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31 OCTOBER 1962

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DIRECT FLIGHT APOLLO STUDY (U)

Volume I:

Two Man Apollo Spacecraft

CONTROL NO. C-72078

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DIRECT FLIGHT APOLLO STUDY

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INTRODUCTION

This report presents the results of a study conducted by McDonnell Aircraft Corporation of the feasibility of performing the manned lunar landing mission by the direct flight mode employing two-man spacecraft. The study was conducted under NASA Contract NASw-522. Corresponding to the three parts of the statement of work, Reference (1-1), the report is divided into three volumes.

Volume I - TWO-MAN APOLLO SPACECRAFT - Presents the results of a design study of a lunar landing spacecraft incorporating a two-man command module similar in external shape to the present Three-Man Apollo command module.

Volume II - GEMINI SPACECRAFT APPLICATION - Presents the results of a study of the modifications required to the Gemini spacecraft to provide it with the capability of performing the lunar landing mission.

Volume III - RESCUE VERSIONS - Presents the results of a study of the modifications required to provide the spacecraft of Volumes I and II with the capability for use in rescue operations.

1. SUMMARY

The results of the studies performed in compliance with Reference (1-1) show that two-man spacecraft can be designed to accomplish the direct flight lunar landing mission well within the Saturn C-5 launch vehicle capability. Either a new two-man design, similar in external shape and in capability to the present Three-Man Apollo, or a Gemini version employing minor modifications is suitable. Gemini versions with more extensive modifications lead to improved performance margins. Advances in the state-of-the-art or extensive engine development programs are not required for any of the spacecraft designs presented. It is also shown that the two-man designs are readily adaptable for unmanned rescue missions.

A useful by-product of the direct flight rescue designs presented is that the propulsion modules which carry the spacecraft to lunar landing are usable, without change, as a lunar cargo carrier, or logistics vehicle, with a useful load capability of about 29,000 pounds.

1.1 Study Ground Rules - The basic ground rules for the study reported herein were established by Reference (1-1). Detailed definition of some of these ground rules and additional ground rules were supplied in References (1-3), (1-4) and (1-5) and verbally by the Office of Systems, Office of Manned Space Flight. The detail criteria established by the contractor for use in the design of the spacecraft are given in Section 2.7. Major ground rules for the part of the study concerned with the use of a Two-Man Apollo type command module are listed below.

A. Launch vehicle is the Saturn C-5, injecting 90,000 pounds to a lunar transfer orbit.

B. Design study to be based as closely as possible on Apollo Mission Guidelines, Reference (1-2).

C. Crew to be provided with stand-up capability.

1.1 (Continued)

D. Flight profile and velocity increments are as given in Reference (1-3).

E. Mission duration is eight (8) days (two and one-half days flight time to moon, one day on lunar surface plus one day contingency, two and one-half days return time plus one day contingency) plus seven days post landing (one day habitable environment plus six days survivable environment).

F. Both cryogenic and noncryogenic propulsion systems to be considered.

G. No weight to be added for radiation or micrometeoroid protection but attention to be given to design details to provide maximum inherent protection.

H. Lunar surface characteristics to be used are as given in Reference (1-6).

I. Atmosphere is 5 psi normal, 3.5 psi emergency, 100% oxygen.

J. First operational flight during the first half of calendar year 1967 shall be assumed.

1.2 Configuration Selection - Studies of spacecraft staging arrangements and propellant combinations, conducted on a parametric basis, indicate the lunar landing mission can be performed under the specified ground rules with a variety of spacecraft configurations. These studies are reported in detail in Section 4.2. In order to further substantiate this conclusion, a single spacecraft arrangement was selected to design in sufficient detail to permit an accurate weight determination. The resulting spacecraft configuration is shown on Figure 1-1.

The major features of the command module are summarized on Figure 1-2. The basic objectives in the design of the command module were to maintain the current Three-Man Apollo external shape (33-degree afterbody half angle and heat shield radius 1.2 times heat shield diameter), provide crew stand-up capability, provide

TWO MAN APOLLO SPACECRAFT

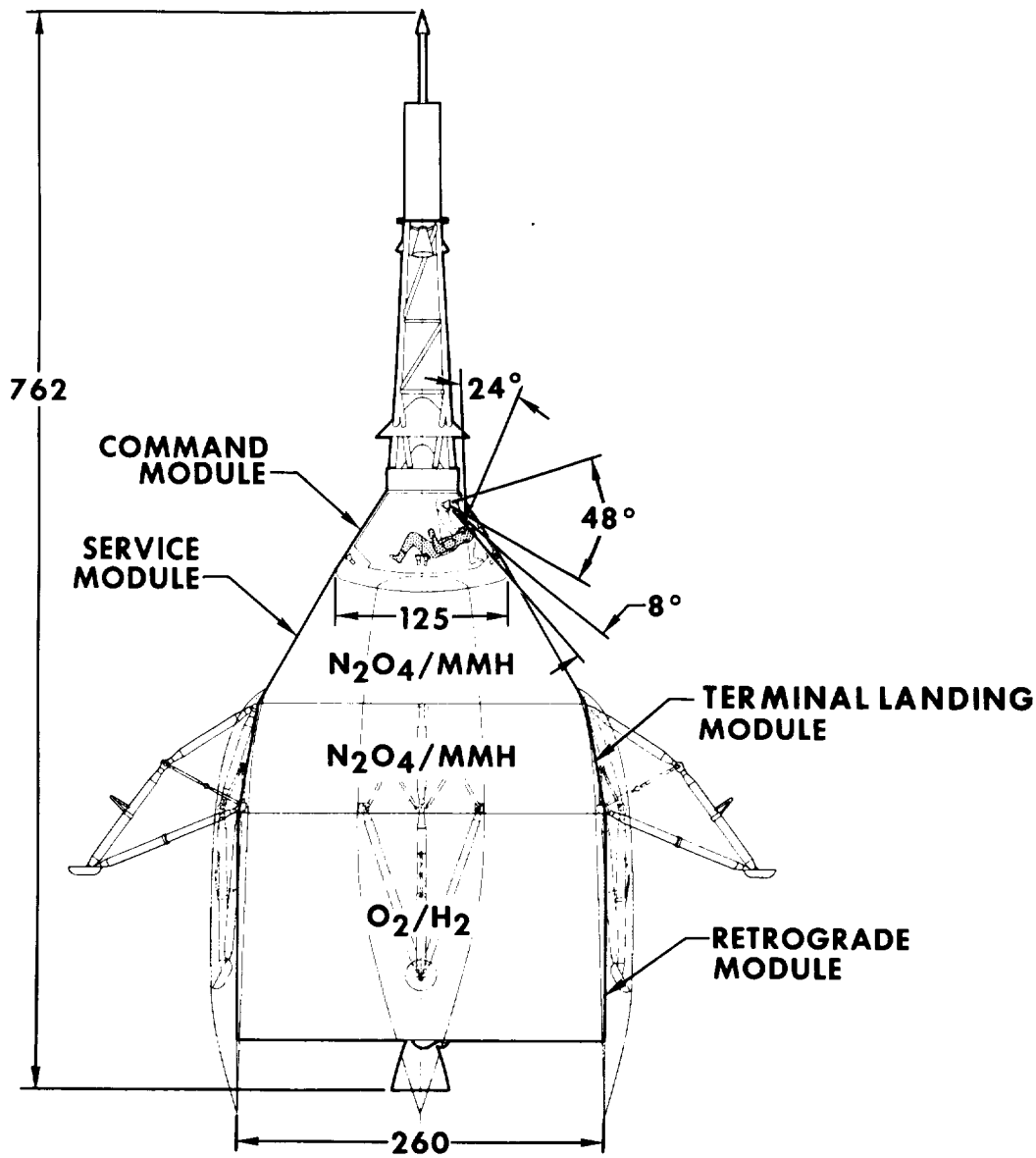
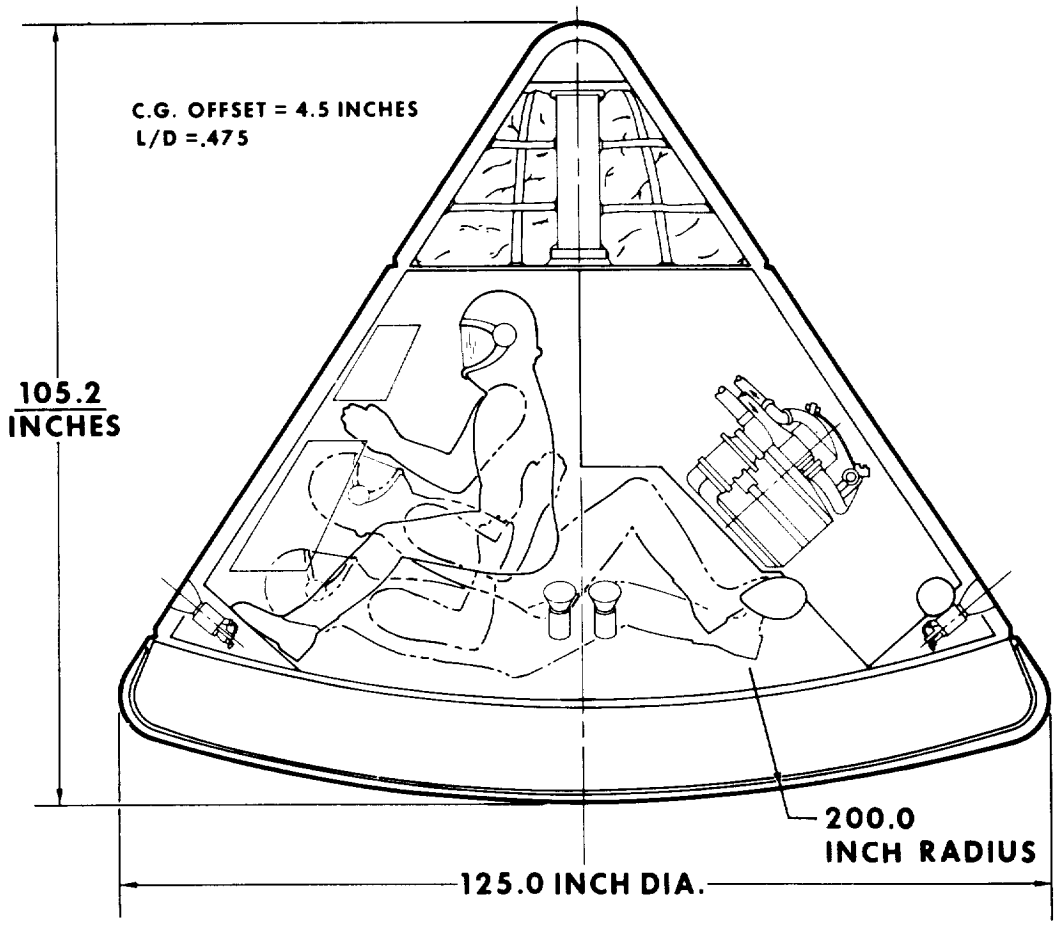


FIGURE 1-1

COMMAND MODULE



VOLUME (185 FT.^{3})

CREW	73 FT.^{3}
EQUIPMENT	100 FT.^{3}
RECOVERY SYSTEM	12 FT.^{3}

HEAT PROTECTION

MAC THERMORAD S-3 MATERIAL
HEATSHIELD - 0.72 IN TO 1.38 IN.
ON 0.010 TITANIUM BACK-UP
AFTERBODY-0.20 TO 0.30 IN.
.016 TITANIUM BACK-UP

INSULATION

FORWARD FACE-.50 IN. MIN-K-1301
AFTERBODY -1.50 IN. RF700

STRUCTURE

TITANIUM SKIN-STRINGER
SINGLE WALL PRESSURE VESSEL

COMMAND MODULE IMPACT ATTENUATION (40 G's)

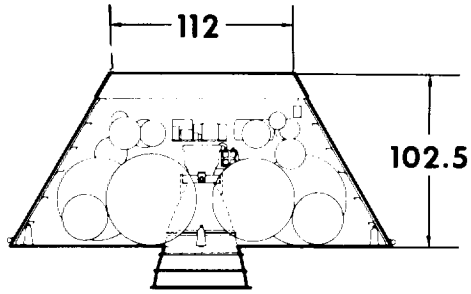
CRUSHABLE TRUSSGRID
9.0 INCH STROKE

SEAT ATTENUATION (20 G's)

6 INCH STROKE

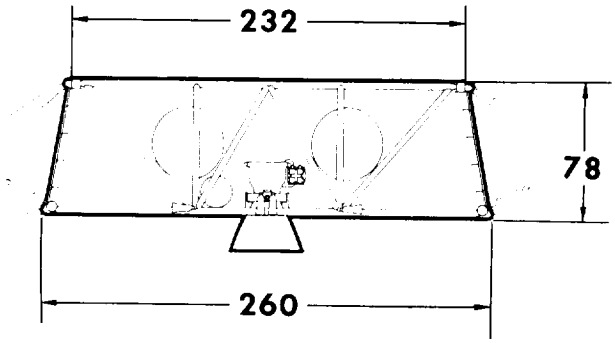
FIGURE 1-2

PROPULSION MODULES



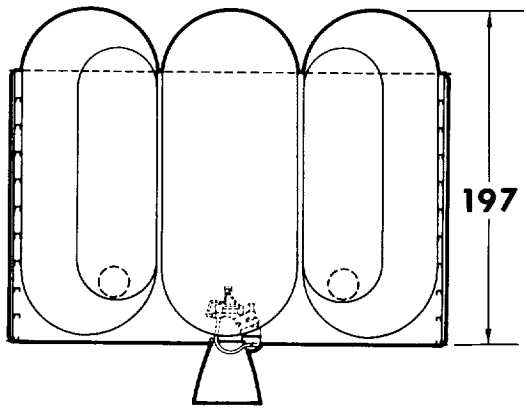
SERVICE

THRUST	11,700 LB.
PROPELLANT	N_2O_4/MMH
LOADED GROSS WT.	23,428 LB.
USEABLE PROPELLANT	19,491 LB.
SERVICE EQUIPMENT	1148 LB.



TERMINAL LANDING

THRUST	11,270 TO 3410 LB.
PROPELLANT	N_2O_4/MMH
LOADED GROSS WT.	6115 LB.
USEABLE PROPELLANT	3380 LB.
LANDING GEAR PLUS CARRY THROUGH	740 LB.



RETROGRADE

THRUST	- 26,500 TO 2,650 LB.
PROPELLANT	O_2/H_2
LOADED GROSS WT.	53,787 LB.
USEABLE PROPELLANT	47,540 LB.

FIGURE 1-3

an L/D of 0.5 for re-entry, and hold the weight to a minimum. Several command module sizes and arrangements were investigated in arriving at the configuration selected. This investigation is discussed in detail in Section 4.1. The study showed that the design objectives of maintaining the Apollo external shape, providing space for one crewman to stand up, while attempting to minimize size (and thus weight), necessitated the use of ballast to obtain a sufficient center of gravity offset for achieving an L/D of 0.5. The configuration selected incorporates what are considered acceptable compromises in the design objectives. "Sit-up" and "stretch-out" capability are provided for both crewmen instead of stand-up capability and the heat shield radius is changed to 1.6 times the heat shield diameter in order to approach the desired L/D with the center of gravity offset attainable without ballast.

External dimensions and major characteristics of the propulsion modules are given on Figure 1-3. The basis for the selection of this staging arrangement in preference to other possible configurations is as follows.

A. Reliability and complexity considerations suggest the use of as fe stages as possible. However, in order to minimize lunar landing gear weight and stability problems at touchdown, it is necessary to stage prior to lunar landing. Also, in order to minimize the possibility of damaging the lunar launch engine, it is desirable to use separate engines for lunar landing and lunar launch. These considerations lead to the three propulsion modules as shown.

B. Since earth storable hypergolic propellants are used in all propulsion systems in the present Three-Man Apollo spacecraft, it is thought desirable to use these propellants in as many of the propulsion systems as possible in the spacecraft under consideration. The initial studies, however, indicated that it probably is not feasible to accomplish the

mission using these propellants throughout. Cryogenic propellants are thus used in the retrograde module.

The retrograde module supplies the velocity increments required during the translunar portion of the mission up to the staging point, approximately 6000 feet above the lunar surface. Final letdown, translation and hover is performed by the terminal landing module. The remaining propulsion requirements, lunar launch through transearth midcourse corrections, are supplied by the service module.

All three of the propulsion stages are based on utilizing state-of-the art engine developments. The engine postulated for the retrograde module is a growth version of the Pratt and Whitney RL10A-3. In the terminal landing and service modules, the engines are based on a reasonable development of any one of three current engine programs; i.e., Aerojet General's 8000 pound thrust Transtage engine; Reaction Motors' Model TD-294 10,000 pound thrust LEM research engine; and a 10,000 pound thrust engine under investigation by Rocketdyne for possible use in the LEM. The engine in the terminal landing module is ablatively cooled, throttleable, and has an expansion ratio of 20:1. The same basic engine is used in the service module with the throttling feature removed and a radiation cooled skirt added to provide an expansion ratio of 40:1.

1.3 Weight Margins - The spacecraft propulsion systems are sized to take full advantage of the specified launch vehicle payload capability. The overall spacecraft propulsion system, then, has a "payload" capability which is the allowable weight of the command module and service module equipment. The comparison between the payload capability of the selected spacecraft configuration, for two propellant combinations, and the estimated command module plus service module equipment weight is given in Table 1-1.

TABLE 1-1

SPACECRAFT WEIGHT MARGINS

PROPELLANT COMBINATION		O_2/H_2 AND N_2O_4/MMH	O_2/H_2 AND OF_2/MMH
PAYLOAD CAPABILITY	LB.	7655	9430
PAYLOAD WEIGHT	LB.	6217	
[COMMAND MODULE	LB.]	[5069]	
[SERVICE MODULE EQUIPMENT*	LB.]	[1148]	
PAYLOAD MARGIN	LB.	1438	3213
PAYLOAD MARGIN	PERCENT	23	52

*TELECOMMUNICATIONS, GUIDANCE AND NAVIGATION, STABILIZATION AND CONTROL, ELCTRICAL POWER AND ENVIRONMENTAL CONTROL SYSTEMS AND STRUCTURAL SUPPORTS

Of the many propellant combinations being studied for space propulsion system applications, a particularly attractive combination is oxygen difluoride (OF_2) and monomethylhydrazine (MMH) because of its high specific impulse and high density impulse. Although a considerable amount of development effort is required in connection with the use of oxygen difluoride, it is believed that it could be available in the specified time period. Since the oxidizer to fuel ratio and

1.3 (Continued)

bulk specific gravity are nearly the same for OF_2/MMH and N_2O_4/MMH, the propulsion systems in the terminal landing module and service module could be designed to be compatible with the use of OF_2 and thus OF_2 could be substituted directly for N_2O_4 when it becomes available. Spacecraft payload capability with this substitution is also shown in Table 1-1.

The contractor was asked during the course of the study to express its opinion as to the confidence that can be placed in the estimated spacecraft weight and the margin that should be allowed for growth. Table 1-2 represents the contractor's best judgment in answer to this request, based on its past experience. The first item in the table is based on the assumption of no change in the design as conceived, i.e., no change in ground rules or design criteria. Under this condition, the actual vehicle weight should not exceed the present estimated weight by more than three percent with the stated probability.

TABLE 1-2

WEIGHT CONFIDENCE AND GROWTH

DESIGN WEIGHT

87-92% PROBABILITY WILL NOT BE EXCEEDED BY MORE THAN 3%

GROWTH THROUGH NECESSARY DESIGN CHANGES

80-85% PROBABILITY WILL NOT EXCEED 8%

OVER-ALL WEIGHT GROWTH

93-96% PROBABILITY CAN BE LESS THAN 20%

1.3 (Continued)

The last two items in the table are based on an analysis of the weight growth history of McDonnell products, the results of which are given in Figure 1-4. Based on the Mercury weight growth, it is believed that the weight growth of the spacecraft under consideration due to technological reasons, results of laboratory tests, more thorough systems analyses, etc. should not exceed eight percent with a probability of 80 to 85 percent. Finally, it is felt that with good weight control procedures, the total weight growth including some changes in spacecraft capability can be held to less than 20% with a probability of 93 to 96 percent.

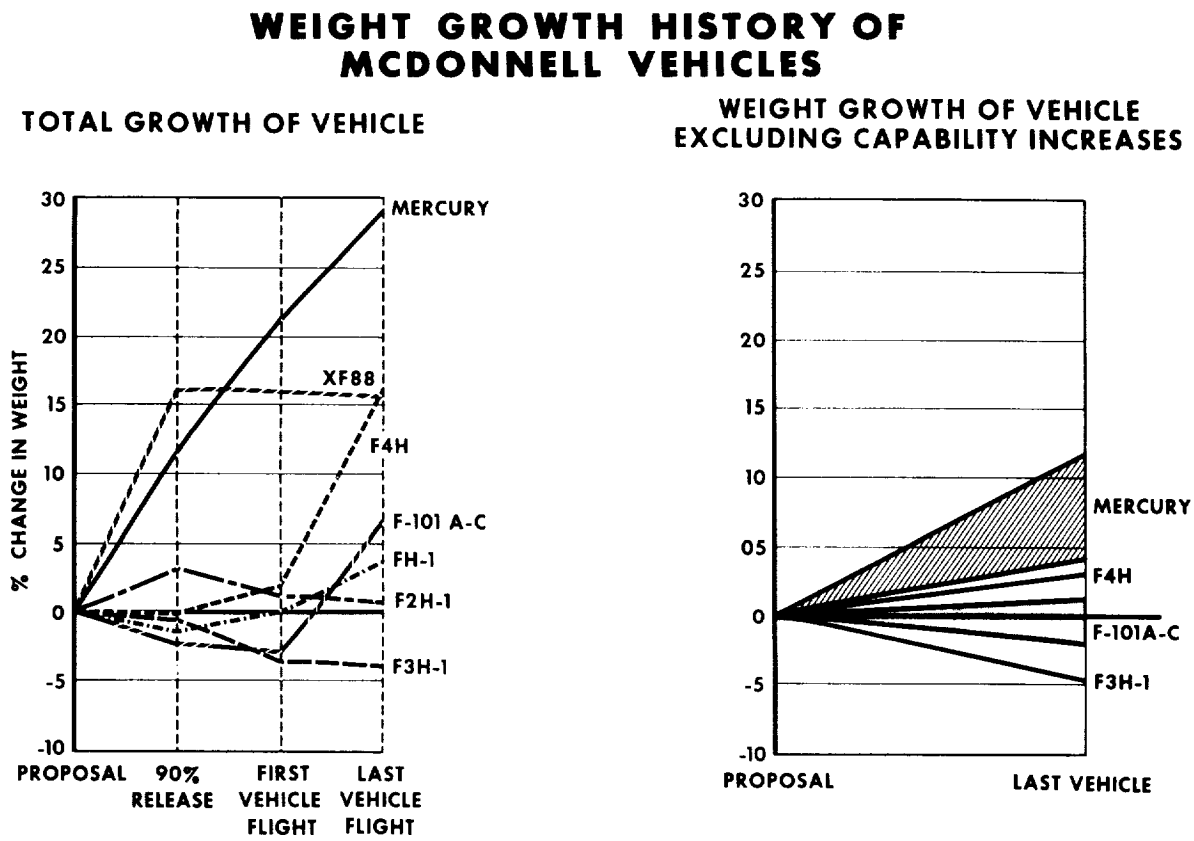


FIGURE 1-4

1.4 Summary Statements - Summaries of the major sections of the remainder of the report (section numbers indicated in parentheses) follow.

Spacecraft Configurations (2) - The Two-Man Apollo spacecraft consists of four basic units, the command module, service module, terminal landing module, and the retrograde module.

Command Module (2.1) - The command module is a 33 degree blunt cone with a 125 inch diameter base. The module has a hypersonic L/D of 0.475, achieved by offset center of gravity. The two crewmen enter and exit the module through a quick opening hatch in the side wall. A tower mounted launch escape system is attached to the conical end of the command module.

Service Module (2.2) - The truncated conical service module, located between the command and terminal landing modules, contains the lunar launch and transearth storable hypergolic propulsion system, electrical power, portions of the environmental control, communication and navigation systems, and a reaction control system.

Terminal Landing Module (2.3) - The terminal landing module with the integral lunar landing gear, is designed to provide a low landing silhouette. The module contains the storable hypergolic propulsion system for hover, translation and lunar landing, and a reaction control system for translunar ullage control and spacecraft attitude stabilization and control.

Lunar Landing Configuration (2.4) - The three modules described in the preceding paragraphs form the lunar landing spacecraft. Visual coverage of the lunar horizon and landing site is afforded the crew through an arrangement of windows, mirrors, and a television display.

1.4 (Continued)

Retrograde Module (2.5) - The cylindrical retrograde module houses the cryogenic propulsion system used for translunar midcourse corrections, injection into lunar orbit, and retrograde to approximately 6000 feet above the lunar surface. At this altitude, the module is jettisoned and final descent accomplished with the lunar landing spacecraft.

Weight Summary (2.6) - The weights of the Two-Man Apollo spacecraft are:

Module	Weight-Pounds
Command Module	5154*
Service Module Equipment	1148
Spacecraft Weight Margin	1438
Service Module	22,280
Terminal Landing Module	6030
Retrograde Module	53,787
Launch Escape System	2600
Landing Gear Fairing	1400
Gross Weight at Launch	93,837
Less (1 - % effective) x (jettisonable items)	-3837
Effective Launch Weight	90,000

*Includes 85 pounds scientific payload transferred to command module prior to lunar launch in normal mission.

Design Criteria (2.7) - The Two-Man Apollo spacecraft is designed in general accordance with the the technical guidelines of References (1-1) and (1-2). Exceptions are taken only where necessary in order to use developed equipment or where weight reductions can be realized without significant degradation in the probability of safe return or mission completion. Emphasis is placed on designing to conservative criteria where specific requirements are not spelled out by the customer.

1.4 (Continued)

Guidance and Navigation (3.1) - The guidance and navigation system for Two-Man Apollo is essentially the same as the system being developed by MIT for the Three-Man Apollo. It consists of an inertial system for attitude memory and acceleration measurements, navigation sensors (optical, radar, and DSIF), a computer for computation of guidance and display signals for nominal and abort trajectories, and electronic interface equipment required to drive appropriate displays. Total weight is 389 pounds, compared with 445 pounds for the Three-Man Apollo.

Stabilization and Control (3.2) - The stabilization and control system translates commands from the pilot or the navigation and guidance system into signals to the reaction control system (RCS) and the main engine gimbal and throttle actuators. Modes ranging from fully manual to fully automatic, depending on crew desires, are provided for both attitude and thrust control with redundancy in equipment, signal path, and torque application. Roll control is provided by the fixed thrust RCS thrust chambers at all times. Pitch and yaw control is obtained by main engine gimbaling during thrusting periods and by the RCS at all other times.

Long-term attitude reference signals are provided by redundant optical sensors (Sun-Horizon-Canopus). Short-term attitude memory is derived from a strapdown inertial unit. Rate gyros provide signals for tight-loop rate stabilization and rate command follow-up. Derived rate stabilization is used to achieve low fuel-consumption attitude hold during long low-disturbance coast phases without requiring direct sensing of the very low rates.

1.4 (Continued)

Manual attitude control is accomplished in a rate command mode by proportional hand controller displacements or in a high-reliability backup mode providing direct command control of the RCS solenoids, gimbal servos, and throttle actuators. A single-pulse direct mode is provided for fine manual rate control.

Environmental Control (3.3) - The environmental control system is identical to the equipment in the Gemini spacecraft with the exception of the volume differences dictated by the requirements for expendables.

Hydrogen and oxygen are stored in the super-critical state in the service module for the fuel cells and the ECS. The tanks are protected by super insulation to eliminate boiloff during all phases of the mission. The hydrogen tanks for the lunar retrograde stage are insulated solely to prevent liquefaction of air during the pre-launch and launch phase while the oxygen tanks require no insulation. Low emissivity coatings are provided on outer surfaces of tanks and inner walls of the module to prevent boiloff during the launch and translunar phases.

Electronic equipment is mounted on cold plates with temperatures being maintained at required levels by a coolant fluid that circulates through the cold plates and the space radiator.

Electrical Power (3.4) - Normal mission power is supplied from six fuel cell stacks located in the service module. Silver-zinc batteries in the command module provide re-entry and recovery power plus back-up for the fuel cells, as required. Additional silver-zinc batteries provide redundant isolated sources for squibs, relays and other transient producing equipment. Both the fuel cells and the batteries are identical to Gemini equipment.

1.4 (Continued)

A detailed electrical load analysis indicates that the mission requirements are 900 watts average. Sufficient fuel has been provided for the full 8 day mission, two days of which are contingency.

Telecommunications (3.5) - The telecommunication system operates with existing ground stations to provide a virtually continuous, two-way voice link between the spacecraft crew and the mission control center. In addition, television and spacecraft data may be transmitted to the earth.

For near-earth operations, primary voice communication is between a 20 watt VHF/AM transceiver, using an omnidirectional antenna, and the Mercury/Gemini ground network. Data transmission is provided by a 20 watt VHF/FM transmitter. Primary communication during the transit and lunar phases is by a 1 or 20 watt VHF transmitter/receiver system, with 4 foot diameter parabolic antennas, which operates in conjunction with the DSIF on earth. This system provides for both voice and data transmission.

Ground tracking of the spacecraft is aided by a C-Band transponder located in the command module. The VHF-DSIF transmitter and receiver operate as a phase coherent transponder to allow two-way doppler and "turn-around" range tracking by the DSIF network.

Full crew intercommunication is included and a relay transceiver provides crew communication during extravehicular operation. Recovery aids include both HF and VHF beacons and transceivers.

Structure (3.6) - Conventional spacecraft structures are employed in all modules, following the proven materials and concepts demonstrated in the Mercury and Gemini designs. Primary structure of each module consists of a semimonocoque shell with reinforcements around cut-outs and fittings to distribute localized

1.4 (Continued)

loads. Titanium is used as the basic shell material in all modules except the service module where beryllium sheet is used for the structural radiator shell.

The command module primary structure is designed to accommodate loads from launch abort, parachute, pressurization and earth landing. Primary structure design below the command module is based on launch conditions.

Heat Protection (3.7) - Re-entry heat protection is conservatively designed for a controlled 9000 nautical mile range, polar re-entry. The ablative material is M.A.C. Thermorad Shield S-3 elastomeric composite, which is being used for the Gemini blunt face heat shield. A 1.15 factor is applied to the predicted heating rates. Nominal material thermophysical properties are used, determined from Project Gemini testing. Total ablative material design weight is 571 pounds.

The launch escape system tower legs are covered with an ablative material in critical areas. Fiberglass fairings protect the lunar landing gear during launch.

Mechanical (3.8) - Wherever possible, mechanical systems are located entirely within pressurized areas and integral static seals are provided.

Crew hatches and doors employ over-center latching for positive and uniform action. All latching mechanisms are contained within the pressurized compartment. Mirrors for auxiliary vision are erected on single hinges and may be manually stowed or jettisoned by the crew.

Separation of the escape tower is accomplished with a three segment clamp ring, connected by explosive bolts, each incorporating dual ignition. Command to service module umbilicals are separated by explosive cutters, backed by spring loaded ejectors located in the service module. Electrical leads are also disconnected within the command module as a safeguard in the event of shorting.

1.4 (Continued)

The landing gear stows against the lunar retrograde structure and is protected during launch by fairings. Erection is accomplished by simple dual torsion springs. Earth impact is attenuated by crushable trussgrid between the heat shield and the pressure bulkhead and by individual seat snubbers.

Propulsion (3.9) - The characteristics of the propulsion systems for the service, terminal landing and retrograde modules are summarized in Table 1-3.

Reaction Control (3.10) - The characteristics of the reaction control systems for the service, terminal landing and retrograde modules are summarized in Table 1-3.

Launch Escape Propulsion (3.11) - The system consists of a tower mounted, solid propellant, stepped thrust rocket motor together with a tower jettison rocket for separating from the command module after escape rocket burnout. Launch escape propulsion system characteristics are presented in Table 1-3. Lateral separation is obtained through thrust alignment to a predetermined eccentricity.

Lunar Landing (3.12) - The lunar landing gear consists of four legs, each having three telescoping members. Impact energy is dissipated by crushing aluminum honeycomb cartridges within each telescoping leg. Dynamic stability during landing is achieved by selection of the relative crushing strength of the honeycomb in each strut element.

Earth Landing (3.13) - Impact energy is dissipated by an 11.5 inch layer of crushable trussgrid material which attenuates the command module to the crew emergency limits for the most severe landing. Viscous dampers on either side of each crew seat provide further attenuation to within the crew normal acceleration limits.

TABLE 1-3

PROPULSION SYSTEMS SUMMARY

LOCATION	SYSTEM	NUMBER OF ENGINES	THRUST LEVEL (VAC. I_{SP} - SECONDS)	PROPELLANTS (O/F RATIO)	TYPE ENGINE	FEED SYSTEM	PROPELLANT QUANTITY
TOWER	LAUNCH ESCAPE	1	*116,400 LB. FOR 1.0 SEC. STEPPED TO 70,000 LB. FOR 2.2 SEC. TAIL-OFF OF .3 SEC. (252)	SOLID			1220 LB.
COMMAND MODULE	REACTION CONTROL PITCH YAW ROLL	 4 4 4	 25 LB. (300 STEADY STATE) 25 LB. (260 PULSED) 50 LB.	N_2O_4/MMH (2.05)	ABLATIVE COOLED	POSITIVE EXPULSION	97 LB. TOTAL USABLE FUEL = 30.5 LB. USABLE OXIDIZER = 62.5 LB.
SERVICE MODULE	LUNAR LAUNCH AND TRANSEARTH INJECTION	1	11,700 LB. (320)	N_2O_4/MMH (2.2)	ABLATIVE & RADIATION COOLED	PRESSURE FED	19,188 LB. TOTAL USABLE FUEL = 5936 LB. USABLE OXIDIZER = 13,060 LB.
	REACTION CONTROL ROLL PITCH YAW	 8 PAIRED ROLL PAIRED ROLL	 25 LB. (300 STEADY STATE) 50 LB. TOTAL (260 PULSED) 50 LB. TOTAL	N_2O_4/MMH (2.05)	ABLATIVE COOLED	POSITIVE EXPULSION	501 LB. TOTAL USABLE FUEL = 162 LB. USABLE OXIDIZER = 333 LB.
	MIDCOURSE CORRECTION	4	100 LB. (300 STEADY STATE) (260 PULSED)	N_2O_4/MMH (2.05)	ABLATIVE COOLED	POSITIVE EXPULSION	
	MAIN PROPULSION PROPELLANT POSITIONING	USE PAIRED MIDCOURSE	200 LB. TOTAL (300 STEADY STATE) (260 PULSED)	N_2O_4/MMH (2.05)	ABLATIVE COOLED	POSITIVE EXPULSION	
TERMINAL LANDING MODULE	LUNAR LANDING	1	11,270 LB. MAX., 3415 LB. MIN. THROTTLEABLE (308)	N_2O_4/MMH (2.2)	ABLATIVE COOLED	PRESSURE FED	2668 LB. TOTAL USABLE FUEL = 826 LB. USABLE OXIDIZER = 1816 LB.
	REACTION CONTROL PITCH YAW ROLL	 4 4 8	 100 LB. (300 STEADY STATE) (260 PULSED)	N_2O_4/MMH (2.05)	ABLATIVE COOLED	POSITIVE EXPULSION	746 LB. TOTAL USABLE FUEL = 242 LB. USABLE OXIDIZER = 496 LB.
	MAIN PROPULSION PROPELLANT POSITIONING	USE PAIRED PITCH OR YAW	200 LB. TOTAL (300 STEADY STATE) (260 PULSED)	N_2O_4/MMH (2.05)	ABLATIVE COOLED	POSITIVE EXPULSION	
RETROGRADE MODULE	LUNAR RETROGRADE	1	26,500 LB. MAX., 2650 LB. MIN. THROTTLEABLE (430)	O_2/H_2 (5.0)	REGENERATIVE COOLED	TURBOPUMP FED	48,153 LB. TOTAL USABLE FUEL = 7923 LB. USABLE OXIDIZER = 39,617 LB.
	MIDCOURSE CORRECTION	1	26,500 LB. MAX., 2650 LB. MIN. THROTTLEABLE (430)	O_2/H_2 (5.0)	REGENERATIVE COOLED	TURBOPUMP FED	

*SEA LEVEL CONDITIONS, $\epsilon = 12$ $P_C = 1600$ PSIA

1.4 (Continued)

Command Module Size and Arrangement (4.1) - Parametric design studies of the command module indicate that provision of "sit-up" capability for the crew is more efficient than "stand-up" capability.

Equipment was arranged in the command module so as to provide an offset center of gravity for aerodynamic trim and accessibility for ease of maintenance. Space was provided for placement of crew in selected positions. The selected arrangement incorporated within the Project Apollo aerodynamic shape results in a maximum lift-to-drag ratio during re-entry of 0.388 instead of the desired 0.50. Several modifications which would increase L/D were investigated. Revision of the heat shield spherical-radius-to-diameter ratio from 1.2 to 1.6 was the most practical solution. This modification results in a lift-to-drag ratio of 0.475.

An airlock is not included in the selected configuration due to the associated weight penalty.

Spacecraft Staging (4.2) - Results are presented of trade-off studies involving number of stages, propellant combinations, and stage-apportioned incremental velocities. The spacecraft selected is based on a compromise among a number of factors. The more important of these are:

A. A minimum number of stages is desirable from a mechanization standpoint.

B. It is not desirable to land and launch with the same module because of possible stage damage during the landing operation.

C. While cryogenics are favored for the large translunar ΔV's, they are considered less desirable than earth storables for the final landing phase because of the ignition system requirement.

1.4 (Continued)

Trajectory Analysis (4.3) - Those phases of the mission that are affected by the spacecraft design are analyzed to verify that the various spacecraft systems are adequate from a flight mechanics standpoint. Maneuvers in the vicinity of the moon and abort capability using the service module propulsion system are emphasized.

Aerodynamics (4.4) - Changes to the basic Apollo configuration to meet the requirement for 0.5 L/D are discussed. Curves of basic parameters for both the abort and re-entry configurations are presented.

Propellant Margin (4.5) - Incremental velocity margins were assigned for each major contingency and apportioned to the various stages and phases of the mission. The results indicate that the ten percent margin specified in Reference (1-3) is realistic, but that individual module margins should range from 8.5% to 18%.

Radiation (4.6) - The minimum inherent shielding provided by the structure and equipment is shown to be 1.9 grams equivalent aluminum per square centimeter for the four cases investigated. The shielding provided to the outboard eye and the skin on the astronaut's back is investigated for the navigation and sleeping positions. Maximum protection to the eye is provided by having the astronaut in the navigation position, i.e., facing the instrument panel, while on duty and facing the heat shield while sleeping.

Meteoroid Protection (4.7) - In the command module, multiple walls, separated by low density insulation, provide inherent meteoroid protection for the pressurized volume and the crew. In the lower modules, various degrees of protection are provided by the multiple-wall effect of tankage and structural shells. However, the open ends of the propulsion modules expose the propellant tanks to direct meteoroid impact. It is shown that the weight associated with increasing

1.4 (Continued)

meteoroid protection is well within the contingencies which are included in the tankage weights to assure a high degree of confidence in the stage performance analysis.

Weight Derivation (4.8) - The weight derivations for the Two-Man Apollo spacecraft are based on analysis of the selected design and extrapolation of existing spacecraft data. Detailed comparisons of the Two-Man Apollo weights with those of Mercury, Gemini and the NAA Apollo spacecraft are made to substantiate the derivations. In cases where extensive extrapolation must be made, detailed analyses are performed and applicable contingencies allowed to assure a high degree of confidence in the estimated weights.

Reliability - Reliability has been a major consideration in configuring each of the spacecraft systems. However, a numerical analysis of the safety and mission reliability of the complete spacecraft has not been performed. In addition to selecting components which are currently in existence or under development, judicious use of redundancies and special system arrangements are made to enhance spacecraft reliability. Some of the systems given particular attention are propulsion, environmental control, electrical power, separation devices, reaction control, and earth landing. Reliability considerations are discussed in the spacecraft system descriptions in Section 3.

2. SPACECRAFT CONFIGURATION

The spacecraft general arrangement for the two-man direct flight lunar landing mission is presented in Figure 2-1. Modular assembly is utilized and consists of five units, i.e., launch escape system, command module, service module, terminal landing module, and retrograde module. A detailed description of the launch escape system is presented in Section 3.11. Descriptions of the remaining four modules are presented in Sections 2.1 through 2.5.

2.1 Command Module - Reference 2-1 indicates the command module should provide a degree of crew mobility allowing stand-up capability. A design objective in the selection of the arrangement is utilization of the Project Apollo re-entry spacecraft aerodynamic configuration, i.e., a 33-degree symmetrical cone and a heat shield spherical radius of 1.2 times the heat shield diameter. A further goal is the realization of a maximum lift to drag ratio of 0.5 during re-entry.

As discussed in Section 4.1, an adequate and more efficient module arrangement is realized by providing a sit-up and stretch-out rather than a stand-up capability. The study also demonstrates that the L/D goals can be met with some saving in weight by deviating from the Apollo shape. In the configuration selected, the heat shield radius to diameter ratio is modified to 1.6 resulting in a trim L/D of 0.475. A more detailed discussion of this compromise is presented in Section 4.4. Incorporation of these modifications results in the module general arrangement presented in Figure 2-2.

For spacecraft design simplicity, the inhabited volume is completely contained in the command module. To minimize weight, the equipment included in the command module is restricted to that required during earth re-entry or which must be accessible to the crew during flight. The following paragraphs are a synopsis of the command module salient features and systems.

TWO-MAN APOLLO SPACECRAFT GENERAL ARRANGEMENT

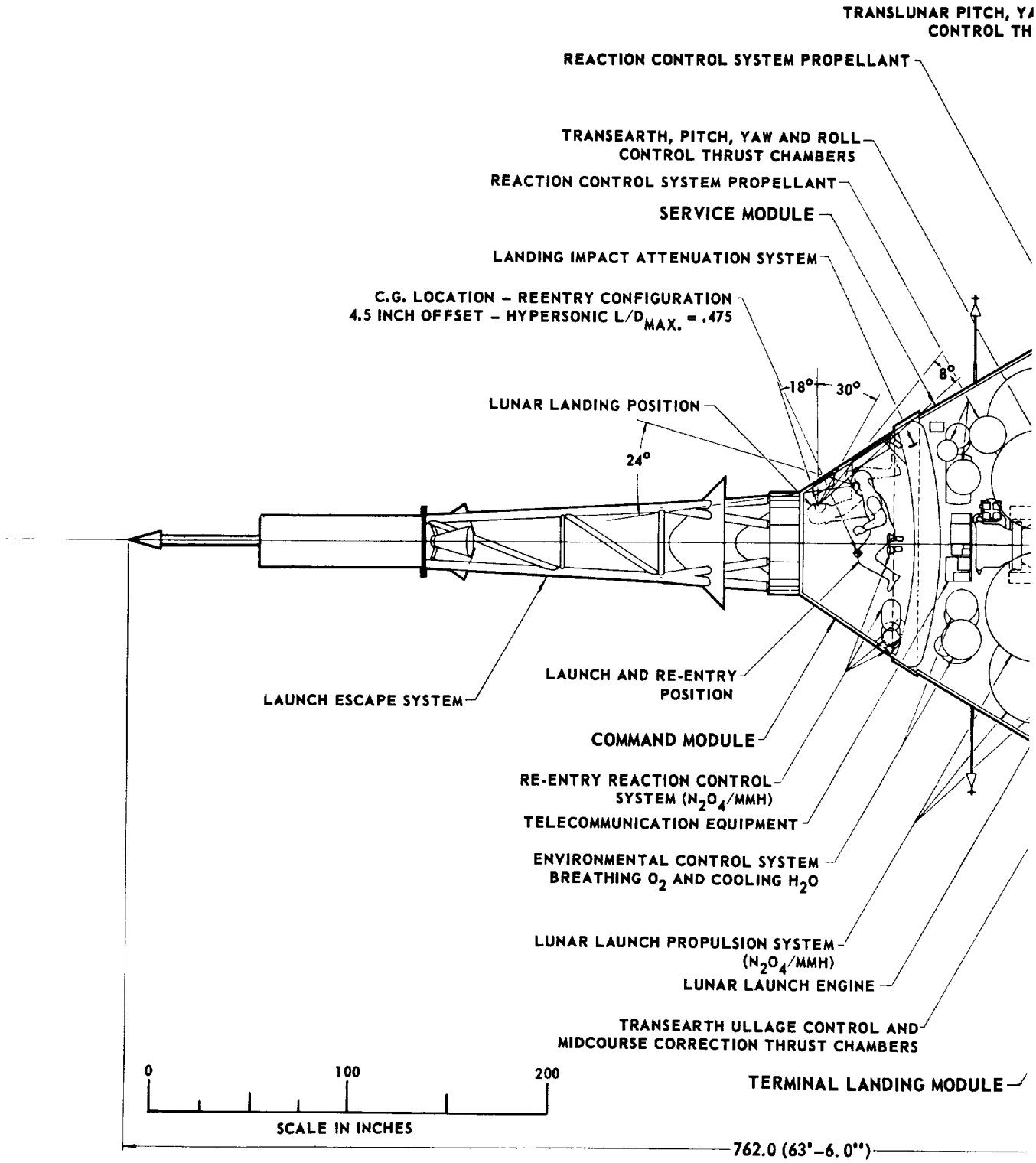
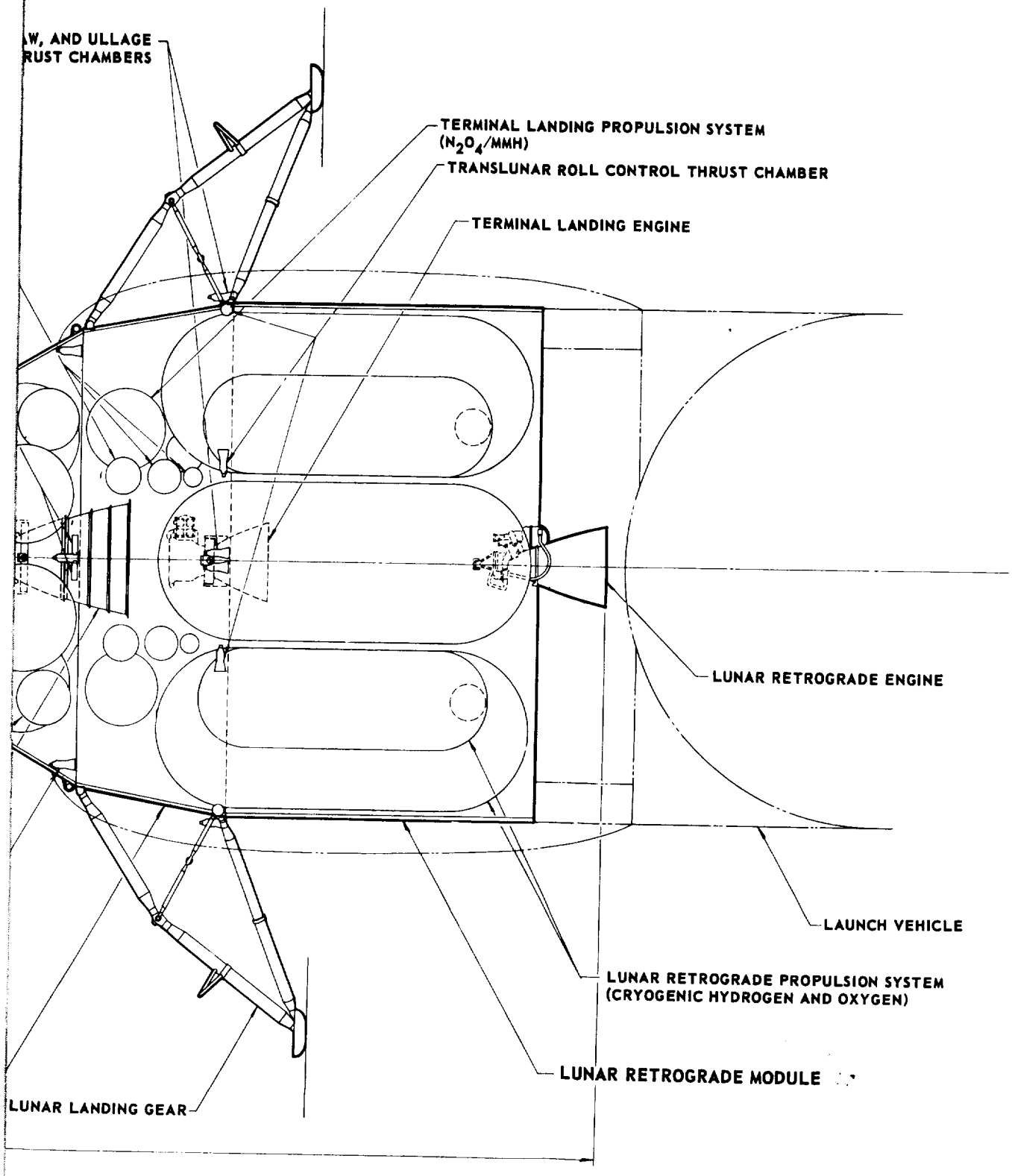


FIGURE 2–1



FOLDOUT FRAME 2

COMMAND MODULE

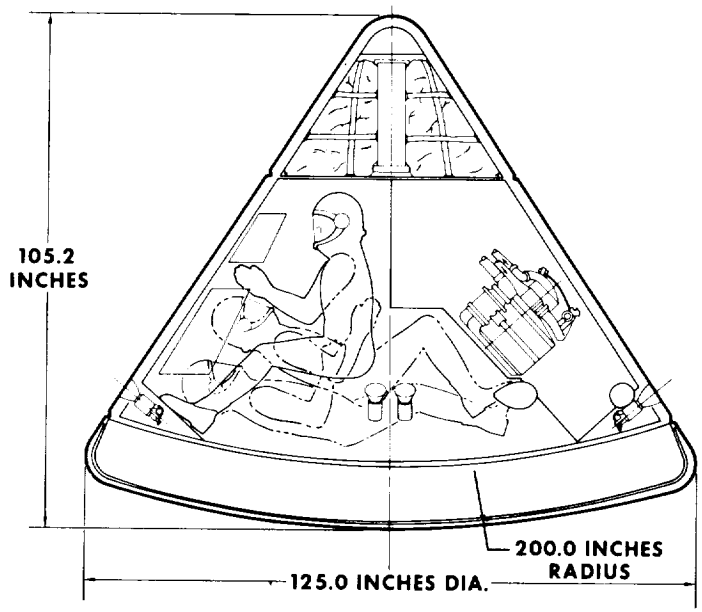


FIGURE 2-2

2.1.1 Trajectory Considerations

A. Tower launch escape system.

B. Trajectory control during re-entry - offset center of gravity (L/D_{max} = 0.475) and roll control.

C. Parachute landing system.

D. Earth landing at preselected ground sites.

2.1.2 General Configurations

A. 33-degree blunt cone; 125 inches diameter at the heat shield.

B. Heat shield R_N/D = 1.6.

C. Four crew positions with appropriate controls and displays - launch and re-entry, lunar landing, navigation, and rest. (Figures 2-3 through 2-5).

D. Normal ingress-egress through quick-opening hatch at pilot station.

LAUNCH AND RE-ENTRY POSITION

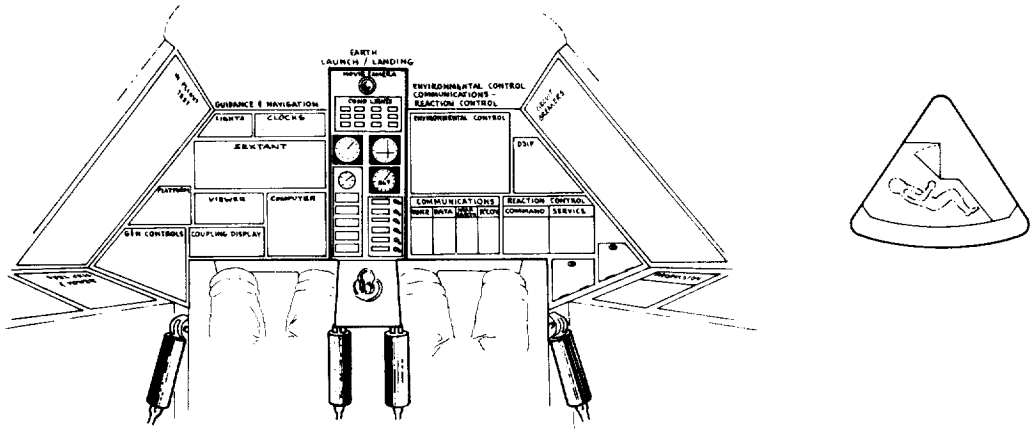


FIGURE 2-3

LUNAR LANDING POSITION

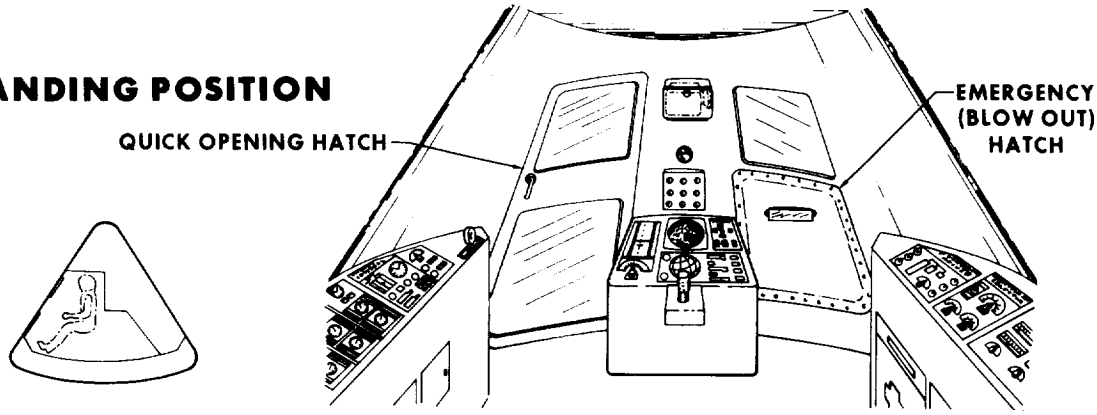


FIGURE 2-4

NAVIGATION POSITION

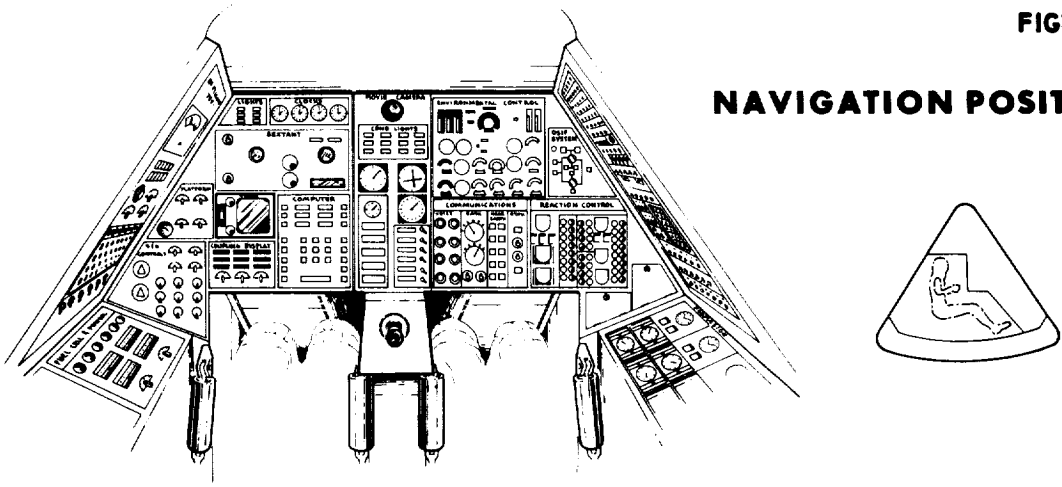


FIGURE 2-5

2.1.2 (Continued)

E. Emergency egress through normal ingress-egress hatch and hatch (blow-out type) at co-pilot station. (Figure 2-4).

F. Two windows in normal ingress-egress hatch, and one window adjacent to emergency hatch.

G. Pilot lunar landing vision - direct through windows augmented by television display.

H. Co-pilot lunar landing vision - direct through window augmented by mirror and television display.

I. Crew couches serve as basic body support for all crew positions (Figure 2-6).

J. Quick disconnect harnesses provide body restraint.

K. Couches supported by hydraulic snubbers and deformable structural elements for earth impact attenuation augmentation.

L. Meteoroid protection provided by heat shielding and structure.

M. Corpuscular radiation protection provided by heat shielding, structure, equipment, and window panes.

N. No airlock provided (Section 4.1).

2.1.3 Structure and Heat Protection.

A. Titanium single skin pressure walls .016 thick.

B. Titanium sheet metal stringers .032 thick.

C. Large pressure bulkhead - 1.0 inch aluminum honeycomb with .032 inner and .040 outer aluminum face plates.

D. Small pressure bulkhead - single skin titanium .016 thick.

E. Outer shingles - .2 to .25 inches M.A.C. S-3 elastomeric composite on titanium single face corrugation panels.

2.1.3 (Continued)

F. Heat shield - .72 to 1.38 M.A.C. S-3 elastomeric composite on .010 titanium panel.

G. Heat shield mounted on 11.5 inch crushable trussgrid shock absorber (9 inch stroke) for impact attenuation.

CREW SUPPORT COUCH ARRANGEMENT

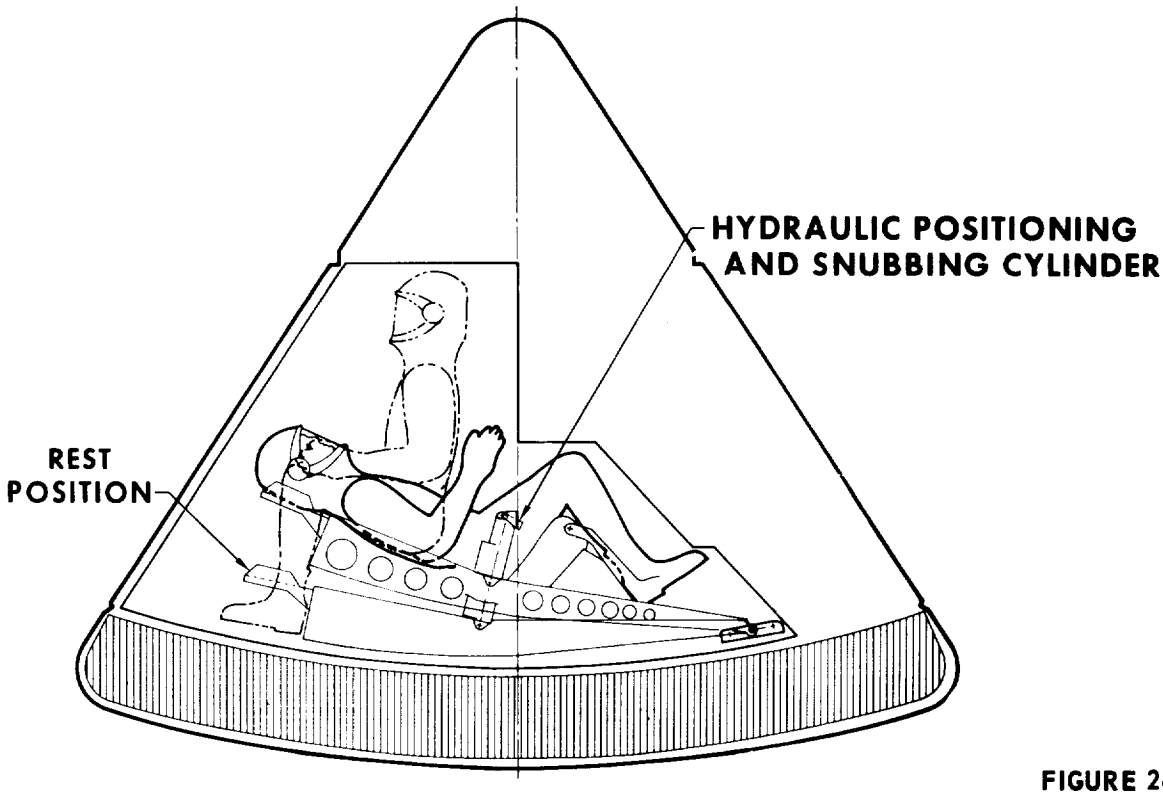


FIGURE 2-6

2.1.4 Environmental Control

A. Atmosphere

1. Pressure - 5 psi normal, 3 1/2 psi emergency.
2. Temperature $70^{\circ} \pm {}^{20^{\circ}}_{10^{\circ}}$ F.
3. Relative humidity 55 ± 20%.
4. Composition - O_2
5. Crew consumption rate - 1.8 lb/man day.
6. Cabin leakage rate - nominal .15 lb/hr.
7. Normal O_2 supply - stored in service module.

2.1.4 (Continued)

8. Emergency and re-entry O_2 supply - 20 lbs. (gaseous) in two containers.
9. CO_2 partial pressure - 7.6 mm Hg maximum.
10. CO_2 production rate - 2.3 lb/man day.
11. CO_2 removal - lithium hydroxide.
12. Odor removal - activated charcoal.
13. Humidity control - condensation and wick type water separation.

B. Temperature Control

1. Heat provided by electrical equipment, metabolic and solar heat.
2. Metabolic heat - 11,300 btu/man/day.
3. Crew cooled by ventilation and evaporation.
4. Atmosphere cooled by OS-139 heat exchanger.
5. Equipment cooled by OS-139 cold plates.
6. Equipment isolated from inhabited volume by thermoshields.
7. OS-139 cooling by service module system.
8. Re-entry cooling - crew by secondary O_2, equipment utilizes thermal capacity.
9. Impact trussgrid insulated by .50 MIN-K-501.
10. Stringers insulated by .20 MIN-K-501 strips.
11. Pressure wall insulated by 1.50 RF-700.

2.1.5 <u>Electrical Power</u>

A. Normal power from service module system.

B. Squib power - three 15 amp-hr. silver zinc batteries.

C. Re-entry and post landing power - five 40 amp-hr silver zinc batteries.

2.1.6 Communication

A. VHF voice transmitter and receivers.

B. Telemetry transmitter.

C. PCM data system.

D. HF and VHF recovery beacons.

E. C-band radar transponder.

2.1.7 Navigation

A. Attitude reference - Sun-Canopus system using sun and star sensors and inertial platform memory. (Canopus sensor located in service module).

B. Transit guidance - manual sextant and digital computer.

2.1.8 Attitude Control

A. Spacecraft transit orientation - roll axis on sun line with command module toward sun. Yaw axis in sun-canopus-spacecraft plane.

B. Attitude orientation held to $\pm$ 5 degrees.

C. Re-entry reaction controls - eight 25 lb. thrust pitch and yaw engines and four 50 lb. thrust roll engines.

D. Propellants - positively expelled MMH and N_2O_4.

2.1.9 Human Support

A. Pressure suit with provisions for sanitation and waste elimination. Worn throughout mission.

B. Food - 2 lb/man day, dehydrated and precooked.

C. Water - 12.5 lb/man day generated by service module electrical system.

D. Waste - body eliminations stored in self-sealing plastic fecal bags.

E. Sanitation - disposable deodorant pads for underarms and crotch, electric razor with vacuum collector.

F. First aid kit.

G. Portable ECS back-pack for extra spacecraft operations.

2.1.9 (Continued)

H. Voice transceiver for crew communication.

I. Personal radiation dosimeters.

J. Radiation garment and overshoes worn over pressure suit.

K. Personal parachute.

L. Personal survival kits.

M. Recovery aids - flashing light, dye marker, shark repellant and sofar bomb.

The arrangement of the equipment within the command module is shown in Figure 2-7. Special design emphasis is placed upon maintainability by providing access which does not require adjacent systems or components to be disturbed. This access is via the normal ingress-egress door augmented by several external panels shown in Figure 2-8. These panels provide access to the re-entry reaction control system, the environmental system canister, and the optical measuring unit.

COMMAND MODULE

EQUIPMENT ACCESS DOORS

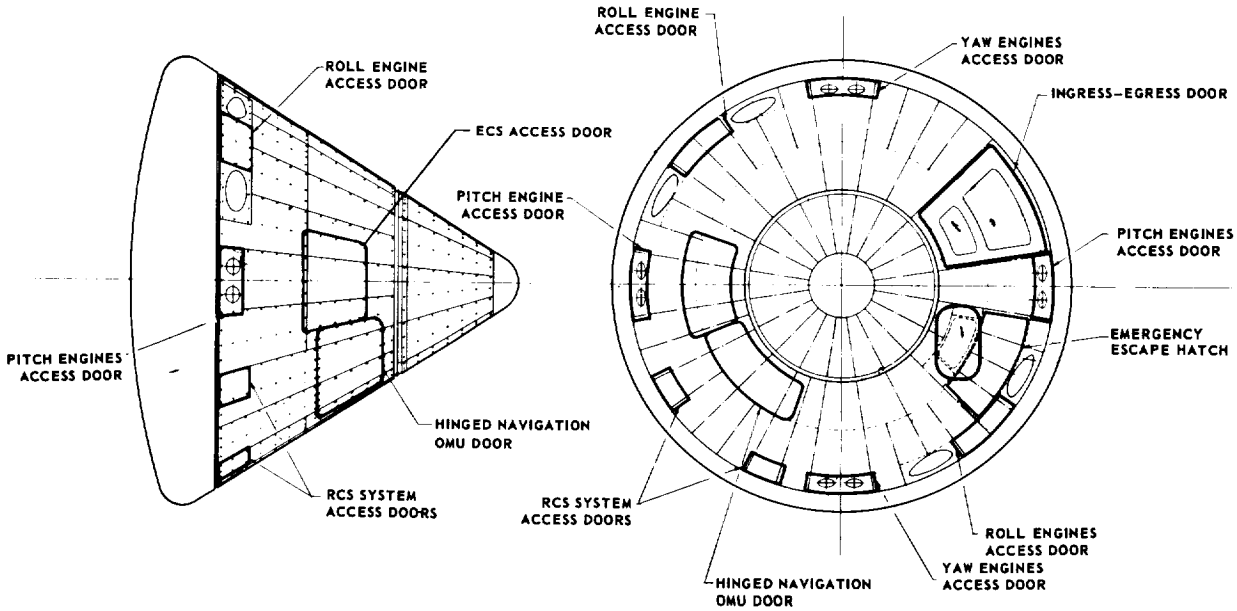
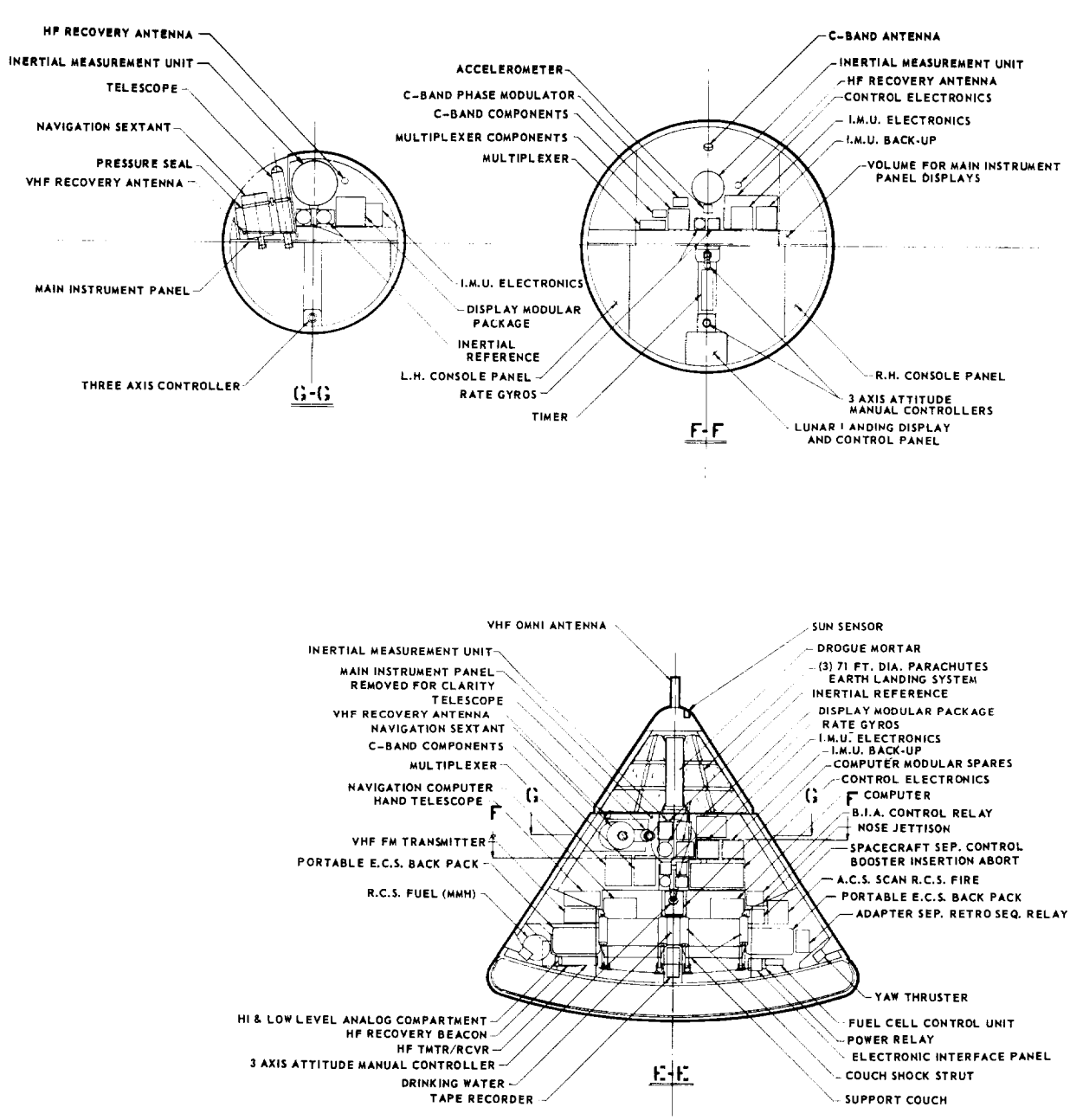


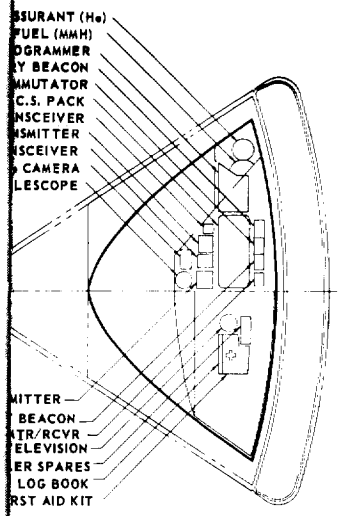
FIGURE 2-8

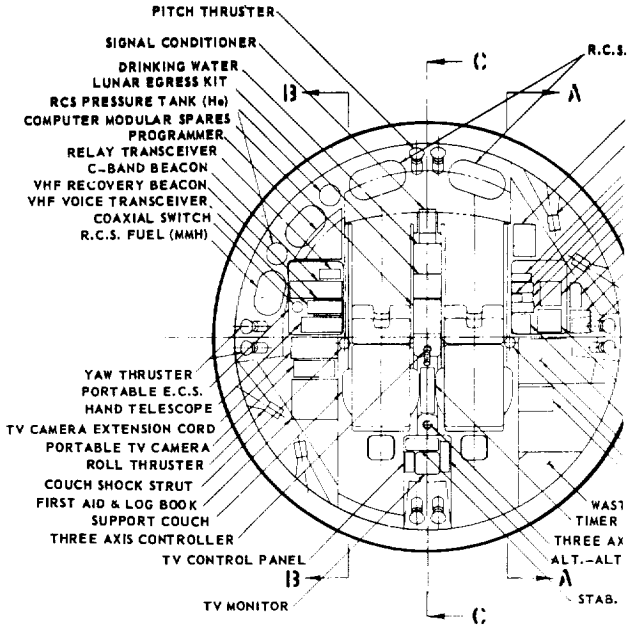
COMMAND MODULE
INTERIOR ARRANGEMENT

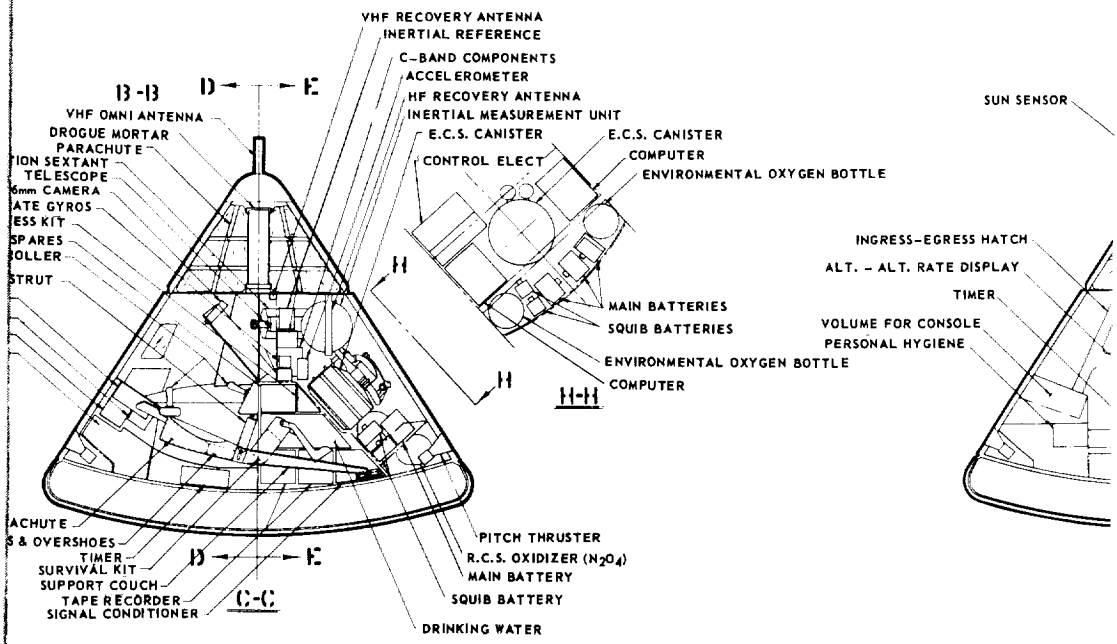


FOLDOUT FRAME 1

FIGURE 2-7







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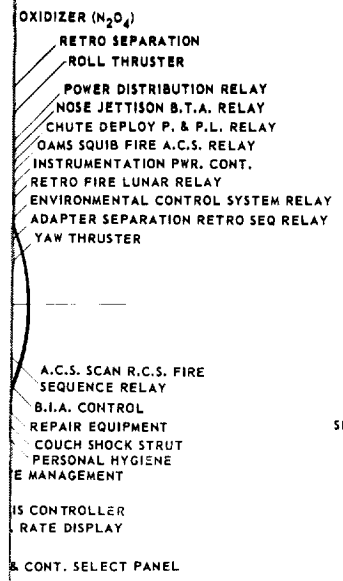
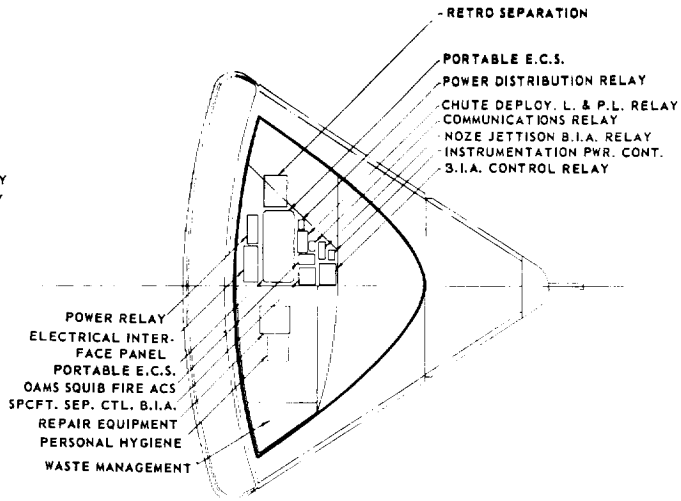
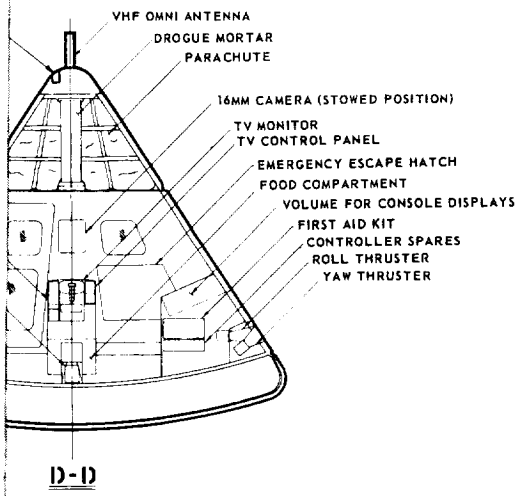
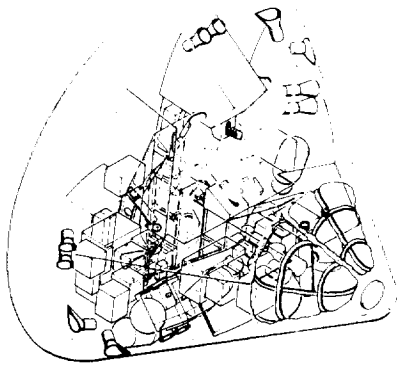




A - A



D - D



FOLDOUT FRAME 3

2.2 Service Module - The service module serves two basic functions; a) it provides space for systems which are required to accomplish the mission of lunar landing and return but are not required for earth re-entry, and b) it houses the lunar launch propulsion system and transearth reaction control system. The interior arrangement of the service module is shown in Figure 2-9. The external shape is a truncated cone, 102.5 inches long, 112 inches in diameter at the command module interface and 232 inches in diameter at the terminal landing module interface. This configuration was selected to obtain the lowest position of launch propellants and the minimum length module. These factors contribute to lowering the lunar landing center of gravity, thereby minimizing lunar landing gear length and weight.

2.2.1 Structure

A. Skin - cross rolled beryllium .010 thick.

B. Stringers - cross rolled beryllium sheet metal .010 thick.

C. Rings - beryllium sheet metal .025 thick.

D. Engine and propellant system support-aluminum tubular truss.

2.2.2 Environmental Control

A. Passive module temperature control - low emissivity coating on inner structural surface and plastic bulkhead at terminal module interface.

B. Command module liquid coolant (OS-139) cooled by coated space radiator on module surface.

C. Command module O_2 supply - supercritically stored (nominal 850 psia).

2.2.3 Electrical Power

A. Two Gemini fuel cell sections at 1.05 KW each.

B. Fuel cell capacity 8500 amps-hrs.

C. Fuel cell cooling by coated space radiator on module surface.

SERVICE MODULE

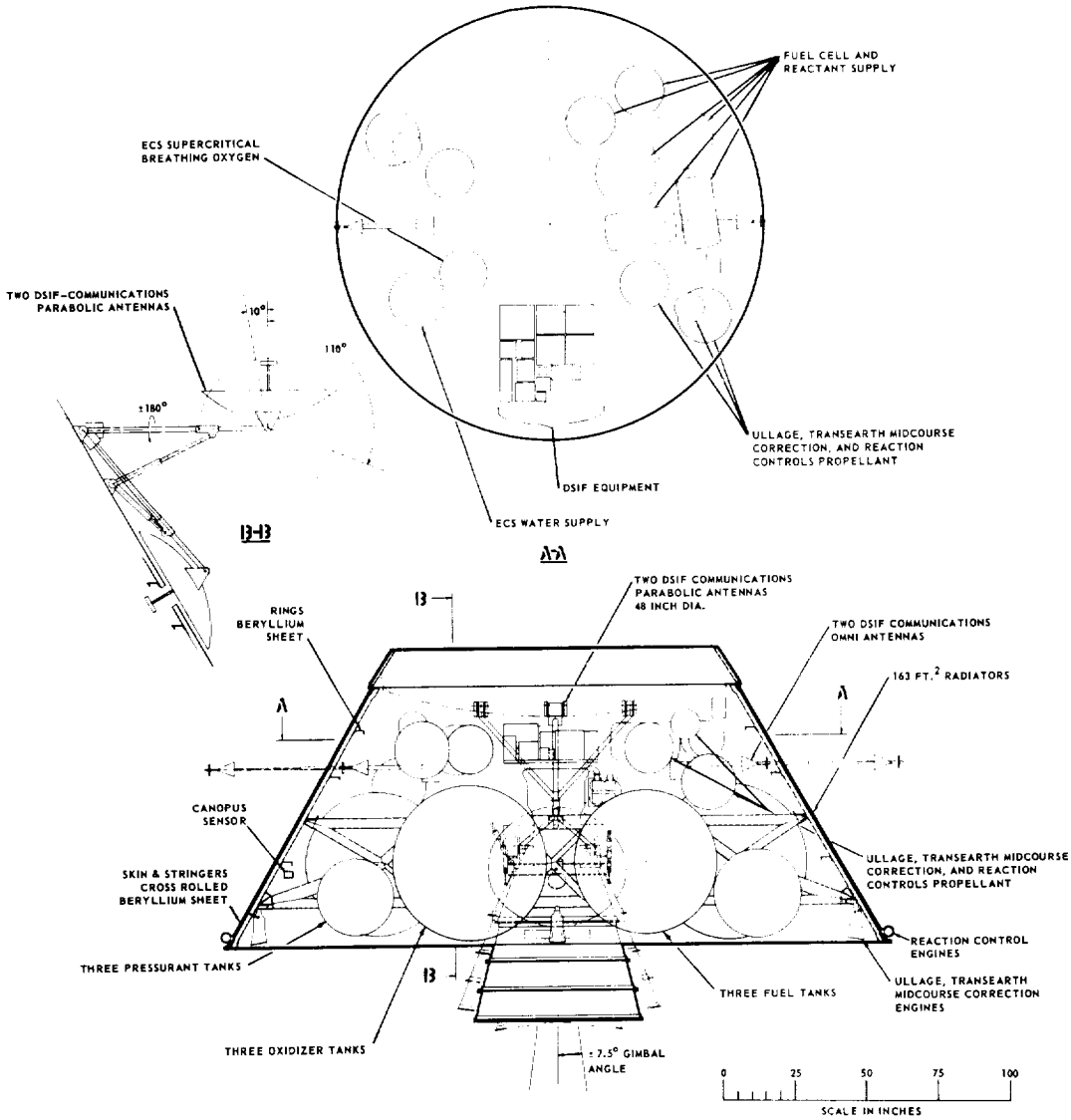


FIGURE 2-9

2.2.4 Electronics

A. DSIF transmitter and receiver for communication and ground based backup navigation system.

B. Externally mounted parabolic antennae (two).

C. Antennae protected during launch by jettisonable fairings.

2.2.5 Reaction Control

A. Four 100 lb. thrust engines for transearth ullage control and midcourse corrections.

B. Eight 25 lb. thrust engines for transearth attitude control.

C. Propellants - positively expelled MMH and N_2O_4.

D. Expulsion pressurant - helium.

2.2.6 Propulsion

A. Single 11,700 pound thrust pressure fed engine.

B. Engine ablative cooled with radiation skirt.

C. Engine gimbaled $\pm$ 7.5 degrees.

D. Propellants - MMH and N_2O_4.

E. Propellant pressurant - helium.

The service module design provides access to all principal groupings of functional systems, plumbing and wiring terminals. Four doors are provided near the command module interface for maintenance prior to launch. Adequate access for passage of an astronaut, dressed in a fully pressurized space suit, is provided via the terminal landing module for maintenance in space and prior to launch.

