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1. SCOPE

This specification establishes the technical performance requirements for the development and design of the Apollo (Block I) Command and Service Module (CSM) System. The CSM System as discussed in this specification is comprised of a Launch Escape Subsystem (LES), a Command Module (CM), a Service Module (SM), a Spacecraft S-IVB Adapter, the associated Ground Support Equipment (GSE), and requisite Trainers. A general configuration of the Block I CSM and Launch Vehicles (LV) are delineated in Figures 1 and 2.

In addition, performance characteristics of the LV and the other items of Government Furnished Equipment (GFE) with which the design of the CSM System is compatible are specified.

The basis for design of the CSM System shall be a Lunar Orbit mission which may be defined as the Lunar Orbit Rendezvous (LOR) mission without the Lunar Excursion Module (LEM) interface. That is, the translunar injected gross weight of CSM System shall include the propellent allotment required for maneuvering if the LEM were attached but the LEM subsystems, their relationship with the CSM subsystems, and the functional flow of data between the LEM and CSM are not included.

1.1 Objectives. -

The objective of the CSM is to transport three men to a lunar orbit and to return them safely to earth.

2. APPLICABLE DOCUMENTS

The following documents form a part of this specification to the extent specified herein:

2.1 Specifications. -

2.1.1 NASA. -

MSFC-PROC-158A 12 April 1964	Soldering electrical connectors (high reliability) procedure for (amendments MSC-ASPO 513 of 10 February 1964)

[redacted]

MSFC 10M 01071 6 March 1961	Environment-protection When Using Electrical Equipment-Within the Areas of Saturn Complexes Where Hazardous Area Exist; procedure for
MSC-EMI-10A 17 October 1963	Addendum to MIL-I-26600, Interference Control Requirements, Aero. Equip.
MSC-GSE-1B 12 February 1964	Apollo Ground Support Equipment General Environmental Criteria and Test Specification
MSC-ASPO-I-4	Apollo Spacecraft Identification and Traceability Specification.

2.1.2 Military. -

MIL-I-26600 2 June 1958	Interference Control Requirements, Aeronautical Equipment
MIL-E-6051C 17 June 1960	Electrical Electronic System Compatibility and Interference Control Requirements for Aeronautical Weapon Systems, Associated Subsystems and Aircraft.

2.1.3 Contractor. -

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

SID 62-1000 17 September 1964	Preliminary Guidance and Navigation System Performance and Interface (P&I) Requirements Specifications
SID 62-1003 17 September 1964	Preliminary NASA Furnished Crew Equipment Interface and Performance Specifications.
SID 62-1001 17 September 1964	Flight Research and Development (R&D) Instrumentation Specification.
SID 63-881 17 September 1964	NASA Manned Space Flight Net (MSFN) Performance and Interface (P&I) Specification - Primary

SID 64-1237 4 September 1964	Vehicle Model Specification, Basic - Block I

2.2 Standards. -

2.2.1 Federal. -

Standard Atmosphere 1962

NASA TN D-595 March 1961	Reference Atmosphere For Patrick AFB
Federal Standard 209 16 December 1963	Clean Room and Work Station Requirements, Controlled Environments

2.2.2 Military. -

MIL-STD-130B 24 September 1962	Identification Marking of U.S. Military Property
MIL-STD-704 6 October 1959	Electrical Power, Aircraft; Characteristics and utilization of
MS 33586A 16 December 1958	Metals; definitions of dissimilar
MIL-STD-803 27 January 1964	Human Engineering Design Criteria for Aerospace Systems and Equipment

2.2.3 Contractor. -

Not applicable

2.3 Drawings. -

ICD 13M20108 20 July 1964	"Instrument Unit to Spacecraft Physical Requirements" (Saturn 1B)
ICD 13M50103 28 May 1964	"Instrument Unit to Spacecraft Physical Requirements" (Saturn V)
ICD 13M20109 24 July 1964	"Spacecraft/"Q"-Ball Physical Requirements" (Saturn 1B)
ICD 13M50123 29 July 1964	Envelope LEM/S-IVB/IU Clearance, Physical

ICD 13M50112 22 July 1964	"Spacecraft/"Q"-Ball Physical Requirements" (Saturn V)

2.4 Bulletins. -

None applicable

2.5 Other Publications. -

2.5.1 NASA. -

NPC 200-2 20 April 1962	Quality-Program Provisions for Space System Contracts

2.5.2 Military. -

AFMTC Pamphlet 80-2, Vol. I 1 October 1963	General Range Safety Plan, Prelaunch Safety Procedure
AFCRL 62-899 July 1962	Two Point Variability of Winds, Vols. I, II and III.
WADC TR 52-321 September 1954	Anthropometry of Flying Personnel - 1950

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

(a) The Definitive Contract, NAS9-150, dated 14 August 1963, Supplemental Agreements and Contract Change Authorizations, (SA/CCA's) thereto of date prior to 1 August 1964.

(b) This specification

(c) SID 64-1237, Vehicle Model Specification - Basic, Block I

(d) Other documents referenced herein

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3. REQUIREMENTS

This section contains the performance and design requirements for the CSM System and will encompass the following:

a. Definition of major elements of the system.

b. Design constraints and standards necessary to assure compatibility of program hardware.

c. The allocation of performance budgets and specified design constraints.

d. Identification of the principal functional interfaces.

e. Identification and use of the Government Furnished Equipment (GFE) forming an integral part of the system.

3.1 General Requirements. - These general requirements are the collection of principles to which the basic technical approach of the CSM subsystems must be responsive. They are the first order criteria from which successive design criteria, performance margins, tolerances, and environments shall be developed.

3.1.1 Launch Vehicle (LV) Performance Requirements. - Propulsion increments involved with the boost phases of the mission will be supplied by NASA furnished Saturn 1B or Saturn V launch vehicles. The CSM system shall be designed compatible with the following interface requirements.

3.1.1.1 Launch Vehicle Attitude Control. - The limit cycle or dead band for the attitude control subsystem of the LV shall be ± 1.0 degree in pitch, roll and yaw at a rate not to exceed 0.05 degrees/second.

3.1.1.2 Propellant Venting. - The SIVB propellant venting shall be continuous and the thrust generated shall not cause any moment that cannot be corrected within the attitude control subsystem dead band.

3.1.1.3 Loads Criteria for CSM SIVB Adapter and Instrumentation Unit IV). - The following maximum flight parameters shall not be exceeded on Block I missions.

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3.1.1.3.1 Load Parameters

α = 6.9 degrees

$q\alpha$ = 5072 Lb/Ft^2 deg. (for flexible body conditions)

Trajectories shall be shaped such that these parameters are not exceeded. (see Figure 46)

The limit axial loads, shears, and bending moments at the CSM/LV interface shall not exceed those shown in the following table for conditions of maximum $q\alpha$:

Load Parameters

Loads x 10^{-3}

Interface	S Shear (lbs.)	P Axial Load (lbs.)	Moment M_A (in-lb)	M Moment ΔM_z (in-lb)	Moment ΔM_y (in-lb)
Adapter/IU	63.0	-269.0	21,800	260	6.3

(Apollo Station 502)

Note: M_A is the moment due to the trajectory and trajectory dispersions. ΔM_z and ΔM_y are fixed direction moments due to the physical design of the vehicle (CG locations, asymmetry, etc.)

3.1.1.3.2 Booster Aero Data. - Loads evaluated for the max $q\alpha$ condition shall use the Booster normal force coefficient and center of pressure as given in Figure 51, and the normal force distribution, for the booster only, as given in Figure 52.

3.1.1.3.3 Booster Control System Criteria. - The system gains for the autopilot are derived by using a combination of the minimum drift and minimum load principles having an auxiliary feedback loop utilizing an angle of attack or an accelerometer sensor.

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3.1.1.3.4 Booster Stiffness. - Distributions of EI and KAG for the Saturn V booster are given in Figures 53 thru 56.

3.1.1.3.5 Booster Weight Distribution. - Weight distributions for the Saturn V booster structure and propellant are given in Figures 57 thru 61.

3.1.1.4 Saturn IB Performance Requirements:

3.1.1.4.1 Payload Capability. - For the Saturn IB missions the LV shall be capable of injecting 32,500 pounds, based on a 8,200 pound Launch Escape Subsystem (LES), into a nominal 100 n.m. Earth orbit.

3.1.1.4.2 Trajectory Requirements, Saturn IB. - The LV shall insert the CSM at cutoff into the particular mission design trajectory within the accuracies defined by the following flight parameters:

Eccentricity, e,	±0.00725
Semi major axis, a.	±10.70 nautical miles
Ascending node,	±0.210 degrees
Inclination,	±0.0839 degrees
True anomaly of the ascending node (If applicable for the particular mission)	±0.830 degrees

3.1.1.5 Saturn V Performance Requirements. -

3.1.1.5.1 Payload Capability. - For the Saturn V missions the LV shall be capable of injecting 90,000 pounds, based on a 8,200 pound LES and a 24,500 pound LEM into a translunar trajectory of the free return type having a nominal vacuum perigee altitude of 21 n.m. with no midcourse corrections required to accomplish the trajectory.

3.1.1.5.2 Trajectory Requirements

a. Parking Orbit. - The LV shall insert the CSM at cutoff into the particular mission design trajectory for the parking orbit within the accuracies defined by the following flight parameters:

Eccentricity, e,	±0.00725	
Semi major axis, a,	±10.70	nautical miles
Ascending node,	±0.210	degrees
Inclination,	±0.0839	degrees
True anomaly of the ascending node (If applicable for the particular mission)	±0.830	degrees

b. Missions. - The LV shall inject the CSM at cutoff into the particular mission design trajectory within the accuracies defined by the following orbital parameters:

Eccentricity, e,	±0.*
Semi major axis, a,	± * nautical miles
Ascending node,	± * degrees
Inclination,	± * degrees
True anomaly of the ascending node (If applicable for the particular mission)	± * degrees

3.1.1.6 Launch Vehicle Mechanical Interfaces. -

3.1.1.6.1 Physical Interfaces

a. CSM SIVB Adapter/IU Interface. - The CSM SIVB Adapter shall structurally and functionally adapt the Service Module to the Launch Vehicle. In the area of

*MSC shall assign values to these parameters that are consistent with the targeting requirements of a lunar mission and with the ΔV budget established for midcourse correction. (Ref. Para. 3.1.1.8.3) interface with the Launch Vehicle, the design of the Adapter and the design of the IU shall meet the requirements of ICD's 13M20108 (Saturn IB) "Instrument Unit to Spacecraft Physical Requirements" (original issue), 13M50103 (Saturn V) "Instrument Unit to Spacecraft Physical Requirements" (original issue). Requirements established by ICD 13M50123 "Envelope, LEM/SIVB/IU Clearance, Physical" (original issue) will be met as required for the Saturn missions involved.

Note: While the effectivity of these documents is limited to missions A201 and A501, the design requirements established therein provide a baseline reference for all Block I Saturn IB and Saturn V missions.

b. "Q" Ball to CSM Interface. - The design of the "Q"-ball and the design of upper end of the ballast enclosure shall meet the requirements ICD's 13M20109 (Saturn IB) "Spacecraft/"Q"-Ball Physical Requirements" (original issue) and 13M50112 (Saturn V) "Spacecraft/"Q"-Ball Physical Requirements" (original issue).

Note: While the effectivity of these documents is limited to missions A201 and A501, the design requirements established therein provide a baseline reference for all Block I Saturn IB and Saturn V missions.

3.1.1.6.2 CSM SIVB Adapter/IU Interface Compartment. -

a. Boost Phase Venting. - During the boost phase the Service Module, Adapter, IU and SIVB forward skirt shall be vented to atmosphere via vents to be located on the SIVB between 122 and 130 inches aft of the Adapter/IU interface. Total vent cross-sectional area shall be 200 sq. inches.

b. Purge Requirements. - Provisions shall be included in the design of the vehicle for a gas purge of the Adapter/IU interface compartment. The purge gas shall be introduced through the umbilicals in the Service Module, Adapter and IU and shall be exhausted via the boost phase vents in the SIVB. The design shall be compatible with an air purge for the control of temperature and working conditions inside the compartment and with a GN_2 purge when an explosion hazard potentially exists.

c. Lower Adapter Access Provisions. - Provisions shall be incorporated in the design of the IU for installation of platforms required for access to the lower LEM area and lower Adapter mounted CSM equipment during ground checkout and servicing operations. These platforms are to be designed and provided by the MSFC. These platforms shall also be designed such that they will provide selected base (leg) attach points and will support the vertical loads only from auxiliary two man platforms.

3.1.1.7 Launch Vehicle Electrical Interfaces. -

3.1.1.7.1 Adapter/IU Interface Provision. - Three type PT00SE-24-61S electrical connectors shall be provided in the Adapter for electrical mating of CSM and Launch Vehicle. The connectors shall be mounted to a bracket attached to the Adapter structure and located 25 inches above the SLA/IU interface, 45 degrees from the -Z axis toward the +Y axis (CSM axes, Table I).

3.1.1.7.2 "Q" Ball Interface Provisions. - Wiring shall be provided from the MSFC furnished "Q" ball to the Adapter/IU interface. Wiring shall be terminated with a ME414-0095-0062 or equivalent connector at the "Q" ball interface and one of the above type PTOOSE-24-61S connectors at the Adapter/IU interface. The interface between Launch Vehicle equipment and the CSM portion of the Launch Vehicle EDS related to the "Q" ball signals is contained in Paragraph 3.1.1.7.5.2.1, a, (3), (f).

3.1.1.7.3 Power Interface. - Electrical interfaces between CSM and Launch Vehicle shall be designed in such a manner that there will be no exchange of electrical power between CSM and Launch Vehicle.

3.1.1.7.4 Signal Interfaces. - For electrical signal interfaces, adequate electrical isolation shall be provided in the interface design so that the effectiveness of any signal crossing the interface will not be deteriorated.

3.1.1.7.5 Launch Vehicle Emergency Detection System (LV-EDS). - The LV-EDS is a system which is operative during boost flight in both the Saturn LV and the CSM system. Its purpose is to detect critical conditions arising from malfunctions within the LV and automatically transmit a signal to the LES to initiate abort action or to provide information to the CSM crew to indicate that an abort may be required. This specification is concerned only with that portion of the LV-EDS which is contained in the CSM (hereafter referred to as LV-EDS) and its relationship with the portion of the LV-EDS which is contained in the Saturn LV. Where reference is made to the LV portion of the LV-EDS, it is so indicated. Since the CSM system will be engaged in missions involving both Saturn IB and Saturn V launch vehicles, the performance and interface requirements for both those vehicles are included in this specification. These requirements are common to both vehicles except where indicated otherwise.

3.1.1.7.5.1 Performance Requirements. - The LV-EDS contained in the CSM system will provide the following capabilities.

3.1.1.7.5.1.1 Astronaut Displays. - In response to signals from the LV and other CSM subsystems, the LV-EDS will display critical conditions to the crew of the CSM.

3.1.1.7.5.1.2 Astronaut Controls. - The CSM will incorporate provisions for astronaut control of the LV-EDS in both the CSM and the LV. These controls will permit the astronaut to:

a. Switch power to or from the LV-EDS system.

b. Enable or disable the automatic abort circuitry or certain portions of that circuitry in the LV.

c. Manually initiate an abort sequence with the LES, or, after LES jettison, the SM propulsion subsystem and, concurrently, command LV active engine cutoff.

3.1.1.7.5.1.3 Automatic LV-EDS Functions. - The LV-EDS will also automatically accomplish the following functions within the CSM.

a. Enable the automatic abort circuitry at the instant of lift-off.

b. Determine through majority voting logic the validity of an automatic abort signal presence in the CSM/LV interface abort circuitry before transmitting an abort command to the LES.

c. Provide an indication to the LV-GSE of an "unsafe" condition in the LV-EDS prior to lift-off. Unsafe being defined as: "attempting to command an abort to the LES which would produce abort action if the automatic abort circuitry were enabled at that time."

d. Disable, in the CSM, the automatic abort circuitry at the instant of LES separation.

e. Initiate the LES abort and shut down the engines on active LV stage upon a valid command from the CSM/LV interface abort circuitry.

3.1.1.7.5.2 Functional Interfaces. -

3.1.1.7.5.2.1 Launch Vehicle Interfaces. - Interchange of LV-EDS signals between the CSM and the LV will be as shown below. The power source to operate these signals shall be as indicated:

a. Launch Vehicle to Command and Service Module Signals. -

(1) Automatic Abort Circuitry. - Loss of power in two out of 3 of the CSM/LV interface automatic abort circuits shall cause abort action to be taken by the LES. CSM power shall be used for this circuitry.

(2) Automatic Enabling of Auto Abort Circuitry at Lift-Off. - A dual redundant signal from the LV-IU to the CSM shall cause the automatic abort circuitry in the CSM to be enabled, i.e., to be switched into a state of operational readiness. CSM power shall be used for this circuitry.

(3) Display Circuits. - CM displays will be activated as shown below on receipt of signals from the LV. These display circuits are normally de-energized prior to signal transmittal.

(a) Engine Status Signals. - A discrete signal from the LV-IU to the CM will indicate the non-thrusting status of each of the active LV engines. Eight signal paths will be provided on Saturn IB missions for use during SIB stage burn and 1 signal path for SIVB stage burn. On Saturn V missions, 5 signal paths will be provided for use during SIC and S-II stage burn and one signal path for SIVB stage burn. The signals for the different stages on each vehicle shall utilize common circuitry. CSM power shall be used for this circuitry.

(b) Excessive Rate Signal. - A discrete signal from the LV-IU to the CM will indicate that the LV rate limit in any of the pitch, roll or yaw planes has been exceeded. CSM power shall be used for this circuitry.

(c) Launch Vehicle Guidance Failure Signal. - A discrete signal from the LV-IU to the CM will indicate that the LV guidance system has failed and that attitude control is lost, (Rate control will still be operative). CSM power shall be used for this circuitry.

(d) Abort Request Signal. - A discrete signal from the LV-IU to the CM will indicate that either the Range Safety Officer has transmitted a Destruct and Engine Cutoff command to the LV or that Launch Control Center (LCC) is indicating an abort necessity. CSM power shall be used for this circuitry.

(e) Lift-Off Signal. - A discrete signal will be transmitted from the LV-IU to the CM to indicate that lift-off has occurred. CSM power shall be used for this circuitry.

(f) Angle of Attack Signal. - An analog signal will be transmitted from the "Q"-ball/CSM interface to provide a continuous readout of an aerodynamic parameter which is a function of angle of attack. The signal from the "Q"-ball interface will be a single signal representing the combined Pitch and Yaw vectors to give a total angle of attack function readout. The display parameter will be differential pressure across the "Q"-ball on the LES. LV power shall be used for this circuitry.

(g) S-II Stage Fuel Pressure Signal (On Saturn V Missions Only). - An analog signal will be transmitted from the LV-IU to provide a continuous readout of S-II fuel tank pressure. LV power shall be used for this circuitry.

(h) SIVB Stage Fuel Pressure Signal (On Saturn V Missions Only). - An analog signal will be transmitted from the LV-IU to provide a continuous readout of SIVB fuel tank pressure. LV power shall be used for this circuitry.

(i) S-II Stage Second Plane Separation Signal (On Saturn V Missions Only). - A discrete signal from the LV-IU will indicate that S-II second plane separation (S-II aft skirt) has occurred. CSM power shall be used for this circuitry.

b. Command and Service Module to Launch Vehicle Signals. -

(1) Launch Vehicle Engine Cutoff Circuitry. - An abort command transmitted to either the LES or SM propulsion systems (after LES jettison) will cause an engine cutoff signal to be transmitted from the CSM to the LV. This signal will consist of loss of power in at least two out of three energized circuits crossing the CSM/LV interface. LV power shall be used for this circuitry.

(2) Astronaut Manual Control Circuitry. - Upon astronaut initiation of the following functions, signals will be transmitted to the LV as indicated.

(a) Two Engine Out Auto-Abort Disable. - When the astronaut commands this function, a signal will be transmitted from the CM to the LV-IU. The interface circuitry shall consist of triple redundant wire paths which become energized when the disabling signal is transmitted. LV power shall be used for this circuitry.

(b) LV Excessive Rates Auto-Abort Disable. - When the astronaut commands this function, a signal shall be transmitted from the CM to the LV-IU. The interface circuitry shall consist of triple redundant wire paths which become energized when the disabling signal is transmitted. LV power shall be used for this circuitry.

(3) LV-EDS Unsafe Signal. - Prior to lift-off, the CM will supply a signal to the LV-IU (for subsequent action in the LV-GSE release ladder to prevent lift-off) in the event the LV-EDS circuitry is in an "unsafe" condition. LV power shall be used for this circuitry.

3.1.1.7.5.2 LV-EDS/CSM SCS/CSM G&N Interfaces. - In addition to the displays which are provided for signals from the LV/CSM interface, the following parameters will be displayed for detection of critical conditions arising from LV malfunctions. These displays will be provided by the CSM SCS, the signals for which will be in turn provided by the CSM G&N subsystem (operating in the monitor mode) as described in SID 62-1000, G&N Performance and Interface Specification.

a. Attitude Error. - A continuous readout of vehicle attitude error shall be provided. For the provision of this readout, it will be necessary that the CSM G&N subsystem be programmed prior to flight in consonance with the planned LV pitch program for first stage boost.

b. Total Attitude. - A continuous readout of vehicle total attitude shall be provided.

c. Angular Rates. - A continuous readout of vehicle angular rates in each of the Pitch, Yaw and Roll planes shall be provided.

