APOLLO DESIGN FEATURES

C. H. Feltz

Assistant Program Manager and Chief Engineer, Apollo
Space and Information Systems Division, North American Aviation, Inc.

Abstract

This paper presents some Apollo design features that were dictated by special problems associated with a manned lunar landing and return mission. Design features primarily attributed to booster limitations, crew safety, and natural mission requirements are discussed. Emphasis is placed on those features considered unique. Examples of specific topics considered are the general designs of the command module, heat shield, environmental control system, service module propulsion system, and Earth landing system.

Introduction

Although unmanned space probes have penetrated into deep space and, in particular, have transmitted information back to Earth regarding our neighboring planet Venus, man's personal venture into space has thus far been confined to Earth-orbital flights. The success of the Mercury program has been phenomenal. Project Gemini is an extension of the Mercury program with a greater number of Earth-orbits, two men in the capsule, and Earth-orbital-rendezvous missions. Projects Mercury and Gemini are logical steps in man's systematic attempts to conquer space, and as such, they are fundamental to future manned space flights extending beyond the gravitation of the Earth.

The next big step after Gemini in the United States manned space program is Project Apollo. Unlike Earth-orbital missions, the Apollo mission to land American astronauts on the Moon and return them to Earth necessitates escaping the Earth to reach the Moon and then escaping the Moon to return to Earth. This jump from manned Earth-orbital missions to manned lunar-landing missions demands propulsion capability far in excess of that ever before required. In addition, mission durations longer than ten days must be anticipated. During this time, the spacecraft and its crew must survive the environment of outer space. The Apollo mission thus imposes severe demands on booster capabilities and introduces many technological and environmental problems that are peculiar to a manned lunar-landing and Earth-return mission.

This paper presents some of the design features dictated by the special requirements of the Apollo mission. In particular, design features primarily attributed to booster limitations, crew safety, and natural mission environment are discussed. Emphasis is placed on those features that are considered unique.

Design Features Arising From Limitations in Available Boosters

The relative sizes of the various launch vehicles that are either in use or considered for use in the United States manned space program are shown in Figure 1. Of these vehicles, only Saturn V or NOVA has the performance capability to fulfill the Apollo objectives. For a direct lunar-landing mission, the NOVA vehicle would be the most desirable from the standpoint of performance, but because of the longer development time and higher cost of the NOVA, NASA selected Saturn V as the Apollo launch vehicle.

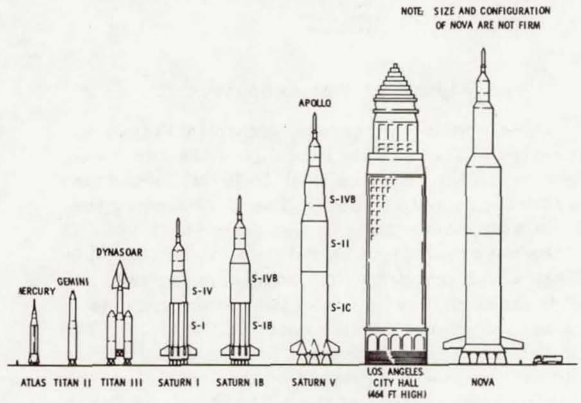


Figure 1. Launch Vehicles

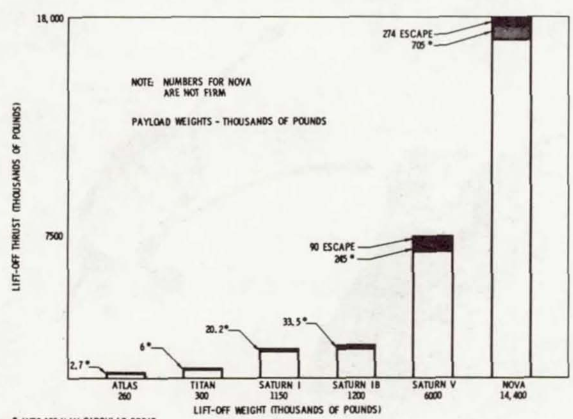


Figure 2. Launch Vehicles—Weight/ Payload/Thrust

The jump from Atlas/Titan to Saturn V is a big one. As shown in Figure 2, Saturn V has an Earth-orbital payload capability approximately 90 times that of the Atlas and 40 times that of the Titan. Although Saturn V is capable of injecting about 90,000 pounds to the Moon, mission requirements of this weight impose severe design restrictions, not only on the spacecraft and associated components, but also on the over-all configuration of the Apollo spacecraft. Because every extra pound that is landed on the Moon and subsequently returned to Earth increases the gross take-off weight by nearly 500 pounds, weight control is a very critical problem. Therefore, some of the design features of the Apollo spacecraft can be attributed primarily to limitations in the performance capabilities of the available boosters.

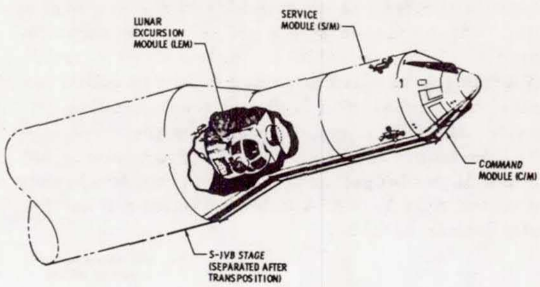


Figure 3. Apollo Spacecraft

The Apollo spacecraft, shown in Figure 3, consists of three basic modules—the command module (C/M), service module (S/M), and lunar excursion module (LEM). The C/M houses the three astronauts going to and from the Moon. It is the only module to be returned to Earth. The S/M, which provides the propulsion for the midcourse corrections and the return trip, is jettisoned prior to Earth entry of the C/M. The LEM houses two astronauts for the lunar-landing and return-to-orbit phases of the mission. The landing gear portion of the LEM is left on the lunar surface, and the remainder is left in lunar orbit after transfer of the astronauts back into the C/M.

