4.3.2.4 Block II spacecraft.- The command and service modules used for all manned missions were of the Block II design (fig. 4-5). Although similar to the Block I spacecraft, a number of changes were made as a result of the program definition study of 1964 and the Apollo I fire in 1967. The major changes are listed in table 4-II. Design changes continued to take place throughout the program as studies and analyses progressed, as hardware failures occurred, and as new requirements developed. Major modifications were made for the final three missions because of expanded requirements for scientific data acquisition from lunar orbit. While these modifications were being implemented, the investigation accruing from the cryogenic oxygen system failure experienced on Apollo 13 dictated additional changes. These changes are also summarized in table 4-II.

4.3.2.5 Block II ground test program.- A considerable number of ground tests were conducted in support of the Block II changes. The test program was not formulated all at once but, rather, was developed over a period of several years as the spacecraft design was reevaluated. The test program embraced the original concept of minimizing flight tests and maximizing ground tests.

4.4 COMMAND AND SERVICE MODULE SYSTEMS DEVELOPMENT AND PERFORMANCE

4.4.1 Introduction

Significant aspects of the development and flight performance of individual command and service module structures and systems are summarized in this section. Brief descriptions of the systems are given where necessary but are not generally included. Complete descriptions of the boilerplate and Block I spacecraft systems are given in references 4-1 through 4-12. The initial Block II command and service module is described in reference 4-13, and subsequent changes are noted in references 4-14 through 4-23. The topics discussed, in some cases, have been treated in greater detail in other individual reports and these are referenced where appropriate.

4.4.2 Structures

The boilerplate flight test vehicles were designed primarily to demonstrate the capability of the launch escape system and to obtain aerodynamic flight data. Therefore, design requirements were to sustain ground and flight loading environments and to present a configuration similar to that of the production flight articles. The Block I and Block II flight spacecraft were designed to sustain normal flight, entry, and recovery loadings, and to provide protection from meteoroids, radiation, and thermal extremes.

Most of the problems encountered in the development and verification of the structure were discovered in the ground test program when the structure failed to meet specified criteria, environment, or loads. Each failure was carefully analyzed, and the specific test criteria were reassessed. In some cases, the reassessment revealed that the test conditions were too severe and should be changed to more realistic conditions. In other cases, structural inadequacies that required design changes were identified. Some modifications were retested, whereas others were certified by analysis. Many of these structural failures were due to inaccurate predictions of load paths and load distribution. The capability of structural analysis methods improved continually during the Apollo program. The structural aspects of the ground and flight test programs as well as significant problems encountered in the test programs and their resolutions are discussed in reference 4-24.

On the Apollo 6 mission, a local structural failure of the spacecraft/lunar module adapter occurred during first-stage boost (ref. 4-12). Approximately 2 minutes 13 seconds after liftoff, abrupt changes of strain, vibration, and acceleration were indicated by onboard instrumentation. Photographs showed objects falling from the area of the adapter; however, the adapter continued to sustain the required loads.

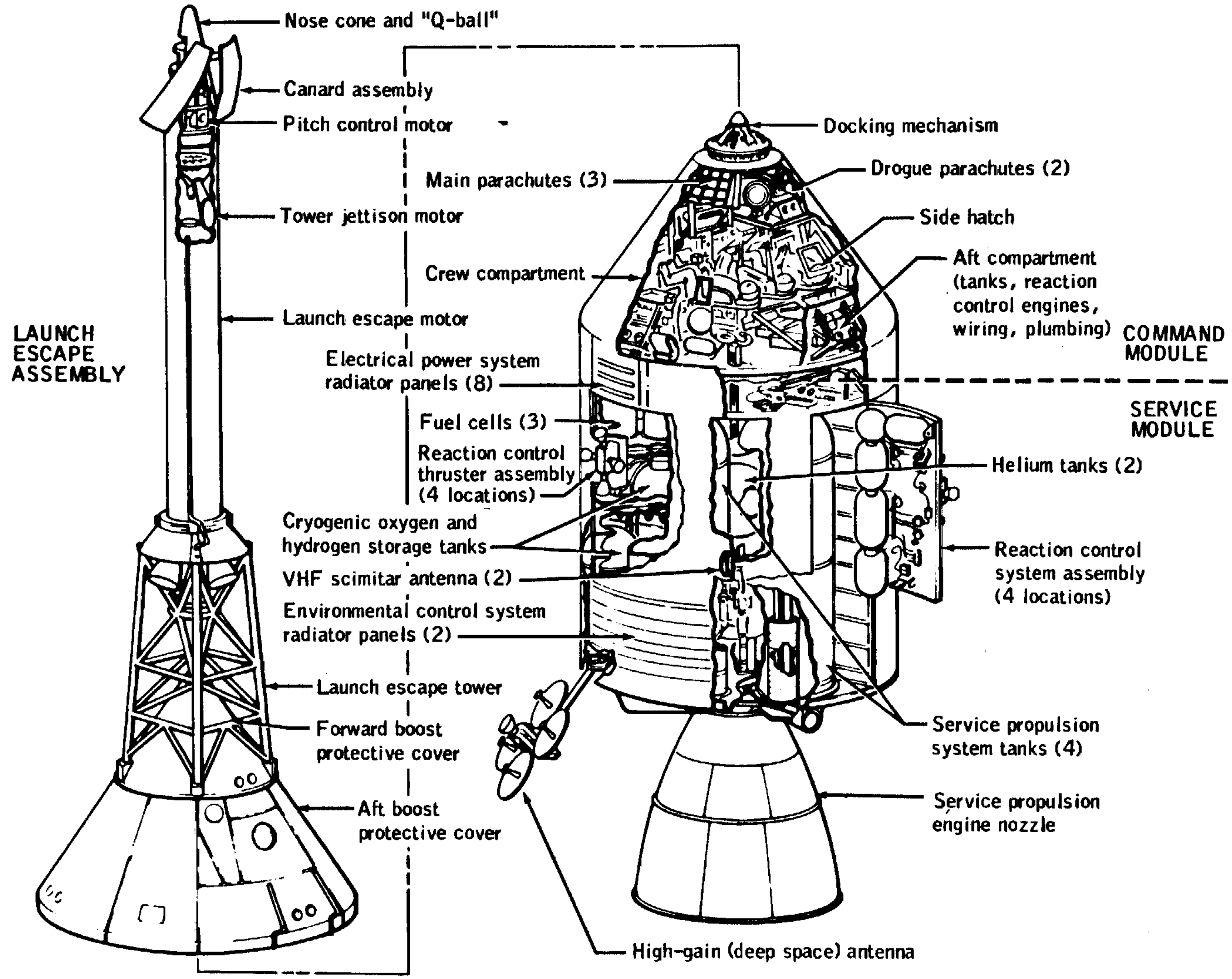


Figure 4-5.- Command and service modules and launch escape system.

TABLE 4-II.- SUMMARY OF MAJOR CHANGES TO COMMAND AND SERVICE MODULE

Function/system	Changes
Changes Resulting From Program Definition Study	
Structures and thermal protection	Forward tunnel structure changed to accomodate docking mechanism and lunar module/command module umbilicals added Antenna protuberances removed from command module Parachute attachment redesigned Command module/service module umbilical relocated Equipment rearranged in service module to provide an empty bay in sector I for later installation of scientific instrument module Micrometeoroid protection added to service module Extravehicular activity provisions incorporated Boost protective cover added Heat shield ablator thickness reduced
Mechanical systems	Docking mechanism added Earth landing system capability improved Unitized couch changed to foldable type and impact attenuation system improved
Thermal control	Changes incorporated for use of passive thermal control
Environmental control	Radiator size increased Selective fluid (water/glycol) freezing and thawing used to accomodate variable heat loads and external environment

TABLE 4-II.- SUMMARY OF MAJOR CHANGES
TO COMMAND AND SERVICE MODULE - Continued

Function/system	Changes
Changes Resulting From Program Definition Study - Continued	
Communications and instrumentation	VHF transceiver redesigned C-band transponder deleted HF recovery transceiver and antenna deleted Electronics packages hermetically sealed with built-in and switchable redundancy
Guidance, navigation and control	Smaller, lighter, and more reliable system used Electronics packages hermetically sealed with built-in and switchable redundancy New entry monitor system scrolls incorporated Flight director attitude indicator redesigned
Propulsion	Service module reaction control system propellant storage capacity increased Size and thickness of service propulsion tanks reduced Service propulsion system main propellant valve control redesigned
Sequential events control	Reliability of events controllers improved Motor switches, instead of relays, used to arm pyrotechnic bus Events controllers added to accomodate lunar module
Crew equipment	Rendezvous and docking aids provided

TABLE 4-II.- SUMMARY OF MAJOR CHANGES
TO COMMAND AND SERVICE MODULE - Continued

Function/system	Changes
Changes Following Apollo I Fire	
Mechanical	Unitized, quick-opening side hatch incorporated Earth landing system modified to withstand opening loads resulting from increased command module weight Uprighting system redesigned as a result of change in the command module center of gravity
Environmental control	Provisions made for nitrogen/oxygen cabin atmosphere prior to launch Rapid cabin repressurization system added High pressure lines changed from aluminum to stainless steel, and joints welded instead of soldered
Electrical	Wiring protection added Harnesses rerouted
Crew station	Use of nonflammable materials expanded
Changes Implemented as a Result of the Apollo 13 Abort	
Cryogenic storage	Oxygen tank redesigned Third oxygen tank installed Isolation valve installed between oxygen tanks 2 and 3 Controls and displays added
Electrical	Lunar module descent stage battery added for emergency power Fuel cell reactant shutoff valves relocated
Crew equipment	Contingency water storage system added

TABLE 4-II.- SUMMARY OF MAJOR CHANGES
TO COMMAND AND SERVICE MODULE - Concluded

Function/system	Changes
Changes Implemented for Apollo 15 and Subsequent Missions	
Structural	Scientific instrument module installed Extravehicular handholds and restraints installed
Mechanical systems	Experiment deployment devices added to the service module
Cryogenic storage	Third hydrogen tank installed
Environmental control	Components added to accomodate extra-vehicular activity
Communications and instrumentation	Scientific data system integrated with existing telemetry system
Crew station	Controls and displays added Additional stowage provided

Extensive study of the photography and other evidence indicated that a large area of the adapter had lost inner facesheet from the honeycomb sandwich panels. Loads and stresses resulting from vibration were determined to be insufficient to initiate such a failure. The investigation was then directed toward determining the range of pressures that could have been trapped in the Apollo 6 adapter sandwich panels, and toward determining the tolerance of the panels to withstand pressure with various degrees of flaws such as adhesive voids and facesheet dents. The degradation effects of moisture and heat exposure on the adhesive strength were also studied and tested. These tests and analyses led to the conclusion that pressure internal to the sandwich panels could have caused the failure, if a large flaw existed. The pressure buildup would have been caused by aerodynamic heating effects on air and moisture trapped in the panel.

The most probable cause of the failure was an abnormal splice assembly, resulting in a facesheet bond too weak for the internal pressure achieved. Sufficient information was developed to verify that deficient assembly techniques had resulted in abnormalities along a panel splice in several of the adapters to be used on subsequent flights.

Before the splice abnormalities were pinpointed, corrective action was taken to reduce pressure buildup in the honeycomb panels and to reduce heat degrading effects on the adhesive. This was done by drilling vent holes in the inner facesheet and covering the outer facesheet with cork. The adapters having splice abnormalities were repaired, and an internal splice plate was eliminated to allow more accurate inspection.

4.4.3 Thermal Management Systems

Management of temperatures within the limits necessary for proper spacecraft systems operation and human occupancy was accomplished by three separate systems: the environmental control system, the thermal control system, and the thermal protection system. The environmental control system is discussed in section 4.4.9. It contained a water/glycol flow system which transferred heat to radiators located on the service module surface and a water boiler for the sublimation of water in the space environment. These functioned as a thermodynamic unit to maintain a habitable cabin thermal environment and to cool electronic equipment located within the cabin. The thermal control system regulated temperatures of the structure and components outside the pressure vessel. The thermal protection system consisted of components which protected the cabin and crew from the entry environment.

Both active and passive means of temperature management were utilized. The active means consisted primarily of the water/glycol flow system and water boiler used for environmental control, as well as electrical heaters. The passive means included: ablative materials that accommodated high heating rates, thermal control coatings, insulations, heat sink materials, and spacecraft orientation.

4.4.3.1 Thermal protection.- The lunar return trajectory of the Apollo spacecraft resulted in an atmospheric entry inertial velocity of over 36 000 feet per second, and this created an aerodynamic heating environment approximately four times as severe as that experienced by either the Mercury or Gemini spacecraft. The induced thermal environment resulting from such an entry necessitated the installation of a heat shield on the command module capable of sustaining, without excessive erosion, the temperatures caused by the high heating rates on the blunt face of the vehicle while preventing excessive substructure temperatures. The concept initially considered consisted of ablative tiles made from phenolic-nylon material bonded to a honeycomb-sandwich substructure made of aluminum. However, in April 1962, recovered heat shields from Mercury spacecraft were found to have experienced debonding of tiled ablative material, and an alternative study was conducted of the ablator insulation method being successfully demonstrated at that time on the Gemini spacecraft. The Gemini heat shield consisted of a fiberglass honeycomb core filled with an elastomeric ablator. Initially, the cells were filled with the ablator by a tamping process, but this caused concern with respect to quality assurance, and the composition of the ablative material was modified so that it could be gunned in a mastic form into the honeycomb cells. Stainless steel was chosen for the substructure in preference to aluminum because of the increased safety provided by the higher-melting-point alloy in the event of a localized loss of ablator.

Unmanned flights provided test verification of the thermal protection system for earth-orbital and lunar-return missions. The measured data obtained from these flights (table 4-III) and from the first two manned flights were used to correlate the analytical models used for the required certification analysis.

Table 4-IV is a summary of the actual entry conditions for the Apollo 8 mission and the Apollo 10 through 17 missions. As indicated in the table, the maximum downrange entry distance was 1497 miles compared with the established Block II design requirement of 3500 miles. The shorter downrange entry distance resulted in a maximum integrated heat load of 28 000 Btu/sq ft, which was appreciably less than the design requirement of 44 500 Btu/sq ft.

4.4.3.2 Thermal control.- The evolution of the thermal control system revealed that mission operational constraints could be used to minimize weight and power requirements. The original concept was that the spacecraft should be insensitive to attitude and position in space. However, unconstrained operational attitudes dictated system design for the worst-case mission environment, which would then have involved the use of such devices as multiple cooling loops and large heaters. The consequences would have been increased spacecraft weight and larger propellant expenditures. After consideration of all aspects of the mission, a plan was developed which made optimum use of the natural space environment to provide passive temperature control. The spacecraft longitudinal axis was aligned normal to the direction of the solar radiation and the spacecraft was rotated about this axis at a nominal rate of 3 revolutions per hour during the translunar and transearth coast phases; the alignment and rotational operations were termed the passive thermal control mode. Another passive thermal control mode was used during sleep periods while in lunar orbit. The command and service module was held in an orientation with solar radiation impinging directly on reaction control system quad B. (The service propulsion system oxidizer sump tank adjacent to quad B acted as a thermal sink.) Utilization of these modes permitted the definition of a large operational envelope in which the spacecraft could function and was used in the planning of each mission to define the thermodynamically related constraints on the vehicles. The flight plan for a nominal mission placed the vehicle in the center of the design envelope in order to maximize its capability to accommodate mission contingencies.

During the evolution of the thermal control design, many tests were conducted to determine insulation performance and installation techniques, thermal control coating properties, coating application processes, thermal shielding performance, and shielding manufacturing techniques. Additional tests were performed to determine the environment to which these materials would be exposed such as rocket engine plume characteristics and aerodynamic heating rates. The results of these tests were used in the development of the thermal mathematical models utilized to determine the adequacy of each thermal control design concept. It was necessary, however, to verify the many assumptions and engineering idealizations which were made in order that the interdependency of the spacecraft structure and systems could be adequately mathematically represented.

Full-scale thermal vacuum tests were performed to provide a means of verifying the spacecraft thermal control system design and the adequacy of the mathematical models used for thermal analysis. Two integrated command and service module prototypes were tested in a thermal vacuum chamber at the Manned Spacecraft Center. Both prototypes (SC-008 and 2TV-1) were exposed to combinations of hot and cold soaks in addition to passive thermal control rolling modes while manned with all systems except the propulsion system operating. In general, the assumptions made in the thermal analyses were found to be conservative (i.e., the measured maximum and minimum temperatures were within the predicted extremes).

No serious problems or anomalies were associated with the thermal control and thermal protection systems on the earth-orbital and lunar missions. The success of the systems can be attributed to the somewhat conservative design philosophy that was adopted and to the rigorous analytical and test certification requirements that were imposed. More detailed information on thermal protection during launch and entry may be found in references 4-25 and 4-26.

TABLE 4-III.- FLIGHT VERIFICATION OF THE THERMAL PROTECTION SYSTEM

Entry conditions	Mission			
	AS-201	AS-202	AS-501	AS-502
Inertial velocity at entry, ft/sec	26 482	28 512	36 545	32 830
Relative velocity at entry, ft/sec	25 318	27 200	35 220	31 530
Inertial flight-path angle at entry, deg . .	-8.60	-3.53	-6.93	-5.85
Range flown, miles	470	2295	1951	1935
Entry time, sec	674	1234	1060	1140
Maximum heating rate, Btu/ft^2/sec	164	83	425	197
Total reference heating load, Btu/ft^2 . . .	6889	20 862	37 522	27 824

TABLE 4-IV.- SUMMARY OF ENTRY CONDITIONS FOR OPERATIONAL LUNAR MISSIONS

Entry conditions	Mission								
	Apollo 8	Apollo 10	Apollo 11	Apollo 12	Apollo 13	Apollo 14	Apollo 15	Apollo 16	Apollo 17 (a)
Inertial velocity at entry, ft/sec	36 221	36 314	36 194	36 116	36 211	36 170	36 096	36 090	36 090
Relative velocity at entry, ft/sec	35 000	34 968	35 024	34 956	34 884	34 996	34 928	35 502	35 502
Inertial flight-path angle at entry, deg	−6.48	−6.54	−6.98	−6.50	−6.49	−6.37	−6.51	−6.49	−6.49
Lift-to-drag ratio	0.300	0.305	0.300	0.309	0.291	0.280	0.290	0.286	0.290
Range flown, miles	1292	1295	1497	1250	1250	1234	1184	1190	1190
Entry time, sec	868	871	929	815	835	853	778	814	801
Maximum heating rate, $Btu/ft^2/sec$	296	296	286	285	271	310	289	346	346
Total reference heating load Btu/ft^2	26 140	25 728	26 482	26 224	25 710	27 111	25 881	27 939	27 939

[a]Data shown are preflight predictions. Actual data were not obtained.

4.4.4 Mechanical Systems

The major mechanical systems incorporated in the command and service modules are discussed in this subsection.

