CHAPTER 3

DESCRIPTION OF APOLLO 13 SPACE VEHICLE AND MISSION SUMMARY

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Discussion in this chapter is broken into two parts. Part 1 is designed to acquaint the reader with the flight hardware and with the mission monitoring, support, and control functions and capabilities. Part 2 describes the Apollo 13 mission and gives a mission sequence of events summary.

PART 1 APOLLO/SATURN V SPACE VEHICLE

The primary flight hardware of the Apollo Program consists of the Saturn V launch vehicle and Apollo spacecraft (fig. 3-1). Collectively, they are designated the Apollo/Saturn V space vehicle (SV). Selected major systems and subsystems of the space vehicle may be summarized as follows.

SATURN V LAUNCH VEHICLE

The Saturn V launch vehicle (LV) is designed to boost up to 300,000 pounds into a 105-nautical mile earth orbit and to provide for lunar payloads of over 100,000 pounds. The Saturn V LV consists of three propulsive stages (S-IC, S-II, S-IVB), two interstages, and an instrument unit (IU).

S-IC Stage

The S-IC stage (fig. 3-2) is a large cylindrical booster, 138 feet long and 33 feet in diameter, powered by five liquid propellant F-1 rocket engines. These engines develop a nominal sea level thrust total of approximately 7,650,000 pounds. The stage dry weight is approximately 288,000 pounds and the total loaded stage weight is approximately 5,031,500 pounds. The S-IC stage interfaces structurally and electrically with the S-II stage. It also interfaces structurally, electrically, and pneumatically with ground support equipment (GSE) through two umbilical service arms, three tail service masts, and certain electronic systems by antennas. The S-IC stage is instrumented for operational measurements or signals which are transmitted by its independent telemetry system.

S-II Stage

The S-II stage (fig. 3-3) is a large cylindrical booster, 81.5 feet long and 33 feet in diameter, powered by five liquid propellant J-2 rocket engines which develop a nominal vacuum thrust of 230,000 pounds each for a total of 1,150,000 pounds. Dry weight of the S-II stage is approximately 78,050 pounds. The stage approximate loaded gross weight is 1,075,000 pounds. The S-IC/S-II interstage weighs 10,460 pounds. The S-II stage is instrumented for operational and research and development measurements which are transmitted by its independent telemetry system. The S-II stage has structural and electrical interfaces with the S-IC and S-IVB stages, and electric, pneumatic, and fluid interfaces with GSE through its umbilicals and antennas.

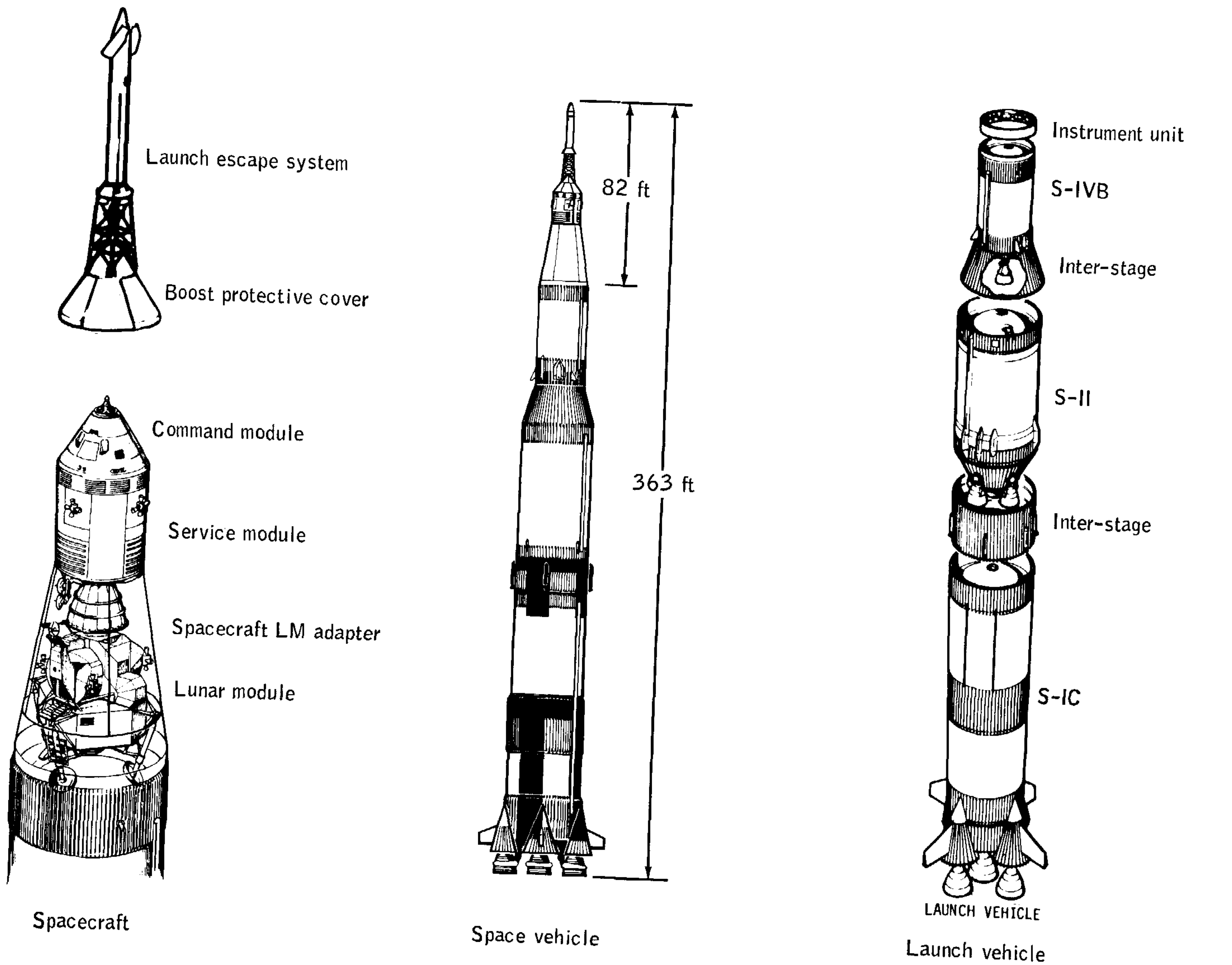


Figure 3-1.- Apollo/Saturn V space vehicle.

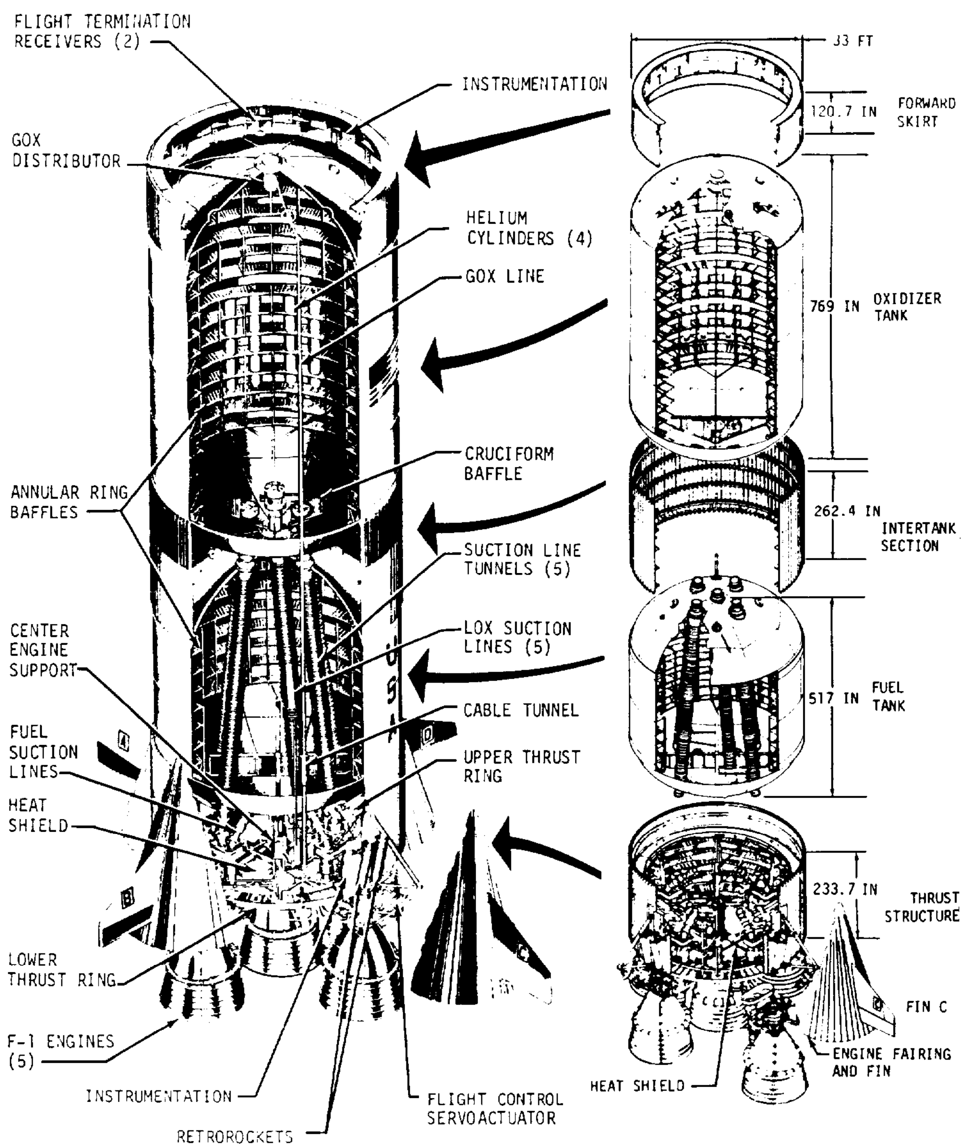


Figure 3-2.- S-IC stage.