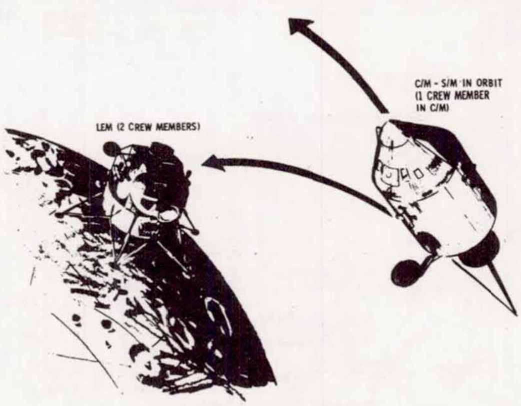


Figure 4. Apollo Approach

Two separate vehicles, each capable of sustaining human lives, are needed to accomplish the lunar-landing mission. These two vehicles are the C/M and the LEM, and their simultaneous existence reflects the decision of the United States to go to the Moon via the lunar-orbital-rendezvous mode. This method was chosen partly because of the limitations in booster capabilities. Figure 4 shows the Apollo approach employing this technique. The LEM is descending to land on the Moon, while the C/M and S/M remain in lunar orbit.

It is possible to eliminate the LEM by going to the Moon via the Earth-orbital-rendezvous (EOR) mode, but two Saturn V launch vehicles and a large spacecraft lunar-landing propulsion unit would be required. In addition, there is the operational problem of having to make two consecutive launches successfully within a specified period of time: one would place a tanker or a booster loaded with cryogenic fuel into an Earth orbit, and the other would place the spacecraft into the proper position for rendezvous with the vehicle in orbit.

From the decision to use the lunar-orbital-rendezvous method, the following design feature was established: the LEM is to be initially transported behind the S/M, and then shortly after translunar injection the LEM is to be transposed and mated with the C/M. This transposition phase of the flight is necessary in order to expose the S/M engine for use in midcourse guidance corrections. Abort requirements make it impracticable to launch with the C/M and LEM mated. A promising scheme for making the transposition and docking is illustrated in Figure 5. The action begins by igniting the four S/M reaction-control-system engines and then blowing off the adapter. Separated from the launch vehicle, the C/M-S/M unit free-flies around to mate with the LEM, which is stabilized by the empty S-IVB stage and its stabilization system. After the mating of the C/M-S/M unit with the LEM, the S-IVB stage is jettisoned, and the Apollo spacecraft proceeds to coast toward the Moon.

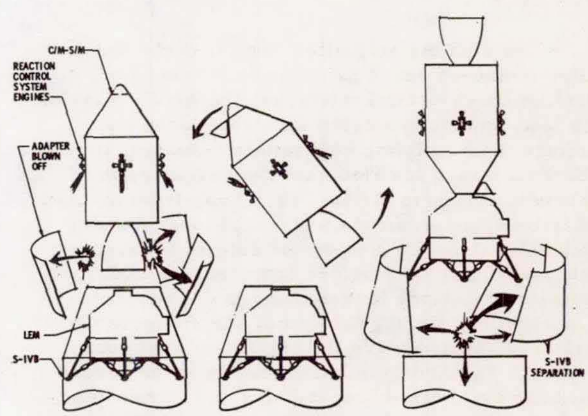


Figure 5. Free Fly-Around Transposition and Docking

It has been stated that the S/M is jettisoned prior to Earth entry of the C/M. Unlike the Mercury and the Gemini vehicles, which require retrothrusting to deorbit for the Earth entry, the C/M, moving with an inertial velocity of approximately 36,000 ft/sec, enters the Earth's atmosphere directly. Partly because of weight limitations, a retropackage is not used to reduce this high velocity. The result is that the C/M must be capable of dissipating the energy (virtually all kinetic) associated with Earth entry in such a manner that the integrity of the spacecraft remains intact and its human occupants unharmed. In addition, the C/M must be capable of correcting guidance errors in order to reach a given landing site. The present C/M, in fulfilling these requirements, presents the following design features.

The C/M is essentially a body of revolution and, with the center of gravity (c.g.) along its longitudinal axis, will develop no aerodynamic lift (Figure 6). By offsetting the c.g., however, the C/M trims at an angle of attack approximating -33 degrees. In this trimmed attitude, the axial force is resolved to yield a lift-to-drag ratio of 1:2. It should be noticed that on this vehicle positive lift is generated at negative angle of attack. The actual c.g. offset is achieved by locating the heavy equipment on one side of the longitudinal axis. This requirement critically restricts the space available for the installation of various components.

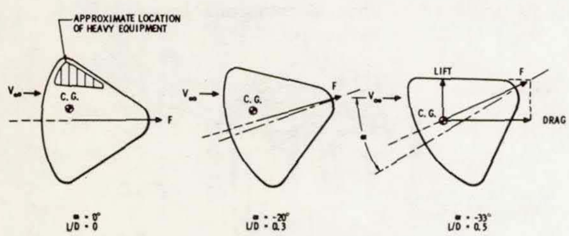


Figure 6. Command Module Aerodynamics

The C/M can be flown by rotating the vehicle about the instantaneous velocity vector. This maneuver, however, forces the lift vector out of a given plane of action so that any effort to maneuver in the vertical plane automatically produces horizontal displacements. Figure 7 shows the C/M with its lift vector fully up, partly tilted to the right (with resulting vertical and horizontal components), and fully down. The four roll reaction-control engines shown in Figure 8 are used to rotate the C/M about the stability axis. Each reaction jet can deliver 100 pounds of thrust. Note that there are 12 reaction-control engines on the C/M. Since only six engines are needed to control roll, pitch, or yaw, the 12 engines represent a completely redundant reaction control system. With a lift-drag ratio of 1:2, the C/M can enter the Earth's atmosphere and maneuver to the landing site from as far out as 5000 nautical miles or as close in as 1400 nautical miles. Figure 9 illustrates the Earth entry range limits.