3.1.1.8 Command and Service Module.

3.1.1.8.1 Command Module (CM). - The CSM shall include a recoverable CM. This module shall contain the communication, navigation, guidance, control, computing, display equipment, etc., requiring crew mode selection. In addition, other equipment required during nominal or emergency Earth landing phases shall be included in the CM. This module shall include features which allow effective crew observation with a field of view for general observation. Equipment arrangements shall allow access for maintenance prior to Earth launch. The CM shall provide for sufficient storage of experimental measurement equipment as specified in SID 62-1001, Flight R&D Instrumentation Specification.

a. Housing. - The CM shall house three crew members during the launch, translunar, lunar orbit, transearth and entry phases.

b. Entry and Earth Landing. - The CM shall be the entry and Earth landing vehicle for both nominal and emergency mission phases.

c. Ingress and Egress. - The side ingress and egress hatch to the CM shall be used during countdown or recovery. No provisions shall be made for extra-vehicular activity during flight.

3.1.1.8.2 Service Module (SM). - An unmanned SM will be provided for all missions. This unmanned module shall contain stores and systems which do not require crew maintenance or direct operation, and which are not required by the CM after separation from the SM. The SM shall house all propulsion subsystems required for midcourse corrections, lunar orbit insertion, lunar orbit maneuvers and transearth injection. The SM will be jettisoned prior to entry into the Earth's atmosphere.

3.1.1.8.3 Command and Service Module Propulsion Increments After SIVB Separation. - After separation of the SIVB, propulsion increments of the CSM shall be supplied by the SPS. For mission comparison purposes, weight reports, etc., the SPS characteristic velocity budget utilized shall be as indicated for the following mission phases.

Mission Phase	Incremental Velocity (FPS)
a. Translunar	
(1) Midcourse	300
(2) Lunar Orbit Injection	3230
(3) ΔV Margin (10%)	353
b. Transearth	
(1) Lunar Orbit Maneuvers	455
(2) Transearth Injection	3610
(3) Transearth Midcourse	300
(4) ΔV Margin (10%)	436

3.1.1.9 Command and Service Module SIV-B Adapter. - The CSM/S-IVB Adapter shall structurally and functionally adapt the SM to the LV.

3.1.1.10 Launch Escape Subsystem (LES). - Provisions shall be made to separate the CM from the LV in the event of failure or imminent failure of the LV during all atmospheric phases.

3.1.1.11 Command and Service Module Subsystems. - The CSM subsystems requirements and subsystem descriptions are contained in SID 62-1237, Vehicle Model Specification - Basic.

3.1.2 Operational Concept. -

3.1.2.1 Manning of Flight. - The CSM shall be designed for manned operation with full utilization of human crew capabilities. Automatic subsystems shall be employed only where they will enhance the performance of the mission.

3.1.2.2 Onboard Command. - The CSM shall have the capability to perform aborts independent of ground based information and command.

3.1.2.3 Flight Crew. - The CSM flight crew shall consist of three men.

3.1.2.3.1 Crew Participation. - The flight crew shall have the capability to control the CSM throughout all flight modes. The flight crew shall participate in navigation, control, monitoring, computing, and observation as required. Status of subsystems shall be displayed for crew monitoring, failure detection and operational mode selection. The CSM shall be designed so that a single crewman will be able to perform all tasks essential to return the CSM in case of emergency.

3.1.2.3.2 Abort Initiation. - Provisions shall be made for crew initiation of all abort modes. Initiation of abort modes by automatic subsystems shall be provided to enhance crew safety.

3.1.2.4 Flight Time Capabilities. -

3.1.2.4.1 Flight. - The Apollo CSM shall be designed to accomplish the Lunar Orbit Mission. The CSM consumables subsystems shall be designed for a nominal mission time of 10.6 days with three of these days in lunar orbit using the ΔV allocations shown in paragraph 3.1.1.8.3. By judicious system management of duty cycles, alternate missions, such as Earth orbital, may be performed within the resultant capabilities of the CSM system.

3.1.2.4.2 Post Flight. - The CM shall provide the crew with a habitable environment for a minimum of 48 hours and a floatation environment of 7 days following a water landing.

3.1.2.5 Earth Landing. - The CSM shall have the capability of initiating a return and Earth-landing maneuver at any time during either lunar or Earth orbital missions. Prior to each flight, a primary water landing site and suitable backup water landing site will be selected for normal mission landing. Additional criteria apply as follows:

3.1.2.5.1 Lunar Orbit Mission. - The CSM shall be capable of landing at the primary water landing site (or at any backup site) upon returning to Earth following a lunar orbital mission. In addition, alternate sites shall be designated so that at least one alternate site on Earth can be reached.

3.1.3 Command and Service Module Performance. - The following sub-paragraphs summarize the nominal performance capabilities of the CM, SM, and S-IVB Adapter.

3.1.3.1 Boost Stabilization. - The effects of windage, aerodynamics, variations of the center of gravity, etc., will be compensated for by the launch vehicle during the boost phase.

3.1.3.2 Trajectories. - The general CSM trajectories shall follow the general requirements shown in paragraph 3.1. After trans-lunar injection, the primary measured CSM positional accuracy shall be provided by the MSFN with the CSM G&N Subsystem serving as a back-up system in accordance with SID 64-881, NASA Manned Space Flight Net Performance and Interface Specification - Primary.

3.1.4 Mission Performance.

3.1.4.1 Flight Plan. - The Apollo mission flight plan for which the CSM is sized shall be as specified in 3.1.4.1.1 and 3.1.4.1.2.

3.1.4.1.1 General Flight Plan Requirements and Characteristics. - The general flight plan requirements and characteristics present the general mission ground rules to which the CSM shall be designed. These ground rules consist of trajectory parameters and operational constraints which shall be considered in overall CSM and subsystem design. The characteristics of the lunar missions are described in the following sub-paragraphs:

a. Launch Site. - All lunar orbital missions shall be launched from Cape Kennedy, Florida. The launch azimuth shall be within limitations set by range safety and tracking considerations. The launch phase for lunar orbital missions begins with S-IC ignition and ends with S-IVB cutoff in Earth parking orbit.

b. Launch Time Window. - Lunar orbital mission flight plans shall include at least a two hour period on the launch date. Launches may be made providing visual reference conditions sufficient for orientation during high altitude abort exist. A launch window shall be provided either by maneuvering the CSM 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 capability is to be provided by a Government Furnished Launch Vehicle from a Government Furnished Launch Complex.

c. Earth Parking Orbit. - The Earth parking orbit phase begins with S-IVB cutoff in orbit and ends with S-IVB relight for translunar injection. The parking orbit altitudes for lunar orbital mission shall be limited to altitudes from 90 to 120 nautical miles. The nominal parking orbit altitude shall be 100 nautical miles. Multiple parking orbits are acceptable but shall be compatible with booster performance and lifetime limitations. The duration of this phase shall not exceed four and one-half hours.

d. Translunar Injection. - The translunar injection phase begins with S-IVB ignition in Earth parking orbit and ends with S-IVB cutoff. Final injection into the translunar trajectory shall be located such that the trajectory can be determined by the MSFN within 15 minutes of translunar injection burnout.

e. Translunar Coast. - The translunar coast phase begins with S-IVB cutoff and ends with SPS ignition for lunar orbit insertion. The translunar trajectory for lunar orbit missions shall be a free return type which has a coast return to the Earth with acceptable entry conditions. The duration of this phase shall be from approximately 59 to 77 hours depending upon the Earth-Moon distance, the inclinations of the geocentric translunar and transearth planes to the Moon's orbit plane, and the injection velocity. The translunar trajectories for lunar orbit missions shall have a nominal pericynthion of 100 nautical miles. The CSM shall include provisions for performing translunar midcourse correction maneuvers.

f. Lunar Orbit Insertion. - Lunar orbit insertion begins with SPS ignition just prior to pericynthion and ends with SPS cutoff in lunar orbit. Insertion will occur over the non-visible portion of the Moon. The CSM shall arrive on a circumlunar trajectory which has a nominal pericynthion altitude of 100 nautical miles. A 5° plane change capability shall be provided within the lunar orbit injection velocity budget for establishing the initial orbit. The maneuver shall be accomplished at the same time as the retro-maneuver for establishing the lunar orbit.

g. Lunar Orbit. - The lunar orbit phase begins with SPS cutoff in lunar orbit and ends with SPS ignition for transearth injection. The nominal lunar orbit altitude shall be 100 nautical miles.

h. Transearth Injection. - Transearth injection begins with SPS ignition in lunar orbit and ends with SPS cutoff. The SM propulsion subsystem shall be capable of providing the necessary propulsion performance to transfer from the lunar orbit to the transearth trajectory. The maneuver required is a function of the characteristic of the parking orbit at the time of injection, the time spent in orbit, and the terminal constraints at perigee which must be satisfied. The terminal constraints which must be satisfied are the Earth atmospheric entry angle, geocentric conic inclination, and the entry epoch. The required entry angle shall be limited such that capture is insured without exceeding the aerodynamic heating or loads limitations. The position of the vehicle at the time of injection will be over the non-visible side of the Moon.

i. Transearth Coast. - The transearth coast phase begins with SPS cutoff and concludes at the entry interface. The duration is determined by the transearth injection conditions and shall range between 60 and 84 hours to allow for return to the primary landing site. The inclination of the transearth trajectory to the Earth's equator and the time of flight shall be used to control the entry in such a way that the entry track will be over planned tracking and recovery areas. The CSM shall include provisions for performing transearth midcourse correction maneuver. Transearth trajectories shall be such that nominal entry for Apollo missions will be with posigrade motion with respect to the Earth to reduce the entry heating and widen the entry corridor.

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j. Entry. - The entry phase begins at the entry interface (nominally 400,000 ft.) and ends at drogue parachute deployment. The CM shall be capable of entry within a 15 nautical mile corridor with a peak deceleration of 7 g during the initial pull-out with a minimum L/D=0.30 and at a parabolic velocity of 36,333 fps when measured in a vacuum at perigee. The maximum range at minimum L/D shall be 2500 n.m. The maximum peak deceleration limit shall not exceed 20 g.

k. Recovery. - The recovery phase covers the time commencing with drogue parachute deployment and ending with touchdown of the CM.

l. Command and Service Module Attitude. - When simultaneous requirements for preferred attitudes exist among CSM functions and subsystems, conflicts shall be resolved in accordance with the following priority:

(1) Crew Safety and operations.

(2) Thermal constraints.

(3) Thrust and lift vector management.

(4) Nuclear protection.

(5) Communications

(6) Conservation of Reaction Control (RCS) Propellant.

(7) Navigation

Attitude control is permissable to eliminate system constraints which would impose successive subsystem requirements.

3.1.4.1.2 Control Weight and Consumables Design Mission. - The following 10.6 day lunar orbit mission timeline shall serve as a basis for provisioning of consumables and for establishing CSM control weights:

Mission Phase	Duration Hours
Prelaunch	10.00
Ascent Phase	0.19
Earth Parking Orbit	4.40
Translunar Injection	0.09
Translunar Coast	77.00

[redacted]

Lunar Orbit Injection	0.09
Lunar Orbit Coast	88.00
Transearth Injection	0.04
Transearth Coast	84.00
Pre-Entry	0.08
Entry	0.50
Recovery	0.17

3.1.4.2 Contingencies. - A contingency situation is the result of any deviation from the mission plan which requires a decision to be made concerning future conduct of the mission. Such deviations can include those concerned with schedule, structural characteristics, vehicle or subsystem performance, crew condition, random natural hazards and others. The CSM individually, and the overall Apollo system, as a whole, shall be capable of resolving contingencies in order to meet the specified probabilities of crew safety and of mission success.

3.1.4.2.1 Design Objective. - Overcoming contingency situations requires operational and performance flexibility. This flexibility shall be provided by the following design objectives:

a. Built-in redundancy

b. Switch-in redundancy

c. Alternate operating modes

3.1.4.2.2 Criteria for Contingency Operation. - Performance requirements for CSM operation under contingency conditions shall be based on the following criteria (listed in approximate order of significance):

a. Adequate crew safety

b. Mission success

c. Adequate fuel margin

d. Minimum response-time criticality

e. Primary landing area

f. Adequate margin for consumables

g. Manned Spacecraft Control Center (MSCC) and MSFN assistance

h. Hardware reliability

i. Minimum number of abort trajectories

j. Minimum flight-plan complexities

k. Performance flexibility

3.1.4.2.3 Contingency Operations. - Crew response to a contingency will comprise, in general, the operations described below.

a. Detection of Contingency. - The crew members shall be alerted to the contingency occurrence by one or more of the following:

(1) CSM displays and controls

(2) Telemetry/communication loops

(3) Telemetry/up-data link

(4) Lack of response to command inputs

(5) Physical sensing by astronaut

(6) Caution and warning display

b. Isolation of Contingency. - To aid the crew in isolation of contingencies, all information required to assure crew safety shall be stored on board the CSM in a readily accessible manner. Pertinent information affecting mission success shall be stored on board where practicable. Complete information at all levels and quantitative predictions of future missions status shall be available from MSCC via MSFN within existing communications capabilities.

c. Evaluation of Contingency. - On-board stored contingency data shall clearly identify contingencies where crew safety may be jeopardized and where time may be a constraining factor.

d. Implementation of Contingency Resolution. - The resolution of all contingencies shall be initiated by the crew. Automatic initiation shall be invoked only when the response time or the complexity of the evaluation and implementation process are beyond reasonable human limitations.

3.1.4.2.4 Abort Factors. - For abort action, the on-board stored contingency data shall normally provide abort-selection criteria including propulsive fuel, time, and landing area.

a. Propellants. - Data listing ΔV requirements for discrete abort trajectories shall be readily available on board. Sufficient conversion data shall be available on board to convert propellant readings to ΔV capabilities.

b. Time. - Time histories for discrete abort trajectories shall be readily available on board. Sufficient information concerning consumable usage rates under varying operational conditions shall also be available on board to enable reasonable predictions on future consumable status. In addition, those contingencies which require a timely response shall be identified in the on-board stored data.

c. Information Retrieval. - On-board stored data shall be in sufficient detail to provide adequate assurance of crew safety even if communications to MSCC are not available. An efficient unambiguous indexing method shall be provided to enable speedy retrieval by the astronauts of adequate information from the on-board stored data.

3.2 Command and Service Module System Design and Performance Criteria

3.2.1 General Design Analysis Criteria. - Design and operational procedures shall be conducted in accordance with rational design principles to include but not be limited to the following:

3.2.1.1 Limit Conditions. - The design limit load envelope shall be established by superposition of rationally deduced critical loads for all flight modes. Load envelopes shall recognize the cumulative effects of additive type loads. No subsystem shall be designed incapable of functioning at limit load conditions.

3.2.1.2 Performance Margins. - Rational margins shall be apportioned to subsystems and components such that the greatest overall design efficiency is achieved within the LV capabilities and implementation criteria constraints.

3.2.1.2.1 Multiple Failure. - The decision to design for single or multiple failures shall be based on the expected frequency of occurrence as it affects subsystem reliability and safety.

3.2.1.2.2 Fail Safe. - Subsystem or component failure shall not propagate sequentially, i.e., design shall "fail safe."

3.2.1.2.3 Design Margins. - All CSM subsystems shall be designed to positive margins of safety except for special cases subject to rational analysis or destructive tests and negotiation with Manned Spacecraft Center (MSC), National Aeronautics and Space Administration (NASA).

3.2.1.3 Performance Criteria

3.2.1.3.1 Command Module Repressurization. - The repressurization subsystem shall be designed for two complete cabin repressurizations and a continuous leak rate as high as 0.2 lbs per hour. Provisions only shall be made for resupply of Portable Life Support Subsystem (PLSS) expendables. Additional supply of the PLSS expendables shall not be required.

3.2.1.3.2 Vacuum Operation of Cabin Equipment. - Vacuum, for design criteria purposes, shall be defined as follows:

For CSM exterior: 7.5×10^{-10} mm hg

For CM interior: 10^{-4} mm hg

For SM interior: 10^{-6} mm hg

3.2.1.3.3 Command Module Reuse. - The CM and internal subsystems shall not be designed for repeated mission reuse after recovery.

3.2.1.3.4 Command Module Water Stability. - CM flotation and water stability characteristics shall be such that the CM will recover from any initial attitude and will float upright with normal egress hatches clear of the water.

3.2.1.4 Command and Service Module Design Criteria. -

3.2.1.4.1 Thermal Resistance. - The CSM modules shall be designed with considerations of thermal resistance to provide best protection without compromise to basic module design.

3.2.1.4.2 Maintenance. - Equipment arrangements, accessibility, and interchangeability features that allow efficient preflight servicing and maintenance shall be given full consideration. Design considerations shall also include efficient mission scrub and recycle procedures.

3.2.1.4.3 Ground Handling. - Full design recognition shall be given to the durability requirements of CSM equipment and subsystems during preflight preparation.

3.2.1.4.4 Ultimate Factor. - The ultimate factor shall be 1.5 applied to limit loads. This factor may be reduced to 1.35 for special cases subject to rational analysis and negotiation with MSC, NASA.

3.2.1.5 Explosive Initiators. - The contractor shall standardize all CSM initiators insofar as possible by utilizing hot wire initiators.

3.2.1.6 Environmental Criteria. - These requirements define the environmental design criteria for the CSM equipment and associated Ground Support Equipment (GSE).

3.2.1.6.1 Command and Service Module and GSE Ground Environments

a. Transportation, Ground Handling, and Storage. - The following conditions represent the natural and induced environmental extremes which may be encountered during transportation, ground handling and storage. Handling GSE shall be capable of operating during exposure to these environments. Other GSE and CSM equipment may be protected by suitable packaging if these environments exceed those experienced during normal operation.

(1) Natural Environments

(a) Temperature

Air transportation	-45 F to + 140 F for eight hours
Ground transportation	-20 F to + 145 F for two weeks
Storage	+25 F to +150 F for three years

(b) Altitude

Air transportation	Up to 35,000 feet for eight hours

	Ground transportation and storage	Up to 6,000 feet for three years
(c)	Humidity	0 to 100 percent relative humidity, including conditions wherein condensation takes place in the form of water or frost for at least thirty days.
(d)	Sunshine	Solar radiation of 360 Btu per square foot per hour for six hours per day for two weeks
(e)	Rain	Up to 0.6 inch per hour for twelve hours
(f)	Sand and dust	As encountered in desert and ocean beach areas, equivalent to 140-mesh silica flour with particle velocity up 500 feet per minute and a particle density of 0.25 grams per cu ft.
(g)	Fungus	As experienced in Florida climate.
(h)	Salt spray	Salt atmosphere as encountered in coastal areas, the effect of which is simulated by exposure to a 5 percent salt solution by weight for 48 hours.
(i)	Ozone	Up to 3 years exposure to 0.05 parts/million concentration.
(j)	Ground winds	These ground wind criteria consist of a description of Cape

Kennedy wind data for the height intervals of 10 to 400 feet.

1. Free Standing - The design wind speeds for structural loading considerations of the CSM are presented in the table below. Wind speed occuring during the strongest wind month at Cape Kennedy, Florida, are less than those presented 99.9 percent of the time.

Height (ft)	Steady State Wind (knots)	Peak Wind (knots)(*)
10	23.0	32.2
30	28.7	40.2
60	32.9	46.1
100	36.5	51.1
200	41.9	58.7
300	45.4	63.6
400	48.1	67.3

(*) 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 for decay to steady state wind speed.

2. Storm Conditions - The 99.9 percent peak wind speeds presented in paragraph 3.2.1.6, (1), (j), 1. may be exceeded during severe thunderstorm or hurricane condition at Cape Kennedy. During such periods, the vehicle must be protected in such a manner that wind loading conditions greater than those for the 99.9 percent winds shall not be experienced by the CSM.

(2) Induced Environment

(a) Shock - as experienced in any direction

Weight (pounds)**	Shock Level (g)	Time (milliseconds)

Less than 250	30	11 ± 1 (half-sine waveform)
250 to 500	24	11 ± 1 (half-sine waveform)
500 to 1,000	21	11 ± 1 (half-sine waveform)
Over 1,000	18	11 ± 1 (half-sine waveform)

**Weight of equipment and package or containers (if any).

(b) Vibration, - Sinoidal as experienced in any direction

Weight (pounds)	5 to 26.5 cps	26.5 to 52 cps (inch DA)	52 to 500 cps
Less than 50	±1.56g	0.043	±6.0g
50 to 300	±1.30g	0.036	±5.0g
300 to 1,000	±1.30g	0.036	
Over 1,000	±1.04g	0.029	

b. Sheltered Environment Areas - These requirements represent the environment design criteria for CSM equipment and GSE both in operating and non-operating condition. The equipment shall be capable of meeting the operating requirements of the applicable performance specification during and after exposure to these environments. Natural and induced environments are combined in this section. The level of environmental control at each Apollo site shall be as indicated in MSC-GSE-1B.

(1) Interior Controlled - An environment in which the temperature, humidity, sand, salt spray, etc., are controlled.

(a) Temperature — +60 F to + 80 F for up to three years

+52 F to +105 F for one hour maximum with environmental equipment out of commission

(b) Oxygen Atmosphere — The following conditions apply to the CM interior:

[redacted]

	95 ± 5 percent by weight oxygen at total pressures up to 14.7 psia for up to 24 hours. Oxygen partial pressure up to 14.7 psia coincident with total pressure up to 21.0 psia for two hours
(c) Humidity	30 percent to 70 percent for up to three years
(d) Sand and Dust	Particle count not to exceed Level 300,000 of Federal Standard 209: No more than 2,000 particles per cubic foot larger than 5 microns. No more than 35 of these larger than 65 microns. No more than 3 of these 35 particles larger than 100 microns.

(2) Interior Uncontrolled - An environment in which the temperature, sand, salt spray, etc., are only partially controlled.

(a) Temperature	+15 F to +105 F for up to three years
(b) Humidity	0 to 100 percent relative humidity, including conditions wherein condensation takes place in the form of water or frost for at least thirty days.
(c) Sunshine	Solar radiation at 360 Btu per square foot per hour for 6 hours per day for 2 weeks.
(d) Sand and dust	As encountered in desert and ocean beach areas, equivalent to 140 mesh silica flour with particle velocity

up 500 feet per minute and a particle density of 0.25 grams per cu. ft.

c. Other Environment Areas - Environments to which certain GSE are exposed, such as the Launch Umbilical Tower (LUT) and Environmental Chamber, shall be as indicated in MSC-GSE-1B.

