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APOLLO MASTER SPACECRAFT SPECIFICATION

(U)

31 OCTOBER 1963

NAS 9-150

Exhibit I Paragraph 4.2

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NORTH AMERICAN AVIATION, INC.
SPACE and INFORMATION SYSTEMS DIVISION

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FOREWARD

This specification represents North American Aviation, Inc., Space and Information Systems Division Engineering design definition below the level of contractor specification requirements in Exhibit B of Contract NAS 9-150. The material contained herein is subject to change at any time within the limits of Exhibit B or other provisions of Contract NAS 9-150.

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ILLUSTRATIONS

TABLES

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APOLLO MASTER SPACECRAFT SPECIFICATION

1. SCOPE

1.1 Scope. - This specification defines the following aspects of the Apollo Spacecraft (SC):

(a) Performance of the SC

(b) Composition and performance requirements of the Command Module (CM), the Service Module (SM), and the Adapter

(c) Performance characteristics of the major systems and modules

(d) Integration of the major systems and modules within the SC

(e) Interfaces between the modules of the SC.

NOTE: For the convenience of the reader, the abbreviations and their meanings used in this specification are repeated in section 6. Section 6 also contains a list of the definitions that apply to this specification.

1.1.1 Objective. - The ultimate objective of this project is to land men on the surface of the moon and return the men safely to earth. The objective of this document is to define the design approaches and operational techniques for transporting a three-man crew to the moon and returning them to earth.

2. APPLICABLE DOCUMENTS

2.1 Applicability. - The following documents form a part of this specification to the extent specified herein:

2.1.1 Government Documents. -

STANDARDS

Military

MIL-STD-130A 8 September 1962	Identification Marking of U. S. Military Property

[redacted]

MS 33586A 16 December 1958	Metals, Definitions of Dissimilar

PUBLICATIONS

National Aeronautics and Space Administration (NASA)

NCP 200-2 15 December 1961	Quality Program Provisions for Space System Contractors

Marshall Space Flight Center (MSFC)

10M01071 6 March 1961	Environment Protection When Using Electrical Equipment Within the Areas of Saturn Complexes Where Hazardous Areas Exist, Procedure for
MSFC-PROC-158 16 February 1962	Soldering Electrical Connections (High Reliability). Procedure for

2.1.2 Non Government Documents.

SPECIFICATIONS

North American Aviation, Inc., Space and Information Systems Division (NAA/S&ID)

SID 62-109	Preliminary Test Plan, Research and Development for Project Apollo, Volumes I, II, III, IV, V
SID 62-154	Apollo Quality Control Program Plan, Volumes I and II
SID 62-203	Apollo Reliability Program Plan
SID 62-1000	Preliminary Guidance and Navigation System Performance and Interface Requirements Specification
SID 62-1002	Preliminary Scientific Instrumentation Performance and Interface Requirements Specification
SID 62-1244	Preliminary Apollo Lunar Excursion Module Performance and Interface Specification
MC 999-0002B	Electromagnetic Interference Control for the Apollo Space System
MC 999-0007	Human Engineering Design Criteria for Spacecraft Systems
MC 999-0017A	Human Engineering Design Criteria for Spacecraft Systems, GSE

C[redacted]

2.2 Precedence. - The order of precedence in case of conflict shall be as follows:

- (a) The contract
- (b) This specification
- (c) Other documents referenced herein.

3. REQUIREMENTS

3.1 Spacecraft. - The SC shall be composed of separable modules so that:

- (a) Effective weight principles can be realized through proper jettisoning of expendable units
- (b) Performance benefits can be obtained by utilizing staging techniques.

The SC concept changes as a function of the particular mission phase. Figures 1 through 14 define the various SC configurations. The SC shall include modules and systems.

3.1.1 Modules. - The SC shall include the following modules:

- (a) CM
- (b) SM
- (c) Lunar Excursion Module (LEM)
- (d) Adapter.

3.1.2 Systems. - The SC shall include the following systems:

- (a) Guidance and Navigation System (G & N)
- (b) **Stabilization** and Control System (SCS)
- (c) Service Propulsion System (SPS)
- (d) Reaction Control System (RCS)
- (e) Launch Escape System (LES)

(f) Earth Landing System (ELS)

(g) Crew System

(h) Environmental Control System (ECS)

(i) Electrical Power System (EPS)

(j) Communication and Instrumentation System (C & I)

(k) In-flight Test System **(IFTS)**

(l) Controls and Displays.

3.2 Materials, Parts, and Processes. - Materials, parts, and processes shall be selected with the following considerations:

(a) Materials, parts, and processes shall be suitable for the purpose intended. Safety, performance, reliability, and maintainability of the item are of primary importance.

(b) Except in those instances where their use is essential, critical materials shall not be used.

(c) Where possible, materials and parts shall be of the kind and quality widely available in supply channels.

(d) When practicable, materials and parts shall be nonproprietary.

(e) When practicable, a choice among equally suitable materials and parts shall be provided.

(f) Whenever possible, single source items shall be avoided.

(g) When practicable, circuits shall be designed with a minimum of adjustable components.

3.2.1 Specifications and Standards. - Materials, parts, and processes shall be selected in the following order of preference, provided coverage is suitable:

(a) Federal specifications approved for use by the NASA

(b) Military specifications and standards (MIL, JAN, or MS)

(c) Other Governmental specifications

(d) Specifications released by nationally recognized associations, committees, and technical societies.

3.2.2 Choice of Standard Materials, Parts, and Processes. - Where applicable, preferred parts lists shall be used. When an applicable specification provides more than one grade, characteristic, or tolerance of a part or material, the standard parts, materials, and processes of the lowest grades, broadest characteristics, and greatest tolerances shall be chosen. However, standard parts, materials, or processes of high grades, narrow characteristics, or small tolerances may be used when necessary to avoid delay in development or production, obvious waste of materials, or unnecessary use of production facilities. The requirements specified for the use of standard parts, materials, or processes shall not relieve the contractor of the responsibility to comply with all performance and other requirements specified in the contract.

3.2.3 Nonstandard Parts, Materials, and Processes. - Nonstandard parts, materials, and processes may be used when necessary to facilitate the design of the particular equipment. However, when such nonstandard items are incorporated in the design, they shall be documented as required by the contract.

3.2.3.1 New Parts, Materials, and Processes. - New parts, materials, or processes developed under the contract may be used, provided they are suitable for the purpose intended. Any new parts, materials, or processes used shall be documented as required by the contract.

3.2.4 Miniaturization. - Miniaturization shall be accomplished to the greatest extent practicable, commensurate with required functions and performance of the system. Miniaturization shall be achieved by use of the smallest possible parts and by compact arrangement of the parts in assemblies. Miniaturization shall not be achieved by means that would sacrifice the reliability or performance of the equipment.

3.2.5 Flammable Materials. - Materials that may support combustion or are capable of causing an explosion shall not be used in areas where the environments or conditions are such that combustion would take place.

3.2.6 Toxic Materials. - Unless specific written approval is obtained from the NASA, materials that produce toxic effects or noxious substances when exposed to CM interior conditions shall not be used.

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3.2.7 Unstable Materials. - Where practicable, materials that are known to emit or deposit corrosive substances, induce corrosion, or produce electrical leakage paths within an assembly shall be avoided.

3.2.8 Fungus-Inert Materials. - Fungus-inert materials shall be used to the greatest extent practicable. Fungus-nutrient materials may be used if properly treated to prevent fungus growth for a period of time, dependent upon their use within the SC. When used, fungus-nutrient materials shall be hermetically sealed or treated for fungus and shall not adversely affect equipment performance or service life.

3.2.9 Metals. - All metals shall be of corrosive-resistant type or shall be suitably protected to resist corrosion during normal service life. Gold, silver, platinum, nickel, chromium, rhodium, palladium, titanium, cobalt, corrosion-resistant steel, tin, lead-tin alloys, tin alloys, Alclad aluminum, or sufficiently thick platings of these metals may be used without additional protection or treatment.

3.2.9.1 Dissimilar Metals. - Unless suitably protected or coated to prevent electrolytic corrosion, dissimilar metals, as defined in Standard MS 33586, shall not be used in intimate contact.

3.2.9.2 Electrical Conductivity. - Materials used in electronics or electrical connections shall have such characteristics that, during specified environmental conditions, there shall be no adverse effect upon the conductivity of the connections.

3.2.10 Lubricants. - Apollo SC lubricants and lubrication shall be compatible with the combined environments in which they are employed. Lubricant material and process specifications will be formulated to prescribe materials and describe application methods.

3.2.11 Special Tools. - The functional components of the SC and component attachments shall be designed so that the use of special tools for assembly, disassembly, installation, and service shall be kept to a minimum. The use of multipurpose tool LS 0287 shall be permitted.

3.3 Design and Construction. - The SC configuration shall be in accordance with Figures 15 and 16.

3.3.1 Modules. -

3.3.1.1 CM. - The CM shall be designed and constructed to sustain normal ground and flight loads, support maximum abort and landing impact loads, provide a mounting surface for all CM systems, provide a vessel for pressurization, decrease the flux density within the capsule due to radiation to the extent practicable by good design practice, provide protection against the damaging effects of meteoroids to the extent practicable by good design practice, provide crew living quarters during various mission phases, and provide thermal protection during the boost and entry phases of the flight. The CM shall be a blunt body developing a hypersonic lift-to-drag ratio (L/D) of approximiately 0.50. The lift vector shall be effectively modulated in hypersonic flight by roll control. The CM shall be designed for equipment location so that the CM center of gravity shall be optimized for the required trim conditions and maneuverability during the entry phase. Illumination of the CM shall be by flood and integral lighting. The gross takeoff weight design objective for the CM shall be 8,500 pounds. This weight shall not include any additional weight required for compatibility with the addition of the LEM. Figures 17, 18, and 19 illustrate CM external dimensions, including the ablative material, structure outline, and center of gravity location versus L/D ratio. CM reference axis shall be in accordance with Table I.

3.3.1.1.1 Entry. - The effects of venting lags in cabin pressure, temperature, and temperature gradients due to aerodynamic heating on material properties, thermal stresses, and deformation shall be considered simultaneously with applied external loads and internal pressures. The CM shall be designed for a limit load of 20 g during entry.

3.3.1.1.2 Earth Landing. - The terminal crew attenuation system of the CM shall be designed to be compatible with the use of parachutes. The landing system shall be designed to safely land the CM on land or water. Any further attenuation required by the crew shall be provided by energy-absorbing devices in the crew support and restraint systems. After landing, the landing system shall provide any necessary flotation, survival, and location aids.

3.3.1.1.3 Land Landing Impact. - Design objectives for land landing impact of the CM shall be as follows:

(a) Crew tolerances to impact accelerations and acceleration onset rates shall not be exceeded during the dynamics of ground contact (see Figure 20).

(b) The following criteria shall govern design of the CM:

  1. The maximum slope in any direction shall be 5 degrees, and the surface shall be smooth and plane.

  2. The coefficient of surface friction shall be 0.8 for design.

  3. The maximum terrain altitude shall be 2,000 feet above sea level.

(c) A maximum horizontal steady wind velocity of 15 knots shall be assumed at 10 feet above ground level to cover the possibility of prevailing winds.

3.3.1.1.4 Water Landing Impact. - Design objectives for water landing impact of the CM shall be as follows:

(a) Crew tolerances to impact accelerations and acceleration onset rates shall not be exceeded.

(b) CM water flotation and water stability characteristics shall be such that the CM shall recover from any initial attitude and shall float upright with normal egress hatches clear of the water. CM seakeeping capability shall provide for a 7-day flotation period for nominal landings. The CM primary structure shall be capable of flotation following the application of dynamic loads during contact.

(c) A 20-knot steady wind shall be assumed to occur at 10 feet above the mean surface level. (See 3.6.2.9.1)

(d) Thermal gradients resulting from sudden water immersion shall be accounted for in the structural design.

3.3.1.1.5 Inboard Profile. - The internal arrangement of the CM shall contain three compartments: crew, forward, and aft.

3.3.1.1.5.1 Crew Compartment. - The crew compartment shall be designed and constructed to house the crew and required systems. Equipment and storage space shall be contained in the following locations, as illustrated in Figures 21, 22, and 23.

(a) Lower equipment bay

(b) Right-hand equipment bay

(c) Left-hand equipment bay

(d) Aft equipment bay

(e) Main display panel

(f) Left-hand forward equipment bay

(g) Right-hand forward equipment bay.

3.3.1.1.5.2 Forward Compartment. - The forward compartment shall be located in an area forward of the crew compartment. This compartment shall be designed to house components of the ELS, RCS, and antennas for the C & I.

3.3.1.1.5.3 Aft Compartment. - The aft compartment shall be located on the aft periphery of the CM. This compartment shall be designed to house components of the RCS, ECS, separation system, and umbilical connections.

3.3.1.1.6 Side Access Hatch. - Normal prelaunch entrance to the CM for the crew shall be through a central hatch. The hatch shall not be obstructed at any stage of space vehicle countdown or flight.

3.3.1.1.7 Forward Access Hatch. - A forward hatch shall be provided to allow ingress and egress to the LEM while in the docked configuration without exposing the crew to the environment of space.

3.3.1.1.8 Windows. - The CM shall have windows to aid in providing maximum use of direct vision during rendezvous, earth landing, and scientfic observations, and for general orientation. These windows shall consist of multiple panes and shall be protected by covers during launch and entry. The covers shall be operated from inside the cabin by the crew members.

3.3.1.1.9 Attachments. - Provisions shall be incorporated into the CM design for attachment of ground handling equipment, the LES, and the SM.

3.3.1.1.10 CM Heat Shield. - The CM heat shield shall be comprised of three major components: the forward, crew, and aft compartment heat shields. The space between the heat shield and the pressure cabin shall be vented to the outside in a region of low pressure during entry. Heat shield materials shall be charring-type ablators. Maintenance and access doors for interior service and actuated covers over windows and guidance equipment shall be provided. Electrical umbilicals, coaxial cables, and hard line connections between the CM and SM penetrate the crew compartment heat shield and leave openings after separation. Two basic elements form the heat shield: ablative material and substructure. The substructure (or outer structure) shall be of brazed PH15-7 Mo or PH14-8 Mo stainless steel, honeycomb sandwich panel construction. The brazed panels shall be welded together to form the large major components. A fiberglass honeycomb matrix shall be bonded to the steel substructure and filled with the charring ablative material, Avcoat 5026-39, an epoxy resin with silica fibers and phenolic microballoons. The substructure-ablator combination shall sustain the applied loads and provide the required thermal protection for the Apollo CM.

3.3.1.1.11 CM Outer Structure. - The CM outer structure shall be the backup structure for the heat shield ablation material.

3.3.1.1.12 CM Inner Structure. - The CM inner structure shall be the primary structure of the CM and shall be the pressure cabin for the crewmen. The structural integrity of the pressure cabin shall not be violated as a result of normal water landings. The cabin shall have a factor of safety of 1.5.

3.3.1.2 SM. - The SM shall be designed and constructed to support body loads and provide a mounting structure for: SM systems; pressure vessels for the EPS and the ECS reactants; and the attachment of Deep Space Instrumentation Facility (DSIF) telecommunications equipment. Space radiators shall be an integral part of the SM outer structural shell. Radiators shall be built into sectors II and IV for the ECS and into sectors I and IV for the EPS. The SM shall be attached to the CM by an aerodynamic fairing. This fairing shall house the CM heat shield and all terminal equipment. SM reference axes shall be in accordance with Table I.

3.3.1.2.1 Propellant Weight. - The maximum usable propellant weight shall be 45,000 pounds. The design objective gross burnout weight is 11,000 pounds maximum.

3.3.1.2.2 Inboard Profile. - The SM internal arrangement shall contain six segments and a center section. (See Figures 24 and 25). Equipment contained in each sector shall be as follows:

(a) Sector I	Cryogenic tankage and smaller items of other equipment
(b) Sector II	SPS oxidizer tank (sump)
(c) Sector III	SPS fuel tank (storage)
(d) Sector IV	Fuel cells, SPS pressurization package, and smaller items of other equipments
(e) Sector V	SPS oxidizer tank (storage)
(f) Sector VI	SPS fuel tank (sump)
(g) Center section	SPS helium storage tanks (two)

The DSIF antenna shall be housed below the lower SM bulkhead and inside the adapter during boost.

3.3.1.2.3 SPS Tank Sizing. - Tank sizing for the SPS shall be apportioned as follows: (Figures are presented as percentages of usable propellant: 30,000 pounds oxidizer, 15,000 pounds fuel.)

	Oxidizer	Fuel
(a) Manufacturer's tolerance	1.58 percent	1.57 percent
(b) Ullage	1.08 percent	0.81 percent
(c) Loading tolerance	1.02 percent	1.02 percent
(d) Mixture ratio shift (oxidizer/fuel)	0.48 percent	0.48 percent
(e) Trapped propellant	0.73 percent	0.63 percent

3.3.1.3 LEM. - The LEM, an associate contractor module, shall be designed and constructed to meet the performance and interface requirements specified in SID 62-1244.

3.3.1.4 Adapter. - The adapter shall be designed and constructed to be a load-supporting structure between the SM and the Saturn V instrument unit. The adapter shall enclose and support the LEM during all mission phases prior to transposition of the LEM to the apex of the CM. The adapter shall be a conical-shaped section. The adapter configuration shall be as shown in Figure 26.

3.3.1.4.1 LEM-Adapter Support. - The LEM support structure shall be attached to the adapter and shall be supplied with the adapter by NAA/S&ID. The internal structure of the adapter shall consist of a frame at each end and a third frame to support the LEM.

The LEM contractor shall provide jettisonable adjustable fittings on the LEM for attachment to the support structure, and provisions for indexing to achieve proper orientation and alignment. Design and fabrication of the adapter and the LEM support structure shall be the responsibility of NAA/S&ID.

3.3.1.4.2 Adapter Access. - Access to the adapter interior shall be provided as shown in Figure 26. The LEM contractor shall provide access to the LEM interior.

3.3.1.4.3 Clearance. - Except for local protuberances between the adapter inner surface and the LEM outer surface, a 30-inch clearance shall be maintained for service access to the LEM, the lower part of the SM, and the upper part of the S-IVB vehicle. All local protuberances from the LEM, such as control nozzles, antennas, egress hatches, and optical devices shall maintain a minimum 3-inch clearance with all adapter and S-IVB structures. A minimum of 10 inches clearance shall be required between the SM engine nozzle exit plane and the LEM docking cone.

3.3.1.4.4 Separation. - The adapter shall be designed so that flexible linear-shaped charges shall break the adapter into a number of panels and disperse the panels to expose the alignment and attenuation cone on the LEM. After the docking maneuver has been accomplished, the remaining portion of the adapter shall be left with the S-IVB booster. LEM separation shall occur at the LEM fittings to the adapter support structure and shall be initiated from the CM.

3.3.1.4.5 Umbilical Lines. - There shall be no electrical, fluid, or pneumatic umbilical lines between the LEM and the CM or SM while the LEM is housed in the adapter.

3.3.1.4.6 Orientation. - The reference axes shall be oriented with respect to the crew members in their normal earth launch position in the CM. (See Table I.) The LEM shall be installed in the adapter with its positive X axis parallel to the positive X axis of the CM. The LEM X axis shall be positive in the launch flight direction. The Y axis shall be normal to the X axis and positive to the right of a CM crewman when the crewman is facing in the positive X direction. The Z axis shall be normal to both the X and Y axes, and shall be positive in the direction of the CM crewman's feet. The rotation of the LEM about the X axis and the possible translational distance from the CM X axis will be defined by NAA/S&ID to provide compatibility of orientation between the LEM and the SC.

3.3.2 Systems. -

3.3.2.1 G&N (NASA-Supplied). - The G&N shall be designed and constructed by an associate contractor designated by the NASA. NAA/S&ID shall be responsible for the interface design, integration of the G&N into the SC, and specification of G&N performance and interface requirements. The G&N shall be designed and constructed to provide steering and thrust control signals for the RCS and SPS through the SCS. The G&N shall be designed to provide storage capability for all precalculated mission trajectory parameters and compute course corrections by comparing these stored parameters with actual position data obtained by navigational sightings taken with the G&N optical equipment or by data up-link from ground radar. The weight of the G&N shall not exceed 350 pounds. This weight shall include the major components of the system, all controls and displays, intraconnecting wiring, spares, and any other SC-installed equipment supplied by the associate contractor.

3.3.2.1.1 Major Components. - The G&N shall include the following major components:

(a) Inertial measurement unit (IMU)

(b) Optical equipment

(c) Computer

(d) Electronics

(e) Controls and displays

(f) Navigation base.

3.3.2.1.2 Accuracy. - The G&N shall be designed for accuracy sufficient to control SC trajectory deviation to a degree consistent with overall fuel budgeting, atmospheric maneuverability, recovery force economics, crew limitations, and control system performance.

3.3.2.1.3 Interfaces. - The G&N shall be designed to assure physical and functional interface compatibility with other Apollo SC systems, including the LEM, ground-based equipment, and the crew in all ground and in-flight modes of operation.

3.3.2.1.4 Thermal Control. - The components of the G&N which require cooling for temperature control shall be designed to operate from the SC coolant system, which consists of cold plate heat exchangers and a water-glycol solution as the heat carrier. The use of quick-disconnect couplings shall be kept to a minimum and shall be used only where the cold plate is an integral part of the component to be cooled.

3.3.2.1.5 Telemetry. - The G&N shall be designed to supply ground and inflight measurements for ground monitor and data acquisition. The system shall be designed to provide these signals to the SC telemetry system and its associated recording equipment.

3.3.2.1.6 Ground Support Equipment (GSE). - The GSE associated with the G&N shall be designed to operate in conjunction with other SC GSE, within checkout procedures, and from the facilities available at the various checkout and countdown locations. The design of ground-handling equipment and the development of component installation procedures shall be coordinated with NAA/S&ID.

3.3.2.1.7 Stabilization and Control. - The G&N shall be designed to supply the SCS with those signals necessary to perform SC maneuvers, to maintain

SC attitude, and to display total attitude. Attitude steering commands shall be proportional to the attitude error angles in both stability and body axes, and the total attitude signals shall be proportional to the total SC attitude angle with respect to a fixed coordinate system. The G&N shall also generate SPS engine ON and engine OFF commands.

3.3.2.1.8 Operation by Crew. - The G&N shall be designed so that the operational procedures associated with the G&N shall not exceed the capabilities and limitations of one human operator for successful operation. Displays and controls shall provide the crew with adequate information on the status and modes of the system. The design of the components and of the installation techniques shall provide ease of accessibility during flight, in any mode of dress, and shall not require the use of special tools.

3.3.2.1.9 In-flight Test. - The G&N shall be designed to supply the IFTS with signals representing conditions that exist within the G&N. These signals shall be used to check system performance, evaluate trends, expedite isolation of failed or marginal subassemblies, and facilitate replacement. All test points shall be signal conditioned at the source, and shall be voltage normalized to a range of 0.0 to 5.0 volts direct current (vdc), negative grounded, at the output termination of the G&N.

3.3.2.1.10 Leakage Rates. - The G&N associate contractor shall supply NAA/S&ID with all information required to establish maximum allowable leakage rates both internal and external to the CM.

3.3.2.2 SCS. - The SCS shall be designed and constructed to provide for angular orientation and stabilization of the SC about three axes under command of the crew, the guidance system, and self-contained standby inertial reference system; provide translational control during transposition and docking; and provide thrust vector control during velocity corrections. The SCS shall also provide display of the various functions.

3.3.2.2.1 Major Components. - The SCS shall consist of the following basic components:

(a) Attitude reference

(b) Rate sensors

(c) Accelerometer

(d) Control electronic assembly

(e) Power supplies

(f) Manual controls

(g) Displays.

3.3.2.3 SPS. - The SC shall incorporate a propulsion system that shall provide all major velocity increments required for maneuvering after SC separation from the boost vehicle and prior to SM separation from the CM. The propulsion system shall be located in the SM and shall be designated as the SPS.

3.3.2.3.1 Major Components. - The SPS shall consist of a pressurized helium storage and distribution system, a propellant storage and distribution system, and a single rocket engine system.

3.3.2.3.1.1 Helium System. - Helium gas, contained at high pressure in dual interconnected tanks, shall be utilized for pressurization of the propellant supplies. Isolation valves shall be provided to preserve helium tank pressure in the event of leakage or component failure, and to shut off helium flow simultaneously with engine shutdown. Primary and secondary pressure regulators shall be incorporated in the helium distribution system. Filters shall be installed, where required, to prevent contamination damage to associated components. Check valves shall be utilized to prevent the contact of oxidizer and fuel in the helium distribution system. Relief valves shall be incorporated to relieve excessive propellant tank pressure. Where necessary, redundant components shall be incorporated to obtain the required overall system reliability. The helium storage tanks shall be designed for a maximum working pressure of 4, 500 pounds per square inch gage (psig).

3.3.2.3.1.2 Propellant System. - The propellants shall be a combination of earth-storable, hypergolic, bipropellant utilizing nitrogen tetroxide (N_2O_4) as the oxidizer and a mixture of 50 percent hydrazine (N_2H_4) and 50 percent unsymmetrical dimethylhydrazine (UDMH) as the fuel. The complete propellant system shall therefore consist of separate storage and distribution subsystems for oxidizer and fuel. Each storage subsystem shall include dual, interconnected tanks. Positive expulsion devices will not be provided. For an engine start during zero g conditions, the propellants shall be forced to the bottom of the tanks by an initial positive acceleration force supplied by the SM RCS. The propellant system shall include a propellant quantity-indicating system and means by which the oxidizer fuel ratio, delivered from the tanks to the engine, may be controlled manually by the crew during flight. Where required, filters and isolation valves shall be provided. Redundant components shall be incorporated, where necessary, to obtain the required overall system reliability. The propellant tanks shall be designed for a maximum working pressure of 240 psig.

3.3.2.3.1.3 Rocket Engine System. - The SPS engine shall be a liquid-fueled, pressure-fed, non-throttleable thrust generator. The engine shall be gimbal-mounted to permit thrust vector control. Chamber cooling shall be accomplished by the use of an ablative-lined combustion chamber fabricated from phenolic refrasil. A radiation-cooled nozzle extension shall be utilized to increase propulsion efficiency. The nozzle shall be designed for an expansion ratio of 60 to 1. Multiple engine propellant valves, arranged in series-parallel redundant pairs, shall be provided. Thrust vector control shall be achieved by electromechanical gimbal actuators, each of which will include redundant electric motors. Other redundant components shall be provided, where necessary, to obtain the required overall system reliability.

3.3.2.3.1.3.1 Engine Chamber Cooling. - The maximum temperature permissible in the engine chamber outer wall, resulting from propellant combustion or post-firing heat soak, shall be 200 degrees F. The chamber shall employ ablative material to provide the required temperature reduction across the chamber wall. The use of insulation between the ablative liner and the outer wall shall be permissible.

3.3.2.3.1.3.2 Abnormal Conditions. - The rocket engine shall be designed for safe operation under abnormal or "off design" conditions. While performance degradation is acceptable under these conditions, safety of operation shall not be compromised.

3.3.2.3.2 Other Provisions. -

3.3.2.3.2.1 Servicing. - Fill-drain couplings shall be provided for the helium and propellant tankage systems. The couplings shall be physically located and indexed to prevent accidental mixing of operating fluids. Vent couplings shall be provided for the venting of propellant vapors and to assure complete filling of the tanks during ground servicing operations. Where possible, ground-to-SC fluid transfer lines shall be disconnected and sealed by manual means. Provisions shall be included for propellant system purging.

3.3.2.3.2.2 Preflight Checkout. - The SPS shall be designed with test points to permit complete functional and leakage testing of the primary and secondary pressure regulators and of all valving in the helium distribution system. In addition, the SCS shall be designed with provisions for leakage testing of the propellant tanks and distribution system prior to servicing. There shall be provisions for checking the sensors employed in the engine gimbal system.

3.3.2.3.2.3 System Installation. - Where possible, a modular concept shall be used in the installation of systems and integral subsystems. Greatest possible accessibility shall be maintained and adverse interactions between systems or components shall be avoided. The number of connection points and traps in propellant lines shall be held to a minimum. Helium and propellant lines shall be welded or brazed, wherever possible, to minimize leakage. The rocket engine shall be mounted in the lower section of the SM, and shall be capable of being installed or removed as a complete unit.

3.3.2.3.3 Propellant System - Engine Interface. - Propellants shall be distributed to engine interface at the inlet conditions specified below:

(a) Nominal steady state operating pressures = 160 pounds per square inch absolute (psia) (oxidizer) 163 psia (fuel)

(b) Propellant temperature = 70 degrees F nominal

(c) Starting mode propellant supply pressure = 180 psia nominal

(d) Oxidizer to fuel mixture ratio = 2:1 nominal

3.3.2.4 RCS. - Both the CM and SM shall be provided with an RCS. The two systems, which shall be functionally and physically independent of each other, are designated as the CM RCS and the SM RCS.

3.3.2.4.1 CM RCS. - The complete CM RCS shall consist of two similar RCS (A and B). The two systems shall be designed to operate simultaneously during normal control operation. In the event of a failure of either system A or system B, the remaining system shall be designed to provide adequate control to safely complete the entry mode of the mission.

3.3.2.4.1.1 Major Components. - Each individual CM RCS (A and B) consists of a pressurized helium storage and distribution system, a propellant storage and distribution system, and six rocket engine systems.

3.3.2.4.1.1.1 Helium System. - Each helium system shall incorporate a single tank in which helium gas is contained for pressurization of the propellant supplies. Prior to initial system pressurization, the helium supplies shall be positively isolated from the distribution systems for the purpose of minimizing overall system leakage prior to the entry mode. Isolation shall be accomplished by normally closed, explosive-operated valves. Solenoid-actuated, helium isolation valves shall also be provided so that in the event of a failure of either system A or system B, the failed system may be deactivated for the remainder of the entry mode. Primary and secondary pressure regulators shall be incorporated in each helium distribution system. Filters shall be installed, where required, to prevent contamination damage to associated components. Check valves shall be utilized to prevent the contact of oxidizer and fuel in each helium distribution system. Relief valves shall be incorporated to relieve excessive propellant tank pressure.

Redundant components shall be incorporated, where necessary, to obtain the required overall system reliability. The helium storage tanks shall be designed to a maximum working pressure of 5,000 pounds per square inch gage (psig).

3.3.2.4.1.1.2 Propellant System. - The propellants shall be a combination of earth-storable, hypergolic, bipropellant utilizing nitrogen tetroxide (N_2O_4) as the oxidizer and monomethylhydrazine (MMH) as the fuel. Each of the propellant systems (A and B) shall therefore consist of separate storage and distribution subsystems for oxidizer and fuel. Each of the storage subsystems shall include a single, positive expulsion tank. The propellant supplies shall be positively isolated from the distribution systems, prior to initial system pressurization, for the purpose of centralizing the propellants during launch operations, and to reduce the period of time the distribution systems are exposed to propellants. Isolation shall be accomplished by burst diaphragm installations. Where required, filters and isolation valves shall be provided. A propellant disposal and purging subsystem shall be provided for the purpose of expelling remaining propellant and helium quantities prior to CM-earth impact. Where necessary, redundant components shall be incorporated to obtain the required overall system reliability. The positive expulsion propellant tanks shall be designed to a maximum working pressure of 360 psig.

3.3.2.4.1.1.3 Rocket Engine System. - The CM RCS engines shall be liquid-fueled, pressure-fed, pulse-modulated, ablative-cooled thrust generators. Each engine shall incorporate a single propellant valve for the oxidizer and a single propellant valve for the fuel. The valves shall be solenoid-actuated and each valve shall incorporate two electrically separated coils. An ablative-cooled nozzle extension, designed for an expansion ratio of 9 to 1, shall be utilized.

3.3.2.4.1.2 Other Provisions. -

3.3.2.4.1.2.1 Servicing. - Fill-drain couplings shall be provided for the helium and propellant tankage systems. The couplings shall be located and indexed to prevent accidental mixing of operating fluids. Vent couplings shall be provided for the venting of propellant vapors and to assure complete filling of the tanks during ground servicing operations. Where possible, ground-to-SC fluid transfer lines shall be disconnected and sealed by manual means. Provisions shall be included for propellant system purging.

3.3.2.4.1.2.2 Preflight Checkout. - The CM RCS shall be designed with test points to permit helium leakage testing of the explosive-operated isolation valves, complete functional and leakage testing of the primary and secondary pressure regulators, and of all other valving in the helium distribution systems. In addition, there shall be provisions for leakage testing of the propellant tanks and distribution systems prior to servicing.

3.3.2.4.1.2.3 System Installation. - Where possible, a modular concept shall be used in the installation of systems and integral subsystems. Greatest possible accessibility shall be maintained and adverse interactions between systems or components shall be avoided. The number of connection points and traps in propellant lines shall be held to a minimum. Helium and propellant lines shall be welded or brazed, wherever possible, to minimize leakage. The roll and yaw control engines and the negative pitch engines shall be located in the aft CM equipment compartment. The positive pitch engines shall be located in the forward compartment, adjacent to the parachute packages.

3.3.2.4.1.3 Propellant System - Engine Interface. - Propellants shall be distributed to the engine interface at the inlet conditions specified below:

(a) Nominal steady state operating pressure = 280 psia

(b) Propellant temperature = 70 degrees F nominal

(c) Starting mode propellant supply pressure = 291 psia nominal

(d) Oxidizer to fuel mixture ratio = 2:1 nominal

3.3.2.4.2 SM RCS. - The complete SM RCS shall consist of four similar reaction control modules located at 90-degree increments around the periphery of the SM. The four modules shall be capable of simultaneous operation during normal control procedures. (See Figure 27.)

3.3.2.4.2.1 Major Components. - Each SM RCS module shall consist of a pressurized helium storage and distribution system, a propellant storage and distribution system, and four rocket engine systems.

3.3.2.4.2.1.1 Helium System. - The helium system of each module shall incorporate a single tank in which helium gas is contained for pressurization of the propellant supplies. Solenoid-actuated, helium isolation valves shall be provided so that in the event of a module failure, the failed module may be deactivated for the remainder of the mission. Primary and secondary pressure regulators shall be incorporated in each helium distribution system. Filters shall be installed, where required, to prevent contamination damage to associated components. Check valves shall be utilized to prevent the contact of oxidizer and fuel in each helium distribution system. Relief valves shall be incorporated to relieve excessive propellant tank pressure. Redundant components shall be incorporated, where necessary, to obtain the required overall system reliability. The helium storage tanks shall be designed to a maximum working pressure of 5,000 psig.

3.3.2.4.2.1.2 Propellant System. - The propellants shall be a combination of earth-storable, hypergolic, bipropellant utilizing nitrogen tetroxide (N_2O_4) as the oxidizer and a mixture of 50 percent hydrazine (N_2H_4) and 50 percent unsymmetrical dimethylhydrazine (UDMH) as the fuel. The propellant system of each module shall therefore consist of separate storage and distribution subsystems for oxidizer and fuel. Each of the storage subsystems shall include a single, positive expulsion tank. Each propellant system shall include a propellant quantity-indicating system, filters, and isolation valves where required. Where necessary, redundant components shall be incorporated to obtain the required overall system reliability. The positive expulsion propellant tanks shall be designed to a maximum working pressure of 248 psig.

3.3.2.4.2.1.3 Rocket Engine System. - The SM RCS engines shall be liquid-fueled, pressure-fed, non-throttleable, radiation-cooled thrust generators, capable of continuous operation, pulse modulation, or any combination of both. Each engine shall incorporate a single propellant valve for the oxidizer and a single propellant valve for the fuel. The valves shall be solenoid-actuated, and each valve shall incorporate two electrically separated coils. A radiation-cooled nozzle extension, designed for an expansion ratio of 40 to 1, shall be utilized. The engine chamber pressure shall be 94 psia nominal during continuous operation.

3.3.2.4.2.2 Other Provisions. -

3.3.2.4.2.2.1 Servicing. - Fill-drain couplings shall be provided for the helium and propellant tankage systems. The couplings shall be located and indexed to prevent accidental mixing of operating fluids. Vent couplings shall be provided for the venting of propellant vapors and to assure complete filling of the tanks during ground servicing operations. Where possible, ground-to-SC fluid transfer lines shall be disconnected and sealed by manual means. Provisions shall be included for propellant system purging.

3.3.2.4.2.2.2 Preflight Checkout. - The SM RCS shall be designed with test points to permit complete functional and leakage testing of the primary and secondary pressure regulators and of all valving in the helium distribution systems. In addition, there shall be provisions for leakage testing of the propellant tanks and distribution systems prior to servicing.

3.3.2.4.2.2.3 System Installation. - Where possible, a modular concept shall be used in the installation of systems and integral subsystems. Greatest possible accessibility shall be maintained, and adverse interactions between systems or components shall be avoided. The number of connection points and traps in propellant lines shall be held to a minimum. Helium and propellant lines shall be welded or brazed, wherever possible, to minimize leakage. Upon installation of the four modules, the rocket engines shall be located in fixed clusters of four on the external surface of the SM with the nozzles canted 10 degrees outboard to reduce exhaust plume impingement.

3.3.2.4.2.3 Propellant System - Engine Interface. - Propellants shall be distributed to the engine interface at the inlet conditions specified below:

(a) Nominal steady state operating pressure = 170 psia

(b) Propellant temperature = 70 degrees F nominal

(c) Starting mode propellant supply pressure = 181 psia nominal

(d) Oxidizer to fuel mixture ratio = 2:1 nominal.

3.3.2.5 LES. - The LES shall be designed to provide crew-initiated or automatic abort capabilities to the CM. The design objective weight of the LES shall be 6,700 pounds maximum, excluding ballast. The overall height shall be 400 ± 0.5 inches.

3.3.2.5.1 Major Components. - The LES shall consist of the following major components:

(a) Launch escape motor

(b) Pitch control motor

(c) Tower jettison motor

(d) Automatic events sequencing system

(e) Tower frame

(f) Skirt

(g) Ballast

(h) Separation system for tower leg restraints

(i) Nose cone

3.3.2.5.1.1 Launch Escape Motor. - The main propulsion motor shall be a solid-fueled rocket motor with four fixed nozzles. The nozzle shall be one of small, two of intermediate, and one of large throat diameter, arranged to provide an appropriate direction of thrust vector. The nozzles shall be canted to minimize impingement of jet plumes on the CM. The tower structure shall not be subjected to plume impingement at low altitudes in order to minimize interference with thrust vector during aborts prior to or shortly after liftoff.

3.3.2.5.1.2 Pitch Control Motor. - The pitch control motor shall be a solid-fueled rocket motor with a fixed nozzle. The motor shall be mounted in the LES nose cone area and placed normal to the X axis to provide a resultant thrust vector in the minus Z direction, and shall be sized to obtain optimum pad abort performance.

3.3.2.5.1.3 Tower Jettison Motor. - The tower jettison motor shall be a solid-fueled rocket motor with two fixed nozzles. The throat diameters of the two nozzles shall differ to provide an appropriate thrust vector direction. The nozzles shall be of a scarfed design and shall exhaust through the sides of the interstage with only a slight projecting lip.

3.3.2.5.1.4 Automatic Events Sequencing System. - The LES shall include an automatic events sequencing system. This system shall be designed to provide automatic activation of system components at predetermined time increments and check-off functions on systems events.

3.3.2.5.1.4.1 Earth Landing Sequencing System. - In the event of launch pad or atmospheric abort, the earth landing sequencing system shall be initiated by tower separation or by a barometric switch, depending on the abort altitude.

3.3.2.6 ELS. - The ELS shall be designed to utilize a multiple parachute technique for decelerating and safely landing the CM following entry from lunar flight or mission abort conditions.

3.3.2.6.1 Major Components. - The ELS shall consist of the parachute subsystem, the ELS sequencer, recovery aids, and the forward heat shield jettison subsystem.

3.3.2.6.1.1 Parachute Subsystem. - The parachute subsystem shall include:

(a) One drogue parachute system, consisting of:

  1. One drogue parachute
  2. One drogue parachute deployment bag
  3. One drogue parachute mortar assembly

4. One drogue parachute riser
5. One drogue disconnect assembly
6. Two drogue mortar pyrotechnic cartridges

(b) Three main landing parachute assemblies, each consisting of:

1. One pilot parachute system, composed of:
    a. One pilot parachute
    b. One pilot parachute deployment bag
    c. One pilot parachute mortar assembly
    d. One pilot parachute riser
    e. Two pilot parachute mortar pyrotechnic cartridges
2. One main parachute pack assembly, composed of:
    a. One main parachute
    b. One main parachute deployment bag
    c. One main parachute riser
    d. Main parachute pack retention flaps

(c) One main cluster disconnect assembly, containing one pyrotechnic main cluster disconnect

(d) One SC main cluster harness assembly

(e) One sequence controller.

3.3.2.6.1.1.1 Pilot and Main Parachute Assembly. - The main parachute assembly shall consist of three individual parachutes simultaneously deployed by three individual mortar-ejected pilot chutes. The main landing parachutes shall be deployed within the operating envelope defined in Figure 28; however, in no case at an altitude less than 7,500 feet for the entry condition, or 2,000 feet for the abort condition.

3.3.2.6.1.1.1.1 Descent Control. - The inflation of two of the three main landing parachutes shall be sufficient to reduce the rate of descent of an 8,500-pound CM to a terminal velocity no greater than 30 feet per second at 5,000 feet altitude.

3.3.2.6.1.1.1.2 Main Cluster Disconnect Assembly. - The main cluster disconnect assembly shall connect the three main parachute risers to the CM main harness assembly and shall disconnect upon signal from the sequence controller. Pyrotechnic means may be employed for disconnect operation.

3.3.2.6.1.1.2 Spacecraft Main Cluster Harness Assembly. - The harness assembly shall attach to the four parachute attach fittings and extend to the main parachute disconnect assembly. The harness assembly shall maintain a continuous loop between the parachute attach fittings after operation of the disconnect.

3.3.2.6.1.2 Sequence Controller. - The sequence controller shall contain time delay switches, relays, and baroswitches as necessary to properly sequence the deployment and disconnect of the various devices related to or contained in the parachute subsystem. Provision for manual override of selected functions shall be incorporated in the sequence controller.

3.3.2.6.1.3 Recovery Aids. - The recovery aids shall provide the location and survival capabilities necessary for safe and prompt recovery of the CM and crew. The individual aids are as follows:

3.3.2.6.1.3.1 SOFAR Bomb. - In the event landing occurs on water, a SOFAR bomb shall be ejected from the CM. The sound resulting from the explosion shall be detectable by sonar installations previously stationed in the general impact area.

3.3.2.6.1.3.2 Dye Markers. - Dye markers shall be provided as an aid for aerial detection over water.

3.3.2.6.1.3.3 Flashing Beacon Light. - A waterproof flashing beacon light shall be provided for aerial detection at night.

3.3.2.7 Crew Systems Equipment. -

3.3.2.7.1 Couches. - Couches shall be designed to provide comfortable support during all mission phases. The seat shall be approximately 24 inches wide with a slight lateral curvature that shall accommodate a crewman in a pressurized suit. The couch shall be covered with a "slow memory" liner to improve load distribution. A comfort liner shall overlay the slow memory liner. The center couch shall be capable of being repositioned for use as a sleeping area. A two-way stretch, open-mesh coverlet shall be provided to draw over the crewman, and hold him in place against the stowed center couch.

3.3.2.7.2 Restraint System. - The restraint system shall be adjustable as required to provide the desired crew restraint and support for the anticipated forces during normal and emergency conditions.

3.3.2.7.2.1 Adjustable Areas. - Couch and restraint equipment shall have sufficient adjustment to provide comfort and attitude change in all postures from sitting at an 86-degree hip angle to full reclining. Suitable adjustment shall be provided for headrest, backrest, and all arm, leg, hand, and foot rests.

3.3.2.7.3 Lighting Equipment.- Lighting equipment of the SC shall consist of lamps and fixtures for floodlighting and integral lighting of the CM, windows, and window filter assemblies. A portable light and a survival and rescue light shall be provided.

3.3.2.7.3.1 Lamp Sources. - Floodlighting sources shall be incandescent lamps.

3.3.2.7.3.2 Lamp Fixtures. - Lamps and fixtures shall be recessed where possible, and shall be located so as to provide even illumination and minimize undesirable reflections and glare.

3.3.2.7.3.3 Lighting System. - The CM shall be illuminated by controllable floodlighting.

3.3.2.7.3.4 Floodlighting. - Floodlighting shall consist of multiple lamps located to provide illumination with a minimum of glare. The brightness shall be capable of graduation from off to full brightness. Separate illumination and control shall be provided for each functional crew station.

3.3.2.7.3.5 Integral Illumination. - Integral illumination shall consist of light sources located within the display to provide the required contrast between indicia and background during periods when low luminance levels are required.

3.3.2.7.3.6 Portable Light. - A high intensity flashlight shall be provided to permit the crew members to view unlighted areas or equipment inside or outside of the CM.

3.3.2.7.3.7 Window Filters. - Window filters shall be provided to control illumination of the CM from exterior sources.

3.3.2.7.4 Crew Accessories. - The crew accessories shall be designed to provide adequate assistance to allow the crew members to record data and perform necessary tasks, and perform manual backup for guidance and navigation functions. The crew accessories shall include:

(a) Map and manual case assembly

(b) Crew map and manual set

(c) Crew logbook and checklist assembly

(d) Crew lapboard assembly.

3.3.2.7.5 Survival Equipment. - Post-landing survival equipment shall include life raft, food, location aids, first aid equipment, and various accessories necessary to support the crew outside the SC for 3 days. Provisions shall be included for removing a 3-day water supply from the SC after landing.

3.3.2.7.6 Personal Equipment. - The crew personal equipment shall include the following:

(a) Space suit assembly— NASA-furnished

(b) Portable life support system(PLSS) — NASA-furnished

(c) Personal radiation dosimeters—NASA-furnished

(d) Umbilical hose

(e) In-flight maintenance belt

(f) Valve-oxygen high pressure charging

(g) Food mouthpiece

(h) Communications assembly—personal (headset, microphone, and constant-wear hat).

3.3.2.7.7 Subsistence and Associated Equipment. -

3.3.2.7.7.1 Nutrition. - Adequate quantities of food and water shall be provided for the crew to maintain an adequate diet throughout the 14-day mission. Food equipment required for food preparation shall be provided.

3.3.2.7.7.2 Food Management Equipment. - Food management shall include storage and preparation of food and disposal of waste food in bags. Plastic film retainers shall cover the individually packaged food products to prevent them from drifting from open storage areas. Reconstitution shall be accomplished with hot or cold water, depending upon the food to be reconstituted, injected by probe into the individual containers.

3.3.2.7.7.3 Waste Management Equipment. - Waste management equipment shall be designed to allow for the collection and storage of liquid and solid human waste in the following manner:

(a) Solid human waste shall be disposed of in defecation-emesis bags.

(b) Gaseous wastes shall be collected from the feces receptacle and the urine disposal system and transmitted to the ECS for processing.

(c) Urine shall be collected, freed of viable bacteria, stored, and ejected overboard at periodic intervals.

(d) A vacuum cleaner shall be provided for retrieval of any spillage.

3.3.2.7.8 Personal Hygiene Equipment. -

3.3.2.7.8.1 Oral Hygiene Supplies. - Oral hygiene supplies shall be designed to provide adequate cleansing of teeth and gums.

3.3.2.7.8.2 Shaver Assembly, Personal. - A completely portable, spring-driven, hand-wound, vacuum-filtered shaver shall be designed to provide well-being and general cleanliness. The shaver shall be designed for storing, in a space other than in the shaving head assembly, the hair shavings accumulated through normal use of the shaver by three men during the 14-day mission.

3.3.2.7.8.3 Defecation-Emesis Bags, Toilet Tissue. - The defecation-emesis bags shall be designed to provide adequate storage facilities for human waste material. The bags shall be interchangeable for either defecation or emesis purposes. Toilet tissue also shall be provided. Defecation-emesis bags shall minimize escape of odors and provide for complete sealing of bags after use.

3.3.2.7.8.4 Cleansing and Deodorizer Pads. - Cleansing pads shall be provided for all-purpose washing needs. Individual pads shall be impregnated with cleanser, disinfectant, and deodorizer to satisfactorily meet sanitation requirements. Individual cleansing pads shall be wrapped in multi-use wrappers.