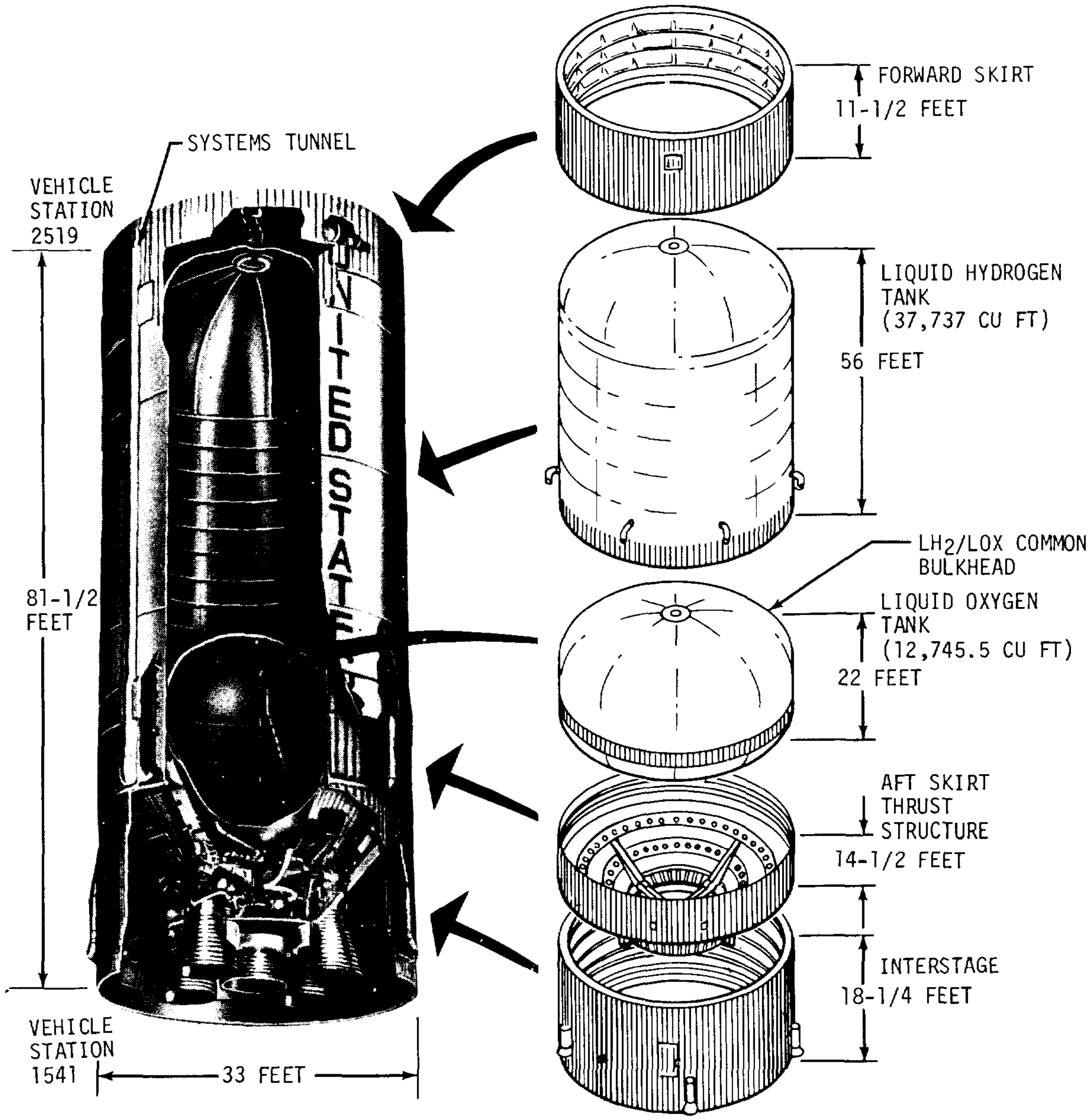


Figure 3-3.- S-II stage.

S-IVB Stage

The S-IVB stage (fig. 3-4) is a large cylindrical booster 59 feet long and 21.6 feet in diameter, powered by one J-2 engine. The S-IVB stage is capable of multiple engine starts. Engine thrust is 203,000 pounds. This stage is also unique in that it has an attitude control capability independent of its main engine. Dry weight of the stage is 25,050 pounds. The launch weight of the stage is 261,700 pounds. The interstage weight of 8100 pounds is not included in the stated weights. The stage is instrumented for functional measurements or signals which are transmitted by its independent telemetry system.

The high performance J-2 engine as installed in the S-IVB stage has a multiple start capability. The S-IVB J-2 engine is scheduled to produce a thrust of 203,000 pounds during its first burn to earth orbit and a thrust of 178,000 pounds (mixture mass ratio of 4.5:1) during the first 100 seconds of translunar injection. The remaining translunar injection acceleration is provided at a thrust level of 203,000 pounds (mixture mass ratio of 5.0:1). The engine valves are controlled by a pneumatic system powered by gaseous helium which is stored in a sphere inside a start bottle. An electrical control system that uses solid stage logic elements is used to sequence the start and shutdown operations of the engine.

Instrument Unit

The Saturn V launch vehicle is guided from its launch pad into earth orbit primarily by navigation, guidance, and control equipment located in the instrument unit (IU). The instrument unit is a cylindrical structure 21.6 feet in diameter and 3 feet high installed on top of the S-IVB stage. The unit weighs 4310 pounds and contains measurements and telemetry, command communications, tracking, and emergency detection system components along with supporting electrical power and the environmental control system.

APOLLO SPACECRAFT

The Apollo spacecraft (S/C) is designed to support three men in space for periods up to 2 weeks, docking in space, landing on and returning from the lunar surface, and safely entering the earth's atmosphere. The Apollo S/C consists of the spacecraft-to-LM adapter (SLA), the service module (SM), the command module (CM), the launch escape system (LES), and the lunar module (LM). The CM and SM as a unit are referred to as the command and service module (CSM).