3.2.1.6.2 Command and Service Module Flight Environments. - These requirements represent the environmental design criteria for the CSM equipment in an operating condition as experienced during the various flight mission phases. The mission phases are as defined in paragraph 3.1.4.1.1. The equipment shall be capable of meeting the operating requirements of the applicable performance specification during and after exposure to these environments.

a. All Mission Phases - These are induced environments which are present for all mission phases.

(1) Temperature — The contractor shall provide temperature requirements for structure, subsystem, and component design for each applicable mission phase.

(2) Oxygen Atmosphere — The following conditions apply to the CM interior: 95 ± 5 percent by weight oxygen for 255 hours. Nominal CM interior atmospheric composition is presented in the following table:

Constituent Gas	Partial Pressure (psia)	% By Vol.	% By Wt.
Oxygen	4.638	92.76	93.49
Carbon Dioxide	0.147(max)	2.94	4.07
Water Vapor	0.215	4.30	2.44

(3) Humidity

The following conditions apply to the CM interior:
0 to 100 percent relative humidity for 255 hours.
40 to 70 percent nominal relative humidity.
95 ± 5 percent relative humidity including conditions where condensation takes place in the form of water, for at least 20 hours.

(4) Corrosive Contaminants

The following condition applies to the CM interior: Salt atmosphere as caused by human perspiration, the effect of which is simulated by exposure to a one percent salt solution by weight for 48 hours.

b. <u>Ascent Phase</u>

(1) Natural Environments

(a) Reference Atmosphere

The reference Earth Atmosphere at Cape Kennedy, Florida, shall be as presented in the following table.

Altitude (feet)	Atmosphere in Accordance with
Up to 101, 705	NASA TN D-595
101, 705 to 295, 272	AFCRL-62-899
Over 295, 272	U.S. Standard Atmosphere, 1962

(b) Ground Winds

The design wind speed for launch of the CSM are presented in the table below. Wind speeds occuring during the strongest wind month at

Cape Kennedy, Florida, are less than those presented 99.0 percent of the time.

Height (ft)	Steady State Wind (knots)	Peak Wind (knots)*
10	18.4	25.8
30	22.9	32.1
60	26.3	36.8
100	29.2	40.9
200	33.5	46.9
300	36.3	50.8
400	38.5	53.9

*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 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.

(c) Winds Aloft. - Spacecraft design shall consider a 99 percentile wind shear (Figure 50) with a 9 meter per second discrete quasi-square wave gust superimposed, such that the total does not exceed a 88 percentile wind speed.

(2) Induced Environments

(a) Pressure	The following condition applies to the CM interior: 14.7 psia nominal decreasing to 6.0 psia
(b) Vibration	Mechanical vibration from all sources of excitation as experienced by the CSM structure. The design vibration levels for various zones of the CSM are presented in Figures 3 through 7.
(c) Acoustics	Acoustic noise resulting from ground reflection and aerodynamic turbulence. The design acoustics levels for various zones of the CSM are presented in Figures 8 through 20.

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(d) Acceleration — The design sustained acceleration levels for the CSM are presented in Figures 21 through 24.

c. <u>Earth Parking Orbit, Translunar Injection, Translunar Coast, Lunar Orbit Insertion, Lunar Operations, Transearth Coast, and Pre-entry Phases</u>

(1) Natural Environments

Electromagnetic Radiation	The source of electromagnetic radiation presented below impinge on the exterior of the CSM in logical combination for a total time up to 255 hours.
Solar Flux	442 Btu/ft^2-hr
Earth Emission	73 Btu/ft^2-hr
Lunar Emission (subsolar point)	287 Btu/ft^2-hr
Lunar Emission (dark side)	5 Btu/ft^2- hr
Earth Albedo	0. 34
Lunar Albedo	0. 124

(2) Induced Environments

(a) Pressure

Location	Pressure	Max. Exposure Time
CSM Exterior	7.5×10^{-10} mmHg	255 hours
CSM Interior (SM and CM Forward and Aft Compartments)	1.0×10^{-6} mmHg	255 hours

Location	Pressure	Max. Exposure Time
CM Interior	6.0 psia decreasing to 5.0 psia	(parking orbit only)
	5.0 ± 0.2 psia (normal)	255 hours
	1.0×10^{-4} mmHg (emergency)	100 hours

(b) Vibration — The design levels for the CSM are presented in Figures 25 and 26.

d. <u>Entry Phase</u>

(1) Natural Environments

Reference Atmosphere — The reference Earth atmosphere for primary and contingency landing sites shall be in accordance with U.S. Standard Atmosphere, 1962.

(2) Induced Environments

(a) Pressure — The following condition applies to the CM interior:
5.0 psia increasing to 5.5 psia (nominal)

(b) Vibration — The design vibration levels for the CM are presented in Figure 5. Uniformly reduced by 10 db.

(c) Acceleration — The design sustained acceleration level is 20 g.

e. Recovery Phase

(1) Natural Environments

(a) Reference Atmosphere	Same as paragraph 3.2.6.2 d, (1).
(b) Sea State	
Wind velocity	16 to 20 knots
Wave height (crest to trough)	0 to 8-1/2 feet first two days-14 to 18 feet next five days
Wave period	5 to 6 seconds
Wave length	125 to 185 feet
Wave velocity	14 to 18 knots

(2) Induced Environments

(a) Pressure	The following condition applies to the CM interior: 5.5 psia (nominal) increasing to 14.7 psia (nominal)
(b) Shock	Terminal peak saw-tooth pulse of 78 g (peak amplitude) with total duration 10 to 15 milliseconds, including decay time

no greater than 10 percent of the total duration. Figures 27 and 28 define the Shock directions.

f. Launch Aborts - Only the equipment necessary for the successful completion of a launch abort shall be capable of meeting the operating requirements of the applicable performance specification during and after exposure to these environments.

(1) Natural Environments

Same as paragraph 3.2.1.6.3b, (1),

(2) Induced Environments

(a) Vibration — Mechanical vibrations from all sources of excitation as experienced by primary structures. The design vibration levels for various zones of the CSM are presented in Figures 29 and 30.

(b) Acoustics — Acoustic noise resulting from aerodynamic turbulence and the launch escape motor. The design acoustics levels for various zones of the CSM are presented in Figures 31 through 36.

(c) Acceleration — The design sustained acceleration level is 20 g.

3.2.1.6.3 Command Module Post Landing Environments. - These requirements represent the environmental design criteria for CM equipment in an operating and a nonoperating condition. Operating equipment is that needed for CM habitability and location. This equipment shall be capable of meeting the operating requirements of the applicable performance specification during exposure to these environments.

a. Natural Environments

(1) Temperature	+15 F to +105 F for up to 48 hours
(2) Altitude	Sea Level
(3) Humidity	Up to 100 percent relative humidity for up to 48 hours.
(4) Sunshine	Solar radiation of 360 Btu per square foot per hour for 6 hours per day.
(5) Rain	Same as paragraph 3.2.1.6.1, a, (1), (e).
(6) Sand and dust	Same as paragraph 3.2.1.6.1, a, (1), (f).
(7) Salt Spray	Same as paragraph 3.2.1.6.1, a, (1), (h).
(8) Sea State	Same as paragraph 3.2.1.6.2, d, (1), (b).

3.2.1.7 Weights. - The weight of the CSM shall be minimum consistent with design requirements and shall not exceed the control weight of 21,200 pounds for the CSM at launch excluding SPS usable propellant of 40,525 pounds.

3.2.1.7.1 Launch Escape Subsystem. - The control weight of the LES shall not exceed 8,200 pounds including ballast.

3.2.1.7.2 S-IVB Adapter. - The control weight of the CSM S-IVB adapter shall not exceed 3,800 pounds. A design weight of 3,900 pounds will be used for trajectory studies and design analyses for a non-LEM carrying vehicle because of 100 pounds of structural members required to replace LEM load carrying members.

3.2.1.7.3 Government Furnished Equipment (GFE). - The following GFE items and associated weights are those used in establishing the above control weights.

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Item	Weight Pounds
Command Module	1,180.6
Guidance & Navigation	(415.0)
Crew Systems	(765.6)
Crew (50-70-90)	528.0
Spacesuits (3)	90.8
Provisions Assembly-Crew Survival (Contents only) Kit #1, #2, #3	68.1
Food Set (incl. Drinking Water Probe) for 10.6 day mission	57.3
Medical Kit - Emergency	2.8
Instrument Set - Physiological Monitor	1.5
Bioinstrumentation	3.8
Radiation Dosimeters	5.5
Communication, Electrical Monitoring & Telemetry (Suit Mounted) (3)	2.4
Constant Wear Garments (6 pr)	5.4

3.2.2 Structural Subsystem. - The CSM Structural Subsystem shall be comprised of the fundamental load carrying structures. Meteroid, radiation and passive heat protection shall be that inherent in the structure designed to carry the loads.

3.2.2.1 Subsystem Requirements

3.2.2.1.1 Structural Loads. - The primary structure of all modules of the spacecraft shall be capable of withstanding all loads resulting from the conditions specified below without requiring pressure stabilization.

a. Launch Phase. - Primary CM structures are to be designed for loads during launch as specified in paragraph 3.1.1.3

b. Entry Phase. - Primary CM structures are to be designed for a limit load of 20g during entry.

c. Noise. - The design shall accommodate sound pressure levels in the respective frequency ranges shown in Figures 8 through 20 and 31 through 36.

d. Vibration. - The effects of the steady and transient inputs shall be combined. The vibration analyses shall recognize the lower damping present in a vacuum. The vibration curves are shown in Figures 3 through 7 and 25, 26, 29 and 30.

e. Dynamic Loading. - The calculation of dynamic loads shall include the effects of engine start, rebound on the pad, lift off transients including ground winds, gusts, and wind shears.

3.2.2.1.2 Pressure Vessel. - The pressure cabin shall be separate from the thermal protection subsystem. The space between the pressure cabin and the thermal protection shall be vented to ambient to limit the collapsing pressures on the pressure vessel. No provisions shall be made for overpressures due to LV explosion except that due to the inherent structural capability resulting from maximum flight loads.

a. Pressure Vessel Limit Loads. - Limit loads shall be obtained with limit pressures. When pressure effects are relieving, pressure should not be used. Limit pressure is defined as the relief valve nominal pressure plus its tolerances and plus hydrostatic head.

b. Pressure Vessel Ultimate Factor. - The ultimate factor shall be 1.50.

c. Pressure Vessel Proof Factor. - The proof factor shall be 1.33 when pressure is applied as a singular load.

3.2.2.2 Subsystem Description

3.2.2.2.1 Launch Escape Tower (LET). - The tower structure shall form the connecting link between the CM and the structural skirt of the launch escape motor, and shall be designed to carry the loads and stresses to which it will be subjected in performing its function of aborting the CM at any point from the launch pad to 30 seconds after ignition of the Saturn S-II. The four main longitudinal members shall terminate at the CM, forming a rectangular pattern. Attachment of each of these four members to the CM shall be by means of explosive bolts, which shall function to detach the tower structure from the CM at the initiation of the jettison command. The launch escape tower shall be protected by an ablative material to prevent overheating.

3.2.2.2.2 Command Module. - The CM physical features shall be defined by aerodynamic and heating performance requirements and crew utility and well being considerations.

a. Geometric Characteristics. - The basic external geometry of the CM is shown in Figure 37. The CM shall be a symmetrical, blunt body developing a minimum hypersonic L/D of 0.30. The L/D vector shall be effectively modulated in hypersonic flight by the use of roll control.

b. Inboard Profile. - Basic arrangements of internal features fundamental to full utilization of the CM geometry shall be as shown in Figures 38, 39, and 40.

(1) Load Attenuation Swept Volume. - The crew shall be suspended on discrete load attenuation devices which normally act on Earth-landing impact.

(2) Crew Space Equipment. - Crew space equipment shall be free of protrusions and snags.

c. Center of Gravity Management. - Consideration shall be given to center of gravity management. Alteration of crew positions may be used for center of gravity management after touchdown.

d. Visibility. - Visibility shall be provided by one window over the crews head in the launch condition. Four additional windows compatible with temperature requirements of the lunar mission shall be provided for use during the flight phase.

e. Access and Egress Hatches. -

(1) There shall be one side hatch provided in the CM to be used for ground access servicing and maintenance. Normal access and egress for the crew and all on-board equipment installation shall be achieved through the side hatch. The capability shall exist for opening of the side hatch on the pad to give unobstructed access to the CM exteriors within 90 seconds.

(2) There shall be another inward opening hatch at the forward end of the crew compartment for use after landing.

f. Entry Thermal Protection. - The CM shall be designed with a thermal protection shell which will insure that the internal environment of the CM will not exceed the design limits of the structure and its enclosed system while entering the Earths atmosphere within the limits of velocities and flight path angles presented in paragraph 3. 1. 4. 1. 1 (j).

3. 2. 2. 2. 3 Service Module. - The SM shall be designed and constructed to support body loads from the SM and Adapter and provide a mounting structure for: SM subsystems, pressure vessels for EPS and ECS reactants, and the attachment for the High Gain S-Band Antenna. Space radiators shall be an integral part of the SM outer shell. The SM reference axes are delineated in Table I.

a. Inboard Profile. - The SM internal arrangement shall contain six segments and a center section. (See Figures 41 and 42.) Equipment contained in each sector shall be as follows:

(1)	Sector I	Cryogenic tankage and smaller items of other equipment
(2)	Sector II	SPS oxidizer tank
(3)	Sector III	SPS fuel tank
(4)	Sector IV	Fuel cells, SPS pressurization package, and smaller items of other equipments
(5)	Sector V	SPS oxidizer tank
(6)	Sector VI	SPS fuel tank
(7)	Center section	SPS helium storage tanks (two)

The high gain S-band antenna shall be housed below the lower SM bulkhead and inside the Adapter. The antenna shall be extended after Adapter - SM separation.

b. SPS Tank Sizing. - Tank sizing for the SPS shall provide a minimum usable propellant storage for 30,000 pounds of oxidizer and 15,000 pounds of fuel.

3.2.2.2.4 Command and Service Module/SIVB Adapter. - The CSM SIVB Adapter shall structurally and functionally adapt the Service Module to the Launch Vehicle. In the area of interface with the Launch Vehicle, design of the Adapter shall meet the requirement of ICD's 13M20108 (Saturn IB) "Instrument Unit to Spacecraft Physical Requirements" (original issue), 13M50103 (Saturn V) "Instrument Unit to Spacecraft Physical Requirements" (original issue). Requirements established by ICD 13M50123 "Envelope, LEM/SIVB/IU Clearance, Physical" (original issue) will be met as required for the Saturn missions involved.

a. Adapter Separation. - The Adapter shall be designed so that separation of the Adapter from the CSM shall be effected by severing the Adapter shell with ordnance devices at a station below the SM/S-IVB Adapter interface. Simultaneously, longitudinal shaped charges shall further separate the shell into hinged panel segments. The lower section of the Adapter shell will be left intact and attached to the S-IVB booster.

3.2.3 Guidance and Navigation Subsystem (G&N). - This subsystem shall be provided by NASA and the requirements for which the G&N subsystem is designed shall be as specified in SID 62-1000, Preliminary Guidance and Navigation System P&I Requirements Specification.

3.2.4 Stabilization and Control Subsystem (SCS). -

3.2.4.1 Subsystem Requirements. - The SCS shall provide for:

a. Flight path stability augmentation for atmospheric aborts.

b. Orientation, attitude control, and entry stabilization and control for extra atmospheric aborts. The subsystem shall accept commands from the guidance subsystem for thrust vector and entry control as specified in SID 62-1000, Preliminary Guidance and Navigation P&I Requirements Specification.

c. Stabilization and control during midcourse flight, and attitude control and orientation for application of midcourse corrections.

d. Stabilization and control during lunar orbit.*

e. Control requirements for entry guidance shall be provided by pitch and roll control.

*The stabilization of the CSM while in Earth Parking Orbit shall be provided by NASA furnished S-IVB.

3.2.4.2 Subsystem Description. - The SCS shall consist of the following basic components:

Attitude reference
Rate sensors
Control electronic assembly

Manual control
Displays
Power supplies

3.2.5 Service Propulsion Subsystem (SPS)

3.2.5.1 Subsystem Requirements. - The SPS shall supply the propulsion increments in the following normal and emergency modes.

a. All major velocity increments (> 10 fps) required for translunar midcourse velocity corrections, for insertion of the CSM into a lunar orbit, for lunar orbit maneuvers, for injection from lunar orbit into the transearth trajectory, and for transearth midcourse velocity corrections.

b. Abort propulsion after jettison of the LES.

3.2.5.1.1 Propellants. - The SPS shall utilize the following earth-storable, hypergolic propellants.

a. Nitrogen tetroxide (N_2O_4) as the oxidizer.

b. A mixture of fifty percent hydrazine (N_2H) and fifty percent unsymmetrical dimethylhydrazine (UDMH) as the fuel.

3.2.5.1.2 Performance. - The subsystem shall have the following performance characteristics.

a. Thrust = 21,900 lbs, nominal in a vacuum.

b. Specific Impulse, I_{sp} = 313 seconds (-3σ value)

c. Continuous operation = 635 seconds maximum

d. Minimum impulse bit = 5000 ± 200 pounds sec.

3.2.5.2 Subsystem Description. - The SPS shall consist of the following components:

3.2.5.2.1 Rocket Engine Subsystem. - The SPS engine shall be a single unit, liquid-fueled, pressure-fed, non-throttleable thrust generator, gimbal-mounted to permit thrust vector control with a maximum gimbal angle of ±8.5 degrees in the X-Y plane and ±6.0 degrees in the X-Z plane with multiple restart capability.

3.2.5.2.2 Propellant Subsystem. - The complete propellant subsystem shall consist of separate storage and distribution subsystems for oxidizer and fuel.

3.2.5.2.3 Helium Subsystem. - Helium gas, contained at high pressure, shall be utilized for pressurization of the propellant supplies.

3.2.6 Reaction Control Subsystem (RCS). - The CSM shall include reaction control subsystems to provide the impulse for attitude control and stabilization. The SM/RCS shall also be capable of minor translational velocity increments.

3.2.6.1 Command Module Reaction Control Subsystem (CM/RCS). - This subsystem shall be used only after separation of the CM from the SM.

3.2.6.1.1 Subsystem Requirements. - The subsystem shall provide three axis control prior to development of aerodynamic moments, roll control during entry, and pitch and yaw damping during entry. A roll acceleration of at least 10°/sec/sec shall be provided during entry. The pitch and yaw maneuver rates during entry shall be at least 7 degrees per second from SM separation to the 0.5 g level and at least 2 degrees per second from the 0.5 g level to recovery. A minimum impulse bit of not more than 2 pound seconds shall be provided. The subsystem shall have the capability to dump unused propellant and pressurant prior to impact. The subsystem shall also be capable of providing three axes control of the CM with the SM attached or the CM alone during all high altitude aborts.

3.2.6.1.2 Subsystem Description. - The RCS shall be pulse modulated, pressure fed, and utilize earth storable hypergolic propellant. Propellant tanks shall be positive expulsion type. The CM shall have two independent RCS systems. Each shall be capable of meeting the total torque and propellant storage requirements. Each subsystem shall consist of helium pressurization propellant storage, distribution, and thrust chamber subsystems. The two subsystems shall be capable of simultaneous or independent operation.

a. Oxidizer. - The oxidizer shall be nitrogen tetroxide (N_2O_4)

b. Fuel. - The fuel shall be monomethylhydrazine (MMH).

3.2.6.2 Service Module Reaction Control Subsystem (SM/RCS)

3.2.6.2.1 Subsystem Requirements. - Translational and roll control of the CSM during all unpowered phases from translunar injection to separation of the CM and SM prior to entry shall be supplied by a reaction control system mounted in the SM. The subsystem shall be capable of performing proportional attitude maneuver rates of 0.1 to 0.65 degrees per second and of supplying a minimum impulse bit of 0.4±0.2 lb/sec. The subsystem shall also be capable of providing the velocity increment required for the following maneuvers.

a. CSM Separation from Boost Vehicle. - The SM/RCS shall provide for separation from the boost vehicle prior to activation of the SPS for post atmospheric abort, translunar injection abort, and after injection into a translunar trajectory.

b. Minor Velocity Increments. - The SM/RCS shall provide for orbital corrections of less than 10 fps after CSM injection into lunar orbit, for reorientation of the CSM prior injection into a transearth trajectory during a translunar trajectory abort, and for separation of CM and SM prior to the entry mode.

c. Emergency Operation. - For emergency operations, the SM/RCS shall be capable of a continuous burn of 500 seconds.

3.2.6.2.2 Subsystem Description. - The RCS shall be pulse modulated, pressure fed and utilize earth storable hypergolic fuel. Fuel tanks shall be positive expulsion type. The subsystem shall consist of helium pressurization propellant storage distribution and thrust chamber subsystems.

a. Oxidizer. - The oxidizer shall be nitrogen tetroxide (N_2O_4).

b. Fuel. - The fuel shall be a mixture of fifty percent hydrazine (N_2H_4) and fifty percent unsymmetrical dimethylhydrazine (UDMH).

3.2.7 Launch Escape Subsystem (LES). - A LES shall be provided.

3.2.7.1 Subsystem Requirements. - The LES shall be capable of separating the CM from the LV in the event of failure or imminent failure of the LV on the launch pad and during all atmospheric phases.

3.2.7.1.1 Performance Criteria. - The CSM shall provide for crew escape from a critically malfunctioning LV from the time of crew insertion into the CM until successful completion of the second booster stage ignition. Crew accelerations incurred during LES abort and entry following abort shall not exceed the emergency limits of Figures 43 through 45.

a. Jettison Capability. - Primary propulsion, trajectory shaping and equipment jettison capability shall be provided.

b. Crew Escape. - The LES shall provide for crew escape from the LV under the following conditions:

(1) For escape prior to and shortly after lift off. The LES shall separate the CM from the LV and propel the CM to an adequate altitude to ensure safe ERS operation. The CM shall be propelled to sufficient range to minimize the effects of wind drift and the plane of the abort trajectory shall be fired nominally in a down range direction.