3.3.2.7.9 Medical Equipment. - Medical equipment shall include surgical equipment, supplies, injectible drugs, first aid equipment, and the devices listed in 3.4.4.8.1.11.2 for inflight measurement of the physiological functions of the astronauts.

3.3.2.7.10 Space Suit Assembly (NASA-Furnished). -

3.3.2.7.10.1 Pressure Garment Assembly. - The pressure garment assembly shall be worn over the constant wear garment for protection of the crewman. The pressure suit and PLSS shall be used for lunar extravehicular exploration and maintenance.

3.3.2.7.10.2 Constant Wear Garment. - The constant wear garment shall be worn under the pressure garment assembly and in the cabin "shirtsleeve" environment. It shall resemble a summer flying suit and shall be worn by the crewmen at all times. The one piece garment shall be the primary clothing for the crewmen and shall integrate functions of comfort, hygiene, and biomonitoring. The constant wear garment shall be free of protruding pockets or attachment hardware. Appropriate openings shall be allowed for urination and defecation, and throwaway pads in the apocrine sweat gland regions. The pads shall provide odor and bacteria control without affecting the pressure suit as a whole. The entire undergarment assembly shall be worn under the pressure garment and shall serve as a carrier for communications, biomonitoring, and other appropriate devices.

3.3.2.7.11 PLSS (NASA-Furnished). - The PLSS shall be used with the space suits during extravehicular operations. A separate system shall be provided for each crew member. This system shall be of the recirculating type. It shall be capable of supporting a crewman for 4 hours independent of the SC, and shall be capable of being recharged. It shall include the communications transceiver that will be used by the crewmen while outside the SC.

3.3.2.8 ECS. - The ECS shall be capable of providing a habitable environment in the Apollo CM.

3.3.2.8.1 Major Components. - The ECS shall include the following:

(a) Pressure suit circuit subsystem

(b) Water-glycol circuit subsystem

(c) CM pressure and temperature control subsystem

(d) Oxygen subsystem

(e) Water supply subsystem.

3.3.2.8.1.1 Pressure Suit Circuit Subsystem. - The pressure suit circuit subsystem shall be designed to automatically control the flow, pressure, temperature, and composition of the pressure suit gas. The pressure suit circuit subsystem, in conjunction with the CM pressure and temperature control subsystem, shall also control these environmental conditions in the CM when any or all of the crew are out of their pressure suits.

3.3.2.8.1.2 Water-Glycol Circuit Subsystem. - The water-glycol circuit subsystem shall be an intermediate heat transfer loop which permits heat to be transferred from the CM interior and the electronic equipment to the space radiators.

3.3.2.8.1.3 CM Pressure and Temperature Control Subsystem. - The CM pressure and temperature control subsystem shall be designed for automatically maintaining the pressure and temperature of the cabin.

3.3.2 8.1.3.1 CM-LEM Docked Position. - Atmospheric control, that is, temperature, humidity, carbon dioxide removal, and gas circulation, shall not be provided for the LEM from the CM ECS.

3.3.2.8.1.3.2 LEM Oxygen. - The CM-SM ECS shall be capable of providing 14.46 pounds of pure oxygen for the LEM, including two pressurizations, leakage losses, and docking losses.

3.3.2.8.1.4 Oxygen Subsystem. - The oxygen subsystem shall be subdivided into two subsystems as described in the following paragraphs.

3.3.2.8.1.4.1 Normal Oxygen Subsystem. - The normal oxygen subsystem shall be designed for controlling the inflow of oxygen from the cryogenic storage subsystem as required for cabin pressurization and metabolic consumption for both the normal and emergency modes.

3.3.2.8.1.4.2 Entry Oxygen Supply Subsystem. - The entry oxygen supply subsystem shall be designed for supplying the oxygen for entry after separation of the SM from the CM. The entry oxygen shall be stored at 900 pounds per square inch (psi) in a single storage tank with a 3-pound capacity. The redundant supply shall be stored in the PLSS.

3.3.2.8.1.5 Water Supply Subsystem. - The water supply subsystem shall be designed to collect, store, and provide water for supplemental and emergency heat transfer operation and collect and store potable water for consumption by the crew members.

3.3.2.8.1.6 ECS and EPS Oxygen. - The oxygen for the ECS and the EPS shall be stored in the supercritical cryogenic state in two tanks. The hydrogen for the EPS shall also be stored in the supercritical cryogenic state in two tanks.

3.3.2.9 EPS. - The EPS shall be designed to supply, regulate, and distribute all electrical power required by the CM and the SM for mission requirements, and for LEM during checkout and monitoring.

3.3.2.9.1 Major Components. - The EPS shall include the following major components:

(a) Fuel cell powerplants

(b) Space radiators

(c) Mechanical accessories

(d) Storage batteries

(e) Electrical power distribution, controls, and protection

(f) Battery charger

(g) Inverters.

3.3.2.9.1.1 Fuel Cell Powerplants. - The fuel cell powerplants, connected in parallel, shall consist of fuel cells also connected in parallel. Each fuel cell shall be designed to be of the low-pressure, intermediate-temperature, Bacon-type, utilizing porous nickel, unactivated electrodes, and aqueous potassium hydroxide as the electrolyte. The fuel cells shall be operated nonregeneratively, utilizing hydrogen and oxygen as the reactants.

3.3.2.9.1.1.1 Operating Pressure and Temperature. - The normal fuel cell operating pressure and temperature shall be approximately 60 psia and 410 to 500 degrees F respectively.

3.3.2.9.1.1.2 Reactant Consumption. - Under normal conditions of operation, with two fuel cell powerplants operating, the specific reactant consumption shall not exceed an average of 0.84 pound per kilowatt-hour (kwh), total hydrogen and oxygen.

3.3.2.9.1.1.3 Water Production. - The water produced by the fuel cell powerplant shall be potable and shall be separated from the hydrogen by a centrifugal water separator and stored. Under normal conditions of operation, water production shall be a total of 2.6 pounds per hour, two modules operating.

3.3.2.9.1.1.4 Ground Start Capabilities. - Self-sustaining reaction within the fuel cell module shall be initiated at a temperature of approximately 410 degrees F. Integral heaters shall be provided to facilitate ground starting. The heaters shall be required during ground operation of the EPS. These heaters shall not overheat units with the fuel cell and its cooling system inoperative. In-flight start capability shall not be provided.

3.3.2.9.1.1.5 Water Contamination Sensing and Control. - A detection and control system shall be provided with each fuel cell powerplant to prevent contamination of the collected water supply.

3.3.2.9.1.1.6 Power Output. - Each fuel cell powerplant shall be designed to have a nominal capacity of 1,500 watts at an output voltage above 27.0 volts direct current (vdc).

3.3.2.9.1.2 Space Radiator. - A space radiator system shall be comprised of radiator panels provided to dissipate waste heat generated in the fuel cells.

3.3.2.9.1.3 Mechanical Accessories. - Mechanical accessories for the system shall include control components, heat exchangers, hydrogen circulators, water separators, piping, isolation valves, and other devices as required.

3.3.2.9.1.3 1 Valve Arrangement. - Adequate valves and controls shall be provided to isolate identical reactant supplies from each other, and from the ECS and fuel cell powerplants. Valve arrangement shall allow flow from any reactant supply to any fuel cell powerplant.

3.3.2.9.1.4 Storage Batteries. - Auxiliary electrical power shall be provided by five zinc-silver oxide storage batteries. Two of these batteries shall be designed to supply all power during the re-entry phase, and one of the remaining batteries plus the reserve of the other two batteries shall supply recovery phase loads. The two remaining batteries shall be designed to supply the redundant pyrotechnic busses.

3.3.2.9.1.4.1 Battery Charger. - The battery charger shall be designed to utilize solid state devices. The battery charger shall recharge the storage batteries, except the pyrotechnic batteries, after normal or emergency discharge and also charge the backpack batteries.

3.3.2.9.1.5 Electrical Power Distribution, Controls, and Protection. - The distribution portion of the EPS shall contain all necessary busses, wiring protective devices, switching, and regulating equipment.

3.3.2.9.1.5.1 System Voltage. - Electrical power shall be generated and distributed at 28 vdc (nominal).

3.3.2.9.1.5.2 Voltage Variance. - The voltage level shall be maintained to prevent variance of more than plus 2 or minus 3 volts from the nominal voltage under all steady state conditions of operation of the fuel cell system without voltage regulators.

3.3.2.9.1.5.3 Alternating Current (AC) Ripple. - All dc busses in the system shall be maintained essentially free of ac ripple.

3.3.2.9.1.5.4 Protection. - Busses and electrical loads shall be selectively protected so that individual load faults shall not cause an interruption of power on the bus to which the load is connected. Likewise, a fault on the nonessential bus shall not cause an interruption of power to the essential bus.

3.3.2.9.1.5.5 Load Grouping. - All electrical loads supplied by the distribution system shall be classified as essential, nonessential, pyrotechnic, or recovery. Essential loads are defined as those loads (except pyrotechnic circuits) that are mandatory for safe return of the SC to earth from any point in the lunar mission. Loads not necessary for the safe return of the SC shall be grouped on the nonessential bus and provision made for disconnecting these loads as a group under emergency conditions. All loads required during the post-landing recovery period shall be supplied by the recovery bus and provision made for manually disconnecting this bus from the essential bus following landing. Redundant busses shall be provided for pyrotechnic circuits, and used to supply only that type load.

3.3.2.9.1.5.6 Power Inversion. - The CM EPS shall be designed to invert some dc power to ac power and shall then regulate, control, and distribute this ac power, except for the ac power required by the G&N system which operates directly from SC dc power.

3.3.2.9.1.5.7 External Power. - Provision shall be made to energize the distribution system from an external source capable of supplying 28 vdc at a maximum current level of 150 amperes through a ground power receptacle located within the CM.

3.3.2.9.1.5.8 Electrical Distribution Panel. - The distribution panel shall be dead front and adequately enclosed or otherwise protected to minimize hazards to the crew, and shall be designed to provide maximum mechanical protection for the electrical system and components. Switching and control shall be accomplished by manually operated circuit breakers or switches in preference to electrically operated contactors, except where use of a remotely controlled device is necessary to reduce the length of large electrical conductors.

3.3.2.9.1.5.8 System Type. - The distribution system shall be a two-wire grounded system; that is, wire and busses shall be employed as the return path for electrical currents, in lieu of using the SC structure for this purpose. The system negative shall be grounded at one point only and shall not be interrupted by any control or switching device.

3.3.2.9.1.5.9 Inverters. - Three static inverters shall supply 115 to 200 volts at 400 cps, and shall be three-phase, Y-connected, ac power. One inverter shall operate with the other two acting as standby units. The inverter shall utilize solid state devices.

3.3.2.10 C&I. -

3.3.2.10.1 Communications Subsystem. -

3.3.2.10.1.1 Major Components. - The major components of the communications subsystem shall include:

(a) Very high frequency (VHF) frequency modulation (FM) transmitter equipment

(b) VHF-amplitude modulation (AM) transmitter-receiver equipment

(c) Unified S-band equipment

(d) S-band power amplifier equipment

(e) C-band transponder equipment

(f) Rendezvous transponder equipment

(g) VHF recovery beacon equipment

(h) High frequency (HF) transceiver equipment

(i) Intercommunications equipment

(j) Television (TV) equipment

(k) Pulse code modulation (PCM) telemetry equipment

(l) Premodulation processor equipment (PPE)

(m) Beacon antenna equipment

(n) VHF/2-kilomegacycle (kmc) omni antenna equipment

(o) VHF recovery antenna equipment

(p) Backup VHF recovery antenna equipment

(q) HF recovery antenna equipment

(r) 2-kmc high gain antenna equipment

(s) Rendezvous transponder antenna equipment

(t) Radome equipment

(u) Forward hatch rotary joints.

3.3.2.10.1.1.1 VHF-FM Transmitter Equipment. - The VHF-FM transmitter shall be designed to provide a means of transmitting PCM telemetry data during near-earth phases of the missions. This transmitter shall operate at a frequency of 237.8 megacycles (mc) with modulation capability of 64,000 bits per second and power output of 10 watts.

3.3.2.10.1.1.2 VHF-AM Transmitter-Receiver Equipment. - The VHF-AM transmitter-receiver shall be designed to provide a means of two-way voice communications between SC and earth during near-earth phases of all missions, between SC and PLSS transceiver, between the CM and the LEM, and between SC and rescue aircraft. In addition, the transmitter-receiver may be used as a backup VHF recovery beacon during recovery. The transmitter-receiver shall be designed for receiving biomedical data from the PLSS transceiver during extra-vehicular activities of the crew, restricted by a 70 kilocycle (kc) intermediate frequency (IF) bandwidth. The transmitter shall operate at a frequency of 296.8 mc at a power output of 5 watts continuous wave (cw). The receiver shall operate at either of two selectable frequencies, 296.8 mc or 259.7 mc.

3.3.2.10.1.1.3 Unified S-Band Equipment. - The unified S-band equipment shall be designed to provide two-way voice communications, emergency keyed transmission, PCM telemetry data and television transmission from the SC, coherent two-way doppler tracking, and turn-around pseudo random noise (PRN) ranging during cislunar phases. During near-earth phases the same operation shall be provided when within line-of-sight of a suitably equipped ground station. The transmitter shall be designed to operate in the 2,200 to 2,400 mc band, with provisions for both crystal-controlled and voltage-controlled oscillator excitation, and have a modulation capability of 1.5 mc FM or phase modulation (PM) at a power output of 250 milliwatts. The receiver shall operate in the 2,000 to 2,200 mc band and shall be a phase-locked double conversion superheterodyne receiver.

3.3.2.10.1.1.4 S-Band Power Amplifier Equipment. - The S-band power amplifier shall be designed to amplify the output of the unified S-band equipment, to provide the required power for transmission in the various modes at distances from near-earth to lunar. This equipment shall be designed to operate in the 2,200 to 2,400 mc band with a radiofrequency (RF) bandwidth of 10 mc at nominal power outputs of either 5.0 or 20 watts.

3.3.2.10.1.1.5 C-Band Transponder Equipment. - The C-band transponder shall be designed to be used for radar tracking during near-earth phases of the mission. The transponder shall be designed to operate in the 5,640 - 5,815 mc band using PM with a capability of 1,300 replies per second of 0.75 plus or minus 0.1 microsecond width pulses of 2.5 kilowatts peak power output. The transponder shall be a comparator type for use with the multiple antennas.

3.3.2.10.1.1.6 Rendezvous Transponder Equipment. - A rendezvous transponder compatible with the LEM rendezvous radar equipment operating at X-band frequency shall be provided to permit radar tracking of the CM-SM by the LEM.

3.3.2.10.1.1.7 VHF Recovery Beacon Equipment. - The VHF recovery beacon shall be provided to aid rescue aircraft in locating the SC during the recovery and post-landing phase. The beacon shall be designed to operate at a frequency of 243 mc using pairs of 10 plus or minus 1 microsecond pulses of 100 watts peak power superimposed on a 2-watt cw carrier.

3.3.2.10.1.1.8 <u>HF Transceiver Equipment.</u> - The HF transceiver shall be provided to aid rescue forces in localizing the SC to an area within which the VHF equipment can operate and locate the SC during the postlanding phase. The HF transceiver shall operate at a single frequency in the 8 to 16 mc band and shall be a 20-watt peak envelope power (PEP) single sideband/5-watt cw compatible AM-HF transceiver. Modulation shall be voice or keyed cw.

3.3.2.10.1.1.9 <u>Intercommunications Equipment.</u> - The intercommunications equipment shall be designed to provide the crew with intercommunication in both pressure suit and shirtsleeve environments, and shall provide flexibility of voice communication over the various SC transmitters and receivers. The intercommunication equipment shall consist of audio center equipment and three audio control units. The equipment shall be compatible with the NASA-furnished headsets that consist of microphones and earphones.

3.3.2.10.1.1.10 <u>Television Equipment.</u> - A television camera shall be designed to be used to provide real time video information for transmission to the GOSS stations. Video information shall be transmitted directly to earth via the unified S-band equipment for observation of astronauts and flight operations. The camera shall be portable, and shall provide an analog type output of 400 total lines per frame at a rate of 10 frames per second.

3.3.2.10.1.1.11 <u>PCM Telemetry Equipment.</u> - The PCM telemetry equipment shall be designed to process SC data into forms suitable for transmission to earth. Conditioned analog data shall be commutated and digitized. These data and other digital data shall be coded into PCM format and transmitted to earth during cislunar phases by the unified S-band equipment. During near-earth phases of all missions, data shall be transmitted over the VHF-FM transmitter. The PCM telemetry equipment shall incorporate provisions for in-flight calibration of PCM equipment. The telemetry equipment shall be designed to be compatible with the carry-on prelaunch automatic checkout equipment (PACE). The accuracy of the PCM telemetry equipment shall be 0.5 percent. Bit rate shall be fixed at 32,000 bits per second with a non-return to zero (NRZ) format except that a minimum data rate mode shall be provided at a bit rate of 1,000 to 2,000 bits per second. The channel capacity shall be as follows:

(a) Analog signal inputs

1.	100 samples per second	14 channels	0 to +5 volts input

2.	50 samples per second	13 channels	0 to +5 volts input
3.	10 samples per second	100 channels	0 to +5 volts input
4.	1 sample per second	75 channels	0 to +5 volts input
5.	1 sample per second	50 channels	0 to 250 millivolts input

(b) Digital signal inputs

1. Parallel inputs

a.	100 samples per second	1 channel
b.	10 samples per second	22 channels

2. Serial inputs

50 samples per second	5 channels

3. 3. 2. 10. 1. 1. 12 <u>PPE</u>. - PPE shall be provided to process and combine the outputs of the intercommunication, data storage, television, and PCM telemetry equipment for the various modes of modulating the RF equipment. The unified S-band equipment voice subcarrier shall be 1. 25 mc and the PCM subcarrier shall be 1. 024 mc. The PPE shall also provide a means to demodulate the 30 kc voice subcarrier from the unified S-band equipment. It shall also provide nine subcarrier oscillators for playback and transmission of nine analog tape channels on the unified S-band link and for relay of information received on the ultra high frequency (UHF)-AM transmitter-receiver via the unified S-band equipment.

3. 3. 2. 10. 1. 1. 13 <u>Beacon Antenna Equipment</u>. - The beacon antenna equipment shall be provided for use with the C-band transponder equipment. The equipment shall consist of four antennas and the transmission cables for operation in the 5, 640 to 5, 815 mc band and shall provide a right-hand circular polarized coverage. Coverage shall be omnidirectional to the maximum extent possible.

3. 3. 2. 10. 1. 1. 14 <u>VHF/2-kmc Omni Antenna Equipment</u>. - The VHF/2-kmc omni antenna equipment shall be provided for use with all VHF communications equipment as well as the unified S-band and power amplifier equipment during all flight phases up to jettison of the forward section for parachute deployment. The equipment shall consist of the VHF/2-kmc omni antenna, VHF multiplexer, two coaxial switches, and the transmission cables. The VHF antenna shall operate at 225 to 300 mc with a minimum

power handling requirement of 40 watts cw plus 100 watts peak pulse power and provide a vertical linear polarized coverage which is, as a design objective, omnidirectional in the Y-Z plane and has a 5 decibel (db) beamwidth of 145 degrees in the longitudinal plane. The 2-kmc antenna shall operate at 2,000 to 2,400 mc and provide a linear polarized coverage which is a design objective, omnidirectional in the Y-Z plane and has a 5 db beamwidth of 135 degrees in the longitudinal plane.

3.3.2.10.1.1.15 VHF Recovery Antenna Equipment. - The VHF recovery antenna equipment shall be provided for use with all VHF equipment from main parachute deployment to earth landing and during the post-flight phase of the mission. The equipment shall consist of the antenna, an actuating mechanism, and transmission cables for operation in the 225 to 300 mc band and provide a vertical linear polarized coverage which is omnidirectional in the Y-Z plane and has a 3 db beamwidth of 90 degrees in the longitudinal plane.

3.3.2.10.1.1.16 Backup VHF Recovery Antenna Equipment. - The backup VHF recovery antenna equipment shall be provided for use with the VHF recovery communications equipment during the post-flight phase of the mission in the event the VHF recovery antenna is damaged during landing. The equipment shall be similar to the VHF recovery antenna equipment.

3.3.2.10.1.1.17 HF Recovery Antenna Equipment. - The HF recovery antenna equipment shall be provided for use with the HF transceiver equipment during the post-flight phase of the mission. The equipment shall consist of the antenna, an actuating mechanism, an impedance matching network, and transmission cables for operation in the 8 to 16 mc band and provide a vertical polarized coverage which is omnidirectional in the Y-Z plane.

3.3.2.10.1.1.18 Two-KMC High Gain Antenna Equipment. - The 2-kmc high gain antenna equipment shall be provided for use with the unified S-band equipment and S-band power amplifier during cislunar phases of the mission. The equipment shall consist of the antenna, two gimbal drives, and servomechanisms, two RF rotary joints, transmission cables, and automatic earth-tracking equipment. The antenna operates in the 2,000 to 2,400 mc band and provides a steerable beam that is right-hand circularly polarized.

3.3.2.10.1.1.19 Rendezvous Transponder Antenna Equipment. - The X-band rendezvous antenna equipment shall be provided for use with the rendezvous transponder.

3.3.2.10.1.1.20 Radome Equipment. - A radome shall be provided to cover and protect the VHF/2-kmc omni antenna from the environments to which the SC is subjected. The radome shall provide 85 percent (minimum) transmission of 225 to 300 mc and 2,000 to 2,400 mc electromagnetic energy.

3.3.2.10.1.1.21 Forward Hatch RF Rotary Joint Equipment. - Forward hatch RF rotary joint equipment shall be provided to serve as transmission lines for the VHF/2-kmc omni antenna at the hinges of the forward hatch. Four RF rotary joints shall be provided.

3.3.2.10.2 Instrumentation Subsystem. - Instrumentation shall be comprised of an operational instrumentation system, NASA-furnished biomedical instrumentation, and GFAE scientific instrumentation.

3.3.2.10.2.1 Operational Instrumentation. - The operational instrumentation system shall be designed to sense, condition, and provide parallel data to the crew display panel, IFTS, ground support checkout system, and to the telemetry system. Each utilizing system shall provide isolation, so that in the event of a failure within a system, the data to other utilizing systems shall not be affected. The data shall be used by the SC crew and ground stations to monitor and evaluate the performance and operation of the various SC subsystems. The operational instrumentation shall consist of sensors, signal conditioning, timing, distribution panels, and magnetic tape recorders. The operational instrumentation shall be designed to provide the following capabilities:

(a) Sense physical parameters that must be displayed, recorded, or transmitted

(b) Standardize electrical signals for convenient displays, recording, and transmission: the instrumentation shall be capable of furnishing signals for simultaneous recording and transmitting; recording for transmission at a later time; recording only; or transmission only.

(c) Provide an in-flight calibration system to monitor the gross state of the telemetry system

(d) Provide a time reference for all time-dependent SC functions except guidance and navigation

(e) Provide means to conveniently distribute the signals to display, recording, and transmission systems

(f) Provide separate sensor and measurement channels for the display of necessary critical flight safety parameters

(g) Store for future readout that data which cannot be conveniently transmitted in real time

3.3.2.10.2.2 NASA-Furnished Biomedical Instrumentation. - The biomedical instrumentation shall be as described in 3.4.4.8.11.2. The data to be transmitted by PCM telemetry shall consist of one 30 cps physiological measurement and six suit parameters requiring 2 cycles per second (cps) response each. This data shall be time-shared over a single PCM channel.

3.3.2.10.2.3 Scientific Instrumentation Equipment (NASA-Supplied). - All NASA-furnished scientific instrumentation shall be integrated into the SC systems by NAA/S&ID. The weight of this equipment shall not exceed 250 pounds, and the volume shall not exceed 10.0 cubic feet. The power requirements shall be specified when equipment requirements are determined, but shall not in any case exceed nominal power available above other systems requirements, as specified in 3.3.2.9.1.1 and subparagraphs. (See SID 62-1002 for interface requirements.)

3.3.2.10.2.3.1 Photographic Equipment. - The photographic equipment shall consist of one 16-mm motion picture camera and one high-resolution still camera, capable of recording events within the CM and LEM and outside the CM and LEM. The movie camera shall be transferred to the LEM for documentation of crew movement on the lunar surface, and for use outside the LEM on the lunar surface. Auxiliary equipment shall include wide-angle lenses, telephoto lenses, and pistol grips for hand-held operation in the CM, LEM, and on the lunar surface. The weight and volume of the photographic equipment shall be included in the scientific instrumentation equipment allocation.

The still camera shall be designed to be compatible for use with the scientific window in the CM, the telescope in the LEM, and the NASA-furnished microscopic equipment on the lunar surface. This camera and associated film supply shall be capable of recording in the ultra-violet or infra-red portions of the spectrum as well as the visible.

3.3.2.10.2.3.2 Gas Chromatograph. - The gas chromatograph shall consist of a helium gas reservoir and analyzing unit capable of determining the atmospheric contaminants in the Apollo vehicle. Auxiliary equipment shall include a sample inlet manifold, necessary valving, and hard lines for sample sensing, carrier gas transfer, and overload discharge. Telemetry inputs for the gas chromatograph shall be in accordance with 3.3.2.10.1.1.11(a). The weight and volume of the gas chromatograph shall be included in the scientific instrumentation equipment allocation.

3.3.2.11 IFTS. - The IFTS shall be designed to provide a means for the SC crew to manually monitor the CM and SM systems and subsystems, determine their status, and provide fault isolation to modules that are replaceable, repairable, or adjustable. The IFTS shall be designed to

isolate a failure in any of these systems to not more than three modules, and a manual voltmeter check at GSE access test points shall indicate which module is faulty.

3.3.2.11.1 Voltage Comparison. - The IFTS shall be designed to have a 225-channel (test point) capability and be hardwired directly to the SM and CM systems and signal conditioners. Each of these analog voltages shall be presented to a comparator that compares the voltage with a reference voltage generated within the IFTS. If the analog input should deviate from within its preselected limits, an out-of-limits indication shall be presented on the IFTS panel.

3.3.2.11.2 Switching. - Switching circuits shall be provided which can route a selected analog input to the IFTS voltmeter where the magnitude and direction of the failure may be determined. The switching circuits shall simultaneously present the analog input to the Apollo PCM telemetry equipment for direct relay of the failure indication to the ground. The analog signal shall be identified to the PCM telemetry equipment with a binary coded word.

3.3.2.11.3 Voltmeter. - The voltmeter shall be designed to have the capability of measuring test points beyond the physical confines of the IFTS but within the accessible areas of the CM.

3.3.2.11.4 Control and Test. - The IFTS shall be designed to be turned on and off, and shall be capable of verifying its own condition. Provision shall also be made to disable individual indicator lamps.

3.3.2.12 Controls and Displays. - Controls and displays shall be provided as required to permit the crew to control the CM and SM systems and monitor the parameters essential to an acceptable mission performance.

3.3.2.12.1 Major Components. - The compositions of the various control-displays groups shall be as described in this paragraph. Primary control-displays locations shall include the following:

(a) Main Display Console

  1. Control station

  2. Center station

  3. Systems management station

(b) Armrest controls

(c) Lefthand console

(d) Righthand console

(e) Navigation station

(f) NASA-furnished automatic guidance equipment controls and displays.

3.3.2.12.1.1 Main Display Console. - The main display console shall be located above the crew couches in the CM. It shall incorporate three operator stations, composed and located as described below.

3.3.2.12.1.1.1 Control Station. - The control station shall be positioned above the control station operator's couch as oriented in the launch position. The station shall contain the following displays and controls:

Description	Quantity
Barometric pressure indicator	1
Event time indicator	1
Abort indicator	1
Gimbal position indicator and attitude set panel	1
Gimbal motor power control group	1
SCS control panel	1
Flight director attitude indicator unit	1
ΔV control panel	1
Warning indicator group	1
Entry monitor display unit	1
Crew safety system controls	1
Master caution indicator group	1
ELS sequencer group	1

Description	Quantity
Computer control and indicator group (NASA-furnished)	1
LES sequencer group	1

3.3.2.12.1.1.2 Center Station. - The center station shall be located in the center of the main display console above the center couch as oriented in the launch position. The station shall contain the following displays and controls:

Description	Quantity
SPS management panel	1
RCS management panel	1
Abort indicator	1
GMT display	1
SM thermal profile indicator	1
Abort indicator	1
Audio control panel	1
24-hour mechanical clock	1
ECS (liquid) system management panel	1
ECS (gas) system management panel	1

3.3.2.12.2.1.1.3 Systems Management Station. - The systems management station shall be positioned above the systems management station operator's couch as oriented in the launch position. The station shall contain the following displays and controls:

Description	Quantity
Electrical power distribution control group	1
EPS (ac) management group	1
EPS (dc) management group	1
Master caution indicator group	1

Description	Quantity
Warning indicator (carbon dioxide partial pressure high)	1
Fuel cell system management panel	1
Cryogenic system management group	1
Abort indicator	1
Communications control panel	1
IFTS on-off control	1
Antenna control group	1
Event timer	1

3.3.2.12.1.2 Armrest Controls. - Provision to accommodate demountable hand controllers shall be incorporated in the armrests of the control station operator's couch and the center station operator's couch. The controllers shall consist of one 3-axis attitude controller and one 3-axis translation controller for each station. When in use, the attitude controller shall be mounted to the right armrest of each couch and the translation controller to the left armrest. Provision shall be made for stowage of the controllers when not in use.

3.3.2.12.1.3 Lefthand Side Console. - The lefthand side console shall be located to the left of the main display console and shall be accessible to the control station operator. The console shall contain the following displays and controls:

(a) Abort sequencer arming indicators and controls

(b) ELS sequencer arming indicators and controls

(c) Post-loading and recovery aids indicators and controls

(d) Lighting controls, lefthand floodlighting

(e) Audio control panel

(f) SCS power control group. (Energy and condition management only are in this category.)

3.3.2.12.1.4 Righthand Side Console. - The righthand side console shall be located to right of the main display console and shall be accessible to the systems management station operator. The console shall contain the following displays and controls:

(a) Motor and power control switching group

(b) Lighting controls, righthand floodlighting

(c) Audio control panel.

3.3.2.12.1.5 Navigation Station. - The navigation station shall be located in the lower equipment bay, positioned to provide optimum operator access. The station shall contain the following display and controls:

(a) Optical measuring unit controls and viewing access equipment (NASA-furnished)

(b) Automatic guidance equipment displays and controls equipment (NASA-furnished)

(c) Projection film viewer equipment (NASA-furnished)

(d) Event time indicator (2) equipment (NAA-furnished)

(e) Mechanical clock equipment (NAA-furnished)

(f) Master caution indicator group equipment (NASA-furnished)

(g) Lighting controls, work area floodlighting equipment (NAA-furnished).

The navigator station shall incorporate provisions to mount one 3-axis attitude controller as described in 3.3.2.12.1.2, and provisions for mounting the astro-sextant cover crank for the optics door.

3.3.2.12.1.6 NASA-Furnished Automatic Guidance Equipment Controls and Displays. - Controls and displays furnished by the NASA for operation of the automatic guidance equipment shall be incorporated in the main display console. It shall be the responsibility of NAA/S&ID to define acceptable limits of panel dimensions, unit weight and volume, and installation design details for this equipment.

3.3.3 Fail Safe. - System or component failure shall not propagate sequentially; that is, design shall "fail safe."

3.3.4 Flight Loads. -

3.3.4.1 Tumbling at Maximum Dynamic Pressure. - The CM LES combination shall be designed for loads arising from tumbling of the escape vehicle following an abort at maximum dynamic pressure during boost. Tumbling shall be considered in both pitch and yaw planes.

3.3.4.2 Dynamic Loading. - The calculation of dynamic loads shall include the effects of engine start, rebound on the pad, and liftoff transients, including ground winds, gusts, wind shears, buffeting, and longitudinal resonance. The coupling of the structural dynamics with the flight control system shall be included in the determination of dynamic loads.

3.3.5 Checkout and Servicing Requirements. - The SC modules, adapter, and stage simulators shall be designed so that system functional checks may be performed on any desired module.

3.3.5.1 Spacecraft Checkout Provisions. - Provisions shall be made for accomplishing all necessary checkout procedures without the necessity of making changes within the SC. Electrical and fluid systems shall not be disconnected to make test measurements if nondisruptive test points can be provided.

3.3.5.2 Combined System Checking. - Provisions shall be made for combined systems checkout as follows:

(a) At the component and subsystem level

(b) At the major system level

(c) Between systems of the same module

(d) Between systems of different modules.

3.3.6 Human Engineering. - Human engineering for the Apollo program shall be in accordance with Specifications MC 999-0007 and MC 999-0017.

3.4 Performance.-

3.4.1 Mission Performance. - Mission performance shall be as follows:

(a) Flight - The CM and SM systems shall be capable of performing at their nominal design performance level for a mission of 14 days without resupply. For lunar landing missions, 7 of the 14 days may be in lunar orbit. The LEM systems shall be capable of performing at their nominal design performance level for a mission of 2 days without resupply. Consideration should be given to extending the LEM capability to 7 days by employing a supply module or resupply vehicle.

(b) Post flight - The CM shall provide the crew a habitable environment for 1 day and a survivable environment for an additional 2 days, for a total of 3 days protection following a water landing.

(c) Earth landing - The SC shall have the capability, by crew control, of initiating a return and earth-landing maneuver at any time during the lunar landing mission. Prior to each flight, a primary ground landing site and a suitable backup landing site shall be selected for normal mission landing. Additional criteria applies as follows:

1. Lunar missions - Alternate landing sites compatible with SC performance capability shall be designated prior to flight so that a landing is possible under abort contingencies. The required numbers of these sites shall be minimized and their locations optimized.

2. Parking orbit - The SC shall be capable of landing at the primary landing site (or at the backup site) from at least three orbits per day. In addition, alternate sites that may involve either land or water landing shall be designated so that at least one alternate site can be reached for a landing from each orbit.

3.4.2 Flight Plan. - The Apollo program ultimate mission flight plan for which the Apollo SC is designed shall be as specified in 3.4.2.1, 3.4.2.2, and 3.4.2.3. The final flight plan and operational considerations for the Apollo lunar landing mission shall be within the constraints of the design parameters of the SC systems defined in this document. Fulfillment of the final flight plan shall also be subject to consideration of design and operational parameters of the NASA-supplied GOSS and design parameters introduced by associate contractors for the Apollo Program. This section is divided into three parts:

(a) The general flight plan requirements

(b) The lunar landing mission trajectory characteristics

(c) A basic flight plan for the lunar landing mission.

3.4.2.1 General Flight Plan Requirements. - The general flight plan requirements shall be as described in the following sub-paragraphs:

(a) Launch site - All earth orbit and lunar missions shall be launched from Cape Canaveral, Florida. The launch azimuths shall be within the limitations set by range safety and tracking considerations.

(b) Launch time window - Lunar mission flight plans must include at least a 2-hour period on launch date in which the mission can be launched at any time during the period or at discrete intervals.

(c) Parking orbit altitude - The nominal parking orbit altitude shall be 100 nautical miles.

(d) Number of parking orbits - Multiple parking orbits are acceptable, but should be compatible with booster performance and lifetime limitations.

(e) Translunar trajectory evaluation - The time required to determine the translunar trajectory will be dependent on the launch date and the planned GOSS capabilities.

(f) Translunar midcourse corrections - The SC shall include provisions for performing translunar midcourse maneuvers.

(g) Lunar orbit - When applicable, the plane change maneuver shall be accomplished at the same time as the retro maneuver for establishing the lunar orbit. The nominal lunar orbit altitude is approximately 80 nautical miles. A 5-degree plane change capability shall be provided for establishing the initial orbit.

(h) Lunar landing - The lunar landing shall be initiated from a lunar orbit. Alternate LEM descent profile techniques, in addition to the equal period orbit concept, which have bearing on overall SC performance are currently being evaluated by both the NASA and NAA/S&ID. The LEM shall separate from the SC and transfer from the circular orbit to an equal period elliptical orbit that does not intersect the moon's surface. The landing, hovering, and final touchdown maneuvers will be performed by the LEM from the elliptical orbit.

(i) Lunar landing site - Mission plans may call for several lunar landing sites. The following factors shall be considered in the choice of a landing site:

1. Propulsion and fuel requirements
2. Maneuvering and hovering capability
3. Communication with GOSS
4. Illumination
5. Temperature of environment
6. Surface texture
7. Ease of identification
8. Length of lunar exploration period.

(j) Rendezvous and docking - The rendezvous and docking maneuvers shall be accomplished by the LEM with the CM and SM taking corrective action as backup to the LEM propulsion system.

(k) Transearth injection - The SM shall be capable of providing the necessary propulsion performance to transfer from the lunar orbit to the transearth trajectory.

(l) Transearth midcourse corrections - The SC shall include provisions for performing transearth midcourse maneuvers. The inclination of the transearth trajectory to the earth's equator shall be compatible with planned tracking stations.

(m) Entry - The CM shall be capable of entry over a nominal 30-nautical mile corridor with a peak deceleration of 10g. The direction of entry shall be with the rotation of the earth.

(n) Spacecraft orientation - Orientation of the SC in all mission phases shall be accomplished as required for proper operation of vehicle systems.

3.4.2.2 Lunar Landing Mission Trajectory Characteristics. - The lunar landing mission trajectory characteristics are presented in the order of the different phases of flight.

(a) Launch time window - A launch window shall be provided either by maneuvering the vehicle or SC to intercept a planned trajectory or by selecting a new trajectory that will satisfy the mission objectives and which will also be obtained at the actual launch time.

Both the lunar trajectory selection and vehicle maneuvering methods shall be developed for obtaining a launch window. This discussion of launch window is limited to lunar trajectory variations. To a first order of approximation, the SC shall be injected into a lunar trajectory from a parking orbit. The parking orbit passes over the earth's surface point, which is formed by projecting the line of centers between the earth and the moon at the time of the closest initial approach of the SC to the moon. The launch window is primarily a function of the permissible azimuth swing for a launch from Cape Canaveral. Figure 28 shows the launch time window and the approximate inclination of the parking orbit as a function of azimuth variations, positive and negative from due east launches from Cape Canaveral. This launch window is independent of lunar declination and can be obtained for lunar injection toward either the south or north.

(b) Parking orbits - A nominal value of earth orbit altitude for lunar missions is 100 nautical miles. The effect of launch delays on the earth track of the parking orbit and the location of the injection point is shown in Figure 29. The extreme orbit paths for a 4-hour launch window are shown as the outside solid lines for the condition where the launch azimuth variation is symmetrical about a due east launch from Cape Canaveral. The inner broken lines show the extremes for a 2-hour launch window. The launch window results from the use of any trajectory between these extremes. The location of the injection points for certain lunar declinations is shown in Figure 29.

(c) Injection - The characteristic velocity requirements for injection into free-return translunar trajectories from a 100-nautical mile earth parking orbit are shown in Figure 30 as a function of the free-return translunar and transearth inclinations to the lunar orbit plane.

(d) Translunar trajectory characteristics - The nominal translunar trajectory for early lunar missions is one which has a coast return to the earth with acceptacle entry conditions. The translunar trajectories for lunar landing missions should have a nominal pericynthion of 80 nautical miles. The inclination of the translunar trajectory plane is a function of the launch azimuth and the moon's position from the ascending node as shown in Figure 31. Varying the inclination of the parking orbit to obtain launch time tolerance, as indicated in this section, will result in a change in the translunar trajectory inclination, which in turn, will have some effect on the inclination of the lunar conic unless plane changes are made during transfer.

The translunar trajectory is tracked with the deep space network. Figure 32 shows the coverage of the existing deep space network at various altitudes. About 15 minutes after injection, the translunar trajectory will be at high enough altitude for tracking.

(e) Lunar orbit - Flight plans require establishing a circular lunar orbit and should nominally be 80 nautical miles altitude for lunar landing missions. For lunar orbit missions, flight plans may call for both circular and elliptical orbits within the limits of propulsion requirements for the 80-nautical mile circular orbit. Impulsive velocity increments for establishing lunar orbits are shown in Figure 33.

(f) Lunar landings - A technique for lunar landing is illustrated in Figure 39. The SC arrives behind the moon on a circumlunar trajectory, a transfer is made to an 80-nautical mile circular orbit about the moon. The SC passes over the landing area once. At the proper position in the orbit, the LEM separates from the SC, and a transfer is made from the circular orbit to an equal period elliptical orbit having a pericynthion of 50,000 feet. The landing run is initiated at 50,000 feet altitude.

(g) Lunar landing site - The lunar landing areas available for free-return missions with translunar trajectories which return to earth with posigrade entry are limited to a band of relative latitudes of approximately ±13 degrees about the orbital plane of the moon. At times during the month, latitudes as high as 19 degrees with respect to the lunar equator plane are obtainable.

(h) Lunar launch - A technique for launch of the LEM from the moon on the return to the CM and SM is to lift off in an essentially vertical maneuver from the local surface and program pitch into an elliptical orbit. The orbit is circularized at apocynthion and rendezvous and docking with the SC initiated immediately. Parking orbits may be required before the transfer and rendezvous occurs.

(i) Transearth - The inclination and the time of flight of the transearth trajectory are used to control the entry in such a way that the entry track will be over planned network facilities and traverse reasonable recovery areas. Inclination of approximately 30 degrees would be compatible with existing facilities and with landing sites in southern Texas, Hawaii, and Australia. The nominal entry for Apollo missions is to be with posigrade motion with respect to the earth to reduce the entry heating and widen the entry corridor.

(j) Entry - The nominal 10g entry corridor for L/D (L/D =0.5) is 38 ± 3 nautical miles. The entry corridor is defined as either the difference in entry path angles or by the difference in perigee radius for those two approach conics which, for a given entry maneuver and vehicle aerodynamic characteristic, result in:

1. a limiting acceleration (undershoot)
2. arrival at a capture boundary (overshoot) during the initial entry phase.

Entry control techniques and sensing uncertainties reduce the corridor to 31 ± 3 nautical miles. Atmospheric variations are latitude-dependent, and if the NASA low-latitude idealized extremes are assumed, the corridor loss is 3 nautical miles. Tolerances in trim L/D (5 percent) reduce the corridor by 1 nautical mile. Thus, for nominal missions where entry is restricted to low latitudes, the maximum entry corridor is 27 ± 3 nautical miles. The locus of the entry point for returning to the primary landing site must lie within 5,000 nautical miles of the landing site.

3.4.2.3 Basic Flight Plan for Lunar Landing Mission. - The following flight plan is presented in the order of the different phases of flight required to accomplish the basic lunar landing mission:

(a) Liftoff conditions - The launch azimuth is 91 degrees.

(b) Parking orbit - Ground tracks for the initial earth orbit having a launch azimuth of 91 degrees are presented in Figure 35. The parking orbit is circular at an altitude of 100 nautical miles.

(c) Parking orbit to translunar - The location of the beginning of the insertion phase may be anywhere along the parking orbit, depending upon the moon's declination. Figure 35 shows earth tracks for an insertion location in the mid-Pacific region.

(d) Translunar and transearth - Figure 36 presents the translunar and transearth trajectories for the inertial earth-moon system. The translunar trajectory shall be such that if no velocity increment is applied, the SC will return to earth at acceptable entry conditions. The pericynthion altitude at the moon is 100 nautical miles. The return of transearth trajectory shown in Figure 37 represents a continuation of the translunar trajectory with a break of 25 hours for landing on the moon and takeoff. The transearth and translunar trajectories combined form a reference circumlunar trajectory with proper correction for the lunar time break.

(e) Lunar orbit - A velocity increment is applied to the SC when approaching pericynthion altitude so that a circular lunar orbit is achieved which will pass over the desired landing site. The landing site is surveyed as the SC passes over this area during its first revolution. For one type of descent profile, as the SC approaches a point nominally 180 degrees from the landing site, the LEM and crew will be separated from the CM to await transfer to an elliptical orbit. As the SC approaches a point 90 degrees from the landing site, the LEM propulsion system provides a velocity impulse to attain the desired elliptical orbit with a pericynthion altitude of 50,000 feet.

(f) Lunar landing - The lunar landing maneuver is initiated at 50,000 feet altitude. The characteristics of the flight plan during the maneuver are presented in Figures 38 and 39. The maneuver ends at an altitude of 1,000 feet, at which time the LEM vertical and horizontal velocity are near zero. The LEM shall have the capability to hover at this altitude and translate 1,000 feet prior to landing.

(g) Lunar launch - The characteristics of the flight plan from lunar takeoff to insertion into the elliptical orbit are presented in Figure 40. Transfer of the LEM from its elliptical orbit to CM and SM circular orbit will be accomplished by the LEM propulsion system.

(h) Rendezvous - The LEM will provide the necessary propulsion performance to transfer from the elliptical orbit to the circular orbit to accomplish the rendezvous maneuver with the CM and SM.

(i) Launch from lunar orbit - The transfer from the 80-nautical mile circular orbit into the transearth trajectory is accomplished by application of a velocity increment at insertion. This velocity increment is provided by the SPS.

(j) Transearth - The transearth trajectory is presented in Figure 37.

(k) Entry - The return perigee altitude is 120,000 feet, the velocity at perigee is 36,320 feet per second, and the reference entry altitude is 400,000 feet. A ground track of the transearth and entry phase of the flight plan is shown in Figure 37. The possible landing area extends along the indicated track, but will be limited to a maximum range of 5,000 nautical miles from the entry point.

3.4.3 Spacecraft Performance. - The following sub-paragraphs summarize the nominal performance capabilities of the CM, SM, LEM, and adapter.

3.4.3.1 Boost Stabilization. - Windage, aerodynamic effects, variations of the center of gravity, etc., will be compensated for by the launch vehicle during the boost phase.

3.4.3.2 Communications. - During all mission operations (except on the far side of the moon), ground communications with, and tracking of, the CM and SM shall be accomplished through the GOSS network.

3.4.3.3 Trajectories. - The general SC trajectories shall be as specified in 3.4.2.2. After translunar injection, the primary measured SC positional accuracy shall be provided by the G&N.

3.4.3.4 Propulsion. - Propulsion increments involved with the ascent, injection into parking orbit, and translunar injection will be supplied by a NASA-furnished Saturn V launch vehicle with the following propulsion capabilities:

Stage	Thrust (pounds)	Specific Impulse (sec)
S-IC	7,500,000 sea level	263.58 sea level
S-II	1,000,000 vacuum	424 vacuum
S-IVB	200,000 vacuum	424 vacuum

After S-IVB separation, major increments of the SC shall normally be supplied by the SPS. The characteristic velocity for each maneuver will vary from mission to mission. However, for mission comparison purposes, load reports, etc., the characteristic velocity budget utilized shall be as indicated during the following mission phases:

	Mission Phase	Incremental Velocity (fps)*
(a)	Translunar midcourse	330
(b)	Lunar orbit injection (including plane change)	3,553
(c)	Lunar maneuvers (including LEM rescue)	618
(d)	Transearth injection	3,853
(e)	Transearth midcourse	330

*Calculated for 80-nautical mile circular orbit, equal period orbit concept with 10-percent margin. Equivalent to the NASA total translunar and transearth ΔV values.

3.4.3.5 Launch Vehicle. - The Saturn V launch vehicle will be the basic launch vehicle for lunar missions. The SC will be injected into an earth parking orbit by a Saturn V capable of achieving a 90- to 400- nautical mile altitude orbit, within safe boost trajectory limits, dependent upon the preprogrammed mission profile.

3.4.3.6 Mission Contingencies. - Contingencies may arise during the course of the mission as a result of human error, equipment failure, unexpected hazards encountered in a hostile environment, or a combination of these elements. Operational procedures and contingency provisions shall be established for the categories that are described in the following subparagraphs.

3.4.3.6.1 Contingency Categories. - Mission contingencies shall be categorized as follows:

(a) Rectifiable

(b) Tolerable

(c) Mission abort.

3.4.3.6.1.1 Rectifiable Contigency. - A rectifiable contingency is a condition that can be corrected without jeopardy to the crew and with no degrading effect on the mission objective.

3.4.3.6.1.2 Tolerable Contingency. - A tolerable contingency is a condition that cannot be corrected. The condition will not jeopardize the crew or degrade the mission objective to an unacceptable level. An alternate mission may be selected when a tolerable contingency occurs.