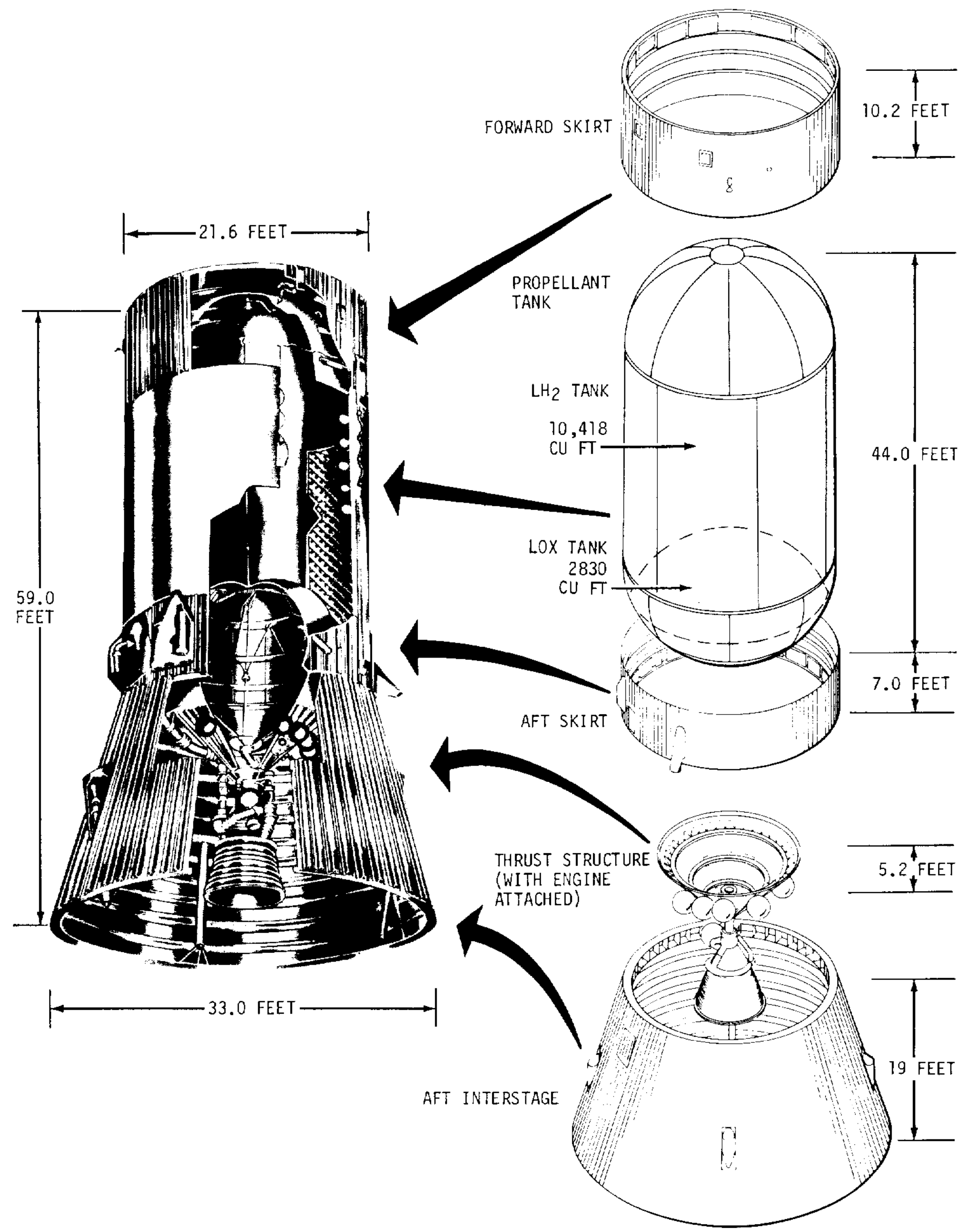


Figure 3-4.- S-IVB stage.

Spacecraft-to-LM Adapter

The SLA (fig. 3-5) is a conical structure which provides a structural load path between the LV and SM and also supports the LM. Aerodynamically, the SLA smoothly encloses the irregularly shaped LM and transitions the space vehicle diameter from that of the upper stage of the LV to that of the SM. The SLA also encloses the nozzle of the SM engine and the high gain antenna.

Spring thrusters are used to separate the LM from the SLA. After the CSM has docked with the LM, mild charges are fired to release the four adapters which secure the LM in the SLA. Simultaneously, four spring thrusters mounted on the lower (fixed) SLA panels push against the LM landing gear truss assembly to separate the spacecraft from the launch vehicle.

Service Module

The service module (SM)(fig. 3-6) provides the main spacecraft propulsion and maneuvering capability during a mission. The SM provides most of the spacecraft consumables (oxygen, water, propellant, and hydrogen) and supplements environmental, electrical power, and propulsion requirements of the CM. The SM remains attached to the CM until it is jettisoned just before CM atmospheric entry.

Structure.- The basic structural components are forward and aft (upper and lower) bulkheads, six radial beams, four sector honeycomb panels, four reaction control system honeycomb panels, aft heat shield, and a fairing. The forward and aft bulkheads cover the top and bottom of the SM. Radial beam trusses extending above the forward bulkhead support and secure the CM. The radial beams are made of solid aluminum alloy which has been machined and chem-milled to thicknesses varying between 2 inches and 0.018 inch. Three of these beams have compression pads and the other three have shear-compression pads and tension ties. Explosive charges in the center sections of these tension ties are used to separate the CM from the SM.

An aft heat shield surrounds the service propulsion engine to protect the SM from the engine's heat during thrusting. The gap between the CM and the forward bulkhead of the SM is closed off with a fairing which is composed of eight electrical power system radiators alternated with eight aluminum honeycomb panels. The sector and reaction control system panels are 1 inch thick and are made of aluminum honeycomb core between two aluminum face sheets. The sector panels are bolted to the radial beams. Radiators used to dissipate heat from the environmental control subsystem are bonded to the sector panels on opposite sides of the SM. These radiators are each about 30 square feet in area.

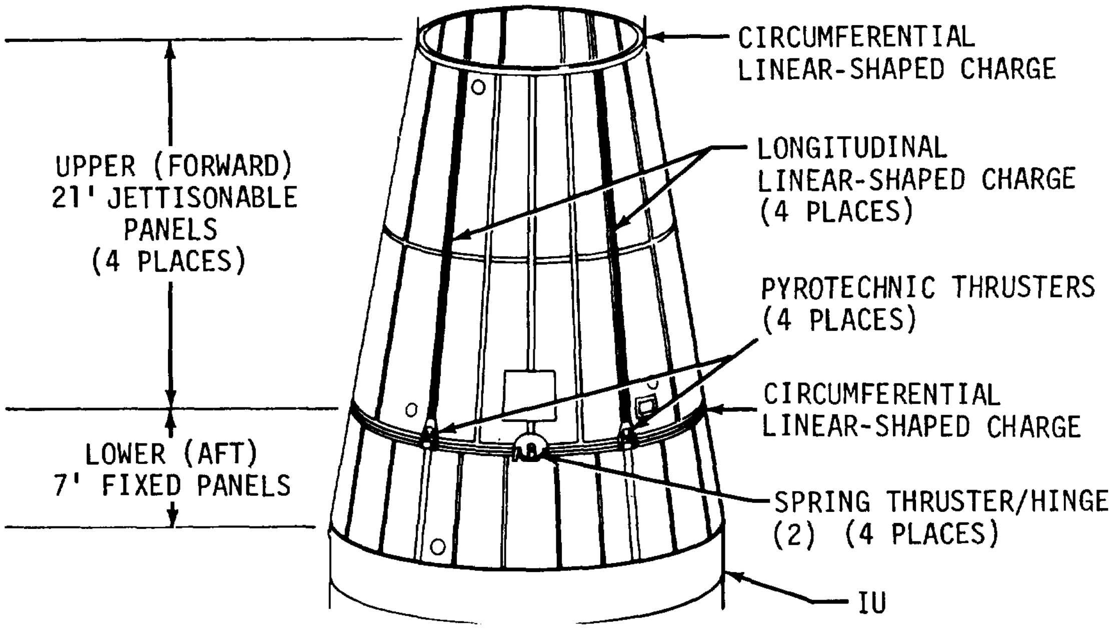
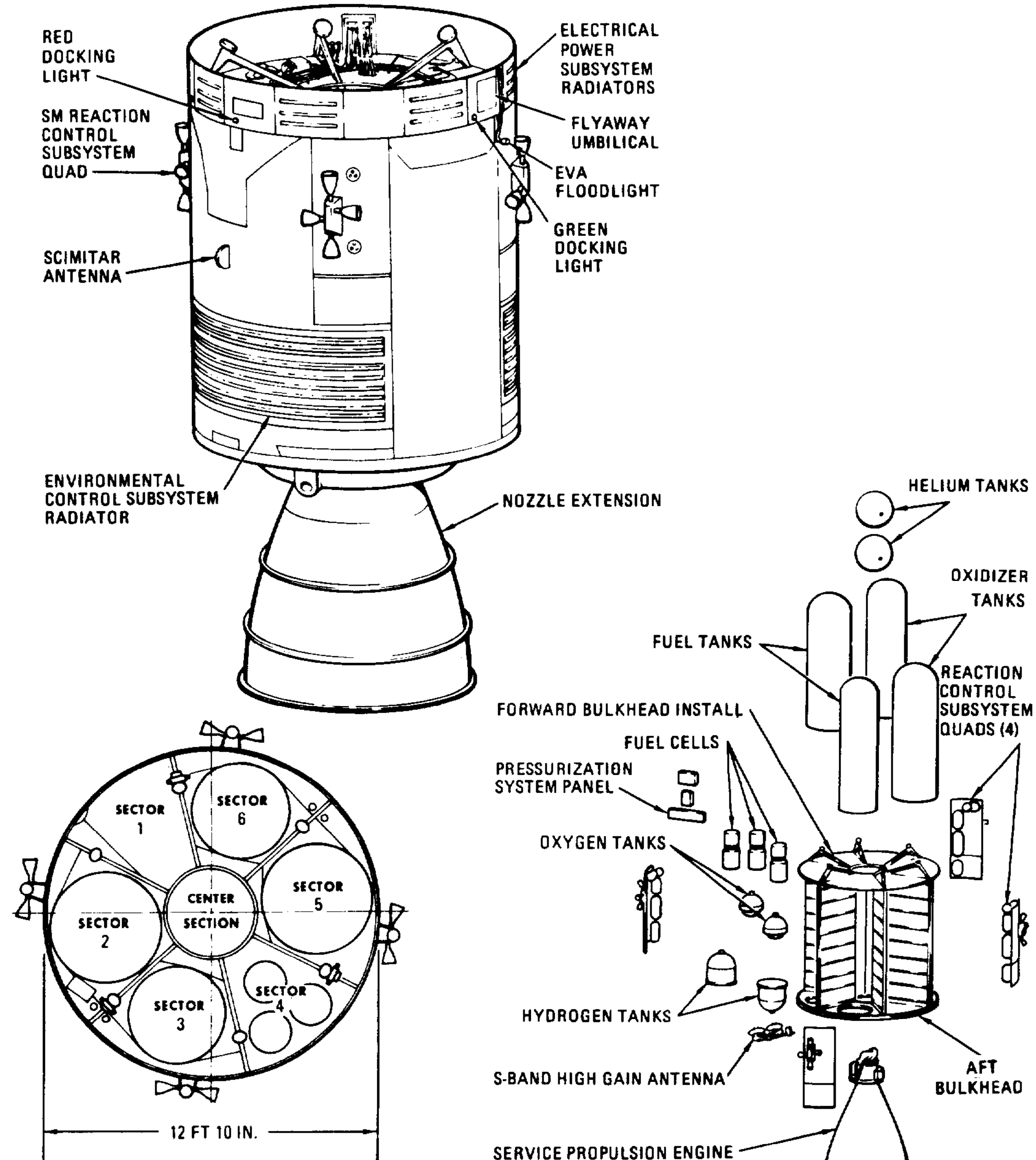


Figure 3-5.- Spacecraft-to-LM adapter.