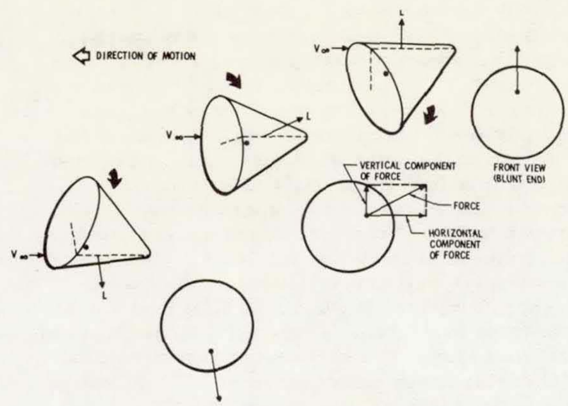


Figure 7. Lift Vector Control

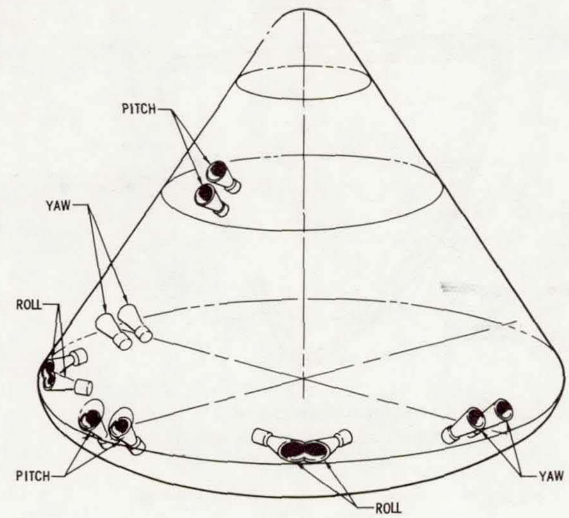


Figure 8. Command Module Reaction-Control-System Engines

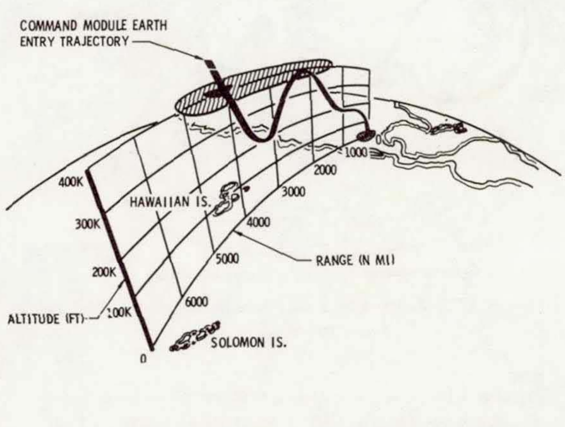


Figure 9. Entry Range Limits

During Earth entry, depending upon the particular trajectory flown—e.g., high deceleration with short flight time or low deceleration with long flight time—the total heat load on the

C/M can vary between approximately 5 to 8 million Btu's. These heat loads are many times larger than those occurring during entry from an Earth orbit. The heat shield being developed for use in the C/M to dissipate the entry heat loads incorporates a fiberglass honeycomb matrix that is bonded to the outer body substructure and then filled with ablative material. This type of construction yields a well-integrated heat shield that can withstand thermal stresses associated with temperatures as low as -260 F. Because of the stringent weight restrictions in the Apollo spacecraft injected payload, the C/M heat shield is tailored in thickness (Figure 10) to the imposed local heat load. The surface temperature of the C/M during Earth entry can reach 5000 F, but the ablator bond line will not exceed 600 F.

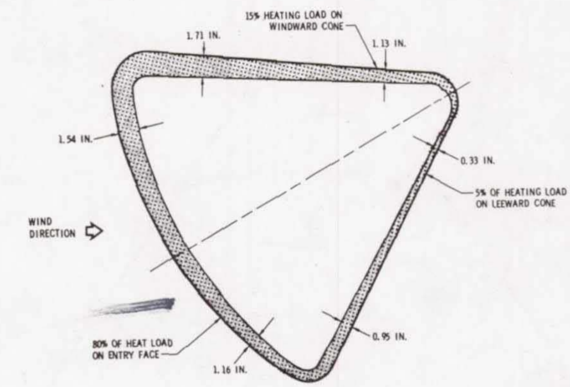


Figure 10. Apollo Command Module Local Heating Load and Heat Shield Thickness

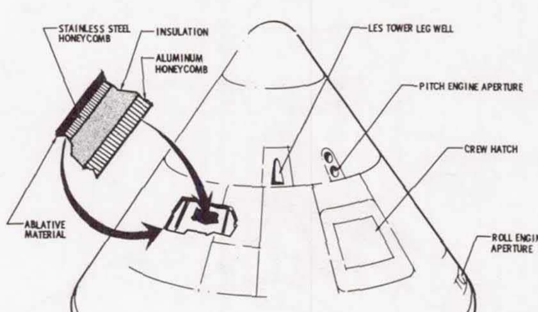


Figure 11. Command Module Exterior Structure

Figure 11 shows a cutaway view of the complete C/M, exposing a cross sectional view of the heat shield and the basic C/M structure. A design feature of this structure is its light-weight, double-shell construction. The outer shell is made of brazed stainless steel honeycomb, and the inner shell (Figure 12) is made of bonded aluminum honeycomb. This inner substructure constitutes the pressure vessel and is maintained at a pressure of 5 psi in a 100-percent oxygen environment for altitudes above 20,000 feet. The two shells are separated by floating fiberglass stringers, and the space between is filled with Q-felt insulation material. Although this type of construction is partially influenced by weight limitations, it is primarily developed from heat transfer considerations. This construction also serves as an effective barrier for meteoroids, trapping any meteoroid that might penetrate the outer layer of the honeycomb structure.