3.4.3.6.1.3 Mission Abort Contingency. - A mission abort contingency is a condition that cannot be corrected and will be hazardous to the crew or will degrade the mission objective to an unacceptable level. When a mission abort contingency arises, the mission shall be aborted in one of the following ways:

(a) Planned abort - The contingency is such that a normal return to earth through a specified entry corridor to a designated landing site is possible.

(b) Controlled emergency abort - The contingency requires a short time return to a specific landing site using any entry corridor. The contingency may arise from equipment failure having a predictable decay rate.

(c) Critical emergency abort - The contingency requires that the CM be returned to any point on earth as soon as possible, under maximum stress conditions if necessary.

3.4.3.6.2 Operational Procedures. - Necessary monitoring equipment shall be provided in the SC for control equipment to permit the crew (SC or ground) to identify contingencies and to take appropriate action. Operational capabilities shall be provided for inflight surveillance of all SC systems that are vital to crew safety and mission success. Surveillance functions shall include:

(a) Detection of contingency conditions

(b) Isolation and evaluation of contingency

(c) Selection of alternate flight or operation modes

(d) Operational transfer to redundant equipment.

3.4.3.6.2.1 Maintenance. - Maintenance during flight shall be limited to those systems accessible to the crew. Maintenance shall include:

(a) Checkout, service, and calibration

(b) Manual adjustment

(c) In-place repairs

(d) Repair by replacement.

3.4.3.6.2.2 Detection of Contingency Conditions. - Monitoring equipment shall be provided for crew surveillance of the following:

(a) Earth-moon space conditions (also ground monitoring control)

(b) SC vehicle operating conditions

(c) Verification of SC systems performance parameters

(d) SC systems malfunction indications.

3.4.3.6.2.3 Isolation and Evaluation of Contingency. - After inital indications of a contingency, planned sequential steps shall be taken by the crew to isolate, evaluate, and rectify or otherwise nullify the cause of the contingency.

3.4.3.6.2.4 Checkout, Service, and Calibration of Equipment. - The crew shall make periodic checkout, servicing, and calibration of the SC operational systems. Effort shall be made to correct deficiencies discovered by these checks.

3.4.3.6.2.5 Manual Adjustment. - The crew shall make manual adjustment of all SC regular or emergency operational controls necessary to correct contingencies.

3.4.3.6.2.6 Selection of Alternate Flight or Operational Modes. - The crew shall consider an alternate flight or operational mode to correct contingencies.

3.4.3.6.2.7 Operational Transfer to Redundant Equipment. - Where provision is made for installation of redundant SC systems, immediate control switchover capability by the crew shall be provided to maintain SC performance continuity.

3.4.3.6.2.8 In-Flight Repairs. - Provisions shall be made for on-board supply of electronic component parts and other small physical hardware. The tools necessary for removal and installation to permit repair of deficient or defective parts by the crew during flight shall also be provided. The crew shall make repairs when considered necessary by indications of possible malfunctions from the control performance monitoring equipment or from visual observation of a possible deficiency by the crew.

3.4.3.6.2.9 Repair by Replacement. - Effort shall be made to repair by replacement when necessary to rectify possible CM systems performance deficiencies.

3.4.3.6.2.10 Verification of Contingency Corrective Measures. - In all cases where any or several contingency corrective measures have been taken by the crew during a mission, subsequent verification of the adequacy of the corrective measures shall be made where possible. Continued observation of the corrected contingency shall be maintained until the conclusion of the mission.

3.4.3.6.3 Mission Contingency Provisions. -

3.4.3.6.3.1 Ascent. - During ascent, the LES shall be capable of separating the CM from the launch vehicle if a mission abort contingency occurs prior to jettison of the LES. The LES shall be capable of being operated through the first stage boost and for the initial few seconds of second stage boost; then the escape system shall automatically be jettisoned. Abort procedures after the jettisoning of the LES shall be accomplished by using the SM propulsion system. During a launch pad abort, the crew shall initiate action to propel the CM to a minimum altitude of 4,000 feet and a lateral distance at apogee of at least 3,000 feet without exceeding emergency crew tolerances or SC structural limits. Parachutes shall be used to lower the crew in the CM safely to earth.

3.4.3.6.3.2 Parking Orbit. - During the parking orbit phase, termination shall result from any abort contingency. The dynamic requirements for abort from earth orbit shall be identical to orbit ejection. The geometric requirements shall be dependent upon the type of abort and the geographic location of acceptable landing sites.

3.4.3.6.3.3 Translunar Coast. - An abort from this phase shall be dependent upon the type of contingency.

3.4.3.6.3.4 Lunar Vicinity. - A mission abort resulting from a contingency in the vicinity of the moon will impose severe performance requirements on the SC propulsion system, especially for the lunar orbital mission.

3.4.4 System Performance. -

3.4.4.1 G&N (NASA-Supplied). -

3.4.4.1.1 Operating Limits. -

3.4.4.1.1.1 Electrical Power. - The G&N shall be capable of operation within design limits from the 28 vdc SC electrical power specified in 3.4.4.10.1.1.

3.4.4.1.2 Performance Characteristics. - The G&N system shall be capable of the following:

(a) Providing a primary inertial reference

(b) Acceleration, velocity, and position determination

(c) Guidance and navigation computation and predication

(d) Providing rotational and translational control inputs to the SCS

(e) Providing rendezvous guidance and control in the event of LEM system malfunction in flight precluding primary mode of rendezvous

(f) Providing crew controls and displays for guidance and navigation and data to GOSS for ground-based comparison with inflight data.

3.4.4.1.3 Primary Modes and Functions. - The G&N shall be capable of performing SC guidance and navigation functions as required during the following flight phases:

(a) Lunar orbit injection

(b) Lunar orbit

(c) Transearth injection

(d) Midcourse ΔV corrections

(e) Transearth and translunar coast

(f) Entry

(g) Abort.

3.4.4.2 SCS. -

3.4.4.2.1 Operating Limits. -

3.4.4.2.1.1 Electrical Power. - The SCS shall be capable of operation within design limits when supplied with the following power:

(a) DC power - 146 watts maximum, 23 kilowatt-hours (336 hours)

(b) AC power - 180 watts maximum, 45 kilowatt-hours (336 hours)

The tolerance shall be as specified in 3.3.2.9.1.5.

3.4.4.2.2 Performance Characteristics. -

3.4.4.2.2.1 Atmospheric Abort. - The SCS shall be capable of rate stabilization of the CM after the LES tower has been jettisoned.

3.4.4.2.2.2 Extra Atmospheric Abort. - The SCS shall be capable of orientation, attitude control, and entry stabilization control. The SCS shall accept commands from the guidance system for thrust vector control and entry control.

3.4.4.2.2.3 Translunar and Transearth. - The SCS shall be capable of stabilization and control during midcourse flight both outbound and inbound. The control technique shall provide fuel economy and shall satisfy all navigation requirements, as well as antenna-pointing requirement. Attitude control and orientation for application of midcourse corrections shall be provided.

(f) The automatic attitude command source shall be the G&N in all G&N modes.

(g) The automatic attitude command source shall be the SCS in all SCS modes.

(h) Manual rate commands shall be provided in the following:

1. G&N attitude control mode
2. SCS attitude control mode
3. SCS local vertical mode
4. SCS entry mode
5. Monitor mode.

(i) Minimum impulse commands shall be provided in the following:

1. SCS attitude control mode
2. G&N attitude control mode.

3.4.4.2.3.2 Monitor Mode. - The monitor mode shall be used when the launch vehicle is in control. Attitude and attitude errors shall be displayed from the G&N and body rates shall be displayed from the SCS. The mechanization of this mode shall be such that the system shall always be ready to accommodate requirements for abort.

3.4.4.2.3.3 G&N Attitude Control Mode. - The G&N attitude control mode shall be used during midcourse and orbital phases when the G N is in control of the SC just prior to switching to a G&N ΔV mode. Attitude error, attitude, and body rate information shall be displayed. Manual input and override capability shall be determined at a later date.

3.4.4.2.3.4 SCS Attitude Control Mode. - The SCS attitude control mode shall be used during midcourse and orbital phases. Attitude error, attitude, and body rate information shall be displayed. Manual input and override capability shall be determined at a later date.

3.4.4.2.3.5 SCS Local Vertical Modes. - The SCS local vertical mode shall be used during orbital phases when the astronaut desires to display attitude, hold attitude, and maneuver with respect to local vertical of the near body.

3.4.4.2.3.6 G&N ΔV Mode. - The G &N ΔV mode shall be used as the primary mode whenever velocity changes are to be made with the SPS. Velocity change, attitude error, attitude, and body rate information shall be displayed. Manual input and override capabilities shall be determined at a later date.

3.4.4.2.3.7 SCS ΔV Mode. - The SCS ΔV mode shall be used as the backup mode whenever velocity changes are to be made with the SM SPS and as the primary mode whenever velocity changes are to be made with the SM RCS. Velocity change, attitude error, attitude, and body rate information shall be displayed. Manual inputs and override capabilities shall be included where required.

3.4.4.2.3.8 G&N Entry Mode. - The G&N entry mode shall be used as the primary entry mode. Attitude error commands in roll only shall be supplied from the G&N coupling display unit (CDU), and rate stabilization only shall be provided in all three axes until the 0.05 g switching within the SCS occurs. Manual override capabilities shall be provided.

3.4.4.2.3.9 SCS Entry Mode. - The SCS entry mode shall be used as the backup entry mode. The astronaut shall use the backup entry display as he controls the SC on a survival trajectory with manual roll commands. Automatic 3-axis attitude control shall be provided until the 0.05 g switch point. Following the 0.05 g switch point, rate stabilization shall be provided in all three axes. Manual rate command capability shall be provided in roll following the 0.05 g switch point.

3.4.4.3 SPS. -

3.4.4.3.1 Operating Limits. - The following paragraphs define the operating limitations for the SPS.

3.4.4.3.1.1 Power Supply. - All electrically actuated system components, with the exception of the propellant quantity-indicating system, shall be capable of operation from the dc power supply specified in 3.4.4.10.1.1. The quantity-indicating system may operate from the dc power supply, the dc and ac power supplies combined, or the ac power supply specified in 3.4.4.10.1.2.

3.4.4.3.1.2 Engine System. - Utilizing propellants at the inlet conditions specified in 3.3.2.3.3, the rocket engine system shall be capable of achieving the following vacuum performance ratings within the limits indicated.

3.4.4.3.1.2.1 Operating Life. - The operating life for the engine shall be a minimum of 750 seconds, without removal for servicing.

3.4.4.3.1.2.2 Thrust. - Initial engine thrust (after 30 seconds) shall be 21,900 pounds ±1 percent and terminal engine thrust (after 750 seconds) shall be 21,900 pounds +10, -1 percent.

3.4.4.3.1.2.3 Specific Impulse. - The minus three sigma value of specific impulse for the engine shall be a minimum of 313 seconds at any time during the operating life of the engine (750 seconds).

3.4.4.3.1.2.4 Duty Cycle. - The engine shall be capable of continuous operation for a period of 635 seconds maximum and a minimum impulse bit of 5,000 ± 200 pound-seconds.

3.4.4.3.1.2.5 Frequency of Operation. - The engine shall be capable of satisfactory operation at any time during a 45-day period subsequent to a non-firing functional checkout.

3.4.4.3.1.2.6 Engine Start. - The engine shall be capable of sustaining a minimum of 50 start cycles during the operational life of 750 seconds.

3.4.4.3.1.2.7 Propellant Supply Temperature. - The engine shall be capable of operating with propellants at the following temperatures (fuel temperature to be within ± 10 degrees F of oxidizer temperature):

(a)	Oxidizer -	30 to 135 degrees F
(b)	Fuel -	40 to 145 degrees F

3.4.4.3.2 Performance Characteristics. - The following paragraphs define the performance characteristics of the SPS.

3.4.4.3.2.1 System Operation. - Engine start and stop shall be signalled by electrical command either from the SC SCS or from crew-controlled override provisions in the CM. The engine propellant valves shall be arranged in series-parallel redundant pairs, to permit engine start or stop in the event of a single malfunction. Propellant flow from the pressurized propellant supply and distribution subsystems shall be divided into parallel oxidizer and fuel lines just upstream of the propellant valve inlets. Fuel shall be used as the operating medium for the valves. The electrically driven gimbal actuators may be controlled by either automatic electrical command from the SC SCS or from manual override crew control in the CM. The redundant electric motors, provided in each actuator, shall be capable of switchover in the event of motor malfunction. Engine position shall be indicated in the CM for crew monitoring.

3.4.4.3.2.2 Engine Chamber Pressure. - During periods of normal operation, the engine chamber pressure shall be 100 psia nominal.

3.4.4.3.3 Primary Modes and Functions. - Upon receiving thrust control signals from the SCS or the manual override controls, the SPS shall be capable of providing the propulsion impulse function, as required, for accomplishment of the normal and emergency SC maneuvers outlined below.

3.4.4.3.3.1 Post-Atmospheric Abort. - Mission abort, during the post atmospheric portion of the launch trajectory, shall be accomplished by the SPS. After the LES has been jettisoned and separation from the boost vehicle has occurred, the SPS shall be capable of providing the velocity increments necessary for SC escape from the boost vehicle, earth orbit injection (if required), and landing point selection.

3.4.4.3.3.2 Earth-Orbit Injection. - The SPS shall be capable of providing (after boost vehicle separation) the propulsion impulse necessary for injection of the SC into a circular earth orbit at low altitude.

3.4.4.3.3.3 Earth-Orbit Transfer. - During the earth-orbital mission, the SPS shall be capable of providing the velocity increment required for transfer of the SC from a low altitude orbit to an orbit at higher altitude. The SPS shall also be capable of providing the propulsion necessary for transfer of the SC from one earth-orbital plane to another.

3.4.4.3.3.4 Earth-Orbit Retrograde. - The SPS shall be capable of providing the SC retrograde velocity required for entry of the CM from the earth-orbital mode.

3.4.4.3.3.5 Trans-Lunar Injection Abort. - Mission abort, during injection of the SC into a translunar trajectory, shall be accomplished by the SPS. After separation from the boost vehicle, the SPS shall be capable of providing the velocity increments necessary for SC escape from the boost vehicle.

3.4.4.3.3.6 Translunar Trajectory Abort. - For an early earth return mission abort during the translunar trajectory, the SPS shall be capable of providing the velocity increments required for injection of the SC into a transearth trajectory and for landing point selection.

3.4.4.3.3.7 Major Midcourse Velocity Corrections. - The SPS shall be capable of providing the velocity increments required when a major midcourse velocity correction for the SC is necessary during either the translunar or transearth trajectory.

3.4.4.3.3.8 Lunar-Orbit Injection and Ejection. - The SPS shall be capable of providing the velocity increments required for injection of the SC into a circular lunar orbit and for SC ejection from the lunar orbit into a transearth trajectory.

3.4.4.3.3.9 Lunar Orbital Maneuvers. - The SPS shall be capable of providing the velocity increments necessary for transfer of the SC from one lunar orbital plane to another. The SPS shall also be capable of providing the velocity necessary for transfer of the SC from one lunar orbit to another and the major velocity increments necessary for SC rendezvous with the LEM.

3.4.4.4 CM RCS. -

3.4.4.4.1 Operating Limits. - The following paragraphs define the operating limits for the CM RCS.

3.4.4.4.1.1 Power Supply. - Each electrically actuated system component shall be capable of operating from the dc power supply specified in 3.4.4.10.1.1.

3.4.4.4.1.2 Engine System. - Utilizing propellants at the inlet conditions specified in 3.4.4.4.1.1, each rocket engine system shall be capable of achieving the following vacuum performance ratings within the limits indicated.

3.4.4.4.1.2.1 Operating Life. - The operating life of the engine shall be a minimum of 200 seconds without removal for servicing.

3.4.4.4.1.2.2 Thrust. - The engine shall be capable of developing a thrust of 91 pounds ±3 percent.

3.4.4.4.1.2.3 Specific Impulse. - The rocket engine shall be capable of developing a specific impulse of at least 265 seconds when operating for periods in excess of 800 milliseconds.

3.4.4.4.1.2.4 Engine Chamber Pressure. - During continuous operation, the engine chamber pressure shall be 150 psia nominal.

3.4.4.4.1.2.5 Minimum Impulse. - The rocket engine shall be capable of providing a minimum impulse of 2 pounds-second.

3.4.4.4.1.2.6 Duty Cycle. - The engine shall be capable of operating at a maximum of 30 cycles per second with a minimum shutdown period of 10 milliseconds between cycles. The engine shall also be capable of continuous operation for a period not to exceed 30 seconds.

3.4.4.4.1.2.7 Engine Start. - The engine shall be capable of sustaining a minimum of 3,000 start cycles during the operational life of 200 seconds.

3.4.4.4.1.2.8 Propellant Supply Temperature. - The engine shall be capable of operating with propellants at the following temperatures (fuel temperature to be with ±5 degrees F of oxidizer temperature):

(a) Oxidizer - 40 to 100 degrees F

(b) Fuel - 40 to 100 degrees F

3.4.4.4.2 Performance Characteristics. - The following paragraphs define the performance characteristics of the CM RCS.

3.4.4.4.2.1 System Operation. - The CM RCS shall employ four engines for roll control, four engines for pitch control, and four engines for yaw control, around the x, y, and z axes respectively. The rocket engines shall be paired to effect rotation about the desired axis with either single or dual system operation. Engine start and stop shall be signalled by electrical command either from the SC SCS or from crew-controlled override provisions in the CM. Propellant flow from the pressurized positive expulsion propellant tanks to the combustion chamber of each engine shall be controlled by the single oxidizer and fuel propellant valves. One of the two coils provided in each valve shall be operated automatically by the SCS, the other shall be operated, as required, by the CM manual override crew controls.

3.4.4.4.3 Primary Modes and Functions. - Upon receiving thrust control signals from the SCS or the rotational hand controller, the CM RCS shall be capable of providing the propulsion impulse function, as required, for accomplishment of the normal and emergency SC maneuvers outlined below.

3.4.4.4.3.1 Normal Entry. - After separation of the CM from the SM and prior to encountering aerodynamic forces and moments, the CM RCS shall be capable of providing three-axes rotational and attitude control to orient and maintain the CM in the entry attitude. During entry, the CM RCS shall be capable of providing the torque required to control roll attitude and to damp roll, pitch, and yaw rates.

3.4.4.4.3.2 Hi-Altitude Abort. - After LES separation during the hi-altitude abort mode, the CM RCS shall be capable of providing the three-axes rate damping torque required to control the CM motion prior to landing system deployment. For any other mission abort mode after LES tower jettison, the CM RCS shall be capable of providing the control functions as required during a normal entry.

3.4.4.5 SM RCS. -

3.4.4.5.1 Operating Limits. - The following paragraphs define the operating limitations for the SM RCS.

3.4.4.5.1.1 Power Supply. - All electrically actuated system components, with the exception of the propellant quantity-indicating system, shall be capable of operating from the dc power supply specified in 3.4.4.10.1.1. The quantity-indicating system may operate from the dc power supply, the dc and ac power supplies combined, or the ac power supply specified in 3.4.4.10.1.2.

3.4.4.5.1.2 Engine System. - Utilizing propellants at the inlet conditions specified in 3.4.4.5.1.1, each rocket engine system shall be capable of achieving the following vacuum performance ratings within the limits indicated.

3.4.4.5.1.2.1 Operating Life. - The operating life of the engine shall be a minimum of 1,000 seconds, without removal for servicing.

3.4.4.5.1.2.2 Thrust. - The engine shall develop a thrust of 100 pounds ± 5 percent.

3.4.4.5.1.2.3 Specific Impulse. - The rocket engine shall develop a specific impulse of 300 seconds when operating for periods in excess of 1 second.

3.4.4.5.1.2.4 Minimum Impulse. - The rocket engine shall be capable of providing a minimum impulse of 0.6 pound-second.

3.4.4.5.1.2.5 Duty Cycle. - The engine shall be capable of operating at a maximum of 25 cycles per second and of starting 10 milliseconds after receiving a shutdown signal. The engine shall also be capable of continuous operation for a period not to exceed 60 seconds.

3.4.4.5.1.2.6 Engine Start. - The engine shall be capable of sustaining a minimum of 10,000 start cycles during the operational life of 1,000 seconds.

3.4.4.5.1.2.7 Propellant Supply Temperature. - The engine shall be capable of operating with propellants at the following temperatures (fuel temperature to be within ± 5 degrees F of oxidizer temperature):

(a) Oxidizer - 40 to 100 degrees F

(b) Fuel - 40 to 100 degrees F.

3.4.4.5.2 Performance Characteristics. - The following paragraphs define the performance characteristics of the SM RCS.

3.4.4.5.2.1 System Operation. - The SM RCS shall employ eight rocket engines for roll control, four engines for pitch control, and four engines for yaw control. Control in each axis shall be supplied by either single or coupled engines with each engine being supplied by the propellant distribution system of the particular module in which the engine is incorporated. Engine start and stop shall be signalled by electrical command either from the SC SCS or from crew-controlled override provisions in the CM. Propellant flow from the pressurized positive expulsion propellant tanks to the combustion chamber of each engine shall be controlled by the single oxidizer and fuel propellant valves. One of the two coils provided in each valve shall be operated automatically by the SCS; the other shall be operated, as required, by the CM manual override crew controls.

3.4.4.5.2.2 Post-Atmospheric Abort. - The SM RCS shall be capable of providing the velocity increment required for SC separation from the boost vehicle, prior to activation of the SPS during a post-atmospheric abort mode.

3.4.4.5.2.3 Orbital Corrections. - The SM RCS shall be capable of providing the minor velocity increments necessary for correction of the orbit, after SC injection into an earth lunar or lunar orbit.

3.4.4.5.2.4 Translunar Injection Abort. - The SM RCS shall be capable of providing the velocity increment required for SC separation from the boost vehicle, prior to activation of the SPS during a translunar injection abort mode.

3.4.4.5.2.5 Translunar Trajectory Abort. - For an early earth return mission abort during the translunar trajectory, the SM RCS shall be capable of providing the three-axes attitude control required for reorientation of the SC prior to injection into a transearth trajectory.

3.4.4.5.2.6 Spacecraft Separation From Boost Vehicle - LEM. - The SM RCS shall be capable of providing the velocity increment required for SC separation from the boost vehicle - LEM configuration, after injection of the SC into a translunar trajectory.

3.4.4.5.2.7 LEM Transposition and Docking. - The SM RCS shall be capable of providing the velocity increments and the three-axis attitude control required for transposition and docking of the SC to the LEM, upon completion of separation from the boost vehicle - LEM configuration during the translunar trajectory.

3.4.4.5.2.8 Spacecraft-LEM Separation From Boost Vehicle. - The SM RCS shall be capable of providing the velocity increment required for SC-LEM separation from the boost vehicle, after transposition and docking of the SC to the LEM during the translunar trajectory.

3.4.4.5.2.9 Minor Midcourse Velocity Corrections. - The SM RCS shall be capable of providing the velocity increments required when a minor midcourse velocity correction for the SC is necessary during either the translunar or transearth trajectory.

3.4.4.5.2.10 Lunar Orbital Maneuvers. - The SM RCS shall be capable of providing the minor velocity increments necessary for SC rendezvous with the LEM.

3.4.4.5.2.11 CM-SM Separation. - The SM RCS shall be capable of providing the velocity increment required for separation of the CM and SM prior to the entry mode.

3.4.4.5.3 Primary Modes and Functions. - Upon receiving thrust control signals from the SCS or the rotational/translational hand controller, the SM RCS shall be capable of providing the propulsion impulse function for accomplishment of the normal and emergency SC maneuvers outlined below.

3.4.4.5.3.1 SPS Maneuvers. - The SM RCS shall be used, in conjunction with the SPS, for those maneuvers in which the SPS is required for major changes in velocity (see 3.4.4.3.3). The SM RCS shall be capable of providing the initial, positive acceleration force required for settling of SPS propellants prior to a zero g SPS engine start. In addition, the SM RCS shall be capable of providing the thrust vectors required for SC roll control during periods in which the SPS is active.

3.4.4.5.3.2 Stabilization and Control. - During a lunar mission, the SM RCS shall be capable of providing the thrust vectors required for three-axis stabilization and attitude control of the SC in all flight phases after translunar injection and prior to CM-SM separation. These flight phases include translunar and transearth trajectories and lunar orbits. In addition, the SM RCS shall be capable of providing three-axes stabilization and attitude control of the SC during earth-orbital missions.

3.4.4.6 LES. -

3.4.4.6.1 Operating Limits. - The following paragraphs define the operating limits for the LES.

3.4.4.6.1.1 Power Supply. - The LES shall be capable of operating from the power supplied by the EPS during all launch pad or atmospheric aborts.

3.4.4.6.2 Performance Characteristics. - The following paragraphs define the performance characteristics of the LES.

3.4.4.6.2.1 System Optimization. - System performance at all possible abort conditions shall be optimized to obtain maximum probability of crew safety.

3.4.4.6.2.2 System Operation. - The crew shall be responsible for the initiation of the LES operation. There will be no responsibility assigned to ground control or automatic systems unless there is insufficient time or information for crew action.

3.4.4.6.2.3 Pad or Boost Abort. - Thrust generated by the launch escape and pitch control motors shall provide for the pad or boost abort operations as defined in 3.4.4.6.3.1 and 3.4.4.6.3.2.

3.4.4.6.2.4 Escape System Jettison. - After successful ignition of the second-stage booster during normal flights, the launch escape tower shall be separated and laterally translated from the space vehicle by the tower jettison motor. Redundant electrical initiators shall be capable of providing for the tower jettison operation. In the event of an abort mode, the LES shall be jettisoned from the CM and propelled away from the CM prior to the initiation of earth landing operations. In the event of a jettison motor failure, the launch escape motor shall be available for system jettison.

3.4.4.6.3 Primary Modes and Functions. - The primary function of the LES shall be to provide an abort capability throughout countdown, first stage boost, and for the first few seconds of second stage burning.

3.4.4.6.3.1 Launch Escape Motor. - A capability for a ±1.0 degree adjustment to the thrust vector alignment shall be provided for the launch escape motor. Means for accurate alignment of the thrust vector relative to the center of gravity shall be provided. The nominal characteristics for the LES motor are as follows:

Resultant Thrust	Resultant Total Impulse
155,000 pounds nominal	515,000 pounds seconds nominal at sea level, 70 degrees F
	(Over first 2.0 seconds of burning at 3,600 feet, 70 degrees F)

3.4.4.6.3.2 Pitch Control Motor. - The operating limitations for the pitch control motor are as follows:

Resultant Thrust	Resultant Total Impulse
minimum - maximum	minimum - maximum
2,400 to 6,000 pounds at sea level, 70 degrees F	1,200 to 3,000 pounds seconds at sea level, 70 degrees F

3.4.4.6.3.3 Tower Jettison Motor. - The operating limitations for the tower jettison motor are as follows:

Resultant Thrust	Resultant Total Impulse
minimum - maximum	minimum - maximum
29,700 to 35,965 pounds at sea level, 70 degrees F	33,400 to 38,500 pounds seconds at sea level, 70 degrees F

3.4.4.6.3.4 Abort Capability. - For escape prior to or shortly after liftoff, the LES shall be capable of propelling the CM to an altitude of at least 4,000 feet and a lateral range at apogee of at least 3,000 feet in the absence of adverse winds or booster malfunctions which generate large initial disturbances. The system shall have the capability of abort in one direction only. This direction will be defined by the orientation of the booster on the launch pad and will be approximately due east of the pad towards the ocean.

3.4.4.7 ELS. -

3.4.4.7.1 Operating Limits. - The parachute subsystems shall be capable of operating within the operational envelope defined in Figure 41.

3.4.4.7.2 Performance Characteristics. - The parachute subsystem shall be capable of meeting the following requirements for an Apollo CM weighing up to 9,500 pounds (including the forward compartment heat shield).

3.4.4.7.2.1 Drogue Parachute. - The drogue parachute shall be mortar deployed within the operational envelope defined in Figure 41; however, in no case at an altitude less than 18,000 feet for the entry condition, or 2,000 feet for the abort condition.

3.4.4.7.2.2 Attitude. - The CM may be in any attitude, other than stabilized apex forward, when the drogue parachute is deployed.

3.4.4.7.2.3 Rotational Rates. - The CM rotational rate about the Y or Z axis at the time the drogue parachute is deployed shall be not greater than 10 degrees per second in the positive or negative directions for the entry condition, or 150 degrees per second in the positive or negative directions for the abort condition.

3.4.4.7.2.4 Orientation. - The deployment of the drogue parachute shall orient the CM to the heat-shield-forward attitude with respect to the relative wind.

3.4.4.7.2.5 Velocity Control. - The drogue parachute shall be capable of bringing the vehicle within the landing parachute operational envelope defined in Figure 41, and shall reduce all body rates to values that ensure satisfactory deployment and operation of the main landing parachutes.

3.4.4.8 Crew Systems. -

3.4.4.8.1 Flight Crew. - The flight crew will consist of three crewmembers designated as astronauts No. 1, 2, and 3, and will perform the following functions:

(a) General flight operations

(b) Systems support operations

(c) Life support operations

(d) Lunar exploratory operations and scientific duties.

3.4.4.8.1.1 Command Operations. - The flight crew will perform command operations by assuming and implementing on-board command of the SC throughout all space modes.

3.4.4.8.1.2 Flight Control Operations. - The flight crew will control or direct control of the SC throughout all flight modes under all normal operating conditions. The control operations are to include such tasks as orientation of SC for boresighting and performing landmark and star fixes, stabilizing SC, and establishing coast attitude, lunar orbit insertion, lunar landing, and lunar orbit rendezvous.

3.4.4.8.1.3 Systems Management. - The flight crew will periodically monitor the status of systems subject to depletion and for analysis, detection, prediction, and remedial procedures as required.

3.4.4.8.1.4 Maintenance. - The flight crew will perform inflight maintenance as required.

3.4.4.8.1.5 Scientific Observations. - The flight crew will make all scientific observations programmed for the specific mission. The capabilities of observing shall include visualizing, recording, utilization of scientific instruments, required maintenance of instruments, and transmission of data.

3.4.4.8.1.6 Work, Rest, and Sleep Tasks. - The flight crew will adhere to work/rest and sleep cycles of the mission task specification designed to provide maximum crew performance reliability.

3.4.4.8.1.7 Biomedical Observations. - The flight crew will utilize the resources and facilities provided in the SC for monitoring health and welfare and in the treatment of illness or injury.

3.4.4.8.1.8 Flight Crew Positions. - The normal flight crew function positions shall be as follows:

	Crew Position	Crew Member
(a)	Control position (left couch)	Astronaut No. 1
(b)	Center position (center couch, stowable)	Astronaut No. 2
(c)	Systems management (right couch)	Astronaut No. 3
(d)	Navigation station (with center couch stowed)	All
(e)	LEM Station (during attached phase only)	All
(g)	Sleep station (center couch while stowed)	All

Crew positions a, b, and c shall be normal for boost and entry phases. Crew positions relative to crew task shall be such that each crew member will have the capability of performing the assigned tasks of any other crew member. The control position shall be the main duty station at which the astronaut maintains complete control of the SC for all normal and emergency conditions.

3.4.4.8.1.9 Crew Space Arrangement. - Crew space arrangement shall be so that display information and controls are in relation to the crew operational functions.

3.4.4.8.1.9.1 Watch Duties. - One of the duty stations shall be the systems management station at which one crewmember stands watch during noncritical mission phases. Some immediate control over all critical systems must be accessible at this station.

3.4.4.8.1.9.2 General Duties. - The general duties of all three astronauts will be as follows:

(a) Preparation for navigational fixes

(b) Orientation of SC for boresightings

(c) Performing star and landmark navigation fixes

(d) Scientific observations

(e) Biomedical observations.

3.4.4.8.1.9.3 Off-Duty Cycle. - The off-duty cycle areas shall provide for:

(a) Eating

(b) Personal hygiene

(c) Sleep and rest

(d) General housekeeping.

3.4.4.8.1.10 Crew Mobility. - One of the crew shall wear a pressurized space suit at all times during normal inflight operations. Mobility while in the pressure suit shall be reasonably unrestricted to allow ingress and egress. Hand dexterity and wrist articulation shall be sufficient to operate controls and use maintenance tools as required by mission activities.

3.4.4.8.1.11 Physiological Response Monitoring. - Physiological response monitoring shall consist of monitoring instrumentation and clinical physiological measuring devices used by the flight crew for monitoring one another.

3.4.4.8.1.11.1 Biomedical Monitoring Instrumentation. - Biomedical monitoring instrumentation shall be used for electromechanical transduction and transmission of physiological parameters from the flight crew to the CM PCM for ultimate telemetering to earth. Sensors and preamplifiers will be supplied by the NASA to measure the following parameters:

(a) Blood pressure

(b) Respiration rate and volume

(c) Temperature

(d) Electrocardiogram.

3.4.4.8.1.11.2 Clinical Physiological Measuring Devices. - Clinical physiological measuring devices shall be used for inflight measurement of physiological functions by the flight crew. The devices include:

(a) Sphygmomanometer

(b) Clinical thermometer

(c) Stethoscope

(d) Metabolic balance device

(e) Pulmonary function device

(f) Cardiovascular measuring device.

3.4.4.8.1.12 Rest Periods. - Crew rest periods will be established to allow for optimum crew performance for the overall mission.

3.4.4.8.1.13 Food. - Crew food provisions shall be sufficient to allow optimum crew performance during all phases of the SC mission.

3.4.4.8.1.14 Waste. - Waste management and facilities shall provide for efficient and sanitary disposal and prevent deterioration of crew duty performance. Waste management will include:

(a) Personal defecation control

(b) Personal urinary control

(c) Sanitary storage of feces and urine

(d) Vomiting control.

3.4.4.8.1.15 Suited Vision. - Suited vision requirements to facilitate visual control in the CM shall be integral with the helmet development. Unrestricted vision in the horizontal of 200 degrees plus 60 degrees up and 60 degrees down from standard line of sight without head movement shall be required. The eye relief distance shall be compatible with the use of the optical instruments provided.

3.4.4.9 ECS. -

3.4.4.9.1 Performance Characteristics. - The ECS system shall be capable of meeting the performance specified herein up to the entry interface. The ECS shall provide a supply of water for controlling the total internal heat load from the entry interface to 100,000 feet altitude. From 100,000 feet to main chute deployment, the ECS shall provide air circulation only. From main chute deployment to touchdown and during the post-landing phase, the ECS shall provide for cabin air circulation.

3.4.4.9.1.1 Post-Flight Period. - The CM ECS shall be capable of providing cabin air circulation for a period of 24 hours after landing on water or land.

3.4.4.9.1.2 Pressurization Subsystem. - The pressurization subsystem shall be capable of maintaining cabin pressure at the following levels:

(a) Normal - 5.0 plus or minus 0.2 psia with a cabin gas leakage rate of 0.2 pound per hour, except in the docked mode

(b) Emergency - 3.5 psia minimum for a period of 5 minutes following a puncture in the cabin wall equivalent to a single 1/2-inch diameter hole

(c) Positive pressure relief - 6.0 plus or minus 0.2 psia above sensed SC ambient pressure

(d) Negative pressure relief - 25-inch water maximum differential pressure between ambient and cabin.

3.4.4.9.1.3 Metabolic. - The ECS shall be capable of maintaining the average daily metabolic requirements for each crew member as specified.

3.4.4.9.1.4 Crew Normal Environment. - The ECS shall be capable of maintaining the following normal crew environments throughout the space mission as specified in 3.6.2.10.1 through 3.6.2.10.5.

3.4.4.9.1.5 Pressure Suit Operation. - The ECS shall be capable of providing a continuous, conditioned oxygen atmosphere to individual pressure suits during pressure suit operation. In event of cabin decompression, the subsystem shall be capable of maintaining a minimum pressure of 3.5 psia within the pressure suits. The pressure suit ventilation flow at 3.5 psia shall be 12 cubic feet per minute through each suit and the maximum flow resistance of each suit shall be 5 inches of water. Individual manual flow control shall be provided for each suit.

3.4.4.9.1.6 Equipment Cooling. - The ECS shall be capable of providing thermal control for equipment. The water-glycol coolant shall be supplied to the equipment in the CM at a temperature between 40 and 70 degrees F. A portion of the water-glycol coolant shall be supplied at 45 plus or minus 3 degrees F for cooling the IMU guidance equipment.

3.4.4.9.1.7 Humidity Control. - The ECS shall be capable of controlling the cabin humidity by condensing and collecting the water in the regenerative circuit loop. The humidity control condensed water shall be pumped into the waste water system by an oxygen pressure displacement pump. The water separator shall be a wick-type separator. The latter shall be a part of the regenerative loop heat exchanger.

3.4.4.9.1.8 Water Storage. - The ECS shall be capable of providing the means for storage of potable water from the fuel cell and from water condensation in the ECS. Water collected from the fuel cells shall be stored in a potable water tank. Water collected by condensation in the ECS and any excess fuel cell water shall be stored in a waste water tank. A separate tank shall be filled with approximately 10 pounds of Freon for cabin and equipment temperature control during the boost phase of the mission. Automatic controls shall be provided for the Freon to water-cooling transition mode.

3.4.4.9.1.9 Potable Water Supply Cooling. - The ECS shall be capable of providing the means for cooling potable water used for metabolic and sanitary purposes.

3.4.4.9.1.10 Cryogenic Storage Subsystem. - The cryogenic storage subsystem shall be capable of supplying oxygen to the ECS and the EPS at a controlled pressure of 900 psia, and hydrogen to the EPS at a controlled pressure of 250 psia. The total quantity of stored hydrogen for the EPS shall be 56 pounds. The total quantity of stored oxygen for the EPS shall be 425 pounds. The reactant quantities shall be based on 522 kwh of electrical power requirements for the Apollo. The total quantity of stored oxygen for the ECS shall be 215 pounds. The above oxygen and hydrogen quantities include oxygen for LEM requirements and hydrogen and oxygen requirements for return during emergency condition on single hydrogen or oxygen tank operation.

3.4.4.9.2 Primary Modes and Functions. - The ECS shall be capable of performing the following functions:

(a) Removing carbon dioxide and odors from the pressure suit circuit, cabin environment, and waste management system by the use of lithium hydroxide and activated charcoal

(b) Storing oxygen and hydrogen required for the ECS and EPS in a single phase condition for the entire mission

(c) Rejecting equipment and crew heat loads to space through a radiator

(d) Providing supplemental and emergency cooling by evaporating water

(e) Maintaining a cabin internal pressure of 5.0 psia during normal system operation and 14.7 psia (ambient) during ground operation

(f) Maintaining a cabin nominal temperature of 75 degrees F with a relative humidity between 40 and 70 percent during normal system operation

(g) Maintaining a suit pressure of 3.5 psia minimum during emergency system operation.

3.4.4.10 EPS. -

3.4.4.10.1 Performance Characteristics. -

3.4.4.10.1.1 DC Voltage Characteristics. - The EPS shall be capable of outputs as follows:

(a) Steady-state load voltage limits: 25 - 30 volts

(b) Transient load voltage limits: 22 to 31 volts with recovery to steady-state limits within 1 second.

(c) Maximum fuel cell output: 1,500 watts (each) at 27 volts

(d) Minimum battery capacity: 25 ampere-hours at a 30-minute discharge rate.

3.4.4.10.1.2 AC Voltage Characteristics. -

(a) Phases: 3-phase displaced 120 ± 2 degrees, Y-connected, with phase rotation of A-B-C referred to the bus phase designation

(b) Steady-state voltage limits: 115 ± 2 volts root mean square (rms), average of 3 phases

(c) Transient voltage limits: 105 - 125 volts rms, 0.05 second recovery

(d) Voltage unbalance: 2 percent maximum deviation of worst phase from average.

(e) Frequency: 400 cps, synchronized to SC clock. For emergency operation, in the event of loss of SC synchronizing signal, the steady state and transient frequency limits shall be 400 ± 7 cps.

NOTE: Clock accuracy from primary source (G&N system) is 1 part in 10 million for a period of 14 days. Accuracy from backup source (communication and instrumentation system) is 2 parts in 1 million for a period of 5 days.

3.4.4.10.2 Primary Modes and Functions. - The EPS shall be capable of functioning as follows:

(a) Three fuel cell powerplants operative: These powerplants shall supply power for all normal loads until SM separation

(b) One fuel cell powerplant inoperative: The remaining two fuel cell powerplants shall supply power for all normal loads until SM separation.

3.4.4.10.2.1 Operating Limits. -

(a) Two fuel cell powerplants inoperative: All nonessential loads shall be removed. The remaining single fuel cell powerplant shall supply power to all essential loads supplemented by the entry batteries for peak loads only.

(b) One fuel cell powerplant inoperative and one hydrogen or oxygen tank inoperative: All nonessential loads removed.

3.4.4.11 C & I. -

3.4.4.11.1 Primary Modes and Functions. - The communications subsystem shall provide the means to accomplish the following:

(a) Convert optical data to electrical signals through the use of TV equipment

(b) Assemble and encode, in suitable form for transmission, the data required by the mission control center (MCC) during SC missions

(c) Provide for voice communication between crew members, and provide for the transfer of audio signals to and from transmitters and receivers

(d) Provide omnidirectional and high gain antennas as required to transmit and receive electromagnetic signals

(e) Provide for reception and relay to ground operational support system (GOSS) of limited biomedical data from the crew during extra-vehicular activities

(f) Transmit voice, telemetry, stored data, emergency keyed transmission, and TV signals to the GOSS stations

(g) Receive and demodulate voice transmissions from the GOSS stations

(h) Receive and transmit in phase-coherence a pseudo-noise type signal to enable theGOSS stations to track the SC in angle, velocity, and range

(i) Provide for two-way voice communication with the crew during extra-vehicular activities

(j) Receive and respond to interrogating radar signals from GOSS

(k) Provide electronic recovery aids in the location and recovery of the SC following entry

(l) Provide two-way voice communications between the CM and the LEM

(m) Receive and respond to interrogation radar signals from the LEM

(n) Provide for emergency-keyed transmission from the SC over the unified S-band link.

3.4.4.12 Controls & Displays. -

3.4.4.12.1 Performance Characteristics. - Performance capabilities of the controls and display equipment shall be as defined in the following subparagraphs.

3.4.4.12.1.1 Flight Control Group. -

3.4.4.12.1.1.1 SCS Controls and Displays. - Controls and displays for the SCS shall have the following performance characteristics:

(a) SCS control panel - This unit shall enable the crew to select the mode of control of the SC, to select the attitude dead-band setting of the SCS, to disable maneuver commands in selected SC axes, to convert any of the three body-mounted attitude gyros (BMAG) into backup rate gyros in case of rate gyro failure, and to manually insert the 0.05g signal by means of a backup switch.

(b) Flight Director Attitude Indicator (FDAI) - This instrument shall display to the crew total SC attitude, attitude errors and body rates, all in each of three axes.

(c) Manual controllers - The rotational hand controllers shall operate the SCS for normal or emergency attitude maneuvers in pitch, yaw, and roll. The translational hand controllers shall operate the SCS for normal translation and ullage maneuvers in X, Y, or Z directions. The translational hand controllers shall also provide for emergency cutoff of the SPS rocket engine.

(d) Attitude set and gimbal position display - This unit shall enable the crew to adjust the attitude error reference coordinates in three axes, to align the FDAI ball to a desired orientation in three axes, to display the relative orientation of the SPS engine gimbals in the pitch and yaw axes, and to command reorientation of the SPS engine gimbals in the pitch and yaw axes.

(e) ΔV Panel - This unit shall enable the crew to set desired ΔV_x magnitude and tail-off correction for thrusting maneuvers, to monitor ΔV_x magnitude attained during thrusting, and to enter normal ignition and cutoff commands for the SPS rocket engine, and emergency ullage commands for the SM RCS.

3.4.4.12.1.1.2 Sequence Controls and Displays. - Controls and displays for execution of launch escape, earth landing, and separation sequences shall be capable of providing positive manual arming and disarming of pyrotechnic devices, effective monitoring of sequence status and condition during execution, and adequate manual hold or emergency initiation capability for automatically sequenced events. The barometric pressure indicator shall provide independent monitoring of baroswitch operation during the earth-landing sequence, and shall indicate pressure-altitude quantitatively under low-altitude, low Mach-number conditions.

3.4.4.12.1.1.3 Display of Rendezvous Information. - The controls and displays system shall be capable of presenting, in real time, not more than five symbolized, continuous analog parameters that depict the rendezvous situation of the CM-SM with respect to the LEM during LEM ascent to lunar orbit. The display parameters shall be based on information furnished by the NASA which defines the types of information to be available from the LEM via radio link.

3.4.4.12.1.1.4 Monitoring of Entry Situation. - The entry monitor display unit shall be capable of displaying the gravity versus time history of the CM and the relative orientation of the aerodynamic lift vector of the CM during entry. This unit shall not be dependent on references within the SCS or G & N for situation information. The display shall enable the crew to effectively monitor the progress of the entry phase of flight during either automatically or manually commanded operations.

3.4.4.12.1.1.5 Caution and Warning Subsystem. - The caution and warning subsystem shall be capable of providing visual and aural notification to the crew of caution and warning conditions in a manner accessible to all crew stations in the CM. In addition to caution and warning functions required for CM and SM systems, the subsystem shall be capable of displaying not more than eight event indications of conditions within the NASA-furnished automatic guidance equipment. The contractor shall be responsible for establishing acceptable event signal characteristics for these functions.

3.4.4.12.1.1.6 Operational Nuclear Radiation Instrumentation Displays. - The following display functions shall be provided for readout of information from the operational radiation instrumentation equipment:

(a) Short-term solar RF alert instrumentation output with provisions for remote alarm set point adjustment

(b) Proton directional detection instrumentation output

(c) Personal dosimeter readout, accumulated dosage (one readout in units of roentgens equivalent man (REM)

NAA/S&ID shall be responsible for establishing acceptable signal characteristics of data from the NASA-furnished personal dosimeters.

3.4.4.12.1.1.7 Miscellaneous Controls and Displays. -

3.4.4.12.1.1.7.1 Thermal Profile Information Display. - This unit shall be capable of displaying continuous analog temperature values from selected structural locations within the CM and SM, adequate to provide thermal profile data for determination of vehicle ambients.

3.4.4.12.2 Primary Modes and Functions. - The several functional groups of controls and displays equipment shall provide the SC crew with information and control capability as defined below.

3.4.4.12.2.1 Flight Control Group. - Controls and Displays for SC flight control shall be adequate for safe and effective control of vehicle maneuvers, presentation of flight situation and procedure information, and execution of staging and docking sequences. This equipment group shall be capable of providing the following control-display functions:

(a) Operation of the SCS

(b) Execution of staging and docking sequences involving the CM-SM separation interfaces

(c) Monitoring of the CM situation during entry

(d) Monitoring of the CM-SM rendezvous situation with respect to the LEM during LEM ascent to lunar orbit

(e) Operation of the automatic guidance equipment by means of equipment furnished by the NASA.

3.4.4.12.2.2 Systems Management Group. - Controls and displays for management of SC CM and SM systems shall be capable of providing the effective energy management, condition management during systems operation, and controls for systems operation, selection of alternate operating modes, and safe shutdown. This equipment group shall be capable of providing the above functions for the following CM and SM systems:

(a) Propulsion

SPS

CM RCS

SM RCS

(b) Electrical Power

Fuel cells

Lighting

AC and dc electrical power distribution

(c) Environmental/cryogenic

ECS (Gas and liquid circuits)

Cryogenic storage system

3.4.4.12.2.3 Communications and Data Control Group. - Controls and displays for communications and data systems shall be capable of providing effective operational control and selection of alternate operating modes for the following communications and data system functions:

(a) Voice communications

(b) Data transmission

(c) Television

(d) Antennas

(e) CTE

3.4.4.12.2.4 Caution and Warning Subsystem. - The caution and warning subsystem shall be capable of notifying the crew, at crew stations in the CM:

(a) Of major malfunctions in the SC CM and SM systems during operation

(b) Of the existence of unsafe conditions in the CM or SM

(c) Of the initiation of abort or other emergency sequences.