SECTOR 2 } SERVICE PROPULSION SUBSYSTEM
SECTOR 3 } OXIDIZER TANKS
SECTOR 4 OXYGEN TANKS, HYDROGEN TANKS, FUEL CELLS
SECTOR 5 } SERVICE PROPULSION SUBSYSTEM
SECTOR 6 } FUEL TANKS
CENTER SECTION - SERVICE PROPULSION ENGINE AND HELIUM TANKS

Figure 3-6.- Service module.

The SM interior is divided into six sectors, or bays, and a center section. Sector one is currently void. It is available for installation of scientific or additional equipment should the need arise. Sector two has part of a space radiator and a reaction control system (RCS) engine quad (module) on its exterior panel and contains the service propulsion system (SPS) oxidizer sump tank. This tank is the larger of the two tanks that hold the oxidizer for the SPS engine. Sector three has the rest of the space radiator and another RCS engine quad on its exterior panel and contains the oxidizer storage tank. This tank is the second of two SPS oxidizer tanks and feeds the oxidizer sump tank in sector two. Sector four contains most of the electrical power generating equipment. It contains three fuel cells, two cryogenic oxygen and two cryogenic hydrogen tanks, and a power control relay box. The cryogenic tanks supply oxygen to the environmental control subsystem and oxygen and hydrogen to the fuel cells. Sector five has part of an environmental control radiator and an RCS engine quad on the exterior panel and contains the SPS engine fuel sump tank. This tank feeds the engine and is also connected by feed lines to the storage tank in sector six. Sector six has the rest of the environmental control radiator and an RCS engine quad on its exterior and contains the SPS engine fuel storage tank which feeds the fuel sump tank in sector five. The center section contains two helium tanks and the SPS engine. The tanks are used to provide helium pressurant for the SPS propellant tanks.

Propulsion.- Main spacecraft propulsion is provided by the 20500-pound thrust SPS. The SPS engine is a restartable, non-throttleable engine which uses nitrogen tetroxide (N_2O_4) as an oxidizer and a 50-50 mixture of hydrazine and unsymmetrical-dimethylhydrazine (UDMH) as fuel. (These propellants are hypergolic, i.e., they burn spontaneously when combined without need for an igniter.) This engine is used for major velocity changes during the mission, such as midcourse corrections, lunar orbit insertion, transearth injection, and CSM aborts. The SPS engine responds to automatic firing commands from the guidance and navigation system or to commands from manual controls. The engine assembly is gimbal-mounted to allow engine thrust-vector alignment with the spacecraft center of mass to preclude tumbling. Thrust-vector alignment control is maintained by the crew. The SM RCS provides for maneuvering about and along three axes.

Additional SM systems.- In addition to the systems already described, the SM has communication antennas, umbilical connections, and several exterior mounted lights. The four antennas on the outside of the SM are the steerable S-band high-gain antenna, mounted on the aft bulkhead; two VHF omnidirectional antennas, mounted on opposite sides of the module near the top; and the rendezvous radar transponder antenna, mounted in the SM fairing.

Seven lights are mounted in the aluminum panels of the fairing. Four lights (one red, one green, and two amber) are used to aid the astronauts in docking: one is a floodlight which can be turned on to give astronauts visibility during extravehicular activities, one is a flashing beacon used to aid in rendezvous, and one is a spotlight used in rendezvous from 500 feet to docking with the LM.

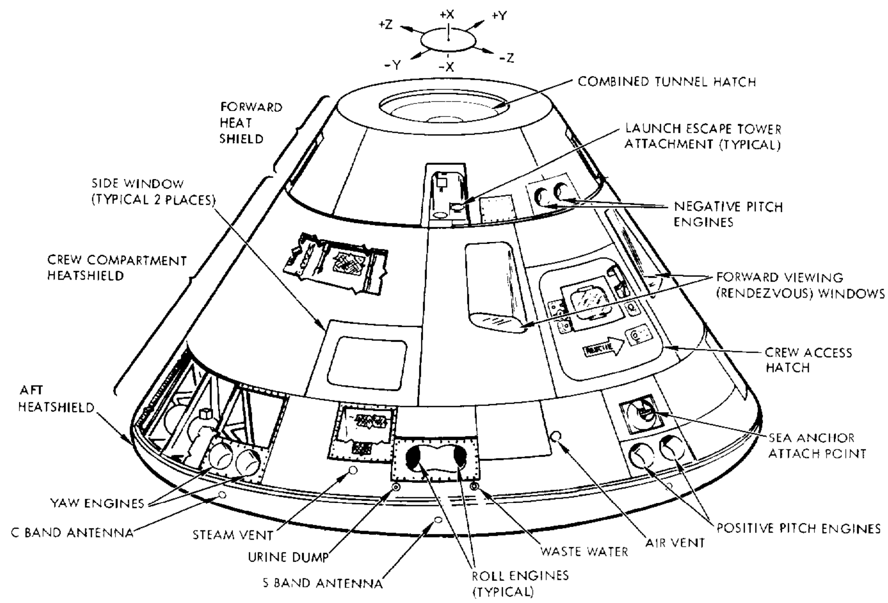
SM/CM separation.- Separation of the SM from the CM occurs shortly before entry. The sequence of events during separation is controlled automatically by two redundant service module jettison controllers (SMJC) located on the forward bulkhead of the SM.

Command Module

The command module (CM) (fig. 3-7) serves as the command, control, and communications center for most of the mission. Supplemented by the SM, it provides all life support elements for three crewmen in the mission environments and for their safe return to the earth's surface. It is capable of attitude control about three axes and some lateral lift translation at high velocities in earth atmosphere. It also permits LM attachment, CM/LM ingress and egress, and serves as a buoyant vessel in open ocean.

Structure.- The CM consists of two basic structures joined together: the inner structure (pressure shell) and the outer structure (heat shield). The inner structure, the pressurized crew compartment, is made of aluminum sandwich construction consisting of a welded aluminum inner skin, bonded aluminum honeycomb core, and outer face sheet. The outer structure is basically a heat shield and is made of stainless steel-brazed honeycomb brazed between steel alloy face sheets. Parts of the area between the inner and outer sheets are filled with a layer of fibrous insulation as additional heat protection.

Display and controls.- The main display console (MDC) (fig. 3-8) has been arranged to provide for the expected duties of crew members. These duties fall into the categories of Commander, CM Pilot, and LM Pilot, occupying the left, center, and right couches, respectively. The CM Pilot also acts as the principal navigator. All controls have been designed so they can be operated by astronauts wearing gloves. The controls are predominantly of four basic types: toggle switches, rotary switches with click-stops, thumb-wheels, and push buttons. Critical switches are guarded so that they cannot be thrown inadvertently. In addition, some critical controls have locks that must be released before they can be operated.



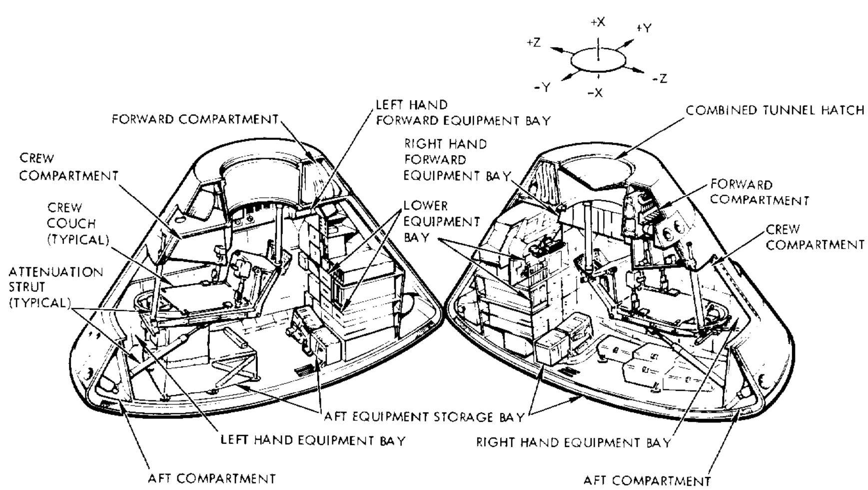


Figure 3-7.- Command module.