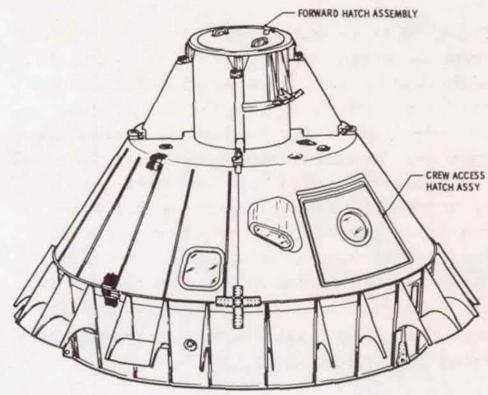


Figure 12. Command Module Inner Structure

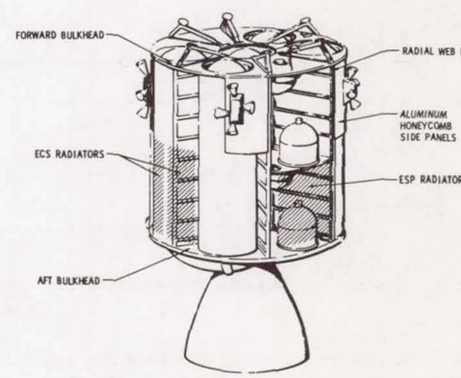


Figure 13. Service Module Structure

The S/M structure (Figure 13) also reflects a light-weight, simple type of construction. The basic structure consists of six equally spaced radial beams that divide the cylindrical S/M into six bays. These bays are used to house various items, such as the fuel and oxidizers for the S/M engine and the fuel cells. Aluminum honeycomb side panels and aft and forward bulkheads are bolted onto the solid aluminum beams to form the outer shell of the S/M. Four radiators, bonded directly to the side panels, are integral parts of the S/M outer structure. Two of these radiators are for dissipating heat from the environmental control system (ECS), and two are for dissipating heat from the electrical power system (EPS). The fuel cells, S/M engine, ECS, and EPS are discussed in the following sections.

Design Features Arising From Requirements for Safety of Astronauts

The United States' philosophy of maximum concern for the safety of the astronauts dictates some operational design features that may or may not be manifested in specific pieces of hardware. Particular examples of nonhardware safety considerations are the circumlunar "free" return trajectory, LEM and C/M-S/M equal-period orbits, and over-all mission abort flexibility.

The circumlunar free return trajectory permits a return to Earth with a minimum change in the velocity vector if an abort is necessary after translunar injection. This means that in the event of a failure of the service propulsion engine, the reaction-control-system engines can be used to correct guidance errors to place the spacecraft into the proper circumlunar trajectory for the free return to Earth. The use of this type of trajectory, together with an Earth-to-Moon transit time of about 70 hours, makes it necessary to land on the Moon in retrograde motion with respect to the natural rotation of the Moon about its axis. Inasmuch as a point on the surface at the equator is moving with a tangential velocity of about 15 ft/sec, the LEM would have to land and take off against this velocity. This is a loss in velocity-change capability of 30 ft/sec. It is a direct consequence of flying such a circumlunar free return trajectory. The free return feature, however, is desirable from a crew safety and morale point of view.

The LEM and C/M-S/M equal-period orbit is a part of the over-all abort flexibility. Its use provides for a possible pickup of an inactive LEM by the C/M-S/M. For example, assume that the spacecraft is orbiting the Moon at 80 nautical miles altitude and that the LEM is ready to deorbit for the lunar landing. (See Figure 14.) A velocity increment of approximately 460 ft/sec toward the center of the Moon is imparted to the LEM. This action injects the LEM into a transfer ellipse that takes it to an altitude of 50,000 feet at perilune, with an orbital period equal to the circular orbital period of the C/M-S/M in its parking orbit. This equal-period orbit provides the LEM with an automatic (without propulsion) rendezvous point with the C/M-S/M in the event of an abort, as well as permits the C/M-S/M to follow the LEM optically down to perilune in a normal mission. For an abort situation, about two hours after the LEM deorbit maneuver, the two vehicles will meet again. The C/M-S/M has chase capability, and if at this time, a 460 ft/sec velocity increment toward the center of the Moon is imparted to the C/M-S/M, it will be placed in the same orbit with the LEM. The C/M-S/M can now actively rendezvous with a disabled LEM.

The over-all mission abort flexibility feature permits the astronauts to abort anytime up to the actual lunar landing. Figure 15 indicates points along the Apollo Earth-to-Moon trajectory where it is possible to abort the mission.