4.4.4.1 Earth landing system.- The earth landing system consisted of three main parachutes, two drogue parachutes, a forward heat shield separation augmentation parachute, and related electromechanical and pyrotechnic actuation components required to decelerate and stabilize the command module to conditions that were safe for landing after either a normal entry or a launch abort. The recovery sequence was initiated automatically by the closure of barometric pressure switches or by manual initiation of time-delay relays.

In addition to stringent program requirements, several specific technical problems, the solution of which required the development of innovative methods and techniques, were encountered. The most severe problem was a continual increase in command module weight. This condition resulted in a major program of redesign and requalification of the Block II earth landing system. The command module weight increases and certain program events are depicted in figure 4-6.

The first three Block I developmental aerial drop tests (single parachutes) were conducted with a parachute constructed from lightweight material and having a minimum of reinforcing tapes. Because major damage was sustained on two of the three tests and because of the first announcement of a command module weight increase, the first modification was made to strengthen and to improve the main-parachute design. The initial changes increased the strength of the structural members of the parachute. These changes caused a significant increase in parachute weight, and the attendant bulk created new problems because limited stowage volume was available. Shortly after the start of main-parachute-cluster tests, modifications had to be made to the main parachutes to change their opening characteristics to achieve more evenly balanced load sharing among the three parachutes, thereby reducing the peak opening loads.

By the time qualification testing of the Block I earth landing system was completed, each system of the spacecraft had progressed to the point that accurate total weight estimates were available. Although the maximum projected weight for a Block II spacecraft was more than the specification value, the overweight condition was not sufficient to justify major design changes in the earth landing system. Therefore, the Block II parachute qualification program was pursued as a minimum-change effort.

During the months immediately following the Apollo I fire, numerous modifications were made to the command module. By mid-April 1967, weight estimates indicated that the projected spacecraft weight had increased to a value greater than that at which the earth landing system could recover the command module with an acceptable factor of safety. The implemented solution consisted of increasing the size of the drogue parachutes and of providing the existing main parachutes with an additional reefing stage. The two changes ensured an adequate factor of safety for the parachutes and the command module structure at the projected recovery weight of 13 000 pounds. Larger drogue parachutes on the heavier command module reduced the dynamic pressure at drogue disconnect/pilot mortar fire to a level near that obtained with the smaller drogue parachutes on the lighter spacecraft. The additional reefing stage in the main parachutes reduced the individual and total main-parachute loads to values no greater than the design loads for an 11 000-pound command module.

In addition to resolving difficult design problems, devising and optimizing component manufacturing and assembling techniques were also necessary to ensure that each part would function properly once it was assembled and installed on the spacecraft. None of the previous space programs required the high density of parachute packing to suit the allotted volume that was necessary in the Apollo program. This requirement necessitated the development of precise techniques for packing the parachutes at very high densities without inflicting damage to the parachute system during packing or deployment. Substitution of steel cables for nylon risers in the parachute system required the development of stowage techniques that provided safe deployment of the cable.

Modifications or procedural changes were made several times in the program because of potentially hazardous conditions that were discovered during mission operations. On the AS-201 mission, the forward heat shield jettisoning system did not provide sufficient energy to thrust the heat shield through the wake of a stabilized command module. To ensure separation, a conventional

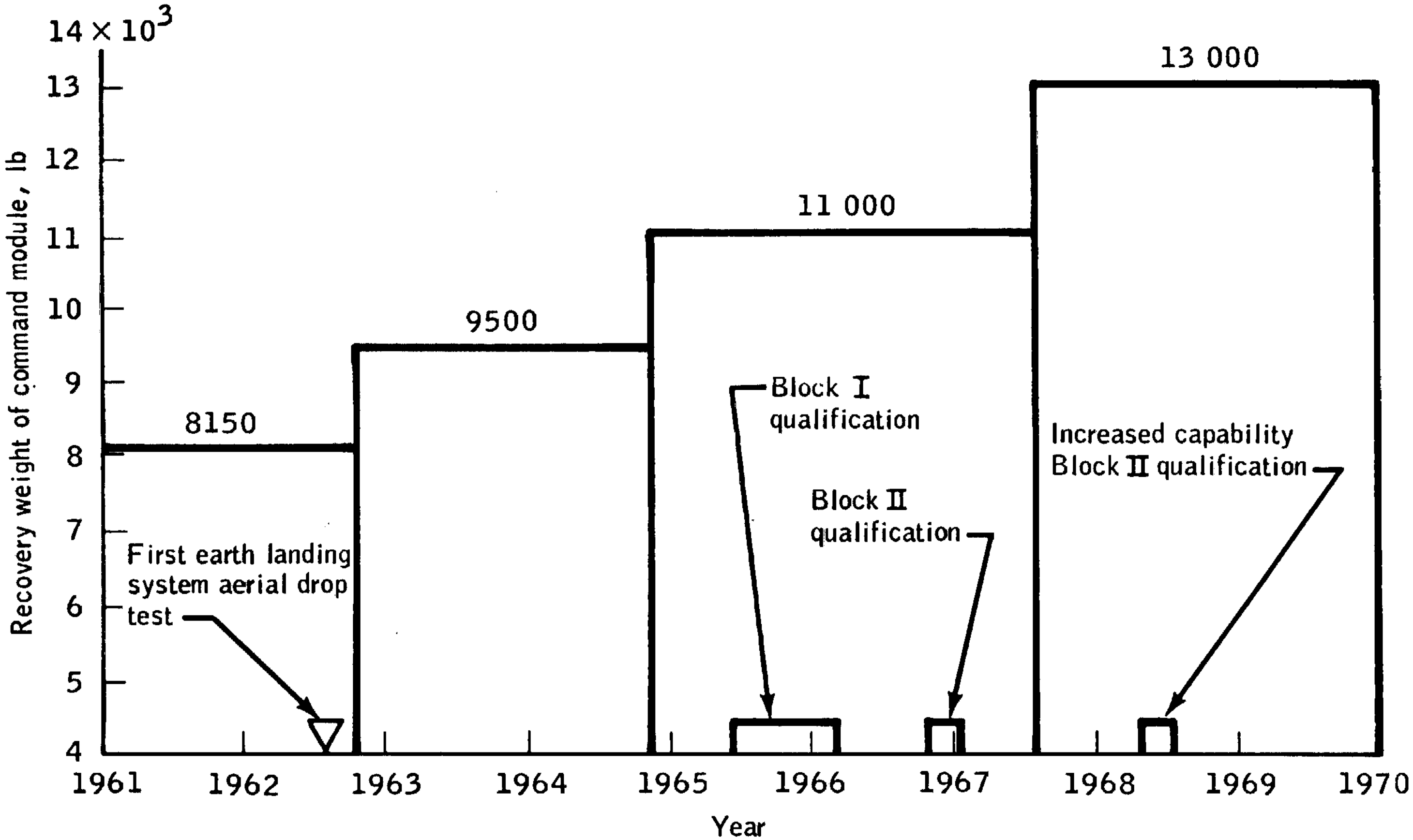


Figure 4-6.- Increases in command module recovery weight.

pilot parachute mortar assembly was mounted in the forward heat shield and was activated by the same signal that initiated the forward heat shield jettisoning devices. On Apollo 4, small burn holes were found in the canopy of a recovered main parachute. Investigation showed that the holes were caused by oxidizer expelled from the command module reaction control system during descent. The condition was corrected by controlling the ratio of fuel to oxidizer loaded on the command module to ensure that the oxidizer would be depleted before the fuel. Although the fuel (monomethylhydrazene) does not degrade nylon, the excess fuel condition was later found to be hazardous as well. One of the main parachutes collapsed during final descent of the Apollo 15 command module. Investigation showed that the most probable cause of the failure was burning fuel coming in contact with the parachute fabric riser. This condition was corrected on the final two missions by retaining excess propellants aboard the command module for normal landings. Also, the propellants were loaded so that there was a slight excess of oxidizer to allow for the low-altitude abort possibility. These problems are discussed in greater detail in references 4-27 and 4-28.

4.4.4.2 Docking mechanism.- The announcement that the lunar landing mission would be accomplished by using the lunar orbit rendezvous technique established the requirement for a docking system that would provide for joining, separating, and rejoining two spacecraft, as well as allowing intravehicular crew transfer. In addition, the Apollo program schedules required that a docking system be selected approximately 2 years before the first Gemini docking mission.

Many design concepts were evaluated, including the Gemini design which was rejected because of its weight. Types of the designs considered included both impact, or "fly-in," systems and extendible systems. The type selected was an impact system consisting of a probe mounted on the forward end of the command module and a drogue installed on the lunar module. The configuration of the Apollo docking system is shown in figure 4-7.

Design of the Apollo docking system began in December 1963 and evolved through a rigorous program of development tests, performance analyses, design studies, and qualification tests. Although many problems were encountered during the development period, most were relatively minor.

Perhaps the primary disadvantage of the system was that it blocked the crew transfer tunnel and, therefore, had to be removable. The original design philosophy had been to simplify the design and reduce the weight of the system. This required that all functions be performed manually by the crew using a special tool or wrench. However, to meet a subsequent requirement to simplify the crew/hardware interface, the complexity of the probe was increased by providing integral, low-force actuation devices, thus reducing the number of manual tasks. These changes were implemented in 1967, after the development test program and after some of the qualification tests of the basic probe assembly had been performed. The development and testing of the system are described in greater detail in reference 4-29.

The docking system was used successfully on nine Apollo missions, as planned. Docking system anomalies occurred only on the Apollo 9 and Apollo 14 missions. During the Apollo 9 mission, difficulties were encountered in undocking the command module from the lunar module and in preparing for lunar-module-active docking. Postflight ground testing demonstrated that both conditions were related and were inherent normal features of the docking probe. The undocking procedure was modified to preclude recurrence of these difficulties. On the Apollo 14 mission, six docking attempts were required to successfully achieve capture latch engagement during the translunar docking phase of the flight. Although the docking system performed successfully for the remainder of the mission, the docking probe was stowed in the command module after lunar orbit rendezvous and was returned with the command module so that a thorough investigation could be conducted. The results of the investigation disclosed two possible causes for the docking problem — one related to the design and one attributed to foreign material restricting mechanical operation. Although a minor design modification was incorporated to preclude such a failure mode for future missions, most evidence indicated that foreign material was the cause of the Apollo 14 anomaly. Additional details of these anomalies are given in references 4-29 and 4-30.

4.4.4.3 Crew support/restraint and impact attenuation systems.- These systems consisted of (1) a three-man couch assembly used to physically support the crew, especially during launch, entry, and landing; (2) a restraint system with a single buckle release; and (3) a shock attenuation system that held the couch in position throughout a mission but allowed couch movement if landing impact forces exceeded a safe level. The attenuation system was developed, primarily, to protect the crewmen in the event of a land landing.

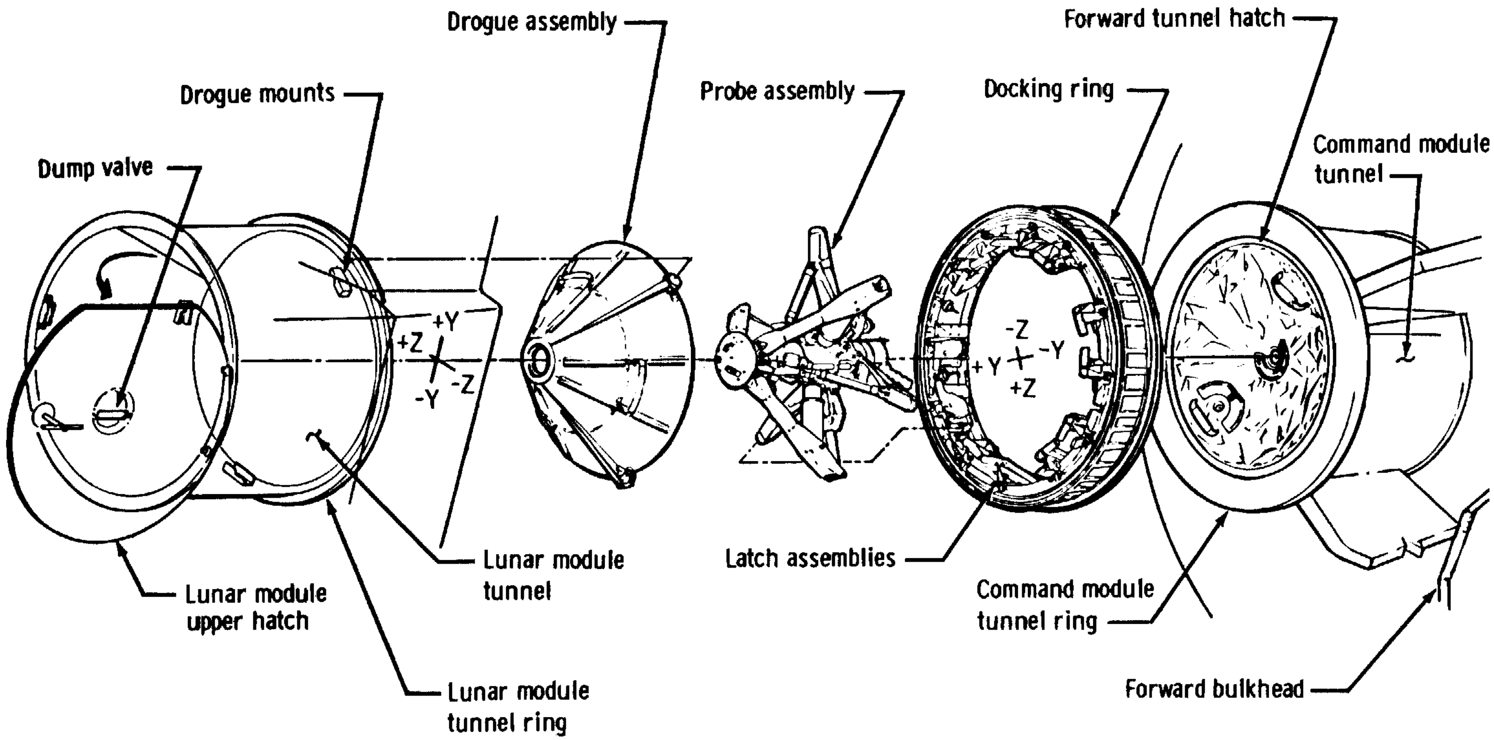


Figure 4-7. - Major assemblies of the docking system.

The original design requirements for the couch and restraint system were based upon the premise that the crewman should be held as rigidly as possible, the then existing philosophy of human impact protection. The prototype couch designed according to this philosophy was excessively massive and impaired the crew's inflight mobility. Subsequent testing reduced the requirement for rigid restraint of the crewman within the deceleration loads specified for the Apollo spacecraft crew couch. The result was a change from the contoured couch concept used in Mercury and Gemini to a universal couch that would fit all crewmen within the 10th- to 90th-percentile sizes. The first couch designed to the new requirements was flown on the Apollo 7 mission.

The Block II redefinition of the Apollo spacecraft emphasized the requirements for more work volume to allow an increase in intravehicular mobility and an open center aisle for side-hatch extravehicular activity by a suited crewman wearing a portable life support system. These requirements could not be met without a major redesign of the unitized couch. Therefore, a new foldable couch was developed and used for all manned missions after Apollo 7.

During the Block II redefinition, because the location of the launch pad and the height of the launch vehicle resulted in a high probability of a land landing from a launch pad abort or a very low altitude abort, the crew couch was made to provide crew protection for land landing. Because the command module did not have facilities for limiting the landing impact, attenuators were required to support the crew couch during all mission phases and to limit the energy transmitted to the crewmen during landing impact. Development efforts resulted in a double-acting, cyclic-strut attenuator which used a unique concept of cyclic deformation of metal to absorb energy. Energy absorption was accomplished by rolling a ring of metal between two surfaces with a separation distance of less than the diameter of the ring thereby causing the ring to continually deform as it rolled. More detailed information on the design of the attenuation system may be found in reference 4-31. Several drop tests were performed to provide a better understanding of the dynamics of the couch and attenuation systems. Data obtained from the tests permitted refinement of the initial impact load to an acceptable rate of acceleration for crew tolerance.

The folding, stowing, and reassembling of the couch in flight were achieved without problems on all missions except Apollo 9 and Apollo 16. During these missions, the crew had some difficulty in reassembling the center body support of the couch.

4.4.4.4 Uprighting system.- Early studies of the command module showed that it had two stable flotation attitudes: stable I (vehicle upright) and stable II (vehicle inverted). The stable II attitude could be attained either by landing dynamics or by postlanding sea dynamics. Allowing the command module to remain in the stable II attitude for more than several minutes was undesirable primarily because the postlanding ventilation system and the location aids were inoperative. The command module could not be configured to be self-righting and still maintain an acceptable aerodynamic lift-to-drag ratio. Therefore, a requirement was established to provide a means of uprighting the command module.

The selected design consisted of three inflatable bags located on the upper deck of the command module, two air compressors, and the associated plumbing and wiring. When use of the system was required, a crewman initiated inflation of the bags by turning on the air compressors. By this action, ambient air was pumped through a series of valves to each of the bags.

