Systems Design Experience from Three Manned Space Programs

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Introduction

SINCE the moment when Sputnik made the world aware of both the specter and the promise of space, many fascinating events have taken place by venturing into "this new ocean."[1] NASA was formed as a part of the National Space Act of 1958.[2] At that time, the contract to build the Mercury spacecraft was awarded, giving birth to the U.S. man-in-space program. After four manned orbital flights, Project Mercury was concluded in less than 5 years from initiation. The Gemini Program was conducted in about the same timespan and was concluded in 1966. The Apollo Program was initiated in 1961 and has only begun the manned lunar exploration program.

Development Programs

The objective of Project Mercury was to place into Earth orbit and safely recover a U.S. astronaut.[3] Implicit in this objective, however, was examination of man's capability to survive and function in the weightlessness of space. Project Mercury demonstrated that the pilot not only could survive the rigors of space but also was a responsive and functional part of the mission operation; therefore, much of the requirement for automation and component redundancy in subsequent spacecraft could be relaxed where manual modes were practical. The Gemini Program was conceived to fill the development void between the limited Mercury flight program and the operational qualification of the Apollo spacecraft. The Gemini Program objectives, therefore, were to advance man's operating ability in space and to verify certain Apollo concepts. Because the lunar-orbit-rendezvous mode had already been selected for Apollo,[4] Gemini was to demonstrate rendezvous[5] and rigid docking with an independent vehicle (the Agena stage) and conduct propulsive maneuvers with either vehicle. Lunar surface exploration required autonomous extravehicular activity; and the Gemini objective, evaluation of man's ability to work outside his spacecraft, was established. The Apollo concept also required a controllable entry vehicle to widen the return corridor and limit acceleration loads; therefore, the Gemini Program adopted the design objective of a lifting body, to be used in Apollo, with limited entry maneuverability. The objective of demonstrating a capability for land landing was later deleted because of insufficient development time. The Apollo goal, initially to land men on the moon and return them safely, has been expanded to include a series of landings at a variety of sites for more fruitful lunar surface exploration. Apollo hardware development relied on Mercury and Gemini performance results,[6,7] and only those technology advances dictated by unique mission requirements were made. A summary of all manned flight objectives and results is presented

John H. Boynton was first introduced to the Apollo concept in 1960 while at General Dynamics where he was a lead engineer in a NASA-funded study to investigate manned lunar landing. Mr. Boynton joined NASA and was responsible for Project Mercury technical documentation from March 1962 to its conclusion. Following Mercury, he coordinated Apollo mission design and joined an engineering task force to determine subsystem design requirements. Mr. Boynton then participated in advanced operations planning as a technical staff assistant to the Director of Flight Operations from 1965 to 1967, when he rejoined the Apollo Program as the Mission Engineer assigned to coordinate all trajectory and mission design aspects of manned lunar flights. In 1968, Mr. Boynton transferred to his current assignment of leading the mission evaluation and technical reporting for manned Apollo flights. Mr. Boynton's two aeronautical engineering degrees were conferred by M.I.T. and Cornell University. He is a member of AIAA.

Kenneth S. Kleinknecht was Manager of Project Mercury from February 1962 to its conclusion. Prior to that time, he was a project engineer at the NACA Lewis Research Center, the Advanced Projects Management Officer responsible for X-15 development at the Flight Research Center, and a Technical Assistant to the Director of the original Space Task Group at the Langley Research Center. In November 1963, Mr. Kleinknecht was appointed Deputy Manager of the Gemini Program. At the conclusion of Gemini, he was named Manager of the Command and Service Modules, Apollo Spacecraft Program, and was recently appointed to his present position of Manager, Skylab Program (formerly AAP). Mr. Kleinknecht has been recognized by NASA through three of its highest awards; the Outstanding Leadership Medal for his management of Project Mercury and the Exceptional Service and Distinguished Service Medals for his contribution to the Apollo Program. He is a member of AIAA.

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in Table 1. Prior to the recent aborted Apollo landing, all manned missions had experienced minor performance anomalies or partial systems failures but each mission had been overwhelmingly successful. This success can be directly attributed to the intensive ground qualification programs and to the attention given to engineering detail in all aspects of systems development, configuration management, and quality control.[8] The cryogenic tank failure during Apollo 13 is now reasonably well understood and is discussed later.

A problem that plagued Mercury development and later reappeared in lunar module design was weight growth. An overweight condition in systems development can be eliminated by either designing lighter weight systems or deleting nonmandatory components, but both approaches can directly affect systems reliability. Over a relatively short development period, the Mercury spacecraft increased in weight by 25%.[6] Although the Gemini spacecraft and the command and service modules also experienced weight growth, the problem was more pronounced on the lunar module.[9] The original design weight of the lunar module was about 22,000 lb, but a more realistic appraisal of system weights as the design progressed affected other systems, such as the landing gear and propellant tanks. The problem became so acute that an intense effort was initiated by the manufacturer[10] to eliminate all unnecessary weight through deletion or redesign, but without comprising systems reliability. The reduction

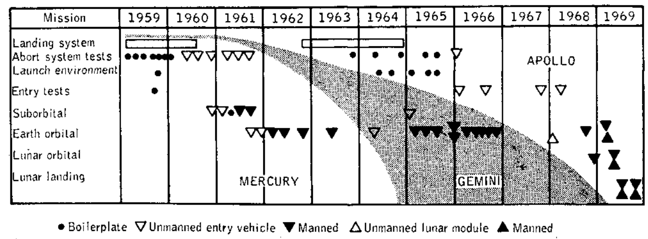

Fig. 1 Flight development summary.

was successful, and the Apollo 11 lunar module weighed 33,684 lb at separation, only 3% above the control weight set 18 months before flight. This weight increase had been verified by analysis to be within safe vehicle performance limits.

The development program for Project Mercury[1,3,11] began in the laboratory before the initial contract was awarded. Included in the various early design studies were wind-tunnel tests of entry shapes, structural analyses of high-temperature metals, and breadboard evaluations of critical electronic designs. Mercury flight testing began early in 1959, as shown in Fig. 1, with landing system tests from aircraft, closely followed with off-the-pad and inflight tests, using the Little Joe solid-fuel booster, of the launch escape system. This escape system used a large solid-rocket motor mounted on a tower above the spacecraft to be ignited in the event of a launch

Table 1 U.S. manned flight summary

Mission	Launch date	Description	Results[a]
			Mercury
MR-3	May 5, 1961	Suborbital qualification	1st U.S. manned space flight; 263-naut miles range.
MR-4	July 21, 1961	Suborbital	Repeat MR-3; spacecraft sank after premature hatch opening.
MA-6	Feb. 20, 1962	3-pass orbital qualification	1st U.S. manned orbital flight; instrumentation failure prompted retention of retropackage; thruster failure required manual entry; drogue deployed early.
MA-7	May 24, 1962	3-pass orbital	Horizontal scanner failure required manual retrofire, during which yaw error caused spacecraft to land 250 naut miles long.
MA-8	Oct. 3, 1962	6-pass orbital	Partially blocked coolant valve delayed suit-temperature stabilization; landed within $4\frac{1}{2}$ naut miles of ship.
MA-9	May 15, 1963	$1\frac{1}{2}$-day orbital	Short circuit in control electronics required manual entry.
			Gemini
III	March 23, 1965	3-pass orbital qualification	Demonstrated launch structural integrity.
IV	June 3, 1965	4-day orbital	1st U.S. extravehicular activity.
V	Aug. 21, 1965	8-day orbital	Qualified rendezvous radar operation.
VI-A[b]	Dec. 15, 1965	1-day rendezvous	1st U.S. closed-loop rendezvous; with spacecraft VII.
VII	Dec. 4, 1965	14-day orbital	Qualification for design duration; 1st controllable entry.
VIII[a]	March 16, 1966	3-day rendezvous	1st rendezvous and docking with another vehicle (Agena); short-circuited roll thruster terminated flight early.
IX-A[b]	June 3, 1966	3-day rendezvous	3 types of rendezvous; extended extravehicular activity.
X	July 18, 1966	3-day rendezvous	Onboard navigation only for rendezvous; 1st docked propulsion maneuvers (Agena); extravehicular activity.
XI	Sept. 12, 1966	3-day rendezvous	1st 1-revolution rendezvous; 1st tethered flight with another vehicle (Agena); extravehicular activity.
XII	Nov. 11, 1966	4-day rendezvous	Gravity-gradient stabilization with tethered Agena.
			Apollo
7	Oct. 11, 1968	11-day orbital qualification of command and service modules	Rendezvous and simulated docking with Saturn S-IVB stage; multispectral Earth photography; 8 propulsion maneuvers.
8	Dec. 21, 1968	Lunar-orbital qualification of command and service molecules	1st manned lunar flight; scientific photographs of lunar farside and Apollo landing zone.
9	March 3, 1969	10-day orbital qualification of spacecraft	Docked maneuvers with descent and service propulsion engines; 1st U.S. independent extravehicular activity; rendezvous.
10	May 18, 1969	Lunar-orbital qualification of spacecraft	Lunar module pass within 8 naut miles of lunar surface, approach photographs of Apollo 11 landing site, rendezvous.
11	July 16, 1969	Lunar landing	1st manned lunar landing; 2-hr 20 min extravehicular activity.
12	Nov. 14, 1969	Precision lunar landing	Touchdown within 600 ft of Surveyor III; two extravehicular excursions of 4 hr each.
13	April 11, 1970	Precision lunar landing	Aborted at 56 hours because of an oxygen systems failure.