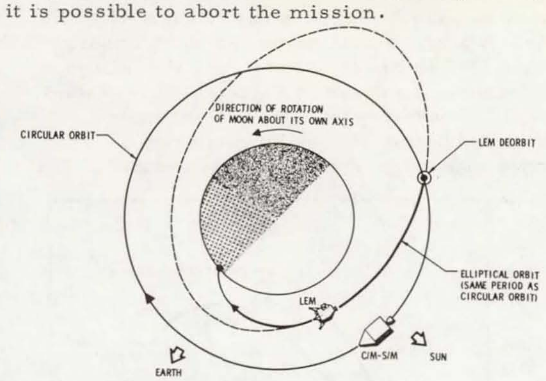


Figure 14. Equal Period Orbits

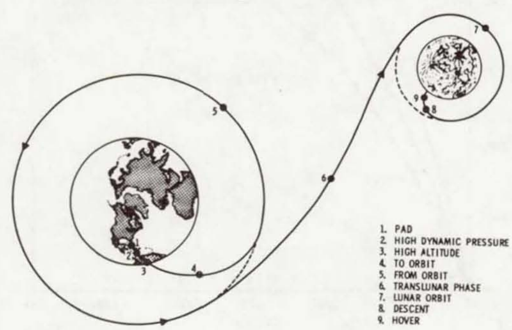


Figure 15. Abort Opportunities

Of the crew safety design features that do manifest themselves in particular pieces of hardware, the most obvious one is the launch escape system (LES). Although the Mercury also utilizes a launch escape rocket, the larger size and more stringent abort requirements for the Apollo make this LES unique. Paraglider and ejection seats are used in the Gemini, but they are considered too heavy for incorporation into the Apollo program. The Apollo LES is designed for abort on the launch pad, during high dynamic pressure, or at high altitude.

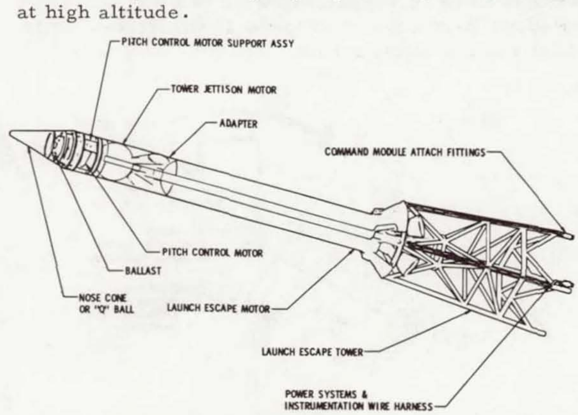


Figure 16. Launch Escape System

Figure 16 shows the basic construction of the LES. Titanium is used for the tower because of its light weight and high structural strength. The thrust of the launch escape motor is about 150,000 pounds. A pitch control motor having an impulse of nearly 1700 lb-sec is used to pitch the LES over for pad abort. As shown in Figure 17, the system is capable of carrying the C/M to a minimum altitude of 4000 feet at 3000 feet downrange. The minimum safe range at touchdown is about 2000 feet.

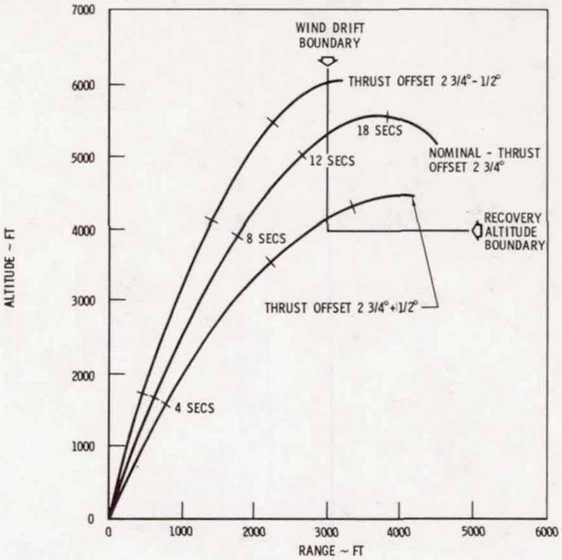


Figure 17. Pad Abort Trajectories

In a normal launch, the LES is jettisoned shortly after ignition of the second stage. Unlike the Mercury, which uses a Marman band for the launch tower separation, the Apollo uses explosive bolts. (These bolts are unique in that there are provisions for loading or unloading the explosive charges.) During LES jettison, there is a possibility that the jet plume might damage the windows of the C/M. Partly because of this reason, but mainly because of the adverse effects from aerodynamic heating during atmospheric exit and entry, the windows have covers. Figure 18 illustrates the C/M window configuration.

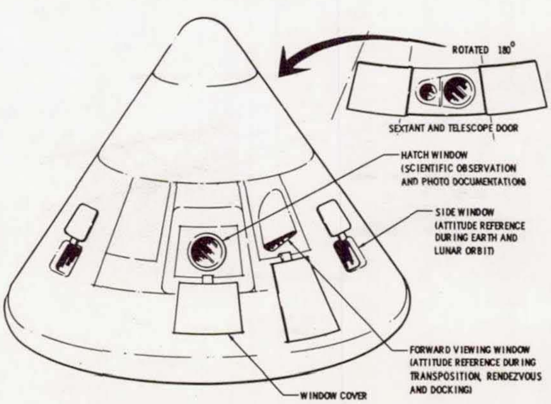


Figure 18. Command Module Window Configuration

For a launch from Cape Canaveral, a high-altitude abort (about 180,000 feet) would force the C/M to land in the ocean. Although San Antonio, Texas, and Woomera, Australia, are being considered for the primary landing sites, the possibility of a water landing requires that the C/M be designed for landing on either land or water. By way of comparison, the Gemini also has a dual landing capability. The Mercury, however, has a water landing capability only. Because of the offset c.g., the C/M has two stable orientations in water. These orientations are shown in Figure 19. As designed, position 1 is the more stable of the two because of the geometry of the C/M and the c.g. location with respect to the water. If the c.g. were low enough or sufficiently offset, the C/M would float in only one orientation.