(2) The abort capability shall provide for critical control and guidance malfunctions which occur simultaneously with lift-off. A nominally performing subsystem shall provide recovery at, or above, ground level for the following booster malfunction conditions and associated parameters for abort initiation.

	Average Booster Divergence Rate Deg/Sec	Attitude Divergence at Abort Initiation Deg.
Condition I	±5.0	±11
Condition II	±2.5	±15

(3) The LES performance perturbations shall be considered to assure a high probability of safe CM recovery following booster malfunctions at lift-off.

(4) During a first 40 seconds following lift-off range safety considerations require escape from a thrusting booster.

(5) After 40 seconds the booster thrust will be terminated automatically at abort initiation.

(6) A minimum separation rate at maximum dynamic pressure shall be characterized by the requirements for attaining a CM separation distance of 350 feet in 3 seconds following CM separation from the SM. A minimum "miss distance" of 800 feet shall be provided for the case of abort at zero degrees angle of attack and stable flight of the booster. The term "miss distance" is defined as the distance between the LES vehicle and the booster/SM components at a time when the LES vehicle crosses a plane which contains the booster and is perpendicular to the booster flight path.

(7) The LES shall be capable of performing its function at the maximum dynamic pressure incurred during the boost (see Figure 46), with abort initiated prior to structural breakup of the LV configuration. A minimum capability shall be provided for abort at conditions described as follows:

Altitude Dynamic Pressure	= 40,000 feet = 750 psf
Condition I (Slow divergence failure)	Attitude rate at abort initiation $\dot{\theta}$ abort = 5 degrees per second, angle of attack and/or sideslip at abort initiation θ abort = ±15 degrees
Condition II (Hard over-gimbals)	Average pitch (or Yaw) acceleration prior to abort initiation ($\ddot{\theta}$ Avg) = 10 degrees per second^2), pitch (or yaw) rate at abort initiation ($\dot{\theta}$ abort) = ±5 degrees/second.

(8) The maximum altitude for LES abort shall be compatible with;

(a) Completion of second stage ignition and separation of jettisoned components

(b) Achieving a dynamic pressure condition permitting utilization of a SM abort. A minimum capability shall include the following parameters:

Altitude 320, 000 feet

Mach No. 8. 0

Dynamic Pressure 0. 5 to 1. 0 psf

Flight Path Angle 20 degrees

3.2.7.1.2 Normal Mission LES Jettison. - A jettison capability shall be provided to separate the LES from the boosters. A sufficient lateral separation distance shall be provided to assure a minimum "miss-distance" of 150 feet when jettison is initiated from a booster at zero angle of attack and nominal pitch rate.

3.2.7.2 Subsystem Description

3.2.7.2.1 Launch Escape Propulsion Section. A solid propellant propulsion section shall be provided to perform the following:

a. Separate the CM from the LV during atmospheric abort

b. Jettison the LES from the CM during normal mission or after atmospheric abort

3.2.7.2.2 Emergency Detection Subsystem (EDS). - Abort shall be initiated manually or automatically by the LV-EDS. Following abort initiation, LES functions shall be automatically controlled by the Automated Sequence Control Subsystem. Manual control of physical functions shall be provided to enhance reliability of the subsystem.

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3.2.8 Earth Recovery Subsystem (ERS). - The CM shall include an ERS to be used under all flight conditions for earth landing requirements.

3.2.8.1 Subsystem Requirements. - The subsystem shall satisfy the following requirements after normal entry, maximum dynamic pressure escape, and pad escape.

3.2.8.1.1 Postentry Stabilization. - Stabilize the CM during postentry descent.

3.2.8.1.2 Design Loads. - The ERS shall be designed for loads resulting from CM gross launch weight of 11,500 pounds with the factor of safety of 1.35.

3.2.8.1.3 Velocity Control. - Reduce the vertical touchdown velocity to not more than 33.5 feet per second at sea level.

3.2.8.1.4 Impact Attenuation. - Reduce impact acceleration such that the CM flotation is not impaired. Any further attenuation required to prevent exceeding the crew emergency acceleration limits delineated in Figures 43 through 45 shall be provided by crewman shock attenuation devices. For design purposes, the CM attitude at impact shall be limited to 30 degrees negative pitch angle.

3.2.8.1.5 Postlanding. - The ERS will provide as auxiliary equipment on the CM the following equipment:

a. CM pick-up sling.

b. Automatic actuated sea dye marker.

c. Flashing beacon light.

d. Exterior recovery party umbilical connection.

3.2.8.1.6 Initiation and Control. - Initiation of all functions shall be capable of being manually controlled except the sea dye marker.

3.2.8.2 Subsystem Description. - The ERS shall consist of 2 FIST type drogue chutes deployed by mortar and a cluster of three simultaneously deployed landing parachutes. Landing parachutes shall be sized such that satisfactory operation of any two of the three will satisfy the vertical velocity requirement.

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3.2.9 Crew Subsystem

3.2.9.1 Subsystem Requirements. - Design and operational procedures shall be in accordance with the crew requirements presented here.

3.2.9.1.1 Crew Size and Number. - The CSM design parameters shall accommodate 3 crew members between the 10th and 90th percentile, as defined in WADC-TR 52-321, Anthropometry of Flying Personnel, for the following dimensions: weight, standing height, sitting height - erect, buttock-to-knee length, knee height (sitting), hip breadth (sitting), shoulder breadth (bideltoid), and arm reach from wall. All other body dimensions shall fall within the 5th and 95th percentiles as defined by WADC-TR 52-321. Percentiles for body dimensions undefined by applicable documents will be estimated. For design purposes, body dimensions will be estimated by appropriate statistical and anthropometric methods.

3.2.9.1.2 Division of Duties. - Crew duty requirements shall be based on cross-training such that each crew member is able to perform tasks performed by other crew members.

3.2.9.1.3 Metabolic Requirements. - The average daily metabolic requirements for each crew man are assumed to be as shown in Table II.

3.2.9.1.4 Environmental Requirements. - The CM interior environment shall be as specified in paragraph 3.2.10.

a. Vision. - Crew vision shall be such that CM control is not adversely affected.

3.2.9.1.5 Decompression Protection. - Pressurized garments (GFE) shall provide protection for crew members in the event of crew compartment decompression. Two crew members shall be capable of donning Pressure Garment Assemblies in five minutes or less. At least one crew member shall wear the Pressure Garment Assembly at all times.

3.2.9.1.6 Food and Water

a. Food. - Provisions for storage of food and associated equipment (GFE) shall be provided within the CM sufficient for a 10.6 day mission.

b. Water. - In addition to the primary source of potable water, a backup supply shall be provided in the survival kit (GFE). Chemicals capable of desalting sea water shall be made available. One pint of fresh water shall be obtained from each 0.16 pounds of chemical.

3.2.9.1.7 Human Waste Control. - Provisions shall be provided for the removal and disposition of gaseous, solid (fecal), liquid human waste within the CM. A manually operated valve shall be provided for periodically venting liquid waste overboard.

3.2.9.1.8 Crew Compartment Lighting

a. A portable light shall be provided which may;

(1) Be used by crewmen during flight for operation of CM subsystems.

(2) Provide interior illumination during the post landing phase.

(3) Be used as a ground signal light with the capability of being seen by recovery forces at a distance of 13 miles in an atmosphere with a transmittance of 70 percent.

b. A window filter assembly capable of attenuating visible solar radiation from 89 percent transmittance to 20 percent. The filter assembly shall also reflect 90 percent of the solar heat.

3.2.9.2 Subsystem Description

3.2.9.2.1 Crew Equipment. - Provisions for the crew equipment are delineated in SID 62-1003, Preliminary NASA Furnished Crew Equipment interface and performance specification.

3.2.9.2.2 Couches. - Couches shall be designed to provide comfortable support during all mission phases. The center crew couch shall be foldable, to the extent required, to provide necessary work space and adequate access by the crew to all regions of the CM as required.

3.2.9.2.3 Restraint Subsystem. - A subsystem of restraints shall be provided for crew support and restraint during normal and emergency mission conditions. The subsystem shall prevent the crew from exceeding the limits delineated in Figures 43 through 45.

3.2.9.2.4 Crew Accessories. All crew accessories shall be provided to assist the crewmen in the performance of tasks under anticipated mission conditions and activities. SID 62-1003, Preliminary NASA Furnished Crew Equipment Interface Requirements specifies the NASA furnished crew equipment.

3.2.9.2.5 Window Filter Assemblies. - A subsystem of filter assemblies shall be provided and installed to attenuate thermal and solar energy entering the CM through the windows as required during the course of the mission.

3.2.9.2.6 Survival Kit Storage. - Provision shall be made for storage and accessibility of 3 sets of Survival Provisions (GFE) in the CM.

3.2.9.2.7 Crew Equipment and Suit Interface. - A subsystem of umbilicals and connectors shall provide electrical power and oxygen circulation for the Pressure Garment Assembly. This equipment shall also include tethering provisions for tools used under weightless conditions.

3.2.9.2.8 Medical Kit Storage. - Provision shall be made for storage of one GFE Medical Kit in the CM.

3.2.9.2.9 Personal Hygiene. - Personal hygiene equipment shall be provided to enable crewmen to perform necessary bodily cleansing during the mission.

3.2.9.3 NASA Furnished Crew Equipment

3.2.9.3.1 Survival Provisions. - NASA-furnished survival provisions shall include:

a. Signal mirror - survival, crewman
b. Sunglasses - survival, crewman
c. Water and container - survival, crewman
d. First aid kit - survival, crewman
e. Machete - survival, crewman
f. Desalting kit - survival, crewman
g. Transceiver - survival, crewman
h. Life vest - survival, crewman
i. Balloon kite - survival, crewman
j. Life raft - one-man survival, crewman
k. Light assembly - survival, crewman
l. Flare set - survival, crewman
m. Sea dye marker
n. Whistle - survival
o. Nylon lanyard
p. Carborundum stone
q. Cottonballs and striker
r. Compass - survival
s. Sewing kit and needles
t. Fishing kit - survival

The NASA-furnished survival provisions shall be packaged in the contractor-furnished survival provisions assembly for storage in the CM.

3.2.9.3.2 Personal Equipment. - The NASA-furnished personal equipment shall include:

a. Pressure Garment Assembly
b. Constant Wear Garment (CWG)
c. GFE Checkout Equipment, as required

3.2.9.3.3 Medical Equipment. - The NASA-furnished medical equipment shall include the following items:

a. Radiation Dosimeter (Set)
b. Dressings - Emergency Medical Kit (Set)
c. Medications - emergency medical kit (Set)
d. Instrument set - clinical monitoring, physiological
e. Instrument assembly - biomedical preamplifier
f. Instrument assembly - biomedical sensors, personal

The NASA-furnished medical equipment shall be stored in the contractor-furnished medical compartment for storage in the CM.

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3.2.9.3.4 Food and Associated Equipment. - The NASA-furnished food and associated equipment shall consist of the following items:

a. Food
b. Mouthpiece, food, personal
c. Probe, water delivery

The NASA-furnished food shall be stored in the contractor-furnished food compartment assembly for storage in the CM.

3.2.10 Environmental Control Subsystem (ECS). - The CSM shall include an ECS which provides a conditioned, "shirtsleeve" atmosphere for the crew; provisions for space suits in event of cabin decompression; thermal control of all CSM equipment where needed; and provisions only for charging the Portable Life Support System (PLSS).

3.2.10.1 Subsystem Requirements

3.2.10.1.1 Cabin Pressure. - The cabin pressure nominal operating limits shall be 5 psia ±0.2. The subsystem shall be capable of maintaining a cabin oxygen partial pressure of at least 3.5 psia for at least 5 minutes following a single one-half inch diameter puncture in the pressure compartment. The emergency unit circuit shall be 3.5 psia minimum.

3.2.10.1.2 Oxygen Partial Pressure. - The oxygen partial pressure nominal limits shall be 233 millimeters of mercury (mm) minimum, and emergency limits shall be 16mm.

3.2.10.1.3 Carbon Dioxide Partial Pressure. - The CO_2 partial pressure nominal limit shall be 7.6mm mercury maximum. In an emergency the limits shall not exceed that given in Figure 47. In the post-landing phase the following values shall be used:

a. CO_2 Production. - Metabolic rate x 0.0045 pounds per man per day. See Table II

b. CO_2 Concentration Allowable. - 16 mm Hg Avg.

3.2.10.1.4 Sweat Rate. - The sweat rate shall be as specified in NASA letter EC6-64-351, Environmental Limits for Apollo Post Landing Phase, dated 17 June 1964.

3.2.10.1.5 Temperature Limits

a. CM Temperature. - The cabin air temperature non-stressed limits shall be 70 degrees F minimum and 80 degrees F maximum. The nominal and emergency limits shall be as presented in Figures 48 and 49 respectively.

b. SM Temperature. - The SM temperatures shall be maintained within safe limits for equipment installed.

3.2.10.1.6 Cabin Relative Humidity. - The cabin relative humidity non-stressed limits shall be 40 percent minimum and 70 percent maximum. The nominal and emergency limits shall be as presented in Figures 48 and 49 respectively.

3.2.10.2 Subsystem Description. - Environmental control shall be accomplished with two air loops, a gas supply section, a thermal control section, and a water management section.

3.2.10.2.1 Regenerative Circuit Loop. - This loop supplies the conditioned atmosphere to the cabin and space suit and shall provide removal of debris and noxious gases and for carbon dioxide absorption. Ventilation flow at 3.5 psia shall be 10 cfm through each space suit with a maximum flow resistance in each space suit of 5 inches of water.

3.2.10.2.2 Cabin Loop. - The loop shall serve to provide cabin ventilation and thermal control during all phases of the mission. Postlanding ventilation shall be accomplished by 5 inch diameter inlet and outlet openings with valves, a fan and directional ducting. For purposes of analyses the following special environmental conditions shall be used:

a. Temperature

Exposure Time *(hr)	Max Air Temp (F)
0 - 2.5	84.4
2.5 - 8	84.4 linear increase to 86.5
8 - 12	86.5

12 - 17	86.5 linear decrease to 84.4
17 - 24	84.4
Repeat	

b. Humidity — 82 percent relative humidity for 48 hours

c. Sunshine

Exposure Time *(hr)	Solar Radiation (Btu/ft^2/hr)
0 - 5	0 linear increase to 306
5 - 9	306
9 - 13	306 linear decrease to 0
13 - 24	0

*Starting at Sunrise.

d. Sea State

Wind Velocity	3 to 28.5 knots
Wave Height (crest to trough)	0.5 to 8.5 feet

3.2.10.2.3 Gas Supply Section. - The primary gas supplies shall be stored as super critical cryogenics in the SM in the same tank as for the EPS. Entry oxygen shall be supplied from a high pressure bottle of gaseous oxygen.

3.2.10.2.4 Temperature Control

a. CM Thermal Control. - Dissipation of the internal thermal load of the CM shall be accomplished by absorbing heat with a circulating coolant and rejecting this heat from a space radiator during certain mission modes. Other cooling subsystems shall supplement or relieve the primary subsystem.

b. SM Thermal Control. - Thermal stabilization of the SM/RCS shall be accomplished by absorbing and dissipating heat with a circulating coolant by the use of a heat sink, electric heater and pump assembly independent of CM thermal control.

3.2.10.2.5 Water Management. - Water shall be collected from the separator and the fuel cell and stored in positive expulsion tanks. The water collected from the fuel cell shall be stored separately and used as the primary source of potable water.

3.2.10.2.6 Safety Features. - All relief valves and other valves which connect the internal pressure vessel to the space environment shall have manual closures and overrides. Filters shall be provided to protect all regulators, control valves, gas analyzers, etc. Relief valves shall be provided to prevent overpressurization of low pressure components. Flow limiting devices shall be provided to prevent excessive use of gas supplies and subsequent depletion of such supplies.

3.2.11 Electrical Power Subsystem (EPS)

3.2.11.1 Subsystem Requirements. - The EPS shall be designed to store energy, generate, supply, regulate, condition, and distribute all electrical power required by the CSM for the full duration of the mission, including the post-landing recovery, but excluding the Portable Life Support Subsystem (PLSS) batteries.

3.2.11.1.1 Power Output. - The EPS shall be capable of generating 575 kwh of electrical energy from fuel cells at a minimum rate of 563 watts and a maximum rate of 1420 watts per cell. In addition 725 watt hours from storage batteries shall be available.

3.2.11.1.2 DC Bus Voltage. - Electrical power shall be generated and distributed at 28 vdc (nominal).

3.2.11.1.3 AC System Voltage. - The ac system shall supply 115/200 volts at 400 cps and shall be three phase type connected.

3.2.11.1.4 AC Ripple. - All dc buses in the subsystem shall be maintained essentially free of ac ripple (as defined in MIL-STD-704, paragraph 31.2) to within a limit of 250 millivolts peak to peak.

3.2.11.1.5 Regulation. - Steady state voltage limits shall be 27.5 volts ±2.5 volts. Transient load voltage limits shall be 21 to 32 volts with recovery time to steady-state limits within one second.

3.2.11.1.6 Load Grouping. - All electrical loads supplied by the distribution system shall be classified as essential, non-essential, pyrotechnic, or recovery. Essential loads are defined as those loads (except pyrotechnic circuits) that are mandatory for safe return of the CSM to earth from any point in the lunar mission. Loads not necessary for the safe return of the CSM shall be grouped on a non-essential 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 a recovery bus and provision made for manually disconnecting the essential bus following landing. Redundant buses shall be provided for pyrotechnic circuits and used to supply only that type load.

3.2.11.2 Subsystem Description

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

(a) Energy Sources

Cryogenic Gas Storage Subsystem
Storage Batteries

(b) Power Generation Equipment

Fuel Cell Subsystem

(c) Power Conversion Equipment

Inverters
Battery Chargers

(d) Power Distribution Equipment

Power Buses, a-c and d-c
Associated Controls

3.2.11.2.2 Location. - The location of each of the above components within the CSM shall be as listed herein. Every effort shall be exercised to minimize equipment size and weight, commensurate with the established requirements and obtaining the highest practicable reliability.

	Location
Fuel cell module and controls	SM
Tank (empty), radiators heat exchangers, piping, valves	SM
Total reactants, plus reserves	SM
Auxiliary batteries	CM
Pyrotechnic batteries	CM
Separation sequencer batteries	SM
Batteries charger	CM
Static inverters	CM
EPS display and control panel	CM

3.2.12 Communication Subsystem

3.2.12.1 Subsystem Requirements. - The CSM shall be capable of transceiving communications between the CSM and Earth at lunar distances. The communication equipment shall be compatible with the equipments used by the MSFN as defined in SID 63-881, NASA MSFN Performance and Interface Specification-Primary.

3.2.12.1.1 Voice Transceiving. - Voice communication capability between the CSM and Earth shall be provided except when the moon blocks the line of sight bearing between the CSM and Earth. At an AM transmission at 2200-2400 mc and reception at 2000 to 2200 mc a capability of transceiving shall also be provided for a HF, 8-16 mc band, and a VHF, FM transmission at 237.8 mc.

3.2.12.1.2 Personnel Communications. - Two-way communication between crew members shall be provided.

3.2.12.1.3 Telemetry. - A capability of telemetering 51,200 bits per second with a non-return to zero format shall be provided with an accuracy of 0.5 percent. A minimum data rate mode shall be provided at a bit rate of 1600 bits per second. This capability shall exist from the time of lift-off until touchdown. Additional requirements include:

a. A digital up-data link operating at 450 mc shall be provided.

(b) Simultaneous operation of functions shall be limited to those noted in SID 64-1237 Vehicle Model Specification - Basic, Block I.

3.2.12.1.4 Television. - The CSM shall contain a capability of transmitting to earth high resolution pictures of near commercial quality in real time and also of monitoring the internal and external scenes in real time. It shall provide an analog output to 320 lines per frame at a rate of 10 frames per second for transmission on the 2200-2400 mc band (S-Band).

3.2.12.1.5 Tracking Transponders. - A subsystem capable of providing reliable tracking signals in the near-Earth-phase of flight and of providing velocity and range tracking to the lunar distance shall be supplied. Near Earth tracking signals ($<$ 8000 nm) shall be in the 5640-5815 mc band (C-Band) and in the 2000 to 2200 mc band for receiving and in the 2200 to 2400 mc band (S-Band) for receiving to the lunar distance.

3.2.12.1.6 Recovery Aids. - After landing, the CM shall have the capability of actively aiding in the recovery operations through voice transmission and reception to a range of 87 nm through a beacon operating at 243 mc (VHF) to aid search and recovery aircraft.

3.2.12.2 Subsystem Description. - The major components of the communications subsystems shall include the following items:

(a) VHF/FM transmitter equipment
(b) VHF/AM transmitter-receiver equipment
(c) Unified S-band equipment
(d) S-band power amplifier equipment
(e) C-band transponder equipment
(f) VHF recovery beacon equipment
(g) HF transceiver equipment

(h) Audio center equipment
(i) Television equipment
(j) PCM telemetry equipment
(k) Premodulation processor equipment
(l) Digital up-data link equipment
(m) Signal conditioning equipment
(n) Central timing equipment
(o) Data storage equipment
(p) Beacon antenna equipment
(q) VHF/2-KMC omni antenna equipment
(r) Recovery antenna equipment
(s) Ancillary equipment
(t) 2-KMC high gain antenna equipment

The subsystem functions and detailed equipment descriptions are contained in SID 64-1237, Vehicle Model Specification, Basic - Block I.