[a] Gemini VIII is the only U.S. manned mission terminated early; all other flights completed successfully.
[b] Gemini VI and IX canceled after failure of Agena target vehicle and launch vehicle, respectively.

catastrophe. The Little Joe program was completed in 1961 and qualified the launch escape system in all critical flight regimes. Using an Atlas launch vehicle, an entry test and an exit heating evaluation were successfully performed on the spacecraft before entering the unmanned orbital flight program. In the Redstone rocket series, three unmanned suborbital flights were conducted before the first manned flight in 1961. In the final unmanned flights of the Redstone and Atlas series, primates were included to assess their physiological response to actual flight conditions. The Project Mercury orbital program began with two three-pass flights and was concluded with a 22-pass mission to evaluate man's response to space for the practical lifetime of the spacecraft.[8]

The Gemini development program[12] involved rigorous component, subsystem, and integrated systems tests. Using both prototype and production systems, these tests were so successful that only two unmanned flights were necessary prior to the first manned mission. All program objectives had been satisfied by Gemini VIII. The decision to abort during Gemini VIII was prompted by a mission rule that specified an expeditious return if an inflight failure occurred which left only one system to perform a critical function (one of two entry reaction control systems had been depleted to arrest a high roll rate).

In Apollo subsystem development,[13] all spacecraft components were ground tested[14,15] within expected operating ranges and at specified limits of heating, vibration, pressure,[16] and duty cycle. Flight testing was reserved for major systems verification after ground tests had fully qualified the equipment for flight. All normal modes of spacecraft operation were demonstrated in flight before the lunar landing attempt except those that required performance of the final descent (Fig. 1). The initial phase of the flight-test program included the development of a launch escape capability using an uprated Little Joe booster (Little Joe II); the launch escape vehicle was separated from the booster under predicted worst-case aerodynamic conditions. The last flight in this test series used a production spacecraft. Parachute landing tests were a key part of these early development flights. Ten unmanned flights using the Saturn I launch vehicle were flown primarily to test launch-vehicle flight systems, but the sixth and seventh flights of this series also tested the spacecraft-separation and launch-escape-tower jettison sequences. Boilerplate spacecraft for exit-heating and structural evaluation were flown prior to the sixth flight. In 1966, two flights, Apollo 2 and 3, were conducted to evaluate the uprated Saturn I launch vehicle for manned flight and to demonstrate the capabilities of service-propulsion-system operation and supercircular entry. The Apollo 4 and 6 flights removed all manning constraints on the Saturn V launch vehicle and demonstrated entry at lunar-return velocities. Critical launch structural data and propulsion system operating information also were obtained. Apollo 5 was an unmanned flight of essentially a complete lunar module to verify systems operation under orbital flight conditions. Previous testing of a fully equipped lunar-module test article in a thermal-vacuum chamber and the Apollo 5 orbital flight qualified the lunar module for manned earth-orbital missions.

Apollo Manned Flight Summary

The Apollo 7 Earth-orbital mission in 1968 proved the operation of the command and service modules for the maximum 11-day lunar mission duration.[17] The crew demonstrated transposition and simulated docking with the S-IVB stage, performed a rendezvous with the S-IVB, and executed eight service propulsion maneuvers of varying duration. Based on the excellent performance during Apollo 7, it was decided to launch Apollo 8, the first manned mission using the Saturn V launch vehicle, in December, 1968, on a lunar-orbital mission. During Apollo 8, all command and service module systems were verified for operation in the lunar environment, and photographs of lunar-surface features improved lunar mapping of prospective landing sites.[18] Apollo 9, the first manned flight of the lunar module, provided Earth-orbital qualification of the entire spacecraft.[19] In the 10-day mission, the lunar module descent engine was successfully fired in the docked configuration, extravehicular operations from both spacecraft were demonstrated, and the first lunar module rendezvous was executed. The ascent stage was then jettisoned, and the ascent engine was remotely fired to propellant depletion. The entire spacecraft was successfully tested in the lunar environment during Apollo 10, and a lunar landing, except for final descent and ascent, was simulated. In particular, the lunar landing time line was verified, and a rendezvous was performed under normal ascent-trajectory

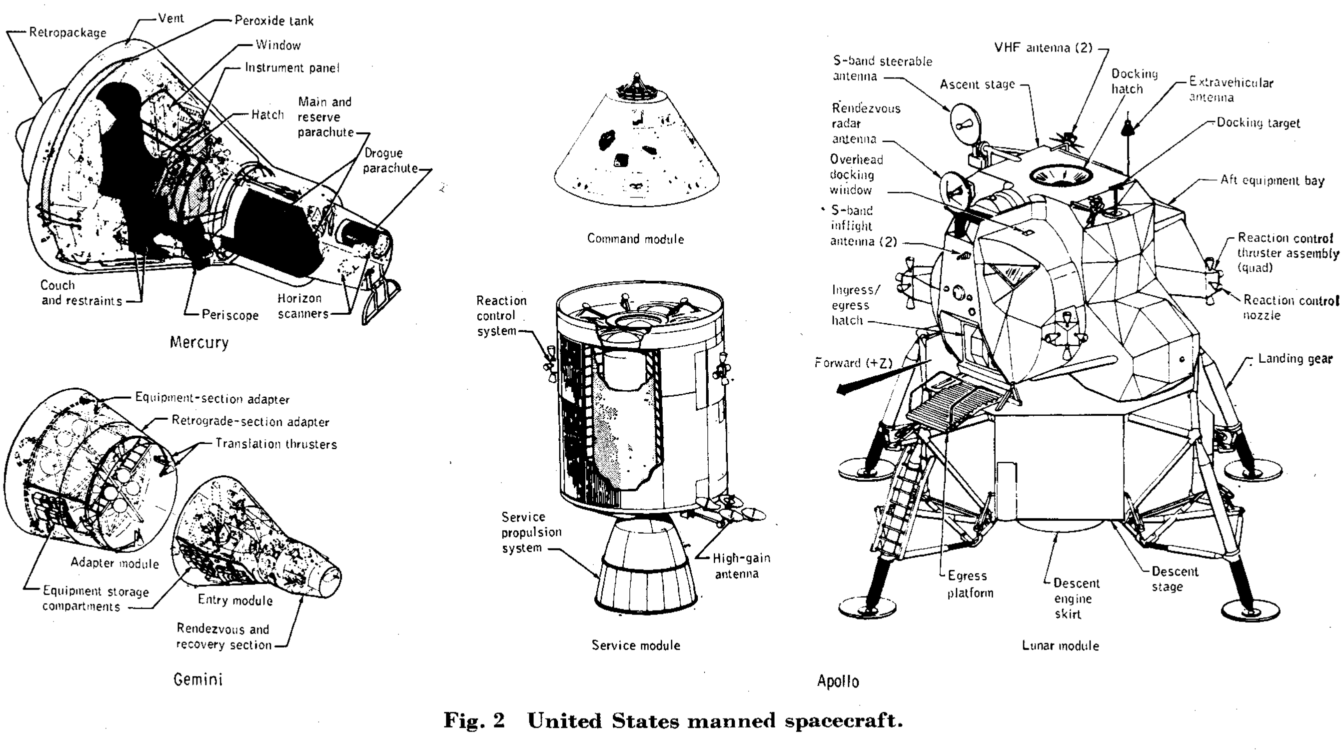

Fig. 2 United States manned spacecraft.

constraints,[20] Apollo 11, the first manned lunar landing, was conducted with extreme precision. All systems performed satisfactorily, and the crew guided the lunar module to a soft landing in the lunar Sea of Tranquility.[21] The two crewmen egressed to the surface, collected soil samples, and deployed three scientific experiments which included a passive seismometer, a solar wind composition shield, and a laser-ranging retroreflector. Performance of the pressurized lunar-surface suits, which allowed unexpected ease of mobility, and of the portable life support systems, was excellent. Lunar module ascent and rendezvous were normal. The Apollo 12 flight verified the feasibility of a nonfree-return translunar trajectory and demonstrated a precise point-landing capability by touching down within 600 ft of a Surveyor spacecraft, landed in 1967. An advanced experiments package was deployed,[22] and documented soil samples and Surveyor components were obtained by the crew in the nearly 8 hr spent traversing the lunar Ocean of Storms. In addition, high-resolution photographs of the Apollo 13 landing site were made from lunar orbit. The Apollo 13 mission was aborted 56 hr after lift-off because of a failure in the cryogenic oxygen storage system, but valuable information was obtained regarding emergency use of lunar module systems.

Spacecraft Design

The retention of program objectives was ensured by the use of certain over-all spacecraft design philosophies.[23] Although most spacecraft design guidelines[24] were ultimately proven to be sound, some are recognized to have been conservative.[11] For example, Mercury spacecraft automation in every function bearing on crew safety was largely unnecessary because of the pilot's effectiveness in executing these functions. In fact, the pilot's use of manual control modes made realization of mission objectives possible in all four manned orbital flights.[25] Manual attitude control was required in three of the orbital flights, and manual adjustment of the cooling circuit was required in the other flight. Table 2 compares certain features for the three spacecraft, which are depicted in Fig. 2.