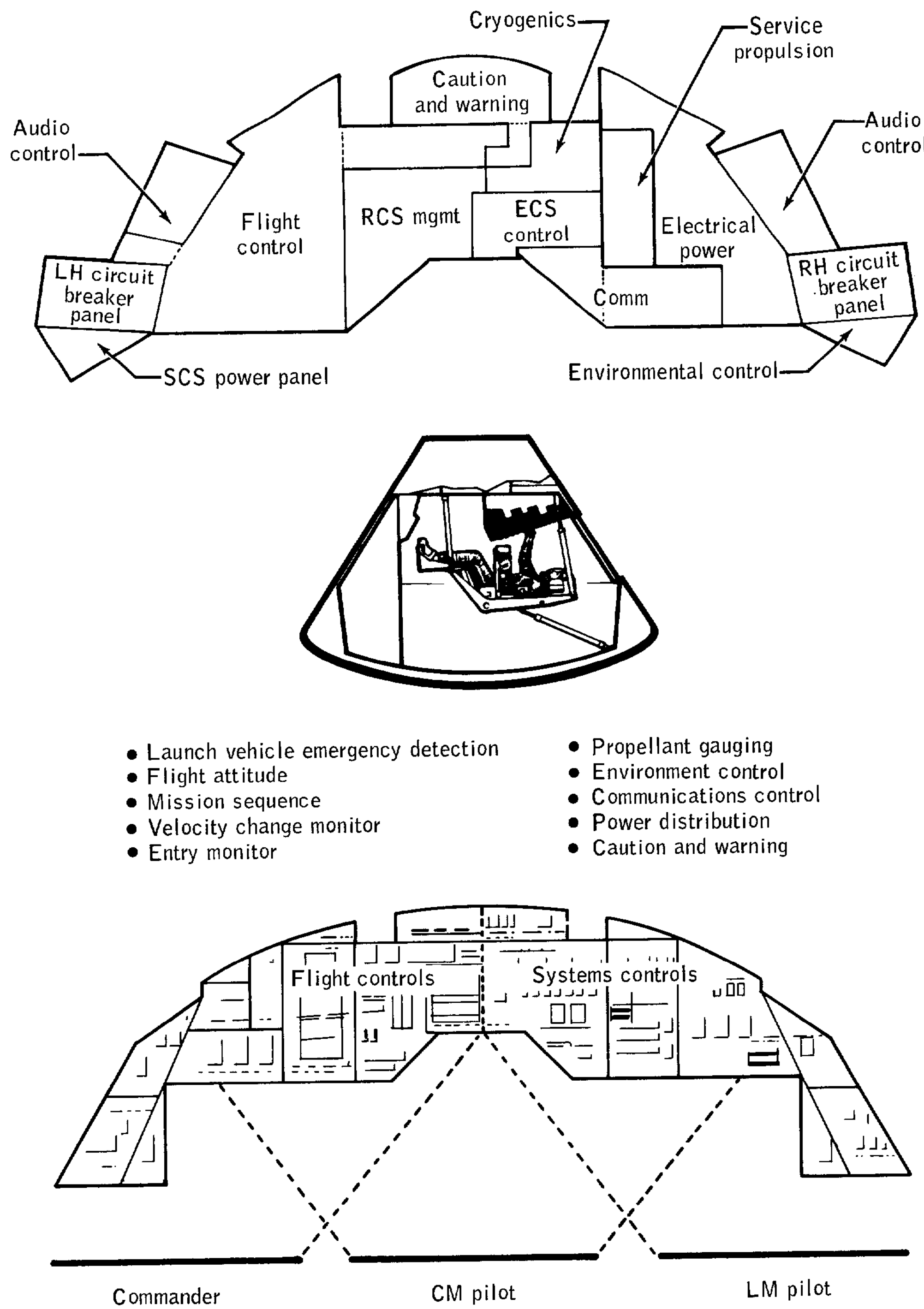


Figure 3-8.- CM main display console.

Flight controls are located on the left center and left side of the MDC, opposite the Commander. These include controls for such subsystems as stabilization and control, propulsion, crew safety, earth landing, and emergency detection. One of two guidance and navigation computer panels also is located here, as are velocity, attitude, and altitude indicators.

The CM Pilot faces the center of the console, and thus can reach many of the flight controls, as well as the system controls on the right side of the console. Displays and controls directly opposite him include reaction control, propellant management, caution and warning, environmental control, and cryogenic storage systems. The rotation and translation controllers used for attitude, thrust vector, and translation maneuvers are located on the arms of two crew couches. In addition, a rotation controller can be mounted at the navigation position in the lower equipment bay.

Critical conditions of most spacecraft systems are monitored by a caution and warning system. A malfunction or out-of-tolerance condition results in illumination of a status light that identifies the abnormality. It also activates the master alarm circuit, which illuminates two master alarm lights on the MDC and one in the lower equipment bay and sends an alarm tone to the astronauts' headsets. The master alarm lights and tone continue until a crewman resets the master alarm circuit. This can be done before the crewmen deal with the problem indicated. The caution and warning system also contains equipment to sense its own malfunctions.

Lunar Module

The lunar module (LM) (fig. 3-9) is designed to transport two men safely from the CSM, in lunar orbit, to the lunar surface, and return them to the orbiting CSM. The LM provides operational capabilities such as communications, telemetry, environmental support, transportation of scientific equipment to the lunar surface, and returning surface samples with the crew to the CSM.

The lunar module consists of two stages: the ascent stage and the descent stage. The stages are attached at four fittings by explosive bolts. Separable umbilicals and hardline connections provide subsystem continuity to operate both stages as a single unit until separate ascent stage operation is desired. The LM is designed to operate for 48 hours after separation from the CSM, with a maximum lunar stay time of 44 hours. Table 3-I is a weight summary of the Apollo/Saturn 5 space vehicle for the Apollo 13 mission.

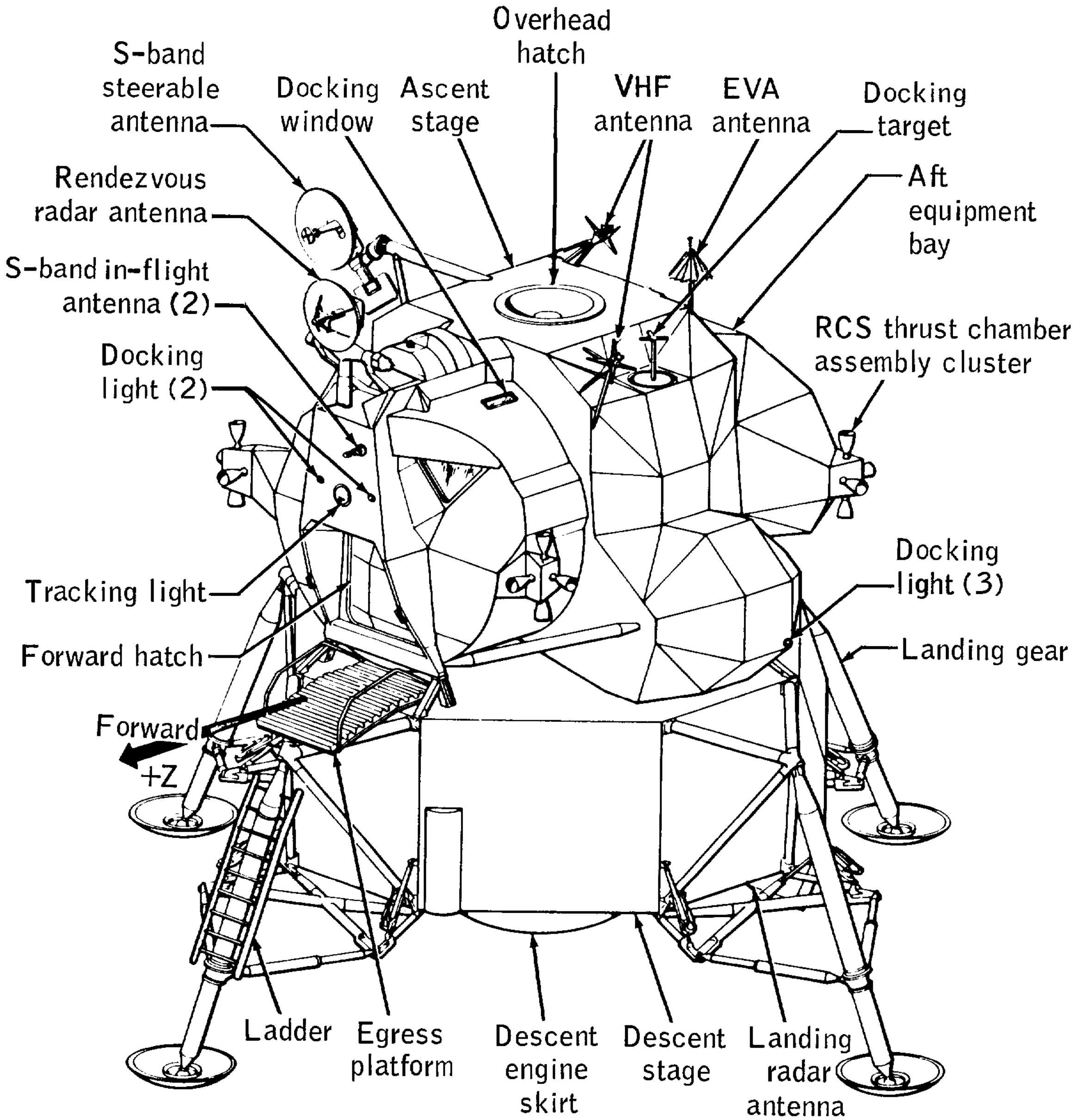


Figure 3-9.- Lunar module.