3.2.13 Instrumentation Subsystem.

3.2.13.1 Subsystem Requirements. - Provisions shall be made on board the CSM System to make, condition, telemeter, and record the following measurements.

a) Pressures
b) Temperatures
c) Flow Rates
d) Accelerations
e) Quantities
f) Angular positions
g) Currents
h) Attitudes

3.2.13.2 Subsystem Description. - The operational instrumentation will be physically and functionally compatible with the other subsystems noted in SID 64-1237, Vehicle Model Specification - Basic, Block I. The R&D instrumentation shall be as noted in SID 62-1001, Flight R&D Instrumentation Performance and Interface Specification. Space provisions only for scientific instrumentation will be provided. The space allocation shall consist of 2.7 cubic feet as delineated in SID 64-1237, Vehicle Model Specification - Basic, Block I.

3.2.14 Displays and Controls (D&C)

3.2.14.1 Subsystem Requirements. - Sufficient depth of information and command access to the CSM subsystems shall be provided to enable the 3 man crew to accomplish the following operations:

(a) Effect manual CSM system management

(b) Safe shutdown of CSM equipment

(c) Select alternate subsystem operating modes

(d) Recognize hazard to crew CSM, launch vehicle or mission and effect mission change if normal system operation cannot be restored.

3.2.14.2 Subsystem Description. - The Display and Control subsystem shall present information to and accommodate control action inputs from the CSM flight crew during the mission as described in section 3.1.4.

The Primary location of the D&C equipment shall be the Main Display Console (MDC), which is located above the crew couches in the CM. Secondary locations of the D&C equipment shall include the right-hand and left-hand side display consoles. Other locations of the D&C equipment shall include the left-hand forward equipment bay, right-hand forward equipment bay and the Navigation Station at the lower equipment bay.

The operational D&C equipment shall be oriented as to establish a Displays and Controls subsystem. The Master Caution and Warning Subsystem and Crew Compartment floodlighting equipment are part of the D&C equipment and shall be provided as defined in SID 64-1237, Vehicle Model Specification, Basic - Block I.

3.2.15 Entry Monitor Subsystem

3.2.15.1 Subsystem Requirements. - The CSM crew shall monitor CM "g" forces, change in velocity, and roll attitude about the stability axis during the entry phase of the mission. The presentation of these parameters shall allow the crew to ascertain that the entry flight path remains within certain critical limits. The CSM crew shall also be provided with adequate information to permit manual override and control at any time during the entry phase, should such action become necessary.

3.2.15.2 Subsystem Description. - The EMS shall contain its own sensors for "g" forces and roll attitude and shall operate independently of all other systems except for primary power.

The EMS shall consist of two sensors (a gyro and an accelerometer) and the following displays:

(a) Threshold indicator lamp

(b) Corridor indicator lamp

(c) Flight Monitor

3.2.16 Automated Sequence Control Subsystem

3.2.16.1 Subsystem Requirements. - Automated sequencing shall be employed to control those functions and events which require greater precision or speed of response than the crew can provide or to relieve the crew of tedious tasks. An interlock between the sequencer subsystem and the GSE shall be provided to prevent inadvertent abort during the pre-launch phase.

3.2.16.2 Subsystem Description. - The automated sequencing subsystem description is delineated in detail in SID 64-1237, Vehicle Model Specification - Basic, Block I. This subsystem shall be capable of performing the proper sequencing of events during ascent, entry, LES Abort, Adapter separation and SPS abort, initiating functions and providing monitor capabilities. An interlock between the sequencer subsystem and the GSE shall be provided to prevent inadvertant abort during the prelaunch phase.

3.2.17 Pyrotechnic Subsystem and Devices

3.2.17.1 Subsystem Requirements. - All Pyrotechnic Subsystem and devices including their associated electrical circuitry shall provide for redundant design through the detonators. No redundancy shall be provided for the explosive charges except in the case of the CM/LES separation devices, the ELS drogue and pilot parachute deployment devices.

3.2.17.1.1 Standard Electro-Explosive Device (EED). - All electrically-actuated pyrotechnic devices shall be fired by the Apollo Standard Initiator (ASI).

3.2.17.2 Standard Detonator Cartridge. - A standard detonator shall be used to initiate all high explosive charges. The detonator shall consist of the ASI hermetically sealed into a cartridge containing a charge which produces a high, order detonation. This cartridge assembly shall be designated as the Apollo Standard Detonator (ASD).

3.2.17.3 Electrical Power Sources. - The firing and logic electrical power sources for pyrotechnic subsystems shall be provided by batteries which are separate and independent from all other CSM power sources and from each other.

3.2.17.4 Firing Circuit Tests. - Provisions shall be made for electrical continuity checkout of all firing circuits after mating of the last electrical connector in the circuit. Special test equipment shall be used for this continuity test to preclude dangerous levels of voltage being inadvertently applied.

3.2.18 Service Module Propellant Dispersal Subsystem. - This pyrotechnic function shall be operable upon receipt of the RF arm and fire signals initiated by the ground Range Safety Officer. The subsystem shall be operable only during that portion of flight prior to LES Tower jettison. The explosive charges shall open the SM main propellant tanks to provide atmospheric dispersal of propellants. Space provisions only shall be provided for explosive charges and devices to disperse contents of the LEM descent stage propellant tanks.

3.3 Reliability Requirements

3.3.1 Mission Success Reliability. - The mission success reliability objective for Apollo shall be 0.90 for a LOR mission followed by the return to earth of the CSM without exceeding the emergency crew limits, given in the design criteria.

3.3.2 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.

3.3.3 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
MSFN	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

The above apportionment considers the use of In-Flight Maintenance and on-board spares. Elimination of these items requires a reapportionment of reliability objectives and a redefinition of reliability goals.

3.4 Electromagnetic Compatibility. - Each assembly shall be electromagnetically compatible with other assemblies in the system, other equipment in or near the LV, associated test and checkout equipment, and to the electro-magnetic radiation of the operational environment. The subsystem shall not be a source of interference that could adversely affect the operation of other equipments or compromise its own operational capabilities. The system shall not be adversely affected by fields or voltages reaching it from external sources, such as improperly suppressed vehicle test and checkout equipment, and nearby radio frequency sources in the operational environment.

3.4.1 Command and Service Module and GSE Equipments. - MIL-I-26600 (USAF), Interference Control Requirements, Aeronautical Equipment, dated 2 June 1958, and MSC-EMI-10A, Addendum to MIL-I-26600, dated 17 October 1963 shall be used as a guide.

3.4.2 Command and Service Module and GSE Subsystems. - Shall be designed in accordance with MIL-E-6051C, Electrical-Electronic System Compatibility and Interference Control Requirements for Aeronautical Weapon System, Associated Subsystem, and Aircraft, dated 17 June 1960.

3.5 Interchangeability. - Mechanical and electrical interchangeability shall exist between like assemblies, subassemblies, and replaceable parts of operating subsystems (electronic, electrical, etc) regardless of the manufacturer or supplier. Non-operating subsystems such as structure need not comply with this requirement. Interchangeability for the purpose of this paragraph does not mean identity, but requires that a substitution of such like assemblies, subassemblies, and replaceable parts be easily effected without physical or electrical modifications to any part of the equipment or assemblies, including cabling, connectors, wiring, and mounting, and without resorting to selection; however, adjustment of variable resistors and trimmer capacitors may be made. In the design of the equipment, provisions shall be made for design tolerances sufficient to accommodate various sizes and characteristics of any one type of article, such as tubes, resistors, and other components having the limiting dimensions and characteristics set forth in the specification for the particular component involved without departure from the specified performance. Where matched pairs are required, they shall be interchangeable and identified as a matched pair or set.

3.5.1 Identification and Traceability. - Apollo identification and traceability shall be in accordance with MSC-ASPO-1-4.

3.6 Ground Support Equipment (GSE). - GSE is defined as the non-flight implements or devices required to checkout, handle, service, or otherwise perform a function in support of the CSM or boilerplate during tests at factory subsequent to manufacturing complete, prelaunch, launch, and post launch operations at the test site, and major development tests such as house CSM tests, propulsion tests and environmental tests.

3.6.1 Support Requirements

3.6.1.1 Operations Supported. - CSM GSE shall support the vehicles during: (1) acceptance, (2) test preparation, (3) test, (4) checkout, and (5) prelaunch checkout. It shall also include such recovery and post launch test items as may be agreed to by the parties.

3.6.1.2 Design Concept. - The GSE design concept delineates four general categories of equipment for supporting servicing, handling, system C/O and testing, and various auxiliary requirements. The equipment design shall be pointed towards remote control utilizing a digital interface with computer analysis and control as well as a direct interface for local/manual control. To as great an extent as practical, similar equipment shall be used to ensure continuity in checkout. Design shall be based on use by skilled technicians.

3.6.1.3 Command and Service Module Checkout Concept. - CM checkout shall consist of:

(a) Local/manual operation

(b) Remote/semi-automatic operation as provided by ACE

(c) Remote/manual operation

The local operation shall be performed with the Acceptance Checkout Equipment - CSM (ACE/CSM) provided by NASA. This remote/automatic operation shall be based on a computer controlled cathode ray tube display system. Conventional test equipment techniques shall incorporate flexibility to easily accommodate frequent changes.

NAA shall design all carry-on equipment for ACE/CSM usage as follows:

(a) All flight hardware equipment based on checkout for the CSM.

(b) All equipment based on checkout of the G&N subsystem.

(c) All equipment required between the CSM and the transmission line interface.

(d) All equipment required for hazardous operations designed for remote/manual operation from protected areas. This equipment shall be designed for specific tests.

3.6.1.4 System Checkout Concept. - All checkout operations performed on systems and subsystems installed in the vehicle shall be performed by checkout equipment having remote manual or automatic capability for malfunction detection and isolation to a replaceable package. Operations performed on systems and subsystems not installed in the vehicle will be accomplished by Bench Test Equipment (BTE) in Downey or Florida. BTE shall be limited to local manual operation.

3.6.1.5 Maintenance Concept. - In-field operations maintenance shall be accomplished by removal and replacement at the "black box" level. System failures and/or malfunctions shall be corrected by removal of the isolated replaceable package to a BTE area for malfunction verification. Electronic packages will then be returned to the supplier. Mechanical components will be repaired by unit replacement, reverified, and made ready for reinstallation. The malfunctioned unit will be returned to supplier. BTE shall also be used for spares certification before installation.

3.6.2 Level of Support. - The level to which CSM GSE shall support the vehicles, operations, and sites specified in Paragraph 3.6.1 is as follows:

3.6.2.1 Test Preparation and Acceptance Area. - Equipment shall be provided in the test preparation and acceptance area to functionally checkout spacecraft subsystems and verify compliance of operational and performance parameters with design requirements. Installation checkout, subsystem functional tests, and integrated systems tests shall be performed. Substitute units shall be provided when required to simulate modules or elements of the system which are not present. Extensive checkout of fuel and cryogenic systems and associated servicing equipment will not be conducted in this area.

3.6.2.2 House Spacecraft. - GSE for House Spacecraft shall perform subsystems and combined subsystems tests for the following purposes:

(a) Engineering development

(b) Field operations checkout (ACE programming and operations)

The remote manual and semi-automatic checkout modes shall be applied to the House CSM operations to develop checkout techniques and operating procedures as ACE capabilities are developed. Servicing, handling and auxiliary equipment shall be provided as required.

3.6.2.3 Prequalification Flight Drop Test Site. - The GSE provided to support these operations shall consist of handling equipment, and limited auxiliary equipment.

3.6.2.4 White Sands Missile Range. - Support for the WSMR abort tests shall be in a local/manual mode and remote/manual mode with checkout equipment having capability for centralized gross system and diagnostic check with checkout equipment having capability as specified in paragraph 3.6.1.5. The support at WSMR shall include control and monitoring during

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pre-launch operations and countdown. The R&D instrumentation checkout equipment and associated checkout equipment shall be installed in a mobile van in place of the Pad Transfer Room. The R&D instrumentation checkout equipment shall be modified as necessary to ensure suitability for its intended end purpose. Auxiliary, handling, and servicing equipment shall be provided as required with checkout equipment specified in paragraph 3.6.13.

3.6.2.5 <u>Las Cruces Propulsion System Development Facility.</u> - Support of test preparation and firing preparation shall be in the local/manual mode with equipment having diagnostic capability as specified in paragraph 3.6.1.5. Servicing equipment required to furnish fluids, propellants, pneumatic pressures for the propulsion, reaction control, and other fluid subsystems shall also be provided with local/manual control capabilities. Engine firing control equipment capable of controlling and monitoring firings in a remote manual mode shall be provided. Handling and auxiliary equipment shall be provided as required.

3.6.2.6 <u>Environmental Test Facility.</u> - Support checkout of the CSM shall be accomplished with ACE-CSM equipment prior to the thermal vacuum test. Handling and auxiliary equipment shall be provided as required. Test operations will be conducted with remote/manual equipment.

3.6.2.7 <u>Atlantic Missile Range.</u> - Equipment shall be provided for the complete functional checkout of the spacecraft and verification of readiness for flight. Equipment at the launch complex shall provide for servicing and preparation of the space vehicle and monitoring and control of the launch operation. Special facilities and equipment shall be provided for static firing of the SPS and RCS systems, and operation and verification of the fuel cell, cryogenic and the environmental control systems. Individual subsystem and integrated systems tests shall be conducted in the operations and control building. Tests and checkout in the operations and control building and at the launch complex shall be designed for use of ACE equipment.

3.6.2.8 <u>Arnold Engineering Development Center (AEDC).</u> - Service Propulsion System tests will be conducted using GSE and STE furnished by the engine subcontractors as well as standard bench type test equipment.

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3.6.2.9 Marshall Space Flight Center (MSFC). - Dynamic & Umbilical Tests will be conducted with bench type test equipment.

3.7 Personnel Training. - A program plan shall be provided for training the flight crew and ground operations personnel in the skills and knowledge required for operation of the Apollo system. The contractor shall support the program with the following categories of trainers.

3.7.1 Subsystems Trainers

3.7.1.1 Trainer Concept. - The CSM Subsystems Trainers, as far as practical, shall provide for the simulation of specified CSN subsystems with sufficient realism for the training of personnel in the functional relationships of CSM components, subsystems, and the procedures of flight subsystems management. The Systems Trainers shall be mobile unit devices operating in a controlled environment. The design configuration will be based on the controls and displays of the first manned earth orbital spacecraft. The trainers shall be capable of accepting selected modes and problems inserted by an instructor.

3.7.1.2 Trainer Items. - The Subsystems Trainers shall consist of the following major items:

(a) Stabilization and Control Subsystem Trainers
(b) Communication Subsystem Trainers
(c) Electrical Power Subsystem Trainers
(d) Environmental Control Subsystem Trainers
(e) Propulsion Subsystems Trainer

3.7.2 Apollo Part Task Trainer (APTT)

3.7.2.1 Trainer Concept. - The APTT shall be a fixed base device operating in a controlled environment and capable of providing training in selected tasks associated with mission segments as follows: launch countdown, launch, earth orbit, translunar, lunar orbit, transearth and entry. The design configuration of the APTT shall be based on the controls and displays of the first manned Earth orbital CSM equipment to simulate mission requirements. Capabilities will be provided in the performance characteristics of the trainer equipment to simulate mission segments of the CSM subsequent to the first manned earth orbital flight including the lunar mission CSM. The APTT shall provide crew training in normal and alternate flight procedures. Malfunctions shall be inserted in the training tasks to require the flight crew to utilize these alternate procedures. Initially at least one malfunction shall be available for each alternate procedure. Additional malfunctions shall be provided on the basis of subsequent analysis to provide a library of malfunctions related to crew actions and alternatives. Malfunctions for which no crew alternative exists shall not be employed.

3.7.2.2 Trainer Items. - The APTT shall consist of the following major equipment groups:

- Simulated CM
- Instructor's Console for Three Instructor Positions
- Computer Complex
  - Digital Computer and Peripheral Equipment
  - Analog Computers and Peripheral Equipment
  - Input - Output Control and Buffer Unit Equipment
- Simulated Command and Service Module Subsystems Equipment
- Visual Simulation for Telescope and Sextant
- Aural Simulation Equipment
- Recording Equipment

3.7.3 Apollo Mission Training Simulators (AMTS):

3.7.3.1 Trainer Concept. - The AMTS shall be a fixed base device operating in a controlled environment and capable of providing training in all tasks associated with continuous mission phases as follows:

(a) Launch countdown, launch, earth orbit, translunar, lunar orbit, lunar rendezvous.

(b) The phases of the mission shall be presented in a continuous fashion without apparent re-programming or switching transients.

(c) The AMTS shall be capable of providing integrated training and operation with the Integrated Mission Control Center (IMCC) and Simulated Checkout and Training System (SCATS).

(d) The crew and ground operations functions shall be derived from training requirements developed from flight crew tasks analyses.

(e) The design configuration of the AMTS shall be based on the controls and displays of the first manned Earth orbital CSM and related AMTS training requirements.

(f) The AMTS shall provide crew training in normal flight procedures and alternate flight procedures. Malfunctions will be inserted in the training tasks to require the flight crew to utilize alternate procedures. Initially, at least one malfunction shall be provided on the basis of subsequent analysis to provide a library of malfunctions related to crew actions and alternatives. Malfunctions for which no crew alternative exists shall not be employed.

3.8 Manned Space Flight Control Center (MSCC) and Manned Space Flight Net (MSFN). - The design configuration of the MSCC and the MSFN must conform to the interface requirements delineated in SID 63-881, NASA Manned Space Flight Net, P&I Specification. This section describes the operation concept for the MSCC and the computational facilities, gives a general description of the initial configuration of the MSFN, and a general outline of the ultimate configuration as currently visualized.

3.9 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.

3.9.1 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.

3.9.2 Choice of Standard Materials, Parts, and Processes. - Where applicable, preferred parts lists shall be used. When an applicable specification provides more than one grade, characteristics, 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.

MSFC-PROC-158A, 12 April 1964, Soldering electrical connectors (high reliability). - Procedure for, as amended by MSC-ASPO 513, 10 February 1960, delineates soldering requirements.

3.9.3 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.

3.9.3.1 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.

3.9.4 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.

3.9.5 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.

3.9.6 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.

3.9.7 Unstable Materials. - Materials which emit or deposit corrosive substances, induce corrosion, or produce electrical leakage paths within an assembly shall be avoided or protective measures incorporated.

3.9.8 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 CSM. When used, fungus-nutrient materials shall be hermetically sealed or treated for fungus and shall not adversely affect equipment performance or service life.

3.9.9 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.

3.9.9.1 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.

3.9.9.2 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.

3.9.10 Lubricants. - The CSM 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.

3.9.11 Special Tools. - The functional components of the CSM 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.

3.9.12 Explosion Proofing. - The entire CSM, including electronic subsystems and rocket motor ignitors, shall be designed to minimize the existence of fire hazards or explosive environment. The subsystems 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 compartmented 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 10M 01071.

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

3.9.14 Connectors. - Wherever practical, all electrical and mechanical connectors (except R&D instrumentation) shall be so designed as to preclude the possibility of incorrect connection.

3.9.15 Ground Support Equipment. - Commercial standards for materials and equipment shall be utilized to the maximum extent possible where such use will not compromise the safety of operations or the meeting of the necessary performance requirements.

3.9.16 Nameplates and Product Markings. - The CSM 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 NPC 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. Documentation requirements shall be as noted in Exhibit I to the Apollo Contract, NAS9-150.

4.1.1 Quality Control. - NAA/S&ID shall establish a quality control plan.

4.2 Reliability. - NAA/S&ID shall establish a reliability program.

4.3 Test. - NAA/S&ID shall establish a qualification test program to determine that the CSM system satisfies the requirements of Section 3 of this specification. The ground rules for establishing this program are as follows.

Test programs for each test phase are to consider the overall mission environmental and operational conditions to be encountered for each grouping of flight vehicles. After these are defined, specific requirements for each vehicle, and in turn each subsystem, are to be established. As a result of this approach, there is to be a definitive test program tailored for each item to be used on an Apollo flight vehicle dependent upon its specific application and previous usage history.

For those subsystems or equipments that are to be utilized on more than one vehicle configuration, the qualification testing is to be so oriented as to satisfy the most severe requirements. In a case where time is of the essence, the qualification tests are to be conducted to satisfy the lesser conditions initially, with additional tests subsequently being run to meet the more stringent requirements.

Sample sizes shall be consistent with the minimum number of articles needed to complete the required tests for Block I qualification. An attempt shall be made to qualify for Block I with a maximum of two equivalent systems (one systems's worth of components for Design Proof tests - Phase A, and one system for Mission Simulation - Life tests) with exceptions taken only where dictated by qualification schedule requirements and the necessity for design margin verification (off-limit tests) related to equipments that influence crew safety (manned flights only).

Repeatability shall not be considered in establishing the number of hardware items or the number of tests required for Block I qualification.

Qualification tests are to consist of a three-part, two-phase program as defined below:

Phase	Part	Test Type	Level of Assembly
A	1 2	Design Proof Off-Limit	Components
B	3	Mission Simulation	Subsystems or functional groups of equipment

Design proof tests will be run at the most severe levels of expected mission environments. In general, these tests are to be sequentially applied, although under certain circumstances combined environments will be utilized. Exposure times will be commensurate with the critical mission phases during which these environments are encountered. Off-limit testing is to verify positive design margins and to verify the equipment's true capability under emergency conditions. Failure modes and fail-safe provisions are also verified by this test. Off-limit testing for Block I hardware is restricted to items essential to crew safety.

Mission simulation - life testing consists of one mission simulation conducted to assure suitable performance, compatibility and the absence of interface problems under natural combinations of critical mission environments, at nominal values. The time of test is commensurate with each phase of the mission, starting with ground handling and transportation of the spacecraft. Availability of test facilities and schedules will establish the general constraints to true simulation, and will restrict environmental combinations.

Qualification testing is to be conducted at the highest practical level of assembly. The level considered practical will depend upon such factors as availability of facilities and the minimum acceptable number of interfaces to be maintained in order to assure a realistic simulation of the mission conditions to be encountered.