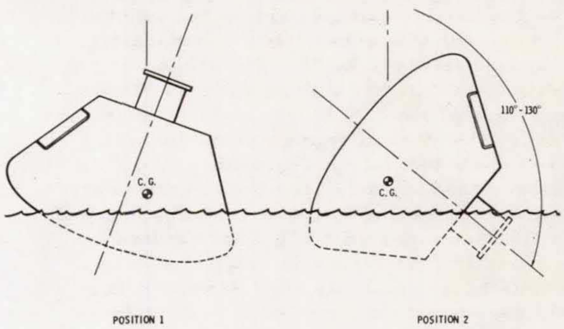


Figure 19. Command Module Flotation Positions

During a high-altitude abort, tumbling may cause the C/M to come in apex forward. In order to eliminate this apex-forward trim point, which is not acceptable from a crew safety point of view, two strakes are installed on the C/M. Although the final size and shape of the strakes are not firm, their approximate geometry and location are shown in Figure 20.

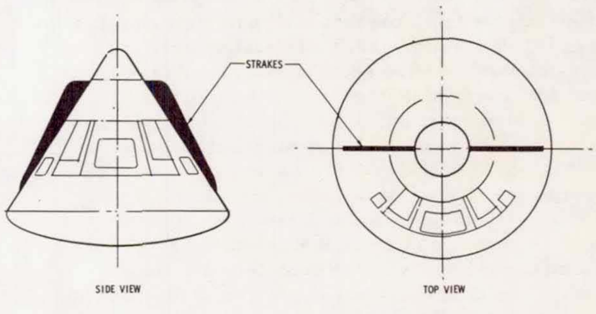


Figure 20. Command Module Strakes

A critical phase of the Apollo mission is the Earth landing of the C/M, whether the landing is being made in connection with an abort or a return from a lunar mission. Whatever the case may be, the Earth landing system must reduce the landing speed of the C/M to assure the safety of the astronauts. Unlike the Mercury, which uses a single main parachute, or the Gemini, which uses a paraglider for the Earth landing, the C/M deploys three main parachutes, any two of which will land the C/M without exceeding emergency limits. The three-chute system was chosen because of its light weight and high reliability.

Figure 21 illustrates the operational sequence of chute deployment. The normal rate of descent of the C/M with all three parachutes deployed will be approximately 24 ft/sec; the emergency descent rate with two parachutes opened will be nearly 30 ft/sec. A couch impact attenuation system is used to reduce the landing impact. As illustrated in Figure 22, the system consists of hollow struts filled with crushable honeycomb that is arranged to fold like a telescope upon landing.

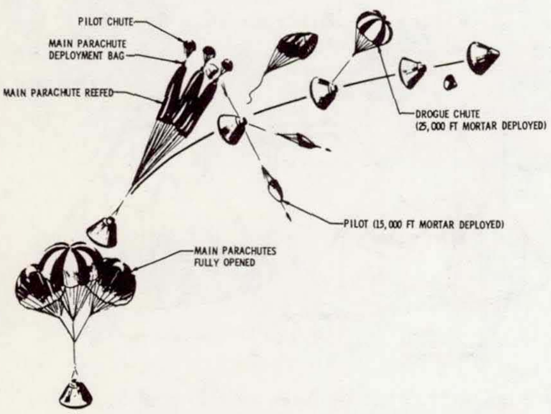


Figure 21. Earth Landing System

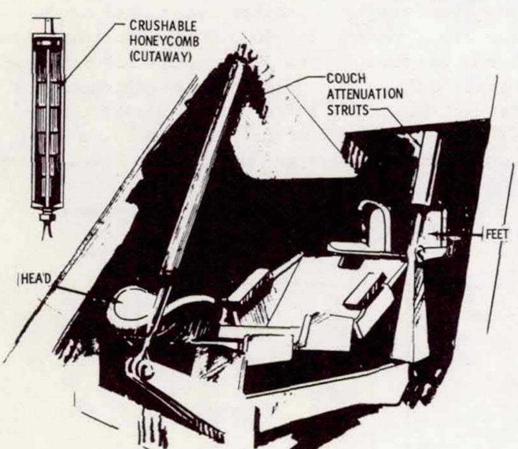
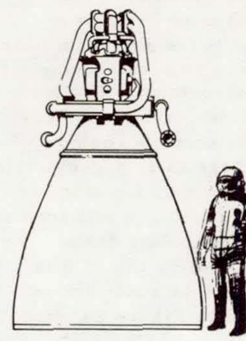


Figure 22. Couch Impact Attenuation System

From the standpoint of mission success (probability of success = 0.900) as well as crew safety (probability of safety = 0.999), a high over-all system reliability is mandatory. One way of assuring high reliability is to incorporate component or system redundancies where practicable. An example is the S/M propulsion engine shown in Figure 23. This is a single swiveled-nozzle engine that must be operable at any time throughout the entire flight. Multi-engine configurations were considered for the S/M, but, based on factors of weight and reliability, it was decided to use a single engine. As shown in Figure 24, the service propulsion propellant system uses a series feed. In order to achieve a high engine reliability, double series and parallel regulator and check valve systems are deployed in the fuel system. This redundancy technique safeguards against possible fail open or fail close situations.



THRUST, NOMINAL (LB)	21,900
I_{SP}, MINIMUM (SEC) (-3σ at 750 SEC)	318.7
AREA RATIO	60:1
MIXTURE RATIO, NOMINAL	2.0
CHAMBER PRESSURE (PSIA)	100
PROPELLANTS N_2O_4/AEROZINE 50	

