunar surface.

2.4.7 DRM Description: Mars Exploration

The Mars Exploration DRM employs conjunction-class missions, often referred to as long-stay missions, to minimize the exposure of the crew to the deep-space radiation and zero-gravity environment while, at the same time, maximizing the scientific return from the mission. This is accomplished by taking advantage of optimum alignment of Earth and Mars for both the outbound and return trajectories by varying the stay time on Mars, rather than forcing the mission through non-optimal trajectories, as in the case of the short-stay missions. This approach allows the crew to transfer to and from Mars on relatively fast trajectories, on the order of 6 months, while allowing them to stay on the surface of Mars for a majority of the mission, on the order of 18 months.

The surface exploration capability is implemented through a split mission concept in which cargo is transported in manageable units to the surface, or Mars orbit, and checked out in advance of committing the crews to their mission. The split mission approach also allows the crew to be transported on faster, more energetic trajectories, minimizing their exposure to the deep-space environment, while the vast majority of the material sent to Mars is sent on minimum energy trajectories. An overview of the mission approach is shown in **Figure 2-8**. As can be seen in **Figure 2-8**, each human mission to Mars is comprised of three vehicle sets, two cargo vehicles, and one round-trip piloted vehicle.

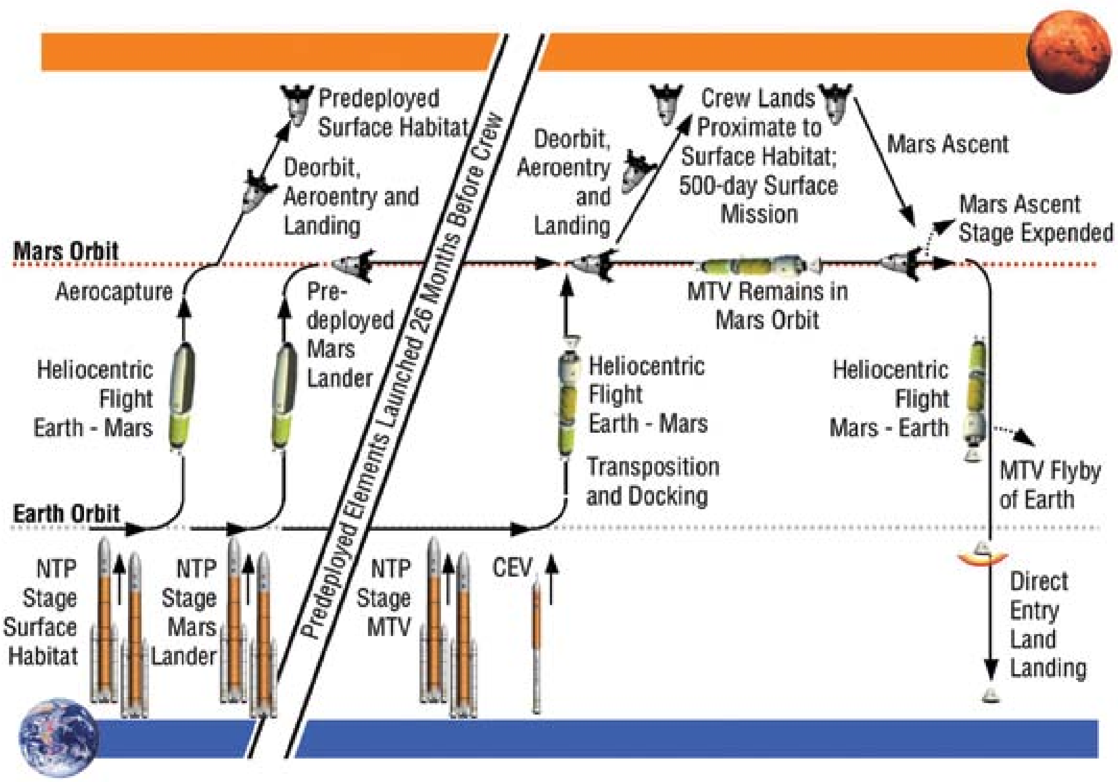


Figure 2-8. Mars Exploration DRM

The scope of the ESAS was only to address the transportation of the crew to a Mars Transfer Vehicle (MTV) in LEO or reentering from the MTV at the conclusion of the Mars mission, and to provide the design of a CaLV with an LEO cargo capacity of 125 mT.

This DRM utilizes the CEV to transfer a crew of six to and from an MTV as part of a Mars mission architecture. The CEV is launched by the CLV into an orbit matching the inclination of the MTV. The CEV spends up to 2 days performing orbit-raising maneuvers to close on the MTV. The CEV crew conducts a standard approach to the MTV and docks. The CEV crew performs a leak check, equalizes pressure with the MTV, and opens hatches. Once crew and cargo transfer activities are complete, the CEV is configured to a quiescent state. Periodic systems health checks and monitoring are performed by Mission Control throughout the Mars transfer mission.

As the MTV approaches Earth upon completion of the 2.5-year mission, the crew performs a pre-undock health check of all entry critical systems, transfers to the CEV, closes hatches, performs leak checks, and undocks from the MTV. The CEV departs the MTV 24 hours prior to Earth entry and conducts an onboard-targeted (ground-validated) deorbit burn. As entry approaches, the CEV maneuvers to the proper entry interface attitude for a direct guided entry to the landing site. The CEV performs a nominal landing at the primary land-based landing site.

3. Ground Rules and Assumptions

At the beginning of the Exploration Systems Architecture Study (ESAS), a number of Ground Rules and Assumptions (GR&As) were established based on management guidance, internal and external constraints, design practices, and existing requirements. The purpose of this section is to summarize those GR&As below.

3.1 Safety and Mission Assurance (S&MA) GR&As

The S&MA GR&As are listed below.

- NASA Procedural Requirements (NPR) 8705.2, Human-Rating Requirements for Space Systems, will be used as a guideline for all architecture design activities. Required deviations from NPR 8705.2 will be noted in the applicable requirements documentation.
- Abort opportunities will be provided throughout all mission phases to the maximum extent possible.
- In the event of an abort from the lunar surface, return of crew to the Earth's surface will take no longer than 5 days—independent of orbital alignment.

3.2 Operations GR&As

The Operations GR&As are listed below.

- The Crew Exploration Vehicle (CEV) will deliver crew to and from the International Space Station (ISS) through ISS end-of-life in 2016.
- The CEV will deliver and return cargo to the ISS through ISS end-of-life in 2016.
- The architecture will separate crew and large cargo to the maximum extent practical.
- The architecture will support ISS up/down mass needs and other ISS requirements, as required, after Shuttle retirement.
- CEV operations will be performed at the Kennedy Space Center (KSC) through clearing of the launch pad structure.
- On-orbit flight operations and in-flight operations for crewed missions will be performed at the Johnson Space Center (JSC).
- Crew and cargo recovery operations from the crew and cargo launches will be managed by KSC with assistance from other NASA and non-NASA personnel and assets as required.
- Architectures will enable extensibility of lunar mission systems to human Mars exploration missions.
- The study will utilize the Mars Design Reference Mission (DRM) known as DRM 3.0, "Reference Mission Version 3.0 Addendum to the Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team EX13-98-036, June 1998."
- The architecture will support lunar global access.
- The architecture will support a permanent human presence on the Moon.
- In-space Extra-Vehicular Activity (EVA) assembly will not be required.
- In-space EVA will only be performed as a contingency operation.
- Human-rated Evolved Expendable Launch Vehicle- (EELV-) derived Launch Vehicles (LVs) will require new dedicated launch pads.

3.3 Technical GR&As

The Technical GR&As are listed below.

- The CEV will be designed for up to a crew of six for ISS missions.
- The CEV will be designed for up to a crew of four for lunar missions.
- The CEV will be designed for up to a crew of six for Mars missions.
- The CEV to support the lunar and Mars exploration missions and the ISS missions will use a single Outer Mold Line (OML) for the entry vehicle.
- Architectures will be designed for the lunar and Mars exploration missions and modified as required to support ISS missions.
- No more than four launches will be used to accomplish a single human lunar mission. This does not include infrastructure launches or supporting logistics.
- The following inert weight contingencies will be used:
  - Zero percent (0%) for existing LV elements with no planned specification change and no anticipated modifications (e.g., Space Shuttle Main Engine (SSME), RS–68, RD–180);
  - Five percent (5%) on existing LV elements requiring minimal modifications (e.g., External Tank (ET), Orbiter aft structure, EELV boosters, upper stages, and shrouds);
  - Ten percent (10%) on new Expendable Launch Vehicle (ELV) elements with direct Shuttle or EELV heritage;
  - Fifteen percent (15%) on new ELV elements with no heritage; and
  - Twenty percent (20%) on new in-space elements with no heritage (e.g., CEV, Lunar Surface Access Module (LSAM)).
- Additional margins and factors of safety include the following:
  - Thirty percent (30%) margin for average power;
  - Two percent (2%) margin for reserves and residuals mass;
  - Two percent (2%) propellant tank ullage fractions for LV stages;
  - Fuel bias of nominal mixture ratio * 0.000246 * usable propellant weight;
  - A 2.0 factor of safety for crew cabins;
  - A 1.5 factor of safety on burst pressure for fluid pressure vessels;
  - A 1.4 ultimate factor of safety on all new or redesigned structures;
  - A 1.25 factor of safety on proof pressure for fluid pressure vessels;
  - Ten percent (10%) margin for rendezvous delta-Vs;
  - One percent (1%) ascent delta-V margin on LVs to account for dispersions;
  - Ten percent (10%) payload margin on all LV payload delivery predictions; and
  - Five percent (5%) additional payload margin on Cargo Launch Vehicle (CaLV) delivery predictions to account for Airborne Support Equipment (ASE).
- Technologies will be Technology Readiness Level-Six (TRL-6) or better by Preliminary Design Review (PDR).

3.4 Cost GR&As

The Cost GR&As are listed below.

- There will be only one CEV contractor after Calendar Year 2005 (CY05).
- There will be no 2008 CEV flight demonstration as originally planned.
- All Life Cycle Cost (LCC) estimates will include best-effort estimates of "full-cost" impacts (including corporate General and Administrative (G&A) at 5%, Center G&A, Center Civil Service salaries, travel, overhead, and Center service pool costs).
- Cost estimates will use 20 percent reserves for development.
- Cost estimates will use 10 percent reserves for operations.
- Cost estimates will use the April 2005 NASA New Start Inflation Index.

3.5 Schedule GR&As

The Schedule GR&As are listed below.

- There is a goal of 2011 for the first CEV human flight to ISS.
- There is a goal of performing the next human lunar landing by 2020—or as soon as practical.

3.6 Testing GR&As

The Testing GR&As are listed below.

- Ground Element Qualification
  - Elements will have ground qualification tests to demonstrate readiness for manned flight.
  - Multi-element integrated tests will be performed to demonstrate readiness for manned flight.
- Element Flight Qualification
  - Qualification of the CEV requires a minimum of one flight demonstrating full functionality prior to crewed flights.
  - Qualification of the LSAM requires a minimum of one flight demonstrating full functionality prior to lunar landing.
  - Qualification of any crewed LV requires three flight tests for human certification prior to crewed flight.
  - Qualification of any CaLV requires one flight test prior to flight of high-value cargo.
- Integrated System Qualification
  - Qualification of the Earth Departure Stage (EDS) for firing while mated to a crewed element requires a minimum of two flights to demonstrate full functionality prior to crewed flight.
  - Lunar mission rehearsal in-space with appropriate architecture elements and crew is required prior to attempting a lunar landing.

3.7 Foreign Assets GR&As

- Foreign assets utilized in LV configurations in this study will be assumed to be licensed and produced in the United States.

4. Lunar Architecture

4.1 Summary and Recommendations

As defined by the Exploration Systems Architecture Study (ESAS), the lunar architecture is a combination of the lunar "mission mode," the assignment of functionality to flight elements, and the definition of the activities to be performed on the lunar surface. The trade space for the lunar "mission mode," or approach to performing the crewed lunar missions, was limited to the cislunar space and Earth-orbital staging locations, the lunar surface activities duration and location, and the lunar abort/return strategies. The lunar mission mode analysis is detailed in **Section 4.2**, **Lunar Mission Mode**. Surface activities, including those performed on sortie- and outpost-duration missions, are detailed in **Section 4.3**, **Lunar Surface Activities**, along with a discussion of the deployment of the outpost itself.

The mission mode analysis was built around a matrix of lunar- and Earth-staging nodes. Lunar-staging locations initially considered included the Earth-Moon L1 libration point, Low Lunar Orbit (LLO), and the lunar surface. Earth-orbital staging locations considered included due-east Low Earth Orbits (LEOs), higher-inclination International Space Station (ISS) orbits, and raised apogee High Earth Orbits (HEOs). Cases that lack staging nodes (i.e., "direct" missions) in space and at Earth were also considered.

This study addressed lunar surface duration and location variables (including latitude, longitude, and surface stay-time) and made an effort to preserve the option for full global landing site access. Abort strategies were also considered from the lunar vicinity. "Anytime return" from the lunar surface is a desirable option that was analyzed along with options for orbital and surface loiter.

The duration, location, and centralization of lunar surface activities were analyzed by first determining the content of the science, resource utilization, Mars-forward technology demonstrations, and operational tests that could be performed during the lunar missions. The study team looked at high-priority landing sites and chose a reference site in order to further investigate the operations at a permanent outpost. With the scientific and engineering activities defined, concept-level approaches for the deployment and buildup of the outpost were created. A comprehensive definition of lunar surface elements and infrastructure was not performed because development activities for lunar surface elements are still years in the future. Therefore, the ESAS team concentrated its recommendations on those elements that had the greatest impact on near-term decisions.

The mission architecture decisions that most greatly affect near-term NASA development activities are mission mode, propulsion system types, and mission duration. The ESAS team recommends the use of an Earth Orbit Rendezvous-Lunar Orbit Rendezvous (EOR–LOR) mission mode. This mission mode, which can be executed with a combination of the launch of separate crew and cargo vehicles, was found to result in a Low Life Cycle Cost (LCC) and the highest crew safety and mission reliability combination. Further, the study found that pressure-fed Liquid Oxygen (LOX)/methane propulsion should be used for the lander ascent stage as well as the Crew Exploration Vehicle (CEV) Service Module (SM), which should be sized to perform the Trans-Earth Injection (TEI) propulsive maneuver for a lunar mission. The study also concluded that the lunar lander should use a LOX/hydrogen throttleable propulsion system for Lunar Orbit Insertion (LOI) and landing. The two-stage lander should include an airlock and be sized to support a 7-day surface mission with four crew members.

4.2 Lunar Mission Mode

The lunar mission mode is the fundamental lunar architecture decision that defines where space flight elements come together and what functions each of these elements perform. Mission mode analysis had its genesis early in the design of the Apollo Program, with notable NASA engineers and managers such as Wernher Von Braun, John Houbolt, Joe Shea, and Robert Seamans contributing to the decision to use LOR as the Apollo mission mode. This study built on the foundation of the Apollo decision but sought to question whether the LOR decision and overall Apollo mission approach were still valid, given new missions requirements and technology.

The ESAS team researched many of the Apollo lunar landing mode comparison studies as well as more recent studies performed by both NASA and industry. One study of interest, performed by a Massachusetts Institute of Technology (MIT)-Draper Laboratory team as part of a "Concept Exploration and Refinement" (CE&R) contract to NASA, suggested that the Apollo mission mode was no longer valid and that NASA should consider "direct return" modes for future human lunar missions. The ESAS team took special note of this study and sought to challenge all of the Apollo mission assumptions.

4.2.1 Previous Lunar Architecture Study Results

Since its inception, NASA has conducted or sponsored numerous studies of human exploration beyond LEO. These studies have been used to understand requirements for human exploration of the Moon and Mars in the context of other space missions and Research and Development (R&D) programs. Each exploration architecture provides an end-to-end mission baseline against which other mission and technology concepts can be compared. The results from the architecture studies were used to:

- Derive technology R&D plans;
- Define and prioritize requirements for precursor robotic missions;
- Define and prioritize flight experiments and human exploration mission elements, such as those involving the Space Shuttle, ISS, and other Space Transportation Systems (STSs);
- Open a discussion with international partners in a manner that allows identification of participants' potential interests in specialized aspects of the missions; and
- Describe to the public, media, and Government stakeholders the feasible, long-term visions for space exploration.

Each architecture study emphasized one or more critical aspects of human exploration in order to determine basic feasibility and technology needs. Examples of architectural areas of emphasis include:

- Destination: Moon ↔ Mars ↔ Libration Points ↔ Asteroids;
- System Reusability: Expendable ↔ Reusable;
- Architecture Focus: Sorties ↔ Colonization;
- Surface Mobility: Local ↔ Global;
- Launch Vehicles (LVs): Existing ↔ New Heavy-Lift;
- Transportation: Numerous stages and technologies traded;
- LEO Assembly: None ↔ Extensive;
- Transit Modes: Zero-gravity ↔ Artificial-gravity;
- Surface Power: Solar ↔ Nuclear;
- Crew Size: 4 ↔ 24; and
- In Situ Resource Utilization (ISRU): None ↔ Extensive.

The ESAS team extensively scrutinized the NASA studies that led to the Apollo Program, most notably studies to determine the shape of the Apollo capsule and the mode used for the Apollo missions. Additionally, the team reviewed the findings of human lunar and Mars mission studies performed over the past 15 years. A summary of these studies is shown in **Table 4-1**.

Office of Exploration (OExP) - 1988 Case Studies Human Expedition to Phobos Human Expedition to Mars Lunar Observatory Lunar Outpost to Early Mars Evolution	**First Lunar Outpost - 1993** **Early Lunar Resource Utilization - 1993** **Human Lunar Return - 1996**
Office of Exploration (OExP) - 1989 Case Studies Lunar Evolution Mars Evolution Mars Expedition	**Mars Exploration Missions** Design Reference Mission Version 1.0 - 1994 Design Reference Mission Version 3.0 - 1997 Design Reference Mission Version 4.0 - 1998 Mars Combo Lander (Johnson Space Center (JSC)) - 1999 Dual Landers – 1999
NASA 90-Day Study - 1989 Approach A - Moon as testbed for Mars missions Approach B - Moon as testbed for early Mars missions Approach C - Moon as testbed for Mars Outposts Approach D - Relaxed mission dates Approach E - Lunar outpost followed by Mars missions	**Decadal Planning Team (DPT)/NASA Exploration Team (NExT) - 2000–2002** Earth's Neighborhood Architecture Asteroid Missions Mars Short and Long Stay **Exploration Blueprint - 2002** **Space Architect - 2003**
America at the Threshold - "The Synthesis Group" - 1991 Mars Exploration Science Emphasis for the Moon and Mars The Moon to Stay and Mars Exploration Space Resource Utilization	**Exploration Systems Mission Directorate (ESMD) 2004–2005**

Table 4-1. Summary of Previous NASA Architecture Studies

4.2.1.1 Summary of Previous Studies

4.2.1.1.1 Office of Exploration Case Studies (1988)

In June 1987, the NASA Administrator established the Office of Exploration (OExP) in response to an urgent national need for a long-term goal to reenergize the U.S. civilian space program. The OExP originated as a result of two significant assessments conducted prior to its creation. In 1986, the National Commission on Space, as appointed by the President and charged by the Congress, formulated a bold agenda to carry America's civilian space enterprise into the 21st century (number 1 in **Section 4.5**, **Endnotes**). Later that year, the NASA Administrator asked scientist and astronaut Sally Ride to lead a task force to look at potential long-range goals of the U.S. civilian space program. The subsequent task force report, "Leadership and America's Future in Space," (number 2 in **Section 4.5**, **Endnotes**) outlined four initiatives which included both human and robotic exploration of the solar system.

In response to the task force report, the OExP conducted a series of studies of human and robotic exploration beyond LEO during the 1987–1988 time frame. These studies ranged in scope and scale with the direct purpose of providing an understanding of the driving mission, technology, and operational concepts for various exploration missions. Four focused case studies were examined: Human Expeditions to Phobos, Human Expeditions to Mars, Lunar Observatory, and Lunar Outpost to Early Mars Evolution.

The case studies were deliberately set at the boundaries of various conditions in order to elicit first principles and trends toward the refinement of future options, as well as to define and refine prerequisites. The objective of this approach was to determine a viable pathway into the solar system and avoid making simple distinctions between Moon or Mars exploration.

Recommendations resulting from the 1988 (number 3 in **Section 4.5**, **Endnotes**) case studies included the following key points:

- A Space Station is the key to developing the capability to live and work in space;
- Continued emphasis on Research and Technology (R&T) will enable a broad spectrum of space missions and strengthen the technology base of the U.S. civilian space program;
- A vigorous life science research-based program must be sustained;
- A heavy-lift transportation system must be pursued with a capability targeted to transport large quantities of mass to LEO;
- Obtaining data via robotic precursor missions is an essential element of future human exploration efforts;
- An artificial gravity research program must be initiated in parallel with the zero-gravity countermeasure program if the U.S. is to maintain its ability to begin exploration in the first decade of the next century; and
- An advanced development/focused test program must be initiated to understand the performance and capability of selected new technologies and systems.

4.2.1.1.2 Office of Exploration Case Studies (1989)

Following the 1988 studies, the OExP continued to lead the NASA-wide effort to provide recommendations and alternatives for a national decision on a focused program of human exploration of the Solar System. Three case studies were formulated during 1989 for detailed development and analysis: Lunar Evolution, Mars Evolution, and Mars Expedition. In addition, a series of special assessments was conducted that focused on high-leverage areas which were independent of the case studies and covered a generally broad subject area with potential for significant benefit to all mission approaches. Special assessments included Power System, Propulsion System, Life Support Systems, Automation and Robotics, Earth-Moon Node Location, Lunar LOX Production, and Launch/On-Orbit Processing.

Results from the 1989 OExP studies were published in the Fiscal Year (FY) 1989 OExP Annual Report (number 4 in **Section 4.5**, **Endnotes**). Key conclusions from the 1989 studies included:

- Mars Trajectories: Human missions to Mars are characterized by the surface stay-time required–short-stay referring to opposition-class missions and long-stay pertaining to conjunction-class Mars missions;
- In-Space Propulsion: All-chemical-propulsive transportation results in prohibitive total mission mass for Mars missions (1,500–2,000 mT per mission). On the other hand, aerobraking utilization at Mars can provide significant mass savings (50 percent) as compared to all-chemical-propulsive transportation. Incorporation of advanced propulsion, such as nuclear thermal rockets or nuclear electric propulsion, can result in mission masses comparable to chemical/aerobraking missions;
- Reusable Spacecraft: Employment of reusable spacecraft is predominantly driven by economic considerations; however, reusing spacecraft requires in-space facilities to store, maintain, and refurbish the vehicles, or the vehicles must be designed to be space-based with little or no maintenance;
- In-Situ Resources: The use of in-situ resources reduces the logistical demands on Earth of maintaining a lunar outpost and helps to develop outpost operational autonomy from Earth; and
- Space Power: As the power demands at the lunar outpost increase above the 100 kWe level, nuclear power offers improved specific power.

4.2.1.1.3 NASA 90-Day Study (1989)

On July 20, 1989, the President announced a major new space exploration vision, asking the Vice President to lead the National Space Council in determining what would be needed to chart a new and continuing course to the Moon and Mars. To support this endeavor, the NASA Administrator created a task force to conduct a 90-day study of the main elements of a human exploration program (number 5 in **Section 4.5**, **Endnotes**). Data from this study was to be used by the National Space Council in its deliberations. Five reference approaches were developed, each of which was based on the President's strategy of "Space Station, Moon, then Mars." Regardless of the reference architecture, the study team concluded that Heavy-Lift Launch Vehicles (HLLVs), space-based transportation systems, surface vehicles, habitats, and support systems for living and working in deep space are required. Thus, the reference architectures made extensive use of the Space Station (Freedom) for assembly and checkout operations of reusable transportation vehicles, ISRU (oxygen from the lunar regolith), and chemical/aerobrake propulsion.

4.2.1.1.4 America at the Threshold – "The Synthesis Group" (1991)

In addition to the internal NASA assessment of the Space Exploration Initiative (SEI) conducted during the NASA 90-Day Study, the Vice President and NASA Administrator chartered an independent team called the Synthesis Group to examine potential paths for implementation of the exploration initiative (number 6 in **Section 4.5**, **Endnotes**). This group examined a wide range of mission architectures and technology options. In addition, the group performed a far-reaching search for innovative ideas and concepts that could be applied to implementing the initiative.

The Synthesis Group's four candidate architectures were Mars Exploration, Science Emphasis for the Moon and Mars, The Moon to Stay and Mars Exploration, and Space Resource Utilization. Supporting technologies identified as key for future exploration included:

- HLLV (150–250 mT),
- Nuclear Thermal Propulsion (NTP),
- Nuclear electric surface power,
- Extra-Vehicular Activity (EVA) suit,
- Cryogenic transfer and long-term storage,
- Automated Rendezvous and Docking (AR&D),
- Zero-g countermeasures,
- Telerobotics,
- Radiation effects and shielding,
- Closed-loop life support systems,
- Human factors research,
- Lightweight structural materials,
- Nuclear electric propulsion, and
- In-situ resource evaluation and processing.

The Synthesis Group also conducted an extensive outreach program with nationwide solicitation for innovative ideas. The Vice President's directive was to "cast the net widely." Ideas were solicited from universities, professional societies and associations, the American Institute of Aeronautics and Astronautics (AIAA), the Department of Defense (DoD) Federal Research Review, the Department of Energy (DoE), the Department of the Interior, and the Aerospace Industries Association, as well as from announcements in the Commerce Business Daily. Nearly 45,000 information packets were mailed to individuals and organizations interested in the SEI, resulting in more than 1,500 submissions. According to a Synthesis Group statement at the time, "The ideas submitted showed innovative but not necessarily revolutionary ideas. The submissions supported a wide range of SEI mission concepts and architectures."

In addition, the Synthesis Group provided specific recommendations for the "effective implementation of the Space Exploration Initiative," including:

- Recommendation 1: Establish within NASA a long-range strategic plan for the nation's civil space program, with the SEI as its centerpiece;
- Recommendation 2: Establish a National Program Office by Executive Order;
- Recommendation 3: Appoint NASA's Associate Administrator for Exploration as the Program Director for the National Program Office;
- Recommendation 4: Establish a new aggressive acquisition strategy for the SEI;
- Recommendation 5: Incorporate SEI requirements into the joint NASA-DoD Heavy-Lift Program;
- Recommendation 6: Initiate a nuclear thermal rocket technology development program;
- Recommendation 7: Initiate a space nuclear power technology development program based on the SEI requirements;
- Recommendation 8: Conduct focused life sciences experiments;
- Recommendation 9: Establish education as a principal theme of the SEI; and
- Recommendation 10: Continue and expand the Outreach Program.

4.2.1.1.5 First Lunar Outpost (1993)

Following the Synthesis Group's recommendations, NASA began the planning for implementation of the first steps of the SEI after completion of the Space Station, namely "back to the Moon, back to the future, and this time, back to stay." This activity was termed the First Lunar Outpost, (number 7 in **Section 4.5**, **Endnotes**) an Agency-wide effort aimed at understanding the technical, programmatic, schedule, and budgetary implications of restoring U.S. lunar exploration capability. Emphasis was placed on minimizing integration of elements and complex operations on the lunar surface and high reliance on proven systems in anticipation of lowering hardware development costs. Key features of the First Lunar Outpost activity included:

- An evolutionary approach with emphasis on minimizing operational complexity;
- Initial missions' reliance on proven operational approaches and technologies;
- Graceful incorporation of advanced operational and technology concepts into downstream missions;
- Initial exploratory lunar landings at a few sites prior to lunar outpost location selection;
- An HLLV with a 200-mT delivery capability;
- A mission strategy of a direct descent to the lunar surface and direct return to Earth;
- Large pre-integrated systems designed for immediate occupancy by the crew of four;
- Simulation of ground and planetary operations for future Mars missions; and
- Engineering evaluation of transportation and surface systems with Mars mission applications.

4.2.1.1.6 Human Lunar Return (1996)

In September 1995, the NASA Administrator challenged engineers at JSC to develop a human lunar mission approach, the Human Lunar Return (HLR) study, which would cost significantly less (by one to two orders of magnitude) than previous human exploration estimates. Key objectives of the HLR activity were to demonstrate and gain experience on the Moon with those technologies required for Mars exploration, initiate a low-cost approach for human exploration beyond LEO, establish and demonstrate technologies required for human development of lunar resources, and investigate the economic feasibility of commercial development and utilization of those resources. The HLR study served as a radical approach from previous missions as evidenced by the "open cockpit" approach for the human lunar landers, reliance on existing small-capacity LVs (Proton and Shuttle), utilization of the Space Station as a staging node, and limited crew size (two) and short duration (3 days on the lunar surface). Activities associated with the HLR effort ended on August 7, 1996—the same day that scientists announced they had found evidence of ancient life in a meteorite from Mars. The HLR study represents the minimum mission approach for a return to the Moon capability.

4.2.1.1.7 Mars Exploration Design Reference Missions (1994–1999)

From 1994 to 1999, the NASA exploration community conducted a series of studies focused on the human and robotic exploration of Mars. Key studies included Mars Design Reference Mission (DRM) 1.0 (number 8 in **Section 4.5**, **Endnotes**), Mars DRM 3.0 (number 9 in **Section 4.5**, **Endnotes**), Mars Combo Lander, and Dual Landers (number 10 in **Section 4.5**, **Endnotes**). Each subsequent design approach provided greater fidelity and insight into the many competing needs and technology options for the exploration of Mars. Key mission aspects of each of these studies included the following:

- <u>Mission Mode:</u> Each of the Mars mission studies during this period employed conjunction-class missions, often referred to as long-stay missions, to minimize the exposure of the crew to the deep-space radiation and zero-gravity environment, while at the same time maximizing the scientific return from the mission. This is accomplished by taking advantage of optimum alignment of the Earth and Mars for both the outbound and return trajectories by varying the stay-time on Mars rather than forcing the mission through non-optimal trajectories as in the case of the short-stay missions. This approach allows the crew to transfer to and from Mars on relatively fast trajectories, on the order of 6 months, while allowing them to stay on the surface of Mars for a majority of the mission, on the order of 18 months.
- <u>Split Mission:</u> The surface exploration capability is implemented through a split mission concept in which cargo is transported in manageable units to the surface, or Mars orbit, and checked out in advance of committing the crews to their mission(s). Emphasis is placed on ensuring that the STSs could be flown in any Mars injection opportunity. This is vital in order to minimize the programmatic risks associated with funding profiles, technology development, and system design and verification programs.
- <u>Heavy-Lift Launch:</u> HLLVs were utilized in each of these studies due to the large mission mass for each human mission to Mars (on the order of the ISS at assembly complete) as well as the large-volume payloads required.
- <u>Long Surface Stay:</u> Emphasis was placed on the surface strategy associated with each mission approach. Use of conjunction-class missions provides on the order of 500 days on the surface of Mars for each human mission.

In order to view its lunar mission design work in the larger context of a future human Mars mission, the ESAS team chose Mars DRM 3.0 as the baseline Mars mission. This choice allowed the ESAS team to choose technologies, spacecraft designs, LVs, and lunar operational demonstrations that were extensible to future Mars missions.

4.2.1.1.8 Decadal Planning Team/NASA Exploration Team (2000–2002)

In June 1999, the NASA Administrator chartered an internal NASA task force, termed the Decadal Planning Team (DPT), to create a new integrated vision and strategy for space exploration. The efforts of the DPT evolved into the Agency-wide team known as the NASA Exploration Team (NExT) (number 11 in **Section 4.5**, **Endnotes**). This team was also instructed to identify technology roadmaps to enable the science-driven exploration vision and establish a cross-Enterprise, cross-Center systems engineering team with emphasis focused on revolutionary, not evolutionary, approaches. The strategy of the DPT and NExT teams was to "Go Anywhere, Anytime" by conquering key exploration hurdles of space transportation, crew health and safety, human/robotic partnerships, affordable abundant power, and advanced space systems performance. Early emphasis was placed on revolutionary exploration concepts such as rail gun and electromagnetic launchers, propellant depots, retrograde trajectories, nanostructures, and gas core nuclear rockets. Many of these revolutionary concepts turned out to be either not feasible for human exploration missions or well beyond expected technology readiness for near-term implementation. Several architectures were analyzed during the DPT and NExT study cycles, including missions to the Earth-Sun Libration Point (L2), the Earth-Moon Gateway and L1, the lunar surface, Mars (short and long stays), near-Earth asteroids, and a 1-year round trip to Mars. Common emphases of these studies included utilization of the Earth-Moon Libration Point (L1) as a staging point for exploration activities, current (Shuttle) and near-term launch capabilities (Evolved Expendable Launch Vehicle (EELV)), advanced propulsion, and robust space power. Although much emphasis was placed on the utilization of existing launch capabilities, the teams concluded that missions in near-Earth space were only marginally feasible, and human missions to Mars were not feasible without a heavy-lift launch capability. In addition, the teams concluded that missions in Earth's neighborhood, such as lunar missions, can serve as stepping-stones toward further deep-space missions in terms of proving systems, technologies, and operational concepts.

4.2.1.1.9 Integrated Space Plan (2002–2003)

During the summer of 2002, the NASA Deputy Administrator chartered an internal NASA planning group to develop the rationale for exploration beyond LEO. This team, termed the Exploration Blueprint team (number 12 in **Section 4.5**, **Endnotes**), performed architecture analyses to develop roadmaps for accomplishing the first steps beyond LEO through the human exploration of Mars. The previous NExT activities laid the foundation and framework for the development of NASA's Integrated Space Plan. The reference missions resulting from the analysis performed by the Exploration Blueprint team formed the basis for requirements definition, systems development, technology roadmapping, and risk assessments for future human exploration beyond LEO. Emphasis was placed on developing recommendations for what could presently be done to affect future exploration activities. The Exploration Blueprint team embraced the "stepping-stone" approach to exploration, where human and robotic activities are conducted through progressive expansion outward beyond LEO. Results from this study produced a long-term strategy for exploration with near-term implementation plans, program recommendations, and technology investments. Specific results included the development of a common exploration crew vehicle concept, a unified space nuclear strategy, focused bioastronautics research objectives, and an integrated human and robotic exploration strategy. Recommendations from the Exploration Blueprint team included the endorsement of the Nuclear Systems Initiative, augmentation of the bioastronautics research, a focused space transportation program including heavy-lift launch and a common exploration vehicle design for ISS and exploration missions, and an integrated human and robotic exploration strategy for Mars.

Following the results of the Exploration Blueprint study, the NASA Administrator asked for a recommendation by June 2003 on the next steps in human and robotic exploration in order to put into context an updated Integrated Space Transportation Plan (post-Columbia) to guide Agency planning (number 13 in **Section 4.5**, **Endnotes**). NASA was on the verge of committing significant funding to programs that would be better served if longer term goals were more evident, including the Orbital Space Plane (OSP), research on the ISS, the National Aerospace Initiative, the Shuttle Life Extension Program, Project Prometheus, and a wide range of technology development throughout the Agency. Much of the focus during this period was on integrating the results from the previous studies into more concrete implementation strategies in order to understand the relationship between NASA programs, timing, and resulting budgetary implications. This resulted in an integrated approach, including lunar surface operations to retire the risk of human Mars missions, the maximum use of common and modular systems including what was termed the "Exploration Transfer Vehicle," Earth orbit and lunar surface demonstrations of long-life systems, collaboration of human and robotic missions to vastly increase mission return, and high-efficiency transportation systems (nuclear) for deep-space transportation and power.

4.2.1.1.10 Exploration Systems Mission Directorate (2004)

On January 14, 2004, the President announced a new Vision for Space Exploration (henceforth referred to here as the Vision). In his address, the President presented a bold, forward-thinking, practical, and responsible vision—one that will explore answers to long-standing questions of importance to science and society and will develop revolutionary technologies and capabilities for the future while maintaining good stewardship of taxpayer dollars.

NASA's Exploration Systems Mission Directorate (ESMD) was created in January 2004 to begin implementation of the President's Vision. ESMD's Requirements Division conducted a formal requirements formulation process in 2004 to understand the governing requirements and systems necessary for implementing the Vision. Included in the process were analyses of requirements definition, exploration architectures, system development, technology roadmaps, and risk assessments for advancing the Vision (numbers 14 and 15 in **Section 4.5**, **Endnotes**). The analyses provided an understanding of what is required for human space exploration beyond LEO. In addition, these analyses helped identify system "drivers" (i.e., significant sources of cost, performance, risk, and schedule variation, along with areas needing technology development).

The requirements development process was initiated through the development of strategic campaigns that represent a range of potential approaches for implementing the Vision, specifically initial lunar missions that support long-term exploration endeavors. These strategic campaigns, often referred to as "architectures," were derived directly from the Vision. The leading candidate architectural options were then studied in some detail in order to understand the sensitivity and relationships between the mission, system, and technology concepts within a feasible option. The analysis activities resulted in what was termed a "Point-of-Departure (POD) Architecture" to be used for further refinement as the Agency progressed toward the

System Requirements Review (SRR). Key features of the ESMD architecture include the following.

- Lunar Landing Sites: Emphasis was placed on developing an exploration architecture that would provide global access for short-duration missions. As the length of stay on the lunar surface was increased (up to 98 days), landing sites were limited to either the lunar poles or the equator due to the desire to retain the ability to return the crew to Earth without the need to wait for proper orbital alignment of transportation elements.
- Mission Mode: Several different staging strategies were examined, resulting in the selection of EOR (required for multiple launches) and LOR. The latter provided the best balance between overall mission mass and the capability for global access for short missions, as well as support for long-duration missions.
- Propulsion: Advanced chemical propulsion was determined to be a key element of the ESMD lunar exploration architecture. Propellant preferences included oxygen/hydrogen propulsion for the outbound mission phase and oxygen/methane for the lunar landing, lunar ascent, and Earth return phases.
- Earth Landing: Direct entry at Earth return was selected with water landing as the primary mode.
- Multi-Mission: Emphasis was also placed on developing a transportation system that could meet a range of other potential mission modes. Although not specifically required by ESMD at the time, the study showed that missions to the ISS, lunar libration points, and a staging point for Mars missions could be accommodated.
- LV: Many different LV concepts were studied, ranging from utilizing existing LVs (EELVs), to heavy-lift concepts derived from existing systems (EELVs and Shuttle), to completely new concepts. During this period, ESMD ruled out new clean-sheet concepts from an affordability perspective but made no other firm decisions on the LV to be used for the exploration architecture.

In addition to its in-house work, ESMD awarded a series of eleven CE&R contracts, with the goal of obtaining a broad set of vehicle concepts, mission architecture designs, and technology rankings from a diverse set of contractors that spanned the continuum from large and familiar aerospace corporations to small and aggressive aerospace entrepreneurs. The contractors were initially given only endpoint milestones and asked to assemble programs of vehicles and missions to achieve the endpoints. Several of the mission concepts produced by the CE&R contractors led directly to options studied further by the ESAS team.

4.2.1.2 Key Findings from Previous Architecture Studies

4.2.1.2.1 Strategy of Progressive Expansion

One common finding from the many previous studies is the stepping-stone approach to exploration that embraces a progressive expansion of human exploration beyond LEO. Under a stepping-stone approach, implementation is initiated with the establishment of key technical capabilities needed for the first step in the journey to deep space. The stepping-stone approach will build the technical capabilities needed for each step with multi-use technologies and capabilities. Each step will build on the previous step to avoid redevelopment of critical systems. The approach capitalizes on progressive exploration capabilities, where the experience and infrastructure gained from each step enables travel to new destinations.

4.2.1.2.2 Earth-to-Orbit (ETO) Transportation—Heavy-Lift

The various architecture studies over the years have emphasized the use of differing LV implementations, ranging from existing or near-existing capabilities, such as the EELV, to newly-designed HLLVs. These studies have shown that, even with the application of advanced technologies, exploration missions require significant initial mass in LEO, on the order of the mass of the ISS at assembly complete (470 mT). Missions in near-Earth vicinity, such as lunar missions, range from 120–220 mT, while Mars missions range from 400–800 mT or more for each human mission. Reducing the number of launches and the corresponding on-orbit assembly requirements can significantly reduce the overall cost and risk of human exploration missions. LV shroud volume is another key requirement for exploration missions. The diameter of the launch shroud has a profound influence on the design of the overall architecture, most importantly on the design of the lander systems. Delivery of the landing vehicle to LEO poses a significant challenge due to its large size (volume) for both lunar and Mars missions and the additional complexity of the aerodynamic shape of the lander required for Mars entry. The range of architecture studies has also shown that crew delivery and return for exploration missions is very similar to ISS crew return needs, and, thus, there is great potential for architecture synergy between LEO and beyond-LEO mission needs for crew delivery.

These studies have shown that architectures in near-Earth space utilizing near-term launch capabilities (e.g., EELV) are marginally feasible, operationally challenging, and very complex, whereas utilizing EELVs for Mars missions is not feasible due to the excessive number of launches required. These previous studies have shown that exploration launch needs (e.g., payload mass and volume) for near-Earth and Mars can be met with concepts evolved from Shuttle systems.

4.2.1.2.3 Crew Transportation—Common Vehicle

Another common thread through each of the exploration studies conducted over the past few years is the recognition of the applicability of a common vehicle design for many of the near-Earth exploration destinations. The performance requirements associated with missions to the ISS, lunar orbit, Earth-Moon libration points, and various Mars mission staging points are very similar in terms of overall mission duration, crew size requirements, and basic transportation payload capabilities. Entry speed at Earth return is the one key discriminator between the various near-Earth destinations, all of which can be satisfied through the choice of a mid-lift/drag vehicle design. Pursuit of this common vehicle design is the key to enabling a robust exploration capability beyond LEO.

4.2.1.2.4 Key Capabilities and Core Technologies

Previous NASA architecture studies have included such destinations as the Moon, near-Earth asteroids, Mars, and the moons of Mars. A review of these previous studies illustrates the existence of a common thread of key capabilities and core technologies that are similar between destinations. All of the technologies listed below do not need to be developed at the same time, but rather should be initiated consistent with the overall implemented exploration strategy.

Human Support

Human support technologies to be developed include the following:

- Radiation Protection: Protecting the exploration crew from both galactic cosmic radiation (remnants from the formation of the universe) and SPEs (solar flares from the Sun);
- Medical Care: Providing advanced medical diagnostic and treatment equipment to the crew in-situ and the corresponding data to the medical teams on Earth;
- Advanced Life Support: Advances in high-reliability, low-maintenance life support technologies are necessary to reduce the consumables required to support early human exploration missions; and
- Human Adaptation and Countermeasures: Advances to counter effects of long-duration space travel including bone decalcification, immune and cardiovascular system degradation, and other deleterious effects.

Transportation

Transportation technologies to be developed include the following:

- Low-Cost, Large Payload ETO: Providing the capability to efficiently and affordably deliver large payloads, in terms of mass and volume, to LEO;
- Advanced Chemical Propulsion: Highly efficient, restartable, and throttleable cryogenic main engines, which provide evolution potential to utilize locally produced propellants;
- Cryogenic Fluid Management (CFM): Providing the capability to manage large quantities of cryogenic fluids such as hydrogen, oxygen, and methane for long periods;
- Hazard Avoidance and Precision Landing: Precision landing and hazard avoidance technologies are also needed for planetary lander vehicles;
- AR&D: Providing the capability to perform rendezvous and docking of multiple elements in remote locations with limited or no support from ground or flight crews;
- Advanced Deep-Space Propulsion: Advanced propulsion concepts including solar electric, nuclear electric, and nuclear thermal propulsion are necessary to reduce the total mission mass for future human missions to Mars; and
- Aeroassist: Providing the capability for entry, descent, and soft landing of large systems is necessary for future human exploration of the Martian surface.

Power Systems

Power systems technologies to be developed include the following:

- Power Storage: Includes power generation, distribution, and control evolving from early exploration capabilities (10s kWe) to longer-term permanent human presence (1 MWe); and
- Power Management and Distribution: Power distribution needs for human exploration include efficient high-power distribution technologies and intelligent, self-diagnosing, and correcting power management and distribution systems.

Miscellaneous

Miscellaneous technologies to be developed include the following:

- Advanced EVA: Technologies that enable routine surface exploration are critical to exploration activities. This includes advanced EVA suits and short- and long-range rovers for surface exploration;
- Advanced Thermal Protection: This includes Thermal Protection Systems (TPSs) that can withstand the temperature extremes of lunar and Mars return missions;

- ISRU: Technologies for "living off the land" are needed to support a long-term strategy for human exploration. Key ISRU challenges include resource identification and characterization, excavation and extraction processes, consumable maintenance and usage capabilities, and advanced concepts for manufacturing other products from local resources; and
- Supportability: Required levels of operational availability and autonomy of spacecraft systems engaged in long-duration human exploration missions will be achieved in a combination of high reliability, adequate redundancy, and maintainability.

4.2.1.3 Applying the Results of Past Studies to ESAS

The ESAS team was fortunate to have the combined wisdom of the aerospace age on which to build. The team established "required reading" documents that included many of the studies performed in the early 1960s in support of the Apollo Program, many of the internal NASA human mission studies performed over the past 15 years, and the results of in-house and contracted studies performed most recently for ESMD. Many of the ESAS team members were steeped in the historical roots of the space program, but also represented the analytical core of modern NASA, with the ability to apply the latest tools and techniques to the analysis of vehicles, flight mechanics, reliability, safety, and cost.

4.2.2 Mission Mode Option Space

The lunar mission mode option space considered the location of "nodes" in both cislunar space and the vicinity of Earth. The study originally considered cislunar nodes at the Earth-Moon L1 libration point, in LLO, and on the lunar surface. Respectively, these translate to Libration Point Rendezvous (LPR), LOR, and Lunar Surface Rendezvous (LSR) mission modes. The study also considered Earth-orbital staging locations in LEO, higher-inclination ISS orbits, and raised-apogee HEO. In all three cases, elements brought together in any type of Earth orbit were generically termed an EOR mission mode. In the case of both cislunar and Earth orbital nodes, a mission type that bypassed a node completely was termed a "direct" mission or the term for the bypassed node was omitted altogether. Therefore, the Apollo missions were "direct" injection from Earth to the Moon, due to there being no EOR activities, and they were LOR at the Moon, owing to the rendezvous of the Command Module and lunar module (LM) following the surface mission. The Apollo mission mode was therefore popularly referred to as LOR.

LPR was eliminated early from the mission mode trade space. Recent studies performed by NASA mission designers concluded that equivalent landing site access and "anytime abort" conditions could be met by rendezvous missions in LLO with less propulsive delta-V and lower overall Initial Mass in Low Earth Orbit (IMLEO). If used only as a node for lunar missions, the L1 Earth-Moon LPR is inferior to the LOR mission mode.

With LPR eliminated, the mission mode question could be illustrated in a simple 2x2 matrix with the axes indicating the existence (or not) of an Earth-orbital and lunar-orbital node. The mission mode taxonomy could then be associated with each cell in this matrix—a mission that required EOR as well as rendezvous in lunar orbit was termed "EOR–LOR." A mission that injected directly to the Moon (bypassing Earth-orbital operations) and returned directly from the surface of the Moon (bypassing lunar-orbital operations) was termed "direct-direct." **Figure 4-1** illustrates the lunar mission mode matrix.

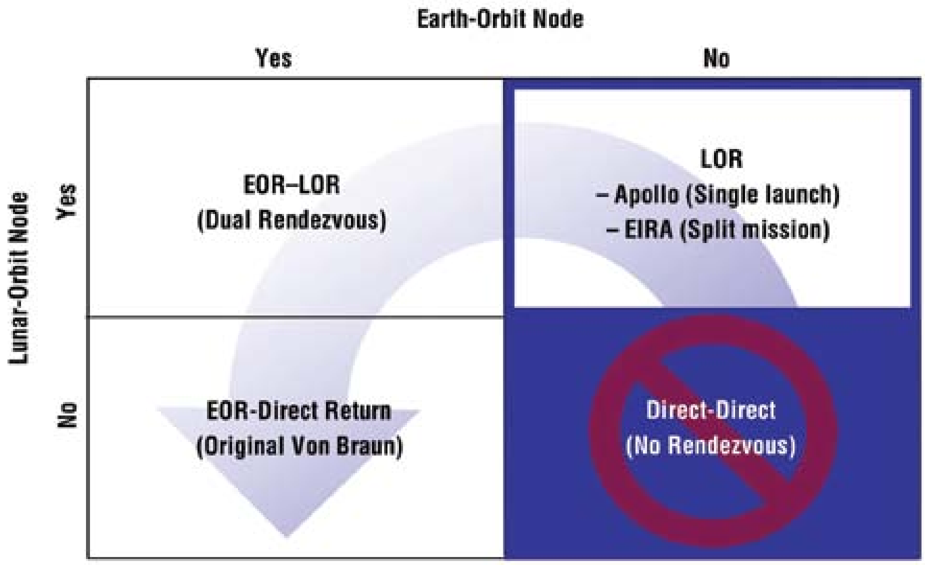


Figure 4-1. Lunar Mission Mode Taxonomy

This matrix becomes clearer when additional descriptions and certain historical lunar missions are added to the respective quadrants. The EOR-direct return mission (lower left-hand quadrant) was the mode favored by Wernher Von Braun early in the Apollo Program, while LOR (upper right-hand quadrant) was the mode eventually chosen. It became clear early in the ESAS analysis that the direct-direct mode (lower right-hand quadrant) would only be possible if the single LV it required had performance capability approaching 200 mT to LEO. Because no LVs of this size were contemplated for this study due to budget and ground operations constraints, direct-direct was eliminated as a mission mode. The three remaining mission modes (LOR, EOR–LOR, and EOR-direct return) will be analyzed in the following sections.

4.2.3 Analysis Cycle 1 Mission Mode Analysis

Mission mode analysis was performed in three cycles, with each cycle resulting in performance, cost, reliability, safety, and other Figures of Merit (FOMs) with which to compare the mission options. At the end of each analysis cycle, decisions were made to eliminate certain mission modes or to perform additional studies to further drive out the differences among the options. A baseline was chosen against which all design options could be compared. The baseline chosen by the ESAS team was a LOR split mission termed the ESAS Initial Reference Architecture (EIRA). The EIRA is explained in more detail in **Section 4.2.3.1**, **Definition of EIRA**.

For the initial analysis cycle, the EIRA mission was compared to EOR–LOR, EOR-direct return, and a variant of EOR–LOR that took the CEV to the lunar surface (similar to a direct return mission) but left the TEI propulsion in lunar orbit. Only the mission mode was varied in this first cycle. In all cases other than the original EIRA, the CEV was a 5.5-m diameter, 25 deg sidewall-angle capsule with 1,400 kg of radiation shielding. All post-TLI propulsion used pressure-fed LOX/methane engines and all lunar landers were expendable, two-stage configurations without airlocks.

The team generated mission performance analysis for each option (IMLEO, number of launches required, and launch margins), integrated program costs through 2025, safety and reliability estimates (probability of loss of crew (P(LOC)), and probability of loss of mission (P(LOM))), and other discriminating FOMs.

4.2.3.1 Definition of EIRA

Prior to beginning Analysis Cycle 1, the ESAS team created an initial reference architecture that would serve as the basis for initial trade studies. The team recognized that this EIRA would likely not be the optimum mission architecture, but would represent a solution that met all of the Ground Rules and Assumptions (GR&As) set forth for the study.

The EIRA timeline showed human crews returning to the Moon in 2018 with up to 7-day sortie missions continuing into the outpost deployment phase in 2020–2021. These initial sorties took a crew of four to any site on the lunar globe and included EVA on each day of the surface stay. The crew of four would explore the lunar surface in two teams of two crew members each, aided by unpressurized rovers for local mobility. Their payload complement would include science packages as well as exploration technology experiments. A minimum of two lunar sortie missions was planned for each year. The EIRA sortie mission is shown in **Figure 4-2**.

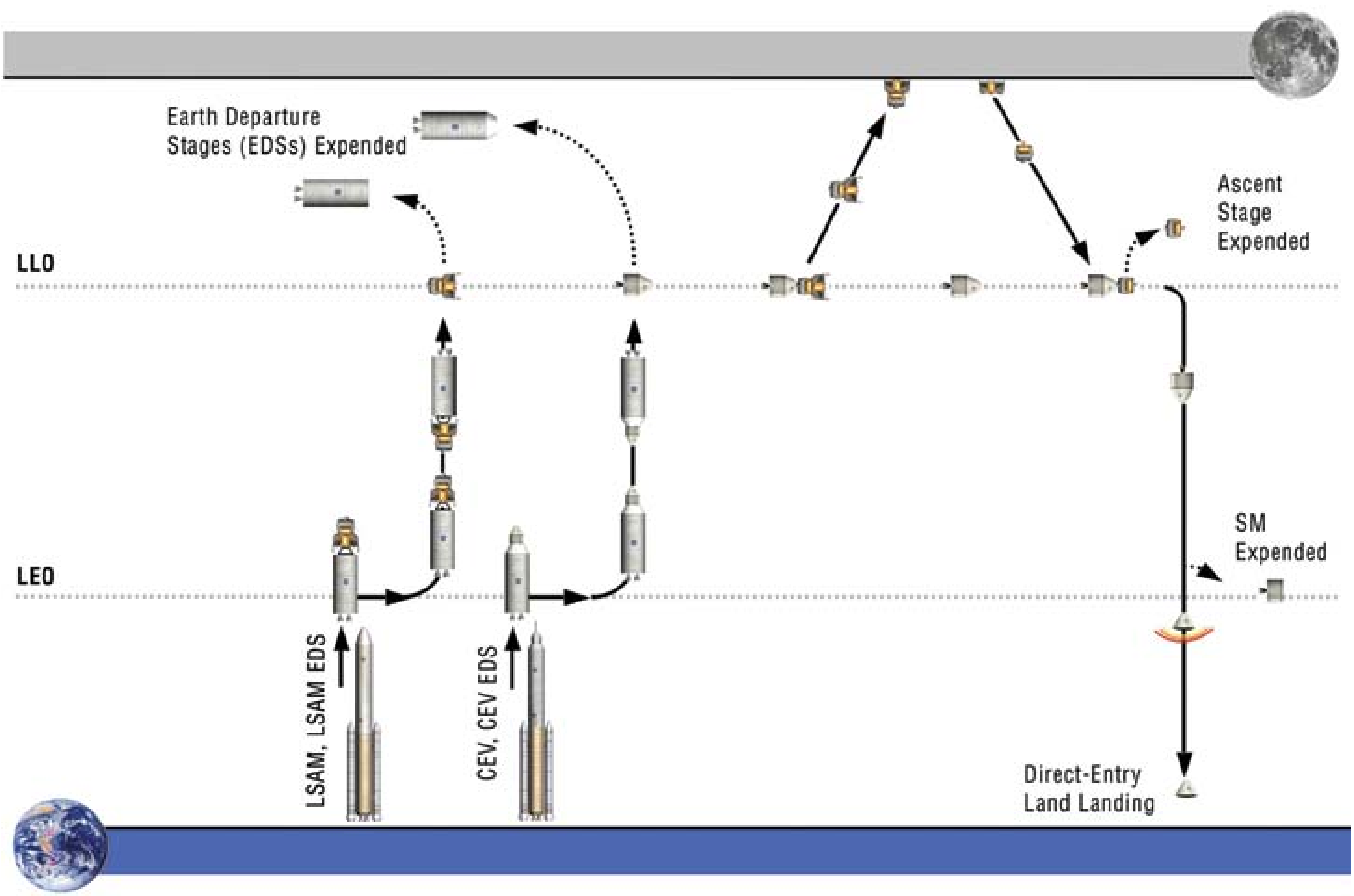


Figure 4-2. EIRA

Beginning in 2020 and extending into 2021, a series of dedicated cargo landers would deliver the elements of a permanent outpost. These elements, including a power system, habitat, and resource utilization equipment, would be deployed with the aid of robotic systems. Just prior to the arrival of the first crew, a "backup" ascent vehicle would be landed at the outpost to give the crew a redundant means of transportation off the lunar surface. The outpost would be in an Initial Operational Capability (IOC) and ready to receive the first crew by 2022.

The first outpost crew would arrive in 2022 for a 6-month rotation on the lunar surface. Subsequent crews would arrive every 6 months thereafter for the duration of the outpost's operational lifetime. As steady-state operation of the outpost would also include a logistics delivery mission spaced midway through each crew rotation, the outpost would receive two crew landings and two logistics landings each year. Logistics flights could deliver up to 15 mT of cargo, including substantial resource utilization hardware and pressurized rovers to increase the crew's mobility range. Outpost crews would continue scientific studies but would concentrate more on resource utilization and demonstration of Mars technologies and operational techniques. The outpost would remain in steady-state operation through at least 2030.

4.2.3.2 Trade Studies

Analysis Cycle 1 trade studies for the lunar architecture were intentionally limited to mission mode differences only. Using the EIRA mission as a baseline, the EIRA–LOR mission was compared to EOR–LOR, EOR-direct return, and a variant of EOR–LOR that would take the CEV to the lunar surface. In three of the four cases, the CEV was increased from a 5.0- to 5.5-m diameter, owing to parallel CEV volumetric and configuration studies that were ongoing at the same time. The mission mode differences also demanded slightly different splits of propulsive maneuvers among the flight elements to balance launch masses. The EOR–LOR variant performed the LOI maneuver using the lander's descent stage, while all other options remained attached to the Earth Departure Stage (EDS) stage throughout the trans-lunar coast in order for that stage to perform LOI. The EIRA mission architecture and four options are highlighted in **Figure 4-3**.

This initial mission mode analysis therefore varied four parameters: the use of a lunar-orbit node (LOR versus a direct return from the Moon), the use of an Earth-orbit node (a "split mission" versus EOR), CEV capsules of varying shapes and volumes, and different TLI/LOI/TEI splits among propulsive elements.

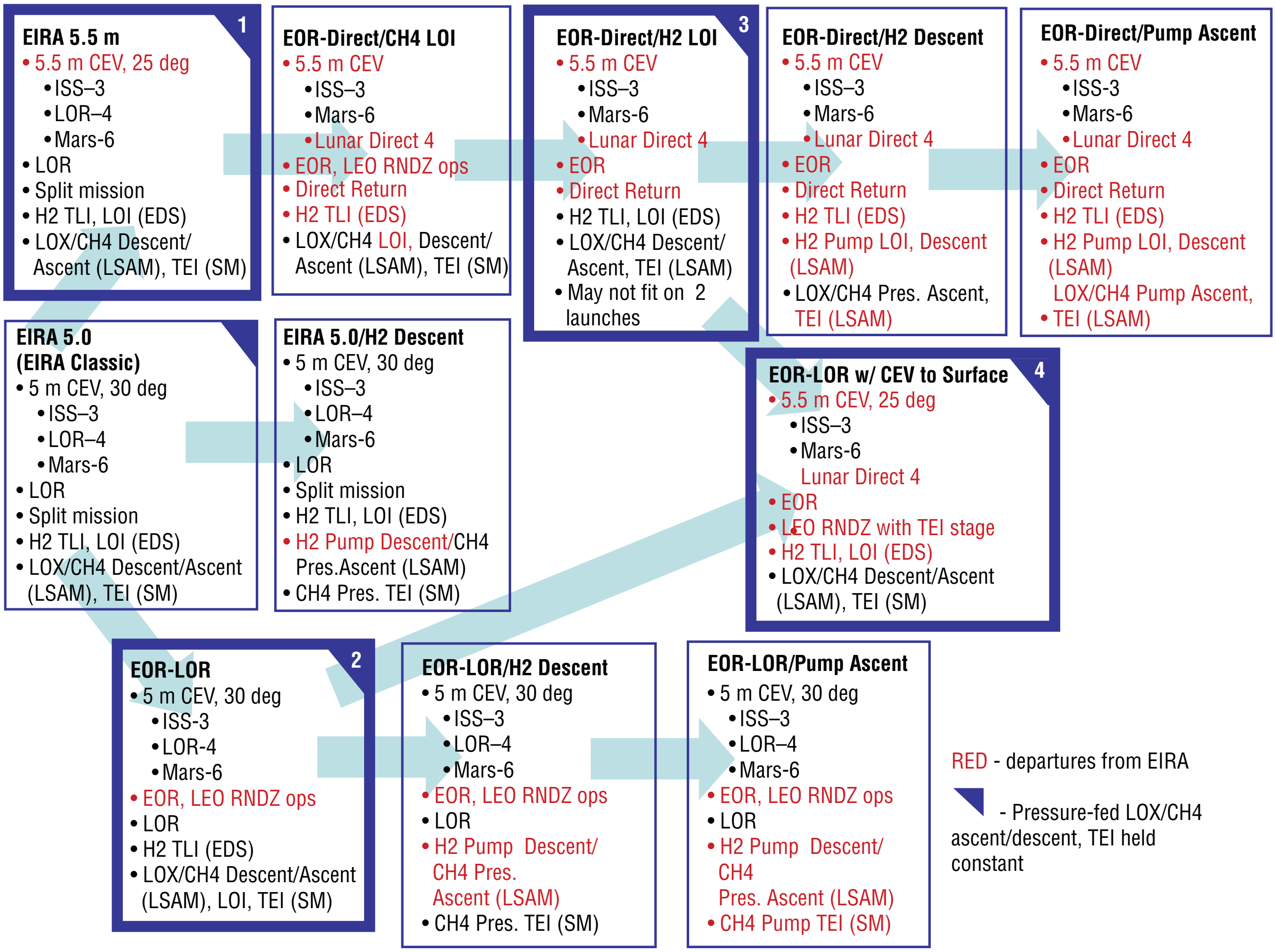


Figure 4-3. Analysis Cycle 1 Mission Architecture Flow

4.2.3.2.1 CEV and Lunar Surface Access Module (LSAM) Volume Studies

An understanding of the mass of the crew-carrying elements of the architecture, the CEV and the LSAM lunar lander, were fundamental to the analysis effort. Analytical tools used by the ESAS team required the pressurized volume of these vehicles as an input. The team investigated historical spacecraft volumes, drew on current human factors research, and involved current astronauts in the evaluation of full-scale mockups to produce recommendations. These results are presented in **Section 5**, **Crew Exploration Vehicle**.

CEV Task Analysis

The initial analysis to determine the minimum net habitable volume in the CEV assumed simultaneous suit donning/doffing of the Shuttle Advanced Crew Escape Suit (ACES) as the baseline. Assumptions were made as to the volume required for one crew member to don/doff the suit, and then this volume was multiplied by 4 for the CEV lunar mission case. This initial analysis also assumed that simultaneous suiting was a stand-alone activity and no other activities would happen while donning/doffing of suits was in progress. Therefore, additional habitable volume for these other tasks was not a consideration in the initial analysis.

A group of 15 astronauts, all with spaceflight experience, including Space Shuttle, Space Station, Mir, and Soyuz, unanimously agreed that simultaneous suit donning/doffing was operationally inefficient, and that assisted suiting is a faster and more efficient use of available volume. This group also agreed the suiting would never be the single activity occupying the entire crew at the same time.

A low-fidelity mockup of the Cycle 4 CEV volume was built at JSC and evaluated by this group of experienced astronauts in a completely subjective fashion to determine if the net habitable volume, as laid out in the point solution of internal layout of systems and seats, seemed sufficient for zero-g activities based on their collective experiences. The measured net habitable volume of the mockup was 14.9 m^3, and there was unanimous agreement that this net habitable volume was sufficient. At this point in the design cycle, the group was unwilling to recommend any volume number smaller than that measured in this mockup—given the uncertainties of where and how systems/seats would be configured and how much of the pressurized volume they would occupy, as well as uncertainties in the operational tasks required for lunar transit missions and the tools necessary to execute those tasks.

4.2.3.3 Performance

The first ESAS architecture design cycle evaluated the performance of four competing human lunar mission architectures. Data generated from this analysis, including vehicle mass properties, mission critical events, and number and type of launches, was subsequently used to inform cost, safety, reliability, and other related FOM comparisons. The first architecture alternative was a mission consisting only of vehicle rendezvous and docking occurring in LLO. This mission mode was selected as the EIRA against which other architectures were measured. The next architecture was a variant from the reference in that the initial rendezvous between the CEV and the LSAM occurred not in lunar orbit but in Earth orbit. Another variant also included vehicle rendezvous and docking in Earth orbit, but the vehicles landed directly on the Moon after leaving LEO and returned directly to Earth. The fourth and final architecture was a variant on the second in that, instead of having two dedicated Crew Modules (CMs) (one for transit to and from the Moon and one for lunar operations), this architecture used a single CM for the entire mission.

As introduced in **Section 4.2.2**, **Mission Mode Option Space**, mission architectures are identified according to the following nomenclature.

- LOR EIRA;
- EOR–LOR;
- EOR-direct return; and
- EOR–LOR with CEV-to-surface.

Subsequent sections of this report describe the performance, operation, and other salient features of these architectures.

4.2.3.3.1 EIRA LOR

The assumed mission mode for the EIRA is a 2-launch "split" architecture with LOR, wherein the LSAM is predeployed in a single launch to LLO, and a second launch of the same vehicle delivers the CEV and crew in lunar orbit, where the two vehicles initially rendezvous and dock. The entire crew then transfers to the LSAM, undocks from the CEV, and performs a descent to the surface. The CEV CM and SM are left unoccupied in LLO. After up to 7 days on the lunar surface, the LSAM returns the crew to lunar orbit and docks with the CEV, and the crew transfers back to the CEV. The CEV then returns the crew to Earth with a direct-entry-and-land touchdown while the LSAM is disposed of on the lunar surface. This mission mode is illustrated in **Figure 4-4**.

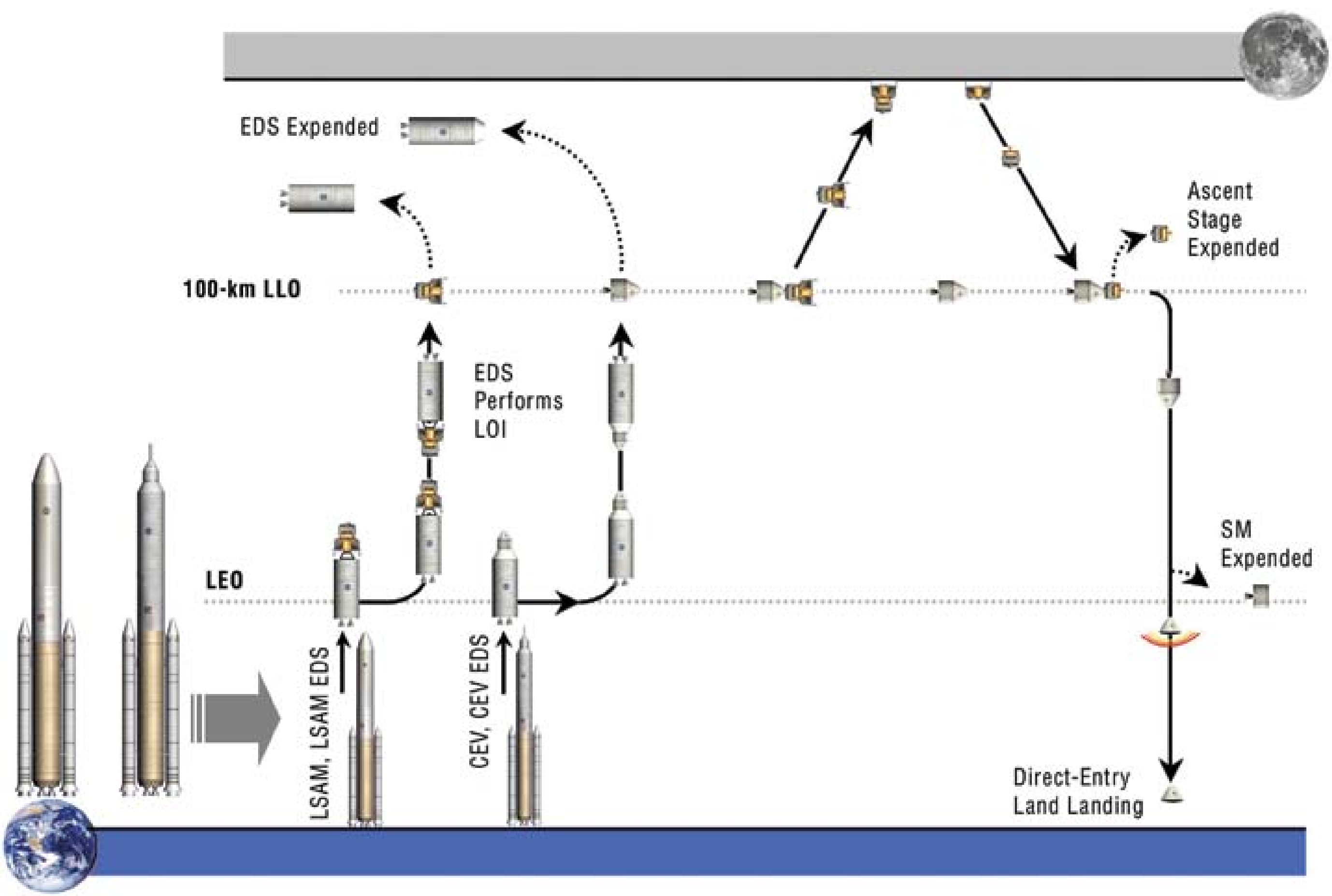


Figure 4-4. EIRA (Analysis Cycle 1)

The Analysis Cycle 1 CEV assumed in the EIRA is a four-person capsule with a base diameter of 5.0 m and 30-deg sidewall angle, providing a total pressurized volume of 22.4 m^3. The CEV provides 47 crew-days of nominal life support capability and includes 5 g/cm^2 of supplemental High-Density Polyethylene (HDPE) shielding on the capsule sidewalls and ceiling for radiation protection. The EIRA CEV SM is an unpressurized cylinder containing the primary vehicle propulsion and power generation systems. An integrated pressure-fed oxygen/methane propulsion system provides 1,772 m/s of orbital maneuvering and reaction control delta-V. For the EIRA, these maneuvers include rendezvous and docking with the LSAM in LLO, a 5-deg contingency ascent plane change, the TEI burn, and mid-course corrections. The SM includes two 22.2-kN (5-klbf) pressure-fed main engines and twenty-four 445-N (100-lbf) reaction control thrusters. The combined CEV mass in LEO following launch is 22,909 kg, with 9,623 kg allocated for the CM and 13,286 kg for the SM. The assumed LV and EDS for the EIRA can deliver a net payload of 29,100 kg to LLO.

The Analysis Cycle 1 LSAM transports four crew from LLO to the lunar surface, supports the crew for up to 7 days on the Moon, and returns the crew to the CEV in LLO. The assumed LSAM configuration includes a separate ascent and descent stage similar to the Apollo LM, with the ascent stage containing the LSAM crew cabin and mounted on top of the descent stage. As in Apollo, the nominal EVA mode is to fully depressurize the ascent stage crew cabin, open the hatch, and egress the vehicle. The crew cabin is a horizontal short cylinder providing 29.2 m^3 of pressurized volume. Propulsion for the ascent stage is similar to the CEV SM in that it uses the same propellants (oxygen/methane) and main/reaction control engines. This propulsion includes two 22.2-kN (5-klbf) pressure-fed main engines and sixteen 445-N (100-lbf) reaction control thrusters to perform 1,882 m/s of ascent and orbital maneuvering delta-V. The total ascent stage mass in LEO is 9,898 kg. The LSAM descent stage provides powered descent for the crew and ascent stage from LLO to the lunar surface. The propulsion system for the descent stage is similar to the CEV SM and includes four 22.2-kN (5-klbf) pressure-fed main engines and sixteen 445-N (100-lbf) reaction control thrusters to perform 1,917 m/s of powered descent and attitude control delta-V. The descent stage also carries 500 kg of mission payload, such as rovers and science equipment, to the surface. The descent stage wet mass is 18,010 kg, and the combined LSAM mass including the ascent stage is 27,908 kg. Since the same EDS used to deliver the CEV to lunar orbit is used for the LSAM, the net LSAM mass limit with 10 percent EDS performance reserve is 29,100 kg. Both the CEV and LSAM have positive mass margins relative to the EDS performance limit, thus making the EIRA a valid 2-launch mission.

The Analysis Cycle 1 EIRA also includes analysis of a larger-volume variant of the CEV. This larger CM assumes a base diameter of 5.5 m and 25-deg sidewall angle to provide a total pressurized volume of 39.0 m^3. With this extra volume, the CM mass increases from 9,623 kg to 11,332 kg, the SM mass increases to 14,858 kg, and the total CEV mass increases to 26,190 kg. The total CEV mass is still within the performance capabilities of the EDS.

4.2.3.3.2 EOR–LOR Architecture

The EOR–LOR architecture (**Figure 4-5**) is functionally similar to the EIRA, with the primary difference that the initial CEV-LSAM docking occurs in LEO rather than LLO. Whereas the EIRA incorporated two smaller EDSs in two launches to deliver the CEV and LSAM to the Moon, the EOR–LOR architecture divides its launches into one launch for a single, large EDS and a second launch for the CEV, crew, and LSAM. The combined CEV and LSAM dock with the EDS in Earth orbit, and the EDS performs TLI. Another difference between the EIRA and EOR–LOR architectures is that the EDS performs LOI for the EIRA. Due to launch performance limitations of the single EDS with EOR–LOR, LOI is instead executed by the CEV for optimum performance. Once the CEV and LSAM reach LLO, this mission mode is identical to the EIRA.

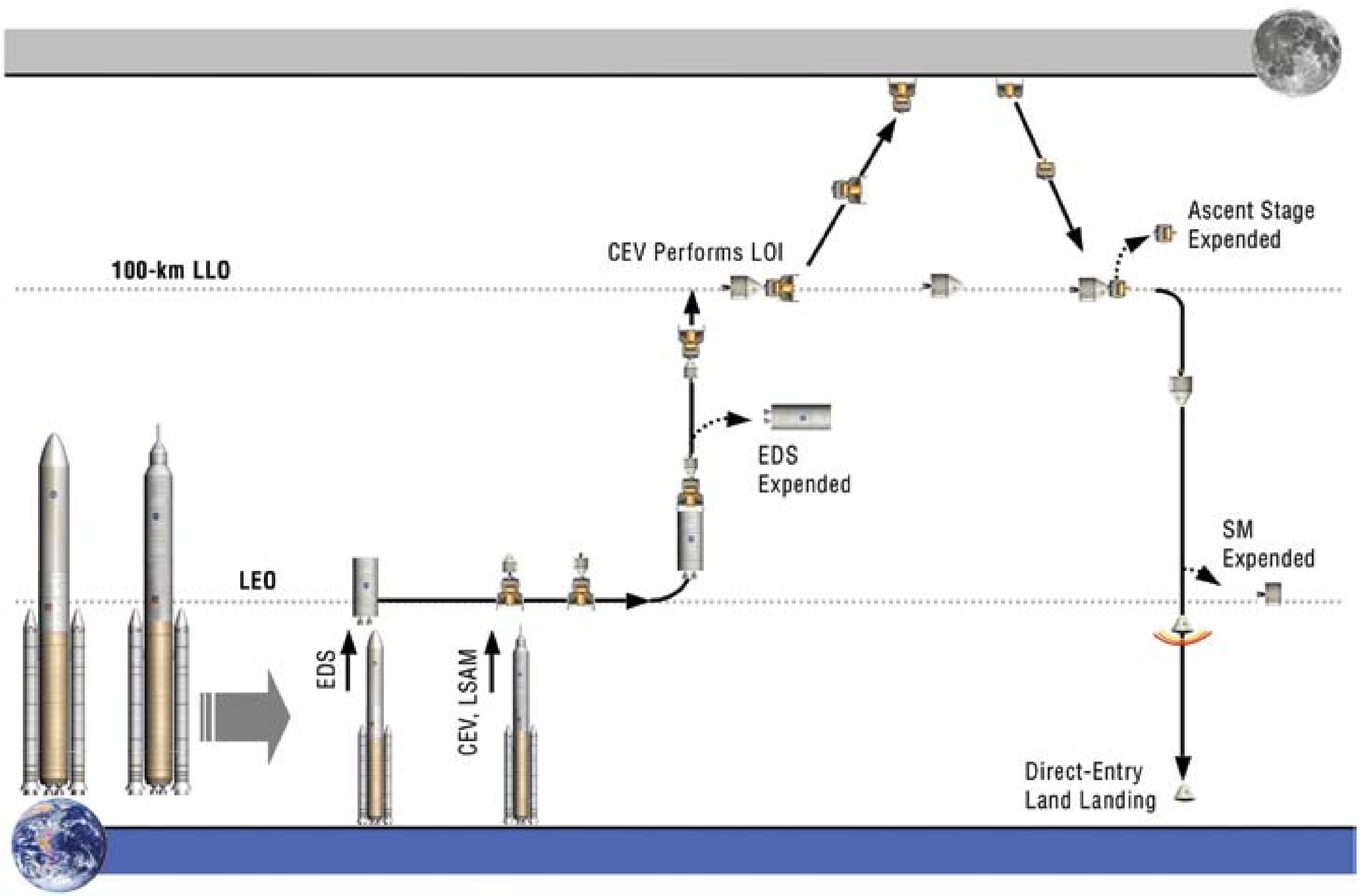


Figure 4-5. EOR–LOR Architecture (Analysis Cycle 1)

The same EIRA 5.0-m CEV CM has been retained for this architecture with one minor modification. The EOR–LOR CEV nominally requires 53 crew-days of life support capability while the EIRA requires 47 crew-days. This is due to the additional rendezvous and docking maneuvering required in LEO with EOR–LOR, whereas the EIRA is a direct injection mission, and the CEV EDS performs TLI within a few hours after reaching orbit.

The SM is also functionally similar to the EIRA with a few notable exceptions. For this architecture, the CEV is needed to perform the LOI maneuver to fit within the EDS TLI capabilities. As the LSAM is attached to the CEV at this point, the required propellant quantity in the SM is significantly higher than the EIRA. The EOR–LOR SM includes four 22.2-kN (5-klbf) pressure-fed main engines to perform 3,161 m/s of delta-V. Major CEV maneuvers in this architecture include transposition and docking with the LSAM in LEO, rendezvous and docking of the combined CEV and LSAM with the EDS, LOI, a 5-deg contingency ascent plane change, TEI, and return mid-course corrections. The SM mass at launch is 49,750 kg, and the total CEV mass is 59,445 kg combined with the CEV CM. This compares to 22,909 kg for the EIRA total CEV mass.

No modifications to the EIRA LSAM are assumed for this architecture. Therefore, the combined LSAM and CEV mass prior to TLI, including docking provisions and subtraction of LEO rendezvous propellant from the CEV, is 83,000 kg. The EDS assumed in this architecture can inject 85,600 kg to TLI, while the CLV can lift 91,300 kg to LEO.

4.2.3.3.3 EOR-Direct Return Architecture

The EOR-direct return architecture analyzed in Analysis Cycle 1 is a significant departure from the previous two options in that there are no rendezvous maneuvers needed to complete the mission once the CEV and LSAM (with no crew volume) depart LEO. LOR architectures rely on leaving some part of the return vehicle (CEV) in LLO while a dedicated lunar landing system transports the crew between lunar orbit and the lunar surface. The EOR-direct return architecture instead carries the entire Earth return system down to the lunar surface, thereby greatly simplifying the mission. Where the previous architectures required two crew cabins (the CEV and LSAM ascent stage crew cabins), the crew spends the entire mission in a single crew cabin (the CEV). However, this flexibility comes at the cost of added architecture mass in LEO. Using the same propulsion assumptions as in the LOR alternatives (pressure-fed oxygen/methane), this architecture requires a third heavy-lift launch to perform each mission.

The assumed mission mode for the EOR-direct return architecture (EIRA) is a 3-launch "all-up" architecture with EOR. Due to the excessive CEV–LSAM mass, each mission requires two EDSs to deliver the vehicles to LLO. The EDSs are launched prior to the crew and automatically docked in LEO. After the crew, CEV, and LSAM launch in the third launch, the vehicles dock to the EDSs and perform TLI. The first EDS is exhausted prior to completing TLI and is separated and disposed. The second EDS completes TLI and performs LOI 4 days later. Rather than undocking from the CEV and leaving the vehicle unoccupied in lunar orbit, the CEV and LSAM both land on the Moon. After up to 7 days on the lunar surface, the CEV returns the crew directly back to Earth with a direct-entry-and-land touchdown. This mission mode is illustrated in **Figure 4-6**.

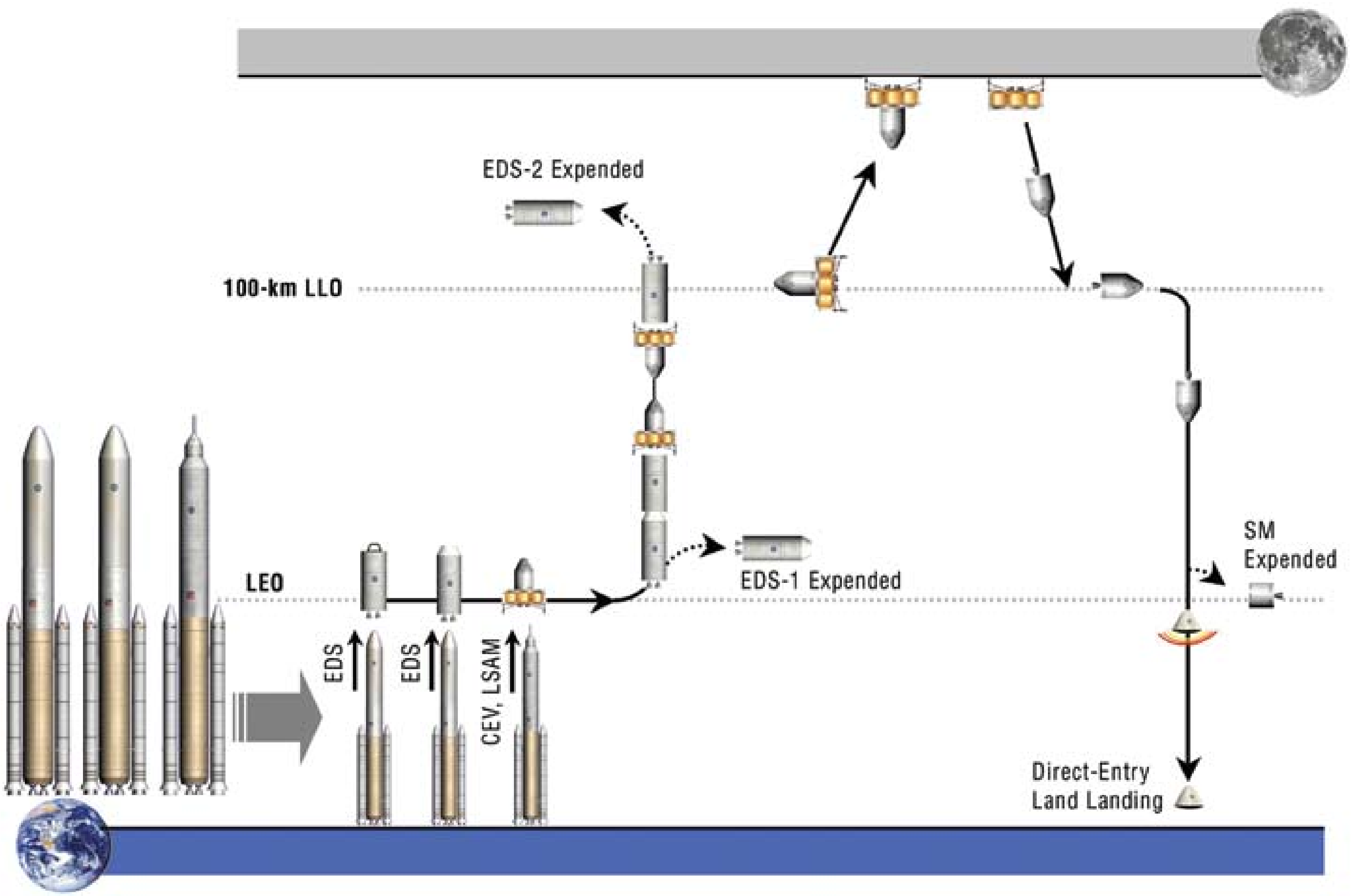


Figure 4-6. EOR-Direct Architecture (Analysis Cycle 1)

The larger 5.5-m, 25-deg sidewall angle CEV introduced in the EIRA description is used for analyzing the EOR-direct return architecture. This vehicle was needed to provide the necessary habitable volume and floor space for operating on the lunar surface for up to 7 days. The configuration provided 39.0 m^3 of pressurized volume. Other modifications include additional displays and controls for landing on and ascending from the lunar surface, a full-cabin depressurization capability for multiple surface EVAs, and additional life support capability for additional crew time spent in the CEV CM. The mass of the EOR-direct return CM is 11,653 kg at launch.

EOR-direct return combines the ascent function of the EIRA LSAM and the TEI function of the EIRA SM into the CEV. Three 44.5-kN (10-klbf) pressure-fed main engines and sixteen 445-N (100-lbf) reaction control thrusters perform 2,874 m/s of ascent, TEI, and attitude control delta-V. The CEV SM mass at launch is 29,642 kg, compared to 14,858 kg for the 5.5-m EIRA SM.

The LSAM in this architecture functionally only includes the descent stage from the EIRA LSAM, as the ascent stage functionality has been moved to the CEV CM and SM. The EOR-direct return LSAM performs powered descent for the crew and CEV from LLO to the lunar surface. The propulsion system for the descent stage uses the same propellants and engines as the CEV SM. It includes five 44.5-kN (10-klbf) pressure-fed main engines and sixteen 445-N (100-lbf) reaction control thrusters to perform 2,042 m/s of LEO rendezvous, powered descent, and attitude control delta-V. The LSAM also carries the same 500 kg of mission payload as in the EIRA. This produces an LSAM wet mass of 47,437 kg, a combined stack mass including the CEV of 88,732 kg for launch, and a trans-lunar-injected mass of 87,235 kg. Since the large EDS used in the EOR–LOR architecture can only deliver 54,700 kg to LLO, a second EDS (and third launch) is required to execute this mission. Adding a second EDS increases the net payload delivery capability to 121,000 kg to LLO, which is well above the CEV–LSAM trans-lunar injected mass.

4.2.3.3.4 EOR–LOR with CEV-to-Surface Architecture

The fourth and final Analysis Cycle 1 architecture is a hybrid between the previous two options. It combines the LOR aspect of the EOR–LOR architecture and single crew volume of the EOR-direct return architecture. Rather than leaving the CEV CM and SM behind in LLO, this architecture separates the two elements, leaves only the SM behind, and uses the CM to operate on the lunar surface. However, as in the EOR-direct return architecture, the combined mass of the CEV and LSAM exceeded the performance capabilities of a single EDS, thus adding a third launch to each mission.

The EOR–LOR with CEV-to-surface architecture (**Figure 4-7**) operates identically to the EOR-direct return mode up to the point of powered descent. At this point, the CEV separates from the LSAM, and the CM separates from the SM. Using a docking module beneath the CM aft heat shield, the CM returns to the LSAM and docks to the ascent stage. The LSAM then transports the crew to the lunar surface for the nominal surface stay (up to 7 days). Once complete, the ascent stage returns the crew and CEV CM to LLO, the vehicles separate, and the CM docks to the SM. Finally, just as in the EOR–LOR architecture, the SM performs a TLI burn and the crew returns to Earth.

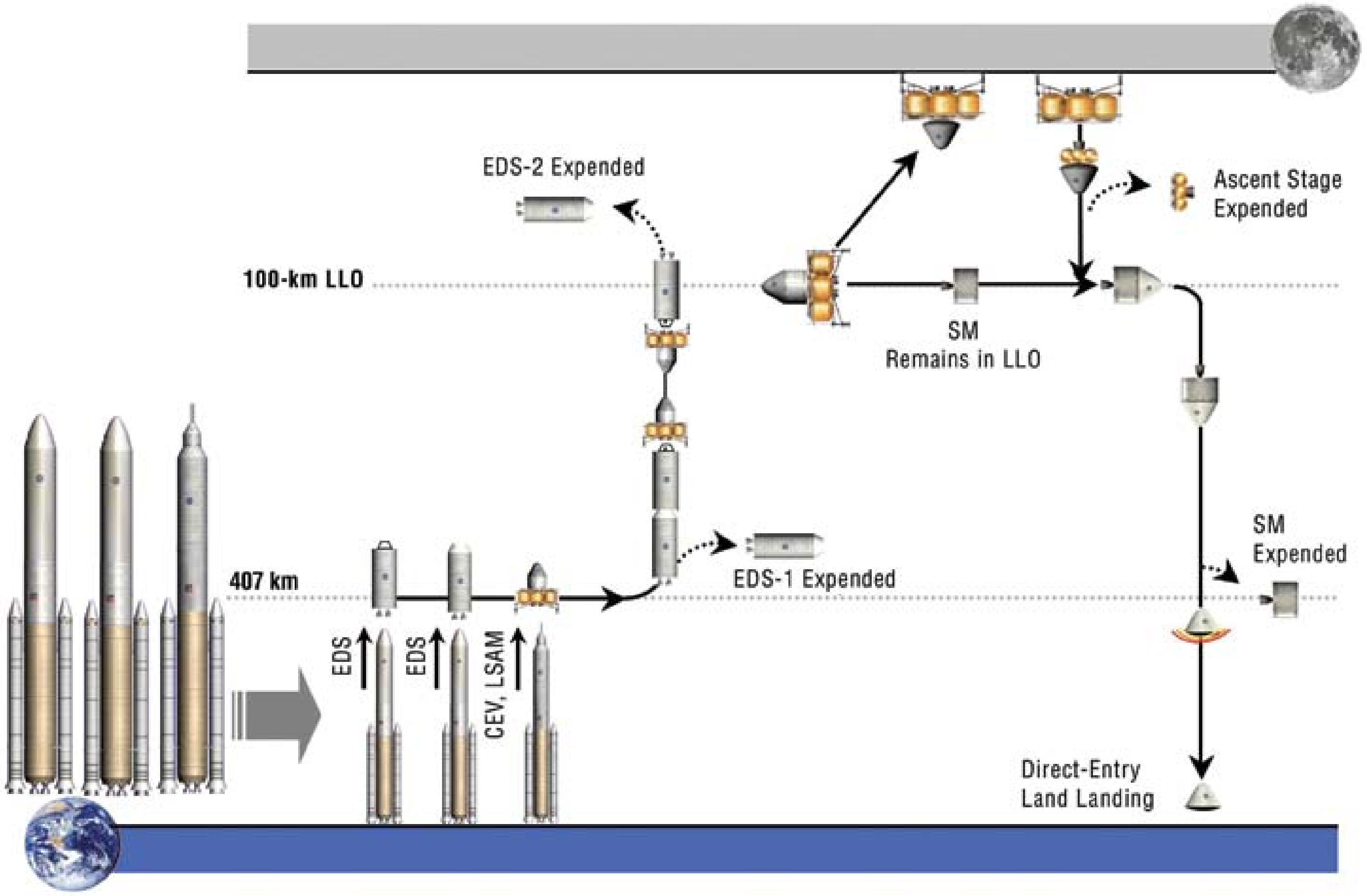


Figure 4-7. EOR–LOR with CEV-to-Surface Architecture (Analysis Cycle 1)

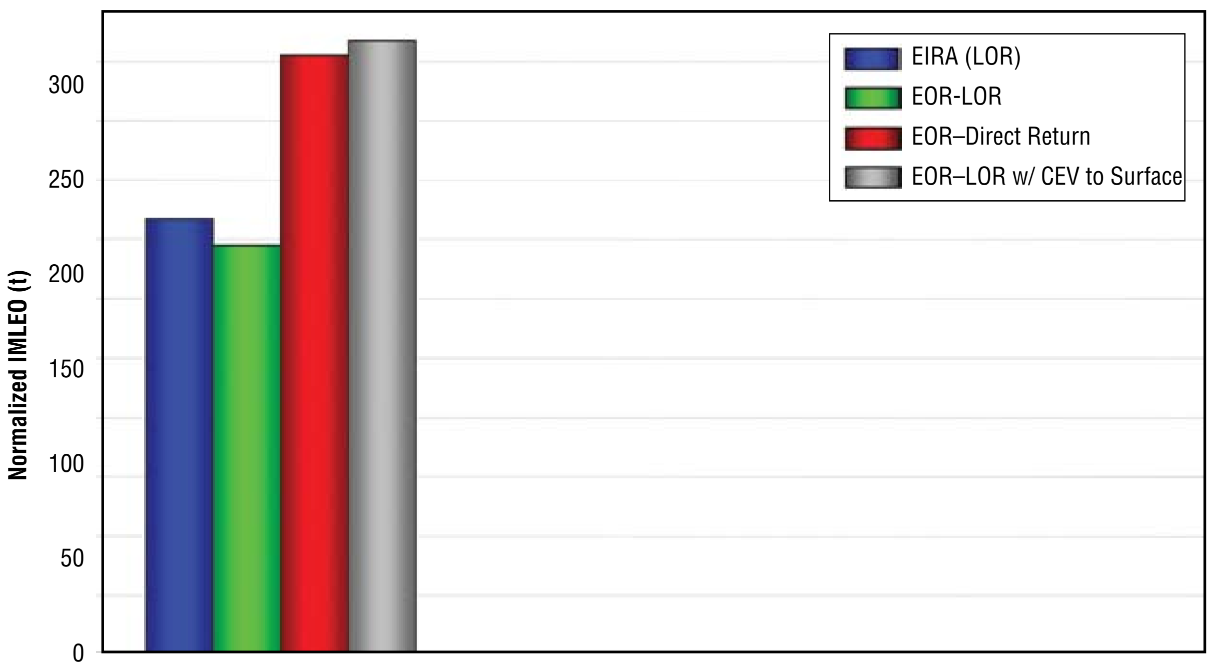
The 5.5-m CEV CM in this architecture is identical to the EOR-direct return architecture, with the addition of four extra crew-days of life support capability. A docking module has also been added to the CEV to facilitate docking to the ascent stage. The SM is functionally similar to the EIRA in that its primary maneuvering capability is for TEI. Additional avionics are included on the vehicle for Command and Control (C&C) while the CM and SM are separated. A single 44.5-kN (10-klbf) main engine and twenty-four 445-N (100-lbf) reaction control thrusters perform 1,612 m/s of TEI and orbital maneuvering delta-V. The masses for the CEV CM, docking module, and SM are 11,871 kg, 1,153 kg, and 11,701 kg, respectively, for a total CEV mass of 24,725 kg at launch.

The LSAM consists of pressure-fed oxygen/methane ascent and descent stages for transporting the CM between LLO and the lunar surface. The descent stage is identical to the EOR-direct return descent stage in number of engines and total delta-V; however, the propellant loading is different due to the lower landed mass. The descent stage's total launch mass is 37,053 kg. The ascent stage consists of two pressure-fed oxygen/methane engines at 44.5 kN (10 klbf) per engine and sixteen 445-N (100-lbf) reaction control thrusters. The total delta-V required is 1,882 m/s for a total ascent stage mass of 14,897 kg.

Total mass at TLI for the CEV and LSAM is 75,635 kg. Since the lunar orbit delivery capability of a single EDS is only 42,700 kg, a second EDS and third mission launch is required to deliver the required mass to the Moon. The EDS performance here is less than in the previous architecture (the performance was 54,700 kg) due to the addition of an LOI plane change on arrival. All LOR options require a plane change to properly align the parking orbit plane for anytime ascent off the surface. As direct return mission modes do not require a rendezvous to return to Earth, this plane change is unnecessary.

4.2.3.3.5 Architecture Performance Comparison

Figure 4-8 provides a normalized total mass comparison of the four architecture modes under consideration in Analysis Cycle 1. As each architecture operates the EDS differently, comparing the alternatives using traditional metrics such as IMLEO is somewhat misleading. For example, the EOR–LOR architecture uses the CEV to perform LOI while the other architectures use the EDS. Also, the latter two architectures that take the CEV to the lunar surface require three launches per mission while the others require two. A more meaningful comparison would be of the margin or vehicle growth potential each architecture provides or a "normalized" IMLEO comparison that assumes the same EDS functionality and number of launches per mission for each option.



Descent Propulsion	LOX/Methane Pressure-Fed	LOX/Hydrogen Pump-Fed	LOX/Hydrogen Pump-Fed
Ascent/TEI Propulsion	LOX/Methane Pressure-Fed	LOX/Methane Pressure-Fed	LOX/Methane Pump-Fed

Figure 4-8. Analysis Cycle 1 Normalized IMLEO Comparison

This analysis assumes the EDS is used to perform TLI and LOI in each architecture and that each is a 2-launch solution. Rather than using a specific EDS mass in LEO, as supplied by the launch vehicle analysts, the EDS mass is estimated for the specific payload using a mass fraction, specific impulse, and required delta-V. Therefore, while each bar in the chart does not correspond to the mission's actual mass in LEO, the relative comparison between the four alternatives is correct. As expected, the EIRA has a slightly higher normalized IMLEO than

EOR–LOR due to its having two EDSs instead of one. EOR–LOR has the lowest normalized IMLEO, while architectures where the CEV goes to the lunar surface (the latter two) are roughly 50 percent higher. The EOR-direct return architecture is heavily penalized for taking the CM to the surface when using pressure-fed oxygen/methane propulsion. Surprisingly, the EOR–LOR with CEV-to-surface architecture has the highest overall normalized IMLEO. One might suspect that eliminating the ascent stage crew cabin and leaving the CEV SM in LLO would at least negate the cost of taking the heavy CM to and from the surface. However, the assumed mission design selected for its positive crew safety aspects requires a relatively large plane change at LOI to align the parking orbit for anytime ascent. The vehicles also require additional propellant and mating interfaces to facilitate the intricate docking and undocking sequences in LLO. These factors cause a significant increase in the required EDS propellant and drive-up the architecture's normalized IMLEO. Later design cycles examined more advanced descent, ascent, and TEI propulsion options to reduce mass and possibly eliminate launches.

4.2.3.4 Figures of Merit

The performance of the baseline EIRA mission was documented in two forms as inputs for the ESAS team. The team was given detailed subsystem mass breakdowns for each of the vehicles in the architecture, each with an estimate for minimum, most likely, and maximum mass as well as an explanation for any subsystem that deviated from the EIRA baseline. The team was also given a summary of mission events, including details for critical events such as engine burn times for major propulsive maneuvers.

The team generated integrated life cycle program costs through the year 2025 that were compared against the EIRA baseline mission. Likewise, the team developed estimates for the P(LOC) and the P(LOM). Detailed safety and reliability analyses are presented in **Section 8**, **Risk and Reliability**, and detailed cost analyses are presented in **Section 12**, **Cost**.

Table 4-2 summarizes the performance, cost, safety, and reliability of Analysis Cycle 1 mission options. IMLEO mass was normalized to account for the fact that different mission modes split propulsive delta-V across the upper stages of LVs and descent stages of landers in different ways. The original EIRA using the 5.0-m diameter CEV and the EIRA with a 5.5-m diameter CEV were analyzed for cost differences only.

Table 4-2. Analysis Cycle 1 Architecture Options Performance, Cost, and Risk Comparison

	Design Cycle 1—Common Assumptions			
	Normalized IMLEO (mT)	Integrated Cost Delta	P (LOC)	P (LOM)
EIRA	183.9	0B	0.07	0.19
#1 EIRA 5.5 m		+1.7B		
#2 EOR–LOR	172.5	+3.0B	0.07	0.20
#3 EOR-Direct Return	252.6	+7.1B	0.04	0.21
#4 EOR–LOR w/CEV-to-Surface	258.9	+18.4B	N/A	N/A

The ESAS team also assessed FOMs other than cost, risk, and performance in order to more completely characterize all major trade studies. FOMs were selected, as described in **Section 2**, **Introduction**, to be relatively independent and comprehensive and were described by direct measures as well as proxy parameters. Extensibility and flexibility were characterized in terms of lunar mission flexibility, Mars mission extensibility, extensibility to other exploration destinations, commercial extensibility, and national security extensibility. Programmatic risk, separate from safety and mission success, was characterized in terms of technology development risk, cost risk, schedule risk, and political risk. Analysis Cycle 1 strengths and weaknesses relative to the FOMs are shown in **Table 4-3**.

Table 4-3. Analysis Cycle 1 Mission Architecture FOMs

Cycle 1 Architecture	Description	Discriminating FOMs	
		Advantages	Disadvantages
EIRA	5-m, 30-deg CEV; LOR split mission; LOX/H2 TLI, LOI; CH4 Descent, Ascent, TEI		
EIRA 5.5 Split Mission	5.5-m, 25-deg CEV; LOR split mission	• No EOR (same as EIRA).	• 2×LOR required for mission success; • 1 critical LOR required for crew survivability; and • No "Apollo 13" lifeboat capability.
EOR–LOR	5.5-m, 25-deg CEV; EOR rendezvous ("all-up"), LOI performed by SM (LOX/CH4)	• One less propulsion stage; • One fewer engine; and • "Apollo 13" backup habitat capability available outbound.	• Greater number of rendezvous required (2×EOR (transposition and docking, dock w/EDS) and 1×LOR) required for mission success; • Critical LOR required for crew survivability; and • Large element rendezvous and docking in LEO.
EOR-Direct	5.5-m, 25-deg CEV; CEV goes direct-to-surface, CEV direct return from surface; 2×EDS stages required for this baseline prop option	• Fewer flight elements (one less crew cabin, one less propulsion stage; • Two fewer vehicle-to-vehicle interfaces; • No LLO rendezvous; no rendezvous required for crew survivability; • Less overall delta-V; and • Greater cargo mass to surface.	• No flight experience with direct return missions; • 2×EDS stages required for this baseline prop option (three launches per mission); • 2×EOR (uncrewed large-element EDS-to-EDS and EDS-to-LSAM) required for mission success; • Initial CEV development scarred for lunar surface missions; • More sensitive to CEV and returned mass growth; • Surface dust control in CEV; • No "Apollo 13" lifeboat capability; and • Ascent stage has less (or no) commonality with CEV ISS SM.
EOR–LOR w/ CEV-to-Surface	5.5-m, 25-deg CEV; CEV goes direct-to-surface, CEV direct return from surface via LOR rendezvous with SM; 2×EDS stages required for this baseline prop option	• Fewer flight elements (one less crew cabin); and • One less engine, one less vehicle-to-vehicle interface.	• 2×EDS stages required for this baseline prop option; • Greater number of rendezvous required—2×EOR (large element uncrewed EDS to EDS, LSAM to EDS) + 2×LOR (CEV to LSAM, crew-survivability critical, complex CEV to SM); • Three launches per mission; • Initial CEV development scarred for lunar surface missions; • Surface dust mitigation on CEV; greater sensitivity to mass growth; • No "Apollo 13" lifeboat capability; and • More sensitive to CEV and returned mass changes.

4.2.3.5 Findings and Forward Work

This initial analysis of mission architecture modes collapsed the option space to a 2x2 matrix that compared Earth-orbit operations and lunar-orbit operations. Within that matrix, the option with no operations in either LEO or LLO (the "direct-direct" mission mode) was eliminated due to its requiring a single launch of more than 200 mT to LEO. A baseline LOR mission architecture was established and LOR, EOR–LOR, and EOR-direct options were compared to it, albeit limited to variations of mission mode only. The CEV for each option was similar, varying only in diameter from 5.0 m to 5.5 m. All options carried 1400 kg of supplemental radiation protection. For each common propulsive event (LOI, lunar descent, lunar ascent, and TEI), propulsion type and technology were held constant across the architectures.

Based on the above assumptions, LOR mission modes result in the lowest IMLEO, lowest cost, and fewest launches. Direct return missions had the lowest P(LOC) but required three launches, resulting in higher P(LOM). The analysis showed that certain mission modes performed poorly using the baseline architecture assumptions, which led the team to investigate more optimized propulsion type and technologies in proceeding analysis cycles. Higher efficiency lander propulsion could reduce direct return options to two launches, thus enabling better comparison of the mission mode.

The poorest performing option investigated in Cycle 1 was EOR–LOR with CEV-to-surface. This option was eliminated from further considerations due to having the highest IMLEO, the highest P(LOM), and the highest cost.

Analysis Cycle 2 was targeted at optimizing mission mode performance with propulsion technology changes, including LOX/H2 descent stages and pump-fed LOX/CH4 ascent stages. Radiation shielding, which accounts for 15 percent of the CEV mass, would be further studied, as would the CEV configuration for surface-direct missions and airlocks for surface operations.

4.2.4 Analysis Cycle 2 Mission Mode Analysis

Initial mission mode analysis results pointed to the need to vary both propulsion technology and spacecraft subsystems to assess the sensitivity of the mission mode to both the linear and exponential variables of the rocket equation. The ESAS team noted that the greatest leverage could be found in CEV systems that necessarily travel round-trip from launch to landing and propulsion systems that occur at or near the end of the mission's series of propulsive events.

The variables examined in the second design cycle included the mass of supplemental radiation shielding applied to the CEV, the split of delta-V maneuvers among propulsive stages, higher-efficiency propulsion systems, and the application of airlocks and split volumes to surface landers. The goal of the analysis was to eliminate all "3-launch solutions" and better optimize each of the mission mode options. In particular, this cycle sought to determine the feasibility of the EOR-direct return mission mode. This particular mission mode was the most stressing to the CEV design, as it required a single crew compartment to perform planetary landing, surface habitation, and planetary ascent functions in addition to all the functions required for an LOR mission.

4.2.4.1 Trade Studies

Analysis Cycle 2 began an in-depth analysis of CEV supplemental radiation shielding and of CEV and LSAM propulsion technology. The ESAS team chose to first trade spacecraft variables that offered the greatest IMLEO savings based on both mass sensitivities and the need to better understand these variables. Mass sensitivities for the three mission modes, as shown in **Table 4-4**, measure the "partial differential" effect of increasing specific vehicle inert masses as a function of the overall system IMLEO mass. Such tables are useful in identifying where the greatest mass savings can be gained via technology investment or increased engineering certainty.

Table 4-4. Lunar Architecture Mass Sensitivities by Mission Mode

IMLEO Mass / Vehicle Mass for:	LOR (EIRA)	EOR–LOR	EOR-Direct
CEV CM	6.1	6.4	14.7
CEV SM	5.1	5.4	12.3
Ascent Stage	8.8	10.1	
Descent Stage	4.8	5.5	4.6
Round-Trip Cargo	12.1	13.4	14.7

Based on the mass sensitivities, the greatest IMLEO mass leverage comes from mass reductions in the LSAM ascent stage, round-trip cargo, or (for the EOR-direct return mode only) the CEV CM itself. Based on this knowledge, the ESAS team undertook a critical study of the CEV CM mass and, in particular, the approximate 1,800 kg of supplemental radiation shielding that was being carried round-trip. Additionally, the LSAM ascent and descent stages offered opportunities to decrease IMLEO through the use of higher specific impulse (Isp) propulsion systems.

4.2.4.1.1 CEV Radiation Protection

The CEV CM will be the primary crew cabin for the majority of the lunar mission. It will contain the crew during launch, Earth-orbital operations, trans-lunar cruise, and in lunar orbit. For LOR missions, the crew will transfer to the LSAM for the duration of surface operations, but will return to the CM for additional lunar orbit operations, trans-Earth coast, and Earth entry. For direct return missions, the crew will remain in the CM for lunar descent, surface operations, and ascent. At a minimum, the crew will spend 9 days in the CM beyond the protection of Earth's magnetosphere in the interplanetary radiation environment.

Ionizing radiation is a major health hazard everywhere in space and on all planetary and satellite surfaces. Galactic Cosmic Rays (GCRs) permeate the galaxy and consist of protons, helium, and high-charge-and-energy ions. Solar Particle Events (SPEs) are dominated by hydrogen and helium ions with energies of several hundred millions of electron volts (MeVs). Albedo neutrons are produced in planetary atmospheres and surfaces and can be a significant source of human exposure. The albedo neutron decay produces electrons and protons that can have long lifetimes when decay is within planetary magnetic trapping regions, giving rise to intense trapped radiation belts.

Ionization leads to direct and indirect injury to the cell genome, resulting in cell death or latent damage that can lead to cancer and other effects. The energy per unit mass, locally deposited by radiation, is quantified as dose. When weighted for the estimated effectiveness of a particular type of radiation, the reference quantity is the equivalent dose. It is essential to recognize that risks are not measured or monitored directly. Instead, radiation quantities are used to estimate the associated risk.

NASA has established limits on the risk that may be incurred by exposure to space radiation. These limits are specified for missions in LEO. The limiting risk for career exposure to space radiation is an increase of 3 percent in the probability of developing a fatal cancer. Thirty-day and annual limits are based on keeping radiation exposure below the threshold level for deterministic effects. Also, NASA has incorporated the requirement to keep exposures "As Low as Reasonably Achievable" (ALARA) in the designs used and the operations conducted in space.

Unlike LEO exposures, which are often dominated by solar protons and trapped radiation, interplanetary exposures may be dominated by GCRs, for which there is insufficient data on biological effects. Consequently, risk prediction for interplanetary space is subject to very large uncertainties, which impact all aspects of mission design. This is especially true since ALARA requirement requires the use of appropriate safety margins, which are directly related to the uncertainty in risk estimates.

The ESMD Space Radiation Program strategy is to develop the knowledge base to accurately project health risks from radiation exposure and to recommend protection requirements. The overall objectives as well as the mission-specific strategies are identified below. Detailed radiation research and protection program objectives for lunar and Mars missions are:

- Ground-based space radiobiology research to establish a knowledge base to set radiation limits, estimate crew risks, and support shielding requirement decisions;
- Ground-based physics research to develop a particle interaction knowledge base, shielding design tools, and materials research;
- Lunar and Mars radiation limit definition;
- Environmental definition;
- Radiation dosimetry and monitoring equipment development; and
- Biological countermeasure development and integration (as needed).

Mission-specific strategies include:

- Use robotic precursor orbital and surface missions to understand the lunar neutron environment, develop reliable area monitors, and establish a high-energy proton capability.
- CEV-to-ISS missions will follow the ISS/STS mission operations model. Strategies include establishing reliable area monitors; integrating the ALARA requirement into the design; recommending the use of carbon composites in vehicle structures, shielding, and components early in the design; and providing recommendations on design optimization.
- Short lunar stay strategies must include integrating the ALARA requirement into the vehicle design and operations; recommending the use of carbon composites in vehicle structures, shielding, and components early in the design; and providing recommendations on design optimization. Sortie times may also be restricted by worst-case SPE definition and EVA suit shielding properties. Local shielding is recommended to minimize risks, and mission planning must consider trade-offs (e.g., habitat shelter shielding versus surface abort).
- Long lunar stay missions will likely require increased shielding over a short stay and the development of strategies to reduce chronic risk and GCR impacts. The inclusion of previous exposures for crew selection also becomes more important (astronauts with prior lunar or ISS missions).

Radiation Limits

NASA relies on external guidance from the National Academy of Sciences (NAS) and the National Council on Radiation Protection and Measurements (NCRP) for establishing dose limits. Due to the lack of data and knowledge, the NAS and NCRP recommended that radiation limits for exploration missions could not be determined until new science data and knowledge was obtained. Lunar radiation limits are being developed by the NCRP and the Chief Health and Medical Officer (CHMO), and there is some expectation that short-term and career limits will change; however, LEO limits were used for this study. The LEO career limit is the probability of 3 percent additional risk of lifetime lethal cancer within a 95 percent confidence interval. The LEO Blood-Forming Organs (BFO) short-term limits are: a 30-day limit of 25 Centigray Equivalent (cGy-Eq) and an annual limit of 50 cGy-Eq. For lunar missions, it is expected that NASA will implement dose limits based on the Risk of Exposure-Induced Death (REID) to replace limits based on Excess Lifetime Risk (ELR) of cancer. Also, information on fatal non-cancer risks, most notably heart disease, is under review. Research on radiation quality and dose-rate effects for heart disease risk is in an early stage; however, for protons of reasonably high dose-rates (>5 cGy/hr), a risk estimate can be made and suggests an increased fatal risk of 50 percent over the risk from fatal cancer alone. Risk projections will be augmented with projection of average life loss for exposure-induced deaths, with approximately 15 years projected for astronauts between the ages of 35–45 for SPE risks.

The ESAS radiation study addressed the relationship between shielding mass, dosage, and crew risk for the CEV. The probability of an event was determined using the two largest events on record for which accurate spectral information is available. The August 1972 event is generally accepted as the benchmark SPE in observable history. The confidence of not exceeding the August 1972 event fluence level above 30 MeV on a 1-year mission near the solar maximum is roughly 97 percent. (Note: High annual fluence levels are usually dominated by the largest event within the year.) To achieve a 99.5 percent confidence level above 30 Million Electron Volts (MeV), one must assume a fluence level about four times the August 1972 event. The probability of an event that would exceed the current LEO limits within any 1-week mission was estimated at 0.2 percent. The estimated probability of an SPE that could cause debilitation (1.5 times the August 1972 event) was estimated at roughly 0.03 percent. A debilitating event was identified as a dose that would cause vomiting within 2 days in 50 percent of the total population. The estimated probability of a catastrophic event (4 times the August 1972 event) causing death within 30 days was estimated at roughly 0.01 percent. These estimates were developed using historical data with no statistical analysis of the frequency distribution of the event.

The Analysis Cycle 2 radiation evaluation involved the analysis of a preliminary computer-aided design (CAD) model of the CEV. The CAD analysis results (shielding files) were used to conduct the final crew risk projections.

Mass sensitivity curves illustrating the reduction in radiation exposure to crew members within the CEV with increasing shield augmentation were calculated for two design case SPEs. Four times the proton fluence (no time dependence) of the August 1972 (King spectrum) event was evaluated, as well as four times the proton fluence of the September 1989 event. It was assumed that only one large design-basis SPE occurred during the specified mission length.

CEV Dose Data and Mass Sensitivity Curve

Since the Cycle 2 design exercise limited the mission length to 7 days, the largest concern for radiation exposure would be from SPEs. Mass sensitivity curves illustrating the reduction in radiation exposure to crew members within a CEV with increasing shield augmentation were calculated for the two SPEs. The internal systems represented in this CEV model were of fairly high fidelity. However, the outer hull of the vehicle was of fairly low fidelity, represented by an aluminum pressure shell and HDPE radiation shield. The areas between the chosen evaluation points and the outside environment that had the lowest radiation shielding consisted only of this aluminum shell and HDPE radiation shield. In general, thin (or lower radiation shielded) areas dominate the resultant radiation exposure to the crew inside the vehicle. These thinly shielded areas and the modeling of the hull and shield likely dominated the exposure estimates for this cycle of analysis.

The analysis was performed by first generating shield distribution files for the vehicle. The generation of these shield files is done by ray-tracing the CAD model. The ray-tracing output describes the amount and thicknesses of material between a chosen point and the outside environment. Two evaluation points were chosen to be consistent with the location of the crew member's torso in the seated position. Best estimates of material composition and density were assigned to the model elements for this evaluation. In addition, a volume representative of an SM was positioned relative to the CEV to approximate the shadow shielding effect that the SM would provide.

A radiation dose calculation was then performed for the skin, eye, and BFO using the equivalent spheres approximation. This approximation assumes a tissue depth of 0.01, 0.3, and 5 cm for the skin, eye, and BFO dose calculations, respectively. It should be noted that use of the equivalent spheres approximation can result in a two-fold overestimation of dose as compared to the more accurate computer-aided manufacturing (CAM) model that will be used in later analysis cycles. Doses were calculated for four times the August 1972 SPE, as well as four times the October 1989 event. These skin, eye, and BFO dose calculations were made for the vehicle with no parasitic shielding as well as with the addition of 5 grams/cm^2 of HDPE. **Table 4-5** shows a comparison to the Apollo Command Module, which corresponds to a thickness of approximately 5 g/cm^2, and a dose calculated using the CAM model.

Table 4-5. Analysis Cycle 2 Radiation Dose Calculations for Aluminum CEV with HDPE Supplemental Shielding

Organ Dose 4× 1972 SPE	Apollo	Aluminum CEV*		CEV + Poly 5 g/cm^2	
Skin (Gy-Eq)	10.36	42.63	47.75	**12.25**	**13.72**
Eye (Gy-Eq)	8.20	**32.54**	**36.44**	**9.71**	**10.87**
BFO (Gy-Eq)	1.39	4.17	**4.67**	**1.56**	**1.73**
Organ Dose 4× 1989 SPE		**Aluminum CEV***		**CEV + Poly 5 g/cm^2**	
Skin (Gy-Eq)		23.40	25.98	**7.10**	**7.88**
Eye (Gy-Eq)		**16.57**	**18.39**	**5.42**	**6.01**
BFO (Gy-Eq)		2.73	**3.03**	**1.29**	**1.40**

*****Note: Two columns for CEV represent two locations within vehicle.*

Figure 4-9 is also represented by a mass sensitivity curve for the BFO dose versus the mass of the HDPE shield. This was made for four times the August 1972 and October 1989 events. The thickness of the radiation shield was varied from 0 to 5 g/cm^2 and plotted according to the corresponding shield mass. The shield at 5 g/cm^2 was effectively at the maximum mass (1,360 kg) allotted for the supplemental radiation shield.

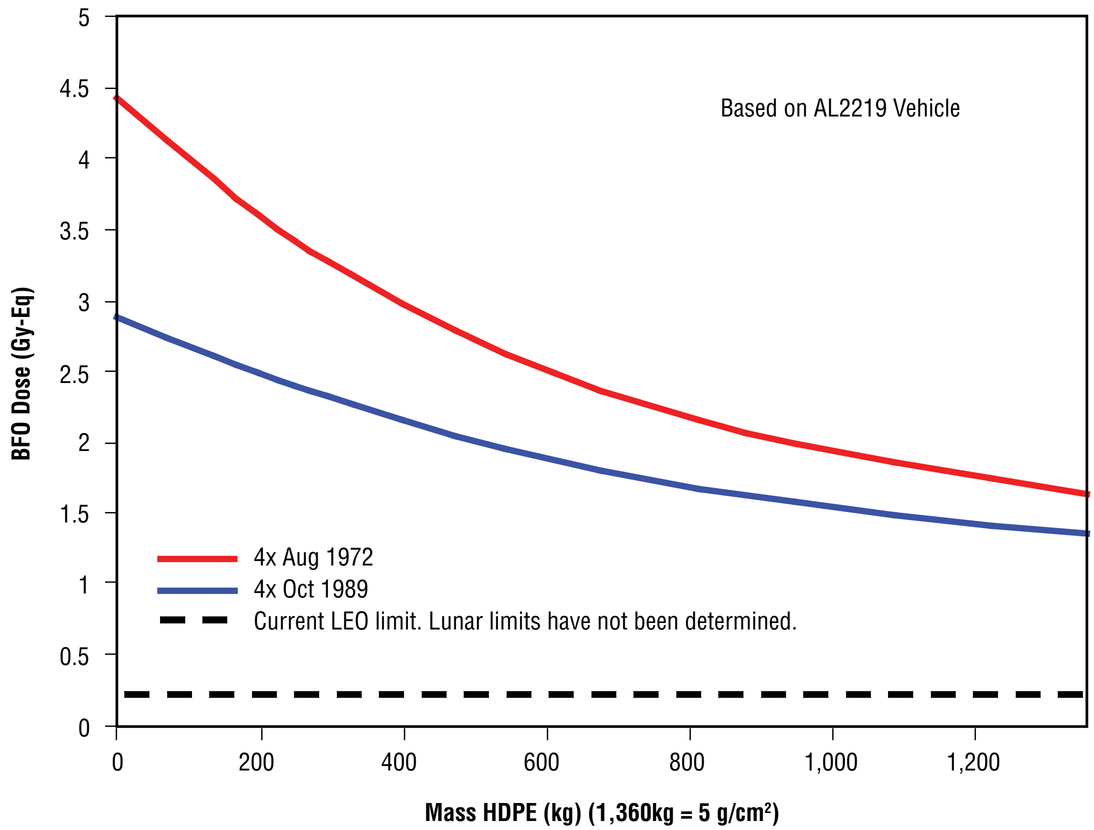


Figure 4-9. Mass of HDPE

CEV Radiation Risks and Shielding

Using the shielding files provided by the above-mentioned analysis, probabilistic estimates of risk and loss-of-life were calculated for a 35- and 45-year-old female as well as a 35- and 45-year-old male. A three-layer version of aluminum or graphite/epoxy, polyethylene, and tissue was employed. No previous occupational radiation exposure was assumed for any of these representative crew members. The current radiation limit is a 3 percent fatal cancer probability within a 95 percent confidence interval. The calculation considers age/gender, radiation quality, SPE dose-rate, shielding materials, and prior ISS/CEV missions. As indicated by **Table 4-6**, all representative crew members exceed the 3 percent probability of fatal cancer risk for the CEV without any supplemental shielding. When 5 g/cm^2 polyethylene is added to the CEV, the risk drops below 3 percent; however, the upper 95 percent confidence interval exceeds the LEO limit.

Table 4-6. Excess Lifetime Cancer Risk for Shielded and Unshielded CEV as a Function of Crew Member Age and Gender

4× 1972 – Equivalent Solar Proton Event – CEV				
Organ Dose	**Aluminum CEV**		**Vehicle + Poly 5 g/cm^2**	
Crew Characteristic	**%Risk**	**95% C.I.**	**%Risk**	**95% C.I.**
Male 35-yr	9.7	[3.4, 17.5]	1.7	[0.5, 4.7]
Male 45-yr	7.5	[2.7, 16.4]	1.3	[0.4, 3.5]
Female 35-yr	12.1	[4.0, 17.6]	2.1	[0.7, 5.9]
Female 45-yr	9.1	[3.2, 17.3]	1.5	[0.5, 4.3]
4× 1989 – Equivalent Solar Proton Event – CEV				
Organ Dose	**Aluminum CEV**		**Vehicle + Poly 5 g/cm^2**	
Crew Characteristic	**%Risk**	**95% C.I.**	**%Risk**	**95% C.I.**
Male 35-yr	6.9	[2.4, 15.8]	2.2	[0.73, 6.0]
Male 45-yr	5.3	[1.9, 13.4]	1.7	[0.57, 4.5]
Female 35-yr	8.6	[2.9, 17.1]	2.8	[0.9, 7.6]
Female 45-yr	6.4	[2.3, 15.3]	2.0	[0.7, 5.6]

C.I. = Confidence Interval

CEV Acute and Late Risks

SPEs represent the greatest concern for radiation exposure during the short-duration lunar missions. For estimating acute risks, calculations using the Nuclear Regulatory Commission (NUREG) fatal accident risk model were performed. The NUREG model is unable to properly evaluate acute risks (mortality or debilitating sickness) below a 10 percent probability because of the uncertainties in sigmoid dose-response curves characteristic of deterministic effects near thresholds. Also, microgravity research suggests that altered immune and stress responses could skew the lower probabilities of dose responses to reduced dose levels complicating the evaluation of acute risk near the threshold (less than 10 percent risk). Depending on the baseline CEV design, acute risks are possible for an event with the 1972 spectral characteristics and two to four times the F(>30 MeV) fluence. Future research and analysis will be needed to establish the correct dose response under these conditions. For a baseline CEV shielded with targeted >2 g/cm^2 of polyethylene shielding, acute effects are unlikely from such events, as shown in **Table 4-7**.

Table 4-7. CEV Acute and Late Risks for Various Depths of HDPE Radiation Shielding

Aluminum Vehicle, 4× 1972 SPE			
HDPE Depth (g/cm^2)	**% Acute Death***	**% Sickness**	**% REID****
CEV-old + 0 g/cm^2	9.5	54	9.1 [3.2, 17.3]
CEV-new + 0 g/cm^2	<1% (***)	<5% (***)	4.4 [1.5, 11.8]
CEV-new + 1 g/cm^2	0	0	3.5 [1.2, 9.7]
CEV-new + 2 g/cm^2	0	0	2.9 [1.0, 8.2]

**Death at 60 days with minimal medical treatment*

***Risk of Cancer death for 45-yr-old females*

****Too close to threshold to estimate*

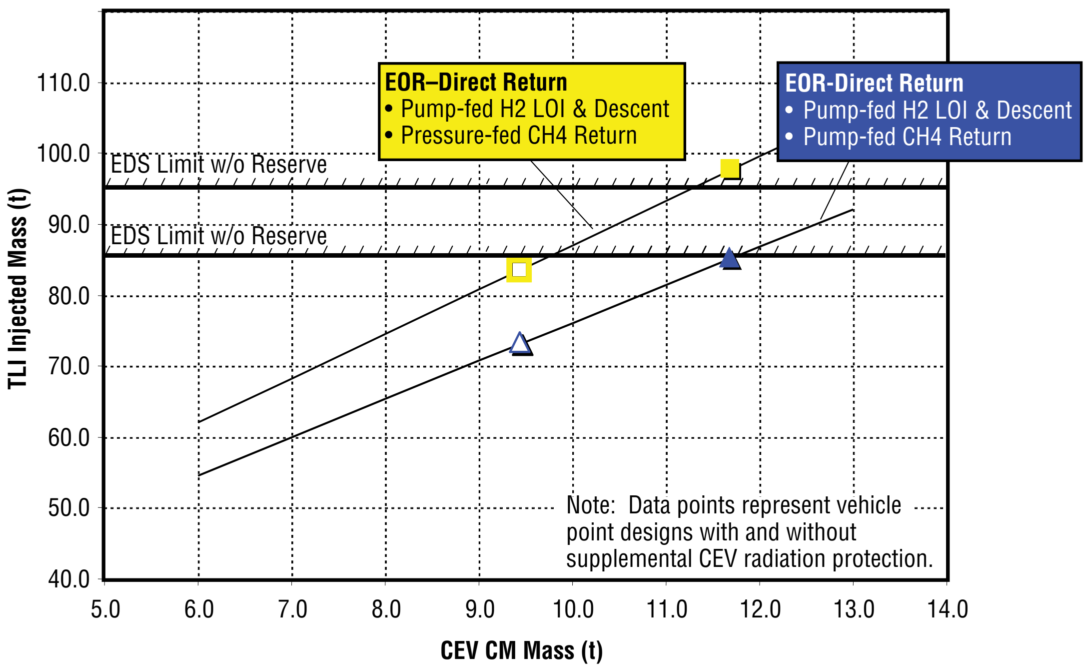


Figure 4-10. TLI Injected Mass versus Crew Compartment Mass, With and Without 5.0 g/cm² Supplemental Radiation Protection

Cycle 2 Radiation Analysis Impact on CEV and Mission Design

The ESAS team reviewed the radiation analysis with an eye toward reducing the supplemental radiation shielding that was resulting in a diminishing benefit to the crew. **Figure 4-10** illustrates the effect of supplemental radiation shielding on injected spacecraft mass. Since the radiation shielding mass is carried round-trip, its mass has one of the greatest mass sensitivity penalties, which identifies it as a candidate for additional analysis. (Refer to **Table 4-4** for more information.) ESAS engineers and safety and risk analysts agreed to proceed into a third analysis cycle utilizing a maximum of 2.0 g/cm^2 of supplemental radiation shielding—the range in which the dose analysis indicated that shielding had the greatest effect. The dose and biological risk data was derived from a 4-times-1972 event that represented a 99.5 percent confidence of not exceeding a fluence level exceeding 30 MeV for a mission duration of 1 year. Therefore, for a 16-day maximum mission (0.04 year duration), the probability for exceeding a 0.01 percent probability of acute death, a 1.9 percent probability of debilitating sickness, and a 3.4 percent probability of excess cancer risk is itself only 0.005. For 5 g/cm^2 of shielding, these values are either zero or approaching zero.

The data presented above led the ESAS team to take a number of actions. First, based on the team's recommendation, the CEV CM would incorporate the use of composite materials, in addition to an aluminum pressure shell, as part of the cross-sectional skin of the vehicle. Secondly, the ESAS team recommended additional analysis to more accurately model the CEV cross-section and to further investigate the range of supplemental radiation shielding from 0 to 2.0 g/cm^2. These decisions would form the basis of the Cycle 3 radiation analysis presented in **Section 4.2.5**, **Analysis Cycle 3 Mission Mode Analysis**.

4.2.4.1.2 LOX/CH4 versus Storable Propellant Trades

Many NASA studies have evaluated propellant combinations of on-orbit propulsion for spacecraft. These include various combinations of Oxygen (O2), Hydrogen Peroxide (H2O2), Nitrous Oxide (N2O), Nitrogen Tetroxide (NTO), and Chlorine Pentafloride (ClF5), together with fuels such as Hydrazine (N2H4), Monomethyl Hydrazine (MMH), Hydrogen, Ethanol (EtOH), Methane (CH4), Propane (C2H6), and Kerosene (RP1). The propellants exhibiting the best overall characteristics from these studies are LO2/LH2, LO2/LCH4, LO2/EtOH, and MMH/NTO. This section will focus on two of these options as the primary alternatives for the CEV SM service propulsion system as well as the ascent propulsion system for the LSAM—LOX/CH4 and MMH/NTO. Storable MMH/NTO systems are well understood and have an extensive operational history; however, LOX/CH4 is of particular interest because it is high-performing, non-toxic, and can be obtained from Martian and lunar in-situ resources (CH4 from the Martian atmosphere and LOX from the Martian and lunar soil).

Performance Comparisons

MMH/NTO propulsion systems are well-characterized, with substantial flight history. These propulsion systems provide the light dry mass systems and good packaging compared to most other propellant combinations. NTO and MMH ignite hypergolically, thus eliminating the need for igniters and reducing system complexity. However, due to the mechanism of hypergolic ignition, the formation of Fuel-Oxidizer Reaction Products (FORPs) can occur for short pulses in a cold vacuum environment, as has been noted on the Shuttle 870-lbf (vacuum) primary thrusters and Apollo-110 lbf (vacuum) attitude control thrusters. In addition, iron nitrates can form in the NTO, which could result in flow decay or valve stiction. Iron nitrate formation is most common in NTO systems when moisture is introduced into the propellant. Nitric acid then attacks iron alloy lines and components, leeching iron from those lines and components to form iron nitrates, which can then be deposited in tight-tolerance flow passages due to pressure and temperature drops as the oxidizer flows through those passages. The deposition of iron nitrate in tight flow passages further reduces the propellant flow through flow passages and can cause stiction in sliding components (i.e., valves). Flow decay can affect engines by causing off-mixture-ratio combustion which, in a worst-case scenario, could result in non-ignition events. Iron nitrate formation is most common in multiuse spacecraft (such as the Shuttle Orbiter) and long-duration spacecraft that contain a significant amount of stainless steel lines and/or components. NTO will freeze at approximately 12°F and is, therefore, not considered to be truly space-storable, since significant heater power is required for in-space operations. NTO and MMH are usually stored and used at temperatures greater than 65°F, and, as a result, significant heater power is required to maintain the bulk propellant temperature and maintain lines and components (especially thrusters) within acceptable temperature bands. NTO and MMH are not synergistic with common fluid systems (e.g., common fluids for propulsion, fuel cell reactants, breathing air, and water production) and In-Situ Produced Propellants (ISPPs) and, therefore, do not benefit from the mass and complexity savings that potentially could be realized through the use of common-fluid spacecraft. NTO and MMH are also highly toxic, thus impacting ground and in-space crew operations. Toxic propellants mostly affect reusable spacecraft, but will also affect single-use spacecraft since the servicing and maintenance of toxic propellant systems require expensive personal protective procedures and gear to be used for those operations. Despite the known negatives, the 40-plus year experience with NTO/MMH propulsion systems establishes NTO/MMH to be the lowest risk option for the successful development of the propulsion system of the CEV SM and LSAM.

Liquid oxygen/methane (LOX/CH4) has no flight history and very limited ground-test history. For the high-risk areas of engines, propellant storage and distribution, and components, the characteristics of LOX/LCH4 must be extrapolated from the limited ground test. However, this limited ground test has shown the combustion performance to be suitable for use as a propellant. LOX/LCH4 is a clean-burning propellant combination. LOX does have an extensive history as a fluid on spacecraft and as propellant on propulsion stages; however, the on-orbit operational experience for LOX systems is limited. The safe handling and safe system design aspect of LOX are well understood. LOX/CH4 does offer higher Isp performance compared to state-of-the-art storables (i.e., NTO/MMH), without the volume increase that is common with LOX/LH2 systems, which results in an overall lower vehicle mass as compared to MMH/NTO propulsion systems. A LOX/LCH4 system uses less power, on the order of 1,000 watts less power, than comparable MMH/NTO propulsion systems, thereby significantly reducing the mass of the spacecraft power system(s). The primary advanced development risk areas that must be addressed for LOX/CH4 propulsion systems are the ignition system, long-duration cryogenic storage, propellant acquisition, and propellant quality management within the distributed propellant feed system. The feasibility of storing LOX for 180 days has been demonstrated in ground tests using a 15,000-lbm capacity flight-weight tank (with all penetrations). These tests achieved 1.3 watts total heat leak for the tank. Basic ignition of O2 and methane has been proven to be feasible; however, the reliable ignition of LOX and methane over the range of propellant conditions and mixture ratios must still be established. During advanced development, the demonstration of robust hardware for spacecraft use is required in order to support vehicle subsystem development. Additional risk mitigation for development will be conducted to allow margin in boil-off and lower Isp for ISS missions, since these missions require less delta-V. LOX/CH4 propulsion systems offer significant gains in spacecraft performance, and the risks of developing a LOX/CH4 system appear to be manageable. However, the tight development schedule for the CEV puts the successful development of a LOX/CH4 system for the SM at risk.

An independent assessment of the ESAS Envision weight and sizing model for the CEV SM, using a more detailed model of the propulsion system, was performed to evaluate the different propellant and configuration options and validate the Envision results. The model considered the mass of the structure, propellant tanks, engine, feed system, boil-off, and power system impacts. The Isp of the engines was generated with a common set of engine parameters (area ratio, chamber pressure, thrust, etc.) and by applying a common efficiency of 94 percent of ideal. Power system sizing information from NASA Glenn Research Center (GRC) was provided to allow the impacts of propulsion system power needs to be evaluated. The results (**Figure 4-11**) showed that LOX/methane provided overall higher performance than MMH/NTO and LOX/LH2 for pressure-fed configuration. The pump-fed options that were evaluated examined integrated service propulsion system and Reaction Control System (RCS) propulsion, as well as separate RCS. If the RCS is separated, additional reserves are required for RCS, as well as redundancy in the service propulsion system, since RCS can no longer perform backup.

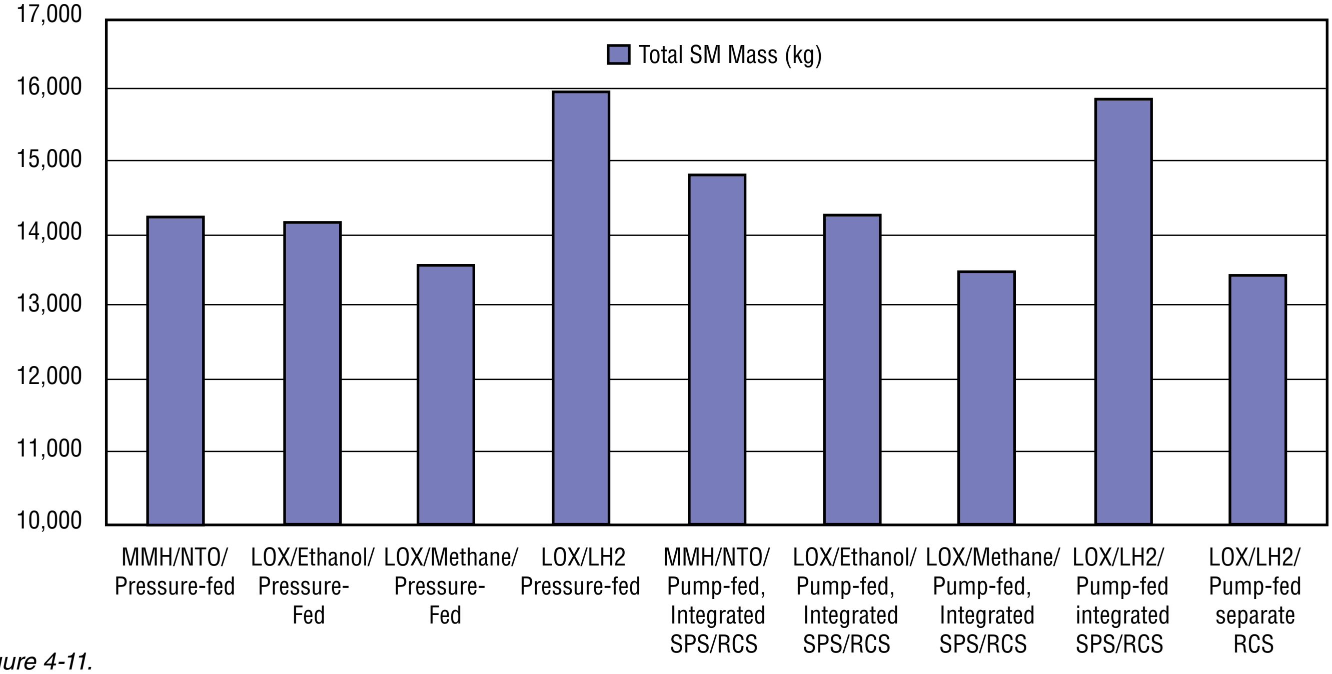


Figure 4-11.
Total SM Mass

The results show that pressure-fed and pump-fed LOX/LCH4 are the lightest mass systems. Pump-fed LOX/LH2 does not perform as well, despite its much higher Isp, due to the increase in structural mass (due to the SM being twice as big), the boil-off for 6 months, the addition of the service propulsion system for redundancy, and a separate RCS. LOX/LH2 is better suited to systems that require more delta-V than an SM. The higher dry mass, volume, and complexity of LOX/LH2 pump-fed systems are the main drivers in selecting an LOX/LCH4 pressure-fed system. The LOX/LH2 stages have the heaviest dry mass, especially the pressure-fed LH2 systems due to the LH2 tank mass at high pressures. The total volume of the propulsion is much larger for LOX/LH2 (by 2.5 times) compared to LOX/LCH4. This has a direct effect on the SM structure. An assessment of the total number of propulsion system components shows comparable totals for all systems except for the LOX/LH2 main system with a separate RCS. For MMH/NTO, the heater components were ignored. This would make it comparable to the other pressure-fed systems. These independent results compare well to the Envision results—within 2 percent.

Reliability Comparisons

The Apollo experience with hypergolic propellants was examined by the ESAS team. The failures indicated similar experiences to Shuttle relative to FORP, although not at the same failure rates. The Shuttle experienced more iron nitrate problems within the NTO system (manifesting itself as flow decay and stiction), although it should be noted that the R4D thruster used during Apollo may have had one oxidizer valve problem. The failures experienced in the Apollo program included four FORP-related engine failures and one oxidizer valve contamination failure.

For LOX/LCH4, the only experience is with ground testing, so its predicted reliability is based on a very limited data set. Key risk areas that have been identified are reliable LOX/LCH4 igniters and thermally efficient RCS propellant feed systems.

Table 4-8 shows the probability of failure modes for the Shuttle Orbital Maneuvering System (OMS). The failure rate is driven by blockages in the feed system, which is usually the result of foreign object debris in the system. For both MMH/NTO and LOX/LCH4, it is assumed that a verification test of the feed line integrity and design pressure drop test capability will be used to screen out blockage failures. **Table 4-8** compares the reliability aspects of these propellant combinations by adjusting the values using qualitative rationale and experienced engineering judgment.

	OMS System		MMH/NTO		LOX/Ethanol/Methane	
	Probability	Contribution	% Adjustment	New Value	% Adjustment	New Value
Failure to Start (igniter vs. hyper problems)	6.38E-07	0%	100%		100%	
Bi-Prop valve	1.10E-05	2%	100%		100%	
Blockage	5.24E-04	84%	10%		10%	
Burnthru	7.86E-07	0%	100%		100%	
Fuel Block	1.06E-05	2%	100%		100%	
Helium	3.43E-05	5%	100%		100%	
Nitrogen	1.01E-05	2%	100%		100%	
OX Block	1.06E-05	2%	100%		100%	
PAD/LAD/PMD	2.32E-05	4%	100%		100%	
Iso Valves	1.32E-06	0%	100%		100%	

Table 4-8. Probability of Failure Based on Shuttle Model and Adjusted for CEV MMH/NTO and LO2/LCH4

The reliability assessment for pump-fed versus pressure-fed was assessed by using pump-fed engine reliability estimates and then deleting components (e.g., engine valves and a thrust chamber assembly) not required for a pressure-fed engine, as shown in **Table 4-9**. The result is that pressure-fed engines are five times more reliable than pump-fed engines.

Table 4-9. Pump-Fed versus Pressure-Fed Engine Reliability

Component	SSME Block 2	RL-10A-4-2	Pressure-fed
Actuators	1.07E-06	10.00%	10.00%
Anti-flood valve	3.58E-09	100.00%	0.00%
Fuel preburner	8.84E-07	0.00%	0.00%
Fuel/Hot gas system	1.20E-05	0.00%	0.00%
Heat exchanger	2.07E-06	0.00%	0.00%
HPFTP/AT	6.99E-05	50.00%	0.00%
HPOTP/AT	4.10E-05	50.00%	0.00%
Igniters	3.58E-09	100.00%	100.00%
LPFTP	1.19E-06	0.00%	0.00%
LPOTP	4.62E-06	0.00%	0.00%
LTMCC	4.69E-05	22.00%	0.00%
Main injector	1.13E-05	22.00%	0.00%
Nozzle	2.82E-05	50.00%	50.00%
Oxidizer preburner	2.44E-06	50.00%	0.00%
Oxidizer system	5.07E-06	100.00%	0.00%
Pneumatic system	7.83E-06	100.00%	100.00%
Powerhead	1.04E-06	100.00%	0.00%
Valves	8.46E-07	100.00%	100.00%
Other risk	4.58E-05	100.00%	10.00%
Failure Rate for 500 sec burn duration	2.82E-04	1.44E-04	2.75E-05

Pressure-fed 5.25 times more reliable than pump-fed

A more detailed reliability and risk assessment for the various in-space propulsion system options that includes an assessment of propellant type, engine-out, pump-fed versus pressure-fed, heritage, and reliability growth is contained in **Section 8**, **Risk and Reliability**.

Restart of Engines After Long Dormant Periods

The restart of engines after long periods of dormancy is driven by the process used to shutdown the engine, the protection provided during mission, and the process used to prepare the engine for restart. The primary issues associated with LOX/LCH4 are how to vent the lines and cavities and potential freezing of propellants. On startup, the cold temperatures in the igniter (less than –120° F) may require heaters. The primary issues with hypergolic propellants are how to shutdown the system for dormancy and safe the lines and engines if wetted. Heaters are required for wetted portions of the system to prevent freezing. Other potential issues are seat swell and flow decay of iron nitrates.

On startup of pressure-fed systems, the lines must be primed with propellants at start box conditions. For hypergolics, it may be required to preheat the propellants with heaters to start conditions. For pump-fed engine startup, the lines are primed with propellants at the start box conditions. For hypergolic engines, there are additional parts (Gas Generator (GG), chamber) for the engine to warm up for ignition. For cryogenic engines such as the RL-10, additional steps to chill-down the pump, condition the igniter temperature, and ensure the proper chamber regeneration circuit temperature for boot strap startup is required.

Engine Throttleability for Descent Stages

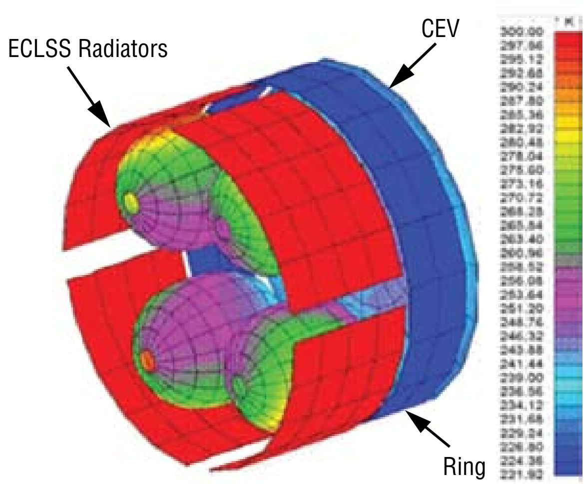
For all throttling engines, it is critical to maintain the injector pressure drops necessary for proper propellant injection and mixing over the throttling range. There are at least two approaches for deep-throttling pressure-fed engines: (1) a sliding pintle to control injection orifice size and (2) dual-circuit injectors. Both approaches have been tested and demonstrated, although additional research is required. In general, engine throttleability does not appear to be a major discriminator between the engine options.

LOX/LCH4 Cryogenic Propellant Management

A review of LOX/LCH4 development history and literature was performed to address cryogenic storage, acquisition, and distribution. In addition, a detailed analysis of the CEV LOX/LCH4 propellant tanks that addressed the environment and the CEV configuration (e.g., radiators) was performed.

Cryogenic propellant handling and storage has been demonstrated for space applications since the 1960s. Some of the more applicable tests have demonstrated flight-weight tanks of 15,000 lbm capacity and heat leaks of 1.3 watts with very low boil-off rates. The recent tank design for Shuttle upgrades for a 10,500 lbm-capacity tank looked at the pressured vessel and Liquid Acquisition Device (LAD) design. Propellant tanks and LADs are critical long-lead items and require significant development and testing in low-g and thermal vacuum tests. The options and issues associated with long-term storage of various cryogenic fluids on the lunar surface were examined. Thermal models of LOX, LCH4, and LH2 tanks were built for the SM. The model shown in **Figure 4-12** predicted 3.3 watts total heat leak. The equivalent boil-off rate for 3.3 watts is 0.15 lbm/hr. The use of H2 in place of CH4 is projected to boil-off at 5 percent per month with this configuration.

- Tanks are surrounded by
  - ECLSS Radiators
  - CEV
  - Engine Plate
- Applied fixed temperatures to
  - ECLSS 300K
  - CEV 300K
- Applied heat load to
  - Engine Plate 200 W/m^2
  - Ring 80 W/m^2
- Tank MLI Temperatures
  - 263K with heat load
  - 253K without heat load
- Sun was not applied



	Engine Shield 253K	Engine Shield 263K	Reference
Insulation heat	1.4	1.6	Scaled from 1997 H2 testing
Strut heat	.93	.93	Utilizing tension straps, existing design
Feed line heat	.6	.6	Modeled
Helium tank	.3	.37	60 layers MLI, coupled to LOX tank
Penetration heat	.1	.1	Small tube and wire harness heat
Total heat (all heat units are watts)	3.3	3.6	

Figure 4-12. Analysis of SM Cryogenic Tank Heat Leak

The cryogenic RCS feed system also requires thermal management. The principle of operation is to use a highly subcooled propellant, such that it can be allowed to warm up by 40°R by the time it gets to the engines, yet still remain a subcooled liquid in the lines. The thruster usage during active periods is approximately 2 lbm/hr, which can absorb approximately than 9.3 watts (32 Btu/hr) into the feedlines and still remain subcooled by 30°R. Therefore, the total heat leak into the feed system from the lines must be less than 9.3 watts (32 Btu/hr). This is achievable, assuming 75 ft for the three manifold feedlines at 0.2 Btu/hr/ft and 0.5 Btu/hr for each of the 16 thrusters and 12 feed system valves. (Note that 9.3 watts is three times the heat leak into the tank. This is felt to be conservative.) Achieving these heat leak rates will be the focus of advanced development tasks. The risk mitigation is for early flights to carry enough margin to vent propellants to condition the feedlines.

Figure 4-13 shows the three redundant RCS manifolds, each with a thermodynamic vent at the end of each manifold that is used to intercept and reject any heat leak. During quiescent periods, there is not enough thruster usage to absorb and reject the heat. There are two possible operational solutions: either (1) vent some propellant at 0.5 lbm/hr thermodynamically to chill the lines or (2) allow the propellant in the lines to turn to gas. This Thermal Vacuum Stability (TVS) gas could be used propulsively for a minimum impulse to control attitude. These modes should be explored in vehicle design trades.

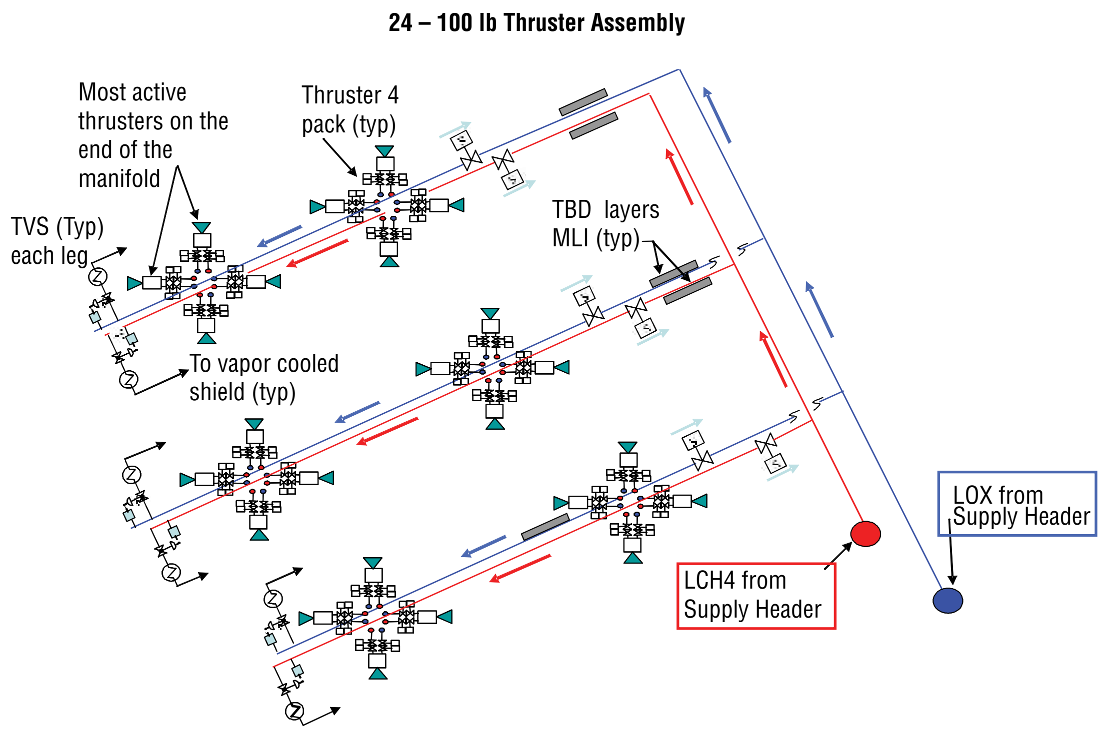


Figure 4-13. RCS Manifolds

Common LADs for use in zero-g use surface tension properties of the fluids to preferentially wet surfaces or screens that lead to the tank outflow port through channels. Fluid acquisition performance is limited by a maximum pressure at which surface tension forces can exclude gas from flowing through the device. This pressure is known as "bubble point," since it is the point where gas bubbles begin to appear on the liquid side of the screen. "Bubble point" is a function of liquid surface tension and pore size. Very fine pore sizes can be obtained with the use of fine-mesh wire screens. Unfortunately, because the pores of the fine-mesh screen are irregular in shape, an analytical calculation of the "bubble point" is not possible. Thus, the fine-mesh screens for LADs require empirical characterization with the liquid of interest. Most of the testing with cryogenic propellants in the literature has been with LH2, since this is the propellant with the lowest surface tension. LOX testing is currently in progress at NASA GRC, and CH4 testing could use the same GRC hardware.

In reviewing CEV operations, it was concluded that the LOX/CH4 RCS will require low-g acquisition systems, while the LOX/CH4 propulsion system could use settling thrusters. However, integration of the RCS and the propulsion system could provide start capability without settling burns. Based on this assessment, a liquid acquisition strategy was proposed, using a compartmented tank similar to the current Shuttle OMS tanks shown in **Figure 4-14**. The upper compartment contains propellant for main burns with acquisition via settling. The upper compartment is unscreened except for a communications window to the lower compartment. The lower compartment contains a series of channels with one side covered in a fine-mesh screen. The channels are located so that some portion of the fine-mesh screen is always in contact with liquid propellant, regardless of gravity environment. The lower compartment contains propellant for the RCS as well as enough propellant to run the main engine from start until propellant settling is achieved.

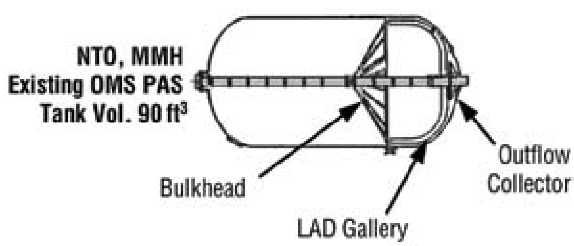


Figure 4-14. OMS Tank Internals Showing LAD Approach

State-of-the-art storable LADs are routinely flown without a low-g flight test for commercial applications. The Shuttle uses screen channel LADs in both RCS and OMS tanks. Shuttle LADs are qualified by a subassembly ground test, a KC–135 test, computer modeling, and flight demonstration during the first four Shuttle missions. Although the Shuttle LADs have operated reliably throughout the Shuttle program, a review of the history revealed several issues with this approach. Technology Readiness Level (TRL) 6 was not achieved until the end of the first four Shuttle mission flights. On-orbit flight operation revealed start transient pressure drops not anticipated by the ground program. The problem was controlled by reducing the number of RCS engines allowed to fire simultaneously. Another issue was that screens required extensive handwork repair to maintain "bubble point" throughout. The Shuttle LADs qualification program also required 7 years to complete.

The approach to qualifying cryogenic LADs for the CEV is based on the approach used to qualify the Shuttle OMS and RCS storable LADs. The same strategy is used for both LOX and methane. Key areas or issues to address are fluid properties for the design region and screen "bubble point" data for the fluids and temperatures of interest (i.e., subcooled LOX viscosities). First, the characterization of fine-mesh screens with LOX would be performed. Secondly, existing equipment would be used for CH4 screen characterizations. Once the characterizations are complete, a new rig for ground testing full-size screen channels will be built. It will be necessary to contract with aerospace manufacturers for LAD build-and-assembly, since capabilities for working with fine-mesh screen and assembly of complex LAD assemblies are limited within NASA. A low-g flight will be required to fully qualify the design. This could be the first flight of the full-up system if the program is willing to take the risk. Risk mitigating factors are that (1) the fluid properties of LOX/CH4 are very similar to storable propellants and (2) a TRL 5 design with design margins was successfully used on the Shuttle program. If the first flight approach is undertaken, it will be critical to build design margins into the tank LAD and then gradually expand the operating envelope.

LOX/LCH4 Engine Ignition and Combustion

The physical properties of LOX/LCH4 combustion were examined to attempt to identify any issues. The ignitability of methane in pure oxygen is suitable for the range used in rocket engine spark igniters. Methane is flammable from 0.66:1 to 20:1. The flame speed is good at 43 cm/sec (ethanol is 45 cm/sec). The minimum ignition energy is 0.28 mj for methane. (Ethanol is 0.15 mj.) (Note: RCS engine igniters use 10 to 90 mj sparks to improve ignition.)

A survey of the engine and component test history for LOX/LCH4 was examined. Upper stage work has primarily been in the form of engine studies with some injector and igniter component testing. No major issues were found, but significant advanced development is required. One major finding is that future LCH4 specifications should ensure that sulphur has been removed from the propellant.

NASA GRC has performed ignition and combustion tests on oxygen and methane. In the early 1980s, Liquid Natural Gas (LNG) test programs performed combustion performance and testing. A hydrocarbon ignition project, shown in **Figure 4-15**, was performed in 1987 and 1988, and a Combustion Wave Ignition (CWI) was performed in 1992. A laser-induced spark ignition test program with Gaseous Oxygen (GOX)/methane was also performed in 1994.

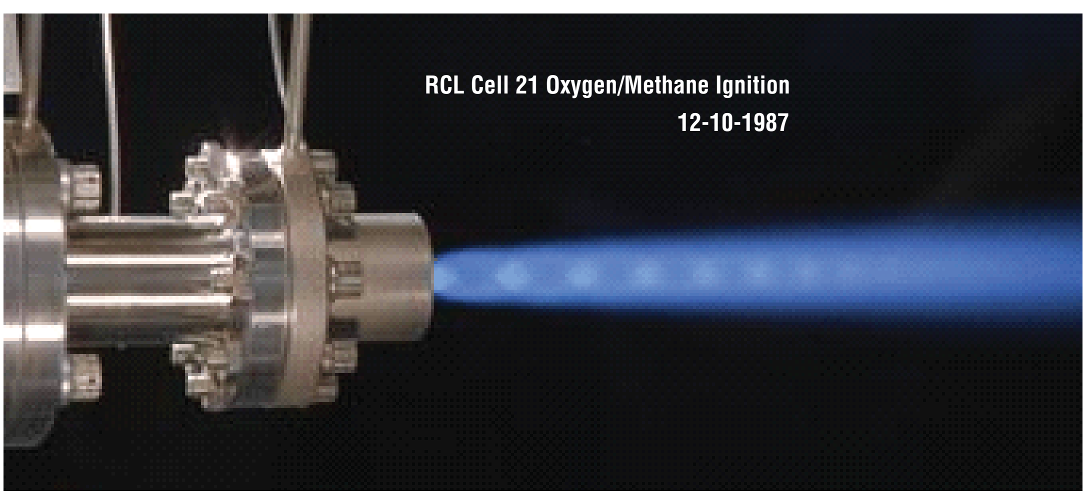


Figure 4-15. Hydrocarbon Engine Test Project

Rocketdyne has performed LOX/LCH4 tests on various contracts primarily related to large engine chambers and gas generators. The High-Pressure LOX/CH4 Injector Program, NASA Marshall Space Flight Center (MSFC) contract NAS8–33206 (1978–1979), examined coaxial and impinging elements for high-pressure LOX/CH4 operation with an existing 40 klb-thrust chamber. The shear coaxial element injector was fabricated, flow-tested, and delivered to MSFC for hot-fire testing. In 1988, MSFC tested the 82-element injector over nine main stage tests at Pc = 1207 to 2381 psia, •/F = 2.48 to 3.49, and C* efficiency = 98.3 to 100.6%.

Rocketdyne, as part of the Methane Heat Transfer Investigation for NASA MSFC contract NAS8–34977 (1985), investigated the cooling and coking characteristics of LNG (92.5 percent methane) using an electrically heated bimetallic tube test apparatus. The project completed 37 tests over a range of heat fluxes Q/A = 1.6 to 85 Btu/in2-sec, coolant velocity = 181 to 781 ft/sec, and coolant pressure = 3,914 to 4,966 psig. This established the Nusselt number correlation of LNG-cooled channel wall combustors. Corrosion of the copper wall was attributed to trace amounts of sulfur in the LNG. The corrosion had an influence on heat transfer and pressure drop. No evidence of coking was seen, even at wall temperatures over 870° F.

Aerojet has experience with gaseous and liquid oxygen-based RCS, primarily with ethanol, but also with some methane tests. Aerojet has developed and delivered the hardware for a 500-lbf GOX/Gaseous Methane (GCH4) system for the X-33 (**Figure 4-16**). Successful ignition was obtained, except under certain conditions. Aerojet has designed a 150-lbf GOX/ethanol thruster, designed and tested a 620-lbf GOX/ethanol thruster, and designed and tested an 870-lbf LOX/ethanol thruster. Several tests with LOX/methane injectors were performed in the 1980s. A LOX/LCH4 30-klbf @ 2,200 psia injector was extensively tested at MSFC.

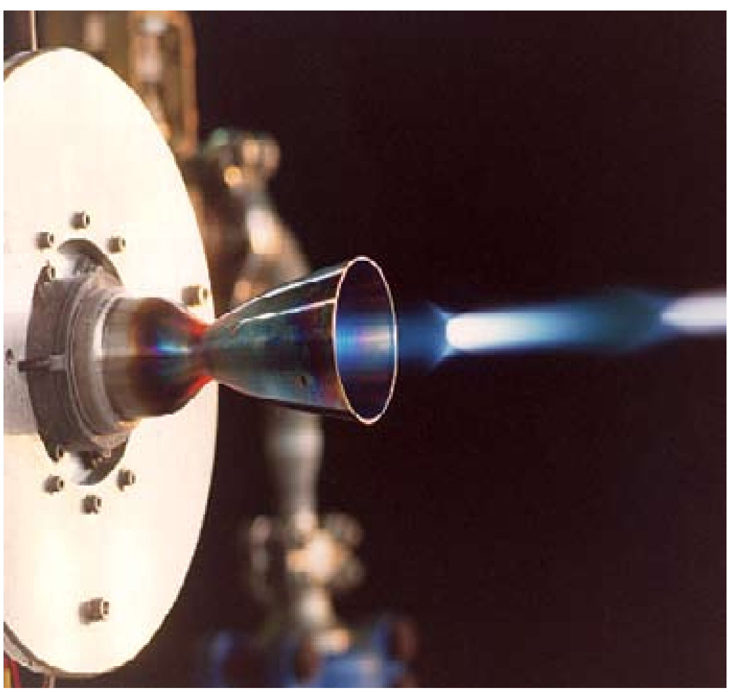


Figure 4-16. 500-lbf GOX/GCH4 Thruster

Russian engineers have designed and extensively tested entire LOX/methane engines of various thrust levels, but none have been flown to date.

LOX/LCH4 Development Issues and Risks

It was determined that, in order to reduce risk for development, a LOX/LCH4 propulsion system must undergo advanced development to a prototype system level of flight-weight pressurization, tanks, feed system, and engines. These advanced development tasks must address cryogenic storage, liquid acquisition, a cryogenic liquid RCS feed system, and engine tests with an emphasis on ignition.

Key highest risk areas include the following:

- Igniter design for LOX/LCH4 ignition over a wide range;
- Flight-weight robust spark plug;
- Cryogenic RCS feed system; and
- Propellant management device.

Moderate risk areas include the following:

- Flight-weight excitor;
- Light-weight pressure vessel (composite overwrapped Aluminum-Lithium (AL-Li));
- Propellant isolation valve;
- Low-heat leak cryogenic tank;
- LOX/LCH4 injector and chamber; and
- Engine valves.

Based on these risks, schedules for a 10-klbs-thrust class, pressure-fed LOX/methane engine system that is throttleable 10:1 for a 2018 human lunar landing were developed. It was determined that a non-throttleable version for the ISS SM in 2011 is feasible, but requires significant advanced development.

4.2.4.1.3 Surface CEV Configuration Studies

One of the three mission modes under consideration, EOR-direct return, requires the CEV to descend to the lunar surface, where it serves as the crew's surface habitat for the duration of the lunar sortie surface mission. This mission mode is potentially attractive because it eliminates the development of a costly second crew cabin for the LSAM; however, it does require the CEV to perform a number of new and unique functions. In addition to serving as the crew cabin for Earth ascent, LEO rendezvous and docking, trans-lunar cruise, lunar orbit operations, trans-Earth cruise, and Earth entry, the EOR-direct return CEV must also be configurable for lunar ascent and descent powered flight and the EVA support and surface habitation functions of the surface mission.

The ESAS team also recognized that the EOR-direct return mission may have advantages in terms of risk and reliability because of the reduced number of operations and vehicles. To fully analyze this mission mode, the team had to compare its cost, performance, and risk characteristics to other mission modes. Like other mission modes, the technical performance of the EOR-direct return mission began with an analysis of the CEV volume required for this distinct mission type. The direct return CEV would require more habitable volume than the LOR-based CEVs because it must accommodate the same crew of four in lunar surface EVA suits in a gravity environment. In the following analysis, this volume was configured two different ways: first, as a single volume where all end-to-end mission functions were performed, and, secondly, as a "split volume" where the surface-specific airlock and EVA functions were split from the remainder of the CEV functions.

EOR-Direct Return Volume Overview
The required equipment volume for the single volume, EOR-direct return CEV is 15.9 m^3, which is somewhat larger than required for the LOR-based LSAM due to the additional provisions for the total mission duration, not just for the lunar portion. The net habitable volume required of the single volume, EOR-direct return CEV is 21.4 m^3, which is exactly the same as for the LOR LSAM, because EVA lunar suit stowage is required. Therefore, the total required pressurized volume is 37.3 m^3. In addition to lunar EVA suit stowage, the other driver of net habitable volume is serial donning of lunar surface EVA suits, requiring 17.3 m^3 of net habitable volume and 33.2 m^3 of total pressurized volume. The larger equipment volume accounts for the larger total pressurized volume of the single volume, EOR-direct return CEV compared to the LOR LSAM, since the net habitable volumes are the same.

Split-Volume EOR-Direct Return CEV Configuration
Configuring a split-volume vehicle where the CEV is taken to the surface, but an airlock module could be left behind following lunar ascent, proved to be a design challenge. One way to approach this challenge is to invert the CEV so that its docking adapter would be oriented toward the descent stage airlock. This would require the CEV to be configured for partial gravity lunar surface operation in a suboptimal apex-nadir configuration, but this configuration would be suitable for dust control. With a potential split-volume, inverted EOR-direct return CEV design, the team developed an internal layout to assess the feasibility of this configuration. Two initial assumptions that governed the layout were that four crew members should be able to stand together during landing and that there was enough horizontal surface area for four people to sleep. This led to an inverted "wedding cake" layout that provided adequate habitable and equipment volume, as given in the volume analysis detailed in **Section 4.2.3.2.1**, **CEV and Lunar Surface Access Module (LSAM) Volume Studies**. Human factors engineers constructed a layout within the pressure vessel defined by the CEV engineers. With this layout, it was believed that the equipment and net habitable volumes determined in the volume analysis would not be readily achieved without reconfiguring the vehicle (i.e., utilizing sleep areas for stowage while the crew was awake). The overall concept, shown in **Figures 4-17** and **4-18**, appeared to still be feasible from an initial layout standpoint.

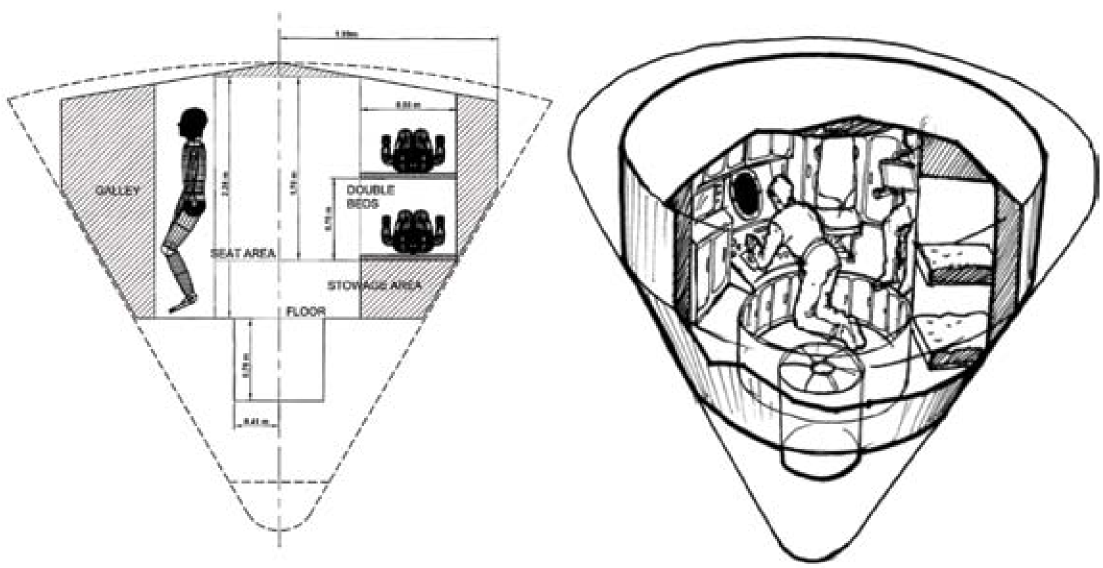


Figure 4-17. Split-Volume EOR-Direct Return CEV Layout

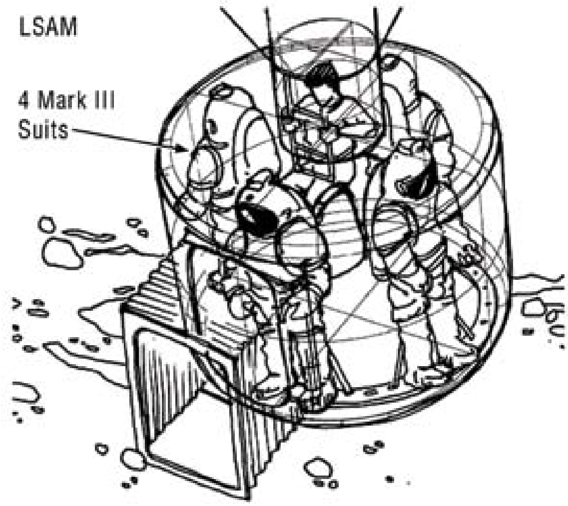


Figure 4-18. Split-Volume EOR-Direct Return CEV Airlock Module

Alternate approaches were investigated with "apex-up" configurations. Placing the airlock on top of the vehicle required it to be far too high off of the lunar surface for safe ingress and egress. Placing the airlock on the side was problematic for CEV Earth ascent and entry configurations.

The team also created layouts for the accompanying airlock module, which would be attached to the bottom of the inverted CEV and used for lunar EVA, including volume for stowage and donning of lunar suits and depress/repress operations. Assumptions for this module were that this would serve as the airlock rather than depressing the entire CEV, lunar suits would most closely resemble the Mark III rear-entry suit, and donning could occur sequentially, with one crew member donning at a time. No CAD models were created for this module, and dimensions were defined by NASA's Habitability and Human Factors Office (HHFO) to achieve minimal volumes required for the given tasks.

Because the initial resulting volume required for the airlock module was large (16.3 m^3 for serial lunar suit donning), the team consulted the experts in EVA operations to explore other lower-volume solutions. It was believed that since only one or two crew members would don suits and depress an airlock at a time, the original four-person layout could be reduced. Several configurations were considered, including two to four one-person suitlocks and a two-person airlock as shown in **Figure 4-19**.

Because the EOR-direct return mission mode is unique, the ESAS team believed it was important to show layouts of the vehicles in different mission phases, including launch, lunar landing, and lunar surface operations. This would illustrate some of the operational complexities of the inverted CEV configuration, which would be "upright" during Earth launch and reentry, but reconfigurable for inverted partial-g operation while on the lunar surface. (Refer to **Figure 4-20.**) These illustrations also point out that a critical docking of the CEV to the lander/airlock module must take place in LEO and must establish all structural, electrical, command, control, and telemetry interfaces between the vehicles.

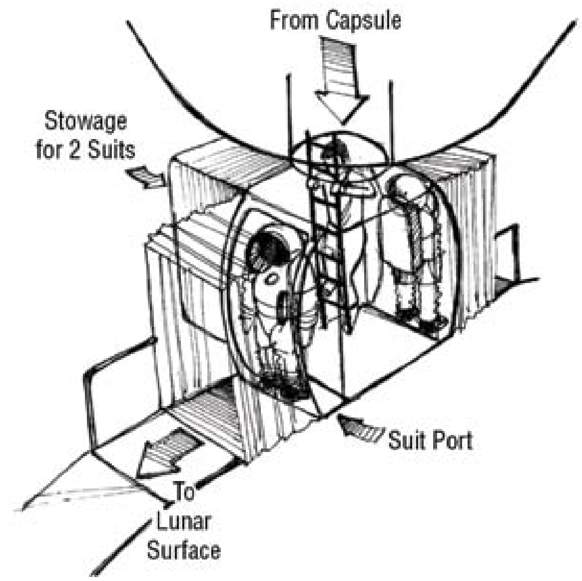


Figure 4-19. Alternative Airlock Module Layout

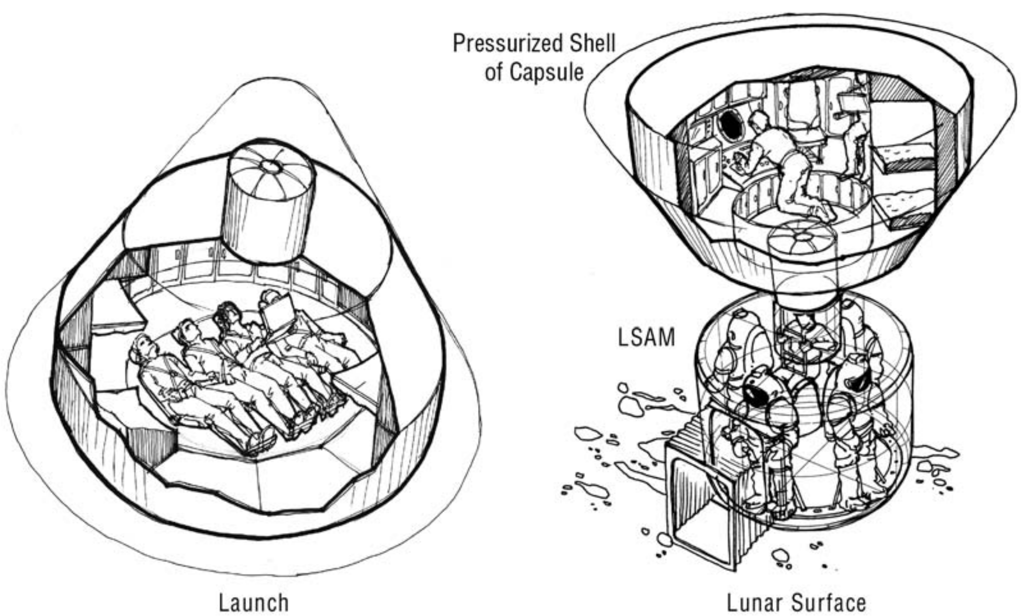


Figure 4-20. Split-Volume EOR-Direct Return CEV + Airlock Module Layout by Flight Phases

Single-Volume EOR-Direct Return CEV Configuration

The team then began to develop a single-volume, "apex-up," internal layout for a 5.5-m/25-deg single-volume CEV, given a pressure shell with a pressurized volume of 36.6 m^3. Because it was believed that depressing the entire module for EVA operations was operationally difficult, studies were conducted to configure an internal airlock within the layout. Lunar dust was identified as a potential problem and an attempt was made to build dust mitigation into the design.

In order to determine the feasibility of this CEV size and shape, specific layout designs were developed, including the location of sleep areas, workstations, airlocks, stowage, and dust mitigation areas. These layouts were intended to be study designs that would represent only one possible solution in order to determine the feasibility of the vehicle volume. Each layout contained a dust mitigation area, with one utilizing a hard wall, and the other utilizing more of a soft wall/curtain concept. Some of the study designs are shown in **Figure 4-21**.

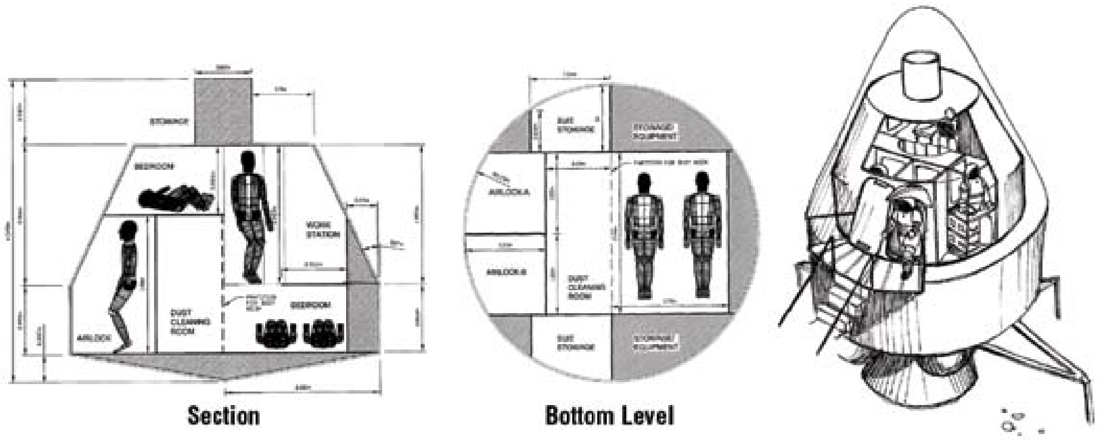


Figure 4-21. Single-Volume CEV Study Design Layouts

Each configuration generally met the volume goals of **Section 4.2.3.2.1**, **CEV and Lunar Surface Access Module**, although the hard wall concept loses some volume usability since the dust area becomes unusable for any other functions. The ESAS team asked for additional layouts that eliminate the internal airlock, use an updated avionics equipment location from the EX model, and eliminate any "hard walled" partitions. The new avionics layout allowed designers to lower half of the floor to increase habitable volume. This layout (**Figure 4-22**) became the basis of comparison for EOR-direct return missions. The final layouts for the single volume, direct-return CEV are shown in **Figures 4-23** and **4-24**.

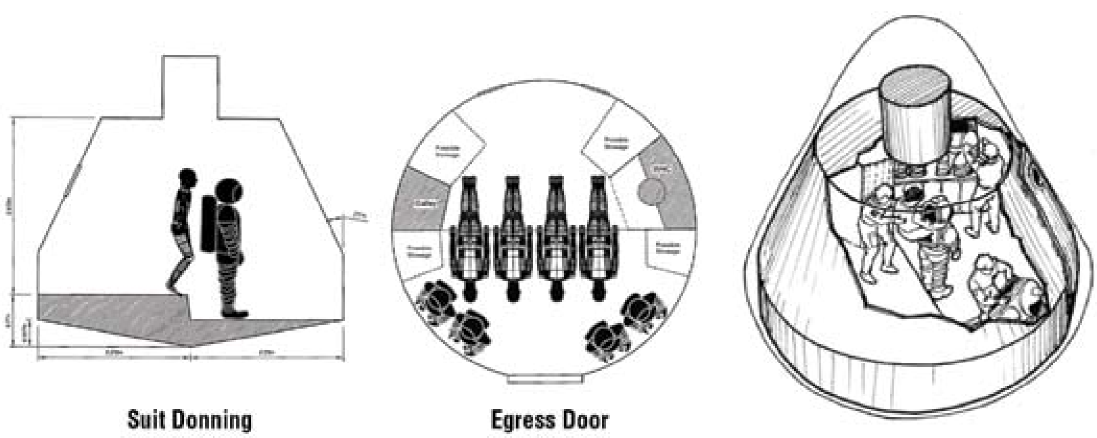


Figure 4-22. Single-Volume EOR-Direct Return Initial Conceptual CEV Layouts

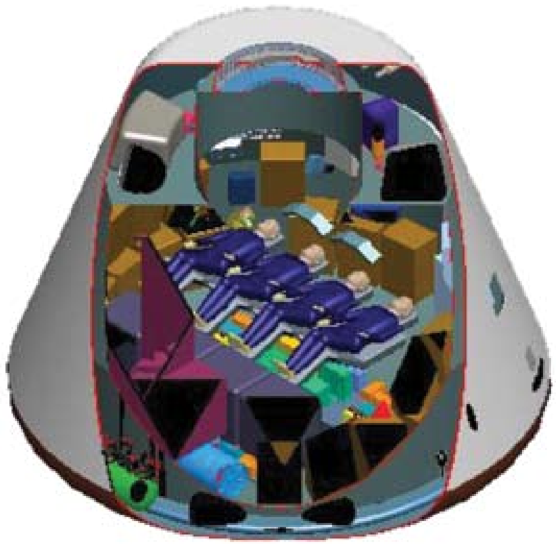


Figure 4-23. EOR-Direct Return CEV CAD Cutaway Showing Four-Crew Earth Launch Configuration

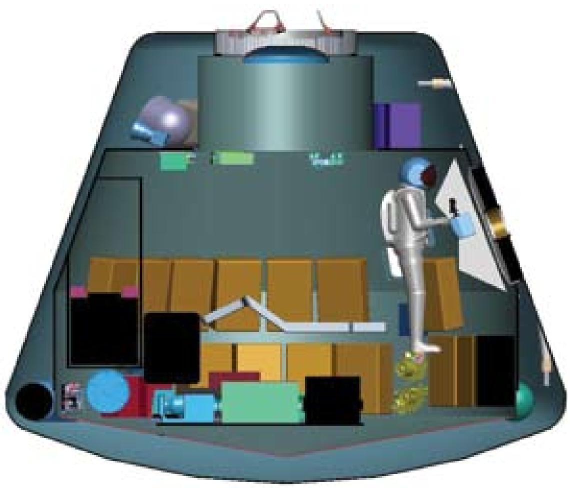


Figure 4-24. EOR-Direct Return CEV CAD Section Showing Lunar Ascent and Descent Pilot Position

Conclusions

The EOR-direct return mission mode has substantial design and risk implications to the CEV design. The surface-direct mission requires greater volume than other mission modes, and this volume could be implemented either in a single- or split-volume configuration. The split-volume configuration offers an intriguing opportunity to deploy a habitable surface element (the airlock module) each time a sortie mission visits the surface, but also subjects the CEV to inverted landing and adds significant weight and complexity. Although the ESAS team did not perform a comprehensive analysis of how the CEV internal arrangement would be reconfigured for lunar ascent, descent, and surface operations, it was clear that this would add substantial design complexity to the vehicle and operational complexity for the crew.

The configuration study also concluded that the integration of an airlock, either within the CEV or supplemental to the CEV, would add complexity and mass to the design. Control of lunar dust has implications in protecting the vehicle subsystems for future use, and the additional airlock mass could make the direct mission mode infeasible for launching on two vehicles. For this reason, the ESAS team recommended a focused study to determine the need for a surface airlock.

4.2.4.1.4 Airlock Trade Study

Process

This study was conducted to generate the information required to support a decision on whether to include an airlock in the LSAM. The information generated during this task drew heavily on findings from recent ESMD studies. Using these studies as a starting point, the ESAS team refined the analysis using inputs from the ESAS lunar architecture.

The ESAS team focused primarily on the hazards and contingencies that an airlock would mitigate. In doing so, the team first agreed on the set of hazards and contingencies to which an airlock was a solution. Next, the team systematically assessed each hazard/contingency, listed alternate solutions, and described the end-state that each solution provided. It was found that an airlock placed on the LSAM would protect against two major classes of problems:

- Constant: Dust, and
- Contingencies: Illness, injuries, suit malfunctions.

Dust is a quantifiable, known problem that must be controlled. The Apollo astronauts found that dust posed a hazard to hardware and crew health. Transcripts and anecdotal comments from Apollo astronauts describe the magnitude of the problem posed by dust, which permeated the crew cabin, covered EVA suits/tools, and soiled the field experiment hardware. It also proved to be a source of respiratory and eye irritation for a number of the crew members.

Contingency scenarios were considered in terms of likelihood, vehicle design/operations options, and mission/crew risk. Many of the contingency situations considered had a low likelihood of occurring. Furthermore, many of the contingencies had either vehicle design and/or operational work-arounds that did not carry the mass and configuration impacts that an airlock would impose. For each work-around, the impacts to mission and crew risks were assessed. Coupled together, this information provided the necessary insight to assess the risk posed by the contingencies. In the end, the decision is largely based on the level of risk that the program is willing to accept.

Finally, while hazard and contingency mitigation was the focus of this study, mention should be made to the operational flexibility that would be gained if an airlock were to be included in the LSAM configuration. The primary operational benefit afforded by an airlock is that it allows for split-crew operations, which can be advantageous in at least two major scenarios. First, it allows one team to conduct an EVA while the other team remains inside the vehicle with suits doffed (perhaps safing the LSAM after landing or prepping the vehicle prior to lunar-surface-departure). Secondly, it allows multiple EVA teams to "split off" in different directions on the lunar surface and return to the LSAM when their team's tasks are complete, without being required to wait for the other team(s) to complete their tasks.

Results

The hazards/contingencies, alternative solutions, and respective end-states that were considered are shown in **Table 4-10**.

Table 4-10. Hazards/ Contingencies, Alternative Solutions, and Respective End-states

Problem	Time Period	Comments	Mitigation or Solution	End-State
Dust	Constant	Solution must meet Spacecraft Maximum Allowable Concentration (SMAC) limits and hardware design specs.	Airlock	Majority of dust is controlled.
			Separate volume (without airlock capabilities)	
			Dust removal method prior to ingress.	
Crew member illness or injury	Prior to donning suit	Needs a solution: too ill/injured to don suit, but not ill/injured enough to merit a return to Earth.	Airlock—crew member remains inside crew cabin.	Ill/injured crew member refrains from EVA. No LOM.
			Internal pressurized volume houses affected crew member.	
			Cease all EVAs until illness/injury subsides.	Mission continuance depends on status of ill/injured crew member.
	During EVA	Crew member needs to return to lander and doff suit.	Airlock—crew member ingresses lander.	Other crew members' activities are unaffected. No LOM.
			Crew member ingresses internal pressurized volume.	
			Operationally constrain EVA traverse distances and distances between EVA teams.	All crew members return to lander together. Mission continuance depends on status of affected crew member.
Suit malfunction	Prior to initiation of lunar descent	Assume problem is too big to fix with available tools/parts.	Airlock—crew member remains inside crew cabin.	All crew, except affected crew member, performs EVA. No LOM.
			Internal pressurized volume houses affected crew member.	
			Affected crew member uses launch/ascent suit.	
			Mission is scrubbed.	Return to Earth
		Assume problem can be fixed with available tools/parts.	Airlock—crew member remains inside crew cabin to fix suit.	All crew members perform EVA; affected crew member joins when ready. No LOM.
			Internal pressurized volume houses affected crew member while fixing suit.	
			Scrub EVA until suit is fixed.	All EVA scrubbed until suit is fixed. No LOM.

Table 4-10. (continued) Hazards/Contingencies, Alternative Solutions, and Respective End-states

Problem	Time Period	Comments	Mitigation or Solution	End-State
Suit malfunction	Prior to egress from lander for EVA	Assume problem is too big to fix with available tools/parts.	Airlock—crew member stays inside crew cabin.	All crew, except affected crew member, performs EVA. No LOM.
			Internal pressurized volume houses affected crew member.	
			Affected crew member remains inside lander using launch/ascent suit.	
			Mission is scrubbed.	Return to Earth.
		Assume problem can be fixed with available tools/parts.	Airlock—crew member fixes suit in crew cabin.	All crew members perform EVA; affected crew member joins when ready. No LOM.
			Internal pressurized volume provides area for crew member to fix suit.	
			Scrub EVA until suit is fixed.	All EVA scrubbed until suit is fixed. No LOM.
	During EVA	Catastrophic suit failure	None	Loss of crew member.
		Assume major failure —can be contained or controlled in order to allow crew member to return to lander, but with no time to spare.	Airlock—crew member ingresses lander.	Other crew members' activities are unaffected. No LOM.
			Crew member ingresses internal pressurized volume.	
			Crew member ingresses lander and dons launch/ ascent suit.	Conditional work-around—time required to doff, repressurize cabin, and switch suits.
			Operationally constrain EVA traverses.	All crew members return to lander together. Mission continuance depends on status of affected crew member.
		Assume minor failure —can be contained or controlled, allowing crew member to return to lander with time to spare.	Airlock—crew member fixes suit in crew cabin.	All crew members perform EVA; affected crew member joins when ready. No LOM.
			Internal pressurized volume provides area for crew member to fix suit.	
			Suit is connected to umbilical to resupply.	Conditional work-around—Applicable if malfunction is consumables-related. Consumables must be available.
			Affected crew member ingresses lander, repressurizes, and dons launch/ascent suit.	Conditional work-around—time required to doff, repressurize cabin, and switch suits.
			Operationally constrain EVA traverse distances and distances between EVA teams.	All crew members return to lander together. Mission continuance depends on status of affected crew member.

In listing the hazards/contingencies, it was necessary to also list the time period during which the hazard/contingencies might occur. For example, it would not have been practical to list "injured crew member" and then try to find mitigation techniques that served as alternatives to airlocks. Instead, it was necessary to define whether the injury occurred in the crew cabin prior to donning an EVA suit or whether the injury occurred while on EVA. The available solutions and end-states to these two problems were different. Additionally, if the latter were to occur, the former would still require a solution for the injured crew member, but not vice versa.

An example of a potential problem that required further definition is in consideration of a "crew member illness or injury." This problem only required a solution if the crew member was too ill or injured to don a suit, but not ill or injured to the point to merit a return to Earth. Obviously, this significantly limited the severity of the illnesses or injuries under consideration and centered the discussion on the likelihood of such an illness or injury. Another example of a problem that required further analysis can be found in the discussion of a "suit malfunction." This problem required that the team examine solutions to malfunctions that were too large to fix, as well as those that could be fixed in-situ. Again, the available solutions and end-states to these two classes of problems were different.

Finally, for each hazard/contingency, viable work-arounds were found. As previously stated, many of these work-arounds did not carry the mass and configuration impacts that an airlock would impose. However, when taken together and when the operational flexibility afforded by an airlock was considered, it was felt that a strong case could be made for the inclusion of an airlock in the LSAM configuration.

Ultimately, the decision to include an airlock in the LSAM configuration is largely dependent on the level of risk the program is willing to accept. As such, it must be made in the context of the key mission parameters and program guidance, as defined by the ESAS team and included below:

- There will be multiple sortie missions in the lunar program.
- All other missions (potentially spanning decades) will be to the vicinity of a habitat or some other pressurized element. In this situation, the crew will egress the lander on landing and ingress the lander at the end of the surface mission. The assumption can be made that, during these types of missions, the lander will not be ingressed nor egressed more than this.
- There will be four EVA crew members.
- The surface mission duration may extend up to 7 days.
- EVA by all crew members is assumed on at least 4 days.
- As currently defined, the lunar sortie surface mission objectives are to perform science, demonstrate the transportation system, opportunistic technology demonstration, and opportunistic surface operations demonstration.

Based on the information in this study, the ESAS team ultimately advised that an airlock be included in the LSAM configuration. Though each of the issues identified in this study (suit malfunction, crew illness, and dust control) could be potentially solved by a combination of cleaning equipment, suit spares, umbilical capability, and operational constraints, the sum of these issues would impose difficult operational requirements on the lunar surface mission. Thus, although a surface airlock is not strictly required, the ESAS team concluded that it is strongly desirable.

The study also concluded that incorporating an airlock into the LSAM crew cabin is easily workable; however, adding an airlock to a CEV is difficult. In particular, adding an airlock to a direct return CEV will consume all available mass margin and further complicate the configuration of the CEV used for this mission mode. An inverted CEV with a separate airlock mounted below the CEV appeared feasible if mass margin allows the addition of a second pressurized volume. In general, airlocks become more essential as the number of ingress/egress cycles increase. The ESAS team concluded that, for a lander that simply rotates crew to an outpost, an airlock is not required; for 7-day sortie-class accessibility, an airlock is strongly desired; and, for an outpost mission, an airlock is essential.

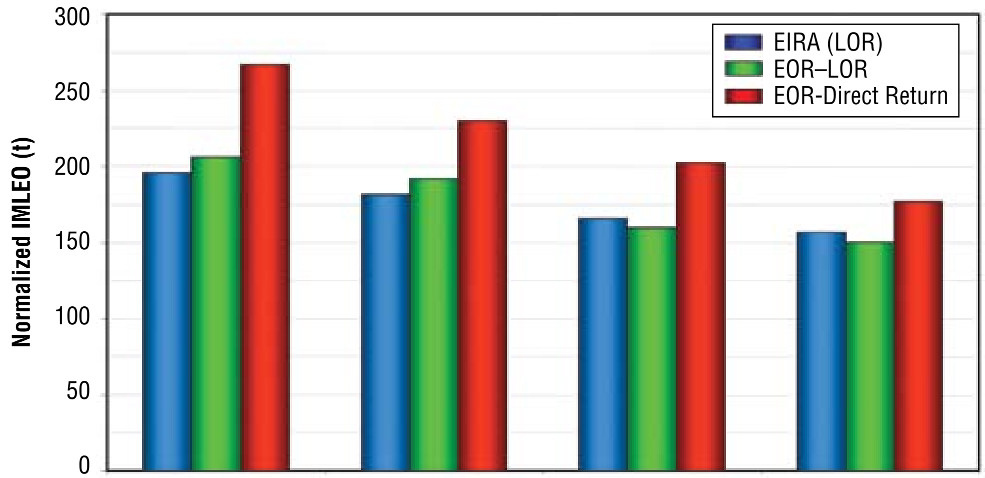
4.2.4.2 Cycle 2 Performance

Based on the FOM analysis comparing the Cycle 1 architectures, the EOR–LOR with CEV-to-surface architecture was eliminated from future consideration. Design Cycle 2 subsequently focused on the remaining three options: (1) LOR (EIRA), (2) EOR–LOR, and (3) EOR-direct return. In this cycle, a uniform CEV capsule shape was also used for comparing architectures. The selected shape was the 5.5-m diameter, 25-deg sidewall angle used in Cycle 1 EOR-direct return. Previously, the two LOR architectures had assumed the smaller-volume 5.0-m diameter, 30-deg shape. Design Cycle 2 also introduced the potential for including more advanced propulsion technology than pressure-fed oxygen/methane in the various architecture elements, as well as an analysis of eliminating the supplemental 5 g/cm^2 of HDPE radiation shielding on the CEV. The advanced propulsion options included using a pump-fed oxygen/hydrogen system on the LSAM descent stage and using pump-fed oxygen/methane on the ascent and return systems.

Figure 4-25 shows a comparison of LOR, EOR–LOR, and EOR-direct return normalized IMLEO for the design options analyzed in Cycle 2. The data on the far left of the figure represents normalized mass using the same assumptions as Cycle 1, with the exception that the two LOR architectures use the same CEV CM shape as EOR-direct return. This change has caused the total mass of these architectures to increase above that which was illustrated previously in **Figure 4-8** of **Section 4.2.3**, **Analysis Cycle 1 Mission Mode Analysis**, while the direct return option has remained relatively constant. The slight increase is due to additional fidelity in the vehicle designs. The remaining three data sets illustrate how architecture mass decreases with increasing propulsion technology and the elimination of supplemental radiation protection. When polyethylene shielding is removed from the CEV (1,860 kg), the total mass for all three architectures decreases, though the greatest reduction is seen in EOR-direct return. This was expected because mass reduction on the CEV has the greatest leverage on the overall architecture when the vehicle transits all the way to the lunar surface and then returns to Earth, rather than only transiting to and from the lunar orbit.

The next set of analyses focused on the effect of incorporating pump-fed oxygen/hydrogen propulsion (rather than pressure-fed oxygen/methane) into the LSAM descent stage. Such a system enables propellant mass savings by significantly increasing the engine Isp and reducing the mass of propellant and pressurization fluid storage. Once again, all three architecture alternatives achieve a significant mass reduction, while EOR-direct return is most affected. Interestingly, this change enables EOR–LOR to now have a slightly lower normalized IMLEO than LOR. In Cycle 1, the lowest mass EOR–LOR mission design required the CEV to perform the LOI maneuver for both the CEV and LSAM. When a pump-fed oxygen/hydrogen system is available on the descent stage, it is more effective to use this stage (rather than the pressure-fed oxygen/methane system) for LOI on the CEV. Finally, the option to use a pump-fed oxygen/methane system is introduced for the ascent and TEI maneuvers in the final set of architecture masses. **Figure 4-25** clearly demonstrates that this particular propulsion technology has a relatively minor impact on the architectures using LOR as compared to the previous design changes and the savings achieved with pump-fed return in EOR-direct return. The effect of slightly higher Isp and lower inert mass with pump-fed systems is more pronounced when the stage in question is performing as much delta-V as possible or is transporting as much mass as possible. The ascent and return function with EOR-direct return is combined into a single stage performing the entire delta-V required, while the total delta-V with LOR is

performed by two stages (with each allocated a portion of the total). Thus, one would expect pump-fed return propulsion to reduce normalized IMLEO for the EOR-direct return architecture more than the other two architectures. **Figure 4-25** also shows that, when all propulsion technologies are applied and supplemental radiation shielding is removed from the CEV, EOR-direct return has an IMLEO only 15 percent higher than LOR or EOR–LOR
Another important metric to use when evaluating competing architectures is the amount of additional mass margin each provides. This metric is a measure of an architecture's flexibility and robustness. For example, mass margin on a given element could be used to cover future

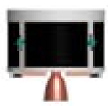


CEV Radiation	+1.8 t Supplemental Polyethylene Shielding	No Supplemental Shielding	No Supplemental Shielding	No Supplemental Shielding
Descent Propulsion	LOX/Methane Pressure-Fed	LOX/Methane Pressure-Fed	LOX/Hydrogen Pump-Fed	LOX/Hydrogen Pump-Fed
Return Propulsion	LOX/Methane Pressure-Fed	LOX/Methane Pressure-Fed	LOX/Methane Pressure-Fed	LOX/Methane Pump-Fed

Figure 4-25. Normalized Architecture IMLEO with Increasing Propulsion Technology

mass growth, add additional capability within the vehicle's subsystems, or transport additional useful cargo to or from a destination. For example, if the CEV radiation protection was eliminated, oxygen/hydrogen was used in the LSAM descent stage, and no elements exceeded their mass allocations (over and above the 20 percent dry weight margin already included), the descent stage could land an additional 6,450 kg of useful cargo on the lunar surface if the EOR–LOR architecture was selected. **Table 4-11** shows the amount of additional mass margin available for any one element in the three architecture alternatives without exceeding allowable launch limits. Only 2-launch architecture solutions were considered and, as illustrated in **Table 4-11**, even when the CEV supplemental radiation shielding is removed, EOR-direct return is still a 3-launch solution for the all-pressure-fed methane propulsion cases. The baseline EOR–LOR mission, with the larger CEV CM than was used in Cycle 1, may also be considered a 3-launch solution because all elements have negative margin. All architectures are 2-launch solutions and have positive margins when pump-fed oxygen/hydrogen is incorporated in the descent stage. However, while LOR and EOR–LOR have robust margins with only this propulsion technology, EOR-direct return also requires pump-fed methane for ascent and return for a comparable level of architecture margin.

Table 4-11. Margins for Cycle 2 Architectures



Technology Switch	CH4 Ascent/TEI	CH4 Ascent/TEI	CH4 Ascent/TEI	Pump-Fed CH4 Ascent/TEI
	CH4 Descent	CH4 Descent	H2 Descent	H2 Descent
	CEV Radiation Protection	No Supplemental Protection	No Supplemental Protection	No Supplemental Protection
CEV CM	+1,330 kg (LOR)	+3,190 kg (LOR)	+3,190 kg (LOR)	+4,100 kg (LOR)
	-1,250 kg (EOR–LOR)	+600 kg (EOR–LOR)	+5,040 kg (EOR–LOR)	+6,730 kg (EOR–LOR)
	N/A (Direct)	N/A (Direct)	+240 kg (Direct)	+1,970 kg (Direct)
CEV SM	+1,600 kg (LOR)	+3,830 kg (LOR)	+3,830 kg (LOR)	+4,910 kg (LOR)
	-1,500 kg (EOR–LOR)	+730 kg (EOR–LOR)	+6,050 kg (EOR–LOR)	+8,060 kg (EOR–LOR)
	N/A (Direct)	N/A (Direct)	+290 kg (Direct)	+2,360 kg (Direct)
Ascent Stage	+330 kg (LOR)	+330 kg (LOR)	+1,740 kg (LOR)	+2,160 kg (LOR)
	-710 kg (EOR–LOR)	+350 kg (EOR–LOR)	+3,230 kg (EOR–LOR)	+4,340 kg (EOR–LOR)
Descent Stage	+1,120 kg (LOR)	+1,120 kg (LOR)	+3,200 kg (LOR)	+4,270 kg (LOR)
	-1,320 kg (EOR–LOR)	+1,140 kg (EOR–LOR) N/A (Direct)	+6,450 kg (EOR–LOR)	+8,060 kg (EOR–LOR)
	N/A (Direct)	N/A (Direct)	+1,300 kg (Direct)	+5,600 kg (Direct)

4.2.4.2.1 Launch EOR–LOR Architecture

The combination of advanced propulsion technology on the LSAM and CEV and additional ascent and injection mass performance with an upgraded CLV introduced another architecture variant in Design Cycle 2. This variant, known as 1.5-launch EOR–LOR, is so named due to the large difference in size and capability of the LVs used in the architecture. Whereas the previous architectures have used one heavy-lift Cargo Launch Vehicle (CaLV) to launch cargo elements and another heavy-lift Crew Launch Vehicle (CLV) to launch the CEV and crew, this architecture divides its launches between one large and one relatively small LV. The 1.5-launch EOR–LOR mission is an EOR–LOR architecture with the LSAM and EDS pre-deployed in a single launch to LEO with the heavy-lift CaLV. A second launch of a 25-mT-class CLV delivers the CEV and crew to orbit, where the two vehicles initially rendezvous and dock. The EDS then performs the TLI burn for the LSAM and CEV and is discarded. On reaching the Moon, the LSAM performs the LOI for the two mated elements, and the entire crew transfers to the LSAM, undocks from the CEV, and performs descent to the surface. The CEV is left unoccupied in LLO. After up to 7 days on the lunar surface, the LSAM returns the crew to lunar orbit, where the LSAM and CEV dock and the crew transfers back to the CEV. The CEV then returns the crew to Earth with a direct- or skip-entry-and-land touchdown while the LSAM is disposed via impact on the lunar surface. The 1.5-launch EOR–LOR architecture is illustrated in **Figure 4-26**.

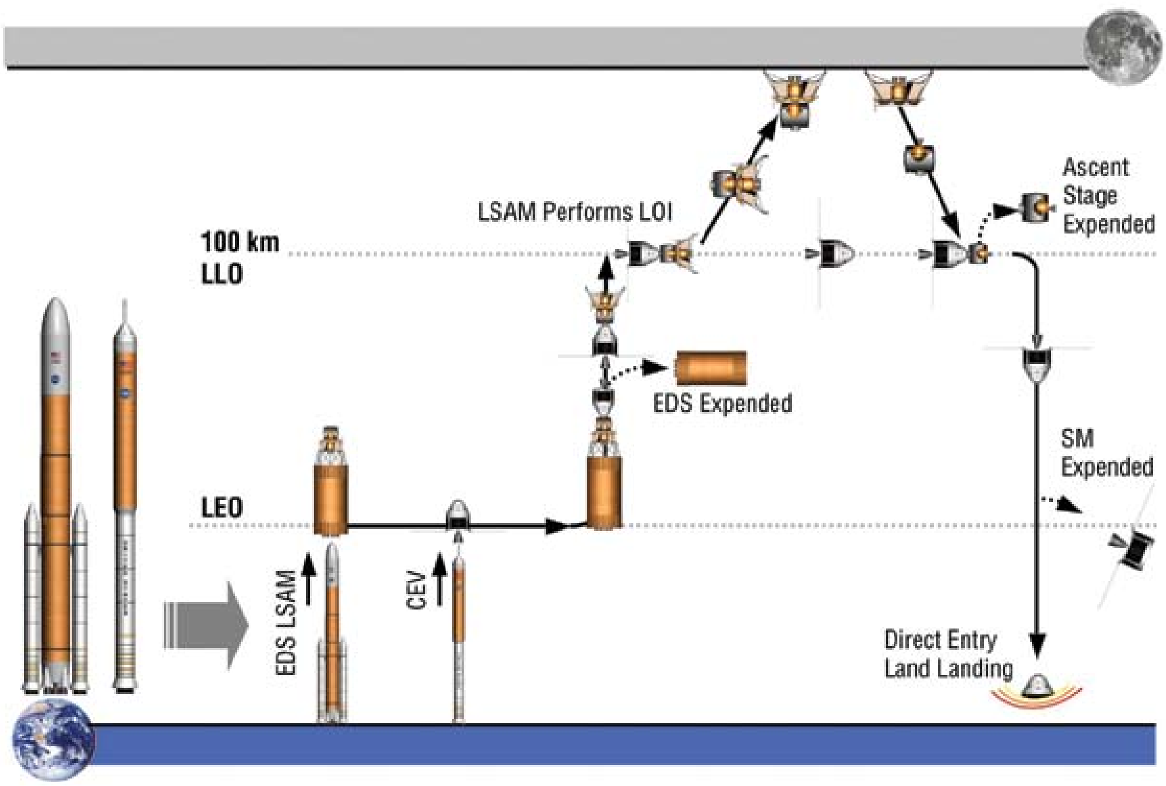


Figure 4-26. 1.5-Launch EOR–LOR Architecture

Similar to EOR-direct return, the 1.5-launch EOR–LOR requires both a hydrogen pump-fed descent stage and the removal of supplemental CEV radiation protection to achieve a 2-launch architecture. Even when these measures are taken, the architecture still has slightly negative EDS performance margin as shown in **Table 4-12**. This table provides a 4x4 matrix of EDS performance margins for the four technology alternatives and four architecture alternatives. The individual launch masses for the LSAM and CEV are also provided.

Table 4-12. EDS Performance Margins

Technology Switch		CH4 Ascent/TEI	CH4 Ascent/TEI	CH4 Ascent/TEI	Pump-Fed CH4 Ascent/TEI
		CH4 Descent	CH4 Descent	H2 Descent	H2 Descent
		CEV Radiation Protection	No Supplemental Protection	No Supplemental Protection	No Supplemental Protection
LOR (EIRA)	**2-launch solution** -5.5-m CEV -5-Segment SRB In-line w/4 SSME	LSAM: 27.9 t CEV: 26.2 t **Required: 27.9 t to TLI+LOI** **EDS Limit: 29.1 t**	LSAM: 27.9 t CEV: 22.2 t **Required: 27.9 t to TLI+LOI** **EDS Limit: 29.1 t**	LSAM: 23.6 t CEV: 22.2 t **Required: 23.6 t to TLI+LOI** **EDS Limit: 29.1 t**	LSAM: 22.6 t CEV: 20.7 t **Required: 22.6 t to TLI+LOI** **EDS Limit: 29.1 t**
EOR–LOR	**2-launch solution** -5.5-m CEV -5-Segment SRB In-line w/ 4 SSME	LSAM: 27.1 t CEV: 64.6 t **Required: 89.9 t to TLI** **Limit: 85.6 t**	LSAM: 27.1 t CEV: 57.9 t **Required: 83.5 t to TLI** **Limit: 85.6 t**	LSAM: 46.1 t CEV: 24.4 t **Required: 69.6 t to TLI** **Limit: 85.6 t**	LSAM: 46.1 t CEV: 24.4 t **Required: 69.6 t to TLI** **Limit: 85.6 t**
	1.5-launch solution -5.5-m CEV -5-Segment SRB In-line w/5 SSME + 2 J2-S -4-Segment RSRB w/1 SSME	**3 Launches Required**		LSAM: 46.9 t CEV: 21.8 t **Required: EDS + 44.9 t to LEO, 65.5 t to TLI** **Limit: 63.8**	LSAM: 42.8 t CEV: 20.3 t **Required: EDS + 42.8 t to LEO, 61.9 t to TLI** **Limit: 65.5 t**
EOR-Direct Return	**2-launch solution** -5.5-m CEV -5-Segment SRB In-line w/4 SSME			LSAM: 50.2 t CEV: 34.9 t **Required: 83.8 t to TLI** **Limit: 85.6 t**	LSAM: 44.1 t CEV: 30.0 t **Required: 73.3 t to TLI** **Limit: 85.6 t**

4.2.4.2.2 Lunar Cargo Transport

Design Cycle 2 also analyzed each architecture's ability to deliver large cargo elements to the lunar surface using the LSAM descent stage and EDS as one-way, uncrewed transportation systems. The LOR and 1.5-launch EOR–LOR architectures are best suited for cargo delivery in a single launch, since the LSAM and EDS are nominally launched together for the crewed mission and one of those two elements performs LOI. The LOR architecture CaLV can land up to 18 t of cargo on the lunar surface in a single launch, while the 1.5-launch EOR–LOR architecture CaLV can land 20.9 t in a single launch. This extra 3 t of capability is due to the larger, higher-performance CaLV. EOR-direct return is instead better suited for delivering cargo in two launches because the LSAM nominally launches separate from the large EDS in the crewed mission. While this architecture does require a second launch to land cargo on the lunar surface, the landed payload capability is nearly doubled to 34.7 t. Finally, EOR–LOR is best suited for either single-launch or 2-launch cargo delivery, depending on the level of propulsion technology assumed. When pressure-fed methane is used in the descent stage, the CEV is used to perform LOI and the descent stage only performs descent from LLO. Therefore, it is more efficient to package the cargo and descent stage with the EDS and only require a single launch for cargo delivery. If pump-fed hydrogen is used instead, the LSAM nominally performs LOI and descent. In this case, the architecture is better suited for 2-launch cargo delivery similar to EOR-direct return. The maximum cargo delivery capability is identical for the two architectures. **Table 4-13** provides the maximum cargo mass for each architecture option.

Table 4-13 . Cycle 2 Architecture Cargo Delivery Capability

<table>
<tr><th colspan="2" rowspan="3">Technology Switch</th><th>CH4 Ascent/TEI</th><th>CH4 Ascent/TEI</th><th>CH4 Ascent/TEI</th><th>Pump-Fed CH4 Ascent/TEI</th></tr>
<tr><th>CH4 Descent</th><th>CH4 Descent</th><th>H2 Descent</th><th>H2 Descent</th></tr>
<tr><th>CEV Radiation Protection</th><th>No Supplemental Protection</th><th>No Supplemental Protection</th><th>No Supplemental Protection</th></tr>
<tr><td>LOR (EIRA)</td><td>2-Launch solution
-5.5-m CEV
-5-Segment SRB In-line w/ 4 SSME</td><td>15.2 t
(Single HLLV Launch)</td><td>15.2 t
(Single HLLV Launch)</td><td>18.0 t
(Single HLLV Launch)</td><td>18.0 t
(Single HLLV Launch)</td></tr>
<tr><td rowspan="2">EOR–LOR</td><td>2-Launch solution
-5.5-m CEV
-5-Segment SRB In-line w/ 4 SSME</td><td>14.4 t
(Single HLLV Launch)</td><td>14.4 t
(Single HLLV Launch)</td><td>34.7 t
(Two HLLV Launches)</td><td>34.7 t
(Two HLLV Launches)</td></tr>
<tr><td>1.5-Launch solution
-5.5-m CEV
-5 Segment SRB In-line w/ 5 SSME + 2 J2-S
-4-Segment RSRB w/1 SSME</td><td colspan="2" rowspan="2">3 Launches Required</td><td>20.9 t
(Single HLLV Launch)</td><td>20.9 t
(Single HLLV Launch)</td></tr>
<tr><td>EOR-Direct Return</td><td>2-Launch solution
-5.5-m CEV
-5-Segment SRB In-line w/ 4 SSME</td><td>34.7 t
(Two HLLV Launches)</td><td>34.7 t
(Two HLLV Launches)</td></tr>
</table>

4.2.4.3 Figures of Merit

The performance analysis contained in the previous section is a good first-order indicator of mission mode viability and was used to examine the three mission modes studied in the second analysis cycle. In order to paint a complete picture of the trade space, however, an examination of additional FOMs is required. Design, Development, Test, and Evaluation (DDT&E), production, and operations costs were calculated for each of the options, as well as integrated LCCs. These detailed results and the methods used are discussed in **Section 9**, **Technology Assessment**. Similarly, safety and reliability calculations were performed for each option in the form of probability of loss of crew (LOC) (P(LOC)) and probability of loss of mission (LOM) (P(LOM)). These detailed results and the methods used are discussed in **Section 8**, **Risk and Reliability**. The ESAS team established a rigorous system to conduct the cost, reliability, and safety studies with common assumptions, performance data, and operational concepts for each of the options. The end results are comparative analyses that accurately capture the relative cost and risk differences among the options.

Figure 4-27 graphs the 20-year LCCs of the various mission options from 2006 to 2025. “Low-tech,” “half-tech,” and “All-tech” on this graph refer to the three propulsion options studied in this analysis cycle—all pressure-fed LOX/LCH4 engines for lander descent and ascent, LOX/LH2 descent with LOX/LCH4 ascent, and LOX/LH2 descent with pump-fed LOX/LCH4 ascent, respectively. The graph suggests that the lander propulsion technology, while having a great effect on performance, has little effect on the program LCC. One anomalous data point that stands out is the “1.5-launch” EOR–LOR mission mode, which differed from the remainder of the options that were costed by using a single Solid Rocket Booster (SRB) for crew launches with a second stage based on a J2–S engine. This result seemed counterintuitive, given that a smaller LV was substituted for a larger booster, and prompted the team to analyze this further in the next analysis cycle.

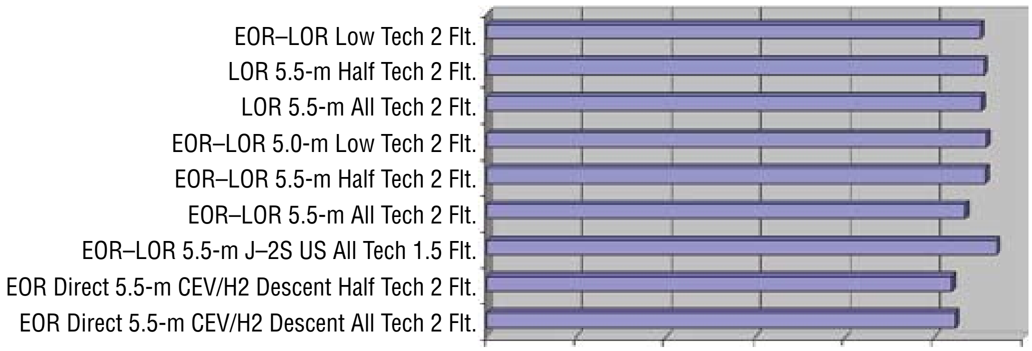


Figure 4-27. Analysis Cycle 2 Mission Mode LCC Comparison through 2025

LOC and LOM analyses provided better separation among the mission mode options. Generally, as more advanced propulsion technology was added, both P(LOM) and P(LOC) increased, as shown in **Figures 4-28** and **4-29**. Comparing mission mode options, EOR-direct return returned the best reliability numbers, likely owing to the fewer number of operations required for this mission mode, though the risk analyses did not take into account any surface operations. EOR–LOR options were shown to be the safest, returning P(LOC) numbers that were on the order of half of LOR missions. The risk metric that captured the ESAS team's attention, however, was the safety and reliability performance of the 1.5-launch EOR–LOR mission relative to any other LOR or EOR–LOR option. For this reason, the team chose to do more in-depth analysis of the 1.5-launch solution.

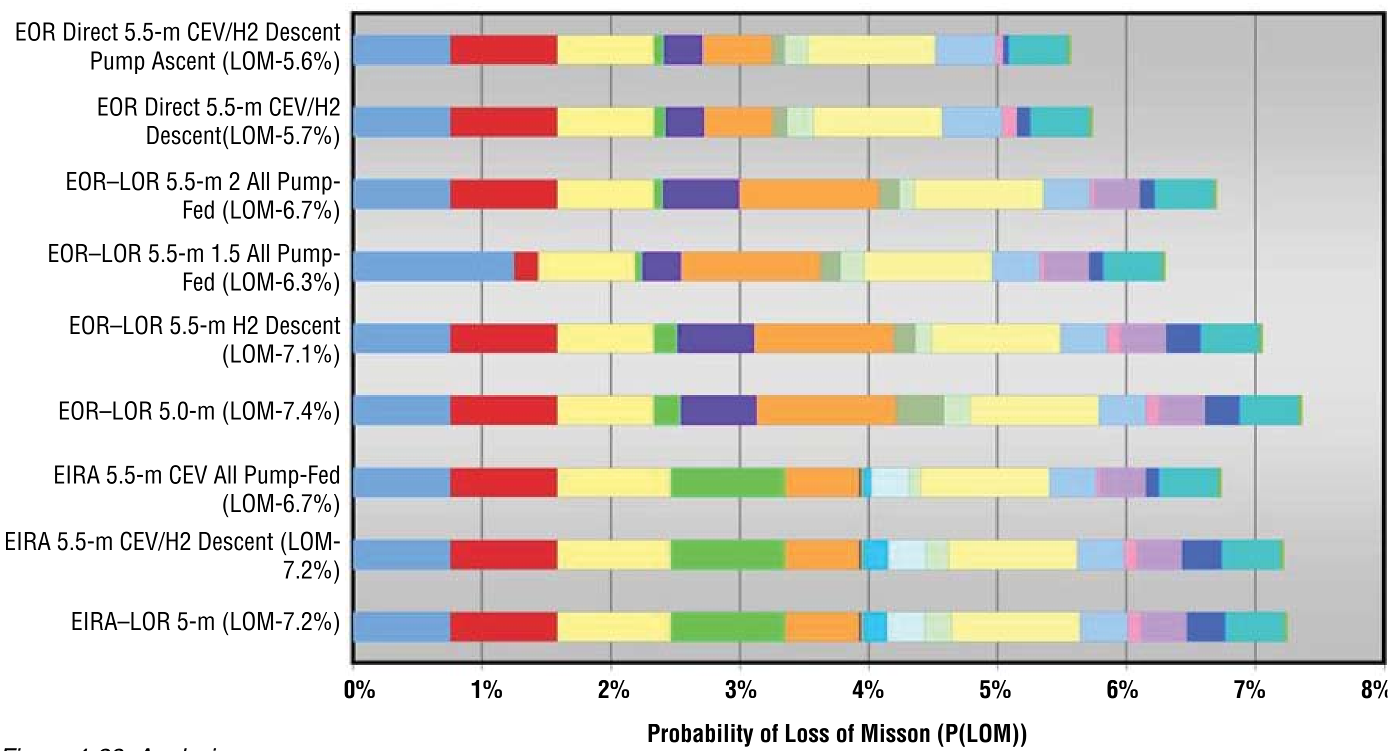


Figure 4-28. Analysis Cycle 2: LOM Comparison

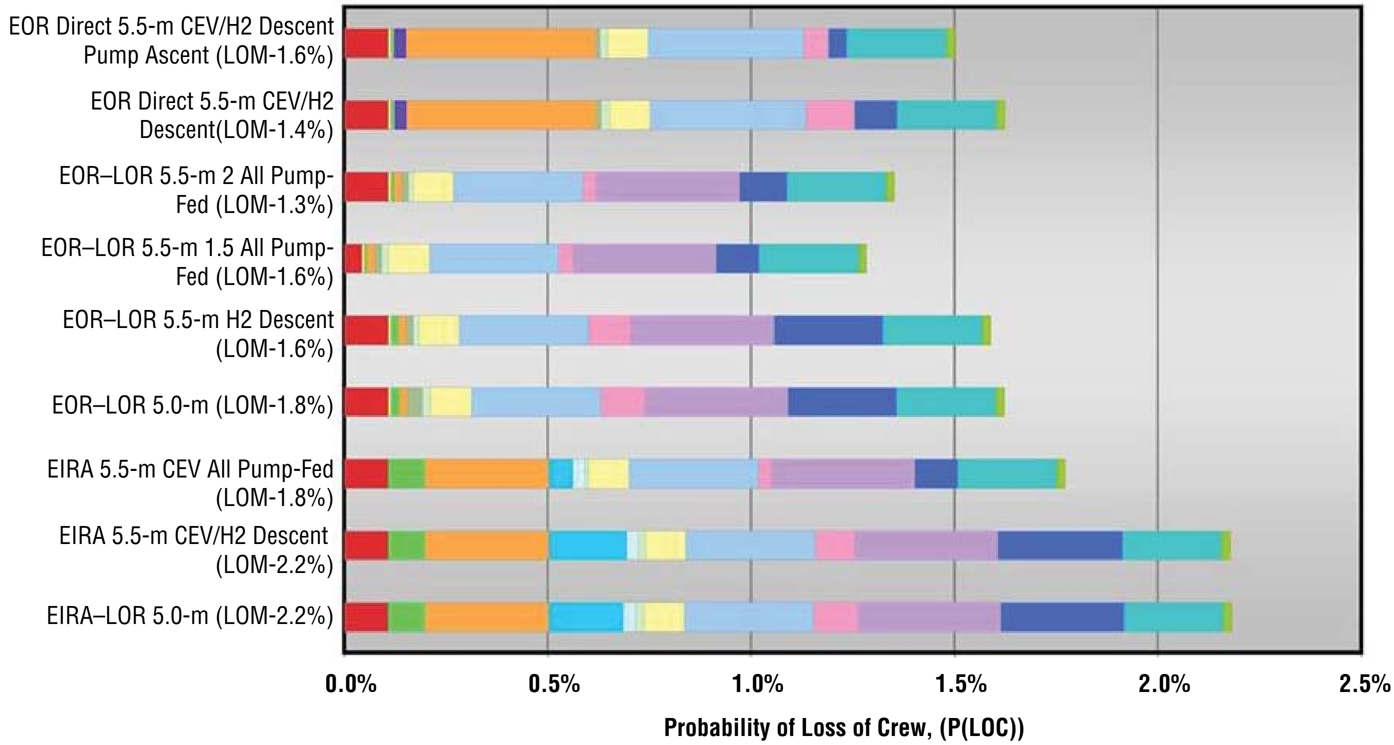


Figure 4-29. Analysis Cycle 2: LOC Comparison

In addition to technical performance, cost, safety, and reliability, the ESAS team also sought to quantify the mission modes with a table of other FOMs, including architecture flexibility, extensibility (to Mars and beyond), effectiveness, and operability. **Tables 4-14** and **4-15** are much-condensed versions of spreadsheets that combined qualitative and quantitative measures for dozens of additional FOMs for the three mission mode options. FOMs that discriminated between the 2-launch and 1.5-launch EOR–LOR options were noted separately in the EOR–LOR row.

Table 4-14. Analysis Cycle 2 Flexibility and Extensibility FOMs

	Pro	Con
LOR	• Mars architecture is also a “split”/orbit rendezvous mission; • Enables single-launch missions to other near-Earth destinations (e.g. Mars Transfer Vehicle (MTV) orbit, LLO, libration points, Geosynchronous Earth Orbit (GEO), etc.); and • More reliable single-launch cargo delivery to lunar surface.	• Largest indivisible surface cargo element limited to <20 t.
EOR–LOR	• Mars architecture is also an orbit rendezvous mission; • The 1.5-launch solution provides more reliable single-launch cargo delivery to lunar surface; and • Provides ability to land large monolithic surface cargo elements.	• Cargo delivery requires two launches and an uncrewed large element AR&D (except 1.5-launch solution); and • 1.5-launch solution has little margin for potential vehicle mass growth with current launch vehicles; and • Requires pump-fed ascent or more expensive, less reliable launchers for additional performance.
EOR-Direct Return	• Larger ascent stage provides greatest leverage for lunar ISRU; and • Provides ability to land large monolithic surface cargo elements.	• Cargo delivery requires two launches and an uncrewed large element AR&D; • EOR-direct return has little margin for potential vehicle mass growth with current LVs; • Requires pump-fed ascent or more expensive, less reliable launchers for additional performance; and • EOR-direct return is highly sensitive to mass growth in the CEV CM and SM.

Table 4-15. Analysis Cycle 2 Effectiveness, Performance, and Operability FOMs

	Pro	Con
LOR	• Lander crew cabin is better configured for supporting sortie missions beyond 7 days; • Relatively simple to incorporate an airlock in LSAM; and • No TLI departure window constraints for outpost missions. (Sortie missions are similar to EOR.)	• Current transportation architecture limited to near-polar or equatorial outpost locations; and • Plane change delta-V costs with “anytime return” capability and global access drastically increase beyond 4–7 days.
EOR–LOR	• Lander crew cabin is better configured for supporting sortie missions beyond 7 days; • Lander volume available on outbound trip for improved habit ability; and • Relatively simple to incorporate an airlock in LSAM → operational flexibility.	• Current transportation architecture limited to near-polar or equatorial outpost locations; • Plane change delta-V costs with “anytime return” capability and global access drastically increase beyond 4–7 days; and • Large, tall descent stage; and • Cargo unloading more challenging.
EOR-Direct Return	• Transportation system delta-V costs independent of landing site location or stay time (global access/anytime return capability).	• Large, tall descent stage; • Cargo unloading more challenging; • Most difficult crew egress path from crew cabin to lunar surface; and • Adding an airlock to the CEV is a very difficult configuration challenge and added mass may exceed available margins.

4.2.4.4 Findings and Forward Work

A tremendous amount of technical, cost, and risk analysis was accomplished by the end of Cycle 2 of the ESAS. Higher-efficiency lander propulsion became of particular interest as performance analysis showed that it enabled both 2-launch direct return and 1.5-launch EOR–LOR solutions. It was recognized that the engine development for the CEV ascent stage should be common to the CEV SM in order to build confidence in a single cryogenic engine that would return crews from the lunar surface. To make this engine available to support a 2011–2012 CEV SM would require a focused engine development program and require the system to be pressure-fed.

The EOR-direct return mission mode was eliminated from further consideration. In the direct return mode, the CEV must operate in, and transition among, 1-g pre-launch and post-landing, hyper-g launch, zero-g orbital and cruise, powered planetary landing and ascent, and 1/6-g lunar surface environments. This added significantly more complexity to a vehicle that must already perform a diverse set of functions in a diverse number of acceleration environments. Additionally, commonality of the SM between lunar and ISS configurations is further reduced in this case. The direct return lunar SM provides lunar ascent and TEI delta-V in excess of 2,400 m/sec, the LOR SM is of the order of 1,850 m/sec, and the ISS mission requires only 330 m/sec. The direct return CEV also requires no docking mechanism since the CEV is the lone crew cabin for the round-trip mission. Conversely, this reduced the commonality from the ISS to the lunar CEV. Ultimately, the ESAS team concluded that the direct return mode entails the greatest number of operability issues and uncertainties, most notably to the configuration of the CEV, and that the complexities of a CEV designed for a surface-direct mission will increase the cost and schedule risks for delivering an ISS-compatible vehicle in the 2011–2012 time frame. The study team eliminated direct return on the basis of CEV complexity, poor margins, greatest number of operability issues and uncertainties, and highest sensitivity to mass growth.

Analysis Cycle 2 began a focus on supplemental radiation protection that was to continue into the final analysis cycle. Radiation shielding was viewed as a risk reduction tool on par with other mission risks, and, based on the statistical modeling presented in this cycle, the decision was made to narrow the range of supplemental radiation protection to 2.0 g.cm^2 or less. The ESAS team also chose to modify the construction of the CEV based on the recommendation of the radiation analysis community. Based on the data presented in this analysis cycle, the skin of the CEV was changed from aluminum to carbon composite due to the superior radiation shielding properties of composites. For follow-on analysis, the ESAS team defined a probable cross-section of the CEV to be used for radiation transport calculations.

Cost, risk, and performance values were calculated for each of the three mission modes, with sensitivities calculated for decreasing supplemental radiation shielding and increasing propulsion efficiency. Direct return missions exhibit the lowest development costs, as well as the lowest P(LOM), but were eliminated from consideration for reasons of operational complexity. All mission modes were within a narrow range for P(LOM) (5.6 to 7.2 percent). The EOR–LOR mission mode showed the lowest P(LOC), and all modes and options were within a low range of 1.3 to 2.2 percent. The EOR–LOR mode has the highest mass margins, with the 2-launch EOR–LOR option having greater margins than the 1.5-launch EOR–LOR option. This 2-launch option closed only when supplemental radiation protection was eliminated and advanced propulsion was enabled for all LSAM stages. The EOR–LOR 1.5-launch option was distinctive in that it required the CEV to be qualified for only one LV (one with an extremely high crew safety) and would retain that CLV for future LEO operations.

Based on the outcome of Analysis Cycle 2, the team selected a focused set of Cycle 3 studies. Of particular interest were studies that could better define the viability of the 1.5-launch EOR–LOR mission mode option. Additionally, the performance of the LOR and EOR–LOR 2-launch solutions would be refined, and updated cost, safety, and reliability FOMs would be generated. Radiation analysis with 0–2 g/cm^2 shielding with an actual CEV cross-section and actual mission durations would be performed to better understand how radiation compared to other hazards to the crew. Global access to any site on the lunar surface and "anytime return" to Earth would be studied as a function of the required LOI and TEI delta-V, respectively, and this propulsive requirement would be incorporated into new propulsion system sizing. Refined CEV configurations and mass estimates would be joined with the updated propulsion sizing, the updated LV sizing, and updated EDS performance calculations to produce a clear picture of spacecraft and LV margins. By the conclusion of Cycle 3, the team was confident that the performance, cost, safety, and reliability analyses would converge so that a lunar mission mode could be recommended.

Finally, Analysis Cycle 3 would continue to investigate lunar lander and lunar surface configurations, with a focus toward investigating alternative LSAM configurations and outpost buildup concepts that could enable sortie-class missions to begin to emplace elements of the permanent outpost infrastructure as part of its cargo. This would challenge the team to analyze split-volume LSAM crew cabins, LSAM cargo packaging, and alternative surface airlock configurations.

4.2.5 Analysis Cycle 3 Mission Mode Analysis

The final cycle of design was focused toward the recommendation of a lunar mission mode. Analysis Cycle 3 continued a number of the Cycle 2 trade studies and undertook additional mission mode analyses emphasizing the EOR–LOR 2-launch, EOR–LOR 1.5-launch, and LOR mission architectures. To better define the performance of the lunar transportation system, the CEV configuration would need to be updated by assessing the sensitivity to supplemental radiation shielding of up to 2 g/cm^2. Additionally, the delta-V requirements for the LSAM descent stage and CEV SM were reevaluated to study the overall performance sensitivities to the requirements of global lunar access and "anytime return." The revised vehicle masses would then be compared to updated LV and EDS designs to assess performance of the different mission modes and evaluate the use of excess performance margins. Each of the updated configurations would then be evaluated to determine differences in cost, P(LOC), P(LOM), and other FOMs to support the final lunar mission mode selection.

During the conduct of the previous analysis cycle, LCCs were calculated for each of the mission mode and technology options. This cost analysis, which is detailed in **Section 12**, **Cost**, was performed assuming an initial combination of sortie missions to a variety of sites and uncrewed cargo missions to construct an outpost, followed by regular crew and cargo missions to maintain a lunar outpost. This particular series of missions was based on an initial balance between sortie missions targeted for short-duration missions at specific sites of scientific interest and the desire for a permanent outpost. Many other implementations of the surface missions are possible, as discussed in **Section 4.3.5**, **Sortie Surface Traffic Model**. This initial approach of sortie-transitioning-to-outpost provided useful data on the relative costs of a sortie-based surface mission architecture versus an architecture that immediately builds up an outpost infrastructure that will support a continued presence of crews. This early cost exercise taught the team that an outpost emplaced in large, monolithic payload blocks could significantly exceed the available budget profile in some years. This was due to a combination of cargo lander development, cargo missions, and the development of the large cargo elements themselves (habitats, nuclear power systems, and science and resource utilization experiments) that would all need to occur simultaneously. Since other options exist to deploy the outpost with and without the use of large cargo landers, studies of alternative buildup concepts were undertaken in this cycle of the study. One such method is to utilize any available excess landed capacity on the crew lander and deploy the outpost in smaller, incremental pieces.

As part of the study of alternative outpost buildup concepts, the alternate configurations of the crew lander were included in the trade space. In addition to excess landed cargo capacity, the cabin of the lander itself offered an opportunity to make use of pressurized volume on the surface that would no longer be used by the crew after departure. This study of alternate lander configurations would begin with the requirements for pressurized crew volume for the 7-day surface stay and the operational need for an airlock, and then investigate how these capabilities could be divided into two areas: (1) those capabilities required for the crew to return to the CEV in lunar orbit and (2) those that could remain on the surface and contribute to the surface infrastructure or be scavenged for resources.

4.2.5.1 Trade Studies

Three major trade studies were undertaken during Analysis Cycle 3 to narrow the mass uncertainty of the CEV and explore alternatives for the deployment of lunar surface outposts. The radiation analysis study begun in Analysis Cycle 2 was updated using a more detailed CEV cross-section and improved analysis tools. Global access and anytime return were studied in detail to understand the sensitivities of delta-V and loiter time. The configuration of the lunar lander was revised to explore packaging of habitable volumes and options for lunar outpost deployment.

4.2.5.1.1 Radiation Study

The Analysis Cycle 3 radiation study focused more heavily on creating the best approximation to the actual CEV in order to generate a more refined and realistic radiation evaluation. The radiation protection and resultant dose/risk associated with the analysis of an analytical CEV model depends highly on how accurately that model reflects the actual design. This higher-fidelity model included the addition of a TPS, composite Outer Mold Line (OML) skin, and insulation to the structure of the vehicle (in addition to the original HDPE radiation shield and aluminum hull). The inclusion of these structures had a significant impact on the amount of radiation shielding the vehicle inherently provided. For comparison purposes, the shield distribution was generated in the same fashion and using the same points as the Analysis Cycle 2 evaluation. Calculations for the historical large SPEs were repeated with the refined CEV configuration and the values of thin HDPE (1 or 2 g/cm^2) augmentations. Results are shown in **Table 4-16**.

Table 4-16. CEV Radiation Shielding Acute and Late Risks for Largest Fluence SPE

Aluminum Vehicle, 4x 1972 SPE			
HDPE Depth (g/cm^2)	% Acute Death*	% Sickness	% REID**
CEV-old + 0 g/cm^2	9.5	54	9.1 [3.2,17.3]
CEV-new + 0 g/cm^2	<1% (***)	<5% (***)	4.4 [1.5, 11.8]
CEV-new + 1 g/cm^2	0	0	3.5 [1.2, 9.7]
CEV-new + 2 g/cm^2	0	0	2.9 [1.0, 8.2]

**Death at 60-days with minimal medical treatment*

***Risk of Cancer death for 45-yr-old females*

****Too close to threshold to estimate*

The probability of acute risk is difficult to estimate accurately for the baseline revised CEV configuration because of lack of radiobiological data at the 0–10 percent probability levels and the potential impacts of immune depression and stress on the dose-response. The addition of HDPE would likely prevent the occurrence of acute risks from a historically large SPE. A statistical analysis of the uncertainties in the acute projections will be needed to properly perform the analysis.

As shown in **Table 4-17**, fatal cancer risk limits or the 95 percent confidence limit requirements would be exceeded for most astronauts with no prior occupational exposure below age 45-yr for the revised CEV with 0 or 1 g/cm^2 polyethylene augmentation shielding. For astronauts with prior ISS exposure, larger constraints will occur. With the 2 g/cm^2 HDPE augmentations, 95 percent confidence limits would be exceeded for a significant fraction of the astronaut population. Higher constraints are possible if fatal non-cancer risks are added to the NASA legal dose limits.

Risk of Exposure Induced Death for 45-yr Females					
Nx1972	Probability (worst-case SPE)	CEV-old with 0 g/cm^2 HDPE	CEV-new with 0 g/cm^2 HDPE	CEV-new with 1 g/cm^2 HDPE	CEV-new with 2 g/cm^2 HDPE
4x	99.1	9.1 [3.2, 17.3}	4.4 [1.5, 11.8]	3.5 [1.2, 9.7]	2.9 [1.0, 8.2]
3x	98.5	6.9 [2.4, 16.0]	3.3 [1.1, 9.2]	2.6 [0.9, 7.4]	2.2 [0.7, 6.2]
2x	97.0	4.7 [1.6, 12.5]	2.2 [0.8, 6.3]	1.7 [0.6, 5.0]	1.5 [0.5, 4.2]
1x	93.0	2.4 [0.8, 6.7]	1.1 [0.3, 3.2]	0.9 [0.3, 2.5]	0.7 [0.2, 2.1]

Table 4-17. Risk of Fatal Cancer for Large SPEs

Risk Leveling

As introduced in **Section 4.2.4**, **Design Cycle 2 Mission Mode Analysis**, the ESAS team adopted a policy of "risk leveling" in order to protect astronaut crews equally from all known sources of injury or death. When applied to radiation dose and effects, the team viewed this risk as having both an acute, short-term effect that could result in LOM and LOC and a long-term effect of excess cancer risk due to exceeding monthly or career dose limits. Acute sickness was conservatively judged to incapacitate the crew to the extent that they could not perform any of their functions, which would lead to LOM and LOC due to their inability to act. The team sought to arrive at a solution that produced near-zero percent probability of acute death or sickness and that did not violate 30-day or career limits for an event with a probability of occurrence equal to that of other LOC risks for a sortie-duration lunar mission. For longer-duration lunar outpost missions, this analysis would be repeated to determine the proper amount of surface habitat shielding required to achieve this same level of protection.

In order to establish the probability of an SPE occurrence that would exceed a fluence of 30 MeV, a 9-day mission duration was chosen as the average length of time a crew would inhabit the CEV during a sortie-class lunar mission. For longer mission durations, these numbers would increase. **Table 4-18** relates the probability of occurrence of a 30 MeV SPE to the biological effects (acute effects and long-term dose) for 0, 1 and 2 g/cm^2 of supplemental HDPE shielding for a 9-day CEV mission. At 2 g/cm^2, all acute effects are zero and long-term doses are within limits until events with a probability of occurrence of 1 in 2,500 (0.04 percent) missions are encountered. With 1 g/cm^2 of shielding, acute effects are again all zero, but 30-day limits are violated once in every 1,428 (0.07 percent) missions. With all supplemental shielding removed, acute health effects begin to appear once in every 1,428 (0.07 percent) missions, while 30-day limits are violated once in every 588 (0.17 percent) missions.

Table 4-18. SPE Risks to Crew (Acute and Long-term Dose) as a Function of Supplemental Shielding for a 9-Day CEV Mission

CEV with 0 g/cm² HDPE						
Nx1972 Event	**F(>30 MeV)**	**% Probability for 9-Day mission**	**Acute Death**	**Acute Sickness**	**Career Limit Violation**	**30-Day Limit Violation**
4x	$2x10^{10}$	0.02	<1%	<5%	Yes	Yes
3x	$1.5x10^{10}$	0.04	0	<1%	Yes	Yes
2x	$1x10^{10}$	0.07	0	0	No (95% Yes)	Yes
1x	$5x10^{9}$	0.17	0	0	No (95% Yes)	No
1% Event	$1.5x10^{9}$	1.00	0	0	No (95% Yes)	No
CEV with 1 g/cm² HDPE						
Nx1972 Event	**F(>30 MeV)**	**% Probability for 9-Day mission**	**Acute Death**	**Acute Sickness**	**Career Limit Violation**	**30-Day Limit Violation**
4x	$2x10^{10}$	0.02	0	0	Yes	Yes
3x	$1.5x10^{10}$	0.04	0	0	No (95% Yes)	Yes
2x	$1x10^{10}$	0.07	0	0	No (95% Yes)	Yes
1x	$5x10^{9}$	0.17	0	0	No	No
1% Event	$1.5x10^{9}$	1.00	0	0	No	No
CEV with 2 g/cm² HDPE						
Nx1972 Event	**F(>30 MeV)**	**% Probability for 9-Day mission**	**Acute Death**	**Acute Sickness**	**Career Limit Violation**	**30-Day Limit Violation**
4x	$2x10^{10}$	0.02	<1%	<5%	Yes	Yes
3x	$1.5x10^{10}$	0.04	0	<1%	Yes	Yes
2x	$1x10^{10}$	0.07	0	0	No (95% Yes)	Yes
1x	$5x10^{9}$	0.17	0	0	No	No
1% Event	$1.5x10^{9}$	1.00	0	0	No	No

In the cases of acute death or sickness, radiation exposure has an effect equal to any other risk which results in LOC. For long-term dose violations, the effect may be an increased probability of lifetime cancer risk to the crew members, but, for the purpose of this analysis, it was conservatively considered to be an LOC risk as well. The complete lunar sortie mission risk analysis is presented in **Section 8**, **Risk and Reliability**, of this report. The analysis details many of the events that could result in LOC, many of which are large-energy change events such as launch, planetary injection or insertion maneuvers, or planetary landings. Other events are lifetime issues associated with vehicle systems. As a group, the individual risks that result in LOC occur in the 1:100 to 1:1,000 range (1.0 to 0.1 percent individual probability of occurrence). To level the probability of radiation risk, a solution was sought that placed the P(LOC) due to radiation within this range (and preferably nearer to 0.1 percent). The ESAS team used **Table 4-18** for the statistical probabilities generated by the radiation community. Per the table, the 1.0 percent probable event has no adverse biological effects for any level of shielding. Similarly, the 0.17 percent SPE has no acute or lifetime biological effect for any level of shielding, including a CEV with no supplemental shielding. It was not until the mission encountered the 0.07 percent probable SPE that the first of the Next Hop Resolution Protocol (NHRP) limits were exceeded.

The ESAS team therefore recommended that, for the modeled cross-section and material choices, no supplemental radiation protection was required for the CEV. With the inherent shielding properties of the CEV structure alone, all radiation effects, less one, show a lower probability of occurrence than equivalent LOC risks; additionally, the one with the greatest probability of occurrence falls within the low end of the range of equivalent LOC events. For the CEV without supplemental shielding, acute effects would occur less than once in every 1,428 missions (<0.07 percent), career dose limits would be exceeded less than once in every 1,428 missions (<0.07 percent), and 30-day dose limits would be exceeded less than once in every 588 missions (<0.17 percent).

Supplemental radiation shielding ultimately has an effect on the performance of the entire transportation system. Any mass associated with the CEV must travel round-trip from Earth to lunar orbit and back. Thus, the performance sensitivity is second only to mass that travels round-trip to the lunar surface. The performance effect of supplemental radiation shielding is shown in **Figure 4-30**. With a performance impact of almost 500 kg for every g/cm^2 of shielding added, the CEV design should seek to minimize supplemental radiation shielding. Additional configuration studies should continue to be performed to further reduce the dose to crews by optimizing the arrangement of crew, fuel, and stowage.

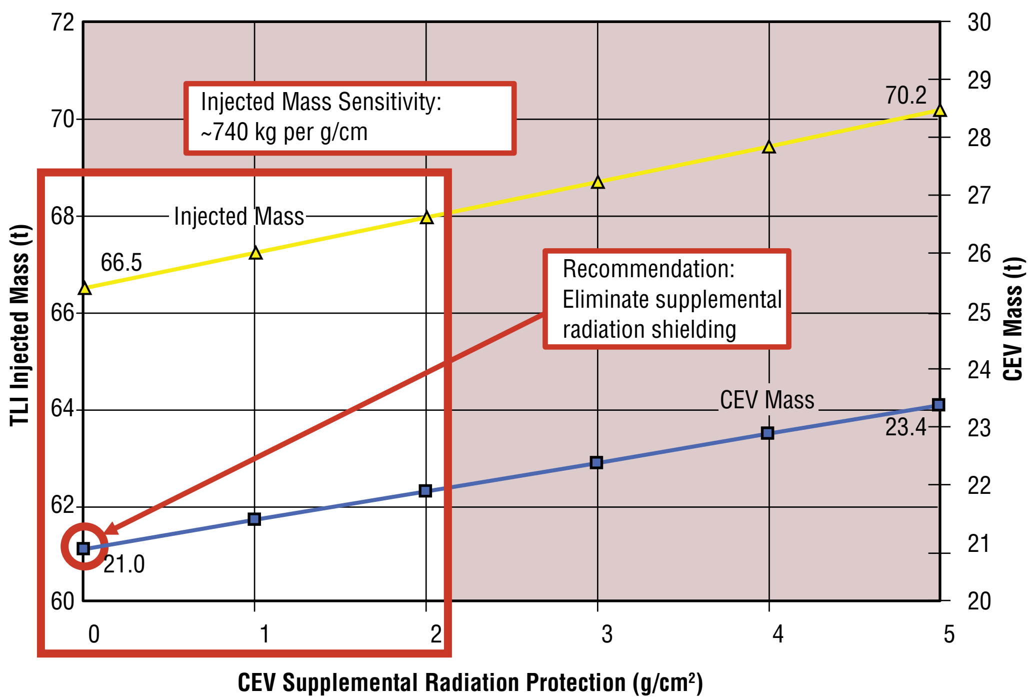


Figure 4-30. 1.5-launch EOR–LOR Sensitivity to CEV Radiation Protection

4.2.5.1.2 Global Access/"Anytime Return"

Another of the Analysis Cycle 3 trade studies was conducted to understand the architecture mass cost of global lunar surface access and "anytime return." Global access was initially interpreted by the ESAS team as the ability of the architecture to conduct a 7-day sortie mission to any location on the lunar surface without requiring any dedicated loiter time in LLO prior to descent and being independent of Earth-Moon system geometry (e.g., for any inclination/declination of the Moon). "Anytime return" was interpreted as giving the crew the ability to return from the lunar surface to Earth independent of orbital plane alignment and within 5 days of an emergency return declaration. These conditions were satisfied in previous architecture design cycles by including sufficient delta-V for LOI and TEI to perform a worst case 90-deg plane change around the Moon.

The global access trade examined the LOI delta-V cost for 7-day sortie missions to various locations on the lunar surface. A list of 10 high-priority sites selected for their perceived scientific and resource utilization value were studied first, followed by a global delta-V map calculated in 10-deg increments of latitude and longitude. The same global map was then calculated assuming the crew could loiter in lunar orbit up to 3 days prior to descent to minimize the LOI plane change cost. The 10 sites are shown in **Figure 4-31**, and each is described in more detail in **Section 4.3**, **Lunar Surface Activities**.

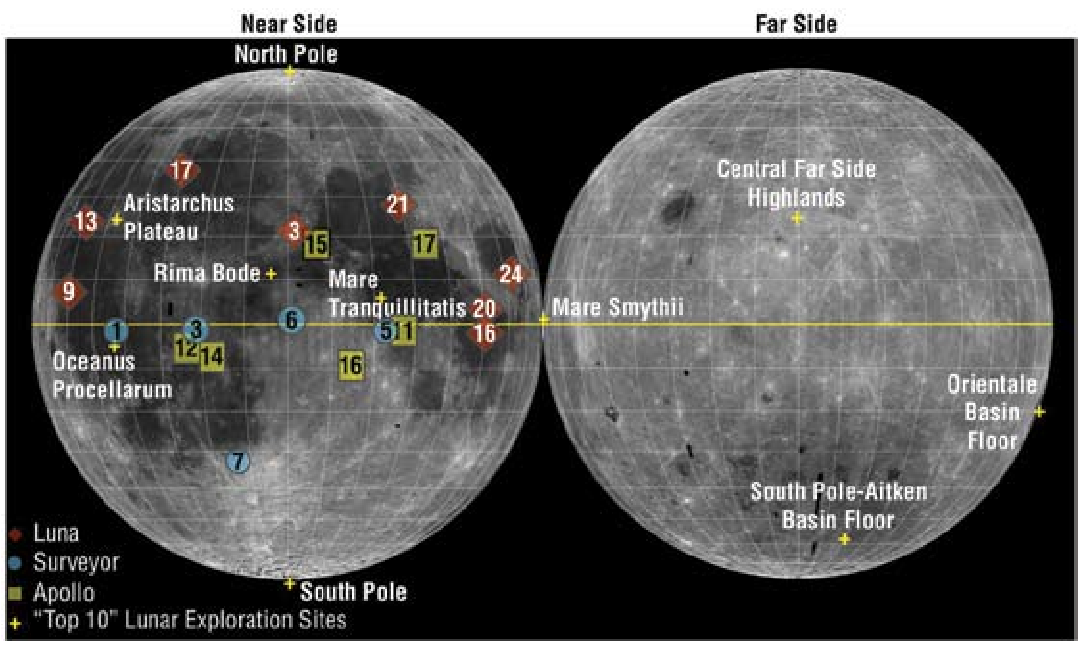


Figure 4-31. Lunar Exploration Sites Chosen as Scientific and Operational Bounding Cases

The following assumptions were used for calculating global access delta-V costs:

- Only outbound portions of the mission were examined.
- The LOI maneuver is a three-impulse sequence to minimize plane change costs.
- The nominal transfer time from TLI to LOI (first maneuver of three-impulse sequence) is 4 days.
- The nominal time from the third maneuver of a three-impulse sequence to descent is 1 day for crew transfer and checkout.
- Earth departure conditions are (1) a LEO parking orbit altitude of 407 km circular, (2) an inclination of 28.7 deg, and (3) the LEO parking orbit ascending node is free and can be selected to minimize delta-V cost.
- A TLI delta-V limit of 3,150 m/s is imposed.
- The arrival epoch is 12/25/2034 to produce a worst-case lunar geometry at arrival. The Moon is at perigee, at its minimum inclination in the 18.6-year metonic cycle (18.3 deg), and at maximum declination (18.3 deg).
- The ascending node and inclination of the lunar parking orbit are selected such that the maximum plane change is minimized for anytime ascent over the 7-day mission.

LOI delta-Vs for 7-day sorties to the top 10 science/resource utilization sites are listed below in **Table 4-19**. The maximum delta-V is 1,078 m/s for a mission to the far side South Pole-Aitken Basin floor. Vehicle sizing for LOI in all design cycles has included 1,390 m/s to protect for a worst-case 90-deg plane change at arrival.

Table 4-19. LOI Delta-V for Top 10 Lunar Sites

LOI Delta-V (m/s)			
Landing Site	**Latitude**	**Longitude**	**Delta-V**
South Pole	89.9 S	180 W	835
Far side SPA floor	54 S	162 W	1,078
Orientale basin floor	19 S	88 W	944
Oceanus Procellarum	3 S	43 W	841
Mare Smythii	2.5 N	86.5 E	826
W/NW Tranquilitatis	8 N	21 E	852
Rima Bode	13 N	3.9 W	851
Aristarchus plateau	26 N	49 W	881
Central far side highlands	26 N	178 E	925
North Pole	89.5 N	91 E	835

Next, a global LOI map (as seen in **Figure 4-32**) was generated for 10-deg increments of landing site latitude and longitude, assuming no additional loiter time in LLO. The global minimum delta-V is 835 m/s for polar or near-equatorial sites, while local delta-V maxima are found near 75°N or S latitudes and 25°E/160°W longitudes. The global maximum delta-V is 1,313 m/s. However, a single mission design technique, described in the assumptions above, was uniformly applied to all possible landing sites. Thus, it may be possible to reduce the maximum LOI delta-V through mission design optimization.

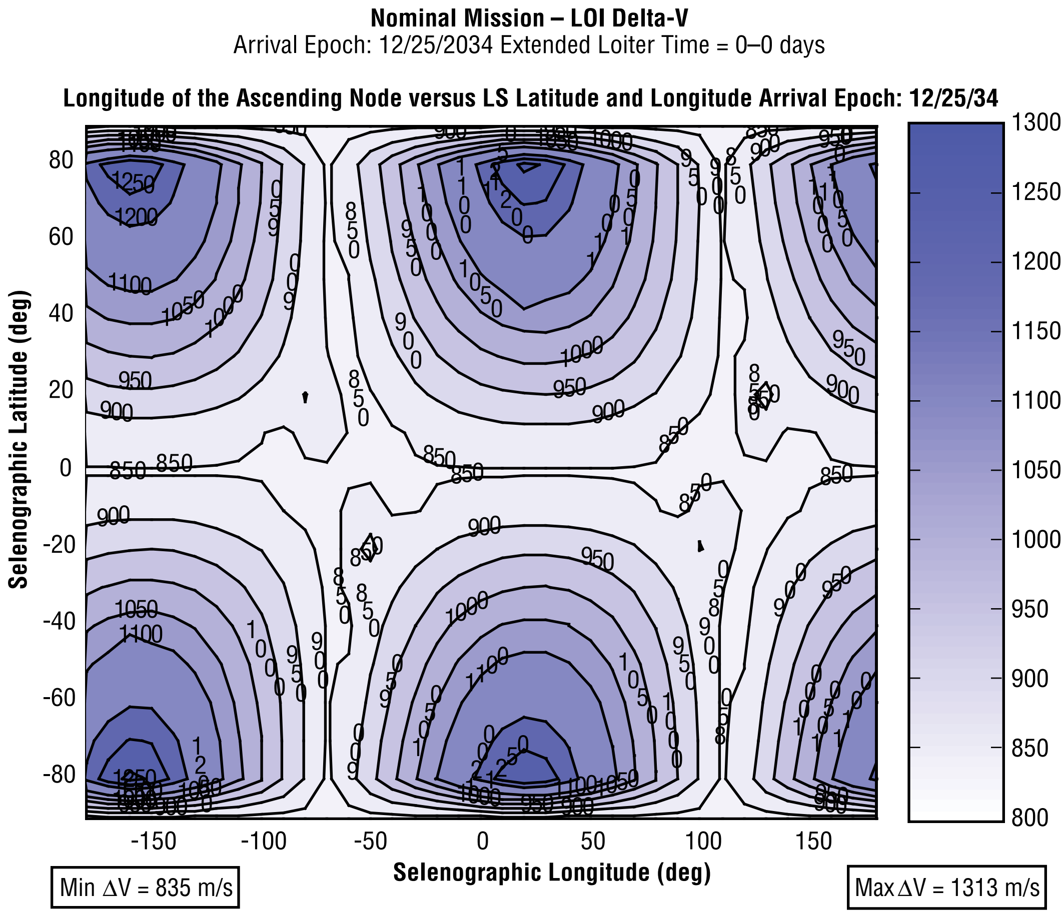


Figure 4-32. Global LOI Delta-V Map, No Loiter Time

Finally, the same global LOI contour map was generated allowing the crew to loiter up to 3 days in LLO to reduce the plane change delta-V cost. The lunar parking orbit ascending node is selected such that, up to 3 days after arrival, the landing site passes underneath the parking orbit plane. This mission design flexibility allows the ascending node change during LOI to be minimized. The net reduction in maximum LOI delta-V is 212 m/s, from 1,313 m/s with no loiter time to 1,101 m/s with up to 3 days loiter time. The LOI map with 3 days of loiter time is provided in **Figure 4-33**.