3.4.4.12.2.5 Miscellaneous Controls and Displays. - In addition to the control-display functions defined above, the following functions shall be provided:

(a) Operational control of the IFTS

(b) Operational control of in-flight calibration

(c) Display of CM-SM thermal profile information

(d) Operation of the film cameras

(e) Radiation instrumentation displays.

3.5 Reliability Requirements. -

3.5.1 Mission Success Reliability. - The mission success reliability objective for the Apollo lunar orbital rendezvous mission shall be 0.90 and shall be interpreted as the achievement of a successful lunar landing and minimum lunar stay period of 2 hours without exceeding the nominal crew limits, given in the design criteria, followed by the return to earth of the SC without exceeding the emergency crew limits given in the design criteria.

3.5.2 Crew Safety Reliability. - The crew safety reliability objective for the Apollo LOR mission shall be 0.999 and shall be interpreted as the probability that the crew shall not have been subjected to conditions greater than the emergency limits given in the design criteria.

3.5.3 Reliability Apportionment. - The reliability objectives for the major Apollo-Saturn systems shall be as delineated below:

Apollo-Saturn Reliability Apportionments

System	Mission Success	Crew Safety
GSE	0.9999	0.99999
GOSS	0.999	0.99999
LAUNCH VEHICLES (defined by the NASA)	0.950	0.99994
CM AND SM	0.9638	0.99958
LEM (defined by the NASA)	0.984	0.9995
APOLLO-SATURN	0.90	0.999

3.6 Environments. -

3.6.1 Transportation, Ground Handling, and Storage. - These criteria represent the natural and induced environmental design criteria associated with transportation, ground handling, and storage. Provisions shall be developed to prevent exposure of SC components to conditions in excess of flight environments. Equipment shall be capable of meeting the requirements of this specification after exposure to these environments.

3.6.1.1 Natural Environments. -

(a)	Temperature (degrees)	
	Air transportation	-20 F minimum to +140 F maximum
	Ground transportation	-20 F minimum to +140 F maximum
	Storage	+25 F minimum to +105 F maximum
	Launch base area	+25 F minimum to +105 F maximum air temperature, plus solar radiation of 360 British thermal units (Btu) per square foot per hour for a 6-hour period each day.
(b)	Altitude	
	Air transportation	Up to 35,000 feet for 8 hours
	Storage	Up to 6,000 feet for 3 years
(c)	Humidity	95 ± 5 percent relative humidity, including conditions wherein condensation takes place in the form of water.
(d)	Sunshine	Solar radiation at 360 Btu per square foot per hour for 6 hours per day for 2 weeks.
(e)	Rain	Up to 0.6 inch per hour
(f)	Sand and dust	As encountered in desert and ocean beach areas, equivalent to 140-mesh silica flour with particle velocity up to 500 feet per minute.
(g)	Fungus	As experienced in tropical climates.
(h)	Salt spray	Salt atmosphere as encountered in coastal areas, the effect of which is simulated by a 20 percent, by weight salt solution for 50 hours.

(i)	Ozone	Up to 3 years exposure to 0.05 part per million concentration.
(j)	Ground winds	These ground wind criteria consist of a description of Cape Canaveral wind data for the height intervals of 10 to 400 feet.

Free standing - The design wind speeds for structural loading considerations of the Apollo SC 99.9 percent of the time during the strongest wind month at Cape Canaveral are presented in the following table:

Height	Steady State Wind (knots)	Peak Wind (knots)*
10	23.0	32.2
30	28.8	40.3
60	33.6	47.0
100	37.5	52.5
200	42.6	59.6
300	46.0	64.4
400	48.3	67.6

Storm Conditions - The 99.9 percent peak wind speeds may be exceeded during severe thunderstorm or hurricane conditions at Cape Canaveral. During such periods, the vehicle must be placed in a service structure, shelter, or tied in such a manner that wind loading conditions greater than those for the 99.9 percent winds shall not be experienced by the vehicle. The vehicle protective structures shall be designed to withstand wind loads resulting from a probable "maximum wind speed" of 108 knots while containing the vehicle. Wind speeds apply from 10- to 400-foot heights.

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*Gust Characteristics:
For the effects of gusts, a linear buildup from the steady state winds to the peak winds will be assumed. The period of this buildup and decay shall be taken as 4 seconds for all height levels; that is, buildup of 2 seconds and 2 seconds for decay to steady state wind speed.

3.6.1.2 Induced Environments. -

(a) Shock — As experienced in each of three mutually perpendicular axes.

Equipment Weight (pounds)	Shock Level (g)	Time (milliseconds)
Less than 250	30	11 ± 1
250 to 500	24	11 ± 1
500 to 1,000	21	11 ± 1
Over 1,000	18	11 ± 1

(b) Vibration — As experienced in each of three mutually perpendicular axes.

Equipment Weight (pounds)	5 to 27.5 cps	27.5 to 52 cps (inch da)	52 to 500 cps
Less than 50	±1.56 g	0.043	±6.0 g
50 to 1,000	±1.30 g	0.036	±5.0 g
Over 1,000	±1.04 g	0.029	±4.0 g

3.6.2 Spacecraft Flight Environments-Mission Phases. - The criteria described in this section represent the environmental conditions and levels to which the SC and its related airborne systems, equipment, and components will be subjected during the various flight mission phases. The equipment shall be capable of meeting the requirements of this specification during or after exposure to these environments.

3.6.2.1 All Mission Environments. - Environmental areas of the SC are represented by zone numbers as indicated in Tables II through V:

(a) Temperature — NAA/S&ID shall provide temperature requirements, in subsystem and component specifications, for each mission phase for each individual design item.

(b) Oxidation	The CM shall be capable of withstanding an atmosphere of 100 percent oxygen at 5 psia for 400 hours (includes ground and flight time), 14.7 pounds psia for 12 hours, and 19.7 psia for 1 hour.
(c) Pressure differential	The CM pressure shell and heat shield shall be designed to store pressure differentials consistent with the design of the SC system and the applicable mission phase.
(d) Hazardous gases - explosion proofing	Design of equipment shall be in accordance with the requirements of MSFC 10M 01071.
(e) Humidity	A relative humidity from 40 to 70 percent is applicable to zone 2, Table III under normal conditions (shirt sleeve environment).

3.6.2.2 Liftoff Environment. -

3.6.2.2.1 Natural Environments. - The 1959 ARDC Standard Atmosphere shall be used as the ambient environmental reference.

(a) Winds	The wind criteria for launching of the Apollo space vehicles consists of a steady-state and a peak wind. The wind gradients below provide a 99-percent launch capability during the strongest wind month.

Steady State Winds (Knots)		Peak Wind (Knots)*
Height	Saturn V (99%)	Saturn V (99%)
10	18.4	25.8
30	22.9	32.1
60	26.4	36.9
100	29.3	41.0
200	33.6	47.0

Steady State Winds (Knots)		Peak Wind (Knots)*
Height	Saturn V (99%)	Saturn V (99%)
300	36.5	51.5
400	38.7	54.2

(b) Sand and dust	same as 3.6.1.1 (f)
(c) Salt spray	same as 3.6.1.1 (h)
(d) Rain	same as 3.6.1.1 (e)

3.6.2.2.2 Induced Environments. -

(a) Humidity	A relative humidity of 95 ± 5 percent is applicable to SM criteria during launch phase only.
(b) Vibration	Refer to Table VI.
(c) Acoustics	Refer to launch and entry acoustical data in Tables II, III, IV, V and VII.

3.6.2.3 Pad Abort Environment. -

3.6.2.3.1 Natural Environments. - The natural environments for pad abort shall be the same as those specified for liftoff in 3.6.2.2.1.

3.6.2.3.2 Induced Environments. -

(a) Vibration	Refer to Table VIII.
(b) Acoustics	Refer to pad abort acoustical data in Tables II, III, IV, V, and VII.
(c) Acceleration and shock	Accelerations shall be compatible with CM weights and escape rocket thrust characteristics.

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* Gust Characteristics:
For the effects of gusts, a linear buildup from the steady-state winds to the peak winds will be assured. The period of this buildup and decay shall be taken as 4 seconds for all height levels; that is, buildup of 2 seconds, and 2 seconds for decay to steady-state wind speed.

3.6.2.4 Boost Environment. -

3.6.2.4.1 Natural Environments. - The only other natural environment considered in this section is the winds aloft-data. The 1959 ARDC Standard Atmosphere shall be used as the ambient environmental reference. An inflight wind profile is shown in Figure 42 for the 95 percent level.

3.6.2.4.2 Induced Environment. -

(a) Vibration	Refer to Table VI.
(b) Acoustics	Same as 3.6.2.2.2 (c)
(c) Acceleration and shock	Accelerations shall be compatible with SC weights and launch vehicle flight characteristics. Thrusts for the launch vehicle stages are as follows:
(1) S-IC	7,500,000 pounds at sea level
(2) S-II	1,000,000 pounds vacuum
(3) S-IVB	200,000 pounds vacuum

3.6.2.5 Hi-Q Abort Environment. -

3.6.2.5.1 Natural Environment. - Same as 3.6.2.4.1.

3.6.2.5.2 Induced Environment. - (See Tables II, III, IV, V, VII, and VIII for acoustic and vibration data.)

3.6.2.6 High Altitude Abort Environment. - Temperature and pressure to be determined.

3.6.2.6.1 Induced Environments. -

(a) Vibration	See Table IX.
(b) Acceleration and shock	Accelerations shall be compatible with SC weights and SM propulsion thrust characteristics.

3.6.2.7 Parking Orbit, Translunar, Lunar Orbit, and Trans-Earth Environments. -

3.6.2.7.1 Natural Environments. -

(a) Nuclear radiation	NAA/S&ID shall perform analytical and experimental shielding studies to determine the inherent shielding capability of the SC. Design activities shall be conducted to reduce the radiation level to a minimum within the CM, to the extent practicable by good design practice. Consideration of the radiation hazard shall be incorporated into the CM and SM design when NAA/S&ID is supplied with radiation protection design criteria by the NASA and as agreed to by the NASA and NAA/S&ID.
(b) Meteoroids	Consideration of the meteorlogical hazard shall be incorporated into the design of the CM and SM when NAA/S&ID is supplied with meteoroid protection design criteria by the NASA and as agreed to by the NASA and NAA/S&ID. Prior to the establishment of this design criteria, limited testing shall be conducted to determine, on a sample basis, the meteoroid protection afforded by the structure.

3.6.2.7.2 Induced Environments. -

(a) Vibration	See Table IX.
(b) Acceleration and shock	Accelerations shall be compatible with SC weights and SM and S-IVB propulsion thrust (200,000 pounds) characteristics.

(c) Pressure

(1) Spacecraft exterior	Normal and Emergency: 7.5 x 10^{-10} mm Hg (minimum) for 336 hours (maximum)
(2) Spacecraft interior-zones 1, 3 and 4	(See Tables III, IV and VII.) Normal and Emergency: 1.0 x 10^{-6} mm Hg (minimum) for 336 hours (maximum)
(3) Spacecraft interior-zone 2, Table III	Normal condition: 5 psia (nominal) Emergency condition: 1.0 x 10^{-4} mm Hg (minimum) for 100 hours (maximum).

3.6.2.8 Earth Entry. -

3.6.2.8.1 Aerodynamic Heating. - Aerodynamic heating limits are listed in Table X for the two limiting design entry trajectories defined in terms of the maximum heating rate, the maximum time-integrated heating load and the flight time associated with this heating load, and the maximum heat shield bond line temperature.

3.6.2.8.2 Natural Environments. - Same as 3.6.2.7.1

3.6.2.8.3 Induced Environments. -

(a) Vibration	See Table VI.
(b) Acoustics	Refer to entry environment acoustical data in Tables II, III, IV, VI, and VII.
(c) Acceleration and shock	Accelerations are as follows: Steep entry 20 g (max.) Shallow entry 10 g (min.)

3.6.2.9 Earth Landing. -

3.6.2.9.1 Natural Environments. -

3.6.2.9.1.1 Ocean Waves. - The design conditions shall be as follows:

Wind velocity	16 to 20 knots

Wave height (crest to trough)	5 to 8 feet
Wave period	5 to 6 seconds
Wave length	125 to 185 feet
Wave velocity	14 to 18 knots

3.6.2.9.2 Induced Environments. -

3.6.2.9.2.1 Acceleration and shock. - Accelerations and shock shall be compatible with the CM weight and the ELS.

3.6.2.10 Crew Environment Requirements. -

3.6.2.10.1 Cabin Pressure. - The cabin pressure nominal limits shall be 4.8 psia minimum and 14.9 psia maximum. The emergency limit shall be 3.5 psia minimum.

3.6.2.10.2 Oxygen Partial Pressure. - The oxygen partial pressure nominal limits shall be 233 millimeters (mm) of mercury minimum, and emergency limits shall be 160 mm mercury minimum.

3.6.2.10.3 Carbon Dioxide Partial Pressure. - The carbon dioxide partial pressure nominal limit shall be 7.6 mm mercury maximum. The emergency limits are presented in Figure 43.

3.6.2.10.4 Cabin Temperature. - The cabin temperature nonstressed limits shall be 70 degrees F minimum and 80 degrees F maximum. The nominal and emergency limits are presented in Figures 44 and 45, respectively.

3.6.2.10.5 Cabin Relative Humidity. - The cabin relative humidity nonstressed limits shall be 40 percent minimum and 70 percent maximum. The nominal and emergency limits are presented in Figures 44 and 45, respectively.

3.6.2.10.6 Radiation. - NAA/S&ID shall prepare a specification for the radiation environment based on the radiation exposure limits set forth in Table XI. Dosage calculations shall be based on the model presented in Figure 46. In the absence of sufficient information to assign a dosage value due to secondary radiation, a value of 50 percent of the primary dose shall be used.

3.6.2.10.7 Noise. - The noise nonstressed limit shall be 80 db overall and 55 db in the 300 to 4,800 cps range. The stressed limit shall be the maximum noise level that will permit communications with the ground and between crew members at all times. The emergency limit is presented in Figure 47.

3.6.2.10.8 Vibration. - The vibration stressed, nonstressed, and emergency limits are presented in Figure 48.

3.6.2.10.9 Sustained Acceleration. - The sustained acceleration limits for eyeballs out, down, and in condition are presented in Figures 49, 50, and 51. The sustained acceleration performance limits are defined as the maximum sustained acceleration to which the crew shall be subjected and still be required to make decisions, and perform hand controller tasks requiring visual acuity.

3.6.2.10.10 Impact Accelerations. - The emergency impact acceleration and onset rate limits to which the crew can be subjected are presented in Figure 20.

3.6.3 Explosion Proofing. - The entire SC, including electronic systems and rocket motor ignitors, shall be designed to minimize the existence of fire hazards or explosive environment. The systems shall be designed to prevent the emission of gaseous vapors that might contaminate the CM during any part of the mission operation. The fuel tanks mounted in the CM shall be compartmentized to prevent ignition in the event that leakage should occur. Where practicable, the various components shall be hermetically sealed or of explosion-proof construction. The rocket motor squibs shall be capable of withstanding an electrical impulse of 1 ampere at 1 watt dc for 5 minutes without detonating. Design of equipment shall be in accordance with MSFC 10M01071.

3.7 Electromagnetic Interference (EMI) Control. - Apollo SC electromagnetic interference control shall be in accordance with Specification MC 999-0002.

3.7.1 EMI Specification Applicability. - The requirements of the EMI specification shall be imposed upon NAA/S&ID design groups and all subcontractors. Coordination will be established with subcontractors and associate contractors for interpretation of the specification, a definitization of the EMI interface between subsystems, a description of the subsystem threshold susceptibility characteristics, a profile of the signal characteristics, and evaluation of all subsystem EMI test data.

3.7.2 Records and Data. - Spectrum charts and records of all interference and susceptibility data shall be maintained and analyses performed to determine potential EMI incompatibility areas. Based upon these analyses, filter requirements, interconnecting wire treatment, and routing shall be established. Compatibility assurance test requirements shall be relaxed or tightened, depending upon these analyses.

3.7.3 Compatibility Assurance. - Combined systems compatibility assurance testing at NAA/S&ID shall be performed upon all flight SC. The extent and scope of each succeeding test shall be influenced and dictated by careful analysis of the preceding test data.

3.8 Interchangeability. - Interchangeability as defined for the Apollo program applies to all completely finished assemblies, components, and parts which shall be readily installed, removed, or replaced without alteration, misalignment, or damage to parts being installed or to adjoining parts. No fabrication operations such as cutting, filing, drilling, reaming, hammering, bending, prying, or forcing shall be required for installation.

3.8.1 Electronic Equipment. - Interchangeability of electronic equipment requires that mechanical and electrical interchangeability shall exist between like assemblies, subassemblies, and parts, regardless of the manufacturer or supplier. Interchangeability, for the purpose of this specification, does not mean identity, but requires that a substitution of such like assemblies, subassemblies, and replaceable parts can be easily effected without physical or electrical modification to any part of the equipment, including cabling, wiring, and mounting, and without resorting to selection; however, adjustment, trimming, or calibration may be made.

3.8.2 Replaceability. - Replaceability, as defined for the Apollo program, applies to parts which may require additional work or operations during installation. This may include additional operations such as drilling, reaming, cutting, filing, trimming, shimming, or other means normally associated with the original assembly into the end item. Replaceable parts shall be designed to permit replacement under field maintenance conditions.

3.8.3 Traceability. - Assemblies, subassemblies, and parts shall be identified or documented so that the history can be traced to the prime source of the individual component or part and the materials used can be traced to the prime supplier.

3.9 Design Objective Weight. - The design objective gross injected weight of the SC, including LEM, shall be 90,000 pounds.

3.10 Nameplates and Product Markings. - The SC and all assemblies, components, and parts shall be marked for identification in accordance with Standard MIL-STD-130.

4. QUALITY ASSURANCE PROVISIONS

4.1 General Quality Assurance Program. - NAA/S&ID shall establish a quality assurance program in accordance with NASA Publication NCP 200-2. Inspections and tests to determine conformance of the system to contract and specification requirements shall be conducted prior to submission of the article to the NASA for acceptance.

4.1.1 Quality Control. - NAA/S&ID shall establish a quality control plan as defined in SID 62-154, volumes I and II.

4.2 NAA/S&ID Reliability Program. - NAA/S&ID shall establish a reliability program in accordance with SID 62-203.

4.3 NAA/S&ID Testing. - NAA/S&ID testing shall be in accordance with SID 62-109, volumes I, II, III, IV, and V.

4.4 NASA Testing. - The NASA shall conduct reliability tests.

4.5 Configuration Management Provisions. -

4.5.1 Change Control. - NAA/S&ID shall maintain an effective configuration control program to control the incorporation of engineering changes affecting engineering orders and drawings, specifications, procurement documents, quality control, inspection and test procedures, process, manufacturing, and operating instructions, and similar documents.

5. PREPARATION FOR DELIVERY

5.1 Preservation, Packaging, and Packing. - Preservation, packaging, and packing shall be in accordance with NAA/S&ID procedures, provided the procedure assures adequate protection in accordance with delivery modes, destinations, and anticipated storage periods.

5.2 Handling. - Handling shall be in accordance with NAA/S&ID procedures.

6. NOTES

6.1 Definitions. -

Absorptance. - Ratio of absorbed radiant energy to the incident radiation. In this specification, absorptance is over the solar spectrum.

Advisory Light. - A placard light used to inform the crew member of a safe or normal configuration, equipment, or system status; operation of essential equipment; or to attract attention and impart information of a routine nature.

Albedo. - Ratio of radiant energy reflected from a planet or satellite to that received by it. A dimensionless decimal quantity equal to or less than 1.

Areas of Illumination Control. - A partial instrument panel area, primary duty station area, or secondary duty station area illuminated by a given set of luminaires.

Automatic Control. - Any device whose function is to activate a mechanism or an equipment without the aid of human control.

Backpack. - A portable life support system providing atmospheric and thermal control.

Biomedical Instruments. - The instrumentation associated with the pickup, recording, and transmitting of physiological data.

Breadboard Model. - An assembly of preliminary circuits and parts employed to prove the feasibility of a device, circuit, equipment, system, or principle in rough or breadboard form, without regard to the eventual overall design or form of parts.

C-Band. - 4,000 to 7,000 mc.

Caution Light. - A placard light used to inform the crew members of an impending dangerous condition requiring attention but not necessarily immediate corrective action, such as "electrical compartment overheated" or "cabin pressure low."

Ceiling Area. - That area above the intersection of the vertical wall and sloping wall or above consoles with the capsule in the launch position. This area begins approximately 30 inches above the capsule floor and side wall intersection.

Clinical Devices. - Devices for making basic physiological measurements indicative of crew well-being, such as thermometers and blood pressure measuring devices.

Combined Stresses. - Combined stresses are those stresses caused by simultaneous action of all factors; for example, direct loading thermal stresses.

Command Functions. - The manual and automatic control of the vehicle during all phases of the mission; the selection, implementation, and monitoring of modes of navigation and guidance; and monitoring and control of key areas of all systems during time-critical periods.

CM RCS. - A control system that provides thrust vectors for three-axis control of the CM. This terminology does not include the associated G&N.

Contrast Ratio. - Contrast ratio is $C = \frac{B_2 - B_1}{B_1}$ where B_1 is the brightness of the background and B_2 is the brightness of the lettering, numbering, or markings.

Console. - As used in GSE nomenclature, a console is a short vertical or sloping front frame upon which standard 19-inch or 24-inch panels are mounted to support instrument indicators and controls. As used in the CM, a console is any front frame upon which are mounted controls and displays, except for the main instrument panel.

Crew Environment Requirements. - Environmental needs to sustain crew life and provide for comfort and efficiency in work, rest, and recreation.

Crew Performance. - The capability of a crew member to perform a required or an assigned task with an acceptable degree of proficiency.

Crew Requirements. - Requirements and provisions necessary to maintain crew health and well-being and to assure effective crew performance.

Crew Safety. - Maintenance of the well-being of the crew within certain specified limits and probabilities.

Crew Status. - The state of well-being, including physical and mental health of the crew, as indicated by measurable physiological changes.

Crew Systems. - The organization, function, and interrelationship of crew members in regard to their responsibilities and performance as a subsystem in support of the SC in performing the specified missions. The arrangement of the CM interior, the crew life support and survival equipment, and miscellaneous comfort items also fall within this category.

Design Burst Pressure. - The maximum limit pressure multiplied by the appropriate ultimate factor of safety.

Design Ultimate Load. - The limit load multiplied by the ultimate factor of safety.

Design Yield Load. - The limit load multiplied by the yield factor of safety.

Design Yield Pressure. - The proof pressure multiplied by the appropriate factor of safety.

Display. - Any device which exhibits, presents, or imparts information to the operator or crew member in any manner. This may be an indicator, or groups of indicators, a label, or a gross machine transmitting motion, vibration, or acoustic information to operator or crew member.

Electrical Heat Load. - The heat load on the ECS resulting from electrical and electronic heat dissipation.

Emergency Controls. - Those controls used during an emergency situation where proper operation is mandatory and where inadvertent operation could result in component destruction or crew member injury. Examples are destruct switches, fire extinguisher switches, and ejection controls.

Emergency Limits. - The limits beyond which there is high probability of permanent injury, death, or incapacity to such an extent that the crew would not perform well enough to survive.

Environmental Heat Load. - The heat load on the ECS which results from heat transfer through the CM wall.

Factor of Safety. - The ratio of the design load on a structure to the limit load.

GSE. - GSE is the equipment required to inspect, test, adjust, calibrate, appraise, gage, measure, repair, overhaul, assemble, disassemble, transport, safeguard, record, store, actuate, service, maintain, launch, and otherwise support an end article. GSE shall include ground-based training equipment, simulation devices, and auxiliary power devices.

HF. - High frequency, 3-30 mc.

Human Engineering. - The determination of man's capabilities and limitations as they relate to the mission environment conditions, crew station provisions, and other equipment he will use, and the application of this knowledge to the planning and design of complete systems, support systems, and operational support equipment so that the reliability and efficiency of the resultant man-machine combinations will be increased.

Human Factors. - The scientific determination of facts about human behavior, the development of systematic methods for considering man in the design of systems, and the application of these facts and methods throughout design. It includes the development and application of procedures and principles for the design of work spaces, equipment, human tasks, training, and human environments.

Human Waste. - Urine, feces, sputum, nasal discharge, vomitus, perspiration, carbon dioxide, and other gases.

Indicator. - Any discrete display device providing a crew member with a qualitative or quantitative indication. This may be a counter, light, mechanical lever movement, or other device which changes color, illumination, or position as a function of another mechanism or condition.

Indicator Light. - A signal light assembly having no markings on the illuminated surface and used to present equipment status or similar information to crew members. The light is not intended as an attention attractor and is usually used in the secondary duty area.

Instrument. - A device for detecting and measuring some physical phenomenon. The output of an instrument may be an indication or a signal to one or more recording devices, or to one or more telemetering devices.

Instrument Panel. - A panel upon which are mounted instruments and their associated displays and controls, such as switches and adjustment knobs for ready scanning and operation by an operator or crew member.

Integral Illumination. - Illumination that originates within a display or indicator device. An integrally illuminated display contains its own light sources.

Life Support System. - The complete system necessary for sustaining life in an alien environment, including atmospheric control, thermal control, food and water provision, waste disposal and sanitation, and radiation protection.

Limit Load. - Maximum calculated load to which the structure will be subjected under specified conditions of operation.

Limit Pressure. - Maximum pressure to which the structure will be subjected under specified conditions of operation. For SC propellant tank design, maximum limit pressure is the maximum relief valve pressure plus hydrostatic head (if applicable), and the minimum limit pressure is the minimum pressure of the propellant under specified conditions of operation plus hydrostatic head (if applicable).

Load Factor. - A body load parameter equal to the ratio of net force applied to the body in a given direction divided by the weight of the body.

Lunar Sub-Solar. - Lunar noon or that point in time of the lunar light period when the sun's rays are perpendicular to the lunar surface at the equator.

Main Instrument Panel. - That instrument panel or collection of instrument panels upon which are mounted the primary attitude and condition indicators for the CM.

Maintainability. - A quality of the combined features of material, design, and installation which permits or enhances the accomplisment of maintenance by personnel of average skill, under the environmental conditions in which the maintenance will be performed. It includes repairability and serviceability, and is a function of the rapidity and ease with which maintenance operations can be performed to avert malfunctions or correct them as they occur.

Maintenance. - The servicing, repair, care, modification, or other action taken to keep material or equipment in, or restore it to, such condition as to meet programmed operational requirements.

Man Heat Load. - The heat load, both sensible and latent, rejected by the crew members to either the ECS or backpack system.

Manual Limits. - Limits within which the crew's environment shall be maintained during normal operations.

CONF[redacted]

Margin of Safety. - The percentage by which the allowable load or stress exceeds the design load or stress.

$$M.S. = \frac{\text{Allowable Load or Stress}}{\text{Design Load or Stress}} - 1.0$$

Master Caution Light. - A placard light used to inform the crew members that one of a number of caution lights has been actuated. It is located at the primary duty station.

Maximum Relief Valve Pressure. - The maximum pressure that a relief valve will permit to exist in the system.

Metabolic Requirements. - Human energy exchange needs, such as oxygen, water and food consumption, and carbon dioxide and leak outputs.

Natural Environment. - The sum total of the external influences, such as conditions of the atmosphere, gravity, and radiations, to which the crew will be exposed throughout the entire course of the mission.

Non-Human Waste. - Food remnants, shaving wastes, and dental and body cleansing wastes, such as tissues, towels, and paste.

Non-Stressed Limits. - The environmental limits to which the crew may be subjected for extended periods of time, such as orbit, lunar transit, and periods subsequent to normal landings.

Noxious Gases. - Gases that are potentially toxic to the crew, including carbon dioxide (CO_2), carbon monoxide (CO), hydrogen sulphide (H_2S), sulphur dioxide (SO_2), methane, indole, skatole, mercaptans, and ozone.

Operating Pressure. - Operating pressure, or working pressure, is the nominal pressure to which components are subjected in service under steady-state conditions. Maximum operating pressure is the operating pressure plus tolerance.

Outgassing. - A condition whereby entrapped gases in solids (metals, plastics, phenolics) are released by pressure (vacuum) resulting in the degradation of the solid and possible emission of toxic compounds. This condition is aggravated by elevated temperatures.

[redacted]

Percentile. - A statistical term denoting the hypothetical person within a population who has a dimension such that a certain percent of the population has smaller dimensions. Thus, the "10th percentile man" has all body dimensions such that 10 percent of the men considered have smaller dimensions, and 90 percent have larger dimensions.

Placard Light. - A signal light assembly having markings on the illuminated surface and used to present information of prime importance to the crew. Placard lights are intended as attention-attracting devices and are usually used in the primary duty station.

Potable Water. - Water of sufficient purity for human consumption.

Primary Duty Station. - Crew position for operation and monitoring of primary displays, controls, and support systems.

Proof Pressure. - Equal to maximum limit pressure multiplied by the appropriate factor of safety and is the reference pressure for establishing acceptance test pressure levels.

Propellant System. - A system that stores and distributes propellants to the associated rocket engines.

Radiation Dosimeters. - Devices for recording amounts of radiation exposure.

Radiobiological Terms. -

(a) Roentgen: That amount of X or gamma radiation whose energy per photon is less than 3 million electron volts (MEV) which produces in 0.001293 grams of dry air, under conditions of electronic equilibrium; for example, one electrostatic unit of ionization charge of either sign.

(b) Absorbed Dose: The amount of energy absorbed in a unit mass of any material, the unit of absorbed dose is the radiation absorbed dose (RAD) which is defined as 100 ergs per gram of any irradiated material.

(c) Relative Biological Effectiveness (RBE): The ratio of the RAD dose of x rays to the RAD dose of any other radiation required to produce an identical biological effect in a particular organ or tissue.

(d) Roentgen Equivalent Man (REM): This is used to express human biological doses as a result of exposure to one or many types of ionizing radiation. The dose in REMS is equal to the absorbed dose in RAD's times the RBE factor of the type of radiation being absorbed. Thus, the REM is the unit of RBE dose.

(e) Radiation Dose: Space ionizing radiation exposure considered acceptable by the NASA, tabulated below.

Table of Space Ionizing Radiation Exposure Dose Limits

Critical Organ	Average yearly dose (RADS)	Maximum permissible single acute emergency exposure (RADS)	Location of dose points
Skin of whole body	125	500	0.07 mm depth from surface of cylinder 2 at highest dose rate point along eyeline
Blood forming organs	50	200	5 mm depth from surface of cylinder 2
Feet, ankles, and hands	175	700	0.07 mm depth from surface of cylinder 3 at highest dose point
Eyes	25	100	3 mm depth from surface on cylinder 1 along eyeline

(f) Blood Forming Organs: Lymph node, bone marrow, spleen, and other tissue related to blood component formation.

Reference Axes. - The reference exes of the SC shall be orthogonal and shall be identified as shown in Table I.

Reliability (Equipment). - The probability that an item will perform satisfactorily for a specified period of time when used in the manner and for the purpose intended.

Reliability (Test). - The degree to which tests can be made to measure consistently, generally expressed as a coefficient of correlation between test scores obtained on the same subjects at a different time.

Reliability Program. - A program established by a system contractor to assure that the reliability requirements fixed by the procuring activity and incorporated in the contract are achieved in the system.

Secondary Duty Station. - Areas for taking navigation fixes, performing maintenance, food preparation, and certain scientific observations.

Shirtsleeve Environment. - The environmental conditions that permit freedom of movement to crew members while wearing no more than one layer of clothing and being independent of special personal breathing apparatus, cooling or heating garments, or other personal environmental control devices.

Space Radiator. - The system component that rejects the heat generated within the SC to the surrounding area by radiation.

Sublimation. - A condition whereby a normal transition of change is bypassed, such as cadmium in a vacuum (or low pressure) going from a solid to a vapor form, bypassing the liquid stage. Structural integrity of the solid may be lost and recondensation on adjacent surfaces may cause serious malfunction. In general, the higher the melting point of a solid the lower the sublimation rate.

Super Critical Cryogenics. - Oxygen and hydrogen stored in the gaseous state at low temperatures and high pressure. Storage pressures are above the critical pressure for the respective gas.

Temperature. - In this specification, the temperature to be used in all analyses is the maximum or the minimum temperature calculated for the given loading condition, whichever is more critical.

Thermal Control. - Temperature control of the electrical, electronic, and mechanical components aboard the SC.

Toilet. - A structure into which crew members can void feces.

UHF. - Ultra high frequency, 300 to 3, 000 mc.

Ultimate Factor of Safety. - The ratio of the design ultimate load on a structure to the limit load.

VHF. - Very high frequency, 30 to 300 mc.

Wall Area. - That vertical area between the floor and ceiling area which begins at the intersection of floor and vertical wall and ends at the intersection of vertical wall and sloping wall or ceiling.

Warning Light. - A placard light used to inform the crew members of the existence or occurrence of a hazardous condition requiring immediate corrective action, such as fire warning or cabin pressure failures.

X-Axis. - The X-axis shall be parallel to the nominal launch axis of the SC, and shall be positive in the direction of initial flight.

Y-Axis. - The Y-axis shall be normal to the X-axis, and positive to the right of a crewman when the crewman is facing toward positive X in the launch position.

Yield Factor of Safety. - The ratio of the design yield load on a structure to the limit load.

Z-Axis. - The Z-axis shall be normal to both the X- and Y-axes and shall be positive in the direction of the crewman's feet.

6.2 Abbreviations. - The following abbreviations and meanings appear in this specification:

ac - alternating current
AM - amplitude modulation
BMAG - body-mounted attitude gyros
Btu - British thermal unit
CDU - coupling display unit
CM - command module
cw - continuous wave
cps - cycles per second
CTE - central timing equipment
C&I - communications and instrumentation system
db - decibel
dc - direct current
DSIF - deep space instrumentation facility
ECS - environmental control system
ELS - earth landing system
EMI - electromagnetic interference

EPS	electrical power system
F	Fahrenheit
FM	frequency modulation
FDAI	flight director attitude indicator
fps	feet per second
GFAE	Government-furnished aeronautical equipment
GMT	Greenwich mean time
G&N	guidance and navigation system
GOSS	ground operational support system
GSE	ground support equipment
HF	high frequency
IF	intermediate frequency
IFTS	in-flight test system
IMU	inertial measuring unit
kc	kilocycle
kmc	kilomegacycle
kwh	kilowatt-hour
L/D	left-to-drag ratio
LEM	lunar excursion module
LES	launch escape system
LOR	lunar orbital rendezvous
mc	megacycle
MCC	mission control center
MEV	million electron volts
mm	millimeter
MMH	monomethylhydrazine
MSFC	Marshall Space Flight Center
NASA	National Aeronautics and Space Administration
NAA/S&ID	North American Aviation, Inc., Space and Information Systems Division
N_2O_4	nitrogen tetroxide

N_2H_4 - hydrazine
PACE - prelaunch automatic checkout equipment
PCM - pulse code modulation
PEP - peak envelope power
PLSS - portable life support system
PM - phase modulation
PPE - premodulation processor equipment
PRN - pseudo random noise
psi - pounds per square inch
psia - pounds per square inch absolute
psig - pounds per square inch gage
RAD - radiation absorbed dose
RBE - relative biological dose
RCS - reaction control system
REM - roentgen equivalent man
RF - radio frequency
rms - root mean square
SC - spacecraft
SCS - stabilization and control system
SM - service module
SPS - service propulsion system
TV - television
UDMH - unsymmetrical dimethyhydrazine
UHF - ultra high frequency
vdc - volts direct current
VHF - very high frequency

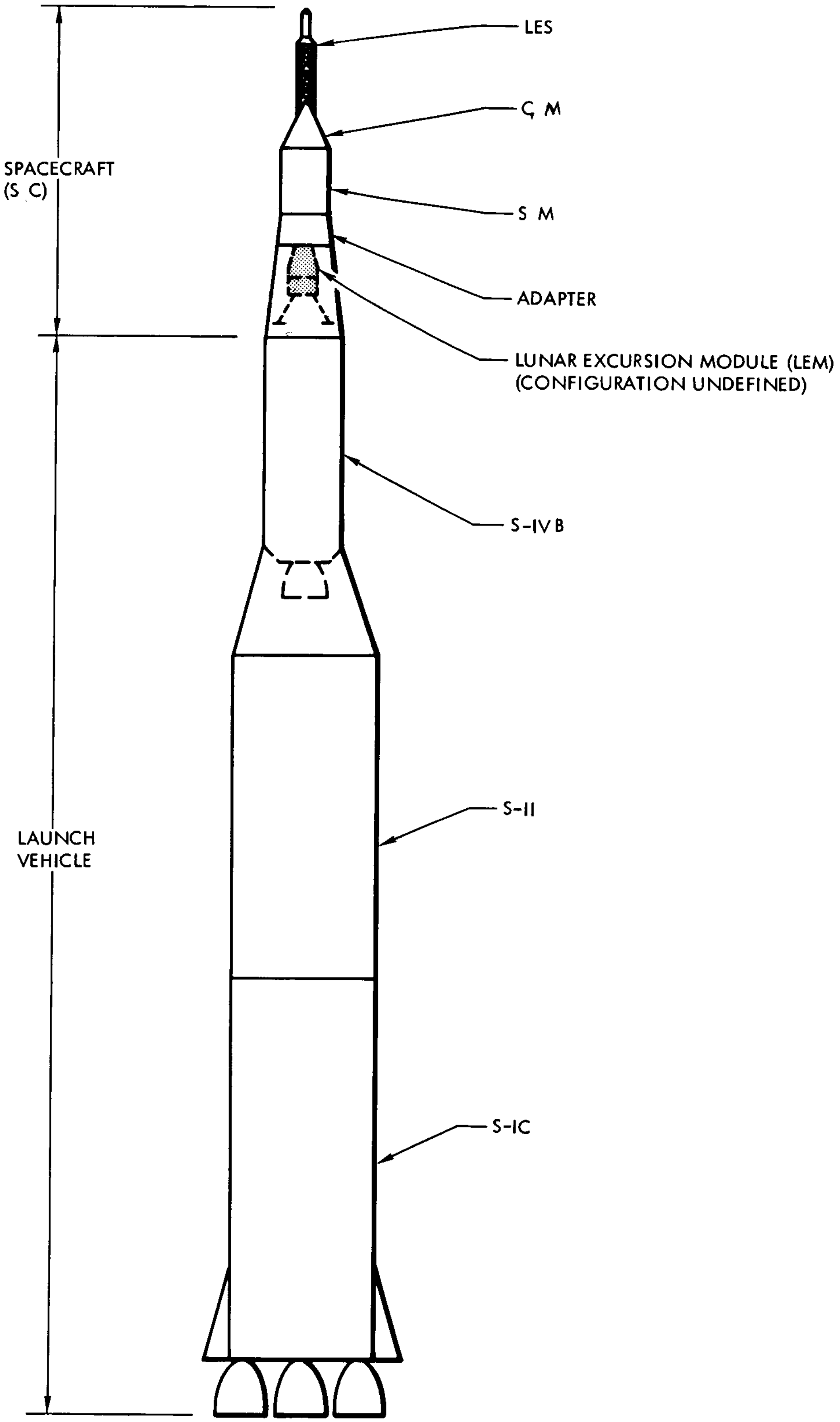

Figure 1 Space Vehicle Configuration Lunar Landing Mission
Launch to S-1C Burnout Nominal Mode

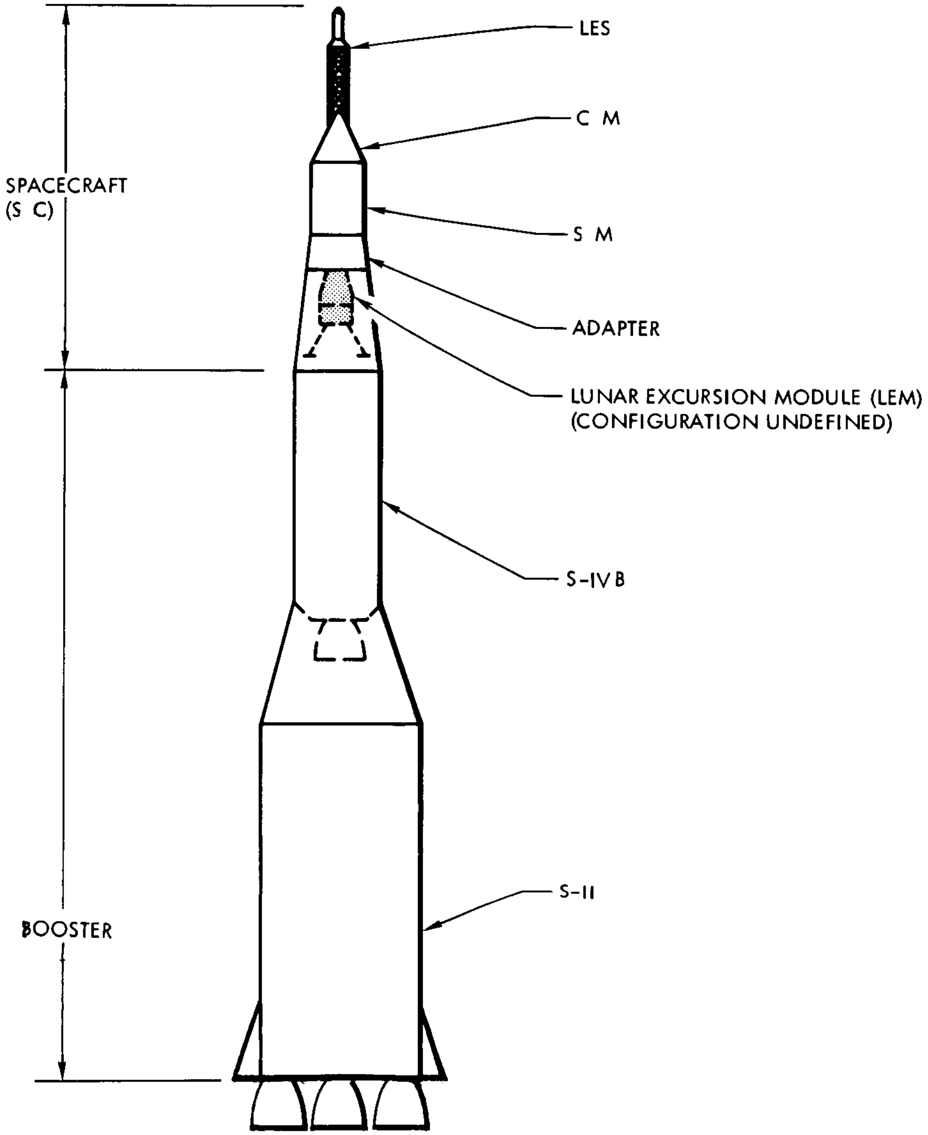

Figure 2 . Spacecraft Configuration S-IC Burnout

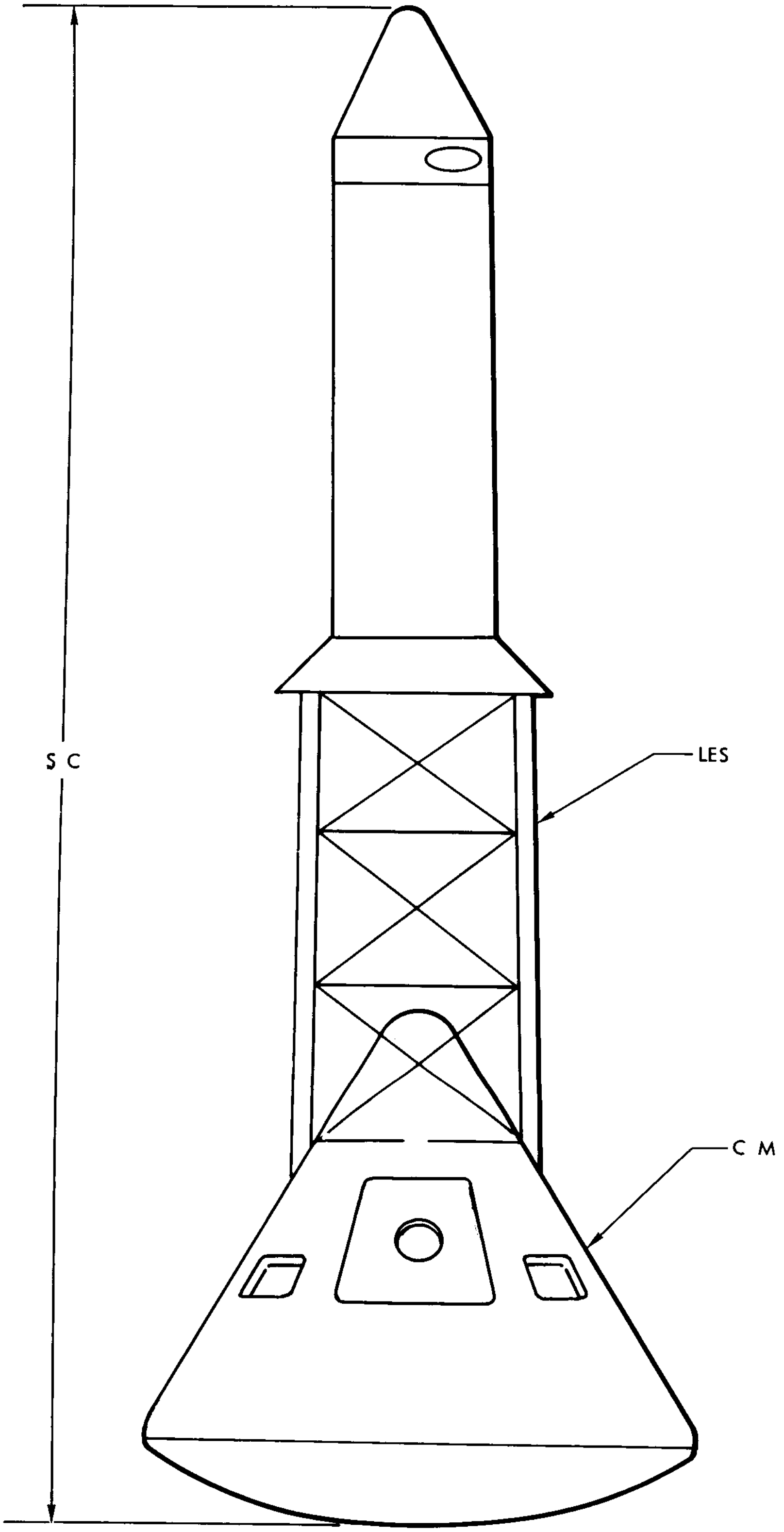

Figure 3. Spacecraft Configuration Launch to Tower Jettison Abort Mode

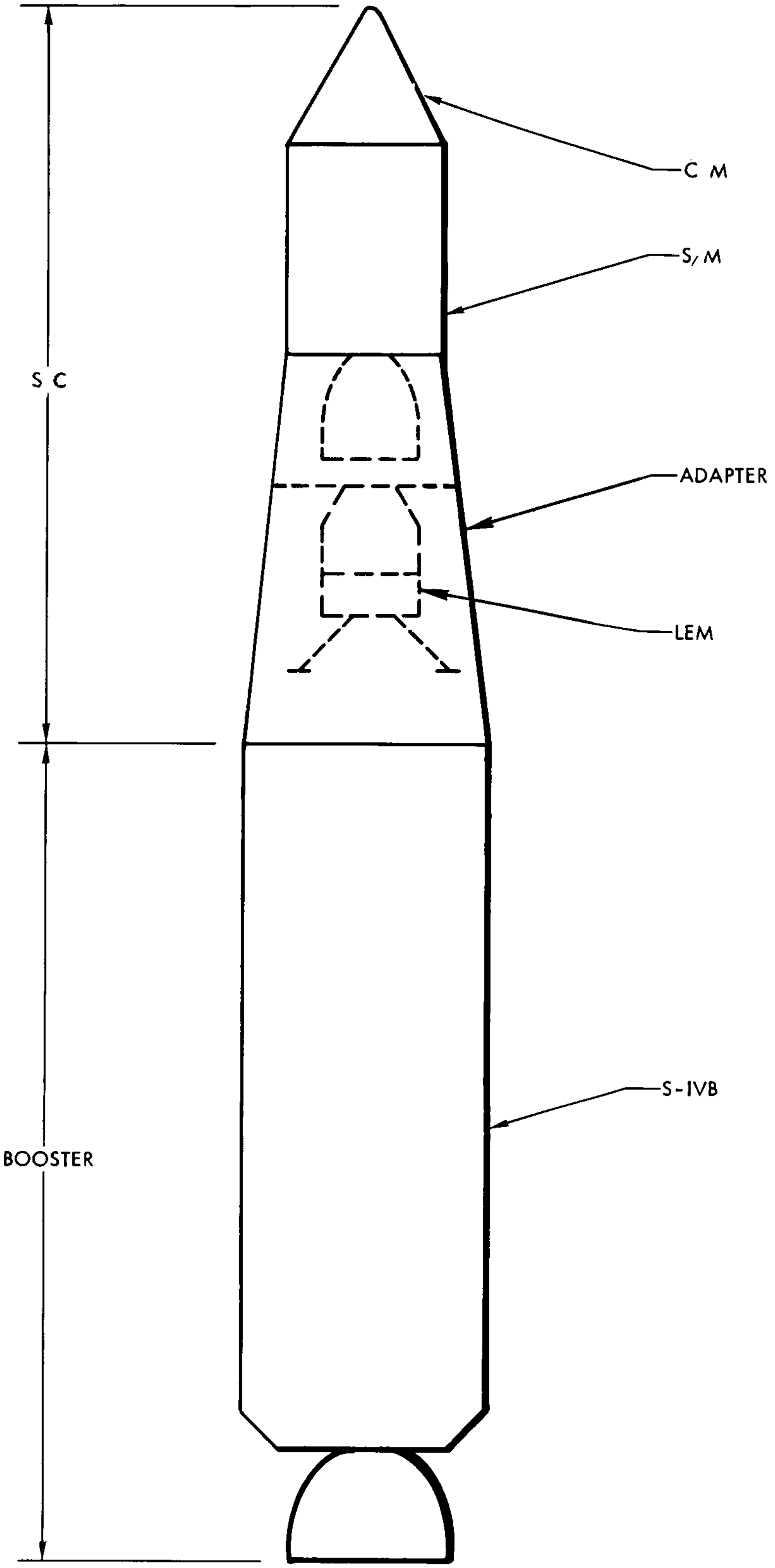

Figure 4 Spacecraft Configuration Tower Jettison to Translunar Injection Nominal Mode