Mercury[26,27]

The Mercury spacecraft, designed to support one man in Earth orbit, was developed with crew safety as the most important consideration. To ensure rapid development, Project Mercury depended heavily on existing technology and hardware,[11] including use of the Redstone and Atlas launch vehicles. System redundancy and the use of manual control modes for critical functions were employed where practical to meet the demanding reliability goals.[24] The exterior shape of the Mercury spacecraft was conical, with a segment of a sphere for the heat shield and a cylindrical afterbody at the apex of the cone. Two crew access hatches were included, one for entrance and egress on the side of the spacecraft and the other for exit through the cylindrical section of the spacecraft. A large window and an instrument panel were provided for crew monitoring of inflight events and system operation. The crew was supported in a contoured couch such that all large in-flight accelerations would be sustained through the astronaut's back. Environmental control was provided by evaporative cooling in two separate circuits, one for the cabin and the other for the pressure suit. A stabilization and control system, using a three-axis gyro package erected by horizon scanners, provided attitude reference for the displays and the two automatic control modes. Two manual control modes were provided for orbital attitude changes and as backups to automatic modes. The initial design incorporated a periscope for attitude reference confirmation prior to retrofire, but this device was later found unnecessary and deleted. Attitude changes were effected through a redundant system of reaction control engines, each axis having two thrust levels, using hydrogen peroxide activated by a catalyst as the sole propellant. Communication systems for voice, telemetry, tracking, and up-link command were installed in redundant units. Electrical power was provided through three silver-zinc batteries, and a complete sequential system initiated all critical spacecraft events. Although many types of atmospheric entry modes had been examined, the simplest concept of purely ballistic drag braking was chosen. Retrofire was effected by three solid-fuel rockets, of which any two were sufficient. It was stipulated that the simple, weight-saving water landing technique be used. The landing system included a ribbon drogue, ringsail main and reserve parachutes, and an impact-attenuation bag extended by the heat shield.

Table 2 Spacecraft comparison

Item	Mercury	Gemini	Command/ service modules	Lunar module
Crew size	1	2	3	2[a]
Insertion weight, lb	3000	8360	63,600[b]	32,650
Height, ft	9.6	19	36.4[b]	23
Maximum diameter, ft	6.2	10	12.8	31
Maximum thrust, lb	6	95	20,500	9,800
Translation, fps	0	800	10,000	13,200

[a] Lunar module supports two of the three command module crewmen.
[b] Command module weight: 12,100 lb; height: 12 ft.

Gemini[28,29]

Reliability experience from Mercury indicated that a failure could propagate from one major system to another when the systems were designed interdependently or when they were installed in a "stacked" manner with common interfaces.[30] As a result, Gemini systems were almost exclusively installed in a modularized fashion outside the inner pressure vessel and were accessible through large panels in the outer skin. Demonstrated crew capability in systems management prompted greater use of manual control, rather than directly automatic systems. What would have been a complicated sequencing system, for example, was quite manageable during design because of the primary manual modes introduced. Systems independence and the simplification afforded by manual modes also helped minimize a problem common to all three programs, that of electromagnetic interference and sneak circuits. The most important Gemini guidelines[30] were those providing maximum application to Apollo.[31]

The Gemini spacecraft consisted of an entry vehicle, including a rendezvous and recovery section, and a special adapter module. The outer equipment-section adapter contained primarily propulsion systems for in-orbit maneuvers and attitude control, fuel cells and reactants for electrical power, most of the communications and instrumentation electronics, and the coolant pumps and most of the thermal radiators used in the environmental control system. The retrograde-section adaptor housed a four-unit retrorocket system, six of the translation thrusters, and the remainder of the thermal radiators. The equipment-section adaptor was jettisoned immediately prior to retrofire, and the retrograde-section adapter was jettisoned prior to entry. The entry vehicle could support a two-man crew for as many as 14 days in Earth orbit and consisted of an outer thermal protection system for entry and an inner pressure vessel for shirtsleeve crew operations and for installation of equipment requiring pressurization. The entry vehicle contained the guidance and control systems for rendezvous and entry, batteries for entry power, a completely independent and redundant reaction control system, a parachute landing system, and the displays, controls, and communications equipment for crew management of onboard systems. Two large crew hatches with windows were provided which could be opened and reclosed in orbit to accommodate extravehicular activity. The crew stations were furnished with special ejection seats to be used in the event of launch or landing aborts. The parachute landing system was packaged in the rendezvous and recovery section, which was designed integrally with the drogue-like docking system on the Agena target vehicle.

Apollo[32,33]

The most significant design philosophy for Apollo systems was the requirement that the spacecraft be capable of autonomous return to Earth in the event of a total failure to communicate with ground stations.[34] This guideline, although conservative, required all navigation and abort calculations that would effect Earth entry to be performed onboard the spacecraft without benefit of ground assistance. Another design objective was to configure systems such that no single failure would result in the loss of the crew. This objective had been implicit, although unstated, in Mercury and Gemini and was satisfied in most instances in Apollo. An interesting design feature in this regard was the capability of the lunar module to support the crew for twice its design lifetime when the redundant oxygen system failed in the Apollo 13 service module, resulting in loss of primary electrical power. Although the lunar-orbit-rendezvous mode was very efficient in terms of spacecraft performance and weight, considerable system duplicity, such as in the guidance and life support systems, was added. For weight reduction, the lunar module was designed as a two-stage vehicle, with the descent stage used as the launching platform for the ascent stage. As in Gemini, manual control modes were primary in most systems except for routine functions or in the few cases where limited human response time could jeopardize crew safety. Because Apollo design was almost concurrent with early flight development, it was impossible to incorporate in initial flight articles all the design changes necessary for the lunar landing.[34] As a result, early spacecraft were categorized as Block I, and the majority of lunar mission design changes were reserved for Block II spacecraft, a management tool that eliminated considerable expense and production delays.

In addition to a launch escape tower and a spacecraft/launch-vehicle adapter, the Apollo spacecraft is composed of three identifiable modules (Fig. 2). The command module, which is the location for primary crew control of the spacecraft, is the only portion intended to return intact. The service module, attached to the blunt face of the command module, contains the service propulsion system for large spacecraft maneuvers and other systems that support operations in the pressurized command module. For example, the fuel cells and cryogenic hydrogen and oxygen supplies are located in the service module; whereas, the power distribution system and three entry batteries are located in the command module. The service module also contains the propellant tanks and engines of the primary reaction control system used to change or maintain attitude prior to entry. The engines are located in four units (quads) of four engines each around the periphery of the service module. The command module houses the guidance, navigation, and control systems; the communications and primary electronics equipment; the life support systems that regulate cabin and suit atmosphere; three couches and other provisions for the crew; a parachute landing system; and the necessary controls and displays. For entry, the command module houses two independently redundant reaction control systems and is covered by an ablative thermal protection system. A special boost-protective cover is attached to the launch-escape tower to minimize launch heating of the command module.

The lunar module is a two-stage vehicle used for descent from lunar orbit to the surface and return to orbit. The descent stage houses the descent propulsion system and four batteries for power during descent and surface operations. This stage is fitted with a four-legged landing gear and has provision for a lunar surface experiments package. The ascent stage has the capability to return over 100 lb of lunar samples and scientific equipment. The ascent stage contains the guidance, navigation, and control systems required for descent, ascent, and rendezvous; a complete communications, display, and control layout for manual operations; a life support system; and two batteries for power after lift-off. The ascent stage also contains a four-quad arrangement of reaction control engines and associated propellants for controlling vehicle attitudes during all separated phases. The two crewmen are supported at their standup duty stations by a special restraint system. A tunnel hatch is provided for intravehicular transfer between spacecraft, and a forward hatch is provided for egress to the lunar surface.

Major Subsystem Development

The greatest advances in technology from Project Mercury for which the Gemini Program was responsible are directly related to the more demanding mission requirements. Because of the advanced system capabilities demonstrated in the Gemini Program, totally new developments for Apollo could be primarily confined to the large propulsion systems required for major maneuvers, entry technology to accommodate lunar-return velocities, a lunar landing gear, and an independent life support system for lunar surface exploration. Although a guidance and control system was developed for the Gemini spacecraft, the increased complexity and autonomity of the lunar landing profile made this system development the greatest single advancement from Gemini technology. The significant development accomplishments made since Project Mercury are summarized in Table 3, and major subsystem development programs are discussed in the following sections according to specific design disciplines.

Entry Shape and Thermal Protection

The Mercury spacecraft shape evolved from a series of wind-tunnel studies that showed that a blunt entry configuration could best accommodate the high convective heating of a hypersonic entry. The conical afterbody originated from the consideration to minimize the total weight of thermal protection by placing as much surface as possible out of the incident flow. The cylindrical section was necessary for attitude stability in low supersonic flow, as well as to provide for convenient packaging of the landing parachutes. The original Mercury shield was a beryllium heat-sink type used only in the suborbital flights.[26] However, entry from orbit would have caused excessive heat soakback with this design, and a fiber-glass shield in a layup construction was developed. For simplicity, the entry was specified to be purely ballistic, with initial attitude stability provided by a preentry rolling maneuver. Since the lunar mission would later require a lifting body to widen the entry corridor,[35] the Gemini spacecraft was designed with a lift capability to support Apollo, as well as provide landing point control.

Performance of the Mercury heat protection system in all flights exceeded original estimates.[6] In the fabrication of the Mercury heat shield, the parallel fiber-glass layups, which were bonded to a structural shield, caused a considerable problem with bonding integrity. Although the manufactured shields were x-rayed to disclose voids in the bonding surface, it was almost impossible to achieve complete surface contact across the entire heat shield. As a result, minor delamination at the bond line was evident in all orbital flights but was never extensive enough to compromise heat-shield performance. For the Gemini and Apollo spacecraft, this bonding problem was eliminated by using a nylon-phenolic honeycomb core bonded to a structural substrate and filled with an ablator.[36] Since the basic Mercury conical shape was retained through the Gemini and Apollo Programs, much of the early wind-tunnel and thermal-analysis results could be transferred to both programs. The greatest unknown during initial design of the Apollo heat shield was the magnitude of the nonequilibrium radiation that would be encountered at near-parabolic entry speeds.