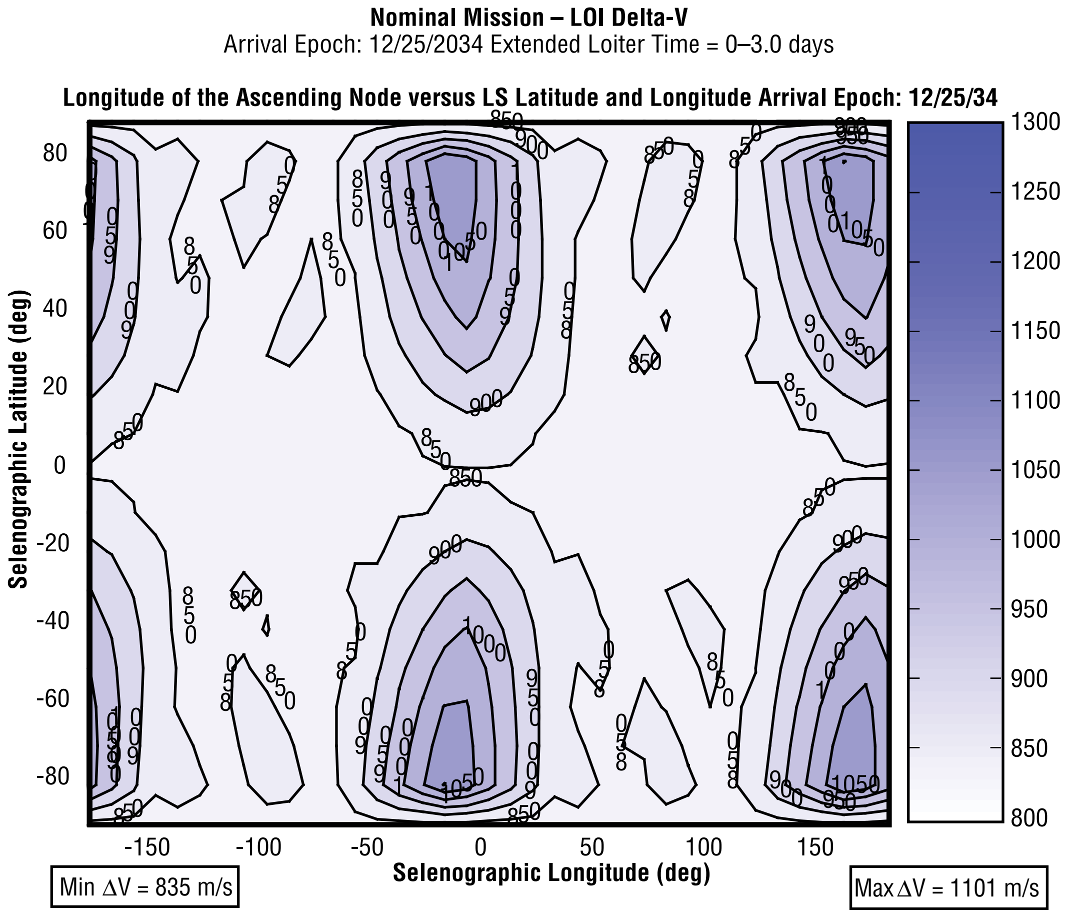


Figure 4-33. Global LOI Delta-V Map, 3-Day Loiter Time

The second part of the global access/"anytime return" trade examined the mass cost of including the capability for the crew to return from lunar orbit to Earth, independent of orbital plane alignment. This capability was manifested as a 90-deg plane change included as part of the TEI maneuvers. To save architecture mass, the CEV and crew could instead loiter in LLO until a more favorable departure opportunity arose. In a worst-case scenario, assuming the first coplanar TEI opportunity was just missed, the crew would have to loiter up to 14 days until the next coplanar opportunity became available. **Figure 4-34** illustrates how CEV total mass, propellant mass, and crew provisions mass each varies as a function of loiter time in LLO (and therefore maximum plane change capability). The CEV "anytime return" capability assumed for architecture sizing purposes is shown on the left side of the figure. Since the crew does not have to loiter at all to return to Earth, the vehicle includes up to a 90-deg plane change for TEI. The maximum CEV injected mass in that case is 22.0 tons (t), while propellant and crew/crew provisions account for 11.8 t and 1.2 t, respectively. The right side of the figure shows the scenario where the CEV is allowed to loiter in lunar orbit until a coplanar TEI opportunity arises, which may last up to 14 days, but eliminates the plane change requirement. Loitering reduces the CEV injected mass to 19.4 t and the propellant mass to 6.3 t, while the crew/crew provisions mass increases to 1.6 t. Because the propellant mass decreases with loiter time faster than crew provisions mass increases, the CEV sees an overall net mass reduction. Injected mass decreases 185 kg per day of loiter in lunar orbit. **Section 4.2.5.2**, **Analysis Cycle 3 Performance**, of this report will further discuss the interaction of TEI and LOI delta-V as it relates to overall mission performance.

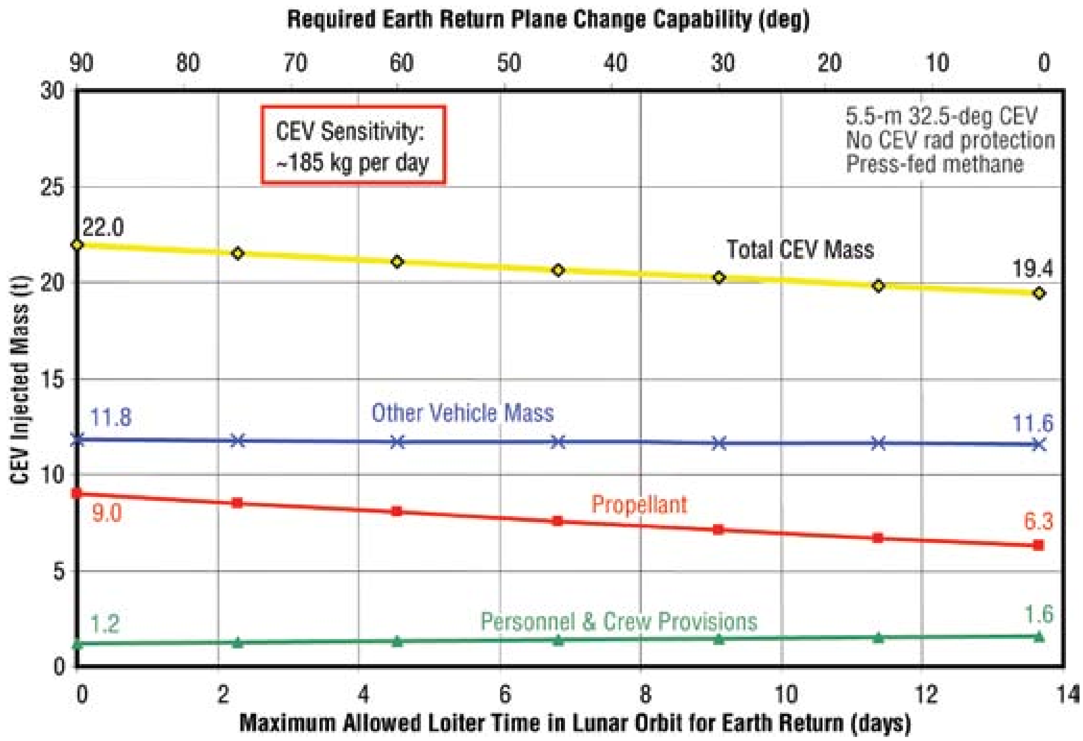


Figure 4-34. CEV Anytime Return Performance

The final part of the trade involved assessing the "cost" of implementing global access and "anytime return." For LOR missions, the propulsive (or mass) "cost" is affected by adding propulsive maneuvers and fixing surface sorties at 7 days. For a fixed TLI of 3,150 m/sec, a maximum three-impulse LOI of 1,390 m/sec is required to achieve global access, which includes a worst-case nodal plane change of up to 90 deg. Departure is from a 407-km circular LEO parking orbit at 28.7 deg inclination. Earth departure Right Ascension of Ascending Node (RAAN) is free and can be adjusted to minimize delta-V cost. The LOI maneuver would set up a "node walking" orbit that would position the CEV for a coplanar LSAM ascent in 7 days. Lunar orbit arrival inclination and Longitude of Ascending Node (LAN) are specified for each landing site latitude and longitude and are based on the minimum departure wedge angle requirement. One day of loiter is assumed in LLO before LSAM landing for LSAM checkout. If the LSAM were to depart the surface early, the CEV would also be given the capability to perform up to a 5-deg nodal and inclination plane change to set up for LSAM rendezvous—a maneuver of 150 m/sec. Finally, anticipating that the CEV may be up to 90 deg out of plane for the return trip to Earth, a 550 m/sec, three-burn nodal plane change was included in addition to the nominal TEI delta-V. The combination of these maneuvers provides all of the required global access and "anytime return" capability, but at the expense of a great deal of excess propulsive capability and additional mass.

The magnitude of these propulsive maneuvers can be decreased through any combination of adjusting the total time spent on the lunar surface, adjusting time spent in post-LOI loiter (prior to lunar descent), and adjusting the time spent in past-ascent loiter prior to TEI. The 7-day surface mission was held constant to preserve surface mission content, and the full nodal plane change at TEI was retained to preserve "anytime return."

To implement "global access," the LOI delta-V was traded against post-LOI loiter time. When loiter time is added to the trade space, the "cost" of global access becomes a function of both LOI delta-V and risk. LOI delta-V as a function of lunar landing site latitude and longitude is shown in **Figure 4-32** and similar delta-V maps were run for pre-descent loiter durations of 0.5 to 7.0 days, in 0.5-day increments. Generally, the maximum LOI delta-V decreases as orbital loiter time increases from zero to 7 days. With a 7-day loiter, LOI reaches a coplanar minimum of 868 m/sec.

Just as the total system mass decreases with additional LLO loiter time, risk increases as a function of increased mission duration. P(LOM) and P(LOC) both increase with loiter time. Due mainly to extended operational timelines of the CEV in lunar orbit, P(LOM) increases at 0.205 percent per day and P(LOC) increases at 0.135 percent per day. **Figure 4-35** illustrates the change of the P(LOC) and P(LOM) for CEV and LSAM vehicle systems for extending orbital loiter prior to lunar descent. LV and propulsion risk probabilities are not included in this figure because they remain constant with respect to loiter time.

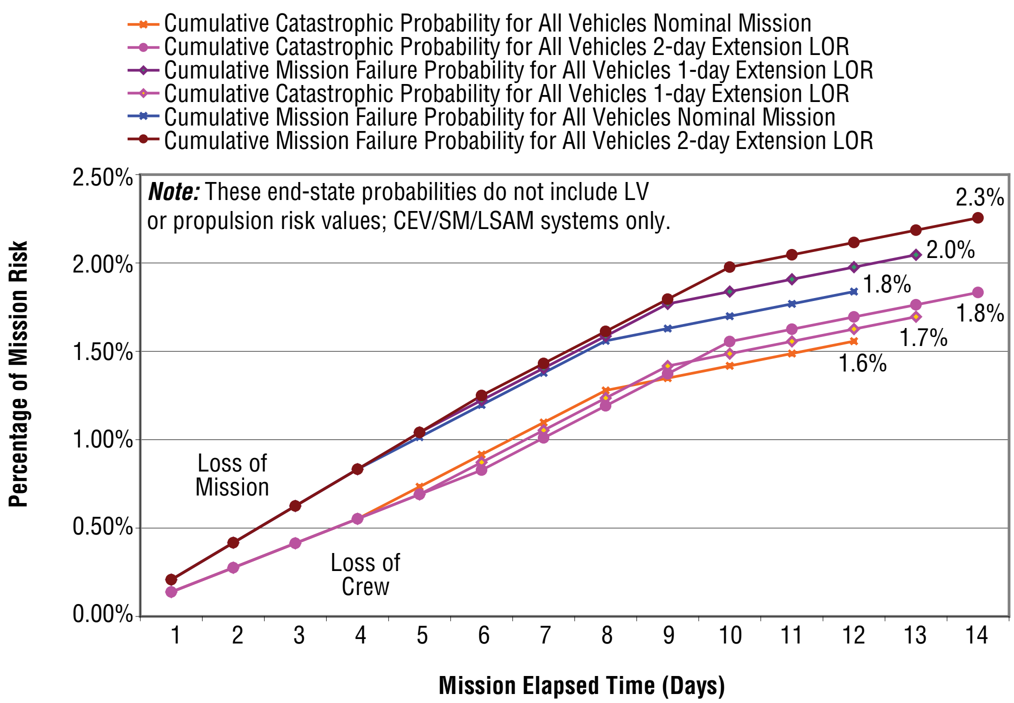


Figure 4-35. Delta P(LOC) and P(LOM) for Extending Orbital Loiter Prior to Lunar Descent

The ESAS team ultimately chose to increase the IMLEO and lander mass margin and to decrease descent stage sizing by requiring the system to accommodate up to a 3-day pre-descent loiter. With this maximum loiter duration, global access can be attained for an LOI delta-V maximum of 1,101 m/sec, as shown previously in **Figure 4-33**. The system would have the capability for immediate access to 84 percent of the lunar surface without additional orbital loiter and could access the remaining surface area by adding no more than 3 days post-LOI orbital loiter. For a maximum 3-day additional loiter, the P(LOC) will incrementally increase 0.4 percent and the P(LOM) will increase 0.6 percent.

4.2.5.1.3 LSAM Configuration Trades

The LSAM configuration trade study was initiated during Analysis Cycle 3 in order to develop a better understanding of the overall system performance and capabilities necessary for the lunar landing portion of the mission. The intent of this study was to develop a higher-fidelity conceptual lander design with special emphasis placed on the numerous competing needs driving the overall vehicle configuration. Focus on LOR mission modes allowed for further optimization of the LSAM configuration, thus providing a better understanding of the relative benefits of differing lunar vicinity segment mission approaches, as well as to determine mission sensitivities to key system design parameters.

The LSAM vehicle provides the key functional capabilities necessary for both the lunar sortie and lunar outpost mission phases. The LSAM must function in a wide range of mission modes, which makes optimization and vehicle configuration development extremely challenging. For instance, surface durations range from 7 active/0 dormant days in the sortie mission phase to 4 active/180 dormant days in the lunar outpost phase. Likewise, in order to reduce the overall development cost and schedule, it is desired to have a common descent stage design for both crew and uncrewed cargo mission modes and commonality of propulsion systems between the LSAM and CEV SM.

Key Functional Requirements and GR&As

The following key functional requirements and GR&As were used for the study, with emphasis placed on ensuring that the architecture approach was consistent with the Cycle 3 ESAS architecture and mission assumptions.

- Mission Mode: Utilize LOR mission approach. In that context, the LSAM may be utilized to perform the LOI maneuver, deorbit, powered descent, hazard avoidance, terminal landing, ascent, and rendezvous. The CEV remains in LLO, which is assumed to be 100 km circular; inclination is landing site dependent.
- Airlocks: All crew landers have airlocks to enable routine exploration of the lunar surface.
- Hardware Reuse: Emphasize leaving hardware behind that can be used for outpost buildup.
- Crew Size: Assume four crew, with all crew traveling to the surface for the entire surface mission duration.
- Surface Duration: Assume up to 7-day sortie missions and up to 180 days during outpost missions.
- LV Shroud: Shroud sizes range from 7.5–10 m. Focus on 8.5 m.
- Commonality: Emphasize a common crew/cargo descent stage concept.
- ISRU: Capable of utilizing locally produced propellants.
- Descent Propellant Type: Utilize hydrogen and oxygen as the propellants for the descent phase of the lunar mission. Hydrogen and oxygen were selected as a technology implementation necessary to reduce the overall mission mass to within reasonable limits.

Review of Previous Lander Design Work

The LSAM design study was initiated with a quick review of the various lunar architecture and mission studies conducted over the past several years, with special emphasis specifically placed on the lander designs. This survey provided a range of vehicle concepts and resulting mission performance as driven by the architecture approach and associated mission requirements. The applicability of each of these studies was considered in terms of the driving mission requirements utilized during that particular study as they applied to the ESAS activity. A review of these previous studies showed that the 2005 ESMD LSAM Phase I study had a high degree of applicability to the current ESAS and was thus utilized as the primary starting point for the LSAM configuration analysis.

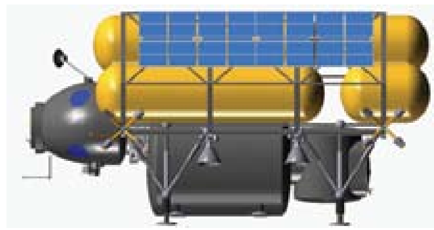
During the early spring of 2005, the ESMD initiated a study of the LSAM as it applied to the current ESMD POD Architecture. The purpose of this task was to conduct a wide range of configuration trades in order to understand the architectural sensitivities and constraints (e.g., launch mass and volume, crew and cargo delivery, crew and cargo unloading, etc.). Results from this study were used to initiate formulation of architecture and element requirements. The study focused on investigating a wide range of vehicle concepts in order to understand the key driving characteristics as they apply to the exploration architecture. Vehicle concepts were studied at a high level in order to drive out the key discriminators to allow further downselection to a limited number of vehicle concepts for further detailed sensitivity and trade study.

LSAM Configuration Considerations

Several different vehicle configurations were investigated in order to span the breadth of configuration options. Combinations of staging approaches, vehicle CM division, and options for leaving vehicle components on the lunar surface for future use were included in the study. The five key configurations utilized for downselection are provided in **Figure 4-36**. Both vertical and horizontal vehicle configurations were considered, as well as both single and dual CMs.

Single-Stage
Single CM
Vertical Design

Single-Stage
Dual Module
Horizontal Design



Two-Stage
Single Module
Vertical Design

Two-Stage
Dual Module – Integrated
Vertical Design

Two-Stage
Dual Module – Integrated
Horizontal Design

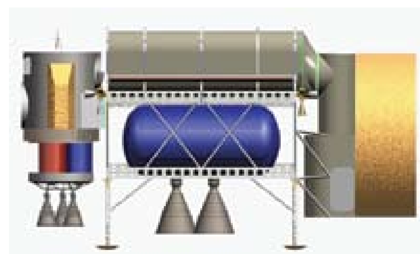


Figure 4-36. Initial LSAM Configurations

LSAM Configuration Selected for Further Assessment

Key FOMs were established to provide guidance for configuration selection. A survey of FOMs utilized for previous exploration architecture and vehicle design studies was performed, and the results were synthesized to drive out those qualities that would provide greater insight into the LSAM design and operational features. The FOMs used for further configuration downselection are provided in **Table 4-20**. The study team utilized a weighted scoring approach to better understand the relative benefits of the various configurations as they specifically apply to the FOMs. The two-stage vertical configurations were selected as the departure points for the ESAS Analysis Cycle 3, since they provided the best configuration for satisfying the identified FOMs.

Table 4-20. Key LSAM FOMs

Key LSAM Figures of Merit	
Overall Mission Performance	**Operations and Risk**
• Shroud diameter	• Crew access to surface
• Launch mass	• Landing stability
• Complexity of launch vehicle attachment	• Flight controllability
• Center of Gravity (CG) offset during launch	• Engine restarts
• Engine-out during landing	• Surface debris hazards on landing
	• Complexity of ascent separation
Outpost Mission Support	• Complexity of deployments
• Cargo unloading complexity	• Complexity of engine-out
• Leaving behind useful assets	
• Payload delivery with the crew	**Development Cost and Schedule**
• Capability of using local propellants (ISRU)	• Complexity of design and manufacturing
• Crew/cargo descent stage commonality	• Technology development cost
	• System commonality with the CEV

Driving Considerations

During the LSAM configuration study, several key driving considerations were identified as key elements impacting the overall vehicle performance. These key driving considerations were very interrelated and required in-depth consideration during the analysis.

Surface Access for Crew and Cargo

Providing the ability for the crew to routinely access the lunar surface is a key discriminator in the overall vehicle configuration selection. Concepts that increase the distance that the crew must traverse from their living quarters to the surface increase overall crew fatigue as well as probability of potential crew injury. Likewise, payload unloading must be considered, especially during the sortie and early outpost missions, when little or no lunar surface support is available. Providing configurations with an airlock integrated within the CM or an airlock split from the CM on another level of the LSAM were considered. This single versus split configuration is depicted in **Figure 4-37**.

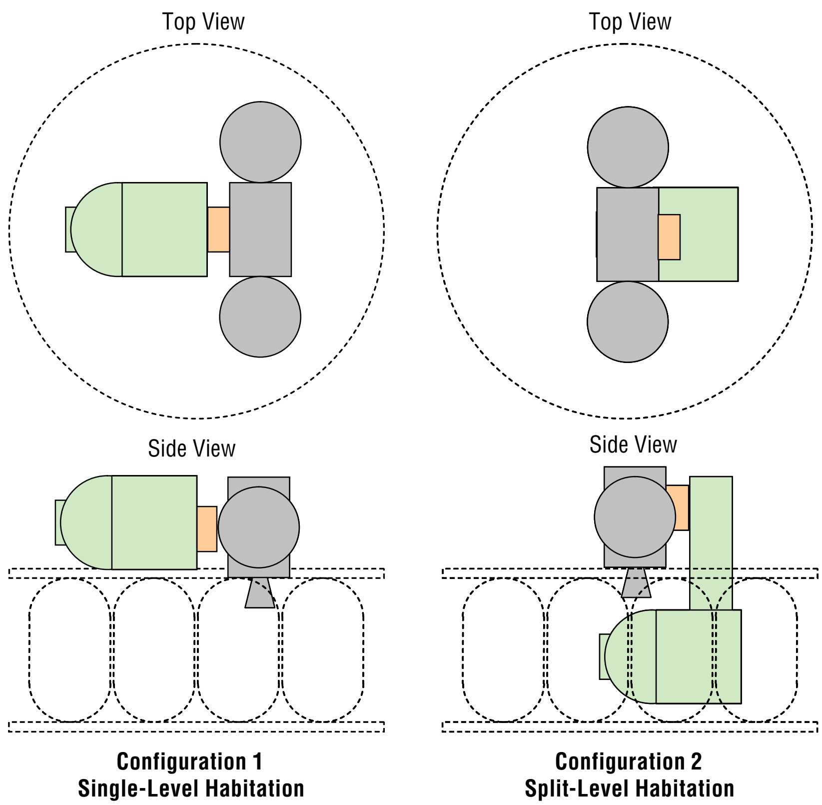


Figure 4-37. Single versus Split Level Habitation Concepts

The single-level habitation module configuration provides quick and easy access of the crew to the ascent stage for both nominal and emergency ascent conditions. In addition, this configuration leaves a cargo bay open on the lower level for payload while keeping an area available on the upper platform as well. Because the CM, including the airlock, is located on the upper level, all EVAs begin at a greater height above the surface.

Because the airlock is located on the lower level, the split-level habitation configuration provides easier crew access to the surface. In addition, the living space is well-protected from radiation because it is integrated among the propellant tanks. Unlike the single-level configuration, the living space on the lower level takes up most of the potential payload cargo space.

Propellant Volume and Tank Configuration

Due to the high propulsive performance required by the vehicle, propellant selection is a key aspect of the overall LSAM configuration. A balance must be found between the performance required and configuration layout of the vehicle. For instance, propellants that provide moderate performance (i.e., hypergolics) package more efficiently than higher-performing propellants (i.e., hydrogen/oxygen), but at the expense of overall increased mission mass. Cycle 2 ESAS analysis (**Section 4.2.4.1**, **Trade Studies**) indicated that higher-performing propellants were necessary to provide the overall mass efficiency for the architecture. This necessitated the use of higher-performing hydrogen and oxygen in the lunar descent leg of the mission. Unfortunately, hydrogen is a very voluminous fuel, requiring large tanks to store. This in turn complicates the overall vehicle packaging and increases the deck height of the landed vehicle.

Likewise, the LV shroud size (namely diameter) has a profound impact on the LSAM overall packaging efficiency and design as shown in **Figure 4-38**. As can be seen in this figure, the LSAM tank layout and descent stage deck height are directly driven by the shroud diameter. As the diameter is reduced, as is shown in the 7-m configuration, deck height is significantly increased and tank packaging becomes complicated. As the diameter is increased, as is shown in the 10-m configuration, deck height is decreased and additional space can be provided for engine intrusions as well as payload capacity.

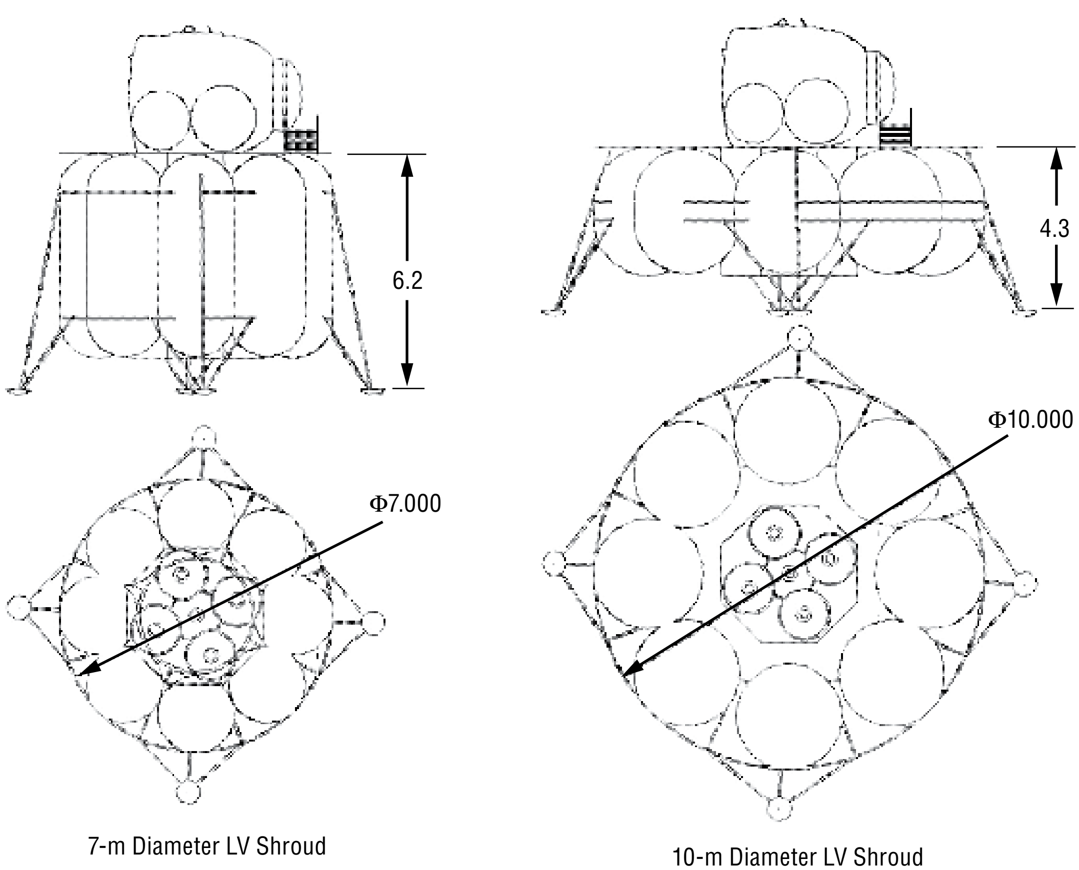


Figure 4-38. LV Shroud Diameter Influence on LSAM Tank Configuration

Common Crew/Cargo Design

One of the overarching guidelines for the study was to investigate a common descent stage design for both crew and cargo missions. Crew safety during ascent necessitates a separation system for the ascent and descent stages that can operate with a high degree of reliability. In addition, the surface strategy studies indicated the need for the delivery of a few large integrated payloads (i.e., surface habitat, pressurized rovers, and nuclear power systems). These payloads require large, unobstructed payload space, combined with the need to be as low as possible to the lunar surface for potential future unloading.

LSAM Crew Cabin Configuration Layout Trades

Several different configuration trades were conducted to determine the impact of varying degrees of segmentation of the living capabilities of the combined descent stage, ascent stage, and other elements. Three distinct types of CM splits were considered, including;

- Minimizing the ascent stage module volume to the greatest extent possible with augmented living space and EVA support for the surface phase of the mission;
- Providing a single CM for all crew support functions while segmenting the airlock functions necessary for routine exploration of the lunar surface; and
- Combining all CM and airlock functions into one element.

In all options, the descent stage provides the necessary transportation function of the ascent stage and living module to the lunar surface. In addition, it provides the LOI propulsion of the LSAM/CEV stack into lunar orbit.

Configuration Concept 1: Minimized Ascent Stage

The emphasis of this configuration layout was to minimize the overall size of the ascent stage to the greatest extent possible. This necessitated the split of the key functional requirements into the following segments, as illustrated in **Figure 4-39**:

- <u>Ascent Stage:</u> The ascent stage provides habitation and crew support during both the descent and ascent phases of the lunar landing mission. It also provides the necessary ascent propulsion for the ascent phase. A docking mechanism for rendezvous with the CEV is provided as well as a retractable attachment for crew transfer to the living module post-lunar-landing. A quick assessment of the interior layout and dimensions required for the crew for both standing and sitting postures was conducted. This assessment included combinations of (1) all crew standing, (2) two crew sitting and two crew standing, and (3) all crew sitting. Human habitability data indicated that the configuration of all crew standing, which was the method utilized during the Apollo missions, provided the minimum overall ascent module layout. This configuration provides approximately 10 m^3 of total pressurized volume and 5.5 m^3 of equivalent habitable volume.
- <u>Living Module/Airlock:</u> The living module supports the crew during the lunar surface phase of the mission. This includes all crew habitation during the surface stay of up to 7 days. This element also provides all necessary EVA support, including access to the surface via an airlock, EVA suit storage, and maintenance. A similar quick assessment of the interior layout of the living module was conducted. An integrated two-person airlock is configured at one end of the living module to provide the crew routine access to the lunar surface. The living module provides approximately 16.7 m^3 of total pressurized volume and 8.7 m^3 of equivalent habitable volume, with the airlock providing another 7.9 m^3 and 5.1 m^3 of pressurized and habitable volume, respectively.

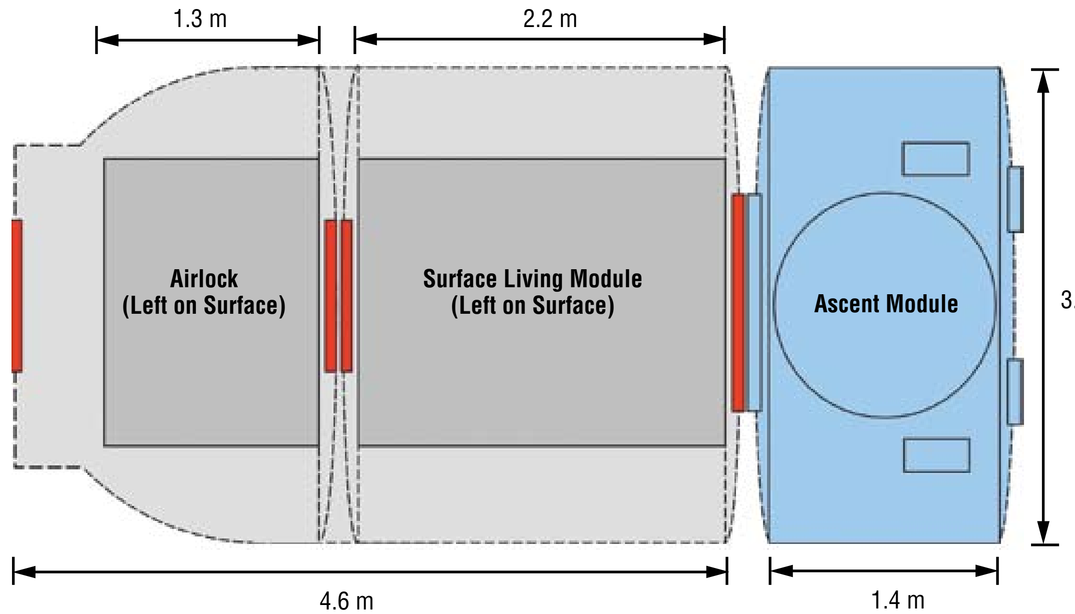


Figure 4-39. LSAM Minimized CM Vehicle Configuration

Configuration Concept 2: Separate Airlock

The emphasis of this configuration layout (**Figure 4-40**) was to provide a single crew module for the sortie missions (descent, surface stay of up to 7 days, and ascent) with a separate airlock. Surface-specific EVA functions, including an airlock as well as other EVA support functions, were provided via an airlock that remains on the lunar surface. This necessitated the split of the key functional requirements into the following segments:

- <u>Living Module/Ascent Stage:</u> The living module/ascent stage provides habitation and crew support during both the descent and ascent phases, as well as the surface phase of the lunar sortie mission. It provides the necessary ascent propulsion for the ascent phase. A docking mechanism for rendezvous with the CEV is provided as well as a retractable attachment to the airlock. This configuration provides approximately 16.7 m^3 of total pressurized volume and 8.7 m^3 of equivalent habitable volume.
- <u>Airlock:</u> A separate two-person airlock is configured at one end of the living module to provide the crew routine access to the lunar surface. A retractable access port provides the necessary separation between the airlock and the CM prior to ascent. The airlock module provides approximately 7.9 m^3 of total pressurized volume and 5.1 m^3 of equivalent habitable volume.

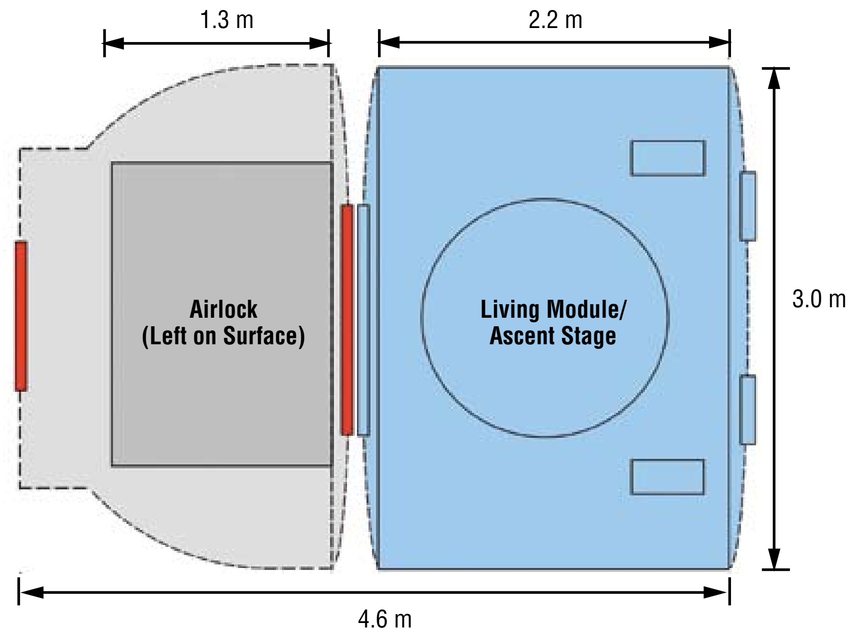


Figure 4-40. LSAM Separate Airlock Configuration

Configuration Concept 3: Combined Module

The emphasis of this configuration layout was on providing a single CM for the sortie missions (descent, surface stay of up to 7 days, and ascent) with an integrated airlock (**Figure 4-41**). This configuration layout provides the necessary ascent propulsion for the ascent phase. A docking mechanism for rendezvous with the CEV is also provided. In addition, an integrated two-person airlock is configured at one end of the living module to provide the crew routine access to the lunar surface. This configuration provides approximately 24.6 m^3 of total pressurized volume and 13.8 m^3 of equivalent habitable volume within the combined module.

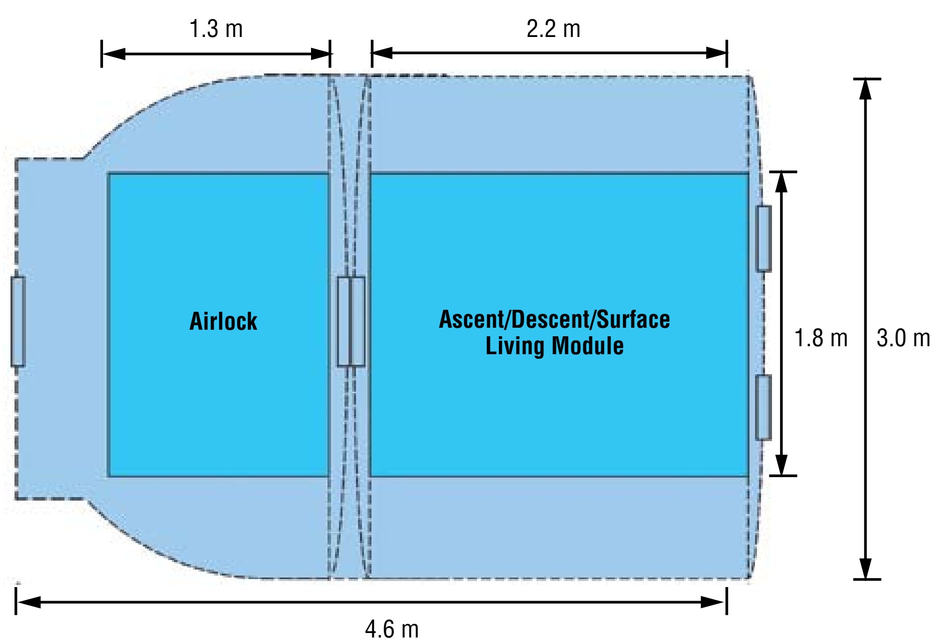
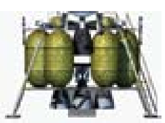
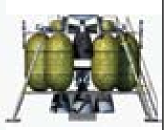


Figure 4-41. LSAM Combined Module Configuration

LSAM Configuration Summary

The Envision spacecraft sizing tool was used to perform quick performance trade assessments of the three vehicle configurations. Mass estimates for the ascent stage, living module, and descent stage performance were developed, a summary of which is provided in **Table 4-21**. As can be seen from this table, all three configurations land approximately the same mass on the lunar surface, in the 10-mT range. It is also interesting to note that integrated vehicle designs, such as the combined approach of Concept 3, provide better overall mass efficiency due to the synergistic design. This efficiency is gained through the elimination of additional redundancy of similar functions required by high degrees of separation of vehicle functions, as well as additional attachments required for segmented pressurized elements. Conversely, Concepts 1 and 2 leave behind potentially useful vehicle assets which could be used for future missions to the same landing site.

Table 4-21. LSAM Configuration Trade Mass Summaries

	Concept 1: Minimized Ascent			Concept 2: Separate Airlock		Concept 3: Combined	
	Ascent Stage	Living Module	Descent Stage	Ascent Stage	Descent Stage	Ascent Stage	Descent Stage
							
1. Structure	604 kg	673 kg	749 kg	712 kg	749 kg	714 kg	749 kg
2. Protection	60 kg	70 kg	275 kg	70 kg	266 kg	70 kg	265 kg
3. Propulsion	624 kg	00 kg	2,003 kg	664 kg	1,915 kg	678 kg	1,908 kg
4. Power	427 kg	260 kg	448 kg	427 kg	448 kg	427 kg	448 kg
5. Control	00 kg	00 kg	00 kg	00 kg	00 kg	00 kg	00 kg
6. Avionics	540 kg	161 kg	145 kg	540 kg	145 kg	540 kg	145 kg
7. Environment	349 kg	526 kg	233 kg	541 kg	177 kg	541 kg	177 kg
8. Other	710 kg	355 kg	576 kg	710 kg	534 kg	355 kg	530 kg
9. Growth	663 kg	409 kg	886 kg	733 kg	847 kg	665 kg	844 kg
10. Non-Cargo	247 kg	644 kg	1,567 kg	836 kg	1,475 kg	845 kg	1,468 kg
11. Cargo*	00 kg	00 kg	4,418 kg	00 kg	1,265 kg	765 kg	500 kg
12. Non-Propellant	41 kg	55 kg	510 kg	60 kg	463 kg	60 kg	463 kg
13. Propellant	3,545 kg	00 kg	29,537 kg	4,278 kg	27,653 kg	4,543 kg	27,489 kg
Dry Mass	**3,979 kg**	**2,455 kg**	**5,316 kg**	**4,396 kg**	**5,081 kg**	**3,990 kg**	**5,066 kg**
Inert Mass	**4,225 kg**	**3,098 kg**	**11,301 kg**	**5,232 kg**	**7,821 kg**	**5,600 kg**	**7,033 kg**
Total Vehicle	**7,811 kg**	**3,153 kg**	**41,348 kg**	**9,570 kg**	**35,937 kg**	**10,203 kg**	**34,985 kg**

* *Cargo includes airlock and/or living module*

The LSAM configuration study investigated basic differences in vehicle design and configuration. Crew size and mission durations were fixed, and limited mission mode and propulsion options effectively narrowed the LSAM trade space. LV shroud diameter was found to dictate the height of the hydrogen tanks needed for the descent stage, while the size of these tanks will impact the opportunities to integrate cargo or habitable volumes into the descent stage. The surface crew cabin could be divided into pressurized segments that could serve as ascent stages or habitable volumes left on the surface, but this separation increases the overall LSAM mass.

ESAS LSAM Configuration

Based on the results of the LSAM configuration trade studies, a combined CM design was chosen as a POD for future lander design studies. This concept was chosen to both provide the required airlock function and simplify ascent and descent stage interfaces. The description and mass property breakouts presented in this section are the result of additional lander analysis and refined subsystem mass estimation using the Envision sizing tool.

It was recognized, however, that returning this large pressurized volume of the combined crew cabin/airlock to lunar orbit does not provide the opportunity to utilize delivered pressurized volumes as elements of an incrementally deployed base. Further design work will refine the LSAM layouts, as well as define approaches for utilizing the remaining surface assets and the necessary resources required to keep those elements active for future use.

Ascent Stage Description

The reference LSAM concept for the ESAS 1.5-launch EOR–LOR architecture is a two-stage, single-cabin lander similar in form and function to the Apollo LM. The LSAM ascent stage, in conjunction with the descent stage, is capable of supporting four crew members for 7 days on the lunar surface and transporting the crew from the surface to lunar orbit. The ascent stage assumes an integrated pressure-fed oxygen/methane propulsion system, similar to the CEV SM, to perform coplanar ascent to a 100-km circular lunar orbit, rendezvous and docking with the CEV, and self-disposal following separation from the CEV. A single 44.5-kN (10,000-lbf) ascent propulsion system and sixteen 445-N (100-lbf) RCS thrusters are used for vehicle maneuvering and attitude control. Spherical ascent stage propellant tanks are sized to perform up to 1,866 m/s of ascent propulsion system and 22 m/s of RCS delta-V.

The LSAM pressure vessel is a horizontal short cylinder 3.0 m in diameter and 5.0 m long to provide 31.8 m^3 of pressurized volume for the crew during lunar operations. A nominal internal atmospheric pressure for the ascent stage of 65.5 kPa (9.5 psia) with a 30 percent oxygen composition has been assumed. The LSAM's notional EVA strategy while on the lunar surface is daily EVA with all four crew members simultaneously egressing the vehicle. For missions lasting beyond 4 days, a rest day between EVAs may be required. Unlike the Apollo LM, the LSAM ascent stage crew cabin includes a bulkhead to partition a section of the pressurized volume, which can serve as an internal airlock. Thus, crew members don their surface EVA suits in the airlock, depressurize the airlock, and egress the vehicle.

Ascent stage power generation capabilities include rechargeable batteries for the 3 hours from liftoff to docking with the CEV. Power generation for all other LSAM operations prior to liftoff is provided by the descent stage.

An illustration of the reference LSAM ascent stage is shown in **Figure 4-42**.

Figure 4-42. LSAM Ascent Stage

Ascent Stage Mass Properties

Table 4-22 below provides overall vehicle mass properties for the LSAM ascent stage. The mass properties reporting standard is outlined in JSC-23303, Design Mass Properties.

Table 4-22. LSAM Ascent Stage Mass Properties

LSAM Ascent Stage	% of Vehicle Dry Mass	Mass (kg)	Volume m^3)
1. Structure	20%	1,025	0
2. Protection	2%	113	1
3. Propulsion	17%	893	11
4. Power	11%	579	1
5. Control	0%	0	0
6. Avionics	8%	385	1
7. Environmental	17%	896	12
8. Other	7%	382	1
9. Growth	17%	855	5
10. Non-Cargo		834	5
11. Cargo		0	0
12. Non-Propellant		131	0
13. Propellant		4,715	0
Dry Mass	**100%**	**5,128**	
Inert Mass		**5,962**	
Total Vehicle		**10,809**	

Descent Stage Description

The LSAM descent stage, shown in **Figure 4-43**, is used in crewed lunar exploration missions to insert the CEV into LLO, land the ascent stage and cargo on the surface, and provide the vehicle's life support and power generation capabilities during an assumed 7-day lunar surface stay. The descent stage uses a pump-fed oxygen/hydrogen main propulsion system to perform LOI and coplanar descent from a 100-km circular lunar orbit. Four 66.7-kN (15,000-lbf) descent propulsion system engines derived from the RL-10 engine family are used for vehicle maneuvering while the ascent stage RCS is used for combined-vehicle attitude control. The descent propulsion system engines are arranged symmetrically around the vehicle centerline at the base of the descent stage.

Six cylindrical hydrogen and two cylindrical oxygen descent stage tanks are included on the LSAM to store the propellant needed to perform up to 1,390 m/s of LOI delta-V with the CEV and ascent stage attached, and 1,900 m/s of descent delta-V with only the ascent stage attached. Although the tanks are sized to hold the maximum propellant quantity needed to perform any possible descent stage mission, the tanks are only filled to the level needed for the specific mission being performed. For example, a long-stay lunar outpost mission to the north pole may only need a minimum-energy coplanar LOI maneuver (845 m/s), while a 7-day sortie to the far side south pole-Aitken Basin Floor may require a much larger LOI due to the plane change at arrival (1,100 m/s). Propellant mass not used for LOI delta-V can maximize the amount of useful cargo mass delivered to the lunar surface. The eight LSAM propellant tanks are mounted around the descent stage in a ring arrangement, leaving two open bays on opposite sides of the stage exterior for surface access and cargo stowage and a circular opening along the vehicle centerline for housing the single ascent stage engine nozzle. In addition to supporting its own propulsion system, the descent stage structure also serves as a support system and launch platform for the ascent stage, provides attachment for a four-leg landing gear system, provides for crew access to the surface, and serves as the attachment point to the EDS.

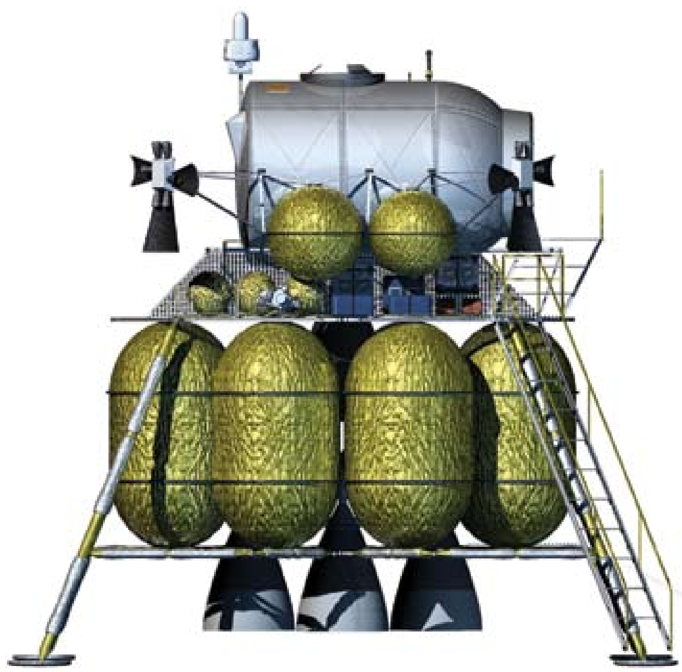


Figure 4-43. LSAM

Three Proton Exchange Membrane (PEM) fuel cells on the descent stage provide LSAM power generation from Earth launch to lunar ascent. Oxygen reactant for the fuel cells is stored in the oxygen propellant tanks, while hydrogen reactant is stored in the hydrogen propellant tanks. The descent stage also contains the gaseous nitrogen, potable water, and water storage systems needed for the mission up to lunar ascent. These systems were included on the descent rather than the ascent stage to avoid the penalty of lifting unnecessary mass back to lunar orbit. Finally, the descent stage provides the mounting location for the active thermal control system radiators. LSAM heat rejection following liftoff from the lunar surface is accomplished using a fluid evaporator system.

Table 4-23 below provides overall vehicle mass properties for the LSAM descent stage assuming a 7-day sortie mission to the far side south pole-Aitken Basin Floor. The mass properties reporting standard is outlined in JSC-23303, Design Mass Properties.

Table 4-23. LSAM Descent Stage Mass Properties

LSAM Descent Stage (Sortie Mission)	% of Vehicle Dry Mass	Mass (kg)	Volume (m³)
1. Structure	18%	1,113	0
2. Protection	1%	88	0
3. Propulsion	38%	2,362	93
4. Power	8%	468	0
5. Control	1%	92	0
6. Avionics	1%	69	0
7. Environmental	5%	281	12
8. Other	10%	640	1
9. Growth	17%	1,023	5
10. Non-Cargo		1,033	5
11. Cargo		2,294	0
12. Non-Propellant		486	0
13. Propellant		25,105	0
Dry Mass	**100%**	**6,137**	
Inert Mass		**9,464**	
Total Vehicle		**35,055**	

4.2.5.1.4 Outpost Deployment Strategies

The above discussion of the LSAM configuration is incomplete without considering how thelander will be used in the deployment of the lunar outpost. **Section 4.3.5**, **Lunar Surface Traffic Model**, introduces lunar surface operations, and **Section 4.3.8**, **Outpost Deployment Studies**, discusses lunar outpost deployment in more depth. **Appendix 4F**, **Alternative Outpost Deployment Options**, explores other alternatives. However, in order to arrive at a lander configuration that will become the basis for the remaining Cycle 3 performance analysis, some discussion of the capabilities of sortie missions and the transition to outpost missions is necessary.

Several outpost deployment strategies were considered, including delivery of outpost elements on large cargo landers and the incremental buildup of an outpost using the excess cargo carrying capacity of lunar sortie missions. The initial ESAS outpost deployment strategy was based on the premise of a short series of 15- to 20-mT landed cargo missions emplacing large, monolithic payloads. This initial outpost deployment approach was to be traded against alternate outpost deployment strategies to assess the degree to which the early sortie missions could be leveraged to enable an incremental deployment of the outpost. As a major element of the outpost, the habitat received special attention, due to its size and difficulty to separate into smaller components. The team was asked to consider if smaller habitable volumes from crewed LSAMs could be used to incrementally construct a habitat using smaller ascent stages, leave-behind crew cabin modules, and deployed payload modules.

Dedicated Cargo Lander Strategy

The initial ESAS outpost deployment strategy deployed the core outpost in three dedicated "cargo" flights and a fourth mission that prepositions a backup LSAM. The fifth flight to the outpost delivers the first crew in an LSAM that will be used to return the second crew complement to Earth. The outpost is completed after five dedicated cargo flights, and is shown in **Figure 4-44**.

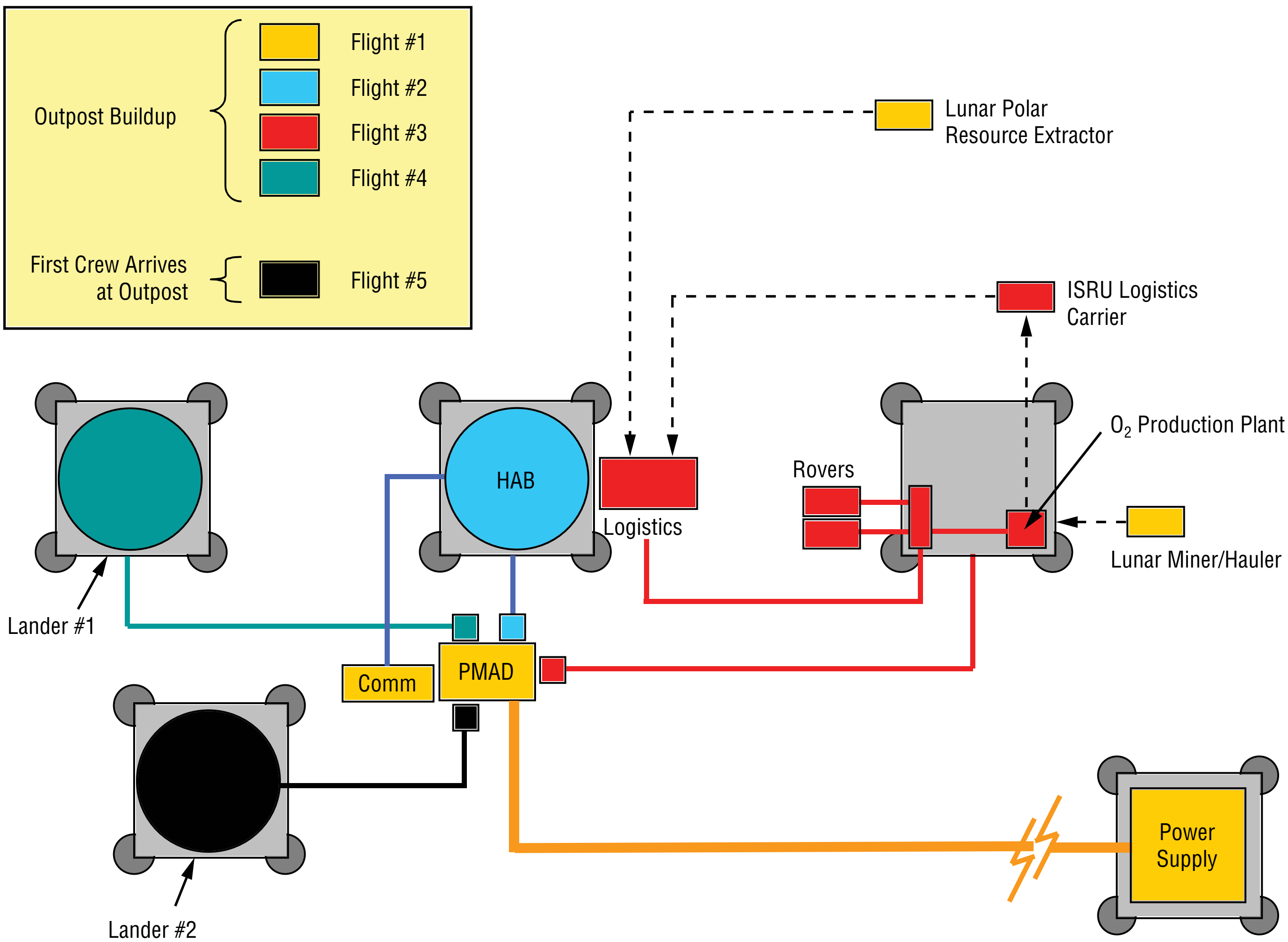


Figure 4-44. Core Outpost Schematic

This initial strategy corresponds to the "pointer" location shown in **Figure 4-54** (which appears later in the report in **Section 4.3.5**, **Lunar Surface Traffic Model**), and serves as a starting point for the analysis of outpost deployment strategies. A number of key assumptions was used by the ESAS team to formulate the initial outpost deployment strategy:

- The outpost IOC will be 2022;
- The outpost deployment phase will be preceded .by a lunar sortie mission phase of approximately 2–4 years; however, sortie missions are not incorporated as part of the outpost deployment strategy;
- The outpost will maintain a continuous, sustained four-crew member presence with crew rotations every 6 months;
- The outpost will be deployed on an elevated feature at a polar region that provides long durations of uninterrupted lighting;
- The outpost is not in continuous view of Earth;
- The descent stage is capable of delivering 15 metric tons (mT) to the lunar surface;
- Deep Space Network (DSN) (flight navigation) and Tracking and Data Relay Satellite (TDRS) (communication) resources are available for continuous support of lunar operations or can be modified to provide this service;
- A nuclear reactor serves as the primary outpost power source; and
- Precursor missions have accomplished the following tasks:
  - Demonstrated ISRU technologies such as O2 production, H2/H2O extraction, and excavation of regolith; and
  - Developed an enhanced lunar gravity potential model.

The purpose of the outpost is to establish an initial set of core lunar surface operating capabilities. Additionally, as mission objectives become more challenging and extensive, surface operations will require an evolved set of surface capabilities. Initial capabilities include the following:

- Enable a continuous, sustained human presence;
- Enable frequent local (3-km radius) and near-field (15-km radius) Extra Vehicular Activities (EVAs);
- Enable in-depth, in-situ data collection and analysis:
  - Field experiment deployments;
  - Lunar geosciences; and
  - Human physiological adaptation.
- Enable ISRU demonstrations/pilot operations:
  - Regolith excavation and transportation;
  - Oxygen production from regolith; and
  - Long-term cryogenic fluid storage and transfer of oxygen.

Evolved capabilities include the following:

- Maintain and grow logistics chain:
  - Landing and traversable zone build-up and clearance; and
  - Lunar-produced logistics augmentation.
- Enable mid-field (30 km radius) and far-field (30+ km radius) EVAs;
- Provide the ability to add additional lunar-based science infrastructure:
  - Space physics; and
  - Astronomy.
- Enable large-scale ISRU production:
  - Large-scale regolith excavation and manipulation;
  - Consumable and propellant production; and
  - Surface construction (pads, berms, roads, etc.).

The design principles that were employed in the creation of the outpost deployment strategy were:

- Landed elements should not be required to move unless absolutely necessary;
- Autonomous activities (e.g., for locomotion or payload manipulation) should only be performed if absolutely necessary;
- Required crew operations for outpost deployment should be limited and simple;
- Landed elements should be delivered on common cargo descent stages;
- Common functions (e.g., power distribution) should be performed by common means; and
- The logistics supply chain should require minimal crew time and robotic manipulation.

"Incremental Build" Strategy

Cost analyses performed in parallel with ESAS identified the predeployed outpost as a major architecture cost driver. This led the team to analyze alternate methods of deploying the outpost, included combining sortie and cargo delivery functions into a single vehicle and delivering the components of the outpost in smaller (2- to 5-mT) elements.

Specifically, the second concept examined whether outpost assets could be built from pieces that were left behind from previous missions. This included examination of pieces that were used by the sortie mission crew but left behind, as well as pieces that were carried as payload on the sortie missions for the sole purpose of future assembly into usable assets.

The same assumptions for outpost deployment listed for the cargo lander strategy were to be used for all outpost deployment alternatives. For the "incremental build strategy," two assumptions were modified as follows:

- The outpost deployment missions can begin as early as the first sortie mission and will increase in capability with each subsequent sortie visit; and
- The outpost will gradually build to a permanent four-crew member presence.

As a first step, the ESAS team examined which outpost assets, if any, made sense to break into modular pieces. Therefore, the team reviewed the manifest of outpost elements, breaking the list into three categories: (1) Less than 2,000 kg; (2) between 2,000 kg and 10,000 kg; and (3) greater than 10,000 kg. The general feeling was that, if an element had a mass less than 2,000 kg, there was no reason to try to break it into smaller pieces. If an element was in the 2,000-kg to 10,000-kg range, it was a potential candidate, but must have extremely good rationale and prove to be relatively simple. If an element had a mass greater than 10,000 kg, it was a candidate that deserved a thorough review. The outpost deployment manifest is as follows:

- Less than 2,000 kg
  - Rover Logistics Box (100 kg),
  - Two Unpressurized Rovers (500 kg),
  - ISRU Lunar Miner/Hauler (600 kg),
  - ISRU O2 Pilot Plant (800 kg),
  - Inchworm (900 kg),
  - ISRU Logistics Carrier (1,000 kg),
  - ISRU Lunar Polar Resource Extractor (1,200 kg), and
  - Power Management and Distribution (PMAD)/Communications Center (1,570 kg);
- 2,000 kg – 10,000 kg
- Greater than 10,000 kg
  - Logistics Module (10,000 kg),
  - Pressurized Rover (10,000 kg) (Evolved Capability),
  - Primary Surface Power Source (11,500 kg), and
  - Habitat Module, not outfitted (15,000 kg).

The elements fell within the two categories: (1) 1,570 kg and less and (2) 10,000 kg or greater. The evolved payload unloader was the one item that fell into the “potential candidate” range, but, since this element is a relatively sizable mobile truck with an integrated scissor-jack lift, it was not conducive to lunar surface assembly operations. Therefore, the team focused on the four items in the “Greater than 10,000 kg” category.

Logistics Module

The logistics module was the best candidate for splitting into separate pieces, but not in the way that was originally envisioned at the beginning of this task (i.e., split into pieces for future assembly). Instead, it was felt that this element could possibly be eliminated altogether, if the assumption regarding the down-mass cargo capability of the descent stage was changed. The original assumption for the descent stage was that it would have a down-mass cargo capability of 15 mT. During a human mission, this meant that a cargo of approximately 500 kg could be carried to the lunar surface, given the mass of the LSAM ascent stage. If the down-mass capability of the descent stage was to be increased by 2,000 kg to 5,000 kg, the potential exists to carry the required logistics for the habitat with each crew increment. To be certain of this conclusion, a detailed assessment of the habitat logistics resupply requirements should be performed.

If the assumption regarding the descent stage's down-mass cannot be made, it is not clear that it would make sense to partition this element into modular pieces. The purpose of the logistics module was to deliver fresh supplies (fluids and pressurized and unpressurized cargo) to the habitat to support each 6-month crew increment (one logistics module per crew increment). While it is true that this element could be assembled on the lunar surface, or could exist as three separate entities, the purpose for doing so is not evident. Cargo missions would still be required to deliver logistical materials to the lunar surface. These logistical materials will already have been packaged on Earth according to their requirements (pressurized, unpressurized, or in holding tanks). Because this packaging would have been needed during the transit phases and, therefore, delivered to the lunar surface, it would not make sense to transfer from one set of packaging to another (the logistics module). Therefore, this idea was not pursued.

Pressurized Rover

The pressurized rover would seem to be a difficult item to build on the lunar surface. Simplistically, a person could envision a pressurized rover as a small habitat placed on a mobile chassis. Therefore, one could envision that sortie mission hardware could be leveraged to construct a pressurized rover on the lunar surface, if a mobile chassis was delivered to the surface and the crewed version of the LSAM had a configuration that left a habitable volume behind on the lunar surface. However, this idea is extremely hard to implement. Many of the same problems associated with using LSAM pieces to construct a habitat also apply to this approach. In brief, some of the major problems are:

- After severing all power, data, fluid, and structural connections with the LSAM descent stage, the habitable module must be unloaded from the descent stage (e.g., crane, placed on wheels, etc.);
- The habitable volume must be moved to the vicinity of the pressurized rover chassis;
- The habitable volume must be lifted (e.g., crane) and placed onto the chassis, and all required connections must be made—power, data, fluid, structural;
- The habitable volume will require all systems (e.g., ECLSS, Thermal Control System (TCS), etc.) that are required to create a surface habitable volume—this means that duplicate systems will be required between the ascent stage and the habitable volume that is left behind;
- The habitable volume's systems must be designed with a significantly longer lifetime than is needed to support sortie missions and must be able to accommodate multiple reuses; and
- No Earth-based integrated validation of the final configuration will be possible.

There is a possibility that the chassis could be pre-integrated with the portion of the LSAM's habitable volume that is left on the lunar surface. However, this has implications as well. The first implication is that, in order to score out the proper mass allocation and interfaces to be able to carry a chassis as an integrated part of the LSAM configuration, much more would need to be known about the pressurized rover during the LSAM design period. This will probably not be the case for a couple of reasons. First, the pressurized rover chassis will probably be designed from lessons learned from the unpressurized rover design after years of operations. Secondly, pressurized rovers are not needed until several years into the lunar surface program; therefore, there will probably not be an element of intense design scrutiny until several years after the LSAM design is underway (or is already developed).

A second implication of this approach is the permanent impact it would have on the LSAM design. As stated previously, the proper mass allocations and interfaces would need to be designed into the LSAM. If this were the case, three options would be present:

- Deliver a mock chassis to the lunar surface on every flight so that the components of the LSAM that depend on interfaces of the chassis (e.g., structural) would have their required support. This results in the delivery of extra mass to the lunar surface that is essentially wasted and would not otherwise be flown;
- Deliver a real chassis to the lunar surface on every flight so that the components of the LSAM that depend on interfaces of the chassis (e.g., structural) would have their required support. This results in the delivery of a usable pressurized rover to the lunar surface with each crewed LSAM—which probably results in developing many more pressurized rovers than required;
- Design the LSAM such that no components of the LSAM are dependent on the interfaces of the chassis, but can accommodate them when flown. This would require a significant engineering effort and a high degree of knowledge about the pressurized rover at the time of the LSAM design efforts. The advantage of this approach is that it would free the chassis' mass allocations for use by other cargo.

None of the three options are ideal. The third option would provide the most efficient use of LSAM capabilities, but requires knowledge of the pressurized rover at an extremely early point in the program, which is probably impractical.

Primary Surface Power Source

The primary surface power source is a potential candidate for splitting into modular pieces, but only if the assumption is made that it is not a nuclear fission power source. If that was the case, there are two ways in which modularizing the power source could be of benefit to the outpost deployment strategy. The first way entails leveraging spent LSAM descent stages with their associated power generation/storage/distribution systems to serve as the outpost's primary power source. The second way potentially eliminates the first outpost deployment flight (delivery of the nuclear power source and PMAD Center) by delivering the power source in smaller pieces along with the crewed LSAM missions.

The first candidate approach involves linking multiple LSAMs together on the lunar surface with power cables. This idea proves to be marginally viable only if the power system's mass allocation can be increased to use the 500-kg cargo down-mass capability of the crewed LSAM missions and to use the descent tanks for storage of fuel cell reactants. This concept was examined as a follow-on activity to the development of the initial outpost deployment strategy task. The team members who performed the analysis found that, if the power system's mass increased to use the entirety of its allocation (800 kg), plus the 500 kg from the cargo down-mass capability of the LSAM, each LSAM could be modified to provide approximately 4 kWe during the lunar day and approximately 2.7 kWe during the lunar night. Because previous outpost power requirement estimates found that 50 kWe is a minimum threshold for full-scale outpost operations, this means that 12–13 LSAMs would need to be linked together, if the outpost could operate on 32–35 kWe during the periods of lunar night (ranging from a few days at select locations at the polar regions to 14 days at most locations on the lunar surface). If a decreased lunar night operating power is not acceptable, 18–19 LSAMs would need to be linked together, providing 72–76 kWe during the lunar day. This approach also means that the cargo-carrying capabilities of the crewed LSAM missions would be completely consumed by power source systems, unless the descent stage's capabilities are increased.

The second candidate approach involves delivering a solar/Regenerative Fuel Cell (RFC) power source in smaller pieces as cargo on the crewed LSAM missions, without trying to connect LSAMs together. As with the first approach, this means that the LSAM's cargo down-mass capabilities are being consumed by power source systems. **Table 4-24** summarizes masses associated with a few 25-kWe and 50-kWe solar/RFC systems that are packaged and pre-integrated on Earth.

Case	Power Level (kWe)	Total System Mass (kg)		
		Gaseous	All Cryogenic	O2 Cyrogenic
Equatorial – Noon, Hot Case	25	18,654	15,800	16,502
45° Lat – Noon, Sun Off-Pointed Case	25	19,069	16,550	16,970
85° Lat – Polar Sun-Tracking Case	25	18,339	15,076	15,931
Shackleton North Rim Only	25	8,986	6,853	8,008
Shackleton North Rim Only	50	15,927	14,970	10,509

Table 4-24. Solar/RFC Total System Masses for Select Locations

The approach for modularizing a solar/RFC system was not studied, but as can be seen in **Table 4-24**, the total system masses for the non-Shackleton Crater locations (25-kWe systems) range from approximately 15 to 20 mT. Therefore, if the assumption limiting the LSAM's cargo down-mass to 500 kg is increased to allow a down-mass capability of approximately 5,000 kg, a the solar/RFC system could be delivered in 5-mT pieces. However, this introduces significant inefficiencies (e.g., mobility systems would be required for each piece), and the design must accommodate a modular approach, which then introduces design complexities. Therefore, the total system masses listed in **Table 4-24** should be expected to grow if a modular approach were to be pursued.

Habitat Module

The habitat received special attention as a part of this effort. The main question was whether habitable volumes from crewed LSAMs could be used to construct the habitat. This would require LSAM configurations with minimized ascent stages and leave-behind habitable modules as part of the outpost deployment strategy. Based on the findings from this extensive effort, the team strongly recommended against this approach.

One of the key findings from this effort was that the habitable modules would be severely deficient in the types of systems that would be incorporated into their design, due to the functional allocation required by the LSAM. Without an independent set of core functions (e.g., life support, thermal control, etc.), the habitable modules were severely lacking and did not provide a good basis from which a habitat could be constructed. However, it was felt that different approaches to habitat modularization might exist that could help decrease early program costs without introducing significant complexities. An example of this might be to develop a fully functional one-story habitat, rather than the originally desired two-story habitat, and add habitable volumes to this habitat as the lunar mission progresses. This would establish a core functional environment on the lunar surface (albeit limited in size) to which the program can build upon as desired. This approach would also allow for a low-risk environment for testing aggressive habitat technologies, such as inflatable add-on modules, at a pace that the program deems acceptable, while always ensuring that there is the core functional habitat available to the crew.

Outpost Deployment Strategy Conclusions
Lunar surface outpost deployment strategy options were studied to determine the order and manifest of the flights required to deploy a core set of lunar surface capabilities for sustained, concentrated lunar operations and to provide for the evolution of the surface capabilities as the lunar program progresses. The cost distribution for different outpost deployment strategies was found to be highly dependent on the time frame over which the outpost is deployed and the number of unique element developments required.

A core set of elements is required in order to enable the key capabilities associated with maintaining a human presence on the lunar surface (e.g., habitat, EVA suits, etc.). Additional elements can be added to the architecture depending on the desire to seek greater degrees of self-reliance (e.g., ISRU) or to seek to operate in operationally challenging areas (e.g., out of constant view of Earth). It should not only be expected that performance requirements will grow or change with time (e.g., expanding EVA traverse capabilities), but they should change to take advantage of a growing set of surface assets and crew availability (e.g., payload unloading strategies).

The ESAS concluded that a reasonable approach for outpost deployment can be achieved by incrementally deploying outpost elements, and incrementally gaining capability over time. **Appendix 4F**, **Alternate Outpost Deployment Strategies**, discusses additional deployment strategies.

4.2.5.2 Analysis Cycle 3 Performance

4.2.5.2.1 1.5- and 2-Launch Architectures

Analysis Cycle 3 focused on comparing the performance of the 1.5-launch EOR–LOR architecture to the 2-launch EOR–LOR and 2-launch LOR architectures after incorporating the final CEV CM shape and the results of the Cycle 3-focused trade studies. The final CEV shape and configuration was a 5.5-m base diameter, 32.5-deg sidewall angle capsule shape with no dedicated radiation shielding over and above that provided by the intrinsic design of the vehicle. The CEV SM and LSAM descent stage tanks are still designed to carry the maximum possible TEI and LOI propellant required; however, the vehicles will only load the propellant required to perform 7-day sortie missions to either the top 10 sites identified or long-stay outpost missions to a near-polar outpost. The LSAM ascent stage, which has a single combined CM, also includes an internal airlock for surface EVA. Pump-fed oxygen/hydrogen is used for the descent stage and pressure-fed oxygen/methane is used on the ascent stage and SM.

Figure 4-45 illustrates how CEV- and LSAM-injected masses vary with mission design requirements. The LSAM mass limit, as constrained by the performance of the CaLV and EDS, is shown as a function of CEV-injected mass. Because the EDS must perform TLI for both the CEV and LSAM, increased CEV mass decreases the LSAM mass that can be launched from Earth and injected to the Moon. The other mass limit comes from the performance capabilities of the CLV. The assumed LV can launch up to a 24.5-mT CEV to the insertion orbit in LEO. Since both lines maintain a 10 percent performance margin on the EDS and LV, if the combination of CEV and LSAM masses for a given mission fall within these limits, the in-space elements have a positive performance margin for additional mass growth or payload delivery capability. Lines of constant LOI delta-V are also plotted in the figure, showing the effect of delta-V and CEV mass on the LSAM, along with different TEI delta-Vs for a particular LOI delta-V. Constant TEI delta-V lines are vertical lines but were omitted in the figure for clarity.

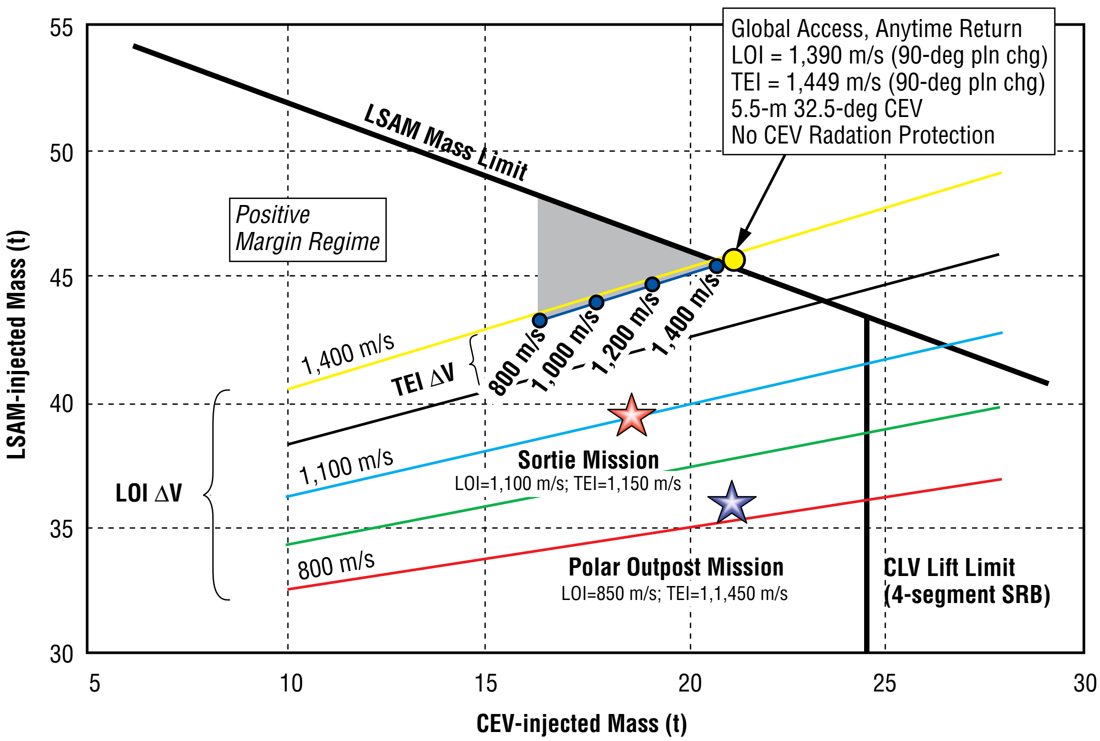
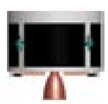


Figure 4-45. LSAM and CEV Mass Constraints for 1.5-Launch EOR–LOR

Three additional points are also plotted in the figure, representing three possible mission designs for the CEV and LSAM. The first point, indicated with a yellow circle, shows the cost of carrying 90-deg plane change capability on the CEV and LSAM for full global access and "anytime return" capability. While the CEV and LSAM propellant tanks are sized to carry that propellant mass, simultaneously including the mass on both vehicles is highly inefficient, as evidenced by the fact that the global access/"anytime return" trade demonstrated that LOI and TEI delta-V could be significantly reduced by simply loitering for a short while in LLO. Nonetheless, the LSAM-injected mass lies on the LSAM mass limit line, demonstrating that this capability is feasible, albeit with no additional performance margin. The second and third points, indicated with red and blue stars, represent the CEV and LSAM mass for 7-day missions to the top 10 sortie sites and long-stay missions to a polar outpost, respectively. Each mission has different TEI and LOI delta-V requirements, though both are comfortably within the LSAM mass and CLV lift limits. The polar outpost mission only requires a minimum energy coplanar LOI but may require a full 90-deg plane change to return to Earth (i.e., TEI) at any point during the long stay on the Moon. The 7-day sorties, though, require a plane change at LOI to set up a lunar parking orbit for minimum-energy anytime-ascent capability during the time on the surface, thus requiring greater LOI delta-V than the outpost mission. At the same time, since the surface mission only lasts 7 days, the full TEI plane change capability included with the outpost mission is not required here. Therefore, the CEV-injected mass is lower for the sortie mission than the outpost mission due to its lower TEI delta-V, while the LSAM-injected mass is higher, owing to its higher LOI delta-V.

Table 4-25 provides performance margins for the three LOR architecture variants using both Design Cycle 2 and Cycle 3 mission design assumptions. In Cycle 2, the descent stage and SM included the propellant mass needed for 90-deg plane changes at LOI and TEI. The CEV CM also included 2 g/cm^2 of supplemental polyethylene radiation shielding. Using these assumptions, the 1.5-launch EOR–LOR architecture had negative performance margins on its in-space elements. When Cycle 3 assumptions for a 7-day sortie are applied instead, the architecture has adequate performance margins and could deliver several tons of additional cargo mass to the lunar surface (if desired), similar to the 2-launch EOR–LOR and LOR architectures. The 1.5-launch EOR–LOR architecture has significantly less margin for the CEV than the others, as it uses the smaller CLV for launching the CEV and crew. The maximum mass margin is provided by the 2-launch EOR–LOR architecture.

Table 4-25. Performance Margins for Cycle 3 Architectures

	Cycle 2 Assumptions *Global Access 4-Day Sortie, "Anytime Return," 2 g/cm^2 Radiation Protection*			**Cycle 3 Assumptions** *Top 10 Site Access for 7-Day Sortie, "Anytime Return," 0 g/cm^2 Radiation Protection*		
Mission Scenario	**1.5-Launch EOR–LOR**	**2-Launch EOR–LOR**	**LOR**	**1.5-Launch EOR–LOR**	**2-Launch EOR–LOR**	**LOR**
CEV Capsule	-475 kg	+4,795 kg	+2,585 kg	+795 kg *(limited by SRB lift & Polar Outpost mission)*	+7,900 kg	+2,920 kg
CEV SM	-580 kg	+5,840 kg	+3,145 kg	+975 kg *(limited by SRB lift & Polar Outpost mission)*	+9,630 kg	+3,600 kg
Ascent Stage	-230 kg	+3,220 kg	+2,150 kg *(LSAM does LOI)*	+1,285 kg	+4,540 kg	+2,815 kg *(LSAM does LOI)*
Descent Stage*	-440 kg	+6,185 kg	+4,140 kg *(LSAM does LOI)*	+2,270 kg	+8,785 kg	+5,410 kg *(LSAM does LOI)*

**Descent stage includes 500-kg landed cargo.*

4.2.5.2.2 Single-Launch Architecture Performance

The Design Cycle 3 studies also included an initial performance analysis of architecture excursions wherein the CEV, LSAM, and EDS are combined into a single launch similar to the Apollo lunar missions. In the ESAS final reference architecture (the 1.5-launch EOR–LOR architecture), the EDS and LSAM are first launched to Earth orbit using the heavy-lift CaLV with CEV and crew following within hours or days with a launch of the smaller SRB-derived CLV. The vehicles rendezvous and dock in LEO and, once mated, the combined stack departs for the Moon. The single-launch option differs from this mission mode in that the EOR and docking phases from the mission are eliminated. Instead of launching on the CLV, the CEV and crew launch with the CaLV on top of the LSAM–EDS stack. Eliminating the rendezvous phase shortens the total mission time by 3 days and reduces the CEV service propulsion system and RCS delta-V budget by approximately 150 m/s. With single-launch architectures, the only time spent in Earth orbit is 1–2 orbit revolutions for vehicle checkout and to phase for lunar departure. Another difference with the single-launch option from the ESAS reference architecture comes with the configuration of the CaLV. The configuration must be modified to allow the CEV to launch on top of the vehicle. The payload shroud enclosing the LSAM in the baseline CaLV is replaced with an LSAM/CEV adapter that supports the gross mass of the CEV during launch and possibly also supports the mass of the LSAM as was done with the Saturn V. After ascent and TLI, the CEV separates from the adapter, the adapter panels are jettisoned to expose the LSAM docking mechanism, and the CEV transposes and docks with the LSAM. The remainder of the single-launch mission functions identically to the reference architecture.

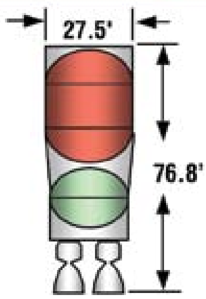
Two single-launch mission options were analyzed to compare the missions' required trans-lunar injected mass against the capability of the modified CaLV. In the first option, the Cycle 3 vehicle designs for the CEV and LSAM were fully retained both in vehicle scale and in subsystem design, while propellant and other consumables were offloaded as needed to fit within the TLI mass constraint. This produced an architecture capable of transporting four crew to the lunar surface for up to 7 days while restricting surface access to near-equatorial landing sites only. In the second single-launch option, a full global surface access capability with anytime return was retained from the ESAS final reference architecture; however, the crew size was reduced from four to two. The Cycle 3 CEV and LSAM designs were likewise reduced in scale to accommodate the smaller crew. The following two sections describe the single-launch options in further detail. Note that the single-launch option analysis was limited to mass comparisons and the options have not been analyzed for risk, cost, or schedule impacts. Impacts will include major launch support structure and CaLV modifications, certification of the CEV and CaLV for human-rated launches on that vehicle, and others to be determined.

Mission Constraints with Current CEV/LSAM

This single-launch option retains the current CEV and LSAM configurations from the ESAS final reference architecture but constrains the mission capabilities to fit both elements on the modified CaLV. The mission still assumes a four-person crew size, 7-day sortie surface duration, and 500-kg cargo delivery to the surface. The CEV and LSAM shapes, scales, and subsystem designs from the reference architecture are also still assumed. Mass savings occur by restricting access to the lunar surface to landing sites located within a few degrees of the lunar equator—sites such as the Mare Smythii basin located at 2.5°N, 86.5°E. Restricted surface access allows the CEV to carry less propellant to perform TEI, thereby reducing its mass. The delta-V for TEI assumed in the reference architecture, approximately 1,150–1,200 m/s, can be reduced to 950 m/s while still retaining full-entry coazimuth control at Earth. The difference between the reference architecture and this single-launch option is that near-equatorial sites do not require large plane changes to align the CEV parking orbit for anytime Earth return, while mid-latitude and near-polar sites do. Near-equatorial sites also require much less delta-V for LOI and, when combined with a lower CEV mass, the required propellant quantity on the LSAM descent stage can be greatly reduced. As with the TEI burn, restricting lunar surface access can eliminate a large plane change during LOI. The required delta-V has been reduced from 1,100 m/s in the final reference architecture to 900 m/s in this restricted-access option.

The combined effect of emplacing mission constraints while retaining the current CEV and LSAM configurations is a required trans-lunar injected mass of 55.6 t. The modified CaLV assumed for the single-launch options, including mass growth, performance reserves, and 10 percent margin, can deliver 54.6 t to TLI. Therefore, given the close match in injected mass requirement to LV capability, this single-launch option was tentatively considered technically feasible, yet it does require a portion of the 10 percent margin to be used to make up the 1.0 t performance shortfall. This mission option is summarized in **Figure 4-46**.

Option	Surface Access	Crew Size	CEV	LSAM	Required TLI Injected Mass
ESAS Cycle 3 Vehicle w/ Restricted Surface Access	7-Day Near-Equatorial Sorties Only	4 Crew, all to lunar surface	• 5.5-m, 32.5-deg CEV • Press-fed CH4 • No radiation protection	• 31m³ pressurized volume (LM = 6.7m³) • Press-fed CH4 ascent, pump-fed H2 descent & LOI	55.6 t



Launch Vehicle 27.3 – EDS + Payload w/ Suborbital Burn: 54.6 t to TLI

Figure 4-46. Single Launch LOR Mission Performance

LSAM/CEV Constraints with Current Mission Requirements

The second single-launch option differs from the first in that, instead of retaining the CEV and LSAM configurations from the reference architecture while restricting surface access to near-equatorial lunar sites, it retains global surface access and anytime return while restricting CEV and LSAM size by reducing the crew size from four to two. As in the reference architecture, a 7-day surface stay time and 500-kg surface cargo delivery is still assumed. This provides an exploration capability comparable to the Apollo program in terms of number of crew on the surface while exceeding its capability in stay time and surface access. The longest Apollo mission, Apollo 17, lasted only 3 days on the surface, while the highest landing site latitude was 26.1°N for Apollo 15.

With the smaller crew size, the pressurized volume of the CEV and LSAM are reduced from 29 and 31 m^3 to 22 and 15 m^3, respectively. The 5.0-m diameter, 30-deg sidewall angle capsule shape from Design Cycle 1 is the assumed CEV shape in this single launch option. Reducing the crew size from four to two produces a trans-lunar injected mass of 55.1 t, compared to a 54.6 t capability with the LV. Again, this option was considered technically feasible according to this preliminary performance analysis while requiring a reduced margin on the LV for the 0.5 t shortfall.

4.2.5.3 Figures of Merit

4.2.5.3.1 Safety and Reliability

The same probabilistic techniques used in previous analysis cycles, and described in **Section 8**, **Risk and Reliability** were repeated for the final set of mission modes and technology options. Three mission modes were analyzed, with three different propulsion technologies applied. In addition to the LOR, EOR–LOR 2-launch, and EOR–LOR 1.5-launch modes, analysis was also performed on a single-launch mission that launched both the CEV and lander atop a single heavy-lift CaLV (the same used for the 1.5-launch solution), much like the Apollo/Saturn V configuration. However, the limited lift capability provided by this approach limited its utility, and it was not examined further. For each of the mission modes, end-to-end single-mission probabilities of LOC and LOM were calculated for (1) a baseline propulsive case using all pressure-fed LOX/methane engines, (2) a case where a LOX/hydrogen pump-fed engine was substituted on the lander descent stage, and (3) a third case where the lander ascent stage engine was changed to pump-fed LOX/methane. **Figures 4-47** and **4-48** illustrate the P(LOC) and P(LOM) for each of these cases.

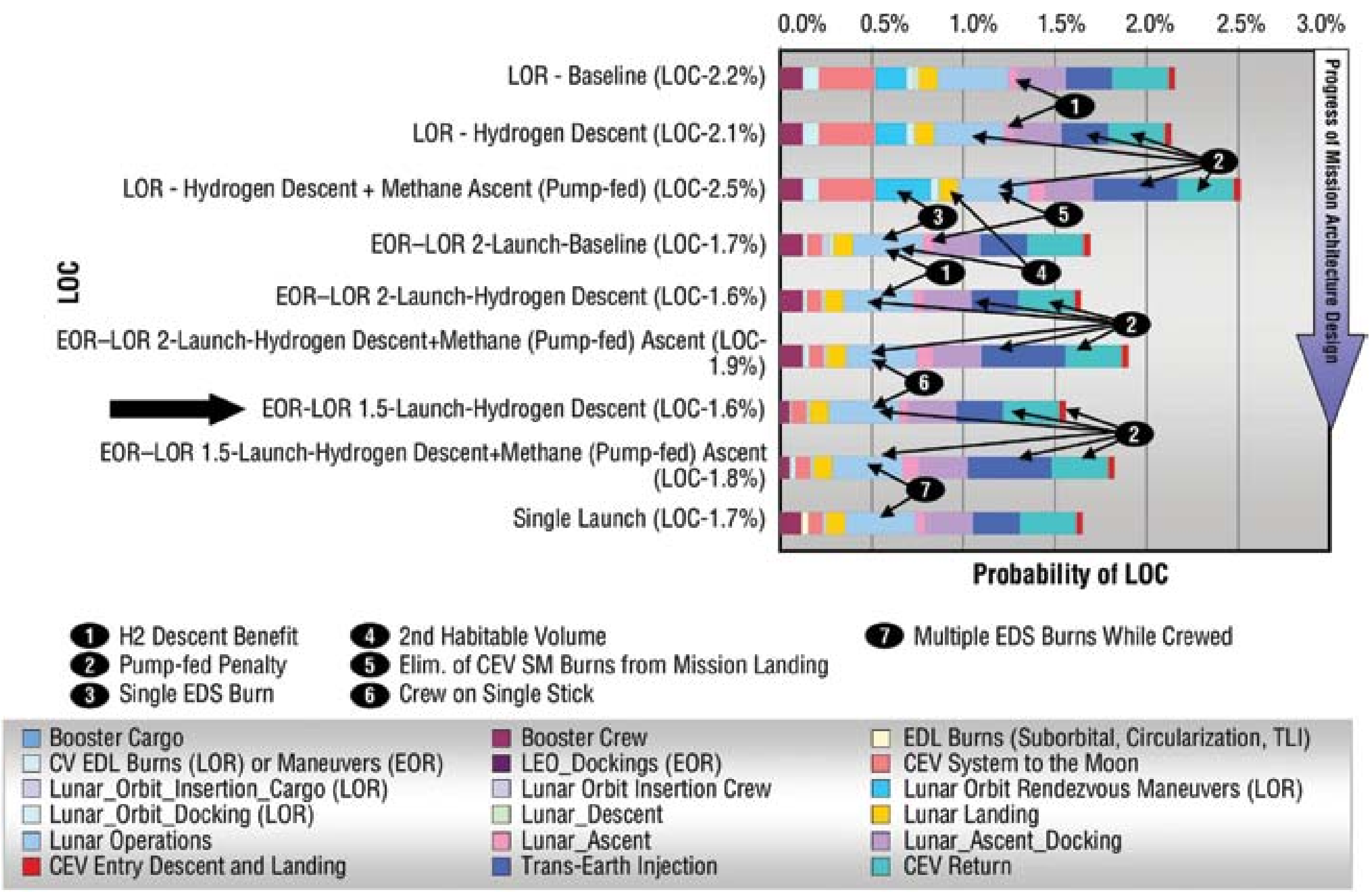


Figure 4-47. LOC Comparison

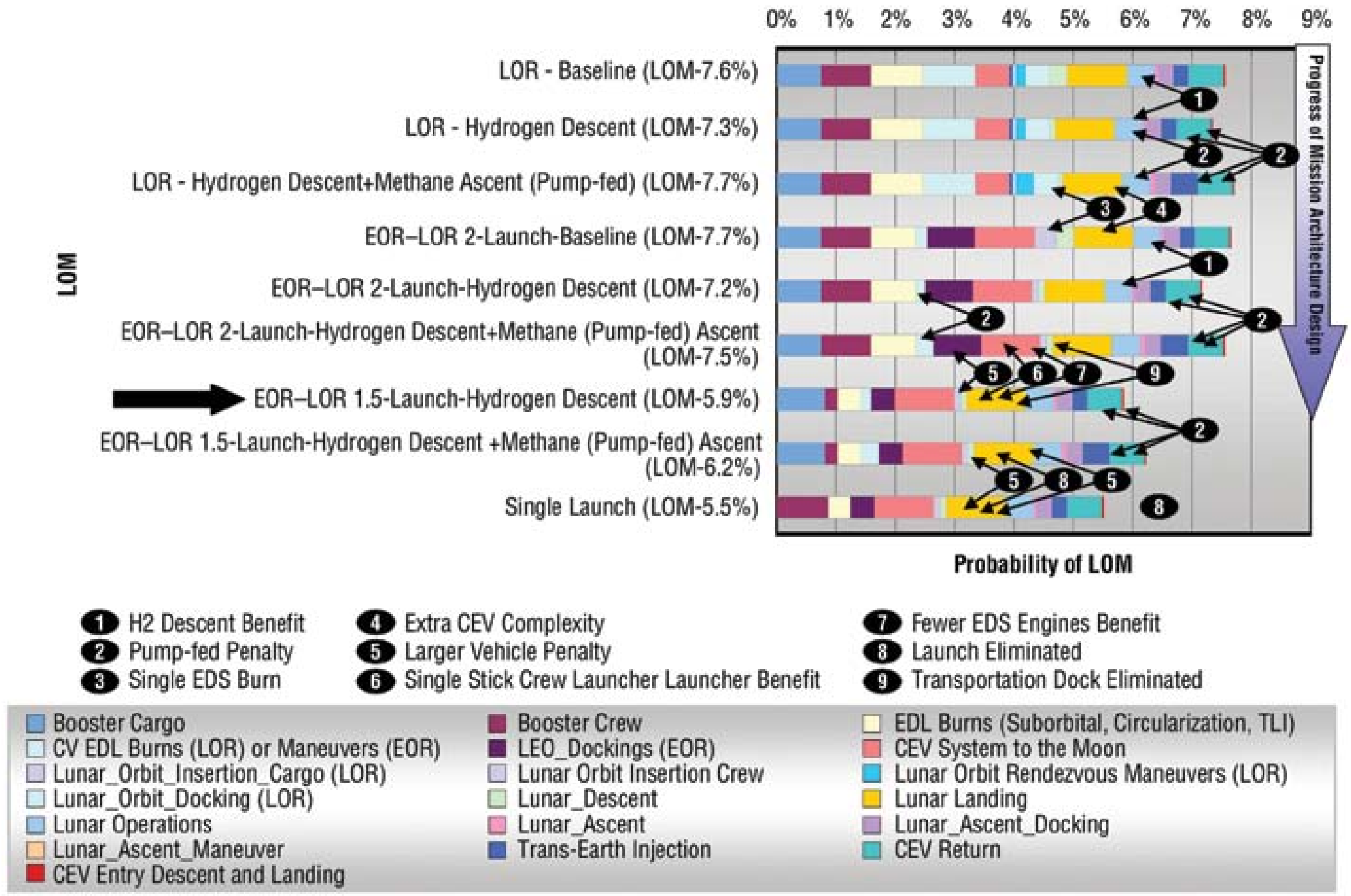


Figure 4-48. LOM Comparison

P(LOC) was dominated by propulsive events and vehicle operating lifetimes. As shown in **Figure 4-47**, LVs varied only slightly between the 2-launch (crew launched on a heavy-lift booster) and 1.5-launch (crew launched on a single SRB CLV) options. The LOR options had added risk due to the lander being sent to the lunar orbit separately from the CEV, and thus not having a back-up crew volume during transit to handle "Apollo 13"-like contingencies. The LOR mission also required the CEV SM to perform an LOI maneuver. Generally, each time a pump-fed engine technology was introduced to replace a pressure-fed system, risk increased, although the LOX/hydrogen engine modeled for the lander descent stage had a high degree of heritage from existing RL–10 engine technology.

When all the mission event probabilities were summed, all mission options fell within a relatively narrow range (1.6 to 2.5 percent), but the difference between the highest- and lowest-risk options approached a factor of two. Missions using the LOR mission mode were the highest risk options, while EOR–LOR 1.5-launch options were the lowest. Missions that utilized a higher-performing LOX/hydrogen lander descent stage scored approximately the same as the baseline option that used pressure-fed LOX/methane, but a change to a pump-fed LOX/methane ascent stage resulted in an appreciable increase in risk. The single-launch option, with its single LV, shorter propulsive segments, and limited surface access ability, was grouped with the lower LOC options. The lowest probability of LOC option was the 1.5-launch EOR–LOR mission using a pump-fed LOX/hydrogen lander descent stage and pressure-fed LOX/methane engines for both the lander ascent stage and CEV SM.

P(LOM) generally followed the same trends as P(LOC). **Figure 4-48** illustrates the reliability benefits of launching crew on the single-SRB CLV, the reduced risk of having a single EDS stage, and the penalties associated with pump-fed engines. LOR and EOR–LOR 2-launch options exhibited the greatest P(LOM), in a range between 7 and 8 percent per mission. The substitution of a LOX/hydrogen lander descent stage engine actually increased mission reliability by adding engine-out performance to the LOI and lunar landing phases of the mission, but further pushing LOX/methane engine technology toward a pump-fed system lowered reliability by eliminating commonality with the CEV SM engine and adding complexity.

The single-launch mission option scored the highest reliability overall, owing mainly to it requiring only a single launch. Of the missions that provide the full lunar landing site access and return capabilities, EOR–LOR 1.5-launch modes were nearly competitive with the single-launch option. Specifically, the EOR–LOR 1.5-launch option using the LOX/hydrogen lander descent stage engines scored the lowest P(LOM) among the full-up mission options. Interestingly, this same mission mode and propulsion technology combination scored the lowest P(LOC) as well.

4.2.5.3.2 Mission Mode Cost Comparison

Figure 4-49 summarizes the LCCs described in detail in **Section 12**, **Cost**. To enable a fair comparison among the options, the complete LCCs, including DDT&E, flight units, operations, technology development, robotic precursors, and facilities, were all included in this analysis. Generally, the choice of mission mode had only a small effect on the LCCs of the exploration program. Of the options modeled, the 1.5-launch EOR–LOR mission using a LOX/hydrogen lander descent stage propulsion system exhibited an LCC that was a few percent less than other options.

4.2.5.3.3 Other Figures of Merit

Analysis Cycle 3 focused primarily on the performance improvements possible for each of the three mission modes and the associated cost, safety, and reliability changes that accompanied these technology or design changes. Fundamentally, the mission modes themselves were not changed during this analysis cycle, so other FOMs such as flexibility, extensibility, effectiveness, performance, or operability did not change from those established in Analysis Cycle 2. For these FOMs, refer to **Tables 4-14** and **4-15** in **Section 4.2.4.3**, **Figures of Merit**.

4.2.5.4 Summary of Analysis Cycle 3 Mission Mode Results

Analysis Cycle 3 was structured to refine the fidelity of spacecraft system descriptions and mass models, investigate technology options that could optimize different modes, and produce performance, risk, and cost data that could be used in combination to recommend a preferred lunar architecture mission mode.

Refinement of the CEV mass model began with an improved understanding of radiation shielding requirements. Based on a revised spacecraft cross-section featuring a composite outer skin, probabilistic event and dose calculations were performed. Based on the short duration of the CEV lunar mission and the leveling of risks to the crew, the ESAS team recommended that the inherent design of the CEV could protect the crew to a degree commensurate with other mission risks and that no supplemental radiation protection was required for the CEV. The reduction of supplemental radiation protection created considerable spacecraft mass margin in each of the mission modes.

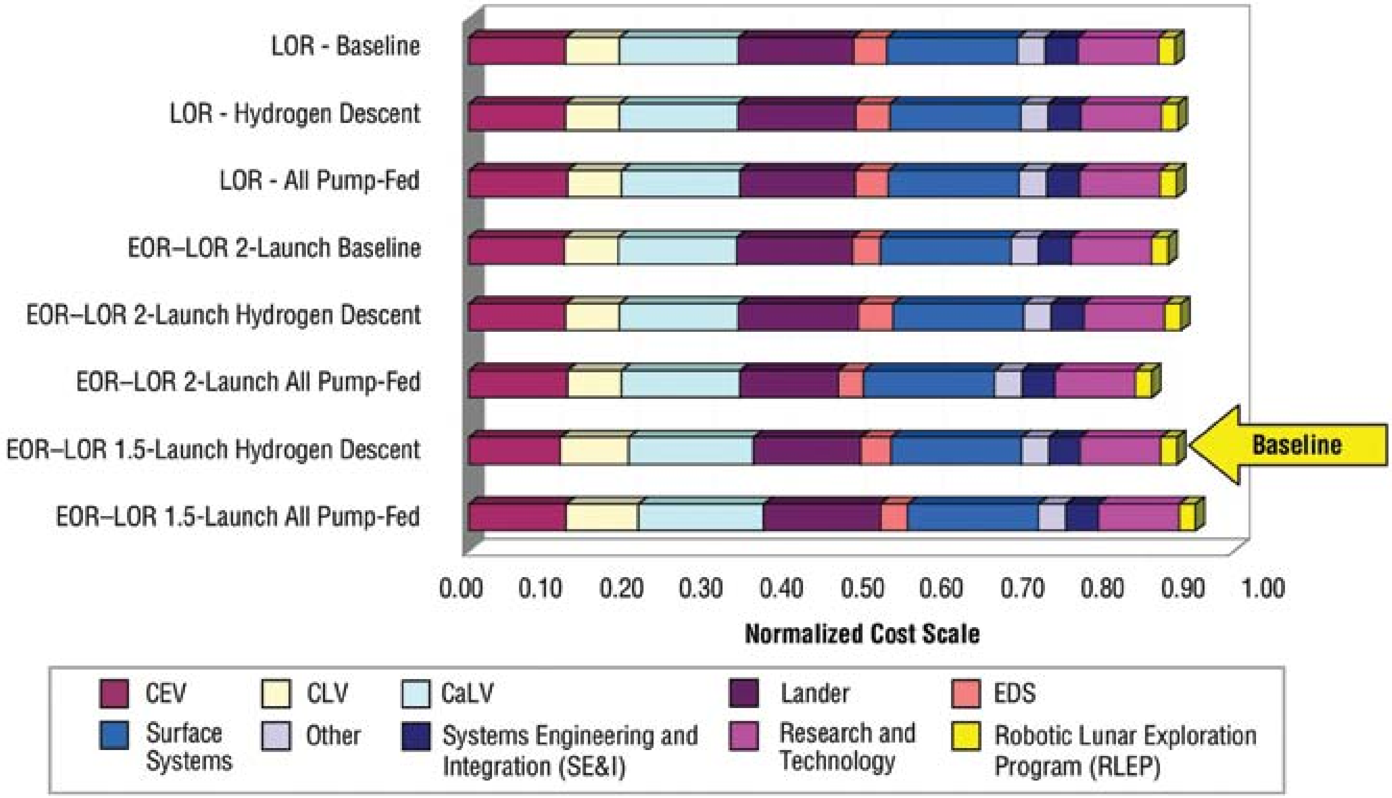


Figure 4-49. Mission Mode LCCs Through 2025

The requirement to return anytime from the surface of the Moon to Earth was the design driver of the SM propulsion system. The CEV SM is common across all of its missions and is sized by the lunar mission application. The lunar mission requires a total of 1,450 m/sec of delta-V, combining a 900 m/sec TEI maneuver and a worst-case 90-deg nodal plane change. This capability enables "anytime return" if the lander is able to perform a coplanar ascent to the CEV. For sortie duration missions of 7 days or less, the CEV's orbital inclination and node will be chosen to enable "anytime return" from any location on the lunar surface. Outpost missions will also have the ability to return anytime the outpost is located at a polar or equatorial site. For other sites, loitering on the surface at the outpost for up to 14 days may be required to enable a coplanar ascent to the orbiting CEV.

Landing at any site on the Moon sizes the magnitude of the LOI maneuver. A nominal 900 m/sec LOI burn enables access to the equator and poles. Additional delta-V is required for nodal plane changes to access other sites up to a maximum of 1,313 m/sec for immediate access to any site on the lunar globe. Another technique that can be used to access any landing site is to loiter in orbit or to use a combination of nodal plane change and orbital loiter. The team ultimately chose the latter combination to balance additional propulsive requirements on the lander descent stage and additional loiter lifetime of the CEV systems. The lander descent stage was sized for a 900 m/sec LOI plus a 200 m/sec maximum nodal plane change, for a total of 1,100 m/sec in addition to lunar descent propulsion. This value allows the system to immediately access all but a small percentage of lunar surface, and those high-latitude, limb sites are accessible with no more than 3 days of post-LOI loiter prior to descent. Limiting LOI to 1,100 m/sec establishes comfortable mass margins for all mission modes.

LOM and LOC probabilities were assessed for each of the mission modes, and P(LOM) and P(LOC) were generated within the mission modes for three discrete lander propulsion options. Of the options studied, the EOR–LOR mode yielded both the lowest P(LOM) and the lowest P(LOC) when flown with a LOX/hydrogen lander descent stage and common pressure-fed LOX/methane propulsion system for both the lander ascent stage and CEV SM. Cost analysis was less definitive, but also showed this same EOR–LOR 1.5-launch option having the lowest cost of all the alternatives studied.

Based on the convergence of robust technical performance, low P(LOC), low P(LOM), and LCCs, the 1.5-launch EOR–LOR using LOX/hydrogen lander descent stage propulsion was selected as the mission mode to return crews to the Moon.

4.2.5.5 Architecture Findings and Recommendations

Based on the analyses performed by the ESAS team, a comprehensive evolutionary architecture was constructed that can successfully perform near-term ISS crew and cargo delivery missions, human missions to the lunar surface, and farther-term human missions to Mars and beyond. The key features that enable the architecture to evolve over time are the design of the CEV, the choice of CLV and CaLV, the selection of technologies (particularly propulsion technologies), and the operations procedures and systems that extend across the destinations. Architecture linkages are shown in **Figure 4-50**.

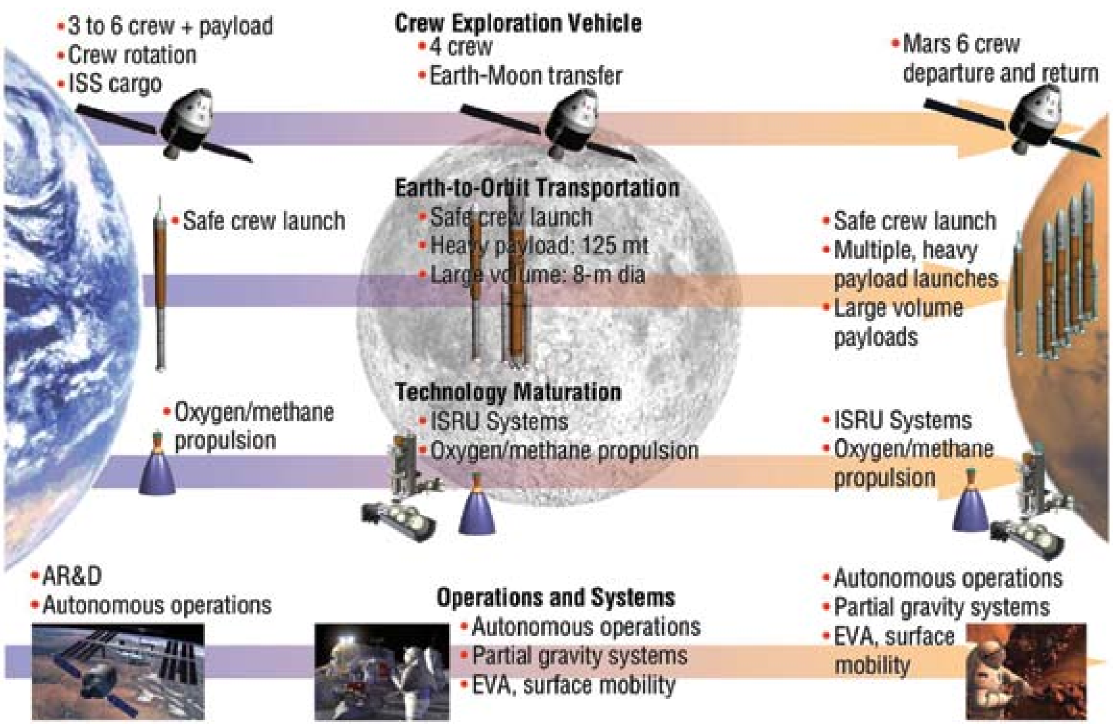


Figure 4-50. ISS→ Moon → Mars Architecture Linkages

The CEV CM was selected to be a 5.5-m diameter blunt-body capsule with a 32.5-deg side-wall angle to retain maximum commonality with Apollo aerothermal databases. The CEV SM will use pressure-fed LOX/methane propulsion that will be common with the lunar lander ascent stage propulsion. The SM will be common for all CEV missions and will be sized for lunar mission TEI (1,450 m/sec delta-V).

The CLV will be derived primarily from components of the Space Shuttle system. The first stage of the CLV will use a four-segment, RSRB. The second stage will be a new design based on a single Space Shuttle Main Engine (SSME), modified for altitude start. The CEV will be fitted with a Launch Escape System (LES) that will provide full-envelope abort capability.

The CaLV will also be based on Shuttle-derived components. It will use twin five-segment RSRBs on either side of an External Tank- (ET-) derived core. The core stage will be powered by five SSME Block II engines. Atop the core stage is an upper stage that burns suborbitally during launch and then serves as the EDS for injecting the CEV and lander to the Moon. The EDS is powered by two J–2S, or equivalent, LOX/hydrogen engines.

The lunar mission will be conducted using a combination of EOR–LOR. A single launch of the CaLV will place the lunar lander and EDS in Earth orbit. The launch of a CLV will follow and place the CEV and crew in Earth orbit, where the CEV and lander/EDS will rendezvous. The combination of the large cargo launch and the single SRM CLV is termed a 1.5-launch EOR–LOR mission. The EDS will then inject the stack on a trans-lunar trajectory and be expended. The lander and CEV are captured into lunar orbit by the descent stage of the two-stage lander, and all four crew members descend to the surface, leaving the CEV operating autonomously in orbit. The two-stage lander uses LOX/hydrogen propulsion for its descent stage and pressure-fed LOX/methane propulsion for ascent. The lander features an airlock and the capability to support up to a 7-day surface sortie. Following the lunar surface mission, the lander's ascent stage returns to lunar orbit and docks with the waiting CEV. The crew transfers back to the CEV and departs the Moon using the CEV SM propulsion system. The CEV then performs a direct-Earth-entry and parachutes to a land landing on the west coast of the United States. The mission is illustrated schematically in **Figure 4-51**.

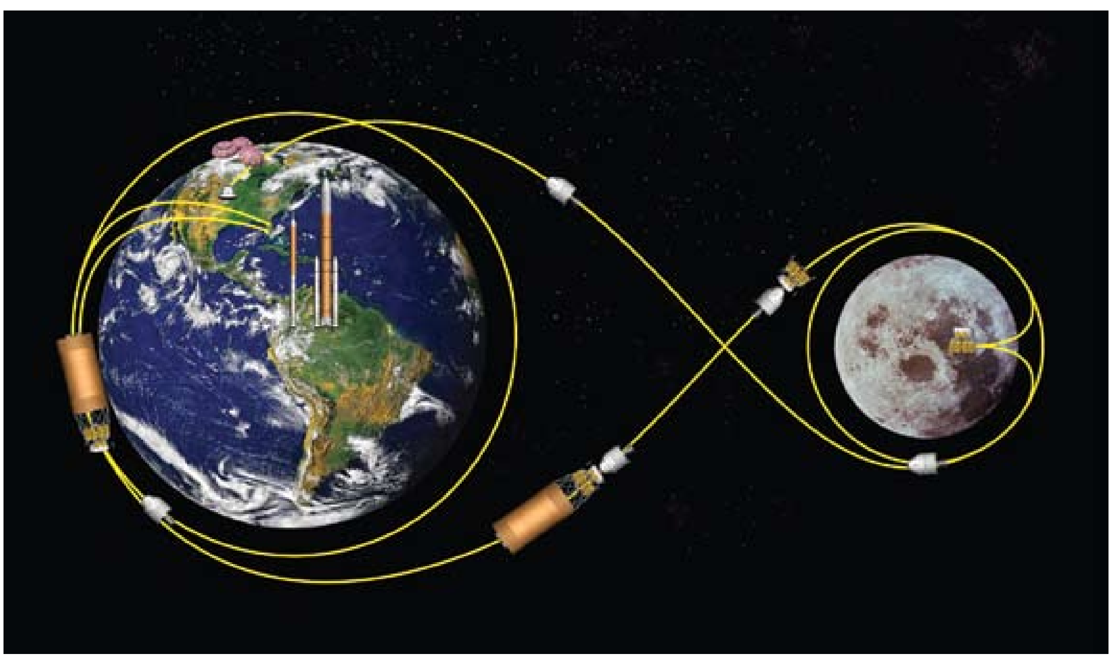


Figure 4-51. EOR–LOR “1.5-launch” Lunar Mission Architecture

4.3 Lunar Surface Activities

In the space mission design process, it is imperative to define the destination activities in order to design the systems necessary to allow those activities to occur. On January 14, 2004, the President provided a strategic vision of space exploration beyond Earth orbit, emphasizing the importance of using the Moon's natural resources to establish an extended human presence on the lunar surface and using the time spent on the surface to develop and test new approaches, technologies, and systems that will allow the further exploration of Mars and other destinations in the solar system. Gaining a better understanding of the Moon and its environment is critical to successfully accomplishing the President's goals. Appropriately, the ESAS team chose to focus lunar surface activities on (1) exploration science, (2) development of lunar resources, and (3) Mars-forward testing of operational techniques and surface system technologies.

4.3.1 Exploration Science

Human return to the Moon both enables and is enabled by science. Gaining a better understanding of the Moon and its environment requires science to be an integral part of the lunar architecture. Moreover, by virtue of the destinations and voyages outlined by the exploration vision, opportunities exist for new and exciting scientific exploration. The return to the Moon enables the study of the Moon as a planetary object, the use of the Moon as a platform for unique scientific observations, and the study of biological systems in an isolated fractional-gravity environment.

Scientific themes for lunar exploration have been developed by the NASA-chartered Lunar Exploration and Analysis Group (LEAG) (**Appendix 4D**). These scientific investigations can be integrated into every phase of the lunar return architecture, including precursor robotic missions, human sorties, and outpost operations. In fact, these phases will be informed and enabled by the scientific investigations undertaken.

Exploration science can be broadly divided into three categories: (1) planetary science of the Moon and Earth-Moon system, (2) use of the Moon and its environment as an observing platform and natural laboratory, and (3) applied science of resource extraction and outpost engineering.

Planetary science uses the natural history recorded in the lunar surface and crust to understand planetary geological processes such as impact, volcanism, and thermal history For example, **Figure 4-52** shows a lunar crew acquiring subsurface samples near the center of a small complex crater through the use of deep drilling equipment to better understand the geologic process of impact cratering. The Moon contains a record of active planetary events between 4.6 and 3 billion years ago. Since then, the Moon has recorded the impact flux in the vicinity of the Earth—a record that can be recovered and read for clues to the impact disruption of the Earth's climate and life suggested from the fragmentary and incomplete terrestrial record. The Moon is also a natural laboratory for processes (especially biological) that may operate differently in a vacuum, fractional gravity, and non-radiation shielded environment. Studies of cell growth and evolution in 1/6 gravity may yield new insights into biological processes in general. An understanding of the long-term effects of space and planetary environments on human physiology and psychology is necessary in order to explore beyond the Moon.

Figure 4-52. Lunar Crew Acquiring Subsurface Samples

The Moon also offers a superb and unique platform to observe the Earth, solar system, and universe at a variety of wavelengths and viewing conditions. The Moon's quiet stable base permits construction of sensitive systems of instruments such as interferometers, and the far side offers a place shielded from interference of the Earth and (occasionally) the Sun. This unique environment permits otherwise unobtainable observations.

One of the key features of the exploration vision is the use of lunar and space resources to create new capability. Science is essential to this endeavor. NASA needs to understand the distribution and state of lunar resources to enable their harvesting and use. Achieving this understanding involves mapping the deposits from orbit, examining and surveying them on the ground, and experimenting with various processes and procedures for their extraction. Robotic precursor missions will investigate the polar regions to map volatiles in the permanently shadowed areas and determine the environment of the poles to identify optimum potential outpost sites. Mining sites must be assayed and prospected to determine the best feedstock locations, and various resource extraction processes must be evaluated to optimize production. The scientific investigations that support these activities will also reveal new aspects of the Moon, thus advancing lunar science as well as supporting the human activities. For these reasons, scientific measurements must be considered during each phase of the human lunar return.

4.3.2 Scientific Themes for Human Lunar Return

These themes include the following:

- Bombardment of the Earth-Moon system:
  - Measure the recorded bombardment history of the inner Solar System;
  - Understand the duration and intensity of "late-heavy bombardment" of early Earth and other planets; and
  - Understand the episodicity of later impactor flux and impactor-induced mass extinctions.
- Lunar processes and history:
  - Determine the composition and structure of the interior of the Moon to understand lunar origin and evolution.
- Scientific resources in the permanently shadowed polar environment:
  - Investigate the unexplored shadowed environment (unlike the equatorial Moon) which is similar to conditions in interstellar space and Oort cloud (silicate grains, cosmic rays Ultra-Violet (UV) radiation, and temperature fluctuations); and

  - Utilize the polar environment as a natural laboratory for understanding environments not otherwise accessible.
- Regolith as a recorder of the Sun's history:
  - Read the 4-billion year record of the sun and galaxy recorded in the lunar regolith; and
  - Investigate the Apollo data that hints at nuclear processes in the Sun not predicted by current models of stellar evolution.
- Biomedicine:
  - Determine fundamental mechanisms causing genomic damage;
  - Establish synergy between lunar expedition and terrestrial biomedical advances; and
  - Develop novel approaches to study and address Earth-based pathogenesis and environmental health hazards.
- Using the Moon's resources:
  - Develop fundamental advances in science associated with resource extraction; and
  - Enhance human exploration capabilities on the Moon, cislunar space, and beyond.
- Astronomy:
  - Observe ultra-deep field; observe first stars to form in the universe beyond the capabilities of the James Webb Space Telescope (JWST); and
  - Utilize the lunar far side; low-frequency radio astronomy opens up new wavelength range.

4.3.3 Lunar Resource Development

Learning how to use the resources of space (materials and energy) is a key feature of the exploration vision. Incorporating the use of these resources into space exploration architectures will make space flight more affordable and expansive. The Moon contains known resources, particularly oxygen and hydrogen, that can be harvested and used. Producing fuel and life-support materials on the Moon will permit the lunar outpost to achieve some measure of self-sufficiency. The skills needed to use these resources will be indispensable as humans travel beyond the Earth-Moon system.

Lunar resources can be classified into three broad categories: (1) materials requiring minimal processing, (2) materials requiring some processing, and (3) energy. In the case of the former, bulk regolith (i.e., the soil that blankets the lunar surface) is a significant resource. It can be used to provide building materials and radiation cover for habitats on the Moon. Regolith can be microwaved into glass, a technique useful for creating pavement for roads and landing areas. Regolith can also be bulldozed into berms to provide blast shielding for landing pads and protect sensitive instruments from flying dust created by human and machine activity. Experiments must be conducted using this bulk regolith to determine its geotechnical properties and ease of handling. However, the regolith is also abrasive and must be studied to learn strategies to mitigate its possible harmful effects to moving machine parts and human ingestion.

Lunar regolith can also be processed to yield a variety of products, the most important of which is oxygen, which makes up at least 40 percent (by weight) of the lunar soil. Although tightly bound chemically, this oxygen can be broken by a variety of chemical reduction processes. Early flight experiments can prove and demonstrate some of these techniques and allow for selection of one technique for general oxygen production. As oxygen is 80 percent of the mass of LOX/hydrogen propulsion systems, the production of lunar LOX would significantly benefit Earth-Moon transportation. Moreover, hydrogen is also present in the lunar regolith as absorbed solar wind gas on the grains of the regolith. This hydrogen can be released if the soil is heated to 700°C. Production of hydrogen by this method implies a significant effort, as such hydrogen is present at very low levels of concentration (roughly 50–100 ppm in the richest deposits).

As a result of robotic missions flown in the 1990s, NASA discovered that the poles of the Moon may contain concentrated amounts of water ice. Confirmation of these deposits, however, awaits the flight of the next round of U.S. and international lunar robotic explorers that will make detailed measurements of the polar environment and deposits. If water ice can be found at the poles in useable quantities, mission flexibility could increase dramatically. Water ice is the most concentrated form of hydrogen and oxygen, and its recovery requires two orders of magnitude less energy than that required to extract the solar wind hydrogen from normal regolith. (See **Table 4-26**.) Thus, in the robotic precursor mission series, emphasis should be placed on early surveys of the poles to confirm or negate the existence of water ice. If water ice is present, a robotic mission should land near the poles and explore these areas in detail on the ground to confirm its presence and characterize the deposits. After such mapping, it will be necessary to experiment with techniques to extract the water ice from the regolith. Such demonstration experiments can be relatively small-scale and conducted on landed robotic missions prior to human arrival. If successful, larger-scale production could proceed at a lunar outpost. (See **Figure 4-53**.)

After the processes described above have been established, consideration should be given to a variety of derived products, including: the metals iron and aluminum (byproducts of oxygen reduction), carbon, nitrogen and sulfur (byproducts of regolith heating), anhydrous glass, and other metals and substances (including platinum group metals and meteoritic components). However, the recovery of these materials should only be considered after the establishment of processes to extract the most important resources.

Energy is abundant at the Moon in the form of incident solar radiation. For most of the Moon, this illumination follows a 28-day diurnal cycle. However, locations that are illuminated for much longer periods of time have been found near the poles (possibly permanently near the north pole). These areas are on the order of a few hundred meters to several kilometers in extent, and solar arrays set up in these zones could provide most of the power required by a lunar outpost. Moreover, such areas always receive grazing incidence angle illumination, making them thermally benign and of nearly constant temperature (approximately 50°C ± 10°C). Particular attention should be paid early in the robotic precursor missions to characterizing these areas and documenting their physical and thermal properties.

Operation	Specific Energy
Excavation of regolith	.01 kWh/kg regolith (electric)
Extraction of water from icy regolith*	2.8 kWh/kg H2O (thermal)
Reduction of SiO2 to Si + O2	5.2 kWh/kg SiO2 (electric)
Electrolysis of water	4.5 kWh/kg H2O (electric)
Extraction of hydrogen from typical regolith**	2,250 kWh/kg H2 (thermal) (~250 kWh/kg H2O)

Assumes 1% ice, heated 100°C above ambient

***Assumes 100 ppm H2, heated 800°C above ambient*

Table 4-26. Lunar Resource Extraction Energy Requirements

Figure 4-53. Lunar Crew at a Permanently Shadowed Crater Near the Lunar South Pole

4.3.4 Mars-forward Testing

Although the Moon and Mars are two very different planetary environments, the operational techniques and exploration systems needed to work and live on both surfaces will have similar strategies and functions. While it is not likely that exact "copies" of Mars-bound systems will be operated or tested on the Moon, it is very likely that components and technologies within those systems ultimately destined for the surface of Mars will undoubtedly find their heritage based on lunar surface operations. Therefore, the philosophy is to do what is proper and required for the lunar environment and use the knowledge, experience, and confidence gained from lunar operations as the foundation for the design of surface systems for Mars and other destinations in the solar system.

Two important operational techniques that should be developed on the Moon are crew-centered control of surface activities and teleoperation of robotic explorers from a central planetary outpost. Mars is very distant from the Earth-Moon system, and due to the speed of light, one-way communications with Mars can take up to approximately 20 minutes. Communications with a human crew on Mars will be hindered by this time delay, and the crew will need to be able to operate in an autonomous mode without constant supervision from Earth. As this is different from the way human space missions have been conducted to-date, the Moon will provide the opportunity to transition from Earth-centered to crew-centered control of daily operations. Likewise, the Mars Exploration Rover (MER) missions, though very successful, have shown the limitations of controlling robotic explorers on Mars from the Earth. Mars and the Moon are too big for a small human crew to explore effectively by themselves, and robotic systems teleoperated by the crew can be used to increase the efficiency of surface activities. Operations on the Moon will develop the synergism of cooperative activity between humans and robots that can be used on planetary surfaces throughout the solar system.

Regardless of which planetary surface human crews are living and working on, similar supporting infrastructure will be necessary. Habitation, power generation, surface mobility (i.e., space suits and roving vehicles), surface communication and navigation, dust mitigation, and planetary protection systems are all common features that will first be provided on the Moon. Repetitive and long-term use of these systems at a lunar outpost will allow the design of these systems and their components to be refined and improved in reliability and maintainability—system traits that will be essential for the exploration of destinations much more distant from Earth. The presence of an atmosphere and a stronger gravitational field on Mars, however, will require modifications to some components of the planetary surface systems used on the Moon, particularly those systems that directly contact the Martian surface or atmosphere. Lunar surface systems that are internal to a pressurized environment, such as a habitat's regenerative life support system, are more likely to be directly applicable to Mars with fewer or no modifications.

4.3.5 Lunar Surface Traffic Model

Lunar architecture capabilities are driven in large part by the duration, location, and centralization of lunar surface activities. The magnitude of surface activities can be represented linearly, coarsely corresponding to a range of "simple" to "complex." The continuum adopted by the ESAS team is shown in **Figure 4-54**. This graphical scale was used to illustrate a multitude of variables and to indicate the design points chosen for the study. The variables represented by this scale include the following:

- Number of sites to be visited (1 → many);
- Location of these sites (constrained latitude/longitude bands → global access);
- Duration of surface activities (approximate week-long sorties → permanently inhabited outpost);
- Centralization of assets (Apollo-class sorties with local mobility → mobile camp with predeployed logistics caches → single outpost w/ regional mobility); and
- Required infrastructure (power, communication, habitation, mobility, resource utilization, and science).

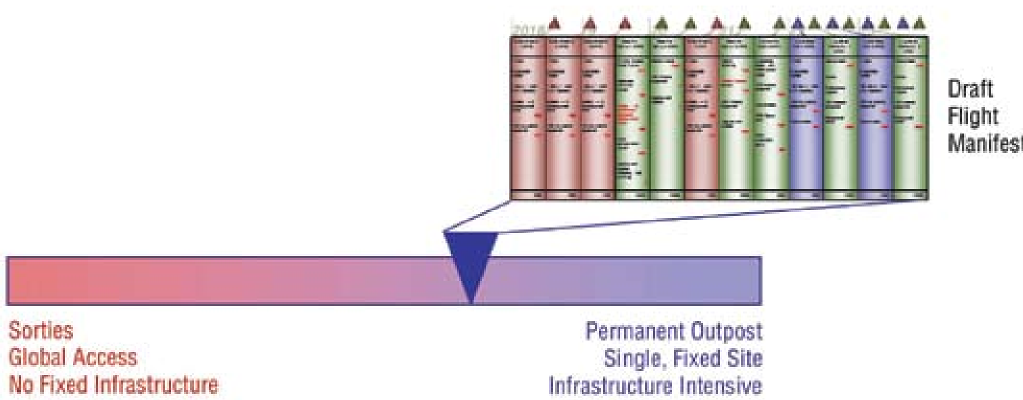


Figure 4-54. Lunar Surface Continuum

An initial strategy was chosen that starts with global-access, short-duration sortie missions beginning in 2018 and transitions quickly to deployment of a permanent outpost that is at an IOC by 2022. This strategy was chosen to enable early missions to test transportation systems and to allow short scientific sorties to a small number of diverse sites and extended development time lines for high-cost outpost systems. It was recognized that this initial strategy was a singular point in the multidimensional duration/location/centralization trade space and would later be modified as cost, risk, and performance of the system was better understood.

The location of the triangle in **Figure 4-54** represents the initial outpost deployment strategy discussed in **Section 4.2.5.1.4**, **Outpost Deployment Strategies**, which is defined by large, dedicated cargo missions that deploy an outpost in a small number of missions. The second deployment option discussed in **Section 4.2.5.1.4**, **Outpost Deployment Strategies**, would plot as another discreet pointer in **Figure 4-54**. This second point along the continuum would represent an outpost deployment strategy that has a greater reliance on sortie missions, but which would also be focused toward a single, revisited site. **Figure 4-54** graphically illustrates that lunar exploration, including the deployment of a lunar outpost, can be accomplished in a great number of ways.

This study has thus far discussed two points on the outpost deployment continuum: (1) dedicated large cargo missions and (2) incremental build using subsequent sortie missions. The ESAS team concluded that other points along the surface activity continuum should be analyzed as well. The initial results of this additional analysis are presented in **Section 4.2.5.1.4**, **Outpost Deployment Strategies**, and **Section 4.3.8**, **Outpost Deployment Studies**. Options can be found in **Appendix 4F**, **Alternate Outpost Deployment Options**.

4.3.6 Landing Sites

Depending on the principal purpose of the mission, there are many possibilities for landing sites on the Moon. Previous studies on future landing sites have emphasized their value to lunar science or for some specialized purpose (e.g., far side telescope installations). The ESAS team considered the many potential requirements for lunar landing site selection and compiled a list of sites on the Moon that illustrates the diversity of scientific and resource opportunities, geographic position, operational considerations, and usefulness. A roster of sites was compiled to explore the various trade spaces needed to understand architectural requirements.

4.3.6.1 Site Selection Consideration

Sites for human missions to the Moon may be selected on the basis of operational, scientific, resource potential, and programmatic considerations. In general, sites that offer many different features to a wide variety of interests are preferred. The more geological diversity a site offers for science, the more attractive it is for exploration. For the extraction of resources, the highest grade “ore bodies” are desired to maximize the product for the minimal investment of time and energy. Fortunately, the Moon is diverse and complicated enough on the scales of human operations that many sites exist that satisfy the requirements of many different and diverse users.

The idea that water ice may exist at the poles of the Moon has gained currency in the last decade as a result of two robotic missions (Clementine and Lunar Prospector) that found evidence for enhanced volatile concentrations associated with the poles. If such water ice exists, the extraction of water from this lunar resource requires at least two orders of magnitude less energy than does the synthesis of water from the hydrogen autoreduction of solar wind gas-saturated regolith. Thus, the polar deposits qualify as “high-grade ore.” Moreover, there is evidence that certain small areas near the poles may be in near-constant sunlight, providing sites that have access to continuous solar power and are also thermally benign.

Ten sites have been selected as examples to explore the ramifications of the site selection trade space. Listing these sites should in no way be construed as an advocacy or an endorsement of any given site as the “favored” site for the lunar outpost. At an appropriate time, NASA will create a process designed to obtain the best possible information on the requirements and needs of the various lunar mission stakeholders in order to pick a landing site that satisfies the most user requirements.

4.3.6.2 Classification of Landing Sites

While sites on the Moon may be classified in a variety of ways, sites were categorized by position (e.g., equatorial, mid-latitude, limb, and polar) for the purpose of the ESAS work to understand the operational difficulties of accessing them and by the similarity or dissimilarity to previously visited Apollo sites (to assess requirements for precursor site knowledge). Ten sites that span the range of these properties were identified, all of which offer significant scientific and operational features that make them worthy of consideration for both sortie and outpost missions to the Moon. (See **Figure 4-55**.)

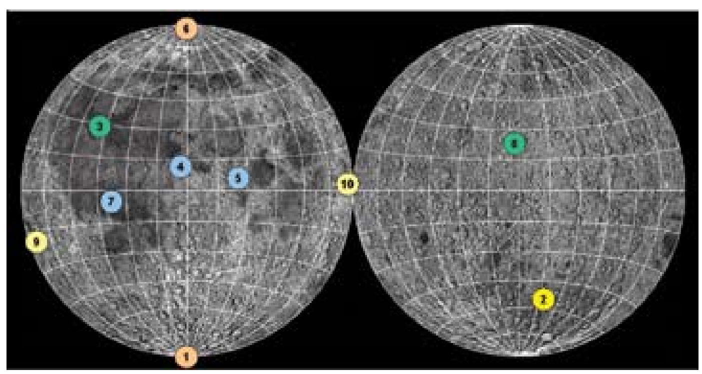


Figure 4-55. Lunar Sites Selected to Explore Architectural Trades

The equatorial sites selected (defined here as those within 30 deg of the equator) are, with one exception, on the near side. They are similar enough in geological age, setting, and physical properties that they can safely be assumed to be very similar to the sites visited by Apollo. This means that surface dust, slope and block distribution, gross topography, and other properties are already fairly well known, at least to the level of being able to successfully plan and conduct a human surface mission. Thus, no robotic precursor missions are required before human visits to such sites. (See **Table 4-27**.)

Sites	Navigation/ Communication	Precision Topography and Local Terrain	Surface Deposit Characterization	Site Environment
Equatorial and low latitude sites	No	Probably not	No	No
Limb sites	Yes	Yes	No	No
Polar sites	Yes	Yes	Yes	Yes

Table 4-27. Classification of Required Precursor Information as a Function of Site Geographical Position

Sites on the eastern and western limbs of the Moon are likewise equatorial and anticipated to have surface physical characteristics similar to previously visited Apollo sites. However, their location on the lunar limbs indicates that these sites have Earth visibility issues due to the longitudinal libration of the Moon. Although these sites can be grossly modeled, detailed information on the exact times and durations of Earth visibility requires detailed site topographic information (data that does not currently exist). Thus, these sites require the acquisition of new, high-resolution topographic data. Such data should be acquired by the forthcoming Lunar Reconnaissance Orbiter (LRO) mission.

Sites on the far side require a relay infrastructure to ensure adequate communications with the Earth, but are otherwise assumed to be similar in physical properties to near-side sites. An exception to this is the absence of Earth light, which is a significant resource for possible night operations on the lunar near side. A single site (South Pole-Aitken basin) is at high latitudes. This site is included not only for its scientific value (e.g., a site on the oldest basin terrain on the Moon), but also to explore the operational trade space of the mission mode decision.

The poles of the Moon are a unique environment. Because the lunar spin axis is essentially normal to the ecliptic (axial tilt 1.7 deg), the Sun is always near the horizon at the poles. This simple relation has profound consequences—not only does it create the permanently shadowed cold traps that may contain water ice, but high peaks and terrain elements near the poles may be in near-constant solar illumination. Analysis of Clementine images show zones of potential continuous light at the lunar north pole and several areas of near-constant illumination (greater than 75 percent of day) near the lunar south pole. Such areas have great value for an outpost site. Not only do they provide a place of near-constant solar illumination (allowing for reliance on solar-voltaic power systems), they are also thermally benign (typically, approximately –50°C ± 10°C) due to the illumination always occurring at grazing angles of incidence. These properties, in addition to their proximity to cold areas containing water ice, make the poles attractive landing site candidates.

However, knowledge of the environment and deposits of the polar regions is extremely limited. The true extent of both lit and dark regions is not known. Temperature estimates of the cold traps are modeled, not observed. The physical and chemical nature of polar volatiles and the regolith containing them, as well as their detailed distribution and state, are unknown. This information must be collected prior to human arrival if use of the unique polar attributes is to be a major mission goal. Thus, the polar sites require the most robotic precursor information (**Table 4-27**), including mapping these relatively unknown areas in detail and characterizing the polar environment and their deposits. A further discussion of precursors to human lunar missions is presented in **Appendix 4I**, **Lunar Robotic Precursor Missions**.

4.3.6.3 Brief Description of the Sample Landing Sites

The following section briefly notes the characteristics, advantages, and drawbacks of the landing sites identified in this study.

- South pole (rim of Shackleton, 89.9°S, 180°W): This area of near-permanent sunlight on the rim provides access to power and proximity to a cold trap (crater interior) that may contain water ice. The site is on the floor of the South Pole-Aitken (SPA) basin, the oldest and biggest impact feature on the Moon. The southern celestial hemisphere is continuously visible.
- SPA basin floor (near Bose, 54°S , 162°W): This site is on the floor of the SPA basin, which possibly exposes the lower crust or upper mantle of the Moon. The site is on the far side of the Moon, out of Earth view, and would require a communications relay system for Earth contact. Observation of the low-frequency radio sky would be possible here.
- Aristarchus Plateau (north of Cobra Head, 26°N, 49°W): This is a diverse site containing unusual rock types, ancient crust, Imbrium basin ejecta, non-mare volcanism, and extensive dark mantling (pyroclastic) deposits. The dark mantle may be good feedstock for ISRU processing (e.g., solar wind hydrogen). There is easy and routine access to this near-equatorial, near-side site.
- Rima Bode (near Vent, 13°N, 3.9°W): There are extensive regional high-Ti dark mantle deposits at this site. The vent system for these ash deposits may contain xenoliths (exotic chunks) of rock from the deep mantle of the Moon. Existing data suggest high-Ti pyroclastic glass may be excellent feedstock for ISRU processing. There is easy and routine access to this near-equatorial, near-side site.
- Mare Tranquillitatis (north of Arago, 8°N, 21°E): High-Ti maria near the landing site for the Apollo 11 mission in 1969. High-Ti basalts are excellent feedstock for ISRU processing. Smooth maria is physically well-characterized and already covered by extensive, high-resolution photography (from the lunar orbiter). There is easy and routine access to this near-equatorial, near-side site.
- North pole (rim of Peary B, 89.5°N, 91°E): This area of near-permanent sunlight on the rim provides access to power and proximity to a cold trap (crater interior) that may contain water ice. The site is on the distal edges of the Imbrium basin ejecta blanket. The northern celestial hemisphere is continuously visible.
- Oceanus Procellarum (inside Flamsteed P, 3°S, 43°W): This mare site is on the western near side. Basalts here appear to be some of the youngest lavas on the Moon, possibly as young as 1 billion years. High-Ti lavas provide excellent feedstock for ISRU processing. There is easy and routine access to this near-equatorial, near-side site.
- Central far side highlands (near Dante, 26°N, 178°E): This highland site is on the central far side of the Moon. The site appears to be on the most ancient, primordial crust of the Moon—the original magma ocean anorthosites. There is Al- and Ca-rich regolith available for ISRU processing. Observation of the low-frequency radio sky would be possible here. This site would require relay satellites for Earth communications.
- Orientale basin floor (near Kopff, 19°S, 88°W): This is a combination highland/mare site on the floor of the youngest major basin on the Moon. Crater Kopff has unusual morphology and may be an endogenically modified impact crater. This site contains both mare and highland regolith feedstock for ISRU processing. The limb site is sometimes out of view of Earth and would require a relay for continuous communications.

- Smythii basin floor (near Peek, 2.5°N, 86.5°E): This mare site is on the floor of the ancient Smythii basin on the eastern limb of the Moon. Mare basalts here are very young (approximately 1–2 billion years) and could be used to study lunar thermal history. This site contains high-Fe mare regolith feedstock for ISRU processing. The limb site is sometimes out of view of Earth and would require a relay for continuous communications.

4.3.7 Sortie Mission Surface Activities

As part of the ESAS effort, a lunar sortie crew DRM was developed to describe the types of activities that will occur on the lunar surface during the initial demonstrations of human exploration beyond Earth orbit. The goals of these short-duration missions are to:

- Conduct scientific investigations using the Moon as a natural laboratory to better understand planetary processes such as impact cratering and volcanism and to understand the integrated effects of low gravity, radiation, and the planetary environment on the human body;
- Conduct ISRU demonstrations, such as regolith excavation, manipulation, and processing; and
- Conduct Mars-forward testing of operational techniques and systems, including field exploration techniques, EVA systems (i.e., space suits, portable life support systems, and surface mobility systems), and dust mitigation and planetary protection approaches.

The guidelines and assumptions used in the development of the lunar sortie crew mission DRM were:

- A crew size of four on the lunar surface with all crew members simultaneously conducting EVAs;
- Surface mission durations of up to 7 days;
- The capability to land anywhere on the Moon;
- Sortie missions directed toward scientifically interesting sites;
- The crew living and working out of their landed LSAM; and
- No predeployed surface assets required to perform a sortie mission.

4.3.7.1 Surface Mission Description

Surface activities on lunar sortie crew missions will be similar to those on the final Apollo J-missions (i.e., Apollo 15, 16, and 17). The emphasis will be on EVAs, where all four crew members don space suits and simultaneously conduct operations on the lunar surface outside of their landed spacecraft. An airlock on the LSAM would allow sick, injured, or fatigued crew members to remain inside the pressurized LSAM while others are conducting EVAs. The crew's activities will focus on exploration science and field work, the deployment and emplacement of long-term scientific experiments, equipment and robotic systems, technology demonstrations of ISRU techniques, and the emplacement of surface infrastructure such as power, communication, and navigation systems to support continuing activities once the crew leaves.

Surface missions lasting 4 days will have EVAs each day. The first (landing) and last (take-off) days will likely have shorter duration EVAs of 4–6 hours, while the middle two days will each have a full 6- to 8-hour EVA period. Longer-duration sortie missions of up to 7 days would likely require at least 1 day of rest without planned EVAs. The crew will mostly work as two separate EVA teams (with each team consisting of two people), although all four crew members could work together on large and/or complex tasks.

While inside the spacecraft, the crew will perform all of the routine aspects of eating, sleeping, resting, and personal hygiene. Intra-Vehicular Activity (IVA) will also include preparing and maintaining EVA space suits, planning for subsequent operations on the lunar surface, and all activities associated with post-landing and prelaunch operations.

Due to the short-duration of lunar sortie crew missions, surface operations would initially be conducted in a fairly traditional manner similar to the Apollo missions. However, as experience is gained or discoveries are made, flexible and evolvable mission operations should replace scripted timelines, with the surface crew allowed some level of autonomy in decision making.

One of the lessons learned from the Apollo missions was that mobility is key to efficient exploration of the lunar surface. The Apollo Lunar Roving Vehicle (LRV) allowed the Apollo 15, 16, and 17 astronauts to travel farther from the LM, observe a more diverse set of terrain features, and collect a greater variety of samples than the previous Apollo missions that were restricted to foot travel only. Surface mobility systems, such as the LRV, will be needed on future sortie missions to allow all crew members to efficiently explore the local area within 15–20 km of their lander. To accommodate the four-person crews, two roving vehicles, each capable of carrying either two people plus payload or four people total, were chosen for this DRM. Alternative combinations of systems could include one roving vehicle plus two single-person mobility systems—similar in function to four-wheel All Terrain Vehicles (ATVs) common on Earth—or four single-person ATVs (one for each crew member). A detailed surface traffic assessment was not conducted for the ESAS.

Some surface assets emplaced on the lunar surface would continue to operate after the crew leaves. Scientific monitoring equipment, robotic systems, and ISRU demonstrations would be teleoperated from Earth. These continuing activities would be supported by long-lived utilities such as power, communication, and navigation systems. It would also be advantageous to continue operating the crew's roving vehicles via teleoperation. This could extend the range of exploration around the landing site from the 15–20 km explored by the astronaut crew to approximately 50 km. To accommodate this, the long-lived utilities emplaced to support the scientific and ISRU experiments would need to be used.

4.3.7.2 Science Investigations

For the ESAS effort, a team of scientists from the LEAG developed a set of scientific investigations that could be conducted on short-duration lunar sortie crew missions. The science investigations can be categorized as field science, emplacement science, teleoperated science, and operational science.

Field science investigations include activities that the crew will perform to explore and gain an understanding of the landing site. These activities will include making a comprehensive site survey of the area within 15–20 km of the LSAM to develop the geologic context of the site. The crew would collect surface samples representing the different terrains, rocks, and regolith materials (which, through several sortie missions to different sites, should result in a full suite of lunar basalts and highland rocks, impact melt sheets/rocks from as many large craters as possible, and samples of multiple regolith layers, including buried “paleoregoliths”). They would also collect subsurface data and samples by drilling, trenching, and using geophysical profiling techniques or by using natural depressions such as impact craters or lava channels to observe exposed stratigraphy.

Emplacement science uses the deployment of scientific packages to collect long-term measurements of the lunar environment and monitor any changes. The data collected will include geophysical information such as seismicity and heat flow, measurements of the solar wind composition and radiation levels at the lunar surface, the composition of the lunar atmosphere (including dust) and how it changes due to human exploration activities, surface temperatures and reflectivity, and Earth-based radio frequency interference. Other data collection will focus on the long-term effects of microgravity and radiation on molecular and cellular microorganisms and verify radiation-shielding model predictions. Another use for deployed scientific packages is to investigate the ability to use the Moon as a platform for astronomy, with prototype deep-field optical telescopes and radio telescope elements.

Teleoperated science involves the use of robotic systems, or the crew roving vehicles once the crew leaves, by operators on Earth. These mobile systems will be used to extend the comprehensive site survey conducted by the human crew during their stay to greater distances from the landing site. Remote sensing techniques will be used to understand the chemistry and mineralogy of the rocks and regolith and measure local magnetic fields and their variation across the surface.

Operational science is primarily focused on assessing the bio-organic “environmental impact” to the Moon due to human presence, which has implications for planetary protection in both forward- and back-contamination.

A notional schedule of scientific investigations conducted during a lunar sortie crew mission is:

- Day 1: Collect contingency surface samples and deploy scientific packages and robotic systems;
- Days 2 and 3: Conduct field science during surface traverses and correct problems with science packages or robotic systems; and
- Day 4 and beyond: Conduct return visits to sites of particular interest or discoveries and correct problems with science packages or robotic systems.

4.3.7.3 Resource Utilization

ISRU technology demonstrations will include the mining, movement, and/or manipulation of the lunar regolith; the chemical processing of the regolith to produce useful materials such as oxygen, hydrogen, and metals; and demonstration of regolith stabilization techniques for constructing roads and/or landing pads. Other technology demonstrations will likely be associated with the crews’ space suits, portable life support systems, and surface mobility systems in preparation for the long-term use of these items at a future outpost facility. Sortie mission ISRU demonstrations are further discussed in **Appendix 4J**, **ISRU**.

4.3.7.4 Required Surface System Capabilities

A number of required surface system capabilities have been identified to allow for successful and efficient lunar sortie crew missions, including:

- LSAMs capable of supporting up to 7 days of crew operations on the lunar surface, including the ability to send all crew members out on EVAs each day;
- Robust EVA systems that support daily operations, including space suits and portable life support systems that can operate in all lunar environmental extremes, suits that are flexible and mobile enough to allow geologic field work and the emplacement of scientific packages to be conducted, and portable life support systems that have an on-surface recharge capability and support an 8-hour EVA day;
- Surface mobility assets (i.e., roving vehicles) that support the crew's ability to range up to 15–20 km away from the lander site, which is over the horizon and beyond the "walk-back" distance of Apollo;
- Robotic systems and/or the crew's roving vehicles that allow post-sortie teleoperation for continued remote/robotic science exploration out to approximately 50 km from the lander site. These systems and/or vehicles are equipped with a basic remote sensing capability (i.e., spectrometers, imaging);
- The ability to acquire subsurface information by drilling into the regolith, equivalent to the 3-m depth accomplished on Apollo;
- Long-lived utilities (i.e., power, communication, and navigation) for deployed science packages, ISRU demonstrations, and teleoperated robotic systems; and
- Orbital communication and navigation infrastructure that allow global access for sortie missions, including the near side, far side, and polar regions. These assets should be part of an early infrastructure development in the Robotic Lunar Exploration Program (RLEP) to ensure readiness for lunar sortie crew missions.

4.3.7.5 Mars-forward Operations and Technologies

The short duration of lunar sortie crew missions will limit operational similarities to Mars surface exploration. However, valuable information will be gained in the areas of geologic field techniques, teleoperation of robotic systems and possibly crew mobility systems, and dust mitigation and planetary protection strategies.

The small number of surface systems required for sortie missions also limits the demonstration of technologies linked to Mars exploration. Technologies that can be demonstrated on lunar sortie crew missions include: oxygen/methane rocket propulsion, EVA suits and portable life support, surface mobility such as unpressurized rovers, long-lived scientific monitoring packages, ISRU technologies, a small long-lived power supply, and sealed sample containers.

However, the majority of lunar science activities will be directly applicable to Mars exploration. This is supported by the recent findings of the Moon-Mars Science Linkage Steering Group (MMSSG), convened in 2004 within the aegis of the Mars Exploration Program Analysis Group (MEPAG), and by the Lunar Exploration Science Working Group (LExSWG) final report in 1995 that stated a primary scientific theme of using the Moon as a natural laboratory for studying planetary processes.

4.3.8 Outpost Deployment Studies

In addition to developing the order and manifest of the flights for initial outpost deployment, a variety of special studies was conducted. These studies focused on key questions regarding logistical and operational capabilities during deployment and sustained operations of the outpost.

4.3.8.1 Surface Power System

A conceptual design study of the various options associated with a lunar surface power system was performed to assess the technology and architectural options associated with deploying a power system on the lunar surface, which would fit in a larger lunar surface architecture. In support of this assessment/study, the following GR&As were established:

- Power load profiles based on the lunar surface outpost deployment strategy described in **Appendix 4F**, **Alternative Outpost Deployment Options**;
- Outpost power was requested by ESAS to be established at an initial capability of 25 kWe and a target of 100 with a minimum 10-year outpost operation;
- Outpost primary power system infrastructure architecture candidates based on two primary power plant options:
  - Fission Surface Power System (FSPS), and
  - Photovoltaic (PV)/RFC;
- Lunar Radioisotope Power System (LRPS) availability limitations and power plant characteristics limit application to supplemental or special-purpose power needs;
- Power system sizing was performed for equatorial, mid-latitude, and polar locations; and
- Basis of estimates:
  - System performance and sizing are based on calculations, and
  - Masses are based on calculations and/or engineering estimates, not detailed designs in most cases.

Using these assumptions, top-level power systems designs were created in order to evaluate the merits of each system. In some cases, trades were also performed within a power system option (e.g., mobile surface reactor versus stationary reactor, coupled with a mobile distribution system). In addition, each top-level power system design option was then evaluated against a set of FOMs in an attempt to determine the optimized power system architecture. The FOMs for this assessment were determined to be safety and mission success, extensibility/flexibility, programmatic risk, and affordability. However, as can be seen in **Table 4-28**, no top-level power system option stood out as the clear optimum solution.

Table 4-28. Power System FOMs

FOMs		FSPS	PV/RFC Power System	Hybrid Power System	LRPS Supplemental power source only; not comparable to first three options
Safety and Mission Success		Robust power; autonomous; mass scales well with power; 2-km keep-out zone	Highly site sensitive; Higher mass except under ideal illumination; graceful degradation	Diversity of sources; more graceful degradation; highly site sensitive	Autonomous; smaller keep-out zone
Extensibility/ Flexibility	**Lunar Flexibility**	Low sensitivity to outpost location	100 kWe mass prohibitive except at ideal site	Low sensitivity to outpost location after FSPS placement	Can be mobile
	Mars Extensibility	Atmosphere will affect outer shell and radiator design	Reduced solar flux yields significantly larger arrays	Atmosphere will affect outer shell and radiator design plus larger solar arrays	Atmosphere will affect outer shell and radiator design
Programmatic Risk	**Technology Development Risk**	Need to develop infrastructure, system not yet at TRL–6, Many design options	System not yet at TRL–6; alternatives to RFC at even lower TRL	System not yet at TRL–6	System not yet at TRL–6, design options
	Cost Risk	Significant infrastructure cost	Substantially lower DDT&E cost	Significant infrastructure cost	Significant costs in increasing heat source availability
	Schedule Risk	If start FY06	If start FY06	If start FY06	If start FY06
	Political Risk	National Environmental Policy Act (NEPA)	None	NEPA	NEPA
Affordability	**Technology Development Cost**	High, ground test reactor	Relatively Low	Two concurrent programs	Relatively low
	DDT&E Cost	$3+B class	$2B class	$4+B class	$1+B class
	Facilities Cost	Cost of nuclear infrastructure	Existing facilities may need mods for 1/6th g simulation	Cost of nuclear infrastructure	Cost of nuclear infrastructure
	Operations Cost	Relatively low	100 kWe system requires most launches	Relatively low	NSRP process for multiple launches
	Cost of Failure	High consequences if first reactor fails	High mass for replacement	Diversity of sources gives redundancy	Existing
Legend:	Best	Few challenges	Moderate challenges	Serious challenges	Potential show stopper

The team concluded that a fission nuclear power system offered the best solution, based on somewhat subjective evaluation of the FOMs coupled with consideration of the lunar architecture as a whole. It was felt that extensibility/flexibility should be held as the primary FOM. This was driven by consideration of the intended purpose of lunar experience as a precursor to the exploration of Mars. In addition, a more flexible power system design could mitigate cost, schedule, and/or technical issues of other lunar architecture elements (e.g., ISRU systems). From that vantage point, a fission surface power which is more extensible to Mars and offers more "graceful" or gradual scaling (i.e., mass increase is minimal as a result of increases in output power (**Appendix 4G, Surface Power System**) to increasing power would bring far more flexibility to the overall lunar architecture. Gradual scaling was considered particularly attractive, considering the low fidelity of the load estimations for various lunar architecture elements and the potential for growth during more detailed design efforts. Consequently, the preferred power system option was determined to be Fission Surface Power System (FSPS).

As a result of the trades and analysis of the various power system architecture options studied during this assessment, several key conclusions were drawn:

- All of the various options are of relatively high mass. The best possible solutions appear to be, at a minimum, in the 10- to 16-mT range to provide 100 kWe (i.e., FSPS) at the outpost and many of the solutions are in the 20-mT range.
- All of the power systems options identified will require significant development to achieve TRL–6, and there do not appear to be any practical solutions using demonstrated technologies. The most “traditional” approach is to use the PV/RFC option, but there are development and life challenges even with these systems that have a significant amount of operational experience in space. There are also several long-lead item areas that will need to be addressed early in the development program, including reactor development (for FSPS) and RFCs (for PV/RFC). In addition, the PV/RFC system will require significant further study to determine the feasibility of the design.
- Each power system option also has its own inherent sensitivity to its location on the lunar surface. As a result, the team strongly recommends obtaining 1 m or better resolution lunar topographic, illumination, and surface/subsurface composition data during either the robotic or sortie missions prior to lunar outpost deployment. It was determined that high landing accuracy in geographic locations (low 10’s of meters) and orientation (with respect to either Sun and/or outpost location) would substantially reduce deployment risk and increase system optimization. This is of particular importance to the PV/RFC system. In addition, the PV/RFC option appears to not be practical for locations outside very limited polar regions, if 100 kWe continuous power is required. This is due to the lack of an effective energy storage system.
- Each of the power systems options was also deemed to be sensitive to the lander system configuration. While this is not deemed a technology issue, there will be a significant engineering/development effort required to integrate these power systems options on a lander. This is due to each option having large deployable structures such as radiators and the PV/RFC option also having a large deployable solar array. In addition, the FSPS option will also have to address separation of the reactor from the lunar habitat. As a result of the reactor masses, it was determined that utilization of a mobile power distribution cart is an appropriate method to achieve this separation. This does, however, increase the deployment complexity of the overall system.
- Each of the options appears to be robust with respect to the 10-year operational lifetime requirement. None of the systems have a great sensitivity to micrometeoroids and are able to achieve the required lifetime. There may be logistics issues, however, associated with the PV/RFC systems, and each option will require further study to address system repair and/or maintenance.

Each power system architecture option was evaluated against a relevant set of FOMs, which, for this assessment, was determined to be safety and mission success, extensibility/flexibility, programmatic risk, and affordability. As shown in Table 4-28, however, no top-level power system option stood out as the clear optimum solution. The team felt that a fission nuclear power system offered the most optimum solution based on an evaluation of the FOMs, coupled with consideration of the lunar architecture as a whole. This was due to the flexibility offered to the entire lunar architecture and to its extensibility to Mars, which was considered as a prime driver to the exploration of the lunar surface.

Other power systems technologies may also play a role in the overall power system architecture. As an example, radioisotope systems make an attractive power system choice for mobility systems or for small isolated experiment elements. In addition, PV systems would be required for start-up of the reactor system. Consequently, each of the technologies addressed in this assessment potentially play a role in the overall architecture on the lunar surface.

The above analysis is based on the assumptions of an outpost deployed by a short series of dedicated cargo landers and the availability of these landers to deliver the surface power system. Alternative outpost deployment methods, such as an "incremental build strategy" that utilizes only sortie missions to deploy an outpost in smaller elements, will require reexamination of the choice of power system.

4.3.8.2 Navigation

Navigation strategies were developed for both lunar landing and surface operations. Both of these types of navigation required the team to develop a set of reference performance requirements that should be reexamined during future efforts to match with any changing mission requirements. The navigation strategies for both types of operations are described in the following sections.

4.3.8.3 Descent and Landing Navigation

There were two top-level requirements around which the descent and landing navigation strategy was formulated, including:

- The first outpost deployment flight shall have a landing precision of ±500 m; and
- All subsequent flights to the vicinity of the future habitat location shall have a landing precision of ±100 m.

Furthermore, the strategy was developed in such a way that it was not dependent on the emplacement of navigational aids by lunar sortie mission crew members. While there is a possibility that precursor missions might be flown to the outpost site, due to the uncertainty associated with the assumption that the robotic missions could be used to emplace hardware (or have a lifetime to act as navigational aids themselves), it was decided that the strategy should reflect an approach that was independent of precursor/sortie mission infrastructure emplacement.

The landing navigation strategy that was developed is based on assistance from the DSN (or similar system) for tracking and orbit determination, lunar surface feature tracking for orbit determination and descent navigation, landing site tracking (including hazard detection) for terminal descent/landing, and possibly an altimeter for landing (necessity depends on hardware used for landing site tracking). It was an assumed requirement that the crew will have the ability to assume control of the LSAM upon landing in order to re-designate a landing site or control the orientation of the spacecraft. It was also recommended that the cargo mission flights maintain this redesignation/reorientation capability while being controlled from Earth. This provides a backup mode of landing hazard avoidance in addition to the autonomous systems that will likely be present.

Given this strategy, it was felt that the LSAM-related hardware required to perform these functions was either presently available or could easily be obtained prior to the date of required usage. The future availability and capability of the DSN or similar systems are still in question.

4.3.8.4 Surface Navigation

For the purposes of this study, the requirements for local (less than 3 km radius from the habitat), near-field (less than15 km radius from the habitat), mid-field (less than 30 km radius from the habitat), and far-field (greater than 30 km radius from the habitat) were established. There were three top-level requirements around which the descent and landing navigation strategy was formulated:

- A crew member's location shall be known relative to the outpost to within 100 m during nominal operations. Position knowledge approaching 10 m is desired. The following scenarios describe nominal field operations:
  - Within 3 km of the outpost, the EVA crew may travel by foot (without a rover);
  - Between 3 km and 15 km of the outpost, at least one unpressurized rover will always be within 200 m of the EVA crew;
  - Between 15 km and 30 km of the outpost, at least two unpressurized rovers will always be within 200 m of the EVA crew.
  - At distances greater than 30 km (and approaching 100 km) of the outpost, at least one pressurized rover will always be within 500 m of the EVA crew.
- During off-nominal conditions, a crew member shall be able to find their way back to within 100 m of the outpost; and
- The location of a pressurized/unpressurized lunar rover shall be known (relative to the outpost) to within 100 m. Position knowledge approaching 10 m is desired.

Given these requirements, the team decided to adopt a strategy that did not require the use of orbital or Earth-based assets. It was felt that sufficient position and heading knowledge could be obtained through the use of "autonomous" navigation systems that are initialized at the beginning of each traverse, similar to Apollo. Multi-day traverses might require reinitialization based on maps, but these types of traverses will not become an issue until pressurized far-field traverses are attempted at some point later in the program.

Orbital navigation systems were considered, but ruled out, for the following reasons. In comparing relative costs, the "autonomous" system is much more competitive in providing, in a timely manner, a navigation system that has the required accuracy. A "Global Positioning System- (GPS-) type system" could be employed, but the costs associated with developing, deploying, and maintaining such a system that provides readily available location and heading data could be prohibitive. While a partial constellation could be deployed, it would incur a performance penalty manifested as a time-delay (i.e., the on-orbit assets would need to perform multiple overhead passes in order to provide the necessary position knowledge). This is an undesirable solution—especially in the case of an EVA team whose rover has failed, in which case the crew members would need to walk back to the habitat. Given the limited life of the crew members' suits, delays in position and heading knowledge would have significant impacts to their ability to return to the habitat safely.

Given the adopted strategy, it was felt that the surface navigation hardware required to perform these functions was either presently available or could easily be obtained prior to the date of required usage. The performance goal to obtain position knowledge approaching 10 m will be a cost driver; however, because this goal is not a requirement, flexibility is available to trade cost versus performance.

It is important to note that none of the landing or surface navigation performance requirements caused a great deal of controversy among the navigation community supporting this portion of the ESAS. However, the solutions should be considered as notional, since much work remains to prove that the performance can be achieved with the stated strategies.

4.3.8.5 Communication

The communication strategy is driven by both the sortie and outpost mission requirements. The sortie missions are intended to have global access to perform operations during short periods of time (up to 7 days) anywhere on the lunar surface. Alternatively, the outpost is intended to have continuous, concentrated operations in a specific location that does not necessarily have a continuous view of Earth.

During both types of missions, it would be highly desirable to maintain constant communication with Earth. Therefore, 24 hours per day/7 days per week (24/7) coverage at the mission site throughout the duration of the mission was treated as a requirement. Furthermore, it was decided that 24/7 global availability should be provided. While this could be relaxed, resulting in a reduced constellation, it would impose restrictions on mission timing, mission duration, and/or mission location. In addition to the communication requirements between the lunar surface and Earth, a requirement from the EVA Project Office imposes the necessity to maintain constant communications between surface crew members, regardless of their distance from the habitat.

In order to provide real-time, global, continuous communications with Earth, a constellation of satellites, deployed in lunar orbit, is needed. This constellation would inherently provide the ability to maintain constant communication between crew members operating on the lunar surface. Additionally, Local Area Networks (LANs) might be established in the areas of high activity (e.g., in the vicinity of the habitat). Available bands will include Ultra-High Frequency (UHF) to provide surface-to-surface audio and low-data rate exchange; S-Band to provide surface-to-surface medium-data rate exchange; X-Band to provide Earth-to-relay vehicle low-data rate exchange; and Ka-Band to provide surface-to-surface and Earth-to-surface high-data rate exchange.

4.3.8.6 ISRU

Although the lunar program was not dependent on the success of the ISRU strategy, ISRU was incorporated into the architecture such that its associated systems were deployed and tested early in the lunar program. The magnitude of consumables that will be pursued through ISRU is an open issue, pending the outcome of economic and technical analysis. However, for the purposes of this study, it was assumed that a capability to produce at least enough O2 to resupply the habitat would be pursued.

The initial lunar surface testing of some ISRU systems and processes was incorporated into the sortie mission strategy. These demonstrations primarily focused on proving the chemical reactions associated with the evolvement of O2 from the lunar regolith. If successful, the first outpost deployment flight will carry a pilot lunar miner/hauler unit and possibly a Lunar Polar Resource Extractor (LPRE) (pending the outcome of H2 availability investigations in the lunar polar region) to the lunar surface. In addition to providing the ability to test the miner/hauler and LPRE early in the program, manifesting the miner/hauler on an early flight will allow it to be used to excavate the terrain at the site of the outpost.

The third outpost deployment flight will deliver a lunar oxygen pilot plant and an ISRU logistics carrier to the lunar surface. This pilot plant will be sized to produce a quantity of consumables ranging from the minimum required to resupply habitat O2 to the maximum necessary to fuel two LSAM ascent stages per year. The logistics carrier will be sized accordingly to transport and deliver the O2 to the appropriate elements. There should be a testing period for these elements prior to inserting them into the critical path to ensure that the process can operate successfully and the required purity of the products can be obtained. At a later date, larger-scale ISRU operations may be undertaken, although their magnitude will be dependent on the applications for which they are intended. More details on the initial outpost ISRU strategy are found in **Appendix 4J**, **ISRU**.

4.3.8.7 EVA Capabilities

Of special interest are the EVA capabilities, since many of the lunar mission objectives are executed through the use of EVA. Significant recent efforts have been undertaken within ESMD studies to establish a baseline set of capabilities and an understanding of the associated support systems. Many details on the evolving lunar EVA strategy can be found in the reports that were generated during the course of the ESMD studies, but some of the major points related to field operations are as follows:

- EVA range (assume two-person nominal rovers)
  - No rover: without a rover, the maximum range available will probably be on the order of 1–3 km;
  - Single rover: The achievable distance possible with a single rover will be constrained by the walk-back distance of 10–15 km; and
  - Two rovers: If a backup rover is provided to the single rover, the walk-back requirement can be avoided. The ultimate distance will then be dictated by either the quantity of consumables carried on the rover or by the crew member-suited physiological guidelines (approximately 8 hours). It is assumed that the physiological limit, which would allow an ultimate range of approximately 20–30 km, is the constraint. This strategy also allows for the ability of two EVA teams to simultaneously explore up to walk-back distances which, in turn, may increase the science return substantially, especially given that there will be relatively short periods of “favorable” EVA time for exploration within a lunar cycle or during the short reconnaissance missions.
- EVA suit: 4-hour uninterrupted operation. Recharge allows 6–8 hour EVA;
- Work Efficiency Index goal: 1.75 (currently 0.43 for ISS); and
- A planetary suit used for nominal surface operations and an in-space suit used as backup, but not designed for repeated surface EVAs.

The initial and evolutionary EVA field capabilities are shown in **Figure 4-56**.

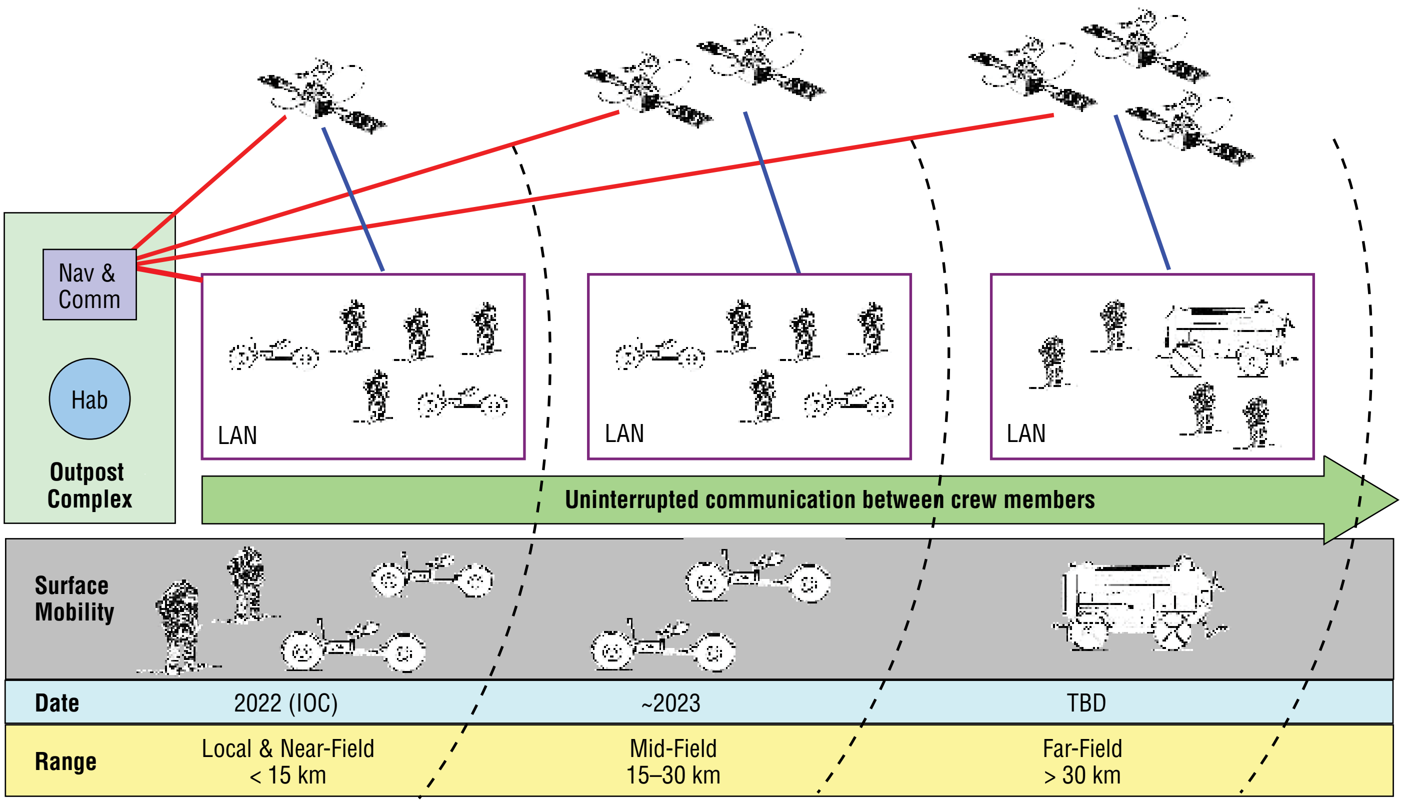


Figure 4-56. Initial and Evolved EVA Field Capabilities

4.3.8.8 Outpost Deployment Conclusions

This lunar surface outpost deployment strategy was developed as part of an overall lunar program strategy led by the ESAS team. Therefore, in order to develop a deployment strategy that was consistent with the overall program strategy, a number of key mission parameters and assumptions (e.g., crew size, mission duration, outpost location, etc.) were adopted.

The primary purpose of developing the outpost deployment strategy was to determine the order and manifest of the flights required to deploy a core set of lunar surface capabilities for sustained, concentrated lunar operations and to provide for the evolution of the surface capabilities as the lunar program progresses. The strategy for outpost deployment was found to be highly dependent on the initial mission parameters and assumptions.

A core set of elements is required in order to enable the key capabilities associated with maintaining a human presence on the lunar surface (e.g., habitat, EVA suits, etc.). Additional elements can be added to the architecture depending on the desire to seek greater degrees of self-reliance (e.g., ISRU) or seek to operate in operationally challenging areas (e.g., out of constant view of Earth). It should not only be expected that performance requirements will grow or change with time (e.g., expanding EVA traverse capabilities), but they should change in order to take advantage of a growing set of surface assets and crew availability (e.g., payload unloading strategies).

The intent of this activity was to gain a general understanding of a reasonable approach for lunar surface operations and outpost deployment. The approach outlined above represents only one point along a continuum of possible outpost deployment strategies. Many of the decisions made within this outpost deployment strategy will be reexamined not only to incorporate final decisions that were made in parallel to this activity, but to also develop the concepts in greater detail and explore their alternatives. (See **Appendix 4F**, **Alternate Outpost Deployment Strategies**, for a discussion of alternative deployment strategies.)

4.3.9 Outpost Mission Surface Activities

As part of the ESAS efforts, a lunar outpost crew mission DRM was developed to describe the types of activities that will occur on the lunar surface during sustained operations at a lunar outpost. The goals of these long-term activities are:

- Take advantage of long-duration human presence to conduct detailed scientific investigations and construct large science facilities;
- Transition ISRU from demonstration to production and incorporation into areas such as life support consumables, spacecraft propellants, and construction materials;
- Understand the integrated effects of long-term low-gravity, radiation, lunar dust, and isolation on the human body in the Moon's environment;
- Conduct Mars-forward testing of operational techniques and planetary surface systems;
- Allow for commercial opportunities to arise that provide services or products related to the operation of the lunar outpost.

The guidelines and assumptions used in the development of the lunar outpost crew mission DRM were: a crew size of four; a crew surface mission duration of 6 months; a crew rotation every 6 months resulting in a permanent human presence on the Moon; uncrewed cargo missions every 6 months (offset from crew missions by 3 months); and the outpost delivered and mostly deployed robotically prior to the arrival of the first outpost crew.

4.3.9.1 Surface Mission Description

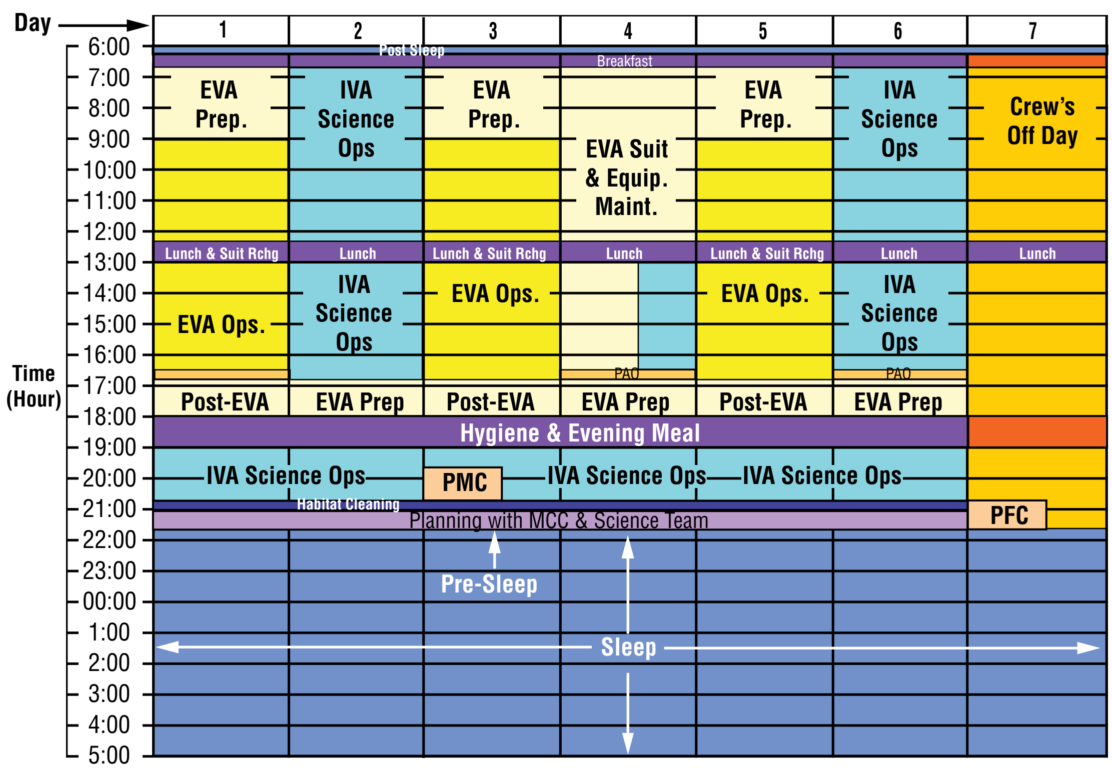
When the initial lunar outpost crew mission arrives at the Moon, the outpost consisting of a habitat, power supply, communication and navigation infrastructure, and surface mobility and robotic systems will already be emplaced. However, it is likely that the crew will have some final assembly and verification tasks to make the outpost operational. Until this process is complete, the four crew members will live in the LSAM. Once the outpost is declared operational, the LSAM will be powered down and kept in a quiescent state for the duration of the 6-month surface mission.

A new outpost crew will arrive at the Moon every 6 months, and there will be a brief period (a few days) where the residing lunar crew will debrief the arriving crew on the status of the outpost and the ongoing research being conducted. During this time, there will be eight people occupying the outpost. After the crew rotation is completed, the departing crew will leave the Moon in the same LSAM that delivered the previous crew. This is operationally similar to the way Soyuz modules are currently changed out at the ISS and ensures that any LSAM does not sit on the Moon for longer than 1 year.

Automated cargo missions will also arrive at the lunar outpost to deliver needed supplies and equipment. One automated cargo mission will land on the Moon every 6 months, with the 6-month centers being offset by 3 months from the crew missions. The outpost crew residing on the Moon when the cargo arrives will be involved in unloading the cargo lander and resupplying the outpost with consumables. Replacement consumables could include life support (air, food, and water), medical supplies, cleaning supplies, personal hygiene items, data storage media, and batteries for small equipment. Modular parts, tools, and supplies to replace or repair failed systems, new science experiments or analytical equipment, and additional specialized robotic systems could also be delivered.

While surface operations during sortie missions were dominated by EVA on the lunar surface, a more balanced schedule of EVA and IVA inside the habitat will occur during crew missions at the outpost (**Figure 4-57**). Sustained EVA over the 6-month outpost crew missions will be limited by the extreme radiation environment on the lunar surface and the accumulated dose each individual crew member receives, the fatiguing nature of EVA operations, space suit maintenance and repair, and portable life support system logistics. Also, while every EVA conducted during sortie missions consisted of all four crew members at the same time, more EVAs during an outpost mission will consist of a single two-person team, with two different teams conducting EVAs on alternating days.

Figure 4-57. Notional Schedule for a Typical Week at a Lunar Outpost



Because of the long-duration of outpost crew missions, the crew's schedule of activities on the lunar surface will not be as "scripted" or tightly controlled as sortie missions. Activities and milestones will be planned more on a weekly basis, rather than the daily basis of sortie missions, and surface operations will gradually transition from Earth-dominated control to local control and crew autonomy. This crew-centered approach to surface operations will be needed on Mars, where the time delay associated with Earth-Mars communications prohibits Earth-based control. Also, as with the ISS, long-duration crews will need "off-duty" and/or "light duty" days where there are no or minimal scheduled activities.

Surface mobility will remain a key asset on the lunar surface at the outpost for all of the same reasons it is important on crew sortie missions. In addition to the 15–20 km exploration range of the unpressurized roving vehicles used on crew sortie missions, outpost crews will also need the use of pressurized rovers, in which crews can operate for days at a time in a "shirt-sleeve" environment away from the outpost. Pressurized rovers would be used much in the same way small submersible craft are used to explore under the seas on Earth. A single pressurized rover could extend the range of human exploration to approximately 50 km away from the outpost, with appropriate planning for crew safety considerations in the event of a rover failure. A second pressurized rover would allow exploration beyond 50 km.

4.3.9.2 Science Investigations

Geoscience activities will include all of the activities conducted during crew sortie missions, but would cover a larger surface area. The longer duration of outpost missions would also allow more detailed studies to be pursued, such as investigating the layered nature and formation process of the lunar regolith. Two new areas of exploration science that will occur at the outpost will involve the crew's use of teleoperated robotic explorers and the ability to perform preliminary chemical and mineralogical analyses on geologic samples. All of the geoscience activities performed on the Moon will demonstrate the equipment and techniques that will enable the efficient geologic exploration of Mars and other destinations in the solar system. Also, the knowledge gained on the basic geologic processes, such as impact and volcanism, will help scientists better understand how these processes occur throughout the solar system.

Other sciences, such as space physics and astronomy, will benefit from the long-duration outpost missions with the ability to build up large-surface infrastructure projects, such as large aperture optical or radio telescopes, or arrays of smaller telescopes acting together as an interferometer.

Life science research and medical operations will allow scientists to begin to understand the long-term effects on the human body of living and working on a planetary body. Medical care techniques such as preventive medicine, telemedicine, and trauma care, countermeasure procedures such as exercise regimens and nutrition, and research on topics such as bone loss, cardiovascular/cardiopulmonary function, skeletal muscle status, and neurological function will improve our ability to keep astronaut crews healthy and productive on long-duration space missions.

4.3.9.3 Resource Utilization

During the outpost missions, the use of in-situ resources will transition from demonstration to incorporation, and ISRU technologies successfully demonstrated on the sortie missions will be scaled up to production-level plants and facilities (**Figure 4-58**). Early activities will include surface construction of berms near landing zones to protect surface assets against regolith blast ejecta from landing spacecraft and roads on highly trafficked paths to reduce the mobilization of lunar dust and the wear on surface mobility systems. Also, the production and storage of life support consumables such as oxygen for the crew's habitat and EVA systems will begin the transition from reliance on Earth-supplied logistics to self-sufficiency on the Moon. As production rates increase, lunar resources will provide the propellants needed by the landing spacecraft, which will lead to basing and servicing of reusable spacecraft on the Moon's surface.

Figure 4-58. Lunar Crew Discussing Outpost Operations With ISRU Facilities in Background

New ISRU capabilities can be demonstrated and incorporated into mission plans on an as-needed and evolutionary basis to lower the cost of outpost missions and demonstrate capabilities that may be required for long-duration stays on the Mars surface. One area that may be critical for long-term lunar operations and trips to Mars where logistics management may be difficult is in-situ manufacturing and repair. This capability includes the ability to fabricate spare parts—especially for high-wear excavation and regolith processing items—and repair techniques for both internal and external hardware. Fabrication processes of interest include additive, subtractive, and formative techniques for multiple feedstock materials (metals, plastics, and ceramics). A "machine shop" capability that includes repair techniques and part characterization may be required. Initially, feedstock from Earth can be used for manufacturing parts; however, resources from regolith to support this capability would be required for permanent surface operations.

The ability to extract metals (iron, titanium, aluminum, etc.) and silicon from regolith is of interest to support in-situ manufacturing and construction capabilities that could be used to lower the cost of infrastructure growth during the outpost phase. Included in this work is development of other manufacturing and construction feedstock, such as concrete, wires, basaltic fibers and bars, metal tubing, etc. Several oxygen extraction concepts can be modified to include additional steps to extract these resources for use in construction feedstock.

Depending on the duration and scope of the outpost phase, the ability to construct landing pads, structures, habitats, observatories, and other infrastructure items of interest from in-situ materials may become important. In-situ fabrication of energy/power generation such as lunar array production may also be of interest for both infrastructure growth and space commercialization potential. Studies and laboratory work have been performed to show that production of solar arrays with minimal Earth consumables is feasible.

A further area of interest during the outpost phase will be internal and external waste recycling. At this time, large amounts of plastic and metal "trash" on the ISS are disposed of on Progress vehicles during atmospheric reentry. These carbon, hydrogen, and metal resources could be critical in supporting in-situ manufacturing and repair capabilities. Also, non-reusable landers surrounding the outpost may be ready sources of carbon, metals, and plastics for in-situ processing. Lastly, reuse of transportation assets will be required for human lunar exploration to become economical in the long-term. Instead of only supplying propellant to ascent stages, deploying a single-stage lander that can be refueled to either return to Earth or travel to another location on the lunar surface would significantly enhance the science return of human lunar operations. For this to occur, production of oxygen (and fuel if possible) would have to be scaled up to between 30 and 60 mT per year. Outpost ISRU capabilities are further detailed in **Appendix 4J**, **ISRU**.

4.3.9.4 Required Surface System Capabilities

To support the diverse set of surface operations described above, the lunar outpost must provide certain capabilities as part of its design. The most critical capability is the ability to support frequent and substantial EVAs. Important EVA surface systems include: (1) space suits that are flexible and lightweight, yet durable and maintainable, and allow for 8 hours of work; (2) an efficient airlock that not only provides access to the surface, but also protects the habitat and crew from lunar dust and provides an area to repair and service the space suits; and (3) surface mobility systems to allow the crew to efficiently explore the region surrounding the outpost.

The human crews will need to be aided in their explorations by robotic systems capable of teleoperation by the outpost crew or operators on Earth. Robotic explorers will provide the planet-wide reconnaissance needed to develop a global understanding of the Moon. Robotic systems will also provide the majority of the ISRU regolith mining and manipulation equipment.

As lunar samples are collected, the ability to analyze those samples with laboratory equipment will become necessary, since the number of samples collected during a long-duration outpost mission will far exceed the amount able to be returned to Earth. Analytical laboratories and equipment will also be needed to support biological investigations concerned with understanding the integrated effects of low gravity, radiation, and dust on the human body, as well as astrobiological investigations associated with planetary protection and the detection of extraterrestrial life.

Exploration of the Moon will not be restricted to the lunar surface. Subsurface exploration will also be vital to understanding the Moon as a whole. Therefore, systems to allow drilling, trenching, and geophysical profiling must be present. Drilling to depths of tens of meters will be needed to reach the base of the regolith and hundreds of meters will be needed to penetrate into the megaregolith. Geophysical techniques will benefit most from stations separated by great distances. While these can be deployed during sortie missions as well, the outpost robotic systems and small, remote, long-lived power and communication systems will be needed to establish and maintain the network of stations.

As outpost activities increase in scale, power on the order of 100 kW will be needed, particularly for ISRU processing plants. This likely will require nuclear power supplies. Habitat systems that will be critical to this endeavor must have a regenerative life support capability to minimize the consumables resupply and radiation protection, thus enabling the habitat to serve as a safe haven in the event of a solar proton event.

4.3.9.5 Mars-forward Operations and Technologies

Most aspects of lunar outpost crew missions will build experience that directly applies to operations on Mars. Short-duration (less than 90 days) and long-duration (approximately 500 days) Mars surface missions can each benefit from the confidence gained in crew operations and system reliability based on the 180-day lunar missions.

Operationally, the most important concepts to be developed during lunar outpost crew missions include crew autonomy and crew teleoperation of robotic exploration systems. It may be desirable at some point in time to implement Mars-like time delays in lunar outpost communications with Earth and daily activities based on a Martian Sol (24 hours, 39 minutes).

From a systems perspective, the most important technologies to be demonstrated on the Moon include oxygen-methane rocket propulsion, long-lived power generation, regenerative life support, teleoperated robotic systems, EVA systems (suits and unpressurized and pressurized roving vehicles), geoscience and bioscience analytical equipment, medical and telemedicine equipment, dust mitigation and planetary protection equipment, and ISRU mining and storage/distribution systems.

Because of the different environments on the Moon and Mars, certain technologies used on the Moon may require some significant modifications before being used on Mars. These include thermal rejection systems for power systems and habitats (Martian atmosphere), ISRU chemical processes (different resources), EVA space suits, portable life support systems, and roving vehicles (higher Mars gravity).

4.3.10 Robotic Precursor Missions

Robotic missions to the Moon should be undertaken prior to human return to the Moon for several reasons. Robotic missions can collect strategic knowledge that permits safer and more productive human missions. Such data includes information on lunar topography, geodetic control, surface environment, and deposits of largely unknown character, such as those of the polar regions. This information can be collected by a variety of spacecraft, including orbiters and soft landers.

In addition to collecting important precursor data, robotic missions can deliver important elements of the surface infrastructure to the eventual outpost site. Such deliveries include exploration equipment (i.e., rovers) and scientific instrumentation (i.e., telescopes). Additionally, since the extraction of resources will be an important human activity on the Moon, robotic precursors can deliver elements of the resource processing infrastructure, including digging, hauling, and extraction equipment. It is likely that mission planners will want to experiment with various processing techniques and methods of extraction, and robotic missions can demonstrate process techniques at small scales in advance of the requirement to put large amounts of infrastructure on the lunar surface. Further discussion of robotic lunar precursor activities is contained in **Appendix 4I**, **Lunar Robotic Precursor Missions**.

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5. Crew Exploration Vehicle

5.1 CEV Overview and Recommendations

One of the keys to enable a successful human space exploration program is the development and implementation of a vehicle capable of transporting and housing crew on Low Earth Orbit (LEO), lunar and Mars missions. A major portion of the Exploration Systems Architecture Study (ESAS) effort focused on the definition and design of the Crew Exploration Vehicle (CEV), the the fundamental element by which NASA plans to accomplish these mission objectives. This section provides a summary of the findings and recommendations specific to the CEV.

While the CEV design was sized for lunar missions carrying a crew of four, the vehicle was also designed to be reconfigurable to accommodate up to six crew for International Space Station (ISS) and future Mars mission scenarios. The CEV can transfer and return crew and cargo to the ISS and stay for 6 months in a quiescent state for emergency crew return. The lunar CEV design has direct applications to International Space Station (ISS) missions without significant changes in the vehicle design. The lunar and ISS configurations share the same Service Module (SM), but the ISS mission has much lower delta-V requirements. Hence, the SM propellant tanks can be loaded with additional propellant for ISS missions to provide benefits in launch aborts, on-orbit phasing, and ISS reboost. Other vehicle block derivatives can deliver pressurized and unpressurized cargo to the ISS.

The ESAS team's first recommendation addresses the vehicle shape. It is recommended that the CEV incorporate a separate Crew Module (CM), SM, and Launch Abort System (LAS) arrangement similar to that of Apollo. Using an improved blunt-body capsule was found to be the least costly, fastest, and safest approach for bringing ISS and lunar missions to reality. The key benefits for a blunt-body configuration were found to be lighter weight, a more familiar aerodynamic design from human and robotic heritage (resulting in less design time and cost), acceptable ascent and entry ballistic abort load levels, crew seating orientation ideal for all loading events, and easier Launch Vehicle (LV) integration and entry controllability during off-nominal conditions. Improvements on the Apollo shape will offer better operational attributes, especially by increasing the Lift-to-Drag (L/D) ratio, improving Center of Gravity (CG) placement, potentially creating a monostable configuration, and employing a lower angle of attack for reduced sidewall heating.

A CM measuring 5.5 m in diameter was chosen to support the layout of six crew without stacking the crew members above or below each other. A crew tasking analysis also confirmed the feasibility of the selected vehicle volume. The pressurized volume afforded by a CM of this size is approximately three times that of the Apollo Command Module. The available internal volume provides flexibility for future missions without the need for developing an expendable mission module. The vehicle scaling also considered the performance of the proposed Crew Launch Vehicle (CLV), which is a four-segment Solid Rocket Booster (SRB) with a single Space Shuttle Main Engine (SSME) upper stage. The CEV was scaled to maximize vehicle size while maintaining adequate performance margins on the CLV.

The CEV will utilize an androgynous Low-Impact Docking System (LIDS) to mate with other exploration elements and to the ISS. This requires the CEV-to-ISS docking adapters to be LIDS-compatible. It is proposed that two new docking adapters replace the Pressurized Mating Adapter (PMA) and Androgynous Peripheral Attachment System (APAS) adapters on the ISS after Shuttle retirement.

An integrated pressure-fed Liquid Oxygen (LOX) and methane service propulsion system/Reaction Control System (RCS) propulsion system is recommended for the SM. Selection of this propellant combination was based on performance and commonality with the ascent propulsion system on the Lunar Surface Access Module (LSAM). The risk associated with this type of propulsion for a lunar mission can be substantially reduced by developing the system early and flying it to the ISS. There is schedule risk in developing a LOX/methane propulsion system by 2011, but development schedules for this type of propulsion system have been studied and are in the range of hypergolic systems.

Studies were performed on the levels of radiation protection required for the CEV CM. Based on an aluminum cabin surrounded by bulk insulation and composite skin panels with a Thermal Protection System (TPS), no supplemental radiation protection is required.

Solar arrays combined with rechargeable batteries were selected for the SM due to the long mission durations dictated by some of the Design Reference Missions (DRMs). The ISS crew transfer mission and long-stay lunar outpost mission require the CEV to be on orbit for 6–9 months, which is problematic for fuel cell reactants.

The choice of a primary land-landing mode was primarily driven by a desire for land landing in the Continental United States (CONUS) for ease and minimal cost of recovery, post-landing safety, and reusability of the spacecraft. However, the design of the CEV CM should incorporate both a water- and land-landing capability. Ascent aborts will require the ability to land in water, while other off-nominal conditions could lead the spacecraft to a land landing, even if not the primary intended mode. However, a vehicle designed for a primary land-landing mode can more easily be made into a primary water lander than the reverse situation. For these reasons, the study attempted to create a CONUS land-landing design from the outset, with the intention that a primary water lander would be a design off-ramp if the risk or development cost became too high.

In order for CEV entry trajectories from LEO and lunar return to use the same landing sites, it is proposed that NASA utilize skip-entry guidance on the lunar return trajectories. The skip-entry lunar return technique provides an approach for returning crew to a single CONUS landing site anytime during a lunar month. The Apollo-style direct-entry technique requires water or land recovery over a wide range of latitudes. The skip-entry includes an exo-atmospheric correction maneuver at the apogee of the skip maneuver to remove dispersions accumulated during the skip maneuver. The flight profile is also standardized for all lunar return entry flights. Standardizing the entry flights permits targeting the same range-to-landing site trajectory for all return scenarios so that the crew and vehicle experience the same heating and loads during each flight. This does not include SM disposal considerations, which must be assessed on a case-by-case basis.

For emergencies, the CEV also includes an LAS that will pull the CM away from the LV on the pad or during ascent. The LAS concept utilizes a 10-g tractor rocket attached to the front of the CM. The LAS is jettisoned from the launch stack shortly after second stage ignition. Launch aborts after LAS jettison are performed by using the SM service propulsion system. Launch abort study results indicate a fairly robust abort capability for the CEV/CLV and a 51.6-deg-inclination ISS mission, given 1,200 m/s of delta-V and a Thrust-to-Weight (T/W) ratio of at least 0.25. Abort landings in the mid-North Atlantic can be avoided by either an Abort-To-Orbit (ATO) or posigrade Trans-Atlantic Abort Landing (TAL) south of Ireland. Landings in the Middle East, the Alps, or elsewhere in Europe can be avoided by either an ATO or a retrograde TAL south of Ireland. For 28.5-deg-inclination lunar missions, abort landings in Africa can be avoided by either an ATO or a retrograde TAL to the area between the Cape Verde islands and Africa. However, it appears that even with 1,724 m/s of delta-V, some abort landings could occur fairly distant from land. However, once the ballistic impact point crosses roughly 50°W longitude, posigrade burns can move the abort landing area downrange near the Cape Verde islands.

5.2 CEV Description

5.2.1 CEV Ground Rules and Assumptions

The following Ground Rules and Assumptions (GR&As) were drafted at the beginning of the ESAS for consistency among the team in studying the ESAS Initial Reference Architecture (EIRA). As the study progressed, some of the assumptions were modified or deleted.

In response to the ESAS charter, the first crewed flight of the CEV system to the ISS was assumed to occur in 2011. The CEV design requirements were, however, to be focused on exploration needs beyond LEO. Therefore, the team started with the existing ESMD Revision E Crew Transportation System (CTS) requirements and assessed these against ISS needs for areas of concern where CEV may fall short of ISS expectations. Any such shortcomings were then examined on a case-by-case basis to determine whether they were critical to performing the ISS support function. If they were found not to be critical, such shortcomings were considered as guidelines and not requirements on the CEV.

The CEV reference design includes a pressurized CM to support the Earth launch and return of a crew of up to six, a LAS, and an unpressurized SM to provide propulsion, power, and other supporting capabilities to meet the CEV's in-space mission needs. Operations at ISS will require the CEV pressurized module to be capable of 14.7 psi. The CEV may launch at a lower pressure but must support equalization with the ISS. The CEV docking system was selected to meet exploration needs and, therefore, was assumed to not be APAS-compatible. This approach will require a docking adaptor to (or in place of) the United States On-orbit Segment (USOS) PMA that remains on ISS.

ISS interfaces to CEV (either direct or through intermediate adaptor) will include:

- Hard-line and Radio Frequency (RF) voice channels (two);
- Basic ECLS System (ECLSS) for habitability air exchange via flexhose—the ISS provides temperature and humidity control and air revitalization capabilities;
- Minimal keep-alive/habitability power provided by the ISS;
- Status telemetry and hard-line command via ISS bent pipe;
- Automated Rendezvous and Docking (AR&D) RF interfaces; and
- Transfer of high-pressure oxygen and nitrogen to ISS airlock.

ISS support assumptions include:

- Two crewed flights per year for crew rotation;
- One uncrewed, unpressurized cargo flight per year; and
- Three uncrewed, pressurized cargo flights per year.

ISS pressurized cargo CEV variant (Block 1B) assumptions include:

- The pressurized cargo module is the crewed CEV CM with seats removed and outfitted with stowage accommodations;
    - Stowage unit size is limited to Shuttle Mid-deck Locker Equivalent (MLE) dimensions compatible with APAS-size hatch.
- The pressurized cargo module supports both up- and down-mass capability (i.e., the module lands and is recovered);

- The AR&D system meets ISS requirements for approach and docking of automated vehicles;
- In addition to dry cargo, the CEV also supports delivery of water, gaseous oxygen, and the transfer of high-pressure oxygen and nitrogen to airlock tanks; and
- The SM provides delta-V for transfer from LV insertion orbit to ISS rendezvous and deorbit from ISS.

ISS crewed CEV variant (Block 1A) assumptions include:

- Same as Block 1B variant, with the following exceptions:
- CEV CM nominally outfitted for three crew plus logistics;
  - Assume the Russians continue to support the ISS with Soyuz—it is considered unrealistic to expect the Russians to stop producing Soyuz.
- The CEV will support a docking as early as Rev3 on flight day 1;
- Assume no less than 6 days of stand-alone free-flight capability;
  - Three days for a flight day 3 rendezvous and docking profile;
  - One contingency rendezvous delay day; and
  - Two contingency post-undock days dwell time for resolving systems problems.
- Option of piloted approach/manual docking based on direct targeting (versus offset targeting used for AR&D case); and
- CEV will support a crew of three docked to the station with hatches closed for up to 48 hours.

ISS unpressurized Cargo Delivery Vehicle (CDV) assumptions include.

- Utilizes the same SM as other blocks;
- Delivers unpressurized cargo to ISS;
- Common Berthing Mechanism (CBM) and grapple fixture for capture and berthing with Space Station Remote Manipulator System (SSRMS); and
- Vehicle expended at the end of the mission.

Lunar CEV variant (Block 2) assumptions include:

- Same as Block 1A variant, with the following exceptions:
- The CEV CM outfitted for four people plus To Be Determined (TBD) cargo;
- Assume no less than 16 days of stand-alone free-flight capability;
- TBD supplemental radiation protection;
- The SM provides delta-V for Low Lunar Orbit (LLO) rendezvous, ascent plane change, and Trans-Earth Injection (TEI); and
- Supports first lunar landing in 2018.

Mars CEV variant (Block 3) assumptions include:

- Same as Lunar Block 2 variant, with the following exceptions:
- The CEV CM outfitted for six people plus TBD cargo; and
- Assume no less than 2 days of stand-alone free-flight capability.

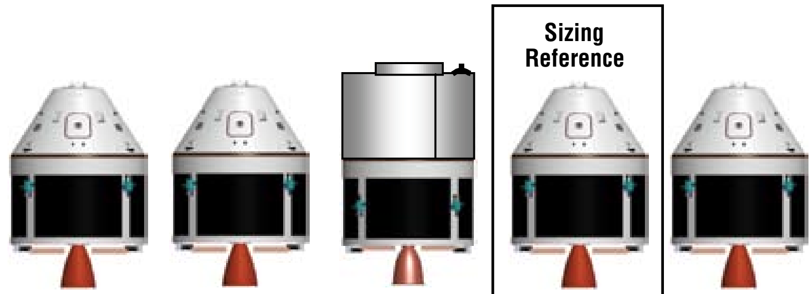
5.2.2 Design Approach

The CEV design was approached with the focus on a lunar polar mission. In addition to optimizing the design for exploration missions, the team also assessed the possible means by which the CEV could access the ISS. The lunar design starting point was very important, as a vehicle optimized for the ISS and then adapted for lunar missions may have a very different outcome. Past studies, such as the Orbital Space Plane (OSP) and the Crew Return Vehicle (CRV), designed vehicles to solely go to the ISS and, therefore, did not address transit out of LEO. The biggest difference with this study is that the CEV does not have a 24-hour medical return mission from the ISS coupled with an emergency evacuation mission that required system power-up in 3 minutes. These requirements would drive vehicle system design and landing site selection. Neither the Space Shuttle nor Soyuz were designed to go to the ISS and meet these requirements, and the CEV is modeled after the capabilities that these two vehicles provide to the ISS. The CEV will be the United States' next human spacecraft for the next 20 to 30 years and should have the flexibility to meet the needs for missions to the Moon, Mars, and beyond.

Vehicle size, layout, and mass were of central importance in this study, because each factors into vital aspects of mission planning considerations. Detailed subsystem definitions were developed and vehicle layouts were completed for a four-crew lunar DRM and a six-crew Mars DRM. The lunar mission was a design driver since it had the most active days with the crew inside. The Mars DRM, which was a short-duration mission of only 1 to 2 days to and from an orbiting Mars Transfer Vehicle (MTV), drove the design to accommodate a crew of six. Ultimately, the CEV CM was sized to be configurable for accommodating six crew members even for an early mission to the ISS.

The different CEV configurations were each assigned a block number to distinguish their unique functionality. The Block 1 vehicles support the ISS with transfer of crew and cargo. The Block 1A vehicle transfers crew to and from the ISS. This vehicle can stay at the ISS for 6 months. Varying complements of crew and pressurized cargo can be transported in the Block 1A CM. The Block 1B CM transports pressurized cargo to and from the ISS. The crew accommodations are removed and replaced with secondary structure to support the cargo complement. The relationship between the Block 1A and Block 1B CMs is similar to that of the Russian Soyuz and Progress vehicles. Unpressurized cargo can be transported to the ISS via the CDV. The CDV replaces the CM with a structural "strong back" that supports the cargo being transferred. The CDV uses the same SM as the other blocks and also requires a suite of avionics to perform this mission. The CDV is expended after its delivery mission. The Block 2 CEV is the reference platform sized to transfer crew to the lunar vicinity and back. Detailed sizing was performed for this configuration and the other blocks were derived from its design. The Block 3 configuration is envisioned as a crewed transfer vehicle to and from an MTV in Earth orbit. The crew complement for this configuration is six. No detailed design requirements were established for this block and detailed mass estimates were never derived.

Design details for each block configuration are discussed in later sections. A mass summary for each block is shown in **Figure 5-1**. Detailed mass statements were derived for each block and are provided in **Appendix 5A**, **CEV Detailed Mass Breakdowns**.



	Block 1A ISS Crew	Block 1B ISS Press Cargo	CDV ISS Unpress Cargo	Block 2 Lunar Crew	Block 3 Mars Crew
Crew Size	3	0	0	4	6
LAS Required	4,218	None	None	4,218	4,218
Cargo Capability (kg)[1]	400	3,500	6,000	Minimal	Minimal
CM (kg)	9,342	11,381	12,200	9,506	TBD
SM (kg)	13,558	11,519	6,912	13,647	TBD
Service Propulsion System delta-V (m/s)	1,544[2]	1,098[2]	330	1,724	TBD
EOR–LOR 5.5-m Total Mass (kg)	22,900	22,900	19,112	23,153	TBD

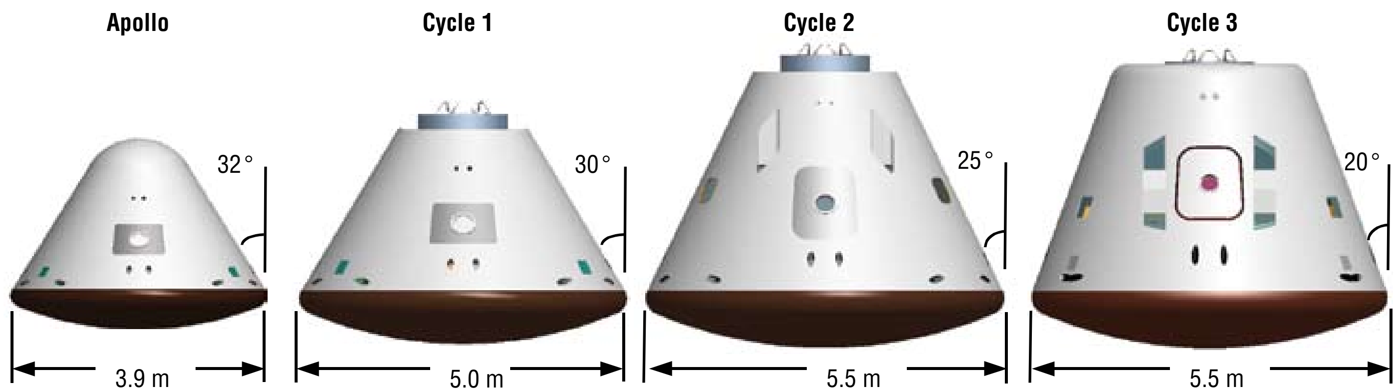
Note 1: Cargo capability is the total cargo capability of the vehicle including Flight Support Equipment (FSE) and support structure.
Note 2: A packaging factor of 1.29 was assumed for the pressurized cargo and 2.0 for unpressurized cargo.
Extra Block 1A and 1B service propulsion system delta-V used for late ascent abort coverage.

Figure 5-1. Block Mass Summaries

The design and shape of the CEV CM evolved in four design cycles throughout the study, beginning with an Apollo derivative configuration 5 m in diameter and a sidewall angle of 30-deg. This configuration provided an Outer Mold Line (OML) volume of 36.5 m^3 and a pressurized volume of 22.3 m^3. The CM also included 5 g/cm^2 of supplemental radiation protection on the cabin walls for the crew's protection. Layouts for a crew of six and the associated equipment and stowage were very constrained and left very little habitable volume for the crew.

A larger CEV was considered in Cycle 2, which grew the outer diameter to 5.5 m and reduced the sidewall angles to 25 deg. Both of these changes substantially increased the internal volume. The pressurized volume increased by 75 percent to 39.0 m^3 and the net habitable volume increased by over 50 percent to 19.4 m^3. The desire in this design cycle was to provide enough interior volume for the crew to be able to stand up in and don/doff lunar EVA suits for the surface direct mission. Most of the system design parameters stayed the same for this cycle including the 5 g/cm^2 of supplemental radiation protection.

Cycle 3 reduced the sidewall angles even further to 20 deg in an effort to achieve monostability on Earth entry. The sidewall angle increased the volume further. Because the increases in volume were also increasing the vehicle mass, the height of the vehicle was reduced by 0.4 m, reducing the height-to-width aspect ratio. This configuration showed the most promise in the quest for monostability, but the proper CG was still not achieved. Analysis in this design cycle showed that the supplemental radiation protection could be reduced to 2 g/cm^2. **Figure 5-2** illustrates the progression of the configurations through Cycle 3 of the study as compared to Apollo and the attached table details the changes in diameter, sidewall angle, and volume. Data for Cycle 4 is also shown and is described in the following paragraphs.



Configuration	Diameter (m)	Sidewall Angle (deg)	OML Volume (m^3)	Pressurized Volume(m^3)
Apollo	3.9	32.5	15.8	10.4
Cycle 1 (EIRA)	5.0	30.0	36.5	22.3
Cycle 2	5.5	25.0	56.7	39.0
Cycle 3	5.5	20.0	63.6	39.5
Cycle 4	5.5	32.5	45.9	30.6

Figure 5-2. CEV Crew Module Sizing Progression

Cycle 4 was the final CEV design cycle and began after the decision was made to no longer consider the lunar surface direct mission. The design implications to the CEV (i.e., difficulty including an airlock and complex operatives) and the low mass margins surrounding the lunar surface direct mission mode were the primary reasons for taking the mode out of consideration. The Cycle 4 CEV was sized for a dual-launch Earth Orbit Rendezvous-Lunar Orbit Rendezvous (EOR–LOR) mission mode where the CEV performs a rendezvous with the Earth Departure Stage (EDS) and LSAM in LEO, stays in lunar orbit while the LSAM descends to the lunar surface, and performs another rendezvous with the LSAM in lunar orbit. No supplemental radiation protection was included in the mass estimates for this design analysis due to results from a radiation study reported in **Section 4**, **Lunar Architecture**.

The resulting Cycle 4 CM shape is a geometric scaling of the Apollo Command Module (**Figure 5-3**). The vehicle is 5.5 m in diameter and the CM has a sidewall angle of 32.5 deg. The resulting CM pressurized volume is approximately 25 percent less than the Cycle 3 volume, but has almost three times the internal volume as compared to the Apollo Command Module. The CEV was ultimately designed for the EOR–LOR 1.5-launch solution, and volume reduction helps to reduce mass to that required for the mission. **Figure 5-4** depicts how vehicle sidewall angle and diameter affect pressurized volume and the resulting design point for each cycle.

The following sections detail the design of the lunar CEV CM, SM, and LAS, as well as the other block variants.

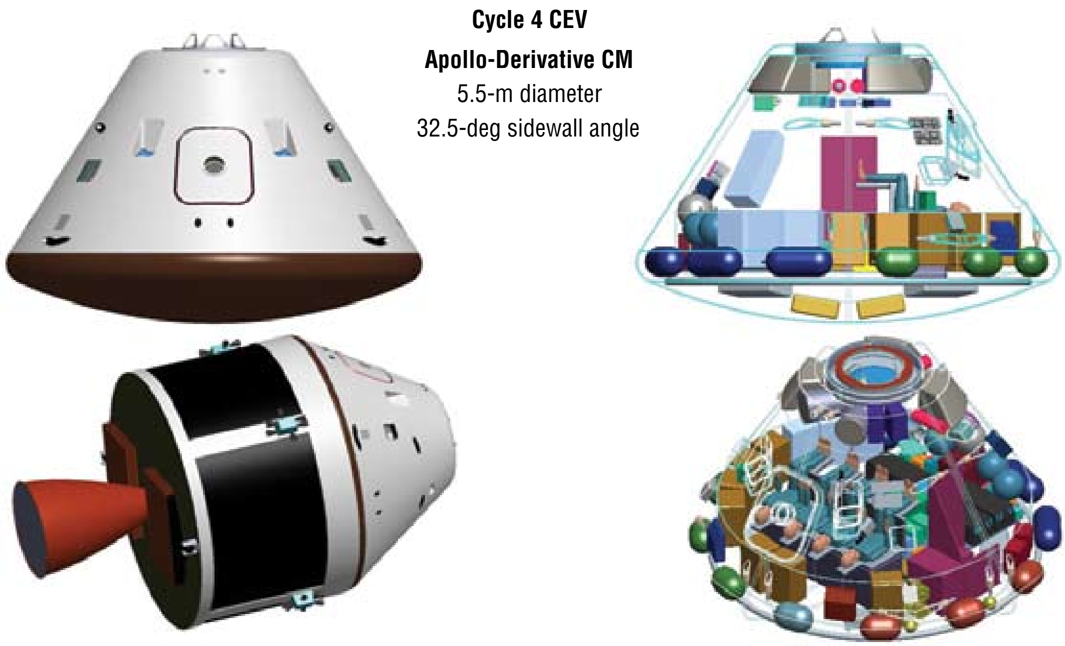


Figure 5-3. Cycle 4 CEV CM

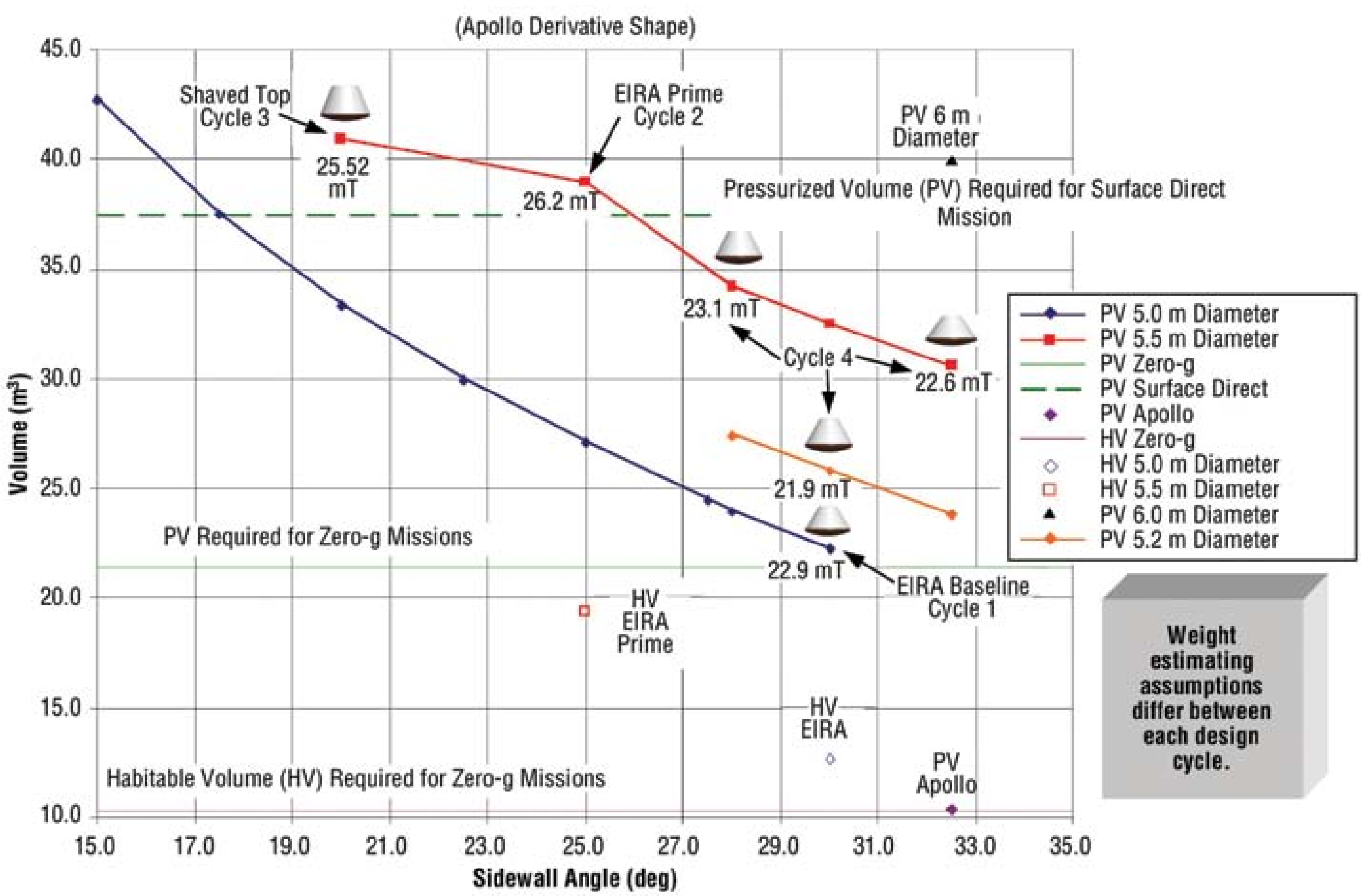


Figure 5-4. CEV Volume Relationships

5.2.3 Block 2 - Lunar CEV

5.2.3.1 Lunar CEV CM

5.2.3.1.1 Vehicle Description

The lunar CEV CM, in conjunction with the SM and LV/EDS, is used to transport four crew members from Earth to lunar orbit and return the crew members to Earth. The CM provides habitable volume for the crew, life support, docking and pressurized crew transfer to the LSAM, and atmospheric entry and landing capabilities. Upon return, a combination of parachutes and airbags provide for a nominal land touchdown with water flotation systems included for water landings following an aborted mission. Three main parachutes slow the CEV CM to a steady-state sink rate of 7.3 m/s (24 ft/s), and, prior to touchdown, the ablative aft heat shield is jettisoned and four Kevlar airbags are deployed for soft landing. After recovery, the CEV is refurbished and reflown with a lifetime up to 10 missions.

A scaled Apollo Command Module shape with a base diameter of 5.5 m and sidewall angle of 32.5 deg was selected for the OML of the CEV CM. This configuration provides 29.4 m^3 of pressurized volume and 12–15 m^3 of habitable volume for the crew during transits between Earth and the Moon. The CEV CM operates at a nominal internal pressure of 65.5 kPa (9.5 psia) with 30 percent oxygen composition for lunar missions, although the pressure vessel structure is designed for a maximum pressure of 101.3 kPa (14.7 psia). Operating at this higher pressure allows the CEV to transport crew to the ISS without the use of an intermediate airlock. For the lunar missions, the CM launches with a sea-level atmospheric pressure (101.3 kPa), and the cabin is depressurized to 65.5 kPa prior to docking with the LSAM.

The lunar CEV CM propulsion system provides vehicle attitude control for atmospheric entry following separation from the SM and range error corrections during the exoatmospheric portion of a lunar skip-entry return trajectory. A gaseous oxygen/ethanol bipropellant system is assumed with a total delta-V of 50 m/s.

Illustrations of the reference lunar CEV CM are shown in **Figure 5-5**.

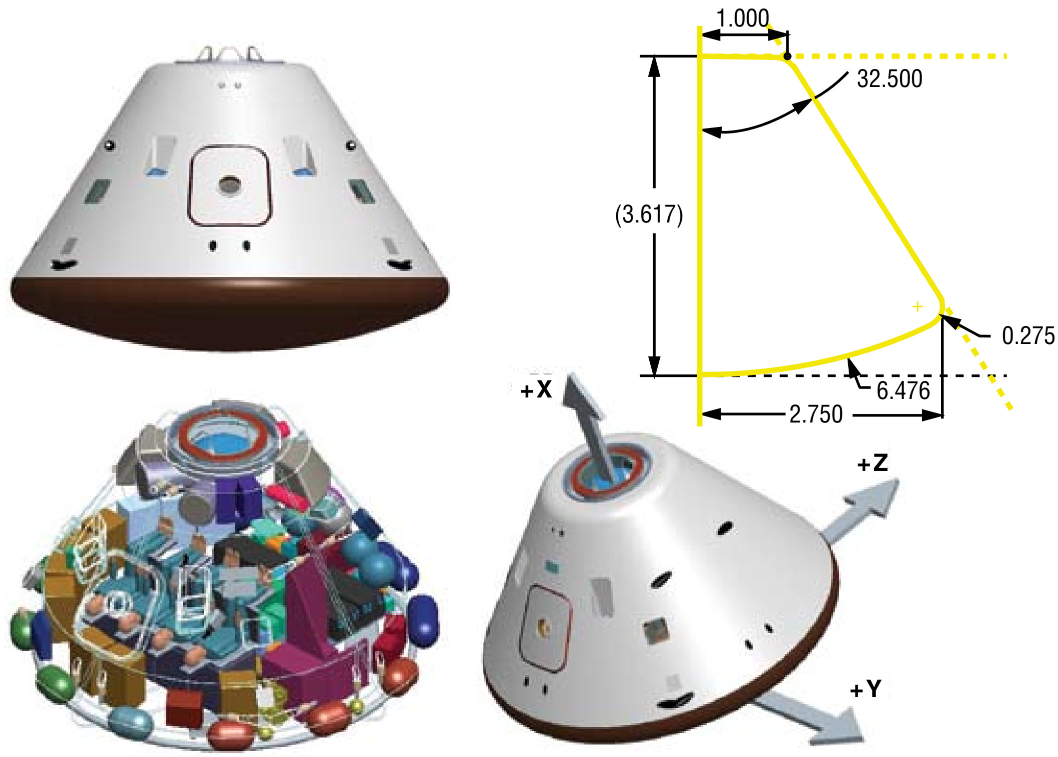


Figure 5-5. Reference Lunar CEV CM

5.2.3.1.2 Overall Mass Properties

Table 5-1 provides overall vehicle mass properties for the lunar CEV CM. The mass properties reporting standard is outlined in JSC-23303, Design Mass Properties. A detailed mass statement is provided in **Appendix 5A**, **CEV Detailed Mass Breakdowns**.

Table 5-1.
Vehicle Mass Properties for the Lunar CEV CM

Lunar CEV CM	% of Vehicle Dry Mass	Mass (kg)	Volume (m^3)
1.0 Structure	23%	1,883	0
2.0 Protection	11%	894	1
3.0 Propulsion	5%	413	1
4.0 Power	10%	819	1
5.0 Control	0%	0	0
6.0 Avionics	5%	435	1
7.0 Environment	14%	1,091	4
8.0 Other	14%	1,159	2
9.0 Growth	17%	1,339	2
10.0 Non-Cargo		821	3
11.0 Cargo		100	1
12.0 Non-Propellant		367	0
13.0 Propellant		184	0
Dry Mass	**100%**	**8,034 kg**	
Inert Mass		**8,955 kg**	
Total Vehicle		**9,506 kg**	

5.2.3.1.3 Subsystem Description

Structure

The CEV CM structure includes vehicle primary structures and consists of the following components:

- Pressure vessel structure,
- Windows, and
- OML unpressurized structure.

The selected shape for the CEV CM is the Apollo Command Module shape scaled in dimension by approximately 141 percent to a base diameter of 5.5 m (18 ft), while the original Apollo Command Module sidewall angle of 32.5 deg has been maintained for this analysis. Selecting this shape provides a total CEV pressurized volume of 29.4 m^3 (1,038 ft^3).

The CEV pressure vessel structure provides habitable volume for the crew and enclosure for necessary systems of the CEV through ascent until rendezvous with the LSAM in LEO, through transit to the Moon and transfer to the LSAM in lunar orbit, and through undocking from the LSAM until reentry and crew recovery on Earth. The CEV CM pressure vessel structure construction is an Aluminum (Al) honeycomb sandwich using materials such as Al 2024 or the equivalent for the face sheets and Al 5052 for the honeycomb core. The mass-estimating method used for estimating pressure vessel structure (including secondary structure) in this assessment was to assume a uniform structure mass per unit area and scale by the external surface area of the pressure vessel. The assumed scaling factor for aluminum honeycomb is 20.3 kg/m^2 (4.15 lb/ft^2) and the surface area of the pressure vessel less windows and hatches is 52.7 m^2. The pressure vessel structure mass for the CEV was designed to withstand a higher 14.7 psia nominal internal cabin pressure required for ISS crew rotation missions instead of the lower 9.5 psia nominal internal pressure for lunar missions.

Five windows are included on the CEV for rendezvous and docking operations, observation, and photography. Two forward-facing windows on the vehicle sidewalls provide a view toward the apex of the CM for docking with the ISS and the LSAM, while two side windows and a fifth circular window located within the side ingress/egress hatch provide additional external views. The windows are double-paned fused silica panels similar to the optical windows on the Shuttle Orbiter.

The OML for the CEV CM is composed of graphite epoxy/Bismaleimide (BMI) composite skin panels similar to those developed for the X–37 Approach and Landing Test Vehicle (ALTV). This structure provides the vehicle's aerodynamic shape and serves as the attachment structure for windward and leeward TPS. The mass-estimating method used for estimating OML structure mass in this assessment was to assume a uniform structure mass per unit area and scale by the external surface area of the outer structure. The assumed scaling factor for composite skin panels, including attachment structure, is 11.6 kg/m^2 (2.38 lb/ft^2), and the surface area of the OML, less windows and hatches, is 66.9 m^2. Graphite epoxy/BMI has a maximum service temperature of 450°K (350°F) for aerothermal analysis.

Protection

The CEV CM spacecraft protection consists of the materials dedicated to providing passive spacecraft thermal control during all mission phases including ascent, ascent aborts, in-space operations, and atmospheric entry, and includes the following components: External TPS and internal insulation.

For the CEV CM, spacecraft protection is the TPS that includes ablative TPS on the windward (aft) side of the vehicle, reusable surface insulation for the external leeward (central and forward) TPS, and internal insulation between the pressurized structure and OML. There are a number of potential materials available for use in the CEV CM protection system and the eventual TPS materials selected will be the result of a rigorous trade study based on performance and cost. Some of these materials may include carbon-carbon, carbon-phenolic, AVCO, Phenolic Impregnated Carbonaceous Ablator (PICA), PhenCarb-28, Alumina Enhanced Thermal Barrier-8 (AETB–8))/TUFI, Advanced Flexible Reusable Surface Insulation (AFRSI, LI-900 or LI-2200, CRI, SLA-561S, cork, and many others.

TPS mass for the present CEV CM concept is scaled from an analysis conducted for a vehicle of the same base diameter but lower sidewall angle and higher mass at Entry Interface (EI). A 5.5-m, 28-deg sidewall concept with a total mass of approximately 11,400 kg requires an aft TPS mass of 630 kg and forward TPS mass of 180 kg. The assumed TPS materials for this analysis were PICA for the aft side and a combination of LI-2200, LI-900, AFRSI, and Flexible Reusable Surface Insulation (FRSI) at equal thicknesses for the central and forward side. The maximum heating rate for the TPS is driven by ballistic entry trajectories at lunar return speeds (11 km/s), and TPS thickness is sized by the total integrated heat load of a skip-entry trajectory. For the lighter 5.5-m, 32.5-deg CM, the 630-kg aft TPS mass from the larger, heavier concept has been retained to provide additional margin, while the central and forward TPS mass has been scaled based on the lower surface area. The current CEV CM mass, including external TPS, is 9,301 kg at atmospheric EI for the nominal lunar mission.

Finally, the mass-estimating method used for internal insulation was to assume Saffil high-temperature fibrous alumina insulation wrapped around the exterior of the CM pressure vessel at a mass penalty of 2 kg/m^2 of surface area. The pressure vessel external surface area is 52.7 m^2.

Propulsion

The CEV CM propulsion consists of an RCS and includes the following components:

- Primary RCS thrusters,
- Primary RCS tanks,
- Primary RCS pressurization,
- Backup RCS thrusters, and
- Backup RCS tanks.

The CEV CM propulsion RCS provides vehicle attitude control following SM separation through atmospheric entry. Following SM separation, the vehicle is reoriented using the primary RCS to a proper attitude for entry; and, during atmospheric flight, the RCS provides roll torque to control the direction of the CM lift vector and to counteract induced spin torques, provides dampening of induced pitch and yaw instabilities, and corrects range dispersions during skip-out portions of a lunar skip return trajectory. A backup, fully independent RCS is also included on the CEV to provide emergency attitude control and a ballistic entry mode in the event of complete loss of primary power and attitude control during entry. A ballistic entry is a non-lifting flight mode where a controlled roll rate is introduced to the vehicle to effectively null the net lift vector, thereby avoiding "lift vector down" flight modes that may exceed maximum crew g-loads and TPS temperature limits during lunar return.

The assumed primary RCS propulsion system for the CEV CM is a Gaseous Oxygen (GOX) and liquid ethanol bipropellant system selected for its nontoxicity and commonality with the life support system's high-pressure oxygen supply system. A similar system has been developed and ground-tested for potential use as a Shuttle Orbiter RCS replacement and for attitude control use on the Kistler K–1 LV. The system consists of twelve 445 N (100 lbf) thrusters arranged to thrust in the pitch, roll, and yaw directions, with two thrusters pointed in each of the six directions (+pitch, –pitch, +roll, –roll, +yaw, –yaw). The assumed Specific Impulse (Isp) for the RCS system is 274 sec at a chamber pressure of 300 psia, oxidizer to fuel mixture ratio of 1.4:1 by mass, and nozzle area ratio of 40:1. The Oxygen (O2) gas for the CM primary RCS and life support system is stored in four cylindrical 5,000 psia graphite composite overwrapped-Inconel 718-lined tanks mounted at the CM base, exterior to the crew pressure vessel. Each tank has an outer diameter of 0.39 m and total length of 0.96 m, and holds 0.092 m^3 (5,553 in^3), or 43 kg, of oxygen. The liquid ethanol for the primary RCS is stored in two cylindrical graphite composite overwrapped-Inconel 718-lined bellows tanks of the same size as the tanks used to store the Nitrogen (N2) gas required for the CEV life support system. Each tank has an outer diameter of 0.39 m and total length of 0.66 m, and holds 0.053 m^3 (3,230 in^3), or 39 kg, of ethanol.

Ethanol tank pressure for the primary RCS is regulated using a high-pressure Gaseous Helium (GHe) pressurization system. Two spherical 6,000 psia tanks hold the required helium gas, 0.4 kg per tank, and have outer diameters of 0.19 m each. The tanks are the same construction as the RCS propellant tanks—graphite composite overwrapped with Inconel 718 liners.

The backup RCS is a fully independent CEV attitude control system and is used to provide emergency vehicle attitude control following complete loss of the primary system. The backup system may be used to reorient the vehicle from an "apex forward" to a "heat shield forward" configuration for entry, or to induce a slow roll rate for an emergency zero-lift ballistic entry flight mode. In the former scenario, the CEV CM, much like the Apollo Command Module, may be bi-stable and have a secondary trim point where the vehicle apex points during entry in the direction of the velocity vector. Such an orientation is clearly undesirable, as the CEV would be unable to withstand the intense heat of atmospheric entry. If the vehicle's CG can be lowered close enough to the aft heat shield, this trim point can be eliminated and the vehicle will have a single trim point (monostable) where the heat shield points toward the velocity vector. Therefore, for a given range of initial vehicle state conditions at entry (e.g., static with apex forward, 3-axis tumbling, etc.), a monostable CEV would eventually trim in the proper orientation due to the pitching moment characteristics of the vehicle. Depending on the initial vehicle state, however, a monostable vehicle may take longer to trim at the proper angle of attack than would be allowed before the onset of induced aerothermal heating exceeded vehicle temperature limits. Thus, while a monostable CEV CM is highly desired, a backup attitude control capability is required. In addition, a monostable vehicle could still trim at an angle of attack that pointed the lift vector down, and, for that possibility, the backup attitude control system can induce a slow, lift-nulling roll rate for a zero-lift ballistic mode.

GOX and liquid ethanol are also used as propellants for the backup RCS. The system, which for simplicity operates in blowdown mode instead of being helium-pressure-regulated, consists of four 445 N (100 lbf) thrusters arranged near the CM apex to thrust in the pitch and roll directions, with two thrusters each pointing in the +pitch and –pitch directions. To induce a roll moment, the +pitch/–Z thruster fires in tandem with the –pitch/+Z thruster, or vice versa. Pitching moments are generated by firing both +pitch or –pitch thrusters in tandem. The backup RCS thrusters are identical to the primary system. Oxygen gas for the CM backup RCS is stored in a single cylindrical 5,000 psia graphite composite overwrapped Inconel 718-lined tank identical to the oxygen tanks for the primary system. The tank has an outer diameter of 0.39 m and total length of 0.96 m. The liquid ethanol for the backup RCS is stored in a single cylindrical graphite composite overwrapped Inconel 718-lined diaphragm tank, again identical to the primary ethanol tanks. The tank has an outer diameter of 0.39 m and total length of 0.66 m.

There are several propellant alternatives to GOX/ethanol also worthy of consideration for the CEV CM propulsion system. These include, but are not limited to, Tridyne, GOX/Gaseous Methane (GCH4), monopropellant hydrazine, monopropellant Hydrogen Peroxide (H2O2), Nitrous Oxide (N2O), Nitrogen Tetroxide (NTO)/Monomethyl Hydrazine (MMH), cold gas nitrogen, and monopropellant Hydroxyl Ammonium Nitrate- (HAN-) based propellants. A warm gas Tridyne system is particularly attractive for the CM but was considered infeasible due to the high delta-V currently associated with the lunar skip-entry.

Power

The power subsystem for the CEV CM encompasses the primary electrical power and distribution and energy storage functions for the CEV and includes the following components:

- Rechargeable Lithium-ion (Li-ion) batteries for primary power,
- 28 Volts Direct Current (VDC) electrical power buses,
- Power control units,
- Remote PCUs, and
- Backup battery.

Four rechargeable Li-ion batteries provide CEV power during LEO and lunar orbit eclipse periods and power following CM–SM separation through landing. These batteries were selected for their high specific energy and volume, low drain rate, long wet life, and good charge retention. The total CM energy storage requirement is 6.0 kW (the CEV's maximum average power for the mission) for 2.25 hours (the time from SM separation to landing). Three batteries are sized to meet this 13.5 kW-hr requirement with a fourth battery included for redundancy. Including power management and distribution losses (10 percent) and a battery depth-of-discharge of 80 percent, each of the four batteries is sized to store a maximum of 223.2 Amp-hr at 28 VDC. Battery mass and volume were estimated using linear scaling factors for rechargeable Li-ion batteries, 100 W-hr/kg and 200 W-hr/L, respectively. The total battery mass was further increased by 10 percent for battery installation.

The four Li-ion batteries, in conjunction with two solar arrays mounted on the SM, provide electrical power to the CEV power distribution system. The primary power distribution system then distributes 28 VDC power to the vehicle across three main distribution buses, with each main bus sized to handle the peak electrical load for two-fault tolerance. CEV average power for the entire mission with crew on board is 4.5 kW, with a peak power of 8 kW. The wiring harness for the electrical power distribution system consists of primary and secondary distribution cables, jumper cables, data cabling, RF coaxial cable, and miscellaneous brackets, trays, and cable ties. Mass for the entire CM wiring harness, including electrical power and avionics wiring and associated items, is estimated at 317 kg.

Power Control Units (PCUs) on the CEV CM monitor and control current from the solar arrays and batteries and distribute power among the vehicle loads. A PCU includes the relays, switches, current sensors, and bus interfaces necessary to control and distribute power, as well as solar array switch modules and battery charge modules for monitoring and regulating output current. There are three PCUs included in the CEV CM (one per bus), with each unit capable of switching 160 amps at 28 VDC continuously (4,500 W) or 285 Amps at 28 VDC over a short duration (8,000 W). PCUs have an estimated mass of 41.1 kg each.

Remote Power Control Units (RPCUs) monitor and control power from the PCUs and distribute 28 VDC power to vehicle loads. Each unit has an estimated mass of 32.6 kg each and three units are included on the CEV CM (one per bus).

The CEV also includes a single rechargeable Li-ion backup battery for emergency power during ballistic entry modes. In the event of complete loss of primary power during entry, the backup battery supplies 500 W of 28 VDC power for 45 minutes.

Control

Items typically included in the spacecraft control category are aerodynamic control surfaces, Thrust Vector Control (TVC), actuators, cockpit controls such as rudder pedals, and others. There are no control components on the CEV CM.

Avionics

The CEV CM avionics subsystem provides Command and Control (C&C) over all CEV operations and consists of the following components:

- Command, Control, and Data Handling (CCDH);
- Guidance and navigation;
- Communications; and
- Cabling and instrumentation.

CCDH includes the components necessary to process and display flight-critical spacecraft data and collect crew input. These components on the CEV CM include: four flight critical computers for implementing dual fault-op tolerant processing, eight data interface units to collect and transmit data, two multifunction liquid crystal displays and two control panel sets to provide a crew interface for system status and command input, and two sets of translational/rotational/throttle hand controllers to provide manual vehicle flight control. Masses for CCDH components are derived from estimates for X–38 or commercially available hardware.

Guidance and navigation comprises the equipment needed to provide on-orbit vehicle attitude information for the CEV, perform vehicle guidance and navigation processing, and execute AR&D. This includes an integrated Global Positioning System (GPS)/Inertial Navigation System (INS), including four space-integrated GPS/INS units, one GPS combiner unit and four GPS antennas; two star trackers; and two video guidance sensors and two Three-Dimensional (3–D) scanning Laser Detection and Ranging (LADAR) units to provide AR&D capability.

The communications and tracking subsystem consists of the equipment for the CEV CM to provide communications and tracking between other architecture elements and to the ground. Information on the communication links will include command, telemetry, voice, video, and payload data. Assumed communications components are: S-band/Search and Rescue Satellite-aided Tracking (SARSAT)/Ultrahigh Frequency Television (UHF) communications systems, network signal processors, information storage units, a Television (TV)/video system, an operations recorder, and a digital audio system. A high data rate Ka-band communications system is included on the SM.

Avionics instrumentation for the CEV CM consists of instrumentation to collect spacecraft health data and includes 120 sensor clusters at 0.29 kg per cluster.

Environment

The CEV environment components consist of the equipment needed to maintain vehicle health and a habitable volume for the crew and include the following:

- Environmental Control and Life Support (ECLS);
- Active Thermal Control System (ATCS); and
- Crew accommodations.

Environmental Control and Life Support (ECLS)

Items included in ECLS are nitrogen storage, oxygen storage, atmosphere supply and control, atmosphere contaminant control, fire detection and suppression, venting and thermal conditioning, water management, and Extra-Vehicular Activity (EVA) umbilicals and support. The assumed cabin pressure for the lunar CEV CM is 65.5 kPa (9.5 psia) with nitrogen and oxygen partial pressures of 43.90 kPa (67 percent) and 19.65 kPa (30 percent), respectively.

The CM includes the atmosphere gases needed for a nominal 13.3 days of crew time in the CEV. Thirty-two (32) kg of Gaseous Nitrogen (GN2) for cabin atmosphere makeup is stored in two cylindrical 5,000 psia graphite composite overwrapped Inconel 718-lined tanks with outer diameters of 0.39 m and lengths of 0.66 m. GOX for one full contingency cabin atmosphere repressurization and nominal crew metabolic consumption (0.8 kg per crew member per day) is stored in the four primary RCS oxygen tanks.

Environment atmosphere supply and control includes the components needed to regulate and distribute oxygen and nitrogen, monitor and control atmospheric pressure, and provide atmosphere relief and venting. Masses and volumes for these items are taken directly from the Space Shuttle Operations Data Book.

The chosen systems to provide atmosphere contaminant control on the CEV CM are a combined regenerative Carbon Dioxide (CO2) and Moisture Removal System (CMRS) for CO2 control, ambient temperature catalytic oxidation (ATCO) for trace contaminant control, and O2/CO2 sensors for atmosphere contaminant monitoring. The mass for the CMRS is scaled from improved Shuttle Regenerative CO2 Removal System (RCRS) heritage data based on the required CO2 removal rate for six crew members, while masses for other atmosphere contaminant control are taken directly from Shuttle heritage components. The CMRS is internally redundant.

Fire detection and suppression on the CEV consists of smoke detectors, a fixed halon fire suppression system, and halon fire extinguishers. Masses and volumes for these components are taken directly from ISS heritage.

Atmosphere venting and thermal conditioning includes cabin fans, air ducting, and humidity condensate separators. Cabin fans and air ducting mass, power, and volumes are scaled from Shuttle data based on the CEV pressurized volume, while the humidity condensate separator is identical to that of the Shuttle.

For the CEV CM, water management includes the tanks and distribution lines necessary to hold potable water for crew consumption, water for the fluid evaporator system, and waste water. Four spherical metal bellows water tanks, pressurized with GN2, are sized to store the mission's potable water supply with a diameter of 0.47 m per tank. The tanks are similar to the Shuttle's potable water tanks and each hold 0.053 m^3 (3,217 in^3) or 53 kg of water. A single waste water tank stores up to 25 kg of waste water and is periodically vented to space. The waste water tank is identical in size and construction to the potable water tanks.

The final components in ECLS are the umbilicals and support equipment needed to support contingency EVAs and suited crew members inside the CEV CM. The assumed EVA method is for the four CEV crew members to don their launch and entry suits, fully depressurize the CEV cabin, and egress from the side or docking hatch in the same manner as was done in the Gemini or Apollo programs. Umbilicals connect the in-space suits to the CM life support system. In the event of an unplanned cabin depressurization, the life support system must support the crew via EVA umbilicals until the internal atmosphere has been restored or the vehicle has returned to Earth. For this, a suit oxygen supply assembly and suit ventilation manifold system has been included.

Active Thermal Control System (ATCS)

Active thermal control for the CEV is provided by a single-loop propylene glycol fluid loop with a radiator and a fluid evaporator system. The fluid loop heat rejection system includes cold plates for collecting CM equipment waste heat, a cabin heat exchanger for atmosphere temperature control (sized for six crew members), a Ground Support Equipment (GSE) heat exchanger for vehicle thermal control while the CEV is on the launch pad, a Liquid-Cooled Ventilation Garment (LCVG) heat exchanger for suit cooling, fluid pumps, fluid lines, and radiator panels. A total heat load of 6.25 kW is assumed for the ATCS, with 5.0 kW collected by the internal cold plates, 0.75 kW collected by the cabin air heat exchanger, and 0.5 kW collected by external cold plates (if necessary). The GSE heat exchanger transfers 6.25 kW of vehicle heat to the ground.

The assumed working fluid for the fluid loop system is a 60 percent propylene glycol/40 percent water blend, selected for its low toxicity and freeze tolerance. Two continuous single-phase fluid loops pump the propylene glycol/water blend through the cabin cold plates and heat exchangers, exit the pressure vessel to external cold plates (if necessary), and finally pump the fluid to the SM radiators where the heat is radiated away. The loop temperature is 308 K prior to entering the radiator and 275 K after exiting. Each loop contains two pumps, with one primary and one backup pump package, and each loop is capable of transporting the entire 6.25 kW heat load. The CEV CM portion of the TCS includes the mass and volume for the pumps, cold plates, lines, and heat exchangers, while the radiators are mounted on the structure of the SM.

The ATCS for the CEV CM also includes a dual-fluid evaporator system to handle peak heating loads in excess of the 6.25 kW maximum capacity of the fluid loop and to reject up to 6 kW of CM waste heat for the 2.25 hours from SM separation to landing. The fluid evaporator system operates by boiling expendable water or Freon R–134A in an evaporator to cool the heat rejection loop fluid, which is circulated through the walls of the evaporator. Generated vapor is then vented overboard. A dual-fluid system for the CEV is required because water does not boil at ATCS fluid loop temperatures and atmospheric pressures found at 100,000 ft or less; therefore, the nontoxic fluid Freon R–134A is used for vehicle cooling from that altitude to the ground. The Apollo Command Module did not provide cooling after water boiling became ineffective; however, that may not be appropriate for the CEV since the vehicle lands on land (i.e., the Command Module relied in part on the water landing for post-landing cooling), the CEV is nominally reusable (i.e., the Command Module was expendable), and the assumed heat load is higher. The mass estimate for the fluid evaporator system is based on the Shuttle Fluid Evaporator System (FES), scaled linearly using the heat capacity of that system. FES water is stored with the ECLSS potable water supply, while Freon R–134A is stored in a single 0.47–m diameter metal bellows tank.

Crew Accommodations
The crew accommodations portion of the CEV CM includes a galley, a Waste Collection System (WCS), Cargo Transfer Bags (CTBs) for soft stowage, and seats. For the galley, a water spigot and Shuttle-style food warmer are included to prepare shelf-stable and freeze-dried packaged foods. The mass for these items is taken from Shuttle heritage equipment. The CEV galley also includes accommodations for cooking/eating supplies and cleaning supplies, which are estimated at 0.5 kg per crew member and 0.25 kg per day, respectively.

The assumed WCS for the CM is a passive Mir Space Station-style toilet/commode with appropriate supplies, a privacy curtain, and contingency waste collection bags. In the Mir-style commode, wastes are deposited in a bag-lined can with a suitable user interface. The bags can then be individually isolated and stored in an odor control container. Alternate methods for waste collection could include urine collection devices (Shuttle), bags (Apollo), an active WCS (Shuttle), or personal urine receptacles.

CTBs are used on the CEV to provide soft stowage capability for crew accommodations equipment. Each CTB holds 0.056 m^3 (2 ft^3) of cargo and 26 bags are required for the vehicle.

For seats, four removable/stowable crew couches are included on the CEV CM for launch and landing with 10 inches of seat stroking under the seats for impact attenuation. Specifically, the seats stroke 10 inches at the crew member's feet, 5 inches at the head, 5 inches above the crew member, and 5.5 inches to the sides. The mass for the crew couches, taken from the Apollo Command Module, is scaled by 133 percent to accommodate a fourth crew member.

Other
CEV CM components included in the "Other" category are:

- Parachutes,
- Parachute structure and release mechanisms,
- Shell heaters,
- Landing airbags,
- Water flotation system,
- Doors and hatches, and
- Docking mechanism.

The CEV CM parachute system is comprised of three round main parachutes, two drogue parachutes, three pilot parachutes, and parachute structure and release mechanisms. Parachutes are packed between the CM pressure vessel and OML near the CEV docking mechanism. The three main parachutes, 34 m (111 ft) in diameter each, are sized to provide a nominal landing speed of 24 ft/s with all three parachutes deployed and a landing speed of 29.5 ft/s with one failed parachute. Main parachutes deploy at a dynamic pressure of 30 psf (10,000 ft altitude and 126 mph sink rate) and have a CM suspended mass of 8,654 kg. The two drogue parachutes, 11 m (37 ft) in diameter, stabilize and decelerate the CEV CM from a deployment dynamic pressure of 78 psf (23,000 ft altitude and 252 mph) to the main parachute deployment at 30 psf. Each drogue parachute is individually capable of slowing the CEV to the desired main parachute deployment sink rate. Once that dynamic pressure is reached, the drogue parachutes are pyrotechnically severed and the main parachutes are simultaneously deployed by the three pilot parachutes. Finally, mass is included in the CEV CM for parachute structure and release mechanisms. This mass is estimated as a fixed percentage (22.5 percent) of the main, drogue, and pilot parachute total mass.

The chosen landing mode for the CEV CM is a land landing with four inflatable Kevlar airbags for impact attenuation. Prior to touchdown, the CM aft heat shield is jettisoned and the airbags are inflated with compressed nitrogen gas. The airbags, which are mounted between the pressure vessel and aft heat shield, include both inner and outer bags, with the outer bags deflating after impact while the inner airbags remain inflated for landing stability. Airbags are sized for a worst-case impact speed of 29.5 ft/s with one failed main parachute. The total impact attenuation system includes the airbags, the airbag inflation system, and airbag controls. One cylindrical high-pressure GN2 tank identical to the ECLSS nitrogen tanks holds the gas used for inflating the airbags, with the tank having an outer diameter of 0.39 m and total length of 0.66 m. The four airbags have a stowed volume of 0.095 m^3 at a packing density of 498 kg/m^3.

A water flotation system is also included in the CEV CM to assure proper vehicle orientation in the event of a water landing. The flotation system allows the CM to self-right for safe vehicle and crew extraction by recovery forces.

The CEV CM also includes miscellaneous doors and hatches for crew access and vehicle servicing. An ingress/egress hatch provides a means for vehicle entry and exit while the vehicle is on the launch pad and is identical in size and mass to the Apollo Command Module hatch (29 inches x 34 inches). As part of the LIDS mechanism, a 32-inch docking adapter hatch provides a secondary egress path from the vehicle and is the means for pressurized crew transfer between two spacecraft. The CEV also includes two passive vent assemblies for purge, vent, and thermal conditioning of enclosed unpressurized vehicle compartments. Finally, umbilical and servicing panels allow for fluid loading on the launch pad.

The other CEV CM component assumed in this category is the androgynous LIDS mechanism for mating with the ISS and other exploration architecture elements. The LIDS on the CEV includes the docking mechanism and LIDS avionics. A flight-qualified LIDS has an estimated mass of 304 kg.

Growth

A 20 percent factor for potential vehicle mass growth is included here, applied to all dry mass components.

Non-Cargo

Non-cargo for the CEV CM consists of the following components:

- Personnel,
- Personnel provisions, and
- Residual propellant.

The CEV CM is capable of carrying four persons to the Moon for lunar exploration missions. A mass estimate for a crew of four is included in the vehicle, assuming the mass (100 kg) of a 95th percentile male crew member.

CEV personnel provisions for the DRM include the following:

- Recreational equipment consists of crew preference items and is estimated at 5 kg per crew member;
- Crew health care includes basic medical, dental, and surgical supplies, and four emergency breathing apparatuses;

- Personal hygiene includes basic hygiene kits and consumables for the mission;
- Clothing includes multiple clothing sets for the four crew members at 0.46 kg per crew member per day;
- Housekeeping supplies include a vacuum, disposable wipes for spills, and trash bags;
- Operational supplies include basic operational supplies estimated at 5 kg per crew member; CEV CM lighting (10 kg); zero-g restraints (12 kg); emergency egress kits for pad aborts at 2.3 kg per crew member; a sighting aid kit for dockings including a Crew Optical Alignment Sight (COAS), binoculars, spotlights, etc. (13 kg); and a crew survival kit including beacons, transponders, a life raft, etc. (44 kg);
- Maintenance equipment includes a basic Shuttle-style in-flight maintenance toolkit;
- Sleep accommodations are zero-g sleep aids estimated at 2.3 kg per crew member;
- EVA suits and spares include Gemini-style launch and entry suits capable of performing emergency EVAs. The assumed EVA mode for the CEV CM is to fully depressurize the CEV pressure vessel with all four crew members donning their EVA suits. Each suit is estimated at 20 kg per crew member; and
- Food for the crew is estimated at 1.8 kg per crew member per day.

Residual propellant on the CEV CM is the trapped ethanol and GOX propellant remaining in the propulsion tanks after completion of the nominal delta-V maneuvers. Residuals for RCS propellants are 2 percent of the nominally consumed propellant. Pressurant is the GHe needed to pressurize the ethanol primary RCS tanks.

Cargo

Cargo for the CEV CM consists of the following components: Ballast.

Ballast mass is included in the CEV CM to ensure a proper vehicle CG location prior to atmospheric entry. The ultimate ballast mass requirement will be the product of a detailed aerodynamic and vehicle mass properties study, but a placeholder mass of 100 kg is included in the CEV CM mass estimate until such analyses can be completed.

Non-Propellant

Non-propellant for the CEV CM consists of the following components:

- Oxygen,
- Nitrogen,
- Potable water, and
- FES water and freon.

Oxygen gas is included in the CEV for breathing gas makeup, contingency EVA consumption, atmosphere leakage and venting, and one contingency full-cabin repressurization. The total oxygen mass requirement is estimated at 64 kg for the lunar mission. An alternative to storing oxygen in the CM would be to use the service propulsion system/RCS oxygen tanks in the SM for shared storage; however, that option was not pursued since the CM primary RCS oxygen tanks provide a convenient source of high-pressure GOX.

The amount of nitrogen gas required for the CEV CM atmosphere is estimated using assumptions for cabin leak rate (0.15 kg/day), waste management and regenerative CO2 system venting, and the number of full-cabin repressurizations (one). A nitrogen partial pressure of 43.9 kPa is assumed, with a total cabin pressurized volume of 29.4 m^3 and cabin temperature of 21°C, for a total nitrogen mass requirement of 32 kg.

CM potable water requirements are estimated to supply (1) water for Intra-Vehicular Activity (IVA) crew water usage, (2) EVA water for contingency EVAs, and (3) water for the CEV CM's water evaporator system. IVA crew water usage for drinking water, food preparation water, and hygiene water is included at a consumption rate of 3.5 kg per crew member per day, with 53 crew-days required for the mission. Consumable water is also included for the ATCS's FES, which is sized to reject 37,800 kJ of heat (35,827 Btu) from the time of SM separation to 100,000 ft. FES water requirements are estimated assuming a heat of vaporization of 2,260 kJ/kg and 20 percent margin for consumables.

Once the CEV reaches an altitude where water boiling is no longer effective, the FES switches to using freon R–134A for cooling. The Freon consumable mass is sized to reject 10,800 kJ of heat (10,236 BTU) from 100,000 ft to post-landing vehicle shutdown. FES freon requirements are estimated assuming a heat of vaporization of 216 kJ/kg and 20 percent margin for consumables.

Propellant
Propellant for the CEV CM consists of the following components: Used RCS propellant.

Primary RCS propellant on the CEV CM is used to reorient the vehicle to a proper attitude for entry and, during atmospheric flight, the RCS provides roll torque to control the direction of the CM lift vector and counteract induced spin torques, provides dampening of induced pitch and yaw instabilities, and corrects range dispersions during skip-out portions of a lunar skip return trajectory. The assumed delta-V for these maneuvers is 10 m/s for entry maneuvering and 40 m/s for skip-out error corrections, with a thruster Isp of 274 sec and initial vehicle mass prior to entry of 9,599 kg. The CEV CM mass includes 100 kg of samples returned from the lunar surface.

The backup RCS propellant is used to reorient the vehicle to a proper trim attitude and induce a roll moment for the emergency ballistic down mode.

5.2.3.2 Lunar CEV SM

5.2.3.2.1 Vehicle Description

The Lunar CEV SM is included in the ESAS exploration architecture to provide major translational maneuvering capability, power generation, and heat rejection for the CEV CM. The SM assumes an integrated pressure-fed oxygen/methane service propulsion system and RCS to perform rendezvous and docking with the LSAM in Earth orbit, any contingency plane changes needed prior to lunar ascent, TEI, and self-disposal following separation from the CM. One 66.7 kN (15,000 lbf) service propulsion system and twenty-four 445 N (100 lbf) RCS thrusters, systems common to both the SM and the LSAM ascent stage, are used for on-orbit maneuvering. The SM propellant tanks are sized to perform up to 1,724 m/s for the service propulsion system and 50 m/s of RCS delta-V with the CEV CM attached and 15 m/s of RCS delta-V after separation. In the event of a late ascent abort off the CLV, the SM service propulsion system may also be used for separating from the LV and either aborting to near-coastline water landings or aborting to orbit.

Two deployable, single-axis gimbaling solar arrays are also included to generate the necessary CEV power from Earth-Orbit Insertion (EOI) to CM–SM separation prior to entry. For long-duration outpost missions to the lunar surface, lasting up to 180 days, the CEV remains unoccupied in lunar orbit. Solar arrays were selected instead of fuel cells or other similar power generation options because the reactant mass requirements associated with providing keep-alive power during the long dormant period for fuel cells became significantly higher than the mass of a nonconsumable system such as solar arrays. The solar arrays use state-of-the-art three-junction Photovoltaic (PV) cells. Finally, the SM composite primary structure also provides a mounting location for four radiator panels. These panels provide heat rejection capability for the CEV fluid loop heat acquisition system.

Illustrations of the reference lunar CEV SM are shown in **Figure 5-6**.

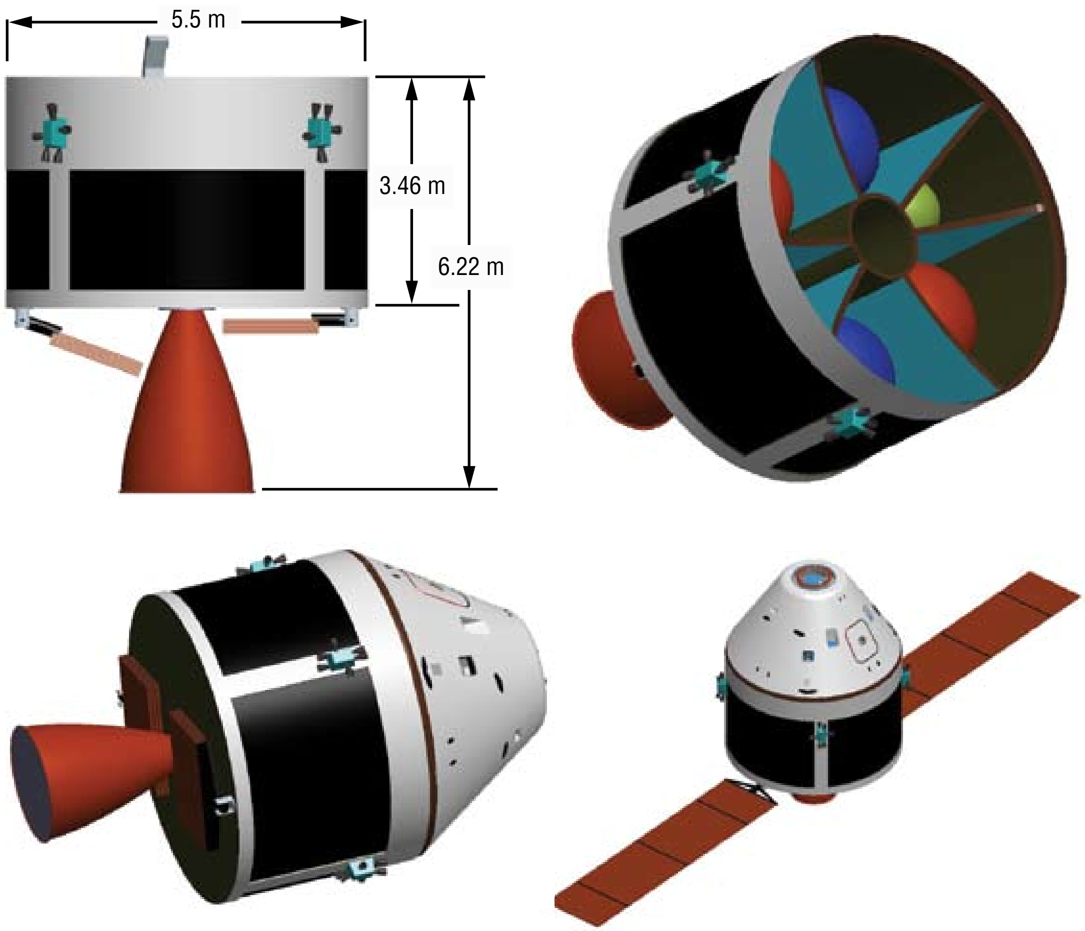


Figure 5-6. Reference Lunar CEV SM

5.2.3.2.2 Overall Mass Properties

Table 5-2 provides overall vehicle mass properties for the SM used for the lunar exploration mission. The mass properties reporting standard is outlined in JSC-23303, Design Mass Properties. A detailed mass statement is provided in **Appendix 5A**, **CEV Detailed Mass Breakdowns**.

Lunar SM	% of Vehicle Dry Mass	Mass (kg)	Volume (m^3)
1.0 Structure	20%	819	0
2.0 Protection	4%	167	1
3.0 Propulsion	36%	1,423	1
4.0 Power	10%	417	1
5.0 Control	0%	0	0
6.0 Avionics	3%	117	1
7.0 Environment	2%	98	4
8.0 Other	7%	290	2
9.0 Growth	17%	666	2
10.0 Non-Cargo		579	3
11.0 Cargo		0	1
12.0 Non-Propellant		0	0
13.0 Propellant		9,071	0
Dry Mass	**100%**	**3,997 kg**	
Inert Mass		**4,576 kg**	
Total Vehicle		**13,647 kg**	

Table 5-2. Overall Vehicle Mass Properties for the SM for the Lunar Exploration Mission

5.2.3.2.3 Subsystem Description

Structure

The CEV SM structure includes vehicle primary structure and consists of the following component: Unpressurized structure.

The CEV SM unpressurized structure provides structural attachment for the CEV power, avionics, and propulsion system components, a mounting location for body-mounted thermal control radiator panels, and an interface for mating to the CEV LV. An SM external diameter of 5.5 m was selected, equal to the diameter of the CEV CM, and the vehicle has a length for the primary structure of 3.46 m. SM structure length was driven by the length of the internal propellant tanks and required acreage for mounting four radiator panels.

The CEV SM is a semimonocoque structure, similar in design and construction to the Apollo SM. Graphite epoxy/BMI composites were selected as the structural material for mass savings, though several aluminum alloys, such as Al 2024 or Al-Li 8090, may also be considered. The mass-estimating method used for composite unpressurized structure mass in this assessment was to assume a power law relationship based on the external surface area of the SM, which is 59.8 m^2. The assumed equation for composites was: Mass = 6.6515 * (surface area)1.1506, where surface area is given in square meters and mass is calculated in kilograms. Mass was further added to the primary structure estimate to account for dedicated tank support structure. This was estimated using a linear relationship of 0.008 kg of tank support structure per kilogram of wet tank mass.