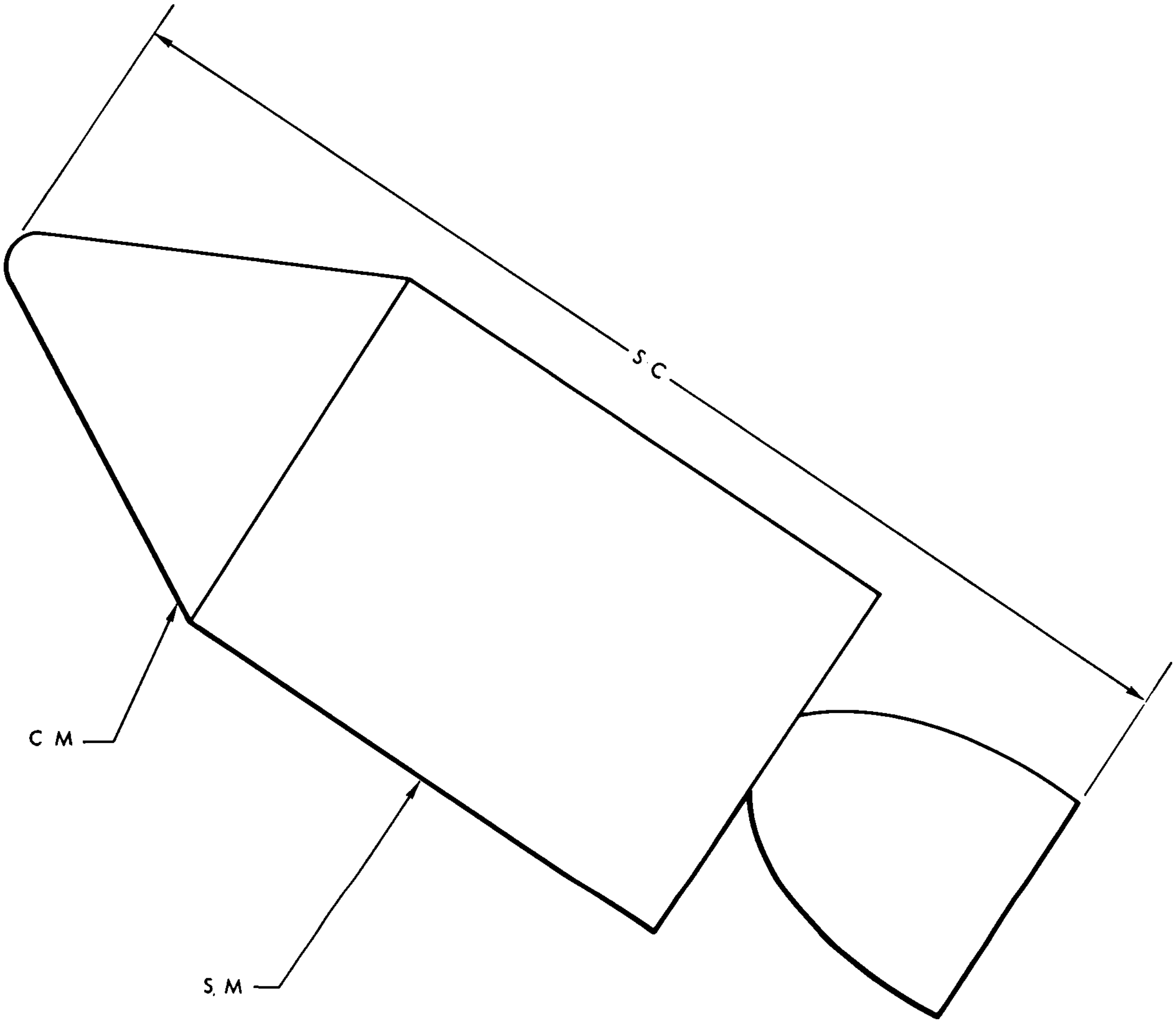

Figure 5. Spacecraft Configuration Tower Jettison to Translunar Injection Abort Mode

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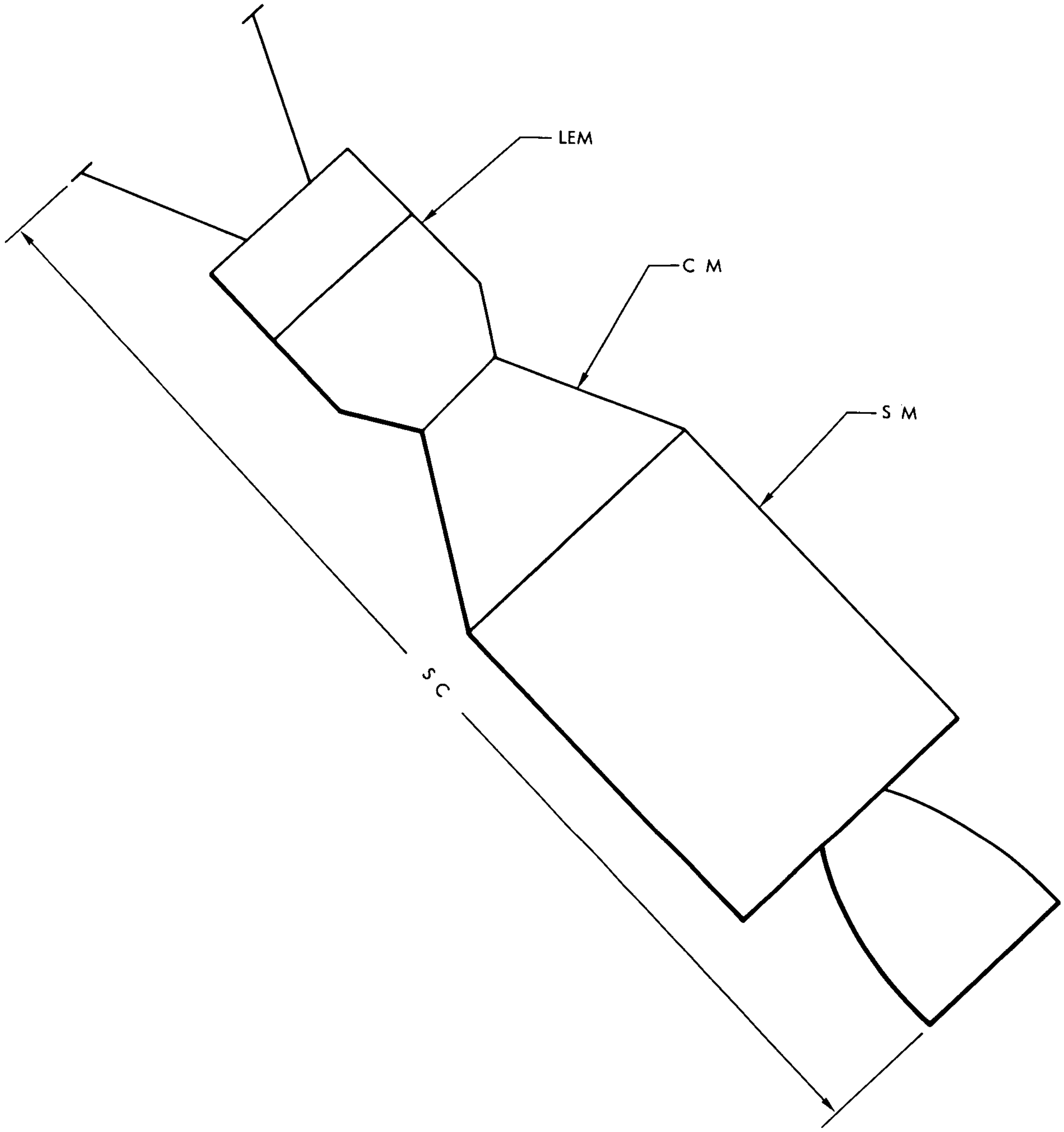

Figure 6. Spacecraft Configuration Translunar Injection to LEM Descent Nominal Mode

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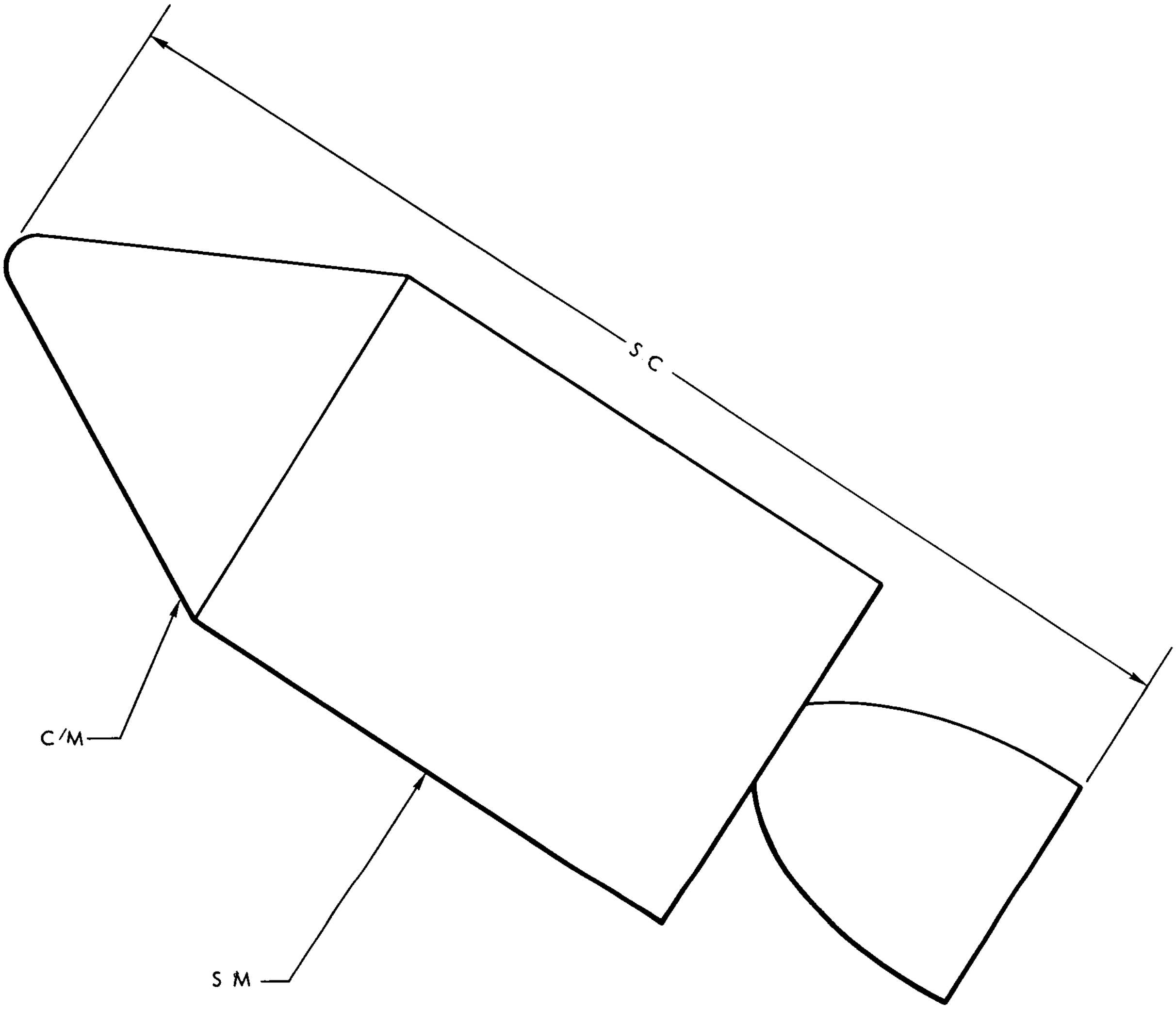

Figure 7. Spacecraft Configuration Translunar Injection to LEM Descent Abort Mode

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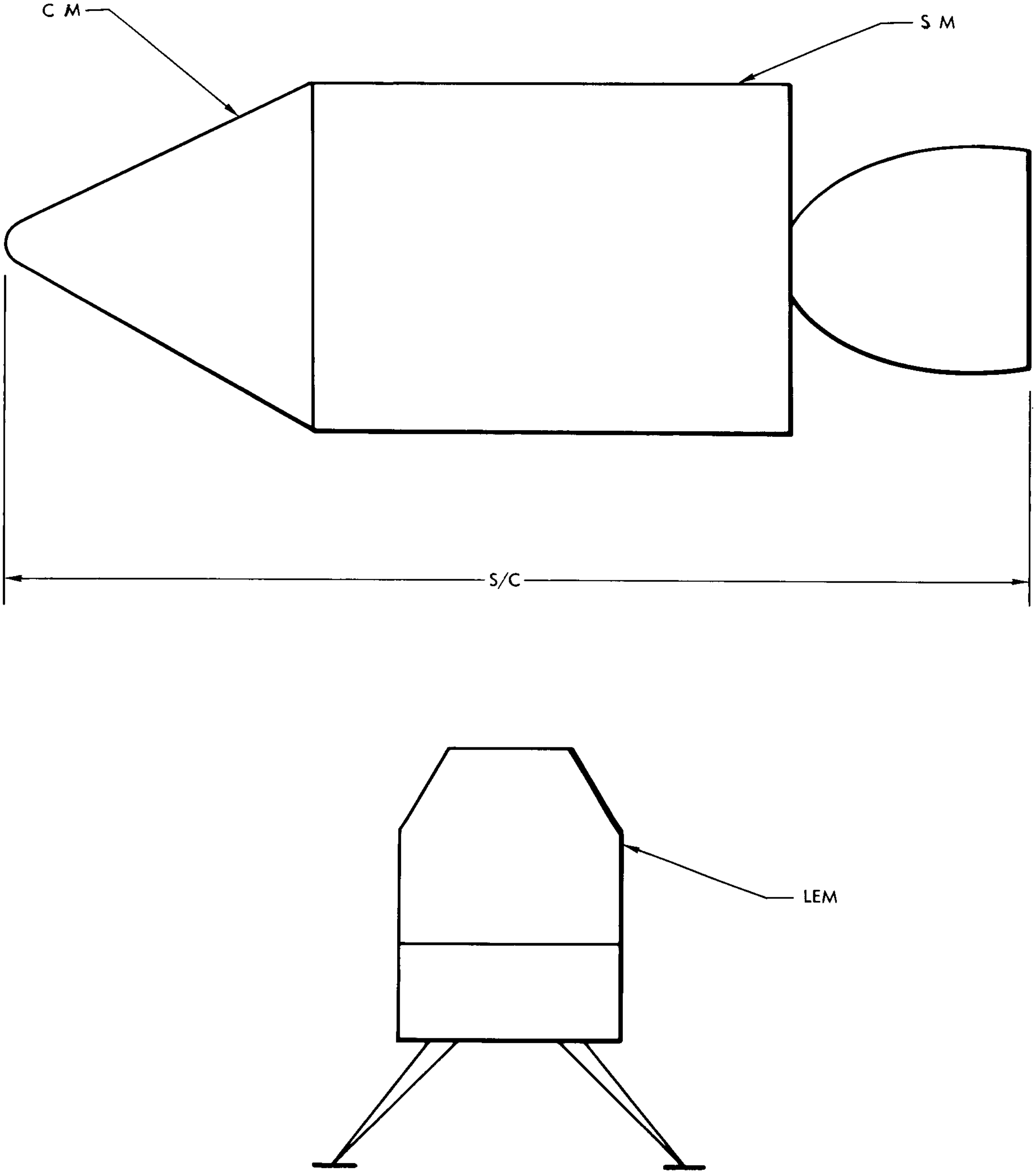

Figure 8. Spacecraft Configuration LEM Descent to Lunar Landing Nominal Mode

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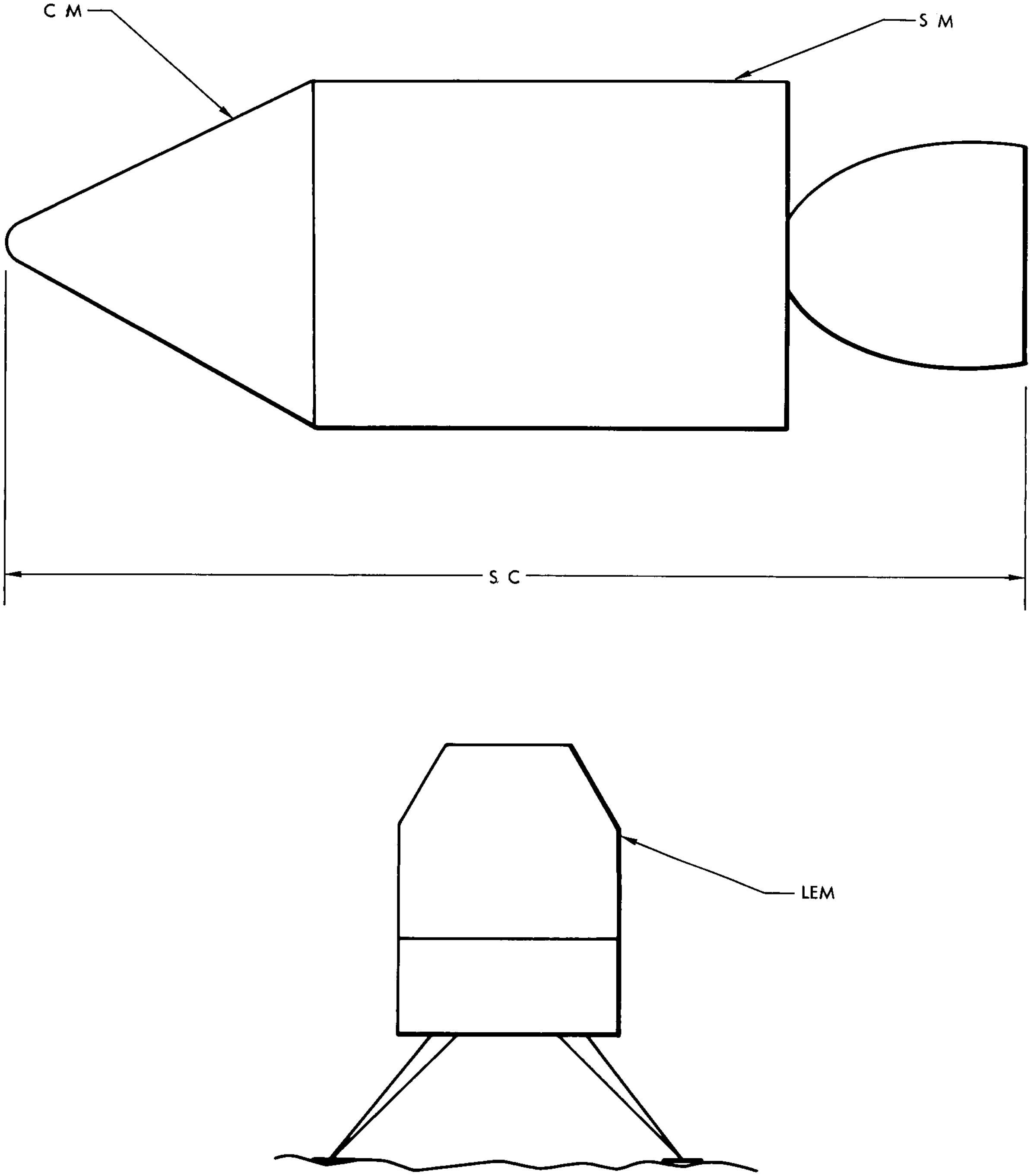

Figure 9. Spacecraft Configuration Lunar Landing to Lunar Launch Nominal and Abort Modes

[redacted]

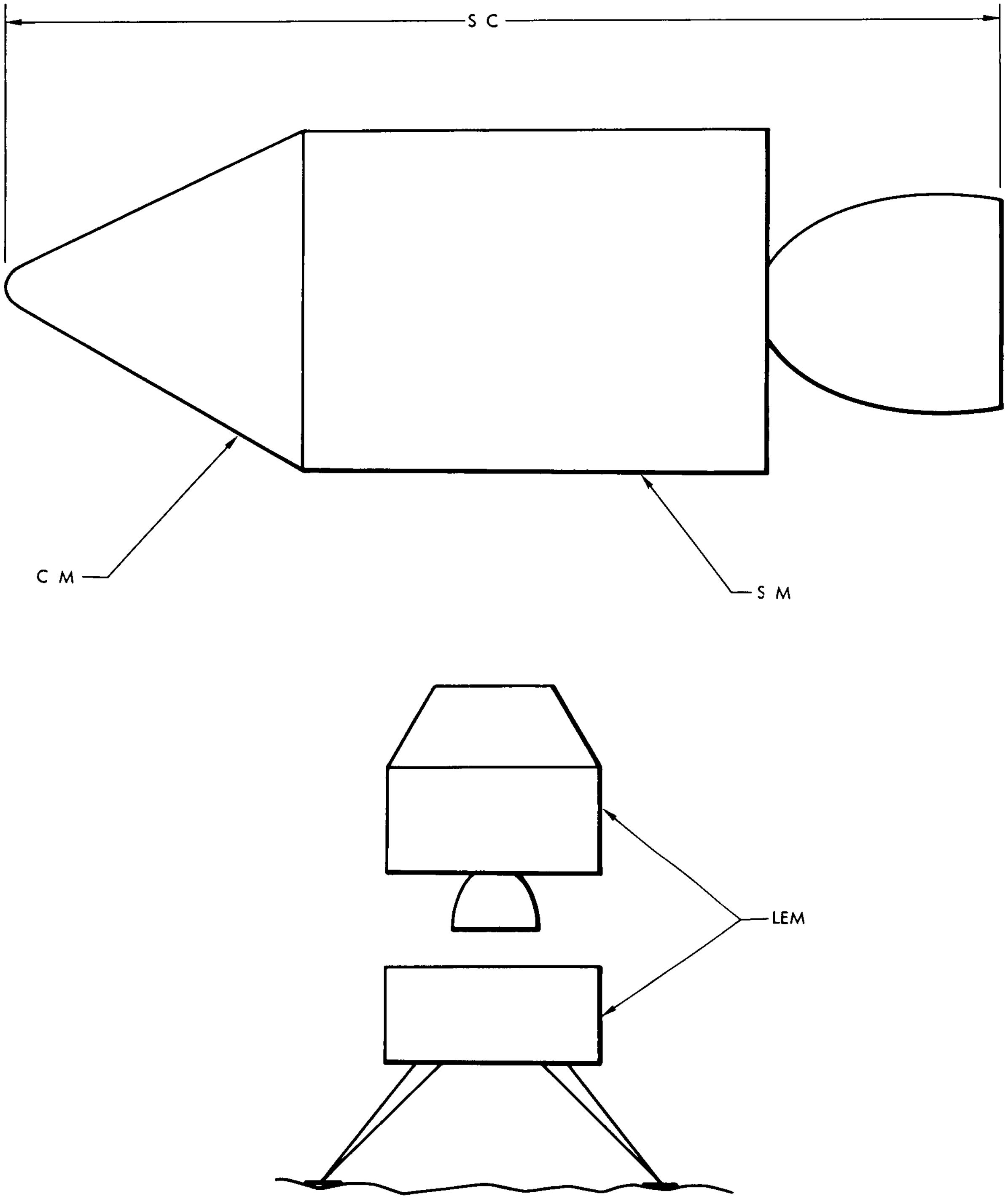

Figure 10. Spacecraft Configuration Lunar Launch to Rendezvous Nominal and Abort Modes

[redacted]

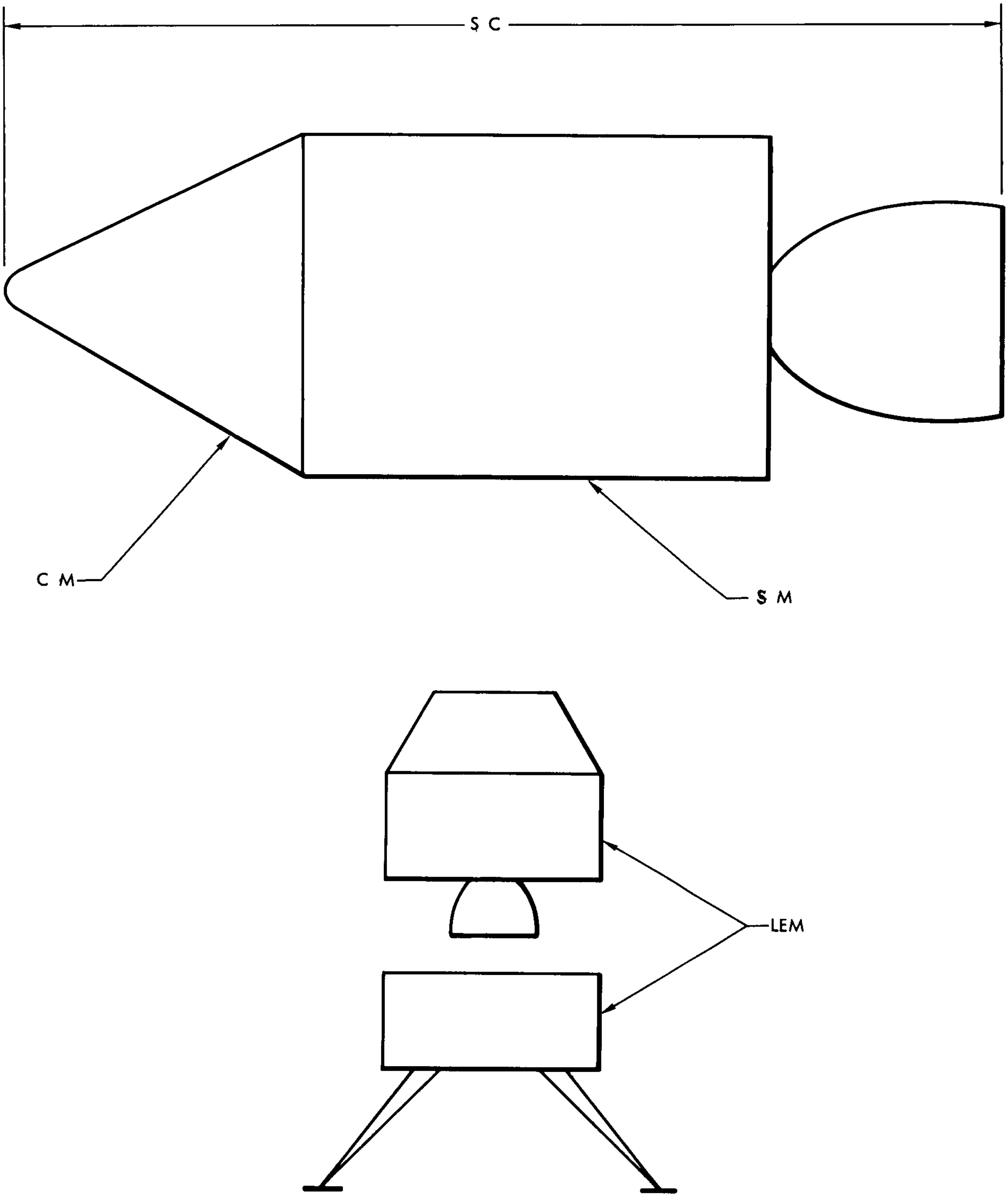

Figure 11. Spacecraft Configuration LEM Descent to Lunar Landing Abort Mode

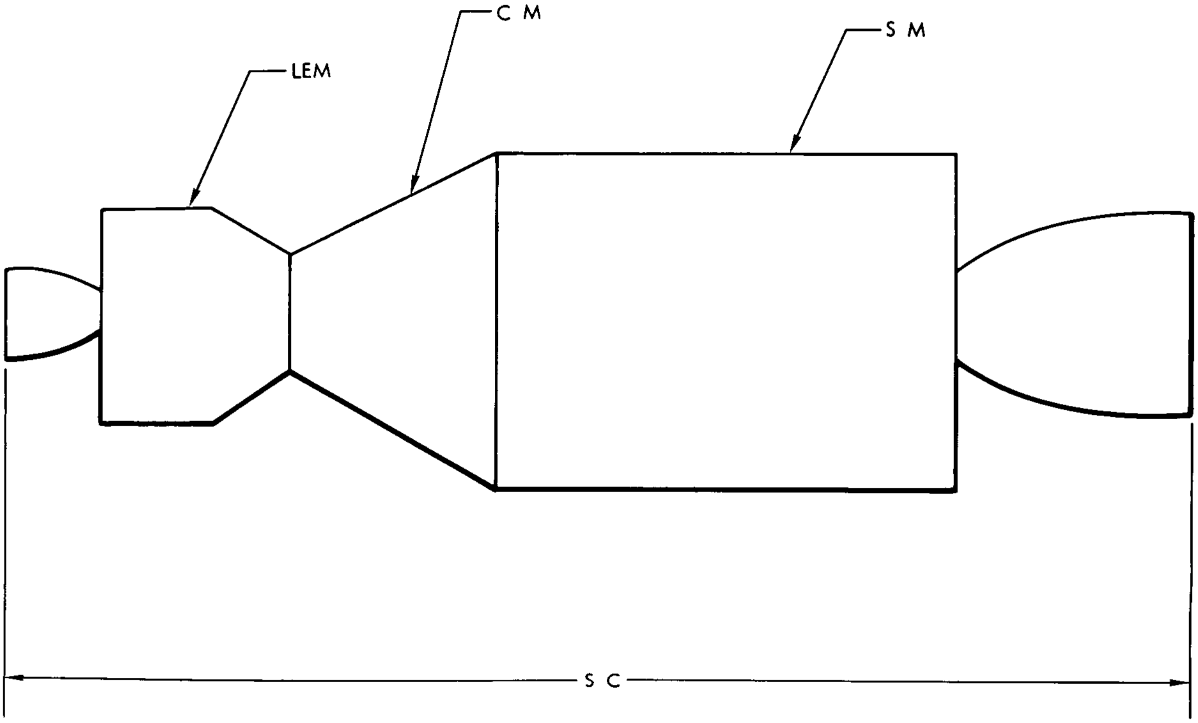

Figure 12. Spacecraft Configuration Rendezvous to LEM Jettison Nominal and Abort Modes

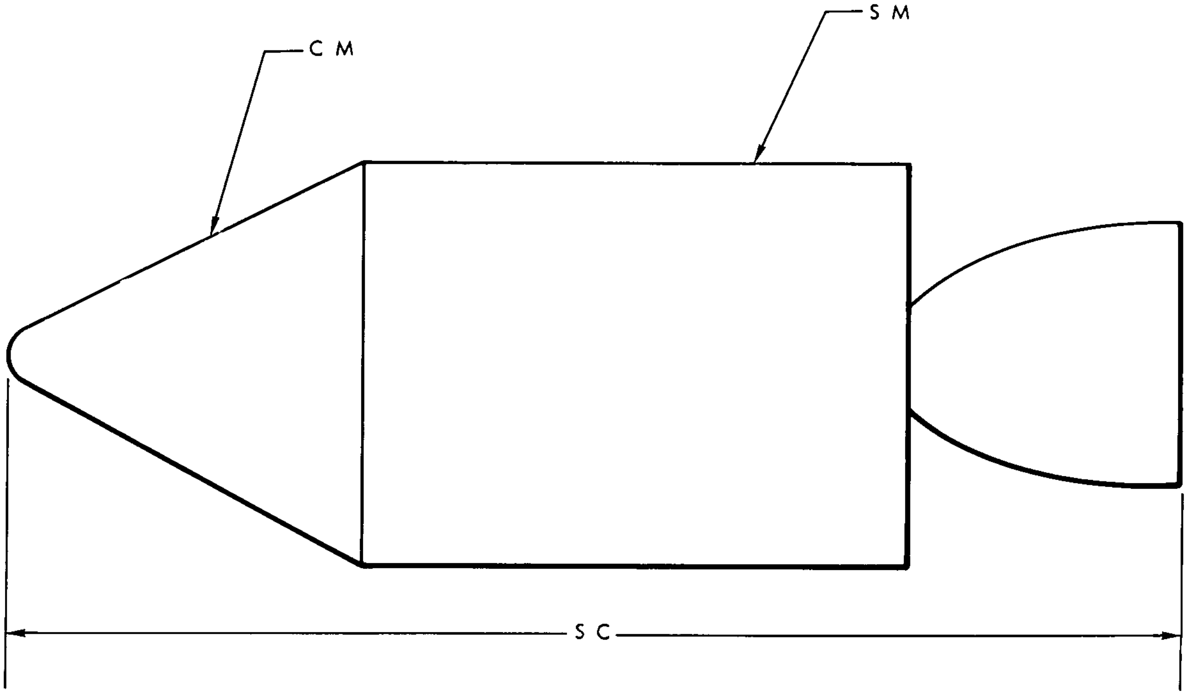

Figure 13. Spacecraft Configuration LEM Jettison to Entry Nominal and Abort Modes

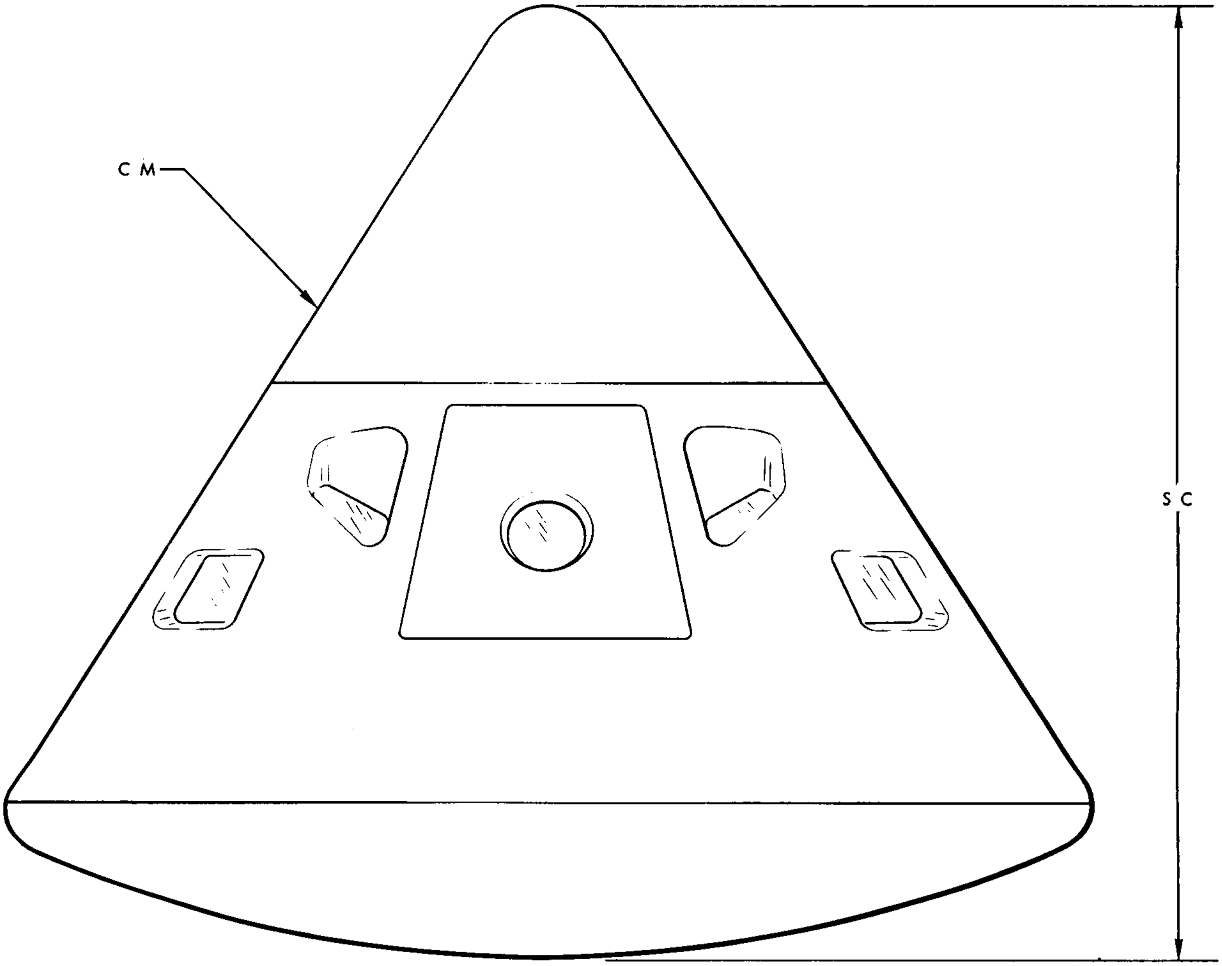

Figure 14. Spacecraft Configuration Entry to Touchdown Nominal and Abort Modes

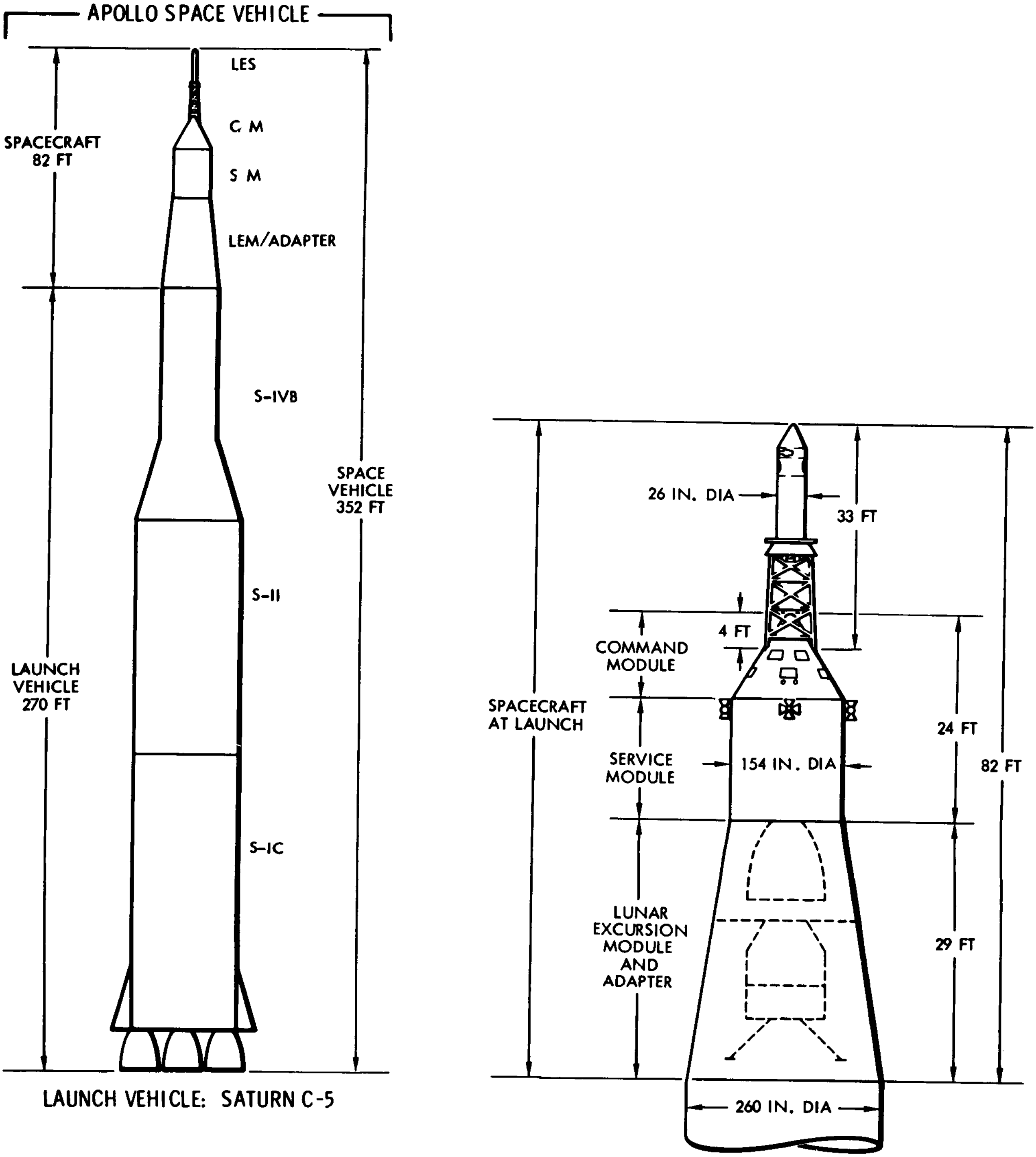

Figure 15. Apollo Space Vehicle

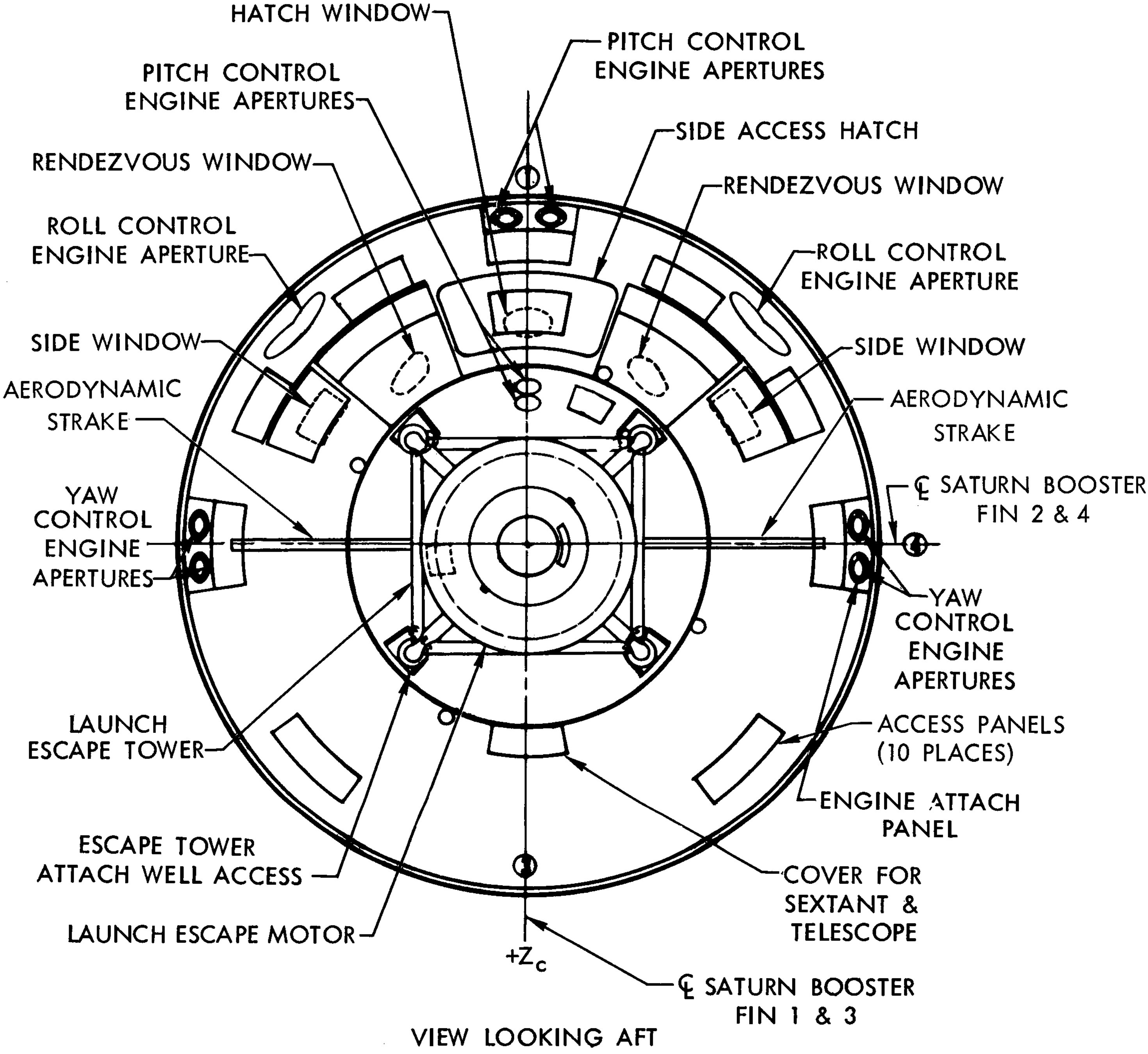

Figure 16. Apollo Space Vehicle

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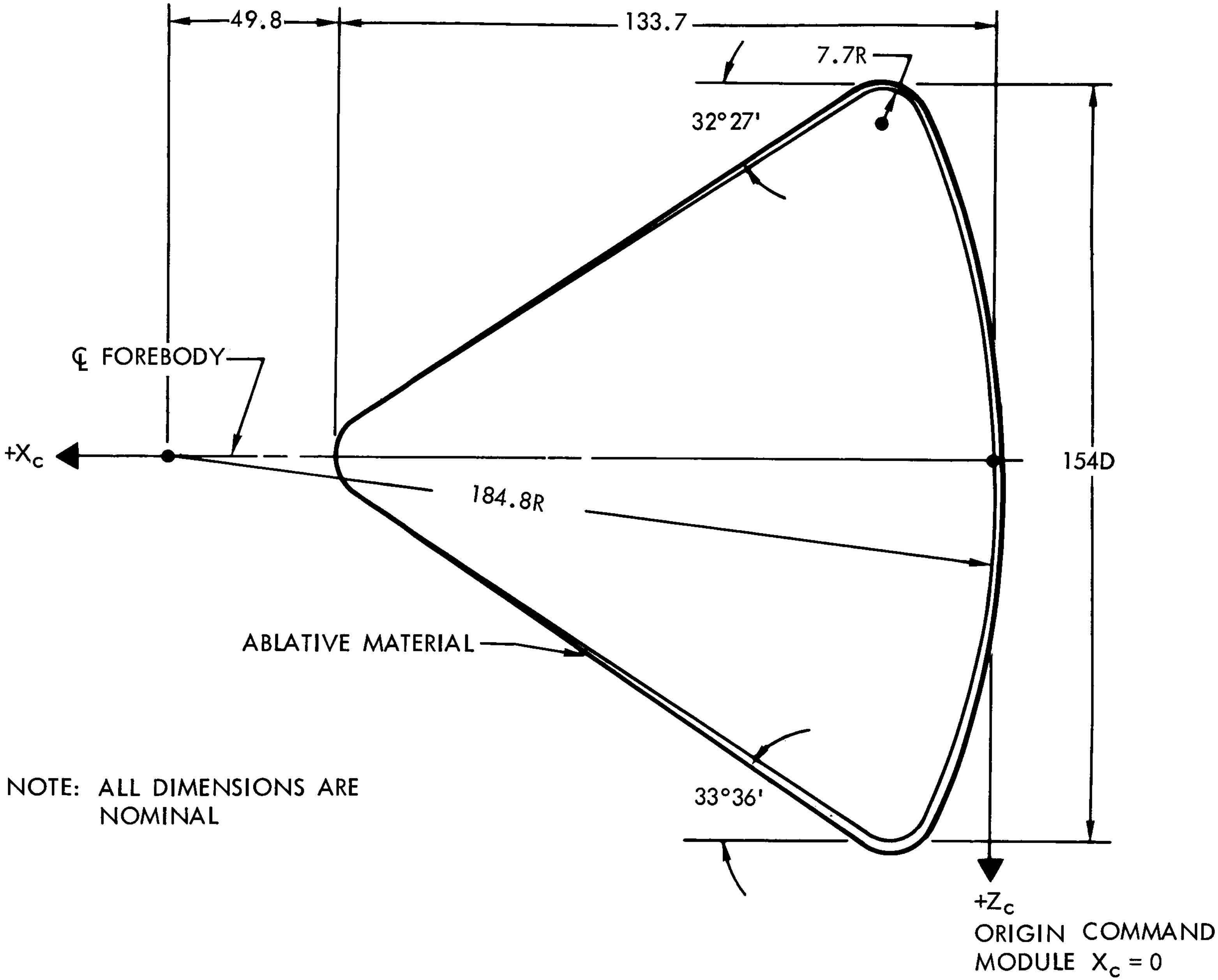