In addition to the overall weight increase, a center-of-gravity shift resulted from the changes made to the command module after the Apollo I fire. Full-scale performance definition tests required by these changes showed that the uprighting capability of the Block II command module was marginal with the two Y-axis bags inflated (one on each side of the upper deck as shown in fig. 4-8). Moreover, a combination of an inflated Y-axis bag and the Z-axis bag (on the side opposite the hatch) resulted in a roll of the command module about its X-axis to a new stable position where uprighting did not occur. Development tests were conducted at the Manned Spacecraft Center to investigate different suspension systems for the bags and to investigate the ability of a smaller Z-axis bag to reduce the roll problem and provide enough buoyancy to assure uprighting. Also, tests were performed to determine the feasibility of two crewmen lowering the center of gravity by moving from the couches to the aft deck. As a result of these tests, the uprighting system was redesigned to provide uprighting capability with any two bags inflated after two crewmen had moved aft. The final configuration was capable of uprighting the command

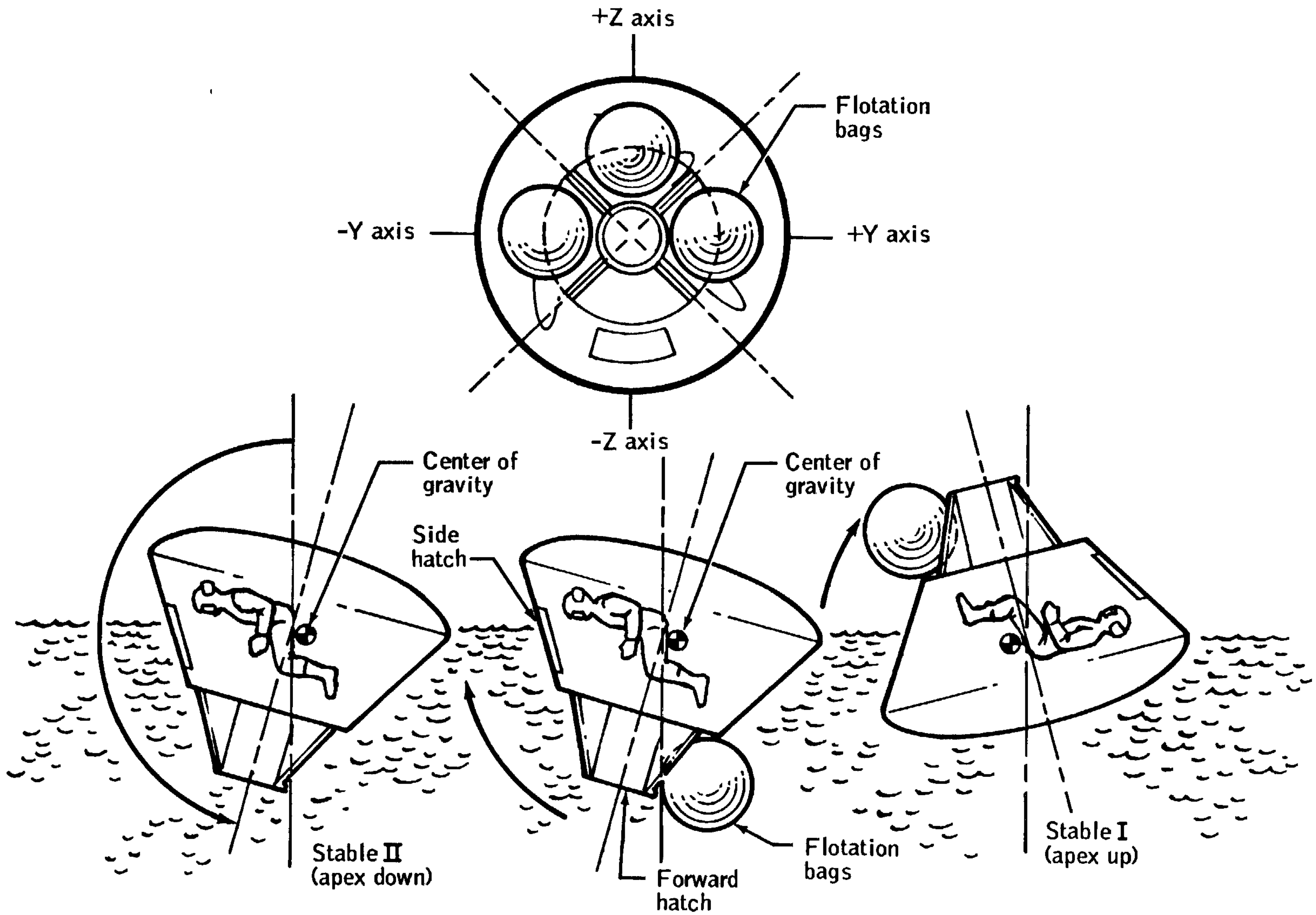


Figure 4-8.- Command module uprighting system.

module in 5 minutes if both compressors and all three bags were operative. With either a failed bag or compressor, 12 minutes was the maximum time required for uprighting. The system could not upright the command module if both a bag and a compressor failed.

The Block II spacecraft was much less stable during water landing than the Block I spacecraft. This lack of stability is attributed to the higher center-of-gravity locations at landing for the Block II spacecraft. All the Block II command module landing centers of gravity and attitudes are plotted on the uprighting capability curve shown in figure 4-9; also shown for reference is the center of gravity of the Block I command module flown on the Apollo 6 mission.

Five of the Block II spacecraft went to the stable II attitude and were uprighted by three bags in approximately 5 minutes. No problems with the system were encountered. The Apollo 7 command module would have been prevented from uprighting if one of the three bags had failed. While the vehicle was in the stable II attitude, water seeped through a faulty hatch valve, and the tunnel was flooded with approximately 400 pounds of water. As can be seen in figure 4-9, the flooded tunnel adversely affected the command module center of gravity; however, because all three bags inflated, the vehicle uprighted. The hatch valve design was changed for all subsequent spacecraft. Additional information on the development and performance of the uprighting system is given in reference 4-32.

4.4.4.5 Side access hatch.- The original Apollo spacecraft side hatch was configured as shown in figure 4-10. An outer ablative hatch provided thermal protection during entry through the earth's atmosphere and an inner pressure hatch sealed the cabin. With this two-hatch design, the hatches maintained the continuity of the structure for predicted loads, thereby reducing the vehicle weight. Although the hatch design fulfilled the program requirements relative to normal ingress-egress and emergency egress, the hatches were awkward to handle in a one-g environment since they were not hinged. In addition, there was no provision to open the inner hatch with the spacecraft pressurized. The tragedy of not having this requirement was demonstrated in the disastrous Apollo I fire.

In the period following the fire, the command module main hatch was redesigned to provide the single-piece, hinged, quick-opening hatch shown in figure 4-11. Although much heavier and more complex, the redesigned main hatch was used without difficulty on all of the Apollo manned missions. Details of the design and development of the hatch are given in reference 4-33.

4.4.4.6 Experiment deployment mechanisms.- To accommodate orbital science equipment on Apollo 15, 16, and 17, one section of the service module was modified to allow installation of a scientific instrument module. The modules for the three missions included a variety of equipment such as cameras and spectrometers. Two of the modules contained a deployable subsatellite. Deployment devices were developed for all three modules to move certain instruments away from the contamination cloud that surrounded the spacecraft or to extend antennas. Figure 4-12 is an artist's concept of the spacecraft in lunar orbit as configured for the Apollo 15 and 16 missions. Figure 4-13 shows the Apollo 17 spacecraft configuration. Problems experienced with several of the deployment mechanisms during flight are discussed in section 3.3.

4.4.5 Cryogenic Storage System

A multiple-tank cryogenic fluid storage system mounted in the service module provided gaseous oxygen and hydrogen to the fuel cell power generation system and metabolic oxygen to the crew via the environmental control system. The system for missions through Apollo 13 contained two oxygen tanks and two hydrogen tanks. This design provided for an emergency return to earth in the event of the loss of a hydrogen tank, an oxygen tank, or both. For Apollo 15, 16, and 17, a third tank was added for both hydrogen and oxygen storage to provide for more extensive operational requirements as well as the contingency requirement. The Apollo 14 system contained only two hydrogen tanks, but a third oxygen tank was added for redundancy after the failure of the Apollo 13 system.

The storage of cryogenic hydrogen and oxygen required judicious selection of pressure vessel materials. A materials screening program led to the selection of type 5Al-2.5 tin-titanium alloy for the hydrogen storage and Inconel 718 alloy for the oxygen storage. These materials were selected because they had attractive combinations of weight, strength, and ductility, and were compatible over the operating temperature ranges.

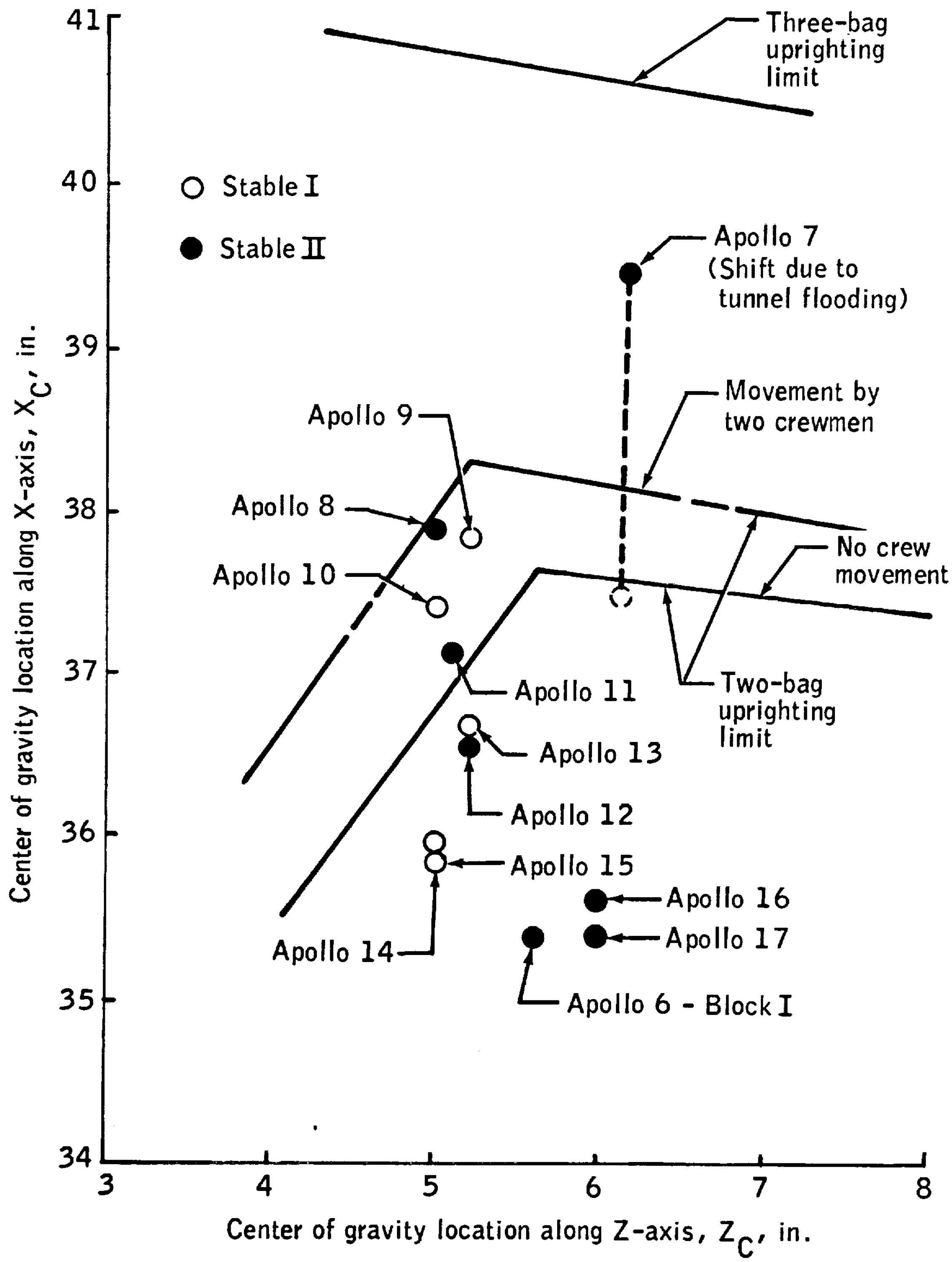


Figure 4-9.- History of landing center of gravity locations and landing attitudes.

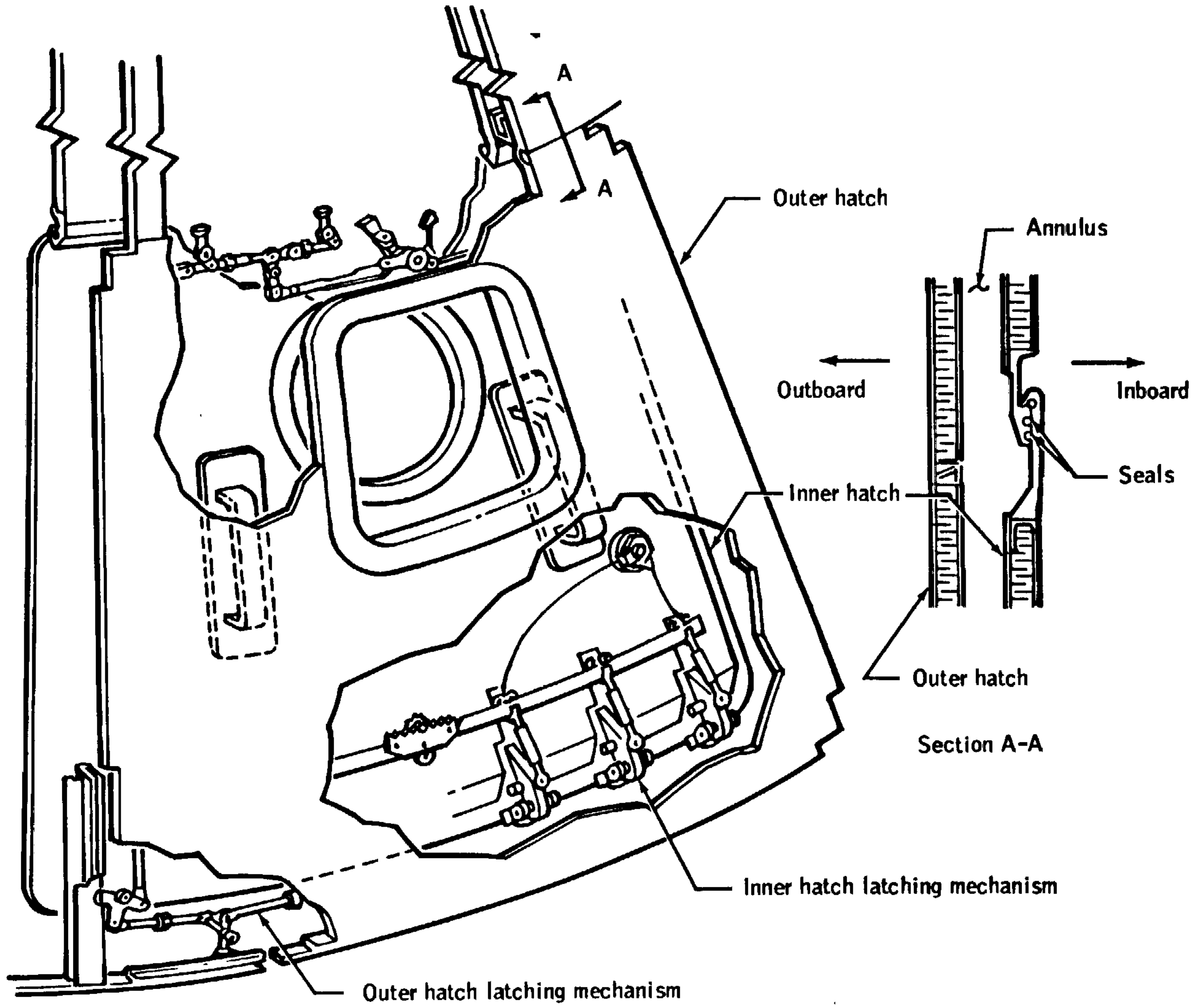


Figure 4-10.- Original side access hatch mechanism.

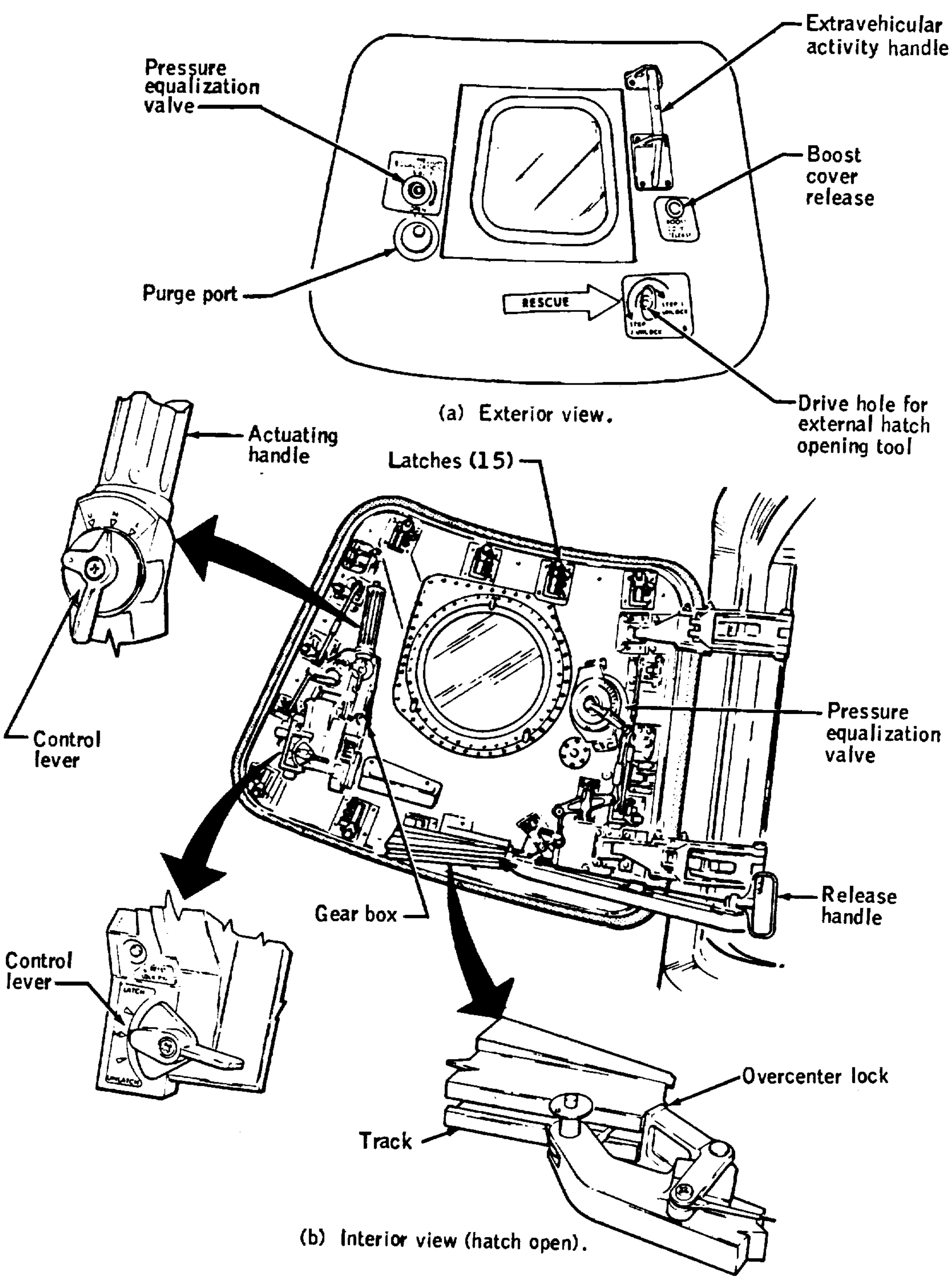
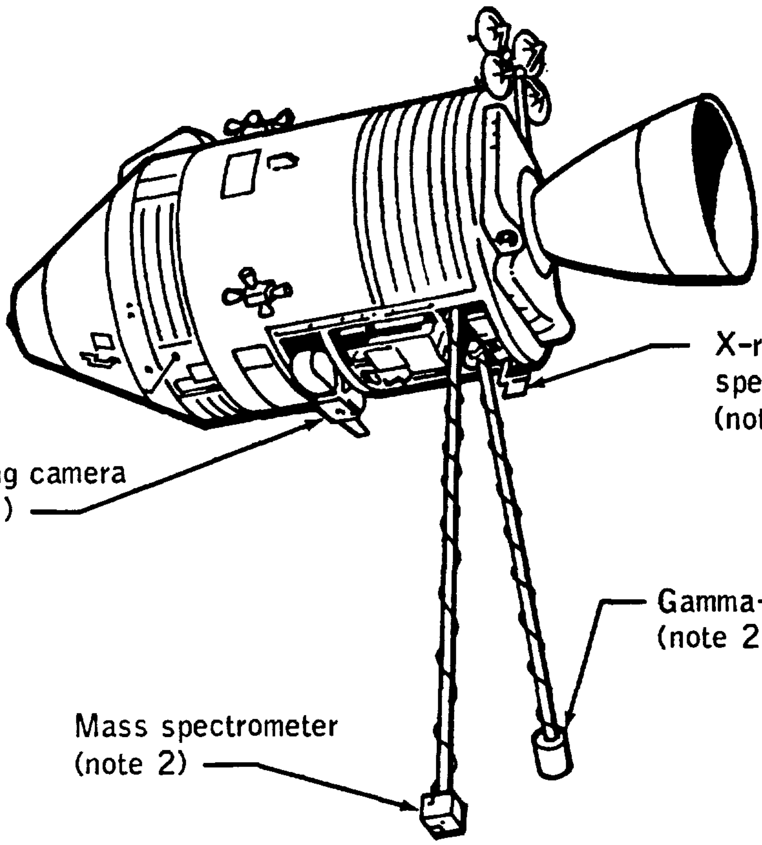