Protection

The CEV SM protection consists of the materials dedicated to providing passive spacecraft thermal control during all mission phases, including ascent and in-space operations, and includes the following component: Internal insulation

The CEV SM contains insulation blankets for passive thermal control. The mass-estimating method used for internal insulation was to assume insulation wrapped around the SM external surface area at a mass penalty of 2 kg/m^2. The unpressurized structure external surface area, including the sidewalls and base heat shield, is 83.6 m^2.

Propulsion

The CEV SM propulsion consists of an integrated service propulsion system/RCS and includes the following components:

- Service propulsion system,
- RCS thrusters,
- Service propulsion system and RCS fuel/oxidizer tanks, and
- Service propulsion system and RCS pressurization system.

The SM propulsion for performing major CEV translational and attitude control maneuvers is a pressure-fed integrated service propulsion system/RCS using LOX and Liquid Methane (LCH4) propellants. This propellant combination was selected for its relatively high Isp, good overall bulk density, space storability, nontoxicity, commonality with the LSAM, and extensibility to In-Situ Resource Utilization (ISRU) and Mars, among other positive attributes. A pressure-fed integrated service propulsion system/RCS was selected for its simplicity, reliability, and lower development cost over other comparable systems. Other tradable propellants for the CEV SM might include bipropellants such as NTO/MMH, LOX/Liquid Hydrogen (LH2), and several other LOX/hydrocarbon propellants such as ethanol or propane. Alternative system configurations might be nonintegrated versus integrated service propulsion system/RCS, pump-fed versus pressure-fed service propulsion system, and common service propulsion system/RCS propellants versus dissimilar service propulsion system/RCS propellants.

The EIRA uses CEV propulsion to rendezvous with the LSAM in LEO, perform any plane changes associated with an emergency anytime return on ascent, and return to Earth from lunar orbit regardless of orbital plane alignment. The assumed delta-Vs for these maneuvers are described below in the CEV SM propellant section.

A single fixed (non-gimbaling) oxygen/methane pressure-fed service propulsion system is included on the SM to perform major translational maneuvers while on-orbit or late-ascent orbits from the LV are necessary. The engine has a maximum vacuum thrust and Isp of 15,000 lbf (66.7 kN) and 363.6 sec, respectively. The regeneratively cooled engine operates at a chamber pressure of 225 psia and an oxygen/methane mixture ratio of 3.6:1 by mass, and has a nozzle expansion ratio of 150:1. The calculated total engine length is 3.41 m, the nozzle length is 2.76 m, and the nozzle exit diameter is 2.01 m. All engine parameters are subject to future optimization trades.

Twenty-four oxygen/methane pressure-fed RCS thrusters are also included for vehicle attitude control and minor translational maneuvers such as terminal approach during rendezvous and docking. Each engine has a maximum vacuum thrust and Isp of 100 lbf (445 N) and 317.0 sec, respectively. The RCS thrusters are film-cooled, operate at chamber pressures and mixture ratios of 125 psia and 3.6:1, and have nozzle expansion ratios of 40:1. As the RCS thrusters operate on liquid propellants, they are able to perform long steady-state burns as a service propulsion system backup, albeit at lower Isp.

Service propulsion system and RCS oxygen/methane propellants are stored in four tanks constructed with Al-Li 2090 liners and graphite epoxy composite overwrappings, with two tanks dedicated per fluid. Each oxygen tank holds 3.49 m^3 or 3,706 kg of subcooled oxygen at a nominal tank pressure of 325 psia and Maximum Expected Operating Pressure (MEOP) of 406 psia. The tanks are cylindrical with external dimensions of 1.80 m for diameter, 2.21 m for overall length, and 0.76 m for dome height. Each methane tank holds 2.63 m^3, or 1,033 kg, of subcooled fluid, has a nominal pressure and MEOP of 325 and 406 psia, respectively, and is cylindrical with external dimensions of 1.80 m for diameter, 1.81 m for overall length, and 0.76 m for dome height.

Oxygen and methane are stored entirely passively on the CEV SM. Each tank includes 60 layers of variable density Multilayer Insulation (MLI) with a total thickness of 0.041 m and a 0.025-m layer of Spray-on Foam Insulation (SOFI), which reduces the average heat leak rate per tank for oxygen and methane to 0.15 and 0.14 W/m^2, respectively. A passive thermodynamic vent system is provided on the tank to periodically vent vaporized propellants. Cryocoolers could be included in the propulsion system to remove the tank heat leak and eliminate propellant boil-off, though such a system would require power and thermal control and would increase tank cost and complexity.

The assumed pressurization system for the SM propellant tanks is GHe stored in two Inconel 718-lined, graphite epoxy composite-overwrapped 6,000 psia tanks. As propellant is consumed, the GHe is distributed to the oxygen/methane tanks to maintain a propellant tank pressure of 325 psia. To minimize helium tank size, the tanks are thermodynamically coupled to the LCH4 tank, thus reducing the helium temperature while stored to 112 K. Each helium tank is spherical with an outer diameter of 1.03 m and holds 86.2 kg of helium.

Power

The power subsystem for the CEV SM encompasses the power generation function for the CEV and includes the following components:

- Triple-junction Gallium Arsenide (GaAs) solar arrays,
- Electrical power distribution, and
- PCUs.

Two 17.9 m^2 (193 ft^2) triple-junction GaAs solar arrays provide CEV power during LEO and lunar orbit operations and during transfer between Earth and the Moon. Each solar array wing is sized to generate the full CEV average power requirement of 4.5 kW with various losses at array end-of-life. Those losses, which include a 90 percent Power Management and Distribution (PMAD) efficiency, 180-day on-orbit lifetime with 2.5 percent degradation per year, 15-deg Sun pointing loss, and 15 percent inherent array degradation, result in arrays theoretically capable of generating 6,167 W in laboratory conditions, assuming a 26 percent maximum conversion efficiency. The beginning-of-life power generation per panel once the CEV is on orbit is 5,242 W. The solar array system includes two array panels, deployment mechanisms, single axis drive actuators, and Sun sensors. Charge control and power conditioning units for the arrays are integrated into the PCUs on the CEV CM. Array system mass for the CEV was estimated for each individual component. Array panel mass was estimated using the array area (17.9 m^2) and a mass scaling factor for state-of-the-art triple-junction GaAs arrays, while other solar array system components were assumed to have masses independent of array power level.

The SM electrical power distribution and control system collects power generated by the solar arrays and distributes it as 28 VDC power to SM loads and the CM power distribution system. CEV average power for the entire mission is 4.5 kW, with the SM distribution system capable of handling a peak power of 8 kW. The wiring harness for the electrical power distribution system consists of primary distribution cables, secondary distribution cables, jumper cables, data cabling, RF coaxial cable, and miscellaneous brackets, trays, and cable ties. Mass for the entire SM wiring harness is estimated at 164 kg.

PCUs on the CEV SM monitor and control power from the solar arrays and distribute power among the vehicle loads. A PCU includes relays, switches, current sensors, and bus interfaces necessary to control and distribute power. There are two units (one primary and one backup) included in the CEV SM, with each unit capable of switching 160 amps at 28 VDC continuously (4,500 W) or 285 amps at 28 VDC over a short duration (8,000 W). PCUs have an estimated mass of 41.1 kg each.

Control

Items typically included in the spacecraft control category are aerodynamic control surfaces, actuators, cockpit controls such as rudder pedals, and others. There are no control components on the CEV SM.

Avionics

The CEV SM avionics subsystem transmits health data and commands between SM components and the CM CCDH system. SM avionics consist of the following components:

- CCDH,
- Communications, and
- Instrumentation.

CCDH on the SM includes four data interface units to collect and transmit health and status data from other SM components. Masses for data interface units are derived from estimates for other commercially available components. A 30 percent installation factor is also included.

The SM also includes a high-gain Ka-band phased array antenna system for sending and receiving high data rate information between Earth and the CEV, though the decision to locate the antenna on the CM or SM is an ongoing trade. The Ka-band antenna is currently mounted near the base (engine) of the SM structure.

Avionics instrumentation for the CEV SM includes 40 sensor clusters at 0.29 kg per cluster.

Environment

The CEV environment components consist of the equipment needed to maintain vehicle health and a habitable volume for the crew and include the following on the SM: ATCS.

Active Thermal Control System (ATCS)

Active thermal control for the CEV is provided by a single-loop propylene glycol fluid loop with radiator and an FES. All ATCS components are mounted in the CM, with the exception of the radiator panels that are mounted on the SM body structure. There are four radiator panels on the SM, each centered 90 deg apart with an area of 7.0 m^2 per panel. The radiator was sized assuming a fluid loop temperature of 275 K exiting the radiator and 308 K entering the radiator. In a worst-case vehicle attitude, two panels are viewing the Sun and two panels are out-of-Sun with a radiation sink temperature of 100 K. The maximum radiator heat load is 8.0 kW.

The assumed coating for the radiator is 10 mil silver-Teflon with a maximum absorptivity of 0.094 and emissivity of 0.888. Radiator panel mass is estimated using total panel area and a radiator mass penalty per unit area of 3.5 kg per m^2.

Other

CEV SM components included in the "Other" category are:

- CEV CM/SM attachment,
- Pyrotechnic separation mechanisms, and
- Doors and hatches.

The CEV CM/SM attachment includes structural mass for physically mating the two vehicles and umbilical lines for sharing power, fluid, and data across the vehicle interface. Mass for this component is estimated by scaling the mass for the Apollo Command Module/SM attachment system. Also included in this category are pyrotechnic separation mechanisms for initiating a mechanical separation of the two vehicles or other SM components. A mass placeholder of 100 kg is included pending further refined analysis.

The SM also includes two passive vent assemblies for purge, vent, and thermal conditioning of enclosed unpressurized vehicle compartments. Umbilical and servicing panels on the SM allow for fluid loading on the launch pad.

Growth

A 20 percent factor for potential vehicle mass growth is included here, applied to all dry mass components.

Non-Cargo

Non-cargo for the CEV SM consists of the following components:

- Residual propellant,
- Propellant boil-off, and
- Pressurant.

Residual propellant on the CEV SM is the trapped oxygen and methane propellant left in the propulsion tanks after completion of the nominal delta-V maneuvers. Residuals for liquid propellants are 2 percent of the nominally consumed propellant.

The LOX and LCH4 used for the SM service propulsion system and RCS are stored entirely passively (i.e., with foam and MLI only); therefore, as heat leaks into the propellant tanks, the cryogenic fluids will slowly vaporize. Vaporized propellant, or boil-off, is vented as it is produced to maintain a nominal tank pressure. Boil-off mass is calculated assuming 60 layers of variable-density MLI per tank and SOFI, a 210 K external environment temperature, and the appropriate heats of vaporization for oxygen and methane.

The assumed pressurization system for the SM propellant tanks is GHe stored in two 6,000 psia tanks. As propellant is consumed, the GHe is distributed to the tanks to maintain a propellant tank pressure of 325 psia. To minimize helium tank size, the tanks are thermodynamically coupled to the LCH4 tank, thus reducing the helium temperature while stored to 112 K.

Cargo

There are no cargo components included on the CEV SM.

Non-Propellant

There are no non-propellant components included on the CEV SM. All non-propellant fluids are stored on the CM.

Propellant

Propellant for the CEV SM consists of the following components:

- Used service propulsion system fuel propellant,
- Used service propulsion system oxidizer propellant,
- Used RCS fuel propellant, and
- Used RCS oxidizer propellant.

CEV total SM service propulsion system/RCS propellant is calculated for four major delta-V maneuvers in the mission. For each maneuver, the assumed service propulsion system Isp is 363.6 sec and the RCS Isp is 317.0 sec.

- The first major maneuver is rendezvous and docking with the LSAM in LEO. The CEV is inserted by the LV upper stage into a 55- x 185-km (30- x 100-nmi) elliptical orbit, while the LSAM and EDS are loitering in a 296-km (160-nmi) circular orbit. The CEV will then rendezvous with the LSAM and dock. The required delta-V for rendezvous and docking is estimated at 119.4 m/s for the service propulsion system and 25.1 m/s for the RCS, while the initial CEV mass prior to the maneuver is 23,149 kg.
- The second major maneuvers are station-keeping in LLO while the crew is on the surface and a contingency 5-deg plane change in the event of a worst-case anytime ascent from a 85-deg latitude landing site. The required delta-V for station-keeping is estimated at 15 m/s for RCS and 156 m/s of service propulsion system delta-V is included for the plane change. The initial CEV mass prior to these maneuvers is 21,587 kg.
- The third major CEV maneuver is TEI from LLO. For a worst-case anytime return from a polar orbit, a 90-deg plane change may first be needed to align the spacecraft's velocity vector with the V-infinity departure vector. The method chosen to accomplish this maneuver is to use a sequence of three impulsive burns, where the first burn raises the CEV orbit apolune from a 100-km orbit to an orbit with a period of 24 hours. The CEV coasts to the correct position to perform the 90-deg plane change and then coasts to perilune to complete TEI. The required delta-V for TEI is estimated at 1,449 m/s for the service propulsion system. This maneuver also includes +/– 90-deg control of the arrival coazimuth at Earth and +/–12-hr control of the nominal 96-hr return time from the third TEI burn. The initial CEV mass prior to the maneuver is 21,057 kg.
- The fourth maneuver is a 10-m/s mid-course correction using an RCS. This is used to correct any errors resulting from an imprecise TEI burn. The initial CEV mass prior to the maneuver is 14,023 kg.
- The fifth and final SM maneuver is to safely dispose of the SM after CM separation. The required RCS delta-V for disposal is 15 m/s, and the initial SM mass prior to the burn is 4,372 kg.

5.2.3.3 Launch Abort System (LAS)

The LAS was sized to pull the CEV CM away from a thrusting LV at 10 g's acceleration. The LAS sizing concept is similar to the Apollo Launch Escape System (LES) in that it is a tractor system that is mounted ahead of the CM. The main difference is that the exhaust nozzles are located near the top of the motor, which will reduce the impingement loads on the CM.

The LAS features an active trajectory control system based on solid propellant, a solid rocket escape motor, forward recessed exhaust nozzles, and a CM adaptor. The motor measures 76 cm in diameter and 5.5 m in length, while eight canted thrusters aid in eliminating plume impingement on the CM. A star fuel grain minimizes motor size and redundant igniters are intended to guarantee the system's start.

The LAS provides abort from the launch pad and throughout powered flight of the booster first stage. The LAS is jettisoned approximately 20–30 seconds after second stage ignition. Further analyses are required to determine the optimum point in the trajectory for LAS jettison. After the LAS is jettisoned, launch aborts for the crew are provided by the SM propulsion system.

The mass for a 10-g LAS for a 21.4 mT CM is 4.2 mT. **Figure 5-7** depicts the LAS on top of the CM.

Figure 5-7. CEV with Launch Abort System

5.2.4 ISS CEV CM (3 Crew with 400 kg Cargo)

5.2.4.1 Vehicle Description

The ISS CEV CM in the ESAS architecture is the Block 1 variant of the lunar CM designed to rotate three to six crew members and cargo to the ISS. The ISS CM is designed largely to support lunar exploration requirements, with a minimal set of modifications made to support ISS crew rotation. Initial mass for the three-crew ISS CM variant is 162 kg less than the lunar CM mass, with the assumed system modifications listed below:

- Removed EVA support equipment for one crew member (–3 kg);
- Sized galley, waste collection consumables, and soft stowage for 18 crew-days instead of 53 crew-days (–19 kg);
- Removed one crew member and sized personnel provisions for 18 crew-days (–238 kg);
- Added ISS cargo (+400 kg);
- Sized oxygen, nitrogen, and potable water for 18 crew-days (–156 kg);
- Sized RCS propellant for smaller vehicle mass and lower delta-V (–145 kg); and
- Less growth allocation for lower vehicle dry mass (–4 kg).

5.2.4.2 Overall Mass Properties

Table 5-3 provides overall vehicle mass properties for the ISS crewed variant of the CEV CM. The mass properties reporting standard is outlined in JSC-23303, Design Mass Properties. A detailed mass statement is provided in **Appendix 5A**, **CEV Detailed Mass Breakdowns**.

Table 5-3. Vehicle Mass Properties for the ISS Crewed Variant of the CEV CM

ISS CEV CM (3 Crew + Cargo)	% of Vehicle Dry Mass	Mass (kg)	Volume (m^3)
1.0 Structure	24%	1,883	0
2.0 Protection	11%	894	1
3.0 Propulsion	5%	413	0
4.0 Power	10%	819	1
5.0 Control	0%	0	0
6.0 Avionics	5%	435	1
7.0 Environment	13%	1,069	3
8.0 Other	14%	1,159	2
9.0 Growth	17%	1,335	1
10.0 Non-Cargo		581	2
11.0 Cargo		500	1
12.0 Non-Propellant		211	0
13.0 Propellant		42	0
Dry Mass	**100%**	**8,008 kg**	
Inert Mass		**9,089 kg**	
Total Vehicle		**9,342 kg**	

5.2.4.3 Subsystem Description

5.2.4.3.1 Structure

The CEV CM structure is identical for the lunar and ISS variants because the lunar CM variant is already designed to withstand an internal cabin pressure of 14.7 psia.

5.2.4.3.2 Protection

The CEV CM spacecraft protection is identical for the lunar and ISS variants because the ISS CM variant uses the ablative aft heat shield designed for the lunar mission.

5.2.4.3.3 Propulsion

The CEV CM propulsion is identical for the ISS and lunar CM variants.

5.2.4.3.4 Power

The power subsystem for the CEV CM is identical for the ISS and lunar variants.

5.2.4.3.5 Control

Items typically included in the spacecraft control category are aerodynamic control surfaces, actuators, cockpit controls such as rudder pedals, and others. There are no control components on the CEV CM.

5.2.4.3.6 Avionics

The CEV CM avionics subsystem is identical for the ISS and lunar variants.

5.2.4.3.7 Environment

The CEV environment components consist of the equipment needed to maintain vehicle health and a habitable volume for the crew and include the following:

- ECLS,
- ATCS, and
- Crew accommodations.

Environmental Control and Life Support (ECLS)

The ISS CEV CM differs from the lunar variant only in that EVA umbilicals and support equipment are included for three crew members rather than four.

Active Thermal Control System (ATCS)

Active thermal control for the CEV is identical for the ISS and lunar variants.

Crew Accommodations

The crew accommodations portion of the CEV CM differs from the lunar variant in that galley equipment, waste collection, and stowage is provided for three crew members (18 crew-days) in the ISS variant versus four crew members (53.3 crew-days) in the lunar mission.

5.2.4.3.8 Other

CEV CM components included in the "Other" category, such as the parachute system, landing system, flotation system, and docking system, are identical for the ISS and lunar CM variants. The ISS CM uses a LIDS docking mechanism for docking to ISS rather than the Shuttle's Androgynous Peripheral Attachment System (APAS) mechanism.

5.2.4.3.9 Growth

Mass growth included on the CEV CM is sized for 20 percent of dry mass.

5.2.4.3.10 Non-Cargo

Mass for personnel and personnel provisions has been reduced on the ISS CM variant to reflect the smaller crew size (three versus four) and shorter mission duration (6 versus 13.3 days).

Residual propellant is estimated at 2 percent of the nominally consumed propellant for the ISS mission.

5.2.4.3.11 Cargo

Cargo for the ISS CEV CM differs from the lunar CM in that 400 kg of pressurized cargo has been added in place of the fourth crew member. The pressurized cargo in Mid-deck Locker Equivalents (MLEs) has a density of 272.7 kg/m^3. Ballast mass for the CM is unchanged at 100 kg.

5.2.4.3.12 Non-Propellant

Mass for oxygen, nitrogen, and potable water has been changed on the ISS CM variant to reflect the smaller crew size (three versus four), shorter mission duration (6 versus 13.3 days), and higher cabin pressure (14.7 versus 9.5 psia).

5.2.4.3.13 Propellant

Propellant for the ISS CEV CM is estimated using a lower delta-V (10 m/s versus 50 m/s), as the lunar skip-entry trajectory is not applicable to the ISS mission. The propellant loading has also changed due to the lower CM mass at entry with the ISS mission. The initial CM mass prior to the maneuver is estimated at 9,335 kg.

5.2.5 ISS CEV CM (Six Crew)

5.2.5.1 Vehicle Description

The ISS CEV CM in the ESAS architecture is the Block 1 variant of the lunar CM designed to rotate three to six crew members and cargo to ISS. The ISS CM is designed largely to support lunar exploration requirements, with a minimal set of modifications made to support ISS crew rotation. Initial mass for the six-crew ISS CM variant is 45 kg more than the lunar CM mass with the assumed system modifications listed below:

- Added EVA support equipment for two crew members (+6 kg);
- Sized galley, waste collection consumables, soft stowage, and seats for six crew and 36 crew-days instead of four crew and 53 crew-days (+31 kg);
- Added two crew members and sized personnel provisions for 36 crew-days (+219 kg);
- Sized oxygen, nitrogen, and potable water for 36 crew-days (–76 kg);
- Sized RCS propellant for larger vehicle mass and lower delta-V (–144 kg); and
- More growth allocation for higher vehicle dry mass (+8 kg).

5.2.5.2 Overall Mass Properties

Table 5-4 provides overall vehicle mass properties for the ISS crewed variant of the CEV CM. The mass properties reporting standard is outlined in JSC-23303, Design Mass Properties. A detailed mass statement is provided in **Appendix 5A**, **CEV Detailed Mass Breakdowns**.

ISS CEV CM (6 Crew)	% of Vehicle Dry Mass	Mass (kg)	Volume (m³)
1.0 Structure	23%	1,883	0
2.0 Protection	11%	894	1
3.0 Propulsion	5%	413	0
4.0 Power	10%	819	1
5.0 Control	0%	0	0
6.0 Avionics	5%	435	1
7.0 Environment	14%	1,129	4
8.0 Other	14%	1,159	2
9.0 Growth	17%	1,346	2
10.0 Non-Cargo		1,038	4
11.0 Cargo		500	1
12.0 Non-Propellant		100	0
13.0 Propellant		43	0
Dry Mass	**100%**	**8,079 kg**	
Inert Mass		**9,217 kg**	
Total Vehicle		**9,551 kg**	

Table 5-4. Vehicle Mass Properties for the ISS Crewed Variant of the CEV CM

5.2.5.3 Subsystem Description

5.2.5.3.1 Structure

The CEV CM structure is identical for the lunar and ISS variants because the lunar CM variant is already designed to withstand an internal cabin pressure of 14.7 psia.

5.2.5.3.2 Protection

The CEV CM spacecraft protection is identical for the lunar and ISS variants because the ISS CM variant uses the ablative aft heat shield designed for the lunar mission.

5.2.5.3.3 Propulsion

The CEV CM propulsion is identical for the ISS and lunar CM variants.

5.2.5.3.4 Power

The power subsystem for the CEV CM is identical for the ISS and lunar variants.

5.2.5.3.5 Control

Items typically included in the spacecraft control category are aerodynamic control surfaces, actuators, cockpit controls such as rudder pedals, and others. There are no control components on the CEV CM.

5.2.5.3.6 Avionics

The CEV CM avionics subsystem is identical for the ISS and lunar variants.

5.2.5.3.7 Environment

The CEV environment components consist of the equipment needed to maintain vehicle health and a habitable volume for the crew and include the following:

- ECLS,
- ATCS, and
- Crew accommodations.

Environmental Control and Life Support (ECLS)

The ISS CEV CM differs from the lunar variant only in that EVA umbilicals and support equipment are included for six crew members rather than four.

Active Thermal Control System (ATCS)

Active thermal control for the CEV is identical for the ISS and lunar variants.

Crew Accommodations

The crew accommodations portion of the CEV CM differs in that galley equipment, waste collection, seating, and stowage is provided for six crew members (36 crew-days) in the ISS variant versus four crew members (53.3 crew-days) in the lunar mission.

5.2.5.3.8 Other

CEV CM components included in the “Other” category, such as the parachute system, landing system, flotation system, and docking system, are identical for the ISS and lunar CM variants. The ISS CM uses a LIDS docking mechanism for docking to the ISS rather than the Shuttle’s APAS mechanism.

5.2.5.3.9 Growth

Mass growth included on the CEV CM is sized for 20 percent of dry mass.

5.2.5.3.10 Non-Cargo

Mass for personnel and personnel provisions has been reduced on the ISS CM variant to reflect the larger crew size (six versus four) and shorter mission duration (6 versus 13.3 days).

Residual propellant is estimated at 2 percent of the nominally consumed propellant for the ISS mission.

5.2.5.3.11 Cargo

The six crew-to-ISS variant of the lunar CEV CM does not carry any cargo to ISS. Ballast mass for the CM is unchanged at 100 kg.

5.2.5.3.12 Non-Propellant

Mass for oxygen, nitrogen, and potable water has been changed on the ISS CM variant to reflect the greater crew size (six versus four), shorter mission duration (6 versus 13.3 days), and higher cabin pressure (14.7 versus 9.5 psia).

5.2.5.3.13 Propellant

Propellant for the ISS CEV CM is estimated using a lower delta-V than the lunar variant (10 m/s versus 50 m/s), as the lunar skip-entry trajectory is not applicable to the ISS mission. The propellant loading is also affected by the higher CM mass at entry with the ISS mission. The initial CM mass prior to the maneuver is estimated at 9,544 kg.

5.2.6 ISS Pressurized Cargo CEV CM Variant

5.2.6.1 Vehicle Description

The ESAS architecture also includes a variant of the ISS CEV CM that may be used to deliver several tons of pressurized cargo to the ISS without crew on board and return an equivalent mass of cargo to a safe Earth landing. This spacecraft is nearly identical to the ISS crew rotation variant, with the exception that the personnel and most components associated with providing crew accommodations are removed and replaced with cargo. Initial mass for the uncrewed ISS CM variant is 2,039 kg greater than the three-crew ISS crew rotation CM, with the assumed system modifications listed below:

- Removed atmosphere contaminant (CO2, etc.) control equipment (–165 kg);
- Removed EVA support equipment (–21 kg);
- Removed galley, WCS, and CTBs (–84 kg);
- Removed mass for personnel and personnel provisions (–580 kg);
- Removed 500 kg of ISS cargo and ballast, and added 3,500 kg of ISS cargo (+3,000 kg);
- Loaded oxygen, nitrogen, and water as needed for the pressurized cargo mission (–64 kg);
- Increased RCS propellant for higher vehicle mass (+8 kg); and
- Less growth allocation for lower vehicle dry mass (–54 kg).

5.2.6.2 Overall Mass Properties

Table 5-5 provides mass properties for the ISS pressurized cargo delivery variant of the CEV. The mass properties reporting standard is outlined in JSC-23303, Design Mass Properties. A detailed mass statement is provided in **Appendix 5A**, **CEV Detailed Mass Breakdowns**.

Table 5-5. Mass Properties for the ISS Pressurized Cargo Delivery Variant of the CEV

ISS CEV Capsule (Pressurized Cargo)	% of Vehicle Dry Mass	Mass (kg)	Volume (m^3)
1.0 Structure	25%	1,883	0
2.0 Protection	12%	894	1
3.0 Propulsion	5%	413	0
4.0 Power	10%	819	1
5.0 Control	0%	0	0
6.0 Avionics	6%	435	1
7.0 Environment	10%	799	3
8.0 Other	15%	1,159	2
9.0 Growth	17%	1,281	1
10.0 Non-Cargo		1	2
11.0 Cargo		3,500	1
12.0 Non-Propellant		147	0
13.0 Propellant		49	0
Dry Mass	**100%**	**7,683 kg**	
Inert Mass		**11,184 kg**	
Total Vehicle		**11,381 kg**	

5.2.6.3 Subsystem Description

5.2.6.3.1 Structure

The CEV CM structure is identical for the crewed and uncrewed ISS variants.

5.2.6.3.2 Protection

The CEV CM spacecraft protection is identical for the crewed and uncrewed ISS variants. The uncrewed CM is designed to return as much cargo to Earth as it delivers to the ISS; however, lunar entry requirements remain the dominant heat load/heat rate case for TPS sizing.

5.2.6.3.3 Propulsion

The CEV CM propulsion is identical for the crewed and uncrewed ISS variants.

5.2.6.3.4 Power

The power subsystem for the CEV CM is identical for the crewed and uncrewed ISS variants.

5.2.6.3.5 Control

Items typically included in the spacecraft control category are aerodynamic control surfaces, actuators, cockpit controls such as rudder pedals, and others. There are no control components on the CEV CM.

5.2.6.3.6 Avionics

The CEV CM avionics subsystem is identical for the crewed and uncrewed ISS variants.

5.2.6.3.7 Environment

Select CEV environment components required for the crew rotation mission are removed from the uncrewed pressurized cargo delivery variant. Changes between the variants are noted below.

Environmental Control and Life Support (ECLS)
The ISS uncrewed pressurized cargo delivery CM differs from the crew rotation variant in that atmosphere contaminant control equipment and EVA umbilicals and support equipment have been removed from the uncrewed CM. Without crew on board, there is no need for the vehicle to remove CO2 from the atmosphere or support EVAs.

Active Thermal Control System (ATCS)
Active thermal control for the CEV is identical for the ISS and lunar variants.

Crew Accommodations
The crew accommodations portion of the CEV uncrewed CM differs in that the galley equipment, waste collection, and crew seating needed for the crewed CM has been removed.

5.2.6.3.8 Other
CEV CM components included in the "Other" category, such as the parachute system, landing system, flotation system, and docking system, are identical for the crewed and uncrewed ISS variants. Using parachutes designed to support the lunar exploration mission, the pressurized cargo CEV lands with three fully inflated main parachutes at 8.2 m/s (26.9 ft/s) and a landed mass of 10,604 kg. For the lunar CEV with one failed chute, the crewed vehicle lands at 8.9 m/s (29.5 ft/s) and landed mass of 8,475 kg.

5.2.6.3.9 Growth
Mass growth included on the CEV CM is sized for 20 percent of dry mass.

5.2.6.3.10 Non-Cargo
Since the pressurized cargo variant of the CEV CM is uncrewed, all mass dedicated to personnel and personnel provisions have been eliminated from the vehicle. The only remaining non-cargo component is residual propellant, which is estimated at 2 percent of the nominally consumed propellant for the ISS mission.

5.2.6.3.11 Cargo
The uncrewed, pressurized cargo delivery CM has been sized to deliver 3,500 kg of pressurized cargo to the ISS in MLEs. The pressurized cargo has a density of 272.7 kg/m^3. Ballast mass for the CM has been removed.

5.2.6.3.12 Non-Propellant
Mass for oxygen and nitrogen is included on the uncrewed pressurized cargo delivery CM to maintain an appropriate pressurized environment for the cargo. Potable water has been removed because of the lack of need without crew on board.

5.2.6.3.13 Propellant
Propellant for the ISS pressurized cargo CEV CM is estimated using the same 10 m/s delta-V as the crew rotation variant; however, the propellant loading has changed due to the different CM mass at entry. The initial CM mass prior to the maneuver is estimated at 11,374 kg.

5.2.7 ISS SM (Off-loaded Lunar SM)

5.2.7.1 Vehicle Description

The ISS SM is identical to the SM designed for lunar exploration, except that propellant is off-loaded to reflect the lower delta-V requirements of ISS crew rotation compared to LOR. Propellant requirements for the ISS SM are estimated based on using the largest vehicle the SM may deliver to the ISS and subsequently deorbit, which is currently the unpressurized CDV. Other potential ISS payloads for the SM are the crewed CEV CM and pressurized cargo CEV; however, these have total masses less than the unpressurized CDV. The CDV has a total mass of 12,200 kg, compared to 9,342 kg for the three-crew CEV, 9,551 kg for the six-crew CEV, and 11,381 kg for the pressurized cargo delivery CEV.

5.2.7.2 Overall Mass Properties

Table 5-6 provides overall vehicle mass properties for the ISS SM, assuming a common lunar SM with off-loaded consumables. The mass properties reporting standard used in the table is outlined in JSC-23303, Design Mass Properties. A detailed mass statement is provided in **Appendix 5A**, **CEV Detailed Mass Breakdowns**.

Table 5-6. Vehicle Mass Properties for the ISS SM

ISS SM for Unpressurized Cargo Carrier (Off-loaded Lunar SM)	% of Vehicle Dry Mass	Mass (kg)	Volume (m^3)
1.0 Structure	20%	819	0
2.0 Protection	4%	167	1
3.0 Propulsion	36%	1,423	15
4.0 Power	10%	417	0
5.0 Control	0%	0	0
6.0 Avionics	3%	117	0
7.0 Environment	2%	98	1
8.0 Other	7%	290	0
9.0 Growth	17%	666	3
10.0 Non-Cargo		882	0
11.0 Cargo		0	0
12.0 Non-Propellant		0	0
13.0 Propellant		2,033	0
Dry Mass	**100%**	**3,997 kg**	
Inert Mass		**4,879 kg**	
Total Vehicle		**6,912 kg**	

5.2.7.3 Subsystem Description

5.2.7.3.1 Structure

The CEV SM structure is identical for the ISS and lunar variants.

5.2.7.3.2 Protection

The CEV SM protection is identical for the ISS and lunar variants.

5.2.7.3.3 Propulsion

The CEV SM propulsion is identical for the ISS and lunar variants because the ISS variant uses the propulsion system designed for the lunar mission and loads propellant as needed to transfer to the ISS.

5.2.7.3.4 Power
The power subsystem for the CEV SM is identical for the ISS and lunar variants.

5.2.7.3.5 Control
Items typically included in the spacecraft control category are aerodynamic control surfaces, actuators, cockpit controls such as rudder pedals, and others. There are no control components on the CEV SM.

5.2.7.3.6 Avionics
The CEV SM avionics subsystem is identical for the lunar and ISS variants.

5.2.7.3.7 Environment
The CEV environment components are identical for the ISS and lunar variants, as the radiator panels are sized for the worst environment conditions of the two missions and are used for either variant.

5.2.7.3.8 Other
CEV SM components are identical for the ISS and lunar variants.

5.2.7.3.9 Growth
Mass growth is the same for either the lunar or ISS SM.

5.2.7.3.10 Non-Cargo
The amount of residual propellant, propellant boil-off, and pressurant included on the SM varies depending on the needs for the ISS or lunar missions. Residual propellant on the CEV SM is 2 percent of the nominally consumed propellant, which is substantially less for the ISS mission owing to the lower total delta-V.

LOX and LCH4 boil-off for the ISS SM has been calculated assuming an average environment temperature of 250 K while docked to the ISS, while lunar mission boil-off was estimated with an average environment temperature of 210 K. The higher temperature is due to the CEV being placed in a non-optimal fixed attitude at the ISS and greater incoming infrared radiation from Earth and the ISS. While loitering in lunar orbit, the CEV can be placed in a more thermally benign attitude configuration, thus reducing propellant boil-off.

The mass of helium pressurant required is identical for the lunar and ISS variants.

5.2.7.3.11 Cargo
There are no cargo components included on the CEV SM.

5.2.7.3.12 Non-Propellant
There are no non-propellant components included on the CEV SM. All non-propellant fluids are stored on the CM.

5.2.7.3.13 Propellant
Propellant for the CEV SM in the ISS unpressurized cargo carrier delivery mission is loaded as needed for that mission's delta-V requirements. CEV total SM service propulsion system/RCS propellant is calculated for three major delta-V maneuvers in the mission. For each maneuver, the assumed service propulsion system Isp is 353.6 sec and the RCS Isp is 307.0 sec. The engine Isp for the ISS SM has been decremented by 10 sec below the level of the lunar variant to allow for suboptimal performance in the early years of the engine life. All ISS mission delta-Vs include 10 percent reserve.

The major SM maneuvers are described below.

- The first major maneuvers are circularization of the CEV insertion orbit and rendezvous and docking with the ISS. The CEV is first inserted by the CEV LV into a 55- x 296-km (30- x 160-nmi) LEO, and, when the CEV coasts to apogee, the SM uses its service propulsion system to circularize its orbit and then rendezvous and dock with the ISS. Maximum ISS altitude is 460 km (250 nmi) for this analysis. The required delta-V for circularization, rendezvous, and docking is estimated at 191.8 m/s for the service propulsion system and 33.5 m/s for the RCS, while the initial CEV mass prior to the maneuver is 19,104 kg.
- The second major maneuvers are undocking from the ISS and deorbit. Deorbit from the ISS is estimated assuming a maximum ISS altitude of 460 km and deorbit perigee of 46 km. The required service propulsion system delta-V for undocking and deorbit is estimated at 137.7 m/s for the service propulsion system and 19.4 m/s for the RCS. The initial CEV mass prior to these maneuvers is 17,204 kg.
- The third and final SM maneuver is to safely dispose of the SM after CM separation. The required RCS delta-V for disposal is 15 m/s and the initial SM mass prior to the burn is 4,224 kg.

5.2.8 ISS Unpressurized CDV

5.2.8.1 Vehicle Description

The ISS CDV was sized to deliver unpressurized cargo to the ISS. The CDV is mainly a structural “strong back” with a CBM for attachment to the ISS. The CDV utilizes the same SM as the other block configurations for transfer from the LV injection orbit to the ISS. Because the avionics for the other CEV variants are located within the CM, an avionics pallet is required for the CDV. This pallet would support the avionics and provide the connection to the ATCS on the SM.

The CDV was sized to transport two 1,500-kg unpressurized Orbital Replacement Units (ORUs) for the ISS. Examples of ORUs include Control Moment Gyroscopes (CMGs) and pump packages. The packaging factor for these ORUs was assumed to be 100 percent; therefore, the trays and secondary support structure for the cargo is estimated to be 3,000 kg, for a total cargo complement of 6,000 kg. The total estimate for the CDV without the SM is 12,200 kg.

Operationally, the CDV would perform automated rendezvous and proximity operations with the ISS and would then be grappled by the SSRMS and berthed to an available port. Two releasable cargo pallets are used to provide structural attachment for the ORUs. The cargo pallets can be grappled by the SSRMS and relocated to the ISS truss as required. Once the cargo has been relocated on the ISS, the CDV would depart from the ISS and perform an automated deorbit burn for burnup and disposal in the ocean.

Illustrations of the reference CDV are shown in **Figures 5-8** and **5-9**.

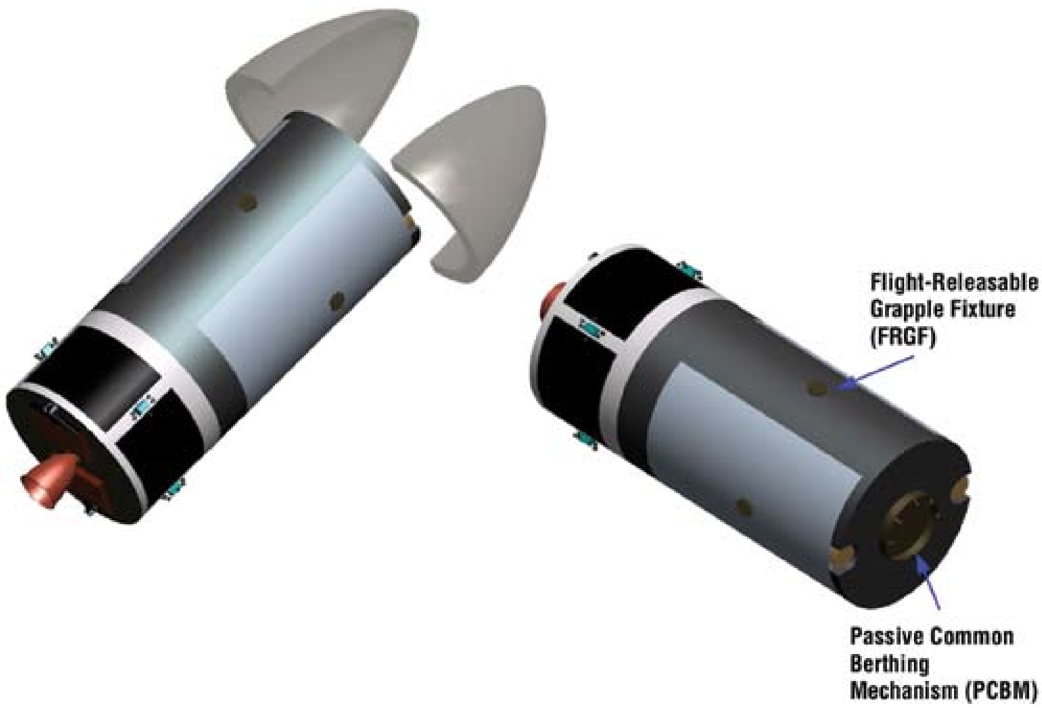


Figure 5-8. CDV

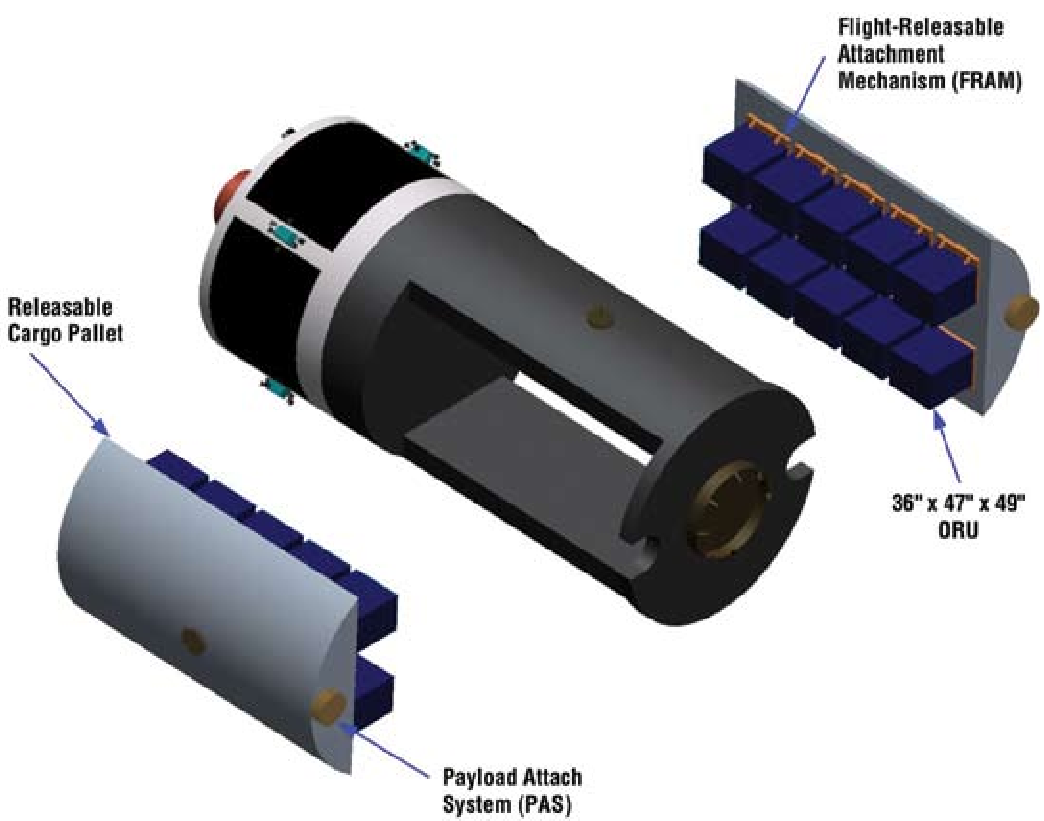


Figure 5-9. CDV Cargo Pallets

5.2.9 Mars Block 3 CEV

5.2.9.1 Vehicle Description

The ESAS reference Mars mission utilizes a Block 3 CEV to transfer a crew of six between Earth and an MTV at the beginning and end of the Mars exploration mission. A Block 3 CEV CM and SM are launched by the CLV into an orbit matching the inclination of the awaiting MTV. The CEV is first injected into a 55- x 296-km altitude orbit while the MTV loiters in a circular orbit of 800–1,200 km altitude. It then takes the CEV up to 2 days to perform orbit-raising maneuvers to close on the MTV, conducting a standard ISS-type rendezvous and docking approach to the MTV. After docking, the CEV crew performs a leak check, equalizes pressure with the MTV, and opens hatches. Once crew and cargo transfer activities are complete, the CEV is configured to a quiescent state and remains docked to the MTV for the trip to and from Mars. Periodic systems health checks and monitoring are performed by the ground and flight crew throughout the mission.

As the MTV approaches Earth upon completion of the 1.5–2.5 year round-trip mission, the crew performs a pre-undock health check of all entry critical systems, transfers to the CEV, closes hatches, performs leak checks, and undocks from the MTV. The CEV departs 24–48 hours prior to Earth entry, and the MTV then either performs a diversion maneuver to fly by Earth or recaptures into Earth orbit. After undocking, the CEV conducts an onboard-targeted, ground-validated burn to target for the proper entry corridor, and, as entry approaches, the CEV CM maneuvers to the proper EI attitude for a direct-guided entry to the landing site. Earth entry speeds from a nominal Mars return trajectory may be as high as 14 km/s, compared to 11 km/s for the Block 2 CEV. The CEV performs a nominal landing at the primary land-based landing site and the crew and vehicle are recovered.

Figure 5-10 shows the Block 3 CEV CM configured to carry six crew members to the MTV.

Figure 5-10. Block 3 CEV CM

5.3 Crew Exploration Vehicle (CEV) Trades

Many trade studies were performed in the development of the CEV design and requirements. Some of these were specific to the CEV and others were more global to the architecture. For example, determining the CM OML shape and internal volume was specific to the CEV, but other trades that addressed propulsion, airlocks, and radiation protection were cross-cutting across the architecture. The following sections describe some of the trades that were performed on the CEV shape, size, systems, and performance.

5.3.1 CM Vehicle Shape

5.3.1.1 Introduction and Requirements

The ESAS team addressed the task of designing the CM vehicle shape. A number of desirable characteristics was identified through requirements allocation and trade studies. The initial goal was to achieve as many of these characteristics as possible with the proper design of an OML shape. These characteristics included:

- Low technical risk for near-term development feasibility;
- Adequate volume to meet the ISS, lunar, and Mars DRMs;
- Satisfaction of acceleration loads across the spectrum of flight conditions within crew limits;
- Efficient dissipation of entry aeroheating loads within existing material temperature limits;
- Adequate crew visibility for rendezvous and docking maneuvers;
- A simple yet robust approach to abort survival in case of primary power or Guidance Navigation and Control (GN&C) failures;
- Land-landing capability for reusability; and
- Highly accurate CONUS landing for ease and minimal cost of recovery and retrieval.

5.3.1.1.1 Monostability

The desire for a simple abort technique led to a goal of producing a vehicle that was monostable. This term implies that the vehicle has only one stable trim angle-of-attack in atmospheric flight. Given enough time, this would guarantee that the vehicle reaches its desired heat shield-forward attitude passively, without assistance from the RCS. The Apollo capsule was not able to achieve monostability due to the inability to place the CG close enough to the heat shield. Conversely, the Soyuz vehicle is monostable, with claims that it is able to achieve its desired trim attitude and a successful reentry with initial tumble rates of up to 2 deg/sec. **Figure 5-11** shows the history of abort ascent and entries that either relied on the monostable characteristic of the vehicle (Soyuz) for survival or would have benefited had the vehicle been monostable.

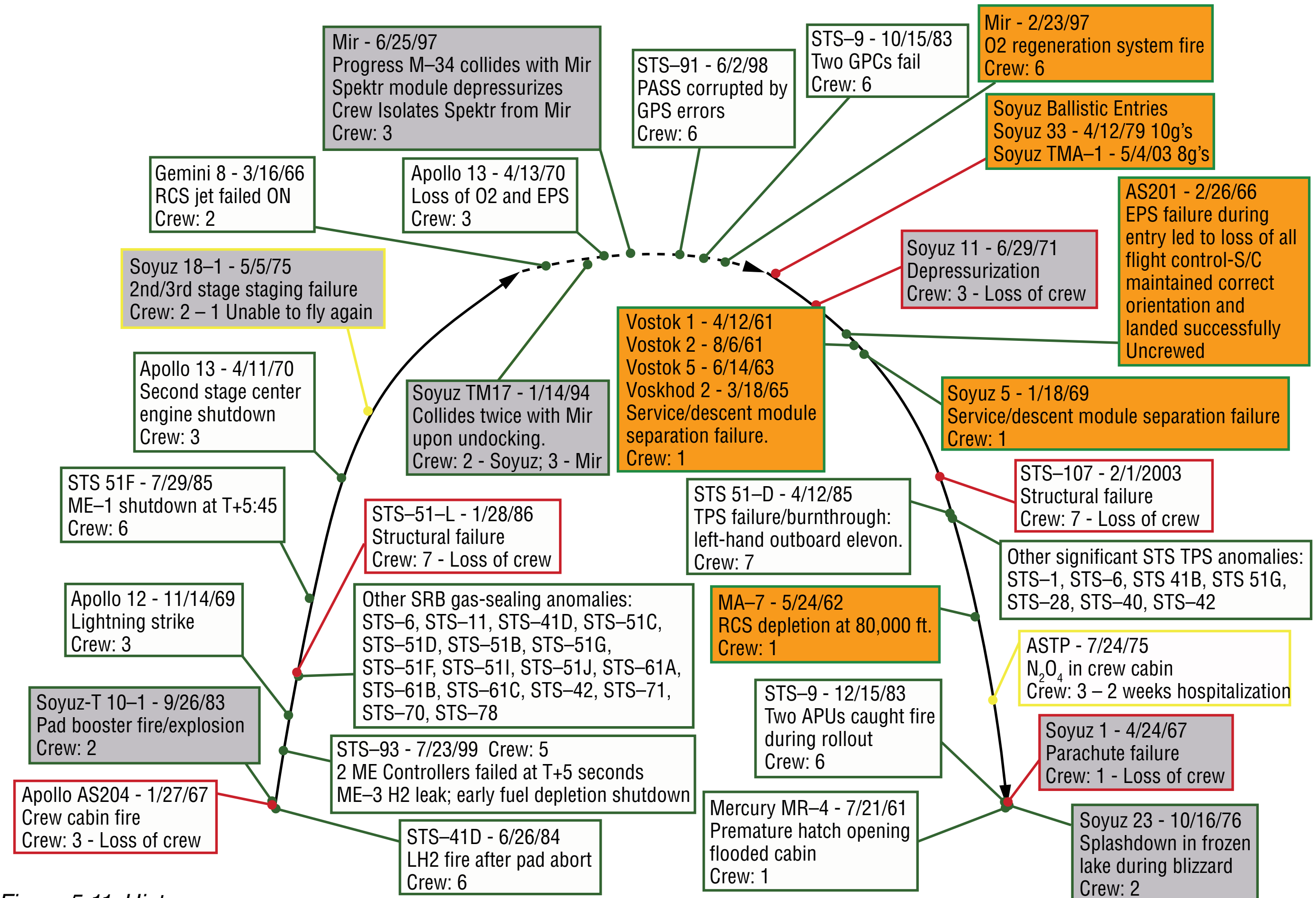


Figure 5-11. History of Manned Capsule Failures

5.3.1.1.2 Ballistic Entry Capability

A second way of achieving a simple abort approach was also examined in detail—to spin-up the vehicle after the proper orientation had been achieved. This spin-up would be a rolling motion about the velocity vector axis so that the lift of the vehicle would gyrate. This would produce a nearly ballistic trajectory through an effective cancellation of the lift vector so that it would have no effect on the entry trajectory. This would allow a vehicle that has lost primary power or control to successfully enter the atmosphere without being stuck in a lift-down roll angle that would exceed crew load limits.

5.3.1.1.3 Lift-to-Drag (L/D) Requirements

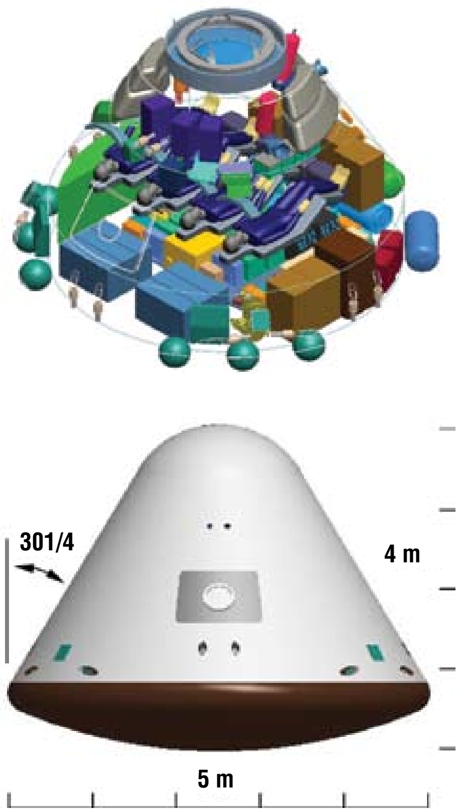
The desire for CONUS land landings led to the requirement for at least a 0.4 L/D ratio. This level of L/D is needed to reach attractive landing sites when returning from the ISS while safely disposing of the SM in the Pacific Ocean. In addition, the 0.4 L/D would aid in the performance of the lunar return skip-entry that was necessary to achieve the CONUS landing sites with a single entry technique. Although not enough time was permitted to perform an accurate quantitative trade study to indicate the minimum necessary L/D, it is known that the more L/D provided will produce a more accurate landing and help minimize the correction burn performed in the middle of the skip-entry maneuver. Further work is required to assess the risk and total viability of the CONUS land-landing approach for both lunar and LEO returns.

5.3.1.2 Blunt Bodies Versus Slender Bodies Trade

The shape study trade was initiated between major vehicle classes. The primary classes considered were capsules (blunt bodies), slender bodies, lifting bodies, and winged vehicles. Winged bodies and lifting bodies (such as X–38, X–24, HL–10, etc.) were eliminated at the outset due to several factors, including: (1) the extreme heating (especially on empennages) these would encounter on lunar return entries, (2) the additional development time required due to multiple control surfaces, and (3) the increased mass associated with wings, fins, and control surfaces which are huge liabilities in that they must be carried to the Moon and back simply for use on entry. Thus, the trade space involved capsules versus slender bodies. It was planned that, after a desirable class of vehicle was selected, the shape would be optimized within that class.

An extensive spreadsheet was designed to compare two applicable, fundamental classes of vehicles—blunt bodies and slender bodies. This spreadsheet attempted to delineate all the important performance, design, and operational differences that could be used as discriminators for selecting one class of vehicles over the other. Categories of evaluation included on the spreadsheet were: crew load directions and magnitudes, LV integration, entry heating, landing sites and opportunities, SM disposal, ballistic entry landing, weather avoidance, aerostability, terminal deceleration systems, landing issues, and additional mission and system requirements. Some of these analyses are presented in more detail below. All flight phases from launch to landing were evaluated for the two classes of vehicles, including three lunar return options: direct-entry, skip-entry, and aerocapture. The spreadsheet is provided in **Appendix 5B**, **CEV Crew Module Shape Trade Data**.

A representative vehicle was chosen in each class for analysis purposes. An Apollo-shape CEV configuration was selected as representative of the blunt-body class as seen in **Figure 5-12**. Both a straight biconic and an ellipsled design from earlier NASA studies were chosen as representative of the slender bodies (**Figure 5-13**). The configuration details of these vehicles can be seen in the first page of the spreadsheet in **Appendix 5B**, **CEV Crew Module Shape Trade Data**. For each of the slender bodies, two variations were analyzed—one without an attached SM and one with an attached SM.



Total Mass:	8,000 kg (17,637 lbs)
Crew Size:	4
Active Duration:	16 days
Passive Duration:	90 days
Pressurized Volume:	22 m^3 (777 ft^3)
Habitable Volume:	12 m^3 (424 ft^3)
Base Diameter:	5 m (16.4 ft)
Max Hypersonic L/D (est.):	0.3
Nominal Return Mode:	Direct Entry
Landing Mode:	Water w/ Contingency Land
Payload:	Crew + 100 kg (220.5 lbs)
Delta-V (Service Propulsion System/RCS):	0/10 m/s (33 ft/sec)
Major Maneuvers:	Aeroentry
Propellant:	Tridyne ($N_2/H_2/O_2$)
Isp:	140 s
Dry Mass Growth:	20%

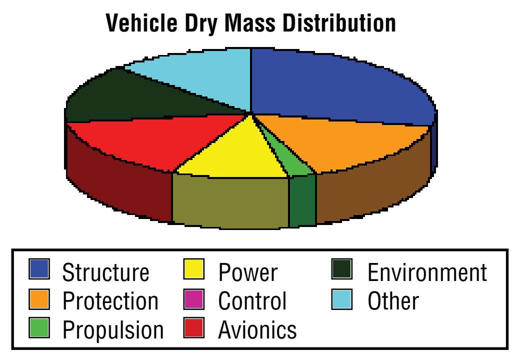


Figure 5-12. Representative Blunt-Body

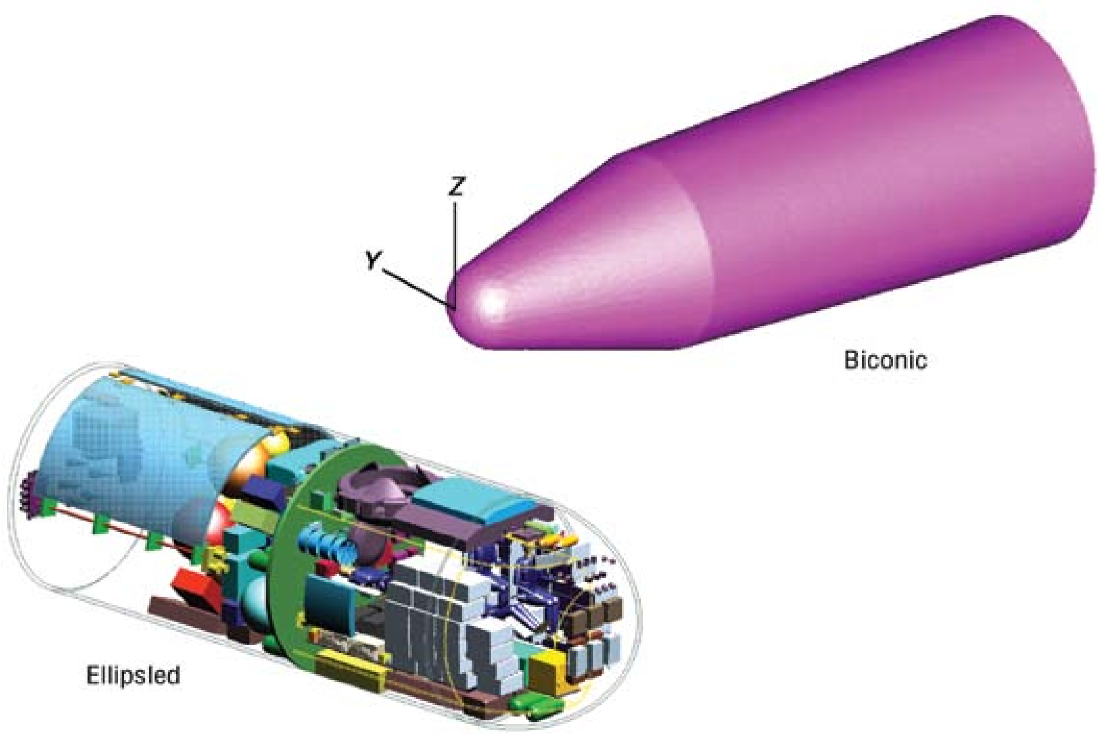


Figure 5-13. Representative Slender Bodies

5.3.1.2.1 Load Directions Analysis

One of the key areas of performance investigation was the area of load directions encountered by the crew in flight. It is important to note that the entry load directions are significantly different between a capsule and a slender body. During entry, the aerodynamic forces on a trimmed blunt-body primarily generate axial loads, as can be seen in **Figure 5-14**. As shown, the majority of the deceleration occurs along the axis of the capsule. This is also the same direction that primary loads are generated during ascent when attached to an LV, during ascent abort, and during landing. Conversely, slender bodies generate primarily normal aerodynamic loads, so that, on entry, the majority of the acceleration occurs normal to the axis of the slender body (**Figure 5-14**). These loads would be 90 deg off from the load direction encountered during ascent or ascent abort. These load directions have implications on the seating orientation of the crew. For a capsule, the logical crew orientation is with their backs parallel to the heat shield. All primary loads would then be carried through the crews' chest towards their backs ("eyeballs in"), which is the most tolerable load direction for a human. For a slender body, the primary load direction changes approximately 90 deg between launch and entry. Thus, either the crew would have to rotate their orientation in flight or a very benign ascent would have to be designed to allow the crew to take the ascent loads sitting up.

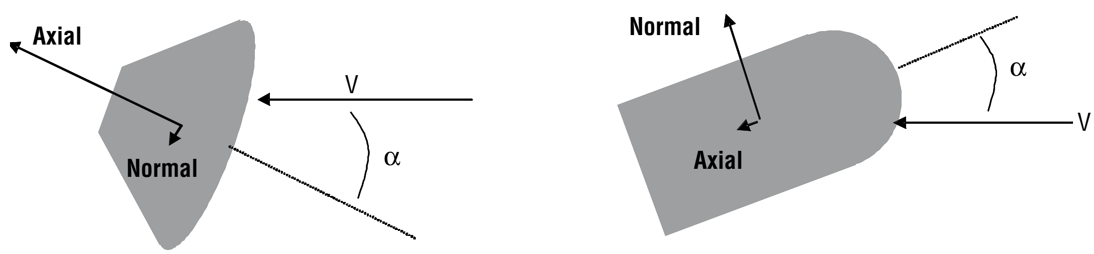


Figure 5-14. Atmospheric Flight (Entry) Aerodynamic Loads Direction

5.3.1.2.2 Load Magnitudes Analysis

During all phases of flight, it is mandatory that accelerations be kept within the crew load limits set forth by the NASA-STD-3000, Volume VIII, Human-Systems Integration Standards document. An example of these limit curves, which are a function of the duration of the load as well as the direction taken in the human body, is shown in **Figure 5-15**. Three limit-curves exist for each of the three human body axis directions. The highest limit-curve is intended for use in abort situations. It represents the maximum loads to ever be applied on the crew with the expectation of survival. The lowest limit-curve applies to crew who have been subjected to zero-gravity or very low gravity for an extended amount of time. The middle curve applies to normal, g-tolerant crew. Each of the vehicle shapes was evaluated in simulations to assess their capacities to meet these limits using the applicable limit-curves. Results can be seen in the spreadsheet in **Appendix 5B**, **CEV Crew Module Shape Trade Data**.

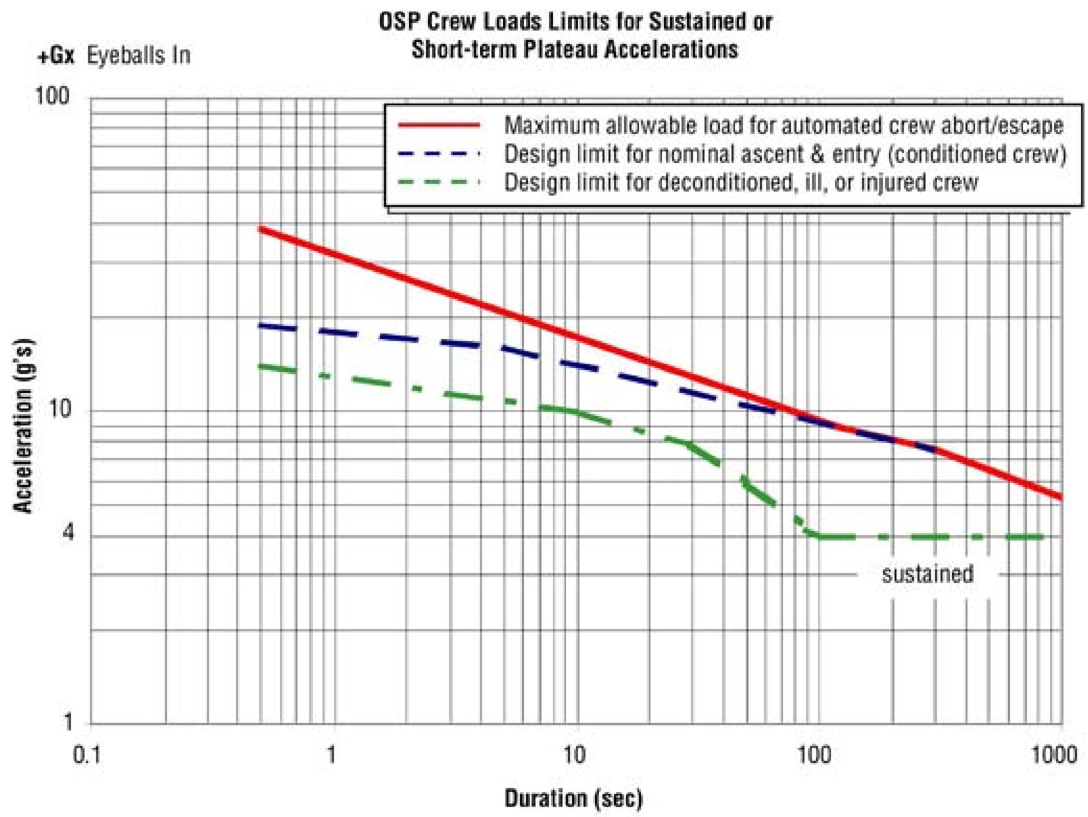


Figure 5-15. Example of NASA Standard 3000 Crew Load Limits

5.3.1.2.3 Aerodynamic Stability Analysis

Another key analysis in the shape trade study involved assessing the inherent aerodynamic stability in the design of the CEV CM as it relates to vehicle shape and CG location. In the presence of an active control system, the natural behavior of a vehicle can be augmented. Still, it is important to design a vehicle that can operate in a passively stable configuration for worst-case situations. An understanding of the stability characteristics of a vehicle cannot be obtained from a single parameter. A number of factors influenced the stability evaluation of the vehicle classes. In this study, monostability (including degree of monostability), pitching moment curve slope ($C_{m\alpha}$), trim$_{\alpha}$, and sensitivity of L/D to CG location were all included. All of these parameters were analyzed, reported, and evaluated for each of the shapes considered.

There are many other important limiting factors that are not related to stability, but are still related to the vehicle aerodynamics. These include CG location placement for desired L/D (which affects systems packaging and landing stability), trajectory range and cross-range capability, loads on vehicle and crew, and heat rates and heat loads (which affect TPS selection and mass). Thus, vehicle trim line information delineating desired CG locations for the proposed L/D was utilized for this analysis, while additional aerodynamic data was supplied to other analysts to perform trades in the other areas.

As discussed earlier, the representative vehicles for the slender bodies included a biconic and an ellipsled configuration. The Apollo capsule was used as the representative for the blunt bodies. The slender body vehicles exhibit a range of L/D ratios much higher than blunt capsules. Thus, the proposed L/D values differed. **Table 5-7** shows the different trim angles and L/D values studied.

	Biconic	Ellipsled	Apollo
L/D	.817	.655	.3
Trim α	40°	40°	19.8°

Table 5-7. L/D Ratios and Trim Angles

The main appeal of the slender bodies is their higher lift capability. **Figure 5-16** shows the CG trim line for the 40-deg angle-of-attack trim for the biconic shape.

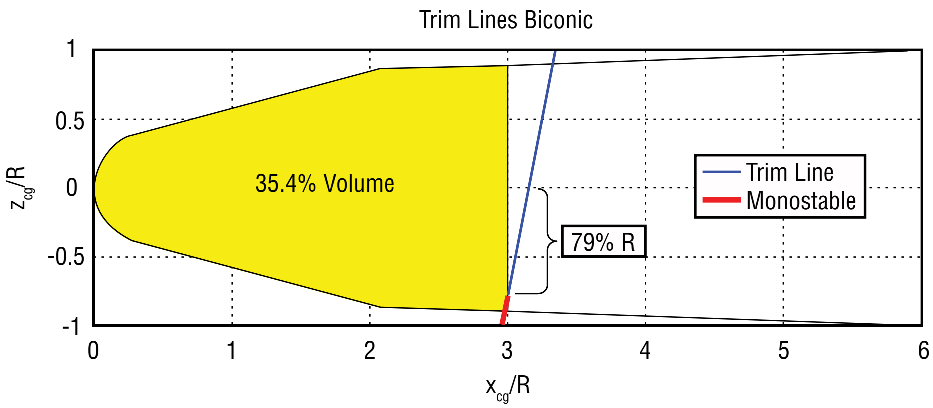


Figure 5-16. Biconic Shape with Trim Line

Figure 5-17 shows that the vehicle, which has an aspect ratio of three, trims with a CG near the center of the vehicle. However, if monostability (one stable trim angle-of-attack) is desired, the required CG location (i.e., the heavy segment of the trim line near the sidewall) is not possible to achieve.

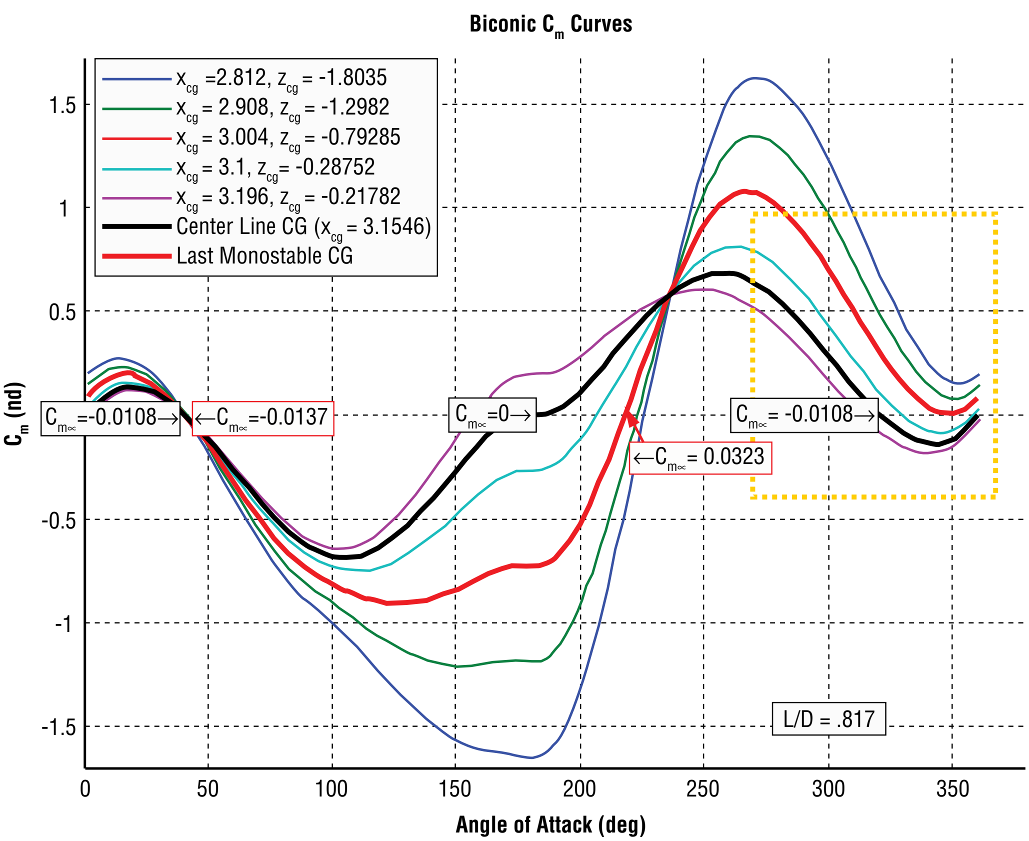


Figure 5-17. C_m Curves for Biconic Vehicle

The plot in **Figure 5-17** shows the pitching moment coefficient (C_m) curves for different points along the blue trim line. The portion of the graph in the gold box highlights the second stable trim point, where the curve crosses zero with a negative slope. The red curve is for a CG at the first monostable point; it is the last curve that does not intersect zero in the gold box. This first monostable CG is the point where the fine blue line ends and the heavy red line begins in the figure.

The data used in this study was generated by a simple, modified Newtonian aerodynamics code. The gold box in **Figure 5-17** is highlighted to indicate the belief that, based on wind tunnel analyses of a related configuration, the second trim point does not really exist in actual flight and would disappear with more robust Computational Fluid Dynamics (CFD) analysis. If the second trim point does exist as these curves suggest and monostability is required, this design is not feasible. However, this does not eliminate slender bodies altogether. Other studies have demonstrated that a bent biconic shape could remove this second trim. The negative aspects to a bent biconic are a loss of symmetry, an increase in configuration complexity, and more volume existing in the opposite direction of the desired CG placement.

The ellipsled vehicle exhibits very similar characteristics to the biconic. The main differences are that the monostable limit is closer to the centerline of the vehicle (37 percent of the vehicle radius as opposed to 79 percent of the radius), and the trim line is less vertical. However, this is still not a realistically achievable CG location to achieve monostability.

The blunt bodies are appealing because they are simpler and have historical precedent. The Apollo capsule vehicle is shown in **Figure 5-18** with two trim lines (0.3 and 0.4 L/D). **Figure 5-19** displays C_m curves for an L/D of 0.3 (CG locations along the solid trim line).

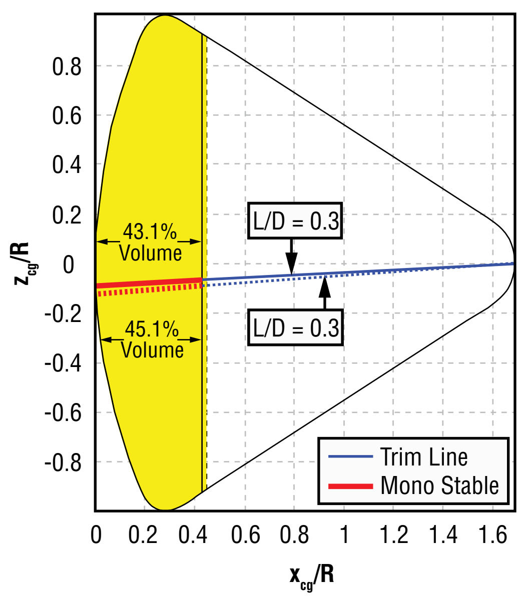


Figure 5-18. Blunt-Body (Apollo) Vehicle: Trim Lines with OML

The Apollo vehicle shows that the trim line is closer to the centerline and gives a larger percent volume that is monostable than the slender vehicles. The location of this trim line is desirable, as it stays close to the centerline throughout the vehicle’s length.

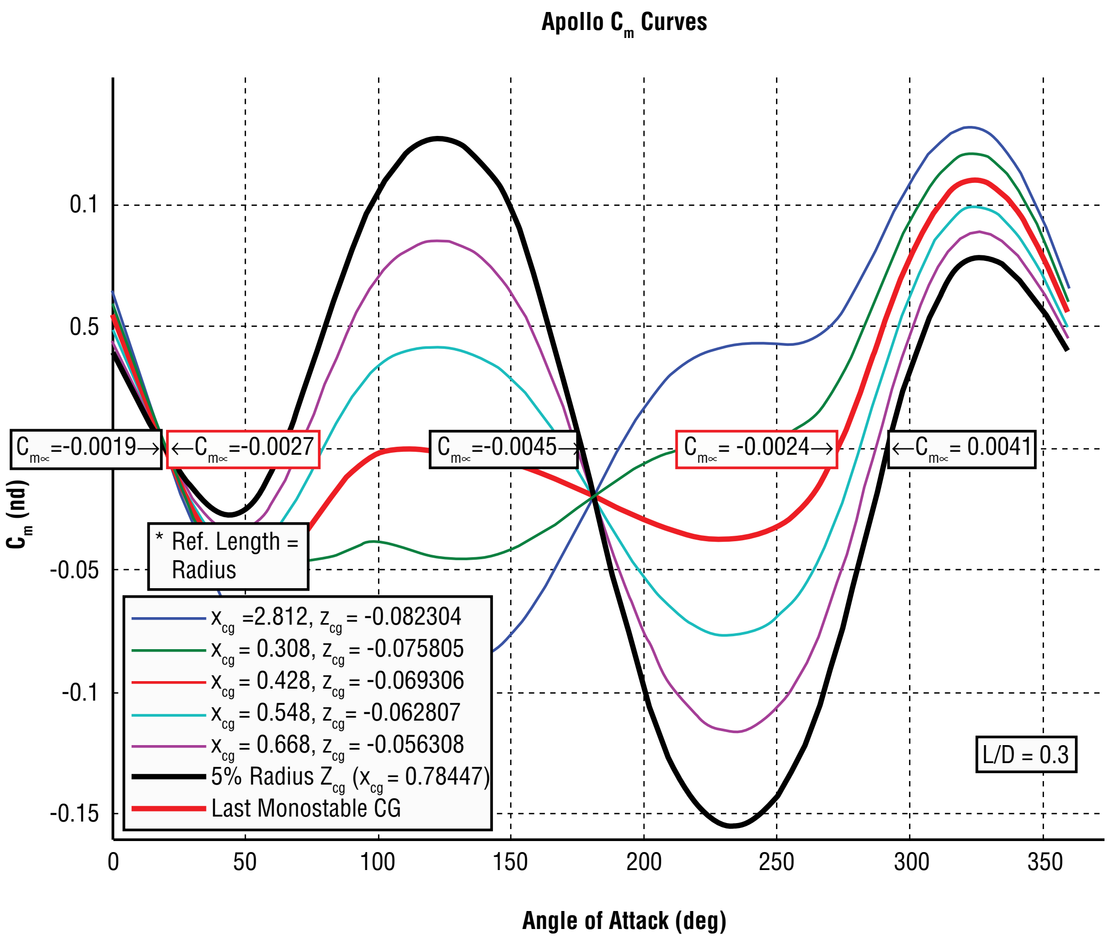


Figure 5-19. Blunt-Body (Apollo) Vehicle: C_m Curves

For comparison purposes, **Table 5-8** shows the stability metrics.

Table 5-8. Stability Comparison Between Slender and Blunt Bodies

		Ellipsled	Biconic	Apollo
Monostability	% Vol	42%	35%	43-45%
	Z_{cg} Offset	37% R	79% R	7% R
L/D Sensitivity	To Z_{cg}	0.001/cm	0.002/cm	0.016/cm
	To X_{cg}	0.002/cm	0.008/cm	0.001/cm

To summarize the aerodynamic stability trade, the Apollo capsule (blunt-body) has more favorable monostability characteristics and the lowest sensitivity to X_{cg} variations, but the least favorable L/D sensitivity to Z-axis CG (Z_{cg}). This is because the trim lines for the capsule are more parallel to the X-axis. For slender bodies, monostability appeared infeasible based on simple Newtonian aerodynamic data, though some existing wind tunnel data suggested it may be better than Newtonian aerodynamic data suggested. In any case, the wind tunnel data would require much more analyses. For blunt bodies, monostability appeared feasible, but the actual Apollo Program could not achieve a CG close enough to the heat shield to produce it. However, it appeared that the capsule shape could be refined to produce an OML that provided monostability, with a CG relatively higher in the capsule than Apollo (i.e., with greater percentage of the OML volume between the needed monostable CG and the heat shield). Obviously, the Soyuz OML has been able to achieve this.

5.3.1.2.4 "Passive," Ballistic Entry Analysis

Another key area of performance investigation was the ability to perform a ballistic entry without an active primary GN&C or power system. In this section, "passive stability" is understood as the capacity of the spacecraft to orient itself to the nominal attitude from an initial off-nominal attitude and/or angular rate without the assistance of an RCS or a stabilizer. Note this requires the vehicle to be monostable, but a backup RCS could also be used to damp rates and/or spin the vehicle for ballistic entry.

This analysis was carried out as a cooperative effort between NASA Centers using both the Six-Degrees of Freedom (6–DOF) simulation tool Decelerator System Simulation (DSS) and the Program to Optimize Simulated Trajectories (POST II) tool. In order to validate both DSS and POST II simulations, a simulation-to-simulation comparison was performed using a scaled Apollo module with excellent comparable results.

Passive stability was investigated for:

- Three shapes:
  - Blunt-body capsule (Apollo),
  - Slender body, biconic, and
  - Slender body, ellipsled.
- Three scenarios:
  - Ascent abort (using CLV - LV 13.1) for worst-case heat rate and heat load cases,
  - Entry from LEO, and
  - Lunar return.

The following specifications for the vehicles were used:

- Blunt-body (Apollo):
  - Actual aerodynamics database,
  - CG on 20-deg alpha trim line (L/D is approximately 0.3),
  - Maximum reasonable monostable position, and
  - $X_{cg}/D = 0.745$; $Z_{cg}/D = 0.04$ (where D is the vehicle diameter).
- Biconic/ellipsled:
  - Modified Newtonian aerodynamics,
  - Secondary trim existed for reasonable CG location listed below,
  - CG on 40-deg alpha trim line,
  - Biconic – $X_{cg}/D = 1.56$; $Z_{cg}/D = –0.0918$; L/D is approximately 0.82, and
  - Ellipsled – $X_{cg}/D = 1.41$; $Z_{cg}/D = –0.0877$; L/D is approximately 0.65.

The assessment of performance includes a heat rate limit criteria and NASA-sanctioned crew load limits criteria. For the heat rate limit criteria, success (or no violation) is declared if, in the time interval when the heat rate is above 20, the attitude oscillations are confined to a safe region. If a small percentage (less than 20 percent) of the oscillations fell outside this heat safe region, no violation would be declared. The heat rate model used in the study was the Detra Kemp Riddell convective model. The radius considered in the model (the corner radius in the blunt-body and the nose radius in the slender bodies) was the smallest one exposed to the flow for each shape.

For the load limit criteria, success is declared if medical maximum allowable load crew limits are not violated in any axis. The medical limits from NASA Standard 3000 are established in charts that show the load in g's as a function of maximum time duration at that load, with each axis having an associated load limit chart. A sample chart depicting the acceleration limit along the X-axis versus the total duration in seconds is shown in **Figure 5-20**.

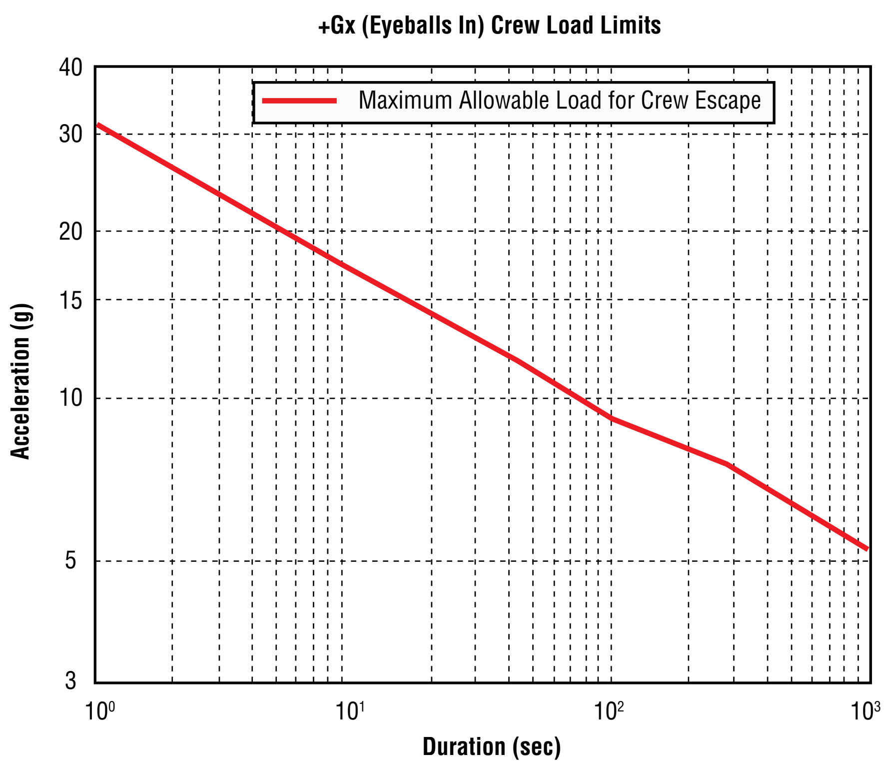


Figure 5-20. Maximum Allowable Load for Crew Escape in the +X Direction (Eyeballs In)

For each scenario and vehicle type, two kinds of 6–DOF tests were run, including:

- With zero initial angular rates and a zero initial sideslip angle, the initial attitude is varied on angle-of-attack (alpha) only.
- With initial attitudes being apex forward in the blunt-body and nose forward in the slender bodies, the initial pitch rate is varied from -10 to +10 deg/s. Yaw and roll rates are initialized to zero.

(Hereafter, these will be referred to as "Test type 1" or "Tt 1" and "Test type 2" or "Tt 2," respectively.)

Conclusions of the trade between slender and blunt bodies' passive stability is summarized in **Table 5-9**. The blunt bodies have slightly more tendency to be able to recover from off-nominal initial attitudes than the slender bodies, but both appear to be able to handle any off-nominal attitude, assuming they are monostable. The axisymmetric slender bodies, however, were shown to require very unreasonable CG locations for monostability.

Table 5-9. Conclusions of the Trade Between Slender and Blunt Bodies' Passive Stability

(Tt = Test type)	Blunt Body (Apollo)	Biconic	Ellipsled
Ascent abort	Tt 1. Acceptable aborts from any initial attitude Tt 2. Acceptable aborts from -2 to +2 deg	*Tt 1. Acceptable aborts from any initial attitude Tt 2. Acceptable aborts from -2 to +1 deg	*Tt 1. Acceptable aborts from any initial attitude Tt 2. Acceptable aborts from 0 to +2 deg
Entry from LEO	Tt 1. Acceptable aborts from any initial attitude Tt 2. Acceptable aborts from -2 to +2 deg	Not performed due to time but assumed similar to the ascent abort case	Not performed due to time but assumed similar to the ascent abort case
Lunar return	High onset of loads and heating associated with lunar return precludes passive stability from working for any appreciable initial attitude rates or off-nominal attitudes		

**These results account for the assumption that the secondary trim point was removed.*

The conclusion for the lunar return cases was later discovered to be an artifact of the analysis technique—multiple cases were skipping or pulling lift-down because the vehicle was not spun-up after the trim attitude was achieved. When proper spin-up of the vehicle is achieved to null the lift vector, results are more favorable.

Introducing a bank rate to null the lift vector effect on the trajectory (ballistic abort) was then investigated. A bank maneuver consists of the rotation of the spacecraft about the velocity vector. This rotation results in gyration of the lift vector, thus producing a ballistic trajectory. An initial bank rate was set via a combination of body axis roll and yaw rates. No damping aerodynamic terms were used, although, at the velocities and altitudes of concern, very little damping would occur in any case. Due to the presence of cross products of inertia and the fact that the principal axis of inertia is not aligned with the trim angle, the initial bank rate oscillates and changes with time—particularly for the slender bodies.

Figures 5-21 and **5-22** show the angle of attack (alpha), sideslip angle (beta), and bank angle time histories with an initial bank rate. The scenario is an ascent abort with worst heat rates at reentry, with the trim attitude being the initial attitude. Initial bank rates are 20 deg in the Apollo and biconic cases and 25 deg for the ellipsled. This is shown as:

Time Histories with Initial Bank Rate = 25 deg/s ellipsled, 20 deg/s biconic.

A comparison between **Figures 5-21** and **5-22** clearly indicates a better performance of the blunt-body with respect to that of the slender bodies. Whereas the Apollo shape is maintaining a reasonable attitude, the biconic and ellipsled shapes are both tumbling at the onset of the simulation. This tumbling is attributed to the principal moments of inertia being nearly aligned with the body axis rather than the trim angle of attack (40 deg away from the principal axis), about which the banking maneuver is being performed.

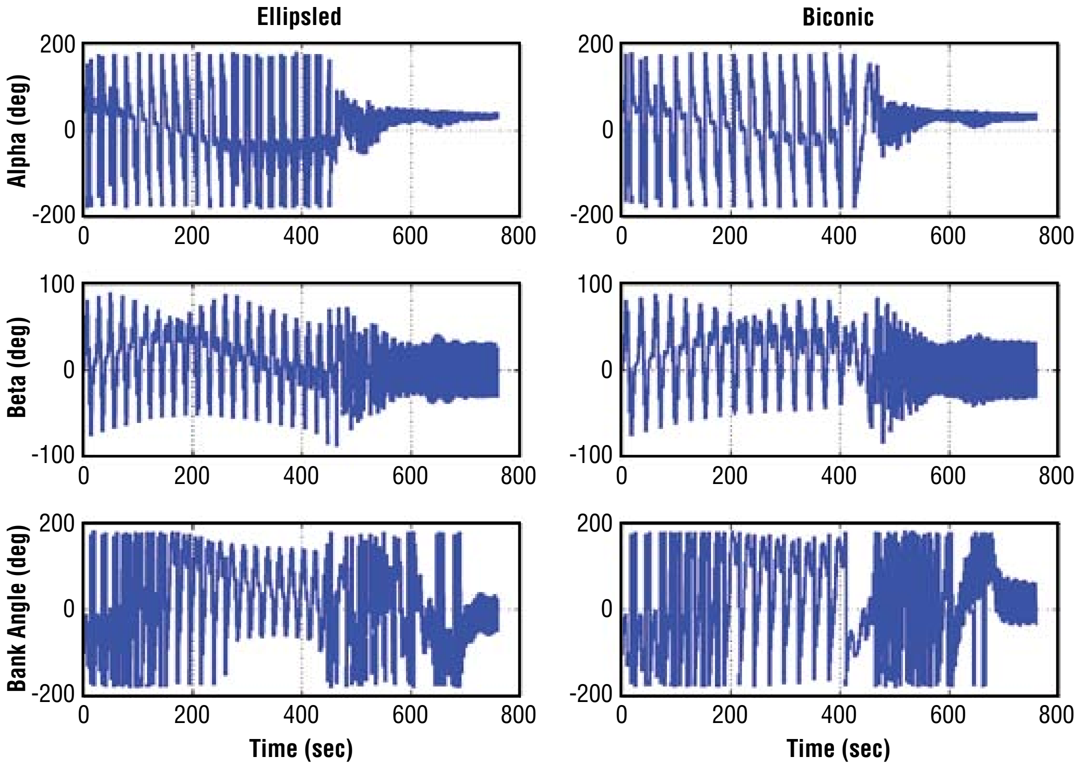


Figure 5-21. Slender Bodies: Angles of Attack (alpha), Sideslip (beta), and Bank

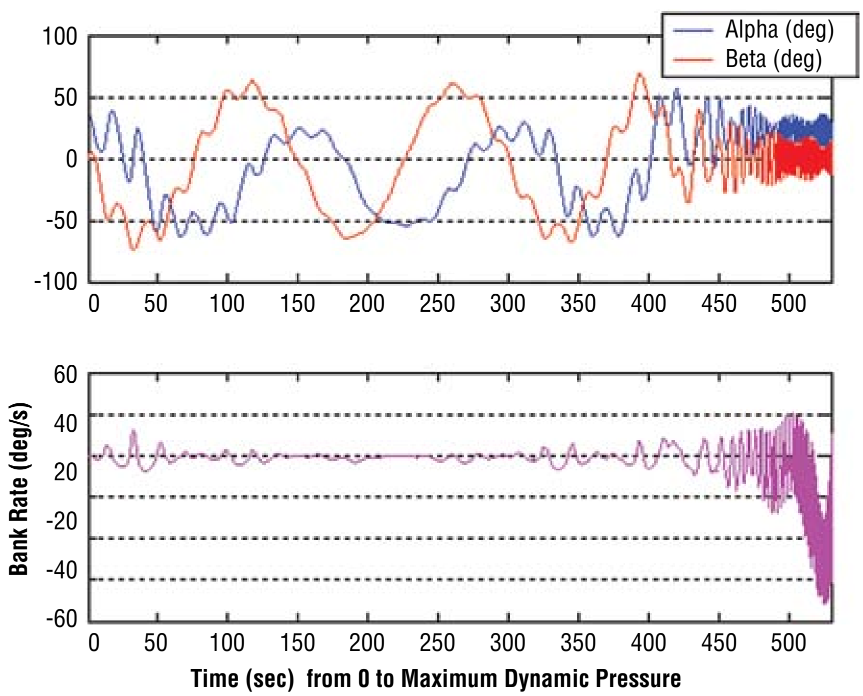


Figure 5-22. Blunt-Body (Apollo): Angles of Attack (alpha), Sideslip (beta), Bank Rate Time Histories with Initial Bank Rate = 20 deg/s (Ascent abort with worst heat rate at reentry. Initial attitude is the trim attitude.)

A lunar return case with initial bank rate was further investigated. As shown in **Figures 5-23** and **5-24**, the presence of cross products of inertia results in larger amplitude of oscillations in alpha, beta, and bank rate for a capsule, although both are acceptable.

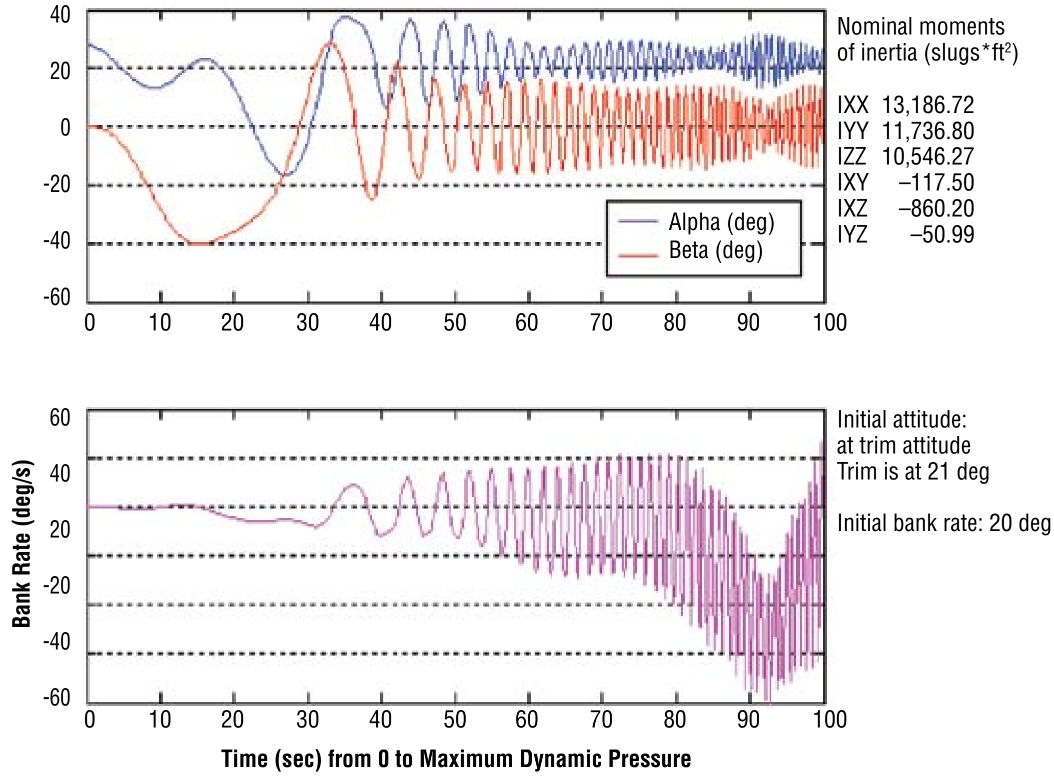


Figure 5-23. Blunt-Body (Apollo) Ballistic Entry with Initial Bank Rate = 20 deg

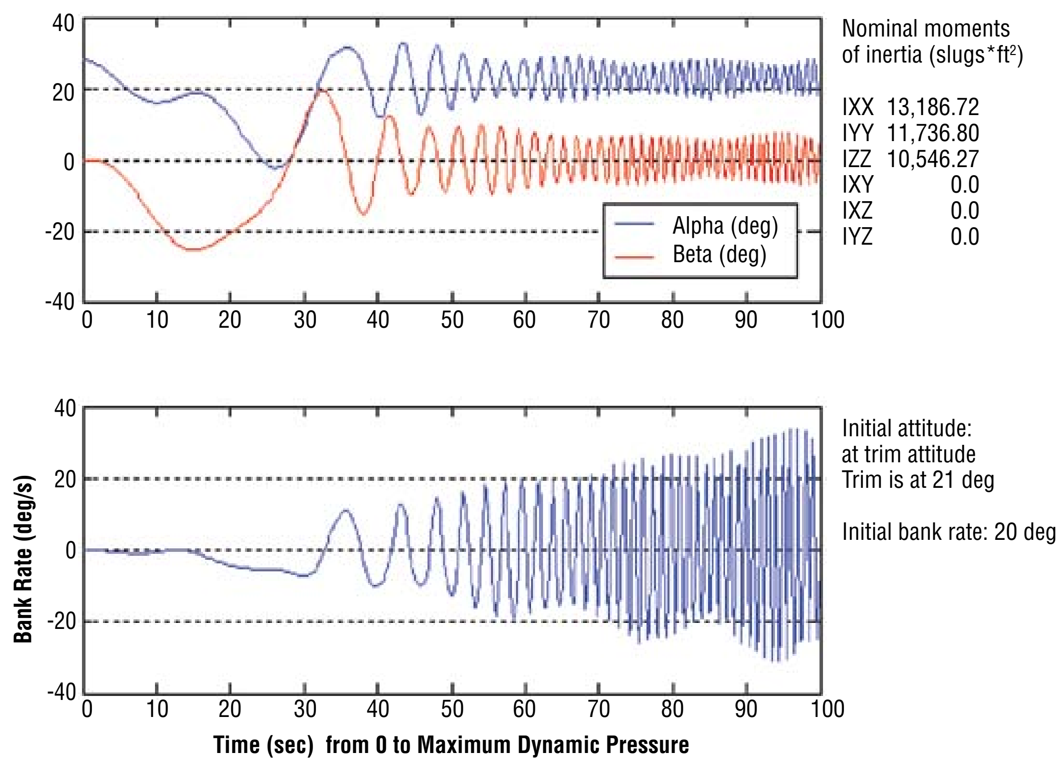


Figure 5-24. Blunt-Body (Apollo) Ballistic Entry with Initial Bank Rate = 20 deg

The tumbling motion of the biconic and ellipsled bodies can be avoided by waiting until there is enough dynamic pressure to fight the effect of the moments of inertia. The effect of a bank rate induced late in the flight in the case of a biconic body is presented in **Figure 5-25**. It is uncertain whether it would be allowable to initiate the bank rate this late in the trajectory under all abort cases. There may be abort situations when it is desirable to have the SM create the bank rate before it separates from the CM. In this case, it appears the slender bodies would have difficulties with dynamics during entry.

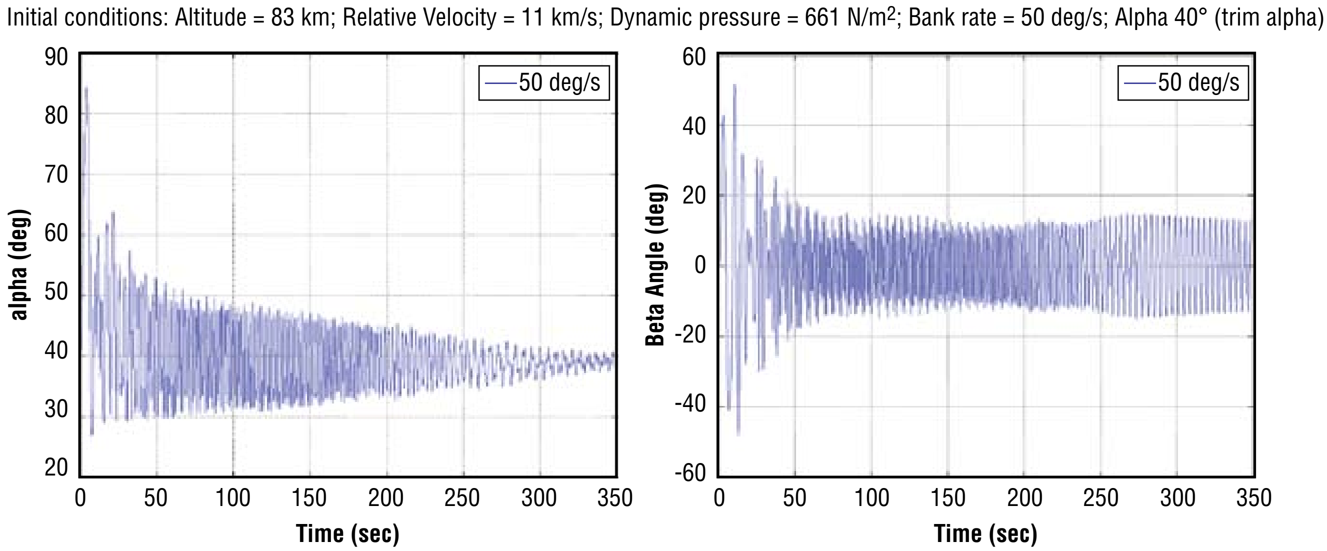


Figure 5-25. Slender Body (Biconic): Angles of Attack (alpha) and Sideslip (beta) Time Histories with Bank Rate Induced Late in the Flight

The conclusions from the trade between slender bodies and blunt bodies using a bank rate to null the lift vector are as follows: Slender bodies are difficult to enter ballistically (without RCS maintenance) unless spin-up occurs very late in the trajectory, after sufficient aerodynamic forces are generated to help stabilize the vehicle. This is due to large inertial cross-coupling. This behavior hinders the ability to spin them up using the SM before entry in case of Command Module RCS total failure. Blunt bodies can be spun up from entry or later.

5.3.1.2.5 Blunt Bodies versus Slender Bodies Comparison Summary

After all performance analyses, simulations, and evaluations were made on the representative vehicles, the spreadsheet in **Appendix 5B**, **CEV Crew Module Shape Trade Data**, was filled out. Key items of discrimination were then flagged as follows:

- Green: a particularly advantageous feature;
- Yellow: a design challenge, operational limitation, or requiring small technology development; and
- Red: a major design challenge, operational impact, or significant technology advancement required.

For the blunt-body, the key benefits were found to be:

- A more familiar aerodynamic design from human and robotic heritage—less design time and cost;
- Acceptable ascent and entry ballistic abort load levels;
- A proven passive, ballistic abort method (as performed on Soyuz);

- Crew seating orientation ideal for all loading events;
- Easier LV integration and controllability;
- TPS not exposed during mission;
- Possible early use of reusable TPS rather than ablator (ISS and LEO missions); and
- If land-landing approach fails, water-landing capability is a known fallback solution.

Major challenges appeared to be:

- Long-range skip-entry or aerocapture techniques must be used to achieve a CONUS land landing from the Moon for anytime return; and
- Land-landing stability (preventing tumbling) and load attenuation may be a significant challenge. (Note that Soyuz tumbles on 80 percent of landings.)

Minor challenges discovered were:

- Requires capsule reshaping or better packaging for CG (compared to Apollo) to achieve a monostable vehicle;
- Requires adequate free-fall time during high-altitude ascent aborts to separate the SM and rotate the capsule to a heat shield-forward attitude;
- Land-landing sites in CONUS must be very near the West Coast for proper SM disposal and potential ballistic abort entry; and
- Land landing generates limitations for ISS return opportunities, which can be solved by proper mission planning and multiple CONUS sites.

For the slender body, the most important benefits were:

- The SM can be integrated and potentially reused, which:
  - Allows use of further inland land-landing sites—at least 550 nmi. (However, this may be extremely limited due to protection for population overflight.)
  - Easily provides necessary delta-V and ECLSS for an aerocapture or skip-entry return.
- The vehicle attitude is pre-set for launch abort, i.e., the vehicle does not need to “flip around” to get the heat shield forward on ascent abort;
- Better separation of alternate landing sites for weather avoidance;
- At least daily land-landing opportunities for routine ISS return or medical mission, although this was not a requirement; and
- Lunar return can land on land in south CONUS using Apollo up-control guidance.

Significant challenges for the slender body were found to be:

- Crew seats (and displays/controls) must rotate 90 deg in flight to achieve proper load direction for ascent versus entry/landing;
- Ballistic ascent abort g-loads are unacceptable to crew survival unless ascent trajectory is significantly depressed like Shuttle; and
- Requires coordinated RCS firings to spin up vehicle properly and may require RCS to maintain banking motion during a ballistic abort due to inertial cross-coupling; hence, no passive re-entry mode would be available during off-nominal entry as Soyuz; and
- Development would take significantly longer and cost more due to added weight and shape complexity.

In addition, minor challenges encountered were:

- TPS exposed on ascent and rest of mission;
- Requires landing orientation control—likely for water or land—and attenuation technology development;
- May require drogue parachute repositioning event (similar problem addressed in X–38);
- Ejection seat design may be challenging to avoid ejection into parachutes;
- Monostable configuration is problematic for axisymmetric shapes—needs detailed aero analysis; and
- Lunar return heating is extremely high (heavy heat shield).

5.3.1.2.6 Blunt Bodies versus Slender Bodies Conclusions

To summarize the results, it appeared that the capsule configurations have more desirable features and fewer technical difficulties or uncertainties than the slender body class of vehicles. Because one of the primary drivers for the selection was the minimal time frame desired to produce and fly a vehicle, the blunt bodies had a definite advantage. All the human and robotic experience NASA has had with blunt bodies has led to a wealth of knowledge about how to design, build, and fly these shapes. A slender, lifting entry body (without wings, fins, or control surfaces) has never been produced or flown by NASA.

The blunt-body has been shown in previous programs to be able to meet the requirements of the LEO and lunar return missions. However, the new desires expressed for a CEV produce some uncertainties and challenges. Perhaps the major concern is the land-landing design challenge, including the skip-entry, sites selection, and impact dynamics. However, the capsule approach has a proven water-landing capability that can be used as a fall-back approach if further studies show the land landing to be too costly, risky, or technically difficult. Another challenge would be to develop a shape to more easily achieve monostability (as compared to Apollo) and achieve more than 0.3 L/D (at least 0.4 L/D). An L/D of 0.4, which appears achievable, is necessary to provide reasonable CONUS land-landing sites in terms of number, size, in-land distance, and weather alternates, and to increase the return opportunities. In addition, it provides lower nominal g-load and better skip-entry accuracy, which reduces skip delta-V requirements. This may result in higher heating on the shoulder and aft side of the vehicle, but this does not appear to be a great TPS concern.

The slender body class of vehicles has several characteristics that create concern about the time required for development. The trade study analysis and spreadsheet results do not indicate that a slender body would be infeasible, simply that there are several concerns and design problems that would require further significant analyses, design iterations, trades, testing, and development. First, they would require substantially more aerodynamic, aerothermodynamic, and TPS design and development work than a blunt-body. Second, the loads directions issue would need to be solved, including potential crew seat rotation, landing orientation control, and landing attenuation. Water-landing impacts and dynamics would need extensive design and test work done. The ascent trajectory would need to be tailored (depressed) to reduce ballistic ascent abort loads due to the fact that slender bodies have high ballistic numbers.

Additionally, the ballistic abort mode problem would need to be solved. At first glance, the slender bodies do not behave dynamically stable when spun up to null the lift vector. They appear to require RCS control or very judicious mass placements for inertias alignment. As an alternative, a configuration with an independent, separable abort capsule could be designed to eliminate the passive, ballistic abort concerns, but this is difficult to design for crew load orientations and difficult to design without adding substantial weight for additional TPS, recovery systems, etc. The ability to integrate an SM into a slender body design is advantageous, but creates an extremely massive entry vehicle and limits descent options to three very large, round chutes due to mass.

The conclusions from the capsules versus slender bodies trade were:

- Using an improved blunt-body capsule is the fastest, least costly, and currently safest approach for bringing lunar missions to reality; and
- Improvements on the Apollo shape will offer better operational attributes, especially with increasing the L/D, improving CG placement feasibility, and potentially creating a monostable configuration.

Based on this preliminary trade study, the class of blunt bodies was selected for further investigation to ultimately define a CEV CM shape.

5.3.1.3 Capsule Shape Trade

5.3.1.3.1 Driving Factors

In the trades between blunt body and slender body classes of vehicles, representative vehicles were adequate for downselect. Within a class, however, optimization requires parameterization. Multiple basic capsule shapes were available to investigate as potential CEV CM OML candidates. The driving factors, particularly for a capsule OML, that resulted from the initial trade study were as follows:

- L/D of 0.4 is required to achieve the necessary range capability between the landing site and SM disposal for the ISS missions, as well as to increase the performance and accuracy of the skip-entry for lunar returns and to reduce delta-V requirements. In addition, increased cross-range capability resulting from increased L/D helps to reduce the number of landing sites and time between opportunities for ISS return;
- Ballistic abort capability, including monostability;
- Satisfaction of acceleration loads across the spectrum of flight conditions within crew limits;
- Feasible, attainable CG requirements;
- Adequate static stability and low sensitivity of L/D to Z_{cg} dispersions (approximately the same or better than Apollo);
- Adequate volume and shape for crewed operations;
- Reusable TPS on the aft-body;
- Low technology requirements; and
- Short development time.

5.3.1.3.2 Axisymmetric Capsule Shape Variations and Effects

The basic capsule shapes shown in **Figure 5-26** were analyzed using a modified Newtonian aerodynamics code. Various shape parameters, such as after-body cone angles, base radii, corner radii, heights, and others, were parametrically changed and evaluated in the aerodynamics generator to assess the effects of these parameters on the desirable criteria. Of primary interest were the sidewall angle (theta), the corner radius (Rc), and the base radius (Rb).

The data quickly indicated the desired path to pursue. The shapes similar to Soyuz could not attain the 0.4 L/D without high angles-of-attack and excessive acreage of after-body sidewall heating. Although after-body TPS could be made to handle the environments (Soyuz and Apollo employed ablative TPS across the entire vehicle), better shapes were available for possibly achieving the desired reusable after-body TPS. The Gemini/Mercury class of shapes showed no significant advantage over plain cones and required more Z_{cg} offset and higher angle-of-attack than plain cones for 0.4 L/D. Although the extended frustum apexes could help increase monostability, a plain cone of the same height was shown to produce more. The Moses-type shapes, while extremely stable with the proper CG, could not attain 0.4 L/D easily. In addition, the crew seating orientation would have to vary to always produce loads perpendicular to crew spines, much as required for the slender bodies. As on the Aero-assist Flight Experiment (AFE) shape, the non-axisymmetric heat shield would produce CG and angle-of-attack benefits but has no flight heritage. The ESAS team decided to leave this AFE shape for further analysis as a potential improvement over a plain conical, axisymmetric shape. The conical, axisymmetric shapes such as Apollo were determined to be preferable since they had the best experience base and aerodynamic familiarity while being capable of producing the desired L/D, monostability, low technology needs, and ease of fabrication due to axisymmetry. Hence, they were found to merit further trade analyses and investigation.

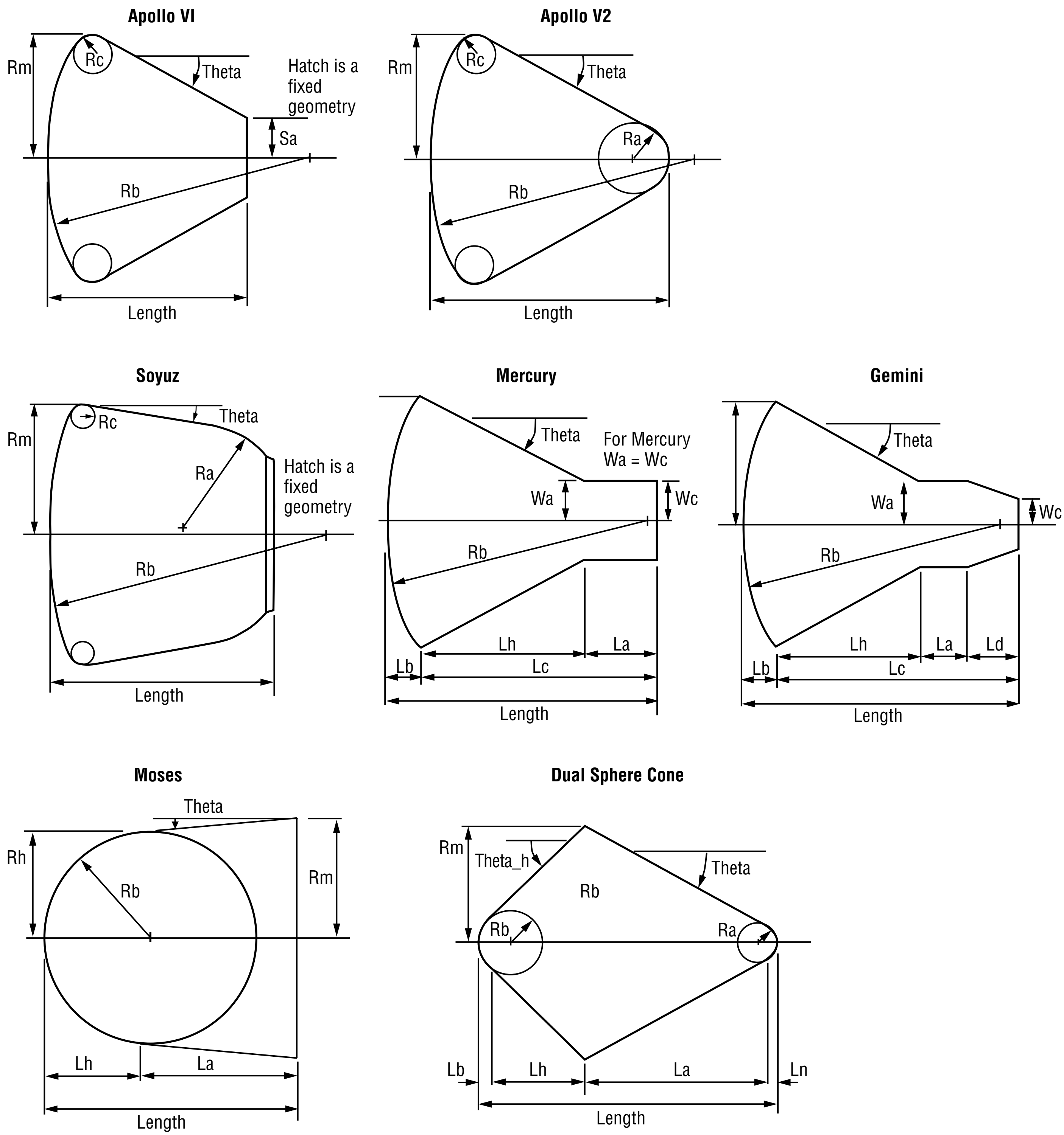


Figure 5-26. Initial Set of Capsule Shapes and Parametric Variables Considered

Figure 5-27 shows an example of how the effect of parameter variations was measured. The figure is a 3–D contour plot. Each intersection point on the colored contour curves represents a different analyzed case. The corresponding value for each case is measured by the values on the Z-axis. In this figure, the quantity of interest is the percent volume below the last monostable CG location (moving away from the heat shield) for 0.4 L/D. This value represents an important quantity for packaging if a monostable vehicle is desired. In this particular figure, a 40 percent contour is also shown—an arbitrary metric for desired volume. An ideal packaging percent volume would be 50 percent if the objects in the vehicle were of uniform density. From this plot, the best vehicle for monostability would have a small sidewall angle (theta), a small base radius, and a large corner radius. Of these three parameters, a corner radius was the largest discriminator, followed by the sidewall angle (theta).

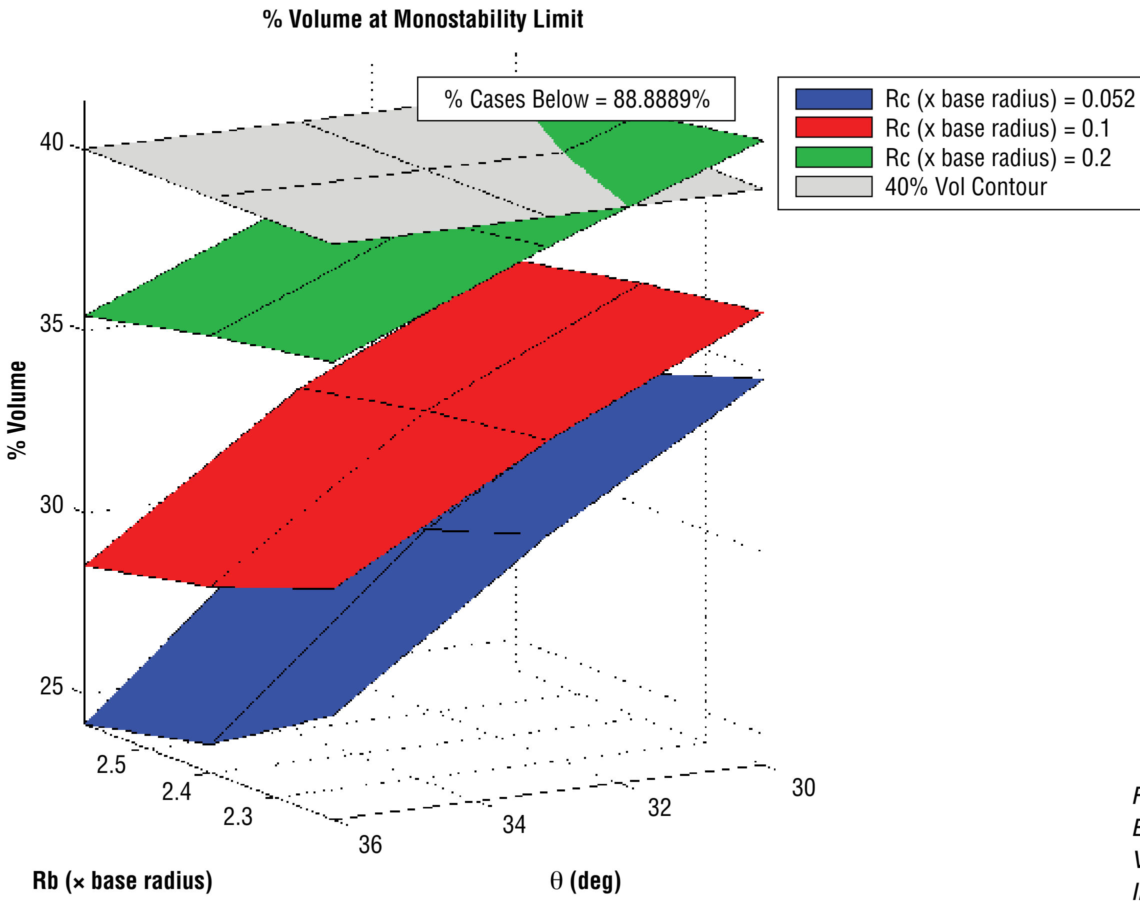


Figure 5-27. Example of Effect of Parameterized Variables on Quantity of Interest (Monostability)

Each of the parameters influenced the important factors in different ways, but all of the blunt-bodied vehicles exhibited similar trends, regardless of their original shape. **Table 5-10** shows the overall affect of this parameterization. The arrows indicate if the parameter (first row) should increase or decrease for a desired quality (left column) to improve.

Table 5-10. Trends Associated with Parameterized Values for Conical Shapes (Based on Constant Height)

	Corner Radius	Base Radius	Cone Angle
To decrease Z_{cg} offset required for 0.4 L/D	↓	↑	↓
To increase % volume below first monostable CG	↑		↓
To increase C_m-alpha magnitude (static stability) at 40% volume CG	↓	↓	
To decrease heat rate	↑	↑	
To decrease sensitivity of L/D to Z_{cg} at 40% volume CG	↑	↓	
To decrease X_{cg}/D at 40% volume CG (effects landing stability)	↓	↑	↑

As illustrated in each column, many of the desired characteristics conflicted with each other. There was no clear variation in a single parameter that would help in all areas. It would require a weighting and compromise of the various desired characteristics to produce a "best" set of vehicle shape parameters. Generally speaking, the desire for monostability corresponded with improved L/D sensitivity to Z_{cg} and heat rate (perhaps two of the least demanding desires), but conflicted with all other (more important) characteristics. Thus, it became difficult to establish an optimal vehicle shape, especially since the requirements for these vehicles were not well defined.

In order to arrive at a desirable sidewall angle, a simultaneous comparison of all parameters was needed. Thus, the vehicles were compared side-by-side with a table of relevant aerodynamic characteristics. The following figures show some of the noted trends.

Figure 5-28 shows how changing only the length affects the vehicle performance characteristics. **Figure 5-29** shows the effect of a changing sidewall angle. A careful study of these vehicles reveals that the length of the aft cone generally has little effect except for one main difference: a longer cone is more monostable. This means there is a greater percentage of the total OML volume below the minimum monostable CG for a longer cone height. Therefore, in theory, the longer cone height OML should be easier to package and attain a monostable condition. If the length is held constant, and the aft cone sidewall angle is changed, the figures show that a smaller (shallower) angle is more monostable. (The CG position for monostability allows a greater percentage volume between the CG position and the heat shield.) However, CG height for constant volume is relatively higher in the vehicle with the smaller sidewall angle. The other parameters vary very little. Other variations were examined, including beveling and rounding of the top of the cone. Besides bringing the trim line only slightly closer to the centerline, the biggest effect of an increase in bevel angle or rounding radius was a decrease in monostability (an undesirable result).

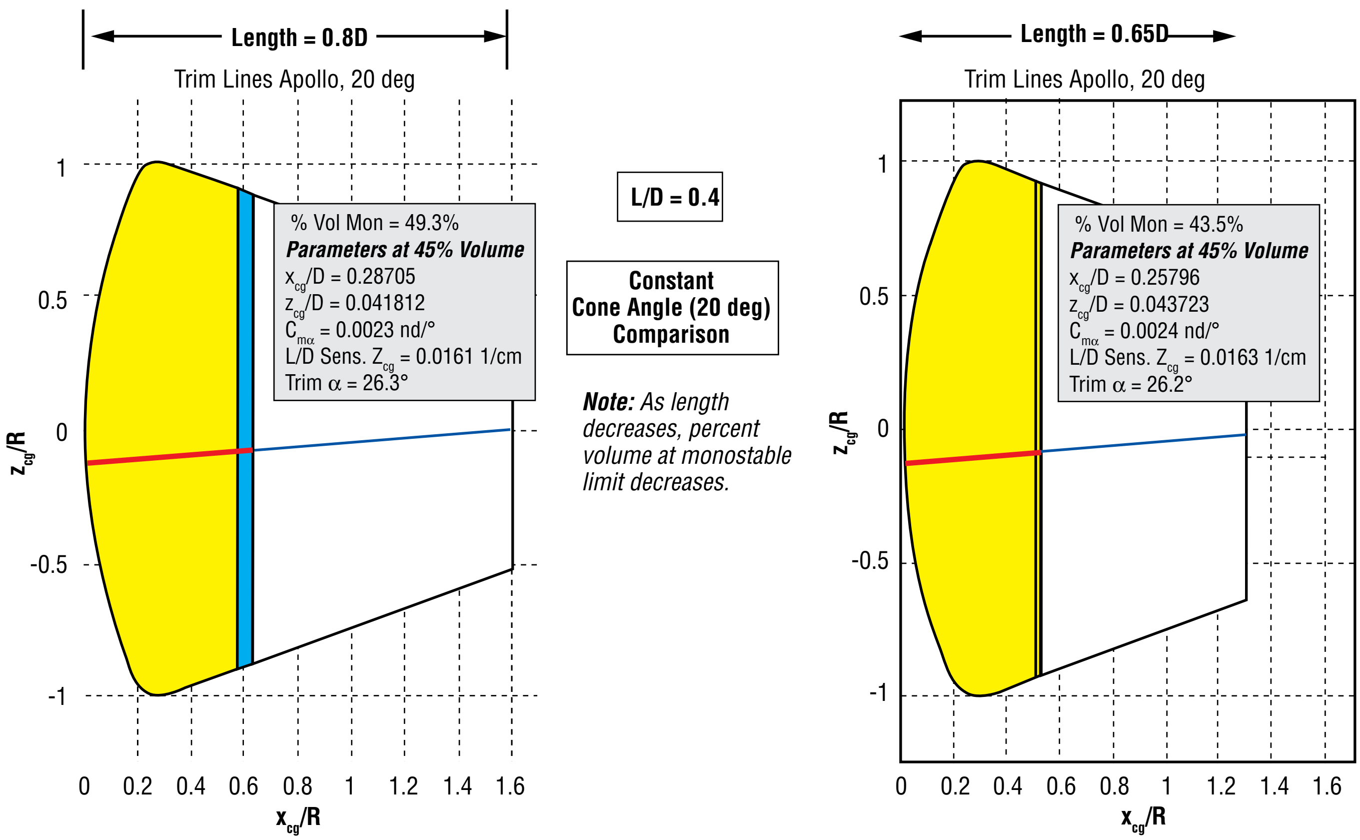


Figure 5-28. Affect of Length on Aerodynamic Characteristics

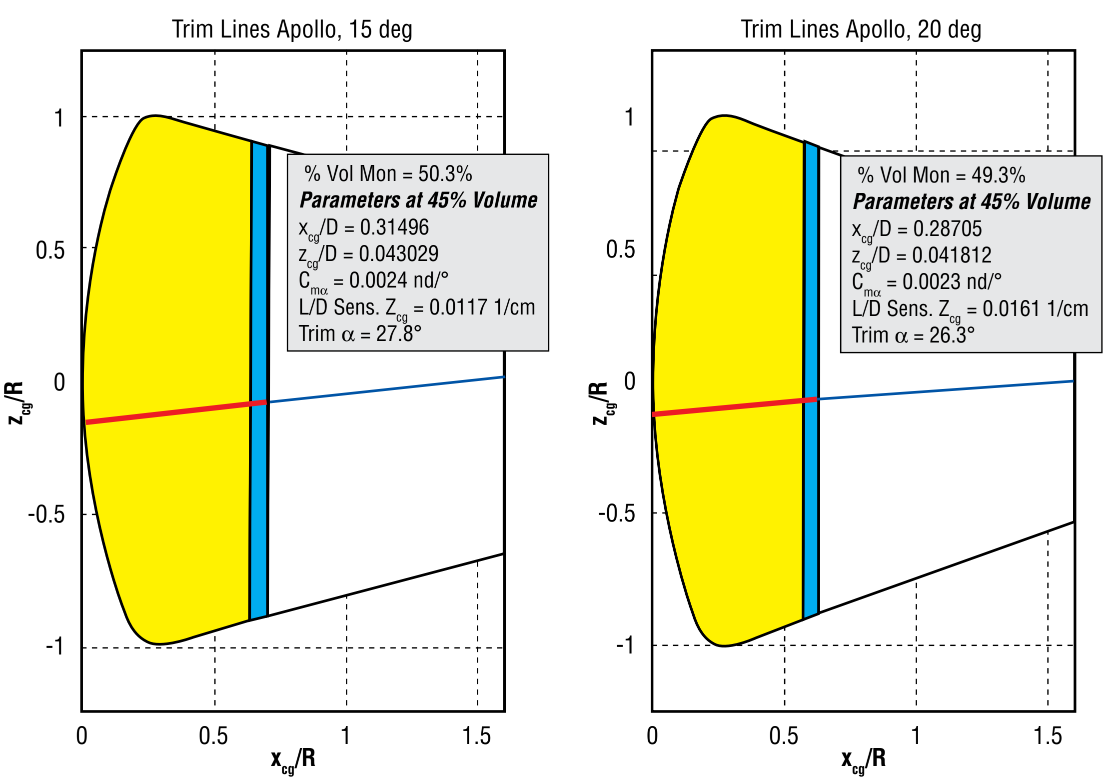


Figure 5-29. Affect of Sidewall Angle on Vehicle Aerodynamics at Constant Length

5.3.1.3.3 Initial Axisymmetric Capsule Shape Downselect

In order to balance the effects of the changing parameters, a baseline vehicle was selected with a shallower cone angle of 20 deg (since this had the least effect on other parameters), with the same base and corner radius as Apollo. This new vehicle trended toward the family of vehicles represented by the Soyuz capsule, which has an even shallower sidewall angle. This vehicle is shown in **Figure 5-30** below. It was estimated that an achievable X-axis center of gravity (X_{cg}) position would lie at or around the 45 percent volume level. In that case, the Z_{cg} offset required for 0.4 L/D would be roughly 0.053 times the diameter. For this shape, the monostable CG position could be as high as the 48.6 percent volume level, which would therefore leave some margin for assured monostability.

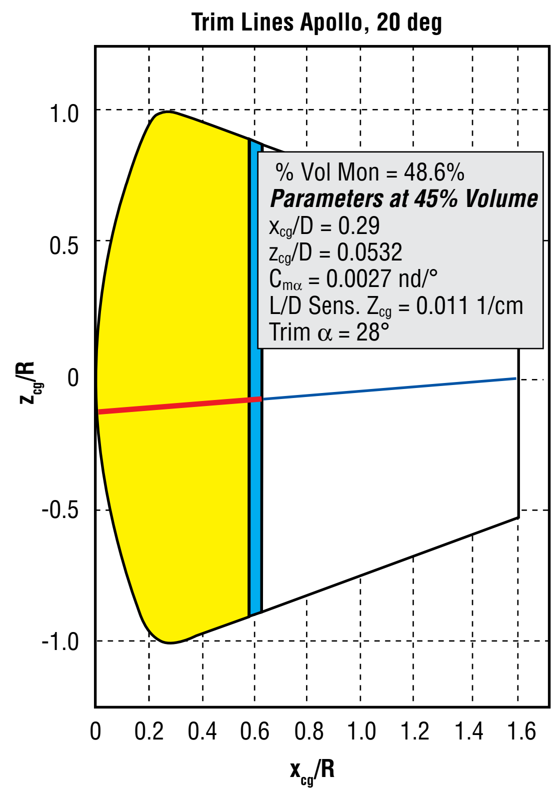


Figure 5-30. Initial Baseline Capsule Data

Figure 5-31 shows the pitching moment coefficient (C_m) curves versus angle-of-attack for this vehicle. The black line shows the C_m curve for the desired CG position at the 45 percent volume level. The red line corresponds to the first monostable CG position at the 48.6 percent volume level. The blue line designates a bi-stable CG position closer to the apex that was arbitrarily chosen for visualization.

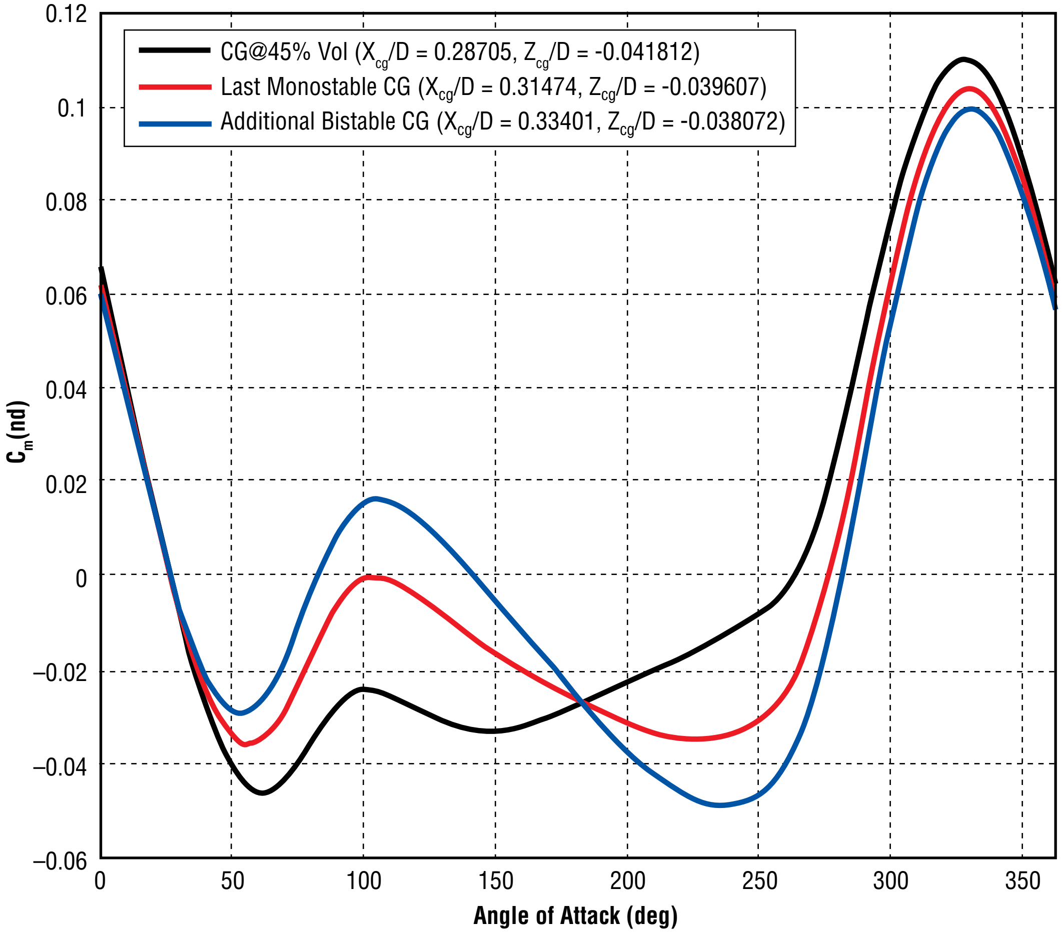


Figure 5-31. Pitching Moment Coefficient Curves for the Baseline Capsule

5.3.1.3.4 Axisymmetric Capsule Shape Variations

One way to achieve the required L/D is to use a nonaxisymmetric shape similar to the AFE shape mentioned previously. A computer-generated shape optimization approach was pursued to attempt to optimize an OML that exhibited some of the desirable characteristics without necessarily being axissymmetric.

The investigation of various "optimized" shapes used the optimization capabilities of the CBAERO computer code. These optimized shapes held the aft-body shape fixed, while the heat shield shape was optimized to meet the trim and L/D constraints. CBAERO permits the very general optimization of the configuration shape, where the actual nodes of the unstructured mesh are used as the design variables. For instance, a typical capsule mesh contained approximately 20,000 triangles and 10,000 nodes. Full shape optimizations were performed where the Cartesian coordinates (x,y,z) of each node were used as design variables. In the example discussed below, there would be 3 x 10,000 = 30,000 design variables.

Often, only the heat shield was optimized, thus reducing the total number of design variables. **Figure 5-32** shows the axisymmetric baseline CBAERO grid. The orange region contains those triangles that lie within the optimization region (2,774 nodes, or 8,322 design variables). **Figure 5-33** shows one optimization result in which L/D was optimized with the moment constrained to zero and the volume held constant. The resultant geometry exhibits a "trim tab" on the upper windward surface, which the optimizer has produced in an attempt to trim the vehicle while maintaining both the required L/D of 0.4 and the vehicle volume. The surface also exhibits some concavities, which may lead to increased heating or other complex effects. More recent optimization studies have imposed constraints on concavities, and it may be desirable to revisit these optimized shapes or start with the AFE baseline.

Figure 5-32. Baseline Axisymmetric Shape CBAERO Grid

Figure 5-33. An Optimization Result from CBAERO Where the Moment was Constrained to Zero and the Volume Held Constant

The engineering level analysis of CBAERO, as well as efficient coding of the gradient process, enables these optimized solutions to be performed with tens of thousands of design variables and multiple constraints in a matter of minutes-to-hours on a typical desktop Personal Computer (PC). The results shown here typically took 100 to 200 design iterations and less than 60 minutes on a PC laptop.

Various candidate designs were shown to meet both the trim and L/D requirements; however, the complexity of the shapes led to the desire to investigate simpler (but nonaxisymmetric) shapes that might obtain similar results.

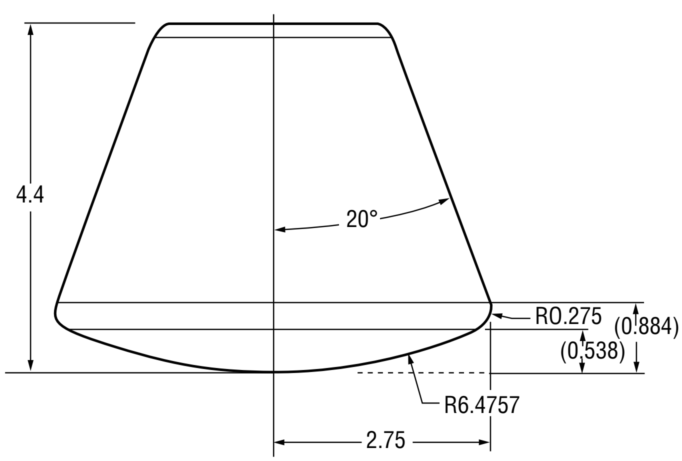
Various rotated heat shield concepts were also investigated to examine their ability to reduce the required "z" offset in the CG to trim the vehicle at the desired L/D of 0.4. The various configurations analyzed were capable of reducing the "z" offset; however, the shapes all failed to meet the required L/D of 0.4.

5.3.1.3.5 Initial Capsule Shape Trade Conclusions

For the initial capsule shape trade study, detailed and extensive analysis of parametric effects and trends of various capsule shapes and features indicated that achieving the desired characteristics was indeed a formidable task. A compromise was made to achieve all of the desired characteristics as closely as possible while minimizing the detrimental effects. The resultant axisymmetric shape (shown in **Figures 5-30** and **5-32**) was a 5.5-m diameter capsule with Apollo heat shield and 20-deg aft-body sidewall angle. The capsule offered large volume (i.e., large enough for surface-direct missions), easily developed axisymmetric shape, the best chance for monostability, L/D = 0.4 with attainable CG, adequate static stability, and low L/D sensitivity to CG dispersions. Nonaxisymmetric shape optimization had shown that this technique could indeed reduce CG offset requirements if needed in the future. Further detailed analysis was then required to further define the performance characteristics of the axisymmetric shape.

5.3.1.3.6 Detailed Aerodynamic Analyses of Initial Baseline Capsule Shape

Once the baseline shape for the CEV was defined as a 5.5-m diameter capsule with Apollo heat shield and 20 deg aft-body sidewall angle (shown in **Figure 5-30**), a number of analyses was conducted to further define the performance and suitability of the selected design. Some specifications are shown in **Figure 5-34**. Data shown in the figure for angle-of-attack and CG location were based on modified Newtonian aerodynamics and were later modified by CFD calculations of the aerodynamics. The CFD aerodynamics give a high-fidelity estimate of the required trim angle and radial CG offset needed for L/D = 0.4. The Newtonian results generally give good estimates of the required trim angle for a given L/D, but underestimate the radial CG offset required to achieve the trim angle. In addition, CFD aerothermodynamics results were used to estimate geometry effects on heating, anchor other predictive tools, and provide input to TPS sizing analyses. Details of aerodynamic CFD analysis, the tools used for CFD aerodynamics, CFD aerothermodynamics, and TPS analysis, as well as the process used for the results presented in this report are included in **Appendix 5C**, **CFD Tools and Processes**.



Baseline CEV Specs
Trim angle-of-attack = 26 deg
Trim L/D = 0.4
Lref = 5.5 m
Sref = 23.76 m^2
Xcg = 0.287D
Zcg = 0.0418D
Lunar return mass = 11,200 kg
ISS return mass = 10,900 kg
Inertias scaled based on Apollo

Figure 5-34. Initial CEV Baseline Specifications

5.3.1.3.7 Detailed TPS Analyses of Initial Baseline Capsule Shape

The general GR&As for the CEV TPS design analysis, modeling assumptions, and data sources are presented below.

Geometry

The OML definition for both the baseline axisymmetric capsule and the AFE-based non-axisymmetric capsule were obtained from the same triangulated surface grid used in the engineering-based aerothermal analysis.

Aerothermodynamics

The aerothermal environments were provided by the CFD-anchored engineering-based CBAERO code (Version 2.0.1). The aeroheating environments consisted of the time history throughout the trajectory of the convective heating (recovery enthalpy and film coefficient), the shock layer radiation heating, and the surface pressure for each surface triangle. No margins on the aeroheating environments were used in the TPS analysis and sizing because conservative margins were used in the TPS analysis.

Trajectory

Both guided entry (nominal) and passive ballistic (abort) trajectories were examined for 8 km/sec (LEO) entry and 11 km/sec and 14 km/sec (Lunar return) entries. In addition, for the 11 km/sec entry, a skipping guided and skipping ballistic trajectory were also provided and analyzed. These trajectories were generated for a nominal L/D ratio of 0.4. Among other flight parameters, these trajectories consisted of the time history of the Mach number, angle-of-attack, and free-stream dynamic pressure. This data was interpolated along each trajectory within the aerothermodynamic database to generate the aeroheating environments.

TPS and Aero-Shell Material and Properties

A summary of the structural and carrier panel aero-shell materials is presented in **Table 5-11**, which includes material selection and representative thicknesses. A similar summary of the TPS materials is presented in **Table 5-12**. For the reusable TPS concepts, the thermal, optical and mechanical properties were taken from the Thermal Protection Systems eXpert (TPSX) online database. A detailed listing of the benefits and concerns associated with each TPS material is given in **Table 5-13**.

Table 5-11. Structural Materials and Thicknesses Analyzed in Studies

Structure Choices	Material	Thickness (cm)
8 km/s aft body	Aluminum 2024	0.2540
	RTV	0.0508
All heat shields, 11 km/s aft body	Graphite Polycyanate	0.0381
	Aluminum Honeycomb	1.2700
	Graphite Polycyanate	0.0381
	RTV	0.0508

The carrier panel aero-shell design consisted of a composite honeycomb panel, with 0.015 inch graphite polycynate face sheets bonded to an 0.5 inches aluminum honeycomb core with a mean density of 8.0 lb/ft^3. The ablator TPS materials were direct-bonded onto the carrier panel using a high-temperature adhesive. The reusable TPS concepts were direct-bonded onto the primary structure (modeled as 0.10 inches Aluminum 2024) using Room Temperature Vulcanized (RTV) adhesive for the blanket concepts, while the ceramic tiles used a Nomex Strain Isolation Pad (SIP) with a nominal thickness of 0.090 inches and two RTV transfer coats.

The TPS split-line definition was generated using the non-conducting wall temperatures, multi-use allowable temperature for the nominal trajectory, and single-use for the abort trajectories. For the LEO return CEV design, the heat shield material selected was the Shuttle-derived high-density ceramic tile (LI–2200), while the aft-body TPS material consisted of existing Shuttle ceramic tile (Reaction-Cured Glass (RCG-) coated LI–900) and flexible blanket systems (AFRSI and FRSI). Temperature limits for these materials are presented in **Table 5-14**.

TPS Choices	Material	Thickness (cm)
FRSI	FRSI	Varying
	DC92	0.0127
AFRSI	EGLASS	0.0279
	QFELT_mquartz	Varying
	ASTRO_quartz	0.0686
	GrayC9	0.0419
LI–900	Nomex SIP	0.2286
	RTV	0.0305
	LI900	Varying
	RCG	0.0305
LI–2200	Nomex SIP	0.2286
	RTV	0.0305
	LI2200	Varying
	TUFT12	0.2540
Silicon Infused Reusable Ceramic Ablator (SIRCA)	Nomex SIP	0.2286
	RTV	0.0305
	SIRCA-15F_V	Varying
	SIRCA-15F_C	0.2540
SIRCA calculated in Fiat	SIRCA-15	Varying
SLA	SLA-561V	Varying
Avcoat	Avcoat	Varying
PICA	PICA-15	Varying
Carbon Phenolic	Carbon Phenolic	Varying
Mid-Density Carbon Phenolic	Carbon Phenolic Mid-Density	Varying
Carbon Facesheet 0.6-cm	Carbon Fiber	Varying
	Carbon Facesheet	0.6000
Carbon Facesheet 1-cm	Carbon Fiber	Varying
	Carbon Facesheet	1.0000
Carbon Phenolic	Carbon Phenolic	Varying

Table 5-12. TPS Materials

Table 5-13. Summary of TPS Material Options and Their Characteristics

	Carbon Phenolic	AVCOAT	PICA	C–C facesheet/Carbon fiberform	Mid-density C–P
Characteristics	Heritage tape-wrapped composite developed by USAF used on BRVs, and as heat shields on Pioneer Venus and Galileo probes (many fabricators)	Filled epoxy novolac in fiberglas-phenolic honey comb used as Apollo TPS (developed by Avco; now Textron)	Phenolic impregnated carbon fiberform used as heat shield on Stardust (developed by Ames, fabricated by FMI)	Carbon-carbon facesheet co-bonded to carbon fiberform insulator used as heat shield on Genesis (developed by LMA, fabricated by CCAT)	Notional developmental material to span the density range 480–960 kg/m^3 by densifying PICA or making low-density carbon phenolic (ongoing development at Ames)
Density, kg/m^3 (lbm/ft^3)	1,441.66 (90)	529 (32)	236 (15)	1,890/180 (118/11)	480.55 (30)
Aerothermal performance limit and failure mode	20,000 W/cm^2 and 7 atm; char spall	700 W/cm^2 and 1 atm; char spall	2,000 W/cm^2 and 0.75 atm; char spall	5,000 W/cm^2 and 5 atm (postulated); strain failure at C–C/insulator interface	5,000 W/cm^2 and 1 atm (postulated); char spall
Attachment to substructure	Fabricated and cured on mandrel; secondary bonding	Honeycomb bonded to structure; cells individually filled with caulking gun	Tile bonded to structure (fabricated as one-piece for Stardust)	Tile bonded to structure (fabricated as one-piece for Genesis)	Multiple options dependent on material architecture. Most likely tiles bonded to structure
Manufacturability and scalability (to 5.5 m)	Not possible to tape-wrap a quality composite with suitable shingle angle at that scale	Pot life of composite may preclude filling all cells and curing on aeroshell of this size	Can be fabricated as tiles, but not demonstrated	Can be fabricated as tiles, but not demonstrated	Most likely to be fabricated as tiles, but not demonstrated
Current availability	Heritage material no longer available; USAF developing new generation using foreign precursor	Not made in 20 years. Textron claims they can resurrect	FMI protoype production	Currently available (CCAT for LMA)	Scalability not an issue if fabricated as tiles
Human-rating status	Space-qualified for uncrewed misssions	Human-rated in 1960s	Space-qualified for uncrewed missions (not tiles)	Space-qualified for uncrewed missions (not tiles)	Developmental
Test facility requirements (include radiation and convective heating)	High-density all-carbon system will be opaque to radiant heating over broad spectrum (Galileo experience)	Opacity over radiative spectrum needs to be evaluated but no facility available	Opacity over radiative spectrum needs to be evaluated but no facility available. Opacity at UV wavelengths demonstrated (lamp tests)	High-density all-carbon system will be opaque to radiant heating over broad spectrum	Mid-density all-carbon system will be opaque to radiant heating over broad spectrum
Test set requirements (experience with range of test conditions/sample sizes)	Strong experience so number of required tests would be relatively low	Extensive ground tests in 1960s, augmented by flight tests and lunar return missions. Radiative heating rates for CEV will be higher	Qualified to 1,600 W/cm^2 and 0.65 atm for Stardust. Issues with tile fabrication/gap fillers has not been evaluated	Qualified to 700 W/cm^2 and 0.75 atm for Genesis. Issues with tile fabrication/gap fillers has not been evaluated	Very limited test data on developmental materials. Issues with tile fabrication/gap fillers has not been evaluated
Radiation (CGR) Protection characteristics	Limited data. Some promise	Unknown	Unknown	Unknown but not expected to be of value	Unknown, but could provide some protection
Material response model status	High fidelity model for heritage material (which is no longer available)	High fidelity model developed under Apollo. Currently, only Ames can utilize	High fidelity model developed for Stardust	Material modeling is straightforward; uniformity of 2-layer contact unknown	Developmental materials; no model currently available but could be scaled from existing models
Micrometeroid/Orbital Debris (MMOD) impact tolerance	Denser materials more robust, glass more forgiving than carbon				
Landing shock tolerance	Heat shield is ejected, so landing shock not important for forebody				
Salt water tolerance (water landing)	Any of these materials would need to be dried out after water landing (or replaced). Denser will absorb less moisture.				

Table 5-13. (Continued) Summary of TPS Material Options and Their Characteristics

	SLA–561	SRAM20	PhenCarb-20	PhenCarb-32	LI–2200
Characteristics	Filled silicone in fiberglas-phenolic honeycomb used as heat shield on Mars Viking, Pathfinder and MER landers. Developed and fabricated by LMA	Filled silicone fabricated by Strip Collar Bonding Approach (SCBA) or large cell honeycomb. Developed and fabricated by ARA	Filled phenolic fabricated by SCBA or large cell honeycomb. Developed and fabricated by ARA	Filled phenolic fabricated by SCBA or large cell honeycomb. Developed and fabricated by ARA	Glass-based tile developed by LMA and used as windside TPS on Shuttle. Several fabricators
Density, kg/m^3 (lbm/ft^3)	256 (16)	320 (20)	320 (20)	512 (32)	352 (22)
Aerothermal performance limit and failure mode	300 W/cm^2 and 1 atm (postulated); char spall	400 W/cm^2 and 0.5 atm (postulated); char spall	800 W/cm^2 and 0.75 atm (postulated); char spall	2,000 W/cm^2 and 1 atm (postulated); char spall	Shuttle-certified to 60 W/cm^2 and 1 atm; glass melt, flow and vaporization at higher heat fluxes
Attachment to substructure	Honeycomb bonded to structure; cells filled by pushing compound into honeycomb	SCBA uses secondary bonding. Compound pushed into cells in honeycomb approach	SCBA uses secondary bonding. Compound pushed into cells in honeycomb approach	SCBA uses secondary bonding. Compound pushed into cells in honeycomb approach	Tile bonded to SIP which is bonded to structure (Shuttle Technology)
Manufacturability and scalability (to 5.5 m)	Pot life of composite may preclude filling all cells and curing on aeroshell of this size	SCBA approach with secondary bonding should scale, but not demonstrated	SCBA approach with secondary bonding should scale, but not demonstrated	SCBA approach with secondary bonding should scale, but not demonstrated	Should scale easily
Current availability	In production (LMA)	Prototype production in small sizes	Prototype production in small sizes	Prototype production in small sizes	Stockpiles of billets at KSC. Manufacturing can be restarted if necessary
Human-rating status	Space-qualified for uncrewed missions (not tiles)	Developmental	Developmental	Developmental	Human-rated for Shuttle
Test facility requirements include radiation and convective heating)	Opacity over radiative spectrum needs to be evaluated but no facility available. Opacity at UV wavelengths demonstrated (lamp tests)	Opacity over radiative spectrum needs to be evaluated but no facility available. Opacity at UV wavelengths demonstrated (lamp tests)	Opacity over radiative spectrum needs to be evaluated but no facility available. Opacity at UV wavelengths demonstrated (lamp tests)	Opacity over radiative spectrum needs to be evaluated but no facility available. Opacity at UV wavelengths demonstrated (lamp tests)	Radiative heating not an issue for Block 1 applications
Test set requirements (experience with range of test conditions/sample sizes)	Qualified to 105 W/cm^2 and 0.25 atm for Pathfinder. Currently being tested to 300 W/cm for MSL	Developmental material currently being tested to 300 W/cm^2 for MSL	Developmental material has been tested to 700 W/cm under ISP	Developmental material has been tested to 800 W/cm under ISP	Gaps and gap fillers need to be tested at higher heat fluxes for CEV Block 1 application
Radiation (CGR) Protection characteristics	Unknown	Unknown, but could provide some protection	Unknown, but could provide some protection	Unknown, but could provide some protection	Unknown
Material response model status	Existing model very limited and not high fidelity. High fidelity model will be developed under ISP	Existing ARA model empirical and limited. High fidelity model will be developed under ISP	Existing ARA model empirical and limited.	Existing ARA model empirical and limited.	High fidelity model for Shuttle regime. Needs to be extended to higher heat fluxes where material may become ablator
MMOD impact tolerance	Denser materials more robust, glass more forgiving than carbon				
Landing shock tolerance	Heat shield is ejected, so landing shock not important for forebody				
Salt water tolerance (water landing)	Any of these materials would need to be dried out after water landing (or replaced). Denser will absorb less moisture.				

Table 5-14. Shuttle TPS Allowable Temperature Limits

TPS Material	Multi-Use Temperature (Kelvin, °F)	Single-Use Temperature (Kelvin, °F)
LI-2200 Ceramic tile	2,000, 3,140	2,000, 3,140
LI-900 Ceramic Tile	1,495, 2,230	1,756, 2,700
AFRSI	922, 1,200	1,256, 1,800
FRSI	672, 750	728, 850

Several candidate ablative materials were investigated for the lunar-return design heat shield, as presented in **Table 5-12**. On the aft-body, Shuttle-derived reusable TPS materials (LI–2200, LI–900, AFRSI and FRSI) were used for regions where the surface temperatures were within the allowable temperature range for a given material.

Initial Conditions
Initial in-depth temperature distribution was assumed to be 70°F (294.26 Kelvin) for both Earth orbit reentry and lunar return entry.

Internal Boundary Conditions
An adiabatic backwall condition was assumed for both the composite aero-shell and primary structure.

Heat Transfer Analysis and TPS Sizing
The TPS sizing analysis was conducted using a transient 1–D "Plug" model. The required TPS insulation thicknesses were computed by a TPS Sizer using the Systems Improved Numerical Differencing Analyzer (SINDA)/Fluid Integrator (FLUINT) software solver for the reusable concepts and the FIAT software code for the ablative TPS materials. For the aft portion of the capsule, a full soak-out condition was imposed for TPS insulation sizing. Because the heat shield for all capsule configurations was assumed to be ejected before landing, a non-soak-out condition (i.e., the heat transfer analysis was stopped at the end of the flight trajectory) was used for the heat shield TPS sizing. For all TPS materials, the required thickness was computed to limit the composite carrier panel and the primary aluminum structure to 350ºF (450 Kelvin).

TPS Analysis
An extensive set of analyses were performed to analyze and size the TPS for ISS, lunar, and Mars mission entry trajectories. A number of trade studies were also conducted. These results are summarized in **Appendix 5C**, **CFD Tools and Processes**.

5.3.1.3.8 Baseline Capsule "Passive" Stability Analysis

A number of analyses was carried out on the initial baseline capsule shape to assess the benefits of monostability versus bistability and the effects of the degree of monostability on a "passive," ballistic entry. The baseline shape on which these analyses were performed is depicted in **Figures 5-29** and **5-33**.

Several arbitrary CG locations were selected (**Table 5-15**), resulting in different pitching moment curves (**Figure 5-35**). Of the six CG locations, five showed different degrees of monostability and one resulted in a bistable vehicle. CG1 is the most monostable and CG5 the least monostable. CG6 represents a bistable configuration. In order to quantify the degree of monostability, each of the CG locations was associated with a parameter—hereafter referred to as "monostability percentage"—that represented the area under the absolute value of its corresponding pitching moment curve as a percentage of that of the Soyuz. Using this method, the range of initial conditions (away from nominal) that each configuration could be able to withstand without any load or heat rate violations could be represented in terms of this monostability percentage.

Two kinds of tests were run for each CG location and the three scenarios described previously (entry from LEO, ascent abort, and lunar return). These tests are described below.

- With zero initial angular rates and zero initial beta, the initial attitude is varied on alpha only from –180 to +180 deg.
- With initial attitude being the trim attitude, the initial pitch rate is varied from –5 to +5 deg/s. Yaw and roll rates are initialized to zero.

The heat rate and crew limits criteria remained the same as those described in the previous trade analysis.

Table 5-15. Selected CEV CG Locations for Passive Stability Evaluation

	Soyuz Monostable	CEV CG1 Monostable	CEV CG2 Monostable	CEV CG3 Monostable	CEV CG4 Monostable	CEV CG5 Monostable	CEV CG6 Monostable
X_{cg}/D	0.375	0.216	0.241	0.268	0.290	0.315	0.340
Z_{cg}/D	–0.0305	–0.0475	–0.0455	–0.0433	–0.0416	–0.0396	–0.0376
Percent Monostability	100	150	125	100	87	77	11

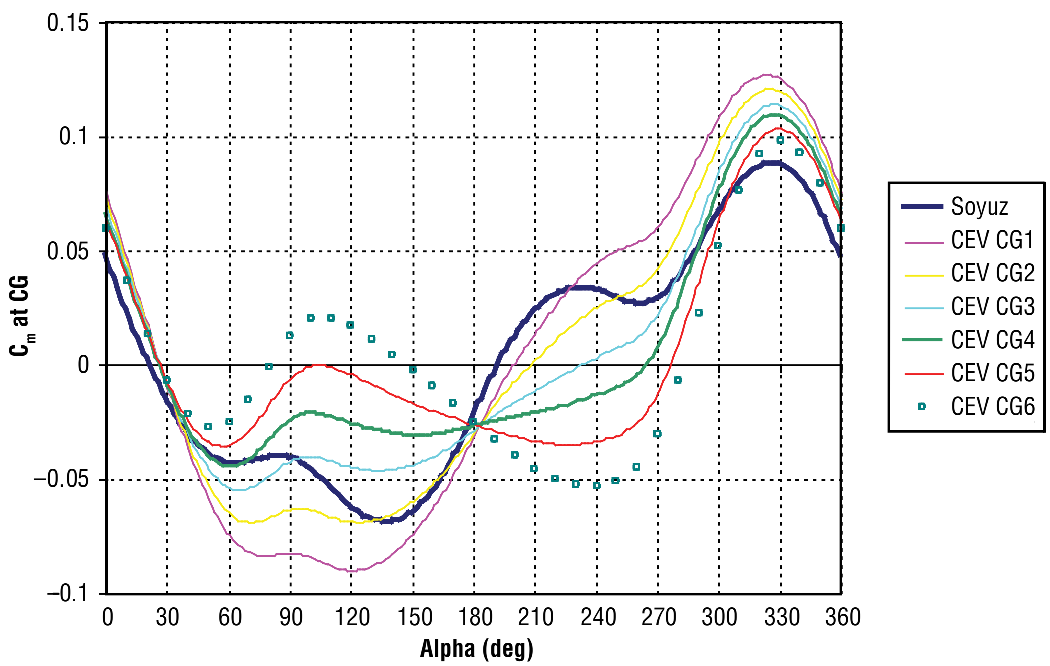


Figure 5-35. Pitching Moment Curves Associated With Each CEV CG Location Used in the Passive Stability Evaluation

The valid ranges for both types of tests for a LEO return are presented in **Figure 5-36**.

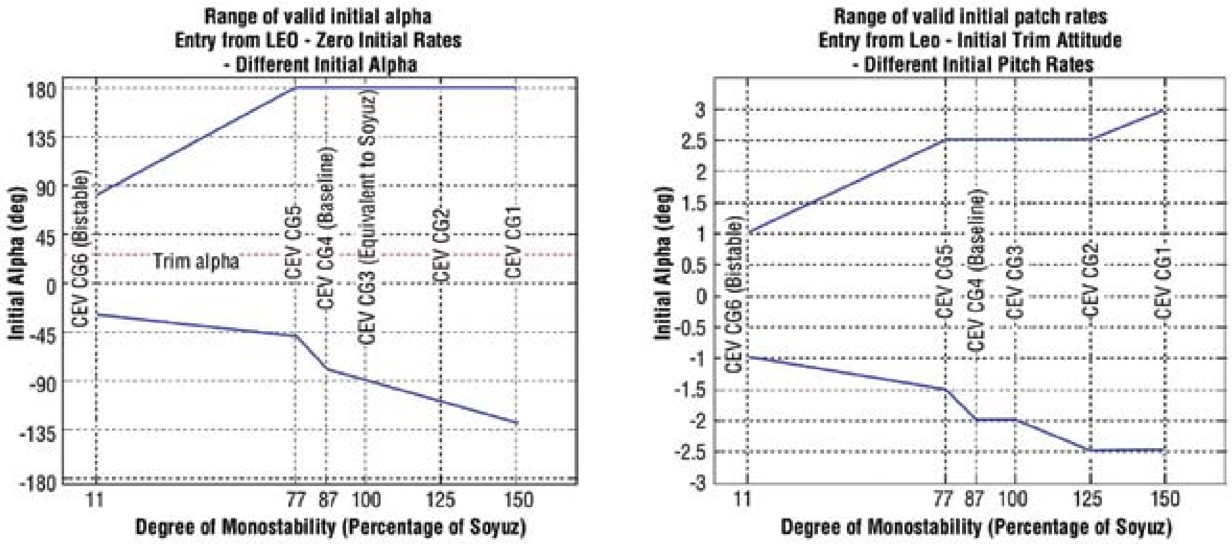


Figure 5-36. Valid Initial Attitudes and Pitch Rates in Entry from LEO versus Different Degrees of Monostability

The valid ranges for Test type 1, for an ascent abort at a trajectory point that produces the worst case heat rates, are presented in **Figure 5-37**. The results for Test type 2 are not easily quantifiable and, therefore, are inconclusive at this point.

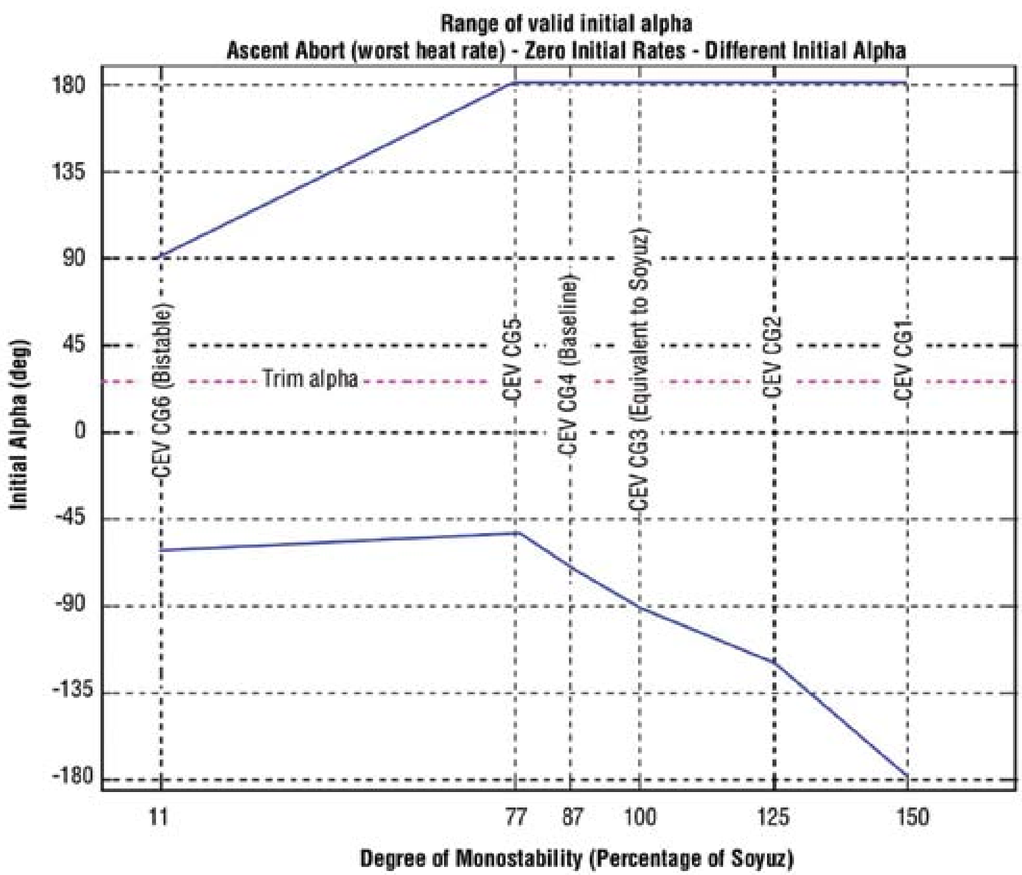


Figure 5-37. Valid Initial Attitudes in Entry from Ascent Abort (Worst Heat Rate) versus Different Degrees of Monostability

In the lunar return case, the L/D characteristics are enough to result in a high number of skip cases for all CGs tested. Therefore, in order to be able to quantify the impact of the degree of monostability in the range of initial conditions that the vehicle could passively recover from, two options were studied. These options were:

- The Z component of the CG location was set to zero. By doing this, the spacecraft was transformed into a ballistic vehicle, permitting the suppression of all the skip cases. The resulting CG locations and monostability percentages are presented in **Table 5-16**. The associated pitching moment curves are depicted in **Figure 5-38**. It can be seen in **Figure 5-38** that the CEV with CG5 becomes a bistable vehicle; therefore, CEV CG6 has been removed from the analysis. The valid ranges of off-nominal initial conditions when Z_{cg} is set to zero are presented in **Figure 5-39**.
- The induction of a spin rate to null the effect of lift, allowed the spacecraft to become close to a ballistic vehicle. A tentative spin rate of 35 deg was imparted before heat rate buildup. In this case, the CG locations are still those of **Table 5-16**. This technique is more realistic in terms of the manner in which a ballistic entry trajectory would actually be achieved. The valid ranges of off-nominal initial conditions when the vehicle is spun up are presented in **Figure 5-40**.

	Soyuz Monostable	CEV CG1 Monostable	CEV CG2 Monostable	CEV CG3 Monostable	CEV CG4 Monostable	CEV CG5 Monostable
X_{cg}/D	0.375	0.216	0.241	0.268	0.290	0.315
Z_{cg}/D	0.0	0.0	0.0	0.0	0.0	0.0
Percent Monostability	100	146	118	87	62	35

Table 5-16. Resulting CEV CG Locations for Ballistic Lunar Return Passive Stability Evaluation

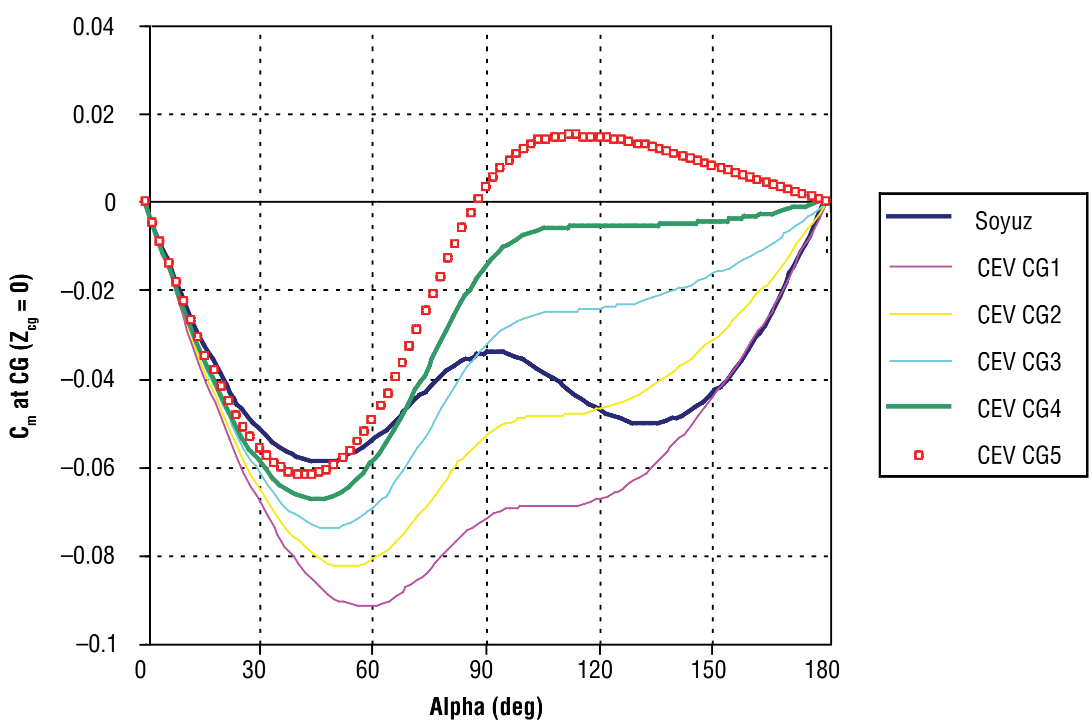


Figure 5-38. Pitching Moment Curves Associated With Each CEV CG Location Used in the Ballistic Lunar Return Passive Stability Evaluation

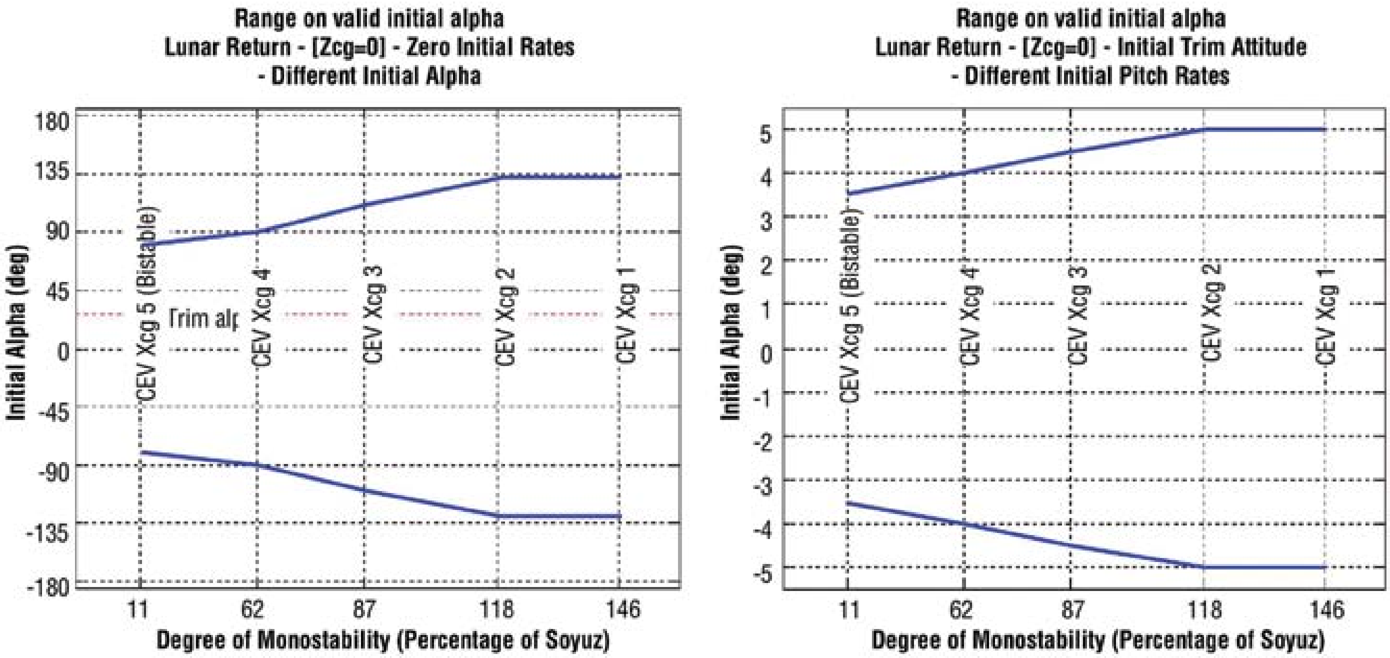


Figure 5-39. Valid Initial Attitudes and Pitch Rates in Ballistic (Z_{cg} = 0) Entry from Lunar Return versus Different Degrees of Monostability

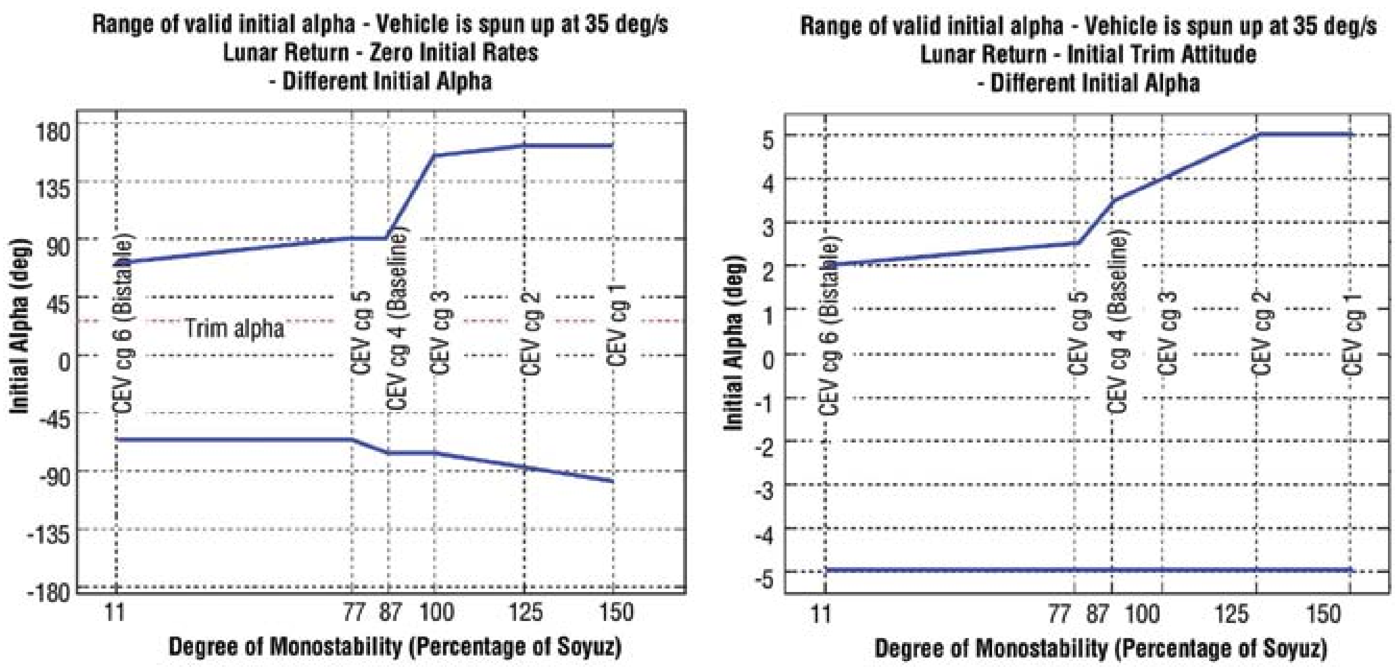


Figure 5-40. Valid Initial Attitudes and Pitch Rates Entry from Lunar Return with Spin Up (35 deg/s) versus Different Degrees of Monostability

5.3.1.3.9 Initial Baseline Capsule Analysis Summary

Detailed CFD investigations of the aerodynamics and aerothermodynamics of the initial baseline capsule validated the initial design results. The trim angle-of-attack for 0.4 L/D was determined to be 28 deg. The vehicle was monostable with up to 49 percent of the volume below the CG. For margin, a desired CG level was established at 45 percent volume, or an X/D location of 0.29. At this location, a Zcg offset of 0.53D would be required, which was approximately the same as required by an Apollo for 0.4 L/D. The vehicle had greater static stability at the desired trim angle than Apollo, and less sensitivity of L/D to a Zcg dispersion than Apollo. Sidewall heating was somewhat influenced by the direct impingement of flow, but only a very small portion of the windward aft-body (near the leading edge corner) would require ablative TPS for the 11 km/sec lunar return velocity. However, a fair amount of LI–2200 was required on the aft-body.

The 6–DOF analysis of the passive, ballistic entry capabilities of the vehicle showed it could handle approximately –90 deg to +180 deg in initial pitch attitude or up to +/–2 deg/sec of initial pitch rate for a LEO entry or ascent abort. For a lunar return, analysis showed that approximately +/– 90 deg initial attitude or +/– 4 deg/sec initial pitch rate could be handled. Even more capability existed if the X_{cg} could be placed lower than the 45 percent volume level.

Some of the attributes of the initial baseline capsule are shown in **Table 5-17**, compared to the actual Apollo with 0.3 L/D, an Apollo with 0.4 L/D, and an AFE shape—all scaled up to the 5.5-m diameter CEV size.

Table 5-17. Comparison of Actual Apollo, Initial Baseline CEV Capsule, and Preliminary AFE-type CEV Parameters




	Apollo (based on flt aero) Actual – 0.3 L/D	**Apollo** (based on flt aero) 0.4 L/D	**Axisym. CEV** (based on CFD) 0.4 L/D	**AFE CEV** (based on CFD) 0.4 L/D
Base radius/D	1.18	1.18	1.18	Original AFE
Corner Radius/D	0.05	0.05	0.05	Original AFE
Cone angle	32.5 deg	32.5 deg	20 deg	20 deg
Height/D (to docking adapter)	0.75 (4.1 m)	0.75 (4.1 m)	0.8 (4.4 m)	0.8 (4.4 m)
α	20 deg	27 deg	28 deg	25 deg
OML Volume	44.3 m_3	44.3 m_3	63.7 m_3	~64 m_3
Xcg/D	0.265 (< 0.22 for monostab.)	0.265 (< 0.23 for monostab.)	0.29 (<0.31 for monostab.)	0.29
Zcg/D	0.038 (> 0.04 for monostab.)	0.05 (> 0.052 for monostab.)	0.053 (> 0.051 for monostab.)	0.032
% Vol below Xcg	55% (< 39% for monostab.)	55% (< 42% for monostab.)	45% (< 49% for monostab.)	~45%
Monostable?	No	No	Yes	Yes
C_m-alpha @ cg	-0.0023	-0.0025	-0.0028	
ΔL/D per ΔZcg	0.022/cm	0.018/cm	0.016/cm	

Note: All are scaled to a 5.5-m diameter.

5.3.1.3.10 Alternative AFE-Type Capsule Shape

The proposed baseline design was disseminated to the systems engineering and aerothermal groups for packaging and TPS estimation, respectively. It became readily apparent that this design could be difficult to package and acquire the desired CG. The primary difficulty rested in attempting to reach the Z_{cg} location. The CG was pushed far off the centerline in order to acquire the desired 0.4 L/D ratio. Thus, in order to keep the general aerodynamics and shape of the baseline vehicle, a slightly modified heat shield, known as the AFE-type, was proposed. The AFE-type shape is intended to bring about two big changes in the aerodynamics. First, it brings the CG closer to the centerline of the vehicle, and secondly, it makes the trim angle-of-attack lower.

The AFE-type shape originated with the AFE of the late 1980s and early 1990s. Although never flown, it offered some advantages over a symmetric blunt-body, particularly in required Z_{cg} offset. The shape was defined by the seven parameters listed below, with the original values shown in parentheses (**Figure 5-41**):

- Cone angle (60 deg),
- Rake angle (73 deg),
- Shoulder turning angle (60 deg),
- Shoulder radius (0.3861 m),
- Nose radius (3.861 m),
- Nose eccentricity, and
- Diameter (3.861 m).

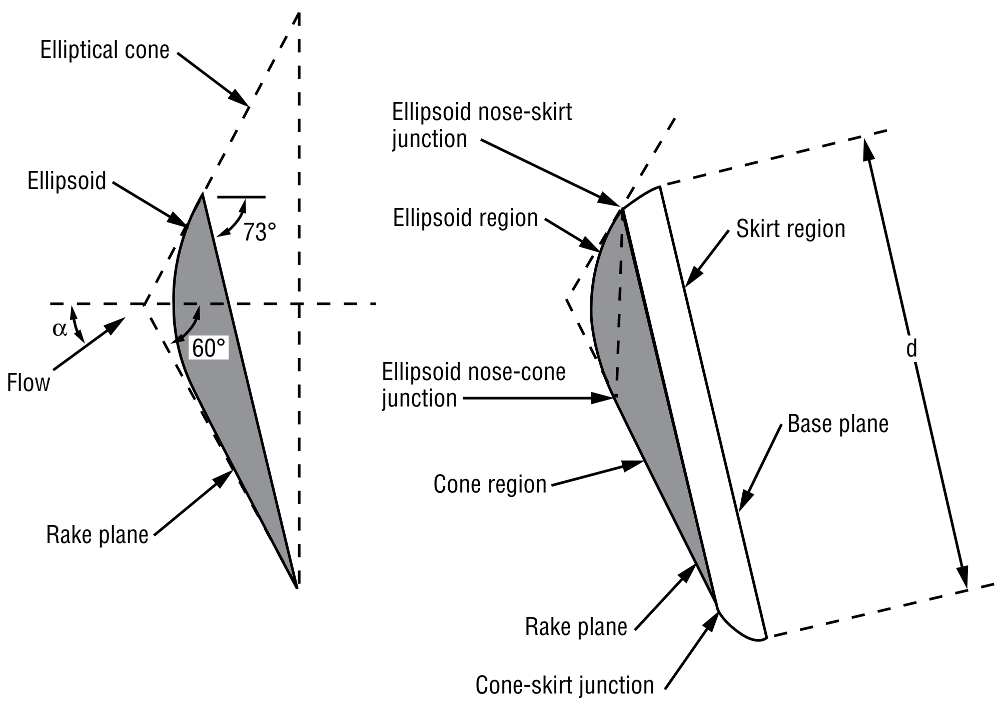


Figure 5-41. AFE-type Shape Parameters

Thus, the AFE-type shape is a well-defined geometry. The design is basically a raked (i.e., cut off at an angle) elliptical cone with a blunted nose (i.e., can be either spherical or elliptic). The rake angle stipulates the position of the blunted nose. If the angle-of-attack is equal to the complement of the rake angle, the velocity vector is aligned with the nose of the heat shield. If the angle-of-attack is smaller than the sidewall angle, the flow will not impinge on the aft cone. The pitch plane elliptical cone angle (for the AFE heat shield) basically determines the thickness of the heat shield. As the difference between the rake and the cone angle increases, the thickness will also increase. Both of these parameters together affect the vehicle aerodynamics. A preliminary analysis of the strengths and weaknesses of employing this shape for the CEV is provided in **Appendix 5C**, **CFD Tools and Processes**, and the geometry tool created by the NASA Ames Research Center (ARC) for AFE-type vehicle model generation is described in **Appendix 5D**, **ARC Geometry Tool for Raked Cone Model**.

5.3.1.3.11 Alternate Proposed CM Shapes

Near the end of the ESAS, it was decided that the direct-to-surface lunar mission architecture would not be prudent. This eliminated the need for a high-volume CEV CM such as the baseline axisymmetric CM shape. In addition, a 1.5-launch solution was selected in which the CEV CM would always be launched on a Shuttle-derived CLV configuration for both LEO and lunar missions. This LV was limited in performance, particularly for the lunar mission and lunar CEV, which created a need to decrease the baseline CEV mass. Because significant mass was created by the extremely large aft-body due to TPS, radiation shielding, and structure, it was desirable to increase the aft-body sidewall angle. In addition, the aft-body flow impingement of the baseline axisymmetric CM shape was not desirable. Finally, the systems packaging at this point had still not achieved the desired CG location for the baseline shape. Although the CG location was low enough to provide monostability, it was not offset far enough to produce the desired 0.4 L/D ratio. All of these factors weighed in against the remaining benefit of the shallow-walled, large aft-body baseline design—the potential monostability. Eventually, the desire for aerodynamic monostability was outweighed by other factors; however, other propulsive or mechanical methods are available to ensure stable ballistic entry, such as employing a flap or RCS jets.

The baseline axisymmetric shape was modified to have a 30-deg back-shell sidewall angle and reduced diameter to 5.2 m. This provided a 2- to 3-deg buffer from the flow direction at a 26–27 trim degree angle-of-attack. The alternative AFE-type vehicle with its 28-deg sidewall angle was already suitable, except for the fact that it was scaled down to a 5.2-m diameter. In addition, its length was decreased to allow for the docking ring diameter and a tighter corner radius was employed to help decrease the Z_{cg} offset requirement. Both changes to the AFE-type shape significantly decreased monostability. These vehicles are shown in **Figure 5-42**.

The C_m curves for these vehicles are shown in **Figure 5-43** at the representative CG locations and monostable limits. The C_m curves are similar, although there is a slight reduction in static stability at the desired trim angle-of-attack of the AFE-type shape compared to Apollo. **Figure 5-44** provides the 0.4 L/D CG trim lines for these configurations. (Note: the significantly reduced Z_{cg} offset requirements of the AFE-type shape.) Both trim lines have roughly equal distance from a representative CG to the monostable CG limit. **Table 5-18** presents some performance specifications for the two vehicles. The overwhelming benefit of the AFE-type configuration is the reduced Z_{cg} offset required for 0.4 L/D, though there is a slight TPS mass cost.

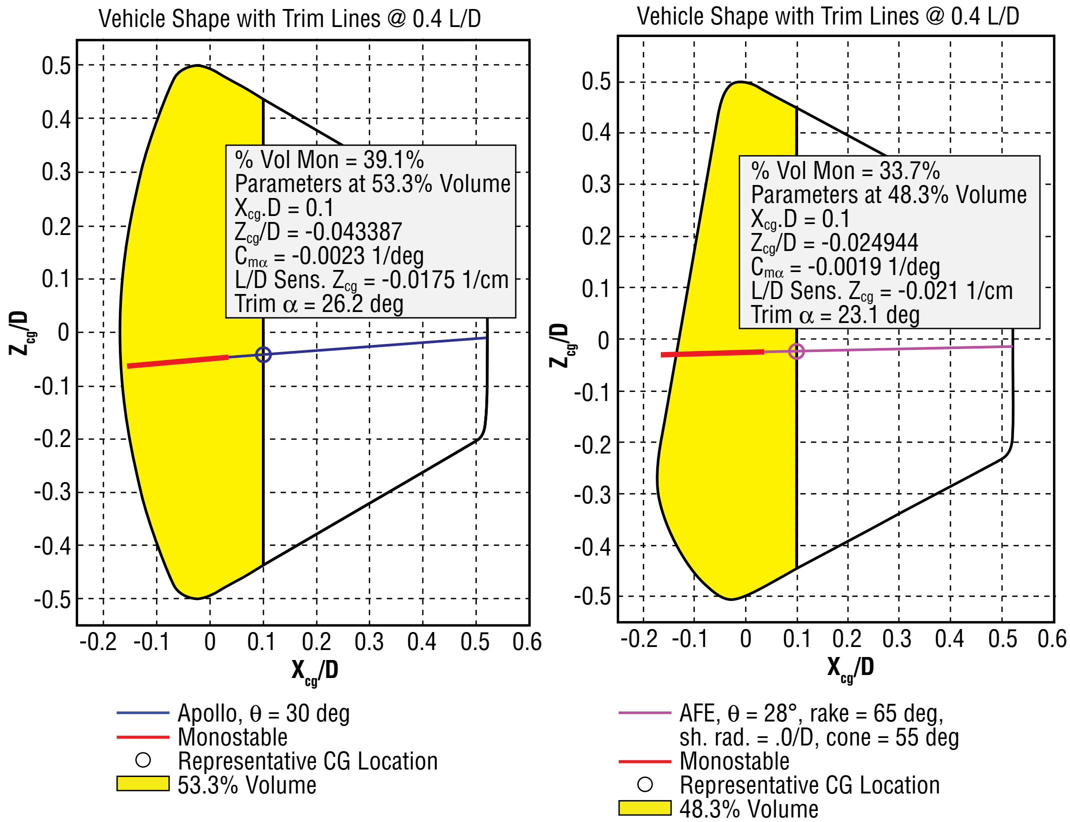


Figure 5-42. Alternate Symmetric and AFE Heat Shield Vehicles

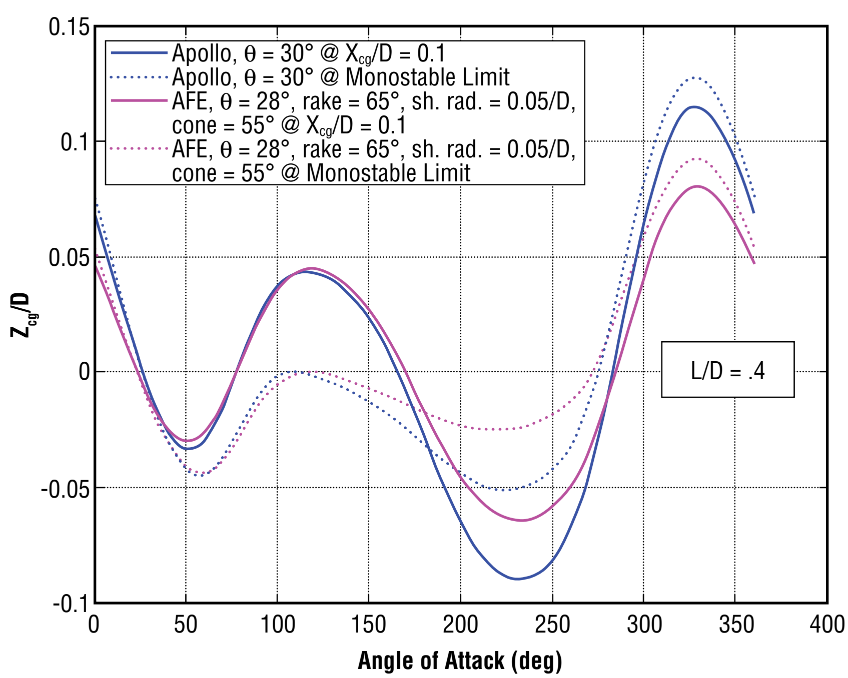


Figure 5-43. C_m Curves for Alternate Vehicles

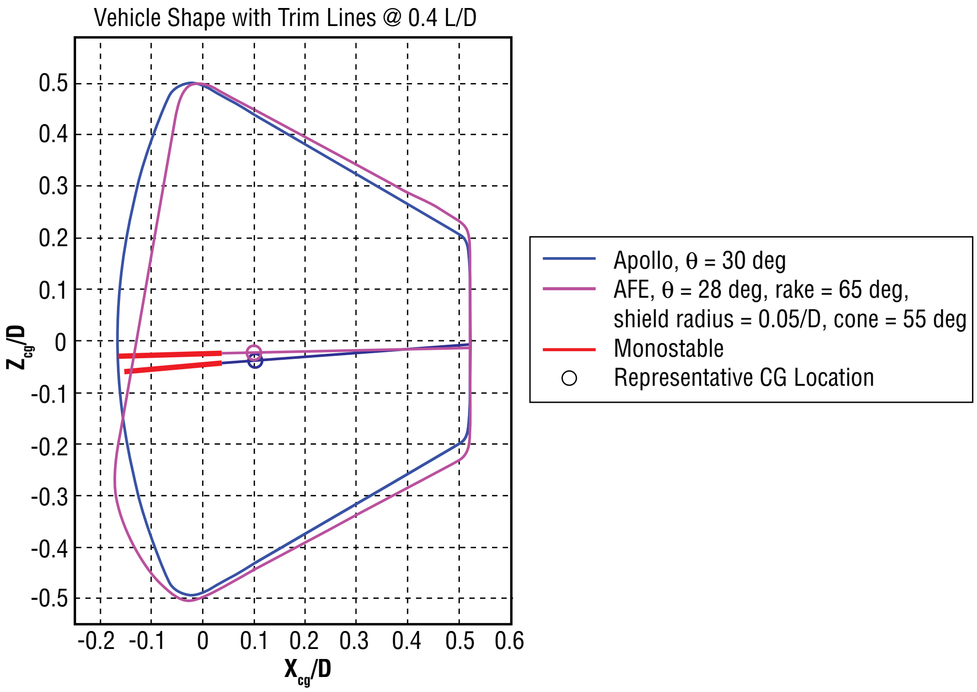
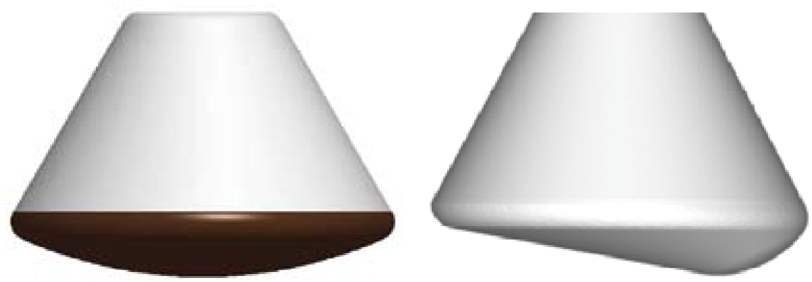


Figure 5-44. Trim CG Lines for Alternate Vehicles

Table 5-18. Comparison of the Alternate Vehicles



Shape Specifics	5.2-m diameter/30-deg sidewall angle/ Apollo heat shield	5.2-m diameter/28-deg sidewall angle/ AFE heat shield
Angle of Attack for 0.4 L/D	27-deg (CFD)	23-deg (CFD) (more margin from afterbody flow Impingment even with larger aftervody)
Zcg offset for 0.4 L/D with Xcg/D=0.1 (0.52 m)*	23 cm (Newtonian**)	13 cm (Newtonian**) (43% decrease)
Heat shield TPS mass *** (non-conservative estimate)	630 kg (5.5 m dia)	690 kg (5.5 m dia) (9% more TPS mass for heat shield)
Monostability Trending	Xcg limit for monostability @ 0.19 m*	Xcg limit for monostability @ 0.19 m* (same distance to a monostable condition)
Sensitivity to Zcg (ΔL/D per ΔZcg)	0.018/cm	0.021/cm (slightly more sensitive to Zcg)
Development complexity	Lower	Higher (20–25% more aero/aerothermo effot)

** Measured from intersection of heat of heat shield and aftbody cone*

***CFD typically increases the Z_{cg} offset requirement by about 0.01D*

****Designed to handle both ballistic direct entry and skip entry*

5.3.1.3.12 Final ESAS CM Shape

Based primarily on packaging and mass issues, the final proposed baseline CEV CM shape was a 5.5-m diameter Apollo (with the original Apollo 32.5-deg sidewall). Thus, the aerodynamics and aerothermodynamics are well known. TPS estimates were made based on the results presented previously using the heat shield data for the axisymmetric baseline shape and the back-shell data for the AFE-type shape. The trimline for this shape was found to be nearly identical to that shown previously for the 30-deg sidewall Apollo. Also, the ballistic entry analyses provided above is still applicable for the most part.

Concern is warranted, however, over the ability to achieve the Z_{cg} offset that will be required to achieve a 0.4 L/D using this shape. However, the alternative AFE-type shape as shown previously would alleviate this concern. The shape working group is continuing to evolve an AFE-type shape that is directly comparable to the proposed 5.5-m diameter Apollo with a 32.5-deg back-shell, with the only difference being in the heat shield shape. Further risk and performance analyses in the areas of landing (land versus water) may ultimately determine which CEV CM shape is selected.

5.3.2 CM Net Habitable Volume Trades

In the history of human spacecraft design, the volume allocated for crew operations and habitability has typically been the remaining excess after all of the LV constraints and vehicle design, weight, CG, and systems requirements were met. As a result, crew operability has often been compromised as crew sizes are increased, mission needs changed, and new program requirements implemented. CM habitability considerations have often been relegated to a second level behind engineering convenience (e.g., putting the galley next to or collocated with the hygiene facility to simplify plumbing). Whereas flight crews have demonstrated a consistent and, at times, heroic resilience and adaptability on orbit, designs of future crew habitable modules should not sacrifice crew operability. NASA should design new vehicles that allow the crew to safely and efficiently execute the mission, not build vehicles that execute a mission which happens to carry crew.

Net habitable volume is defined for this study as the pressurized volume left available to the crew after accounting for the Loss of Volume (LOV) due to deployed equipment, stowage, trash, and any other structural inefficiency that decreases functional volume. The gravity environment corresponding to the habitable volume must also be taken into consideration. Net habitable volume is the volume the crew has at their disposal to perform all of their operations. In order to estimate the net habitable volume requirement for the CEV for each phase of flight, this study first looked at the crewed operations required in the spacecraft, what operations must be done simultaneously, how many crew members might be expected to perform each operation, how long each operation might last, how often each operation might be required during the mission, the complexity of the task, and the potential impact to the task by vehicle structure, shape, and gravity environment. The analysis took into account the entire spacecraft pressurized volume and the estimated volume and layout of internal systems equipment and stowage volumes by mission type and phase. Pressurized and net habitable volumes of previous and current spacecraft were used for comparison. Full-scale rough mockups were made for the internal volumes of both the CEV CM and LSAM to assist in the visualization and evaluation process.

The initial goal of the study was to determine the minimum net habitable volume required for the CEV for each DRM. However, without more definition of systems and structural requirements (e.g., how much volume seat stroke, plumbing, cables, and wiring will require), a specific volume number was difficult to derive. Using the mockups, the ESAS team determined a rough estimate of minimum net habitable volume. More detailed analysis may find ways to be more efficient in the design of internal systems and structure; however, requirements for systems and volumes not currently anticipated may also be added in the future, which will compromise the net habitable volume for the crew.

Full-scale high-level mockups of the CEV interior configurations being traded allowed the ESAS team to visualize the impacts of using the CEV as a single vehicle to take crew all the way to the lunar surface and as part of a set of vehicles for the lunar exploration mission where the CEV remains in LLO. The ESAS team provided the designs it felt best supported the requirements of launch, on orbit, and entry. The team also provided best available estimates of both equipment volumes and required task volume.

The number of crew, mission duration, task/operations assumptions, and volume discussions for each of the CEV DRMs are described in the following sections.

5.3.2.1 ISS Crew/Cargo Mission

The CEV will carry three to six crew members to the ISS with nominally a day of launch rendezvous, but, in the worst case, taking 3 days to get to the ISS. Returning from the ISS to Earth will nominally take 6 hours; however, in a contingency this could take a day or more. The crew will not need to exercise, will not require a functional galley, will not conduct planned EVA, will not perform science activities, but will still require privacy for hygiene functions. Consumables required for this mission will be minimal. The CEV and launch and entry suits will be capable of contingency EVA, but, for the ISS mission, it is anticipated that the vehicle would return to Earth or stay at the ISS if a contingency EVA was required. The vehicle and the launch and entry suits will support contingency cabin depressurization to vacuum. The CEV will remain docked to the ISS for a nominal period of 6 months. The CEV will support safe haven operations while docked to the ISS and provide nominal and emergency return of the crew that arrived at the ISS in the vehicle.

Since ascent and descent are the main activities in the CEV for this DRM, seats may not require stowing, and the CM interior will probably not require significant reconfiguration for on-orbit operations. The lunar DRMs will drive minimum net habitable volume for the CEV; therefore, the volume required for the ISS DRM was not examined in detail since the lunar DRM net habitable volume requirement is larger than that required for the ISS DRM.

5.3.2.2 Lunar Mission – CEV Direct to the Lunar Surface

The CEV will carry a crew of four on a 4- to 6-day Earth-to-Moon trip, with up to 7 days on the surface and 4–6 days return. All systems and equipment must function in a variety of environments and orientations (e.g., 1-g ground/pad prelaunch operations, up to 4-g ascent operations, zero-g on-orbit operations, one-sixth-g lunar surface operations, and up to 15-g worst-case Earth reentry/abort environments). The crew will need to exercise, both enroute and on the lunar surface, will require private hygiene capability and a galley, and will need to reconfigure the volume for on-orbit operations, including rendezvous and docking with other exploration elements. All crew members must be able to stand up simultaneously in the vehicle on the lunar surface. The CEV and the launch and entry suits will support contingency EVA operations. Lunar surface suits and support equipment will be carried in the CEV and must be accessible by the crew after landing on the lunar surface. An airlock is required on the lunar surface.

The critical task driving the required volume in this DRM was the volume needed for four crew to don, doff, and maintain the lunar surface EVA suits in partial gravity. The volume sensitivity to both simultaneous and serial suit donning and doffing was evaluated. Utilizing graphics analysis, direct measurement, and indirect measurement of suited operations, a rough estimate of a critical “open area” of net habitable volume of approximately 19 m^3 was derived.

5.3.2.3 Lunar Mission, CEV Left in Lunar Orbit

The CEV will carry a crew of four on a 4- to 6-day Earth-to-Moon trip and remain in orbit uncrewed while the entire crew spends time on the lunar surface in an ascent/descent module (LSAM). The CEV will rendezvous with the LSAM in LEO, and the LSAM volume will be available as living space for the crew on the way to the Moon. The on-orbit assumptions for this DRM are the same as the previous DRM. After the lunar stay, the ascent module will rendezvous with the CEV in lunar orbit and be discarded once the crew has transferred to the CEV. Only the volume in the CEV will be available to the crew for the 4- to 6-day return trip to Earth. Lunar surface suits and support equipment will be carried in the LSAM. An airlock will be required in the LSAM for lunar surface operations.

For this scenario, the donning and doffing of launch and entry suits was the major volume driver, with a minimum required critical “open area” of net habitable volume of 8 (TBR) m^3.

5.3.2.4 Mars Missions

The CEV will carry a crew of six to an MTV in Earth orbit. The time the crew spends in the CEV is expected to be less than 24 hours. The CEV will remain attached to the Mars vehicle for the transit to Mars (6 months), then remain in Mars orbit with the transit vehicle while the crew is on the Martian surface (18 months), and remain with the transit vehicle for the Earth return (6 months). The crew will reenter the CEV for the last 24 hours of the return trip to Earth. The requirements for habitability and operations for this DRM are the same as the ISS DRM.

5.3.2.5 CEV Split Versus Single Volume

A considerable amount of time was spent analyzing the advantages and disadvantages of a CEV split versus single volume. Separating the CEV volume into a CM used primarily for ascent and entry and a mission module that could be sized and outfitted for each particular mission has operational advantages depending on the mission to be supported. Also, separation of the mission module with the SM after the Earth deorbit burn provides the lightest and smallest reentry shape.

The difficulty in minimizing the ascent/entry volume of the vehicle became a driving factor because this volume must accommodate a maximum crew of six for the Mars return mission. Once the ascent/entry volume for six was determined, all other DRM crew sizes by definition will fit in this volume. A CEV sized for the six-crew DRM is the minimum size for the ascent/entry module.

The study found a single volume, which is less complex from a build-and-integrate standpoint, to be more mass-efficient and volume-efficient for a given mass. A larger single-volume vehicle also has lower entry heating and g’s as a result of a larger surface area, and thereby lower ballistic coefficient, than a smaller ascent/entry split volume. A mission module was determined to not be required for the ISS and the Mars return DRMs and was of limited value to the lunar DRM, if the single volume is large enough, and the CEV is not taken all the way to the lunar surface.

Finally, the cost and LV analyses determined that the split volume case would be higher cost (building two versus one module) and require a larger throw capability on the booster for the same net habitable volume. Based on these factors, the ESAS team decided that a single volume CEV sized for the six-crew ISS and Mars DRM would provide sufficient volume for both the four-crew lunar DRM and the three-crew ISS DRM.

5.3.3 Airlock Trades

5.3.3.1 Airlock Design Considerations

Early in the ESAS, a proposal was made by the operational community to incorporate an airlock into the CEV design. Depending on the configuration, this requirement could have significant design implications. Because the mass and volume implications of an airlock affect the size and layout of the CEV, justification of the need was addressed.

Integration of an airlock into the CEV design is complex. Non-inflatable airlocks are massive and require significant volume. Inflatable airlocks are not as heavy, but the support system requirements are the same or larger. Inflatable airlocks also bring the risk of not being able to be retracted, thus requiring jettison capability before reentry.

5.3.3.2 Zero-g Missions

The first question to be answered is whether or not the DRMs require an airlock. For missions to the ISS, the CEV docks with the station and returns to Earth. The CEV is only active for 2 to 3 days at a time during transit. Contingency EVAs are not even required for this mission. For lunar EOR/LOR missions, the CEV docks with the LSAM which then goes to the lunar surface. This mission does require contingency EVA capability that can be accomplished with a cabin depressurization. For the Mars DRM, the CEV docks with an MTV in LEO. As in the ISS missions, the CEV in this scenario is only active for 1 or 2 days. This mission does have a possible contingency EVA requirement, which could be accomplished with a cabin depressurization.

5.3.3.3 Lunar Surface Direct Mission

The only mission scenario for the CEV that could significantly benefit from an airlock is the lunar surface-direct mission, in which the CEV is taken all the way to the lunar surface. This mission would require an airlock. Without an airlock, the entire CEV would have to be depressurized, and all four crew would require Extra-vehicular Maneuvering/Mobility Units (EMUs), even if only two crew members performed an EVA. A separate airlock could be left on the lunar surface with all or portions of the EVA equipment, which would reduce the dust issue in zero-g flight. Several concepts were studied for this mission scenario, but further study would be required. The concepts studied show different arrangements for the crew during ascent/entry and for surface operations that have difficult issues to be resolved (e.g., what functionality is within the CM versus the airlock). Since the lunar surface-direct mission is no longer being considered, the requirement for a CEV airlock on the lunar surface shifts to the LSAM.

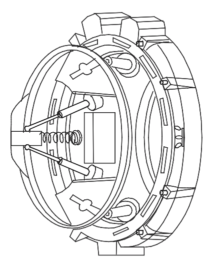
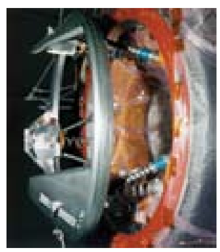
5.3.3.4 Recommendation

An airlock is not required for any of the current zero-g CEV DRMs. The ascent/entry volume is adequate for an entire mission profile, and a disposable airlock module would increase development and recurring costs.

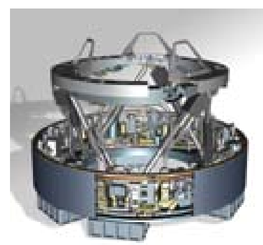
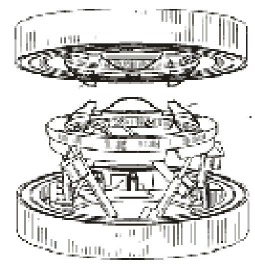
5.3.4 Docking Mechanism/ISS Docking Module Trades

As indicated in the President's Vision for Space Exploration, the completion of the ISS is a high priority for the Agency and the U.S. aerospace community. As such, CEV access to the ISS is of primary importance, and the mechanism and operations required for mating to the ISS must be factored into the CEV design and operations concept. Also, as stated in the Vision for Space Exploration, there is a need to develop systems and infrastructure that are enabling and allow for an affordable and sustainable exploration campaign. As such, it has been determined that systems developed in support of the CEV ISS missions should be compatible with other exploration missions (e.g., docking of CEV and LSAM).

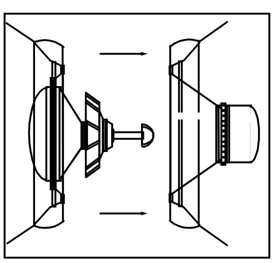
The three mating systems currently available for the U.S. Space Program are: the U.S. CBM, the Russian APAS docking mechanism, and the Russian Drogue-Probe docking mechanism. The study researched these options as they presently exist and also explored possibilities for optimizing each through adaptation and modification. The study also assessed a next-generation docking/berthing mechanism being developed at the NASA Johnson Space Center (JSC) called LIDS. The four mating concepts are depicted in **Figure 5-45**.

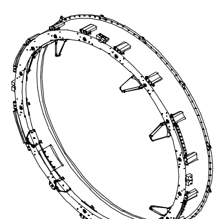
Androgynous Peripheral Docking System (APAS)
Weight: 1,250 lbs (mech/avionics/lights/hatch/ Comm/ranging sys)
Max OD: 69˝ dia
Hatch Pass Through: 31.38˝ dia
Source: JSC-26938, "Procurement Specification for the Androgynous Peripheral Docking System for the ISS Missions";
OSP ISS Port Utilization Study; Final Version, Nov. 8, 2002

New Mating System based on LIDS/ADBS [1]
Weight: est. 870 lbs (mech/avionics/hatch)
Max OD: 54˝ dia (X-38 CRV scale)
Hatch Pass Through: 32˝ dia
Source: ADBS Project
[1] LIDS/ADBS in development




Russian Probe/Cone (P/C)
Weight: 1,150 lbs (mech/avionics/lights/hatch/ Comm/ranging sys)
Max OD: 61˝ dia
Hatch Pass Through: 31.5˝ dia (approximate)
Source: Energia; OSP ISS Port Utilization Study; Final Version, Nov. 8, 2002




Passive Common Berthing Mechanism (PCBM)
Weight: 900 lbs
(mech/avionics/lights/hatch/ Comm/ranging sys/grapple fixture)
Hatch Pass Through: 54˝ square
Max OD: 86.3˝ dia
Source: SSP 41004, Part 1, "Common Berthing Mechanism to Pressurized Elements ICD" & SSP 41015, Part 1, Common Hatch & Mechanisms To Pressurized Elements ICD; OSP ISS Port Utilization Study; Final Version, Nov. 8, 2002

Figure 5-45. Docking/ Berthing Mechanisms

The two Russian docking mechanisms are complex, do not support berthing operations, and have performance limitations that create dynamically critical operations, increasing risk for missions, vehicles, and crews. With respect to their current usage on the ISS (i.e., in LEO), these limitations are manageable, and consideration of wholesale upgrade and replacement for existing vehicles and programs is not practical. However, after factoring in technical limitations, level of fault tolerance, reliance on foreign suppliers, and the requirement for application beyond the ISS and LEO, it became clear to the ESAS team that existing docking solutions were inadequate.

The ISS berthing mechanism does not support docking dynamics because it requires a robotic arm to deliver and align mating interfaces; therefore, all berthing operations would require involvement of the crew, which is incompatible with lunar applications and autonomous mating operations. Additionally, preliminary CEV architectural sizing has determined that the diameter of the CBM is too great to fit the current CEV configuration, further eliminating it for potential consideration for the CEV.

During the study, it was confirmed that all three existing systems failed to meet dual-fault tolerance requirements for critical operations and those for time-critical release, which are very important for an emergency or expedited separation. While both docking mechanisms provide nominal hook release and a pyrotechnic backup, the Space Shuttle Program accepts the use of a 96-bolt APAS release via a 4-hour EVA to satisfy dual-fault tolerance requirements. CBM-powered bolts do not operate fast enough to support expedited release because of the threaded bolt and nut design, and they are operated in groups of four to prevent binding and galling during unthreading. The CBM uses a pyrotechnic to provide one-fault tolerance for release.

Additionally, all three systems contain uniquely passive and active (male and female) interfaces that are not fully androgynous, offer limited mission mating flexibility, and each has a specific, narrow, operational range of performance for use. **Figure 5-46** depicts the dispositions of the various presented solutions and their associated issues.

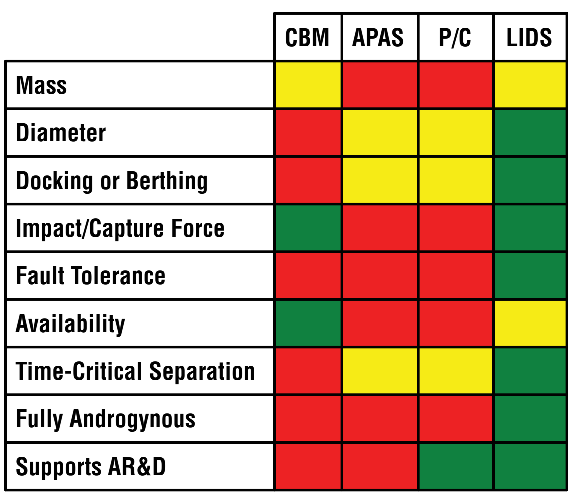


Figure 5-46. Various Solutions and Associated Issues

These facts indicate that the development of a modular, generic mating infrastructure is a key element needed for the success of CEV and other future NASA exploration missions and programs.

Many of the issues associated with existing systems have been well understood for more than a decade. Since the early 1990s, in response to mitigating these issues, the NASA Advanced Docking Berthing System (ADBS) project has been developing the LIDS as a smaller, lighter, low-impact mating system to reduce the dynamics required for and the risks associated with mating space vehicles. The ADBS project has focused on the development and testing of a low-impact mating system that incorporates lessons learned from previous and current mating systems to better meet future program requirements. As a result, it has been established that an advanced mating system built around low-impact characteristics is feasible and will help ensure meeting anticipated future mating system requirements. **Figure 5-47** depicts the LIDS mechanism in detail.

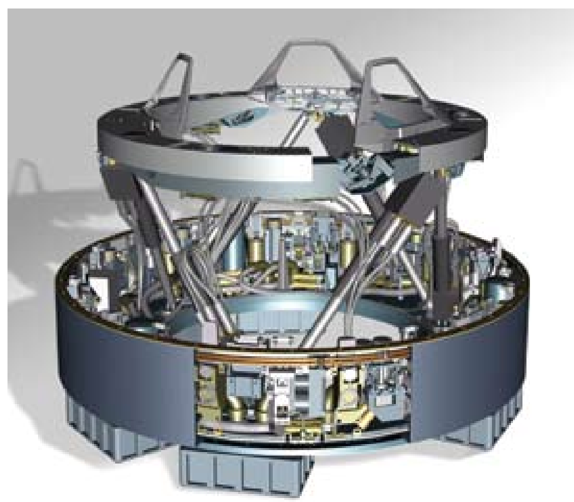


Figure 5-47. LIDS Docking/Berthing Mechanism

Through the course of this study it was also established that over the last decade, except for the LIDS development, no other U.S. activity has been occurring to develop a human-rated, crew transfer mating system. Currently, the project is funded under ESMD's Technology Maturation Program. Of primary concern was the ability of the technology to meet the accelerated CEV schedule and, in response, the ADBS/LIDS project has performed credible planning that demonstrates it can bring the TRL to the level required to support the accelerated CEV schedule. As such, it is recommended that NASA continue the LIDS development for the CEV, but use both the CEV and planned future exploration requirements to develop a mating mechanism and operations approach to form the basis of a standardized mating element that can be used as a key component in new exploration program architectures.

When developing a new mating system, an understanding of the ISS mating ports and locations becomes critical. During the assessment of existing mating options, it was established that the two existing ISS Primary Mating Adapters (PMAs) ports used as the primary and secondary docking ports for the Shuttle would be available (following Shuttle retirement) for modification or replacement and could then be used for CEV docking. However, after assessing the inability of the APAS to meet CEV and exploration requirements, it is recommended that the LIDS mechanism be incorporated onto an adapter, enabling near-term CEV/ISS use as well as supporting near-term commercial ISS cargo needs. By adapting LIDS to the ISS, this will also allow the LIDS development to proceed focused on requirements from the broader exploration activities and not just those associated with using existing ISS mating hardware.

Study trades indicate that developing a small LIDS-to-ISS adapter to configure the ISS for LIDS mating operations will allow continued accessibility through a direct-docking of the visiting vehicles and Remote Manipulator System (RMS) berthing and unberthing to easily relocate attached vehicles.

The trades have also shown that the adapter could be delivered as a new "PMA" requiring more payload bay space in a Shuttle launch or be designed as a small adapter taking up less space in a future Shuttle flight. A small adapter would also lend itself to be able to "piggy-back" on the first CEV flight should Shuttle launches or payload bay space be unavailable. RMS grappling and berthing would be required to install the adapter in this scenario. An additional scenario was evaluated using a small LIDS-to-APAS adapter to be attached to a PMA, but this requires the adapter and its delivery vehicle to deal with the force-intensive active APAS and its air-cooled avionics pallet, all of which makes this scenario less attractive than other options.

Based on the trade study, the ESAS team's recommendation for the docking mechanism is to develop the LIDS into a common interface for all applicable future exploration elements. Currently already in development at NASA/JSC, the LIDS could be completed and inserted onto the vehicle as Government-Furnished Equipment (GFE) for the early CEV-to-ISS missions. The docking adapter would subsequently be developed to convert the ISS docking points into LIDS interfaces following additional ISS port utilization trades, Shuttle launch and payload bay availability assessments, detailed design studies, and requirements definition.

5.3.5 Landing Mode/Entry Design

5.3.5.1 Summary and Introduction

The choice of a primary landing mode—water or land—was driven primarily by a desire for land landing in the CONUS for ease and minimal cost of recovery, post-landing safety, and reusability of the spacecraft. The design of the CEV CM will need to incorporate both a water- and land-landing capability to accommodate abort contingencies. Ascent aborts can undoubtedly land in water and other off-nominal conditions could lead the spacecraft to a land landing, even if not the primary intended mode. In addition, the study found that, if a vehicle is designed for a primary land-landing mode, it can more easily be altered to perform primarily water landings than the inverse situation. For these reasons, the study attempted to create a CONUS land landing design from the outset, with the intention that, if the risk or development cost became too high, a primary water lander would be a backup design approach.

5.3.5.2 Return for ISS Missions

5.3.5.2.1 Landing Site Location Analysis

A landing site location analysis was performed for the CEV conceptual design that compares the 0.35 L/D (100 nmi cross-range) and the 0.40 L/D (110 nmi cross-range) vehicles. The focus of this study was to show where acceptable landing sites can be located with respect to the SM disposal area. The SM is assumed to be unguided, and its entry state vector unaltered from that of the CEV, except for the small separation maneuver. The SM debris ellipse, which encompasses a track approximately 900 nmi long from toe-to-heel, must not infringe on land areas. This SM footprint was derived from multiple previous studies at NASA JSC, including the Assured Crew Return Vehicle (ACRV), Soyuz Crew Return Vehicle, X–38, and Orbital Space Plane (OSP) projects. It is based on detailed analyses of, and actual data from, SM-type breakups.

Three landing sites that meet the SM disposal guidelines were analyzed: Edwards Air Force Base (AFB) in California, Carson Flats in Nevada, and Moses Lake in Washington. Vandenberg AFB in California was originally considered as a prime site, but the landing area does not meet minimum size requirements (5–6 nmi diameter). Moses Lake and Carson Flats have not been surveyed as actual NASA landing areas, but have been considered in previous studies. Present satellite photos show that they meet the minimum size requirement with a high probability that they have acceptable terrain for landing. Moses Lake resides near Larson AFB (closed in 1966) and Carson Flats is located near a Naval Target Area. It is highly recommended that these sites be investigated in more detail to assess their viability.

Figure 5-48 represents the 0.35 L/D case, which shows that the SM debris limits the landing site locations to no further than 350 nmi east of the Pacific Ocean (including a 25-nmi safety margin for all U.S. coast lines). This boundary line was computed by using the entry aerodynamic flight characteristics for this vehicle design. The results show that Edwards AFB is accessible only on the ascending passes and that Carson Flats and Moses Lake were very near the safety limits, thus making them marginal for off-nominal approaches.

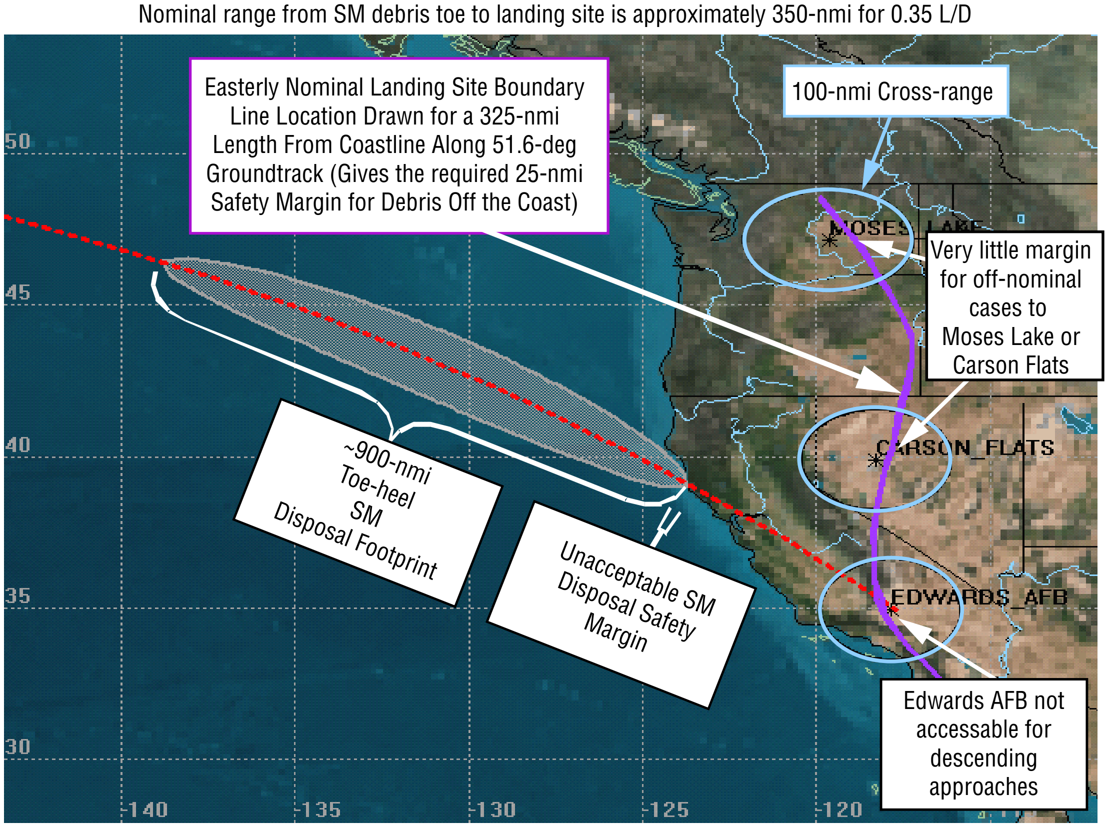


Figure 5-48. Maximum Landing Site Boundary for a 0.35 L/D CM Returning from a 51.6-deg Space Station Orbit

Figure 5-49 shows the 0.40 L/D case, which has an SM debris limit boundary line of 500 nmi (including the 25-nmi safety zone). All three landing sites are shown to have adequate accessibility on both ascending and descending passes without concern for SM debris. There is a safety margin available from the SM debris area to the coast of at least 100 nmi for all three sites. Based on this analysis, an L/D of 0.4 was determined to be desirable for the CEV CM design.

Nominal range from SM debris toe to landing site is approximately 500-nmi for 0.4 L/D

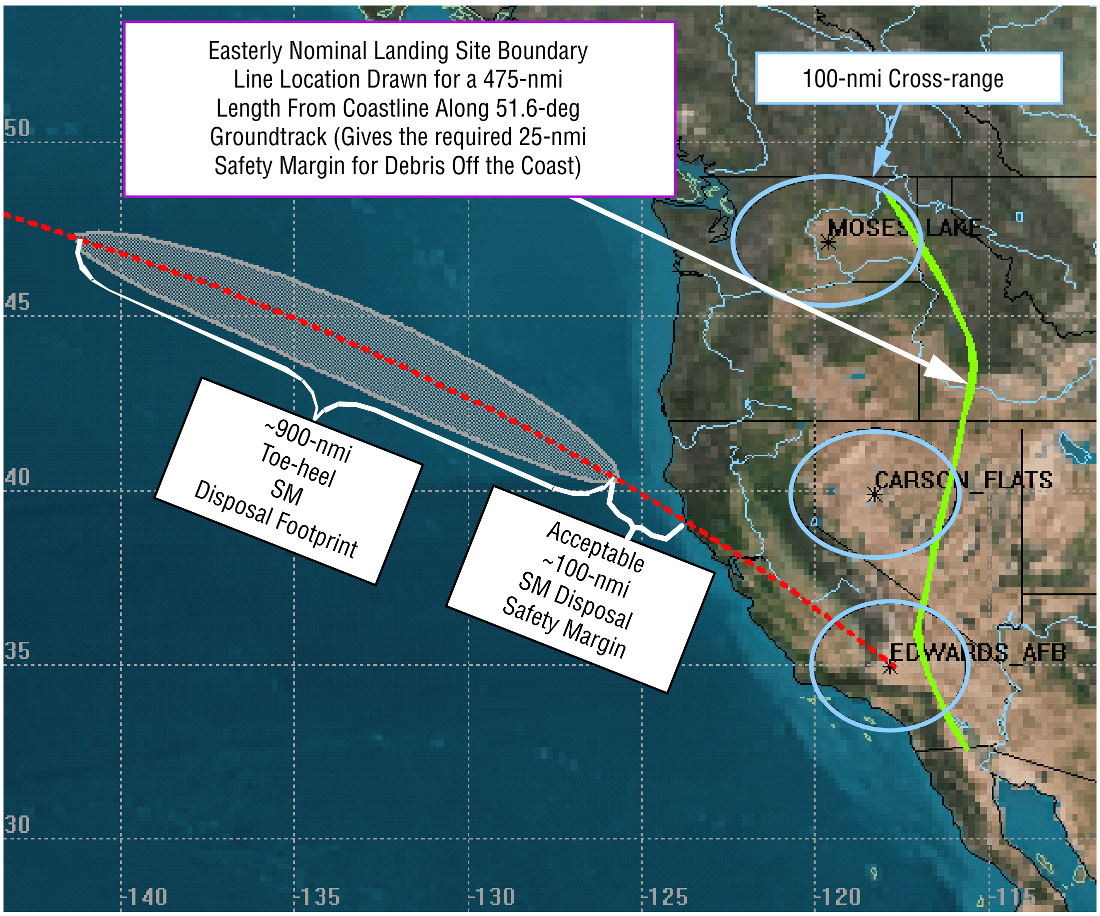


Figure 5-49. Maximum Landing Site Boundary for a 0.4 L/D CM Returning from a 51.6-deg Space Station Orbit

5.3.5.2.2 Landing Site Availability Analysis for 0.4 L/D CEV CM

The objective of this study was to find average and maximum orbital wait times for landing opportunities considering the three different CONUS landing sites located in the western U.S. for a 0.4 L/D CM. The three sites chosen were Edwards AFB, Carson Flats, and Moses Lake.

The trajectory profile used in the analysis is derived from an ISS real-time state vector with an altitude of approximately 207 nmi. This orbit is at the lower end of what is considered nominal, but is well within the operational range of many of the ISS activities.

The nominal orbital wait times, as well as ones that are phased (a procedure that lessens the wait time by shifting the node favorably—with a possible delta-V penalty), were included in this study. Results are shown in **Table 5-19** with supporting plots in **Figures 5-50** through **5-53**. Phasing implies inserting the CEV into a higher or lower orbit, then waiting to achieve a landing opportunity sooner than if one had remained in the circular ISS orbit. Phasing maneuvers can be used when considering the overall propellant budget. For this study, an additional delta-V of 250 ft/sec was assumed available over the normal propellant budget required for the deorbit from ISS altitude.

Table 5-19. Average and Maximum Wait Times for Deorbit Opportunities from 207-nmi Orbit

	Nominal Opportunities		Phasing Maneuver Opportunities	
Landing Site	Average Orbital Wait Time (hrs)	Maximum Orbital Wait Time (hrs)	Average Orbital Wait Time (hrs)	Maximum Orbital Wait Time (hrs)
Edwards AFB (CA)	39	71	18	28
Carson Flats (NV)	35	71	17	31
Moses Lake (WA)	21	28	15	23
All Sites Considered	10	28	8	21

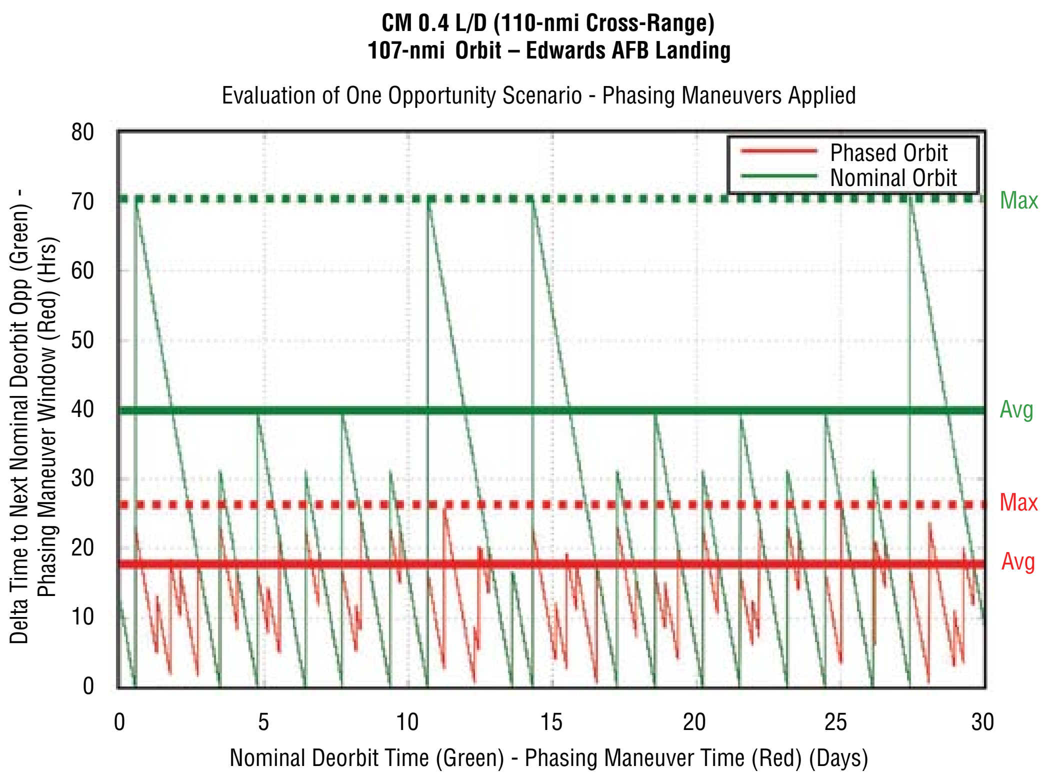


Figure 5-50. Edwards AFB Deorbit Opportunities

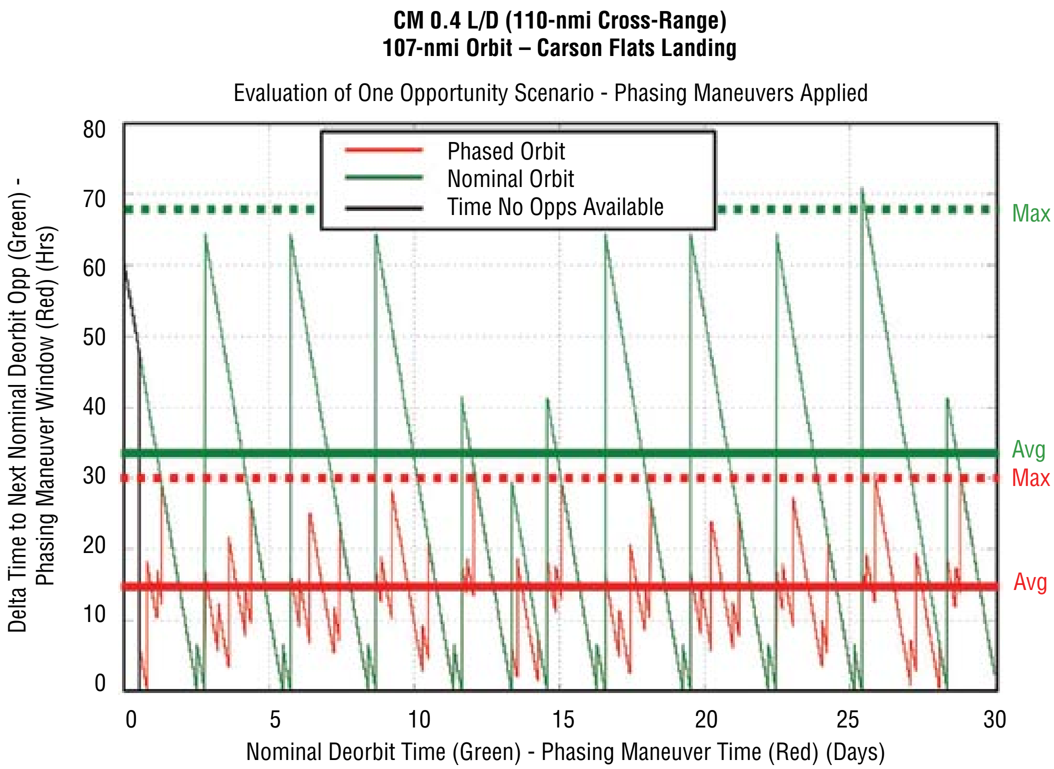


Figure 5-51. Carson Flats Deorbit Opportunities

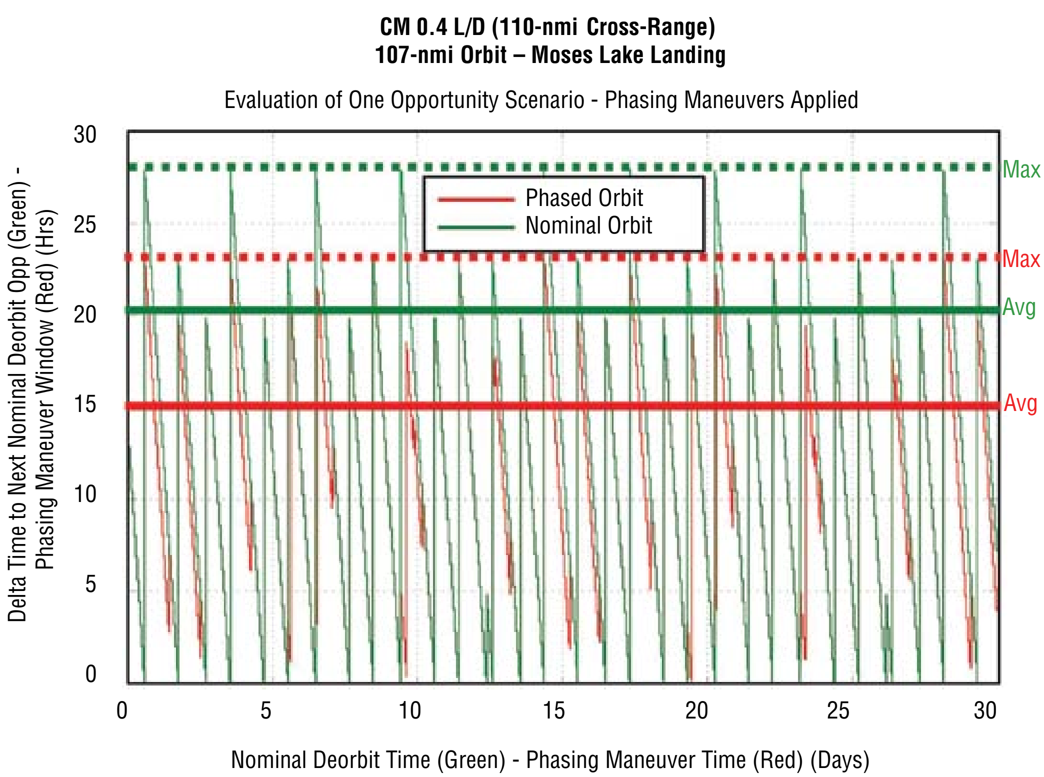


Figure 5-52. Moses Lake Deorbit Opportunities

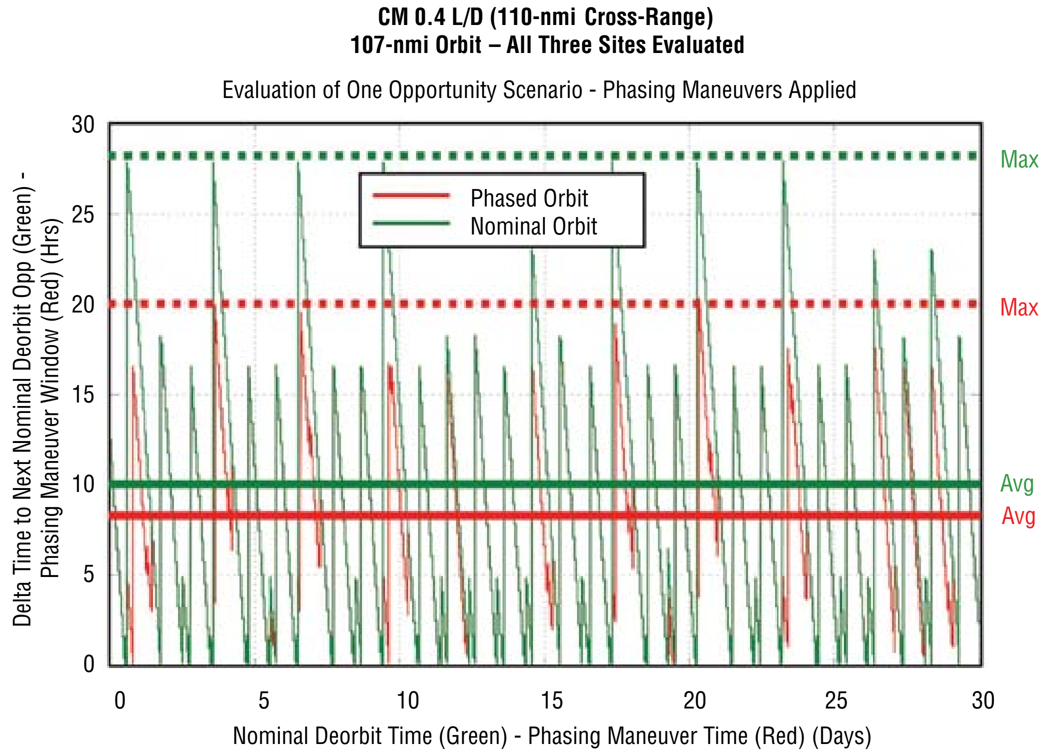


Figure 5-53. Deorbit Opportunities for All Three Sites Combined

It should be noted that the vehicle's operational altitude, vehicle cross-range capability, and site latitude location will change the landing opportunity wait times. Also, consideration of densely populated areas along the ground track to the landing site will have to be a part of a detailed safety analysis in the site selection process. At the present time, an acceptable orbital wait time requirement for the CEV has not officially been determined. Previous program studies such as X–38 and OSP only addressed the maximum wait time allowed for medical emergencies (18 hours).

Results show that the average orbital wait time for the nominal case for Moses Lake was 21 hours. This is considerably less than either Edwards AFB (39 hours) or Carson Flats (35 hours). The gap is even wider for the maximum wait time cases. However, if all three sites are considered together, the average time lowers to 10 hours and the maximum to 28 hours. If phasing is used, almost all times are reduced considerably, with the exception of combining the three sites together. In that case, the average wait time is reduced by only 2 hours and the maximum by 7 hours.

As a general rule, the higher the north or south latitude of the site, the more opportunities are available. This makes Moses Lake a good candidate as a potential landing site. However, there are other important factors that must be considered. The possibility of a water landing should be seriously considered as an option since it would alleviate many of the problems presented in this analysis.

The plots in **Figures 5-50** through **5-53** show both the nominal (green) and the phased (red) deorbit opportunities. All nominal landing opportunities are plotted for the entire mission segment, as are the predicted phasing opportunities, which are based on a current time using a maximum allowable dwell time of 36 hours. The Y-axis shows delta time to the next opportunity in hours, and the X-axis is the mission elapsed time in days, which shows the approximate time that the deorbit opportunity needs to be performed. It should be noted that a single landing site opportunity scenario was used, as opposed to one that includes a backup site, since this information does not need to be addressed at the present time.

5.3.5.2.3 Entry Trajectory for CEV CM Returning from ISS

Process

An evaluation of the CEV returning from the ISS was conducted as part of the ESAS. A simplified CEV vehicle model was used in the 4-DOF Simulation and Optimization of Rocket Trajectories (SORT). The vehicle model consisted of an L/D of 0.4 which included constant lift-and-drag coefficients as well as a constant ballistic number throughout the entry. A complete list of the simplified CEV model can be seen in **Table 5-20**.

Lift Coefficient	0.443
Drag Coefficient	1.11
L/D	0.4
Aeroshell Diameter (m)	5.5
Mass (kg)	10,900
Ballistic Number (kg/m^2)	413.32

Table 5-20. Simplified CEV Model

All entry scenarios were flown assuming two entry techniques, guided and ballistic (spinning). The guided trajectories were all flown using the Apollo Final Phase Guidance (AFPG) logic to converge on a range target. This guidance was used for all Apollo reentries, and a derivative is currently being slated as the Mars Science Laboratory (MSL) entry guidance. The ballistic entry cases were flown at the same angle-of-attack as the guided cases, which produced the same amount of lift; however, the vehicle was given a constant spin-rate (bank-rate) to null out the lift force.

Each of the two entry modes had its own set of constraints for the entry design to accommodate. An ISS return mission had the following constraints for a nominal guided entry:

- The g-load profile experienced during entry had to be less than the maximum limits for a deconditioned crew member. (Limits are provided in **Appendix 5E**, **Crew G-Limit Curves**.)
- The vehicle had to fly at least approximately 450 nmi more range than the SM disposal to ensure proper disposal of the SM in the Pacific Ocean.
- The vehicle had to converge on the target within 1.5 nmi using the current chute-deploy velocity trigger.

Ballistic entry constraints were:

- The g-load profile experienced during the ballistic entry had to be less than the maximum crew limits for an abort scenario. (Limits are provided in **Appendix 5E**, **Crew G-Limit Curves**.)
- The ballistic vehicle must land in the Pacific Ocean.

For an ISS return mission, the primary design parameters are the EI flight-path angle and the entry guidance design. Even though each entry technique had its own constraints, the flight-path angle chosen for the guided mission also had to accommodate the ballistic entry mission. Therefore, different constraints were applied to each entry technique, but both sets of constraints had to be satisfied with a single flight-path angle and entry guidance design. The flight-path angle and guidance design were adjusted until all nominal constraints were met. The associated ballistic case was then examined with the same flight-path angle to confirm that all ballistic constraints were met.

Results

The CEV trajectory for the ISS return mission met all nominal mission constraints. Assuming the nominal guided entry may become a ballistic entry in an abort scenario, the ballistic entry was also confirmed to meet all ballistic constraints. The entry flight-path angle that met all constraints was found to be –2.0 deg. This correlates to an inertial velocity at EI of 8 km/s, and the guidance design reference trajectory at 52 deg bank. **Figures 5-54** through **5-56** depict the nominal guided entry trajectory.

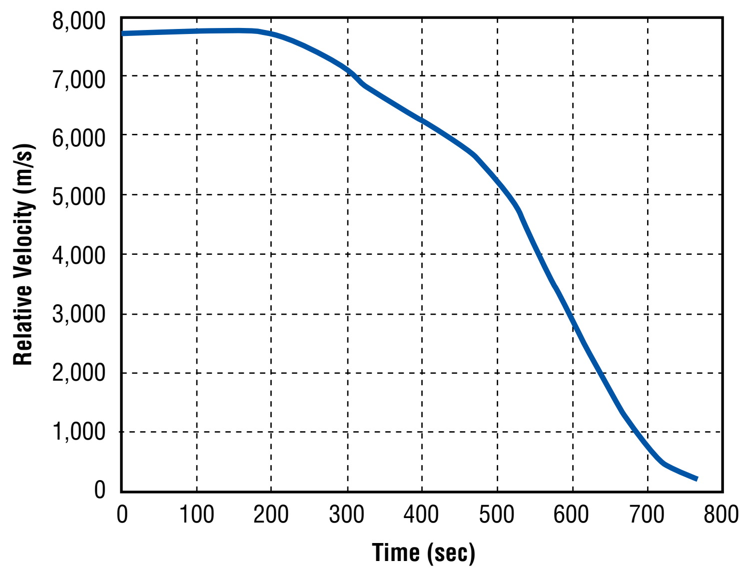


Figure 5-54. Nominal Guided – Relative Velocity Profile

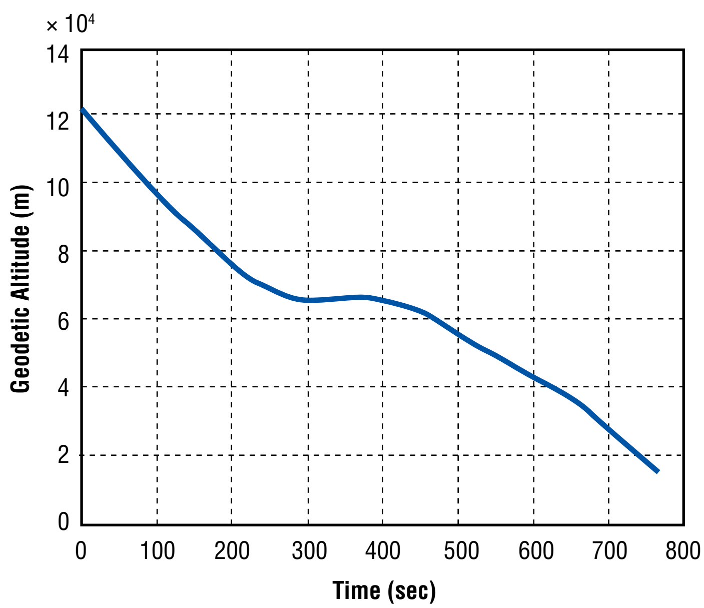


Figure 5-55. Nominal Guided – Altitude Profile

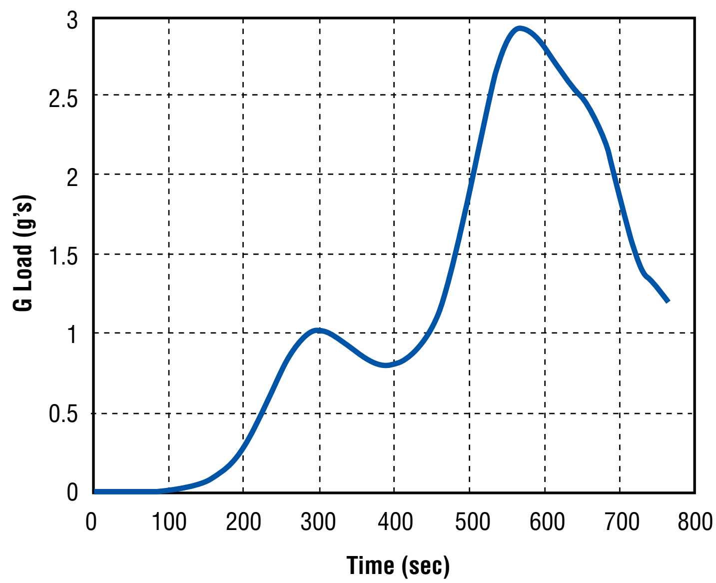


Figure 5-56. Nominal Guided – g-Load Profile

The ballistic entry (spinning) trajectory is shown in **Figures 5-57** through **5-59**).

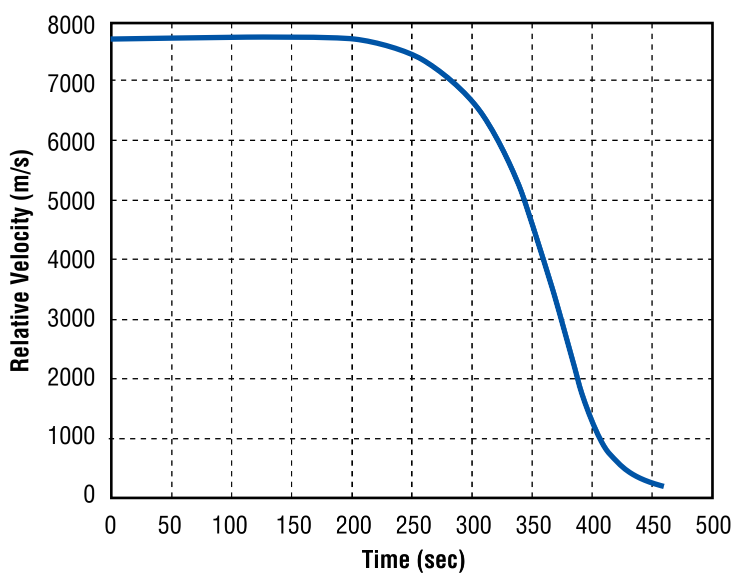


Figure 5-57. Ballistic Entry – Relative Velocity Profile

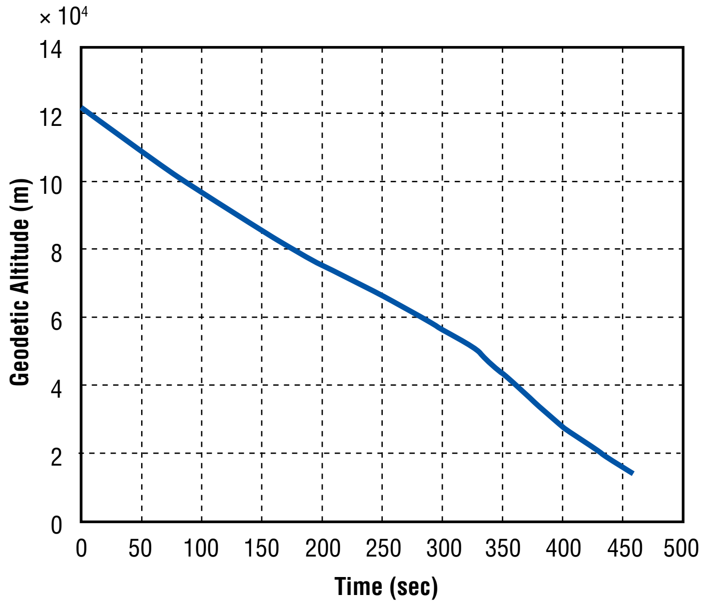


Figure 5-58. Ballistic Entry – Altitude Profile

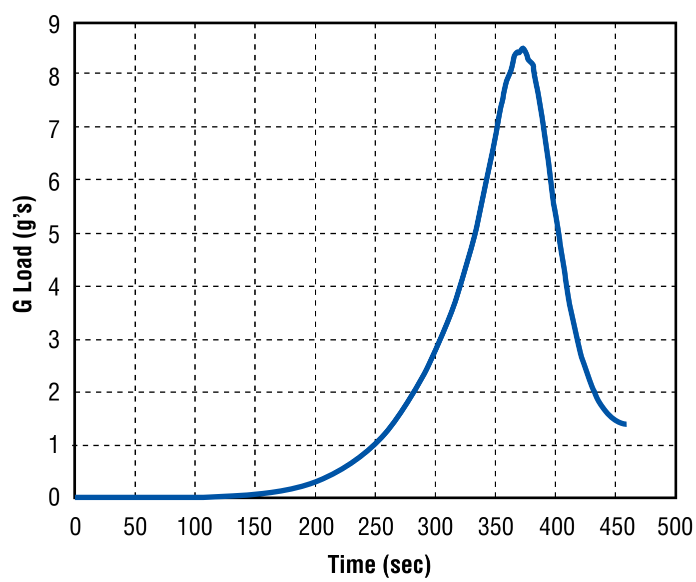


Figure 5-59. Ballistic Entry – g-Load Profile

Once the initial design analysis was completed, a corridor analysis was conducted using the nominal flight path angle and guidance design. The goal of a corridor analysis is to understand the overall capability of the vehicle to converge on the target and stay within constraints. The process starts by setting up the nominal guidance design and entry flight-path angle. The trajectory is then dispersed by steepening or shallowing the entry flight-path angle along with +30 percent of the atmospheric density for the steep case and –30 percent for the shallow case. The guided entry simulation is run using the nominal guidance design with the trajectory dispersions to confirm the vehicle's ability to still achieve the target and stay within constraints. The bounds of the corridor are determined when the vehicle no longer achieves the target or a trajectory constraint is not met. The corridor analysis revealed a corridor size of approximately 1 deg, which is sufficient, with margin, for the ISS return mission.

The ISS return mission was designed using an undispersed trajectory; thus, the design had to have margin so that the constraints would still be met when dispersions were applied. In order to confirm that all constraints would be met when dispersions were applied, a Monte Carlo analysis was conducted for both the nominal guided and ballistic entries. The Monte Carlo analysis is a statistical analysis meant to encompass all possible dispersions that may be encountered during a real-world entry. The Monte Carlo analysis included dispersions in the initial state at EI (including flight-path angle), aerodynamic uncertainties, atmosphere disturbances, and ballistic number uncertainties. For the analysis, 2,000 entry cases were simulated that applied different dispersion levels in each of the areas previously listed. The Monte Carlo analysis was used to confirm that all constraints could be met for the nominal guided and ballistic missions within the dispersed (real-world) environment. **Figure 5-60** shows a histogram plot of the maximum g-loads experienced during each of the 2,000 cases.

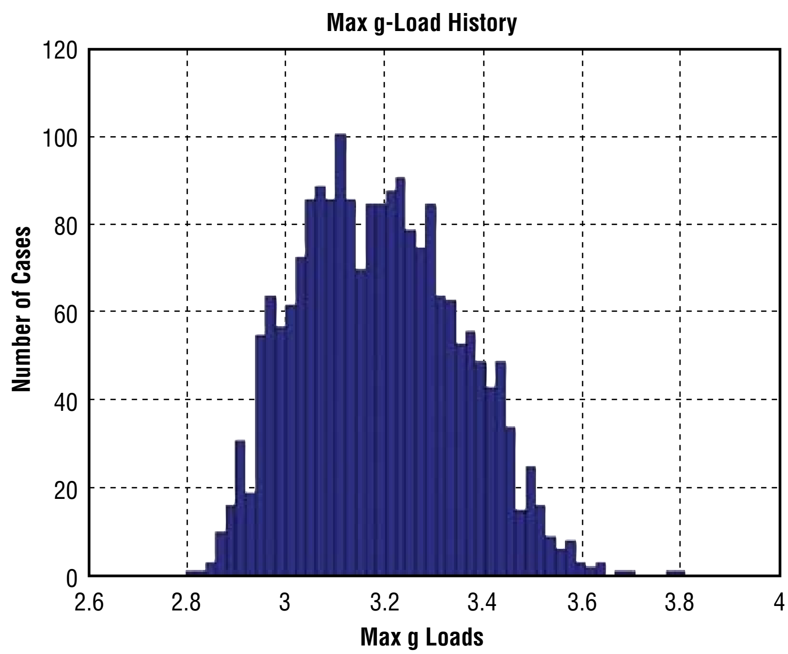


Figure 5-60. Nominal Guided Monte Carlo g-Load Histogram

As can be seen from the g-limit curves in **Appendix 5E**, **Crew G-Limit Curves**, a deconditioned crew member can withstand a 4-g load sustained (greater than 100 sec) in the X-axis ("eyeballs in") direction. Since the maximum g-load achieved for all 2,000 cases was 3.8 g's, it can be confirmed that all nominal guided cases are within the g-load limits for a deconditioned crew member.

Figure 5-61 shows the chute deploy accuracy for the same 2,000 cases.

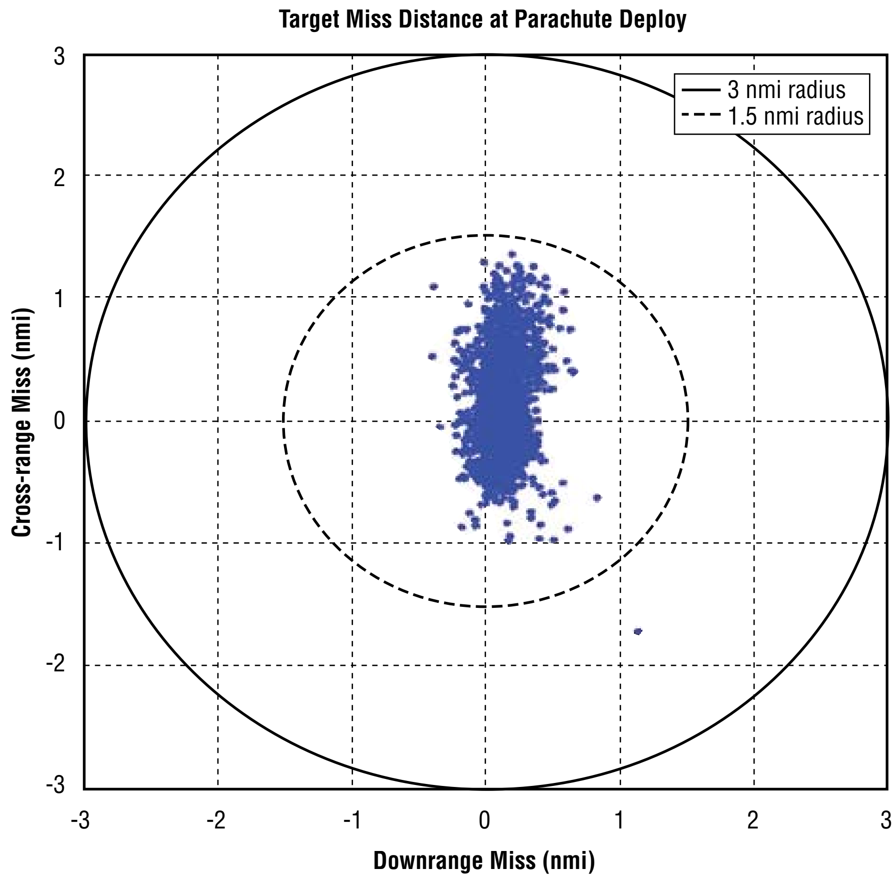


Figure 5-61. Nominal Guided Monte Carlo Target Miss Distance

All cases, except for one, are within the 1.5-nmi constraint. However, this is with a single iteration through the guidance design process. With further detailed design, this case could be brought to within 1.5 nmi. Also, the chute deploy trigger is based solely on velocity. With a more advanced chute deploy trigger and near-target guidance technique, it is believed that the target miss distance could be improved to be within 0.5 nmi or better. Based on those two assumptions, the range convergence constraint of 1.5 nmi was considered to be achieved.