TABLE 3-I.- APOLLO 13 WEIGHT SUMMARY (WEIGHT IN POUNDS)

Stage/module	Inert weight	Total expendables	Total weight	Final separation weight
S-IC	288000	4746870	5034870	363403
S-IC/S-II interstage	11464	---	11464	---
S-II stage	78050	996960	1075010	92523
S-II/S-IVB interstage	8100	---	8100	---
S-IVB stage	25050	236671	261721	35526
Instrument unit	4482	---	4482	---
Launch vehicle at ignition 6,395,647				
Spacecraft-LM adapter	4044	---	4044	---
Lunar module	9915	23568	33483	*33941
Service module	10532	40567	51099	**14076
Command module	12572	---	12572	**11269 (Landing)
Launch escape system	9012	---	9012	---

* CSM/LM separation
** CM/SM separation

TABLE 3-I.- APOLLO 13 WEIGHT SUMMARY (WEIGHT IN POUNDS) - Concluded

Stage/module	Inert weight	Total expendables	Total weight	Final separation weight
Spacecraft at ignition 110,210				
Space vehicle at ignition			6505857	
S-IC thrust buildup			(-)84598	
Space vehicle at lift-off			6421259	
Space vehicle at orbit insertion			299998	

Main propulsion.- Main propulsion is provided by the descent propulsion system (DPS) and the ascent propulsion system (APS). Each system is wholly independent of the other. The DPS provides the thrust to control descent to the lunar surface. The APS can provide the thrust for ascent from the lunar surface. In case of mission abort, the APS and/or DPS can place the LM into a rendezvous trajectory with the CSM from any point in the descent trajectory. The choice of engine to be used depends on the cause for abort, on how long the descent engine has been operating, and on the quantity of propellant remaining in the descent stage. Both propulsion systems use identical hypergolic propellants. The fuel is a 50-50 mixture of hydrazine and unsymmetrical-dimethylhydrazine and the oxidizer is nitrogen tetroxide. Gaseous helium pressurizes the propellant feed systems. Helium storage in the DPS is at cryogenic temperatures in the super-critical state and in the APS it is gaseous at ambient temperatures.

Ullage for propellant settling is required prior to descent engine start and is provided by the +X axis reaction engines. The descent engine is gimbaled, throttleable, and restartable. The engine can be throttled from 1050 pounds of thrust to 6300 pounds. Throttle positions above this value automatically produce full thrust to reduce combustion chamber erosion. Nominal full thrust is 9870 pounds. Gimbal trim of the engine compensates for a changing center of gravity of the vehicle and is automatically accomplished by either the primary guidance and navigation system (PGNS) or the abort guidance system (AGS). Automatic throttle and on/off control is available in the PGNS mode of operation.

The AGS commands on/off operation but has no automatic throttle control capability. Manual control capability of engine firing functions has been provided. Manual thrust control override may, at any time, command more thrust than the level commanded by the LM guidance computer (LGC).

The ascent engine is a fixed, non-throttleable engine. The engine develops 3500 pounds of thrust, sufficient to abort the lunar descent or to launch the ascent stage from the lunar surface and place it in the desired lunar orbit. Control modes are similar to those described for the descent engine. The APS propellant is contained in two spherical titanium tanks, one for oxidizer and the other for fuel. Each tank has a volume of 36 cubic feet. Total fuel weight is 2008 pounds, of which 71 pounds are unusable. Oxidizer weight is 3170 pounds, of which 92 pounds are unusable. The APS has a limit of 35 starts, must have a propellant bulk temperature between 50° F and 90° F prior to start, must not exceed 460 seconds of burn time, and has a system life of 24 hours after pressurization.

Electrical power system.- The electrical power system (EPS) contains six batteries which supply the electrical power requirements of the LM during undocked mission phases. Four batteries are located in the descent stage and two in the ascent stage. Batteries for the explosive devices system are not included in this system description. Postlaunch LM power is supplied by the descent stage batteries until the LM and CSM are docked. While docked, the CSM supplies electrical power to the LM up to 296 watts (peak). During the lunar descent phase, the two ascent stage batteries are paralleled with the descent stage batteries for additional power assurance. The descent stage batteries are utilized for LM lunar surface operations and checkout. The ascent stage batteries are brought on the line just before ascent phase staging. All batteries and busses may be individually monitored for load, voltage, and failure. Several isolation and combination modes are provided.

Two inverters, each capable of supplying full load, convert the dc to ac for 115-volt, 400-hertz supply. Electrical power is distributed by the following busses: LM Pilot's dc bus, Commander's dc bus, and ac busses A and B.

The four descent stage silver-zinc batteries are identical and have a 400 ampere-hour capacity at 28 volts. Because the batteries do not have a constant voltage at various states of charge/load levels, "high" and "low" voltage taps are provided for selection. The "low voltage" tap is selected to initiate use of a fully charged battery. Cross-tie circuits in the busses facilitate an even discharge of the batteries regardless of distribution combinations. The two silver-zinc ascent stage batteries are identical to each other and have a 296 ampere-hour capacity at 28 volts. The ascent stage batteries are normally connected in parallel for even discharge. Because of design load characteristics, the ascent stage batteries do not have and do not require high and low voltage taps.

Nominal voltage for ascent stage and descent stage batteries is 30.0 volts. Reverse current relays for battery failure are one of many components designed into the EPS to enhance EPS reliability. Cooling of the batteries is provided by the environmental control system cold rail heat sinks. Available ascent electrical energy is 17.8 kilowatt hours at a maximum drain of 50 amps per battery and descent energy is 46.9 kilowatt hours at a maximum drain of 25 amps per battery.

MISSION MONITORING, SUPPORT, AND CONTROL

Mission execution involves the following functions: prelaunch checkout and launch operations; tracking the space vehicle to determine its present and future positions; securing information on the status of the flight crew and space vehicle systems (via telemetry); evaluation of telemetry information; commanding the space vehicle by transmitting real-time and updata commands to the onboard computer; and voice communication between flight and ground crews.

These functions require the use of a facility to assemble and launch the space vehicle (see Launch Complex), a central flight control facility, a network of remote stations located strategically around the world, a method of rapidly transmitting and receiving information between the space vehicle and the central flight control facility, and a real-time data display system in which the data are made available and presented in usable form at essentially the same time that the data event occurred.

The flight crew and the following organizations and facilities participate in mission control operations:

a. Mission Control Center (MCC), Manned Spacecraft Center (MSC), Houston, Texas. The MCC contains the communication, computer display, and command systems to enable the flight controllers to effectively monitor and control the space vehicle.

b. Kennedy Space Center (KSC), Cape Kennedy, Florida. The space vehicle is launched from KSC and controlled from the Launch Control Center (LCC). Prelaunch, launch, and powered flight data are collected at the Central Instrumentation Facility (CIF) at KSC from the launch pads, CIF receivers, Merritt Island Launch Area (MILA), and the downrange Air Force Eastern Test Range (AFETR) stations. These data are transmitted to MCC via the Apollo Launch Data System (ALDS). Also located at KSC (AFETR) is the Impact Predictor (IP), for range safety purposes.

c. Goddard Space Flight Center (GSFC), Greenbelt, Maryland. GSFC manages and operates the Manned Space Flight Network (MSFN) and the NASA communications (NASCOM) network. During flight, the MSFN is under the operational control of the MCC.

d. George C. Marshall Space Flight Center (MSFC), Huntsville, Alabama. MSFC, by means of the Launch Information Exchange Facility (LIEF) and the Huntsville Operations Support Center (HOSC) provides launch vehicle systems real-time support to KSC and MCC for preflight, launch, and flight operations.

A block diagram of the basic flight control interfaces is shown in figure 3-10.

Vehicle Flight Control Capability

Flight operations are controlled from the MCC. The MCC has two flight control rooms, but only one control room is used per mission. Each control room, called a Mission Operations Control Room (MOCR), is capable of controlling individual Staff Support Rooms (SSR's) located adjacent to the MOCR. The SSR's are manned by flight control specialists who provide detailed support to the MOCR. Figure 3-11 outlines the organization of the MCC for flight control and briefly describes key responsibilities. Information flow within the MOCR is shown in figure 3-12.

The consoles within the MOCR and SSR's permit the necessary interface between the flight controllers and the spacecraft. The displays and controls on these consoles and other group displays provide the capability to monitor and evaluate data concerning the mission and, based on these evaluations, to recommend or take appropriate action on matters concerning the flight crew and spacecraft.