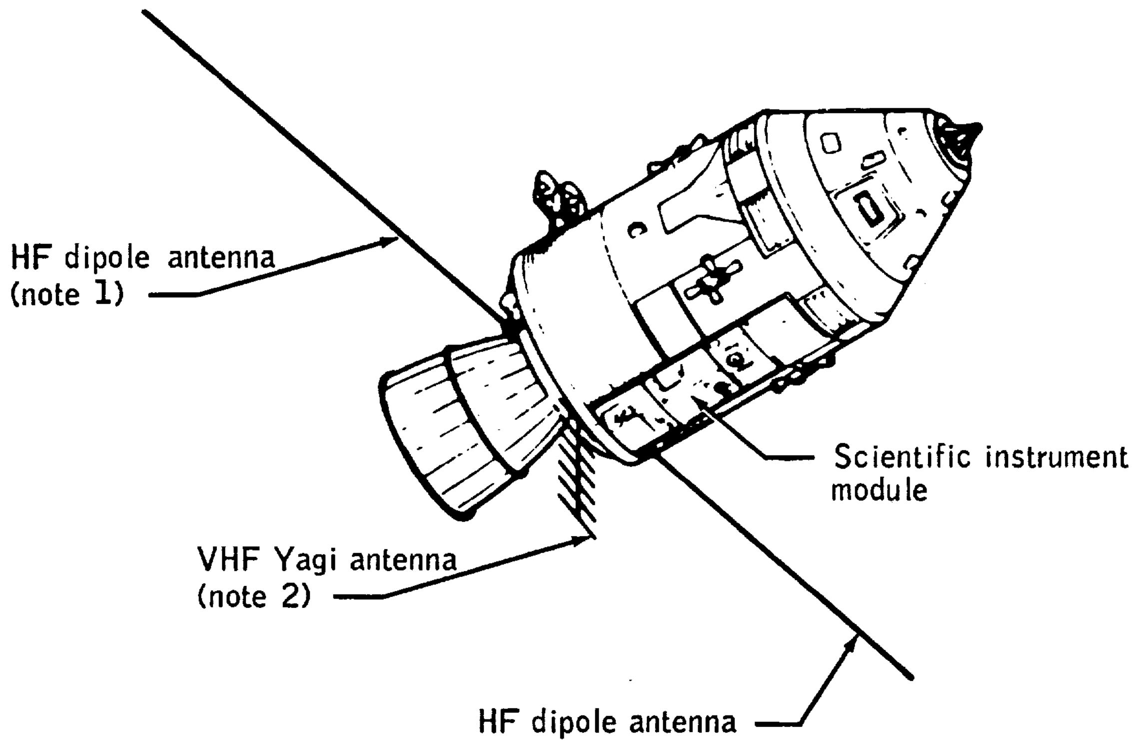
Figure 4-11.- Block II side hatch configuration.



Notes

1. Mapping camera extended and retracted on rails with linear ball bushing, driven by screw with ball nut.
2. Gamma-ray spectrometer and mass spectrometer extended and retracted on spiral-wrapped steel tape booms, driven by dual tape reels.
3. Experiments were protected from optical and thermal control surface degradation during firing of adjacent service module reaction control system firings by mechanically actuated covers.

Figure 4-12.- Apollo 15 command and service module in lunar orbit configuration.



Notes

1. HF dipole antenna extended and retracted as interleaving steel tapes, driven by dual tape reels. Combined length when extended was approximately 80 feet.
2. VHF Yagi multiple element antenna extended once and locked as a rigid, spring-loaded, hinged assembly. Not retractable. Extended length 103.8 inches.

Figure 4-13.- Apollo 17 command and service module with antennas extended for lunar sounder experiment.

Several titanium alloy pressure vessel problems occurred in the early developmental stages. These were (1) overly large grain size (which was eliminated by a vendor change) and (2) premature failure during proof-pressure testing caused by a phenomenon known as creep. Increased wall thickness of the pressure vessel allowed certification of the vessel for flight. Other problems resulted from hydride formation on various welds, dissimilar metals joining, and quality control of electron-beam welding. In all cases, a materials or process change was found to adequately resolve the problem.

Uniform depletion of the tank content was necessary so that, at any time during a mission, emergency quantities of fluid were available in each tank. Equal depletion was maintained by internal heaters. The original design for the heaters was a concentric aluminum sphere that was perforated to reduce weight. The heater element was a high-resistance film (electrofilm) sprayed over the aluminum sphere.

High heat rates from small areas can result in zones of fluid adjacent to the heater with significant temperature and density gradients. Vehicle accelerations can suddenly mix these thermally stratified zones and, under some fluid conditions, significant pressure decays can result. The potential problem of thermal stratification was circumvented by the installation of a fan and heater combination instead of using the coated aluminum spheres. In this design, a fan was installed at each end of a perforated, cylindrical tube, and the heater element was brazed in a barberpole manner around the tube. As a result of the Apollo 13 failure, however, the fan motors were removed from the tanks to reduce potential ignition sources (fig. 4-14). In the final configuration, heat was transferred by natural convective processes.

The method of insulating the tanks was developed through extensive analysis and was optimized by a comprehensive test program. Tests were conducted on removable outer shells that were clamped together; then the entire assembly was placed in a vacuum chamber. This configuration permitted rapid modification of the test article. These tests led to the conclusion that a vapor-cooled shield was required to achieve the specified thermal performance. The vapor-cooled shield provided an intermediate cold boundary layer within the insulation. The oxygen tank had eight sequences of insulation, and the hydrogen tank had 28 sequences. One sequence consisted of six layers: three of aluminum foil (each 0.0005 inch thick), two of paper, and one of fiberglass. All of the tanks were vacuum jacketed. A monocoque outer shell was selected, and a thickness of 0.020 inch was found to withstand the buckling stresses brought about by the 1-atmosphere load.

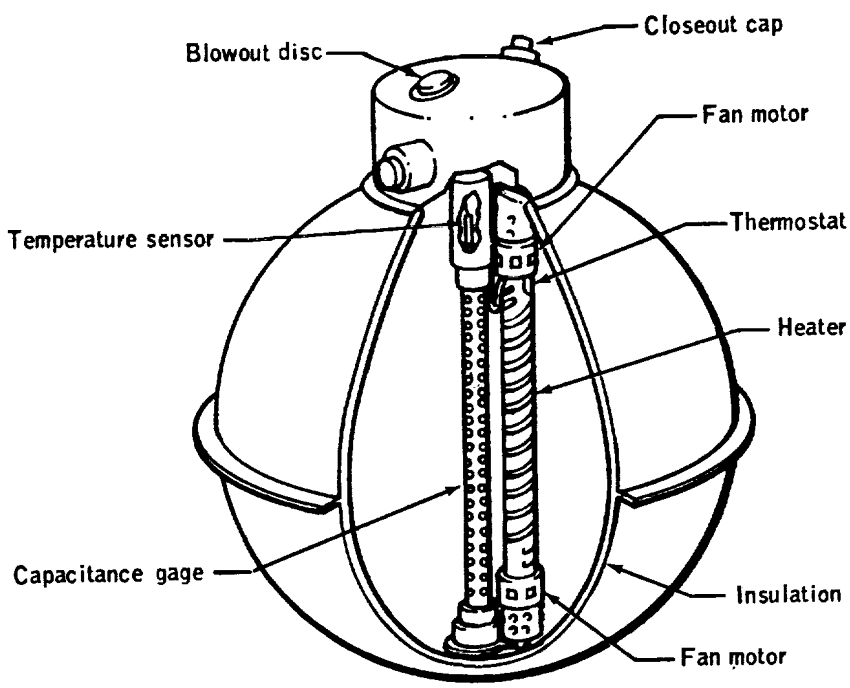
By far the most serious flight problem was the one that occurred during the Apollo 13 mission when oxygen tank 2 failed at almost 56 hours into the mission and caused the loss of the entire cryogenic oxygen system. An accident investigation board determined that two protective tank heater thermal switches failed closed during an abnormal detanking procedure prior to flight. Subsequent fan motor wire insulation damage caused a fire in one of the oxygen tanks and subsequent loss of the system. The changes made as a result of the investigation, in addition to the elimination of the fan motors, included reducing or eliminating internal materials with relatively low burning points (such as magnesium oxide, silicone dioxide, and Teflon).

The development of the cryogenic storage system resulted in significant technological developments for cryogenic applications, particularly in fabrication and welding of pressure vessel shells, metallurgy associated with titanium creep and hydride formation, application of vapor-cooled shields in high-performance insulation, and vacuum acquisition and retention. Most of these advances are directly applicable to other required cryogenic developmental programs. Additionally, preflight analytical predictions and subsequent correlations with flight data have contributed much information on heat transfer and stratification of cryogens at low-gravity levels. Reference 4-34 provides more detailed information regarding the development and performance of this system.

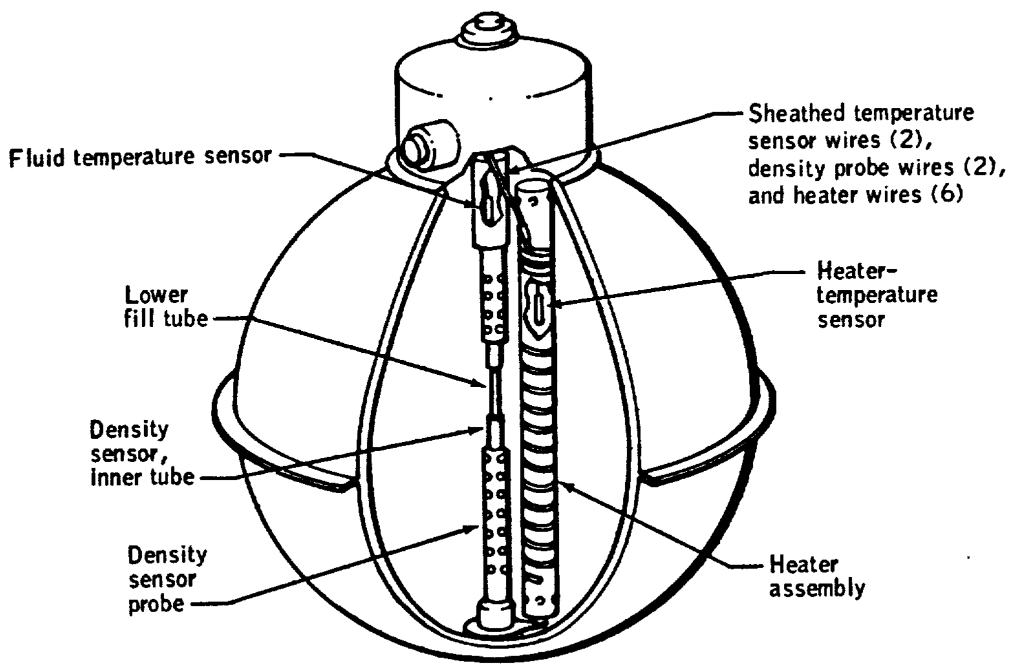
4.4.6 Electrical Power System

The electrical power system consisted of a fuel cell power generation system, a battery power system, and a power conversion and distribution system. The development and performance of each system is discussed.

4.4.6.1 <u>Fuel cells</u>.- The fuel cells provided all electrical power required by the command and service module from launch to command module separation prior to entry, although batteries



(a) Configuration before Apollo 14.



(b) Configuration for Apollo 14 and subsequent spacecraft.

Figure 4-14.- Oxygen tank.

were available for power augmentation such as might be required during service propulsion engine firings.

Before the selection of a power system to meet the requirements of the Apollo program, various nuclear, chemical, and solar energy devices were considered. The fuel cell system was selected because of its favorable developmental status, relatively light weight, and great operational flexibility. Following the selection of fuel cells for the primary power generation system, the mission electrical energy requirements were defined and specified as 575 kilowatt-hours of energy from three fuel cells at a minimum rate of 563 watts per hour and a maximum rate of 1420 watts per hour per fuel cell.

The system contained three fuel cell modules, each having four distinct sections: an energy conversion section (the basic cell stack), a reactant control section, a thermal control and water removal section, and the required instrumentation. Figure 4-15 shows one of the modules. The fuel cells consumed hydrogen and oxygen from the cryogenic storage system and produced electrical energy, water, and heat. The electrical energy was produced at a nominal 28 volts dc and was distributed, conditioned, and used throughout the command and service module. The water was stored in tanks in the command module and used for drinking and cooling. The heat produced by the fuel cells was rejected by means of radiators around the upper part of the service module.

The available fuel cell technology at the beginning of the program was inadequate to fabricate an operable system that would be reliable under the expected mission conditions. The more significant problems encountered in the development of the flight system are discussed.

An early developmental problem was leakage of electrolyte at the periphery of the unit cell (fig. 4-16). The electrolyte is highly concentrated potassium hydroxide, a very corrosive solution that is difficult to contain. The use of Teflon as a seal material and the incorporation of design improvements eliminated the leakage problem.

The two half cells (electrodes) that formed the single-cell assembly (fig. 4-16) were composed of dual-porosity sintered nickel formed from nickel powder that was pressed into sheets. The liquid-electrolyte/gas-reactant interface was maintained within the sintered nickel by means of a controlled 10.5-psi pressure differential between the electrolyte and the reactant compartments. If either the hydrogen or oxygen gas pressure was more than 2.5 psi below or 15 psi above the electrolyte pressure, a breakdown of the liquid/gas interface was possible. In the early design stages, many electrolyte leaks developed that allowed potassium hydroxide to enter the gas-reactant cavities. As a result, individual cells failed to maintain an electrical load. The manufacturing procedure was changed to obtain a more uniform porosity of the nickel electrodes, thus increasing the bubble pressure and decreasing susceptibility to flooding. Also, a coating of lithium-impregnated nickel oxide was added to the electrolyte side to inhibit oxidation. These modest improvements helped, but the fundamental problem of ground test cell flooding caused by gas pressure imbalance remained throughout the program. This ground test operational defect was minimized by improved ground support equipment, better gas distribution systems, improved test procedures, and more careful handling.

The fuel cell showed signs of internal shorting during qualification testing. The cause was the formation of nickel dendrites between the electrodes due to electrochemical reaction. The reaction rate was found to be dependent upon temperature and time. Therefore, operational procedures were changed to minimize fuel cell operation during cell buildup and launch checkout. The problem did not recur after this change.

An accumulator was provided as part of the water/glycol coolant system to maintain a constant coolant pressure without regard to the volumetric changes due to coolant temperature variations. This pressure was controlled by a flexible bladder that imposed a regulated nitrogen blanket pressure on the coolant system. During ground tests using boilerplate spacecraft 14, thermal expansion of the water/glycol extended the accumulator bladder to its dimensional limit, causing the coolant pressure to increase. A larger accumulator was added to production fuel cells and no problems were encountered thereafter.

The electrolyte, 80 percent potassium hydroxide, was a porous solid at ambient temperature. Therefore, small quantities of reactant gases could permeate the electrolyte as it dried and hardened during shutdown of the fuel cell. The early method of shutdown depressurization was to open the reactant-gas purge valves and thereby rapidly reduce cell pressure. When the cells were rapidly depressurized, the expansion of the trapped gases could break the bond between the electrode

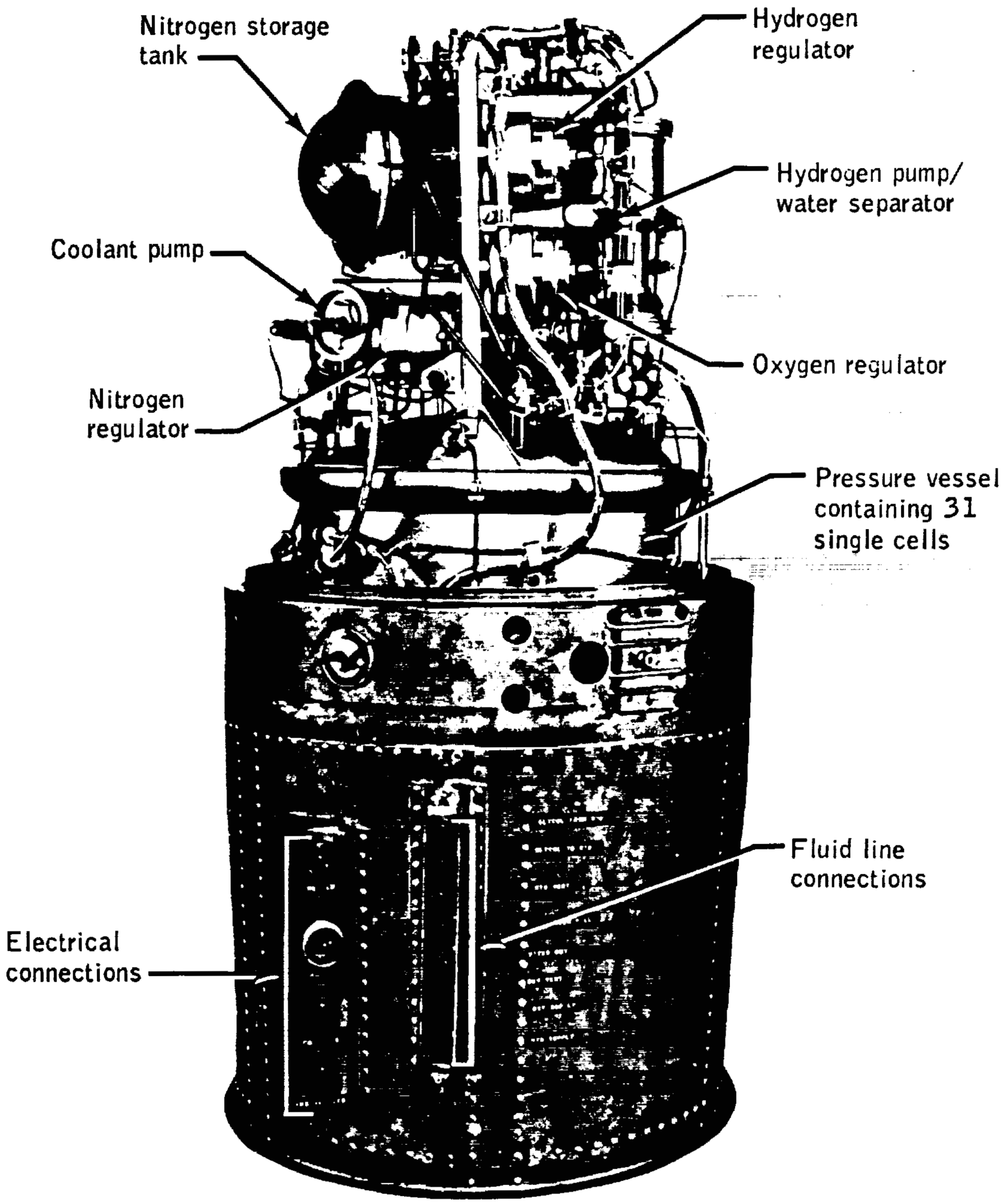


Figure 4-15.- Apollo fuel cell module.