2.3 Terminal Landing Module - The terminal landing module contains the propulsion systems for translunar ullage positioning, translunar attitude control, hover, translation and final descent to the lunar surface. The module includes the lunar landing gear and the scientific equipment. At the termination of the lunar stay, the module serves as the launch platform for the transearth spacecraft.

The general arrangement of the terminal landing module is shown in Figure 2-10. The external shape is a truncated cone, 78 inches long, 232 inches in diameter at the service module interface and 260 inches in diameter at the retrograde module interface. This length was selected as a minimum compatible with efficient lunar landing gear support structure. Further, this minimum length contributes to reducing landing gear length and weight.

A synopsis of the terminal module systems and features is given in the following paragraphs.

2.3.1 Structure

A. Skin - titanium .010 thick.

B. Stiffeners - titanium corrugation .010 thick.

C. Rings - titanium .025 thick.

D. Engine, propellant, landing gear, and scientific gear support - aluminum tubular truss.

E. Four externally mounted lunar landing gear.

F. Each gear has three shock struts utilizing crushable aluminum honeycomb as energy absorber.

G. During earth launch gear are retracted against side of terminal and retrograde modules and are protected by jettisonable fairings.

TERMINAL LANDING MODULE

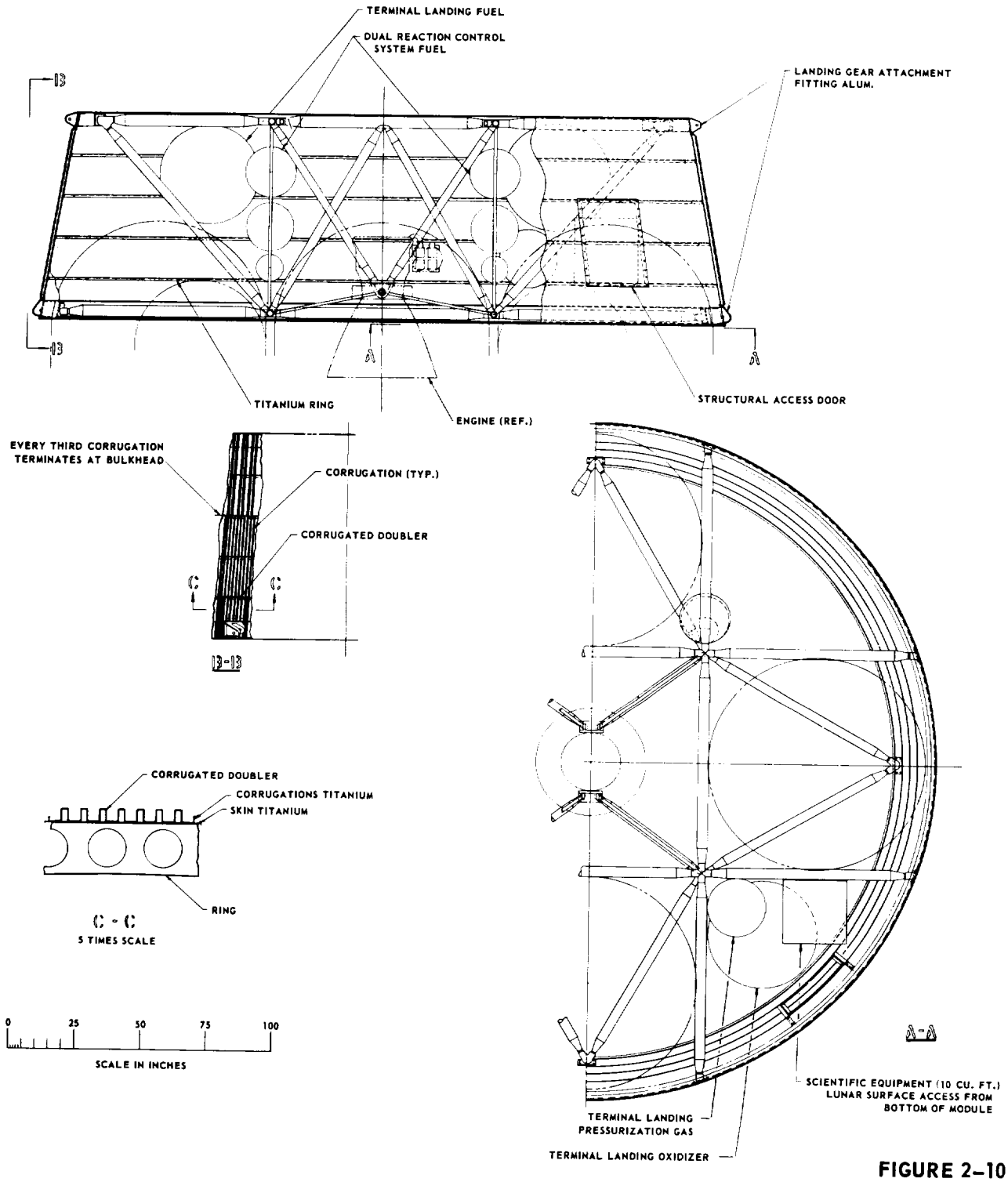


FIGURE 2-10

2.3.2 Reaction Control

A. Sixteen 100 pound thrust engines for translunar attitude and ullage control.

B. Propellants - positively expelled MMH and N_2O_4.

C. Propellant pressurant - helium.

2.3.3 Propulsion

A. Single, 11,270 pound maximum thrust, pressure fed engine.

B. Throttling ratio - 3.3:1.

C. Engine ablative cooled.

D. Engine gimbaled $\pm$ 6 degrees.

E. Propellants - MMH and N_2O_4.

F. Propellant pressurant - helium.

Access into the terminal module, before retrograde module staging, is accomplished through two doors in the side wall. On the lunar surface, the interface at the retrograde module is exposed and provides adequate passage for an astronaut dressed in a fully pressurized space suit.

2.4 Lunar Landing Configuration - The modules described in the preceding three sections joined together, form the lunar landing spacecraft. This configuration is shown in Figure 2-11.

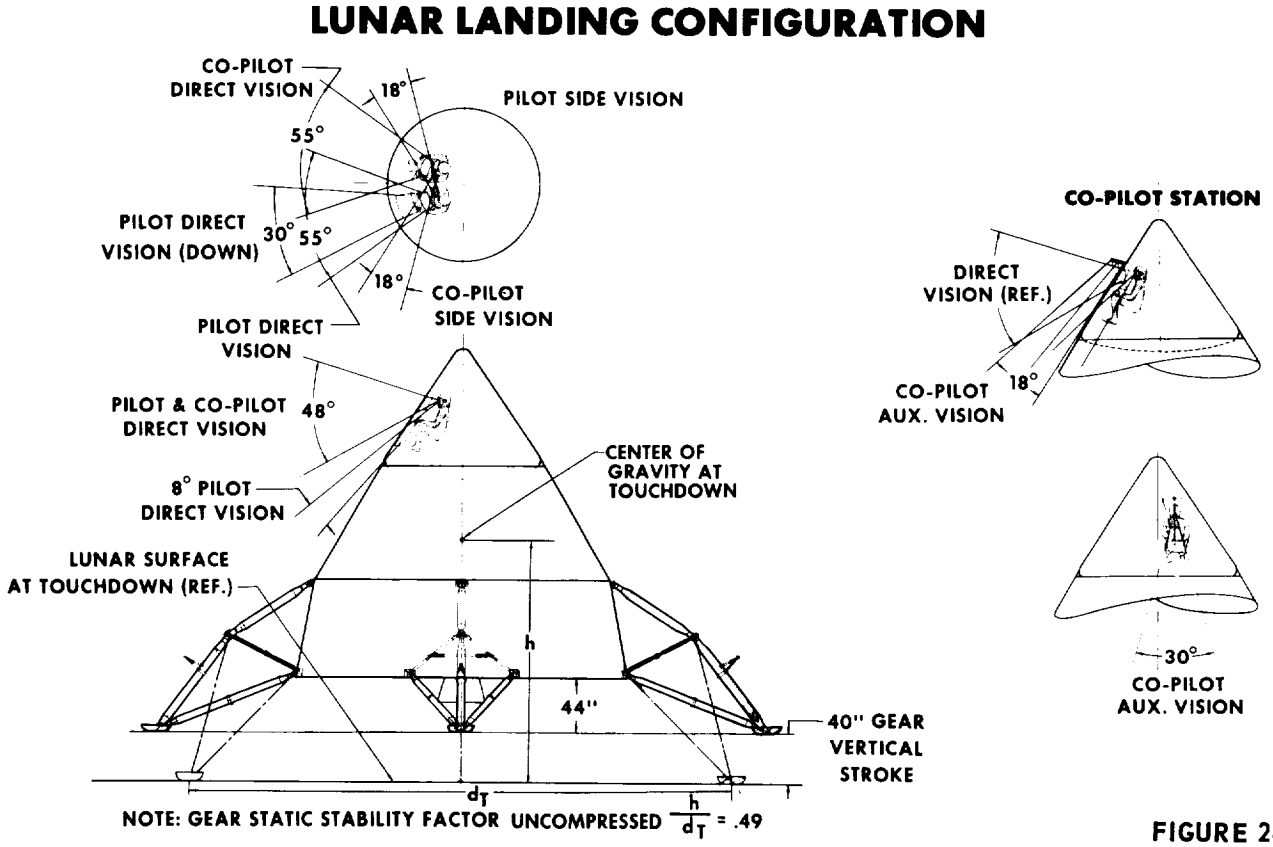


FIGURE 2-11

The command module windows afford the pilot direct visual coverage of the horizon and the landing area immediately beneath one of the four landing gears. The co-pilot has a direct window view and a mirror view of a portion of the landing area. A television presentation of the lunar surface immediately beneath the spacecraft is provided as a visual aid to the pilot and co-pilot.

The landing gear is configured to provide a minimum of 44 inches of clearance between the spacecraft structure and the lunar surface. A digital computer analysis of the landing gear was conducted to define the landing stability characteristics. The results are discussed in Section 3.12.

At lunar launch, the terminal landing module is separated from the command module - service module combination and serves as the launch platform for the transearth spacecraft configuration.

2.5 Retrograde Module - The retrograde module provides the spacecraft propulsion for translunar midcourse corrections, injection into lunar orbit, retrograde and braking to approximately 6,000 feet above the lunar surface. The arrangement of this 260 inch diameter, 160 inch long cylindrical module is shown in Figure 2-12.

The following paragraphs are a synopsis of the retrograde module salient features.

2.5.1 Structure

A. Skin - titanium .010 thick upper half of module, .016 thick lower half of module.

B. Stiffeners - titanium corrugations .010 thick upper half of module, .016 thick lower half of module.

C. Rings - titanium sheet metal .025 thick.

D. Propellant tank upper support 3.0 inch aluminum honeycomb bulkhead with .032 aluminum face plates.

E. Engine and tank lower supports - aluminum tubular truss.

2.5.2 Propulsion

A. Single, 26,500 pound maximum thrust, pump fed engine.

B. Throttling ratio 10:1.

C. Engine regeneratively cooled.

D. Engine gimbaled ± 6 degrees.

E. Propellants - cryogenic hydrogen and oxygen.

F. Fuel pressurant - heated hydrogen.

G. Oxidizer pressurant - heated helium.

RETROGRADE MODULE

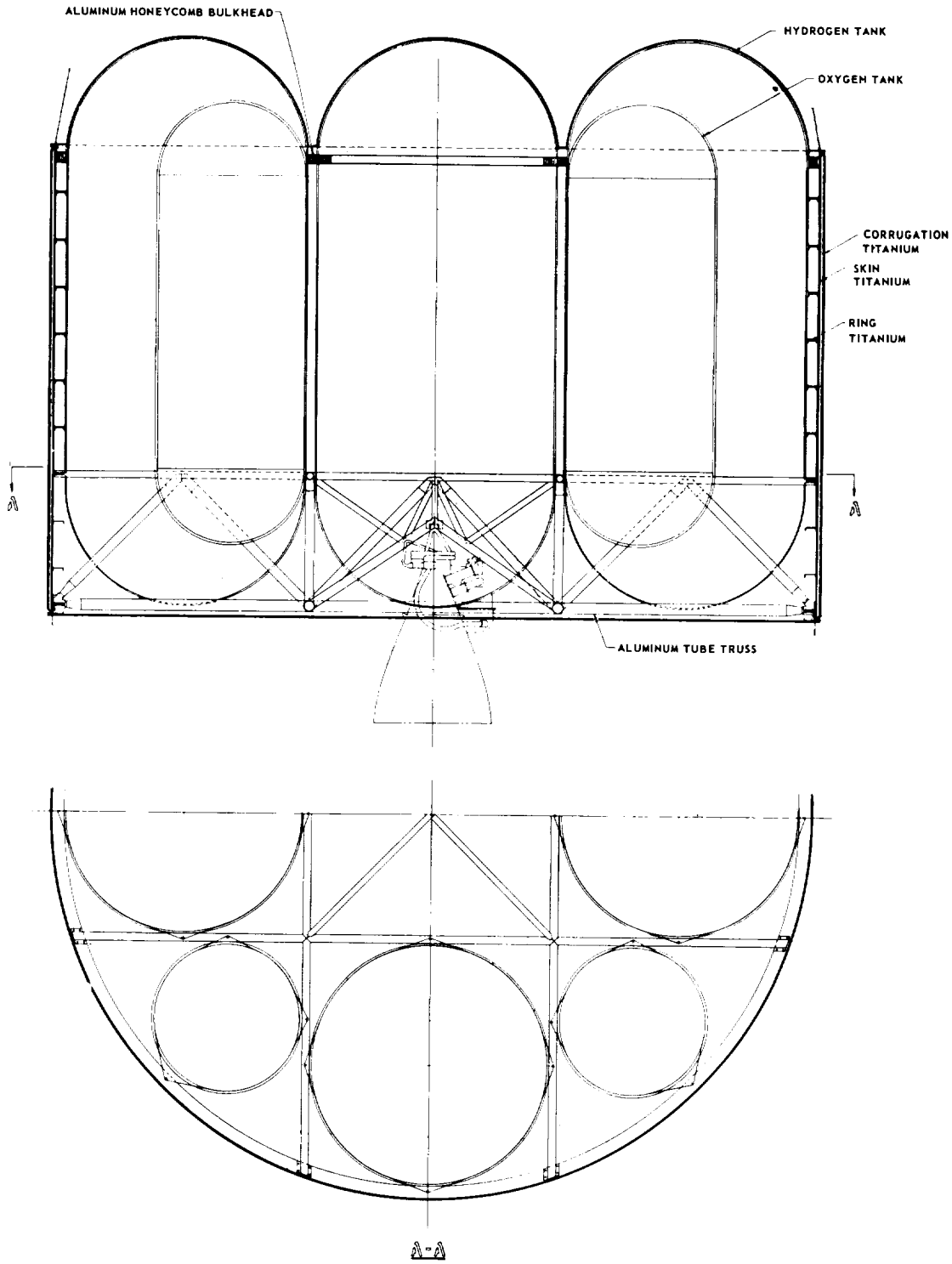


FIGURE 2-12

2.6 Weight Summary - The spacecraft weight summary is presented in Table 2-1. The service module, terminal landing module and retrograde module designs are based on the Saturn C-5 capability of injecting 90,000 pounds effective payload to a lunar transfer orbit. The command module weights and service module equipment weights are based on the spacecraft as configured in Figure 1-1. The margin existing between the propulsive systems capability and the command module service module equipment weight is shown as "spacecraft weight margin".

The design mission is the lunar landing "touch and go" abort prior to usage of any lunar rest E.C.S. expendables or transfer of 85 pounds of scientific payload to the command module. Designing to the lunar launch weight of this abort mission rather than the lighter nominal mission reduces the spacecraft weight margin by 39 pounds. The design weights of the re-entry vehicle are established from the heavier weight corresponding to the nominal mission.

The physical characteristics of weight center of gravity and moments of inertia throughout the nominal mission history are shown in Table 2-2. The "spacecraft weight margin" is considered to be part of the command module for balance and inertia calculations. The weight history of the lunar landing "touch and go" abort is shown in Table 2-3.

TABLE 2-1

SPACECRAFT WEIGHT SUMMARY

GROUP	WEIGHT – POUNDS		
COMMAND MODULE (RE-ENTRY CONDITION)			5154
STRUCTURAL SYSTEMS		1675	
BASIC STRUCTURE	1042		
HEAT SHIELD	633		
CREW SYSTEMS		905	
COMMUNICATIONS AND INSTRUMENTATION		767	
DISPLAYS	348		
TELE-COMMUNICATIONS	419		
GUIDANCE AND NAVIGATION		297	
STABILIZATION AND CONTROL		204	
REACTION CONTROL		210	
ELECTRICAL POWER		321	
ENVIRONMENTAL CONTROL		304	
EARTH LANDING		386	
SCIENTIFIC EQUIPMENT (TRANSFERRED FROM TERMINAL MODULE AT LUNAR SURFACE)		85	
SERVICE MODULE EQUIPMENT			1148
STRUCTURAL SUPPORTS		54	
TELE-COMMUNICATIONS		249	
GUIDANCE AND NAVIGATION-STABILIZATION CONTROL		13	
ELECTRICAL POWER		456	
ENVIRONMENTAL CONTROL		376	
SPACECRAFT WEIGHT MARGIN			1438
SERVICE MODULE			22280
STRUCTURE		1079	
PROPULSION SYSTEM		1294	
PROPELLANT (INCL. TRAPPED)		19188	
EQUIPMENT		80	
REACTION CONTROL SYSTEM		639	
TERMINAL LANDING MODULE *			6030
STRUCTURE		724	
PROPULSION SYSTEM		456	
PROPELLANT (INCL. TRAPPED)		2668	
REACTION CONTROL AND ULLAGE SYSTEM		1005	
EQUIPMENT (INCL. 165 LB. SCIENTIFIC GEAR)		437	
LANDING GEAR		740	
RETROGRADE MODULE			53787
STRUCTURE		2364	
PROPULSION SYSTEM		3289	
PROPELLANT (INCL. TRAPPED)		48020	
EQUIPMENT		114	
LAUNCH ESCAPE SYSTEM (L.E.S.)			2600
LANDING GEAR FAIRING			1400
GROSS WEIGHT AT LAUNCH			93837
LESS (1-% EFFECTIVE) LAUNCH EXPENDABLES			-3837
EFFECTIVE LAUNCH WEIGHT			90000

*LESS 85 POUNDS SCIENTIFIC EQUIPMENT TRANSFERRED TO COMMAND MODULE AT LUNAR SURFACE.

TABLE 2-2

SPACECRAFT MISSION HISTORY PHYSICAL CHARACTERISTICS (NORMAL MISSION)

CONFIGURATION	WEIGHT – POUNDS							CENTER OF GRAVITY(1)			MOMENTS OF INERTIA(2)	
	COMMAND MODULE	SERVICE MODULE EQUIP.	SERVICE MODULE	TERMINAL LANDING MODULE	RETRO MODULE	LES & FAIRINGS	TOTAL	X	Y	Z	ROLL	PITCH & YAW
ON PAD LAUNCH	6507	1148	22280	6115	53787	4000	93837	232.0	0	0	142,500	305,200
LESS: L.E.S. (1-% EFF.)						-2478	-2478					
FAIRING (1-% EFF.)						-1335	-1335					
ECS EXPENDABLES		-24					-24					
EFFECTIVE LAUNCH WEIGHT							90000					
LESS: L.E.S. (% EFF.)						-122	-122					
FAIRING (% EFF.)						-65	-65					
E.C.S. EXPENDABLES		-1					-1					
RCS FLUIDS				-53			-53					
SEPARATION DEVICES					-82		-82					
BEGIN TRANSLUNAR							89677	242.7	0	0	142,300	256,100
LESS: ULLAGE & MIDCOURSE CORRECTION				-178	-1992		-2170					
BEGIN RETROGRADE MANEUVER							87507	244.2	0	0	133,600	247,300
LESS: ULLAGE AND R.C.S. PROPELLANT				-358			-358					
RETROGRADE PROPELLANT					-19772		-19772					
LUNAR ORBIT WEIGHT							67377	264.3	0	0	92,200	191,500
LESS: ULLAGE AND RCS PROPELLANT				-80			-80					
TRANSFER AND RETRO PROPELLANT					-25776		-25776					
WEIGHT AT STAGING							41521	319.6	0	0	38,700	81,800
LESS: TRAPPED PROPELLANT					-480		-480					
PRESSURE GASES					-172		-172					
ECS EXPENDABLES		-16					-16					
INERT					-5513		-5513					
BEGIN TERMINAL PHASE							35340	348.7	0	0	28,300	31,400
LESS: PROPELLANT				-2642			-2642					
RCS				-69			-69					
LUNAR TOUCHDOWN							32629	354.7	0	0	25,800	26,900
LESS: INERT				-2650			-2650					
ECS EXPENDABLES		-147					-147					
EQUIPMENT TRANSFER	+85			-85			0					
LUNAR LAUNCH							29832	362.4	0	0	20,900	19,900
LESS: PROPELLANT-LAUNCH			-18957				-18957					
RCS AND ULLAGE			-16				-16					
BEGIN TRANSEARTH							10859	398.3	0	0	5,100	6,900
LESS: MIDCOURSE CORRECTION			-376				-376					
RCS			-103				-103					
WEIGHT AT STAGING							10380	400.8	0	0	4,700	6,400
LESS: INERT		-943	-2828				-3771					
ECS EXPENDABLES		-17					-17					
WEIGHT AT EARTH RE-ENTRY-PROPULSION CAPABILITY							6592	426.0	0	-4.5	1,600	2,000
LESS: SPACECRAFT WEIGHT MARGIN	-1438						-1438					
EARTH RE-ENTRY WEIGHT							5154	426.0	0	-5.8	1,200	1,500
LESS: ABLATED MATERIAL	-271						-271					
CHUTE COVER AND EXPENDABLES	-66						-66					
PARACHUTE DEPLOYMENT WEIGHT							4817	425.8	0	-6.2	1,200	1,400
LESS: CHUTES AND EXPENDABLES	-317						-317					
IMPACT WEIGHT	4500						4500	423.4	0	-6.0	1,100	1,200

(1) X STATION + 403 IS THEORETICAL INTERSECTION OF CONICAL SECTION AND FRONT FACE OF HEAT SHIELD, Y AXIS ECCENTRICITIES ARE NEGLIGIBLE, NEGATIVE Z IS TOWARD ASTRONAUTS FEET.

(2) MOMENT OF INERTIA IN SLUG – FEET SQUARED ABOUT THE C.G.

TABLE 2-3

SPACECRAFT MISSION WEIGHT HISTORY
LUNAR LANDING TOUCH AND GO ABORT

CONFIGURATION	WEIGHT – POUNDS						
	COMMAND MODULE	SERVICE MODULE EQUIP.	SERVICE MODULE	TERMINAL LANDING MODULE	RETRO MODULE	LEG & FAIRINGS	TOTAL
ON PAD LAUNCH	6507	1148	22280	6115	53787	4000	93837
LAUNCH THROUGH LUNAR LANDING							
SEE TABLE 2.6-1		-41		-3380	-53787	-4000	-61208
LUNAR TOUCHDOWN							32629
LESS: INERT				-2735			-2735
ECS EXPENDABLES		-					
EQUIPMENT TRANSFER	-			-			
LUNAR LAUNCH							29894
LESS: PROPELLANT-LAUNCH			-18996				-18996
RCS AND ULLAGE			-16				-16
BEGIN TRANSEARTH							10882
LESS: MIDCOURSE CORRECTION			-376				-376
RCS			-103				-103
WEIGHT AT STAGING							10403
LESS: INERT		-1090	-2789				-3879
ECS EXPENDABLES		-17					-17
WEIGHT AT EARTH RE-ENTRY – PROPULSION CAPABILITY							6507
LESS: SPACECRAFT WEIGHT MARGIN	-1438						-1438
EARTH RE-ENTRY WEIGHT							5069
LESS: ABLATED MATERIAL	-271						-271
CHUTE COVER AND EXPENDABLES	-66						-66
PARACHUTE DEPLOYMENT WEIGHT							4732
LESS: CHUTES AND EXPENDABLES	-317						-317
IMPACT WEIGHT	4415						4415

2.7 Design Criteria - The detail criteria established by the contractor for use in the design of the Two-Man Apollo Spacecraft is in general accordance with the technical guidelines of References (2-1) and (2-2). Where exceptions are taken, no significant compromise in the probability of safe return or mission completion is believed to be incurred. Emphasis is placed on designing to conservative criteria where specific requirements are not spelled out by the customer.

2.7.1 Mission Criteria - In establishing the spacecraft weights, propellant reserves are provided for all phases of the mission with a 10% ΔV margin for each phase as shown in Table 2-4 which is based on Reference (2-3). Provisions are made for a total of 200 hours mission time which includes a 24 hour contingency on the lunar surface and a 24 hour flight time contingency. In addition, a one week post landing recovery time is provided. Table 2-5 presents a breakdown of mission time apportionments.

TABLE 2-4

ΔV REQUIREMENTS

Translunar ΔV = 3883 ft/sec	
Midcourse	300
Retro into 80 na. mi. lunar orbit	3130
6° simultaneous plane change	100
10% allowance	353
Lunar Orbit to Lunar Surface ΔV = 7462 ft/sec	
Hohmann to 50,000 ft.	123
Descent to 1,000 ft. altitude	5961
Hover and translation	700
10% allowance	678
Lunar Launch ΔV = 6474 ft/sec	
Launch to 50,000 ft. circular orbit	5885
10% allowance	589
Transearth ΔV = 4281 ft/sec	
Transearth injection	3592
Midcourse	300
10% allowance	389

Total ΔV = 22,100 ft/sec

TABLE 2-5

MAXIMUM MISSION TIME APPORTIONMENTS

Launch to Parking Orbit	.18 hrs.
Parking Orbit Duration	1.5 hrs.
Parking Orbit to Injection	.10 hrs.
Translunar Coast	60.0 hrs.
Lunar Orbit Injection	.07 hrs.
Lunar Orbit Duration	3.0 hrs.
Lunar Landing	.14 hrs.
Lunar Stay Time (24 hr. contingency)	48.0 hrs.
Lunar Launch	.10 hrs.
Lunar Orbit Duration	3.0 hrs.
Transearth Injection	.04 hrs.
Transearth Coast (24 hr. contingency)	84.0 hrs.
Earth Re-entry and Landing	.9 hrs.
Post Landing Recovery (Habitable environment for 24 hrs. and survivable environment for additional 144 hrs.)	168.0 hrs.

2.7.2 Structural Design Criteria

A. Design factors for loads and heat effects

1. An ultimate factor of 1.5 applied to limit loads

2. A burst factor of 1.5 applied to all main fuel and oxidizer tanks and the reaction control system tanks in the terminal landing and service modules. A burst factor of 2.0 applied to all propellant pressurant tanks, the command module reaction control system tanks, and all other tanks not specifically excepted.

2.7.2 (Continued)

3. A 1.15 factor applied to predicted heating rates to establish design heating rates.
4. Design temperatures are based on design heating rates.
5. Combine ultimate loads and predicted temperatures or heating inputs.
6. Combine limit loads and design temperatures or heating inputs.
7. No primary structures require pressure stabilization.
8. Tank pressures are not used to relieve tank support loads.
9. Cabin burst pressure 15 psi ultimate.
10. No additional material provided for meteoroid protection.

B. Boost Phase

1. Maximum launch dynamic pressure of 820 psf at 10 degrees angle of attack.
2. Hard over gimbal failure of booster.
3. Tumbling abort at maximum dynamic pressure.
4. Sound pressure levels of 166 db in the frequency range 4 to 9600 cps emanating from the launch escape propulsion system during launch and abort.

C. Lunar Landing

1. Maximum contact velocities of 10 fps vertically and 5 fps laterally.
2. Maximum lunar surface slope of 15 degrees.
3. Maximum spacecraft tilt of 10 degrees from local vertical.
4. Surface friction coefficient from zero to infinity.
5. Vertical load factor of 4.5 earth g's for a four footed touchdown.
6. Bearing strength of lunar surface 200 psi.

D. Earth Re-entry

1. Maximum re-entry load factor of 20 g's.
2. Maximum heating and maximum heating rates combined with load factors at these conditions.

E. Earth Landing

1. Loads from main parachutes as determined by optimum reefing with any rational order of deployment.
2. Sink speed of 30 fps with two of three main parachutes at 5000 ft. altitude.
3. Wind velocity of 30 fps on land and 51 fps on water.
4. Crushable trussgrid system providing attenuation to 40 g combined with individual seat snubbers providing further attenuation to 20 g.
5. Equipment in cabin tied down for 80 g ultimate longitudinally and 40 g ultimate laterally.

2.7.3 Guidance and Navigation System Design Criteria

A. Complete on-board capability independent of earth tracking and communications.

B. Command and control decisions retained by crew.

C. System redundancy achieved by different systems capable of performing the same function permitting crew cross checking or selection, and repair capability of critical systems.

D. Earth tracking via DSIF with communications link to provide redundant data for cross checking and additional back-up of on-board system.

E. Guidance information display for abort at any time during mission.

2.7.4 Stabilization and Control System Design Criteria

A. Redundant manual control modes available at all times:

1. Direct command (hard wire to gimbal actuator or to RCS valves, either on-off or single pulse mode).

2.7.4 (Continued)

2. Rate command mode with redundant rate gyros.

B. Automatic modes available upon crew selection:

1. Tight control modes using sensed rate; dead bands and maneuver rates adjustable.

2. Low duty cycle derived rate modes for coast phases.

C. Standby inertial reference capable of driving displays.

D. Redundant gimbal actuators.

E. Redundant RCS jets where necessary to assure that no single jet failure results in loss of control along any axis.

F. Control from either right or left crew station.

2.7.5 Environmental Control System Design Criteria

A.	Metabolic	
	O_2 consumption	1.8 lb/man day
	CO_2 production	2.3 lb/man day
	Heat production	11300 BTU/man day
	Drinking H_2O required	12.5 lb/man day
B.	H_2O production	
	Perspiration and respiration	10.0 lb/man day
	Urine	3.0 lb/man day
	Feces	0.5 lb/man day
C.	Cabin leakage	.15 lb/hr
D.	Primary O_2 storage	Supercritical
E.	Secondary O_2 rate	.10 lb/man min
F.	Cabin atmosphere	O_2 at 5 psi (nom.)
G.	Cabin temperature	$70^{\circ}F$ to $80^{\circ}F$
H.	CO_2 removal	Li OH
I.	Odor removal	Activated charcoal

2.7.6 Electrical Power System Design Criteria

A. Six fuel cell stacks provided, any four of which can carry normal load and any three can carry the emergency load, in conjunction with the main batteries.

B. Two parallel sets of supercritical cryogenic hydrogen and oxygen reactant tanks are provided, thus allowing mission completion with reduced power in event of a single system failure.

C. Fuel cells (primary system)

1. Provisions for 900 watt load for 8 day mission
2. Nominal loads 800 watt average, 1620 watt peak
3. Emergency loads 1134 watts peak

D. Emergency power level assumed to be 70 percent of normal

E. Main batteries

1. Five 1250 watt-hour batteries provided
2. Nominal loads 1080 watts peak and 6527 watt-hours total energy
3. Emergency 756 watts peak

F. Squib-control batteries

1. Three 420 watt-hour batteries provided
2. Nominal loads 398 watt-hours total energy

G. Two redundant squib busses and one control bus, all powered from squib-control batteries, provided to isolate transient producing loads from the main bus.

H. External power supplied prior to launch

I. Fuel cell oxygen supply used as back-up for ECS oxygen

2.7.7 Telecommunications System Design Criteria

A. Provide voice communication, TV picture, and telemetry data transmission, crew intercommunication, tracking aids, rescue and recovery aids.

2.7.7 (Continued)

B. In flight maintenance capability

C. Redundancy of antenna system, voice signal flow paths, and equipment.

D. Conventional design

E. Continuous earth spacecraft communication, except from behind the moon, with circuit margin to provide for circuit degradation or unexpected propagation losses.

2.7.8 Heat Protection Design Criteria

A. Launch heating

1. Patrick AFB atmosphere

2. Transition Reynolds Number = 100,000

B. Spaceflight heating

1. Solar constant = 443 BTU/ft^2

2. Earth is 452°R black body

3. Earth albedo = 35%

C. Lunar surface heating

1. Lunar surface at sunset = 425°R

2. Lunar surface at midnight = 214°R

3. Lunar surface at noon = 705°R

4. Lunar albedo = 7%

D. Re-entry heating

1. 1959 ARDC atmosphere

2. Design re-entry for heat loads assumes:

initial flight path angle (@ 400,000 ft)	= -4.9°
initial velocity (@ 400,000 ft)	= 36,000 ft/sec
orbit inclination	= 90°
Angle of attack	= -30°

2.7.8 (Continued)

lift drag ratio	= 0.5
downrange (from 400,000 ft)	= 9000 na. mi.

3. M.A.C. Thermorad Shield S-3 ablative composite (used on Gemini)
4. A 1.15 factor on predicted heating rates to establish required ablative material

2.7.9 Service Module Propulsion Design Criteria

A. Earth storable hypergolic bipropellants

B. Pressure fed system using a cold He gas pressurization system

C. Nozzle cooled by ablation and radiation

D. Single engine

E. Thrust vector control accomplished by gimbaling engine

F. Thrust to weight ratio at lunar launch of 0.4

2.7.10 Terminal Landing Module Propulsion Design Criteria

A. Earth storable hypergolic bipropellants

B. Pressure fed system using a cold He gas pressurization system

C. Nozzle cooled by ablation

D. Single engine

E. Thrust vector control accomplished by gimbaling engine

F. Throttling ratio of 3.3 to 1.0

2.7.11 Retrograde Module Propulsion Design Criteria

A. Cryogenic O_2 and H_2 propellant

B. Pump fed system

C. Nozzle regeneratively cooled

D. Thrust to weight ratio at lunar descent of 0.4

E. Single engine

F. Thrust vector control accomplished by gimbaling engine

G. Throttling ratio of 10.0 to 1.0

2.7.12 Command Module Reaction Control System Design Criteria

A. Earth storable hypergolic propellant

B. Positive expulsion using a cold H_e gas pressurization system

C. Nozzles cooled by ablation

D. Eight 25 pound and four 50 pound thrusters

E. Two independent subsystems

F. System provides

1. Spacecraft orientation prior to re-entry

2. Pitch and yaw damping during re-entry

3. Roll maneuvers during re-entry for trajectory control

2.7.13 Service Module Reaction Control System Design Criteria

A. Earth storable hypergolic propellant

B. Positive expulsion using a cold H_e gas pressurization system

C. Nozzles cooled by ablation

D. Eight 25 pound and four 100 pound thrusters similar to Gemini

E. Two independent subsystems

F. System provides

1. Roll control during lunar launch

2. Pitch, yaw, and roll control during lunar orbit

3. Propellant positioning prior to transearth injection

4. Transearth midcourse correction

5. Transearth pitch, yaw, and roll control

2.7.14 Terminal Landing Module Reaction Control System Design Criteria

A. Earth storable hypergolic propellant

B. Positive expulsion using a cold H_e gas pressurization system

C. Nozzles cooled by ablation

D. Sixteen 100 pound thrusters similar to Gemini

2.7.14

E. Two independent subsystems

F. System provides

1. Pitch maneuver of the spacecraft and S-IV B booster stage in earth parking orbit
2. Limit cycle control of the spacecraft and S-IV B booster stage in earth parking orbit
3. Pitch, yaw, and roll maneuvers during translunar coast phase
4. Roll control during retrograde module engine operation
5. Minor midcourse correction
6. Roll control during lunar landing
7. Propellant positioning prior to retrograde module engine operation

2.7.15 Launch Escape System Design Criteria

A. A minimum peak altitude of 5000 feet and a minimum range at touchdown of 3000 feet following off-the-pad aborts.

B. Adequate separation under all flight conditions to avoid blast damage from an exploding booster.

C. Positive static stability under all expected Mach number conditions.

D. Emergency crew tolerances not to be exceeded.

2.7.16 Human Factors Criteria

A. Life support - atmospheric and temperature environment, accelerations, noise, and vibration controlled to within required limits

B. Man-machine integration

1. Give crew active management of all vehicle systems
2. Employ man to improve reliability and allow for flexibility and adaptability to new situations
3. Emphasize manual operation of spacecraft systems

2.7.16 (Continued)

4. Use automatic equipment for speed or precision or to relieve crew of tedious tasks

5. Design crew functions so that they may be performed reliably under all conditions

6. Provide for inflight and lunar surface maintenance

C. Human engineering

1. Design to accommodate 10 to 90 percentile man

2. Employ anthropometric data in design and arrangement of seats and all man operated equipment

3. Design couches for best attitude during various phases such as launch or re-entry, navigation, lunar landing and rest

4. Design controls for ease of operation and displays to present information in appropriate form

5. Establish acceptable levels of illumination, noise, vibration, and other factors affecting human performance

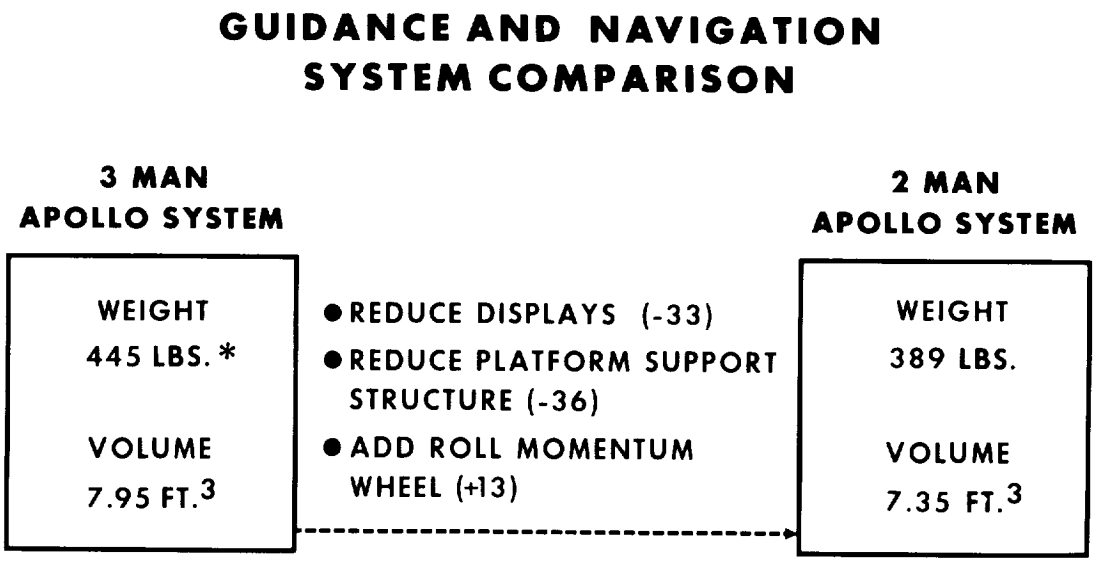
D. Radiation

1. No additional radiation protection is provided

3. SPACECRAFT SYSTEMS

3.1 Guidance and Navigation

3.1.1 Summary - The guidance and navigation system for the Two-Man Apollo Spacecraft is the system being developed by MIT for the Three-Man Apollo except for the changes indicated in Figure 3-1.



*INCLUDES TRACKING AND LANDING RADARS FROM LEM

FIGURE 3-1

Displays are reduced by 33 pounds because remote panels with duplicate displays are not necessary. The inertial platform and optical measurement unit are located nearly side by side, thereby allowing a 36 pound reduction in interconnecting support structure. A roll momentum wheel (13 pounds) is added to reduce roll rate during navigational sightings in the transearth phase from 0.04 degrees per second (provided by the reaction control system) to 0.01 degrees per second (determined by MIT as required for sextant operation). The material presented herein is based on data from References 3-1 through 3-16.

3.1.2 Functions - The following functions are performed with the guidance and navigation system:

3.1.2 (Continued)

A. Monitoring of earth launch, parking orbit, and translunar injection.

B. Determination of midcourse velocity corrections based on navigational measurements and guidance computations.

C. Guidance and navigation for lunar approach, orbit and landing, including computations and display information for periselene adjustments, retrograde into lunar orbit, orbital adjustment, and lunar landing.

D. Determination of lunar launch time, the attitude program for ascent, and computation of transearth injection conditions.

E. Measurement and computation to assure safe re-entry and for aerodynamic correction of the re-entry trajectory.

F. Provision of information for procedures to be followed in the event the mission must be aborted.

3.1.3 System Description - The guidance and navigation equipment consists of an inertial system for attitude memory and acceleration measurements, navigation sensors (optical, radar, and DSIF), a computer for computation of guidance and display signals for normal and abort trajectories, and electronic interface equipment required to drive appropriate displays. Weights for major components are presented in Figure 3-2. Weights are compared in detail with Three-Man Apollo weights in Figure 3-3. Power requirements are presented in Figure 3-4.

Inertial System - The inertial system consists of an inertial measurement unit (IMU) with characteristics presented in Figure 3-5, and the associated electronics including servo amplifiers and power supplies. It is in operation during earth launch (to permit crew monitoring of booster operation), lunar landing, lunar launch, and earth re-entry. It is normally off during midcourse to conserve life and power.

During each phase of the mission in which the inertial system is used, it can be shown that accelerations, velocities, and spacecraft headings all lie close to

GUIDANCE AND NAVIGATION SYSTEM WEIGHTS

SUBSYSTEM	WEIGHT-LBS.
INERTIAL SYSTEM	84
OPTICAL MEASUREMENT UNIT	35
COMPUTER	110
RADARS	89
VIEWER AND COUPLING DISPLAY UNIT	15
MISCELLANEOUS	56
TOTAL VOLUME = 7.35 FT.^{3}	TOTAL WEIGHT = 389 LBS.

FIGURE 3–2

GUIDANCE AND NAVIGATION DETAIL WEIGHT SUMMARY

COMPONENT	3-MAN APOLLO			2-MAN APOLLO		
	LOCATION	WT.-LB.	VOL.-FT.^{3}	LOCATION	WT.-LB.	VOL.-FT.^{3}
INERTIAL SYSTEM						
IMU	CM	60	1.00	CM	60	1.00
IMU ELECTRONICS	CM	24	0.60	CM	24	0.60
OPTICAL MEASUREMENT UNIT						
SEXTANT	CM	27	0.60	CM	27	0.60
TELESCOPE	CM	8	0.20	CM	8	0.20
COMPUTER SYSTEM						
COMPUTER	CM	100	1.60	CM	100	1.60
COMPUTER SPARES	CM	10	0.10	CM	10	0.10
RADARS						
TRACKING RADAR	LEM	55	1.00	TLM	55	1.00
ANTENNA-TRACKING RADAR	LEM	—	18" DIA.	TLM	—	18" DIA.
LANDING RADAR	LEM	17	1.00	TLM	17	1.00
ANTENNA-LANDING RADAR	LEM	17	14" DIA.	TLM	17	14" DIA.
DISPLAYS AND CONTROLS						
MAP AND VISUAL DISPLAY	CM	9	0.15	CM	9	0.15
COUPLING DISPLAY UNIT	CM	6	0.05	CM	6	0.05
DISPLAY AND CONTROL-FORWARD	CM	15	0.18	—	—	—
DISPLAY AND CONTROL-CENTER	CM	15	0.18	—	—	—
DISPLAY AND CONTROL-RIGHT	CM	3	0.04	—	—	—
MISCELLANEOUS						
NAVIGATION BASE	CM	46	0.40	CM	10	0.10
STRUCTURAL SUPPORTS	CM	8	0.10	CM	8	0.10
ELECTRICAL PROVISIONS	CM	18	0.60	CM	18	0.60
ENVIRONMENTAL CONTROL	CM	4	0.10	CM	4	0.10
SUN SENSOR	SM	3	0.05	SM	3	0.05
ROLL MOMENTUM WHEEL	—	—	—	CM	13	0.10
TOTALS		445	7.95		389	7.35

FIGURE 3–3

WATT-HOUR REQUIREMENTS - GUIDANCE AND NAVIGATION

EQUIPMENT	UNIT WATTS	LAUNCH (0.2)	INJECTION (2)	TRANSLUNAR (56)	MIDCOURSE CORRECTION (4)	LUNAR ORBIT (2)	LUNAR LANDING (1.3)	LUNAR REST (48)	LUNAR LAUNCH (4)	TRANSEARTH (80)	MIDCOURSE CORRECTION (4)	RE-ENTRY (0.5)	POST LANDING (24)
INERTIAL MEASUREMENT UNIT OPER. STBY.	138/ 44	27.6 (.2/0)	276 (2/0)	2460 (0/56)	176 (0/4)	276 (2/0)	179 (1.3/0)	–	552 (4/0)	3510 (0/80)	176 (0/4)	69 (.5/0)	–
IMU ELECTRONICS	122	24.4 (0.2)	244 (2)	–	488 (4)	244 (2)	159 (1.3)	–	488 (4)	–	488 (4)	61 (0.5)	–
OPTICAL MEASUREMENT UNIT	150	–	300 (2)	1650 (11)	600 (4)	300 (2)	–	–	300 (2)	1650 (11)	600 (4)	–	–
COMPUTER	47	9.4 (0.2)	94 (2)	–	188 (4)	94 (2)	67	–	188 (4)	–	188 (4)	24 (0.5)	–
TRACKING RADAR	60	–	–	–	–	120 (2)	78 (1.3)	–	–	–	–	–	–
LANDING RADAR	100	–	–	–	–	–	130 (1.3)	–	–	–	–	–	–
MAP AND VISUAL DISPLAY	50	–	100 (2)	550 (11)	200 (4)	100 (2)	–	–	100 (2)	550 (11)	200 (4)	–	–
COUPLING DISPLAY UNIT	78	15.6 (0.2)	156 (2)	–	312 (4)	156 (2)	100 (1.3)	–	312 (4)	–	312 (4)	39 (0.5)	–
ROLL MOMENTUM WHEEL	64	–	–	–	–	–	–	–	–	192 (3)	–	–	–

NOTE: FIGURES IN PARENTHESES ARE HOURS, EITHER FOR PHASE OR FOR OPERATION OF EQUIPMENT

FIGURE 3-4

APOLLO INERTIAL MEASUREMENT UNIT

THREE GIMBALS:
1) OUTER GIMBAL – ROLL: 360°
2) MIDDLE GIMBAL – YAW: ± 65°
3) INNER GIMBAL – PITCH: 360°

PICKOFFS:
1) TYPE: TWO-SPEED SYNCHRO SYSTEM
2) ACCURACY: 15 ARC SECOND R.M.S.

INITIAL ALIGNMENT ACCURACY:
1) OVER-ALL: 1 TO 2 ARC MINUTES
2) ASSUMES: 1) SYNCHRO ACCURACY = 15 ARC SECONDS
2) SEXTANT ACCURACY RELATIVE TO STARS = 10 ARC SECONDS
3) STRUCTURE (OMU TO IMU) = 30 ARC SECONDS

THREE GYROS:
1) TYPE: 25 IRIG (INERTIAL RATE INTEGRATING GYRO)
2) BIAS DRIFT: 0.15 DEG./HOUR
3) ACCELERATION-SENSITIVE DRIFT DUE TO ACCELERATION ALONG INPUT AXIS: 0.15 DEG./HOUR/G
4) ACCELERATION SQUARED-SENSITIVE DRIFT TERM ABOUT THE INPUT AXIS AND SPIN REFERENCE AXIS: 0.15 DEG./HOUR/G^2

THREE ACCELEROMETERS:
1) TYPE: 16 PIP (PULSE INTEGRATING PENDULUM) MADE BY SPERRY
2) SCALE FACTOR ERROR: 100 PPM
3) ACCELERATION SENSITIVE ERROR: 1 PPM/G

TEMPERATURE CONTROL:
1) OPERATING: 55°F
2) STANDBY: 55°F TO 200°F

FIGURE 3-5

3.1.3 (Continued)

some particular plane. The orientation of this plane is a characteristic of the particular phase. The IMU platform is aligned prior to each use of the inertial system. The IMU inner axis is oriented along the pole of the particular plane in which that flight phase is expected to occur.

Optical Measurement Unit (OMU) - The OMU permits the astronauts to make manual navigational measurements onboard the spacecraft. It consists of a scanning telescope and sextant with characteristics shown in Figure 3-6. The telescope is

APOLLO OPTICAL MEASUREMENT UNIT

SCANNING TELESCOPE (SCT)

1) SHAFT AXIS ROTATION: 360°
2) TRUNION AXIS ROTATION: −90° TO +90°
3) MAGNIFICATION (CONTINUOUSLY VARIABLE): 1X TO 3X
4) EFFECTIVE APERTURE: 0.2" TO 0.4"
5) FIELD: 60° TO 20°

SEXTANT (SXT)

1) APERTURE: 1.6"
2) FIELD: 1.9 DEGREES
3) MAGNIFICATION: 28X
4) TRANSMISSION OF STAR SIGHT LINE THRU BEAM SPLITTER: 80%
5) TRANSMISSION OF PLANET SIGHT LINE THRU BEAM SPLITTER: 10%
6) STAR PRISM ANGULAR FREEDOM ABOUT TRUNION AXIS: +20° TO −70°
7) LANDMARK PRISM ANGULAR FREEDOM ABOUT TRUNION AXIS: +70° TO −20°
8) SHAFT AXIS ANGULAR FREEDOM: 360°
9) ACCURACY OF PICKOFF MEASURING ANGLE BETWEEN STAR AND LANDMARK PRISMS: 10 ARC SECONDS OR BETTER.

FIGURE 3-6

used for acquisition of a target: star, planet edge, planet landmark, or landing site. It has a single line-of-sight that can be redirected with two degrees of freedom. The sextant is used to measure star elevation angles and to align the IMU.

The most accurate method for onboard midcourse navigation is measurement of angles between stars and landmarks on the earth and moon. A second less accurate method uses the angles between stars and the closest points on the planet's horizon

3.1.3 (Continued)

(i.e., the edge of the moon or a specific scattering brightness in the earth's illuminated atmosphere). Where the earth is used as a reference, this method must be employed if there is unusual cloud coverage. The expected star-planet elevation angle is set on the sextant 5 to 10 minutes ahead in time. The astronaut pushes a button when the images coincide and the angle is inserted into the computer. Several such measurements provide the data for the determination of the position of the spacecraft during midcourse flight.

Considering the tracking capabilities of the eye and the magnification of the optics, angular rates of the spacecraft must be less than 0.01 degrees per second during sightings. Improvements in the sextant design may eventually involve addition of an automatic tracking capability to the star set of optics. Hence, the star image could be automatically stabilized regardless of spacecraft rates so long as the star remained in the sextant field of view. In addition, a rate-integrating gyro could be used as a reference to maintain stability about the sextant shaft axis.

Computer (Apollo Guidance Computer - AGC) - The computer is provided to monitor boost-orbit-injection trajectories, and to determine trajectory corrections and abort trajectories. Inputs to the computer are from accelerometers, platform and sextant angle transducers, the radars, and the crew via the manual keyboard. Outputs from the computer are displayed to the crew for trajectory and attitude control. Outputs are fed to the telemetry system for transmission to the DSIF. Characteristics of the Apollo computer are summarized in Figure 3-7.

Radars (RAD) - A tracking radar and a doppler radar are provided to make navigational measurements prior to and during lunar landing. The tracking radar measures altitude relative to the lunar surface during lunar orbit and measures range and attitude relative to a transponder during lunar descent. The doppler radar measures

3.1.3 (Continued)

APOLLO COMPUTER (RAYTHEON)

TYPE:
1) ORGANIZATION: GENERAL PURPOSE, BINARY, SERIAL WITH PARALLEL WORD TRANSFER, FIXED-POINT ARITHMETIC
2) CONSTRUCTION: 22 WELDED, ENCAPSULATED, REPLACEABLE STICK MODULES.

MEMORY:
1) TYPE: RANDOM ACCESS, NON-DESTRUCTIVE READ-OUT
2) VOLATILE MEMORY: 1024–16 BIT WORDS IN ERASEABLE FERRITE CORE MEMORY
3) WIRED MEMORY: 10,240–16 BIT WORDS IN WIRED ROPE MEMORIES
4) TOTAL: 11,264–16 BIT WORDS

WORD MAKEUP (16 BITS):
1) DATA: 1 FOR SIGN, 14 FOR DATA, 1 FOR PARITY
2) INSTRUCTION: 3 FOR OPERATING CODE, 12 FOR ADDRESS, 1 FOR PARITY

ARITHMETIC TIMES:
1) ADD, SUBTRACT, (DOUBLE PRECISION): 400 MICROSECONDS
2) MULTIPLICATION (DOUBLE PRECISION): 2000 MICROSECONDS
3) WORD TRANSFER: 5 MICROSECONDS

CLOCK RATES:
1) BIT RATE: 200 KC

INPUTS:
1) DISCRETES: 90
2) COUNTERS: 20
3) INTERRUPT: 20

OUTPUTS:
1) DISCRETES: 60
2) COUNTERS: 20

FIGURE 3–7

altitude and three components of velocity during lunar landing. Characteristics of the Apollo radars are summarized in Figure 3-8.

Map and Visual Display (MVD) - A film viewer is provided for display of maps, charts, procedures, etc. It is located on the display panel used for navigation.

Coupling Display Unit (CDU) - The coupling display unit is provided to couple the attitude signals received from the IMU with the attitude commands from the computer, such that error signals are determined for transmission to the autopilot. It consists of three servo driven resolver chains.

Roll Momentum Wheel - The roll momentum wheel is provided to reduce roll rates during navigational sightings on the transearth phase of the mission from 0.04 degrees per second (provided by the reaction control system) to below 0.01 degrees per second (determined by MIT as required for sextant operation).

APOLLO RADARS

TRACKING RADAR:

1) FREQUENCY: X-BAND
2) OPERATION: PULSE (NON-COHERENT) CONICAL SCANNING
3) ANTENNA: 18" TRACKING, 2 GIMBALS AND 2 GYROS
4) VOLUME: 1 CUBIC FOOT
5) WEIGHT: 55 POUNDS
6) POWER: 60 WATTS
7) PEAK RF POWER: LUNAR ORBIT: 10 KW
   LUNAR DESCENT-NON COOPERATIVE: 5 KW
   LUNAR DESCENT-COOPERATIVE: 250 W
8) ALTITUDE: 86 N.M.
9) ALTITUDE ACCURACY: ± 0.1%
10) RANGE (SLANT TO LANDING SITE): COOPERATIVE: 54 N.M.
    NON-COOPERATIVE: 17 N.M.
11) RANGE ACCURACY: ± 1%
12) ANGLE: ± 30° AFT FROM THE SPACECRAFT AXIS OF SYMMETRY TO THE LANDING SITE BY IMU WITH ACCURACY OF ± 1°.
13) RANGE RATE: 2300 FT./SEC. TO 328 FT./SEC. DETERMINED BY DIFFERENTIATION OF MEASURED RANGE USING SMOOTHING TIME OF SEVERAL SECONDS.

TRANSPONDER:

1) LOCATION: LUNAR LANDING SITE
2) ANTENNA: 180° BEAM WIDTH
3) SIZE: 0.5 CUBIC FEET
4) WEIGHT: 11 LB.
5) POWER: 20 W
6) RF POWER: 5 W

DOPPLER RADAR:

1) FREQUENCY: X-BAND
2) OPERATION: DOPPLER WITH ALTITUDE CAPABILITY (ICW)
3) ANTENNA: 14" DIELECTRIC LENS, 3 FEEDS FIXED TO SPACECRAFT
4) VOLUME: 1 CUBIC FOOT
5) WEIGHT: 34 LB. (17 LB. ANTENNA INCLUDED)
6) POWER: 100 W
7) RF POWER: 1 W
8) MEASUREMENT: ALTITUDE AND THREE COMPONENTS OF VELOCITY DURING LUNAR LANDING.
9) ALTITUDE: 2300 TO 23 FEET
10) ALTITUDE ACCURACY: ± 5 FEET
11) VERTICAL VELOCITY: 165 FT./SEC. TO 3 FT./SEC.
12) HORIZONTAL VELOCITY: 490 FT./SEC. TO 3 FT./SEC.
13) VERTICAL VELOCITY ACCURACY: 1 FT./SEC.

FIGURE 3-8

3.1.4 System Operation - The chart below indicates the use of the navigation sensors and the inertial system through the various mission phases:

PHASE	PRIMARY	SECONDARY
Midcourse	DSIF	Sextant
Lunar Orbit	Sextant	DSIF
Lunar Landing	Radars* - Inertial	Sextant* - Inertial
Lunar Launch	Sextant* - Inertial	
Earth Re-entry	DSIF* - Inertial	Sextant* - Inertial

*Provide Initial Conditions and Updating

In the event of a failure of the IMU, the strapdown IMU (included as part of the stabilization and control system) is used. Operation with the strapdown IMU is summarized in Figure 3-9.

OPERATION WITH STRAPDOWN IMU
(FAILURE OF PRIMARY IMU)

PHASE	OPERATION
LUNAR LANDING	ABORT TO LUNAR ORBIT
LUNAR LAUNCH	BOOST INTO HIGHER-ALTITUDE LUNAR ORBIT USE DSIF INFORMATION FOR TRANSEARTH INJECTION
EARTH LANDING	RE-ENTER USING "g" -PROGRAM (ACCEPT LARGE LANDING DISPERSIONS)

NOTES:

1) STRAPDOWN IMU CONSISTS OF 3 STRAPPED-DOWN GYROS AND ONE LONGITUDINAL ACCELEROMETER.

FIGURE 3-9

3.1.4 (Continued)

During midcourse, DSIF data are more accurate than data provided by the onboard sextant, assuming that the onboard system follows Battin's method (Reference 3-14) with 40 translunar sightings being taken using a sextant with an instrument accuracy of 9 arc seconds. Hence, the use of DSIF data, when available, provides a means to conserve fuel used for midcourse corrections. Accuracies for the DSIF and onboard system are plotted in Figure 3-10 (based on References 3-14 and 3-16).

NAVIGATION ACCURACY
DSIF VS. SEXTANT

ASSUMES THAT ABILITY TO MEASURE MIDCOURSE ΔV'S IS PERFECT

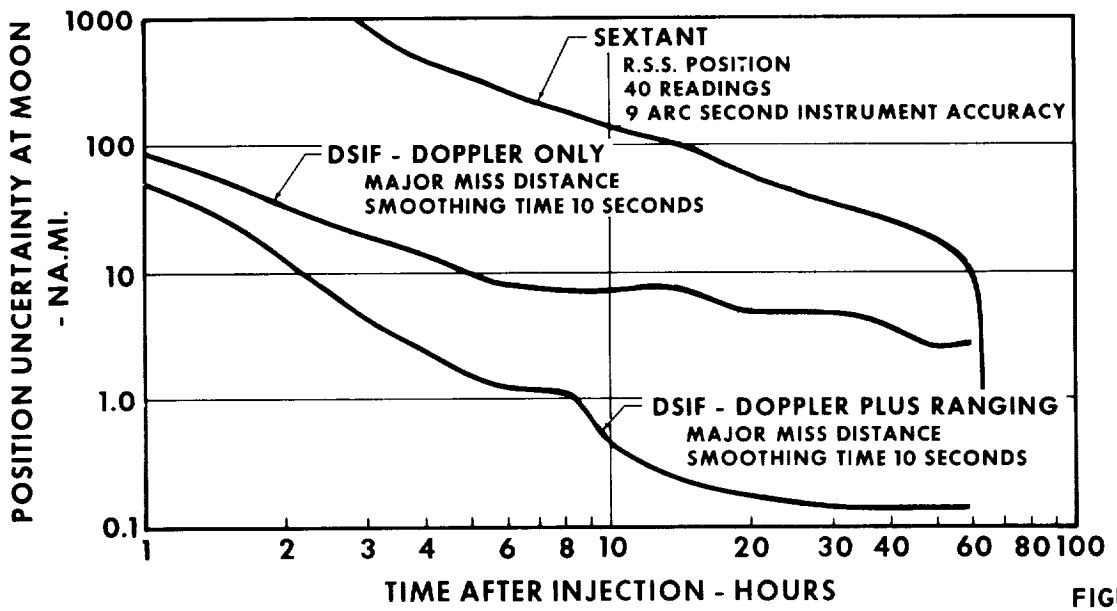


FIGURE 3-10

The criteria used by Battin to determine the time for an observation is called the "Miss Distance Reduction Ratio". For an observation to be made, a significant reduction in the potential miss distance at the moon must result. Values of 0.1 and 0.5 for Miss Distance Reduction Ratio are assumed to determine the number of midcourse observations that are made. The criteria is used prior to earth launch to establish a program listing angles and times when observations are worthwhile. In accordance with this program, the spacecraft is reoriented first in roll and second (if necessary)

3.1.4 (Continued)

in pitch to place the proper star and planetary body in the field of view of the optical measurement unit. After the observation, the spacecraft is returned to its nominal attitude (nose to sun; Canopus-oriented in roll).

The criterion used to determine the time for a velocity correction is called the "Velocity Correction Uncertainty Ratio". Values of 0.2 to 0.3 are assumed to determine the times at which velocity corrections should be made. If a correction is worthwhile, the spacecraft is reoriented first in roll and second in pitch using the strapdown inertial system as the attitude reference. The integrated output of a longitudinal accelerometer is used to control the magnitude of the ΔV that is applied.

In lunar orbit, navigation using the sextant involves measurement of the angle to a known landmark relative to the onboard inertial system. Measurements of star occultation times and star elevation angles provide additional data for navigational computations. Altitude information is provided by the tracking radar. Selection of the landing site with the telescope (or tracking radar if a transponder is used) provides data for computation of retrograde time for landing. If no transponder is located at the landing site, the inertial system (corrected by the doppler radar at low altitude) is used for navigation during the descent.

If a transponder is used, the guidance system provides steering and thrust control to place the spacecraft at a point approximately 6000 feet altitude and near the transponder with zero velocity. Propellant consumption is minimized in following the trajectory calculated by the computer. Measurements that are required to compute the trajectory are:

A. Range to transponder (from Tracking Radar)

B. Angle between horizontal and line-of-sight to transponder (from horizon scanner and tracking radar antenna)

3.1.4 (Continued)

C. 3 velocity components (from doppler radar referenced to inertial attitude reference)

The geometry, equations, and mechanization are discussed in detail in the guidance and navigation section of Volume III.

Prior to lunar launch, sextant measurements are used to compute position and align the platform. The lunar launch consists of a vertical flight phase during which the yaw axis of the vehicle is rolled into the desired orbital plane followed by a pitch program into lunar orbit. During lunar orbit, navigational sightings are taken to establish the time for initiation of injection into the transearth trajectory.

Prior to re-entry, initial conditions (position and velocity components) are transmitted to the computer from the DSIF and the sextant is used to align the platform. During re-entry, aerodynamic correction of the trajectory is achieved by commanding the roll orientation of the lift vector. This orientation is computed from inertial measurements and touchdown point prediction equations.

3.2 Stabilization and Control

3.2.1 Summary - The stabilization and control system (SCS) for the Two-Man Apollo Spacecraft is similar to the system being developed by Minneapolis Honeywell for the Three-Man Apollo (References 3-17 and 3-18). Changes which affect weight are indicated in Figure 3-11. Deletion of control modes associated with rendezvous reduces the control electronics unit by 10 pounds. Two Canopus sensors (10 pounds) are added to provide a backup mode for alignment of the IMU or IMU strapdown in the event of failure of the sextant. The displays are reduced by 6 pounds since the Two-Man Apollo does not require remote panels with duplicate displays.

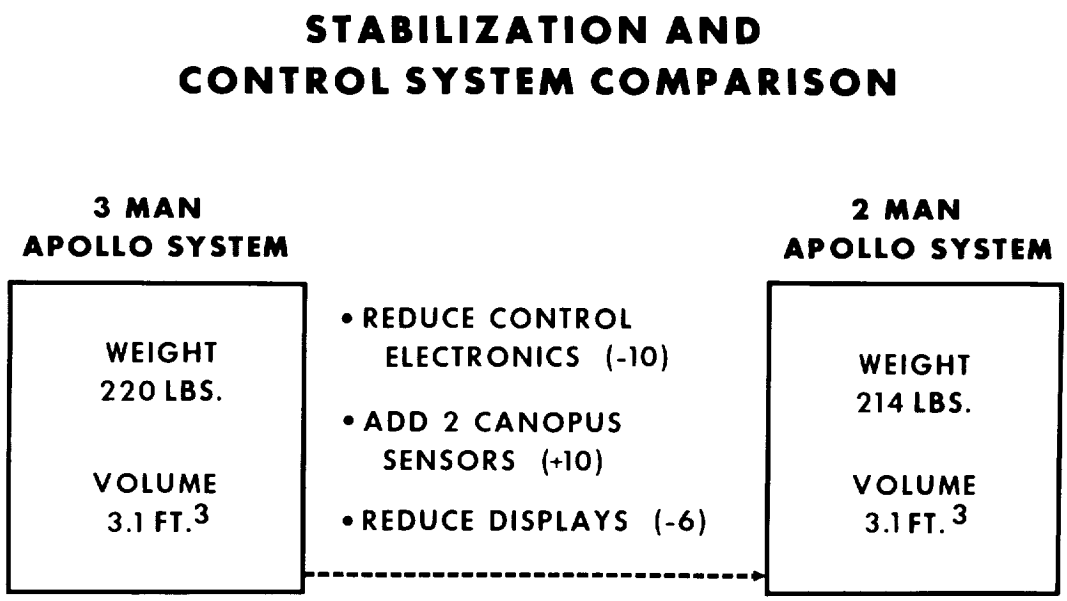


FIGURE 3-11

3.2.2 Operational Functions - The SCS, in conjunction with the Reaction Control System (RCS) and the gimbaled main engines, controls the attitude of the spacecraft and the thrust of the throttlable engines. Functional modes of operation are as follows:

A. Direct command; available at all times as a high-reliability backup mode for both angular and linear acceleration commands.

3.2.2 (Continued)

B. Manual angular rate command; primary manual attitude control mode

C. Sensed rate automatic control; provides precise, well-damped attitude response to commands from guidance and navigation system (GNS) and tight attitude hold against disturbance torques

D. Derived rate automatic control; provides low duty cycle attitude hold for crew relief during long coast phases

E. Automatic linear acceleration control; varies thrust according to command from GNS during lunar landing maneuver

More details on the features of the various modes are given in Section 3.2.4.

3.2.3 System Description - The SCS consists of an IMU strapdown unit for short term attitude memory and longitudinal acceleration measurement, rate sensors, a control electronics unit for mechanization of control logic, nongyroscopic attitude sensors (Sun-Horizon-Canopus) for long-term attitude reference, and a display modular package to condition control signals to drive the displays. Selection of control modes is done manually by switches on the control panel. Weights for each of these subsystems are listed on Table 3-1 and compared in detail with weights for the Three-Man Apollo system on Table 3-2. Power requirements are listed on Table 3-3.

A. IMU Strapdown - This unit serves as an attitude memory during vehicle orientation maneuvers required for navigation sightings and for midcourse velocity corrections. It measures the longitudinal accelerations during the midcourse correction thrusting and provides the necessary cut-off signal. Thus the primary IMU in the GNS is not used during the long transit phases. The strapdown IMU serves as a backup to provide an alternate method of accomplishing the large ΔV maneuvers during which the primary IMU is normally operating. It consists of three body-mounted rate-integrating gyros and a longitudinal accelerometer. Reference orientation is

TABLE 3–1

STABILIZATION AND CONTROL SYSTEM WEIGHTS

SUBSYSTEM	WEIGHT-LBS.
IMU STRAPDOWN	17
RATE SENSORS	13
CONTROL ELECTRONICS	122
SUN-HORIZON-CANOPUS SENSORS	12
DISPLAY MODULAR PACKAGE	34
MISCELLANEOUS	16
TOTAL VOLUME = 3.1 FT.^{3}	TOTAL WEIGHT = 214 LBS.

TABLE 3–2

STABILIZATION AND CONTROL DETAIL WEIGHT SUMMARY

COMPONENT	3-MAN APOLLO			2-MAN APOLLO		
	LOCATION	WT.-LB.	VOL.-FT.^{3}	LOCATION	WT.-LB.	VOL.-FT.^{3}
IMU STRAPDOWN						
INERTIAL REFERENCE	CM	12	0.30	CM	12	0.30
LONGITUDINAL ACCELEROMETER	CM	5	0.03	CM	5	0.03
RATE SENSORS						
RATE GYRO PACKAGE	CM	11	0.10	CM	11	0.10
RATE GYRO SPARE	CM	2	0.05	CM	2	0.05
CONTROL ELECTRONICS						
CONTROL ELECTRONICS UNIT	CM	90	1.10	CM	80	1.00
CONTROL ELECTRONICS SPARE	CM	12	0.10	CM	12	0.10
POWER SUPPLY	CM	15	0.20	CM	15	0.20
POWER SUPPLY SPARE	CM	15	0.20	CM	15	0.20
SUN-HORIZON -CANOPUS SENSORS						
SUN SENSOR (2)	CM	2	0.05	CM	2	0.05
HORIZON SENSOR (2)	(REF. UNDER DISPLAYS)			(REF. UNDER DISPLAYS)		
CANOPUS SENSOR (2)	—	—	—	SM	10	0.20
DISPLAYS AND CONTROLS						
DISPLAY MODULAR PACKAGE	CM	34	0.40	CM	34	0.40
SELECT PANEL	CM	6	0.10	—	—	—
MISCELLANEOUS						
STRUCTURE	CM	4	0.03	CM	4	0.03
ELECTRICAL PROVISIONS	CM	10	0.40	CM	10	0.40
ENVIRONMENTAL CONTROL	CM	2	0.04	CM	2	0.04
TOTALS		**220**	**3.10**		**214**	**3.10**

3.2.3 (Continued)

STABILIZATION AND CONTROL SYSTEM POWER REQUIREMENTS

EQUIPMENT	UNIT WATTS	LAUNCH (0.2)	INJECTION (2)	TRANSLUNAR (56)	MIDCOURSE CORRECTION (4)	LUNAR ORBIT (2)	LUNAR LANDING (1.3)	LUNAR REST (48)	LAUNCH LUNAR (4)	TRANSEARTH (80)	MIDCOURSE CORRECTION (4)	RE-ENTRY (0.5)	POST LANDING (24)
IMU STRAPDOWN	9	4.5 (0.2)	18 (2)	504 (56)	36 (4)	18 (2)	12 (1.3)	–	36 (4)	72 (80)	36 (4)	4 (0.5)	–
RATE SENSORS	36	7.2 (0.2)	72 (2)	2010 (56)	144 (4)	72 (2)	46.8 (1.3)	–	144 (4)	2880 (80)	144 (4)	18 (0.5)	–
CONTROL ELECTRONICS	10	2 (0.2)	20 (2)	560 (56)	40 (4)	20 (2)	13 (1.3)	–	40 (4)	800 (80)	40 (4)	5 (0.5)	–
POWER SUPPLIES	50	10 (0.2)	100 (2)	2800 (56)	200 (4)	100 (2)	65 (1.3)	–	200 (4)	4000 (80)	200 (4)	25 (0.5)	–
OPTICAL SENSORS/ELECTRONICS	24	4.8 (0.2)	48 (2)	1345 (56)	96 (4)	48 (2)	31 (1.3)	–	96 (4)	1920 (80)	96 (4)	12 (0.5)	–
DISPLAY MODULAR PACKAGE	100	20 (0.2)	200 (2)	1100 (11)	400 (4)	200 (2)	130 (1.3)	–	400 (4)	1100 (11)	400 (4)	50 (0.5)	–

NOTE: FIGURES IN PARENTHESES ARE HOURS, EITHER FOR PHASE OR FOR OPERATION OF EQUIPMENT

established from the nongyroscopic attitude sensors. Orientation changes based on the strapdown IMU short-term attitude memory are normally accomplished in a roll-pitch, rather than yaw-pitch, sequence of gyro torquing in order to minimize fuel consumption.

B. Rate Sensors - Three rate gyros provide the basic follow-up signal for manual rate command modes and the damping signal for the sensed rate automatic control modes. The rate gyros are energized throughout the mission, even though their signals are not used in the coast phase modes, so that the modes requiring rate signals are always available without any warmup delay. A spare rate gyro is provided for replacement if a failure should occur during the mission.

C. Control Electronics - This unit provides the circuitry to sum, shape, and switch sensor and command input signals to control the operation of the

3.2.3 (Continued)

RCS jets, engine gimbal actuators, and engine throttling. There are three channels, one for each axis, containing the signal limiters, voltage amplifiers which also act as summing points, gain selection circuits, switching amplifiers for driving RCS solenoid valves, jet selector switches, etc. necessary for the various control modes. Section 3.2.4 describes these modes in more detail. Other functions of the electronics unit include provisions for signal conditioning during in-flight checkout, monitoring, and telemetry, and power switching for various functions initiated by discrete commands from the crew or the GNS computer.

D. Nongyroscopic Attitude Sensors - Long term attitude reference is provided by sun, horizon, and canopus sensors. Each of these sensors is duplicated for reliability. During translunar and transearth phases, the sun sensor provides the basic signals for orienting the spacecraft roll axis in space, and the canopus sensor is the reference for vehicle orientation about this axis. In lunar orbit, the horizon sensor orients the spacecraft yaw axis, and gyrocompassing supplies the third axis signal. The canopus sensors can also be used for IMU alignment in the event of failure of the GNS sextant. An alternate roll reference signal is available from pickoffs on the gimbals of the DSIF antennas, which are directed toward earth by a closed loop control around an earth tracker mounted on the antenna.

E. Display Modular Package - This unit performs the signal conditioning, power amplification, and impedance matching necessary to drive stabilization and control displays.

3.2.4 Control Modes

A. Direct Command - Motion of the pilot's hand controller closes switches, which send power directly to the RCS jet valves, or supplies proportional signals

3.2.4 (Continued)

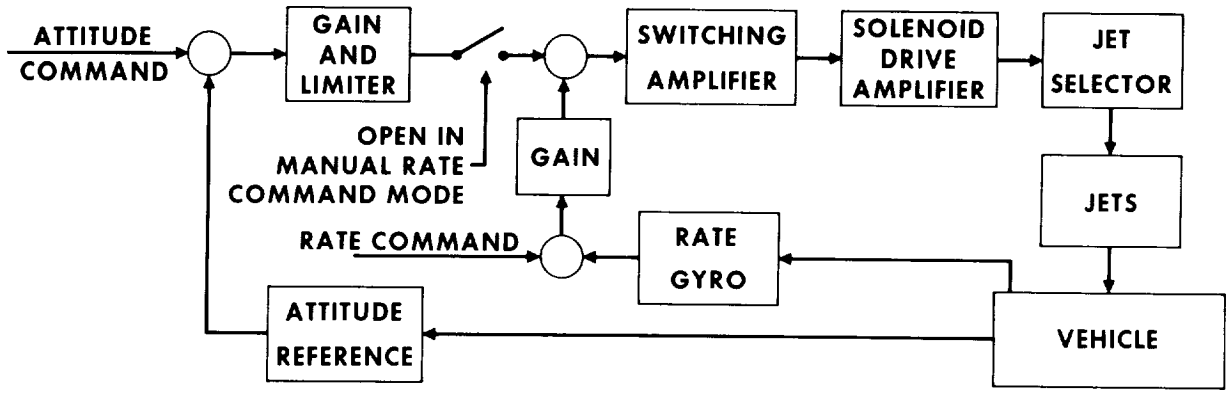
to the main engine gimbal or throttle actuators. No switching logic rate gyro signals or other artificial stabilizing loops are involved. The RCS signal can be either on-off or discrete minimum width pulses. To command a pulse, the pilot moves the controller in the desired direction and depresses a trigger switch located on the controller. Repeated actuation of the trigger gives repeated pulses. Use of the pulse mode enables the pilot to exercise fine control over vehicle rate, thereby minimizing fuel expenditure and allowing more accurate navigational sightings.

B. Rate Command - The preferred mode for manual control of the spacecraft attitude is the rate command mode in which signals proportional to hand controller deflection are combined, in the electronics unit, with rate gyro signals and the difference is sent through amplifiers to the appropriate gimbal actuator or RCS jet. Torque is applied until the rate gyro signal matches the commanded rate signal from the controller. In the gimbaled engine modes, the torque is proportional to the rate error, and hence, the vehicle rate response approaches the rate command exponentially. The constant torque applied in RCS modes drives the rate linearly until it differs from the commanded value by less than the deadband setting of the switching amplifier.

C. Sensed Rate Automatic Attitude Control - In this mode, attitude commands from the GNS are compared with attitude reference signals, and the difference is passed through a limiter in the electronics unit. This limited difference represents a rate command, and causes the same type of control action as manual rate commands. Figure 3-12 shows a simple block diagram of the resulting control loop with RCS and illustrates the corresponding phase plane switching boundaries. As the various staging operations occur, the

3.2.4 (Continued)

SCHEMATIC FOR SENSED RATE RCS CONTROL MODES



PHASE PLANE CONTROL BOUNDARIES

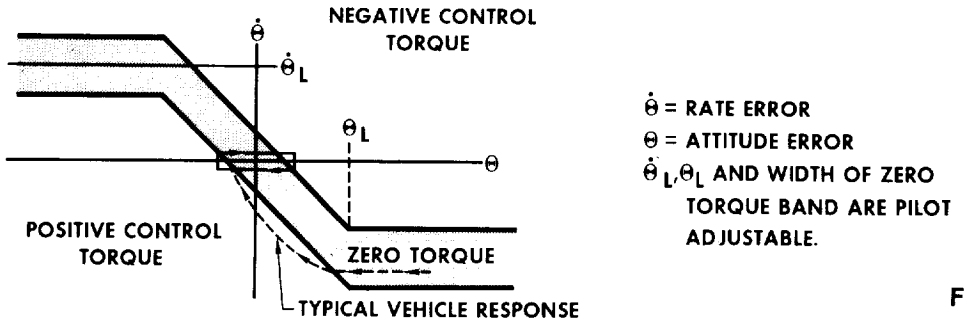


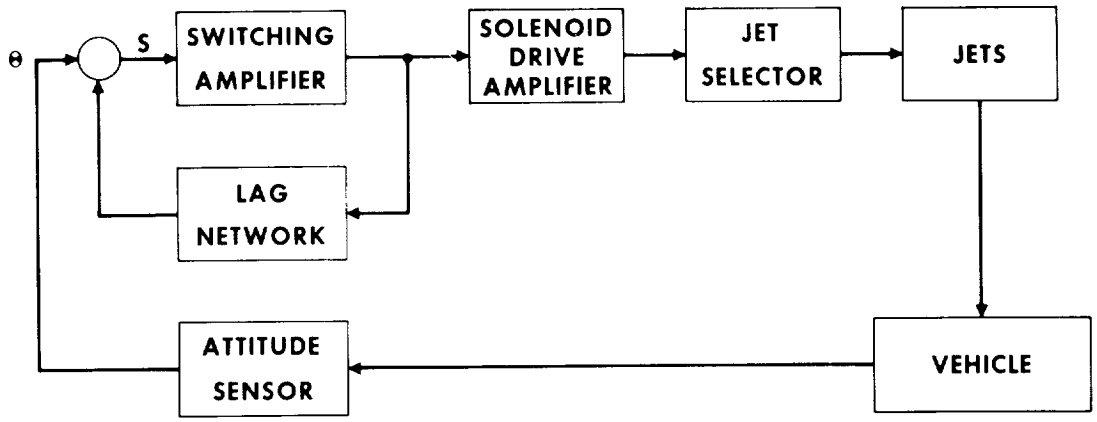
FIGURE 3-12

slopes and break points of these boundaries are adjusted, by altering the gains, to produce near-optimum response for the appropriate torque-to-inertia ratio. With gimbaled engines, the control loop is the same with the switching amplifier and solenoid valve driver being replaced by a power amplifier and gimbal actuator.

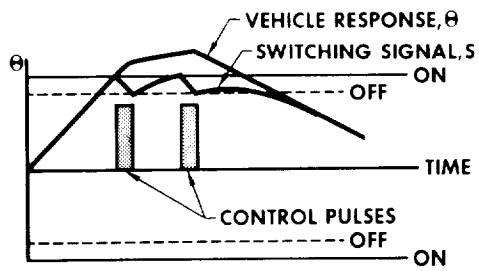
D. Derived Rate Automatic Attitude Control - Figure 3-13 illustrates the principle of this mode, which is utilized for long-term attitude hold. The switching amplifier output is essentially in phase with control torque and resulting acceleration, since disturbing torques are small in comparison. The switching amplifier signal is conditioned by a long-time-constant lag network producing a signal which approximates the integral of acceleration, or vehicle rate change. This signal is used to produce the necessary damping in lieu of an ordinary sensed rate signal. The advantage of the

3.2.4 (Continued)

CONTROL LOGIC FOR DERIVED RATE MODES



CONTROLLED RESPONSE



S IS COMBINATION OF θ AND LAGGED TORQUE SIGNAL.
WHEN S REACHES "ON" LEVEL, LATCHING CIRCUIT HOLDS TORQUE COMMAND ON FOR MIN. PULSE TIME.
SWITCHING LEVEL, GAIN, AND LAG TIME CONSTANT ARE PILOT ADJUSTABLE

FIGURE 3-13

derived rate scheme is simplicity and freedom from adverse effects at low signal levels. The no-disturbance limit cycle will always converge to one pulse at each boundary. For large initial rates the scheme produces repeating pulses which eventually brings the spacecraft attitude within the deadband at a reduced rate.

E. Automatic Linear Acceleration Control - The portion of the control electronics unit associated with this mode commands the throttle actuator to drive in the proper direction until measured acceleration equals that commanded by the GNS.

3.2.5 Control Torque Devices - The criteria for selecting RCS jet sizes are shown on Table 3-4. These criteria were arrived at by somewhat arbitrary engineering judgment and do not necessarily represent stringent requirements. One exception is the command module roll acceleration requirement of 10 deg/sec.^{2}, Reference 3-19.

3.2.5 (Continued)

TABLE 3-4

RCS JET SIZE DESIGN CRITERIA

- MIN. ANG. ACCELERATION, DEG./SEC.^{2}

	ROLL	PITCH	YAW
TRANSLUNAR	1.0	0.2	0.2
TRANSEARTH	2.0	1.0	1.0
RE-ENTRY	10.0	5.0	5.0

- ROLL DURING HOVER: 180 DEG. IN 15 SEC.
- ROLL AT LUNAR TAKE-OFF: 90 DEG. BEFORE 1000 FT. ALT.
- MAX. RATE CHANGE FOR SINGLE PULSE IN TRANSIT PHASES: .1 DEG./SEC.
- LINEAR ACCELERATION FOR MIDCOURSE ΔV, FT./SEC.^{2}: 1.0 MIN. ; 5.0 MAX.
- NO VARIABLE OR DUAL-LEVEL RCS JETS

This is met with ample margin, as shown by the estimated available control accelerations listed on Table 3-5. With SCS gains chosen to give a time constant of 1 second, the gimbaled engine pitch and yaw angular accelerations during the hover phase fall in the satisfactory region of the handling qualities criterion of Reference 3-20, as shown by Figure 3-14. The roll acceleration provided by the RCS jets is also shown, but as a line rather than a point because the on-off operation is not consistent with the concept of a specific time constant. This line falls in the acceptable region for a wide range of time constants. Redundancy of control torque application is provided on all axes by using dual gimbal actuators and by arranging the RCS jets in such a manner that, with proper jet selection, failure of any single jet does not result in loss of control.