Figure 23. Service Propulsion Engine Configuration

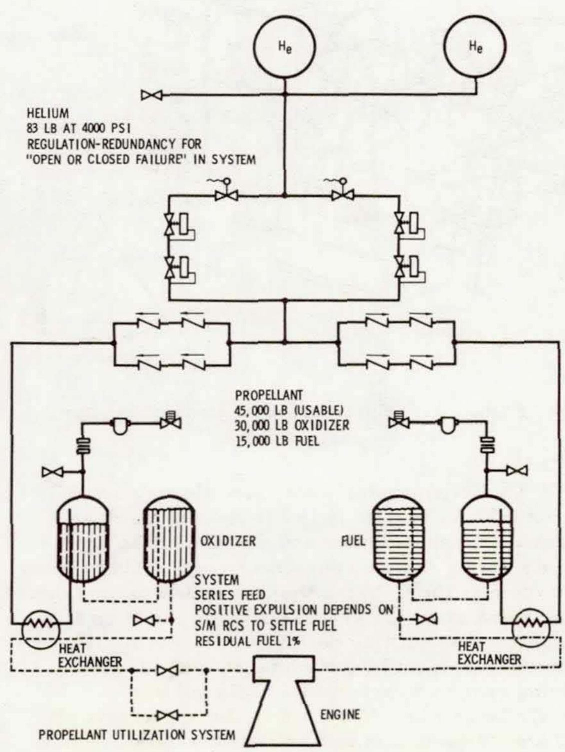


Figure 24. Service Propulsion System Schematic

Design Features Arising From Encounter With Natural Mission Environment

This section covers design features that stem from the important problems of how to sustain life during a space mission and how to survive the natural mission environment. These features are discussed here because they arise from basic needs rather than from considerations which cope with special emergency measures as discussed earlier.

One of the foremost human needs on a lunar mission is the maintenance of life with reasonable comfort. Because of the long duration of the voyage, the Apollo spacecraft must provide a habitable environment for the three astronauts for at least ten consecutive days. This requirement is satisfied by the use of an environmental control system (ECS) of a sophisticated, multifunctional design. Figure 25 illustrates some of the components of the ECS and indicates their approximate location in the C/M. The two major functions of the ECS are the control of temperature and atmosphere in the C/M cabin and the cooling of the electronic equipment. Specifically, the ECS is required to maintain a shirt sleeve environment inside the C/M. As indicated in Figure 26, five major loops make up the ECS; i.e., the suit atmospheric control, the cabin temperature control, the oxygen supply, the water management, and the coolant transportation loop.

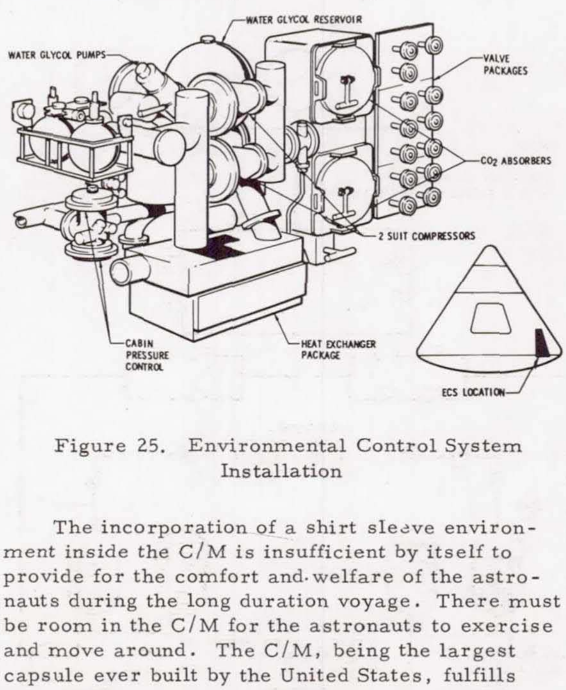


Figure 25. Environmental Control System Installation

The incorporation of a shirt sleeve environment inside the C/M is insufficient by itself to provide for the comfort and welfare of the astronauts during the long duration voyage. There must be room in the C/M for the astronauts to exercise and move around. The C/M, being the largest capsule ever built by the United States, fulfills this requirement by providing 80 cubic feet of living space per astronaut. This volume is relatively large when compared to the approximately 60 and 40 cubic feet per astronaut available in the Mercury and Gemini capsules, respectively. Figure 27, a cross sectional view of the C/M, illustrates the living area.

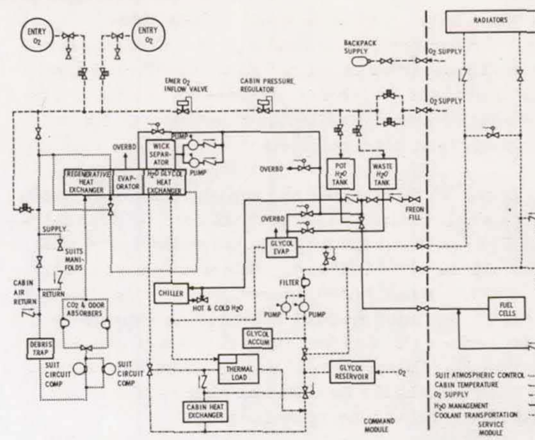


Figure 26. Environmental Systems

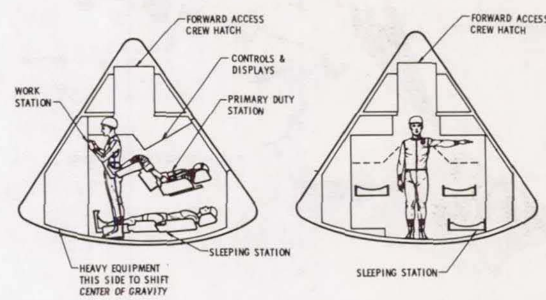


Figure 27. Living Area—Command Module

With the three astronauts aboard, the need for an adequate supply of potable water is obvious. Unlike the Mercury, in which a specific amount of water is carried aboard the capsules for drinking purposes only, a major portion of the drinking water for the Apollo astronauts is derived from the fuel cells located in the S/M. The fuel cells produce potable water as they generate electricity.