NOTE: ALL DIMENSIONS ARE NOMINAL

Figure 17. Command Module External Dimensions including Ablative Material

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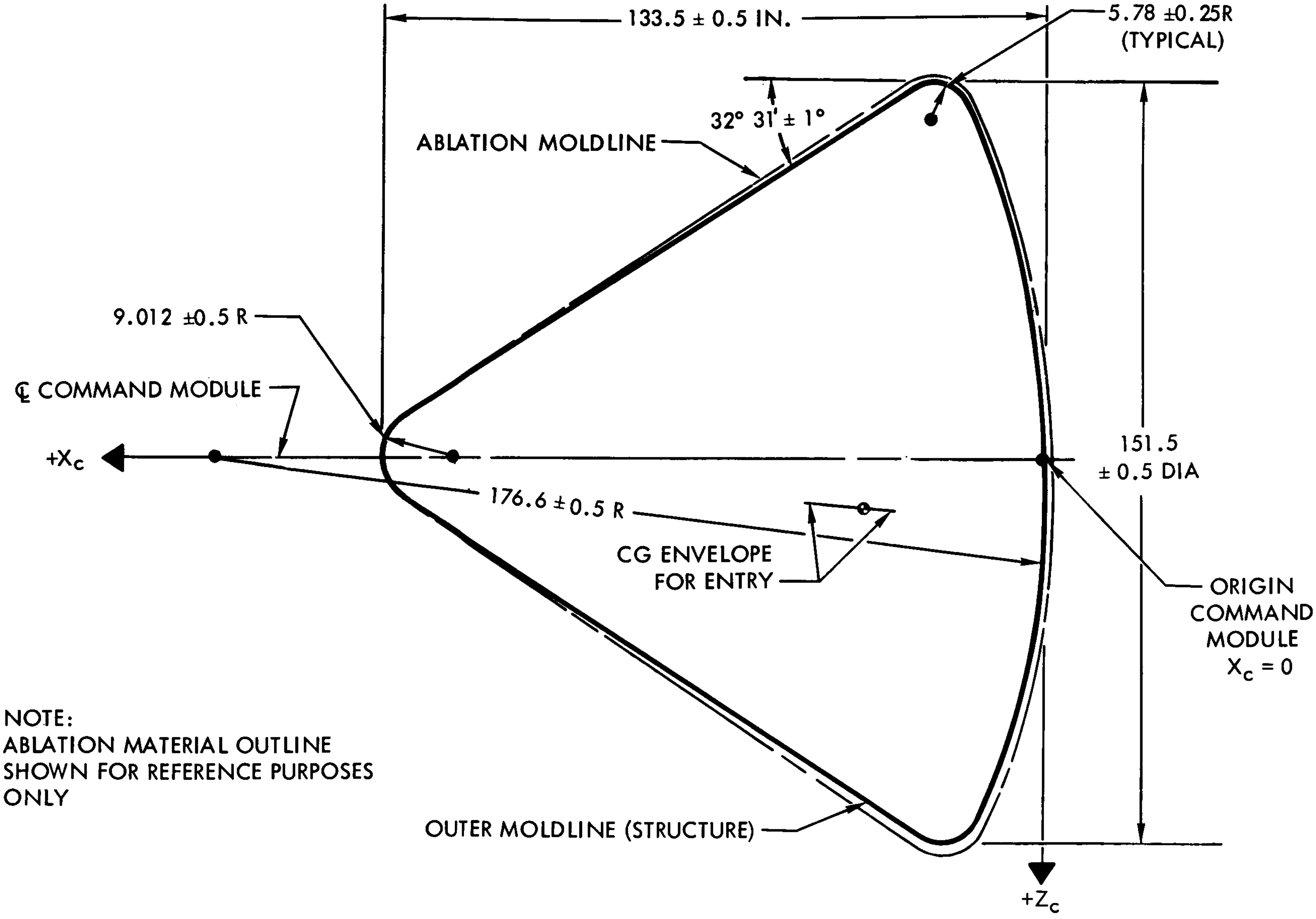

NOTE:
ABLATION MATERIAL OUTLINE
SHOWN FOR REFERENCE PURPOSES
ONLY

Figure 18. Command Module Structure Outline

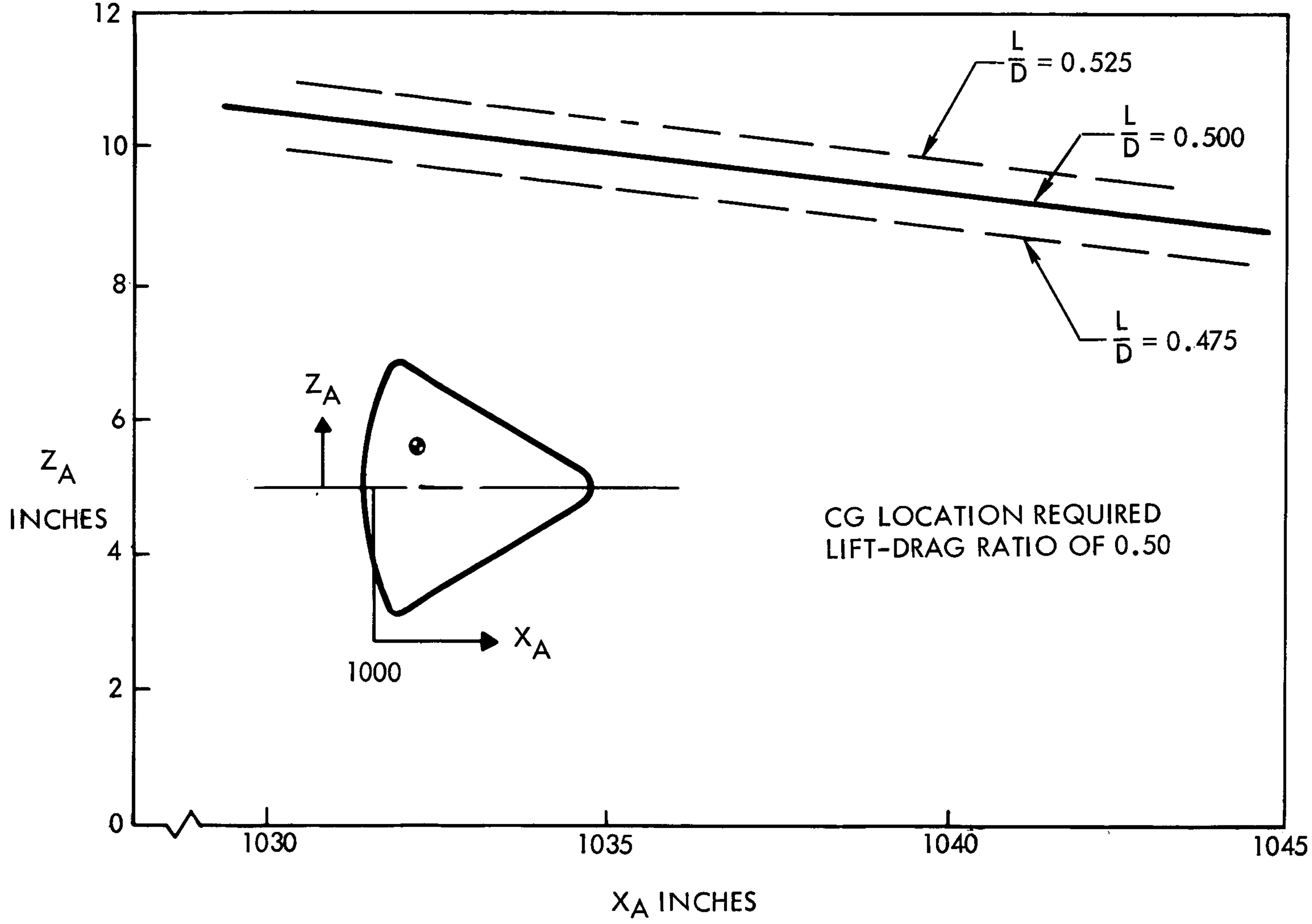

Figure 19. CG Location vs. Lift-Drag Ratio

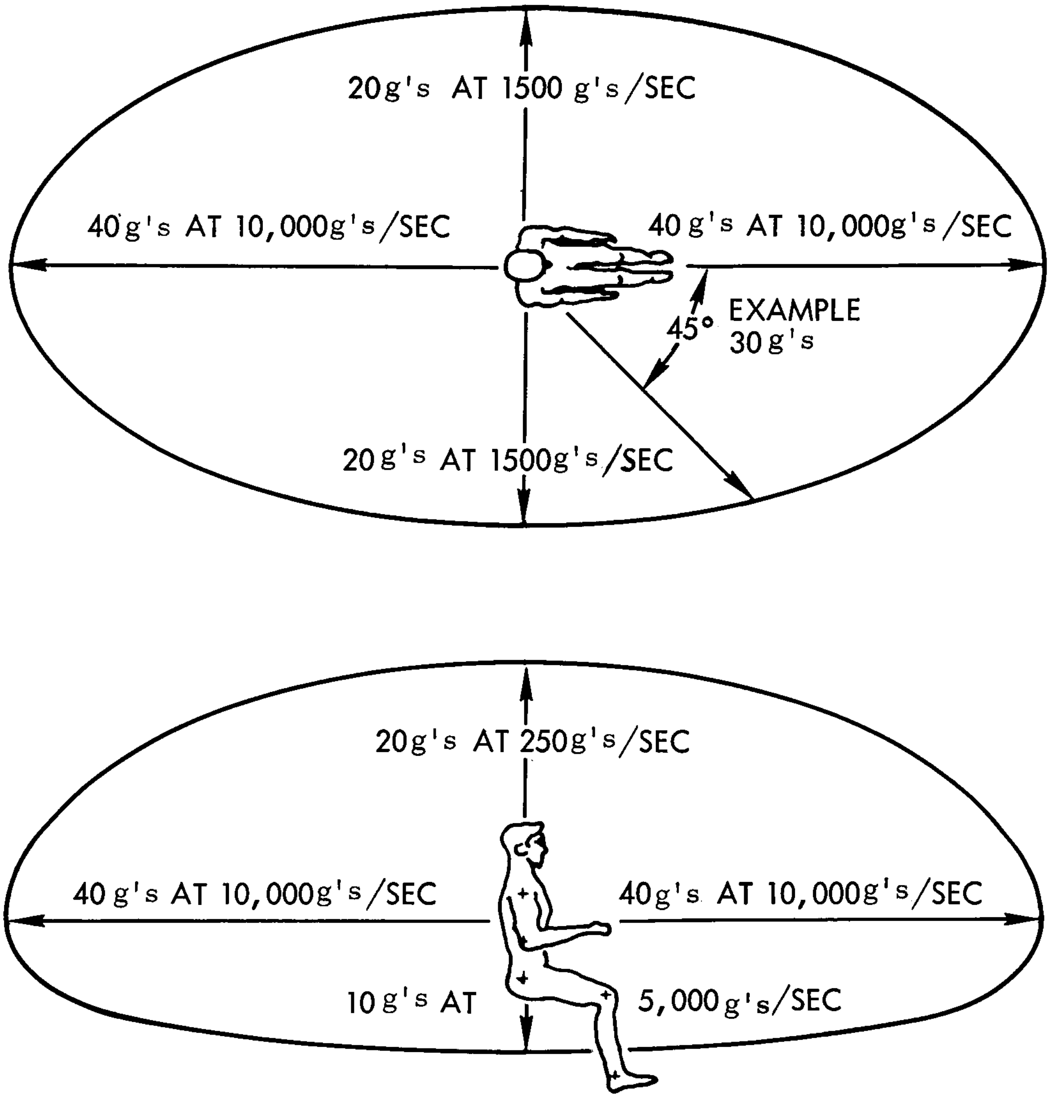

Figure 20. Maximum Impact Tolerance

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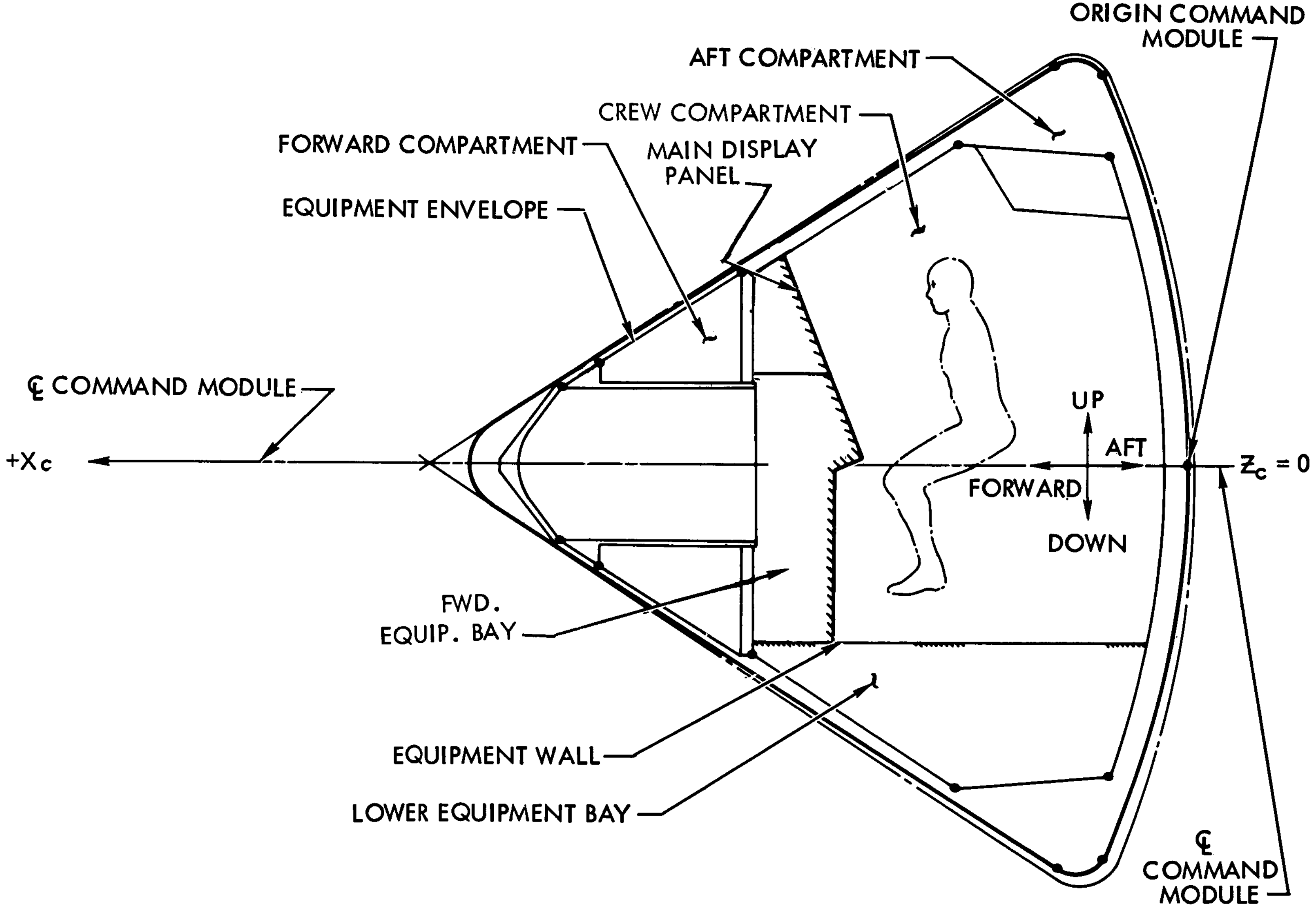

Figure 21. Area Designations Side View

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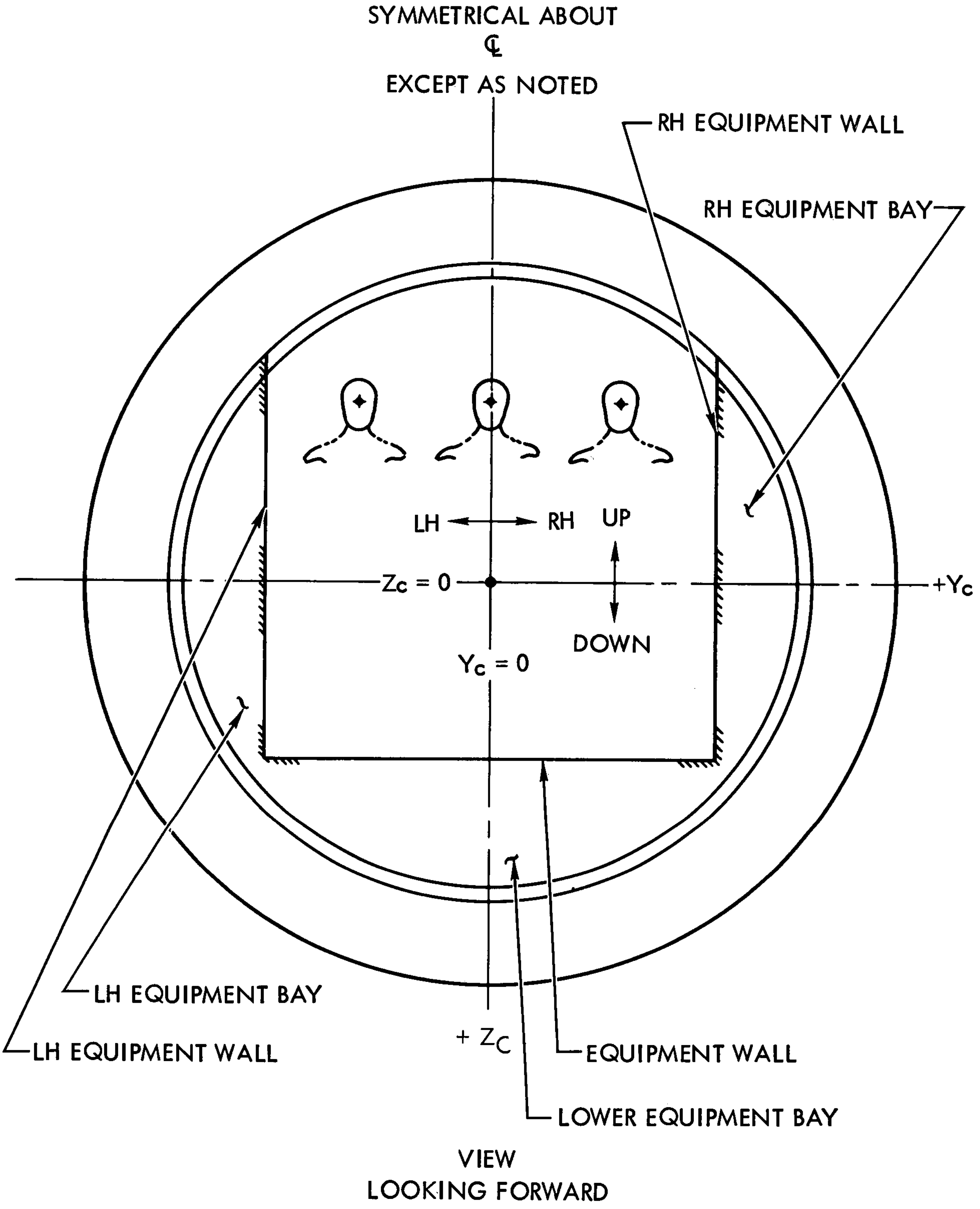

Figure 22. Area Designations

[redacted]

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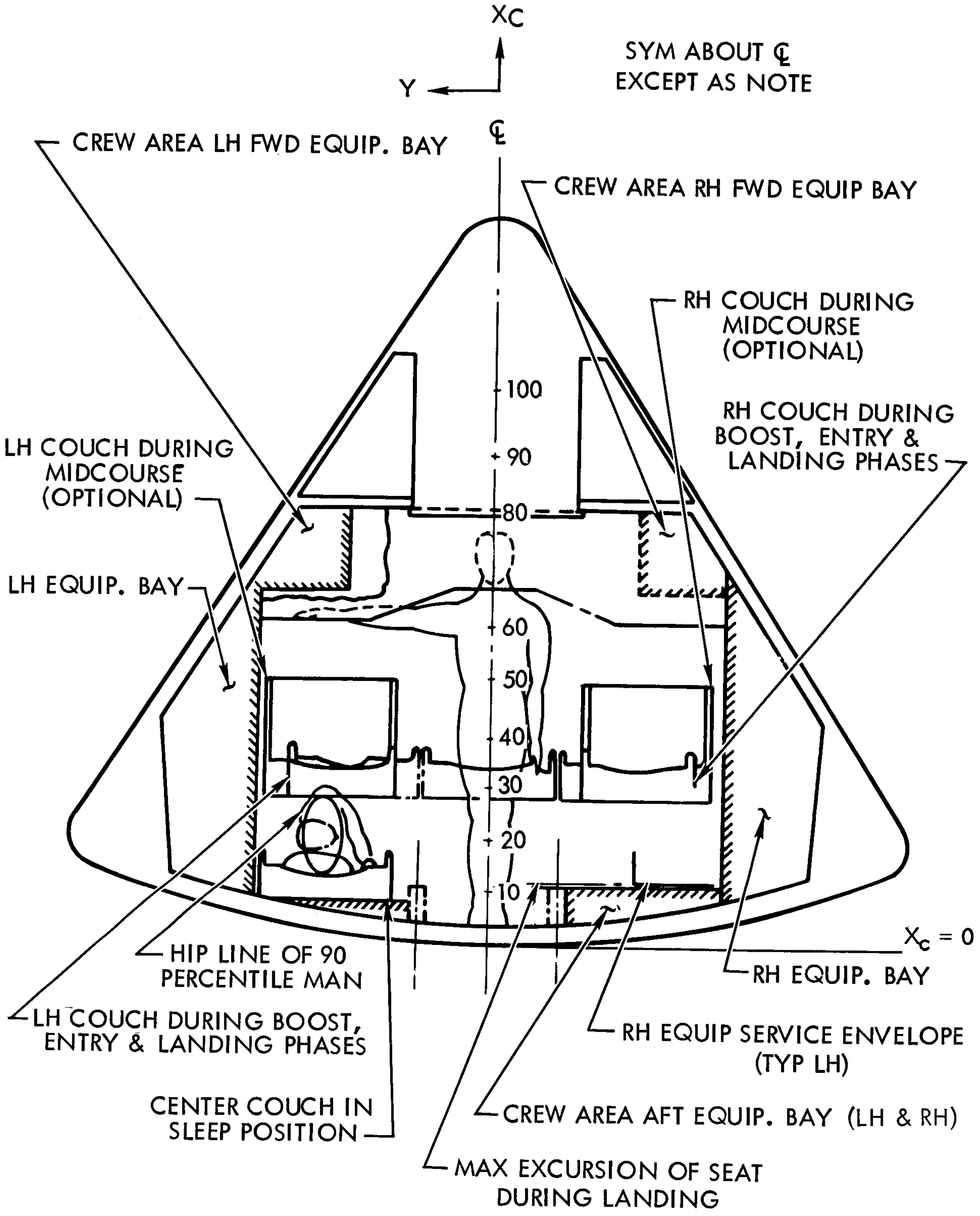

VIEW LOOKING TOWARD LOWER END OF CREW AREA

Figure 23. Area Designations

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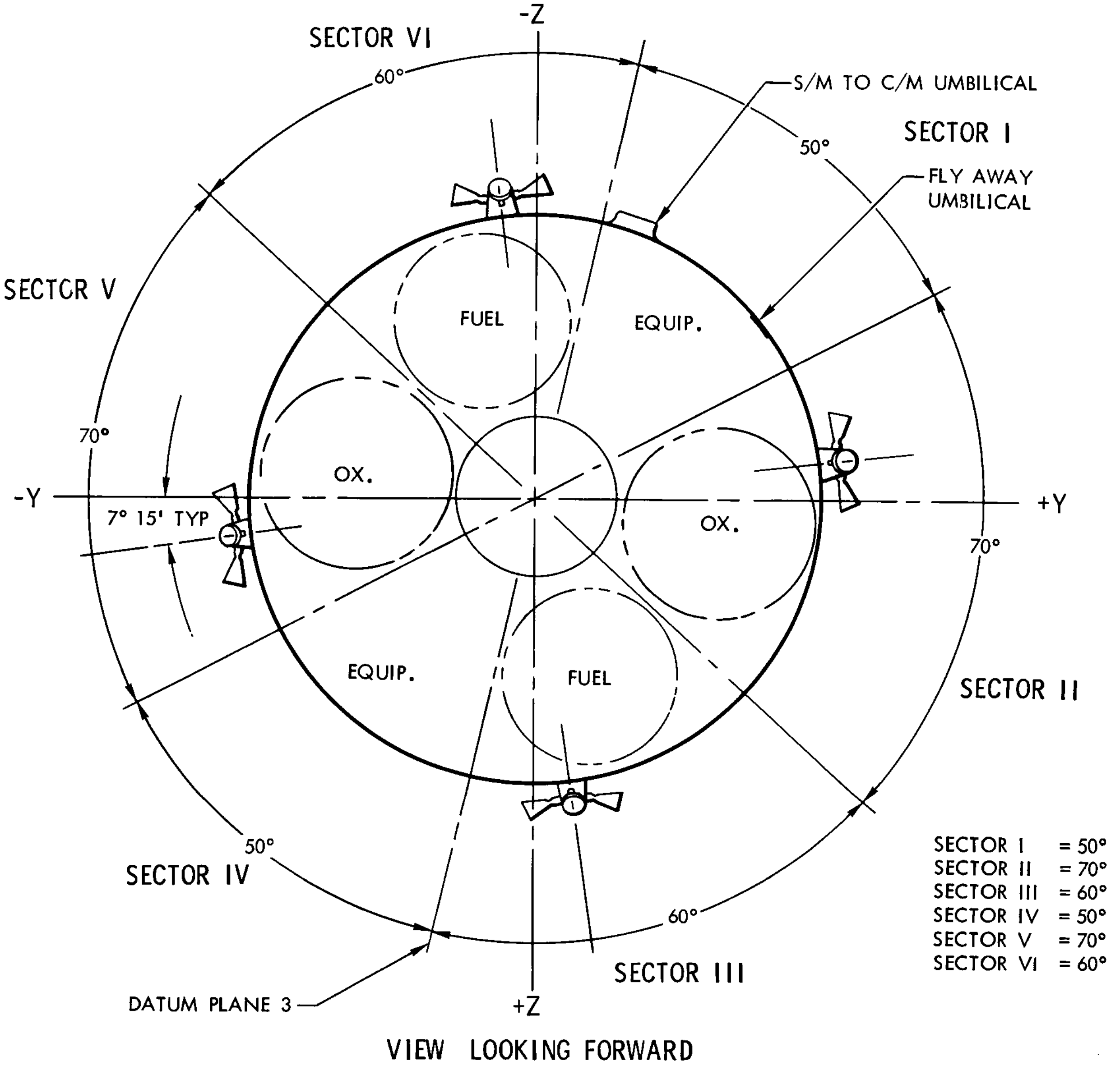

Figure 24. Service Module Inboard Profile

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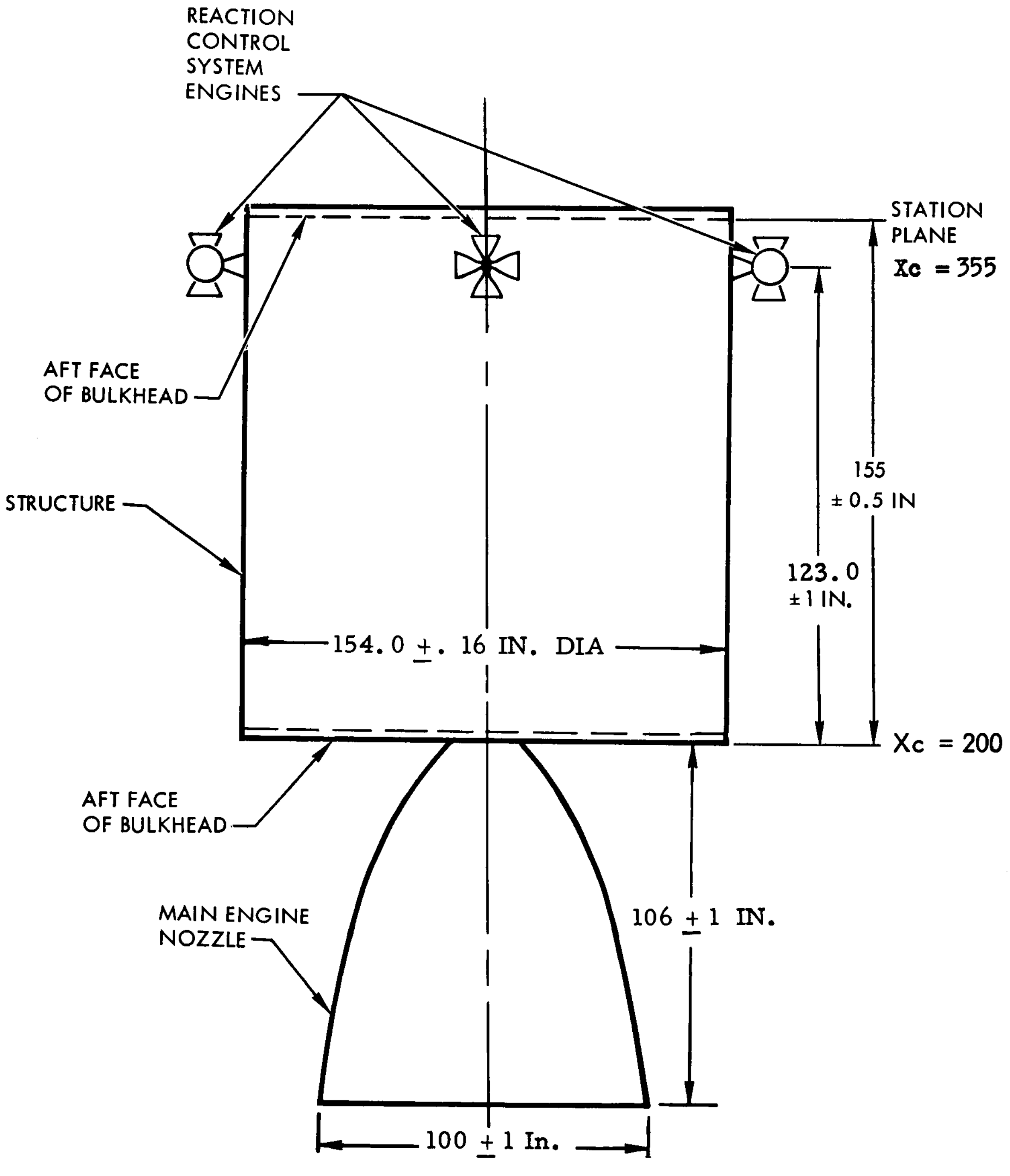

Figure 25. Service Module Inboard Profile

CO[redacted]

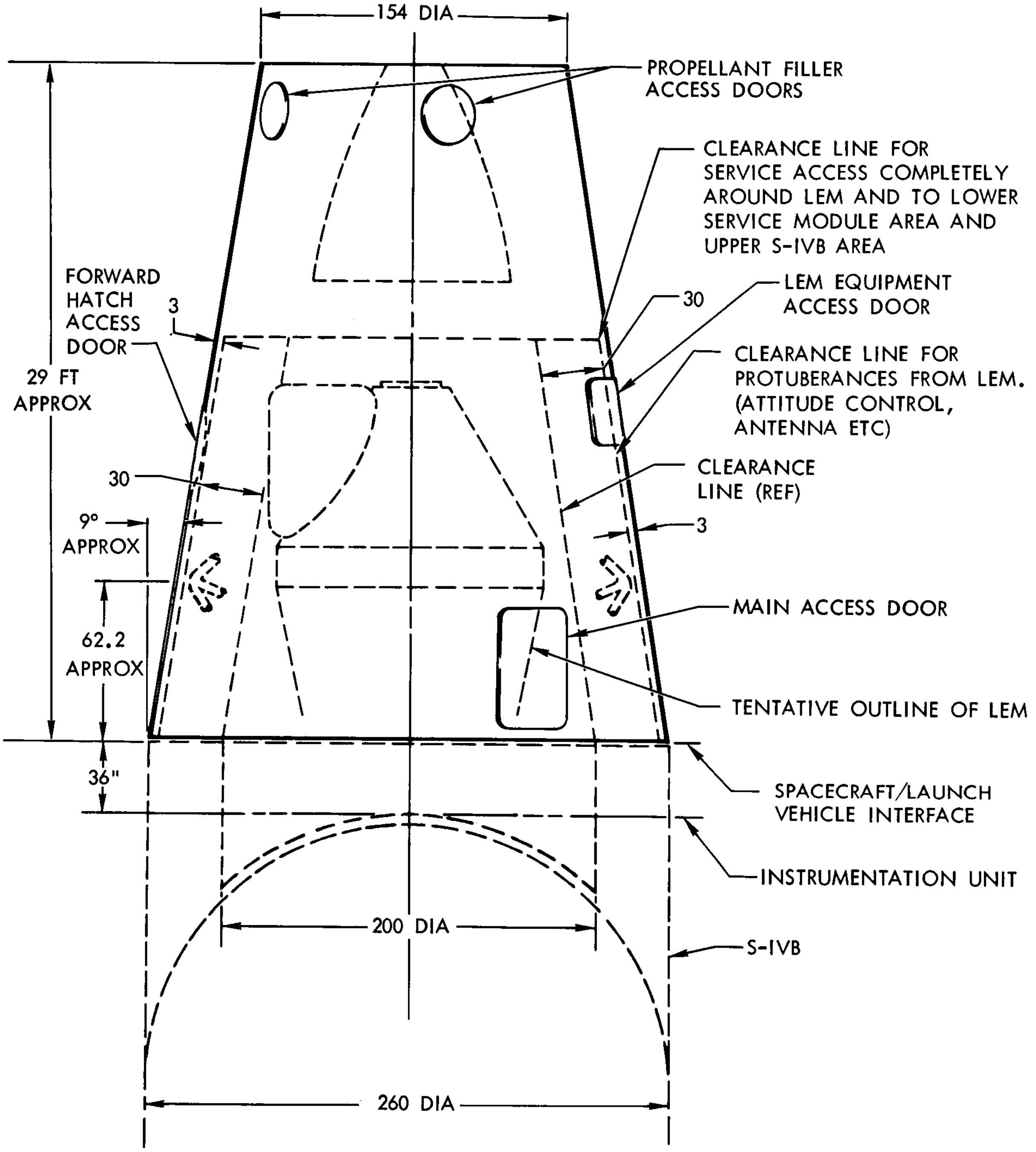

Figure 26. Apollo Spacecraft Adapter

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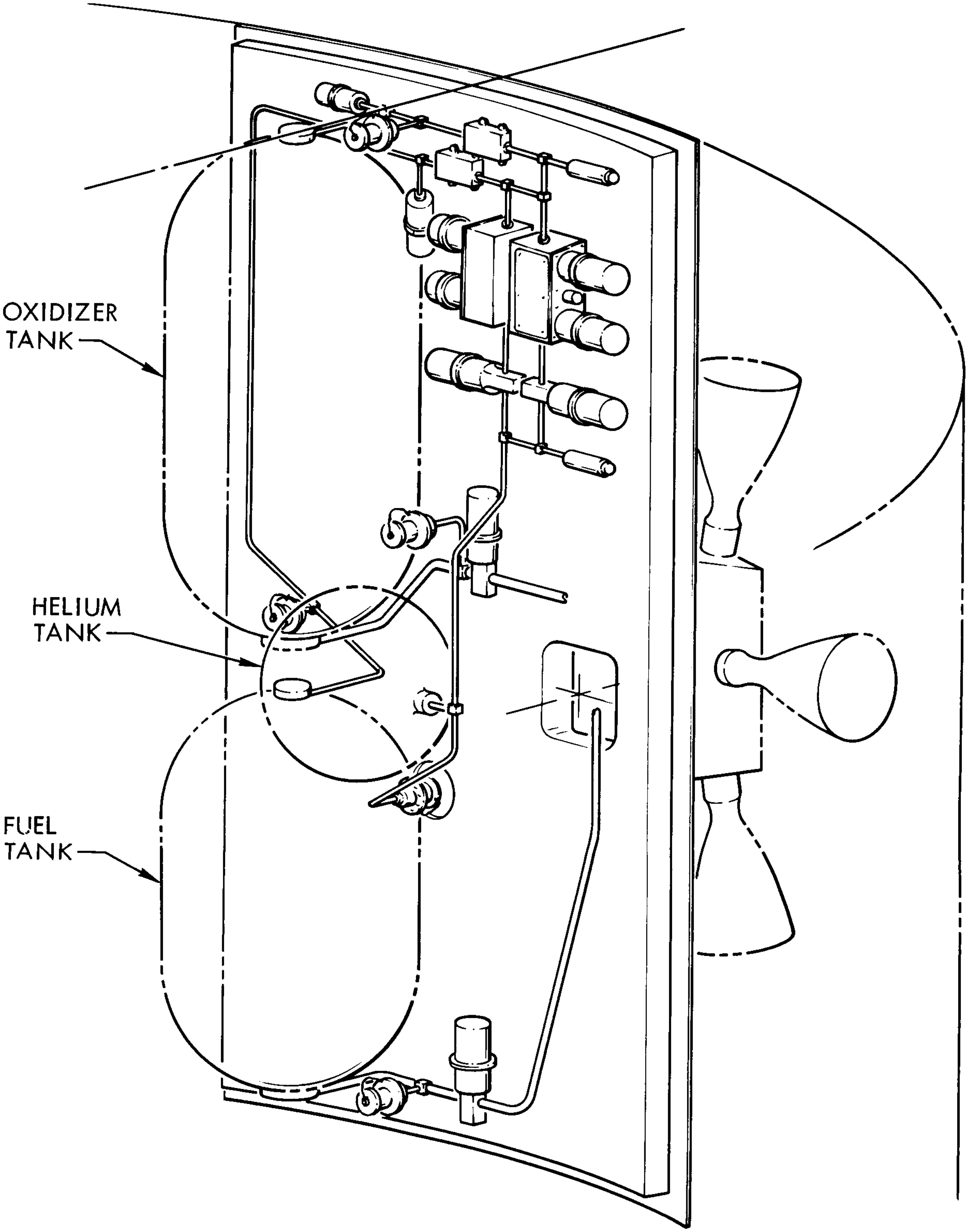

Figure 27. SM RCS Module

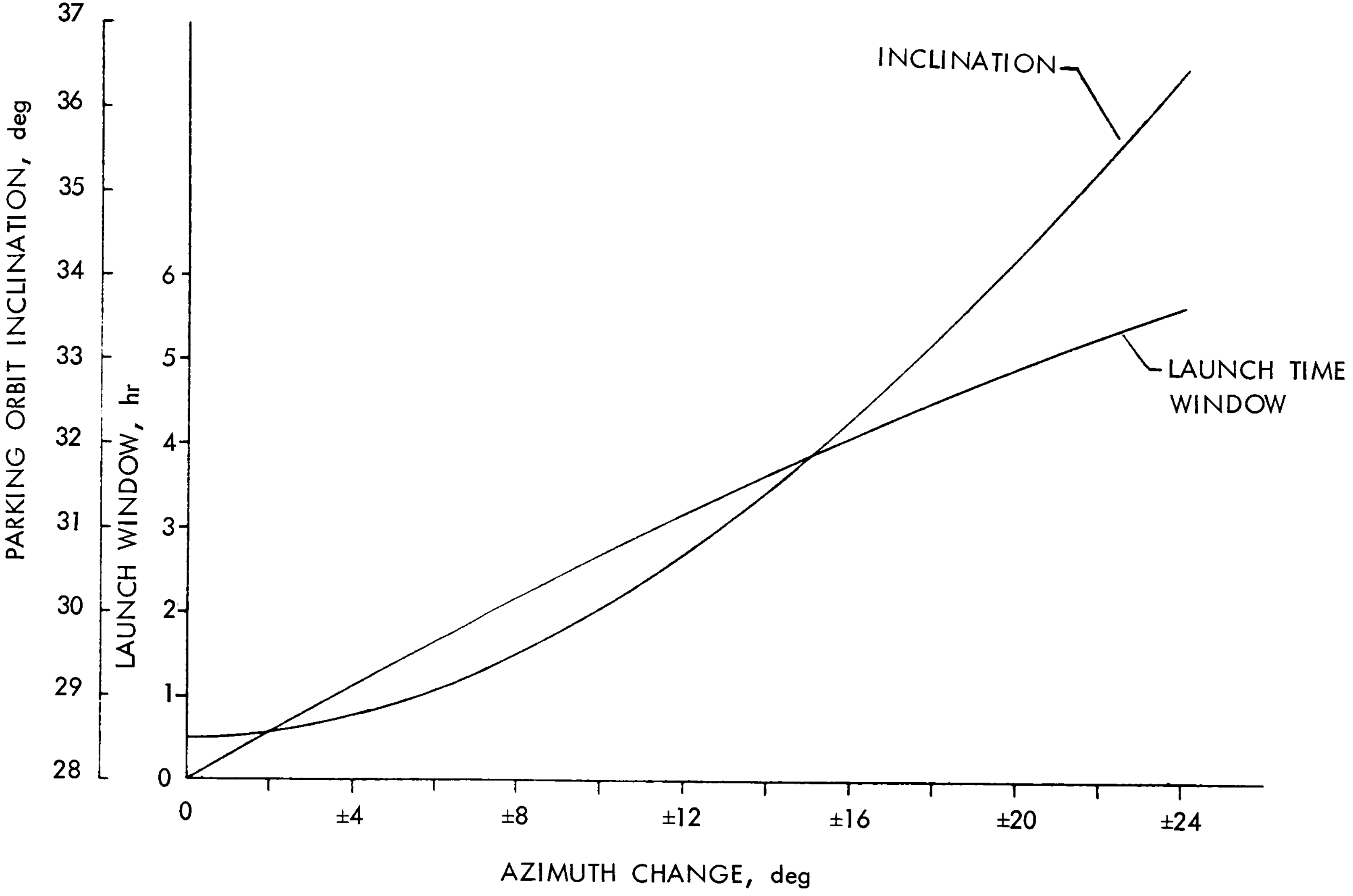

Figure 28. Launch Time Window for Variation in the Translunar Trajectory

[redacted]

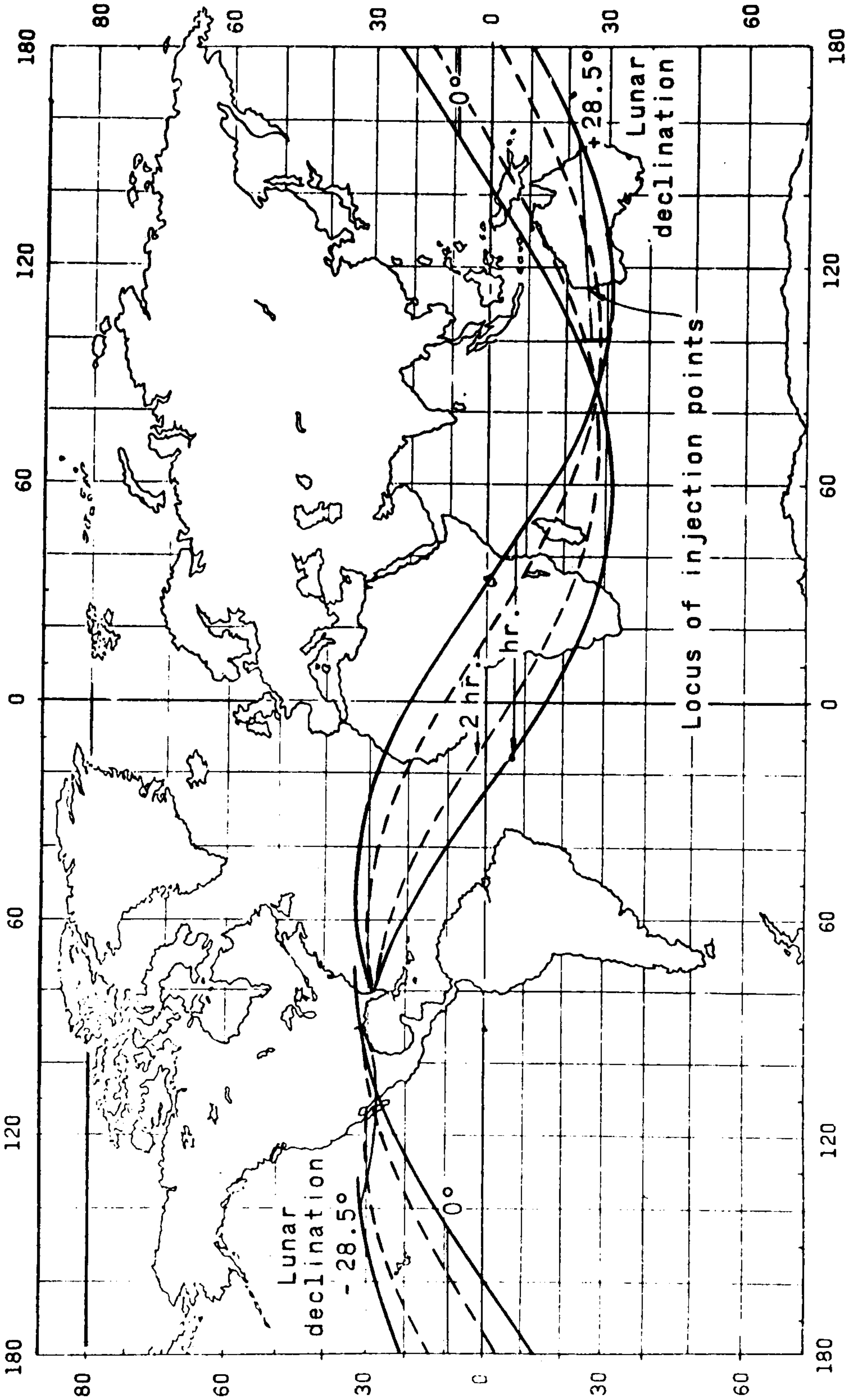

Figure 29. Parking Orbit Boundaries for Launch Time Window

[redacted]