As with Apollo, the Gemini spacecraft used an offset center of gravity to obtain lift,[35] but the angles of attack and associated temperatures for a Gemini entry were small enough that the radiative-type shingles used on the Mercury afterbody could similarly be retained. Use of these shingles permitted the modularized installation of systems with access through panels. The Gemini heat shield used a silicone elastomer with glass eccospheres for the ablator, and the Apollo heat shield employs a low-density resin (epoxy novalac) with phenolic microballoons for improved low-temperature characteristics. The eccospheres and microballoons were added to reduce ablator density and over-all spacecraft weight. The severe thermal conditions of a lunar-return entry required a rounding of the heat-shield edge to minimize local heating effects.

The exterior surface of the stainless-steel Apollo substrate[37] (Fig. 3) is covered by the ablator in a thickness that varies with the local thermal loading. Once the honeycomb cells have been filled, the ablator is oven cured, inspected, and contoured in a three-dimensional milling machine. In sizing the thermal protection system for Apollo,[38,39] extremes of the probable guided-entry trajectories were examined to determine conditions of maximum heating rate and heat load, based on a lift-to-drag ratio of 0.5, a trim angle of attack of 33°, and an entry speed of 36,333 fps. The maximum permissible bond-line temperature was 600°F. As a result, the spacecraft entry limits (Fig. 4) were the overshoot boundary, defined by a 5000-mile range, and the undershoot boundary, defined by a structural limit of $20g$ acceleration. Later, as the command module design progressed, it became impossible in packaging systems to obtain a center-of-gravity offset sufficient for a 33° angle of attack; therefore, a new lift-to-drag ratio of 0.35 was assumed. Also, to reduce ablator thickness and weight, the ranging requirement was decreased to 3500 miles. As a result, both boundaries of the entry corridor[40] were altered, and the heat shield was modified accordingly.

The heat-of-ablation approach to the Apollo heat-shield analysis was considered expedient for initial design purposes. But a more complete analysis[41] based on a charring model and an energy balance at the char surface was necessary to verify the final design. To support heat-shield development, numerous ground tests were conducted, but the results were limited by the scale of the tests. Four full-scale flights (Fig. 1) were therefore conducted to qualify the thermal protection system. The first two flights were at supercircular entry speeds, corresponding to orbital return extremes, and the final two flights demonstrated performance at lunar-return speeds. Data from these flights permitted a description of thermal protection dynamics with a single analytical model. Although the Apollo thermal protection system has not been

Table 3 Major subsystem comparison

Design concept	Mercury	Gemini	Apollo: Command and service module	Apollo: Lunar module
Entry shape	blunt cone with cylindrical afterbody	same as Mercury	blunt-cone command module	
Thermal protection	fiber-glass ablator on blunt face; high-temperature shingles on afterbody	same as Mercury except for silicone elastomer ablator	ablator similar to Gemini of varying thickness around entire command module	multilayer reflective insulation
Launch escape	solid-fuel rocket mounted on tower	ejection seats to 45,000 ft, retrorockets above 15,000 ft	same as Mercury	
Life support	pure O_2 atmosphere, water evaporators for cooling	same atmosphere; radiator for primary cooling, secondary evaporators	same as Gemini	pure O_2 atmosphere; water sublimators for cooling
Attitude control	hydrogen peroxide monopropellant; redundant systems	hypergolic propellants; ablatively cooled engines; redundant entry systems	same as Gemini except radiatively cooled engines in redundant coast system	same as service module coast system
Maneuver propulsion	none	95-lb thrusters using same propellant as attitude control	20 500-lb thrust service propulsion system, pressure-fed hypergolics	3500-lb ascent and 10 500-lb-thrust throttleable descent engines
Retrograde propulsion	three solid-fuel rockets	four solid-fuel rockets	maneuver propulsion used	
Onboard control	body-mounted gyro stabilization; horizon scanner reference	inertial platform (four gimbal); horizon scanners; digital computer; rendezvous radar	inertial platform (three gimbal); digital autopilot and computer; optical alignment; VHF ranging	same as command module but with rendezvous and landing radars
Electrical power	three silver-zinc batteries	fuel cells with batteries for backup and entry	same as Gemini	four descent and two ascent batteries
Communications	UHF, VHF voice and PAM telemetry; C- and S-band tracking; command link	same as Mercury except PCM telemetry	unified S-band for all functions, including television; VHF voice	same as command module plus extravehicular communications
Landing system	one drogue; one main and one reserve parachute; landing bag	one drogue; one main parachute; crushable structure	two drogues; three main parachutes; crushable structure, stroke couch	four-leg landing gear with crushable honeycomb
Pressure suit	backup to cabin atmosphere control	extravehicular type with spacecraft umbilical	Extravehicular with umbilical	extravehicular with independent life support

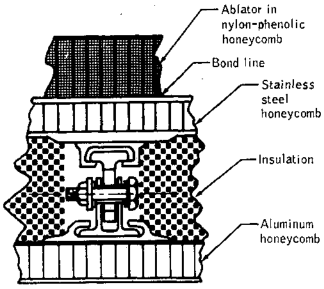

Fig. 3 Apollo heat-shield structure.

tested to its design limits, significant design conservatism is apparent from all flight results.

Life Support

Development of life support systems throughout the three programs was probably the most consistent of all subsystems and therefore required the least number of significant advances in technology. Nevertheless, problems, such as the presence of condensation in the cooling circuits and the inability to extract the condensation, have appeared in all three spacecraft. While the science of heat rejection was well understood before Mercury, the mechanics of implementing compact and efficient environmental control for the crew and equipment have presented a continuing development effort.[42] In the Mercury spacecraft, the thermal analysis and cooling-system design were not as conservative as in the design for other systems; cooling system performance margins were therefore not sufficient to handle peak thermal loads.[6] Even under normal loads, the environmental control system was operated in a close adjustment range, thus increasing the importance of control instrumentation. In the suit cooling circuit, excessive waterflow resulted in freezeup of the heat exchanger and flow rates any lower were at times inadequate to cool the crewman.[25,43]

As mission durations increased, water evaporation as a primary means of heat rejection was abandoned in favor of more efficient radiative systems. The large Gemini adapter module and the Apollo service module were particularly well suited to installation of radiators. Since radiators are ineffective during launch and unavailable during entry, water evaporators were incorporated into both entry vehicles as secondary cooling devices. These evaporators are also serviceable, if required, at other times in the mission to supplement the radiators in handling peak thermal loads. The primary heat rejection system in the lunar module is a water-type device because radiators are not effective near the lunar surface.

The most serious inflight problem with the Mercury life support system occurred during the third manned orbital

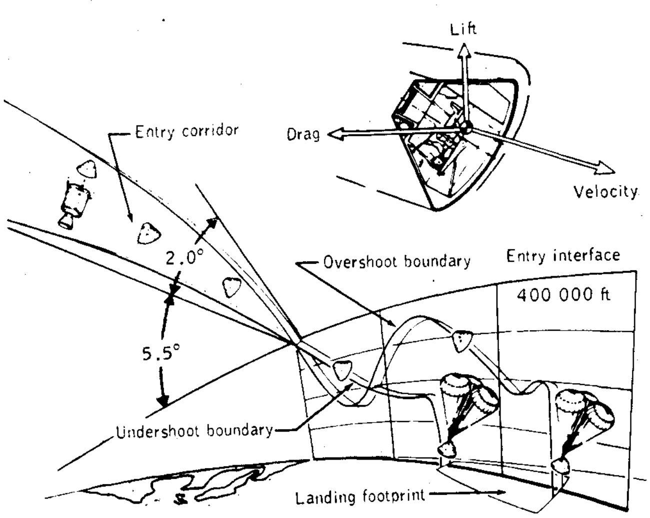

Fig. 4 Lunar-return entry boundaries.

flight, the first extension of orbital mission duration. The suit-inlet temperature failed to stabilize after orbital insertion, and initial control-valve adjustments were unsatisfactory.[43] While consideration was being given to terminating the flight early, the pilot methodically brought the system under control. A postflight inspection revealed that excessive dried lubricant had clogged the tiny coolant control valve and changed the valve flow characteristics from the preflight calibration. This partial clogging demonstrated early in manned flight development not only the need to avoid the use of small orifices in system design, but also the need for controlling seemingly incidental processes such as valve lubrication.

The Gemini life support system operated satisfactorily throughout the 10 Gemini missions, including the record 14-day flight of Gemini VII. As with other systems, qualification of the environmental control system was demonstrated largely at the system level, rather than with separate components, because of the close interaction of all subsystems. Qualification tests were followed by long-duration reliability tests in an altitude chamber. These simulations revealed that the heat transfer from the lithium hydroxide canister to ambient was greater than expected, and the increased heat flow caused chilling of the gas stream, resulting in unwanted condensation of water. The condensate collected and reduced the life of the chemical materials in the canister by 45%. The lithium hydroxide canister was subsequently redesigned to incorporate insulation between the chemical bed and outer shell. Probably the most serious challenge in the design of the Gemini life support system was the accommodation of new extremes in metabolic and electronic-system thermal loading. Unlike the Mercury spacecraft, Gemini life support required provision for strenuous extravehicular operations, as well as widely changing equipment operating cycles. Minor modifications were made throughout the Gemini flight program to make the radiator cooling circuit more efficient under these loading extremes.[28]

The environmental control systems in the Apollo command[44] and lunar modules[33] are similar but independent in their operation. The command module system was designed to control electronic equipment temperatures and to support three men for 14 days in a pressurized oxygen atmosphere.[45] The system also removes trace contaminants and metabolic carbon dioxide. The main applications to Apollo from previous programs were primarily in the methods of metabolic condensate collection and water evaporator control.[6,30] The knowledge gained from observing liquid behavior at zero gravity led to the incorporation of a wick-type porous-plate condensate separator, instead of the sponge-type collector used previously. Also, difficulties experienced with the Mercury evaporators prompted the use of a backpressure evaporation temperature control and of thermistor sensing of excess water. During coast phases, thermal control of all Apollo systems was made more effective by introducing a slow rolling motion about the spacecraft longitudinal axis, perpendicular to the sun line, to distribute solar energy more evenly.