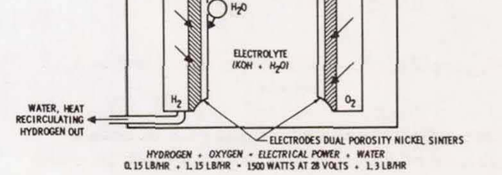


Figure 28. Fuel Cell

Figure 28 illustrates the basic principles of the fuel cells. There are three fuel cells and three batteries. These units constitute the electrical power sources (Figure 29). While it is clearly desirable to have all three fuel cells operating, any two of these cells will satisfy the mission requirements. The three batteries located in the C/M are for use during Earth entry, but they can be used at anytime in the event of an emergency.

SUBSYSTEM	STATE, OPERATIONS
FUEL CELLS	ALL 3 MODULES OPERATING
INVERTERS	1 OF 3 OPERATING
BATTERIES	ALL 3 OFF LINE

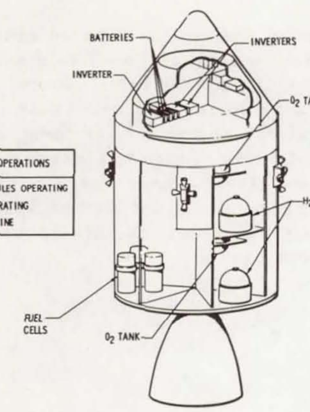


Figure 29. Electrical Power System

The Apollo spacecraft, traveling to and from the Moon, is placed in a radiation environment that can produce surface temperature variations from 250 to -290 F, depending upon the orientation of the spacecraft to the sun. Lengthy exposure to these temperatures can be avoided by properly controlling the orientation of the vehicle. This method of solution, however, is not desirable, and the spacecraft is consequently being designed to withstand temperature extremes for various orientations of the vehicle with respect to the sun.

In addition to the requirements for a habitable spacecraft, there also exists a requirement for suitable communication with the Earth, which is essential to the well-being of the astronauts as well as to mission success. The various antenna equipment located in the C/M and S/M are illustrated in Figure 30. For distances greater than 40,000 miles from the Earth, the 2-kmc high-gain antenna is used in transmitting signals to the Deep Space Instrumentation Facilities (DSIF) located at Goldstone, California; Woomera, Australia; and Johannesburg, Africa. The vhf omniantenna is used with the Ground Operational Support System (GOSS) for near-Earth communication. The frequencies will be the same as those now used on the present GOSS complex for Mercury. A design feature of the communication system is that voice communication between the spacecraft and the Earth is available almost continuously. Blind spots will occur during certain phases of Earth operations and when the spacecraft is traversing the back side of the Moon.

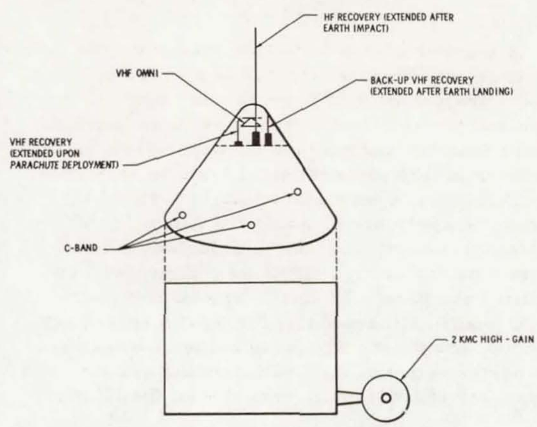


Figure 30. Antenna Equipment

A final design feature to be presented in this paper is the personal communication assembly (Figure 31). The assembly consists of a bump hat, a microphone with amplifier, and earphones. It is worn by the astronauts when they are not in their spacesuits. Identical microphones and earphones are incorporated in the helmet of the spacesuit. These components are compatible with hardwire or wireless communication equipment. Communication within the cabin is achieved through the intercommunication system, using a hardwire plug-in. Radio frequencies are used for voice communication exterior to the spacecraft. This personal communication system is especially required during the actual exploration of the Moon. It is mandatory that the astronauts, one of whom will be walking on the lunar surface, be in voice contact with one another.

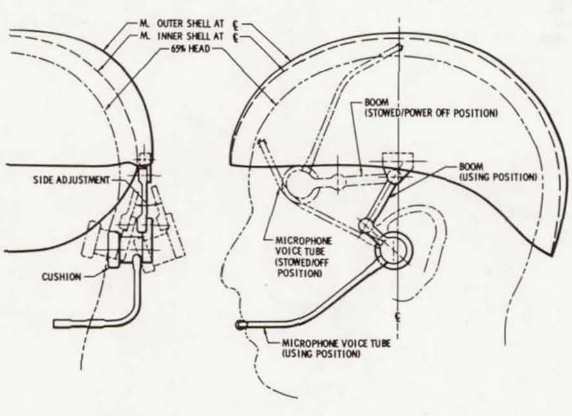


Figure 31. Communication Assembly—Personal

Concluding Remarks

A number of Apollo design features have been discussed to illustrate the broad spectrum of the Apollo spacecraft design problems. Not all the technical problems have been covered. Each design feature, before final incorporation, must endure stringent experimental tests to verify its acceptability. There will be flight tests of the launch escape system, using the Little Joe II booster to investigate aborts at high dynamic pressures and at high altitudes. There will be Saturn I and Saturn IB Earth-orbital missions for flight qualification tests of the Apollo spacecrafts. Aircraft drop tests are being made to investigate the performance of the Earth landing system, and drop tests of boilerplate versions of the C/M are being made to assess landing impact loads. The flotation and stability of the C/M have been explored by dropping and towing boilerplate versions of the C/M in water.

Some of the design features presented undoubtedly will be modified as a result of new experimental data and information. In addition, some new problems will arise that will dictate other design features. Therefore, the design must be flexible enough to incorporate changes as needed. At this time, there is no known technical reason why the United States cannot successfully complete the Apollo mission within the present decade.