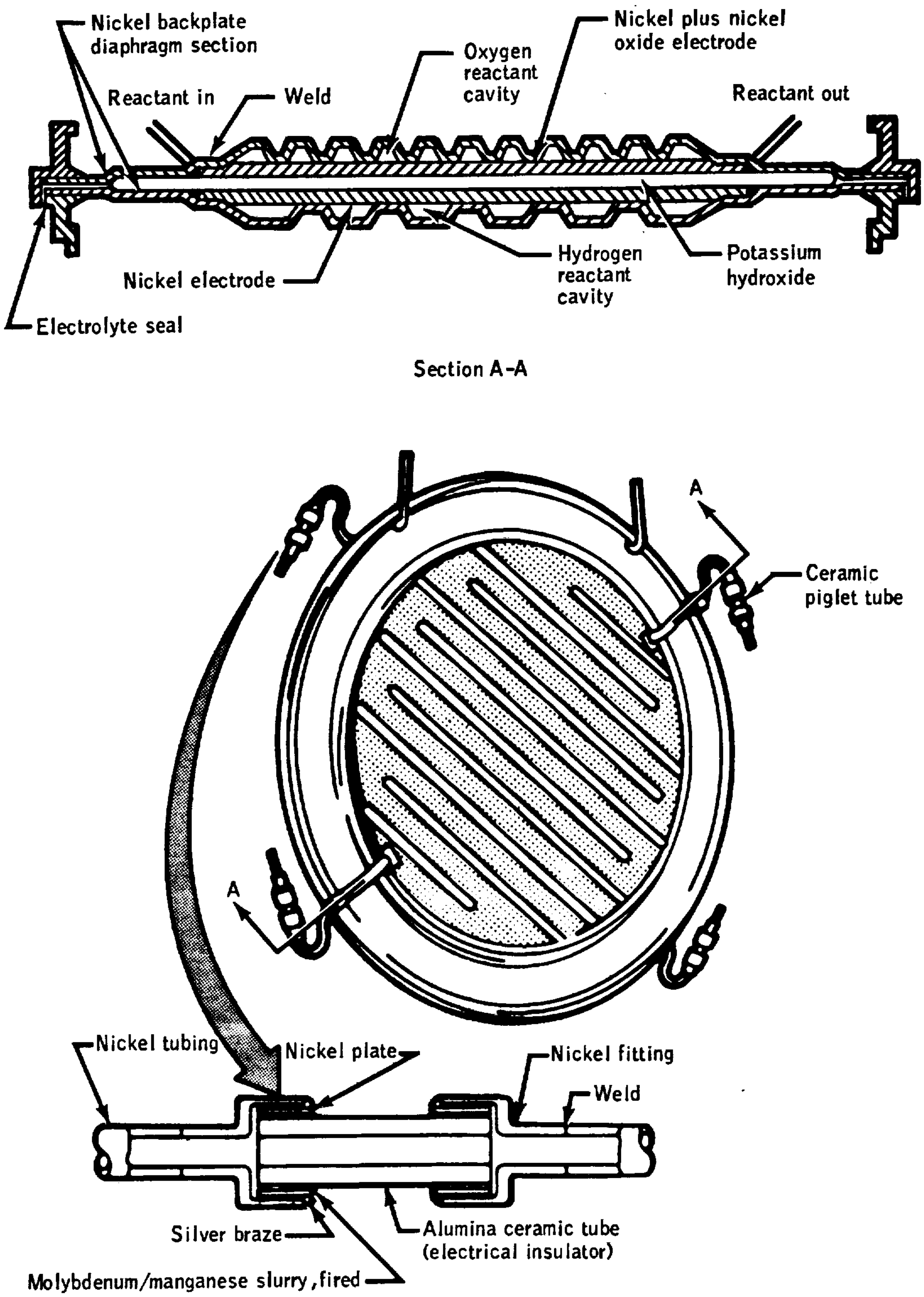


Figure 4-16.- Single-cell assembly.

and the solidified potassium hydroxide. On restart of the cell, the trapped reactant gas formed a bubble between the electrolyte and the electrode. This reduced the active electrode area and caused a decrease in performance. Careful adherence to a controlled, slow repressurization of the cell reactant gases eliminated the bubbles, because the trapped gases could diffuse out slowly from the solidifying electrolyte.

During acceptance tests, several of the water/glycol pumps tended to stick on initial fuel cell start. Examination of failed pumps showed that, during a final flush-and-dryout procedure before storage, a residue was left on the shaft. Therefore, the shaft could not rotate because the pump had a low (4 inch-ounce) starting torque. After the water/glycol pumps were started, the residue would dissolve and the failure did not occur during operation. A new rinse-and-dryout procedure was adopted and the residue problems ended.

Two reactant purge ports, one for hydrogen and one for oxygen, were provided on each fuel cell to allow purging of impurities (nonreactant gases) that accumulated in internal cell reactant cavities. During testing with airframe 008, water vapor condensed and froze at the purge opening under extreme thermal conditions, preventing further hydrogen purging. To correct the condition, two heaters, connected electrically in parallel for redundancy, were added to subsequent flight vehicles. The heaters were activated 20 minutes before a fuel cell hydrogen purge and turned off 10 minutes after the purge was terminated.

Only one fuel cell problem was encountered during the Block I command and service module flights. Cooling system temperature excursions observed during the AS-202 mission were found to be caused by inadequate radiator coolant loop servicing, which permitted gas bubbles to remain in the system and caused the coolant pump to cavitate. Improved servicing and checkout procedures corrected the problem.

Condenser exit temperature problems were experienced on most of the early Block II flights through Apollo 10. The fuel cell condenser served as a means of controlling the humidity of the fuel cell hydrogen loop; the condenser exit temperature determined the position of a secondary coolant loop bypass valve and was, therefore, a prime determinant of the thermal condition of the fuel cell. The combination of coolant, corrosion inhibitors, and aluminum plumbing caused the formation of a gelatinous product over long dormant periods. The formation of this product in the coolant loop on Apollo 7 and 9 affected secondary loop bypass valve performance. Servicing procedures were revised to service the coolant system at the Kennedy Space Center as late as possible prior to launch and to sample the coolant loop fluid. The radiators were hand-vibrated and flushed if any of the coolant samples were questionable. The fuel cells for Apollo 11 and subsequent spacecraft were retrofitted with Block I bypass valves which were shown by tests to be less susceptible to contamination than the improved Block II valves.

Another condenser exit temperature problem was observed on the Apollo 10 mission. The condenser exit temperature oscillated well out of the normal control band during lunar orbit and caused repeated caution and warning system alarms. Investigation showed (1) that the fuel cell thermal control system was marginally stable under certain conditions of high loads and low radiator temperatures such as those experienced during lunar orbit dark-side passes and (2) that thermal oscillations could be induced if the system was adequately "shocked." This was simulated in ground tests by alternately stopping and starting the coolant pump while in the proper fuel cell operating conditions. Analysis determined that the shock, or trigger, for the inflight oscillations was the result of water slugging out of the condenser in large subcooled quantities rather than in the uniformly sized droplets that had always been observed in ground operations. Although nothing could be done to prevent this zero-gravity phenomenon from recurring (which it did, several times), procedures were developed to stop the oscillations when they occurred, and the circumstances necessary to develop oscillations were carefully avoided whenever possible. Temperature oscillations were not observed on flights after Apollo 10.

The ingestion of hydrogen gas into the drinking water caused discomfort to the crewmen until a hydrogen gas separator was developed and added to the drinking water system. This device removed a sufficient amount of hydrogen from the water so that it was no longer a serious problem to the crewmen.

The fuel cell proved to be a reliable and versatile electrical power generation device in the Apollo program. The fuel cell operated satisfactorily during spacecraft launch/boost vibration and in the space environment, and met all electrical demands imposed on it. When problems did occur, the redundancy of the fuel cells prevented catastrophic results, and the extreme operational flexibility of the system usually permitted operation in modes that obviated or minimized the likelihood of recurrent failures. Additional information on the fuel cell power generation system is contained in reference 4-35.

4.4.6.2 Batteries.- As stated previously, batteries were used to augment the fuel cells during periods of high current demand. Battery power was also used (1) to supply low-level loads that had to be isolated from the main buses, (2) to supply electrical power after jettisoning of the service module, and (3) to provide power for pyrotechnic devices.

The battery complement on manned missions through the Apollo 13 mission consisted of five silver-oxide/zinc batteries located in the command module. Three of these (entry-and-postlanding batteries) were rated at 40 ampere-hours each and were rechargeable. The remaining two (pyrotechnic batteries) were each capable of supplying approximately 2 ampere-hours of energy. (The specified capability was 0.75 ampere-hour.) As a result of the Apollo 13 cryogenic oxygen system failure, an auxiliary battery having a capacity of 400 ampere-hours was installed in the service module for the Apollo 14 and subsequent missions. This battery could have provided 12 kilowatt-hours of emergency energy and could have been connected to the command module main buses through the distribution system for fuel cell 2.

The requirements established for the Apollo command and service module batteries were well within the existing state of the art for batteries; hence, no unique problems were identified or experienced during battery development and qualification during short-time unmanned flights.

The only significant battery problems on the operational flights resulted from the use of a relatively new type of nonabsorbent separator (Permion 307) in the command module entry-and-postlanding batteries and from failure to verify the effectiveness of the battery-charging system for those batteries. These two factors jointly resulted in severe undervoltage on the command module main buses at command module/service module separation during the Apollo 7 mission. The final solution of these problems for the flight of Apollo 11 was achieved by reverting to the originally used absorbent cellophane separator material and by raising the output voltage of the command module battery charger. With the possible exception of an auxiliary battery in the unmanned Apollo 6 flight (there was insufficient data to prove a battery failure), no command and service module battery failure occurred in any flight. Reference 4-36 contains more detailed information on battery performance.

4.4.6.3 Power conversion and distribution.- Two systems for power conversion and distribution were designed and flown during the Apollo program. The first was used in the launch escape vehicle test program conducted at the White Sands Missile Range. The design philosophy for this system was based upon returning performance data for evaluation. Thus, the design was quite simple and NASA facilities were used for fabrication. The second was the operational system used on the manned Apollo flights. Since high reliability was required to assure crew safety, this system was more complex and was fabricated by the command and service module contractor.

a. System for early development flights: Off-the-shelf hardware components that had been qualified on previous space programs were used in the launch escape vehicle test program. This assured early delivery and low cost as well as giving a high probability that these assemblies would pass the Apollo environmental qualification tests after having been installed in higher level assemblies. As an example, the relays, connectors, wire, current shunts, and fuses were qualified in the Mercury program. The fuse holders were qualified in the X-15 aircraft program.

All loads were protected by fuses except those that were essential to the primary mission objectives. The philosophy was that, if a load was of secondary importance and could short circuit, a fuse should be in the line to remove the shorted load from the bus, thereby allowing the other loads to operate properly. If the load was of primary importance, however, a short circuit could cause the loss of the primary mission objectives and, thus, it did not matter whether the load was fused or not. Also, since reliability analysis showed that a fuse would be one more series element that could fail, the loads of primary importance were not fused. In addition to the hardware selection and circuit design considerations, redundancy and fail-safe techniques were used, good wiring practives were followed, and good quality control was maintained. No flight failures occurred in this system.

b. Operational system: The operational system used on the manned Apollo flights took longer to design because, with the manned mission requirement, a stricter design philosophy was followed to assure crew safety.

Although a great deal of the required system reliability was achieved through the design itself, performance was enhanced by extensive testing of individaul components, separate assemblies, the total distribution system, and the entire vehicle. In addition to evaluation and design proof tests, a random production sample of each component was subjected to a series of electrical, mechanical, and environmental tests before certifying that part for flight. Finally, various selected parameters of each component or assembly were measured during acceptance testing before installation of the components into higher level assemblies.

The direct-current distribution, designed around two isolated main buses, received power from any combination of three fuel cells and/or three command module batteries. Redundant loads were connected to each bus, nonredundant critical loads were connected to both buses through isolation diodes, and noncritical loads were connected to either bus as required to equalize the loads. This configuration prevailed until the Apollo 13 failure highlighted the need for the additional battery that was installed in the service module for the Apollo 14 through 17 missions.

Based on the experience of the Mercury and Gemini programs, wherein it was demonstrated that many of the inflight tasks did not need to be automated, automatic functions in the electrical power system were kept to a minimum. The only functions that were automated were those which had to be initiated faster than a crewman could react. For instance, the power system was designed to connect the command module batteries to the buses automatically in the event of a pad abort.

The alternating-current distribution system contained circuitry to disconnect a bus from its source automatically if the voltage became too high. This was necessary because electrical equipment, especially semiconductor devices, can be damaged by instantaneous excessive voltage. The alternating-current sensor and associated circuitry therefore monitored each alternating-current bus for voltage and current. If the voltage became too low or the current too great, the sensor signaled the crew, notifying them of the need for action. If the circuit sensed an abnormally high voltage, the circuit automatically removed the affected bus from the inverter and signaled the crew regarding the changeover.

Distribution of alternating current was achieved through a system similar to that of the direct-current system. Three static dc-to-ac three-phase inverters provided alternating-current for the vehicle, each phase furnishing 115-volts at 400 hertz. Although each of the inverters was capable of providing 1250 volt-amperes of power, more power than was required for the entire vehicle, three were installed for increased reliability. During normal operation, two inverters supplied power while the third inverter remained on standby.

Alternating and direct current were used to provide power to the battery charger used to charge the three command module entry-and-postlanding batteries. To provide maximum reserve power for emergencies and for recovery aids after landing, the batteries were recharged as soon as possible after each use.

Fuses, circuit breakers, and sensors were all used so that faulty loads could be removed from the bus and the sources protected from downstream failures. Mission success and crew safety demanded that failures not be allowed to propagate to other areas and that the sources and buses be protected.

The electrical power distribution system performed satisfactorily throughout the flight program. Further information may be obtained from reference 4-37.

4.4.7 Propulsion Systems

The command and service module propulsion systems consisted of the service propulsion system, used for major velocity changes, and two separate reaction control systems, one in the command module and one in the service module.

4.4.7.1 Service propulsion system.- Early requirements for the service module included vernier and main propulsion systems for a direct lunar landing profile. The main propulsion system was to consist of several identical solid-propellant motors which would provide thrust for translunar abort and lunar ascent. A separate module was to be designed that would provide for terminal descent. These requirements were changed early in 1962 to specify a single service module engine. Earth-storable liquid hypergolic propellants were to be used by the new system, which could include single or multiple thrust chambers. The service propulsion system was to be capable of providing for abort after jettison of the launch escape system, for launch from the lunar surface, and for midcourse corrections during earth return.

When the lunar orbit rendezvous mode was selected over the direct lunar landing mode in July 1962, the service propulsion system requirements were reduced to provide for midcourse corrections, lunar-orbit insertion, and transearth injection. The final service propulsion system design had a single pressure-fed-liquid rocket engine which used nitrogen tetroxide as the oxidizer and hydrazine (Aerozine-50) as the fuel. The propellants were stored in four tanks located in the service module. The tank pressurant gas was helium, which was supplied from two bottles located in the center bay of the service module. Isolation valves, check valves, and regulators for the helium supply system were mounted on a panel in one of the service module bays. A propellant utilization and gaging system was used to maintain the correct oxidizer-to-fuel ratio for the engine.

In the early stages of system development, materials and processes were investigated. Material-properties research was conducted to determine the emissivities of nozzle and nozzle-coating materials. Tube brazing and weld techniques were improved by means of propellant-metal compatibility studies and brazing-welding metallurgical investigations. Thrust chamber ablative materials were selected after the completion of laboratory tests that limited the materials list before thrust chamber testing. Laboratory studies were conducted on 42 potential thrust chamber material samples; the studies included high-temperature vacuum tests and thermal- and structural-properties investigations. Seal materials for propellant equipment were selected after investigation of elastomer and pseudoelastomer compounds to screen for propellant compatibility, swell, creep, resilience, and other seal properties.

Zero-gravity propellant motion problems were investigated by means of theoretical and experimental research in fluid mechanics. The goals of this research were new modeling and scaling techniques for earth simulation of zero-gravity effects on the propellant and an improvement in the understanding of fundamental phenomena.

The complete propulsion system was subjected to a test program using heavyweight test rigs and a flight-type system at the White Sands Test Facility. These tests were conducted at ambient conditions and explored the full range of potential system use. The engine was qualified primarily through simulated altitude testing at the Arnold Engineering Development Center.

Throughout the engine development and qualification phase, many configuration changes occurred as a result of knowledge gained in the test programs. One of the more significant changes resulted in the incorporation of a baffled injector to reduce the risk of combustion instability.

Inflight testing was the final phase of the service propulsion system development. Qualification of the system under all space-operational conditions was attempted during the ground test program. However, the impracticability of simulating all space conditions in ground tests prevented complete demonstration of system performance. Thus, the service propulsion system was used conservatively in the early flights. As the flight program progressed, the complexity of operating modes and system demands were increased.

Several notable problems were encountered during the flight program. First, the propellant gaging system, while operating as designed, was not matched to the system in a manner that allowed a direct reading of actual propellants without correction throughout the mission. This required interpretation of indicated quantities by system specialists and was a source of crew irritation on several missions. Secondly, incomplete bleeding of gas trapped in the engine and feedlines during propellant loading resulted in unusual start transients on the Apollo 8 mission. Improved engine bleed provisions were incorporated on later spacecraft. In another case, the engines were re-orificed to eliminate unbalance between the propellant flow rates. Prior to the re-orificing, the propellant utilization valve was used to correct the unbalance. These and other problems noted during the operational phase of the program are discussed in more detail in reference 4-38.

The most significant lesson that was learned from the service propulsion system development program was the need to first develop basic technology for propulsion systems before initiating full-scale hardware designs. Besides the anticipated technical problems such as engine performance and combustion instability, schedule delays were experienced during hardware development, and these delays generally were associated with the high reliability requirements of the Apollo program and the lack of experience with the propellants and their effects on materials.

4.4.7.2 Reaction control systems.- Initially, the reaction control system capabilities were to include attitude control, stabilization, propellant settling for the aforementioned vernier propulsion system, and minor velocity corrections. The system was to be pulse modulated and pressure fed, and was to use storable hypergolic propellants. When these requirements were changed to delete the vernier propulsion system, a requirement was added to provide (1) ullage maneuvers (propellant settling) for the service propulsion system and (2) a deorbit capability to back up that of the service propulsion system. The redundant system concept was also expanded such that the command module reaction control system consisted of two independent systems and the service module reaction control system consisted of four independent systems, each having a four-engine cluster (quad).