Inadequate water demand caused repeated evaporator dryout during the first four manned Apollo missions.[44] This condition was attributed to a false sensing of water presence around the sensors in zero gravity; however, the radiators were sufficient to maintain a habitable environment without the water evaporators. A modification to the evaprator corrected the operating deficiency, and the system performed well during Apollo 11 and 12.

Although the basic technology that existed during the design of the lunar module environmental control system was applicable, the operating concept chosen differed from that of previous spacecraft. The lunar module required an environmental control system that could remain quiescent for 3 days and then be activated for 2 or 3 days. This requirement dictated a need to preserve expendables while meeting stringent performance requirements during a relatively short operating life. The system design was made more efficient by using water sublimators, instead of traditional evaporators, for heat rejection to take advantage of the latent heat of fusion. A sublimator, however, is poorly adapted to on-off switching, and several obstacles[46] were encountered in developing an operable system. Manufacturing of the porous plate for the sublimators was found to be a difficult task. Heat transfer fins could not be brazed onto the plates consistently without causing essential pores to be sealed. The problem was corrected by discretely welding the fins to the plates. Performance of the lunar module life support systems has been excellent, particularly in its backup role during Apollo 13.

The development of crew support equipment for the Apollo Program stemmed largely from experience gained during Mercury and Gemini. The most significantly new mission requirement for crew equipment pertained to lunar surface exploration. Other items, such as food, tools, waste management, and crew restraints, were outgrowths from earlier development. All Gemini extravehicular activity[7,47] was accomplished by using a life support umbilical; whereas, Apollo extravehicular activity required two crewmen to be equipped independently of the spacecraft for greater than 4 hr of lunar surface activity. To accomplish this goal, an extravehicular mobility unit (Fig. 5) was developed to include the pressure suit, a backpack, and a remote control unit, in addition to an integrated communications transceiver. The backpack consists of a portable life support system, that supplies oxygen and coolant water to a newly developed liquid-cooling garment,[48] and a backup oxygen system, that can provide 30 min of regulated flow for contingency cooling and breathing. This system was successfully tested during Apollo 9[19] and functioned as expected during Apollo 11 and 12 lunar surface operations.[21] Suit mobility was reported as excellent.

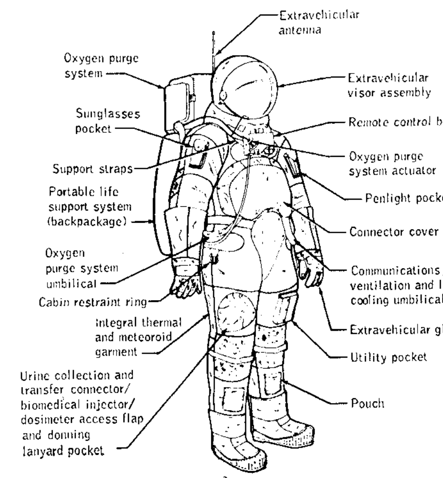

Fig. 5 Extravehicular mobility unit.

Propulsion

Propulsion system development did not become a major part of manned spacecraft design until the Apollo Program, even though the Gemini spacecraft had orbit maneuvering capability. The Gemini system employed essentially a scaled-up reaction control thruster with a maximum thrust rating of 95 lb. This engine was ablatively cooled and, therefore, had a limited lifetime when used for continuous thrusting. The retrograde maneuvers in both Mercury and Gemini were performed with independent solid-fuel rockets, the same type of propulsion used in the Mercury and Apollo launch escape systems[49] for reliability reasons. None of these solid-fuel systems has ever presented a real development problem.

The Mercury reaction control thrusters were a constant source of concern. Some units exhibited intermittent behavior, and others were direct failures. Early development tests, under the limitations of Earth gravity, had failed to reveal these problems, but a major thruster redesign late in the program proved successful. This modification[6] altered the thermal characteristics so that heat soakback was reduced and orifice contamination was eliminated.

In Gemini V and VII, two yaw engines became inoperative[28] late in the mission. Since these engines were not recoverable, postflight inspection was impossible. A probable cause for the inoperative engines was temporary freezing of the oxidizer lines since other thrusters had failed in a similar manner but later operated satisfactorily. In general, the attitude and translation thrusters performed effectively in all Gemini orbital flights.

The three primary propulsion systems[50] in the Apollo spacecraft are located in the service module and in the lunar module descent and ascent stages. The service propulsion system is required to supply the propulsive impulse for lunar orbit insertion, lunar module rescue (if required), trans-Earth injection, and any large midcourse corrections.[51] A previous effort that contributed technology to the design of the service propulsion system was a 2200-lb-thrust pressure-fed engine used in an unmanned satellite program. All three propulsion systems require multiple starting, and the descent engine must also be throttleable. Simplicity in multiple starting was achieved in all three systems through the use of an ablatively cooled engine pressure fed by hypergolic propellants. Because of component redundancy (e.g., descent engine, Fig. 6), no single component malfunction in the propellant feed system will cause engine failure.

A problem common to each propulsion system during engine development was that of obtaining acceptable performance while maintaining injector/thrust-chamber compatibility and engine combustion stability.[13] Except for the descent engine, which appears to be inherently stable, combustion stability was not easily achieved. Self-induced combustion instabilities occurred in the development of both the ascent engine and the service propulsion engine. Baffles were incorporated into the design of the injectors for both engines, and with a series of modifications to injector configurations, the instability problems were eventually solved.

Even with discrete baffles, such severe combustion instability problems were encountered with the ascent engine injector that a parallel development effort was initiated. The originally developed injector incorporated three baffles, a center ring, and a groove-type combustion stabilizer. The

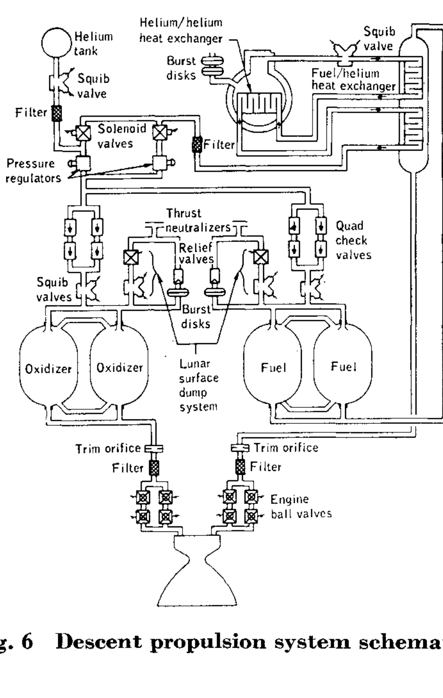

Fig. 6 Descent propulsion system schematic.

parallel development, from which the final configuration was selected, produced an injector with three slightly longer baffles, plus acoustic cavities around the injector periphery. Stability tests, which consisted of detonating an explosive charge in the combustion chamber to perturb the combustion process, were conducted on all engines to ascertain proper recovery within a 20-msec time period. These tests ultimately served to qualify all three engines for flight.

Attainment of engine performance was a continuous tradeoff with achievement of satisfactory thrust-chamber durability. Injectors with excellent performance exhibited detrimental thrust-chamber gouging or uneven throat erosion. As a result, the original service propulsion quadlet-pattern injectors were changed to a doublet configuration, but at the cost of engine performance. A later change in the propellant mixture ratio and refinements in the injection pattern were necessary to produce acceptable engine performance. Durability of the ascent-engine thrust chamber was achieved by improving propellant distribution from the injector and by the use of film cooling.

Two of the three systems exhibited serious start-transient problems, in that either the peak pressure force exceeded the initial design limits or the rate of pressure increase was too great. The start-transient problem was solved in the service propulsion engine by using only one bank (bore) of the series-parallel propellant (ball) valves (e.g., descent engine, Fig. 6) for the first 5 sec after ignition, thereby limiting the pressure overshoot. The use of one valve bank was not acceptable for starting the ascent engine because of the crew safety implications. Although an extensive program was conducted to improve the start-transient characteristic for this engine, an acceptable solution was not found. Subsequent analysis indicated that the spacecraft structure could sustain the start transient, but a weight penalty was incurred when the thermal shielding was increased in thickness to accommodate impingement forces. Performance of all three engines has been essentially flawless during six manned flights.

In the Apollo reaction control engines, the major development problem was to design a system in the service module with a long continuous-thrust lifetime, in addition to a multiple-restart capability. In the Gemini spacecraft, the engine nozzles were ablatively cooled and were therefore relatively life limited when used for pure translation.[28] The reaction control engines for the service module were designed to be radiatively cooled and have demonstrated burn durations exceeding 1000 sec in ground tests. In all six manned Apollo missions, no failure of a service module reaction control thruster occurred. The command module reaction control system, which uses engines nearly identical to Gemini translation thrusters, has also operated satisfactorily.