Qualification for manned flights shall be completed prior to sign-off of the DD250 for the flight vehicle.

Qualification planning is to be based upon success during the test program. No provisions are to be included in the initial planning phases for schedule or cost contingencies, associated with failures.

Prior to the initiation of any tests, investigations are to be conducted to assure that similar usage history is not available which would allow qualification by similarity.

In addition to the previous items, the following ground rules apply to the ground support equipment:

1. Qualification is to be completed prior to usage with the first manned flight vehicle.

2. There shall be no new procurement for qualification testing. Equipments intended for field usage shall be

subjected to the qualification tests and then refurbished prior to final application.

3. The qualification program shall be in consonance with the requirements of MSC, GSE-1B.
4. Mission critical GSE only shall be qualification tested.

4.4 Configuration Management Provisions.

4.4.1 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 operation 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 Reference Axes: - The reference axes of the CSM shall be orthogonal and shall be identified as shown in Table I.

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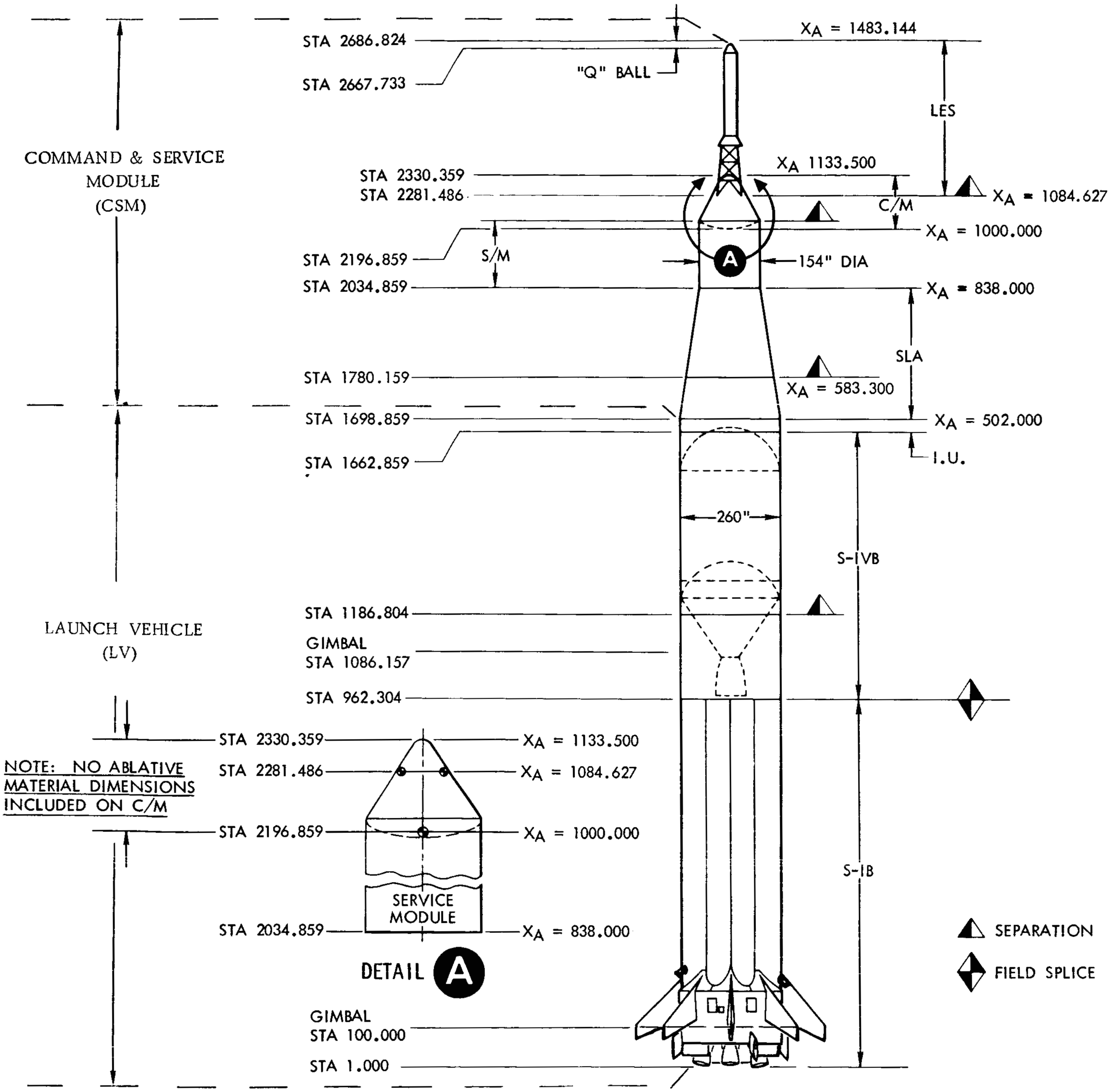

Figure 1. Saturn C-IB Configuration

[redacted]

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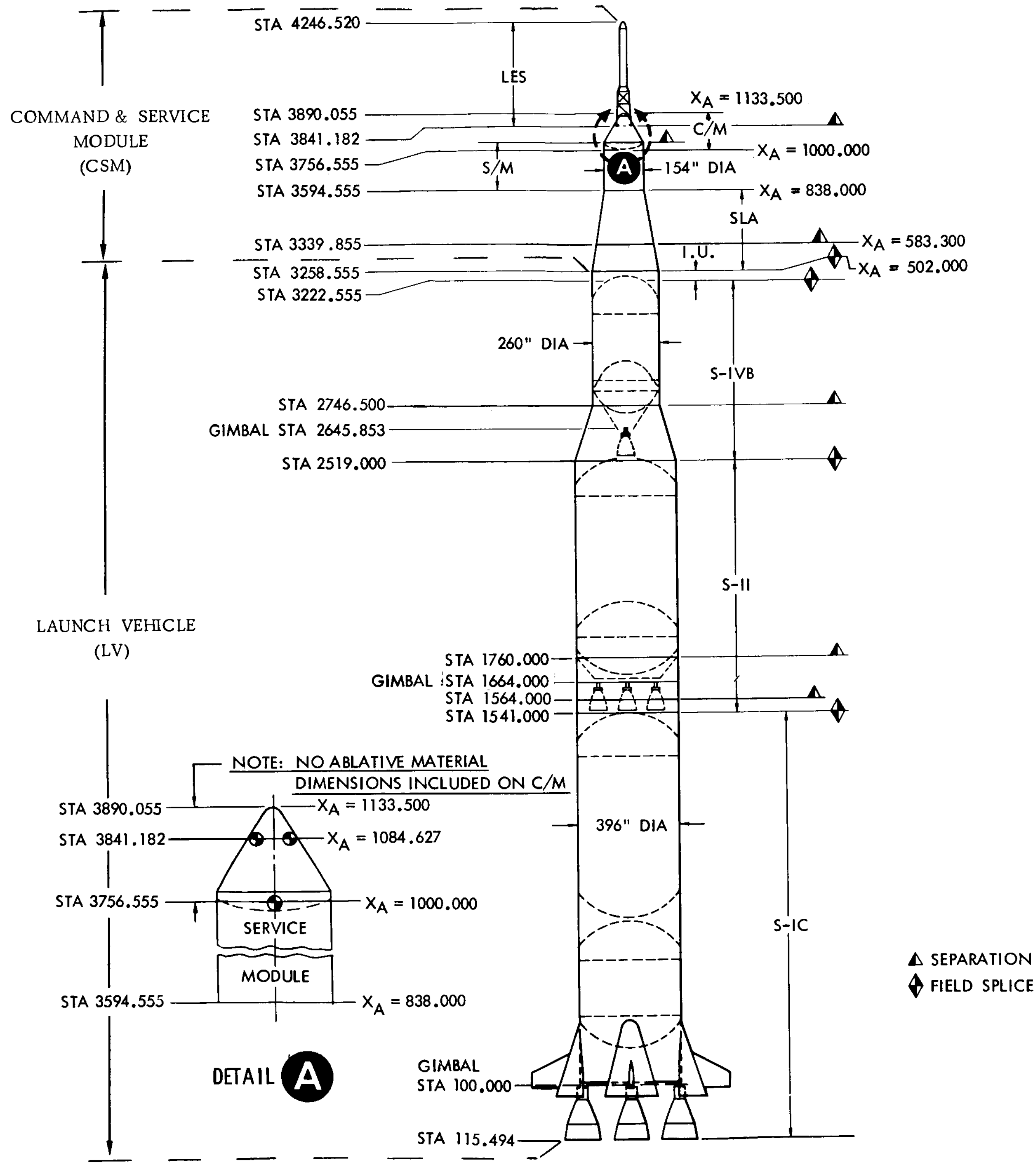