The analysis for disposing of the SM in the Pacific Ocean was conducted with only a single trajectory meant to determine where the toe of the debris footprint would land relative to the target landing site. The trajectory associated with the debris toe was designed to include sufficient margin in order to represent the worst case that would come from a Monte Carlo analysis. The debris toe trajectory was given an original ballistic number of approximately 463 kg/m^2 (95 psf) and transitions to approximately 600 kg/m^2 (123 psf) at a 300,000 ft altitude. Throughout the entire entry, the debris piece was assumed to produce 0.075 L/D, which would extend the range of the toe trajectory even farther. This was determined to be very conservative and could be used to represent a worst case from a Monte Carlo analysis. The debris toe trajectory was found to land approximately 500 nmi uprange of the nominal landing target, which meets the nominal mission constraint of 450 nmi with some margin. Based on this analysis, it can be confirmed that a CEV ballistic entry would also land at least 500 nmi uprange of the target landing site, placing it in the Pacific Ocean. This is because the CEV ballistic number is less than the toe ballistic number, resulting in less range flown, and the ballistic CEV will be spinning, thus nulling the lift force and resulting in less range flown. Therefore, the ballistic constraint of landing in the ocean is met.

A Monte Carlo analysis was also conducted assuming a ballistic entry; however, in the interest of time, only 100 cases were run instead of 2,000. This will result in less confidence in the statistical analysis, but should still allow a general trend to be established and a good approximation of the maximum value if 2,000 cases had been simulated. The histogram plot of the maximum g-loads is shown in **Figure 5-62**.

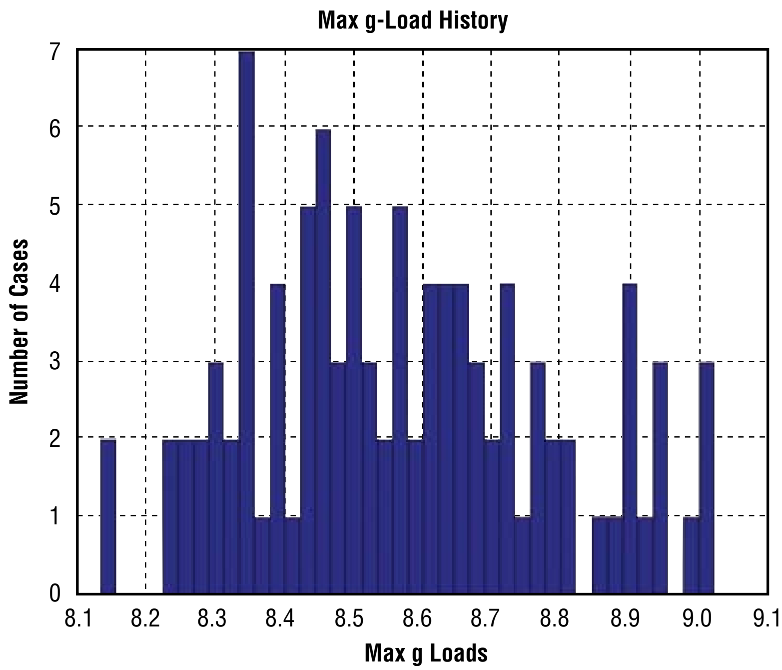


Figure 5-62. Ballistic Entry Monte Carlo G-Load Histogram

The histogram charts show a maximum g-load of roughly 9 g's. It is believed that a 2,000-case Monte Carlo would result in a maximum g-load of roughly 9.2 g's. An assessment of the g-load profile was conducted against the maximum g-load limits for an abort scenario and found to be within the limits in **Appendix 5E**, **Crew G-Limit Curves**.

All ISS return mission constraints were met with single-case trajectory designs and later confirmed with Monte Carlo analysis. Further analysis could be conducted to strengthen the confidence in the ballistic entry scenario. Analysis could also be conducted with updated models that would more accurately model the CEV capability for EI targeting, aerodynamic uncertainty, and navigation capability. Based on this first iteration approach at an entry design with the CEV, acceptable entry trajectories can be designed and flown to meet all entry constraints for a nominal guided and ballistic entry.

5.3.5.3 Return from Lunar Missions CEV Entry Trajectories

5.3.5.3.1 Landing Mode Skip-Entry Technique Description

The "skip-entry" lunar return technique provides an approach for returning crew to a single CONUS landing site anytime during a lunar month. This is opposed to the Apollo-style entry technique that would require water or land recovery over a wide range of latitudes, as explained in the following sections. This section will discuss the top-level details of this technique, as well as the major technological and vehicle system impacts.

The skip-entry trajectory approach is not a new concept. The original Apollo guidance was developed with skip trajectory capability, which was never used because of navigation and control concerns during the skip maneuver. The Soviet Union also used skip trajectories to return Zond robotic vehicles to a Russian landing site. Considerable analysis was completed in the 1990s to investigate the long-range capability of vehicles in the 0.5 L/D class, which, at that time, was considered the minimum L/D required to enable accurate skip trajectory entry capability. Skip-entry in its current formulation for the ESAS effort differs in two ways from previous approaches for capsule vehicles. First, the inclusion of an exoatmospheric correction maneuver at the apogee of the skip maneuver is used to remove dispersions accumulated during the skip maneuver. Secondly, the flight profile is standardized for all lunar return entry flights. Standardizing the entry flights permits targeting the same range-to-landing site trajectory flown for all return scenarios, stabilizing the heating and loads that the vehicle and crew experience during flight. This does not include SM disposal considerations that must be assessed on a case-by-case basis.

The Standardized Propulsive Skip-Entry (SPASE) trajectory begins at the Moon with the targeting for the TEI maneuver. The vehicle is placed on a trajectory that intercepts EI (121.9 km, 400,000 ft) at Earth at the correct flight-path angle, latitude, time (longitude), range, and azimuth to intercept the desired landing site. **Figure 5-63** shows the geometry and the resulting ground tracks at two points, 11,700 km (6,340 nmi) and 13,600 km (7,340 nmi) antipode range, along two Constant Radius Access Circles (CRACs). The antipode is targeted to slide along the desired CRAC during the lunar month, fixing the range to the desired landing site. The flight-path angle, longitude, and azimuth are controlled via the TEI maneuver back at the Moon, establishing the required geometry to accomplish the return entry flight. The Moon is shown at a maximum declination of ±28.6 deg. The entry vehicle enters the atmosphere at lunar return speed (approximately 11.1 km/sec) and then steers to a desired exit altitude and line-of-apsides. Currently, this altitude is approximately 128 km (420,000 ft). During the coast to apogee, the navigation system is updated via GPS communication. Just before apogee of the skip orbit, a correction burn is executed using small engines on the capsule to correct for dispersions (if required) accumulated during the skip phase of flight. This maneuver steers the vehicle to an optimal set of conditions (flight-path angle and range) at the second entry point. The second entry is initiated at LEO entry speeds. The vehicle enters the atmosphere a second time and steers to the desired landing site location. The change in targeting to the shallow side of the entry corridor for the first entry enables the skip trajectory to be safely designed within guidance capability and remains a distinct difference between targeting direct-entry versus skip-entry.

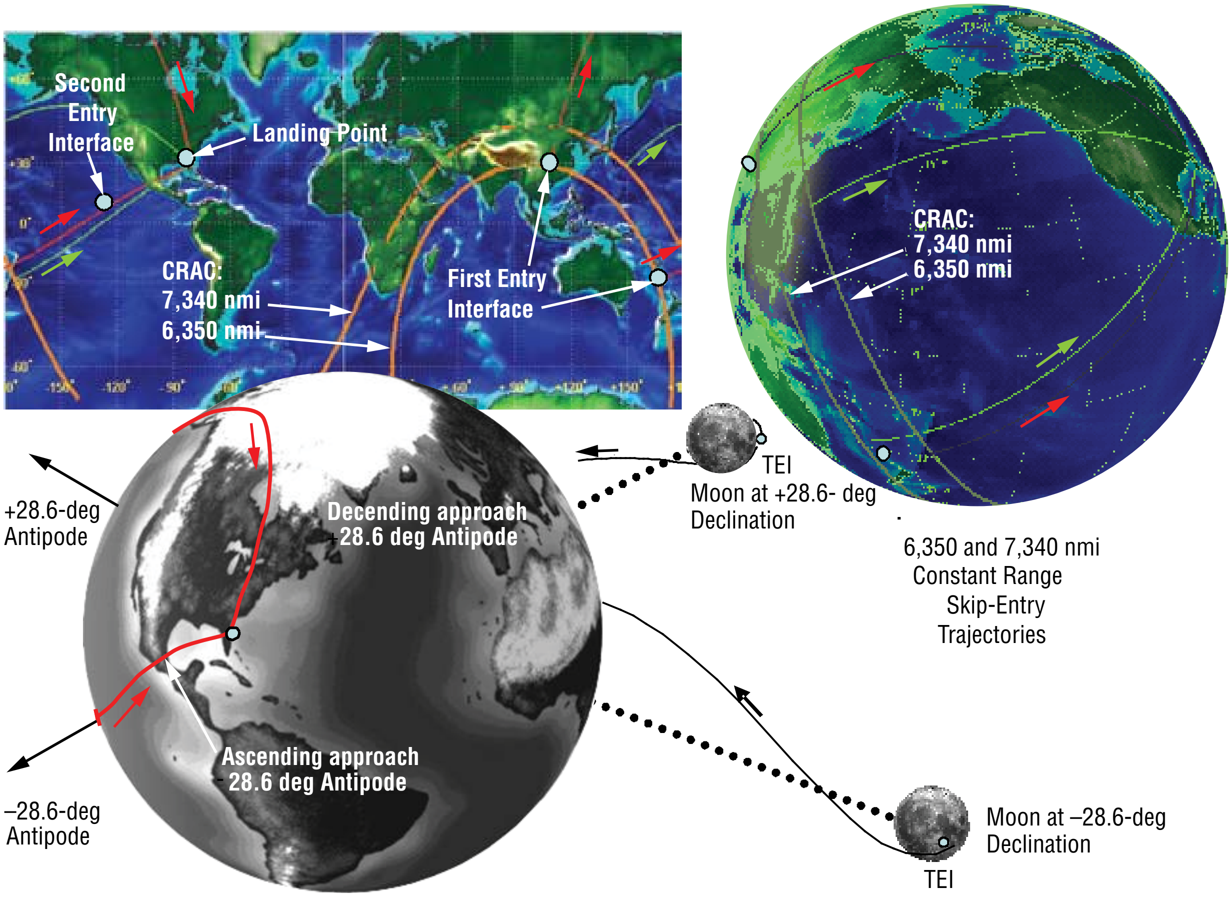


Figure 5-63. SPASE Entry Design Concept

Several state-of-the-art guidance algorithms are currently used for steering the vehicle. The generic vehicle design with 0.3 L/D used in this preliminary analysis is shown in **Figure 5-64**. The vehicle is controlled by steering the lift vector via a bank angle about the relative velocity vector. The angle-of-attack is fixed by appropriately designing the vehicle CG. The Hybrid Predictive Aerobraking Scheme (HYPAS) is used for steering the vehicle during hypersonic skip flight. The Powered Explicit Guidance (PEG) is used for the exoatmospheric correction maneuver. The Space Shuttle Entry Guidance (SEG) is used for steering the hypersonic and supersonic phases of the second entry. Finally, the Apollo Entry Guidance (AEG) is used for steering the supersonic and transonic flight phases down to parachute deployment. Ballistic chutes are released at a 6–km (20,000–ft) altitude.

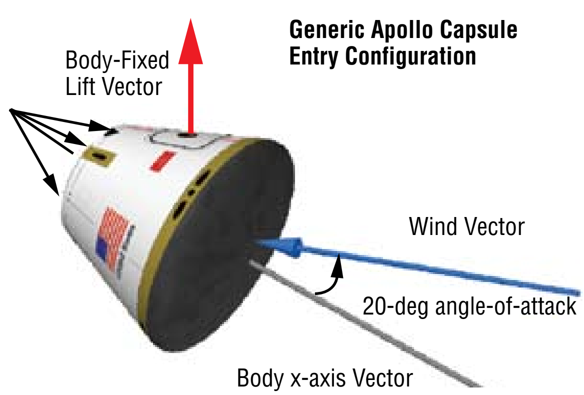


Figure 5-64. Generic SPASE Entry Capsule Concept

Figure 5-65 shows the time line of events for a 7,340-nmi CRAC SPASE flight to NASA's KSC for the Moon at +28.6-deg declination and the antipode at –28.6 deg. Note that the entire entry phase from first entry to landing is completed in less than 40 minutes. **Figures 5-66** through **5-70** provide trajectory plots for nominal flight, and **Figures 5-71** through **5-75** provide trajectory plots for a 100-case Monte Carlo. The Monte Carlo used Global Reference Atmospheric Model (GRAM) atmosphere and winds, initial state, weight, and aerodynamics uncertainties, with perfect navigation.

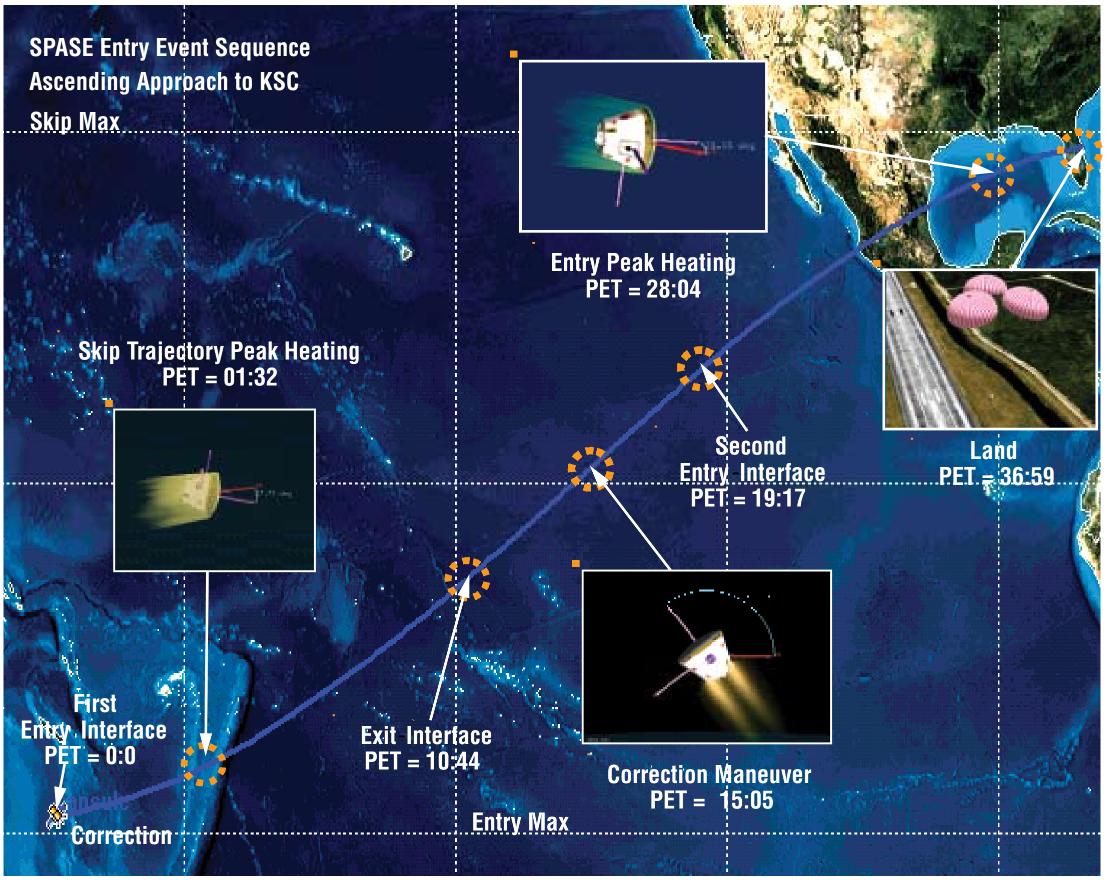


Figure 5-65. SPASE Nominal Flight Entry Event Sequence, (times shown in minutes: seconds)

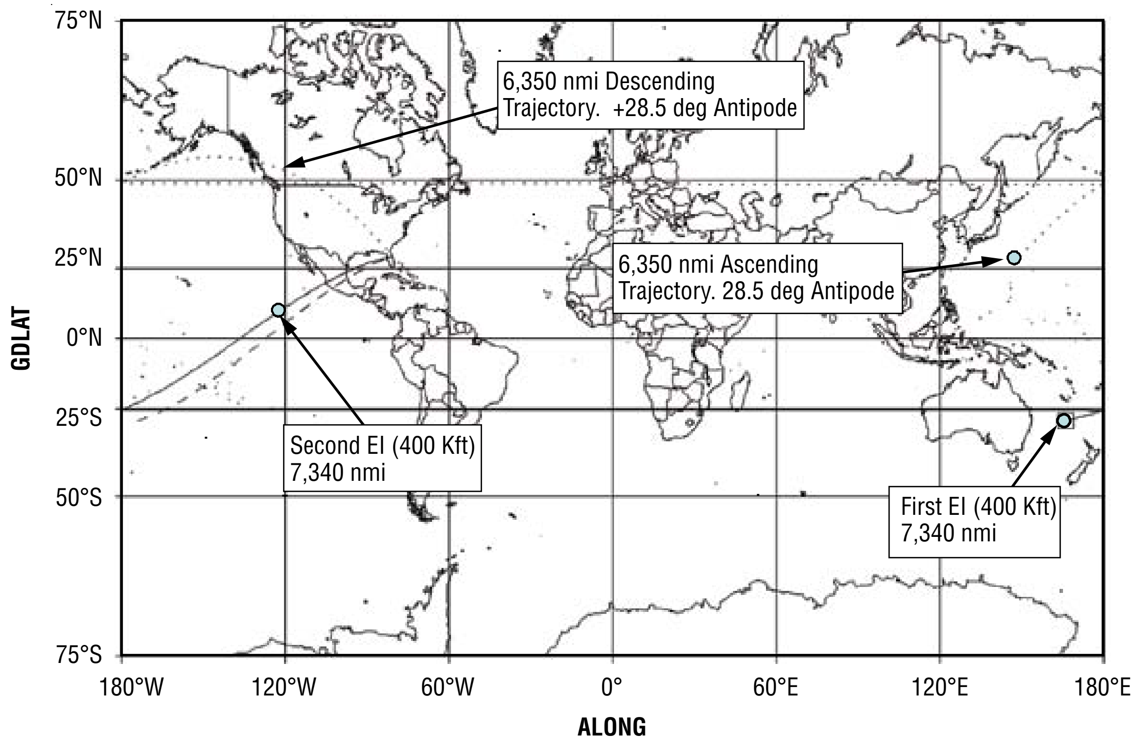


Figure 5-66. SPASE Nominal Flight Groundtrack to KSC

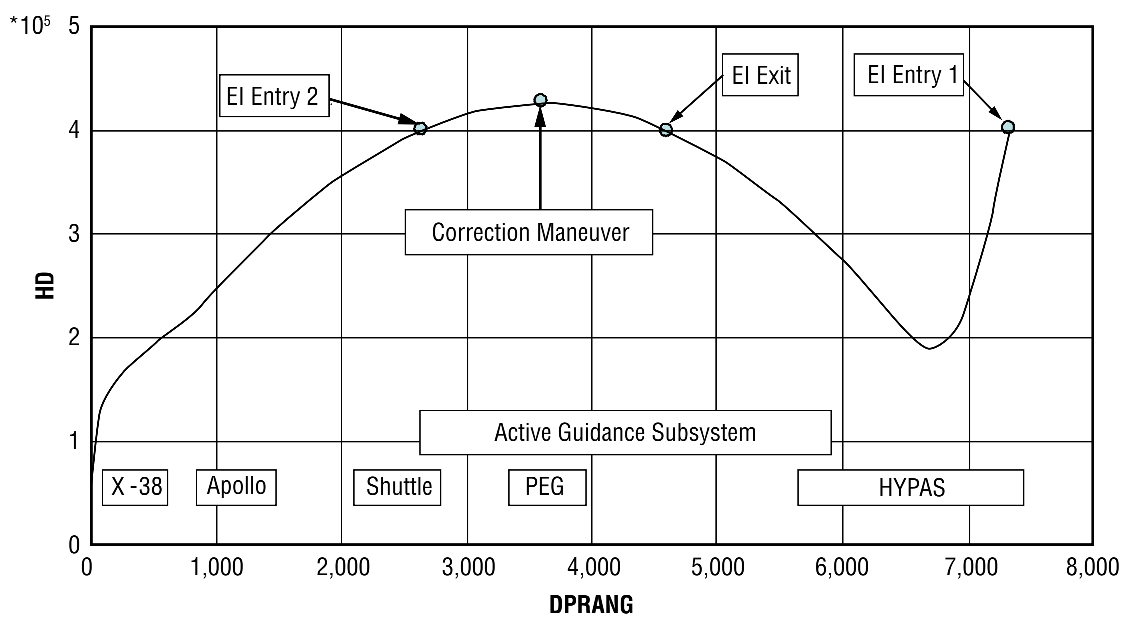


Figure 5-67. SPASE Nominal Flight Geodetic Altitude (ft) versus Range (nmi) to Landing Site

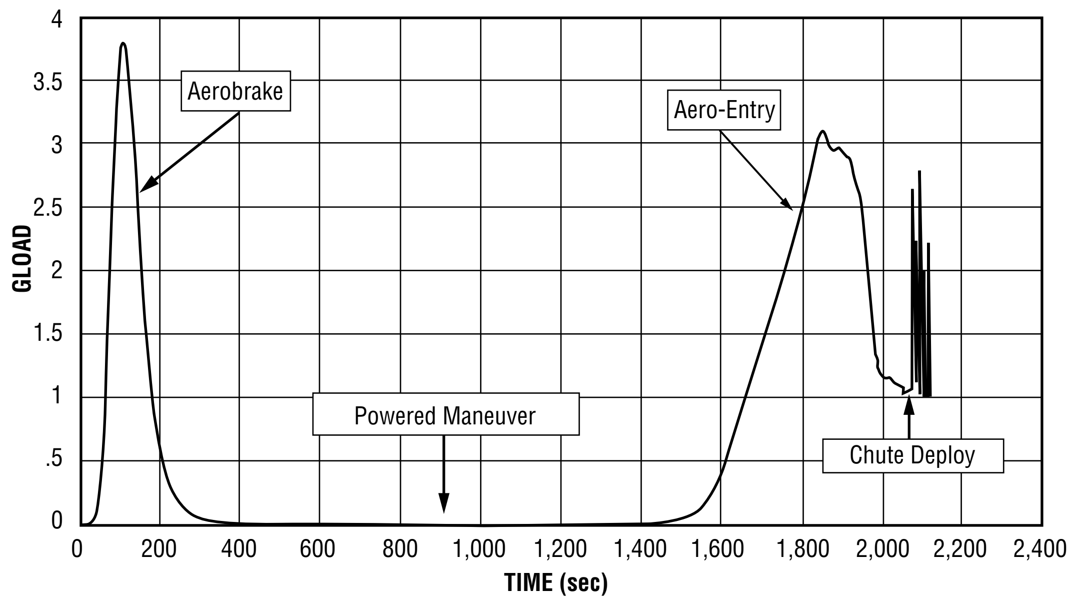


Figure 5-68. SPASE Nominal Flight Total Aerodynamic/Propulsive Acceleration (g's) versus Time (sec)

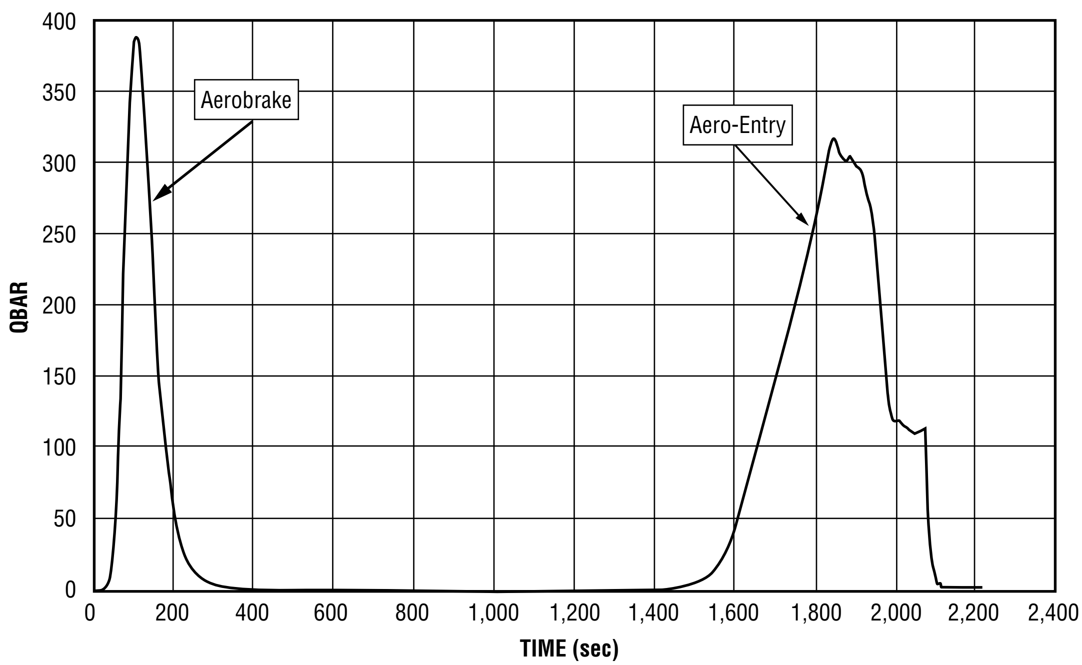


Figure 5-69. SPASE Nominal Flight Dynamic Pressure (psf) versus Time (sec)

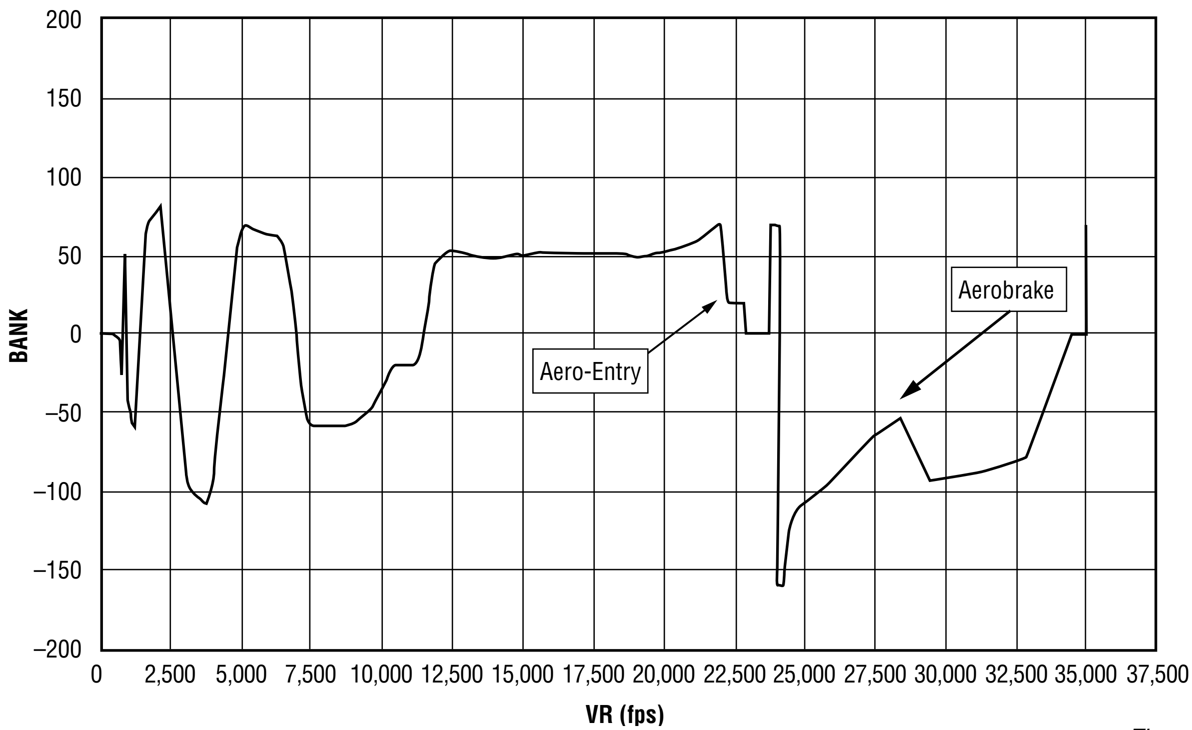


Figure 5-70. SPASE Nominal Bank Angle (deg) versus Relative Velocity (fps)

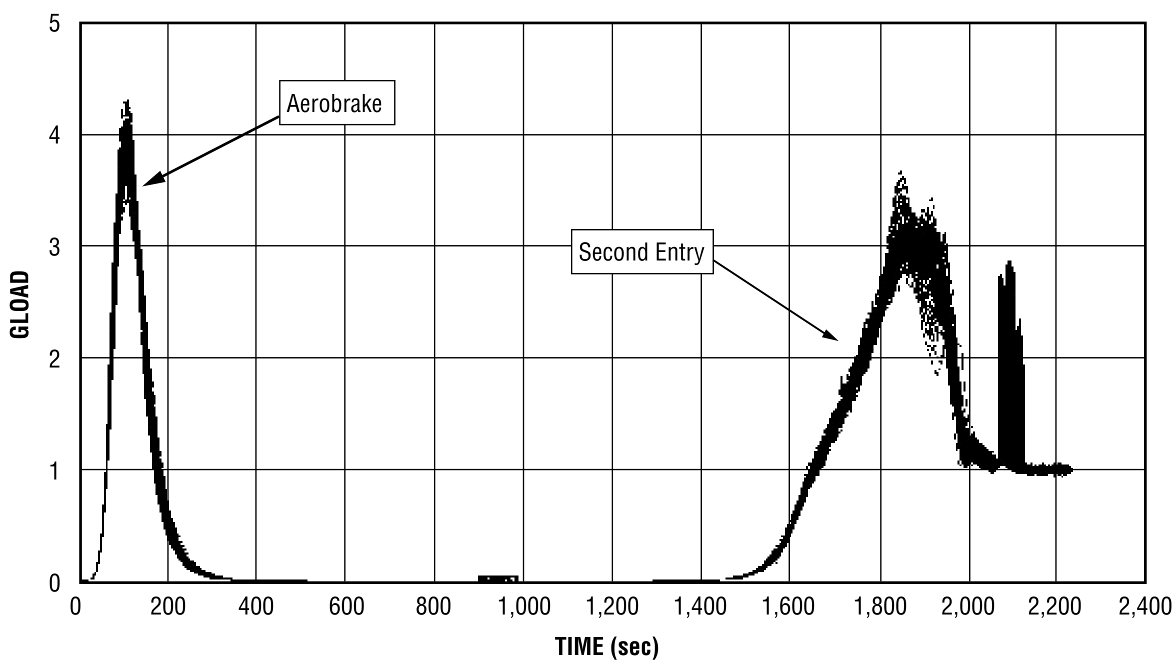


Figure 5-71. SPASE Dispersed Flight Acceleration (g's) versus Time (sec) (100 Cases)

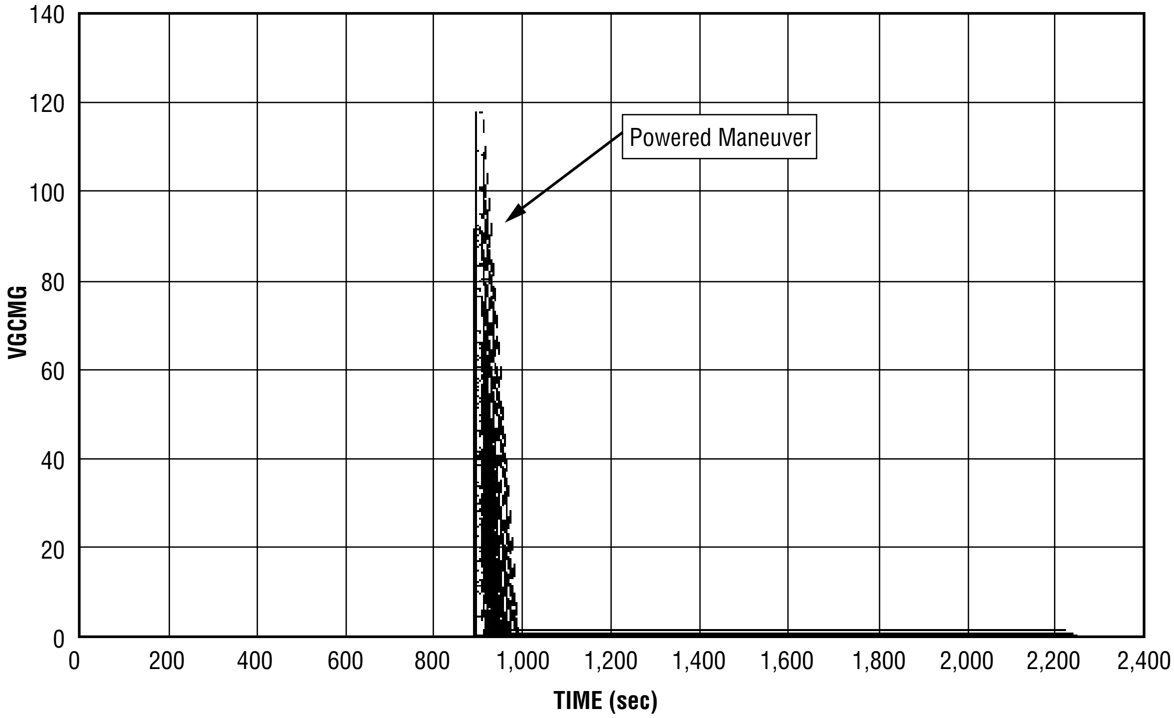


Figure 5-72. SPASE Powered Maneuver Delta Velocity Required (fps) versus Time (sec) (100 Cases)

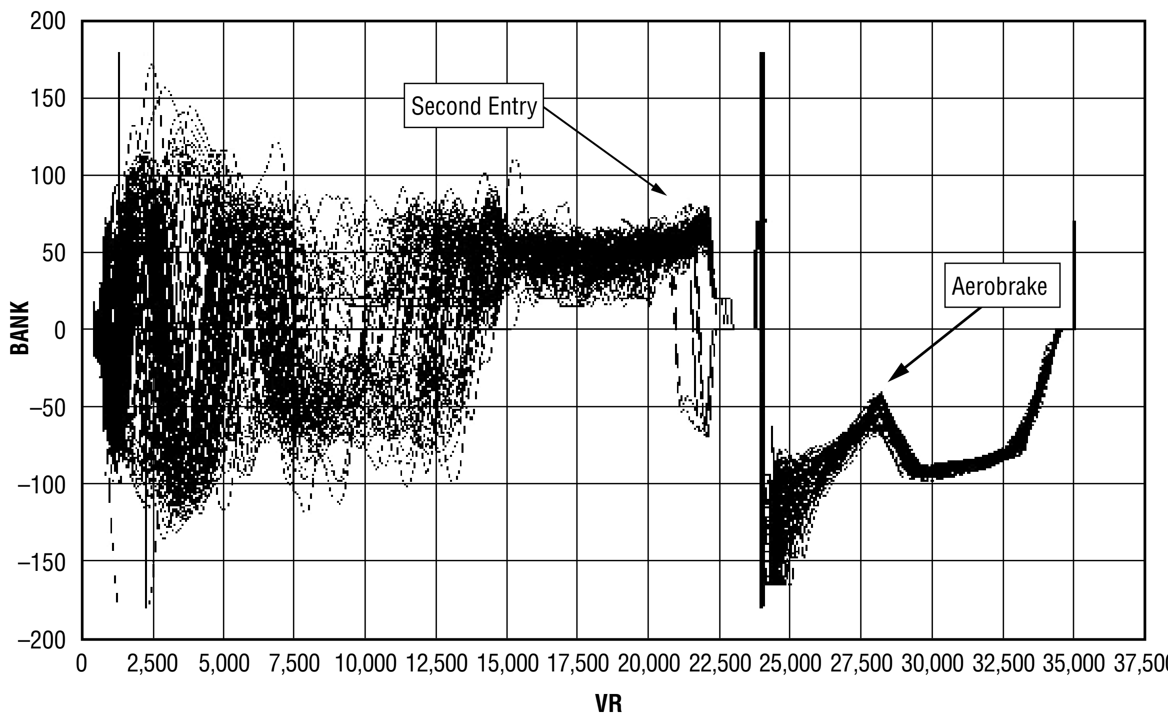


Figure 5-73. Bank Angle (deg) versus Relative Velocity (fps)–Dispersed Flight (100 Cases)

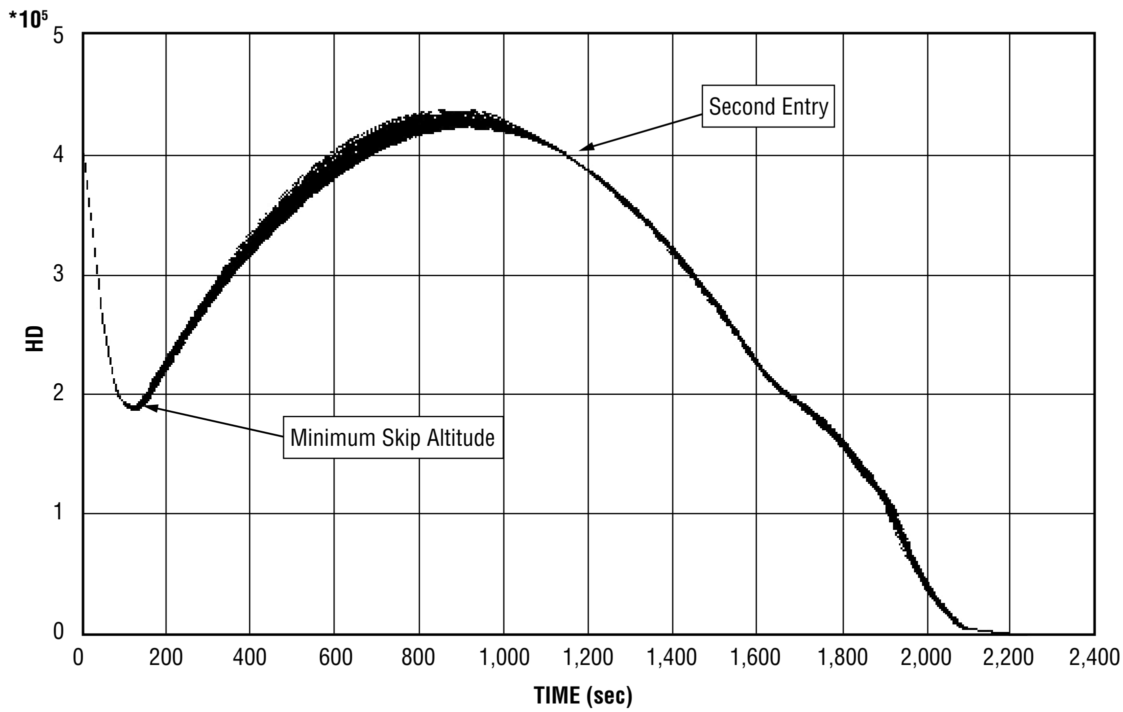


Figure 5-74. Geodetic Altitude (ft) versus Time (sec)–Dispersed Flight (100 Cases)

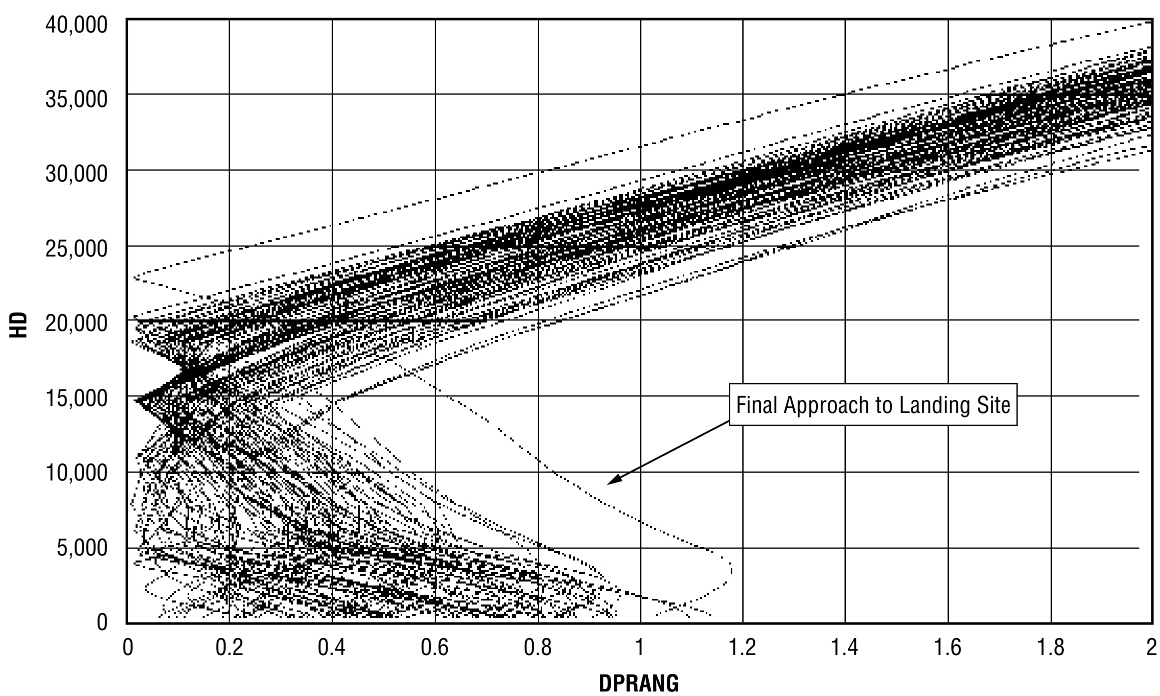


Figure 5-75. Geodetic Altitude (ft) versus Range at Final Approach (nmi)–Dispersed Flight (100 Cases)

5.3.5.3.2 Direct-Entry Versus Skip-Entry Comparison

This section will compare a lunar return direct-entry flight to a skip-entry flight. The vehicle used in this comparison will be an Apollo-style capsule with a ballistic number of 106 psf and a L/D ratio of 0.3. The drag coefficient is 1.29. The entry speed will be 36,309 fps at an EI altitude of 400,000 ft. The flight-path angle for the direct-entry flight is –6.65 deg and –6.0 deg for the skip-entry. The difference in nominal flight-path angle at EI is the most distinct difference in the targeting between skip-entry and direct-entry. The direct-entry flight is targeted to ensure capture of the vehicle and protect against skip-out of the entry vehicle. The skip-entry vehicle is designed to skip-out and, therefore, is biased into the skip side of the entry corridor. The vehicle must target lift-vector up during a majority of the skip phase to achieve the low-altitude skip target. Biasing the skip targeting to the steep side of the skip corridor is required to ensure that the vehicle will ballistically enter in case a failed control system and vehicle spin-up is required. This steep targeting is also required to ensure that the SM will ballistically enter and impact into a safe water location.

The Apollo-style direct-entry requires water- or land-landing over a wide range of latitudes. The antipode defines a vector connecting the Moon and Earth at time of lunar departure and closely approximates the landing site for a direct-entry mission. The lunar inclination, and, therefore, antipode varies from 28.6 deg to 18.3 deg over an 18.6-year lunar cycle. Therefore, depending on the lunar cycle, appropriate recovery forces or ground landing zone would have to be available within this antipode range approximately 3 days from lunar departure. For an L/D of approximately 0.3, this implies a landing site or recovery ship within 2,200 km of the EI location, or 200 km of the antipode location.

The guidance bank angle command is used to steer the entry vehicle to drogue chute deployment. The target range is 1,390 nmi for the direct-entry mission. The 1969 version of Apollo guidance is used for modeling the direct-entry flight. For the skip-entry trajectory, the HYPAS aero-braking guidance algorithm is used for the skip phase of the skip-entry flight. The Powered Explicit Guidance (PEG) algorithm is used for the exoatmospheric flight phase. For the second entry, the hypersonic phase of the SEG is used. Finally, the final phase of the AEG algorithm is used for sub-mach-5 flight to chute deployment. The target range for the skip-entry flight under analysis is 13,600 km (7,340 nmi) from EI.

The intent of this section is to quantify the trajectory differences between flying a 0.3 L/D vehicle using the standard Apollo-style direct-entry versus a skip-entry method. As can be noted from the plots in **Figures 5-76** and **5-77**, the skip-entry method provides a lower heat rate but higher heat loads than the direct-entry method. The skip-entry trajectory also has a "cooling off" period following the first aerobrake maneuver before the second entry. This will allow the heat pulse absorbed during the aeropass to soak into the structure and must be accounted for in the TPS design. The dramatic difference in range flown from EI is the most distinct difference between the trajectories. This not only extends the flight time but greatly extends the distance between the location of the SM disposal footprint. It also locates the ballistic abort CM location close to the perigee of the lunar approach orbit. The great distance between the SM disposal location could be advantageous for inland landing site locations or populated over-flight geometries; however, the great distance between the ballistic abort landing point and the nominal landing point would necessitate a mobile Search and Rescue Force to recover the crew and vehicle from the abort landing location.

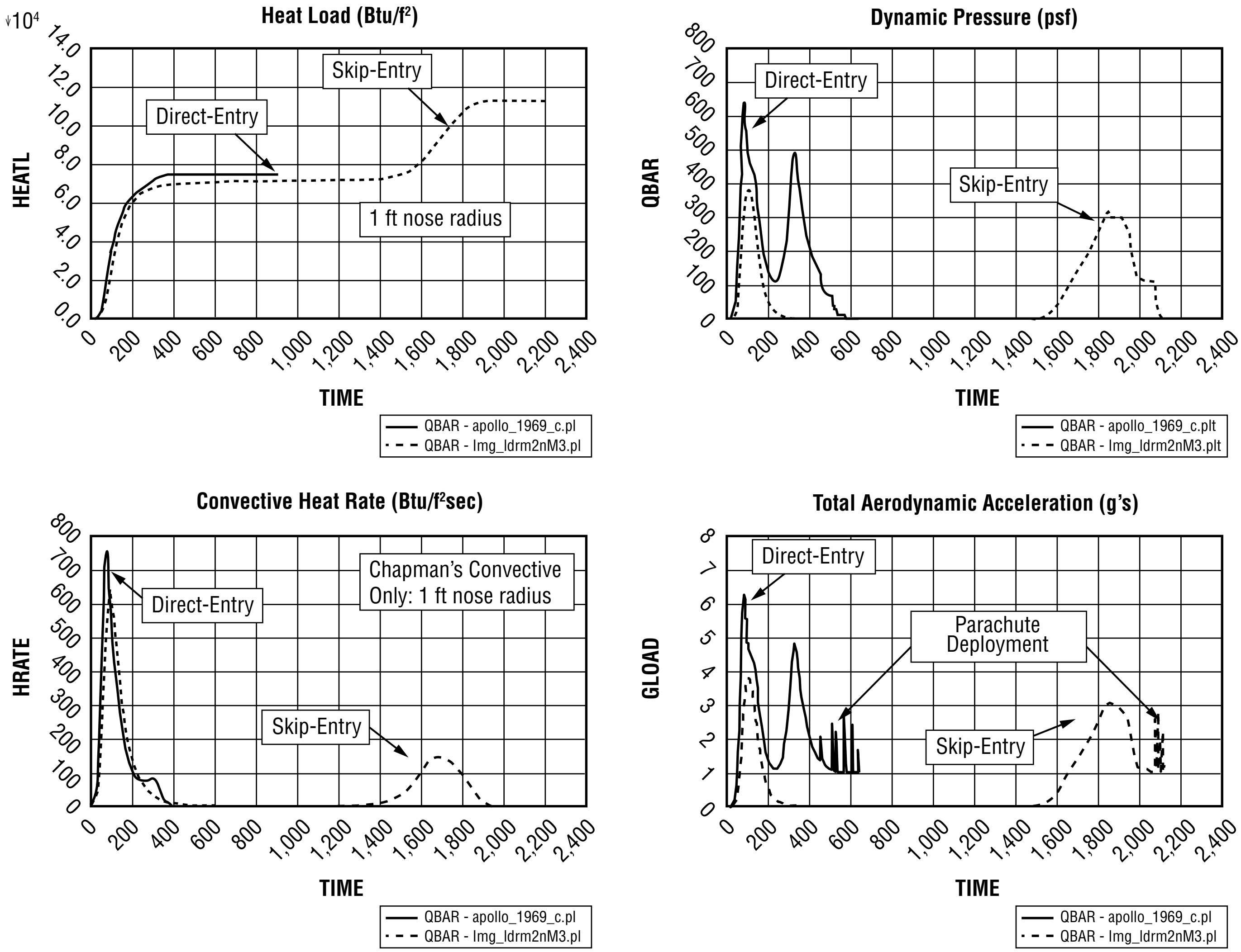


Figure 5-76. Trajectory Comparisons Direct versus Skip-Entry

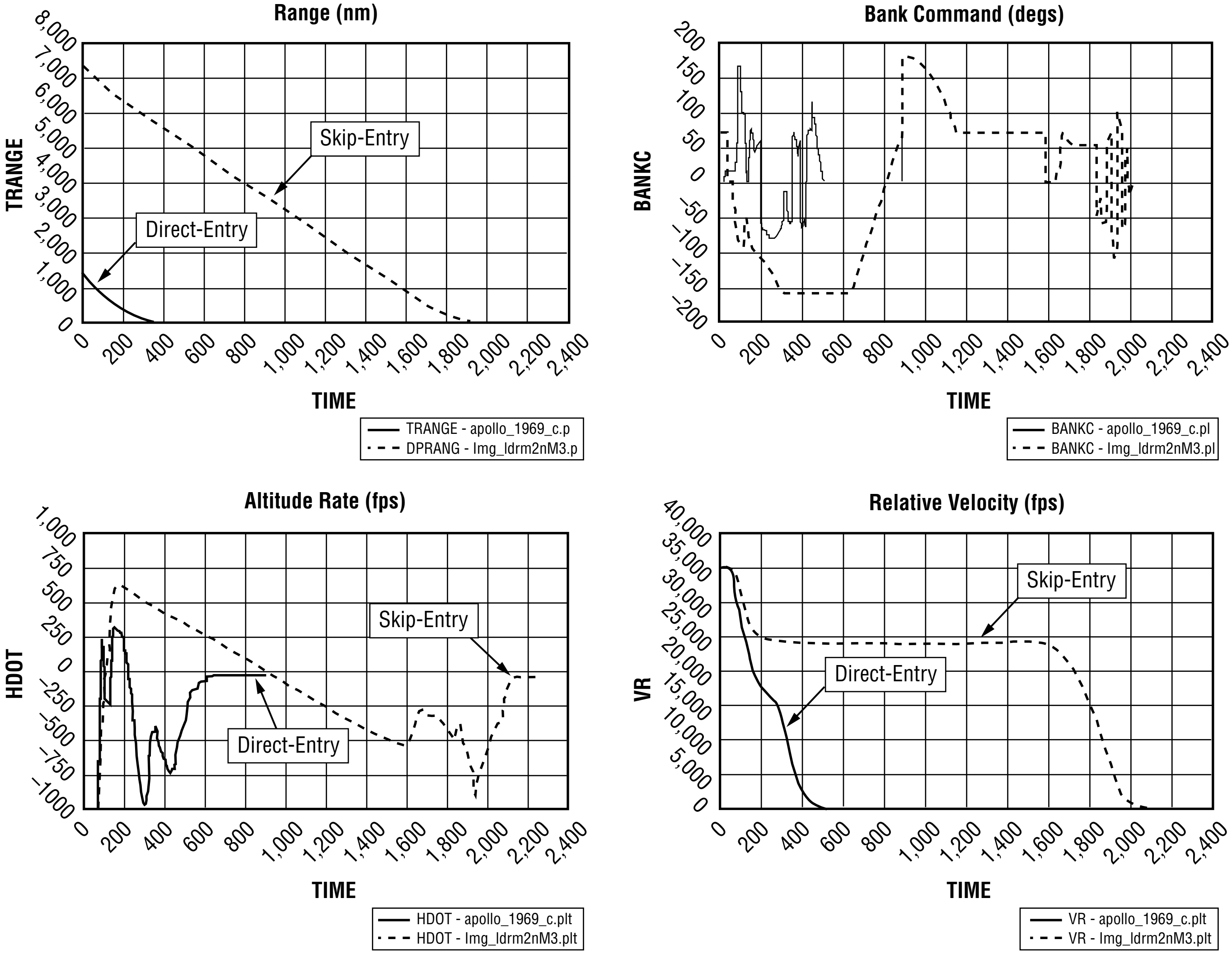


Figure 5-77. Trajectory Comparisons Direct versus Skip-Entry

5.3.5.3.3 Skip-Entry Vehicle Configuration Comparison

This section will provide a comparison of three different vehicle comparisons (**Figures 5-78** to **5-85)** for a skip-entry trajectory. The vehicles considered will be a capsule (L/D = 0.3, ballistic number = 64 psf), a biconic (L/D = 0.82, ballistic number = 199 psf), and an ellipsled (L/D = 0.66, ballistic number = 197 psf). Targeting was completed that ensured the proper amount of energy is depleted for an exoatmospheric apogee altitude of approximately 420,000 ft. This implied a capsule EI flight-path angle of –5.83 deg, a biconic EI flight-path angle of –6.94 deg, and an ellipsled EI flight-path angle of –6.5 deg. The steeper flight-path angles required for the ellipsled and biconic are a result of the higher ballistic number and the increased lift acceleration used for exit-phase targeting. The entry conditions for all vehicles simulate a lunar return with an inertial entry velocity of 11.1 km/sec (36,300 fps).

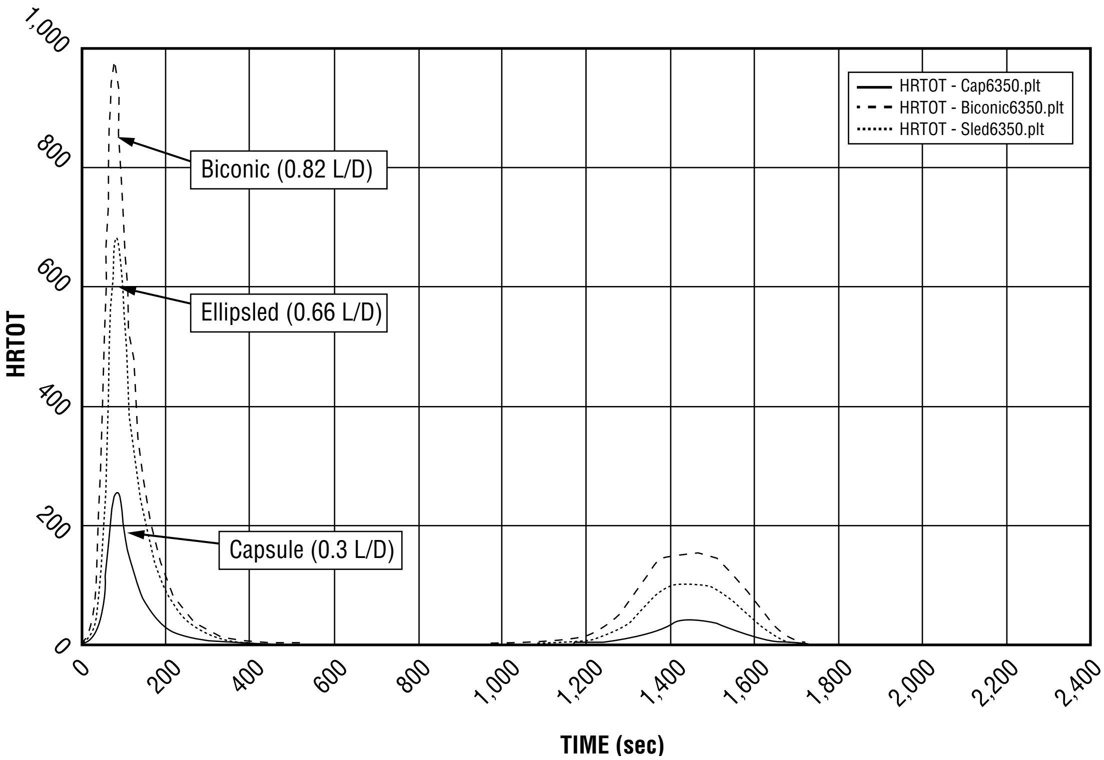


Figure 5-78. Total Heating - Radiative Plus Convective (1-ft Radius Sphere)

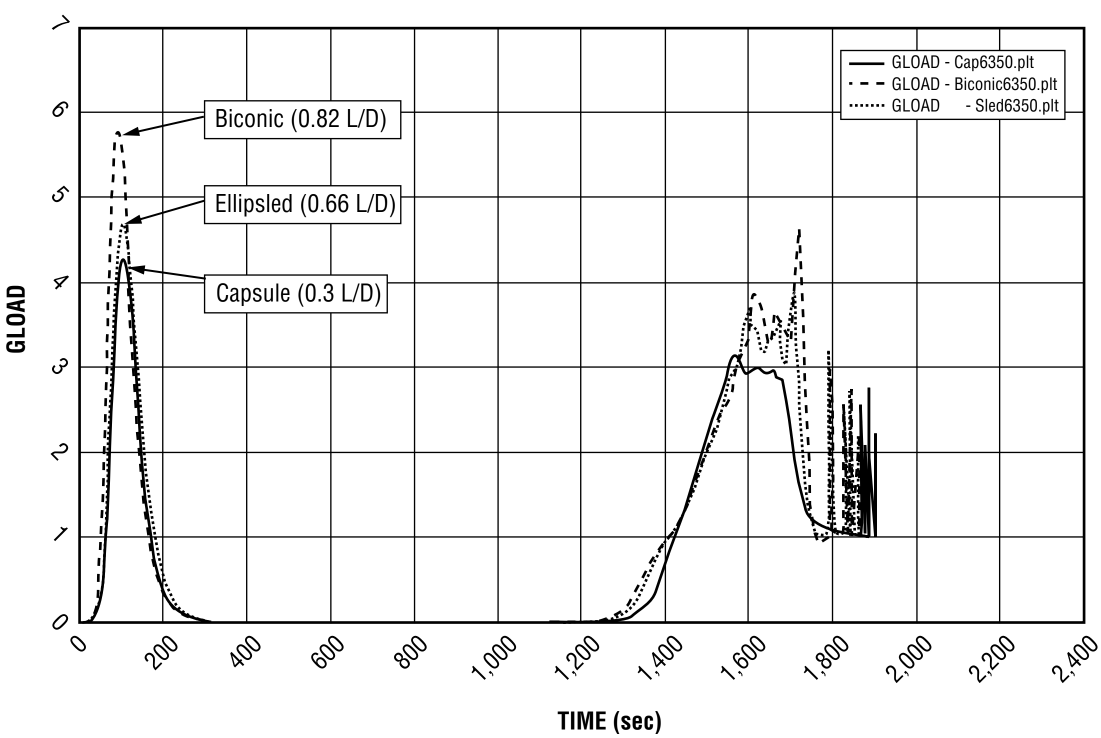


Figure 5-79. Total Aerodynamic Acceleration (g's) versus Time from EI (sec)

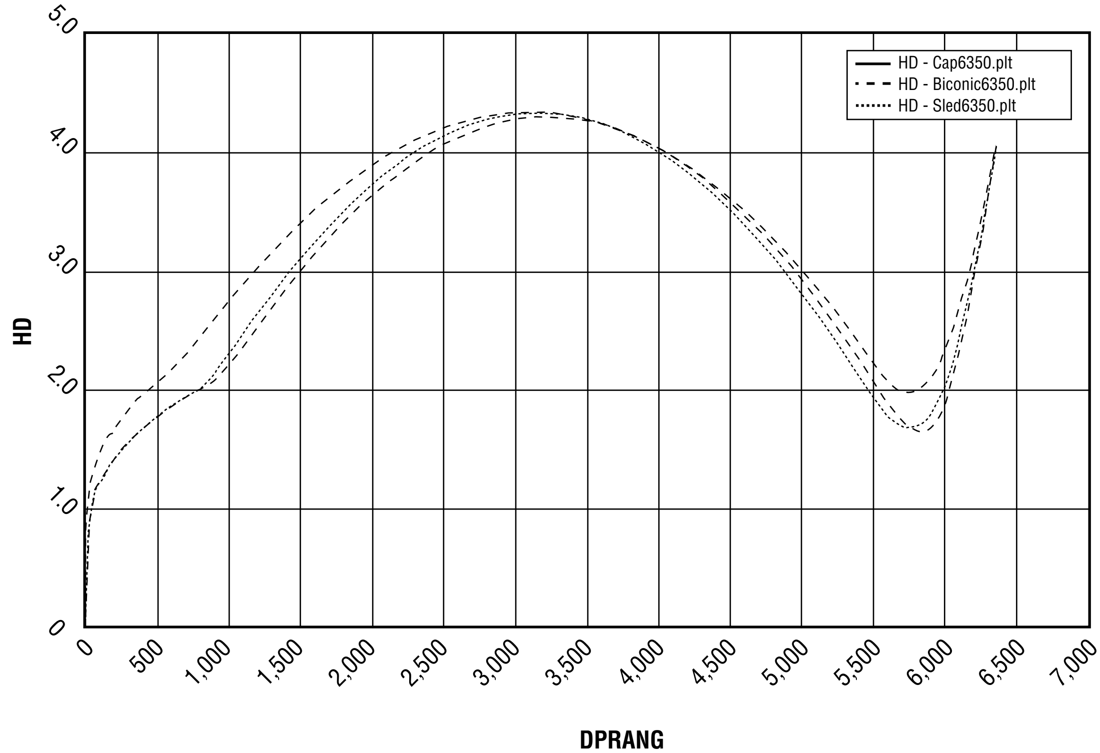


Figure 5-80. Geodetic Altitude (ft) versus Range-to-Target, DPRANG (nmi)

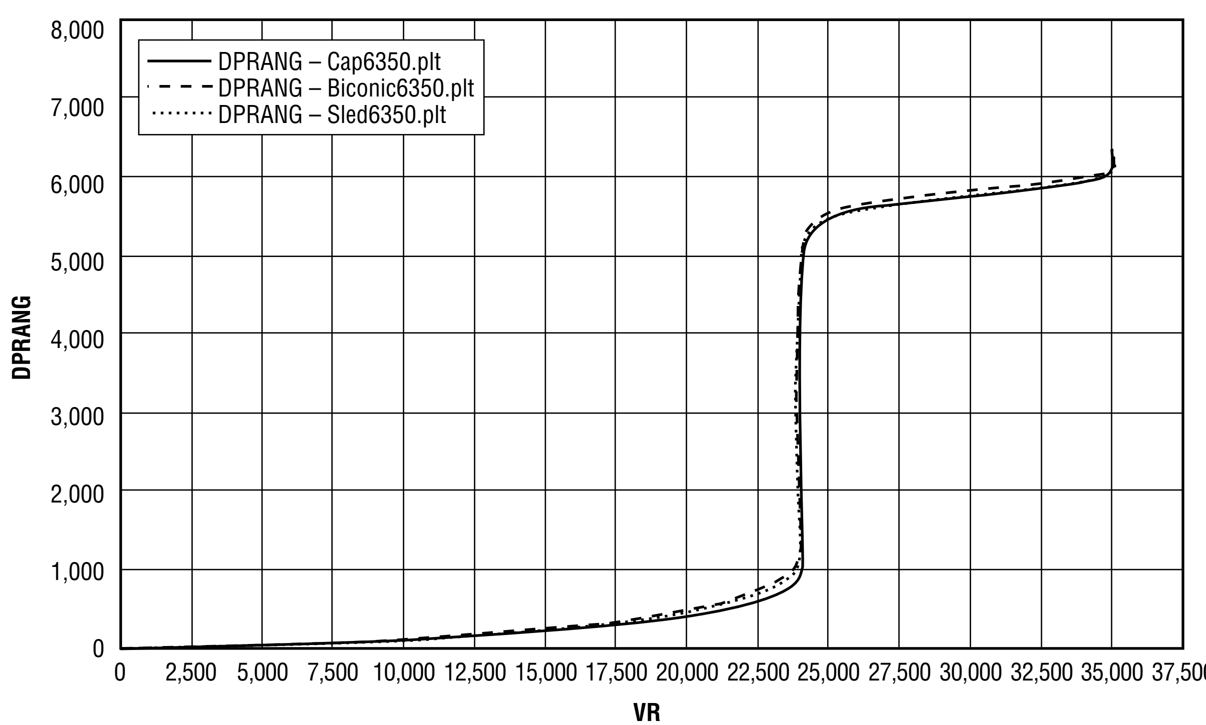


Figure 5-81. Range-to-Target (nmi) versus Relative Velocity (fps)

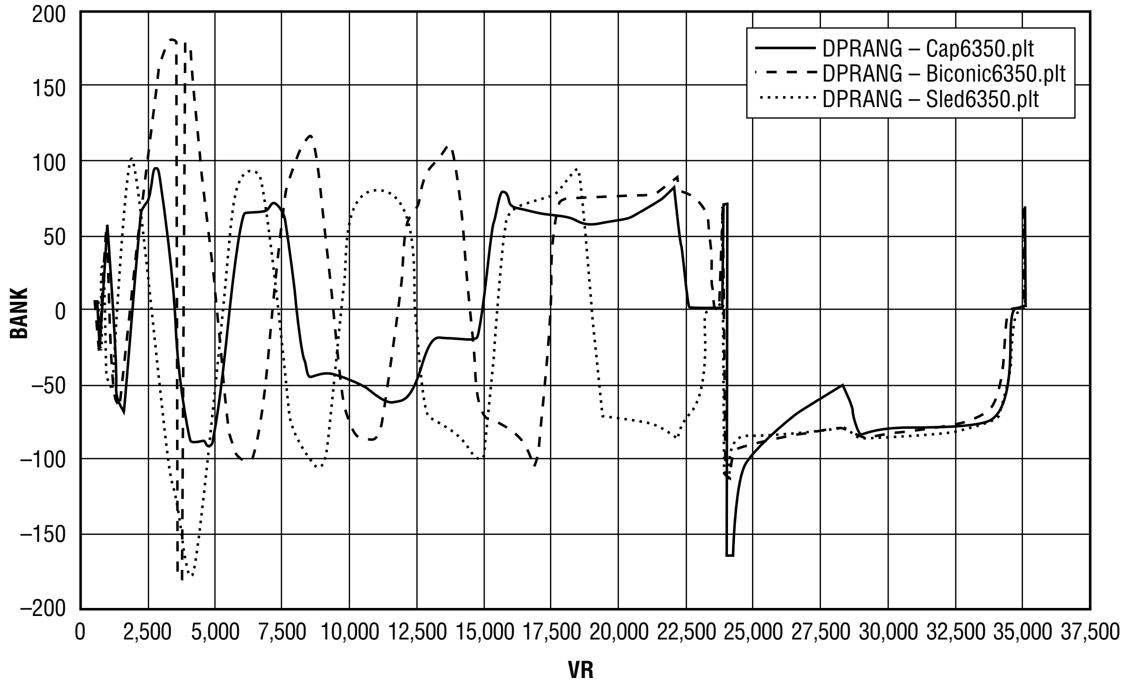


Figure 5-82. Bank Angle versus Relative Velocity (fps)

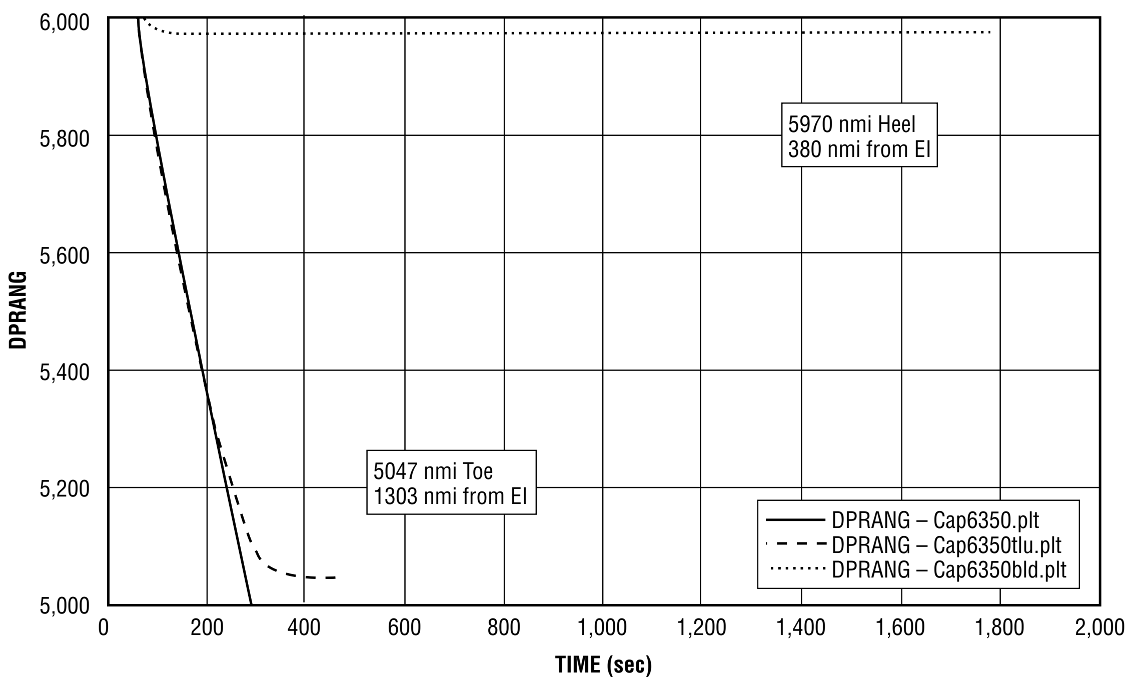


Figure 5-83. Capsule SM Footprint

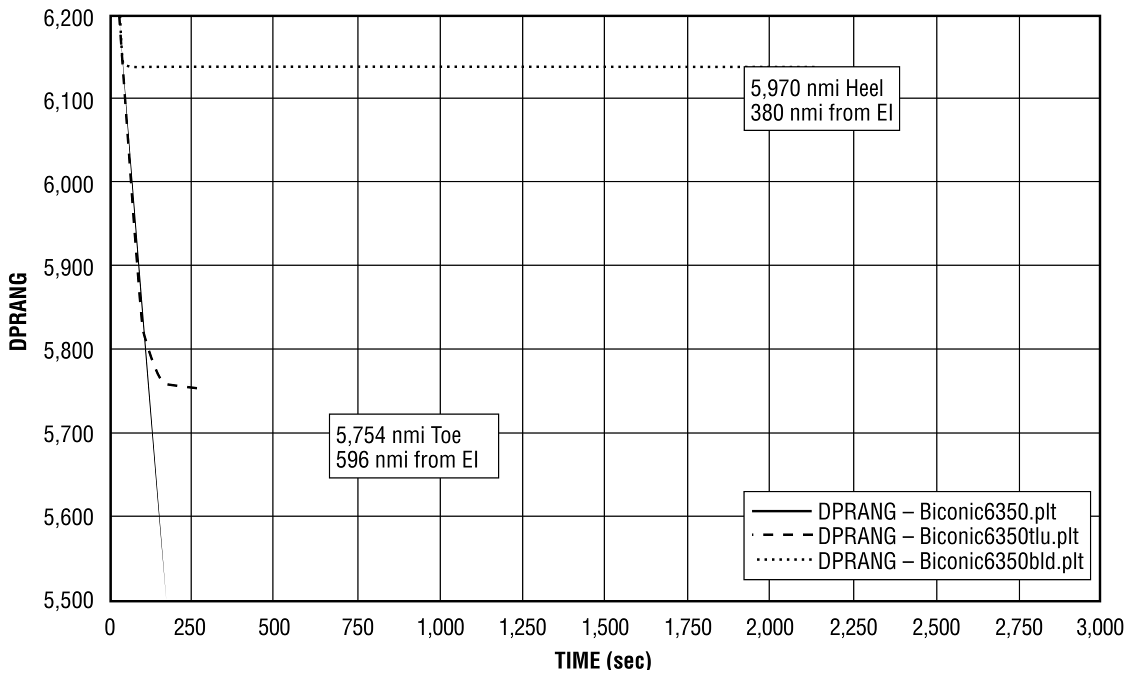


Figure 5-84. Biconic SM Footprint

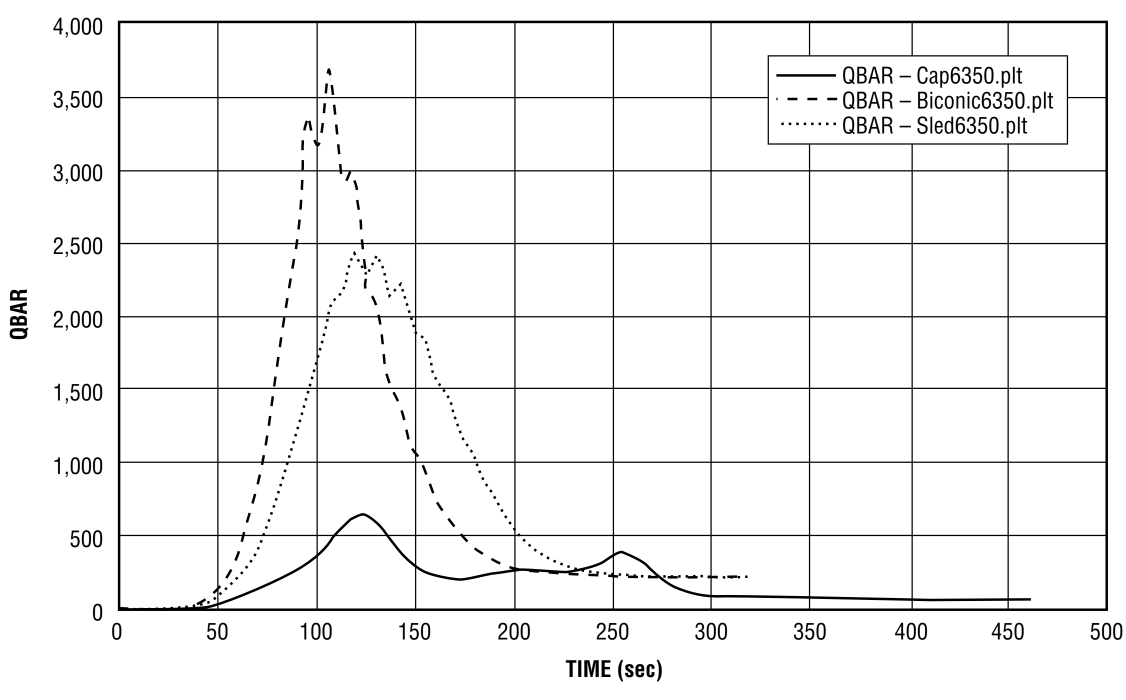


Figure 5-85. Ballistic Entry Comparison - Dynamic Pressure (psf) versus Time

5.3.5.3.4 Skip-Entry Vehicle (0.3 and 0.4 L/D) Site Accessibility

This section provides information on the site accessibility for the 0.3 and 0.4 L/D vehicles. **Figures 5-86** and **5-87** provide the footprint comparisons and the strategy for controlling the approach azimuth to the landing site using co-azimuth control during the TEI maneuver at the Moon. (Note that the footprint can be rotated about the antipode by controlling the entry azimuth.) This technique permits an alignment of the approach geometry with the desired landing site. For direct-entry scenarios, this permits alternatives for approaching the landing site for SM disposal considerations, or perhaps populated over-flight concerns. For SPASE trajectories, this enables the antipode and the landing site alignment to achieve the desired landing site. Note that the 0.4 L/D vehicle provides more than 500 km of additional direct-entry footprint than the 0.3 L/D vehicle. This has important implications for achieving direct-entry inland landing sites while maintaining the required coastal SM disposal clearance.

EI, Vacuum Perigee (VP), and the entry footprint are all interrelated via the entry design process (**Figures 5-86** and **5-87**). The antipode is fixed to the landing site at the time of lunar departure. However, VP moves relative to the antipode, and, therefore, to the landing site, by as much as 430 km over ±12 hours of flight time variation. This variation in flight time is controlled by the TEI maneuver and is required to allow the Earth to spin into proper entry orientation. The amount of flight time variation required to achieve the desired Earth-relative longitude is not known until lunar departure; therefore, as much as 430 km of footprint must be "reserved" to account for the flight time variation. If this is not done, an opportunity could arise where the footprint would lie outside of the desired landing site.

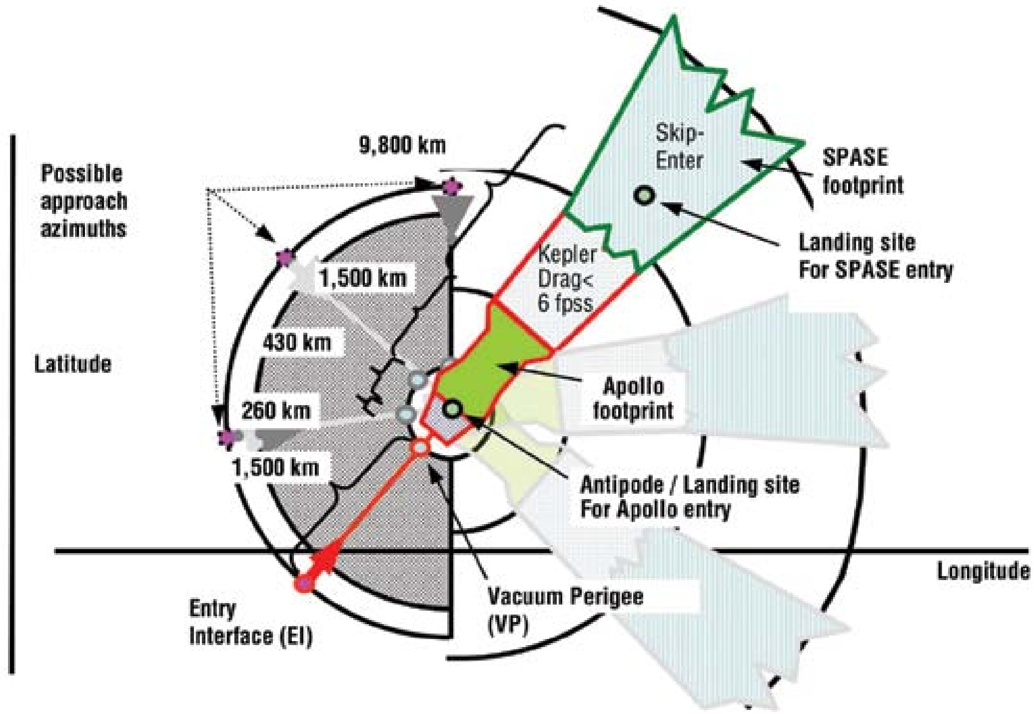


Figure 5-86. Entry Footprint with Co-azimuth Control (Direct-Entry and Skip-Entry), 0.3 L/D

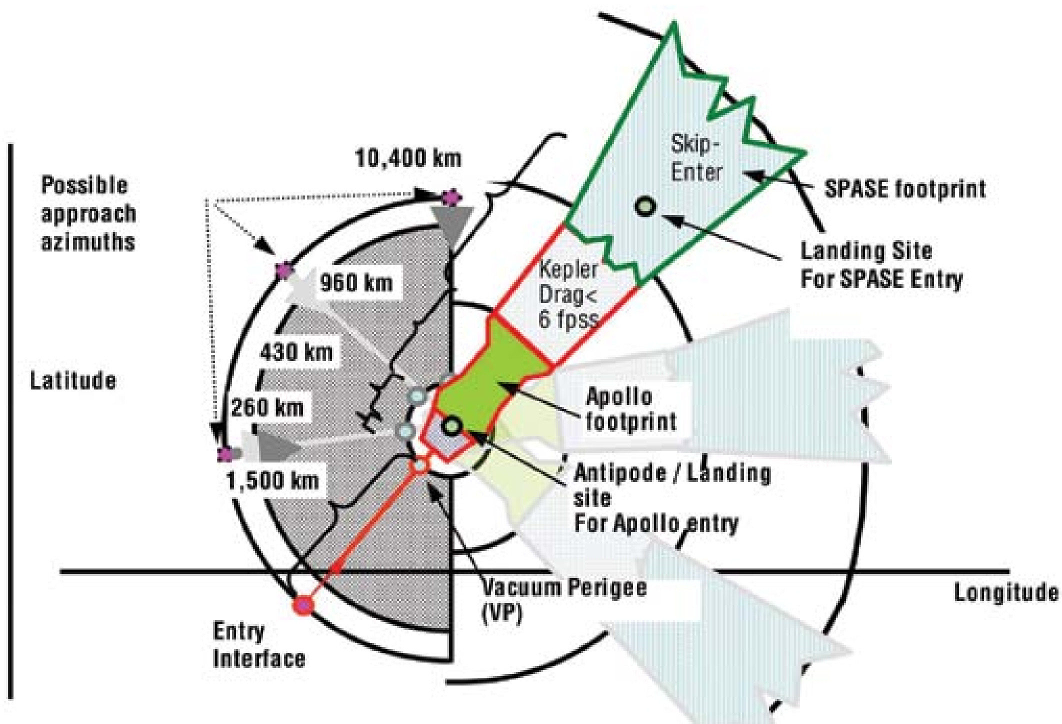


Figure 5-87. Entry Footprint with Co-azimuth Control (Direct-Entry and Skip-Entry), 0.4 L/D

The original Apollo guidance formulation provides for achieving long-range targets via a "Kepler" phase of guidance, which was exercised in only one test flight and never operationally flown due to concerns with controlling the up-control Kepler skip errors. **Figure 5-88** demonstrates that at least 9,200 km of range is required to achieve the Vandenberg landing site when the antipode is at maximum southerly location (–28.6 deg).

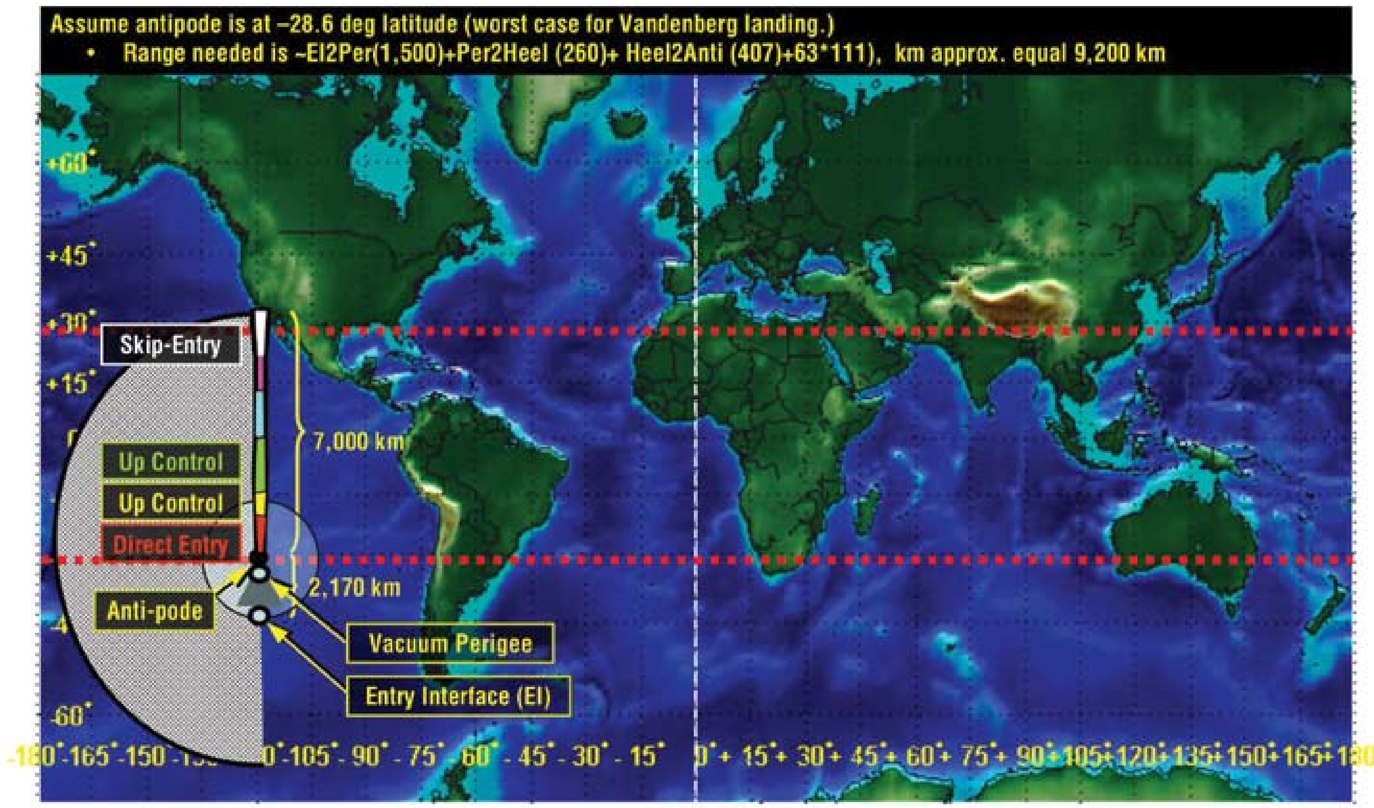


Figure 5-88. Flight Range Required to Reach Vandenberg AFB

Figures 5-89 and **5-90** provide the site accessibility of Vandenberg AFB for a 0.3 L/D capsule vehicle for different range flights. The current Apollo guidance provides an access circle of approximately 1,000 km, taking into account the loss of footprint due to the affect of ±12 hour flight time variation on the relative position of the landing site and the antipode. (Note that the original Apollo guidance capability would currently provide no access to Vandenberg AFB for maximum antipode in the ±18.3 deg cycle and less than 1 day for the ±28.6 deg cycle.) Each successive range circle increases the accessibility for the Vandenberg landing site until a range of 5,900 nmi for the ±18.3 deg cycle, or 9,300 nmi for the ±28.6 deg cycle, provides full-month coverage.

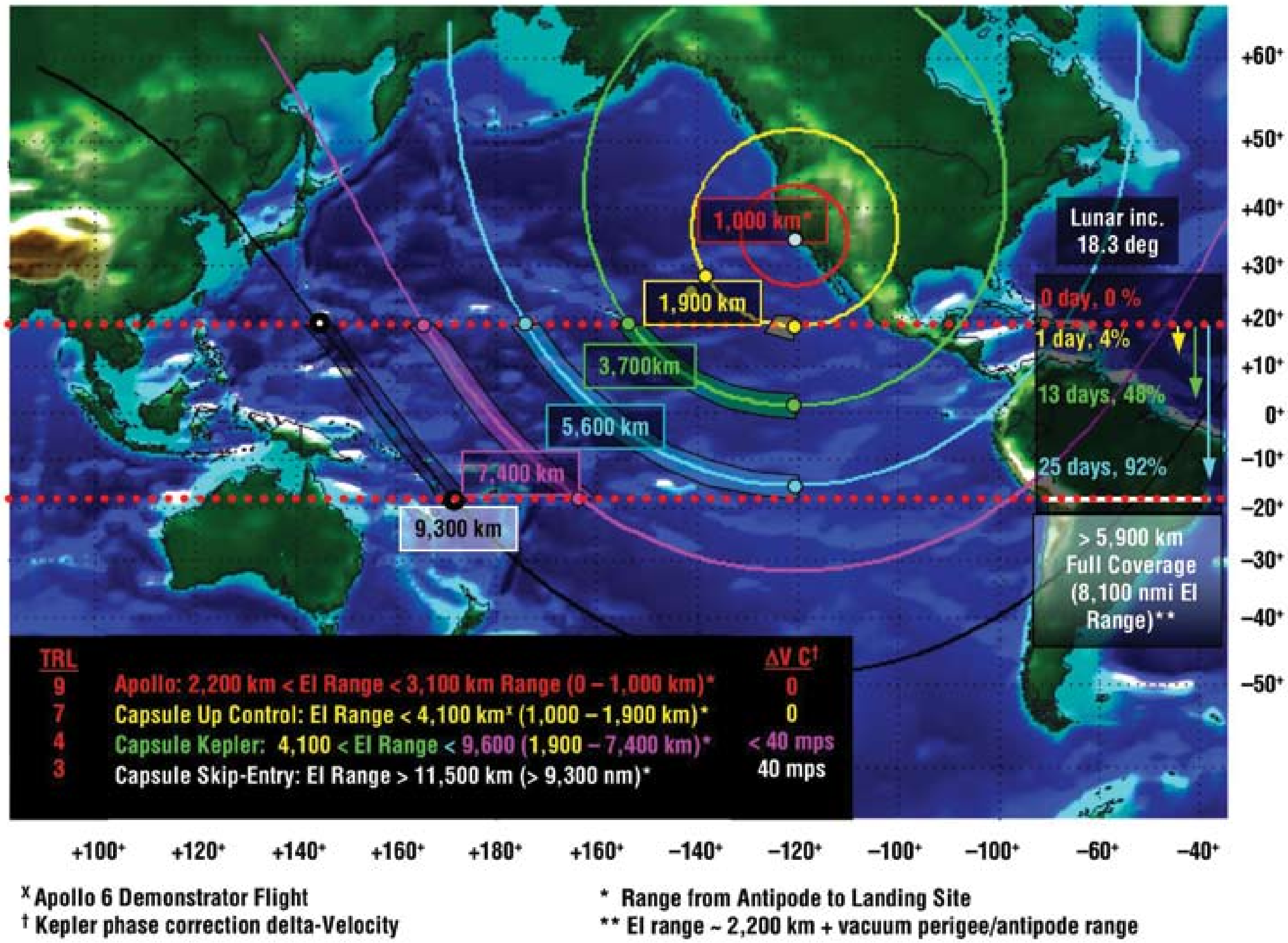


Figure 5-89. Vandenberg AFB Site Accessibility (0.3 L/D Capsule, ±18.3 deg Lunar Inclination

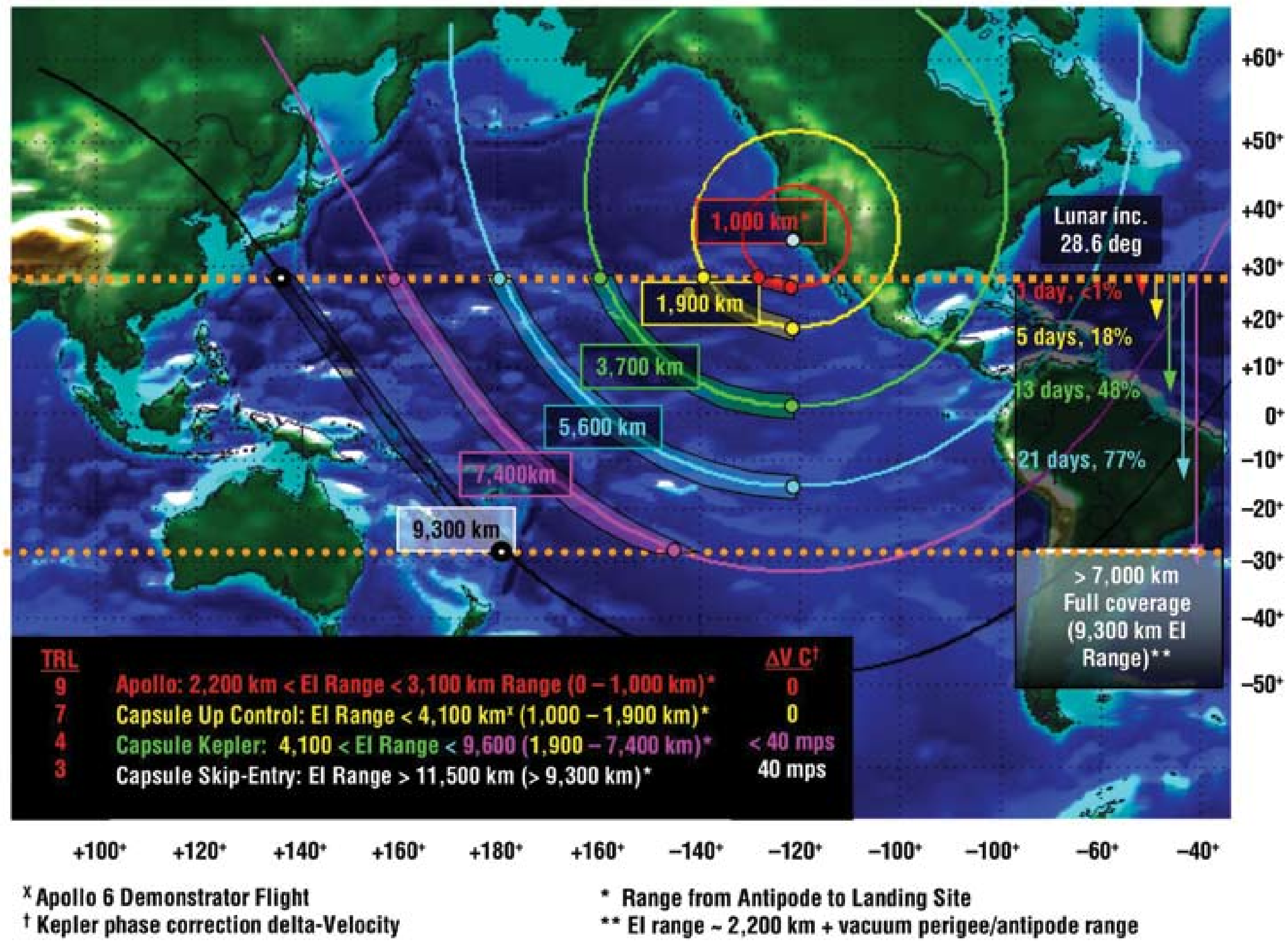
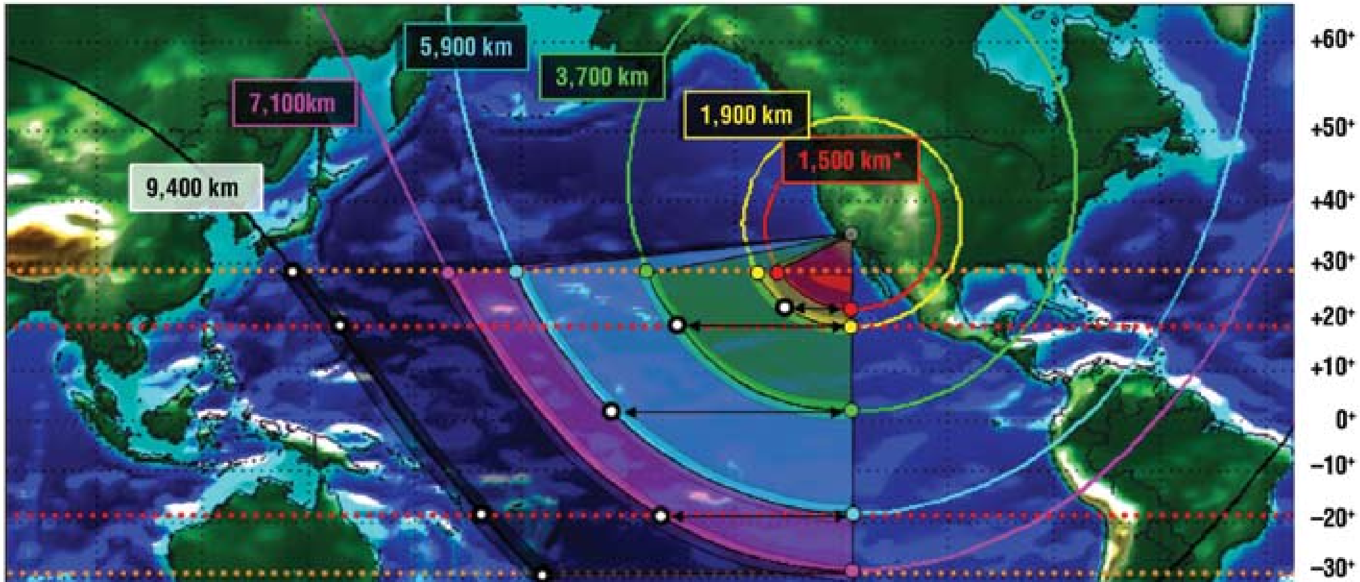


Figure 5-90. Vandenberg AFB Site Accessibility (0.3 L/D Capsule, ±28.6 deg Lunar Inclination)

Figure 5-91 provides Edwards AFB site access for a 0.4 L/D capsule vehicle. Even with the extended range provided by the increased L/D, Edward's site accessibility is not possible for the most northerly +18.3 deg antipode location using standard direct-entry Apollo guidance (1,500 km range).



Increased Guidance Complexity	Lunar Antipode (deg)	Entry Technique	EI Range (km)	Antipode Range (km)	Landing Site Access 18.3 deg incl.	(Days/month)** 28.6 deg incl.	ΔV C† (mps)
	28.6 to 21.6	Direct	3,300	1,100	0	2	0
	21.6 to 18.3	Up-Control#	4,100	1,900	<1	5	0
	18.3 to 1.6	- Kepler	5,900	3,700	12	13	0
	1.6 to -18.3	Kepler-Skip	8,100	5,900	Total	22	<40
	-18.3 to -28.6	Kepler-Skip	9,300	7,100	Total	Total	<40
	-28.6	Skip	>11,600	9,400	Total	Total	40

† Kepler phase correction delta-Velocity
# Significant positive altitude rate direct entry

* 0.4 L/D provides - 1,500 km Footprint (0.3 provided 960 km): J. Broome
** One sidereal month = 27.32 days

Figure 5-91. Edwards AFB Site Accessibility, 0.4 L/D Capsule (±28.6 deg, ±18.3 deg, Lunar Inclination

5.3.6 SM Propulsion Trades

A wide variety of SM propulsion trades were performed prior to selecting a LOX/methane pressure-fed system that has a high degree of commonality with the LSAM ascent stage. These trade studies and their results are presented in **Section 4**, **Lunar Architecture**.

5.3.7 Radiation Protection Trades

Detailed radiation analyses were performed on various CEV configurations to assess the need for supplemental radiation protection for lunar missions. These analyses and conclusions are presented in **Section 4**, **Lunar Architecture**.

5.4 Ascent Abort Analyses for the CEV

5.4.1 Summary

This analysis examines ascent aborts for a number of different CEV and LV combinations and focuses on total loss-of-thrust scenarios after jettison of the LAS. The general goal is to determine the abort options that might reasonably exist for various points in the ascent and characterize the CEV entry environment (e.g., in terms of loads and temperatures).

For the major portion of the analysis, the CEV is an Apollo-like capsule with a corresponding SM. SM delta velocities (ΔV's) are assessed from 330 to 1,732 m/s (1,083 to 5,682 fps) and Thrust-to-Weight (T/W) ratios from 0.38 to 0.17. Ascents to both 51.6 deg and 28.5 deg inclinations are considered.

The focus of the later portion of the analysis is on a Shuttle-derived LV: a four-segment SRB with a single SSME upper stage, LV 13.1. The sensitivity of abort coverage and abort mode boundaries to variations in available ΔV and T/W are key factors that received appropriate emphasis. Other important factors include the minimum operating altitude of the thrusting CM and SM (i.e., can they safely perform in the 335-kft altitude region where the effects of aeroheating cannot be ignored?) and the ignition delay of the propulsion system (i.e., how quickly can the CM/SM separate and maneuver to burn attitude?). These factors are particularly important for LV 13.1, because the ascent trajectory is quite depressed. Abort coverage will not be good if, for example, the CM/SM cannot perform safely below approximately 340 kft, has a T/W of less than 0.2, cannot ignite the propulsion system fairly quickly, and is launched on a very depressed ascent trajectory. This analysis tries to quantify the effects of all of these factors.

The analysis sought to define near-optimal abort coverage by using numerically optimized pitch profiles during thrusting phases. The intent was to try to avoid limitations that available guidance algorithms might impose. New guidance algorithms may well be needed to automatically target and fly some of the abort trajectories from this analysis.

The results indicate a fairly robust abort capability for LV 13.1 and a 51.6 deg mission, given 1,200 m/s of ΔV, a T/W of at least 0.25, a CM/SM minimum operating altitude of 335 kft, and the ability to initiate propulsion system burns in about one-third the time budgeted for Apollo. (Apollo budgeted 90 sec to initiate posigrade burns and 125 sec for retrograde burns.) Abort landings in the mid-North Atlantic can be avoided by either an ATO or posigrade TAL south of Ireland. Landings in the Middle East, the Alps, or elsewhere in Europe can be avoided by either an ATO or a retrograde TAL south of Ireland. At 28.5 deg, landings in Africa can be avoided by either an ATO or a retrograde TAL to the area between the Cape Verde islands and Africa. However, it appears that even with 1,732 m/s of ΔV, some abort landings could occur fairly distant from land. However, once the ballistic impact point crosses roughly 50°W longitude, posigrade burns can move the abort landing area downrange near the Cape Verde islands.

The next section will briefly introduce some of the various abort modes, including a summary of the Apollo abort modes. Key assumptions will also be discussed. Subsequent sections will then review the detailed results, beginning with the Shuttle-derived boosters, followed by the Evolved Expendable Launch Vehicles (EELVs). Lastly, results for two different lifting bodies will be reviewed that address mostly abort loads and surface temperatures. Some results from earlier analyses are also presented to illustrate the effect of dispersions and other operational considerations.

5.4.2 Introduction

The Apollo literature on ascent aborts has quite proven useful to these studies. **Figure 5-92** presents a summary of the abort modes for Apollo 11. Four abort modes are identified. Mode I covers aborts using the LAS. Mode II aborts are simple, unguided lift-up entries, terminated when the landing area begins to impinge on Africa. Mode III uses lift reduction and retrograde thrust to land short of Africa. Mode IV is a contingency orbit insertion (or ATO in Shuttle jargon). A large ATO capability exists, especially with use of the S-IVB stage. Interestingly, the abort plan did not include use of posigrade thrust to target some aborts off Africa. For this CEV analysis, use of posigrade thrust is considered for suborbital abort modes like TAL.

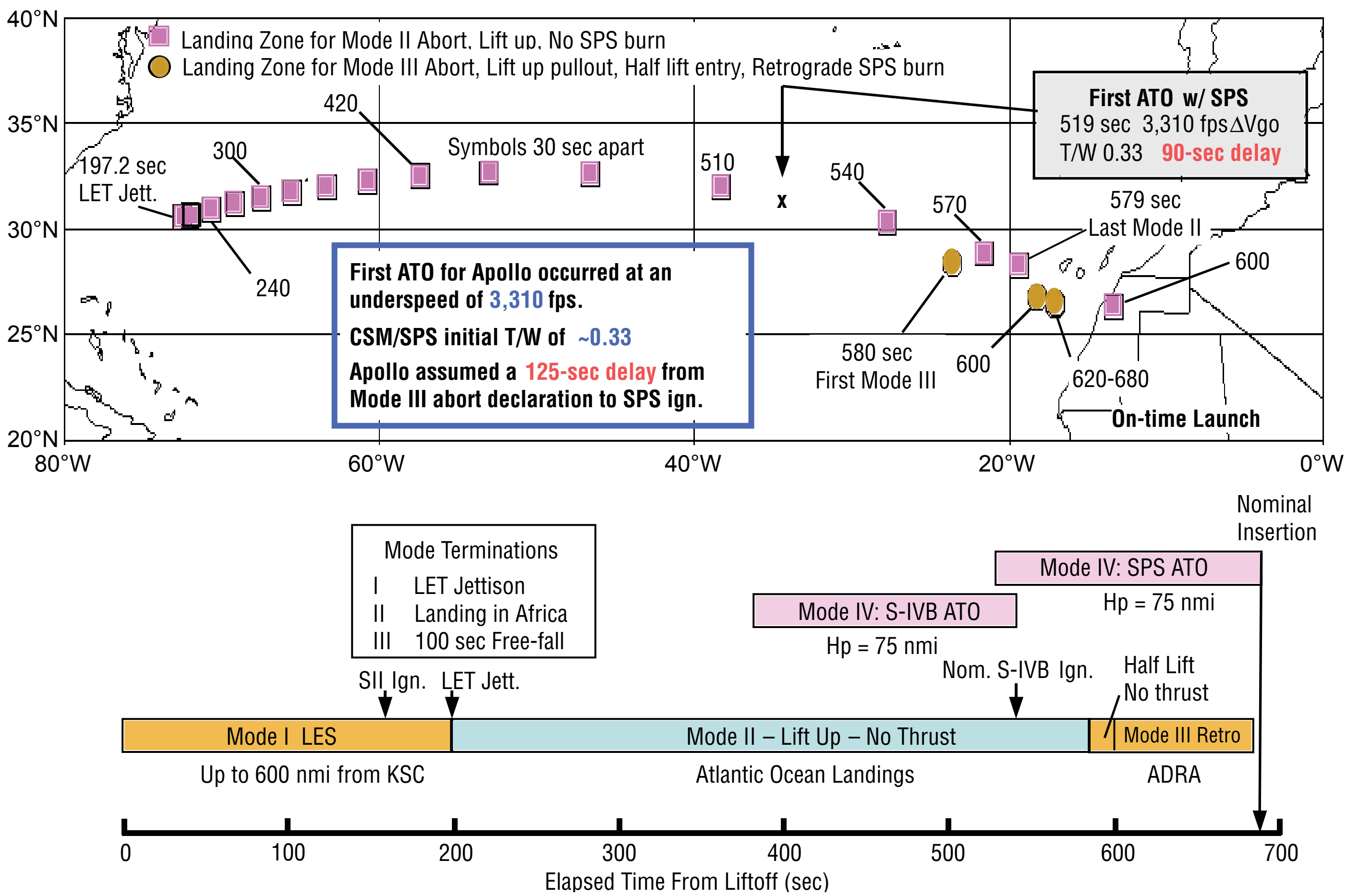


Figure 5-92. Apollo 11 Abort Modes

A key parameter in the Apollo ascent abort analyses is "free-fall time" to 300 kft altitude. For instance, Mode II aborts require 100 sec of free-fall time from abort declaration to 300 kft altitude on the abort-entry trajectory. This amount of time is budgeted to terminate thrust on the LV, separate the CM/SM from the stack, separate the CM from the SM, and then orient the CM for entry. Likewise, Mode III aborts require 100 sec of free-fall time from termination of the Service Propulsion System (SPS) burn to 300 kft on the abort-entry trajectory. While no specific free-fall time requirement has been established for the CEV, the parameter has been included in the analyses. It is a useful parameter for assessing the reasonableness of abort scenario timelines from ascent trajectories with varying amounts of loft.

Figure 5-92 identifies other guidelines for abort time lines. Ninety seconds are budgeted for startup of the SPS engine for Mode IV aborts (ATO). One-hundred-twenty seconds are budgeted for startup of the Mode III retrograde burns. This CEV study took the approach of initially using a much more aggressive time line (20 sec for SPS startup) and assessing the sensitivity to larger delays.

5.4.3 Assumptions and Methodology

Key assumptions are made relative to aerodynamics and the estimation of surface temperatures. Where possible, the Apollo aerodynamic database is used. For a capsule with 0.3 L/D, the Apollo angle of attack (α) versus Mach tables for the Command Module are used with an angle-of-attack of 160 deg. For ATO studies, the free-molecular coefficients for the Command and Service Module are used. For vehicles with an L/D other than 0.3, aerodynamics are typically modeled with a coefficient of L/D and the given reference area.

Figure 5-93 presents the methodology for estimating surface temperature. The approach has provided reasonable surface temperature estimates for preliminary assessment purposes. The results are evaluated relative to the single mission limit for TPS materials developed for the Shuttle and X–38 (i.e., 3,200–3,300°F for the C/SiC-coated Reinforced Carbon-Carbon (RCC) developed for the X–38.

The 1976 Standard Atmosphere is used, with no winds.

SORT is used to define the trajectories for the analyses. SORT is a versatile 3-DOF simulation tool that is controlled by the Aerosciences and Flight Mechanics Division at NASA's JSC.

Heat rates use the modified Detra-Kemp-Riddell (DKR) formulation embedded within a Newton-Rhapson temperature convergence.

T_{surf} is the estimated surface temperature in degrees Rankine.

Modified DKR Heat Rate for an Equilibrium Wall Temperature

$$\dot{Q} = \frac{17600}{\sqrt{R_N}} \sqrt{\frac{\rho}{\rho_o}} \left(\frac{V_{fs}}{26000} \right)^{3.25} \left(\frac{H_{stag} - H_{wall}}{H_{stag} - 129.6} \right)$$

V_{fs} = freestream velocity (fps)

ρ = atmospheric density (slugs/ft^3)

$H_{wall} = 0.24 * T_{Surf}$

$H_{Stag} = \left(V_{fs}\right)^2 / 50073.12$

Radiation Equilibrium Temperature: T_{surf}

$$T_{Surf} = \left(\frac{\dot{Q}}{0.85 * S_{bc}} \right)^{\frac{1}{4}}$$

$S_{bc} = 4.75833 \times 10^{-13}$

Emissivity = 0.85

Effective nose radius (R_N) for capsule derived from "ACRV Landing Accuracy Study" NASA Memo EG3/9104-24, 4/4/91: R_N = 8 ft for a 14 ft diameter Apollo capsule

For this study, R_N (in ft) = 8 x capsule diameter (in ft) / 14

Figure 5-93. Methodology for Estimating Surface Temperature

ATOs are defined using two burns. The pitch profile to raise apogee consists of a constant segment followed by a linear segment. To raise perigee, a single linear segment is used. A SORT optimizer is used to define profiles for minimal ΔV. The optimizer defines the value for the constant segment, the transition time to the linear segment, the slope and length of the linear segments, and the coast period between burns. ATOs are targeted to a 100- x 100-nmi circular orbit to provide a 24-hour orbital life.

5.4.4 Shuttle-Derived Vehicles (SDVs)

Ascent aborts are analyzed for several different Shuttle-Derived Vehicles (SDVs): in-line crew vehicles with four- and five-segment SRBs (LV 13.1 and 15, respectively) and an in-line crew/cargo vehicle with five-segment SRBs and four SSMEs on the tank (LV 26). The LV numbers correspond to those defined in the LV data summary. These numbers are typically included on the figures to aid booster identification (usually contained in parentheses).

Ascent trajectories for the three boosters are presented in **Figure 5-94**. Subsequent sections will first address the loads, estimated surface temperature, free-fall time characteristics, and impact points for "Mode II" aborts from the various boosters. This will be followed by a discussion of preliminary abort mode boundaries and the sensitivities to T/W and other factors.

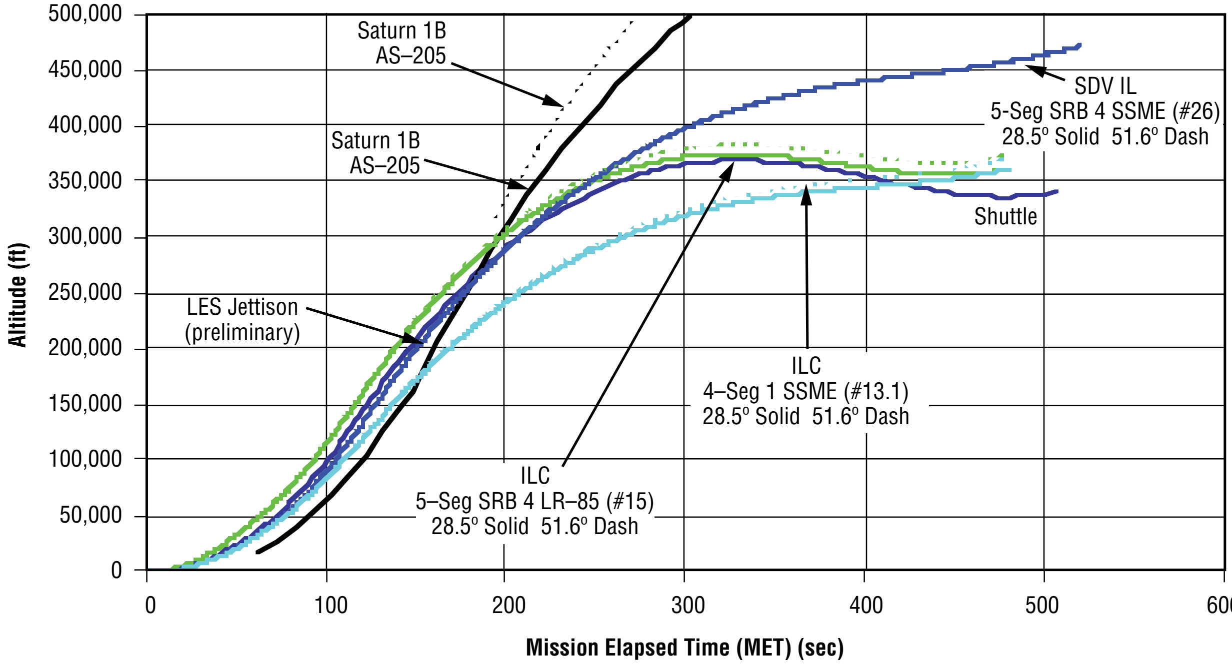


Figure 5-94. Ascent Trajectories for SDVs

5.4.4.1 Loads, Surface Temperature, Free-fall Time, and Impact Points

Peak loads, estimated maximum surface temperatures, and free-fall time are presented for ballistic (i.e., nullified lift) and lift-up aborts in **Figures 5-95** to **5-99**. Data from the Apollo Program are included for ballistic and lift-up abort loads and free-fall time for comparison. Data for CEV aborts from a representative Shuttle trajectory are also included.

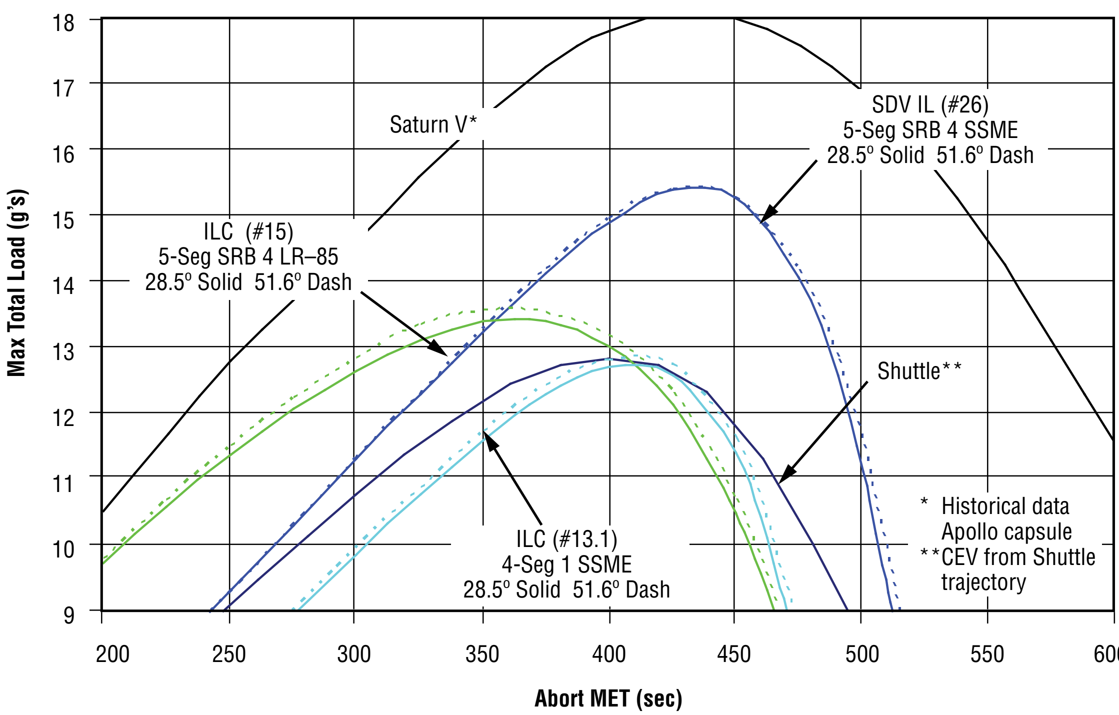


Figure 5-95. Ballistic Loads for Aborts from Shuttle-Derived Boosters

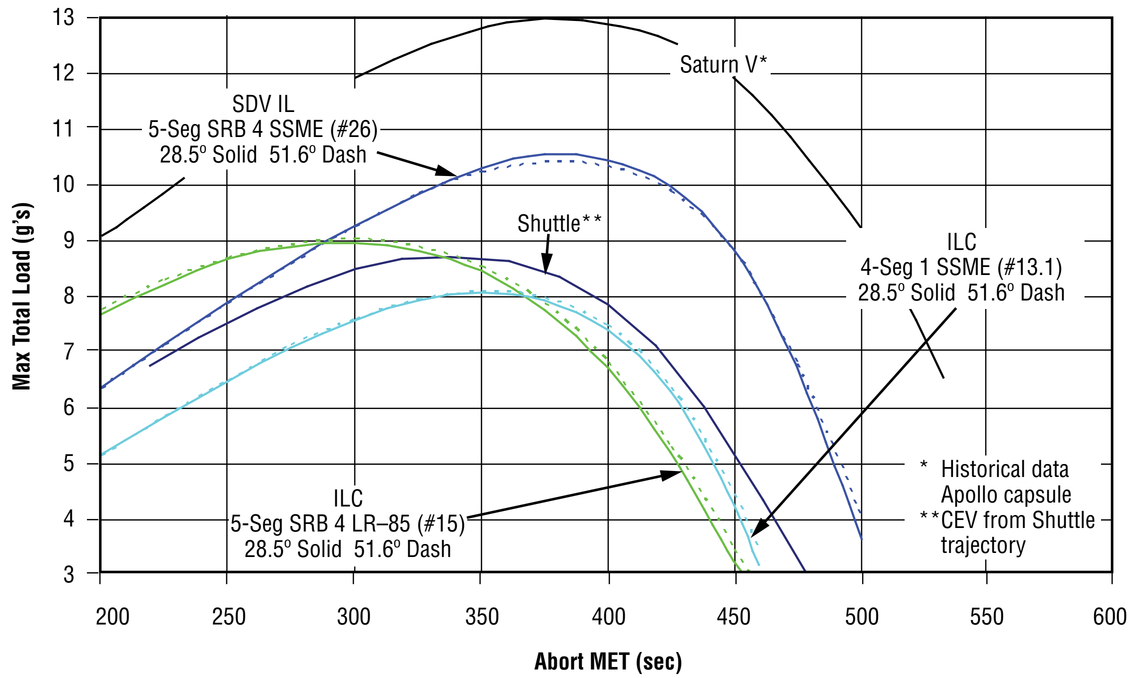


Figure 5-96. Lift-up Loads for Aborts from SDVs

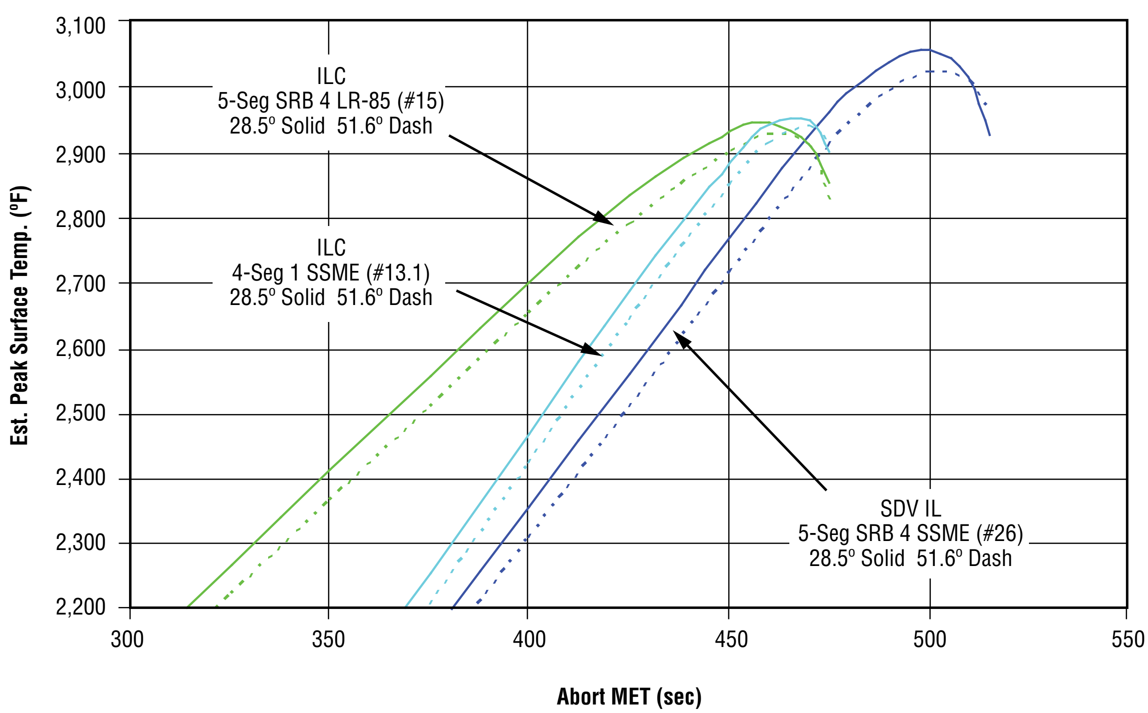


Figure 5-97. Maximum Surface Temperatures for Ballistic Aborts from Shuttle-Derived Boosters

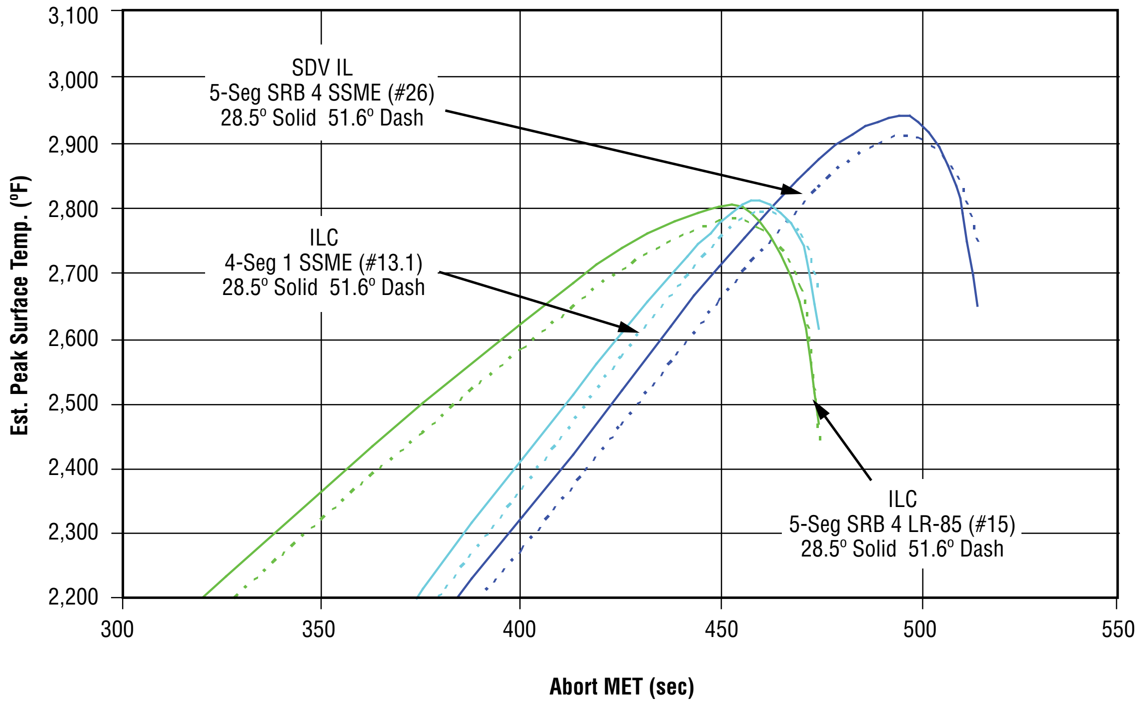


Figure 5-98. Maximum Surface Temperatures for Lift-up Aborts from Shuttle Derived Vehicles

Data at 28.5 deg are for the lunar CEV (Block 2) with an L/D of 0.3 and a ballistic number of 81 psf. Data at 51.6 deg are for the ISS CEV (Block 1) with an L/D of 0.3 and a ballistic number of 67 psf. (Note that, in general, the abort parameters for the depressed LV 13.1 trajectory are lower than for the other, more lofted ascent trajectories). The lower loads and temperatures are obviously a benefit. However, since the LAS most likely will not be available after approximately 240 sec (and possibly earlier), the limited amount of free-fall time before encountering the atmosphere could be an issue. Free-fall times near 50 sec indicate abort scenarios that probably deserve more attention.

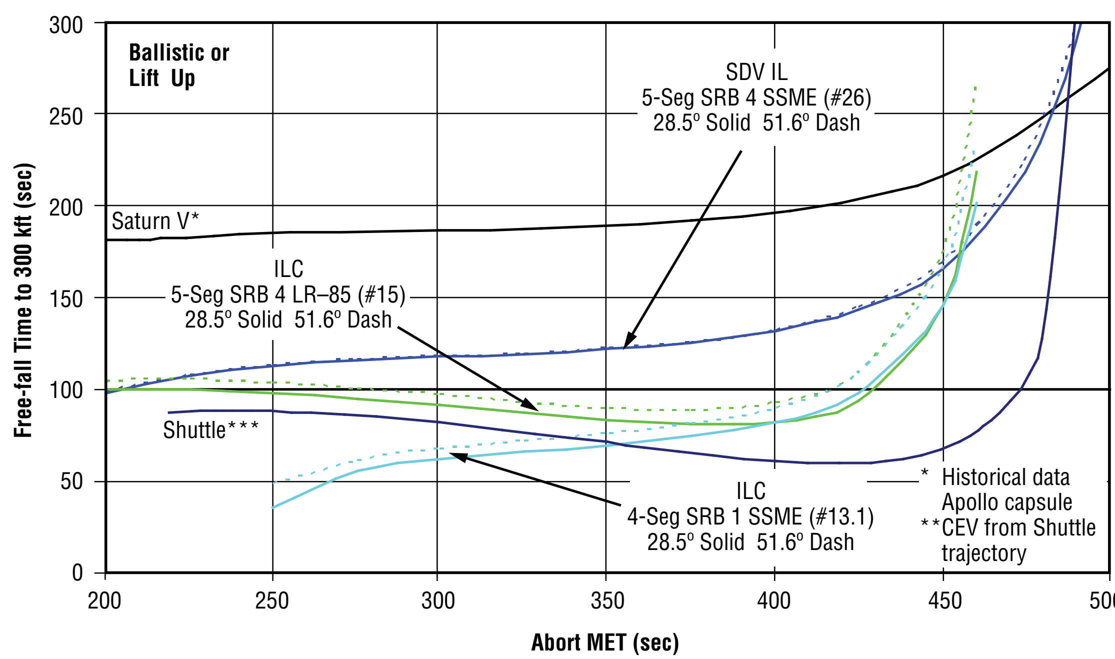


Figure 5-99. Free-fall Time for Aborts from SDVs

Load durations for the worst-case ballistic aborts are well within the crew limits for escape. (Note the duration histories relative to the red line on **Figure 5-100**.) Estimated maximum surface temperatures are within the single mission limits for TPS materials developed for the Shuttle and X–38. However, higher fidelity aeroheating analyses are needed to confirm this data.

Alternate capsule designs evolved during the analysis. **Figure 5-101** compares abort loads and estimated surface temperatures for the Cycle 1 and Cycle 2 CEV CMs and LV 13.1. The Cycle 2 capsule has a higher ballistic number (87 psf versus 67 psf for the ISS versions). This causes the ballistic loads and surface temperatures to increase slightly. (The ballistic loads are also driven up slightly by the increase in L/D from 0.3 to 0.4.) For the lift-up aborts, the increased L/D helps the loads and appears to almost cancel the effect of the increased ballistic number on the temperatures for the lift-up aborts. **Figure 5-102** indicates that the load durations for the worst-case ballistic aborts are well below the crew limits.

Ballistic impact points for the Cycle 1 capsules (ballistic numbers of 67 and 81 psf) are presented in **Figure 5-103.** The high T/W ratios (for second stage) limit North Atlantic abort landings to approximately 3–5 percent of the ascent trajectory. Powered abort options (discussed below) were also examined to totally avoid the North Atlantic and other undesirable landing areas along the 51.6 deg inclination ground track.

It is worth noting the ATO times on **Figure 5-103** for 28.5 deg. The first ATO of LV 15 has a significantly lower "under speed" (i.e., the velocity magnitude short of the nominal engine cutoff velocity). Although this LV was not carried forward in the later analyses, it is worth noting the impact on ATO of the negative altitude rate during the later portion of the trajectory. (Note that a minimum operating altitude of 345 kft was used for this comparison; it is difficult to meet this limit with a trajectory shaped like the one for LV 15.) A higher Second-Stage Engine Cutoff (SECO) altitude will bring ATO performance for LV 15 closer to that of LV 13.1.

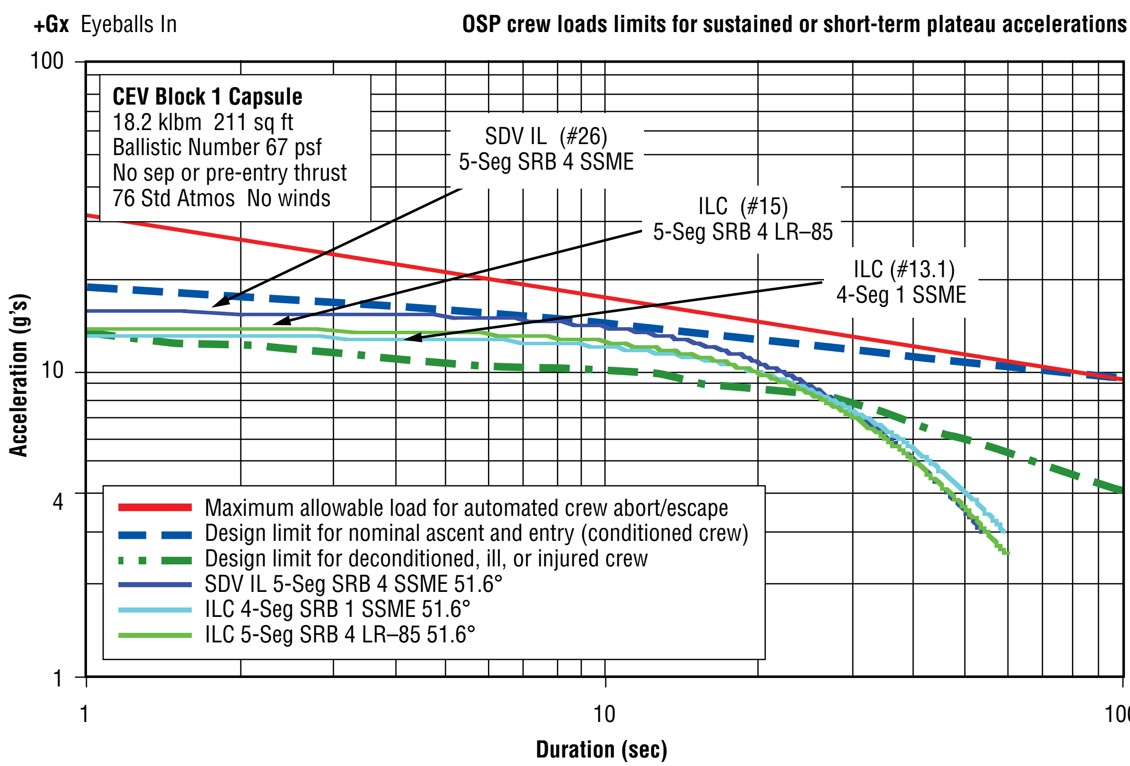


Figure 5-100. Worst Case Ballistic Load Durations versus Crew Limit Lines

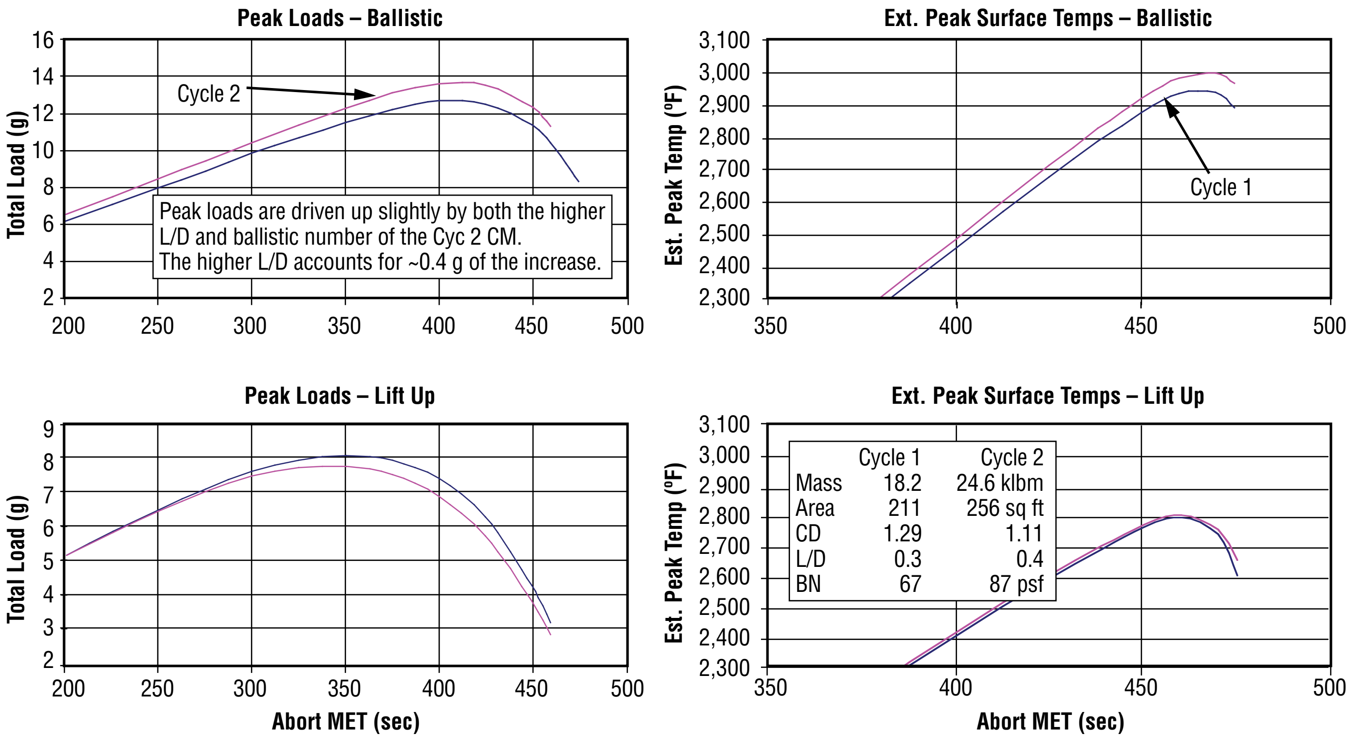


Figure 5-101. Comparison of Aborts for Cycle 1 and 2 ISS CEV and LV 13.1

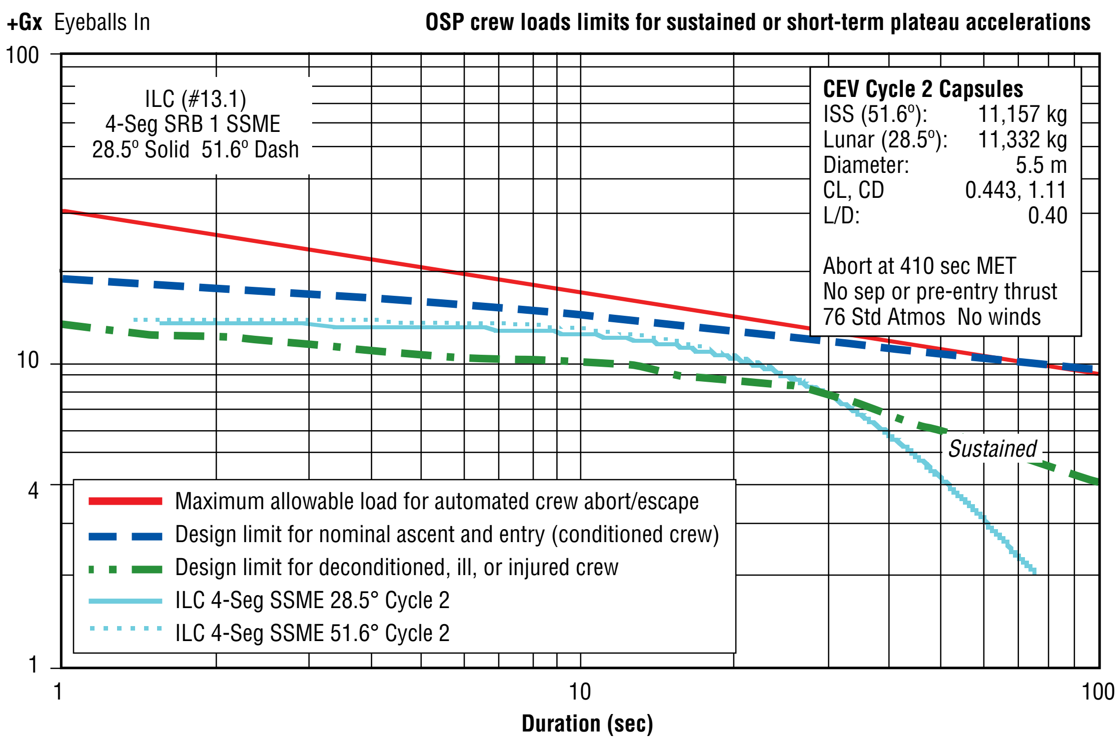
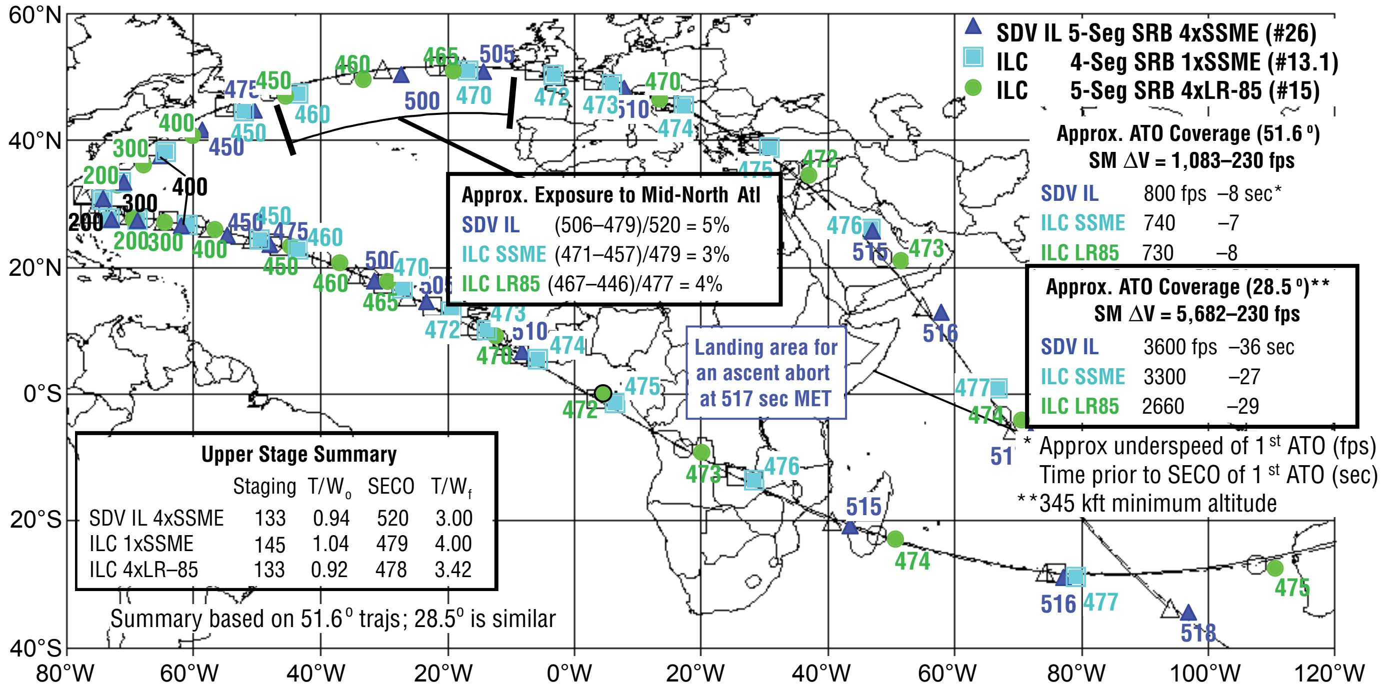


Figure 5-102. Worst Case Ballistic Load Durations for Cycle 2 CEV and LV 13.1

5.4.4.2 Abort Mode Assessments for 51.6-deg Inclination

5.4.4.2.1 Abort Modes for Two Different Propulsion System Configurations

Abort modes were initially assessed for an ISS CEV (Block 1) with the baseline and an alternate propulsion system delta-V and thrust: 330 m/s and 44.5 kN, and 1,200 m/s and 66.75 m/s (1,083 fps and 10 klbf, and 3,937 fps and 15 klbf). Abort modes for LV 13.1 were assessed for both propulsion system configurations, while LV 26 only used the baseline configuration. This latter study was undertaken to understand the effects of the depressed ascent trajectory for LV 13.1. The effect of various T/W levels and propulsion system ignition delays was briefly studied for LV 13.1.



Landing areas are indicated for ballistic abort entries with no separation or post-abort thrust

Figure 5-103. Ballistic Landing Areas for SDVs

Figure 5-104 presents the results for LV 13.1 and the baseline propulsion system configuration. The turquoise symbols and time tags indicate landing areas for no thrust-and-lift (i.e., a ballistic entry). The first ATO occurs at 472 sec—corresponding to the ballistic landing symbol near Ireland and England—indicating that the ATO abort mode avoids abort landings in the Alps and Middle East. The red symbols and time tags indicate landing areas for a retrograde burn that minimizes downrange, combined with a "half lift" entry at a 60-deg bank. The landing areas are shifted well to the left when all available ΔV is used. The implication is that for METs between 472 sec and SECO (479 sec), retrograde burns of a lesser magnitude can target a landing area south of Ireland (in a manner similar to the way Apollo targeted a landing area near the Canary Islands for Mode III aborts). This provides another potential abort mode for avoiding the Alps and the Middle East, but will require a more thorough examination since the free-fall time is only approximately 50 sec for the 472-sec abort and an aggressive-maneuver time line is used for the retrograde burn. The green symbols and time tags indicate landing areas for a posigrade burn that maximizes downrange, combined with a full-lift entry. The landing area for a 462-sec abort is in northern France. If the retrograde burn-abort mode were available at 462 sec (note the red square with a landing area near Newfoundland), landings in the middle of the North Atlantic could be avoided by landing on either side of the Atlantic. However, a very short free-fall time after the burn (17 sec) does not make this abort appear practical.

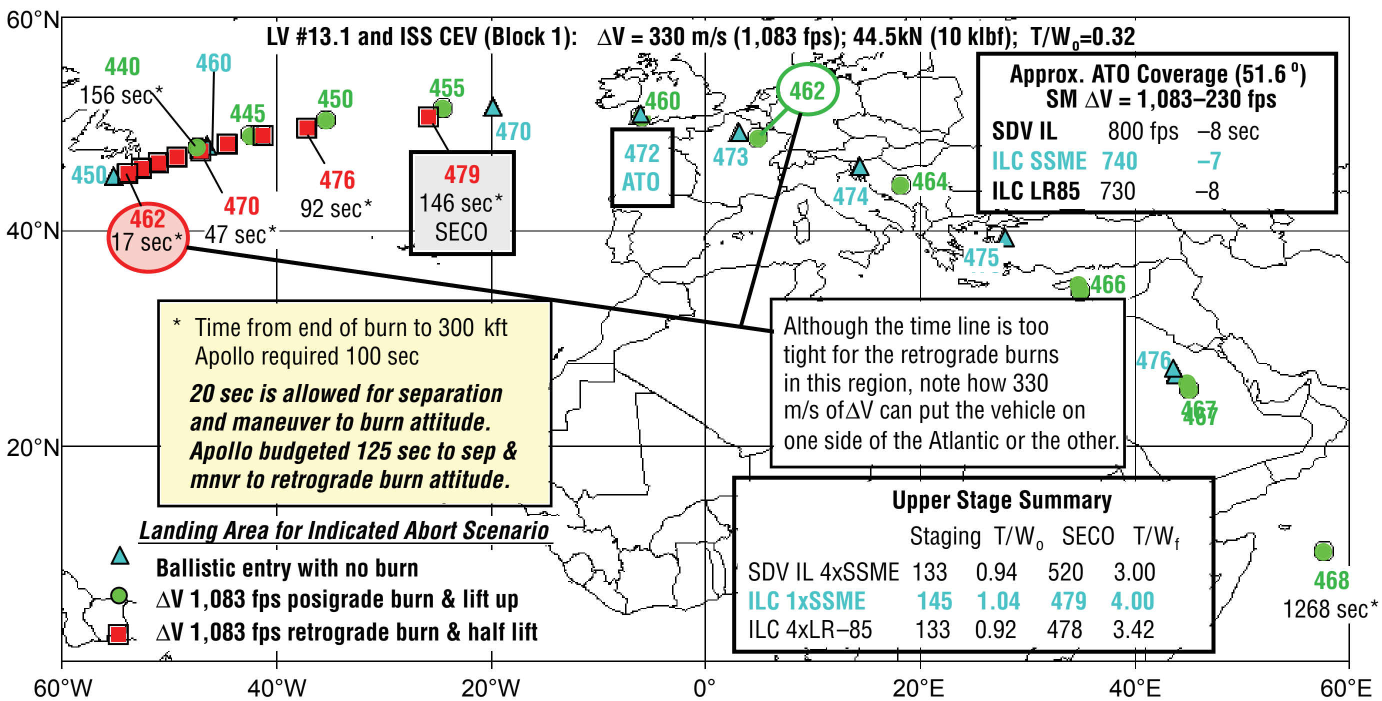


Figure 5-104. Effect of 330 m/s on Landing Areas for LV 13.1

A similar analysis was performed for LV 26 to assess the effect of a more lofted ascent trajectory (**Figure 5-105**). (Note how landings are possible on either side of the North Atlantic for aborts at 490–491 sec). For the more lofted ascent, the 490-sec retrograde abort has 69 sec of free-fall time from the end of the burn. Given a CEV with a robust RCS that allows a quick separation and maneuver to retrograde burn attitude, this abort mode may be feasible. Another important observation is that the first ATO does not provide protection from landing in the Alps; the first ATO is at 512 sec, which corresponds to the ballistic landing area in Bosnia. This is not due to the lofted ascent trajectory, but rather due to the 3-g maximum acceleration for LV 26 (versus 4-g limit for LV 13.1).

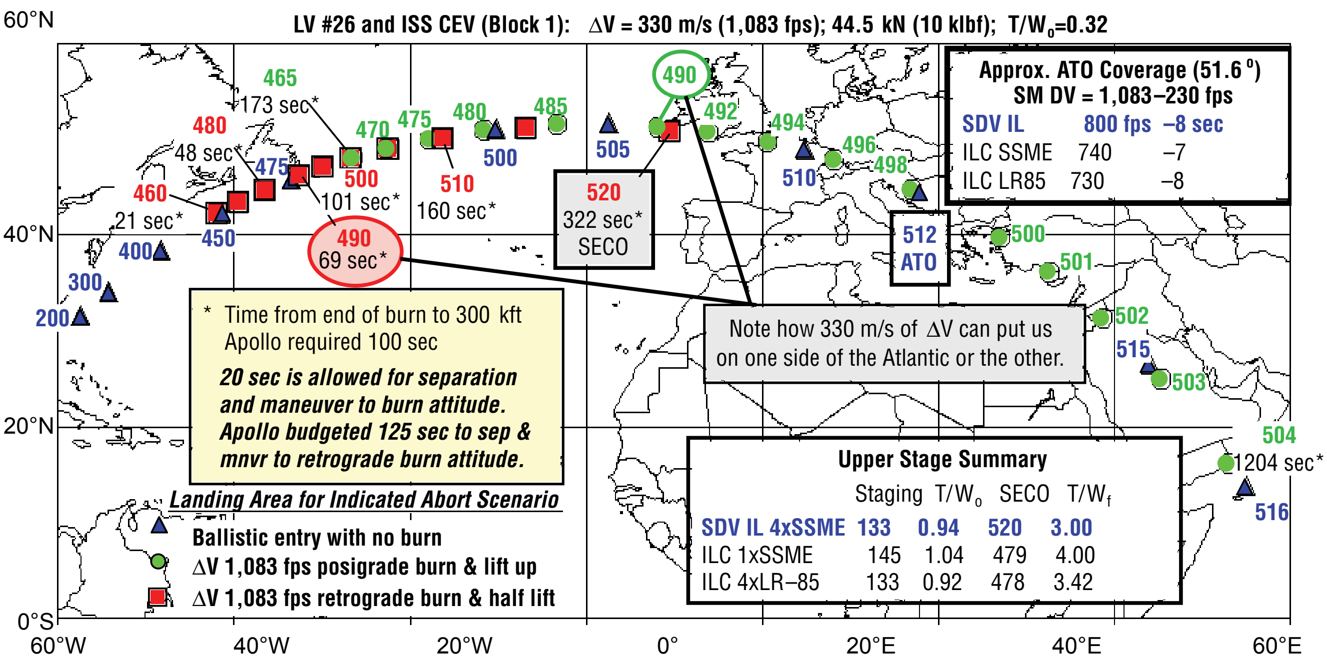
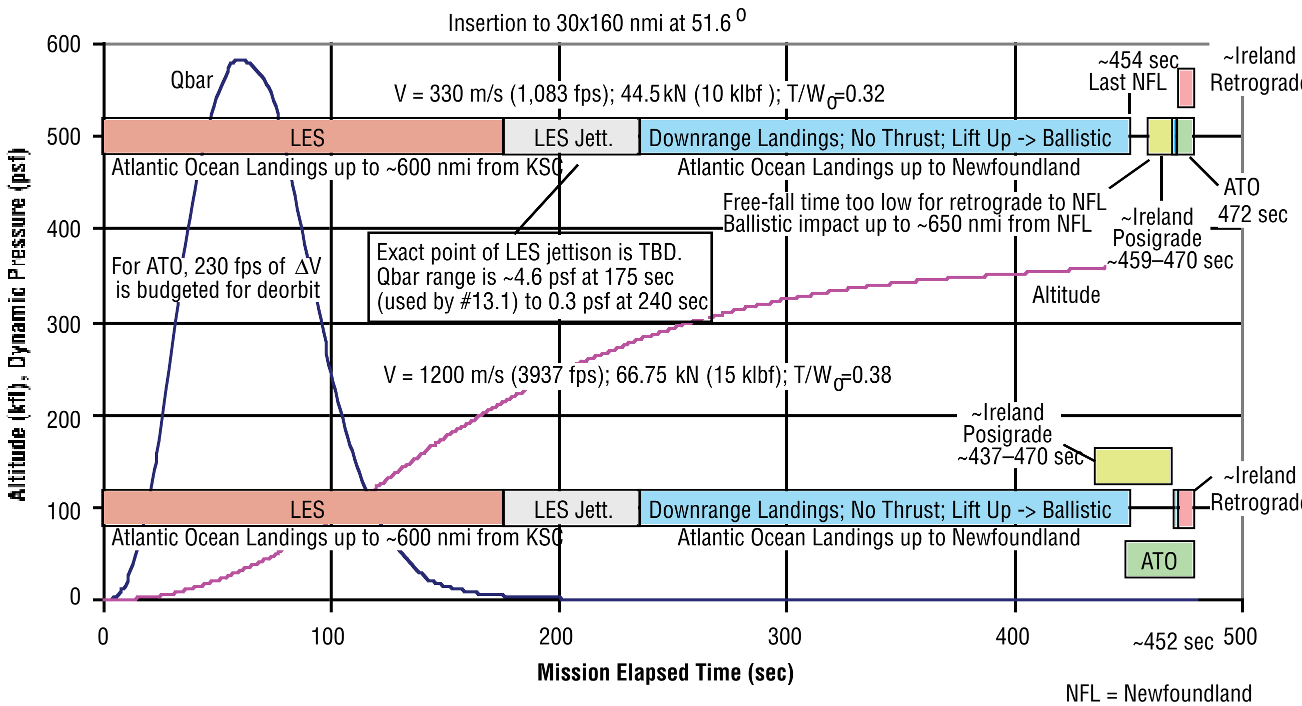


Figure 5-105. Effect of 330 m/s on Landing Areas for LV 26

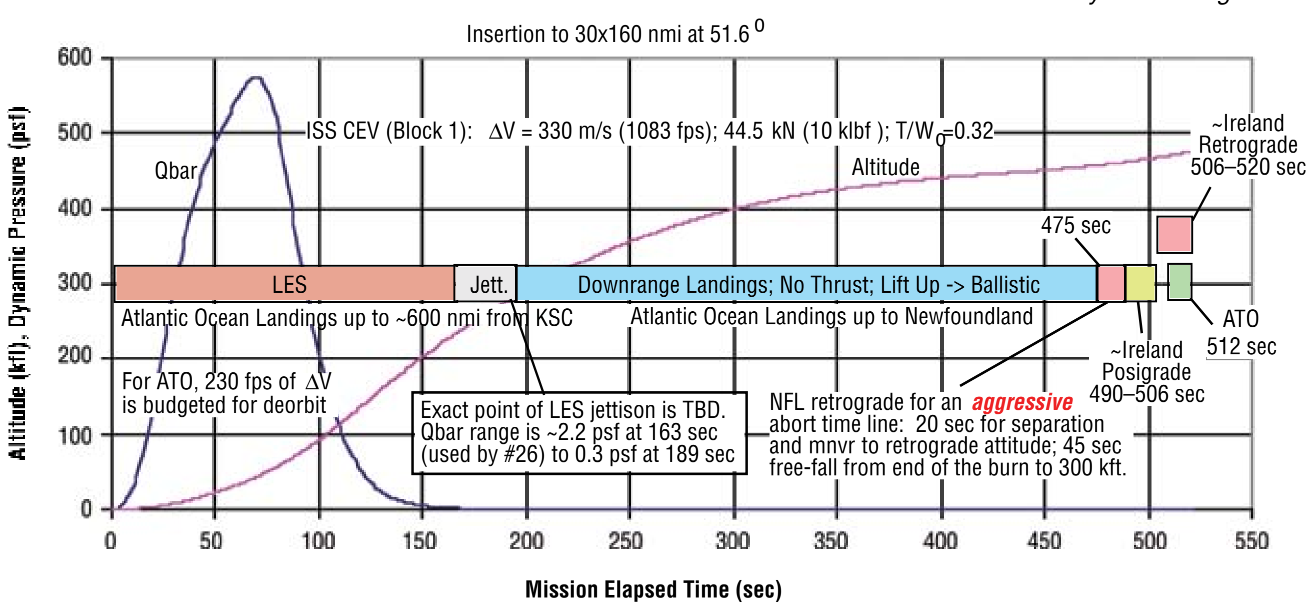
The results of these two analyses are summarized in the top half of **Figure 5-106** and **5-107**. The conclusion is that, with a more lofted ascent trajectory, the North Atlantic and other undesirable landing areas can be avoided with a limited amount of ΔV, if the CEV RCS is robust enough to separate and maneuver to burn attitude quickly and if the CEV propulsion system can ignite quickly.

The bottom half of **Figure 5-106** summarizes the LV 13.1 abort modes for the alternate propulsion system configuration. To summarize briefly, this configuration provides two abort modes for avoiding a landing in the middle of the North Atlantic, in the Alps, or in the Middle East: ATO and a posigrade TAL, or ATO and a retrograde TAL, respectively.



NFL = Newfoundland
Nom. insertion: 479.3 sec

Figure 5-106. Abort Modes for Launch Vehicle 13.1 with the ISS CEV and Two Propulsion System Configurations



NFL = Newfoundland
Nominal insertion: 520 sec

Figure 5-107. Abort Modes for LV 26 and the ISS CEV with Limited OMS Propellant

5.4.4.2.2 Abort Mode Sensitivities to T/W and Propulsion System Ignition Delay

Figure 5-108 presents the sensitivity of TAL and ATO opportunities to variations in T/W for aborts from LV 13.1. The study assumes that 200 m/s (3937 fps) of propulsion system ΔV is available. (For ATO, 70 m/s is reserved for deorbit.) The horizontal limit line at approximately 453.5 sec indicates the point in the ascent when the distance from Newfoundland to the ballistic landing area begins to increase. The limit line is meant to provide a rough indication of the T/W required to avoid the middle of the North Atlantic with either an ATO (T/W ≈ 0.26) or a TAL (T/W ≈ 0.16). For T/Ws below approximately 0.21, selection of the first TAL time begins to be driven by having enough free-fall time from the end of the burn to the beginning of atmospheric entry. This study assumes the Apollo guideline of 100 sec of free-fall to 300 kft. Also, maintaining altitude above the assumed minimum operating altitude of the thrusting CM/SM (335 kft) is very important at these T/W levels. For ATO, the thrust pitch angle must be increased to maintain altitude, introducing a "steering loss" to the velocity gain. This effect is more apparent in **Figure 5-109**; as T/W drops below approximately 0.25, the rate of loss of ATO coverage begins to accelerate.

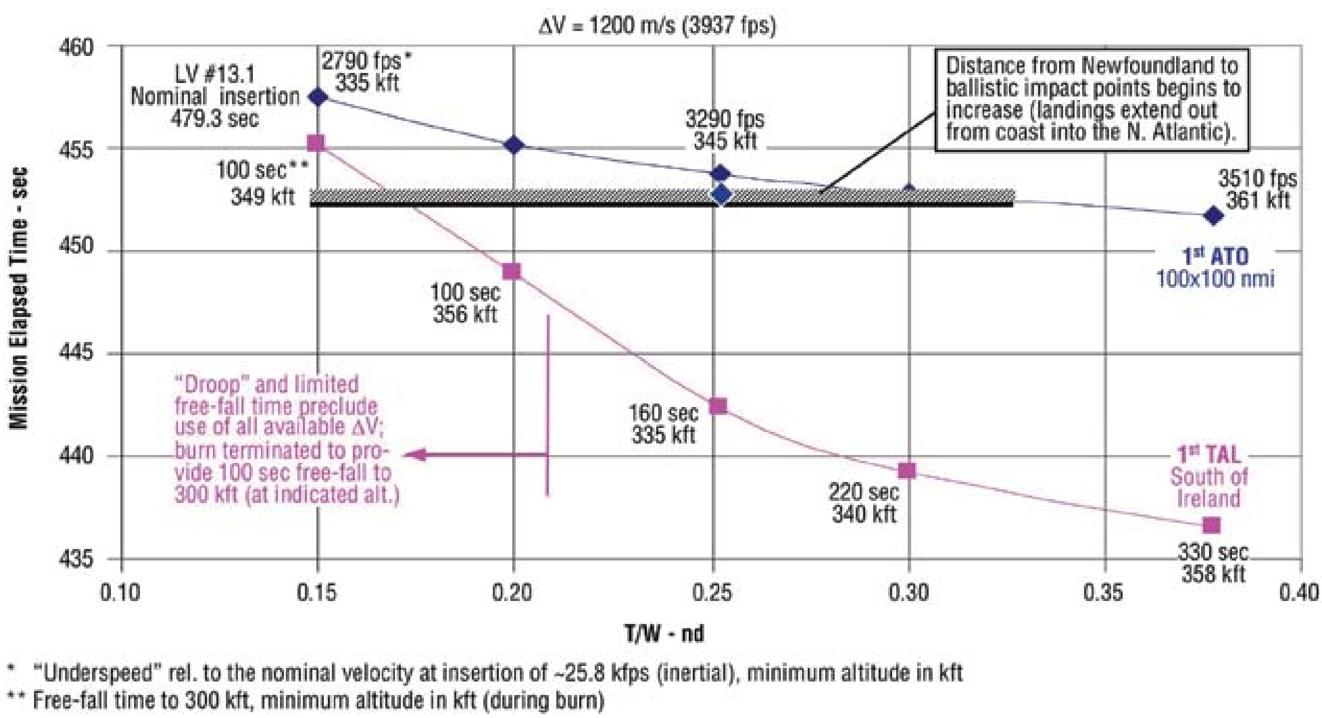
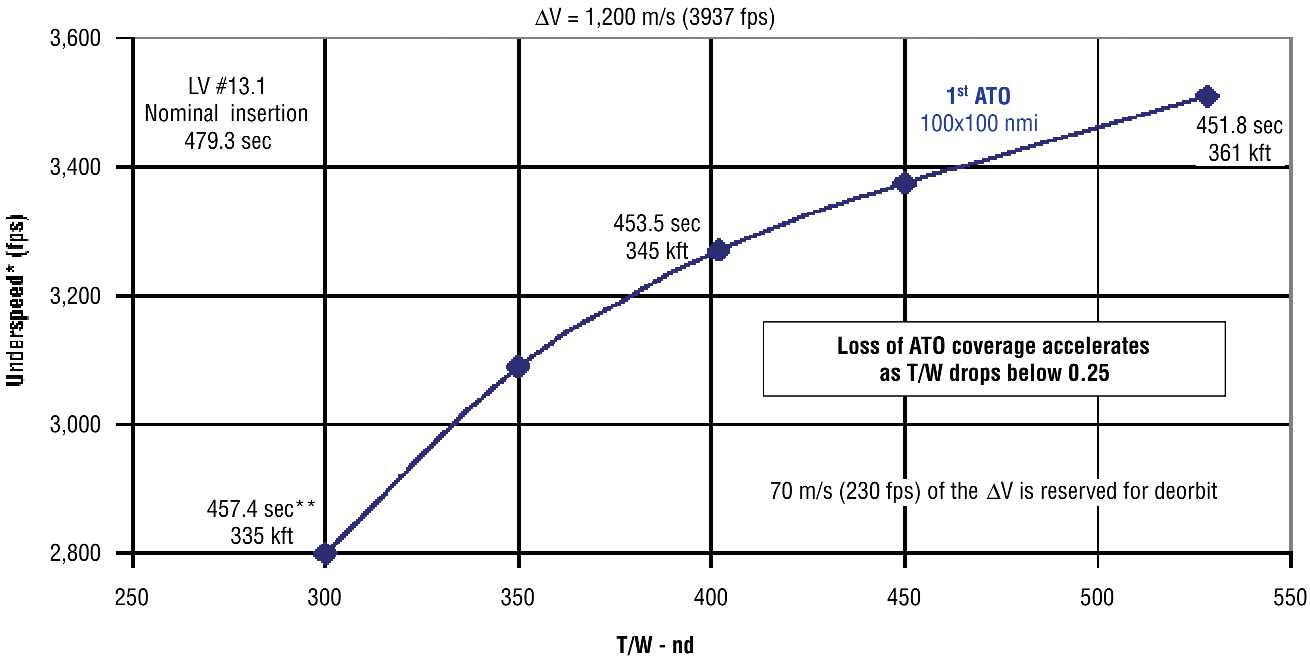


Figure 5-108. Sensitivity of First TAL and ATO to T/W for LV 13.1

The effect of propulsion system ignition delay on ATO coverage is presented in **Figure 5-110**. First ATOs are defined for delays from 20 to 80 sec for two T/W levels. The loss of ATO accelerates when the minimum operating altitude constraint gains prominence. The sensitivity to the propulsion system ignition delay is slightly less for the higher T/W.

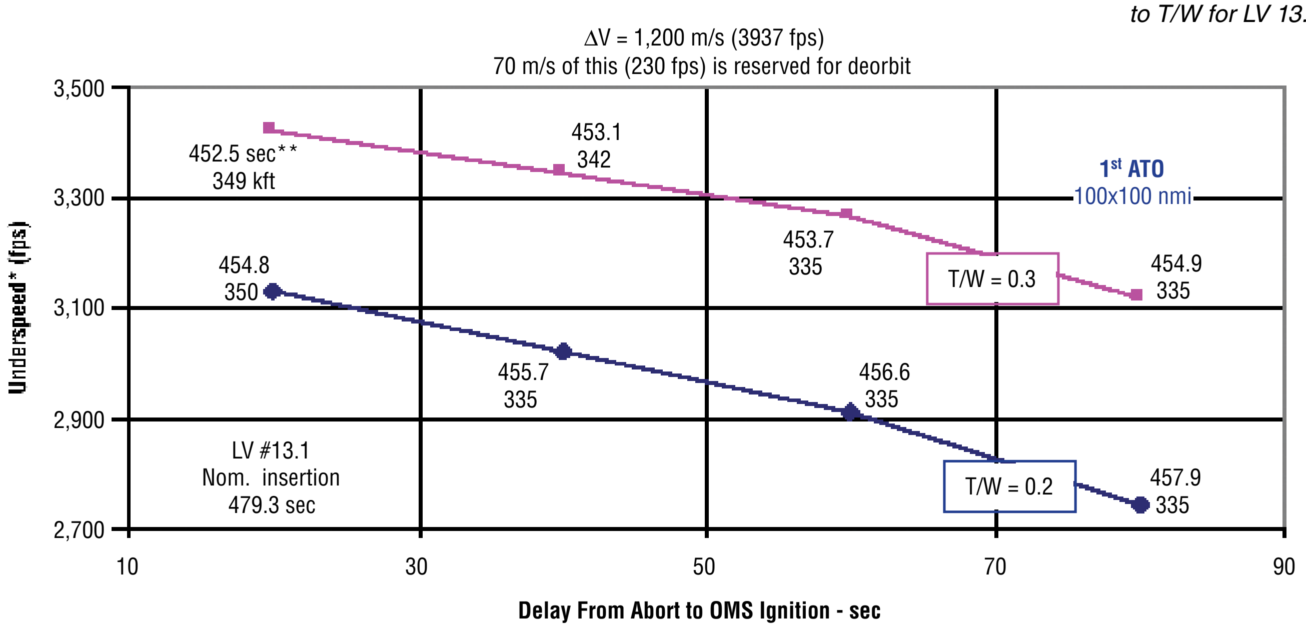
It is interesting to note that, individually, the sensitivity to T/W or propulsion system ignition delay is not that significant (plus or minus a couple of seconds), but, taken together, they become more significant. The abort coverage for LV 13.1, with a given ΔV, can be significantly lessened given a low T/W, a propulsion system that takes as long as Apollo to ignite, and an SM that cannot operate below approximately 340 kft.



* Relative to the nominal velocity at insertion of approximately 25.8 kfps (inertial)

** MET of 1st ATO in sec, minimum altitude in kft

Figure 5-109. Sensitivity of First ATO to T/W for LV 13.1



* Relative to the nominal velocity at insertion of approximately 25.8 kfps (inertial)

** MET of 1st ATO in sec, minimum altitude in kft (constrained to be greater than 335 kft)

Figure 5-110. Sensitivity of First ATO to Propulsion System Ignition Delay for LV 13.1

5.4.4.3 Abort Mode Assessments for 28.5-deg Inclination

Potential ascent abort modes for 28.5-deg inclination launches are shown for LV 13.1 and 26 in **Figures 5-111** and **5-112**, respectively. For posigrade and retrograde suborbital maneuvers, a recovery area is assumed between the Cape Verde Islands and Africa. Posigrade burns can access the recovery area once the ballistic impact point passes roughly 50°W longitude. The significance of this is that some abort landing areas will be far from land, even with the use of propulsion system thrust.

The effect of Earth oblateness should be noted: for the due east missions, the oblate Earth "rises up" during the ascent (the Earth radius increases); whereas, at 51.6 deg, the oblate Earth falls away. This phenomenon seems to explain the apparent reduction in the posigrade down-range abort capability at 28.5 deg. While not readily apparent from the abort mode diagrams, the down-range abort capability at 28.5 deg occurs significantly closer to the ATO abort boundary than at 51.6 deg. This oblateness effect should also impact the ATO boundary for LV 13.1, where minimum altitude is a concern. However, the effect probably is less than 300 fps of under-speed, which is the difference between LV 13.1 and 26. (Refer to **Figure 5-113**.) This effect could be negated by targeting the 28.5-deg engine cut-off at a higher altitude than 51.6°. The Space Shuttle Program used this strategy, targeting Main Engine Cutoff (MECO) 5 nmi higher when due-east missions were flown.

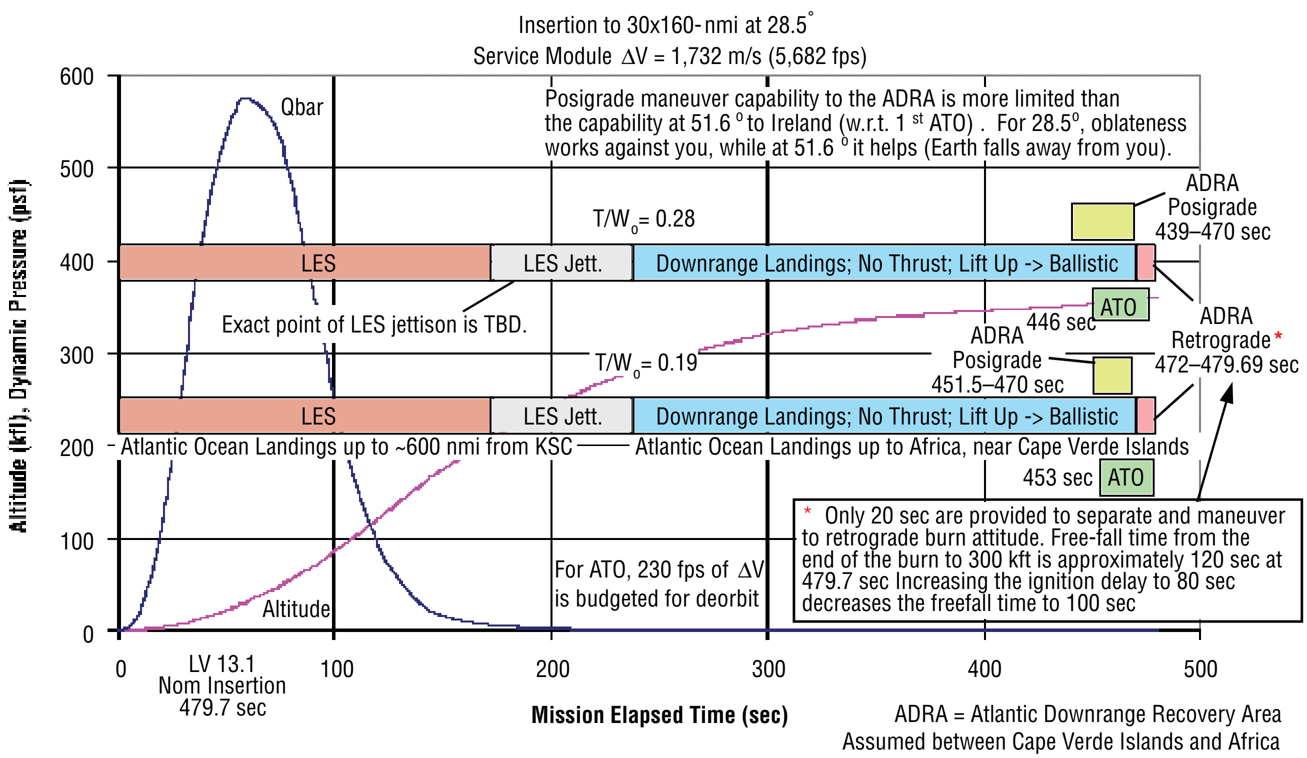
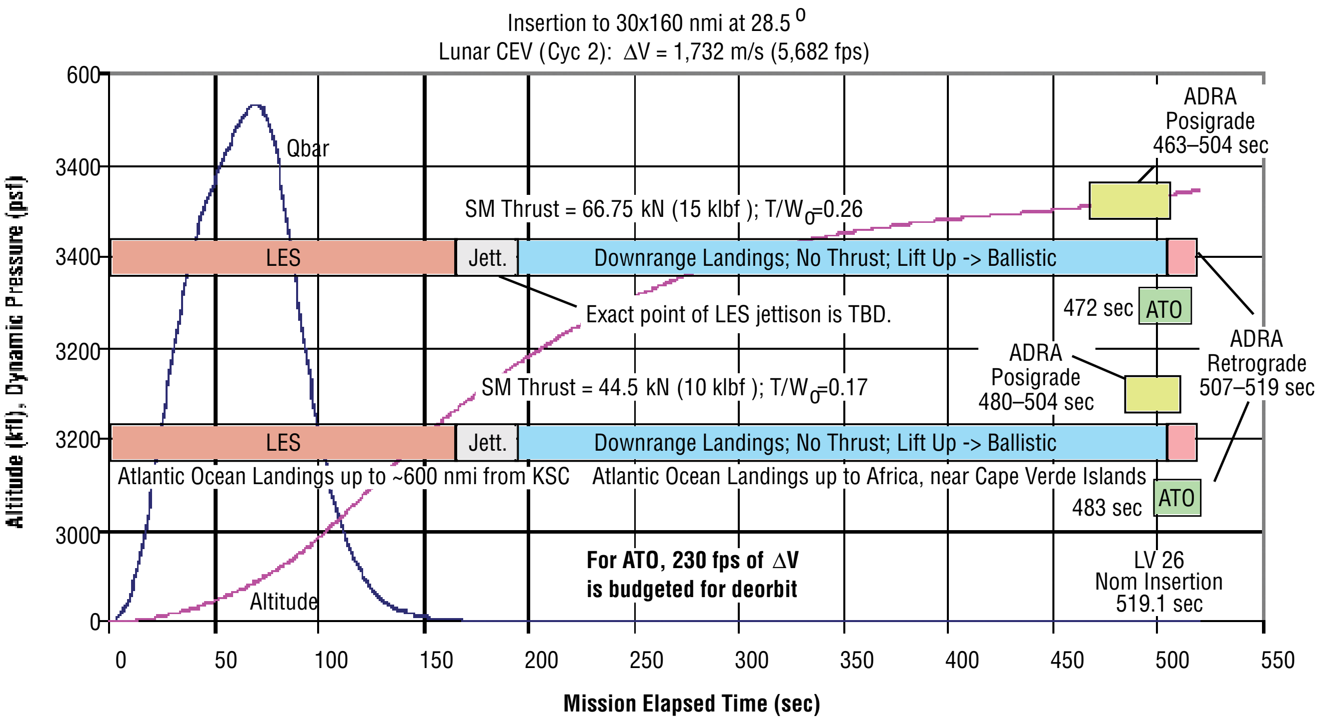


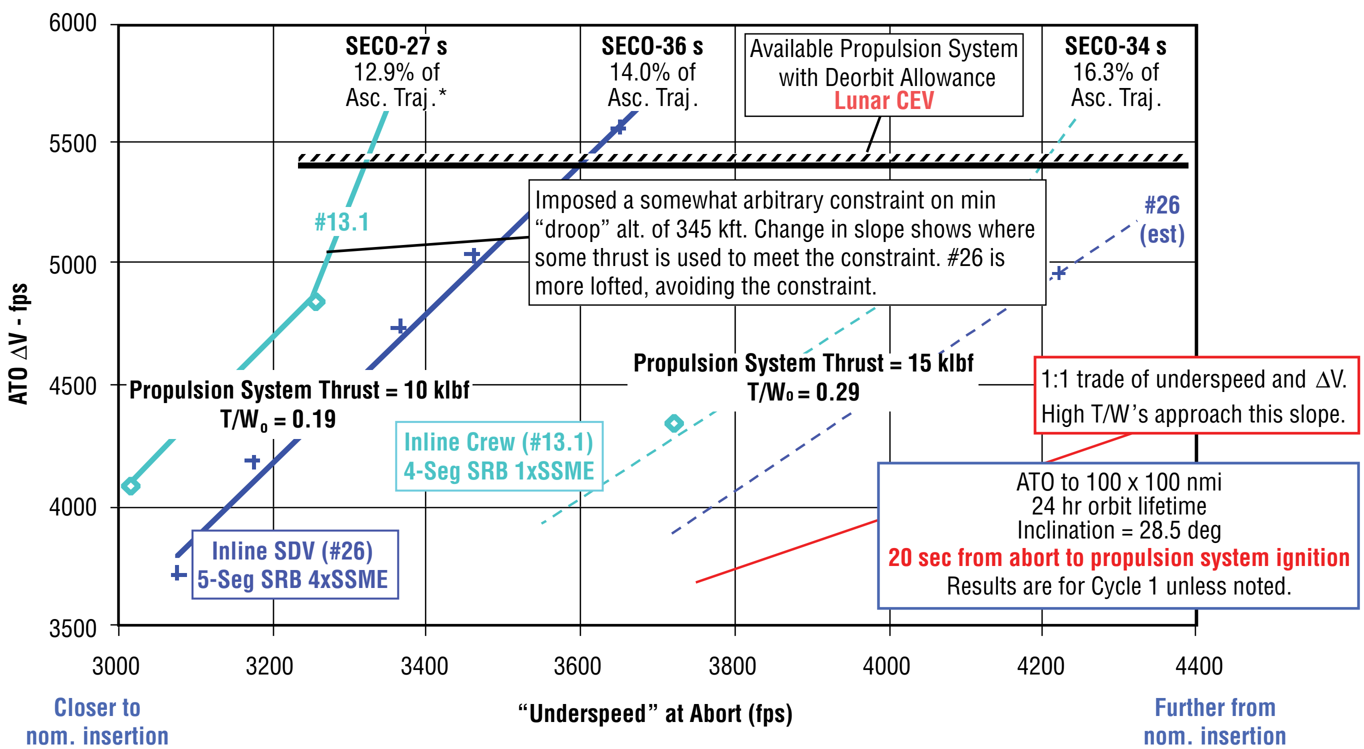
Figure 5-111. Abort Modes for LV 13.1 at 28.5 deg



ADRA = Atlantic Downrange Recovery Area
Assumed between Cape Verde Islands and Africa

Figure 5-112. Abort Modes for LV 26 at 28.5 deg

Figure 5-113 presents a comparison of the ATO ΔV requirement for the LV 13.1 and LV 26 for two T/W levels. The data is presented as a function of abort "under-speed" (i.e., the velocity magnitude short of the nominal engine cutoff velocity). This is a useful parameter for comparing different LVs with different acceleration levels. Because LV 13.1 accelerates at 4g, and LV 26 at 3g, comparison of ATO times relative to nominal engine cut-off can be misleading. One can roughly convert from the under-speed domain to the time domain using the acceleration limits: approximately 100 and 130 fps2 for 3 g and 4 g, respectively (a 1,000 fps under-speed is roughly 10 sec prior to engine cutoff for a 3-g limit). Several interesting trends are presented on **Figure 5-113**. First, the benefit of higher T/W increases for earlier ATOs, which have larger under-speeds. (Note how the different slopes of the T/W curves cause them to diverge as the under-speed increases.) The earlier aborts provide more time for the larger gravity and steering losses of the lower T/W to accumulate. Conversely, the curves converge for smaller under-speeds, indicating that the effect of different T/W and ascent trajectory lofting diminishes as aborts occur closer to nominal engine cutoff. There is also a break point in the curves for LV 13.1. This particular study assumed a minimum operating altitude for the CSM of approximately 345 kft. The slope of the curve increases when the abort gets long enough that the altitude "droops" to the minimum. At that point, more thrust must be "diverted" upwards, making the burn less efficient. Since LV 26 has a more lofted ascent trajectory, this problem occurs at larger under-speeds than are shown on **Figure 5-113**.



Relative to the nominal velocity at insertion of ~25.8 kfps (inertial)

Figure 5-113. Comparison of ATO Delta-V Requirements at 28.5 deg

6. Launch Vehicles and Earth Departure Stages

6.1 Introduction

The United States has embarked on a plan to explore the solar system, both by humans and robotic spacecraft, beginning with a return to the Moon. These first efforts will be followed by human missions to Mars and other locations of interest. A safe, reliable means of human access to space is required after the Space Shuttle is retired in 2010. As early as the mid-2010s, a heavy-lift cargo requirement in excess of 100 mT per flight will be required in addition to the crew launch capability to support manned lunar missions and follow-on missions to Mars.

6.1.1 Charter/Purpose

The Exploration Systems Architecture Study (ESAS) team developed candidate Launch Vehicle (LV) concepts, assessed these concepts against the ESAS Figures of Merit (FOMs) (e.g., cost, reliability, safety, extensibility), identified and assessed vehicle subsystems and their allocated requirements, and developed viable development plans and supporting schedules to minimize the gap between Shuttle retirement and the Crew Exploration Vehicle (CEV) Initial Operational Capability (IOC). The team was directed to develop LV concepts derived from elements of the existing Expendable Launch Vehicle (ELV) fleet and/or the Space Shuttle. A principal goal was to provide an LV capability to enable a CEV IOC in 2011. The team also strived to provide accurate and on-time support and consultation to meet overall ESAS objectives.

The ESAS team was tasked to provide clear recommendations to ESAS management concerning the most advantageous path to follow in answering the following questions:

- Which overall launch architecture provides the most viable options and paths to achieve the stated goals for safe crew transport to Low Earth Orbit (LEO) (Crew Launch Vehicle (CLV)) and meets lift requirements for exploration cargo (Cargo Launch Vehicle (CaLV))?
- What is the preferred CLV concept to provide safe and rapid human access to space after Shuttle retirement in 2010?
- What is the preferred heavy-lift CaLV capable of meeting lunar mission lift requirements and evolving to support Mars missions?
- What is the preferred option for transporting crew for exploration missions beyond LEO?
- What is the best launch option for the robotic exploration effort?
- What is the best launch option for delivering cargo to the International Space Station (ISS) subsequent to Shuttle retirement?

Specifications and analysis results for each of the LV options assessed are provided in **Appendix 6A**, **Launch Vehicle Summary**.

6.1.2 Methodology

The findings of previous studies, particularly the Exploration Systems Mission Directorate (ESMD) Launch Vehicle Study in 2004, had concluded that, while new "clean-sheet" LVs possessed certain advantages in tailoring to specific applications, their high development costs exceeded available budgets and lengthy development schedules would lead to a significant crew transport gap after Shuttle retirement. Therefore, ESAS management directed the team members to use existing LV elements, particularly engines, as much as practicable and to emphasize derivative element designs. New design elements were acceptable where absolutely necessary, but had to be clearly superior in safety, cost, and performance to be accepted. The Payload Fairings (PLFs) for the cargo vehicles are a prime example of a required new element. No existing PLF could accommodate the mass or volume requirements of some of the lunar vehicle elements currently under consideration.

Analysis tasks and technical assessments were focused in several key areas. Considerable effort was expended by the ESAS team to identify, assess, and document applicable vehicle systems, subsystems, and components that were candidates for use in the ELV- and Shuttle-derived vehicle concepts. This information was also used in the generation and assessment of viable development schedules and cost analysis. The team provided key input from the system assessment for safety and reliability analysis. The team developed candidate CLV and CaLV concepts for the study through parametric sizing and structural analysis, and assessed vehicle lift capability and basic induced environments through the generation of three-Degrees-of-Freedom (3-DOF) point-mass trajectory designs anchored by the sizing, structural, and subsystem assessment work. Output of the vehicle concept development work was forwarded to the operations, cost, and reliability/safety groups for use in their analyses. The ESAS team conducted analyses to determine the optimum range for Earth Departure Stage (EDS) main engine thrust levels, EDS configuration layouts, and other supporting analyses.

Candidate LV concept development was governed by the study's overall ground rules and guidance from results of previous studies, including the ESMD Launch Vehicle Study, the ESMD Analysis of Alternatives (AOA), the ESMD Human-Rating Study, and several smaller studies—all of which were conducted in the 12 months preceding the inception of ESAS. The results of the ESMD Concept Exploration and Refinement (CE&R) studies were also evaluated and considered as part of the study. Previous interactions and exchanges with various teams from industry were incorporated, and the ESAS team also conducted and included ongoing interactions with industry teams during the study. Findings from in-house studies conducted in support of the Orbital Space Plane (OSP) Program were also used where applicable. Heritage documentation from the Apollo-Saturn and Space Shuttle Programs were consulted and utilized. Where no data was available for a particular payload-class vehicle, known vehicle elements such as engines, strap-on solids, and strap-on liquid boosters were used to generate representative concepts from LEO payload classes of interest to this study. The candidate concept was initially sized and then flown on a simulated optimized trajectory to assess its performance and to generate data to support an initial structural assessment. The results of the trajectory and the structural analysis were input into a follow-on sizing analysis, which provided an updated vehicle data set. This process was repeated until trajectory results and sizing results agreed within a specified tolerance. The results of this analysis were submitted to the operations, cost, and reliability/safety analysis groups for use in their assessments. Concepts were assessed using the ESAS FOMs provided in **Section 2**, **Introduction** and **Appendix 2E**, **ESAS FOM Definitions**.

The technical requirements for human rating were derived from NASA Procedural Requirements (NPR) 8705.2a, Human-Rating Requirements for Space Systems. The document applies human rating at the system level—identifying the system as LV and spacecraft. Allocation between the LV and spacecraft is provided for in subsequent system requirements documents for the elements. For this study, NPR 8705.2A is the basis for evaluating all ESAS LV concepts to ascertain the modifications and design approaches necessary for carrying crew to Earth orbit.

A depiction of the LV architecture analytical flow is shown in **Figure 6-1**. The ESAS team process flow is shown in **Figure 6-2**.

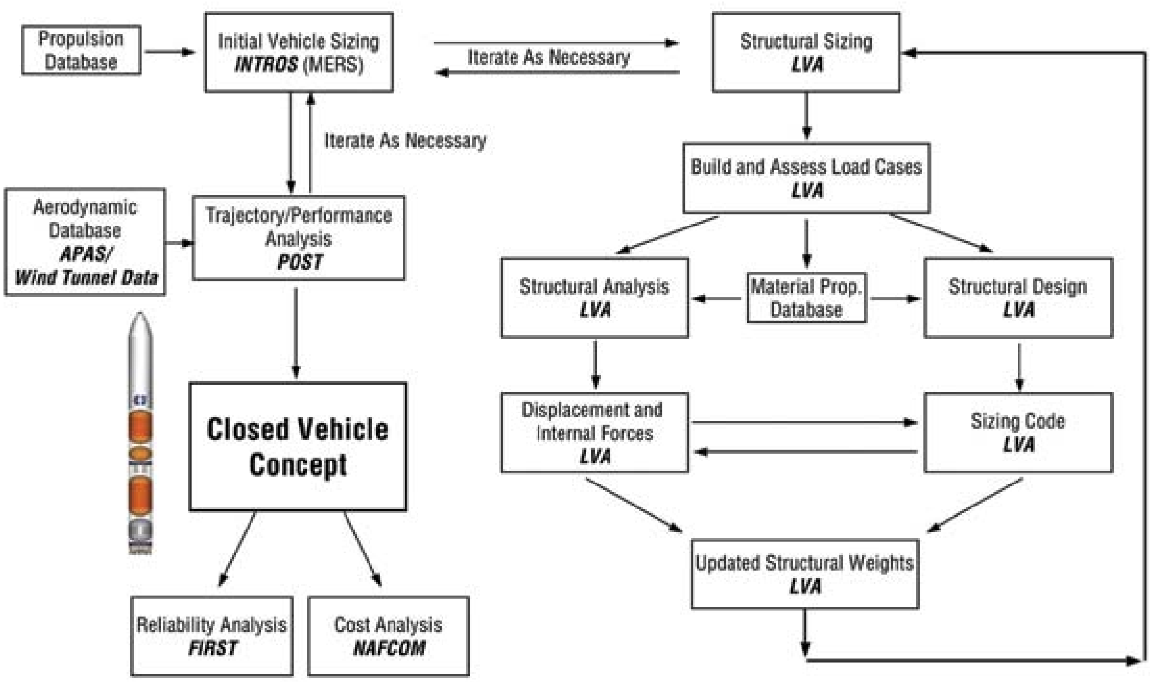


Figure 6-1. Vehicle Conceptual Sizing and Performance Analysis Flow for Earth-to-Orbit (ETO) LVs

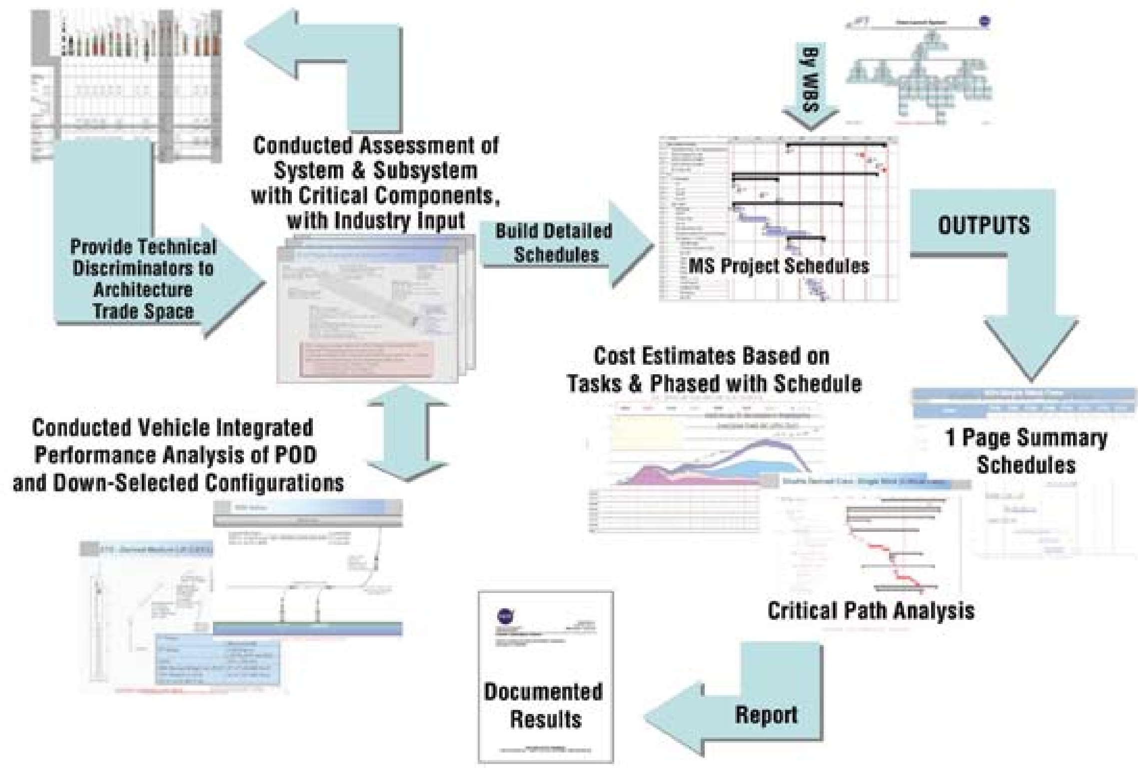


Figure 6-2. ESAS Team Process Flow

The analytical approach taken by the ESAS team was to use the sizing and trajectory data, along with the cost data from the cost group and subsystem data from discipline experts, and synthesize it for the ESAS team. This synthesis process included identifying real limits and risks for key subsystems such as main engines. It also involved characteristic and data comparisons between candidate stages and subsystems. Trends and observations were then reported to the ESAS team.

The conceptual analysis flow for EDS is shown in **Figure 6-3**.

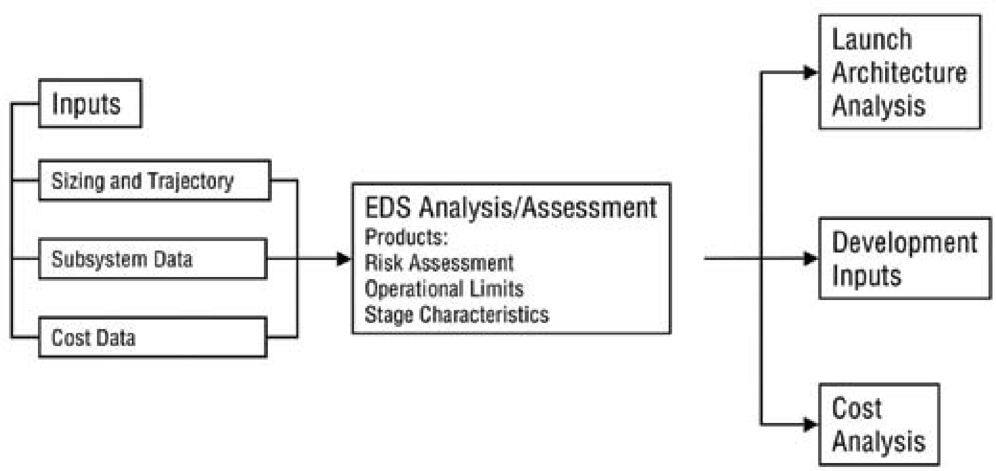


Figure 6-3. Conceptual Analysis Flow for EDS

6.1.3 Recommendations

6.1.3.1 Recommendation 1

Adopt and pursue a Shuttle-derived architecture as the next-generation launch system for crewed flights into LEO and for 125-mT-class cargo flights for exploration beyond Earth orbit. After thorough analysis of multiple Evolved Expendable Launch Vehicle- (EELV-) and Shuttle-derived options for crew and cargo transportation, Shuttle-derived options were found to have significant advantages with respect to cost, schedule, safety, and reliability. Overall, the Shuttle-derived option was found to be the most affordable by leveraging proven vehicle and infrastructure elements and using those common elements in the heavy-lift CaLV as well as the CLV. Using elements that have a human-rated heritage, the CaLV can enable unprecedented mission flexibility and options by allowing a crew to potentially fly either on the CLV or CaLV for 1.5-launch or 2-launch lunar missions that allow for heavier masses to the lunar surface. The Shuttle-derived CLV provides lift capability with sufficient margin to accommodate CEV crew and cargo variant flights to ISS and potentially provides added services, such as station reboost.

The extensive flight and test databases of the Reusable Solid Rocket Booster (RSRB) and Space Shuttle Main Engine (SSME) give a solid foundation of well-understood main propulsion elements on which to anchor next-generation vehicle development and operation. The Shuttle-derived option allows the Nation to leverage extensive ground infrastructure investments and maintains access to solid propellant at current levels. Furthermore, the Shuttle-derived option displayed more versatile and straightforward growth paths to higher lift capability with fewer vehicle elements than other options.

The following specific recommendations are offered for LV development and utilization.

6.1.3.2 Recommendation 2

Initiate immediate development of a CLV utilizing a single four-segment RSRB first stage and a new upper stage using a single SSME. The reference configuration, designated LV 13.1 in this study, provides the payload capability to deliver a lunar CEV to low-inclination Earth orbits required by the exploration architectures and to deliver CEVs configured for crew and cargo transfer missions to the ISS. The existence and extensive operational history of human-rated Shuttle-derived elements reduce safety risk and programmatic and technical risk to enable the most credible development path to meet the goal of providing crewed access to space by 2011. The series-burn configuration of LV 13.1 provides the crew with an unobstructed escape path from the vehicle using a Launch Abort System (LAS) in the event of a contingency event from launch through Earth-Orbit Insertion (EOI). Finally, if required a derivative cargo-only version of the CLV, designated in this report as LV 13.1S, can enable autonomous, reliable delivery of unpressurized cargo to ISS of the same payload class that the Shuttle presently provides.

6.1.3.3 Recommendation 3

To meet lunar and Mars exploration cargo requirements, begin development as soon as practical of an in-line Shuttle-derived CaLV configuration consisting of two five-segment RSRBs and a core vehicle with five aft-mounted SSMEs derived from the present External Tank (ET) and reconfigured to fly payload within a large forward-mounted aerodynamic shroud. The specific configuration is designated LV 27.3 in this report. This configuration provides superior performance to any side-mount Shuttle-derived concept and enables varied configuration options as the need arises. A crewed version is also potentially viable because of the extensive use of human-rated elements and in-line configuration. The five-engine core and two-engine EDS provides sufficient capability to enable the 1.5-launch solution, which requires one CLV and one CaLV flight per lunar mission—thus reducing the cost and increasing the safety/reliability of each mission. The added lift capability of the five-SSME core allows the use of a variety of upper stage configurations, with 125 mT of lift capability to LEO. LV 27.3 will require design, development, and certification of a five-segment RSRB and new core vehicle, but such efforts are facilitated by their historical heritage in flight-proven and well-characterized hardware. Full-scale design and development should begin as soon as possible synchronized with CLV development to facilitate the first crewed lunar exploration missions in the middle of the next decade.

6.1.3.4 Recommendation 4

To enable the 1.5-launch solution and potential vehicle growth paths as previously discussed, NASA should undertake development of an EDS based on the same tank diameter as the cargo vehicle core. The specific configuration should be a suitable variant of the EDS concepts designated in this study as EDS S2x, depending on the further definition of the CEV and Lunar Surface Access Module (LSAM). Using common manufacturing facilities with the Shuttle-derived CaLV core stage will enable lower costs. The recommended EDS thrust requirements will require development of the J–2S+, which is a derivative of the J–2 upper stage engine used in the Apollo/Saturn program, or another in-space high performance engine/cluster as future trades indicate. As with the Shuttle-derived elements, the design heritage of previously flight-proven hardware will be used to advantage with the J–2S+. The TLI capability of the EDS S2x is approximately 65 mT, when used in the 1.5-launch solution mode, and enables many of the CEV/LSAM concepts under consideration. In a single-launch mode, the S2B3 variant can deliver 54.6 mT to Trans-Lunar Injection (TLI), which slightly exceeds the TLI mass of Apollo 17, the last crewed mission to the Moon in 1972.

6.1.3.5 Recommendation 5

Continue to rely on the EELV fleet for scientific and ISS cargo missions in the 5- to 20-mT lift range.

6.1.4 Recommended Launch System Architecture Description

6.1.4.1 Crew Launch Vehicle (LV 13.1)

The recommended CLV concept is derived from elements of the existing Space Shuttle system and is designated as ESAS LV 13.1. It is a two-stage, series-burn configuration with the CEV positioned on the nose of the vehicle, capped by an LAS that weighs 9,300 lbm (pounds of mass). The vehicle stands approximately 290 ft tall and weighs approximately 1.78M lbm at launch. LV 13.1 is capable of injecting a 24.5-mT payload into a 30- x 160-nmi orbit inclined 28.5 deg and injecting 22.9 mT into the same orbit inclined 51.6 deg.

Stage 1 is derived from the Reusable Solid Rocket Motor (RSRM) and is composed of four field-assembled segments, an aft skirt containing the Thrust Vector Control (TVC) hydraulic system, accompanying Auxiliary Power Units (APUs), and Booster Separation Motors (BSMs). The aft skirt provides the structural attachment to the Mobile Launch Platform (MLP) through four attach points and explosive bolts. The single exhaust nozzle is semi-embedded and is movable by the TVC system to provide pitch and yaw control during first-stage ascent. The Space Transportation System (STS) forward skirt, frustrum, and nose cap are replaced by a stage adapter that houses the RSRB recovery system elements and a roll control system. Stage 1 is approximately 133 ft long and burns for 128 sec. After separation from the second stage, Stage 1 coasts upward in a ballistic arc to an altitude of approximately 250,000 ft, subsequently reentering the atmosphere and landing by parachute in the Atlantic Ocean for retrieval and reuse similar to the current Shuttle RSRB.

Stage 2 is approximately 105 ft long, 16.4 ft in diameter, and burns Liquid Oxygen (LOX) and Liquid Hydroxen (LH2). (This was changed to 5.5 m in diameter at the close of the ESAS.) It is composed of an interstage, single RS–25 engine, thrust structure, propellant tankage, and a forward skirt. The interstage provides the structural connection between the Stage 1 adapter and Stage 2, while providing clearance for the RS–25 exhaust nozzle. The RS–25 is an expendable version of the current SSME, modified to start at altitude. The thrust structure provides the framework to support the RS–25, the Stage 2 TVC system (for primary pitch and yaw during ascent), and an Auxiliary Propulsion System (APS) that provides three-axis attitude control (roll during ascent and roll, pitch, and yaw for CEV separation), along with posigrade thrust for propellant settling. The propellant tanks are cylindrical, with ellipsoid domes, and are configured with the LOX tank aft, separated by an intertank. The LH2 main feedline exits the Outer Mold Line (OML) of the intertank and follows the outer skin of the LOX tank, entering the thrust structure aft of the LOX tank. The forward skirt is connected to the LH2 tank at the cylinder/dome interface and acts as a payload adapter for the CEV. It is of sufficient length to house the forward LH2 dome, avionics, and the CEV Service Module (SM) engine exhaust nozzle. Stage 2 burns for approximately 332 sec, placing the CEV in a 30- x 160-nmi orbit. After separation from the CEV, Stage 2 coasts approximately a three-quarter orbit and reenters, with debris falling in the Pacific Ocean.

6.1.4.2 Cargo Launch Vehicle (LV 27.3)

The ESAS LV 27.3 heavy-lift CaLV is recommended to provide the lift capability for lunar missions. It is approximately 357.5 ft tall and is configured as a stage-and-a-half vehicle composed of two five-segment RSRMs and a large central LOX/LH2-powered core vehicle utilizing five RS–25 SSMEs. It has a gross liftoff mass of approximately 6.4M lbm and is capable of delivering 54.6 mT to TLI, or 124.6 mT to 30- x 160-nmi orbit inclined 28.5 deg.

Each five-segment RSRB is approximately 210 ft in length and contains approximately 1.43M lbm of Hydroxyl Terminated Poly-Butadiene (HTPB) propellant. It is configured similarly to the current RSRB, with the addition of a center segment. The operation of the five-segment RSRBs is much the same as the STS RSRBs. They are ignited at launch, with the five RS–25s on the core stage. The five-segment RSRBs burn for 132.5 sec, then separate from the core vehicle and coast to an apogee of approximately 240,000 ft. They are recovered by parachute and retrieved from the Atlantic Ocean for reuse.

The core stage carries 2.2M lbm of LOX and LH2, approximately 38 percent more propellant than the current Shuttle ET, and has the same 27.5-ft diameter as the ET. It is composed of an aft-mounted boattail that houses a thrust structure with five RS–25 engines and their associated TVC systems. The RS–25 engines are arranged with a center engine and four circumferentially mounted engines positioned 45 deg from the vertical and horizontal axes of the core to provide sufficient clearance for the RSRBs. The propellant tankage is configured with the LOX tank forward. Both the LOX and LH2 tanks are composed of Aluminum-Lithium (AL-Li) and are cylindrical, with ellipsoidal domes. The tanks are separated by an intertank structure, and an interstage connects the EDS with the LH2 tank. The core is ignited at liftoff and burns for approximately 408 sec, placing the EDS and LSAM into a suborbital trajectory. A shroud covers the LSAM during the RSRB and core stage phases of flight and is jettisoned when the core stage separates. After separation from the EDS, the core stage continues on a ballistic suborbital trajectory and reenters the atmosphere, with debris falling in the South Pacific Ocean.

6.1.4.3 Earth Departure Stage (EDS S2B3)

The recommended configuration for the EDS is the ESAS S2B3 concept, which is 27.5 ft in diameter, 74.6 ft long, and weighs approximately 501,000 lbm at launch. The EDS provides the final impulse into LEO, circularizes itself and the LSAM into the 160 nmi assembly orbit, and provides the impulse to accelerate the CEV and LSAM to escape velocity. It is a conventional stage structure, containing two J–2S+ engines, a thrust structure/boattail housing the engines, TVC system, APS, and other stage subsystems. It is configured with an aft LOX tank, which is comprised primarily of forward and aft domes. The LH2 tank is 27.5 ft in diameter, cylindrical with forward and aft ellipsoidal domes, and is connected to the LOX tank by an intertank structure. A forward skirt on the LH2 tank provides the attach structure for the LSAM and payload shroud. The EDS is ignited suborbitally, after core stage separation, and burns for 218 sec to place the EDS/LSAM into a 30- x 160-nmi orbit, inclined 28.5 deg. It circularizes the orbit to 160 nmi, where the CEV docks with the LSAM. The EDS then reignites for 154 sec in a TLI to propel the CEV and LSAM on a trans-lunar trajectory. After separation of the CEV/LSAM, the EDS is placed in a disposal solar orbit by the APS.

In connection with the sizing and performance predictions of the various EDS and LV combinations, the ESAS team explored the mission functional requirements on the EDS, such as using suborbital burning to place the payload into LEO. Using this approach, the EDS functions as a third stage for launch and as payload, as it will eventually perform the TLI burn.

6.1.5 Section Content Description

This section of the report offers the following products:

- Lift requirements and trade study and analytical results (**Sections 6.3**, **Lift Requirements**, and **6.4**, **LV and EDS Performance System Trades**);
- CLV and CaLV concept descriptions with cost and development schedule assessments (**Sections 6.5**, **Crew Launch Vehicle**, and **6.6**, **Lunar Cargo Vehicle**);
- EDS assessment (**Section 6.7, Earth Departure Stage**);
- An assessment of system safety and reliability (**Section 6.8**, **LV Reliability and Safety Analysis**);
- Vehicle subsystem descriptions and assessments (**Section 6.9**, **LV Subsystem Descriptions and Risk Assesments**);
- A discussion of conclusions drawn from the conduct of the study (**Section 6.10**, **LV Development Schedule Assesment**);
- A set of recommendations for the CLV, CaLV, EDS, and launch system support for robotic exploration and ISS resupply (**Section 6.11**, **Conclusions**).

A set of appendices (**6A–6H**) containing data summaries and a design assessment of the LV 13.1 CLV is provided separately.

6.2 LV Ground Rules and Assumptions

The LV Ground Rules and Assumptions (GR&As) used by the ESAS team are a subset of those provided previously in **Section 3**, **ESAS Ground Rules and Assumptions**, and are summarized in that section.

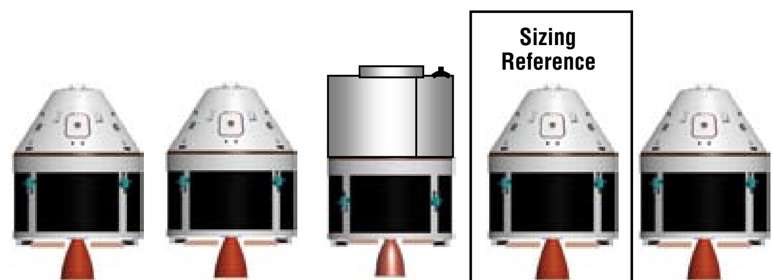
6.3 Lift Requirements

6.3.1 Lunar Missions

The lunar architecture lift requirements involve launching a lunar CEV, an LSAM, and the EDS. The CEV Crew Module (CM) provides a protective environment for the crew during ascent (including aborts), serves as the crew habitat during the lunar mission, and provides the Thermal Protection System (TPS) and recovery system to safely return the crew to Earth at the end of the mission. The CEV Service Module (SM) provides the propulsion system for the Trans-Earth Injection (TEI) burn to return the crew to Earth, life support consumables, power, and other systems required for the lunar mission. The EDS is an in-space rocket stage that burns during the final phase of the CaLV ascent to inject the EDS and the connected LSAM into orbit. The CEV will be placed in orbit by the CLV. The LSAM is attached to the CEV during lunar transit to provide an alternate crew habitat and serves as the primary crew habitat. Also, it provides propulsion and other systems for descent, landing, and ascent at the Moon. Additional details of lunar missions, including specific Design Reference Missions (DRMs), are contained in **Section 2**, **Introduction**.

6.3.2 CEV

The CEV is being considered for access to ISS in three variants, with additional variants for lunar and Mars missions. The block mass summaries for these variants are shown in **Figure 6-4**.



	Block 1A ISS Crew	Block 1B ISS Press Cargo	CDV ISS Unpress Cargo	Block 2 Lunar Crew	Block 3 Mars Crew
Crew Size	3	0	0	4	6
LAS Required	4,218	None	None	4,218	4,218
Cargo Capability (kg)[1]	400	3,500	6,000	Minimal	Minimal
CM (kg)	9,342	11,381	12,200	9,506	TBD
SM (kg)	13,558	11,519	6,912	13,647	TBD
Service Propulsion System delta-V (m/s)	1,544[2]	1,098[2]	330	1,724	TBD
EOR–LOR 5.5-m Total Mass (kg)	22,900	22,900	19,112	23,153	TBD

Note 1: Cargo capability is the total cargo capability of the vehicle including Flight Support Equipment (FSE) and support structure.
Note 2: A packaging factor of 1.29 was assumed for the pressurized cargo and 2.0 for unpressurized cargo.
Extra Block 1A and 1B service propulsion system delta-V used for late ascent abort coverage

Figure 6-4. Block Mass Summaries

The crewed (and possibly uncrewed, pressurized cargo) versions of the CEV carry an LAS consisting of a monolithic Solid Rocket Motor (SRM) that provides an acceleration of at least 10 g's for 3 sec to propel the CEV CM away from a malfunctioning CLV. The LAS projects from the forward end of the CEV CM. It is jettisoned 30 sec after the CLV second stage has ignited. The LAS is used for regions of the ascent flight where high dynamic pressures exist, and where major events, such as staging, occur. The time for LAS jettison was chosen as a point in the ascent where dynamic pressure was dwindling, and the second-stage engine was operating fully in main stage. The baseline LAS total lift mass requirement is 4,152 kg (9,155 lbm). The CEV carries enough propellant to enable transatlantic and Abort-To-Orbit (ATO) options, which are addressed in **Section 5**, **Crew Exploration Vehicle**. The CEV has the potential capability to ATO during the final minute of powered ascent flight, which means that, if the LV could not safely deliver the CEV to orbit even after expending its flight performance reserve propellant (which covers approximately 100 m/s of underspeed), the CEV could place itself in a safe 24-hour orbit, from which the crew would return to Earth.

Additional details of ISS missions, including specific DRMs, are contained in **Section 2**, **Introduction**.

All systems are required to develop a plan that addresses the human-rating system requirements specified in NPR 8705.2A, Human-Rating Requirements for Space Systems, especially the following:

- Specifications and standards,
- Two-fault tolerants systems,
- Crew-system interactions,
- Pad emergency egress,
- Abort throughout the ascent profile,
- Software common cause failures,
- Manual control on ascent, and
- Flight Termination System (FTS).

6.3.3 Launch Window Impacts

When launching for a rendezvous with the ISS or another on-orbit vehicle, additional constraints are placed on the mission. This has an impact on the available launch times and lift capabilities. The first launch of a mission buildup will not be restricted to a specific orbit plane. The inclination will be predetermined, but the ascending node is not determined by the rendezvous requirements. Any subsequent launches must perform the rendezvous missions and must be launched into the orbit plane of the first component.

The effect of the Earth's rotation and the need to launch into the required orbit plane as the launch site rotates past the target orbit is shown as a payload penalty in **Figures 6-5** and **6-6**. **Figures 6-7** and **6-8** show the penalty as a percentage of the payload for each total launch window duration. In this study, the subsequent launches are allowed to optimize the launch azimuth as well as perform yaw steering after the first stage separates. The reference trajectory does not allow the yaw steering.

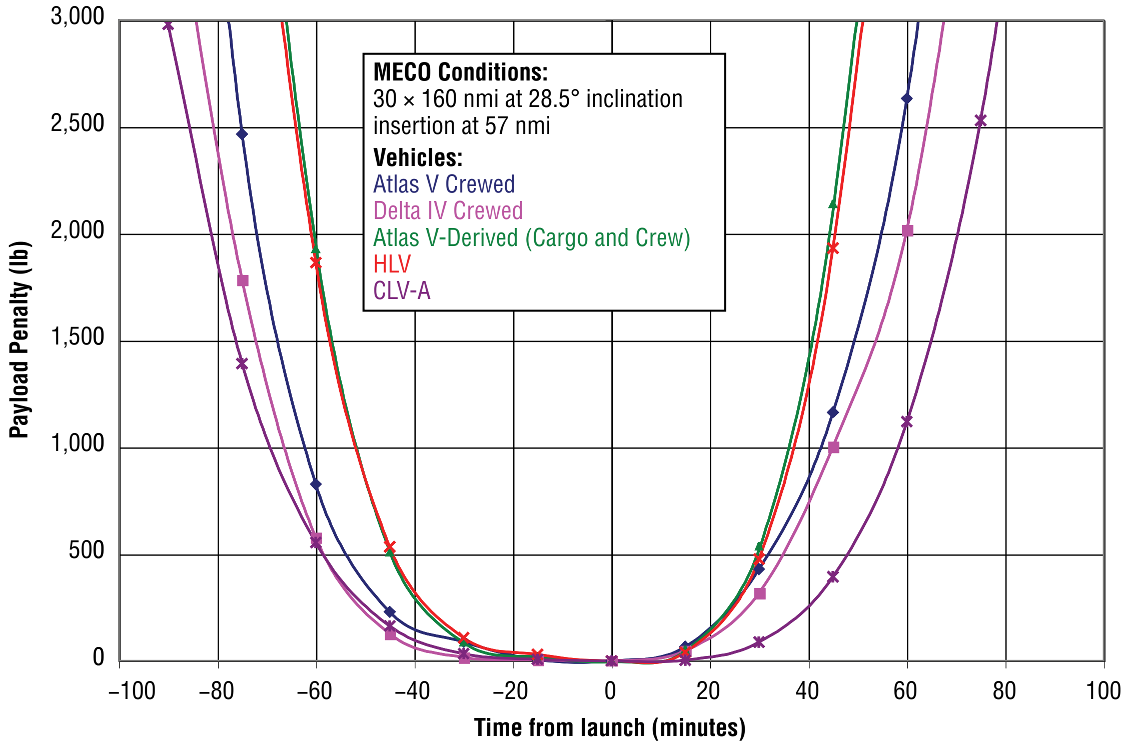


Figure 6-5. Launch Window Payload Penalties (Lunar Due East Launch)

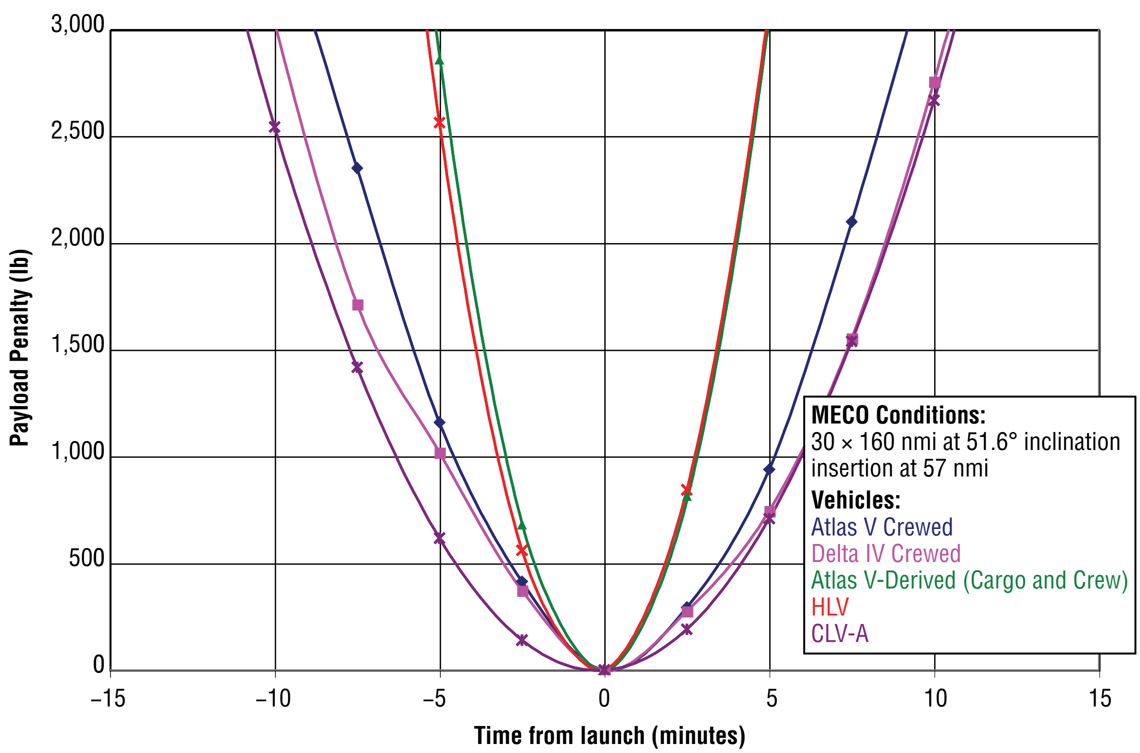


Figure 6-6. Launch Window Payload Penalties (ISS Mission)

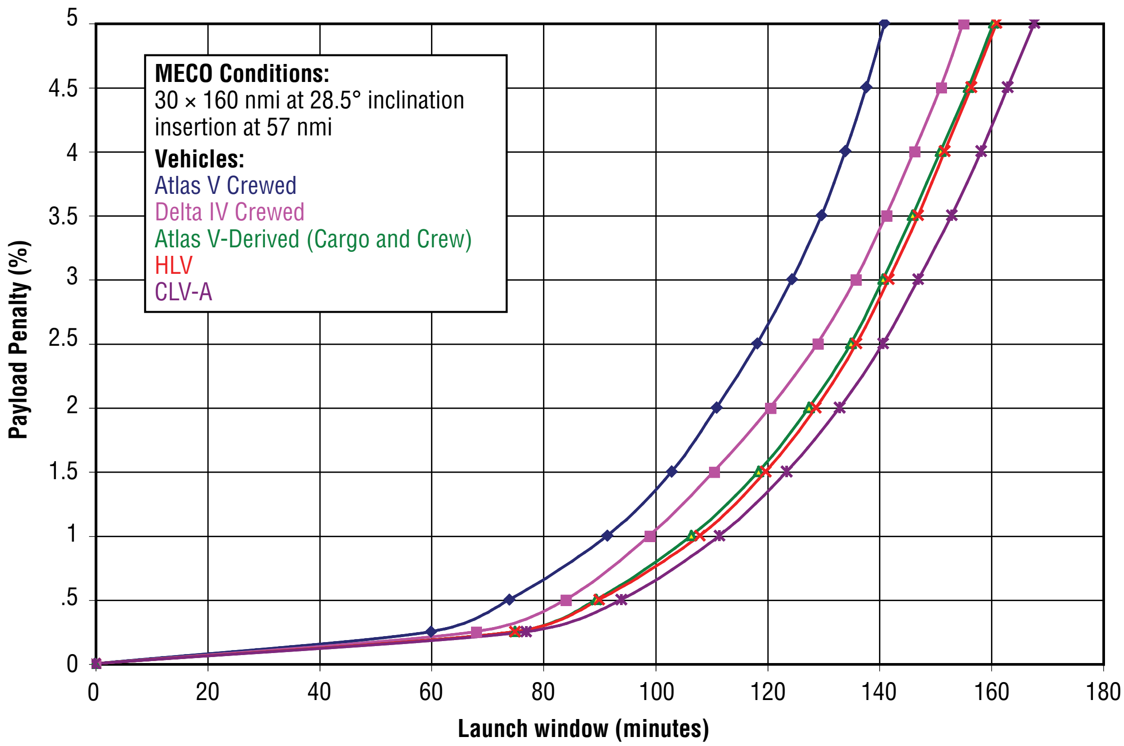


Figure 6-7. Percentage of Payload Lost (Lunar Due East Launch)

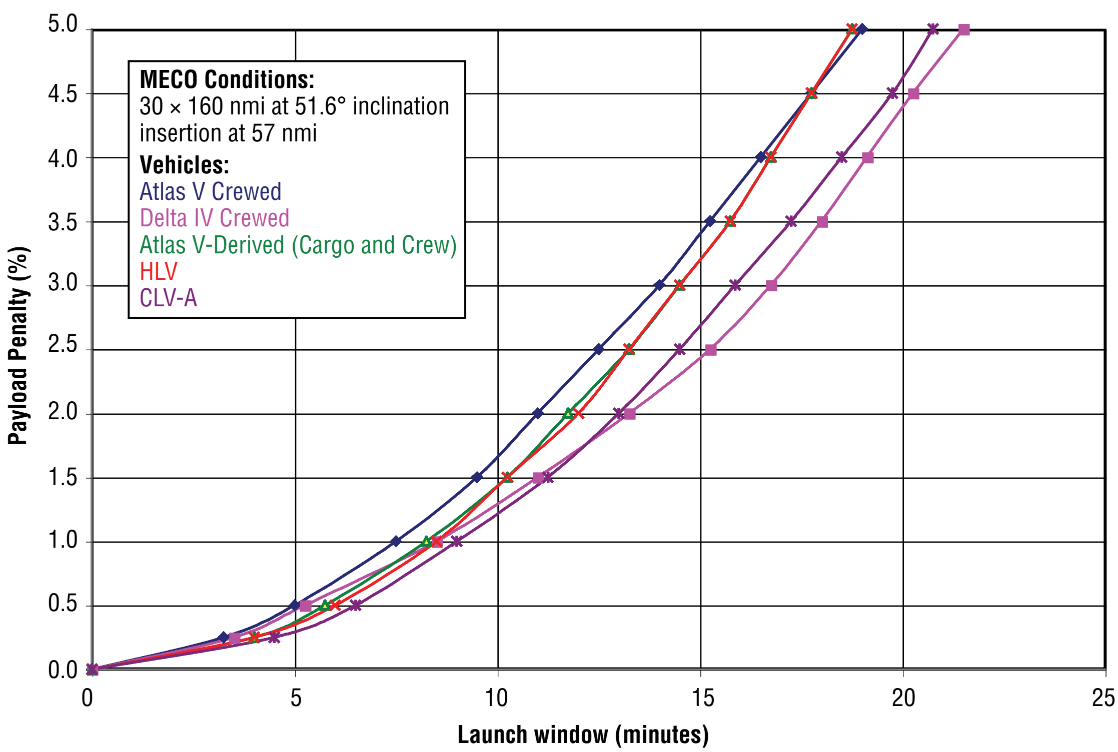


Figure 6-8. Percentage of Payload Lost (ISS Mission)

The vehicles represented in **Figures 6-5** through **6-8** were sized by the ESAS team and used as points of departure (PODs). **Table 6-1** shows the relationship to the study's nomenclature.

Vehicle	Study Nomenclature
Atlas V Crewed	Vehicle 2
Delta IV Crewed	Vehicle 4
Atlas Evolved (Crew+Cargo)	Vehicle 7.5
Heavy-Lift Vehicle (HLV) (CaLV)	Vehicle 27
CLV-A	Vehicle 15 (Results will be identical for LV 13.1)

By launching into a slightly higher inclination, the launch window for a due east mission can be increased with little additional payload penalty. The payload penalties for the two-stage CLV (LV 15) are shown in **Figure 6-9**. The penalty for each total launch window duration is provided in **Figure 6-10**. When the vehicle is launched into the 29.0 deg inclination, two launch opportunities are present within a short period of time. These opportunities represent the ability to launch into either the ascending leg of the orbit or the descending leg. This produces the payload penalty oscillation seen in **Figure 6-9**. Similar analysis was conducted for the in-line CaLV (LV 27). The results are shown in **Figures 6-11** and **6-12**. The penalties for this vehicle are greater than for the CLV.

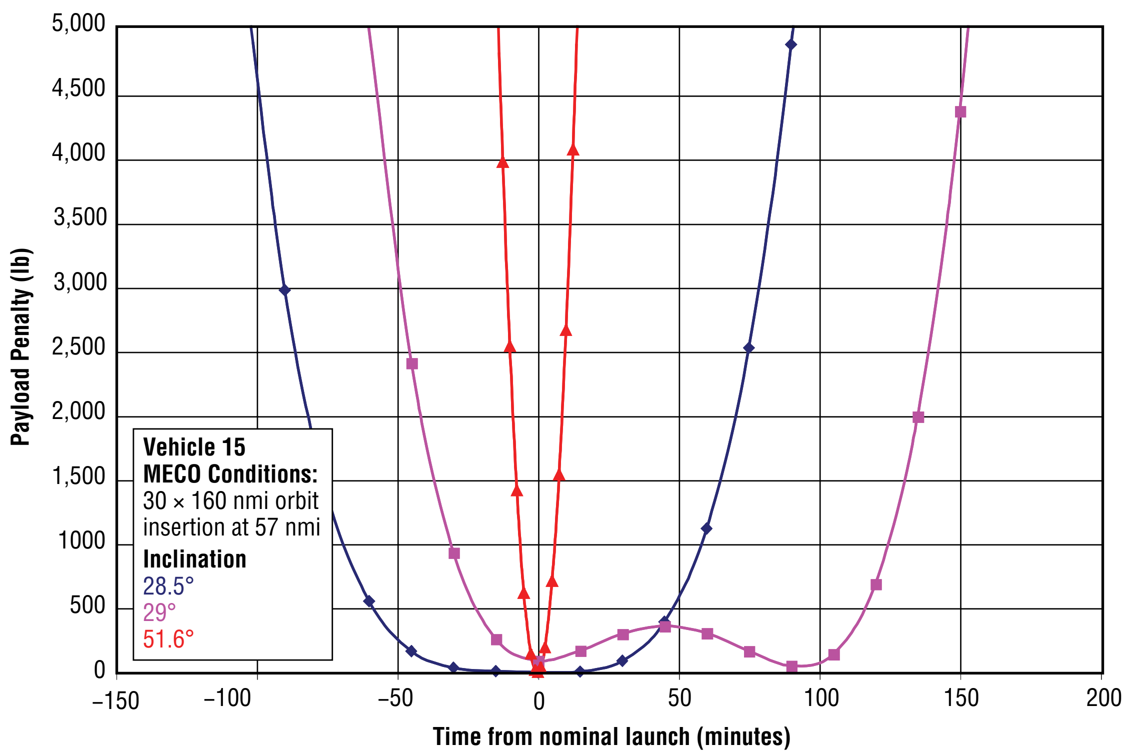


Figure 6-9. Payload Penalty for LV 15

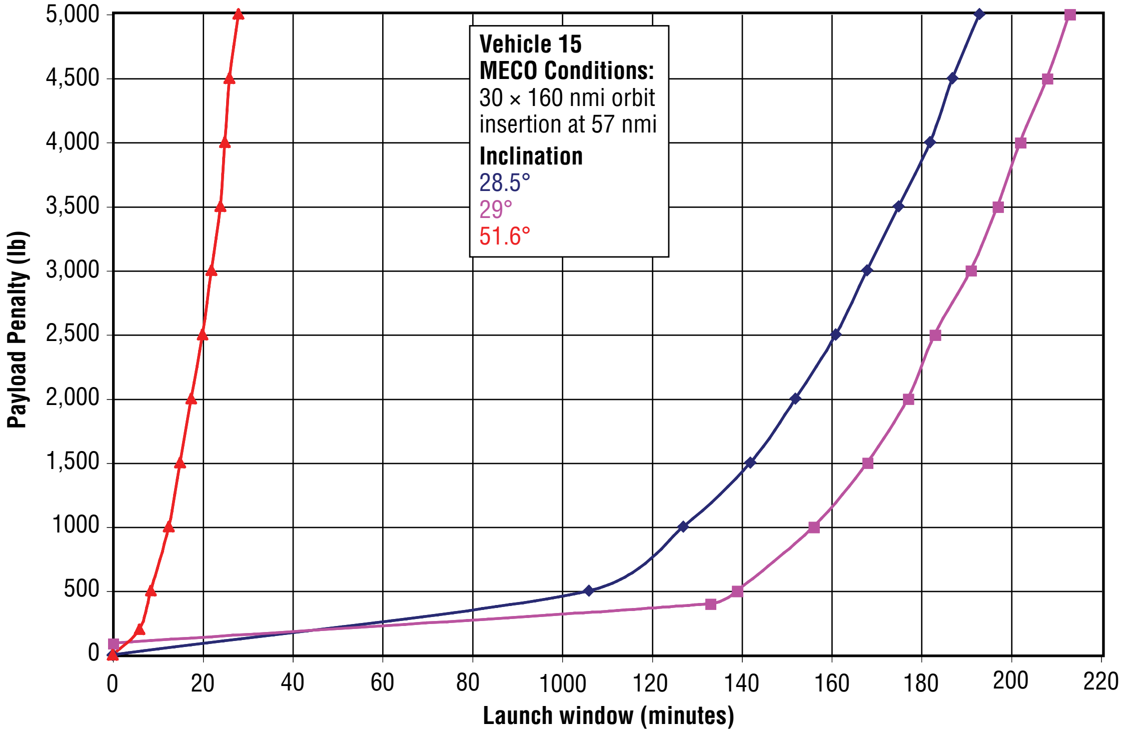


Figure 6-10. Launch Window Duration for LV 15

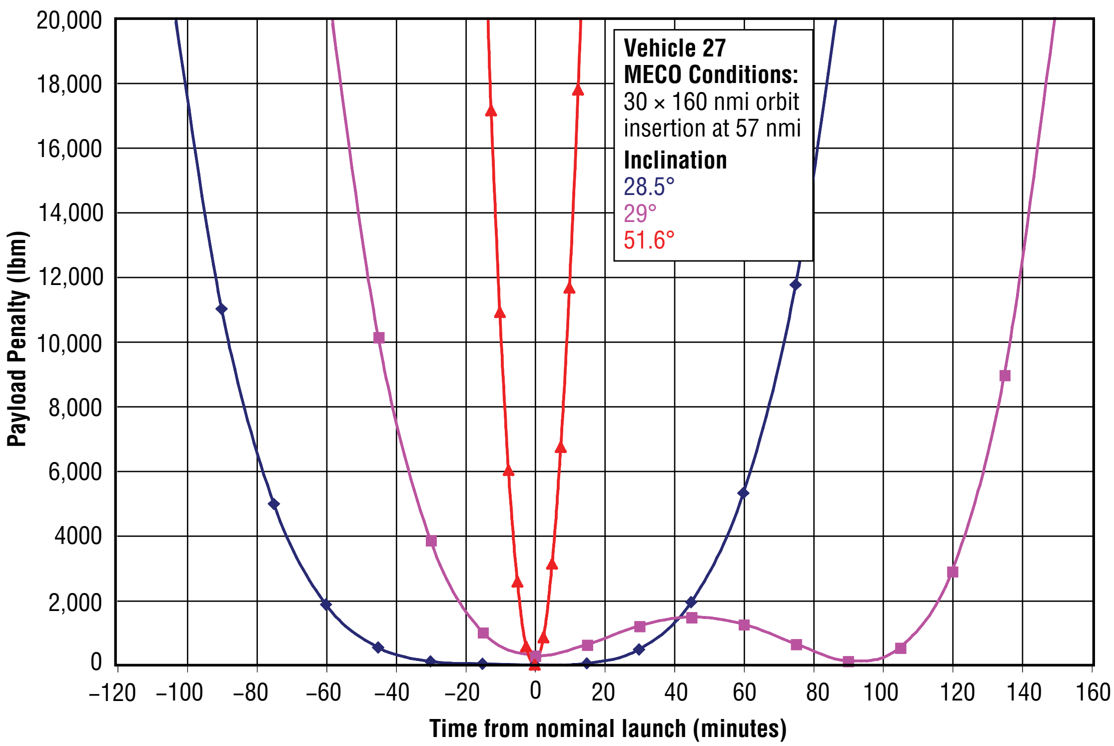


Figure 6-11. Payload Penalty for LV 27

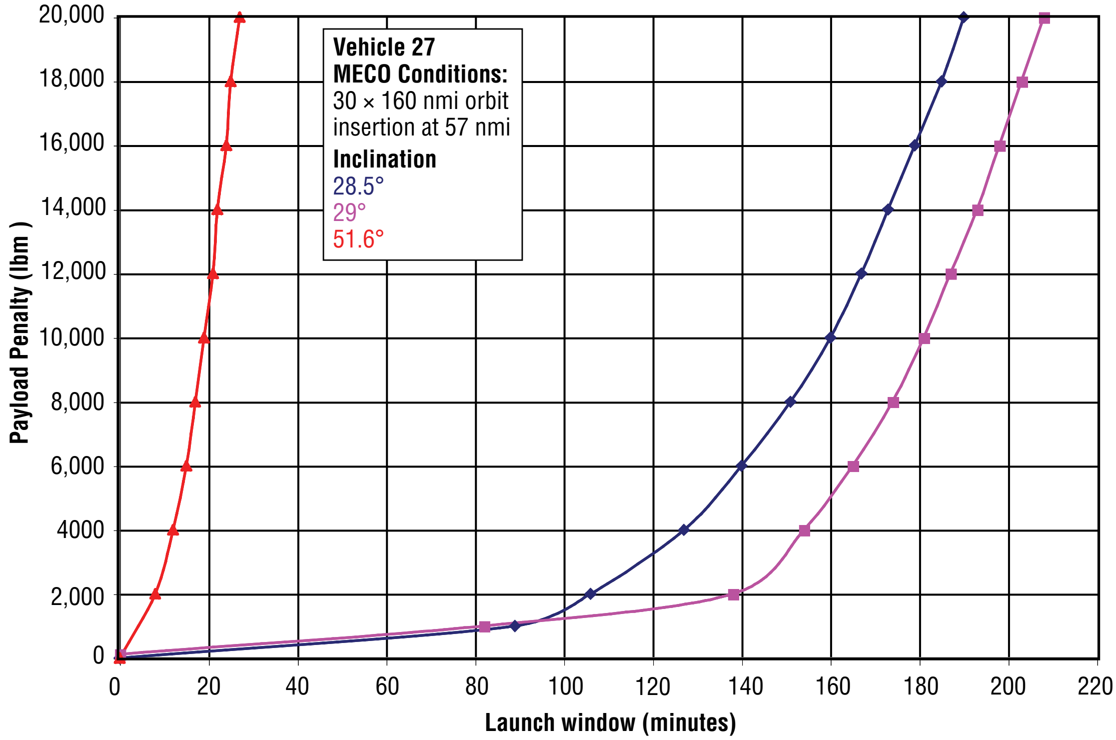


Figure 6-12. Launch Window Duration for LV 27

6.4 LV and EDS Performance System Trades

6.4.1 Launch Trade Tree Description

The options for LVs have become increasingly complex as technical strides are made in materials and systems design. The broad trade space currently available for ETO transportation for crew and cargo is shown in **Figure 6-13**.

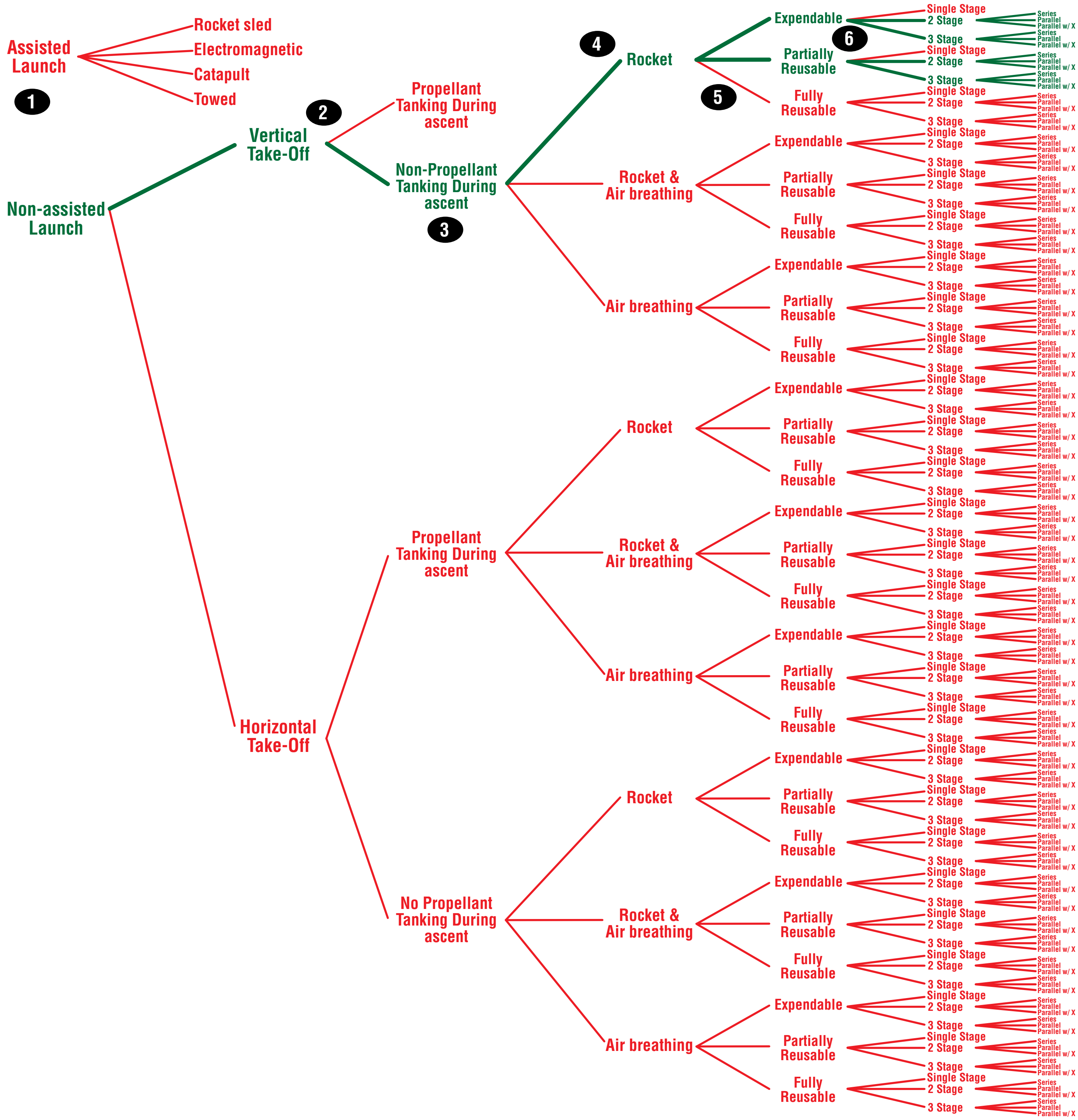


Figure 6-13. Possible Range of Launch Trade Study

In order to arrive at a set of manageable trade options, an objective evaluation must consider the external influences on the concept decision process as well as the technical influences. Prime examples of external influences are:

- Cost: How much is it going to cost to build and field the new system, and how much will it cost to operate?
- Schedule: When is this new capability needed?

Technical influences will include:

- Safety,
- Reliability,
- Available infrastructure,
- Technology level,
- Mission, and
- Crew or cargo requirements.

Many of these influences are interrelated, such as the influence of the availability of infrastructure on the upfront cost to field the new system. For the ESAS, the launch architecture was considered as a whole through the concept-level trade tree. The crew and cargo transportation systems would be treated as an integrated system to take advantage of commonality between systems. Therefore, a common overall launch architecture was defined. A gross examination of the overall trade tree resulted in the branch shown in **Figure 6-14** as the focal point for further consideration, or "pruning."

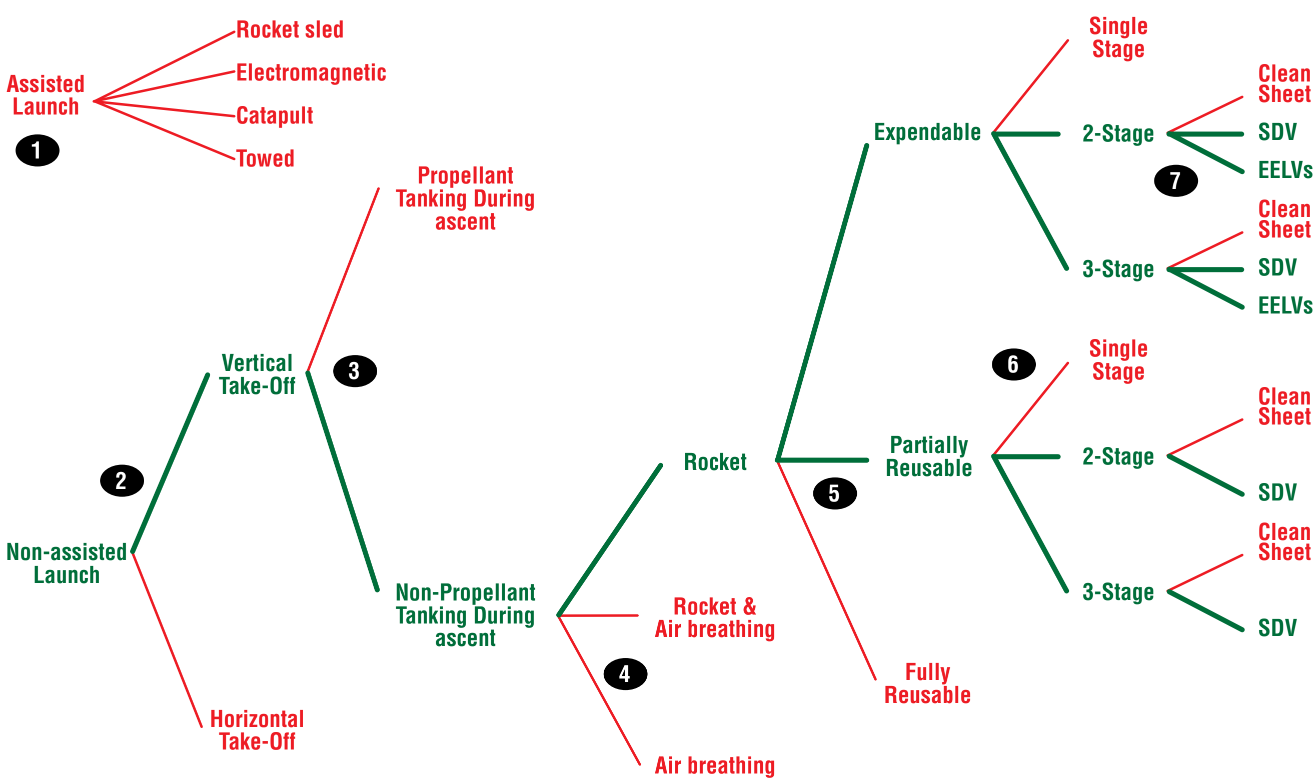


Figure 6-14. Integrated Trade Tree Pruning Rationale

The decision points of the branch are described below, with the subsequent study decisions and supporting rationale.

- Non-assisted versus Assisted Takeoff: Assisted launch systems (e.g., rocket sled, electromagnetic sled, towed) on the scale necessary to meet the payload lift requirements are beyond the state-of the-art for near-term application. Therefore, Non-assisted Takeoff was chosen.
- Vertical versus Horizontal Takeoff: Current horizontal takeoff vehicles and infrastructures are not capable of accommodating the gross takeoff weights of concepts needed to meet the payload lift requirements. Therefore, Vertical Takeoff was chosen.
- No Propellant Tanking versus Propellant Tanking During Ascent: Propellant tanking during vertical takeoff is precluded due to the short period of time spent in the atmosphere (1) to collect propellant or (2) to transfer propellant from another vehicle. Therefore, No Propellant Tanking was chosen.
- Rocket versus Rocket and Air Breathing versus Air Breathing: Air breathing and combined cycle (i.e., rocket and air breathing) propulsion systems are beyond the state-of-the-art for near-term application and likely cannot meet the lift requirements. Therefore, Rocket was chosen.
- Expendable versus Partially Reusable versus Fully Reusable: Fully reusable systems are not cost-effective for the low projected flight rates and large payloads. Near-term budget availability and the desire for a rapid development preclude fully reusable systems. Therefore, Expendable or Partially Reusable was chosen.
- Single-stage versus 2-Stage versus 3-Stage: Single-stage concepts on the scale necessary to meet the payload lift requirements are beyond the state-of-the-art for near-term application. Therefore, 2-Stage or 3-Stage was chosen.
- Clean-sheet versus Derivatives of Current Systems: Near-term budget availability and the desire for a rapid development preclude clean-sheet systems. Therefore, Derivatives of Current Systems was chosen.

Note that the decision rationale is a combination of external and technical influences. The selected architecture is derived from existing launch systems and possesses the following attributes:

- Multistage,
- Expendable or partially reusable,
- Rocket-powered in all stages,
- Carries all of its required propellant from liftoff, and
- Takes off vertically with no assist from ground-based thrust augmentation.

With these features selected, two candidate existing launch systems were identified as having the potential to meet the ESAS requirements:

- Derivatives from the family of EELVs, and
- Derivatives from the Space Shuttle system.

The options sets were kept pure (i.e., elements of the Shuttle were not "mixed and matched" with elements of EELV) with a few exceptions. RS–68 engines were substituted for SSMEs to evaluate the performance difference. J–2S+ engines were used on both EELV and Shuttle-derived options. Findings from previous studies were examined at the beginning of the study to focus efforts on those concepts that provide the greatest potential for meeting the study goals. For example, the ESMD Launch Vehicle study considered several ET-derived CLV concepts that were near-Single-Stage-to-Orbit (SSTO) vehicles, which used an ET-derived first stage with four SSMEs, coupled with a very large CEV SM or small kick stage to inject the CEV into orbit. These concepts were not considered in the ESAS due to their poor performance (i.e., they did not meet the ESAS lift requirements). Also, very large cargo vehicles that used four Shuttle RSRBs were not considered due to the enormous cost of modifying the present launch infrastructure, the Quantity-Distance (QD) safety considerations in the Vehicle Assembly Building (VAB), and because the very high LEO performance of such vehicles was excessive for the intended application. Payload shroud concepts were common, and some cargo vehicle options used the same diameter core vehicle as the ET to take advantage of existing tooling at the Michoud Assembly Facility (MAF). For more information on all LVs assessed, see **Section 6.5**, **Crew Launch Vehicle**, and **Appendix 6A**, **Launch Vehicle Summary**.

6.4.1.1 CaLV Tree

Specific CaLV configuration selections were made based on a variety of practical considerations. Strap-on boosters were a part of each architecture option. Accordingly, strap-on boosters with a central core stage were selected as a POD. The set of major trades for the CaLV is provided in **Figure 6-15**.

6.4.1.2 CLV Tree

Specific CLV configuration selections were also made based on a variety of practical considerations. The set of major trades for the CLV is provided in **Figure 6-16**.

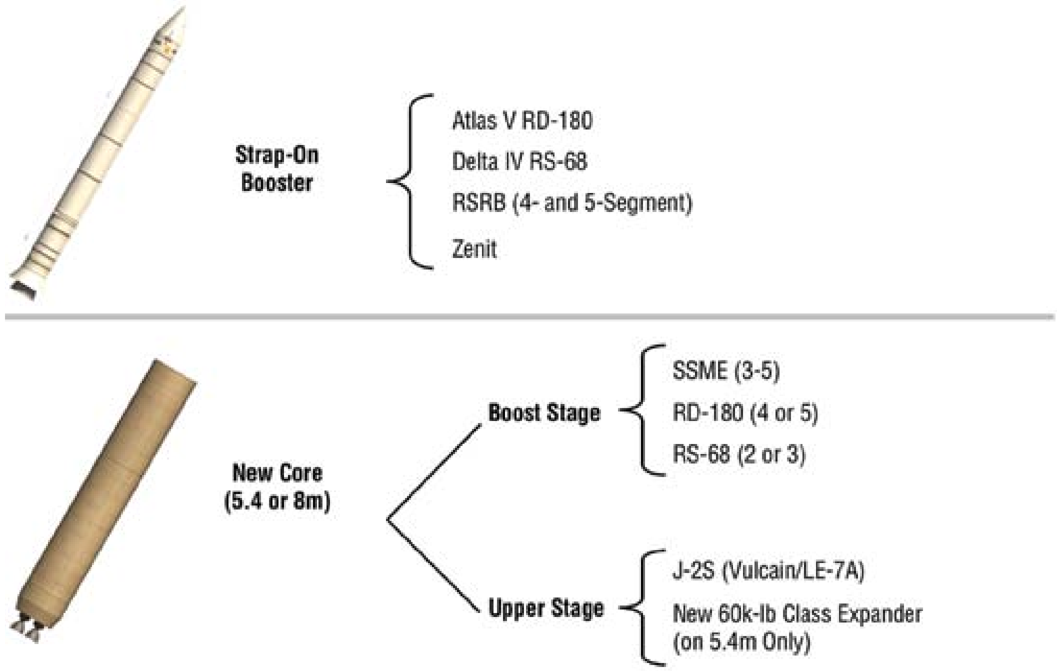


Figure 6-15. Lunar CaLV Trade Tree

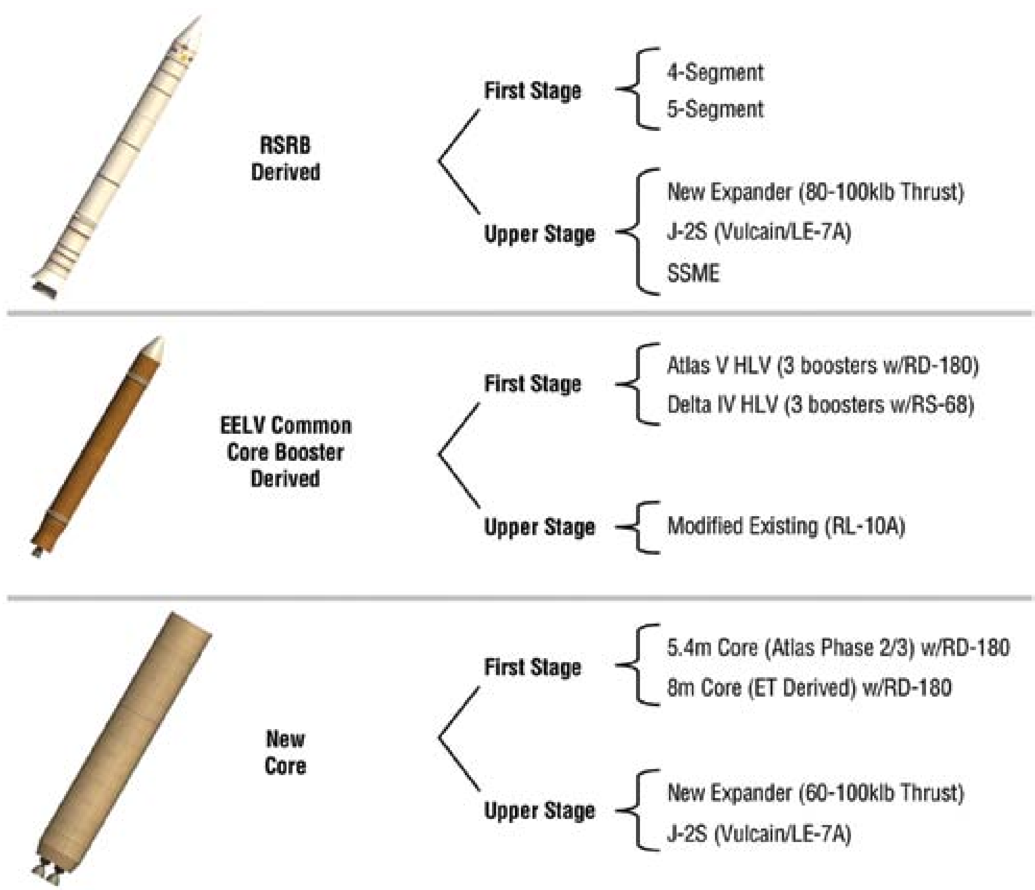
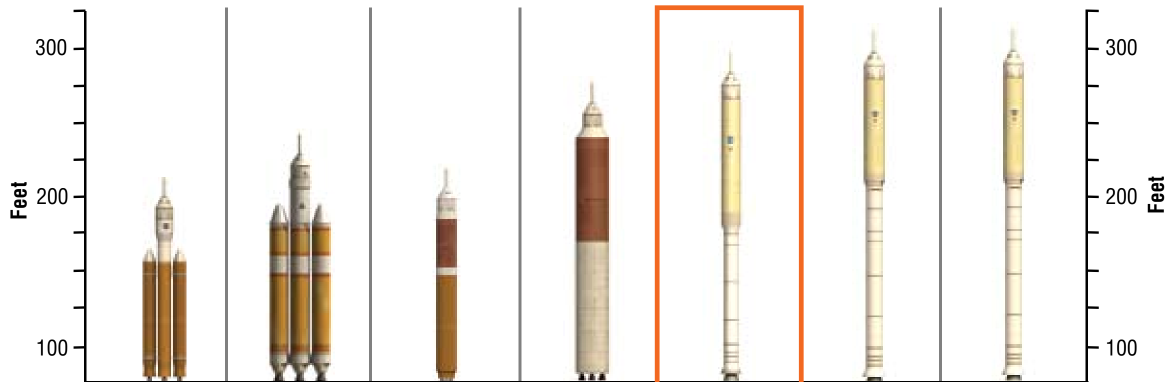


Figure 6-16. Crew (LEO) Trade Tree

6.4.2 LV Trades Overview

6.4.2.1 Crew Launch Vehicle

A summary of the most promising CLV candidates assessed and key parameters is shown in **Figure 6-17**. (Note: cost is normalized to the selected option.)



	Human-Rated Atlas V/New US	Human-Rated Delta IV/New US	Atlas Phase 2 (5.4-m Core)	Atlas Phase X (8-m Core)	4 Segment RSRB with 1 SSME	5 Segment RSRB with 1 J–2S	5 Segment RSRB with 4 LR–85
Payload (28.5°)	30 mT	28 mT	26 mT	70 mT	25 mT	26 mT	27 mT
Payload (51.6°)	27 mT	23 mT	25 mT	67 mT	23 mT	24 mT	25 mT
DDT&E*	1.18**	1.03	1.73**	2.36	1.00	1.3	1.39
Facilities Cost	.92	.92	.92	.92	1.00	1.00	1.00
Average Cost/Flight*	1.00	1.11	1.32	1.71	1.00	.96	.96
LOM (mean)	1 in 149	1 in 172	1 in 134	1 in 79	1 in 460	1 in 433	1 in 182
LOC (mean)	1 in 957	1 in 1,100	1 in 939	1 in 614	1 in 2,021	1 in 1,918	1 in 1,429

LOM: Loss of Mission LOC: Loss of Crew US: Upper Stage RSRB: Reusable Solid Rocket Booster

* *All cost estimates include reserves (20% for DDT&E, 10% for Operations), Government oversight/full cost; Average cost/flight based on 6 launches per year.*

** *Assumes NASA has to fund the Americanization of the RD–180. Lockheed Martin is currently required to provide a co-production capability by the USAF.*

Figure 6-17. Comparison of Crew LEO Launch Systems

The EELV options examined for suitability for crew transport were those derived from the Delta IV and Atlas V families. The study focused on the heavy-lift versions of both Delta and Atlas families, as it became clear early in the study that none of the medium versions of either vehicle had the capability to accommodate CEV lift requirements. Augmentation of the medium-lift class systems with solid strap-on boosters does not provide adequate capability and poses an issue for crew safety regarding small strap-on Solid Rocket Motor (SRM) reliability, as determined by the OSP–ELV Flight Safety Certification Study report, dated March 2004. Both vehicles were assessed to require modification for human rating, particularly in the areas of avionics, telemetry, structures, and propulsion systems.