Onboard Guidance and Control

State-of-the-art advances[52,53] from Mercury to Apollo are most profound in the category of onboard guidance and control. In Mercury, the basic requirement was to provide an attitude reference system that permitted automatic deorbit and entry in the event of a contingency. The Mercury stabilization and control system used body-mounted gyros to provide attitude reference and to maintain orientation within a specified dead band. Manual modes were also provided which permitted direct attitude control by the pilot. In the first manned orbital mission, intermittent failures in the small reaction control thrusters necessitated pilot control of attitudes for entry. In the second flight, a bias in the pitch horizon scanner required a manual retrofire, during which an error in yaw attitude caused the spacecraft to overshoot the target by 250 miles.[6] In the last manned orbital mission, a short circuit in the control electronics[6] completely disabled the autopilot and necessitated astronaut control of both the retrofire and the entry maneuvers.[25] The effective way in which all pilots were able to control their vehicle prompted the incorporation of primary manual control modes in both the Gemini and Apollo spacecraft.

The design requirements for the Gemini guidance and control system stemmed largely from the rendezvous[54] and controlled entry objectives.[55] The rendezvous operation required both onboard navigation and computation capabilities.[56,57] The navigation function was satisfied by a four-gimbal inertial platform, and the computation capability was provided by a small digital computer. The computer was designed with a self-checking function to prevent procedural errors and input overloads, a feature ultimately found to be very important to Apollo design. To improve over-all navigation accuracy, a rendezvous radar was incorporated.[28,29] The precise rendezvous maneuvers and the various manual control modes required new display concepts for the crew.

The mission requirements placed on the design of Apollo guidance, navigation, and control systems are evident in each phase of the lunar landing mission. During launch, the command module guidance system is required to display critical data to the crew for monitoring the launch trajectory. These data, consisting of altitude, velocity, and vehicle attitude, are generated and displayed by the command module computer. A backup interface with the launch vehicle guidance system is

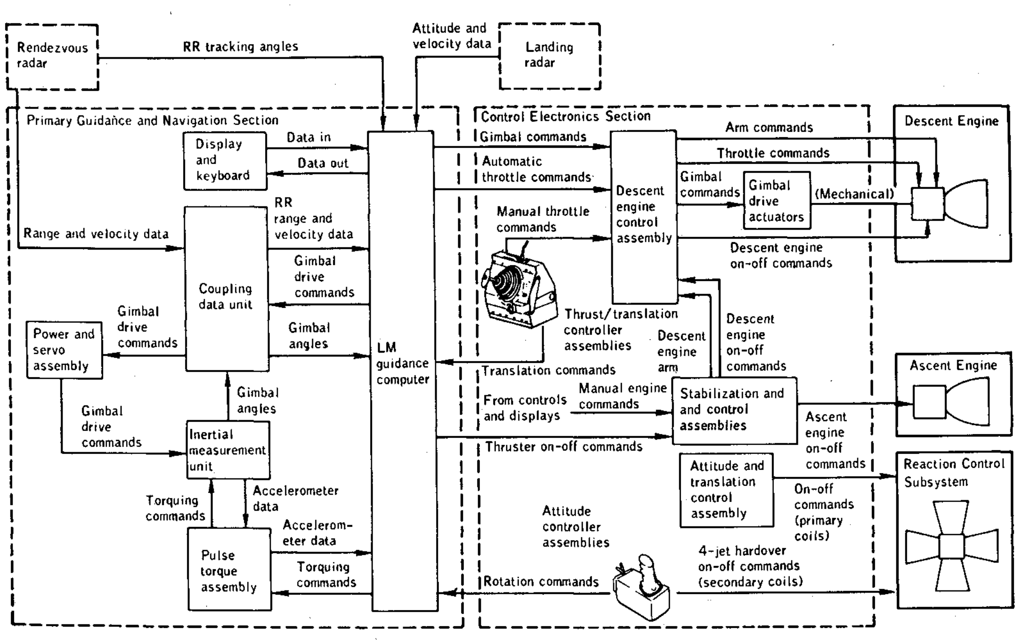

Fig. 7 Lunar module primary guidance schematic.

provided between the respective inertial platforms so that, for certain types of launch-vehicle failures, the spacecraft can provide navigation data to complete orbital insertion.

In the Earth-orbit, translunar, and trans-Earth coast phases, the command module inertial platform is required to produce navigation data necessary to maintain a valid onboard state vector (position and velocity). To meet this requirement, an inertial-platform alinement capability was needed. During Earth parking orbit, angles to known landmarks are entered into the computer, and during cislunar flight, sightings of stars and of either the lunar or Earth horizon are used. A special navigation computer program is then used to calculate a new state vector by applying a mathematical filtering technique. The navigation system is also designed to accept and process state-vector updates from the ground produced from Earth-based tracking data.

Satisfactory functioning of the command module guidance system is most important immediately prior to and during lunar-orbit insertion. The guidance system must project the state vector to the desired pericynthian, or lunar arrival, condition and determine the specific maneuver required in order to enter a 60 × 170-naut mile orbit. This inertion maneuver requires the longest duration firing of the service propulsion engine; therefore, the maneuver produces the largest center-of-mass shift and the greatest three-dimensional lunar gravity effects. The spacecraft must, therefore, be guided to precise cutoff conditions, followed 4 hr later by a brief circularization burn. These two maneuvers and trans-Earth injection are performed behind the moon and out of sight of Earth tracking stations.

The role of the guidance and control systems in the lunar module is to perform the descent to[58] and the ascent from the lunar surface, followed by a precise rendezvous sequence. Since lunar module guidance[59] must be precise because of profile intricacies, a landing radar and a rendezvous radar were incorporated to provide a continuous update of the respective relative positions. Data from these radar systems are compared with at least one independent source by the computer for validity. Figure 7 depicts the operational paths for the lunar module primary guidance system, which is functionally comparable to the command module system. Because the guidance function is so critical, an abort guidance system was added to perform ascent, rendezvous, or an abort from descent if a primary system failure occurs.

The trans-Earth injection maneuver and any subsequent midcourse corrections are targeted to achieve the desired conditions at Earth entry. These conditions, sometimes referred to as the entry corridor (Fig. 4), are defined as a 2° range of acceptable flight-path angles for a given velocity at a 400,000 ft altitude. The flight-path angle constraints are established by the undershoot and overshoot trajectory boundaries. During entry, roll attitude commands are generated by the guidance system to achieve the desired entry range by modulating the lift vector. As a backup to the primary guidance system, an entry monitor system was incorporated to permit manual takeover[60] along a more simplified trajectory (constant acceleration).

The development of the Apollo guidance, navigation, and control systems was aided by experience derived from the Polaris Program in the use of gyros, accelerometers, and platform electronics. The major new development tasks were the optics design[61] and computer program construction.[62] Also, the use of a digital autopilot in the Apollo spacecraft represented another innovation in manned spacecraft guidance.[63] Therefore, the crew interfaces for these design features and the associated software were major development hurdles.

In the development of Apollo guidance and navigation hardware, only the fixed-memory (hardwired) storage of the guidance computer presented a major hardware obstacle.[64] The manufacturing procedures required to hardwire permanent memory posed a serious difficulty early in the program, and the leadtime in defining individual flight software[65] for this process was at first excessive. However, later improvements in the manufacturing process reinstated much of the desired schedule flexibility.

Table 4 Apollo midcourse corrections

Mission	Total midcourse velocity change required, fps: Command module	Lunar module
8	4.8	...
9	...	1.4
10	1.6[a]	3.35
11	4.7	2.3

[a] Applied 3 hr before entry to adjust to center of corridor.

For both Apollo systems, the major development obstacle was the definition and resolution of each item of mission software. The computer programs necessary to operate on all mission parameters required definition within the constraints of memory size and man-machine limitations. Following definition, each set of software required functional verification within the tightly woven matrix of digital operations.

In all Apollo missions, system accuracy and response were better than had been anticipated. The greatest success in command module guidance was probably the Earth landing accuracy achieved in Apollo 8 when the spacecraft landed at lunar-return conditions within 1 mile of the target point.[18] In Apollo 10, the trans-Earth injection was performed so precisely that no midcourse corrections were required to enter the narrow entry corridor. As a direct measure of systems performance, the trans-Earth and rendezvous midcourse correction values are presented in Table 4. The precision with which the Apollo 11 and 12 landings were performed confirms the satisfactory development of the lunar module guidance and control technology. Ironically, it was a self-checking feature of the lunar module computer[62] that caused concern during the Apollo 11 descent. An overload in computing caused by parallel but nonmandatory, rendezvous radar calculations resulted in several computer alarms. Information available to ground controllers, however, indicated safe continuance despite the loss of minor displayed data. An effective man-machine interface in the descent guidance routine made possible the use of judgment in continuing the mission where an abort might otherwise have been required. Precision operation both in ground-based navigation and onboard control permitted the Apollo 12 lunar module to land within 600 ft of the Surveyor spacecraft.

Electrical Power Generation

Silver-zinc batteries satisfactorily supplied the total primary electrical power for the 36-hr maximum duration of the Mercury spacecraft.[6] The only serious electrical problem encountered during a Mercury flight occurred in MA-9 when an unprotected power plug short circuited because of the free water in the cabin.[6] Gemini mission durations ranged from a few minutes to 14 days. Although the use of fuel cells was originally planned for all Gemini flights, late availability of the fuel cells forced the use of batteries as the primary power source on several of the early missions. Because batteries are an inefficient source of power for periods exceeding 4 days (Fig. 8),[66,67] the longer term fuel cells were required to support

Table 5 Gemini power source comparison[67]

Source	Weight, lb: 2 days	Weight, lb: 2 wk	Volume, ft^3: 2 days	Volume, ft^3: 2 wk
Batteries	647	3296	4.3	22.2
Solar cells	739	739	40	40
Fuel cells	279	497	8	15

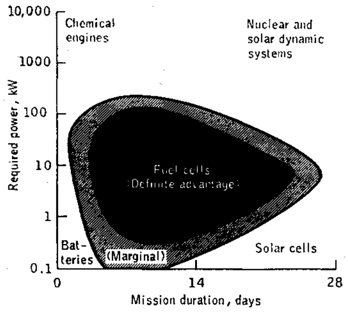

Fig. 8 Power source comparison.[67]

the 8- and 14-day flights. A comparison, shown in Table 5, of weights and volumes for three candidate power sources clearly shows the advantage of fuel cells for a 2-week mission.