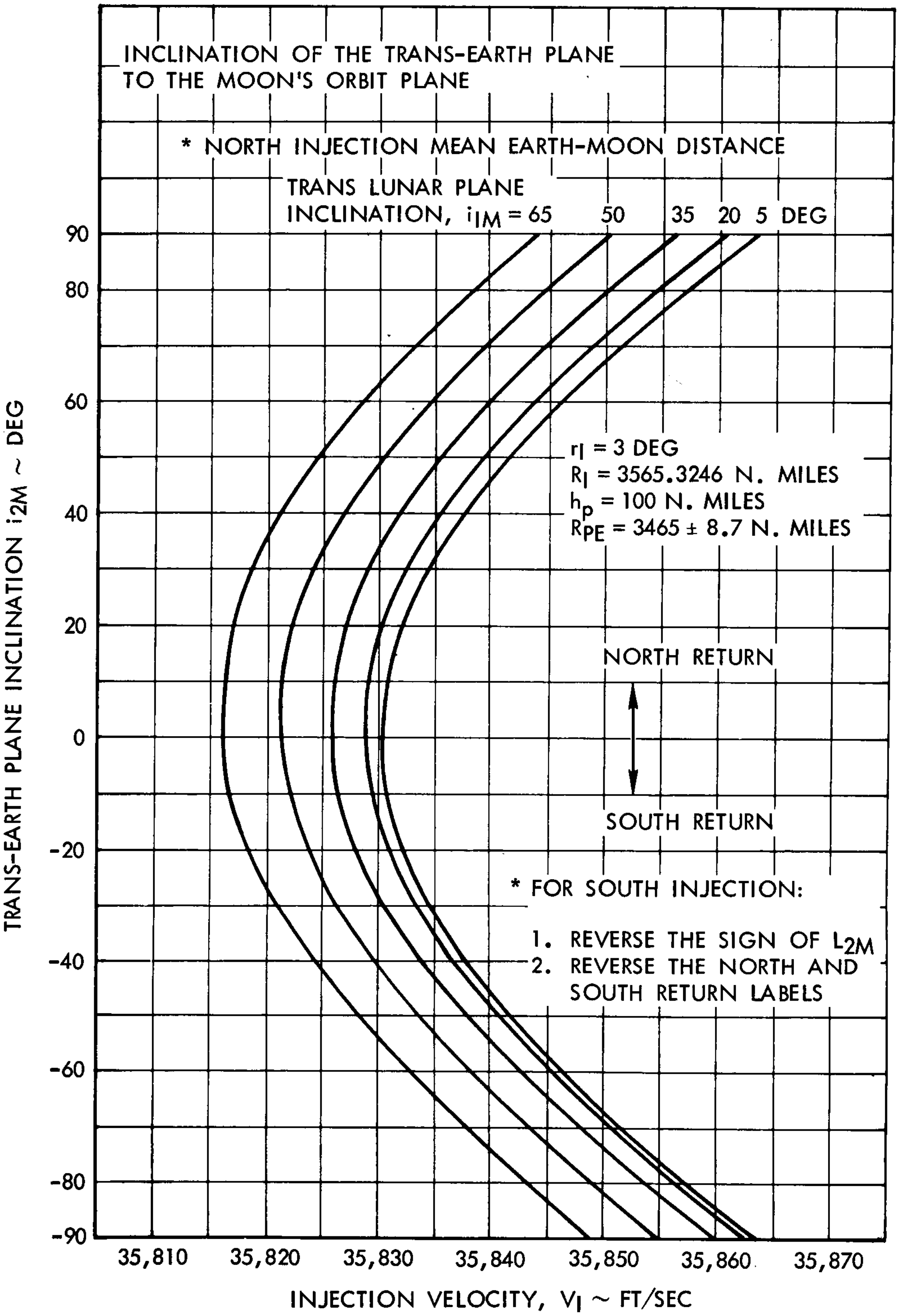

Figure 30. Inclination of the Trans-Earth Plane to the Moon's Orbit Plane

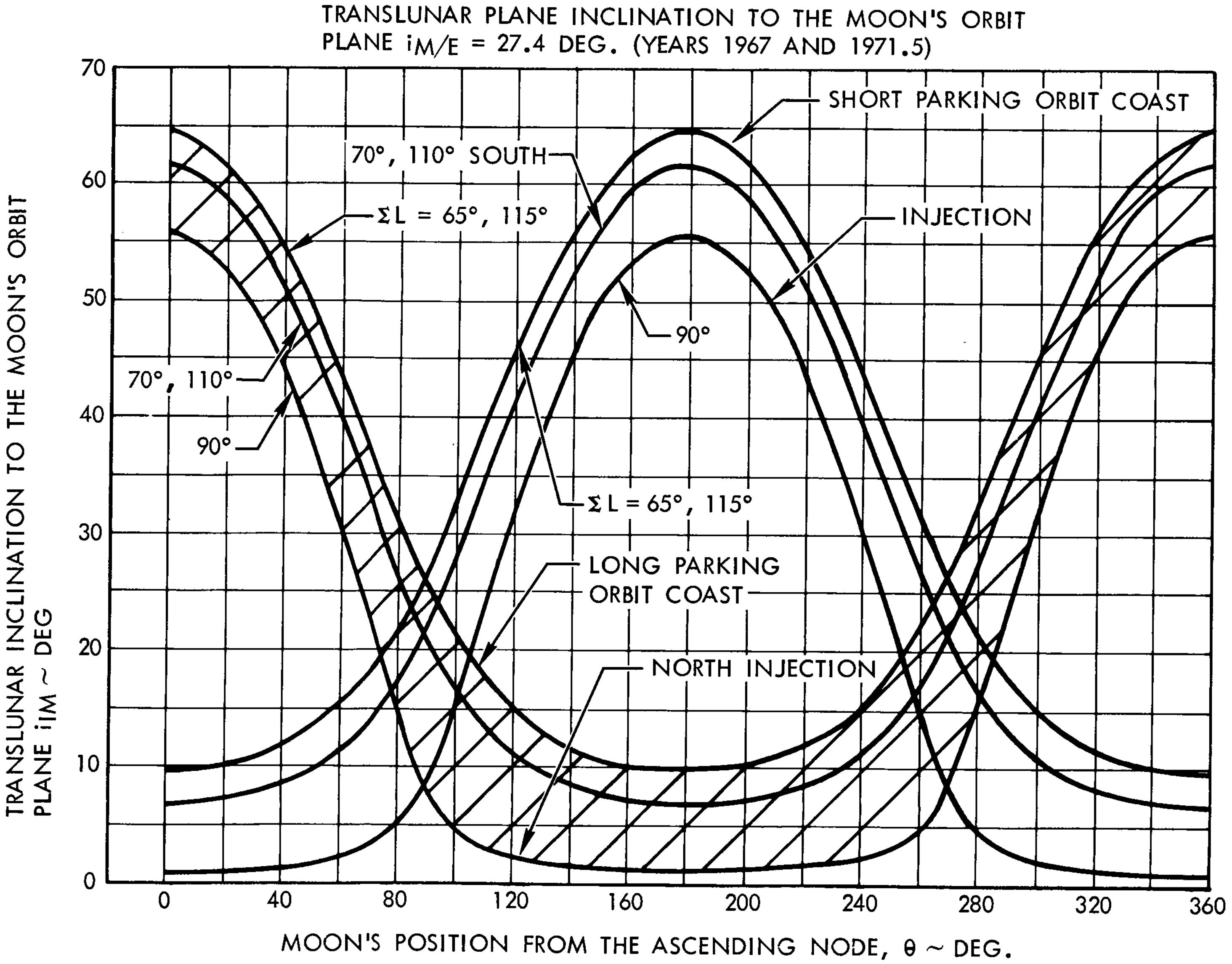

Figure 31. Translunar Plane Inclination to the Moon's Orbit Plane $i_{M/E}$ = 27.40 deg. (1967 and 1971.5)

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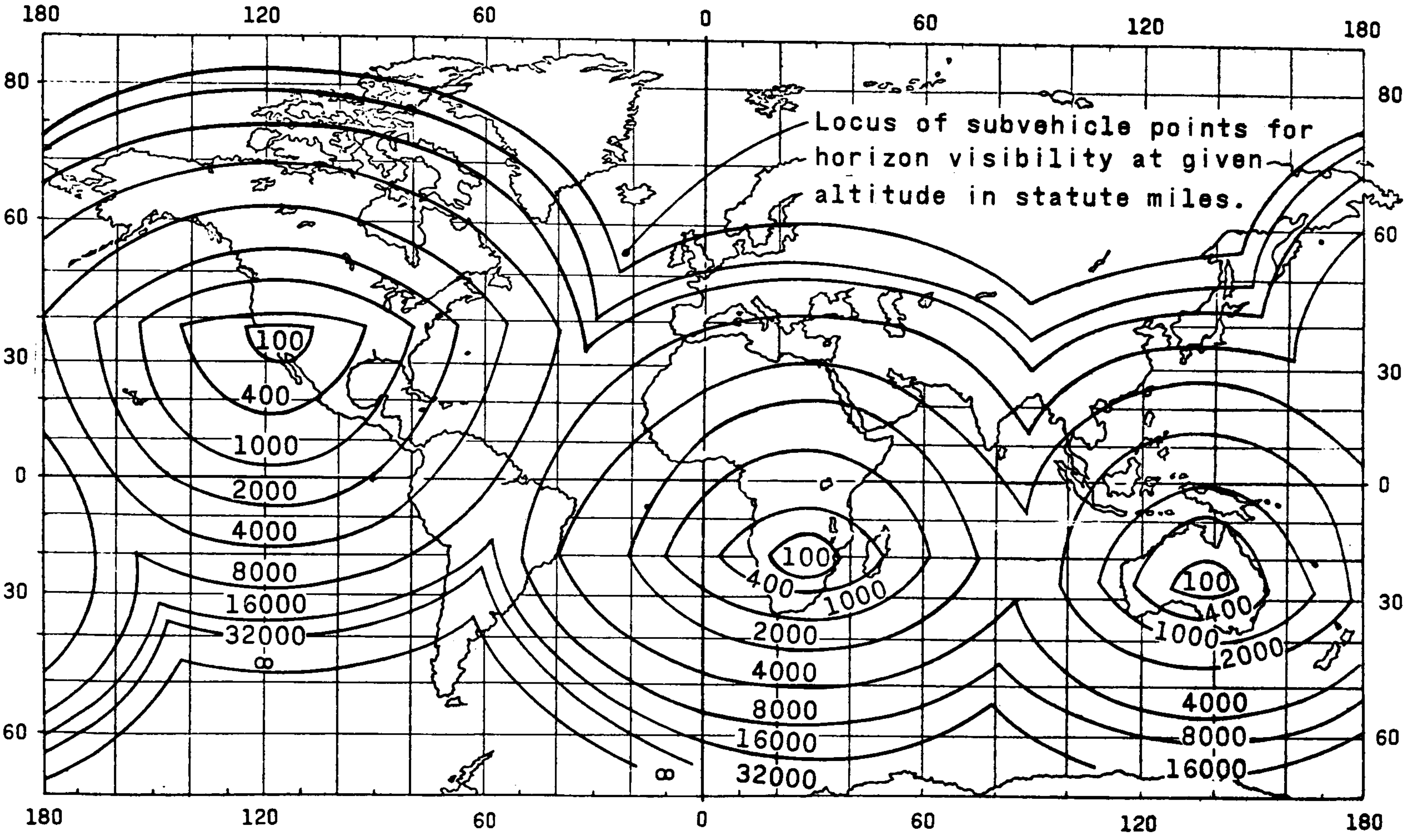

Locus of subvehicle points for horizon visibility at given altitude in statute miles.

Figure 32. Deep Space Station Coverage Plots for a 5° Terrain Horizon Mask

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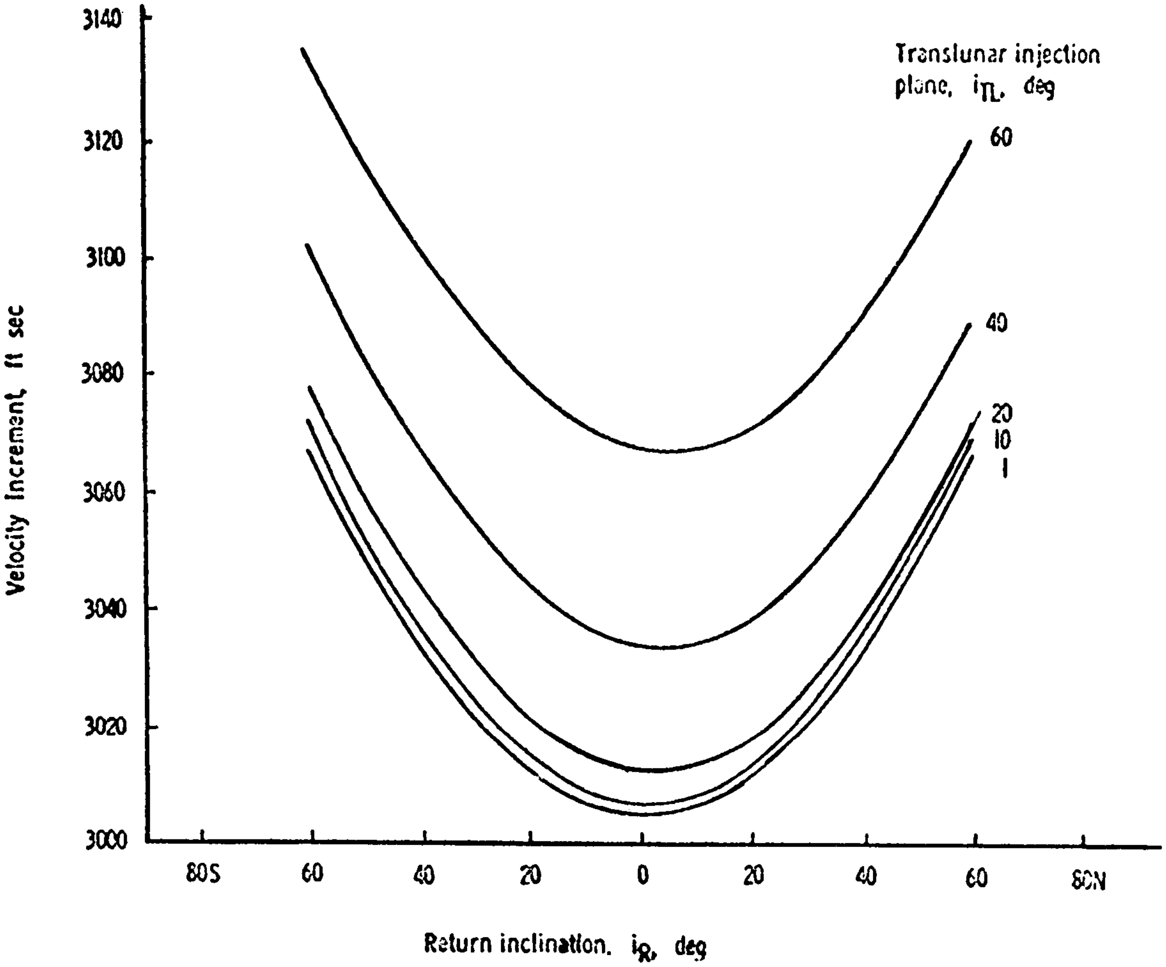
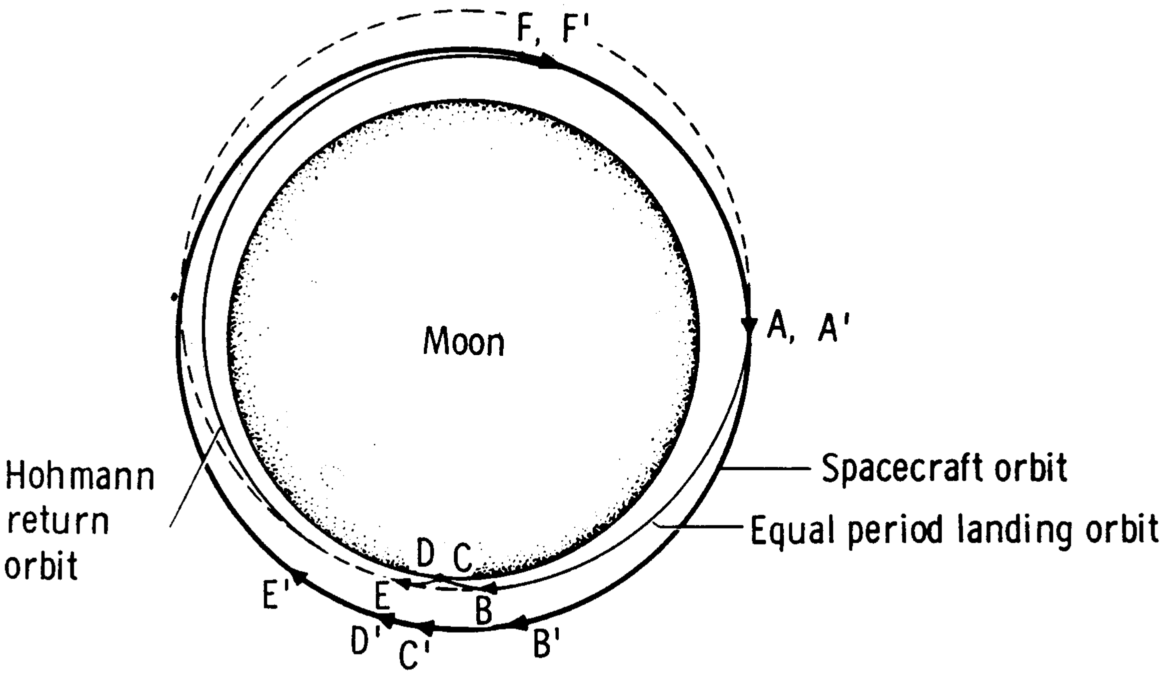


Figure 33. Velocity Increments for Establishing 80 N. M. Lunar Circular Orbits from Free Return Circumlunar Trajectories

[redacted]

CO[redacted]



A. Seperation of Lunar Module from Spacecraft
B. Initiation of landing maneuver
C. Start of hover
D. Abort from hover or takeoff from lunar surface
E. Insertion into return orbit
F. Rendezvous of Spacecraft and Lunar Excursion Module

Figure 34. Lunar Landing Technique Via Equal Period Transfer

[redacted]

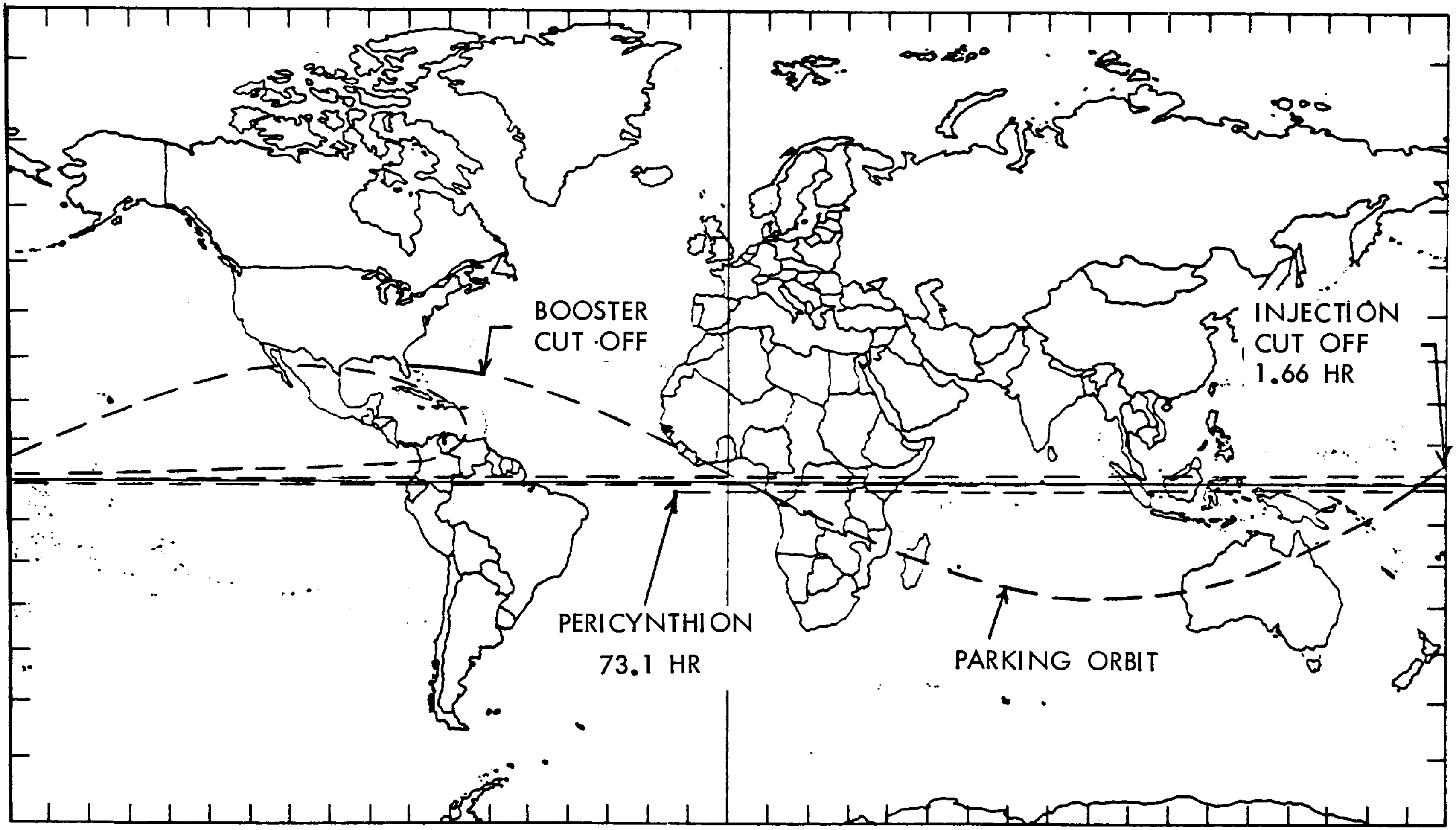


Figure 35 Earth Track for Lunar Flight Plan

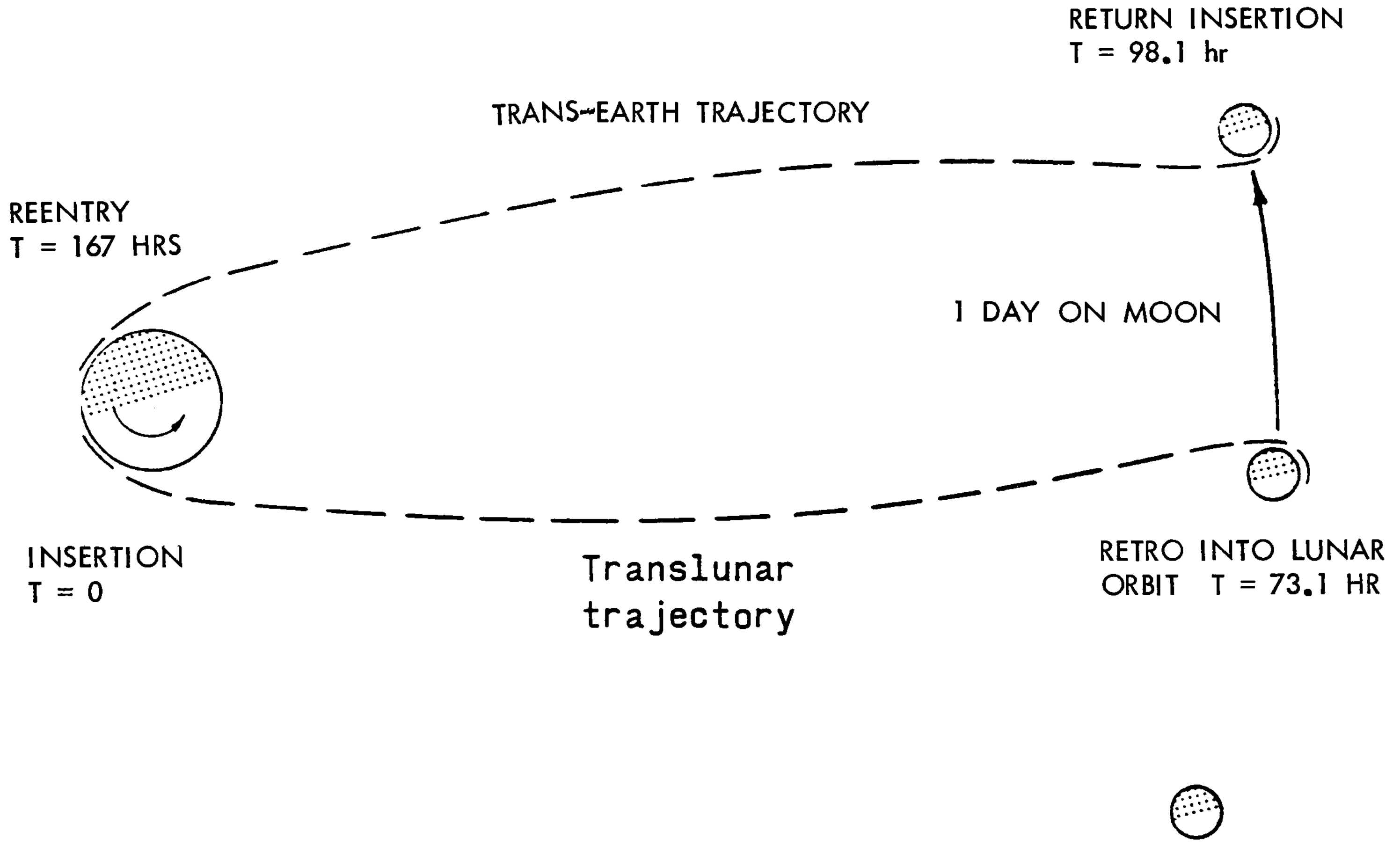


Figure 36. Translunar and Trans-Earth Trajectories Shown in the Inertial Earth-Moon System

[redacted]

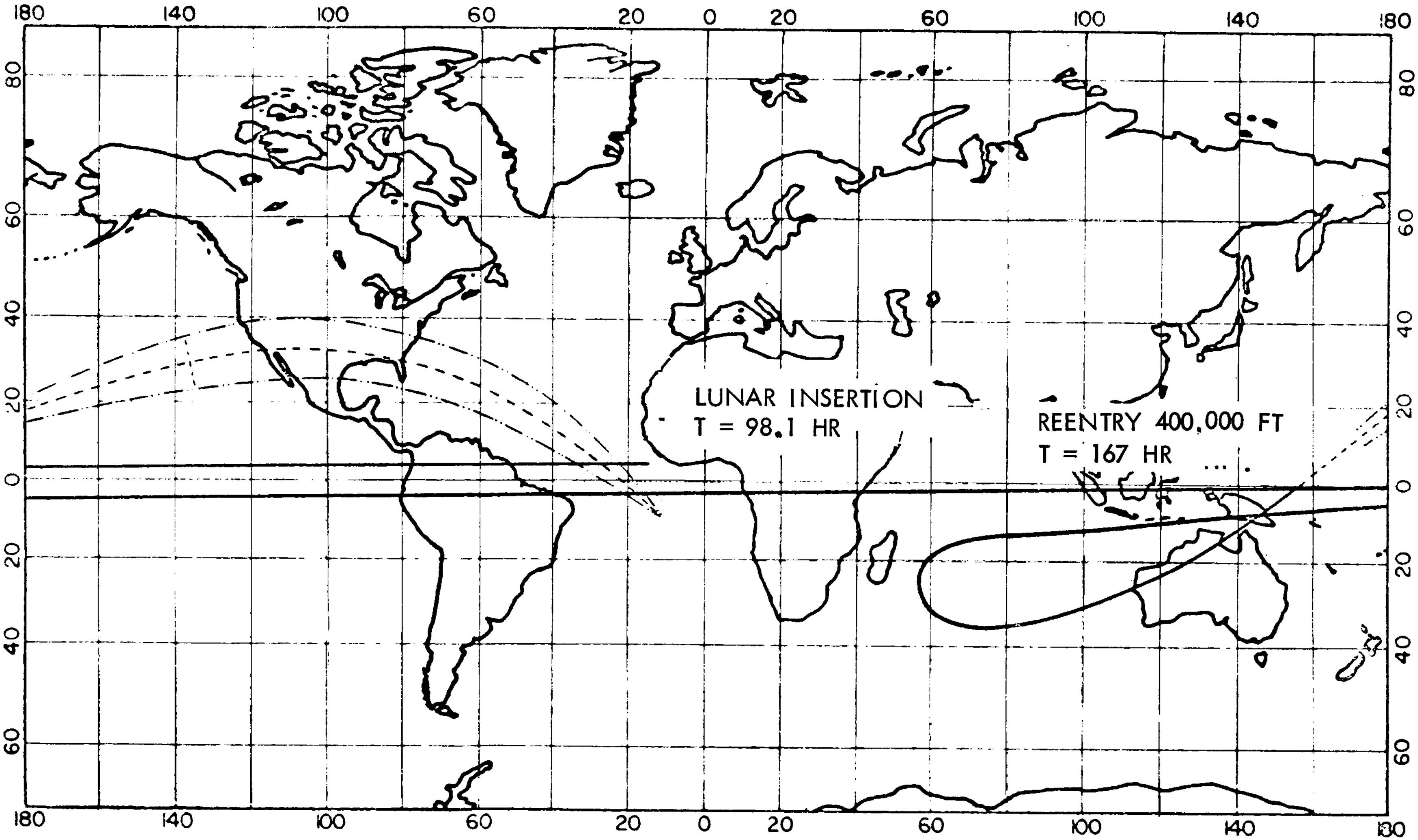


Figure 37. Earth Track of Trans-Earth and Reentry Phases

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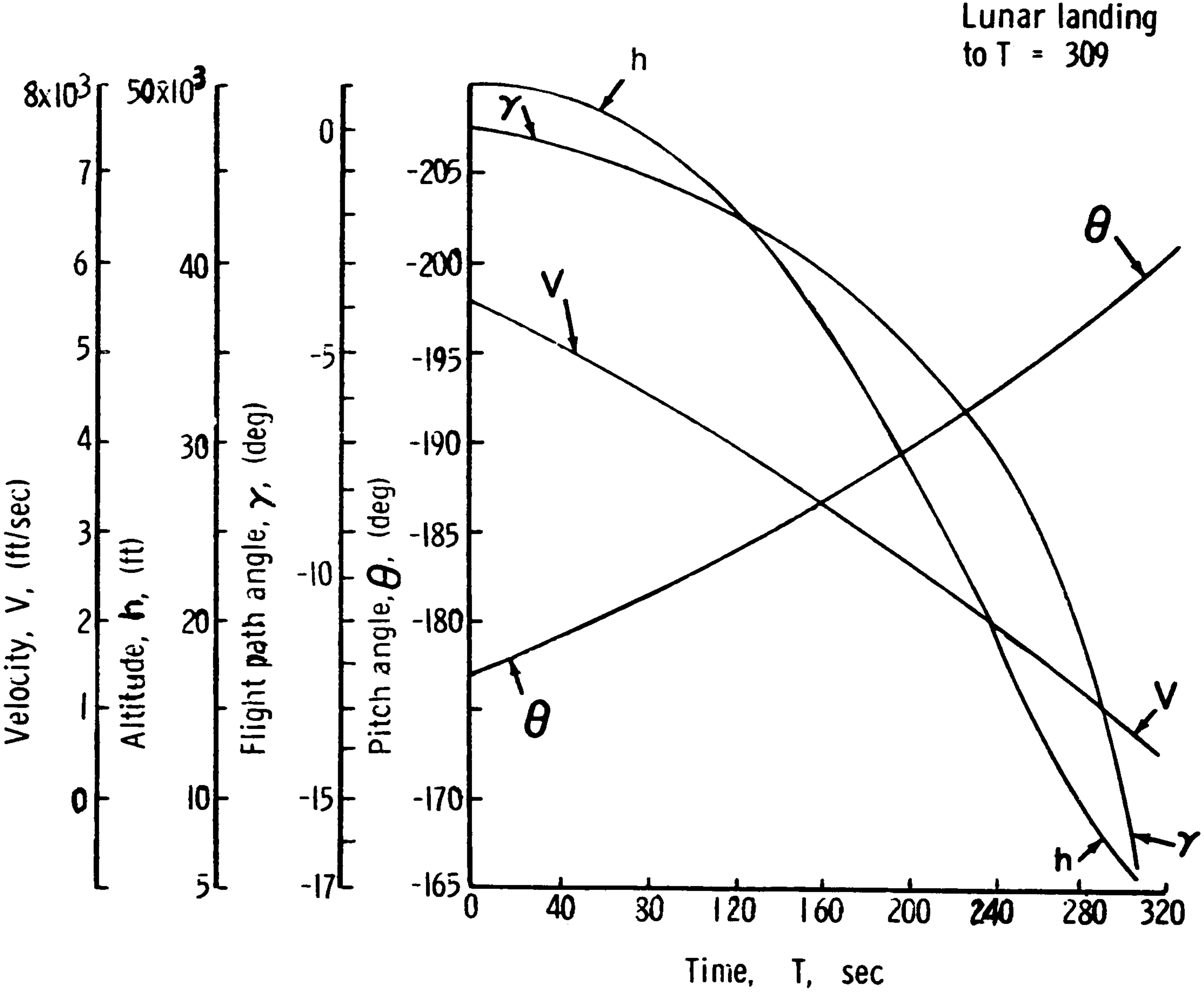

Figure 38. Lunar Landing Trajectory $I_{sp} = 305$, $T W_o = 0.4$

[redacted]

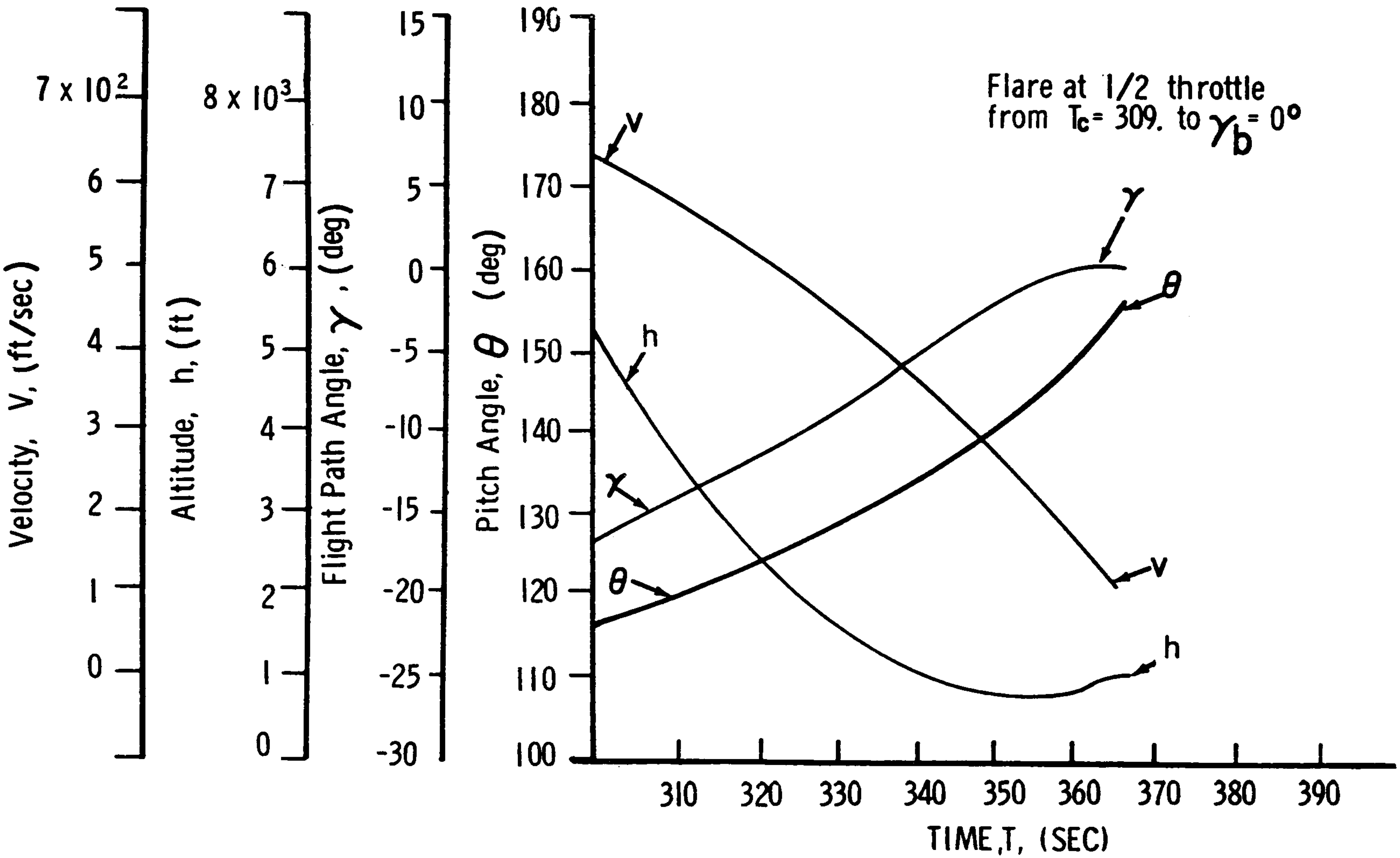

Figure 39. Lunar Landing Trajectory

[redacted]

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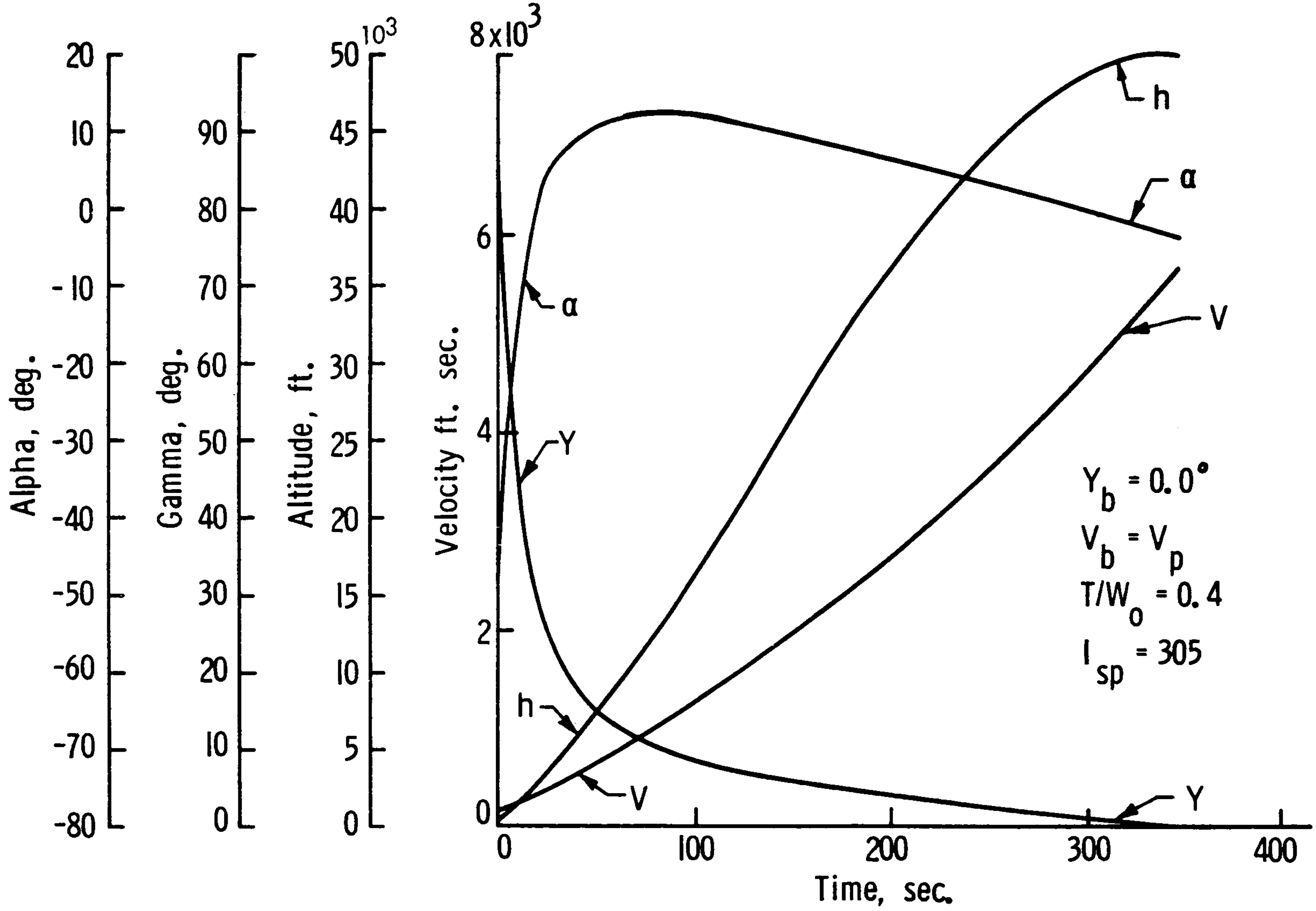

Figure 40. Time History for Optimum Lunar Launch to 50,000 Ft. Pericynthion, Apocynthion Altitude = 600,000 Ft.

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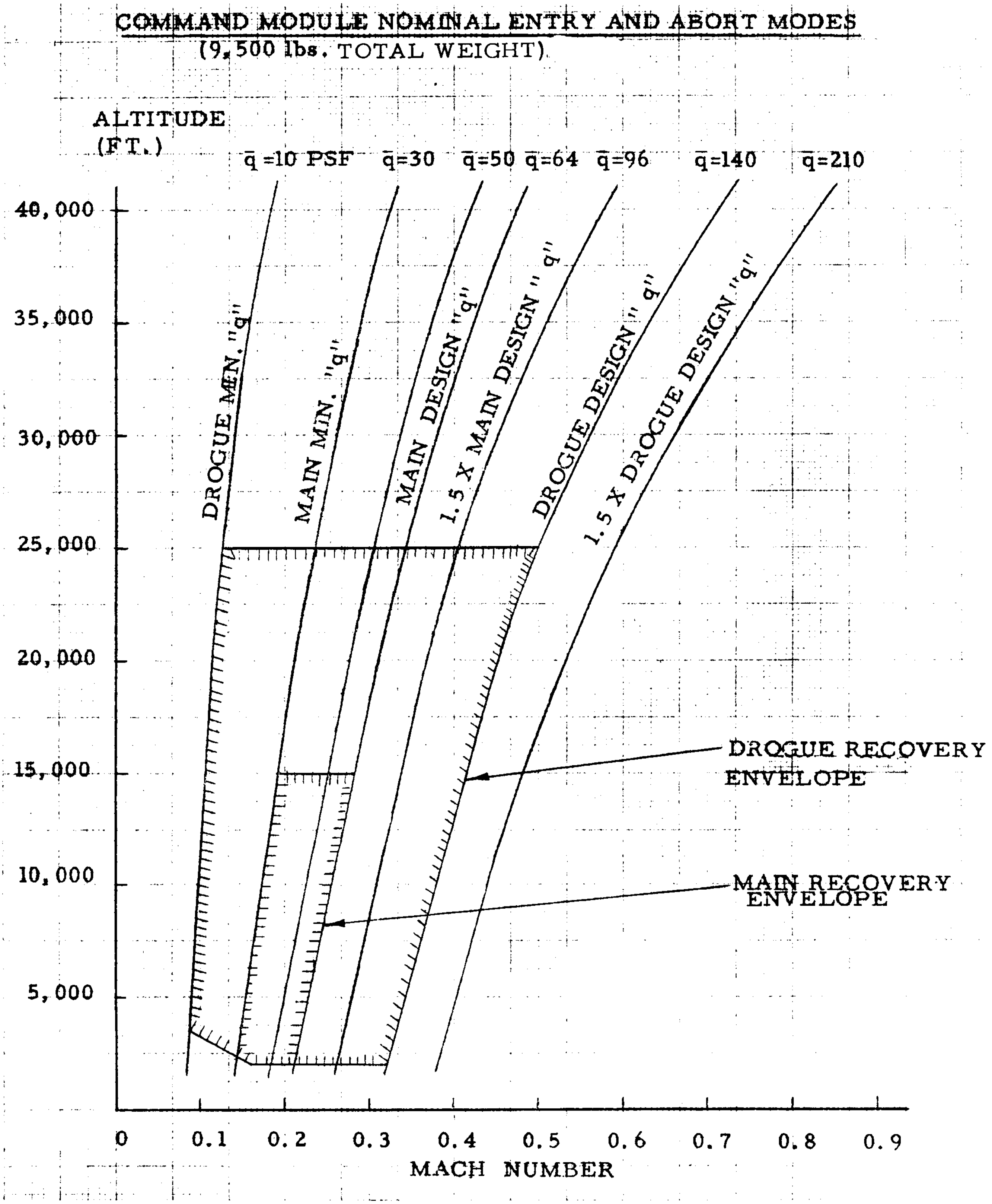

Figure 41. Parachute Recovery Envelope

[redacted]

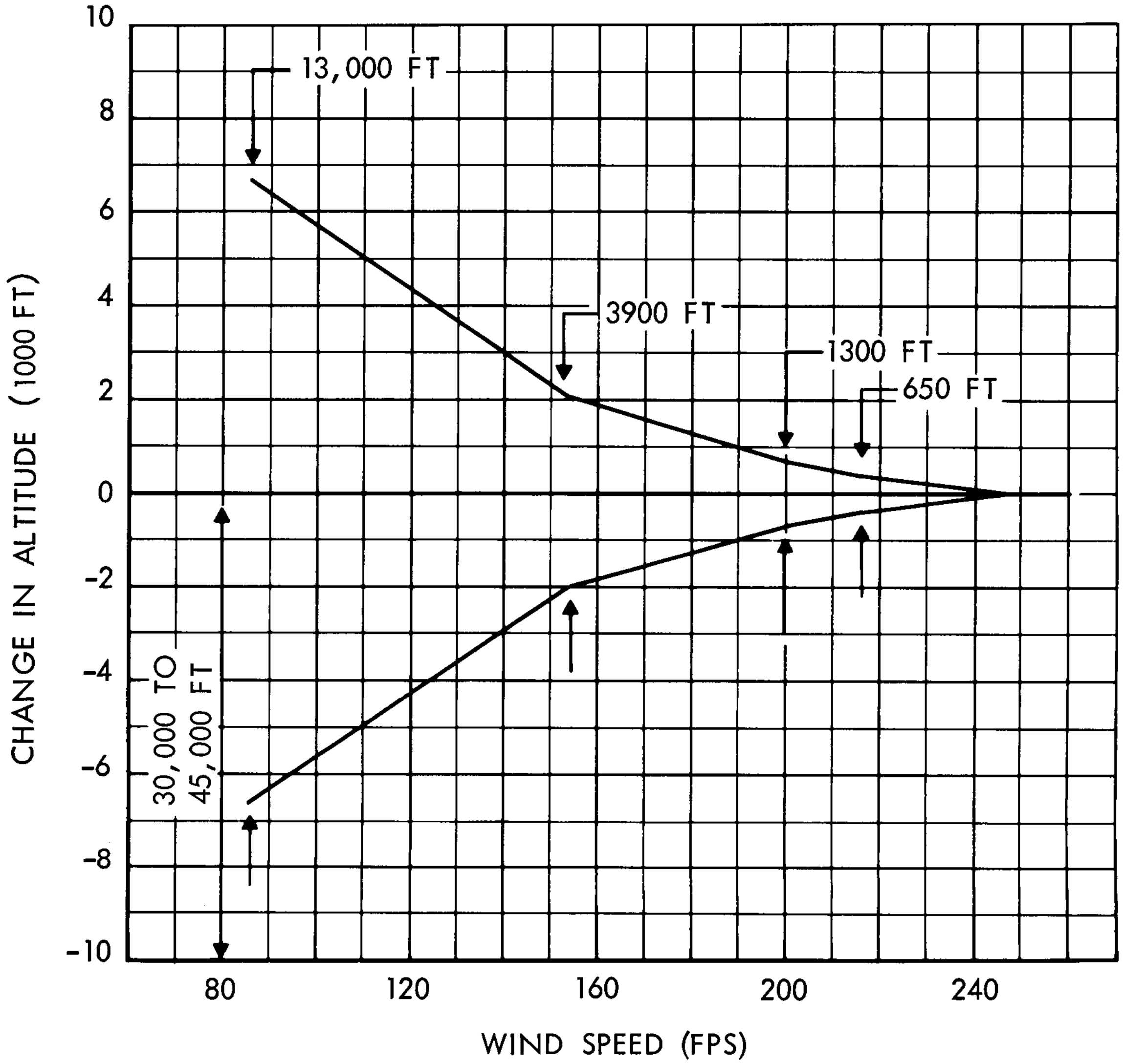

Figure 42. Wind Shear Profile

[redacted]