The basic design of the command and service module reaction control systems was not changed appreciably from the original concepts. The only major change to the service module reaction control system was to increase the propellant storage capacity of the Block II system by adding one additional fuel tank and one additional oxidizer tank to each quad assembly.

In each service module reaction control system assembly, high-pressure (4150 psia) helium was stored in a spherical titanium alloy tank. The helium flowed through two-way solenoid-controlled isolation valves to regulators. After being regulated to the desired working pressure (181 psia), the helium passed through check valves and into the gas side of the propellant tanks. Pressure relief valves were provided between the check valves and the propellant tanks to prevent overpressurization of the tanks. The propellant forced from the propellant tanks by the collapsing bladders flowed through solenoid-controlled isolation valves and in-line filter assemblies into the engine assemblies. Each of the four engines on each quad was a pulse-modulated, radiation-cooled, 100-pound thrust unit. The service module reaction control system also included heater assemblies and controls to maintain safe operating temperatures in the systems, many access ports for checkout and servicing, and an instrumentation system, including a propellant quantity gaging system, to monitor system performance.

The command module reaction control system was similar to the service module system with the following exceptions. It had two rather than four independent assemblies, each capable of providing entry control. The system also had pyrotechnic, normally closed, helium isolation valves rather than solenoid valves. These valves were opened just before entry and no provision was made for isolating the helium supply. To provide sealing of the system before use, burst-disk-type isolation valves were installed in the propellant feedlines between the tanks and the solenoid-type propellant isolation valves. The limited-life engines were ablatively cooled. The command module reaction control system also had provisions for interconnecting the two redundant systems. Additionally, the propellants and the pressurizing gas could be dumped rapidly in case of an abort.

Although none of the components were off-the-shelf items, most of them were state of the art. For these, the development program was rather straightforward and usually consisted of (1) tests of pre-prototype hardware to define the design, (2) a design verification test of prototype hardware to verify design adequacy, and (3) qualification tests to demonstrate the adequacy of production hardware.

In addition to the component tests, a considerable number of system-level tests was conducted. Several of the system-level tests constituted a part of the formal certification. The system-level evaluations included system performance demonstration tests, vibration tolerance demonstration tests, and thermal vacuum tests. A detailed discussion of the system-level test program is contained in reference 4-39.

A certification and qualification test program was conducted for each component in the command and service module reaction control system. These tests included a demonstration of the capability to withstand exposure to temperature, vacuum, vibration, shock, propellants, and acceleration conditions, and demonstrations of operational capability such as functional cycling, proof pressure tests, leakage tests, and pressure-drop tests. Tests were also conducted to demonstrate tolerance to particulate contamination and to determine the quantity of contaminants generated. Additionally, selected components were tested under conditions that were more severe than those expected during flight, including vibration to 1.5 times the normal qualification levels and pressurization to the component burst point. A number of problems encountered during these tests necessitated modifications or imposed operational limitations.

Two hardware failures occurred during flight missions in the service module reaction control system and five in the command module reaction control system. There were also five electrical-type failures, all on the command module reaction control system. Because many of these failures occurred on early missions that were flown at the same time that the qualification and system-level ground tests were being conducted, the failures were not unique to flight experience. Those failures experienced only in flight are discussed in the following paragraphs.

Apollo 7 postflight tests revealed that the command module reaction control system propellant isolation valves would not latch in the closed position. The tests showed that if the valve was closed at the time of system activation, the valve bellows were damaged to the point of causing the failure. The corrective action was to open the isolation valve before the systems were activated.

During Apollo 9 and several subsequent missions, some of the service module reaction control system propellant isolation valves inadvertently closed during separation of the spacecraft from the S-IVB launch vehicle stage. Investigative testing revealed that the pyrotechnic shock was sufficient to cause the valve to close but did not damage the valve. The valves were simply reopened and no further corrective action was required.

Another flight failure involved the interface between the reaction control system and the parachute system. As discussed previously, small holes were found in the canopy of a recovered main parachute on an early flight. These holes were caused by raw oxidizer which was expelled from the command module reaction control system after the fuel was expended during the propellant depletion firing after entry. (The firing was accomplished after the main parachutes were deployed.) On the Apollo 7 mission, the depletion firing was not accomplished and the excess propellants were left on board. For the Apollo 8 mission, the command module reaction control system was loaded with an excess of fuel so that, during the depletion firing, the oxidizer would be expended before the fuel; the firing was satisfactorily accomplished. On the Apollo 15 mission, several riser lines on one of the main parachutes failed. Investigative testing demonstrated that burning fuel from the depletion firing caused the parachute failure. Consequently, the Apollo 16 and Apollo 17 command modules were landed with the excess propellants on board.

The last corrective action brought about a hazardous situation that occurred during postflight deservicing of the Apollo 16 command module. On previous flights, essentially no residual propellants were left on board. However, the deservicing procedures used on these earlier missions were also used for the Apollo 16 command module, which had about 200 pounds of residual propellants on board. During the offloading of the oxidizer, an incorrect ratio of neutralizer to oxidizer resulted in an explosion that destroyed the deservicing cart. After Apollo 16, the deservicing procedures and ground support equipment were changed so that the fuel and oxidizer were put in separate containers and neutralization was accomplished at a remote site.

In retrospect, certain problem areas were common to many of the component development efforts. Recommendations to minimize the impact of the problems on future programs are as follows.

The initial component function design specifications often were more stringent than was necessary because actual requirements were not known. In some cases, the specification requirements were the projected limits of the state of the art at the anticipated time of use. As the requirements were defined more fully, there was hesitancy to relax the specifications, which might have resulted in some unnecessary and, perhaps, unfruitful efforts. An intensive effort should be made to define requirements accurately as early as possible. Also, as a relaxation in requirements become evident, the specification should be relaxed if cost or schedule savings can be realized.

A lack of compatibility of the system and its components with the propellants was a recognized problem early in the Apollo program. The major deterrent to efficient resolution of the problem was the unavailability of elastomeric materials that were compatible with propellants under long-duration exposure. A problem that was not recognized until considerably later in the program involved the incompatibility of the system and components with the flush fluids (or combinations of flush fluids) and propellants. At such time that compatibility of a system and its components with fluids is established, all fluids and mixtures of fluids that might be introduced into the system should also be established. Particular attention should be given to determining the specific fluids that might be used during manufacturing and checkout of the system and its components when the materials are selected. Provisions for adequate drying of systems should be made and verified if fluid mixing cannot be tolerated.

Cleanliness control was a problem because of the many small orifices and the close tolerances of moving parts. Assembling a clean system was difficult, and the need for component removal and replacement further increased the probelm. To minimize the problem, filters were added to protect components that had an unusually high failure rate because of contamination. On future programs, all components should be designed to be as insensitive to contamination as possible. Additionally, such components should be protected by integral filters. A further recommendation is that, if fluids are reverse-flowed through any component during a flushing or filling operation, both the inlet and outlet ports on the component should be protected against contamination. If large quantities of contaminants are expected, filters should also be provided at the fluid source.

A considerable number of unnecessary and costly situations occurred during the development and qualification tests, because the production of components was well underway before the test programs were completed, particularly during the system-level tests. Corrective action for problems that existed during these programs almost always involved the retrofit of production units and the modification of completed systems. Some problems were tolerated because of the extensive vehicle rework that would be required for corrective measures. These shortcomings were compensated for by either tolerating higher rejection rates or modifying operating procedures. Only limited changes were made to the systems as a result of these late tests. Consequently, the test results did little for the development of more reliable systems but, rather, were useful in instilling confidence in equipment or defining operating constraints. A further recommendation, therefore, is that extensive efforts be made to integrate the test program schedules with the master production schedules. Specifically, the overall schedule should be adjusted to provide time to implement the production hardware changes dictated by the test program.

4.4.8 Guidance, Navigation, and Control System

The functions of the guidance, navigation, and control system may be divided as follows:

a. Navigation is the process of determining spacecraft position and velocity at a given time in a basic reference coordinate system. The position and velocity data for a given time are referred to as a state vector.

b. Guidance and control are the functions that furnish commands to the engines to change or correct vehicle trajectory and to control vehicle attitude. The engines are controlled automatically in some modes and by the crew in other modes.

The two basic system configurations were referred to as Block I and Block II. The Block I system was designed when the command and service modules were to be landed on the moon. To achieve the system reliability required by this plan, spare units were to be carried on board, and inflight maintenance was to be performed. However, inherent problems existed in this concept that were never really solved, such as moisture getting into electrical connectors during changeout. The adoption of the lunar orbit rendezvous plan provided a logical time to change to the Block II configuration which, because of redundant paths, negated the inflight maintenance requirement and thereby avoided the connector problem. The Block II system was smaller, lighter, and more reliable than the Block I design. Another advantage was that the primary guidance systems for the command module and the lunar module could be nearly alike. The Block I system was flown on unmanned missions only. Therefore, this discussion pertains primarily to the Block II system.

The Block II configuration has a primary and secondary guidance and control system as illustrated in figure 4-17. Although navigation could be performed on board with the primary system, the primary method of navigation was to use data transmitted from the Mission Control Center, and the onboard system served as a backup. The redundancy of the Block II system assured that no single failure would cause total loss of any function.

The primary guidance, navigation, and control system consisted of inertial, optical, and computer systems. The inertial system provided a three-gimbal gyroscopically stabilized platform upon which three accelerometers were mounted, one for each orthogonal axis. Any rotational motion of the spacecraft about the platform was detected by the gyros and measured by resolvers built into the gimbals. Attitude information could thus be continuously sent to the computer. The three integrating accelerometers detected translational acceleration of the spacecraft and provided continuous velocity information to the computer. The inertial system also contained the electronics and power supplies required by the guidance and control system.

The optical system consisted of a sextant, a telescope, and associated electronics. Optical sightings were made on celestial bodies and on earth or lunar landmarks to accurately determine inertial position. When an optical sighting was made, a set of data consisting of time, spacecraft attitude, and optics pointing angles was recorded by the computer. By taking successive sightings, navigation data were obtained to solve the navigation equations.

The computer system received input data from the inertial and optical systems and manual commands from the crew through a hand controller. Operating on these inputs, the system solved navigation equations, generated on-off commands to the 16 attitude thrusters and the main engine, generated steering commands to the engine gimbal actuators, and generated appropriate control and display data. The computer contained a digital autopilot to control the vehicle during all flight phases. Three types of attitude control were available: automatic maneuvering to any desired attitude, maintenance of a desired attitude within selectable limits, and manual control by the crew through the use of rotation and translation hand controllers. During thrusting maneuvers, the autopilot automatically generated commands to the engine gimbal actuators to keep the thrust vector aligned with the center of gravity of the spacecraft. Engine ignition and cutoff commands were issued to achieve the desired velocity changes for that maneuver. During earth entry, the system automatically performed entry navigation and guided the spacecraft to a safe landing by controlling vehicle attitude to achieve the desired aerodynamic lift vector.

The secondary system consisted of attitude control, attitude reference, and thrust vector control systems, and the required displays and controls. The attitude control system received manual commands from two rotation and two translation hand controllers, and data from two body-mounted rate and attitude gyro packages. Operating on these inputs, this system generated on-off commands to the 16 attitude thrusters to maintain the desired attitude and perform the desired maneuvers. The attitude reference system provided display information and maintained an inertial attitude reference. It could be aligned to the primary guidance system inertial platform or to its own control panel thumbwheel settings. Total attitude, attitude errors, and spacecraft attitude rate were displayed on either one of two flight director attitude indicators. The thrust vector control assembly provided two backup modes of controlling the engine gimbal actuators during thrusting maneuvers if the primary system failed. An automatic mode and a manual mode were provided. Command inputs were routed to one of two servo systems which positioned the redundant gimbal actuators. Had a failure occurred in the primary system autopilot, servo system, or actuator, the crew could have switched to the secondary guidance system, servo system, and actuator.

The design and development of the primary guidance, navigation, and control system evolved from error analyses performed on early missile trajectories. The Polaris inertial guidance system concept was thought to be adequate to accomplish the Apollo program. Error analyses determined that moderate errors in the inertial instruments (gyros and accelerometers) could be tolerated because of the inflight realignment capability of the inertial system. The Polaris system was therefore modified and repackaged as necessary. The modifications provided (1) inflight alignment capability, (2) a general purpose computer, (3) mode selection by the crewmen, and (4) inflight maintenance capability (later deleted). Studies were made of strapdown guidance systems and of three-gimbal versus four-gimbal systems before the final configuration was determined.

The computer was developed through three configurations: the first was primarily for research and development, the second for unmanned flight, and the third for manned flight. The software was changed as required to meet specific requirements. The flexibility of the software proved to be a great asset late in the program.

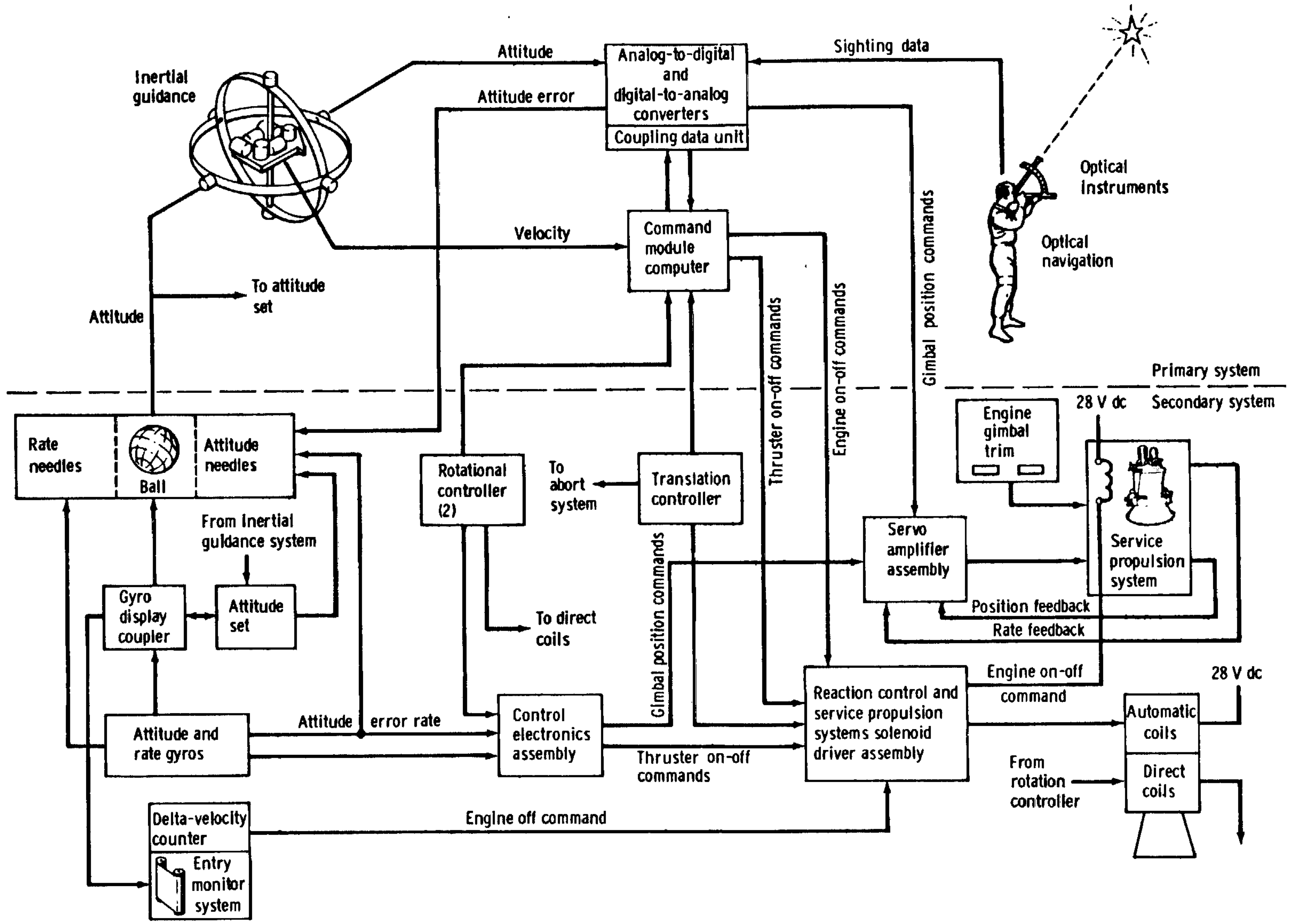


Figure 4-17. - Block II Command and service module guidance, navigation, and control system.

The optical system, at one time, included a map and data viewer and a star-tracker/horizon-photometer assembly. The map and data viewer was intended to display information such as flight plan data, checklists, and maps on rolls of film that could be projected. The viewer was deleted because of cost and schedule implications. The star-tracker/horizon-photometer assembly was intended to track celestial bodies automatically and to aid in tracking lunar and earth landmarks. This assembly was deleted because of cost and schedule impacts, and a rate-aided tracking routine that used computer software and existing optics equipment was substituted.

The test program consisted of four basic phases: development, qualification, acceptance, and installation and checkout. Functional, environmental, and evaluation tests were performed on material, parts, and components during the development test phase. Environmental and performance evaluation tests were performed on production parts, assemblies, subsystems, and systems during the qualification test phase. In general, systems were tested to nominal mission levels, whereas subsystems and below were tested to the stress level for critical environments. Acceptance tests and installation tests to specified limits were conducted to verify acceptable systems performance.

The performance of the Block I and Block II primary and secondary systems was excellent. The anomalies that did occur were of a minor nature and most were circumvented by workaround procedures.

The most significant anomaly that occurred in the primary system was in the inertial system. A voltage transient occurred when a set of relay contacts was transferring a voltage. The transient was electromagnetically coupled to other wiring within the electronics package and resulted in an erroneous indication to the computer that the inertial attitude reference had been lost. The crew reestablished the inertial reference by taking star sightings.

The most significant anomaly that occurred in the secondary system was in the redundant engine gimbal actuator assembly. An open gimbal rate feedback circuit caused unexpected oscillation of the engine gimbal. The oscillation was detected in the redundant servo system while the pilot was performing preignition checks which verify the primary and secondary servo systems.

A good indication of system performance of the inertial and optical systems was available from realignment data. Realignment of the inertial platform was performed periodically during each flight to correct for the very small drift rate of the gyros. The realignment was accomplished by sighting on two known stars using the sextant. The computer compared the measured angle between the stars to the known value and displayed a star angle difference to the crew. The star angle difference was an indication of sighting error (instrument error plus operator error). A 1-sigma value sighting error had been computed for each lunar mission. The largest value was 0.016 degree, and the 1-sigma value for eight lunar missions (Apollo 13 excluded*) was 0.011 degree. This compared well with the error analysis value of 0.012 degree for the two-star alignment procedure.