TABLE 3-5

ANGULAR ACCELERATION CONTROL CAPABILITY

MISSION PHASE	ROLL ACCELERATIONS (DEG./SEC.^{2})	TORQUE CONFIGURATION	PITCH, YAW ACCELERATIONS (DEG./SEC.^{2})	TORQUE CONFIGURATION
EARTH PARKING ORBIT	1.25	4-100 LB. TRANSLUNAR ROLL JETS FIRING	.0445	2-100 LB. TRANSLUNAR ROLL JETS FIRING
TRANSLUNAR	1.74	4-100 LB. TRANSLUNAR ROLL JETS FIRING	.236	1-100 LB. TRANSLUNAR PITCH-YAW-ULLAGE JET FIRING
HOVER	8.05	4-100 LB. TRANSLUNAR ROLL JETS	12.7	(GIMBALLED ENGINE) 5° FROM C.G.
LUNAR TAKEOFF	2.48	4-25 LB. TRANSEARTH ROLL JETS FIRING	4.7	(GIMBALLED ENGINE) 5° FROM C.G.
TRANSEARTH	9.9	4-25 LB. TRANSEARTH ROLL JETS FIRING	1.64	2-25 LB. TRANSEARTH ROLL JETS FIRING
RE-ENTRY	16.4	2-50 LB. RE-ENTRY ROLL JETS FIRING	6.7	2-25 LB. RE-ENTRY PITCH & YAW JETS FIRING

COMPARISON OF ANGULAR ACCELERATION CAPABILITY WITH NASA HANDLING QUALITIES CRITERION

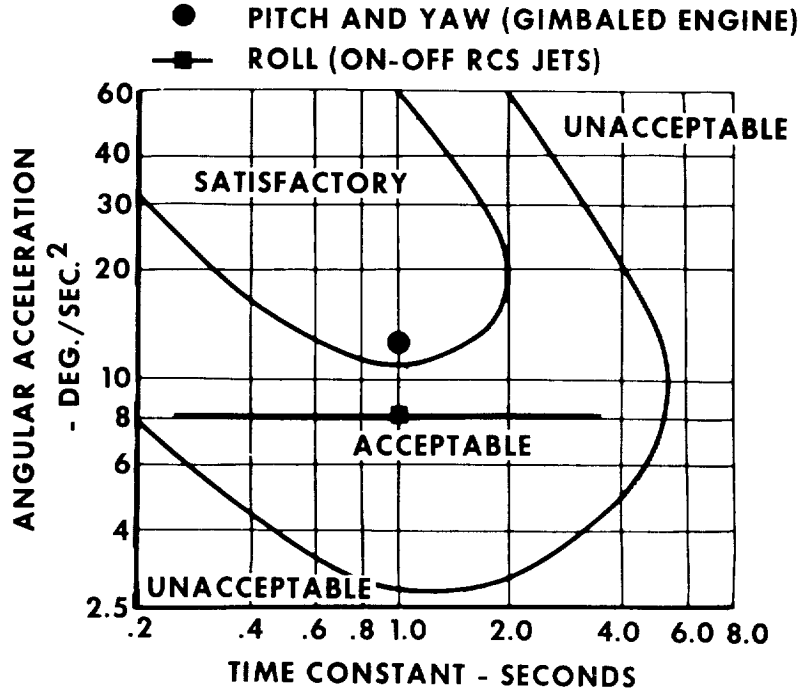


FIGURE 3-14

3.3 Environmental Control

3.3.1 Crew and Equipment - The environmental control system (ECS) is essentially identical to that used in the Gemini spacecraft with the quantities of expendables adjusted for the 8-day lunar mission. The cabin pressure is maintained at a nominal 5 psi with O_2 as the gaseous medium. This concept is an extension of the Project Mercury environmental control system which has been proven in orbital flight and by exhaustive altitude chamber tests. The ECS provides the following:

A. Pressure control of the cabin and pressure suits

B. Conditioned gas suitable for breathing by and ventilation of the crew

C. Equipment temperature control

D. Water management

E. Reliable operation with redundant components where required

Electronic equipment temperatures are controlled by a coolant circulated through the equipment mounting plates. Individual items of hardware are passively controlled by insulation where necessary and electrical heaters are utilized in specific instances where lower temperature limits are approached.

3.3.2 Environmental Control System Description - The primary components of the ECS are shown schematically in Figure 3-15. The command module contains the closed-loop circuit, the cabin circuit and the secondary oxygen supply. The crewmen are positioned in parallel in the closed-loop circuit and a flow of conditioned oxygen at a nominal inlet temperature of 60°F and a rate of 10 cfm at 5 psi is introduced to each pressure suit to provide ventilation and breathing oxygen. An important feature of the closed-loop environmental control system design is that approximately three-fourths of the metabolic heat is removed by the evaporation of perspiration from the body surface. This is a perfectly comfortable environmental situation for the crewmen, however, it does require that the torso garments be

3.3.2 (Continued) **PRIMARY COMPONENTS OF THE ENVIRONMENTAL CONTROL SYSTEM**

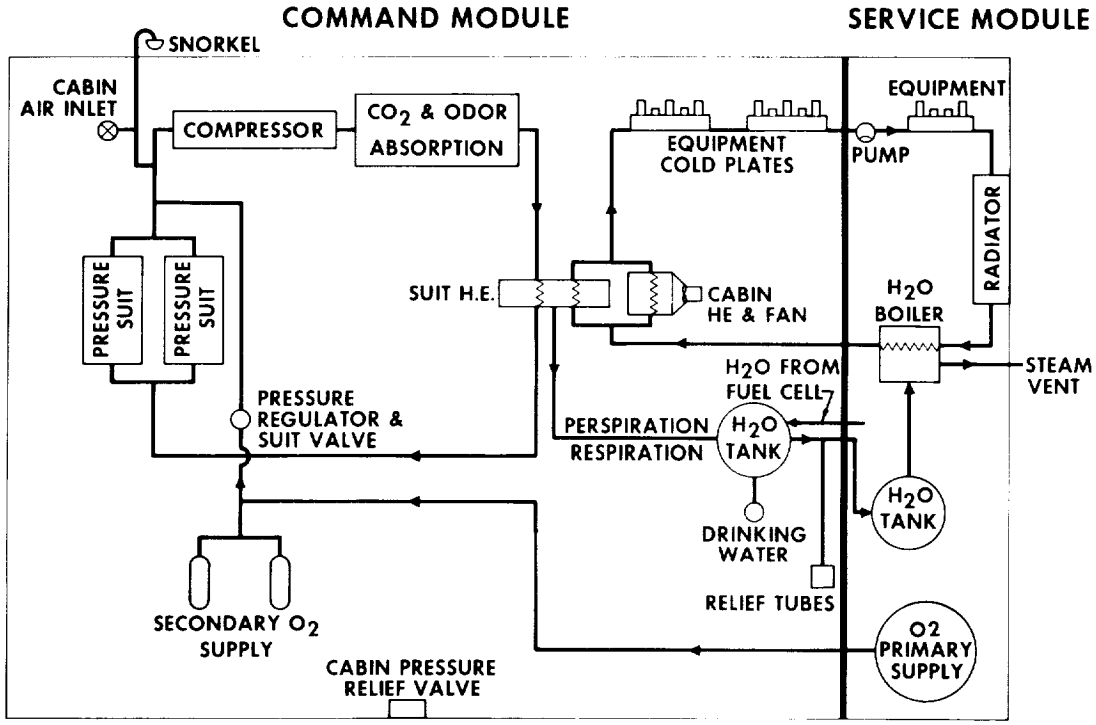


FIGURE 3-15

continuously worn and ventilated. The relatively large body surface evaporative rate requires that the crewman water intake be sufficient to maintain a normal body water balance. Approximately 12.5 pounds of drinking water per day per man is required to maintain the correct body water balance for a metabolic rate of 11,300 BTU/day/man. The food processed in the body is assumed to supply one pound of water per day. The design criteria used in this vehicle are delineated in Section 2.7.5. A compressor draws the gas mixture (O_2, CO_2, H_2O vapor) from the suits and forces it through a LiOH canister where the CO_2 is removed and odors absorbed. At the exit of the LiOH canister the CO_2 partial pressure is 1 mm Hg or less. A heat exchanger removes heat from the gas stream and condenses the water vapor before returning the gas to the pressure suit inlets. The gas flows are individually throttled by the crewmen and the gas temperature is selected by adjustment of the heat exchanger coolant flow rate.

3.3.2 (Continued)

The cabin pressure is controlled by a valving system similar in operation to the type used on Project Mercury. A cabin heat exchanger and fan are utilized to cool and circulate the cabin gas.

The water generated from perspiration and respiration that is condensed by the heat exchanger is removed from the closed loop circuit by a wick-type separator and transferred to a storage tank where it may be utilized for drinking or cooling as required. A separate system is used to transfer urine to the water boiler in the service module. The secondary oxygen supply is used as the cooling and breathing O_2 supply for the crewmen during earth re-entry and is stored in the gaseous state at 7500 psi in redundant bottles.

The ECS components in the service module consist of a supercritical primary O_2 supply, the OS 139 coolant pumps, a water storage tank, water boiler, and ground cooling heat exchanger.

An integral structural portion of the service module wall constitutes an external radiator surface that serves as the heat sink for all heat generated in the vehicle. The surface radiation characteristics, size and angular arrangement are selected to prevent freezing or overheating of the coolant or internal systems. Certain phases of the mission require that the water boiler be utilized to supplement the performance of the radiator.

3.3.3 Pre-launch Operation - After the hatches are sealed, an external ground purge connection is used to purge the cabin with O_2. A pressure check of the cabin is performed. The ECS and required electronic equipment are activated. Ground cooling is provided by a system which circulates a cold fluid through a ground cooling heat exchanger in the service module. The normal capsule coolant loop operates to remove heat from the suit and cabin heat exchangers and the electronic equipment. The radiator is by-passed by the OS 139 coolant during this phase of the operation.

3.3.3 (Continued)

In the retrograde module, the hydrogen tank insulation spaces are ventilated continuously with helium to prevent liquefaction of air. The tank compartment is vented with dry air to prevent formation of ice on outer surfaces of the H_2 tank insulation or the O_2 tank surfaces.

The O_2 and H_2 tanks that supply the ECS and fuel cells are protected by super insulation to prevent venting during the pre-launch operation.

3.3.4 Launch and Translunar - The pressure control components of the ECS perform during the launch phase in the same manner as in Project Mercury.

The radiator is by passed by the coolant fluid and cooling is achieved by boiling water. This mode of operation continues for 45 to 50 minutes until the radiator has cooled sufficiently from the aerodynamic heating encountered during launch to permit satisfactory cooling operation. At this time, the by-pass valving is positioned to flow the coolant through the radiator. In remaining portions of the mission, the water boiler is utilized only to supplement the radiator for cooling.

The orientation of the vehicle during the translunar phase is with the small end of the command module pointing toward the sun. This orientation has the advantage that the radiator can be designed with an angular configuration that permits adequate cooling performance without coolant freezing. A 163 ft^2 radiator is required on the half cone angle of 30°. An additional benefit is achieved by this orientation since the retrograde module, which contains the H_2 and O_2 propulsion tanks, receives only a minimal amount of heat from the sun. The heat losses to space from the H_2 and O_2 tanks are sufficient to prevent boiloff of either fluid and eliminate the need for tank insulation during this phase of the mission. Analysis of the configuration has also shown that no freeze-out of O_2 occurs.

3.3.5 Lunar Stay - When the cabin is pressurized, the operation of the ECS on the lunar surface is identical to the translunar phase. The water boiler is used to supplement the radiator to a greater or lesser extent depending upon the selected lunar landing site. Figure 3-16 indicates the effect of lunar landing site on the initial water weight required at launch. The system is designed to provide the amount of water necessary to permit cooling during the most severe two day period. This peak period corresponds to a touchdown point at 355° from the subsolar point and a subsequent two day stay. Eleven pounds of water are required by the vehicle ECS for this condition assuming that the crew stays in the vehicle.

EFFECT OF SELECTION OF LUNAR LANDING SITE ON INITIAL WATER REQUIREMENT

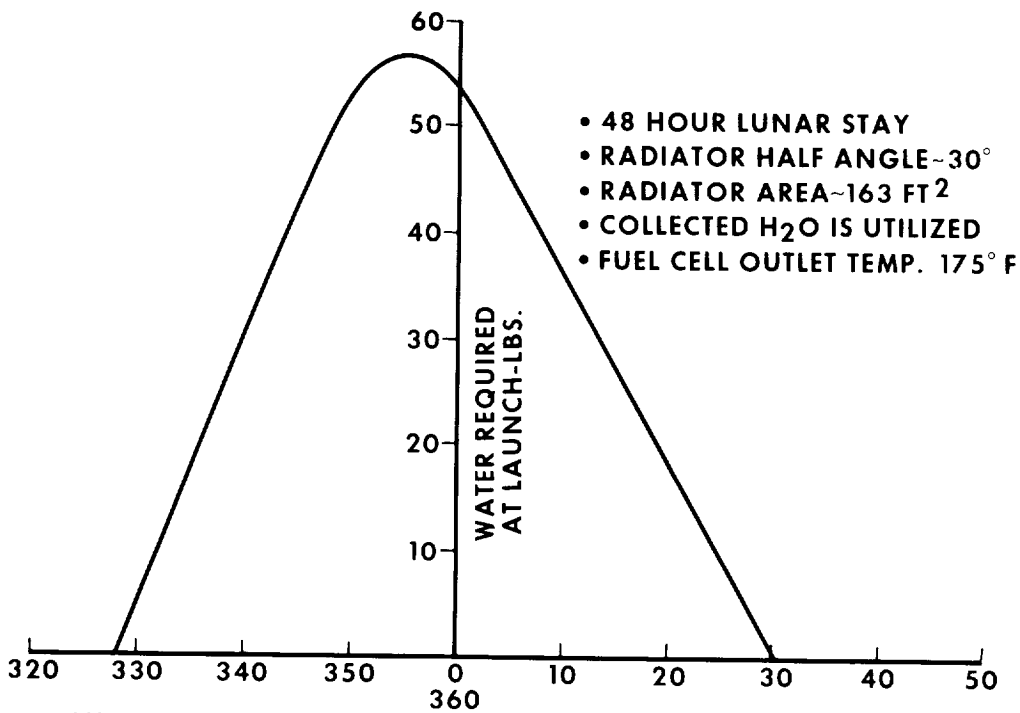


FIGURE 3-16

The portable environment control systems require 16.24 pounds of water which is available in the command module water tank.

Excursions on the lunar surface require a transfer from operation on the vehicle to the portable ECS. This is accomplished with the cabin pressurized by disconnecting the hoses from the suit and coupling the free ends together. Attachment is made to

3.3.5 (Continued)

the portable ECS lines and the portable O_2 supply and fan activated. The cabin is slowly decompressed and the exit effected. Re-entering the vehicle is accomplished by reversing the procedure. Provisions for replenishing the portable ECS oxygen are included in the primary supply.

Thermal analysis has shown that the earth storable propellants and the supercritical gas storage containers in the service module will not be adversely affected by a two day stay at any equatorial point on the lunar surface. However, temperature control of water lines and individual valves may be necessary for certain landing sites. Provisions are incorporated for electrical heating of such items.

Table 3-6 summarizes the water inventory for all phases of the mission.

TABLE 3-6

WATER INVENTORY

- LUNAR LANDING AT SITE REQUIRING MAXIMUM H_2O USAGE
- TWO DAY LUNAR STAY
- 30 DEG. CONICAL HALF ANGLE ON RADIATORS

MISSION PHASE *	COMMAND MODULE WATER TANK (LBS.)	SERVICE MODULE WATER TANK (LBS)
EARTH LAUNCH	40	16
TRANS LUNAR	34	0
LUNAR ORBIT	40	42
LUNAR REST	40	45
LUNAR LAUNCH	0	0
TRANS EARTH	2	0
POST LANDING	40	0

* QUANTITIES SHOWN ARE AT START OF EACH INDICATED MISSION PHASE

3.3.6 Transearth and Earth Re-entry - The thermal environment of the vehicle is maintained within acceptable limits in the same manner as in the translunar phase. The radiator serves as the heat sink until the service module is separated prior to re-entry into the earth's atmosphere. When the service module separation occurs (8 minutes prior to reaching 400,000 feet) the secondary O_2 supply is utilized at a rate of 0.1 lb/man-minute to achieve cooling during re-entry. The electrical equipment provides thermal capacity for absorbing the heat generated during this time period.

The cabin pressure is maintained at the nominal 5 psi during re-entry and at approximately 20,000 feet a snorkel valve is opened to admit ambient air into the suits and cabin. In the case of land landings, the crewmen will be required to leave the vehicle for a sufficient length of time to allow the structure to cool down.

3.3.7 Reliability Considerations - To provide acceptable performance in the event of component failure, an independent supercritical pressure and temperature control has been incorporated in the RSS for ECS usage and has been paralleled with the normal ECS O_2 supply. Valving has been included for isolation during normal operation and to permit supply and control from either source if an emergency occurs. Redundant suit circuit compressors are incorporated with manual switch-over in the event of failure. Also, two completely independent coolant circuits have been provided with satisfactory operation of either being sufficient.

3.3.8 Deviations from the Project Apollo Statement of Work - Deviations from certain of the requirements stated in Reference 3-19 are necessary. The deviations

3.3.8 (Continued)

are stated below and are considered justified either by directive from the NASA or by the nature of the developed systems proposed for use in the Two-Man Apollo vehicle.

The paragraph numbers used below refer to the paragraph numbers in Reference 3-19.

Paragraph - 3.2.1.1.2.1.4

Requirement - Means of egress to free space without decompression of the entire Command Module shall be provided.

Two-Man Apollo Design - During egress the entire Command Module must be depressurized since no airlock is incorporated in the vehicle.

Paragraph - 3.2.1.2.4.2

Requirement - A "shirtsleeve" environment shall be provided during all flight phases.

Two-Man Apollo Design - A "shirtsleeve" environment will not be provided. The ECS system requires that the torso be continuously ventilated.

Paragraph - 3.2.1.2.5.1

Requirement - 14-day mission without resupply

Two-Man Apollo Design - 8-day mission without resupply. (Reference 3-21)

Paragraph - 3.2.1.2.5.2

Requirement - The command module shall provide the crew a habitable environment for one day following earth landing.

Two-Man Apollo Design - For land landing, command module must be vacated until wall "cool down". Water landing will provide an immediate one-day habitable environment.

Paragraph - 3.2.3.4

Requirement - The repressurization system shall be designed for two complete cabin repressurizations, a minimum of 18 airlock operations, and a continuous

3.3.8 (Continued)

leak rate as high as 0.2 lbs. per hour.

Two-Man Apollo Design - The repressurization system is designed for 9 complete cabin repressurizations, and a continuous leak rate of 0.15 lb. per hour. There is no airlock.

Paragraph - 3.2.5.2

Requirement - Water consumption is 6.0 lb/day/man. (This includes water in food).

Two-Man Apollo Design - Water consumption is 13.5 lb/day/man. (This includes water in food).

Paragraph - 3.4.2.8.1.1

Requirement - Total cabin pressure (O_2 and N_2 mixture) 7 $\pm$ 0.2 psia.

Two-Man Apollo Design - Total cabin pressure (pure O_2) 5 psia (Nominal).

Paragraph - 3.4.2.8.2.1.1.2

Requirement - Noxious gases shall be removed by activated charcoal and a catalytic burner with the latter provided with a regenerative heat exchanger. A gas analyzer shall be provided.

Two-Man Apollo Design - Noxious gases shall be removed by activated charcoal. A catalytic burner is not provided. A CO_2 measuring device will be provided.

Paragraph - 3.4.2.8.2.1.1.3

Requirement - The system shall provide for two, parallel, isolated, lithium hydroxide canisters.

Two-Man Apollo Design - Lithium hydroxide canister is not in parallel.

Paragraph - 3.4.2.8.2.1.1.4

Requirement - The loop shall be provided with three, parallel, isolated blowers. 12 cfm thru each suit at 3.5 psia. Each pressure suit connection shall have a bypass which will permit individual manual flow control.

3.3.8 (Continued)

Two-Man Apollo Design - The loop shall be provided with two, parallel, isolated blowers. 10 cfm through each suit at 5 psia. Manual flow control will be provided by manual flow control valves upstream of the suits.

Paragraph - 3.4.2.8.2.1.1.5

Requirement - A regenerative heat exchanger shall be provided for the crew to control inlet air temperature. A water evaporator shall also be provided for cooling of circulating air in event of loss of coolant.

Two-Man Apollo Design - A coolant flow control valve regulates the pressure suit and cabin temperatures. A redundant coolant loop is provided.

Paragraph - 3.4.2.8.2.1.1.6

Requirement - The condensed water vapor shall be removed by either of two, parallel, isolated separators. Air-driven centrifugal water separators shall be developed for use. The development of a sponge-type water separator shall be pursued until the desired use of the centrifugal separator is unquestioned.

Two-Man Apollo Design - A wick-type water separator shall be integrated with the suit heat exchanger.

Paragraph - 3.4.2.8.2.1.2.1

Requirement - Atmospheric Circulation - The loop shall be provided with two fans, either of which is capable of circulating the required flow.

Two-Man Apollo Design - The "in suit" design of this ECS eliminates the dependence on cabin circulation and therefore one cabin fan is considered sufficient.

3.3.8 (Continued)

Paragraph 3.4.2.8.2.2.1

Requirement - The gas supply shall have a 50 percent excess capacity over that required for normal metabolic and leakage needs.

Two-Man Apollo Design - 10 lb. of O_2 is provided for leakage contingency. A connection to the fuel cell O_2 is provided for backup.

Paragraph - 3.4.2.8.2.3.2

Requirement - The liquid coolant flow shall be maintained at fixed rate by one of three hermetically sealed, constant speed pumps. The redundant reservoir accumulator shall allow for a complete recharging of the liquid coolant. Provisions shall be made for a gas check before recharging in the event of a rupture to allow for isolation of leakage zones and to re-establish system integrity.

Two-Man Apollo Design - The liquid coolant flow shall be maintained at a fixed rate by one of two hermetically sealed constant speed pumps. A completely redundant coolant loop and pump system is provided.

Paragraph - 3.4.2.8.2.3.3.5

Requirement - Lunar Landing - A refrigeration system shall be used to provide low temperature liquid coolant to the coolant loop.

Two-Man Apollo Design - A water boiler system shall be available to supplement the radiator for cooling during the lunar stay.

Paragraph - 3.4.2.8.2.3.3.6

Requirement - During re-entry, the thermal load shall be cooled by water evaporation in the liquid coolant heat exchanger.

Two-Man Apollo Design - During re-entry, the men shall be cooled in their suits by secondary O_2 flow. Equipment shall be cooled passively.

3.3.8 (Continued)

Paragraph - 3.4.2.8.3

Requirement - The water collected from the fuel cells shall be stored separately.

Two-Man Apollo Design - The water collected from the fuel cells shall be stored together with reclaimed perspiration.

3.4 Electrical Power - The design objectives for the electrical power system in the Two-Man Apollo Spacecraft are to provide adequate DC power for all electrical loads throughout the mission and to provide a simple system with a minimum number of components and automatic devices.

The system selected as best fulfilling these objectives employs Gemini type fuel cells and batteries in the arrangement shown by the system schematic in Figure 3-17. Any four of six fuel cell stacks can carry the nominal load and any three of six stacks can carry the emergency load. Any one main battery can supply the re-entry load plus the nominal recovery load for approximately 4.3 hours. All three squib-control batteries share the control bus loads with two providing redundant power for squib devices. The total weight of the electrical power system is 776.8 lbs., distributed as shown in Section 4.8.

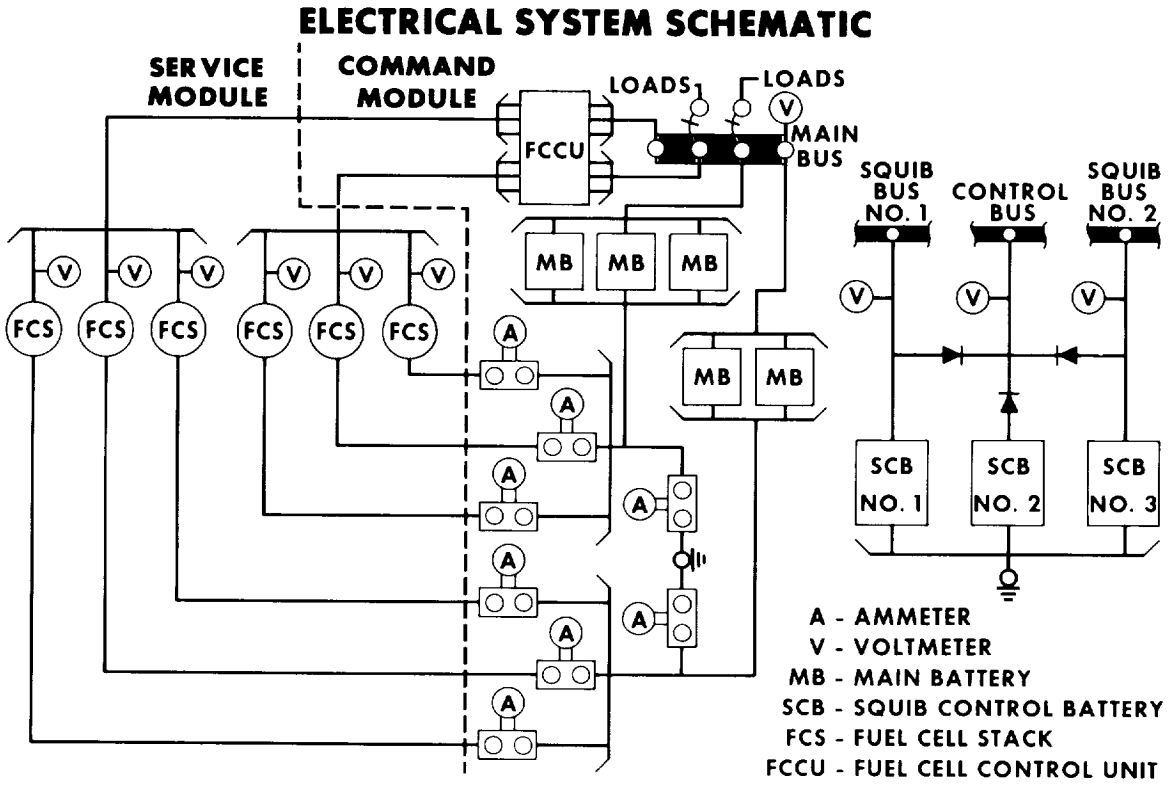


FIGURE 3-17

3.4.1 Power System Comparison - Four power systems have been compared, the ion exchange membrane fuel cell, Bacon fuel cell, Cryhocycle and H_2-O_2 turbine system. The results of the comparison are illustrated in Figure 3-18. The bases for comparison are as follows:

A. System parameters derived from recent hardware studies, reflecting current state of the art.

B. Power level of 2.1 KW peak and energy requirement of 900 watts for 192 hours duration.

C. Equal water provided from each system, either produced or stored.

Based on the results of the comparison, the system selected as the primary electrical power source utilizes ion exchange membrane fuel cells, identical to those used in the Gemini vehicle and supplied by General Electric.

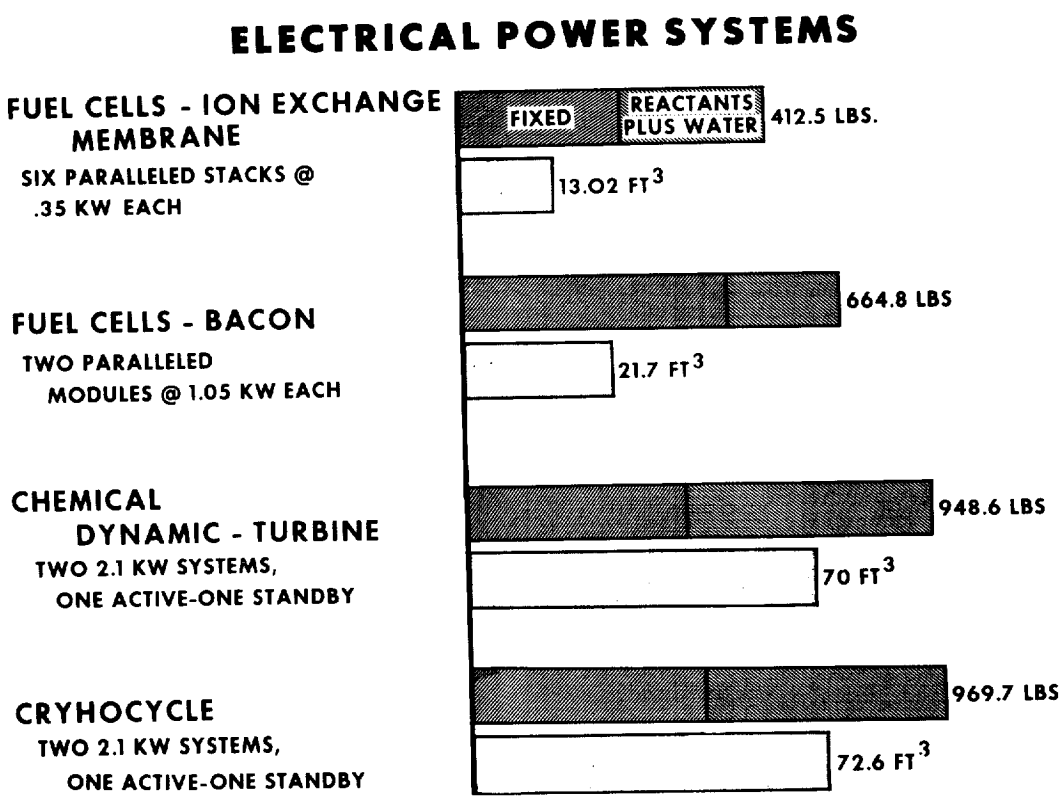


FIGURE 3-18

3.4.1 (Continued)

Both types of chemical dynamic systems considered provide cooling capacity for spacecraft equipment as well as providing electrical power and thus, in order to provide a valid comparison, the fixed equipment weights and volumes shown for the fuel cell systems include the spacecraft radiators. The reactant consumption for the chemical dynamic systems is based on providing the electrical power requirements and not the entire spacecraft cooling requirement. The large volumes required by both dynamic systems is the result of greater total reactant consumption compared to fuel cells and the percentage of hydrogen used (100% for Cryhocycle and approximately 70% for the turbine). No weight penalty for additional spacecraft structure to house the tanks is included.

The reactant consumption for both types of fuel cells compared is essentially equal. The difference in fixed weight results from the increased complexity of the Bacon type, which includes circulating hydrogen, water separation from hydrogen and electrolyte concentration control. No allowance is included for external insulation on the Bacon cells to maintain operating temperature at low loads.

The ion-exchange membrane fuel cells differ from the Bacon type as follows:

A. ECS coolant used for cooling instead of circulating hydrogen.

B. Static water removal instead of dynamic centrifugal separation from circulating hydrogen.

C. Solid electrolyte with no controls instead of liquid electrolyte requiring concentration control.

D. Operating temperature of 130-175°F instead of 400-500°F.

E. Start-up time in seconds instead of hours.

F. Parasitic power is 5 watts/section instead of 100 watts/section.

The G.E. fuel cells provide a greater estimated probability of successfully providing the required power than the Bacon cells. This is primarily because the

3.4.1 (Continued)

G.E. units contain fewer single components whose failure can preclude further fuel cell module (section) usage. If a stack component fails in a G.E. fuel cell, only 1/6 of the total fuel-cell-generated power is lost, whereas a failure of a stack component precludes further usage of one complete Bacon section, resulting in a 50% power loss. The added complexity of the electrolyte concentration control, the water separation and the temperature control systems of the Bacon cell increases the probability of a failure and the loss of output from a complete module.

The results of a reliability study of the fuel cells examined, including the reactant supply systems, are shown in Figure 3-19.

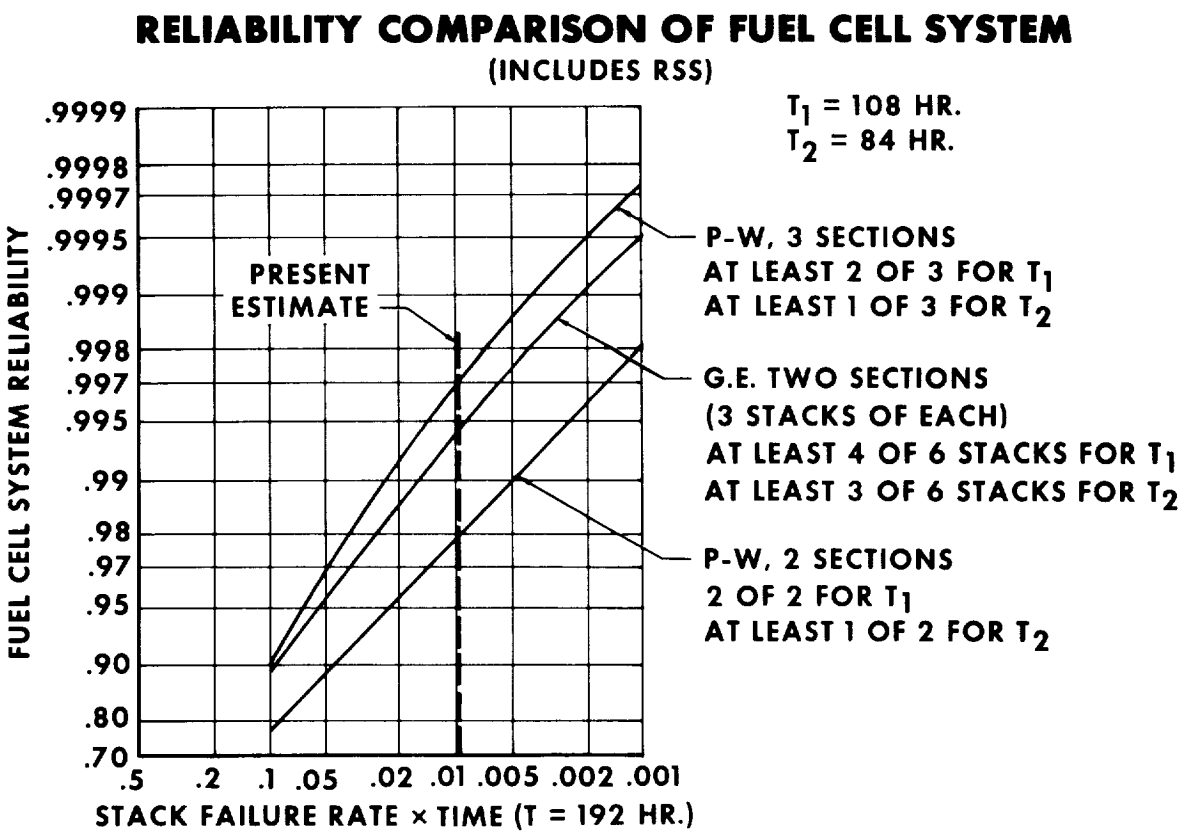


FIGURE 3-19

3.4.1 (Continued)

For comparison, a three-section Bacon cell arrangement is also included. The assumptions for this study are:

A. The stack failure rate is the same for both the G.E. cell and the Bacon cell.

B. To meet the vehicle power requirements, at least two sections of the Bacon cell must function for the first 108 hours; at least one section is required for the succeeding 84 hours.

C. For the G.E. cell, at least 4 stacks must function for the first 108 hours; at least 3 stacks must function for the remaining 84 hours.

D. The reliability of the reactant supply system is the same for the three cases shown.

3.4.2 Normal and Emergency Power Spectra - The total average electrical load level during each mission phase is shown in Figure 3-20, along with fuel cell power output capability as a function of the number of operating stacks. Table 3-7 summarizes individual system load requirements. No allowance is included for growth of equipment load requirements. It is assumed that load increases will be compensated by improvement in fuel cell performance capability. The emergency power level is considered to be 70% of the normal power level based on a detailed load analysis. Momentary peak load requirements, such as occur at turn-on of electronic gear, lights, etc. are supplied from the fuel cells and, if necessary, from the main batteries. Load requirements for squibs, relay coils, reaction control system solenoids, etc. (assumed to be 8/14 of the 14-day Gemini load) are not included in the above figures, since they are powered from isolated squib-control batteries.

ELECTRICAL LOAD PROFILE

1.7 KW PEAK, 172.5 KW-HR@900 WATT AVERAGE

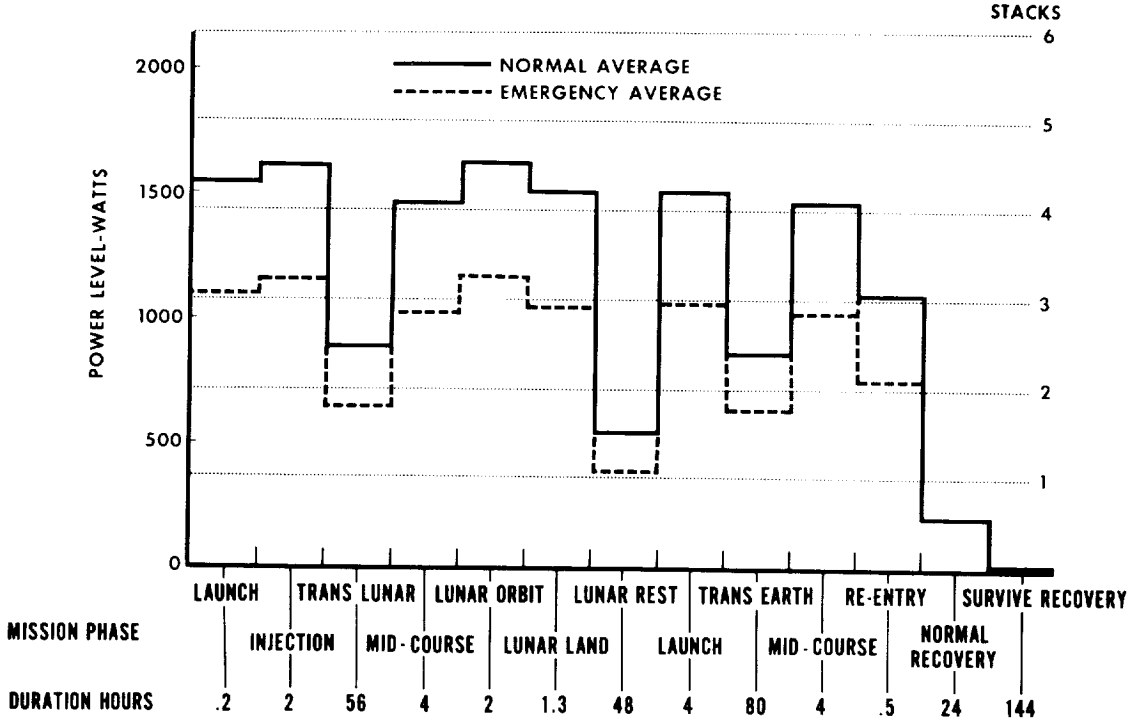


FIGURE 3–20

TABLE 3–7

ELECTRICAL LOAD SUMMARY - WATTS (AVERAGE)

EQUIPMENT	EARTH LAUNCH	INJECTION	TRANSLUNAR	MID-COURSE CORRECTION	LUNAR ORBIT	LUNAR LAND	LUNAR REST	LUNAR LAUNCH & INJECTION	TRANSEARTH	MID-COURSE CORRECTION	RE-ENTRY	RECOVERY NORMAL	RECOVERY SURVIVE
COMMUNICATIONS	330	329	187	286	298	257	127	286	187	286	290	49	14
LIGHTING	33	33	33	33	33	33	33	33	33	33	33	33	
ENVIRONMENTAL CONTROL	290	279	283	279	280	279	289	279	283	279		100	
NAVIGATION & GUIDANCE	440	627	125	532	687	590		557	113	532	428		
STABILIZATION & CONTROL	225	225	143	223	223	223		223	137	223	224		
MISCELLANEOUS	235	101	133	101	103	101	103	102	106	101	100	15	

3.4.3 Basic Generating Principle and Operating Modes - The basic power generation system consists of Gemini fuel cells and batteries. The power capability of the fuel cells is 2.1 KW. The five main batteries provide 1250 watt-hours each and the three squib-control batteries 420 watt-hours each. The comparison between battery capacity and estimated load is shown on Figure 3-21.

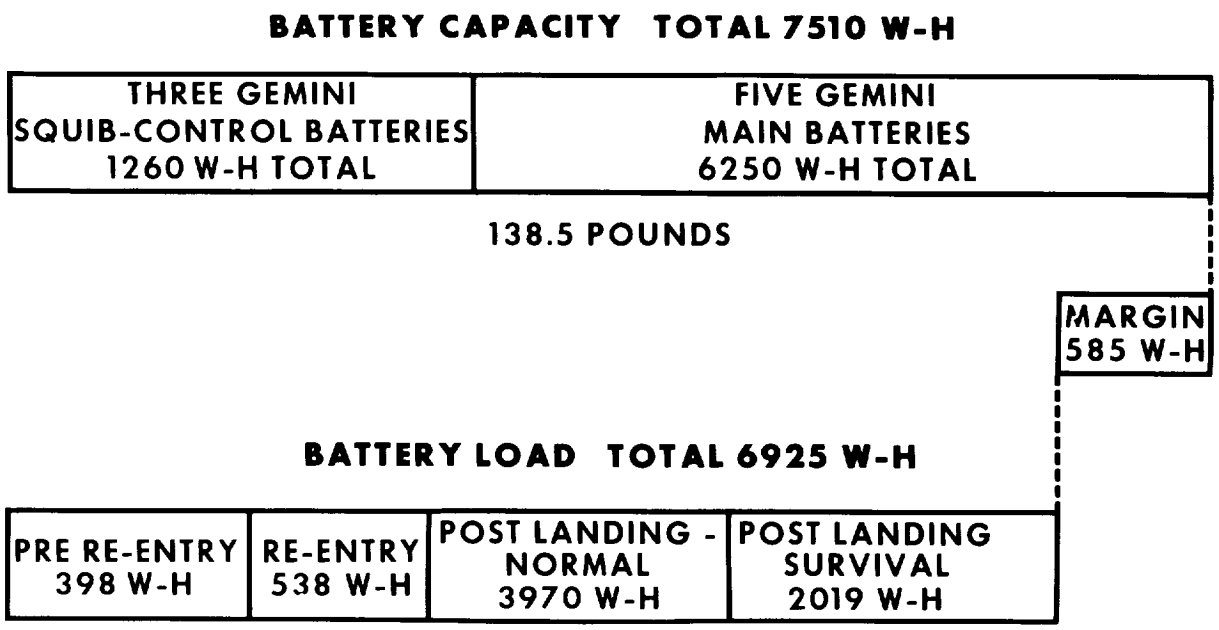


FIGURE 3-21

The 2.1 KW fuel cell output is derived from six 31-cell stacks, electrically paralleled, each rated at 350 watts. Circuitry is provided to monitor the current from each stack and compare it to the average stack current. In the event any stack current drops to an unacceptable level, indicator lights signal the crew. An ammeter-selector switch combination is provided for manual crew monitoring of stack performance. Manual controls are provided to remove any stack, without affecting the operation of the remaining stacks.

Sufficient reactants are provided for fuel cell operation at a level of 900

3.4.3 (Continued)

watts for the 8 day mission duration. Two tanks for each reactant, hydrogen and oxygen, are provided. Cross plumbing is provided between the reactant pressure regulators so that both fuel cell sections can be operated from one dual pressure regulator in the event the other regulator fails. A schedule of reduced loading and combined use of the fuel cells and battery capacity is planned to permit safe crew return from any point in the mission after the loss of reactant supply from one tank.

All batteries are attached to cold plates, maintained at 85°F from launch to re-entry, to ensure full output on demand and minimize loss of capacity.

Both the Gemini fuel cells and batteries are currently undergoing qualification testing and will have qualified Gemini flight status when required for the direct lunar flight.

3.4.4 Distribution and Regulation - The power distribution system is shown in Figure 3-17 and is essentially the same as the Gemini arrangement. Isolated squib-control busses, from which transient producing equipment such as relays, solenoids and squibs are powered, are used so that the main bus, from which sensitive equipment derives power, is essentially free from excessive disturbances.

Diodes are not used in series with the fuel cell stacks, so that the effective capacitance of the fuel cells is utilized to maintain voltage stability and avoid the power loss associated with diode use. The contactors used to connect each stack to the main bus are provided with reverse current windings.

The main bus and control bus voltage regulation is 26 ± 4 V.D.C. The two squib busses have a voltage variation of 13 volts during squib firings to 30 volts at no load.

AC power, and DC power required at different levels from those provided by the fuel cells and batteries, is provided internally in the using equipment.

3.4.4 (Continued)

External power is used during pre-launch operation on earth to power all busses. At a pre-selected time, the fuel cells and squib batteries are brought on the line.

3.4.5 Control Modes - The control of the electrical power sources and loads is manual. Hermetically sealed, switch-type circuit breakers are used for control of equipment and protection of wiring.

Bus voltages and fuel cell stack voltages are monitored by the crew through the use of a voltmeter-selector switch combination.

Main bus current is continuously displayed, with fuel cell stack currents individually monitored, when desired, with an ammeter-selector switch combination.

Fuel cell reactant quantities and pressures are continuously displayed.

3.4.6 Maintenance and Serviceability - All components of the electrical power system are installed with adequate accessability provided for checkout. Sealed modular construction is used to preclude possible damage during handling.

Test points, accessible from the outside of the spacecraft, are provided for monitoring system operation during check-out operations.

3.4.7 Deviation from Project Apollo Statement of Work - Deviations from several requirements stated in Reference 3-19 are necessary. The deviations are stated below and are considered justified either by directive from the NASA or by the nature of the developed systems proposed for use in the 2-Man Apollo spacecraft.

The paragraph numbers used below refer to the paragraph numbers in Reference 3-19.

Paragraph - 3.2.1.2.5.1

Requirement - Provisions shall be made for 14 days operation without re-supply. For lunar landing mission, 7 of 14 days may be on lunar surface.

Two-Man Apollo - Provisions are made for 8 days operation without resupply, including 2 days allowed for lunar stay.

3.4.7 (Continued)

Paragraph - 3.4.2.9.1.2

Requirement - Fuel cell system shall consist of 3 H_2-O_2 fuel cell modules. Three silver-zinc primary batteries, rated at 3000 W-H/battery @ 10 hour rate @ 80°F shall be provided.

Two-Man Apollo - Fuel cell system consists of 2 H_2-O_2 fuel cell sections (6 stacks). Five silver-zinc primary batteries rated at 1250 W-H/battery @ 10 hour rate @ 85°F and three silver-zinc primary batteries rated at 420 W-H/battery @ 10 hour rate @ 85°F are provided.

Paragraph - 3.4.2.9.1.3

Requirement - The fuel cells and controls shall be located in the service module.

Two-Man Apollo - The fuel cell sections and reactant supply system are located in the service module. The fuel cell control unit is located in the command module.

Paragraph - 3.4.2.9.1.4.1

Requirement - For normal conditions, the entire electrical load shall be supplied by 3 fuel cell modules in parallel. Primary batteries would be maintained fully charged under this condition.

Two-Man Apollo - For normal conditions, the entire electrical load shall be supplied by 6 fuel cell stacks in parallel. Primary batteries would not be discharged under this condition.

Paragraph - 3.4.2.9.1.4.2

Requirement - For emergency conditions, 1 module failed, the entire electrical load shall be assumed by 2 fuel cell modules. Primary batteries would be maintained fully charged under this condition.

3.4.7 (Continued)

Two-Man Apollo - For emergency conditions, 1 stack failed, the entire electrical load shall be assumed by 5 fuel cell stacks. Primary batteries will not be discharged under this condition.

If 2 fuel cell stacks fail, the entire electrical load shall be assumed by 4 fuel cell stacks with batteries supplying power above the capability of 4 stacks. (Approx. 30% of capacity removed).

Or, if 2 fuel cell stacks fail, the crew shall program loads to an emergency level (70% of normal). The remaining 4 fuel cell stacks will provide all emergency level loads with no battery capacity required.

Paragraph - 3.4.2.9.2.1.1

Requirement - The fuel cell modules shall be low pressure, medium temperature Bacon type.

Two-Man Apollo - The fuel cell stacks shall be low pressure, low temperature ion exchange membrane type.

Paragraph - 3.4.2.9.2.1.2

Requirement - Each fuel cell module shall have nominal capacity of 1200 watts at output of 28 VDC and current density such that 50% overloads can be continuously supplied.

Two-Man Apollo - Each fuel cell section (3 stacks) shall have nominal capacity of 1050 watts at 22.5 VDC and current density such that 20% overload can be supplied for 5 seconds.

Paragraph - 3.4.2.9.2.1.3

Requirement - Fuel cell pressure and temp. - 60 psia and 425-500°F

Two-Man Apollo - Fuel cell pressure and temp. - 16 psia and 100-175°F

3.4.7 (Continued)

Paragraph - 3.4.2.9.2.1.6

Requirement - Fuel cells to contain integral heaters to bring cells up to 275°F for start-up at which time the fuel cell shall be self sustaining.

Two-Man Apollo - No heaters required for ion-exchange fuel cells. Start-up in seconds.

Paragraph - 3.4.2.9.2.2.2

Requirement - System voltage shall be 28 VDC nominal.

Two-Man Apollo - System voltage shall be 26 VDC nominal.

Paragraph 3.4.2.9.2.2.3

Requirement - Voltage regulation - ± 2 volts from nominal.

Two-Man Apollo - Voltage regulation - ± 4 volts from nominal.

Paragraph 3.4.2.9.2.3.1

Requirement - Tanks shall provide for 14 days fuel cell and ECS operation. Nitrogen storage to be provided for both fuel cells and ECS.

Two-Man Apollo - Separate tank provided for fuel cells and ECS, for 8 days. No nitrogen required by either fuel cells or ECS.

Paragraph - 3.4.2.9.2.3.2

Requirement - Tank volumes shall be based on fuel cell requirements plus 10% plus ECS requirements as applicable.

Two-Man Apollo - Separate fuel cell and ECS tanks provided with reactants for an 8 day mission, 2 days of which are contingencies.

3.5 Telecommunication System

3.5.1 Summary - The telecommunication system for the Two-Man Apollo spacecraft is similar to that presently being developed for the Three-Man Apollo. The characteristics of the Two-Man system, however, have been established independently utilizing References 3-2 and 3-22 through 3-26 as major sources of data. The weight of the principal elements are summarized in Table 3-7 with the weights for the Three-Man system, as derived from Reference 3-15, included for comparison.

TABLE 3-7

TELECOMMUNICATION SYSTEM WEIGHT COMPARISON

SUBSYSTEM	3-MAN APOLLO	2-MAN APOLLO
DSIF SUBSYSTEM	60 LB.	39 LB.
NEAR-EARTH SYSTEMS	42	54
INTERCOM	5	5
RECOVERY		6.8
DATA PROCESSING	74	65
SPARES		20
INSTRUMENTATION	147	166
ANTENNAS	130	104.4
WIRING, ENVIRONMENTAL & STRUCTURAL PROVISIONS	101	210
TOTAL WEIGHT	584 LB.	670.2 LB.
TOTAL VOLUME	———	10.408 FT.^{3}

3.5.2 System Requirements - The telecommunication system provides:

A. Continuous voice communication between the spacecraft and earth.

B. Transmission of telemetry and television data to earth.

C. Voice communication between crew members during extravehicular operation.

D. Tracking of the spacecraft during all mission phases.

E. Location and rescue aids during the recovery phase.

3.5.2 (Continued)

In arriving at a definition of the system, these guidelines have been followed:

A. Redundancy or spare modules for critical equipment to provide backup for component failures.

B. Redundant signal flow paths for voice communications.

C. Ample circuit margin to allow for system degradation or unexpected propagation loss.

D. Conventional design techniques to minimize development requirements.

3.5.3 System Description

A. DSIF - The Deep Space Instrumentation Facility (DSIF) supplies all spacecraft communications during the transit and lunar phases. The onboard equipment includes redundant receivers, transmitters, and power amplifiers with an available RF power output of one or twenty watts, Figure 3-22. This equip-

TELECOMMUNICATION SYSTEM FUNCTIONAL BLOCK DIAGRAM

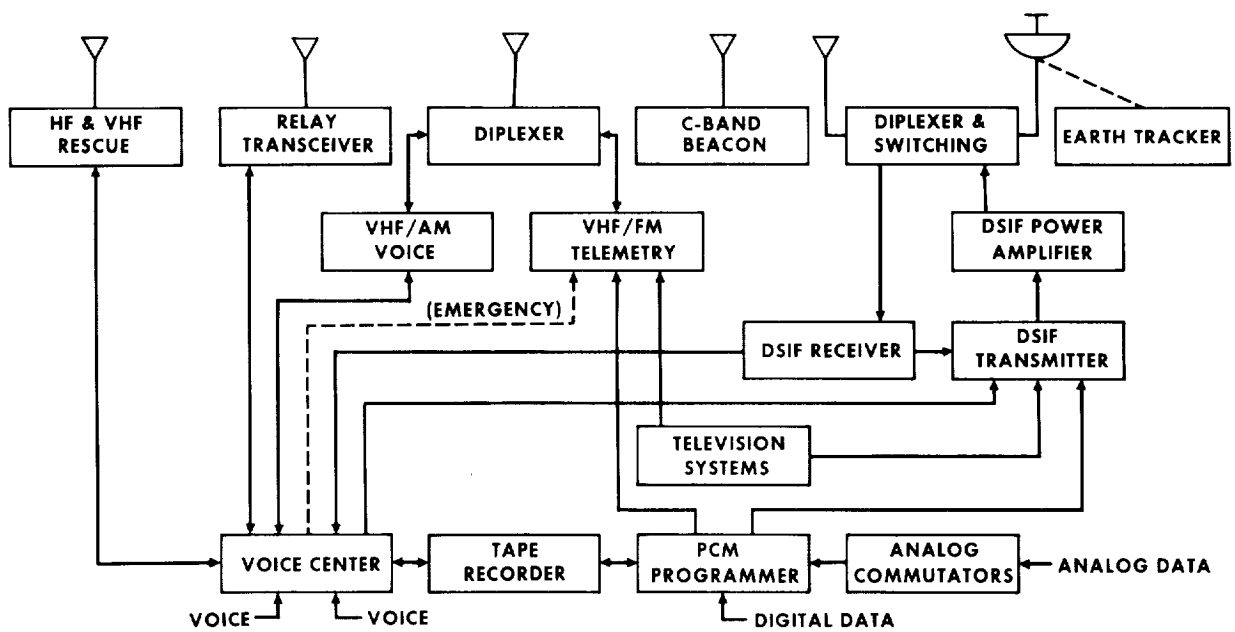


FIGURE 3-22

3.5.3 (Continued)

ment operates in conjunction with the three station DSIF, utilizing 85' parabolic antennas. The receiver and transmitter operate together to form a coherent transponder to allow precise doppler tracking of the RF carrier and "turn-around" range tracking of the DSIF range code. Antennas for the system include an omnidirectional system for use near the earth, and two, four foot diameter parabolic antennas, capable of automatic earth tracking which are used during the majority of the transit and lunar phases of the mission.

Data transmission from the spacecraft is in the form of PCM-PM modulation of a subcarrier channel. Data may be transmitted at a maximum rate of approximately 200,000 bits per second in a data-only mode. Voice transmission to and from the spacecraft is in the form of analog FM modulation of another subcarrier channel. At the same time, data transmission is continued at a reduced rate.

Periodic transmission of television data at a rate of two frames per second is achieved by analog FM modulation of the transmitted carrier. Voice and data transmission are interrupted during television transmission. Television data may be continually displayed to the crew in a closed circuit form.

B. Near-Earth - The near-earth systems operate in conjunction with the Mercury/Gemini ground network to provide spacecraft tracking and communication during the periods immediately following launch and before re-entry, when continuous communications with the DSIF is not possible. Voice communication is provided by the VHF/AM voice transceivers. Telemetry data is transmitted during these phases by means of the VHF/FM transmitters. Both operate

3.5.3 (Continued)

through a stub antenna, located at the apex of the command module cone, which will provide nearly omnidirectional coverage. Each utilizes a 20 watt transmitter to insure reliable communication to the point at which continuous contact is established with the DSIF. Radar tracking is facilitated by means of a 2500 watt C-Band beacon operating in conjunction with the AN/FPS-16 or the AN/FPQ-6 radar. The beacon antenna consists of three cavity backed helical elements, equally spaced around the command module.

C. Data Processing - The data processing system includes the data commutators, PCM programmer and tape recorders. Up to 600 data channels can be combined into a single PCM train for transmission to earth. Data in digital form, bilevel sequence signals and high or low level analog data are processed. Two tape recorders provide a capability for storage of voice and data when the spacecraft is not in direct contact with earth, as well as permanent storage of wide-band information such as vibration data taken during launch and re-entry.

D. Instrumentation - Instrumentation is provided to monitor the performance and status of the spacecraft systems and environment, and includes the telemetry sensors, associated power supplies and signal conditioning. Two television cameras allow the crew to observe remote and external objects not accessible to direct visual examination. In addition, two 16 mm motion picture cameras and one 35 mm camera provide a permanent record of crew activities, mission events, and lunar features of interest.

E. Other Systems - Voice communication during extravehicular operation is achieved by means of a relay transceiver which together with belt-pack

3.5.3 (Continued)

transceiver allows a crewman outside the vehicle to communicate to earth or with the other crewman.

After earth landing, voice communication is provided by an HF transceiver and the primary VHF voice transceivers. A VHF pulse rescue beacon, and tone modulation of the HF transceiver provide means of radio frequency direction finding to aid rapid location and recovery of the spacecraft.

3.5.4 Equipment Characteristics - A summary of the physical characteristics of the equipment is given in Table 3-8. An electrical load summary is contained in Table 3-9.

TABLE 3-9

TELECOMMUNICATION ELECTRICAL LOAD - WATT HOURS

MISSION PHASE / SUBSYSTEM	SYSTEM LOAD WATTS	LAUNCH (0.2)	INJECTION (2)	TRANSLUNAR (56)	MIDCOURSE CORRECTION (4)	LUNAR ORBIT (2)	LUNAR LANDING (1.3)	LUNAR REST (48)	LUNAR LAUNCH (4)	TRANSEARTH (80)	MIDCOURSE CORRECTION (4)	RE-ENTRY (0.5)	POST LANDING NORMAL (24)	POST LANDING SURVIVAL (144)
DSIF SUBSYSTEM	69.5	3.9	39	1372	278	139	90.3	1032	278	1960	278	—	—	
NEAR- EARTH SYSTEM	140	28	280	—	—	—	—	—	—	—	—	70	120	
INTERCOM	9	1.2	12	336	24	12	2.8	360	24	480	24	3	144	
RECOVERY	36	—	—	—	—	—	—	—	—	—	—	—	865	2019
DATA PROCESSING	35	11	110	2070	220	74	72	1800	220	2960	220	17.5	—	
INSTRUMENTATION	93	17.1	171	4996	500	250	163.3	2853	502.4	7120	500	42.9	48	
ANTENNA SYSTEMS	53	4.6	46	1680	120	120	—	144	120	2400	120	12	—	
PHASE TOTALS		65.8	658	10454	1142	595	333.4	6089	1144.4	14920	1142	145.2	1177	2019
MISSION TOTAL						39,884.8 WATT HOURS								

*NOTE: FIGURES IN PARENTHESES DENOTE PHASE TIME IN HOURS

TABLE 3-8

TELECOMMUNICATIONS DETAIL WEIGHT SUMMARY

COMPONENT	3-MAN APOLLO			2-MAN APOLLO		
	LOCATION	WT.-LB.	VOL.-FT.^{3}	LOCATION	WT.-LB.	VOL.-FT.^{3}
DSIF SUBSYSTEM						
DISF RECEIVER TRANSMITTER	SM	30		SM(2)	9	.210
DSIF POWER AMPLIFIER	SM	30		SM(2)	30	.550
NEAR EARTH SYSTEMS						
VHF AM TRANSCEIVER	CM	8		CM(2)	20	.350
VHF FM TRANSMITTER	CM	20		CM(2)	20	.210
C-BAND TRANSPONDER	CM	14		CM(1)	13	.186
COAX SWITCH				CM(1)	1	.003
INTERCOM						
RELAY TRANSCEIVER	CM	3		CM (1)	3	.031
HEADSET	CM	2		CM(2)	2	
RECOVERY						
HF RECOVERY BEACON	CM	4		CM(1)	4	.076
VHF RECOVERY BEACON	CM	2		CM(1)	2.8	.047
DATA PROCESSING						
TELEMETRY	CM	38				
LL ANALOG COMMUTATOR				CM(2)	6	.070
LL ANALOG COMMUTATOR				SM(2)	6	.070
HL ANALOG COMMUTATOR				CM(4)	12	.140
HL ANALOG COMMUTATOR				SM(2)	6	.070
PROGRAMMER				CM(1)	7	.203
RECORDER	CM	36		CM(2)	28	.463
SPARES	CM	19		CM	20	.463
INSTRUMENTATION						
SENSORS	CM	18		CM	18	.500
SENSORS	SM	25		SM	25	.500
TV SYSTEM	CM	30		CM(1)	12	.145
TV SYSTEM				SM(1)	12	.145
CAMERAS & MAGAZINE	CM	20				
16 mm CAMERAS				CM(2)	20	.148
35 mm CAMERAS				CM(1)	5	.111
TELESCOPE	CM	54		CM(1)	5	.028
SIGNAL CONDITIONING				CM	21	.328
SIGNAL CONDITIONING				SM	7.5	.250
TIMING				CM	7.5	.194
PATCH PANELS				CM	12	.055
POWER SUPPLY				CM	14	.014
POWER SUPPLY				SM	7	.014
ANTENNA SYSTEMS						
ANTENNAS & TRANSMISSION	CM	54				
DSIF DISH ANTENNA	SM	32				
ANTENNA DRIVE	SM	44				
DSIF ANTENNA & DRIVE				SM(2)	50	
EARTH TRACKER				SM(2)	6	.042
2 KMC OMNI ANTENNA				SM (2)	0.5	.029
C-BAND ANTENNA				CM(3)	1.5	.003
PHASE MODULATOR				CM(1)	1.7	.026
VHF OMNI ANTENNA				CM(1)	4	.007
MULTIPLEXER				CM(1)	9	.151
VHF RECOVERY ANTENNA				CM(1)	1.25	.014
HF RECOVERY ANTENNA				CM(1)	4	.010
ASSOCIATED COMPONENTS				CM	16.5	.320
ASSOCIATED COMPONENTS				SM	10.0	.200
MISCELLANEOUS						
STRUCTURAL SUPPORTS	CM	12		CM	12	.239
STRUCTURAL SUPPORTS	SM	40		SM	40	.818
ENVIRONMENTAL CONTROL	CM	20		CM	20	.407
ELECTRONIC INTERFACE	CM	8		CM	8	.151
ELECTRICAL PROVISIONS	CM	9*		CM	90	1.800
ELECTRICAL PROVISIONS	CM	12*		SM	40	.818
TOTALS		**584**			**670.25**	**10.408**

*APPARENTLY THE REMAINDER OF NAA WIRING FOR TELECOMMUNICATIONS IS CHARGED ELSEWHERE.

3.6 Structure - Structural design is based on the criteria presented in Section 2.7. The command module primary structure is designed by loads from launch abort, parachute, pressurization and earth landing conditions. Equipment within the command module is supported for earth impact shock loads. Primary structure below the command module is designed by launch conditions. The lunar landing gear and back-up truss and local areas of the lunar landing module are designed by lunar landing loads. Internal structural trusses within each module are designed by inertia loads from tanks and equipment and propulsion unit thrust loads. Structural design loads for launch and abort conditions are shown in Figures 3-23, 3-24 and 3-25.

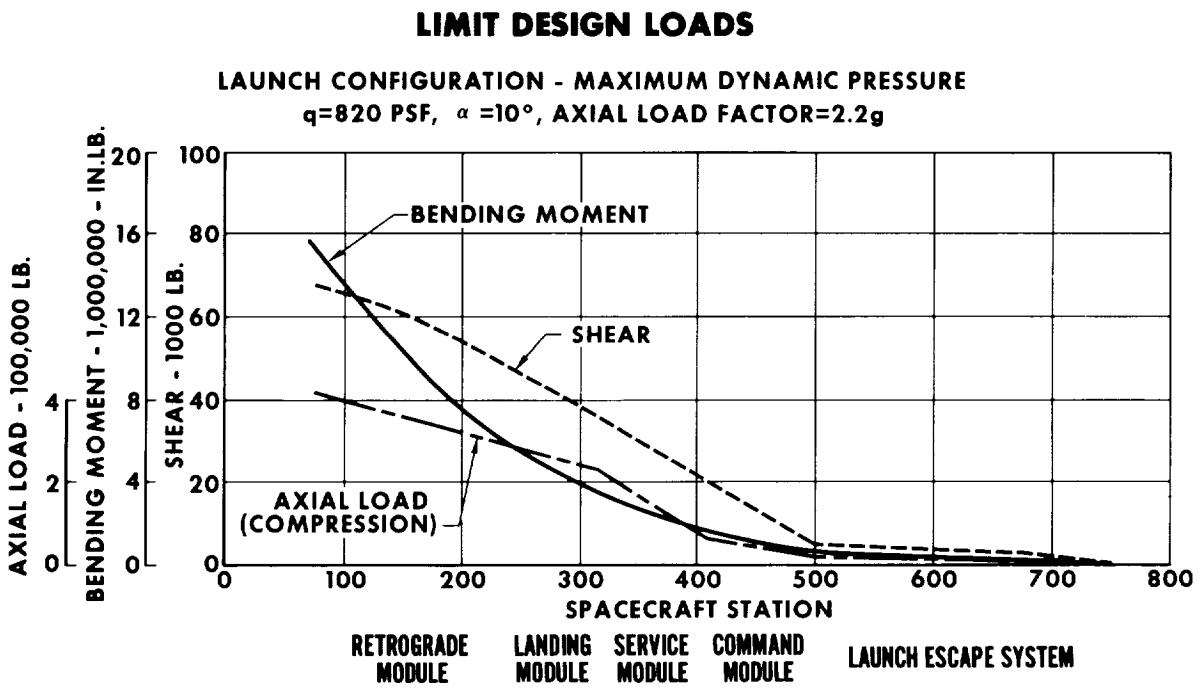


FIGURE 3-23

LIMIT DESIGN LOADS

LAUNCH CONFIGURATION - HARD OVER GIMBAL
q=40 PSF, AXIAL LOAD FACTOR=5.0g

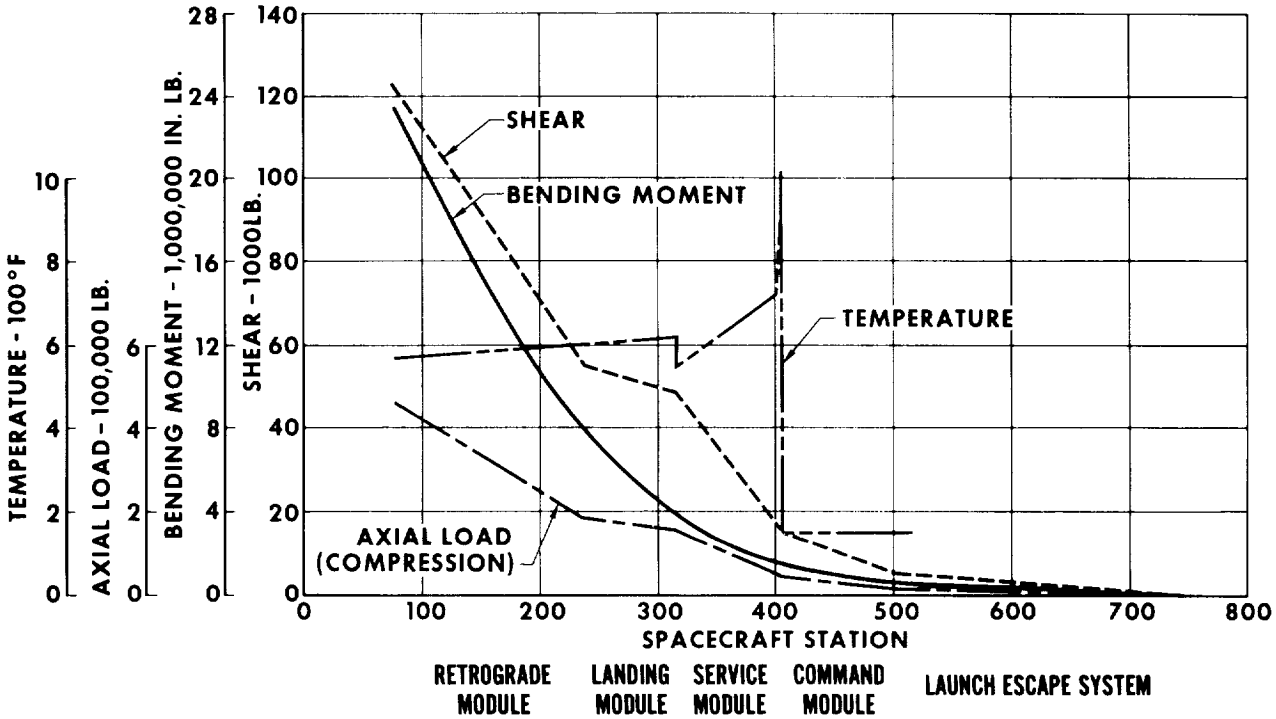


FIGURE 3-24

LIMIT DESIGN LOADS

TUMBLING ABORT

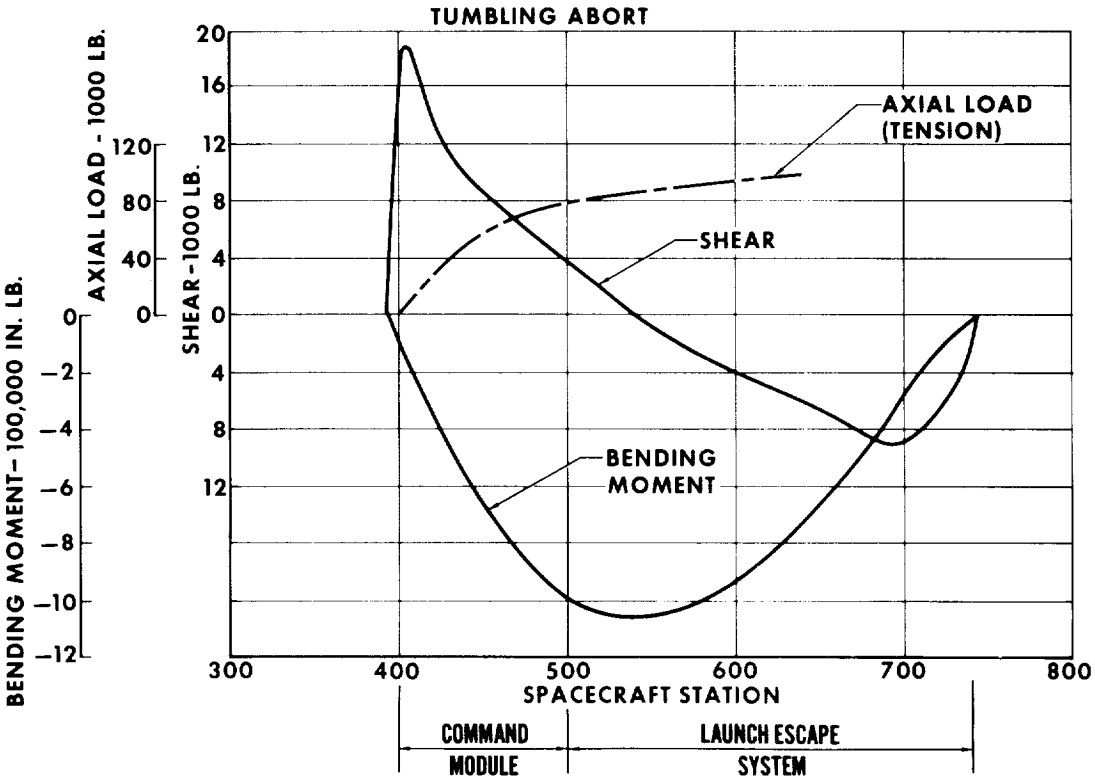


FIGURE 3-25

3.6.1 Structural Description - Conventional spacecraft structures are employed in all modules with materials and concepts demonstrated in the Mercury and Gemini designs. Beryllium sheet is used for the construction of the service module structural radiator shell. Primary structure of each module consists of a semi-monocoque shell with local reinforcements around cut-outs and fittings to distribute localized loads.

3.6.2 Command Module - A comparison study has shown that no distinct weight advantage exists for either aluminum or titanium for the command module primary structure. Titanium is selected for the basic conical shell structure since it is easily fabricated by proven techniques and possesses a measure of protection against local hot spots which might occur during re-entry.

A basic conical skin gage of 0.016 A-110AT is required to withstand internal cabin pressurization (15 psi ultimate) and to carry the flexural shear stresses resulting from launch and abort loads. Longitudinal structural elements are 0.032 A-110AT hat sections designed to withstand overall bending and axial loads of the launch and abort conditions, combined with external collapsing pressures occurring during a tumbling abort at maximum dynamic pressure. External pressure is not relieved by cabin pressurization. Machined titanium rings and framing members for cut-outs are welded to the conical skin, providing a pressure seal with a minimum dependence on gasket type seals or sealing compounds. Established fabrication procedures permit the use of minimum gage structures with no loss of strength due to welding.

The maximum stringer spacing is 8 inches. Ablation panels, consisting of an ablative layer bonded to single-faced double skin corrugations of 0.008 titanium, span the distance between stringers. Low density insulation between the ablation panels and the structural shell, plus Min-K insulation at the panel support points, maintain reasonable cabin temperatures. The large pressure bulkhead is an aluminum honeycomb sandwich structure designed to redistribute the uniform crushing pressures

3.6.2 (Continued)

from the trussgrid material used for earth impact attenuation. In order to withstand cabin pressurization loads, the plane small pressure bulkhead is made of two .010 titanium sheets, one beaded, and is additionally strengthened by angle stiffeners. Pressurization loads are carried as bending in the beaded webs and stiffeners.

Command module structure is illustrated in Figure 3-26.

3.6.3 Service Module - Constructed of 0.020 beryllium sheet, but otherwise conventional in structural arrangement with skins, stringers and rings, the conical shell functions both as primary structure and radiator surface. Coolant tubes of 1100-S aluminum extrusions are riveted to the beryllium stringers with Eccobond adhesive between the faying surfaces for increased thermal conductivity. Beryllium is chosen to withstand the launch temperatures and because of its high thermal conductivity.

Internal aluminum tube trusses, insulated from the hot structural shell, support equipment, fuel and propulsion units.

Figure 3-27 shows the structural configuration of the service module.

3.6.4 Terminal Landing Module - Corrugated 0.010 titanium over an inner 0.010 titanium skin form the primary structural shell which is designed primarily by launch conditions. Formed titanium rings stiffen the shell.

Lunar landing gear loads are distributed to the structural shell by an internal trusswork of aluminum tubes which also supports the terminal landing engine, propellant system and scientific equipment. Machined titanium rings at each end of the module provide for manufacturing splices and separation devices.

The terminal landing module structure is shown in Figure 3-28 and the landing gear in Figures 3-29 and 3-30.