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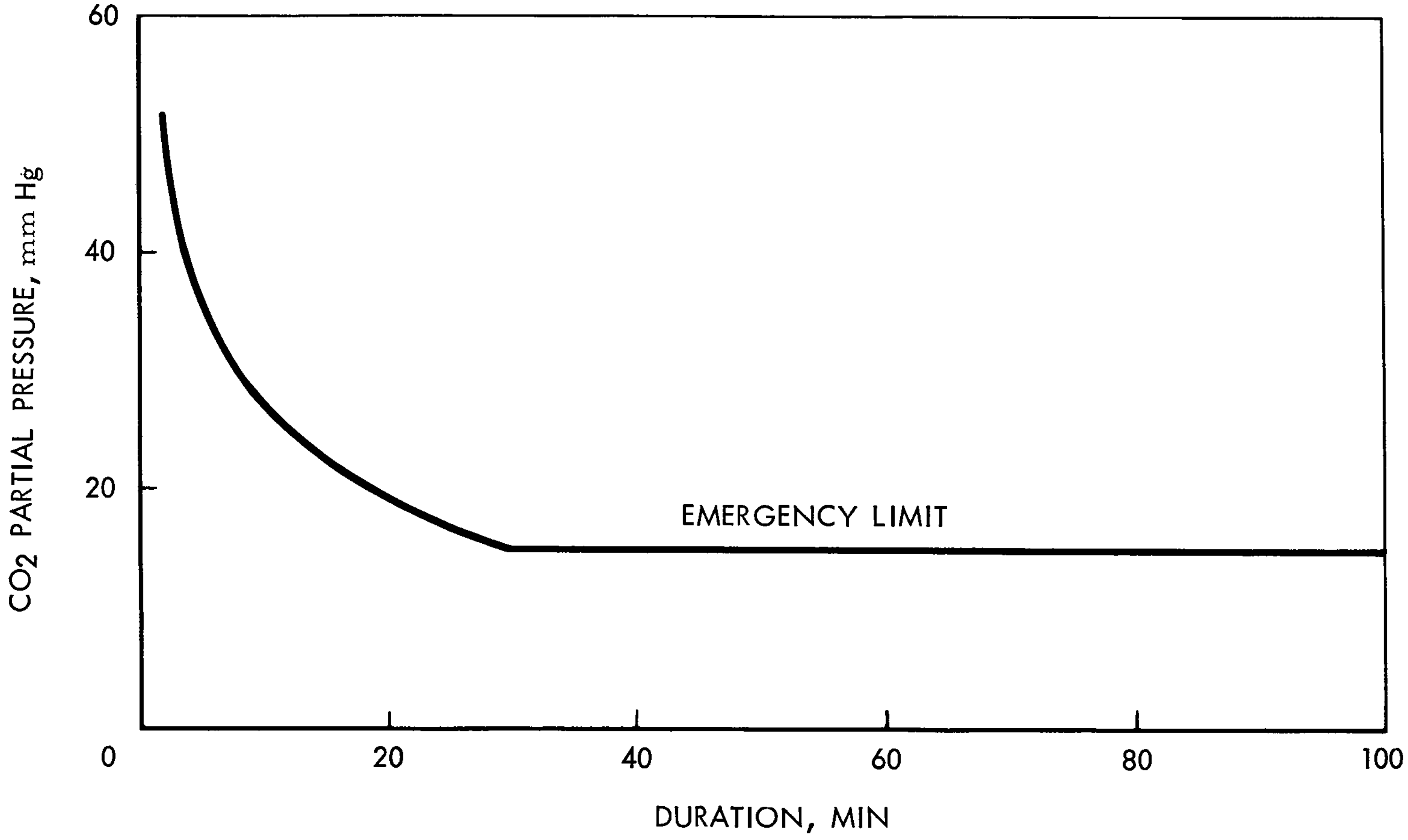

Figure 43. Emergency Carbon Dioxide Limit

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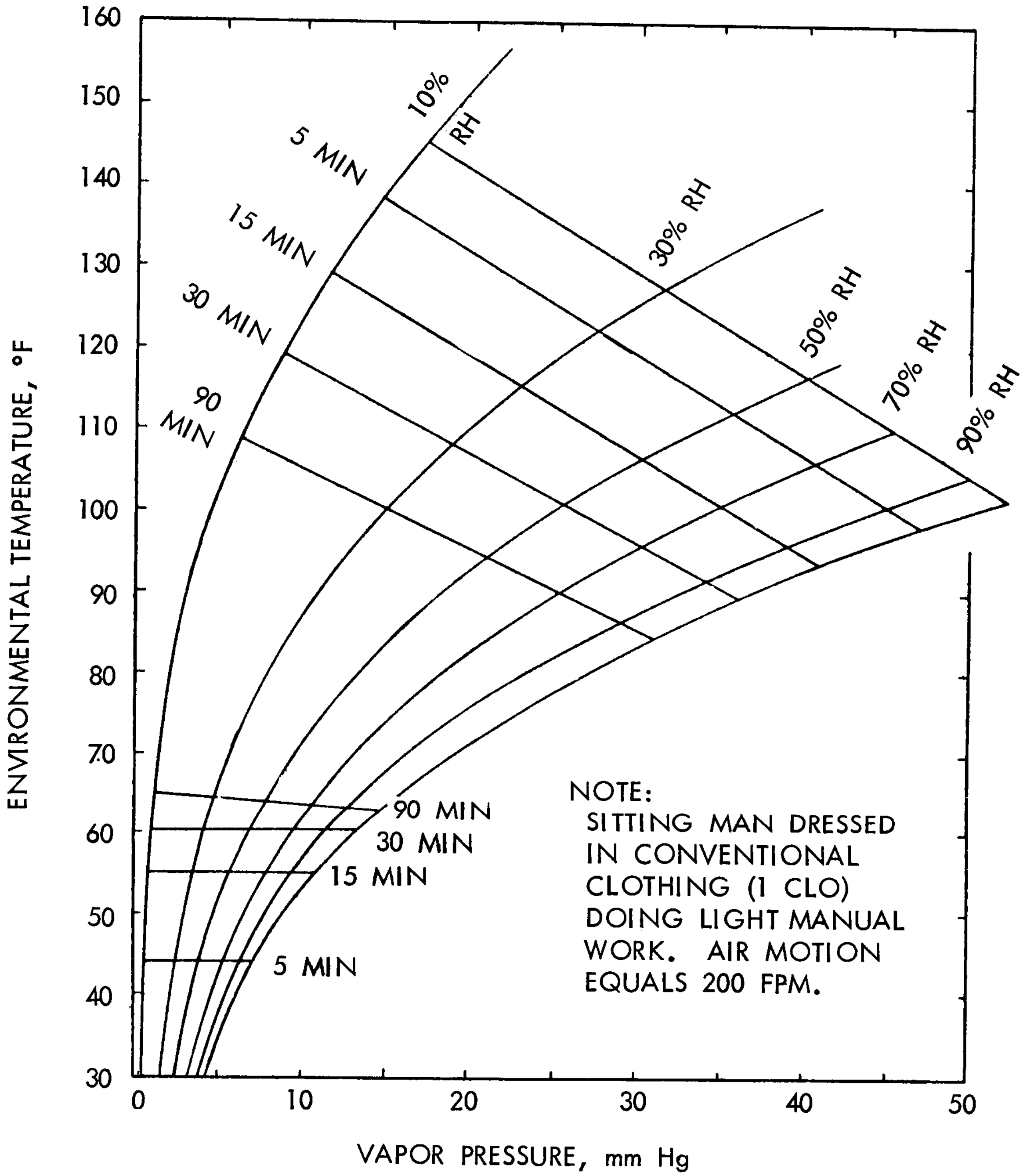
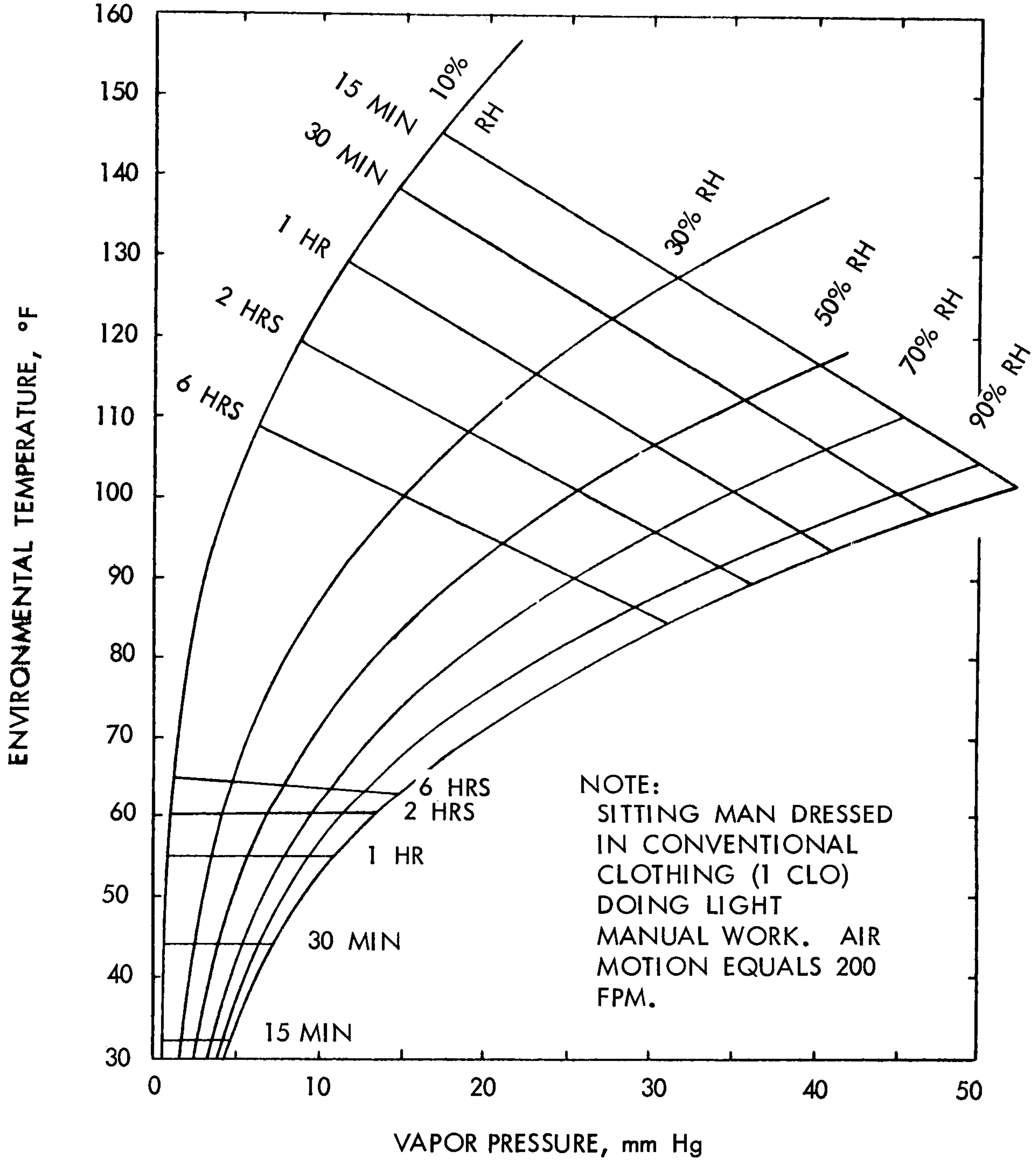


Figure 44. Temperature and Humidity Nominal Limit



NOTE:
SITTING MAN DRESSED IN CONVENTIONAL CLOTHING (1 CLO) DOING LIGHT MANUAL WORK. AIR MOTION EQUALS 200 FPM.

Figure 45. Temperature and Humidity Emergency Limit

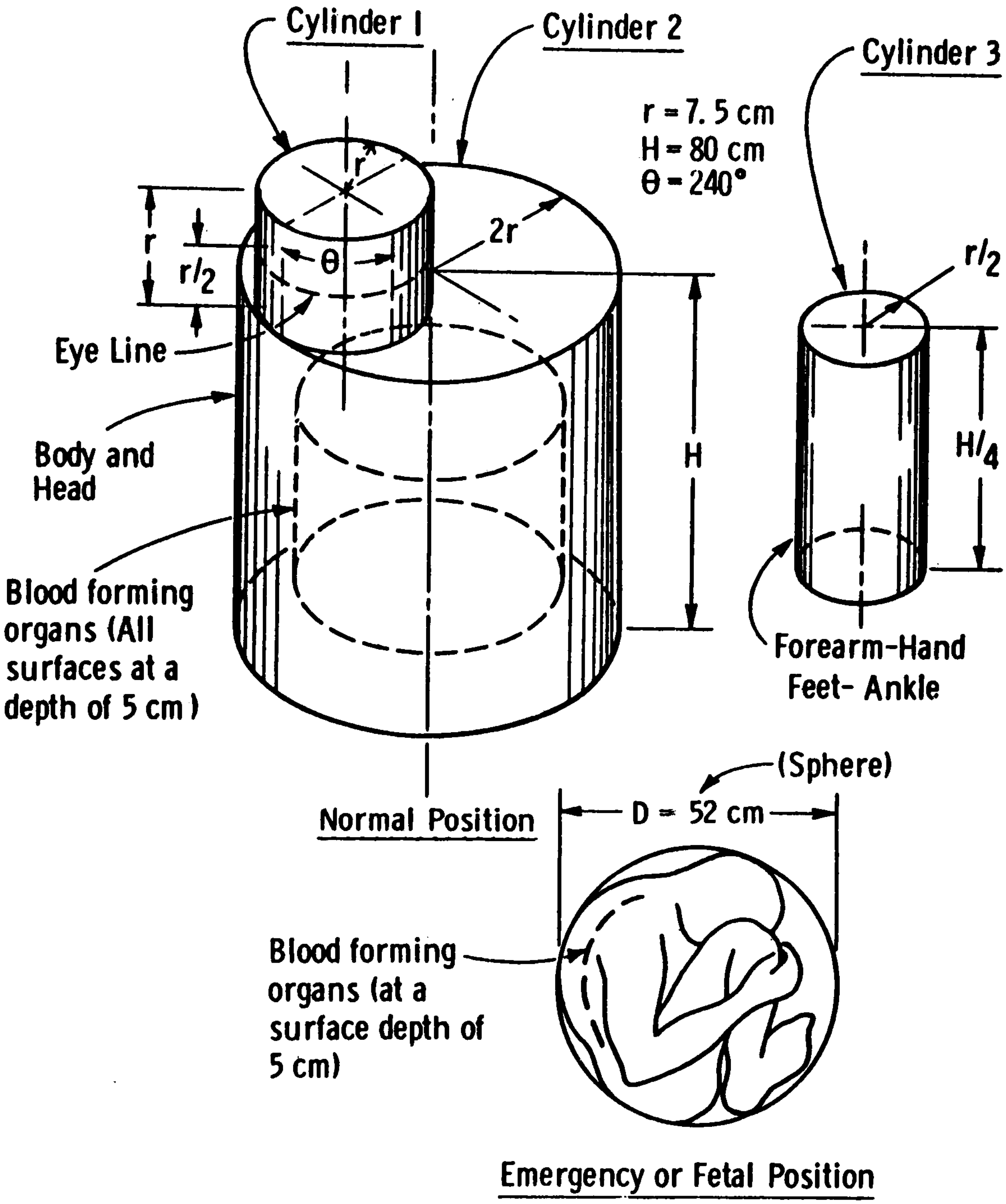


Figure 46. Models of the Radiation Standard Man

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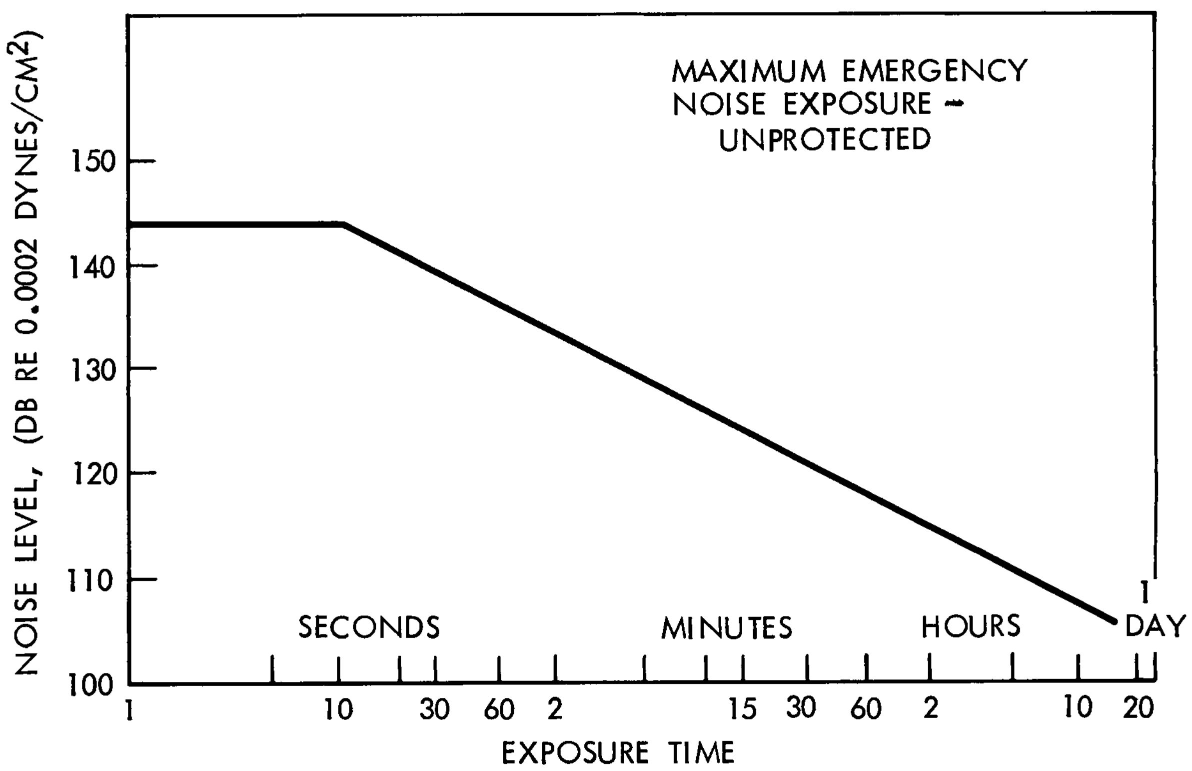

Figure 47. Noise Tolerance, Emergency Limit

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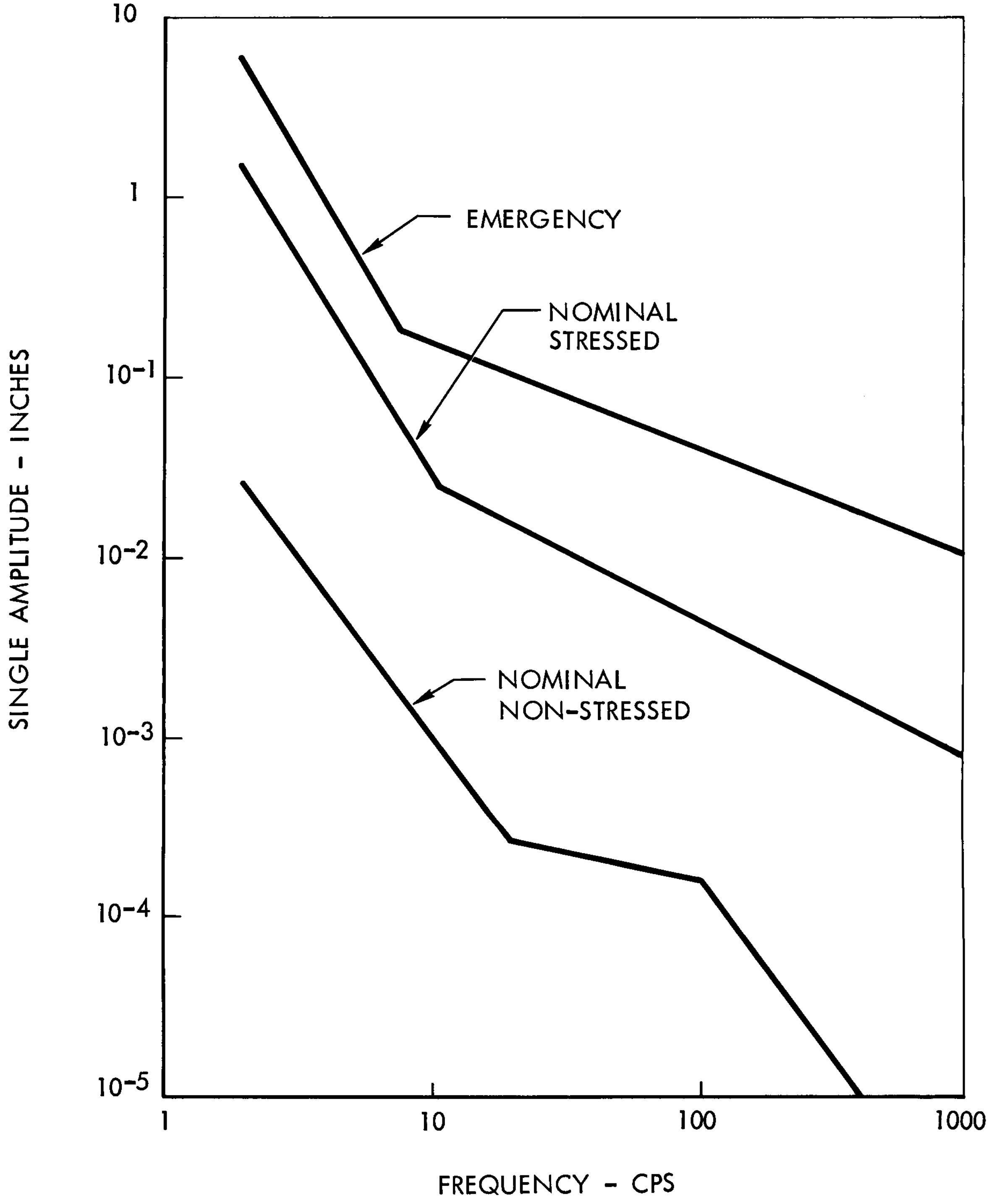

Figure 48. Vibration Limits

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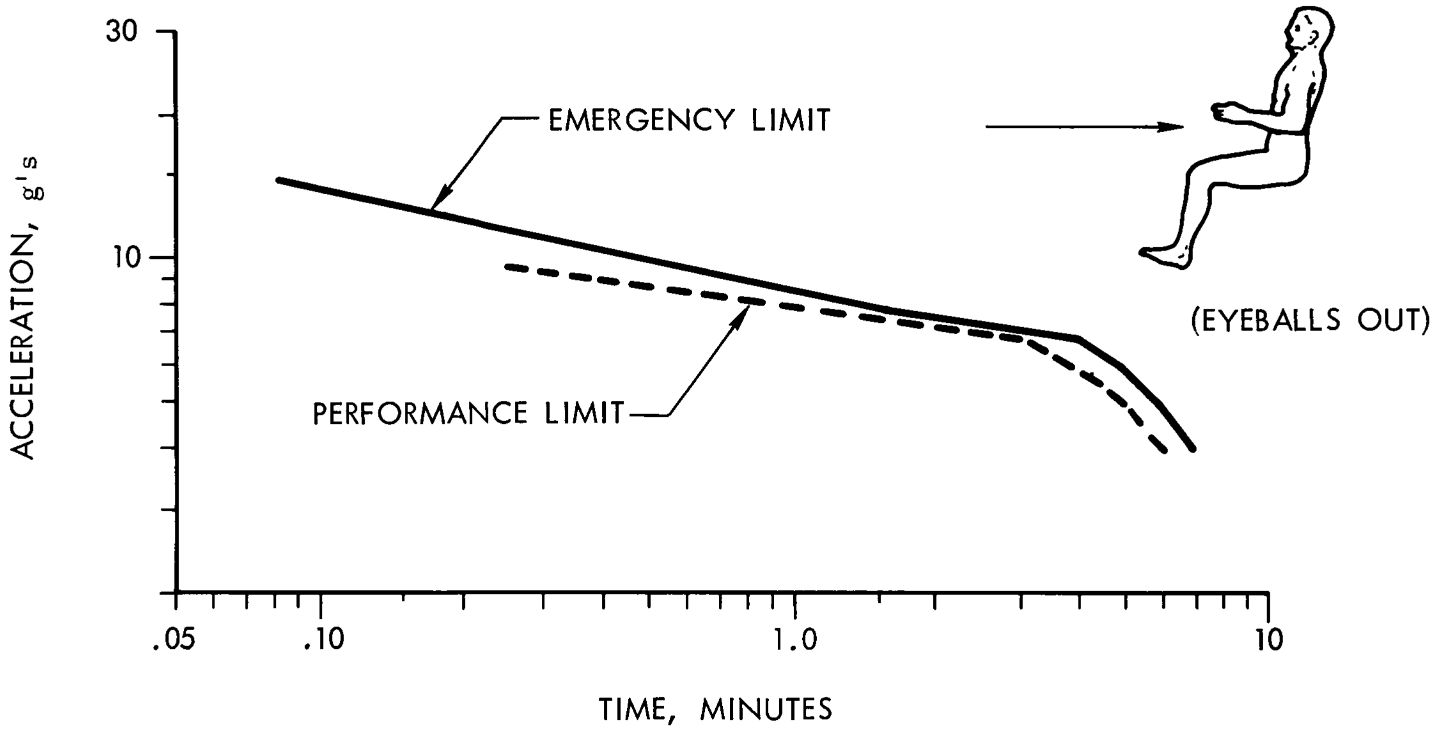

Figure 49. Sustained Acceleration

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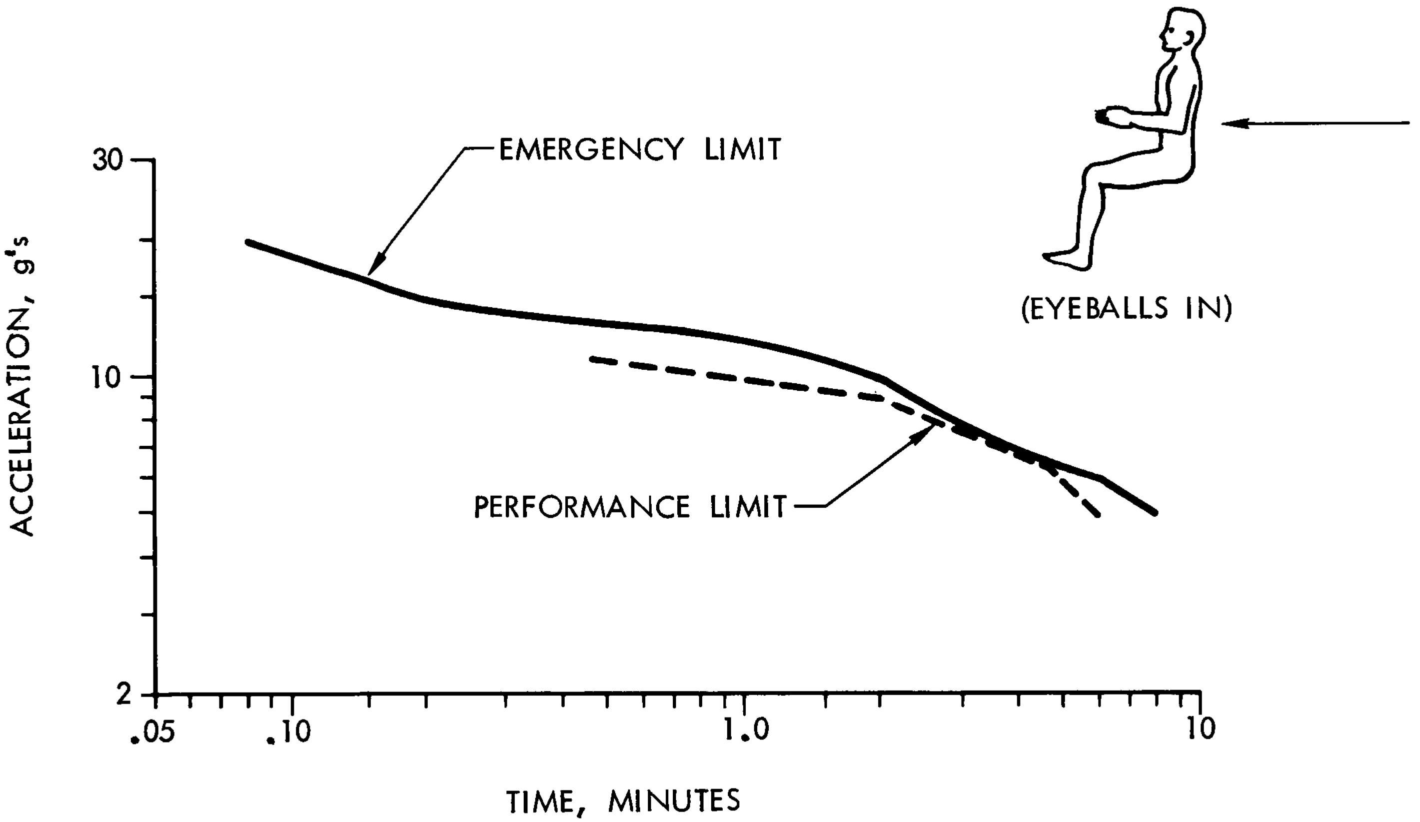

Figure 50. Sustained Acceleration

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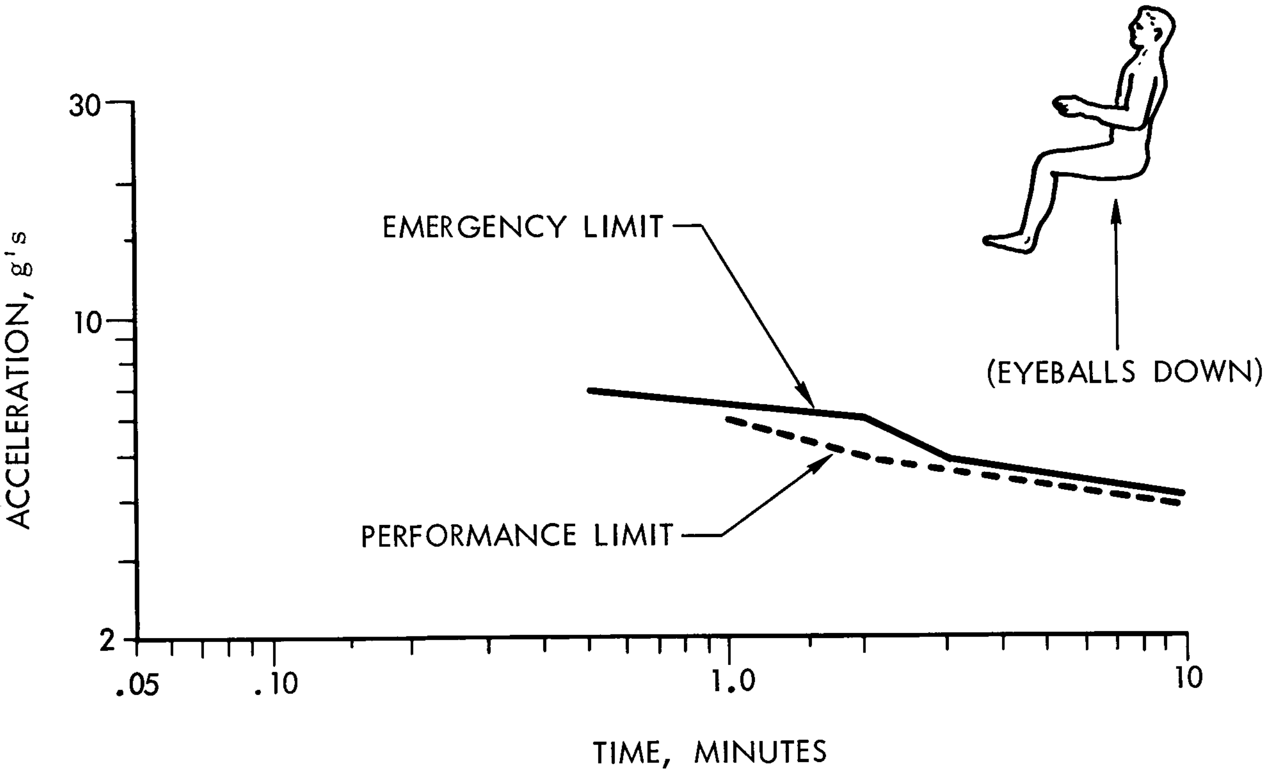

Figure 51. Sustained Acceleration

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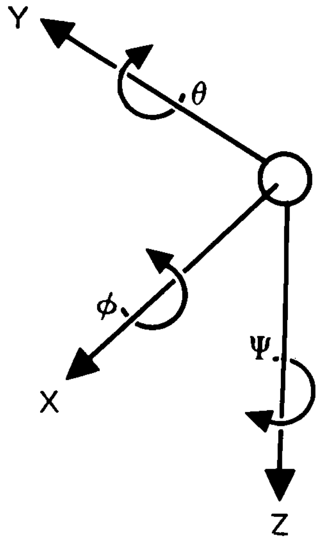
Table I. Reference Axes, Spacecraft

Positive direction of axes and angles (forces and moments) are shown by arrows. (When launch vehicle is at a launch angle of 90°, the positive "X" direction is vertically upwards.)



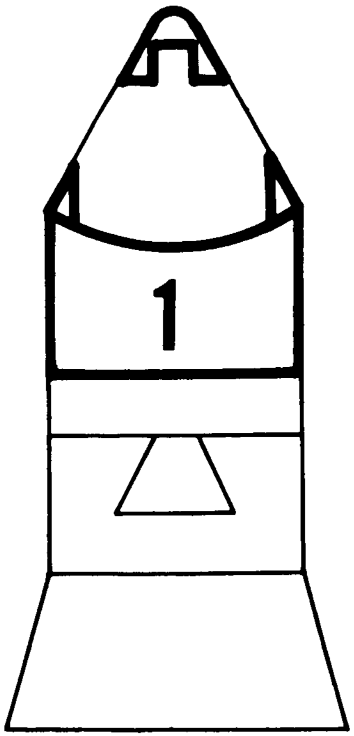
Axis		Moment About Axis		
Designation	Symbol	Designation	Symbol	Positive Direction
Longitudinal	X	Rolling	L	Y ⟶ Z
Lateral	Y	Pitching	M	Z ⟶ X
Normal	Z	Yawing	N	X ⟶ Y

Force	Angle		Velocities	
(Parallel to Axis Symbol)	Designation	Symbol	Linear (Components along Axis)	Angular
X	Roll	ϕ	U	p
Y	Pitch	θ	V	q
Z	Yaw	ψ	W	r

[redacted]

Table II. Acoustics

ZONE 1



OCTAVE BAND	LAUNCH & ENTRY	PAD ABORT & HI - Q ABORT	SPACE OPERATIONS
	SPL (DB - INTERNAL)	SPL (DB - INTERNAL)	
4.7 - 9.4 CPS	145	144	DNA
9.4 - 18.8 CPS	153	152	
18.8 - 37.5 CPS	150	151	
37.5 - 75 CPS	150	150	
75 - 150 CPS	147	150	
150 - 300 CPS	145	149	
300 - 600 CPS	140	144	
600 - 1200 CPS	132	145	
1200 - 2400 CPS	125	147	
2400 - 4800 CPS	120	144	
4800 - 9600 CPS	116	142	
OVERALL	158	158	

[redacted]

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ZONE 2

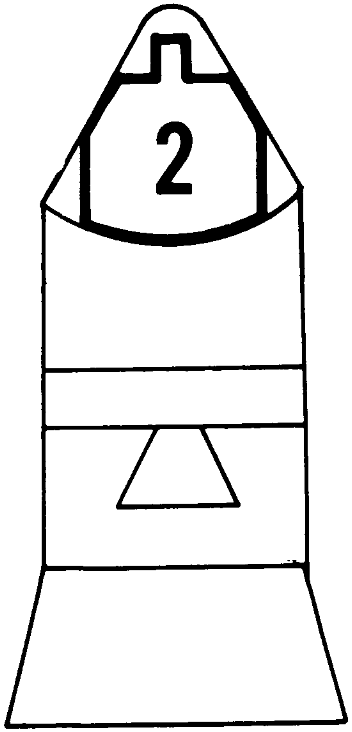

Table III. Acoustics

OCTAVE BAND	LAUNCH & ENTRY	PAD ABORT & HI - Q ABORT	SPACE OPERATIONS
	SPL (DB - INTERNAL)	SPL (DB - INTERNAL)	
4.7 - 9.4 CPS	131	131	DNA
9.4 - 18.8 CPS	132	132	
18.8 - 37.5 CPS	133	135	
37.5 - 75 CPS	133	134	
75 - 150 CPS	131	134	
150 - 300 CPS	127	131	
300 - 600 CPS	122	129	
600 - 1200 CPS	116	130	
1200 - 2400 CPS	109	129	
2400 - 4800 CPS	101	126	
4800 - 9600 CPS	95	122	
OVERALL	140	142	

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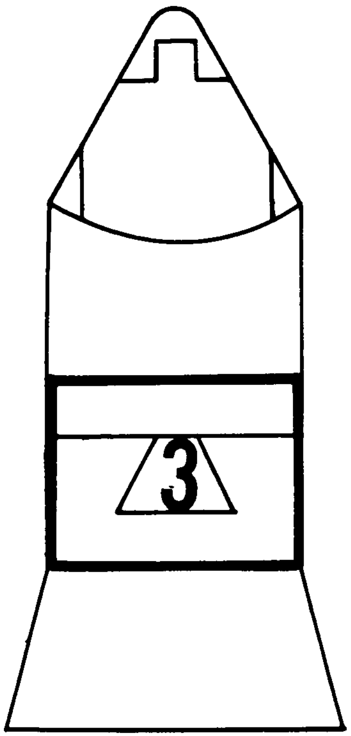
Table IV. Acoustics

ZONE 3



OCTAVE BAND	LAUNCH	SPACE OPERATIONS
	SPL (DB - INTERNAL)	
4.7 - 9.4 CPS	137	DNA
9.4 - 18.8 CPS	141	
18.8 - 37.5 CPS	142	
37.5 - 75 CPS	143	
75 - 150 CPS	140	
150 - 300 CPS	136	
300 - 600 CPS	132	
600 - 1200 CPS	122	
1200 - 2400 CPS	119	
2400 - 4800 CPS	113	
4800 - 9600 CPS	107	
OVERALL	148	

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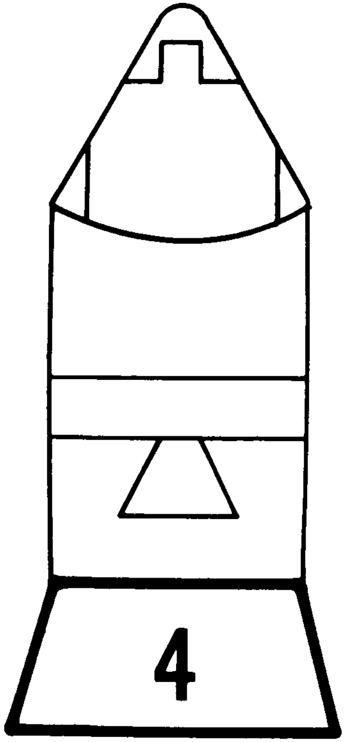
Table V. Acoustics

ZONE 4



OCTAVE BAND	LAUNCH & ENTRY	SPACE OPERATIONS
	SPL (DB - INTERNAL)	DNA
4.7 - 9.4 CPS	136	
9.4 - 18.8 CPS	142	
18.8 - 37.5 CPS	141	
37.5 - 75 CPS	141	
75 - 150 CPS	138	
150 - 300 CPS	134	
300 - 600 CPS	130	
600 - 1200 CPS	123	
1200 - 2400 CPS	116	
2400 - 4800 CPS	110	
4800 - 9600 CPS	104	
OVERALL	147	

[redacted]

Table VI. Vibration—Launch and Entry Environments

	Zone 1	Zone 2	Zone 3	Zone 4
Random (Linear relates to plot on log-log scale)	5 to 50 cps, Linear increase from 0.007 to 0.122 g^2/cps	5 to 50 cps, Linear increase from 0.0063 to 0.0095 g^2/cps	5 to 50 cps, Linear increase from 0.005 to 0.086 g^2/cps	5 to 50 cps, Linear increase from 0.004 to 0.025 g^2/cps
	50 to 150 cps, constant 0.122 g^2/cps	50 to 150 cps, constant at 0.095 g^2/cps	50 to 150 cps, constant at 0.086 g^2/cps	50 to 150 cps, constant at 0.025 g^2/cps
	150 to 2000 cps, Linear decrease from 0.122 to 0.007 g^2/cps	150 to 2000 cps, Linear decrease from 0.095 g^2/cps to 0.0035 g^2/cps	150 to 2000 cps, Linear decrease from 0.086 g^2/cps to 0.005 g^2/cps	150 to 2000 cps, Linear decrease from 0.025 g^2/cps to 0.005 g^2/cps
Sinusoidal (g peak) (Linear relates to plot on log-log scale)	5 to 100 cps, Linear increase from 0.3 g to 10 g	5 to 100 cps, Linear increase from 0.3 g to 8.5 g	5 to 100 cps, Linear increase from 0.3 g to 7 g	5 to 120 cps, Linear increase from 0.3 g to 5 g
	100 cps to 2000 cps, constant at 10 g	100 to 300 cps, constant at 8.5 g	100 to 2000 cps, constant at 7 g	120 to 2000 cps, constant at 5 g
		300 cps to 2000 cps, Linear decrease from 8.5 g to 5 g		

NOTE: Vibration levels shown are for basic structure and for small rigidly-mounted packages. For rigidly-mounted packages with weights in excess of 10 pounds, the vibration levels at

[redacted]

Table VI. **Vibration (continued)**

NOTE (Cont): frequencies above one-half the first local vibration mode will diminish in accord with the following: $A_2 = A_1 (1.5 - 1/2 \log 10\ W_2)$ where A_1 = noted vibration level (g) from Table W_2 = package weight (lb), and A_2 = vibration level of the package.

Zone 1- Forward and Aft C/M Equip. compartments and from Xa 910 to Xa 1018 on S/M

Zone 2- Crew Compartment Interior

Zone 3- S/M and Adapter from Xa 745 to Xa 910

Zone 4- Adapter- Aft of Xa 745

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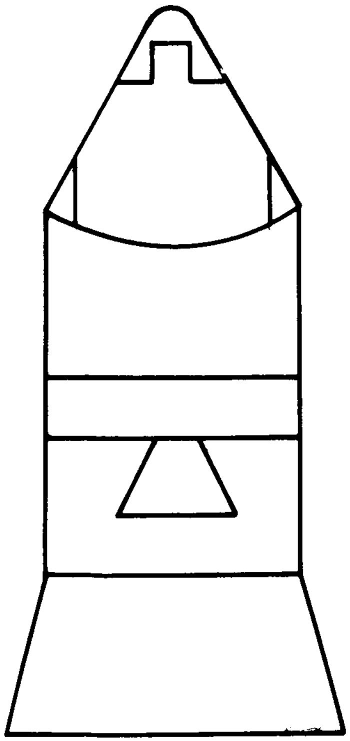
Table VII. Acoustics

EXTERNAL



OCTAVE BAND	LAUNCH & ENTRY	PAD ABORT & HI - Q ABORT
	SPL (DB - EXTERNAL)	SPL (DB - EXTERNAL)
4.7 - 9.4 CPS	155	154
9.4 - 18.8 CPS	163	162
18.8 - 37.5 CPS	164	164
37.5 - 75 CPS	161	161
75 - 150 CPS	158	161
150 - 300 CPS	155	159
300 - 600 CPS	150	157
600 - 1200 CPS	145	158
1200 - 2400 CPS	139	159
2400 - 4800 CPS	133	157
4800 - 9600 CPS	129	155
OVERALL	168.5	171

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Table VIII. Vibration—Hi "Q" Abort and Pad Abort

	Zone 1	Zone 2
Random (Linear relates to plot on log-log scale)	5 to 60 cps, Linear increase from 0.006 to 0.155 g^2/cps	5 to 60 cps, Linear increase from 0.006 g^2/cps to 0.13 g^2/cps
	60 to 500 cps, constant at 0.155 g^2/cps	60 to 200 cps, constant at 0.13 g^2/cps
	500 to 2000 cps, Linear decrease from 0.155 to 0.035 g^2/cps	200 to 2000 cps, Linear decrease from 0.13 g^2/cps to 0.006 g^2/cps
Sinusiodal (g peak) (Linear relates to plot on log-log scale)	5 to 15 cps, Linear increase 0.3 to 2 g	5 to 15 cps, Linear increase 0.3 to 2 g
	15 to 100 cps, Linear increase 2 g to 11 g	15 to 100 cps, Linear increase 2 g to 11 g
	100 to 1000 cps, Linear increase 11 g to 25 g	100 to 2000 cps, constant 11 g
	1000 to 2000 cps, constant 25 g	

NOTE: Vibration levels for pkg in excess of 10 pounds, see note on Launch and Entry Environments.

Table IX. Vibration - Space Operation (SPS operating)

	Zone 1	Zone 2	Zone 3
Random (Linear relates to log-log plot)	5 to 100 cps, Linear increase from 0.007 g^2/cps to 0.052 g^2/cps	5 to 100 cps, Linear increase from 0.007 g^2/cps to 0.04 g^2/cps	5 to 100 cps, Linear increase from 0.0086 g^2/cps to 0.07 g^2/cps
	100 to 550 cps, constant 0.052 g^2/cps	100 to 200 cps, constant 0.04 g^2/cps	100 to 550 cps, constant at 0.07 g^2/cps
	550 to 2000 cps, Linear decrease to 0.035 g^2/cps	200 to 2000 cps, Linear decrease to 0.015 g^2/cps	550 to 2000 cps, constant at 0.05 g^2/cps
Sinusoidal (g peak) (Linear relates to log-log plot)	5 to 300 cps, Linear increase from 0.16 g to 5.2 g 300 to 2,000 cps 5.2 g	5 to 300 cps, Linear increase from 0.16 g to 3.5 g 300 to 2,000 cps 3.5 g	5 to 400 cps, Linear increase from 0.23 g to 9 g 400 to 2,000 cps 9.0 g

NOTE: (1) For packages in excess of 10 pounds, see note on Launch and Entry Environments.

(2) When SPS is non-operating, the vibration levels will be insignificant.

(3) RCS engine operation will cause vibration in zone 2 at approximate levels shown, but will be insignificant in other zones.

[redacted]

Table X. Aerodynamic Heating

Trajectory	γe (deg)	Entry Velocity(fps)	Load(g's)	Flight Time from 400,000 ft to 100,000(sec)	Heat Flux* (B/ft^2 sec)	Integrated Heat Load* (B/ft^2)	Bond Line Temp(°F)
Structural load limit design	-10	36,200	20	292	1161	43,700	600
Maximum heat load design traj	-5.5	36,200	2.14	1706	465	138,100	600

* For maximum heating point on body.

[redacted]

Table XI. Radiation Exposure Limits

Critical organ	Maximum permissible integrated dose (rem)	RBE (rem rad)	Average yearly dose (rad)	Maximum permissible single acute emergency exposure (rad)	Location of dose point *
Skin of whole body	1,630	1.4	233	500[1]	0.07-mm depth from surface of cylinder 2 at highest dose rate point along eyeline
Blood-forming organs	271	1.0	54	200	5-cm depth from surface of cylinder 2
Feet, ankles, and hands	3,910	1.4	559	700[2]	0.07-mm depth from surface of cylinder 8 at highest dose point
Eyes	271	2[3]	27	100	3-mm depth from surface on cylinder 1 along eyeline

*See figure 6.

[1] Based on skin erythema level

[2] Based on skin erythema level but these appendages believed to be less radiosensitive

[3] Slightly higher RBE assumed since eyes are believed more radiosensitive