Both Atlas- and Delta-derived systems required new upper stages to meet the lift and human-rating requirements. Both Atlas and Delta single-engine upper stages fly highly lofted trajectories, which can produce high deceleration loads on the crew during an abort and, in some cases, can exceed crew load limits as defined by NASA Standard (STD) 3000, Section 5. Depressing the trajectories flown by these vehicles will require additional stage thrust to bring peak altitudes down to levels that reduce crew loads enough to have sufficient margins for off-nominal conditions. Neither Atlas V nor Delta IV with their existing upper stages possess the performance capability to support CEV missions to ISS, with shortfalls of 5 mT and 2.6 mT, respectively.

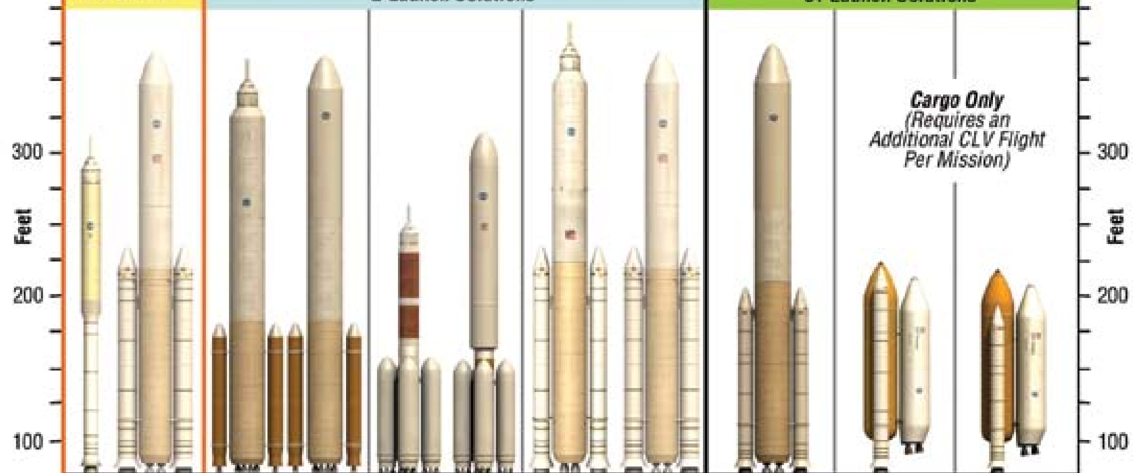
Another factor in both vehicles is the very low Thrust-to-Weight (T/W) ratio at liftoff, which limits the additional mass that can be added to improve performance. The RD–180 first-stage engine of the Atlas HLV will require modification to be certified for human rating. This work will, by necessity, have to be performed by the Russians. The RS–68 engine powering the Delta IV HLV first stage will require modification to eliminate the buildup of hydrogen at the base of the vehicle immediately prior to launch. Assessments of new core stages to improve performance as an alternative to modifying and certifying the current core stages for human rating revealed that any new core vehicle would be too expensive and exhibit an unacceptable development risk to meet the goal of the 2011 IOC for the CEV. Note the EELV costs shown in **Figure 6-17** do not include costs for terminating Shuttle propulsion elements/environmental cleanup. Finally, both the EELV options were deemed high-risk for a 2011 IOC.

CLV options derived from Shuttle elements focused on the configurations that used an RSRB, either as a four-segment version nearly identical to the RSRB flown today or a higher-performance five-segment version of the RSRB using HTPB as the solid fuel. New core vehicles with ET-derived first stages (without Solid Rocket Boosters (SRBs)) similar to the new core options for EELV were briefly considered, but were judged to have the same limitations and risks and, therefore, were not pursued. To meet the CEV lift requirement, the team initially focused on five-segment RSRB-based solutions. Three classes of upper stage engine were assessed—SSME, a single J–2S+, and a four-engine cluster of a new expander cycle engine in the 85,000-lbf vacuum thrust class. However, the five-segment development added significant near-term cost and risk and the J–2S+/expander engine could not meet the 2011 schedule target. Therefore, the team sought to develop options that could meet the lift requirement using a four-segment RSRB. To achieve this, a 500,000-lbf vacuum thrust class propulsion system is required. Two types of upper stage engine were assessed—a two-engine J–2S cluster and a single SSME. The J–2S option could not meet the 2011 target (whereas the SSME could) and had 6 percent less performance than the SSME-based option (LV 13.1). The SSME option offered the added advantages of an extensive and successful flight history and direct extensibility to the CaLV with no gap between the current Shuttle program and exploration launch. Past studies have shown that the SSME can be altitude-started, with an appropriate development and test program.

The 13.1 configuration was selected due to its lower cost, higher safety/reliability, its ability to utilize existing human-rated systems and infrastructure, and the fact that it gave the most straightforward path to a CaLV.

6.4.2.2 Cargo Launch Vehicle

A summary of the most promising CaLV candidates and key parameters is shown in **Figure 6-18**. (Note: Cost is normalized to the selected option.) The requirement for four or less launches per mission results in a minimum payload lift class of 70 mT. To enable a 2- or 1.5-launch solution, a 100- or 125-mT class system, respectively, is required.



	1.5-Launch	2-Launch Solutions			3+ Launch Solutions		
	5 Segment RSRB In-Line with 5 SSME Core - Cargo	Atlas Phase X (8-m Core)	Atlas Phase 3A (5.4-m Core)	5 Segment RSRB In-Line with 4 SSME Core	4 Segment RSRB In-Line with 3 SSME Core	5 Segment RSRB Side-mount with 3 SSME	4 Segment RSRB Side-mount with 3 SSME
Payload to 28.5	106 mT (125 mT w/ upperstage)	95 mT	94 mT	97 mT	74 mT	80 mT	67 mT
Lunar LV DDT&E*	1.00	1.29	.59	.96	.73	.80	.75
CLV+Lunar Crew/Cargo DDT&E (family op)*	1.00 (4 Seg RSRB w/1 SSME)	1.26 (Atlas V)	1.02 (Phase 2)	.98 (4 Seg RSRB w/1 SSME)	.83 (4 Seg RSRB w/1 SSME)	1.03 (5 Seg RSRB w/J2S-LR85)	.85 (4 Seg RSRB w/1 SSME)
Lunar LV Facility DDTE	1.00	1.33	2.25	1.00			
CLV+Lunar Facilities Cost	1.00	1.12	1.56	1.00			
Lunar LV Average Cost/Flight*	1.00	1.08	1.19	.87	.78	1.13**	1.13**
LOM – Cargo (mean)	1 in 124	1 in 71	1 in 88	1 in 133	1 in 176	1 in 172	1 in 173
LOC (mean)	1 in 2,021	1 in 536	1 in 612	1 in 915	1 in 1,170	N/A	N/A

* All cost estimates include reserves (20% for DDT&E, 10% for Operations), Government oversight/full cost; Average cost/flight based on 6 launches per year.

**Production costs are higher than in-line due to production of separate side-mount cargo carrier

Figure 6-18. Lunar Cargo Launch Comparison

EELV-derived options for the CaLV included those powered by RD–180 and RS–68 engines, with core vehicle diameters of 5.4 and 8 m. No RS–68-powered variant of an EELV-derived heavy-lift cargo vehicle demonstrated the capability to meet the lunar lift requirements without a new upper stage and either new large liquid strap-on boosters or Shuttle RSRBs. The considerable additional cost, complexity, and development risk were judged to be unfavorable, eliminating RS–68-powered CaLVs. Hydrocarbon cores powered by the RD–180 with RD–180 strap-on boosters proved to be more effective in delivering the desired LEO payload. Vehicles based on both a 5.4-m diameter core stage and an 8-m diameter core were analyzed. A limitation exhibited by the EELV-Derived Vehicles (EDVs) was the low liftoff T/W ratios for optimized cases. While the EELV-derived CaLVs were able to meet LEO payload requirements, the low liftoff T/W ratio restricted the size of EDS in the suborbital burn cases. As a result, the Earth-escape performance of the EELV options was restricted. The 5.4-m core CaLV had an advantage in Design, Development, Test, and Evaluation (DDT&E) costs, mainly due to the use of a single diameter core derived from the CLV which was also used as a strap-on booster. However, the CLV costs for this option were unacceptably high. (See **Section 6.4.2.1**, **Crew Launch Vehicle**.) In addition, there would be a large impact to the launch infrastructure due to the configuration of the four strap-on boosters (i.e., added accommodations for the two additional boosters in the flame trench and launch pad). Also, no EELV-derived concept was determined to have the performance capability approaching that required for a lunar 1.5-launch solution. Finally, to meet performance requirements, all EELV-derived CaLV options required a dedicated LOX/LH2 upper stage in addition to the EDS—increasing cost and decreasing safety/reliability.

The Shuttle-derived options considered were of two configurations: (1) a vehicle configured much like today's Shuttle, with the Orbiter replaced by a side-mounted expendable cargo carrier, and (2) an in-line configuration using an ET-diameter core stage with a reconfigured thrust structure on the aft end of the core and a payload shroud on the forward end. The ogive-shaped ET LOX tank is replaced by a conventional cylindrical tank with ellipsoidal domes, forward of which the payload shroud is attached. In both configurations, three SSMEs were initially baselined. Several variants of these vehicles were examined. Four- and five-segment RSRBs were evaluated on both configurations, and the side-mounted version was evaluated with two RS–68 engines in place of the SSMEs. The J–2S+ was not considered for use in the CaLV core due to its low relative thrust and the inability of the J–2S+ to use the extended nozzle at sea level, reducing its Specific Impulse (Isp) performance below the level required. No variant of the side-mount Shuttle-Derived Vehicle (SDV) was found to meet the lunar lift requirements with less than four launches. The side-mount configuration would also most likely prove to be very difficult to human rate, with the placement of the CEV in close proximity to the main propellant tankage, coupled with a restricted CEV abort path as compared to an in-line configuration. The proximity to the ET also exposes the CEV to ET debris during ascent, with the possibility of contact with the leeward side TPS, boost protective cover, and the LAS. The DDT&E costs are lower than the in-line configurations, but per-flight costs are higher—resulting in a higher per-mission cost. The side-mount configuration was judged to be unsuitable for upgrading to a Mars mission LEO capability (100 to 125 mT). The in-line configuration in its basic form (four-segment RSRB/three-SSME) demonstrated the performance required for a 3-launch lunar mission at a lower DDT&E and per-flight costs. Upgrading the configuration with five-segment RSRBs and four SSMEs in a stretched core with approximately one-third more propellant enables a 2-launch solution for lunar missions, greatly improving mission reliability. A final variation of the Shuttle-derived in-line CaLV was considered. This concept added a fifth SSME to the LV core, increasing its T/W ratio at liftoff, thus increasing its ability to carry large, suborbitally ignited EDSs. LV 27.3 demonstrated an increased lift performance to enable a 1.5-launch solution for lunar missions, launching the CEV on the CLV and launching the LSAM and EDS on the larger CaLV. This approach allows the crew to ride to orbit on the safer CLV with similar Life Cycle Costs (LCCs) and was selected as the reference. This configuration proved to have the highest LEO performance and lowest LV family nonrecurring costs. When coupled with the four-segment RSRB/SSME-derived CLV (13.1), Loss of Mission (LOM) and Loss of Crew (LOC) probabilities are lower than its EELV-derived counterparts.

6.4.3 EDS Performance Trades

Four variations of EDS missions were examined against four representative CaLVs. The LVs were:

- LV 25: Shuttle-derived in-line CaLV with two four-segment RSRBs and three SSMEs;
- LV 27 (and variants): Shuttle-derived in-line CaLV with two five-segment RSRBs and four SSMEs (five SSMEs on LV 27.3);
- LV 30: Shuttle-derived in-line CaLV with two five-segment RSRBs and four SSMEs and an upper stage with two J–2 engines; and
- LV 7.4: EELV-derived with two Atlas V strap-on boosters, a 5-RD–180 core vehicle with a 4-J–2S+ upper stage.

The mission trade variations were the four paired combinations of:

- Suborbital burn versus no suborbital burn, and
- Payload versus no payload.

A summary of coupled LV/EDS performance capabilities appears in **Appendix 6C**, **Launch Vehicles and EDS Performance Sizing**. The results of the EDS performance trades indicated that there were numerous EDS/LV combinations that would work for 2- and 3+-launch solutions for lunar missions. In assessing the 1.5-launch solution, a large, suborbitally ignited EDS capable of carrying an LSAM proved to be the most advantageous from a performance and cost perspective. The basic 1.5-launch EDS concept, S2B3/4/5, when coupled with LV 27.3, allows a 45 mT LSAM to be delivered with it to orbit. No other CaLV provided this capability. The addition of the fifth SSME and the large EDS eliminated the need for a separate upper stage and EDS. The high T/W of LV 27.3 (approximately 1.45) is a key factor in enabling the 1.5-launch solution.

6.4.4 Number of Launches and On-Orbit Assembly Assessment

6.4.4.1 Synopsis

To assess the merits and pitfalls of the number of launches required for exploration missions, an analysis was conducted of the key parameters: LV availability (including launch scrub recycle time and mission window), LV reliability, and automated rendezvous subsystem reliability. Concatenation of these parameters as a function of the number of LVs was evaluated using LOM (i.e., the failure to successfully complete one mission out of a number of missions) as the FOM. The results showed that, for any combination of parameters based on history, a very small quantity of very Heavy-Lift Launch Vehicles (HLLVs) was the path to acceptable values of LOM.

6.4.4.2 Problem Statement

Mission success for crewed lunar missions depends on three significant processes. First, launch availability relates to the architecture's ability to provide on-time launch of each of the mission elements. Second, LV reliability relates to the architecture's ability to successfully fly each LV to the destination orbit and release the payload, either cargo or crew. Third, Automated Rendezvous and Docking (AR&D) reliability relates to the architecture's ability to successfully conduct the on-orbit integration of all elements that require automated link-up prior to initiation of the lunar mission. The concatenation of these three processes provides a first-order estimate of the architecture's likelihood to succeed. This estimate is measured by LOM. This estimate does not include all aspects of the total mission; rather, it is truncated for the purpose of this analysis at the point of lunar departure.

6.4.4.3 Analysis Process

The following sections discuss each of the processes, explain the analytical methodology, provide results for each of the processes, and develop observations related to the combined results.

6.4.4.3.1 Availability Analysis

Availability is the probability that any LV in the chain required for a lunar mission will fly in the planned launch window. Many outside conditions and design features affect the value of availability. The lunar payload and LV must be designed to support rapid integration and yet require minimal support on the launch pad, including a rapid and reliable ability to be fueled. Both the range and the LV must be tolerant of a variety of weather conditions: temperature, winds both at the surface and at altitude, cloud cover, and lightning. The Russian Soyuz vehicle and infrastructure represents a good example. Finally, the range must have the ability to be rapidly and reliably reconfigured to minimize the time required to support each launch, whether lunar cargo or crew, ISS crew or cargo, or other Department of Defense (DoD) or commercial missions.

Although the ability to launch as scheduled contributes to the likelihood that a lunar mission is successfully launched, there are other embedded parameters that significantly influence the LOM measure. These significant parameters include the number of launches required, the mission window required for the launch of all elements, and the "recycle" time required following a scrub to prepare for the next launch attempt. This analysis assumed that the individual launch availability varies from 60 percent to 90 percent, that the number of launches could be as many as 10 to 20 (but greater than 1), that 4 to 5 months could be needed to put all elements in orbit, and the "mission clock" starts after the first successful launch. Assuming that each subsequent launch requires 2 weeks of preparation, the key independent parameter was the average recycle time. Since recycle times are due to diverse consequences and could vary from 1 day (weather-related) to 2 years (design-related), this analysis assumed an average over all scrubs of 3 to 7 days.

The time lines of two different architectural solutions (**Figure 6-19**) illustrate the implications of these parameters, where the individual LV availability was assumed to be 100 percent. The first architecture requires two launches over an 18-week period with 7 days on average between launch scrubs to provide all lunar mission elements. As can be seen in **Figure 6-19**, 14 launch scrub/recycle events still leave sufficient time to have both launches occur within the mission window. Conversely, a "9-launch" architectural solution allows for zero scrubs to meet the 4-month mission window.

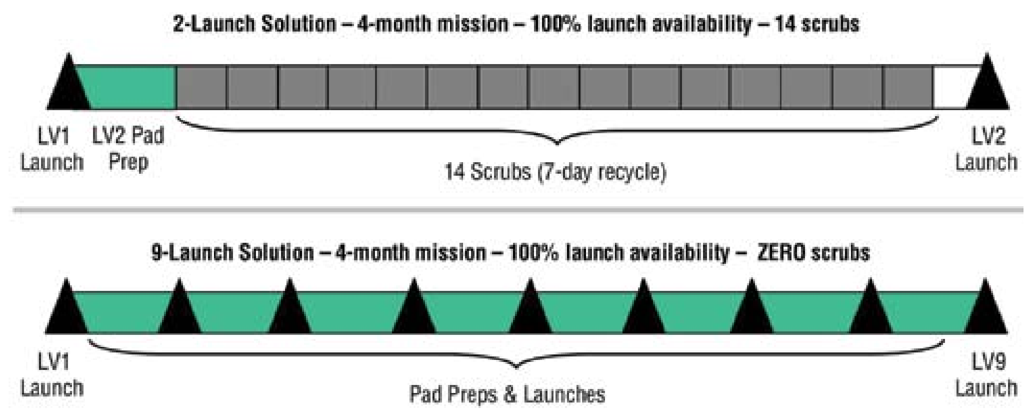


Figure 6-19. Description of 2-Launch versus 9-Launch Solutions

A summary of the analysis for the above assumptions is shown in **Figure 6-20**, where the measure is the cumulative probability of successfully launching all mission elements.

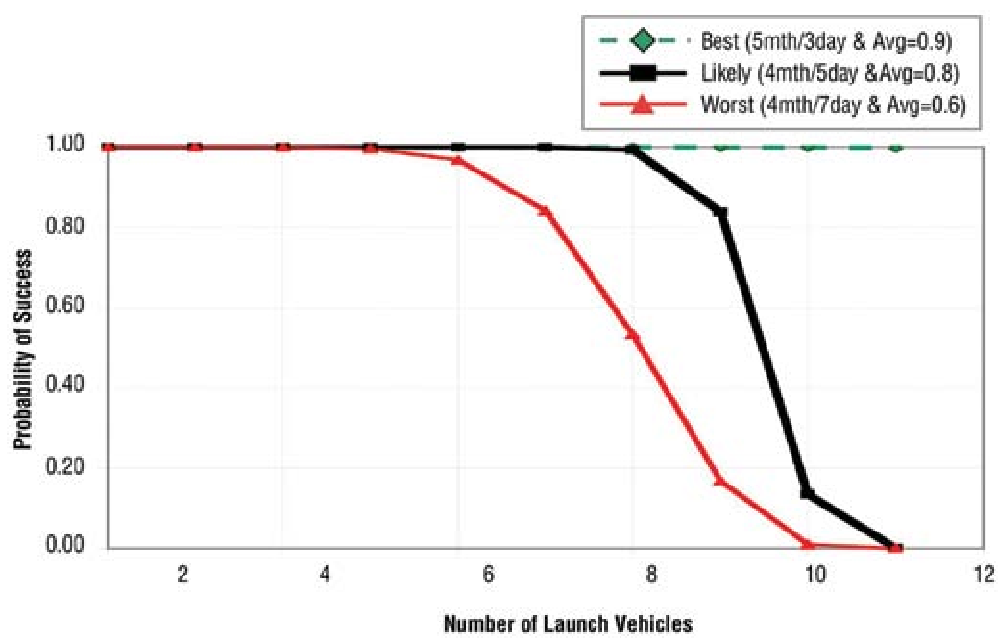


Figure 6-20. Probability of Mission Completion

Clear impacts can be observed from these data:

- Availability favors a fewer number of launches. If large numbers of launches are needed to support a lunar mission, the probability of success rapidly diminishes. Indeed, there are certain combinations of assumptions in which it is not possible to statistically achieve eight or more launches.
- The range would likely have to be dedicated to lunar mission support configuration until all elements are launched due to the time-critical nature of the on-orbit cryogenic propellants used and the nature of scrubs and recycles. Given the multiplicity of Eastern Range customers, this restriction would be undesirable, yet vital for lunar mission success. This restriction is exacerbated if multiple yearly lunar missions are considered.
- Twin launch pads would shorten the mission window and the range dedication.
- A dedicated range for lunar traffic models greater than one annual mission would be desirable.

6.4.4.3.2 LV Reliability Analysis

Although the reliability of specific CLV and CaLV configurations was analyzed parametrically, for the purposes of this analysis of number of launches, historical LV reliabilities were used. LV reliability varies significantly depending on the system: Soyuz reliability is approximately 97 percent over more than 1,000 launches, Delta 2 has 98 percent reliability in 100+ launches, the Shuttle demonstrated launch reliability of 99 percent, and Pegasus has less than 90 percent demonstrated reliability. Statistically, the chaining of launches using historical averages results in the LOM shown in **Figure 6-21**. For 10 LVs of current demonstrated reliability, the LOM due only to LV reliability would be one failed mission in 5 to 10—undesirable in terms of the expense of the launched assets lost. However, as the concatenation of the significant parameters show, LV reliability is not the dominant term and contributes the least to the overall LOM result.

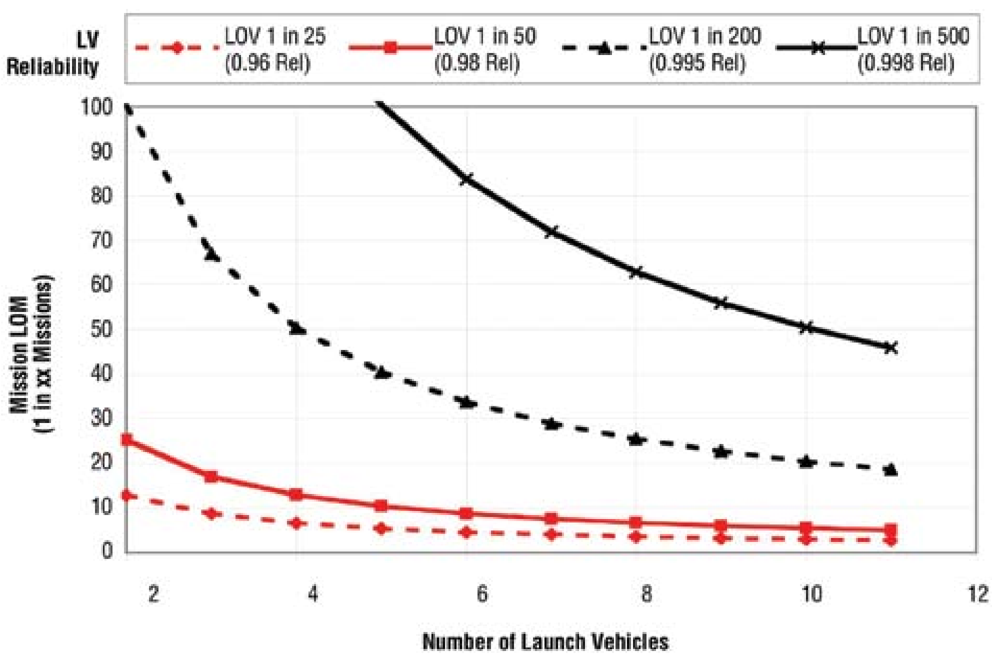


Figure 6-21. LOM Due Only to Average LV Reliability

6.4.4.3.3 AR&D Reliability Analysis

In most multi-launch vehicle lunar architectures, some of the mission elements must be linked without the presence of human aid, just as when Progress docks with ISS. An AR&D system, illustrated functionally in **Figure 6-22** below, is quite complex. As a flight element of the host in-space element, the system must plan for the orbital rendezvous path with contingencies, continuously measure with increasing precision the position of its host relative to the target, execute the guidance through propulsion on the chaser, communicate and display state and status data to many users, and make contact with the target that finally results in docking.

Element	Function
Sensor	Reliably Measure Where You Are Relative to Target
GN&C Msn Mgr	Control Planned Rendezvous Event Plan/Replan Rendezvous Event
Comm	Communicate Data Reliably Thruout Useable Range
Visualization	Graphically Support Critical Decisions
Docking	Reliably Connect Both Vehicles

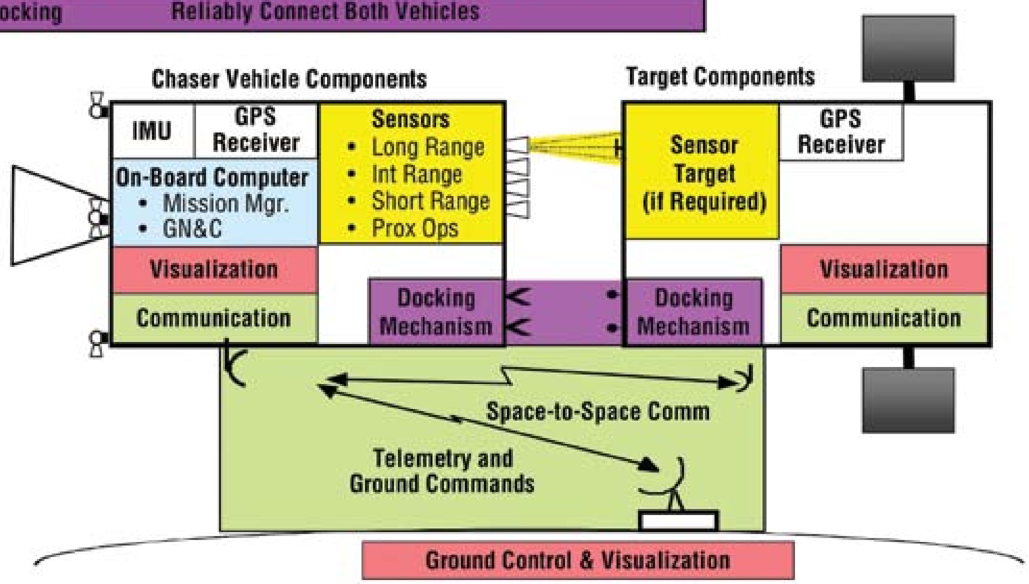


Figure 6-22. An AR&D System's Chaser and Target Components

Although the U.S. has conducted several flight experiments (with more planned), Russia has the only AR&D operationally proven system—Kurs. If the Kurs reliability data is carefully examined to focus on those dockings that were successful only in the automated mode, the reliability of this subsystem is approximately 85 percent. Assuming that the subsystem reliability is only a function of mechanical systems (i.e., that software does not contribute to reliability), a representative reliability allocation to lower-level subsystems can be developed, as shown in **Table 6-2**. When chained across several events, an 85 percent AR&D reliability would not support a viable lunar mission scenario. An unrealistic AR&D subsystem reliability of 99.95 percent (1 failure in 2,000 operations) causes orders of magnitude increase in the Mean Time Between Failure (MTBF) of lower-level subsystems. Based on existing technologies and projected improvements, an AR&D reliability of 95 percent appears realistic, given the hardware and software complexity and operational environment.

Table 6-2. Lower-level Subsystems Reliability Allocation

If AR&D Subsystem Reliability = 0.8500					If AR&D Subsystem Reliability = 0.9995				
Then the Subsystem Reliability Allocation Might Be	3 Dissimilar Sensors	Non-Redundant Control System	Common System	Docking System (10 min On Time)	Then the Subsystem Reliability Allocation Might Be	3 Dissimilar Sensors	Non-Redundant Control System	Common System	Docking System (10 min On Time)
MTBF (hr)	1,500	4,000	7,500	1,000	MTBF Increase Factor	3	100	667	100
# of 12-hour Missions	20	300	600	80	Factor Increase in # of 12-hour Missions	15	10	67	100

6.4.4.3.4 Concatenated Analysis

The previous three sections identified the significant parameters associated with lunar mission preparation. The mission success calculation for the phases prior to leaving Earth orbit requires a concatenation (chain product) of these parameters to determine the statistical LOM. Due to the number of variables, this discussion will focus on three cases that combine these variables into an "optimistic" case, a "most-likely" case, and a "worst-case" expectation. The independent variables include LV average reliability and the number of launches in an architecture. The analysis then assumes that an irrecoverable mission event causes an LOM. Irrecoverable events occur whenever there is an inability to launch all mission elements within the scheduled window, whenever a launch fails to deliver the payload to the destination orbit, or whenever two elements that require AR&D are unsuccessful in the automated mode.

The "optimistic" case results, shown in **Figure 6-23**, were developed to allow the sensitivity of LV reliability to be observed. Unfortunately, to achieve these results, unrealistic values for launch availability of 90 percent, a 3-day average for schedule recycles, a lengthy 5-month mission window, and AR&D automated reliability of 99.95 percent are required. Even a 10-launch architecture, for example, results in an LOM of 1 in 5 for EELV-like reliabilities and 1 in 10 for Shuttle-like reliability. Therefore, launch availability and AR&D reliability are obviously driving parameters that focus the architecture solution toward minimum numbers of launches.

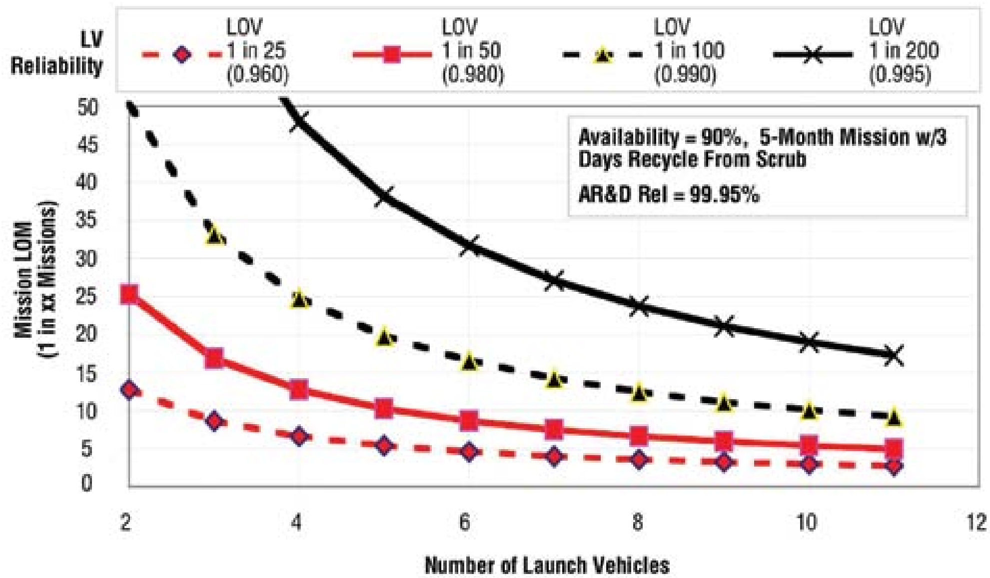


Figure 6-23. LOM Due to LV Reliability, Launch Availability, and On-Orbit Integration (Optimistic Case)

The "most-likely" case results, shown in **Figure 6-24**, should be achievable within current technology projections. Launch availability was assumed to be 80 percent with a 5-day average schedule recycle duration, a 4-month mission window, and an AR&D automated reliability of 95 percent. Here, LV reliability has a reduced role. The curves begin to develop a significant "knee" at a 3-launch architecture.

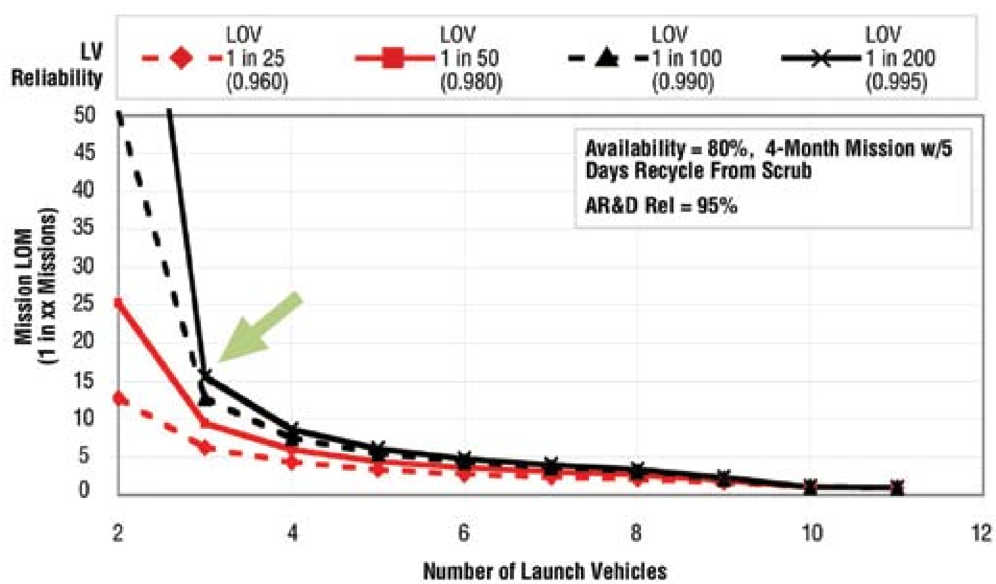


Figure 6-24. LOM Due to LV Reliability, Launch Availability, and On-Orbit Integration (Likely Case)

The "worst-case" results, shown in **Figure 6-25**, approximates Shuttle performance by assuming a launch availability of 60 percent with a 7-day average schedule recycle duration, a 4-month mission window, and Kurs-like AR&D automated reliability of 85 percent. For this case, LV reliability plays a significant role for all 2-launch solutions and the curves begin to develop a significant "knee" at a 3-launch architecture with a LOM of 1 in 5. The combination of docking reliability and inability to fit the launches within the mission window causes an LOM of nearly every attempt for architectures requiring more than eight launches and of every other attempt when six launches are required.

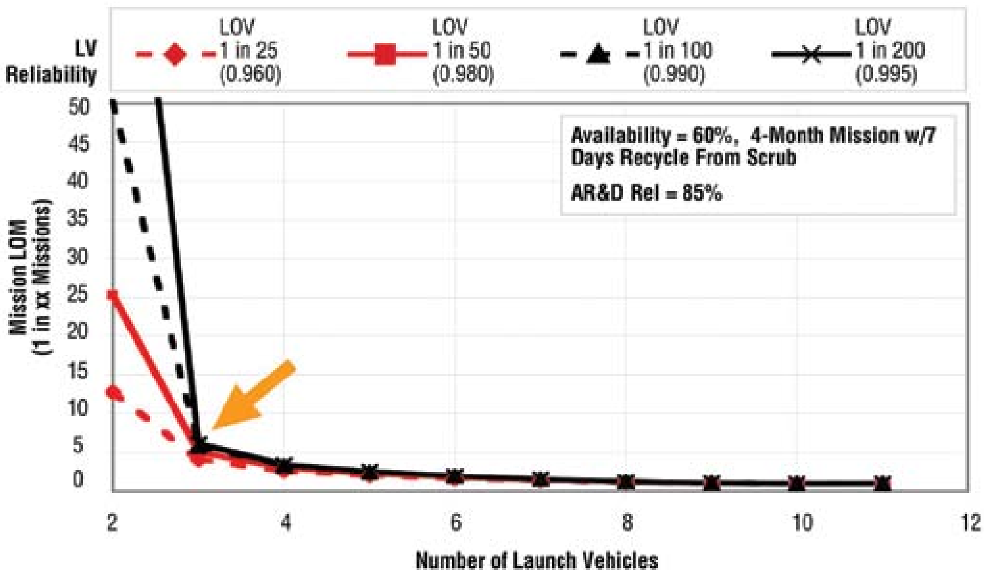


Figure 6-25. LOM Due to LV Reliability, Launch Availability, and On-Orbit Integration (Worst Case)

6.4.4.4 Summary of Results

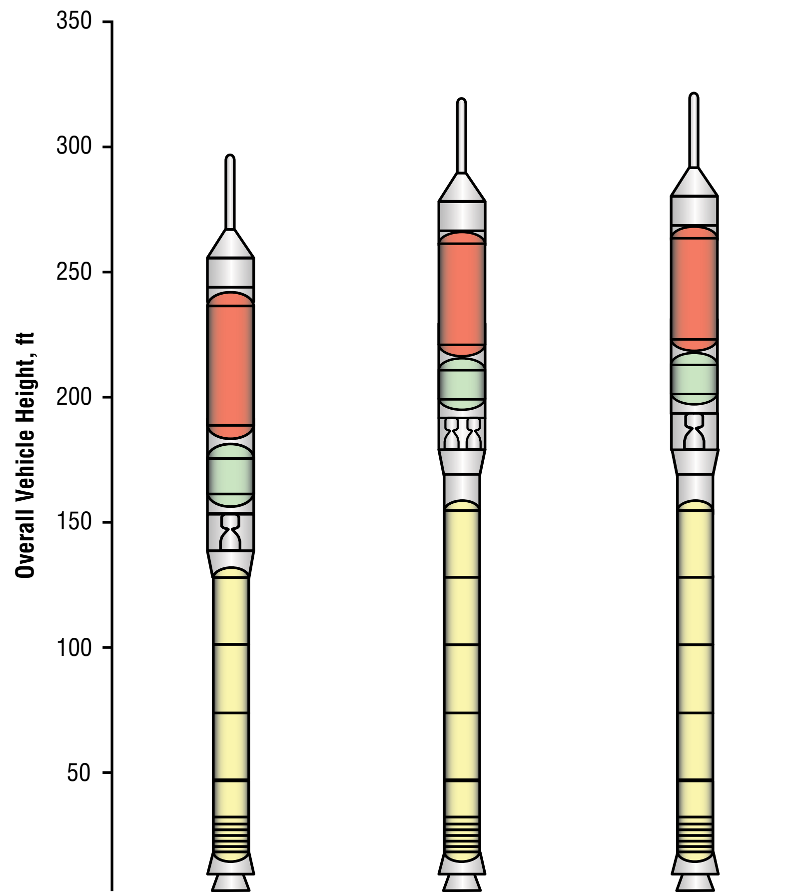
Listed below are the results of the assessment of number of launches and on-orbit assembly:

- Launch scrubs are unfortunately a fact of rocketry. The average time between attempts is as much a function of weather as hardware and software glitches. Reducing hardware complexity reduces scrubs and recycles.
- AR&D operational systems do not currently provide reliable automated performance; only the near presence of human backup pilots on either ISS or in the crew cabin allows the Kurs system to provide high reliability.
- Existing ranges have other, equally time-critical customers. Dedicating a range configuration to support many launches for a single yearly lunar mission is improbable, and expecting the range to support multiple yearly missions can only occur if the range is dedicated to exploration.
- The architecture should limit the numbers of launches to a few (i.e., two) vehicles capable of lifting very heavy payloads. This approach allows adequate time to accommodate vehicle/payload integrations and launch scrub/recycles, minimizes the need for automated rendezvous, and supports exploration traffic growth without requiring a dedicated range.

6.5 Crew Launch Vehicle

An array of options was assessed to determine their individual abilities to meet the stated requirements for the CLV. Those that most closely support the necessary demands are provided here. The remaining CLV options that were not evaluated further are discussed in more detail in **Appendix 6A**, **Launch Vehicle Summary**.

Table 6-3. Shuttle-Derived CLV Options Assessed in Detail



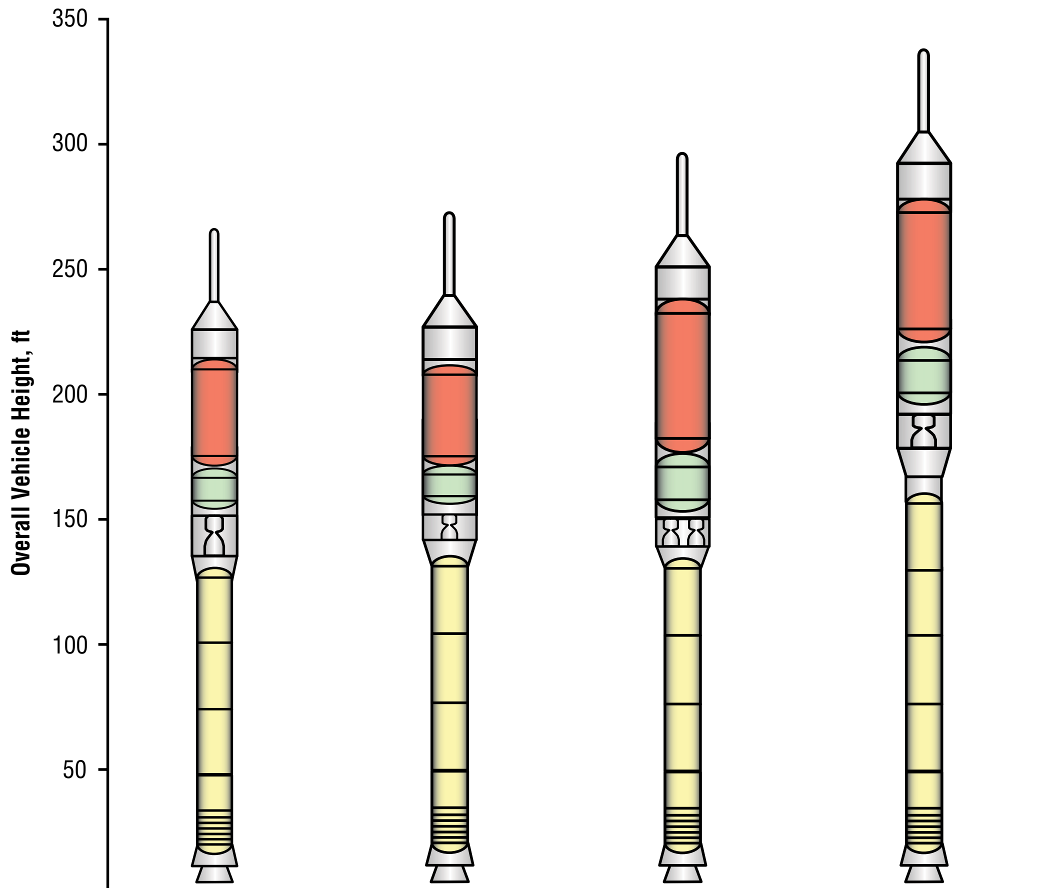
		13.1	15	16
Vehicle Name		**4-Segment SRB with 1 SSME Crew**	**5-Segment SRB with 4 LR–85 Crew**	**5-Segment SRB with 1 J–2S+ Crew**
Payload 28.5 Deg Inc*	**Units**			
Lift Capability	mT	27.2 mT	29.9 mT	28.7 mT
Net Payload	**mT**	**24.5 mT**	**27.0 mT**	**25.8 mT**
Payload 51.6 Deg Inc*				
Lift Capability	mT	25.4 mT	28.1 mT	27.0 mT
Net Payload	**mT**	**22.9 mT**	**25.3 mT**	**24.3 mT**
General Parameters				
Overall Height	ft	290.4 ft	309.4 ft	311.8 ft
Gross Liftoff Mass	lbm	1,775,385 lbm	2,029,128 lbm	2,014,084 lbm
Liftoff Thrust/Weight	G	1.38 g	1.77 g	1.78 g
Second Stage Thrust/Weight	G	1.03 g	0.91 g	0.77 g

**Delivered to 30X160 nmi Orbit*

6.5.1 Candidate LV Options Summary

Table 6-3 provides the four Shuttle-derived options (LV 13.1, LV 15, and LV 16) that werer assessed in detail in this study, including their anticipated dimensions, payload capabilities, and other parameters. The table also includes data for LV 14, LV17.1, LV 17.2, and LV 19.1, which were initially assessed.

Table 6-3. Other Shuttle-Derived Options Initially Assessed

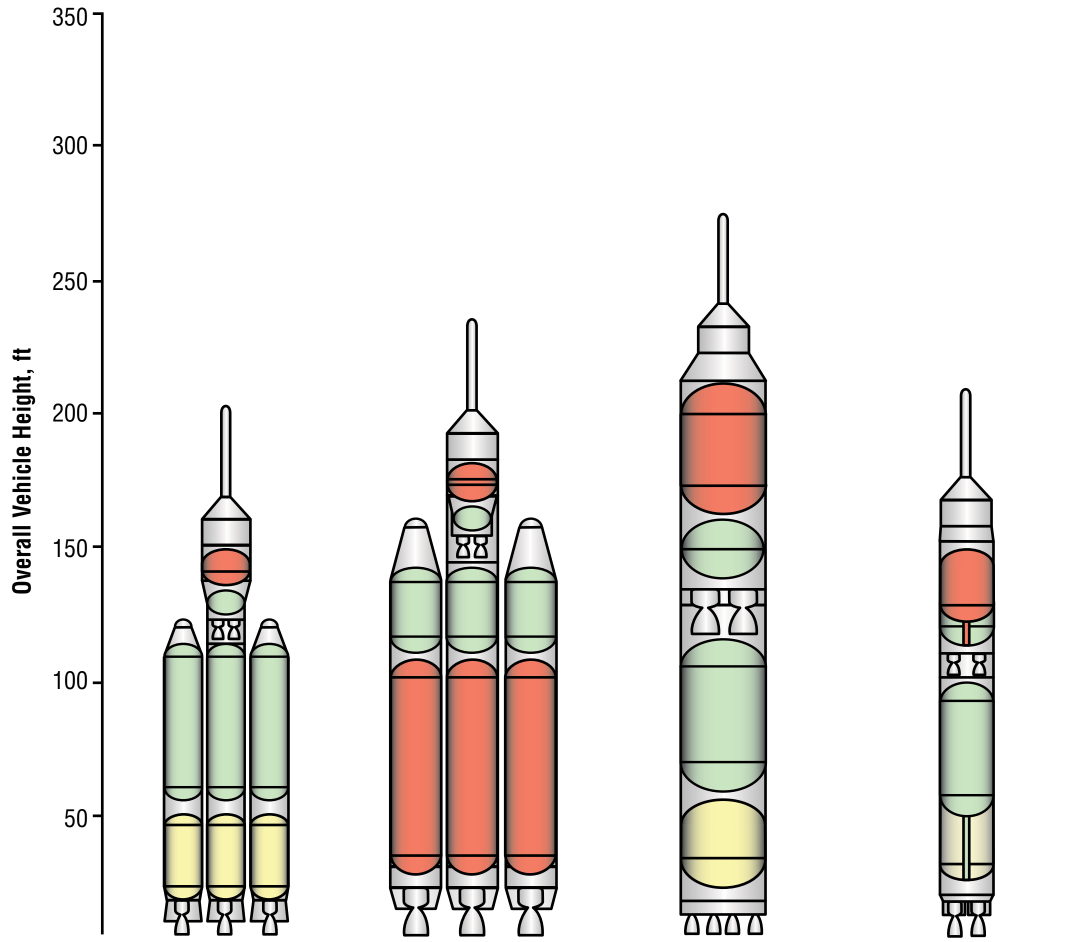


Vehicle Name		14 4 Segment RSRB with 1 J-2S+ Crew	17.1 4-Segment RSRB w/ 1 J–2S (5.5m)– Crew	17.2 4-Segment RSRB w/ 2 J–2S (5.5m)– Crew	19.1 5-Segment RSRB w/ 1 SSME (5.5m) – Crew
Payload 28.5 Deg Inc*	**Units**				
Lift Capability	mT	21.6 mT	18.6 mT	25.3 mT	36.7 mT
Net Payload	**mT**	**19.5 mT**	**16.8 mT**	**22.8 mT**	**33.0 mT**
Payload 51.6 Deg Inc*					
Lift Capability	mT	20.3 mT	17.4 mT	23.6 mT	34.5 mT
Net Payload	**mT**	**18.2 mT**	**15.7 mT**	**21.2 mT**	**31.0 mT**
General Parameters					
Overall Height	ft	267.4 ft	262.9 ft	293.1 ft	329.1 ft
Gross Liftoff Mass	lbm	1,621,814 lbm	1,623,852 lbm	1,813,730 lbm	2,198,812 lbm
Liftoff Thrust/Weight	G	1.51 g	1.51 g	1.35 g	1.63 g
Second Stage Thrust/Weight	G	0.85 g	0.81 g	1.03 g	0.91 g

**Delivered to 30X160 nmi Orbit*

Table 6-4 provides the same information for the four EELV-derived CLV options (LV 2, LV 4, LV 5.1, and LV 9) that were assessed in detail. Also included is data for LV 1 and LV 3.1, which were initially assessed..

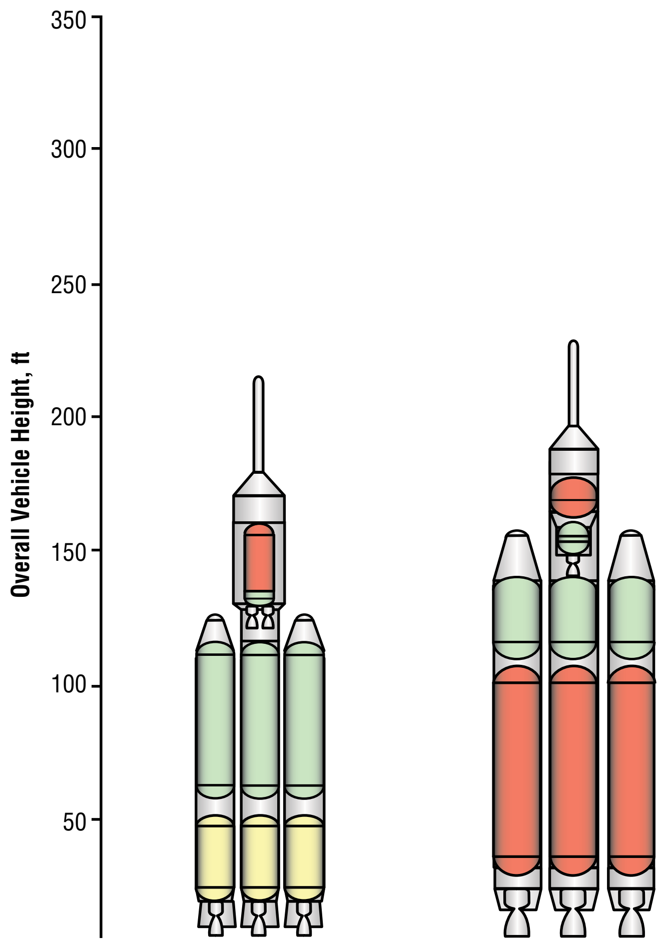
Table 6-4. EELV-Derived CLV Options Assessed



		2	4	5.1	9
Vehicle Name		**Atlas V Heavy New Upper Stage Crew Human Rated**	**Delta IV Heavy New Upper Stage Crew Human Rated**	**Atlas Evolved (5RD–180 & 4 J–2S+) Crew**	**Atlas Phase 2 Crew**
Payload 28.5 Deg Inc*	**Units**				
Lift Capability	mT	33.4 mT	31.6 mT	78.3 mT	28.8 mT
Net Payload	**mT**	**30.0 mT**	**28.4 mT**	**70.4 mT**	**25.9 mT**
Payload 51.6 Deg Inc*					
Lift Capability	mT	29.5 mT	25.5 mT	73.7 mT	27.3 mT
Net Payload	**mT**	**26.6 mT**	**22.9 mT**	**66.4 mT**	**24.5 mT**
General Parameters					
Overall Height	ft	199.1 ft	228.6 ft	265.6 ft	205.7 ft
Gross Liftoff Mass	lbm	2,189,029 lbm	1,698,884 lbm	3,577,294 lbm	1,409,638 lbm
Liftoff Thrust/Weight	G	1.18 g	1.17 g	1.20 g	1.22 g
Second Stage Thrust/Weight	G	0.57 g	0.59 g	1.14 g	0.91 g

**Delivered to 30X160 nmi Orbit*

Table 6-4. Other EELV-Derived Options Initially Assessed



		1	3.1
Vehicle Name		**Atlas V Heavy Crew Human Rated**	**Delta IV HLV Crew Human Rated**
Payload 28.5 Deg Inc*	**Units**		
Lift Capability	mT	26.3 mT	26.5 mT
Net Payload	**mT**	**23.7 mT**	**23.9 mT**
Payload 51.6 Deg Inc*			
Lift Capability	mT	19.9 mT	22.5 mT
Net Payload	**mT**	**17.9 mT**	**20.3 mT**
General Parameters			
Overall Height	ft	207.3 ft	224.9 ft
Gross Liftoff Mass	lbm	2,170,687 lbm	1,663,255 lbm
Liftoff Thrust/Weight	G	1.19 g	1.20 g
Second Stage Thrust/Weight	G	0.37 g	0.19 g

**Delivered to 30X160 nmi Orbit*

6.5.2 FOM Assessments

6.5.2.1 Shuttle-Derived Systems

A summary of the FOMs assessments for the Shuttle-derived CLV candidate vehicles is presented in **Table 6-5**. The assessment was conducted as a consensus of discipline experts and does not use weighting factors or numerical scoring but rather a judgment of high/medium/low (green/yellow/red) factors, with high (green) being the most favorable and low (red) being the least favorable.

Table 6-5. Shuttle-Derived CLV FOMs Assessment Summary

	LV	Shuttle-derived CLV		
		4-Segment RSRB with 1 SSME	5-Segment RSRB with 4 LR-85s	5-Segment RSRB with 1 J–2S+
		13.1	15	16
FOMs	Probability of LOC	1 in 2,021	1 in 1429	1 in 1,918
	Probability of LOM	1 in 460	1 in 182	1 in 433
	Lunar Mission Flexibility			
	Mars Mission Extensibility			
	Commercial Extensibiity			
	National Security Extensibility			
	Cost Risk			
	Schedule Risk			
	Political Risk			
	DDT&E Cost	1.00	1.39	1.30
	Facilities Cost	1.00	1.00	1.00

The Shuttle-derived options were assigned favorable (green) ratings in the preponderance of the FOMs, primarily due to the extensive use of hardware from an existing crewed launch system, the capability to use existing facilities with modest modifications, and the extensive flight and test database of critical systems—particularly the RSRB and SSME. Each Shuttle-derived CLV concept exceeded the LOC goal of 1 in 1,000. The use of the RSRB, particularly the four-segment, as a first stage provided a relatively simple first stage, which favorably impacted LOC, LOM, cost, and schedule risk. The introduction of a new upper stage engine and a five-segment RSRB variant in LV 15 increased the DDT&E cost sufficiently to warrant an unfavorable (red) rating. The five-segment/J–2S+ CLV (LV 16) shares the DDT&E impact of the five-segment booster, but design heritage for the J–2S+ and the RSRB resulted in a more favorable risk rating.

Applicability to lunar missions was seen as favorable (green), with each Shuttle-derived CLV capable of delivering the CEV to the 28.5-deg LEO exploration assembly orbit. Extensibility to commercial and DoD missions was also judged favorably (green), with the Shuttle-derived CLV providing a LEO payload capability in the same class as the current EELV heavy-lift vehicles.

The five-segment RSRM/one-SSME SDV CLV variant (LV 19.1) was not considered in the final selection process because it had performance significantly in excess of that required for the ESAS CEV concepts. However, it is viewed as a viable follow-on upgrade. LV 17.2, with a four-segment first stage and 2 J–2s in the upper stage, was not selected because it does not support maintaining the SSME needed for the cargo vehicle. Its performance was below that needed for using a single SSME, and it was judged not capable of being ready for flight by 2011, and was high risk for being ready in 2012. LV14 variant using a four-segment RSRM first stage and a single J–2S+ in the upper stage did not meet the CLV performance goals and was dropped from consideration.

6.5.2.2 EELV-Derived Systems

A summary of the FOMs assessment for the EELV CLV candidate vehicles is presented in **Table 6-6**. The assessment was conducted using the same rating system as for the Shuttle-derived systems.

Table 6-6. EELV-Derived CLV FOMs Assessment Summary

	LV	EELV-derived CLV			
		Atlas V HLV New Upper Stage Human-Rated	Atlas Evolved Crew	Atlas Phase 2 Crew	Delta IV HLV New Upper Stage Human-Rated
		2	5.1	9	4
FOMs	Probability of Loss of Crew	1 in 957	1 in 614	1 in 939	1 in 1,100
	Probability of Loss of Mission	1 in 149	1 in 79	1 in 134	1 in 172
	Lunar Mission Flex-ibility				
	Mars Mission Extensibility				
	Commercial Extensibiity				
	National Security Extensibility				
	Cost Risk				
	Schedule Risk				
	Political Risk				
	DDT&E Cost	1.18	2.36	1.73	1.03
	Faciities Cost	0.92	0.92	0.92	0.92

For the EELV-derived vehicles, the FOMs for flexibility for lunar missions and extensibility to commercial and DoD applications scored well. Because the Delta IV and Atlas V heavy-lift LV families were originally designed for DoD and commercial applications, particularly Geosynchronous Transfer Orbit (GTO) missions, the development of a new upper stage would only improve their capabilities in these areas.

Most EELV-derived CLVs came close to the goal of 1 in 1,000 LOC, but with less margin than the RSRM-derived options. The Atlas Phase 2 and Atlas-evolved CLVs utilize new multi-engine first stages, which require new tankage, avionics, and Main Propulsion Systems (MPSs). Of these two (5.1 and 9), the Atlas Phase 2 ranked higher for LOC, due to the lesser complex-ity of its first stage, with two engines. The human-rated Atlas V and Delta IV HLV CLVs with new upper stages (2 and 4) were evaluated to be safer and more reliable than the multi-engine first stage options, but the more complex strap-on staging event introduced failure modes that impacted LOC and LOM.

No EELV CLV candidate was judged to exhibit a favorable (green) rating for the risk incurred relative to cost and schedule. The determination was made that these options would be higher risk for a CEV IOC by 2011. The Atlas V HLV and Delta IV HLV share common risk areas of significant rework and modification for human rating and the development of a new multi-engine upper stage. The fact that Delta IV HLV has flown, while Atlas V HLV has not, would impact its relative cost and schedule risk. The Atlas Phase 2 would also require a new upper stage engine—adding to the cost and schedule risk.

The modified Delta IV and Atlas V HLV vehicles were evaluated to be favorable (green) in DDT&E costs, largely due to design heritage. Facilities modifications were judged to be in a similar scope to those required for a Shuttle-derived LV, and rated favorable (green). The new core options (5.1 and 9) have very high DDT&E costs, resulting in a low (red) rating.

6.5.3 Detailed Assessment Summary

6.5.3.1 Descriptions of Selected CLV

CLV variant, LV 13.1, (**Figure 6-26**) is a two-stage, series-burn LV for CEV launch. The first stage is a four-segment RSRB with Polybutadiene Acrylonitride (PBAN) propellant. The concept was designed with a 10 percent reduction in the burn rate of the four-segment RSRB to reduce the maximum dynamic pressure the LV achieves on ascent. Earlier configurations similar to LV 13.1 with smaller LOX/LH2 second stages experienced maximum dynamic pressures greater than 1,000 psf. It was deemed desirable for crewed launches that this parameter be reduced to more benign conditions. Therefore, the reduced burn rate for the four-segment RSRB was implemented for all two-stage configurations of this type. (Later studies have shown this modification will not be required to achieve a reasonable maximum dynamic pressure.) The second stage for LV 13.1 is LOX/LH2 with one SSME for propulsion. This vehicle is flown to 30- by 160-nmi orbits at inclinations of 28.5 deg and 51.6 deg and inserted at an altitude of 59.5 nmi. The SSME is run at a throttle setting of 104.5 percent. The purpose of this analysis was to evaluate the performance of the SSME, modified for altitude-start, as an upper stage engine in comparison to a modified J–2S (J–2S+) engine.

6.5.3.2 Performance Summary

The net payload capability of LV 13.1 is 24.5 mT to a 30- by 160-nmi orbit at a 28.5 deg inclination. The net payload to 30- by 160-nmi at a 51.6 deg inclination is 22.9 mT. No GR&As were violated for this LV analysis. Special considerations required to analyze this vehicle included: (1) SSME was ignited at altitude and (2) a 10 percent reduction in the burn rate for the four-segment RSRB. (However, later more detailed assesments have shown this modification will not be necessary.)

Figure 6-26. LV 13.1 General Configuration

6.5.3.2.1 Vehicle Sizing

The mass properties for the second stage of LV 13.1 are shown in **Table 6-7**, calculated using the Integrated Rocket Sizing Program (INTROS). The mass properties for the four-segment RSRB were used as delivered with only two modifications. The current Solid Rocket Booster (SRB) nosecone was removed and an interstage was added to complete the vehicle configuration.

Table 6-7. LV 13.1 INTROS Mass Summary

Mass Properties Accounting		
Vehicle: Four-Segment SRB with 1 SSME Crew – Blk 2 Stage: Second (1 SSME)		
Item	Mass Subtotals	Mass Totals
	lbm	lbm
Primary Body Structures	17,147	
Secondary Structures	960	
Separation Systems	136	
TPSs	75	
TCSs	1,198	
MPS	12,501	
APS	203	
Power (Electrical)	1,868	
Power (Hydraulic)	415	
Avionics	513	
Miscellaneous	126	
Stage Dry Mass Without Growth		**35,142**
Dry Mass Growth Allowance	3,455	
Stage Dry Mass With Growth		**38,597**
Residuals	3,610	
Reserves	2,747	
In-flight Fluid Losses	69	
Stage Burnout Mass		**45,022**
Main Ascent Propellant	360,519	
Engine Purge Helium	41	
Reaction Control System (RCS) Ascent Propellant	300	
Stage Gross Liftoff Mass		**405,882**
Stage: First (Four-Segment SRB)		
Stage Burnout Mass		**188,049**
Main Ascent Propellant	1,112,256	
Stage Gross Liftoff Mass		**1,300,305**
Net Vehicle Total		
Payload	59,898	
LAS	9,300	
Upper Stage(s) Gross Mass	405,882	
Net Vehicle Gross Liftoff Mass		**1,775,385**

6.5.3.2.2 Structural Analysis

Figure 6-27 shows the CLV structural configuration. The loads plot (**Figure 6-28**) is a combined worst-case including prelaunch, liftoff, maximum dynamic pressure (max q), and maximum acceleration (max g). The compression loads show a major jump where the LOX tank loads are integrated into the outside structure. The bending moment shows a steady increase from the tip progressing aftward.

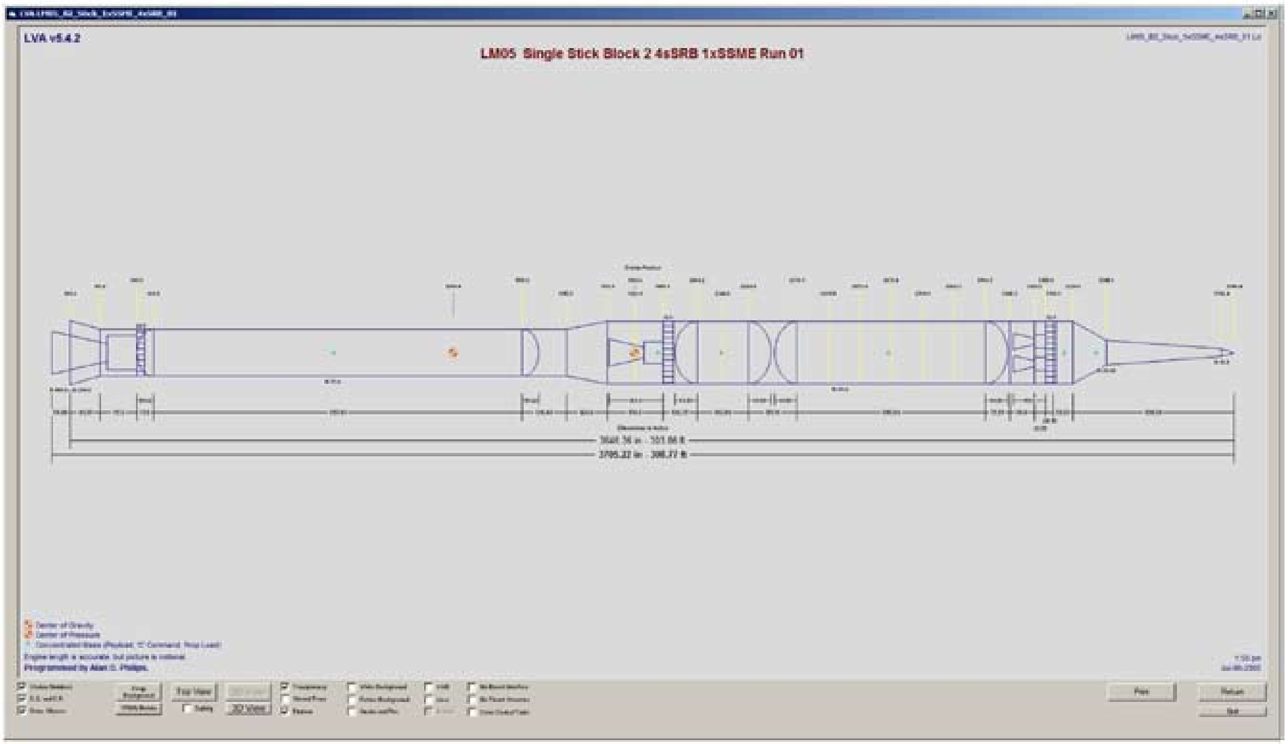
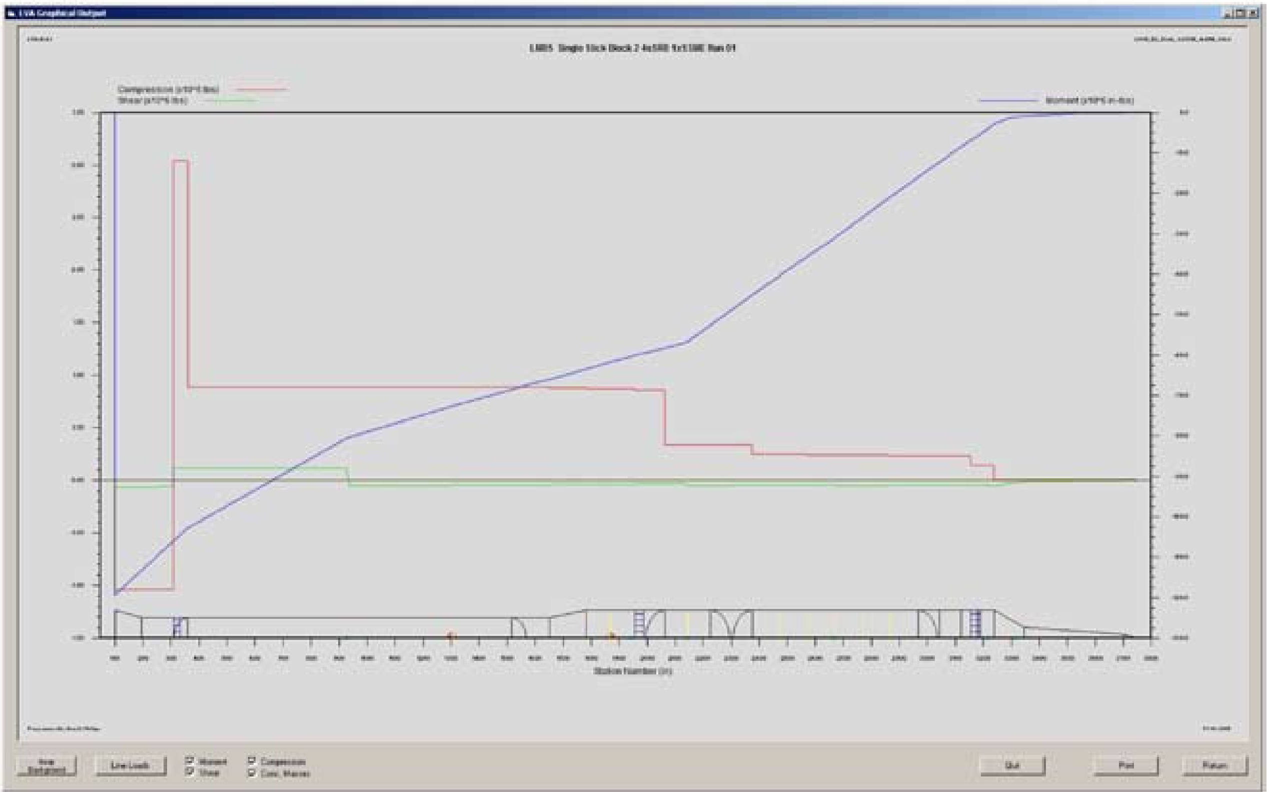


Figure 6-27. CLV Structural Configuration

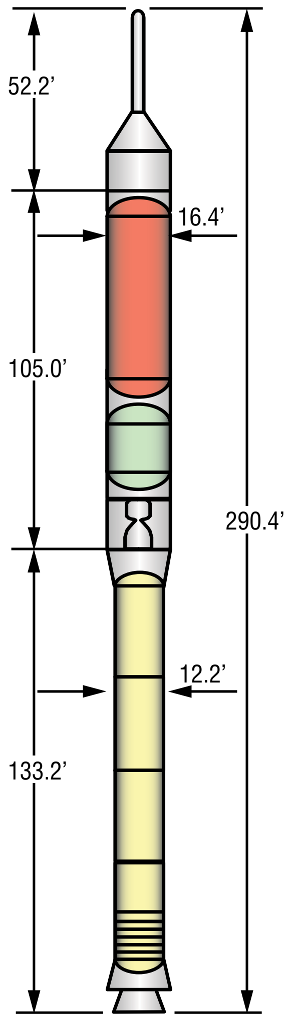


6.5.3.2.3 Flight Performance Analysis and Trajectory Design

Figure 6-28. CLV Structural Loads Analysis Results

The closed-case trajectory summary results and LV characteristics are shown in **Figure 6-29**. Selected trajectory parameters are shown in **Figures 6-30** through **6-33**. The vehicle exhibits a 1.38 T/W ratio at liftoff. The maximum dynamic pressure is 576 psf at 59.2 sec in the flight. The maximum acceleration during the first stage is 2.26 g's and is 4.00 g's during the second stage. Staging occurs at 145.3 sec into the flight at an altitude of 166,694 ft and Mach 4.16. The T/W ratio at second-stage ignition is 1.03. Orbital injection occurs at 478.7 sec at 59.5 nmi.

4-Segment SRB with 1 SSME Crew



Vehicle Concept Characteristics

GLOW	**1,775,385 lbf**
Payload	5-m diameter CEV
Launch Escape System	9,300 lbm
Booster Stage (each)	
Propellants	PBAN
Useable Propellant	1,112,256 lbm
Stage pmf	0.8554
Burnout Mass	188,049 lbm
# Boosters / Type	1 / 4-Segment SRM
Booster Thrust (@ 0.7 secs)	3,139,106 lbf @ Vac
Booster Isp (@ 0.7 secs)	268.8 sec @ Vac
Second Stage	
Propellants	LOX/LH2
Useable Propellant	360,519 lbm
Propellant Offload	0.0 %
Stage pmf	0.8882
Dry Mass	38,597 lbm
Burnout Mass	45,022 lbm
# Engines / Type	1 / SSME
Engine Thrust (100%)	469,449 lbf @ Vac
Engine Isp (100%)	452.1 sec @ Vac
Mission Power Level	104.5%
Delivery Orbit	30 x 160 nmi @ 28.5°
Delivery Orbit Payload	59,898 lbm 27.2 mT
Net Payload	53,908 lbm 24.5 mT
Insertion Altitude	59.5 nmi
T/W @ Liftoff	1.38
Max Dynamic Pressure	576 psf
Max g’s Ascent Burn	4.00 g
T/W Second Stage	1.03
Delivery Orbit	30 x 160 nmi @ 51.6°
Delivery Orbit Payload	56,089 lbm 25.4 mT
Net Payload	50,480 lbm 22.9 mT

Summary data for reference mission (30 × 160 nmi @ 28.5):

liftoff to SRM staging
max SRM accel = 2.26

time of max Q = 59.24 sec
max Q = 576 psf
mach = 1.13

after SRM jettison (core only)
tstg = 145.30 sec
alt@stg = 166,694 ft
mach@stg = 4.16

dynp@stg = 20 psf
dV1 = 8,430 ft/s
max core f/w = 4.00

LES jettison @t = 175.3 sec
alt @ jettison = 211,660 ft

at MECO / orbital insertion
time to MECO = 478.7 sec
MECO altitude = 361,539 ft
dVt = 30,046 ft/s

Figure 6-29. LV 13.1 Summary

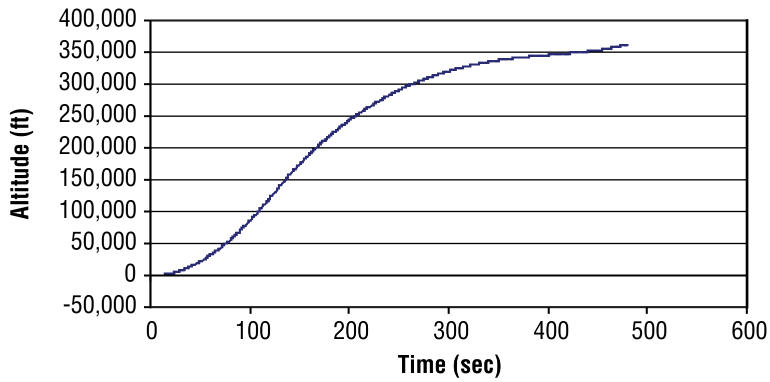


Figure 6-30. Altitude versus Time

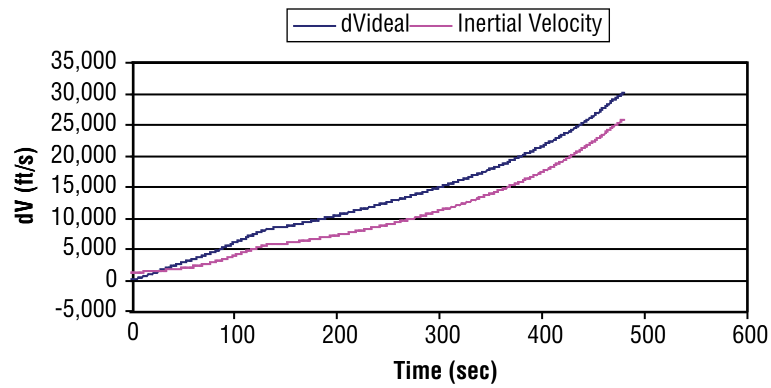


Figure 6-31. Velocity versus Time

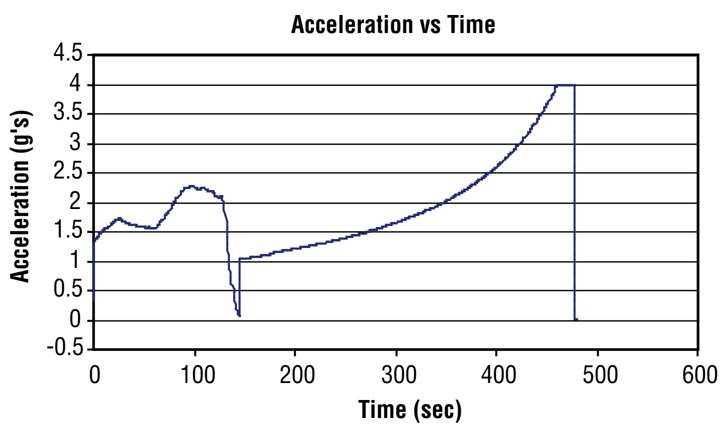


Figure 6-32. Acceleration versus Time

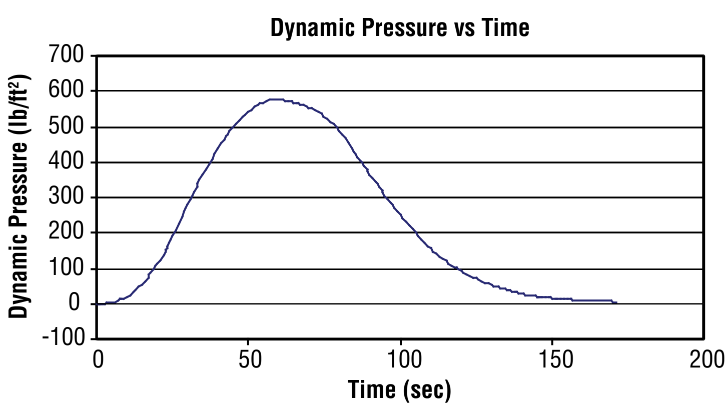


Figure 6-33. Dynamic Pressure versus Time

6.5.3.3 Cost Analysis Assumptions for CLVs (LV 13.1, LV 15, LV 16)

6.5.3.3.1 Inputs

The booster stage for these CLVs is either a four-segment RSRB or a five-segment RSRB. The four-segment RSRB is in production today. While the five-segment will draw heavily from the four-segment, some DDT&E will be needed.

Upper stages are used to deliver the payload to the desired orbit. In general, all of the upper stages are considered new designs using existing technology.

Structure and Tanks

Both metallic and composite intertanks, interstages, and thrust structures have been used on various programs. Design and manufacturing capabilities exist today. The critical elements will be the development of the separation system, a new interstage, and the payload adapter. Material is either 2219 aluminum or AL-Li. Shrouds are made of graphite-epoxy panels, based on Titan and Delta IV designs. Structures and tanks are well understood with sufficient manufacturing capability in existence. All structures have similar subsystems to EELV, Shuttle, or ET. The NASA and Air Force Cost Model (NAFCOM) cost estimate assumptions assumed a new design with similar subsystems validated in the relevant environment. Full testing and qualification will be required.

MPS—Less Engine

The MPS will take significant heritage from the existing SSME MPS subsystem. However, a new design is needed to accommodate one SSME. NAFCOM cost estimates assumed a new design with similar subsystems validated in the relevant environment. Full testing and qualification will be required.

Both the J–2S and LR–85 engines are equivalent to new engines, due to the length of time that has passed since the J–2 was in production, and the LR–85 is currently on paper only. Each will take heritage from the previously existing engine, but the MPS on the upper stage will be new. NAFCOM cost estimates assumed a new design with similar subsystems validated in the relevant environment. Full testing and qualification will be required.

Engine – SSME

Altitude-Start SSME

A 1993 study (NAS8–39211) and a 2004 Marshall Space Flight Center (MSFC) study examined the Block 2 engines for altitude-start. Both studies determined altitude-start will require minor changes, but is considered straightforward. Specialized testing for certification to the environment will be required. Development and certification of altitude-start for the Block 2 RS–25d engine is needed. The cost estimate is based on SSME historical actuals, vendor quotes, and estimates. It also assumes the Shuttle Program continues to pay the fixed cost of infrastructure through Shuttle termination.

Current Inventory SSME

At the conclusion of the STS Program, there will be 12 Block 2 (RS–25d) engines in inventory if the 28-flight manifest occurs, or 14 engines in inventory with a 16-flight manifest. In either case, the program plans to use at least 12 of the existing Block 2 assets for the early flights. Assembly, handling, and refurbishment of the existing engines and conversion of the reusable engine for upper stage use will be needed. Excluded from these costs are any sustaining engineering or Space Shuttle Program (SSP) hardware refurbishment. These early flights will incur some operations costs, which are yet to be determined.

Minimal Changes for Expendable Applications SSME

In addition to the minor changes required to altitude-start the SSME (RS–25d), it is desirable to make some engine improvements to lower the unit cost and improve producibility. Suggested improvements include low-pressure turbomachinery simplifications; a new controller; a Hot Isostatic Press (HIP) bonded Main Combustion Chamber (MCC); flex hoses to replace flex joints on four ducts; and simplified nozzle processing. In addition, process changes would be incorporated to eliminate inspections for reuse and accommodate obsolescence of the controller. Development and certification of these minimal changes is designated SSME RS–25e. The estimate is based on SSME historical actuals, vendor quotes, and estimates.

Engine: J–2S

Two different variants of the J–2S were analyzed for this study. The first assumed a design as close as possible to the original Apollo-era J–2S. The second variant was a J–2S redesign, specifically designed for optimal reliability and low production costs. Either could be used with a larger area ratio nozzle. Once again, cost analysis was performed using a bottom-up approach. All production costs were derived assuming a manufacturing rate of six engines per year.

Engine: LR–85

LR–85 is a conceptual design engineered to meet derived requirements from the program Human-Rating Plan. Production of the LR–85 was assumed to use domestic production capabilities. Parametric analysis was performed on the engine using the Liquid Rocket Engine Cost Model (LRECM). Major cost drivers to this model are the Isp and thrust. Options are available to include heritage from older engines.

Appropriate rate curves were applied to both manufacturing and refurbishment to reflect dynamics of the engine production rates with respect to the largely fixed nature of the costs. Theoretical First Unit (TFU) costs from NAFCOM or vendor data were used as a baseline point in the analysis. Historic RS–68, RL–10, and SSME data was also used to help generate Productivity Rate Curves (PRCs).

Avionics and Software

The avionics subsystem must support Fail Operational/Fail Safe vehicle fault tolerant requirements. Upon the first failure, the vehicle will keep operating. The second failure will safely recommend an abort. Crew abort failure detection and decision-making capabilities have been demonstrated and are ready for flight. All architectures will meet these requirements, either by adding a modification for instrumentation redundancy for the EELV health management system, or by providing the capabilities through the new design of the avionics for Shuttle-derived configurations.

Avionics hardware is divided into Guidance, Navigation, and Control (GN&C), and Command, Control, and Data Handling (CCDH). GN&C provides for attitude control, attitude determination, and attitude stabilization. CCDH provides all the equipment necessary to transfer and process data; communication for personnel, as well as spacecraft operations/telemetry data; and instrumentation for monitoring the vehicle and its performance. Both systems are tied together through the LV software system. LV hardware requirements are well understood.

During the benchmarking activity for NAFCOM, it was discovered that the Cost Estimating Relationships (CERs) for avionics were significantly different from the contractors' data. This difference led to NAFCOM developers reviewing the database and statistical analysis of the avionics CERs. One result of this exercise was to drop very old avionics data points as unrepresentative of modern avionics. In addition to the CER adjustment, the avionics Mass Estimating Relationships (MERs) used in the INTROS LV sizing program were revised. Previous MERs were derived from STS data, Centaur stage data, Shuttle C, Heavy-Lift Launch Vehicle (HLLV), and other studies, leading to a much heavier weight input into NAFCOM than would be expected with modern electronics. In recent years, avionics have changed considerably due to advances in electronics miniaturization and function integration. State-of-the-art avionics masses are considerably less than what was previously used in INTROS. Revised MERs were developed for GN&C, actuator control, Radio Frequency (RF) communications, instrumentation, data management/handling, and range safety. The revised MERs were used within NAFCOM as one input into the multivariate CERs.

The core booster does not guide and control the ascent. This function is in the upper stage. Core booster avionics include translators, controllers, Analog-to-Digitial (AD) converters, actuator control, electronics, and sufficient CCDH hardware to interface with the upper stage. The upper stage avionics control ascent, separations, and flight. Upper stage avionics hardware includes the Inertial Measuring Unit (IMU), processors, communication, telemetry, and instrumentation. Software provides the separation commands, software for general flight, mission-specific flight algorithms, and launch-date-specific software.

Software also provides the commands that control the vehicle, viewed as one entity for the LV. As such, the software estimate is not divided between the core and upper stage. Software is normally located on the upper stage since it is the upper stage that controls the ascent of the LV. The software estimate for the LVs is based on a detailed breakdown of the functional requirements, which is provided in **Table 6-8**.

Table 6-8. Functional Breakout of Software Lines-of-Code (SLOC) Estimates

Events Manager (50 Hz) (approximately 500 to 1,000 SLOC)
Manage Events Sequencer
Manage Events Updates
Navigation Manager (50 Hz) (approximately 8,000 to 15,000 SLOC)
Provide Translational Navigation Estimates
Provide Rotational Navigation Estimates
Guidance Manager (1 Hz) (approximately 15,000 to 25,000 SLOC)
Ascent Mode Provide Open-Loop Guidance Provide Closed-Loop Guidance Provide Circularization Guidance **Abort Mode** Provide Ascent Abort (IIP) (50 Hz) (Flight planning for avoiding undesirable landing areas using reduced capability) *Note: This could contain added capability; currently no defined requirements.*
Control Manager (50 Hz) (approximately 8,000 to 15,000 SLOC)
Manage Stage Separation Control
Manage Ascent Vehicle Control
Manage RCS Control

Table 6-8. (continued) Functional Breakout of Software Lines-of-Code (SLOC) Estimates

Command and Data Manager (50 Hz) (approximately 28,000 to 40,000 SLOC)
Initialize Software
Initialize Hardware
Provide Payload Interface
Provide Sensor Interface (GPS, INS, Gyro)
Provide Telemetry Data
Provide Ground Interface
Provide Engine Controller Interface
Provide Upper Stage Controller Interface
Provide Booster Interface Unit Interface
Provide TVC Controller Interface
Provide Flight Termination System Interface
Note: This assumes a limited fault detection and notification/recovery capability.
Time Manager (50 Hz) (approximately 1,500 to 2,000 SLOC)
Provide Time
Power Manager (25 Hz) (approximately 2,500 to 4,000 SLOC)
Provide Power System Management
Vehicle Management Software (110K SLOC ± 50%)
Abort Management System (70K SLOC ± 50%) Trajectory Replan Requests (10K SLOC) • Engine Operation • Stage Separation Status Payload (10K SLOC) • Abort Conditions • Health Indications Determination of Proper Scenario (50K SLOC) • Burn Remaining Engines Longer • Separate Upper Stage Early
Launch Pad Interface (15K SLOC ± 50%) Data Gathering Communication with Launch Pad—ability to diagnose health of engine Fault Identification on Vehicle
Onboard FTS Tracking (25K SLOC ± 50%) Trajectory Following RT Position Monitoring Compare Position Monitoring Abort Scenario Updates • Trajectory Modifications • Flight Termination Delay Communication with Range Safety to Request Flight Termination
Total Flight Software SLOC estimate: 48,500 to 102,000 **Vehicle Management included: 55,000 to 165,000** **Total: 103,000 to 267,000**

Note: This estimate does not include Backup Flight Software (BFS). BFS estimated at 45,000 SLOC.

Software estimates are based on the above maximum SLOC, using the Software Estimation Model (SEER–SEM) tool for software estimation, planning, and project control. SEER–SEM is a recognized software estimation tool developed by Galorath Incorporated for use by industry and the Government.

Shuttle-Derived Avionics Hardware
The GN&C and CCDH subsystems for Shuttle-derived LVs are considered new designs. Because the subsystems and software are new, integrated health management and human-rating requirements are incorporated from the start. The avionics hardware assumed a new design with existing technology.

Shuttle-Derived Software
All Shuttle-derived software is considered a new software development, incorporating the functions identified above. The maximum SLOC estimate were used with the SEER-SEM model to arrive at a deterministic software estimate.

Other Subsystems
The basic thermal systems are ½- to 1-inch Spray-on Foam Insulation (SOFI), with cold plates and insulation for passive cooling of equipment and avionics. No new technology is planned. Heritage has normally been given to the thermal subsystem because it is well understood and used on existing systems today.

Electrical power is provided by silver-zinc batteries with a redundancy of two. Conversion, distribution, and circuitry are considered new designs with state-of-the-art technology. Hydraulic power is fueled by hydrazine, which is used in LVs today.

RCSs, when used, are the same type as those used in the Shuttle. Range safety will require modifications to the flight termination system to add a time-delay for abort. Human-rating requirements may require the removal of the autodestruct capability. All of these subsystems are similar to those already in existence, either on EELVs or Shuttle, and have been validated in the relevant environment. Full qualification and testing is estimated for all crew and cargo vehicles.

6.5.3.3.2 DDT&E
The lowest cost option, as shown in **Table 6-9**, uses the existing four-segment RSRB and the modified SSME. Of the two five-segment configurations, the vehicle that uses only one J–2S engine is cheaper than the vehicle that requires four LR–85s.

6.5.3.3.3 Production
LV 13.1, LV 15, and LV 16 are single SRB-based crew vehicles, with either a four- or five-segment booster modified from the current Shuttle SRBs. As described above, the modifications will enable the integration of the booster with an upper stage. The recurring production costs of these three concepts are very close and are within the accuracy of the model. Although the four-segment SRM is slightly cheaper to refurbish than the five-segment version (the cost of refurbishing and reloading a single motor segment is relatively small), the cost of the Expendable Space Shuttle Main Engine (eSSME) equipped upper stage more than offsets this savings, so that LV 13.1 has the highest recurring production cost.

6.5.3.3.4 Launch Operations

All of these concepts require the stacking of either a four- or five-segment SRB with a modified forward skirt and an interface to the interstage. The SRM segments are refurbished in the same manner as in the current Shuttle operation (described previously in **Section 6.5.3.3.3**, **Production**). A portion of the interstage is also a refurbished item. The upper stage, upper stage engine, and part of the interstage are newly manufactured hardware. The launch operations activities include receipt, checkout, stacking and integration, testing, transport to the launch pad, pad operations, and launch. As shown in **Table 6-9**, The cost of launch operations is lowest for LV 16 and greatest for LV 15. However, the difference at six flights per year is slight.

Phase	Relative Cost Position		
Vehicle	13.1	15	16
DDT&E	1.00	1.39	1.30
Production	1.00	0.92	0.93
Operations	1.00	1.03	0.85
Facilities	1.00	1.00	1.00

Table 6-9. Relative Comparison of SDV Crew Vehicle Costs

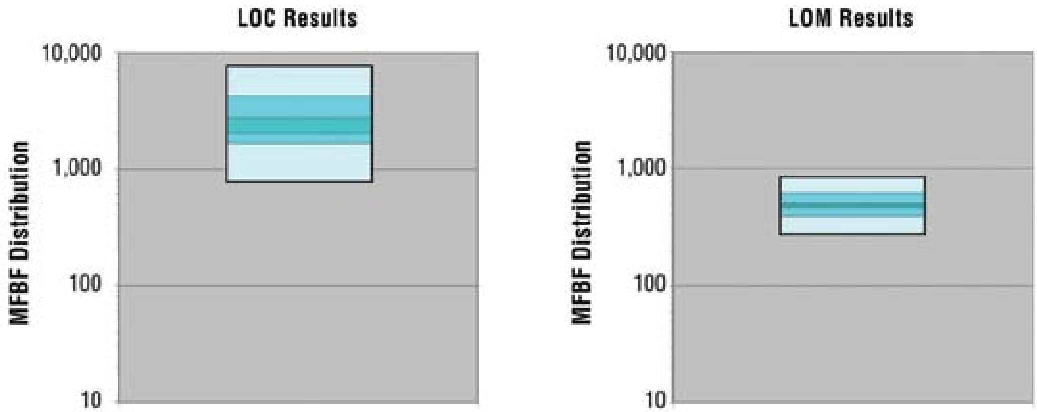
6.5.3.3.5 Facilities

The facilities costs include modifications to the Mobile Launch Platform (MLP), VAB, and launch pad to accommodate the different profile and footprint of the in-line SRB configuration. The cost is the same for all three concepts, as shown in **Table 6-9**.

6.5.3.4 Safety/Reliability Analysis (LV 13.1)

The Flight-Oriented Integrated Reliability and Safety Tool (FIRST) reliability analysis tool was used to determine the LOM and LOC estimates for the four-segment SRB with one SSME (RS–25) upper stage CLV (LV 13.1). These estimates were based on preliminary vehicle descriptions that included propulsion elements and a Space Shuttle-based LV subsystem with updated reliability predictions to reflect future testing and design modifications and a mature LV 13.1. A very simple reliability model using point estimates was used to check the results. A complete description of both models is included in **Appendix 6D**, **Safety and Reliability**. Likewise, a complete description of how reliability predictions were developed for the individual LV systems that were used in the analyses is provided in **Appendix 6D**, **Safety and Reliability**. LV 13.1 LOM and LOC estimates are shown in **Figure 6-34**. The results are for ascent only, with LOC calculated assuming an 80 percent Crew Escape Effectiveness Factor (CEEF) for catastrophic failures and a 90 percent CEEF for noncatastrophic failures. Also, the model applied a Command Module CEEF = 0 percent, but this may prove to be overly conservative as CM designs evolve. Other key assumptions included:

- No mission continuance engine-out capability on upper stage;
- Because second-stage engine shutdown or failure to start (altitude-start) is catastrophic to the vehicle, the model applies a CEEF of 80 percent;
- No mission continuance engine-out capability; and
- SSME is operated with current redlines enabled, but adjusted for altitude-start.



LOC 95th	LOC 75th	LOC (mean)	LOC 50th (median)	LOC 25th	LOC 5th
1 in 775	1 in 1675	1 in 2021	1 in 2711	1 in 4200	1 in 7610

LOM 95th	LOM 75th	LOM (mean)	LOM 50th (median)	LOM 25th	LOM 5th
1 in 273	1 in 394	1 in 460	1 in 500	1 in 626	1 in 850

Figure 6-34. LV 13.1 LOC and LOM Estimates

The reliability analysis used Space Shuttle Probabilistic Risk Assessment (PRA) data as a baseline that was reviewed by propulsion engineers to incorporate potential upgrades for this vehicle. This led to the propulsion system reliability estimates shown in **Table 6-10**.

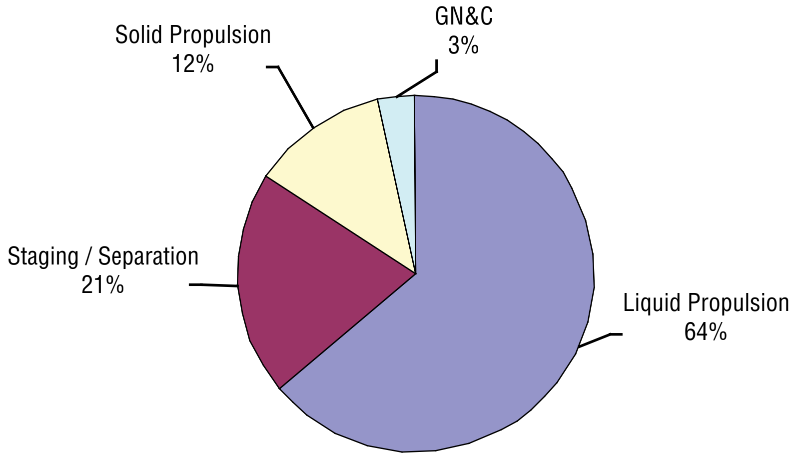
Table 6-10. LV 13.1 Propulsion System Failure Probabilities

Engine	Failure Probability (Cat)	Failure Probability (Ben)	Failure Probability (Start)	CFF	Error Factor
SSME	2.822E-04	1.482E-03	3.000E-04	16.0%	2.6
RSRB (4-segment PBAN)	2.715E-04	N/A	N/A	N/A	1.7

The single RSRB reliability estimate is described in **Section 6.8.1**, **Reusable Solid Rocket Boosters**.

A key area for future analysis is the SSME altitude-start failure probabilities. With limited analysis time, these estimates were based on expert opinion and limited historic data. Aside from the specific altitude-start failure probabilities, it was assumed that the startup period, from ignition to full stable thrust, is instantaneous and the probability of catastrophic (uncontained) failure during engine startup is negligible. The altitude-start failure estimate for the upper stage engine was made based on preliminary engineering estimates for altitude-starting an SSME Block 2. Rocketdyne test data was updated assuming a 99 percent fix factor for startup problems resulting in a failure probability per engine of 1 in 661. Also, it was assumed that altitude-start redlines would be, for the most part, inhibited during the altitude-start sequence since a failure to start could be just as catastrophic to the vehicle as an uncontained engine failure. Further, it was assumed that a rigorous test program would be able to reduce the SSME altitude-start risk. These assumptions led to an altitude-start estimate of 3.0E-04, or 1 in 3,333, for a mature altitude-started SSME.

Figure 6-35 shows the LV 13.1 subsystem risk contributions. The risk is dominated by the second-stage engine ("liquid propulsion").



Notes: Percentages are based on the mean LOM failure probability. SSME burntime is 336 seconds.

	Mean Failure Probability	MFBF (1 in)
Liquid Propulsion	1.3824E-03	723
Solid Propulsion	2.7151E-04	3,683
GN&C	7.4112E-05	13,493
Staging/Separation	4.4804E-04	2,232
LOM (Loss of Mission)	**2.1745E-03**	**460**
LOC (Loss of Crew)	**4.9473E-04**	**2021**

Engine	Reliability (Cat)	Reliability (Ben)	Reliability (Start)	CFF	Error Factor
SSME	2.822E-04	1.482E-03	3.000E-04	16.0%	2.6
RSRB (4 Segment PBAN)	2.715E-04	N/A	N/A	N/A	1.7

Notes: Cat and benign based on default 515 second mission. Start risk is per demand. Error factor = 95th/50th.

Figure 6-35. LV 13.1 Subsystem Risk Contributions

To check these results, a simple mean reliability model was developed and is provided in **Appendix 6D**, **Safety and Reliability**. The model calculates FOMs by multiplying subsystem reliabilities. The MTBF results of this model for LV 13.1 yielded LOC = 2,855 and LOM = 516, which compare favorably with the results from FIRST, LOC = 2,021 and LOM = 460, thus affirming the reliability estimates for LV 13.1.

In addition, a preliminary sensitivity study was performed to investigate the LOC sensitivity to altitude-start reliability to CEEF. (See **Appendix 6D**, **Safety and Reliability**). Results indicate that reasonable and appropriate increases in the CEEF values applied to Delayed Catastrophic Failures (DCFs) events (altitude-start and noncatastrophic engine shutdown) allow for significant variations in altitude-start reliability without compromising LOC. It is recommended that, as the LV 13.1 design matures and the subsystem reliability estimates gain more certainty, detailed abort analyses replace the simplified CEEF estimates used in this study.

6.5.3.5 Schedule Assessment

A detailed development schedule (**Section 6.10**, **LV Development Schedule Assessment**, was developed for the ESAS Initial Reference Architecture (EIRA) CLV (five-segment RSRB with an upper stage using a new expander cycle engine). The CLV schedule for the EIRA Shuttle-derived option resulted in a predicted launch date of the first human mission in 2014. The critical path driver was the LR–85 new rocket engine for the upper stage. In order to meet the 2011 launch date requirement, an engine with a very short development time was needed. This requirement was met using the existing SSME modified for an altitude-start or an RL–10. The RL–10 was ruled out because of its low thrust level. The J–2 or J–2S could not support the 2011 launch date requirement.

6.5.4 Human-Rating Considerations for EELV

The EELV Program was intended to provide for a reliable access for commercial and military payloads, hence considerations for flying crew were never factored into the original design of the vehicles. The Mercury and Gemini Programs used vehicles originally designed for other purposes for launching crews to orbit. In order to accomplish crewed operations, major modifications were performed to provide for increased reliability, redundancy, failure detection and warning, and removing hardware not necessary for the crew launch mission. The same considerations would be required to utilize the EELV fleet to launch crew to LEO.

6.5.4.1 Human-Rating Requirements Drivers

The main requirement drivers from NPR 8705.2A, Human-Rating Requirements for Space Systems, are:

- Specifications and standards,
- Two-fault tolerant systems,
- Crew-system interactions,
- Pad emergency egress,
- Abort throughout the ascent profile,
- Software common cause failures,
- Manual control on ascent, and
- FTS.

The EELV fleet was built primarily to company standards and processes. The EELV was developed to “high-level” system requirements, and few aerospace industry design practices and standards were imposed. At the time the program was implemented, high reliability was to be demonstrated with multiple commercial launches before committing Government payloads. In response to the collapse of the commercial launch market (and resulting loss of demonstrated and envisioned reliability gains), Government mission assurance was ramped up with support from the Aerospace Corporation and the National Reconnaissance Office (NRO). The new CY2005 Buy III EELV contract will now include Government mission assurance requirements and standards. For EELV, these standards would need thorough evaluation and approval against NASA standards and processes to be used for flying crewed missions, with changes and additions implemented to close known gaps in requirements.

One of the most important requirement drivers is the requirement for two-fault tolerance to loss of life or permanent disability. NPR 8705.2A also states that abort cannot be used in response to the first failure. This implies that the LV must be at least single-fault tolerant, and, for subsystems that are required for abort, it must be two-fault tolerant. EELV will require upgrades in certain areas to achieve single-fault tolerance.

In order to fly crew for any launch system, the crew must have certain situation awareness and be able to react to contingencies based on that awareness. As such, NPR 8705.2A contains many requirements that deal with the crew’s ability to monitor health and status and take appropriate actions as a result of that status, if required. This will require upgrades in the EELV avionics architecture to accommodate an interface with the spacecraft as well as to be able to accept commands from the crew. For the LV, these commands will primarily be for contingency situations and will be for events such as abort initiation, retargeting (i.e., ATO), and response to other contingencies. Manual control is also a response to a contingency, although its use would primarily be limited to second-stage operations, where structural and thermal margins allow manual control. The form of manual control would be the subject of future trade studies and could range from a classical “yoke” control to a series of discrete commands that allow retargeting and ATO scenarios.

Another important requirement is to provide for successful abort modes from the launch pad through the entire ascent profile. This will require the EELV to be modified to provide the data necessary for abort decision-making. The modifications to the EELV may also require a computer and software for making the decision, or the decision-making may reside with the spacecraft. The means for providing for abort decision-making is a subject for a future trade study. Regardless of the outcome of that trade, significant effort on the LV will be required for health management and abort decision-making.

Other requirements, such as protection against software common-cause failures and FTSs, are not as extensive, but require some effort on the LV to implement. Protection against common-cause software failures can take several forms and is discussed in NPR 8705.2A. In the case of FTS, the EELVs can use the autodestruct command with lanyard pull devices to initiate an FTS event. Human spaceflight has never used autodestruct, and the utility of using these devices needs to be examined. Lanyard pulls allow the booster (first stage) to not have a dedicated receiver and command decoder unit, because it is able to accept the commands from the second stage and capable of autodestruct in the event of an inadvertent separation. Removal of the autodestruct may require addition of a dedicated receiver and command decoder unit on the first stage.

6.5.4.2 EELV Modifications for Human-Rating Summary

The Atlas V HLV with the new upper stage and Delta IV HLV with a new upper stage were considered for assessing modifications for flying crew. In some cases, detailed assessments were possible, while, in others, only the type of issues and resultant potential modifications were identified, depending on the fidelity of data available from the commercial launch provider. In either case, the goal of the analysis was to make reasonable judgments to provide valid cost assessments and ascertain potential schedule issues. (Refer to **Appendix 6F**, **EELV Modifications for Human-Rating Detailed Assessment**, for more information.

6.5.4.2.1 Atlas V HLV with New Upper Stage

Avionics and Software

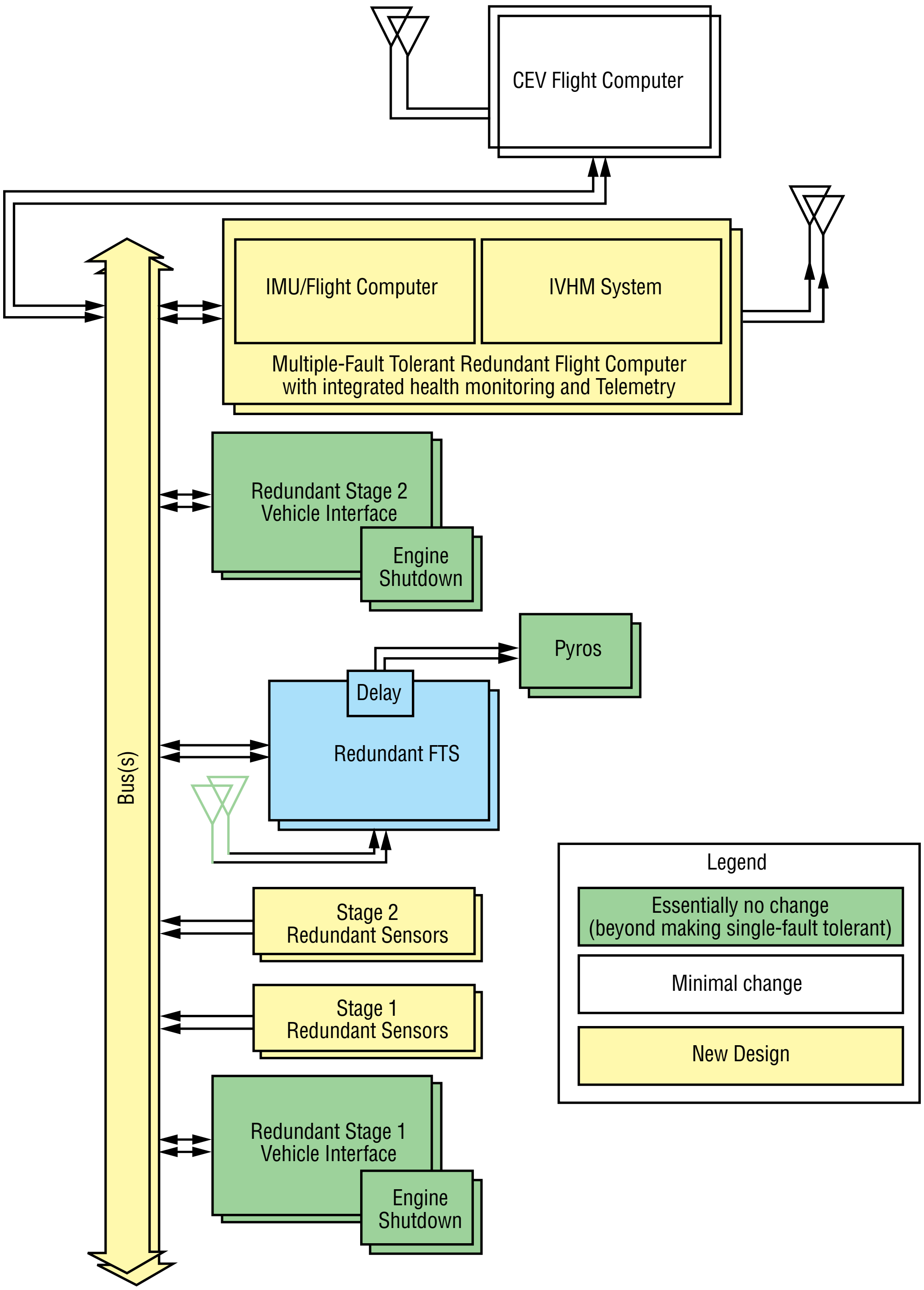
The avionics and software for the vehicle was assumed to be primarily new; however, some heritage in the GN&C area from the existing Atlas vehicle was assumed.

Launch Vehicle Health Management (LVHM) implementation as a fully integrated system is shown in **Figure 6-36**.

The core avionics meets the minimum single-fault tolerant requirement. Those elements needed for abort are two-fault tolerant.

The SLOC for a new build avionics system was estimated as follows:

- Events Manager (50 Hz) 500 to 1,000 SLOC;
- Navigation Manager (50 Hz) 8 to 15 thousand software lines of code (KSLOC);
- Guidance Manager (1 Hz) 15 to 25 KSLOC (both ascent and abort modes);
- Control Manager (50 Hz) 8 to 15 KSLOC;
- Command and Data Manager (50 Hz) 25 to 40 KSLOC;
- Time Manager (50 Hz) 1,500 to 2,000 SLOC;
- Power Manager (25 Hz) 2,500 to 4,000 SLOC;
- Vehicle Management Software (55 to 165 KSLOC); and
- Total SLOC = 103 to 267 KSLOC.



EELV Heritage with dual fault tolerant FC design (single fault-fly mission),
Plus integrated Health Monitoring and Abort Capability built into EELV FC to add Abort on second fault capability:
(Full redesign of FC and FC SW required. EELV FC capable of commanding abort as well as being controlled from CEV). EELV FC acting as its own backup computer.

Figure 6-36. Generic LVHM implementation

The large range in values is due to the vehicle management software, which incorporates the LVHM, Fault Detection, Isolation, and Recovery (FDIR), and abort decision-making. At present, there is uncertainty concerning the extent of LVHM that will be required, which will be the subject of future trade studies.

First Stage Main Propulsion

The primary focus of the effort was to examine changes required to the RD–180 for use in a human-rated system. The RD–180 was required to be built with U.S. production capability.

Second-Stage MPS

The second-stage MPS is new; however, modifications were assumed necessary for the RL–10A–4–2 engine to meet reliability and human-rating requirements.

Engine modifications were examined by considering the reliability enhancement program, along with consultation with vendors and discipline experts. The results were used to bound the cost estimates.

Structure

NPR 8705.2A imposes as an applicable document NASA–STD–5001, Structural Design and Test Factors of Safety for Spaceflight Hardware. This standard requires all structural Factors of Safety (FSs) for tested structures to be greater than 1.4. The commercial EELVs were designed to structural FSs of 1.25. NASA has taken exception to NASA–STD–5001 for FSs of less than 1.4 for well-defined loads. The process involves analyzing the load contribution (static versus dynamic) in assessing the required FS. For the purposes of bounding the problem in assessing costs for the modification of a structure, the criteria was used that for any structure with margins of less than 0.05 for an FS of 1.25, redesign would be required for EELV. Margins were assessed for actual flight loads. Since the Atlas has not flown in the heavy configuration, the 552 configuration (5-m core with five solids) was used for this assessment. The analysis results were used to bound the cost estimates for structural modification.

6.5.4.2.2 Delta IV with New Upper Stage

Avionics and Software

The basic Delta IV avionics system is single-fault tolerant, but some minor modifications were assumed. LVHM implementation was similar to the approaches previously discussed for the Atlas with a new upper stage vehicle where the LVHM function was integrated into the LV avionics (**Figure 6-39**). For the purposes of cost estimation, SLOC estimates were considered the same as for the Atlas case with a new upper stage.

Delta IV Booster MPS

The primary consideration for the Delta IV booster MPS was the upgrades for the RS–68 engine. Engine and MPS modifications were examined by considering the reliability enhancement program in consultation with vendors and discipline experts. The results were used to bound the cost estimates.

Upper Stage MPS

The upper stage MPS was assumed to be a new design utilizing the RL–10A–4–2 engine modified as discussed in **Section 6.5.4.2.1.3**, **Atlas V HLV with New Upper Stage**.

Structure

The Delta IV structure was evaluated using the same procedure as described for the Atlas V. As-flown margins of the Delta IV HLV booster were used for this assessment. The upper stage structure was all assumed new. The analysis results were used to bound the cost estimates for structured modifications.

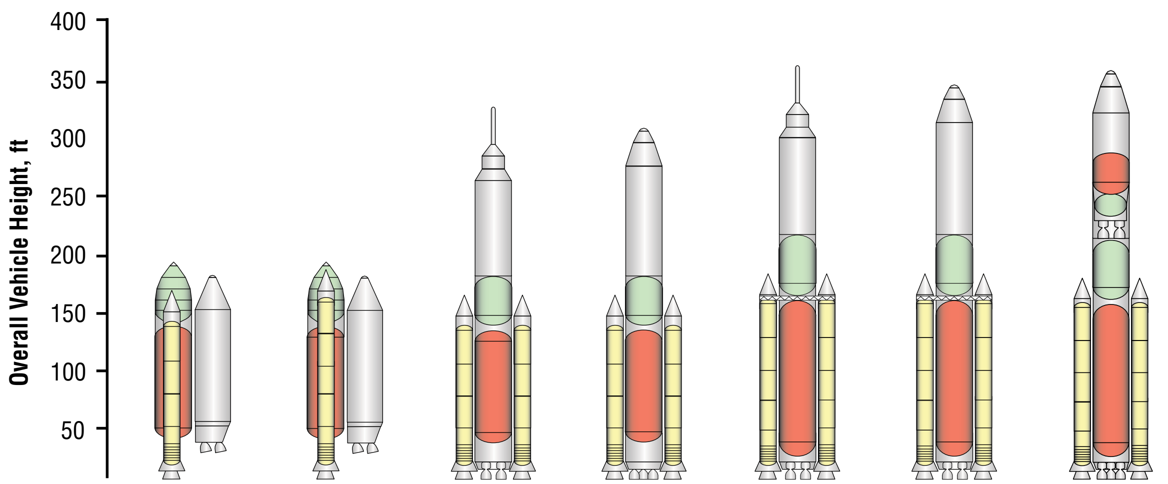
6.6 Lunar Cargo Vehicle

As with the CLV, many possible launch systems were examined during the study to meet the stated requirements of the CaLV. These options were narrowed down to the following candidates. The architectures that were not evaluated further are discussed in **Appendix 6A, Launch Vehicle Summary**.

6.6.1 Candidate LV Options Summary

Table 6-11 shows the Shuttle-derived lunar options assessed (options assessed in detail and other options initially assessed), including their dimensions, payload capabilities, and other parameters. **Table 6-12** provides the same information for the EELV-derived options.

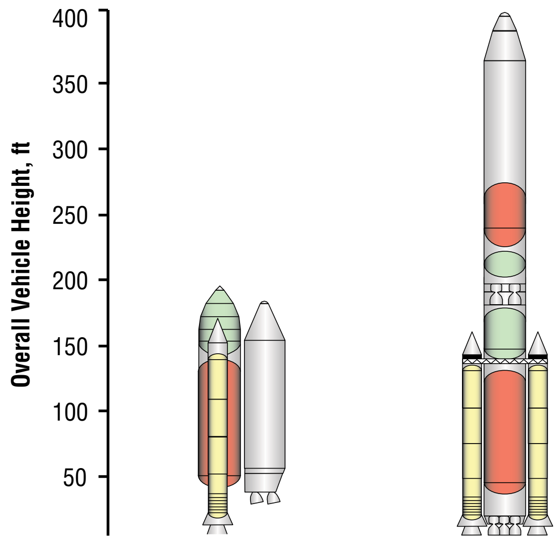
Table 6-11. Shuttle-Derived Lunar Options Assessed in Detail



		20	21	24	25	26	27	27.3 with EDS
Vehicle Name		**4-Segment SRM Side Mount SDV**	**5-Segment SRM Side Mount SDV**	**4-Segment SRB plus 3 SSME Core Crew + Cargo**	**4-Segment SRB plus 3 SSME Core Cargo**	**5-Segment SRB plus 4 SSME Core Crew +Cargo**	**5-Segment SRB plus 4 SSME Core Cargo**	**5-Segment SRB plus 5 SSME Core Plus EDS Cargo**
Payload 28.5 Deg Inc*	**Units**							
Lift Capability	mT	78.5 mT	93.5 mT	82.1 mT	86.5 mT	107.4 mT	113.8 mT	148.3 mT
Net Payload	**mT**	**66.7 mT**	**79.5 mT**	**73.9 mT**	**73.5 mT**	**91.3 mT**	**96.7 mT**	**126.0 mT**
Payload 51.6 Deg Inc*								
Lift Capability	mT	73.1 mT	87.5 mT	77.0 mT	81.1 mT	100.3 mT	106.8 mT	Not Evaluated
Net Payload	**mT**	**62.2 mT**	**74.4 mT**	**69.3 mT**	**69.0 mT**	**85.3 mT**	**90.8 mT**	**Not Evaluated**
General Parameters								
Overall Height	ft	183.8 ft	183.8 ft	315.9 ft	303.2 ft	355.0 ft	342.3 ft	357.6 ft
Gross Liftoff Mass	lbm	4,544,392 lbm	5,294,308 lbm	4,537,794 lbm	4,545,168 lbm	5,984,103 lbm	5,993,890 lbm	6,393,975 lbm
Liftoff Thrust/Weight	G	1.52 g	1.57 g	1.54 g	1.54 g	1.47 g	1.46 g	1.43 g
Second Stage Thrust/ Weight	G	0.97 g	0.94 g	0.96 g	0.95 g	0.93 g	0.93 g	0.84 g

**Delivered to 30X160 nmi Orbit*

Table 6-11 . Other Shuttle-Derived Options Initially Assessed



Vehicle Name		22 Shuttle Derived Side-mount 4-Seg. SRM & 2 RS–68	29 4-Segment SRBs 3RS–68 & 4 J–2S + Cargo
Payload 28.5 Deg Inc*	**Units**		
Lift Capability	mT	52.7 mT	108.2 mT
Net Payload	**mT**	**44.8 mT**	**91.9 mT**
Payload 51.6 Deg Inc*			
Lift Capability	mT	47.9 mT	102.4 mT
Net Payload	**mT**	**40.7 mT**	**87.1 mT**
General Parameters			
Overall Height	ft	183.8 ft	399.7 ft
Gross Liftoff Mass	lbm	4,492,706 lbm	5,401,018 lbm
Liftoff Thrust/Weight	G	1.58 g	1.44 g
Second Stage Thrust/Weight	G	1.05 g	1.09 g

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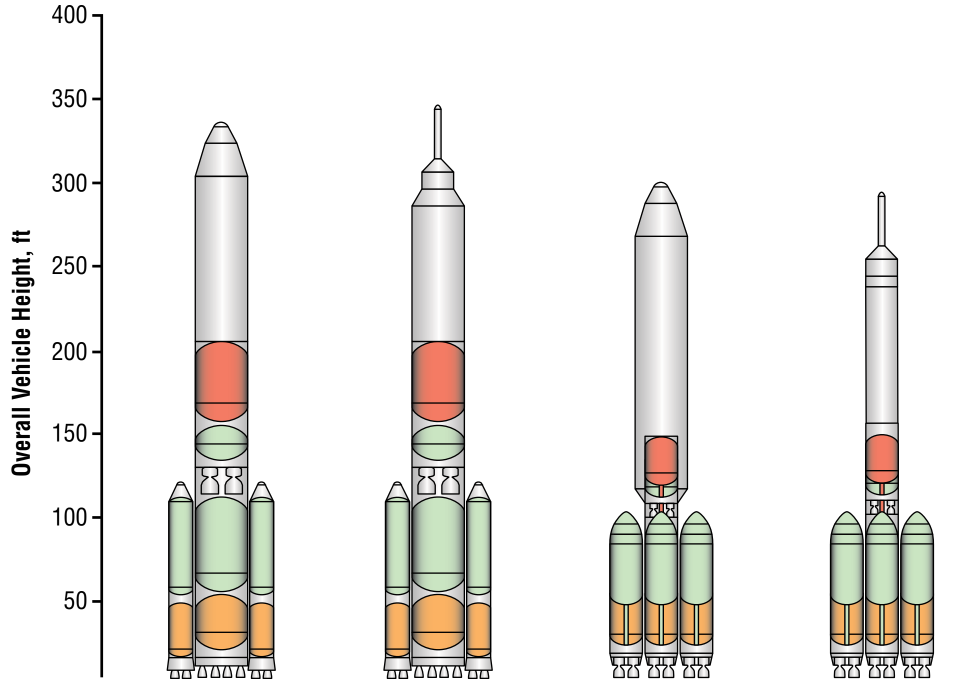
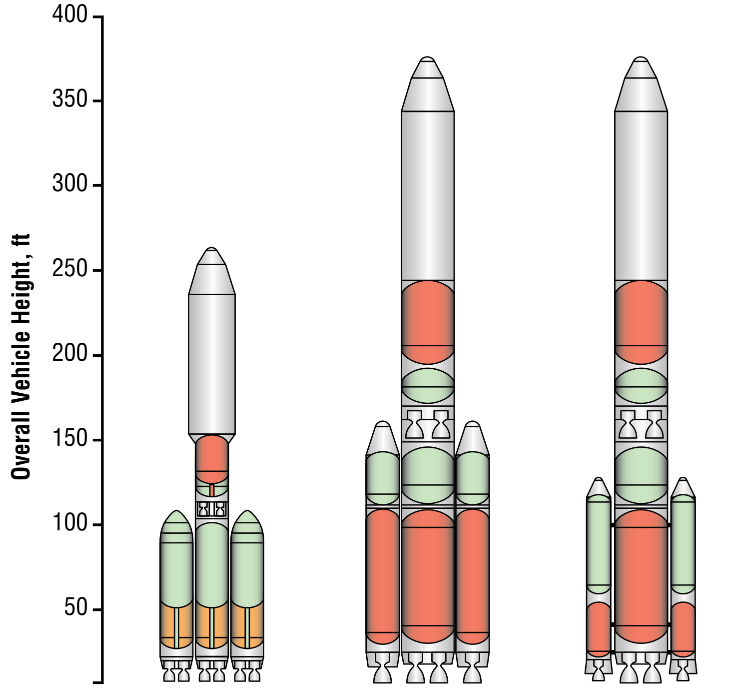
Table 6-12. EELV-Derived Lunar Options Assessed in Detail



		7.4	7.5	11	11.1
Vehicle Name		**Atlas Evolved (8m Core) + 2 Atlas V Boosters Cargo**	**Atlas Evolved (8m Core) + 2 Atlas V Boosters Crew + Cargo**	**Atlas Phase 3A (5m CBC) Cargo**	**Atlas Phase 3A Crew + Cargo**
Payload 28.5 Deg Inc*	**Units**				
Lift Capability	mT	111.9 mT	110.3 mT	110.4 mT	106.6 mT
Net Payload	**mT**	**95.1 mT**	**93.7 mT**	**93.8 mT**	**90.6 mT**
Payload 51.6 Deg Inc*					
Lift Capability	mT	106.1 mT	104.2 mT	104.4 mT	100.3 mT
Net Payload	**mT**	**90.2 mT**	**88.6 mT**	**88.8 mT**	**85.3 mT**
General Parameters					
Overall Height	ft	334.6 ft	347.6 ft	295.7 ft	290.1 ft
Gross Liftoff Mass	lbm	5,004,575 lbm	4,995,071 lbm	6,222,816 lbm	6,195,750 lbm
Liftoff Thrust/Weight	G	1.21 g	1.21 g	1.39 g	1.39 g
Second Stage Thrust/Weight	G	1.05 g	1.06 g	0.56 g	0.53 g

**Delivered to 30X160 nmi Orbit*

Table 6-12. Other EELV-Derived Options Initially Assessed



		10	28	28.1
Vehicle Name		**Atlas Phase 2 – Cargo**	**4 RS–68 Core + 4 J–2S + & 2 Delta IV Boosters Cargo**	**4 RS–68 Core + 4 J–2S + & 2 Atlas V Boosters Cargo**
Payload 28.5 Deg Inc*	**Units**			
Lift Capability	mT	73.6 mT	58.2 mT	64.1 mT
Net Payload	**mT**	**62.6 mT**	**49.5 mT**	**54.5 mT**
Payload 51.6 Deg Inc*				
Lift Capability	mT	69.5 mT	54.8 mT	60.6 mT
Net Payload	**mT**	**59.1 mT**	**46.6 mT**	**51.5 mT**
General Parameters				
Overall Height	ft	252.9 ft	368.5 ft	368.5 ft
Gross Liftoff Mass	lbm	3,811,194 lbm	3,207,626 lbm	3,601,955 lbm
Liftoff Thrust/Weight	G	1.36 g	1.24 g	1.22 g
Second Stage Thrust/Weight	G	0.64 g	1.19 g	1.17 g

**Delivered to 30X160 nmi Orbit*

6.6.2 FOMs Assessments

6.6.2.1 Shuttle-Derived Systems

A summary of the FOM assessment for the Shuttle-derived CaLV candidate vehicles is presented in **Table 6-13**. The assessment was conducted as a consensus of discipline experts and does not use weighting factors or numerical scoring but rather a judgment of high/medium/low (green/yellow/red) factors, with high (green) being the most favorable and low (red) being the least favorable.

Table 6-13. Shuttle-Derived Cargo Vehicle FOMs Assessment Summary

		Shuttle-derived CaLV				
		4-Segment RSRB Side-mount Cargo	5-Segment RSRB Side-mount Cargo	4-Segment RSRB In-line SDV Cargo	5-Segment RSRB/4 SSME Core In-line SDV Cargo	5-Segment RSRB/5 SSME Core In-line SDV Cargo Variant
	LV	20	21	24/25	26/27	27.3/13.1
FOMs	Probability of LOC	N/A	N/A	1 in 1170	1 in 915	1 in 2,021
	Probability of LOM	1 in 173	1 in 172	1 in 176	1 in 133	1 in 124
	Lunar Mission Flexibility					
	Mars Mission Extensibility					
	Commercial Extensibility	N/A	N/A	N/A	N/A	N/A
	National Security Extensibility	N/A	N/A	N/A	N/A	N/A
	Cost Risk					
	Schedule Risk	N/A	N/A	N/A	N/A	N/A
	Political Risk					
	DDT&E Cost (family)	.85	1.03	.83	0.98	1.00
	Facilities Cost (family)	N/A	N/A	N/A	1.00	1.00

The Shuttle-derived options rated moderate to favorable for LOC and favorable (green) for "family" DDT&E cost, largely due in each case to extensive use of flight proven hardware with extensive flight and test databases. A "family" DDT&E cost is derived for a CaLV that draws heavily from a CLV concept for some elements (e.g., booster engines). Essentially, this means that a development task is not repeated and paid for twice. For cost risk, the four-segment RSRB side-mount and in-line Shuttle-derived CaLVs were judged to be favorable (green), because the only new element to be developed is the cargo carrier. Five-segment RSRB development and new four- and five-SSME cores for LV 26/27 and LV 27.3, respectively, drive the cost risk for these vehicles to the yellow rating. No commercial or DoD extensibility was envisioned. The limitations of the side-mounted configuration in carrier vehicle geometry and payload lift capability restrict their extensibility for Mars missions, as well as flexibility for lunar missions to a lesser extent. No side-mounted SDV is capable of a 2-or-less lunar launch mission scenario. The four-segment/three-SSME SDV, LV 24/25, is not capable of launching lunar missions with two or less launches either. Favorable (green) rankings were given to the five-segment RSRB in-line SDV variants, LV 26/27/27.3, which possess the versatility required to accommodate changing lunar and Mars spacecraft architectures, because their configurations can accommodate a variety of payload geometries and increase lift capability relatively easily. LV 27.3, with five-segment RSRBs and five SSMEs in the core vehicle, enables the 1.5-launch solution (in conjunction with 13.1), which allows the crew to go to orbit on a CLV and have only one CaLV flight for the EDS and LSAM to LEO. Facilities costs were rated favorable (green) for the in-line Shuttle-derived CaLV variants, due to their continued extensive use of NASA Kennedy Space Center (KSC) Launch Complex (LC) 39. See **Section 7**, **Operations**, for more details on operations.

6.6.2.2 EELV-Derived Systems

A summary of the FOMs assessment for the EELV CaLV candidate vehicles is presented in **Table 6-14**. The assessment was conducted in the same manner as that for the Shuttle-derived vehicles.

	LV	EELV-Derived CaLV	
		8-m Core/RD–180/ 2 Atlas V Boosters w/ Upper Stage	Atlas Phase 3A (5.4-m CBC)
		7.4/7.5	11/11.1
FOMs	Probability of LOC	1 in 536	1 in 612
	Probability of LOM	1 in 71	1 in 88
	Lunar Mission Flexibility		
	Mars Mission Extensibility		
	Commercial Extensibiity	N/A	N/A
	National Security Extensibility	N/A	N/A
	Cost Risk		
	Schedule Risk	N/A	N/A
	Political Risk		
	DDT&E Cost (family)	1.26	1.02
	Facilities Cost (family)	1.12	1.56

Table 6-14. EELV-Derived Cargo Vehicle FOMs Assessment Summary

Both EELV CaLV concepts rated unfavorable (red) for LOC as they do not approach the 1-in-1,000 goal. The use of multi-engine stages, multiple strap-on boosters, and relatively low-to-moderate design heritage from existing systems all were major contributors. Low T/W for LV 7.4/7.5 and limitations due to the 5.4-m core vehicle diameter for LV 11/11.1 limits the ability of each to provide flexibility to future lunar missions and extending the use of either vehicle for Mars missions. The four strap-on boosters with a central core configuration, LV 11/11.1, dictates the need for new facilities, as no present launch infrastructure at KSC can accommodate this configuration. LV 7.4/7.5 is more conventional in geometry, with two strap-on boosters, which could be accommodated with modification to KSC LC 39. No projected commercial or DoD missions require the use of this class of LV, so no FOM rating was applied. While both vehicles support a 2-launch lunar mission solution, neither demonstrated the ability to enable the 1.5-launch solution, necessitating each vehicle to be human rated. The delta CaLV options (LV 28 and LV 28.1) did not meet the threshold payload performance of 70 mT and were dropped from further consideration. Options using RSRBs required upper stages to meet performance goals. The Phase 3 Atlas, LV 11/11.1, draws more design heritage from CLV (option 9) and, as a result, demonstrates a more favorable family DDT&E cost than LV 7.4/7.5, with the 8-m core. However, the CLV costs for this option were unacceptably high. (See **Section 6.5.2.2**, **EELV-Derived Systems**) Use of an 8-m stage diameter for a CLV to derive family DDT&E costs adversely affects the initial CLV cost, which results in little or no overall savings. The Atlas Phase 2 variant utilizing two liquid strap-on boosters offered no advantages to the Atlas 3A, other than having a simpler configuration that reduced the scope of the launch infrastructure modifications, but still required the associated cost of a dedicated CLV.

6.6.3 Detailed Assessment Summary

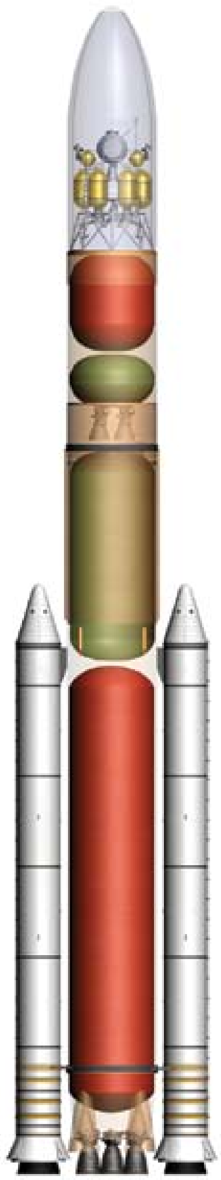


Figure 6-37. LV 27.3 General Configuration

6.6.3.1 Description of Selected LV

The preferred CaLV concept, LV 27.3, (**Figure 6-37**) is a 1.5-stage parallel-burn LV with an EDS that is optimized for cargo to TLI. This is an in-line Shuttle-derived concept that uses ET-diameter tankage and structure for the core and EDS. The general configuration is two solid strap-on boosters connected to a LOX/LH2 core stage. The two solid strap-on boosters are five-segment RSRBs (HTPB propellant). The LOX/LH2 core stage uses five SSMEs for propulsion. The EDS is LOX/LH2 with two J–2S+ engines and is burned suborbitally in the concept. (Later studies indicate this may be able to be reduced to one J–2S+ engine). This vehicle is flown to a 30- by 160- nmi orbit at an inclination of 28.5 deg and inserted at an altitude of 78.3 nmi. The SSMEs are run at a throttle setting of 104.5 percent. The J–2S+ engines of the EDS stage are operated at a 100 percent power level during the suborbital and TLI burns.

6.6.3.2 Performance Summary

The net payload capability of LV 27.3 plus EDS to TLI is 54.6 mT for the maximum TLI payload carried from liftoff (no orbital rendezvous).

Two other EDS cases were considered for this vehicle. In Case 1, LV 27.3 was assumed to have a 42.8-mT LSAM attached to the EDS at launch. The EDS propellant load was reoptimized for this case. The EDS with LSAM attached then rendezvoused with a CEV on orbit at a 160-nmi circular orbit that weighed 19.1 mT, for a total cargo stack mass of 61.9 mT in orbit. The EDS with LSAM and CEV then performed a TLI burn with the remaining EDS propellant. The TLI net payload capability for this case was determined to be 68.6 mT, which is 6.7 mT greater than the required delivery mass of 61.9 mT.

In Case 2, LV 27.3 was assumed to have a 44.9-mT LSAM attached to the EDS at launch. The EDS propellant load was also reoptimized for this case. The EDS with LSAM attached then rendezvoused with a CEV on orbit at 160 nmi circular orbit that weighed 20.6 mT, for a total cargo stack mass of 65.5 mT in orbit. The EDS with LSAM and CEV then performed a TLI burn with the remaining EDS propellant. The TLI net payload capability for this case was determined to be 66.9 mT, which is 1.4 mT greater than the required delivery mass of 65.5 mT. A graphical representation of TLI payloads and the relative masses of the LSAM and CEV are shown in **Figure 6-38**.

No GR&As were violated for this LV analysis.

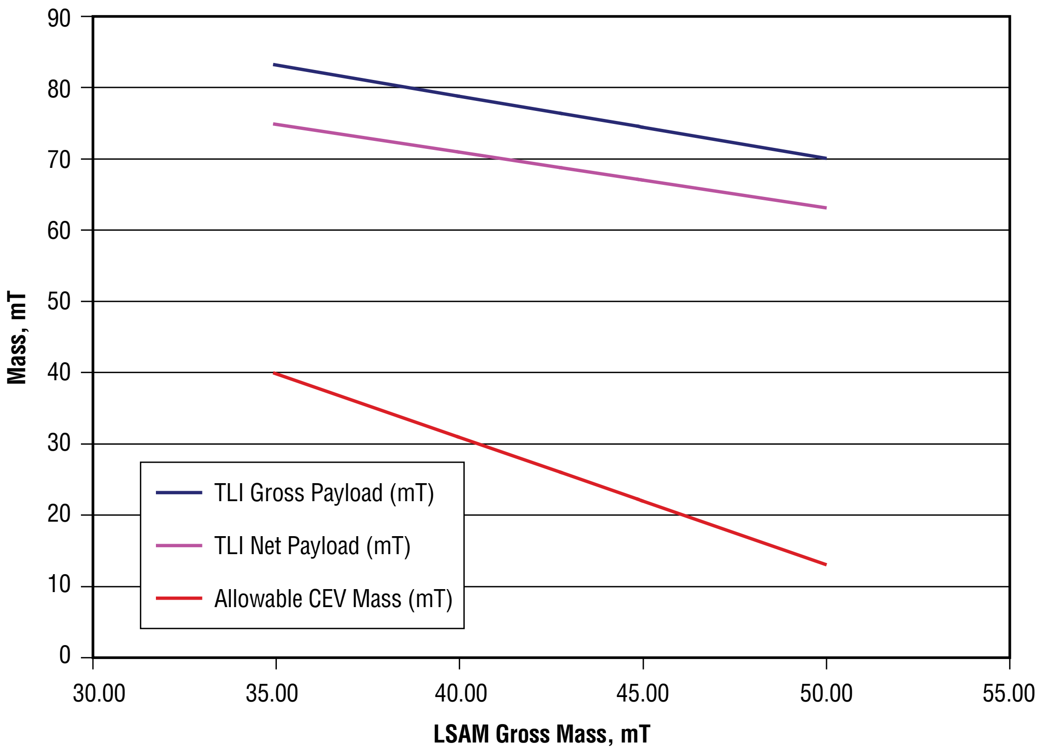


Figure 6-38. 1.5-Launch Solution Allowable Masses as a Function of LSAM Mass

6.6.3.2.1 Vehicle Sizing

The mass properties for the core stage and the EDS of LV 27.3 are shown in **Table 6-15**.

Table 6-15. Launch Vehicle 27.3 INTROS Mass Summary

Mass Properties Accounting Vehicle: In-line Cargo – 5 SSME, 2 J–2S+, Five-Segment RSRB Stage: Strap-on Solid (Five-Segment RSRB)		
Item	**Mass Subtotals** Primary lbm	**Mass Totals** lbm
Stage Burnout Mass		**221,234**
Main Ascent Propellant	1,434,906	
Stage Gross Liftoff Mass		**1,656,140**
Stage: EDS (2 J–2S+)		
Primary Body Structures	19,592	
Secondary Structures	2,436	
Separation Systems	199	
TPS	317	
TCS	1,482	
MPS	12,642	
Power (Electrical)	1,413	
Power (Hydraulic)	404	

Table 6-15. Launch Vehicle 27.3 INTROS Mass Summary (continued)

Mass Properties Accounting Vehicle: In-line Cargo – 5 SSME, 2 J–2S+, Five-Segment RSRB Stage: Strap-on Solid (Five-Segment RSRB)		
Item	**Mass Subtotals**	**Mass Totals**
	Primary	
	lbm	lbm
Avionics	430	
Miscellaneous	131	
Stage Dry Mass Without Growth		**39,046**
Dry Mass Growth Allowance	3,599	
Stage Dry Mass With Growth		**42,645**
Residuals	5,309	
Reserves	628	
In-flight Fluid Losses	59	
Stage Burnout Mass		**48,640**
Main Ascent Propellant	457,884	
Engine Purge Helium	52	
Stage Gross Liftoff Mass		**506,576**
Stage: Core Stage (5 SSME Blk 2)		
Primary Body Structures	102,965	
Secondary Structures	3,789	
Separation Systems	3,898	
TPS	574	
TCS	5,373	
MPS	58,015	
Power (Electrical)	2,922	
Power (Hydraulic)	1,804	
Avionics	670	
Miscellaneous	573	
Stage Dry Mass Without Growth		**180,583**
Dry Mass Growth Allowance	14,413	
Stage Dry Mass With Growth		**194,997**
Residuals	16,676	
Reserves	3,323	
In-flight Fluid Losses	262	
Stage Burnout Mass		**215,258**
Main Ascent Propellant	2,215,385	
Engine Purge Helium	251	
Stage Gross Liftoff Mass		**2,430,894**
Payload	133,703	
Payload Shroud	10,522	
Upper Stage Gross Mass	506,576	
Strap-ons, Gross Mass	3,312,279	
Vehicle Gross Liftoff Mass		**6,393,975**

6.6.3.2.2 Structural Analysis

The loads plot is a combined worst-case including liftoff, maximum dynamic pressure (max q), and maximum acceleration (max g). The tie-down loads are assumed to be carried by the RSRBs, as with the current Shuttle system. The compression loads show a major jump where the LOX tank loads are integrated into the outside structure, with a quick reduction of the loads where the introduced SRB loads counteract the compression. The bending moment shows a steady increase from the tip of the vehicle to the liftoff Center of Gravity (CG), then a steady decrease back to zero, as expected from an in-flight case. **Figure 6-39** shows the structural configuration of the CaLV, while **Figure 6-40** summarizes the structural load analysis.

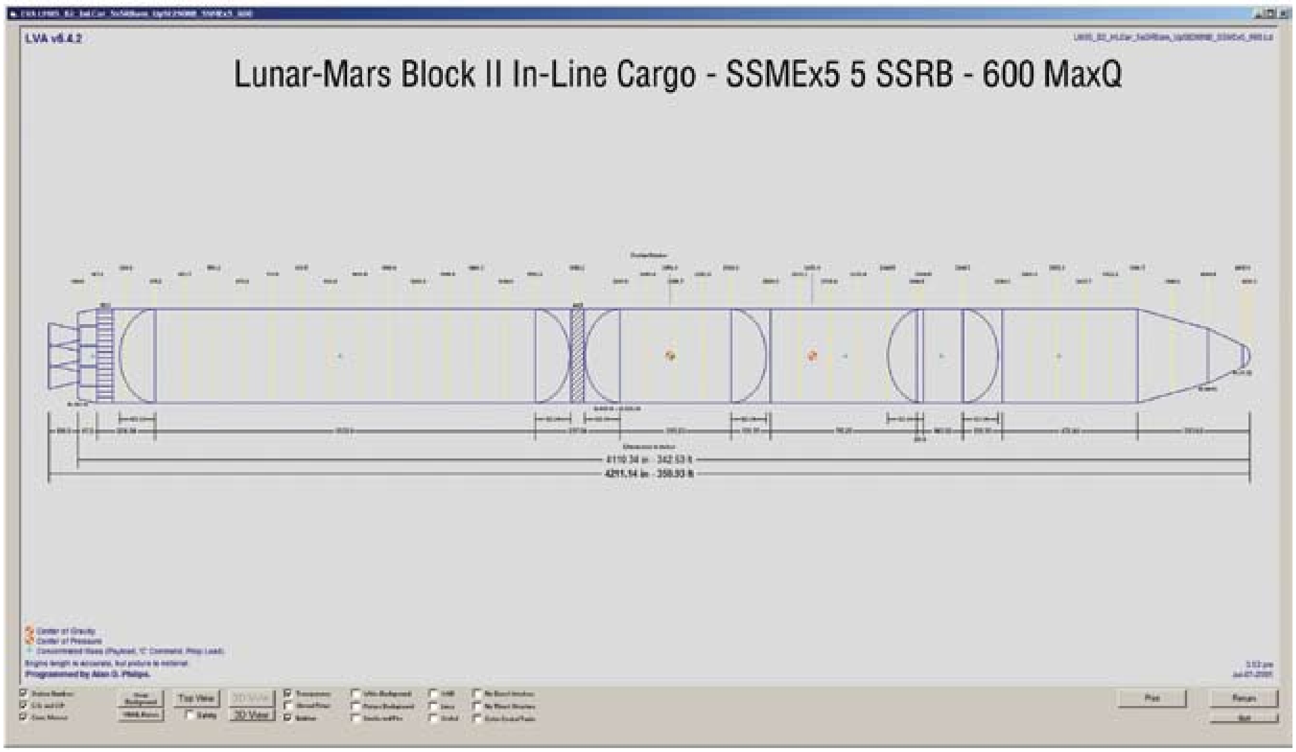


Figure 6-39. CaLV Structural Configuration

Considerable effort was used to optimize this particular vehicle for payload to TLI. A max q of 600 lb/ft^2 was used (as calculated by the Program to Optimize Simulated Trajectories (POST) tool) instead of the 750 normally used for this class of vehicle. Also, a more efficient conical thrust structure was used for the EDS instead of the standard cruciform. The core was also re analyzed for the loads in this particular case.

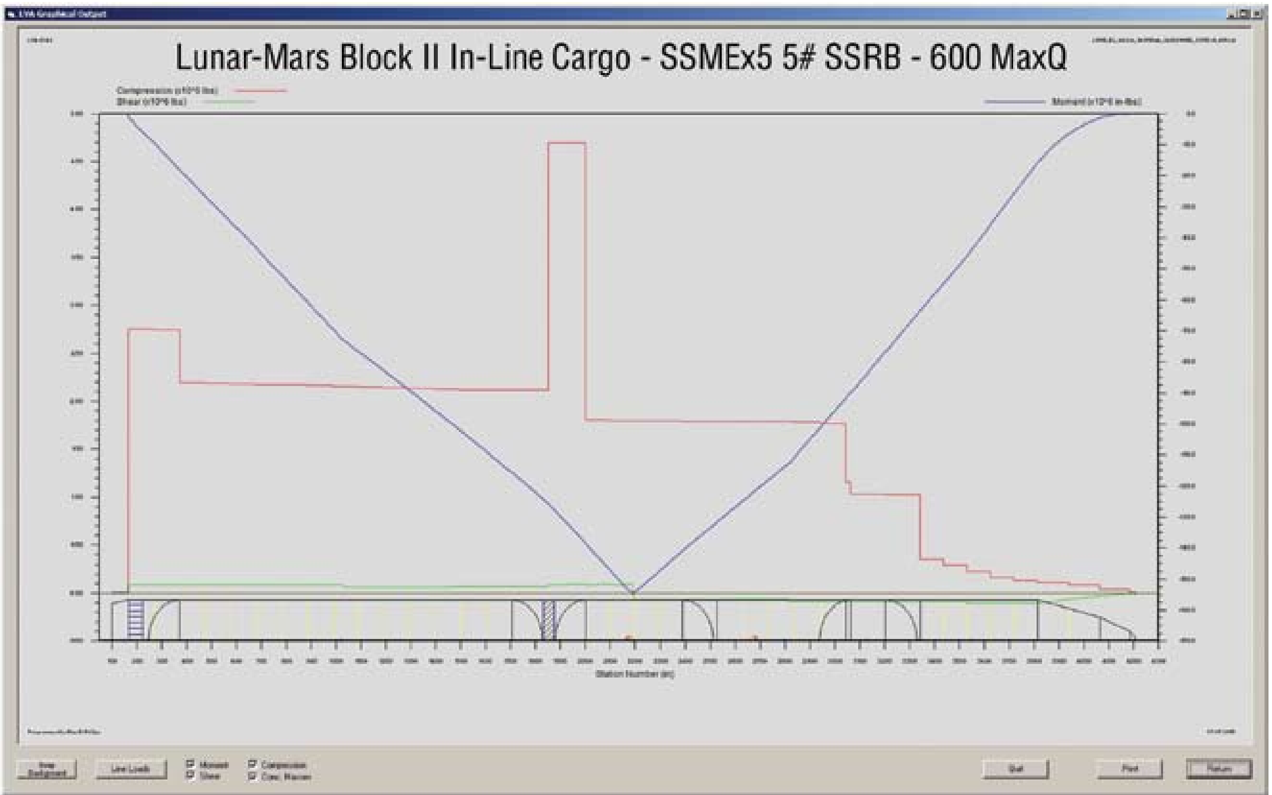
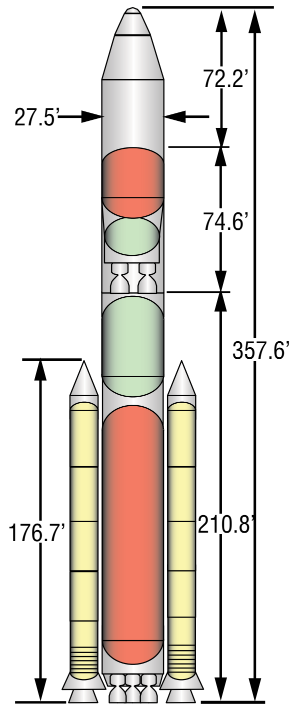


Figure 6-40. CaLV Structural Loads Analysis Results

6.6.3.2.3 Flight Performance Analysis and Trajectory Design

Major events in the trajectory and LV characteristics are shown in **Figure 6-41**. The analyses of the four EDS case studies are shown in **Figures 6-42** and **6-43**. Selected trajectory parameters are shown in **Figures 6-44** through **Figure 6-47**. This vehicle T/W ratio at liftoff is 1.43. The vehicle reaches a maximum dynamic pressure of 561 psf at 72.7 sec. The maximum acceleration with boosters attached is 2.32 g's, while the core hits a max of 2.83 g's before burnout, and the EDS stops accelerating at 1.46 g's prior to Main Engine Cutoff (MECO). The five-segment SRBs separate 132.52 sec into the burn at an altitude of 154,235 ft and Mach 3.9. The core burns out at 408.2 sec, having reached an altitude of 408,090 ft at Mach 12.1. From this point, the EDS ignites and burns 264,690 lb of propellant to reach orbit. The T/W ratio of the core after SRB separation is 1.04, and 0.84 after EDS ignition. Orbital injection occurs 626 sec after liftoff at 78.3 nmi.

5 SSME Core & 5-Segment SRB + 2 J-2S + EDS Cargo



Vehicle Concept Characteristics

GLOW	**6,393,975 lbf**
Payload Envelope L x D	39.4 ft x 24.5 ft
Shroud Jettison Mass	10,522 lbm
Booster Stage (each)	
Propellants	HTPB
Useable Propellant	1,434,906 lbm
Stage pmf	0.8664
Burnout Mass	221,234 lbm
# Boosters / Type	2 / 5-Segment SRM
Booster Thrust (@ 0.7 sec)	3,480,123 lbf @ Vac
Booster Isp (@ 0.7 sec)	265.4 sec @ Vac
First Stage	
Propellants	LOX/LH2
Useable Propellant	2,215,385 lbm
Propellant Offload	0.0 %
Stage pmf	0.9113
Dry Mass	194,997 lbm
Burnout Mass	215,258 lbm
# Engines / Type	5 / SSME Blk 2
Engine Thrust (100%)	375,181 lbf @ SL 469,449 lbf @ Vac
Engine Isp (100%)	361.3 sec @ SL 452.1 sec @ Vac
Mission Power Level	104.5 %
Earth Departure /Upperstage	
Propellants	LOX/LH2
Useable Propellant	457,884 lbm
Propellant Offload	0.0 %
Stage pmf	0.9039
Dry Mass	42,645 lbm
Burnout Mass	48,640 lbm
# Engines / Type	2 / J-2S+
Engine Thrust (100%)	274,500 lbf @ Vac
Engine Isp (100%)	451.5 sec @ Vac
Mission Power Level	100.0 %

Delivery Orbit	30 x 60 nmi @ 28.5°	
Del. Orbit Payload	326,896 lbm	148.3 mT
Net Payload	277,862 lbm	126.0 mT

LEO payload Optimized Thru Propellant Offload in EDS of 40%

Delivery Orbit	TLI (EDS Suborbital Burn)	
Gross Payload	133,703 lbm	60.6 mT
Net Payload	120,333 lbm	54.6 mT

Closed Case Summary Data for Reference Mission (30-160 nmi @ 28.5):

Liftoff to SRM staging
- f/wo = 1.43 (@ t = 1 sec)
- max RSRM accel = 2.32
- time of max Q = 72.7 sec
- throttle @ bucket = no change
- max Q = 561 psf
- mach = 1.52

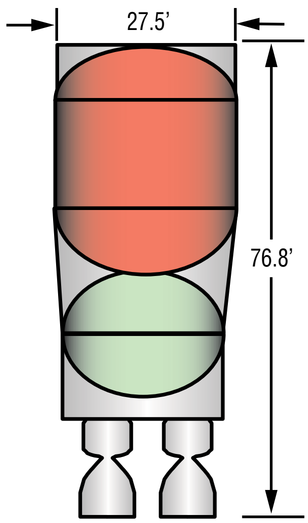
After SRM jettison (Core stg1 + stg2)
- tstg = 132.52 sec
- alt @ stg = 154,235 ft
- mach @ stg = 3.85
- dynp @ stg = 27 psf
- dv1 = 8,058 ft/s
- f/w1 = 1.041
- max stg1 f/w = 2.83

Shroud Jettison @t = 447.0 sec
- alt @ jettison = 431,200 ft

After Stg1 jettison (stg2 only)
- tstg = 408.2 sec
- alt @ stg = 408,090 ft
- mach @ stg = 12.12
- dynp @ stg = 0 psf
- dv1 = 22,656 ft/s
- f/w1 = 0.844
- max stg2 f/w = 1.46

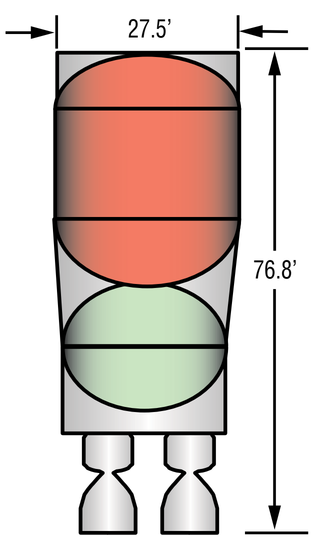
At MECO / Orbital Insertion
- time to MECO = 625.9 sec
- MECO altitude = 475,827 ft
- dvt = 30,386 ft/s

Figure 6-41. LV 27.3 Summary



Vehicle Concept Characteristics		
EDS+PL Gross @ Liftoff	**640,282 lbf**	
EDS Gross @ Liftoff	**545,924 lbf**	
EDS Stage		
Propellants	LOX/LH2	
Useable Propellant @ Liftoff	495,128 lbm	
Useable Propellant @ 160 nmi cir.	223,826 lbm	
Stage PMS	0.9070	
Dry Mass	44,314 lbm	
Burnout Mass	50,741 lbm	
# Engines / Type	2/J–2S+	
Engine Thrust (100%)	274,500 lbf @ Vac	
Engine Isp (100%)	451.5 sec @ Vac	
Mission Power Level	100.0%	
TLI Delivery		
CEV @ Liftoff	44,754 lbm	20.3 mT
LSAM Payload	94,358 lbm	42.8 mT
CEV Payload	42,108 lbm	19.1 mT
Margin Payload	31,480 lbm	14.3 mT
Gross Total Payload	167,946 lbm	76.2 mT
Net Payload	151,152 lbm	68.6 mT
Net Allowable CEV Mass	56,794 lbm	25.8 MT

Figure 6-42. Case 1: LV 27.3 SP + EDS (42.8 mT LSAM)



Vehicle Concept Characteristics		
EDS+PL Gross @ Liftoff	**640,281 lbf**	
EDS Gross @ Liftoff	**541,294 lbf**	
EDS Stage		
Propellants	LOX/LH2	
Useable Propellant @ Liftoff	490,744 lbm	
Useable Propellant @ 160 nmi cir.	219,443 lbm	
Stage PMS	0.9066	
Dry Mass	44,118 lbm	
Burnout Mass	50,494 lbm	
# Engines / Type	2 / J-2S+	
Engine Thrust (100%)	274,500 lbf @ Vac	
Engine Isp (100%)	451.5 sec @ Vac	
Mission Power Level	100.0%	
TLI Delivery		
CEV @ Liftoff	48,061 lbm	21.8 mT
LSAM Payload	98,988 lbm	44.9 mT
CEV Payload	45,415 lbm	20.6 mT
Margin Payload	19,500 lbm	8.8 mT
Gross Total Payload	163,903 lbm	74.3 mT
Net Payload	147,513 lbm	66.9 mT
Net Allowable CEV Mass	48, 525 lbm	22.0 mT

Figure 6-43. Case 2: LV 27.3 SP + EDS (44.9 mT LSAM)

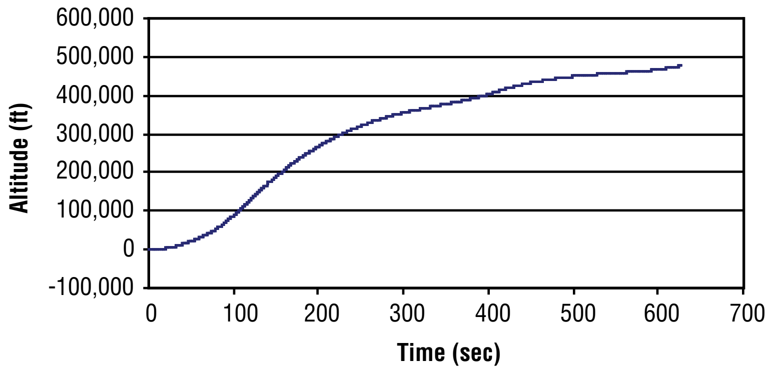


Figure 6-44. Altitude versus Time

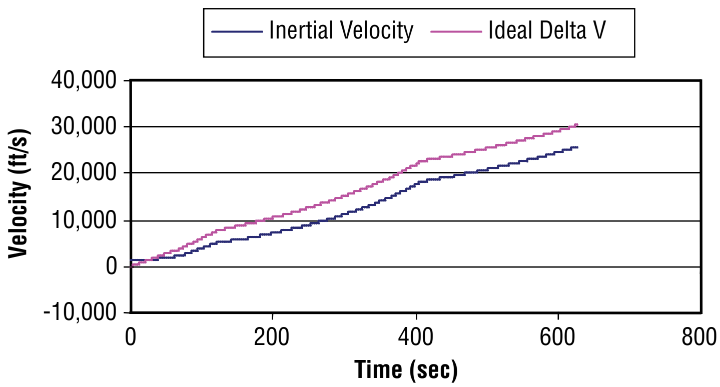


Figure 6-45. Velocity versus Time

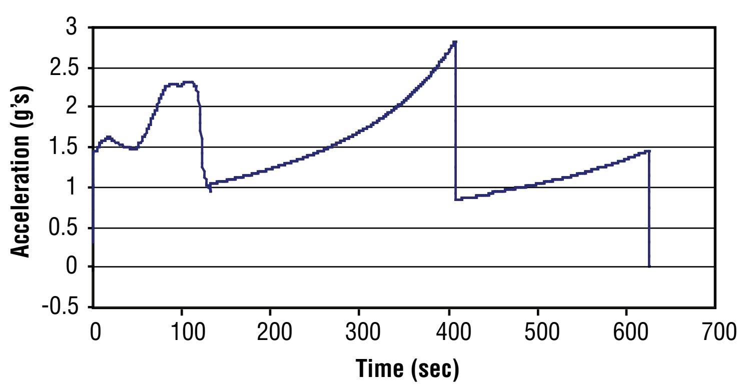


Figure 6-46. Acceleration versus Time

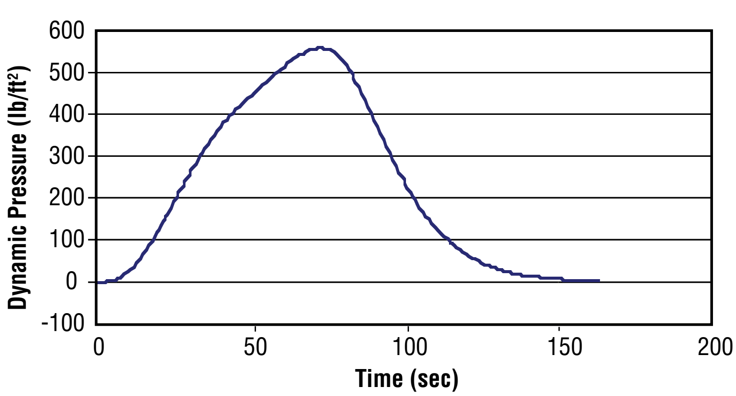


Figure 6-47. Dynamic Pressure versus Time

6.6.3.3 Cost Analysis Assumptions for CaLVs

6.6.3.3.1 Inputs – Core Stage

Structure and Tanks

All structures and tanks are considered a new design, but with no new technology. The stage diameter is the same as the ET. Materials are either 2219 aluminum or AL-Li. Shrouds are made of graphite-epoxy panels, based on Titan and Delta IV designs. Structures and tanks are well understood with sufficient manufacturing capability in existence. All structures are similar to EELV and ET and have been validated in the relevant environment. All vehicles will, however, require full testing and qualification.

Main Propulsion System

The MPS will take significant heritage from the existing SSME MPS subsystem. However, a new design is needed to accommodate the varying number of SSMEs. Cost estimates assumed a new design with similar subsystems validated in the relevant environment. Full testing and qualification will be required.

Engine–SSME

In addition to the minor changes required to altitude-start the SSME (RS–25d), it is desirable to make some engine improvements to lower the unit cost and improve producibility. Suggested improvements include: low-pressure turbomachinery simplifications, a new controller, an HIP bonded MCC, flex hoses to replace flex joints on four ducts, and simplified nozzle processing. In addition, process changes would be incorporated to eliminate inspections for reuse and accommodate obsolescence of the controller.

The next step in the evolution of the SSME for exploration involves improvements for low-cost manufacturing and operations for a fully expendable SSME. Improvements include a channel wall nozzle, simplified high-pressure pumps, and a cast and simplified powerhead. The estimate is based on SSME historical costs, vendor quotes, and estimates.

Avionics and Software

The avionics subsystem must support Fail Operational/Fail Safe vehicle fault-tolerant requirements, meaning that, upon occurrence of the first failure, the backup to the failed system will keep the vehicle operating nominally. Upon a second failure, the subsystem will safely recommend an abort. Crew abort failure detection and decision-making capabilities have been demonstrated and are ready for flight. All architectures will meet these requirements, either by adding a modification for instrumentation redundancy for the EELV health management system, or providing the capabilities through the new design of the avionics for Shuttle-derived configurations.

Avionics hardware is divided into GN&C and CCDH. GN&C provides for attitude control, attitude determination, and attitude stabilization. CCDH provides all the equipment necessary for transfer and processing of data; communication for personnel, as well as spacecraft operations/telemetry data; and instrumentation for monitoring the vehicle and its performance. Both systems are linked through the LV software system. LV hardware requirements are well understood.

The core booster does not guide and control the ascent. This function is controlled by the upper stage. Core booster avionics includes translators, controllers, AD converters, the actuator control, electronics, and sufficient CCDH hardware to interface with the upper stage. The upper stage avionics controls ascent, separations, and flight. Upper stage avionics hardware includes the IMU, processors, communications, telemetry, and instrumentation. Software provides separation commands and includes general flight, mission-specific flight algorithms, and launch-date-specific software.

Software also provides the commands that control the vehicle, viewed as one entity for the LV. As such, the software estimate is not divided between the core and upper stage. Software is normally located on the upper stage, because the upper stage controls the ascent of the LV. The software estimate for the LVs is based on the same detailed breakdown of the functional requirements, shown in **Table 6-8**.

Software estimates are based on the maximum SLOC, using the SEER–SEM tool for software estimation, planning, and project control. SEER–SEM is a recognized software estimation tool developed by Galorath Incorporated for use in industry and the Government.

Shuttle-Derived Avionics Hardware
The GN&C and CCDH subsystems for Shuttle-derived LVs are considered new designs. Because the subsystems and software are new, integrated health management and human-rating requirements are incorporated from the start. The avionics hardware assumed a new design with existing technology.

Shuttle-Derived Software
All Shuttle-derived software is considered new software development, incorporating the functions identified above. The maximum SLOC estimates were used with the SEER–SEM model to arrive at a deterministic software estimate.

Other Subsystems
The basic thermal systems are ½- to 1-inch thick SOFI, with cold plates and insulation for passive cooling of equipment and avionics. No new technology is planned.

Electrical power is provided by silver-zinc batteries with a redundancy of two. Conversion, distribution, and circuitry are considered new designs with state-of-the-art technology. Hydraulic power is fueled by hydrazine, which is used in LVs today.

RCSs, when used, are the same type as currently used in the Shuttle. Range safety will require modifications to the flight termination system to add time-delay for abort. Human-rating requirements may necessitate the removal of the autodestruct capability. All of these subsystems are similar to those already in existence, either on EELVs or the Shuttle, and have been validated in the relevant environment. Full qualification and testing is estimated for all crew and cargo vehicles.

For the side-mount Shuttle-derived vehicles, the existing ET is used.

6.6.3.3.2 DDT&E

The lowest cost options, as shown in **Table 6-16**, from this group of vehicles are the four-segment RSRB in-line with three SSMEs and the four-segment RSRB side-mounted SDV. The five-segment in-line SDV follows next. The most expensive DDT&E is LV 27.3. This vehicle includes an EDS used as an upper stage, which is included in the cost estimates.

Table 6-16. Relative Comparison of Shuttle-Derived Cargo Vehicle Costs

Phase	Relative Cost Position						
Vehicle	20	21	24	25	26	27	27.3
DDT&E	0.75	0.80	0.73	0.73	0.96	0.96	1.00
Production	1.33	1.33	0.88	0.88	0.96	0.96	1.00
Operations	1.07	1.07	0.96	0.96	1.00	1.00	1.00
Facilities	1.00	1.00	1.00	1.00	1.00	1.00	1.00
Number of launches for lunar mission	3+	3+	3	3	2	2	1.5

6.6.3.3.3 Production

LV 20/21, 24/25, 26/27, and 27.3 are four- or five-segment RSRB in-line or side-mounted configurations using modified/evolved Shuttle flight hardware. The recurring production costs of these four families of concepts are relatively close. The four-segment RSRB is slightly less expensive to refurbish than the five-segment version (i.e., the cost of refurbishing and reloading the two additional motor segments is relatively small), so that the main differences in cost relate more to the total number and TFU costs of other hardware pieces that must be produced (and integrated), such as separate tanks (for the side-mounted concepts) and engines in particular. SSMEs are significant drivers of production costs, thus the greater the number on the vehicle, the greater the production costs. As shown in **Table 6-16**, LVs 20 and 21 (side-mounts) have the greatest annual production cost at six flights per year, followed by LV 27.3 and LV 26/27, while the least expensive configuration to produce is LV 24/25.

6.6.3.3.4 Operations

All of these concepts require the stacking of either two four- or five-segment SRBs similar to the current Shuttle configuration. The SRM segments are refurbished in the same manner as in the current Shuttle operation. Core stages and engines are new manufacturable items. The launch operations activities include receipt, checkout, stacking and integration, testing, transport to the launch pad, pad operations, and launch. As shown in **Table 6-16**, the cost of launch operations is lowest for LV 24/25 and greatest for LV 20/21, because of the greater number of elements to be integrated. However, the difference at six flights per year is slight.

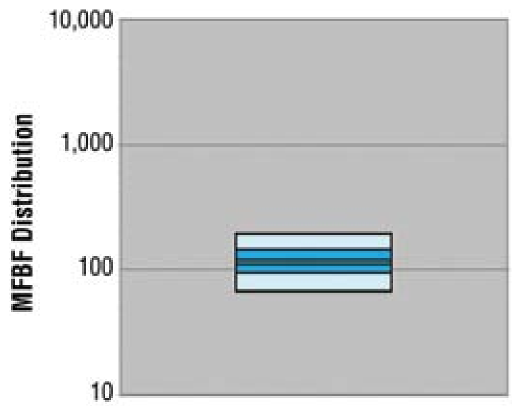
6.6.3.3.5 Facilities

The facilities costs include modifications to the MLP, VAB, and the launch pad to accommodate the different profile and footprint of the in-line configuration. The side-mounted concepts would require little modification. The facilities cost is greatest for LV 27.3. The relative cost position of the vehicles for DDT&E, production, launch operations, and facilities is summarized in **Table 6-16**. Detailed cost estimates are provided in **Section 12**, **Cost**.

6.6.3.4 Safety/Reliability Analysis (LV 27.3)

The same tool as previously discussed for LV 13.1 was used to determine the CaLV in-line core with five SSMEs and two five-segment RSRBs (LV 27.3) LOM estimates. These estimates were based on preliminary vehicle descriptions that included propulsion elements and Space Shuttle-based LV subsystems reliability predictions. A simple reliability model using point estimates was used to validate the results. A complete description of both models is included in **Appendix 6D**, **Safety and Reliability**. The LV 27.3 LOM estimates are shown in **Figure 6-48**. The LOM results are for ascent only. Other key assumptions included:

- No mission continuance engine-out capability;
- Engine shutdown is just as catastrophic to the vehicle as an uncontained failure; and
- SSMEs operated with current redlines inhibited. A 10 percent risk reduction of the overall LOM mean estimate is assumed due to the redlines being inhibited.



LOM 95th	LOM 75th	LOM (mean)	LOM 50th (median)	LOM 25th	LOM 5th
1 in 68	1 in 96	1 in 112	1 in 120	1 in 147	1 in 194

Note: LOM mean = 1 in 124 applying 10% risk reduction due to inhibiting engine redlines.

Figure 6-48. LV 27.3 Cargo Variant LOM Estimates

The reliability model used Space Shuttle PRA data that was reviewed by propulsion engineers to incorporate potential upgrades for this vehicle. This led to the propulsion system reliability estimates in **Table 6-17**.

Table 6-17. Launch Vehicle 27.3 Propulsion System Failure Probabilities

Engine	Failure Probability (Cat)	Failure Probability (Ben)	Failure Probability (Start)	CFF	Error Factor
SSME	2.822E-04	1.482E-03	N/A	16.0%	2.6
RSRB (5-Segment HTPB)	3.484E-04	N/A	1.278E-05	N/A	1.8

The use of dual RSRBs is assumed the same as the current STS configuration. The Space Shuttle PRA data for the four-segment PBAN RSRBs was modified for the incorporation of the five-segment HTPB RSRBs. See **Section 6.8**, **LV Reliability and Safety Analysis**, for a description of the methodology used for determining the reliability of the other RSRB configurations used in the study.

The payload shroud reliability used in the ESAS was generated from a separate off-line analysis. The complete reliability analysis results are presented in **Section 6.8**, **LV Reliability and Safety Analysis**.

Figure 6-49 and **Table 6-18** show the LV 27.3 subsystem risk contributions. The vehicle risk is dominated by the multiple SSMEs on the core stage.

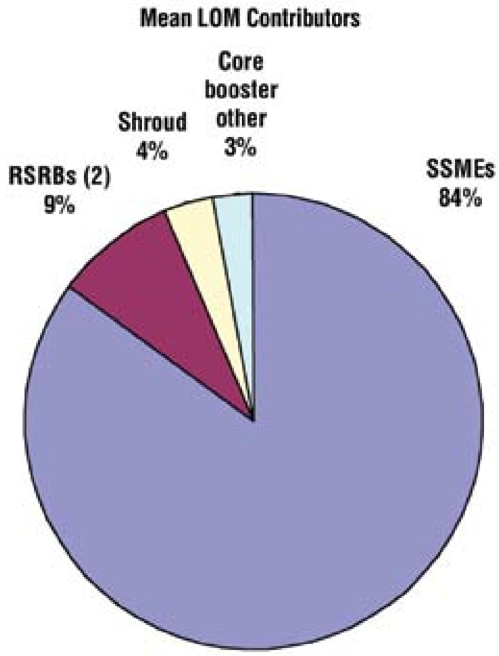


Figure 6-49. LV Subsystem Risk Contributions

Table 6-18. LV Subsystem Risk Contributions

	Mean Failure Probability	MFBF
RSRB (2)	5.7437E-04	1 in 1,741
RSRB Separation	2.1219E-04	1 in 4,713
Core Booster Engine Instantaneous Catastrophic Failure (ICF)	1.1075E-03	1 in 903
Core Booster Engine Benign Failure (BGN)	6.4955E-05	1 in 31,736
Core Booster APU	3.1510E-05	1 in 31,736
Core Booster TCS	1.0800E-09	1 in >1,000,000
Core Booster PMS	1.8401E-04	1 in 5,435
Core Booster TVC	2.3633E-05	1 in 42,314
Shroud	3.2464E-04	1 in 3,080
LOM	8.9246E-03	1 in 112

Note: LOM mean is 1 in 124 assuming 10% reduction due to inhibiting redlines. MFBF = Mean Flights Between Failures

To validate these results, a simple mean reliability model was developed. The model calculates FOMs by multiplying reliabilities for this quick illustration. The results of this model (LOM = 1 in 120 (1 in 133 with 10 percent risk reduction for inhibiting engine redlines)) compare favorably with the results of the FIRST model (LOM = 1 in 112 (1 in 124 with 10 percent risk reduction for inhibiting engine redlines)). This affirms the reliability estimates for LV 27.3. Complete results are provided in **Appendix 6D**, **Safety and Reliability**.

6.6.3.5 Schedule Assessment

There were no detailed development schedules generated for the CaLV options because they have a much later IOC than the CLV. The consensus was that the more clean-sheet EELV-derived design would require a longer development time than the Shuttle-derived solutions due to using well-characterized heritage systems (i.e., SRB, SSME). The additional upper stage required for the EELV concepts was also considered a driving factor. Assuming the traffic model for the first flight to be in 2017, the development would likely be 6–8 yrs, depending on the chosen option.

6.6.4 Cost Analysis Assumptions for Launch Families

The cost estimates developed for the family assessments continued to use NAFCOM for production of the DDT&E and TFU costs. However, rather than costing each vehicle as an independent, stand-alone concept, the family approach assumed an evolved methodology. Each family develops a CLV first. The first LV in the family will lift crew plus a limited amount of cargo per launch to the ISS. The second vehicle developed within the family will be used to lift heavy cargo and, in some families, crew also. Its development takes credit, wherever possible, for any development costs already paid for by the crew vehicle (engine development, software development, etc.). The cargo vehicle in the family may take some heritage credit where the subsystem is similar to the crew vehicle (i.e., thermal), thus reducing the development cost of the cargo vehicle. The discussion below deals with the DDT&E costs of the vehicle only. Facilities and test flight costs are not included. All launch family options using the Shuttle-derived CLV (LV 13.1) realize an additional savings from not incurring keep-alive costs for the SSME and RSRB facilities from STS retirement to CaLV development.

6.6.4.1 1.5-Launch Solution (LV 13.1 Followed by LV 27.3)

In the 1.5-launch solution family for lunar missions, the crew vehicle is the four-segment RSRB with a new upper stage using the SSME. The evolved vehicle in this family is an in-line HLV. The ET-based core uses five SSMEs, with two five-segment RSRBs as strap-ons. As an evolved vehicle from the crew vehicle, the cargo vehicle pays the development cost to make the SSME fully expendable. The crew vehicle paid for altitude-start and minimal changes to lower cost. In addition, some of the crew vehicle software can be either modified or reused. Test software, database software, and time/power management are a few of the functions that fall into this category. These savings are somewhat offset by the fact that the cargo vehicle must incur the development cost of the five-segment RSRB. The evolved cargo vehicle saves development costs as compared to stand-alone estimates. It should be noted that the cargo vehicle uses an EDS. This EDS is not included in the costs.

6.6.4.2 2-Launch Solution using Four-Segment RSRB as Core for Crew (LV 13.1 Followed by LV 26/27)

The 2-launch lunar mission solution family also begins with the four-segment RSRB with an SSME upper stage crew vehicle. The crew vehicle development pays for improvements to the SSME (altitude-start and minimal changes to lower cost). The evolved cargo vehicle is an ET-based core with four SSMEs and the five-segment RSRBs as strap-ons. The evolved vehicle pays the development cost to make the SSME fully expendable. In addition, some of the crew vehicle software can be either modified or reused. Test software, database software, and time/power management are a few of the functions that fall into this category. These savings are somewhat offset by the fact that the cargo vehicle must incur the development cost of the five-segment RSRB. The evolved cargo vehicle saves development costs as compared to stand-alone estimates.

6.6.4.3 2-Launch Solution Atlas Phase X (LV 2 Followed by LV 7.4/7.5)

This 2-launch lunar mission solution starts with a crewed version of the Atlas V HLV configuration that is human rated. The Atlas Phase X vehicle has an 8-m core stage, which uses five RD–180 engines. Two one-engine Atlas V boosters are used as strap-ons. The new upper stage uses four J–2S+ engines. Since it is known at the start of development that the Atlas Phase X vehicle will be used both as a crew and cargo vehicle, development costs include the shroud development. Some additional test hardware and testing will be needed to test for both missions. This provides savings over development of two similar vehicles.

6.6.4.4 2-Launch Solution Atlas Phase 3A (LV 2 Followed by LV 11)

This 2-launch lunar mission solution starts with a crewed version of the Atlas V HLV configuration that is human rated. This human-rated Atlas V has one RD–180 in a 4.3-m core, with four RL–10–4–A engines in the upper stage. The Atlas Phase 3A vehicle has a 5-m core stage, but uses two RD–180 engines. This new core is then used as four strap-ons in the follow-on vehicle. The new upper stage on the follow-on vehicle uses four LR–60 engines. Many of the subsystems will receive only minor changes for the new follow-on vehicle. More modifications will be needed for the MPSs due to the increased number of engines in the core and the new engines in the upper stage. This provides savings over development of two vehicles.

6.6.4.5 2-Launch Solution Atlas Phase 3A (LV 9 Followed by LV 11)

The crew vehicle of this 2-launch lunar mission solution family begins with the Atlas Phase 2, where the 5-m core stage uses two RD–180 engines. The new upper stage uses four new LR–60 engines. The follow-on cargo vehicle takes the 5-m core from the crew vehicle as the core of the cargo vehicle. With some minor development for separation systems and attachments, this same core is used for four strap-ons to provide the additional lift required for cargo. A new shroud is also developed for the cargo vehicle. The upper stage is essentially the same upper stage as the crew vehicle. A full structural test article was included. This family approach produced cost savings.

6.6.4.6 3-Launch Solution Four-Segment RSRB (LV 13.1 Followed by LV 25)

This 3-launch lunar mission solution family begins with the four-segment RSRB with an SSME upper stage crew vehicle. The crew vehicle pays for improvements to the SSME (altitude-start and minimal changes to lower cost). The cargo vehicle is an in-line ET-based core using three SSMEs. The crew vehicle paid for altitude-start and minimal changes to lower cost. In addition, some of the crew vehicle software can be either modified or reused. Test software, database software, and time/power management are a few of the functions that fall into this category. Attached to this core are two four-segment RSRBs and a new shroud. The family shares the SSMEs and the four-segment RSRBs, allowing for savings over separate estimates.

6.6.4.7 3-Launch Solution Four-Segment RSRB (LV 13.1 Followed by LV 20)

This 3-launch lunar mission solution family begins with the four-segment RSRB with an SSME upper stage crew vehicle. The crew vehicle pays for improvements to the SSME (altitude-start and minimal changes to lower cost). The cargo vehicle is a Shuttle-derived side-mounted configuration. The ET and two four-segment RSRBs provide boost capability. The ET is in production today. A new payload carrier using four SSMEs will be developed to carry cargo and will be attached to the side of the ET. This evolved approach saves money over estimating the vehicles separately.

6.6.4.8 3-Launch Solution Five-Segment RSRB (LV 15 Followed by LV 21)

This 3-launch lunar mission solution family starts with a five-segment single RSRB in-line crew vehicle. The new upper stage uses four new LR–85 expander cycle engines. The cargo vehicle is a Shuttle-derived side-mounted configuration. The ET and two five-segment RSRBs provide boost capability. The ET is in production today. A new payload carrier using four SSMEs will be developed to carry cargo. The payload carrier is attached to the side of the ET. Since the crew vehicle does not use SSMEs, the CaLV must incur the total cost of development of the engine and sustainment of the production capacity. The five-segment RSRB development cost is included in the crew vehicle. This evolved approach allows for limited cost savings over stand-alone estimates.

6.7 Earth Departure Stage

6.7.1 Summary of EDS Trades

The ESAS EDS studies covered three separate key trades: (1) the number of EDSs required to accomplish the lunar missions, (2) potential commonality between the EDS and the LV upper stages, and (3) the thrust level and number and type of engines on the EDS.

The EDS was initially considered to provide the propulsion delta-V function for four different mission phases of a lunar mission: LEO circularization from 30- by 160-nmi to 160-nmi circular, TLI, Lunar Orbit Insertion (LOI), and, finally, the lunar orbit plane change maneuver. **Figure 6-50** displays these mission phases. A fifth phase was added later to consider using the EDS during launch to place the payload including EDS into a 30- by 160- nmi orbit. The assumed delta-Vs (DVs) for each of these maneuvers are shown in **Table 6-21**. The delta-V for the LEO burn varied depending on LV performance.

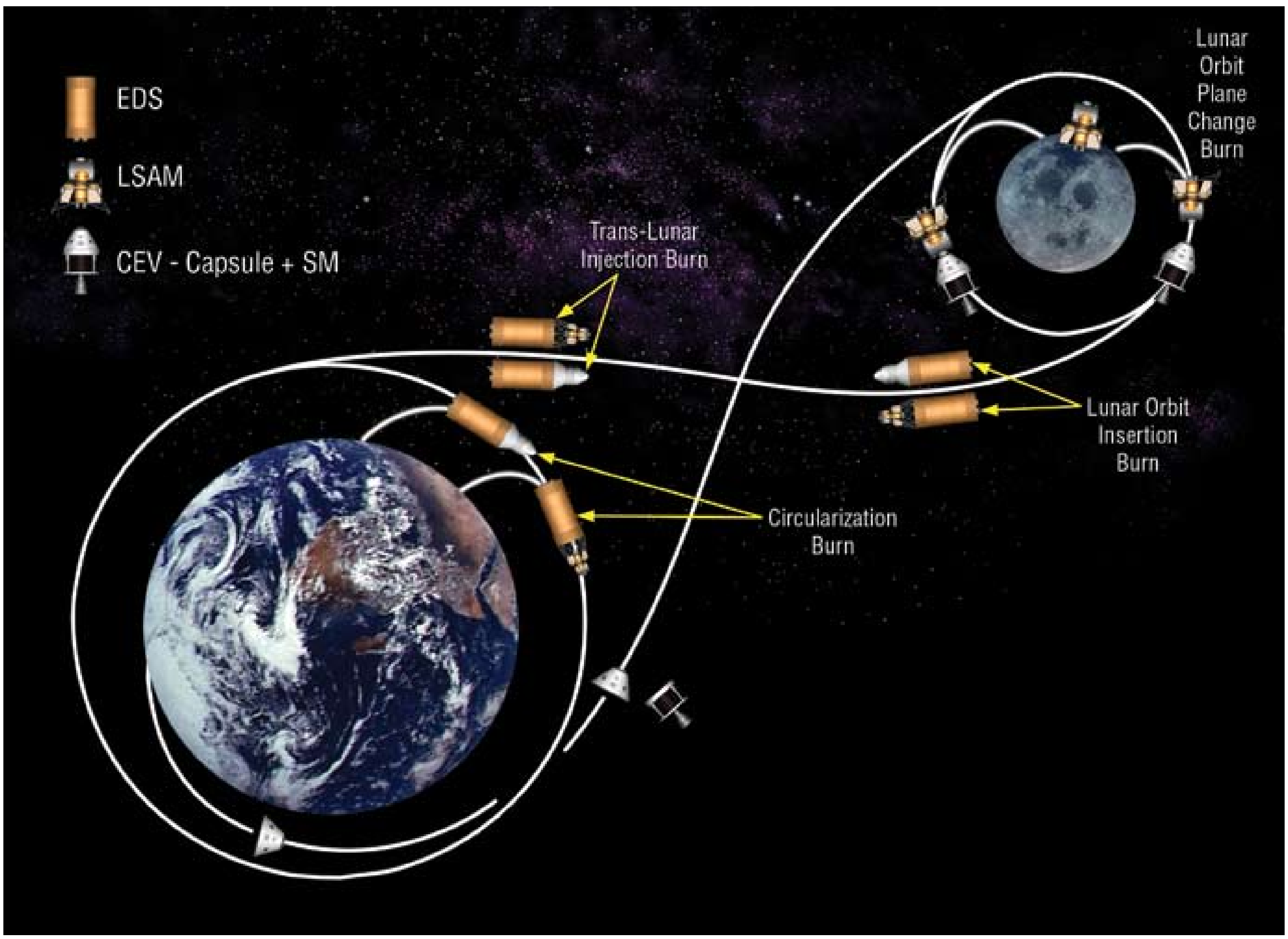


Figure 6-50. Potential EDS Mission Functions

Circ DV	78.6 m/sec (258 ft/sec) (30 x 160 nmi to 160 nmi)
TLI DV	3,120 m/sec (10,236 ft/sec)
LOI DV	890 m/sec (2,920 ft/sec)
Plane Change DV	510 m/sec (1,673 ft/sec)
Total Max DV	4,599 m/sec (15,087 ft/sec)

Table 6-19. Assumed Delta-Vs for Potential Mission Functions

6.7.2 Number of EDSs Required to Accomplish Lunar Mission

The objective of this trade was to define the bounds that set the desired number of EDSs for the EIRA lunar mission. To focus this trade, three key questions were identified that must be answered, including:

- What are the LV limits of mass to LEO?
- What are the in-space mission element masses for EIRA lunar mission (CEV, LSAM)?
- What effect does the concept of operations have on EDS design? For example, will the mission use split parallel flights to deliver the LSAM and CEV to the Moon or use a single-shot flight with combined LSAM and CEV in LEO?

The products from this trade consist of the data on the bounding constraints for selecting the number of EDSs per mission.

The LV limits were identified from the LV trades. Early in the vehicle study, it was decided to limit the number of launches to accomplish a mission to four launches. Later it was decided to discard vehicle options that could not lift at least 70 mT. **Figures 6-51** and **6-52** show the size relationship between the EDS and its lunar delivery capability with the identified LV limits.

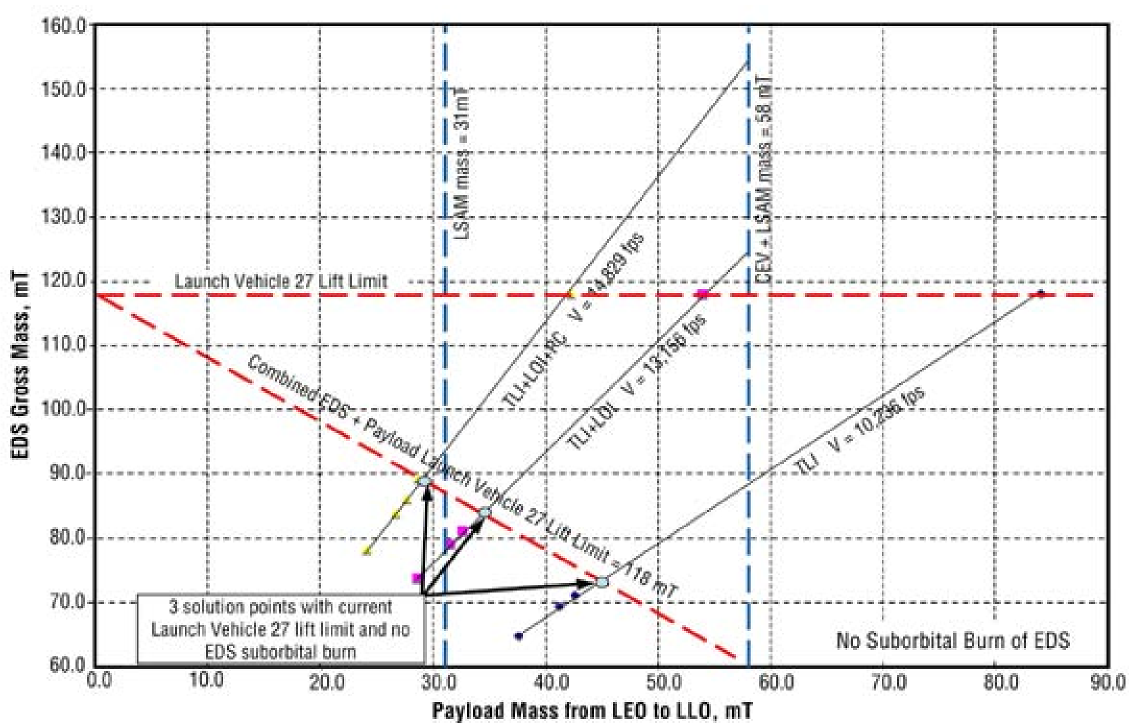


Figure 6-51. EDS Gross Mass Versus Payload Mass from LEO to Moon

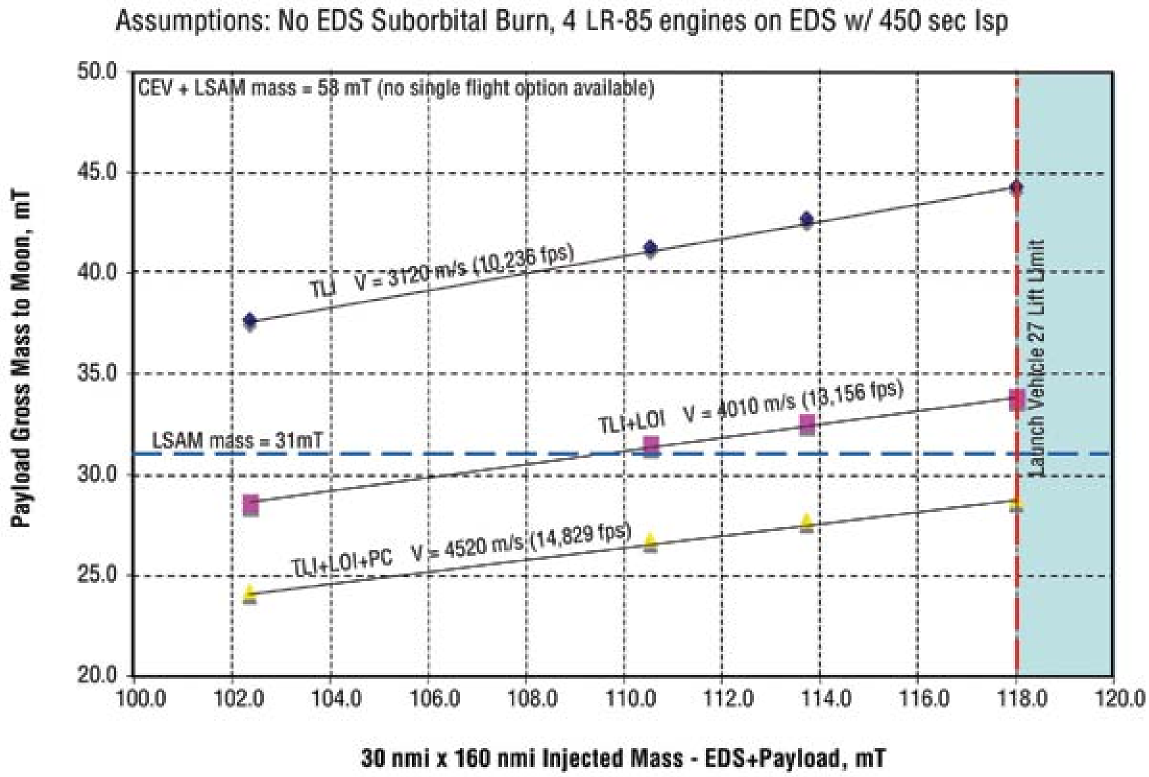


Figure 6-52. Payload Gross Mass to Moon Versus Total Injected Mass EDS + Payload

The EIRA masses for the CEV and the LSAM are 27 mT and 31 mT, respectively. The concept of operations for the EIRA architecture assumes two parallel flights with the LSAM carried with one EDS, and the CEV carried with another EDS. An alternative architecture approach uses one EDS pushing both CEV and LSAM in one all-up flight to the Moon, or it could use two EDSs burning in series with the CEV and LSAM together. With the LV lower lift limits that were set earlier, this means that no more than two EDSs are required per lunar mission.

There are several launch scenarios available to put up all the elements assembled for each mission. **Figure 6-53** depicts three potential launch solution sets for various mission concepts of operations. The actual sizes of the EDS depend on the specific LV option selected for the cargo and crew launches. **Figures 6-54** and **6-55** show the sizes of several options based on specific LV option lift constraints and the mission architectures described in **Section 4.2**, **Lunar Mission Mode**. A number of LV/EDS combinations were explored, providing essential architecture and vehicle trade sensitivities for the ESAS team.

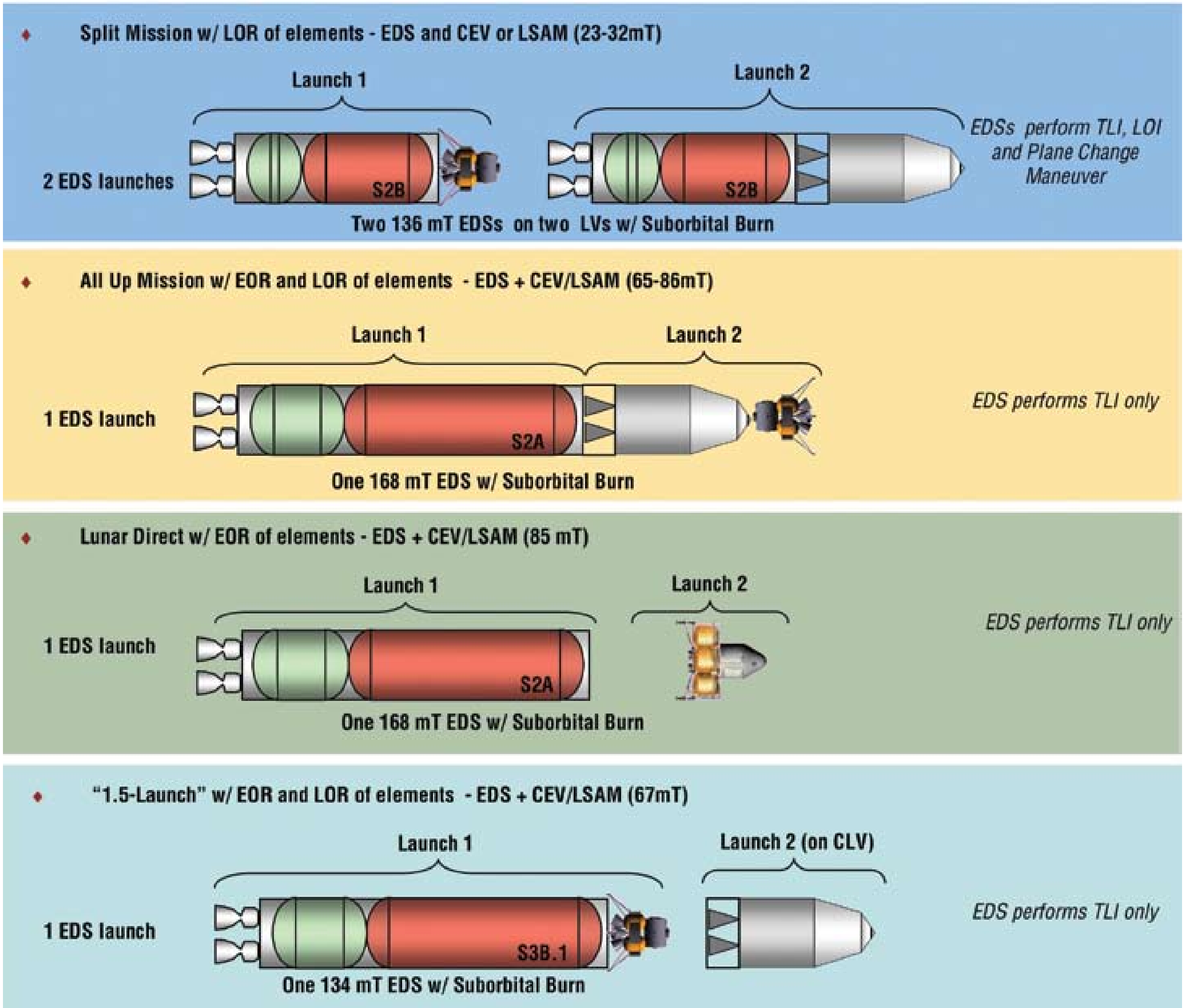
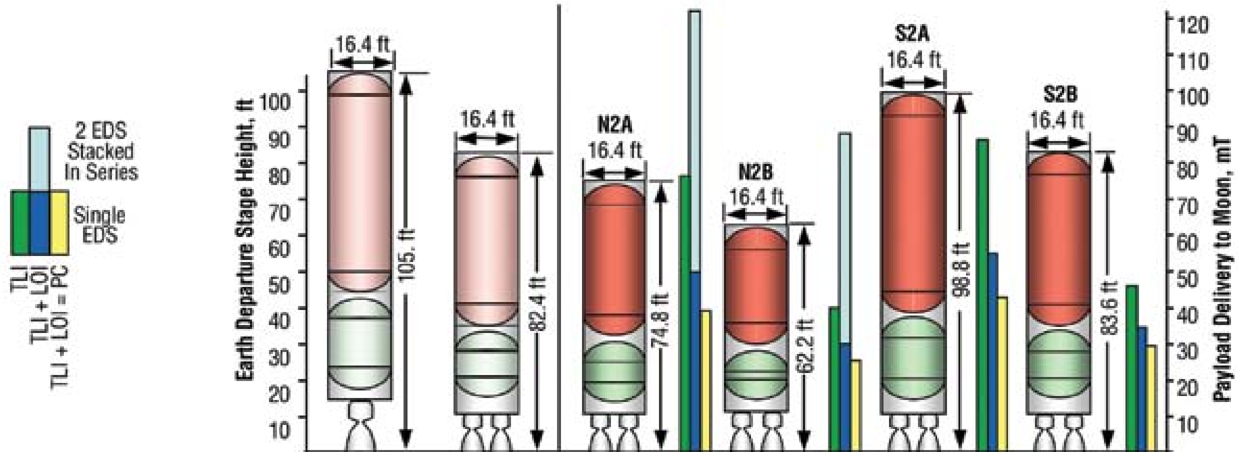
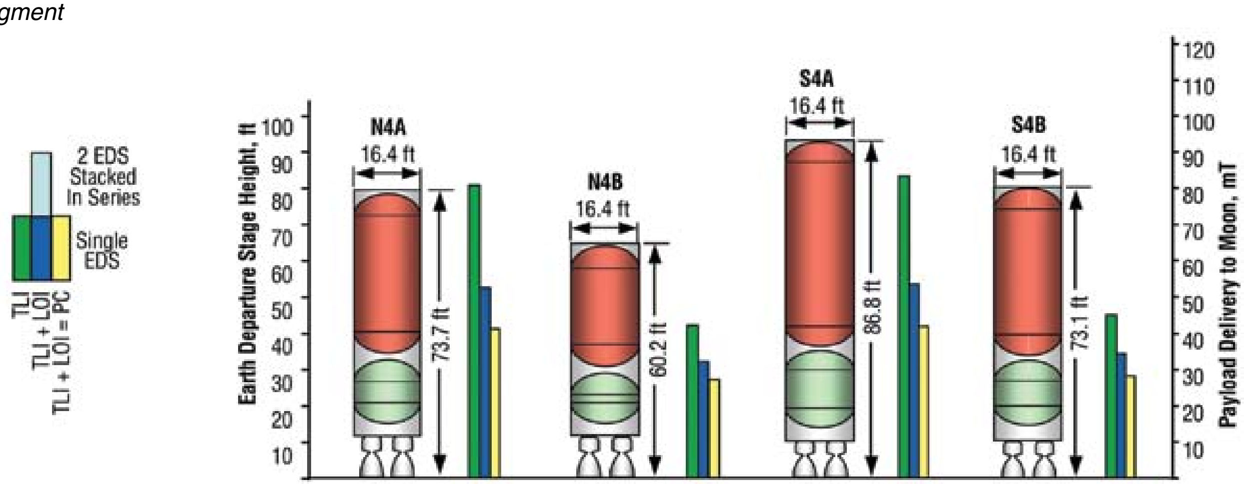


Figure 6-53. Potential Launch Solution Sets for Various Mission Concepts of Operations



	4-Seg RSRM w/ SSME Crew US	5-Seg RSRM w/ LR-85 Crew US	EDS No Payload at Launch*, No Suborbital Burn	EDS w/ Payload at Launch, No Suborbital Burn	EDS No Payload at Launch*, w/ Suborbital Burn	EDS w/ Payload at Launch, w/ Suborbital Burn
Stage Gross Mass @ Liftoff	405,882 lbm	296,198 lbm	260,224 lbf	196,937 lbf	370,430 lbm	299,190 lbf
Useable Propellant @ Liftoff	360,519 lbm	260,093 lbm	228,182 lbm	168,433 lbm	332,401 lbm	265,344 lbm
Useable Propellant @ 160nmi cir.	na	na	228,182 lbm	168,433 lbm	254,623 lbm	187,563 lbm
Stage pmf	0.8882	0.8781	0.8769	0.8553	0.8973	0.8869
Dry Mass	38,597 lbm	30,966 lbm	29,213 lbm	26,324 lbm	34,112 lbm	30,623 lbm
Burnout Mass	45,022 lbm	36,075 lbm	32,016 lbm	28,485 lbm	37,991 lbm	33,816 lbm

Figure 6-54. EDS Sizes Launched on In-Line SDV with 5-Segment RSRB



	EDS No Payload at Launch*, No Suborbital Burn	EDS w/ Payload at Launch, No Suborbital Burn	EDS No Payload at Launch*, w/ Suborbital Burn	EDS w/ Payload at Launch, w/ Suborbital Burn
Stage Gross Mass @ Liftoff	255,011 lbf	192,065 lbf	314,275 lbm	250,128 lbf
Useable Propellant @ Liftoff	219,710 lbm	165,048 lbm	279,321 lbm	218,980 lbm
Useable Propellant @ 160nmi cir.	219,710 lbm	165,048 lbm	230,426 lbm	170,084 lbm
Stage pmf	0.879	0.859	0.889	0.876
Dry Mass	28,020 lbm	24,884 lbm	31,585 lbm	28,416 lbm
Burnout Mass	30,772 lbm	26,999 lbm	34,922 lbm	31,123 lbm

Figure 6-55. E Launched on Evolved Heavy Lift 8-m Core Vehicle

6.7.3 Potential Commonality between EDS and LV Upper Stages

A previous study performed during the summer of 2004 validated the potential benefits and feasibility of commonality between the EDS and the LV upper stages. The preliminary assessment data indicated that there are several possible evolutionary development scenarios that could save significant money and reduce program risk.

The objective of this study was to build on the earlier commonality study and identify specific commonality elements for the various candidate LV upper stages and the EDS. During this study, EDS configuration benefits were examined for tank and stage diameter commonality and propulsion system element commonality, and the cost savings were analyzed for several vehicle families. **Figure 6-56** shows three of the families examined, and **Table 6-20** shows the relative DDT&E costs for the various EDSs with and without commonality. **Table 6-21** shows the potential common elements that could be used between an LV upper stage and the EDS. **Table 6-22** shows new stage elements that will be required for an EDS if evolved from an LV upper stage.

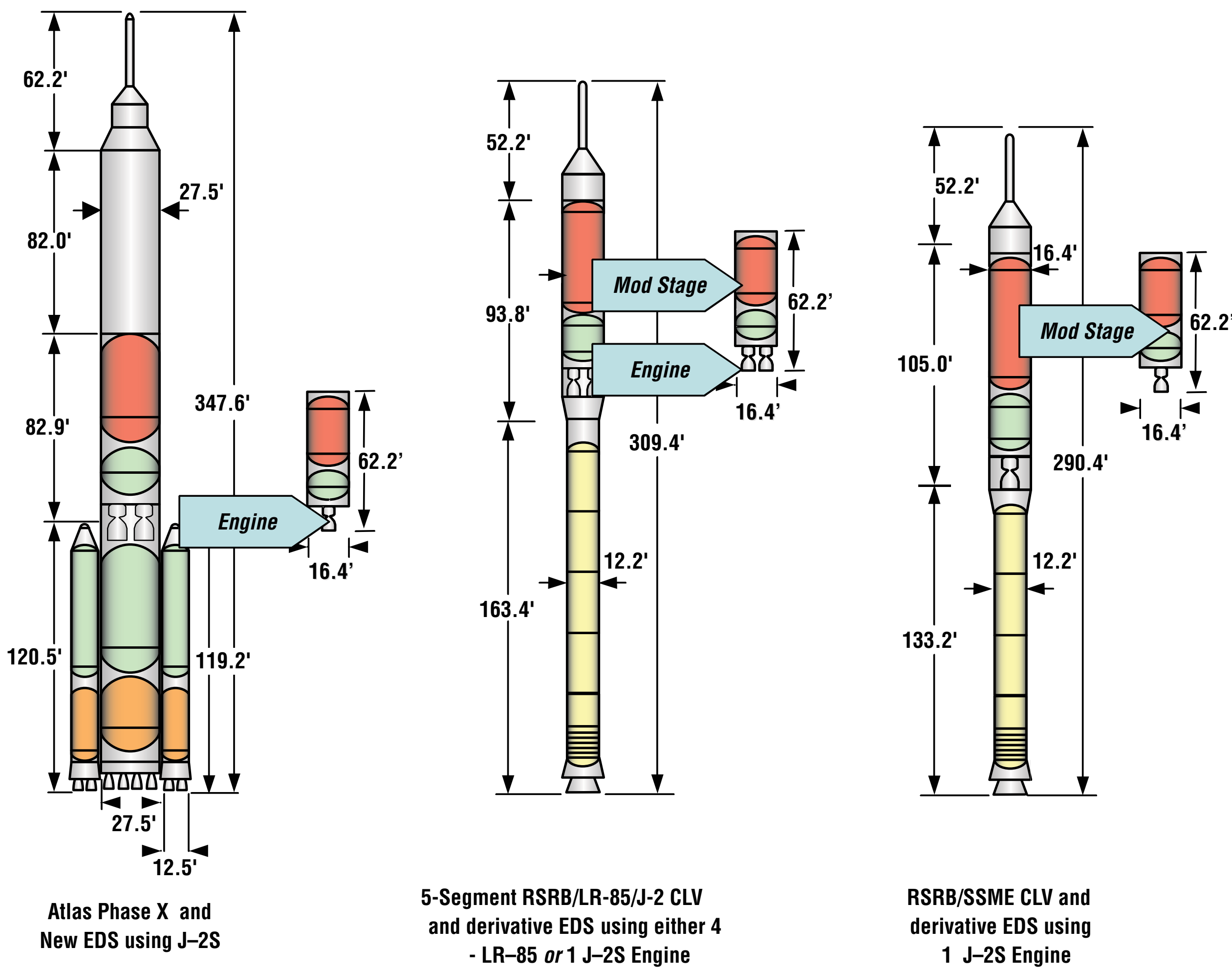


Figure 6-56. LV/EDS Commonality Family Combinations

Table 6-20. EDS Development Cost Comparison Between Clean-sheet and Family Commonality

WBS Element	EDS Development Cost					
	Atlas Phase X EDS Family		5-Segment RSRB/LR–85/ J–2 CLV EDS Family		RSRB/SSME CLV EDS Family	
	Clean-sheet DDT&E	Family DEV DDT&E	Clean-sheet DDT&E	Family DEV DDT&E	Clean-sheet DDT&E	Family DEV DDT&E
EDS	2.630	1.598	2.630	1.931	2.630	1.000
EDS Subsystems	0.306	1.624	3.062	2.281	3.062	1.000
Structures and Mechanisms	1.620	1.620	1.620	1.000	1.620	1.000
TCS	1.000	1.000	1.000	1.000	1.000	1.000
MPS	1.000	1.000	1.000	1.000	1.000	1.000
Reaction Control Subsystem	0	0	0	0	0	0
Electrical Power and Distribution	8.823	8.823	8.823	1.000	8.823	1.000
CCDH	3.245	3.245	3.245	1.000	3.245	1.000
GN&C	3.627	3.627	3.627	1.000	3.627	1.000
Software	1.000	1.000	1.000	1.000	1.000	1.000
Range Safety	0	0	0	0	0	0
Liquid Rocket Engine- 1 J–2	6.636	0.307	6.636	6.636	6.636	1.000
EDS System Integration	1.480	1.527	1.480	1.000	1.480	1.000
Integration, Assembly, and Checkout (IA&C)	1.000	0.892	1.000	1.000	1.000	1.000
System Test Operations (STO)	1.000	1.000	1.000	1.000	1.000	1.000
Ground Support Equipment (GSE)	1.000	1.000	1.000	1.000	1.000	1.000
System Engineering and Integration (SE&I)	2.012	2.161	2.012	1.000	2.012	1.000
Program Management	1.689	1.778	1.689	1.000	1.689	1.000
Fee	2.630	1.597	2.630	1.930	2.630	1.000
Program Support	0	0	0	0	0	0
Contingency	0	0	0	0	0	0
Vehicle Level Integration	2.626	1.596	2.626	1.930	2.626	1.000

Table 6-21. Potential EDS Common Elements with a LV Upper Stage

Launch infrastructure
Production and handling infrastructure
Adapters, PLF, and separation system
Avionics
Tank sections (cylinder plugs could enable multiple lengths)
Aft umbilicals for simplified ground operations
Aft thrust structure
Engine mounts and gimbals
Propulsion systems (main engine and feed system)

Avionics Basic avionics could be the same with upgrades for radiation hardening of avionics and power.	
Cryogenic Fluid Management Passive cryogenic TPS [Mission Peculiar Kit (MPK) insulation (sun shield, Multi-Layer Insulation (MLI)].	
TPS/Structures Micrometeoroid/Orbital Debris (MMOD) and radiation protection. Delta development cost for loads differences. Delta production cost for tank barrel length.	
Propulsion (main engine, RCS, pneumatics and feed system) Production may have different number of same engines. May require new Main Propulsion Test Article (MPTA). RCS requirements for Upper Stage and EDS are very different. EDS has commonality with other in-space elements and In-Situ Resource Utilization (ISRU), which could drive propellant selection as well as long duration.	

Table 6-22. New EDS Elements Required if Evolved from LV Upper Stage

While examining common elements, some specific questions arose regarding the use of the SSME. First, could the SSME be used on an upper stage with an altitude-start? Second, could the SSME be used on the EDS with a suborbital burn and then restarted on orbit for the TLI burn? Finally, if the EDS evolved from the CLV upper stage that uses an SSME, what MPS changes would be required to replace the SSME with a J–2S?

Several studies have been conducted to address the first question. One study performed in 1993 looked at the Phase 2 version of the SSME and concluded that it was feasible to use the engine in an altitude-start arrangement. The most recent in 2004 looked at the current Block 2 version. The goal of each study was to minimize any required modifications to the current SSME configuration and operation to reduce risk. The development test program for SSME modifications would need to address at least three key operational issues: (1) engine thermal conditioning required for start, (2) engine pre-start purging, and (3) engine start sequence modifications caused by the different environments. **Table 6-23** lists a number of modifications that would be required along with the related specific analysis and testing. In summary, SSME altitude-start for use as an upper stage engine is feasible with reasonable risk.

Modifications Modify augmented spark igniter orifice. Software updates for start sequence modifications (e.g., low main oxidizer valve ramp rate). Additional analysis required to determine operational modifications for LCCs, redlines, purges, chill down, etc.
Interface Requirement Pushback Require propellant settling motors, similar to SII (Saturn 2) and SIVB (Saturn V Third Stage), to ensure quality propellants at engine inlet. Recommended minimum propellant inlet pressures : LH2 36 psi; LOX 40 psi. Higher inlet pressures reduce start risk.
Development and Verification for Low Inlet Conditions During Altitude-Start Analyze Augmented Spark Igniter (ASI) to verify combustion ignition at low pressure (test in small vacuum facility). - Develop and demonstrate transients on NASA Stennis Space Center (SSC) Test Stand A1 with low inlet pressure(relocate existing small volume run tank to low evaluation). - Certify with altitude test. Ensures that issues associated with priming the MCC and nozzle cooling circuit are resolved (i.e., sensitivity of SSME start to fuel side oscillations that could result in high fuel side turbine temperature spikes).

Table 6-23. SSME Modifications to Enable Altitude-start

While it is feasible to use the SSME as an upper stage with altitude-start, it is not recommended at this time to use the SSME with an in-space restart. The SSME is an intricate and sensitive system. After each operation, the engine requires extensive drying using purges to remove any moisture that accumulates from the combustion products of LOX and LH2. Additionally, pre-start thermal conditioning would be a post-operation concern. Time would have to be allowed for the engine to cool down and be reconditioned for a restart. Additional thermal concerns would arise from the in-space radiation heating of the SSME hardware and how it would affect the sensitive start sequence. All of this would require that additional LOX and LH2 be carried on the stage, along with helium and nitrogen for purging. Also, after flight, the engine typically goes through an extensive inspection and verification process to ensure that it is acceptable for another flight. This would not be possible with an in-space restart.

The ESAS team investigated why the SSME was so sensitive to restart when the J–2/J–2S engine was able to perform this function for the Apollo/Saturn vehicle. The SSME is a dual preburner staged combustion cycle engine versus the Gas Generator (GG) or tap-off cycle for the J–2/J–2S engine. Propellant conditioning for the dual preburner cycle is more complicated. Proper propellant conditions are necessary for three separate synergistic ignition systems, while the GG cycle has only two that are independent, and the tap-off cycle has only one. The J–2 series was also designed with a spin assist start system, while the SSME uses a tank head start and "bootstraps" up. ("Bootstrapping" is an engine start procedure whereby there is enough stored energy in the form of pressure to initiate and maintain the engine ignition procedure.) The SSME was also designed with boost pumps to allow for lower tank pressures and, thus, to save vehicle weight. The J–2 series did not require boost pumps. On a complicated synergistic cycle like the SSME, the boost pumps have a significant effect on the start sequence. Although it is not technically impossible to use the SSME in this manner, it is considered to be a very high-risk and costly development effort.

6.7.4 Thrust Level and Number of Engines on the EDS

This last trade study objective was to determine if there is an optimum thrust for a common main engine/propulsion system for crew and/or cargo upper stage and EDS applications for current LV configurations. The key questions addressed for this study included:

- What is the EDS total thrust requirement?
- What is the EDS individual engine thrust requirement?
- What limits, if any, are there on engine cycle?
- What is the benefit, if any, to engine-out?
- How does the optimum EDS engine thrust work on the upper stage options for the various CLVs and CaLVs?

The output consisted of data on the bounding constraints for selecting the number and thrust level for EDS main engines.

The number and size of the EDS main engines depends on LV trajectory parameters, architecture approach, and engine design parameters. The trajectory parameters that affect this trade are the GN&C accuracy requirements and acceleration limits, and the EDS gravity losses that are set by the EDS initial stage T/W ratio. The architecture approach, such as split parallel TLI flights or a single all-up TLI flight, determines the total mass that is pushed at a given time to the Moon. Additionally, the use of the EDS to perform other mission functional requirements, such as LOI, TEI, or lunar orbit plane change maneuvers, also affects the trade results. The engine design parameters include maximum and minimum burn times, throttle requirements, and envelope. Programmatic boundaries, such as commonality with LV upper stages and whether to use existing engines, were also considered.

Figure 6-57 shows the potential LOX/LH2 engines that could be used for either an upper stage or the EDS. Technical descriptions and other information about the key candidates are discussed in **Section 6.9**, **LV Subsystem Descriptions and Risk Assessments**.

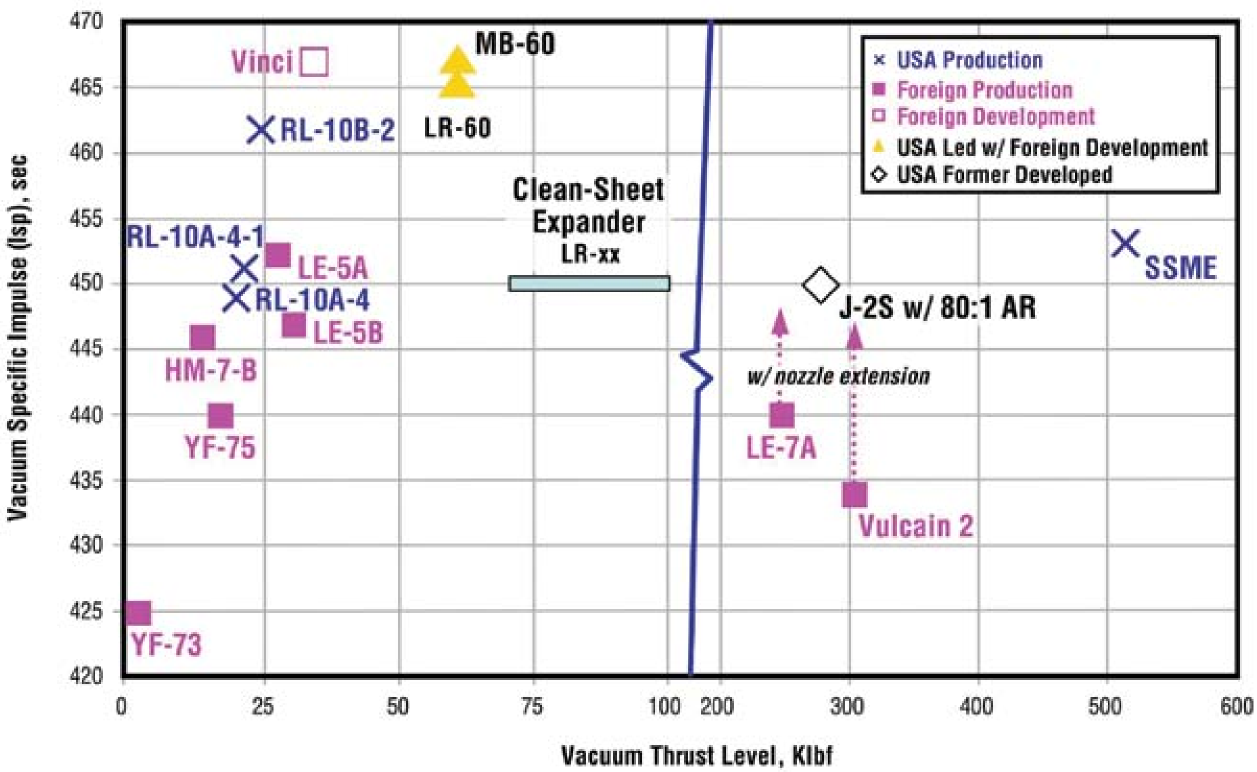


Figure 6-57. Candidate LOX/LH2 Engines for an Upper Stage and Earth Departure Stage Main Engine

From the LV conceptual design trade study, there are nine stage thrust classes. These are shown in **Table 6-24**. The stage thrust can be accomplished by several means. For example, the thrust requirement could be satisfied with a single engine or by multiple engines. For multiple engines, the stage could have engine-out or not. If the various combinations are determined for both the upper stages and the EDS, the data can be compared to determine the most probable desired thrust level. This comparison is shown in **Table 6-25**.

Table 6-24. LV Upper Stage Thrust Requirements

Stage Thrust (lbf)	Number of Engines	Type	Nominal Engine Thrust	Engine Power Level % to Achieve Stage Thrust
24,750	1	RL-10B-2	24,750	100.0
44,600	2	RL-10A-4-2	22,300	100.0
66,900	4	RL10A-4-2	22,300	75.0
240,000	4	LR–60	60,000	100.0
274,500	1	J–2S with 80:1 AR*	274,500	100.0
340,000	4	LR–85	85,000	100.0
400,000	4	LR-100	100,000	100.0
490,847	1	SSME	469,710	104.5
1,098,000	4	J–2S with 80:1 AR*	274,500	100.0

** Modified LE–7A or Vulcain 2 are international alternatives to J–2S.*

Table 6-25: Comparison of Upper Stage and EDS Thrust Requirements

			Split Parallel Flights w/72—180 klbf Thrust							Single TLI Flight Single EDS w/176—440 klbf						
		Number of Engines	1 NEO	4 NEO	4 NEO	3 NEO	3 NEO	2 NEO	2 NEO	1 NEO	4 NEO	4 NEO	3 NEO	3 NEO	2 NEO	2 NEO
		EDS Engine Thrust Levels, klbf	72–180	18–45	24–45	24–60	36–60	72–90	176–440	176–440	44–110	58–110	58–147	88–147	88–220	176–220
Stage Thrust Class, lbf	Engine Combinations	US Engine Thrust Levels, klbf														
24,750	Single Engine NEO	24.8		•	•	•										
44,600	Single Engine NEO	44.6		•	•	•	•	•			•					
	2 Engines NEO	22.3		•												
66,900	Single Engine NEO	66.9						•			•	•	•			
	4 Engines NEO	16.7														
	4 Engines EO	22.3		•												
	5 Engines EO	16.7														
240,000	Single Engine NEO	240.0								•						
	4 Engines NEO	60.0				•	•	•			•	•	•			
	4 Engines EO	80.0	•					•	•		•	•	•			
	5 Engines EO	60.0				•	•	•			•	•	•			
274,500	Single Engine NEO	274.5								•						
	4 Engines NEO	68.6						•								
	4 Engines EO	91.5	•								•	•	•	•	•	
	5 Engines EO	68.6						•								
340,000	1 Engine NEO	340.0								•						
	4 Engines NEO	85.0	•					•	•		•	•	•			
	4 Engines EO	100.0	•										•	•	•	
	5 Engines EO	85.0	•					•	•		•	•	•			
400,000	Single Engine NEO	400.0								•						
	4 Engines NEO	100.0	•								•	•	•	•	•	
	4 Engines EO	133.3	•										•	•	•	
	5 Engines EO	100.0	•								•	•	•	•	•	
490,847	1 Engine NEO	490.8														
	2 Engines NEO	245.4								•						
	4 Engines EO	163.6													•	
	5 Engines EO	122.7	•										•	•	•	
1,098,000	4 Engines NEO	274.5								•						
	5 Engines EO	274.5								•						

Upper Stage Engine-Out accomplished by operating all engines throttled until one shuts down.
Upper Stage Engine-Out calculated by Stage thrust/(number of engines–1)
EDS Engine-Out accomplished by operating all engines at full thrust and accepting the lower thrust.
EDS Engine-Out calculated by minimum thrust for mazimum engine burn of 600 sec.

The bounding trajectory parameter values that affect thrust were defined during the study. The upper limit on vehicle stage thrust is dictated by the GN&C accuracy requirement and any maximum acceleration load limits. The GN&C accuracy requirement sets the minimum engine burn time for each main propulsion burn maneuver at approximately 200 sec. The lower limit on vehicle stage thrust is dictated by the need to minimize the gravity losses. Increased gravity losses mean the need for additional propellant, and, thus, growth of the vehicle mass. **Figure 6-58** shows the delta-V loss as a function of stage initial T/W ratio. From this curve, the recommended lower limit on T/W ratio is 0.4. This value can then be multiplied to the total EDS plus payload mass to calculate the lower thrust limit for the EDS.

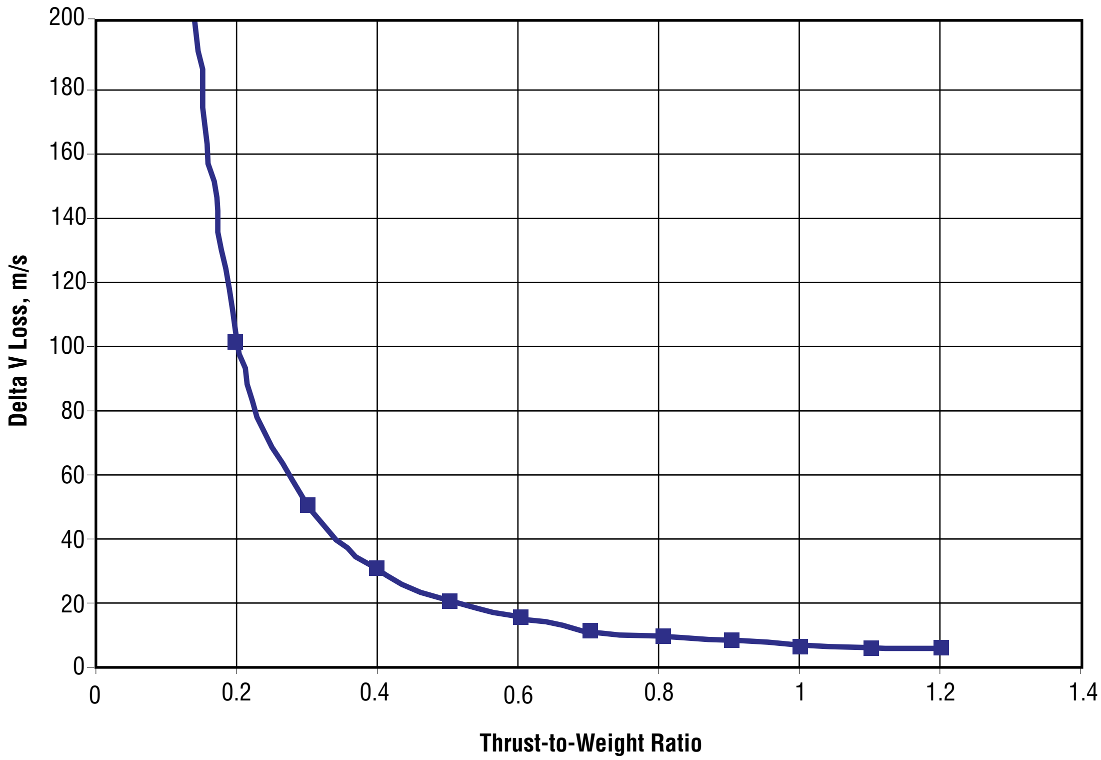


Figure 6-58. Gravity Losses for an EDS Single Burn from 400-km Circular LEO to TLI

The engine design parameters were set for the study based on historical experience. The maximum burn time has historically been approximately 600 sec. From a practical basis, the minimum burn time is set by the engine startup and shutdown sequences and this is generally approximately 5 sec. The engine throttle limit for pump-fed systems has been approximately 20 percent of full power, but this has only been demonstrated on a technology demonstration unit. The operational experience is with the SSME at 65 percent Rated Power Level (RPL). The 65 percent RPL value is actually 59 percent of the full power level of the SSME, which is 109 percent RPL.

When all these defining boundaries are taken into account, the stage and individual engine thrust levels can be set. **Figure 6-59** again shows the delta-V losses plotted against the stage T/W ratio, but with the addition of the defining limits. **Figure 6-59** also has the delta-V curves for the separate functional burns plotted instead of as one composite as in **Figure 6-58**. Combining the information of **Table 6-25** and **Figure 6-59** yields the information in **Figure 6-60**, which shows the various engine solutions for the range of EDSs being considered. An important observation from **Figure 6-60** led to a decision late in the study to have only the EDS perform the TLI burn. There was not a stage thrust level or individual engine combination that would allow the EDS to perform the LOI and plane change maneuvers without violating at least one of the bounding limits.

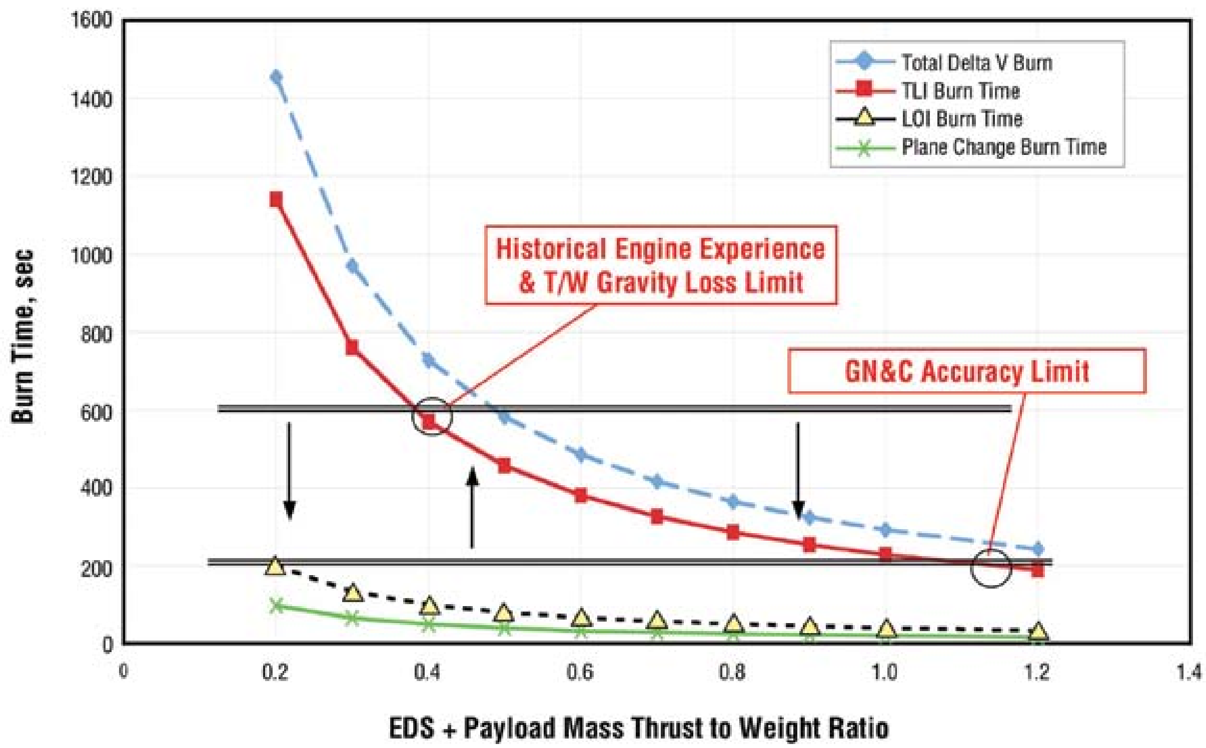


Figure 6-59. Bounding Limits on EDS Thrust and Engine Thrust

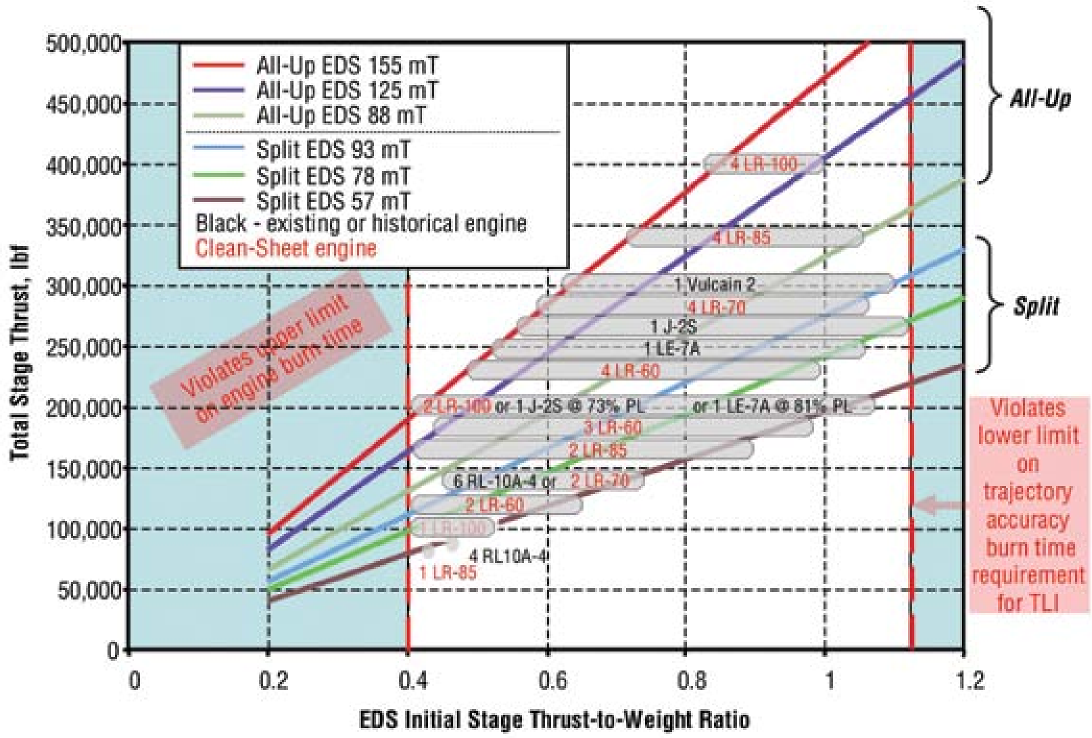


Figure 6-60. EDS Thrust Values

Note: *All Up EDS delivers 58 mT payload*
Split EDS delivers 31 mT payload

6.7.5 EDS Reliability Model

Appendix 6D, **Safety and Reliability**, outlines a stage reliability model that was developed to support the ESAS activity. The reliability model was applied to the 1.5-launch configuration described previously. **Figure 6-61** shows the results of the model runs for the three engine burns required for the 1.5-launch configuration EDS. The results are broken out to indicate the contribution from MPS, engine start failures, and engine main-stage failures. The first two burns are the suborbital and circularization burns and do not have engine-out capability. Burn number 3 shows the best reliability due to engine-out capability for the TLI burn.

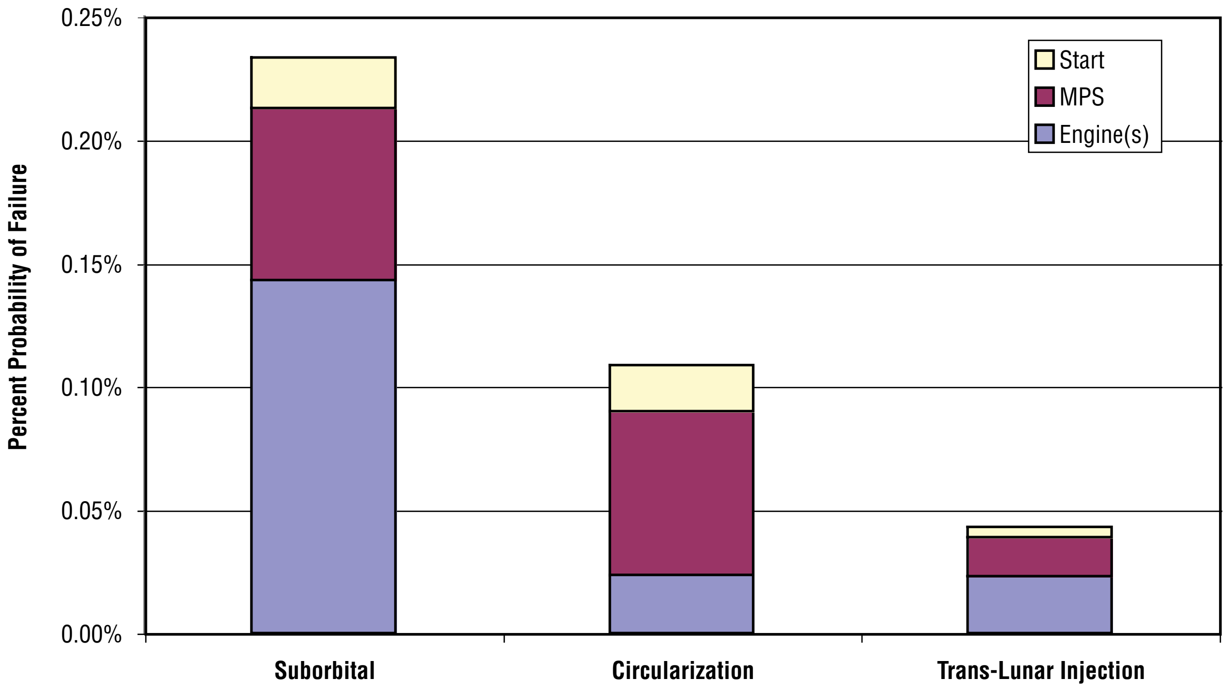


Figure 6-61. 1.5-Launch Configuration EDS Reliability Results

6.7.6 Results Summary

The final results from the EDS thrust and number of engine trade study are summarized in this section. A number of solutions work for both all-up and split missions. The all-up-mission EDS could use modified existing engines or designs (J–2S+, LE–7A, Vulcain 2) or multiple clean-sheet expander cycle engines (70- to 100-klbf of thrust). The split-mission EDS could again use modified existing engines or designs (RL–10A–4, 73-percent J–2S+, 81-percent LE–7A) or multiple clean-sheet expander cycle engines (70- to 100-klbf of thrust). If the EDS performs a suborbital burn, then maximum thrust would be set by this requirement, not the in-space TLI burn. Additionally, an assessment of potential engine-out benefits may set the number and thrust of engines. Finally, as mentioned previously, there is no solution that can satisfy GN&C minimum burn time for LOI and lunar plane change delta-V maneuvers with a single-engine propulsion for TLI. T/W values less than 0.2 (initial EDS + payload T/W) would be required. The required thrust ranges from 27 klbf for split missions to 67 klbf for all-up missions. If the GN&C accuracy is compromised and the minimum burn time dropped to the 100s, then the required thrust range becomes 53 to 133 klbf.

There are two possible solutions. Use multiple engines on the EDS (at least four) and only burn one of them for LOI and plane change; or the preferred approach of changing the delta V split between elements and using the CEV/LSAM propulsion for LOI and plane change burns.

6.8 LV Reliability and Safety Analysis

Reliability and safety analyses were performed on all of the LV propulsion components. These components consist of RSRBs, liquid propellant rocket engine systems, the MPS, the ET, APUs, separation systems, and the payload shroud. Each component is discussed in the following sections, along with some additional RSRB safety considerations.

6.8.1 Reusable Solid Rocket Boosters

6.8.1.1 Overview

The similarity model (FIRST) analysis uses the data from the Space Shuttle Quantitative Risk Assessment System (QRAS) 2000 failure estimates for the Space Shuttle four-segment RSRB pair using PBAN propellant. The QRAS model is composed of separate estimates for the RSRM and the SRB failures, and the ESAS (FIRST) model maintains this separation in the similarity analyses for the RSRBs in the study. The model allows users to select the propellant (PBAN or HTPB), the number of segments in a RSRM, and the number of RSRBs on the vehicle; then it provides a probability of failure per mission for the selection. ESAS examined vehicles using a single four-segment RSRB or a pair of five-segment RSRBs. Several corrections and assumptions were made to the QRAS RSRB for the ESAS vehicle models to account for the varying number of segments in the RSRM, the different propellants, the use of a single SRB (with grossly different impacts from a start failure than that for a Shuttle-like design), and modifications necessary for an in-line separation. The methodologies for determining the reliability of the SRBs used in the study are detailed in the following sections.

6.8.1.2 Similarity Analyses

QRAS failure probabilities of every component for a pair of RSRMs and SRBs were halved to represent a single RSRB. The single- and dual-RSRM/SRB component reliability estimates from QRAS are provided in **Appendix 6D**, **Safety and Reliability**. ESAS used failure probabilities of all RSRB components "as is," with the exceptions described below.

6.8.1.2.1 Five-Segment RSRB

To account for the addition of an RSRM center segment for the five-segment RSRM, the QRAS probability of failure calculations are augmented with the addition of risks associated with an additional case field joint and case factory joint for the extra center segment, and scaling other failure values proportionally for the difference in propellant quantities, motor length, and exposed areas. Additionally, the use of the new propellant HTPB was included in RSRM modeling for the five-segment RSRB. Discussions with ATK Thiokol led to a preliminary conclusion that HTPB is considered intrinsically more reliable than the nominal PBAN propellant because its strain capability is 30 to 50 percent greater. This increased capability is due primarily to its smaller modulus of elasticity (2,000 psi versus 3,300 psi). Given the consensus that HTPB provides superior reliability and that researched material properties are consistent with this conclusion, the ESAS model (FIRST) quantifies improvements in safety resulting from HTPB with a 50 percent risk reduction to the QRAS failure mode "Propellant Energy" for the five-segment RSRM with HTPB propellant.

6.8.1.2.2 Single RSRB

The primary differences in RSRB-related failures on a vehicle with a single RSRB compared to the use of side-mount pairs on the Shuttle design are due to startup and separation.

For the two side-mount RSRBs on the Shuttle, any single RSRM startup failure would result in the catastrophic loss of the entire Space Shuttle vehicle stack due to the unbalanced thrust. For vehicles in this study with a single SRB, such as the in-line configuration, an RSRM startup failure would be merely a hold-down event. To account for this, the ESAS model (FIRST) removes the QRAS RSRM failure contributor "Igniter and Main Propellant Ignition" for a single RSRB in-line design.

Because the separation modes and mechanisms for a single RSRB in-line design are significantly different than for side-mount RSRBs, the portion of the QRAS RSRB estimate dealing with Shuttle RSRB separation is removed and applied independently to the modeling of separation systems. The QRAS SRB failure designations removed are "Separation System" and "Booster Separation Motor (BSM)."

6.8.1.3 RSRB Failure Probabilities

Table 6-26 represents the combined risk of the SRB and RSRM to form the RSRB risk values along with start probabilities and Error Factors (EFs). Note that the catastrophic rates do not contain the separation risk for the SRB, because the ESAS model calculates it outside the RSRB model, nor does it contain the startup risk (listed separately). The ESAS model accepts the EFs accompanying the original QRAS data but applies a higher EF for the newer propellant HTPB in the model.

EF is an abbreviated presentation of the uncertainty in a probabilistic distribution. Defined as the ratio of the 95th percentile to the 50th, EFs may be generated from historical data for components, or, in the case of components with inadequate historical data, EF may be estimated using similarity analyses and engineering judgment. In this case, a higher EF was assumed because less is known about the HTPB propellant and no test data was available.

Booster Type	P_{ICF}	P_{BEN}	P	CFF	EF
RSRB (4-Segment PBAN)	2.72E–04	N/A	1.28E–05	N/A	1.7
RSRB (5-Segment PBAN)	5.76E–04	N/A	1.28E–05	N/A	1.7
RSRB (4-Segment HTPB)	2.71E–04	N/A	1.28E–05	N/A	1.8
RSRB (5-Segment HTPB)	2.74E–04	N/A	1.28E–05	N/A	1.8

Table 6-26. RSRB Multi-Segment Risks for Two Propellants

Figure 6-62 shows how the model alters the RSRB QRAS data. Note that the various separation models are included in the flow and in the overall number.

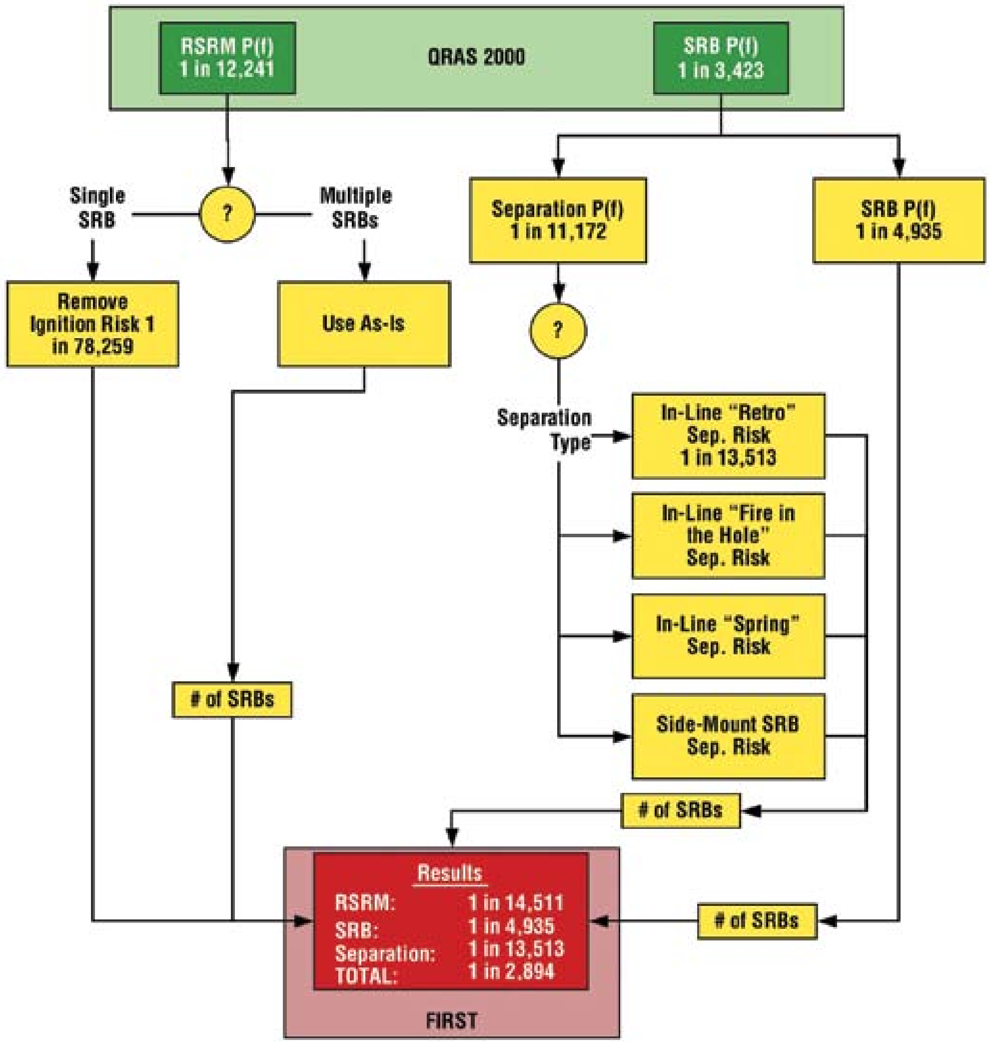


Figure 6-62. RSRB Risk Development from QRAS Results

6.8.2 Liquid Propellant Rocket Engine Systems

6.8.2.1 Overview

This safety and reliability summary delineates the trades and analyses that the ESAS team conducted. Benign and catastrophic failure probabilities and catastrophic failure fractions were generated as metrics in the analyses.

Relevant terms used in the study include:

- System Safety Analysis: The application of management and engineering principles, methods, models, and processes to optimize the safety (prevention of death and injury to personnel and protection of valuable assets) throughout the life cycle of the system.
- Reliability: Probability that an item will perform a required function under stated conditions for a stated period of time.
- Safety: The elimination of hazardous conditions that could cause death, injury, or damage to people or valuable property.

- Risk: The likelihood that an undesirable event will occur combined with the magnitude of its consequences.
- Catastrophic Failure: An immediate uncontained failure that leads to energetic disassembly of the engine and vehicle.
- Benign Failure: A failure that does not directly lead to loss of engine or vehicle, but may lead to an abort situation.
- Catastrophic Failure Fraction (CFF): The ratio of the catastrophic failure rate to the total probability of benign and catastrophic failures (CFF = Catastrophic/[Benign + Catastrophic]), reported as a fraction.

Propulsion systems and their components are heavy contributors to the unreliability of an LV. **Figure 6-63** shows element results from the Shuttle QRAS PRA, and **Figure 6-64** reflects component contributions to unreliability from SSME. It is notable, due to the nature of the QRAS PRA, that benign engine failures were not included in the analysis. Inclusion of benign failures would significantly increase the predominance of propulsion systems in unreliability predictions for the STS.

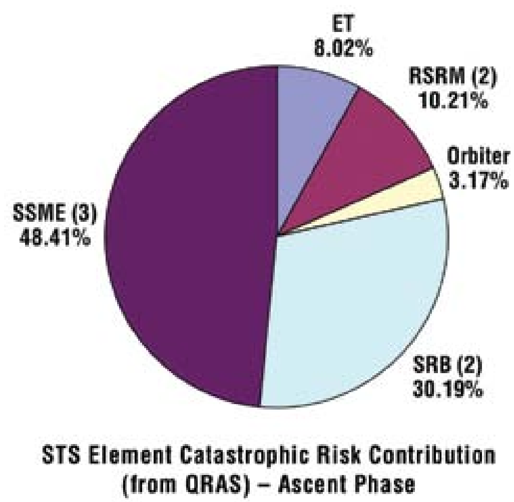


Figure 6-63. STS QRAS PRA Element Contributions

The analysis process involved a bottom-up review of the candidate engines, compared against SSME, which served as the baseline. SSME reliability is reflected by the QRAS data—a PRA conducted on all components of the engine down to part and failure mode level. The QRAS data applies to catastrophic failures and is a mix of very detailed probabilistic models and of top-level failure indicators (e.g., unsatisfactory condition reports). It is the most detailed failure assessment of any rocket engine in the world and reflects analysis and demonstrated data from approximately 20 years of Shuttle flights and approximately 30 years of propulsion system test and operations.

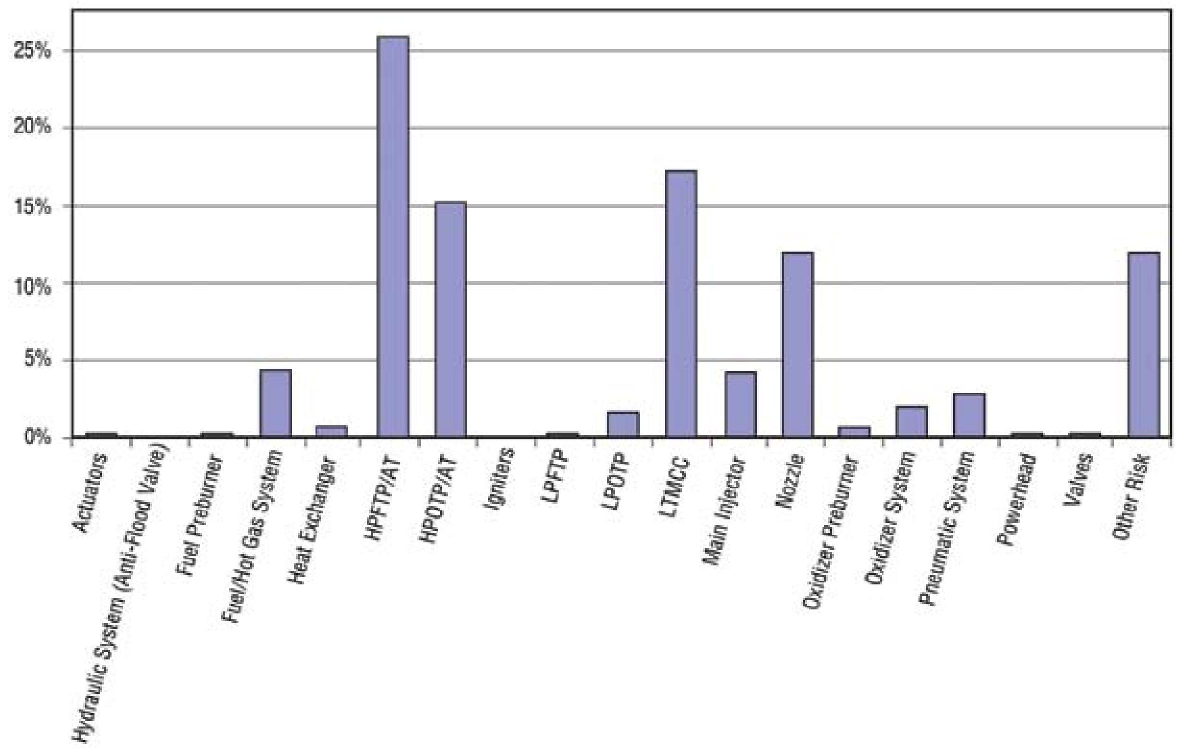


Figure 6-64. Component Contributions from SSME

Figure 6-65 describes the process the ESAS team used. The team reviewed all available data on the candidate engines, including configuration, background, Failure Modes and Effects Analyses (FMEAs), critical items lists, reliability predictions, and collected risk items (i.e., known problems, lack of data issues, human-rating issues) to support a bottom-up assessment. Again, the SSME PRA established the baseline for comparison. All candidate engines were compared, component-by-component, by component experts against this baseline. The SSME PRA failure probabilities were adjusted through expert opinion into metrics for catastrophic failure probability, benign failure probability, and CFF. The failure probabilities were modified with the guidance of expert opinion to reflect the design and environment of the candidate engine hardware. The probabilities were modified at the lowest possible level, where more detail was available and the judgment more direct. While quantified failure probability metrics were calculated, relative rankings of the candidate engines against SSME were the real result, given that the probabilities were modified by comparison and based on expert opinion. With an analysis of this type, it is easy to bias against the baseline, in this case, the SSME. It is easier to remove risk from the baseline in cases where no comparable component exists (e.g., no low-pressure pump) than it is to add in an accurate amount of risk for a new component.

The failure probabilities generated by the expert comparative assessment were considered to be mature estimates because the SSME reliability results from more than 1 million seconds of engine testing. New design or restart engines, such as the LR–85 or J–2S, cannot be considered mature. Historically, engines demonstrate significant reliability growth after first flight. Thus, the candidate engine failure probabilities were adjusted based on the SSME reliability growth experience. Using this approach, all the newer engines had their probabilities adjusted based on SSME growth. Also, all new engines were given credit for being ready for first flight; it was assumed they had achieved 100,000 sec of test time, roughly the SSME experience.

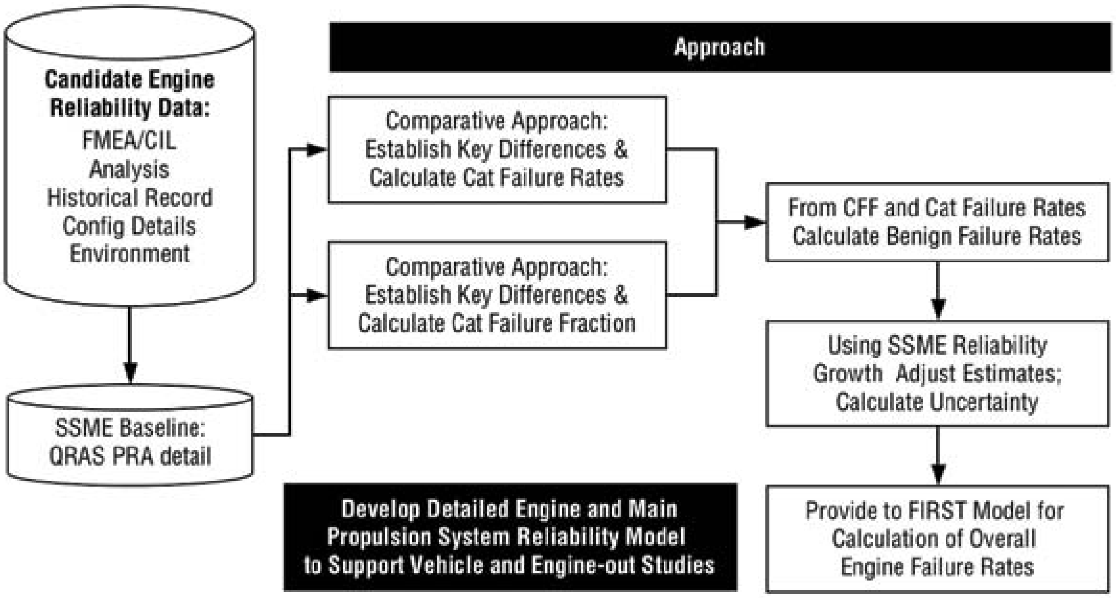


Figure 6-65. Reliability Assessment Methodology

These failure probabilities were used in the LV similarity model (FIRST). In this model, failure rates are derived and run in a time-based simulation. Algorithms accounting for thrust and size are also in the model.

Finally, the team also developed an engine and MPS reliability model and exercised it in support of vehicle and engine-out studies. MPS (valves, feedlines, etc.) reliability is a significant contributor to overall vehicle reliability and must be considered in any vehicle studies.

6.8.2.2 LV Liquid Rocket Engine Systems Methodology (FIRST)

For liquid rocket engines, three types of MPS failure modes are modeled in FIRST: (1) ICF, (2) DCF, and (3) BGN. The models used in FIRST for each of these is described in **Appendix 6.D**, **Safety and Reliability**, along with sections explaining the contributors to catastrophic vehicle failures. These sources include MPS failure through five root causes: (1) catastrophic failure of an engine, (2) loss of engine thrust, (3) loss of TVC in more than one-third of the total number of engines, (4) loss of stable propellant feed, and (5) engine altitude-start failure (for series-burn vehicles). The general vehicle level mean probability of failure due to MPS failure is computed by aggregating the element level failure probabilities together.

6.8.2.3 Comparative Analysis Results

Candidate engines for this phase of comparison included the RL–10, LR–60, LR–85, and LR–100 (all expander cycles), the J–2S (tap-off cycle), the RD–180 (single OX-rich preburner, staged combustion cycle), and the RS–68 (GG cycle). The baseline for comparison, the SSME, is a dual fuel-rich preburner, staged combustion cycle. **Figures 6-66** and **6-67** reflect the theoretical reliability benefits for the different cycles. For example, in **Figure 6-66**, LOX-rich combustion provides for more energy being released at the same temperature than in a fuel-rich preburner, thus there is a reliability benefit to the LOX-rich combustion if it can be accomplished at a lower temperature and still meet performance needs. This kind of information reflects overall cycle benefits independent of design specifics and provides for a sanity check on the relative rankings of the analysis.