COMMAND MODULE
STRUCTURAL ASSEMBLY

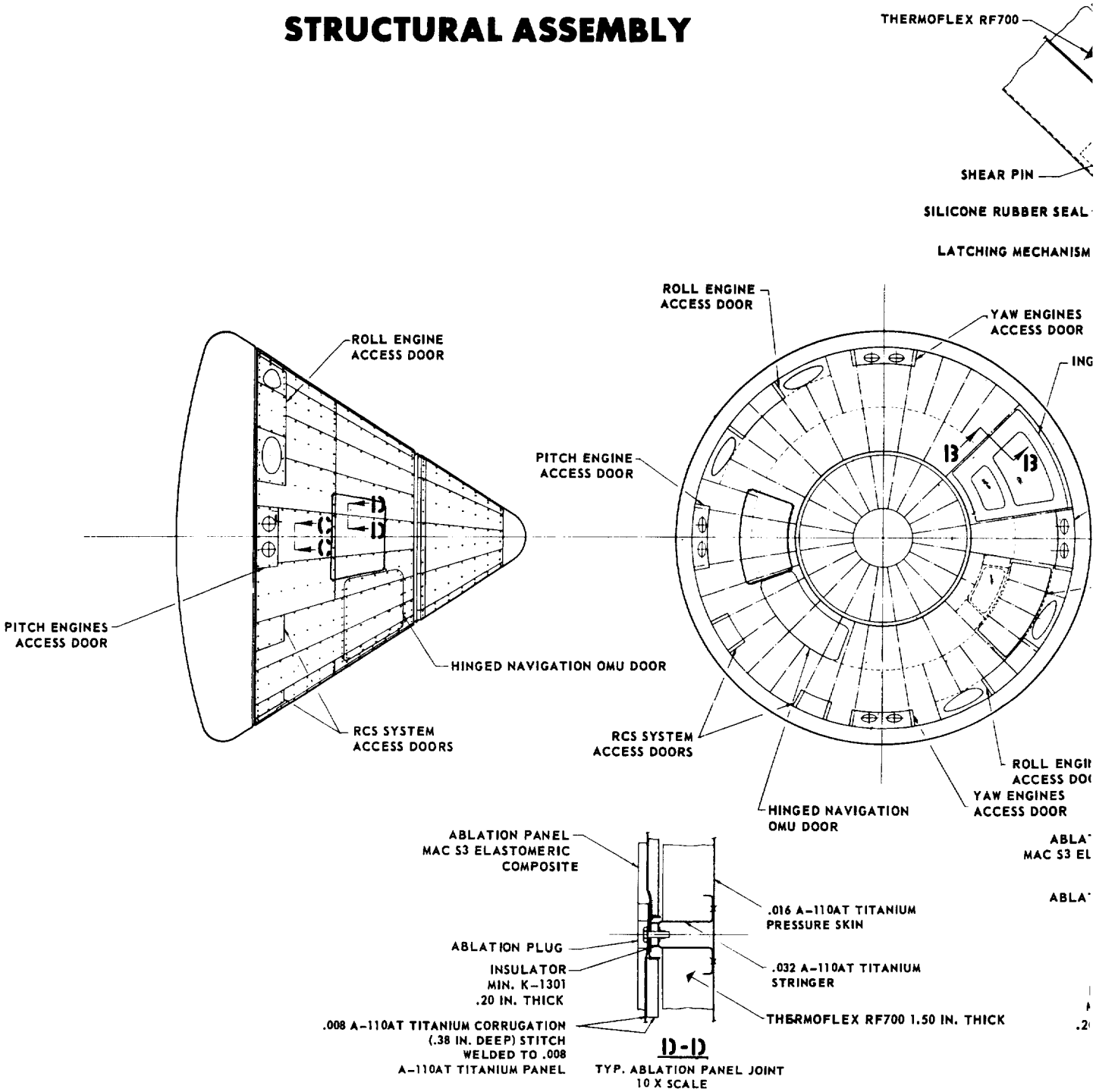
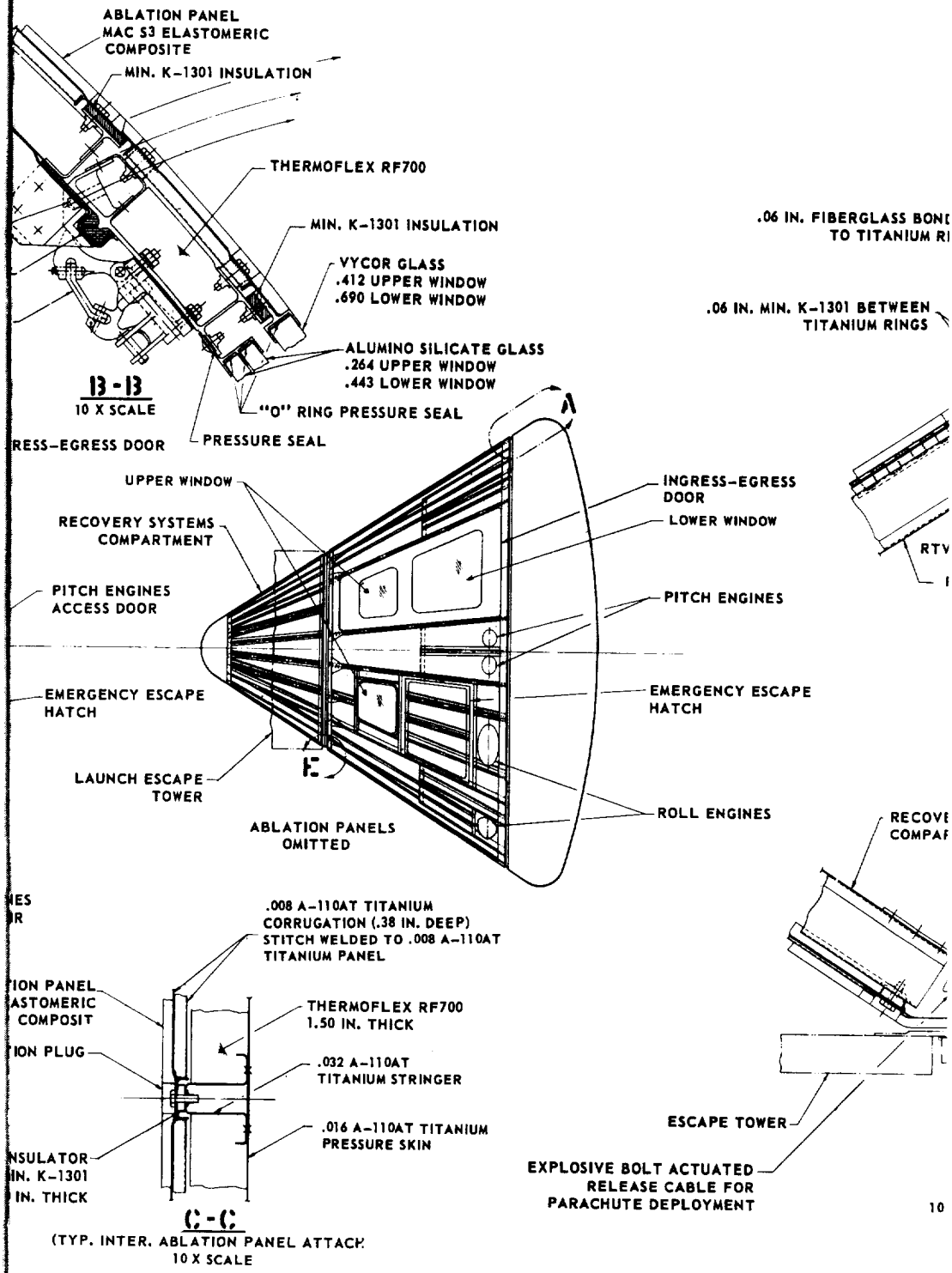


FIGURE 3-26

FOLDOUT FRAME 1



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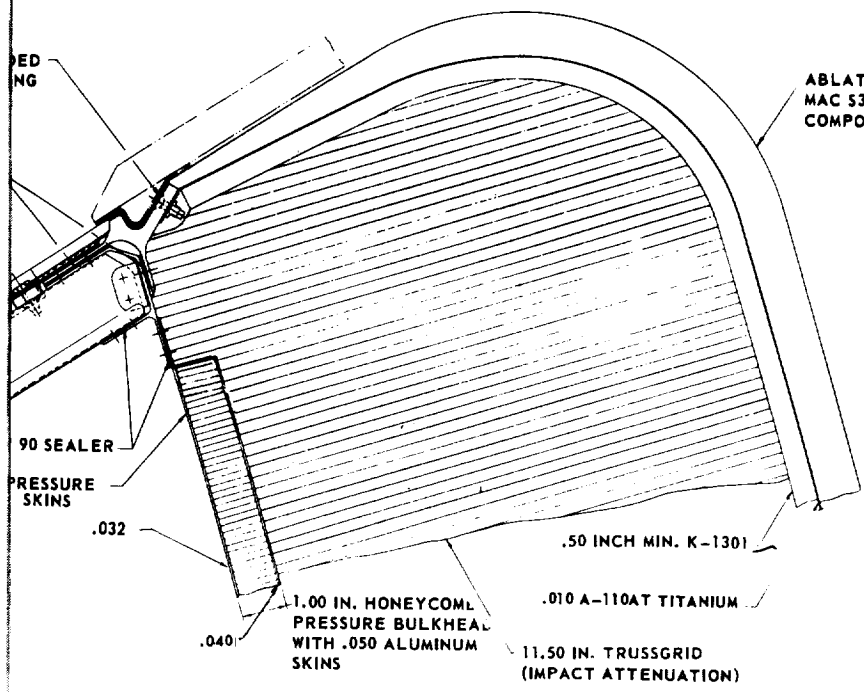

A
10 X SCALE

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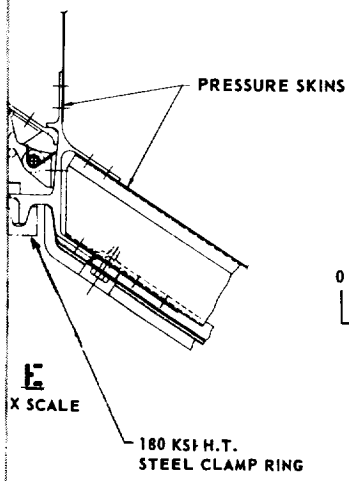
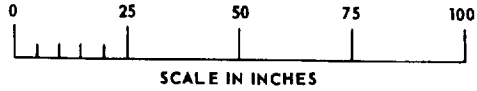
RTMENT

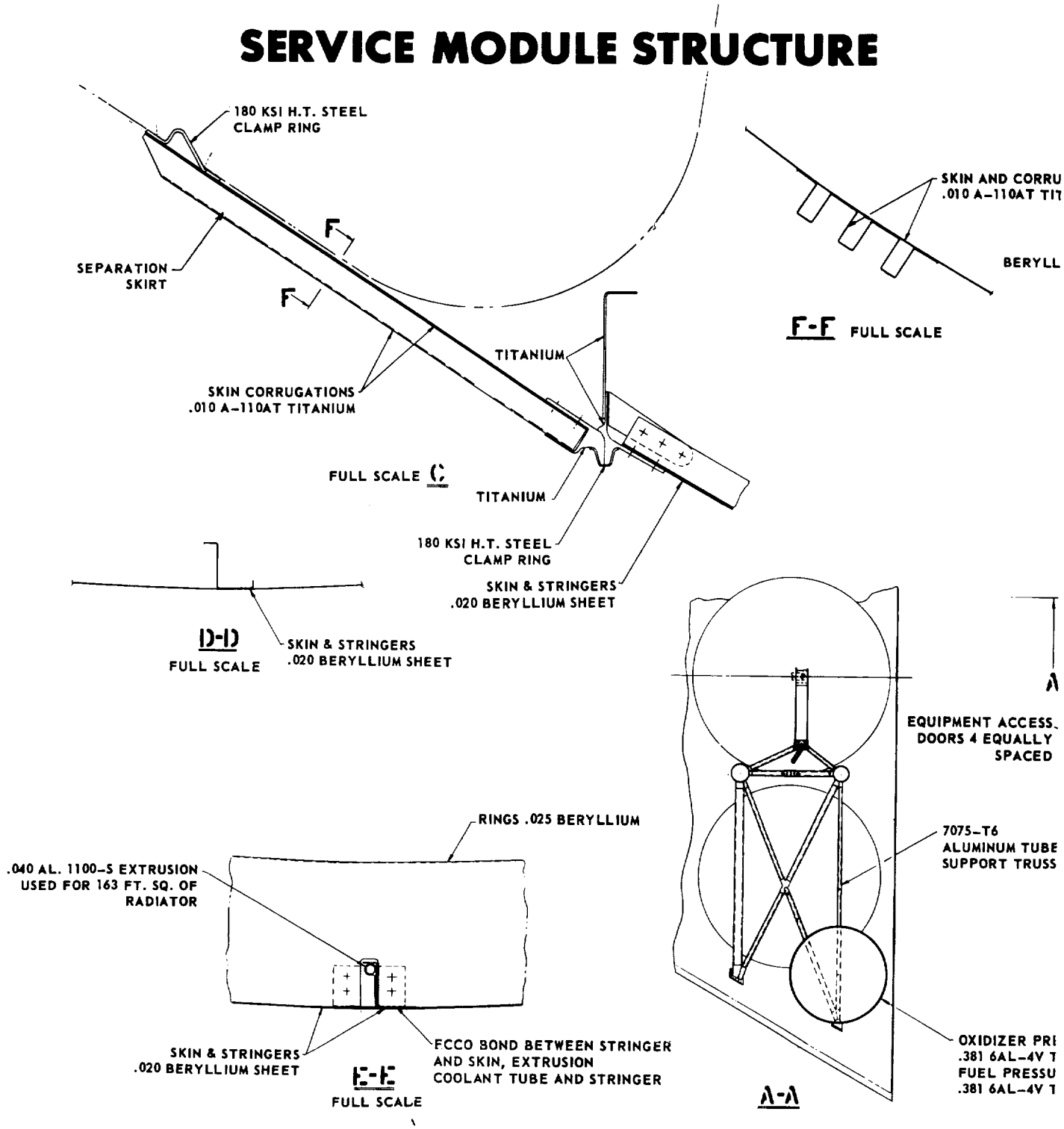


E
X SCALE



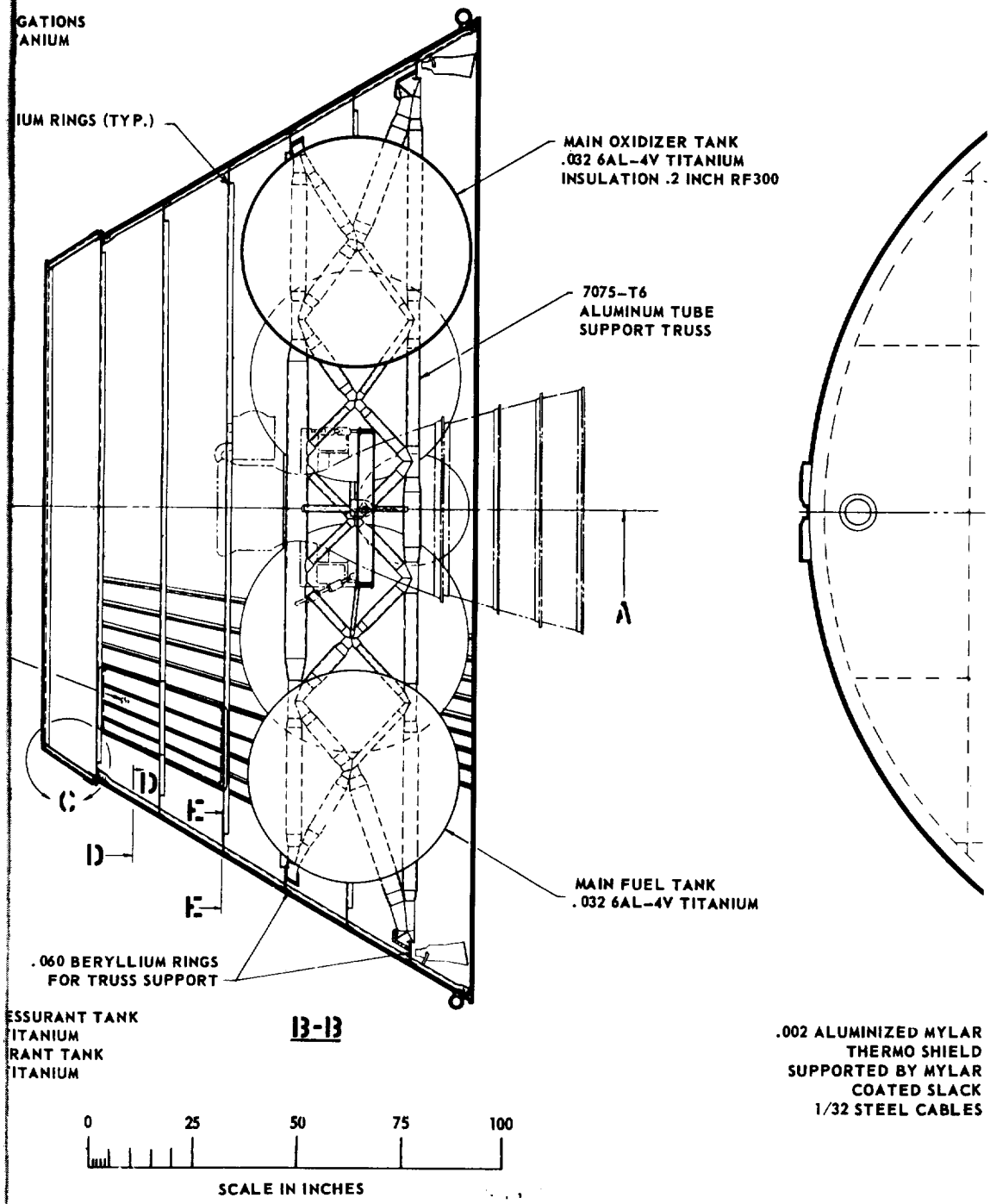
FOLDOUT FRAME 3

SERVICE MODULE STRUCTURE



FOLDOUT FRAME 1

FIGURE 3-27



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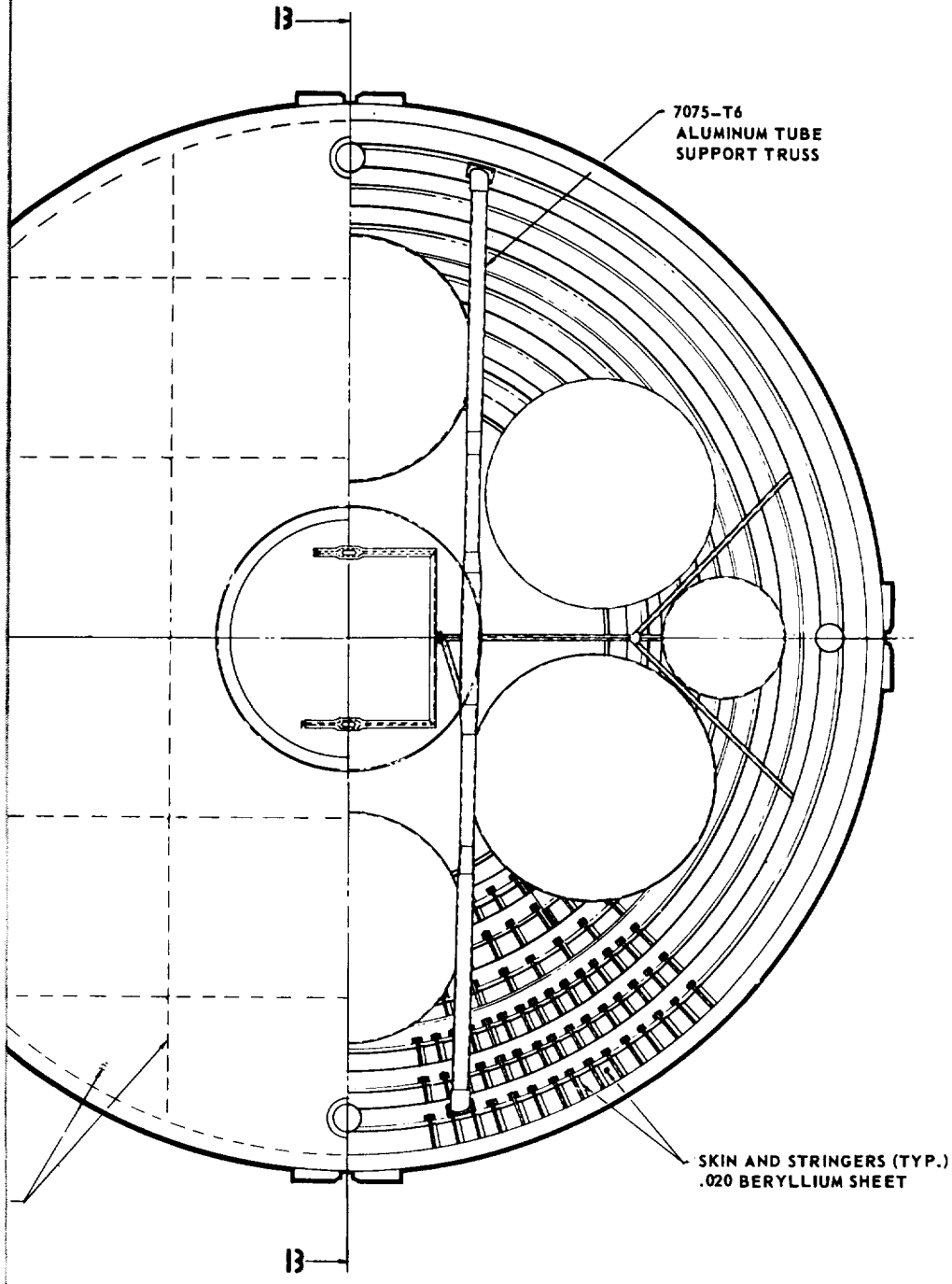
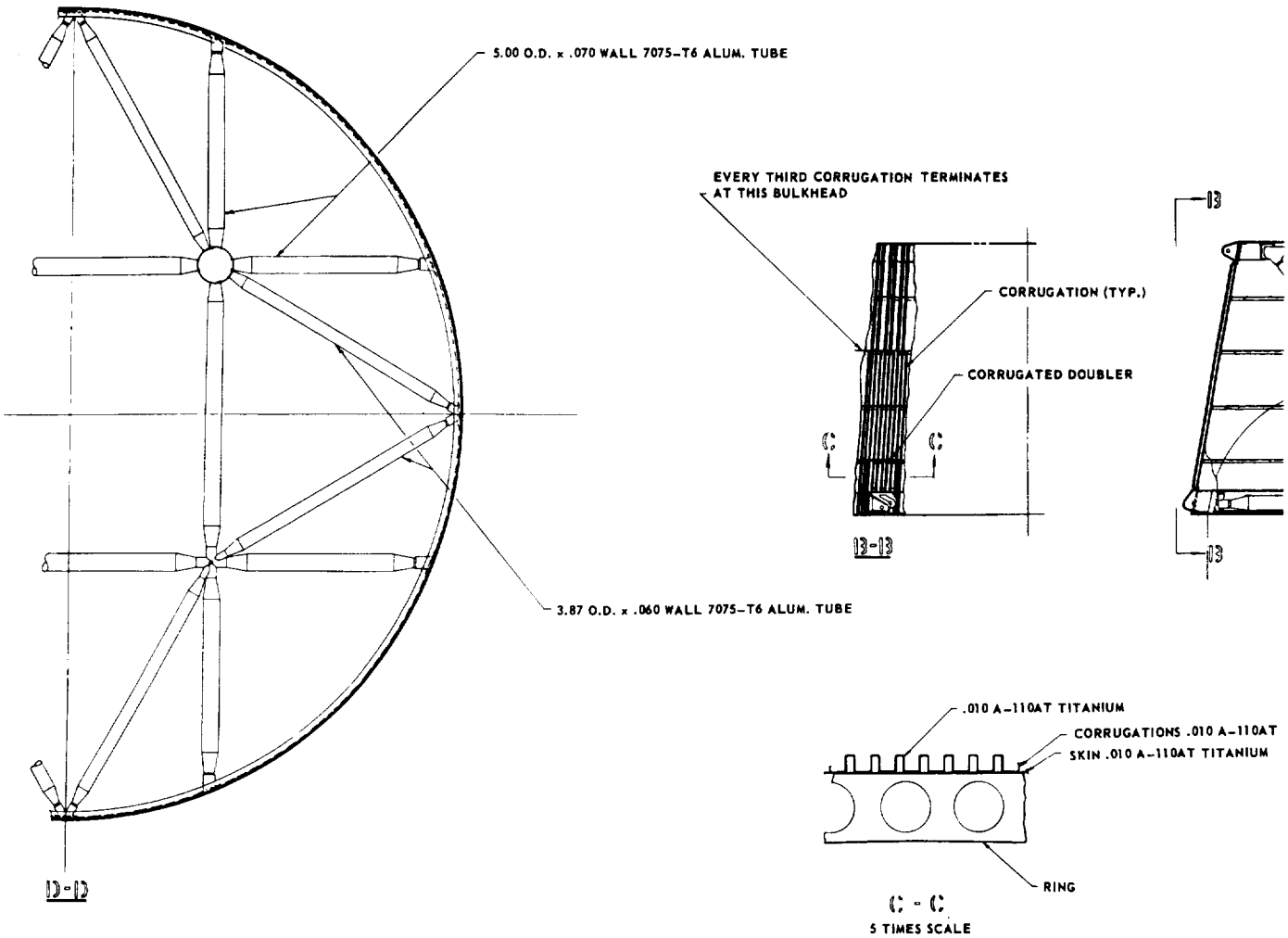


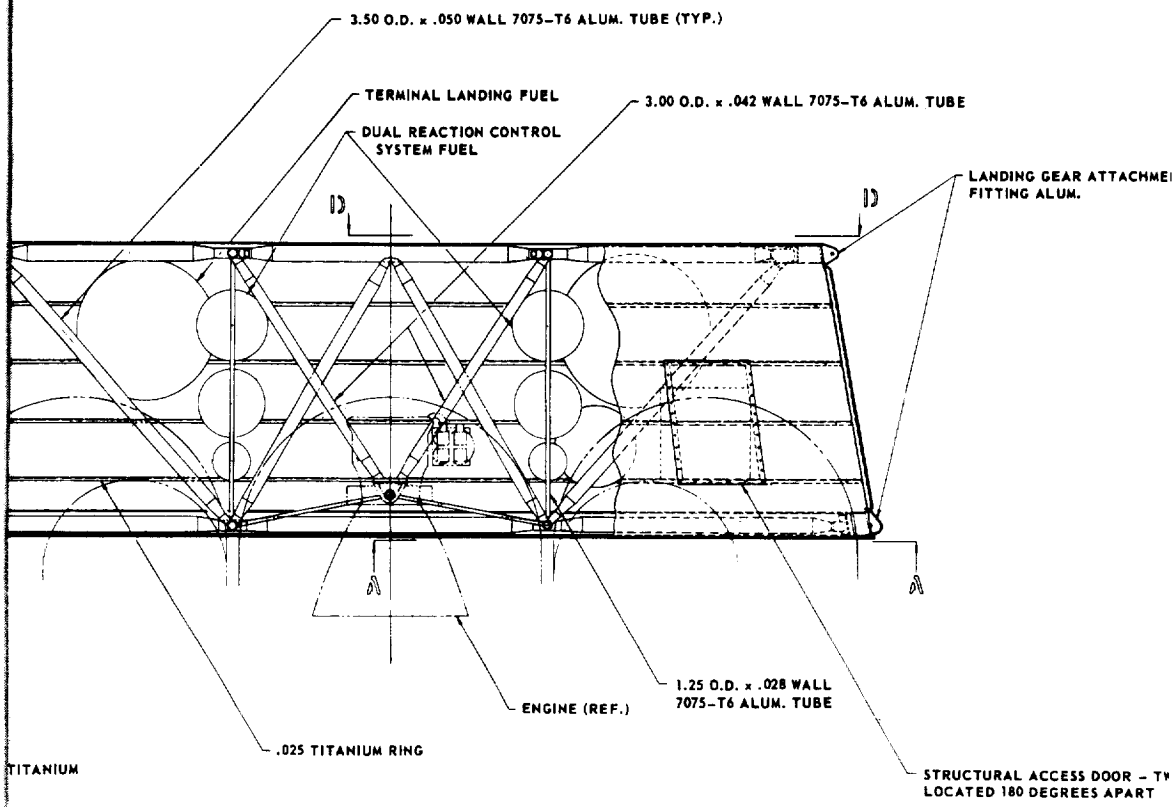
FOLDOUT FRAME 3

TERMINAL LANDING MODULE STRUCTURAL ASSEMBLY

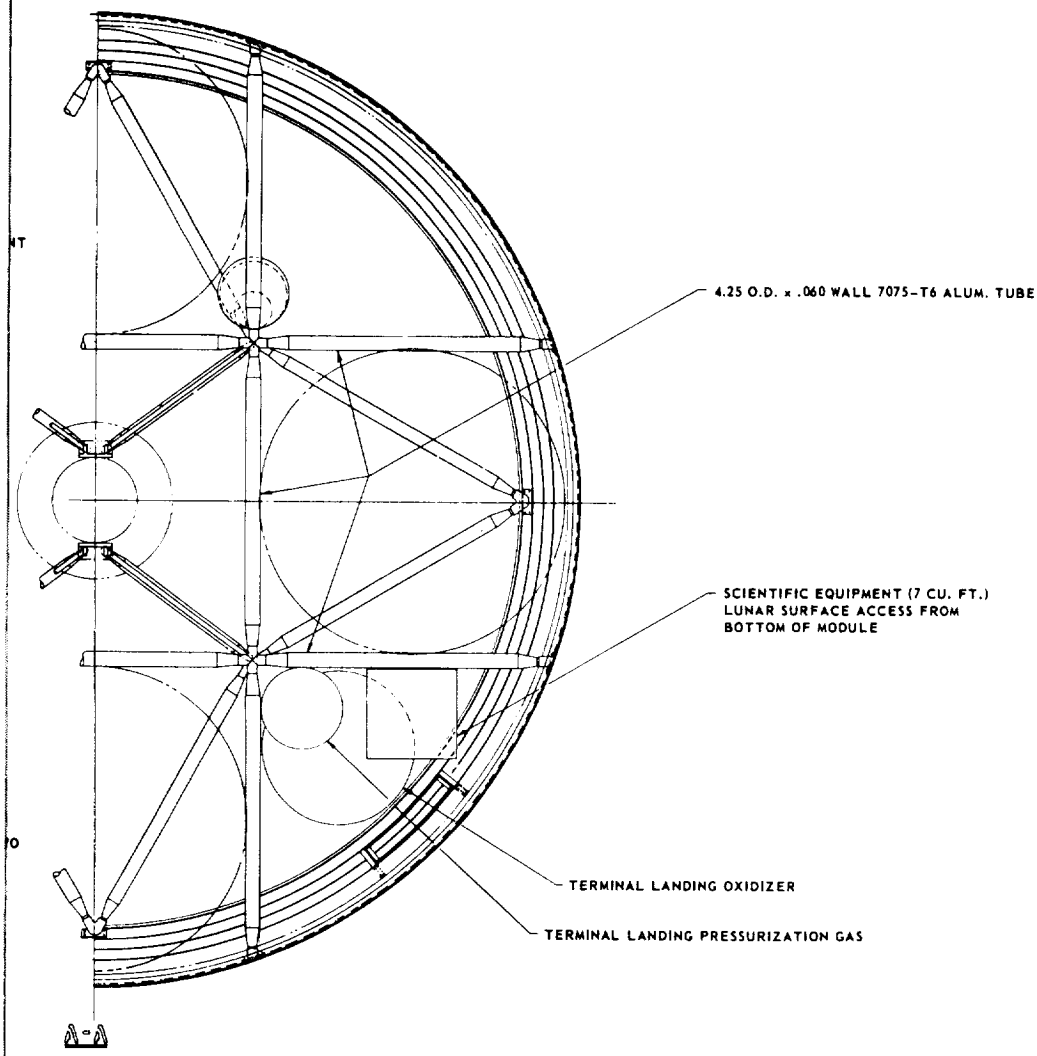


FOLDOUT FRAME 4

FIGURE 3-28



FOLDOUT FRAME 5



FOLDOUT FRAME 6

LUNAR LANDING GEAR
AND BACK-UP STRUCTURE

GEAR SHOWN FULLY COMPRESSED

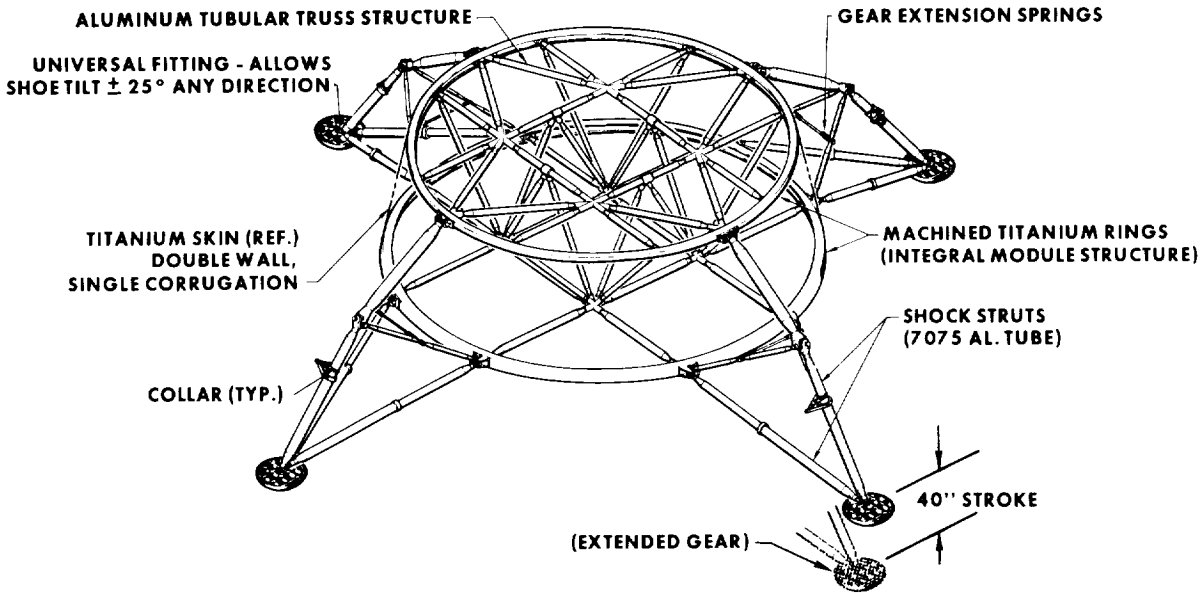


FIGURE 3–29

MAIN STRUT ASSEMBLY DETAIL
(SHOWN FULLY COMPRESSED)

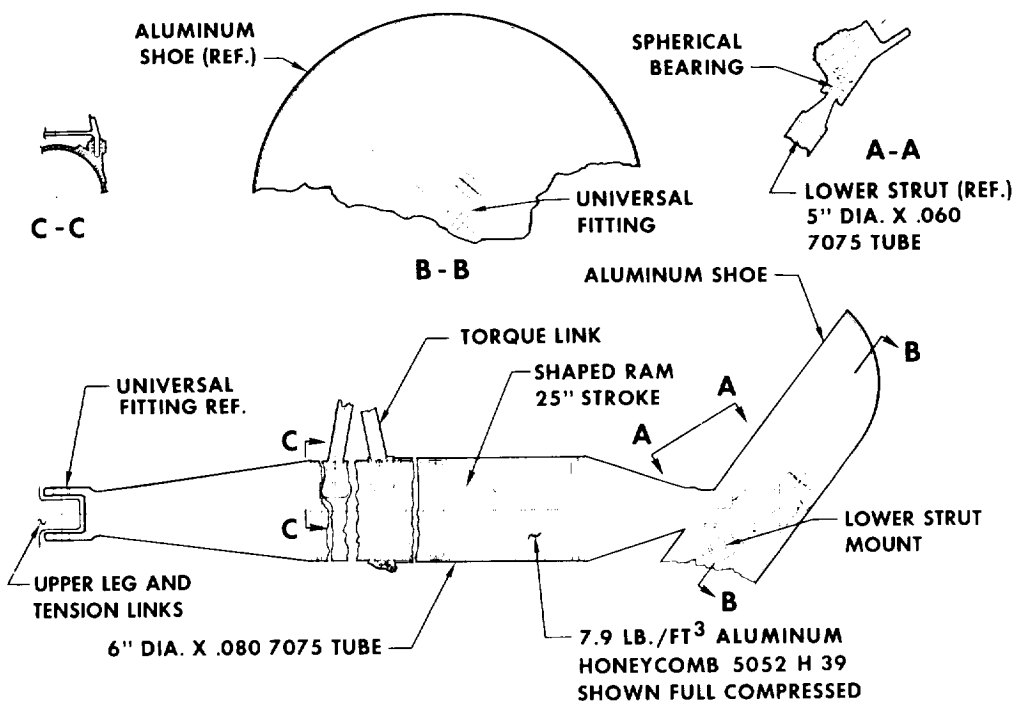


FIGURE 3–30

3.6.5 Retrograde Module - Launch loads determine the design of the cylindrical structural shell which consists of a corrugated 0.010 to 0.016 outer titanium skin over a smooth inner 0.010 to 0.016 titanium skin, supported by formed titanium rings. Fuel and oxidizer tanks and the thrust chamber are supported on an aluminum tube truss arrangement at the lower end and an aluminum honeycomb sandwich bulkhead at the upper end. Titanium machined rings are employed at each end to provide for manufacturing splices, attachment to the booster and separation devices. The retrograde module structure is shown in Figure 3-31.

3.6.6 Separation Joints - The escape tower is attached to the command module by a three segment clamp ring, similar to the Project Mercury design.

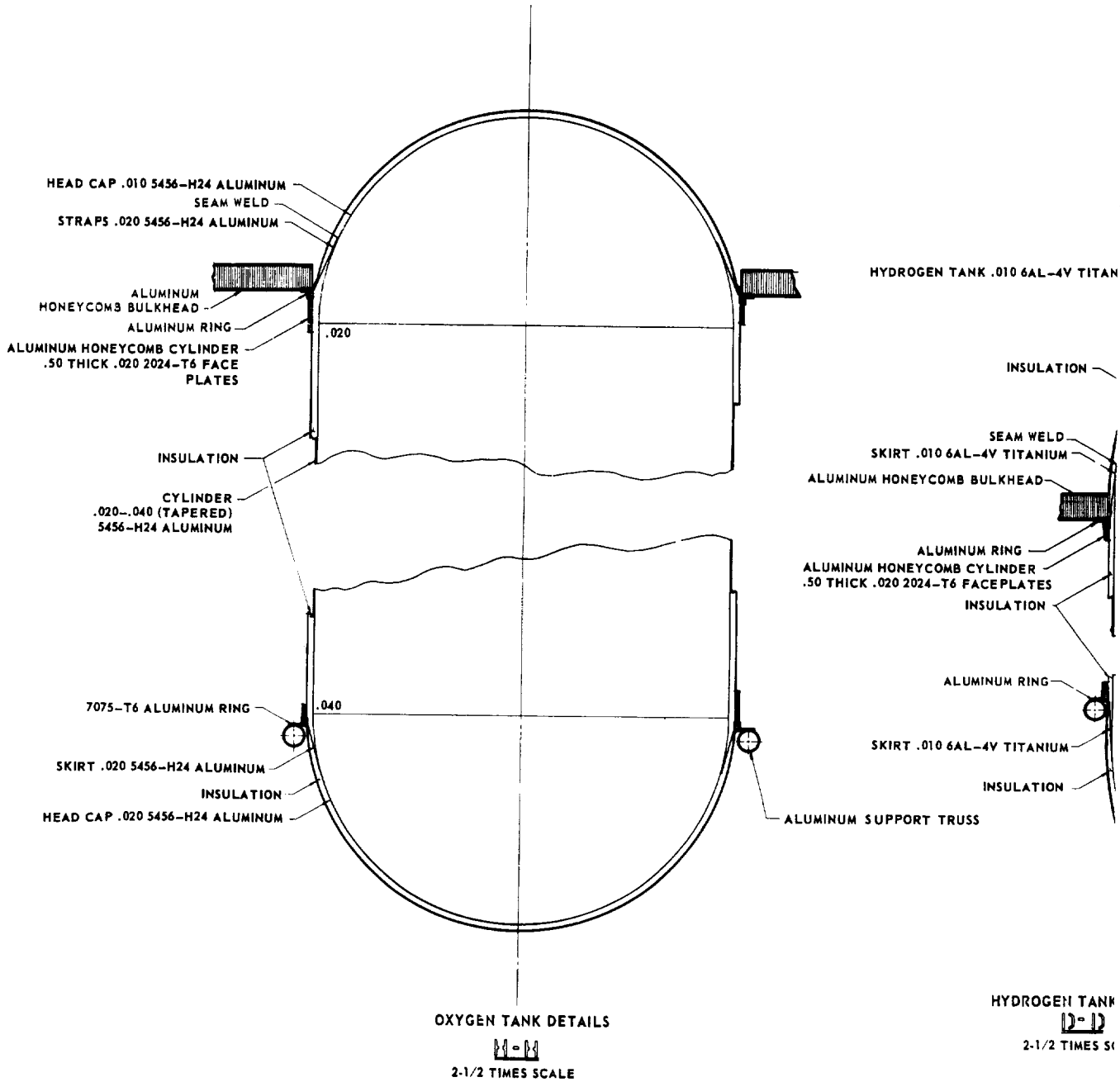
A three segment structural skirt between the command module and service module bridges across the front face heat shield and earth impact attenuation system. Attached to either module by a three segment clamp ring, the segmented skirt permits separation by falling away from the large diameter heat shield.

The remaining module separation joints are dual flexible linear shaped charges (FLSC) similar to the configurations developed for Project Gemini. The structure cut by the FLSC will be single thickness mold line elements of the machined match rings.

3.6.7 Tankage - Titanium 6AL4V is the material selected for most of the tankage systems. Exceptions are the use of high strength alloy steel (180,000 psi heat treat) for the service module ECS and fuel cell oxygen, and the use of 5456 aluminum for the liquid oxygen tanks in the retrograde system. A burst pressure factor of 2 is used for all tankage except the main propellant systems and RCS propellant tankage in the service and terminal landing modules for which a burst factor of 1.5 is used. Because all tankage is isolated from heat inputs, room temperature allowables are used in all but the cryogenic tanks, where advantage is taken of the increased strength at extremely low temperatures.

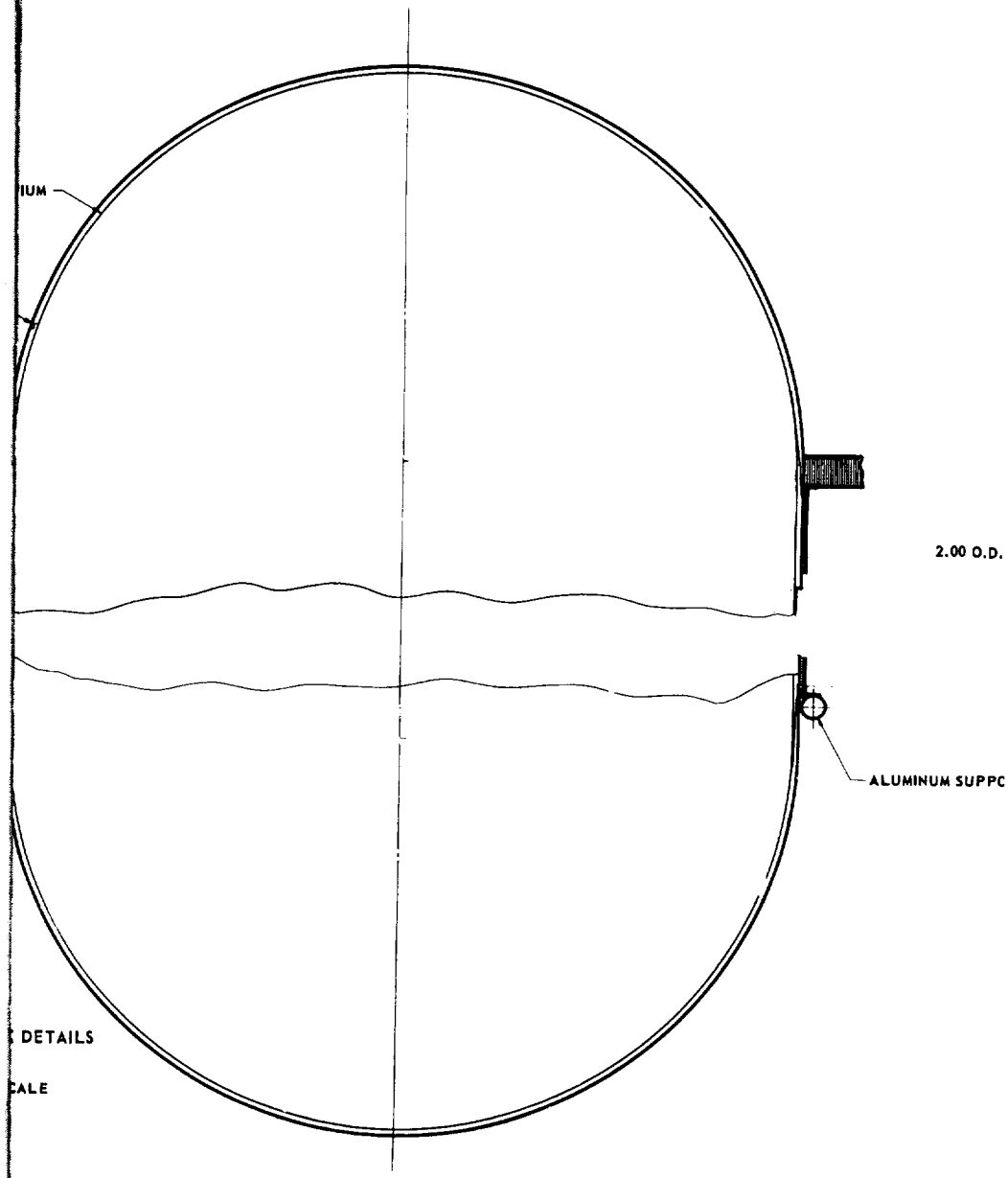
Ⓐ

RETROGRADE MODULE
STRUCTURAL ASSEMBLY



FOLDOUT FRAME

FIGURE 3-31



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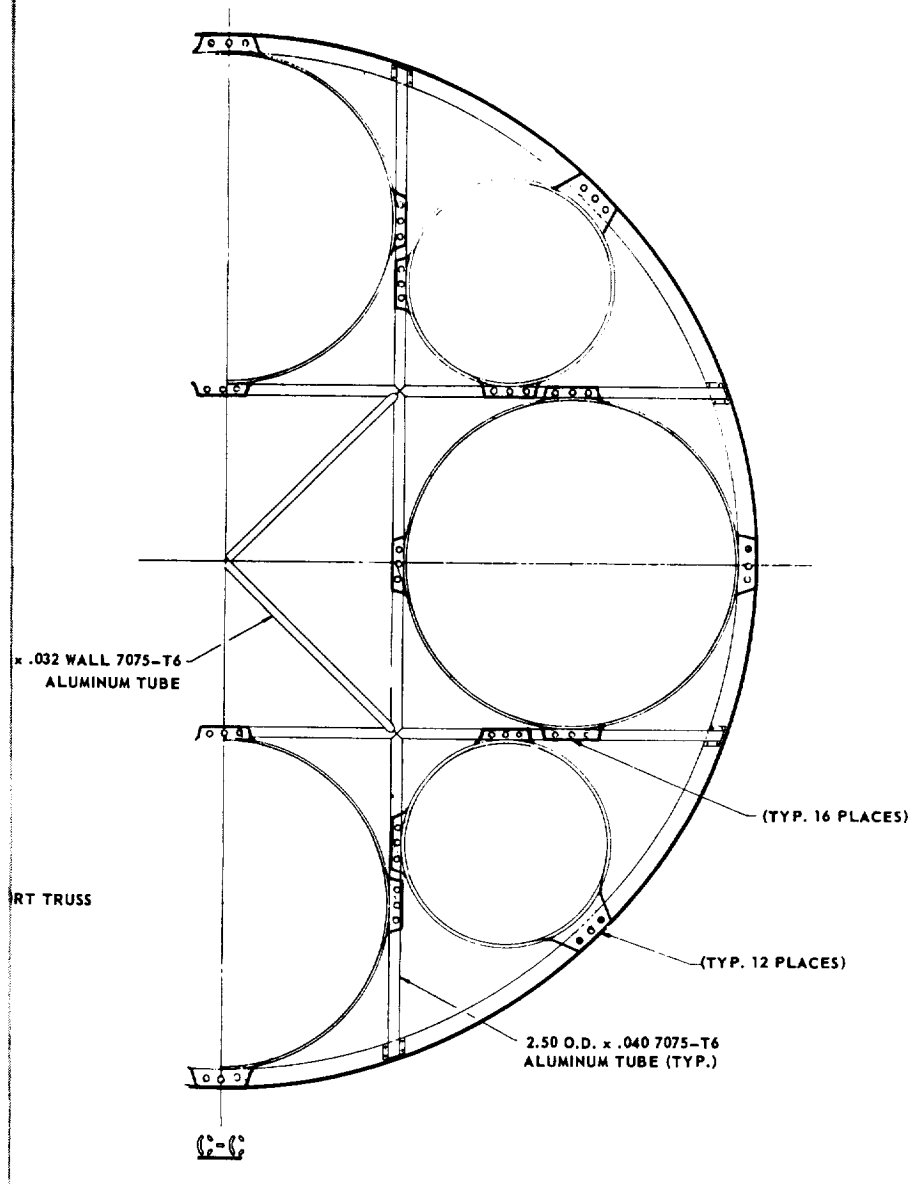

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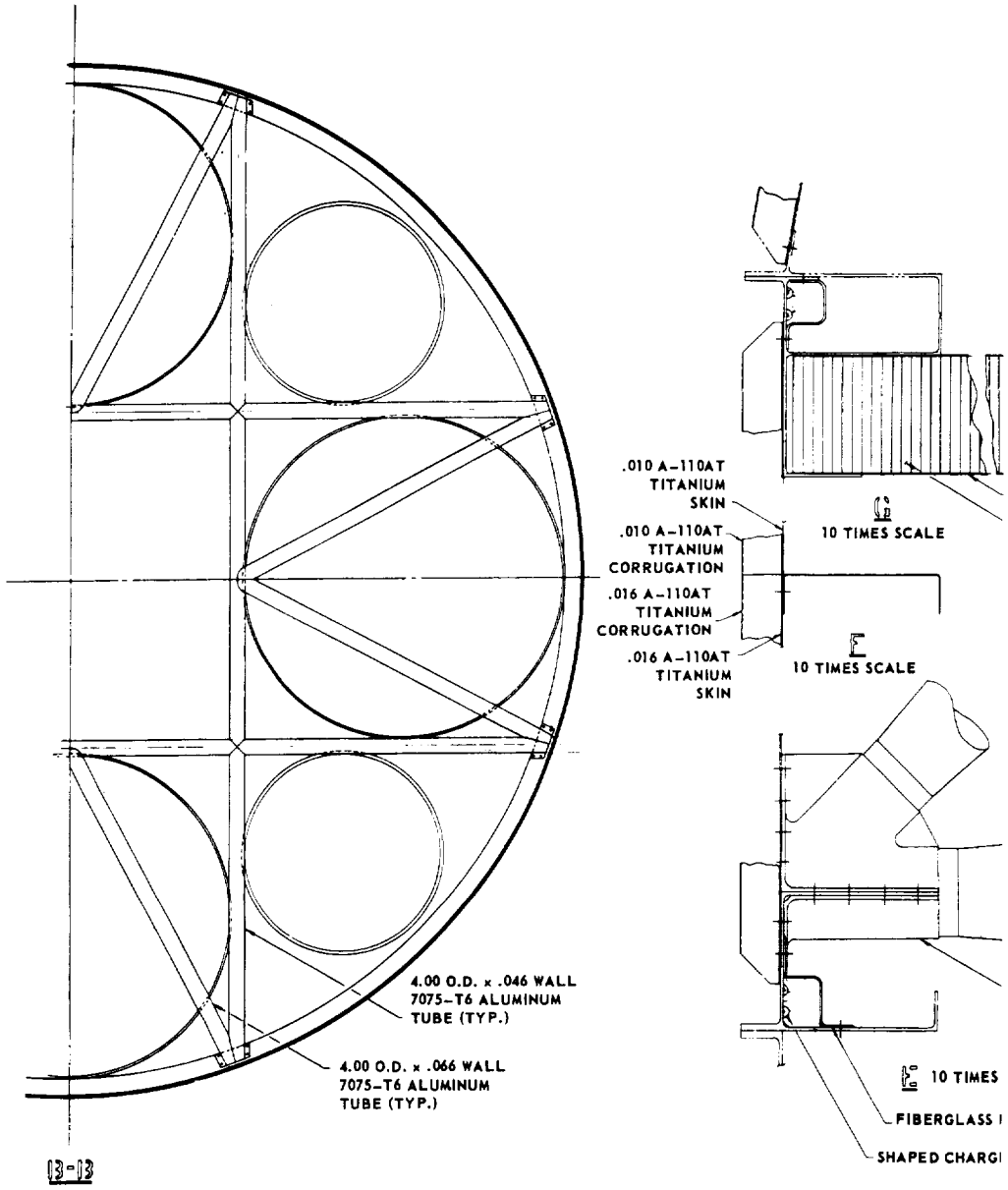
RETROGRADE MODULE
STRUCTURAL ASSEMBLY



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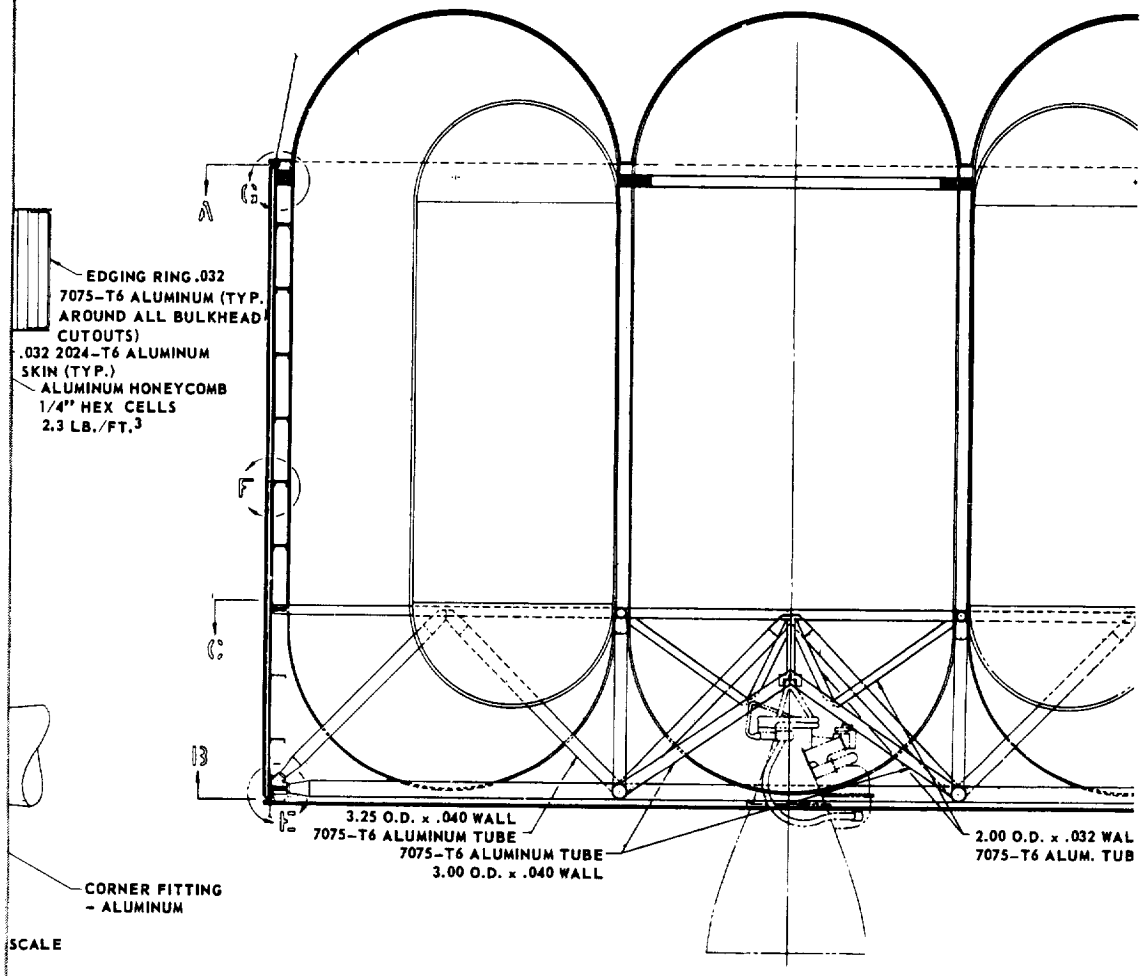

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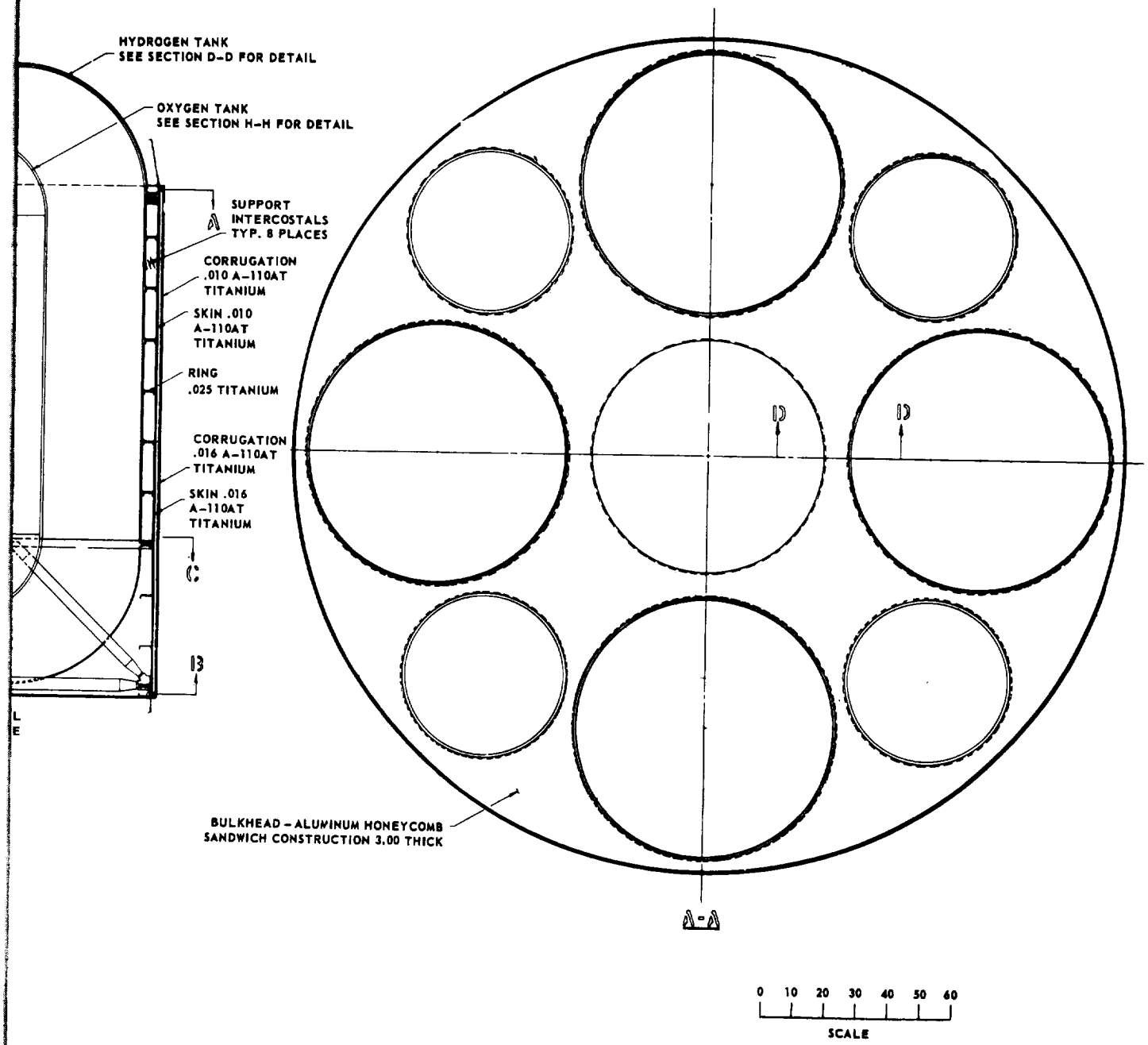


FOLDOUT FRAME 6

FIGURE 3-31

3.7 Heat Protection - The external heat protection system of the command module is conservatively designed for a long range polar re-entry. The following paragraphs contain a discussion of the methods used in establishing the heat protection design.

3.7.1 Launch - Structural temperatures are determined for critical areas using the Eckert reference temperature method of Reference 3-28, assuming a local transition Reynolds number of 100,000. Escape tower legs are covered with an ablative material as required, based on Project Mercury experience. In order to obtain a high efficiency of the Project Gemini type skin-stringer radiator design, and to provide adequate strength at the temperatures encountered during launch, the service module external skins and stringers are made of beryllium. The radiators are coated to provide an emissivity of 0.8 or greater, which results in a predicted maximum temperature of 800°F during launch.

3.7.2 Space Flight - External heat loads are determined from heat inputs based on solar, earth and moon reflected and emitted radiation, spacecraft orientation, surface emissivity and solar absorptivity. External temperatures are controlled by surface coatings. These coatings, discussed in Section 3.3, provide passive environmental control of some of the equipment located inside the service, terminal landing, and retrograde modules.

3.7.3 Re-entry - Re-entry heat loads are greatly influenced by the amount of longitudinal range desired and the initial re-entry angle. The convective heat load becomes greater with increasing range or increasing inclination angle of the re-entry, as shown by Figure 3-32. Gaseous radiative heat loads increase as the initial re-entry angle or inclination angle of the re-entry increases. The selected re-entry heat protection design condition is a shallow, 9000 nautical mile range re-entry having a 90° inclination angle. The steep re-entry, with high gaseous

3.7.3 (Continued)

RE-ENTRY HEAT LOADS

- TRAJECTORIES BASED ON W/C_DA=50.9 PSF & L/D=.5 (ROLL MODULATED)
- HEAT LOAD BASED ON STAGNATION POINT OF 16.7 FT. RADIUS HEMISPHERE
- i=INCLINATION ANGLE OF RE-ENTRY

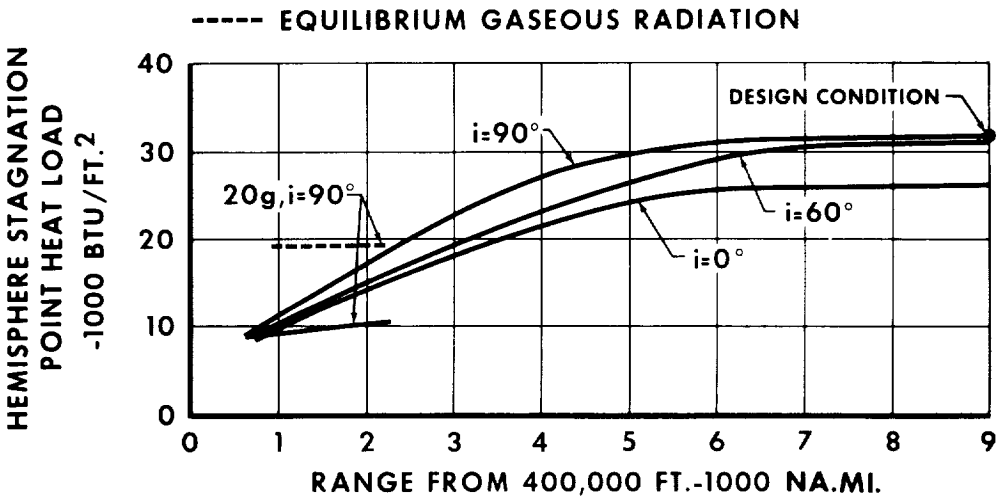


FIGURE 3-32

radiation heating, is not a critical design condition. Atmospheric variations increase the heat load less than 15% above the design condition. Re-entry heat loads following an abort do not exceed the design limits.

A. Re-entry Heat Loads - Stagnation point convective and gaseous radiative heating rates are calculated for a hemisphere with a radius equal to the radius of curvature of the blunt front face. The convective heating rates are computed by the Fay and Riddell method of Reference 3-29 for a 1000°R surface temperature. The equilibrium gaseous radiative heating rates are computed from Avco tabulated data found in Reference 3-30. The non-equilibrium gaseous radiative heating rates are estimated from available NASA-Ames tests data, (References 3-31 and 3-32), with no vehicle size factor included. The various heating rate histories for the design re-entry condition are shown on Figure 3-33. Figure 3-34 presents the heating rate histories for the steep (20 "g") re-entry.

HEATING RATE HISTORIES
RANGE = 9000 NA.MI.

- TRAJECTORY BASED ON W/C_DA = 50.9 PSF & L/D = .5 (ROLL MODULATED)
- HEATING RATES BASED ON STAGNATION POINT OF 16.7 FT. RADIUS HEMISPHERE
- 90° INCLINATION ANGLE OF RE-ENTRY

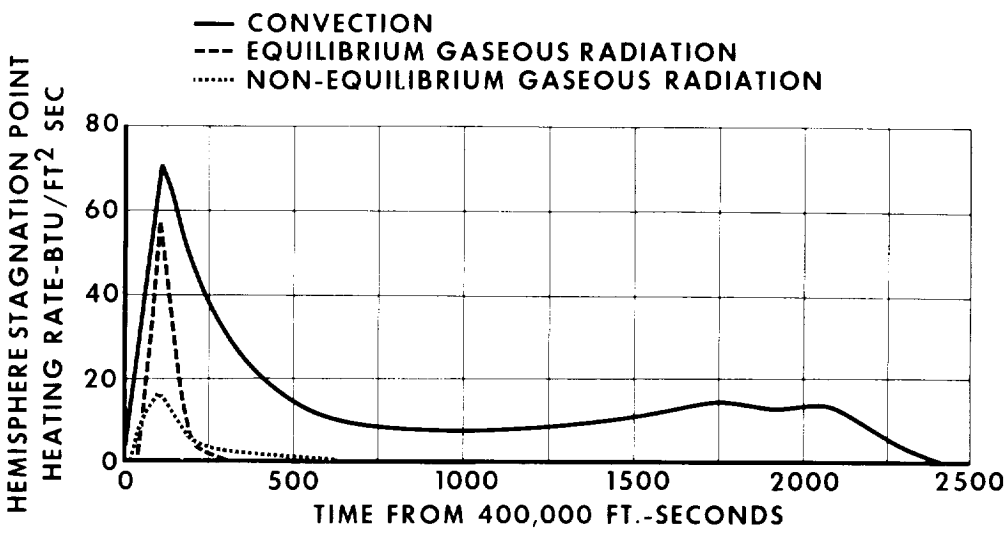


FIGURE 3-33

HEATING RATE HISTORIES
20_g RE-ENTRY

- TRAJECTORY BASED ON W/C_DA = 50.9 PSF & L/D = .5
- HEATING RATES BASED ON STAGNATION POINT OF 16.7 FT. RADIUS HEMISPHERE
- 90° INCLINATION ANGLE OF RE-ENTRY

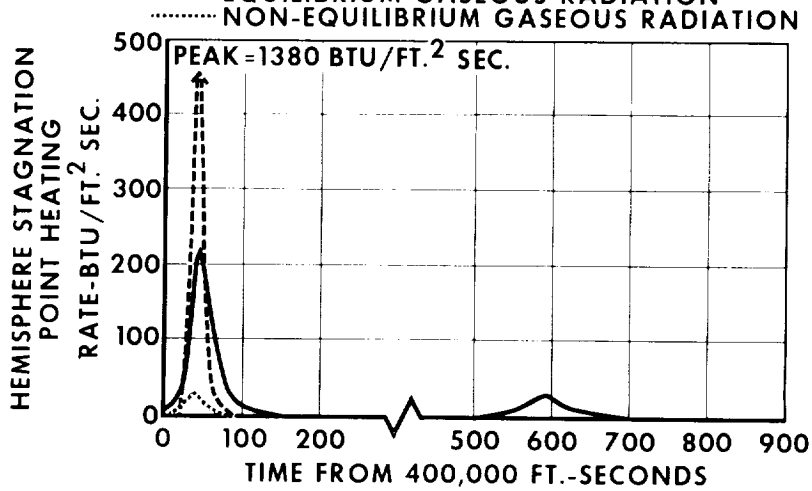


FIGURE 3-34

3.7.3 (Continued)

The convective heating distribution of the front face, presented in Figure 3-35, is determined from available wind tunnel test data (M.A.C., Cornell, NASA and others) in conjunction with the analytical method of Rose, Kemp and Detra (Reference 3-33). The heat load increase due to a

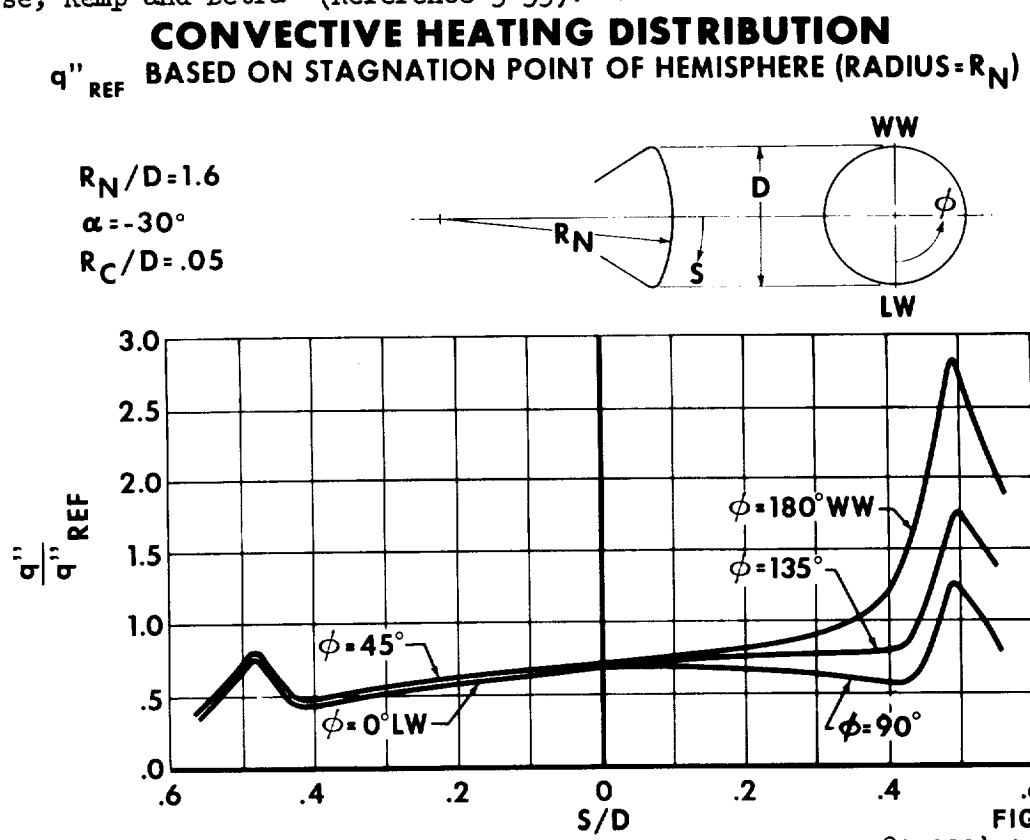


FIGURE 3-35

turbulent boundary layer (transition Reynolds number of 180,000) is negligible. The afterbody convective heating distribution, shown in Figure 3-36, is based on wind tunnel test results of the Apollo shape, and test results from Mercury and other blunt nose conical afterbody models. The windward side test results agree closely with theoretical laminar flow calculations.

Gaseous radiation distributions on the front face are based on shock shapes extended from those calculated by the method of Kaattari in Reference 3-34. Equilibrium radiation distributions are shown in Figure 3-37. These

ESTIMATED AFTERBODY CONVECTION DISTRIBUTION

q''_{REF} BASED ON STAGNATION POINT OF HEMISPHERE (RADIUS=R_N)

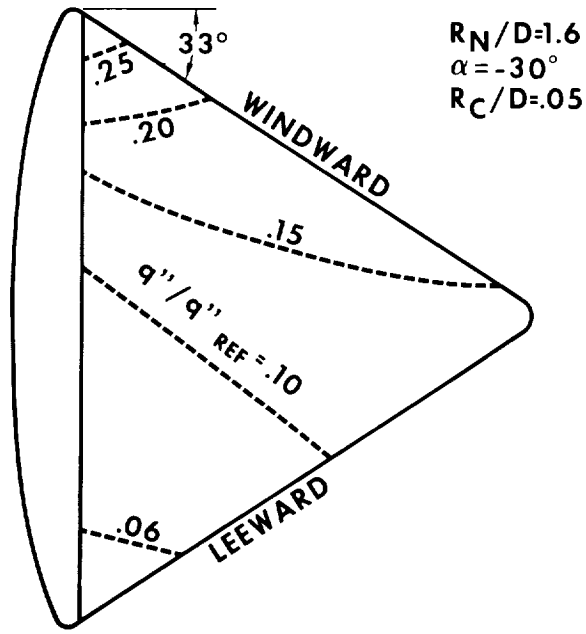


FIGURE 3-36

EQUILIBRIUM RADIATIVE HEATING DISTRIBUTION

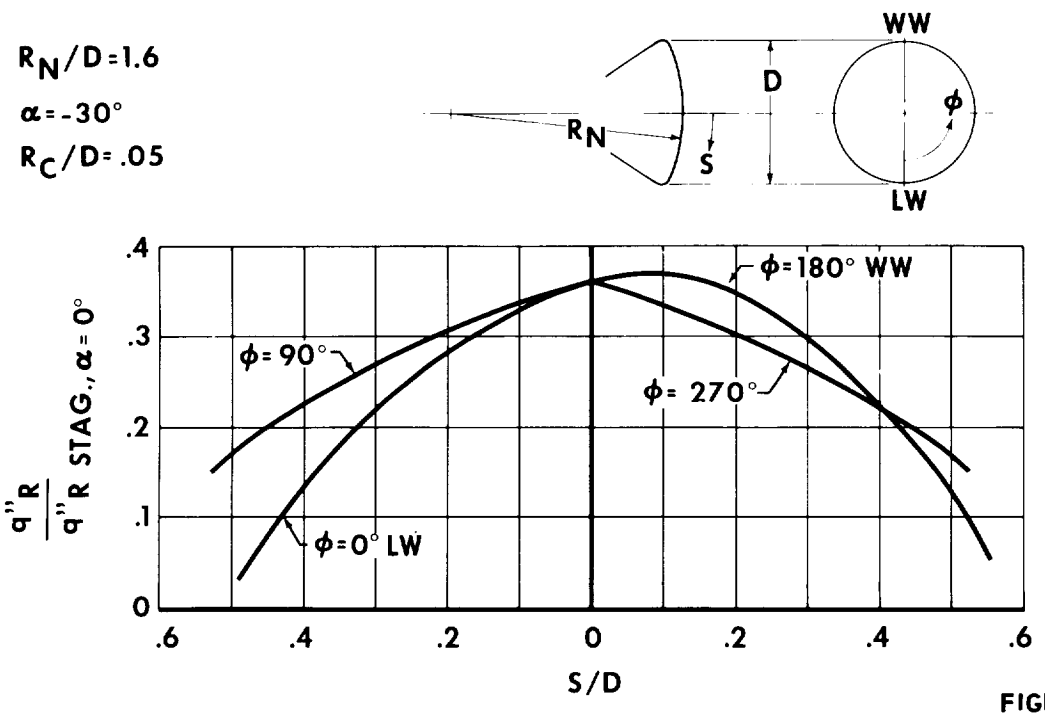


FIGURE 3-37

3.7.3 (Continued)

are determined by assuming a linear temperature distribution between the shock and body at a nominal superorbital velocity and altitude. The non-equilibrium radiation distribution is established by the velocity component normal to the shock.

B. Re-entry Heat Protection - The ablative material on the command module is M.A.C. Thermorad Shield S-3. This material is the same as used on the Project Gemini front face.

1. Ablative Material Selection - McDonnell has tested approximately 60 internally developed and commercially available materials. The most promising materials have been tested at all the test conditions shown in Figure 3-38. The high efficiency of the M.A.C. Thermorad Shield S-3 elastomeric composite is shown in Figure 3-39, which presents the results of a "standard" test. Many of the thermophysical and mechanical property tests have been completed for the Project Gemini re-entry environments.

2. Heat Protection Distribution and Weight - Command module ablative material distributions are determined with the following considerations:

   a. A 1.15 factor on predicted heating rates.

   b. Nominal thermophysical properties based on available test results.

   c. Material behavior in the combined convection and gaseous radiation environment based on information from NASA-Ames investigations.

The required front face ablative material distribution is shown in Figure 3-40 for the design re-entry heat load. The ablative material design thickness is adequate to provide a bondline temperature limit of 800°F. This is compatible with the titanium backup structure. The steep 20 "g" re-entry (largest gaseous radiation heat load) is not the ablative material

HEATING RATE AND ENTHALPY OF ABLATIVE MATERIAL TESTING

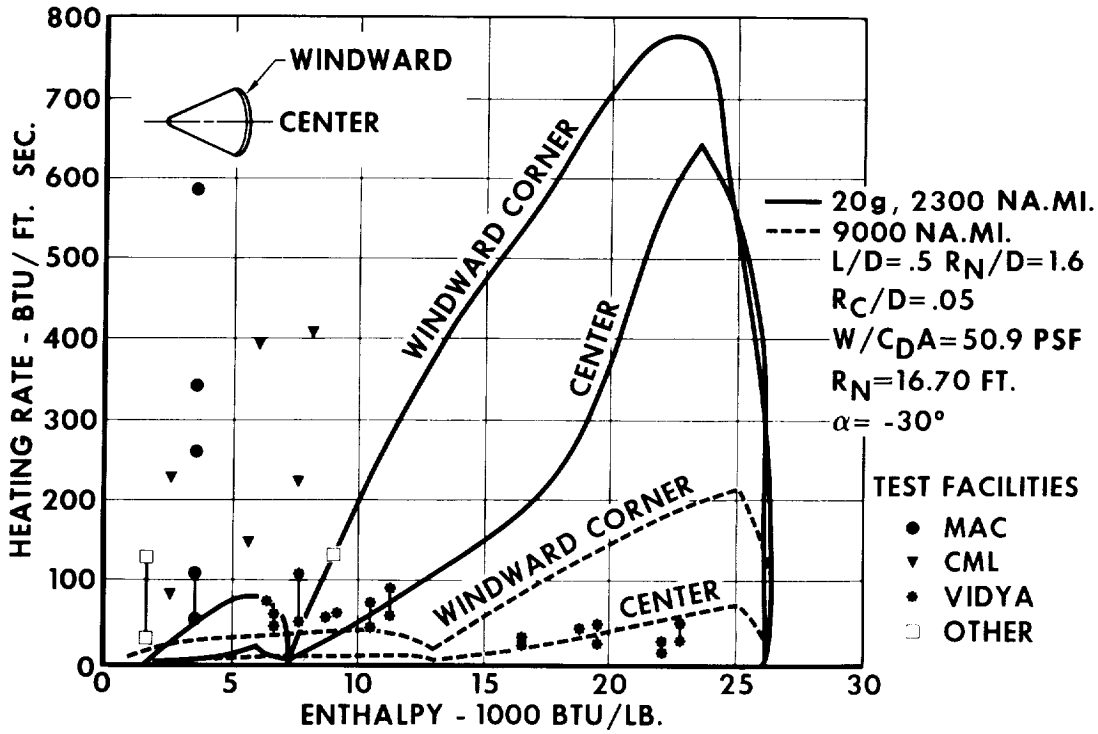


FIGURE 3-38

BONDLINE TEMPERATURE RISE FOR SEVERAL ABLATIVE MATERIALS

INITIAL WEIGHT = 4 PSF
MEAN HEATING RATE = 56.1 BTU/FT.^{2} SEC.
MEAN ENTHALPY = 3444 BTU/LB.

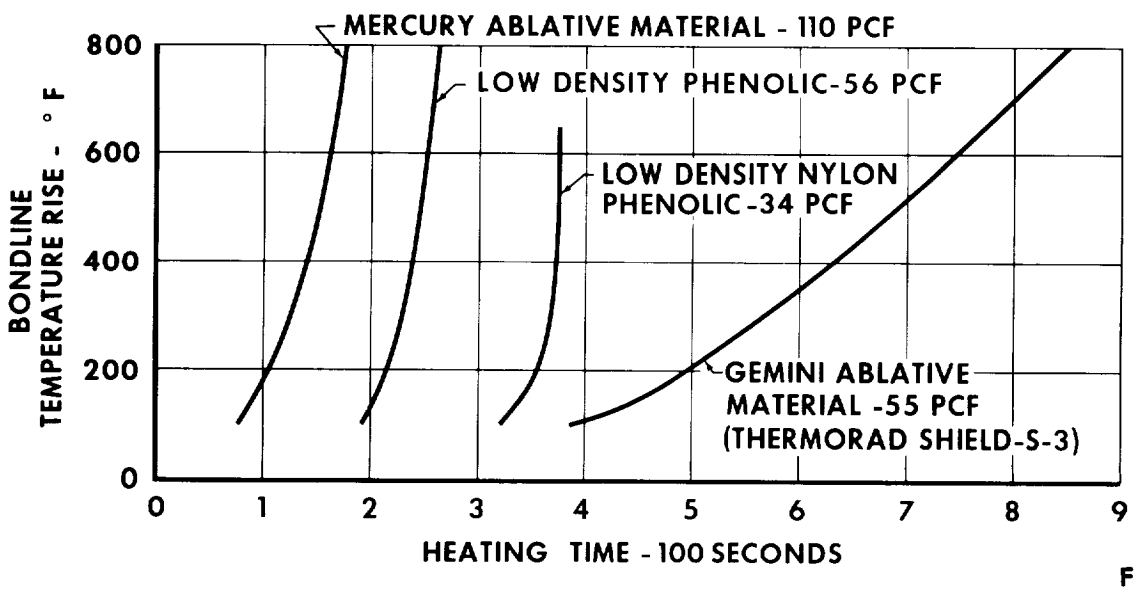


FIGURE 3-39

3.7.3 (Continued)

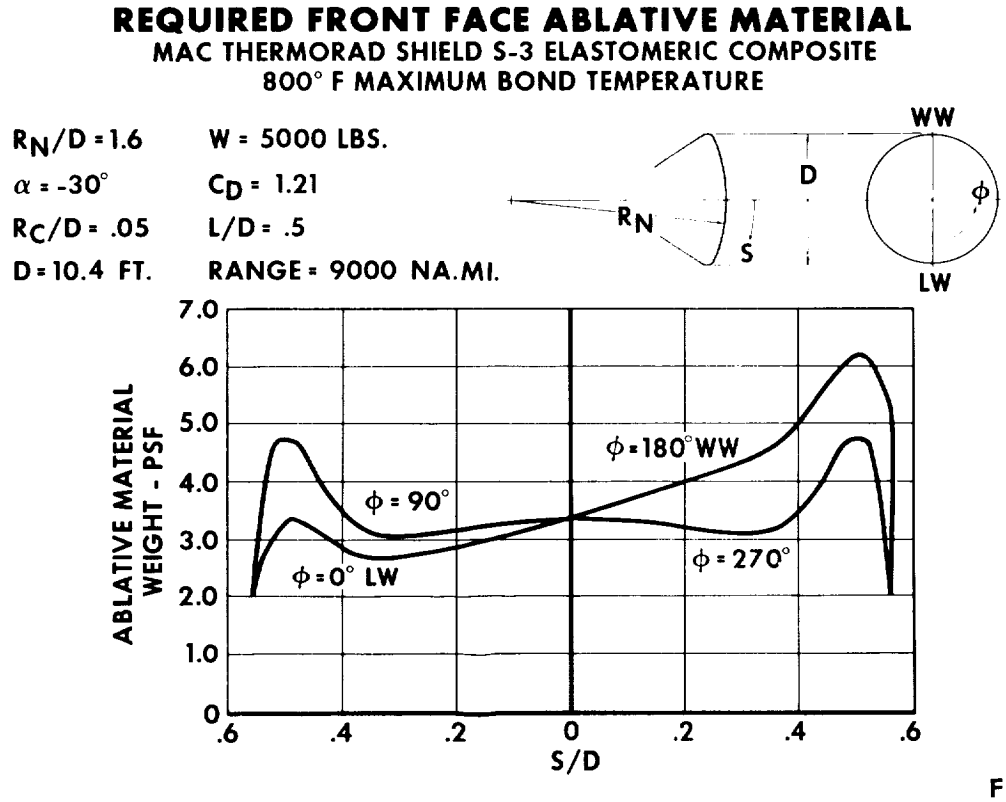


FIGURE 3-40

design condition. The ablation material thickness as shown in Figure 3-40 is the theoretical thickness distribution that is computed from the design heat input. In practice, the shield is designed with linear variations between various points on the shield. The actual design thickness is indicated at several points on the front face and afterbody in Figure 3-41. The shield weight is determined from the actual shield thickness variation.

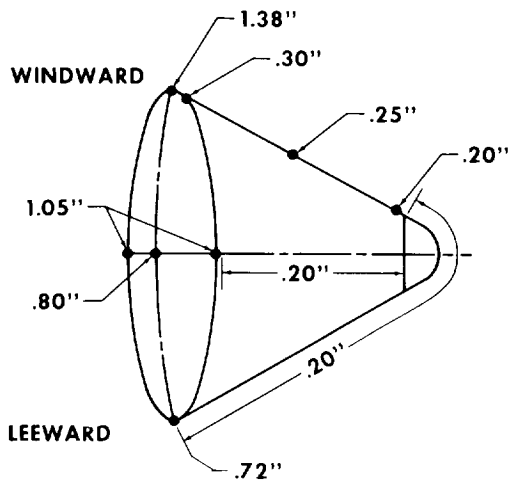
Figure 3-41 also presents a weight summary of the ablative material and insulation design. The 1/2 inch MIN-K 1301 limits the trussgrid impact attenuation material to a maximum design temperature of 300°F. The cabin environment is limited to a maximum temperature of 160°F during flight with 1 1/2 inches of Thermoflex insulation on the afterbody. Due to the insulation effect of the afterbody ablative material, the cabin environment

3.7.3 (Continued)

COMMAND MODULE

HEAT PROTECTION WEIGHT SUMMARY

ABLATIVE MATERIAL DESIGN



WEIGHT SUMMARY

FRONT FACE	
ABLATIVE MATERIAL (MAC THERMORAD SHIELD S-3, 55 PCF)	438 LB.
INSULATION (.5" MIN-K 1301, 20 PCF)	82 LB.
AFTERBODY	
ABLATIVE MATERIAL (MAC THERMORAD SHIELD S-3, 55 PCF)	133 LB.
INSULATION (1.5" RF 700, 7 PCF & 1.5"RF 300, 3 PCF)	102 LB.
TOTAL	755 LB.

FIGURE 3-41

rises above the emergency temperature limits specified in Reference 3-19 following a long range re-entry. If a long range re-entry is made, the astronauts are required to vacate the command module on land landings, using caution in their egress in a manner similar to that required by the Project Mercury astronauts when the small bulkhead hatch is used for egress. Water landings will quench the command module side-wall temperatures and alleviate the problem.

3.8 Mechanical - Major mechanical systems were investigated in sufficient detail to provide sound bases for weight estimates and to assure operational feasibility of system concepts. Mercury and Gemini experience is heavily relied upon in system selections. Design complexity is avoided in favor of simpler, although heavier, methods.

The problem of cold molecular welding is not completely defined, but is recognized as a potential detail problem area. The following study philosophies are adopted regarding this phenomenon:

A. Where possible, mechanical systems are arranged to circumvent or eliminate the problem by locating within a pressurized area or individual sealing of critical units.

B. Areas inconvenient to pressurize are designed to minimize bearing pressures and coating mating surfaces with Everlube 811 (a Moly-disulfide coating). Project Gemini uses this coating and although testing in hard vacuum is still incomplete, preliminary results are satisfactory.

C. Data relating bearing pressures, temperatures, time and vacuum conditions from various research sources are incomplete and contradictory. Therefore, the preliminary M.A.C. test data, supported by design margins and the use of special lubricant coatings are used as study criteria.

3.8.1 Escape Tower Separation - Separation of the escape tower, shown in Figure 3-42 is accomplished with a three segment clamp ring similar to that used on Project Mercury. The segments are connected by explosive tension bolts, each incorporating dual ignition. Release of a single bolt releases the clamp ring. The tower umbilical is separated by a guillitine cutter at the time of separation rocket ignition. The tower is shimmed at the truss-skirt interface for fine thrust alignment and the clamp ring is moly-disulfide coated and graphite film lubricated to assure uniform clamping during assembly.

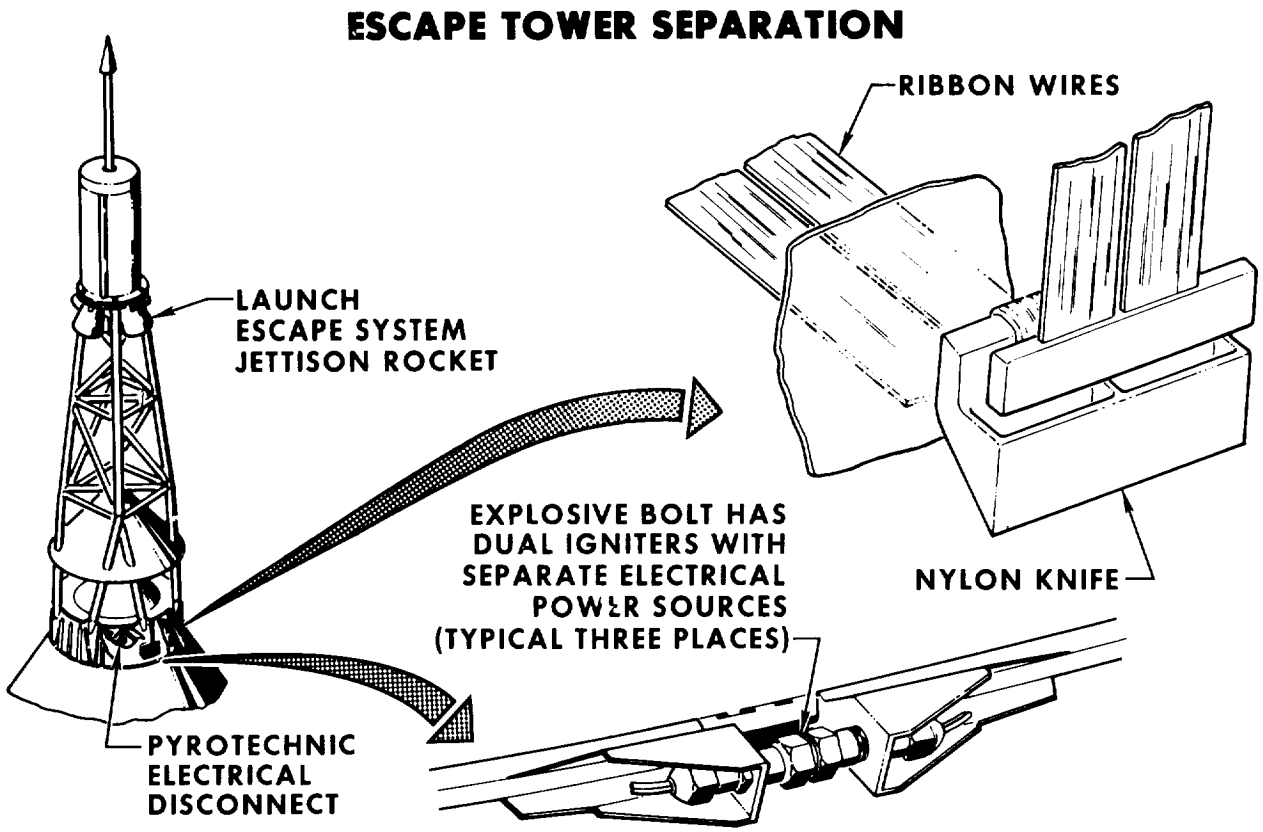


FIGURE 3-42

3.8.2 Crew Hatches and Doors - (Figure 3-43) Over-center latching, similar to that designed for Project Gemini, is incorporated to provide positive action and quick release. All latching mechanisms are contained within the pressurized compartment. All bearings and hinges are dry film lubricated (Everlube 811) and are generally of dissimilar metals. The hatches are easily opened and latched manually by the crew from either inside or outside the vehicle. A latch safeguard feature is provided to prevent inadvertant release.