Problems concerning crew safety and mission success are identified to flight control personnel in the following ways:

a. Flight crew observations

b. Flight controller real-time observations

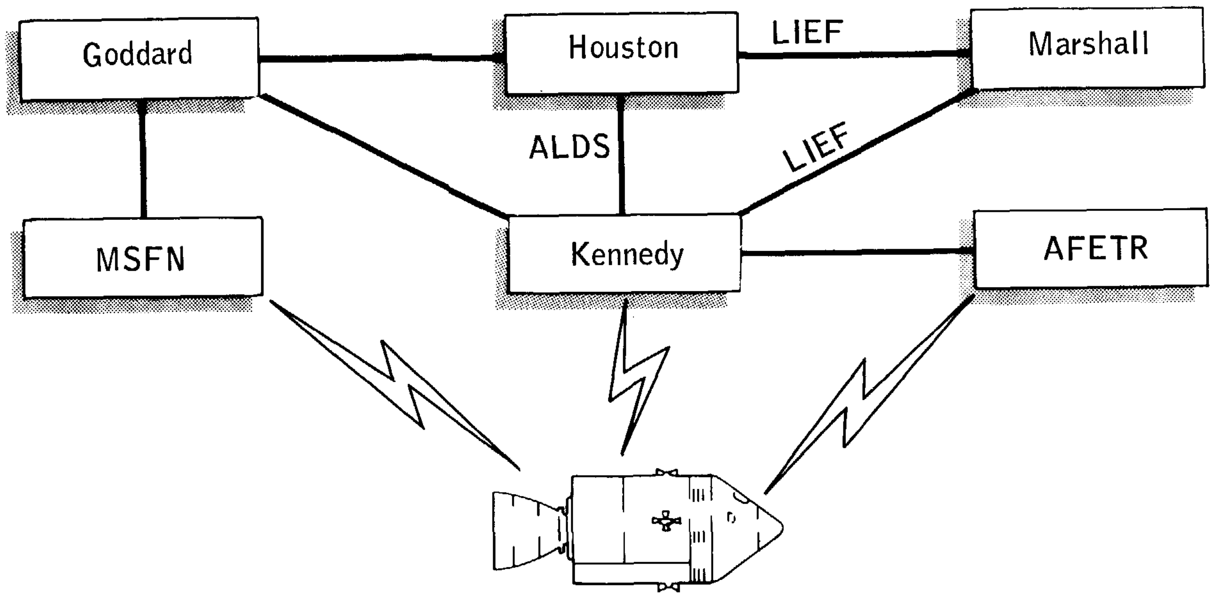
c. Review of telemetry data received from tape recorder playback

d. Trend analysis of actual and predicted values

e. Review of collected data by systems specialists

f. Correlation and comparison with previous mission data

g. Analysis of recorded data from launch complex testing



ALDS - Apollo Launch Data System
LIEF - Launch Information Exchange Facility

Figure 3-10.- Basic telemetry, command, and communication interfaces for flight control.

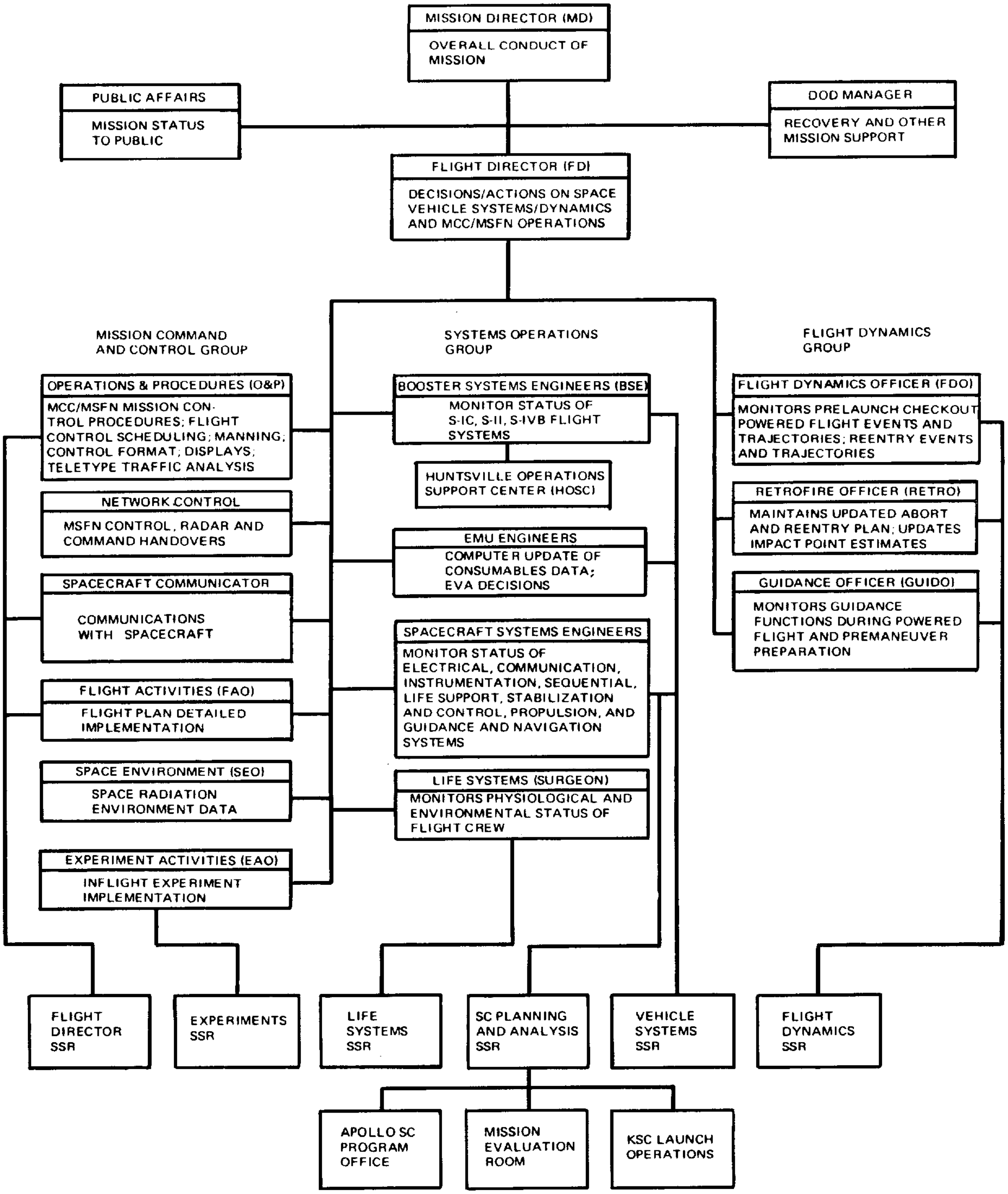


Figure 3-11.- Mission Control Center organization.

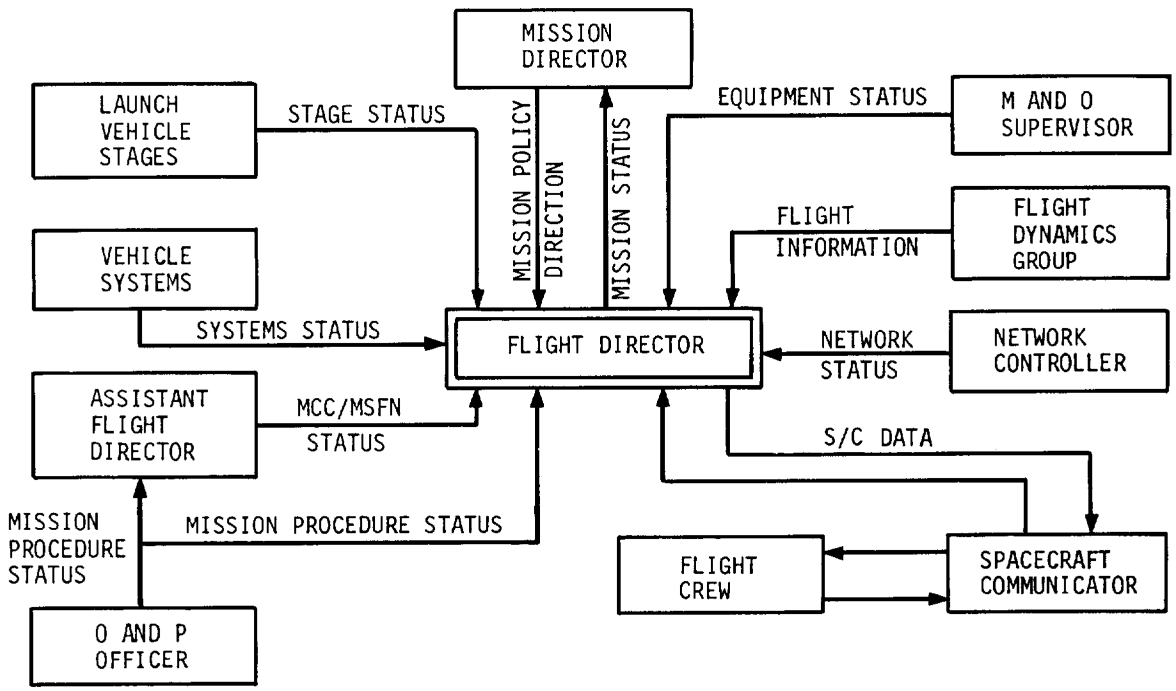


Figure 3-12.- Information flow within the Mission Operations Control Room.