Figure 2. Saturn V LOR Configuration

VIBRATION TIME HISTORY
ATMOSPHERIC FLIGHT

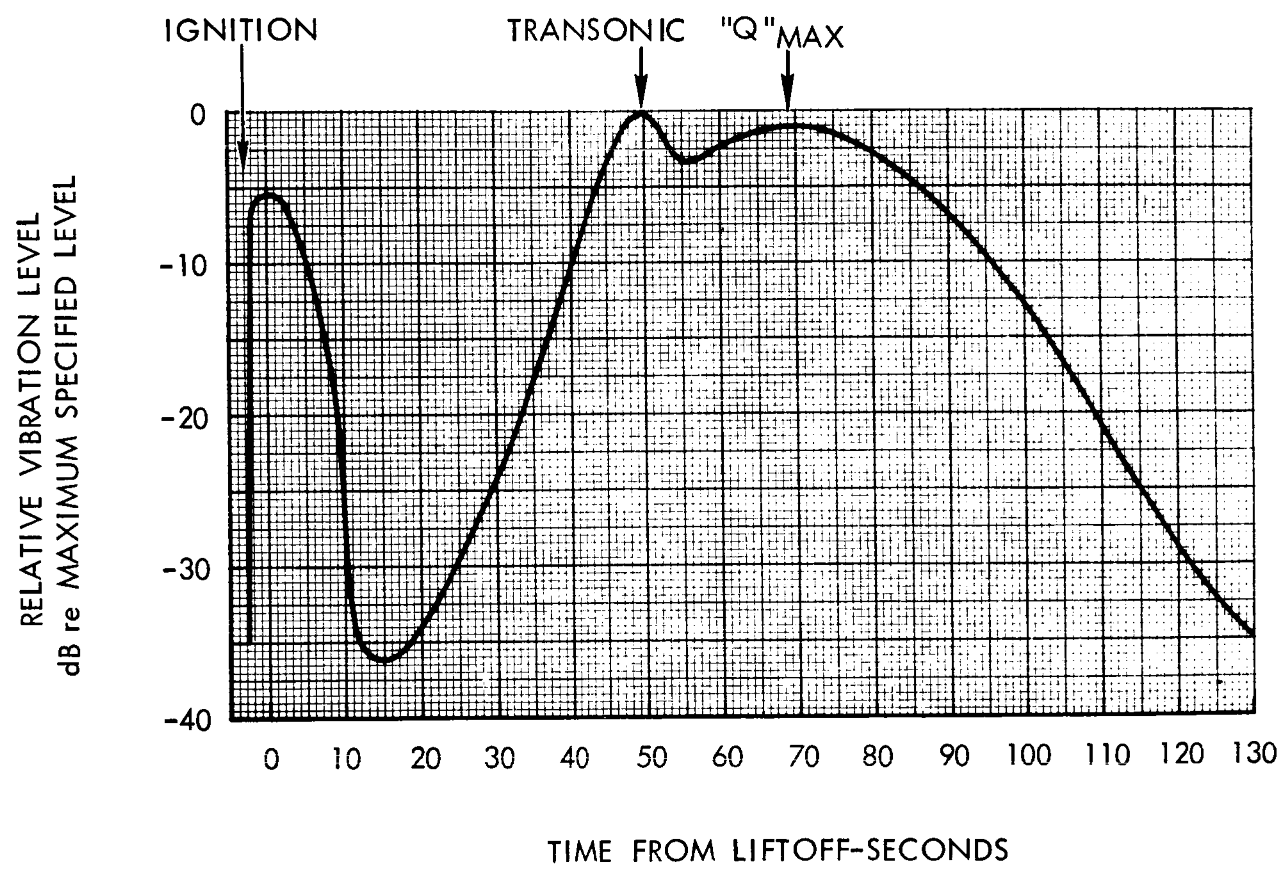

Figure 3. Vibration Time History - Atmospheric Flight

~~CONFIDENTIAL~~

VIBRATION
LAUNCH ESCAPE SYSTEM
ATMOSPHERIC FLIGHT

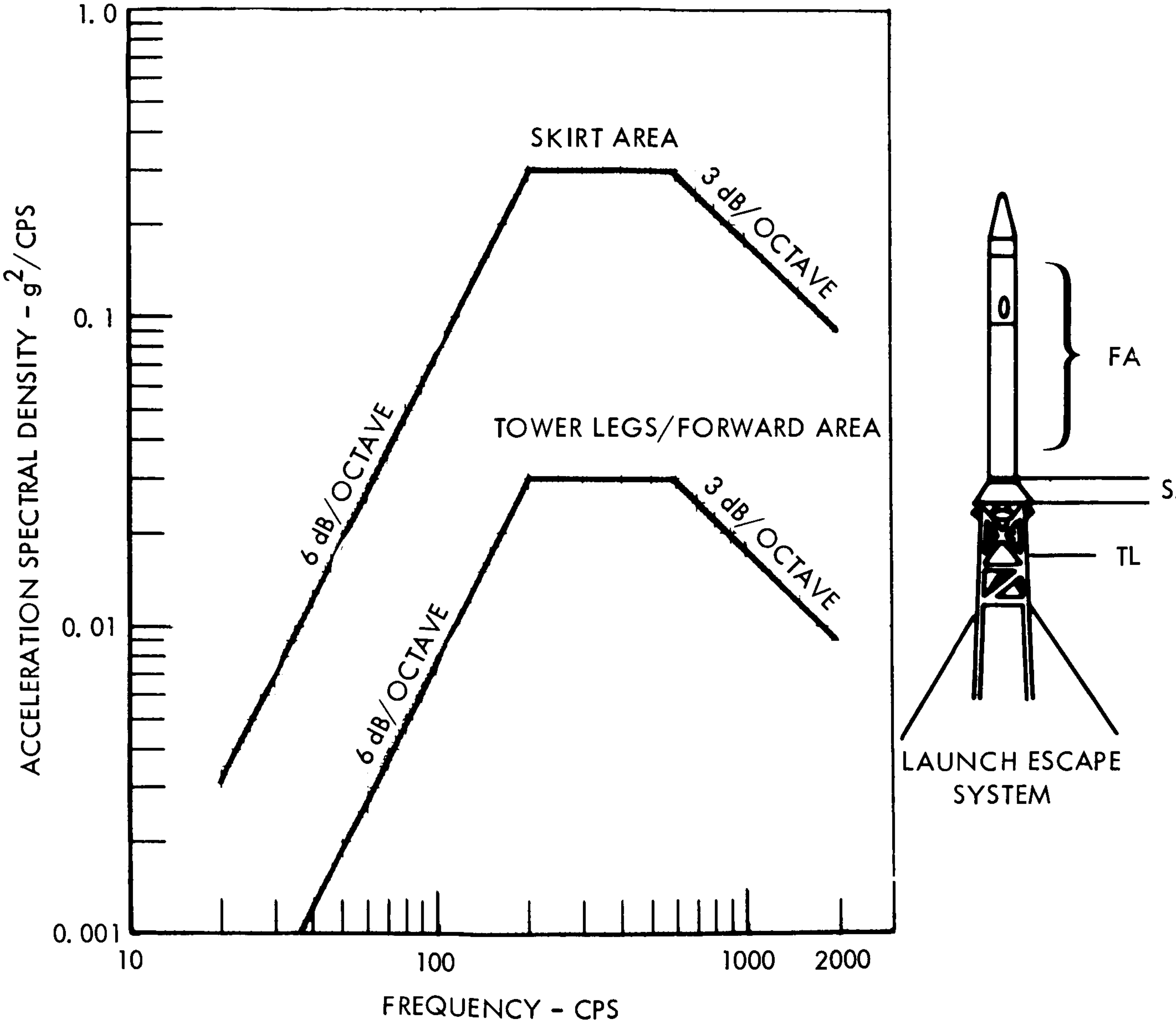

Figure 4. Vibration LES - Atmospheric Flight

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VIBRATION
COMMAND MODULE

ATMOSPHERIC FLIGHT

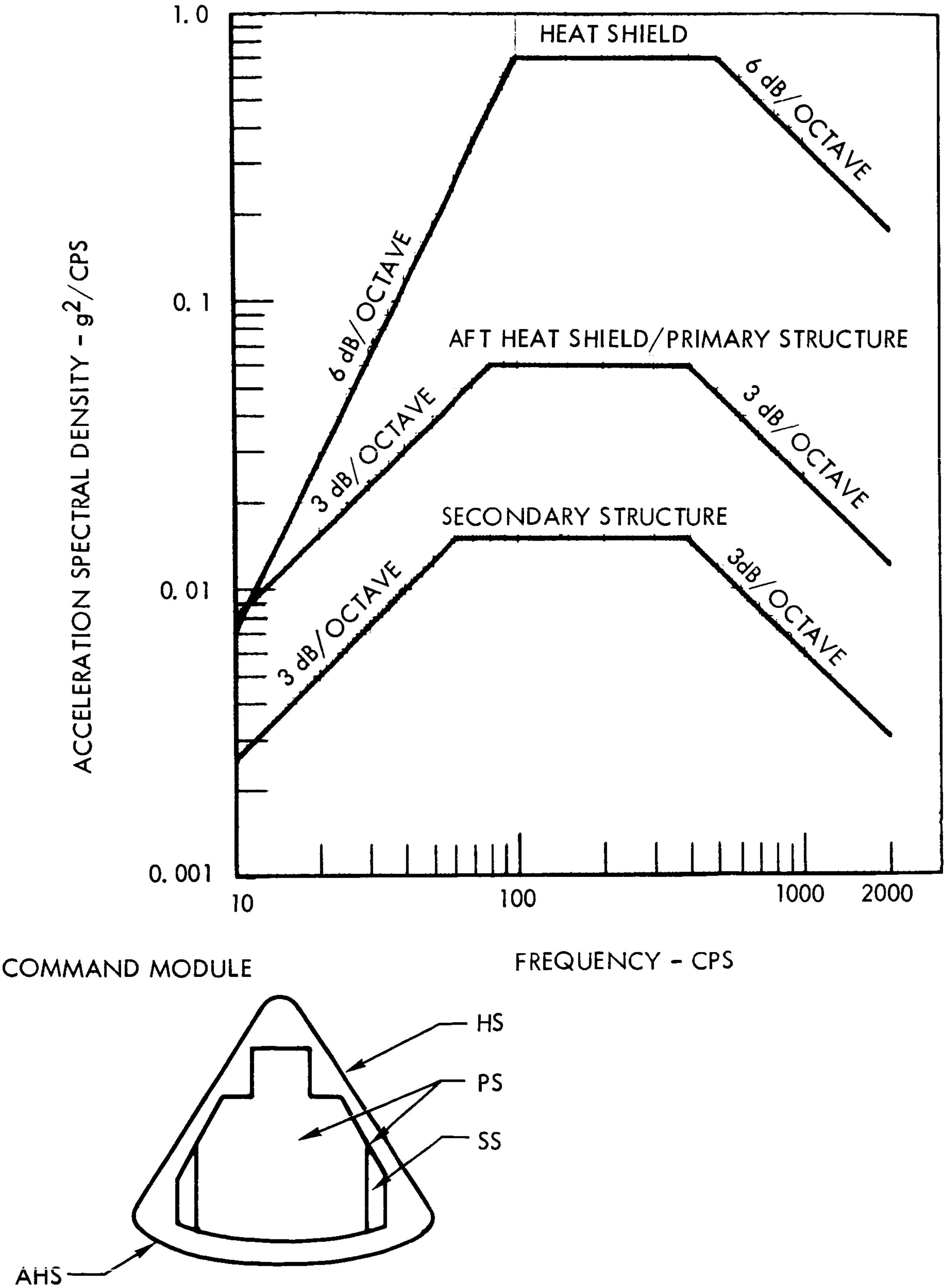

Figure 5. Vibration CM - Atmospheric Flight

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VIBRATION
SERVICE MODULE
ATMOSPHERIC FLIGHT

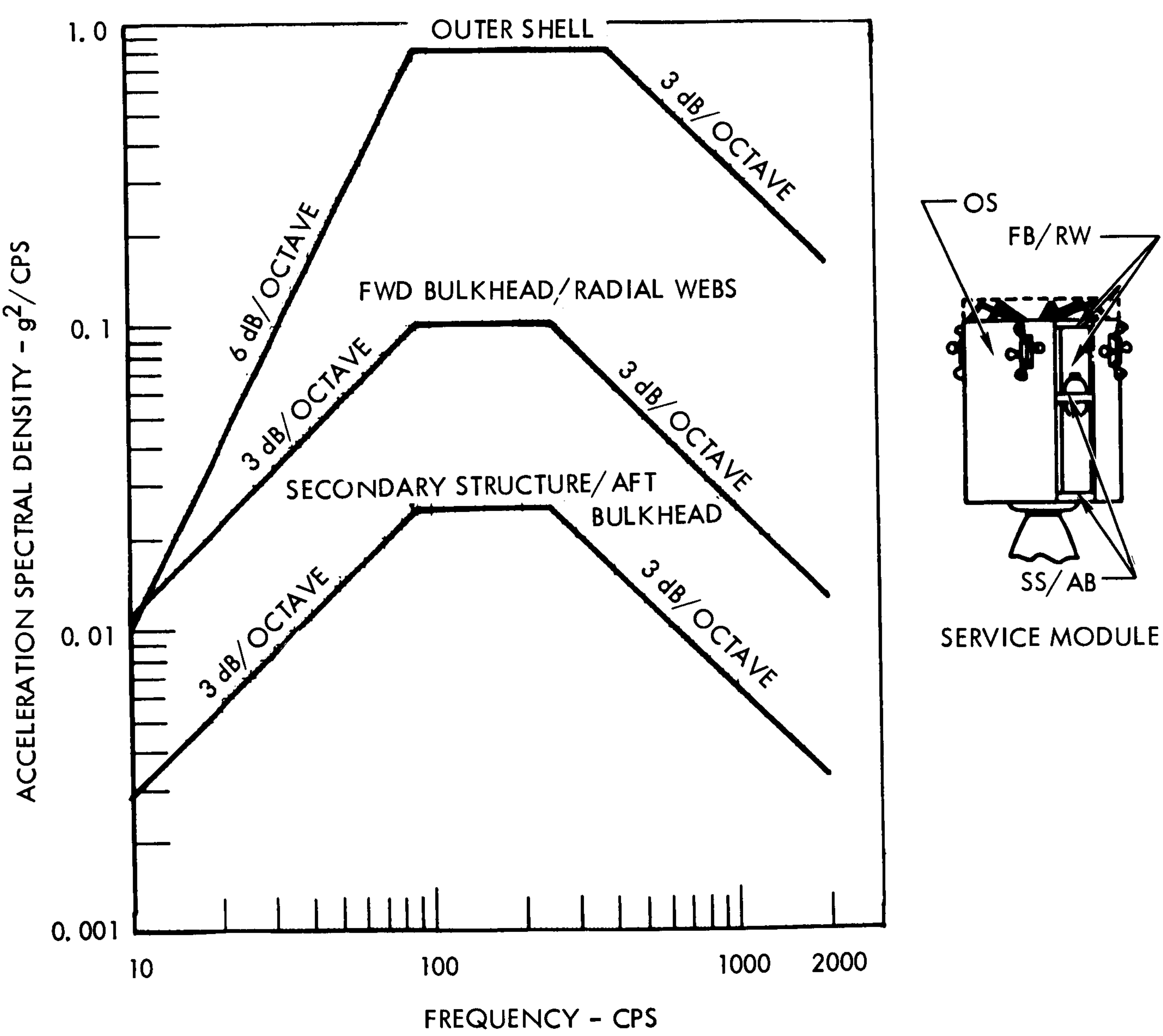

Figure 6. Vibration SM - Atmospheric Flight

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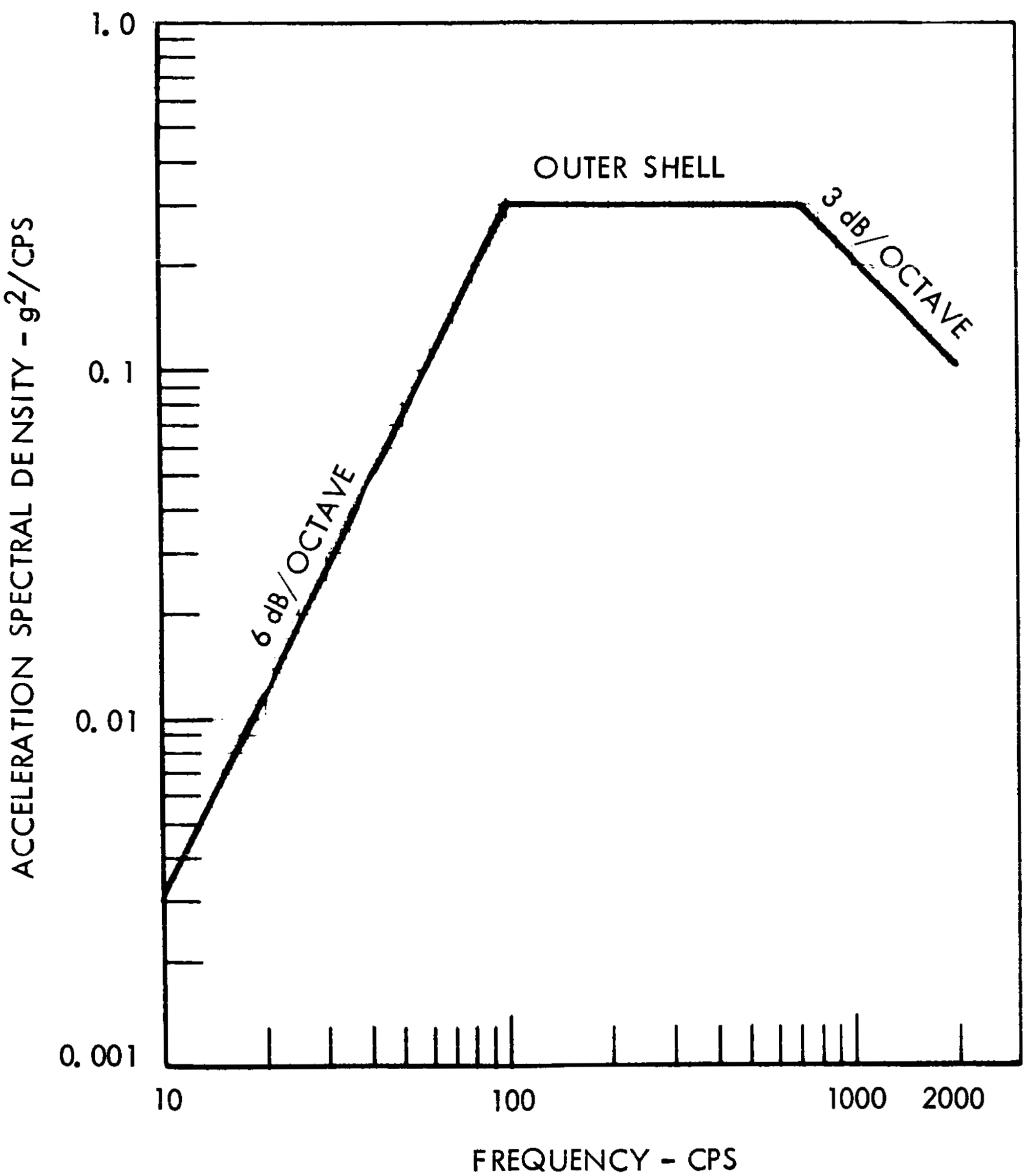
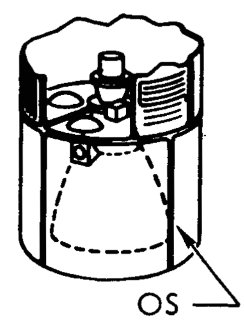
CON[redacted]

VIBRATION
SERVICE MODULE ADAPTER
ATMOSPHERIC FLIGHT





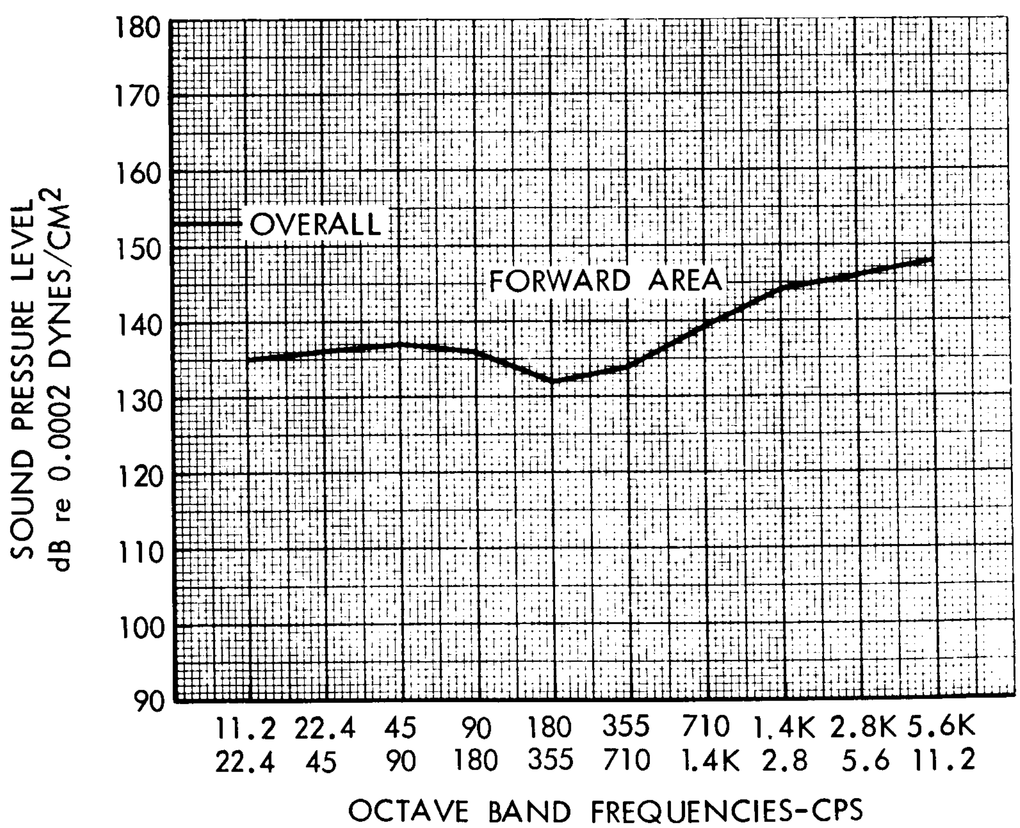
SM ADAPTER

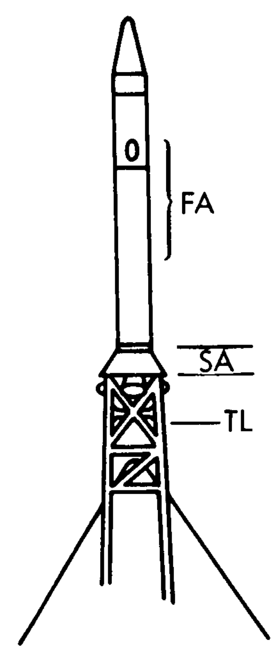
Figure 7. Vibration SM Adapter - Atmospheric Flight

[redacted]