3.8.3 Umbilicals - Figure 3-44 illustrates the separation methods of the umbilicals. Command to service module electrical and other umbilicals are separated by explosive cutters and spring loaded ejectors located in the service module.

3.8.4 Control Linkages - No mechanical control linkages are anticipated outside the pressurized compartment.

3.8.5 Erectable Mirrors - All external mirrors are erected and positioned immediately following departure from the atmosphere by spring devices and are manually stowed or jettisoned by the crew after use.

3.8.6 Lunar Landing Mechanisms - Figure 3-45 shows the lunar landing system in the extended position. All joints and bearings are protected by suitable coating materials. The gear folds against the spacecraft structure and is protected from heating during launch by large shallow fairings which also cover the DSIF antennas and the attitude control thrust chambers. After exit from the atmosphere, the gear is extended to the lunar landing position by dual torsion springs and over-center locks. No strut extension, compression, or rotation that might complicate gear erection, is necessary during unfolding. A detailed description of the struts are given in Section 3.12.

LUNAR LANDING GEAR

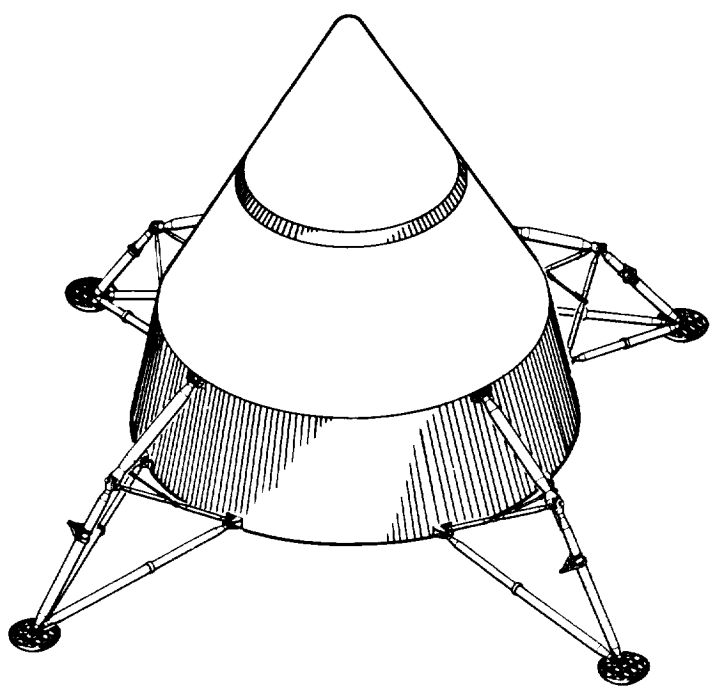


FIGURE 3-45

HATCHES AND DOORS

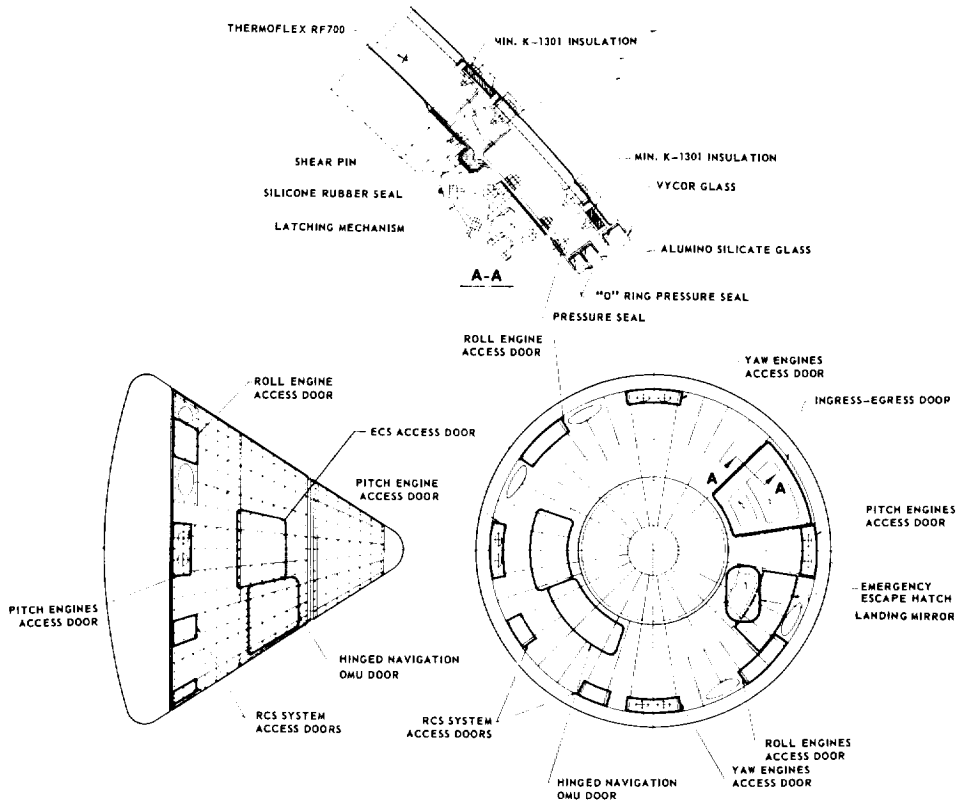


FIGURE 3-43

COMMAND MODULE TO SERVICE MODULE UMBILICALS

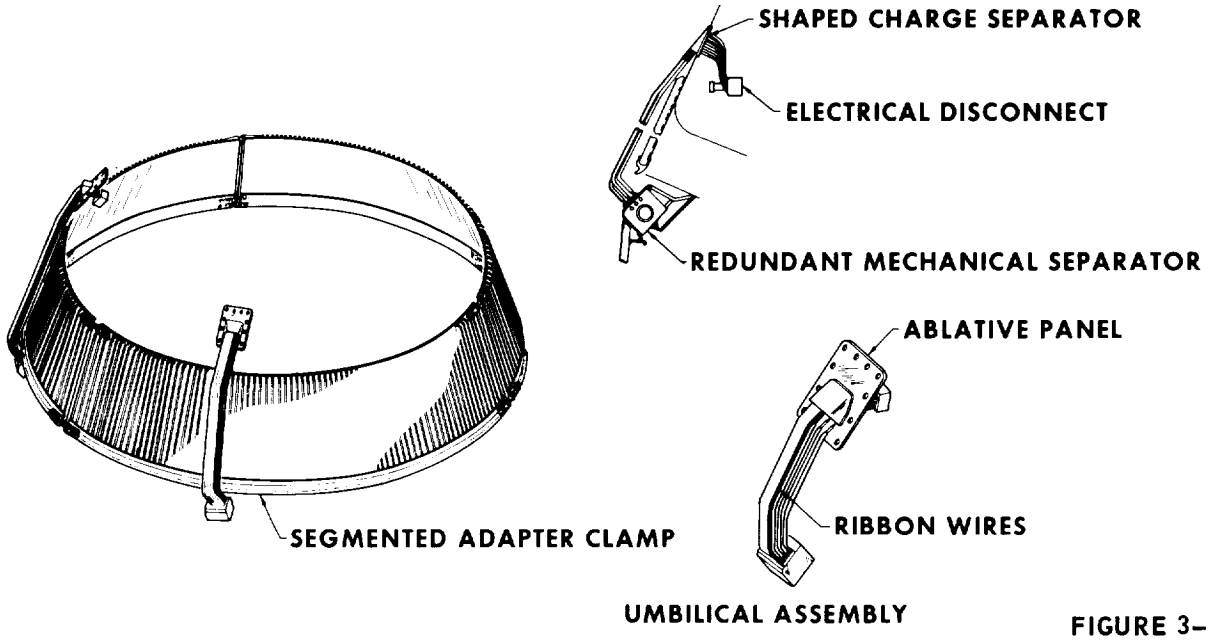


FIGURE 3-44

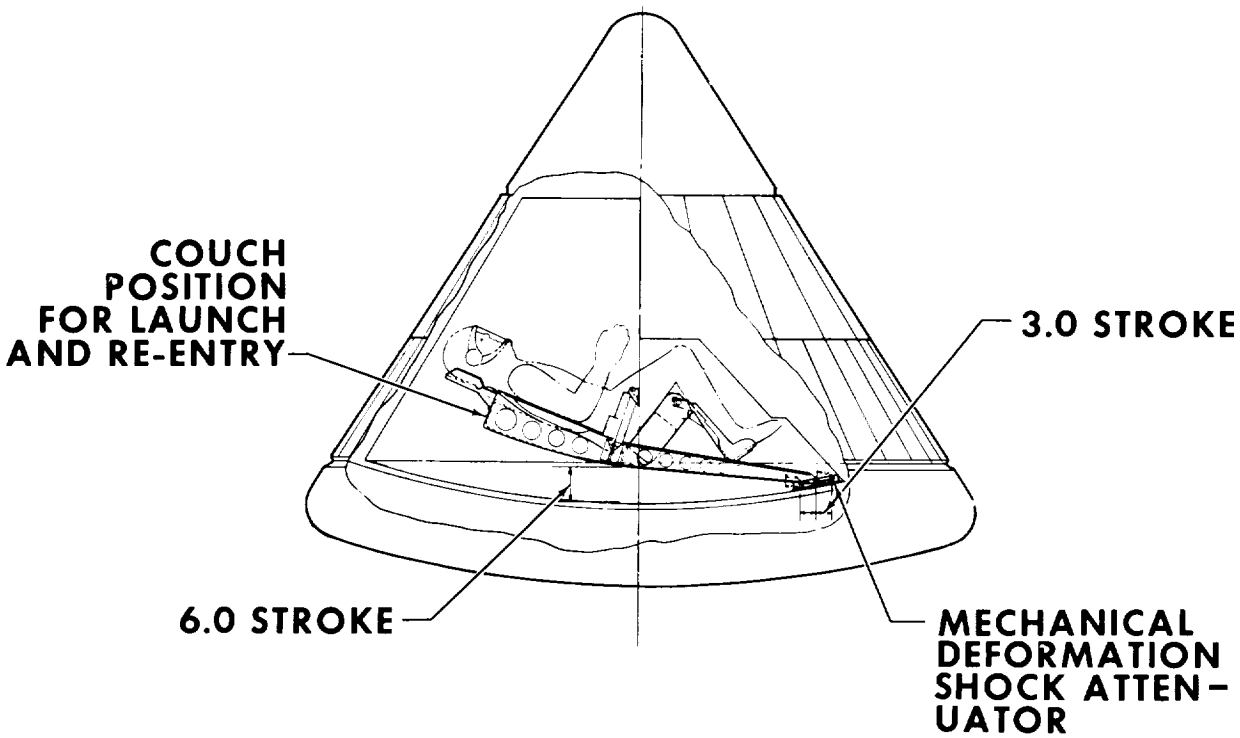
3.8.7 Earth Landing System - The earth landing system consists of three 71 foot main parachutes that are deployed simultaneously by a drogue parachute or a reserve pilot parachute. A detailed description of this system is presented in Section 3.13.

Both the drogue and reserve pilot parachutes are deployed by separate guns which have the dual function of blowing off the main parachute cap. This cap is attached to the command module structure by three spring loaded latches as shown in Figure 3-26. At parachute deployment, the latches are released by an explosive bolt actuated cable which girds the recovery systems compartment.

3.8.8 Earth Impact Attenuation - Figure 3-46 illustrates the earth impact attenuation system. Distributed crushable trussgrid located between the heat shield and the pressure bulkhead attenuates the command module to 40 g maximum. Individual seat snubbers further attenuate the seats to 20 g transverse and 10 g spinal for normal landings and maintain the crew within tolerable limits in the event of partial landing system failure. The snubbers allow for manual stowage of the seats against the pressure bulkhead during other operational phases. No problems are anticipated with the hydraulic snubbers because of the inherent inspection and maintenance features of their location within the pressurized crew compartment. A more detailed description of this system appears in Section 3.13.

3.8.9 Pyrotechnic Systems - Ignition of pyrotechnics is by conventional methods similar to those used in Project Mercury and those designed for Project Gemini. Adequate no-fire safety is provided by a relatively high no-fire current level, grounded shielding, and shorting to ground of all bridge-wire circuits.

COMMAND MODULE
IMPACT SYSTEM ATTENUATION



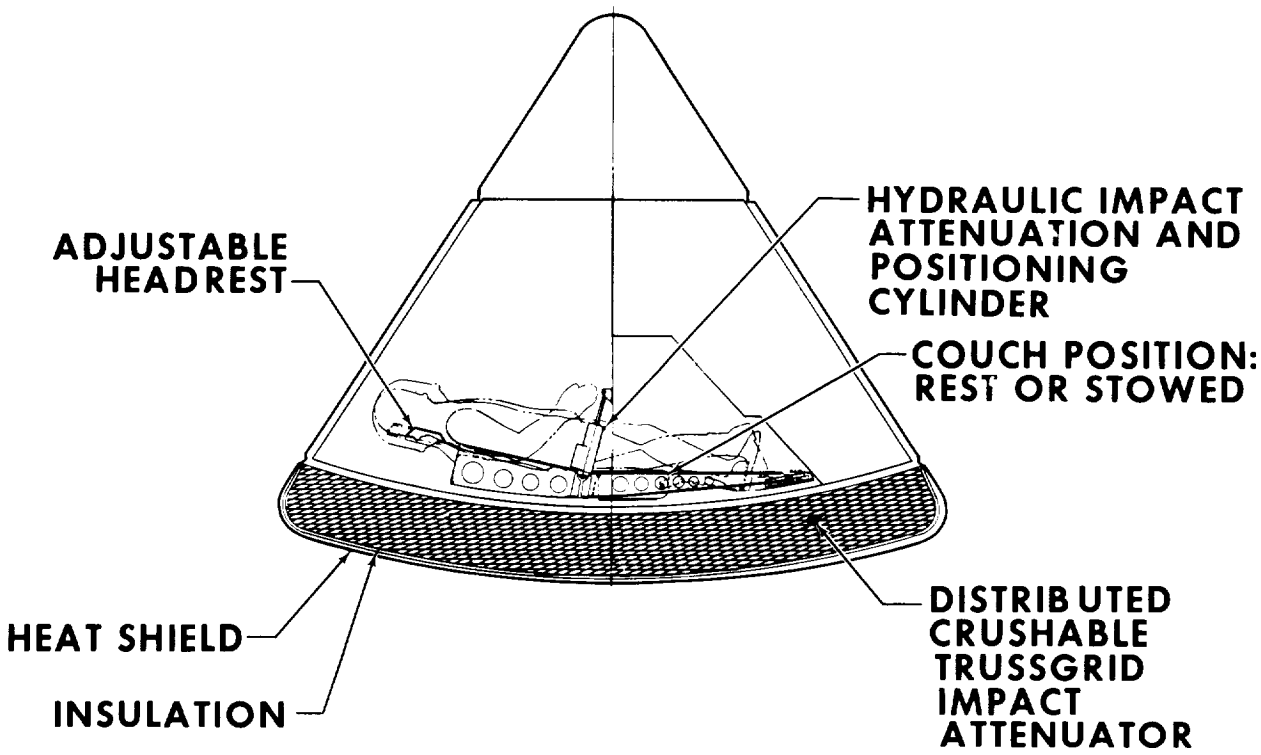


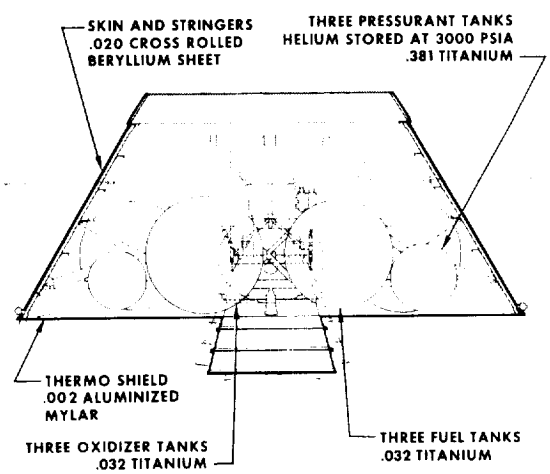
FIGURE 3-46

3.9 Propulsion

3.9.1 Service Module Propulsion System

3.9.1.1 General - The service module propulsion system is designed to perform the lunar launch and transearth injection. It also supplies the impulse for any abort after the escape tower is jettisoned. A single redundantly gimbaled fixed thrust ablative cooled engine with a radiation cooled extension, operating on earth storable hypergolic propellants, is utilized. Figure 3-47 presents the basic propulsion system details. The incremental velocity requirements are as specified in Reference 3-27. The functions of ullage positioning, attitude control, and trans-earth midcourse correction are accomplished by the service module reaction control system (RCS) described in Section 3.10.2. Table 3-10 summarizes the V requirements and the propellant required for each function.

SERVICE MODULE DETAILS



FUNCTION

LUNAR LAUNCH
TRANSEARTH INJECTION
TRANSEARTH ULLAGE POSITIONING
TRANSEARTH MIDCOURSE
ATTITUDE CONTROL FOR LAUNCH AND TRANSEARTH

WEIGHT

LOADED GROSS WT.	23,428 LB
BURNOUT WT.	3937 LB.
USABLE PROPELLANTS	19,491 LB.
TRAPPED PROPELLANTS	198 LB
SERVICE EQUIPMENT	1148 LB
STAGE MASS FRACTION	.875*

*EXCLUDES SERVICE EQUIPMENT

ENGINE DATA

THRUST	11,700 LB
COOLING METHOD	ABLATIVE AND RAD.
P_C	100 PSIA
ε	40:1
I_{SP} (VAC)	320 SEC.
O/F	2.2
WEIGHT	373 LB.
GIMBAL ANGLES	±7.5°

FIGURE 3-47

TABLE 3-10

Function	ΔV-ft/sec.	W_p-lbs.
Lunar Launch	6474	13925
Transearth Ullage Positioning	7.00	11*
Transearth Injection	3944	5071
Transearth Midcourse Correction	330	365*

*Propellant contained in service module reaction control system

The basic design criteria used in the selection of the service module propulsion system are:

A. Use of storable propellants having an advanced development background.

B. A pressurized system for simplicity of operation.

C. Helium used as pressurant for minimum over-all system weight.

D. Compatible materials for the fluids involved.

E. Pressure regulator redundancy to provide protection for any mode of primary regulator failure.

F. Motor-operated propellant shutoff valves to conserve propellant in the event of downstream leakage.

G. A burst diaphragm-relief series arrangement to assure zero leakage prior to activation of the pressure relief valve.

H. Filters located downstream from normally closed cartridge valves and burst diaphragms.

I. Redundant linked bipropellant valves on the thrust chamber assembly to assure positive operation.

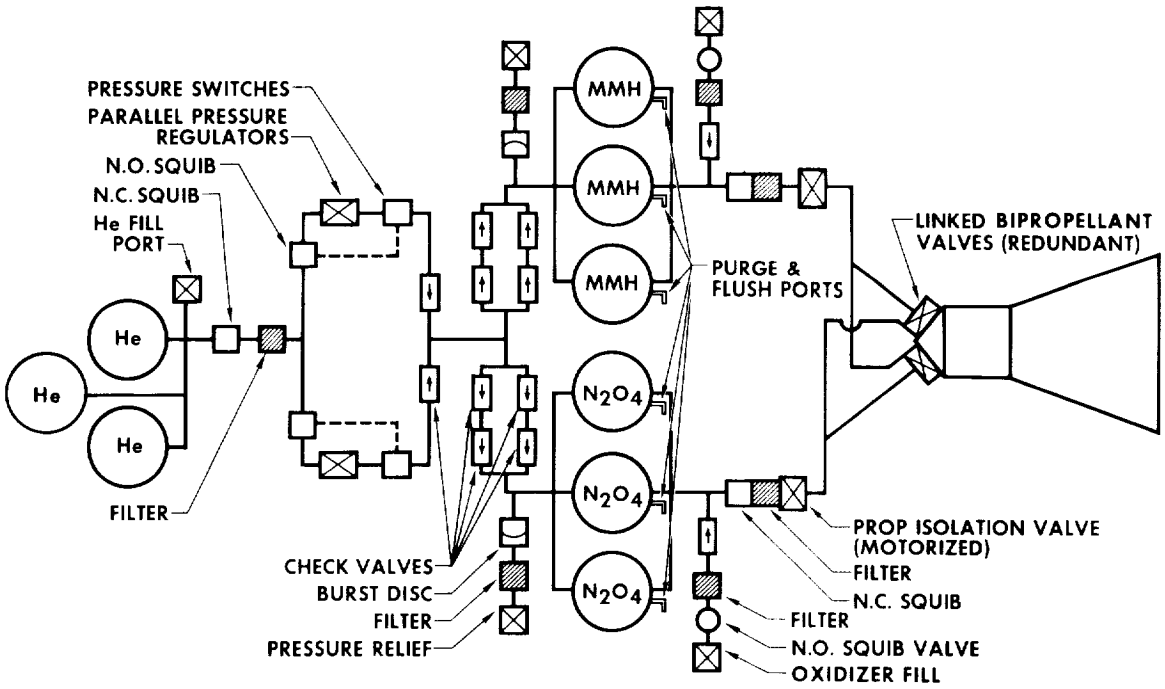
J. Thrust-to-weight ratio of 0.4 based on Reference 3-20.

3.9.1.1 (Continued)

The engine selected for the service module is designed to be interchangeable with the engine utilized in the terminal landing module with the exception of expansion ratio and throttling capability. The nozzle is constructed such that the expansion ratio may be altered by the addition or removal of a radiation cooled skirt. This "common" design eliminates the need for two engine development programs.

Ground checkout provisions are included in the system inert weight calculations but are not shown on the schematic, Figure 3-48. Functional checks of all components can be made using gas or test fluids.

SERVICE MODULE
PROPULSION SYSTEM SCHEMATIC



GROUND SERVICE CHECKOUT CONNECTIONS ARE PROVIDED, BUT NOT SHOWN SCHEMATICALLY

FIGURE 3-48

3.9.1.2 Propellant Characteristics - The propellant combination chosen for the service module propulsion system is nitrogen tetroxide (N_2O_4) and monomethylhydrazine (MMH). This combination has been used extensively in the rocket industry and is currently being used on Project Gemini. Similar propellant combinations are being employed in the Agena and Titan vehicles, therefore, field handling and servicing techniques are well established. Table 3-11 presents the pertinent characteristics of the propellants.

TABLE 3-11

Propellant Characteristics

(N_2O_4 - MMH)

Oxidizer/Fuel Mixture Ratio	2.2
Propellant Bulk Specific Gravity	1.196
Flame Temperature - ^{o}F	5293
Theoretical Shifting Equilibrium Vacuum Specific Impulse (ϵ = 40)-sec.	339
Theoretical Density Impulse - $\frac{lb\text{-}sec.}{ft^3}$	25,300
Nominal Delivered Shifting Equilibrium Vacuum Specific Impulse (ϵ = 40)-sec.	320
Oxidizer Freezing Point - ^{o}F	+12
Fuel Freezing Point - ^{o}F	-63
Normal Oxidizer Boiling Point - ^{o}F	+70
Normal Fuel Boiling Point - ^{o}F	+189

3.9.1.3 Propellant Supply System - The two basic types of propellant supply systems, pump fed and pressure fed, were evaluated in preliminary studies. The pressure fed system is chosen for the service module because of its inherent

3.9.1.3 (Continued)

reliability and simplicity of operation. Pressurization systems utilizing gas generators, exothermic fluid reaction and cold gas storage were considered with the latter being selected because of its simplicity and advanced state of development. Helium gas is chosen as the pressurant due to the large weight advantage in comparison to nitrogen in systems of this size. Positive expulsion bladders are not provided in this system. Ullage positioning is provided by lunar gravity at launch and the service module RCS during the transearth portion of the mission.

3.9.1.4 Pressurization System - The basic pressurization system for the service module consists of three helium storage bottles, a normally closed squib valve, parallel pressure regulators, check valves, and tank pressure relief valves. A parallel pressure regulator system is provided which automatically guards against a regulator failure through the use of pressure sensing switches and normally open pyrotechnic valves. In the event of a fail-closed regulator, the parallel regulator will accomplish the regulation task. In the event of a fail-open regulator, the pressure switch will sense the pressure increase and close the normally open pyrotechnic valve, allowing continued regulation by the remaining regulator. The pressurization gas characteristics are shown in Table 3-12.

The pressurization gas storage tank is fabricated of 6AL4V titanium and is designed for a nominal working pressure of 3000 psia at 530°R with a burst factor of 2.0. The tank is fabricated by drop forging in halves, welding together, then final machining the required mounts and bosses.

TABLE 3-12

Helium Pressurization Gas Characteristics

Loading Pressure (PSIA)	3000
Loading Temperature (^{o}F)	68
Density @ standard conditions ($\#/Ft^3$)	.01039
Specific heat at constant pressure ($Btu/\#-^{o}R$)	1.25
Specific heat ratio	1.667
Normal boiling point (^{o}R)	8
Density @ loading conditions (P = 3000 psia, T = $68^{o}F$) lb/ft^3	1.92

3.9.1.5 Propellant Tankage - The propellant is stored in six spherical tanks; three oxidizer tanks and three fuel tanks. The tanks are constructed of 6AL4V titanium and are designed for a nominal working pressure of 200 psia with a burst factor of 1.5. The tanks are fabricated by either drop forging or hydro-spinning and welding.

A low emissivity coating is provided on the outside of the oxidizer tanks to maintain the desired temperature range.

3.9.1.6 Engine Design and Performance - The main service module engine has a vacuum thrust level of 11,700 pounds with a nozzle expansion ratio of 40:1. The chamber pressure is 100 psia. Redundant hydraulic actuators are provided on the gimbals for pitch and yaw attitude control.

The method of thrust chamber cooling was selected after evaluation of regenerative cooling, full radiation cooling, and ablative cooling, taking into consideration the interchangeability discussed in Section 3.9.1.1, vehicle installation and overall system weight. The large pressure drop in a regenerative cooled engine is undesirable when using a pressurized feed system. The use of a fully radiation cooled engine would be desirable from the standpoint of engine weight but

3.9.1.6 (Continued)

is not utilized because high insulation weights on the adjacent propellant tanks and equipment would be incurred due to the partially submerged installation. The selected design employs full ablation cooling to an area ratio of 20:1 (20:1 is the area ratio employed in the terminal landing module engine) and a radiation cooled skirt to an area ratio of 40:1. The engine is installed such that the radiation cooled nozzle extension has a sufficient spaceview factor for proper cooling. The engine is constructed of oriented phenolic-refrasil ablative material with a fiberglass structural wrap. The radiation cooled extension is titanium. Several currently programmed engines which might be employed in this application are Aerojet General's 8000 pound thrust 624A Transtage engine; Reaction Motors Division Model TD-294 10,000 pound thrust LEM research engine; and a 10,000 pound thrust engine under investigation by Rocketdyne also for possible use in the LEM.

A study was made to determine optimum or near optimum chamber pressures to be utilized in the service and terminal landing modules. The results of this study are presented graphically in Figure 3-49. Depending on the thrust level and propellant loading, the optimum P_c for the storable systems is seen to be in the 50-150 psia range. With the earth storable propellant loadings required, 100 psia is considered near optimum and is selected for both systems. Assumptions and supporting information were:

A. Helium gas with initial conditions @ 3000 psia and 530°R is used as the pressurizing media.

B. The relationship of pressurization system, gas and bottle weight to propellant tank pressure used is as given in Figure 3-50.

C. Propellant tank weights were calculated as a function of pressure and volume using titanium as the material.

D. Thrust chamber weights versus chamber pressure and thrust level were obtained from subcontractor data.

CHAMBER PRESSURE OPTIMIZATION

N_2O_4/MMH

OF_2/MMH

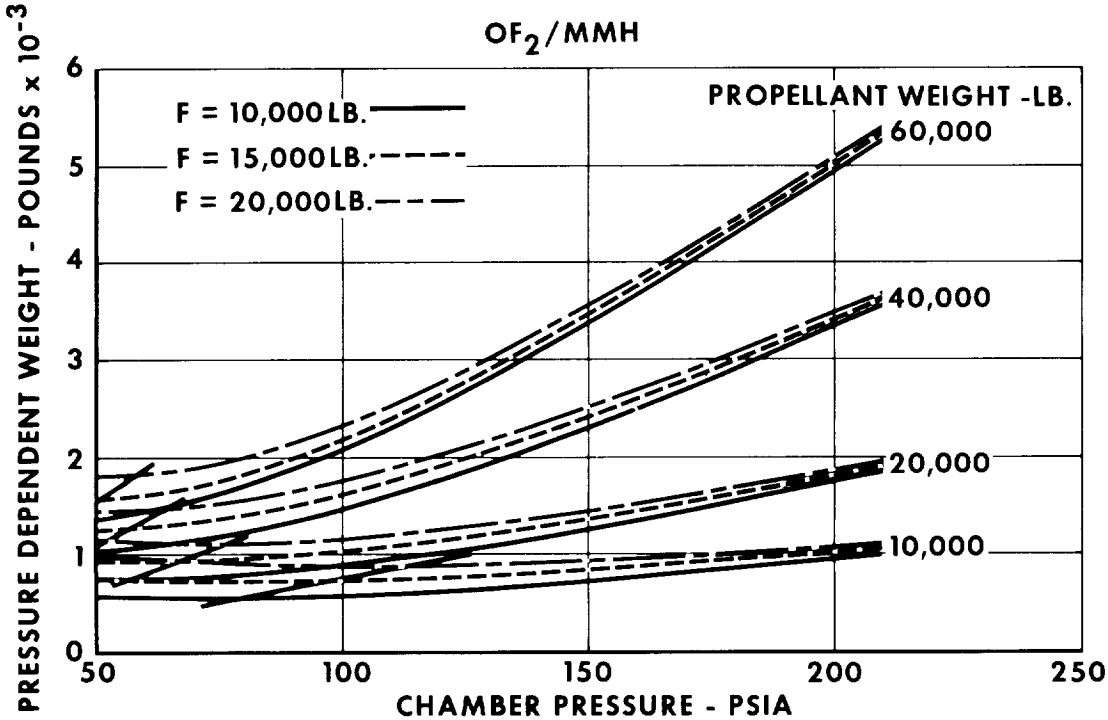


FIGURE 3-49

PRESSURIZATION SYSTEM WEIGHT PER UNIT VOLUME

HELIUM GAS P_I = 3000 PSIA
T_I = 530 °R
K = 1.667
ISENTROPIC EXPANSION

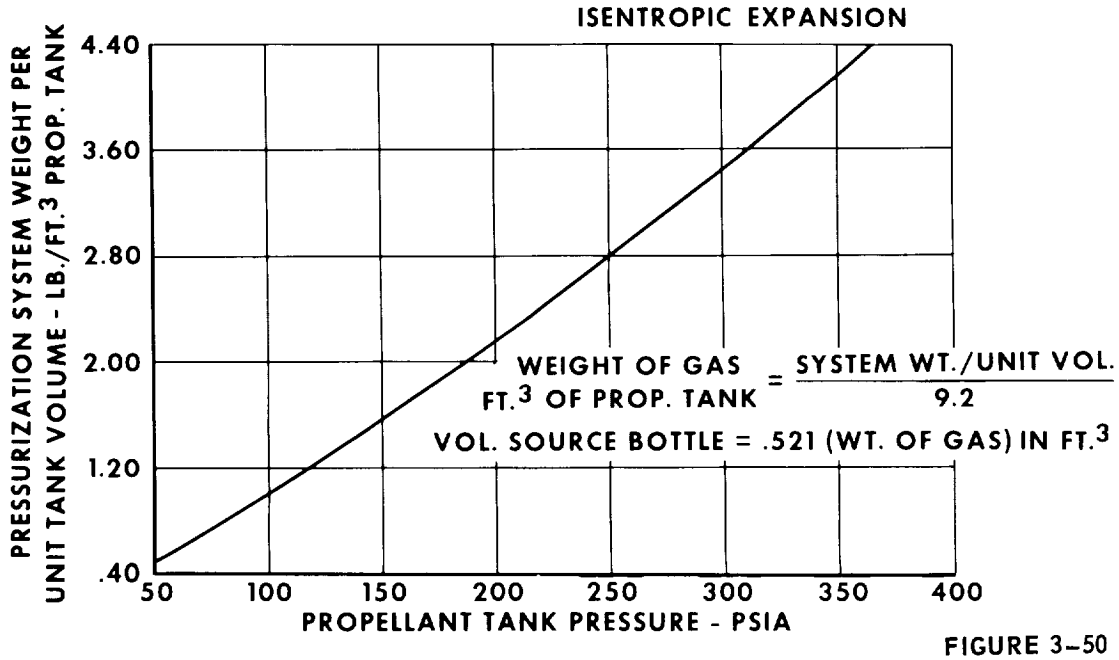


FIGURE 3-50

3.9.1.6 (Continued)

The mixture ratio of 2.2 is chosen to provide the capability for conversion from earth storable propellants to OF_2/MMH with no change in the propellant tankage. Figures 3-51 and 3-52 illustrate that the selected mixture ratio is near optimum from an I_{sp} point of view and the corresponding flame temperature is only approximately 30°F higher than for the RCS thrust chambers being developed for Project Gemini. The reduced heat transfer rates in larger thrust chambers considerably reduces the effects of the increase in operating temperature.

The nominal specific impulse, as determined from an industry survey for the type of thrust chamber and the mixture ratio employed, is 320 seconds.

SPECIFIC IMPULSE VARIATION WITH MIXTURE RATIO N_2O_4/MMH

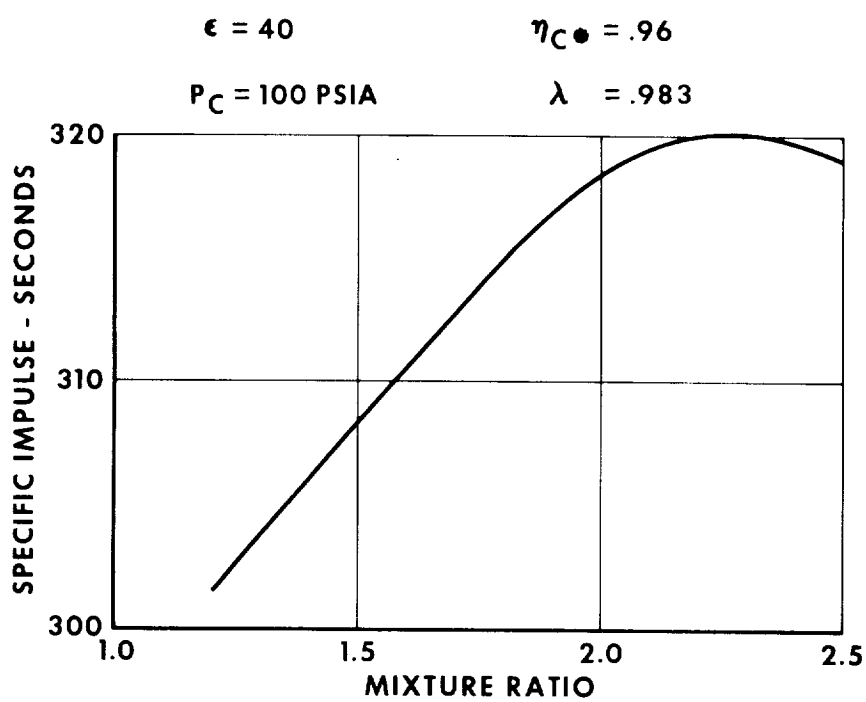


FIGURE 3-51

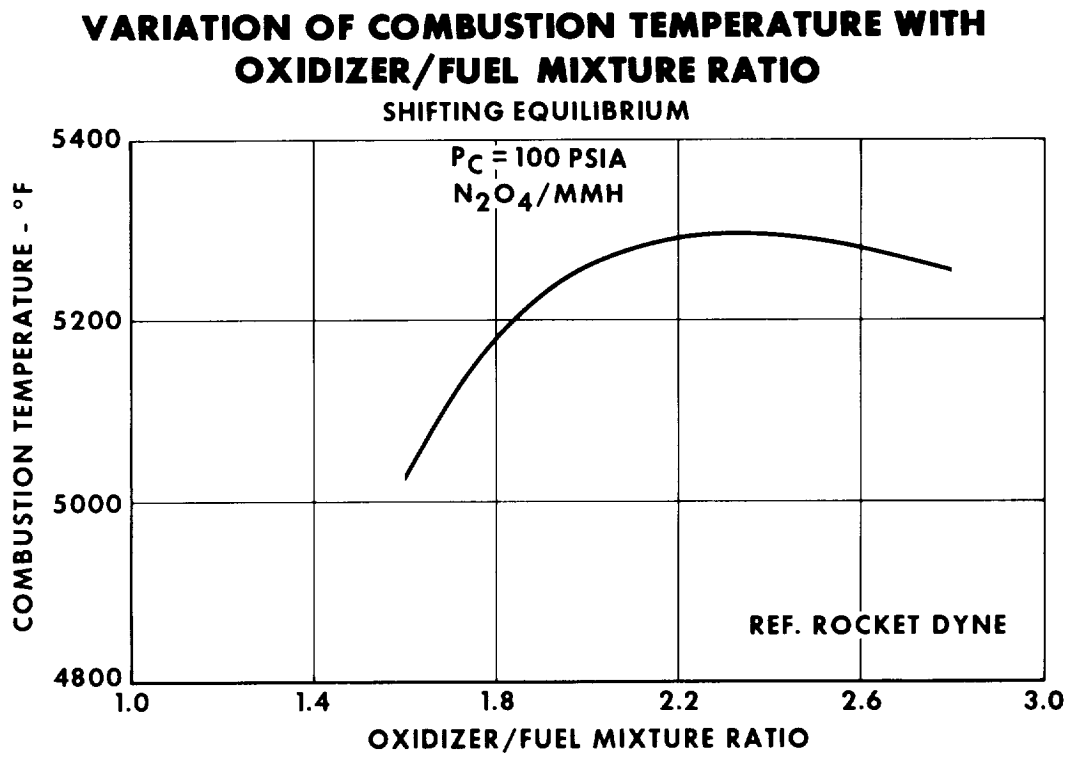


FIGURE 3-52

3.9.1.7 System Description and Operation - A schematic of the system selected for the service module is shown in Figure 3-48. As discussed in Section 3.9.1.4, the pressurization system consists of three helium storage bottles, parallel regulators and parallel-series check-valves to prevent propellant vapors from mixing in the lines. The propellant tanks are manifolded together and the fuel and oxidizer are retained in their respective tanks by a normally closed squib valve until system operation is initiated. An isolation valve is incorporated in the lines to conserve propellant in the event of a downstream leak. Linked bi-propellant injector valves are utilized on the thrust chamber to insure satisfactory operation of the engine. Filters are incorporated downstream of the normally closed pyrotechnic valves and burst discs to insure that foreign particles do not enter downstream components.

3.9.1.6 (Continued)

Filters are also included in all ground test connections to prevent foreign material from entering the system during servicing.

The system is activated by firing the normally closed squib valves in the pressurant and propellant lines. Opening the solenoid actuated bipropellant valves on the thrust chamber allows propellant flow to the combustion chamber where hypergolic ignition occurs.

The pressure losses between the propellant tank and engine are estimated on the basis of Gemini component losses with allowances for differences between the systems. These losses are summarized on the following page.

3.9.1.7 (Continued)

Injector ΔP	50.0 psi
Propellant filter	4.5
Line loss (including tank manifolding)	5.5
Propellant valve and thrust chamber calibrating orifice	40.0
Total	100.0 psi

Based on these pressure losses, the propellant tank operating pressure is established at 200 psia.

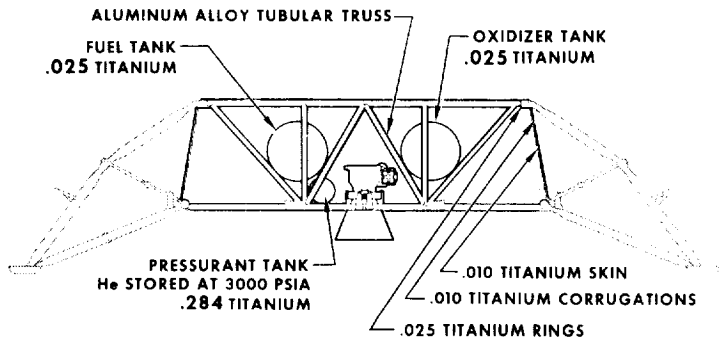
It is recognized that some difficulty may be experienced with spacecraft stability during service module engine start-up due to over pressures or flow deflection caused by "firing-in-the-hole". While this condition has not been analyzed, the internal structural and equipment only sparsely occupy the terminal landing module such that the flow from the service module engine is relatively unobstructed after the retrograde module is jettisoned. For lunar launch it is anticipated that the terminal landing engine could be removed if necessary.

3.9.2 Terminal Landing Module

3.9.2.1 General - The propulsion system of the terminal landing module is designed to perform the final descent, hover, and translation maneuvers. A single gimbaled variable thrust ablative cooled engine is utilized which operates on earth storable hypergolic propellants. The ΔV requirement is 770 ft./sec. as specified in Reference 3-26. Figure 3-54 presents the basic propulsion system details.

The design criteria used in the selection of the propulsion system are similar to those employed in the service module design, discussed in Section 3.9.1.1, with the exception of required thrust-to-weight ratio and that throttling is required.

TERMINAL LANDING MODULE DETAILS



FUNCTION	WEIGHT		ENGINE DATA	
TRANSLUNAR ULLAGE POSITIONING	LOADED GROSS WEIGHT	6115 LB.	THRUST (MAX.)	11,270 LB.
			THRUST (MIN.)	3410 LB.
TRANSLUNAR ATTITUDE CONTROL	USABLE PROPELLANTS	3380 LB.	COOLING METHOD	ABLATIVE
	TRAPPED PROPELLANTS	34 LB.	P_C	100 PSIA
FINAL DESCENT, HOVER AND TRANSLATION	LANDING GEAR PLUS CARRY-THROUGH	740 LB.	ε	20:1
			I_{SP} (VAC.)	308 SEC.
	STAGE MASS FRACTION	.660*	O/F	2.2
			GIMBAL ANGLE	6°
			WEIGHT	266 LB.

*EXCLUDES LANDING GEAR, PLUS CARRY THROUGH AND SCIENTIFIC EQUIP.

FIGURE 3-54

3.9.2.1 (Continued)

A throttling ratio of 3 to 1 is considered readily attainable within present state-of-the-art and was used in the analysis of the terminal landing propulsion system. For this range, a minimum thrust-to-weight ratio (earth reference) of about 0.1 results in a minimum propellant consumption as discussed in Section 3.9.2.6. The corresponding maximum thrust-to-weight ratio required results in a thrust level nearly the same as that of the service module engine. On this basis it was decided to modify the service module engine for the terminal landing module application. The modifications include removing the titanium radiation cooled skirt and addition of a throttleable injector.

Ground checkout provisions are included and are similar to those in the service module.

3.9.2.2 Propellant Characteristics - The propellant combination chosen for the terminal landimg module is nitrogen tetroxide and monomethyhydrazine. The performance characteristics are identical to those of the service module, Table 3-11, with the exception of I_{sp} and theoretical density impulse. For an expansion ratio of 20:1 these values are 308 seconds (nominal delivered I_{sp}) and 23,000 lb-sec/ft^3, respectively.

3.9.2.3 Propellant Supply Systems - Considerations in the selection of the propellant supply system are the same as for the service module (discussed in Section 3.9.1.3). A helium gas pressurization system is chosen for the terminal landing module. Positive expulsion is not utilized because the propellants will be properly positioned at the completion of main retrograde and prior to ignition of the terminal landing engine.

3.9.2.4 Pressurization System - The pressurization system is similar to that utilized in the service model, discussed in Section 3.9.1.4.

3.9.2.5 Propellant Tankage - The propellants are stored in two spherical 6AL4V titanium tanks. These tanks are designed for a 200 psia working pressure with a burst factor of 1.5. The tanks are fabricated by the techniques described in Section 3.9.1.5.

3.9.2.6 Engine Design and Performance - The terminal engine size was selected on the basis of throttling ratio and propellant consumption required to make the final descent. A throttling ratio of 3 to 1 is considered readily attainable within present state-of-the-art and was selected for this system. With this throttling ratio a minimum thrust-to-weight ratio (earth reference) of 0.1 results in minimum propellant consumption as shown in Figure 3-55. A higher throttling range would result in some reduction in propellant requirements for final descent, as shown on Figure 3-56, and therefore, would be a desirable goal of the engine development program.

The maximum required engine thrust arrived at by the above analysis is compatible with utilizing the same basic engine as that used in the service module. The radiation skirt is removed, in order to reduce the height of the lunar landing spacecraft configuration and thus gear weight, and throttling capability is added. The reduction in engine performance associated with the removal of the radiation skirt, expansion ratio reduced from 40:1 to 20:1, is more than offset by the reduction in spacecraft weight.

Utilizing the service module engine in the manner indicated and maintaining a minimum thrust-to-weight ratio of 0.1 results in a maximum thrust of 11,270 pounds and a throttling ratio fo 3.3:1. The required engine burning time in terms of maximum and minimum thrust are shown in Figure 3-57 as a function of staging attitude.

SELECTION OF LUNAR LANDING PROPULSION THRUST-TO-WEIGHT RATIO

VERTICAL VELOCITY = 0 FT./SEC.
THROTTLING RATIO = 3

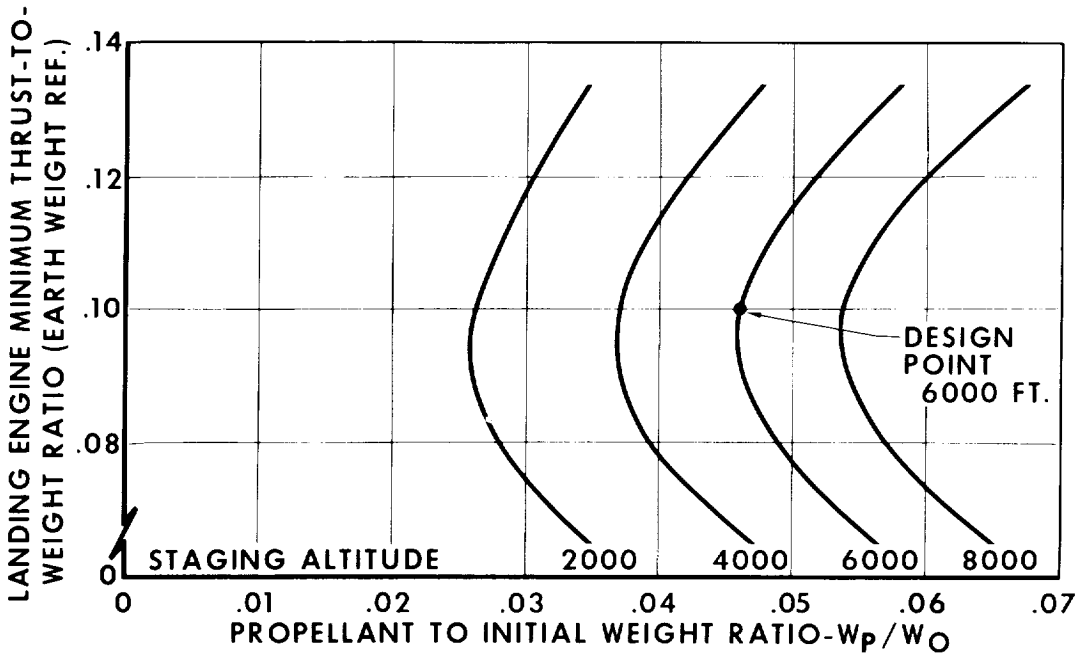


FIGURE 3-55

THROTTLING RATIO EFFECTS ON PROPELLANT REQUIREMENTS

INITIAL VERTICAL VELOCITY = 0 FT./SEC.

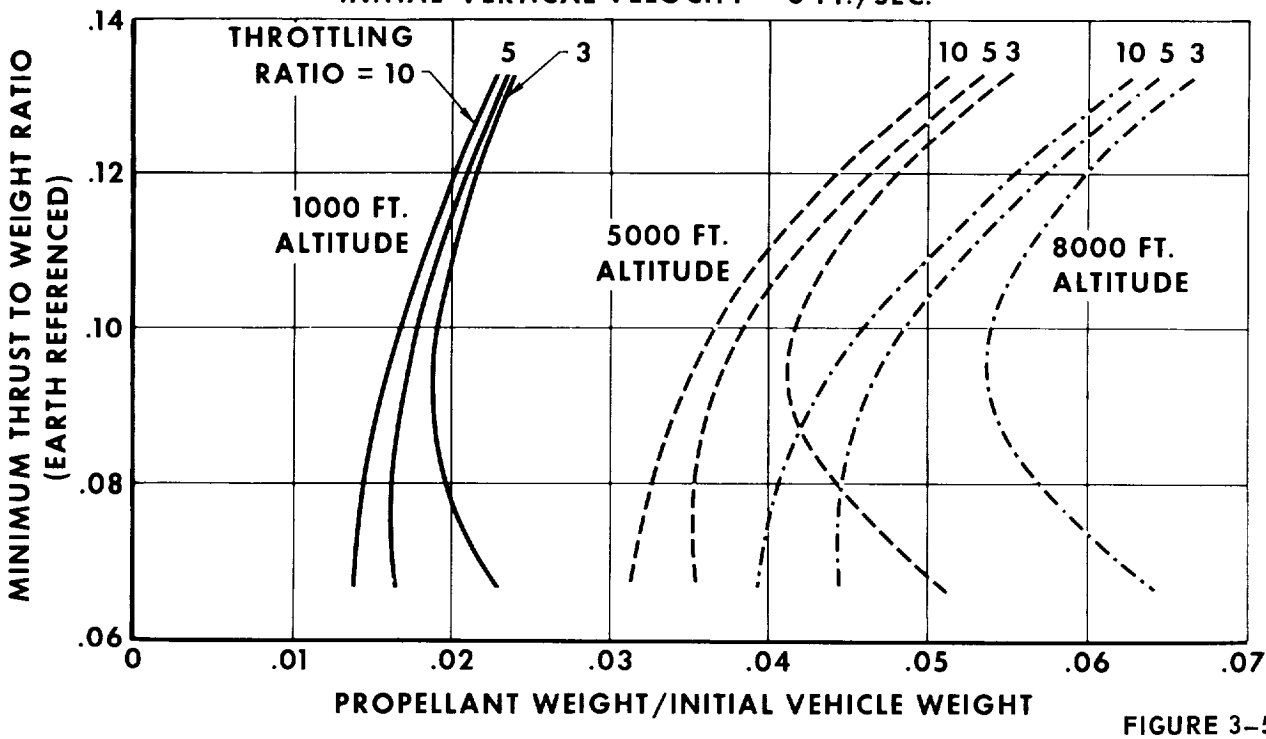


FIGURE 3-56

LANDING ENGINE BURNING TIME

F_{MAX} = 11,270 LB. F_{MIN} = 3,410 LB.

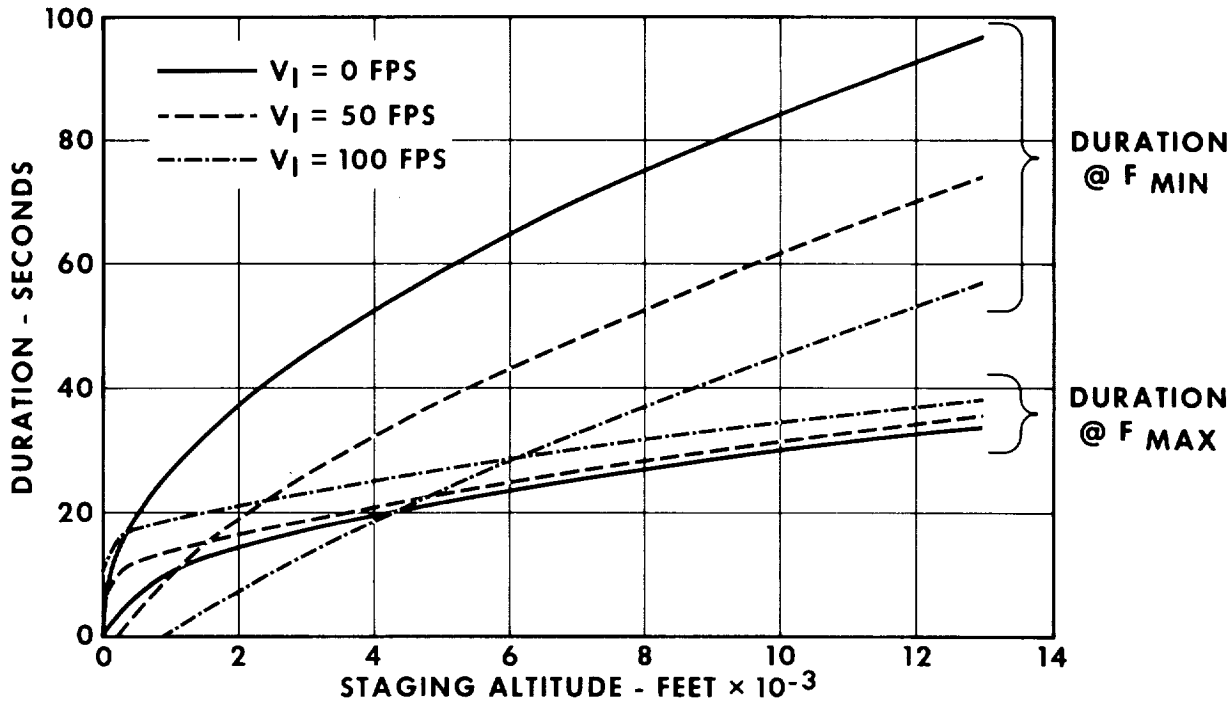


FIGURE 3-57

Throttling a rocket engine throughout the range specified can be accomplished a number of ways; (1) variable area injector, (2) variable area injector with separate flow control valve (being developed by STL), or (3) helium injection (as proposed by Rocketdyne on LEM). These approaches appear to be the more promising methods. A definite selection of a method was not made during the study but several techniques were reviewed to establish the feasibility of the throttling range required.

The discussion in Section 3.9.1.6 is applicable to the selection of chamber pressure, mixture ratio and specific impulse.

3.9.2.7 System Description and Operation - A schematic of the system selected for the terminal landing module is shown in Figure 3-58. The system is similar to the one employed in the service module discussed in Section 3.9.1.7.

TERMINAL LANDING MODULE
PROPULSION SYSTEM SCHEMATIC

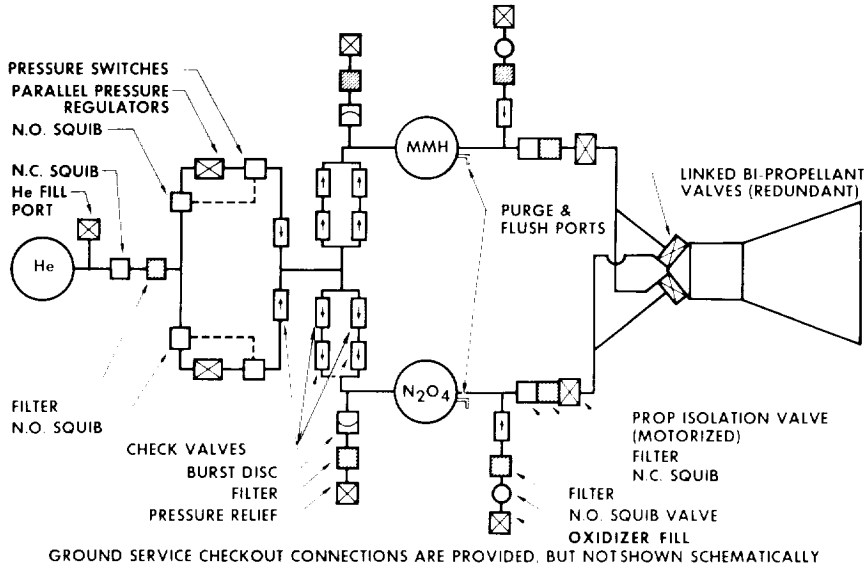


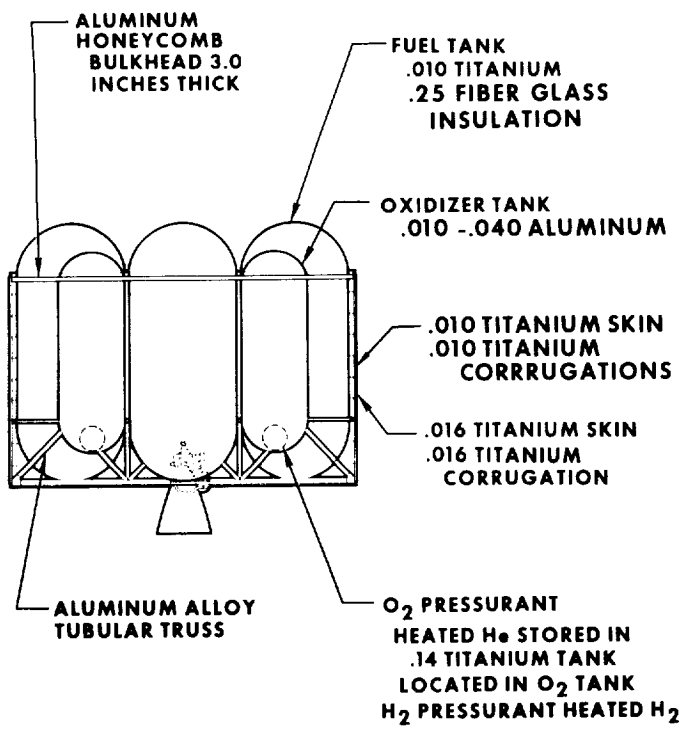
FIGURE 3-58

The terminal descent phase is accomplished in the following manner. At the completion of the descent from orbit, the main retrograde module is jettisoned and the terminal engine ignited. The spacecraft descends with the engine running at minimum thrust level to the altitude where near maximum thrust is required to bring the velocity to zero approximately 100 feet above the lunar surface. The spacecraft attitude is essentially vertical during the descent phase; pitch and yaw control being provided by the gimbaled engine and roll control by the reaction control system located in the terminal landing module. The hover phase is accomplished by throttling the main engine. Translation is achieved by tilting the vehicle with the gimbaled engine, and applying the necessary thrust to maintain the desired horizontal accelerations combined with the required hover thrust.

3.9.3 Retrograde Module

3.9.3.1 General - The retrograde module propulsion system performs the functions of translunar midcourse correction, insertion into lunar orbit, rejection from the lunar orbit and the final retrograde to the selected staging altitude. An uprated turbopump-fed, regeneratively cooled Pratt and Whitney RL 10A-3 engine, operating with liquid oxygen and liquid hydrogen propellants is utilized. Figure 3-59 presents the propulsion system details. The ΔV requirements and the usable propellant loadings for the various module functions are summarized in Table 3-13.

RETROGRADE MODULE DETAILS



FUNCTION

TRANSLUNAR MIDCOURSE CORRECTIONS
LUNAR ORBIT INJECTION
MAIN LUNAR LANDING RETROGRADE

WEIGHT

LOADED GROSS WEIGHT	53,787 LB
USEABLE PROPELLANTS	47,540 LB
TRAPPED PROPELLANTS	480 LB
STAGE MASS FRACTION	.884

ENGINE DATA

THRUST (MAX.)	26,500 LB
THRUST (MIN.)	2650 LB
COOLING METHOD	REGENERATIVE
P_C	535 PSIA
ϵ	40:1
I_{SP} (VAC.)	430 SEC
O/F	5
GIMBAL ANGLE	$\pm 6°$
WEIGHT	370 LB

FIGURE 3-59

3.9.3.1 (Continued)

TABLE 3-13

	ΔV^* - fps	W_p - Lbs.
Translunar Midcourse Correction	311	1992
Lunar Orbit Injection and 6° Plane Change	3547	19772
Lunar Orbit Rejection	129	624
Main Retrograde to Staging Point	6551	25152
	Total	47540

*37 fps is provided by the terminal landing module RCS for ullage positioning.

The design criteria used in the selection of the retrograde propulsion system are:

A. Engine selection predicated on existing or logical growth-version engines.

B. High performance propellant combination with advanced development experience.

C. Materials compatible with the combination selected.

D. Turbopump engine to reduce tankage and overall stage weight.

E. Pressurization system for the net positive suction head requirements that is light and has adequate development experience.

F. Optimum insulation weight to reduce pre-launch boiloff rates to an acceptable level.

Ground checkout provisions are included in the system weight but are not shown in the schematic, Figure 3-62.

3.9.3.2 Propellant Characteristics - The propellant combination chosen for the main retrograde propulsion system is liquid oxygen and liquid hydrogen. The pertinent propellant characteristics are presented in Table 3-14.

3.2.3.2 (Continued)

TABLE 3-14

Oxidizer/Fuel Mixture Ratio	5
Theoretical Vacuum Specific Impulse - Sec (Shifting equilibrium $\epsilon = 40$)	456
Delivered Vacuum Specific Impulse - Sec ($\epsilon = 40$)	430
Flame Temperature - ^{o}F	5435
Oxidizer Boiling Point - ^{o}F	-297
Fuel Boiling Point - ^{o}F	-423.3
Oxidizer Freezing Point - ^{o}F	-362
Fuel Freezing Point - ^{o}F	-435
Bulk Specific Gravity at Saturated Cond. (14.7 psia)	.325

3.9.3.3 Propellant Supply System - A pump-fed propellant transfer system is chosen from weight considerations. Some propellant pressurization is required, however, to satisfy the turbopump net positive suction head requirements of the selected engine.

Prior to the use of the retrograde engine, the propellant is positioned by the reaction control system located on the terminal landing module. The propellant weight for ullage positioning is estimated on the basis of six firings of the reaction control thrust chambers.

3.9.3.4 Pressurization System - A nominal amount of propellant tank pressurization is required to meet the net positive suction head (NPSH) requirements of the turbopumps. Engine manufacturer data indicates that 4 psia and 2 psia NPSH is sufficient for the oxygen and hydrogen pumps respectively. It is assumed that, the propellants are in a saturated condition during loading, venting to sea level ambient and a 5 psi drop in the tank manifolds is realistic. From these considerations, a propellant tank pressure of 30 psia is chosen, allowing an increase in vapor pressure due to heat input of 6 psi for the oxygen and 8 psi for the hydrogen.

3.9.3.4 (Continued)

The pressurization system for the module is similar to that employed on the Saturn S-IV. Heated hydrogen to pressurize the hydrogen tanks is extracted from the regenerative cooling tubes on the engine. Heated helium is utilized to pressurize the oxygen tanks. The helium is stored at 3000 psi in tanks within the liquid oxygen tanks and is heated by heat exchangers mounted on the RL 10A-3 engine. A factor of 3.0 is incorporated in the pressurant weight requirements to allow for cooldown during coast periods.

3.9.3.5 Propellant Tankage - The propellant is stored in eight cylindrical tanks, four for the oxidizer and four for the fuel. The fuel tanks are constructed of 6Al4V titanium and the oxidizer tanks of aluminum. Linde SI-4 insulation is applied to the tanks to eliminate liquefaction of air around the tanks prior to launch and to minimize boiloff during the launch and translunar phases of flight. During the pre-launch period, the hydrogen tank insulation spaces are ventilated continuously with helium to prevent liquefaction of air. The tank compartment is vented with dry air to prevent formation of ice on outer surfaces of the H_2 tank insulation or the O_2 tank surfaces. The insulation necessary for pre-launch propellant storage is sufficient to reduce inflight boiloff to negligible amounts. The location of the tanks within a structural skin enclosure contributes substantially to the reduction of heat input to the tanks.

3.9.3.6 Engine Design and Performance - The selection of a turbopump system instead of a pressurized propellant feed system for the final configuration is based on overall system weight and spacecraft envelope considerations.

The engine utilized is an uprated version of the Pratt & Whitney RL 10A-3, currently in the final phases of development for the Centaur and Saturn programs. A summary of the engine characteristics is shown in Figure 3-59.

3.9.3.6 (Continued)

The maximum thrust level is chosen on the basis of the 0.4 thrust-to-weight ratio at initiation of retrograde, given in Reference 3-20. The engine has the flexibility of being throttleable over a 10:1 range, which is advantageous for making midcourse corrections. Trajectory studies indicate a need for throttling, only to about 30% of maximum thrust for trajectory control during main retrograde. The engine is gimbaled with two degrees of freedom to provide spacecraft attitude control in pitch and yaw while the engine is operating. The gimbal arrangement is an adaptation of the Pratt & Whitney system with a separate battery electrical power source located within the module. Allowances for similar gimbal arrangement systems are included in the weights of the service and terminal landing modules. Roll control is provided by the terminal module reaction control system. A propellant utilization system is provided with a mixture ratio controller on the engine to effect near-simultaneous depletion of the main retrograde propellants.

A study was made using one, two, and three P & W RL 10A-3, 15,000 pound thrust engines to perform the retrograde module functions. The V requirements for lunar retrograde from 50,000 feet are taken from Reference 3-20 for the thrust-to-weight ratios resulting from the various engine configurations studied. The effect of the number of main retrograde RL 10A-3 engines on the command module weight is summarized in Table 3-15.

TABLE 3-15

Number of RL 10A-3 Engines Operating	Command Module Weight-Lbs.
1 of 1	-413
1 of 2	-534
1 of 3	-644
2 of 2	-106
2 of 3	-202
3 of 3	-230

3.9.3.6 (Continued)

It can be seen that the weight penalty associated with single engine performance is large. In order to avoid adding the extra propellant weight, at least two operating engines are required. The probabilities of two out of two and two and out of three engines performing successfully have been examined as a function of single engine reliability. The results are shown in Figure 3-60. The probability of

ENGINE CONFIGURATION RELIABILITY COMPARISON

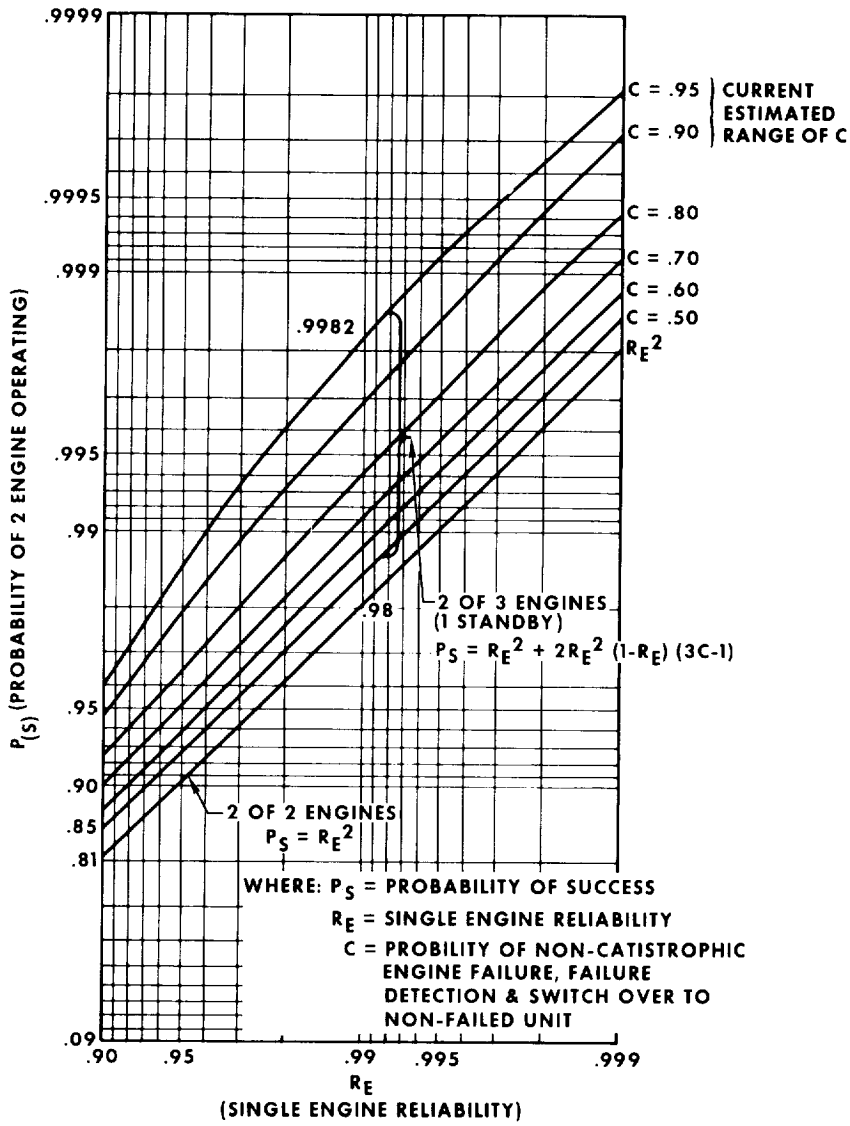


FIGURE 3-60

3.9.3.6 (Continued)

success of the three engine arrangement is dependent on a factor designated as C, which is the product of the probabilities of failure detection, failure isolation, and non-catastrophic failure occurrence. The comparison is shown for a range of C which is estimated to be about .9 and .95.

For a single engine reliability of .99 the two of two arrangement has a probability of success of about .98. This is low enough that it is impractical not to provide the propellant for single engine operation. The three engine arrangement for the same single engine reliability of .99 and a C factor of .925, has a probability of .9975 that two engines will be operative. This is considered sufficiently high to justify not including propellant for single engine operation. Because the thrust level of the RL 10A-3 engine is substantially below that which is required for minimum propellant consumption, it is apparent that at least three RL 10A-3 engines should be employed.

The three engine arrangement shown in the previous comparison considered two engines active with one in standby mode. This configuration was selected over the three engine active arrangement based on the following considerations. The two systems are essentially the same except the one standby engine arrangement requires that two additional normally closed shut-off valves be actuated if engine failure occurs. For the three engine active configuration, the probability that at least two will operate successfully is given by:

$$P_{S\ active} = R_E^3 + 3\ R_E^2\ (1-R_E)C$$

where R_E = Per Engine Reliability

C = (Probability of failure detection)(Failure isolation)(Probability that failure is non catastrophic)

Similarly, for the two active, one standby configuration, the probability that two engines will function is given by:

3.9.3.6 (Continued)

$$P_{S\ standby} = R_E^2 + 2\ R_E^2\ (1-R_E)\ C\ R_{SV}^2$$

where R_{SV}^2 = Probability that two normally closed valves will function

The reliabilities of these two configurations are equal when $R_{SV}^2 = \frac{3C-1}{2C}$. Figure 3-61 shows this relationship. The reliability of the shut-off valve considered here (squib operated) is estimated to be at least 0.999, which results in a substantial preference for 2 active, one standby arrangement.

3.9.3.7 System Description and Operation - A schematic drawing of the main retrograde propulsion system is shown in Figure 3-62. The system operates basically as follows. The oxygen and fuel fill vents are opened prior to initiating the fill operation through the fill valves. Subsequent to filling, a top-off procedure is followed until just prior to launch. During this period, the helium bottles are filled and allowed to stabilize. At launch, the top-off operation is ceased and the tank ambient vents are closed. The propellant vapor pressure is allowed to rise, by heat addition, to the vent-relief setting of 21 psia on the oxygen and 23 psia on the hydrogen. Immediately prior to system operation, the vent-relief valves are closed and the normally closed squib valve is actuated, allowing the oxygen tank to be pressurized to 30 psia. Opening the main propellant valves to the engine allows liquid hydrogen to be heated in the regenerative cooling tubes, a portion of which is bled through a regulator to pressurize the hydrogen tank to 30 psia. Engine ignition is accomplished by electrical spark. During engine operation, helium stored at liquid oxygen temperature, is heated for more efficient utilization before pressurizing the oxygen tank.

The engine manufacturer states that propellant losses due to engine chill-down prior to start-up are being reduced in the current development program and will be essentially zero by 1965. Therefore, no consideration of these losses is made in the system analysis.

STANDBY VS. ACTIVE REDUNDANCY

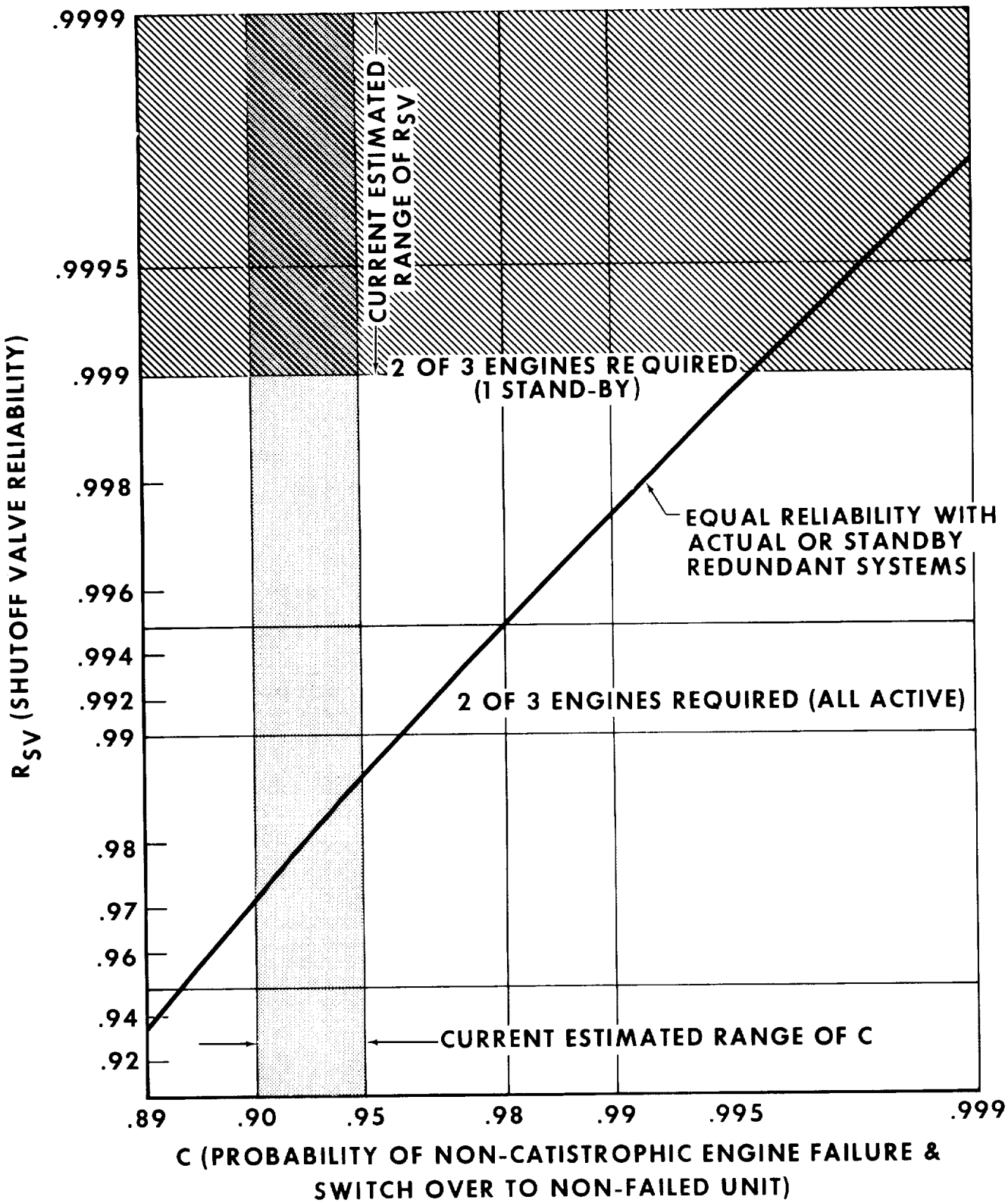


FIGURE 3–61

RETROGRADE MODULE
PROPULSION SYSTEM SCHEMATIC

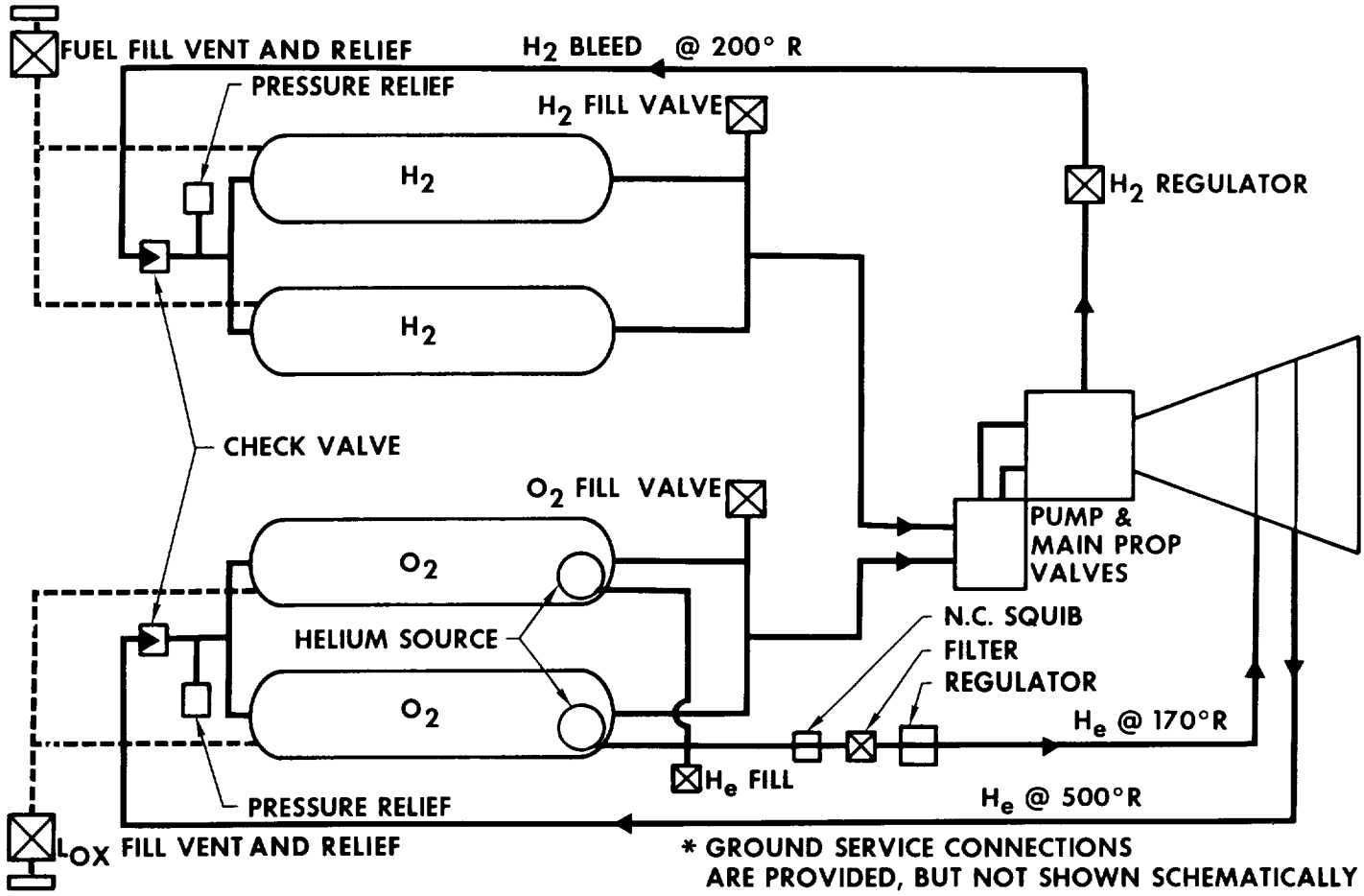


FIGURE 3-62

3.9.4 Deviations to the Apollo Statement of Work - The following deviations are taken to the respective paragraphs in Reference 3-19 in the design of the propulsion systems.

Paragraph - 3.4.2.2.3.1

Requirement - Helium tanks shall be positively sealed by redundant valves prior to use to prevent leakage.

Two-Man Apollo - Only one pyrotechnic valve is used.

Requirement - Propellant utilization control is required to maintain proper oxidizer/fuel ratio.

Two-Man Apollo - Only the main retrograde engine has propellant utilization control. All other engines are adjusted to trim during acceptance testing.

Requirement - If multiple fuel and oxidizer tanks are used, each tank should be emptied prior to use of next tank.

Two-Man Apollo - Multiple tanks are used but are manifolded, not placed in series.

Paragraph 3.4.3.4

Requirement - The lunar landing module utilizes a composite propulsion system both main retrograde and lunar landing.

Two-Man Apollo - The lunar landing module is divided into two modules:

A. Main retrograde with one engine for translunar midcourse and gross velocity increments required for lunar orbiting and lunar retrograde.

B. Terminal landing module with a single engine for final descent, hover and translation.

Paragraph - 3.4.3.4.1.1

Requirement - The velocity increments required for translunar midcourse control will be provided by the landing engine.

3.9.4 (Continued)

Two-Man Apollo - Translunar midcourse is provided by the main retrograde engine.

Paragraph - 3.4.3.4.1.2

Requirement - Velocity vector control and vernier velocity control during lunar orbit injection will be provided by the landing engine.

Two-Man Apollo - Control is provided during lunar orbit injection by the gimbaled main retrograde engine.

Paragraph - 3.4.3.4.1.3

Requirement - Velocity vector control and vernier velocity control will be provided by the landing engine during lunar orbit retrograde.

Two-Man Apollo - Same as 3.4.3.4.1.2

Paragraph - 3.4.3.4.2.1

Requirement - The lunar retrograde engines should be pressure fed.

Two-Man Apollo - The lunar retrograde engine is pump fed.

Paragraph - 3.4.3.4.2.2

Requirement - The lunar landing engine will utilize the same propellant supply as the lunar retrograde engines.

Two-Man Apollo - The lunar landing engine does not use the same propellant supply as the lunar retrograde engines.

3.10 Reaction Control

3.10.1 Command Module - The command module reaction control system (RCS) provides the required spacecraft attitude stabilization and control during re-entry and touchdown. The RCS incorporates ablation cooled thrust chambers and earth storable hypergolic propellants which are positively expelled by gaseous helium at an initial storage pressure of 3000 psia, regulated to a system operating pressure of 300 psia.