PART 2. APOLLO 13 MISSION DESCRIPTION

PRIMARY MISSION OBJECTIVES

The primary mission objectives were as follows:

Perform selenological inspection, survey, and sampling of materials in a preselected region of the Fra Mauro Formation.

Deploy and activate an Apollo Lunar Surface Experiments Package (ALSEP).

Develop man's capability to work in the lunar environment.

Obtain photographs of candidate exploration sites.

Table 3-II lists the Apollo 13 mission sequence of major events and the time of occurrence in ground elapsed time.

TABLE 3-II. - APOLLO 13 MISSION SEQUENCE OF EVENTS

Event	Ground elapsed time (hr:min:sec)
Range zero (02:13:00.0 p.m. e.s.t., April 11)	00:00:00
Earth parking orbit insertion	00:12:40
Second S-IVB ignition	02:35:46
Translunar injection	02:41:47
CSM/S-IVB separation	03:06:39
Spacecraft ejection from S-IVB	04:01:03
S-IVB APS evasive maneuver	04:18:01
S-IVB APS maneuver for lunar impact	05:59:59
Midcourse correction - 2 (hybrid transfer)	30:40:50
Cryogenic oxygen tank anomaly	55:54:53
Midcourse correction - 4	61:29:43
S-IVB lunar impact	77:56:40
Pericynthion plus 2-hour maneuver	79:27:39
Midcourse correction - 5	105:18:32
Midcourse correction - 7	137:39:49
Service module jettison	138:02:06
Lunar module jettison	141:30:02
Entry interface	142:40:47
Landing	142:54:41

Launch and Earth Parking Orbit

Apollo 13 was successfully launched on schedule from Launch Complex 39A, Kennedy Space Center, Florida, at 2:13 p.m. e.s.t., April 11, 1970. The launch vehicle stages inserted the S-IVB/instrument unit (IU)/ spacecraft combination into an earth parking orbit with an apogee of 100.2 nautical miles (n. mi.) and a perigee of 98.0 n. mi. (100-n. mi. circular planned). During second stage boost, the center engine of the S-II stage cut off about 132 seconds early, causing the remaining four engines to burn approximately 34 seconds longer than predicted. Space vehicle velocity after S-II boost was 223 feet per second (fps) lower than planned. As a result, the S-IVB orbital insertion burn was approximately 9 seconds longer than predicted with cutoff velocity within about 1.2 fps of planned. Total launch vehicle burn time was about 44 seconds longer than predicted. A greater than 3-sigma probability of meeting translunar injection (TLI) cutoff conditions existed with remaining S-IVB propellants.

After orbital insertion, all launch vehicle and spacecraft systems were verified and preparation was made for translunar injection (TLI). Onboard television was initiated at 01:35 ground elapsed time (g.e.t.) for about 5.5 minutes. The second S-IVB burn was initiated on schedule for TLI. All major systems operated satisfactorily and all end conditions were nominal for a free-return circumlunar trajectory.

Translunar Coast

The CSM separated from the LM/IU/S-IVB at about 03:07 g.e.t. Onboard television was then initiated for about 72 minutes and clearly showed CSM "hard docking," ejection of the CSM/LM from the S-IVB at about 04:01 g.e.t., and the S-IVB auxiliary propulsion system (APS) evasive maneuver as well as spacecraft interior and exterior scenes. The SM RCS propellant usage for the separation, transposition, docking, and ejection was nominal. All launch vehicle safing activities were performed as scheduled.

The S-IVB APS evasive maneuver by an 8-second APS Ullage burn was initiated at 04:18 g.e.t. and was successfully completed. The liquid oxygen dump was initiated at 04:39 g.e.t. and was also successfully accomplished. The first S-IVB APS burn for lunar target point impact was initiated at 06:00 g.e.t. The burn duration was 217 seconds, producing a differential velocity of approximately 28 fps. Tracking information available at 08:00 g.e.t. indicated that the S-IVB/IU would impact at 6°53' S., 30°53' W. versus the targeted 3° S., 30° W. Therefore, the second S-IVB APS (trim) burn was not required. The gaseous nitrogen pressure dropped in the IU ST-124-M3 inertial platform at 18:25 g.e.t. and the S-IVB/IU no longer had attitude control but began tumbling slowly.

At approximately 19:17 g.e.t., a step input in tracking data indicated a velocity increase of approximately 4 to 5 fps. No conclusions have been reached on the reason for this increase. The velocity change altered the lunar impact point closer to the target. The S-IVB/IU impacted the lunar surface at 77:56:40 g.e.t. (08:09:40 p.m. e.s.t. April 14) at 2.4° S., 27.9° W., and the seismometer deployed during the Apollo 12 mission successfully detected the impact. The targeted impact point was 125 n. mi. from the seismometer. The actual impact point was 74 n. mi. from the seismometer, well within the desired 189-n. mi. (350-km) radius.

The accuracy of the TLI maneuver was such that spacecraft midcourse correction No. 1 (MCC-1), scheduled for 11:41 g.e.t., was not required. MCC-2 was performed as planned at 30:41 g.e.t. and resulted in placing the spacecraft on the desired, non-free-return circumlunar trajectory with a predicted closest approach to the moon on 62 n. mi. All SPS burn parameters were normal. The accuracy of MCC-3 was such that MCC-3, scheduled for 55:26 g.e.t., was not performed. Good quality television coverage of the preparations and performance of MCC-2 was received for 49 minutes beginning at 30:13 g.e.t.

At approximately 55:55 g.e.t. (10:08 p.m. e.s.t.), the crew reported an undervoltage alarm on the CSM main bus B. Pressure was rapidly lost in SM oxygen tank no. 2 and fuel cells 1 and 3 current dropped to zero due to loss of their oxygen supply. A decision was made to abort the mission. The increased load on fuel cell 2 and decaying pressure in the remaining oxygen tank led to the decision to activate the LM, power down the CSM, and use the LM systems for life support.

At 61:30 g.e.t., a 38-fps midcourse maneuver (MCC-4) was performed by the LM DPS to place the spacecraft in a free-return trajectory on which the CM would nominally land in the Indian Ocean south of Mauritius at approximately 152:00 g.e.t.

Transearth Coast

At pericynthion plus 2 hours (79:28 g.e.t.), a LM DPS maneuver was performed to shorten the return trip time and move the earth landing point. The 263.4-second burn produced a differential velocity of 860.5 fps and resulted in an initial predicted earth landing point in the mid-Pacific Ocean at 142:53 g.e.t. Both LM guidance systems were powered up and the primary system was used for this maneuver. Following the maneuver, passive thermal control was established and the LM was powered down to conserve consumables; only the LM environmental control system (ECS) and communications and telemetry systems were kept powered up.

The LM DPS was used to perform MCC-5 at 105:19 g.e.t. The 15-second burn (at 10-percent throttle) produced a velocity change of about 7.8 fps and successfully raised the entry flight path angle to -6.52°.

The CSM was partially powered up for a check of the thermal conditions of the CM with first reported receipt of S-band signal at 101:53 g.e.t. Thermal conditions on all CSM systems observed appeared to be in order for entry.

Due to the unusual spacecraft configuration, new procedures leading to entry were developed and verified in ground-based simulations. The resulting timeline called for a final midcourse correction (MCC-7) at entry interface (EI) -5 hours, jettison of the SM at EI -4.5 hours, then jettison of the LM at EI -1 hour prior to a normal atmospheric entry by the CM.

MCC-7 was successfully accomplished at 137:40 g.e.t. The 22.4-second LM RCS maneuver resulted in a predicted entry flight path angle of -6.49°. The SM was jettisoned at 138:02 g.e.t. The crew viewed and photographed the SM and reported that an entire panel was missing near the S-band high-gain antenna and a great deal of debris was hanging out. The CM was powered up and then the LM was jettisoned at 141:30 g.e.t. The EI at 40,000 feet was reached at 142:41 g.e.t.

Entry and Recovery

Weather in the prime recovery area was as follows: broken stratus clouds at 2000 feet; visibility 10 miles; 6-knot ENE winds; and wave height 1 to 2 feet. Drogue and main parachutes deployed normally. Visual contact with the spacecraft was reported at 142:50 g.e.t. Landing occurred at 142:54:41 g.e.t. (01:07:41 p.m. e.s.t., April 17). The landing point was in the mid-Pacific Ocean, approximately 21°40' S., 165°22' W. The CM landed in the stable 1 position about 3.5 n. mi. from the prime recovery ship, USS IWO JIMA. The crew, picked up by a recovery helicopter, was safe aboard the ship at 1:53 p.m. e.s.t., less than an hour after landing.

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