The Gemini fuel cells were hydrogen-oxygen acidic-type cells which used an ion-exchange membrane with catalytic electrodes.[29] Problems were encountered in both the cryogenic reactant supplies and the fuel-cell stacks during some Gemini flights, but subsequent modifications produced a fully operational system late in the program. For example, thermal leaks and a failed heater circuit were experienced in Gemini V, and the inability to vent hydrogen in Gemini VII caused water buildup in one cell which sharply reduced its performance. Over-all design and performance of the Gemini fuel cells were satisfactory and provided an important base line for Apollo development.

As in Gemini, the Apollo power generation system was designed to store electrical energy both in batteries and in the form of cryogenic reactants for the fuel cells.[68] Command module power is also provided to the lunar module during translunar coast for operating critical electronics. Three Bacon-type fuel cells (Fig. 9) provide primary electrical power for the command and service modules. The average power level required from each cell is 600 w; however, in a partially powered-down mode, the spacecraft can return to Earth from any point after loss of two fuel cells. Water for crew consumption is also produced as a byproduct of the fuel cell reaction.

The cryogenic storage system supplies the reactants to the fuel cells and metabolic oxygen to the life support system. These fluids are stored in two hydrogen and two oxygen tanks that are sized so that a safe return is always possible if one tank in each supply is lost. The three command module batteries accommodate various sensor loads and share fuel cell loads during periods of high current drain, such as in service propulsion maneuvers. These batteries also provide the total command module power from service module separation (15 min before entry) through landing as well as for post-landing operations. For reliability, two separate batteries are provided to activate pyrotechnic devices. The lunar module batteries were substituted at a weight penalty for the originally planned fuel cells, primarily for reliability considerations. Pyrotechnic devices are also fired by two batteries identical to those used in the command module.

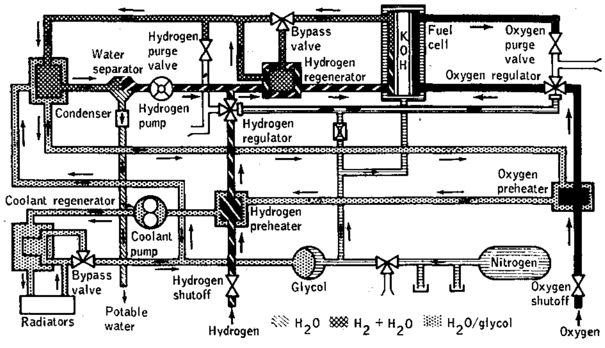

Fig. 9 Fuel cell schematic.

The fuel cells were the first major subsystem to be qualified in the Apollo Program, although several early problems were encountered. Numerous design concepts of electrode material, membrane/pore configuration, and bonding techniques were evaluated prior to selection of an electrode design. The development of electrolyte seals that would retain shape and maintain sealing capability at 500°F was successful. The major problems encountered in cryogenic system development resulted from the structural design and fabrication of the pressure vessels. Proof-pressure tests conducted revealed premature tank failures caused by titanium creep. The problem was solved by reducing the design stress on these tanks to 75% of the yield strength. Exposure of pure titanium tubing to hydrogen resulted in a titanium hydride formulation, which caused spalling of the metal and eventual leaks. Changing the tubing material from titanium to stainless steel eventually solved the problem.

In the Apollo flight program, the command module batteries developed internal shorting because of zinc penetration of the permion separator material. This unique material was selected instead of the widely used cellophane separators because of an initially specified temperature requirement of 120°F. Since this temperature level was never observed during flight, the cellophane separators were incorporated in Apollo 10 and subsequent batteries with excellent results. The fuel cells satisfactorily provided primary power for two unmanned and six manned missions. However, a few anomalies occurred which placed operational constraints on individual cells. The occurrence of a high condenser-exit temperature caused a fuel cell to be placed on an intermittent open circuit during portions of the Apollo 7 and 9 missions. All spacecraft have exhibited condenser-exit-temperature oscillations in varying degrees, but these oscillations are not divergent and have not affected system performance. During Apollo 11, a short circuit opened the circuit breaker for a fuel cell hydrogen pump and thus restricted its operation. On the Apollo 7 mission, an electromagnetic interference problem that caused the inverters to be removed from the bus was evident in the cryogenic fan-motor switch. The problem was circumvented by employing manual activation of the fan motors. Probably the greatest testimony to comprehensive power system design was the ability of the command module power distribution system to properly switch over to battery power when the Apollo 12 space vehicle sustained two potential discharges during launch which dropped out fuel cell power.[21]

During Apollo 13,the sudden loss of pressure in one of the two cryogenic oxygen tanks resulted in the eventual loss of pressure in the other tank and consequently all fuel cell electrical power. This condition prompted an immediate abort of the mission. Exhaustive post-flight testing and examination of flight data resulted in the conclusion that the most probable failure source was an abnormal detanking procedure during a routine preflight test at the launch site which caused thermal overload switches in a tank heater circuit to fuse together. The abnormal procedure was required because the tanks could not be emptied in the normal manner for an unknown reason. This shorted condition probably damaged wiring to a fan inside the tank which is in the vicinity of the heater wires. The heaters are used to increase tank pressure when required, and the fans are used periodically to maintain a near-constant density of the fluid. It is then hypothesized that the fan wiring shorted and caused ignition of certain materials within the tank which ultimately raised the temperature and pressure of the fluid beyond the structural limit of the tank or one of its components. The sudden rupture somewhat damaged the remaining tank or its plumbing, and it gradually lost pressure. The rupture of the first tank also resulted in an over-pressure in the associated service module bay which separated the outer skin panel. A satisfactory redesign to avoid this electrical malfunction is still under development.

Communications and Instrumentation

The design and development of the communications and instrumentation systems[32,33] are discussed concurrently since the systems are interdependent. The fundamental design requirements for these two systems in the Apollo spacecraft were as follows: 1) to collect systems status data for onboard storage and display and for transmission to Earth; 2) to provide voice communications between the docked spacecraft, between the two separated vehicles and the extravehicular astronauts, and between the vehicles and the Earth; 3) to provide independent ground tracking capability of each spacecraft; 4) to provide transmission of television pictures; 5) to provide acceptance of real-time commands from Earth; and 6) to provide beacon transmissions for recovery location assistance.

The communications systems in the Mercury and Gemini spacecraft were conservatively designed and used conventional voice transceivers, telemetry transmitters, and C- and S-band pulse-type radar transponders to fulfill similar requirements.[6,29] These initial equipment configurations satisfactorily supported the Earth-orbital missions with only minor problems. It became obvious early in the Apollo design that previous approaches would not suffice for a lunar landing because of the great ranges involved. Instead, the Apollo design adopted the concept used by the Jet Propulsion Laboratory for lunar and planetary flights.[13] This concept involved putting all required spacecraft-to-Earth functions in a phase-coherent transmission system operating at S-band frequencies. By using subcarrier oscillators, the resultant system (unified S-band) could conveniently handle all required functions, including extremely accurate tracking by virtue of the coherency in the system. For voice communications between the separated spacecraft and the extravehicular crewmen, the more conventional VHF transceivers were used because of the short ranges and simple hardware involved. Late in the development program, it became necessary to provide backup command module tracking of the lunar module. Instead of another complex rendezvous radar, the existing VHF voice system was modified to permit reception and retransmission in the lunar module of a combination of digital tones. This capability in turn allowed accurate determination of relative range and range rate by command module VHF systems.

The instrumentation system was designed to provide onboard display of system data by using a variety of standard transducers. Data were sent to Earth either directly or from storage after a delay (e.g., after passage behind the moon). To satisfy storage requirements, a recorder with time coding and a 14-track capacity was developed which accepted both analog and digital data. Instrumentation support of all flights has been excellent.

For the first time, extravehicular communications equipment without a hardline transmission link was required in order to transmit limited systems and biomedical data, as well as to afford two-way voice communications. This design objective was most easily handled by installing a small VHF transceiver on the back package (Fig. 5) and providing a relay-to-Earth capability from either spacecraft. All communications systems have performed well, including during Apollo 11 and 12 lunar surface operations.

Two major problems were encountered in the development of the communications and instrumentation systems; the establishment of the final flight measurement lists and the ground development of the two high-gain antennas. The definition of the measurement lists required more time than had been anticipated because of systems complexity and the many organizational interfaces involved. The first major approach to the design of the high-gain antennas was a disk-rod array (candelabra), but this configuration was found to be extremely difficult to implement. Subsequently, the more conventional parabolic configuration was adopted for both vehicles; a single dish for the lunar module and a four-unit array for the service module (Fig. 2). Circuit margins in both systems have been equal to or better than preflight predictions, and the automatic acquisition and tracking feature of these antennas has been somewhat erratic but satisfactory.