3.10.1.1 Design Criteria - The command module reaction control system was evolved using the following design criteria:

A. Earth storable propellants with advanced development background.

B. Pure roll couples.

C. The reaction control system is comprised of two functionally independent subsystems.

D. Maximum utilization of hardware developed on other space programs for minimum development time and cost.

E. Filtering of pressurants and propellants for contaminant protection.

F. Pressurant storage pressure optimized for minimum pressurant system weight.

G. Helium gas pressurant for minimum weight.

H. Compatible corrosion resistant steels for lines, valves, fittings and component parts except where impractical, such as valve seals, burst diaphragms, valve poppets, etc.

I. All propellant expulsion bladders are of laminated teflon.

J. All tanks are titanium for high strength to weight ratio.

K. Multiple component packaging wherever practical for weight reduction, leakage path elimination, installation accommodation, reduced volume, and increased service and test point accessibility.

L. Maximum component interchangeability between the command, service and terminal landing modules.

3.10.1.1 (Continued)

M. Manually controlled pressure regulator by-pass to assure system operation in the event of a regulator failure.

N. Series - parallel check valves for flow path assurance and maximum protection against back-flow leakage and prevention of accidental propellant vapor mixing in pressurant system.

O. Burst diaphragm - relief valve series arrangement for zero leakage assurance prior to overpressure venting and minimum pressurant loss after system relief actuation.

P. No external leak paths through dynamic seals.

Q. All service connections controlled by multiple seal manual valves for maximum leakage protection.

R. Pressurant and propellant tank fluids contained by positive metal seal squib operated valves for zero leakage prior to use.

S. Purge and fill connections provided on each tank for servicing.

T. The design life of the ablative cooled thrust chambers includes allowances for all anticipated usage.

U. All thrust chamber assemblies of the same thrust level are identical.

V. Attitude stabilization requirements of Section 3.2.

3.10.1.2 Interchangeability - System components which perform identical functions are interchangeable with one another, not only in this module but also with the service and terminal landing modules.

3.10.1.3 Ground Checkout - Ground checkout consists of leakage measurements and functional checks of all components using gas at suitable pressures. Manual valves and ground test connections are provided to perform these functions but are not shown in the schematic Figure 3-63.

COMMAND MODULE
REACTION CONTROL SYSTEM SCHEMATIC

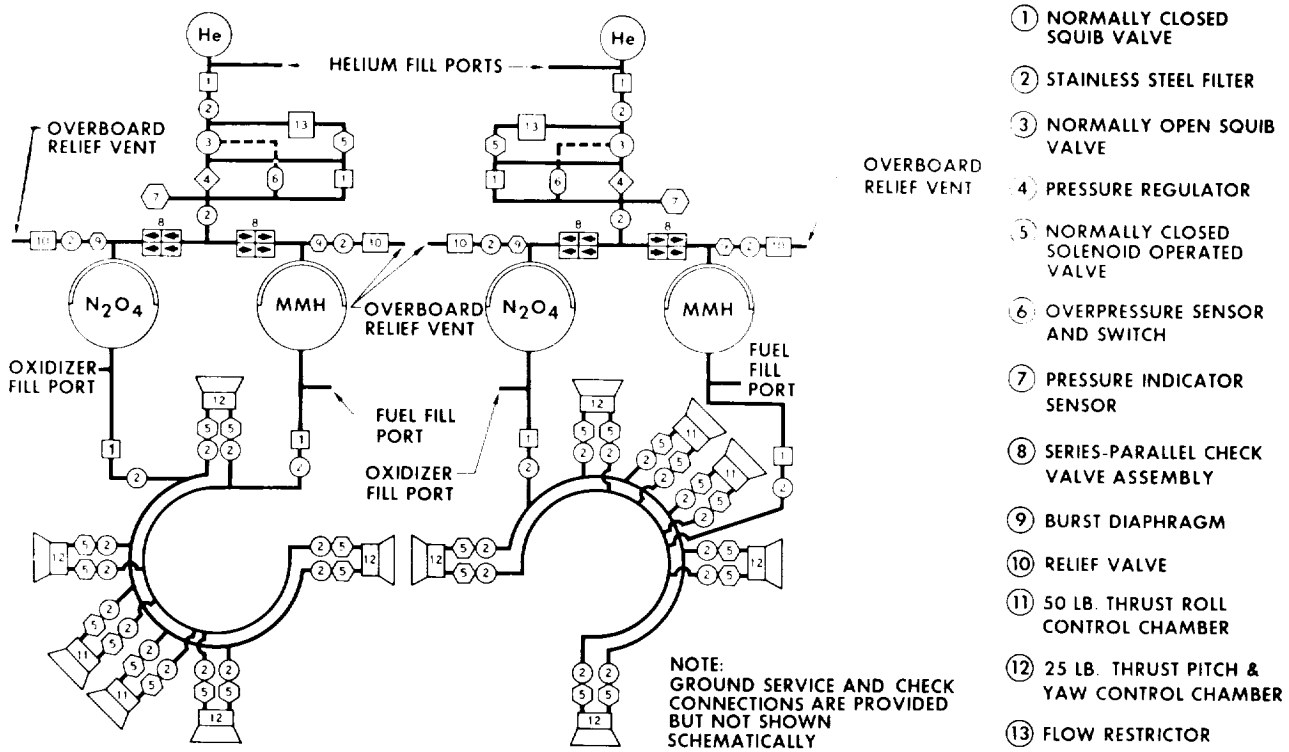


FIGURE 3-63

3.10.1.4 Propellant Characteristics - The propellant combination is nitrogen tetroxide (N_2O_4) as the oxidizer and monomethylhydrazine (MMH) as the fuel. The characteristics of the peopellants are shown in Table 3-16. The effect of oxidizer to fuel ratio on delivered specific impulse is shown in Figure 3-64. The fuel and oxidizer conform to specification MIL-P-27403 and MIL-P-26539A, respectively. Extensive use of this propellant combination in existing space programs permits utilization of proven handling techniques and protective measures.

TABLE 3-16

Reaction Control System Propellant Characteristics

Oxidizer - Fuel Ratio	2.05
Theoretical Shifting Equilibrium Steady State Vacuum Specific Impulse, I_{sp} - Seconds	
(ϵ = 40:1; P_c = 150 psia)	318
Delivered Specific Impulse - I_{sp} - Seconds	
Steady State ($t_B \geq 80$ ms)	300
Pulsed (t_B = 20 ms)	260
Specific Gravity	
At Loading Conditions	
(P = 14.7 psia; T = 68°F)	
Fuel (MMH)	.878
Oxidizer (N_2O_4)	1.445
At System Maximum Tank Design Temperature (165°F)	
Fuel (MMH)	.825
Oxidizer (N_2O_4)	1.310
Vapor Pressure at Maximum Tank Design Temperature (165°F)	
Fuel (MMH)	9.9
Oxidizer (N_2O_4)	120
Freezing Point, °F	
Fuel (MMH)	-63
Oxidizer (N_2O_4)	12
Normal Boiling Point (P = 14.7 psia), °F	
Fuel (MMH)	189
Oxidizer (N_2O_4)	70
Boiling Point at System Pressure (300 psia), °F	
Fuel (MMH)	420
Oxidizer (N_2O_4)	212

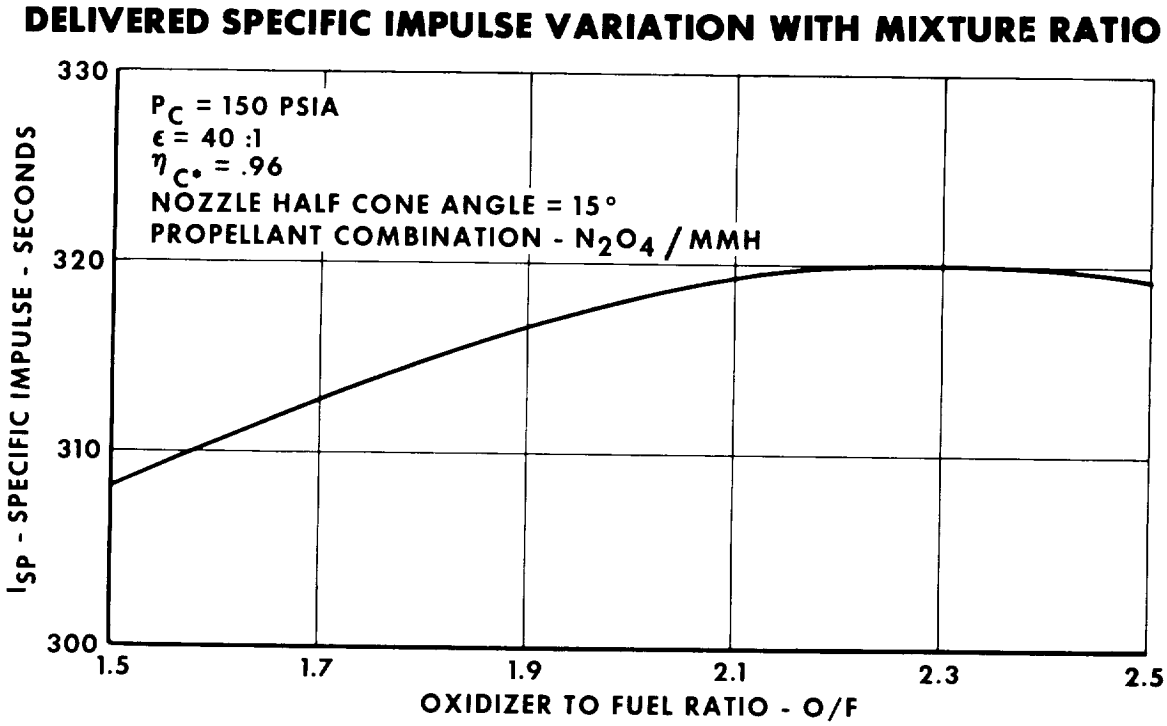


FIGURE 3-64

3.10.1.5 Propellant Supply System - The propellant supply system, shown schematically in Figure 3-63, consists of the pressurization system, the propellant tankage and the associated lines and controls for propellant distribution to the RCS thrust chambers.

3.10.1.6 Pressurization System - The pressurization system employs one spherical helium storage tank for each subsystem, normally closed squib valves, pressure regulators, control valves, filters, check valves, pressure sensors, pressure relief provisions and the associated plumbing required to distribute the pressurizing gas to the propellant tanks. Helium is chosen in preference to nitrogen as the pressurant for weight considerations. It is stored at 3000 psia and regulated to the system operating pressure of 300 psia.

The tanks are fabricated from 6AL4V titanium alloy, designed for a burst pressure factor of 2.0 times the maximum system operating pressure. Titanium

3.10.1.6 (Continued)

alloy was selected for its high strength to weight ratio at elevated temperatures. Drop hammer forging or explosive forming in sections and welded joints will be employed.

The volume of each helium tank is 244 cubic inches and the helium weight is .27 pounds at design pressure. The physical characteristics of helium are summarized in Section 3.9.1.4.

3.10.1.7 Propellant Tanks - The propellant is stored in four spherical tanks, one fuel and one oxidizer tank per subsystem. They are positive expulsion type tanks with teflon expulsion bladders. The bladders are installed through the stand-pipe openings in the bottoms of the tanks and are held in place by sealed flanges. A flow path is provided through the propellant portion of the tanks for bleeding, purging, and drying. The tanks are fabricated of 6AL4V titanium and are designed for a normal working pressure of 300 psia and a burst pressure factor of 1.5. The propellant loading is based upon the loading criteria shown in Table 3-17. The tanks contain 97 pounds of propellant. The allocation of this propellant is 12 pounds for pre-re-entry maneuvers, 81 pounds for re-entry requirements and a 4 pound allowance for unusable residuals. The fluid volume capacity is 704.5 cubic inches for the oxidizer tank and 544.5 cubic inches for the fuel tank. The charged fluid volume capacity at 165°F for the design propellant weight is 690 cubic inches for the oxidizer tank and 532.5 cubic inches for the fuel tank.

3.10.1.8 Thrust Chamber Design - All thrust chamber assemblies are installed with the nozzle exit flush with the command module mold line. Each thrust chamber assembly consists of two thrust chamber propellant valves, two calibration orifices, a fuel and oxidizer injector system, a combustion chamber and an expansion nozzle. The propellant valves are quick-acting, normally closed solenoid valves which open upon application of an appropriate electrical signal, to permit flow of oxidizer or

TABLE 3-17

RCS PROPELLANT LOADING CRITERIA

ANGULAR MANEUVERS:

PHASE	NUMBER	AXIS	RATE, DEG/SEC.
EARTH PARKING ORBIT	1	PITCH	0.25
TRANSLUNAR	40	COMBINATION	0.5
LUNAR ORBIT	20	ROLL	1.0
NEAR MOON	3	ROLL	10.0
TRANSEARTH	40	COMBINATION	5.0
PRE RE-ENTRY	10	ALL	2.0
RE-ENTRY	40	ROLL	20.0

VELOCITY CHANGES:

TRANSLUNAR	5 FPS	(FINE CONTROL)
TRANSEARTH	330 FPS	(TOTAL)
ULLAGE	6.0 FPS FOR EACH OF 6 STARTS	

LIMIT CYCLING:

20 MSEC. PULSES; 5 DEG. MAX. DEADBAND;
5 MIN. MAX SPACING

EARTH PARKING ORBIT: 3 HRS

TRANSLUNAR & TRANSEARTH: 72 HRS EACH

PRE RE-ENTRY: 2 HRS

MISCELLANEOUS:

ROLL TRIM FOR .6 IN C.G. OFFSET &
5 DEG. AVERAGE GIMBAL DEFLECTION

RE-ENTRY PITCH AND YAW DAMPING:
30 LBS. BASED ON SIMULATOR STUDIES

PERFORMANCE MARGIN: 30 PERCENT PLUS
ARBITRARY ALLOWANCE FOR UNSPECIFIED
MANEUVERS; 60 LB. TRANSLUNAR;
20 LB. TRANSEARTH

TRAPPED PROPELLANT: 1 PERCENT FOR
TERMINAL LANDING & SERVICE MODULES;
4 PERCENT FOR COMMAND MODULE

3.10.1.8 (Continued)

fuel to the specific injector. The injector utilizes jets which impinge the fuel and oxidizer on one another for controlled mixing and good combustion efficiency. The calibration orifices are trim devices used in conjunction with the propellant valves and the injectors to adjust the flows to the design levels. The trim orifices are adjusted and fixed during acceptance testing. The combustion chambers and the nozzles are lined with ablative material and insulation material to control the external wall temperature to less than 500°F. Seals are provided between the thrust chamber nozzle and the command module skin to prevent the backflow of exhaust gas into the spacecraft.

Regenerative cooled thrust chambers were eliminated because of insufficient cooling associated with the low flow rates as well as fluid dynamic problems.

Radiation thrust chambers, although the lightest in weight and least complex, are not utilized because of the submerged installation requirements.

The relative position of the command module reaction control system thrust chambers is shown in Figure 3-65.

3.10.1.9 Thrust Chamber Performance - Typical response characteristics of the command module RCS thrust chambers based on tests results from the Gemini Program are shown in Figure 3-66. The signal pulse is of positive square wave form. The propellant performance characteristics are presented in Section 3.10.1.4. Figure 3-67 shows the effect of pulsed operation upon the delivered specific impulse as a function of pulse width.

REACTION CONTROL SYSTEMS SUMMARY

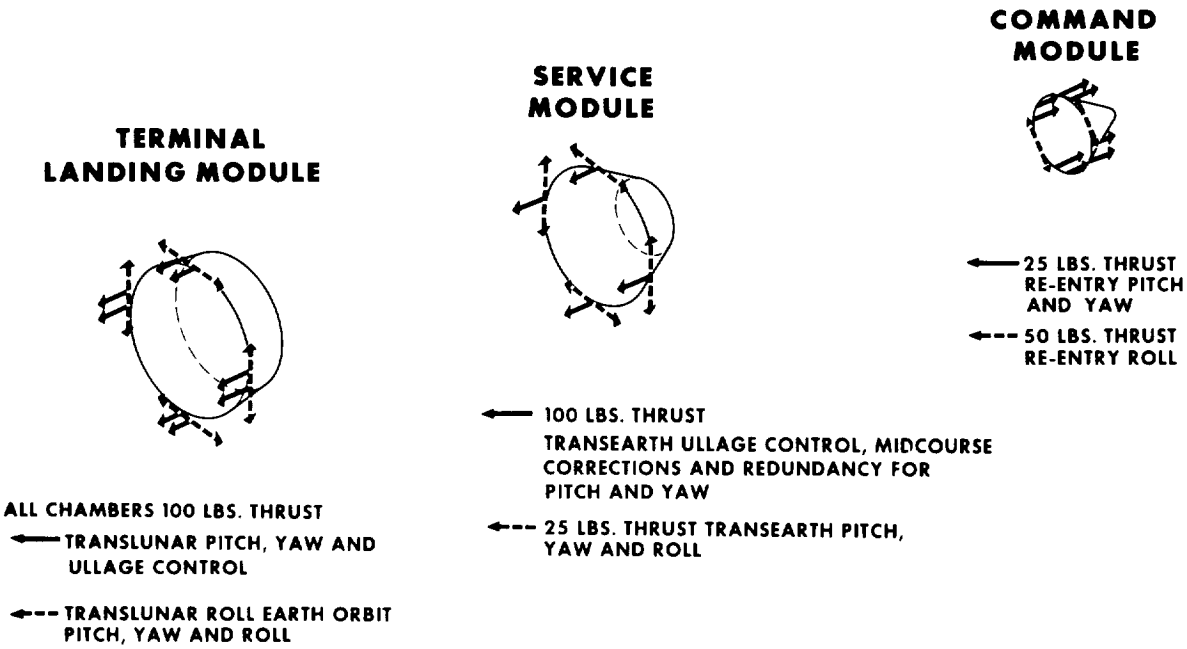


FIGURE 3-65

TYPICAL REACTION CONTROL THRUST CHAMBER RESPONSE CHARACTERISTICS

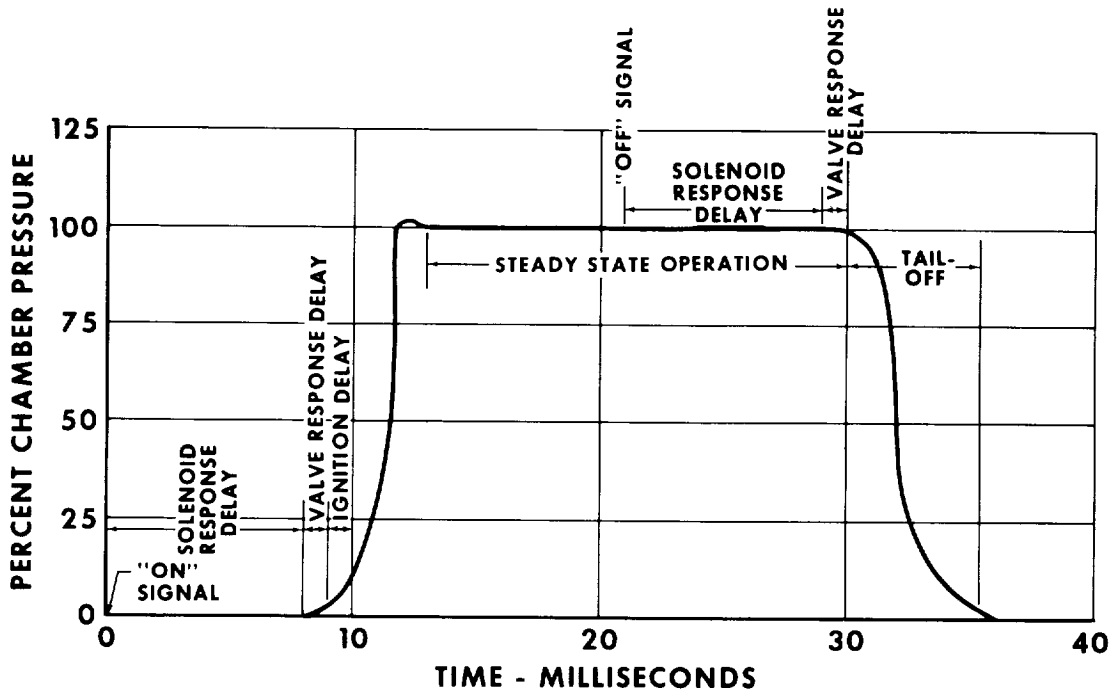


FIGURE 3-66

TYPICAL EFFECT OF PULSE WIDTH ON DELIVERED SPECIFIC IMPULSE

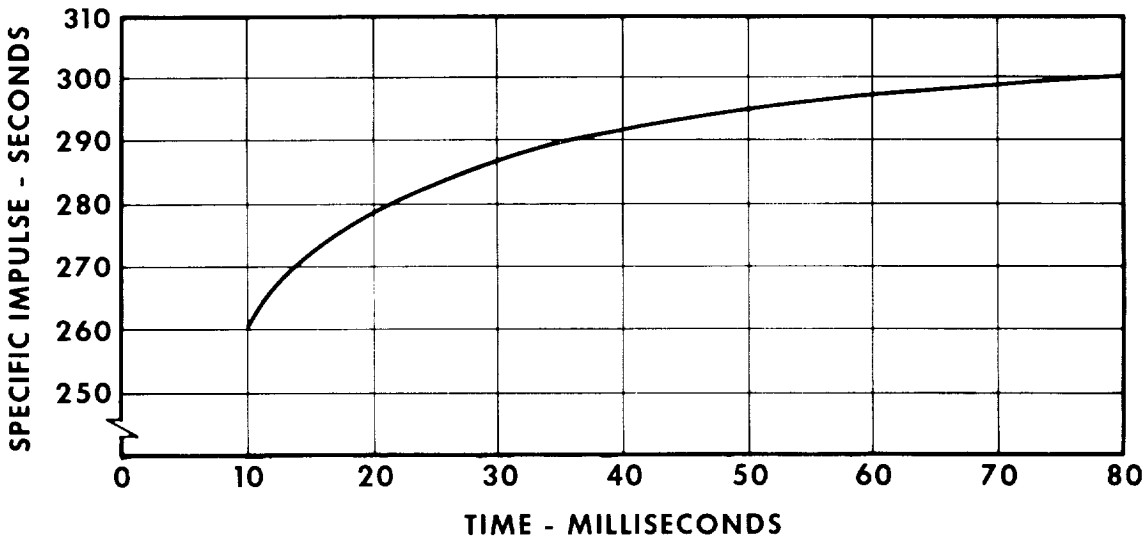


FIGURE 3-67

3.10.1.10 Description and Operation - The command module RCS system provides the following control functions:

A. Pre-re-entry spacecraft orientation.

B. Re-entry pitch and yaw damping.

C. Re-entry roll maneuvers for touchdown dispersion control.

The two independent systems shown schematically in Figure 3-63 are identical and provide functional redundancy.

Pitch and yaw control is provided by eight 25 pound thrust ablation cooled thrust chambers. Roll control is provided by four 50 pound thrust ablation cooled thrust chambers. Table 3-18 presents the design characteristics of the thrust chambers.

TABLE 3-18

Thrust Chamber Design Characteristics

	Pitch & Yaw	Roll
Thrust Level (lbs.)	25	50
Expansion Ratio	40:1	40:1
Chamber Pressure (psia)	150	150
Diameters (in.)		
Nozzle Throat	.337	.472
Nozzle Exit	2.13	2.99
Combustion Chamber	.70	.944
Lengths (in.)		
Nozzle	2.68	3.75
Overall	5.5	7.5
Assembly Weight (lbs.)	3	6

The system is activated by firing the normally closed squib valve in the helium line of each RCS subsystem (Figure 3-63). The gaseous helium, stored at 3000 psia, is reduced to a system operating pressure of 300 psia through the pressure regulator. An overpressure sensor monitors the regulator downstream pressure and automatically initiates a normally open squib valve, upstream of the pressure regulator, in the event of a regulator malfunction in the open position. A solenoid operated by-pass valve permits manual pressurization of the propellants through the regulator if the overpressure sensor has closed the primary flow path. If the regulator fails in the closed position, a normally closed squib valve, located downstream of the solenoid operated valve, permits by-passing the regulator to provide manual control of the propellant pressurization. Series-parallel check valves are located in the pressurant lines upstream of the fuel and oxidizer tanks. Following system charging and checkout, the normally closed squib valves in the propellant lines are activated to ready the systems for use. Selective actuation of the solenoid operated injector valves provides the required attitude control thrust. Filters are located immediately downstream of all normally closed squib valves. Propellant supply system overpressure protection is provided by a burst diaphragm-relief valve combination. The burst diaphragm is located between the

3.10.1.10 (Continued)

pressure relief valve and the propellant pressurization system to protect against relief valve leakage prior to initial system overpressure. A filter between the burst diaphragm and pressure relief valve provides contamination protection to the relief valve from possible burst diaphragm fragments. The pressure relief valves are designed to open at propellant tank proof pressure and fully reseat at system operating pressure.

Figure 3-68 shows the command module reaction control system installation. The system operating pressure of 300 psia is based upon a combustion chamber pressure of 150 psia, an injector pressure drop of 75 psi and a 75 psi pressure loss through the system lines, fittings and other associated equipment. In case of malfunction in either of the two normally operating systems, the one remaining system is capable of performing a satisfactory re-entry, but not without some loss in touchdown accuracy. Table 3-19 presents the propellant allocation requirements for emergency operation, based upon the assumptions cited in the table. This table also shows the comparison of required emergency usable propellant to the total usable propellant for normal operation.

TABLE 3-19

Propellant Allocation for Emergency Operation With One System

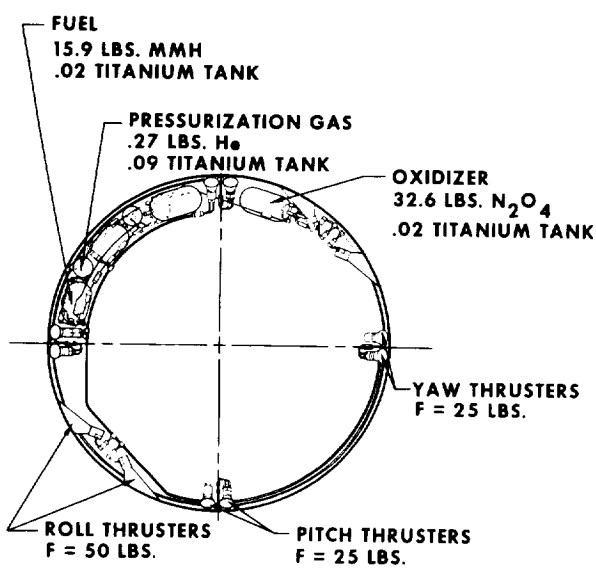
Stabilization and Control	
Pre-re-entry	9 lb.
Re-entry	38 lb.
Total usable for Emergency Operation	47 lb.
Total usable for Normal Operation	93 lbs.

Assumptions:

1. Delete 30% margin
2. Use 10 roll maneuvers at 20°/sec to accomplish safe re-entry instead of 40 roll maneuvers.

COMMAND MODULE
REACTION CONTROL SYSTEM INSTALLATION

NOTES:
1. POSITIVE EXPULSION TANKS
2. DUAL SYSTEMS FOR RELIABILITY
3. SYSTEM WEIGHT 210 LBS.
   USEABLE PROPELLANT 93 LBS.
   TRAPPED PROPELLANT 4 LBS.
4. THRUST CHAMBER CHACTERISTICS
   ABLATIVE COOLED
   ϵ 40:1
   P_c 150 PSIA
   I_{SP} STEADY STATE 300 SEC.
   I_{SP} PULSING 260 SEC.



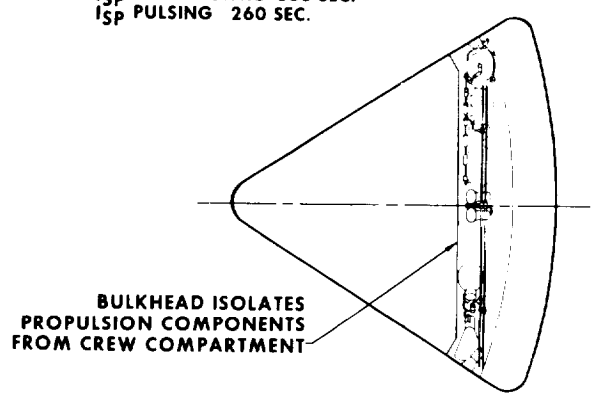


FIGURE 3-68

3.10.2 Service Module - The service module reaction control system (RCS) supplies attitude stabilization and control, main engine propellant positioning and midcourse corrections during lunar launch and the transearth mission phases. The propellants, pressurization gas, propellant expulsion method and system pressures are the same as those of the command module RCS discussed in Section 3.10.1.

3.10.2.1 Design Criteria - In addition to the design criteria presented in Section 3.10.1, the following design objective applies to this RCS system:

A. Thrust chambers have multiple function capabilities, i.e. attitude control, propellant positioning and midcourse correction.

3.10.2.2 Interchangeability - The major functional components of the service module RCS are interchangeable with one another and are interchangeable with similar components in the command and terminal landing modules except for the pressurant and propellant tanks.

3.10.2.3 Ground Checkout - Ground service and test point provisions are provided for component functional tests as discussed in Section 3.10.1.

3.10.2.4 Propellant Characteristics - The propellants in the service module RCS are the same as those utilized in the command module RCS. The characteristics are discussed in Section 3.10.1.

3.10.2.5 Propellant Supply System - The propellant supply system is comprised of the following components in each of two independent systems; a pressurant tank, pressure regulator, a normally closed solenoid operated regulator by-pass valve, pressure sensors, a propellant tank, an oxidizer tank, and the associated lines and valves for propellant flow control and distribution to the thrust chambers. The reaction control system is shown, schematically, in Figure 3-69. The materials used are discussed in Section 3.10.1.

3.10.2.6 Pressurization System - The pressurant used is gaseous helium at an initial storage pressure of 3000 psia. Each of two helium tanks have an internal

3.10.2.6 (Continued)

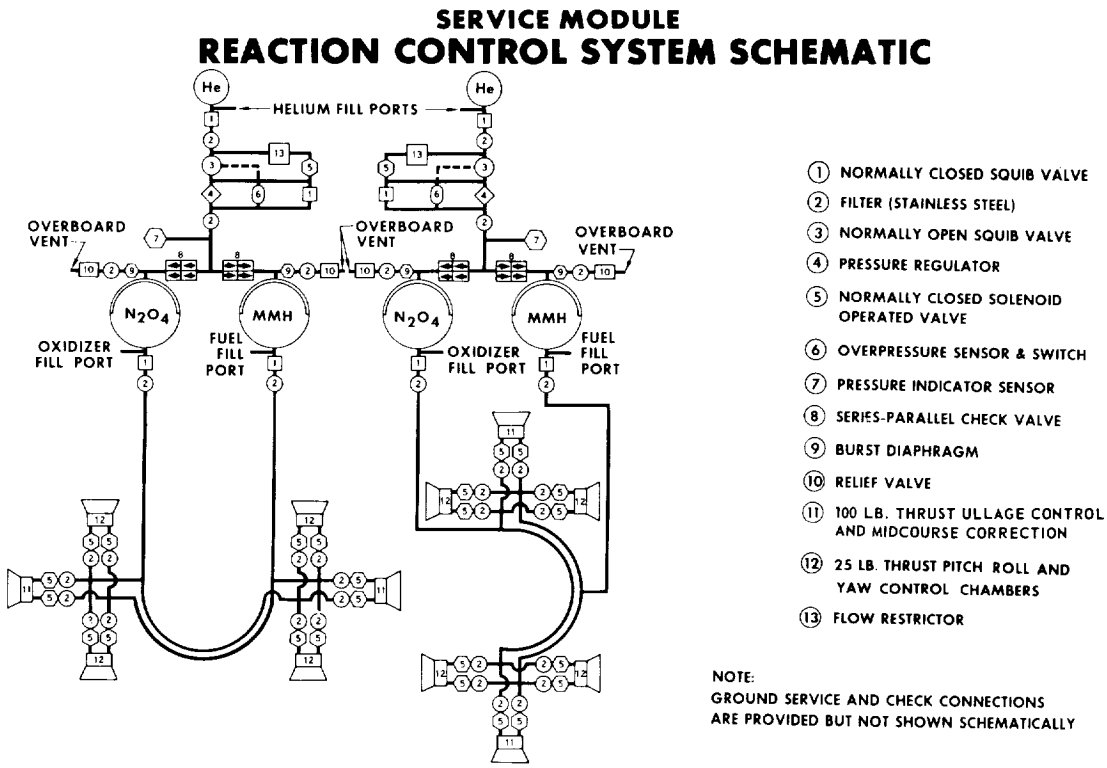


FIGURE 3-69

volume of 1249 cubic inches and contain 1.4 pounds of helium when charged to the storage pressure. The tank wall thickness is 0.15 inch which provides for a burst pressure of 200 percent of the initial storage pressure. The tanks are fabricated of the same materials and by the same methods described in Section 3.10.1.

3.10.2.7 Propellant Tankage - The fuel and oxidizer tanks for the service module RCS are spherical and have the capacities and design characteristics shown in Table 3-20. The propellant utilization is shown in Table 3-21. The fabrication method, expulsion method, and material selection is discussed in Section 3.10.1.

3.10.2.8 Thrust Chamber Design - The thrust chambers of the service module RCS are located near the base of the service module as shown in Figure 3-70. The description of the thrust chamber assembly discussed in Section 3.10.1 is applicable, however, seals are not required for installation in the service module.

TABLE 3-20

Propellant Tank Capacity and Design Characteristics

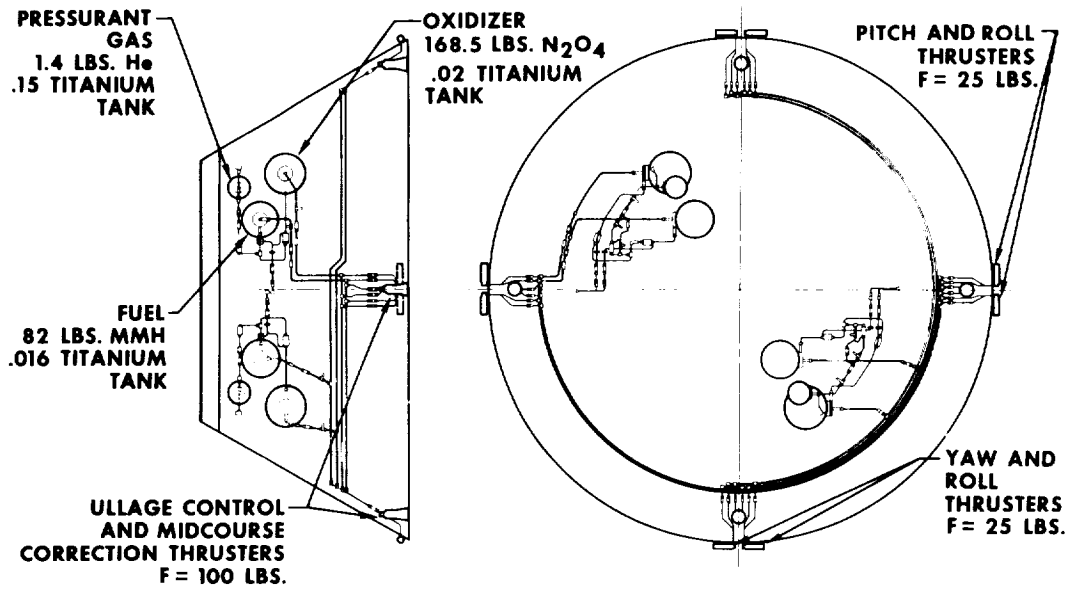
Item	Fuel	Oxidizer
Fluid Weight (Lbs.)	82	168.5
Internal Volume (in^3)	2808	3634
Burst Factor (based on system operating pressure)	1.5	1.5
Operating Pressure (psia)	300	300
Wall Thickness (in.)	.016	.020
Design Temperature (°F)	165	165
Material	6 AL 4V Titanium	6 AL 4V Titanium
Expulsion Efficiency (%)	99	99

TABLE 3-21

Service Module RCS Propellant Utilization

Function	Propellant Quantity Lbs.
Stabilization and Control	
Transearth Coast	98
Lunar Orbit	5
Roll Control During Main Engine Firing	16
Transearth Midcourse Correction	365
Main Propulsion Propellant Positioning	11
Total Propellant Utilized	495
Trapped Propellant	6
Total Propellant Loading	501

SERVICE MODULE
REACTION CONTROL SYSTEM INSTALLATION



NOTES:
1. POSITIVE EXPULSION TANKS
2. DUAL SYSTEMS FOR RELIABILITY
3. SYSTEM WEIGHT 639 LBS.
   USEABLE PROPELLANT 495 LBS.
   TRAPPED PROPELLANT 6 LBS.
4. THRUST CHAMBER CHARACTERISTICS
   ABLATIVE COOLED
   ϵ 40:1 P_c 150 PSIA
   I_{SP} STEADY STATE 300 SEC
   I_{SP} PULSING 260 SEC

FIGURE 3-70

3.10.2.9 Thrust Chamber Performance - Response characteristics and propellant performance characteristics described in Section 3.10.1 are applicable.

3.10.2.10 Description and Operation - The following functions are performed during the lunar launch and transearth phases of the mission:

A. Roll control during main engine thrusting.

B. Main propulsion propellant positioning in lunar orbit.

C. Pitch, yaw and roll control during coast periods.

D. Transearth midcourse corrections.

The RCS is composed of two separate subsystems as shown in Figures 3-69 and 3-70. The relative location of the thrust chambers is presented in Figure 3-65. Each subsystem is capable of performing the transearth mission functions, but with some degradation in midcourse correction accuracy and a reduced number of sighting maneuvers. Table 3-22 presents the emergency propellant requirements as compared to the normal, dual system propellant loading. Each subsystem has four 25 pound thrust

3.10.2.10 (Continued)

TABLE 3-22

Propellant Allocation for Emergency Operation with One System

Stabilization and Control	
Transearth Coast	16 Lb.
Lunar Orbit	4 Lb.
Roll Control During Main Engine Operation	12 Lb.
	32 Lb.
Transearth Midcourse Corrections	194 Lb.
Main Propulsion Propellant Positioning	7 Lb.
Total Useable for Emergency Operation	233 Lb.
Total Useable for Normal Operation	495 Lb.

Assumptions:

1. Delete 30% margin and 20 lb. unspecified margin.
2. Use DSIF for transearth guidance and navigation.
3. Use optimistic allotment for main propulsion propellant positioning which is 71% of conservative allotment.
4. Lower transearth midcourse correction allotment by degrading probability of success to 90%. (The 90% is somewhat conservative since the DSIF should increase guidance accuracy.)

roll control thrust chambers and two aft firing 100 pound thrust midcourse correction thrust chambers in addition to the rest of the components previously described. Figure 3-64 illustrates the manner in which the RCS is utilized. Roll control during coast is provided by four of the eight 25 pound thrust roll control thrust chambers. Pitch and yaw control is provided by utilizing paired roll thrust chambers. Four aft firing 100 pound thrust chambers provide propellant positioning and midcourse corrections as well as redundancy in pitch and yaw. During main engine operation for lunar launch and transearth injection, pitch and yaw control is accomplished with the gimbaled main engine.

3.10.2.10 (Continued)

Table 3-23 presents the design characteristics of the 100 pound thrust chambers used for midcourse corrections.

TABLE 3-23

Thrust Chamber Design Characteristics

Thrust (lbs.)	100
Expansion Ratio	40:1
Chamber Pressure (psia)	150
Diameters (in.)	
Nozzle Throat	.674
Nozzle Exit	4.26
Combustion Chamber	1.25
Lengths (in.)	
Nozzle	5.35
Overall	11.0
Assembly Weight	10

The system activation and operation is identical to the command module RCS, described in Section 3.10.1.

3.10.3 Terminal Landing Module - The reaction control system in the terminal landing module provides the thrust required for vehicle attitude stabilization and control and main propulsion propellant positioning during the earth orbit, translunar and lunar landing phases of the mission. This RCS has the same pressurant, propellants and system operating pressures as the command module RCS described in Section 3.10.1.

3.10.3.1 Design Criteria - The design criteria presented in Section 3.10.1 and 3.10.2 are applicable to this RCS.

3.10.3.2 Interchangeability - Components having the same functions are interchangeable with one another and are interchangeable with similar components in the service module RCS excepting the pressurant and propellant tanks.

3.10.3.3 Ground Checkout - Ground service and test point provisions are provided as discussed in Section 3.10.1.

3.10.3.4 Propellant Characteristics - The propellants used are identical to those for the service and command modules.

3.10.3.5 Propellant Supply System - The components of the terminal landing module RCS consist of two pressurant tanks, squib actuated valves, pressure sensors, check valve assemblies, burst diaphrams, relief valves, two oxidizer tanks, two fuel tanks, and the required lines and fittings for propellant distribution to the sixteen 100 pound thrust RCS thrust chamber assemblies. These components are assembled in two separate subsystems, shown schematically in Figure 3-71. The materials used are discussed in Section 3.10.1

3.10.3.6 Pressurization System - The terminal landing module RCS pressurization system has 2.1 pounds of helium in each of two 1862.5 cubic inch spherical tanks when the tanks are charged to a pressure of 3000 psia at 68°F. The tank wall thickness is .173 inch which is compatible with a design burst pressure factor of 2.0. The tank material and fabrication method are described in Section 3.10.1.

TERMINAL LANDING MODULE
REACTION CONTROL SYSTEM SCHEMATIC

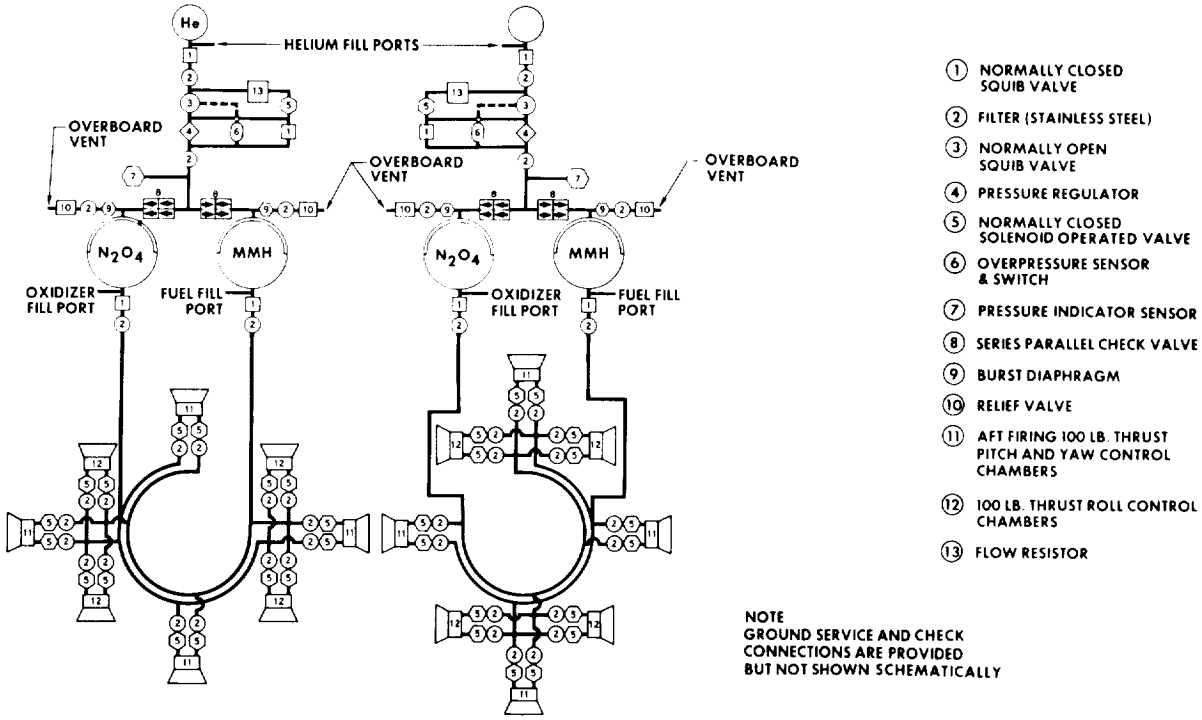


FIGURE 3-71

3.10.3.7 Propellant Tankage - The material and fabrication technique for the two fuel and two oxidizer tanks is the same as the command module RCS propellant tanks, discussed in Section 3.10.1. The tanks are spherical and have the capacities and design characteristics shown in Table 3-24 The propellant utilization is shown in Table 3-25.

3.10.3.8 Thrust Chamber Design - The terminal landing module RCS thrust chambers are identical to the 100 pound thrust midcourse correction thrust chamber assemblies discussed in Section 3.10.2. Figure 3-72 shows the location of these thrust chambers near the base of the terminal landing module.

3.10.3.9 Description and Operation - The terminal landing module RCS supplies the thrust required for the following functions during the earth parking orbit, translunar and lunar landing phases of the mission.

TABLE 3-24

Propellant Tank Capacity and Design Characteristics

Item	Fuel	Oxidizer
Fluid Weight (Lbs.)	122.5	250.5
Internal Volume (in^3)	4168.5	5402
Burst Factor (based upon system operating pressure)	1.5	1.5
Operating Pressure (psia)	300	300
Wall Thickness (in)	.016	.020
Design Temperature (°F)	165	165
Material	6 AL 4V Titanium	6 AL 4V Titanium
Expulsion Efficiency (%)	99	99

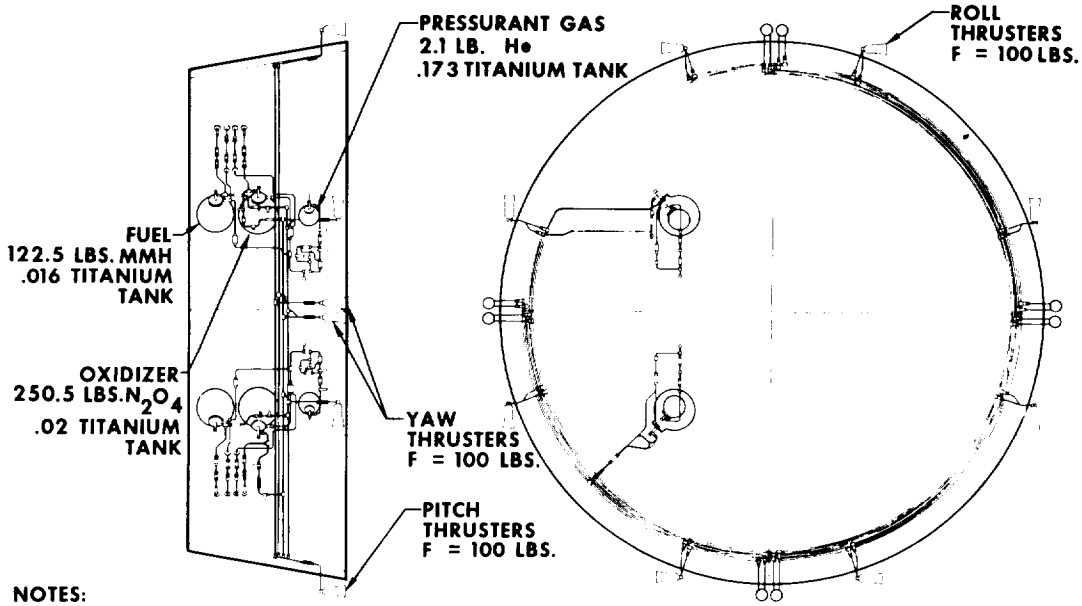
TABLE 3-25

Terminal Landing Module RCS Propellant Utilization

Function	Propellant Quantity Lbs.
Stabilization and Control	
Earth Parking Orbit	53
Translunar Coast	288
Lunar Orbit	27
Roll Control During Main Engine Firing	44
Main Propulsion Propellant Positioning	326
Total Propellant Utilization	738
Trapped Propellant	8
Total Propellant Loading	746

3.10.3.9 (Continued)

TERMINAL LANDING MODULE
REACTION CONTROL SYSTEM INSTALLATION



NOTES:
1. POSITIVE EXPULSION TANKS
2. DUAL SYSTEMS FOR RELIABILITY
3. SYSTEM WEIGHT 1005 LBS.
   USEABLE PROPELLANT 738 LBS.
   TRAPPED PROPELLANT 8 LBS.
4. THRUST CHAMBER CHARACTERISTICS
   ABLATIVE COOLED
   ϵ 40:1 I_{sp} STEADY STATE 300 SEC.
   P_c 150 PSIA I_{sp} PULSING 260 SEC.
5. ROLL THRUSTER USED IN PITCH AND YAW DURING EARTH HOLDING ORBIT

FIGURE 3-72

A. One pitch maneuver during earth orbit while the S-IVB booster stage is attached.

B. Main propulsion propellant positioning during the translunar phase.

C. Spacecraft pitch, yaw and roll control when main retrograde or terminal landing engines are not operating.

D. Roll control during main retrograde and terminal landing engine operation.

E. Roll control during lunar landing.

The two independent subsystems of the RCS are shown in Figures 3-71 and 3-72. Each subsystem is capable of performing the RCS functions during the translunar and lunar landing phases of the mission, but degradation of performance will be incurred with single system operation. Table 3-26 lists the emergency propellant requirements as compared to the normal, dual system propellant loading and the assumptions used. The sixteen thrust chambers, 100 pounds thrust each, operate at a chamber pressure of 150 psia.

3.10.3.9 (Continued)

TABLE 3-26

Propellant Allocation for Emergency Operation with One System

Stabilization and Control	
Earth Parking Orbit	41 Lb.
Translunar Coast	39 Lb.
Lunar Orbit	21 Lb.
Roll Control During Main-Engine Operation	34 Lb.
	135 Lb.
Main Propulsion Propellant Positioning	230 Lb.
Total Useable for Emergency Operation	365 Lb.
Total Useable for Normal Operation	738 Lb.

Assumptions:

1. Delete 30% margin and 60 lb. unspecified margin.
2. Use DSIF for translunar guidance and navigation.
3. Do not use RCS for translunar midcourse correction refinement.
4. Use optimistic allotment for main propulsion propellant positioning which is 71% of conservative allotment.

During the earth orbit phase, the eight roll thrust chambers provide attitude stabilization about the three vehicle axes, as shown in Figure 3-65. After separation of the S-IVB booster stage, the spacecraft center of gravity shifts very near to the plane containing the roll thrust chambers, rendering them ineffective in pitch and yaw. The eight aft-firing ullage control thrust chambers are then utilized for the pitch and yaw maneuvers while the retrograde module is still attached. During main engine thrusting, pitch and yaw control is accomplished with the gimbaled engine. Following separation of the retrograde module, the eight roll thrust chambers provide the roll control during the landing phase. Pitch and yaw control is obtained by gimbaling the lunar landing engine. For control of the main engine propellant position, two opposing pairs of the aft-firing thrust chambers are

3.10.3.9 (Continued)

used. Pairs of aft-firing thrust chambers provide a minor midcourse correction of 5 fps or less, if required.

The method of system activation and operation discussed in Section 3.10.1 is the same as the method used for this system. The thrust chamber assembly design characteristics are the same as those presented in Table 3-21.

3.10.4 Deviations to the Apollo Statement of Work - The following deviations are taken to the respective paragraphs in the Apollo Statement of Work, Reference 4-4 in the design of the Two-Man Apollo vehicle.

Paragraph - 3.4.2.3.1.2.1

Requirement - Two independent systems - each capable of meeting total torque and propellant storage requirements.

Two-Man Apollo - Two systems are provided - each capable of meeting total torque requirements but only one propellant supply is provided.

Paragraph - 3.4.2.3.2.2.1

Requirement - The service module roll thrusters will be used for pitch and yaw maneuvering whenever the lunar landing module is attached.

Two-Man Apollo - The service module does not provide attitude control during lunar landing.

Paragraph - 3.4.3.3

Requirement - The reaction control system in the service module provides attitude control for the lunar landing mission.

3.10.4 (Continued)

Two-Man Apollo - The service module does not provide attitude control during lunar landing.

3.11 Launch Escape - Separation of the command module from the launch vehicle in the event of a malfunction during the time from final countdown until approximately twenty seconds after second stage ignition is accomplished by the launch escape propulsion system (LEPS). This system consists of a tower-mounted, solid propellant, stepped-thrust rocket motor providing an initial thrust-to-weight ratio of 14:1 and a burning time of 3.5 seconds, together with a jettison rocket for separation of the tower from the capsule after escape rocket burnout.

3.11.1 System Performance - System performance is predicated on the following requirements:

A. A minimum peak altitude of 5000 feet and minimum range at touchdown of 3000 feet following off-the-pad aborts.

B. Adequate separation under all flight conditions to avoid blast damage from an exploding booster.

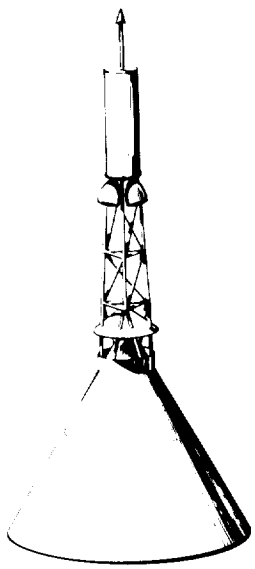
C. Positive static stability under all expected Mach number conditions.

D. Emergency crew tolerances must not be exceeded.

The first requirement determines the stepped-thrust characteristics, total impulse and initial thrust eccentricity while the second determines the initial thrust-to-weight ratio. Requirements C and D serve as constraints on the overall system design.

Figure 3-74 summarizes the major parameters of the LEPS.

LAUNCH ESCAPE PROPULSION SYSTEM



ROCKET MOTOR CHARACTERISTICS

PROPELLANT:	POLYBUTADIENE /AMONIUM PERCHLORATE/ALUMINUM
I_{SP}:	242 LB.-SEC/LB. (SEA LEVEL)
W_F:	1204 LB.
I_{TOTAL}:	274,500 LB.-SEC. (AXIAL)
T_1:	116,400 LB.
T_2:	70,000 LB.

THRUST ECCENTRICITY FOR LATERAL RANGE: ..... 0.3 IN.

LOAD FACTOR TIME HISTORY (SEA LEVEL):

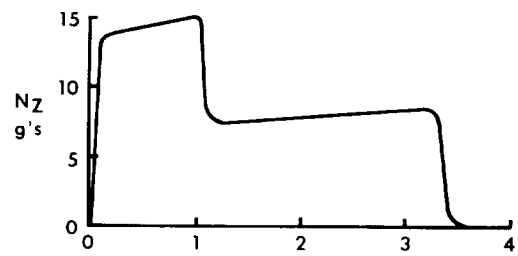


FIGURE 3-74

3.11.2 Rocket Motor Characteristics - The rocket motor design has the following characteristics:

Thrust	See Figure 3-75
Burning Time	See Figure 3-75
Total Impulse (axial)	274,500 lb.-sec.
Number of Nozzles	4
Nozzle Cant Angle	20 deg.
Nozzle Expansion Ratio	12
Port/Throat Ratio	8
Propellant Mass Fraction	.735
Chamber Pressure	1600/958 psia

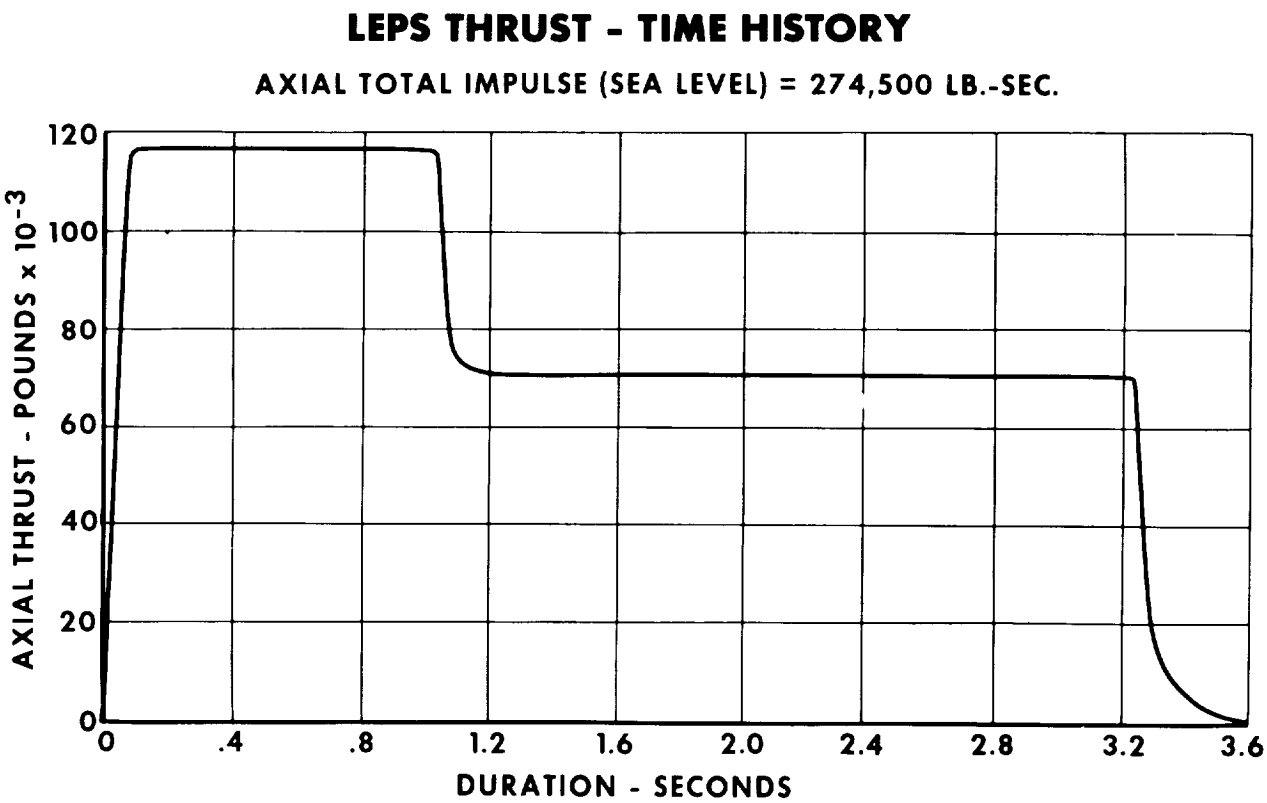


FIGURE 3-75

3.11.2 (Continued)

The stepped thrust-time characteristic is achieved by varying propellant grain geometry, thus enabling the grain to be made from a single propellant. The grain is an eight point double, or stepped, "wagon wheel" configuration. This design provides a net volumetric loading efficiency of approximately 55%. Motor geometry is based on a cylindrical thrust chamber with 2:1 elliptical end closures. Case weight is estimated using 180 K psi yield-stress steel at the nominal 70°F chamber pressure of 1600 psia and factors of 1.5 (safety), 1.25 (ignition spike) and 1.114 (pressure excursion between 70°F and 160°F).

The propellant is a polybutadiene/ammonium perchlorate/aluminum composite with physical and ballistic properties based on 70°F and 1000 psia expanded to sea level as follows:

Characteristic Velocity, C*	5150 ft./sec.
Specific Impulse, I_{sp}	242 sec.
Burning rate, r	.65 in./sec.
Pressure Exponent, n	.32
Flame Temperature, T_c	5500°F
Temperature Coefficient of Pressure	0.12%/°F
Density	.062 lb./in.^{3}
Tensile Strength	139 psi
Elongation	49%

A detailed specification of the tower jettison motor is not presented, however, studies indicate its total impulse requirements to be 2% of the LEPS total impulse. This percentage is used as a size and weight factor in the configuration analyses.

3.12 Lunar Touchdown - The lunar landing gear shown in Figure 3-29 consists of four individual legs, each consisting of three struts terminating at universal-mounted flat bearing pads (Figure 3-30).

The gear is designed to provide impact attenuation and stability for the range of impact parameters outlined in the design criteria (Section 2.7).

Impact energy is dissipated by crushing aluminum honeycomb in each of the telescoping struts. Relative crushing strength of each honeycomb cartridge is selected to control the resulting load line for vertical reacting forces with high inward acting friction. Digital calculations, using a three-degree-of-freedom program indicate that positive dynamic stability exists with the honeycomb distribution used. The maximum vertical load factor is 4.5 g's resulting from a four-point landing on a high friction surface. The lunar landing configuration is illustrated in Figure 2-11, Section 2.4.

3.13 Earth Landing System - The earth landing system consists of a parachute system to reduce vertical landing velocities and an attenuation system to reduce landing impact shock to an acceptable level. The system is designed to satisfy the criteria given in Section 2.7.

3.13.1 Parachute System - Three reefed main parachutes are deployed simultaneously by a drogue parachute or a reserve pilot parachute. During atmospheric aborts, the drogue is deployed to stabilize the command module, however, this is not required during normal stabilized re-entry. The crew is provided with the option of by-passing the drogue sequence and directly activating the reserve pilot parachute for main parachute deployment.

Both the drogue and reserve pilot parachutes are deployed by separate guns which also have the function of blowing off the main parachute compartment cap. In the drogue sequence, the risers are snubbed, permitting the stabilizing action without deployment of the main parachutes. Release of the snubbed risers allows the drogue to deploy the three main parachutes simultaneously. In the reserve pilot parachute sequence, the crew directly activates the reserve pilot parachute for main parachute deployment, by-passing the drogue sequence. Upon landing impact, the main parachutes are jettisoned. The drogue and pilot parachute guns have double cartridges, either of which is adequate to provide successful operation. Each cartridge has dual electrical firing circuits. A mechanically fired squib is used to provide cartridge ignition back-up for the drogue parachute gun.

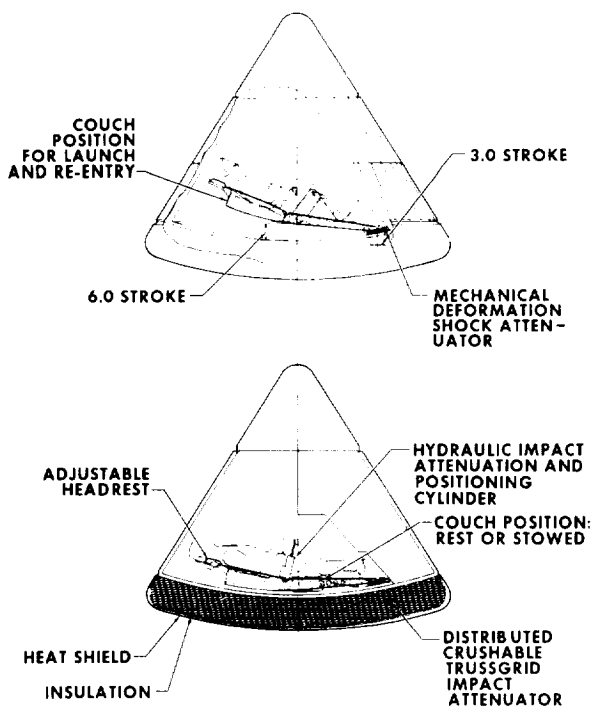
3.13.2 Impact Attenuation - Vertical impact energy is dissipated by crushing an 11.5 inch layer of low density trussgrid material between the large pressure bulkhead and the front face heat shield. The geometric relationship of the low center of gravity and large heat shield diameter results in the ability to resist overturning moments for friction coefficients up to .50.

3.13.2 (Continued)

Command module load factors are limited to 40 g's for the emergency landing condition, i.e., at 5000 feet with two parachutes. The effective area of crushing trussgrid material varies linearly with stroke as the spherical face of the heat shield flattens upon impact. This feature results in a load factor build up rate which is well within crew limits. Trussgrid material is chosen over honeycomb material because of its ability to crush uniformly and resist collapsing under conditions of high side loading.

Additional attenuation to nominal limits is provided for each crew member by viscous dampers and deformable structural elements supporting each side of the crew seat. Hydraulic cross-coupling of the pressure and return side of each damper insures coordinated motion. The impact attenuation system is shown in Figure 3-76.

EARTH IMPACT ATTENUATION SYSTEM



- DISTRIBUTED CRUSHABLE TRUSSGRID ATTENUATES COMMAND MODULE TO 40g MAX.
- INDIVIDUAL SNUBBERS ATTENUATE SEATS TO 20g TRANSVERSE, 10g SPINAL FOR NORMAL LANDING
- TRANSVERSE LOAD FACTOR ON CREW LESS THAN 40g WITH ONE OF THE FOLLOWING EMERGENCY CONDITIONS:
  - FAILURE OF ONE RECOVERY PARACHUTE
  - WATER LANDING
  - FAILURE OF SEAT SNUBBERS TO OPERATE
- TRUSSGRID CHARACTERISTICS
  - 11.5 INCH SOLID HEIGHT
  - 9 INCH EFFECTIVE STROKE
  - CRUSHABLE MATERIAL WEIGHT - 112 LB.
  - DENSITY 1.6 LB./FT.^{3}

FIGURE 3-76

4. SUPPORTING STUDIES AND INFORMATION

4.1 Command Module Size and Arrangement - The command module aerodynamic configuration is that of the Project Apollo re-entry spacecraft with modifications appropriate for the two-man module. These modifications are described in Section 4.4.

Although Reference 4-1 indicates the command module should have crew stand-up capability, parametric design studies of habitable volume requirements were conducted to determine the practicability of this constraint on the interior arrangement. In these studies the crew were positioned such that:

A. Acceleration vector is approximately normal to the trunk, during launch.

B. Deceleration vector is approximately normal to the trunk, and the head is adjacent to the emergency egress hatch, during re-entry.

C. Head is adjacent to and facing windows with trunk parallel to lunar gravity, during lunar landing.

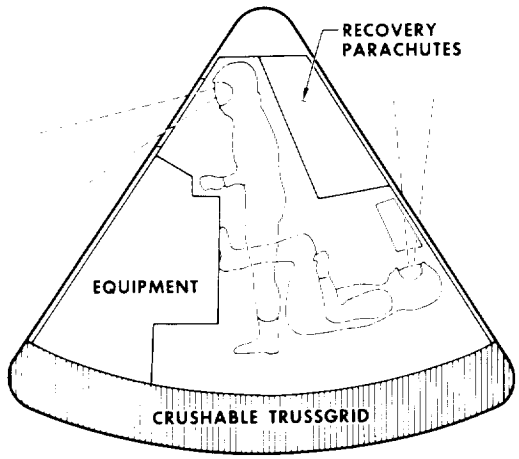
D. Body is supine and normal to lunar gravity, during rest.

Crew stations were superimposed, wherever possible, to reduce total habitable volume. Two of the more attractive arrangements from these studies are shown in Figure 4-1, i.e. stand-up and sit-up arrangements.

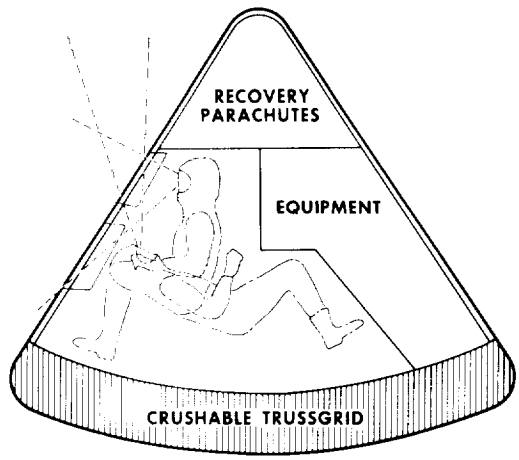
The sit-up configuration has several advantages over the stand-up arrangement, e.g.: a) with modules of equal size, both crewmen can simultaneously assume the sit-up position while only one can stand, b) all windows can be located on the leeward side, c) the equipment can be more effectively arranged for obtaining center of gravity offset, and d) a more efficient recovery system installation can be made. For these reasons the sit-up arrangement is selected.

A study was made to determine the advantage, if any, of including an airlock. The results are presented in Figure 4-2. The number of lunar surface repressuri-

ALTERNATE ARRANGEMENTS



STAND-UP ARRANGEMENT



SIT-UP ARRANGEMENT

FIGURE 4–1

AIRLOCK CONSIDERATIONS

1. AIRLOCK VOLUME = 25 FT.^{3}
2. TOTAL PRESSURIZED CABIN VOLUME = 130 FT.^{3}
3. WEIGHT OF AIRLOCK = 50 LBS.
4. WEIGHT ADDED DUE TO INCREASE IN CABIN SIZE = 300 LBS.

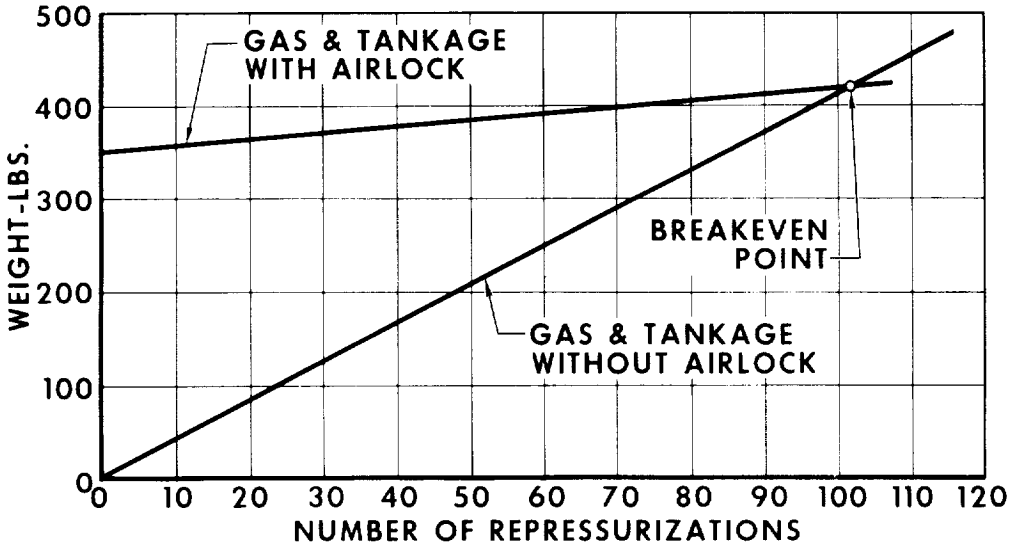


FIGURE 4–2

4.1 (Continued)

zations is nine, resulting in a 300 pound weight penalty with the airlock.

Three criteria are used in locating equipment within the command module. In order of priority these are: a) obtain the maximum center of gravity offset with respect to the geometric center line, b) locate components subject to inflight repair and calibration accessible to the crewmen, and c) provide accessibility for pad maintenance such that adjacent systems or components need not be disturbed. These considerations yield the arrangement shown in Figure 4-3.

To verify that this equipment installation is realistic, the resulting packing density, compared to similar manned spacecraft, is shown on Figure 4-4.

The selected arrangement incorporated within the Project Apollo aerodynamic shape results in a maximum lift-to-drag ratio during re-entry of .388 instead of the desired .50. A parametric comparison of various possible modifications was made and the results are shown in Figure 4-5 . These include the addition of ballast, changing the heat shield spherical radius-to-diameter ratio, increasing the command module size, revising the crew location, and combinations of all four. Revision of the heat shield radius ratio is shown to give the lightest configuration.

Reference 4-1 requests the variation of command module weight with size be determined between the limits of the selected configuration and the Project Apollo spacecraft. Figure 4-6 shows this variation.

As a correlation point, the comparable weight from Reference 4-2 is also plotted. The corrections from a three to a two man craft which were used to demonstrate correlation are listed in Table 4-1.

COMMAND MODULE INTERIOR ARRANGEMENT

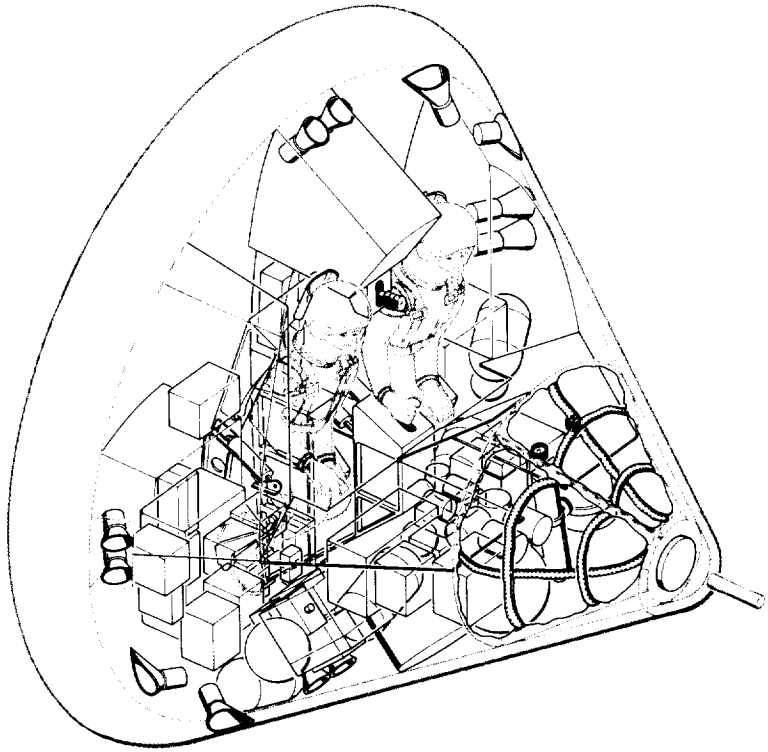


FIGURE 4-3

COMMAND MODULE EQUIPMENT

PACKING DENSITY

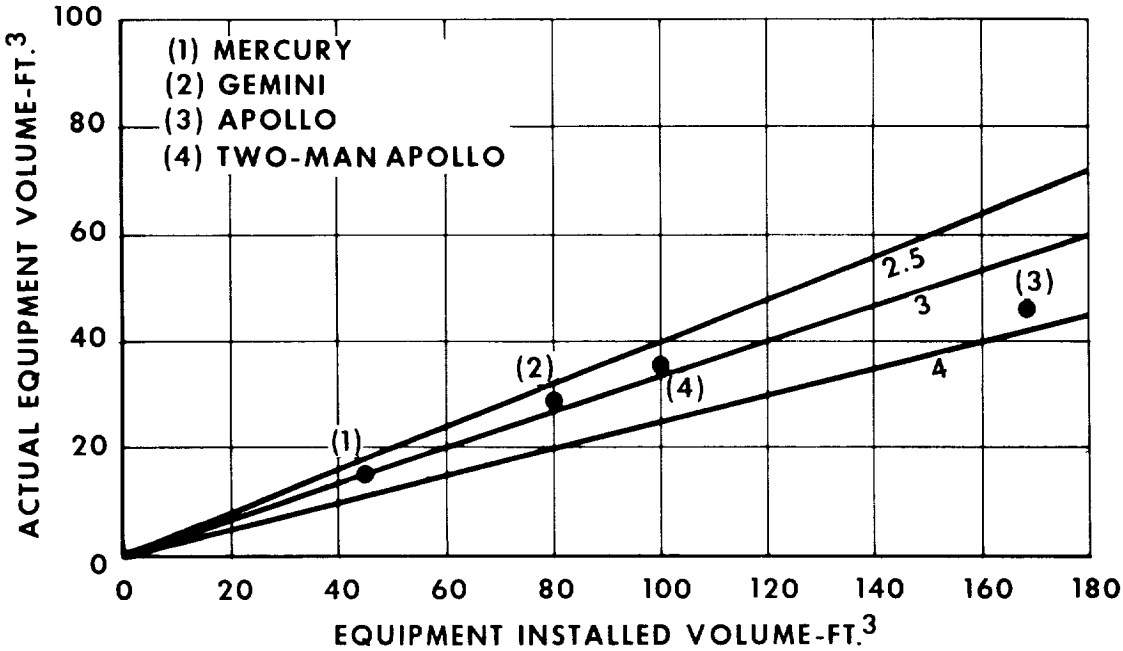


FIGURE 4-4

C.G. OFFSET CONSIDERATIONS

33° CONE ANGLE
0.5 L/D
.32 Z/D

--------- BALLAST REQUIRED - CREW RELATIONSHIP TO WINDOW MAINTAINED

——— BALLAST REQUIRED - CREW RELATIONSHIP TO SIDE OPPOSITE WINDOW MAINTAINED

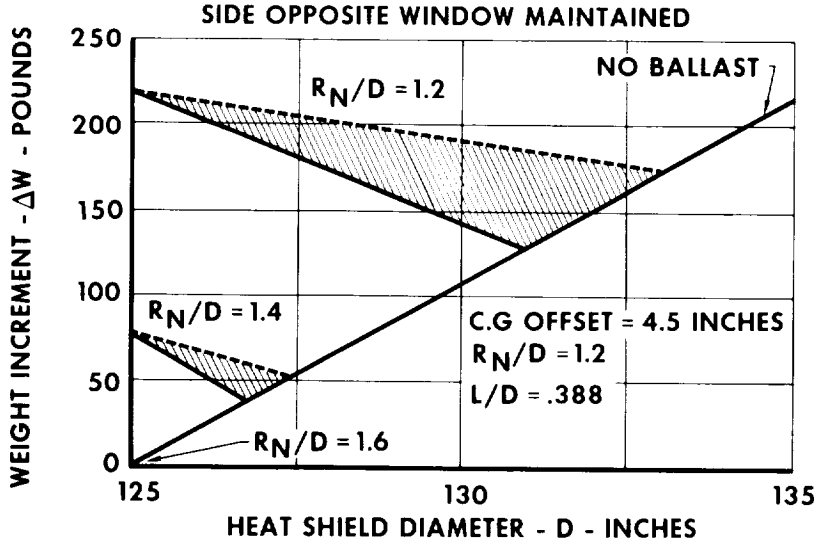


FIGURE 4-5

WEIGHT VS. VOLUME

(33° CONE ANGLE)

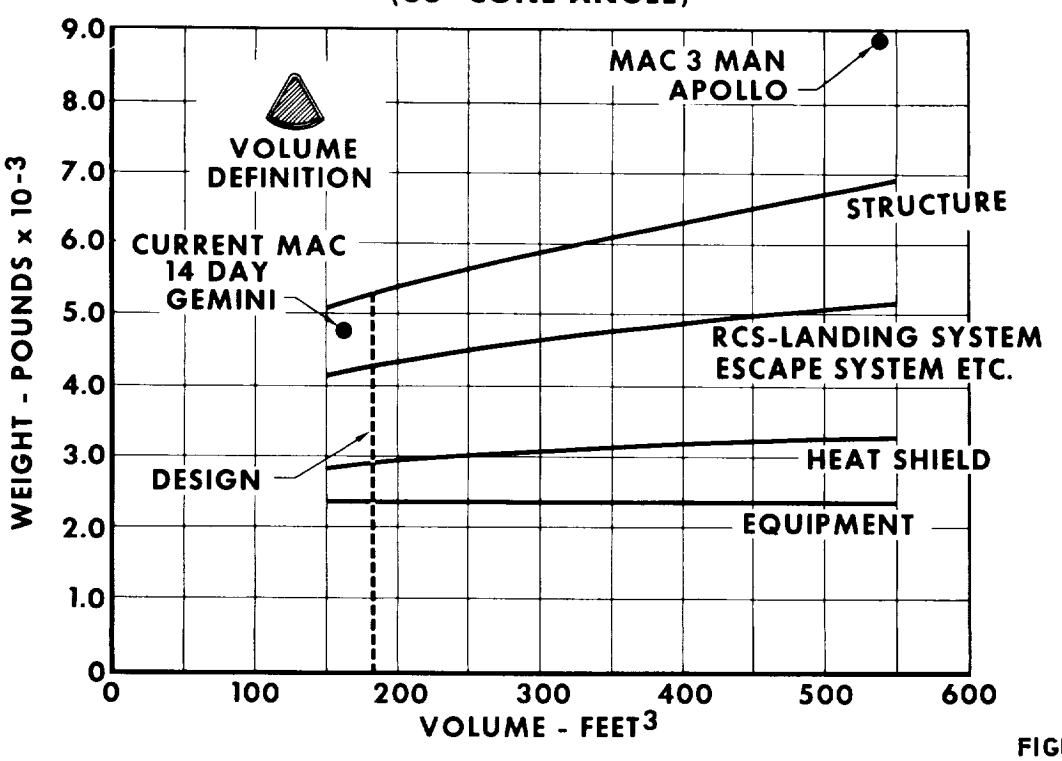


FIGURE 4-6

TABLE 4-1

WEIGHT CONSIDERATIONS
3 MAN TO 2 MAN APOLLO

	ΔW LBS.	WEIGHT LBS.
COMMAND MODULE-3 MAN APOLLO MAC PROPOSED		8889
MAINTAIN VEHICLE SIZE		
SIMPLER HATCH-WINDOW SYSTEM	−50	
REMOVE AIRLOCK	−82	
REDUCTION WITH LOWER LANDING WEIGHT	−456	
REDUCED T.V. & CAMERA COVERAGE	−100	
REDUCED ELECTRICAL POWER & SYSTEMS	−219	
REDUCED C-BAND POWER	−40	
REDUCED - FOOD, WATER & ECS REQUIREMENTS	−242	
REMOVE 1 CREWMAN, GEAR & SEAT	−332	
REMOVE RADIATION PROTECTION	−356	
REDUCED SCIENTIFIC GEAR	−165	
MISCELLANEOUS	−48	
COMMAND MODULE-2 MAN APOLLO (13' DIA.)		6799

4.2 Spacecraft Staging

4.2.1 General - A study was conducted to define the vehicle staging arrangement considering (1) command module weight, (2) utilization of present state-of-the-art propellants with consideration for future growth potential, and (3) module functional operation. The basic assumptions used in the study are given in Figure 4-7. These assumptions were adhered in order that direct comparisons could be made between any of the vehicle configurations examined.