From the sighting data, the computer calculated the small angular position errors of the platform caused by the small gyro drift rates. For eight lunar missions (Apollo 13 excluded*), a 1-sigma drift rate of the command module system was 0.00765 degree per hour. This value compared well with the specification value of 0.030 degree per hour. Accelerometer bias errors (erroneous velocity output when no input acceleration is applied) were equally small. The average bias error for the Block II command module system was 0.00239 foot per second per second.

The performance of the digital autopilot during all thrusting maneuvers of the Apollo program was excellent. The digital autopilot guided the vehicle during thrust maneuvers to achieve a targeted velocity-to-be-gained. The residual velocity-to-be-gained after engine cutoff was an indication of overall system performance. Residuals were caused by accelerometer errors, gyro errors, computational errors, or engine thrust errors. The worst-case velocity residual of the Block II system was 4.4 feet per second. This was attributed to helium ingestion in the engine propellant, which caused a momentary low-thrust condition. Typically, the velocity residuals were on the order of 0.3 foot per second or less.

*Because of operational constraints, normal realignment procedures could not be followed. Consequently, the inaccuracies were larger than would normally be expected and the data were excluded from the calculation of the 1-sigma values.

The performance of the computer was flawless. Perhaps the most significant accomplishment during Apollo pertaining to guidance, navigation, and control was the demonstration of the versatility and adaptability of the computer software. For instance, the crews gained additional confidence in the digital autopilot with each mission. During the last mission, a special software procedure was used in lunar orbit to maintain precise spacecraft pointing attitudes, despite having normally used attitude thrusters turned off. The only consistent method of initiating the passive thermal control mode was to use a software routine, which was modified slightly to accomplish special results. Workaround procedures, called erasable memory programs, were used time and again to accomplish special jobs and lighten crew tasks. Hardware modification to accomplish these changes would not have been feasible.

As stated earlier, the Mission Control Center provided the primary navigation mode. However, the onboard computer and the sextant and telescope did provide onboard navigation capability. Cislunar navigation (to and from the moon) was demonstrated, particularly during the Apollo 8 and 10 missions. Star-horizon optical sightings were made using the earth and moon horizons. Postflight analysis of these data verified the crew's capability to navigate to the moon, compute the lunar-orbit-insertion maneuver, and place the vehicle in a safe lunar orbit. The same navigation technique was used to demonstrate the crew's capability to return to earth and to accomplish a safe earth landing.

In lunar orbit, the intended navigation technique was to use the telescope to track known or unknown landmarks. In practice, the sextant, which was a more accurate instrument, was normally used because a computer routine called rate-aided optics was available. This routine made the sextant tracking task much easier. Postflight analysis of data from the landmark tracking navigation technique demonstrated the capability to successfully compute a transearth injection maneuver.

For detailed discussions of the development and performance of guidance, navigation and control systems, see references 4-40 through 4-49.

4.4.9 Environmental Control System

The three major functions of the environmental control system were atmospheric control, thermal control, and water management. Six systems operating in conjunction with each other provided these functions.

a. The oxygen system controlled the oxygen flow within the command module, stored a reserve supply of oxygen for use during entry and emergencies, regulated the pressure applied to components of the oxygen system and pressure suit circuit, controlled cabin pressure, controlled pressure in the water tanks and water/glycol reservoir, and provided for purging of the pressure suit circuit.

b. The pressure suit circuit system provided the crew with a continuously conditioned atmosphere. With this system, suit gas circulation, pressure, and temperature were automatically controlled, and debris, excess moisture, odors, and carbon dioxide were removed from both the suit and cabin gases.

c. The water system supplied water for drinking, food reconstitution, and evaporative cooling. Water produced by the fuel cells was pumped into a potable water storage tank. Waste water (primarily perspiration condensed by the suit heat exchanger) was stored in a waste water tank and distributed through the control valves of the water/glycol evaporators. Waste water could be augmented by excess potable water for evaporative cooling. If the water production rate exceeded the usage rate, water was dumped overboard.

d. The water/glycol system provided cooling for the pressure suit circuit, the potable water chiller, and the spacecraft equipment mounted on coldplates. The system also heated and cooled the cabin atmosphere. Temperature control was obtained by the circulation of a mixture of water and ethylene glycol through primary and secondary coolant loops. The temperature of the heat-transport fluid was controlled either by radiators or by glycol evaporators.

e. The waste management dump system provided for dumping urine and excess water overboard and venting the waste storage compartment.

f. The postlanding ventilation system provided a means of circulating ambient air through the command module cabin after landing.

To provide the high degree of reliability required for lunar missions, the system was designed with redundant components, backup systems, and alternate modes of operation. For example, parallel system regulators and relief valves were contained in a single housing and had isolation selector valves. Suit compressors, condensate pumps, and cabin fans had separate backup units. The primary coolant system contained redundant pumps, and a secondary coolant system with radiators, evaporator, and cabin and suit heat exchangers was provided. However, some electronic components were not serviced by the secondary loop. Also, the secondary radiators could not reject sufficient heat for a normal mission and were therefore considered a contingency system.

Major changes to the environmental control system during the development program included the redesign of the coldplates, control of the glycol evaporator, and the composition of the cabin gas during preflight operations. More detailed information is given in references 4-50, 4-51, and 4-52.

The most significant change to the Block II environmental control system was the addition of hardware for extravehicular activity from the command module. A 10-pound-per-hour oxygen purge system was added to supply suit pressure, breathable atmosphere, and thermal control to the extravehicular crewman in the event of an emergency. For normal operation, the spacecraft suit circuit system regulated the upstream pressure through a 25-foot umbilical hose to an orifice assembly attached to the extravehicular crewman's pressure suit. Flow was regulated by a suit outlet relief valve which controlled suit pressure at 3.75 psia. The other two crewmen were supported by the spacecraft pressure suit circuit while the cabin was depressurized.

With the exception of the Apollo 13 oxygen source failure, the oxygen system operated satisfactorily throughout the entire flight program. Cabin pressure relief and regulation were maintained at nominal values of 6 and 5 psia, respectively, and all scheduled cabin repressurizations were accomplished without incident. No emergency pressure regulation was required. Inflight cabin leakage varied from about 0.10 pound per hour to 0.02 pound per hour with improvement noted in the later vehicles.

The pressure suit circuit system also generally performed acceptably and met mission requirements. As confidence was gained in the dependability of the spacecraft cabin environment, fully suited operation was eventually limited to the launch and lunar module jettison events. Depressurized cabin operations were handled routinely, and no emergency suit circuit conditions were encountered. Carbon dioxide removal, obtained by alternately replacing the lithium hydroxide elements on a nominal 72 man-hour rotation, was satisfactory. Carbon dioxide partial pressure seldom rose above an indicated 3 torr. Some excessive element swelling due to moisture absorption was noted during solo crewman operation on one lunar flight. Procedures were subsequently incorporated to prevent recurrence of the problem.

Water servicing of the sintered, porous metal plate in the suit heat exchanger proved to be a major system problem. Gas breakthrough and/or degraded flow rate led to extensive ground testing to better understand the physical phenomena involved and to develop an adequate wetting technique. Humidity control and water removal were satisfactory under the flight-imposed coolant loop conditions, and no evidence of gas breakthrough or flow degradation was observed during a mission.

The water system proved to be a source of both positive and negative crew reaction. On the plus side, the hot water provided for food reconstitution was greatly appreciated and was noted as a considerable improvement over the cold food available on earlier spacecraft. On the minus side, gas in the potable water caused problems in filling the food and water bags and in the digestive processes of individual crewmen. The gas consisted of hydrogen from the hydrogen-saturated fuel cell water and oxygen (used to pressurize the water tanks) permeating through the tank bladders. A silver-palladium tube separator was installed to remove hydrogen. To remove the oxygen, a gas separator cartridge assembly that used hydrophobic and hyrophyllic membranes was added for attachment to the water supply ports. This membrane assembly met with only limited success.

Additional crew problems occurred during the daily water sterilization procedures when separate chlorine and buffer solutions were injected into a port in the water system. Leakage at the port was noted during the Apollo 15 mission and breakage of the bags containing the solutions increased during the later missions. Revised assembly methods eliminated the port leakage.

The water/glycol coolant system provided adequate thermal control in spite of several hardware failures. Built-in manual operating modes were successfully used to replace the normal automatic control. Early glycol evaporators showed tendencies to dry out under low heat loads and were reserviced by the crew. Subsequent modifications, which included the previously mentioned relocation of the wetness sensors and trimming of the surrounding sponges, provided satisfactory units. After the radiator system demonstrated acceptable heat rejection, evaporator operation was limited to launch and entry periods only. The radiator and flow-control system provided typical heat rejection in the range of 4000 to 5000 Btu/hr.

Noise from the cabin fans was considered objectionable by the crews, and use of the fans was discontinued on the later flights except to remove lunar dust from the cabin environment.

During the Apollo 16 mission, the automatic controller for the command module water/glycol temperature control failed. Manual positioning of the mixing valve was successfully accomplished by the crew.

The addition of lunar orbital science experiments to the later spacecraft required holding attitudes during experiment operation in lunar orbit which resulted in undesirable radiation environments for the space radiators. Also, operation of the glycol evaporators was undesirable because of possible contamination of the experiment lenses and fields of view, and because of the propulsive reaction of the vehicle. Therefore, during each lunar orbit, spacecraft temperatures cyclically rose to levels from 70° to 85° F rather than being controlled to 50° F, maximum, as intended by design.

On early flights, checks of redundant components were performed each night during the mission. On later flights, the secondary coolant loop and oxygen regulator checks were performed in earth orbit and a secondary coolant loop check was performed just before lunar orbit insertion. Nightly checks were eliminated. No redundant component failure was detected by an inflight check. The only redundant component that may have failed during the Apollo missions was a main oxygen regulator isolation valve which failed closed due to shearing of the actuation handle pivot pin. The failure, however, was believed to have actually occurred after the flight.

An area of deviation from the intended procedure was the use of the glycol evaporator only in earth orbit until the radiators cooled down from the launch heating, and during chilldown for entry. This resulted in higher cabin temperatures during certain fixed attitudes and excessive temperature cycling that ranged from 45° to 85° F during lunar orbit. As a result, condensation occurred on cold surfaces after the higher temperatures of the cycles because the dew point temperature is directly related to the coolant temperature.

Other minor deviations from designed operating modes were (1) use of the carbon dioxide absorber elements for more than 72 man-hours and (2) use of the coolant temperature control valve in manual mode, at a higher temperature than the normal automatic 45° F, to increase cabin temperature and crew comfort. This action was taken because of attitude holds in transearth coast which prevented exposure of the radiators and side structures to the sun and resulted in lower overall temperatures.

When ground thermal vacuum tests indicated that intermittent, automatic overboard relief of excess water might result in dump nozzle freeze-up, a manual method of dumping was developed and used successfully in flight. On later missions, half of the redundant relief valve was removed, and the manual method was simplified by dumping directly through the normal water dump nozzle.

During several of the later missions, urine was stored for medical experiments and dumped overboard only once a day. Crystals which formed in the stored fluid caused plugging of the regular in-line system filter. A special high-capacity, open-cell polyurethane core filter was developed and used successfully for dumping stored urine on subsequent flights.

4.4.10 Displays and Controls

The displays and controls system served as the interconnecting link between the crew and the spacecraft. The interior and exterior lighting devices and the malfunction detection devices (known as the caution and warning system) were also a part of the system. The system contained toggle switches, event indicators, electrical meters, panel assemblies (some of which had electroluminescent lighting overlays), rotary switches, pushbutton switches, digital timers (mission timers and event timers), and several other types of control and indicating devices. The types and numbers of devices varied from mission to mission because of different mission requirements.

Many problems became evident during the system developmental phase and much testing and evaluation was required to produce the flight-qualified components for final vehicle installation. With only a few exceptions, identical components were used in the Block I and Block II vehicles.

One of the problems encountered during the development phase was the unsuccessful use of a hermetically sealed snap-action switch unit in conjunction with an unsealed mechanical toggle actuator. The toggle switch was pressure-sensitive and functioned erratically. The toggle switch finally used on Block II vehicles was a completely hermetically sealed unit. A number of discrepancies was encountered during the development of the hermetically sealed switch. For example, extra pieces and parts were found inside the switch, poor welds were observed, and inverted contact buttons on internal terminal posts were found. In spite of the poor preflight record, only one switch of this type failed in flight.

Other items with which problems were encountered during the development and test phases were electrical indicating meters, event indicators, interior floodlights, mission timers, and potentiometers. The electrical indicating meters and the event indicators contained internal contaminants which caused the movements to bind excessively. The interior floodlights had several development problems, some of which were not solved until after the third manned flight. The use of starting diodes that were of better quality and operated at higher voltages corrected the condition that caused the earlier lamp failures. Another corrective action was a change in the lamp-use procedure. Restricted use of the secondary lamps in the dim mode vastly extended the life of those lamps. The Block II mission timer had a solder joint breakage problem because of the difference in expansion rates between internal components and the potting compound. A redesign of the timer reduced the solder joint problem. In addition, the glass faces of some timers cracked. This condition was corrected by a design change to the case seal which had been stressing the glass. The mission timer problems started with Apollo 7 and continued sporadically until the redesigned unit was introduced on the Apollo 14 mission. The potentiometer problem was isolated to a shaft that was being deformed under load and breaking or overriding an internal stop, as well as giving erratic resistance readings. The corrective action was to install a bearing support and an external stop for the shaft and to require a calibration curve with each potentiometer delivered by the vendor.

Because of the thorough development and test program, the flight displays and control system problems were minimal. Some examples of the problems encountered during flight and corrective actions taken are as follows. On the Apollo 15 mission, a shorted filter capacitor tripped a circuit breaker which made some of the lower equipment bay lights and the guidance and navigation display keyboard unusable. Installation of a fuse in the offending circuit prevented recurrence of this problem. There were several instances of poor performance of the event timer during flight. Erratic timing and obscuring of the timer numerals by paint particles resulted from mechanical wear and friction.

Very few changes were made in the displays and controls system during the flight program except to accommodate changes made in other systems. These were usually the addition of items such as switches, circuit breakers, or meters. However, following the oxygen tank failure on the Apollo 13 mission, several changes were made. First, the oxygen tank fan and thermostat controls were removed and two switches were added to connect the auxiliary battery power supply to the distribution system and activate an isolation valve between oxygen tanks 2 and 3. Secondly, the reactant valves in the hydrogen and oxygen lines of all tanks were coupled to the caution and warning system as well as to the event indicators. Finally, the indicator circuitry was changed to indicate when either valve was closed rather than to indicate when both were closed. Additional information on the development and performance of the controls and displays is given in reference 4-53.

4.4.11 Communications System

The communications system included the equipment required for voice communications, data operations, tracking and ranging, and onboard television transmission. The system included both VHF and S-band equipment to accommodate the various radio frequencies used in air-to-ground transmissions.

Voice communications included spacecraft intercommunications between crewmen, hardline two-way voice communications with the Launch Control Center through the service module umbilical during the prelaunch period, inflight two-way voice communications with the Manned Space Flight Network (later designated the Space Flight Tracking and Data Network) by VHF/AM and S-band systems, and postlanding voice communications with recovery ships and aircraft.

Data operations included time-correlated voice tape recording of flight crew comments and observations; S-band transmission of real-time or stored telemetry data; and S-band reception of updata (guidance and navigation data, timing data, and real-time commands) from the Space Flight Tracking and Data Network.

As with other systems, the communications system had a major design change point that divided the development program into Blocks I and II. Although certain functional design changes were made for the Block II communications system, the basic change was from a mechanical standpoint. Inflight-replaceable modular-type equipment was replaced with sealed units that had built-in and switchable redundancy where required to meet program objectives. The Block I and Block II communications systems differed in three major aspects.

a. Equipment that was not considered necessary to the lunar landing mission was deleted from the Block II spacecraft.

b. Deficiencies that were noted in the Block I design were corrected in the Block II design.

c. New equipment was added to the Block II system because of the requirement for combined lunar module/command and service module operations.

The deleted equipment consisted of a VHF/FM transmitter and a C-band transponder. Functions of this equipment (data transmission and ranging) were absorbed by S-band equipment. In addition, a high-frequency transceiver and antenna were also removed from the program.

Electrical wiring problems were experienced during the Mercury 9 flight wherein contaminants (water, urine, sweat, etc.) migrated to exposed electrical terminals in the zero-g environment. These problems led to the decision to seal all Apollo electrical wiring and connectors. However, the Block I Apollo hardware was already designed and was being built in accordance with the inflight maintenance concept. This meant that many module-to-black-box connectors and many self-mating black-box-to-spacecraft connectors were required. The attempt to make connectors and modules humidity proof was lengthy, sometimes futile, and practically eliminated any possibility of inflight maintenance. The Block II design change involved repacking the crew compartment equipment into completely sealed units and incorporating built-in and switchable redundancy, as well as backup modes, to achieve the desired reliability and to satisfy the lunar rendezvous mission requirements.

The development of the individual equipment parameters was based on the total communications system requirements. The interface parameters defined in the equipment specifications were validated and verified in laboratory system tests conducted by the major subcontractor as part of the ground test program. Further laboratory tests were performed at the Manned Spacecraft Center to verify overall system compatibility with the Space Flight Tracking and Data Network and the lunar module. However, development and qualification were performed on the basis of individual equipment tests.

Flight tests were performed to ensure that the system would meet the requirements of space operations. Unmanned flights qualified the portion of the system that was required for manned earth-orbital flights. The manned earth-orbital flights, together with supporting laboratory evaluation, qualified the system for the lunar mission operations.

The major problem area in the design, development, and production of the communications system hardware was the S-band high gain antenna. The high gain antenna was the pacing item of communications equipment and underwent extensive redesign to correct for major deficiencies and failures experienced during its development and qualification. As a result, the antenna could not be flown on the Apollo 7 mission as originally planned, and it was necessary to waive the qualification requirement and install the antenna assembly at the launch site to permit its use on the Apollo 8 mission. However, operation during the Apollo 8 mission was considered satisfactory. Data obtained during this mission were valuable in developing procedures and as a reference for evaluating high gain antenna performance during subsequent missions.

The equipment malfunctions that were experienced throughout the program are mentioned here, and additional details may be obtained from the mission reports referenced.

Apollo 9: On one occasion, the updata link would not accept commands until the decoder logic was reset by cycling the spacecraft uptelemetry switch from the NORMAL to OFF to NORMAL positions (ref. 4-15).

Apollo 12: Problems experienced during the Apollo 12 mission were poor VHF voice quality during lunar module ascent and rendezvous and an occasional decrease in S-band signal strength when operating through the high gain antenna. These problems are discussed in reference 4-18.

Apollo 13: Difficulty was experienced in obtaining high gain antenna acquisition and subsequent tracking (ref. 4-19).

Apollo 14: Communications system problems were (1) poor VHF performance for voice and ranging during lunar module ascent and rendezvous and (2) the high gain antenna failure to acquire and track properly at various times during the mission (ref. 4-20).

Apollo 16: On two occasions, the updata link did not accept commands until the decoder logic was reset. This condition was the same as that experienced on Apollo 9 (refs. 4-15 and 4-22). A second problem was that, on one occasion, the high gain antenna failed to operate properly in the reacquisition/narrow-beamwidth mode until the logic had been reset by momentary selection of the manual mode by the crew (ref. 4-22).