Landing System

For Project Mercury, the altitudes and Mach numbers encountered in entry from orbit were much higher than for any previous manned vehicle, and a significant advance in parachute-landing-system technology was required. Applicable previous experience in parachute design was largely confined to the ejection seats and landing systems of high-performance aircraft, but this technology was of limited value in view of the severe design requirements involved. The drogue, which was deployed first in the landing sequence while the spacecraft was still in supersonic flight, was essentially a new development. Hundreds of aircraft drop tests were conducted on the ribbon-drogue configuration eventually adopted. The risers and suspension lines, as well as the canopy, required qualifications for the thermal and shock conditions involved at deployment.[69,70] One of the most difficult dynamic problems during landing-system design was the variety of drogue deployment attitudes and attendant structural loads. This variety of initial conditions became evident from an analysis that indicated marginal aerodynamic stability at the lower supersonic Mach numbers. Consequently, the drogue was designed for deployment at altitudes reaching 70,000 ft to augment natural vehicle stability.

The Mercury main parachute used a more conventional design, but weight growth of the spacecraft during design and development continually added new constraints and redesign problems in the landing system.[6] The main parachute, which required packaging in a confined area of the spacecraft cylindrical section demonstrated a large performance margin during initial design. This fact, combined with the inclusion of the landing-shock attenuation bag in the landing-system development program, made accommodation of spacecraft weight growth possible. All parachutes were initially reefed to limit the opening shock. One problem encountered in the design and operation of the reefing systems was the jagged reefing cutters which caused premature disreefing, but this problem was solved in an intensive flight qualification program. Although the landing-bag concept was sound, it was difficult to design a bag that satisfied the expected ranges of water and contingency land-landing conditions. The heat shield was released to deploy the bag, and the cable restraint system experienced breakage problems caused by incessant wave action after landing.[23]

Although the paraglider, a major landing-system advance, was planned for use in Gemini, early weight and development problems forced cancellation of this concept in favor of more conventional parachutes.[23,30] The crew ejection seats were originally adopted both to back up a paraglider failure and to provide launch abort capability. Once the parachutes were selected, the ejection seats provided a backup to a possible main parachute failure, making a reserve parachute unnecessary. One advancement evident in the Gemini landing system stemmed from the paraglider concept and involved a deployment scheme whereby the spacecraft touched down in an attitude slightly pitched up from horizontal, the normal flotation position, instead of vertically as in Mercury and Apollo.[23] This more favorable attitude eliminated the severe landing shock caused by the blunt heat shield. The Gemini main parachute, which had a fully inflated diameter of 84 ft, was probably near the practical sizing limit of an efficient single-parachute configuration.[29] For the Apollo command module, which weighs approximately 50% more than the Gemini spacecraft at landing, three main parachutes, each of which also measures 84 ft, were required. An original design constraint that the spacecraft land safely on any two main parachutes eliminated the need for a reserve parachute, as well as provided lower normal impact velocities. The Apollo landing system uses a three-stage reefing sequence to minimize opening-shock loads,[69,70] instead of the two-stage technique used in previous spacecraft. The main parachutes in all three spacecraft were developed from a ringsail concept, but small improvements were made in each design.

The attenuation of landing shock, particularly for a contingency land landing, was one of the more difficult design challenges in all spacecraft. The development problems encountered with the Mercury landing bag forced incorporation of a more integral attenuation system in Apollo.[71] Gemini had the advantage of landing at an attitude that permitted clean water entry to minimize loads, and crushable spacecraft structure was able to accommodate the remaining water impact loads. The ejection seats negated designing for the highly restrictive land-landing contingency. In the Apollo landing system, the basic structure is designed to attenuate most loads[72] upon land landing, and the couches are fitted with support struts that can stroke proportionally with very large impact loads. Since the command module has two stable flotation attitudes, a system for uprighting the spacecraft after it assumes the undesirable apex-down position was developed. On four of the seven manned flights, the command module has rotated to the apex-down attitude, and the uprighting system has successfully operated in each case In all three flight programs using production spacecraft, neither landing system has ever functioned unsatisfactorily.

Mechanical System Development

Mechanical system development in Mercury was always reasonably well understood, and most hardware of this type presented no serious problems during the flight program.[6,8] One of the most surprising anomalies occurring in Mercury mechanical systems was the failure of four of the five umbilicals, connected to the retropackage, to separate in the final flight.[6] A thorough investigation revealed that the pyrotechnic units normally used to separate attached umbilicals were test devices and did not contain the main powder charge. The separator which worked normally came from a different set of units, all of which contained the proper charge. Separation of only one umbilical was sufficient to effect retropackage jettison, which clearly demonstrates the need for both redundant mechanical functions and rigorous configuration and quality control.

In Gemini, the two most challenging mechanical-system developments were the docking devices and the dual hatches.[23] The docking system was developed and proved through exhaustive ground simulations[73] and presented no actual problems in the flight program. Interestingly, docking objectives on the Gemini IX mission were not accomplished because of a stuck cable in the deployment mechanism of the Agena nose fairing. The hatch doors caused concern with closing during the first few extravehicular operations, but slight modifications to the gear box and latches eliminated this difficulty.[23]

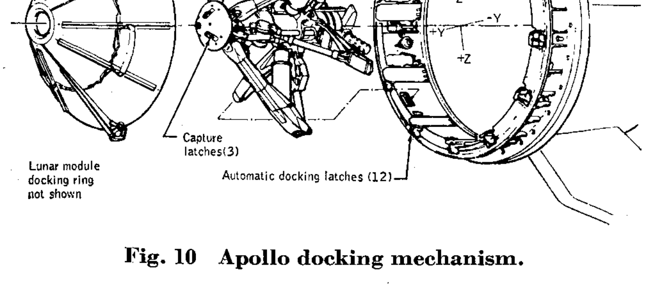

Fig. 10 Apollo docking mechanism.

The major new developments for Apollo mechanical systems were the docking devices in each vehicle and the lunar module landing gear. The docking system had the added requirement of permitting crew transfer between the two spacecraft, and the landing gear was a new development in the manned space program. Other mechanical systems, such as the hatches, the crew couches, and various deployable items, were more easily developed.

The original design of the command module side hatch consisted of two separate units (one each for the inner and outer structures) to minimize weight and heat-shield discontinuities. A subsequent requirement for rapid egress during prelaunch operations[74] resulted in a redesign to a single outward-opening hinged unit. This redesign was facilitated by a better definition of the heating environment from actual ablator performance. The Gemini spacecraft used rapid-operating hatch doors to accommodate crew ejection, and much of this technology was applied to Apollo hatch design. The pressure-seal cross section is indentical to the Gemini configuration, and the latches are similar but enlarged.

The lunar-orbit-rendezvous concept requires a docking subsystem (Fig. 10) to rigidly couple the command and lunar modules for performance of propulsion maneuvers by either vehicle. The docking interface must also permit crew transfer between the two spacecraft through a pressurized tunnel. Although the exact designs of the Apollo and Gemini docking mechanisms are quite different, similar techniques were used. Both systems use a capture-type latch to attenuate the contact energy[75] through hydraulic devices and to then retract the system to a rigid connection. The design of a satisfactory mechanical interface that could accommodate transfer of the crew in hard pressure suits was a difficult task. Extensive mockup evaluations in a water-immersion facility and in aircraft flow in parabolic arcs were conducted to simulate and evaluate zero-gravity effects. Manual functions were designed to require only one hand for operation in order to permit use of the other hand for stabilization. Apollo flight results have shown docking-system operation and hatch removal and reinstallation easier than expected.

The lunar module landing gear (Fig. 2) provides energy absorption and assures stability within the design envelope of lunar landing conditions,[76] based on a model of lunar surface conditions and control-system characteristics. A foldable landing gear was required to permit lunar module stowage within the spacecraft/launch-vehicle adapter. The contribution of previous space programs to landing gear development was almost nonexistent, especially since the program schedule required early establishment of a suitable lunar surface model before Lunar Orbiter and Surveyor results were known. The landing gear design uses crushable aluminum honeycomb in the primary struts for energy absorption. The development of struts that could provide energy absorption under close-tolerance loading conditions within structural limitations was a difficult problem. The honeycomb elements are currently precrushed to eliminate the initial peak load required. The landing gear was designed for landing stability within an envelope of expected variations in contact velocity and attitude.[77] The model lunar surface was assumed to have sliding coefficients of friction from 0.4 to 1.0, and the effective maximum slope at touchdown was assumed to be 12°. Interfering protuberances and depressions of up to 24 in. were also considered. Results of the two lunar landings to date indicate that the landing gear design has more than adequate performance margins.

Concluding Remarks

A brief summary of the design and development experiences during the 10-year span of U.S. manned space involvement has been given. In choosing to review each major subsystem, only the highlights could be discussed. The perspective of this overview, however, affords certain conclusions, many of which have been previously stated in this and other papers. Though the conservatism of certain design guidelines might be criticized by some, it was this kind of foresight, for example, that yielded three incredibly successful flight programs. The establishment of intricate management organizations, facilities, and procedures for accommodating increasingly complex spacecraft, designed and developed by thousands of people, has paid off. Constant attention to engineering detail in all aspects of design development has produced man's first venture to another celestial body. Only time and human application will prove the worth of this accomplishment.

The lunar landings on July 20 and November 19, 1969, are only the beginning. Additional lunar exploration flights will be made to surface areas of greater scientific interest, and use will be made of more sophisticated sensing equipment. The Skylab Program, the next major manned space effort, will make direct use of Apollo hardware to study the effects of long-duration crew exposure in Earth orbit, in addition to several interesting scientific objectives. The Apollo Program now includes provisions for extending the lunar exploration time and providing surface mobility devices to the crew. For these missions, an empty bay in the service module will be fitted with special equipment for scientific reconnaissance from lunar orbit. The eventual space-station and planetary exploration programs of tomorrow will make use of a proven Apollo technology, if not of the Apollo hardware. Each manned space effort, from the first Mercury suborbital flight to the lunar landings, has served as a reminder that our greatest limitation is imagination.

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