STAGING STUDY ASSUMPTIONS

- ENGINES ARE PRESSURE FED
- MIDCOURSE CORRECTIONS MADE WITH N_2O_4/MMH
- PROPELLANTS STORED IN SPHERES
- O_2/H_2 AND F_2/H_2 SYSTEMS BASED ON 70 PSIA CHAMBER PRESSURE AND 105 PSIA TANK PRESSURE
- OF_2/MMH AND N_2O_4/MMH SYSTEMS BASED ON 100 PSIA CHAMBER PRESSURE AND 140 PSIA TANK PRESSURE
- ABLATIVE COOLED THRUST CHAMBERS WITH RADIATION SKIRTS FROM AN EXPANSION RATIO OF 15

FIGURE 4-7

The selection of chamber pressure for stages using earth storable propellants is discussed in Section 3.9.1.6. The selection of chamber pressure for the stages using cryogenic propellants (F_2/H_2 and O_2/H_2) is made on a similar basis. In this case, however, spherical aluminum tanks are used (in place of titanium for earth storable propellants) with insulation for 90 hours space storage. The results from the study using O_2/H_2 are presented in Figure 4-8. These results are also considered applicable for stages using F_2/H_2. For the pressure range examined, down to 50 psia, no minimum is shown. This is due to the strong influence of tank weight with

4.2.1 (Continued)

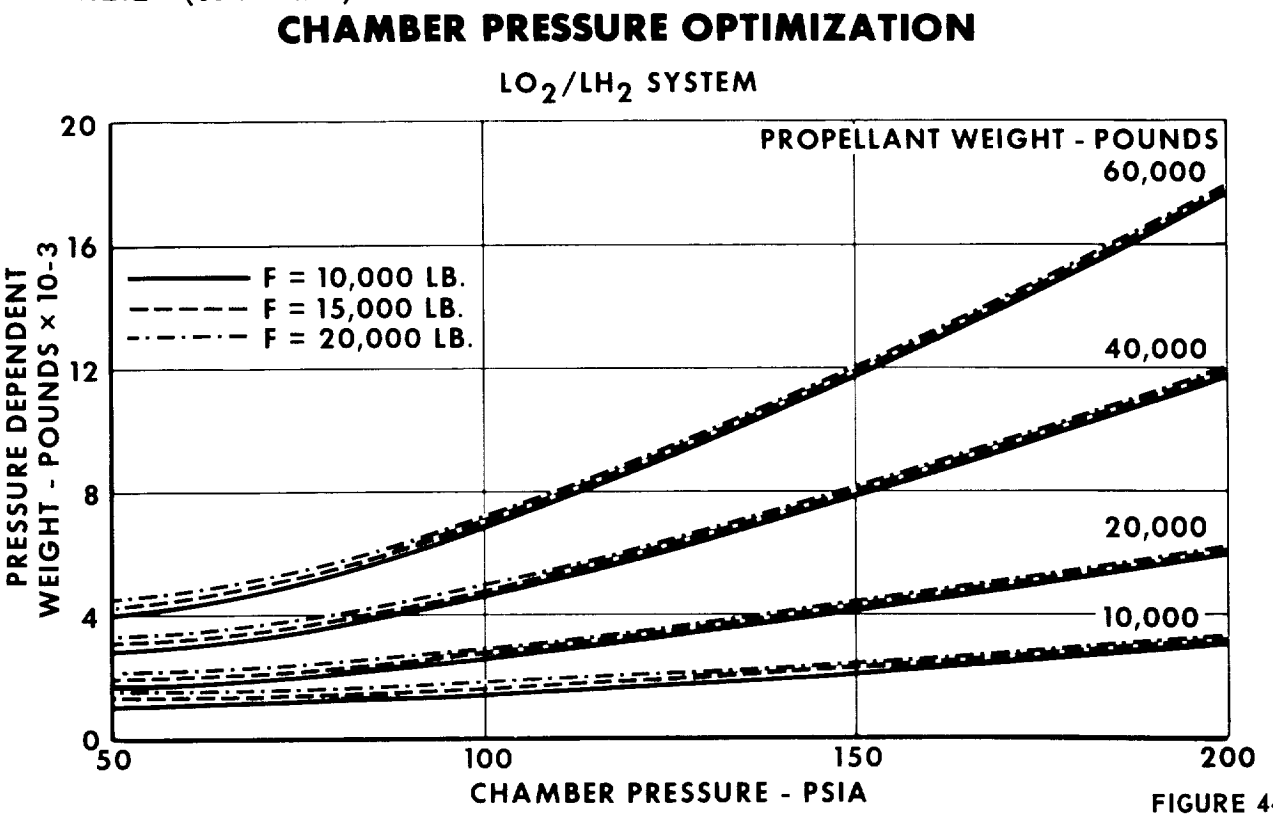


FIGURE 4-8

cryogenic propellants. A somewhat arbitrary choice of 70 psia was made at this point and used throughout the parametric studies.

The propellant characteristics for the staging study are given in Figure 4-9. A comparison of propellant volumes required to perform a given incremental velocity is also given for purposes of comparison.

The incremental velocities used for the staging study are specified in Reference 4-3 and are repeated in Table 4-2. The NASA 10% contingency on ΔV is apportioned to the individual function to be performed.

4.2.2 Staging Arrangements - Five of the more attractive staging arrangements examined, were selected for comparison and are shown in Figure 4-10. Cryogenic propellants are used only for the early phase of the mission up to the terminal phase of lunar landing in four of the configurations. In one arrangement cryogenic propellants are also used for lunar landing. The individual stage weights for each

PROPELLANT CHARACTERISTICS

PROPELLANT COMBINATION / PARAMETER	F_2-H_2	O_2-H_2	OF_2-MMH	N_2O_4-MMH
OXIDIZER/FUEL MIXTURE RATIO	13	4.5	2.5	2.2
PROPELLANT BULK SPECIFIC GRAVITY	.621	.309	1.26	1.196
FLAME TEMPERATURE, °F	≈ 8000	5380	6670	5293
THEORETICAL VACUUM SPECIFIC IMPULSE (ε=40:1, SHIFTING EQUILIBRIUM) - I_{SP}, - SEC.	467	455	406	339
THEORETICAL DENSITY IMPULSE - I_D - LB-SEC/FT.^{3}	18,150	8,772	31,919	25,300
DELIVERED VACUUM SPECIFIC IMPULSE (ε=40:1, STEADY STATE OPERATION) - I_{SP}, SEC.	449	430	387	320
OXIDIZER FREEZING POINT, °F	-365	-362	-371	+12
FUEL FREEZING POINT, °F	-435	-435	-63	-63
ALLOWABLE OXIDIZER TEMPERATURE AT TANK DESIGN PRESSURE	-263°F @ 105 PSIA	-255°F @ 105 PSIA	-168°F @ 140 PSIA	+172°F @ 140 PSIA
ALLOWABLE FUEL TEMPERATURE AT TANK DESIGN PRESSURE	-407°F @ 105 PSIA	-407°F @ 105 PSIA	+350°F @ 140 PSIA	+350°F @ 140 PSIA
PROPELLANT VOLUME REQUIRED FOR ΔV=10,575, FPS AND INITIAL WEIGHT=88,867 LBS., FT.^{3}	1467	3079	767	894

FIGURE 4-9

TABLE 4-2

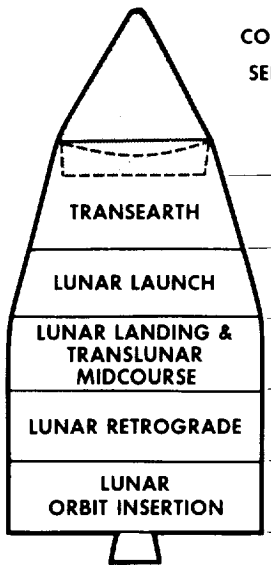
ΔV Requirements

Function	ΔV ft/sec
Translunar Midcourse Corrections	330
Retrograde into 80 N.M. Lunar Orbit	3443
6° Simultaneous Plane Change	110
Hohmann to 50,000 Ft.	135
Descent to 1,000 Ft. Altitude	6557
Hover and Translation	770
Launch to 50,000 Ft. Circular Orbit	6474
Transearth Injection	3951
Transearth Midcourse Corrections	330

4.2.2 (Continued)

PROPULSION SYSTEM STAGING

TOTAL ΔV = 22,100 FPS



WT.-LBS.	I	II	III	IV	V
COMMAND MODULE	5937	5907	5735	5746	6265
SERVICE EQUIPMENT	2118	2118	2118	2118	2118
EFF. TOWER	165	165	165	165	165
TOTAL	8220	8190	8018	8029	8548
TRANSEARTH	19,854	5,172	19,567	28,594	20,489
LUNAR LAUNCH		13,871			
LUNAR LANDING & TRANSLUNAR MIDCOURSE	8,914	8,928	8,576		
LUNAR RETROGRADE	27,894	52,347	52,347	51,882	35,645
LUNAR ORBIT INSERTION	23,271				23,471

N_2O_4/MMH O_2/H_2

FIGURE 4-10

of the five configurations is shown in the respective block of the array.

Configuration III was selected, for more detailed examination, as a compromise between a number of factors. The more important of these are:

A. A minimum number of stages is desirable from a mechanization standpoint.

B. It is not desirable to land and launch with the same module because of possible stage damage during the landing operation.

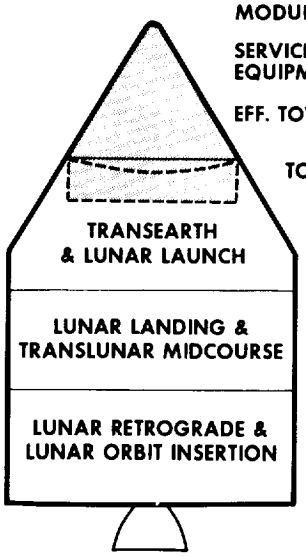
C. While cryogenics are favored for the large translunar ΔV's, they are considered less desirable than earth storables for the final landing phase because of the ignition system requirement.

4.2.3 Effect of Propellant Combinations on Configuration III - The Configuration III vehicle is used to determine the effects of various propellant combinations on the allowable command module weight. The basic ground rules set forth in Figure 4-7 are applied to this phase of the study. Figure 4-11 summarizes the effect of

4.2.3 (Continued)

CONFIGURATION III
EFFECT OF PROPELLANT COMBINATIONS

TOTAL ΔV 22,100 FPS



WT.-LBS.	III	III-a	III-b	III-c	III-d	III-e	III-f
COMMAND MODULE	5735	4285	5967	7825	7882	7284	6436
SERVICE EQUIPMENT	2118	2118	2118	2118	2118	2118	2118
EFF. TOWER	165	165	165	165	165	165	165
TOTAL	8018	6568	8250	10,108	10,165	9567	8719
TRANSEARTH & LUNAR LAUNCH	N_2O_4/MMH	N_2O_4/MMH	N_2O_4/MMH	N_2O_4/MMH OF_2/MMH	N_2O_4/MMH OF_2/MMH	N_2O_4/MMH O_2/H_2	N_2O_4/MMH
LUNAR LANDING & TRANSLUNAR MIDCOURSE	N_2O_4/MMH	N_2O_4/MMH	N_2O_4/MMH	OF_2/MMH N_2O_4/MMH	OF_2/MMH N_2O_4/MMH	O_2/H_2 N_2O_4/MMH	N_2O_4/MMH
LUNAR RETROGRADE & LUNAR ORBIT INSERTION	O_2/H_2	N_2O_4/MMH	OF_2/MMH	O_2/H_2	OF_2/MMH	O_2/H_2	F_2/H_2

FIGURE 4-11

propellant combinations on the vehicle payload. From this analysis the relative advantages of the oxygen difloride/monomethylhydrazine (OF_2/MMH) and fluorine/hydrogen (F_2/H_2) propellant systems can be seen.

The OF_2/MMH combination is a logical growth potential substitution for a propulsion system designed originally for N_2O_4/MMH because (1) the tanks sized for N_2O_4/MMH at a mixture ratio of 2.2 are in the same proportion as required for OF_2/MMH at a mixture ratio of 2.5, and (2) materials compatibility problems of the two propellant combinations are similar in nature. Where differences exist, the N_2O_4 system can be designed to the OF_2 criteria, thus allowing the substitution to be made with minimum change at a later time. The effect of substituting OF_2/MMH in the service module of Configuration III is shown on Figure 4-12 with the second order effects of optimizing tank and structural weights.

4.2.4 Propulsion Module Mass Fraction Comparison - The weight of inert

4.2.4 (Continued)

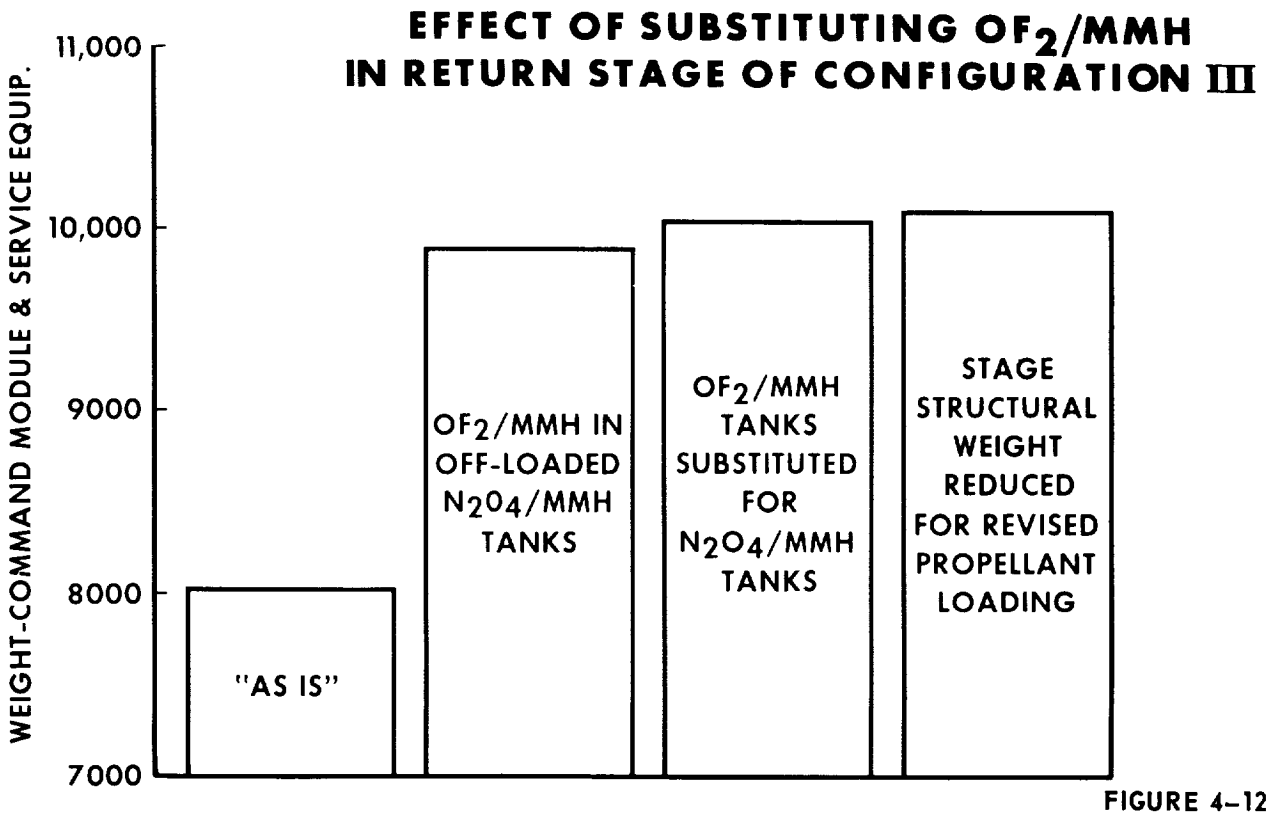


FIGURE 4-12

hardware items in the propulsion modules (i.e., engine, tanks, support structure, lines, valves, pressurization, gimbal actuators) is defined for each configuration by computation using a design layout and through utilization of information on similar hardware items from such programs as Gemini, Titan, Centaur, Agena, and Mercury. The propulsion mass fraction of the stage was then computed. Similar propellant mass fractions were obtained for a number of existing launch vehicles, both pump fed and pressure fed. These data were then compared in Figure 4-13 as a function of size, defined by total propellant weight. This comparison was made to substantiate that the computed propellant mass fractions for the stages being examined were not in the over optimistic region when compared with todays hardware. The terminal landing module mass fraction for the various arrangements is not shown since it is quite inefficient as a propulsion stage, a mass fraction in the range of .6 to .65, due to the low propellant requirements.

PROPULSION MODULE MASS FRACTION COMPARISON

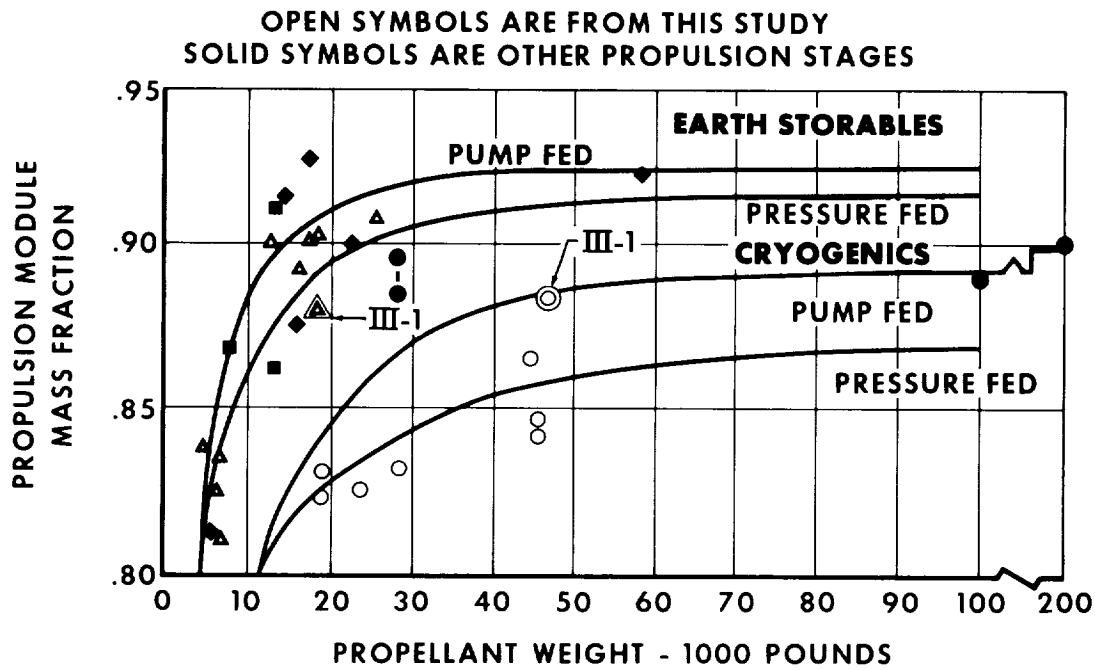


FIGURE 4-13

4.2.5 Terminal Staging Altitude Determination - The altitude at which main retrograde module staging is accomplished and the terminal landing module activated is chosen as the result of a study considering the main retrograde trajectory, the abort condition at staging with the service module engine, and guidance and control system dispersions. The desired minimum altitude above the lunar surface during an abort after main retrograde staging is arbitrarily chosen as 1000 feet. The results of this study are summarized in Figure 4-14. The terminal staging altitude for subsequent studies is considered to be 6000 feet.

4.2.6 Terminal Landing Capability - The available capability of the terminal landing module with a propellant supply based on a final descent from 6000 feet with an initial vertical velocity of zero and a landing engine sized as discussed in Paragraph 3.9.2.6 with a throttling ratio of 3:1 is shown in Figure 4-15. A continuous thrusting vertical trajectory program is followed; the engine is

TERMINAL STAGING ALTITUDE DETERMINATION

ASSUMPTIONS:

1. VERTICAL VELOCITY AT STAGING POINT = 0 FT/SEC
2. MAIN RETRO ENGINE THROTTLED 10/1 : $T/W_{O_{max}} = .4$
3. LUNAR LAUNCH ENGINE $T/W_O = .4$
4. ABORT CONDITION CAN BE ANALYZED AND CORRECTIVE ACTION TAKEN IN 20 SECONDS

SUMMARY:

1. DESIRED MINIMUM ALTITUDE DURING ABORT		1000
2. Δh TO ARREST VEHICLE VELOCITY AFTER 20 SECOND TIME DELAY		1800
3. GUIDANCE & CONTROL SYSTEM DISPERSIONS		
A. ACCELEROMETER ERROR (.1% & .03 FPS^2)	1641	
B. PLATFORM ALIGNMENT ERROR (.1 DEG.)	1742	
C. INITIAL POSITION AND VELOCITY ERRORS (2000 FT. & 2 FPS)	2370	
D. LUNAR GRAVITY UNCERTAINTY (.1%)	354	
RSS		3440
TERMINAL STAGING ALTITUDE		6240

FIGURE 4-14

TERMINAL LANDING WINDOW PROPELLANT QUANTITY FOR 6000 FT. ALT., $V_1 = 0$

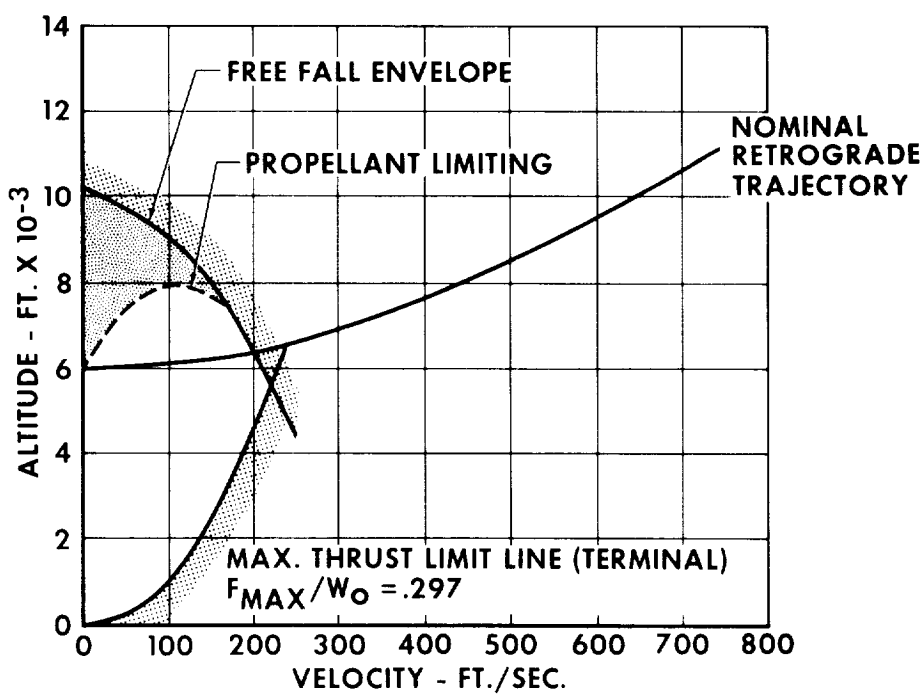


FIGURE 4-15

4.2.6 (Continued)

operated at minimum thrust level until the altitude and vertical velocity that requires application of near maximum thrust to arrest the vehicle velocity prior to touchdown on the lunar surface is reached. The propellant supply required for final descent from 6000 feet and V = 0 will also allow a final descent to be made from 8000 feet at an initial velocity of 100 feet/sec. The propellant requirements for a constant thrusting program are a strong function of the gravitational losses incurred during the final descent, these being a maximum at the V = 0 initial condition.

The nominal retrograde trajectory shown in Figure 4-15 is a plot of total vehicle velocity versus altitude, while the propellant limiting line, maximum thrust capability line, and free fall envelope are based on vertical velocities only. The nominal retrograde trajectory can be followed closely from a range of initial conditions by utilizing the closed loop control and engine throttling technique discussed in Section 4.3.2. Should the main retrograde function be completed at slightly higher altitudes than the nominal, free fall can be employed to bring the vehicle to a point within the propulsion system capability envelope.

4.3 Trajectory Analysis - A detailed description of the flight mechanics of the direct flight lunar landing mission is not attempted in this section since these studies are available in current literature. This discussion is directed toward analyzing those phases of the mission that are affected by the spacecraft design and verifying that the systems, as designed, are adequate from a flight mechanics standpoint. Emphasis is placed on maneuvers in the vicinity of the moon and on the abort capability of the service module. Four general areas are considered; initial retrograde from the translunar trajectory to terminal staging, landing, launch, and abort.

4.3.1 Lunar Retrograde to Terminal Staging Altitude - The basic retrograde maneuver consists of three phases; establish near circular orbit about the moon, establish low altitude pericynthion near the landing site, descend from pericynthion to terminal staging altitude. The first two maneuvers are accomplished by applying thrust in a tangential or near tangential direction until the desired velocity and flight path angle are obtained.

The descent to staging altitude can be accomplished in a variety of ways. One that appears promising combines continuous thrust let-down from a low pericynthion with range and altitude controlled in a closed-loop scheme which utilizes vehicle attitude and variable thrust as control parameters. A description of this control scheme is contained in Section 4.3.2.

Staging altitude is determined through an analysis of the altitude dispersions resulting from initial position and guidance system uncertainties together with allowances for pilot reaction time in recognizing the malfunction and initiating abort procedures. Figure 4-14 is a summary of the assumptions used and results of the staging altitude analysis.

4.3.2 Trajectory Control During Main Descent - Practical considerations make it essential to provide the capability for reaching the desired staging

4.3.2 (Continued)

conditions, both velocity and position, from a wide range of initial conditions. It is also essential to have a simple means of correcting the effects of off-nominal thrust-to-weight ratio and control errors. These requirements dictate a variable thrust main retrograde engine. With this capability the equations of motion can be written so that they have simple solutions of the desired form. The thrust commands are then the difference between the desired accelerations and the natural accelerations. This procedure yields the following equations for in-plane control of the main retrograde.

$$T_r = (G/r^2) - r\dot{\theta}^2 - (3/2)(h-h_N)\left(\frac{\dot{\theta}}{\theta_N-\theta}\right)^2 - (2\dot{h} + \dot{h}_N)\left(\frac{\dot{\theta}}{\theta_N-\theta}\right.$$

$$T_\theta = 2\dot{h}\dot{\theta} - \frac{1}{2}\, r\dot{\theta}^2/(\theta_N-\theta)$$

where T_r = radial component of thrust per unit mass

T_θ = horizontal component of thrust per unit mass

G = gravitational constant times mass of moon

r = radial distance from center of moon

θ = in-plane angular travel

h = altitude

Subscript N denotes desired end conditions to be achieved when horizontal velocity goes to zero.

Out-of-plane control can be achieved by commanding a lateral thrust proportional to the out-of-plane displacement of the target site.

With thrust applied according to the equations given above, the trajectory equations have simple, closed form solutions which show that the desired end conditions result, regardless of initial conditions. Control may begin at any convenient point, e.g., after an initial period of open-loop thrusting during which the range and velocity are decreased in some grossly correct fashion. Figure 4-16

4.3.2 (Continued)

shows that simply applying maximum thrust along the local horizontal for a significant portion of the trajectory (160 seconds, over 2200 fps velocity reduction) actually reduces the propellant requirement. This is because the closed-loop control generally commands the engine to throttle back as the site is approached, hence, the average thrust-to-weight ratio is increased by the initial period of open-loop maximum thrust. During this 160 second period the range (distance from initiation of thrust to target point) decreases by about 116 nautical miles, but the altitude decreases less than 14,000 feet. Hence, starting altitude is not critical from a geometric standpoint. Figure 4-16 also shows that it is not critical for propellant economy. Range and altitude errors built up during the open-loop phase are automatically corrected out in the closed-loop phase. Propellant requirements to allow for error correction are not excessive, e.g., the increased characteristic

EFFECT OF PERICYNTHION ALTITUDE ON CHARACTERISTIC VELOCITY REQUIRED FOR MAIN DESCENT MANEUVER

INITIAL VELOCITY: PER TRANSFER FROM 80 N.M.
FINAL VELOCITY: ZERO AT 6000 FT. ALTITUDE
MAX. $(T/W_E)_o = .4$

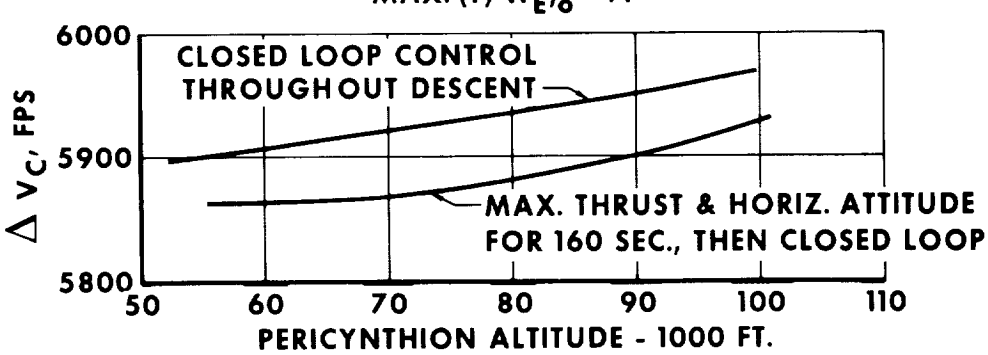


FIGURE 4-16

4.3.2 (Continued)

velocity required for a 10 nautical mile range adjustment, made after 160 seconds of open-loop maximum thrust, is only about 50 feet per second.

4.3.3 Staging Altitude to Lunar Surface - Results of studies related to this phase of the direct lunar landing mission are presented in Section 4.2.

4.3.4 Launch from Lunar Surface - The effect of initial thrust-to-weight ratio on the velocity losses during launch to lunar orbit is presented in Figure 4-17. The trajectory assumes a vertical rise to an altitude of 1000 feet and subsequent optimum in-plane maneuvering to orbit injection. Figure 4-17 indicates that the launch design thrust-to-weight ratio of 0.4 is near optimum. Figure 4-18 presents the effect of lunar orbit injection errors on the pericynthion of the resulting orbit. These data indicate that an overspeed of approximately 35 feet per second at injection is required to minimize the effect of injection errors.

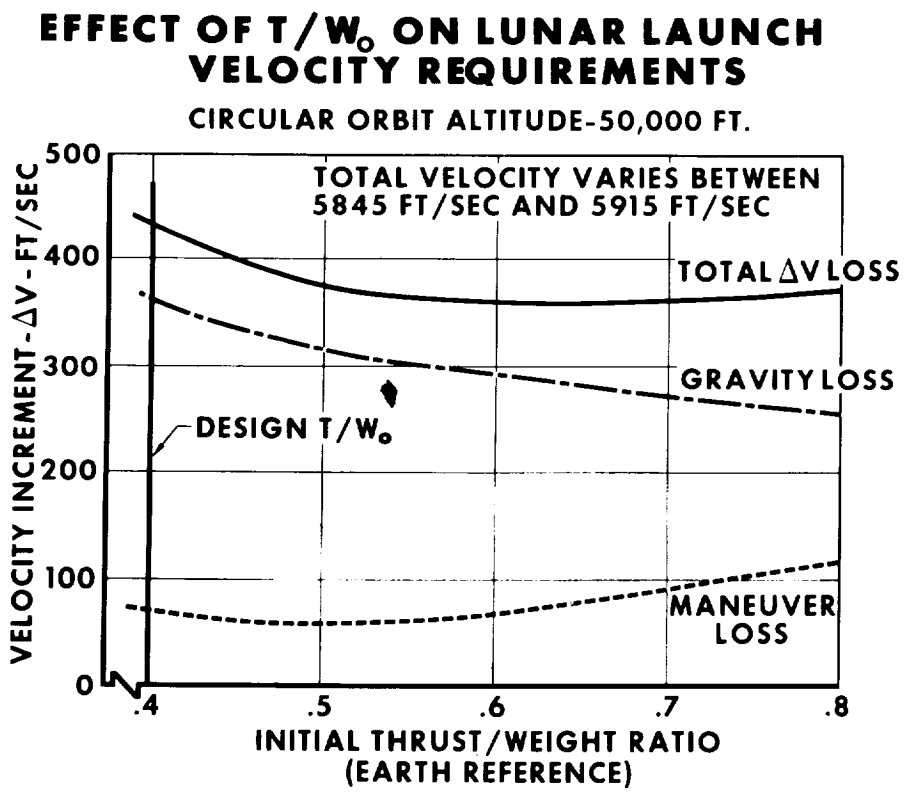


FIGURE 4-17

LUNAR ORBIT INJECTION SENSITIVITY

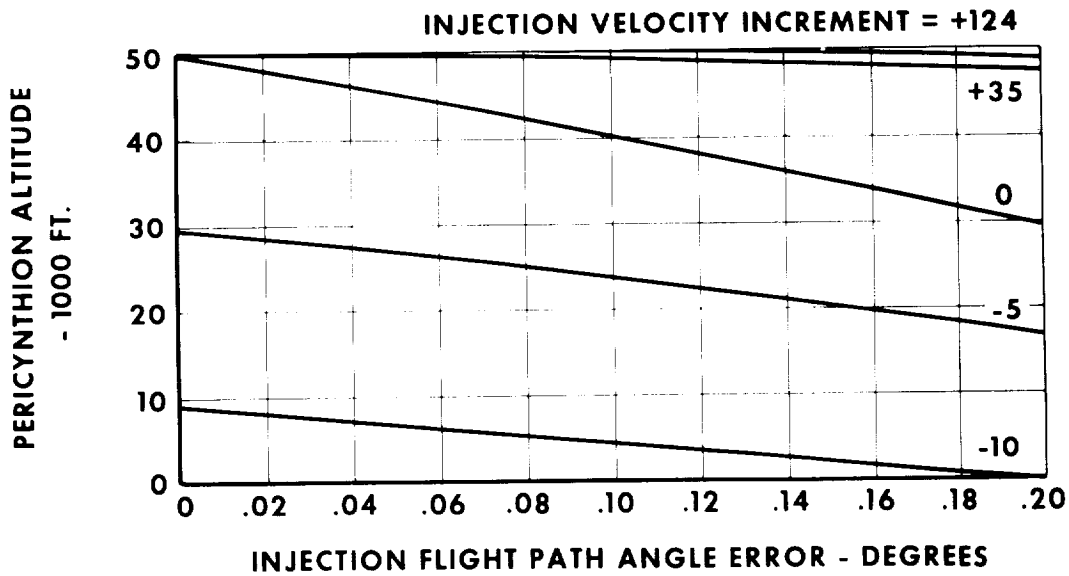


FIGURE 4-18

Apocynthion resulting from this overspeed is approximately 200,000 feet where an additional impulse of 35 feet per second is required to circularize the lunar parking orbit.

4.3.5 Abort - The abort considerations from countdown until injection into the earth parking orbit fall into two general categories - abort with the LEPS and abort after LEPS jettison. The criteria and design of the LEPS are discussed in Section 3.11.

For separation of the command module from the launch vehicle after LEPS jettison three sources of impulse are available; service module, terminal landing module and retrograde module. Of the three, the service module has the highest initial thrust-to-weight ratio, sufficient ideal velocity available for all abort conditions (in excess of 10,500 feet per second) and means of positive separation from the command module prior to re-entry, therefore, it is selected as the abort propulsion module.

4.3.5 (Continued)

At all times after LEPS jettison the launch vehicle thrust-to-weight ratio exceeds that of the service module thus requiring shutdown of the launch engines before separation can be effected. Studies indicate that service module thrusting times up to ten seconds have little effect on the apogee altitudes of the spacecraft after abort and that the probability of recontact with the launch vehicle is lessened by long thrusting times. Ten seconds is the prescribed thrusting time for separation.

Figure 4-19 shows the apogee conditions resulting from aborts from a nominal Saturn C-5 launch to parking orbit. Also shown on the figure are the 10 g and 20 g recovery ceilings. From the nominal trajectory it can be seen that no difficulty is experienced in keeping the re-entry load factors following abort below the emergency limit of 20 g's.

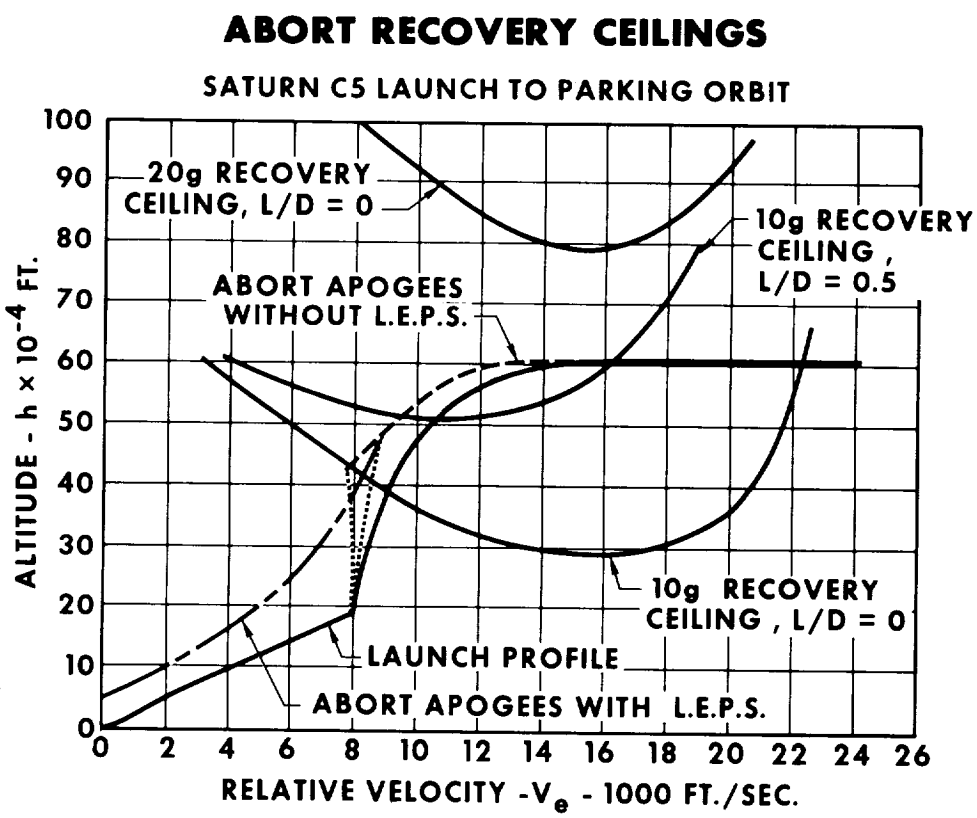


FIGURE 4-19

4.3.5 (Continued)

Another factor considered in the abort analysis is the touchdown location. For the nominal trajectory of Figure 4-19 and typical lunar mission launch azimuths from Cape Canaveral, an abort initiated at a relative velocity greater than 23,000 feet per second will land on the African continent. Since the vehicle is near orbit velocity under these conditions, the service module can easily be used to continue the injection into the parking orbit and retrograde and recovery can be effected after one orbit around the earth. For launches to the northeast from Cape Canaveral the recovery area after one orbit can be in the Atlantic Missile Range but for launches to the southeast the recovery area is in the Pacific Ocean.

Should an abort be necessary during the acceleration to translunar injection, a retrograde maneuver into a higher altitude earth orbit can be performed. Figure 4-20 indicates the service module is capable of returning the command module to earth orbit for aborts at inertial velocities less than 33,200 feet per second with reserves for retrograde. Use of the retrograde module in addition to the service module does not appreciably extend the orbit return limit since its low thrust-to-weight ratio becomes ineffective at near injection velocities.

Providing suitable recovery sites for quick-return aborts becomes difficult for velocities beyond which circular orbit can no longer be re-established. Trajectory studies show that azimuth or plane change maneuvers at the time of abort are ineffective for landing point control. An effective cross-path displacement due to earth rotation can be achieved by controlling the time of re-entry. This is done by proper selection of the initial retrograde magnitude followed by a small velocity change at apogee to establish permissable re-entry conditions. This technique is chosen for aborts at velocities greater than the orbit return limit in order to control the command module to a preselected recovery area. The velocity capability of the service module is adequate to perform these maneuvers.

IDEAL VELOCITY REQUIREMENT ABORT FROM TRANSLUNAR INJECTION

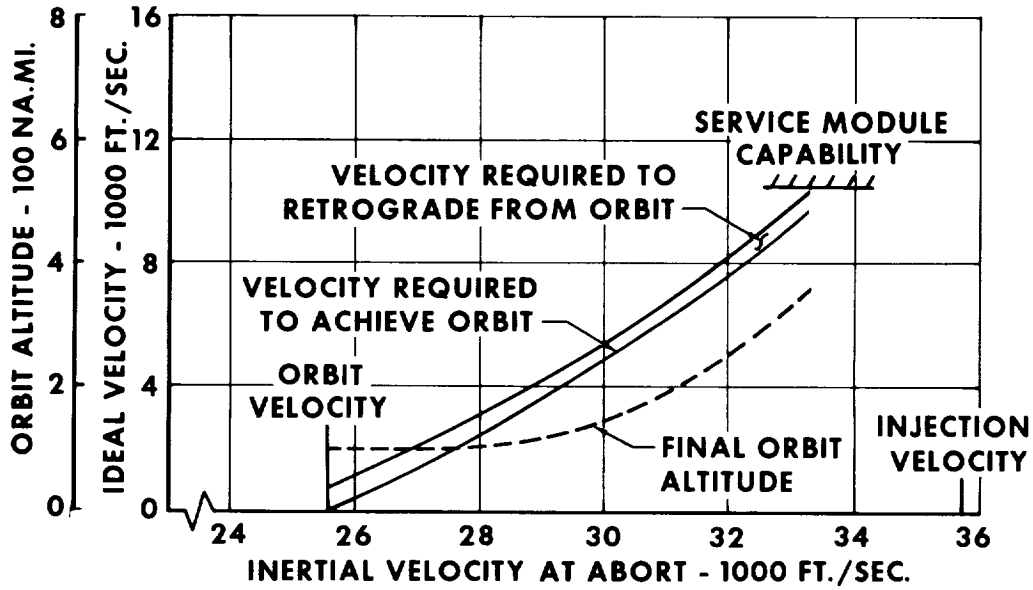


FIGURE 4-20

Aborts after translunar injection are accomplished in the same manner as those immediately prior to injection, that is, the time to re-entry is controlled by the magnitude of an initial retrograde impulse in order to achieve re-entry location control. Beyond approximately 40 hours after injection no time saving is realized by the abort maneuver over completion of a circumlunar mission.

The abort during retrograde into lunar orbit is accomplished by completion of the retrograde maneuver with the service module. Subsequent transearth injection is accomplished as in the normal mission.

Aborts during lunar landing are executed by maneuvering back into the lunar parking orbit with the service module. Figure 4-21 illustrates simulated abort profiles from a typical landing. The minimum altitude during the recovery maneuver is 2500 feet. Figure 4-22 presents the ideal velocity required for maneuver back to orbit, and indicates that sufficient ideal velocity is available to complete the transearth injection.

ABORTED LUNAR LANDING PROFILES

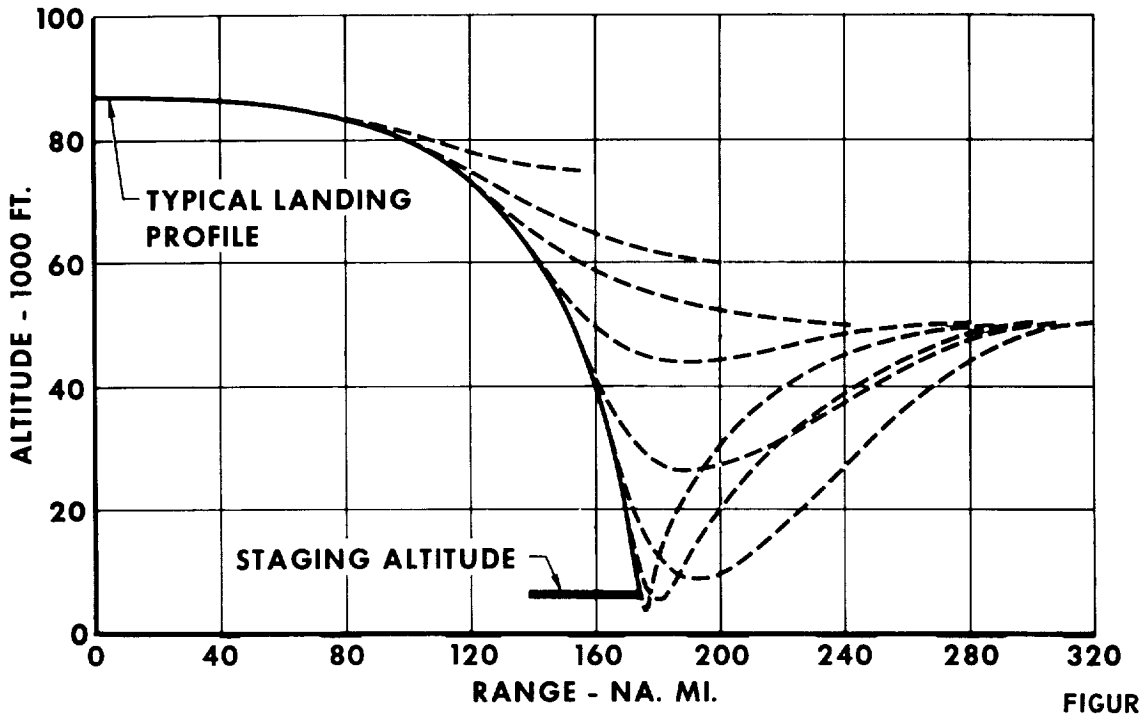


FIGURE 4-21

IDEAL VELOCITY REQUIREMENT ABORT FROM LUNAR LANDING

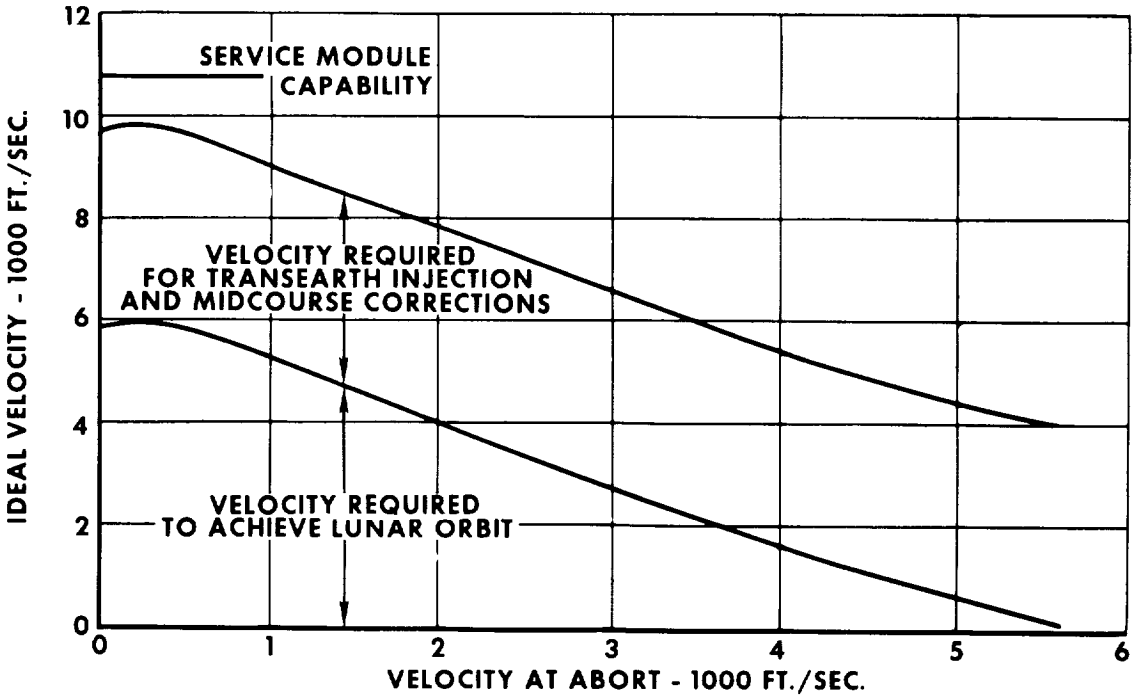


FIGURE 4-22

4.3.5 (Continued)

Figures 4-23 and 4-24 present the altitude increment and ideal velocity required to bring the vehicle to zero vertical velocity in the event of abort at terminal staging. This plot shows that sufficient time is available to recognize an abort requirement and initiate service module operation.

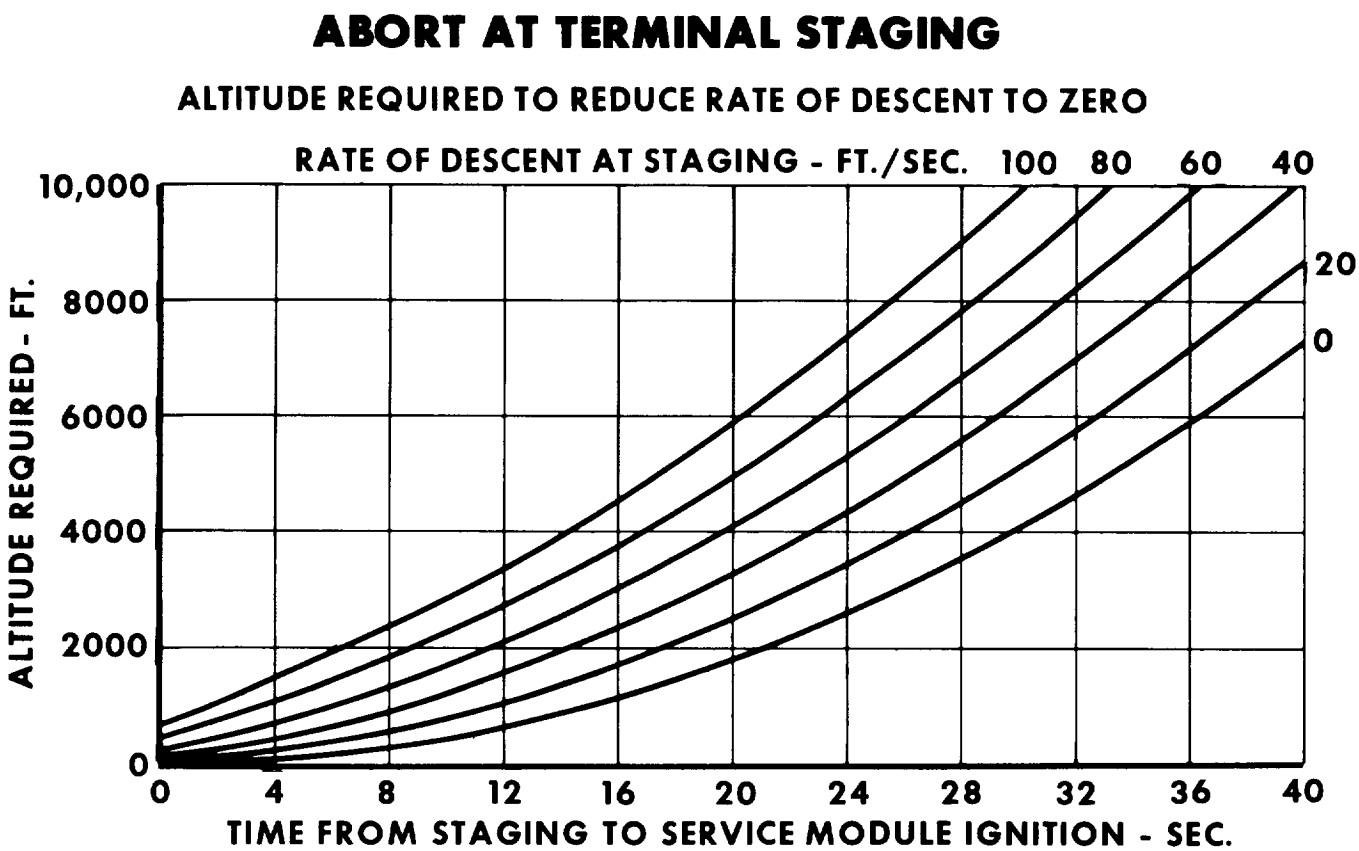


FIGURE 4-23

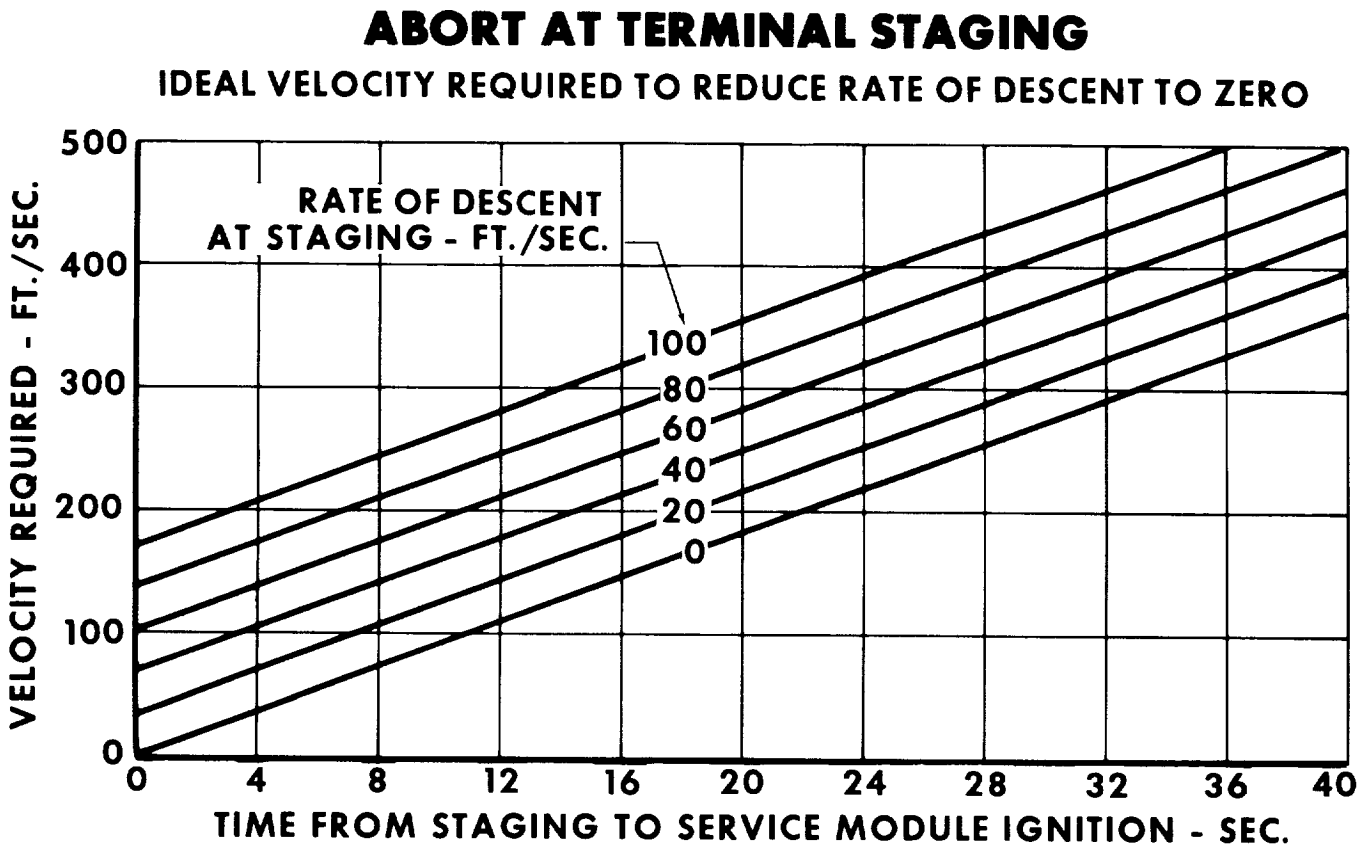

ABORT AT TERMINAL STAGING
IDEAL VELOCITY REQUIRED TO REDUCE RATE OF DESCENT TO ZERO
RATE OF DESCENT
AT STAGING - FT./SEC.
100
80
60
40
20
0
VELOCITY REQUIRED - FT./SEC.
500
400
300
200
100
0
0
4
8
12
16
20
24
28
32
36
40
TIME FROM STAGING TO SERVICE MODULE IGNITION - SEC.

FIGURE 4-24

4.4 Aerodynamics - The original configuration considered for the two-man command module was a scaled-down version of the Three-Man Apollo command module. Early in the study it was realized that, because of limitations on crew placement and equipment arrangement imposed by the reduced volume, the offset center-of-gravity (c.g.) required to trim to a lift-to-drag ratio (L/D) of 0.5 would be difficult to obtain. Studies were undertaken to determine what modifications could be made to the configuration, other than add ballast or simply increasing the size and thereby the weight, to achieve the desired L/D.

Of the possible changes to the basic Apollo shape (cone angle, heat shield radius of curvature, corner radius, and external flap surfaces) the modification that produces the maximum reduction in c.g. offset requirements for the minimum change to the command module is heat shield radius of curvature. Figure 4-25 shows the L/D versus c.g. offset for three values of heat shield radius to capsule diameter ratio, R_N/D. These curves were obtained from wind tunnel data for the R_N/D of 1.2 adjusted by modified Newtonian theory to the other values shown.

On the basis of these results, a design value for R_N/D of 1.6 is selected. The final weight and balance yields a c.g. offset of 4.5 inches. This with the associated longitudinal c.g. position of about .304D gives a trim L/D of 0.475.

Figures 4-26, 4-27 and 4-28 present aerodynamic coefficients for the command module in the re-entry and abort configurations respectively.

Figure 4-29 is a summary of the c.g. offset required to trim to L/D = 0.5 for $R_N/D = 1.6$. Variations are shown for corner radius ratio, R_c/D, and longitudinal c.g. position and indicate that further gains in L/D are available for minor capsule modification.

SENSITIVITY OF LIFT TO DRAG RATIO (L/D) TO C. G. OFFSET AND HEAT SHIELD FACE RADIUS (R_N/D)

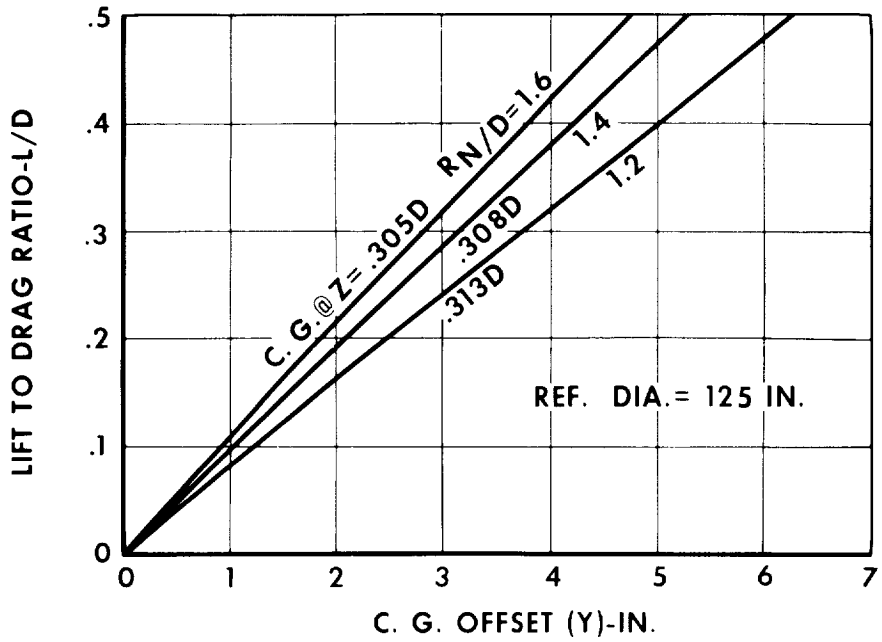


FIGURE 4-25

RE-ENTRY CONFIGURATION AERODYNAMIC CHARACTERISTICS VS. ANGLE OF ATTACK

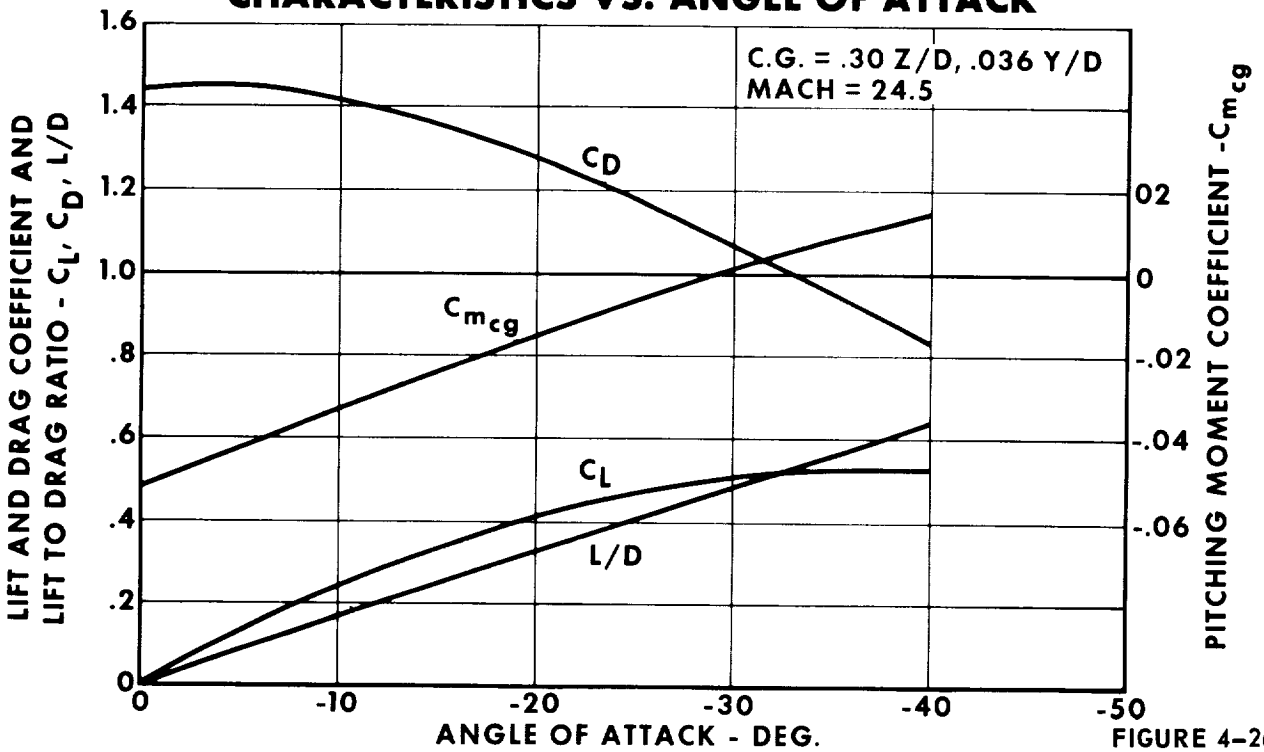


FIGURE 4-26

STATIC STABILITY IN ABORT CONFIGURATION AFTER ROCKET BURNOUT

CG @ .60 Z/D, .028 Y/D

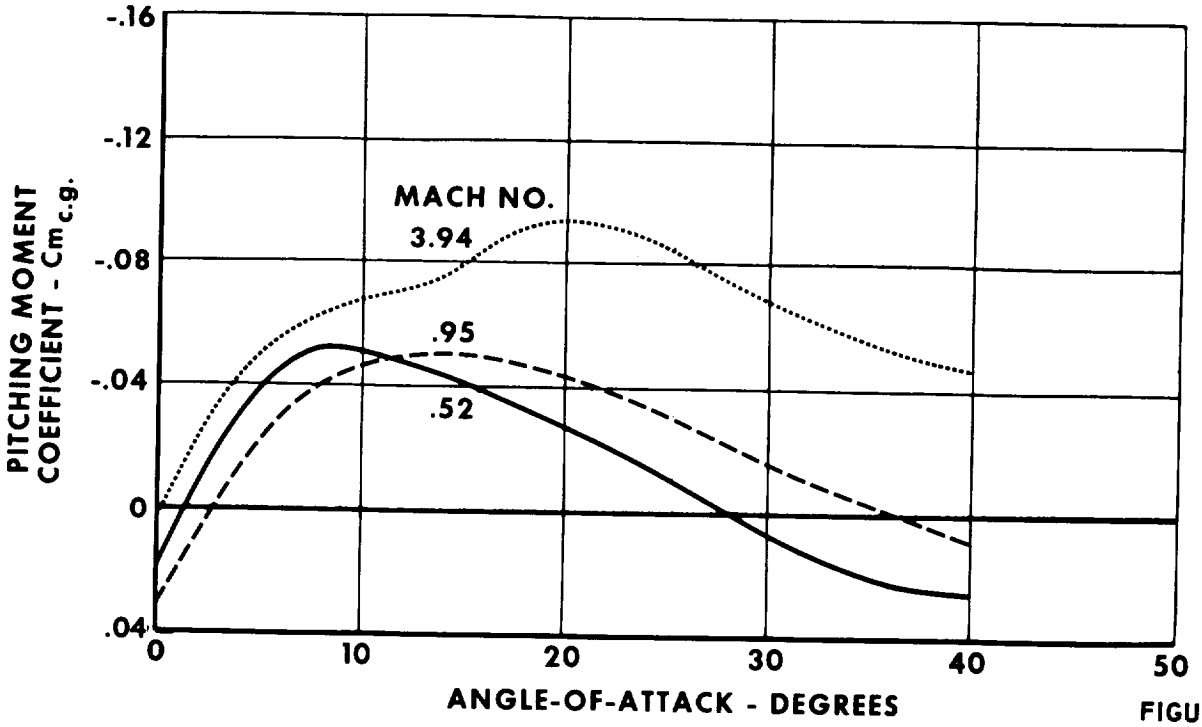


FIGURE 4-27

DRAG COEFFICIENT vs. ANGLE OF ATTACK FOR ABORT CONFIGURATION

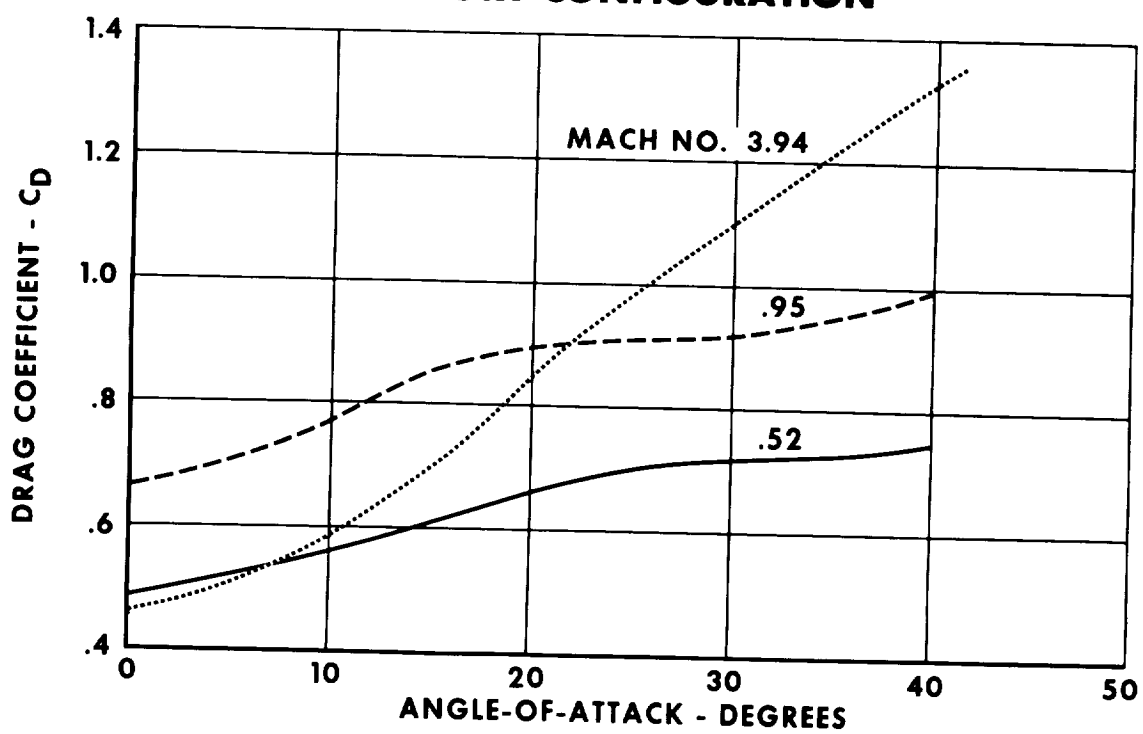


FIGURE 4-28

EFFECT OF CORNER RADIUS (R_C/D) ON OFFSET C.G. REQUIRED FOR L/D = 0.5

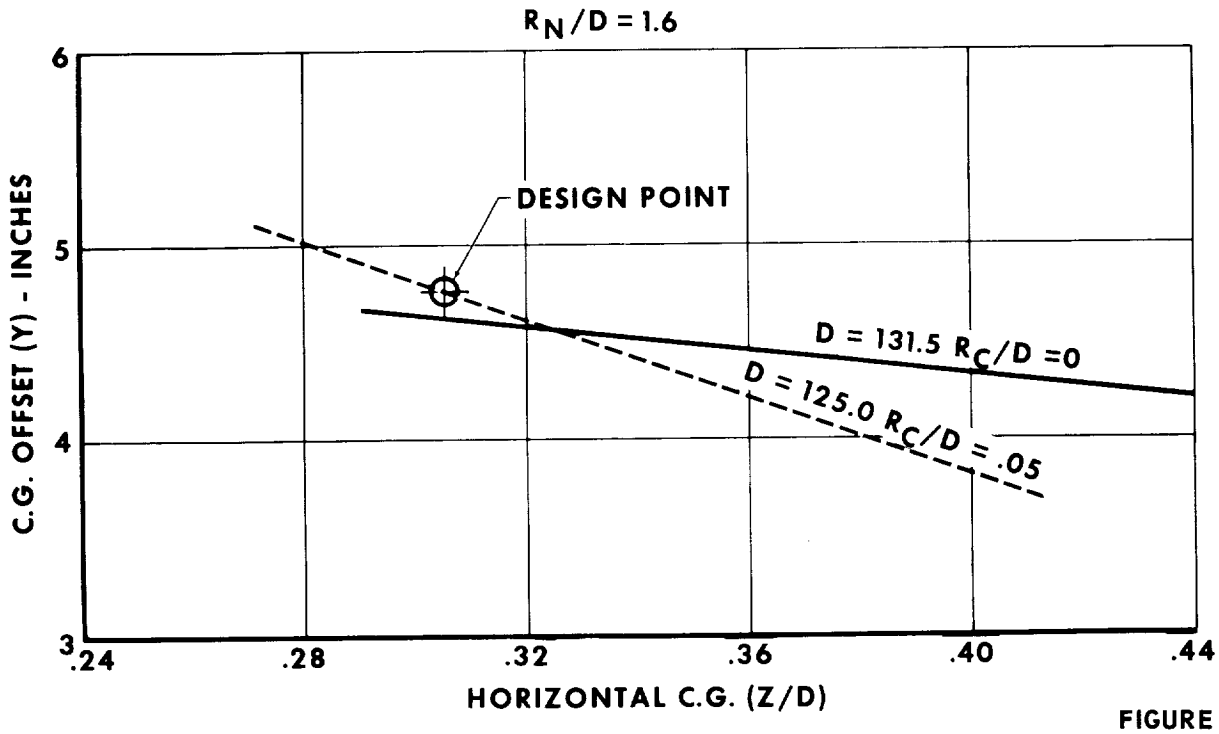


FIGURE 4-29

4.5 Propellant Margins - The incremental velocities specified in Reference 4-3, used for the spacecraft design, have a ten percent pad. This margin is to cover contingencies such as propulsion dispersions, wider range in lunar landing locations, variation in trip time, spacecraft weight variations, and guidance errors. The ten percent is applied to each mission phase and does not necessarily allocate appropriate propellant loading throughout the stages.

An assessment of the possible sources of contingency and the magnitude of such contingencies has been made by stage and mission phase. These contingencies, listed in Table 4-3, were applied to configuration III-1 and the resulting increments of velocity computed. These results are summarized in Table 4-4. On a direct summing basis the net velocity increment is 11.1 percent. In this analysis the propulsion effects, guidance and control errors, man-in-the-loop, weight variations are considered "worst-on-worst" in arriving at the total required contingency. It is realized that this is probably not a completely valid approach to the problem. Assuming that the root-sum-square approach is a more nearly true case, the total effect for each module becomes:

Retrograde Module	4.25%
Terminal Landing Module	11.10%
Service Module	7.39%

Total Vehicle Net ΔV Margin = 6.01%

The actual case is probably somewhere between the two analyses, therefore, the 10% margin as quoted by the NASA in Reference 4-3 is considered sufficient for vehicle performance calculations using nominal propellant performance and inert weights.

TABLE 4-3

ΔV MARGIN ANALYSIS

	FUNCTION	BASIC ΔV	MAC CONTINGENCIES	
			G & C	MAN-IN-LOOP
RETROGRADE MODULE	MIDCOURSE	300 FPS		
	RETRO INTO 80 N.M. ORBIT	3130 FPS	0.2%	1.0%
	6° PLANE CHANGE	100 FPS	0.2%	1.0%
	HOHMANN TO 50,000 FEET	123 FPS		1.0%
	DESCENT TO STAGING ALT.	5961 FPS	2.0%	1.0%
			NET = 1.305%	NET = .97%
TERMINAL LANDING MODULE	FINAL DESCENT HOVER & TRANS.	700		10.0%
SERVICE MODULE	LAUNCH TO 50,000 FEET	5885	1.0%	1.0%
	TRANSEARTH INJECTION	3592	0.2%	1.0%
	MIDCOURSE	300		
			NET = .675%	NET = .97%

INERT WEIGHT VARIATION: 3%

PROPELLANT UTILIZATION FACTORS

FACTOR	TRANSLUNAR & MAIN RETRO	LUNAR LANDING MODULE	ATTITUDE CONTROL-LUNAR LANDING MODULE (BLADDER)	MAIN SERVICE MODULE PROPULSION	ATTITUDE CONTROL IN SERVICE MODULE (BLADDER)	ATTITUDE CONTROL IN COMMAND MODULE (BLADDER)
LOADING TOLERANCE	.5%	.5%	.5%	.5%	.5%	.5%
MIXTURE RATIO SHIFT						
TEMP. EFFECTS, BLADDER ΔP CHANGE, TRIM ORIFICE ADJ. (RSS)		2.3%	2.4%	2.3%	2.4%	2.4%
M.R. CONTROLLER	.5%					
TRAPPED PROPELLANT						
TANK	1.0%	1.0%	1.0%	1.0%	1.0%	1.0%
LINES & MANIFOLDS	1.0%	1.0%	3.0%	1.0%	2.5%	5.0%
TOTAL PROPELLANT FACTOR (BASED ON USABLE)	3.0%	4.8%	6.9%	4.8%	6.4%	8.9%

4.5 (Continued)

TABLE 4-4

ΔV MARGIN ALLOTMENT-%

	RETROGRADE MODULE	TERMINAL LANDING MODULE	SERVICE MODULE
PROPULSION: SPECIFIC IMPULSE	2.38	2.33	2.18
PROPELLANT UTILIZATION	2.94	4.09	6.71
GUIDANCE & CONTROL	1.31	—	.68
MAN-IN-LOOP	.97	10.00	.97
INERT WEIGHT	1.04	1.19	1.72
TOTAL - INCLUDING INTERACTION EFFECTS	8.92	18.56	12.74

TOTAL VEHICLE NET ΔV MARGIN 11.1%
TOTAL VEHICLE ROOT-SUM-SQUARE ΔV MARGIN = 6.01%

The guidance and control errors and the effect of man-in-the-loop are evaluated on the basis of the equipment in use during each function. The man-in-the-loop effect is predominant during the lunar terminal descent phase where hover and translation maneuvers may be required. The inert weight variation assumed for the study is substantitated by experience on Mercury and Gemini. The propulsion contingencies considered are propellant loading tolerance, mixture ratio shift, propellant tank bladder differential pressure changes, engine trim orifice accuracies, specific impulse variations and trapped propellants throughout the various systems. It is assumed that a mixture ratio controller and propellant utilization system will be utilized on the main retrograde stage, thus, any mixture ratio shifts due to temperature or pressure effects, will be automatically compensated for. For the remaining systems, the effects of temperature, bladder ΔP change and trim orifice adjustment are considered to be independent and random

4.5 (Continued)

in nature; therefore, these effects are root-sum-squared for the analysis. The trapped propellant in lines and manifolds is based on actual vehicle designs (Gemin, Apollo, Centaur) and varies as a function of module size, number of thrust chambers, and line routing. The propellant trapped within the tanks is assumed to be 1% whether the analysis is based on the NASA specified 10% ΔV contingency or on the M.A.C. derived contingencies and thus does not enter into the final answer. Minimum guaranteed propellant specific impulse values were obtained from various engine manufacturers for the propellant combinations involved. From this survey and M.A.C. judgment of data authenticity, the values of specific impulse shown in Table 4-5 are chosen for this study.

TABLE 4-5

Propellant Specific Impulse

	Propellant Combination	Main Stage Steady State	Main Stage Throttling	Reaction Control Steady State	Reaction Control Min. Pulse
Nominal I_{sp}	O_2/H_2	430	430	----	----
	N_2O_4/MMH	320, $\epsilon = 40$ 308, $\epsilon = 20$		300 $\epsilon = 40$	260
3σ Low I_{sp}	O_2/H_2	420	420	----	----
	N_2O_4/MMH	313, $\epsilon = 40$ 300, $\epsilon = 20$	305, $\epsilon = 40$ 293, $\epsilon = 20$	300 $\epsilon = 40$	260

4.6 Radiation - The ground rules for the study, Section 1.1, stipulate that no radiation protection material be added over and above the protection already furnished by structure and equipment - the inherent shielding. In Table 4-6 the inherent shielding is summarized for two points on the astronaut's body and for two positions of the astronaut. The values shown are calculated by considering the line of sight areal density (thickness times density), usually referred to as thickness, of the various materials for approximately 40 solid angles. The equivalent thickness of aluminum is determined from proton range-energy relationships for the various materials and the results are shown in terms of aluminum thickness. The equipment contribution to the inherent shielding is calculated by determining the line of sight distance through the equipment bays for each of the solid angles, determining an areal density using an equipment density of 24 pounds per cubic foot and assuming the equipment to be as effective as aluminum. There exist voids between equipment through which radiation can arrive by penetrating only the continuous structure. This has been accounted for by assuming the installed equipment has a porosity factor of 20% and, therefore, does not shield 20% of the solid angles through equipment bays. The parachutes, which are tightly packed, are assumed to be continuous. The shielding provided by the body has been included in the thickness shown in Table 4-6. The shielding effect of the second astronaut has not been included due to the uncertainty of his location relative to the point considered on the first astronaut.

The two positions chosen for analysis are the navigation and sleeping positions. Figure 4-30 shows the associated major radiation windows. If a solar event should be encountered, it is assumed that the astronauts would locate themselves in one or the other of these positions which tend to use the inherent shielding to maximum advantage in protecting the eyes. These positions are

4.6 (Continued)

TABLE 4-6

INHERENT RADIATION SHIELDING

ASSUMPTIONS

1. PERSONAL PARACHUTES OVER EMERGENCY HATCH
2. ASTRONAUTS IN SPACE SUIT WITH VISOR CLOSED
3. ASTRONAUT SLEEPS FACING HEAT SHIELD
4. NO SHIELDING PROVIDED BY PARTNER

NOTE: SELF SHIELDING BY BODY INCLUDED

POINT CONSIDERED	NAVIGATION POSITION		SLEEPING POSITION	
	FRACTION SOLID ANGLE	SHIELDING g/cm² ALUMINUM	FRACTION SOLID ANGLE	SHIELDING g/cm² ALUMINUM
OUTBOARD EYE	.204	2.2	.051	2.6
	.081	7.7	.259	7.6
	.466	19.4	.529	16.7
	.157	37.4	.121	40.5
	.092	>50.0	.040	>50.0
SKIN ON CENTER OF BACK	.146	2.1	.186	1.9
	.134	6.7	.194	9.1
	.231	16.5	.207	22.5
	.237	34.0	.399	32.5
	.252	> 50.0	.014	>50.0

selected since, as shown in Reference 4-4, the eyes have the minimum allowable dose. The navigation position is compatible with tasks the crew must perform which occupy most of their duty time such as monitoring instrumentation, taking navigation fixes and making midcourse trajectory corrections. Sleeping facing the heat shield, which is the best position for eyes, should impose no hardship on the astronauts.

As seen in Figure 4-30, the major radiation windows are located on one side of the command module. This is characteristic of re-entry vehicles which utilize an offset center of gravity for trimming to angle of attack. The personal parachutes are moved from the normal stowage in the seats to the emergency hatch after translunar injection in order to reduce the radiation window. The required offset center of gravity is not affected since the parachutes are returned to the seats during re-entry. Although it has not been studied in detail, it appears possible to use other equipment in the same manner but with less effectiveness.

RADIATION WINDOWS

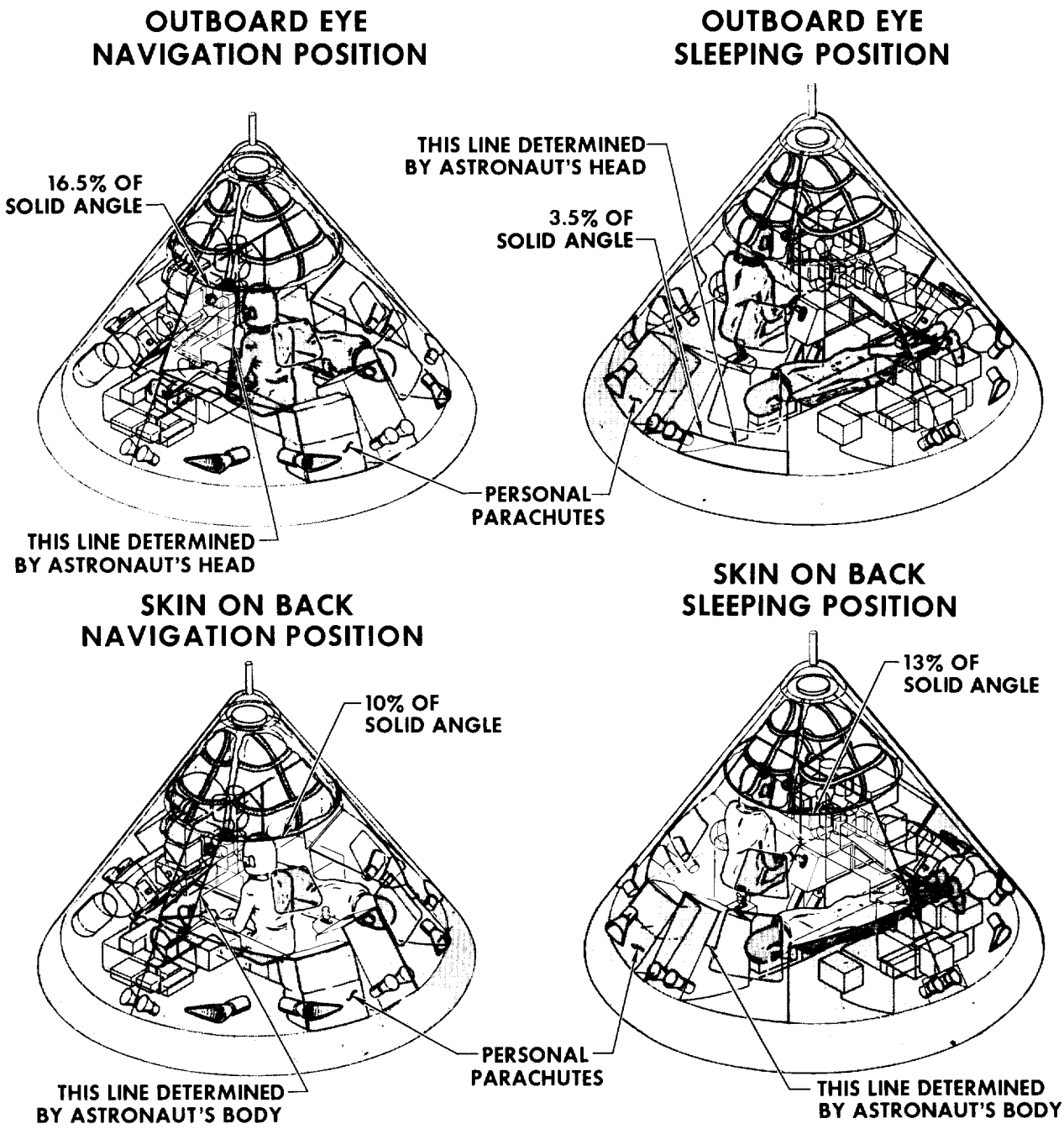


FIGURE 4-30

4.7 Meteoroid Protection - In the command module, multiple walls, separated by low density insulation and covered by ablation material, provide inherent meteoroid protection for the pressurized cabin and its occupants. In the propulsion modules, various degrees of meteoroid protection are afforded the tankage by the multiple wall effect of tank plus structural shell. The open lower ends of the retrograde module and service module during translunar and transearth coasting flight respectively expose the thin walls of propellant tanks to direct meteoroid impact with no bumper protection.