	Dual Preburner Staged Combustion (Fuel-Rich)	Single Preburner Staged Combustion (LOX-Rich)	Gas Generator (Fuel-Rich)
Cycle Description	2 preburners – turbine drive gases stay in system	1 LOX-rich preburner; in LOX-rich, more energy released at same temp	1 gas generator; turbine gases dumped overboard
Complexity	Most complex – highest performance – capable of very high combustion temps	Complex – can lead to lower temps in engine	Less complex than staged combustion; lower performance
Environment	Highest temps and pressures; high pump speeds	Lower temps throughout engine due to LOX-rich combustion; LOX-rich issues	Lower temps and pressures than staged combustion
FMs, Effects & CFF	Many high critically FMs and catastrophic effects; highest ratio cat/benign	LOX-rich concerns with materials and combustion	Cycle more benign than staged combustion

Reliability Improvement Due to Cycle

Figure 6-66. Benefits of Boost Engine Cycles

	Dual Preburner Staged Combustion (Fuel-Rich)	Tap-Off Cycle	Expander Cycle
Cycle Description	2 preburners - turbine drive gases stay in system	No GG or preburner, turbine drive gases tapped off of chamber	No GG or preburner, turbine drive gases collected from regen chamber and nozzle
Complexity	Most complex - highest performance - capable of very high combustion temps; concerns with air-start	Higher performing than expander - higher turbine drive temps	Simplest - lowest performance; propellant not wasted; tolerant to throttling; chamber pressure restricted
Environment	Highest temps and pressures; high pump speeds	Similar to expander but high temps tapped off chamber	Lower temps and pressures; lower concern with combustion stability
FMs, Effects & CFF	Many high critically FMs and catastrophic effects; highest ratio cat/benign	Similar to expander but high temps in hot gas system for turbine drive	Heat absorption is through benign process - few critical FMS and effects

Reliability Improvement Due to Cycle

Figure 6-67. Benefits of Upper Stage Engine Cycles

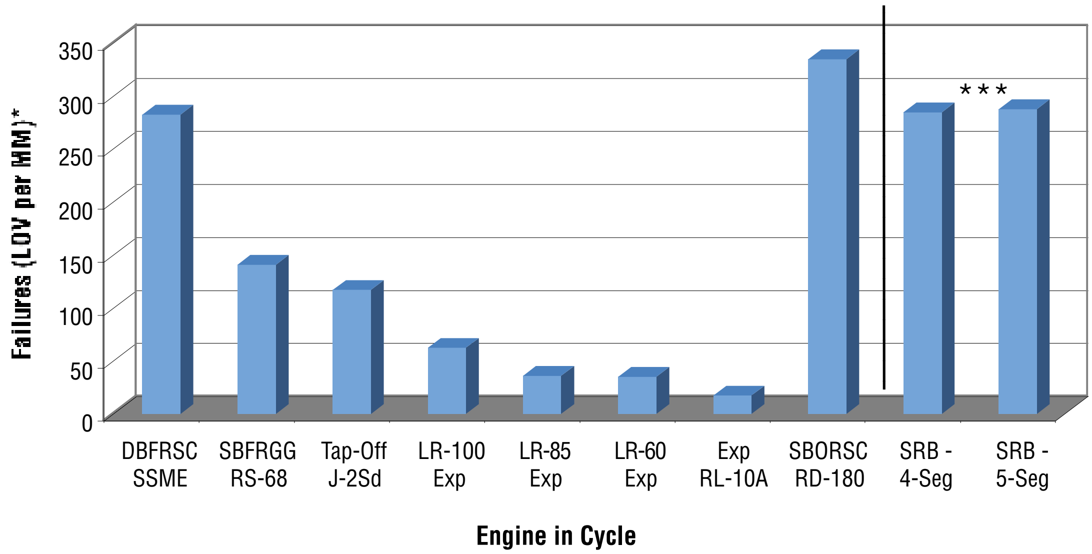
Results of the comparative analysis are provided in **Appendix 6G**, **Candidate Vehicle Subsystems**, for the different engine candidates. Again, all assessments were made against the SSME. An example of supporting QRAS data for the comparison is presented in **Table 6-27**. The comparative results include the SSME risk by component and the engine rationale for changing the risk as derived by expert opinion. An example of the supporting data for this process is provided in **Table 6-27**, which reflects the level of detail available for the comparison from QRAS. Piece part failure mode and mechanism data were available to identify true risk concerns in each component. For example, turbine end vane failure due to thermal concerns causing Low Cycle Fatigue (LCF) and fracture contribute approximately 10 percent of the risk in the current High-Pressure Fuel Turbopump (HPFTP). Such information was used by the experts in comparing differences across engines. Data at this lowest level made for better comparisons and supported this bottom-up comparative analysis approach.

The percentages of risk reduction or increase were rolled up into an overall catastrophic reliability probability. The CFF was also generated from the expert comparative approach and presented in **Table 6-27**.

Table 6-29. SSME Failure Modes and Causes

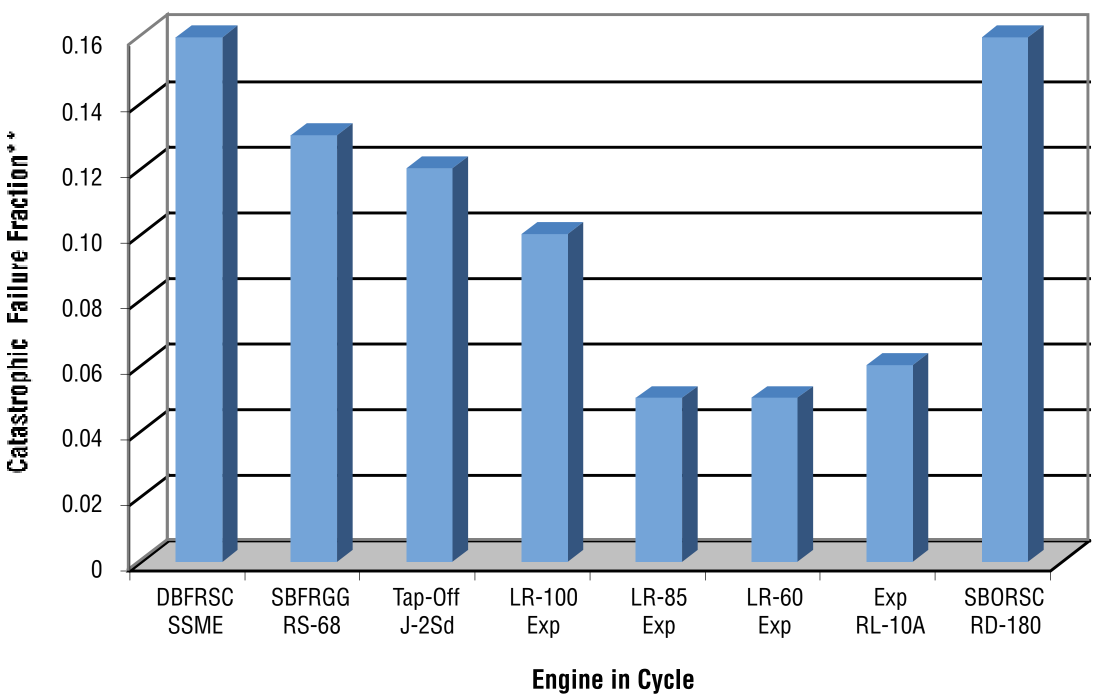
Component / Failure Modes / Causes	Component Contribution %	Failure Modes % Contribution	Description
HPFTP/Alternate Turbopump (AT)	24.7725%		Could cause a turbine disk assembly failure, generating over-speed, burst, and case penetration leading to fire and explosion.
Turbine Disk Assembly Failure		14.39%	
Design (LCF/fracture due to vibration, thermal growth, material/manufacturing defect, overspeed, rub, loss of cooling)			
Second-stage Turbine Vane Failure		10.46%	Thermal gradients/stresses may induce cracks in the second stage vanes. Dynamic loading from preburner gas flow can aggravate cracks to the point of vane rupture. Vane rupture will release mass into the flow path and result in impacts with turbine blades. The loss of blades due to impact from vane material does not imply Loss of Vehicle (LOV)/LOC.
Design (LCF/fracture due to thermal gradient/stress)			
Second-stage Turbine Blade Failure		10.10%	Cracking of the blade due to material defects in conjunction with loading caused by thermal transients or other loads can lead to liberation of blade material. The liberated material could impact other blades, causing those blades to fracture, resulting in high temperatures due to a LOX-rich environment. This could lead to LOV; however, the HPFTP/AT has demonstrated the capability of surviving such a scenario with a benign shutdown.
Design (LCF/fracture due to material defects/loading)			
Interstage Seal (1–2 Damper) Failure		9.21%	The interstage seals function to control leakage of propellants between pump stages and to dampen vibrations in the rotating machines. The interstage seals function to control leakage of propellants between pump stages and to dampen vibrations in the rotating machinery. Clearance anomalies can result in reduced pump speed and/or high vibrations.
Design (LCF/fracture due to clearance anomaly)			
First-stage Turbine Vane Failure		8.91%	Thermal gradients/stresses may induce cracks in the first-stage vanes. Dynamic loading from preburner gas flow can aggravate cracks to the point of vane rupture. Vane rupture will release mass into the flow path and result in impacts with turbine blades. The loss of blades due to impact from vane material does not imply LOV/LOC.
Design (LCF/fracture due to thermal gradient/stress)			
First-stage Turbine Blade Failure		8.61%	Cracking of the blade due to material defects in conjunction with loading caused by thermal transients or other loads can lead to liberation of blade material. The liberated material could impact other blades, causing those blades to fracture, resulting in high temperatures due to a LOX-rich environment. This could lead to loss of vehicle; however, the HPFTP/AT has demonstrated the capability of surviving such a scenario with a benign shutdown.
Design (LCF/fracture due to material defects and loading/thermal trainsients)			
First Blade Outer Gas Seal (BOGS) Hook Failure		5.07%	Cracking of the first BOGS hook caused by thermal transients or other loads can lead to the BOGS dropping into the blade path, causing blade failure. The impact to the blades can result in high temperatures due to a LOX-rich environment. This could lead to LOV; however, the HPFTP/AT has demonstrated the capability of surviving such a scenario with a benign shutdown.
Design (LCF/fracture due to thermal transient/vibratory stress)			
Pump Discharge Housing Failure		4.59%	The primary concern of a pump discharge housing structural failure is loss of hydrogen flow and an external fire due to hydrogen leakage. The HPFTP/AT design eliminates many of the concerns of housing failures by eliminating many of the welds of the previous HPFTP that may precipitate potential structural failures.
Design (LCF/fracture due to stresses)			

From the catastrophic failure probability and the CFF, the benign reliability probability was derived. These catastrophic failure probabilities (by engine) and the CFF results are summarized in **Figures 6-68** and **6-69**. All comparisons were made using information on engines as currently available.



* From FIRST Model with MSFC input; component lack of data issues with RD-180
** SSME-based – Assumes SSME-like red-line system.
*** SRB assessment from FIRST model

Figure 6-68. Engine Catastrophic MTBFs



* From FIRST Model with MSFC input; component lack of data issues with RD-180.
** SSME-based - Assumes SSME-like redline system
***SRB assessment from FIRST model.

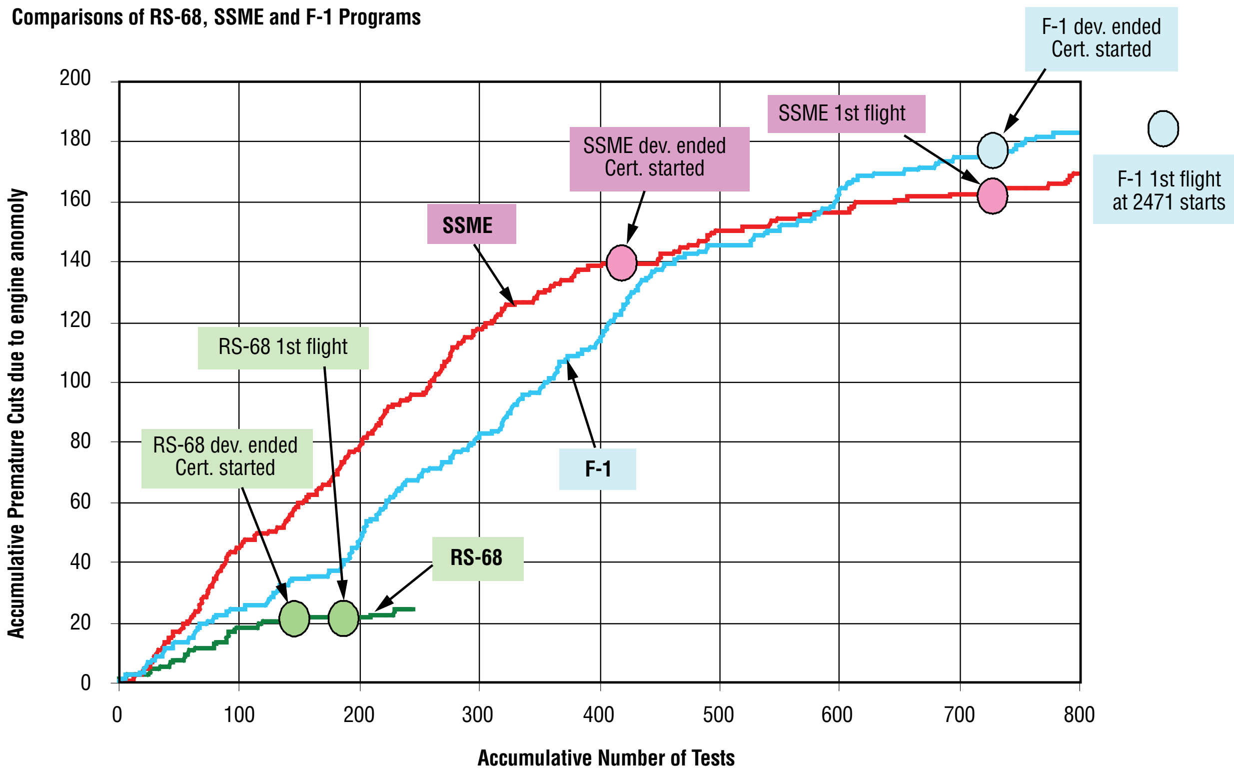
Figure 6-69. Engine Catastrophic Failure Fractions

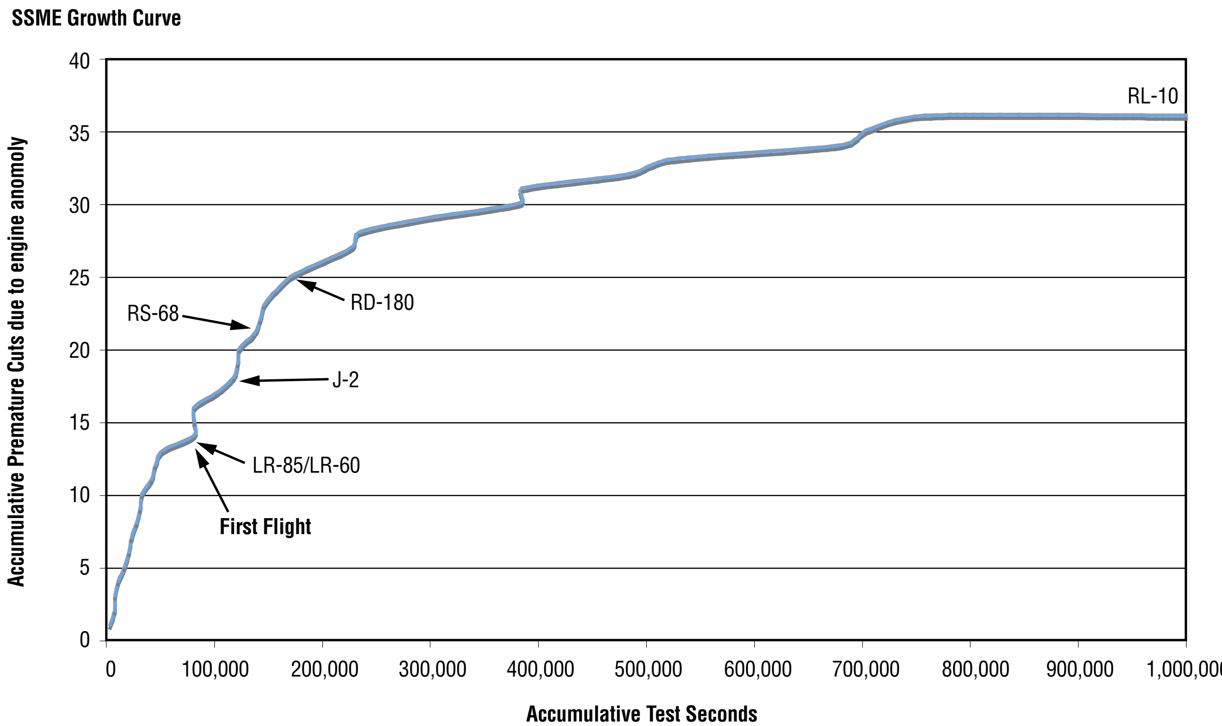
The results in **Figure 6-68** reflect FIRST model output with MSFC-generated engine input. There was a concern about a lack of data (i.e., no FMEA, lack of certain component data) on the RD–180. In such a case, the risk on the comparable SSME component went unchanged. Also, the SSME PRA data included a redline system that supported engine cutoff under certain conditions. This means that the other comparative engine reliability data implicitly includes a similar redline system. In **Figure 6-68** the SRB reliability is included only for comparison purposes. This also reflects results from the FIRST model. In actuality, the SRB includes much more than an engine. The SRB has the MPS equivalent of a liquid propulsion system, as well as structure, TVC, and recovery systems.

6.8.2.4 Reliability Growth Modeling

After considering the similarity analyses performed by comparing a fully developed SSME to other engines, there was still the problem of dealing with the relative maturity of the engines. This problem was overcome by applying a reliability growth model to the engine values. Because everything was compared to a fully mature SSME, the resulting values for the other engines based on the similarity analysis were also "mature" values. However, because most of these engines were far from being mature, the "true" value for these immature engines was found by "backing out" the mature values using the SSME reliability growth curve. A detailed discussion of the reliability growth modeling and associated uncertainty is contained in **Appendix 6D**, **Safety and Reliability**.

Baseline comparative results were presented in **Figures 6-68** and **6-69**. The top image within **Figure 6-70** presents the associated reliability growth curves of SSME, F–1, and RS–68. The adjustments to the baseline numbers associated with the reliability growth of the SSME are presented in **Figure 6-71**. The SSME demonstrates a typical growth curve—steep at the beginning and flatter as the curve goes over the "knee." This reflects fewer failures during the same amount of testing over time and demonstrates the engine reliability improvement as testing reduces failures and test/fail/fix cycles correct problems. The bottom image of **Figure 6-70** reflects the starting point for the reliability estimates of the candidate engines. All engines were given credit for having roughly 100,000 sec of test time at first flight. Additional test time was added as appropriate. Various versions of the RL–10 have more than 2 million sec of test time, and, thus, this engine is considered very mature.





**All new/restart engines at first flight equivalent of 100k + test time.*

Figure 6-70. Engine Reliability Growth Curves and Cumulative Failures by Test Seconds and Number of Tests

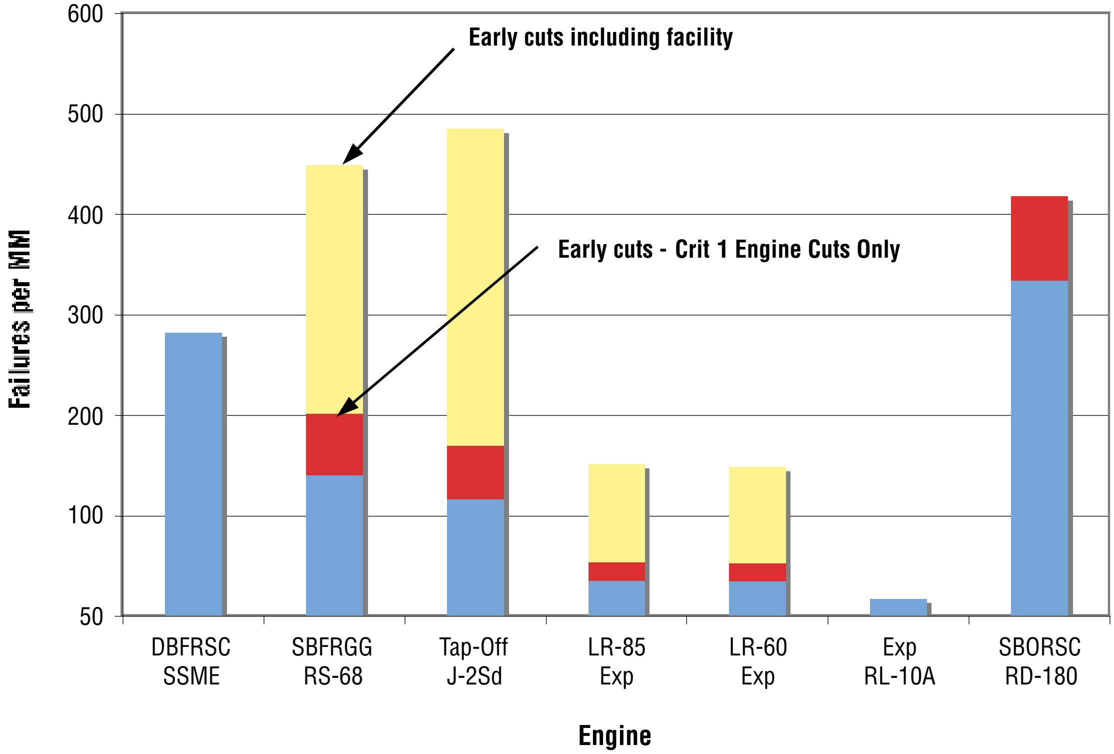


Figure 6-71. Adjustments to Baseline Estimates

The analysis provided in **Figure 6-71**, as discussed previously, utilized the SSME hot fire database. Two runs were completed and are presented in **Figure 6-71**. The red bar represents early engine-related cuts that reflect Criticality 1 failures only. The overall red and yellow bar reflects results from both early engine and facility-related cuts. Though the latter estimates were not analyzed any further, they could provide insight into engine and MPS failures or overall operational system reliabilities.

Finally, in terms of reliability growth, the RD–180 was assumed to need 50 percent regrowth with limited insight to become a mature engine. This is necessary given the "Americanization" that is required to support the insight needed for comparison to the SSME.

To this point, the reliability assessment estimates generated have been point estimates. Point estimates with a measurement of uncertainty are better assessment metrics. The uncertainty in the candidate systems should be assessed. The uncertainty will be higher in newer systems. Since the comparative approach used with point estimates was based on a comparison with SSME, a similar approach was used with the uncertainty estimates. The estimates are based on SSME values with adjustments to the amount of test history.

Figure 6-72 presents the logic behind the uncertainty estimates. Candidate engines with limited test time will have higher levels of uncertainty. Through selection of mature components, hardware development, and testing, the uncertainty was reduced to levels more closely approximating the SSME baseline, which has the most design and test data. The uncertainty can be represented with the 95th and 5th percentile, mean and median. Thus, appropriately, the reliability point estimates and uncertainties both reflect the influence of reliability growth.

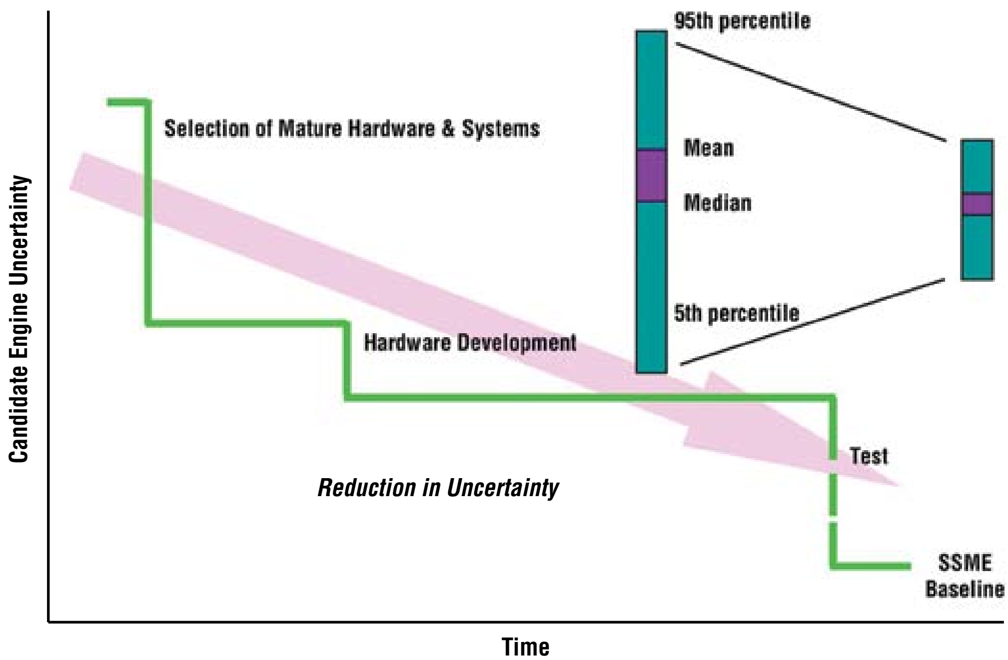


Figure 6-72. Point Estimates And Uncertainty

Table 6-28 provides the resulting measures of uncertainty, including percentiles and the EF, associated with the candidate engines. This approach was again based on reliability growth, where the extent of engine testing determines the size of the EF. Uncertainty is lowest for mature engines (e.g., SSME at 2.6; consistent with QRAS estimates) and highest for new engines (e.g., LR–60 at 9.22). As stated earlier, all new engines were given 100,000 sec of test time to be ready for first flight.

Table 6-28. Uncertainty Measures

Engine	Failures/ MM	MTBF	Test Time	Eq Missions	Equivalent Failures	EF	Sigma	Mu	5th	Median	Mean	95th
RL–10	17.30	57799	1000225	1942	17.00	2.60	0.58	–11.13	5.62E–06	1.46E–05	1.73E–05	3.80E–05
LR–60	34.84	28702	100000	194	1.70	9.22	1.35	–11.18	1.52E–06	1.40E–05	3.48E–05	1.29E–04
LR–85	35.61	28082	100000	194	1.70	9.22	1.35	–11.15	1.55E–06	1.43E–05	3.56E–05	1.32E–04
LR–100	62.29	16053	100000	194	1.70	9.22	1.35	–10.60	2.72E–06	2.50E–05	6.23E–05	2.31E–04
RS–68	140.39	7123	134131	260	2.28	6.95	1.18	–9.57	1.01E–05	7.01E–05	1.40E–04	4.87E–04
J–2S	116.86	8557	126000	245	2.14	7.01	1.18	–9.76	8.26E–06	5.80E–05	1.17E–04	4.07E–04
RD–180	417.00	2398	140000	388	2.38	5.12	0.99	–8.27	4.98E–05	2.55E–04	4.17E–04	1.30E–03
SSME	282.22	3543	1000225	1942	17.00	2.60	0.58	–8.34	9.17E–05	2.38E–04	2.82E–04	6.20E–04

6.8.3 MPS Models

The ESAS team applied two different MPS models. The simplified MPS model used for LV reliability LOM and LOC predictions in this study is part of an LV analysis tool with models for all other subsystems. Although the incorporated model did not support MPS trade studies like the general MSFC MPS model described in **Appendix 6D**, **Safety and Reliability**, it did allow the rapid assessment of many complex LVs against a uniform standard using strictly standardized methodologies to support objective reliability comparisons at the vehicle level. A general liquid propulsion reliability model was also developed and applied to both LV and in-space propulsion stages. The model assumes a generic MPS architecture and relies exclusively on the Space Shuttle PRA for quantifying the failure events captured in the model. Each model's purpose and methodology is described in **Appendix 6D**, **Safety and Reliability**.

The MPS reliability model was developed to support the LV selection trades. Selected stage configurations with burn times were provided for analysis. **Figure 6-73** shows the results of the analysis of these configurations. **Figure 6-73a** shows the results for LV stages using engine failure parameters corresponding to a first build engine reliability. **Figure 6-73b** shows the results for LV stages using engine failure parameters corresponding to mature engine reliability. Note that the SSME and the RL–10 are both operational and mature; hence, the first build and mature failure probabilities are the same for these engines.

The results are broken out to indicate: (1) only the effects of the engine cluster, (2) the combination of the engine cluster and the MPS, and (3) the combination of the engine cluster, the MPS, and failure to start. Note that for boost stages, hold-down at launch is assumed. Thus, start failures are not included for boost stages. Results indicate that MPS failures can contribute from 15 to 50 percent of the overall stage unreliability depending on engine reliability and the number of engines. Start failures can contribute from 10 to 40 percent of the overall stage reliability, again depending on engine reliability and the number of engines. For upper stages, results indicate that engine-out is a preferred capability. The bulk of the added MPS modes are benign, leading to an engine shutdown. The added redundancy appears to easily absorb these failures.

Results strongly indicate that, if engine-out is feasible (configuration, packaging, performance, etc.), then engine-out capabilities provide a significant reliability benefit. MPS and avionics unreliability may overwhelm the gain in reliability from using a pressure-fed engine. Stage development should include considerable efforts to improve MPS and avionics reliability.

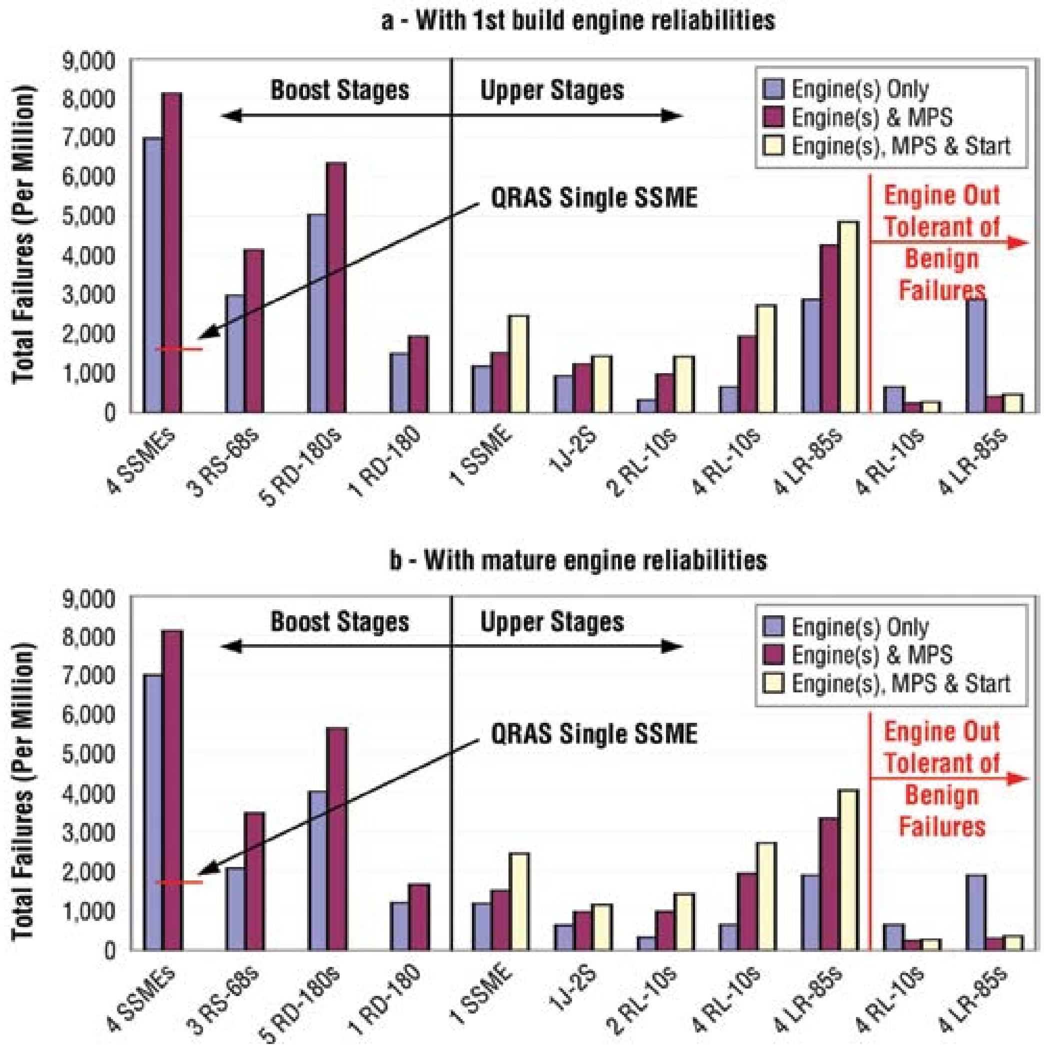


Figure 6-73. Launch Vehicle Stage Reliability Results

6.8.4 External Tank

The risk of failure for an ET, such as that used on the Space Shuttle to provide propellant to the SSMEs, is included in FIRST, while internal propellant tank risk is ground-ruled based on engineering judgment.

The probability of failure of an ET is based on the QRAS 2000 Space Shuttle estimates and is 1 in 6,442 MFBFs. This risk has been applied to Shuttle-derived side-mount vehicles only. This is a low-level contributor (on the order of 5 to 10 percent) of total vehicle risk.

6.8.5 Auxiliary Power Units

APUs provide power for valve actuation, TVC, general power, etc. They are used on core boosters, strap-on boosters, and upper stages. The risk estimates used in FIRST for the APU are based on top-level analysis of the state-of-the-art Space Shuttle system proportioned by the ascent time and the system fuel type. Hydrazine APUs were used on vehicles containing the SSME, RS–68, J–2, RL–10, LR–60, and LR–85. The RD–180 has a self-contained hydraulic system; therefore, no additional APU risk is required for this engine. Proton Exchange Membrane (PEM) fuel cells were chosen as the APU type for the CM.

It is assumed that today's Space Shuttle APUs are considered state-of-the-art, and, therefore, Shuttle APU risk estimates are used for hydrazine-powered APUs on vehicles in this study. However, the integration of hydrazine APUs into a new vehicle does add some level of uncertainty, thus the EF for this type of APU was increased by 50 percent (from the QRAS 2000 estimates) to a factor of 7.5. The results from the 1997 QRAS Study for the Space Shuttle APU yield a catastrophic failure probability on ascent of 3.95E–5 (for a 515-sec burn time).

Non-hydraulic APUs are advanced power systems powered by PEM fuel cells. They are an advanced technology designed with high reliability and negligible risk. Using engineering judgment, the system risk is set to 1 in 1 million and is given an EF of 10 in FIRST.

The risk contribution from the APU subsystem is assessed in FIRST as follows:

Step 1: The APU risk for the Space Shuttle on ascent is divided by the Space Shuttles' ascent time (515 sec on ascent) yielding a per-second ascent risk estimate of 7.67E–8.

Step 2: For APU ascent risk, the element ascent time (the amount of time from liftoff when the APU is powered on until the element separates from the vehicle) is multiplied by the per-second risk estimate from above. The number of engines does not affect the APU failure probability. Further, APU risk exposure time begins at liftoff, so the APU risk associated with an upper stage would also take into account the exposure time between liftoff and upper stage engine start.

6.8.6 Thermal Control System

The TCS is present on elements including the core boosters, strap-on boosters, upper stage, and CM. The risk estimate is adopted from a report that studied LOM failures of the active TCS, which considered only the ascent phase. The estimated failure probability for the TCS is 1.08E–9 per element, with an associated EF of 10. When multiple elements exist, the failure probability is increased by the number of elements present.

6.8.7 Separation Systems

The model (FIRST) provides two types of element separation mechanisms, in-line and strap-on, to accommodate a number of LV configurations. The following sections detail the separation options, their availabilities, and their contributions to vehicle risk.

The in-line separation model provides risk estimates for all LV elements configured in series. It is available as a separation option to all core boosters. Three different methods of in-line element separation are available: fire in the hole, spring, and retro. However, only the retro was used for this study. Retro is an in-line stage separation mechanism that uses retro rockets on the lower stage to separate the two stages after explosive bolts have been blown. Retro separation reliability estimates are patterned after the Titan second-stage separation using QRAS 2000 data from Shuttle SRB separation components.

Probabilities of failure for all of the in-line separation models were reached by decomposing QRAS 2000 results for Shuttle SRB separation into two main components: explosive separation bolts and BSM clusters. Each Shuttle SRB uses one forward separation bolt and three aft separation bolts, and one BSM cluster is used at both the forward and aft separation points to push the SRB away from the ascending vehicle after separation. Using the reliability estimates for forward and aft bolts and BSMs, an average failure rate was determined for a single separation bolt and a single BSM. The per-unit failure rates are shown in **Table 6-29**.

Table 6-29. Per-Unit Failure Rates

Component	5th	50th	Mean	95th
1 Bolt	4.097E–06	1.129E–05	1.456E–05	3.558E–05
1 BSM Cluster	5.621E–06	1.330E–05	1.574E–05	3.332E–05

Using these component reliabilities, a model separation system was constructed for each of the above mentioned in-line separation methods by aggregating the appropriate number of bolts and thrusting elements, as shown in **Table 6-30**.

Table 6-30. In-Line Separation Risks

Method	Number of Bolts	Number of BSM Clusters	P(f)	Error Factor
Fire in the Hole	4	–	1 in 17,165	3.15
Retro	4	1	1 in 13,513	3.00
Spring	4	–	1 in 17,165	3.15

The strap-on separation model provides risk estimates for LV elements in a side-mounted, or parallel, configuration. Strap-on booster separation is the designated separation mechanism for side-mounted, multi-segment SRBs. It consists of three aft attachment points and one fore attachment point with exploding bolts separating the booster from a core stage. BSM solid rocket clusters at the fore and aft attachments provide the separation force.

The probability of separation failure for a strap-on booster is based on QRAS 2000 four-segment Shuttle SRB results. The QRAS SRB separation results are used as-is for four-segment SRBs, and have also been scaled to accommodate five-segment SRBs. For the addition of a center segment to the four-segment baseline configuration, booster separation bolt risk is increased by 15 percent, and BSM risk is increased by 25 percent. The resultant probabilities of separation failure are provided in **Table 6-31**.

Table 6-31. Strap-On Separation Risks

Number of Segments	P(f)	Error Factor
4	1 in 11,173	2.15
5	1 in 9,425	2.15

6.8.8 Payload Shroud Reliability Analysis

At the beginning of the study, the Payload Shroud Reliability model used a 1 in 891 probability of failure per launch, based on historical data displayed in **Table 6-32**. This model included the possibility of any type of payload shroud failure (structural, failure to separate, and inadvertent separation).

Table 6-32. Historic LV Data Used for Payload Shroud Failure Estimate

LV	Total Launch Attempts (1986–Dec. 31, 2003)	Launch Attempts Not Including Vehicle Failures Before PLF Separation Could Begin	PLF Separation Failures	Successful Launches Since Failure Occurred
Ariane	135	133	0	
Athena	7	6	1	1
Atlas	94	93	0	
Delta	116	113	0	
H-Series	20	18	0	
Long March	59	56	0	
Pegasus	34	33	0	
Proton	165	165	0	
Soyuz/Molniya	454	453	**2***	80
Taurus	6	6	0	
Titan	60	57	0	
Tsiklon/Dnepr	115	114	0	
		1,247	**3(1.4*)**	
	Total PLF Attempts	1,247		
	Total PLF Failures	1.4		
		0.001122694		
		1 in 891		

**Failures due to manufacturing process. These failures are discounted assuming an 80% fix factor, hence each failure counts as only 0.2 of a failure.*

The ESAS team directed an improved shroud risk estimate using a physics-based process for estimating shroud risk rather than historical statistical estimates. After research and investigation of shroud design technology and methods, it was determined that historical data remains the most accurate method for predicting shroud structural failures because:

- Shroud loads, material, required factors of safety, payload size and weight, etc., are inputs that lead to a shroud's physical characteristics.
- In conceptual or preliminary stages of vehicle designs:
  - Shrouds (and other structures) are designed by analysis, and
  - Designs are not evaluated for reliability at given loads or environments, rather loads and environments determine shroud design.
- Each shroud is tailored to a particular payload and trajectory, and, if the shroud material, payload, or trajectory (aero loading) changes, the shroud design changes, (i.e., one shroud design would be just as reliable as another shroud design that was developed using the same design tool and process).
- For preliminary design, worst-case loads and minimum material strengths are assumed.
- Slight changes in material strength may significantly alter predicted structural reliability.
- Small variations in aerodynamic loads may significantly alter a shroud's predicted reliability.

The ESAS team confirmed that attempting to relate predicted shroud reliability to design factors of safety or shroud physical characteristics at the current level of design detail would be very inaccurate. Based on this conclusion, the existing shroud model was refined by distinguishing the historical structural failures from those caused by inadvertent separation or failure to separate. Structural failure probability based on historical data could be combined with a PRA of the other two failure modes for a typical shroud design. A shroud failure fault tree is provided in **Figure 6-74**.

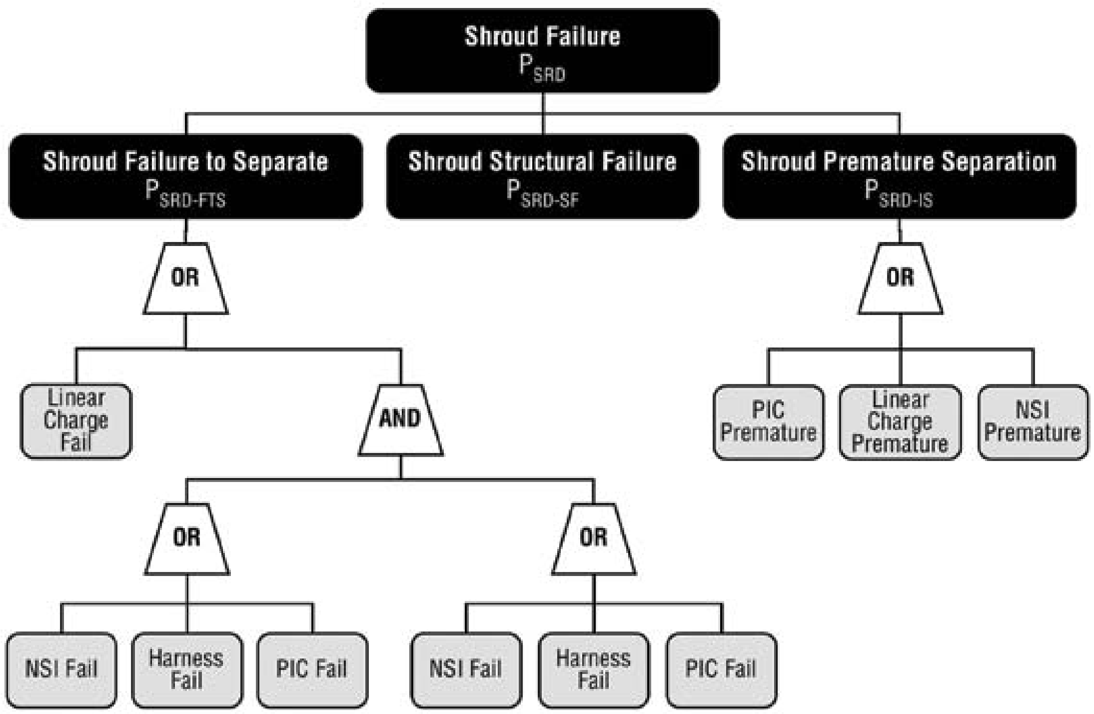


Figure 6-74. Payload Shroud Fault Tree

The revised historical model is summarized in **Table 6-33** below.

Table 6-33. Revised Payload Shroud Historical Data Set

International Launches 1986 – 2003	
Launches	1,247
Structural failures	Soyuz/Molniya - 2 x 0.2 (80% fix factor)
Failure Rate	0.4/1,247 ~ 1/3,100

The failure to separate model is provided in **Table 6-34**. This model provided a more realistic estimate for payload shroud failure (1 in 3,080) and was used to estimate LOM for all the payload LVs in the ESAS.

Baseline Data
Calculations
Final Result

Table 6-34. Updated Payload Shroud Reliability Estimate Model

Baseline Component Failure Probabilities	Failure to Separate			Premature Separation[3]		
	Mean		EF[6]	Mean		EF[6]
Pyrotechnic Initiator Controller (PIC)	1 in 18,707[1]	5.35E–05	1.015	1 in 1,000,000	1.00E–06	3.930
NASA Standard Initiators (NSIs)	1 in 375, 375[2]	2.66E–06	9.100	1 in 1,000,000	1.00E–06	3.930
Linear Charge	1 in 1,000,000[3]	1.00E–06	3.930	1 in 2,000,000	5.00E–07	3.930
Wire Harness/System	1 in 3,105[4]	3.22E-04	2.416	N/A	N/A	N/A

Shroud Structural Failure		
1 in 3,118[5]	**3.21E–04**	6.30
	9.9968E–01	

Calculations	Failure to Separate		Premature Separation	
	Mean		Mean	
Success Probability per Chain	1–(1 in 2,644)	9.9960E–01	1 in 400,000	**9.9900E–01**
Failure Probability for two Parallel Chains	1 in 6,991,428	1.4300E–07	N/A	N/A
Success Probability w/ two Chains and Linear Charges	1– (1 in 874,866)	**9.9900E–01**	N/A	N/A
Probability of Failure	**1 in 874,866**	**1.1430E–06**	1 in 400,000	**2.4990E–06**

Primary Failure Modes Calculated Probabilities	Failure Probability	
	Mean	
Failure to Separate	1 in 874,866	**1.1430E–06**
Premature Separation	1 in 400,000	**2.4999E–06**
Structural Failure	1 in 3,118	**3.2077E–04**

Shroud Probability of Failure	Mean		EF
	1 in 3,080	**3.2464E–04**	**6.05***

1. *PIC reliability based on demonstrated reliability—analysis is found in the Space Shuttle Analysis Report (SSMA–02–006 November 20, 2002) titled: Pyrotechnic Initiator Controller (PIC) Reliability and Maintenance Analysis.*
2. *NSI failure rate is based on 125,000+ firing without a failure. This data is based on lot acceptance firings through the years of Gemini, Apollo, (ASTP), and Shuttle. The 1 in 375,375 Mean is CARPEX-generated based on 0 failures in 125,000 trials (1/3 rule).*
3. *Linear charge failure rate and all premature separation failures assumed very unlikely due to system design. Failure rates for these are engineering judgment.*
4. *Wire harness reliability based on three partial failures of redundant wiring harness in the STS history (9,314 commanded firings) per SSMA–02–006, researched by Jeremy Verostico (Pyros/PIC JSC Safety and Mission Assurance (S&MA) Science Applications International Corporation (SAIC)).*
5. *Structural failure based on demonstrated reliability—all international launches 1986–2006. (See Para. 5 above.)*
6. *Error factors were calculated using CARPEX.*

6.8.9 Additional RSRB Safety Considerations

6.8.9.1 Introduction

This section addresses, in more detail, the inherent characteristics that drive the predicted reliability and survivability (as described previously in **Section 6.8.1**, **Reusable Solid Rocket Booster**, of Shuttle-derived RSRBs as applied to the in-line CLV (LV 13.1) configuration. Because the CLV is currently in the conceptual phase of development, this assessment is based on evidence from a variety of sources, including the flight history of similar systems as well as relevant analytical modeling activities. Although there is uncertainty in the specific values of reliability and survivability, the results of this assessment indicate that the Shuttle-derived RSRB is a reliable concept whose failures are survivable, particularly in comparison to the liquid-core EELV alternative.

The value of the Shuttle-derived RSRB can be summarized in the following factors:

- Simple, Inherently Safe Design: The human-rated RSRB (post-51–L) first stage has been matured over 88 Shuttle flights (equivalent to 176 single RSRM flights);
- Historically Low Rates of Flight Failure: Only the Challenger event marred a perfect record of 226 SRB flights. This results in a 0.996 launch success rate (combined 50 flights of the SRM and 176 flights of the RSRM);
- Design Robustness: Test results and physics-based simulations show the SRB LV design is robust and resistant to crew adverse catastrophic failure, even for the most severe failure modes;
- Non-Catastrophic Failure Mode Propensity: SRB history and SRB design features suggest gradual failures that are less likely to threaten the crew;
- Process Control and Inspection: The proposed design offers benefits of propulsion suppliers with mature process control and inspection systems to minimize in-factory and post-manufacturing human error, a significant contributor to the current launchers' risk; and
- Failure Precursor Identification and Correction: The design capitalizes on the significant failure precursor identification and elimination from recovery and post-flight inspection of the recovered SRBs.

6.8.9.2 RSRB Description

The Shuttle RSRB is a four-segment, steel-case propulsion system that provides a peak sea-level thrust of 2,900,000 lbf and burns for 123 sec. At the end of burn, which is at 150,000 ft and a velocity of 4,500 ft/sec, it separates from the Shuttle and splashes down in the Atlantic, some 122 nmi downrange from the launch site. From there, it is towed back by recovery ships for refurbishment and it may be reused for up to 20 launches. The RSRB weighs 1,255,000 lb, of which 1,106,000 lb is solid propellant. It has an igniter in the forward segment and a nozzle at the aft segment. The igniter ignites the propellant inner surface, which burns at an engineered rate into the propellant volume. The basic elements of the RSRB are shown in **Figure 6-75**.

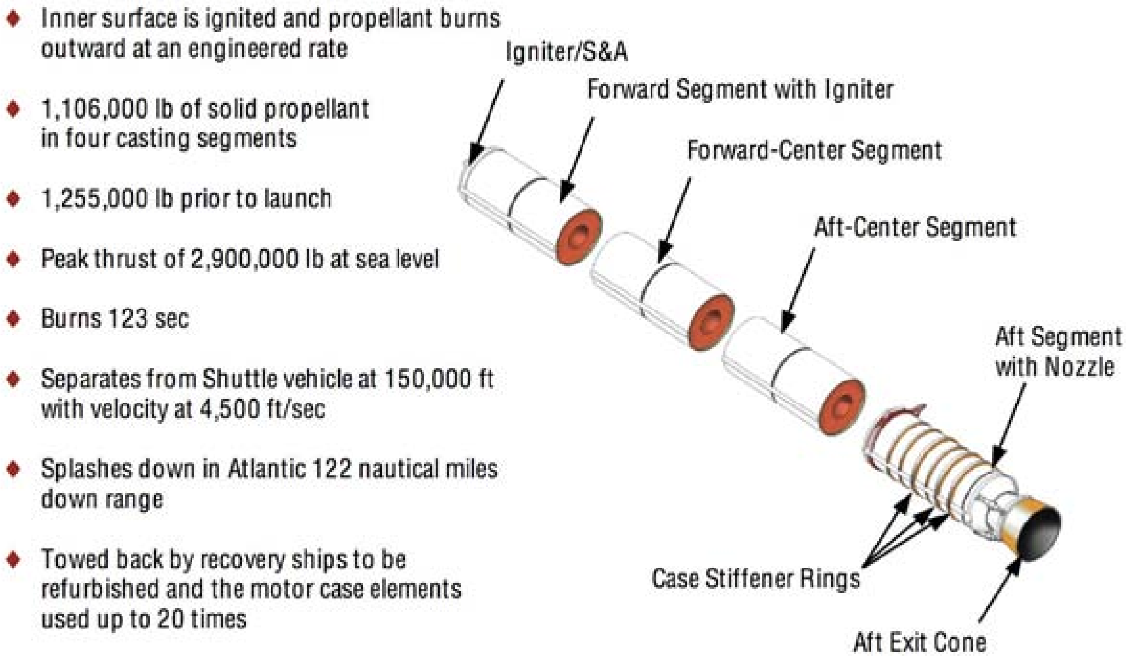


Figure 6-75. Basic Elements of a Shuttle RSRB

Table 6-35 presents the composition of the RSRB propellant. It is important to note that the propellant is not an explosive, and extensive testing has demonstrated the inability of the propellant to detonate, even under extreme accident conditions such as those produced by motor fallback. It is classified as Hazard Division (HD) 1.3 by the DoD and Department of Transportation (DoT), which, by definition, identifies the major hazard as mass fire.

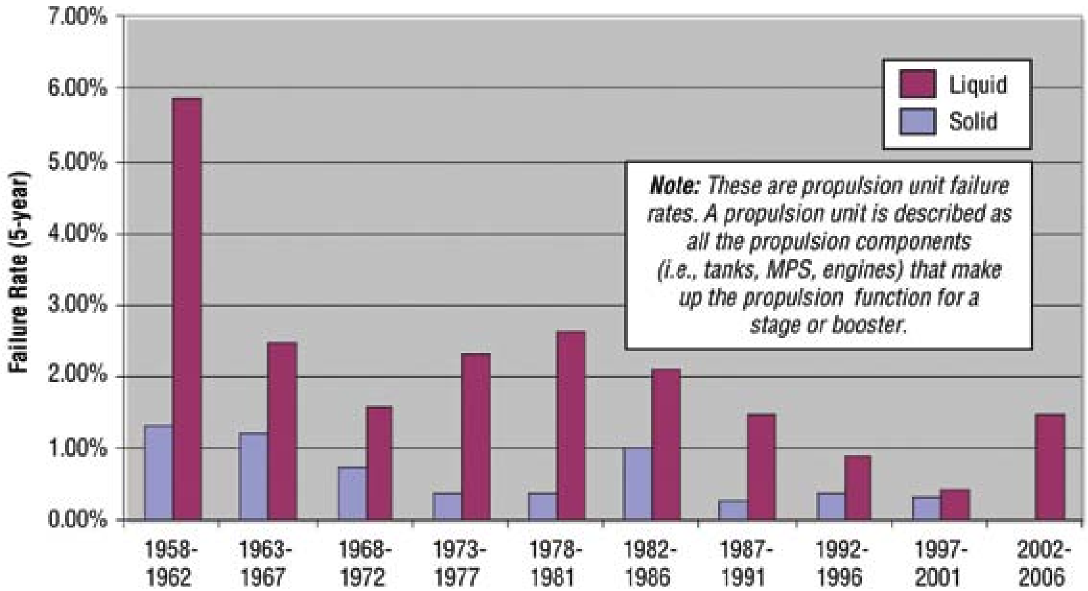
Table 6-35. Composition of RSRB Propellant

Raw Material	Function	Main Grain Percentage
Ammonium Perchlorate	Oxidizer	69.70
Aluminum Powder	Fuel	16.00
PBAN Polymer	Binder	12.10
Epoxy Resin	Curing Agent	1.90
Iron Oxide	Burn Rate Catalyst	0.30

6.8.9.3 Solid Propulsion History

Assessment of Shuttle-derived RSRB reliability (and survivability) begins with a general review of propulsion history to develop baseline values to which it can be compared. Although the historical record is a valuable information set, caution must be used when extrapolating from past failures of systems that reflect varying degrees of similarity to the RSRB.

The most elementary distinction that can be made between propulsion systems is the gross propellant type (i.e., solid versus liquid propellant). As shown in **Figure 6-76**, solid propulsion systems have historically had lower rates of failure than liquid systems. From 1958 to the present, there have been 39 failures out of 2,133 attempts for liquid propulsion systems, whereas there have been 20 failures out of 3,535 attempts for solid propulsion systems. This corresponds to historical failure probabilities of 0.57 percent for solids and 1.8 percent for liquids, a factor of 3.2.



Overall Propulsion Unit Failure Rates from 1963-2004		Attempts	Failures
Liquid Propulsion	**1.83%**	**2,133**	**39**
Solid Propulsion	**0.57%**	**3,535**	**20**

RSRM Influences on Reliability & Crew Survivability for a Human Rated SDLV
Nov 2004, David Hawkins, ATK Thiokol

Figure 6-76. Failure Rates of Solid and Liquid Propulsion Systems

A more focused historical comparison can be made by looking at the failure history of large-throat SRBs, including Titan IV, Ariane V, and Shuttle SRB. A summary of these flights is presented in **Table 6-36** and shows an overall history of 3 failures out of 362 attempts, for a failure probability of 0.83 percent. However, the data also shows a marked difference between the failure rate of the Shuttle SRB and the Ariane and Titan SRBs. Historically, the Shuttle SRB has been 3.3 times more reliable than Titan IV and Ariane V (taken together), similar to the factor of 3.2 between solids and liquids, generally. This difference may be partially due to the larger number of Shuttle SRB flights, but it also suggests that Shuttle SRB reliability is achieved by design features other than just propellant type.

Table 6-36. Large-Throat SRB Flight Failure History

Large-Throat SRB Attempts and Failures (Flight Demonstrated Only)				
Vehicle	**Flights**	**Attempts (Flights x 2)**	**SRB Failures**	
Shuttle	113	226	1	
Total	**113**	**226**	**1**	**1 in 226**
Titan IV A	22	44	1	
Titan IV B	14	28		
Titan 34D	15	30	1	
Ariane V	17	34		
Total	**68**	**136**	**2**	**1 in 68**

Shuttle RSRBs have reliability drivers that are unique among SRMs. First, they have been designed from the ground up for human flight and, as a result, have been designed to greater margins of safety and built under more stringent process controls than other systems. Also, they are recoverable, which has allowed for failure precursor identification and correction, which has contributed to continual design improvements throughout the program. **Figure 6-77** highlights some of the safety enhancements that have resulted from post-flight inspection. As shown in **Figure 6-78**, post-flight inspection (along with the post-Challenger redesign to the RSRM) has been an integral part of an aggressive ongoing program of reliability improvement for the Shuttle RSRB.

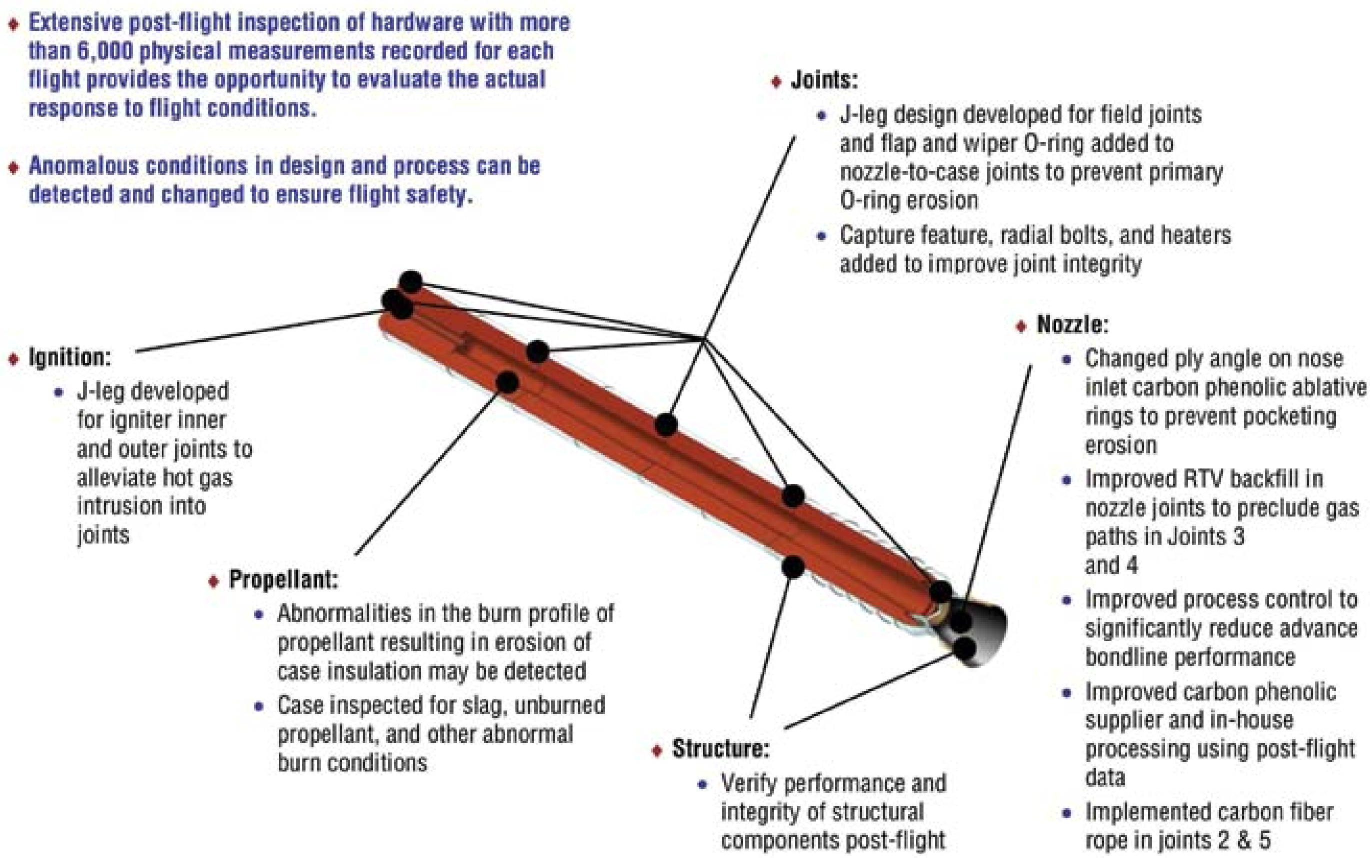


Figure 6-77. Safety Enhancements from Post-Flight Inspection

It is worth noting the significant design differences between a boost stage using Shuttle RSB and a boost stage using small strap-on solid motors such as the Delta Castor IV and Delta 2 GEM. These small strap-on motors have entered the human spaceflight debate as a result of the OSP–ELV Human Flight Safety Report Certification Study, which recommended against their use for crewed flight. The basis for this recommendation was that, although small strap-on solids are individually reliable (estimated at 0.9987), failures of these motors serve as undetectable initiators of liquid core explosion, which requires an estimated 2 sec of abort warning to escape. Additionally, since small strap-ons provide relatively little delta-V, multiple strap-ons are often required for performance reasons, multiplying their overall risk. The combination of reasonably high cumulative risk and their (assumed) undetectable failure modes makes them incompatible with crew survivability.

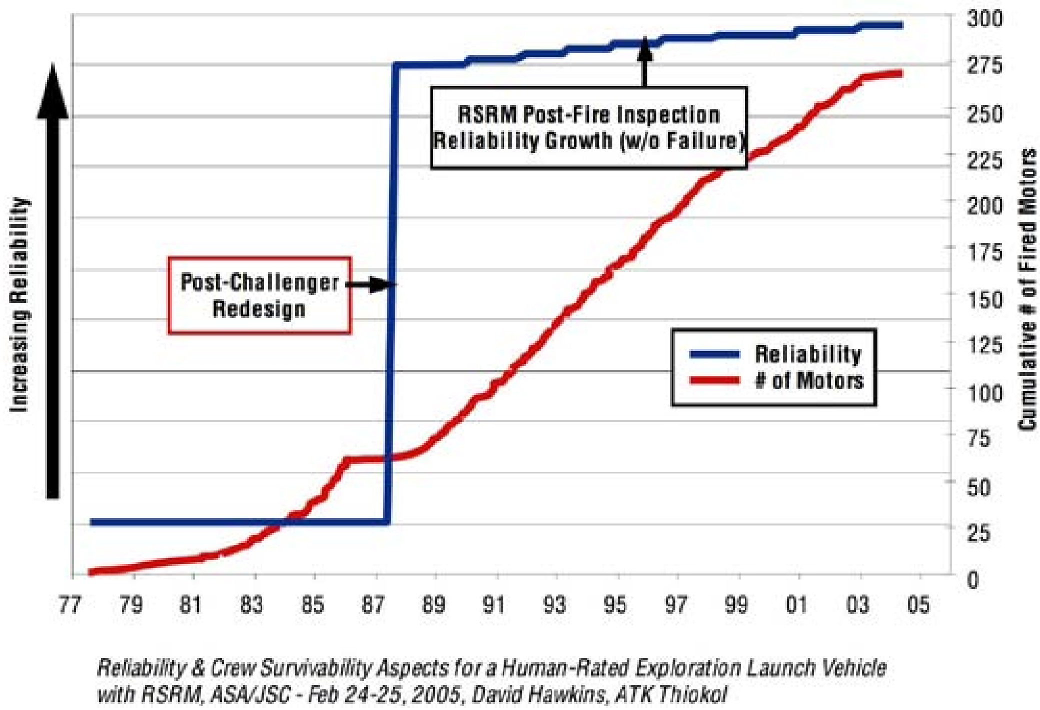


Figure 6-78. Historical Increase in RSRB Reliability

These concerns do not apply to the Shuttle-derived CLV configuration because a single RSRB provides the entirety of the first-stage performance; hence, the issue of cumulative risk over multiple units is not applicable. In fact, the RSRB failure risk replaces liquid core risk, as opposed to adding to it, as is the case for strap-ons. Moreover, as is discussed in **Section 6.8.9.5**, **RSRB Survivability**, the RSRB does not have the explosive potential of a liquid core stage, so the 2-sec abort warning requirement does not apply. Finally, RSRB failures are detectable (as demonstrated by the Challenger accident).

6.8.9.4 Solid versus Liquid Reliability Estimates

Predictions of future reliability must go beyond the raw data of history and capture the improvements in design, production, and management that tend to make the next flight more reliable than the average of the previous flights. A number of analyses have been completed or are in process to address this issue, including QRAS and the ongoing Shuttle PRA. Both analyses indicate that the Shuttle RSRM catastrophic failure rate is approximately three times less than that of the SSME, the only man-rated U.S. liquid propulsion system currently operating. However, the SSME also has a significant non-catastrophic failure probability of 1 in 640, resulting in a total SSME failure rate that is roughly 20 times higher than RSRM. Comparison of man-rated solid and liquid vehicle reliability estimates has also been performed using FIRST, which includes man-rated EELV reliability estimates, as shown in **Figure 6-79**.

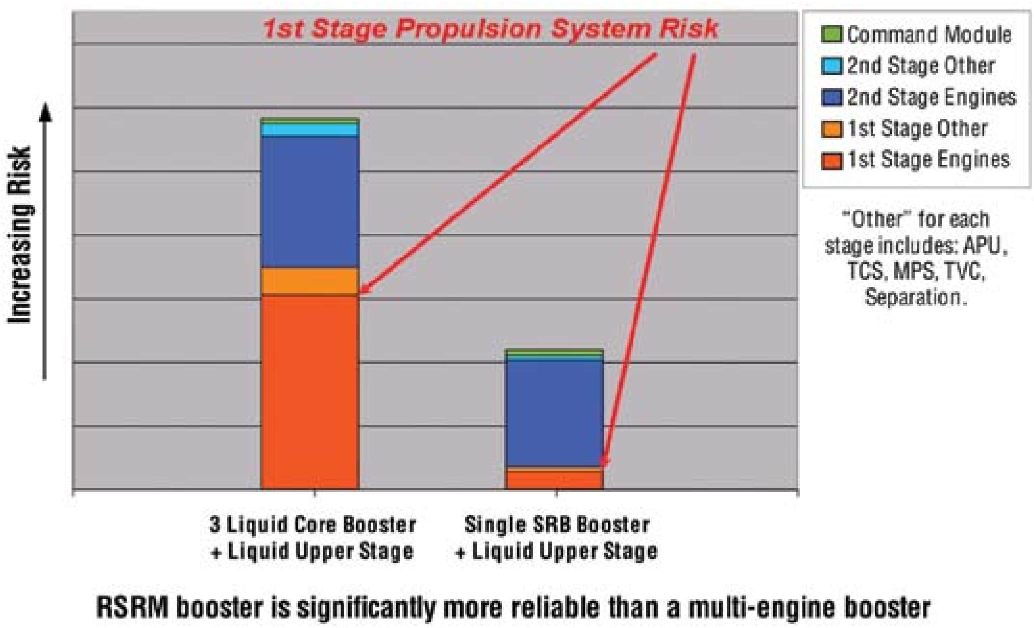


Figure 6-79. Comparison of Man-Rated Vehicle Risk Forecasts

6.8.9.5 RSRB Survivability

Although propulsion system reliability is a primary concern, especially with respect to crewed vehicles, crew survival is assured by an effective abort system that functions successfully in cases of LV failure. Survivability depends on the vehicle's ability to detect failures as they unfold and initiate abort early enough to remove the crew from the hazardous environment of the failing LV. Assessment of survivability requires an understanding of the detectability of each failure mode and the accident stresses they produce.

6.8.9.6 RSRM Failure Environments

RSRM failures can be broadly categorized into two risk significant case failure types: case breach and case rupture. Case breach entails release of hot gases from the internal volume and has a number of potential consequences. It may initiate other failures of systems in the proximity of the breach, as was the case for Challenger, where a joint leak impinged on the ET, causing it to explode. It may create an imbalanced thrust, which may or may not be within the capability of the TVC system to counter. It may also be benign.

Case rupture entails the large-scale release of chamber pressure and can occur as a result of extreme pressure buildup due to large pieces of solid propellant breaking off and clogging the exhaust bore or due to a crack or flaw within the solid propellant that increases the burn area, creating extreme pressure gradients. It is important to note that the propellant does not explode, and a fireball is not created; chunks of burning propellant are released, and an overpressure wave is produced as a result of the rapid depressurization of the chamber volume.

Of the two types of failure, case rupture creates the more extreme environment. However, even this bounding environment is significantly more benign than that of a liquid stage containing large quantities of propellant. **Figure 6-80** presents a comparison of the explosive energies of the two systems as a function of time. The left axis indicates the Trinitrotoluene (TNT) equivalent, representing the amount of TNT that would produce an explosion of comparable size. ("TNT equivalent" is a common method of normalizing explosive yields from a variety of materials under a variety of conditions and allows the use of empirical TNT overpressure equations.) The axis on the right indicates the critical distance associated with the TNT equivalent. The critical distance is the required distance from the explosion origin in order not to exceed the CEV overpressure design limit—a nominal value of 10 psi was used for this analysis. It can be seen from **Figure 6-80** that the maximum explosive potential of the RSRM is six times less than that of Delta IV or Atlas V. It can also be seen from the In-Line Configuration (ILC) CLV icon on the right that the maximum RSRM critical distance is less than the distance between the forward RSRM segment and the CEV, indicating that abort lead time may not be needed for RSRM failures (assuming they do not propagate to the upper stage). This is not the case for the Delta IV, which has a critical distance that exceeds the entire height of the ILC CLV.

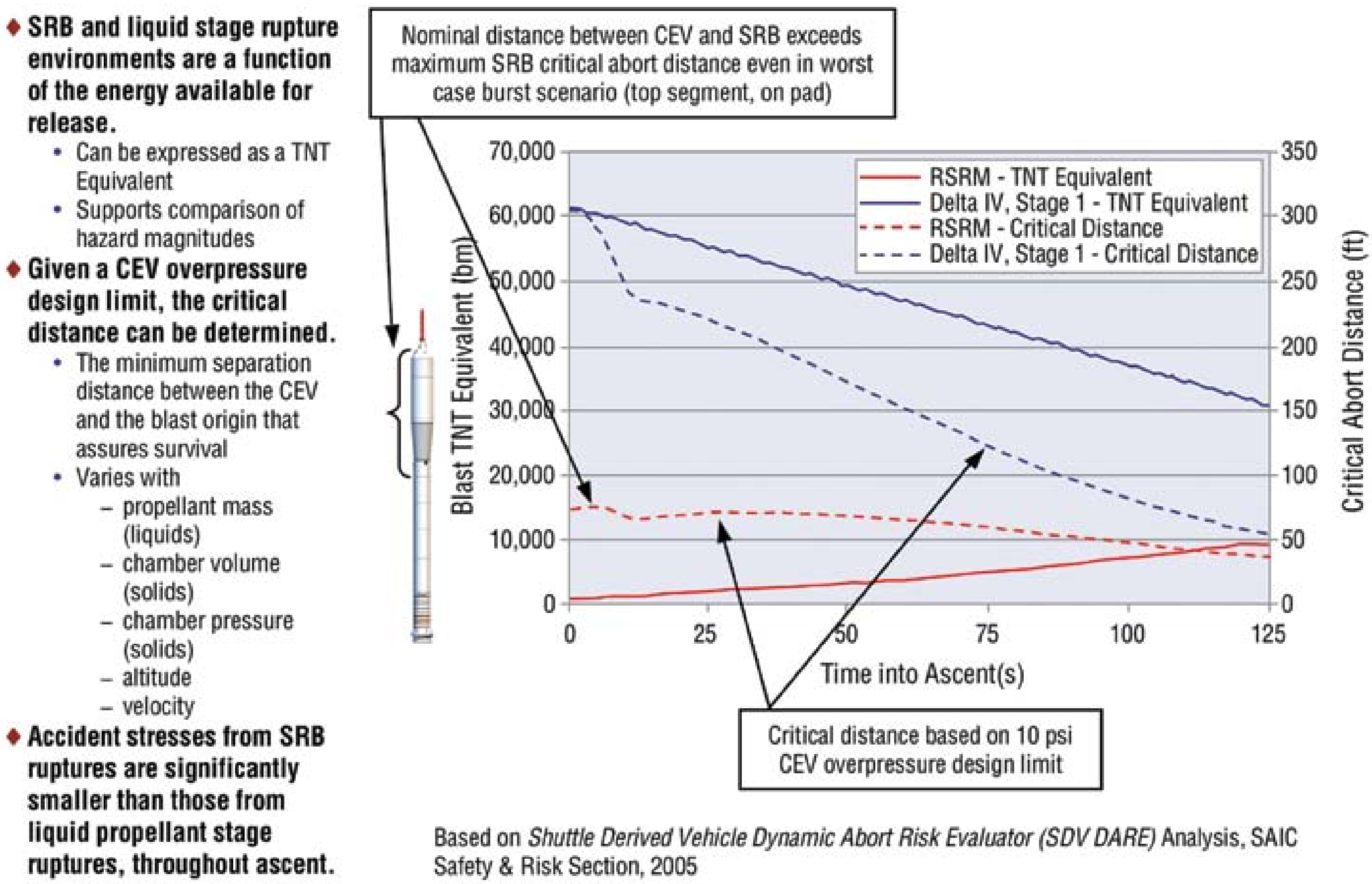


Figure 6-80. Comparison of RSRM and Liquid Stage Rupture Environments

This finding is corroborated by a high-resolution Computational Fluid Dynamics (CFD) analysis that was completed using a spectrum of rupture and flight conditions and showed that the overpressure experienced by the CEV on the stack would be negligible. Results of this study are shown in **Figures 6-81** and **6-82**.

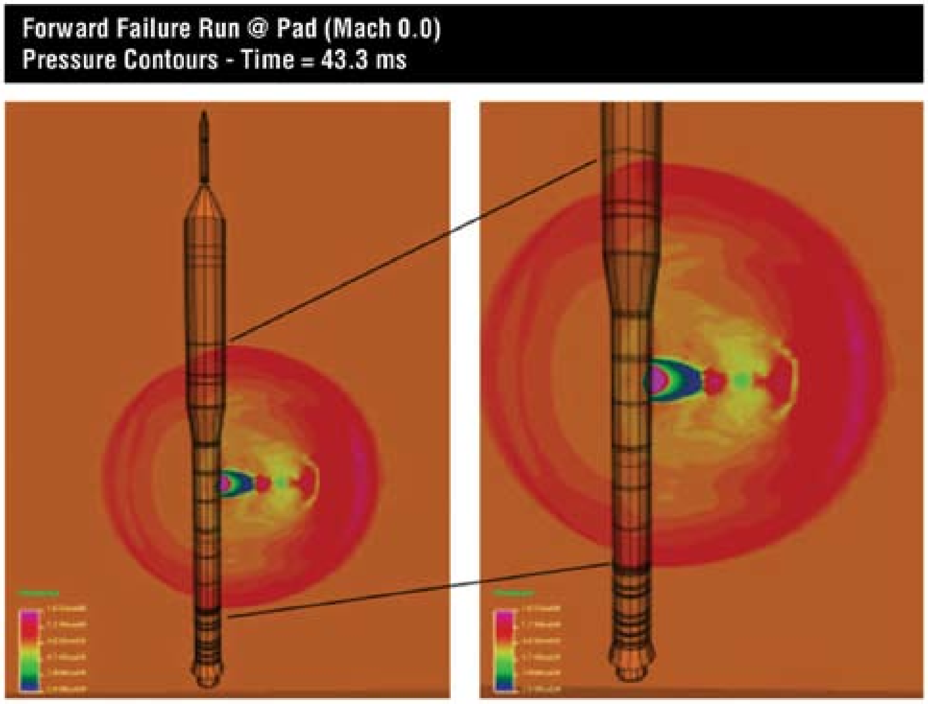


Figure 6-81. Worst-Case Condition of a Forward Segment Rupture on the Pad

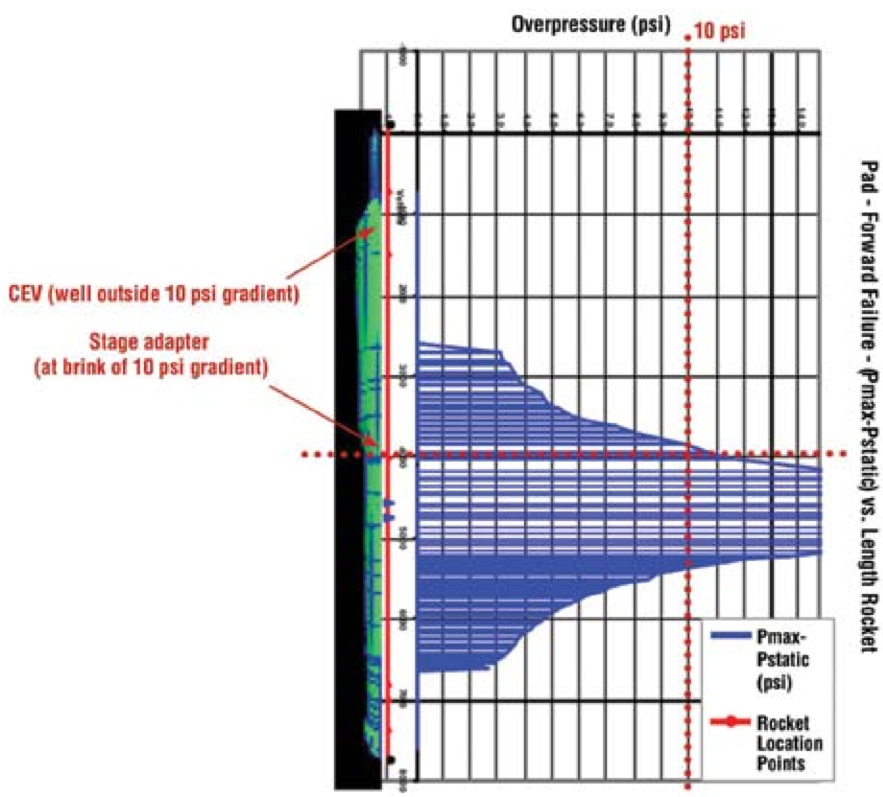


Figure 6-82. Overpressure as a Function of Distance from Rupture Origin

There is the additional consideration of failure propagation to the upper stage, which may exacerbate the stresses on the CEV as it attempts an abort. The propagation potential is failure mode dependent, and there is always the possibility of designing for more graceful failure (e.g., designing the aft SRB segment to rupture preferentially) or hardening against failure stresses (e.g., designing the interstage to withstand RSRM burst overpressures). The primary qualitative difference between SRB and liquid stage propagation potential is that bounding SRB accident stresses are likely to be amenable to mitigation (e.g., hardening), whereas bounding liquid stage accident stresses would likely overwhelm any practical mitigation.

6.8.9.7 Abort Lead Time

To develop a sense of the abort lead times that might be achieved by an Integrated Vehicle Health Management (IVHM) system against the RSRM failure modes, a so-called "Zero Warning Failure" study was conducted whereby detection lead time ranges were assigned to each RSRM failure mode identified in Shuttle PRA Version 1. The results of this study are summarized in **Table 6-37**, which shows that 44 percent of the RSRM risk is associated with very long abort lead times, while the remaining 56 percent is associated with lead times between 0–5 sec.

Table 6-37. Abort Lead Time Estimates for RSRM Failure Modes

RSRM Failure Modes	Potential Detection Method	Reaction Time(s)
Joints		
BB Rotor Joints Failure	Infrared (IR) Camera; Thermocouple (TC)	5–130
Flex Bearing Joint Failure	TC; Pressure Transducer (P)	30–130
Field Case Joint Failure	IR, Visual; Burn Through Wire; TC; P	5–130
Igniter Case Joint Failure	IR, Visual; Burn Through Wire; TC; P	5–130
Igniter Joint Failure	IR Camera	5–130
Nozzle Joint 1 Failure	Burn Through Wire; TC	5–10
Nozzle Joint 2 Failure	TC	5–130
Nozzle Joint 3 Failure	TC; P	5–130
Nozzle Joint 4 Failure	Burn Through Wire	5–130
Nozzle Joint 5 Failure	Burn Through Wire	20–130
Nozzle Case Joint Failure	Visual; Burn Through Wire; TC; P	5–130
OPT Joint Failure	IR; TC	5–130
Safe and Arm (S&A) Gasket Joint Failure	IR; TC	5–130
SII Joints Failure	IR; TC	5–130
Structure		
Case Segment Structural Failure	P; Strain Gauge	0–5
Igniter Structural Failure	P; Strain Gauge	0–5
Nozzle Housing Structural Failure	Strain Gauge	0–5
Thermal		
Case Insulator Failure	Visual; Burn Through Wire; TC; P	5–130
Igniter Insulation Failure	P; Strain Guage	5–130
Nozzle Phenolics Failure	Only Aft Exit Cone—Visual; TVC	5–130
Performance		
Nozzle Failure	Only Aft Exit Cone—Visual; TVC	5–130
Motor Propellant Failure	P	0–5

Note: Current RSRM only has Pressure Transducers (P) in the forward end of the motor.

It is important to note that the range of 0–5 sec is very wide and does not indicate that the failure mode is unabortable. Instead, the analysis focuses attention on those failure modes that require additional investigation with respect to survivability, because the potential accident environment hazards cannot be dismissed solely on the grounds of lead time alone. Instead, the question becomes whether the critical distances exceed the actual CEV separation distances for the specific accident stresses and lead times associated with each of the four identified "zero warning" failure modes. This question was addressed using the Space Shuttle Dynamic Abort Risk Evaluator (DARE) abort modeling tool, which probabilistically samples abort lead times over the ranges provided previously, comparing the resulting separation distances and critical distances to generate an abort failure probability. The result is a mean SRB abort ability from an ILC CLV configuration of 95 percent, which includes a failure-mode-specific assessment of the potential for propagation to the upper stage. Because the 0–5 sec range is large, uncertainty is relatively high; nevertheless, the conclusion is that RSRB failures are likely to be detectable with sufficient time margin for successful abort.

6.8.9.8 Summary

History and analysis shows that RSRB failures are detectable and far less severe than catastrophic liquid engine failures, affording comparatively benign environments for crew escape. An estimated 95 percent of RSRB failures are detectable with sufficient time for successful abort. Moreover, the probability of RSRM failure is far less than the alternative liquid propulsion solutions. This is due in part to the relative simplicity of solid propulsion systems versus liquid systems, but also as a result of the conservative design and manufacturing standards under which the RSRM is built, and the extensive post-flight inspection process that has contributed to reliability growth through numerous upgrades. **Table 6-38** presents a summary comparison between the ILC and triple-core EELV configurations.

Table 6-38. Boost Stage Safety Comparison

Reliability Drivers	Single-Stick SRB	Shuttle	EELV (Triple Core)
Simplicity	Single element	2 SRBs plus 3 staged combustion engines	3 engines (with 2 turbopumps), 3 feedback control systems, (1 staged combustion), 3 propellant management systems, 3 purging systems
Dynamics (moving parts)	1 TVC	5 TVCs, 6 high-performance turbopumps with Preburners	3 TVCs, 6 turbopumps, 3 throttle valves, numerous prop management valves
Understanding of the environment (margin)	226 flight operations, with post-flight inspection	113 flight operations	1 EELV HLV flight, conflagration during Delta launch, LOX-rich environment (RD–180)
Process control and feedback	Post-flight inspection, production process controls	Post-flight inspection, production process controls	No post-flight inspection; Rely on process control in flight (redlines)
Survivability Drivers			
Trajectory (g loads on abort reentry)	Crew escape system: Flat trajectory with mild g-loads	No crew escape system	Crew escape system: Requires more lofted trajectory with higher loads on crew (more so with Delta); can be mitigated with new upper stage
Detection lead time	~95% sufficient	N/A	84% to 90% sufficient
Accident environment	Low overpressure explosive environments, thrust augmentation typically leads to slow single-stack breakup	SRB thrust augmentation leads to immediate breakup and potential propellant mixing/conflagration	Potentially high overpressure/conflagration environments, thrust augmentation or engine shutdown can lead to interactions between cores

6.9 LV Subsystem Descriptions and Risk Assessments

6.9.1 Overview of Options Compared

The ESAS team made a comparative risk assessment of CLV options for ISS (and transition paths to EDS and CaLV for lunar missions). Both Shuttle-derived and EELV-derived configurations were assessed. The analysis focused on subsystems evaluations, including development risks and development schedule assessment. The following subset represents the CLV options studied by the team:

- Shuttle-derived options:
  - Five-segment RSRB with four LR–85 (new expander cycle) upper stage engines,
  - Four-segment RSRB with one RS–25d (altitude-start SSME) upper stage engine,
  - Five-segment RSRB with one RS–25d upper stage engine,
  - Five-segment RSRB with one J–2S upper stage engine, and
  - Five-segment RSRB/ET variant with four RS–25d engines.
- EELV-derived options:
  - Atlas V (new upper stage) with three RD–180 booster engines and four RL–10 upper stage engines, and
  - Delta IV (new upper stage) with three RS–68 booster engines and four RL–10 upper-stage engines.

Each system was analyzed relative to its earliest availability date; first, second, and third level schedule critical path; development risk; acquisition strategy; mitigation schedule; and extensibility to EDS and CaLV. **Appendix 6G**, **Candidate Vehicle Subsystems**, provides a more detailed summary of the analysis results by system.

6.9.2 Shuttle-Derived Assessments

For the five-segment RSRB with four LR–85 upper stage engines (designated LV 15), a new upper stage with the proposed clean-sheet expander cycle engine was the major concern. The earliest availability of such a system was determined to be 2014, with possible schedule mitigation to 2013. Critical path items include the LR–85 upper stage engine, upper stage MPS, avionics, and flight software. The system was found to be extensible to the EDS, with partial extensibility to the HLLV. Overall development risk was determined to be medium, based on the need for an entire new upper stage and an upper stage engine.

For the preferred CLV using a four-segment RSRB with one RS–25 upper stage engine (designated LV 13.1), the major concern was that it would be a new large upper stage design. The schedule of 2013 was seen as attainable, with mitigation strategies applied for availability in 2011. The primary critical path was the MPS, followed by avionics, flight software, and the four-segment RSRB. The system was found to be partially extensible to the EDS. The RS–25 is not extensible to the EDS, but it is extensible to the HLLV. This vehicle concept is shown in **Figure 6-83**.

Figure 6-83. LV 13.1 Concept

Several commonality options exist between the four-segment RSRB/one RS–25 upper stage and the envisioned EDS. Significant cost savings may be realized by this commonality. For example:

- Adapters, payload fairing, and separation system,
- Launch infrastructure,
- Production and handling infrastructure,
- Avionics (basic avionics could be the same),
- Tank (tank cylinder plugs enable multiple lengths),
- Umbilicals (aft umbilicals for simplified ground operations),
- Aft thrust structure,
- Engine mounts and gimbals, and
- Propulsion (main engine, feed system).

Overall development risk was scored as low, based on the availability of critical existing booster and RS–25 assets.

For the five-segment RSRB with one RS–25 upper stage engine, the same major concerns apply. Schedule availability of 2013 could potentially be mitigated to 2012. Critical path items were the same, except for the noted five-segment RSRB. The same acquisition strategy as the four-segment RSRB was applied. As was shown above for the four-segment RSRB with one RS–25, the system was determined to be partially extensible to the EDS. The RS–25 and the five-segment RSRB are extensible to the HLLV. Again, development risk was determined to be low, because it leveraged existing assets or upgrades.

In analyzing the five-segment RSRB with one J–2S upper stage engine (designated LV 16), the major concern was a new upper stage and upper stage engine development. Schedule availability of 2014 could be mitigated to 2012. The primary critical path item was the J–2S engine, followed by the MPS, avionics, and flight software. This configuration was found to be extensible to the EDS and partially extensible to the HLLV. Overall development risk was scored as medium, due to engine redevelopment and certification.

6.9.3 EELV-Derived Assessments

The Atlas V outfitted with three RD–180s on the core and a new upper stage outfitted with four RL–10 engines (designated LV 2) was a major concern, because it is a modified vehicle with the need to Americanize and human rate the RD–180 booster engines, in addition to the margin, reliability, and safety upgrades needed to human rate the RL–10 and current vehicle designs. The schedule availability in 2014 could be mitigated to late 2012. The primary critical path driver was the RD–180, followed by the MPS and the RL–10. This vehicle was not extensible to the EDS, but was partially extensible to the HLLV (boosters and core propulsion). The development risk was scored as high. See **Section 6.5.4.2**, **EELV Modifications for Human-Rating Summary** for more details.

The major concerns with the Delta IV outfitted with three RS–68s and a new upper stage with four RL–10 engines (designated LV 4), are the new upper stage and human rating the RS–68, RL–10, and overall vehicle. The 2013 schedule could be mitigated to early 2012. Critical path items included the RL–10 engine, the MPS, and the RS–68 engine, in that order. This vehicle was not extensible to the EDS or to the HLLV. Overall development risk was assessed to be medium. See **Section 6.5.4.2, EELV Modifications for Human-Rating Summary**.

The results of the assessments are contained in **Appendix 6F**, **EELV Modifications for Human-Rating Detailed Assessment**, and contain extensive company-proprietary data.

6.9.4 Summary Assessment of the RS–25 as an Upper Stage Engine

The RS–25, shown in **Figure 6-84**, was recommended by the ESAS team as the most viable upper stage engine for the following primary reasons:

- The RS–25 is a technically feasible upper stage engine and was considered a low-risk approach.
- The RS–25 engine is a practical near-term engine schedule solution because it could be developed and certified in approximately 3 years and meets the calendar year 2011 first human launch date.
- The Rough Order of Magnitude (ROM) DDT&E costs to certify the present configuration for upper stage use are reasonable and much less than certification costs of a new engine.

Significant SSME flight hardware would be available to support early upper stage development and would provide a major cost savings. If the Shuttle flight manifest remains at 28, there are 12 engines available at the end of 2010. If the Shuttle manifest is cut to 16, an additional 2 engines are available, for a total of 14 available in early 2009. It is likely that one of three current development engines could be made available in 2007 to begin testing. The time needed to build long-lead components, such as nozzles, is approximately 5 years. The ESAS team assumption for this study is that there are 16 Shuttle flights remaining and there will be 14 Block 2 flight engines available for CLV use. Additional flight engines would be required to meet the proposed manifest, but bringing production to current capacity of six engines per year is possible, and this rate can support the proposed manifest.

Figure 6-84. RS–25 (Altitude-Start SSME)

6.9.4.1 RS–25 Altitude-Start Evaluation

A single RS–25 (Block 2 SSME) is recommended for the upper stage engine. The overall goal is to minimize modifications to the current configuration and operation. Since the engine must operate with no gravity head, the design goal is to minimize propellant tank pressures. The engine start would be at approximately launch-plus-2 minutes.

RS–25 modifications and objectives of the test program would address engine thermal conditioning, engine prestart purging, and engine start sequence, including achieving sufficient oxidizer inlet pressure in the absence of an oxidizer gravity head.

Although starting an RS–25 at altitude presents a risk, an evaluation conducted in 1993 looked at using the Phase 2 SSME for altitude-start. The results indicated that a new start sequence would be required and that higher turbine temperature spikes would be provided. A 2004 study indicated that additional start sequence updates would be required and the inlet conditions were not optimal. Further refinement is in progress. The overall conclusion is that RS–25 altitude-start for an upper stage application is feasible.

6.9.5 Space Shuttle SRB, Four-Segment SRB Derivative

More than 200 four-segment SRBs have been flown on the Space Shuttle Program, with a total of 42 SRM static-test firings, 18 of which are RSRM tests and ongoing production. The Shuttle Program currently has reusable assets for flight beyond 2020 with the current four-segment configuration, which is shown in **Figure 6-85**.

Figure 6-85. SRB Four-Segment Configuration in Production

Status is as follows:

- Four-segment production is performed at ATK Thiokol. KSC has supported up to 19 motors (8 flight sets and 3 static tests).
- Six production RSRM flight sets have been built. An additional 23 sets are available in the current contract scope.
- SRB hardware deliveries are set to support the Shuttle to 2010, but can be extended.
- This option has minimal “keep alive” issues.

Other considerations for use as a CLV include enhancements that may be required as intermediate block upgrades, including motor insulation material obsolescence, recovery systems, and propellant upgrades and nozzle extension for increased Isp. The development plan is based on minimal burn-rate reduction for dynamic pressure reduction and minor propellant grain modifications.

Motor specifications are given in **Table 6-39**. Given these parameters, this system is capable of delivering the performance needed for a CLV.

Table 6-39. Four-Segment SRB Performance Specifications

Propellant	PBAN
Total Isp (M lbf/sec)	296.3
Chamber pressure (psia)	625
Maximum Thrust (lbf)	3,331.400
Burn Time (sec)	123.5
Burn Rate (in/sec)	0.368
Initial T/W	1.52

The development schedule goal for this approach puts first human flight in 2011. This assumes that new avionics will be required (a 3.5-year schedule driver) and that the first crewed flight hardware delivery will be in 2010. This approach also assumes that there will be cost synergies gained from contract bridging to Shuttle production, with minimal "keep alive" costs due to current Shuttle hardware production projected to system retirement in 2011. This schedule is based on a production rate of 10 or more motors per year, with a capability for a total of 19.

Risks, Opportunities, and Watches (ROW) for the four-segment SRB development are listed in **Table 6-40**.

Table 6-40. Four-Segment SRB Development Risk Summary

Area	ROW	Notes
Asset Transition	Watch	Schedule and cost assumptions are based on existing hardware migration. Lead times for design and manufacturing of new case hardware is a key driver.
Avionics	Watch	Schedule driver for early flight. Aggressive with or without Shuttle hardware migration.
Obsolescence	Watch	Obsolescence historically required vigilance and continual funding. Planning should remain in place to address obsolescence issues.
System Design	Watch	Schedule assumes slight design changes and accelerated review/manufacturing. Does not address significant changes due to new loads, controls, etc.

The goal of using the SRB for the CLV is to take advantage of an existing booster with little risk to the manufacturing schedule and cost. Overall, development risk is low with utilization of existing assets and experience. Facilities and hardware risk is low, without significant vendor ramp-up. **Table 6-41** categorizes ROW items related to required changes.

Component	ROW	Notes
System	Opportunity	Mature design, experienced staff, and existing test stands with 150+ four-segment firings. Analytical tools and skills in place to support minor design changes.
Structures	Watch	Preliminary assessment shows margin for structures and joints.
Insulation	Watch	Chrysotile replacement certification for four-segment allows block upgrade without additional test program.
Separation System	Watch	Currently qualifying ATK as a new source for BSM. Design change is to be determined.
Avionics	Risk (Low)	Replacement/upgrade of outdated parts necessary during the life of the Exploration Program.
Recovery System	Watch	Minimal design change.

Table 6-41. Four-Segment SRB Change Risk Summary

The Space Shuttle system is a significant asset, with existing RSRM and SRB hardware in inventory that can be transitioned to the CLV for a cost and schedule benefit. Attrition rates are less than 10 percent. Both the booster and motor can support an 8-flight set throughput per year. Production supports the Shuttle transition with 10 to 14 extra motors built at the end of a 16-Shuttle mission schedule (at the current rate). This assumes current inventory and contract structure, four-segment baseline (no additional hardware needed), production ramping up to 10 motors per year from ATK beginning in 2006, refurbishment of hardware, and no additional attrition. A key factor is that no upgrades are needed to current capability.

Obsolescence and vendor issues are workable. Only one key obsolescence issue is not being addressed by the current Space Shuttle Program: the closing of the RSRM case segment manufacturing and heat-treatment facility. Relocation/reconstitution has a 2.5-year schedule to production. However, this affects only the five-segment SRB. In addition, SRB forward- and aft-skirt vendors no longer exist. Based on an SRB study conducted in 2000, it would take approximately 3 years to qualify a new vendor. Currently, the RSRM insulation (chrysotile) replacement activity in work is included in the RSRM Program Operating Plan through 2010. Although not required for the current manifest, it will be needed for flights in 2010 and beyond. The RSRM nozzle is a rayon material, with enough material available for 68 additional nozzles. Qualification of a new vendor is captured in the current Space Shuttle Program cost.

6.9.6 Upper Stage and Interstages Subsystems

6.9.6.1 Upper Stage RCS

The goal of a new upper stage RCS is to meet requirements for crew missions. A conceptual schematic is shown in **Figure 6-86**.

The upper stage RCS configuration of the reference LV 13.1 CLV is based on a hypergolic R4D-based architecture. Propellants are Monomethylhydrazine (MMH) and Nitrogen Tetroxide (NTO) (Mon-3), with a pressure-fed thruster configuration. System pressure is approximately 50 to 300 psia, and RCS thrust level is approximately 100 lbf. The upper stage RCS represents the state-of-the-art in space propulsion capability. Qualified vendors are available for RCS development. Considerations for use in a human-rated system include use of components that either have been human rated (such as the R4D thrusters), can accommodate human rating with new development (propellant and pressurant tanks), or are derived from existing state-of-the-art capabilities (other components). Test issues include availability of test stand and early test stand preparation. Overall, development risk is low, as shown in **Figure 6-86**.

Goal – New upper stage to meet mission requirements for crew

Upper Stage RCS Configuration
- POD: Hypergolic R4D Based Architecture
- Propellants:
  - MMH and NTO (Mon-3)
  - Pressure-Fed Thruster Configuration
  - System Pressure~ 50-300 psia
  - RCS Thrust Level ~100 lbf

History/Status
- State-of-the-Art Space Propulsion Capability
- Qualified vendors available for RCS development

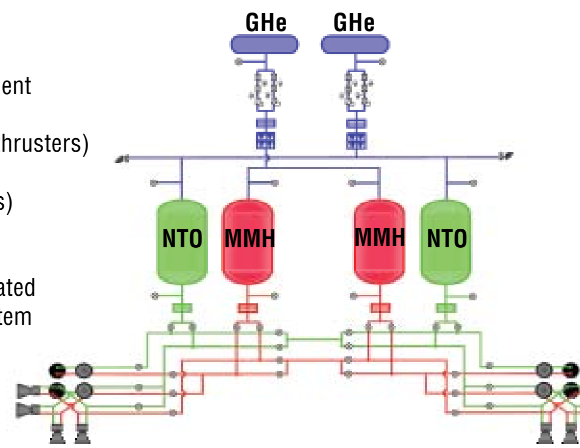
Considerations for use in Human-Rated System
- Components either have human-rating (R4D thrusters) or can accommodate human-rating with new development (propellant and pressurant tanks) or derived design development (components)

Development Path/Issues
- Typical development cycle can be accommodated in identified schedule with aggressive subsystem and hardware start
- High parts count necessitates early start

Risk
- Overall Low Development Risk

Test Issues
- Availability of test stand and earlytest stand preparation

Conceptual Schematic



Component	Low Risk
System	RCS subsystem design is similar to SOA designs and capabilities.
Tankage	Existing vendor base with new tank development, but within technology and SOA technical basis.
Thrusters	Existing vendor base with SOA thrusters.
Components	SOA or SOA-derived components.
Avionics	SOA avionics boxes.

Figure 6-86. Upper Stage RCS Conceptual Schematic

6.9.6.2 Upper Stage Structures

The upper stage structural elements of the reference LV 13.1 CLV consist of the following load-bearing structures: LOX/LH2 tanks, intertank, forward skirt, aft skirt, and thrust structure. **Figure 6-87** shows the breakout of the upper stage structural elements. A systems tunnel is provided as a secondary structure.

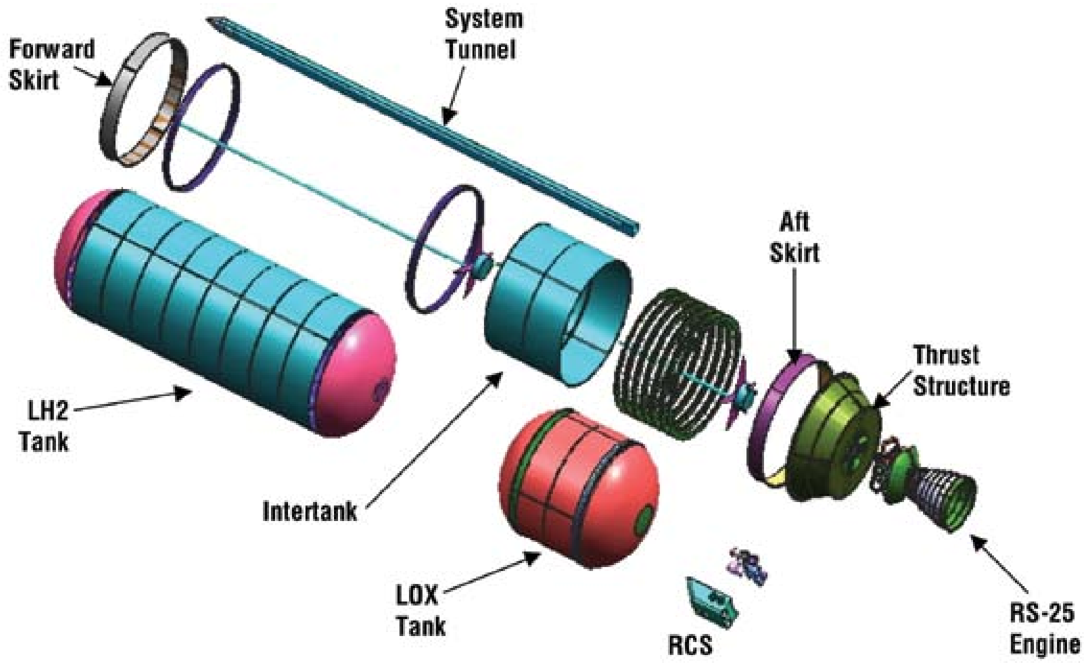


Figure 6-87. Upper Stage Structural Elements

6.9.6.2.1 Tanks and Primary Structures

The LH2 tank and the LOX tank provide the storage for the fuel and oxidizer for the MPS. The initial design was sized to carry 35,000 gallons of LOX and 93,000 gallons of LH2. The intertank is a cylindrical structure that serves as the load-bearing interface between the two tanks (LOX and LH2). It has integral stiffeners and can serve for the mounting of avionics, pressure bottles, and other elements.

The upper stage forward skirt provides the interface and load path for the LOX tank and the payload adapter. This cylindrical or conical structural element is integrally stiffened and may provide additional mounting for avionics. The aft skirt is the transitional structural interface between the LH2 tank and the thrust structure. It also provides attach points for other elements, such as the RCS.

The thrust structure is the primary load-carrying conduit for loads resulting from engine operations. It also provides the mounting surface for the engines.

Each of the primary structures leverages existing technology to minimize risks. Optimization of the stiffening components of each subassembly could act to decrease weight while satisfactorily meeting the established structural strength requirements.

During prelaunch, the TPS controls the propellant boil-off, stratification, and loading accuracy; prevents air liquefaction and ice/frost formation; maintains acceptable interface conditions; and provides acceptable structural margins at liftoff. Selection of TPS materials to be used on the various components is based on the above requirements and manufacturing concerns, such as TPS closeouts and TPS debris requirements.

Several key trades for the upper stage structural elements should be performed, including:

- Material selection;
- Fabrication techniques and methodology;
- Component- versus system-level test and verification;
- Open versus closed intertanks and interstages;
- Common/nested bulkhead versus separate tankage;
- Tank geometry effects on interstage to intertank purge requirements and umbilical requirements;
- Selection of TPS materials to be used on the various components based on ice/frost, stratification, cryogenic heat leaks, air liquefaction, TPS closeouts, and TPS debris requirements;
- Single versus multiple systems tunnel configuration;
- Instrumentation selection and redundancy approaches;
- Potential failure response and detection architectures;
- Optimized ground checkout strategies; and
- Hardware commonality and cost reduction evaluations.

Upper stage structures driving factors included material selection, stiffened panel configuration selection, and component integration selection. Below is a list of the key upper stage assumptions:

- Components to be designed and tested to 1.4 factor of safety (FS);
- Aluminum 2219 material used in sizing effort;
- Isogrid panel configuration for all barrel and cylinder components;
- Classic y-ring component integration with friction-stir weld; and
- Isogrid panel interstage cylinder.

6.9.6.2.2 Interstages and Secondary Structure

The interstages consist of several integral pieces of structural hardware that are necessary to connect the primary structures together into a total CLV. Specifically, the interstage elements connect the primary structures of the spacecraft/payload to the upper stage and the upper stage to the booster. This hardware is designated as the spacecraft/payload adapter, the interstage, and the forward frustum. In addition, the system tunnels and flight termination system are also included in the interstages. **Figure 6-88** shows the interstages structural elements within the upper stage.

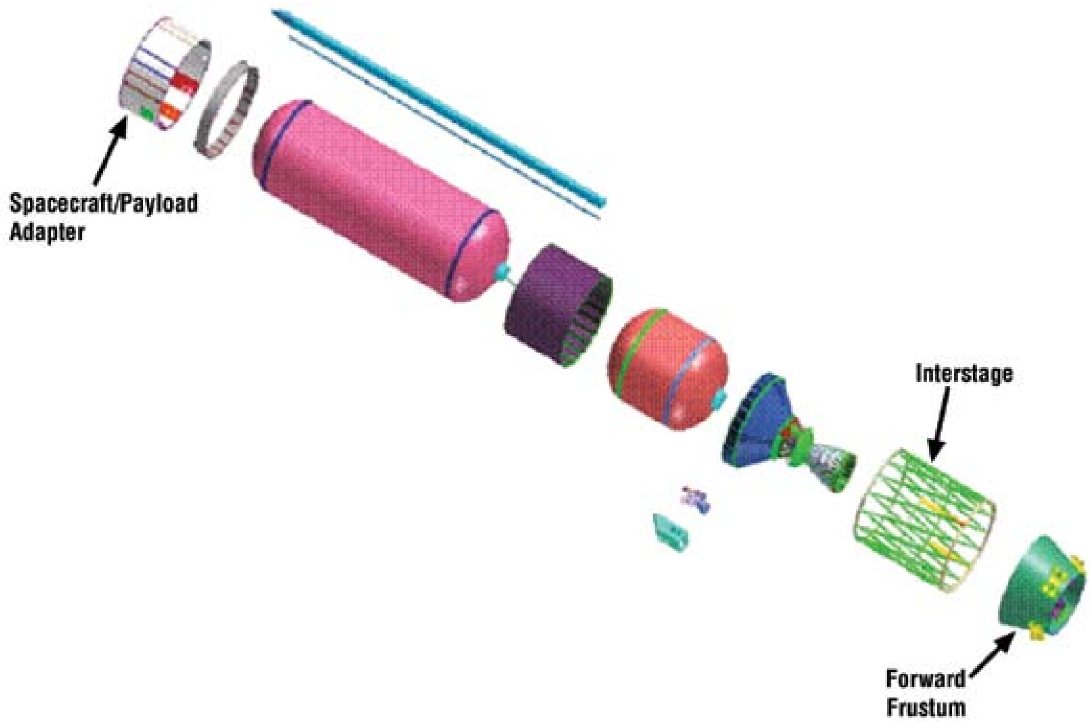


Figure 6-88. Interstages Structural Elements Within the Upper Stage

The forward frustum hardware consists of the forward frustum structure and its subsystems that include the booster RCS, the booster recovery system, booster avionics packaging, purge and vent, and any associated Government-Supplied Equipment (GSE). The booster RCS is shown in greater detail in **Figure 6-89**. It is a blow-down hydrazine system mounted as four replaceable units with four 900-pound thrusters each. It will be used for CLV roll control during ascent.

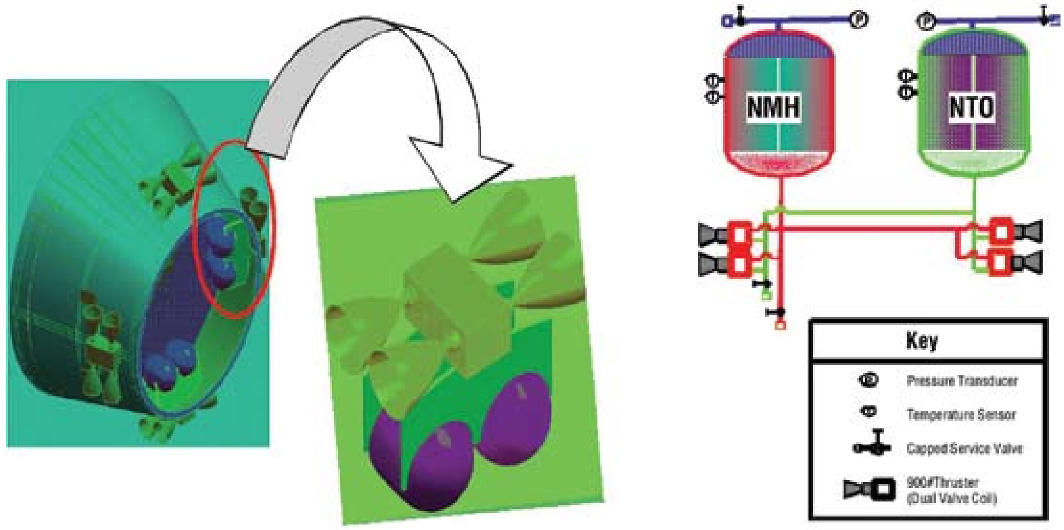


Figure 6-89. Booster RCS

The interstage itself is a cylindrically shaped structure that connects to the upper stage's thrust structure (forward) and the booster stage's forward frustum (aft). Its primary function is for the separation of the upper stage and booster stage. This segment, whether open truss work or a closed reinforced cylindrical shell, is key to the vehicle's mission. It not only provides the mechanism to withstand launch and flight loads, but, because the thrust structure houses the upper stage's engine and the forward frustum houses the booster RCS and other avionics hardware, the interstage's purge and vent system must be designed to facilitate proper operation of these subsystems. The ullage settling motors are also housed within the interstage.

The separation of the booster and upper stage takes place through the separation systems within the interstage structure. One such separation system, that of the booster from the interstage, will initially separate at the aft end of the interstage, with the interstage structure remaining connected to the upper stage. The second separation system is that of the interstage and the upper stage. This will jettison the interstage structure away from the upper stage after the engine reaches 100 percent thrust. The separation approach will leverage work completed for the Saturn Program and make use of separation concepts currently employed for Shuttle operations and ELVs. **Figure 6-90** shows the interstage structural element.

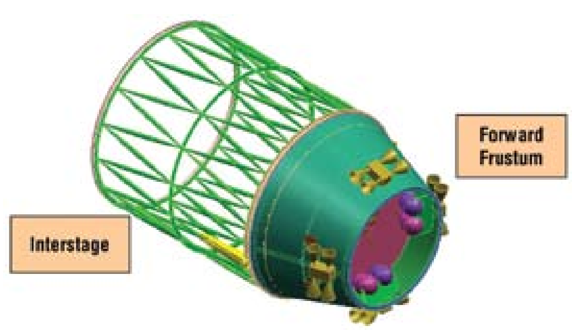


Figure 6-90. Interstage Structural Element Attached to the Forward Frustum

The Spacecraft/Payload Adapter (SPA) structural element is shown in **Figure 6-91**. The SPA is approximately 83 inches long with a 217-inch diameter. After receiving CEV configuration data, including the length of the CEV nozzle, this length was initially set to 105 inches. The SPA has a rigid connection to the upper stage and a separation system interface to the spacecraft. It also contains most of the avionics for control of the LV. On the pad, it will require a purge and an electrical umbilical, and it includes an access door. The baseline assumption is that passive cooling of the avionics is adequate. The SPA provides the mechanical and electrical interfaces between the CEV and the LV and also provides the appropriate accommodations for the LV avionics system. The SPA hardware consists of the spacecraft/payload structure, the upper stage avionics, the separation system, purge and vent, and any associated GSE.

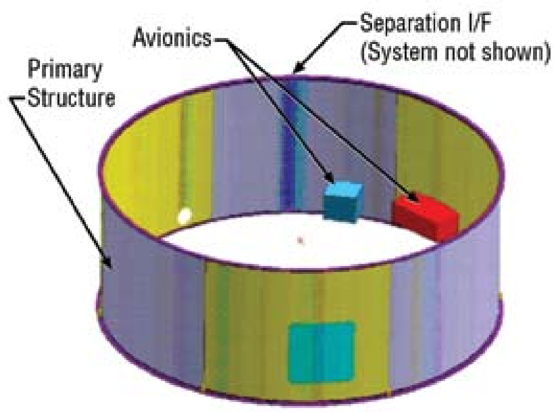


Figure 6-91. SPA Structural Element

6.9.6.3 Upper Stage and Interstage Schedule

The upper stage and interstages are not currently on the program critical path; however, there are several areas that are considered the schedule drivers for structures: (1) requirements; (2) major reviews of Preliminary Design Review (PDR) and Critical Design Review (CDR); (3) tooling modification for 5.5-m tank fabrication; (4) hardware manufacturing; (5) assembly and integration of primary structure and secondary structures with various subsystems required for Prototype Test Article (PTA)/Static Test Article (STA); and (6) modal and structural testing.

The upper stage major milestones and high-level development schedule are as follows:

- March 2009: Upper stage structures delivery to MPTA;
- May 2009: Upper stage structures delivery for STA;
- July 2009: Upper stage structures delivery for the second Risk Reduction Flight (RRF–2)/ Certification Flight 1;
- September 2009: Upper stage structures delivery for RRF–3/Certification Flight 2; and
- November 2009: Upper stage structures delivery for first human flight unit.

Major components of the interstage structures that may support RRF–1 will include the forward frustum, booster recovery system, booster RCS for roll control, and separation systems for the upper stage to booster. The interstages major milestones and high-level development schedule include:

- March 2009: Interstages structures delivery to STA;
- To Be Determined: Interstages structures delivery to RRF–1;
- July 2009: Interstages structures delivery for RRF–2/Certification Flight 1;
- September 2009: Interstages structures delivery for RRF–3/Certification Flight 2; and
- November 2009: Interstages structures delivery for first human flight unit.

These milestones and the overall schedule are discussed in more detail in **Section 10**, **Test and Evaluation**, and **Section 11**, **Integrated Master Schedule**.

6.9.6.4 Upper Stage and Interstages Risks

Table 6-42 describes the key ROWs for the upper stage Primary and Secondary (PS) structure, along with the Interstage (IS) structures. Overall CLV Program risk for the development of this subsystem is recognized as low to medium due to the clean-sheet design, Government-led design and development through PDR, and the baselined 5.5-m tankage (driven by CEV interfaces) driving new fabrication tooling. Offsetting these potential risks is the utilization of existing fabrication processes and techniques.

Table 6-42. Upper Stage and Interstages Structures Risk Summary

Risk ID	Area	ROW	Description
PS–1	Clean-Sheet Design and Development Timeline	Watch	Clean-sheet DDT&E cycle is longer than using existing or modified system. Fluctuating vehicle requirements could extend development cycle.
PS–2	5.5-m Tank Fabrication	Watch	5.5-m tank fabrication is not synergistic with current 5.0-m tank fabrication performed for EELV. Ability to facilitate for production capability while sustaining current Shuttle ET fabrication requirements may impose program risk.
PS–3	Composite Structures	Watch	Current structures assumptions represent low-risk material selection. Decision to migrate to composite structure with low TRL would introduce additional program risk.
IS–1	Separation Systems	Risk (Medium)	Two separation systems are needed to complete mission objectives. Leverage existing systems to facilitate design, but integration issues remain. Debris generation models and alternate separation methods studies needed.
IS–2	Potential to use existing subsystem hardware designs or derived designs	Opportunity	Lengthy DDT&E can be avoided through the use of Shuttle hardware or derived hardware designs and qualification (where applicable).
IS–3	Transition to Prime Contractor	Risk (Low)	Transition of the interstage DDT&E at PDR could cause schedule slips from extended contract negotiations/ award, requirements creep, etc.
IS–4	Government-led Activity	Risk (Medium)	Ability of the Government to perform initial program design and development phases while continuing to support Space Shuttle Program manifest.

6.9.7 Avionics Subsystem

6.9.7.1 Avionics Subsystem Description

The avionics subsystem for the baseline CLV, LV 13.1, is depicted in **Figure 6-92**. As shown in the diagram, the CLV avionics systems physically partitions into three primary vehicle elements: upper stage, boost stage, and the upper stage main engine.

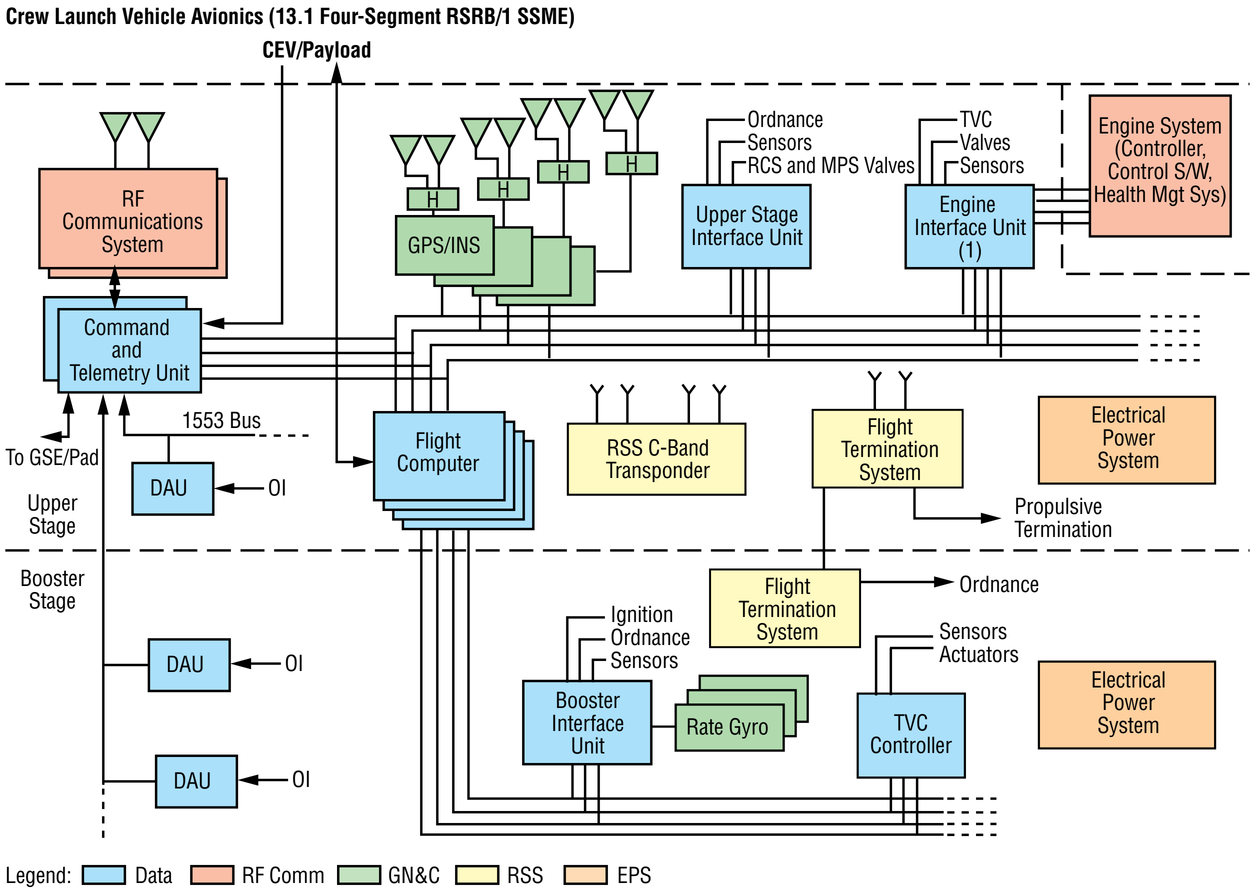


Figure 6-92
Conceptual CLV Avionics Architecture

6.9.7.2 Avionics Subsystem Development

The current element-level development philosophy is that the boost stage element will require minimal modifications of the proposed avionics. Potential changes for this element, with respect to avionics, will be driven by either the propulsion engineers, the health management requirements, or designing out obsolete components. The upper stage element will be the most significant piece of DDT&E for the team. The ground rule is to utilize heritage subsystems to minimize development risk. The main engine for the upper stage is a heritage engine, and the avionics associated with the engine have a defined evolutionary path from the existing SSME.

The primary function of the CLV avionics is to safely guide and control the propulsion stages of the CLV and lift the CEV/CDV into the defined mission orbit. Flight avionics will consist of component subsystems, such as command and data handling, flight software, sensors and instrumentation, video, communications, vehicle management, power systems, electrical integration, and electrical GSE.

The avionics system will interface with the CEV, CDV, payloads, and ground support systems. These interfaces will be defined in program documentation, such as Interface Control Documents (ICDs) and Interface Requirements Documents (IRDs). Depending on program-level documentation structuring, there may be an Interface Definition Document, which would define the complete capability of the interface.

The key features of the conceptual avionics architecture are a traditional approach with heritage electronics that provides for a low-risk development; a practical vehicle management system with health function focusing on crew abort management and on board flight termination; a fail operational/fail safe avionics system architecture where the second major component failure safely recommends crew abort; an onboard range tracking function with the goal of eliminating dependency on the current Air Force ground-tracking sites and associated cost; and an independent flight control capability from the CEV. Additionally, the LV 13.1 conceptual avionics architecture lends itself to a high degree of commonality with the LV 27.3 CDV avionics. A design goal was to make the interfaces with the launch pad and CEV/CDV as clean and loosely coupled as possible. Although the avionics architecture depicts these interfaces as such, system-level requirements may drive these interfaces to be more complex. The initial design currently has no plan to distribute power across the interface between CLV and CEV/CDV.

6.9.7.3 Avionics Schedule

The team will implement a traditional, but accelerated, requirements development plan. Accelerated requirements development introduces risk and the possibility that avionics requirements development may be inconsistent with vehicle requirements. However, this approach optimizes the overall avionics development effort. The avionics system requirements lag the vehicle element and System Requirements Reviews (SRRs), and the avionics component SRRs and PDRs feed other vehicle system-level PDRs and CDRs.

Parallel development during requirements, preliminary design, and critical design phases will be necessary to achieve major program milestones. The avionics major milestones include:

- March 2009: Avionics Delivery for MPTA;
- August 2009: Avionics Delivery for RRF–1;
- December 2009: Avionics Delivery for RRF–2;
- April 2010: Avionics Delivery for RRF–3; and
- August 2010: Avionics Delivery for ISS–1.

These milestones and the overall schedule are discussed in more detail in **Section 10**, **Test and Evaluation**, and **Section 11**, **Integrated Master Schedule**. The avionics subsystem is not currently on the CLV program critical path; however, there are five major areas that are considered the schedule drivers for avionics: (1) flight software; (2) GN&C hardware; (3) Global Positioning System (GPS)/Inertial Navigation System (INS); (4) GN&C rate-gyro assembly; and (5) the flight computer.

6.9.7.4 Avionics Risk

The avionics development plan will follow a traditional avionics architecture approach and utilize existing avionics technologies for subsystem development, resulting in minimized risk when compared to a new technology development approach. However, all new avionics will be developed for this vehicle and will be subject to some low to medium risks identified in **Table 6-43** below.

Table 6-43. Avionics Risk Summary

Title	Risk Level	Risk Description
Avionics System	Low	Traditional avionics architecture with heritage electronics augmented by practical vehicle management. Avionics system requirements development lags vehicle system requirements and is susceptible to inevitable change. Software is a long-lead item tied to operations philosophy. Test program becomes compressed and potentially jeopardized.
Software	Medium	Software will be a critical path item. This software architecture will be challenged with requirements for (1) human rating, (2) vehicle management, and (3) operations concept.
Redundancy	Watch	Redundancy management is implemented across subsystem interfaces (operational, software, electrical, and mechanical) and becomes quite intricate. Requirements and testing are essential.
Electrical and Electronic Engineering (EEE) Parts	Watch	Part choices and selection are limited for space-rated electronic parts and usually require long-lead procurements.
Vehicle Management	Opportunity	Practical vehicle management provides three major vehicle functions: crew abort management, onboard flight termination system, and pad interface diagnostics.
Engine Controller (Delivered with Engine System)	Medium	The engine controller hardware and software will be a schedule risk based on previous engine experience. Engine health management is included in the engine controller.

6.10 LV Development Schedule Assessment

6.10.1 Schedule Approach

The requirements given to the ESAS team were based on three driving requirements: (1) first crewed flight to ISS in 2011; (2) the ESAS Traffic Model shown in **Figure 6-93**; and (3) the human-rating requirements derived from NPR 8705.2A, Human-Rating Requirements for Space Systems.

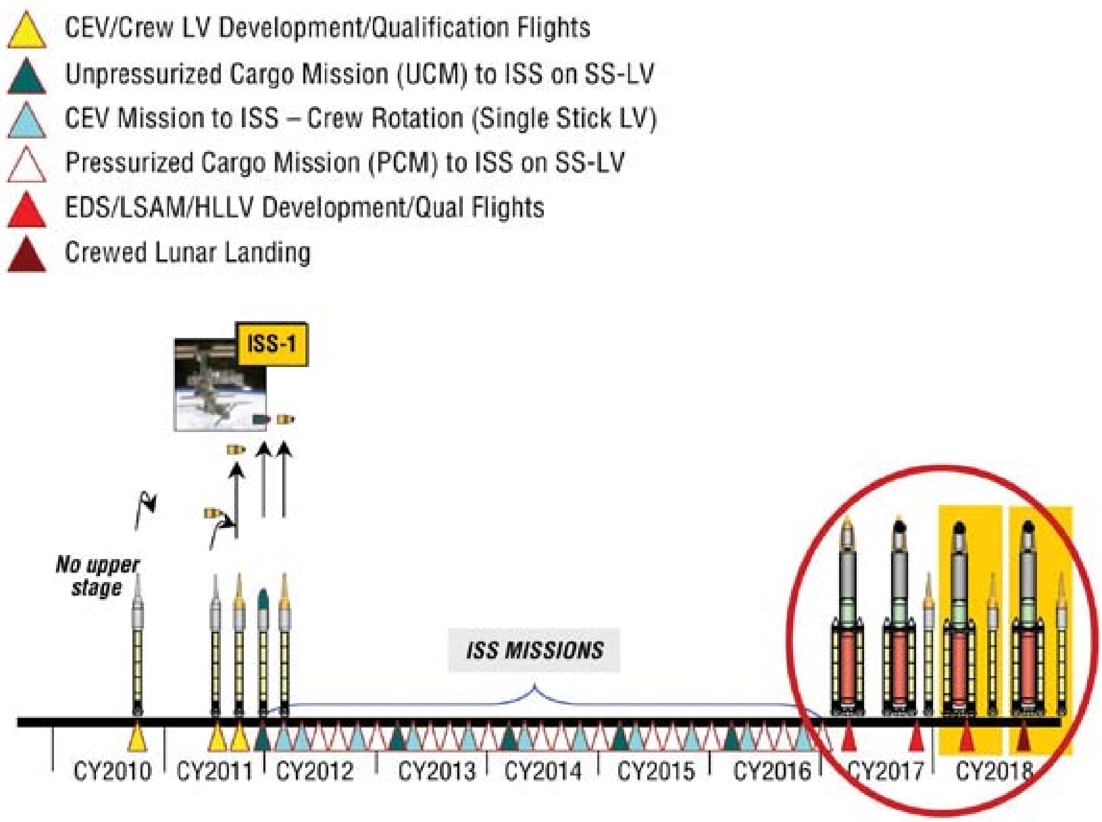


Figure 6-93. ESAS Traffic Model

The team's approach to the schedule was to build a detailed development Integrated Master Schedule (IMS) with all long-lead critical design, fabrication, and test tasks for the CLV from the EIRA in a logically linked Microsoft Project schedule. Although high-level CaLV schedules were developed, detailed focus was on the CLV because it is a near-term, critical-path item in the ESAS architecture. This entailed engaging engineers who have experience in developing flight hardware and software systems, using their expert judgment to define the tasks, task durations, and task relationships for each subsystem and SE&I activities necessary to design and develop the EIRA CLV (i.e., five-segment RSRB with an upper stage using a new expander cycle engine). The schedule feasibility for other alternatives was performed using a comparison approach by modifying the EIRA CLV bottom-up development schedule details (e.g., replace one engine schedule with another). The tasks were organized by a team WBS, shown in **Figure 6-94**.

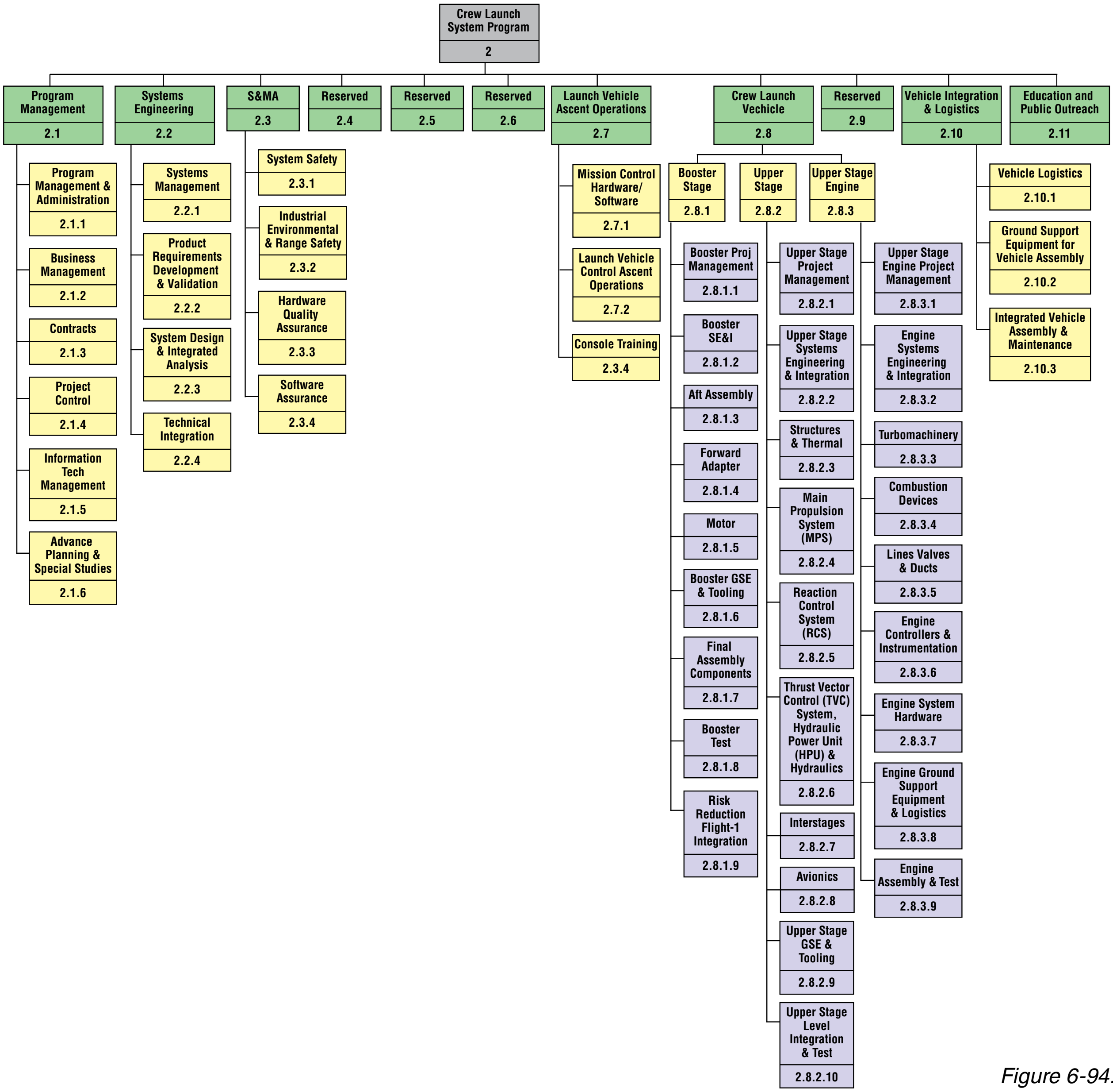


Figure 6-94. Team WBS

The CLV options evaluated by the team were:

- Shuttle-derived options: In-line four-segment or five-segment RSRB with a new upper stage with various engines (LR–85, J–2, J–2S, RL–10), and
- EELV options: Delta IV and Atlas V with new upper stage.

Once a detailed CLV IMS was built, the team evaluated the above alternatives against the driving requirements while also assessing the development feasibility of each of the proposed alternatives. For instance, the detailed schedule showed that a new upper stage engine development (LR–85) was the critical path driver for the EIRA CLV. The team then looked at alternatives to the engine development, such as J–2, SSME, and the RL–10. The schedule analysis focused on meeting a 2011 first human flight to LEO. Other technical and programmatic FOMs were being evaluated in parallel (cost, technical performance, and reliability). The initial detailed schedule construction and analysis revealed a launch date for first mission of no earlier than 2014 for EIRA CLV. In evaluating alternatives, it should be noted that many tasks were common or very similar for the Shuttle-derived CLV options, such as avionics, SE&I and structures, and MPSs. The engine proved to be the significant schedule discriminator among Shuttle-derived CLV options.

6.10.2 IMS for the Selected CLV (LV 13.1)

The IMS consists of subprojects developed by the WBS leads and their supporting engineering disciplines at the subsystem level that are then mapped to the CLV WBS. These subprojects were logically linked into a master schedule. The schedule is discussed in more detail in **Section 10**, **Test and Evaluation**, and **Section 11**, **Integrated Master Schedule**.

The integration logic of the IMS was built around the flow down of requirements to the component level.

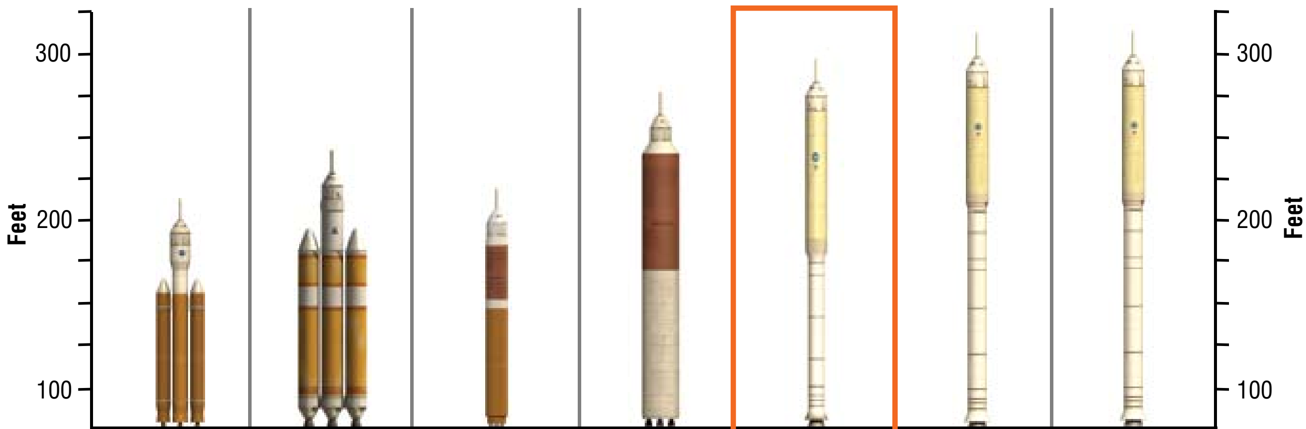
- LV system requirements are developed through an SE&I process resulting in an LV SRR.
- Requirements flowdown to the system elements (i.e., booster, upper stage, interstages) results in system element SRRs 3 months after the LV SRR.
- System element requirements flowdown to the subsystem level results in subsystem SRRs 3 months after system-element SRRs.
- Subsystem requirements flowdown to the components results in component requirements reviews, which then begin the preliminary design phase for each component.
- After the preliminary design phase, a PDR is conducted for each component.
- Component PDRs flow back up to a subsystem level and, 3 months later, a subsystem PDR is conducted.
- Subsystem PDRs flow back up to a system-element level and, 3 months later, a system-element PDR is conducted.
- System-element PDRs flow back up to the LV System and, 3 months later, the LV System PDR is conducted.
- The Critical Design and Design Certification phases follow the same path as the PDR, starting at the component and flowing back up to the LV Systems.

The primary critical path is driven by requirements flowing down from the LV System to system elements to subsystems to components. The driving component is in design cycles for valves and actuators for the MPS. MPS CDR drives the fabrication of feedlines for the MPS. The MPS is the schedule driver for integration of the upper stage MPTA. The MPTA is scheduled to perform 166 days of propulsion testing to qualify the upper stage. After the upper stage is qualified, greenrun tests are performed on the upper stages for RRF–2 and RRF–3 prior to shipment to KSC. These flights and the overall schedule are discussed in more detail in **Section 10**, **Test and Evaluation**, and **Section 11**, **Integrated Master Schedule**.

6.11 Conclusions

6.11.1 Crew Launch Vehicle

The Shuttle-derived in-line RSRB vehicle, LV 13.1, using a four-segment RSRB and a SSME-powered upper stage, provides the best option for meeting exploration crew transport goals, ISS crew transfer requirements, and ISS cargo resupply requirements. A summary of candidate CLVs and key parameters is shown below in **Figure 6-95**.



	Human-Rated Atlas V/New US	Human-Rated Delta IV/New US	Atlas Phase 2 (5.4-m Core)	Atlas Phase X (8-m Core)	4 Segment RSRB with 1 SSME	5 Segment RSRB with 1 J–2S	5 Segment RSRB with 4 LR–85
Payload (28.5°)	30 mT	28 mT	26 mT	70 mT	25 mT	26 mT	27 mT
Payload (51.6°)	27 mT	23 mT	25 mT	67 mT	23 mT	24 mT	25 mT
DDT&E*	1.18**	1.03	1.73**	2.36	1.00	1.3	1.39
Facilities Cost	.92	.92	.92	.92	1.00	1.00	1.00
Average Cost/Flight*	1.00	1.11	1.32	1.71	1.00	.96	.96
LOM (mean)	1 in 149	1 in 172	1 in 134	1 in 79	1 in 460	1 in 433	1 in 182
LOC (mean)	1 in 957	1 in 1,100	1 in 939	1 in 614	1 in 2,021	1 in 1,918	1 in 1,429

LOM: Loss of Mission   LOC: Loss of Crew   US: Upper Stage   RSRB: Reusable Solid Rocket Booster

* *All cost estimates include reserves (20% for DDT&E, 10% for Operations), Government oversight/full cost; Average cost/flight based on 6 launches per year.*

** *Assumes NASA has to fund the Americanization of the RD–180. Lockheed Martin is currently required to provide a co-production capability by the USAF.*

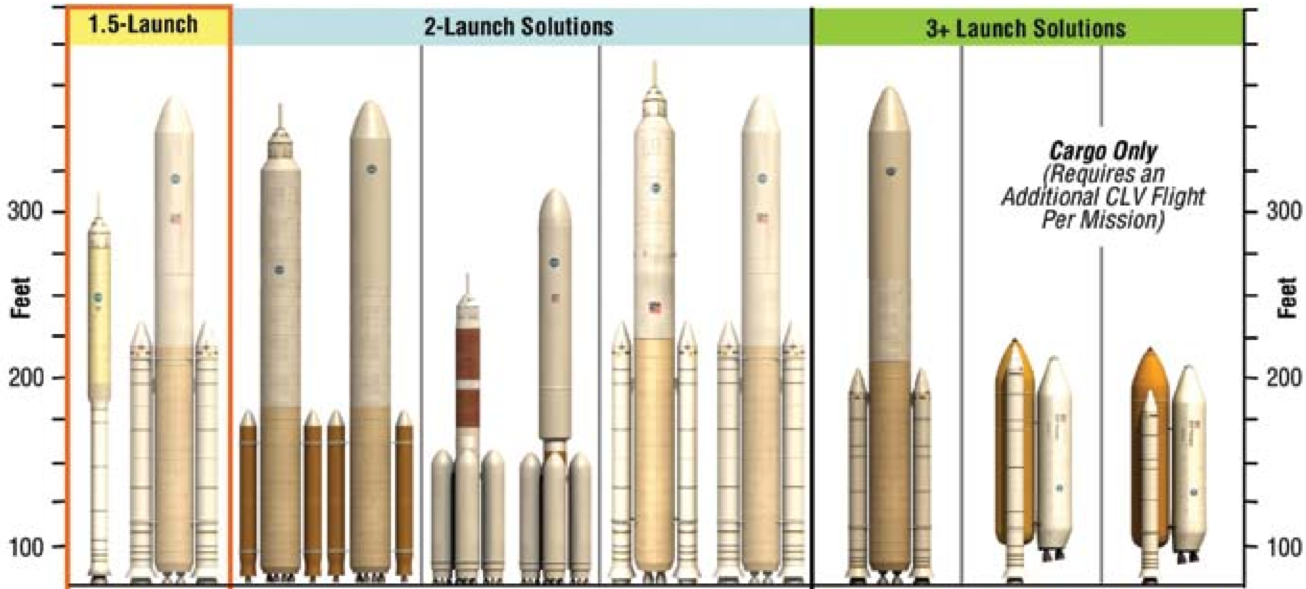
Figure 6-95. Comparison of Crew LEO Launch Systems

LV 13.1 exhibited the lowest predicted LOC probability and the lowest DDT&E cost of the options assessed. It provides a net delivered payload to exploration assembly orbits and ISS with sufficient margins to accommodate anticipated CEV masses. Infrastructure costs are slightly higher, approximately 8 percent, than EELV-derived options, and average cost per flight is comparable to EELV options (depending on flight rates) and 4 percent higher than other SDV options evaluated. Additionally, LV 13.1 provides key elements, particularly propulsion systems, vital to the development of the CaLV for lunar and Mars exploration. It also maintains the Nation's access to solid propellant production at current levels.

6.11.2 Cargo Launch Vehicle

The CaLV concept determined to offer the best option for meeting exploration goals is the Shuttle-derived in-line vehicle, LV 27.3, using two five-segment RSRBs and five SSMEs in the ET-diameter core vehicle. A summary of candidate CaLVs and key parameters is shown in **Figure 6-96**.

LV 27.3 is the only heavy-lift CaLV in the study trade space that enabled the "1.5-launch" solution for lunar missions for anticipated CEV and LSAM masses without the requirement to develop a two-stage core vehicle. The 125-mT lift capability increases mission safety and reliability by minimizing on-orbit assembly and multiple rendezvous and docking events. It exhibits LOM and LOC probabilities higher than EELV-derived options and has fewer discrete elements to develop than options derived from EELV elements. Previous studies did not show any advantage to new clean-sheet concepts, and, in fact, found them to be of significantly higher risk and cost, while not providing any advantage in lift capability, safety, or mission success. Only one 2-launch solution option, LV 27, exhibits a lower per-flight cost than LV 27.3, and it is the four-SSME core vehicle on which LV 27.3 is based. The LV family DDT&E is within 2 percent of the lowest 2-launch solution vehicle. Comparisons with 3+-launch solutions show savings of other options in both individual and family DDT&E costs, but per-mission costs would be significantly higher. The Shuttle-derived side-mounted vehicles provide the most commonality with the current Shuttle, but have significantly less lift capability (requiring at least four launches), exhibit higher production costs due to the carrier vehicle, and exhibit the least straightforward evolutionary path to Mars exploration lift requirements. The Shuttle-derived side-mounted is not considered to be a viable crewed configuration due to the requirement of the configuration to place the CEV within 10 feet of the LOX tank and a more-obstructed path away from the vehicle in the event of a launch abort. The in-line Shuttle-derived CaLV configuration provides enhanced safety for a crew (if needed), a straightforward upgrade path for Mars missions, and higher mission reliability for a small (2 percent) additional investment upfront for the CLV/CaLV combined development of LV 13.1 and LV 27.3, as compared to the lowest 2-launch solution combined option.



	5 Segment RSRB In-Line with 5 SSME Core - Cargo	Atlas Phase X (8-m Core)	Atlas Phase 3A (5.4-m Core)	5 Segment RSRB In-Line with 4 SSME Core	4 Segment RSRB In-Line with 3 SSME Core	5 Segment RSRB Side-mount with 3 SSME	4 Segment RSRB Side-mount with 3 SSME
Payload to 28.5	106 mT (125 mT w/ upperstage)	95 mT	94 mT	97 mT	74 mT	80 mT	67 mT
Lunar LV DDT&E*	1.00	1.29	.59	.96	.73	.80	.75
CLV+Lunar Crew/Cargo DDT&E (family op)*	1.00 (4 Seg RSRB w/1 SSME)	1.26 (Atlas V)	1.02 (Phase 2)	.98 (4 Seg RSRB w/1 SSME)	.83 (4 Seg RSRB w/1 SSME)	1.03 (5 Seg RSRB w/J2S-LR85)	.85 (4 Seg RSRB w/1 SSME)
Lunar LV Facility DDTE	1.00	1.33	2.25	1.00			
CLV+Lunar Facilities Cost	1.00	1.12	1.56	1.00			
Lunar LV Average Cost/Flight*	1.00	1.08	1.19	.87	.78	1.13**	1.13**
LOM – Cargo (mean)	1 in 124	1 in 71	1 in 88	1 in 133	1 in 176	1 in 172	1 in 173
LOC (mean)	1 in 2,021	1 in 536	1 in 612	1 in 915	1 in 1,170	N/A	N/A

* All cost estimates include reserves (20% for DDT&E, 10% for Operations), Government oversight/full cost; Average cost/flight based on 6 launches per year.

**Production costs are higher than in-line due to production of separate side-mount cargo carrier

Figure 6-96. Lunar Cargo Launch Comparison

6.11.3 Earth Departure Stage

The EDS concept for LV 27.3, EDS S2B4, can deliver 125 mT to 160 nmi LEO when used as an upper stage. This provides growth to anticipated Mars mission launch masses with no additional DDT&E expenditure.

6.11.4 Integrated Launch System Considerations

The combined LV 13.1/27.3 architecture development approach provides the highest potential for meeting a CEV IOC of 2011 and a CaLV IOC of mid-2010s due to these attributes:

- Requires no new engine system development for the CLV;
- Relies on the most extensive, human-rated U.S. operational database in history for the CLV propulsion elements;
- Requires that only the CLV (LV 13.1) be human rated, while preserving the option to human rate the CaLV (LV 27.3).
- Facilitates CaLV development by using two of the three required engine/motor systems needed for LV 27.3 and its EDS;
- Minimizes keep-alive costs and schedule issues for SSME and RSRB by continuing in production and launch/recovery operations; and
- Both LV 13.1 and LV 27.3 vehicles will draw from the same, existing ground infrastructure.

6.11.5 Final Considerations

The combined development of LV 13.1 and LV 27.3 pairs the most reliable, safest CLV with the most extensible, most reliable, and highest performing CaLV. The development of the CLV based on LV 13.1 will provide the most straightforward, structured progression to the 1.5-launch solution lunar architecture, while providing the lowest risk CLV development to acquire and maintain crewed access to LEO and the ISS. The 1.5-launch solution CaLV provides payload performance to TLI exceeding that of the Saturn V of the 1960s with minimal development and certification of critical flight elements. The use of the SSME in the core stage of the CaLV allows this high performance without the requirement for an upper stage (beyond the EDS) for LEO. The use of key elements from the current Shuttle system allows a straightforward path to human rating of the CLV.

7. Operations

7.1 Ground Operations

The Exploration Systems Architecture Study (ESAS) team addressed the launch site integration of the exploration systems. The team was fortunate to draw on expertise from members with historical and contemporary human space flight program experience including the Mercury, Gemini, Apollo, Skylab, Apollo Soyuz Test Project, Shuttle, and International Space Station (ISS) programs, as well as from members with ground operations experience reaching back to the Redstone, Jupiter, Pershing, and Titan launch vehicle programs. The team had a wealth of experience in both management and technical responsibilities and was able to draw on recent ground system concepts and other engineering products from the Orbital Space Plane (OSP) and Space Launch Initiative (SLI) programs, diverse X-vehicle projects, and leadership in NASA/Industry/Academia groups such as the Space Propulsion Synergy Team (SPST) and the Advanced Spaceport Technology Working Group (ASTWG).

7.1.1 Ground Operations Summary

The physical and functional integration of the proposed exploration architecture elements will occur at the primary launch site at the NASA Kennedy Space Center (KSC). In order to support the ESAS recommendation of the use of a Shuttle-derived Cargo Launch Vehicle (CaLV) and a separate Crew Launch Vehicle (CLV) for lunar missions and the use of a CLV for ISS missions, KSC's Launch Complex 39 facilities and ground equipment were selected for conversion. Ground-up replacement of the pads, assembly, refurbishment, and/or processing facilities was determined to be too costly and time-consuming to design, build, outfit, activate, and certify in a timely manner to support initial test flights leading to an operational CEV/CLV system by 2011. (Reference **Section 12**, **Cost**.) The ESAS team also performed a detailed examination of Launch Vehicle (LV) options derived from the Evolved Expendable Launch Vehicle (EELV) configurations options in support of the study. The results of those analyses and a technical description of the vehicle configurations considered can be found in **Section 6**, **Launch Vehicles and Earth Departure Stages**. For a description of the EELV-derived concepts of operation, refer to **Appendix 7A**, **EELV Ground Operations Assessment**. **Section 12**, **Cost**, provides the cost estimation results.

For similar cost- and schedule-related reasons, conversion of key facilities at KSC's Industrial Area is recommended for Crew Exploration Vehicle (CEV) spacecraft assembly and integration. The existing capabilities for human spacecraft processing are such that there was found to be no need to spend large amounts of resources to reproduce and construct new facilities to support the CEV.

The ESAS team began its architectural definition by defining reference concepts of operations that addressed the following: (1) CEV spacecraft assembly and checkout; (2) CLV and heavy-lift CaLV assembly; (3) CLV and CaLV space vehicle integration and launch operations; and (4) recovery and refurbishment operations of the reusable Solid Rocket Booster (SRB) and Crew Module (CM) elements. Indirect functions and infrastructures (e.g., facilities maintenance, flight and ground system logistics, support services, and sustaining engineering) were defined to support these operations as outlined in detail in **Appendix 7B**, **Concepts of Operations and Reference Flows**.

Cost-estimation analyses, which are detailed in **Section 12**, **Cost**, addressed the direct operations costs as well as the aforementioned infrastructure functions. Contemporary management and operations methods were assumed with no credit taken in the estimates for incorporating new methods. While it is anticipated that opportunities to incorporate new methods will be seized upon, the ESAS team believes NASA should demonstrate savings as the program progresses rather than promise savings at the program's outset. Advanced concepts for improving NASA's annual support costs for infrastructure consolidation, more efficient work control systems, and more advanced Command and Control (C&C) systems are addressed below.

The level of nonrecurring conversion work and recurring launch processing work will be highly dependent on the complexity of the flight system interfaces with the ground systems. Ground architecture conversion costs, conversion schedule, and annual recurring operations costs are highly dependent on the management process controlling the number and complexity of the flight-to-ground interfaces. Wherever possible and whenever practical, the study team searched for means to control these interfaces in the ESAS concepts and searched for innovative means to manage and contain their growth in the ESAS requirements emerging from the study. The following is a list of ESAS operability design drivers for management and control during design:

- Total number of separate identified vehicle systems;
- Total number of flight tanks in the architecture;
- Number of safety-driven functional requirements to maintain safe control of systems during flight and ground operations;
- Number of unplanned tasks;
- Number of planned tasks;
- Total number of required ground interface functions;
- Total number of active components;
- Number of different required fluids;
- Total number of vehicle support systems with element-to-element interfaces;
- Number of flight vehicle servicing interfaces;
- Number of confined/closed compartments;
- Number of commodities used requiring Self-Contained Atmospheric Protection Ensembles (SCAPE), medical support, and routine training;

- Number of safety-driven limited access control operations;
- Number of safing operations at landing;
- Number of mechanical element mating operations (element-to-element and element-to-ground);
- Number of separate electrical supply interfaces;
- Number of intrusive data gathering devices; and
- Number of Criticality 1 (Crit-1) system and failure analysis modes.

Additional detail, including key benchmarks, is provided in **Appendix 7C**, **ESAS Operability Design Drivers**.

The ESAS team also imposed a requirement for the gradual removal of hazardous and toxic commodities in time for the lunar program. This requirement states: "The Exploration Architecture subsystems, which require new development, shall not use expended toxic commodities." It further states "that the Exploration Program will develop a plan for legacy subsystems to eliminate use of any expended toxic commodities." (Refer to **Appendix 2D**, **ESAS Architecture Requirements**.) Specification of these commodities in flight systems requires expensive infrastructure capable of safely conducting such operations. The use of toxic commodities requires ground personnel working with these systems to wear special SCAPE suits, complicates launch facility design, slows down processing cycle times, imposes personnel hazards, and drives up infrastructure and logistics support costs. Examples are provided in **Appendix 7D**, **Toxic, Hazardous Operations Impacts**. While the ESAS cost estimates assumed that hazardous and toxic propellant servicing may be required for the initial LV, a technology integration plan was developed to eliminate the use of toxic commodities for the lunar missions and beyond as outlined in **Section 9**, **Technology Assessment**.

The ESAS team also recommended quantitative methods for managing and controlling critical flight and ground system design characteristics that pose ground operations and support risks (both safety hazard and cost risks) similar to the way flight system weight and flight performance are managed during traditional design processes. This "design-for-support" approach complements the traditional requirement for the launch site to support the design and is intended to create a more effective architecture for NASA that is safer, simpler, more affordable, and more dependable to develop, operate, and sustain.

7.1.2 Reference Ground Architecture Description

7.1.2.1 Flight System Assumptions

The ESAS examined various space vehicles for replacement of the Shuttle Orbiters as a means for human access to space. Much like today's Shuttle system, the ESAS reference architecture is also a partially reusable system and was selected from an array of options after a careful review of various risk factors, including crew safety, performance, and overall economy.

The chosen ESAS reference mission architecture calls for a "1.5-launch solution" for crewed lunar missions that use a Shuttle-derived CaLV to launch a Lunar Surface Access Module (LSAM) attached to an Earth Departure Stage (EDS). This is followed by the launch of a single four-segment SRB-derived CLV and a new upper stage propelled by a single Space Shuttle Main Engine (SSME) fed by Liquid Oxygen (LOX) and Liquid Hydrogen (LH2). The trade studies leading to the selection of this approach and a detailed definition of the selected elements are provided in **Section 4**, **Lunar Architecture** and **Section 6**, **Launch Vehicles and Earth Departure Stages**.

A critical element of the proposed architecture is the launching of the CEV on top of the CLV rather than a side-mounted approach. The integrated CEV spacecraft is composed of a reusable CM, an expended Service Module (SM) that houses support services such as power and in-space propulsion, and a Launch Abort System (LAS) that is nominally jettisoned and allows the crew a safe option for avoiding catastrophic LV events. A lunar CEV capsule can accommodate a nominal crew of four personnel. Configured for an ISS mission or future Mars mission scenarios, the CM can support a crew of six. The trade studies leading to the selection of this approach and a detailed definition of the systems are provided in **Section 5**, **Crew Exploration Vehicle**.

7.1.2.2 Concept of Operations Overview and Approach

The approach used to define the operations concept was to identify generic launch site functions and relate these to each flight hardware element and major assembly. Previous engineering efforts in this area, led by KSC, were used in this study. A more detailed definition of the generic functions drawn on by the ESAS team is provided in **Appendix 7E**, **Generic Ground Operations and Infrastructure Functions**.

The approach of relating the flight elements and major assemblies to the generic ground operations is depicted pictorially for the CEV/CLV space vehicle in **Figure 7-1** and for the LSAM/CaLV space vehicle in **Figure 7-2**. The figures show the flight hardware elements arriving at the launch site on the left side. From left to right, each figure follows the hardware conceptually as it arrives and goes through the various functions of a launch operation. The concept of operations diagram provides a structure to define: (1) the arrival concept; (2) the flight element receiving, assembly, and/or storage concept; (3) the vehicle integration concept; (4) the launch concept; and (5) the post-launch element recovery and reuse concept (if applicable). More detailed descriptions of the launch operations concepts in **Figure 7-1** and **Figure 7-2** are found in **Section 7.1.2.2.3**, **Reference Architecture Ground Processing Description.**

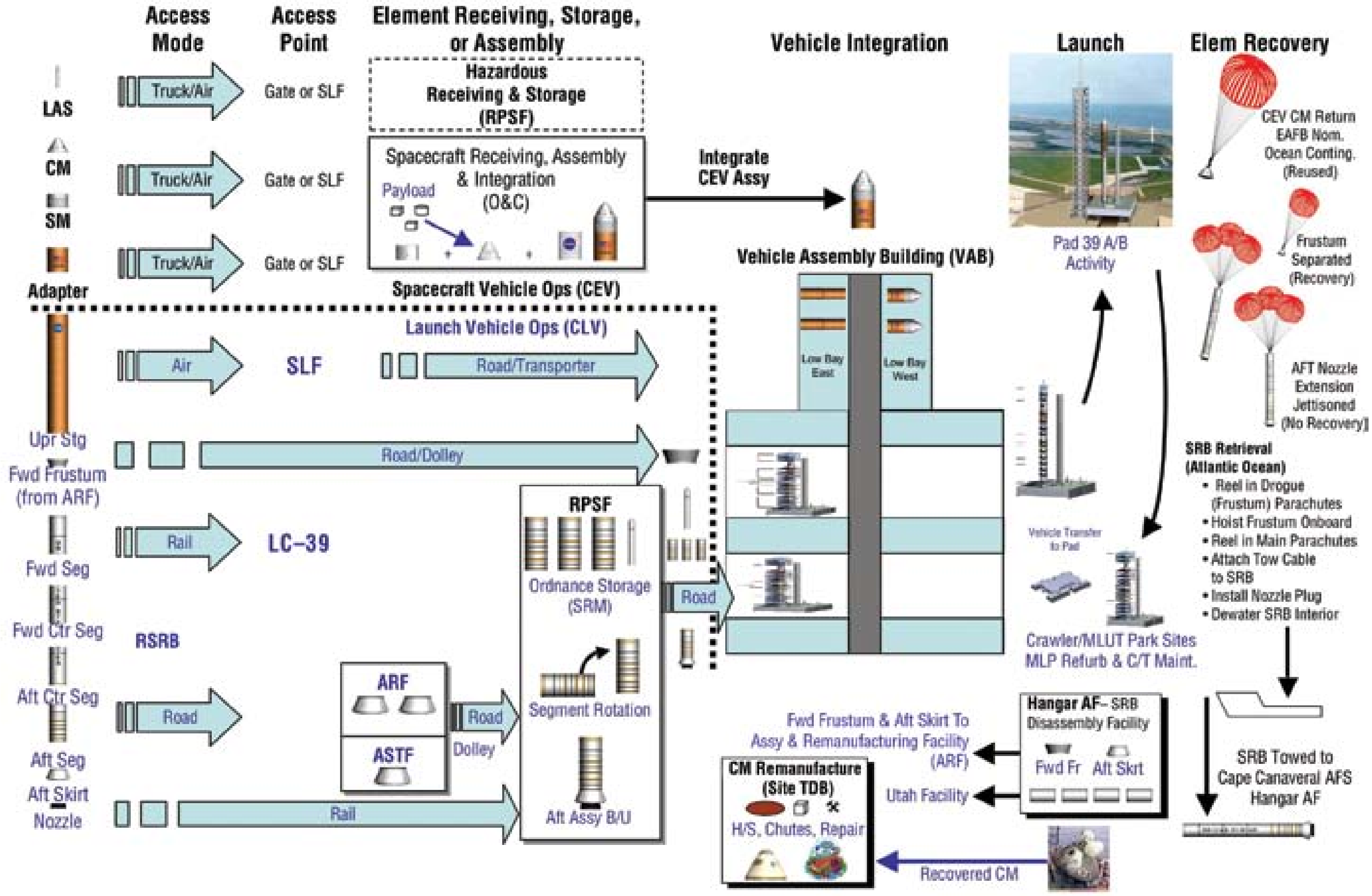


Figure 7-1. Defining the CEV/CLV Operations Concept

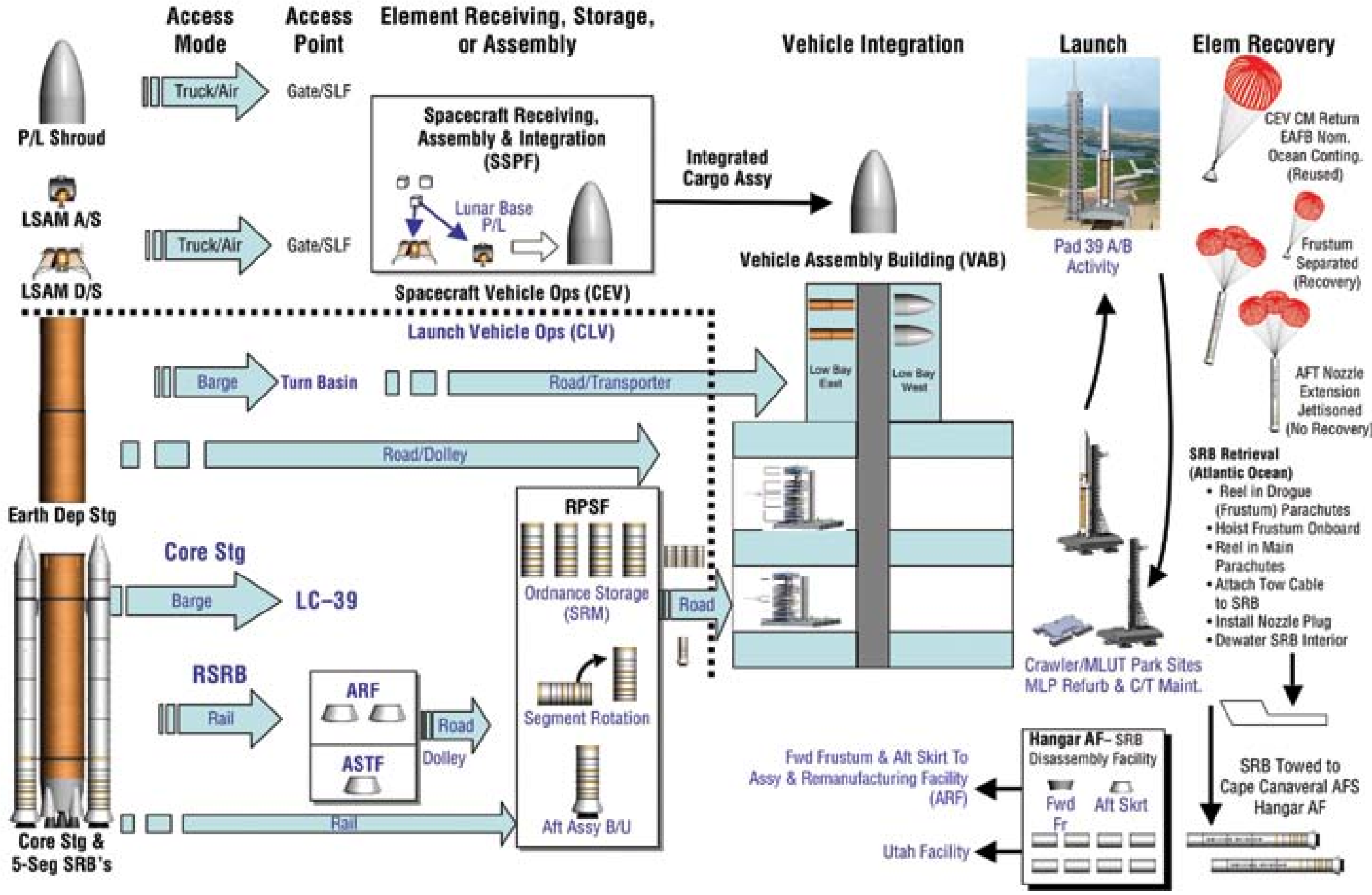


Figure 7-2. Defining the LSAM/CaLV Operations Concept

The association of the generic processing operations (i.e., operations headings along the top of the figures) with specific ESAS flight hardware elements and assemblies under consideration enabled a structured process for defining ESAS-specific tasks, leading to a preliminary operations concept. The fundamental objective of the operations concept is to define a safe and efficient ground-based process that produces routine and safe human space flights for the flight crew and high-value cargo. Creating an efficient operations concept involves working with the vehicle design teams to minimize the number and complexity of flight elements to help reduce the resulting required ground functions and, therefore, avoid the traditional accumulation of ground tasks from the beginning. This part of the study required insight into potential interfaces between the proposed flight hardware elements and the resulting accumulated need for ground facilities, Ground Support Equipment (GSE), and software. Thus, as various LV and spacecraft concepts were assessed, identifying potential ground interfaces was an important task in the ESAS effort.

The importance of spending the time to conduct an analysis of flight-ground interfaces and the trading the resulting system concepts was recognized early in the history of human space flight: "…the von Braun team preached and practiced that rocket and launch pad must be mated on the drawing board, if they were to be compatible at the launching. The new rocket went hand in hand with its launching facility." (Moonport: A History of Apollo Launch Facilities and Operations, NASA SP–4204; Benson & Faherty, 1978).

Analyzing flight-to-ground interfaces produced several design characteristics as important discriminators in the study. The ground interface sensitivities depended on the number of flight elements, the relative complexity of the proposed upper stage, whether or not the current SRB aft skirt interfaces are maintained, and the requirement for local manual access at the pad for SRB Safe and Arm (S&A) device operation prior to launch. The number and type of main propulsion engines and turbo pumps, the engine start cycle, engine operating cycle, and tank arrangements were assessed for resulting upper stage subsystem and ground operations complexity. Recent benchmarking assessments of various engine/propulsion designs by the Government/industry/academia SPST helped to assess relative operational complexity and dependability.

As an example of the importance of this type of LV-to-launch site compatibility analysis, one EELV-derived heavy-lift concept under consideration would have required a flame deflector width and depth so large that even the Pad 39 flame trench would have needed to be greatly enlarged. This would have in turn required a mobile launcher so wide that Vehicle Assembly Building (VAB) high bay dimensions would be called into question and a different crawler transporter would be required to straddle the trench, thus requiring a new crawler way. Upon further study, it was obvious that the LV system concept was incompatible with reasonable ground architecture investments and study constraints.

7.1.2.2.1 Revisiting the "Clean Pad" Concept

The ESAS team had previously conducted assessments of various launch concepts and determined that the integrate-transfer-launch concept, or "mobile launch" concept, was the preferred approach for the Shuttle-derived concepts. What was recommended, however, was a KSC Complex 39 mobile launch concept with less overall accumulated infrastructure to operate and maintain.

7.1.2.2.2 Complex 39 Historical Background

The origin of the mobile launch concept with a clean pad goes back to at least the German rocket team led by Wernher von Braun during the 1930s and the Second World War. Adopted for tactical missile operations, the clean pad design approach has been continually pursued as an objective in the U.S. for larger-scale space flight since the Air Force's Advanced Launch System (ALS) studies of the 1980s. This design approach assumes that prelaunch assembly and servicing of the LV and spacecraft occur away from the launch point and that only propellant loading and final countdown operations are required, without the need for large access and auxiliary service equipment, subsequent to positioning for launch. If a failure requiring intrusive personnel access to the space vehicle occurs, the vehicle is quickly rolled back to its assembly and servicing facility. The design characteristics essential to making this approach work are simple automated vehicle-to-ground interfaces, minimal personnel access requirements at the pad, and dependable flight hardware and launch pad systems. The clean pad mobile launch concept was, in fact, the original design approach for the Apollo-Saturn Launch Complex 39, as shown in **Figure 7-3**. However, due to late design issues associated with Apollo spacecraft servicing, a massive Mobile Service Structure (MSS), shown in **Figure 7-4**, complicated the pad operations and maintenance for Apollo launches. During the course of this study, it was often helpful to the team to refer to Apollo-Saturn launch operations. **Appendix 7F**, **Apollo-Saturn V Processing Flows** is provided as an example of a processing flow of an Apollo-Saturn space vehicle and its countdown.

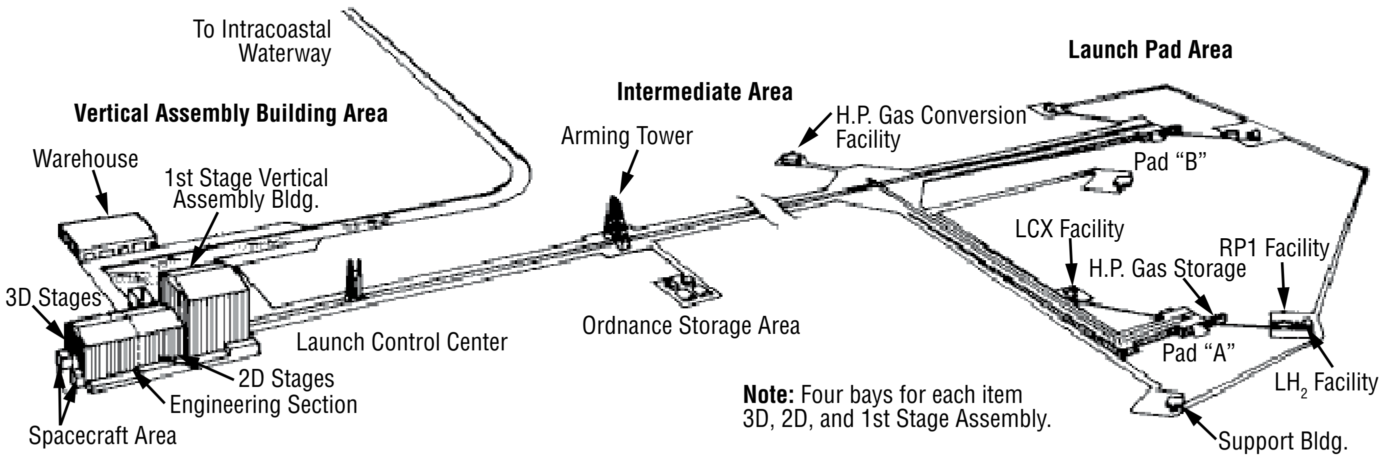


Figure 7-3. Saturn Mobile Launch Concept, July 1961

Figure 7-4. Saturn-Apollo MSS

A second opportunity for NASA to implement the clean pad approach during the conversion of Launch Complex 39 for the Shuttle program was again complicated by late pad access requirements that drove the design of a very complex Rotating Service Structure (RSS) fixed permanently to the apron. Additionally, many of the flight-to-ground propellant service arms and umbilicals were removed from the mobile platform and fixed to a permanent tower on the pad (the Fixed Service Structure (FSS)). The entire ground design began to revert back to the older approach of assembling and mating pad-to-vehicle interfaces at the pad while still maintaining the same VAB infrastructure and an internally complex Mobile Launch Platform (MLP). For example, what had been portable hypergolic servicers during Apollo became permanent equipment fixed within the pad perimeter topped by open canopies. What ultimately emerged at Complex 39 as the vehicle needs became more defined was more than a relocation of infrastructure from the Mobile launcher to the pad—it was simply more total infrastructure.

Since a modern, large-scale clean pad design was successfully implemented on Pad 41 for the USAF/Lockheed Martin Atlas V launch system, the ESAS team concluded that NASA should also return to its original desire to build a clean pad at Complex 39. The chosen concepts use a near-clean pad by building a much less complicated tower for personnel access, lightning protection, and flight crew emergency egress—all relatively passive in design when compared to complex propellant umbilicals and swing-arms. In the ESAS concepts, these services are affixed to a Launch Umbilical Tower (LUT) that is, in turn, attached to and MLP. Not only is the space vehicle assembled indoors, the space vehicle-to-ground interfaces are also mated, checked, and prepared for launch within the protection of the VAB. This approach returns the VAB to its original purpose of serving as an enclosed facility for personnel to prepare a space vehicle prior to a short and highly automated set of actions at the launch pad.

7.1.2.2.3 Reference Architecture Ground Processing Descriptions

Descriptions of the reference ground operations approach for both the CEV/CLV and the LSAM/CaLV are provided below. The cost estimates for launch site labor and facility needs assumed the worst-case, most facility-intensive, work-intensive flows found in **Appendix 7B**, **Concept of Operations and Reference Flows**, while the reference-targeted approach is the more streamlined set of processes depicted in **Figure 7-1** and **Figure 7-2**.

Reference CEV/CLV Processing Description

While the CLV pad concept draws on Saturn-Apollo, much of the ground hardware will come from the Space Shuttle program. Three Space Shuttle MLPs are available to convert to CLV Mobile Launch Umbilical Towers (MLUTs). Additionally, the FSS can be simplified and extended to accommodate the CLV. More detailed design activity must occur before final decisions are made on whether the personnel access and crew emergency egress functions of the FSS should be combined with the umbilical tower and located on the MLUT (as was done during Apollo), or whether program constraints dictate that it should remain fixed to the pad (its current configuration for Shuttle).

The ESAS team has recommended a reference operations concept for the CEV/CLV space vehicle. A reference processing flow is described below for crewed flights and follows the numbered elements in **Figure 7-5**.

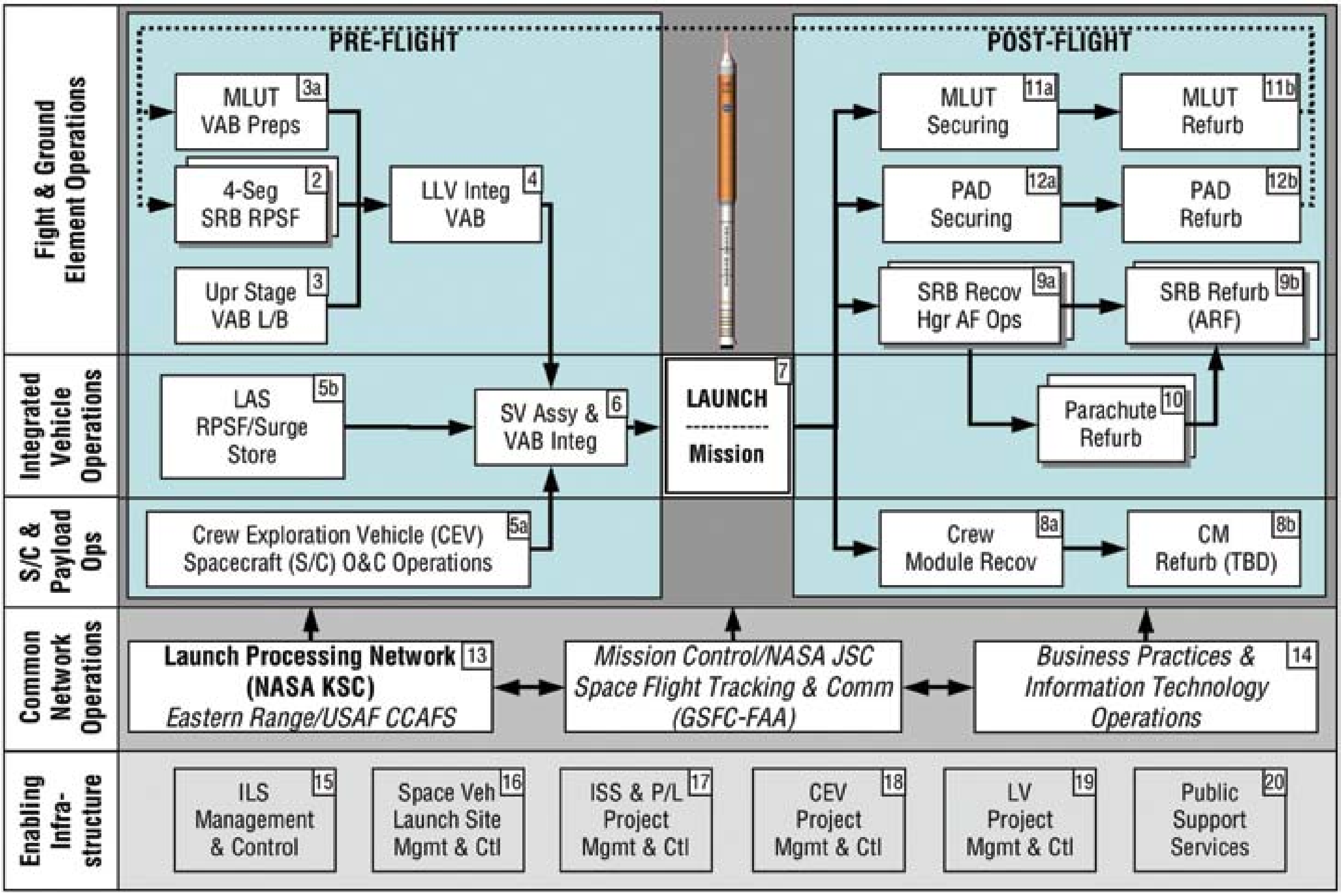


Figure 7-5. CEV/CLV Ground Operations Architecture

Reference Processing Flow for Crewed Flights

1. One of the converted Shuttle MLPs, now a MLUT, is positioned in a VAB Integration Cell (VAB High Bay).
2. A four-segment Solid Rocket Motor (SRM) arrives by rail at the Rotation Processing and Surge Facility (RPSF), along with the refurbished forward frustum, SRB aft skirt, and a new nozzle extension for aft assembly buildup.
3. The CLV upper stage arrives at the VAB Low Bay equipped with a pre-fired single SSME.
4. A single SRB is stacked on the MLUT, followed by the mating of the upper stage. The adapter/forward frustum area is also set up for any necessary purges. The upper stage is mated to a single SRB.

5a. The CEV, composed of a CM and a SM, arrives at KSC's Operations and Checkout (O&C) Building for assembly, testing, and spacecraft integration. The integrated spacecraft assembly is then prepared for transport to the VAB (assuming no toxic propellants or any commodities requiring hazardous SCAPE operations are required—if so, then the spacecraft assembly may have to go through another hazardous facility in KSC's Industrial Area).

5b. The LAS arrives at Complex 39 for temporary ordnance storage (RPSF) until needed for final CLV element mating in the VAB Integration Cell.

6. The space vehicle assembly and integration process mates the CEV spacecraft to the LV and the LAS to the CEV spacecraft to form the integrated CEV/CLV space vehicle. All launch preparations, short of final pad propellant loading and ordnance installation that cannot be performed in the VAB, are completed. Hazardous operations for toxic Reaction Control System (RCS) assembly/servicing (if required) could occur here with facility clear restrictions, but require a thorough hazard analysis to be performed. The CLV space vehicle is then transferred to Pad 39 (A or B).

7. Launch operations include the mating of the MLUT to pad propellant and gas systems (with all flight-to-ground mates having occurred in the VAB). Personnel access for final ordnance hook-ups and flight crew ingress also occurs at the Pad. The clean pad design approach minimizes the amount of work content that occurs at the Pad. Final propellant loading, flight crew ingress, and countdown lead to CLV departure.

8a. The CM is land-recovered (Edwards Air Force Base (EAFB) prime is reference) with two land contingencies and ocean-recovery contingencies for launch and reentry aborts. Any required CM safing for transport to the refurbishment site is also accomplished.

8b. CM refurbishment includes heat-shield removal and replacement, parachute system restoration, post-flight inspections and troubleshooting, and the return of the CM to a launch processing state compatible with spacecraft integration. The location for this function could be KSC's O&C facility or a local off-site facility.

9a. A single four-segment SRB is ocean-recovered and returned to Hangar AF at Cape Canaveral Air Force Station for wash-down and disassembly, as is done with Shuttle SRB recovery.

9b. Disassembled SRB components are refurbished in a similar fashion as the Space Shuttle SRBs. SRM segments are returned to Utah for remanufacturing, while the forward frustum and aft skirt assembly are sent to the Assembly/Remanufacturing Facility (ARF) near Complex 39.

10. SRB parachutes, and possibly the CEV recovery chutes, are sent to the Parachute Refurbishment Facility in KSC's Industrial Area.

11a. Following CLV departure, the MLUT undergoes safety inspections while preparations for post-launch ground crew access are provided.

11b. The MLUT is restored and reserviced in preparation for the next flight.

12a. Following CLV departure, the Pad undergoes safety inspections while preparations for post-launch ground crew access are provided.

12b. Pad systems (kept to a minimum in the clean pad design approach) are restored and propellant systems re-serviced in preparation for the next flight.

Reference LSAM/CaLV Processing Description

A reference processing flow for the lunar cargo launch is described below and follows the numbered elements in **Figure 7-6**.

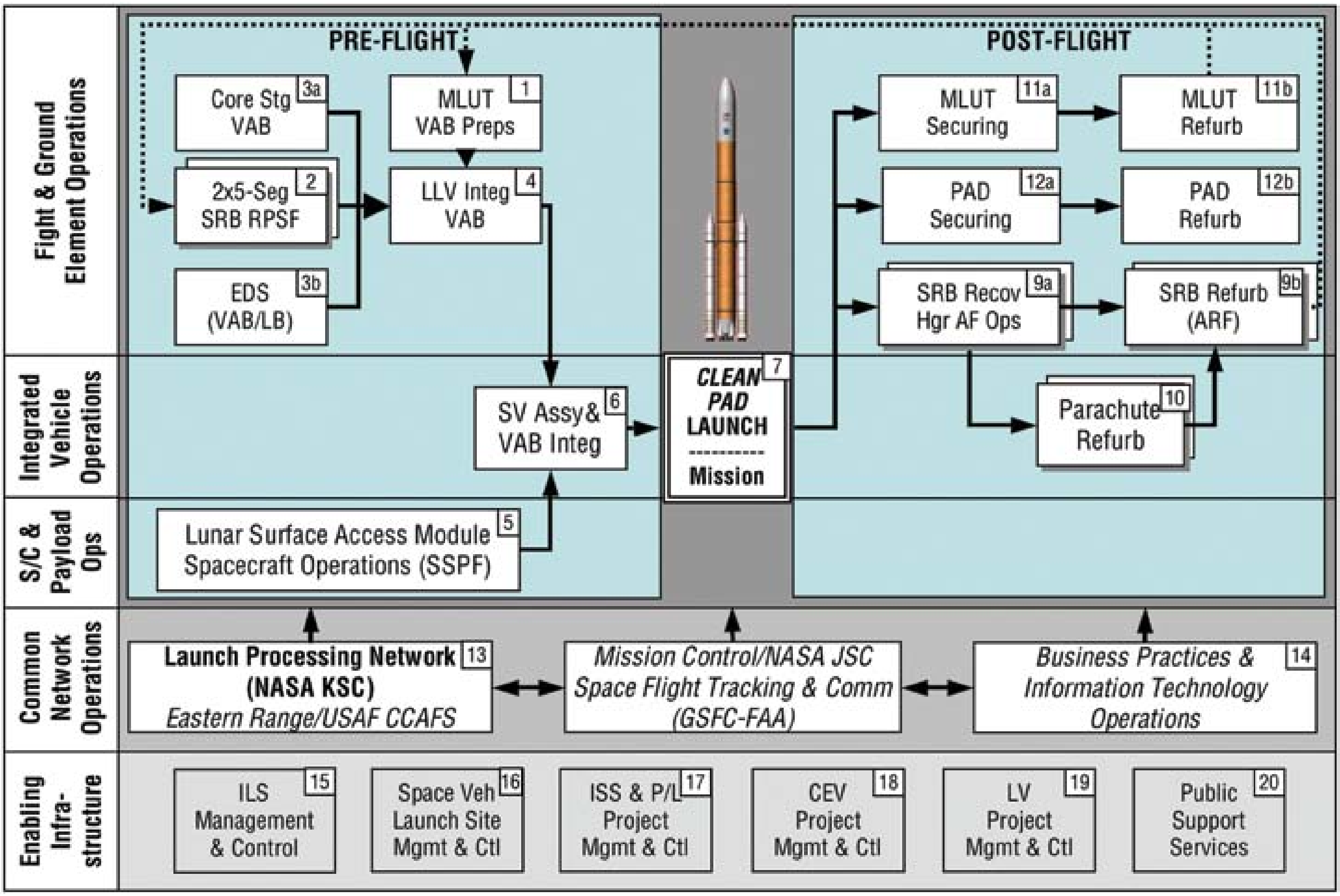


Figure 7-6. LSAM/CaLV Ground Operations Architecture

Reference Processing Flow for Lunar Cargo Launch

1. A newly designed MLUT is positioned in a VAB Integration Cell (VAB High Bays).
2. Two five-segment SRMs arrive at the RPSF along with refurbished SRB aft skirts, nozzles, and forward assemblies for segment rotation and buildup.

3a The core stage arrives in the VAB transfer aisle ready for mating to the MLUT. Five SSMEs arrive having already been built into stage and pre-fired and pre-tested at the factory. No stand-alone engine operations are assumed.

3b. The Earth Departure Stage (EDS) arrives at the VAB's Low Bay equipped with two J–2S multi-start engines. The stage is assumed to arrive with the engine and interstage adapter preassembled.

4. CaLV assembly and integration occurs in a VAB Integration Cell (or High Bay) with LV connections to ground services and integrated checks occurring in the VAB. The SRB stacking occurs and the core stage is mated to the MLUT, followed by the mating of

the EDS/interstage adapter assembly. The LV is mated to ground services through the MLUT in the VAB. The CaLV is assumed to require no hazardous SCAPE operations per the recommended ESAS requirements.

5. The LSAM arrives at KSC's Industrial Area for assembly, testing, and encapsulation in the CaLV forward shroud and is then prepared for transport to the VAB.
6. Space Vehicle Assembly and Integration mates an encapsulated shrouded LSAM spacecraft to the CaLV in the VAB integration cell. All launch preparations except final pad propellant loading and countdown occurs within the confines of the VAB (similar to many Saturn/Apollo tasks and Atlas V Operations).
7. Launch operations include the mating of the MLUT to pad propellant and gas systems through auto-couplers (all flight-to-ground mates having occurred in the VAB). Personnel access for final ordnance hook-ups and flight crew ingress also occur at the Pad. The clean pad design approach minimizes the amount of work content. Final propellant loading and countdown lead to the CaLV departure.
8. [There is no CM processing on the uncrewed CaLV configuration.]

9a. Two five-segment SRBs are ocean-recovered and returned to Hangar AF at Cape Canaveral Air Force Station for wash-down and disassembly.

9b. The disassembled SRB components are refurbished in a similar fashion as the Space Shuttle SRBs. The SRM segments are returned to Utah for remanufacturing, while the forward and aft skirt assemblies are sent to the ARF near Complex 39.

10. SRB parachutes are sent to the Parachute Refurbishment Facility in KSC's Industrial Area.

11a. Following CaLV departure, the MLUT undergoes safety inspections while preparations for post-launch ground crew access are provided.

11b. The MLUT is restored and re-serviced in preparation for the next flight.

12a. Following CaLV departure, the Pad undergoes safety inspections while preparations for post-launch ground crew access are provided.

12b. Pad systems (kept to a minimum in the clean pad design approach) are restored and propellant systems reserviced in preparation for the next flight.

7.1.3 Launch Facility and Equipment Conversions

7.1.3.1 Development Schedule and Flight Test Manifest Assumptions

A preliminary analysis by the ESAS team provided a concept for conversion of the pads and MLPs to support the flight test program. Little or no modification is required for the mobile launcher for the first LC–39 flight (Risk Reduction Flight-1 (RRF–1)) because the flight configuration is largely composed of mass simulators and flown without crew. Therefore, assembly and integration can occur within the VAB, and pad personnel access can be confined to SRB S&A operations provided by mobile heavy equipment access. Little or no tear-down of the current pad (currently envisioned to be Pad B due to timing of Pad B long-term refurbishment) FSS and RSS are required to support this test flight. Some modifications are required for the mobile launcher for the second and third flights (Risk Reduction Flight-2 (RRF–2) and Risk Reduction Flight-3 (RRF–3) because servicing of the CLV upper stage is required. Assembly and integration occurs within the VAB and pad personnel access is still confined to SRB S&A operations provided by mobile heavy equipment access. Tear-down of the current pad RSS occurs after RRF–1 and after extensions to the FSS with associated personnel access provisions have been installed and activated. Additionally, the MLUT and associated systems are installed and routed throughout the MLUT internal structure while auto-couplers between the MLUT and pad are installed and certified for use. The final CEV/CLV space vehicle flight-ground system launch configuration accommodates both unmanned crew and cargo flights from Launch Complex 39. (see **Figure 7-7**.)

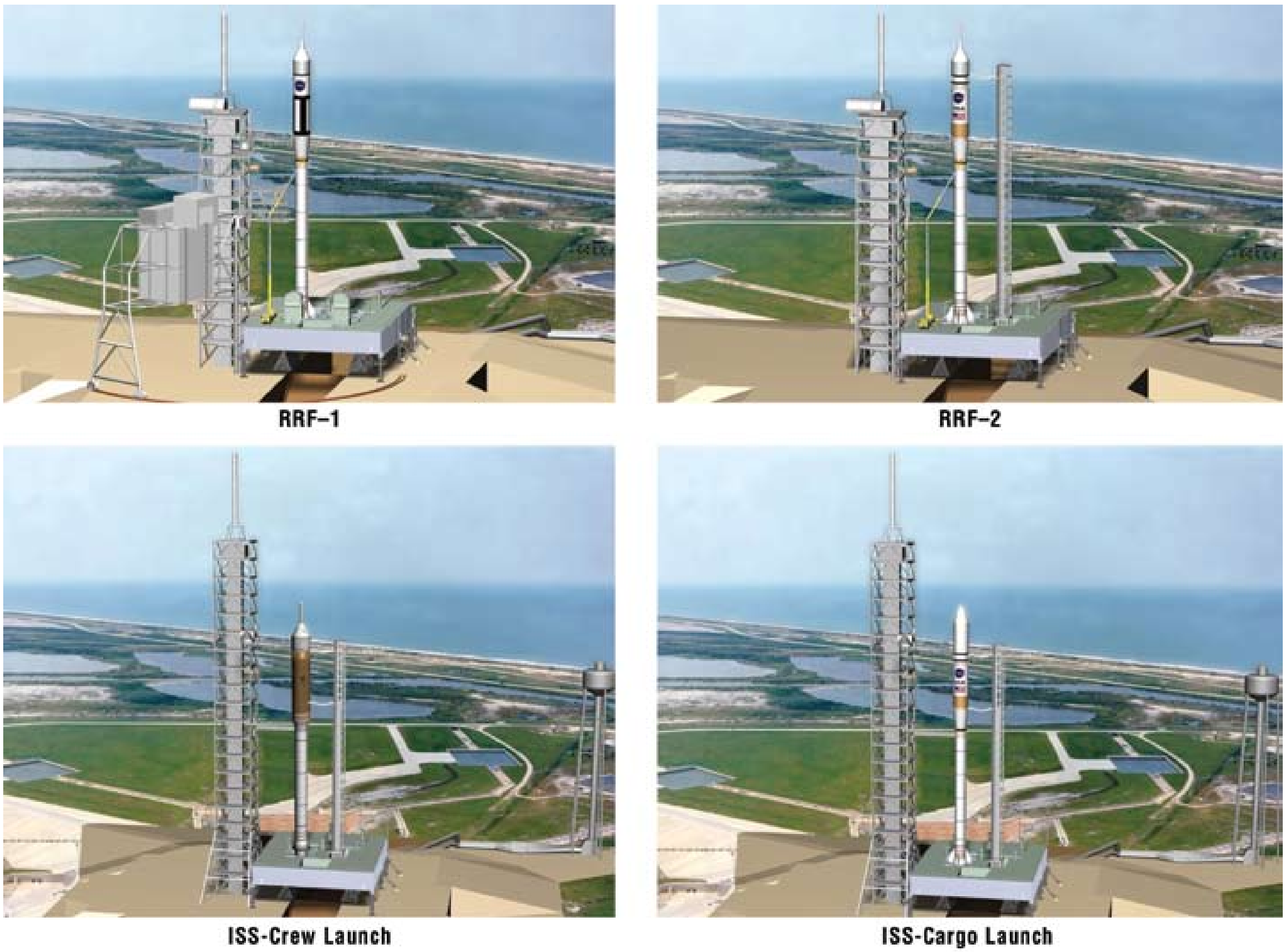


Figure 7-7. Concept for CLV Pad Conversions through ISS Operation

A reference launch pad transition scenario envisions first launching the test and evaluation flights from Pad B (since it is the next pad scheduled to go off-line for long-term maintenance) with personnel access provided from the current FSS, most fluid and electrical services transferred to the converted Shuttle MLUT, and the RSS removed. This would be an interim step toward an LC–39 clean pad and would allow the current Space Shuttle to continue departures to the ISS from Pad A. For the lunar missions, Pad A would be "stripped down" to accept the new mobile launcher design, which would now have a more functional umbilical/access tower—similar in function to the Apollo-Saturn launcher. The method of crew emergency egress for such an approach has several design alternatives to be determined. This has been a classical launch site design issue for human space flight. Pad B could then be reconfigured to the new clean pad design at Pad A. In summary, the reference concept is to have two clean pads (39A and 39B) that can eventually accommodate both the CLV and CaLV configurations. It was not clear within the ESAS time frame for analyzing the 1.5-launch solution whether a universal mobile launcher design could accommodate both the CLV and CaLV. Cost estimates assumed that three converted Shuttle MLPs for the CLVs and two new mobile launchers for the CaLVs (a total of five mobile launchers) would be required for the program.

7.1.3.2 Spacecraft Processing Facility Conversions

7.1.3.2.1 CEV Architecture and Mission

The CEV architecture consists of three primary flight elements: CM, SM, and LAS. For spacecraft ground processing, the CM is assumed to be reusable, provided to the spacecraft assembly and integration area in a safe, nonhazardous condition that requires no propellant or ordnance handling in the processing facility.

The SM is assumed to be expendable and does not require hypergolic propellant servicing in the reference case. (Hazardous hypergolic propellant servicing was explored as a contingency during spacecraft integration prior to delivery for LV integration. The safety implications of this scenario, while accounted for in the cost estimates, will require a more thorough hazard and safety analysis). The LAS is expendable and arrives with solid fuel and is most likely stored in Complex 39 near the SRB segments. There is also the possibility of storing and mating as part of the Industrial Area spacecraft integration process, but this also requires a thorough safety and handling analysis.

The assumed Industrial Area processing analysis supports an annual launch rate of six CEVs with two crew missions, three pressurized cargo missions, and one unpressurized cargo mission per year on 2-month centers. It is also assumed that crew access is required at the pad.

7.1.3.2.2 CEV Infrastructure Assumptions

The CM and SM are assumed to be assembled, serviced, and integrated in a nonhazardous and/or hazardous facility, as required to support the CEV flight element design. Parallel clean work areas are required for processing a minimum of four CEV systems. Vacuum chamber testing may be required for CMs during flow.

The SM may require fueling in a hazardous facility. The LAS will be stored and processed in a hazardous ordnance facility. The worst case CEV spacecraft integration infrastructure assumes assembly of the CM, SM, and LAS performed in hazardous facilities. Integration with the CLV is performed in the VAB High Bay. Although crew access and emergency egress is required at the pad, propellant loading or servicing is not.

7.1.3.2.3 Operations and Checkout (O&C) Building Modifications and Development

The ESAS reference ground processing architecture assumes the O&C Building at KSC's Industrial Area is modified for CEV element processing and integration. (See **Figure 7-8**.) The final spacecraft element integration facility will be largely determined by the nature of the spacecraft subsystems and their hazardous processing requirements.

The O&C facility concept incorporates an open-floor design that is compatible with mobile Ground Support Equipment (GSE) to support CEV and/or other spacecraft hardware. The processing concept also incorporates standard services including compressed air, power, gases, vents, and instrumentation, and others as required to support the CEV flight systems. The concept also upgrades cranes to support the new program. It is envisioned that development of new common GSE for CEV processes and ISS interface testing will occur. This will also allow incorporation of state-of-the-art technology developments for fluids, avionics, and mechanical GSE. If required, the O&C vacuum environmental chambers will be verified for compatibility with program needs.

Figure 7-8. CEV Footprint in KSC's O&C Building

7.1.3.2.4 Space Station Processing Facility (SSPF) Modifications and Development

No facility modifications are planned for the initial CEV/CLV program. New GSE development for ISS interface testing may be performed. This SSPF is envisioned to continue support of the ISS program. Assets may transition over to support the LSAM as flight hardware and prototypes arrive at the launch site.

7.1.3.2.5 Vertical Processing Facility (VPF) Modifications and Development

The Vertical Processing Facility (VPF) is envisioned to be modified to support the CEV program if off-line hazardous integration and fueling of the CEV is performed in the Industrial Area (as opposed to launch pad servicing). This will require removal of platforms and other fixed GSE; incorporation of open floors compatible with mobile GSE to support CEV and/or other spacecraft; incorporation of standard services including compressed air, power, gases, vents, instrumentation and other services; accommodations made for CEV and/or other spacecraft fueling and ordnance installation; incorporation of technology development for fluids, avionics, and mechanical GSE; and development and acquisition of common GSE for CEV hazardous processing.

7.1.3.2.6 LSAM/SSPF Modifications and Development

Use of the SSPF is the ESAS reference concept for LSAM processing. The concept expands the work area from 8 to 12 footprints, includes a canister operations area and adds cranes and a new airlock, and develops new GSE for lunar spacecraft checkout and integration.

Since the LSAM is launched on a separate EDS that does not require a long-length shroud, it may be possible to use the SSPF or extend it to the east to perform final LSAM encapsulation in the payload fairing of a Heavy-Lift Vehicle (HLV). This would be sent as an integrated spacecraft package from KSC's Industrial Area to Launch Complex 39 for integration with the Heavy-Lift Launch Vehicle (HLLV). Precedence for this is found in the Skylab program.

7.1.3.3 CLV Facility/Equipment Conversions

7.1.3.3.1 CLV Architecture

The CLV architecture consists of a single four-segment Reusable Solid Rocket Booster (RSRB) first stage and a LOX/LH2 upper stage with a single SSME that is modified for altitude-start. The SRB is assumed to be reusable in the same manner as the Space Shuttle

Program. The CLV Upper Stage is expendable—with the SSME pre-fired, assembled, and checked-out prior to delivery and launch site acceptance. The launch rate is assumed to be six per year on 2-month centers.

7.1.3.3.2 CLV Infrastructure Assumptions

No additional processing areas, facilities, or GSE are assumed to be required for SRB operations. Some accommodation changes are expected due to the differences in forward frustum design. Launch Complex 39 RPSF operations are retained. Hangar AF SRB retrieval operations are retained and hazardous hydrazine de-servicing operations are maintained for the near-term only. The upper stage is assumed to require minimal processing in the VAB Low Bay where element receiving and acceptance and prestacking operations occur. Minimal infrastructure modifications are required. If toxic/hazardous RCS are employed on the CLV, it is assumed that those elements arrive preloaded. The hazardous, toxic propellant loading may be performed off-site and assembled onto the appropriate stage in the VAB (pending safety and hazard analysis). No hypergolic propellant loading system is envisioned in the VAB, on the mobile launcher, or at the pad—nor was such a system budgeted for in the ESAS ground architecture. It is also assumed that no clean room is required for upper stage processing or payload encapsulation at the VAB.

Three MLPs and two crawler-transporters are required to maintain the ISS tempo of six flights per year with periodic long-term restoration downtime required for these ground elements. The VAB is assumed to perform stacking and payload integration. For this capability, two VAB high bays (High Bays 1 and 3) are required for integration and are modified to support access and vehicle servicing and assembly requirements. The current Quantity Distance (QD) restriction of 16 SRB segments in the VAB applies, although ESAS has initiated a NASA reassessment of this requirement. The current 16-segment restriction is not believed to be a major restriction with the ESAS 1.5-launch solution for the two lunar mission per year rate. Two launch pads are assumed to be required, with crew access and emergency egress required at the pad. Main propellant loading is performed at Pad 39. No toxic or hazardous hypergolic loading or servicing is required at Pad 39. SCAPE operations are not envisioned in the ESAS reference concept at NASA's Complex 39.

The study assumes that an Apollo-like MSS is not necessary for the ESAS concepts if the nontoxic requirement is adhered to, or if toxic systems employed can avoid on-pad loading and servicing from the opposite side of the umbilical tower. The study also assumes that the CEV design and location of pyrotechnic arming locations can avoid the problems encountered in the Apollo Program by likewise locating pyrotechnic arming, or other functions requiring late manual access at the Pad, via crew access arms or from the base level of the mobile launcher (Moonport: A History of Apollo Launch Facilities and Operations, Chapter 13 "From Arming Tower to Mobile Service Structure," NASA SP-4204; Benson & Faherty, 1978).

An example of ground system design trades for modification of Launch Complex 39 is shown in **Table 7-1**. The objective of the concept trades is to determine how to accommodate fluid and propellant services and how best to provide personnel access.

	Design Options					'Goals'
Requirements	**Mod FSS**	**Mod MLP**	**New MLP**	**Hybrid**	**Hybrid**	**Clean Pad**
Lightning Protection	FSS	Faraday	Faraday	Faraday	Faraday	LUT
Emergency egress	Slidewire	Track	Chute	Slidewire	Chute	Safe/Low Ops
Crew access OAA	FSS	LUT	LUT	FSS	FSS	LUT
LOX/LH2 servicing	FSS	LUT	LUT	LUT	LUT	LUT
SRB S&A Access	FSS	LUT	LUT	FSS	LUT	LUT
MLP Structure	STS	Mod STS	New	Mod STS	New	New
VAB Changes	Low	High	High	Low	Medium	-
Schedule Availability	Late	Late	Late	RRF-1	Con-1	Con-1
Cost	Medium	Medium	Medium	High	Medium	Affordable

Table 7-1. Example Pad/MLP/VAB System Trades

The current Space Shuttle configuration of Pad 39 provides fixed services at the launch pad, with a mobile platform carrying the space vehicle to the pad to connect up to those services. As previously mentioned in the clean pad approach, both services and access can be attached to the mobile platform, as was done with the Apollo-Saturn configuration.

The ESAS reference is currently a hybrid approach, where the services are provided on a MLUT with minimal personnel access at the pad provided through a modification and extension of the current Space Shuttle FSS. The Shuttle pad current RSS is no longer required because late-payload assembly and integration to a side-mounted space vehicle will not occur at the pad, and hypergolic servicing of side-mounted RCS will not be required.

Elevated vehicle personnel access at the pad is assumed for SRB Safe and Arm (S&A) operations, as well as for late flight crew stowage, flight crew ingress, and emergency flight crew egress. Ground services to the CEV/CLV space vehicle include hardwire command and data paths, electrical power, nitrogen purging of the aft skirt (if hydrazine is maintained on the SRB Thrust Vector Control (TVC) system), cryogenic loading, upper stage and interstage conditioning and purging, and spacecraft propellant and gas system servicing. Some of the services may be performed in a local/manual mode in the VAB or in a remote/manual or remote/automated mode from a control center.

Lightning protection for the space vehicle will also be accommodated at the pad. Several alternative design concepts for this function are available, including a tall lightning mast mounted on the LUT or a permanently pad-mounted system of "Faraday Cage" towers.

Each of these trades must be integrated to complete the ground architecture, including the VAB, mobile launcher, and pad system designs. The availability of Space Shuttle MLPs, pads, and VAB Integration Cells (high bays) are factored into the initial ISS CEV/CLV space vehicle ground support architecture and related costs.

In order to perform the space vehicle integration for the CEV/CLV, the VAB's extensible platforms will require modification. These large facilities translate in and out to provide access for personnel to perform on-vehicle assembly, any local-manual flight element servicing operations, and final closeout activities prior to space vehicle rollout.

The ESAS team provided a conceptual design for the redesign of these VAB extensible access platforms as shown in **Figure 7-9**.

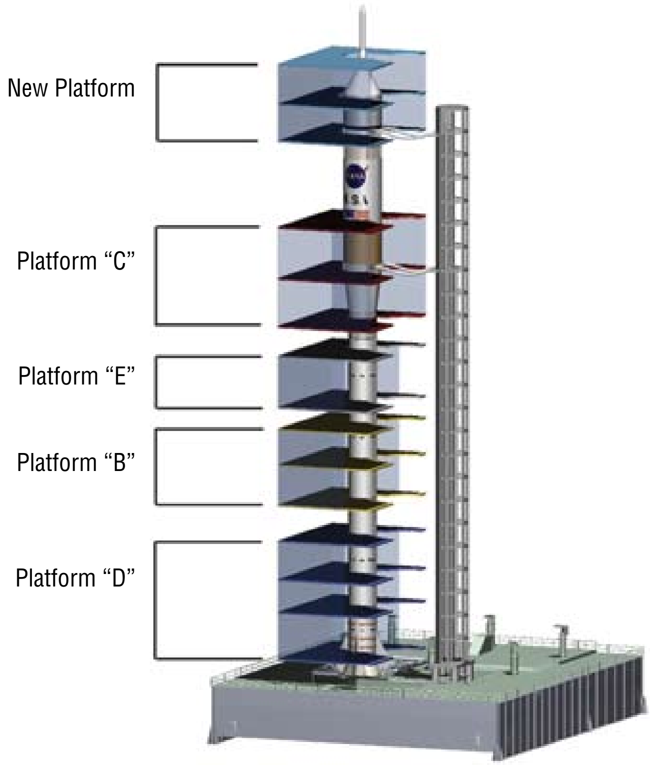


Figure 7-9. Vehicle Assembly Building High Bay Extensible Platform Concept

7.1.3.4 CaLV Facility/Equipment Conversions

7.1.3.4.1 CaLV Architecture

The CaLV architecture consists of two five-segment SRBs, a liquid core stage with five SSMEs, and an EDS with two J–2S multi-start engines. The SRBs are assumed to be reusable in the same manner as the Space Shuttle Program and the CLV. The launch rate is assumed to be two CaLV’s per year on 6-month centers.

7.1.3.4.2 CaLV Infrastructure Assumptions

No additional processing areas, facilities, or GSE are assumed to be required for five-segment SRB operations. Some minor changes are expected due to the five-segment, versus four-segment, configuration. Launch Complex 39 RPSF operations are retained. Hangar AF SRB retrieval operations are retained, and hazardous hydrazine deservicing operations are maintained for the near-term only. The EDS is assumed to require minimal processing in the VAB Low Bay where element receiving and acceptance and prestacking operations occur. Minimal infrastructure modifications are required. No toxic/hazardous RCS are assumed to be employed on the CaLV. The hazardous, toxic propellant loading may be performed off-site and assembled onto the appropriate stage in the VAB (pending safety and hazard analysis). No hypergolic propellant loading system is envisioned in the VAB, on the mobile launcher, or at the pad. Nor was such a system budgeted for in the ESAS ground architecture. It is also assumed that no clean room is required for EDS processing or payload encapsulation at the VAB.

Two new MLUTs and two new crawler-transporters are required to maintain a lunar campaign tempo of two flights per year with periodic long-term restoration downtime required for these ground elements. The VAB is assumed to perform stacking and encapsulated payload integration. For this capability, one VAB high bay is required for integration and is modified to support access and vehicle servicing and assembly requirements. The current QD restriction of 16 SRB segments in the VAB applies, although ESAS has initiated a NASA reassessment of this requirement. The current 16-segment restriction is not believed to be a major restriction, even with one five-segment pair for the CaLV and a single-four segment CLV in stacking to support the two lunar mission-per-year rate. Two CLV/CaLV launch pads (39-A and 39-B) are assumed to be required. Main propellant loading is performed at Pad 39.

In order to accommodate the space vehicle integration for the LSAM/CaLV, one of the VAB's extensible platforms will require modification. These large facilities translate in and out to provide access for personnel to perform on-vehicle assembly, any local-manual flight element servicing operations, and final closeout activities prior to space vehicle rollout.

7.1.4 Cost Estimation Approach

For the cost estimation approach and results, reference **Section 12**, **Cost**.

7.1.5 Special Topics

7.1.5.1 Design Process Controls to Manage Inherent Complexity and Dependability

The ESAS team, working under a tight architecture definition time constraint, diligently worked to alleviate the level of ground operations work. This was done by searching for space vehicle configurations with the least number of practical stages, the fewest number of engines, and the fewest number of different engines where practical. During the ESAS effort, specific design characteristics, such as the toxicity of fluids, the number of different fluids, and the number of separate subsystems were used by the team. (Reference **Appendix 7C**, **ESAS Operability Drivers**.)

In order to contain the ground operations costs (both development and recurring operations and support), it is vital that these design characteristics be quantified and baselined at the start of the program. These parameters should then be managed through a tracking system that includes a means of surfacing deviations from the baseline to high-level program management and NASA independent program assessment. If no constraints other than on weight and performance are applied to the design process, the ground operations costs will be difficult to control.

7.1.5.2 Integrated Logistics Support and Affordable Supply Chain

For the integrated logistics support and affordable supply chain analysis, reference **Section 12**, **Cost**.

7.1.5.3 Reuse of SRB and CM

7.1.5.3.1 SRB Reuse Opportunities

The ESAS effort determined that the reuse of the SRBs is more economically viable than continually producing the segments, aft skirts, etc. (Reference **Section 12**, **Cost**). The ground operations processes for SRB reuse are envisioned to follow the same process and have the same infrastructure as that of the Space Shuttle program. This section describes some architectural options for consideration to improve Life Cycle Cost (LCC) and the throughput performance capability.

The SRB's TVC system involves two major subsystems: a 3,000-psi hydraulic actuation subsystem and a toxic hydrazine Hydraulic Power Unit (HPU) that converts the stored chemical energy in the hydrazine monopropellant to mechanical shaft power for a hydraulic pump by means of a catalytic bed that generates hot-gas exhaust expanded through a turbine drive. (See **Figure 7-10**.)

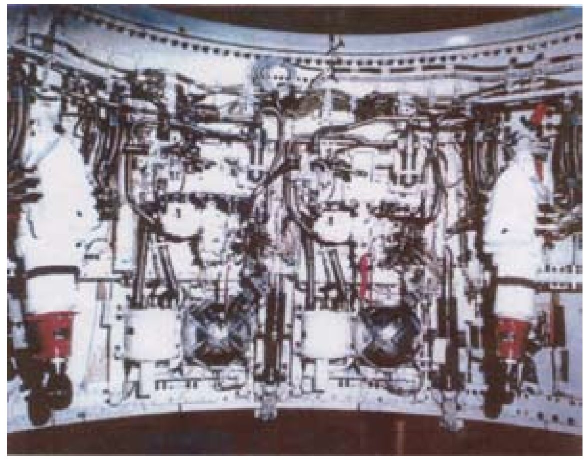


Figure 7-10. SRB Aft Skirt Thrust Vector Control Installation

The toxic hydrazine loaded into the SRB aft skirts drives expensive, hazardous, and time-consuming SCAPE operations in a number of areas across the nation. A dedicated aft skirt safing and disassembly facility is required at Cape Canaveral's Hangar AF SRB Disassembly Area. After the nearly year-long process of remanufacturing the aft skirt components and aft skirt resurfacing, inspections, and checks at NASA MSFC's ARF located within the KSC property, the completed aft skirt is transferred to another dedicated facility to hot-fire test the hydrazine-powered assembly, where more SCAPE operations occur. The system is then drained of the toxic hydrazine, safed, secured, and prepared for delivery to KSC's RPSF for SRB aft booster assembly. Final hydrazine loading of the SRB TVC system occurs at the launch pad just prior to launch. As with all SCAPE operations, this is a hazardous operation under local manual control with remote monitoring of the operations with fire and medical rescue support available. Hazardous operations are also repeated at the manufacturing site for motor firings involving the TVC that may occur. For a more detailed description of the process, reference **Appendix 7D**, **Toxic, Hazardous Operations Impacts**.

Some alternatives should be explored to eliminate the SCAPE operations hazards. Much preliminary engineering has already been accomplished on these alternatives. For example, NASA conducted a multi-center Electric Actuation Technology Bridging Study during the early 1990s that looked at all-electric solutions involving replacing the distributed hydraulic actuators with either Electro-Mechanical Actuators (EMAs) or battery-powered self-contained Electro-Hydrostatic Actuators (EHAs). Replacement of the similar hydrazine-powered APU on the Orbiter with an electric actuator and the SRB HPU with a high-pressure cold-gas blow-down system was also engineered in the late 1990s. Both of these nontoxic APU/HPU solutions should be resurrected for consideration in the CLV, and the self-contained hydraulic actuator should be considered for the five-segment solid qualification program.

7.1.5.3.2 Crew Module Reuse Considerations

Economic analysis by the ESAS team indicates that NASA baseline reuse of the CM. (Reference **Section 12**, **Cost**.) NASA KSC experience in reuse of space flight hardware shows that prediction of unplanned work levels, resulting direct costs and infrastructure support costs, and prediction of reuse turnaround times are very difficult and require broad uncertainty bands around such predictions. The reality is that NASA will not know the true outcome until full-scale, fully functional hardware systems, subsystems, and parts go through the ascent, on-orbit, entry, landing and recovery, and ground remanufacturing environments.

During the design phase, it is important for the end-items to be specified with design life parameters that are appropriate for the above environments and for the individual components, rather than assuming a design life commensurate with the airframe (as was done with the Space Shuttle). When allocating the design life, the amount of ground power-on time, power-on/off cycles, tank/system pressurization cycles, and so forth are highly important in containing the level of unplanned work (or even planned work in the case of limited life items).

A reusable CM needs known structural margins designed in and verified before delivery to avoid intrusive and time-consuming structural inspections. This means that Structural Test Articles (STAs) should be loaded and tested to destruction, as is done with reusable aerospace vehicles. The key to eliminating unwanted tests and inspections is to gain engineering confidence by investing in such tests and test articles. This approach has been assumed in the ESAS cost estimation results (**Section 12**, **Cost**).

Additionally, the CEV program should invest early in a thorough, first-class Maintenance Engineering Analysis (MEA) that is factored into the overall system and subsystem designs and again into the end-item specification process—with a design certification and buy-off process before delivery. In order to meet the first CEV deliveries, there will be a tendency for the manufacturer to push for completion of final assembly and other “minor” manufacturing tasks that tend to accumulate at the launch site. Experience has proven during human space flight programs (particularly the Shuttle Orbiter) that this tendency destroys launch site flow planning and creates net wasted time and effort. Adherence to a clear contractual final delivery plan is a must to meet the CEV/CLV test and evaluation objectives and schedules.

Provided below is a quick overview of the type of work that may be required for CM refurbishment. The work tasks below assume the ESAS reference CEV with accommodations for up to a crew of six. It further assumes that the CM has already been recovered (land or ocean) and arrives at a refurbishment/remanufacturing facility.

Top-level refurbishment categories may include the following:

- Facility/Equipment Preps and Setups for CM Refurbishment;
- CM Handling and Positioning, Connection to Services, Gaining Access, and Protection;
- CM Post-Flight Safing;
- CM Post-Flight Inspections and Servicing;
- CM De-servicing;
- CM Unplanned Troubleshooting and Repair;
- CM Modifications and Special Tests;
- CM Reconfiguration;

- Closeout for CM Delivery/Turnover; and
- CM Refurbishment Facility and Equipment Periodic Maintenance.

A more detailed list of tasks is shown in **Table 7-2**.

Table 7-2. CM Refurbishment Work Tasks

CM Refurbishment and Recertification for Launch Processing	
Facility Preps for CM Refurbishment	Functional verification of ground systems prior to vehicle arrival
	Servicing and staging of ground systems prior to vehicle arrival
	Contamination control preps and setups
CM Handling and Positioning, Connection to Services, Gaining Access, and Protection	CM transport and alignment in refurbishing stands
	Ground access kit positioning
	Connection to facility electrical, data, fluid, and gas services
	Establish protective enclosures and install/remove flight vehicle protective covers
	Removal of external flight equipment covers/panels to gain access
	Open CM hatches
CM Post-Flight Safing	Hazardous fluid and pyro/ordnance safing
	Establish CM purges for personnel/CM safety
CM Systems Deservicing	Landing bag system rework
	Parachute recovery system rework
	Other CM mechanical systems recalibration, lubrications, etc.
	Routine replacement of environmental control filters and window cavity purge desiccants, etc.
	Flight crew systems deservicing
	Passive thermal protection routine replacement (known limited life)
	Thermal Protection System (TPS) seal replacements as required
	Fluid drain and deservicing
	Application of blanket pressures for transport and delivery, if required
	Navigation and instrumentation component servicing and calibrations
	Routine replacement of expendable and limited life CM components
	Application of CM electrical power and avionics systems health monitoring
CM Post-Flight Inspections and Checkout	CM structural integrity inspections and mechanism functional verifications
	Propellant, fluids, and gas system leak checks, functionals, and inspections
	CM powerup, switch lists, functional checks, and onboard software updates and checks
CM Mission Reconfiguration	Reconfigure TPS and chutes
	Remove unique mechanisms (e.g., seats, cargo restraints, etc.) from previous mission/Install unique mechanisms for upcoming mission and recertify for delivery
	CM pressurized cabin locker/stowage area, displays and controls, etc., reconfiguring, cleaning, and recertification for delivery

CM Refurbishment and Recertification for Launch Processing	
CM Unplanned Troubleshooting and Repair	CM Line Replaceable Unit (LRU) troubleshooting, replacement, disposition of failed and suspected failed components
	Troubleshooting and repair of leaks
	Fluid/pneumatic system decontamination and cleaning
	Electrical cable and connector troubleshooting, retest, and repair
	Unplanned troubleshooting or replacement of TPS hardware (not routine replacements)
	Unplanned structural repair/refurbishment
	Repair of ducts, tubes, hoses, mechanisms, and thermal/pressure seals
	Troubleshooting and repair of ground support equipment
CM Modifications and Special Tests	Flight equipment modifications/upgrades and mandatory Material and Process (M&P) changes
	Special tests, fleet system, and component cannibalizations
Closeout for CM Delivery and/or Further Spacecraft Integration	Removal of ground services, umbilicals, and personnel access equipment
	CM system closeouts for SM/spacecraft integration
	Final CM cleaning and preps for delivery
CM Refurbishing Facility and Equipment Periodic Maintenance	Interval maintenance of CM refurbishment equipment
	CM refurbishment facility and system modifications and M&P changes

Table 7-2. CM Refurbishment Work Tasks (Continued)

Segregation of the normal CM preflight preparations (i.e., CEV spacecraft assembly and integration and servicing) from the CM refurbishment function should be considered. To better ensure that design corrective action occurs for unplanned, nuisance, and high-maintenance surfaces during the CEV test and evaluation period, it is recommended that the design center initially take responsibility for the refurbishment process similar to the way NASA Marchall Space Flight Center (MSFC) controls the SRB process locally in the ARF at the Cape. That function, however, should be collocated with launch operations resources. Once refurbishment operations and design modifications to improve component dependability have stabilized, a smooth transition of the refurbishment function to the launch operations center can occur. It is also recommended to provide the CM development contractor with incentives during the CEV acquisition process to activate an aggressive maintainability-by-design corrective action process that demonstrates to NASA the maximum benefit that can be obtained by CM reuse by the end of the flight test program.

7.1.5.4 Command and Control (C&C) Concepts

7.1.5.4.1 Background

Unique stovepipe Command and Control (C&C) systems are traditionally built for the various ground operational elements within a program. This approach leads to the proliferation of independent systems with duplicative functionality, logistics requirements, and multiple sustaining engineering organizations. Development, operations, and maintenance of these individual systems both complicate operations and result in high LCCs.

7.1.5.4.2 Goals and Objectives

The primary goal of the ground C&C concept is to take advantage of the commonality that exists across ground C&C systems and to reduce program cost of ownership through large-scale reuse of software across Constellation ground operational systems.

7.1.5.4.3 Technical Approach

Software and hardware product lines are rapidly emerging in the commercial marketplace as viable and important development paradigms that allow companies to realize order-of-magnitude improvements in time-to-market, cost, productivity, quality, and other business drivers. A software and hardware product line is a set of software-intensive systems that share a common, managed set of features that satisfy the mission needs and are developed from a common set of core assets.

Constellation C&C Product Line

The technical approach is based on the application of the product line strategy to program ground operations C&C systems. Rather than stovepipe C&C systems, the approach focuses on developing a family of related systems. The Constellation C&C product line is intended to provide the foundation for the system family capable of supporting vehicle/spacecraft integration, launch site processing, and mission operations.

The context of the product line is characterized at a high level in **Figure 7-11**, where the external interfaces are described in the green boxes, operations are described in the blue boxes, and the product line systems are described in the circle. The connecting lines represent the major external interfaces for the product line systems. The product line systems provide those control and monitoring capabilities for the LVs, spacecraft, and GSE necessary to support program operational needs. The capabilities of the product line systems include services for real-time data visualization, data processing, data archival and retrieval/analysis, end-item simulation, configuration, and mission customization. Data products exported and/or ingested from external data repositories and business information such as email and process documents are exchanged with the Management Information System (MIS) infrastructure.

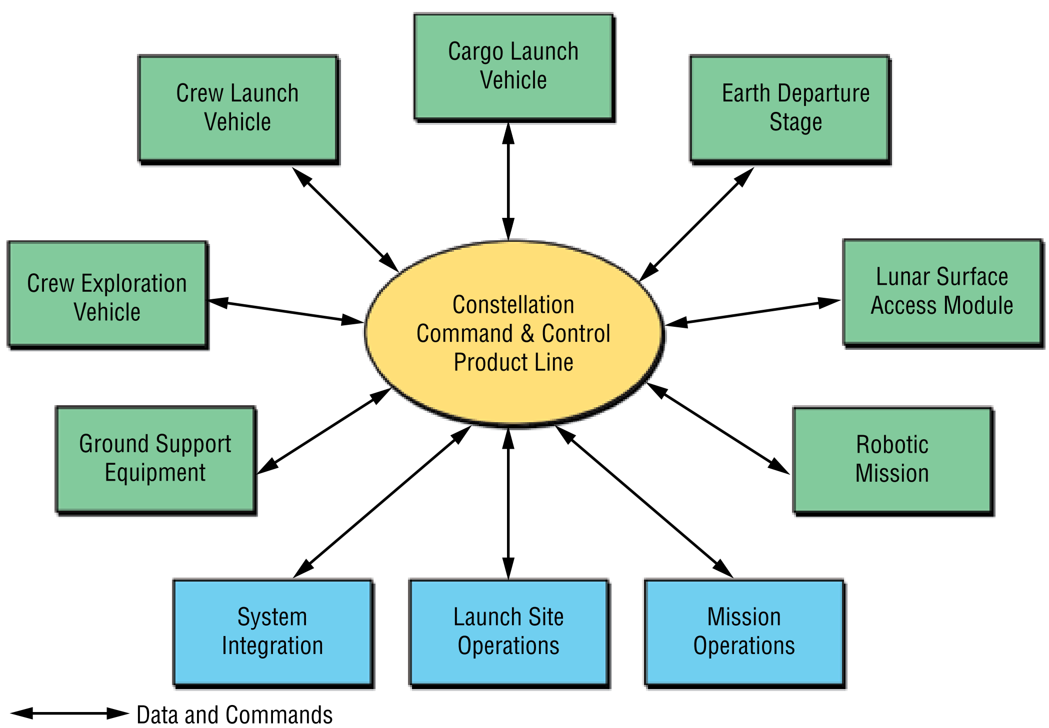


Figure 7-11. Constellation C&C Product Line Systems

System Context

The context of a product line system is characterized at a high level in **Figure 7-12**, where the product line services are represented in the circle and site/mission specific elements necessary to support operations are represented interfacing to the services.

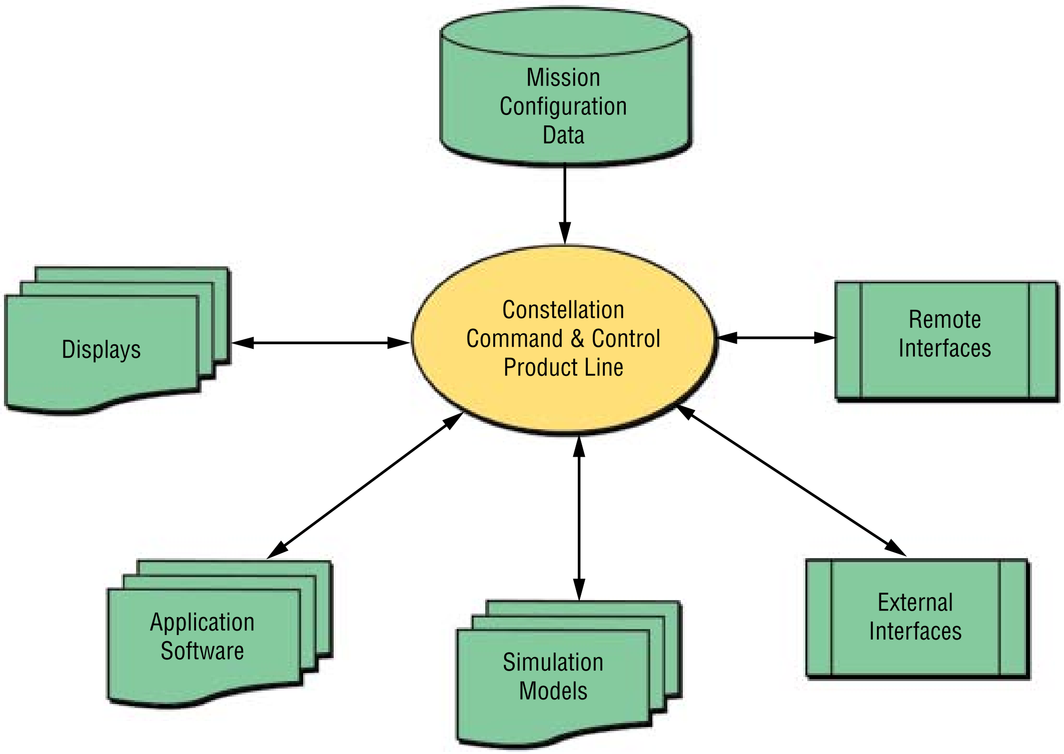


Figure 7-12. Product Line System Context

The product line services provide those capabilities common across Constellation C&C systems. These services include the following capabilities:

- C&C system configuration for operations;
- Real-time control and infrastructure monitoring;
- Web-based access for remote users;
- Data processing and command routing;
- Archival, retrieval, and support of the analysis tools;
- System management and redundancy; and
- Simulation infrastructure for external interfaces simulations.

Site and mission-specific functionality provides those capabilities unique to a specific operational activity. These services include the following capabilities:

- Mission Configuration Data: The repository of all information required to configure the ground systems to support operations;
- Displays and Applications: The system-unique (i.e., CEV, CLV, CaLV, EDS, and LSAM) capabilities required for functional verification, system integration, ground processing, troubleshooting, servicing, and launch processing. Also included are those capabilities required to support the planning and scheduling of communications, resource, and mission activities;
- Simulation Models: The system-unique behavior models required for software certification and training (launch team, mission operations team, and crew);
- External Interface: The interfaces necessary to collect and distribute vehicle, spacecraft, and GSE measurement data and initiate effecter and/or state commands; and
- Remote Interfaces: The interfaces necessary for remote users to monitor operational activities.

7.2 Flight Operations

7.2.1 Exploration Mission Operations

In order to perform a comprehensive cost, schedule, and performance analysis as part of this study, the following basic assumptions, guidelines, and ground rules were used to relate to all aspects of exploration mission operations. These are discussed in the following order within this section:

- Scope of mission operations,
- Mission design and activity planning,
- Crew and flight controller training,
- Flight mission execution and support,
- Mission support segment infrastructure,
- Communications and tracking networks, and
- Mission operations infrastructure transition and competency retention.

7.2.1.1 Scope of Mission Operations

Successful human space flight operations are founded on the program management, operations, and sustaining engineering "triad," providing a robust system of checks and balances in the execution of each human mission. This principal is fundamentally true regardless of program scale or nationality. Although small-scale programs, in which personnel may play multiple roles, quite often blur the lines of distinction between the three elements, the lines of distinction are still functionally present. **Table 7-3** summarizes the functional responsibilities by phase of mission operations development and summarizes the function of each group as it relates to the essential flight planning, training, and execution process.

The roles and responsibilities of each element of the triad comprise the total definition of "mission operations" as it pertains to cost of operations. Management provides program requirements to the operations community and sets priorities for each flight or mission. Operations provides the facilities, tools, training, and flight control personnel to plan and execute real-time operations of the assigned mission per the program's requirements. Engineering provides the spacecraft design expertise to validate that operational plans and procedures fall within the limits and capabilities of the flight vehicle.

Differences between human spaceflight and other spacecraft operations are rooted in:

- The differences in scale with respect to complexity of in-space elements;
- The mitigation of risk to the human element as well as to the facility itself; and
- The scale of multinational relationships where international partners are involved.

Table 7-3. Program Management, Operations and Engineering Sustaining "Triad"

	Management	Operations	Engineering
Plan	• Provide prioritized requirements for each increment/expedition • Provide definition of logistics requirements	• Plan and coordinate each increment/expedition • Develop crew procedures • Develop detailed EVA tasks • Develop detailed robotics tasks	• Define vehicle constraints on operations • Validate operations plans based on vehicle design limitations • Identify changes to logistics and maintenance requirements • Validate procedures based on vehicle design
Train	• Identify mission-unique tasks to be trained • Assembly • Maintenance • Utilization	• Development of training capability and curriculum • Train flight controllers and flight crews on vehicle systems operations • Train all EVA and robotics operations • Provide flight crews language training • Train flight crews and science investigators on science experiment operation	• Support development of new training content
Fly	• Provide management oversight of daily operations • Provide requirements-related and top-level tradeoff decision-making	• Real-time decision-making • Command and control all human elements of the exploration architecture • Coordinate among multiple control centers and numerous worksites to plan and execute daily plans • Build and maintain facilities to interface with all elements of the exploration architecture	• Lead resolution of in-flight anomalies with flight systems • Validate (as required) short-term plans based on vehicle design limitations • Build and maintain flight software and onboard crew displays

While there have been major technological advances in the 40-plus years of human space flight related to the tools of the trade, the fundamental functions necessary to plan, train, and fly a human space flight mission have not changed. Barring major improvements in the technologies of the vehicles involved well beyond the capabilities reasonably attainable through the year 2025, this fundamental process should hold true for many years to come. The following summarizes the primary functional responsibilities in the "plan-train-fly" sequence of activities:

7.2.1.1.1 Plan – Mission Design and Activity

- System design phase support by advocating for operability;
- Trajectory analysis and design;
- Flight planning and crew timeline scheduling;
- Systems and integrated procedures development;
- Flight and ground segment software development;
- C&C systems development and reconfiguration; and
- Operations procedures development and maintenance.

7.2.1.1.2 Train – Crew and Flight Controller

- Provide crew and flight controller training at the major training facilities;
- Develop simulators, mockups, and part task trainers;
- Define lesson and facility development; and
- Support certification of critical personnel.

7.2.1.1.3 Fly – Mission Execution

- Provide flight directors, flight controllers, and a Mission Control Center (MCC) operations support team;
- Provide an MCC, integrated planning system, and communications and tracking network; and
- Perform MCC functions during flight execution:
  - Crew communications and health monitoring,
  - Anomaly response and resolution,
  - Ascent/entry support and abort prediction,
  - Mission timeline planning and modification,
  - Spacecraft housekeeping, command, and control,
  - Engineering data gathering and archiving, and
  - Trajectory and rendezvous planning and execution.

7.2.1.2 "Plan" – Mission Design and Activity Planning

The keys to success in human space flight are found in meticulous and comprehensive mission design and activity planning. It is in this phase that every detail of mission execution is scrutinized, studied, and negotiated among the operational stakeholders. Facilities must be designed and implemented to train the operators (crew and flight controllers) and support mission execution through control center systems and their supporting communications networks. Mission planning occurs under the overall guidance and direction of the flight director, who will conduct the real-time mission. Specialists from each discipline are assigned to perform and support the following technical areas:

- Ground segment platform development
  - Flight controller workstation,
  - Facility design and development,
  - Command, control, and communications, and
  - Information Technology (IT).
- Technical data acquisition for spacecraft systems
  - Command and telemetry,
  - Software/firmware logic,
  - Instrumentation, and
  - Nominal/off-nominal performance.

- Intradiscipline operating concepts formulation
  - Crew displays and controls,
  - Ground segment displays and controls,
  - Flight rules governing responses to off-nominal system performance and contingency situations,
  - Crew and ground segment procedures for C&C,
  - Nominal operating procedures,
  - Off-nominal procedures, and
  - Backup/contingency procedures.
- Interdiscipline operating concepts formulation
  - Team interaction protocols;
    - Flight controller to program,
    - Flight controller to engineering, and
    - Flight controller to partner control center.
  - Flight rules governing integrated responses to multidiscipline contingency situations; and
  - Crew and ground segment integrated procedures for C&C.
- Participation in testing of flight elements
  - Flight segment stand-alone testing of flight element;
  - Flight segment element-to-element testing; and
  - Closed-loop testing with ground segment.
- Flight planning and production
  - Mission planning and design,
  - Trajectory design,
  - Crew activity planning,
  - Ground-controlled activity planning,
  - Flight software reconfiguration (flight-specific command and telemetry definitions), and
  - Command and telemetry format definition.

The integrated team works with the program elements to provide definition to these areas and develop detailed plans, procedures, and mission rules. The flight director uses a common forum known as “flight techniques” in the integration of these activities.

The operations community must be involved in the Design, Development, Test and Evaluation (DDT&E) of all exploration system space flight elements. Operations personnel (flight crew and controllers) should play an active role during the design process to ensure that systems designs meet operational objectives. The operations personnel provide insight to program management to ensure that the design meets operational needs. Interaction between the development and operations personnel will occur throughout the design process and will come to a focus during significant design reviews. Operations personnel should also participate in requirements verification processes. As preliminary operational procedures are developed, they should be validated on the hardware by operations personnel and flight crew. Working relationships developed between the operations and engineering personnel will carry over into the mission execution phase.

7.2.1.3 "Train" – Crew and Flight Controller Training

As the mission planning phase nears completion, the integrated team of crew and flight controllers applies the results in a set of plans and operational products (i.e., procedures and mission rules) that can then allow training specialists to develop both generic (i.e., all-mission) and flight-specific training. The following summarizes the scope of this area of emphasis.

Training facilities/platforms development includes part task trainers, mock-ups, Virtual Reality (VR) trainers, dynamic simulators, engineering simulators, and integrated mission simulators.

Training content development requires significant progress in systems operations concepts and product development. This development also examines the roles of crew versus ground personnel in systems operation.

Crew training consists of classroom instruction, workbooks, part task trainers, and Computer-Based Training (CBT). This training may be conducted as stand-alone (intracrew coordination, drill systems knowledge, and skills) or integrated with the ground segment.

Flight controllers training consists of classroom instruction, workbooks, part task trainers, and CBT. This training includes part task trainers (drill systems knowledge, skills, and expertise), integrated simulations (team coordination and console management), and certification.

7.2.1.4 "Fly" – Mission Execution

Exploration missions will have multiple transit and orbital vehicles and operating surface installations, with several vehicles and installations operating simultaneously. Human and robotic exploration resources will have to be simultaneously managed, while international and commercial resources are likely to require simultaneous management as well. As a result, the exploration program will have the challenge of defining an efficient and appropriate C&C architecture.

7.2.1.4.1 Guiding Principles

The ESAS team envisions a set of guiding control principles for exploration missions. Crewed vehicles should nominally be controlled by the crew independent of Earth and be capable of being under Earth control independent of the crew. Empty and support vehicles can be controlled by Earth or by crewed vehicles in proximity. Robotic vehicles can be controlled by Earth or by crewed vehicles as appropriate for their function. For example, robotic orbiting assets would be primarily controlled from Earth, and teleoperated rovers would be controlled by crews when crews are in proximity. All vehicles will be self-sustaining without Earth or crew intervention between critical events for up to 2 days for lunar missions or 2 weeks for Mars missions, including in the event of single failures of any system(s). Vehicles will maintain command ability after such events. All vehicles will operate autonomously during critical events.

7.2.1.4.2 Command and Control

One central authority should direct all assets in accomplishing the mission. While control may be distributed to leverage existing operational capability and infrastructure, decision-making authority must be centralized. While encouragement of international participation is an acknowledged Level 0 requirement, operational C&C of the crewed elements of human exploration missions will remain NASA's responsibility.

Transitions of C&C should be minimized for efficiency and risk control. Transfer of mission control from one control center to another may be done if appropriate system and operational expertise exists at both locations. Transfer of command from one control center to another may be done when the assets join or depart a larger segment of the mission. Appropriate C&C requirements and architectures will be established and maintained at each step in the program definition.

7.2.1.4.3 Operational Roles of the Crew and Ground Segment

Based on the above level of systems autonomy, the following provides a concept for the split of operational responsibilities between the exploration crew and the ground segment:

The exploration crew will:

- Exercise on-scene authority to make major changes to mission plans or content in situations where time does not permit ground segment consultation;
- Optimize ground-developed plans based on on-scene developments;
- Exercise C&C of dynamic phases (dockings, landings, departures, etc.);
- Perform preventive maintenance as required to keep spacecraft systems functioning within operational limits;
- Perform corrective maintenance;
- Provide functional redundancy to selected autonomous and ground-controlled operations; and
- Perform in-situ science investigations guided by a ground-based science program.

The ground segment will:

- Provide a central authority for authorizing major changes in mission plan or content—time permitting;
- Provide daily planning recommendations to the crew;
- Be capable of exercising C&C of all vehicles through all phases of flight. During dynamic phases of flight, control will be exercised through onboard automation and sequencing as appropriate;
- Provide operations and engineering expertise related to spacecraft systems operations;
- Perform systems trend monitoring and develop troubleshooting recommendations for systems faults that fall beyond the scope of onboard procedures or techniques;
- Provide software maintenance as required to keep spacecraft systems functioning within operational limits;
- Provide a strategic science plan responsive to exploration and discovery; and
- Provide crew psychological support to the extent allowed by communications technology.

7.2.1.4.4 Role of Automation

Communications time delays inherent in missions beyond Earth-Moon space must be accounted for when considering contingency cases and integrated responses to system malfunctions. **Figure 7-13** suggests a methodology for determining the appropriate application of onboard automation. The exploration architecture element design should be guided by the following principles:

- Onboard automation is appropriate for functions that cannot be practically managed by ground segment or crew intervention, or where automation significantly simplifies the effort required by the operator to manage the spacecraft systems;
- Automated functions should be applied across all elements in a consistent fashion, such that the operator does not have to account for which element or module they are in before they interpret their situation; and
- Onboard automation should be minimized or avoided entirely for functions where ground segment or crew intervention is adequate (e.g., non-time-critical functions on scales of hours or longer).

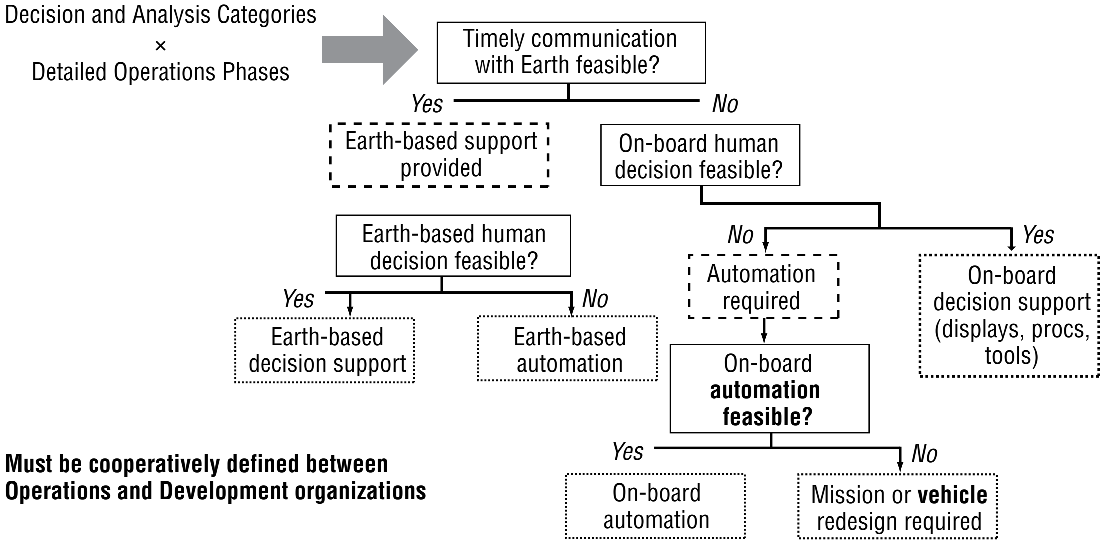


Figure 7-13. Role of Automation

7.2.1.5 Mission Support Segment Infrastructure

7.2.1.5.1 Dealing with Multiple Spacecraft

The ground infrastructure for operations, including facilities, simulators, and control centers, will have to simultaneously monitor, control, and simulate multiple spacecraft. Various combinations of Earth-orbiting spacecraft, lunar and Mars transit/orbiting/descending/ascending spacecraft, and surface habitats may be operating simultaneously. Separate control centers will be staffed and operated for each dedicated mission under a unified command and distributed control architecture. Simulators will be utilized to train crews on multiple spacecraft and be available for real-time failure support. Networks should be able to receive telemetry and communication from simultaneously operating spacecraft and transmit to the applicable control centers. Launch site facilities will provide storage capacity for multiple flight elements that are not in the mission processing flow. An overview of mission operations support infrastructure is provided in **Figure 7-14**.

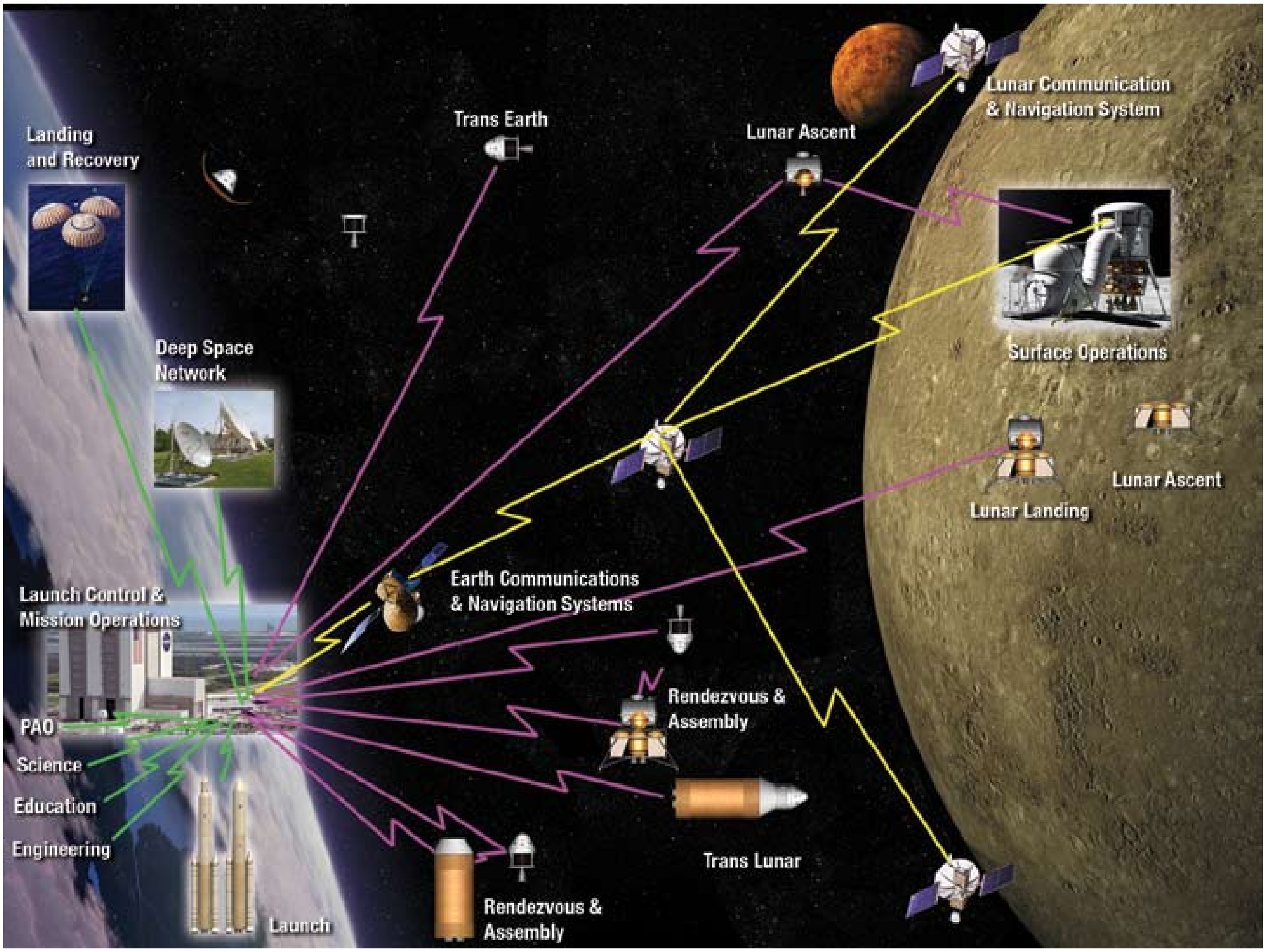


Figure 7-14. Mission Operations Communications Overview

7.2.1.5.2 Common Architecture and Integration

Multiple control centers and facilities will be operating for exploration missions with spacecraft operating in Earth orbit, lunar orbit, Mars orbit, and in transit to and from the surface of both celestial bodies. To reduce costs, all facilities should be designed with common architectures (processors, work stations, etc.) to the maximum extent practical. This will provide additional advantages, as facility redundancy will be increased and procurement simplified. The control centers should provide functions for both short- and long-term mission objectives. Ascent and descent to and from the surface will be highly dynamic phases but will occur in relatively short time frames. Surface operations and coast phases will be more quiescent and longer in duration. The control center architecture should have the capability to support both short- and long-term mission phases. While the same control center may be utilized for multiple mission types and phases, the ones operating simultaneously should be integrated in terms of communications.

7.2.1.5.3 Network Asset Management

During selected periods, or in response to failures, the communications network must be available to support operations 24 hours per day. The network should provide voice communications, spacecraft telemetry, command load, and operational message uplink/downlink capability to cover both scheduled and contingency communications. Contingencies include spacecraft systems failures requiring MCC notification or future assistance, solar flare alerts to the crew, etc. In addition, vehicles from multiple programs and missions will be operating simultaneously. This will require well-coordinated scheduling of communications and tracking network assets to avoid conflicts in nominal operations and prioritize allocation of assets during contingency operations.

7.2.1.5.4 Ground Data Management

A Ground Data Management System (GDMS) will be a collaborative environment that globally ties NASA, contractors, and suppliers together to the maximum extent practical while forming a key part of the systems data management concept. The GDMS concept comprises multiple databases that have the ability to exchange/access data in the most efficient manner possible, whether electronically or by other means. The GDMS should serve as the backbone for planning, process management, requirements management, problem reporting, configuration management, metrics capture, telemetry management, space flight element health management, ground systems health management, and data archival.

7.2.1.5.5 Communications and Tracking

A supporting communications and tracking infrastructure will be required to implement the C&C function. The evolution and deployment of this infrastructure must be integrated with the primary mission architecture. The communications systems should support a tiered hierarchy of operations, including:

- Local control (by flight crew) of deployed assets and EVA in proximity to the crewed elements;
- Local control with remote monitoring of deployed assets from within the crewed elements with ground support; and
- Remote monitoring and control of deployed assets via ground-based systems.

This hierarchy must support the following mission types and flight regimes:

- Near-Earth operations: launch support, early Earth-orbit operations, rendezvous, in space operations, lunar transit, and landing operations;
- Lunar Operations: orbit insertion, landing, surface operations, and Earth transit; and
- Mars Operations: Mars transit, Mars orbit insertion, surface operations, ascent, and Earth transit.

Given the 20-plus year span of this activity, the communications infrastructure should be evolutionary and seamless. Sizing and evolution of the infrastructure must envelope the total mission requirements in support of all active elements, including built-in redundancy for onboard and ground system anomalies, while supporting multiple and simultaneous lunar, Low Earth Orbit (LEO), and Martian operations. Adequate bandwidth for autonomous operations for all near-Earth and lunar activities must be provided. The autonomous activities can be conducted as demonstrations in support of initial robotic operations, including dockings, maneuvers, and transit operations. Intervehicle communication should be accessible by various vehicles and the ground crew to support onboard anomaly investigation and performance assessments of the various elements.

7.2.1.6 Human Spaceflight Transition and Competency Retention

Consistent with the above description of the primary phases of mission operations, a qualitative assessment of how existing mission operations facilities and core competencies would transition from their current Shuttle/ISS focus into supporting exploration needs was compiled. The following summarizes the strategies for each phase of operations, planning, training, and flying.

7.2.1.6.1 Mission Design and Activity Planning

The strategy for vehicle development support calls for DDT&E to be supported by experienced Shuttle/ISS controllers and mission support personnel with expertise in applying lessons learned and operations best practices for vehicle design.

While ascent and entry flight design for CEV is expected to be greatly simplified compared to the Shuttle, lunar trajectory analysis and design, precision surface landing, and abort core competencies must be redeveloped.

Core competencies in flight planning and crew time line scheduling should carry over directly to CEV lunar missions. Flight and ground software development competencies in systems and integrated procedures will be sustained by the ISS. C&C systems development and reconfiguration will be required as CEV flight experience accumulates.

7.2.1.6.2 Crew and Flight Controller Training

The strategy for training facilities is to retire Shuttle simulators and mock-ups after Shuttle standdown, develop CEV/exploration-unique simulators and mockups, continue sustaining ISS simulator and mockups to the end of the ISS program, develop new EVA trainer mockups for the Neutral Buoyancy Laboratory (NBL), and develop a surface analog facility for training surface operations.

7.2.1.6.3 Flight Execution

The strategy for flight execution is to have the flight control team makeup driven by mission and vehicle design while being founded on the essentials of human space flight. The operations team will be the smallest size that the technology, systems, and mission requirements will allow.

The MCC will require pre-Shuttle-retirement facility development to support early test flights (2009). Integrated planning system tools from the Shuttle can evolve to meet exploration needs, and significant amounts of Shuttle flight software can be reused for exploration missions.

7.2.1.7 Mission Operations Cost Estimate

Based on the above description, cost estimates were generated for conducting exploration flight operations through 2025 and are provided in **Section 12**, **Cost**, of this report. The following Ground Rules and Assumptions (GR&As) were used:

- Unique operations preparation and support to development are required for each new space vehicle or major upgrade of a space vehicle.
- LV performance margins will be maintained to avoid significant replanning of operational missions.
- Recurring ISS missions will use a single stable CEV, LV configuration, and mission design.

- Existing JSC MCC will be used with limited modification for all human missions and test flights of human-rated vehicles. Telemetry and command formats are assumed to be compatible with existing MCC capabilities.
- Recurring fixed costs for MCC use will be shared by the exploration architecture after the Shuttle stops flying in proportion to facility utilization.
- New development is required for training simulators, as potential for reuse of existing simulators is very limited.
- Simple mission planning and operations are assumed for test flights and CEV flights to the ISS.
- Complex and highly integrated mission plans are required for initial lunar sorties.
- Simple and quiescent surface operations are assumed for extended lunar stays.
- Crew and ground tasks are considerably simpler than those of the Shuttle during critical mission phases.

The fundamental findings of this costing activity were:

- Mission operations cost is largely dependent on the number of unique space vehicles and annual crewed flight rate.
- Mission operations cost is generally independent of the number of launches involved in a single crewed mission.
- Mission operations cost is generally independent of the launch architecture.

7.2.1.8 Communications Architecture Study Assumptions

As part of this study, the ESAS team established a set of assumptions related to operational support infrastructure needed to conduct CEV flights to and from the ISS and the lunar campaigns through 2025. These assumptions were based largely on the work of NASA's Space Communications Architecture Working Group (SCAWG).

LEO needs were assumed to be satisfied by the existing Space Network Tracking and Data Relay Satellite System (TDRSS) constellation through 2025, assuming the planned TDRSS upgrade deployment occurs in the 2015 time frame.

Similar to Apollo, initial lunar sortie missions will be supported by Earth ground stations. This assumption is based on no identified requirement for communications coverage for critical maneuvers performed on the lunar backside. The SCAWG recommendations call for a Ka-band-array on the ground for providing high-bandwidth support. If exploration objectives are limited to polar and near-side sorties, the ground-based Ka-array capability should be sufficient when augmented with a few S-band ground antennae. This concept is already in development as an upgrade program for the Deep Space Network (DSN) scheduled to begin in 2008. The ESAS assumptions would augment this capability through additional ground site investments in capacity and control systems to make the system available for use on lunar missions by 2018. **Figure 7-15** depicts one possible architecture for supporting lunar sortie missions.

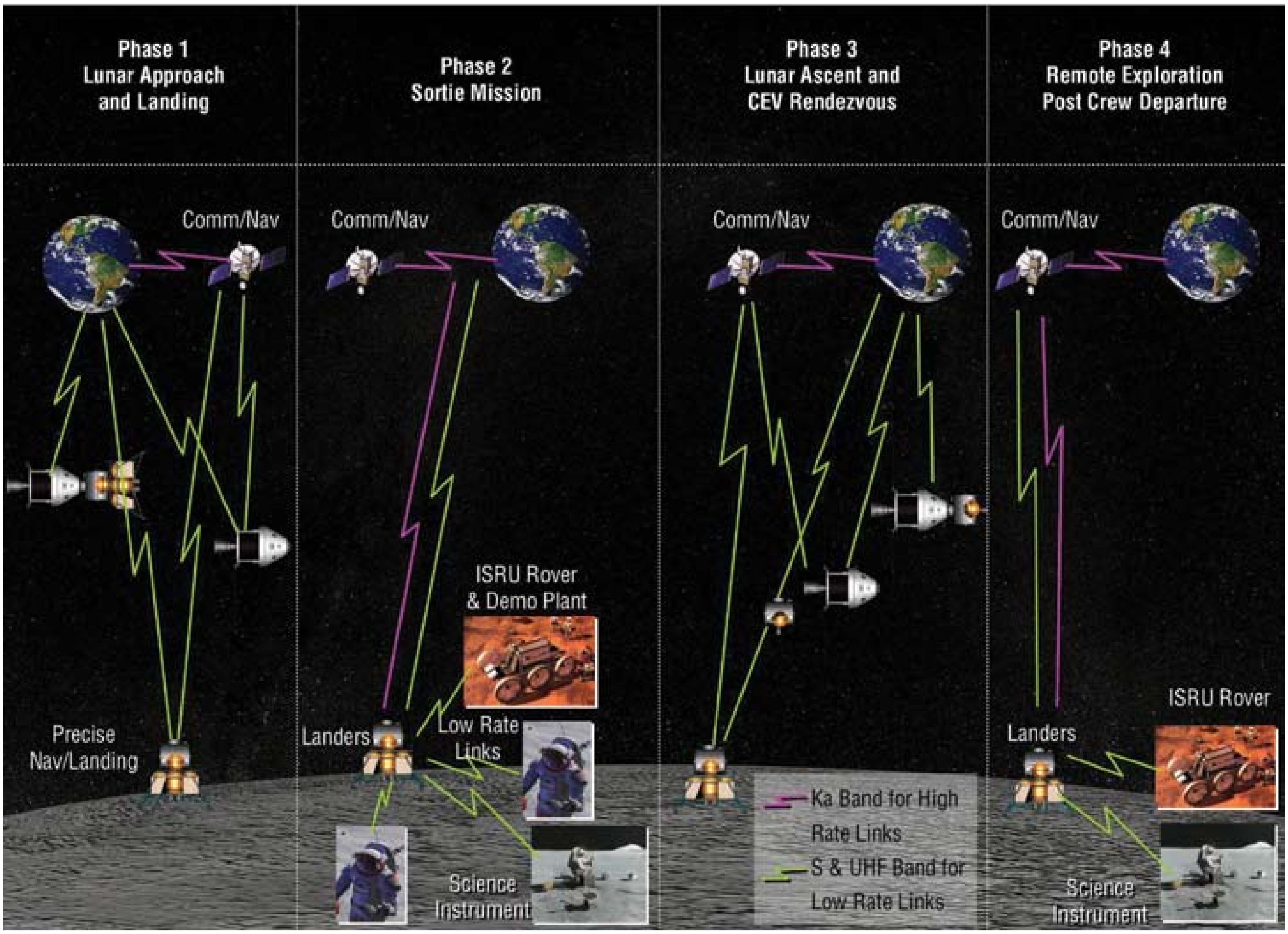


Figure 7-15. Lunar Sortie Surface Communications

A requirement for communications and navigation that supports lunar far-side operations will exist to enable global lunar access for surface operations. To meet this requirement, a lunar relay satellite constellation will be needed. SCAWG assessments have converged on the recommendation to place a constellation of three relay spacecraft with the orbital apoapsis beneath the south pole, thus increasing the viewing, or dwell time, above that region. For example, phasing the spacecraft can ensure that two of three satellites are within view of the south pole. Deployment of such a system should be closely coupled with the design of a lunar campaign (initial global sorties versus outpost buildup at single site, etc.). **Figures 7-16** and **7-17** depict one possible architecture for supporting lunar outpost missions.