Information obtained during the missions was fed back into the operational procedures and the ground test program. The high gain antenna was the major area in which ground tests were changed. A special system-level high gain antenna thermal/functional acceptance screening test, introduced prior to the Apollo 15 mission, was instrumental in identifying an antenna gimbal radio-frequency rotary joint design deficiency that was not detected during development or acceptance testing.

As the result of flight experience, changes were incorporated in the areas of crew-adjustable controls for VHF squelch and for microphone placement. Training simulator fidelity was improved and the crews were briefed and trained to recognize and correct idiosyncrasies and problems previously experienced in flight. The area of antenna management was improved by the incorporation of high gain antenna gimbal angle and mode switch telemetry, updating procedures, and developing a look-angle display for determining optimum up-link command times. The command and service module communications system is discussed further in references 4-54, 4-55, and 4-56.

4.4.12 Instrumentation System

The instrumentation system of the command and service modules consisted of data acquisition and storage components and central timing equipment. Transducers and signal conditioners were located throughout the spacecraft, each in the proximity of the parameter to be measured. On a typical manned spacecraft, about 125 parameters were measured by this system, which interfaced with all other systems. Sensors were provided to measure pressure, temperature, quantity, flow, attitude, attitude change rate, voltage, current, frequency, radio power, vibration, strain, acoustic noise level, acceleration, heat shield char, ablation and heat flux, nuclear particle flux, biomedical parameters, and to perform gas analysis of the spacecraft atmosphere. There were different measurements for each spacecraft because the mission objectives were different for each flight and instrumentation emphasis changed as experience was gained. The data storage equipment was a magnetic tape recorder large enough to hold all data generated by the spacecraft while out of communications with a ground station. This condition occurred when the direct line between the spacecraft and ground station was occluded by a portion of the earth or moon. The central timing equipment provided timing signals to other systems, including elapsed time from launch, to the telemetry system.

In some cases, instrumentation hardware was integrated with other systems and delivered to the prime contractor already installed. Such items were not considered a part of the instrumentation system, per se, and are not included in this discussion.

The first stage in the instrumentation development process was the establishment of measurement requirements. An instrumentation equipment list was then compiled and procurement activity was undertaken to obtain the items on the equipment list. As the hardware was developed, it was subjected to testing that provided assurance that the hardware (1) could perform in the operational environment to which it would be subjected, (2) could conform to the accuracy requirements of its specification, and (3) could reasonably be expected to last as long as necessary. Design proof tests, qualification tests, off-limits tests to destruction, and accuracy determination were performed on each type of measurement device. After passing these tests, the hardware was subjected to acceptance testing, pre-installation testing, testing after installation on the spacecraft, and system checkout.

Because of the extensive testing, nearly all the following deficiencies were discovered early in the program.

a. A rather high rate of rejection at pre-installation inspection

b. Mechanical damage by personnel working in the spacecraft

c. Susceptibility of some instruments to radio-frequency interference

d. Calibration changes

e. Instability of output

The high rate of rejection was found to be caused by a difference between the acceptance test procedure used at the vendor's plant before shipment and the pre-installation test procedures performed at the prime contractor's plant. This was solved by making the two procedures identical, including the fail/pass criteria. The mechanical damage problem was solved by providing appropriate precautionary instructions to the manufacturing and checkout personnel. Susceptibility to interference was reduced to an acceptable level by changing the electrical grounding techniques. Calibration shifts and instability of output were both traced to oscillations of the scaling amplifiers and regulators within the signal conditioners and were eliminated by the addition of small shunt capacitors.

The tape recorder used for data storage was initially designed and built to the requirements of the reference lunar mission; the recorder had to be modified for the earth-orbital missions and the lunar-orbital science missions. The first modification, to meet the requirements of the earth-orbital missions, consisted of strengthening the transport mechanism to extend its specified life from 14 to 200 hours. The second modification, for the lunar-orbital science missions, added a digital channel for mission scientific data and doubled the recording time capacity.

The central timing equipment was modified to provide a serial time code output for the scientific experiment hardware and data system, in addition to the original parallel output.

Very few flight failures occurred. From Apollo 7 through Apollo 17, there were three cases in which the measurement hardware produced no output, three cases of noisy outputs from which data could be derived by averaging, and three cases in which the output was slightly out of tolerance. These nine cases represent only about 0.6 percent of the instrumentation system hardware flown on the 11 spacecraft.

The data storage equipment operated thousands of hours without data loss except for a few minutes during the entry of the Apollo 10 command module and about a half minute during transearth coast of the Apollo 15 command and service module. The Apollo 10 data loss was caused by deformation of the tape recorder case due to the pressure increase of entry; strengthening the case corrected this condition for later flights. The half-minute loss of data on Apollo 15 was traced to the tape leader material, which had transferred to the first few feet of the magnetic tape. This problem was corrected for later missions by carefully wiping the first few feet of the tape and leader material before installing the magnetic tape.

Several recommendations for instrumentation systems may be made from the experience derived from the Apollo program. A realistic approach to measurement requirements and to the accuracy actually needed makes it possible to instrument for almost any operational parameter. Attempts to provide large numbers of exotic measurements at unattainable accuracies merely waste time and money. In nearly all measurements, an overall accuracy of plus or minus 5 percent will suffice. A workable ground rule for establishing the number of measurements is that one measurement at each point in each system where a change of physical state occurs is necessary and sufficient. Simple hardware redundancy is not as effective in protecting against instrumentation hardware failures as is a matrix of measurements whereby data missing due to hardware failures can be derived from other measurements. Flexibility to change measurements by deletion, addition, and substitution should be built in from the beginning.

The development and performance of the command and service module operational instrumentation system is discussed in greater detail in reference 4-57.

4.5 LUNAR MODULE DEVELOPMENT PROGRAM

4.5.1 Introduction

The decision to utilize a lunar rendezvous mission technique was made in July 1962, and the contract for the design and development of the lunar module was awarded four months later. The lunar module was unique in that it was the first manned spacecraft which was specifically designed for operation totally outside of the earth's environment. Based on the mission plan, the spacecraft was designed to (1) land two astronauts on the moon from lunar orbit, (2) support lunar surface exploration and the deployment of scientific experiments, and (3) return the astronauts and lunar samples to the command and service module in lunar orbit.

No parallel equivalent to the command module Block I and Block II development philosophy existed in the lunar module development, although the lunar module was reconfigured in the late stages of the Apollo program to accommodate an extended lunar stay capability. Unlike the command module development program, the lunar module development program emphasized ground tests and minimized unmanned flight development tests. As planned, LM-1 was the sole unmanned lunar module which was flight tested with operative systems. In all, only three production lunar modules were flight tested prior to the Apollo 11 lunar landing mission (see sections 2.3 and 2.4) and there were no active boilerplate flight items in the program. The general configuration of the lunar module is shown in figure 4-18.

4.5.2 Test Articles and Ground Test Program

The lunar module development program utilized a series of ground test vehicles for establishing the production configuration and man-rating the flight vehicles. In increasing order of development complexity, the types of vehicles employed were mockups (M series), test modules (TM series), and lunar module test articles (LTA series). In some instances, the total lunar module configuration was simulated; however, in other instances, only the area of test interest was simulated. The following paragraphs identify the test articles and indicate the types of ground test programs that they supported.

4.5.2.1 Mockups.- Five lunar module mockups were constructed during the course of the development program. A wooden mockup, designated M-1, was constructed for the purpose of studying the ascent stage cabin configuration requirements. M-3 was an ascent and descent stage external

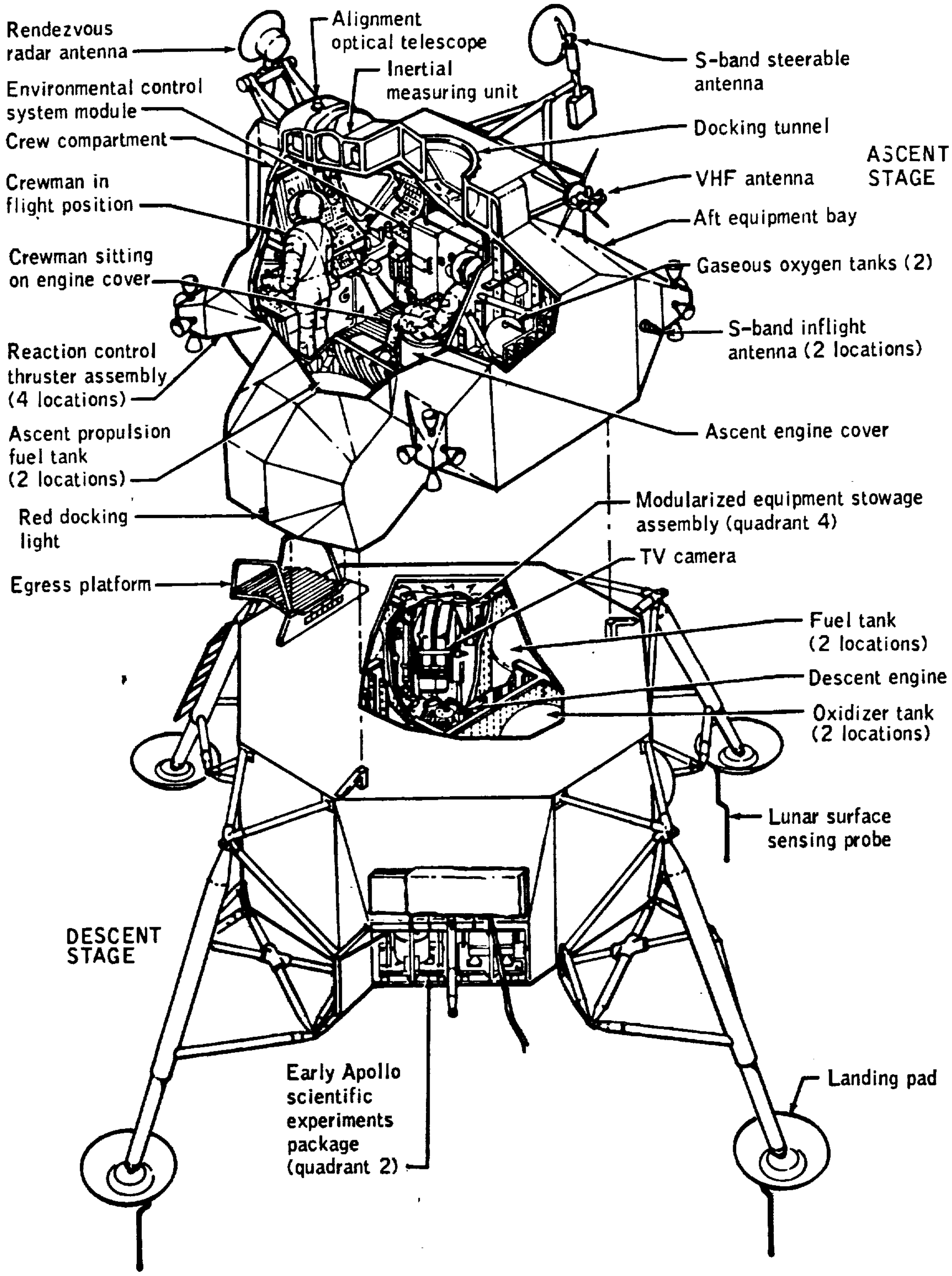


Figure 4-18.- Lunar module configuration for initial lunar landing.

configuration article. It was used for verification of the spacecraft/launch vehicle adapter interface and for facility verification. The M-4 mockup was constructed to study the descent stage engine compartment requirements. M-5 was a mockup for the evaluation of the spacecraft equipment installation. Mockup M-6 was developed to support new flammability test requirements imposed after the Apollo I fire.

4.5.2.2 Test Models.- Sixteen test models were used in the lunar module development program. Most of the test models were specialized for specific investigations and were not complete ascent and descent stage configurations. These models were used for such things as crew visability and mobility studies (TM-1), radio frequency tests (TM-3), pyrotechnic studies of ascent/descent stage separation (TM-4), lightweight descent stage landing studies and stowage reviews (TM-5), rendezvous radar antenna tests (TM-6 and TM-7), landing radar tests (TM-8), reaction control system plume impingement tests (TM-9), battery installation thermal tests (TM-13), docking tunnel tests (TM-14), descent stage thermal tests (TM-15 and TM-17), and descent stages structural tests (TM-16).

4.5.2.3 Lunar Module Test Articles.- Eight lunar module test articles were constructed. The LTA-B article was used solely to provide ballast, in the form of the lunar module configuration, for the Apollo 8 mission. The LTA-1 test article was used for ground testing the lunar module electrical and electronic systems and to verify the checkout procedures which were developed for flight spacecraft. Like all of the LTA series, LTA-1 was constructed, inspected, and tested by the same controlled process as a production flight vehicle. Also, this test article was designed in parallel with the LM-1 unmanned flight vehicle, but had an earlier forward hatch configuration. Test article LTA-2 was first used to test the response to the launch vehicle vibration environment. It was later refurbished and used as payload ballast for the Apollo 6 launch vehicle. LTA-3 was a static and dynamic structural test article. Designed in parallel with LM-3, the LTA-3 test article was a product of the so-called super weight improvement program which was implemented for LM-3 and subsequent vehicles to decrease and control the growing lunar module weight. The LTA-5 test bed was a complete descent stage and was used for descent stage propulsion testing at the White Sands Test Facility. Man-rating testing was performed on LTA-8 in the Space Environment Simulation Laboratory at the Manned Spacecraft Center (sec. 11.4). This test article was essentially the same as the LM-1 spacecraft. Originally built as a test article for use by the command and service module prime contractor, LTA-10 was later used on the unmanned Apollo 4 mission as instrumented ballast for the launch vehicle. The LTA-11 test vehicle supported the extended lunar stay requirements for the Apollo 15, 16 and 17 missions, and was used as a drop test vehicle in conjunction with the testing of the lunar roving vehicle.

4.5.3 Unmanned Flight Test Program

The Apollo 5 mission featured the unmanned flight testing of the first production lunar module, designated LM-1. As an unmanned vehicle, LM-1 had both automatic and remote-controlled programming capability to operate the active onboard systems. The LM-2 vehicle was produced as a "sister ship" to LM-1, but had optional manned/unmanned flight capability. Originally intended to be used as the first manned lunar module on Apollo 8, it was diverted to support the ground test program in the Manned Spacecraft Center's vibro-acoustic test facility after Apollo 8 became the command-and-service-module lunar orbital mission.

4.5.4 Manned Vehicles

The lunar module development program was continued during the production of the flight spacecraft by the continual updating of flight hardware to reflect changes indicated from mission experience and new program requirements. The most program-effective single step was the aforementioned super weight improvement program. This program employed some of the most sophisticated engineering design and manufacturing techniques used to date in the production of manned spacecraft.

4.5.4.1 Apollo 9 through Apollo 14 Lunar Modules.- The vehicles used in the Apollo 9 and Apollo 10 missions were developed for use in earth orbit and lunar orbit and, as such, had numerous differences from the lunar landing spacecraft. Table 4-V indicates the major differences.

TABLE 4-V.- SUMMARY OF MAJOR CHANGES TO LUNAR MODULE

Function/System	Changes
Changes Implemented for Apollo 9 and Apollo 10 Missions (LM-3 and LM-4)	
Structures	Doublers added to upper deck of descent stage. Apollo lunar surface experiment package and modular equipment stowage assembly mass simulated. Descent battery support structure modified to mount two batteries in quadrant I and two batteries in quadrant IV. Emergency detection relay box support structure modified to mount one box on ascent stage and one box on descent stage. Crushable honeycomb inserts added to landing gear leg assemblies.
Thermal control, passive	Insulation lightened by reducing number of layers of insulation in blankets. Window shade material thermal capability increased from 200° to 300° F.
Pyrotechnics	Electro-explosive devices batteries and relay boxes relocated, one mounted on ascent stage and one mounted on descent stage. Number of circuit interrupters reduced from three to two (LM-4).
Electrical power	Four descent stage batteries relocated. Descent electrical control assembly modified to allow command module to power ascent stage alone.
Instrumention	Development flight instrumentation deleted (Apollo 10 only).
Communications	Digital uplink assembly added to replace digital command assembly. Ranging tone transfer assembly added for command and service module/lunar module VHF ranging.
Radar systems	Landing radar modified for earth orbital mission and lunar orbital mission, per respective flights.

TABLE 4-V.- SUMMARY OF MAJOR CHANGES TO LUNAR MODULE - Continued

Function/System	Changes
Changes Implemented for Apollo 9 and Apollo 10 Missions - Concluded (LM-3 and LM-4)	
Guidance and control	Ascent engine arm assembly modified to allow unmanned abort guidance system firing. Alignment optical telescope weight reduced. Reaction control system thruster-on time was increased for a given input signal.
Descent propulsion	Helium explosive valve reinforced by adding an external braze.
Ascent propulsion	Rough combustion cutoff assembly deleted. Propellant tank support cone installation changed from rivets to bolts. Relief valves modified to gold braze with notched poppet step.
Environmental control	Suit circuit assembly changed from titanium to aluminum for better fan operation. Primary sublimator feedline solenoid valve deleted in water management system.
Changes Implemented for Apollo 11 Through Apollo 14 Missions (LM-5 Through LM-8)	
Structures	Provisions added for scientific equipment package. Modular equipment stowage assembly added in quadrant IV of descent stage. Docking structure, descent stage shear webs and base heat shield modified as part of weight reduction program. Quadrant IV modified to support modular equipment transporter (LM-8 only). Forward landing gear surface sensing probe removed and length of remaining probes increased.

TABLE 4-V.- SUMMARY OF MAJOR CHANGES TO LUNAR MODULE - Continued

Function/System	Changes
Changes Implemented for Apollo 11 Through Apollo 14 Missions - Continued (LM-5 Through LM-8)	
Thermal control, passive	Descent stage base heat shield changed from Kapton to Kel-F to prevent landing radar interference. One layer each of nickel foil and Inconel foil added to landing gear struts. Landing gear insulation reduced for weight savings of 27.2 pounds. Thickness of forward hatch outer shielding increased.
Electrical power	Descent stage batteries modified by adding potting insulation across top of cells and providing an overboard vent manifold for cell vent valves. Manifold vent valve and core vent valve added to control differential pressure across cell cores (LM-8).
Instrumentation	Ascent propulsion system helium tanks temperature measurements deleted and redundant pressure measurements added. Temperature measurements added to ascent stage water lines and descent propulsion system engine ball valves.
Communications	Extravehicular activity antenna and S-band erectable antenna added. Television camera stowed on modular equipment stowage assembly.
Radar	Crew control added to break lock and search for main beam of landing radar; circuitry provided to prevent computer strobing pulse from appearing as two pulses. Override switch added to rendezvous radar for primary or secondary gyro select; heaters added to gyro assemblies.
Guidance and control	Primary guidance and navigation control function to descent engine gimbal drive actuators changed from brake to constant damping. Primary guidance program changed to allow return to automatic control for landing in the event that dust obscured visibility. Ascent engine arming assembly removed from control electronics.