Meteoroid penetration is computed in accordance with the criteria summarized in Figure 4-31. The probabilities of no meteoroid penetration for the basic structural arrangement and the weight associated with increased meteoroid protection are shown for the service and retrograde modules. The retrograde module

METEOROID PROTECTION

CRITERIA

FREQUENCY DISTRIBUTION PER
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PENETRATION EQUATION

$$\frac{P}{d} = \gamma \left(\frac{\rho_p}{\rho_t}\right)^{1/3} \left(\frac{V_p}{C_t}\right)^{1/3}$$

WHERE

P = DEPTH OF PENETRATION
d = PROJECTILE DIAMETER
γ = A CONSTANT = 4
ρ_p = PROJECTILE DENSITY
ρ_t = TARGET DENSITY
V_p = PROJECTILE VELOCITY
C_t = SPEED OF SOUND IN TARGET MATERIAL

PROTECTION

ITEM	PROBABILITY OF NO PENETRATION (p)	ADDED WEIGHT FOR p = .990	ADDED WEIGHT FOR p = .999
COMMAND MODULE - CREW	.99911	—	—
PRESSURE WALL	.9978	—	54 LB.
SERVICE MODULE TANKAGE	.7320	13.5 LB.	128 LB.
TERMINAL LANDING MODULE TANKAGE	.99979	—	—
RETROGRADE MODULE TANKAGE	.0590	254 LB.	2950 LB.
(REDUCTION IN ALLOWABLE COMMAND MODULE WEIGHT)		(60.0 LB.)	(722 LB.)

FIGURE 4-31

4.7 (Continued)

tank weight includes a contingency of 824 pounds to assure that a high confidence tank weight is included in the stage performance analysis. This weight can be allocated for meteoroid penetration or other purposes and is in excess of the computed weight of the stage as designed. The 254 pounds for achieving .99 probability is well within this pad.

4.8 Weight Derivation - Weights are derived by three methods.

A. Extrapolation of existing hardware weight data. Mercury and Gemini data are utilized wherever applicable with NAA Apollo, Titan and Saturn data being used in specific areas.

B. Detail analysis of layout drawings, system schematic drawings and structural components including non-optimum factors based on the contractors past experience.

C. Vendor estimates of certain equipment.

The weight estimates are deliberately conservative, assuming current state-of-the-art materials and designs as defined by Mercury and Gemini, or, in specific cases, items under development for existing programs.

Comparisons of the derived Two-Man Apollo weights with Mercury, Gemini, and NAA Apollo are included to substantiate the overall analytical procedure. The command module subsystems and service module equipment are grouped in accordance with NAA Apollo weight breakdowns from Reference 4-7. The propulsion modules are not compared directly because of functional differences.

Mercury data is from the weight and balance statement for Spacecraft 19, the last three-orbit spacecraft, (Reference 4-5). Gemini weight data is current for the 14 day, no rendezvous provisions, spacecraft (Reference 4-6).

In some instances, because of uncertainties in grouping individual components, the NAA Apollo system weights used in other sections of this report to derive Two-Man Apollo system weights do not agree with Reference 4-7 but reflect vendor estimates of current system status. However, it is believed that a more valid total spacecraft weight comparison is obtained if NAA Apollo data from Reference 4-7 are used without modification in this section.

Table (4-7) presents a summary of Mercury, Apollo, NAA Apollo and Two-Man Apollo spacecraft weights.

TABLE 4-7

SPACECRAFT GROUP WEIGHT SUMMARY

GROUP	MERCURY C.M.	GEMINI C.M.	GEMINI S.M.	NAA APOLLO C.M.	NAA APOLLO S.M.	2-MAN APOLLO C.M.	2-MAN APOLLO S.M.
STRUCTURE	574.16	1319.44	53.60	2530		1042.4	54.0
HEAT SHIELD	307.59	227.00		1320		633.3	
NAVIGATION AND GUIDANCE		206.63		310	3	297.0	3.0
STABILIZATION AND CONTROL	88.14	40.40	9.24	220		204.0	10.0
REACTION CONTROL	187.21	192.70		405		209.7	
EARTH LANDING	334.07	793.81		631		386.0	
ELECTRICAL	348.05	300.14	523.69	496	1816	320.5	456.3
DISPLAYS	134.61	173.91		490		348.0	
TELE-COMMUNICATIONS	211.63	188.23	106.88	352	213	419.3	249.0
ENVIRONMENTAL CONTROL	142.13	339.56	369.51	417	588	304.0	376.0
CREW SYSTEM	326.64	929.09		1499		905.2	
SCIENTIFIC EQUIPMENT				250		85.0	
C.M. & S.M. EQUIPMENT TOTAL	2654.23	4710.91	1062.92	8920	2620	5154	1148
BALLAST & MF. VARIATIONS	62.00						
SPACECRAFT WEIGHT MARGIN						1477	
RETROGRADE SYSTEM	271.19	348.63					
FAIRINGS & ADAPTERS & SEPARATION	189.68	113.57				1400	
LAUNCH ESCAPE PROPULSION SYSTEM	1088.42					2600	
SERVICE MODULE							
STRUCTURE		256.20		3614		1079	
REACTION CONTROL SYSTEM		495.06		1179		639	
MAIN PROPULSION SYSTEM				43367		20443	
EQUIPMENT						80	
LANDING STAGE						6030	
TRANS-LUNAR & RETRO STAGE						53787	
TOTAL WEIGHT AT LAUNCH	4265.52	6987.29		59700		93837	

4.8.1 Command Module Structure - Command module structural system weights are calculated for major components such as sidewalls, bulkheads, attach rings and heat shield.

4.8.1.1 Structural System - Structural criteria and descriptions are given in Sections 2.1, 2.7, 3.6 and 3.7. Unit weight comparisons of the entire structural system are:

	Mercury	Gemini	NAA Apollo	Two-Man Apollo
Structural System - pounds	(881.75)	(1546.44)	(3850)	(1675.7)
Structure-Basic - pounds	574.16	1319.44	2530	1042.4
Heat Shielding - pounds	307.59	227.00	1320	633.3
Outer ML Wetted Area - sq. ft.	130.2	211.8	441.5	249.9
Gross Unit Weight - psf	6.8	7.3	8.7	6.7
Useable Volume - cu. ft.	84	165	539	185
Gross Unit Weight - pcf	10.5	9.4	7.2	9.1

Table 4-8 presents a comparison of Mercury, Gemini, NAA Apollo and Two-Man Apollo structural systems. The derivation of the major structural component weights follows:

Sidewalls

	Mercury	Gemini	Two-Man Apollo
Area-Sidewalls - sq. ft.	86.2	149.2	108.9
Area-Windows and Hatches - sq. ft.	- 6.0	- 22.8	- 4.1
Area-Net - sq. ft.	80.2	126.4	104.8
Weight-Gross-Current - pounds	82.94	252.78	
Add Portion of Chute Provisions	14.40		
Remove Contour Break Ring	-11.55	- 8.54	
Remove Flotation Bag Provisions	- 3.73		
Remove Paraglider and Gear Supports		- 13.16	
Remove Docking Provisions		- 9.66	
Remove Equipment Access Door Increment		- 24.00	
Remove Landing Gear Door Complications		- 10.00	
Weight-Basic Sidewalls - pounds	82.06	187.42	131.0
Unit Weight - psf	1.02	1.48	1.25
Unit Weight-Chute Compartment Sidewall - psf			0.90
Area-Chute Compartment Sidewall - sq. ft.			29.1
Weight-Chute Compartment Sidewall - pounds			26.0
Total Sidewall Structure Weight - pounds			(157.0)

TABLE 4-8

STRUCTURAL SYSTEM WEIGHT COMPARISON

STRUCTURE - BASIC	MERCURY C.M. 574.16	GEMINI C.M. 1319.44	NAA APOLLO C.M. (1) 2530	TWO-MAN APOLLO C.M. 1042.4	TWO-MAN APOLLO S.M. 54.0
SIDEWALLS	(82.94)	(252.80)		157.0	
SKIN AND STIFFENERS	30.98	39.64			
STRINGERS	40.41	79.52			
RINGS	11.55	78.54			
EQUIPMENT ACCESS DOORS		55.10			
AFT BULKHEAD	41.89	(61.35)		149.0	
FORWARD BULKHEAD	16.44	(15.42)		15.0	
TOWER/PARAGLIDER ATTACH	5.57	(5.73)	861	13.0	
FOREBODY ATTACH.	17.74	(10.50)		35.0	
HATCHES AND WINDOWS	(57.39)	(352.10)		(247.4)	
HATCH-EMERGENCY EGRESS	16.06			16.0	
HATCH-EJECTION		297.45			
PERISCOPE/UMBILICAL	9.48	10.00		8.0	
LANDING GEAR		10.00			
OBSERVATION WINDOWS	29.82			141.9	
E.C.S. BAY-TOTAL		33.45			
E.C.S. BAY-INCREMENT				10.0	
EQUIPMENT-INCREMENT				15.0	
MISCELLANEOUS	2.03	1.20		5.1	
HATCH-HINGED-INCREMENT				51.4	
SHINGLES/ABLATION BACKUP	113.78	216.39	1211	80.0	
INSULATION	51.18	174.62	260	209.0	
CHUTE BACKUP	35.85	-		35.0	
HOISTING	3.86	7.70		6.0	
CLAMP RING ATTACH.	7.49	20.00		4.2	
EQUIPMENT MOUNTING	31.69	(55.80)		46.0	
CABIN WALL		47.80			
EQUIPMENT BAY		8.00	198		
PARAGLIDER AND GEAR BACKUP	-	47.53		-	
PAINT AND SEALANT	5.00	11.70		5.8	
NOSE ANTENNA & JETTISONED STRUCT.	70.42	35.26		-	
MISCELLANEOUS STRUCTURE	(32.92)	(52.54)		40.0	
INTERNAL STRUCTURE	27.52	29.33			
BALLAST MOUNTING		1.00			
SNORKEL INLET	3.60	3.60			
MISCELLANEOUS	1.80	17.19			
FIXED ANTENNA		1.42			
HEAT SHIELD	307.59	227.00	1320	633.3	
FOREBODY	307.59	(227.00)		(479.9)	
FOREBODY ABLATED		24.06			
FOREBODY ABLATION RETAINED		72.19		437.6	
FOREBODY ABLATION RING		33.98			
MOUNTING PROVISIONS		4.08		16.6	
FOREBODY BACKUP		(76.12)		(25.7)	
HONEYCOMB & TITANIUM RING		76.12			
TITANIUM SKIN				25.7	
AFTERBODY				(153.4)	
ABLATION MATERIAL				133.4	
MOUNTING PROVISIONS				20.0	
GROWTH ALLOWANCE		16.57			

(1) DISTRIBUTION OF NAA APOLLO WEIGHT IS UNKNOWN

4.8.1.1 (Continued)

A unit weight of 1.25 pounds per square foot is selected based on Mercury and Gemini unit weights. The higher Gemini unit weight results from the large access doors and is only partially applicable to the Two-Man Apollo. A unit weight of 0.90 pounds per square foot is representative of the lightly loaded sidewall structure of the chute compartment.

Detail Calculation Check

Item	Weight-Pounds
.016 titanium wall	36.0
Twenty .032 (1.1 x 1.75 x .80 x 1.75 x 1.1) hats, total length Twenty .032 (1.1 x 1.75 x .80 x 1.75 x 1.1) hats, 1/2 length	58.3
Calculated Sidewall Weight	(94.3)
Unit weight-calculated-94.3 lbs/104.8 sq. ft. = 0.90 psf	
The resulting non-optimum-material factor of 1.39 (1.25 psf/.90 psf) is conservative based on Mercury values of 1.12 to 1.20	

Aft Bulkhead - The impact attenuation system imparts a 30 pound per square inch crushing load to the aft bulkhead. Aluminum honeycomb is selected as the basic bulkhead structure.

	Mercury	Gemini	Two-Man Apollo
Misc. Attachment, Sealant, Supports - pounds	13.89	20.51	23.1
Skin and Attach Ring - pounds	28.00	40.89	(125.9)
Honeycomb-Wide Span Sections - psf			(1.85)
.032 - .040 Aluminum Faces - psf			1.04
Core - 1 inch @ 7.9 pcf - psf			0.66
Glue and Edging - psf			0.15
Honeycomb - Short Span Section - psf			1.20
Weight - Honeycomb - pounds			
(1.85 psf) (39.3 sq. ft.)			72.7
(1.20 psf) (26.0 sq. ft.)			31.2
Attach Ring - Titanium - .38 sq. in. - pounds			22.0
Weight - Aft Bulkhead - pounds	(41.89)	(61.40)	(149.0)

4.8.1.1 (Continued)

Figure 4-32 presents a weight/area comparison of bulkheads.

Forward Bulkhead

	Mercury	Gemini	Two-Man Apollo
Area - sq. ft.	8.6	11.4	12.5
Weight - pounds	16.44		
Less Escape Hatch Increment	- 8.56		
Weight - Basic - pounds	(7.88)	(15.42)	(15.2)
Unit Weight - psf	.92	1.35	1.25

The Gemini forward bulkhead is complicated by the attachment to a non-symmetrical pressure vessel.

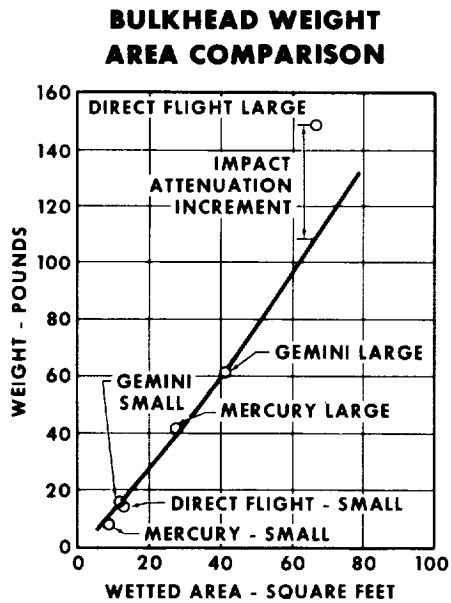


FIGURE 4-32

Tower Attach Ring - The tower attach ring is extrapolated from Mercury tower data utilizing a ratio of capsule weights as representative of the attach ring loads.

Item	Derivation	Weight-Pounds
Mercury Ring		5.6
Basic Ring	$5.6 \left(\frac{5100 \text{ pounds}}{2800 \text{ pounds}}\right)^{.6}$	8.0

4.8.1.1 (Continued)

Item	Derivation	Weight-Pounds
Allowance	Attach area heat protection	5.0
Tower Attach Weight		(13.0)

Forebody Attach Ring - 0.62 sq. in. titanium ring (35.0)

Hatches and Windows - Glass thicknesses are calculated from strength requirements.

Item	Weight-Pounds
Observation Window - 2.7 sq. ft.	(83.3)
2 Panes Alumino-Silicate Glass 0.443 inches thick	31.4
1 Pane Vycor Glass 0.69 inches thick	21.1
Frame Weight - Extrapolated from Mercury @ 0.39 lbs/inch	30.8
View Windows - two windows @ 0.95 sq. ft. area	(58.6)
4 Panes Alumino-Silicate Glass 0.264 inches thick	13.2
2 Panes Vycor Glass 0.412 inches thick	8.8
Frame Weight @ 0.39 lbs/inch	36.6
Emergency Egress Hatch - 2 @ 8 pounds	(16.0)
Extrapolated from Mercury - 16.1 pounds less 8.1 pounds basic structure	
Ingress/Egress Hatch - The latching and hinging portions of the Gemini egress hatch are ratioed by perimeters of the two doors.	(51.4)
Sill, Seal, and Corner Fittings	12.3
Torque Box	5.6
Hinge Beam	5.5
Latch Support Fittings	7.1
Hinge	2.2
Flipper Door and Associated Complications	16.2
Latch Mechanism	12.5
Reduce 15% with elimination of ejection loads	- 10.0
E.C.S Access Door - Incremental Weight (for bolt-on door)	(10.0)
Equipment Access Door - Incremental Weight	(15.0)
Umbilical	(8.00)
Miscellaneous Allowance	(5.1)

Ablation Back-Up - 133.9 square feet of single-faced, double-skin corrugation of .008 titanium (1.08 non-optimum) (80.0)

Insulation

Item	Weight-Pounds
Forebody (98.2 sq. ft.) (0.50 in. Min-K @ 20 pcf)	81.5
Afterbody - Cabin (104.8 sq. ft.) (1.5 in. RF 700)	91.2

4.8.1.1 (Continued)

Item	Weight-Pounds
Afterbody - Chute (29.1 sq. ft.) (1.5 in. RF 300)	11.1
Afterbody Stringers 0.20 in. Min-K @ 20 pcf on each stringer	4.2
Installation and allowance for special areas such as RCS nozzle cutouts and stringer attachments	21.0
Weight - Total	(209.0)

Chute Provisions - Extrapolated from Mercury

Item	Derivation	Weight-Pounds
Mercury Chute Provisions	35.85 pounds less 14.40 pounds of sidewall	21.45
Chute Provisions	$\frac{4900 \text{ pound ldg weight}}{2480 \text{ pound ldg weight}}^{.6}$ 21.45	35.0
Weight - Total		(35.0)

Hoist Provisions - Extrapolated from Mercury using capsule weight as parameter (6.0)

Clamp Ring Attach - Calculated weight of 0.06 fiberglass ablation cover on forebody attach ring (4.2)

Equipment Mounting - The weight of equipment mounting structure is based on Mercury and Gemini basic shelving. Reference is made to Figure 4-33. (46.0)

Paint and Sealant - (5.8)

Miscellaneous Structure - Estimated from Mercury and Gemini (40.0)

4.8.1.2 Heat Shielding - The detail criteria for the ablation material and methods of calculating required thickness are presented in Section 3.7.

Items	Descriptions	Weight-Pounds
Forebody Ablation	M.A.C. Thermorad 5-3, thickness 0.72 to 1.38 inches, area 110.8 sq. ft.	437.6
Forebody Back-up	.010 titanium skin	25.7

4.8.1.2 (Continued)

Items	Descriptions	Weight-Pounds
Forebody Mounting	0.15 psf for bonding, mounting screw plugs and other complications	16.6
Afterbody Ablation	M.A.C. Thermorad 5-3, thickness 0.20 to 0.30 inches area 134.8 sq. ft.	133.4
Afterbody Mounting	0.15 psf	20.0
Total		(633.3)

SUPPORT STRUCTURE - EQUIPMENT

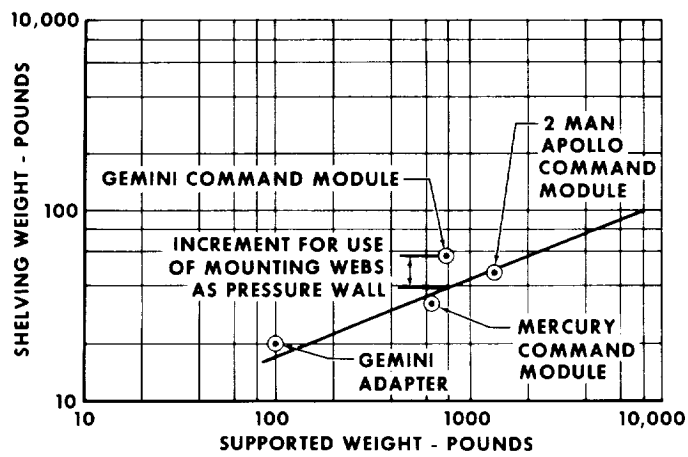


FIGURE 4-33

4.8.2 Command and Service Module Equipment

4.8.2.1 Guidance and Navigation System - The guidance and navigation equipment weights are based on the system being developed for the NAA Apollo vehicle with the changes described in Section 3.1. Table 4-9 presents the weight breakdown for the Mercury, Gemini, NAA Apollo and Two-Man Apollo guidance and navigation systems. The landing radar weights are included in the terminal landing stage. Compared to the Gemini computer, the Two-Man Apollo computer weight estimate is conservative and is a source of potential weight reduction.

NAVIGATION-GUIDANCE SYSTEM WEIGHT COMPARISON

NAVIGATION AND GUIDANCE	MERCURY C.M.	GEMINI C.M. 206.63	GEMINI S.M.	NAA APOLLO C.M. 310	NAA APOLLO S.M. 3	TWO-MAN APOLLO C.M. 297.0	TWO-MAN APOLLO S.M. 3.0
INERTIAL SYSTEM							
PLATFORM		31.77		60		60.0	
PLATFORM POWER SUPPLY		42.63					
PLATFORM ELECTRONICS		44.21				24.0	
POWER SERVO				20			
COMPUTER SYSTEM							
DIGITAL COMPUTER		61.51		100		100.0	
COMPUTER SPARES						10.0	
OPTICAL MEASUREMENT SY							
SEXTANT				27		27.0	
TELESCOPE				8		8.0	
DISPLAYS AND CONTROLS							
MAPS AND VISUAL DISPLAY		1.80		9		9.0	
COUPLING DISPLAYS				6		6.0	
FORWARD LOWER				15			
RIGHT HAND				3			
LOWER CENTER				15			
MISCELLANEOUS							
NAVIGATION BASE				17		10.0	
STRUCTURAL SUPPORTS		9.14		8		8.0	
CIRCUITRY		10.01		18		18.0	
E.C.S. PROV.		5.56		4		4.0	
SUN SENSOR					3		3.0
ROLL MOMENTUM WHEEL						13.0	

4.8.2.2 Stabilization and Control - Table 4-10 presents the weight breakdown of Mercury, Gemini, NAA Apollo and Two-Man Apollo stabilization and control systems.

The control electronics package of Gemini weighs 30.14 pounds, including its power supply, compared to the NAA Apollo weight of 102 pounds without power supply. The weight of the control electronics package for the Two-Man Apollo spacecraft is based on subtraction of the rendezvous control mode from the NAA Apollo package, a reduction of 10 pounds.

4.8.2.3 Reaction Control System - The reaction control system weights for the Two-Man Apollo are extrapolated from Gemini data. Table 4-11 presents a comparison of these weights with those of the other spacecraft.

4.8.2.4 Earth Landing System - The earth landing system consists of a cluster of three parachutes and a crushable honeycomb impact attenuation system. The design landing weight is 4900 pounds. A comparison of the Two-Man Apollo earth landing system weights with those of the other spacecraft is given in Table 4-12.

TABLE 4-10

STABILIZATION AND CONTROL SYSTEM WEIGHT COMPARISON

STABILIZATION AND CONTROL SYSTEM	MERCURY	GEMINI		NAA APOLLO		TWO-MAN APOLLO	
	C.M. 88.14	C.M. 40.40	S.M. 9.24	C.M. 220	S.M.	C.M. 204.0	S.M. 10.0
MANUAL RCS CONTROL RODS	15.87						
CANOPUS SENSOR							10.0
SUN SENSOR				2		2.0	
ATTITUDE GYRO'S	14.78						
IMU STRAPDOWN							
INERTIAL REFERENCE				12		12.0	
ACCELEROMETER				5		5.0	
ACCELEROMETER	.61						
RATE GYRO	6.03	8.50		11		11.0	
RATE GYRO SPARE				2		2.0	
CONTROL ELECTRONICS							
AMP-CAL UNIT	31.28						
CONTROL ELECTRONIC		15.90	7.24	90		80.0	
CONTROL ELECTRONIC SPARE				12		12.0	
INVERTER		7.00					
POWER SUPPLY				15		15.0	
POWER SUPPLY SPARE				15		15.0	
RATE DAMPER UNIT	5.25						
SELECTOR PANEL				6			
DISPLAY MODULAR PKG.				34		34.0	
STRUCTURAL SUPPORTS	1.28			4		4.0	
CIRCUITRY	13.04	9.00	2.00	10		10.0	
ENVIRONMENTAL CONTROL				2		2.0	

TABLE 4-11

REACTION CONTROL SYSTEM WEIGHT COMPARISON

REACTION CONTROL SYSTEM		MERCURY C.M. 187.21	GEMINI C.M. 192.70	NAA APOLLO C.M. 405	TWO-MAN APOLLO C.M. 209.7
	AUTO SY. 81.09	MAN. SY. 106.12			
PRESSURIZATION SYSTEM	(10.29)	(10.26)	(29.26)	(37)	(26.0)
GASS	1.71	1.71	2.80	2	.5
BOTTLE	5.35	5.35	4.40	9	3.8
VALVING, PLUMBING FITTINGS	3.23	3.20	20.26	20	20.0
PLUMBING FITTINGS			1.80	6	1.7
PROPELLANT- SYSTEM	(36.56)	(47.77)	(90.88)	(221)	(107.8)
PROPELLANT-USABLE	23.40	32.00	70.00	200	93.0
PROPELLANT-TRAPPED	2.73	2.40	4.00	10	4.0
PROPELLANT-TANKAGE	10.43	13.37	16.88	11	10.8
THRUST CHAMBERS	5.66	15.44	36.80	78	48.0
INSTALLATION, VALVING, CIRCUITRY	28.58	22.91	35.76	69	27.9
QUICK FIXES		9.74			

4.8.2.4 (Continued)

TABLE 4-12

EARTH LANDING SYSTEM WEIGHT COMPARISON

EARTH LANDING SYSTEM	MERCURY C.M. 334.07	GEMINI C.M. 793.81	NAA APOLLO C.M. 631	TWO-MAN APOLLO C.M. 386.0
MAIN CHUTE/CHUTE CLUSTER	(67.86)		348	(222.0)
CANOPY & RISER	57.21			207.0
BAG AND INSTALLATION	10.61			15.0
RESERVE CHUTE	(67.11)			
CANOPY AND RISER	57.25			
BAG AND INSTALLATION	9.86			
DROGUE CHUTE	(11.51)		66	19.0
CHUTE	5.56			
MORTAR BAG AND INSTALLATION	5.95			
PILOT CHUTE	(4.42)	15.13		6.0
CANOPY	1.67			
INSTALLATION	2.75			
CONTROLS/BAROSTATS	4.12		14	6.0
MOUNTING	2.32			
PARAGLIDER		519.22		
CIRCUITRY	18.42		12	10.0
IMPACT ATTENUATION SYSTEM	144.01		179	112.0
RECOVERY SYSTEM	(14.30)	(6.55)	12	(11.0)
RECOVERY LIGHT	4.40	3.75		4.4
SOFAR	4.36			
DYEMARKER	3.54	2.80		2.8
CHAFF/SHARK REPELLANT	.60			2.4
MOUNTING AND CIRCUITRY	1.40			1.4
SKID GEAR ASSY.		252.91		

The weight of the main chute is obtained from Figure 4-34. Remaining portions of the chute system are based on Mercury weights. With the simpler simultaneous deployment system, circuitry weight is less than for Mercury.

The impact attenuation system consists of an 11.5 inch thick crushable truss-grid at 1.6 pounds per cubic foot. (1.6 pcf) (70.0 cu. ft.) = 112 pounds. Recovery system unit weights are based on Mercury and Gemini data.

4.8.2.5 Electrical System - The weights are based on the Gemini system with appropriate modifications for the shorter duration, higher power level and additional propulsion module circuitry requirements of the Two-Man Apollo spacecraft. Section 3.4 presents a detailed comparison of the NAA Apollo Bacon fuel cell system and the selected Gemini ion-exchange fuel cell system and shows the weight difference (NAA Apollo (1620 pounds) and Two-Man Apollo (364.9 pounds)) to be rational.

4.8.2.5 (Continued)

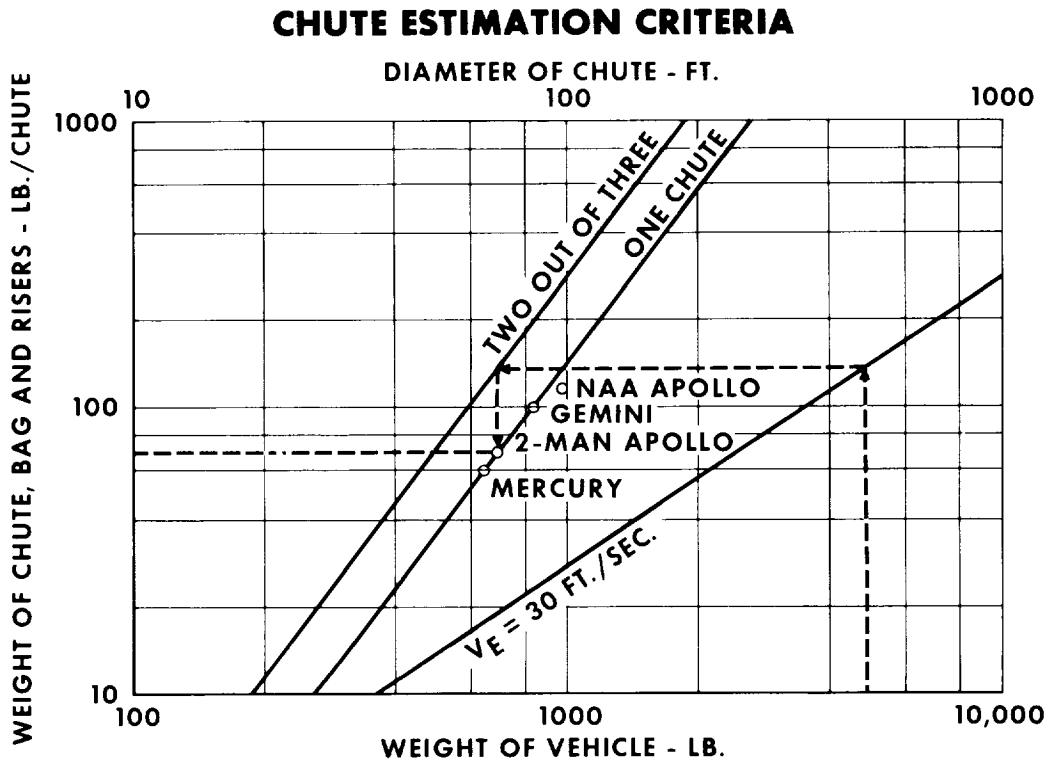


FIGURE 4-34

The comparison of Mercury, Gemini, NAA Apollo and Two-Man Apollo electrical system weights is presented in Table 4-13.

The command module batteries are designed to conservative Gemini criteria. Substitution of Apollo-type batteries with higher watt-hours per pound rating is a potential weight reduction.

4.8.2.6 Display System - The display system weights shown in Table 4-14 are taken from NAA Apollo data where applicable.

4.8.2.7 Telecommunications System - Unit weights for the telecommunications system are derived by independent analysis of each requirement. The system weight is compared in Table 4-15 with other similar equipment.

4.8.2.8 Environmental Control System - The component weights for the environment control system are based primarily on existing Gemini hardware.

The weights of the major items changed are derived as follows:

TABLE 4-13

ELECTRICAL SYSTEM WEIGHT COMPARISON

ELECTRICAL SYSTEM	MERCURY C.M. 348.05	GEMINI C.M. 300.14	GEMINI S.M. 523.69	NAA APOLLO C.M. 496	NAA APOLLO S.M. 1816	TWO-MAN APOLLO C.M. 320.5	TWO-MAN APOLLO S.M. 456.3
BATTERIES	(203.01)	(138.50)		(102)		(138.5)	
MAIN	152.20	110.00		65		110.0	
SQUIB/RESERVE/STANDBY	50.00	28.50		33		28.5	
INSTALLATION	.81						
CHARGERS				4			
INVERTERS	24.90			84			
FUEL CELL SYSTEM		(8.98)	(445.06)		(1620)	(9.0)	(364.9)
FUEL CELLS			119.20		781		119.2
FUEL CELL REACTANTS			202.25		550		150.8
REACTANT CONTAINERS			114.61		278		85.9
PLUMBING			9.00		11		9.0
CONTROLS		8.98				9.0	
PYROTECHNIC INITIATION				45	45		
CIRCUITRY - POWER	(25.46)	(33.63)	(27.72)	(118)	96	(33.6)	(27.7)
POWER - AC	5.47			17			
POWER RELAYS, FUSES, ETC.	3.99	29.73				29.7	
DIODE PANEL	1.88	3.90				3.9	
CIRCUITRY DC POWER	14.12		27.72				27.7
CIRCUITRY AND DISTRIBUTION				101			
UMBILICAL	2.95	3.00	2.00			3.0	2.0
DISCONNECTS		8.00	2.40			8.0	7.2
CIRCUITRY UTILITY/GENERAL	(46.30)	78.77	7.38	127		93.8	7.4
EXTERNAL POWER/GROUND TEST	12.15						
TOWER SEPARATE	4.74						
ESCAPE ROCKET FIRE	3.64						
PYLON JETTISON FIRE	1.20						
ADAPTER SEPARATION FIRE	8.95						
ADAPTER SEPARATION SENSE	.19						
ABORT CIRCUITRY	6.70						
EMERGENCY HOLD	2.21						
CONTROL AND MISCELLANEOUS	6.52						
CIRCUITRY - PROPULSION	(27.95)	20.62			3.0	26.0	8.0
RETRO FIRE	4.70						
RETRO HEAT	1.96						
RETRO SEPARATE	4.95						
POSI FIRE	3.41						
UMBILICAL SEPARATE	1.89						
RETRO FIRE SENSE	2.12						
CONTROL AND MISCELLANEOUS	8.92						
INSTALLATION PARTS	17.48	8.64	39.13	20	52	8.6	39.1

TABLE 4-14

DISPLAY SYSTEM WEIGHT COMPARISON

DISPLAY SYSTEM	MERCURY C.M. 134.61	GEMINI C.M. 173.91	NAA APOLLO C.M. 490	TWO-MAN APOLLO C.M. 348.0
COMMUNICATIONS & INSTRUMENTATION	(1.30)	(6.30)	(56)	(26.0)
TELECOMMUNICATIONS			8	8.0
COMMUNICATION CONTROL			8	8.0
AUDIO/VOICE CONTROL PANEL	1.30	6.30		
OPTICAL VIEW PLATE			30	
TV MONITOR			10	10.0
GUIDANCE AND NAVIGATION	(59.35)	(21.94)	(66)	(58.0)
DISPLAYS AND CONTROLS			52	
RADAR CONTROLS			14	
PLATFORM GROUP		1.49		6.0
SXT AND SCT GROUP				10.0
VIEWER	1.77			4.0
EARTH PATH INDICATOR	3.98			
PERISCOPE	45.26			
COMPUTER GROUP		4.88		5.0
PRINTER				2.0
CLOCK GROUP	8.34	11.57		12.0
INSTRUCTION TEMPLATE				2.0
RADAR GROUP				6.0
INCREMENTAL VEL. IND.		4.00		
CDU GROUP				4.0
POWER CONTROLS				6.0
CAUTION & ABORT LIGHTS				1.0
STABILIZATION AND CONTROL	(18.79)	(45.90)	(90)	(57.0)
CONTROLLER - EARTH/LUNAR LANDING				10.0
3 AXIS HAND CONTROLLER	2.78	4.50		
MANUAL ROTATIONAL, YAW, TRANSLATE			32	
THRUST CONTROL - LUNAR LANDING				3.0
SELECT PANEL				
MODE SELECT - LUNAR LANDING			3	2.0
FLIGHT DIRECTOR		5.60	12	2.0
FLIGHT DIRECTOR INDICATOR		16.80	19	10.0
ANGULAR RATE INDICATORS	7.63			
ADJUST PANEL				6.0
FLIGHT INDICATORS (THRUST CONTROL)			12	
TRANSLATIONAL VELOCITY				6.0
ENGINE GIMBAL ANGLE				6.0
HORIZON SCANNER	8.38	19.00	12	12.0
LAUNCH, RE-ENTRY AND ABORT	12.58	12.77	25	25.0
UTILITY CONTROLS AND DISPLAYS	(5.84)	(23.52)	(42)	(27.0)
ENVIRONMENTAL	3.99	13.62	12	12.0
POWER DISTRIBUTION	1.05	7.60	12	3.0
FUEL CELL REACTOR PANEL			12	2.0
PROPULSION	.80	2.30	6	10.0
IN-FLIGHT TEST			85	50.0
ELECTRICAL PROVISIONS	21.62	37.18	111	70.0
ENVIRONMENTAL CONTROL PROV.			15	
STRUCTURE	15.13	26.30		35.0

TABLE 4-15

TELECOMMUNICATIONS SYSTEM WEIGHT COMPARISON

TELECOMMUNICATIONS	MERCURY C.M. 211.63	GEMINI C.M. 188.23	GEMINI S.M. 106.88	NAA APOLLO C.M. 352	NAA APOLLO S.M. 213	TWO-MAN APOLLO C.M. 419.3	TWO-MAN APOLLO S.M. 249.0
DSIF SUBSYSTEM							
RECEIVER/TRANSMITTER					30		9.0
POWER AMPLIFIER					30		30.0
NEAR EARTH SYSTEM							
HF VOICE	3.22						
UHF/VHF AM	3.55	6.00		4		20.0	
VHF FM	6.36	4.62	2.31	10		20.0	
C-BAND TRANSPONDER	11.80	7.49		14		13.0	
COAXIAL SWITCH	1.78					1.0	
S-BAND	12.20						
COMMAND RECEIVER	9.50						
DIGITAL COMMAND			24.65				
ACQ AID BEACON			1.13				
RECOVERY							
UHF/HF RESCUE BEACON	2.90	2.80		2		2.8	
HF RECOVERY VOICE	2.33	3.50	3.50	4		4.0	
HF RECOVERY VOICE	2.30						
SUPER SARAH	3.81						
DATA PROCESSING							
TELEMETRY				38			
PROGRAMMER	2.75					7.0	
HL COMMUTATOR		2.40	2.40			12.0	6.0
LL COMMUTATOR		1.60	1.60			6.0	6.0
RECORDER	13.76	12.00		36		28.0	
PCM MULTIPLEXING		11.00					
INTER COMMUNICATIONS	1.95	1.69				3.0	
SPARES				19		20.0	
INSTRUMENTATION							
SENSORS	7.78	4.05	.06	18	25	18.0	25.0
TV SYSTEM				30		12.0	12.0
CAMERAS	15.06			20		25.0	
TELESCOPE				54		5.0	
SIGNAL CONDITIONING	20.86	11.75	4.03			21.0	7.5
TIMING DEVICES						7.5	
PATCH PANELS						12.0	
POWER SUPPLY		14.00	7.00			14.0	7.0
ANTENNA SYSTEMS				54			
DSIF ANTENNA AND DRIVE					76		50.0
EARTH TRACKER							6.0
ASSOCIATED COMPONENTS							5.0
2 KCI OMNI							.5
ASSOCIATED COMPONENTS							5.0
DISCONE ISOLATOR	1.20						
C-BAND ANTENNA/S-BAND	6.15	1.70	.85			1.5	
PHASE MODULATOR		1.70				1.7	
VHF OMNI		.20	1.00			4.0	
MULTIPLEXER	6.34	3.50	1.70			9.0	
DIPLEXER	.41						
UHF OR VHF RECOVERY	.81	3.20	2.00			1.3	
HF RECOVERY	4.95	7.13				4.0	
ASSOCIATED COMPONENTS		2.70	.61			16.5	
MISCELLANEOUS COMPONENTS	3.01						
STRUCTURAL SUPPORT	21.76			12	40	12.0	40.0
ENVIRONMENTAL CONTROL				20		20.0	
ELECTRONIC INTERFACE				8		8.0	
CIRCUITRY	45.09	72.10	25.00	9	12	90.0	40.0

4.8.2.8 (Continued)

Item		Derivation	Weight
O_2 Primary			(103.1)
Repressurizations	-	9 repressurizations @ 130 cu. ft. each	32.8
Leakage	-	.15 lb/hour	28.8
Usage	-	1.8 lb/man-day	28.8
Back Pack Recharge	-		2.5
Contingency			10.2
LiOH Unit			(71.0)
LiOH	-	2.3 lbs. CO_2/man-day 1.08 lbs LiOH/lb. CO_2 90% efficient absorption	44.0
Cannister	-	extrapolated from Gemini	27.0
Cold Plates			
Gemini	-	current total (all groups)	53.7
		increases for additional equipment	10.3
Two-Man Apollo	-	total cold plates	64.0
Cold Plates	-	grouped in telecomm., electrical, guidance and nav. and stab. and control	26.0
Cold Plates	-	grouped to environmental and system	38.0
Re-entry O_2	-	0.1 lbs/man-minute for 50 minutes with 100% redundancy	20.0
Re-entry O_2 Bottle	-	extrapolated from Mercury bottle	48.8

The weight comparison of the environmental control systems is shown in Table 4-16.

4.8.2.9 Crew Systems - The crew system weights are based on the NAA Apollo and Gemini systems. Items that are different from NAA Apollo or Gemini are:

Item		Derivation	Weight
Seat	-	An examination of the Mercury and Gemini seat designs as they apply to the Two-Man Apollo gives reasonable confidence that the Two-Man Apollo seat can be built as light as Gemini, less catapult. The seat weight includes the seat snubbing system.	158.4
Food	-	1.25 pounds/man-day includes packaging	27.5
Water System	-	The water system inventory is included in Section 3.3.	40.0

The weight comparison of the crew system is shown in Table 4-17.

TABLE 4-16

ENVIRONMENTAL CONTROL SYSTEM WEIGHT COMPARISON

ENVIRONMENTAL CONTROL SYSTEM	MERCURY C.M. 142.13	GEMINI C.M. 339.56	GEMINI S.M. 369.51	NAA C.M. 417	APOLLO S.M. 588	TWO-MAN APOLLO C.M. 304.0	TWO-MAN APOLLO S.M. 376.0
SUIT PRESSURE AND BREATHING	(69.06)	(158.32)	(153.79)	248	203	(120.7)	(150.3)
O_2	4.00		106.00				103.1
O_2 BOTTLE	9.21		38.92				38.3
O_2 VALVING	5.68		8.87				8.9
COMPRESSORS	3.80	9.40				9.4	
CO_2 ABSORBER	13.13	108.60				71.0	
VALVING, DUCTING	33.24	40.32				40.3	
EMERGENCY O_2	(15.75)	(61.08)		29		(73.2)	
O_2	4.00	14.00				20.0	
O_2 BOTTLE	9.21	42.68				48.8	
VALVING, INSTALLATION	2.54	4.40				4.4	
EQUIPMENT COOLING	(17.18)	(92.81)	(180.87)	89	127	(82.8)	(180.8)
COMPRESSOR	.74	14.50	61.64			14.5	61.6
HEAT EXCHANGER	2.20	14.10	23.60			14.1	23.6
VALVING, INSTALLATION	3.88		18.12				18.1
COOLING FLUID		15.76	38.31				38.3
COLD PLATES	10.36	32.45	9.20			38.0	9.2
COOLING LINES		16.00	30.00			16.2	30.0
WATER SYSTEM			7.00	28			
AIR LOCK PROV.				4			17.0
POST LANDING VENT SYSTEM	11.88	3.17				3.2	
RADIATORS					228		
CABIN PRESSURE RELIEF	5.05	3.34				3.3	
MOUNTING	17.59	10.84	23.85	12	10	10.8	23.9
CIRCUITRY	5.62	10.00	4.00	7	20	10.0	4.0

TABLE 4-17

CREW SYSTEM WEIGHT COMPARISON

CREW SYSTEM	MERCURY C.M. 326.64	GEMINI C.M. 929.09	NAA APOLLO C.M. 1499	TWO-MAN APOLLO C.M. 905.2
ASTRONAUT	180.00	360.00	528	352.0
BALLAST PROVISIONS	1.20	3.00		
PRESSURE SUIT	24.43	58.60	90	60.0
SURVIVAL KIT - COLLECTIVE			53	
SURVIVAL KIT - INDIVIDUAL	29.37	53.00	42	53.0
BACK PACKS			60	60.0
BACK PACK RECHARGE				22.0
PERSONAL CHUTES		46.00	60	40.0
EGRESS KIT		26.40		
SEAT INCL. RESTRAINT AND PADS	70.10	158.40	300	158.4
SEAT CATAPULT		108.66		
FIRST AID PACK		10.00	14	4.0
FOOD - DRY		39.50	92	27.5
WATER SYSTEM		20.44	(56)	(47.4)
WATER	8.25		54	40.0
CONTAINER			2	7.4
RELIEF PROVISIONS		8.00	29	5.0
WASTE MANAGEMENT			2	
PERSONAL HYGIENE		18.60	26	18.6
NUCLEAR INDICATOR		7.00	40	7.0
IN-FLIGHT MAINTAIN DISPLAY			15	15.0
BIOMEDICAL INSTRUMENTATION	9.47	6.00	5	2.3
RECREATION AND EXERCISE			25	
PERSONAL COMMUNICATIONS		1.69	9	6.0
LIGHTING	3.82	3.80	4	4.0
OVERSHOES			2	1.0
CONSTANT WEAR GARMENT			9	
SOLAR RADIATION GARMENT			7	7.0
SLEEPING AND PRIVACY			10	1.0
LAP BOOK, BOARD, MAPS AND CASES			8	1.0
EXTRA-VEHICULAR ACCESSORIES			12	12.0
ELECTRICAL PROVISIONS			1	1.0

4.8.2.10 Support Structure - Service Module Equipment - The service module equipment group includes all the non-propulsion systems and their mounting structure. The sidewalls, separation ring and propulsion systems are carried as service module weights.

Item	Weight-Pounds
Module Supports	27.3
Structural Mounting Provisions	5.0
Radiator Installation Parts	7.7
Electrical, E.C.S. Umbilical and Connections	4.1
Fairings	9.5
Service Module Equipment Support Structure	(53.6)

4.8.2.11 Miscellaneous - Scientific equipment of 250 pounds is located in the terminal landing module at launch.

The launch escape system is based on McDonnell Apollo studies and extrapolations of the Mercury tower. Ballast to obtain satisfactory aerodynamic stability is included.

The landing gear fairing weight is based on an average unit weight of structure of 1.4 pounds per square foot plus miscellaneous attachments and jettison systems.

4.8.3 Propulsion Module Weights - Results of the weight derivations for the propulsion modules structure and equipment are summarized in Table 4-18. Structural weights are derived from both estimating procedures and detail calculation.

Sidewall and support structure comparisons are shown in Figure 4-35 and Figure 4-36 respectively. These curves are based on data and experience from Mercury, Gemini and Apollo. Detail sidewall weight calculations are shown in Table 4-19. The results are plotted on Figure 4-35 and agree closely with the correlation curves. Support structure calculations are given in Tables 4-20, 4-21, 4-22 and results are plotted on Figure 4-36. Since the landing module structure serves both as support structure and landing gear carry-through, two points are shown. The upper point represents the total weight of support plus carry-through structure while the lower

4.8.3 (Continued)

TABLE 4-18

PROPULSION MODULE WEIGHT SUMMARY

	SERVICE MODULE	TERMINAL LANDING MODULE	RETROGRADE MODULE
STRUCTURE	(1079)	(724)	(2364)
SIDEWALLS	389	556	1403
SUPPORT STRUCTURE	489	95	797
FAIRING/SEPARATION DEVICES	201	73	164
PROPULSION SYSTEM	(20482)	(3124)	(51309)
PROPELLANT (USABLE)	18996	2642	47540
PROPELLANT (TRAPPED)	192	26	480
PROPELLANT (TANKAGE)	271	40	2375
PRESSURANT TANKAGE	498	66	86
PRESSURANT GAS	67	9	172
THRUSTER	373	266	370
PLUMBING AND CIRCUITRY	85	75	286
REACTION CONTROL AND ULLAGE SYSTEM	(639)	(1005)	
PROPELLANT (USABLE)	495	738	
PROPELLANT (TRAPPED)	6	8	
PROPELLANT TANKAGE	17	21	
PRESSURANT TANKAGE	20	30	
PRESSURANT GAS	3	4	
THRUSTERS	64	160	
PLUMBING, CIRCUITRY, MISC.	34	44	
EQUIPMENT	(80)	(522)	(114)
PROPELLANT GAGING	12	8	29
DESTRUCT SYSTEM		10	10
INSTRUMENTATION, ELECTRICAL, MISC.	68	113	75
SCIENTIFIC EQUIPMENT AND CONTAINER		284	
LANDING RADAR AND ANTENNA		34	
TRACKING RADAR		55	
ANTENNA WIRING AND MOUNTS		18	
LANDING GEAR (INCLUDING CARRY-THROUGH)		(740)	
TOTAL MODULE WEIGHTS AT LAUNCH	(22280)	(6115)	(53787)

point represents only that support structure which is independent of landing gear loads. A similar condition is noted on the landing gear correlation curve, Figure 4-37. If the total gear and carry-through plus support structure weight from the correlation curve is compared with the total calculated weight for these same items it is noted that the calculated values are about 200 pounds heavier. This is due to conservative initial analyses and much of this 200 pounds can be eliminated through structural optimization and detail design. However, the conservative, calculated value has been used for all aspects of this study.

The landing gear correlation curve, Figure 4-37, is based on Surveyor, Prospector, and Apollo data. Calculated values are shown in Table 4-23. The calculated landing gear weight of 740 pounds includes all major trusswork in the landing module since this structure is designed by landing gear loads.

SIDEWALL WEIGHT ESTIMATION CRITERIA

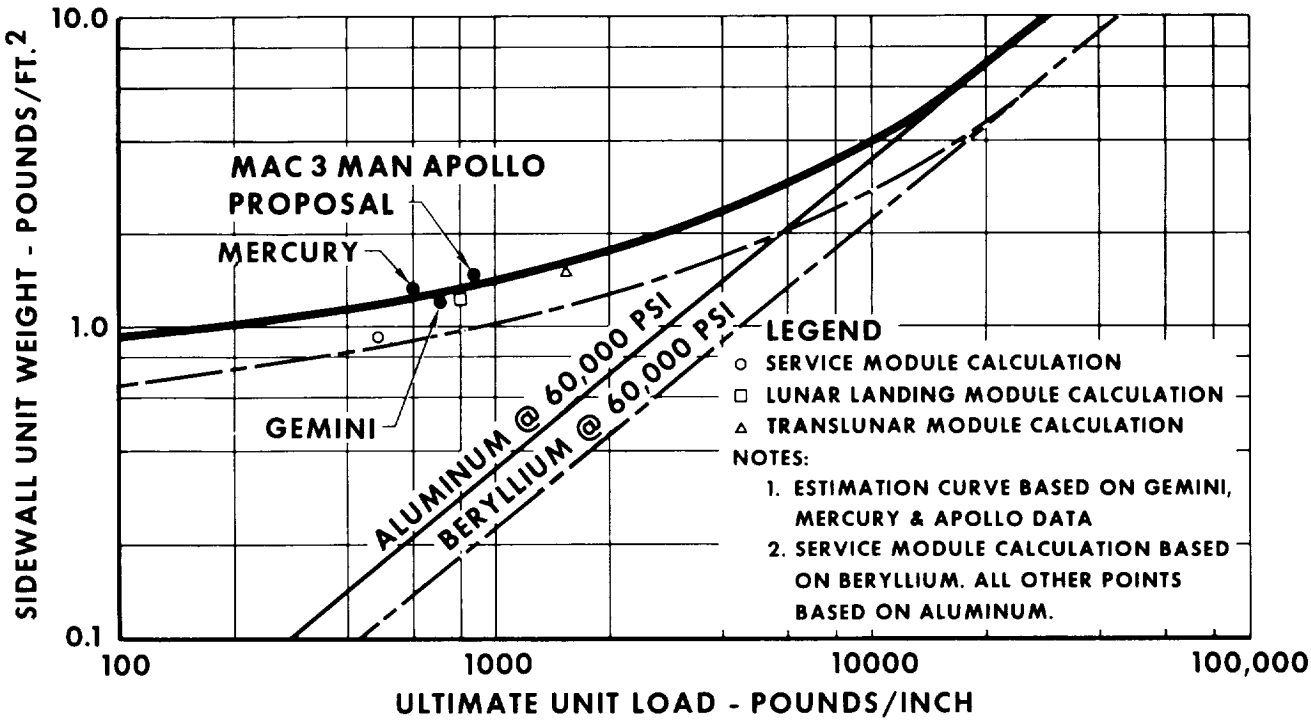


FIGURE 4-35

SUPPORT STRUCTURE ESTIMATION CRITERIA

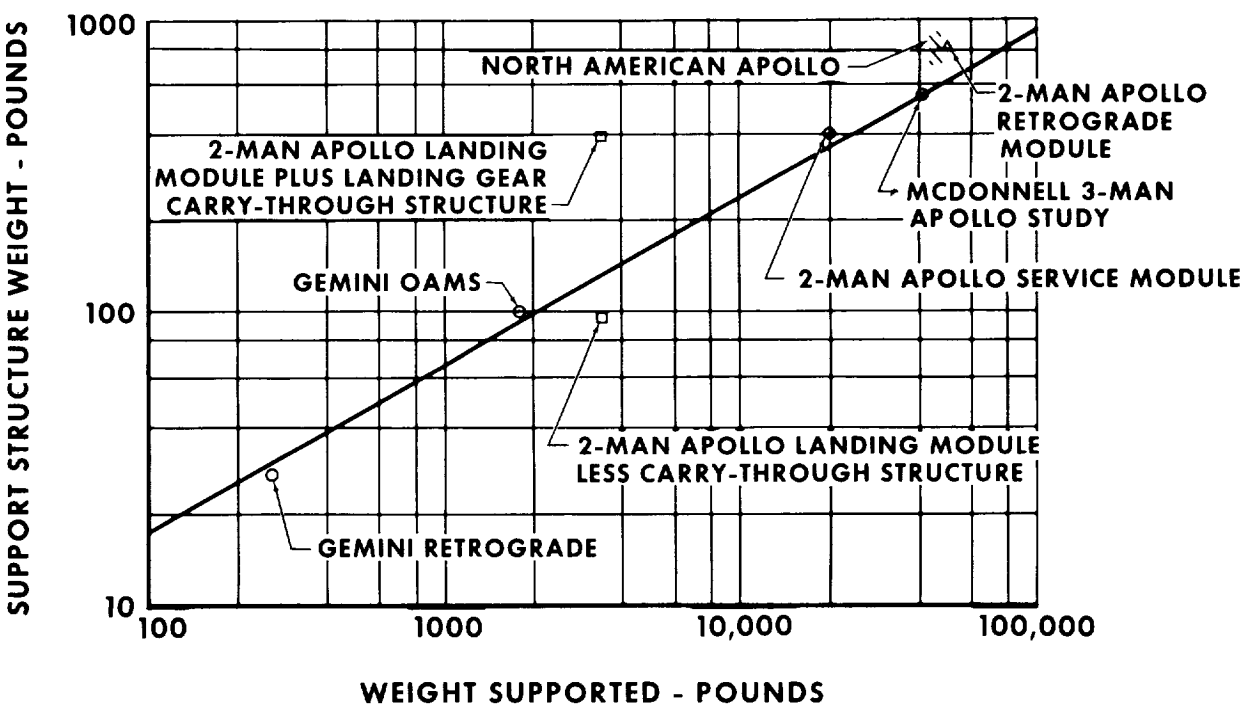


FIGURE 4-36

TABLE 4-19

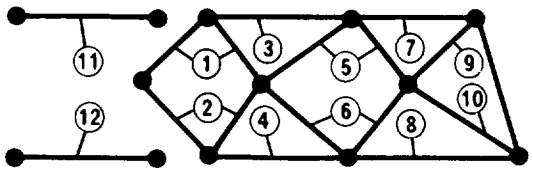
SIDEWALL STRUCTURE CALCULATIONS

ITEM	CRITERIA	WEIGHT
SERVICE MODULE		
SKIN	.020 BERYLLIUM X 404.5 SQUARE FT.	75.6
STRINGERS	72 CONTINUOUS, 72 HALF-WAY UP, 1.65 X 1.3 X .020 BERYLLIUM ANGLES	43.2
STRINGER FLANGES	43.2 POUNDS X 5 PERCENT	2.2
RADIATOR TUBES	1 INCH X .04 ALUMINUM FLANGE, .3 INCH DIAMETER, .040 ALUMINUM TUBE OVER 163 FT.^{2}	40.6
RINGS TYPICAL (4 REQ'D)	1 X 1 X .025 BERYLLIUM ANGLE AND 4 X 1 X .025 BERYLLIUM ANGLE	48.4
RING, TRUSS MTG., UPPER	.877 IN.^{2} TITANIUM "I" SECTION X 175 IN. DIAMETER = 78.8 LB. BUT 66.7 LB. IS CHARGED TO SUPPORT STRUCTURE	12.1
RING, TRUSS MTG., MIDDLE	1 X 4 X .060 TITANIUM CHANNEL X 195 IN. DIAMETER = 36.1 LB. BUT 24.0 LB. IS CHARGED TO SUPPORT STRUCTURE	12.1
RING, TRUSS MTG., LOWER	.798 IN.^{2} TITANIUM "I" SECTION X 210 IN. DIAMETER = 86.2 LB. BUT 74.1 LB. IS CHARGED TO SUPPORT STRUCTURE	12.1
BLAST FLANGE	4 X .08 BERYLLIUM PLATE X 230 IN. DIAMETER	15.1
FAIRING ATTACHMENT	ESTIMATE	20.0
TENSION LUGS	ONE PER 18 INCHES X 230 IN. DIAMETER X .071 LB. EACH	3.0
ACCESS DOORS	EQUIVALENT OF SEVEN 42" X 30" DOORS AT 5.0 LB. EACH	35.0
CONTINGENCY	15%	69.6
	TOTAL SERVICE MODULE SIDEWALLS WEIGHT/SURFACE AREA = .96 LB./FT.^{2}	(389.0)
LANDING MODULE		
SKIN	.010 TITANIUM X 428 SQUARE FT.	101.3
CORRUGATIONS	570 CORRUGATIONS, 1.0 X .5 X .010 TITANIUM X 80 INCHES LONG WITH EVERY THIRD CORRUGATION REMOVED IN UPPER HALF OF MODULE	225.4
RINGS TYPICAL (4 REQ'D.)	1 X 4 X .025 TITANIUM CHANNEL X 3030 INCHES	74.6
RING, CARRY THROUGH, UPPER	.48 IN.^{2} TITANIUM "I" SECTION X 230 IN. DIAMETER = 57 LB. BUT 38.3 LB. IS CHARGED TO SUPPORT STRUCTURE	18.7
RING, CARRY THROUGH, LOWER	.183 IN. TITANIUM SECTION X 258 IN. DIAMETER = 24.3 LB. BUT 5.6 LB. IS CHARGED TO SUPPORT STRUCTURE	18.7
TENSION LUGS	3 POUNDS AT UPPER RING AND 5 POUNDS AT LOWER	8.0
ACCESS DOORS	EQUIVALENT OF FOUR 42" X 30" DOORS AT 4.5 LB. EACH	18.0
FAIRING ATTACHMENT	ESTIMATE	15.0
CONTINGENCY		76.2
	TOTAL LANDING MODULE SIDEWALLS WEIGHT/SURFACE AREA = 1.3 LB./FT.^{2}	(555.9)
RETROGRADE MODULE		
SKIN	UPPER HALF .010 TITANIUM, LOWER HALF .016 TITANIUM, X 907 SQUARE FEET	278.5
CORRUGATIONS	550 CORRUGATION, 1.0 X .5 X .010 TITANIUM IN UPPER HALF AND 1.0 X .5 X .016 TITANIUM IN LOWER	653.0
RINGS TYPICAL (8 REQ'D.)	1 X 4 X .025 TITANIUM CHANNEL X 258 IN. DIAMETER	159.5
LOWER TANK PICKUP RING	.30 IN.^{2} TITANIUM SECTION X 258 IN. DIAMETER = 39.9 LB. BUT 19.9 LB. IS CHARGED TO SUPPORT STRUCTURE	20.0
UPPER AND LOWER END RINGS	.45 IN.^{2} TITANIUM SECTIONS X 250 IN. DIAMETER (2) = 119.8 LB. BUT 79.8 LB. IS CHARGED TO SUPPORT STRUCTURE	40.0
TENSION LUGS	5 POUNDS AT UPPER RING AND 10 POUNDS AT LOWER	15.0
ACCESS DOORS	EQUIVALENT OF SIX 42" X 30" DOORS AT 6.8 LB.	40.8
CONTINGENCY		196.0
	TOTAL RETROGRADE MODULE SIDEWALLS WEIGHT/SURFACE AREA = 1.55 LB./FT.^{2}	(1402.8)

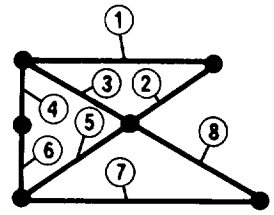
TABLE 4-20

SERVICE MODULE

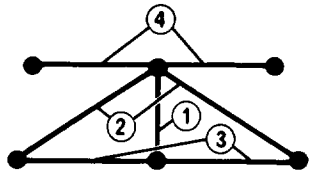
SUPPORT STRUCTURE



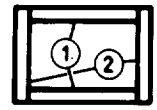
UNIT "A" – 4 REQD.



UNIT "B" 2 REQD.



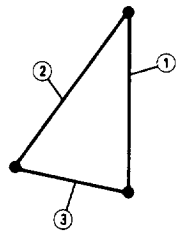
MOTOR MOUNTING TRUSS



MOTOR MOUNTING COLLAR

ITEM		CRITERIA	WEIGHT
UNIT A – (4 REQUIRED)	1	2.00 x .035 ALUMINUM TUBE x 20.5" LONG (8)	3.5
	2	2.75 x .035 ALUMINUM TUBE x 19.4" LONG (8)	4.6
	3	5.00 x .063 ALUMINUM TUBE x 32·9" LONG (4)	13.0
	4	3.25 x .049 ALUMINUM TUBE x 33.2" LONG (4)	6.6
	5	3.50 x .042 ALUMINUM TUBE x 23.5" LONG (8)	8.6
	6	1.75 x .028 ALUMINUM TUBE x 23.5" LONG (8)	2.9
	7	3.50 x .049 ALUMINUM TUBE x 40.0" LONG (4)	17.0
	8	3.5 x .049 ALUMINUM TUBE x 55.7" LONG (4)	11.8
	9	3.0 x .035 ALUMINUM TUBE x 25" LONG (4)	6.6
	10	4.5 x .058 ALUMINUM TUBE x 39.5" LONG (4)	25.8
	11	5.0 x .058 ALUMINUM TUBE x 28" LONG (2)	5.1
	12	4.0 x .058 ALUMINUM TUBE x 28" LONG (2)	4.0
UNIT B – (2 REQUIRED)	1	2.5 x .040 ALUMINUM TUBE x 62" LONG (2)	3.8
	2	1.25 x .028 ALUMINUM TUBE x 38" LONG (2)	.8
	3	1.00 x .028 ALUMINUM TUBE x 31" LONG (2)	.5
	4	1.00 x .028 ALUMINUM TUBE x 14" LONG (2)	.2
	5	1.50 x .042 ALUMINUM TUBE x 31" LONG (2)	1.2
	6	.75 x .022 ALUMINUM TUBE x 14" LONG (2)	.1
	7	1.00 x .022 ALUMINUM TUBE x 78" LONG (2)	1.1
	8	2.50 x .040 ALUMINUM TUBE x 53" LONG (2)	3.3
MOTOR MOUNTING TRUSS	1	1.65 x .035 ALUMINUM TUBE x 14" LONG (2)	.5
	2	1.55 x .028 ALUMINUM TUBE x 30" LONG (4)	1.6
	3	1.75 x .035 ALUMINUM TUBE x 27" LONG (4)	2.0
	4	1.125 x .028 ALUMINUM TUBE x 27" LONG (4)	1.0
MOTOR MOUNTING COLLAR	1	TWO CHANNELS, .865 IN.^{2} AREA x 36" LONG ALUMINUM	6.2
	2	TWO "I" SECTION, .834 IN.^{2} AREA x 36" LONG ALUMINUM	6.0
TANK STRAPS		4" x .032 TITANIUM CIRCUMFERENCIAL STRAPS FOR 16 TANKS	35.2
TANK BRACE		2.3 POUNDS PER BRACE x EQUIVALENT OF 10 REQD.	23.0
JOINT REINFORCING		1.5 POUNDS PER JOINT x EQUIVALENT OF 42 REQD.	63.0
SUPPORT RINGS		66.7 LB. + 24.0 LB. + 74.1 LB. OF RINGS CHARGED TO SUPPORT STRUCTURE	164.8
CONTINGENCY			65.0
		TOTAL SERVICE MODULE SUPPORT STRUCTURE	(488.8)

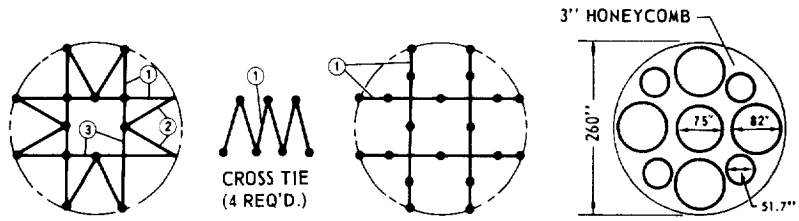
TABLE 4-21
TERMINAL LANDING MODULE
SUPPORT STRUCTURE



MOTOR MOUNTING TRUSS
4 REQUIRED

ITEM	CRITERIA	WEIGHT
MOTOR MOUNTING TRUSS ①	1.25 X .028 ALUMINUM TUBE X 82.7" LONG (4)	3.6
②	3.0 X .042 ALUMINUM TUBE X 72.3" LONG (4)	11.3
③	1.25 X .028 ALUMINUM TUBE X 52.1" LONG (4)	2.2
TANK STRAPS	4" X .032 TITANIUM CIRCUMFERENTIAL STRAPS FOR 9 TANKS	14.7
TANK BRACE	2.3 POUNDS EACH X EQUIVALENT OF FOUR REQ'D.	9.2
SUPPORT RINGS	38.3 LB. PLUS 5.6 LB. OF RINGS CHARGED TO SUPPORT STRUCTURE	43.9
CONTINGENCY		10.0
	TOTAL LANDING MODULE SUPPORT STRUCTURE*	(94.9)
	*MAJOR TRUSS WORK IS DESIGNED BY LANDING GEAR LOADS AND IS CHARGED TO LANDING GEAR	

TABLE 4-22
RETROGRADE MODULE
SUPPORT STRUCTURE



INTERIM BULKHEAD — LOWER BULKHEAD — UPPER BULKHEAD

ITEM	CRITERIA	WEIGHT
INTERIM BULKHEAD ①	4" X .0464" ALUMINUM TUBE X 79" LONG (8)	32.4
②	4" X .067" ALUMINUM TUBE X 90" LONG (8)	59.4
③	4" X .056" ALUMINUM TUBE X 86" LONG (4)	23.8
CROSS TIE ①	3.25 X .040 ALUMINUM TUBE X 62" LONG (24)	60.0
LOWER BULKHEAD ①	2.00 X .040 ALUMINUM TUBE X 244" LONG (4)	24.0
UPPER BULKHEAD	2.3 LB./FT.^{3} ALUMINUM HONEYCOMB WITH .032 ALUMINUM SIDE PANELS AND 3 X .5 X .063 ALUMINUM CHANNEL EDGING AROUND CUTOUTS AND PERIMETER	263.1
THRUSTER SUPPORT	FOUR 3 X .040 X 67" AND EIGHT 1.5 X .040 X 60" ALUMINUM TUBES	17.8
OXYGEN TANK INTERCOSTALS	4" X 100" X .5" ALUMINUM HONEYCOMB (4)	8.0
HYDROGEN TANK INTERCOSTALS	4" X 112" X .25" ALUMINUM HONEYCOMB (4)	4.4
OXYGEN TANK INTERNAL PRESSURANT SUPPORTS	4 TANKS AT 3 LB. PER TANK	12.0
PRESSURANT TANK STRAPS	4" X .032 TITANIUM STRAPS	4.5
JOINT REINFORCING	EQUIVALENT OF 32 JOINTS AT 1.5 LB. EACH	48.0
SUPPORT RINGS	19.9 LB. PLUS 79.8 LB. OF RINGS CHARGED TO SUPPORT STRUCTURE	99.7
CONTINGENCY	25 PERCENT	139.3
	TOTAL LUNAR RETROGRADE MODULE SUPPORT STRUCTURE	(796.4)

SPACECRAFT LANDING GEAR WEIGHT

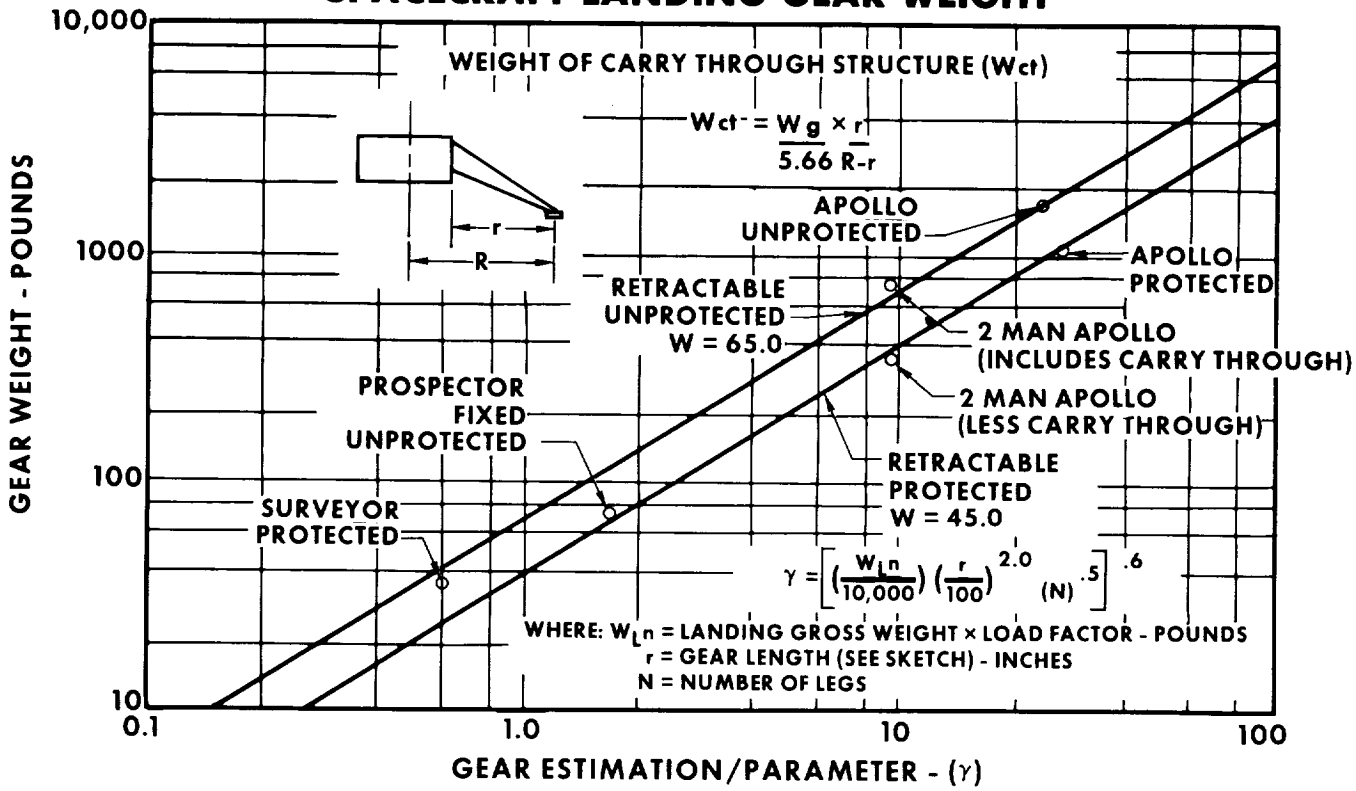
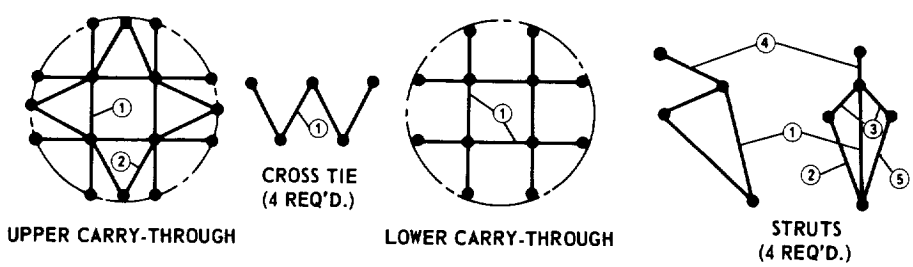


FIGURE 4-37

TABLE 4-23

LANDING GEAR WEIGHT CALCULATIONS



ITEM		CRITERIA		WEIGHT
UPPER CARRY-THROUGH	1	5.0 X .070 ALUMINUM TUBE X 215" LONG	(4)	94.4
	2	3.875 X .060 ALUMINUM TUBE X 80" LONG	(8)	46.4
CROSS TIE	1	3.50 X .050 ALUMINUM TUBE X 93" LONG	(16)	81.6
LOWER CARRY-THROUGH	1	4.25 X .060 ALUMINUM TUBE X 244" LONG	(4)	78.0
STRUTS (4)	1	6.0 X .080 ALUMINUM TUBE X 120" LONG	(4)	77.2
		7.9 LB./FT.^{3} CRUSHABLE HONEYCOMB	(4)	14.0
	2	5.0 X .060 ALUMINUM TUBE X 126" LONG	(4)	50.8
		7.9 LB./FT.^{3} CRUSHABLE HONEYCOMB	(4)	10.4
	3	2.0 X .025 ALUMINUM TUBE X 148" LONG	(8)	9.2
	4	6.0 X .070 ALUMINUM TUBE X 73" LONG	(4)	38.4
	5	5.0 X .060 ALUMINUM TUBE X 126" LONG	(4)	50.8
		7.9 LB./FT.^{3} CRUSHABLE HONEYCOMB	(4)	10.4
GEAR FOOT		4 REQUIRED AT 5 POUNDS EACH		20.0
SKIN BEEF-UP		15 INCH X .010 TITANIUM OVER 6 CORRUGATIONS (8 PLACES)		5.0
LANDING GEAR FITTINGS		12 REQUIRED AT 1 POUND EACH		12.0
JOINT REINFORCING		2.0 POUNDS PER JOINT X EQUIVALENT OF 22 JOINTS		44.0
CONTINGENCY				97.0
		TOTAL LANDING GEAR		(739.6)

4.8.3 (Continued)

Detail calculations for the service-module-to-command-module fairing and for the shaped-charge separation devices used in the various modules are contained in Table 4-24. The separation devices are based on Gemini designs plus a 20 percent contingency.

Criteria for the derivation of propellant weights are discussed in the applicable sections under propulsion systems. Rocket engine weights are based on vendor estimates.

Tables 4-25, 4-26, and 4-27 contain propellant tank and pressurant tank calculations. The liquid oxygen tanks in the retrograde module are aluminum; all other propellant and pressurant tanks are titanium. The retrograde module propellant tank weight is based on 30 pounds per square inch operating pressure with a 1.5 burst factor. The landing and service module main propellant tanks are based on 200 pounds per square inch operating pressure and a burst factor of 1.5. Reaction control and ullage positioning system propellant tanks are based on 300 pounds per square inch operating pressure and 1.5 burst factor. All pressurant tanks are based on 3000 pounds per square inch operating pressures and a burst factor of 2.0.

Propulsion system plumbing and circuitry weights are based on detail considerations of size and quantity of lines, fittings, squib valves, ground connections, propellant shut-off valves and wiring.

Equipment item weights are, in most cases, based on comparisons with known equipment. Equipment lists for various existing boosters have been reviewed in order to estimate probable equipment requirements in addition to those that are known or specified. Based on past experience, allowances for non-optimum material, omission of some small parts and routine complications are included in the detail calculations. The retrograde module tankage weight includes a contingency of 824 pounds. Propulsion module equipment weight includes a 50% contingency. These large allowances are included to insure 80-90% probability of meeting or bettering total spacecraft weight estimates.

TABLE 4-24

FAIRING/SEPARATION DEVICE CALCULATIONS

ITEM	CRITERIA	WEIGHT
FAIRING, COMMAND MODULE TO SERVICE MODULE		
SKIN AND CORRUGATIONS	.010 TITANIUM SKIN AND CORRUGATIONS, 39.4 FT.^{2} AT .76 POUNDS/FT.^{2}	30.0
UPPER RING	.24 IN.^{2} STEEL x 116 IN. DIAMETER	24.7
FAIRING	.12 IN.^{2} FIBERGLASS x 116 IN. DIAMETER	2.8
LOWER RING	.57 IN.^{2} TITANIUM x 131 IN. DIAMETER	37.8
SEPARATION RING	.42 IN.^{2} TITANIUM x 131 IN. DIAMETER	27.8
LOWER CLAMP RING	.08 IN.^{2} STEEL x 133 IN. DIAMETER	9.5
FAIRING	.15 IN.^{2} FIBERGLASS x 133 IN. DIAMETER	3.0
EXPLOSIVE BOLTS & FTGS.		8.0
SHIELDS, CABLES, MISC.		57.4
	TOTAL FAIRING WEIGHT	(201.0)
LANDING MODULE SEPARATION DEVICE	232 IN. DIAMETER SHAPE CHARGE AT 0.1 LB./IN.	(73.0)
RETROGRADE MODULE SEPARATION DEVICE (2 REQD.)	260 IN. DIAMETER SHAPE CHARGE AT 0.1 LB./IN., ONE AT TOP AND ONE AT BOTTOM OF MODULE	(164.0)

TABLE 4-25

SERVICE MODULE

TANKAGE CALCULATIONS

ITEM	CRITERIA	WEIGHT
MAIN PROPELLANT TANKS	ALL TANKS ARE SPHERICAL, MATERIAL IS TITANIUM	
OXIDIZER	3 REQUIRED, 57" DIAMETER, 200 PSI OPERATING PRESSURE, 1.5 BURST FACTOR	130.0
FUEL	3 REQUIRED, 50" DIAMETER, 200 PSI OPERATING PRESSURE, 1.5 BURST FACTOR	96.0
CONTINGENCY	20%	45.0
	TOTAL SERVICE MODULE PROPELLANT TANKAGE	(271.0)
MAIN PRESSURANT		
MAIN PRESSURANT	3 REQUIRED, 33" DIAMETER, 300 PSI OPERATING PRESSURE, 2.0 BURST FACTOR	480.0
CONTINGENCY		18.0
	TOTAL SERVICE MODULE PRESSURANT TANKAGE	(498.0)
R.C.S. PROPELLANT		
OXIDIZER	2 REQUIRED, 19" DIAMETER, 300 PSI OPERATING PRESSURE, 1.5 BURST FACTOR	9.0
FUEL	2 REQUIRED, 17.4" DIAMETER 300 PSI OPERATING PRESSURE, 1.5 BURST FACTOR	8.0
	TOTAL R.C.S. PROPELLANT TANKAGE	(17.0)
R.C.S. PROPELLANT	2 REQUIRED, 13" DIAMETER, 3000 PSI OPERATING PRESSURE, 2.0 BURST FACTOR	20.0
	TOTAL R.C.S. PRESSURANT TANKAGE	(20.0)

TABLE 4-26

TERMINAL LANDING MODULE

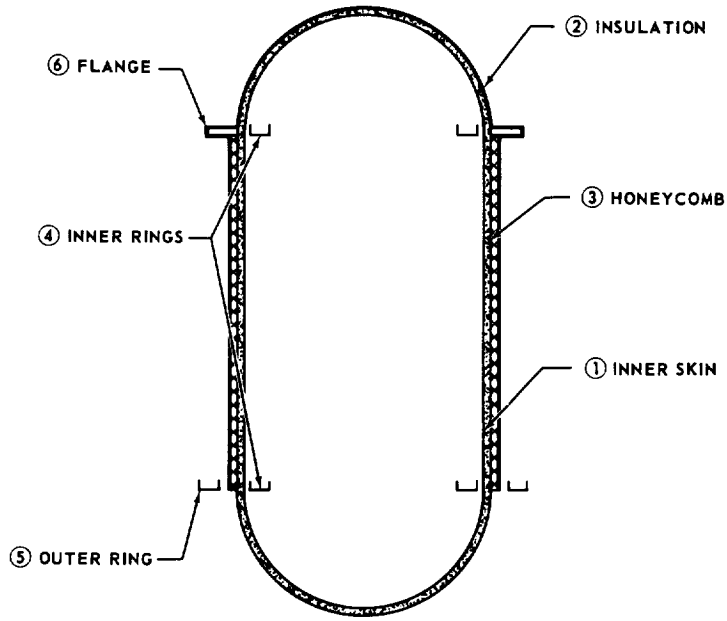
TANKAGE CALCULATIONS

ITEM	CRITERIA	WEIGHT
MAIN PROPELLANT TANKS		
OXIDIZER	1 REQ'D. 42" DIA., 200 PSI OPERATING PRESSURE 1.5 BURST FACTOR	19
FUEL	1 REQ'D. 38" DIA., 200 PSI OPERATING PRESSURE 1.5 BURST FACTOR	14
CONTINGENCY	20 PERCENT	7
	TOTAL MAIN PROPELLANT TANKAGE	(40.0)
MAIN PRESSURANT TANKAGE		
MAIN PRESSURANT	1 REQ'D., 24.4" DIA., 3000 PSI OPERATING PRESSURE, 2.0 BURST FACTOR	64.0
CONTINGENCY		2.0
	TOTAL MAIN PRESSURANT TANKAGE	(66.0)
RCS PROPELLANT		
OXIDIZER	2 REQ'D., 22" DIA., 300 PSI OPERATING PRESSURE 1.5 BURST FACTOR	11.0
FUEL	2 REQ'D., 20" DIA., 300 PSI OPERATING PRESSURE 1.5 BURST FACTOR	10.0
	TOTAL RCS PROPELLANT TANKAGE	(21.0)
RCS PRESSURANT	2 REQ'D., 15" DIA., 3000 PSI OPERATING PRESSURE, 2.0 BURST FACTOR	30.0
	TOTAL RCS PRESSURANT TANKAGE	(30.0)

TABLE 4-27

RETROGRADE MODULE

TANKAGE CALCULATIONS



TYPICAL CRYOGENIC TANK

CRITERIA		ITEM	WEIGHT
HYDROGEN TANK (4 REQD.)			
INNER SKIN	①	.010 UPPER DOME, .010 LOWER DOME, .010 SIDES (TITANIUM) x (4)	322.4
INSULATION	②	341 FT.^{2} SURFACE x .5 LB./FT.^{3} INSULATION, 1/4 INCH THICK PLUS .04 LB./FT.^{2} FOR GLUE x (4)	68.8
HONEYCOMB SLEEVE	③	2.3 LB./FT.^{3} ALUMINUM HONEYCOMB WITH .020 SIDE PANELS (4)	525.6
INNER RINGS	④	TWO 1 x 3 x .020 TITANIUM CHANNELS x (4)	32.0
OUTER RINGS	⑤	ONE 1 x 3.5 x .040 ALUMINUM CHANNEL x (4)	24.0
FLANGE	⑥	1.5 INCH x .020 ALUMINUM x 84.6" DIAMETER x (4)	3.2
OXYGEN TANK (4 REQD)			
INNER SKIN	①	.010 UPPER DOME, .020 LOWER DOME, .02–.04 TAPERED SIDES (ALUMINUM) x 4	236.0
INSULATION	②	NOT REQUIRED	
HONEYCOMB SLEEVE	③	2.3 LB./FT.^{3} ALUMINUM HONEYCOMB WITH .020 SIDE PANELS (4)	290.0
INNER RINGS	④	TWO 1 x 3.5 x .040 ALUMINUM CHANNELS x (4)	25.6
OUTER RINGS	⑤	ONE 1 x 4 x .050 ALUMINUM CHANNEL x (4)	21.2
FLANGE	⑥	1.5 INCH x .020 ALUMINUM x 53.8" DIAMETER x (4)	2.0
CONTINGENCY			824.2
		TOTAL PROPELLANT TANKAGE	(2375.0)
PRESSURANT TANKAGE (4 REQD.)		4 REQD., 17" DIA., 3000 PSI OPERATING PRESSURE, 2.0 BURST FACTOR + 3 POUND CONTINGENCY	(86.0)

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