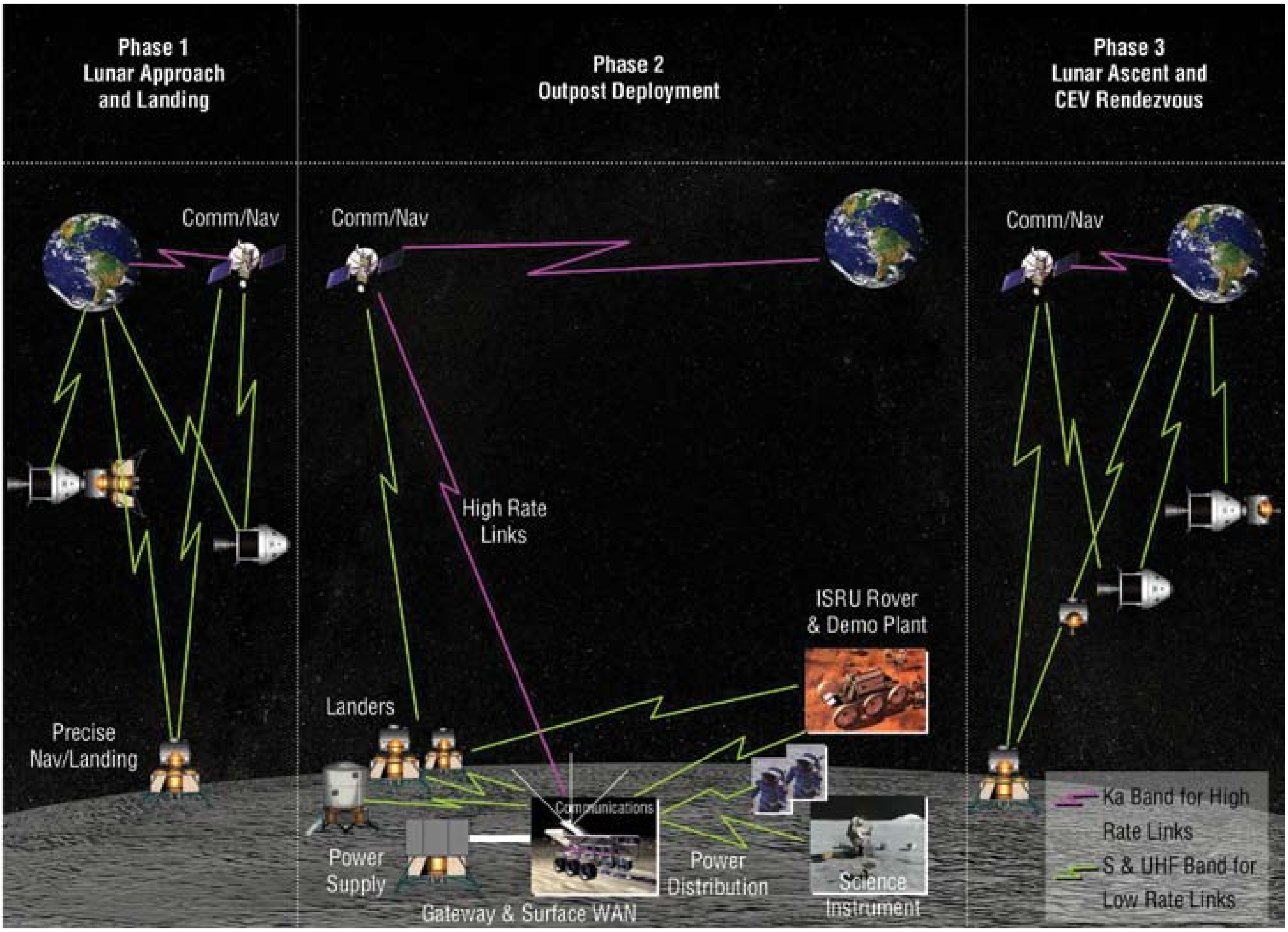


Figure 7-16. Lunar Outpost Surface Communications

During the course of this study, the initial costing of lunar relay spacecraft was based on a TDRSS derivative. The SCAWG also defined a "small sat" concept that is scaleable according to operational needs and would involve gradual capability buildup at a much lower cost. The ESAS team recommends that the SCAWG investigate alternatives to procurement of dedicated LVs for deployment, perhaps integrating them instead onto EDSs of early lunar sortie missions.

Augmentation of the early lunar robotic programs should be aggressively pursued. The Lunar Reconnaissance Orbiter (LRO) and subsequent orbiters should carry independent communications and navigation relay packages designed with at least a 5- to 10-year lifetime. This augmentation builds on capabilities successfully demonstrated on Mars requirements for lunar robotic precursor orbiters. Robotic (and crewed) lunar landers should be augmented to assure communication and navigation relay after primary science/engineering objectives are completed.

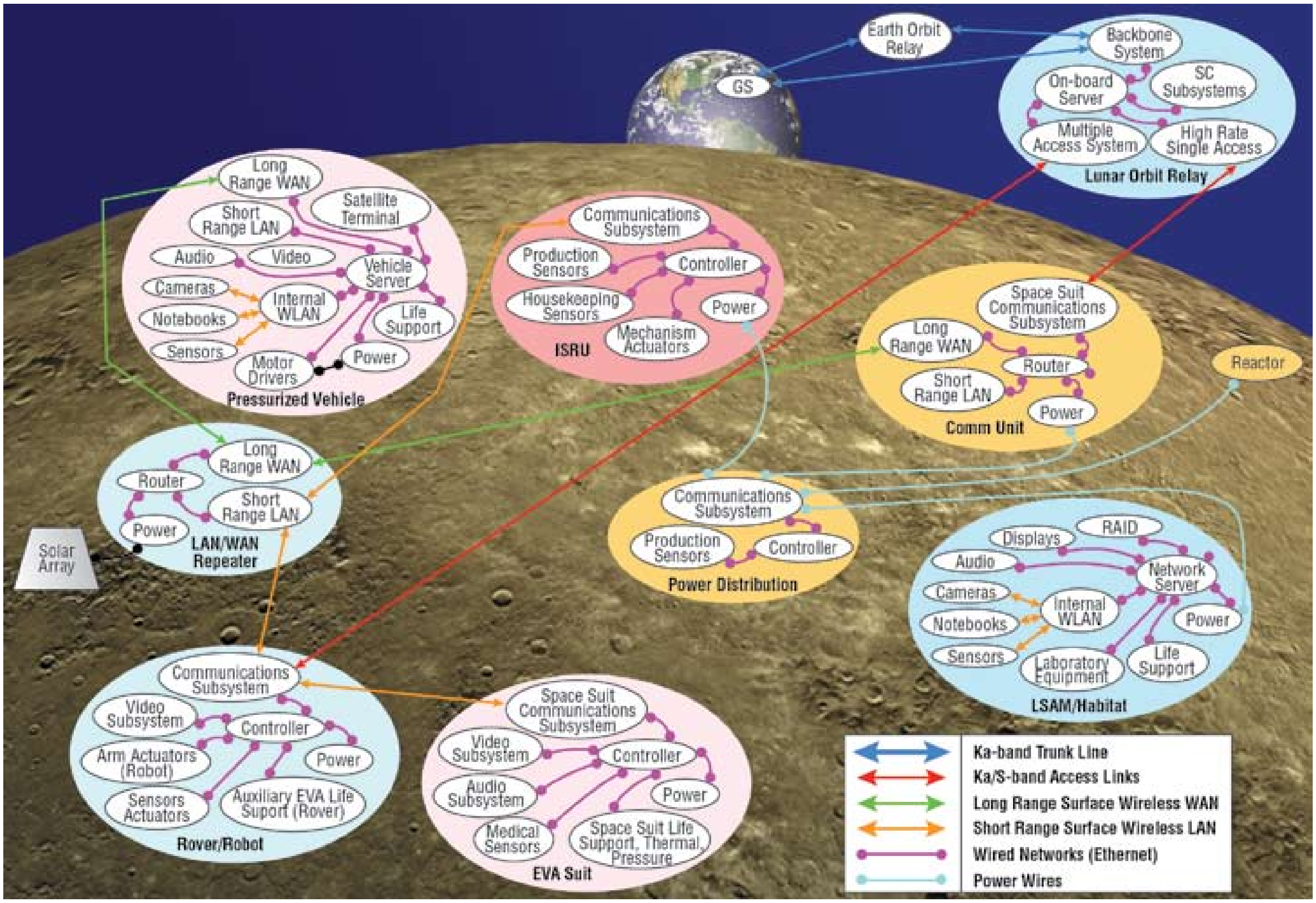


Figure 7-17. Communication and Navigation Networks

7.2.2 Operational Considerations for Effective Exploration Architectures

"Space systems engineering is the art and science of developing an operable system capable of meeting mission requirements within imposed constraints, including (but not restricted to) mass, cost, and schedule."

– M. Griffin,

Space Vehicle Design

2nd Edition

The feedback loop and linkages between operational performance and system design for future exploration missions are particularly important if NASA is to maximize learning during the long-term development process that will be required. Achieving the ambitious space exploration goals will require optimal learning from each mission and the ability to apply lessons learned to future missions and system development cycles. The Agency will need to create an organizational control structure for Safety and Mission Assurance (S&MA) that is self-reflective, self-analytical, sustainable, and capable of compiling and applying lessons learned.

With this goal in mind, the ESAS team addressed the following topics related to the overall approach to effective exploration architecture definition:

- Flight Safety and Mission Reliability,
- Design for Human Operability,
- Commonality,
- Maintainability,
- Interoperability,
- Habitability,
- Supportability,
- Environmental Considerations,
- EVA,
- Communications and Tracking, and
- Software.

7.2.2.1 Flight Safety and Mission Reliability

7.2.2.1.1 Design Simplicity

Design simplicity is a prime criterion for design trade-offs and involves minimization of moving parts and interdependence on other systems, ease of operation, and maintenance by the crew. Simple systems require less operations attention and fewer operator constraints, necessitate less training, and enhance reliability for long-duration missions. However, to paraphrase Albert Einstein, "everything should be made as simple as possible, but not simpler." For example, there will be functions that necessitate adding complexity during the design phase in order to achieve the needed performance and reliability.

7.2.2.1.2 Design Robustness

All architecture elements should allow for safe execution and operation toward completion of all primary mission objectives in the presence of any single credible systems failure. Crew safety shall be assured for any two independent credible failures sustained at any point in the mission. Robustness can be achieved through redundancy, generous design margin, or demonstrated high reliability.

7.2.2.1.3 Redundancy

High-reliability systems are preferred over multiple levels of redundancy where possible to reduce system complexity. Where redundancy is implemented, dissimilar and full-capability systems are preferred. Minimum requirement and performance backup systems are less preferable than full-capability systems. Redundant paths such as fluid lines, electrical wiring, connectors, and explosive trains should be located to ensure that an event that damages one path is least likely to damage another. All systems incorporating an automated switch-over capability must be designed to provide operator notification of the component malfunction, confirm that proper switch-over has occurred, and confirm that the desired system is online.

7.2.2.2 Design for Human Operability

All sensing components associated with enabling the crew to recognize, isolate, and correct critical system malfunctions for a given vehicle should be located onboard that vehicle and be functionally independent of ground support and external interfaces. Two independent instrumented cues are required for any major change in the nominal mission plan. The source of these cues can be space vehicle displays, alerts, and downlink telemetry. Cues are not independent if space vehicle and ground indications are from the same sensor. Redundant sensors are required if two independent cues of a failure cannot be obtained. No requirement should exist for more than two active sensors for a particular parameter on an individual system.

The crew should have the ability to intervene and override any onboard decision regardless of sensor indications. A central objective of the sensor systems is to facilitate the situational awareness of both the crew and remote operators on Earth or another vehicle. The design should allow the operator to make a rapid assessment of the current situation, including the exposure and investigation of off-nominal states.

The crewed vehicle design should allow for the crew to provide functional redundancy to the automated and Earth-in-the-loop systems where practical. Examples of this functionality include orbit determination, maneuver design and execution, and rendezvous and docking operations without the aid of ground control. Crewed vehicles should require crew consent for irreversible actions where practical with respect to human reaction and decision times. Examples of this include a commitment to injection, deorbit, and trajectory correction maneuvers. In the event the need for uncrewed operation arises due to crew incapacitation or maximizing effective use of crew time, the ground crew should be able to control all crewed vehicle-critical functionality.

Crewed vehicles will provide the flight crew with insight, intervention capability, control over vehicle automation, authority to enable irreversible actions, and critical autonomy from the ground. Display and control interfaces will be simple and intuitive. Presentation of onboard systems' status information to the crew should be done in a consistent manner across all flight elements and should be based on common, well-documented practices of measure, iconography, and graphical standards. System design must preclude any failure mode requiring unreasonably swift human action to prevent a catastrophe.

7.2.2.3 Commonality

Commonality at the component and subsystem level should be applied to and across all elements of the exploration system. Strict adherence to commonality will minimize training requirements, optimize maintainability (particularly on long-duration missions), and increase operational flexibility. Design for commonality and standardization of hardware and hardware interfaces will simplify provisioning of spares, minimize the number of unique tools and amount of unique test equipment, and enable substitution between elements. This applies to hardware at all levels and among all architecture elements, including power and data buses, avionics circuit card assemblies, electronic components, and other assemblies such as pumps, power supplies, fans, fasteners, and connectors.

Commonality applies to hardware components and similar software functions across the elements. Vehicle subsystems should be designed so that consumable items can be interchanged with other common subsystems on the overall vehicle and other vehicles. Unattended systems should not have catastrophic failure modes requiring immediate human intervention.

7.2.2.4 Design for Human Operability

7.2.2.4.1 Maintainability

Systems and hardware must be designed to simplify maintenance operations and optimize the effective use of maintenance resources. The mass and volume of spares and other materials required for maintenance and the overall effect on system availability must also be considered. Standard design approaches to simplifying maintenance operations should be employed. These approaches include reduction of the need for EVA maintenance to the greatest degree possible, ensuring easy access to all items that may require maintenance, unambiguous marking of lines and connectors, and implementation of the minimum number of standard interfaces for transfer of power, liquids, gases, and data. System design should ensure that pre-maintenance hazard isolation is restricted to the item being maintained. Maintenance impacts to other operations should be minimized. Due to the time and distance effects on the logistics of re-supply and the effects of hardware failure on long-duration mission risk, hardware should be designed from the initial design phase for ease of repair and maintenance. This involves a shift from the historic LRU philosophy of spacecraft to a philosophy of having the capability for disassembly and repair of the failed unit.

7.2.2.4.2 Interoperability

Systems (or components of systems) should be interoperable with similar systems or components in other architecture elements where possible. Common standards should be established for power systems, operating environment envelopes, consumable or replaceable components, displays and controls, software, communication capabilities and protocols, and other systems attributes. This approach will minimize training requirements, enhance the usability of portable or transferable equipment, and reduce logistics requirements.

7.2.2.4.3 Supportability

The logistics footprint required to support exploration missions must be minimized. Strategies to achieve this objective include broad implementation of commonality and standardization at all hardware levels and across all elements, repair of failed hardware at the lowest possible hardware level (as determined on a case-by-case basis by detailed analyses), and manufacture of structural and mechanical replacement components as needed. Prepositioning of logistics resources (spares and consumables) should be used to distribute logistics mass across the architecture elements and reduce mission risk by staging critical assets at the destination prior to committing human crew. Utilization of in-situ surface resources for production of propellants, breathing gases, and possibly consumable water could significantly reduce the mass that must be delivered to planetary surfaces.

A comprehensive inventory management system that monitors and records the locations and quantity of logistics items should be implemented. The system should accommodate crew interaction while also performing routine audits and item tracking without active crew involvement.

7.2.2.5 Environmental Considerations

The vehicle design should minimize environmentally induced constraints on ground and flight operations while also minimizing sensitivity to extreme variations in both natural and induced environmental conditions. Hardware should be able to survive long periods with no power and be able to return to operation from such a frozen state. Space weather information and alerts should be received directly from the weather sentinels by operational vehicles without having to be processed by Earth stations. Space weather alerts should be transmitted automatically. Crewed vehicles should be equipped with space weather sensors for radiation event alerts where practical. Space radiation should only be accounted for in the design to a risk level commensurate with other sources of risk to crew safety.

7.2.2.6 Habitability

Habitability must be a prime consideration in the design of all crew vehicles and habitats. Convenient engineering design solutions (functional adjacencies) must not compromise habitability. Habitable volumes will provide a pressurized, temperature- and humidity-controlled atmosphere for the crew for all nominal phases of flight except Earth launch and landing. Habitable volumes will also provide protection for the crew against failures that would compromise the habitable environment. During long-duration missions, habitable volumes must allow for simultaneous activities such as sleeping, eating, performing hygiene functions, and exercising, and for off-duty activities with separate and dedicated volumes. Privacy for sleeping, hygiene, and off-duty times is essential.

7.2.2.7 Extra-Vehicular Activity (EVA)

7.2.2.7.1 Habitat Assembly

As the primary focus of planetary operations should be toward exploring the planetary surface(s), EVAs should be minimized in design. Although a crew transportable rover is not envisioned for these EVAs since these tasks will be located within a short distance from the habitat, a small rover to assist with tool transportation may be desirable.

7.2.2.7.2 Habitat Maintenance

The majority of maintainable items should be located internal to the habitat in order to expedite maintenance activities.

7.2.2.7.3 Surface Exploration

The surface mobility systems for a lunar outpost should be sized to support up to five EVAs per week. These EVAs, while initially local to the habitat, will eventually require rovers (pressurized for excursions greater than approximately 10 km) with EVA resupply capability and possible EVA way stations to support EVA resupply in case of a walk-back scenario. This will also require a lightweight and highly mobile suit that can withstand the dust contamination concerns and will be of sufficient crew comfort to support this EVA frequency. This phase of operation may drive the need for a suit separate from the launch and entry/contingency EVA suit in order to withstand the dust, abrasion, radiation, and thermal environments and offer pressurized suit comfort and a self-contained life support system.

7.2.2.7.4 Logistics or In-Situ Resource Utilization (ISRU) Retrieval

EVAs may be required to retrieve the resupply capabilities from logistics modules if those modules are launched and landed separately from the habitat. ISRU may require EVA to transport the manufactured resources back to the end-item location.

7.2.2.7.5 Distance from Habitat

If the tasks are located farther than the emergency return walking distance (approximately 30 minutes), way stations to provide suit consumable resupply should be provided. This would not only enable refill of the suits in an emergency situation to complete the walk back to the habitat airlock, but could also extend the EVA for those tasks without spending an inordinate amount of time continuing translation back to the airlock for resupply. These way stations would not necessarily need to be a permanent infrastructure, but could be laid down on the way out and picked up on the way back during the sortie (which would somewhat constrain the return path). If a particular translation route is frequently used, it may be cost-effective with regards to EVA time to leave these stations in place.

Another method to provide this resupply capability would be to allocate a consumable resupply on the rover, although the distance would still be constrained to a full suit resupply for walk-back with no way stations available (either on the crew transport rover or smaller tool-carrying rovers). A string of way stations might be required if the traverse distance from the airlock exceeds 1 hour. If a rover is used, a string of way stations might be required to maintain a walk-back capability. If no way stations are implemented, the rover distance from the habitat may be constrained by the distance a crew member can walk without resupply.

7.2.2.7.6 Airlocks and Dust Mitigation

Dust will be one of the most problematic issues for both lunar and Martian EVAs. Although the perception of the nuisance factor of dust varied between Apollo crews, all crews except Apollo 11 experienced some issues with lunar dust. The longer a crew was on the lunar surface (including multiple EVAs per mission) and the more intricate a particular mission's EVAs were, the more dust-related problems were experienced. Apollo J-missions, which spent 3–4 days on the lunar surface and conducted three EVAs on each mission, experienced the most problems with lunar dust. Dust got into any unclosed or unsealed volume through almost any size hole, including suit pockets, sample storage bags, nooks and crannies on the Lunar Roving Vehicle (LRV), internal mechanisms of cameras, and onto thermal blankets of surface experiments and communications systems.

The effects of lunar dust were specifically:

- Fouling of mechanical systems such as tools, suit parts, and rovers;
- Degrading thermal control by changing the reflectivity of external coverings;
- Degrading optical surfaces;
- Abrasion of glove coverings and other equipment;
- Irritation of the skin, particularly the hands and glove wear points (e.g., knuckles and fingernails); and
- Irritation of the mucous membranes and eyes.

Despite increased cleaning efforts as part of later J-missions, no cleaning methods were completely effective at removing dust from suit parts prior to entering the Lunar Module (LM). Each lunar mission from Apollo 12 onward experienced problems associated with tracking dust into the LM. While brushing down suits, kicking off boots on the LM ladder, and jumping up and down on the LM footpad prior to ascending the LM ladder were all somewhat effective, no mission was able to completely prevent the introduction of dust into the LM cabin, regardless of how enthusiastically the cleaning methods were applied.

Although the pressure garment was effective at keeping dust out of the interior of the suit, later missions introduced dust into the suit on the second and third EVAs after crew members doffed suits and picked up dust on hand and foot coverings. Also, Velcro to assist in crew restraint during zero-g operations covered the LM floor. Velcro proved particularly effective at collecting lunar dust, which was picked up by the crew and introduced into the suit through handling and donning. The Apollo 17 crew's general consensus was that their suits would not have been able to perform a fourth EVA.

Although the LM Environmental Control and Life Support System (ECLSS) was able to clean the airborne dust from the LM atmosphere on the lunar surface during crew sleep periods and remove floating dust from the atmosphere after LM ascent stage orbit insertion, there was still noticeable dust in the LM atmosphere when the Command/Service Module (CSM) and LM docked. Apollo 17 crew in particular noticed on-orbit eye, nose, and throat irritation when helmets and gloves were doffed prior to transfer of cargo from the LM to the CSM. This not only impacts the suit functionality (bearings, visors, etc.), but also potentially internal operations if not sufficiently handled. Several potential available options include utilizing a suit-washing station prior to bringing the suits into the habitable volume or utilizing an airlock arrangement so that suits are seldom brought inside (e.g., a suit port concept). A combination of both of these may be required, as the suits should eventually be brought inside for maintenance or return to the lander vehicle.

7.2.2.7.7 Surface Habitat Pressure and Atmosphere

In order to increase the margin of crew safety and the work efficiency index (so that the frequency of exploration EVAs can be performed), it is imperative to have a low habitat pressure combined with an increase in oxygen partial pressure. Reducing cabin pressure and increasing oxygen partial pressure can allow for conservative pre-breathe protocols, thus reducing crew day-length concerns and offering a high degree of protection against Decompression Sickness (DCS). Although some amount of final in-suit pre-breathe will be required for all credible suit pressures, pre-breathing can be minimized with the proper cabin atmosphere selection. Pre-breathe protocols should be simple and should not require dedicated exercise equipment and complicated infrastructure with multiple single-point failures. In addition to the resulting habitable design impacts, all long-term crew health countermeasures should take this reduced pressure and increased partial pressure environment into consideration.

7.2.2.7.8 EVA Transfer to Ascent Vehicle

The EVA to transfer the crew back to the planetary ascent vehicle from the habitat is a scheduled EVA for safe crew return and is, therefore, of the highest criticality. As such, this phase may drive the amount of EVA suit redundancy.

7.2.2.8 Communications and Tracking

The following summarizes the recommended attributes of an effective communications support infrastructure for exploration missions:

- Be capable of sustaining uninterrupted, multi-year activity involving multiple vehicles;
- Provide operational capability to support simultaneous near-Earth and lunar or near-Earth and Martian operations;
- Limit the number of unique communication subsystems on the various operational elements through commonality in communication components and data standards;
- Allow for modular upgrades over the life of the program;
- Provide flexibility for contingency ground operations so that no site, tracking station, or facility is a single point of failure; and
- Ensure that C&C interactions between launch processing facilities, MCC, and key element integration sites are consistent and have the appropriate diversity in routing and reliability for the missions operations.

7.2.2.8.1 Vehicle Telemetry and Command

The infrastructure should provide near-continuous telemetry and command with all elements. Communications systems on all vehicles will continuously transmit engineering and health information and are always receptive to commands, either directly to and from Earth or through relay assets. Omnidirectional antennae should provide emergency link capability to avoid dependence on accurate pointing and attitude control in off-nominal conditions where practical.

All vehicles should provide redundant command reception capability, including the ability to remotely restart the element's primary computers. Vehicle safe modes will automatically transmit a 911 signal to help ascertain vehicle status and location. This will be a minimal keep-alive communications capability to help locate assets from Earth, lunar, or Martian operations.

7.2.2.8.2 Voice Communications
The integrated architecture should provide at least two primary channels of operational full-duplex voice communications per crewed mission activity. At least one additional independent backup voice channel should also be provided. The capability for voice communication between the crew and ground segment should be provided when appropriate.

7.2.2.8.3 Video Communications
The integrated architecture should provide at least one channel of commercial quality video to the ground segment. At least one channel of bidirectional video should be provided for operational use and at least one additional channel of bidirectional video should be provided for crew psychological support.

7.2.2.8.4 Intervehicle Communication
The concept that the monitoring and control of a vehicle from another vehicle should be as identical as possible to the monitoring and control of that vehicle from Earth is a guiding operational concept for communications. This level of monitoring and control between vehicles should be available whenever the two vehicles are involved in coordinated operations such as rendezvous, formation transit, and surface exploration. Communications links between such vehicles that provide the availability and bandwidth needed to support the operations activities are required. Relay communications satellites or stations should be provided in the architecture where line-of-sight is not available between such vehicles.

7.2.2.8.5 Communications Security
Near-Earth command links for robotic, manned, and autonomous operations should be protected and secure. Earth access systems and facilities must also be secured.

7.2.2.8.6 Space Weather Monitoring
The infrastructure should ensure that space weather information and alerts can be received by operational vehicles directly from the weather sentinels without having to be processed by Earth stations. Space weather alerts should be transmitted automatically. Crewed vehicles should be equipped with space weather sensors for radiation event alerts.

7.2.2.8.7 Ground Testing Support
Pre-flight test and integration of the vehicles and ground systems should use the same communications systems and mission control facilities used in flight. Earth test beds used for diagnosis and response testing while the corresponding vehicles are in flight should also be monitored and controlled using flight-like communications and should be directly operable by the mission control facility.

7.2.2.8.8 Radio Frequency (RF) Spectrum Assignment
A key goal is to ensure compatibility among all U.S. and/or international elements. RF and optical spectrum frequencies that provide for growth and isolation from interference must be assigned.

7.2.2.8.9 Navigation Support

The navigation architecture should address two distinct operational regimes:

- In-space navigation and guidance support for critical in-space maneuvers between the Earth's surface and the exploration destination; and
- Surface navigation assets to support human and robotic operation on destination surfaces, including navigation aids for precision landing and the tracking of asset position on the surface.

Both radiometric tracking using external resources (such as Earth) and RF or optical tracking using only intravehicle resources can independently provide the required accuracy to achieve the mission objectives. Navigation support for critical in-space maneuvers should be highly robust and reliable. Ground-based systems are preferred over space-based systems in terms of robustness, thus providing a greater number of assets with the greatest flexibility and robustness in tracking network diversity.

7.2.2.9 Software

In a recent study of the role of software in space flight mishaps, many cultural and managerial flaws manifested themselves in the form of technical deficiencies, including:

- Inadequate system and software engineering,
- Inadequate review activities,
- Ineffective system safety engineering,
- Inadequate human factors engineering, and
- Flaws in the test and simulation environments.

Software standards are required to avoid the cost of supporting dissimilar systems and architectures. Similar software functions across all architectural elements, such as Fault Detection, Isolation, and Recovery/Reconfiguration (FDIR/R) or other vehicle management applications, should be developed to a common set of standards.

Computer advancements, the emergence of highly reliable decision-making algorithms, and the emphasis on efficiency make an increased use of automated systems possible. However, full automation is often not practical for some human space flight applications. The program must weigh the DDT&E cost of placing functions on board (including factors such as design flexibility, verification/validation of flight software, and sustaining engineering during flight operations) against the cost of performing functions on the ground for functions where reaction time is compatible with light-time communications delays.

Software design and architecture should support the capability for rapid changes according to changing program and operational needs. This capability should support major version-level updates as well as both prelaunch and in-flight, time-critical, small-scale fixes and parameters. The design cycle in ISS was so extensive that in-flight changes or operational workarounds were almost always required by the time the operations community had access to the final software versions. The software should be modular and provide a capability to turn a function on and off as needed. The design must ensure minimum total system impact from programming changes and additions.

Additionally, software design should offer flexibility to allow the incorporation of upgraded and/or new LRUs with minimal impact, such as use of industry standard interfaces and preserving performance margin. For example, Shuttle upgrade studies with a Global Positioning System (GPS) and Space Integrated GPS Instrumentation (SIGI) have demonstrated the difficulties of integrating recent technologies with older flight software. Rather, a balance must be found between how much human operators trust automation and how much benefit and cost-savings automation provides. This balance may result in an intermediate level of automation somewhere between full-computer responsibility and full-human responsibility. For example, the Space Shuttle Program has found that the appropriate balance of computer and human authority gives launch abort authority to human launch controllers when there is enough time to make a decision. Similarly, computers are given launch abort authority for some time-critical failures.

Distributed control systems should change software states based not only on events (e.g., commands completing), but also on telemetry from lower-tiered units. For multi-tiered systems, command validation at each tier should only include parameters controlled by that tier. Commands should be passed to lower tiers to verify other parameters under their control, with the response then passed back accordingly. Message integrity is the only validation that should be performed at all levels.

Onboard crew should have a method to view any cyclic data parameter. Any parameter that might need to be changed should not be hard-coded. This includes timeout values, Caution and Warning (C&W) limits, etc. These parameters should be changeable by command as well as by some method to set defaults and make large changes at one time. The process for creating and releasing reconfiguration data should accommodate quick-turnaround changes. Testing requirements should be defined in advance to identify cases where the products do not need to be tested on a simulation of the vehicle and, therefore, can be developed more quickly. All hardware interface controls should have a software override/equivalent in case of failure.

8. Risk and Reliability

8.1 Summary

The risk and reliability assessment of the Exploration Systems Architecture Study (ESAS) was an integral element of the architectural design process. Unlike traditional turnkey assessments used to evaluate results independently derived by designers, the risk assessment approach used in this study allowed designers to examine risk trades concurrent with the design process. This approach resulted in an architecture that met vehicle and mission requirements for cost and performance, while ensuring that the risks to the mission and crew were acceptable. This integrated approach to risk-informed design gave designers a risk-centric view of mission architecture and vehicle design to complement their traditional performance-centric view. This complementary perspective allowed them to see, among other things, that the local risk penalties incurred with some high-performance options might produce greater reliability throughout the overall architecture. That is, as the mission architectures evolved, assessments showed that, while certain element risks might increase, the overall mission risk could decrease by choosing the right combination of these dependent elements.

Figures 8-1 and **8-2** show the general evolution of mission architectures with component risk defined. The order of architectures considered flow from top to bottom, roughly representing the manner in which the ESAS architectural investigation proceeded. In general, the risk of Loss of Mission (LOM), as well as Loss of Crew (LOC), decreased as the risk assessment guided the architecture design process. As shown in **Figures 8-1** and **8-2**, while certain trades resulted in individual penalties, the proper combination of trades generally resulted in an overall lower risk of LOM. The single-launch mission resulted in the lowest risk of LOM. However, in this case, LOC penalties for the Launch Vehicle (LV) (as shown in **Figure 8-2**) and performance limitations, in terms of landed mass on the lunar surface, prompted designers to select the Earth Orbit Rendezvous-Lunar Orbit Rendezvous (EOR–LOR) 1.5-launch hydrogen descent, pressure-fed ascent option with the lowest LOC risk and LOM risk approximately equal to that of the single-launch architecture. **Figures 8-1** and **8-2,** and the specific trade studies and results summarized in them, will be discussed in more detail in later sections.

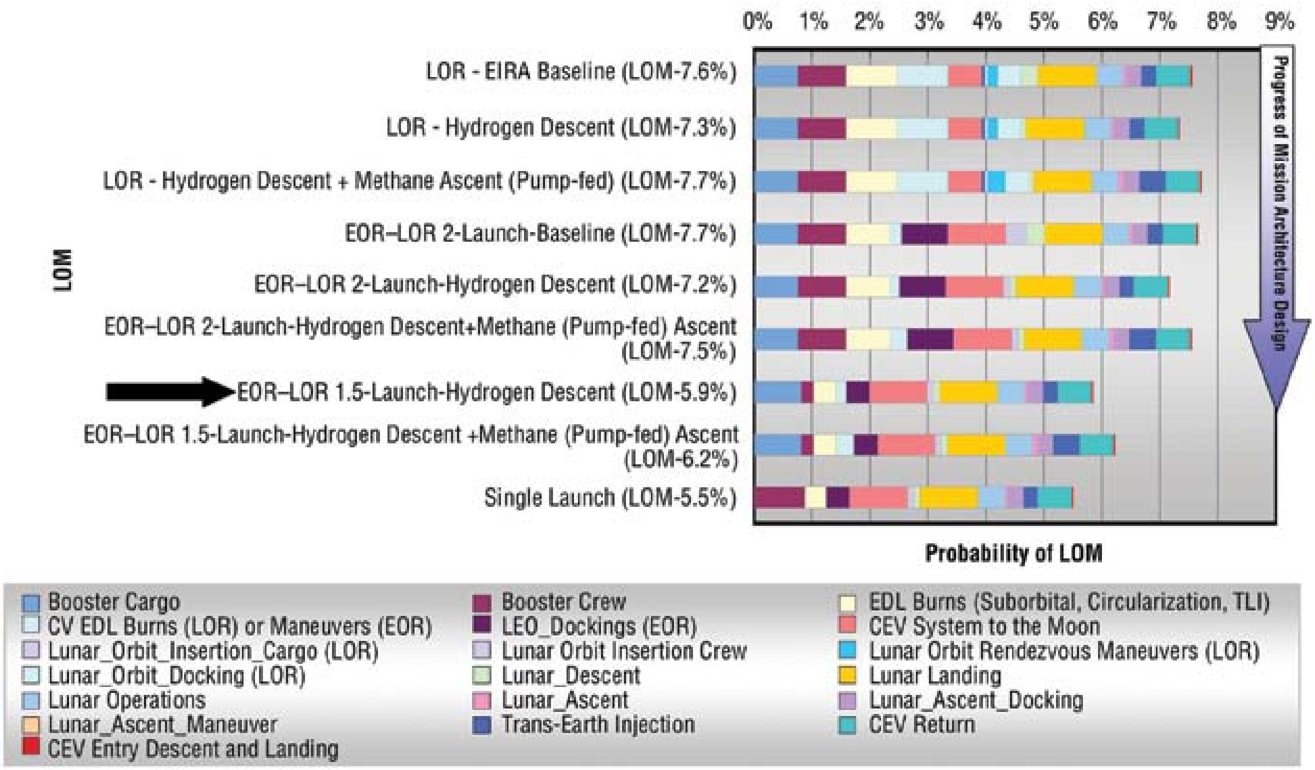


Figure 8-1. Comparison of All (

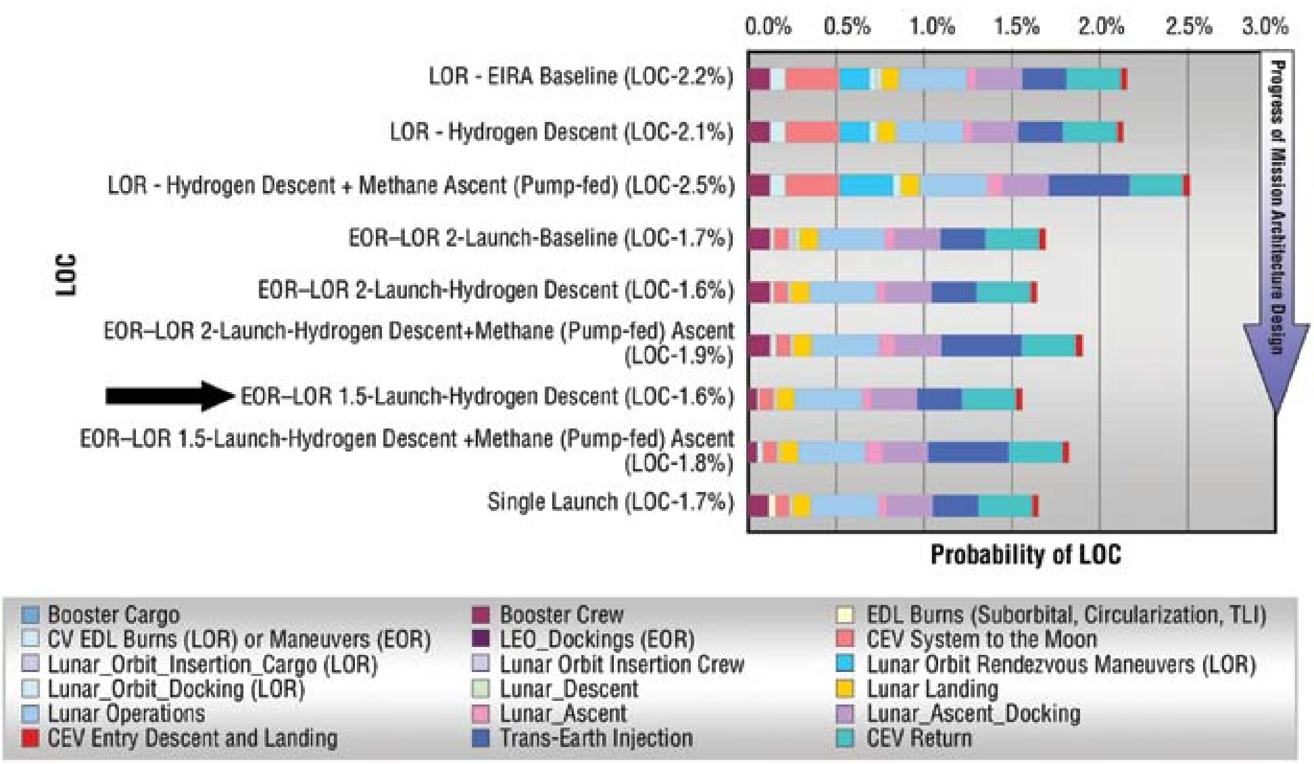


Figure 8-2. Comparison of All Cases for LOC

The risk and reliability portion of the ESAS focused on identifying “differences that made a difference” in architectural risk. The conceptual nature of proposed vehicle designs and the analysis of the mission scenarios at this stage in the process made it essential to identify the architecture-discriminating issues that would drive the risk of the program. The quantification of the individual discriminating risks was also important to ensure that concentration of risk reduction in one area did not compromise the overall risk level of the design and result in increased risk in other areas. This focus allowed program resources (i.e., mass, time, dollars) to be spent in a manner consistent with their importance to the overall architecture, not just their individual importance. Determining these drivers allowed the ESAS team to select the missions and vehicles to create architectures that would produce the highest likelihood of mission success with the least risk to the safety of the crew. Using analysis tools such as the Screening Program for Architecture Capability Evaluation (SPACE) tool (**Appendix 8A**, **SPACE Background**) and the Flight-oriented Integrated Reliability and Safety Tool (FIRST) (**Appendix 8B**, **FIRST Background**), the risks of mission and vehicle elements were quantified, and the top drivers were determined from these results. Classifying the top drivers was intended to provide guidance to future analysts indicating where to properly focus their analytical efforts and suggesting the level of resolution required for the models.

Results from quantitative risk and reliability analysis were an important input to decision-making during the design process. These results provided concrete ways to compare relative risks and to inform the design decision makers of the risk consequences of their decisions. Key programmatic decisions that were influenced by the risk assessment results included:

- Choice of lunar mission mode: The significant safety benefit of a second habitable volume in EOR missions was demonstrated in the analysis. This benefit supported the decision to use this mission mode. The safety benefit of the 1.5-launch mission architecture was demonstrated as well, due to its use of a single Solid Rocket Booster (SRB) for crew launch.
- Choice of Crew Launch Vehicle (CLV): The risk analysis demonstrated the significant benefit of the single SRB launcher with a single Space Shuttle Main Engine (SSME) upper stage, albeit recognizing the residual risk due to the SSME air-start requirement. The risk assessment supported the designer’s intuition that the simplest possible system developed from the most mature propulsion elements was superior to other design choices.
- Choice of propulsion systems: The need for reliable propulsion systems for return from the lunar surface requires the propulsion systems for lunar ascent and Trans-Earth Injection (TEI) to be as simple as possible and to employ systems that are mature and have the potential for achieving acceptable reliability. The risk analysis quantified the benefit of maturing these systems during International Space Station (ISS) missions, thereby suggesting that the same propulsion system be used for both applications. This led to the elimination of pump-fed Liquid Oxygen (LOX)/methane systems for the Lunar Surface Access Module (LSAM) ascent stage because the pump-fed system would not likely be ready in time for the ISS missions. The designers discovered that a single-engine ascent stage was a preferable option to a double-engine system because the geometry and physics of the design would make it difficult to achieve a balanced single-engine ascent on a multiple-engine system. The failure predominance of the propellant supply and delivery portion of a pressure-fed system with an ablative combustion chamber and nozzle also suggested the dubious nature of the risk benefits of engine-out in the LSAM ascent stage. Finally, the analysis demonstrated that, although possibly less reliable than a hypergolic

system, the LOX/methane system could be developed in time and with sufficient reliability for the mission. The additional performance benefit of a mature LOX/methane system, along with the choice of a pump-fed LOX/hydrogen engine for LSAM descent, provided the launch mass capability to enable the 1.5-launch architecture, thus allowing for crew launch on the single-stick SRB, which has the lowest LOC probability. The LOX/methane system was also desirable to eliminate the operability issues related to hypergols and to enable the use of in-situ methane on Mars and oxygen on the Moon and Mars. The crew safety and mission success benefits provided to the overall architecture showed the individual local reliability benefits of a hypergolic Crew Exploration Vehicle (CEV) propulsion system would be overwhelmed by architectural benefits of the higher performance, albeit less mature, LOX/methane option. The use of a higher performance pump-fed LOX/hydrogen engine on the LSAM descent stage would increase performance of the engine that enables the 1.5-launch solution. The analysis also led to the elimination of the LSAM descent stage Reaction Control System (RCS), thereby simplifying the design and adding margin.

- Elimination of unnecessary radiation shielding from the CEV: Quantification of risk from radiation led to the elimination of over 1,000 kg of radiation shielding from the CEV, with a reduction in CEV mass of 2.4 mT and a reduction of injected mass by 3.7 mT. This mass enabled the 1.5-launch mission without requiring the use of a pump-fed ascent stage on the LSAM and provided more margin for the design. The 1.5-launch solution sensitivity to CEV radiation shielding is shown in **Figure 8-3**.
- Relaxation of the requirement for aerodynamic monostability: Monostability ensures that the CEV will aerodynamically trim in a single attitude. However, the requirement for monostability, in the context of the entire system, is only one way to achieve the goal of safe trim during reentry given a loss of primary flight controls. Because the monostability requirement adversely constrains the Outer Mold Line (OML) of the CEV, the requirement was relaxed to allow alternate means of maximizing architecture safety levels.
- Definition of acceptable risk: The risk assessment demonstrated that the risk of a lunar mission is significant, but it could be controlled to a level similar to what is accepted on Shuttle missions today. NASA must acknowledge this risk and execute the program accordingly. In addition, the analysis suggested that crew missions to the ISS may be at least 10 times safer than the Shuttle once the CEV service propulsion system is matured, despite the fact that the first several test missions might incur larger initial risk.

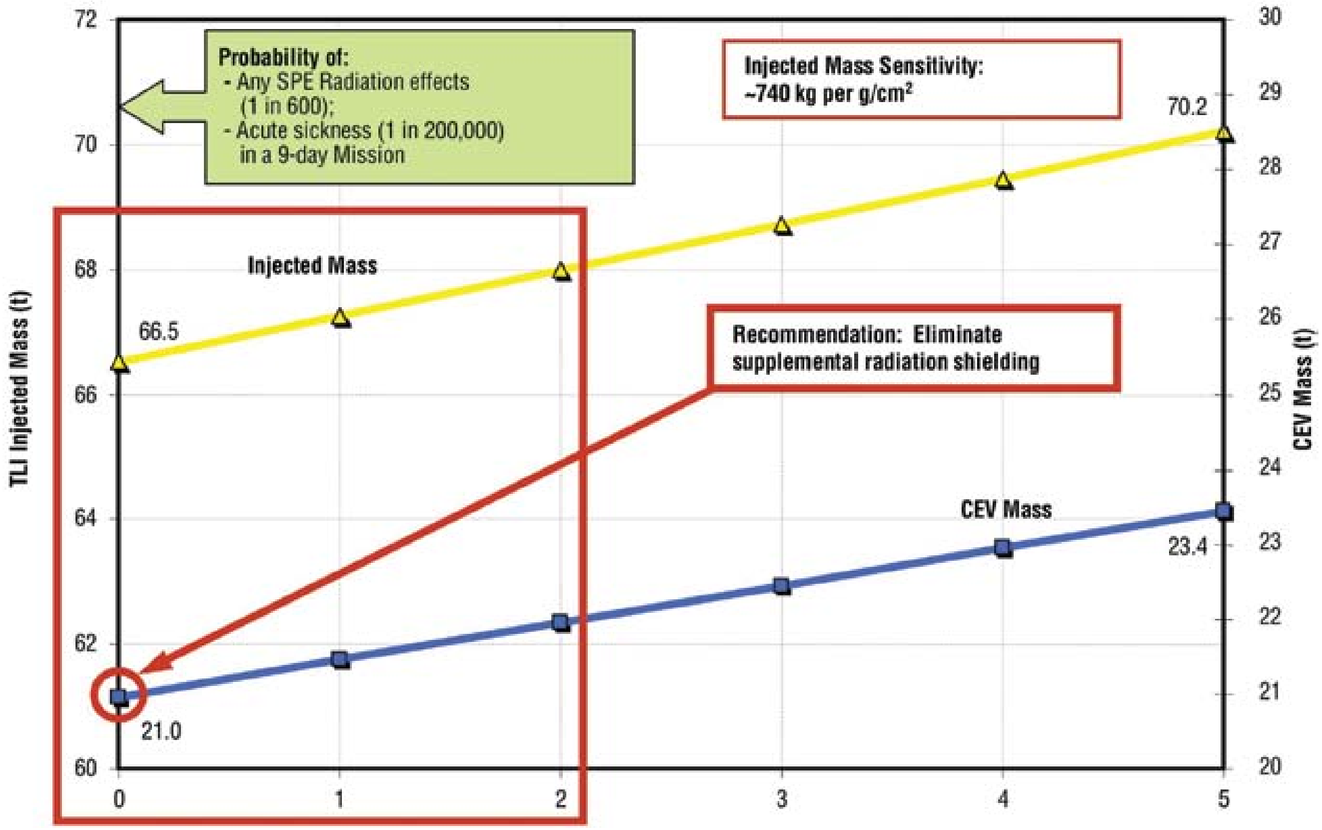


Figure 8.3. 1.5-Launch Solution Sensitivity to CEV Radiation Shielding

Risk assessment results were used to determine the highest-risk flight phases of the ESAS architecture. Pre-mission risks by flight phase are shown in **Figures 8-4** and **8-5**.

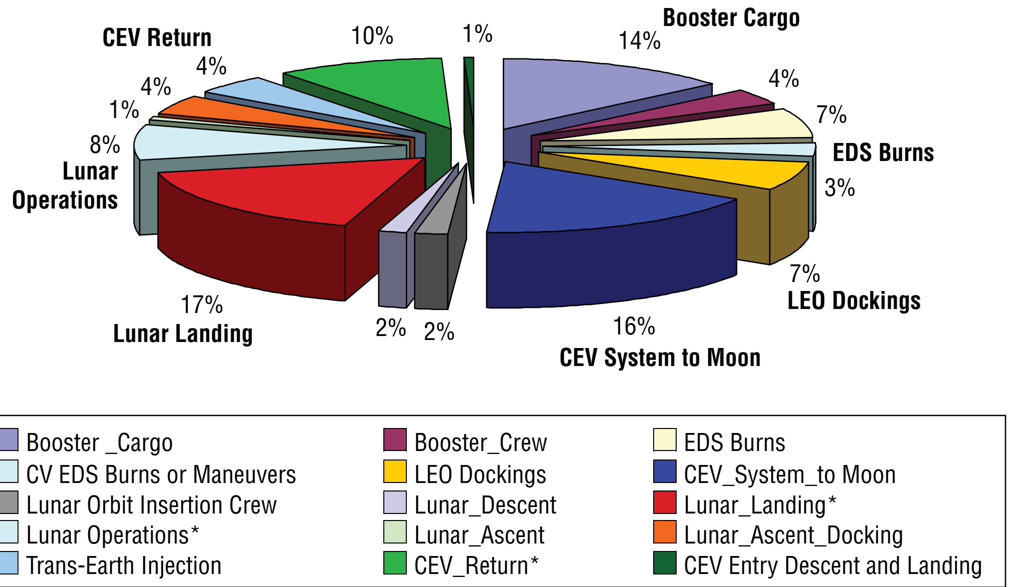


Figure 8-4. LOM Contributors for EOR–LOR 1.5-Launch Mission

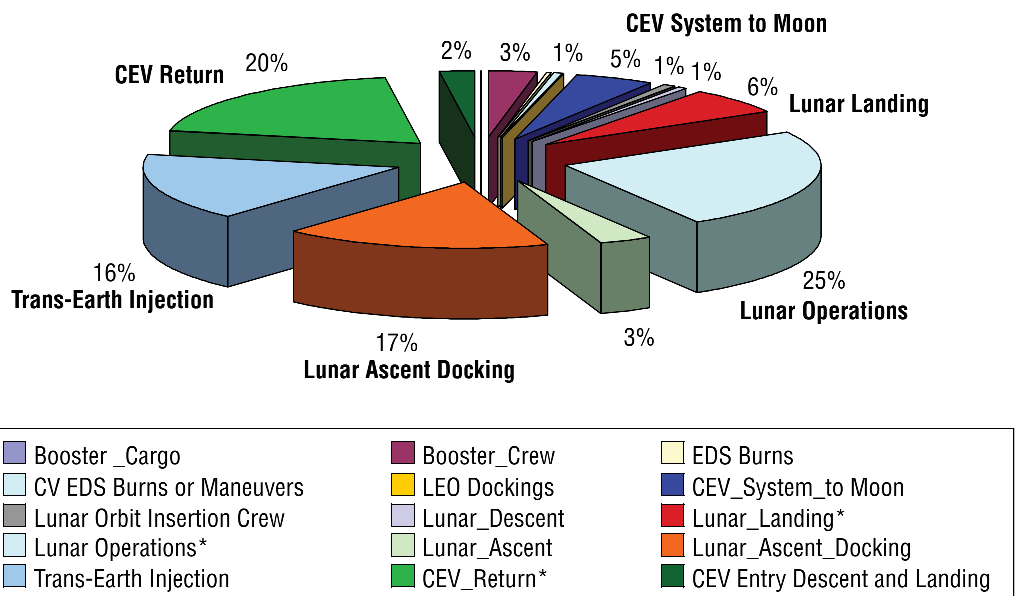


Figure 8-5. LOC Contributors for EOR–LOR 1.5-Launch Mission

The top ten risk drivers were determined to be:

- LOX/methane engine development;
- Air start of the SSME;
- Lunar-Earth reentry risk;
- Crew escape during launch;
- Liquid Acquisition Devices (LADs) in the CEV service propulsion system;

- Lunar vehicle LOX/hydrogen throttling on descent;
- Integration of the booster stage for the Heavy-Lift Vehicle (HLV);
- J–2S development for the Earth Departure Stage (EDS);
- Unmanned CEV system in lunar orbit; and
- Automated Rendezvous and Docking (AR&D).

These identified risks should be examined and tracked carefully as the architecture design and development progresses. Additional risks will certainly be added in the future. Vigilance will be needed throughout the program to assure that other risks remain low.

8.1.1 LOX/Methane Engine/RCS Development

The development of the LOX/methane engine was recognized as one of the largest architectural risks during the course of the ESAS. No LOX/methane engine has had any flight test experience and there has been only a limited number of Russian ground tests. The LOX/methane system was desirable from a performance perspective and also to eliminate the operability issues related to hypergols and to enable the use of in-situ methane on Mars and oxygen on the Moon and Mars. The choice of the simple pressure-fed design over the higher performance, but more complex, pump-fed alternative for the LOX/methane engine should significantly increase the likelihood of the engine maturing in time to meet the 2011 CEV launch date and, ultimately, to rapidly reach a high plateau reliability. Despite this forecasted eventual high reliability, the lack of heritage and flight history suggests an initially low level of maturity. In turn, this lack of maturity is reflected in a low initial forecasted success likelihood of 80 percent. This low initial value suggests that a significant test and flight program to ISS should be planned to lower this risk to the plateau value. In particular, supporting analysis using a Bayesian predictive model suggests that the engine would be forecasted to require 19 flights before this plateau is reached. (See **Appendix 8C**, **Reliability Growth**.) The forecasted growth curve of the LOX/methane engine reliability as a function of the number of test flights is shown in **Figure 8-6**.

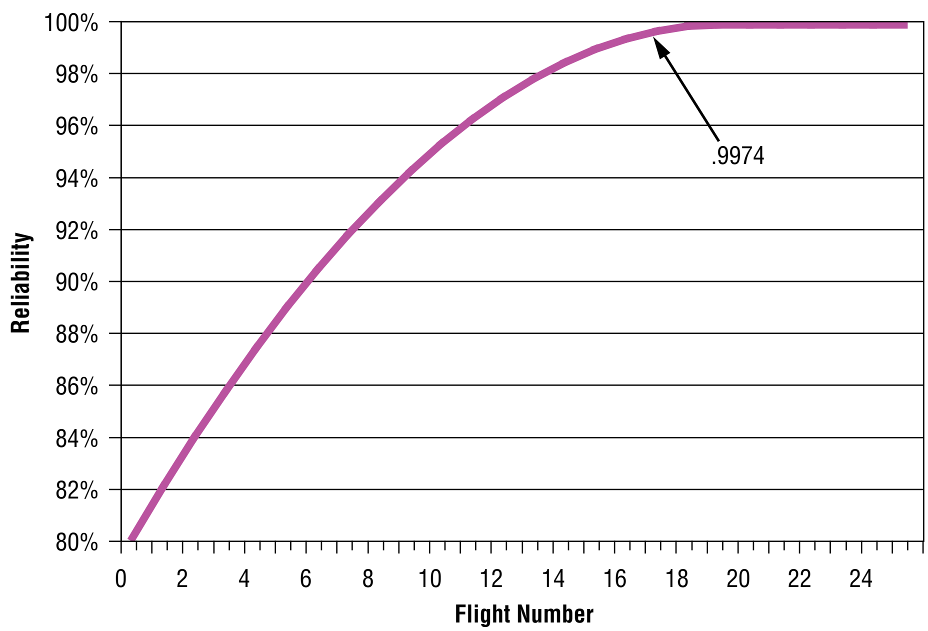


Figure 8-6. Reliability Growth of LOX/Methane Engine

The LOX/methane-based RCS has development and mission risks of its own. Existing RCSs do not require igniters. The liquid propellant for the thrusters will require conditioning prior to each firing. This will present a challenge to the RCS designers in the development of the RCS propellant supply and manifolding scheme. The propellant lines are small and would extend some distance from the tanks without proper manifolding. If individual propellant lines are used for the thrusters, leakage becomes an issue, while shared lines have the potential for common cause failures. All of these issues must be carefully considered by the designers in the ultimate RCS design development.

8.1.2 Air Start of the SSME

The SSME is a fuel-rich, combined-cycle, pump-fed LOX/hydrogen engine with significant maturity, heritage, and strong test- and flight-proven reliability. However, the CLV upper stage requires the SSME to start in flight. SSME air starts have never been demonstrated on the ground or in flight. The upper stage SSME test program will include simulated vacuum starts to aid in maturing the system. However, there is always the risk that exact conditions at staging and ignition may not be adequately simulated on the ground. The air-start function is considered moderately complex with no heritage, making it a risk driver. The initial reliability was estimated to be 70 percent due to the possibility of unknown risks. However, because of the significant SSME heritage, the system is expected to mature rapidly and reach plateau reliability in five flights. The reliability growth of an SSME air start is shown in **Figure 8-7**.

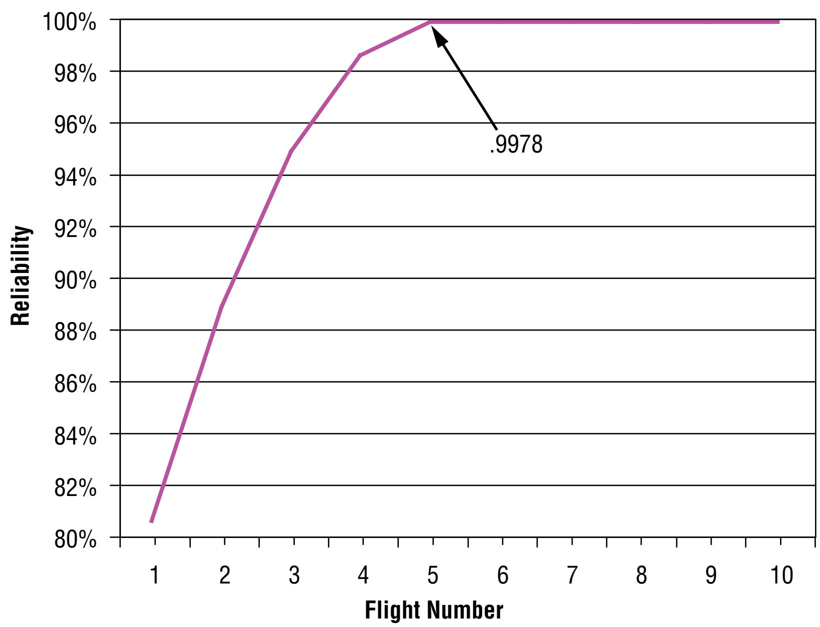


Figure 8-7. Reliability Growth of SSME Air Start (Pump-fed LOX/Hydrogen)

8.1.3 Lunar-Earth Reentry Risk

The Thermal Protection System (TPS) is responsible for the vehicle integrity throughout reentry. Although ablative heat shields were successfully developed for the Apollo program, the development was not problem-free. **Figure 8-8** shows the many repair plugs that had to be added during the manufacture of the Apollo VII heat shield.

Figure 8-8. Apollo VII Heat Shield (with Repair Plugs)

In the case of the CEV versus Apollo, the much larger area of the CEV suggests significant development of the TPS would be required in spite of existing Apollo heritage. The development of the TPS for the CEV would require the certification of the manufacturing process and the ability to recreate the Apollo material. Initial analysis indicates that the performance benefits of a new material would be worth the extra effort required, instead of recertifying Apollo material. However, analysis has shown that the additional development step is not something that should be taken for granted in terms of schedule. Certifying an existing material generally leaves the methane engine development as the leading risk driver, but, if technology development is needed for the TPS, then the TPS becomes the dominant schedule driver in technology development.

Improvements in Computational Fluid Dynamics (CFD) will allow tests and simulations of vehicle reentry to be modeled to further understand this risk. Fortunately, significant progress has been made since Apollo in the area of CFD simulations of reentry conditions. **Figure 8-9** shows an initial simulation that was performed to model the contours of constant axial velocity experienced by the CEV on reentry. Such accurate representations of reentry physics were unavailable during the Apollo era and would be expected to be extremely helpful during CEV TPS design development.

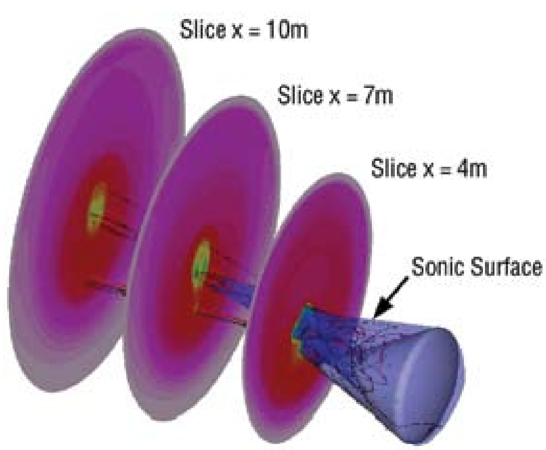


Figure 8-9. Contours of Constant Axial Velocity

8.1.4 Crew Abort During Launch

The loads applied to the vehicle on ascent, as well as other aspects of the accident environment, can pose a high risk to the crew if an abort is required. To analyze this risk, physical and logical simulations and tests are required. The required physical simulations would include: fracture and breach physics, cloud interaction and combustion physics, Navier-Stokes CFD analysis of escape and recovery, and an integrated comprehensive evaluation of both logical sequences and physical environment. **Figure 8-10** shows an example of a model of the normal aerodynamic loads applied to a vehicle on ascent. Such accurate representations of the ascent aerodynamics require the use of advanced CFD codes, representative geometric models of the vehicles, and technically adequate methodology to tie the geometry to the physics. In addition, construction of pressure and velocity profiles requires significant computational capability to represent the ascent accurately. The addition of fracture models mapping to internal motor or engine conditions, the combustion physics, and the fracture fragment propagation in the air stream makes the problem even more challenging.

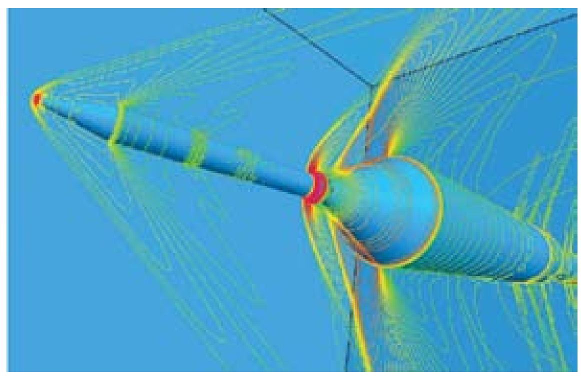


Figure 8-10. Aerodynamic Loads on Steady-State Ascent

8.1.5 LADs in the CEV Service Propulsion System

The LADs in the CEV service propulsion system will require much testing and certification to meet the required level of reliability. The Space Shuttle uses screen channel LADs in both RCS and service propulsion system tanks. Key issues are fluid properties for the design region, screen bubble point data for fluids, and modeling of temperatures of interest (i.e., subcooled LOX viscosities). A review of the history of LADs revealed several issues with their use, including the fact that the Shuttle LADs qualification program took 7 years to complete. **Figure 8-11** shows the Space Shuttle service propulsion system tank internals, including the dividing bulkhead and LAD gallery in the lower compartment.

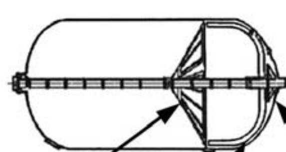


Figure 8-11. Space Shuttle Service Propulsion System Tank Internals

8.1.6 Lunar Vehicle LOX/Hydrogen Throttling on Descent

For all throttling engines, it is critical to maintain the injector pressure drops necessary for proper propellant injection and mixing over the throttling range without causing instabilities in combustion. This requirement creates a substantial risk to the LOX/hydrogen engine on the lunar vehicle for descent. One of the approaches that can be used for deep-throttling, pressure-fed engines is a sliding pintle to control engine orifice size. While experience on the LEM descent engine indicates a pintle can be used, the LEM descent engine was fueled with hypergols. No previous sliding pintle applications were found for a hydrogen-fueled engine. However, if a sliding pintle development proves problematic, there are alternative approaches that have been used successfully on at least one hydrogen-fueled engine. One throttling application was the RL–10 throttling approach shown in **Figure 8-12**. The throttling experience with the RL–10, using dual throttling valves, was used on the Delta Clipper Experimental (DC–X) program. This approach suggests alternatives that might be employed in addition to the sliding pintle. However, regardless of the approach taken, this is still an area that represents a risk to the mission and should be analyzed further.

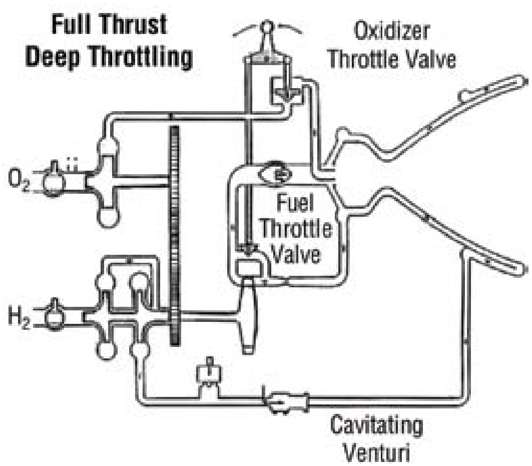


Figure 8-12. RL–10 Throttling Approach

8.1.7 Integration of Booster Stage for the Heavy-Lift Vehicle

The integration of the booster stage engines for the heavy-lift Cargo Launch Vehicle (CaLV) (**Figure 8-13**) is an element that poses a fair amount of risk. This risk is driven by the integration of two five-segment Reusable Solid Rocket Boosters (RSRBs) with five SSME cores for the booster stage. Integration risk is prominent because the SSMEs themselves are mature and reliable, as are the Shuttle SRBs, albeit in a four-segment design. Possible risks include engine propellant manifolding, thrust imbalance, thrust vectoring, and the possible interaction between the two Reusable Solid Rocket Boosters (RSRBs) and the liquid core, as well as residual uncertainties due to the addition of a fifth segment to the SRB.

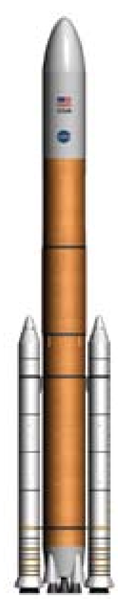


Figure 8-13. CaLV Propulsion System Integration

8.1.8 J–2S Development for the EDS

The use of a J–2S engine for an Earth Departure Stage (EDS) is an area of high risk because a J–2S engine has never been flown. The J–2S (J–2 simplified) was designed to replace the Saturn vehicle upper stage J–2 engines. While the J–2S replaces the J–2's gas generator engine cycle with a simpler tap-off engine cycle, the development program for the J–2S ended in 1972. The J–2S is more than just a paper engine, however. It has significant ground test experience and was almost certified for flight. However, the J–2S development was not completely trouble-free. There were some problems with the tap-off cycle and the engine had no flight experience. Thus, the estimated time of 4 years for qualification, fabrication, and testing of the engine poses a significant risk to the program. A test firing of a J–2S engine is shown in **Figure 8-14**.

Figure 8-14. J–2S Test Firing

8.1.9 Unmanned CEV System in Lunar Orbit

For the first time, the mission will require leaving an uncrewed vehicle in lunar orbit for an extended period with eventual crew return. This vehicle must be operationally ready and must perform reliably after its quiescent period when called upon. It is expected that this risk will be effectively mitigated by the early CEV flights to the ISS since the CEV (**Figure 8-15**) is likely to remain quiescent at the ISS for even longer periods than would be required for lunar missions.

Figure 8-15. CEV

8.1.10 Automated Rendezvous and Docking (AR&D)

The final lunar mission architecture selected does not require AR&D. Pressurized cargo delivery to the ISS will require some level of AR&D; however, ISS crew will be available to provide backup capability. Other lunar missions that were considered did use AR&D. In many of these missions, the risk presented from AR&D was a driver. Even a manned vehicle docking with a passive vehicle involves significant risk as shown in **Figure 8-16**. However, it is expected that the experience gained from early CEV missions to the ISS would substantially mitigate the risk.

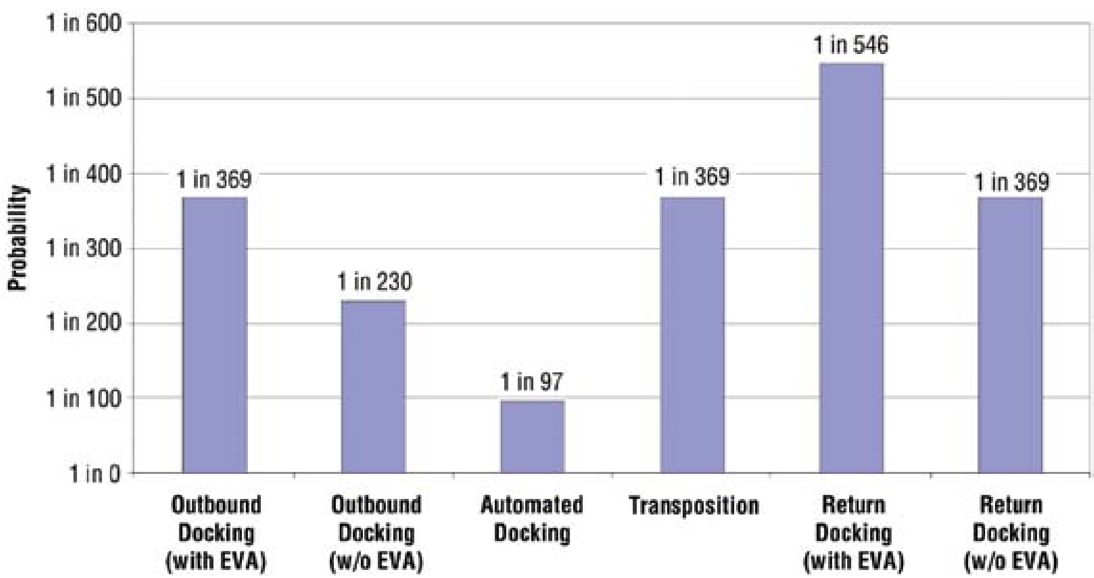


Figure 8-16. Docking Comparisons for Lunar Missions Probability of Docking Failure

8.2 Methodology

The risk assessment methodology for this project was based on a variety of techniques developed over the past few years (References 1 and 2 in **Section 8.7**, **References**). The top-down, scenario-based risk assessment approach utilized by this study is a complex process that incorporates many sources of information to produce a representative analysis. This approach combines modules that represent risk drivers in a transparent fashion so that design teams can easily understand risks, and analysts can quickly generate models. An intensive review of heritage information back to Apollo, past risk assessments, and interaction with vehicle designers and operations experts was performed by experienced analysts to identify risk drivers for ISS and lunar missions. The risk drivers of individual mission elements were combined into models for the specifics of each mission implementation. This initial risk and reliability analysis does not claim to quantify exact estimates of the reliability, instead its goal is to arrive at reasonable estimates that can be used to identify "differences that make a difference." Once these elements are identified, more analysis may be performed if a more exact estimate is required.

Three fundamental mission types were analyzed: (1) LOR, (2) EOR, and (3) direct missions. Also, three alternative propulsion system configurations were analyzed: (1) all pressure-fed LOX/methane, (2) LOX/hydrogen pump-fed lunar descent stage, and (3) pump-fed LOX/methane engines on the CEV Service Module (SM) and lunar ascent elements. Analysis showed the mission modes and propulsion options were the fundamental drivers for the risk assessments. (See **Appendix 8D**, **Mission and ISS Models**.) Once these elements were established, the risk drivers could be assembled and quantified. Once the missions were modeled, an integrated campaign model was developed to assess the integrated risks of the program.

A key aspect of the analysis was the development of maturity models for the early stages of the program. The traffic model for the campaign was combined with the maturity model to account for the benefit of flight operations on later flights. In particular, the maturing of the LOX/methane pressure-fed engine was used as a means of returning from the Moon. The campaign risk model can be used to understand the risks of missions of space flight. These risks can be combined with consequence models to understand the impacts of the risks on achieving NASA objectives and to develop strategies for coping with failures that are likely to occur. These models can discount program and performance costs based on the likelihood of accidents and quantification of risk to the overall program—such as LOC, loss of a key asset, or program cancellation due to unexpected poor performance.

The process architecture is illustrated in **Figure 8-17**. The mission description provided the fundamental basis for the analysis. The mission designers worked with design engineers to create missions that were physically realizable based on the mass and performance capabilities of feasible systems. Abort options were identified as part of the mission design. The mission elements were iterated until a mission could be described in terms of actual systems consisting of vehicles capable of being produced and launched on a feasible launcher. The mission description identified vehicles (propulsion system type, vehicle systems, redundancy). This information was used to create modular element reliability models tailored to key mission events (i.e., engine burns, rendezvous and docking, landing, Earth entry) and associated time durations (if applicable). The missions are described in **Section 8.3**, **Model Elements.**

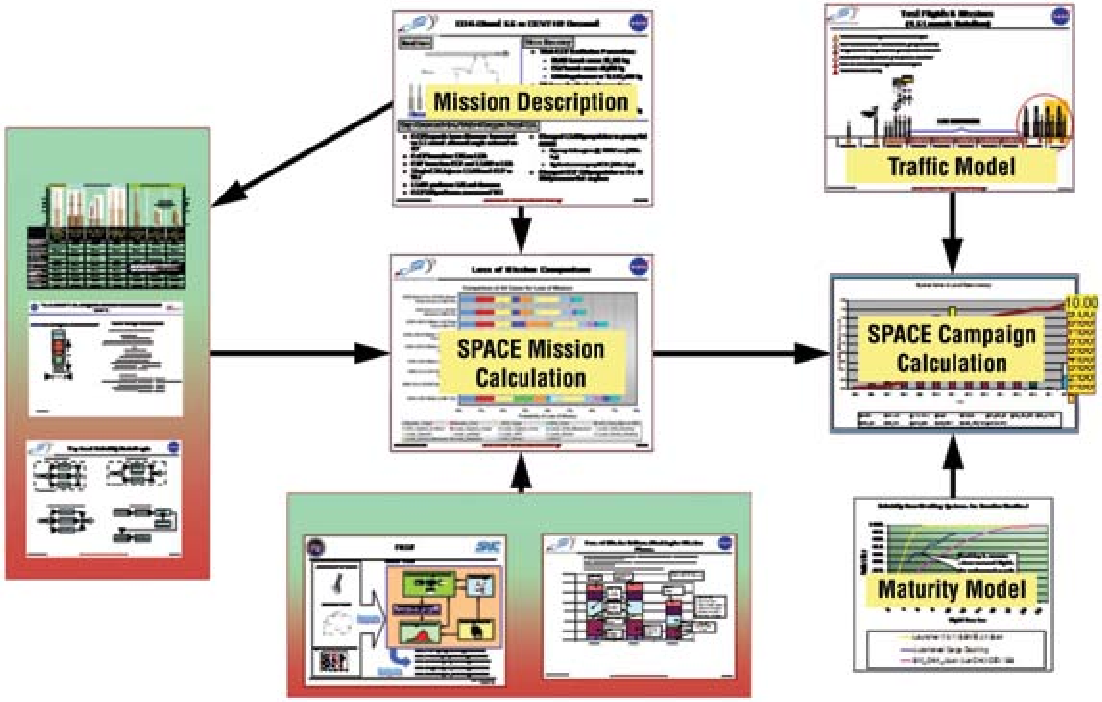


Figure 8-17. Elements of the Risk Model

Heritage information and analysis identified elements that were most likely to contribute to mission risk. These elements are shown in **Table 8-1** for each of the mission modes.

Table 8-1. Mission Elements for Mission Modes

Phase	Mission Element	Mission Mode		
		LOR	EOR	Direct
Launch	Booster Cargo	Variable**	Variable	Variable
	Booster Crew	Variable	Variable	Variable
Low Earth Orbit (LEO) Operations	EDS Burns (Suborbital, Circularization, Trans-Lunar Injection (TLI))	Variable	Variable	Variable
	Crew Vehicle EDS Burns (LOR) or Maneuvers (EOR)	Variable	Variable	Variable
	LEO_Dockings (EOR)	N/A	Variable	Variable
Transit	CEV System to the Moon	Variable	Variable	Variable
Lunar Orbit	Lunar_Orbit_Insertion_Cargo (LOR)	Variable	N/A	N/A
	Lunar Orbit Insertion Crew	Variable	Variable	Variable
	Lunar Orbit Rendezvous Maneuvers (LOR)	Variable	Variable	Variable
	Lunar_Orbit_Docking (LOR)	Variable	N/A	N/A
Lunar Descent	Lunar_Descent	Variable	Variable	Variable
	Lunar_Landing*	Constant***	Constant	Constant
Lunar Operations	Lunar Operations	Variable	Variable	Variable
Lunar Departure	Lunar_Ascent	Variable	Variable	Variable
	Lunar_Ascent_Docking	Variable	Variable	N/A
	Trans-Earth Injection (TEI)	Variable	Variable	Variable
Return	CEV_Return	Variable	Variable	Variable
Entry, Descent, and Landing System (EDLS)	CEV Entry Descent and Landing	Constant	Constant	Constant

** Indicates use of placeholder value as a conservative reliability estimate.*
*** "Variable" indicates element reliability changes with each mission mode.*
**** "Constant indicates element reliability does not change with each mission mode.*

The launch phase contains the booster vehicles. The reliability of the boosters was dependent on the number and type of engines and burn times. Crew survival was dependent on the type of failure mode (i.e., immediate without warning or delayed with sufficient warning for a crew escape system to actuate). Engine-out capability was examined and rejected due to performance requirements, while crew escape was a part of all launcher designs. The details of the launcher analysis are described in **Section 8.3.1**, **Launch Vehicles**.

The next mission phase was Low Earth Orbit (LEO). The LOR mission mode only requires a circulation and TLI burns for the CEV and LSAM. The EOR and direct missions require rendezvous and manual docking of the elements before a single TLI burn. EOR missions with two launches require an additional transposition and docking maneuver for the LSAM and CEV prior to docking with the EDS. All of the activities in this mission phase have abort capability. The reliability of the in-space propulsion systems is documented in **Section 8.3.2**, **In-Space Propulsion Systems**, and the docking maneuver reliability is documented in **Section 8.3.4**, **Reliability Estimates for the Rendezvous and Docking of the CEV and Lunar Mission Architecture Elements**. The reliability of abort capability was estimated at 90 percent. This reliability is judged to be conservative given the proximity of the CEV to Earth and will be refined when a detailed analysis of failure modes is performed.

The transit phase represents the operation of the vehicles on the way to the Moon. The mission modes are configured differently. The main effect of this configuration is the availability of a second habitable crew volume (the LSAM) for the EOR missions, which allows for an Apollo-13-type of abort capability. Recovery after the CEV failure is assessed for each failure mode. The reliability and recovery of the CEV is documented in **Section 8.3.3**, **Mission Elements – CEV, SM, and LSAM Systems Probability Estimates**.

Lunar orbit includes the Lunar Orbit Insertion (LOI) burns for the vehicles and maneuvering and docking for the LOR missions. These events are quantified based on the engine burns employed for specific propulsion systems. Most of the missions employed either the EDS or LSAM for these maneuvers, leaving the SM service propulsion system as a backup for abort. The docking activities were quantified in **Section 8.3.4**, **Reliability Estimates for the Rendezvous and Docking of the CEV and Lunar Mission Architecture Elements**, and the propulsion maneuvers were modeled in **Section 8.3.2**, **In-Space Propulsion Systems**.

The lunar descent phase includes the engine burns for lunar descent documented in **Section 8.3.2**, **In-Space Propulsion Systems**. Abort from lunar descent is possible for all mission modes using the upper stage of the LSAM for LOR and EOR missions and the SM service propulsion system for the direct missions. A reliability of 99 percent was estimated for this event. The reliability of initiating an abort, given engine or landing failure, is assessed to be 90 percent. A more detailed assessment will be performed when additional details of the mission are specified.

The lunar operations phase was modeled for the LSAM for the EOR/LOR missions and the CEV for the direct missions. These models are documented in **Section 8.3.4**, **Reliability Estimates for the Rendezvous and Docking of the CEV and Lunar Mission Architecture Elements**. This analysis considered the operability and recovery failures of the crew habitat. No Extra-Vehicular Activities (EVAs) were modeled. Additional detail will be added to the model when specific activities and modes are identified. Crew survivability after an LOM failure was quantified in a similar fashion to the transit failures, by assessing the ability to return to the CEV for each LOM failure mode. Effects of risk from extension of lunar stay time are modeled in **Section 8.3.5**, **Lunar Surface Stay Risk Change**.

The lunar departure phase includes ascent from the lunar surface and TEI burn by the propulsion systems quantified in **Section 8.3.2**, **In-Space Propulsion Systems**. There is no diverse backup for these events, so LOM failures lead directly to LOC. LOR and EOR missions require rendezvous and docking which is documented in **Section 8.3.4**, **Reliability Estimates for the Rendezvous and Docking of the CEV and Lunar Mission Architecture Elements**.

The return portion of the mission is represented by the CEV modeled in **Section 8.3.3**, **Mission Elements – CEV, SM, and LSAM Systems Probability Estimates**. There is no diverse backup for these events, so LOM failures lead directly to LOC.

The final phase of the mission is the Earth Entry, Descent, and Landing (EDL). This mission phase is assessed in **Section 8.3.6**, **CEV Stability Impacts on Crew Safety During Entry**. There is no diverse backup for these events, so LOM failures lead directly to LOC.

The SPACE model was used to combine the results of the analysis of each element within each mission. The SPACE model directly captured the risk results for each mission element. These elements were identified by event name and then identified in an element database. The SPACE model also provided a way to capture specific engine burn definitions for each stage. These values were input into the FIRST propulsion model which returned the probability that failure occurred in the specified engine burn. The SPACE model extracted the burn failure probabilities from the propulsion database. LOM and LOC probabilities were calculated by summing the event probabilities together. The LOC probability was calculated by multiplying the LOM failure probability by the conditional probability that there is a fatality given the LOM risk. Since the ISS missions matured most of the critical hardware used for the lunar mission, the SPACE model for lunar missions did not vary. A sample SPACE mission model is shown in **Table 8-2**.

Table 8-3. SPACE Campaign Model

EOR–LOR 1.5-launch - Hydrogen Descent						
Generic Event	**Mission Event**	**Event_Name**	**Number**	**LOM**	**Fatal**	**LOC**
Booster _Cargo	Booster _Cargo	Booster-27-3 Cargo	1	0.81%	0%	0.00%
Booster_Crew	Booster_Crew	Booster-13.1	1	0.22%	23%	0.05%
EDS_Cargo	S2–44.9 TLI Burn Eng Out 1.5-Launch	S2–44.9	1,2,3	0.40%	1%	0.00%
EDS_Crew	SM Rendezvous and Docking	SM_EOR_H2 Descent	1,2	0.17%	5%	0.01%
LEO_Dock	LEO_Dock	DOC_Man_Pass	1	0.40%		0.00%
CEV_System_to Moon	CEV/lander trans-lunar coast	CEV_LSAM_EOR_LOR	1	1.00%	8%	0.08%
Lunar_Capture_Cargo			0	0.00%		0.00%
Lunar_Capture_Crew	LSAM perform lunar capture	LSAM_EOR_H2 Descent_Descent	1,2,3	0.11%	10%	0.01%
Lunar_Orbit_Maneuvers			0	0.00%		0.00%
Lunar_Orbit_Docking			0	0.00%		0.00%
Lunar_Descent	LSAM lunar descent 4 5K 2 burns	LSAM_EOR_H2 Descent_Descent	4,5	0.10%	10%	0.01%
Lunar_Landing*	Lander/CEV lunar landing	LSAM-Landing	1	1.00%	10%	0.10%
Lunar_Ops**	Surface mission – 96 hours, 4 EVAs	Lunar_OPS EIRA_EOR_LOR	1	0.47%	82%	0.39%
Lunar_Ascent	Lunar_Ascent	LSAM_EOR_H2 Descent_Ascent	1	0.05%	100%	0.05%
Lunar_Ascent_Docking	Lunar_Ascent_Docking	DOC_Man_Ascent	1	0.26%	100%	0.26%
Lunar_Ascent_Maneuver	Lunar_Ascent_Maneuver			0.00%		0.00%
Lunar_Departure	CEV "ascent stage" (SM) performs TEI burn, CEV trans-Earth coast	SM_EOR_H2 Descent	3,4,5	0.25%	100%	0.25%
CEV_Return	CEV trans-Earth coast	CEV_Return_EIRA_EOR_LOR	1	0.58%	53%	0.31%
EDLS	CEV direct Earth entry	CEV-EDLS	1	0.04%	100%	0.04%

** Indicates use of placeholder values as conservative reliability estimates.*
*** Does not include EVA risk.*

Failure probabilities of ISS missions changed with time as key systems matured. ISS missions were characterized as Launch, Orbital Maneuvers to Station, and Docking (manual for crew, automated for cargo). With the exception of the manual docking mission, a maturity model was used to address the risk of ISS operations while the CEV matured. A maturity model was developed for many different technologies. When the final missions were identified, specific vehicle maturity models were developed. This analysis is documented in **Section 8.6**, **Forward Work**.

The SPACE campaign model integrates all the models together to provide a risk profile for the integrated program. The SPACE campaign model is shown in **Table 8-3**. The traffic model was used to call out the individual missions for each year. CEV missions are captured and mapped into the elements of the maturity model and used to calculate the reliability of each mission as the elements mature. The LOM risk is calculated from the individual mission risk models and integrated with the maturing elements to calculate the expected number of lost missions per year. The LOC model calculates the probability of LOC for each mission for each year. The probabilities are then converted to reliabilities and multiplied together to calculate the probability of total success. The probability of failure is the complement of the probability of success.

Table 8-3. SPACE Campaign Model

Mission Flight Rates														
Missions	**2005**	**2006**	**2007**	**2008**	**2009**	**2010**	**2011**	**2012**	**2013**	**2014**	**2015**	**2016**	**2017**	**2018**
Shuttle	1	3	5	5	3	3	0	0	0	0	0	0	0	0
HTV (H2)	0	0	0	0	1	1	1	1	1	1	1	1	0	0
ATV (Ariane)	0	1	1	1	1	1	1	1	1	1	1	0	0	0
Soyuz	0	1	1	2	2	2	1	0	0	0	0	0	0	0
Progress	0	0	0	3	3	4	5	0	0	0	0	0	0	0
CEV_DEV_SO	0	0	0	0	1	1	2	0	0	0	0	0	0	0
CEV_DEV_ORB	0	0	0	0	0	0	2	0	0	0	0	0	0	0
ISS_UnPress	0	0	0	0	0	0	1	0	1	1	1	1	0	0
CEV_ISS	0	0	0	0	0	0	1	2	2	2	2	2	0	0
ISS_Pres	0	0	0	0	0	0	0	3	3	3	3	3	0	0
Con-1	0	0	0	0	0	0	0	0	0	0	0	0	1	0
Con-2	0	0	0	0	0	0	0	0	0	0	0	0	1	0
Con-3	0	0	0	0	0	0	0	0	0	0	0	0	0	1
Con-4	0	0	0	0	0	0	0	0	0	0	0	0	0	1
Maturity Model														
SM_Orbit_Ajust (LOXCH4)/CEV ISS	20.0%	20.0%	20.0%	20.0%	20.0%	20.0%	20.0%	12.7%	5.9%	1.2%	0.3%	0.3%	0.3%	0.3%
Launcher (13.1)	30.0%	30.0%	30.0%	30.0%	30.0%	30.0%	30.0%	1.5%	0.2%	0.2%	0.2%	0.2%	0.2%	0.2%
Docking_Auto_ station	10.0%	10.0%	10.0%	10.0%	10.0%	10.0%	10.0%	7.2%	2.3%	2.0%	2.0%	2.0%	2.0%	2.0%
Loss of Mission Risk														
Shuttle	0.01	0.03	0.05	0.05	0.03	0.03	–	–	–	–	–	–	–	–
HTV (H2)	–	–	–	–	0.29	0.27	0.26	0.25	0.24	0.23	0.22	0.21	–	–
ATV (Ariane)	–	0.06	0.05	0.05	0.04	0.03	0.03	0.03	0.02	0.02	0.02	–	–	–
Soyuz	–	0.01	0.01	0.02	0.02	0.02	0.01	–	–	–	–	–	–	–
Progress	–	–	–	0.12	0.12	0.16	0.20	–	–	–	–	–	–	–
CEV_DEV_SO	–	–	–	–	0.01	0.19	0.21	–	–	–	–	–	–	–
CEV_DEV_ORB	–	–	–	–	–	–	0.44	–	–	–	–	–	–	–
ISS_UnPress	–	–	–	–	–	–	0.33	–	0.08	0.03	0.02	0.02	–	–
CEV_ISS	–	–	–	–	–	–	0.19	0.22	0.08	0.02	0.01	0.01	–	–
ISS_Pres	–	–	–	–	–	–	–	0.37	0.14	0.08	0.07	0.07	–	–
Con–1	–	–	–	–	–	–	–	–	–	–	–	–	0.03	–
Con–2	–	–	–	–	–	–	–	–	–	–	–	–	0.03	–
Con–3	–	–	–	–	–	–	–	–	–	–	–	–	–	0.05
Con–4	–	–	–	–	–	–	–	–	–	–	–	–	–	0.06
Total Incidents	**0.01**	**0.11**	**0.22**	**0.46**	**0.96**	**1.68**	**3.35**	**4.21**	**4.78**	**5.16**	**5.51**	**5.83**	**5.88**	**5.99**
Loss of Crew Risk														
Shuttle	1.0%	3.0%	5.0%	5.0%	3.0%	3.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%
Soyuz	0.0%	0.3%	0.3%	0.5%	0.5%	0.5%	0.3%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%
CEV_ISS	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	1.9%	2.2%	0.8%	0.2%	0.1%	0.1%	0.0%	0.0%
Con-3	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.6%
Con-4	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	1.5%
Total Success	**99.0%**	**95.8%**	**90.8%**	**85.8%**	**82.8%**	**79.9%**	**78.2%**	**76.5%**	**75.9%**	**75.7%**	**75.6%**	**75.5%**	**75.5%**	**75.5%**
Probability_LOC	**1.0%**	**4.2%**	**9.2%**	**14.2%**	**17.2%**	**20.1%**	**21.8%**	**23.5%**	**24.1%**	**24.3%**	**24.4%**	**24.5%**	**24.5%**	**24.5%**

8.3 Model Elements

8.3.1 Launch Vehicles

A team led by the MSFC Safety and Mission Assurance Office (S&MA) assessed more than 30 LV concepts to determine LOM and LOC estimates. Evaluations were based on preliminary vehicle descriptions that included propulsion elements and Shuttle-based LV subsystems. The team ensured that every analysis used a strictly uniform methodology for combining vetted failure rates and probabilities for each subsystem.

Assessment results were validated using available LV reliability estimates and a simple point-estimate reliability model. Complete descriptions of the analyses methodology, results evaluations, and assessment validations are provided in **Appendix 6D**, **Safety and Reliability**. A complete description of how the team developed reliability predictions for each LV system considered in the similarity analyses is provided in **Section 6.8**, **LV Reliability and Safety Analysis**.

The stochastic LOC and LOM distributions for each of the CLV results are shown graphically in **Figures 8-18** and **8-19**, respectively. **Figures 8-20** and **8-21** show similar graphics for lunar CaLV LOC and LOM. Detailed LV reliability information is provided in **Sections 6.5**, **Crew Launch Vehicle**, **6.6**, **Lunar Cargo Vehicle**, **6.8**, **LV Reliability and Safety Analysis**, and **Appendix 6D**, **Safety and Reliability**.

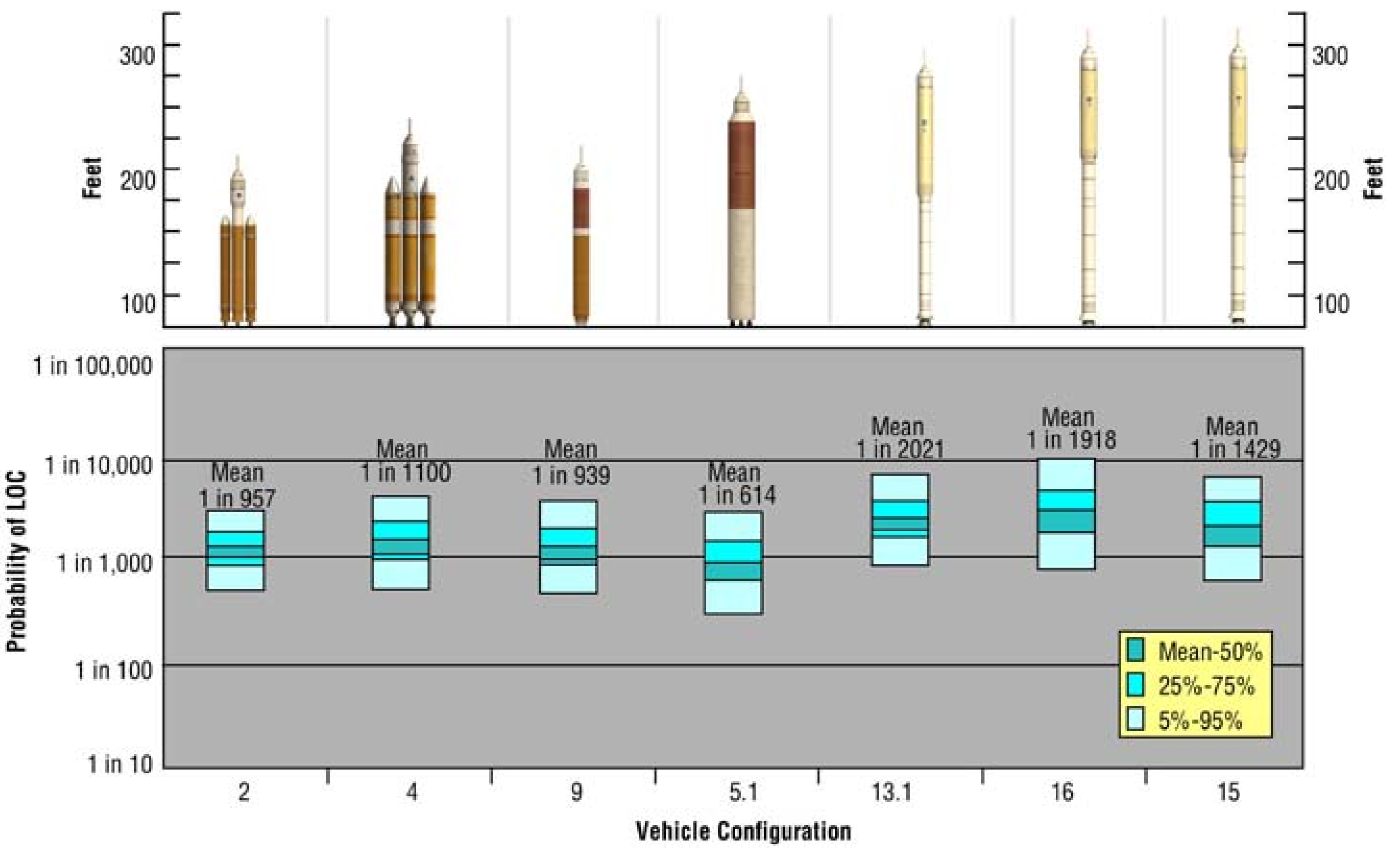


Figure 8-18. CLV LEO Launch Systems LOC

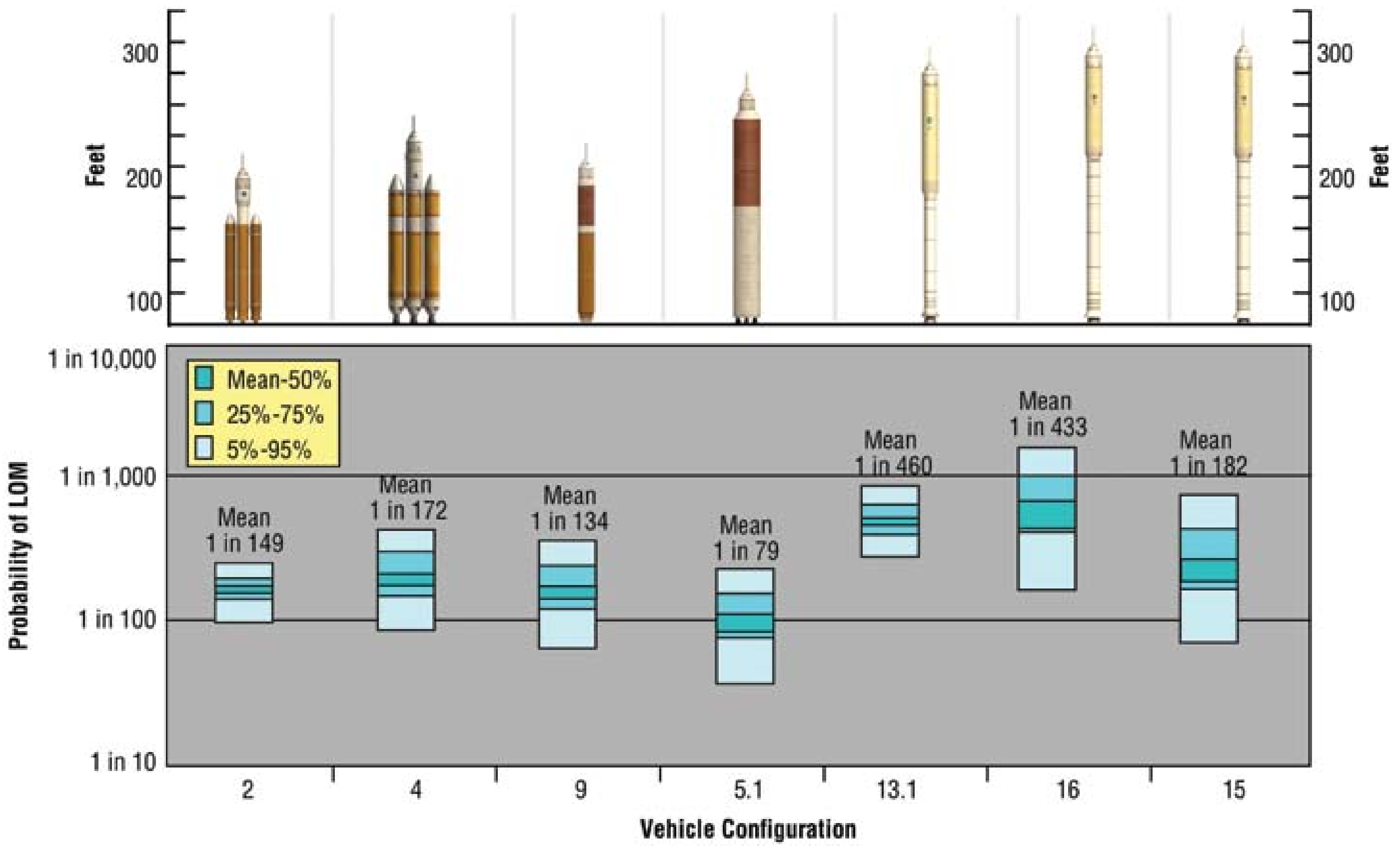


Figure 8-19. CLV LEO Launch Systems LOM

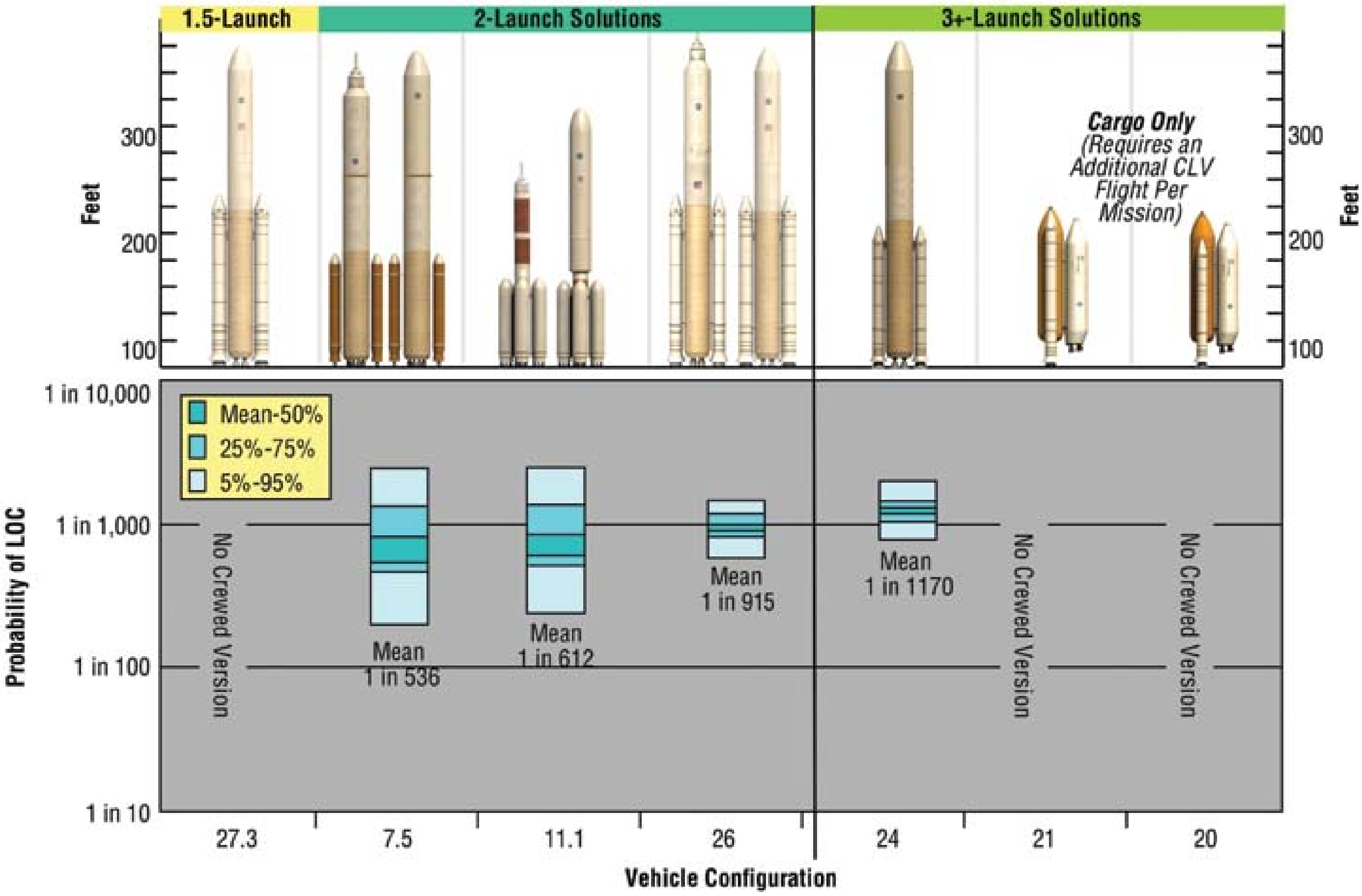


Figure 8-20. Lunar CaLV Launch Systems LOC

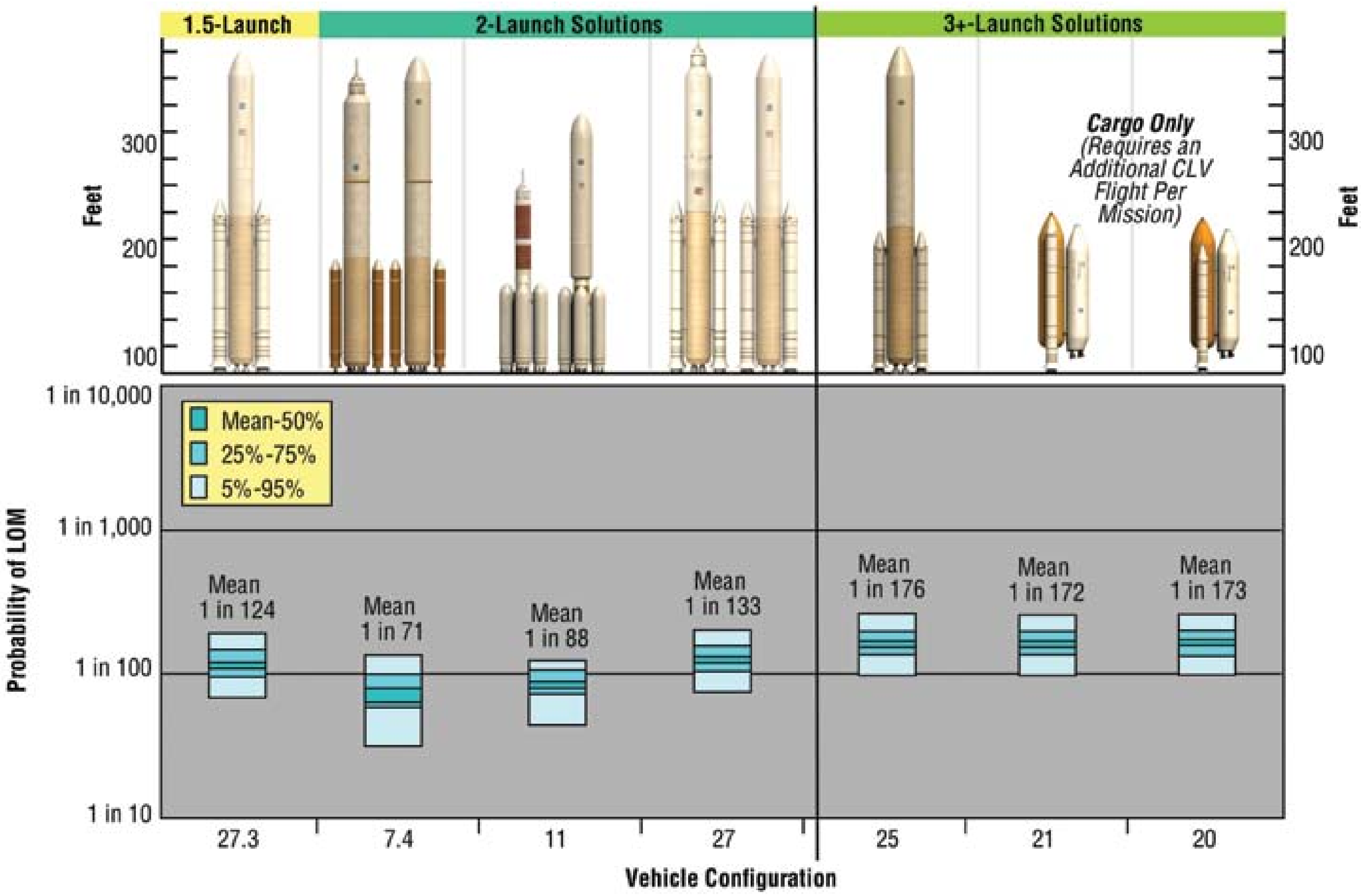


Figure 8-21. Lunar CaLV Launch Systems LOM

8.3.2 In-Space Propulsion Systems

A liquid propulsion system reliability model was developed in support of the ESAS. Reliability trades on number of engines, engine cycle, propellant type, and engine-out scenarios were performed. The model was then used to predict the propulsion stage reliability for specific in-space architecture.

The liquid propulsion system reliability model reflects a systems approach to reliability modeling, i.e., the model simulates an engine in a propulsion system that includes Main Propulsion System (MPS) elements and avionics elements. **Figure 8-22** shows a schematic of the modeled liquid propulsion system. The schematic shows the engine boundaries. For pressure-fed configurations, the engine boundaries contain injector, chamber, nozzle, and igniters. For pump-fed configurations, the engine boundaries also contain the turbopumps and engine propellant valves.

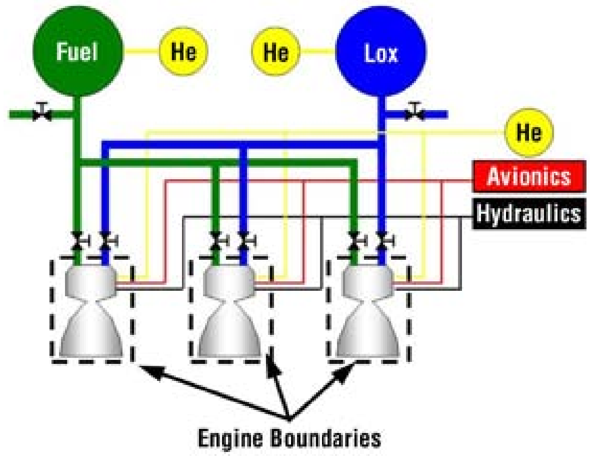


Figure 8-22. Liquid Propulsion System Schematic

The model reflects those physical elements that would have a significant contribution to stage reliability. For example, an engine purge system is indicated because of the potential requirement for restart. However, while a fill-and-drain system would be present physically, such a system would be verified and latched prior to launch commit and, therefore, is not modeled here. Note that, because the engine interface requirements are not known, the avionics, pneumatics, and hydraulic subsystems are modeled as grouped elements. For the purpose of this assessment, MPS refers to the non-engine components of the propulsion system including avionics, hydraulics, pneumatics, and propellant-feed systems.

The liquid propulsion system reliability model described here is an event-driven, Monte Carlo simulation of the schematic shown in **Figure 8-22**. For each event, the cumulative failure distribution is randomly sampled to obtain a time-to-failure. The time-to-failure is compared to mission burn time. If the time-to-failure is less than the burn time, a failure is recorded. The event logic for the reliability model is shown in **Figure 8-23**. Note that parallel events indicate that a failure in any one path is a system failure, as indicated on the event tree as an "OR" failure scenario. **Figure 8-23** shows the top-level events where the engine cluster is modeled in parallel with failures in the purge system and external leakage events. **Figure 8-23** shows the breakdown of the cluster where each engine is modeled along with support systems. It also shows the further breakdown of the engine support systems to include the avionics, pneumatics, and hydraulics provided to the engines. **Figure 8-23** shows the sequence of events modeled at the individual engine level to include isolation valve failures and engine start and main stage failures. All steps must be successful for a successful engine burn.

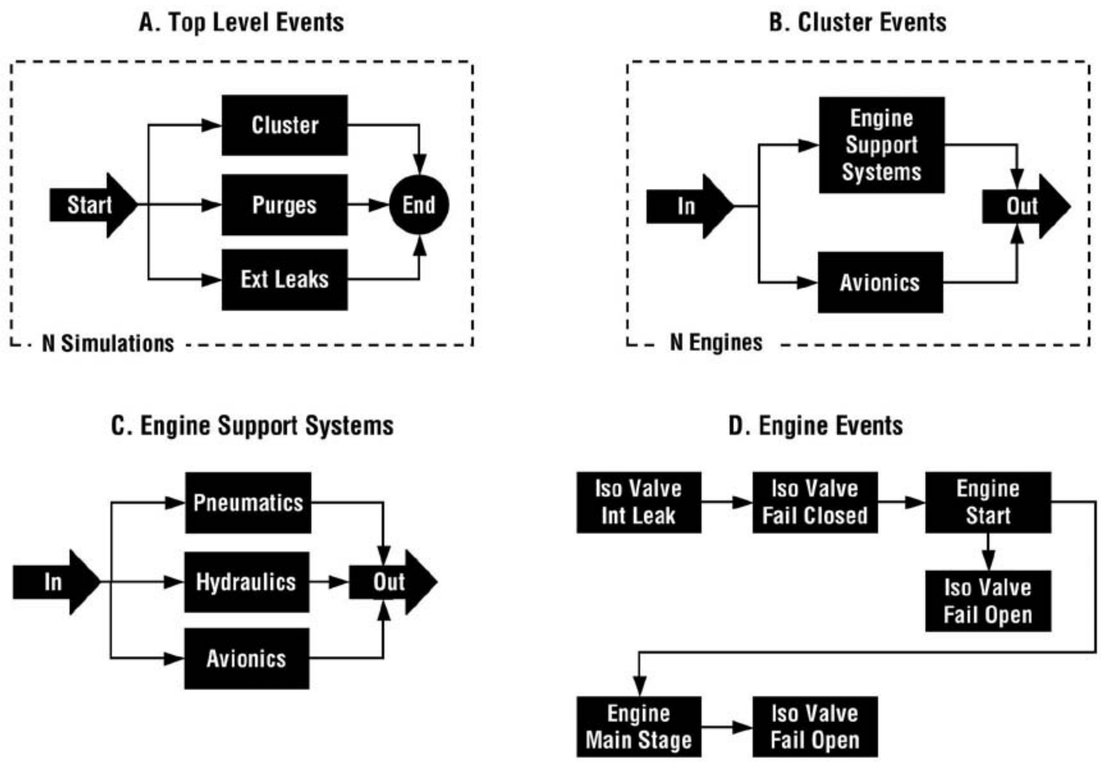


Figure 8-23. Event Logic Model

For engine-out cases, if a first benign failure is recorded, then the burn time is scaled by the ratio of the original number of engines divided by the number of operational engines remaining. The time-to-failure for the remaining operational engines are compared to this new extended burn time. If the time-to-failure of any one of the remaining operational engines is less than the new extended burn time, then a stage failure is recorded.

8.3.2.1 Data Sources and Event Quantification

The data source for quantifying the non-engine events is the Space Shuttle Probabilistic Risk Assessment , Iteration 2.0 (Reference 3 in **Section 8.7**, **References**). The one exception is that the avionics failure rates for the Space Shuttle Orbiter were not available; the engine controller failure rates for the SSME were used instead. **Table 8-4** shows the failure parameters that were used for quantifying the non-engine failure events. The effect of using Space Shuttle data to quantify event probabilities is that Space Shuttle design and operational philosophies are inherently assumed.

Table 8-4. Non-Engine Failure Event Parameters

Event	Number Per Engine	Distribution Type	Distribution Parameters
Purge Valve Failure	2	Weibull	Shape = 0.5 Scale = 8.02×1012
External Leakage	6	Weibull	Shape = 0.5 Scale = 1.73×1012
Pneumatic System Failure	1	Weibull	Shape = 0.5 Scale = 5.12×1018
Hydraulic System Failure	1	Weibull	Shape = 0.5 Scale = 5.12×1018
Avionics System Failure	1	Weibull	Shape = 0.5 Scale = 1.14×1011
Isolation Valve – Internal	2	Demand	Mean = 3.15×10^{-6}
Isolation Valve – Fail Open	2	Demand	Mean = 3.88×10^{-4}
Isolation Valve – Fail Closed	2	Demand	Mean = 2.23×10^{-4}

For pump-fed engine cycles, a similarity analysis using SSME as a baseline was performed to obtain main stage engine failure rates. The isolation valve failures events represented valves with redundant actuation (i.e., SSME valves). The similarity analysis provided main stage engine catastrophic failure probability per second and the catastrophic failure fraction. For pressure-fed engine cycles, the Space Shuttle Orbital Maneuvering System (OMS) was used as a baseline. The data source for the failure rates for the OMS was the Space Shuttle Probabilistic Risk Assessment, Iteration 2.0 (Reference 3 in **Section 8.7**, **References**). For a single OMS thruster, a catastrophic failure probability of 1.03×10^{-6} is predicted for a typical four-burn mission. Each burn was assumed to be 200 sec. This results in a per-second catastrophic failure probability of 9.72×10^{-9}. **Table 8-5** shows the engine failure parameters used for this study.

Table 8-5. Engine Failure Parameters

Engine	Pstart	Pcat/s (First Launch)	Pcat/s (Mature)	CFF
In Space Stages				
LH–10K	0.0001	–	1.89^{-07}	0.05
LH–15K	0.0001	–	1.97^{-07}	0.05
LH–20K	0.0001	–	2.03^{-07}	0.05
LM–10K Pump	0.0005	–	1.89^{-07}	0.05
LM–15K Pump	0.0005	–	1.97^{-07}	0.05
LM–20K Pump	0.0005	–	2.03^{-07}	0.05
LM–XK Pressure-fed	0.0005	–	9.72^{-09}	0.25

The difference in Pstart (probability of engine start) between the hydrogen and methane engines is an impact of the propellant. LOX/methane flammability limits are only 58 percent as wide as LOX/LH2. Thus, tighter mixture ratio control is required for LOX/methane systems. Additionally, the minimum ignition energy for LOX/methane is an order of magnitude higher than for LOX/LH2. Thus, higher performing spark igniters are required for LOX/methane. A technology development program would most likely reduce the fail-to-start probability for a LOX/methane system. However, until such a program can be completed, the benefit of such a program cannot be incorporated into the analysis. The contribution to stage unreliability of main stage engine failures is much less for the pressure-fed configurations as compared to the pump-fed configurations. This is in keeping with the data from the Space Shuttle, which indicated that pressure-fed engines are more reliable.

8.3.2.2 Architecture Case Study

Many alternative propulsion options were considered during the ESAS. The possibility of engine-out capability was considered for pump-fed stages due to the significantly higher failure rates for pump-fed systems. However, because of the smaller mass and volume of the LSAM ascent stage and the restrictions on thrust vectoring the ascent engine(s), it would be physically difficult to implement engine-out, even if there were theoretical reliability benefits. That is, there would be great difficulty maintaining the thrust vector with one engine because of thrust imbalances without a significant increase in RCS capability. Therefore, the only stage that had engine-out was the LOX/hydrogen LSAM descent stage.

A reliability case study of the in-space propulsion stages for the 1.5-launch configuration was performed. **Figure 8-24** shows the results of the architecture case study both by stage and burn.

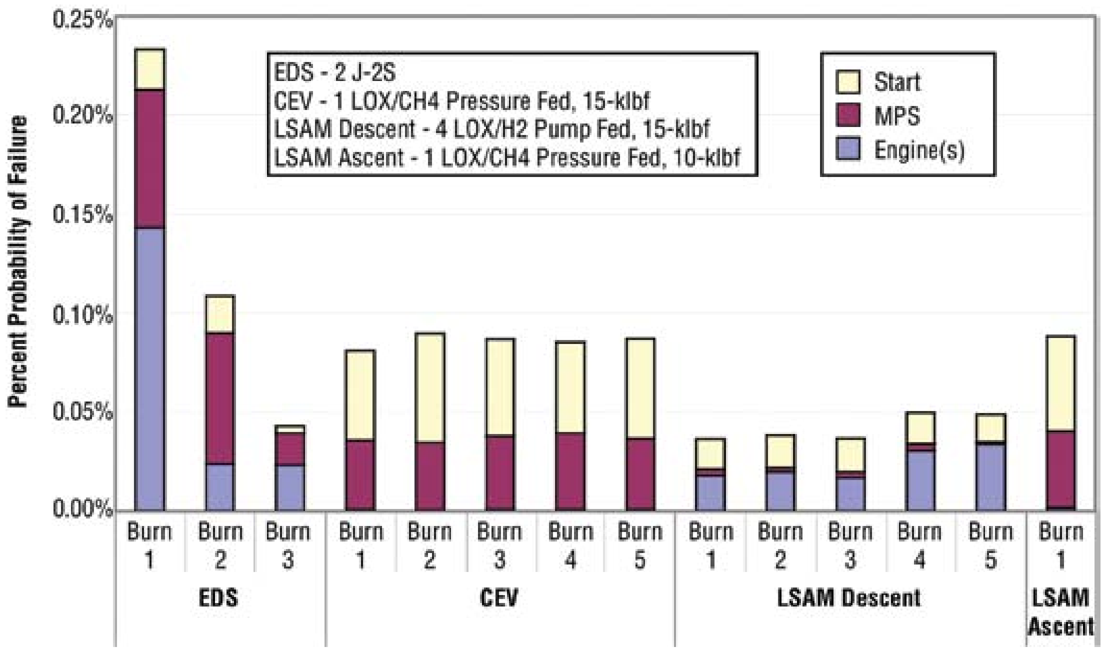


Figure 8-24. Propulsion Stage Reliability Results for the 1.5-Launch Configuration

The risk associated with first burn of the EDS is dominated by failure of the two engines operating without engine-out (61 percent). The remaining risk for the first burn of the EDS is associated with the MPS (30 percent) and engine-fail-to-start (9 percent). The second burn of the EDS is a short burn, with MPS comprising the bulk of the risk (61 percent). The third burn of the EDS incorporates an engine-out capability, with the dominant risk being the two engines (53 percent) and the MPS (37 percent).

The risk of the LOX/methane stages was dominated by failure to start (55 percent) and failure of the MPS propellant isolation valves to open and close (45 percent). Consideration was given to adding redundancy to the valves, but the additional complication in the system was seen to be adding additional failure modes that would offset the benefit.

The LSAM descent stage with engine-out was dominated by double start failures (30 to 45 percent) and catastrophic engine failures (45 to 70 percent) because engine-out effectively eliminates benign engine and MPS (isolation valve) failures from contributing to system failure. It was assumed that the gap between Lunar Orbit Insertion (LOI) and lunar descent burns would not affect engine reliability during descent. Some architectures employed two pressure-fed LOX/methane engine systems without engine-out capability. The start and MPS failures (no engine-out) caused this configuration to be approximately a factor of 3 less reliable than the LOX-fed system with engine-out.

Results indicate that pressure-fed configurations are significantly more reliable than pump-fed configurations with like number of engines and no engine-out capabilities. Results also indicate that detailed vehicle/mission/architecture design studies are needed to determine if the benefits of additional redundancy (engine-out, redundant valves, etc.) will have significant reliability benefit.

8.3.3 Mission Elements – CEV, SM, and LSAM Systems Probability Estimates

For this study, the subsystem descriptions supporting the failure rate estimates were obtained directly from members of the ESAS team, with a significant level of interaction and iteration. Failure rate estimates for these subsystems are derived whenever possible from other space subsystem applications; otherwise, surrogate data is used. Each estimate is a failure-per-hour unless the mission event is a demand; then the failure probability is listed as a demand (e.g., parachute deployment). No attempt is made in this assessment to develop exact failure rates for every system, nor is there an uncertainty applied to the numbers. Therefore, this data should not be assumed to be detailed subsystem failure rates. The intent is to apply an estimate for a top-level mission architecture comparison. Propulsion system dormant states are given failure rates for estimated dormant failures (e.g., leakage of propellant). Engineering judgment is employed to provide sanity checks for the probability values. LV and propulsion burn failure rates are estimated in **Section 8.4**, **Architecture Summary**.

For mission phases where the mission element is uncrewed, the element is assumed to be quiescent, or in a powered-down state, where some of the subsystems are removed from the list. The LSAM is considered quiescent for Case 1 outbound to lunar orbit until docking with the CEV/SM, and also in Case 2 outbound while docked with the CEV/SM. The CEV/SM is only quiescent in Cases 1 and 2 when it is unoccupied in lunar orbit. The estimate of nominal mission phases times are: 24 hours in LEO, 96 hours for lunar transit, 24 total hours for lunar orbit operations, 96 hours lunar surface mission, and 96 hours return-to-Earth—for an approximate 14-day mission. Data is selected from the list for the different mission phases depending on the subsystems that are assumed to be operational during those phases.

The CEV, SM, and LSAM elements reliabilities are aggregated for three basic mission cases:

- Case 1: CEV/SM parallel transit Moon with a quiescent pressurized LSAM, docking in lunar orbit for crew transfer, and LSAM to surface with quiescent CEV/SM in lunar orbit.
- Case 2: CEV/SM launched together, docking with a quiescent pressurized LSAM in LEO, and quiescent CEV/SM in lunar orbit.
- Case 3: CEV/SM docking with an unpressurized LSAM in LEO and direct mission with CEV/SM/LSAM.

Figures 8-25 and **8-26** show the results for LOM and LOC. The LOM includes the LOC within its results. Cases 1 and 2 are virtually equivalent for LOM risk because a failure of a subsystem may simply preclude the completion of the mission. However, Case 2 has a much lower potential LOC risk since it has the LSAM as a “lifeboat” should a critical failure occur with the CEV/SM. Case 3 appears to have a better chance for mission success simply because of the fewer number of subsystems in the LSAM. For LOC, Cases 2 and 3 are roughly equivalent where the LSAM “lifeboat” capability of Case 2 equals the reduced number of subsystems and eliminated return docking of Case 3. Redundancy is taken into account by, in most cases, allowing for a two-fault tolerant system with the critical systems.

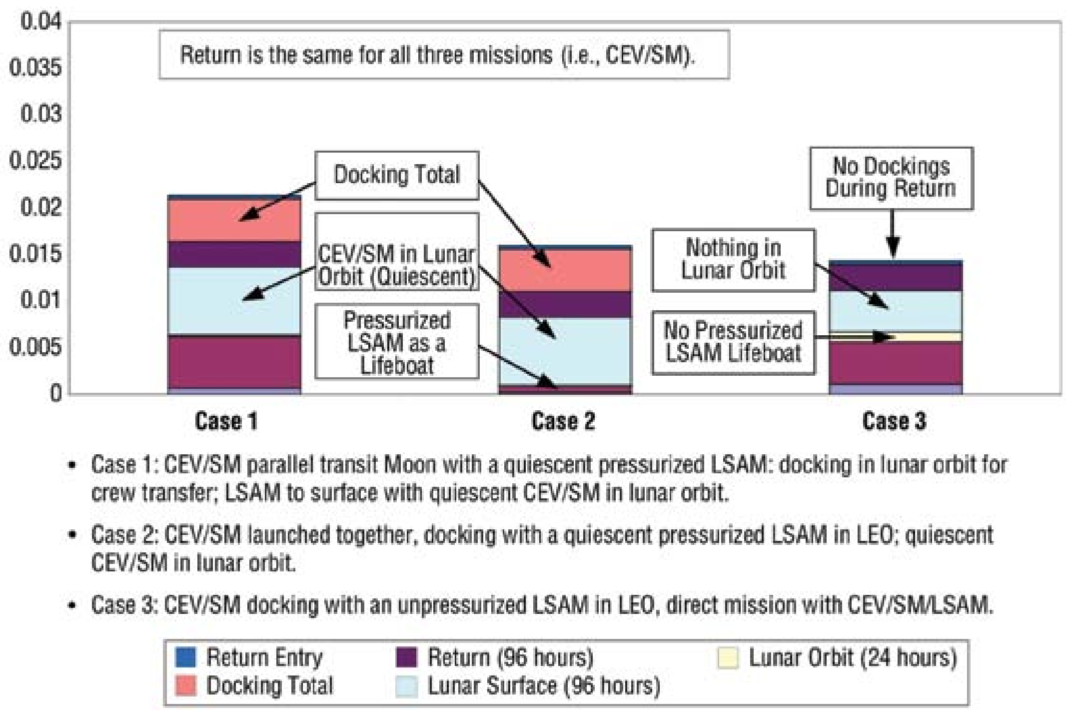


Figure 8-25. Catastrophic Failures Aggregation for CEV/SM and LSAM Mission Phases

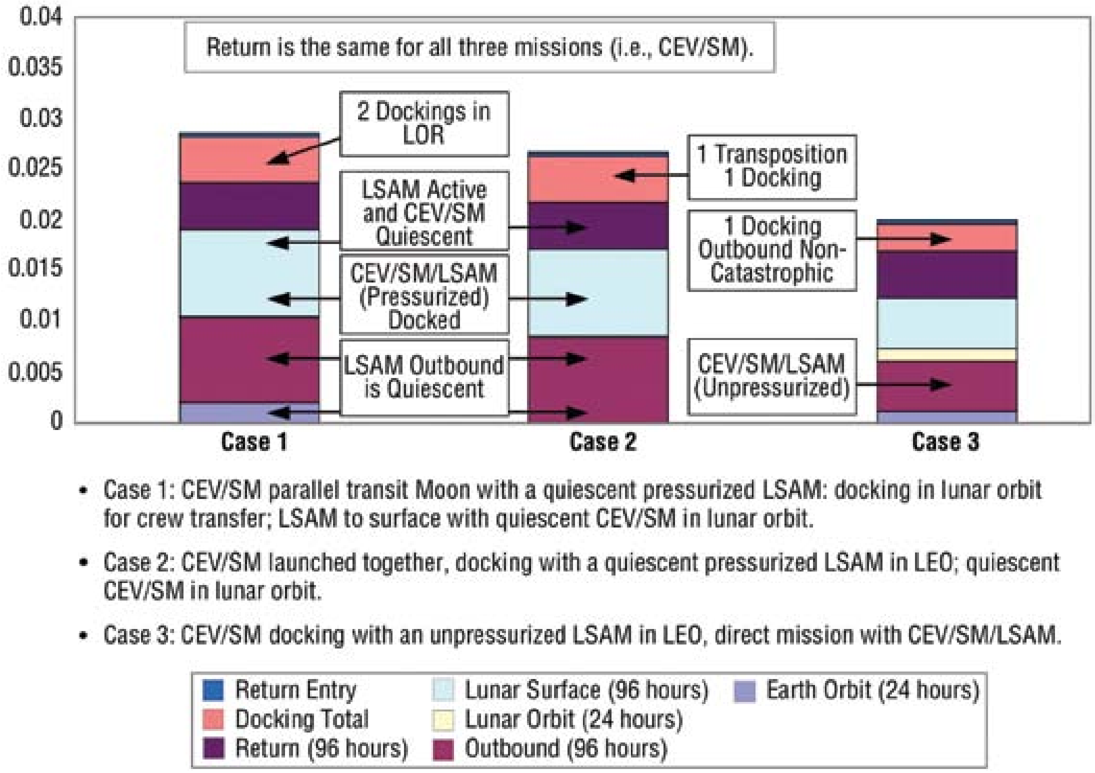


Figure 8-26. LOM Failures Aggregation for CEV/SM and LSAM Mission Phases

8.3.4 Reliability Estimates for the Rendezvous and Docking of the CEV and Lunar Mission Architecture Elements

Historical accounts of rendezvous and docking by spacefaring nations provide an assessment of the reliability of conducting these sequences of events. However, the differences between the Russian Space Agency and NASA rendezvous-and-docking mission success requires a more in-depth review of the failures, precursors to failure, and rendezvous and docking technology. Both agencies have had a relatively large number of precursors to failure; however, U.S. missions have succeeded in applying contingency and malfunction procedures to achieve a 100-percent success rate. The conceptual lunar missions using the CEV will require rendezvous and docking maneuver events that are conducted in Earth or lunar orbit. Given the 2-launch solution and an LOR for the return mission, there are two types of rendezvous and docking maneuvers that will take place with differing contingency measures. These maneuvers are shown in **Figure 8-27**.

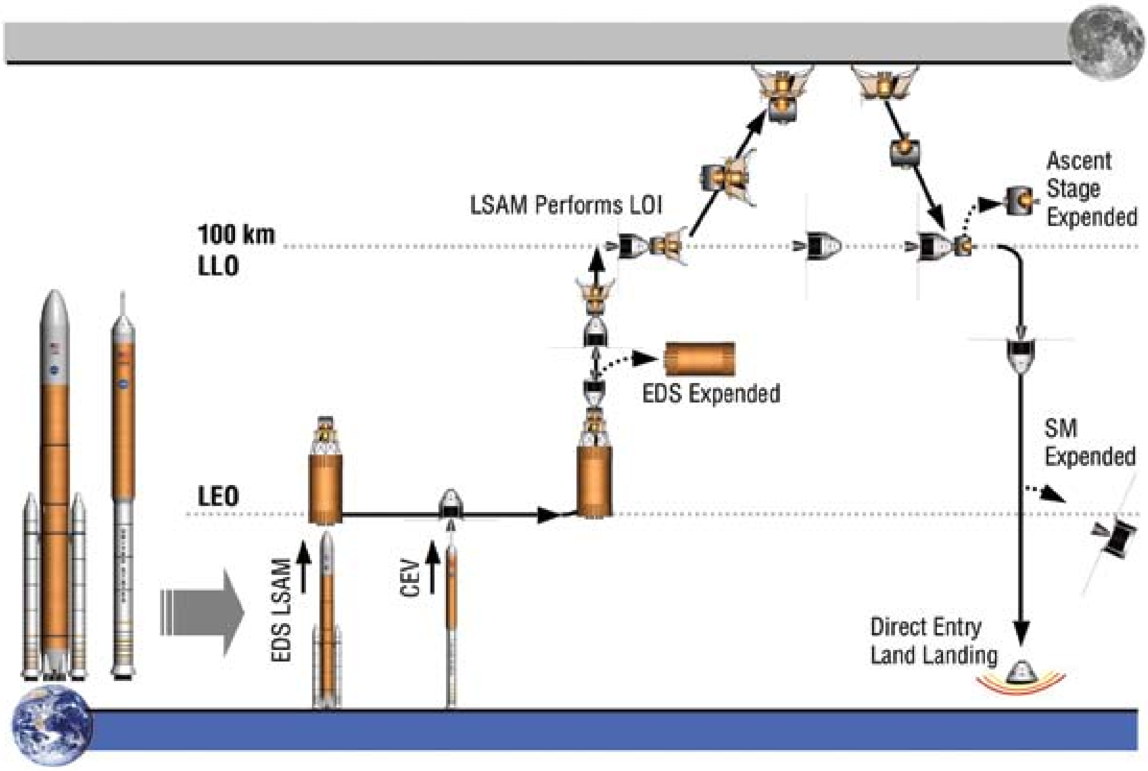


Figure 8-27. Lunar Mission Requiring Rendezvous and Docking Maneuvers

This analysis will assess the risk of failing to rendezvous and dock in Earth orbit and on the return mission. If a failure to dock occurs on the outbound leg of the mission, the mission may abort and return to Earth, depending on the perceived risk of continuing the mission. If an initial failure to dock occurs on the return leg, many more contingency procedures will likely be attempted to save the crew.

8.3.4.1 Rendezvous and Docking Mission Sequence

For this analysis, a mission sequence for a mission rendezvous and docking is developed in the form of a mission Event Sequence Diagram (ESD) and, in turn, converted to an event tree. Even though there are many detailed steps that must be accomplished prior to a hard docking, these steps can be broken down into any acceptable level of resolution. In this case, the steps are: (1) Rendezvous, (2) Proximity Operations, and (3) Docking. (An introduction to rendezvous and docking can be found in References 4 and 5 which are identified in **Section 8.7**, **References**. (See **Appendix 8E**, **Reliability Estimates for the Rendezvous and Docking of the CEV and Lunar Mission Architecture Element**.) In past missions, failed docking mechanisms or processes have been replaced by exceptional events, especially in the proximity operations and dockings. Rendezvous techniques are well established and are conducted by either an approach from below the target vehicle's orbit or from above. Primary systems used to conduct the rendezvous maneuver include the data processing system, electrical power distribution and control, digital autopilot, Star Tracker, Ku-band radar, translational and rotational hand controllers, cameras, Inertial Measurement Units (IMUs), general-purpose computer, and crew optical alignment sight. Each of these subsystems has some form of redundancy, which is referred to as the secondary rendezvous. These redundancies may be like or unlike. **Figure 8-28** shows an ESD of a typical rendezvous and docking maneuver with contingencies for failed events in the outbound leg of the lunar mission. In an ESD, arrows to the right are considered a success of the event and arrows downward are considered a failure. The ESD can be directly converted into an event tree.

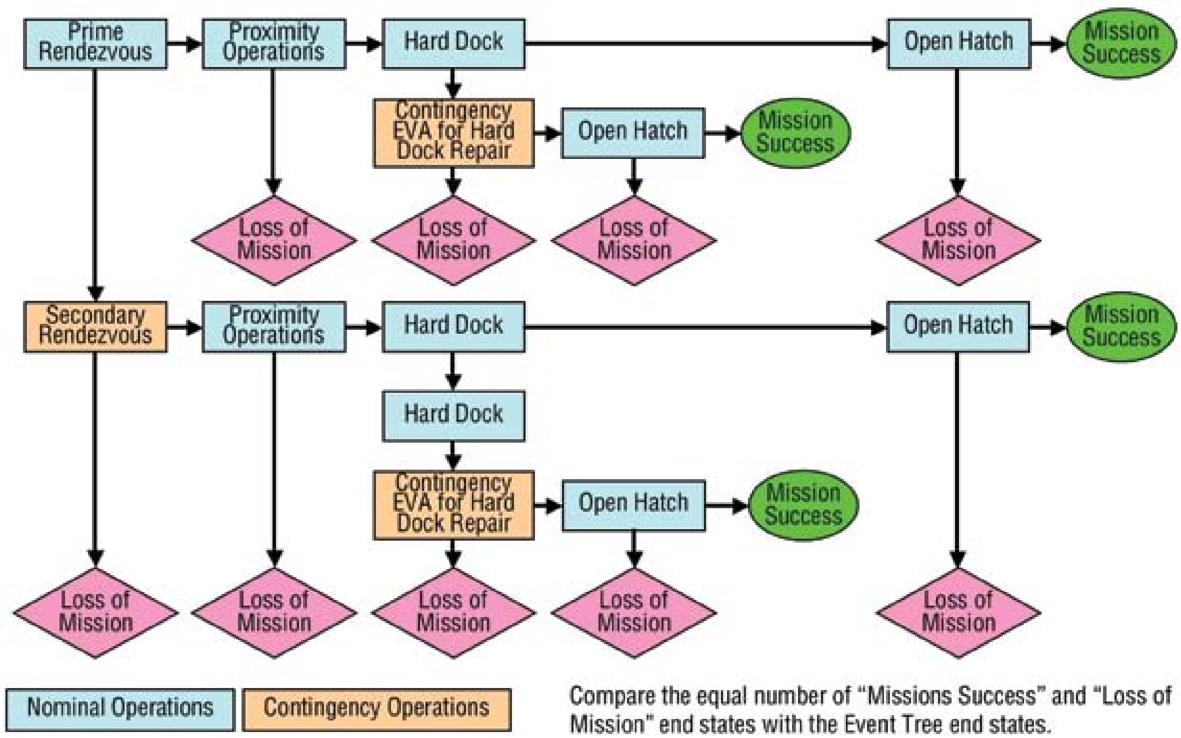


Figure 8-28. ESD for a CEV Lunar Mission Outbound Rendezvous and Docking

In the event of an unstable target vehicle in the return mission (e.g., a CEV that is oscillating due to a malfunctioning thruster or some propulsive venting), requirements may be in place to dampen the oscillations remotely, or the crew could take action to arrest the CEV using EVA. This would likely not be easy since it would require a set of thrusters in the EVA backpack that may not be required for a nominal mission. Contingency EVA procedures will likely be in place for anomalous conditions during the return mission that may not be attempted during the outbound leg of the mission. Returning from the Moon could take this propensity to an extreme if numerous failures occur. The crew will try many more options to get back to the CEV even with a seemingly unattainable transfer. When these challenges are included in the return mission, many more options will be included in the flight rules, and the ESD will appear as in **Figure 8-29**. Most of the events in this ESD are for contingency EVA which is employed to stabilize or repair the docking mechanisms or open the hatch between the elements.

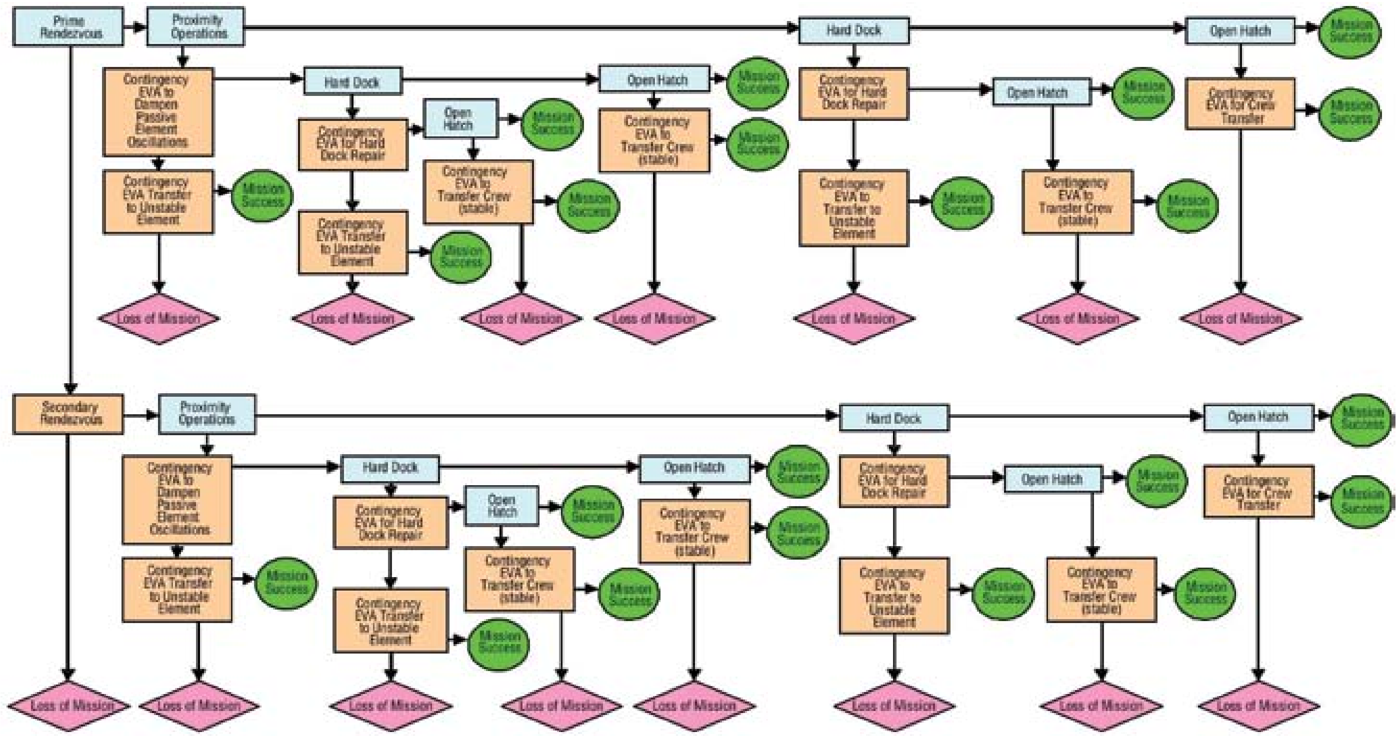


Figure 8-29. ESD for a CEV Lunar Mission Return Rendezvous and Docking

8.3.4.2 Rendezvous

The prime rendezvous event is assumed to have the same technology as the current Space Shuttle, i.e., the Ku-band antenna used in the radar mode to track the target vehicle. Should a failure of the Ku-band occur, the rendezvous can be conducted using Star Trackers or with upload commands from Mission Control. The Shuttle rendezvous radar supplies range, range rate, and angular measurements. The radar system does contain single point failures. In the event of radar failure, two Star Trackers are available to conduct the rendezvous, and the crew is trained in these procedures. In this model, the Ku-band is assumed to be the primary rendezvous equipment, and the Star Trackers are assumed to be the secondary rendezvous equipment. Data exists for each of these components; however, the history of failure and the available failure rate is not that consistent. One Ku-band failure in the radar mode has occurred during STS–92, and one Star Tracker failed during STS–106. There have been other Ku-band antenna failures, but these failures would not have affected a rendezvous. A "resource-rich" spacecraft such as the Space Shuttle is equipped with redundant systems and more than adequate margin in thermal control, power storage and generation, propellant, on-board computer capacity, and communications bandwidth. These redundant systems and margin work together to enhance mission success and to lower mission risk in the presence of failures and off-nominal conditions. This approach has a lot to do with the success rate of NASA rendezvous and dockings. Therefore, the actual Ku-band and Star Tracker data for the prime and secondary rendezvous will be used in this assessment. With the existing failures over 113 missions, the values are 0.991 for the Ku-band antenna and 0.99992 for the redundant Star Trackers.

8.3.4.3 Proximity Operations

Since there are several unlike redundancies for proximity operations, the probability for failure will be accounted for with an "and" gate in the event tree of the Ku-band radar, crew optical alignment sight, Ku-band radar, hand-held lidar (i.e., Light Detection and Ranging), trajectory control sensors, ranging-rulers and overlays, and the laptop computers used to process data. The failure rates for these components are shown in **Table 8-6** (Reference 6 in **Section 8.7**, **References**). A Global Positioning System (GPS) may be included in this list of navigation instruments; however, current GPS technology is not accurate enough to use for proximity operations. The resulting probability of failure of all proximity operations navigation instruments calculated at 6.4^{-7} is quite low; however, there are very few common cause failures with these instruments. The next driving system risk will likely be the RCS.

Table 8-6. Proximity Operation Navigation Instruments

Proximity Operation Instruments	Failure Rate for 113 Shuttle Flights
Ku-band Radar	0.00885
Trajectory Control Sensor	0.0619
Hand-held Lidar	0.0442
Star Tracker	0.0265
Probability (product of all failure rates)	6.4^{-7}

Current vernier RCS failure rates for all Shuttle missions indicate a failure rate of 4.42 percent for an RCS jet off (i.e., a failure of the RCS jet to burn when commanded); however, there are several primary jets in each axis (Reference 6 in **Section 8.7**, **References**). The vernier jets are not redundant, but the primary jets have multiple redundancies that could be used to abort proximity operations. For example, in the case of a closing rate and range where a collision is imminent (possibly due to a crew failure), a failure of all RCS jets in the forward firing axis can be estimated. For the CEV, a required two-fault tolerant system and a non-hypergolic thruster system will reduce the probability of failure for the RCS because the Shuttle hypergolic system is susceptible to leakage from the oxidizer. Even with a 4.42-percent failure rate, the resulting probability of system failure is 3.8^{-6} with three additional levels of redundancy. So, the next candidate for failure of proximity operations is human error.

The process for rendezvous and docking is tightly controlled by mission control, but it can be assumed that the crew is performing the proximity operations on their own. There is no data for proximity operations for human error, so the Shuttle PRA estimate for the crew failure to fly the Heading Alignment Circle (HAC) is used as the surrogate data. The HAC procedure is performed just prior to landing, and the value is estimated at 4.6^{-5} by a flight crew representative from the Shuttle PRA. The result is a summation of the values above ($6.4^{-7} + 3.8^{-6} + 4.6^{-5} = 5.0^{-5}$), which is still quite low.

When rendezvous and docking are assessed, however, it should be apparent that two vehicles are involved, and that the target vehicle cannot always be assumed to be stable. In June 1997, a power failure aboard the Mir space station was reported in which the ship's computer disconnected from the control system overnight after some critical batteries ran low. A month later, the stabilizing gyroscopes that point the Mir toward the Sun shut down temporarily. Still later, the Mir lost power after a vital computer cable was accidentally disconnected, sending the Mir into free drift.

Leaky thrusters or uncontrolled venting can also send a vehicle into oscillations. In a case involving STS–72, a remote-manipulator-system-deployed satellite experienced unexpected propulsive venting that caused trajectory dispersions. On STS–52, there was a strong correlation between periods of increased RCS propellant consumption to maintain attitude and operation of the Flash Evaporator System (FES). Later analysis showed that FES impingement on the elevons produced a pitch moment. Gemini 8 was probably very close to disaster when a reaction control thruster was stuck open. Based on historical failures, the ESAS team estimated that a half a failure will occur for 200 rendezvous missions and that there is a probability of 1/400 for encountering an unstable target vehicle. When this is substituted into the other probabilities, the resulting value is 0.99745 for a proximity operations success.

8.3.4.4 Docking and Hard Docking

Several docking mechanisms have been used throughout spaceflight history, ranging from the simple to the very complex. Currently, the CEV is planning to employ a Low-Impact Docking System (LIDS). Of course, there is no direct docking failure history for this mechanism, so other docking systems must be assessed. All docking systems employ some type of mechanical latch and a motorized mechanism to pull the two vehicles together. Soyuz docking systems using probe and cone have an estimated failure rate of 3.27^{-3}, which is probably reasonable given the scattering of initial docking failures that have occurred from Gemini through Shuttle and from Soyuz through Progress, and including all of the initial failed capture and berthing events. The ESAS team estimated a probability of success of approximately 0.9967.

8.3.4.5 Contingency EVA

In a review of the previous Skylab mission where a contingency EVA was performed to resolve a serious docking problem and other contingency EVAs conducted for capturing satellites, it is obvious that U.S. space missions have gone to great lengths to ensure mission success, even when aborting the mission was certainly an option. There have been cases where planned EVAs were performed to capture a rotating satellite (e.g., Westar and Palapa II). For the CEV, this capability is not required; however, there will be contingencies for the return rendezvous since the only way to get back to Earth is with the CEV/SM. The only EVA where the Lunar Excursion Module (LEM) and Command Module were attached was Apollo IX, when the lunar rendezvous was tested in LEO. Both Command Module and LEM hatches were open, with crew members emerging from each hatch at the same time, but no transfer was made outside the spacecraft, even though it probably could have been done. Even with the possibilities so high, the estimate of a contingency EVA transfer probability of success to a docked vehicle is 50 percent and is 25 percent for a drifting vehicle. As discussed before, there may also be some failure modes of the docking hardware that can be repaired by EVA, as with Skylab.

8.3.4.6 Automated Docking

The only automated docking maneuvers of any statistical note are the Russian Progress vehicles, which began in 1978 with Progress 1 docking with Salyut 6 (though the first automated docking was with a Soyuz vehicle in 1975). The docking mechanism is the same probe and cone as on the Soyuz. Very little detailed history exists for the Progress vehicle in the early years of rendezvous and docking, and all of the failures listed in historical accounts have occurred since 1990. This is probably an artifact of the data collection process because more recent dockings have been covered by non-government news media representatives who have access to this information. The set of data indicates that, of 102 Progress rendezvous and dockings, there have been 93 successes, 8 significant anomalies (which include collision with space station equipment), and 1 failure (the infamous collision with the SPEKTR module on Mir). NASA has performed many automated or remotely controlled rendezvous but few, if any, automated dockings. The model was used with an assumed Ku-band antenna, but no Star Trackers, which effectively removes the secondary rendezvous option. When the model was implemented, the probability of success is similar to the Progress results, which are approximately 0.99.

8.3.4.7 Results of the Rendezvous and Docking Model

The inclusion of contingency events to ensure the success of rendezvous and docking is assumed because great lengths have been employed in past missions to achieve mission success. In many cases, ground support and simulations have also contributed to mission success. This support cannot be ignored, and, with enough resources, docking success should not be a major risk driver. **Figure 8-30** shows a comparison of probability of docking failure for lunar missions.

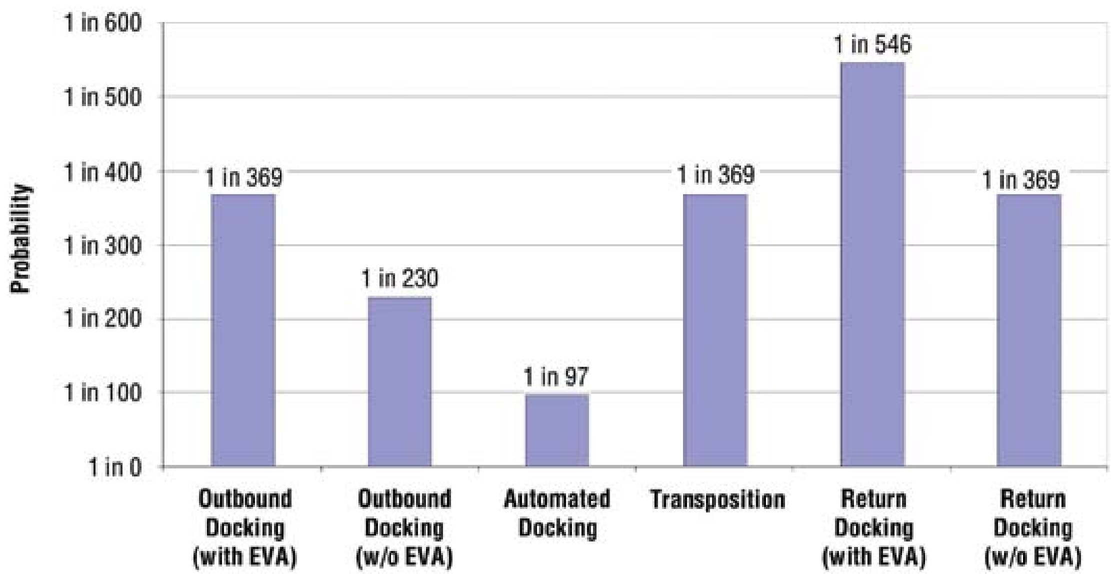


Figure 8-30. Docking Comparisons for Lunar Missions Probability of Docking Failure

8.3.5 Lunar Surface Stay Risk Change

The lunar surface stay time will affect the total mission risk in terms of how long the LSAM and the CEV/SM remain in standby for the return trip. The LSAM is launched prior to the CEV/SM and loiters until the CEV/SM arrives and docks. Given an initial checkout prior to TLI, the LSAM will likely be powered down to a quiescent or semi-dormant state. For a typical sortie mission, the pressurized LSAM will become active just prior to undocking from the CEV/SM which will, in turn, become quiescent as the crew leaves to burn to the surface. For expediency, the system probability of failure inputs were assumed as a λ failure rate, with the system probabilities as simply a summation of the probability of failure per hour for all systems involved in the mission. Because the probability values are relatively low, this approach gives an adequate approximation. Most of the system failures were less than 1^{-5} failures per hour or better. For this mission, the probability of LSAM vehicle failure was 4.86^{-5} per hour and the CEV/SM vehicle quiescent failure rate was 6.51^{-5} per hour. Multiplying the summation of these two by 24 gives the 2.73^{-3} per-day result. The catastrophic fraction is the percentage of system failures that would result in a failure of the crew to return to Earth. **Figure 8-31** presents the mission failure probabilities over an extended mission.

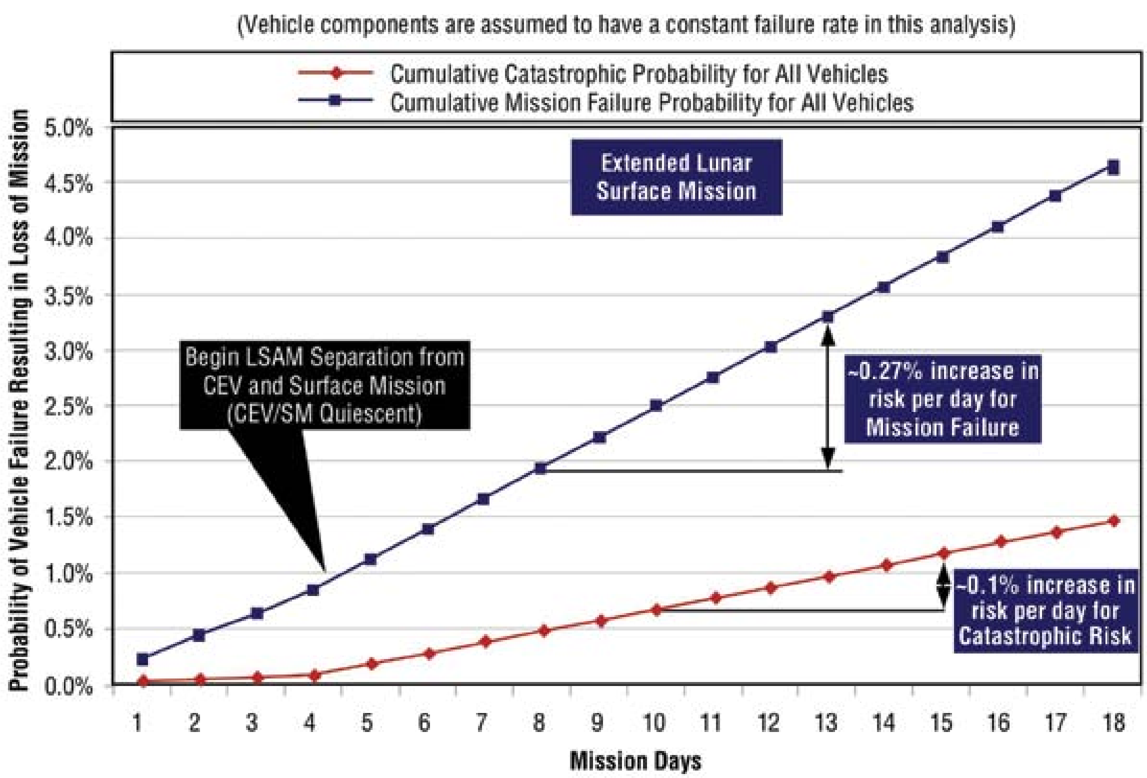


Figure 8-31. Mission Failure Probabilities Over an Extended Mission

8.3.6 CEV Stability Impacts on Crew Safety During Entry

One of the issues arising frequently for crew return is that of monostability of the CEV entry vehicle (Crew Module (CM)). In this context, a monostable CEV will only aerodynamically trim in one attitude, such that the vehicle would always be properly oriented for entry (similar to Soyuz). Requiring a CEV to be inherently monostable results in a CEV OML with weight and packaging issues. This study looked at how much benefit, from a risk standpoint, monostability provides so that the costs can be traded within the system design. In addition, this study looked at additional CEV systems that are required to realize the benefits of monostability and considered systems that could remove the need to be monostable. (See **Appendix 8F**, **CEV Stability Impacts on Crew Safety During Entry**.)

This study consists of two parts: a flight mechanics stability element and a risk assessment. The two pieces are combined to analyze the risk impact of CEV stability. The risk assessment was performed using the simple event tree shown in **Figure 8-32** representing the pivotal events during the entry mission phase. Each pivotal event was assigned a success probability determined from historical reliability data. In addition, mitigations to key pivotal events were modeled using the results from the stability study.

The success probabilities for the ballistic entry were determined from the aerodynamic stability study outlined below. In the event tree, the “Perform Ballistic Entry” event mitigates the “Perform Entry” (attitude and control) event, while the “Land and Recover from Ballistic Entry” event replaces the “Land and Recover” event should a ballistic entry occur.

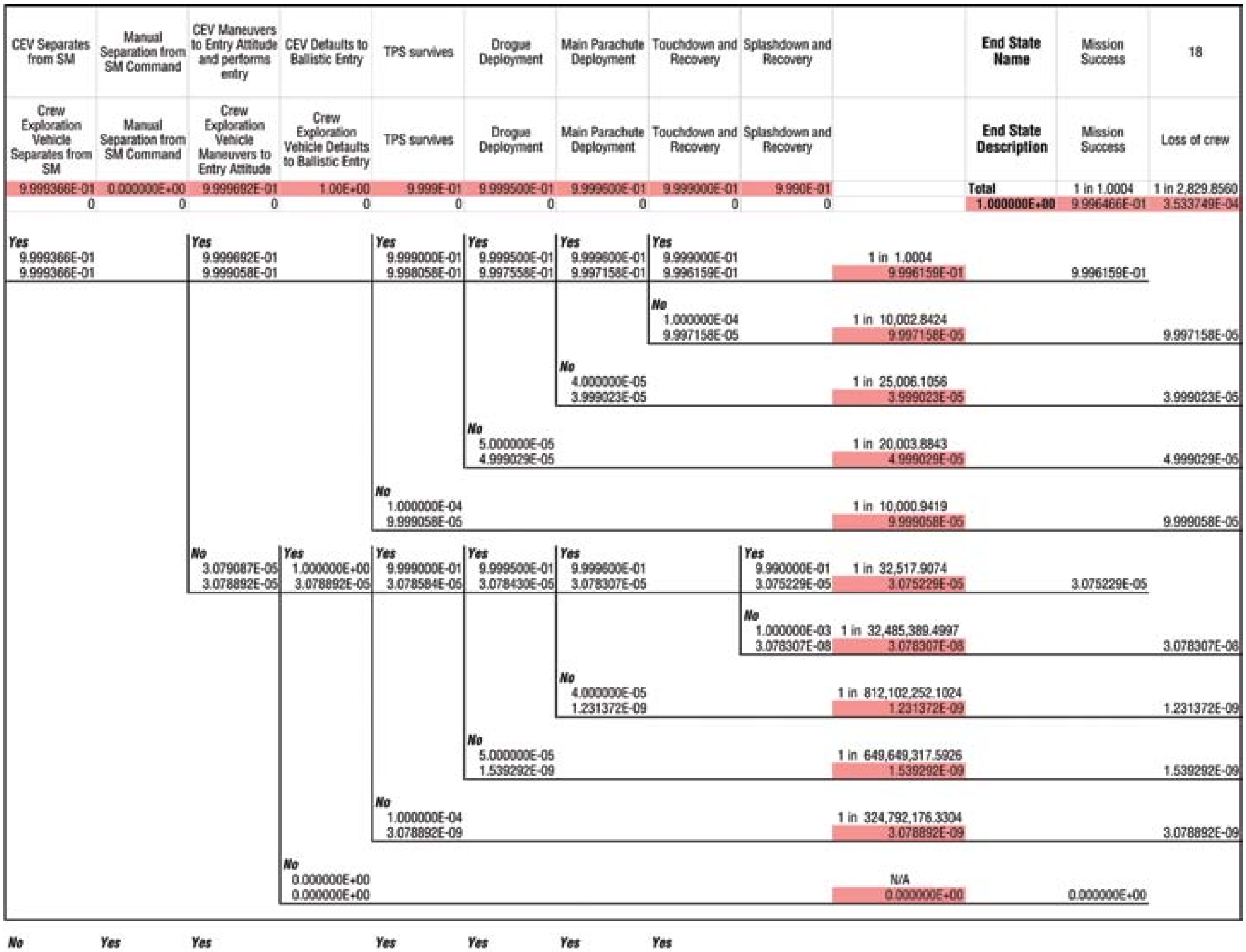


Figure 8-32. Entry, Descent, and Landing Event Tree

8.3.6.1 Results

The analysis process was used to generate range limits for load and heat rate violations for atmospheric entry from LEO/ISS. Initial attitudes from 180 to –180 deg were analyzed with no initial heat rate. In addition, initial pitch rates from –5 to 5 deg/sec were simulated at a constant initial attitude.

In addition to the initiating failures, there are additional actions that must occur to realize the risk benefit of strong stability. Most importantly, the CEV must roll to modulate the lift vector and establish a ballistic entry trajectory. If the lift vector is not rotated, a skip or excessive gravity dive can occur. This requires the use of RCS, which would be unavailable in the event of an RCS or total power loss, or an alternate means of roll initiation. To perform the roll initiation, a navigation aid must exist to indicate when to activate the roll or to initiate the roll automatically. These systems imply some form of active power source as well.

There are several options for systems to perform the above functions. The simplest system would include a solid, or cold jet, spin motor and a mark on the CEV window to orient the crew. If the only navigation aid was the visual window mark, the crew would have to visually determine the CEV attitude and activate the spin motor. This would require a conscious and coherent crew. The simplicity of this system could make it incredibly reliable, but the approach depends on monostability, or an alternate way to reach primary trim, to guarantee success.

An alternate approach would be based on four small, opposing cold jets near the apex of the CEV. These jets would also depend on a navigation aid such as the window mark or a backup gyroscope. The additional benefit of this system is that some pitch control would be available, reducing the need for monostability. This system could also be very reliable, but the operation itself is more complex. In addition, the weight of this system would be higher that the simplest system due to tanks, plumbing, controllers, etc.

A complete backup power system, RCS, and avionics system could be included, but the weight and packaging constraints would make this unattractive. This system would remove virtually all monostability requirements, because the vehicle could fly the nominal mission, as long as the system is completely independent to rule out any common cause failures.

All of these systems would mitigate the loss of primary power, avionics, and RCS. The specific system used will depend on the overall design trade results. If, for example, monostability drives the design to severely compromise the CEV, then a system that eliminates the need for monostability would add value. On the other hand, if the CEV is not largely hindered by the inclusion of monostability, the simplest system may be desirable.

For the risk analysis, the two simplest systems were assumed to have a reliability of 1.0; for the complete backup system, the reliability was assumed to be the same as for the original system, which was 0.99997.

8.3.6.2 Lunar Return

For the lunar return scenario, the same pivotal events were used in the event tree as in the LEO/ISS case. Events like separating the CEV from the SM, chute deployment, landing, and recovery are exactly the same in both cases; thus, the same reliability numbers were used. The Thermal Protection System (TPS) could be different, though the systems will likely be designed to the same margin. For this study, the TPS reliability number was held constant for the two missions. The RCS would endure more firings for the lunar case, so the failure rate was doubled to reflect a higher usage. The effective combination of these factors resulted in the same LOC probability of 1 in 2,780 for the lunar return entry.

The initial study results showed that the relative effect of CEV stability did not provide any measurable benefit for lunar cases. Further examination of the results showed that a large number of entry conditions resulted in skips due to the analysis approach. In this case, the initial attitude drives the results for the most stable cases, so there is little difference in the initial rate ranges. For the bistable case, the initial attitude and pitch rate limits are limiting, and a greater difference is seen. In addition, there are possible trajectories, with a short initial skip phase, which could be survivable if the CEV was tumbling. Predicting how likely this would be requires simulation of the vehicle heating as it enters the atmosphere. These computations would be sensitive to the specifics of the CEV OML, as well as the TPS distribution. For the current study, it was assumed none of the cases were survivable, but it is worth further investigation when more design definition exists. The actions required to perform ballistic entries in the event of primary system failures are the same as discussed above. The implications on the CEV design remain the same and will not be repeated here.

8.3.6.3 High-altitude Abort

High-altitude abort occurs after the ejection of the escape tower during the ascent phase. For a large portion of this mission phase, gravity and heat rate limits do not apply because the vehicle's speed is below the critical values. For a representative Shuttle-derived mission (500 sec in length), the escape tower would be ejected at approximately 150–200 sec of Mission Elapsed Time (MET). Entry gravity limits become important for mission aborts around 400–450 sec MET, and heat rate is an issue around 450–500 sec MET.

As with the previous cases, the absolute abort effectiveness, estimated at 89 percent using the event tree and numbers from **Appendix 8F**, **CEV Stability Impacts on Crew Safety During Entry**, does not change with the degree of stability. An LOC probability can be computed by assuming an upper stage failure probability of 1 in 625 (Reference 1 in **Section 8.7**, **References**). An 89-percent abort system leads to an LOC risk of 1 in 5,680 for this phase. These absolute figures are included for context only and need to be revisited once the design details are established.

8.3.6.4 Low-altitude Abort

The low-altitude abort regime is fundamentally different from the modes discussed so far. The aero forces are largely coupled and damping derivatives are important. For this reason, the simulation approach used in this study does not apply. In addition, a low-altitude abort would likely depend on an escape tower to perform the abort. The escape tower would shift the Center of Gravity (CG) and change the aerodynamic characteristics of the CEV and tower. The tower could be used as a stabilizing device, as used on Apollo, or could be ejected after the escape so strong stability could add some benefit. Unlike the previous cases, gravity load and heat rate limits are not of concern to a low-altitude abort. The primary concern with a tumbling, or improperly trimmed, CEV stems from the need to deploy the drogue chute. The explosive charge should have no problem propelling the drogue into the freestream, but some minor risk would occur from the possibility of the drogue lines wrapping around the CEV and interfering with the use of the primary chutes. It is suggested that the drogue would deploy in nearly every condition that would cause the CEV to attain its desired attitude.

8.3.6.5 Monostablility Summary

The benefit of stability manifests in two ways: monostability and strength of the attractor. Monostability relates to the lack of the possibility of an off-design trim, while the strength of the attractor (stability) determines the likelihood of trimming to the possible states within the required time.

In this study, stability effects were studied for lunar and LEO/ISS entries as well as high-altitude aborts. The extension to low-altitude aborts was mentioned but not quantitatively evaluated.

For LEO/ISS and lunar returns, the absolute benefit of strong monostability is negligible in terms of the LOC estimates. However, if the CEV does enter into a situation where an off-nominal entry is required, a strong primary trim attractor can result in aerodynamic positioning of the CEV in a safe-entry attitude between 6 and 80 percent of the time, depending on the strength of the attractor and the initial conditions. The expected benefit is in the range of 30 to 50 percent. Analysis approach refinements midway through the study suggest that LEO/ISS benefits may be larger than the current results suggest, and should be requantified if the success rate becomes a safety driver.

High-altitude aborts are energetically similar to LEO/ISS entries but originate from a different initial situation. Only initial attitude was investigated as a driving parameter for high-altitude aborts due to limited study time. It is expected that the benefit of strong monostability will be similar to the LEO/ISS results, but the range of potential initial pitch rates could be much higher.

Realization of strong stability benefits requires additional actions, e.g., putting the CEV into a roll to nullify the net lift of the capsule for a ballistic entry. Three approaches were discussed to accomplish ballistic entries, and the choice for a specific application depends on the overall vehicle trade space.

8.4 Architecture Summary

The risk analysis was performed in a number of iterations in concert with the design cycles of the study. Initially, placeholder values based on expert opinion were used. These values were updated as more information about the design was quantified, and detailed models were produced. The level of detail of the models was chosen based on the determination of the degree each would affect the architecture's overall risk. This concept focused on identifying "differences that make a difference." Lunar mission modes and ISS mission modes were analyzed in this study.

8.4.1 Lunar Missions

The initial study considered three mission modes: LOR, EOR, and EOR (direct). Each of these mission modes was evaluated with alternative levels of technology that enabled the missions to be launched on fewer or smaller launchers. The study had a set of ground rules which eliminated missions that required more than four launches due to the inherent unreliability of these concepts. Mission modes requiring three launches were eventually eliminated from consideration due to their cost and reliability issues (i.e., multiple launches, AR&D). Cost and mission reliability considerations tended to correlate with one another because simpler, mature systems have higher reliability and lower cost.

The mission modes are shown in **Figure 8-33**. The initial reference mission was LOR. In this mission mode, the crew and LSAM travel separately to lunar orbit where they dock. At this point, the mission becomes like Apollo, except the CEV is uncrewed. The LSAM is activated and descends to the lunar surface. At the end of the lunar stay, the LSAM ascends and docks with the CEV/SM which then returns to Earth. Risk drivers for this mission are:

- Two EDSs;
- Crew must be launched on a complex heavy vehicle; and
- The separation of the LSAM from the CEV eliminates the opportunity for the LSAM to serve as a safe haven during the trip to the Moon (as was done with Apollo 13). The CEV/SM is required to make additional burns in lunar orbit to rendezvous and dock with the LSAM. If the engine fails during these burns, the crew will be stranded in lunar orbit.

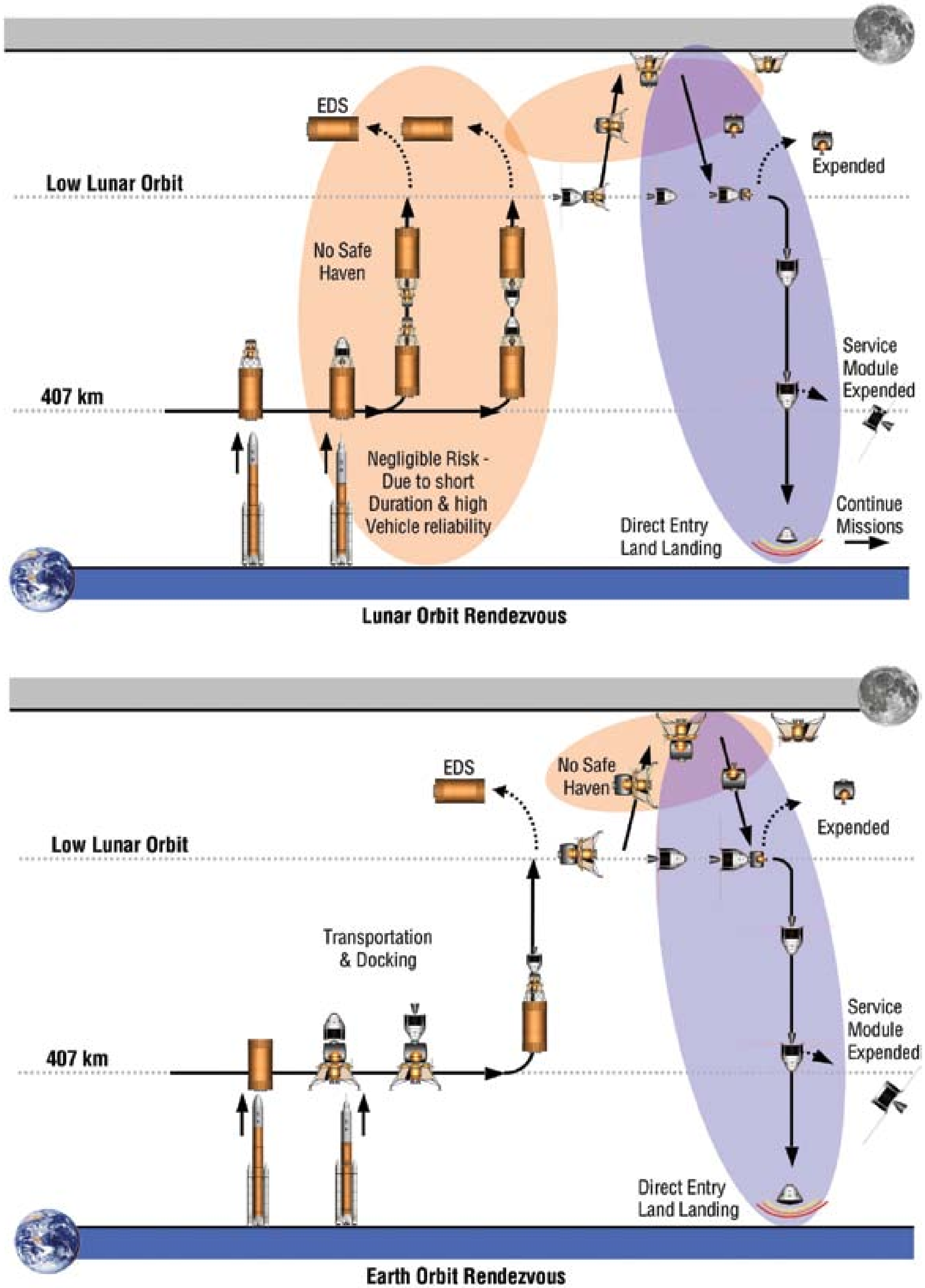


Figure 8-33. Lunar Mission Modes

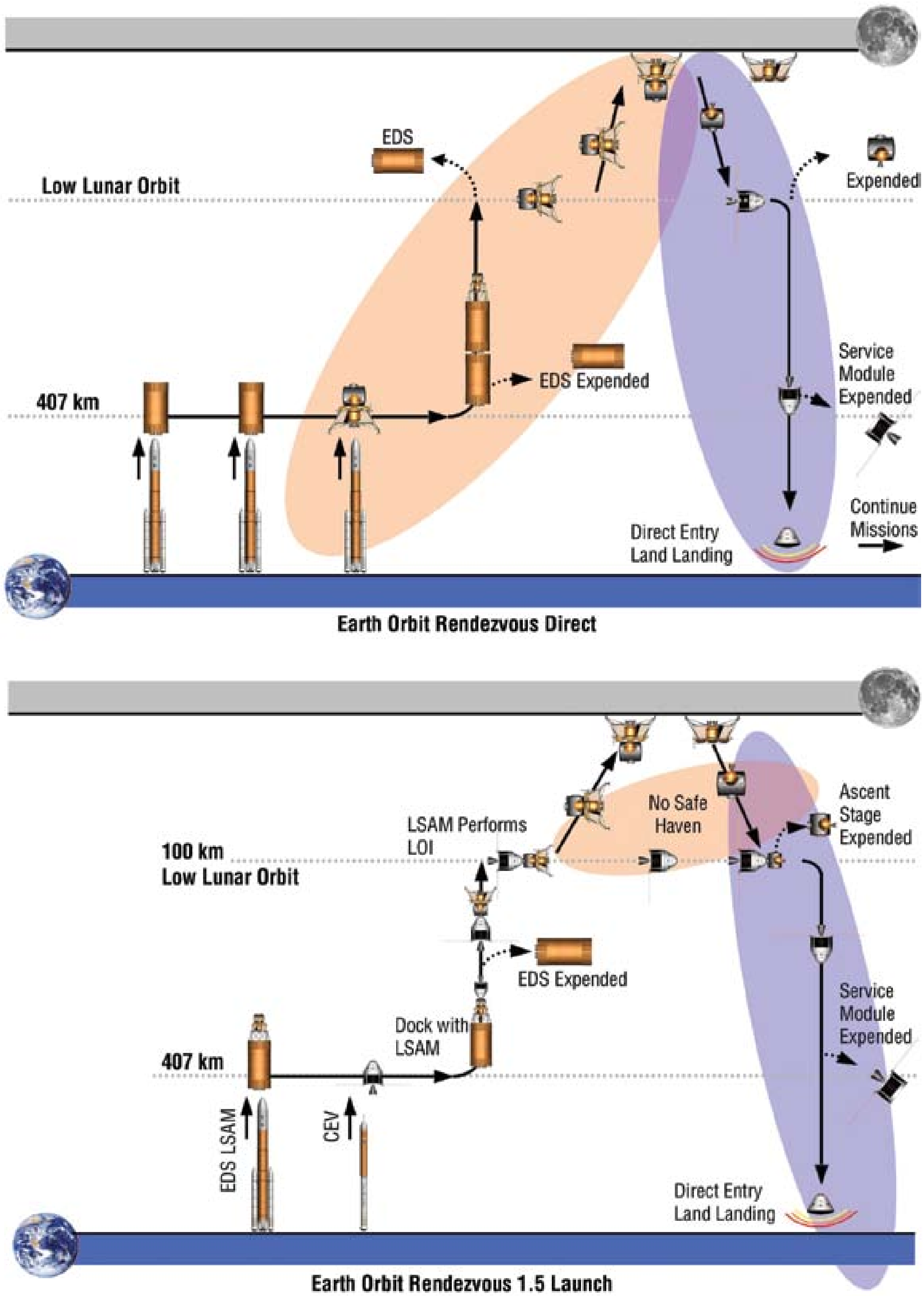


Figure 8-33. Lunar Mission Modes (continued)

The EOR mission is similar to the LOR mission except that the vehicle is assembled in LEO and a single EDS burn is required. Once in lunar orbit, the EOR mission is the same as LOR after docking. The EOR mission has several safety benefits over the LOR mission, including:

- LSAM is a potential safe haven during outbound legs;
- CEV/SM burns for rendezvous occur in LEO; and
- Reduced launch requirements, including:
  - Possibility for using the ISS CLV, thereby reducing the launch risk to the crew and eliminating a docking maneuver, and
  - Launching the mission on a single vehicle eliminates a docking maneuver, but increases risk to the crew due to the larger vehicle.

The EOR direct mission is the simplest in terms of mission events. The mission is assembled in LEO, proceeds directly to the Moon, and lands. It returns directly from the lunar surface, eliminating the need for docking on the return. However, this mission requires a third launcher and an AR&D, or higher performance of the propulsion on the in-space vehicles to achieve two launches. Risk drivers for this mission are:

- A single volume for the crew;
- A third launch with AR&D for low-performance vehicles; and
- Larger LVs and EDSs.

The additional risk from these aspects more than offsets the benefit of eliminating the need to rendezvous and dock with the SM on the return from the lunar surface, resulting in higher overall risk.

The mission mode preferred by this study was the EOR mission with the crew and CEV/SM being launched on the ISS CLV, and the LSAM and EDSs being launched on a heavy launcher (the 1.5-launch EOR mission). Risk impacts of this mode are:

- The ISS CLV is the safest, most reliable launcher with considerable experience in servicing the ISS; and
- Elimination of a transposition and docking maneuver for assembling the stack when the LSAM is launched with the CEV.

The risk analysis of each mission was developed from the individual events occurring in each type of mission. The direct mission modes were eliminated. The final analysis considered nine mission alternatives for the LOR and EOR mission modes. These alternatives explored the risks and benefits of increasing performance of the in-space propulsion stages, which is the key mission driver. Increasing performance of the in-space elements allows the same mission to be mounted with less mass delivered to orbit, thereby simplifying the mission.

The risk analysis also considered radiation risk and Micrometeoroid/Orbital Debris (MMOD) risk. These risks are moderated by the relatively short time the vehicle is exposed during ISS and lunar missions, and by the fact that the CEV has significant inherent shielding for these events. An analysis of the CEV radiation shielding requirements is contained in **Section 4, Lunar Architecture**. The shielding requirements for the CEV will cause these hazards to be controlled to a level where they will not affect overall risk. Spacecraft in LEO are threatened by the impact of either meteoroids or MMOD. The probability of being struck by MMOD is dependent on the geometry of the vehicle. The results of the assessment of the probability of being struck by an MMOD and this causing Loss of Vehicle (LOV) are shown below in **Table 8-7**.

	MMOD Probability of No LOV Damage	MMOD Risk	Odds of MMOD Impact exceeding LOV failure criteria
Twelve 6-month missions (6 years from 2011 through 2016)	0.980	2.0%	1 in 50
Requirement for 12 missions	≥ 0.992	≤ 0.8%	Better than (≤) 1 in 120

Table 8-7. Cumulative MMOD Risk for Multiple Missions to the ISS

The risk analysis analyzed the missions for LOM and LOC. The results of this analysis are shown in **Figures 8-34** and **8-35**, respectively. The EOR mission with 1.5 launches and pressure-fed engines on the CEV SM and lunar ascent stage have the lowest mission and crew risk.

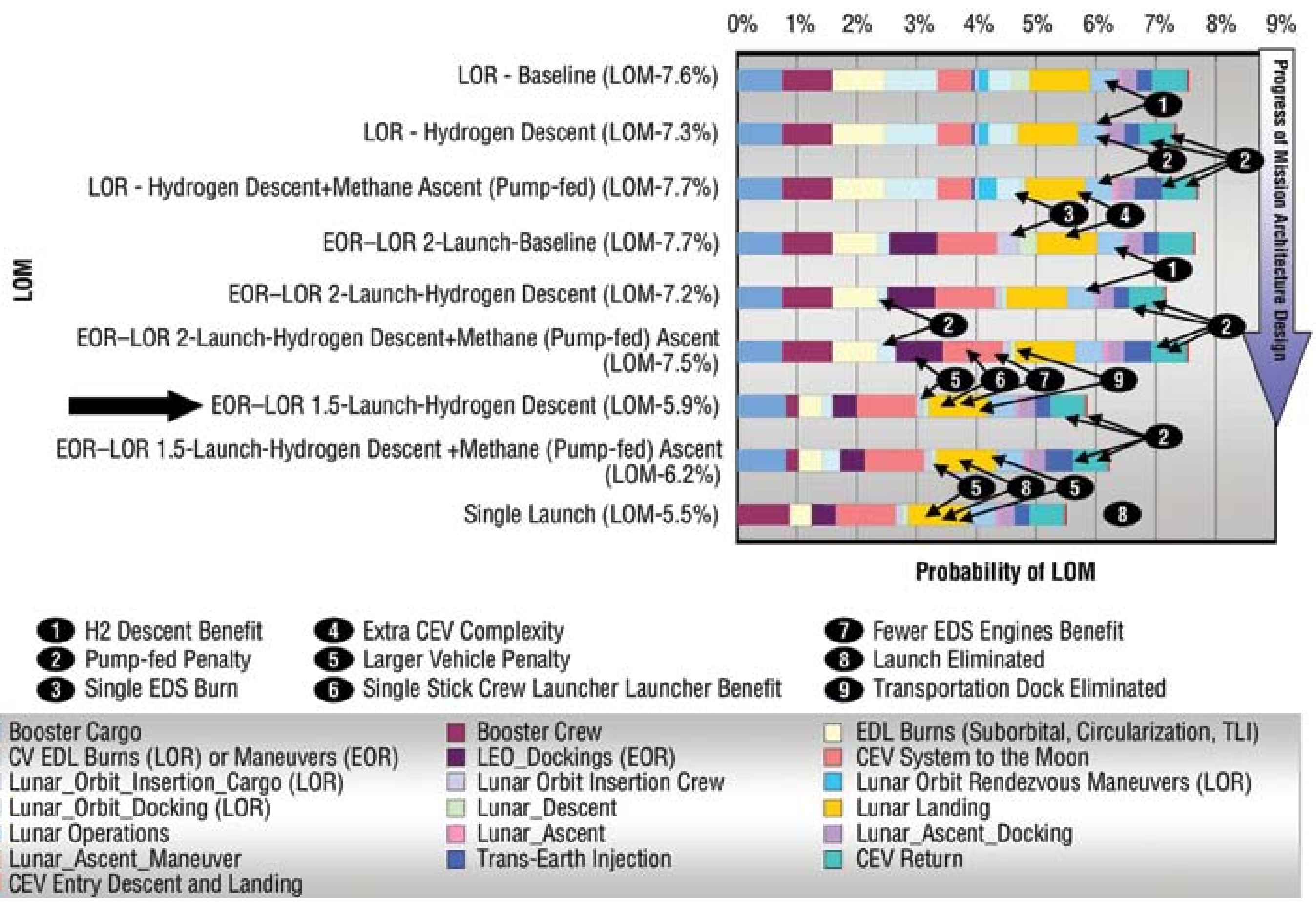


Figure 8-34. Comparison of LOM Risk for LOR and EOR Mission Alternatives

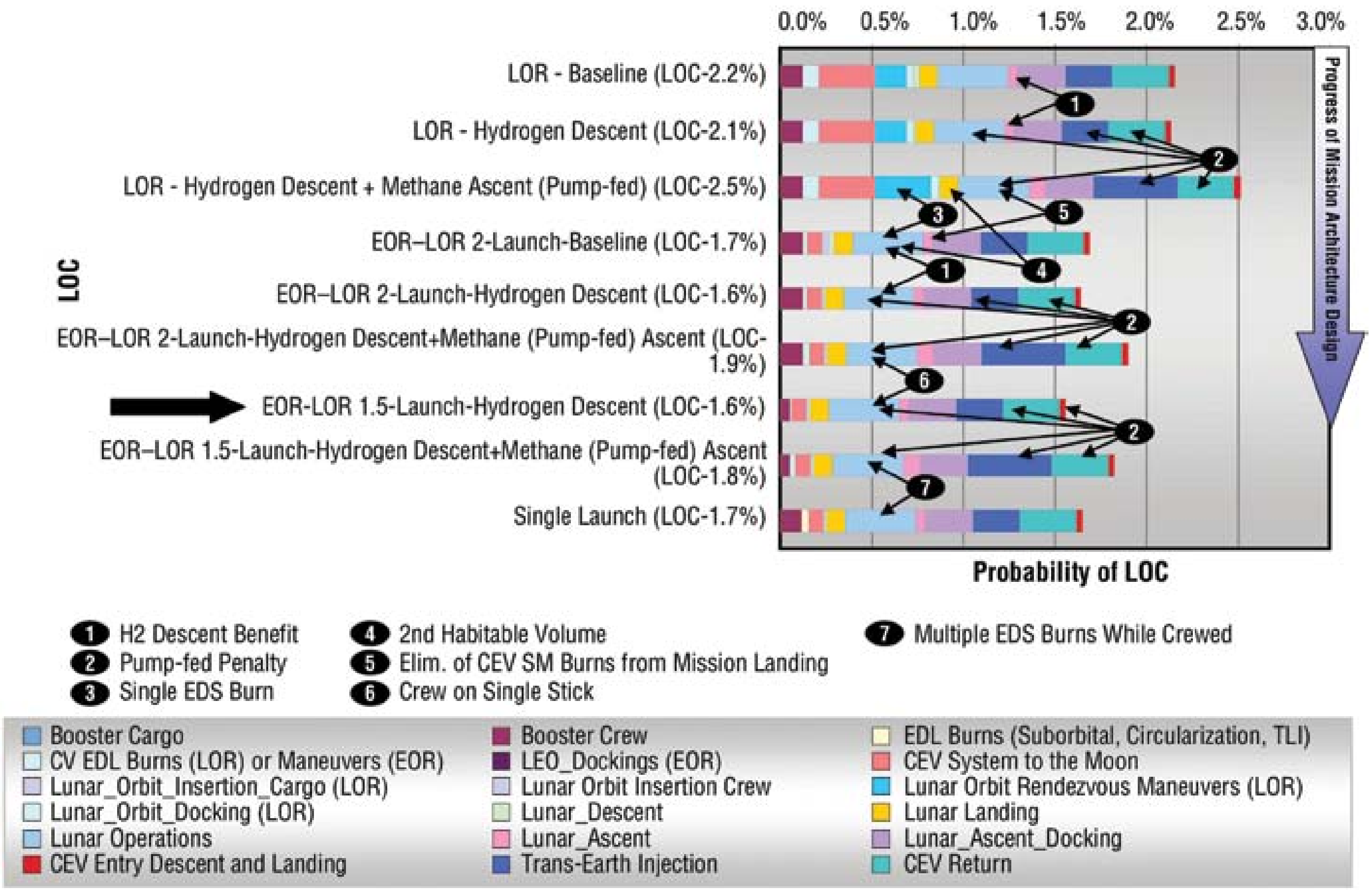


Figure 8-35. Comparison of LOC Risk for LOR and EOR Mission Alternatives

Annotations in the figures identify places where the risks vary. The application of the LOX/hydrogen engine (1) with engine-out capability on the LSAM descent stage is more reliable than two non-redundant pressure-fed LOX/methane engine systems. Replacing the pressure-fed LOX/methane engine on the SM and LSAM (2) increases risk because pump-fed engines are inherently less reliable than pressure-fed engines. Engine-out capability for these stages, however, presented a packaging problem that could not be solved. Changing the mission mode from LOR to EOR (3) eliminates an EDS burn, thereby increasing reliability. Combining the LSAM with the CEV (4) increases complexity, causing LOM risk to increase, but the extra habitable volume reduces LOC. This mode also eliminates the CEV/SM burns in lunar orbit for rendezvous and docking, shown in **Figure 8-35**. If the engine fails during this maneuver, the CEV will be marooned in lunar orbit. Combining the LSAM with the EDS and launching the crew on the single SRB requires slightly larger stages (5), with corresponding increase in risk for the larger vehicle. This risk is offset by replacing one launcher with the higher-reliability single-stick SRB (6). The 1.5-launch solution also employed an EDS with two J–2S engines rather than four LR–70 engines (7), further reducing risk. The 1.5-launch solution also elimiates a transportation and docking maneuver in LEO (4). Combining all the vehicles into a single launch reduced LOM risk by eliminating the single-stick CLV (8), but increased LOC risk by putting the crew on a larger, more complex vehicle (5).

Table 8-8 highlights the risk contributors for the preferred EOR mission with 1.5 launches (pump-fed LSAM descent hydrogen engines with engine-out, pressure-fed LSAM methane ascent and CEV SM engines). The yellow mission elements are the key drivers for LOM. The events in red indicate where the mission does not have a diverse abort mode on the lunar surface and on the return.

Table 8-8. EOR 1.5-Launch Mission with Pressure-fed SM and LSAM Ascent Engines

Phase	Mission Element	EOR–LOR 1.5-launch—Hydrogen Descent and Methane (pump fed) Ascent		
		LOM	Fatal	LOC
Launch	Booster _Cargo	124	–	–
	Booster_Crew	460	4	2,021
LEO Ops	EDS_Cargo	252	145	36,506
	EDS_Crew	332	10	3,319
	LEO_Dock_Man or ARD	250	–	–
Transit	CEV_System_to Moon	100	13	1,250
Lunar Orbit	Lunar_Capture_Cargo	–	–	–
	Lunar_Capture_Crew	905	10	9,046
	Lunar_Orbit_Maneuvers	–	–	–
	Lunar_Orbit_Docking	–	–	–
Lunar Descent	Lunar_Descent	1,018	10	10,178
	Lunar_Landing*	100	10	1,000
Lunar_OPS**	Lunar_OPS**	213	1	259
Lunar Departure	Lunar_Ascent	1,089	1	1,089
	Lunar_Ascent_Docking	381	1	381
	Lunar_Ascent_Manauver	–	–	–
	Lunar_Departure	218	1	218
Return	CEV_Return	172	2	325
EDLS	EDLS	2,830	1	2,830
		16		**55**
	Probability of Failure	6.2%		1.8%
	Reliability	**94%**		**98%**

** Indicates use of placeholder values as conservative reliability estimates.*
*** Does not include EVA risk.*

Figure 8-36 shows the breakout of LOM contributors. The risk contributors for this mission are relatively evenly distributed. The most significant LOM risks for this mission are the launch of the HLV, the CEV systems on the way to and from the Moon, and the lunar landing. The next highest risks include the docking maneuvers in Earth and lunar orbit.

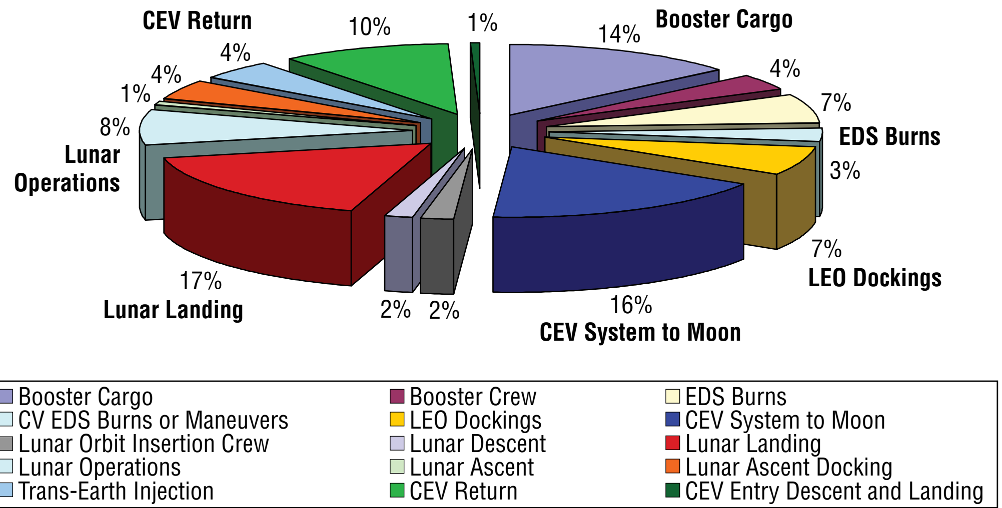


Figure 8-36. LOM Contributors for EOR 1.5-Launch Mission

The LOC risk breakout is shown in **Figure 8-37**. LOC is dominated by mission elements occurring after lunar landing where there are no diverse backups. These elements are the operations on the lunar surface; the ascent docking lunar departure return cruise; and entry, descent, and landing. Typically, the crew launcher would be a contributor, but the high reliability of the single-stick CLV significantly reduces this risk.

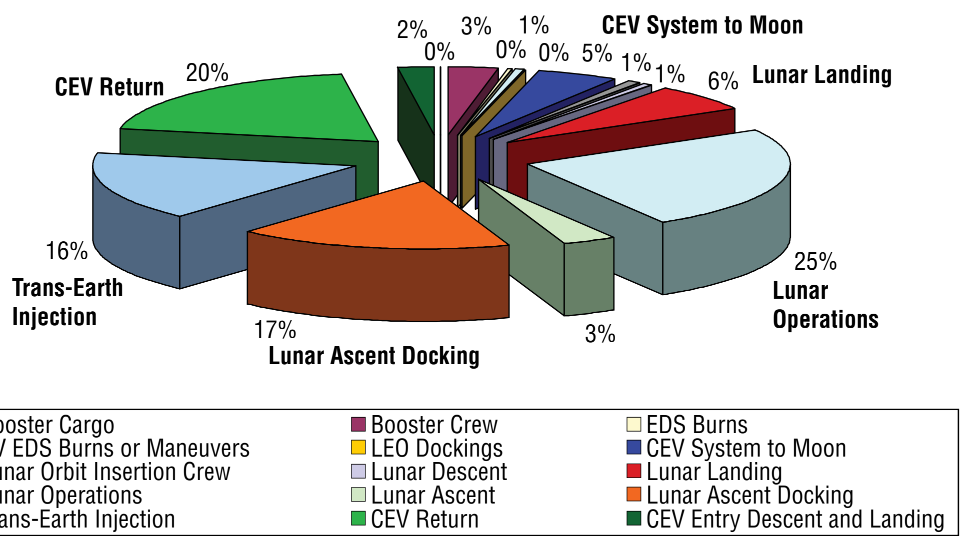


Figure 8-37. LOC Risk Contributors

Spaceflight will remain risky for the foreseeable future. The results of this analysis show that a lunar mission can be developed with acceptable, but not negligible, risk to the crew. The key factor causing the risks to be as low as they are is the application of existing technology for most mission elements and the extensive flight experience gained by operating critical CEV/SM in support of the ISS. Early failures of the CEV/SM system will occur in LEO, with simple abort options, rather than in lunar orbit with no possibility of return. This process helps mitigate the most significant source of risk to space systems, which is often referred to as unknown "unknowns."

8.4.2 ISS Missions

Missions for servicing the ISS are to be performed by derivatives of the CLV and CEV/SM system. For crewed missions to ISS, the CEV/SM is identical to the lunar mission. Pressurized cargo missions will require an automated docking capability similar to Progress. A simple mission model was developed from the CLV and the SM propulsion stage, combined with manual docking maneuver, and EDLS. The LOM and LOC results for the mature vehicle after 19 launches are shown in **Figures 8-38**, **8-39**, and **8-40**. Initially, the CEV/SM, and CLV will have higher failure rates due to the immaturity of the SSME air start (matures over 5 missions) and the LOX/methane engine of the SM (matures over 19 missions).

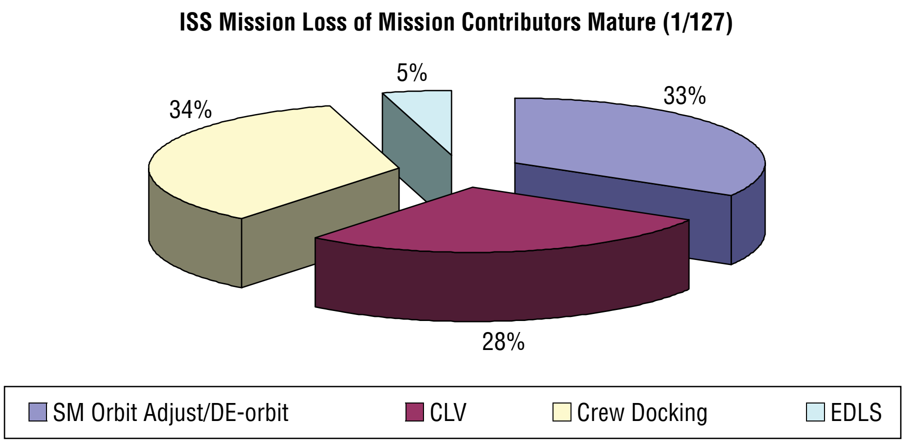


Figure 8-38. LOM Contribution for Mature Vehicle Crewed Missions

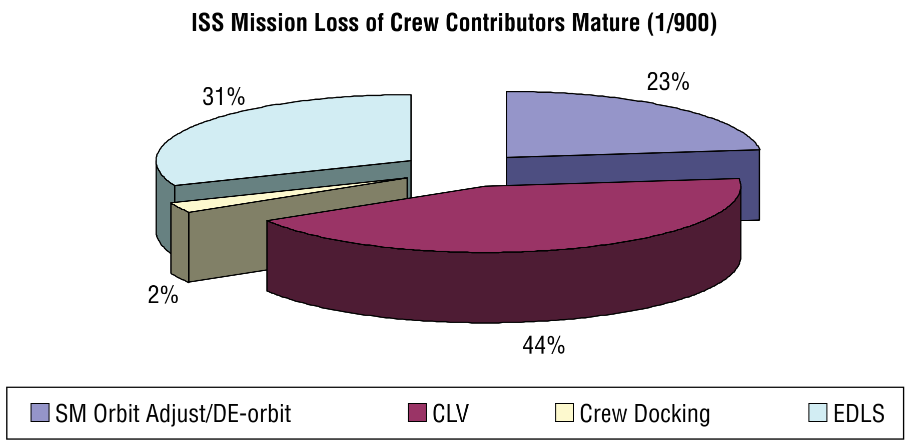


Figure 8-39. LOC Contribution for Mature Vehicle

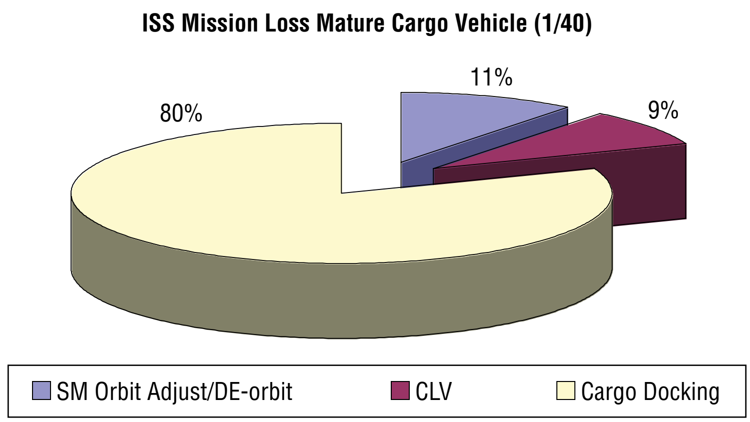


Figure 8-40. LOM Contributors for Cargo Vehicle

The cargo missions include automated docking maneuvers similar to the Progress, the Automated Transfer Vehicle (ATV), and the H–11 Transfer Vehicle (HTV). The LOM contribution for the mature cargo vehicles is shown in **Figure 8-38**.

The ISS mission model included in the architecture study includes the effect of maturity based on the actual traffic for the particular architecture. This effect is shown in **Section 8.5, Cumulative Campaign Summary**. LOM risk for the cargo vehicle is dominated by failure of the automated docking process.

8.5 Cumulative Campaign Summary

The risks of the lunar missions discussed in **Section 8.4**, **Architecture Summary**, were developed in the context of the NASA manned spaceflight program. This study recognized the importance of ISS missions in maturing the reliability of the most critical systems for the lunar mission (CEV, CEV/SM, and lunar ascent). This maturation process puts a significant burden of coping with failures on the ISS, but provides a tremendous opportunity for reliability growth of these systems (if NASA chooses to recognize this risk, learn from this experience, and continue flying if failures occur). The integrated mission model indicates that there is a significant likelihood that failure will occur, and analysis has shown that early crewed CEV missions will be riskier than the Shuttle. With the ISS cargo missions, the CEV reaches maturity in 2015 and is safer than the Space Transportation System (STS) by the third crewed flight. Moving the crewed flights within the schedule has a significant effect on the estimated risk.

The results of the analysis are shown in **Table 8-9**. The upper portion of the table shows the planned flight schedule for crew and cargo missions to the ISS and Moon. The maturity model shows how the key technologies mature during the process. The risk for LOM and LOC is shown in the bottom of the table.

Table 8-9. Cumulative Campaign Results

Mission Flight Rates														
Missions	**2005**	**2006**	**2007**	**2008**	**2009**	**2010**	**2011**	**2012**	**2013**	**2014**	**2015**	**2016**	**2017**	**2018**
Shuttle	1	3	5	5	3	3	0	0	0	0	0	0	0	0
HTV (H2)	0	0	0	0	1	1	1	1	1	1	1	1	0	0
ATV (Ariane)	0	1	1	1	1	1	1	1	1	1	1	0	0	0
Soyuz	0	1	1	2	2	2	1	0	0	0	0	0	0	0
Progress	0	0	0	3	3	4	5	0	0	0	0	0	0	0
CEV_DEV_SO	0	0	0	0	1	1	2	0	0	0	0	0	0	0
CEV_DEV_ORB	0	0	0	0	0	0	2	0	0	0	0	0	0	0
ISS_UnPress	0	0	0	0	0	0	1	0	1	1	1	1	0	0
CEV_ISS	0	0	0	0	0	0	1	2	2	2	2	2	0	0
ISS_Pres	0	0	0	0	0	0	0	3	3	3	3	3	0	0
Con-1	0	0	0	0	0	0	0	0	0	0	0	0	1	0
Con-2	0	0	0	0	0	0	0	0	0	0	0	0	1	0
Con-3	0	0	0	0	0	0	0	0	0	0	0	0	0	1
Con-4	0	0	0	0	0	0	0	0	0	0	0	0	0	1
Maturity Model														
SM_Orbit_Ajust (LOX/CH4)/CEV ISS	20.0%	20.0%	20.0%	20.0%	20.0%	20.0%	20.0%	12.7%	5.9%	1.2%	0.3%	0.3%	0.3%	0.3%
Launcher (13.1)	30.0%	30.0%	30.0%	30.0%	30.0%	30.0%	30.0%	1.5%	0.2%	0.2%	0.2%	0.2%	0.2%	0.2%
Docking_Auto_station	10.0%	10.0%	10.0%	10.0%	10.0%	10.0%	10.0%	7.2%	2.3%	2.0%	2.0%	2.0%	2.0%	2.0%
Loss of Mission Risk														
Shuttle	0.01	0.03	0.05	0.05	0.03	0.03	–	–	–	–	–	–	–	–
HTV (H2)	–	–	–	–	0.29	0.27	0.26	0.25	0.24	0.23	0.22	0.21	–	–
ATV (Ariane)	–	0.06	0.05	0.05	0.04	0.03	0.03	0.03	0.02	0.02	0.02	–	–	–
Soyuz	–	0.01	0.01	0.02	0.02	0.02	0.01	–	–	–	–	–	–	–
Progress	–	–	–	0.12	0.12	0.16	0.20	–	–	–	–	–	–	–
CEV_DEV_SO	–	–	–	–	0.01	0.19	0.21	–	–	–	–	–	–	–
CEV_DEV_ORB	–	–	–	–	–	–	0.44	–	–	–	–	–	–	–
ISS_UnPress	–	–	–	–	–	–	0.33	–	0.08	0.03	0.02	0.02	–	–
CEV_ISS	–	–	–	–	–	–	0.19	0.22	0.08	0.02	0.01	0.01	–	–
ISS_Pres	–	–	–	–	–	–	–	0.37	0.14	0.08	0.07	0.07	–	–
Con-1	–	–	–	–	–	–	–	–	–	–	–	–	0.03	–
Con-2	–	–	–	–	–	–	–	–	–	–	–	–	0.03	–
Con-3	–	–	–	–	–	–	–	–	–	–	–	–	–	0.05
Con-4	–	–	–	–	–	–	–	–	–	–	–	–	–	0.06
Total Incidents	**0.01**	**0.11**	**0.22**	**0.46**	**0.96**	**1.68**	**3.35**	**4.21**	**4.78**	**5.16**	**5.51**	**5.83**	**5.88**	**5.99**
Loss of Crew Risk														
Shuttle	1.0%	3.0%	5.0%	5.0%	3.0%	3.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%
Soyuz	0.0%	0.3%	0.3%	0.5%	0.5%	0.5%	0.3%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%
CEV_ISS	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	1.9%	2.2%	0.8%	0.2%	0.1%	0.1%	0.0%	0.0%
Con-3	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.6%
Con-4	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	1.5%
Total Success	**99.0%**	**95.8%**	**90.8%**	**85.8%**	**82.8%**	**79.9%**	**78.2%**	**76.5%**	**75.9%**	**75.7%**	**75.6%**	**75.5%**	**75.5%**	**75.5%**
Probability_LOC	**1.0%**	**4.2%**	**9.2%**	**14.2%**	**17.2%**	**20.1%**	**21.8%**	**23.5%**	**24.1%**	**24.3%**	**24.4%**	**24.5%**	**24.5%**	**24.5%**

The integrated LOM risk for the traffic model is shown in **Figure 8-41**. This shows that the manned missions are a small contributor to the total mission losses. The LOM estimate is dominated by the HTV due to the estimated unreliability of the Japanese HII launcher. The CEVs are less reliable during their early missions, but improve dramatically after 2013. The ATV is a small contributor because it flies only once a year and is relatively mature. This result indicates that it would be prudent for NASA to develop a method to cope with failures and be able to return to flight as soon as possible. It would be wise to treat all early flights as test flights and thoroughly examine anomalies, perhaps even having a preconvened accident investigation board ready to investigate and close out incidents.

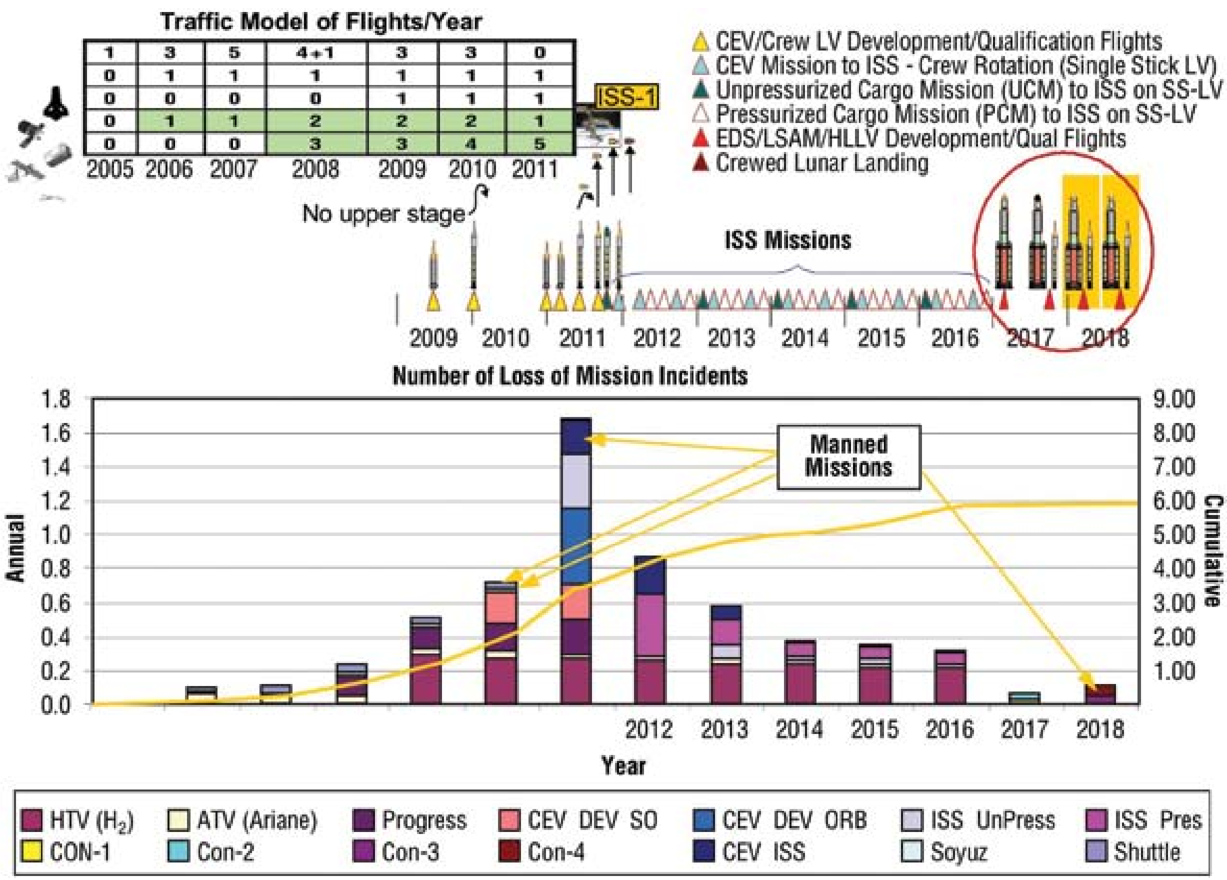


Figure 8-41. Probability of LOM Per Year

Figure 8-42 shows the integrated LOC results. This result indicates that STS launches present the greatest risk to the crew. The CEV missions to ISS are initially risky, but become small after the first 3 years. A close look at the maturity model shows how the ISS cargo missions are effective in lowering risk for the crew since they share the same SM.

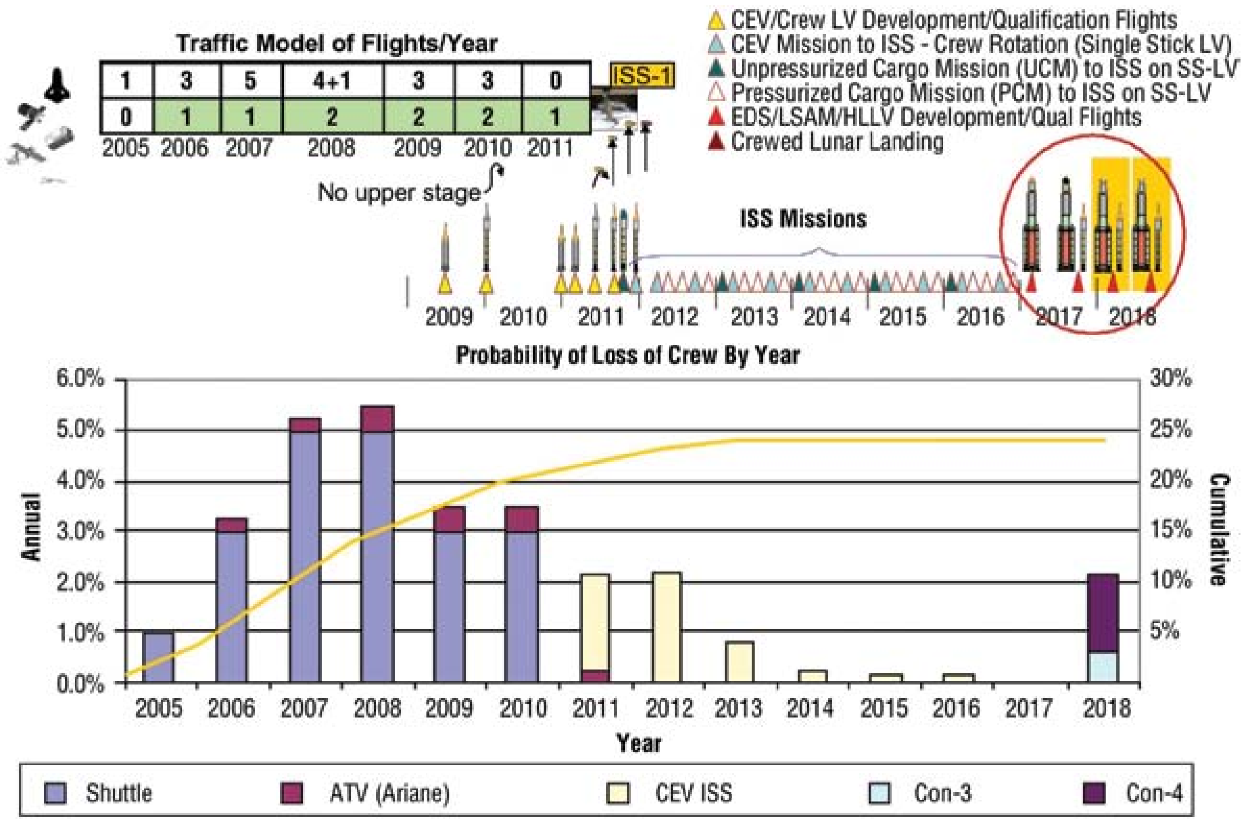


Figure 8-42. Probability of LOC Per Year

Figure 8-43 illustrates how ISS cargo missions aid in the maturation of the CEV crewed vehicle since they share the same SM. The upper curve shows crewed flights only, with no cargo and two test flights. The bottom curve shows the current schedule, which is two test flights, one cargo flight, and then alternating crewed and cargo flights (two and four per year, respectively). In either case, it takes five flights, in addition to the two test flights, to surpass the Shuttle safety level of 1 in 100.

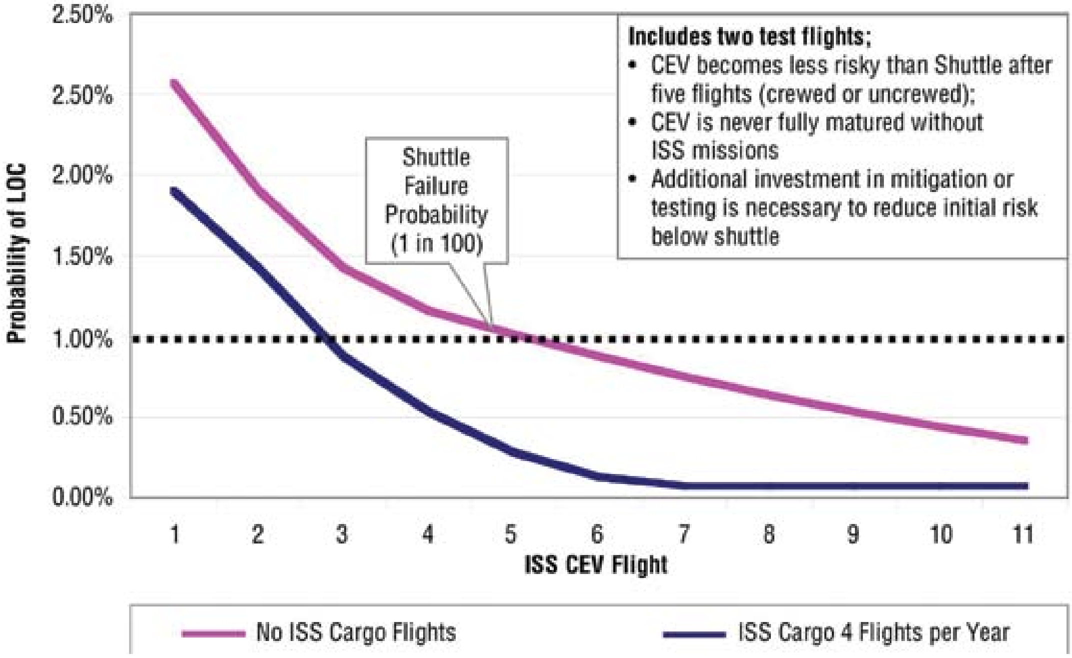


Figure 8-43. Benefit of ISS Cargo Flights on Crew Safety

If the schedule is followed, the first crewed flight would have three maturity flights (two test and one cargo) before it flies. Therefore, it would be less than twice the Shuttle risk, approximately 1 in 50. If there were no cargo flights beforehand, the risk of the first crewed flight after the two test flights would be approximately 1 in 40, or approximately 2.5 percent (2.5 times the Shuttle).

The additional cargo flights allow the system to mature at a faster rate, achieving a factor of 10 improvement over the Shuttle in the seventh flight. Without the cargo flights, the CEV is only about 3 times better than the Shuttle at the end of the 11 ISS missions.

8.6 Forward Work

By organically including quantitative risk assessment into the design process, the ESAS was able to perform complex trades across multiple Figures of Merit (FOMs) and arrive at a solution that effectively blended performance and risk within time and budget constraints. These blends applied technology to enable safer mission modes and reduced mass in areas that were overprotected. This organic process can be applied at any level. SPACE models can be developed to assure that lunar basing strategies account for potential failures during the campaign and effectively blend in new capabilities without undue increases in risk. This blending process will allow the benefits of system maturity to be applied to developing new systems. This process can also be applied at the systems and component level of the architecture. Employing these techniques in developing requirements for crew escape could help NASA develop a balanced design that is focused on risk drivers.

As the SPACE models are developed to higher and lower levels, they can be combined with development risk models and cost model results to provide an integrated view of the overall program. This view will allow NASA to make decisions on an integrated basis such that the program is structured to reduce the vulnerability to failure, balance the resources used to prevent failure, and assure that resources and activities are organized to maximize return given the inherent uncertainties.

Most importantly, this approach can be applied to enhance the decision-making process within the concept of an "Observation, Orientation, Decision, Action (OODA) Loop" (Reference 7 in **Section 8.7**, **References**) shown in **Figure 8-44**. This process can be applied by NASA to create a decision-making environment that will allow NASA to cope with the uncertainty in space programs. This is done by improving the capability to apply heritage information with information gleaned from unfolding circumstances within an integrated analytical framework that is agile enough to allow the synthesis of multiple responses that affect cost, risk, performance, and schedule. Currently, NASA is hobbled by complex analytical tools that make it difficult to explore a decision space effectively. The quality of the analysis is perceived to increase with additional fidelity. However, as fidelity increases, the interactions between model elements grows exponentially, and it becomes impossible to analyze more than a few design reference architectures or missions, even with the current significant growth in computational power. By using the SPACE analysis process, model fidelity is increased where the increase in fidelity provides insight to the decision at hand. The analytical framework must be simple enough and flexible enough to provide answers at an appropriate level of detail as both the environment and questions themselves change.

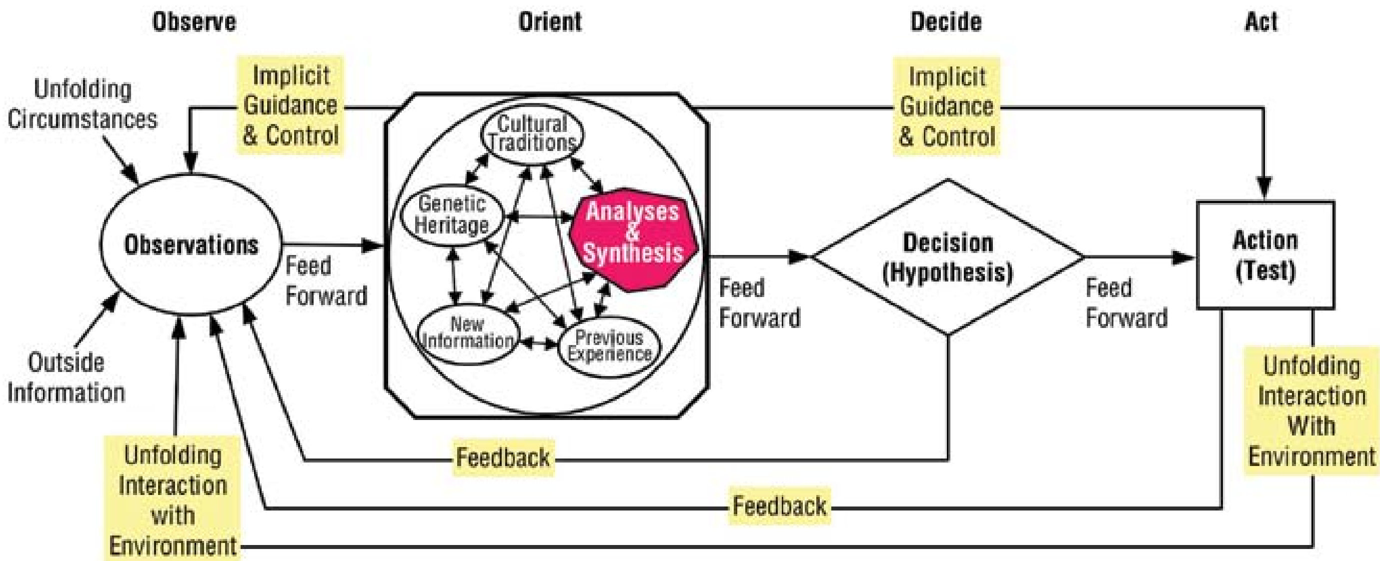


Figure 8-44. Boyd's OODA Loop

An initial application of this approach is the decision-making process regarding the interaction between the Shuttle manifest to the station (i.e., number of flights and content) and the progress being made on the CEV. Given the current uncertainty in the projections of equipment reliability and the lack of a probabilistic model for the performance of the ISS, decisions might be made that will result in either too many or too few logistics flights to the ISS. Since each Shuttle flight is so precious to NASA and its partners, integrated models that can adapt to new information will be extremely valuable. These models can capture operating experience of ISS equipment and project that reliability into the context of ISS operability with different sparing strategies and gaps in logistic flights. These models can be combined with development risk models to determine the likelihood and consequences of gaps between Shuttle termination and CEV missions. If this model is updated on a continuous basis, the program will be able to assimilate new information from both sides and make decisions that will most effectively apply the Shuttle assets.

8.7 References

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3. NASA, Space Shuttle Probablistic Risk Assessment, Iteration 2.0, Space Shuttle Program, February 2005 (Draft).
4. United Space Alliance, Introduction to Rendezvous Guidance, Navigation and Control.
5. Rendezvous GPO Console Handbook.
6. JSC – 49626, February 2003, Space Shuttle Rendezvous and Proximity Operations Experience Report.
7. Col John Boyd, USAF (Ret) coined the term "OODA loop" describing the process of Observation, Orientation, Decision, Action, used to dominate an adversary in an uncertain environment. (See http://www.d-n-i.net/second_level/boyd_military.htm.)

9. Technology Assessment

9.1 Summary

The Space Exploration Vision set forth by President Bush cannot be realized without a significant investment in a wide range of technologies. Thus, key objectives of the Exploration Systems Architecture Study (ESAS) are to identify key technologies required to enable and significantly enhance the reference exploration systems and to prioritize near-term and far-term technology investments. The product of this technology assessment is a revised Exploration Systems Mission Directorate (ESMD) technology investment plan that is traceable to the ESAS architecture and was developed by a rigorous and objective analytical process. The investment recommendations include budget, schedule, and center/program allocations to develop the technologies required for the exploration architecture.

This section summarizes the results of this assessment, including the key technologies required to support the new architecture. The three major tasks of the technology assessment were: (1) to identify what technologies are truly needed and when they need to be available to support the development projects; (2) to develop and implement a rigorous and objective technology prioritization/planning process; and (3) to develop ESMD Research and Technology (R&T) investment recommendations about which existing projects should continue and which new projects should be established.

The following are the major Ground Rules and Assumptions (GR&As) used for the assessment:

- All technology developments shall be directly traceable to architecture requirements.
- Mission reference dates for R&T planning shall be:
  - 2011 Crew Exploration Vehicle (CEV) human flight to the International Space Station (ISS);
  - 2018 goal of human mission to the Moon including landing, but no later than 2020; and
  - 2022 goal of permanent human presence on Moon.
- Technologies shall be developed to Technology Readiness Level Six (TRL–6) or better by Preliminary Design Review (PDR), the reference dates for which shall be:
  - 2007 PDR for CEV and Crew Launch Vehicle (CLV);
  - 2012 PDR for initial lunar mission elements; and
  - 2017 PDR for lunar permanent human presence mission elements.
- The Prometheus Nuclear Systems Technology (PNST) shall receive a funding profile for this study of $100M in FY06 and $50M in FY07–11 followed by significant increases.
- Ten percent (10%) of each program budget shall be reserved for program management.
- The budget shall not include funds for earmarks.
- Legislated requirements (e.g., Small Business Innovative Research (SBIR)) shall be preserved.
- Relevant ISS flight research payloads shall be preserved.
- Funding wedges shall be included for future lunar and Mars R&T requirements.

The ESAS technology assessment determined that technology development projects are needed in 12 major areas:

- Structures and Materials,
- Protection,
- Propulsion,
- Power,
- Thermal Controls,
- Avionics and Software,
- Environmental Control and Life Support (ECLS),
- Crew Support and Accommodations,
- Mechanisms,
- In-Situ Resource Utilization (ISRU),
- Analysis and Integration, and
- Operations.

The final result of the technology assessment is a recommended reduction in the overall funding of ESMD R&T of approximately 50 percent. **Figures 9-1** and **9-2** show the before- and after-budget profiles.

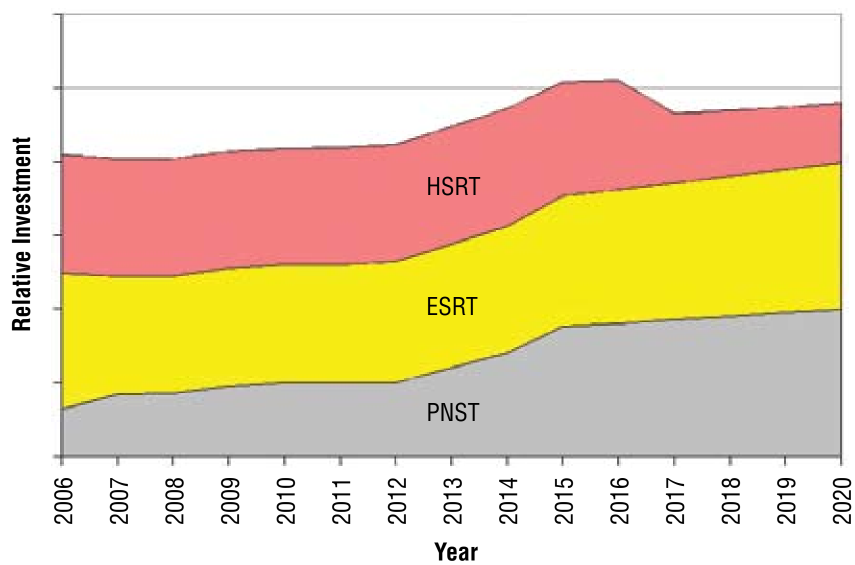


Figure 9-1. FY06–FY19 Original Funding Profile

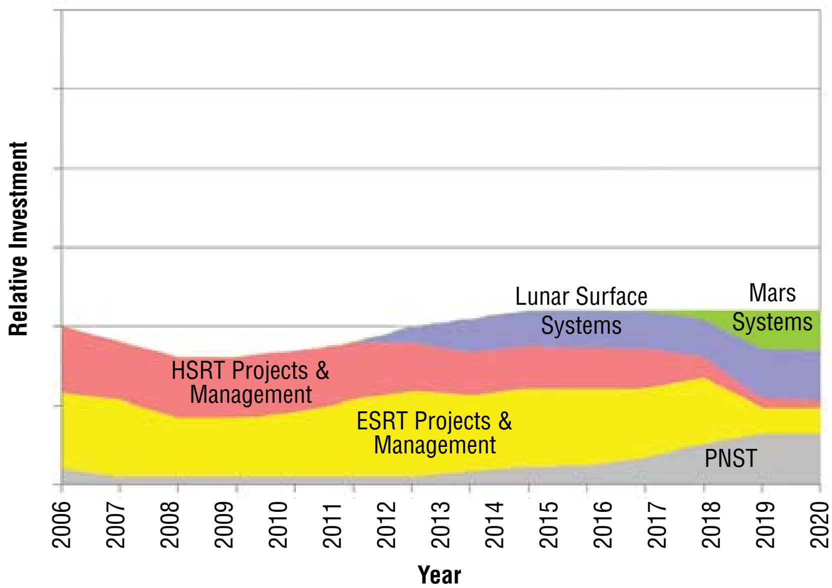


Figure 9-2. FY06–FY19 ESAS Recommended Funding Profile

The funding profile includes 10 percent management funds and approximately 30 percent of liens due to prior agency agreements (e.g., Multi-User System and Support (MUSS), the Combustion Integrated Rack (CIR), and the Fluids Integrated Rack (FIR)) and legislated requirements (e.g., SBIR, Small Business Technology Transfer (STTR)).

Seven key recommendations arose from the technology assessment:

- ESMD should share costs with SOMD for MUSS, CIR, and FIR.
- ESMD should transfer the Alpha Magnetic Spectrometer (AMS) to the Science Mission Directorate (SMD) to compete for funding with other science experiments.
- ESMD should quickly notify existing Exploration Systems Research and Technology (ESRT) projects not selected by ESAS that they will be not receive funding beyond FY05.
- ESMD should move Systems Analysis and Tool Development activities (and budget) to a directorate level organization—no longer in ESRT.
- Key ESAS personnel should work with ESMD to facilitate implementation. (Many technologies require immediate commencement on an accelerated schedule.)
- ESMD should develop a process for close coordination between architecture refinement studies and technology development projects. Technology projects should be reviewed with the flight element development programs on a frequent basis to ensure alignment and assess progress.
- ESMD should develop a process for transitioning matured technologies to flight element development programs.

9.2 Technology Assessment Process

The above recommendations were developed through a rigorous and objective process consisting of the following: (1) the identification of architecture functional needs; (2) the collection, synthesis, integration, and mapping of technology data; and (3) an objective decision analysis resulting in a detailed technology development investment plan. The investment recommendations include budget, schedule, and center/program allocations to develop the technologies required for the exploration architecture, as well as the identification of other investment opportunities to maximize performance and flexibility while minimizing cost and risk. More details of this process are provided in **Appendix 9A**, **Process**.

The ESAS technology assessment involved an implementation team and an Agency-wide Expert Assessment Panel (EAP). The team was responsible for assessing functional needs based on the ESAS architecture, assembling technology data sheets for technology project(s) that could meet these needs, and providing an initial prioritization of each technology project's contribution to meeting a functional need. This involved key personnel working full time on ESAS as well as contractor support and consultation with technology specialists across NASA, as needed.

The EAP was a carefully balanced panel of senior technology and systems experts from eight NASA centers. They examined the functional needs and technology data sheets for missing or incorrect entries, constructed new technology development strategies, and performed technology development prioritization assessment using the ESAS Figures of Merit (FOMs) for each need at the architecture level. They provided internal checks and balances to ensure even-handed treatment of sensitive issues.

All results were then entered into spreadsheet tools for use by the ESAS team in analyzing technology investment portfolio options. During the final step of the process, the ESAS team also worked with ESMD and the Administrator's office to try to minimize Center workforce imbalance.

9.3 Architecture R&T Needs

This assessment was performed in parallel with the architecture development, requiring the whole ESAS team to coordinate closely to ensure that the technology assessment captured the latest architecture functional needs. The functional needs were traced element by element, for each mission, in an extensive Excel file. These needs were the basis for the creation of the technology development plans used in the assessment. Thus, all technology development recommendations were directly traceable to the architecture. This analysis indicated that R&T development projects are needed in the following areas:

- Structures and Materials,
- Protection,
- Propulsion,
- Power,
- Thermal Controls,
- Avionics and Software,
- ECLS,
- Crew Support and Accommodations,
- Mechanisms,
- ISRU,
- Analysis and Integration, and
- Operations.

These areas are described below. Each area's section contains the description of its functional needs, the gaps between state-of-the-art and the needs, and the recommended developments. There is a more detailed write-up for each recommended technology development project listed in **Appendix 9B**, **Technology Development Activity Summaries**.

9.3.1 Structures and Materials

The ESAS architecture could potentially meet mission needs using aluminum alloys and state-of-the-art materials. However, reduction in structural mass translates directly to additional up-and-down mass capability that would facilitate logistics and increase science return for current and future mission phases. Reductions in structural mass could also offset growth in other systems. Lightweight structures that provide structural load-bearing support, radiation protection, and possibly other combinations of protection such as thermal or Micrometeoroid/ Orbital Debris (MMOD) are also desirable from an architecture robustness perspective.

Simultaneous proof-of-concept demonstrations for integrated lightweight structures and research into advanced materials and structures for future missions to the Moon and Mars must be conducted. A series of building-block demonstrations of key components, elements, and subsystems should be conducted with appropriate validation testing. In order to support lunar surface systems, near-term activities should culminate in a large-scale integrated structures demonstration in relevant environments by 2012 (TRL–6). Critical investments include:

- Novel, multifunctional design concepts, including modularity;
- Integrated system performance (including deployment) in relevant environments of core structural modules;
- Durable flexible materials, including Nextel, aliphatic polymers and polyurethane, tailoring for redundant load paths and self-healing;
- Organic materials, including polymer matrix composites;
- Advanced aluminum alloys, titanium alloys, super alloys, refractory alloys, and metal matrix composites;
- Hybrid organic/metallic composites;
- Integrated thermal management;
- Lightweight radiation protection, including use of advanced materials (boron composites);
- Advanced sensors for structural and environmental monitoring, including embedded fiber optic and acoustic sensors, and other integrated, autonomous sensing technologies; and
- Structures that can adapt to dynamic environmental conditions and mission changes. This includes self-healing materials, redundant structural architectures, and active, embedded sensing and control.

9.3.2 Protection

Protection is a category of capabilities that provide protection of an element and its contents from environments, both natural and self-induced. These capabilities include thermal protection, radiation protection, and lunar dust/environment mitigation. Protection is a key area with respect to mission success and safety and warrants considerable investment. The ESAS architecture requires that the CEV Crew Module (CM) be capable of performing entry into the Earth's atmosphere at Earth-orbital, lunar-return, and Mars-return velocities. A Thermal Protection System (TPS) requires materials specifically designed to manage aerothermal heating (heat flux, dynamic pressure) experienced during hypersonic entry, for both nominal and abort scenarios. A single architecture may require both reusable and single-use materials. Only ablators can meet maximum requirements; they are designed to sacrifice mass under extreme heating efficiently and reliably. Reusable materials that preserve the Outer Mold Line (OML) can meet requirements for lower heating locations. The Apollo ablative TPS (AVCOAT–5061) no longer exists. Qualification of new or replacement materials will require extensive analysis and testing.

It is well known that the primary sources of radiation exposure in space are Galactic Cosmic Rays (GCRs) and Solar Particle Events (SPEs). However, due to a number of independent variables associated with these sources of radiation, there is considerable uncertainty about the total shielding required for long-duration missions. Research is needed to confidently predict the shielding capabilities of various materials and spacecraft components along with corresponding research to understand crew exposure limits. Most hydrocarbon-based composites have value as radiation shielding; thus, many materials (e.g., ones developed for lightweight structures) may also be useful for radiation protection.

Apollo lunar Extra-Vehicular Activity (EVA) experience has shown lunar dust to be problematic with respect to seals and mechanical systems. Significant research into dust tolerant airlock systems needs to be performed. Durable and robust materials and systems for airlock structure and seals that include dust mitigation capabilities must be developed. Enhanced durability and dust exclusion technologies for application to EVA surface suit outer protection and pressure seals for both suit and airlock systems are also needed. These technologies must have long-term durability, be damage-tolerant, provide dust-exclusion capabilities, and be nonflammable/oxygen-compatible.

9.3.3 Propulsion

The ESAS architecture requires a variety of propulsion technologies to be evolved or developed in support of ISS and lunar missions. In order to maximize safety during ground processing, launch, and space operations, nontoxic propellants were chosen when possible. Larger, nontoxic monopropellant thrusters need to be developed and human rated for the Launch Vehicle (LV) upper stages, the CM, and, perhaps, the Lunar Surface Access Module (LSAM) ascent stage. Specific propulsion research in support of nontoxic propellants includes developing and demonstrating technologies to enable change in LV upper stage Reaction Control Systems (RCSs) from hydrazine to a nontoxic alternate to enable safe/efficient launch operations, infrastructure reduction, performance improvement, logistics reduction, and potential commonality between main and auxiliary propellants. Critical technologies to enable Tridyne-based attitude control propulsion are required, along with 50- to 100-lbf thrusters to support the CEV CM and other applications.

Missions to the ISS prior to return to the Moon can be accommodated assuming current Space Shuttle Main Engine (SSME) production rates and utilization of SSMEs from existing inventory for the CLV upper stage. The Cargo Launch Vehicle (CaLV) requires two J–2 engines on each upper stage. Research is needed to get the SSME and J–2s to the point to where they can be produced economically for the lunar missions. The SSME is an extremely efficient and capable engine but is expensive and takes years to produce using current production methods. The NASA/U.S. Air Force (USAF) Integrated Powerhead Demonstrator (IPD) is a research activity that has demonstrated production methods that can significantly reduce production bottlenecks and reduce engine part count by up to an order of magnitude, while reducing costs. Applying these methods to the SSME will enable cost and production goals to be met for the lunar missions. A subset of these methods can also be applied to making the J–2s more affordable to produce in support of the lunar missions.

The architecture requires a high Specific Impulse (Isp) propulsion system for the Service Module (SM) and lunar ascent that yields high reliability without significant propellant boil-off issues. A propulsion system developed to perform both functions can also reduce costs. A human-rated 5- to 20-klbf pressure-/pump-fed Liquid Oxygen (LOX)/Methane (CH4) in-space engine and propulsion system is required for the SM and the LSAM ascent stage.

The architecture requires the fueled CEV/SM to remain at the ISS for up to 6 months with the ability to leave the ISS within minutes of notification. Thermal conditioning to enable long-term storage of cryogens will be required, along with the ability to have propellant acquisition after dormant periods in zero gravity. Development and demonstration of critical technologies for cryogenic storage for CEV and outpost surface elements (i.e., LSAM, regenerative fuel cells, ISRU reactant storage) are needed—key fluids are LOX, CH4, and Liquid Hydrogen (LH2). Primary research needs include:

- Tank systems, including: Liquid Acquisition Devices (LADs), passive thermal and pressure control, prototype pressure vessel demonstration, low-gravity mass gauging, active thermal control, and Cryogenic Fluid Management (CFM) integrated system demonstration;
- Main engine systems, including: ignition and combustion characterization, long-life ignition system, and bi-propellant valves;
- RCS engine systems, including: RCS thrusters, Electro-Mechanical Actuators (EMAs) inlet valve for RCS, and RCS chamber materials; and

- RCS feed systems, including: Helium (He) pressurization tank, He control system, prototype cryogenic propellant isolation valve, RCS feed system design and LOX test, CH_4 feed system test, CH_4 specifications and hazards, and integrated RCS demonstration.

The LSAM requires a high-performance, lightweight descent propulsion system that is highly throttleable for the descent module. A LOX/LH2 system maintains adequate margin while also providing a path for utilization of in-situ-produced propellants and eventual LSAM reusability. The LSAM descent stage requires a moderate thrust (5- to 20-klbf) pump-fed, deep-throttling engine. A pump-fed, hydrogen-fueled engine was chosen because of its high Isp and mass savings as compared to a pressure-fed system. This allows the LSAM to perform the circularization burn upon arrival at the Moon, while also maximizing the LSAM cargo delivery capability. The same engines need the capability to restart for the lunar descent with the ability to throttle down to 10 percent of total thrust. As a lunar outpost is established, there is potential to use lunar oxygen and perhaps hydrogen to refuel and reuse the landers. This would require the engines to be capable of many restarts. These new engine capabilities need to be developed, and the RL–10 can be used as the basis for the development.

The ESAS architecture does not address the Mars phase in detail, but it is recognized that traditional chemical propulsion cannot lead to sustainable Mars exploration with humans. Nuclear Thermal Propulsion (NTP) is a technology that addresses the propulsion gap for the human Mars era. NTP's high acceleration and high Isp together enable fast transit times with reasonable initial mass in Low Earth Orbit (LEO). Primary areas of work to be performed in support of future Mars mission include:

- Retire risks and develop high-temperature fuels and materials for NTR operation.
- Identify ground test plans and required facility development. Options include containment with effluent treatment to scrub rocket exhaust of fission products, or use of tunnels at the Nevada Test Site (NTS) to trap exhaust.
- Perform systems analysis to define requirements and engine/system trades (cycle, thrust, Thrust-to-Weight (T/W), Isp).
- Examine feasibility issues including engine clustering, shielding, testing strategy, engine cycle, and use of existing engine components.

9.3.4 Power

Significant gaps exist in power capabilities that are on the critical path to enabling human exploration beyond Earth orbit. The ESAS architecture desires nontoxic fluids to reduce ground processing facility requirements and to increase safety for the crew. Hydrazine (toxic) is currently used to drive the Solid Rocket Booster (SRB) and SSME Auxiliary Power Units (APUs). Research into nontoxic power generation for ESAS LVs is required. ESAS architecture elements, including the CEV, LSAM, and surface systems, require long-life/high-capacity/high-density energy storage on the order of 5 to 10 kW. Lithium ion batteries are required to be human rated at load profiles that are currently higher than state-of-the-art. Fuel cell systems provide power largely independent of environment (solar incidence), which allows greater mission flexibility and will typically provide larger power levels for less total mass for short-duration missions. The ESAS architecture requires advanced fuel cells to meet LSAM and surface system design margins. Radioisotope power sources are a technology option to meet LSAM and surface system mission requirements in support of long-duration surface missions. An outpost will require power of at least 25 kWe, with more required for

ISRU. A technology option for providing outpost power is surface solar arrays, but this option requires some research for array deployment on the 1/6th-g lunar surface. Fission power is a technology option, especially if there are periods of darkness at the outpost location. Power Management and Distribution (PMAD) for a lunar surface power infrastructure will be a new and challenging capability due to the temperature environment and distributed outpost environment.

9.3.5 Thermal Controls

Heat transfer fluids must be selected early for the CEV Active Thermal Control Subsystem (ATCS) and for all other subsequent vehicles because hardware designs are fluid-dependent. Thermal control fluids are desired that not only have good thermophysical properties, but also are safe for use inside the cabin of a vehicle and in radiators. Fluids must be nontoxic, nonflammable, compatible with the Environmental Control and Life Support System (ECLSS), and have freezing temperatures that allow for use in radiators.

The ESAS architecture features lunar surface destinations that have thermal environments much different than deep space. The impact of dust and surface operations and vehicle integration must be incorporated into the thermal system design. Advanced technology development for heat rejection for short-duration missions to the surface of the Moon include:

- Lightweight radiators made from advanced materials. (Significant mass savings are possible—lightweight radiators are predicted to save over 300 kg for lunar missions.)
- Radiators integral to vehicle structure. (Structural radiators may provide as much as a 40 percent mass savings over body-mounted radiators.)
- Coatings and materials for improved performance and dust resistance.
- Evaporative heat sinks for specific mission elements (e.g., lunar descent/ascent or post landing). These have not undergone development since the 1970s; advances include reduced mass, improved controllability, expanded operating range, and increased life.

Advanced technology development for heat rejection is required for long-duration missions to the surface of the Moon, across the harsh conditions of the lunar day and night. Studies on the effects of lunar dust on radiator performance should be performed. Heat pumps are required to elevate the temperature of radiators. Due to the high-temperature environment, lunar missions near the equator during the day cannot use vertically oriented radiators to reject heat into the environment and would require large horizontal radiators. A heat pump enables the use of vertical radiators and greatly reduces radiator size for horizontally mounted radiators. Two-phase ATCSs have been shown to require less power and mass for applications with high-heat loads and long-transport distances. This is because two-phase heat transfer not only uses the sensible energy of the working fluid, but also the latent energy. This provides higher heat transfer coefficients and enables lower mass flow rates. Two-phase systems are desirable for an outpost or base with high internal heat loads.

9.3.6 Avionics and Software

The ESAS architecture requires advanced Integrated Systems Health Management (ISHM) and autonomy beyond that currently utilized in the ISS and Shuttle programs to increase crew safety, increase performance through enhanced autonomy, and reduce operations costs via in-situ diagnostics and mission support. Enhanced ISHM will be required to facilitate lunar outpost activities. The architecture has elements operating from the surface of Earth to the surface of the Moon and back. The radiation environment in space and on the lunar surface can cause electronics to fail in numerous ways. Research is required to make electronics more robust in this environment either with circuit design and/or with shielding. Current crewed system elements have miles of copper wire and data buses that were designed decades ago. It is imperative that research and implementation of advanced crewed spacecraft network solutions be undertaken to increase reliability and robustness and decrease system mass. A substantial amount of new flight software will also need to be developed. A significant amount of the effort associated with this software will be for verification and validation. Enhanced processes and methodologies for developing, validating, and verifying the ESAS element software are needed to enhance safety and reliably and reduce costs.

The ESAS architecture requires the CEV to perform rendezvous and docking with the ISS, the LSAM stack in LEO, and the LSAM in Low Lunar Orbit (LLO) after return from the lunar surface. The CEV performs Automated Rendezvous and Docking (AR&D) when the CEV CM is serving as a pressurized cargo carrier to the ISS. In all other circumstances, the rendezvous and docking is either piloted or facilitated by a human-controlled berthing procedure (unpressurized ISS cargo delivery). Vehicle position, velocity, acceleration, attitude, and attitude rate measurement and estimation are required. The ESAS architecture features a lunar outpost that is gradually built through a series of sortie missions to the outpost location. These sortie missions will be both piloted and automated and will require precision landing and hazard avoidance to ensure outpost deliveries are located properly. The architecture features a near-anytime return capability from the lunar surface to accommodate contingencies. To accommodate these lunar return contingencies, a “skip-entry” guidance system and associated avionics/software need to be developed that can allow the CM to deflect off of the upper atmosphere to phase reentry profiles.

The ESAS architecture requires high data rate communications to support in-space and surface operations. The three primary needs are mission contingency support, science interaction, and public outreach. Sortie and outpost locations may require additional relay antennas or spacecraft. Possible lunar mission sites include some permanent dark regions in craters near the lunar poles. These targets are of interest for scientific and ISRU potential. Low-temperature electronics are needed to enable the sensors, probes, robots, and, eventually, large regolith machinery that may journey into the crater shadows. The architecture features four crew members available for simultaneous EVA while at the lunar outpost for up to 180 days. This is a significant leap beyond the EVA capability that Apollo had and is an opportunity to perform a significant amount of surface science. Technologies associated with science instruments require additional investment. These include: sample acquisition; in-situ chemical, physical, and biological inspection and analysis; sample handling and processing; and sample return. Current modes of on-orbit operations feature different hardware radios for different applications and frequencies. This requires the crew to have access to several different radios to perform certain functions. A software digital radio designed for ISS and lunar sorties may have significant impacts on productivity and hardware requirements for those and future missions.

9.3.7 Environmental Control and Life Support

Technologies for ECLS currently exist for crewed sorties to LEO. This technology is implemented in the ISS and Shuttle systems and sometimes can be large, massive, and unreliable. Research is necessary to reduce mass and volume requirements of the systems while also addressing increased reliability and the lunar surface environment.

The CEV will require atmospheric management technology investments to (1) improve volume efficiency of Lithium Hydroxide (LiOH) by advanced packaging and formulation; (2) reduce mass, volume, thermal, and power requirements of air revitalization system by combining Carbon Dioxide (CO2), moisture, and trace contaminant removal into single vacuum swing system; and (3) identify/develop improved adsorbents and chemisorbents for vacuum swing systems. Additional technology investments for atmospheric management in support of lunar sortie missions include: (1) low-maintenance techniques for removing particulate matter including planetary dust from process air streams, (2) technologies and methods to isolate lander/habitat from external dust contamination, and (3) improvements to multifunctional CO2, humidity, and trace contaminant systems for planetary surface use. In order to support long-duration lunar outpost missions, technologies are needed for the reduction of system consumables. These technologies include regeneration of filters for removal of particulates, alternative low-power/temperature systems for removal and recovery of CO2 using advanced amines and nonamine sorbants, and alternative organic contaminant removal technologies including regenerable adsorbents and thermal and photocatalytic oxidation.

Advanced air and water recovery systems for the CEV and lunar missions are needed to reduce the overall supplies of air and water necessary to sustain humans beyond Earth. These technologies will provide for efficient life-sustaining functions inside spacecraft and planetary surface habitats by decreasing mass, expendables, resupply, energy, volume, heat rejection, and crew time. Some specific needs include: (1) improved pretreatment for urine and stabilization of waste water for longer missions, (2) improved potable water treatment for longer missions, and (3) improved water storage tanks to reduce mass and with considerations for radiation protection. These technologies would improve operability and reliability, and reduce operating buffers and system consumables.

Advanced environmental monitoring and control technologies are required to support crewed lunar missions. Updates to material flammability standards for partial gravity are needed. An integrated suite of reliable environment monitors to detect events and maintain environmental contaminant limits needs to be produced and validated. Information and control systems that provide crew with pertinent environment information that guides actions and design information for mixed human/automated fault recovery are needed, as well as lunar-transit and surface-fire scenarios and training.

9.3.8 Crew Support and Accommodations

Crew support and accommodations include EVA systems, accommodations for crew escape, crew health systems, habitability systems, and radiation exposure management. The ESAS architecture requires the CEV to have EVA capability for all crew members in support of contingencies. An in-space suit is required that can be used for EVA with an umbilical from the CEV. The suit also needs to support emergency depressurization on launch and entry. Current shuttle pressure suits cannot support an EVA. A robust and highly reliable crew escape system to minimize loss of crew is required. This includes an integrated solution that goes across the LV, escape tower, CEV, in-space suit, and the crew accommodations in the CEV. Technology investments are required for an in-space EVA suit system and associated infrastructure support for crew survival from emergency vehicle depressurization. These technologies include a pressure garment with integral EVA capabilities, tools/mobility aids (tethers, etc.) necessary to perform in-space contingency EVA tasks from the CEV, survival equipment for abort conditions, vehicle support equipment required to interface the in-space EVA suit with the CEV, and equipment/ground support facilities required to test/verify in-space EVA systems. The in-space suit and its associated support equipment are an integral part of the crew escape system. Technology needed to accommodate crew escape include foolproof and rapid failure detection capability to detect pad fallback/reconnect at first motion during liftoff. Other technology needs include launch and entry pressure suits with thermal protection and cooling, flexible (constant volume) joints, and helmets. Safety equipment requirements include parachutes and water survival equipment such as life rafts, life jackets, and search-and-rescue Global Positioning System (GPS) beacons for operation by deconditioned crew. A lunar surface EVA suit and associated systems are needed. Shuttle EVA suits are designed for zero gravity and cannot tolerate the lunar surface environment. Apollo suits are no longer available and are not designed for the cold polar environments or with any embedded radiation protection.

Technology for crew health care systems currently exists for crewed sorties to LEO. Some of this technology is implemented in the ISS and Shuttle systems and can be large, massive, and unreliable. Research is needed to reduce mass and volume requirements of the systems while also addressing increased reliability and the lunar surface environment. The architecture requires a system of crew health tools to enable crew performance for surface operations for lunar missions that span both short- and long-duration stays. Technology development is needed to: (1) mitigate identified biomedical risks to ensure capability of crew to perform missions, (2) stabilize and treat for minor medical events and evacuation for selected major medical events, (3) integrate exercise and EVA pre-breathe countermeasures, (4) develop exposure limits for mission and tool design, and (5) advance state-of-the-art technology for vacuum exposure and volume/mass limits.

Habitability systems for lunar sortie missions include the galley (stored-food system), solid-waste management (including trash), crew accommodations, and human factors engineering. Technology investments are required to provide acceptable crew accommodations within tightly constrained vehicle mass and volume; enhanced galley operations in partial gravity and reduced pressure; waste stabilization, volume reduction, and storage; updated human systems interfaces; and reduction in potential for human error-induced mission failures. Technologies that increase crew efficiency and reduce fatigue need to be developed along with those that yield an improvement in maintainability and operational flexibility.

Research that enhances radiation exposure management is needed in support of the lunar outpost due to the long mission times outside of Earth’s magnetic field. The radiation environment is extremely dynamic (a continuous flux of GCRs punctuated by intense fluxes from SPEs). Long-term dosages of GCRs can lead to long-term crew health issues, and SPEs can cause acute radiation sickness. Crew exposures must be managed in real-time to keep them within limits. This requires technology investments to refine nowcasts (i.e., short-term forecasts) of solar outbursts on the sun, forecasts of “all-clear” periods, and accurate forecasts of dose rates versus times at the Moon. It also requires technology developments to enhance active dosimeters and radiation monitors that accompany the crew and report their data back to Earth in real-time. Software development is also needed for modeling the data and training for real-time exposure management.

9.3.9 Mechanisms

Mechanisms perform element operations through moveable, deployable, or articulating devices. They include devices to facilitate landing, docking, and element deployment.

The ESAS architecture currently features a land touchdown for the CEV CM. Technologies for human-rated main chutes and supersonic drogues to enhance landing accuracy need to be developed. Reducing the final impact to acceptable levels requires a touchdown decelerator such as an airbag or retro-rocket in addition to the main parachutes. An integrated system test of the CM recovery systems is required for human rating. This includes deployment of chutes and any type of terminal descent system that would be consistent with a nominal or contingency recovery.

The architecture requires that all crewed elements utilize a common and robust docking mechanism. Shuttle/ISS heritage docking systems (Androgynous Peripheral Attachment System (APAS), Probe and Cone (P/C)) require significant docking impulses that would drive CEV design. Those same systems are not manufactured in the United States. Technology development is required to take current Low-Impact Docking System (LIDS) concepts to the point where they can be incorporated into the CEV and LSAM designs.

The architecture builds its lunar outposts through a series of sortie missions. This leads to a need for a slow build up of smaller components that will require transportation and assembly. Much of this could be done autonomously or via teleoperation. Technology research into surface system deployment methodologies and mechanisms is required. Potential lunar targets also include permanent dark regions in craters near the lunar poles. These targets are of interest for scientific and ISRU potential. Low-temperature mechanisms are needed to enable the sensors, probes, robots, and, eventually, large regolith machinery that may journey into the crater shadows.

9.3.10 In-Situ Resource Utilization (ISRU)

The ESAS architecture has two primary goals for lunar exploration. The first is developing and demonstrating the capabilities needed for humans to go to Mars and the second is lunar science. ISRU is a blend of science and the development of exploration capabilities. Specific requirements for ISRU will change based on what future lunar robotic probes may discover on the surface, but the benefits of reduced logistics and extended mission durations associated with ISRU are highly desirable.

All lunar ISRU processing and construction requires excavation and handling of lunar regolith. Demonstration of effectiveness and regolith abrasiveness and wear characteristics is required before full-up use of ISRU in the outpost phase. Excavation and handling demonstrations of interest include: excavation and trenching down to at least 1 m, berm building up to 3 m in height (for engine plume debris and radiation shielding), and area clearing/leveling for landing area preparation and road construction for dust mitigation. Also, low-gravity dust, regolith handling, and transport characterization testing is required.

The regolith on the Moon contains approximately 45 percent oxygen by mass. Most oxygen extraction methods are applicable to multiple sites of interest for future exploration. Oxygen production for life support and ascent/hopper propulsion during the Outpost phase could significantly reduce the cost, risk, and delivered mass of outpost missions, while increasing mission effectiveness. Demonstration of process efficiency and life characteristics is required before full use during the outpost phase. Until hydrogen/water extraction from lunar poles is demonstrated, extraction of solar wind hydrogen/methane volatiles from regolith should be pursued. Demonstrations should be low-mass and low-cost to allow easy packaging. Early oxygen extraction techniques developed for lunar sortie and initial lunar outpost activities will be of the simplest and lowest risk possible, which usually equates to low extraction efficiency. The ability to evaluate higher risk but higher efficiency/payback techniques is of interest, especially if production levels rise and/or duration of operations is extended (e.g., hydrofluoric acid reduction). Support hardware developed for initial oxygen production hardware should be utilized to the maximum extent possible.

The Lunar Prospector has shown that significant quantities of hydrogen exist at the lunar poles, but the form of hydrogen is unknown (i.e., hydrogen, water, ammonia, methane). Hydrogen and water are extremely important for long-term life-support and propulsion needs. It is critical that a demonstration (1) characterize the form and concentration of hydrogen present, (2) characterize the regolith and environment in the shadowed crater, (3) operate for an extended period in an approximately 40–K-temperature environment, and (4) demonstrate a scaleable extraction and separation concept before the outpost phase. Commonality with Mars water extraction techniques is desired.

9.3.11 Analysis and Integration

The ESAS architecture has identified operational scenarios and crew flight regimes that have not been modeled since the Apollo era. Significant analytical tool development will be required to support mission design, development, and operations along with identification and implementation of analytical standards to facilitate cross-Agency analysis. Trade studies that assess changes in configuration, operations, or technologies to adjust to fluctuating margins and requirements will be needed continuously as designs and technologies mature. Significant cost savings and schedule robustness can be obtained by increasing analysis throughout the program cycle to (1) support key architecture decisions, (2) determine optimal technology investment portfolios, and (3) assess alternative programmatic and architectural “off-ramps” prior to when a contingency may occur.

Investments in analytical tool methodologies, analysis integration, and quantitative technology assessment are required to support the implementation of the ESAS architecture across NASA during the coming decades. Specifically, investments are required to: (1) identify, modify (or develop) and integrate appropriate analytical capabilities to quantitatively model the exploration architecture, missions, systems and technologies; (2) apply and/or develop integration standards to facilitate consistent and defensible analysis and design; and (3) develop and apply a verification, validation, and accreditation approach, while leveraging existing proven tools to the maximum extent possible.

In addition to, and parallel with the above, investments in the application of the analytical methodologies are required to drive analysis capability requirements and yield information critical to the success of the ESAS architecture. These analytical applications include: (1) technology analysis and portfolio assessment supported by investments in technology information collection and management, portfolio development, assessment and recommendations, and ongoing validation of technology development projects and associated impacts on architecture; (2) architecture modeling and analysis supported by investments in advanced concept development and assessment, technology impact assessment, and FOM assessments; and (3) data integration and report development that enable decision makers to rapidly extract significant information.

9.3.12 Operations

The ESAS architecture sets the foundation for exploration systems for the next 30 years. In order to be sustainable and robust, the architecture and its associated elements need to incorporate supportability as a design philosophy from the start. This will be especially important as distances and durations increase.

Technology investments to facilitate forward-commonality and interchangeability of CEV systems hardware with other architecture elements are needed. ISS demonstration of technologies to reduce the outpost logistics footprint will be needed, and continued collaboration with the Department of Defense (DoD) for leveraging common needs for repair and manufacturing is required. Specific technology development/demonstration needs include reprogrammable/reconfigurable systems, ISS demonstration of enhanced repair technologies, ISS demonstration of enhanced maintenance information management capabilities, ISS demonstration of key capabilities for on-demand production of spares, automated work control processes for ground processing/logistics, surface robotic systems for maintenance and repair, enhanced maintenance information management capabilities, techniques for reducing ground processing costs, and robust, damage tolerant, self-repairable systems.

The architecture requires human-system interaction beyond that which is currently utilized in the ISS and Shuttle programs. This is necessary to increase crew safety, increase performance, and reduce operations costs. Technology investments are needed to enhance reliable, real-time data and command interface between humans and systems. This includes research into effective forms of shared control between intelligent systems and humans. Technologies are also required for robotic assistance for humans and the intelligent systems technologies for enabling effective interactions between the robotics and humans. Additional technology investments include: (1) highly reliable dexterous manipulators for hostile environments; (2) multi-modal systems/robots with variable autonomy (autonomous/teleoperable to full human control); and (3) reliable personnel tracking.

Technologies that enable surface operations with respect to transportation of logistics and surface mobility require additional investments. An unpressurized vehicle that can support four crew for a 7-day sortie mission, can be reused, can potentially be operated robotically when uncrewed, can survive 4 years of continuous operation, and is capable of 30-km distance required for the architecture. Technology needs in support of surface mobility include: (1) highly durable, highly reliable, and long-life systems; (2) durable mechanisms and power train; (3) tribology for durable and long life; (4) recharging/refueling capability with extended range; (5) operations in extreme/hostile environment (temperature, dust, radiation); (6) simple maintenance; (7) teleoperations and autonomy; (8) high bandwidth communications; (9) multi-modal teleoperations and autonomy; and (10) robotic operation at enhanced speeds.

9.4 Recommendations

As a result of the technology assessment, it is recommended that the overall funding of ESMD for R&T be reduced by approximately 50 percent to provide sufficient funds to accelerate the development of the CEV to reduce the gap in U.S. human spaceflight after Shuttle retirement. This can be achieved by focusing the technology program only on those technologies required to enable the architecture elements as they are needed and because the recommended ESAS architecture does not require a significant level of technology development to accomplish the required missions. Prior to the ESAS, the technology development funding profile for ESMD is as shown in **Figure 9-1** (included previously in this section). The ESAS recommendations for revised, architecture-driven technology development is as shown in **Figure 9-2** (included previously in this section).

Figures 9-3 through **9-5** show, respectively, the overall recommended R&T budget broken out by program with liens, functional need category, and mission. "Protected" programs include those protected from cuts due to statutory requirements or previous commitments.

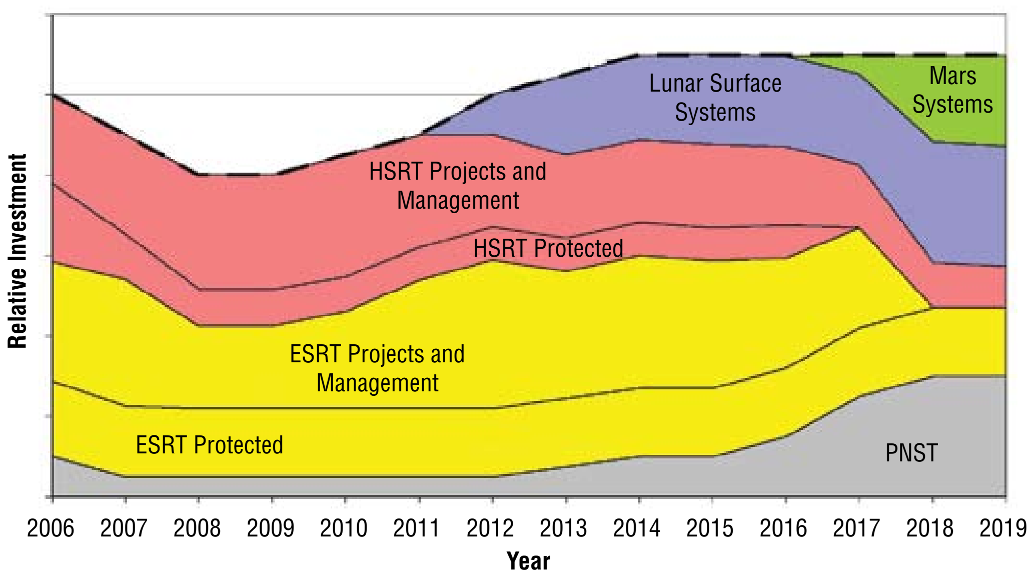


Figure 9-3. Overall Recommended R&T Budget Broken Out by Program with Liens

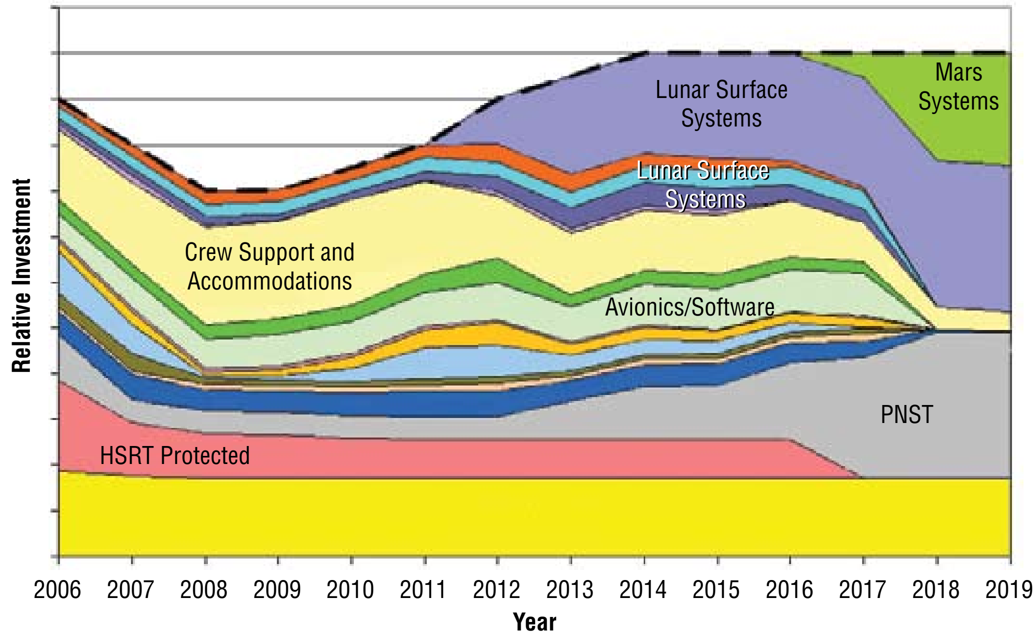


Figure 9-4. Overall Recommended R&T Budget Broken Out by Functional Need Category

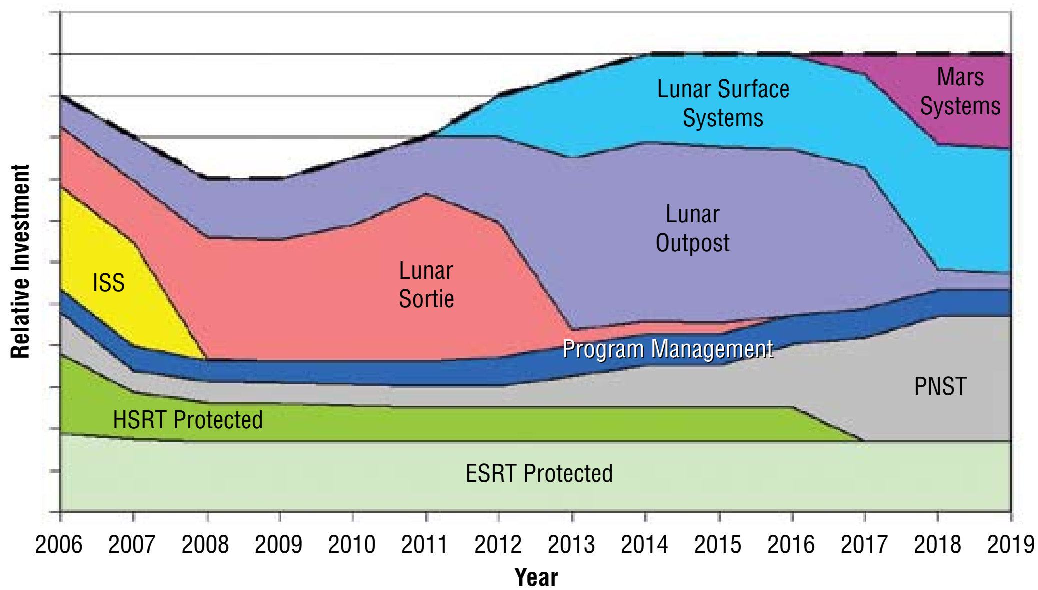


Figure 9-5. Overall Recommended R&T Budget Broken Out by Mission

The existing funding profile includes 10 percent management funds and approximately 30 percent of liens due to prior Agency agreements (e.g., MUSS, the CIR, and the FIR) and legislated requirements (e.g., SBIR, STTR).

The final recommended technology funding profile was developed in coordination with the ESAS cost estimators using the results of the technology assessment. The following seven key recommendations arose from the technology assessment:

- ESMD should share costs with the SOMD for MUSS, CIR, and FIR. MUSS, CIR and FIR are all ISS operations activities and, as such, should not be bookkept in ESMD R&T. Funds were identified in the recommended budget; however, cost-sharing plans should be implemented to ensure these facilities are efficiently operated.
- ESMD should transfer the AMS to the SMD to compete for funding with other science experiments. The AMS may be of scientific importance, but does directly contribute to meeting ESMD R&T needs. Therefore, it should be moved to SMD for consideration with other science missions.
- ESMD should quickly notify existing ESRT projects not selected by ESAS that they will receive no funding beyond FY05. If work on the existing ESRT projects not selected for continuation is not stopped in FY05, there will be a potential for significant FY06 funds required to cover the contracts. Accordingly, appropriate notice must be provided as soon as possible to ensure efficient transition.
- ESMD should move Systems Analysis and Tool Development activities (and budget) to a directorate-level organization—no longer in ESRT. These system analysis and tool development functions should not be buried in multiple disparate organizations. While each organization will require its own analytical capabilities, a focal point should be established at the directorate level to ensure consistency in the ground rules, assumptions, and analytical methodologies across ESMD. This will ensure decision makers are provided "apples-to-apples" analysis results. These activities are also required to handle "what-if" studies and strategic analysis actions to provide greater stability in the development programs (i.e., development programs can focus on their work and avoid the disruption of frequent strategic studies and issue analyses).
- Key ESAS personnel should work with ESMD to facilitate implementation. Many technologies require immediate commencement on an accelerated schedule to meet aggressive development deadlines. Key ESAS personnel should also work with ESMD to ensure the analytical basis supporting ESAS recommendations is not lost, but carefully preserved and refined to improve future decisions.
- ESMD should develop a process for close coordination between architecture refinement studies and technology development projects. Technology projects should be reviewed with the flight element development programs on a frequent basis to ensure alignment and assess progress.
- ESMD should develop a process for transitioning matured technologies to flight element development programs. Experience shows that technologies have a difficult time being considered for incorporation into development projects due to uncertainty and perceived risk. The technologies identified in this assessment are essential for the architecture and, therefore, a structured process for transitioning them must be implemented to ensure timely integration into development projects with minimal risk and uncertainty.

The key technology development project recommendations from the study are shown in **Table 9-1**.

Table 9-1. Technology Project Recommendations

Number	ESAS Control Number	Program	Category	New Projects
1	1A	ESRT	Structures	Lightweight structures, pressure vessel, and insulation.
2	2A	ESRT	Protection	Detachable, human-rated, ablative environmentally compliant TPS.
3	2C	HSRT	Protection	Lightweight radiation protection for vehicle.
4	2E	HSRT	Protection	Dust and contaminant mitigation.
5	3A	ESRT	Propulsion	Human-rated, 5–20 klbf class in-space engine and propulsion system (SM for ISS orbital operations, lunar ascent and TEI, pressure-fed, LOX/CH4, with LADS). Work also covers 50–100 lbs nontoxic (LOX/CH4) RCS thrusters for SM.
6	3B	ESRT	Propulsion	Human-rated deep throttleable 5–20 klbf engine (lunar descent, pump-fed LOX/LH2).
7	3C	ESRT	Propulsion	Human-rated, pump-fed LOX/CH4 5–20 klbf thrust class engines for upgraded lunar LSAM ascent engine.
8	3D	ESRT	Propulsion	Human-rated, stable, nontoxic, monoprop, 50–100 lbf thrust class RCS thrusters (CM and lunar descent).
9	3F	ESRT	Propulsion	Manufacturing and production to facilitate expendable, reduced-cost, high production-rate SSMEs.
10	3G	ESRT	Propulsion	Long-term, cryogenic, storage and management (for CEV).
11	3H	ESRT	Propulsion	Long-term, cryogenic, storage, management, and transfer (for LSAM).
12	3K	ESRT	Propulsion	Human-rated, nontoxic 900-lbf Thrust Class RCS thrusters (for CLV and heavy-lift upper stage).
13	4B	ESRT	Power	Fuel cells (surface systems).
14	4E	ESRT	Power	Space-rated Li-ion batteries.
15	4F	ESRT	Power	Surface solar power (high-efficiency arrays and deployment strategy).
16	4I	ESRT	Power	Surface power management and distribution (e.g., efficient, low mass, autonomous).
17	4J	ESRT	Power	LV power for thrust vector and engine actuation (nontoxic APU).
18	5A	HSRT	Thermal Control	Human-rated, nontoxic active thermal control system fluid.
19	5B	ESRT	Thermal Control	Surface heat rejection.
20	6A	ESRT	Avionics and Software	Radiation hardened/tolerant electronics and processors.
21	6D	ESRT	Avionics and Software	Integrated System Health Management (ISHM) (CLV, LAS, EDS, CEV, lunar ascent/descent, habitat/Iso new hydrogen sensor for on-pad operations).
22	6E	ESRT	Avionics and Software	Spacecraft autonomy (vehicles & habitat).
23	6F	ESRT	Avionics and Software	Automated Rendezvous and Docking (AR&D) (cargo mission).
24	6G	ESRT	Avionics and Software	Reliable software/flight control algorithms.
25	6H	ESRT	Avionics and Software	Detector and instrument technology.
26	6I	ESRT	Avionics and Software	Software/digital defined radio.

Table 9-1. Technology Project Recommendations (continued)

Number	ESAS Control Number	Program	Category	New Projects
27	6J	ESRT	Avionics and Software	Autonomous precision landing and GN&C (Lunar & Mars).
28	6K	ESRT	Avionics and Software	Lunar return entry guidance systems (skip entry capability).
29	6L	ESRT	Avionics and Software	Low temperature electronics and systems (permanent shadow region ops).
30	7A	HSRT	ECLS	Atmospheric management - CMRS (CO2, Contaminants and Moisture Removal System).
31	7B	HSRT	ECLS	Advanced environmental monitoring and control.
32	7C	HSRT	ECLS	Advanced air and water recovery systems.
33	8B	HSRT	Crew Support and Accommodations	EVA Suit (including portable life suppport system).
34	8E	HSRT	Crew Support and Accommodations	Crew healthcare systems (medical tools and techniques, countermeasures, exposure limits).
35	8F	HSRT	Crew Support and Accommodations	Habitability systems (waste management, hygiene).
36	9C	ESRT	Mechanisms	Autonomous/teleoperated assembly and construction (and deployment) for lunar outpost.
37	9D	ESRT	Mechanisms	Low temperature mechanisms (lunar permanent shadow region ops).
38	9E	ESRT	Mechanisms	Human-rated airbag or alternative Earth landing system for CEV.
39	9F	ESRT	Mechanisms	Human-rated chute system with wind accommodation.
40	10A	ESRT	ISRU	Demonstration of regolith excavation and material handling for resource processing.
41	10B	ESRT	ISRU	Demonstration of oxygen production from regolith.
42	10C	ESRT	ISRU	Demonstration of polar volatile collection and separation.
43	10D	ESRT	ISRU	Large-scale regolith excavation, manipulation and transport (i.e., including radiation shielding construction).
44	10E	ESRT	ISRU	Lunar surface oxygen production for human systems or propellant.
45	10F	ESRT	ISRU	Extraction of water/hydrogen from lunar polar craters.
46	10H	ESRT	ISRU	In-situ production of electrical power generation (lunar outpost solar array fabrication).
47	11A	ESRT	Analysis and Integration	Tool development for architecture/mission/technology analysis/design, modeling and simulation.
48	11B	ESRT	Analysis and Integration	Technology investment portfolio assessment and systems engineering and integration.
49	12A	ESRT	Operations	Supportability (commonality, interoperability, maintainability, logistics, and in-situ fab.)
50	12B	ESRT	Operations	Human-system interaction (including robotics).
51	12C	ESRT	Operations	Surface handling, transportation, and operations equipment (Lunar or Mars).
52	12E	ESRT	Operations	Surface mobility.

10. Test and Evaluation

10.1 Approach

Architecture Design, Development, Test, and Evaluation (DDT&E) schedule, costs, and risk are highly dependent on the integrated test and evaluation approach for each of the major elements. As a part of the Exploration Systems Architecture Study (ESAS), a top-level test and evaluation plan, including individual flight test objectives, was developed and is summarized in this section. The test and evaluation plan described here is derived from the Apollo Flight Test Program of the 1960s.

A more detailed test and evaluation plan will be based on detailed verification requirements and objectives documented in specifications and verification plans. In order to support schedule, cost, and risk assessments for the reference ESAS architecture, an integrated test and evaluation plan was developed to identify the number and type of major test articles (flight and ground) and the timing and objectives of each major flight test, including facilities and equipment required to support those tests. This initial plan is based on the Apollo Program and the ESAS Ground Rules and Assumptions (GR&As)—including the human-rating requirements from NASA Procedural Requirements (NPR) 8705.2A, Human-Rating Requirements for Space Systems.

10.2 Ground Rules and Assumptions

ESAS GR&As establish the initial set of key constraints to testing. Although all ESAS GR&As are considered, the specific ones listed below are particularly significant, as they deal with schedule and testing/qualification assumptions.

- The crew launch system shall facilitate crew survival using abort and escape. There will be three all-up tests of the Launch Abort System (LAS).
- Qualification of the Crew Launch Vehicle (CLV) requires three flight tests for human certification prior to crewed flight.
- Qualification of the Crew Exploration Vehicle (CEV) requires a minimum of one flight demonstrating full functionality prior to crewed flights.
- The first CEV human flight to the International Space Station (ISS) will occur in 2011.
- The CEV will support crew to ISS through ISS end-of-life (2016).
- Qualification of the Earth Departure Stage (EDS) for firing while mated to a crewed element requires a minimum of two flights to demonstrate functionality prior to crewed flight.
- Qualification of the Lunar Surface Access Module (LSAM) requires a minimum of one flight demonstrating full functionality prior to a lunar landing.
- Lunar mission rehearsal in-space with appropriate architecture elements and crew is required.
- There is a goal of performing the next human lunar landing by 2020—or as soon as practical.

The ESAS team also considered the guidance of NPR 8705.2A, Human-Rating Requirements for Space Systems. These requirements are identified in **Table 10-1**.

Table 10-1. Significant Human-Rating Requirements Pertaining to Test and Evaluation

1.6.7 Software Testing	"1.6.7.2—Flight software shall, at a minimum, be tested using a flight-equivalent avionics test-bed operating in a real-time, closed-loop test environment (Requirement 34357)."
1.6.8 Flight Testing	"1.6.8.1—In Volume III of the Human-Rating Plan, the Program Manager shall document the type and number of flight tests that will be performed across the mission profile under actual and simulated conditions to achieve human-rating certification (Requirement 34360)."
C.3 Applicability of Requirements	"C.3.1—Human-rating requirements are applicable to any system which transports or houses humans or interfaces with other systems which transport or house humans. Therefore, many uncrewed elements may also be subject to these requirements. For example, currently the expendable launch vehicle is not used in concert with a human-rated system, and so these requirements do not apply. However, if an expendable launch vehicle is used as part of a crewed launch system, human-rating requirements apply."
C.10 Crew and Passenger Survival	"C.10.3.4—Crew escape systems require extensive testing and analysis to verify the functional envelope and environment for system utilization, as well as detailed tests and assessments to ensure the system does not cause a fatality or permanent disability. Due to the dynamic and unpredictable nature warranting the use of crew escape systems, complete verification by integrated flight test is impossible. Crew escape systems may never be considered as a leg of redundancy."

10.3 Apollo as a Reference

The Apollo Program included the development of the Apollo spacecraft, the Saturn V Launch Vehicle (LV), and the Lunar Module (LM), as well as the development of test and launch facilities, launch and mission operations, and the tracking system. The Saturn V LV, Apollo spacecraft, and LM developments were examined to capture the testing approach of each. A summary of the Apollo test program is shown in **Figure 10-1**.

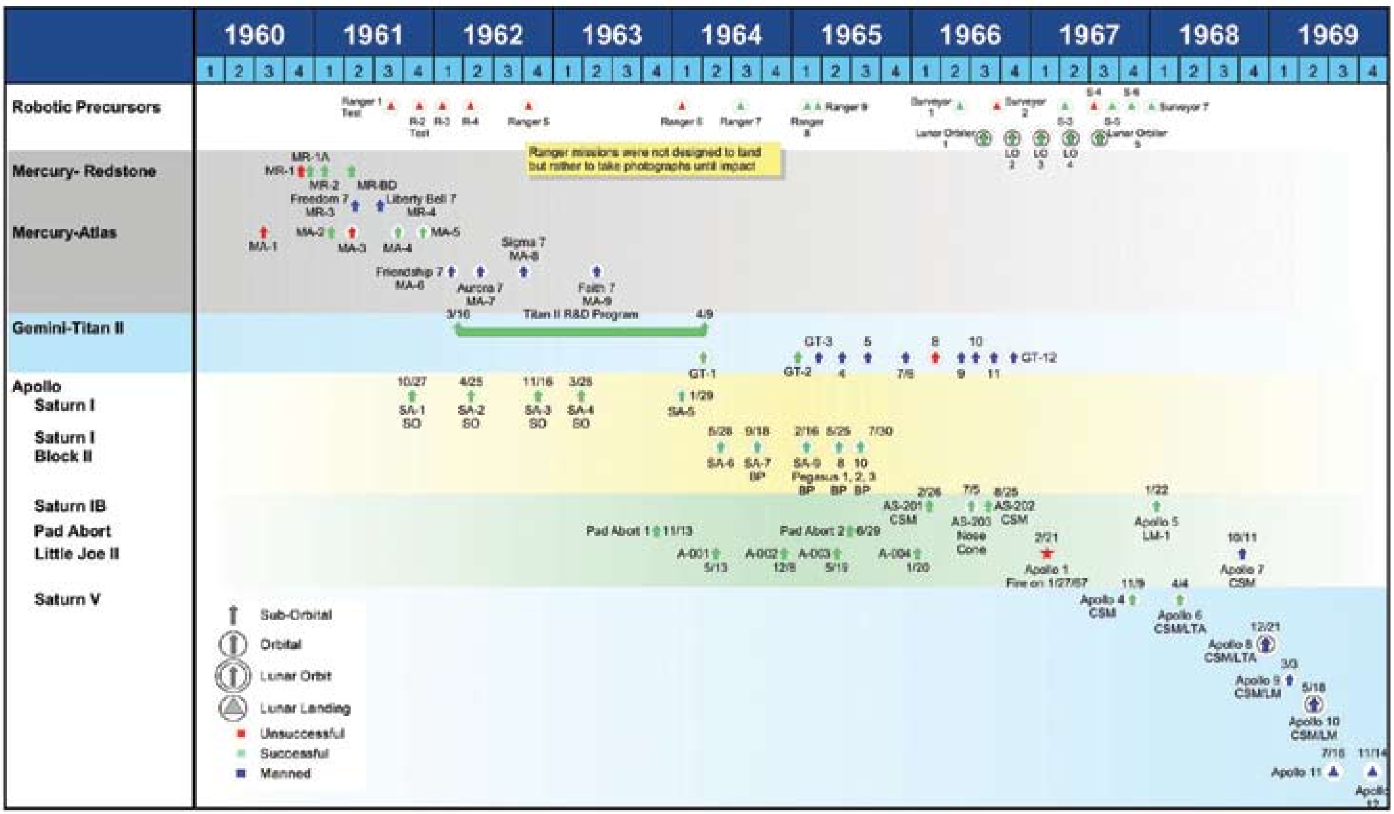


Figure 10-1. Apollo and Other Early NASA Milestones

The relationship between the ESAS test program and the Apollo test program is diagrammed in **Figure 10-2**.

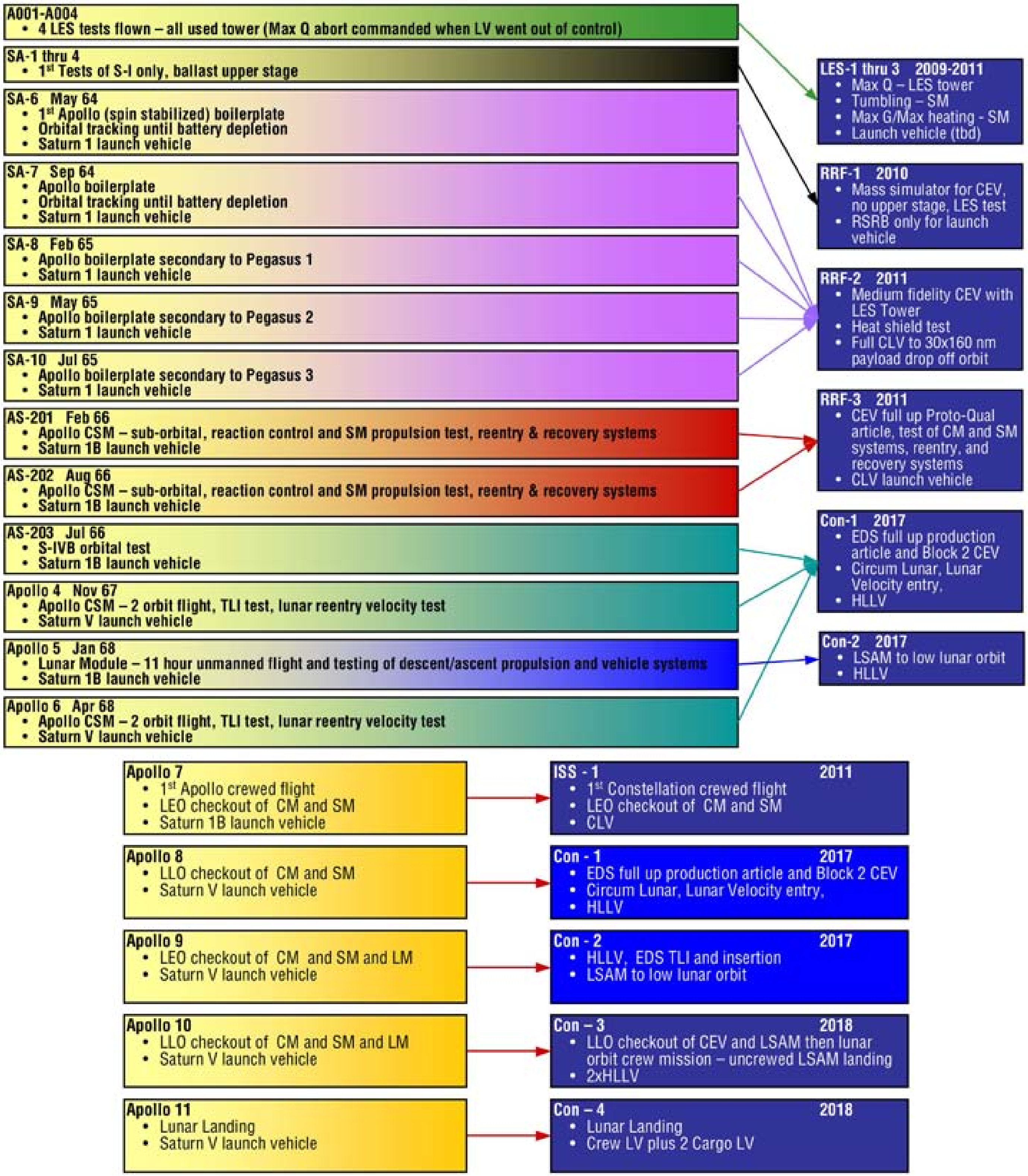


Figure 10-2. Apollo-Constellation Flight Tests Comparisons

10.4 ESAS Flight Test Program

Figure 10-3 reflects the summary flight test program based on the Shuttle-derived solution using a Solid Rocket Booster- (SRB-) derived booster for the CLV and a Space Shuttle Main Engine (SSME-) SRB-derived Cargo Launch Vehicle (CaLV) with Earth Orbit Rendezvous (EOR) and Lunar Orbit Rendezvous (LOR). Test descriptions for these flights with preliminary test objectives are provided in the remainder of this section.

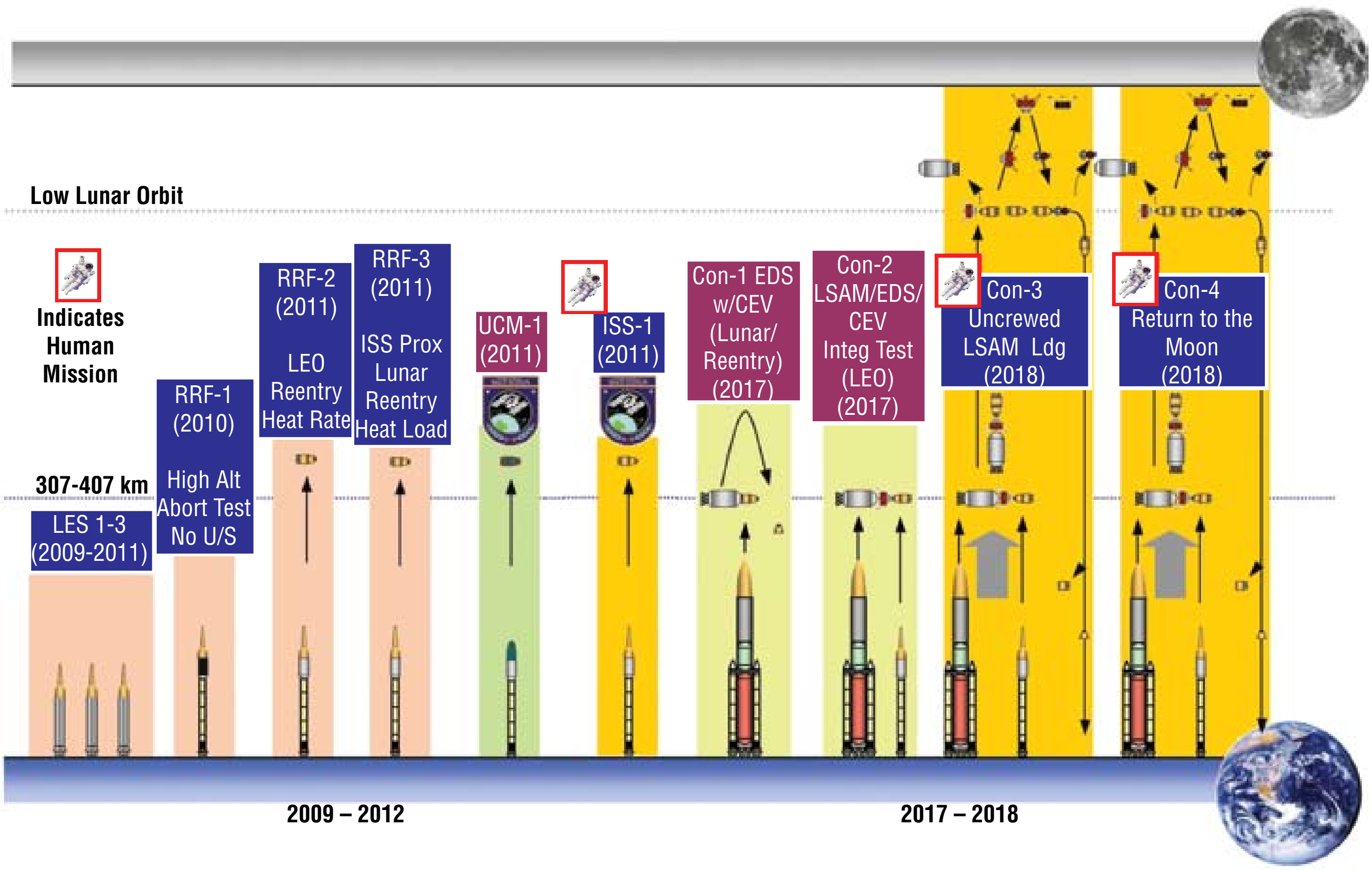


Figure 10-3. Flight Test Program Overview

10.4.1 Pad Abort - 1 (PA–1)

Space Vehicle: Crew Module (CM) Flight Test Article (FTA) (30-percent CM), surrogate Service Module (SM) interface.

> **Systems:** Launch escape tower test article, Outer Mold Line (OML)/mass representative CM, power and communications system for telemetry, parachute system, landing impact attenuation system and recovery system, flight test instrumentation package.

Launch Vehicle: Not Applicable (N/A)—Ground structure to replicate conditions of test.

Objectives:

- Demonstrate satisfactory Launch Escape System (LES) performance in zero velocity, zero-tilt angle conditions.
- Demonstrate ability of LES to detect and initiate pad abort.
- Demonstrate CM/SM interface separation and pyrotechnics performance.

- Demonstrate LES control system capability to deliver CM to safe lateral distance and altitude location for worst-case pad explosion.
- Verify predictions on LES/CM integrated structural dynamics during launch pad abort and validate prediction models.
- Determine vibration and acoustic environment in the CM during low pad abort.
- Obtain data on thermal effects during boost of the launch escape motor plume impingement on the CM.
- Demonstrate LES tower jettison.
- Demonstrate operation of parachute system for low-altitude deployment.
- Demonstrate performance of impact attenuation system during pad abort.
- Demonstrate mission support facilities and operations needed for emergency crew recovery during launch abort.

Parameters: Zero altitude, zero velocity, zero angular rates.

Constraints:

- Early test that may not encompass mass of Block 2 CM.
- Start can be accelerated if chutes/landing are not part of these tests.
- Recovery system performance is critical since refurbishment and reuse of the CM article is planned.

10.4.2 Pad Abort - 2 (PA–2)

Space Vehicle: CM FTA (30-percent CM), surrogate SM interface.

Systems: Launch escape tower qualification unit, OML/mass representative CM, power and communications system for telemetry, parachute system, land-landing impact attenuation system and recovery system, flight test instrumentation package.

Launch Vehicle: N/A – Ground structure to replicate conditions of test.

Objectives:

- Demonstrate satisfactory LES performance in zero altitude, and dynamic angle change at (To Be Determined (TBD)) degrees per second.
- Verify capability of LES to detect and initiate pad abort.
- Verify CM/SM interface separation and pyrotechnics performance.
- Verify LES control system capability to deliver CM to a prescribed landing location that is a safe lateral distance and altitude location for worst-case pad explosion and does not endanger facilities or personnel.
- Verify vibration and acoustic environment predictions in the CM during low pad abort.
- Verify CM acceleration predictions during abort, chute deployment, and impact.
- Verify models on thermal effects during boost and during impingement of the launch escape motor plumes on the CM and the launch escape tower.
- Verify LES tower jettison system performance and safe disposal of tower.
- Verify operation of parachute system for low-altitude deployment.
- Demonstrate water landing.

- Demonstrate mission support facilities and operations needed for emergency crew recovery during launch abort with water impact.

Parameters: Zero altitude, zero velocity, angular rates TBD.

Constraints:

- Early test that may not encompass mass of Block 2 CM.
- It is assumed this test will be conducted at a location that permits a water landing—the test can meet all key objectives with a land landing as accomplished in PA–1.

10.4.3 Launch Escape System - 1 (LES–1) (Ascent Abort Test)

Vehicle: CM FTA (60-percent CM), surrogate SM interface.

Systems: Launch escape tower qualification unit, OML/mass representative CM, Command and Control (C&C) sufficient to initiate abort, power and communications system for telemetry, parachute system, land-landing impact attenuation system and recovery system, flight test instrumentation package.

Launch Vehicle: TBD LV.

Objectives:

- Demonstrate SM-executed (upper stage failures) abort modes at envelope conditions.
- Verify launch-escape power-on stability for abort in maximum dynamic pressure region (max q) with power-on tumbling (loss of control) conditions approximating emergency detection subsystem limits.
- Verify aerodynamic stability characteristics of escape configuration for ascent and abort conditions and collect data on dynamic pressure on CM during abort.
- Verify LES control system capability to recovery from tumbling condition.
- Verify launch escape tower/CM structural integrity during highly stressing abort condition.
- Demonstrate proper separation of CM from SM interface surrogate (reliability).
- Demonstrate capability of escape system to propel CM to predetermined distance from LV (reliability).
- Demonstrate launch escape tower separation following abort (reliability).
- Demonstrate satisfactory operation of parachute, landing system, and recovery systems (reliability).

Parameters: Altitude range—based on LV performance.

Constraints:

- Early test that may not encompass mass of Block 2 CM.
- CM is planned to be reused in subsequent ascent abort tests, so performance of parachutes and landing system is critical.

10.4.4 Launch Escape System - 2 (LES–2) (Ascent Abort Test)

Vehicle: CM FTA (60-percent CM), SM FTA.

Launch Vehicle: TBD LV.

Systems: Launch escape tower production unit, OML/mass representative CM with TPS with development unit heat shield, C&C sufficient to initiate abort, power and communications system for telemetry, CM parachute system, land-landing impact attenuation system and recovery system, flight test instrumentation package. SM equipped with development propulsion system and Reaction Control System (RCS) and associated avionics and structure.

Objectives:

- Verify satisfactory performance of the LES using an SM-initiated abort (after launch escape tower jettison) during max tumble (upper stage loss of flight control) at CEV flight mass.
- Verify separation and steering by SM and CM control system away from a tumbling upper stage to predetermined distance from the LV.
- Demonstrate SM propulsion system and RCS responsiveness to abort command, engine start, throttle up, and operation in a highly dynamic flight environment.
- Verify CM/SM structural integrity during abort from vehicle tumbling condition.
- Demonstrate normal launch escape tower separation under nominal ascent conditions abort (reliability).
- Demonstrate proper separation of CM from SM (reliability).
- Demonstrate CM passive stability following separation from SM and TPS performance under atmospheric heating.
- Demonstrate satisfactory operation of parachute, landing, and recovery systems (reliability).

Parameters: Altitude—desirable to replicate conditions early in upper stage burn.

Constraints:

- Early test may be inadequate for Block 2 CM mass and SM propulsion system performance.
- This test demands high-fidelity SM propulsion system and RCS.
- CM is planned to be reused in subsequent ascent abort tests, so performance of parachutes and landing system is critical.

10.4.5 Launch Escape System - 3 (LES–3) (Ascent Abort Test)

Vehicle: CM FTA (60-percent CM), SM FTA.

Launch Vehicle: LV.

Systems: Launch escape tower flight unit, OML/mass representative CM with development unit heat shield, C&C sufficient to initiate abort, power and communications system for telemetry, CM parachute system, land-landing impact attenuation system and recovery system, flight test instrumentation package. SM equipped with development propulsion system and RCS and associated avionics and structure.

Objectives:

- Verify satisfactory performance of the LES using an SM (or LES through analysis) initiated abort in lofted trajectory case leading to maximum gravity and Thermal Protection System (TPS) heating case.
- Verify separation and steering by SM/CM control system away from an upper stage and into a less-stressing reentry trajectory (maximum turning case).
- Demonstrate SM propulsion system and RCS responsiveness to abort command, engine start, throttle up, and operation (reliability).
- Demonstrate normal launch escape tower separation under nominal ascent conditions, including separation of the boost protective cover by the tower jettison motor and jettison of the forward heat shield by the thrusters (if these two structures are required) (reliability).
- Demonstrate proper separation of CM from SM (reliability).
- Verify CM passive stability following separation from SM and TPS performance under atmospheric heating.
- Demonstrate satisfactory operation of parachute and recovery systems (reliability).
- Demonstrate water landing and recovery force operations needed for emergency crew recovery during launch abort with water impact (deep water).

Parameters: Altitude—replicate conditions leading to max G and maximum heating case.

Constraints:

- Early test that may not encompass mass of Block 2 CM.
- This test demands high fidelity SM propulsion system and RCS.
- CM requires development unit heat shield (minimum).

10.4.6 Risk Reduction Flight - 1 (RRF–1) CLV Development Flight

Space Vehicle: CEV.

Systems: LES production, instrumented OML representative CEV test article with TPS and telemetry downlink capability.

Launch Vehicle: Four-segment Reusable Solid Rocket Booster (RSRB).

Systems: Flight article with a full first stage, simulated mass representative upper stage, flight instrumentation, parachute recovery system.

Objectives:

- Demonstrate CLV booster/first-stage subsystems and avionics performance.
- Demonstrate TVC and roll control performance in first-stage open-loop flight.
- Collect data to validate/anchor/update performance, acoustics, coupled loads, dynamics, and aerodynamic models for CLV and CEV.

- Demonstrate booster and upper stage separation.
- Demonstrate booster parachute systems and recovery.
- Demonstrate overall CLV first-stage performance.
- Obtain data on CEV environment during first-stage burn including dynamic pressure.
- Demonstrate ability of launcher anomaly detection system to identify failure of upper stage (dummy) to initiate abort with LES tower.
- Demonstrate LES performance following ascent on first stage in actual flight environment (reliability).
- Verify performance of RSRB water recovery system.
- Demonstrate mission support facilities and operations needed for launch, mission conduct, and water recovery of RSRB.

Parameters: Based on Stage 1 performance.

Constraints: First-stage test only.

10.4.7 RRF–2/CLV Certification Flight

Space Vehicle: CEV.

Systems: Production LES, ProtoQual 80-percent fidelity CM, 90-percent fidelity SM with telemetry downlink capability.

Launch Vehicle: Four-segment RSRB first-stage with single SSME upper stage.

Systems: Full-up integrated LV with launch abort detection system, SM adapter ring, final flight control system configuration, and full flight test instrumentation.

Objectives:

- Demonstrate upper stage propellant loading and management.
- Verify TVC and roll control performance in first stage.
- Demonstrate upper stage air start and flight control authority.
- Verify booster and upper stage separation dynamics.
- Verify predictions on full LV coupled loads and dynamics including the upper stage.
- Evaluate performance of emergency launch abort detection system.
- Verify performance of booster parachute and water recovery systems.
- Demonstrate all CLV subsystems and avionics performance in full function flight through staging and orbital insertion.
- Demonstrate nominal jettison of LAS tower (reliability).
- Validate performance predictions and models for SM.
- Demonstrate propulsion, maneuvering, navigation, and flight control functions of CEV.
- Demonstrate rendezvous with “virtual ISS” up to point of mating.
- Demonstrate simulated target acquisition for automated rendezvous and mating system checkout including collision avoidance function.
- Demonstrate power production and distribution subsystem performance.
- Demonstrate active thermal control functions.

- Demonstrate communication and remote commanding functions.
- Demonstrate SM deorbit, separation, and disposal.
- Demonstrate nominal-controlled OR failed-to-ballistic entry modes.
- Demonstrate land landing precision, touchdown systems, and recovery.

Parameters: Approximately 450 km (ISS-compatible) circular orbit, 51 deg. Mission duration is 24 hours (TBR).

Constraints: Assumes non-methane propulsion system for Block 1 SM.

10.4.8 RRF–3/CLV Certification Flight

Space Vehicle: ProtoQual Block 1 CEV (last preproduction CEV) with full SM.

> **Systems:** Production launch escape tower system, all CM/SM systems including solar panels and deployment mechanisms, Automated Rendezvous and Docking (AR&D), propulsion (non-methane) flight test instrumentation package, CM reentry, and recovery systems.

Launch Vehicle: CLV (second full certification flight).

> **Systems:** CLV production article with flight test instrumentation.

Objectives:

- Verify upper stage propellant loading and management.
- Validate crewed flight countdown procedures (CLV and CEV).
- Demonstrate Thrust Vector Control (TVC) and roll control performance in first stage (reliability).
- Verify upper stage air start and flight control authority.
- Demonstrate booster and upper stage separation dynamics (reliability).
- Verify performance of emergency launch abort detection system (monitoring mode).
- Demonstrate performance of booster parachute and water recovery systems (reliability).
- Verify all CLV subsystems and avionics performance in full function flight through staging and orbital insertion.
- Demonstrate nominal jettison of LAS tower (reliability).
- Verify propulsion, maneuvering, navigation and flight control functions of CEV.
- Verify relative navigation, targeting/display systems, and automated rendezvous and mating with ISS—assumes ISS outfitted with new adaptor.
- Verify capability of ISS crew to monitor, pilot, and abort rendezvous and mating.
- Verify quiescent (mated) operations with ISS, including interface functions and CEV subsystem performance (power, thermal, communication, etc.).
- Verify electromagnetic environments of CEV and ISS are compatible.
- Validate nominal and emergency ingress and activation procedures and functions.
- Verify crew accommodation (Environmental Control and Life Support (ECLS), etc.) functions.
- Demonstrate SM deorbit, separation, and disposal (reliability).
- Demonstrate performance for faster/steeper controlled entry. (Use residual SM propellant.)

- Demonstrate land-landing precision, touchdown systems, and recovery (reliability).
- Demonstrate operation of and recovery cycle for launch pad and support facilities (reliability).
- Demonstrate mission operations systems and procedures needed for launch and mission conduct (reliability).
- Demonstrate parachute and landing system performance and recovery force operations (confidence).
- Verify suitability of CM for refurbishment and reflight.

Parameters: Approximately 450 km (ISS-compatible) circular orbit, 51 deg. Mission duration is 24 hours (TBR).

Constraints: Assumes ISS outfitted with new adaptor otherwise limited to proximity operations.

10.4.9 Unpressurized Cargo Module - 1 (UCM–1)

Space Vehicle: UCM ProtoQual unit with CEV-derived SM.

Systems: All UCM/SM systems including power, avionics, AR&D (docking or berthing TBD), flight test instrumentation payload.

Launch Vehicle: CLV (can support certification of LV).

Systems: CLV production article with flight test instrumentation, qualification shroud.

Objectives:

- Verify predictions of LV coupled loads and vehicle dynamics with shroud.
- Verify and anchor aerodynamic models and wind-tunnel data for entire LV with shroud.
- Demonstrate jettison of shroud.
- Verify LV guidance subsystems performance with shroud.
- Verify CLV upper stage/UCM separation characteristics and parameters meet requirements.
- Verify operation and performance of all UCM avionics, rendezvous systems, and propulsion systems.
- Demonstrate AR&D system by conducting rendezvous and proximity operations with ISS (reliability).
- Verify electromagnetic environments of CEV and ISS are compatible.
- Verify ground processing and flight operations for UCM and UCM/CLV configurations.

Parameters: Approximately 450 km (ISS compatible) circular orbit, 51 deg. Mission duration is 24 hours (TBR).

Constraints:

- Only non-production UCM.
- Assumes ISS outfitted with new adaptor.
- Assumes substantial commonality with CEV systems.
- ESAS test and evaluation studies did not address qualification of shroud.

10.4.10 ISS–1 – Crewed Flight

Space Vehicle: Block 1 Production CM with SM and LES.

Systems: Production system, all CM/SM systems including AR&D, flight test instrumentation payload.

Launch Vehicle: CLV (fourth full-up flight)

Systems: CLV production article with flight test instrumentation.

Objectives:

- Verify CLV and CEV performance with human crew.
- Verify operation of all CEV subsystems with full human crew.
- Demonstrate rendezvous and docking with ISS.
- Demonstrate mission operations and full recovery force employment for human mission.

Parameters: Approximately 450 km (ISS-compatible) circular orbit, 51 deg. Mission duration is 24 hours (TBR).

Constraints: None.

Assumptions: No assessment of reuse of the CEV was made in this test and resulting flight program. Fundamental assumptions must be developed addressing:

- Whether ISS–1 delivers a crew that starts the 6-month on-orbit crew cycle;
- Whether the CEV for ISS–1 remains docked to ISS for 6 months to serve as a lifeboat; and
- The number of times a CEV is reused and the associated rationale for that assumption.

10.4.11 Constellation-1 – Uncrewed Mission (Low Earth Orbit (LEO))

Heavy-Lift Launch Vehicle - 1 (HLLV–1)/Earth Departure Stage - 1 (EDS–1)/CEV (Block 2)

Space Vehicles:

CEV

Systems: RRF–3 CM refurbished to Block 2 configuration with lunar mission avionics, subsystems and heat shield. Production LES and Block 2 SM with high Specific Impulse (Isp) propulsion system (assume methane) and other lunar mission subsystems.

EDS

Systems: Full EDS development flight test article and flight test instrumentation, restartable, with J–2 engine and avionics common with upper stages of CLV. Also includes all EDS autonomous operations avionics, telemetry systems, attitude control, extended flight power systems, and propellant management system for storing cryogenic propellants (Liquid Oxygen/Hydrogen (LOX/H2)) for multiple weeks.

Launch Vehicles:

HLLV (first certification flight for crewed operations)

Systems: HLLV development article (initial flight) with full flight test instrumentation. The configuration of the HLLV is based on Space Transportation System (STS-) derived elements including two five-segment RSRB boosters (common with the CLV) and five SSMEs on the base of an STS-derived LOX/H2 tank. Vehicle is an in-line configuration with the RSRBs attached to the main propellant tank and the payload carried atop it.

Objectives:

HLLV

- Verify propellant loading and management.
- Verify TVC and roll control performance of vehicle.
- Verify predictions on LV coupled loads and vehicle dynamics.
- Verify and anchor aerodynamic models and wind-tunnel data for entire LV.
- Demonstrate upper stage air start and flight control authority.
- Verify RSRB and core separation dynamics.
- Verify core and upper stage separation dynamics.
- Verify predictions on full LV coupled loads and dynamics including the upper stage.
- Demonstrate operation of and recovery cycle for launch pad and support facilities.
- Demonstrate mission operations needed for launch and mission conduct.

EDS

- Demonstrate structural integrity and verify loads and dynamic characteristics during propulsion system burns and validate mode models with solar power system deployed (if used).
- Demonstrate multiple restarts with CEV and validate control models by performing attitude control as integrated stack.
- Conduct burns after 14-day cold soak and verify propellant management and storage during extended on-orbit operations.
- Conduct burn to depletion to verify shutdown characteristics.
- Verify overall vehicle performance and thermal predictions and operation of all EDS subsystems including solar panel deployment and operation (if used) supporting certification for crewed use (Flight 1).
- Demonstrate flight operations needed for conduct of EDS mission.

CEV

- Verify performance of all Block 2 subsystems including autonomous operations modes and methane SM propulsion system.
- Collect data on ascent environment and dynamics with HLLV and during staging events.
- Verify integrity of EDS/CEV flight configuration during burns of EDS propulsion system.
- Collect data on environmient and dynamics during EDS burns including SM solar arrays.
- Validate CEV thermal models in stacked configuration and verify thermal compatibility.
- Verify electromagnetic environments of CEV and EDS are compatible.
- Verify CEV/EDS separation and associated interface terminations and dynamics.
- Verify CM heat shield performance during entry at lunar return velocity.
- Collect entry environmental data.
- Demonstrate mission operations during extended autonomous mission operations.

Parameters: 307- to 407-km circular orbit, 28.5 deg. Mission duration is 14 days minimum representing short lunar stay mission.

Constraints: Assume test program for Block 1 LES sufficient for Block 2 CEV.

Mission Profile:

- HLLV launch.
- HLLV jettisons spent RSRBs.
- HLLV shroud is jettisoned.
- HLLV delivers EDS/CEV to 30- x 160-nmi phasing orbit.
- HLLV upper stage and EDS/CEV separate.
- EDS performs burn(s) to circularize orbit (307–407 km).
- EDS conducts RCS burns while docked with CEV to verify controllability.
- CEV maintains attitude control during EDS burn to demonstrate controllability in contingency mode.
- EDS enters quiescent operations mode to demonstrate 2-week delay between Constellation element launches.
- EDS conducts burn to depletion to verify system performance predictions and put CEV on Earth-intersecting trajectory.
- CEV jettisons EDS and CEV conducts multiple Main Propulsion System (MPS) burns to verify Block 2 SM performance.
- CEV SM performs burn to depletion to accelerate the CM toward lunar return velocity (approximately 11 km).
- CEV conducts reentry at lunar return velocity for normal land recovery.

10.4.12 Constellation-2 – Uncrewed Mission (LEO)

HLLV/EDS/LSAM

CLV/CEV

Space Vehicles:

EDS

Systems: Full production article EDS with subset flight test instrumentation.

LSAM

Systems: Full LSAM ProtoQual FTA and full flight test instrumentation. Throttleable, CEV-derived methane engine, full set of vehicle systems to support crewed flight, propellant management system for storing cryogenic propellants (LOX/methane) for multiple weeks.

CEV

Systems: Full production Block 2 CEV with subset flight test instrumentation.

Launch Vehicles:

HLLV (second certification flight for crewed operations)

Systems: Production HLLV with subset of flight test instrumentation and new shroud.

CLV

Systems: Full production article with five-segment RSRB as tested on ISS resupply missions.

Objectives:

HLLV

- Demonstrate propellant loading and management (reliability).
- Demonstrate TVC and roll control performance of vehicle (reliability).
- Demonstrated performance and interaction of five SSME core engines and SSME/RSRB interaction during flight conditions (reliability).
- Demonstrate upper stage air start and flight control authority (reliability).
- Demonstrate RSRB and core separation dynamics (reliability).
- Demonstrate core and upper stage separation dynamics (reliability).
- Validate overall vehicle performance predictions and verify compatibility of HLLV, EDS, and LSAM.
- Demonstrate mission operations needed for launch and mission conduct.

CLV (five-segment)

- Demonstrate propellant loading and management of upper stage (reliability).
- Demonstrate TVC and roll control performance of vehicle (reliability).
- Demonstrate all staging events (reliability).
- Verify predictions on LV coupled loads and vehicle dynamics in shroud configuration with Block 2 CEV.
- Validate overall vehicle performance predictions and verify compatibility of five-segment CLV and Block 2 CEV.
- Demonstrate mission operations needed for launch and mission conduct.

EDS

- Demonstrate multiple restarts with LSAM and validate control models by performing attitude control as integrated stack with the CEV.
- Conduct burns after 14-day cold soak and verify propellant management and storage during extended on-orbit operations.
- Verify thermal predictions in EDS/LSAM/CEV configuration.
- Verify overall vehicle performance supporting certification for crewed use (Flight 2).

LSAM

- Verify performance of all subsystems (power, thermal, propulsion, avionics, Environmental Control and Life Support System (ECLSS)) including autonomous operations modes and methane propulsion system.
- Collect data on ascent environment and dynamics with HLLV and during staging events.
- Conduct LSAM RCS burns to verify LSAM/EDS configuration dynamics and controllability for AR&D with CEV and to validate LSAM control system models.
- Demonstrate LSAM navigation and control systems for conducting an AR&D with the CEV.
- Verify integrity of EDS/LSAM/CEV flight configuration during burns of EDS propulsion system.

- Collect data on environment and dynamics during EDS burns including LSAM solar arrays deployed.
- Conduct LSAM RCS burns to verify LSAM/CEV configuration dynamics and controllability and to validate LSAM control system models.
- Validate LSAM thermal models in stacked configuration with EDS and verify thermal compatibility.
- Verify electromagnetic environments of LSAM, CEV, and EDS are compatible.
- Verify LSAM–EDS separation and associated interface terminations and dynamics.
- Verify operation of descent stage jettison during propulsion system operation (fire-in-the-hole abort).
- Demonstrate LSAM controllability and operations after jettison of descent stage.
- Demonstrate mission operations during extended autonomous mission operations.

CEV Block 2

- Verify performance of all subsystems (power, thermal, propulsion, avionics, ECLSS) including autonomous operations modes and methane propulsion system.
- Collect data on ascent environment and dynamics with CLV and during staging events.
- Demonstrate CEV navigation and control systems for conducting an AR&D with the LSAM/EDS stack.
- Verify integrity of EDS/LSAM/CEV flight configuration during burns of EDS propulsion system.
- Conduct CEV RCS burns to verify LSAM/EDS/CEV configuration dynamics and controllability and to validate CEV control system models.
- Validate CEV thermal models in stacked configuration with LSAM and verify thermal compatibility.
- Verify LSAM/CEV separation and associated interface terminations and dynamics.
- Verify operation of MPS during burn to depletion.
- CEV conducts reentry at lunar return velocity for normal land recovery.
- Demonstrate mission operations during extended autonomous mission operations.

Parameters: 307- to 407-km circular orbit, 28.5 deg. Mission duration is 14 days minimum representing short lunar stay mission.

Constraints: None

Mission Profile:

- HLLV launch.
- HLLV jettisons spent RSRBs.
- HLLV jettisons shroud.
- HLLV delivers EDS/LSAM to 30- x 160-nmi phasing orbit.
- HLLV upper stage and EDS/LSAM separate.
- EDS performs burn(s) to circularize orbit (307–407 km).
- LSAM conducts RCS burns docked with EDS to verify integrity of configuration.

- EDS conducts RCS burns while docked with LSAM to verify controllability.
- LSAM maintains attitude control during EDS burn to demonstrate controllability in contingency mode.
- EDS enters quiescent operations mode to demonstrate 2-week delay between Constellation element launches.
- CLV launches with CEV.
- CLV staging followed by LES jettison.
- CLV delivers CEV to 30- x 160-nm phasing orbit.
- CEV conducts SM main propulsion burn to circularize.
- LSAM/EDS conduct active rendezvous and docking with CEV.
- CEV and LSAM/EDS undock and separate.
- CEV conducts active rendezvous and docking with LSAM/EDS.
- EDS conducts multiple burns to demonstrate control dynamics of integrated CEV/LSAM/EDS stack including with burns using LSAM and CEV RCS for control in a contingency operation. EDS changes orbit from circular to highly elliptical.
- EDS is jettisoned from LSAM/CEV stack and conducts burn to depletion supporting crewed certification and leading to destructive reentry.
- LSAM and CEV alternate control of vehicle to verify controllability during propulsion system burns.
- LSAM and CEV undock and the LSAM conducts simulated descent abort and transition to ascent system.
- CEV performs simulated autonomous maneuver to retrieve the LSAM; then the LSAM and CEV rendezvous and dock.
- LSAM conducts a burn to depletion of propulsion system to accelerate CEV for entry.
- CEV conducts SM burns to accelerate for lunar return entry velocity.
- LSAM conducts destructive reentry.
- CEV conducts reentry at lunar return velocity for normal land recovery.

10.4.13 Constellation-3 – Crewed Mission (Lunar Orbit)

HLLV/EDS/LSAM

CLV/CEV

Space Vehicles:

EDS

Systems: Full production EDS (launch with LSAM and orbit circularization achieves three flight certification for crewed operations).

LSAM

Systems: Full-production LSAM.

CEV

Systems: Full-production Block 2 CEV.

Launch Vehicles:

HLLV (launch with EDS/LSAM is third certification flight for crewed operations)

Systems: Full-production HLLV with shroud.

CLV (launch with CEV)

Systems: Full-production CLV.

Objectives:

- Verify and demonstrate performance of all lunar mission elements in LEO and lunar orbit.
- Demonstrate CEV and LSAM/EDS AR&D in LEO and lunar orbit (reliability).
- Verify CEV autonomous operation in lunar orbit.
- Verify LSAM operation in lunar orbit with full crew complement.
- Demonstrate and rehearse LSAM accessibility and systems performance by conducting EVA in lunar orbit.
- Demonstrate automated, uncrewed LSAM descent to lunar surface and landing.
- Verify LSAM systems operations on lunar surface.
- Demonstrate LSAM launch and ascent from lunar surface to AR&D with CEV.
- Demonstrate CEV AR&D with LSAM in lunar orbit in rescue mode.
- Assess LSAM survival unattended in lunar orbit.
- Verify mission operations plans and procedures for lunar landing mission.

Parameters:

- LEO: 307- to 407-km circular orbit, 28.5 deg.
- Lunar orbit: 100- x 500-km polar orbit with 0 deg inclination.
- Duration: 5-day lunar orbit mission. Total mission duration from first element launch to CEV landing is 25 days.

Constraints: None—all elements must operate as planned for lunar landing.

Mission Profile:

- EDS/LSAM launch on HLLV.
- CEV launches on CLV 14 days later.
- EDS/LSAM conduct AR&D with CEV.
- Crew fully checks out LSAM.
- EDS/LSAM/CEV performs Trans-Lunar Injection (TLI) burn.
- EDS conducts lunar orbit injection burn for LSAM/CEV and is jettisoned.
- LSAM and CEV rendezvous and dock.
- Full crew transfers to LSAM and completes vehicle checkout.
- Crew conducts EVA from LSAM to demonstrate vehicle ingress and egress and simulate lunar surface operations.
- Crew splits between CEV and LSAM; then the LSAM and CEV undock and conduct proximity operations while crew conducts complete checkout of LSAM.
- LSAM and CEV dock and full crew transfers to CEV.

- LSAM undocks and performs burn to initiate descent toward lunar surface.
- LSAM conducts lunar landing at Constellation 4 landing site and remains on surface for 2 days (TBR).
- LSAM jettisons descent stage and ascends to AR&D with CEV.
- Crew splits between LSAM and CEV and crew checks systems after descent and landing.
- LSAM and CEV undock and crew practices lunar orbit maneuvering in LSAM ascent stage.
- LSAM and CEV dock and crew returns to CEV.
- LSAM powered down for long-duration life testing in lunar orbit.
- CEV conducts Trans-Earth Injection (TEI) burn and direct entry recovery.

10.4.14 Constellation-4 – Crewed Mission (Lunar Landing)

HLLV/EDS/LSAM

CLV/CEV

Space Vehicles:

EDS

Systems: Full-production EDS.

LSAM

Systems: Full-production LSAM.

CEV

Systems: Full-production Block 2 CEV.

Launch Vehicles:

HLLV

Systems: Full-production HLLV with shroud.

CLV

Systems: Full production CLV.

Objectives: Return Americans to the Moon.

Parameters:

- LEO: 307- to 407-km circular orbit, 28.5 deg.
- Lunar orbit: 100- x 500-km polar orbit, 0 degrees inclination.
- Duration: 4-day lunar surface stay. Total mission duration from first element launch to CEV landing is 25 days (TBR).

Constraints: It is assumed on the first lunar landing with crew that the CEV will retain a subset of the normal crew complement.

Mission Profile:

- EDS/LSAM launch on HLLV.
- CEV launch on CLV 14 days later.
- EDS/LSAM conduct AR&D with CEV.
- Crew fully checks out LSAM.
- EDS/LSAM/CEV performs TLI burn.
- EDS conducts lunar orbit injection burn for LSAM/CEV and is jettisoned.
- Full crew transfers to LSAM and completes vehicle checkout.
- Crew splits between CEV and LSAM; then the LSAM and CEV undock and conduct proximity operations while crew conducts complete checkout of LSAM.
- LSAM performs burn to initiate descent toward lunar surface.
- LSAM conducts lunar landing and remains on surface for 4 days (TBR).
- LSAM jettisons descent stage and ascends to AR&D with CEV.
- LSAM and CEV dock and crew returns to CEV.
- LSAM powered down for long-duration life testing in lunar orbit.
- CEV conducts TEI burn and direct entry recovery.

11. Integrated Master Schedule

11.1 Introduction and Ground Rules

The Exploration Systems Architecture Study (ESAS) Integrated Master Schedule (IMS) is an integrated, logically connected set of activities and milestones. There are 22 separate individual schedule networks that make up the IMS. The major individual logic schedule networks are:

- ESAS Systems Engineering and Integration (SE&I),
- Crew Exploration Vehicle (CEV),
- Crew Launch Vehicle (CLV),
- Cargo Launch Vehicle (CaLV),
- In-Space Transportation Systems (includes the Earth Departure Stage (EDS) and Lunar Surface Access Module (LSAM)),
- In-Space Support Systems (includes communication and navigation activities),
- Launch Operations and Ground Support Systems, and
- Mission Operations.

Key Ground Rules and Assumptions (GR&As) are listed below:

- The ESAS IMS reflects the recommended launch scenario for the ISS and lunar missions/flights.
- The ESAS IMS includes all activities from the test and evaluation plan outlined in **Section 10**, **Test and Evaluation**.
- The ESAS IMS includes activities associated with all CEV Block options.
- The CEV Crew Module (CM) is reusable.
- The Automated Rendezvous and Docking (AR&D) module is reusable.
- The CEV Service Module (SM) is expendable.
- The CEV Launch Escape System (LES) is expendable.
- Design, manufacturing, assembly, and test for CEV will continue until the first crewed CEV mission to the International Space Station (ISS) (ISS–1).

11.2 Summary IMS

The schedule shown in **Figure 11-1** summarizes the activities/milestones associated with the ISS flights/missions and the initial lunar landing flights/missions in 2018. The more detailed ESAS IMS is provided in **Appendix 11A**, **Detailed Integrated Master Schedule**.

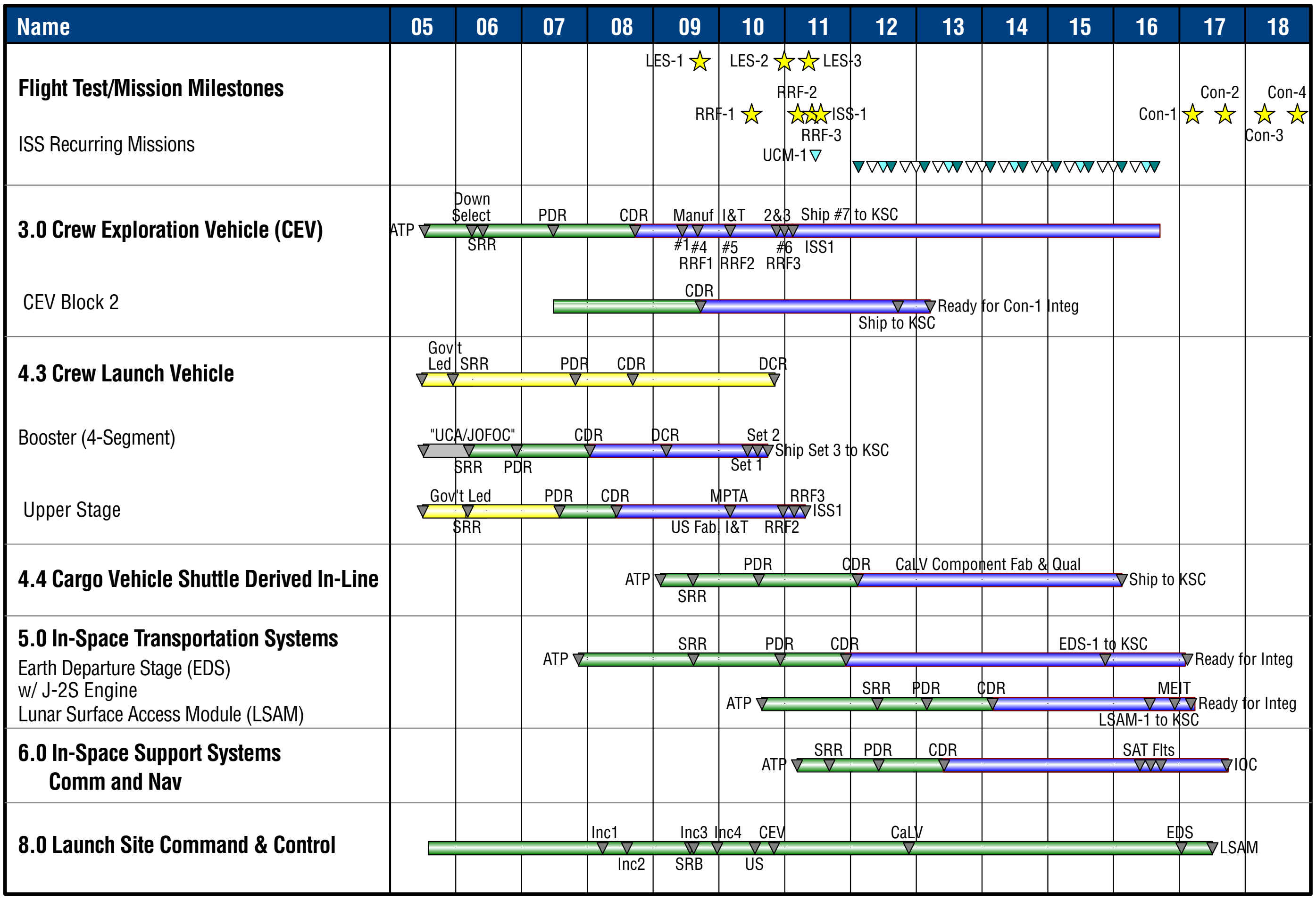


Figure 11-1.
Summary IMS

12. Cost

12.1 ESAS Cost Analysis Context

Current NASA cost projections for the Exploration Vision are based on the Exploration Systems Architecture Study (ESAS) recommended architecture. The estimates are based on parametric cost models, principally the NASA and Air Force Cost Model (NAFCOM). The cost analysis attempted to be conservative. For example, NAFCOM assumes the historical levels of requirements changes, budget shortfalls, schedule slips, and technical problems. If the Vision program maintains stable requirements, is provided timely funding, and incurs fewer technical issues (due to the simplicity and heritage of the approach), cost should be lower than historical norms. Cost credits were not taken for such outcomes, nor do the estimates reflect desired commercial activities that might develop needed cargo and crew services

On the other hand, the ESAS was a Phase A concept study. The designs will mature as in-house and contractor studies proceed. Costs will be revisited at Systems Requirement Review (SRR) and Preliminary Design Review (PDR), including non-advocate independent cost estimates. A firm commitment estimate is not possible until PDR.

Finally, critical procurement activity is currently underway and Government cost estimates are being treated as sensitive information. Accordingly, all cost results are provided in the procurement-sensitive appendix, **Appendix 12A**, **Procurement-Sensitive Cost Analysis**.

12.2 Major Cost Conclusions

The ESAS effort considered a wide trade space of space transportation, space vehicle, and ground infrastructure options that are discussed throughout this report. The final recommended architecture resulted, in large part, from selections made on the basis of cost. First, Shuttle-derived Launch Vehicles (LVs) were found to be more economical, both in nonrecurring and recurring cost terms, than the other major alternatives considered—various configurations of Evolved Expendable Launch Vehicle (EELV)-derived launchers. Specifically, the most economical Crew Launch Vehicle (CLV) was found to be the four-segment Solid Rocket Booster (SRB) in-line vehicle with a Space Shuttle Main Engine (SSME) upper stage. Shuttle-derived in-line Heavy-Lift Launch Vehicles (HLLVs) were also found to be more economical than their EELV-derived counterparts. The Crew Module (CM) part of the Crew Exploration Vehicle (CEV) was baselined to be reusable, which resulted in significant Life Cycle Cost (LCC) savings. Using the (CEV) and CLV combination to service the International Space Station (ISS) results in an average annual cost that is approximately $1.2B less than the current cost of using the Shuttle to service the ISS. Various lunar mission modes were considered and costed. While the direct-to-the-lunar-surface mission mode resulted in lowest overall cost, the Earth Orbit Rendezvous–Lunar Orbit Rendezvous (EOR–LOR) option was actually selected for other reasons and was only marginally higher in cost. A CEV and CLV funding profile for first flight in 2011 was recommended that offers an acceptable cost and schedule confidence level, but exceeds planned budgets in some years. Beyond 2011, the development of the lunar LV and other elements again results in a relatively modest over-budget situation in certain years that can be addressed by additional design-to-cost approaches. Subsequent to the ESAS effort, NASA baselined a 2012 first flight for CEV and CLV, which allows the program to be accomplished within available budget.

12.3 Top-Level Study Cost Ground Rules

The major ESAS cost assumptions are listed below:

- Cost is estimated in 2005 dollars in full cost (including civil service and corporate General and Administrative (G&A)).
  - Cost is converted to inflated ("real year") dollars only for the "sand chart" budget overviews.
- Cost reserves of 20 percent for Design, Development, Test, and Evaluation (DDT&E) and 10 percent for production and operation cost are included.
  - Probabilistic cost risk analysis performed later in the study verified that this reserve level is acceptable.
- Cost estimates are formulation estimates and, as such, are considered preliminary.
- Any cost estimates supplied by contractors are vetted by independent Government estimators.
- All cost estimates reflect today's productivity levels and modern engineering processes.
- The costs include all civil service salaries and overheads and all Government "service pool" costs ("full cost" in NASA terms).

The cost estimates include all LLC elements from DDT&E through operations.
These elements are listed below:

- DDT&E;
- First flight unit;
- Test flight hardware costs;
- Hardware annual recurring cost (split between fixed and variable);
- Operations capability development;
- Facilities and facilities Maintenance and Operations (M&O);
- Hardware operations costs:
  - Flight operations (fixed and variable);
  - Launch operations (fixed and variable); and
  - Sustaining engineering, spares, and logistics.
- Flight and ground software;
- Full cost adds:
  - Civil servant; and
  - Support contractors.
- Reserves.

Ground test hardware, test flight hardware, and test operations were all included to be consistent with the test plan reported separately in this report. Early operations capability development at the launch and mission control sites at NASA Kennedy Space Center (KSC) and Johnson Space Center (JSC) were included. Production and operations costs were book-kept as annual fixed cost and variable cost-per-flight to properly account for rate effects across varying flight rates per year.

12.4 Cost-Estimating Participants

As shown in **Figure 12-1**, several NASA Centers and NASA Headquarters (HQ) participated in the cost-estimating activities. JSC estimated the CEV, landers, surface systems, Launch Escape System (LES), and mission operations. NASA Marshall Space Flight Center (MSFC) was responsible for all space launch and transportation vehicle development and production costs. KSC estimated the launch facilities and launch operations costs. NASA Glenn Research Center (GRC) provided the lunar surface power systems cost estimates. The costs were integrated by the ESAS team at NASA HQ, which also handled the interface with the Shuttle/ISS configuration. Final budget integration and normalization was done by NASA HQ.

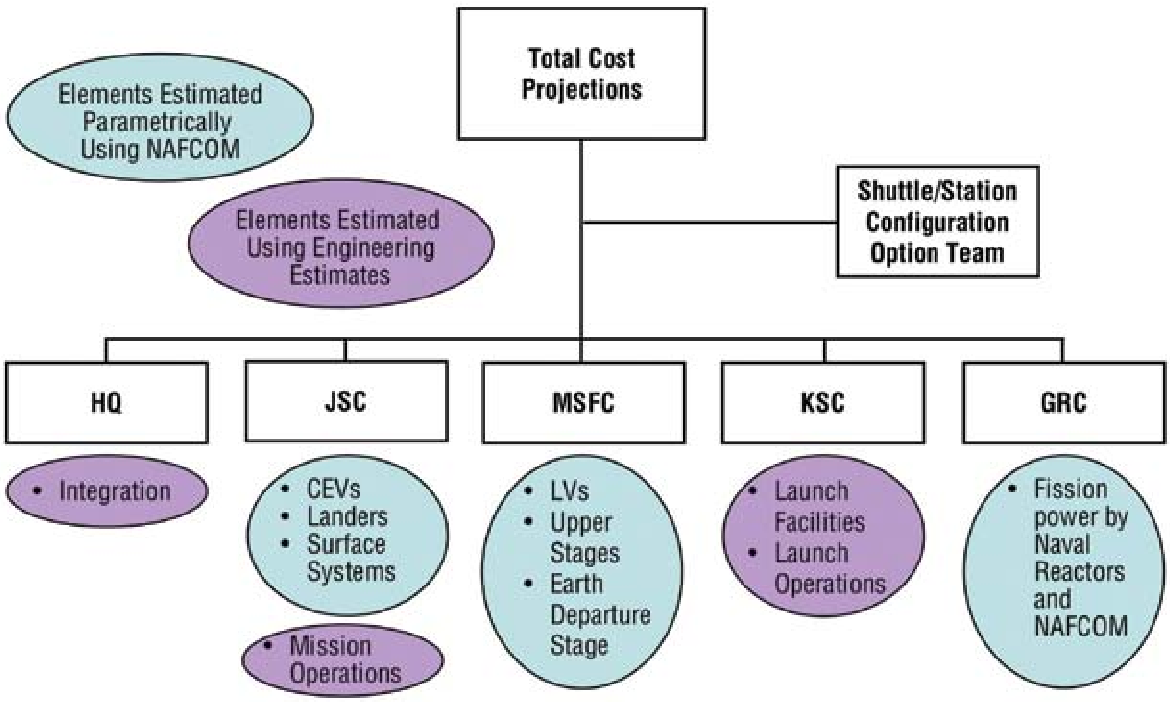


Figure 12-1. Cost-Estimating Participants and Approaches Used

12.5 Top-Level Summary of Methodologies Employed

12.5.1 DDT&E and Production Costs

The cost estimates were calculated using both parametric and engineering estimating approaches. Most parametric estimates were performed with NAFCOM, which is a basic parametric cost-estimating tool widely used in the aerospace sector. NAFCOM is a NASA-managed model currently being maintained by a Government contractor for NASA. As shown in **Figure 12-2**, the model is based on a relatively large database of approximately 122 historical projects including LVs and spacecraft. Recent model improvements include changes as a result of internal statistical assessments and benchmarking activities with aerospace industry contractors.

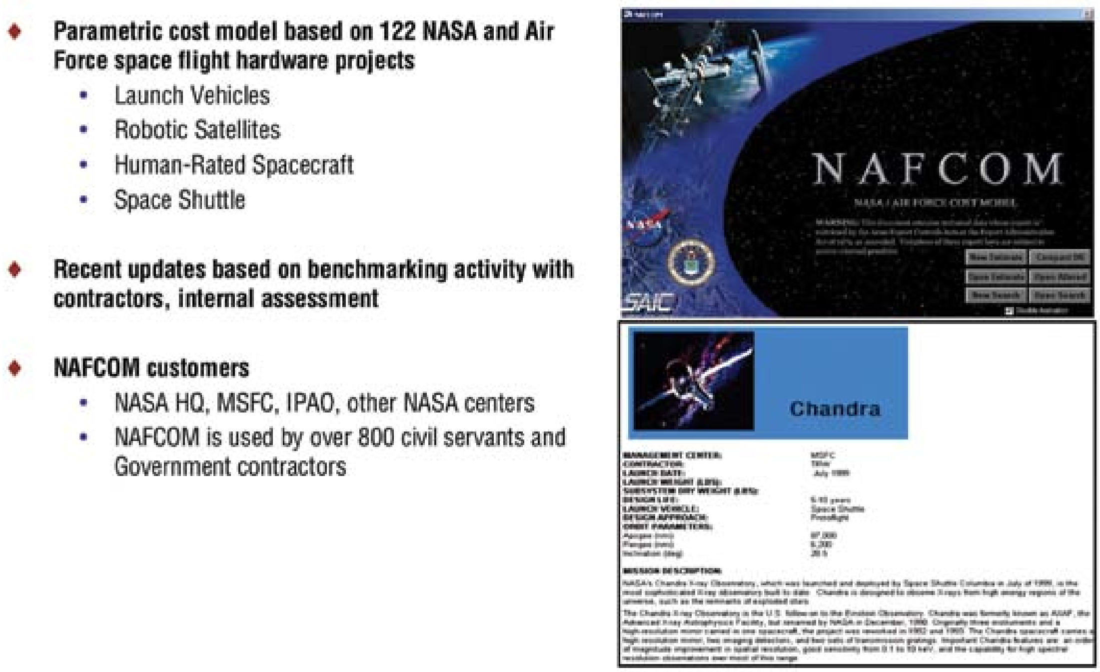


Figure 12-2. NAFCOM

The NAFCOM database is the basis for the multivariable Cost-Estimating Relationships (CERs) in NAFCOM. The model allows the user to use a complexity generator approach, which is essentially a multivariable regression CER approach or, alternately, a specific analogy approach in which the user selects the historical data points from the database for calibration purposes. Sample data from the NAFCOM database is shown in **Figure 12-3**.

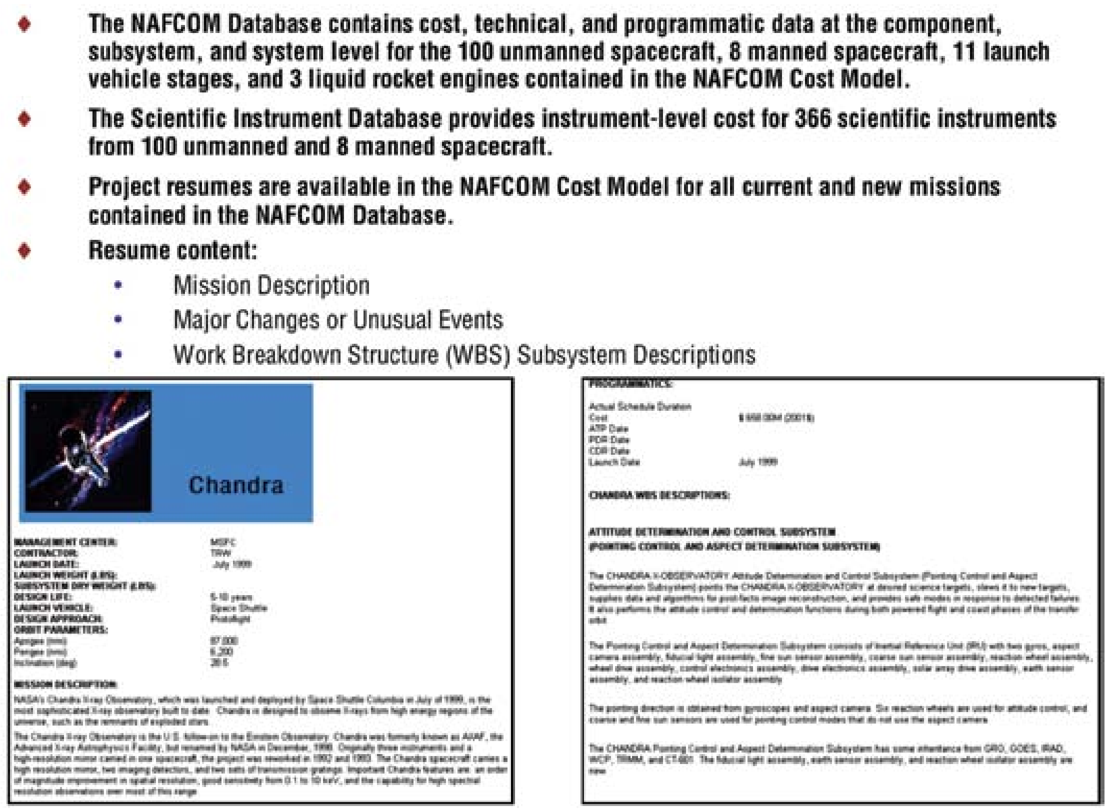


Figure 12-3. Sample from the NAFCOM Database

12.5.2 Operations Costs

ESAS operations analysis of affordability used a combination of cost-estimation methods including analogy, historical data, subject matter expertise, and previous studies with contracted engineering firms for construction cost estimates. The operations affordability analysis relied on cost-estimating approaches and was not budgetary in nature, as budgetary approaches generally have extensive processes associated with the generation of costs, and these budgetary processes cannot easily scale to either architecture-level study trades in a broad decision-making space or to trading large quantities of flight and ground systems design details in a short time frame. The operations cost-estimating methods used in the ESAS are attempts at fair and consistent comparisons of levels of effort for varying concepts based on their unique operations cost drivers.

12.6 Productivity Improvements Since Apollo

ESAS cost estimates account for productivity improvements since Apollo. On average, the American economy has shown an approximate 2 percent productivity gain for the period of 1970 to 1990. Subsequent to 1990, average productivity has been higher due to the continuing effects of the Information Technology (IT) revolution.

The Bureau of Labor Statistics (BLS) has a productivity data series for "Guided Missiles & Space Vehicles (SIC Code 3761)" for the period of 1988 to 2000. Over that period, this series has shown an average productivity gain of 2.47 percent. Independently, internal NASA estimates have shown that an approximate 2.1 percent productivity gain can be derived from the data behind NAFCOM. This data includes historical Apollo hardware costs. It is reasonable to question whether the BLS data is valid for earlier years because the effects of the IT revolution began impacting costs in aerospace design and manufacture around the mid-1980s.

Assuming that the internal NASA estimates are a better estimate as to the average annual rate of productivity gain in the aerospace industry over the period of 1970 to 2005, but that the BLS data reflects trends in the period 1985 to 2005, it can be estimated that, prior to about 1985, productivity gains averaged approximately 1.58 percent. This gives an overall average productivity gain of 2.1 percent for the whole period.

NAFCOM accounts for the productivity gains discussed above for most subsystems by embedding a time variable in the CERs, which is modeled as the development start date. Some NAFCOM CERs do not include the time variable due to statistical insignificance. For example, in the regressions for Main Propulsion System (MPS), engines, system integration, and management, the development start date was not statistically significant. With all else being equal, NAFCOM would predict that most Apollo flight hardware developed 50 years later would cost 33 percent less to develop and 22 percent less to produce than in 1967. Some subsystems, such as avionics, have an opposing trend of increasing cost over time due to increased functional requirements. In addition, NAFCOM allows the modeling of other specific engineering and manufacturing technology improvements that further reduce estimated cost.

12.7 Comparison to Apollo Costs

The cost estimates of the ESAS architecture accounted for the productivity gains previously discussed. In addition, because Shuttle-derived hardware is being used in the estimate, a cost savings is seen as compared to the Apollo Program's development of the Saturn V. Another factor to consider is that the proposed ESAS architecture is significantly more capable than the Apollo architecture. Apollo placed two crew members on the lunar surface for a maximum of 3 days, whereas the ESAS architecture places four crew members on the surface for a maximum of 7 days. This is factor of 4.6 times more working days on the lunar surface per sortie mission. The ESAS CEV also has three times the volume of the Apollo Command Module. This additional capability does not come at a great additional cost.

The historical cost in current 2005 dollars for the Apollo Program through the first lunar landing (FY61–FY69) was approximately $165B. The $165B figure includes all civil service salaries and overheads and all Government "service pool" costs. The ESAS architecture has an estimated total cost of $124B through the first lunar landing (FY06–FY18). As shown in **Figure 12-4**, costs were estimated conservatively with the inclusion of $20B for ISS servicing by CEV. Currently, NASA is planning to use commercial crew and cargo services to service the ISS which could further reduce cost.

The factor of 4.6 gain in capability and factor-of-three improvement in volume can be attained for less cost than the historical costs of Apollo. It should be noted that the ESAS architecture also allows access to the entire lunar surface, whereas Apollo was confined to the equatorial regions, and the ESAS architecture allows anytime return from any lunar location, thus providing still more capability over the Apollo capability.

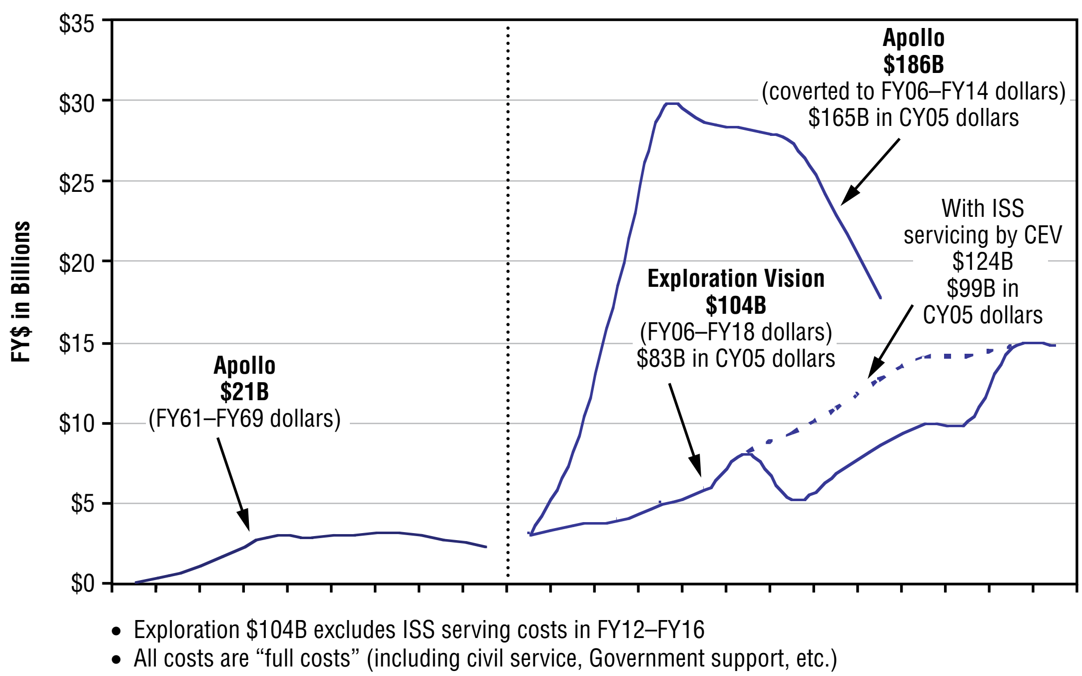


Figure 12-4. Comparison of Apollo Costs to Exploration Vision

12.8 Cost Integration

12.8.1 Full Cost "Wrap"

As shown in **Figure 12-5**, the cost of civil service and institutional costs has, over the long term of NASA's history, equated to an approximate 25 percent "wrap factor" on procurement costs. In the ESAS, this 25 percent "parametric" factor was used where a detailed engineering estimate of civil service and institutional costs was not available.

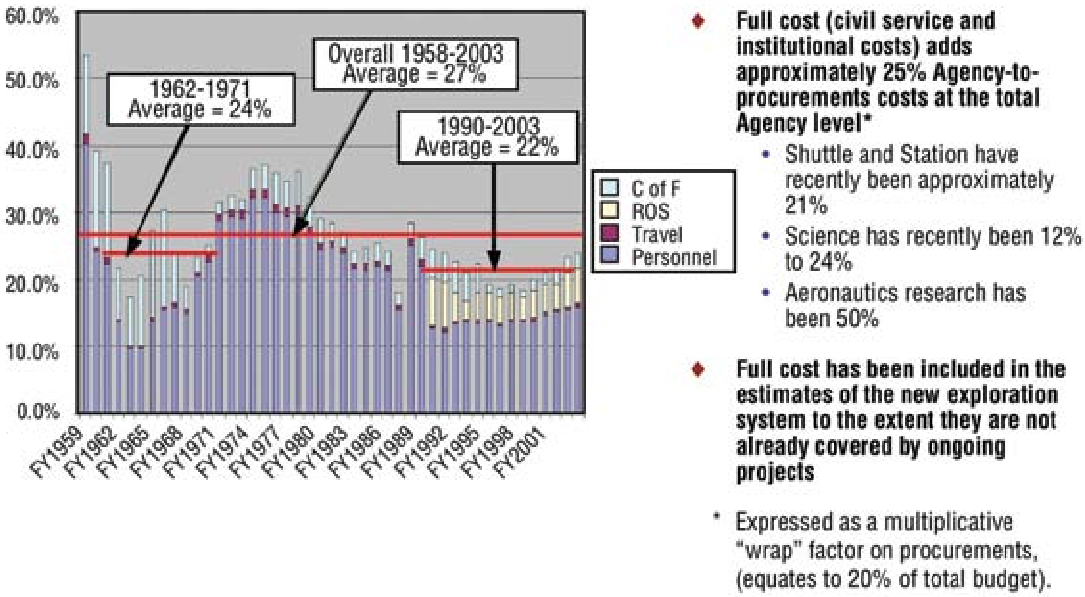


Figure 12-5. Full Cost "Wrap"

12.8.2 Research and Technology Cost Estimates

The Research and Technology (R&T) budget includes Exploration Systems Research and Technology (ESRT), Human Systems Research and Technology (HSRT), and the Prometheus Nuclear Systems Technology (PNST) programs. The Prometheus program has been entirely replaced with a nuclear surface power technology program. This program is focused on a low-cost, low-mass, high-power capability for use on the Moon or Mars. The ESRT and the HSRT programs will be focused on projects with direct application to the ESAS-recommended architecture. Near-term focus for HSRT will be on ISS applications that provide information on effects of long durations in zero- or low-gravitational environments, risks due to radiation exposure, radiation protection measures, and advanced space suit technologies. Near-term focus of ESRT is on the following: heat shield applications; propulsion technologies for CEV and lander, especially Liquid Oxygen (LOX)/methane engines; Low-Impact Docking System (LIDS); airbag and parachute landing systems; precision landing Guidance, Navigation, and Control (GN&C); In-Situ Resource Utilization (ISRU) technologies; lunar surface mobility systems; and Integrated System Health Management (ISHM).

12.8.3 Systems Engineering and Integration Cost Estimate

The Systems Engineering and Integration (SE&I) cost estimate was derived by examining the costs for several previous human spaceflight programs. As a percentage of all other costs, the SE&I cost averaged approximately 7 percent in those programs. The ESAS cost is estimated as 7 percent of total cost until it reaches a staffing cap at 2,000 people. In all prior human space flight programs, the peak workforce level for SE&I was between 1,000 and 1,500 people. Capping the Exploration estimate at 2,000 people is a conservative assumption that should be attainable. There were no additional reserves added to the cost of SE&I for that reason.

12.8.4 Robotic Lunar Exploration Program (RLEP) and Other Costs

NASA's RLEP program costs are bookkept on the future ESAS budget profiles because it is an integral part of NASA's human and robotic exploration program.

12.8.5 Other Costs

Several smaller cost elements and elements that provide common support across all elements have been included in the "other" category of the ESAS cost estimates. The elements included in this category are: Mission Control Center (MCC) common systems/software, Launch Control Center common systems/software, In-Space Support Systems, crew medical operations, Flight Crew Operations Directorate support, flight crew equipment, Space Vehicle Mockup Facility (SVMF), neutral buoyancy laboratory support, nontraditional approaches for providing space flight transportation and support, ISS crew and cargo services from international partners, and Michoud Assembly Facility (MAF) support during transition from Shuttle to Exploration LV production. These costs were predominantly estimated by direct input from the performing organizations. Most are the result of assessing current expenditures for the Shuttle and ISS programs and extrapolating those results to the Exploration program. The major element within the In-Space Support Systems is the lunar communication constellation. This estimate was taken from a communication system working group study from March 2005. In the case of the nontraditional approaches, there is a set-aside in these estimates of approximately $600M to cover entrepreneurial options between 2006 and 2010. The ISS crew and cargo services were estimated by pricing Soyuz flights at $65M per flight and Progress flights at $50M per flight. The Soyuz and Progress estimates were coordinated with the ISS Program Office.

12.8.6 Cost Risk Reserve Analysis

The ESAS team performed an integrated architecture cost-risk analysis based on individual architecture element risk assessments and cost-risk analyses. Cost-risk analyses were performed by the cost estimators for each major element of the Exploration Systems Architecture (ESA). Cost estimates for the CLV, Cargo Launch Vehicle (CaLV), CEV, landers, rovers, and other hardware elements were developed using NAFCOM. Risk analysis was provided by the cost estimators using the risk analysis module within NAFCOM. NAFCOM uses an analytic method that calculates top-level means and standard deviations and allows full access to the element correlation matrix for inter- and intra-subsystem correlation values. Facility modification and development, ground support, Michoud Assembly operations, R&T, and other costs were estimated using other models or engineering buildup. Risk for some of these elements was assessed by reestimating using optimistic and pessimistic assumptions and modeling the three estimates as a triangular risk distribution. The integrated cost from these estimates was transferred to an "environment" known as the Automated Cost-Estimating Integrated Tools (ACEIT). ACEIT is a spreadsheet-like environment customized for cost estimating that has the capability to perform comprehensive cost-risk analysis for a system-of-systems architecture.

The cost-risk analysis was performed subsequent to the presentation of the cost estimate to the NASA Administrator, Office of Management and Budget (OMB), and the White House. This was due to the limited time of the ESAS and the high number of alternative configurations costed. Hence, for these initial estimates, the ESAS team decided to include a 20 percent reserve on all development and 10 percent reserve on all production costs, approximately $1.5B total, to ensure an acceptable Confidence Level (CL) in the estimate to arrive at a total estimate of $31.2B through the first lunar flight.

The cost-risk analysis identified the 65 percent CL estimate ($31.3B) as equivalent to the ESAS team's point estimate with 20 percent/10 percent reserves. The cost-risk analysis confirmed that the cost estimate presented earlier was an acceptable confidence. Note that risk analysis was performed only through 2011; most cost risk is post-2011.

12.9 Overall Integrated Cost Estimate

In general, the cost estimates for all elements of the ESAS were provided by cost-estimating experts at the NASA field centers responsible for each element. These estimates were provided to the ESAS team in FY05 constant dollars including NASA full cost wraps. The team applied the appropriate reserve levels for each estimate to achieve a 65 percent CL and spread the costs over the appropriate time periods to complete the LLC estimates. Nonrecurring costs were provided as "Estimates at Completion" (EAC). Recurring costs were provided in fixed cost and variable cost components. In general, the nonrecurring costs were spread over time using a distribution curve based on historical cost distributions from many previous human space flight and large scientific NASA programs. The prior programs were converted to percent-of-time versus percent-of-total-cost curves. A beta distribution curve that closely traced the historical data was then used by the ESAS team for spreading the individual element nonrecurring costs.

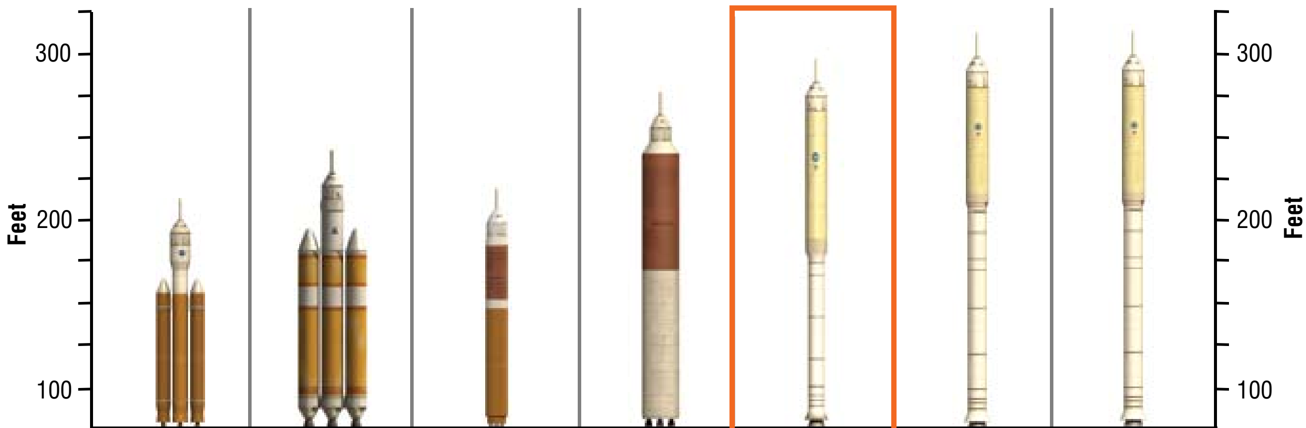
For all of the elements described below, the integrated cost estimate includes the DDT&E estimate for the hardware, production of flight hardware, provision of flight test articles, facility modifications at the NASA operational sites and engine test sites, operational costs for processing hardware at the launch site, mission operations to train crew and ground personnel, initial spares lay-in, logistics for processing the reusable components, and sustaining engineering for reusable components once production has ended.

12.9.1 ESAS Initial Reference Architecture

In order to compare various approaches and options for the study, a baseline departure point was selected by the ESAS team. This baseline was referred to as the ESAS Initial Reference Architecture (EIRA).

12.9.2 Launch Vehicle Excursions from EIRA

Excursions from EIRA were examined for both the CLV and the CaLV. **Figures 12-6** and **12-7** provide the top-level comparison of costs for the most promising LV options assessed during this study.



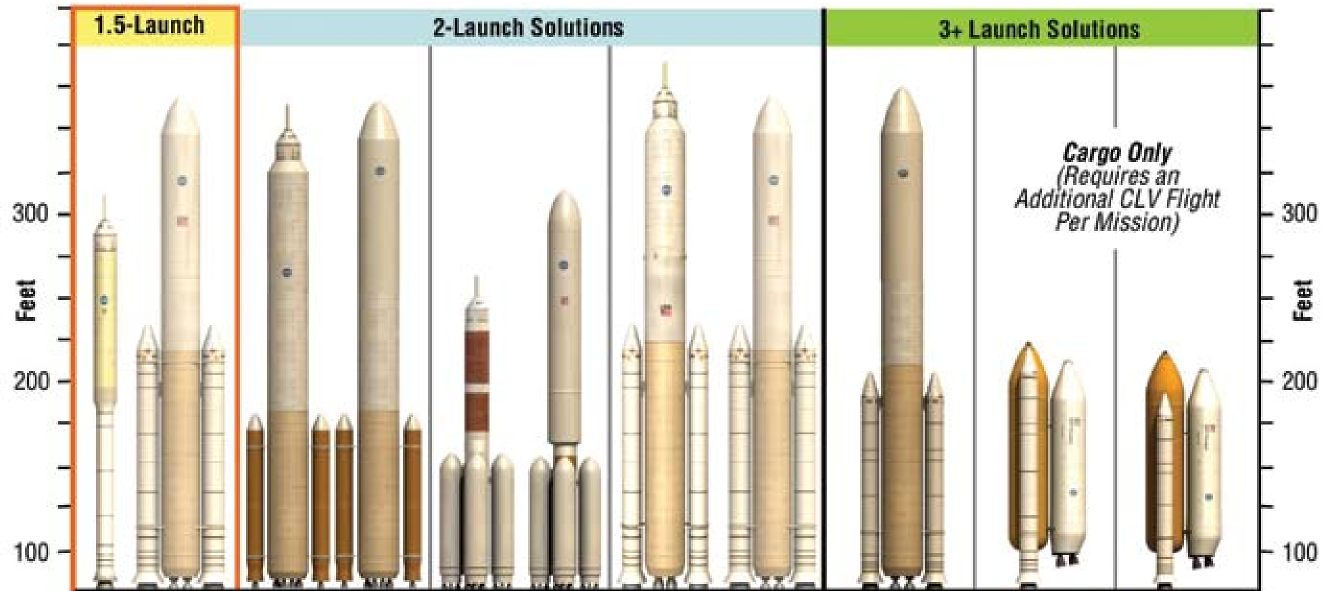
	Human-Rated Atlas V/New US	Human-Rated Delta IV/New US	Atlas Phase 2 (5.4-m Core)	Atlas Phase X (8-m Core)	4 Segment RSRB with 1 SSME	5 Segment RSRB with 1 J–2S	5 Segment RSRB with 4 LR–85
Payload (28.5°)	30 mT	28 mT	26 mT	70 mT	25 mT	26 mT	27 mT
Payload (51.6°)	27 mT	23 mT	25 mT	67 mT	23 mT	24 mT	25 mT
DDT&E*	1.18**	1.03	1.73**	2.36	1.00	1.3	1.39
Facilities Cost	.92	.92	.92	.92	1.00	1.00	1.00
Average Cost/Flight*	1.00	1.11	1.32	1.71	1.00	.96	.96
LOM (mean)	1 in 149	1 in 172	1 in 134	1 in 79	1 in 460	1 in 433	1 in 182
LOC (mean)	1 in 957	1 in 1,100	1 in 939	1 in 614	1 in 2,021	1 in 1,918	1 in 1,429

LOM: Loss of Mission LOC: Loss of Crew US: Upper Stage RSRB: Reusable Solid Rocket Booster

* *All cost estimates include reserves (20% for DDT&E, 10% for Operations), Government oversight/full cost; Average cost/flight based on 6 launches per year.*

** *Assumes NASA has to fund the Americanization of the RD–180. Lockheed Martin is currently required to provide a co-production capability by the USAF.*

Figure 12-6. Comparison of Crew LEO Launch Systems



	5 Segment RSRB In-Line with 5 SSME Core - Cargo	Atlas Phase X (8-m Core)	Atlas Phase 3A (5.4-m Core)	5 Segment RSRB In-Line with 4 SSME Core	4 Segment RSRB In-Line with 3 SSME Core	5 Segment RSRB Side-mount with 3 SSME	4 Segment RSRB Side-mount with 3 SSME
Payload to 28.5	106 mT (125 mT w/ upperstage)	95 mT	94 mT	97 mT	74 mT	80 mT	67 mT
Lunar LV DDT&E*	1.00	1.29	.59	.96	.73	.80	.75
CLV+Lunar Crew/Cargo DDT&E (family op)*	1.00 (4 Seg RSRB w/1 SSME)	1.26 (Atlas V)	1.02 (Phase 2)	.98 (4 Seg RSRB w/1 SSME)	.83 (4 Seg RSRB w/1 SSME)	1.03 (5 Seg RSRB w/J2S-LR85)	.85 (4 Seg RSRB w/1 SSME)
Lunar LV Facility DDTE	1.00	1.33	2.25	1.00			
CLV+Lunar Facilities Cost	1.00	1.12	1.56	1.00			
Lunar LV Average Cost/Flight*	1.00	1.08	1.19	.87	.78	1.13**	1.13**
LOM – Cargo (mean)	1 in 124	1 in 71	1 in 88	1 in 133	1 in 176	1 in 172	1 in 173
LOC (mean)	1 in 2,021	1 in 536	1 in 612	1 in 915	1 in 1,170	N/A	N/A

* All cost estimates include reserves (20% for DDT&E, 10% for Operations), Government oversight/full cost; Average cost/flight based on 6 launches per year.
**Production costs are higher than in-line due to production of separate side-mount cargo carrier

Figure 12-7. Lunar Cargo Launch Systems Comparison

One of the first options examined was the use of human-rated EELVs. From a cost perspective, the EELV-derived CLVs were approximately equivalent to the Shuttle-Derived Vehicles (SDVs). They were eliminated from further consideration primarily as a result of the reliability and safety analysis and because they did not offer a significant cost advantage over the more highly reliable SDVs. EELV derivatives for the CaLV were more costly than the SDVs and were eliminated on that basis.

In the case of the EIRA Shuttle-derived CLV, some of the highest cost items for development were the changes to a five-segment Reusable Solid Rocket Booster (RSRB) and the new LR–85 upper stage engine. Several options were examined to change the upper stage engine to a J–2S or an SSME. Using the SSME as an upper stage engine allows the use of the four-segment RSRB on the CLV instead of the five-segment version. This defers the cost of the five-segment development until needed for the CaLV. The lowest-cost and shortest-schedule CLV option is the current four-segment RSRB combined with a minimally changed SSME used for the upper stage. This configuration provides a highly reliable and safe vehicle. Since it is almost entirely derived from existing Shuttle components, it has the highest likelihood of meeting the desired launch date in 2011. It is the ESAS-recommended option for both ISS support and lunar missions.

For the CaLV, several excursions were examined to try to minimize the number of launches needed to complete a lunar mission and also to try to get a CaLV to pair with the highly reliable RSRB-derived CLV. In order to use the RSRB-derived CLV, the CaLV needs to provide more lift capability than the EIRA configuration. Options to add an upper stage to the vehicle were examined as well as options that use the Earth Departure Stage (EDS) during the ascent stage. The lowest cost option that allowed continued use of the RSRB CLV was the five-SSME core stage with five-segment strap-on RSRBs, while using the EDS for a burn during the ascent stage. This is the ESAS-recommended option for the CaLV. When coupled with the RSRB-derived CLV, this is called the “1.5-launch solution.”

One additional option was examined to try to reduce the total LLC of LVs. In this option, the CLV was eliminated and the HLLV was designed from the beginning for use as both a CLV and a CaLV. While this option has the best total LLC, it is very expensive in the near-term. Secondly, this option would represent excessive risk to meeting the desired 2011 launch date with many significant development activities needed. It also scores worse than the RSRB-derived CLV for reliability and safety. These results are presented in more detail in **Section 6.11**, **Conclusions**.

12.9.3 Cost Excursions from EIRA

12.9.3.1 "One Content Change at a Time" Excursions from EIRA

In addition to looking at LV options, the ESAS team looked at several other candidates to either reduce the costs in the critical periods or to improve the technical aspects of the EIRA baseline. **Table 12-1** provides a summary for several options that were examined.

Table 12-1. "One Content Change at a Time" Cases

Cost Case	Crew to ISS LV	Lunar Cargo and Crew LV	EDS	Rendezvous Option and CEV Diameter	Lunar Launches Per Mission	Lunar Program	Schedule	Technology Program
1A (EIRA) Low Tech	5-segment SRB LR-85 US	SDV 8-m Core 5-segment SRBs (with SSMEs)	Heritage Modified from US (LR-85s)	0 (LOR Split Mission) 5M CEV	2	Sorties + Base	2011 2018	Full
1E Low Tech	5-segment SRB LR-85 US	SDV 8-m Core 5-segment SRBs (with SSMEs)	Heritage Modified from US (LR-85s)	0 (LOR Split Mission) 5M CEV	2	**Sorties Only Deferred Base**	2011 2018	Full
1F Low Tech	5-segment SRB LR-85 US	SDV 8-m Core 5-segment SRBs (with SSMEs)	Heritage Modified from US (LR-85s)	0 (LOR Split Mission) 5M CEV	2	Sorties + Base	**2012** 2018	Full
1J Low Tech	5-segment SRB LR-85 US	SDV 8-m Core 5-segment SRBs (with SSMEs)	Heritage Modified from US (LR-85s)	0 (LOR Split Mission) 5M CEV	2	Sorties + Base	2011 2018	**Reduced**
1M Low Tech	5-segment SRB LR-85 US	SDV 8-m Core 5-segment SRBs (with SSMEs)	Heritage Modified from US (LR-85s)	0 (LOR Split Mission) 5M CEV	2 **Modified Test Plan**	Sorties + Base	2011 2018	Full

Changes from EIRA indicated by bold text.

Cost Case 1E was introduced to reduce the cost problem in the out-years. This option eliminated the long-stay base requirements and limited missions to short-duration sorties only. Cost Case 1F slipped the first CEV/CLV flight from 2011 to 2012. While it helps the near-term significantly, it is considered undesirable except as a last resort by the NASA Administrator because of the gap it introduces in the U.S. human space flight capabilities between Shuttle retirement and first CEV capability. Cost Case 1J significantly reduces the R&T budgets by focusing the activity on the needs of the ESAS-recommended program content. This option provides significant benefit in both the near-years and the out-years. Cost Case 1M implements a change to the flight test plan that was decided for technical reasons as a more reasonable test approach. It eliminates one of two previously planned Low Earth Orbit (LEO) tests of the CEV, lander, and the EDS. It retains both an LEO flight test mission and a lunar flight test mission, in which an unmanned lander goes to the lunar surface and returns for a rendezvous with the CEV. The deleted flight test mission saves the production of the CEV, lander hardware, and the LVs.

12.9.3.2 Rendezvous and Propulsion Technology Options

In support of the lunar architecture mission mode trade studies, several options were identified to vary the rendezvous locations for the CEV and lander. The initial rendezvous could either occur in Low Lunar Orbit (LLO) per the EIRA assumptions or they could initially rendezvous in LEO. The LEO rendezvous was preferable from an operational, safety, and reliability perspective because any problems with the rendezvous would occur in close proximity to the Earth and would allow better contingency options. The second major rendezvous occurs when the lander returns from the surface of the Moon. In the EIRA, the lander returns from the lunar surface and rendezvous with the CEV in LLO. Another option is to take the CEV to the lunar surface; then the return to Earth does not require a rendezvous at all. The CEV may go directly from the lunar surface to an Earth return trajectory. This option was referred to as a "direct return." In the course of examining these options, additional options were introduced to change the technology level of the engines used for the CEV and lander. The EIRA assumes pressure-fed LOX/methane engines for the CEV Service Module (SM), lander descent stage, and lander ascent stage. The first set of options changed the lander descent engines to pump-fed LOX/hydrogen. The second set of options changed the CEV and lander ascent engines to pump-fed LOX/methane, in addition to using LOX/hydrogen engines for the lander descent stage. The lowest cost options were the lunar direct-return missions that required the pump-fed engines in all applications. The next lowest cost and the ESAS recommendation was the EOR–LOR case with pump-fed LOX/hydrogen engines on the lander descent stage and retaining pressure-fed LOX/methane engines for the CEV and lander ascent. The lunar direct-return cost was much lower due to the elimination of the habitable volume and crew systems on the lander ascent stage. These were replaced by the CEV going all the way to the lunar surface. The ascent stage of the lander was also eliminated by using the SM capabilities for ascent propulsion from the lunar surface. These cost advantages were offset by reduced safety and reliability due to the loss of the redundant habitable volume provided by the lander. Having both the CEV and the lander as separable crew habitation space was desirable from a crew survival perspective and for operational flexibility. The results of these rendezvous and engine technology options are shown in **Figure 12-8**. These results are presented in more detail in **Section 4.2.5**, **Analysis Cycle 3 Mission Mode Analysis**.

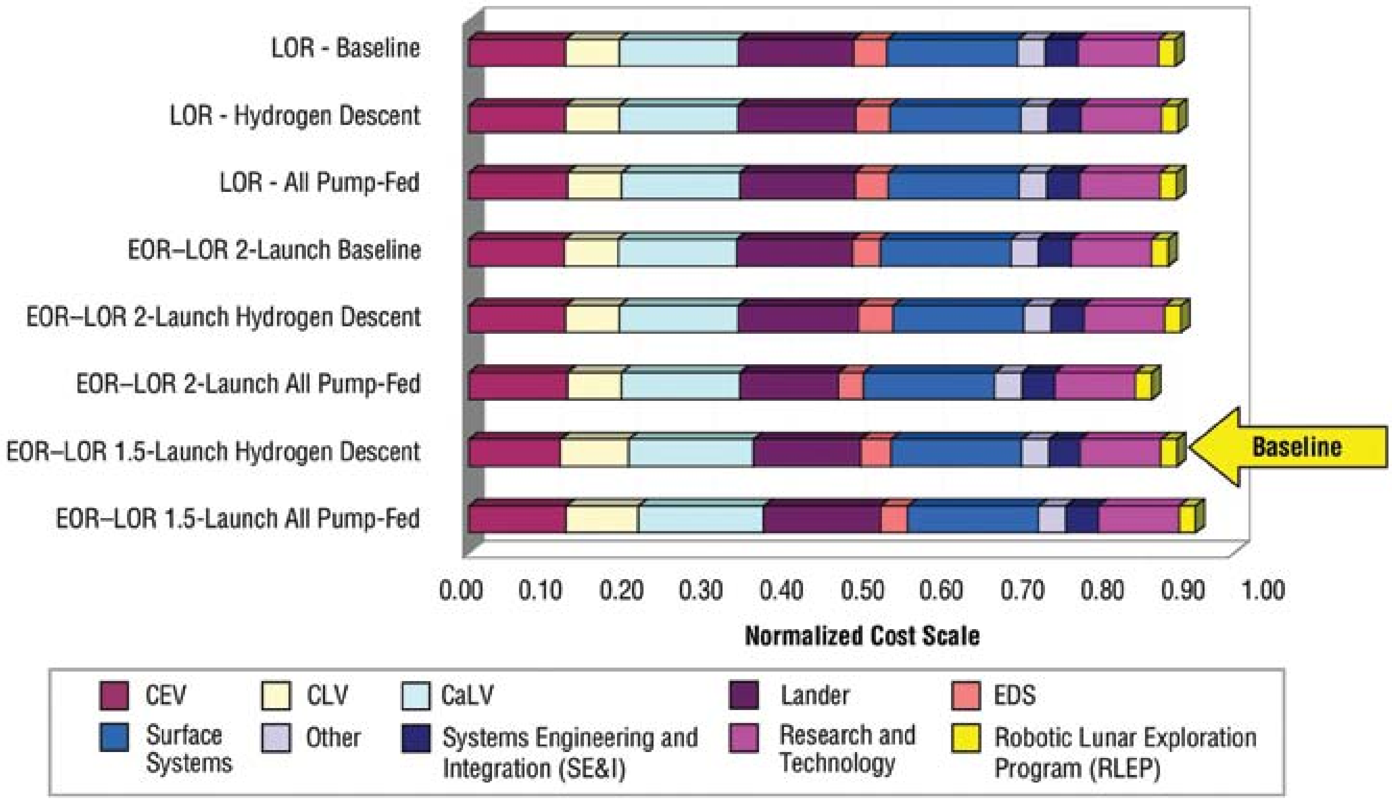


Figure 12-8. Rendezvous and Propulsion Technology Options

12.9.4 Final Cost Projections

As part of the President's Vision for Space Exploration, NASA was provided with a budget profile through FY11. The ESAS final cost recommendation assumed a funding requirement that exceeded this guideline in the years FY08–10, assuming the CEV first flight were to occur in 2011.

The ESAS recommendation retains the schedule objectives of the EIRA (i.e., 2011 first ISS mission and 2018 first human mission to the lunar surface). The recommendation includes the use of the 5.5-m CEV, the reduced R&T budget, and the modified test plan. It also includes reductions to the lunar outpost but not complete elimination of the outpost. The recommendation includes a minimal outpost consisting of both unpressurized and pressurized rovers, solar electrical power instead of nuclear power, lunar surface Extra-Vehicular Activity (EVA) suits, and small ISRU technology demonstrators. It eliminates the dedicated habitation module and large-scale ISRU production capability demonstrators. The recommended architecture is somewhat above the currently available budget in many years, but is considered a prudent achievable plan to replace human space flight capability and to begin to develop the necessary transportation infrastructure to provide a meaningful exploration capability. Subsequent to the ESAS effort, NASA baselined a 2012 first flight for the CEV and CLV, which allows the program to be accomplished within available budget.

12.10 CEV, Lunar Surface Access Module (LSAM), Lunar Surface Systems, EVA Systems, and Mission Operations Cost Estimates

12.10.1 Crew Exploration Vehicle (CEV) Acquisition Costs

12.10.1.1 Scope

The CEV cost estimates include the costs for the crewed CEV (Crew Module (CM), SM, and Launch Abort System (LAS)), the uncrewed CEV, and the unpressurized Cargo Delivery Vehicle (CDV). The CEV CM and SM have ISS and lunar variants. Because the ISS variant is the first to be developed, it carries the majority of the development cost. Lunar variants assume a significant degree of commonality with the ISS variant.

12.10.1.2 Methodology

All CEV estimates were prepared using NAFCOM. Hardware estimates were generally completed at the subsystem level or component level if detailed information was available. Cost estimates include the DDT&E cost and the production cost of the first Flight Unit (FU).

12.10.1.3 Ground Rules and Assumptions (GR&As)

The cost GR&As are included below.

- Cost estimates are in millions of FY05 dollars and include 10 percent fee and 4 percent vehicle integration.
- Hardware estimates were based on weight, selected analogies, design and development complexity, design heritage, flight unit complexity, system test hardware quantity, quantity next higher assembly, production quantity, and learning curve.
- System Test Hardware (STH) quantities were based on inputs from the Test and Verification Plan. All STH was allocated to the ISS versions of the CEV, since the lunar version was assumed to be identical. Production costs were based on a 90 percent Crawford learning curve. Production quantities were based on the projected mission manifest through 2030. For the uncrewed CEV capsule, and lunar variants of the CEV capsule and SM, the learning curve was assumed to start where the previous production run ended.
- Software estimates were based on previous estimates for the Orbital Space Plane (OSP). Software estimates for the CEV were output as a subsystem in NAFCOM to capture the additional system integration costs.
- The system integration costs were based on the average of three analogies: Gemini, Apollo Command and Service Module (CSM), and X–34. The Gemini and Apollo CSM analogies represent crewed capsule developments, while the X–34 was used to represent modern systems engineering and program management methods.

12.10.1.4 Alternative CEV Architectures

Several alternative CEV architectures were studied. These alternatives were estimated by variations from the basic EIRA cost estimate. Variations were created by changing the rendezvous mode and propulsion options. In addition, some alternatives have new technology options, such as fuel cells instead of solar arrays. The alternative architectures were assumed to perform the same number of missions.

12.10.1.5 Reusable Versus Expendable Capsule Trade Study

A trade study was performed to compare the LLC of a reusable CEV capsule to an expendable CEV capsule. The study looked at numerous variables that would influence the cost difference between the two options, including: flight unit cost, missions per vehicle, percentage of hardware replaced per mission, missions per year, and learning curve rate.

The trade study assessed the range of values for each input variable and identified the most likely values for each input. The study concluded that a reusable CEV capsule could save approximately $2.8B compared to an expendable capsule over the life of the program.

12.10.2 Lunar Surface Access Module (LSAM)

12.10.2.1 Scope

Estimates for the LSAM include the crew ascent vehicle, crew descent stage, and cargo descent stage.

12.10.2.2 Methodology

The cost-estimating methodology used for the LSAM was similar to the method used for the CEV.

12.10.2.3 Ground Rules and Assumptions

The GR&As used for the LSAM were similar to those used for the CEV.

12.10.2.4 Alternative LSAM Architectures

Several alternative LSAM architectures were studied. These alternatives were estimated by variations from the basic EIRA cost estimate. Variations were created by changing the rendezvous mode and propulsion options. In addition, some alternatives have new technology options, such as pump-fed versus pressure-fed engines. The alternative architectures were assumed to perform the same number of missions.

12.10.3 Lunar Surface Systems

12.10.3.1 Scope

The scope of the lunar surface systems includes hardware that is intended to operate primarily on the lunar surface, except for power systems, EVA systems, and crew equipment, which are addressed in later sections. The lunar surface systems include surface vehicles, surface modules, construction and mining systems and vehicles, and manufacturing and processing facilities.

12.10.3.2 Methodology

The cost-estimating methodology used to estimate the lunar surface systems was similar to the method used to estimate the CEV and LSAM. The data provided by the technical team was top-level notional design information that included system mass with functional descriptions. A Work Breakdown Structure (WBS) was developed for each surface system element with the aid of the ESAS design engineers and past studies. The estimates are preliminary and require updates as the designs mature.

12.10.3.3 Ground Rules and Assumptions

The GR&As used to estimate the lunar surface systems were similar to those used to estimate the CEV and LSAM.

12.10.4 Fission Surface Power System (FSPS)

12.10.4.1 Scope

The operational scenario calls for a stationary Fission Surface Power System (FSPS) power plant landed with a mobile Station Control Electronics (SCE) cart. The FSPS remains on the lander. The mobile SCE is off-loaded from the fixed power plant and drives to the designated site while deploying the power transmission cable. Startup and verification of the power system is performed prior to landing the habitat near the mobile SCE. The habitat provides the power interface to the mobile SCE.

12.10.4.2 Methodology

The design of the FSPS was based on the Prometheus FSPS-Lunar "Task 3 Report, Revision 8," dated June 10, 2005. Several power levels were studied in addition to the baseline power level of 50 kWe.

12.10.4.2.1 Reactor/Power Conversion Subsystems

Reactor and power conversion subsystem costs were derived from the Project Prometheus Naval Reactors prime contractor cost input. The estimate included costs for development, two ground test reactors, and two flight units. Costs for the second ground test reactor and flight unit have been removed and costs were converted from FY06 dollars to FY05 dollars. These costs had not been approved by Department of Energy (DoE) Naval Reactors as of the time of this estimate and are considered conservative. The estimate scope includes development and flight hardware for all reactor and power conversion subsystems/components, materials test and evaluation, and ground test reactor/facilities development and operations.

12.10.4.2.2 Balance of Power System/Mobility System

NAFCOM and GRC Boeing Task 26, CERs for Advanced Space Power and Electric Propulsion Systems, dated June 2005, were used to develop the Phase C/D cost estimates for the balance of the power system and the mobility system. NAFCOM was used to estimate the mechanical subsystem for the power plant (using manned-mission-type analogies) and the mobility system subsystems (using unmanned-planetary-type analogies and Earth-orbiting-mission-type analogies). GRC Boeing Task 26 was used to estimate the heat rejection subsystem for the power plant, transmission cable, and the station control electronics.

12.10.4.3 Ground Rules and Assumptions

The key GR&As associated with the FSPS estimate were as follows:

- The estimate was for surface power only (e.g., no nuclear propulsion related work). The costs do not include NASA HQ program management, risk communications, NASA National Environmental Policy Act (NEPA) support, KSC FSPS facility requirements, and corporate G&A.
- The reactor and power conversion subsystems were sized for 100 kWe for all power options considered.
- The mobile SCE concept mobility system was sized to transport only the power control electronics and cable from the lander to a remote location approximately 2 km away.
- Non-nuclear technology development costs were based on Prometheus requirements and do not change with a change in power level.
- Nuclear technology and facilities development costs include the ground test reactor and facilities and reactor module materials test and evaluation.
- Phase A/B DoE Naval Reactors costs were estimated at 7 percent of the Naval Reactors Phase C/D costs. Phase A/B prime contractor costs were estimated at 14 percent of prime contractor Phase C/D costs due to increased complexity associated with human-rating and integration complexity associated with multiple Government/contractor entities.
- The ground test reactor will continue to operate throughout the mission life.
- Government insight/oversight full costs were estimated at 10 percent of non-nuclear costs and 5 percent of DoE Naval Reactors costs for technology, Phase C/D, and Phase E. Government insight/oversight full costs were estimated at 30 percent of total Phase A/B costs.
- Phase C/D power system and mobility system risk ranges were based on results of a NAFCOM subsystem risk analysis. The cost estimate was the 50 percent confidence value from NAFCOM with risk turned on. Narrow risk ranges for the Phase A/B and Phase C/D are the result of using conservative estimating information and techniques.
- Cost phasing was based on assumed activities to be performed in each phase of the program. Minimum cost-level requirements that may be necessary for DoE Naval Reactors work have not been assumed. These requirements may change the phasing but not the total FY05 dollars required.
- Three test hardware items were included for all SCE subsystems/components and the heat rejection subsystem has two test hardware items.
- Mobility system power was assumed to be provided by a 3-kWe station control electronics solar array for transit and reactor startup.
- Ground test reactor facilities were assumed to be operational by 2014. Operations during the Phase C/D period were included in the Phase C/D costs.

12.10.5 Extra-Vehicular Activity (EVA) Systems

12.10.5.1 Scope

The EVA system consists of three major elements: (1) EVA suits, (2) EVA tools and mobility aids, and (3) vehicle support systems. The EVA suit system consists of the life support systems and pressure garments required to protect crew members from ascent/entry, in-space, and planetary environmental and abort conditions.

The EVA tools and mobility aids consist of the equipment necessary to perform in-space and planetary EVA tasks, and include items such as drills, hammers, ratchets, walking sticks, vehicle handrails, and foot restraints. The vehicle support system consists of the equipment necessary to interface the EVA system with the vehicles. It includes items such as mounting equipment, recharge hardware, and airlock systems.

Associated with each of the three major EVA elements are ground support systems. The ground support systems include the equipment and facilities required to test and verify the EVA development and flight systems.

For the purpose of this cost estimate, it was assumed that each phase of the exploration mission architecture will require a unique EVA system. Though these systems may be based on a common architecture, for cost purposes, they were considered separately.

12.10.5.1.1 EVA System I

The EVA System I will include the delivery, by 2011, of an in-space suit and the associated equipment necessary to support launch, entry, and abort scenarios and contingency EVA from the CEV and other Constellation vehicles. The EVA System I is required for all crewed missions, regardless of destination.

12.10.5.1.2 EVA System II

The EVA System II will include the delivery, by 2017, of a surface suit and the associated equipment necessary to support surface exploration during the lunar sortie phase. The EVA System II is required for short-term lunar missions, starting with the CEV/LSAM LEO integrated test flight.

12.10.5.1.3 EVA System III

The EVA System III will include the delivery, by 2022, of an enhanced surface suit and the associated equipment necessary to support surface exploration during steady-state lunar outpost operations. The EVA System III will be based on System II and will include those upgrades and modifications necessary for longer planetary missions.

12.10.5.2 Methodology

For each of the areas listed above, procurement costs were based on historical data from previous EVA efforts or derived from bottom-up estimates. Institutional costs were derived as a percentage of the procurement cost. The percentage chosen was dependant on the specific activity. For instance, the civil service involvement during the complicated DDT&E phase of the EVA suit was assumed to be higher (30 percent), while civil service involvement during the more straightforward ground processing phase was assumed to be lower (15 percent).

The following cost breakdowns were provided for each of the EVA systems.

- A total nonrecurring cost was provided for the DDT&E for each of the major elements (suit system, tools and mobility aids, vehicle support system, and ground support system). An estimated time required for completion was provided along with the total cost.
- A per-unit recurring cost was provided for production of each of the major elements (suit system, tools and mobility aids, vehicle support system, and ground support system).
- A yearly cost was provided for the sustaining engineering associated with the overall EVA system. This effort includes activities such as failure analysis and correction and discrepancy tracking.
- A yearly cost was provided for the ground processing associated with the overall EVA system. This effort includes ground processing for both flight and training activities.

12.10.5.3 Ground Rules and Assumptions

The key GR&As associated with the EVA systems estimate are as follows:

- All costs were estimated in 2005 dollars.
- Estimates for operations activities were not provided as part of the EVA estimate. Instead, they were provided as part of the mission operations cost estimate.

12.10.6 Mission Operations

12.10.6.1 Scope

Mission operations includes the control centers, training simulators and mockups, processes, tools, and personnel necessary to plan the missions, train flight crews and flight controllers, and support the flights and missions from the ground. Mission operations are complementary to ground operations at the launch site, which was estimated separately. Mission operations support the flight crew, the costs of which were estimated separately.

12.10.6.2 Methodology

In August 2004, budget analysts from six flight operations areas convened to develop mission operations cost estimates to support the Constellation Development Program. The product was a run-out of costs for each launch/mission and an estimated duration and spread among the following four cost categories: (1) operations support to vehicle development, (2) new or modified facility capability, (3) mission preparation, and (4) mission execution. Assumptions and results were documented and reviewed with NASA HQ management and independent groups.

The estimates presented in this report were a product of applying the mission-specific cost templates to a manifest of ESAS launches to produce an integrated cost.

12.10.6.3 Ground Rules and Assumptions

The key GR&As associated with the mission operations estimate were as follows:

- Unique operations preparation and operations support to development are required for each new space vehicle or major upgrade of a space vehicle.
- LV performance margins will be maintained to avoid significant recurring planning optimization of operational missions.
- Recurring ISS missions will use a single stable CEV and LV configuration and mission design.
- The existing NASA JSC MCC will be used with limited modification for all human missions and test flights of human-rated vehicles. Telemetry and command formats were assumed to be compatible with existing MCC capabilities.
- Recurring fixed cost for MCC use was shared by Exploration after Shuttle Orbiter stops flying, in proportion to facility utilization.
- New development was required for training simulators because the potential for reuse of existing simulators is very limited.
- Simple mission planning and operations were assumed for test flights and CEV to ISS.
- Complex, highly integrated mission plans are required for initial lunar sorties.
- Simple, quiescent surface operations are assumed for extended lunar stays.
- Crew and ground tasks are considerably simpler than for Shuttle during critical mission phases.
- Mission operations cost is largely dependent on the number of unique space vehicles and annual crewed flight rate.
- Mission operations cost is generally independent of the number of launches involved in a single crewed mission.
- Mission operations cost is generally independent of the launch architecture.

12.10.7 Neutral Buoyancy Laboratory (NBL) and SVMF Operations

12.10.7.1 Scope

For the SVMF, the estimate includes development of CEV mockups representing each of the configurations: crewed to ISS, uncrewed to ISS, and crewed to the Moon. It also includes LSAM descent stage, both crewed and cargo, and an ascent stage. A lunar rover was included in the cost estimate as well as upgrades to the partial gravity simulator for surface EVA training.

For the Neutral Buoyancy Laboratory (NBL), the mockup development includes a CEV mockup for contingency EVA training and a mockup for water egress and survival training. The NBL will need facility modifications to support the new EVA suits. Two suits were assumed: an ascent/entry/abort/contingency EVA suit and a surface EVA suit.

Training EVA and launch/entry suits for either facility were not included in this assessment. Also, science package training hardware was not included in this assessment.

12.10.7.2 Methodology

The cost estimates for the SVMF and NBL were based on experience supporting the ISS and Shuttle programs.

12.10.7.3 Ground Rules and Assumptions

The key GR&As associated with the NBL/SVMF operations estimate were as follows:

- The development and delivery schedule estimated a mockup delivery date of 2 years before the first flight and a 2-year development process.
- The fidelity for the mockups was assumed to be similar to the existing high-fidelity Shuttle and ISS mockups. This was reflected in the "most-likely" cost. The high cost estimate would be valid if more fidelity is required or if the vehicles are more complex than reflected in the initial reference architecture. The low cost estimate can be realized if engineering or qualification hardware is available to augment the training mockups.
- The cost assessment included a sustaining cost for each mockup after it was delivered that is consistent with the sustaining cost of current Shuttle and ISS mockups.
- Manpower estimates included instructor and flight control personnel for crew systems and EVA. A mix of civil servant and contractor personnel were assumed for these jobs. It was assumed that, early in the program, there would be a higher ratio of civil servants to contractors than there would be later in the program.

12.10.8 Flight Crew Operations

12.10.8.1 Scope

The cost estimate for flight crew operations includes estimates for the Astronaut Office, Vehicle Integration and Test Office (VITO), and Aircraft Operations Division (AOD). Astronaut Office personnel include astronauts, technical support engineers, astronaut appearance support, IT support, schedulers, and administrative and secretarial support. The VITO provides critical support at KSC during test and integration of flight hardware and during launch flows as representatives of the crew.

Aircraft operations include maintaining and flying the T–38 aircraft used by all astronauts to develop the mental and manual skills required to fly safely and successfully in a spacecraft. It also includes all the personnel to serve as flight instructors and as engineering support for the aircraft, as well as an Aviation Safety Office. The portion to be retained for Exploration training will most closely resemble the T–38 aircraft program of today.

12.10.8.2 Methodology

12.10.8.2.1 Astronaut Office

The number of astronauts is driven by the need to support crew mission assignments, provide flight crew support for operations development and technical issues, provide (non-crew) mission support, and support educational outreach to the public pertaining to the NASA mission and goals.

There is a minimum office size required to maintain the appropriate skill sets and experience within the Astronaut Corps. It is critical to maintain crew members with spaceflight experience, including those with experience in developing operational concepts for EVA, robotics, rendezvous, docking, controlling and maintaining a spacecraft and its systems, and other crew activities.

For this estimate, an Excel spreadsheet was used to estimate attrition, astronaut candidate selections, and military-to-civil-servant conversion. The current size of the Astronaut Office was used as the initial value. Estimates of attrition were based on historical values, and the variable of astronaut selection was used to stabilize the number of astronauts at approximately 60 by 2016. There is variation from year to year with the addition of each new astronaut class and with attrition. After 2016, classes of approximately 12 were required every 3 years to maintain the number of astronauts between 56 and 64. Until 2010, astronaut support was divided between Shuttle, ISS and Exploration programs. Following this period, the support was divided between ISS and Exploration. Beginning in 2016, only the Exploration program was supported.

The Shuttle Retirement Change Request (CR) was used to estimate the number of support engineers for the Astronaut Office. This was a convenient source of reference for procurement contractor support and civil service support for the Shuttle program. Due to the complex and multiple elements required for Exploration (i.e., support of multiple elements for lunar missions and eventual support of Mars strategy), estimates were made that support needs to begin in FY06 and ramp up to values greater than the current Shuttle support numbers by 2016. Contractor support for astronaut exploration activities will also begin in FY06. Total office support in these areas was shared with the other programs prior to FY16.

Expedition Readiness Training (ERT) for exploration astronauts is currently in the planning stages. It is envisioned that there will be challenging training situations to hone leadership and survival skills and to provide a basis for serious evaluation of the astronaut candidates and assigned crew members. This will include travel to outdoor leadership field exercises to assess leadership skills, travel to Mars analog sites to assess operational concepts, hardware concept development, and suitability for training and further expedition leadership training. There are numerous site possibilities for these activities.

12.10.8.2.2 Vehicle Integration and Test Office (VITO)

The Vehicle Integrated Test Team (VITT) has a long history of providing critical support at KSC during test and integration of flight hardware and during launch flows as representatives of the crew. This support will begin to ramp up to support CEV in FY07. VITT members also travel to contractor sites to inspect hardware under development to provide critical input regarding hardware crew interface standards while changes can be made more easily.

12.10.8.2.3 Aircraft Operations Division (AOD)

A percentage of aircraft operations support that includes maintaining and flying the T–38 aircraft should be shared by the Exploration Program. This cost estimate shows it beginning at a low level in FY06 and ramping up in proportion to the percentage of astronauts dedicated to Exploration support.

Discussions are underway concerning aviation analog training, with the objective of providing situations requiring time-critical and, perhaps, life-critical decisions in a real-life environment. This may include a variety of aircraft; however, details were not yet available as the planning is in the very preliminary stages. The goal is to stay within the T–38 portion of the AOD operations budget as it exists today. Based on the Shuttle Retirement CR numbers, the current T–38 program costs were prorated based on the percentages of astronauts dedicated to exploration activities from FY06 through FY25.

Support includes civil servants who serve as instructors and research pilots and who provide other engineering support. Contractor procurement support includes engineering support, aircraft maintenance, and other support staff. The total procurement costs based on the Shuttle Retirement CR estimates also include T–38 operating costs. Annual funding was included for aircraft modifications and other uncertainties.

12.10.8.3 Ground Rules and Assumptions

The key GR&As associated with the flight crew operations estimate were as follows:

- All costs are in FY05 dollars.
- All contractor travel was included in procurement costs.

12.10.9 Medical Operations

12.10.9.1 Scope

Medical operations include the following functions:

- Medical operations (direct mission support);
- Astronaut health (rehabilitation and conditioning, as well as Flight Medicine Clinic (FMC) and human test support);
- Flight surgeons;
- Shuttle-Orbiter-Medical-System- (SOMS-) like support;
- Crew-Health-Care-System- (CHeCS-) like support;
- Training;
- Contingency;
- Radiation health office;
- Behavioral health and performance;
- Documents and requirements integration;
- Flight medical testing;
- Environmental monitoring;
- Clinic laboratory; and
- Pharmacy.

12.10.9.2 Methodology

Driving assumptions for the basis of this estimate are:

- Lunar sortie missions are similar to Shuttle missions with respect to medical operations level of effort (MCC support, systems complexity, training templates, etc.).
- Similarly, lunar outpost missions are similar to ISS missions for medical operations.
- CEV missions between 2011 and 2016 are only for ISS crew rotation (no non-ISS CEV missions). Therefore, minimal medical operations support is required for CEV MCC console support, training, contingency, environmental and crew medical testing, and documentation support.
- Medical kit provisioning for CEV-to-ISS and lunar sorties will be similar in scope to the SOMS.
- The complement of medical supplies and equipment at the lunar outpost will be similar in scope (complexity, capabilities, consumables, etc.) to the ISS CHeCS.

12.10.9.3 Ground Rules and Assumptions

The key GR&As associated with the medical operations estimate were as follows:

- Costs were estimated initially in FY05 dollars (based on Space Transportation System (STS) and ISS experience), and then a 3.5 percent inflation rate was applied.
- Medical operations costs will begin in 2017 for pre-mission activities and for bridging staff and capabilities after ISS program termination.
- Costs assume little international participation. An international partnership arrangement similar to ISS would add costs for medical coordination with partners.
- ISS ends in 2016.
- The astronaut corps will be reduced in size after 2011, which affects outside medical bill costs.
- Supporting laboratories are assumed to continue to have multiple funding sources such that operations products can be purchased as required.
- Food provisioning was not included.
- Development, production, certification, and sustaining engineering of medical hardware was not included.
- For CEV to ISS, preflight environmental monitoring is performed similar to Shuttle. In-flight and post-flight support is reduced to less than half. The net result when combined with ongoing ISS support is 40 percent of the cost of Shuttle environmental monitoring.

12.10.10 Flight Crew Equipment (FCE)

12.10.10.1 Scope

Flight Crew Equipment (FCE) includes the following crew escape equipment: the pressure suits, hardware processing costs, training events, and other associated content. It also includes the provisioning (food) and associated integration requirements. The estimate includes crew equipment requirements such as electronics, cameras, medical kits, and laptop computers. The estimate includes the parachute packing and testing requirement, including laboratory calibration. The FCE estimate also includes allowances for subsystem management support and new development/modifications (lockers, cables, batteries, etc).

12.10.10.2 Estimating Methodology

The estimates were derived from analogies of existing expenditures for the Space Shuttle and ISS programs.

12.10.10.3 Ground Rules and Assumptions

Based on FY05 planning values provided from the Flight Crew Equipment (FCE) Office personnel at NASA JSC, the value of the annual Space Shuttle Program (SSP) support was assumed as a base value. The current assumption for the SSP requirements is seven crew times five flights per year. While this would infer some fluctuation in supporting the ISS/lunar manifest, a significant portion of this capability was considered fixed and, therefore, applicable fidelity required for the ESAS effort.

A set of values was then added to the base for the ISS variable requirements. The initial value was consistent with the ISS-supported portion of the ESAS manifest and later was doubled to account for lunar outpost operations, including provisioning requirements for lunar crew for 6-month periods.

The current budget baseline values associated with SSP non-prime content such as parachute packing were added. Finally, a wedge was included to approximate the subsystem management/sustaining engineering requirements including a small value for new development items. This value was based in part on current Internal Task Agreement (ITA) support for the NASA JSC Engineering Directorate and estimates provided by the NASA JSC FCE manager.

12.11 Launch Vehicles (LVs) and Earth Departure Stages (EDSs) Cost Estimates

12.11.1 Scope

The LV estimates include the cost for the core booster stage, the upper stage, and any strap-ons applicable to the configuration. Both CLVs and CaLVs were estimated. EDSs were also estimated. There was an extensive trade assessment performed with regards to the LVs and EDSs. Concerning the LV alone, over 36 different variations were assessed during the study. The LV trade space represented a large cross-section of alternatives consisting of both EELVs and SDV configurations. Potential EELV-derived families included the Delta IV and Atlas V Heavy and Atlas Phase 2/Phase 3/Phase X growth vehicles. Shuttle-derived families include four- and five-segment SRBs with new upper stages, External Tank (ET) with side-mounted cargo carriers, and new heavy-lift launch families based on the diameter of the ET.

In support of the 36+ vehicles and 12 EDSs that were evaluated during this study, various engine trades were performed. The scope of engine trades ranges from maximum reuse of existing engines to newly developed engines. The EDS is part of the mass the LV must lift to orbit, and, therefore, is viewed as a payload to the LV. The selected EDS configuration has two J–2S+ engines. EDS estimates were performed adhering to the same GR&As as used for the LV crew upper stages. EDS configurations with no heritage were assessed primarily for cost sensitivities with regard to the type of engine used. With regards to nonrecurring cost, appropriate heritage gained from the crew vehicle upper stage was identified at the subsystem level.

The details of these trades, vehicle descriptions, and study results are contained in **Section 6**, **Launch Vehicles and Earth Departure Stages**.

12.11.2 Methodology

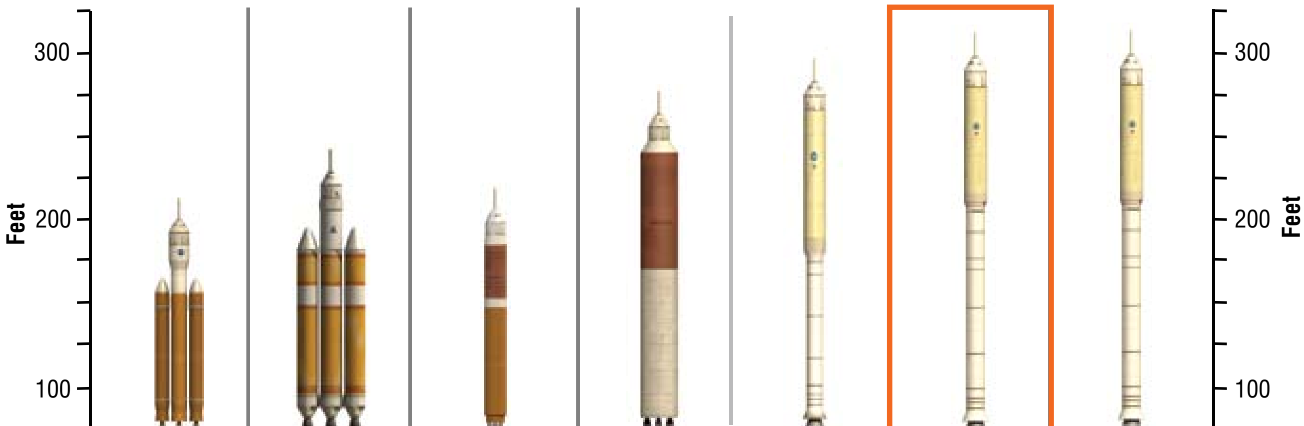
All LV and EDS acquisition cost estimates were prepared using NAFCOM. Hardware estimates were generally performed at the subsystem level or component level if detailed information was available. Software estimates were performed using SEER, with estimates for lines of code based on required functionality. Acquisition cost estimates include the DDT&E cost and the production cost of the first Theoretical Flight Unit (TFU). TFU cost is defined as the cost to produce one unit at a rate of one per year.

12.11.3 Ground Rules and Assumptions

The total LV was estimated, except for the crew LES. Shuttle program and contractor inputs for Shuttle elements and engines were used where applicable, after verification and adjustment for content by program and engineering assessments. All LV option costs include a Structural Test Article (STA) and a main propulsion test article. Total DDT&E also includes three test flights for crewed vehicles and one test flight for cargo vehicles. Required facilities are included for both crew and cargo vehicles. Vehicle physical integration of stages into a complete LV was an additional 4 percent of DDT&E, based on NASA experience. A standard fee of 10 percent was used, and a 20 percent reserve was added to each vehicle estimate. U.S. Government oversight of 25 percent was included for the full cost accounting factor. The full cost accounting factor includes civil service salaries, travel, infrastructure upkeep, utilities, security, cost of facilities, and corporate G&A. Facilities costs are based on engineering assessments of infrastructure requirements. When contractor inputs were available, Government estimates were compared and reconciled with those inputs.

12.11.4 Results

The cost estimates for the most promising lunar CLVs/CaLVs are presented in **Figures 12-9** and **12-10**. The seven crew LEO launch systems consist of four EELV-derived configurations and three Shuttle-derived configurations. The selected CLV (Vehicle 13.1) utilizes Shuttle Reusable Solid Rocket Boosters (RSRBs) as the core stage, with a new upper stage utilizing the SSME. This selection provides a low-cost solution for the crew LEO mission elements and meets the primary consideration in the selection of the CLV for safety/reliability of the system. It also has the ability to meet the early schedule dictated by the need to support ISS beginning in 2011.



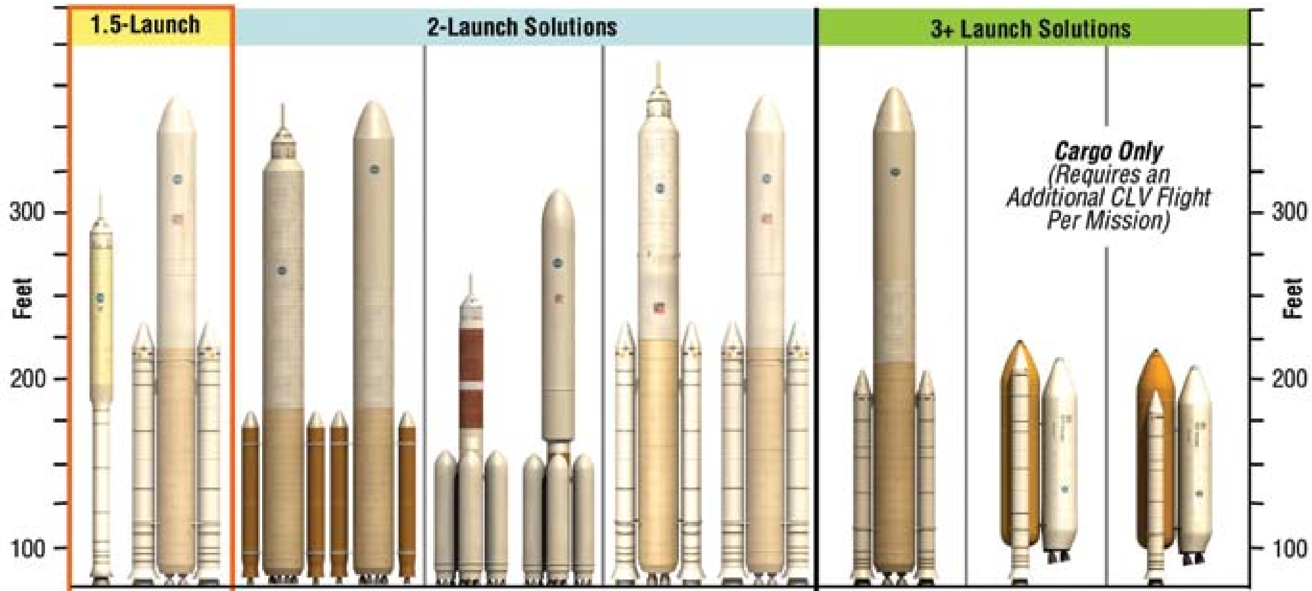
	Human-Rated Atlas V/New US	Human-Rated Delta IV/New US	Atlas Phase 2 (5.4-m Core)	Atlas Phase X (8-m Core)	4 Segment RSRB with 1 SSME	5 Segment RSRB with 1 J–2S	5 Segment RSRB with 4 LR–85
Payload (28.5°)	30 mT	28 mT	26 mT	70 mT	25 mT	26 mT	27 mT
Payload (51.6°)	27 mT	23 mT	25 mT	67 mT	23 mT	24 mT	25 mT
DDT&E*	1.18**	1.03	1.73**	2.36	1.00	1.3	1.39
Facilities Cost	.92	.92	.92	.92	1.00	1.00	1.00
Average Cost/Flight*	1.00	1.11	1.32	1.71	1.00	.96	.96
LOM (mean)	1 in 149	1 in 172	1 in 134	1 in 79	1 in 460	1 in 433	1 in 182
LOC (mean)	1 in 957	1 in 1,100	1 in 939	1 in 614	1 in 2,021	1 in 1,918	1 in 1,429

LOM: Loss of Mission LOC: Loss of Crew US: Upper Stage RSRB: Reusable Solid Rocket Booster

* *All cost estimates include reserves (20% for DDT&E, 10% for Operations), Government oversight/full cost; Average cost/flight based on 6 launches per year.*

** *Assumes NASA has to fund the Americanization of the RD–180. Lockheed Martin is currently required to provide a co-production capability by the USAF.*

Figure 12-9. Comparison of Crew LEO Launch Systems



	5 Segment RSRB In-Line with 5 SSME Core - Cargo	Atlas Phase X (8-m Core)	Atlas Phase 3A (5.4-m Core)	5 Segment RSRB In-Line with 4 SSME Core	4 Segment RSRB In-Line with 3 SSME Core	5 Segment RSRB Side-mount with 3 SSME	4 Segment RSRB Side-mount with 3 SSME
Payload to 28.5	106 mT (125 mT w/ upperstage)	95 mT	94 mT	97 mT	74 mT	80 mT	67 mT
Lunar LV DDT&E*	1.00	1.29	.59	.96	.73	.80	.75
CLV+Lunar Crew/Cargo DDT&E (family op)*	1.00 (4 Seg RSRB w/1 SSME)	1.26 (Atlas V)	1.02 (Phase 2)	.98 (4 Seg RSRB w/1 SSME)	.83 (4 Seg RSRB w/1 SSME)	1.03 (5 Seg RSRB w/J2S-LR85)	.85 (4 Seg RSRB w/1 SSME)
Lunar LV Facility DDTE	1.00	1.33	2.25	1.00			
CLV+Lunar Facilities Cost	1.00	1.12	1.56	1.00			
Lunar LV Average Cost/Flight*	1.00	1.08	1.19	.87	.78	1.13**	1.13**
LOM – Cargo (mean)	1 in 124	1 in 71	1 in 88	1 in 133	1 in 176	1 in 172	1 in 173
LOC (mean)	1 in 2,021	1 in 536	1 in 612	1 in 915	1 in 1,170	N/A	N/A

* All cost estimates include reserves (20% for DDT&E, 10% for Operations), Government oversight/full cost; Average cost/flight based on 6 launches per year.

**Production costs are higher than in-line due to production of separate side-mount cargo carrier

Figure 12-10. Lunar Cargo Launch Systems Comparison

The most promising CaLV configurations are shown in **Figure 12-9**. The selected configuration for heavy lift (Vehicle 27.3) has several advantages. First, the 1.5-launch solution evolves from the CLV, using an updated SRB and the existing SSMEs. It also reduces the total amount of launches per mission. This selection keeps alternate access to space by maintaining the CLV. In addition, the five-segment RSRB in-line SSME core option offers substantially greater lift capacity over four-segment options at modest additional DDT&E.

12.11.5 Operations Cost Model and Recurring Production Costs

12.11.5.1 Operations Cost Model (OCM)

OCM is an Excel-based, parametric model developed for the estimation of space launch systems operations costs. For the purpose of modeling in OCM, launch system operations are defined as those activities that are required to deliver a payload from a launch site on the Earth's surface to LEO. The OCM WBS cost elements represent the full complement of products and services potentially required to operate an LV. The cost elements are arranged into four segments: Program (P), Vehicle (V), Launch Operations (L), and Flight Operations (F). The individual WBS cost elements are assigned to one of these four segments. Estimating cost for every WBS cost element is not required, nor is it necessarily expected. For instance, an unmanned vehicle would not be expected to have costs for F7 Crew Operations or V2 Reusable Hardware Refurbishment. **Figure 12-11** shows the WBS cost element arrangement.

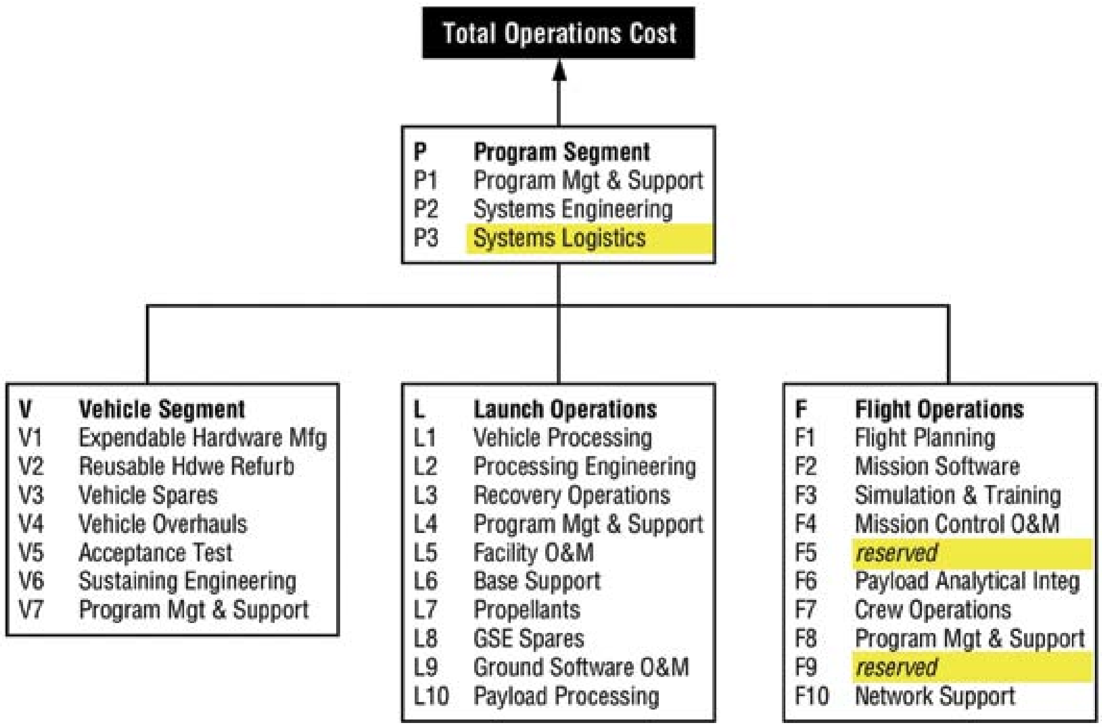


Figure 12-11. OCM Cost Element WBS

For this analysis only, the Vehicle Segment of OCM was used to estimate the recurring production costs of flight hardware elements. Launch operations costs, as defined above, were estimated by NASA KSC personnel, while flight operations costs were estimated by NASA JSC personnel. Program segment costs and full cost accounting were included by adding 25 percent wraps and 10 percent reserve to the other operations cost estimates. The detailed cost estimates for new or modified launch facilities and Ground Support Equipment (GSE) were prepared also by NASA KSC personnel.

12.11.5.2 General Assumptions and Ground Rules

In general, a conservative approach was adopted. Production of hardware for all architectural configurations was estimated as for manned systems, whether the vehicle was designated as "crew" or "cargo."

Production costs include both those for the manufacture of new expendable hardware and the refurbishment of reusable hardware. Production costs are estimated using a TFU cost, either obtained from NAFCOM or derived from historical or vendor data, and applying rate curves using the OCM to estimate the annual production costs. The use of rate curves in these estimates is critical to capture the effects of variances in production (or flight) levels in the launch industry. The entire Shuttle program is an example of how fixed costs can dominate. Regardless of the actual flight rate, the budget remains constant, in large measure because the extensive staff of trained, skilled, and experienced personnel at all levels must be retained during periods of low activity in order to sustain the capacity of the system. Expendable systems behave in a similar manner. When fixed costs predominate, they must be spread over the units produced to recoup the expenditures with revenue. Lower production levels mean higher prices for the items produced. The spread is not linear but is best reflected by a power function model called the rate curve. This is entirely different from a learning curve in which the effects over time cease to have much impact. The cost of a single production unit is found by the following equation:

Average Unit Cost = TFU x (number of units produced/unit of time) ^ (Log_2(Rate Curve %)).

However, a more useful equation is a linear approximation of the power curve. In the OCM, this is derived by estimating the annual operations/production cost at four production levels (e.g., 2, 4, 6, and 8 units per year) which are plotted as Cost on the Y-axis and Number of Units Produced on the X-axis. A "best-fit" linear approximation is then constructed through these four points. The Y-intercept is the fixed production/operations cost, and the slope of the line is the variable cost per unit of output (or marginal cost.) The linear approximation of the rate curve power function is useful because an analyst can then estimate the annual operation cost for any output for any year of operation using the following equation:

Annual Cost = Fixed Cost + Number of Units of Output x Variable Cost Per Unit.

Fixed and variable costs may be aggregated at the segment level if desired. The annual cost for any production or flight rate for any hardware element can be estimated from the fixed and variable costs using the above equation. This allows the analyst to estimate annual production costs in the face of variations in production or flight rates from year to year.

12.11.6 Flight Hardware Production (Manufacturing and Refurbishment)

Flight hardware production costs, including both the manufacture of new expendable hardware (core, upper stage, engines) and the refurbishment of reusable hardware Solid Rocket Motor (SRM)/SRB, part of the interstage), were estimated using the OCM as described above. Cost estimates were based on the concept of the "ongoing" concern, that is, for the period of time under analysis, production facilities would operate from year to year with the same capacity regardless of how many items were actually produced. Implicit in this assumption is the idea that the staff of trained, experienced, and skilled technicians, engineers, and managers does not vary with changes in demand. This assumption allows the use of fixed and variable costs to estimate the annual cost of production and the average unit cost of hardware for a given year.

12.11.7 Family Assessment for Reference 1.5-Launch Solution Architecture (LV 13.1 Followed by LV 27.3)

The cost estimates developed for the family assessments used NAFCOM for calculation of the DDT&E and TFU costs. However, rather than costing each vehicle as an independent stand-alone concept, the family approach assumed an evolved methodology. Each family develops a CLV first. The first LV in the family will lift crew and a limited amount of cargo per launch. The second vehicle developed within the family will be used to lift cargo and, in some families, crew also. Its development takes credit, wherever possible, for any development costs already paid for by the crew vehicle (i.e., engine development, software development, etc.). If full development costs cannot be applied to the CLV, the second CaLV in the family may take some heritage credit, where the subsystem is similar to the CLV (i.e., thermal), thus reducing the development cost of the CaLV. The discussion below deals with the DDT&E costs of the vehicle only. Facilities and test flight costs are not included in the provided dollars. All other GR&As remain the same.

In the 1.5-launch solution family, the CLV is the four-segment RSRB used as the booster, with a new upper stage using the SSME. DDT&E costs were estimated at $3.4B. The evolved vehicle in this family is an in-line heavy-lift CaLV. The ET-based core uses five eSSMEs, with two five-segment RSRBs as strap-ons. As an evolved vehicle from the CLV, the CaLV pays the development cost to make the SSME fully expendable. The CLV paid for altitude start and minimal changes to lower cost. In addition, some of the CLV software can be either modified or reused. Test software, database software, and time/power management are a few of the functions that fall into this category. These savings are somewhat offset by the fact that the CaLV must incur the development cost of the five-segment RSRB. The evolved CaLV saves $0.9M in development costs as compared to stand-alone estimates. It should be noted that the CaLV uses an EDS as an upper stage. This EDS is not included in the costs.

12.12 Launch Site Infrastructure and Launch Operations Cost Estimation

12.12.1 Purpose

The purpose of this subtask of the ESAS was to estimate launch site operations nonrecurring infrastructure costs, such as facilities and GSE, and future recurring launch operations costs. The ESAS team developed numerous potential architectures, which included detailed assessments and descriptions of systems and subsystems for the major elements of the architectures. Launch operations insight was provided by the ESAS team to allow decision-making to proceed with architecture-level launch operations factors properly analyzed and integrated into the life-cycle perspective of the study.

12.12.2 Team

The ESAS team members that contributed or generated costs had the support of numerous other KSC personnel depending on the insight required and the tasks at hand. Team member backgrounds included Level 4 cost estimation competency and experience in previous NASA advanced studies and projects, and Apollo, Shuttle, ISS, or ELV past and current subsystem management or technical experience. Work experience in advanced projects included OSP, Next Generation Launch Technology (NGLT), Space Launch Initiative (SLI), diverse X-vehicle projects, and numerous other architectural studies going back through NASA history into the 1980–90s. Additionally, membership experience included product lifecycle management and obsolescence management.

12.12.3 Approach

ESAS operations analysis of affordability used a combination of cost estimation methods including analogy, historical data, subject matter expertise, and previous studies with contracted engineering firms for construction cost estimates. The operations affordability analysis relied on cost-estimating approaches and was not budgetary in nature, as budgetary approaches generally have extensive processes associated with the generation of costs, and these budgetary processes cannot easily scale to either architecture-level study trades in a broad decision-making space or to trading large quantities of flight and ground systems design details in a short time frame. The operations cost-estimating methods used in the ESAS are attempts at fair and consistent comparisons of levels of effort for varying concepts based on their unique operations cost drivers.

12.12.4 Risks

Numerous risks exist that (1) might alter an ESAS cost estimate, (2) could be significant but were not addressed within the ESAS charter, or (3) reasonably warrant attention in future refinements. These risks include:

- Numerous Space Shuttle Program deferred infrastructure maintenance costs.
- Operational deployment costs for providing for water landing/abort recovery capability for each CEV flight. Margin was allowed in the current operations cost estimate for these operations, but detailed analysis is required to add confidence to eventual operational cost expectations.

- Failure modes may add operational complexities, e.g., an upper stage sizable leak/rupture that would endanger the SRB casings and structural integrity. The identification of such modes as a design evolves and the subsequent operational "mitigation" can drive operational costs upwards in unexpected ways.
- The cost behavior for center institutional costs as Space Shuttle elements retire (Orbiters), are modified (SRBs, SSMEs), and, in some cases, reappear years later (ET-derived core stages) introduces the risk that certain costs will surface in implementation as having a heavier component of fixed costs transferred to the new systems than has been estimated. The conservatism and methodology of the current estimate addresses this but cannot entirely eliminate such risks.

12.12.5 Analysis

12.12.5.1 Launch Site Infrastructure

Cost included in the launch site infrastructure cost estimates are:

- Architectural and Engineering (A&E) and design contract costs and the construction contract costs through construction acceptance (i.e., motor bump tests, wiring ring out, Heating, Ventilating, and Air Conditioning (HVAC) test and balance, etc.); SE&I costs; building outfitting costs (i.e., telephone and communication systems, furniture, movable office partitions, building IT cable distribution systems); and facility activation (facility turned over to operations). GSE and Command, Control, and Checkout systems and consoles are not included. The only additional costs to be added at a higher level are the Other Burden Costs (OBCs) (i.e., Center service pool distributions, Center G&A, and Corporate G&A).

12.12.5.2 Launch Site Operations

Costs included in the launch site recurring or "operations" cost estimates are:

- Civil service and contractor (prime and subcontractors) for (1) logistics and GSE, (2) propellant, and (3) launch operations, inclusive of the following: processing; systems engineering support; facility Operations and Maintenance (O&M); command, control, and checkout center O&M, inclusive of instrumentation; modifications (as an annual allotment, used as required); sustaining engineering; program support (procurement, etc.); communications; base operations support/O&M; weather support; payload integration; and (4) payload processing and Multi-Element Integrated Test (MEIT).

The launch site recurring costs estimation methodology approaches are as shown in **Figures 12-12** and **12-13** for Shuttle-derived and EELV-derived systems, respectively.

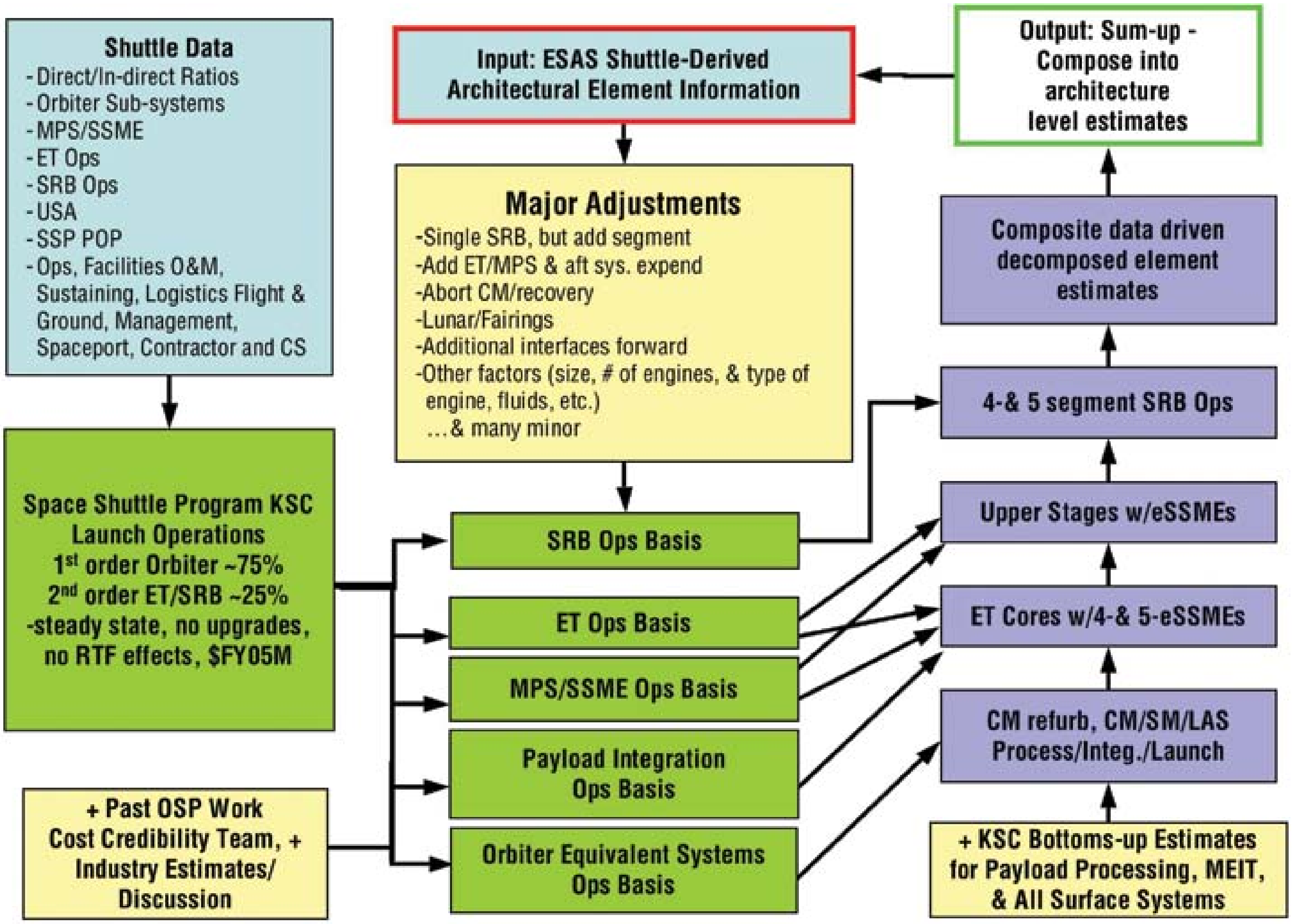


Figure 12-12. Launch Site Operations Cost-Estimating Methodology for Shuttle-Derived ESAS Architectures

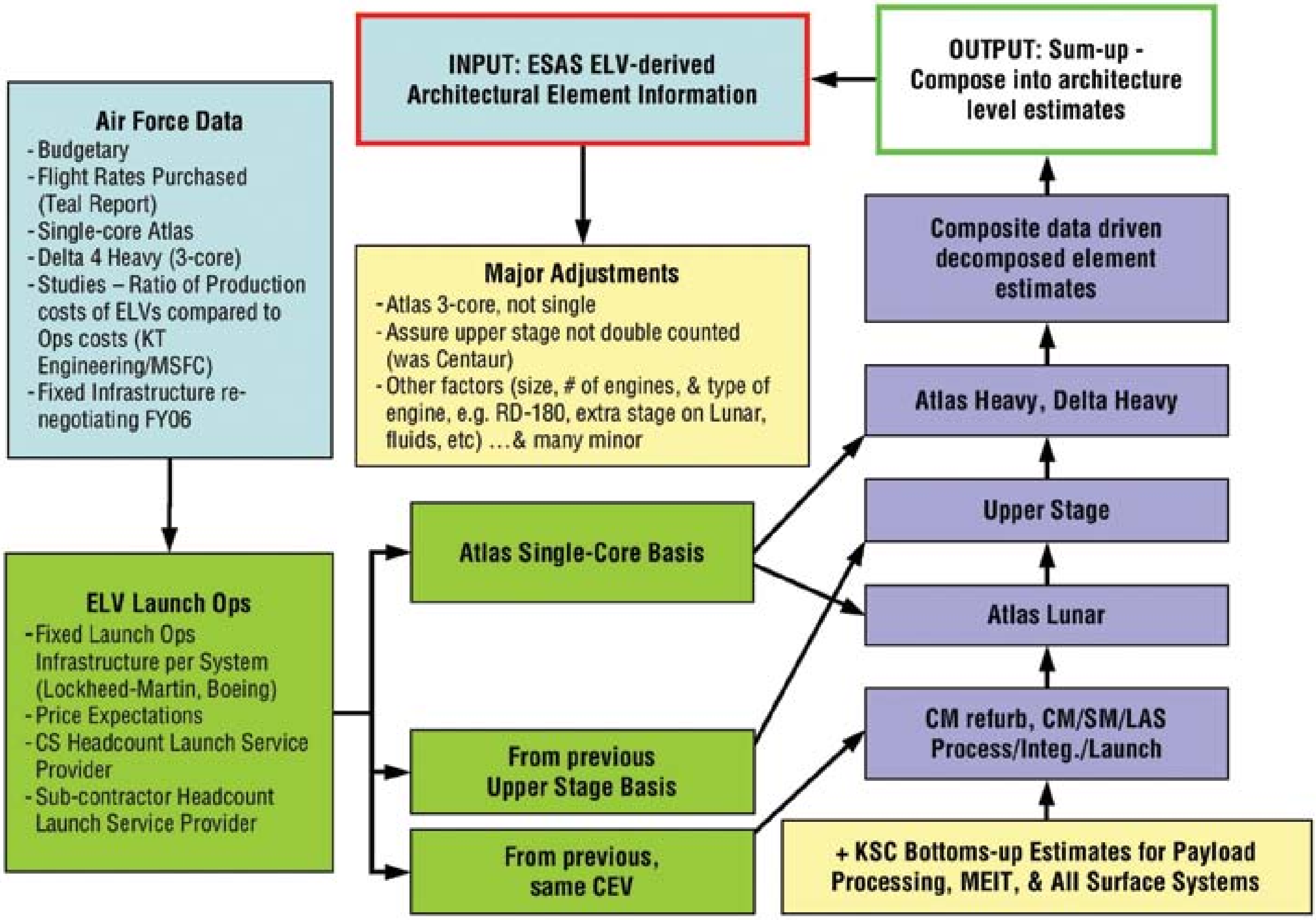


Figure 12-13. Launch Site Operations Cost-Estimating Methodology for EELV-Derived ESAS Architectures

12.12.6 Recommended Opportunities for Improvement in Operations Costs

The full cost estimates developed have not considered some areas that offer opportunities for significant improvements.

- Hypergols should be eliminated at an architectural level across the CEV and LV elements. The need is to create highly operable systems that improve over current systems operations in regard to costs, safety of ground personnel, and overall responsiveness of the system to flight rate demands. A generation of systems has evolved that has deferred such an evolution to nontoxic systems. The elimination of hypergols would begin with newer elements such as the CEV and the upper stage, and would continue as upgrades to SRB- and SSME-related systems (power systems). Then, eventual elimination of hydraulic systems and the implementation of simpler electric actuated systems would become possible, leading to further operability improvements.
- A supply chain improvement study and initiative should be pursued to better understand opportunities and define better ways of doing business. Such an initiative should be based on established Business Process Reengineering (BPR) and IT that are widely employed in the private sector. Improvements in supply chain management would address the areas and the interactions among Integrated Logistics Concepts (ILC) material and information flows, requirements management systems, work control and verification, ground process scheduling, program-level manifesting, corrective action systems, improvement systems, data management systems, sustaining and technical support, procurement, and financial systems. Together, BPR and IT advances and an improved integration of the host of other common network operations and enabling functions, an initiative looking across all supply chain functions, may offer significant opportunities for improvement that must be quantified and defined.
- A hardware-, subsystem-, and system-level reliability improvement initiative is required. An initiative is immediately required to control and improve on the nature of aerospace "small unit buy" (high variance) systems, while still maintaining and meeting unique requirements.
- A much more detailed review of the CEV ground processing activities, including refurbishment and reuse of the CM, is required. Such an analysis would go below architecture into potential subsystems at least to a level 4 of "what-if" definition. Such analysis would feed directly into SRRs and PDRs in 2006–07.

13. Summary and Recommendations

13.1 Lunar Architecture Recommendations

The lunar architecture defined by the Exploration Systems Architecture Study (ESAS) integrates mission mode analysis, flight element functionality, and the activities to be performed on the lunar surface. An integrated analysis of mission performance, safety, reliability, and cost led to the selection of a preferred mission mode, the definition of functional and performance requirements for the vehicle set, and the definition of lunar surface operations. Additionally, the analysis looked back to examine how the Crew Exploration Vehicle (CEV) and Crew Launch Vehicle (CLV) would be used to transport crew and cargo to the International Space Station (ISS), and looked forward to define the systems that will carry explorers to Mars and beyond, in order to identify evolutionary functional, technological, and operational threads that bind these destinations together.

13.1.1 Mission Mode

The ESAS team recommends a combination of Earth Orbit Rendezvous (EOR) and Lunar Orbit Rendezvous (LOR) as the preferred lunar mission mode. The mission mode is the fundamental lunar architecture decision that defines where space flight elements come together and what functions each of these elements perform. The EOR–LOR mode is executed with a combination of the launch of separate crew and cargo vehicles, and by utilizing separate CEV and lander vehicles that rendezvous in lunar orbit. This mission mode combined superior performance with low Life Cycle Cost (LCC) and highest crew safety and mission reliability.

The lunar mission mode study initially considered a wide variety of locations of transportation "nodes" in both cislunar space and the vicinity of Earth. Initial analyses eliminated libration point staging and direct return mission options, leaving the mission mode analysis to investigate a matrix of low lunar (LOR) and Earth orbital (EOR) staging nodes.

13.1.2 Mission Sequence

The ESAS team recommends a mission sequence that uses a single launch of the Cargo Launch Vehicle (CaLV) to place the lunar lander and Earth Departure Stage (EDS) in Earth orbit. The launch of a CLV will follow and place the CEV and crew in Earth orbit, where the CEV and lander/EDS will rendezvous. The combination of the large cargo launch and the CLV is termed a "1.5-launch EOR–LOR" mission. Following rendezvous and checkout in Low Earth Orbit (LEO), the EDS will then inject the stack on a trans-lunar trajectory and be expended. The lander and CEV are captured into lunar orbit by the descent stage of the two-stage lander, and all four crew members transfer to the lander and descend to the surface, leaving the CEV operating autonomously in orbit. The lander features an airlock and the capability to support up to a 7-day surface sortie. Following the lunar surface mission, the lander's ascent stage returns the crew to lunar orbit and docks with the orbiting CEV. The crew members transfer back to the CEV and depart the Moon using the CEV Service Module (SM) propulsion system. The CEV then performs a direct-Earth-entry and parachutes to a land landing on the west coast of the United States.

13.1.3 Lunar Surface Activities

Recommended lunar surface activities will consist of a balance of lunar science, resource utilization, and "Mars-forward" technology and operational demonstrations. The architecture will initially enable sortie-class missions of up to 7 days duration with the entire crew of four residing and performing Extra-Vehicular Activities (EVAs) from the lunar lander.

The ESAS team recommends the deployment of a lunar outpost using the "incremental build" approach. Along with the crew, the lander can deliver 500 kg of payload to the surface, and up to 2,200 kg of additional payload if the maximum landed capacity is utilized. This capability opens the possibility of deploying an outpost incrementally by accumulating components delivered by sortie missions to a common location. This approach is more demanding than one that delivers larger cargo elements. In particular, the habitat, power system, pressurized rovers, and some resource utilization equipment will be challenging to divide and deploy in component pieces. The alternative to this incremental approach is to develop a dedicated cargo lander that can deliver large payloads of up to 21 mT.

The study team defined high-priority landing sites that were used to establish sortie mission performance. Of those sites, a south polar location was chosen as a reference outpost site in order to further investigate the operations at a permanent outpost. A Photovoltaic (PV) power system was chosen as the baseline power system for the outpost.

13.1.4 Propulsion Choices and Delta-V Assignment

The ESAS team examined a wide variety of propulsion system types and potential delta-V allocations for each architecture element. It is recommended that the CaLV's upper stage will serve as the EDS for lunar missions and will perform the Trans-Lunar Injection (TLI) propulsive maneuver. The descent stage of the lunar lander was selected to perform Lunar Orbit Insertion (LOI) and up to 200 m/s of plane change using Liquid Oxygen (LOX)/hydrogen propulsion. The lunar lander descent stage will perform a coplanar descent to the surface using the same engine that performed LOI, and the crew will perform the surface mission while the CEV orbits autonomously. The lunar lander ascent stage will perform a coplanar ascent using LOX/methane propulsion that is common with the CEV SM propulsion system. The SM will perform up to a 90-deg plane change and Trans-Earth Injection (TEI) with full co-azimuth control (1,450 m/s total delta-V).

Pump-fed LOX/hydrogen propulsion was selected for the lunar descent stage because of the great performance, cost, and risk leverage that was found when the lunar lander descent stage propulsion efficiency was increased by the use of a LOX/hydrogen system. To achieve a high-reliability lunar ascent propulsion system, and to establish the linkage to in-situ propellant use, common pressure-fed LOX/methane engines were chosen for the CEV SM and lunar ascent stage propulsion systems.

13.1.5 Global Access

It is recommended that the lunar architecture preserve the option for full global landing site access for sortie or outpost missions. Landing at any site on the Moon sizes the magnitude of the LOI maneuver. A nominal 900-m/s LOI burn enables access to the equator and poles, and a maximum of 1,313 m/s is required for immediate access to any site on the lunar globe. The architecture uses a combination of orbital loiter and delta-V to access any landing in order to balance additional propulsive requirements on the lander descent stage and additional orbital lifetime of the CEV systems. The lander descent stage was sized for a 900-m/s LOI plus a 200-m/s maximum nodal plane change, for a total of 1,100 m/s in addition to lunar descent propulsion. This value allows the crew to immediately access 84 percent of the lunar surface, and to have full global access with no more than 3 days loiter in lunar orbit.

13.1.6 Anytime Return

It is recommended that the architecture provide the capability to return to Earth in 5 days or less for sortie missions at any site on the lunar globe. The requirement to return anytime from the surface of the Moon to Earth was the design driver of the SM propulsion system. The lunar mission requires a total of 1,450 m/s of delta-V, combining a 900-m/s TEI maneuver, a worst-case 90-deg nodal plane change, and Earth entry azimuth control. This capability enables "anytime return" if the lander is able to perform a coplanar ascent to the CEV. For sortie duration missions of 7 days or less, the CEV's orbital inclination and node will be chosen to enable a coplanar ascent. Outpost missions will also have anytime return capability if the outpost is located at a polar or equatorial site. For other sites, loitering on the surface at the outpost may be required to enable ascent to the orbiting CEV.

13.1.7 Lunar Lander

The recommended lunar lander provides the capability to capture itself and the CEV into lunar orbit, to perform a plane change prior to descent, and to descend to the lunar surface with all four crew members using a throttleable LOX/hydrogen propulsion system. On the lunar surface, the lander serves as the crew's habitat for the duration of the surface stay, and provides full airlock capability for EVA. Additionally, the lander carries a nominal payload of 500 kg and has the capability to deliver an additional 2,200 kg to the lunar surface. The lander's ascent stage uses LOX/methane propulsion to carry the crew back into lunar orbit to rendezvous with the waiting CEV. The lander's propulsion systems are chosen to make it compatible with In-Situ Resource-Utilization- (ISRU-) produced propellants and common with the CEV SM propulsion system.

13.1.8 ISS-Moon-Mars Connections

Evolutionary paths were established within the architecture to link near-term ISS crew and cargo delivery missions, human missions to the lunar surface, and farther-term human missions to Mars and beyond. The key paths that enable the architecture to evolve over time are the design of the CEV, the choice of CLV and CaLV, the selection of technologies (particularly propulsion technologies), and the operational procedures and systems that extend across the destinations. The CEV is sized to accommodate crew sizes up to the Mars complement of six. The CLV was chosen to be a reliable crew launch system that would be the starting point of a crew's journey to the ISS, Moon, or Mars; and the CaLV was chosen, in part, to deliver 100-mT-class human Mars mission payloads to LEO. Propulsion choices were made to link propulsive elements for the purpose of risk reduction, and to enable the use of future ISRU-produced propellants. These propellant choices are further linked to the ISRU technology experiments to be performed on the planetary surfaces. Finally, EVA systems and mission operations will be developed to share common attributes across all ISS, lunar, and Mars destinations.

13.2 CEV Recommendations

One of the key requirements to enable a successful human space exploration program is the development and implementation of a vehicle capable of transporting and housing crew on LEO, lunar, and Mars missions. A major portion of the ESAS effort focused on the design and development of the CEV, the means by which NASA plans to accomplish these mission objectives. This section provides a summary of the findings and recommendations specific to the CEV.

It is recommended that the CEV incorporate a separate Crew Module (CM), SM, and Launch Abort System (LAS) arrangement similar to that of Apollo, and that these modules be capable of multiple functions to save costs. The CEV design was sized for lunar missions carrying a crew of four. Also, the vehicle was designed to be reconfigurable to accommodate up to six crew for ISS and future Mars mission scenarios. The CEV can transfer and return crew and cargo to the ISS and stay for 6 months in a quiescent state for nominal end of mission, or return of crew at anytime in the event of an emergency. The lunar CEV design has direct applications to ISS missions without significant changes in the vehicle design. The lunar and ISS configurations share the same SM, but the ISS mission has much lower delta-V requirements. Hence, the SM propellant tanks can be loaded with additional propellant for ISS missions to provide benefits in launch aborts, on-orbit phasing, and ISS reboost. Other vehicle block derivatives can deliver pressurized and unpressurized cargo to the ISS.

The ESAS team's next CEV recommendation addresses the vehicle shape. Using an improved blunt-body capsule for the CM was found to be the least costly, fastest, and safest approach for bringing ISS and lunar missions to reality. The key benefits for a blunt-body configuration were found to be less weight, a more familiar aerodynamic design from human and robotic heritage (resulting in less design time and cost), acceptable ascent and entry abort load levels, crew seating orientation ideal for all loading events, easier Launch Vehicle (LV) integration, and, improved entry controllability during off-nominal conditions. Improvements on the Apollo shape will offer better operational attributes, especially by increasing the Lift-to-Drag (L/D) ratio, improving Center of Gravity (CG) placement, creating a more stable configuration, and employing a lower angle of attack for reduced sidewall heating.

A CM measuring 5.5 m in diameter was chosen to support the layout of six crew without stacking the crew members above or below each other. A crew tasking analysis also confirmed the feasibility of the selected vehicle volume. The recommended pressurized volume for the CM is approximately three times that of the Apollo Command Module. The available internal volume provides flexibility for future missions without the need for developing an expendable mission module. The vehicle scaling also considered the performance of the proposed CLV, which is a four-segment Solid Rocket Booster (SRB) with a single Space Shuttle Main Engine (SSME) upper stage. The CEV was scaled to maximize vehicle size while maintaining adequate performance margins on the CLV.

It is recommended that the CEV utilize an androgynous Low-Impact Docking System (LIDS) to mate with other exploration elements and to the ISS. This requires the CEV-to-ISS docking adapters to be LIDS-compatible. It is proposed that to develop small adapters to convert ISS interfaces to LIDS. The exact implementation is the course of further study.

An integrated pressure-fed LOX/methane service propulsion system/Reaction Control System (RCS) is recommended for the SM. Selection of this propellant combination was based on performance and commonality with the ascent propulsion system on the Lunar Surface Access Module (LSAM). The risk associated with this type of propulsion system for a lunar mission can be substantially reduced by developing the system early and flying it to the ISS. There is high risk in developing a LOX/methane propulsion system by 2011, but development schedules for this type of propulsion system have been studied and are in the range of hypergolic systems.

Studies were performed on the levels of radiation protection required for the CEV CM. Based on an aluminum cabin surrounded by bulk insulation and composite skin panels with a Thermal Protection System (TPS), no supplemental radiation protection is recommended.

Solar arrays combined with rechargeable batteries were recommended for the SM due to the long mission durations dictated by some of the Design Reference Missions (DRMs). The ISS crew transfer mission and long-stay lunar outpost mission require the CEV to be on-orbit for 6 to 9 months, which is problematic for fuel cell reactants.

The choice of a primary land-landing mode was primarily driven by a desire for land landing in the Continental United States (CONUS) for ease and minimal cost of recovery, post-landing safety, and reusability of the spacecraft. However, it is recommended that the design of the CEV CM should incorporate both a water- and land-landing capability. Ascent aborts will require the ability to land in water, while other off-nominal conditions could lead the spacecraft to a land landing, even if not the primary intended mode. However, a vehicle designed for a primary land-landing mode can more easily be made into a primary water lander than the reverse situation. For these reasons, the study attempted to create a CONUS land-landing design from the outset, with the intention that a primary water lander would be a design off-ramp if the risk or development cost became too high.

In order for CEV entry trajectories from LEO and lunar return to use the same landing sites, it is recommended that NASA utilize skip-entry guidance on the lunar return trajectories. The skip-entry lunar return technique provides an approach for returning crew to a single CONUS landing site anytime during a lunar month. The Apollo-style direct-entry technique requires water or land recovery over a wide range of latitudes. The skip-entry includes an exoatmospheric correction maneuver at the apogee of the skip maneuver to remove dispersions accumulated during the skip maneuver. The flight profile is also standardized for all lunar return entry flights. Standardizing the entry flights permits targeting the same range-to-landing site trajectory for all return scenarios so that the crew and vehicle experience the same heating and loads during each flight. This does not include SM disposal considerations, which must be assessed on a case-by-case basis.

For emergencies, it is recommended that the CEV also include an LAS that will pull the CM away from the LV on the pad or during ascent. The LAS concept utilizes a 10-g tractor rocket attached to the front of the CM. The LAS is jettisoned from the launch stack shortly after second-stage ignition. Launch aborts after LAS jettison are performed by using the SM service propulsion system. Launch abort study results indicate a fairly robust abort capability for the CEV/CLV and a 51.6-deg-inclination ISS mission, given 1,200 m/s of delta-V and a Thrust-to-Weight (T/W) ratio of at least 0.25. Abort landings in the mid-North Atlantic can be avoided by either an Abort-To-Orbit (ATO) or posigrade Trans-Atlantic Abort Landing (TAL) south of Ireland. Landings in the Middle East, the Alps, or elsewhere in Europe can be avoided by either an ATO or a retrograde TAL south of Ireland. For 28.5-deg-inclination lunar missions, abort landings in Africa can be avoided by either an ATO or a retrograde TAL to the area between the Cape Verde islands and Africa. However, it appears that, even with 1,724 m/s of delta-V, some abort landings could occur fairly distant from land. However, once the ballistic impact point crosses roughly 50°W longitude, posigrade burns can move the abort landing area downrange near the Cape Verde islands.

13.3 Launch System Architecture Recommendations

13.3.1 Recommendation 1

Adopt and pursue a Shuttle-derived architecture as the next-generation launch system for crewed flights into LEO and for 125-mT-class cargo flights for exploration beyond Earth orbit. After thorough analysis of multiple EELV- and Shuttle-derived options for crew and cargo transportation, Shuttle-derived options were found to have significant advantages with respect to cost, schedule, safety, and reliability. Overall, the Shuttle-derived option was found to be the most affordable by leveraging proven vehicle and infrastructure elements and using those common elements in the heavy-lift CaLV as well as the CLV. Using elements that have a human-rated heritage, the CaLV can enable unprecedented mission flexibility and options by allowing a crew to potentially fly either on the CLV or CaLV for 1.5-launch or 2-launch lunar missions that allow for heavier masses to the lunar surface. The Shuttle-derived CLV provides lift capability with sufficient margin to accommodate CEV crew and cargo variant flights to ISS and potentially provides added services, such as station reboost.

The extensive flight and test databases of the RSRB and SSME give a solid foundation of well-understood main propulsion elements on which to anchor next-generation vehicle development and operation. The Shuttle-derived option allows the Nation to leverage extensive ground infrastructure investments and maintains access to solid propellant at current levels. Furthermore, the Shuttle-derived option displayed more versatile and straightforward growth paths to higher lift capability with fewer vehicle elements than other options.

The following specific recommendations are offered for LV development and utilization.

13.3.2 Recommendation 2

Initiate immediate development of a CLV utilizing a single four-segment RSRB first stage and a new upper stage using a single SSME. The reference configuration, designated LV 13.1 in this study, provides the payload capability to deliver a lunar CEV to low-inclination Earth orbits required by the exploration architectures and to deliver CEVs configured for crew and cargo transfer missions to the ISS. The existence and extensive operational history of human-rated Shuttle-derived elements reduce safety risk and programmatic and technical risk to enable the most credible development path to meet the goal of providing crewed access to space by 2011. The series-burn configuration of LV 13.1 provides the crew with an unobstructed escape path from the vehicle using an LAS in the event of a contingency event from launch through EOI. Finally, if required, a derivative cargo-only version of the CLV, designated in this report as LV 13.1S, can enable autonomous, reliable delivery of unpressurized cargo to ISS of the same payload class that the Shuttle presently provides.

13.3.3 Recommendation 3

To meet lunar and Mars exploration cargo requirements, begin development as soon as practical of an in-line Shuttle-derived CaLV configuration consisting of two five-segment RSRBs and a core vehicle with five aft-mounted SSMEs derived from the present ET and reconfigured to fly payload within a large forward-mounted aerodynamic shroud. The specific configuration is designated LV 27.3 in this report. This configuration provides superior performance to any side-mount Shuttle-derived concept and enables varied configuration options as the need arises. A crewed version is also potentially viable because of the extensive use of human-rated elements and in-line configuration. The five-engine core and two-engine EDS provides sufficient capability to enable the "1.5-launch solution," which requires one CLV and one CaLV flight per lunar mission—thus reducing the cost and increasing the safety/reliability of each mission. The added lift capability of the five-SSME core allows the use of a variety of upper stage configurations, with 125 mT of lift capability to LEO. LV 27.3 will require design, development, and certification of a five-segment RSRB and a new core vehicle, but such efforts are facilitated by their historical heritage in flight-proven and well-characterized hardware. Full-scale design and development should begin as soon as possible synchronized with CLV development to facilitate the first crewed lunar exploration missions in the middle of the next decade.

13.3.4 Recommendation 4

To enable the 1.5-launch solution and potential vehicle growth paths as previously discussed, NASA should undertake development of an EDS based on the same tank diameter as the cargo vehicle core. The specific configuration should be a suitable variant of the EDS concepts designated in this study as EDS S2x, depending on the further definition of the CEV and LSAM. Using common manufacturing facilities with the Shuttle-derived CaLV core stage will enable lower costs. The recommended EDS thrust requirements will require development of the J–2S+, which is a derivative of the J–2 upper stage engine used in the Apollo/Saturn program, or another in-space high-performance engine/cluster as future trades indicate. As with the Shuttle-derived elements, the design heritage of previously flight-proven hardware will be used to advantage with the J–2S+. The TLI capability of the EDS S2x is approximately 65 mT, when used in the 1.5-launch solution mode, and enables many of the CEV/LSAM concepts under consideration. In a single-launch mode, the S2B3 variant can deliver 54.6 mT to TLI, which slightly exceeds the TLI mass of Apollo 17, the last crewed mission to the Moon in 1972.

13.3.5 Recommendation 5

Continue to rely on the EELV fleet for scientific and ISS cargo missions in the 5- to 20-mT lift range.

13.4 Technology Assessment Recommendations

As a result of the technology assessment, it is recommended that the overall funding of the Exploration Systems Mission Directorate (ESMD) for Research and Technology (R&T) be reduced by approximately 50 percent to provide sufficient funds in order to accelerate the development of the CEV to reduce the gap in U.S. human spaceflight after Shuttle retirement. This can be achieved by focusing the technology program only on those technologies required to enable the architecture elements as they are needed and because the recommended ESAS architecture does not require a significant level of technology development to accomplish the required missions. Prior to the ESAS, the planned technology development funding profile for ESMD was as shown in **Figure 13-1**. The ESAS recommendations for a revised, architecture-driven technology development is as shown in **Figure 13-2**.

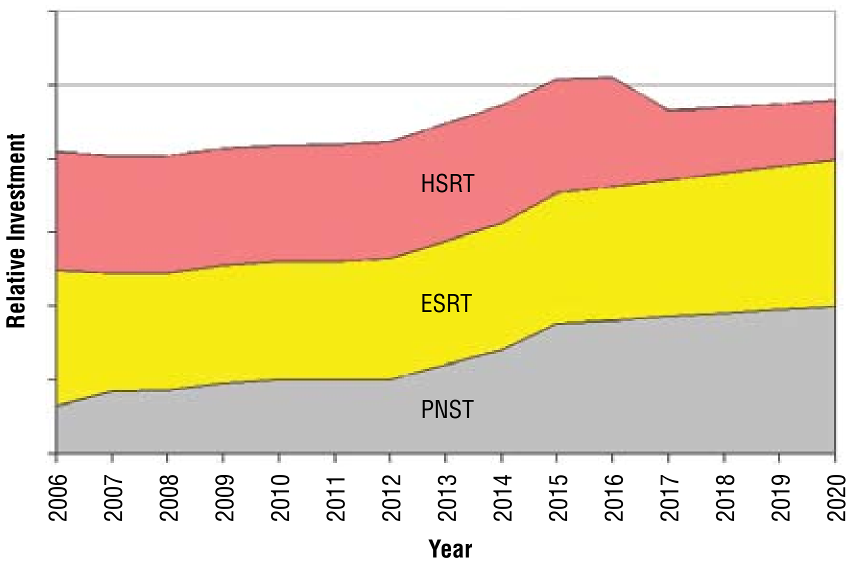


Figure 13-1. FY06–FY19 Original Funding Profile

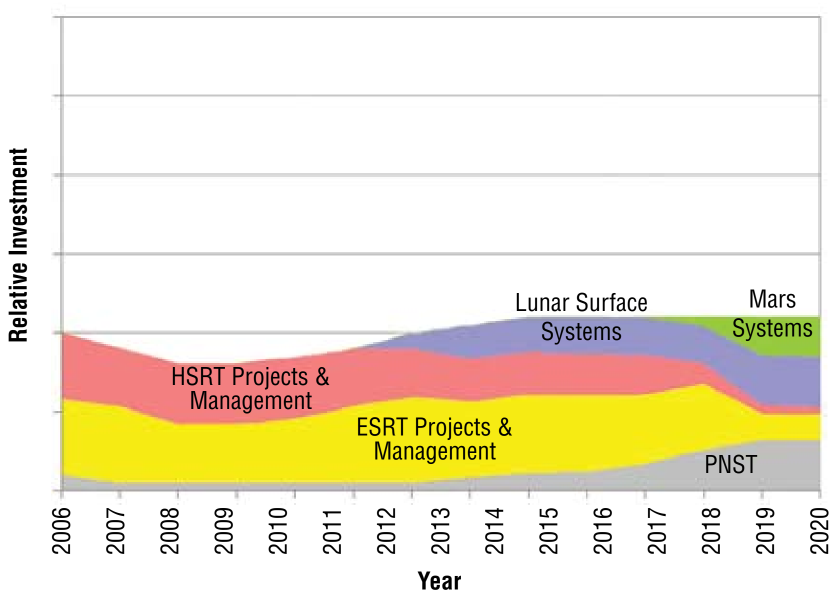


Figure 13-2. FY06–FY19 ESAS-Recommended Funding Profile

Figures 13-3 through **13-5** show, respectively, the overall recommended R&T budget broken out by program with liens, functional need category, and mission. “Protected” programs include those protected from cuts due to statutory requirements or previous commitments.

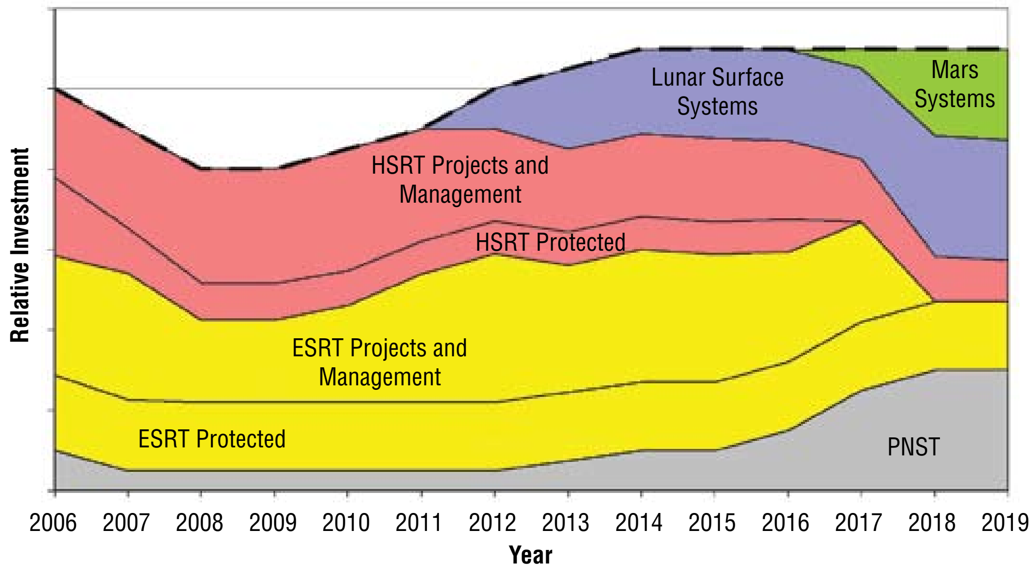


Figure 13-3. Overall Recommended R&T Budget Broken Out by Program with Liens

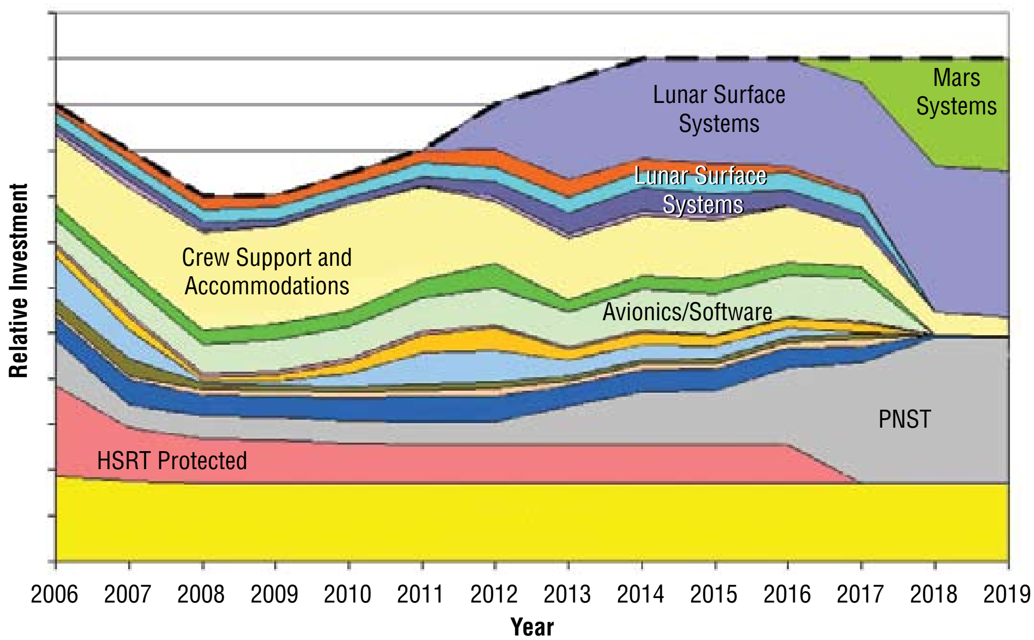


Figure 13-4. Overall Recommended R&T Budget Broken Out by Functional Need Category

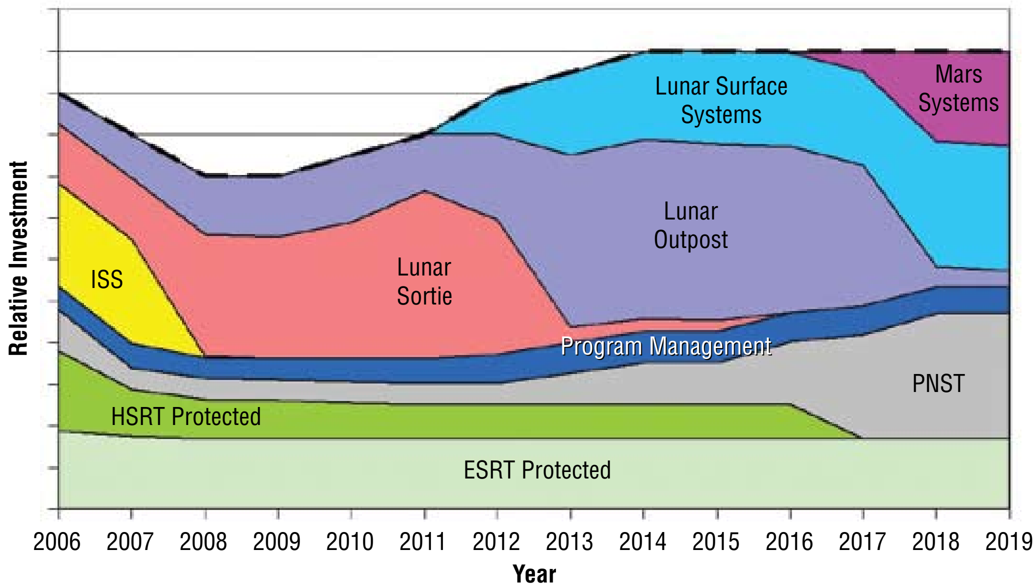


Figure 13-5. Overall Recommended R&T Budget Broken Out by Mission

The existing funding profile includes 10 percent management funds and approximately 30 percent of liens due to prior Agency agreements (e.g., Multi-User System and Support (MUSS), the Combustion Integrated Rack (CIR), and the Fluids Integrated Rack (FIR)) and legislated requirements (e.g., Small Business Innovation Research (SBIR), Small Business Technology Transfer (STTR)).

The final recommended technology funding profile was developed in coordination with the ESAS cost estimators using the results of the technology assessment. The following seven key recommendations arose from the technology assessment:

- ESMD should share costs with the SOMD for MUSS, CIR, and FIR. MUSS, CIR, and FIR are all ISS operations activities and, as such, should not be funded by ESMD R&T. Funds were identified in the recommended budget; however, cost-sharing plans should be implemented to ensure these facilities are efficiently operated.
- ESMD should transfer the Alpha Magnetic Spectrometer (AMS) to the Science Mission Directorate (SMD) to compete for funding with other science experiments. The AMS may be of scientific importance, but does directly contribute to meeting ESMD R&T needs. Therefore, it should be moved to SMD for consideration with other science missions.
- ESMD should quickly notify existing Exploration Systems Research and Technology (ESRT) projects not selected by ESAS that they will receive no funding beyond Fiscal Year 2005 (FY05). If work on the existing ESRT projects not selected for continuation is not stopped in FY05, there will be a potential for significant FY06 funds required to cover the contracts. Accordingly, appropriate notice must be provided as soon as possible to ensure efficient transition.

- ESMD should move Systems Analysis and Tool Development activities (and budget) to a directorate-level organization—no longer in ESRT. These system analysis and tool development functions should not be buried in multiple disparate organizations. While each organization will require its own analytical capabilities, a focal point should be established at the directorate level to ensure consistency in the ground rules, assumptions, and analytical methodologies across ESMD. This will ensure decision makers are provided "apples-to-apples" analysis results. These activities are also required to handle "what-if" studies and strategic analysis actions to provide greater stability in the development programs (i.e., development programs can focus on their work and avoid the disruption of frequent strategic studies and issue analyses).
- Key ESAS personnel should work with ESMD to facilitate implementation. Many technologies require immediate commencement on an accelerated schedule to meet aggressive development deadlines. Key ESAS personnel should also work with ESMD to ensure the analytical basis supporting ESAS recommendations is not lost, but rather carefully preserved and refined to improve future decisions.
- ESMD should develop a process for close coordination between architecture refinement studies and technology development projects. Technology projects should be reviewed with the flight element development programs on a frequent basis to ensure alignment and assess progress.
- ESMD should develop a process for transitioning matured technologies to flight element development programs. Experience shows that technologies have a difficult time being considered for incorporation into development projects due to uncertainty and perceived risk. The technologies identified in this assessment are essential for the architecture and, therefore, a structured process for transitioning them must be implemented to ensure timely integration into development projects with minimal risk and uncertainty.

The key technology development project recommendations from the study are shown in **Table 13-1**.

Table 13-1. Technology Project Recommendations

Number	ESAS Control Number	Program	Category	New Projects
1	1A	ESRT	Structures	Lightweight structures, pressure vessel, and insulation.
2	2A	ESRT	Protection	Detachable, human-rated, ablative environmentally compliant TPS.
3	2C	HSRT	Protection	Lightweight radiation protection for vehicle.
4	2E	HSRT	Protection	Dust and contaminant mitigation.
5	3A	ESRT	Propulsion	Human-rated, 5–20 klbf class in-space engine and propulsion system (SM for ISS orbital operations, lunar ascent and TEI, pressure-fed, LOX/CH4, with LADS). Work also covers 50–100 lbs nontoxic (LOX/CH4) RCS thrusters for SM.
6	3B	ESRT	Propulsion	Human-rated deep throttleable 5–20 klbf engine (lunar descent, pump-fed LOX/LH2).
7	3C	ESRT	Propulsion	Human-rated, pump-fed LOX/CH4 5–20 klbf thrust class engines for upgraded lunar LSAM ascent engine.
8	3D	ESRT	Propulsion	Human-rated, stable, nontoxic, monoprop, 50–100 lbf thrust class RCS thrusters (CM and lunar descent).
9	3F	ESRT	Propulsion	Manufacturing and production to facilitate expendable, reduced-cost, high production-rate SSMEs.
10	3G	ESRT	Propulsion	Long-term, cryogenic, storage and management (for CEV).
11	3H	ESRT	Propulsion	Long-term, cryogenic, storage, management, and transfer (for LSAM).
12	3K	ESRT	Propulsion	Human-rated, nontoxic 900-lbf Thrust Class RCS thrusters (for CLV and heavy-lift upper stage).
13	4B	ESRT	Power	Fuel cells (surface systems).
14	4E	ESRT	Power	Space-rated Li-ion batteries.
15	4F	ESRT	Power	Surface solar power (high-efficiency arrays and deployment strategy).
16	4I	ESRT	Power	Surface power management and distribution (e.g., efficient, low mass, autonomous).
17	4J	ESRT	Power	LV power for thrust vector and engine actuation (nontoxic APU).
18	5A	HSRT	Thermal Control	Human-rated, nontoxic active thermal control system fluid.
19	5B	ESRT	Thermal Control	Surface heat rejection.
20	6A	ESRT	Avionics and Software	Radiation hardened/tolerant electronics and processors.
21	6D	ESRT	Avionics and Software	Integrated System Health Management (ISHM) (CLV, LAS, EDS, CEV, lunar ascent/descent, habitat/Iso new hydrogen sensor for on-pad operations).
22	6E	ESRT	Avionics and Software	Spacecraft autonomy (vehicles & habitat).
23	6F	ESRT	Avionics and Software	Automated Rendezvous and Docking (AR&D) (cargo mission).
24	6G	ESRT	Avionics and Software	Reliable software/flight control algorithms.
25	6H	ESRT	Avionics and Software	Detector and instrument technology.
26	6I	ESRT	Avionics and Software	Software/digital defined radio.

Table 13-1. Technology Project Recommendations (continued)

Number	ESAS Control Number	Program	Category	New Projects
27	6J	ESRT	Avionics and Software	Autonomous precision landing and GN&C (Lunar & Mars).
28	6K	ESRT	Avionics and Software	Lunar return entry guidance systems (skip entry capability).
29	6L	ESRT	Avionics and Software	Low temperature electronics and systems (permanent shadow region ops).
30	7A	HSRT	ECLS	Atmospheric management - CMRS (CO2, Contaminants and Moisture Removal System).
31	7B	HSRT	ECLS	Advanced environmental monitoring and control.
32	7C	HSRT	ECLS	Advanced air and water recovery systems.
33	8B	HSRT	Crew Support and Accommodations	EVA Suit (including portable life suppport system).
34	8E	HSRT	Crew Support and Accommodations	Crew healthcare systems (medical tools and techniques, countermeasures, exposure limits).
35	8F	HSRT	Crew Support and Accommodations	Habitability systems (waste management, hygiene).
36	9C	ESRT	Mechanisms	Autonomous/teleoperated assembly and construction (and deployment) for lunar outpost.
37	9D	ESRT	Mechanisms	Low temperature mechanisms (lunar permanent shadow region ops).
38	9E	ESRT	Mechanisms	Human-rated airbag or alternative Earth landing system for CEV.
39	9F	ESRT	Mechanisms	Human-rated chute system with wind accommodation.
40	10A	ESRT	ISRU	Demonstration of regolith excavation and material handling for resource processing.
41	10B	ESRT	ISRU	Demonstration of oxygen production from regolith.
42	10C	ESRT	ISRU	Demonstration of polar volatile collection and separation.
43	10D	ESRT	ISRU	Large-scale regolith excavation, manipulation and transport (i.e., including radiation shielding construction).
44	10E	ESRT	ISRU	Lunar surface oxygen production for human systems or propellant.
45	10F	ESRT	ISRU	Extraction of water/hydrogen from lunar polar craters.
46	10H	ESRT	ISRU	In-situ production of electrical power generation (lunar outpost solar array fabrication).
47	11A	ESRT	Analysis and Integration	Tool development for architecture/mission/technology analysis/design, modeling and simulation.
48	11B	ESRT	Analysis and Integration	Technology investment portfolio assessment and systems engineering and integration.
49	12A	ESRT	Operations	Supportability (commonality, interoperability, maintainability, logistics, and in-situ fab.)
50	12B	ESRT	Operations	Human-system interaction (including robotics).
51	12C	ESRT	Operations	Surface handling, transportation, and operations equipment (Lunar or Mars).
52	12E	ESRT	Operations	Surface mobility.

14. Architecture Roadmap

As outlined previously, the Exploration Systems Architecture Study (ESAS) team developed a time-phased, evolutionary architecture approach to return humans to the Moon, to service the International Space Station (ISS) after Space Shuttle retirement, and to eventually transport humans to Mars. The individual elements were integrated into overall Integrated Master Schedules (IMSs) and detailed, multi-year integrated Life Cycle Costs (LCCs) and budgets. These detailed results are provided in **Section 11**, **Integrated Master Schedule**, and **Section 12**, **Cost**. A top-level roadmap for ESAS architecture implementation is provided in **Figure 14-1**.

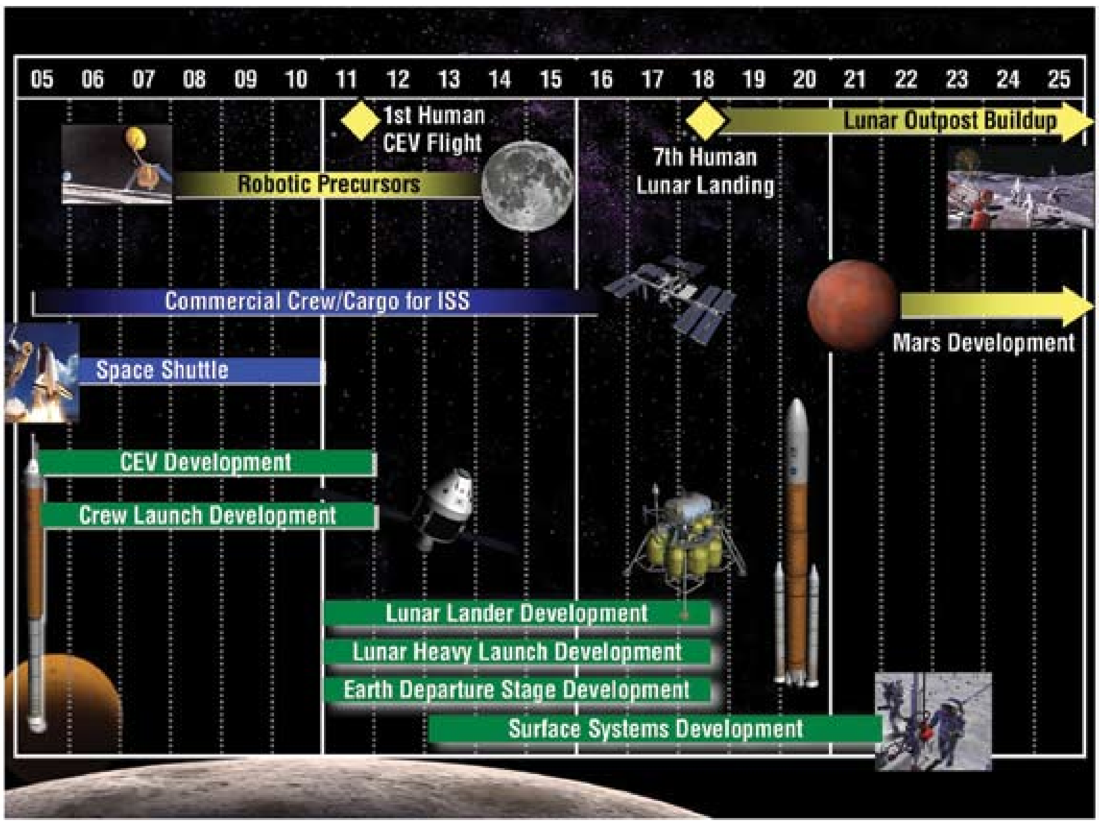


Figure 14-1. ESAS Architecture Implementation Roadmap

In this implementation, the Space Shuttle would be retired in 2010, using its remaining flights to deploy the ISS and, perhaps, to service the Hubble Space Telescope (HST). Crew Exploration Vehicle (CEV) and Crew Launch Vehicle (CLV) development would begin immediately, leading to the first crewed CEV flight to the ISS in 2011. Options for transporting cargo to and from the ISS would be pursued in cooperation with industry, with a goal of purchasing transportation services commercially. Lunar robotic precursor missions would begin immediately with the development and launch of the Lunar Reconnaissance Orbiter mission and continue with a series of landing and orbiting probes to prepare for extended human lunar exploration. In 2011, development would begin of the major elements required to return humans to the Moon—the Lunar Surface Access Module (LSAM), Cargo Launch Vehicle (CaLV), and Earth Departure Stage (EDS). These elements would be developed and tested in an integrated fashion, leading to a human lunar landing in 2018. Starting in 2018, a series of short-duration lunar sortie missions would be accomplished, leading up to the deployment and permanent habitation of a lunar outpost. The surface systems (e.g., rovers, habitats, power systems) would be developed as required. Lunar missions would demonstrate the systems and technologies needed for eventual human missions to Mars.

15. Architecture Advantages

The Exploration Systems Architecture Study (ESAS) team examined a wide variety of architecture element configurations, functionality, subsystems, technologies, and implementation approaches. Alternatives were systematically and objectively evaluated against a set of Figures of Merit (FOMs). The results of these many trade studies are summarized in each major section of this report and in the recommendations in **Section 13**, **Summary of Recommendations**.

Although many of the key features of the architecture are similar to systems and approaches used in the Apollo Program, the selected ESAS architecture offers a number of advantages over that of Apollo, including:

- Double the number of crew to the lunar surface;
- Four times the number of lunar surface crew-hours for sortie missions;
- A Crew Module (CM) with three times the volume of the Apollo Command Module;
- Global lunar surface access with anytime return to the Earth;
- Enabling a permanent human presence at a lunar outpost;
- Demonstrating systems and technologies for human Mars missions;
- Making use of in-situ lunar resources; and
- Providing significantly higher human safety and mission reliability.

In addition to these advantages over the Apollo architecture, the ESAS-selected architecture offers a number of other advantages and features, including:

- The Shuttle-derived launch options were found to be more affordable, safe, and reliable than Evolved Expendable Launch Vehicle (EELV) options;
- The Shuttle-derived approach provides a relatively smooth transition of existing facilities and workforce to ensure lower schedule, cost, and programmatic risks;
- Minimizing the number of launches through development of a heavy-lift Cargo Launch Vehicle (CaLV) improves mission reliability and safety and provides a launcher for future human Mars missions;
- Use of a Reusable Solid Rocket Booster (RSRB-) based Crew Launch Vehicle (CLV) with a top-mounted Crew Exploration Vehicle (CEV) and Launch Abort System (LAS) provides an order-of-magnitude improvement in ascent crew safety over the Space Shuttle;
- Use of an Apollo-style blunt-body capsule was found to be the safest, most affordable, and fastest approach to CEV development;
- Use of the same modular CEV CM and Service Module (SM) for multiple mission applications improves affordability;
- Selection of a land-landing, reusable CEV improves affordability;
- Use of pressure-fed Liquid Oxygen (LOX)/methane propulsion on the CEV SM and Lunar Surface Access Module (LSAM) ascent stage enables In-Situ Resource Utilization (ISRU) for lunar and Mars applications and improves the safety of the LSAM; and
- Selection of the “1.5-launch” Earth Orbit Rendezvous–Lunar Orbit Rendezvous (EOR–LOR) lunar mission mode offers the safest and most affordable option for returning humans to the Moon.

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17. Acronyms and Abbreviations

°C . . . . Degrees Celsius

°F. . . . . Degrees Fahrenheit

°R . . . . Degrees Rankine

24/7 . . . . 24 Hours/day, 7 days/week

2–D . . . . Two-Dimensional

3C . . . . Command, Control, and Checkout

3–D . . . . Three-Dimensional

3–DOF . . . . . . Three-Degrees of Freedom

6-DOF . . . . Six-Degrees of Freedom

A&E . . . . Architectural and Engineering

ACEIT. . . . . . . Automated Cost-Estimating Integrated Tools

ACES . . . . Acceptance and Checkout Evaluation System

ACP. . . . . Analytical Consistency Plan

ACRN . . . . Assured Crew Return Vehicle

ACRV . . . . Assured Crew Return Vehicle

AD. . . . . Analog to Digital

ADBS . . . . Advanced Docking Berthing System

ADRA. . . . . . . Atlantic Downrange Recovery Area

AEDC . . . . Arnold Engineering Development Center

AEG . . . . Apollo Entry Guidance

AETB . . . . Alumina Enhanced Thermal Barrier

AFB. . . . . Air Force Base

AFE. . . . . Aero-assist Flight Experiment

AFPG . . . . Apollo Final Phase Guidance

AFRSI. . . . . . . Advanced Flexible Reusable Surface Insulation

AFV . . . . Anti-Flood Valve

AIAA . . . . American Institute of Aeronautics and Astronautics

AL. . . . . Aluminum

ALARA . . . . . As Low As Reasonably Achievable

AL-Li . . . . Aluminum-Lithium

ALS. . . . . Advanced Launch System

ALTV . . . . Approach and Landing Test Vehicle

AMS . . . . Alpha Magnetic Spectrometer

AMSAA . . . . . Army Material System Analysis Activity

AOA . . . . Analysis of Alternatives

AOD . . . . Aircraft Operations Division

APAS . . . . Androgynous Peripheral Attachment System

APS. . . . . Auxiliary Propulsion System

APU . . . . Auxiliary Power Unit

APU . . . . Auxiliary Propulsion Unit

AR&D. . . . . . . Automated Rendezvous and Docking

ARC . . . . Ames Research Center

ARF . . . . Assembly/Remanufacturing Facility

ASE. . . . . Airborne Support Equipment

ASI . . . . Augmented Space Igniter

ASTWG . . . . . Advanced Spaceport Technology Working Group

ASTP. . . . . Advanced Space Transportation Program

AT . . . . Alternate Turbopump

ATCO . . . . Ambient Temperature Catalytic Oxidation

ATCS . . . . Active Thermal Control System

ATO . . . . Abort-To-Orbit

ATP. . . . . Authority to Proceed

ATS. . . . . Access to Space

ATV . . . . Automated Transfer Vehicles

ATV . . . . All Terrain Vehicle

B . . . . Billion(s)

BEN . . . . Benign

BFO. . . . . Blood-Forming Organs

BFS . . . . Backup Flight Software

BGN . . . . Benign Failure

BLS . . . . Bureau of Labor Statistics

BMI. . . . . Bismaleimide

BOGS . . . . Blade Outer Gas Seal

BPR. . . . . Business Process Reengineering

BRCU . . . . Booster Remote Control Unit

BSM . . . . Booster Separation Motor

Btu. . . . . British Thermal Unit

C . . . . Carbon

C&C . . . . Command and Control

C&W . . . . Caution and Warning

C.I.. . . . . Confidence Interval

C2H6 . . . . Propane

C_3. . . . . Earth Departure Energy

CAD . . . . Computer-Aided Design

CaLV . . . . Cargo Launch Vehicle

CAM . . . . Computer-Aided Manufacturing

CBC . . . . Common Booster Core

CBM . . . . Common Berthing Mechanism

CBT. . . . . Computer-Based Training

CCAFS . . . . . . Cape Canaveral Air Force Station

CCB . . . . Common Core Booster

CCDH . . . . Command, Control, and Data Handling

CDF. . . . . Cumulative Distribution Function

CDR . . . . Critical Design Review

CDV . . . . Cargo Delivery Vehicle

CE&R . . . . Concept Exploration and Refinement

CEEF. . . . . Crew Escape Effectiveness Factor

CER. . . . . Cost Estimating Relationships

CEV . . . . Crew Exploration Vehicle

CFD. . . . . Computational Fluid Dynamics

CFF . . . . Catastrophic Failure Fraction

CFM . . . . Cryogenic Fluid Management

CFO. . . . . Chief Financial Officer

cg or CG . . . . . Center of Gravity

cGy-Eq . . . . . . Centigray Equivalent

CH4 . . . . Methane

CHeCS . . . . . . Crew Health Care System

CHMO . . . . . . Chief Health and Medical Officer

CIL . . . . Critical Items List

CIR . . . . Combustion Integrated Rack

CL . . . . Confidence Level

ClF5. . . . . Chlorine Pentafloride

CLV . . . . Crew Launch Vehicle

CM . . . . Crew Module

cm . . . . Centimeters

CMG . . . . Control Moment Gyroscope

CMRS . . . . Carbon Dioxide (CO2) and Moisture Removal System

CO2 . . . . Carbon Dioxide

COAS . . . . Crew Optical Alignment Sight

Comm . . . . Communication

CONUS . . . . . . Continental United States

CR . . . . Change Request

CRAC . . . . Constant Radius Access Circle

Crit-1 . . . . Criticality 1

CRV . . . . Crew Return Vehicle

CSM . . . . Command and Service Module

CTB. . . . . Cargo Transfer Bag

CTS . . . . Crew Transportation System

CWI. . . . . Combustion Wave Ignition

CY . . . . Calendar Year

DARE . . . . Dynamic Abort Risk Evaluator model

DAV . . . . Descent/Ascent Vehicle

DBFRSC . . . . . Dual Preburner Fuel Rich Staged Combustion

DCF. . . . . Delayed Catastrophic Failure

DCS. . . . . Decompression Sickness

DC–X . . . . Delta Clipper Experimental Program

DDCmplx . . . . Design and Development Complexity

DDInher . . . . . Design and Development Inheritance

DDT&E. . . . . . Design, Development, Test, and Evaluation

deg. . . . . Degrees

Demo. . . . . Demonstration

DKR . . . . Detra-Kemp-Riddell

DoD. . . . . Department of Defense

DoE . . . . Department of Energy

DOF . . . . Degrees of Freedom

DoT . . . . Department of Transportation

DPT. . . . . Decadel Planning Team

DRM . . . . Design Reference Mission

DSN . . . . Deep Space Network

DSS . . . . Decelerator System Simulation

DV. . . . . Delta Velocity (i.e., change in velocity)

EAC. . . . . Estimates at Completion

EAFB . . . . Edwards Air Force Base

EAP. . . . . Expert Assessment Panel

EAS. . . . . Engine Air Start

ECLS. . . . . Environmental Control and Life Support

ECLSS . . . . . . Environmental Control and Life Support System

ECU . . . . Engine Control Unit

EDL. . . . . Entry Descent and Landing

EDLS. . . . . Entry, Descent, and Landing System

EDS. . . . . Earth Departure Stage

EDV . . . . EELV-Derived Vehicle

EEE . . . . Electrical and Electronic Engineering

EELV. . . . . Evolved Expendable Launch Vehicle

EF . . . . Error Factor

EHA . . . . Electro-Hydrostatic Actuator

EI. . . . . Entry Interface

EIRA. . . . . ESAS Initial Reference Architecture

ELR. . . . . Excess Lifetime Risk

ELV . . . . Expendable Launch Vehicle

EMA . . . . Electro-Mechanical Actuators

EMU . . . . Extra-vehicular Maneuvering/Mobility Unit

EO . . . . Engine-Out

EOI . . . . Earth-Orbit Insertion

EOR. . . . . Earth Orbit Rendezvous

ERT . . . . Expedition Readiness Training

ESA . . . . Exploration Systems Architecture

ESAS. . . . . Exploration Systems Architecture Study

ESD . . . . Event Sequence Diagram

ESMD . . . . Exploration Systems Mission Directorate

ESRT. . . . . Exploration Systems Research and Technology

eSSME . . . . . . Expendable Space Shuttle Main Engine

ET . . . . External Tank

ETO. . . . . Earth-to-Orbit

EtOH . . . . Ethanol

EVA. . . . . Extra-Vehicular Activity

FAR. . . . . Federal Acquisition Regulation

FC . . . . Flight Computer

FCE . . . . Flight Crew Equipment

FDIR/R . . . . . . Fault Detection, Isolation, and Recovery/Reconfiguration

FES . . . . Fluid Evaporator System

FEV . . . . Flash Evaporator System

FIR . . . . Fluids Integrated Rack

FIRST . . . . Flight-Oriented Integrated Reliability and Safety Tool

FIV . . . . Fuel Isolation Valve

Flts . . . . Flights

FLUINT . . . . . Fluid Integrator

FM. . . . . Failure Mode

FMC . . . . Flight Medicine Clinic

FMEA . . . . Failure Modes and Effects Analysis

FMHR. . . . . . . Free Molecular Heating Rate

FOM . . . . Figure of Merit

FORP . . . . Fuel-Oxidizer Reaction Product

FRAM. . . . . . . Flight-Releasable Attachment Mechanism

FRGF. . . . . Flight-Releasable Grapple Fixture

FRSC. . . . . Fuel-Rich Staged Combustion

FRSI . . . . Flexible Reusable Surface Insulation

FS . . . . Factor of Safety

FSE . . . . Flight Support Equipment

FSPS . . . . Fission Surface Power System

FSS . . . . Fixed Service Structure

ft . . . . Feet

ft/s . . . . Feet Per Second

FTA . . . . Flight Test Article

FTINU . . . . . . Fault Tolerant Inertial Navigation Unit

FTP . . . . Fuel Turbopump

FTS . . . . Flight Termination System

FU . . . . Flight Unit

FY . . . . Fiscal Year

g. . . . . Acceleration due to gravity (32 ft/sec)

G&A . . . . General and Administrative

g/cm^2 . . . . Grams Per Square Centimeter

GaAs . . . . Gallium Arsenide

GCH4 . . . . Gaseous Methane

GCR . . . . Galactic Cosmic Ray

GDMS. . . . . . . Ground Data Management System

GEM . . . . Graphite Epoxy Motor

GEO . . . . Geosynchronous Earth Orbit

GFE. . . . . Government-Furnished Equipment

GG. . . . . Gas Generator

GHe. . . . . Gaseous Helium

GLOW. . . . . . . Gross Liftoff Weight

GN&C. . . . . . . Guidance, Navigation, and Control

GN2. . . . . Gaseous Nitrogen

GOX . . . . Gaseous Oxygen

GPS. . . . . Global Positioning System

GR&A. . . . . . . Ground Rules and Assumptions

GRAM . . . . . . Global Reference Atmospheric Model

GRC . . . . Glenn Research Center

GSE. . . . . Government-Supplied Equipment

GSE. . . . . Ground Support Equipment

GSFC–FAA. . . Goddard Space Flight Center – Federal Aviation Administration

GT. . . . . Gemini-Titan

GTO . . . . Geosynchronous Transfer Orbit

H2 . . . . Hydrogen

H2O. . . . . Water

H2O2. . . . . Hydrogen Peroxide

HAC . . . . Heading Alignment Circle

HAN . . . . Hydroxyl Ammonium Nitrate

HD. . . . . Hazard Division

HDPE . . . . High-Density Polyethylene Shielding

He . . . . Helium

HEO . . . . High Earth Orbit

HEX . . . . Heat Exchanger

HHFO . . . . Habitability and Human Factors Office

HIP . . . . Hot Isostatic Press

HLLV . . . . Heavy-Lift Launch Vehicle

HLR . . . . Human Lunar Return

HLV. . . . . Heavy-Lift Vehicle

HPFTP . . . . . . High-Pressure Fuel Turbopump

HPFTP/AT . . . High-Pressure Fuel Turbopump/Alternate Turbopump

HPOTP/AT . . . High-Pressure Oxidizer Turbopump/Alternate Turbopump

HPU . . . . Hydraulic Power Unit

HQ. . . . . Headquarters

HSRT . . . . Human Systems Research and Technology

HST. . . . . Hubble Space Telescope

HTPB . . . . Hydroxyl Terminated Poly-Butadiene

HTV . . . . H-11 Transfer Vehicle

HV. . . . . Habitable Volume

HVAC . . . . Heating, Ventilating, and Air Conditioning

HYPAS . . . . . . Hybrid Predictive Aerobraking Scheme

I&T . . . . Integration and Test

I/F . . . . Interface

IA&C . . . . Integration, Assembly, and Checkout

ICD . . . . Interface Control Document

ICF . . . . Instantaneous Catastrophic Failure

IDT . . . . Integrated Discipline Team

ILC . . . . In-Line Configuration

ILC . . . . Integrated Logistics Concepts

IMLEO . . . . . . Initial Mass in Low Earth Orbit

IMP . . . . Integrated Mission Program

IMS . . . . Integrated Master Schedule

IMU . . . . Inertial Measuring Unit

in . . . . Inch(es)

INS . . . . Inertial Navigation System

INTROS . . . . . Integrated Rocket Sizing Program

IOC . . . . Initial Operational Capability

IPAO . . . . Independent Program Assessment Office

IPD . . . . Integrated Powerhead Demonstrator

IR . . . . Infrared

IRD . . . . Interface Requirements Document

IS. . . . . Interstages

ISHM . . . . Integrated Systems Health Management

Isp . . . . Specific Impulse

ISPP. . . . . In-Situ Produced Propellant

ISRU . . . . In-Situ Resource Utilization

ISS. . . . . International Space Station

IT. . . . . Information Technology

ITA . . . . Internal Task Agreement

ITAR . . . . International Traffic and Arms Regulations

IVA . . . . Intra-Vehicular Activity

IVHM . . . . Integrated Vehicle Health Management

JOFOC . . . . . . Justification for Other Than Full and Open Competition

JSC . . . . Johnson Space Center

JWST . . . . James Webb Space Telescope

K . . . . Thousands

K . . . . Kelvin

kg. . . . . Kilograms

KJ . . . . Kilojoule

klb . . . . Kilopound

klbf . . . . Thousand Pounds Force

km . . . . Kilometers

km/s. . . . . Kilometers Per Second

kN . . . . Kilonewtons

kPa . . . . Kilopascals

KPP . . . . Key Performance Parameters

KSC. . . . . Kennedy Space Center

KSLOC . . . . . . Thousand Software Lines of Codes kWe . . . . Kilowatts Electric

L/D . . . . Lift-to-Drag (ratio)

LAD . . . . Liquid Acquisition Device

LADAR. . . . . . Laser Detection and Ranging

LAN . . . . Local Area Network

LAN . . . . Longitude of Ascending Node

LaRC. . . . . Langley Research Center

LAS. . . . . Launch Abort System

lb . . . . Pound(s)

lbf . . . . Pounds Force

lbm . . . . Pounds Mass

LC . . . . Launch Complex

LCC. . . . . Life Cycle Cost

LCF . . . . Low Cycle Fatigue

LCH4. . . . . Liquid Methane

LCVG . . . . Liquid Cooled Ventilation Garment

LCX . . . . Launch Complex X

LEAG . . . . Lunar Exploration and Analysis Group

LEM . . . . Lunar Excursion Module

LEO. . . . . Low Earth Orbit

LES . . . . Launch Escape System

LExSWG. . . . . Lunar Exploration Science Working Group

LH2. . . . . Liquid Hydrogen

Li. . . . . Lithium

LiDAR . . . . . . Light Detection and Ranging

LIDS . . . . Low-Impact Docking System

LiOH . . . . Lithium Hydroxide

LLO. . . . . Low Lunar Orbit

LLV . . . . Lunar Launch Vehicle

LM . . . . Lunar Module

LNG . . . . Liquid Natural Gas

LOC . . . . Loss of Crew

LOI . . . . Lunar-Orbit Insertion

LOM . . . . Loss of Mission

LOR . . . . Lunar Orbit Rendezvous

LOV . . . . Loss of Vehicle

LOX . . . . Liquid Oxygen

LPFTP/AT . . . . Low-Pressure Fuel Turbopump/Alternate Turbopump

LPOTP/AT . . . Low-Pressure Oxidizer Turbopump/Alternate Turbopump

LPR . . . . Libration Point Rendezvous

LPRE. . . . . Lunar Polar Resource Extractor

LRB. . . . . Liquid Rocket Booster

LRECM . . . . . Liquid Rocket Engine Cost Model

LRO. . . . . Lunar Reconnaissance Orbiter

LRPS. . . . . Lunar Radioisotope Power System

LRU. . . . . Line Replaceable Unit

LRV. . . . . Lunar Roving Vehicle

LSAM . . . . Lunar Surface Access Module

LSC . . . . Launch Services Contractor

LSR . . . . Lunar Surface Rendezvous

LTMCC. . . . . . Large-Throat Main Combustion Chamber

LUT. . . . . Launch Umbilical Tower

LV . . . . Launch Vehicle

LVA . . . . Launch Vehicle Architecture

LVHM. . . . . . . Launch Vehicle Health Management

M. . . . . Million(s)

m . . . . Meters

M&O. . . . . Maintenance and Operations

M&P . . . . Material and Processing

m/s. . . . . Meters Per Second

m^3 . . . . Cubic Meters

MA . . . . Mercury-Atlas

MAF . . . . Michoud Assembly Facility
mbo . . . . Stage Burnout Mass
MCC . . . . Main Combustion Chamber
MCC . . . . Mission Control Center
mdry . . . . Stage Dry Mass
MDU . . . . Master Data Unit
ME . . . . Main Engine
MEA . . . . Maintenance Engineering Analysis
MECO . . . . Main Engine Cutoff
MEIT . . . . Multi-Element Integrated Test
MEOP . . . . Maximum Expected Operating Pressure
MEPAG . . . . . . Mars Exploration Program Analysis Group
MER . . . . Mass Estimating Relationship
MER . . . . Mars Exploration Rover
MET . . . . Mission Elapsed Time
MeVs . . . . Millions of Electron Volts
MFBF . . . . Mean Flights Between Failure
mgross . . . . Stage Gross Liftoff Mass
mgross-veh . . . Vehicle Gross Liftoff Mass
MIS . . . . Management Information System
MIT . . . . Massachusetts Institute of Technology
MLE . . . . Mid-deck Locker Equivalent
MLI . . . . Multilayer Insulation
MLP . . . . Mobile Launch Platform
MLUT . . . . Mobile Launch Umbilical Tower
MLV . . . . Medium-Lift Vehicle
MM . . . . Mission Mode
MMH . . . . Monomethyl Hydrazine
MMOD . . . . . . Micrometeoroid/Orbital Debris
MMSSG . . . . . Moon-Mars Science Linkage Steering Group
MON-3 . . . . . . Multiple Oxides of Nitrogen – 3%

MPK . . . . Mission Peculiar Kit

MPPF . . . . Multi-Purpose Processing Facility

MPS . . . . Main Propulsion System

MPTA . . . . Main Propulsion Test Article

MR . . . . Mercury-Redstone

MS. . . . . Microsoft®

MSFC . . . . Marshall Space Flight Center

MSL . . . . Mars Science Laboratory

MSS . . . . Mobile Service Structure

mT . . . . metric ton(s)

MTBF . . . . Mean Time Between Failure

MTV . . . . Mars Transfer Vehicle

MUSS . . . . Multi-User System and Support

MWe . . . . Megawatts Electric

N . . . . Newtons

N/A . . . . Not Applicable

N2 . . . . Nitrogen

N2H4. . . . . Hydrazine

N2O. . . . . Nitrous Oxide

NAFCOM . . . . NASA and Air Force Cost Model

NAS . . . . National Academy of Sciences

NASA . . . . National Aeronautics and Space Administration

Nav . . . . Navigation

NBL . . . . Neutral Buoyancy Laboratory

NCRP . . . . National Council on Radiation Protection and Measurements

NEO . . . . No Engine-Out

NEPA . . . . National Environmental Policy Act

NExT. . . . . NASA Exploration Team

NGLT . . . . Next Generation Launch Technology

NHRP . . . . Next Hop Resolution Protocol

nmi . . . . nautical mile

NPD . . . . NASA Program Directive

NPG . . . . NASA Procedure and Guideline

NPOE . . . . NPO Energomash

NPR. . . . . NASA Procedural Requirements

NRO . . . . National Reconnaissance Office

NSI . . . . NASA Standard Initiative/Initiator

NTO . . . . Nitrogen Tetroxide

NTP. . . . . Nuclear Thermal Propulsion

NTS. . . . . Nevada Test Sites

NUREG . . . . . Nuclear Regulatory Commission

O&C . . . . Operations and Checkout

O&M. . . . . Operations and Maintenance

O2 . . . . Oxygen

OBC . . . . Other Burden Costs

OCM . . . . Operations Cost Model

OExP. . . . . Office of Exploration

OMB . . . . Office of Management and Budget

OML . . . . Outer Mold Line

OMS . . . . Orbital Maneuvering System

OODA . . . . Observation, Orientation, Decision, Action

Ops . . . . Operations

ORCA . . . . Ordnance Remote Control Assembly

ORSC . . . . Oxygen-Rich Staged Combustion

ORU . . . . Orbital Replacement Unit

OSP . . . . Orbital Space Plane

OTP. . . . . Oxygen Turbopump

P . . . . Pressure Transducer

P(LOC) . . . . . . Probability of Loss of Crew

P(LOM). . . . . . Probability of Loss of Mission

P/C. . . . . Probe and Cone

PA . . . . Pad Abort

PAO . . . . Public Affairs Office

PAS . . . . Primary Ascent System

PAS . . . . Payload Attach System

PBAN . . . . Polybutadiene Acrylonitride

PC . . . . Plane Change

PC . . . . Personal Computer

PCBM . . . . Passive Common Berthing Mechanism

PCM . . . . Pulse Code Modulated

PCM . . . . Pressurized Cargo Mission

PCU. . . . . Power Control Unit

PDR. . . . . Preliminary Design Review

PEG . . . . Powered Explicit Guidance

PEM . . . . Proton Exchange Membrane

PIC . . . . Pyrotechnic Initiator Controller

PICA . . . . Phenolic Impregnated Carbonaceous Ablator

PLF . . . . Payload Fairing

PMA . . . . Pressurized Mating Adapter

PMA . . . . Primary Mating Adapter

PMAD. . . . . . . Power Management and Distribution

PMS. . . . . Propulsion Management System

PNST. . . . . Prrometheus Nuclear Systems Technology

POD . . . . Point of Departure

POST. . . . . Program to Optimize Simulated Trajectories

ppm . . . . Parts Per Million

PRA. . . . . Probabilistic Reliability Assessment

PRC . . . . Productivity Rate Curve

PS . . . . Primary and Secondary

psf . . . . pounds per square foot

psi . . . . Pounds Per Square Inch

psia . . . . Pounds Per Square Inch Absolute

PTA . . . . Prototype Test Article

PV . . . . Photovoltaic

PV . . . . Pressurized Volume

PV/RFC. . . . . . Photovoltaic/Regenerative Fuel Cell

PV/W. . . . . *Formula:* (Pressure · Volume)/Weight

QD. . . . . Quantity Distance

QRAS . . . . Quantitative Risk Assessment System

R&D . . . . Research and Development

R&T . . . . Research and Technology

RAAN. . . . . . . Right Ascension of Ascending Node

RCC. . . . . Reinforced Carbon-Carbon

RCG . . . . Reaction-Cured Glass

RCRS . . . . Regenerative CO2 Removal System

RCS . . . . Reaction Control System

RDU . . . . Remote Data Unit

REID. . . . . Risk of Exposure-Induced Death

RF . . . . Radio Frequency

RFC. . . . . Regenerative Fuel Cell

RFP . . . . Request for Proposals

RLEP. . . . . Robotic Lunar Exploration Program

RMS . . . . Remote Manipulator System

ROM . . . . Rough Order Magnitude

ROW . . . . Risks, Opportunities, and Watches

RP . . . . Rocket Propellant

RP1 . . . . Kerosene

RPCU . . . . Remote Power Control Unit

RPL . . . . Rated Power Level

RPSF . . . . Rotation Processing and Surge Facility

RRF. . . . . Risk Reduction Flight

RRGU . . . . Redundant Rate Gyro Unit

RSRB . . . . Reusable Solid Rocket Booster

RSS . . . . Rotating Service Structure

RTV. . . . . Room Temperature Vulcanized

S&A . . . . Safe and Arm

S&MA. . . . . . . Safety and Mission Assurance

SAIC . . . . Science Applications International Corporation

SARSAT . . . . . Search and Rescue Satellite-aided Tracking

SBIR . . . . Small Business Innovation Research

SCAPE . . . . . . Self-Contained Atmospheric Protection Ensembles

SCAWG. . . . . . Space Communications Architecture Working Group

SCBA . . . . Strip Collar Bonding Approach

SCE . . . . Station Control Electronics

SDILV . . . . Shuttle-Derived In-line Launch Vehicle

SDLV. . . . . Shuttle-Derived Launch Vehicle

SDV. . . . . Shuttle-Derived Vehicle

SE&I . . . . Systems Engineering and Integration

sec . . . . Second(s)

SECO . . . . Second Stage Engine Cutoff

SEER–SEM. . . SEER Software Estimation Model

SEG. . . . . Space Shuttle Entry Guidance

SEI. . . . . Space Exploration Initiative

SFT . . . . Single Fault Toerant

SHAB . . . . Surface Habitat

SiC. . . . . Silicon Carbon

SIGI. . . . . Space Integrated GPS Instrumentation

SII . . . . Saturn Second Stage

SINDA . . . . . . Systems Improved Numerical Differencing Analyser

SIP. . . . . Strain Isolation Pad

SIRCA. . . . . . . Silicon Infused Reusable Ceramic Ablator

SIVB . . . . Saturn V Third Stage

SLF . . . . Shuttle Landing Facility

SLI. . . . . Space Launch Initiative

SLOC . . . . Software Lines of Code

SM. . . . . Service Module

SMAC . . . . Spacecraft Maximum Allowable Concentration

SMD . . . . Science Missions Directorate

SOA. . . . . State-of-the-Art

SOFI . . . . Spray-On Foam Insulation

SOMD. . . . . . . Space Operations Missions Directorate

SOMS . . . . Shuttle Orbiter Medical System

SORT . . . . Simulation and Optimization of Rocket Trajectories

SPA . . . . Spacecraft/Payload Adapter

SPA . . . . South Pole-Aitken (basin on the Moon)

SPACE. . . . . . . Screening Program for Architecture Capability Evaluation

SPASE. . . . . . . Standardized Propulsive Skip Entry

SPE . . . . Solar Particle Event

SPF . . . . Single Point Failure

SPS . . . . Service Propulsion System

SPST . . . . Space Propulsion Synergy Team

SRB. . . . . Solid Rocket Booster

SRM . . . . Solid Rocket Motor

SRR. . . . . System Requirements Review

SSC . . . . Stennis Space Center

SS–LV. . . . . . . Single-Stick Launch Vehicle

SSME . . . . Space Shuttle Main Engine

SSP . . . . Space Shuttle Program

SSPF . . . . Space Station Processing Facility

SSRMS . . . . . . Space Station Remote Manipulator System

SSTO. . . . . Single-Stage-to-Orbit

STA . . . . Static Test Article

STA . . . . Structural Test Article

STD. . . . . Standard

stg . . . . Stage

STH. . . . . System Test Hardware

STO . . . . System Test Operations

STPPO. . . . . . . Space Transportation Programs and Projects Office

STS . . . . Space Transportation System

STTR. . . . . Small Business Technology Transfer

SUF . . . . Startup Failure

SV . . . . Space Vehicle

SVMF . . . . Space Vehicle Mockup Facility

SW. . . . . Software

T/W . . . . Thrust-to-Weight (ratio)

TAL. . . . . Trans-Atlantic Abort Landing

TBD . . . . To Be Determined

TBR. . . . . To Be Resolved

TC . . . . Thermocouple

TCA. . . . . Thrust Chamber Assembly

TCS . . . . Thermal Control System

TDRSS . . . . . . Tracking and Data Relay Satellite System

TEI . . . . Trans-Earth Injection

TEP/ODE . . . . Thermal Equilibrium Program/One Dimensional Equilibrium

TFU. . . . . Theoretical First Unit

TIFF . . . . Throttle-Insensitivity Failure Fraction

TLI . . . . Trans-Lunar Injection

TNT . . . . Trinitrotoluene, i.e., CH3C6H2(NO2)3

TPS . . . . Thermal Protection System

TPSX. . . . . Thermal Protection System Expert

TRL. . . . . Technology Readiness Level

Tt 1/2. . . . . Test Type 1, Test Type 2

TV . . . . Television

TVC. . . . . Thrust Vector Control

TVS. . . . . Thermal Vacuum Stability

U.S. . . . . United States

UCA . . . . Undefinitized Contract Actions

UCM . . . . Unpressurized Cargo Mission

UHF . . . . Ultrahigh Frequency

URCU . . . . Upper Stage Remote Control Unit

US . . . . Upper Stage

USAF . . . . United States Air Force

USOS . . . . United States On-orbit Segment

UV. . . . . Ultra-Violet

VAB . . . . Vehicle Assembly Building

VDC . . . . Volts Direct Current

VIPA . . . . Vehicle Integrated Performance Analysis Team

VITO. . . . . Vehicle Integration and Test Office

VITT . . . . Vehicle Integrated Test Team

VP . . . . Vacuum Perigee

VPF . . . . Vertical Processing Facility

VR. . . . . Virtual Reality

WAN . . . . Wide Area Network

WBS . . . . Work Breakdown Structure

WCS . . . . Waste Collection System

ΔV . . . . Delta Velocity, i.e., change in velocity