ACOUSTICS

LAUNCH ESCAPE SYSTEM
ATMOSPHERIC FLIGHT





LAUNCH ESCAPE SYSTEM

Figure 8. Acoustics LES - Atmospheric Flight - Forward Area

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ACOUSTICS
LAUNCH ESCAPE SYSTEM
ATMOSPHERIC FLIGHT

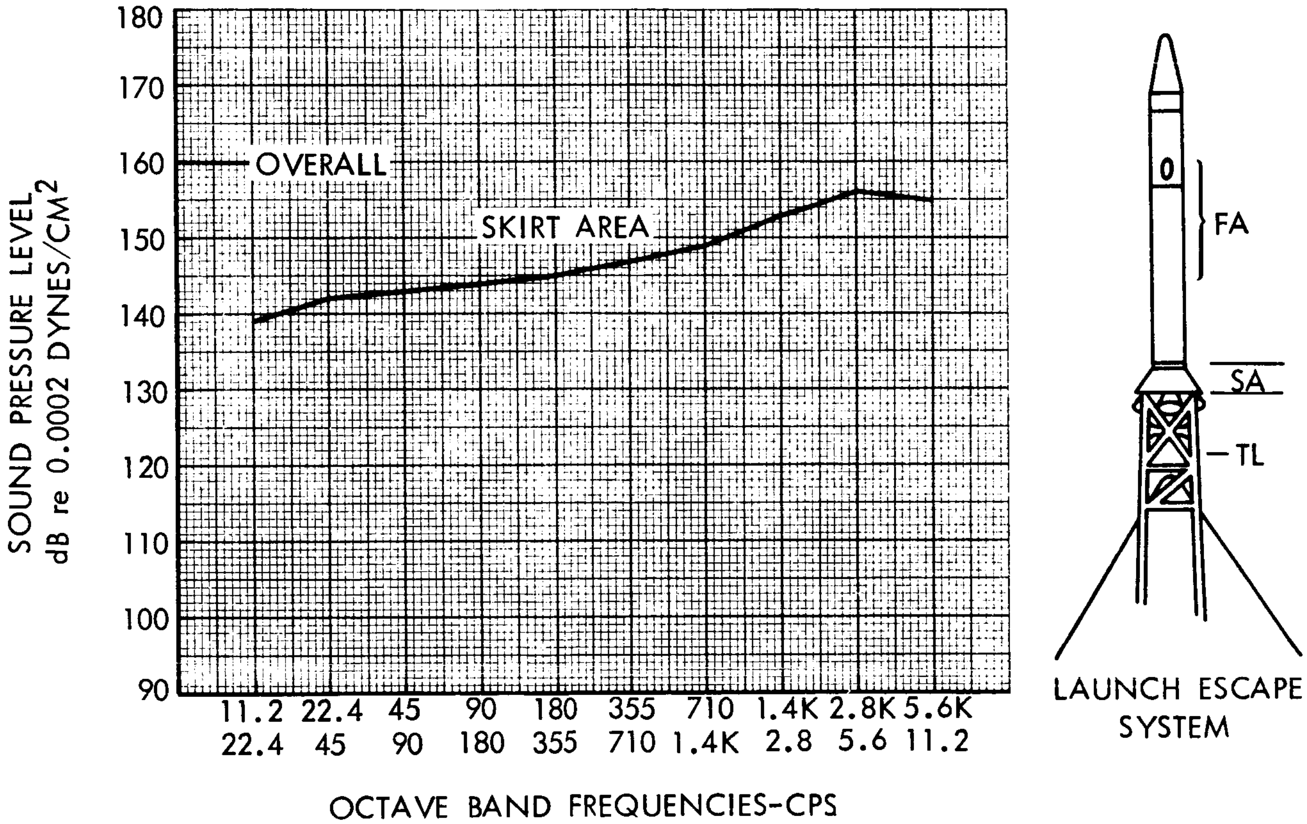

Figure 9. Acoustics LES - Atmospheric Flight - Skirt Area

~~CONFIDENTIAL~~

ACOUSTICS
COMMAND MODULE
ATMOSPHERIC FLIGHT

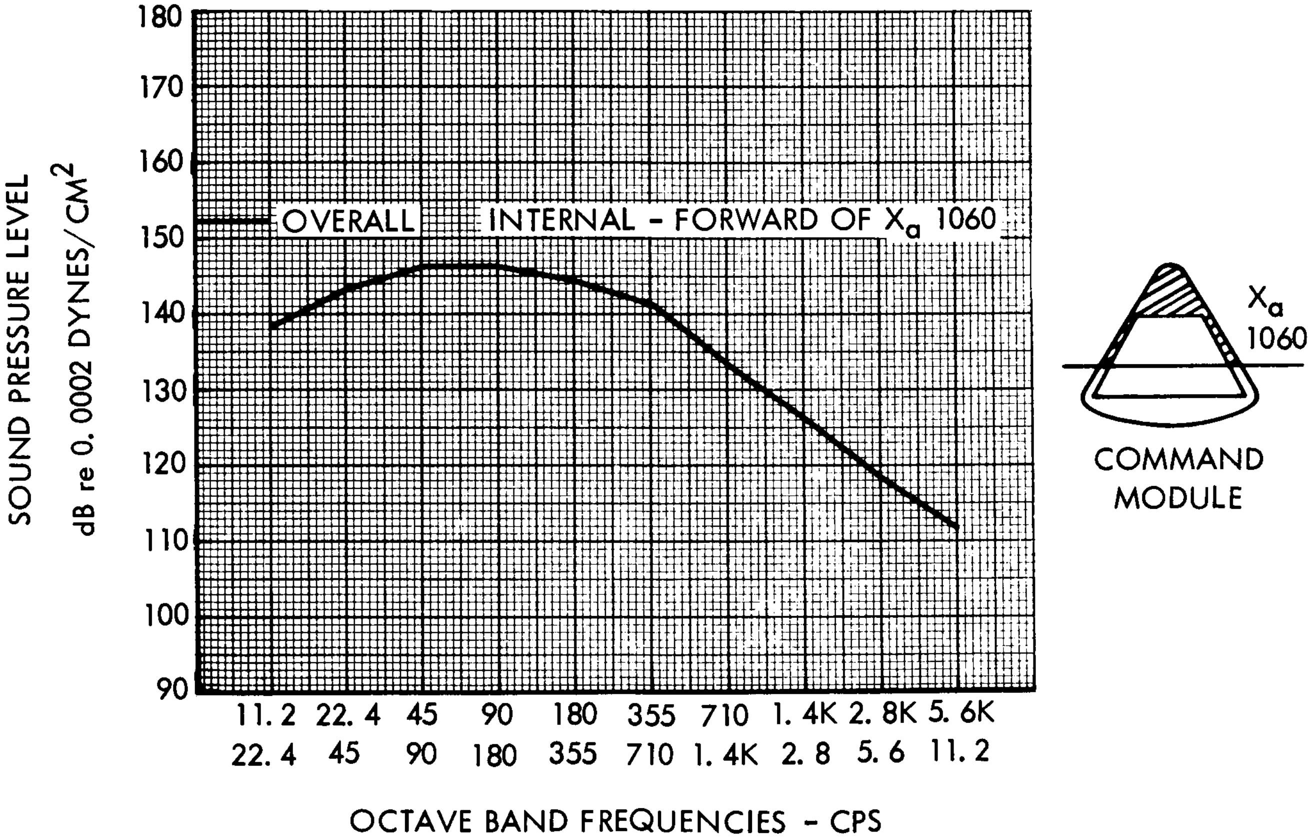

Figure 10. Acoustics CM - Atmospheric Flight - Internal - Forward Xa1060

ACOUSTICS

COMMAND MODULE
ATMOSPHERIC FLIGHT

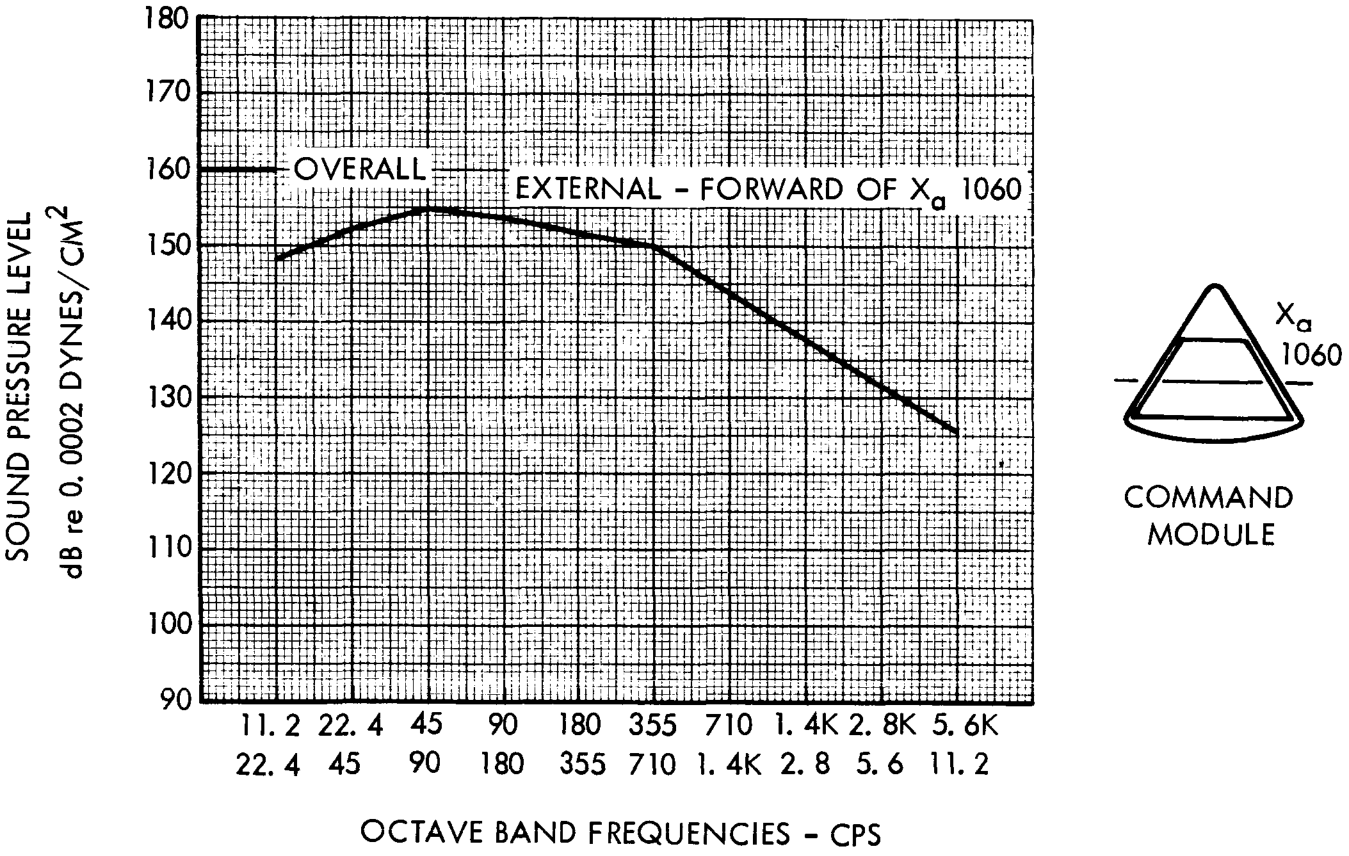

Figure 11. Acoustics CM - Atmospheric Flight - External - Forward Xa1060

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ACOUSTICS

COMMAND MODULE

ATMOSPHERIC FLIGHT

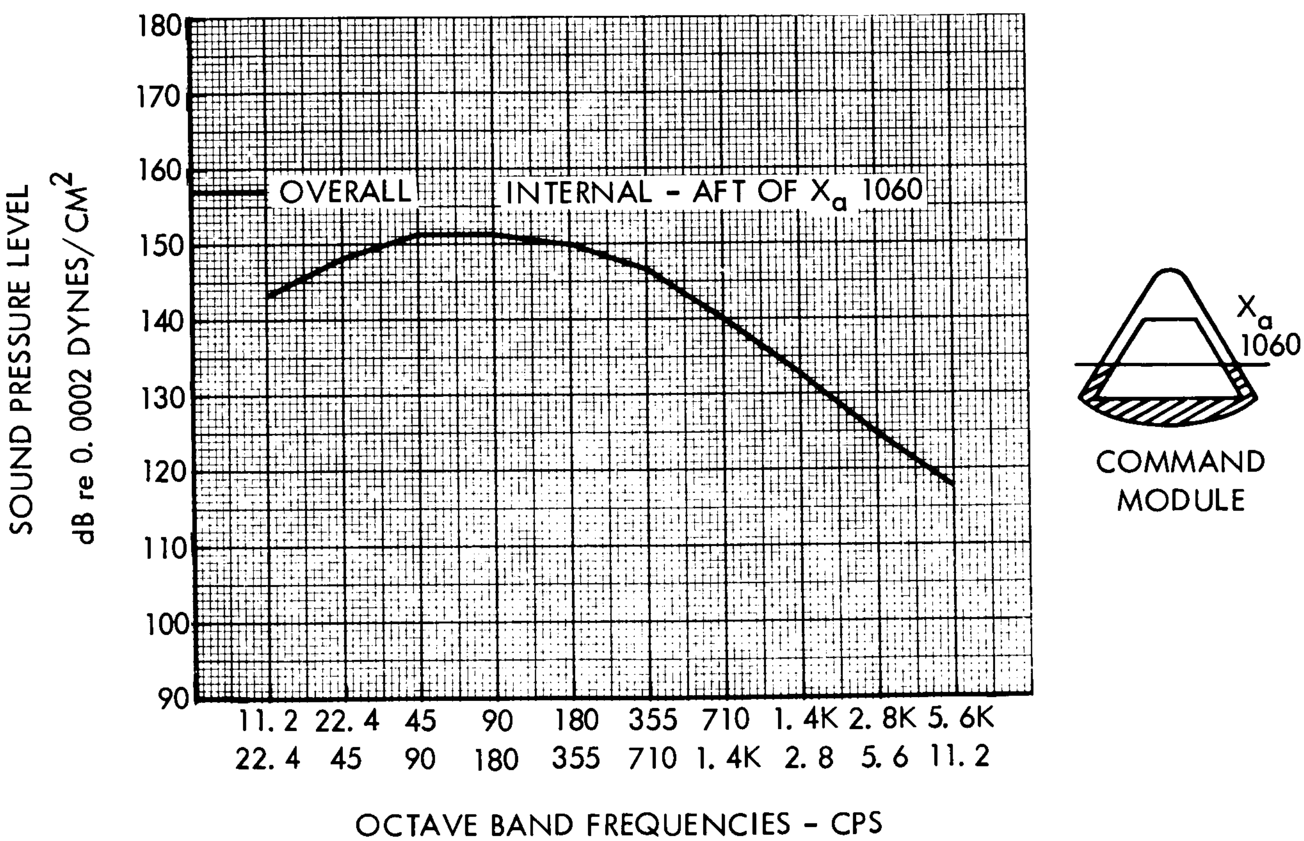

Figure 12. Acoustics CM - Atmospheric Flight - Internal - Aft Xa1060

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ACOUSTICS

COMMAND MODULE
ATMOSPHERIC FLIGHT

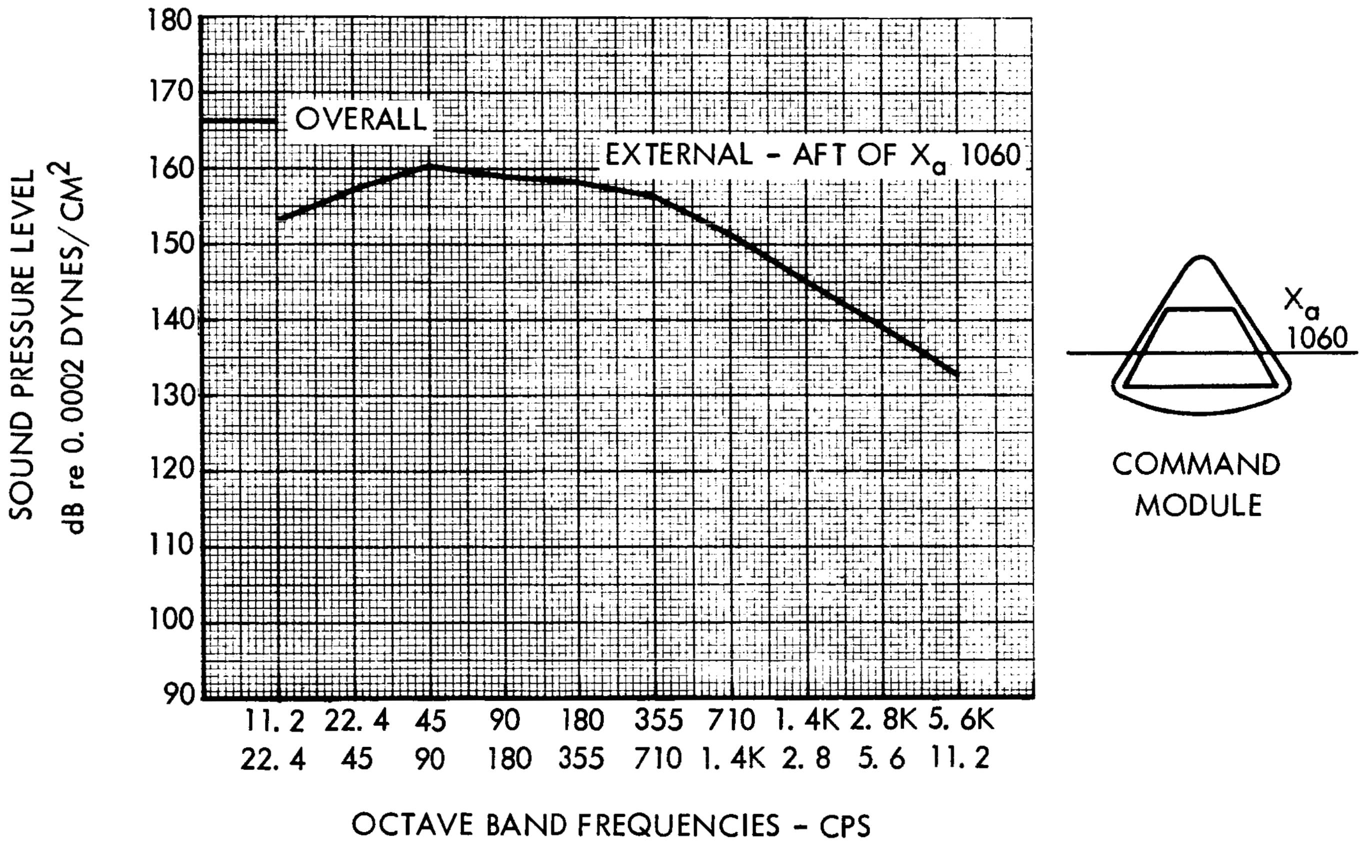

Figure 13. Acoustics CM - Atmospheric Flight - External - Aft Xa1060

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ACOUSTICS
COMMAND MODULE
ATMOSPHERIC FLIGHT

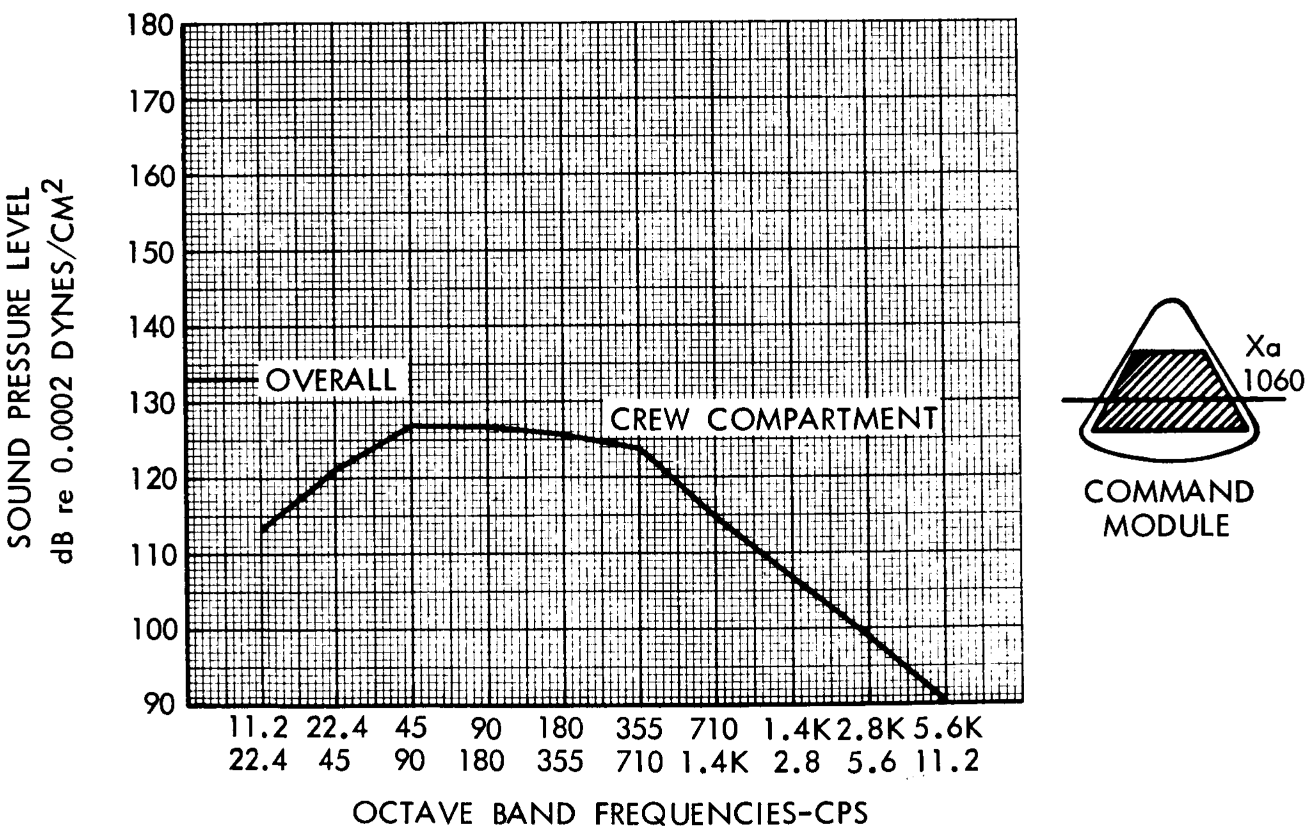

Figure 14. Acoustics CM - Atmospheric Flight - Crew Compartment

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ACOUSTICS

SERVICE MODULE
ATMOSPHERIC FLIGHT

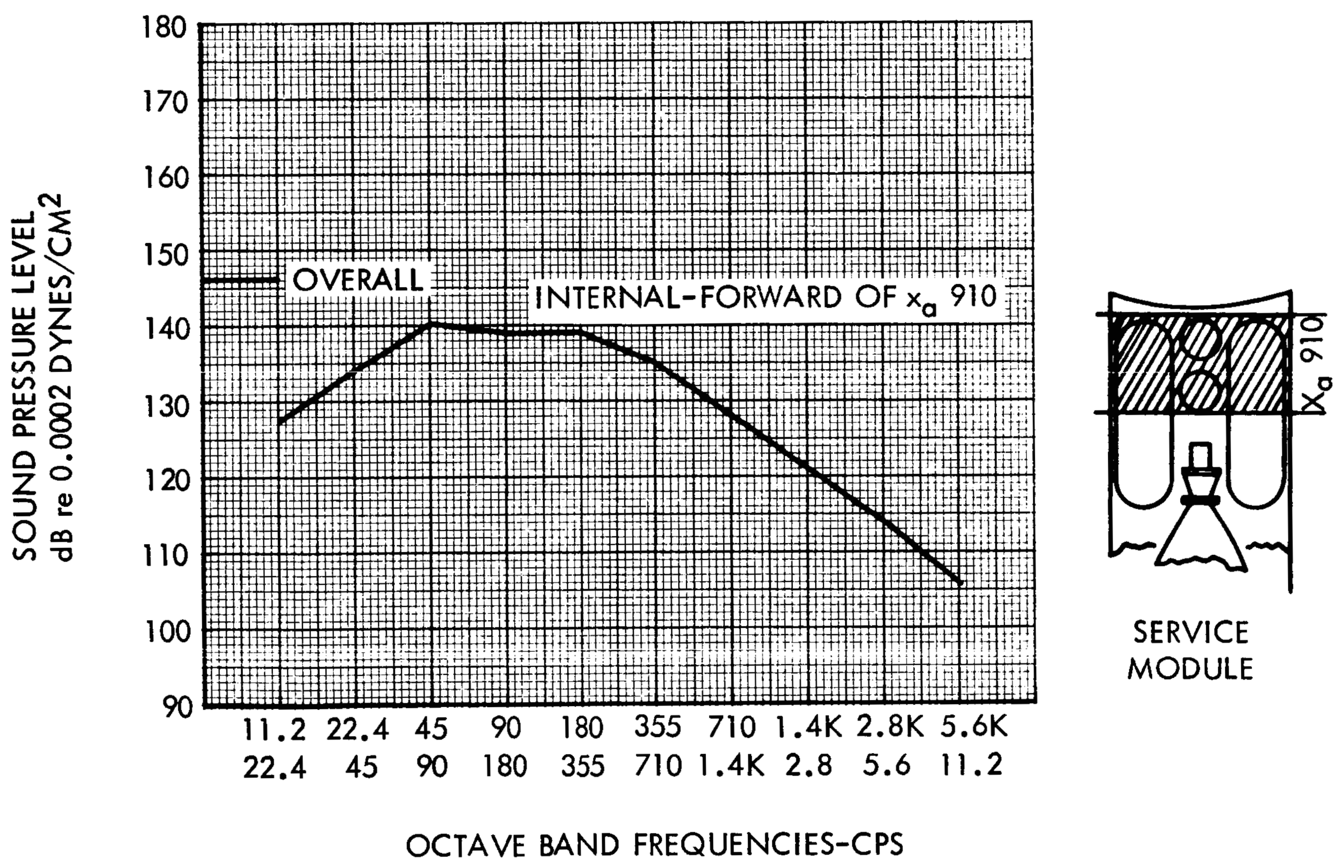
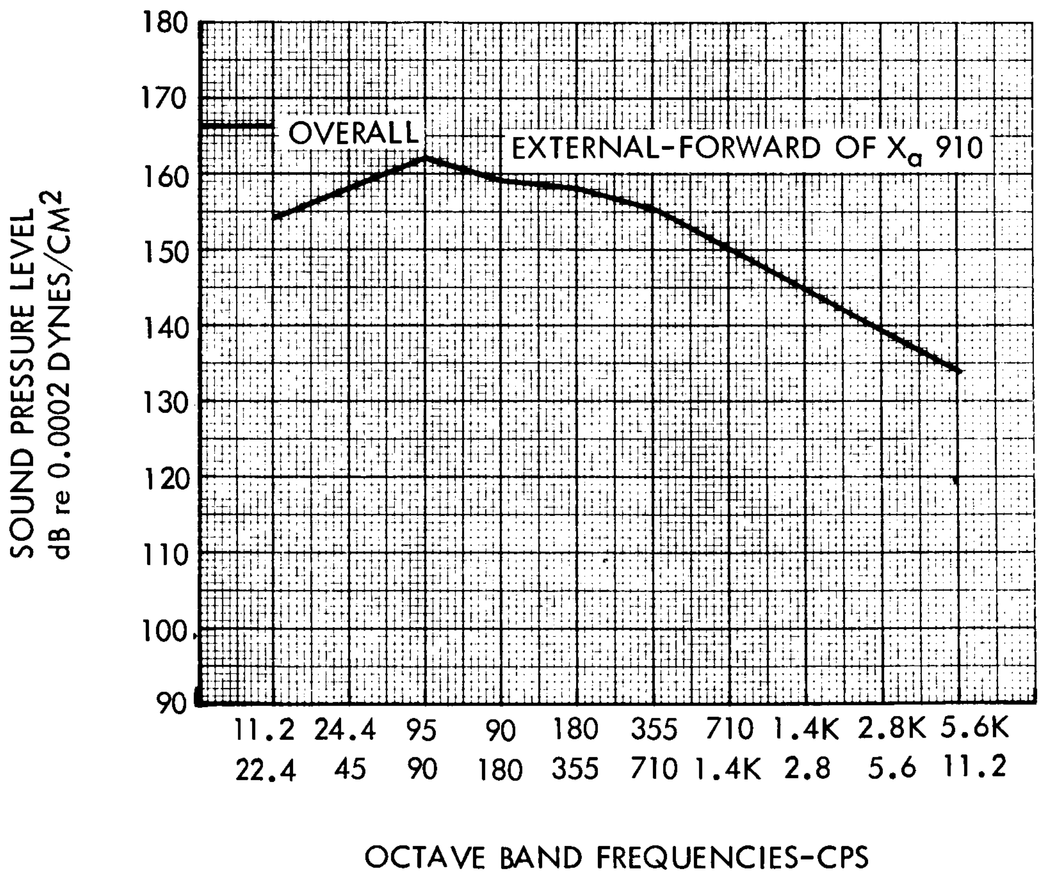


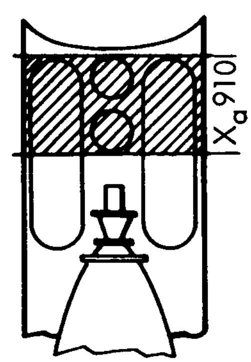
Figure 15. Acoustics SM - Atmospheric Flight - Internal - Forward Xa910

CONFIDENTIAL

ACOUSTICS

SERVICE MODULE
ATMOSPHERIC FLIGHT





SERVICE MODULE

Figure 16. Acoustics SM - Atmospheric Flight - External - Forward Xa910

CON[redacted]

ACOUSTICS

SM RCS ENGINE & PANEL
ATMOSPHERIC FLIGHT

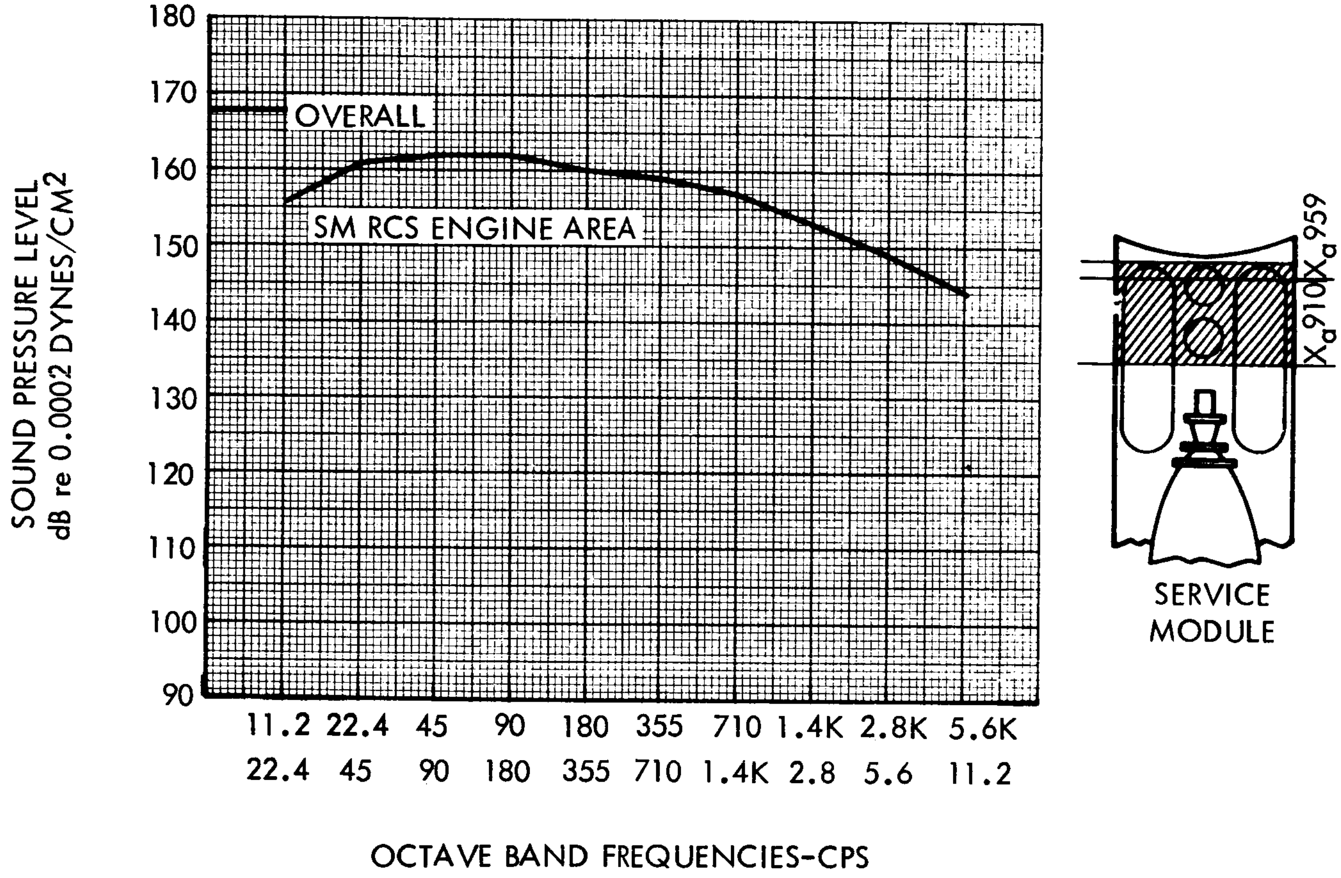


Figure 17. Acoustics SM/RCS Engine & Panel - Atmospheric Flight

ACOUSTICS
SERVICE MODULE
ATMOSPHERIC FLIGHT

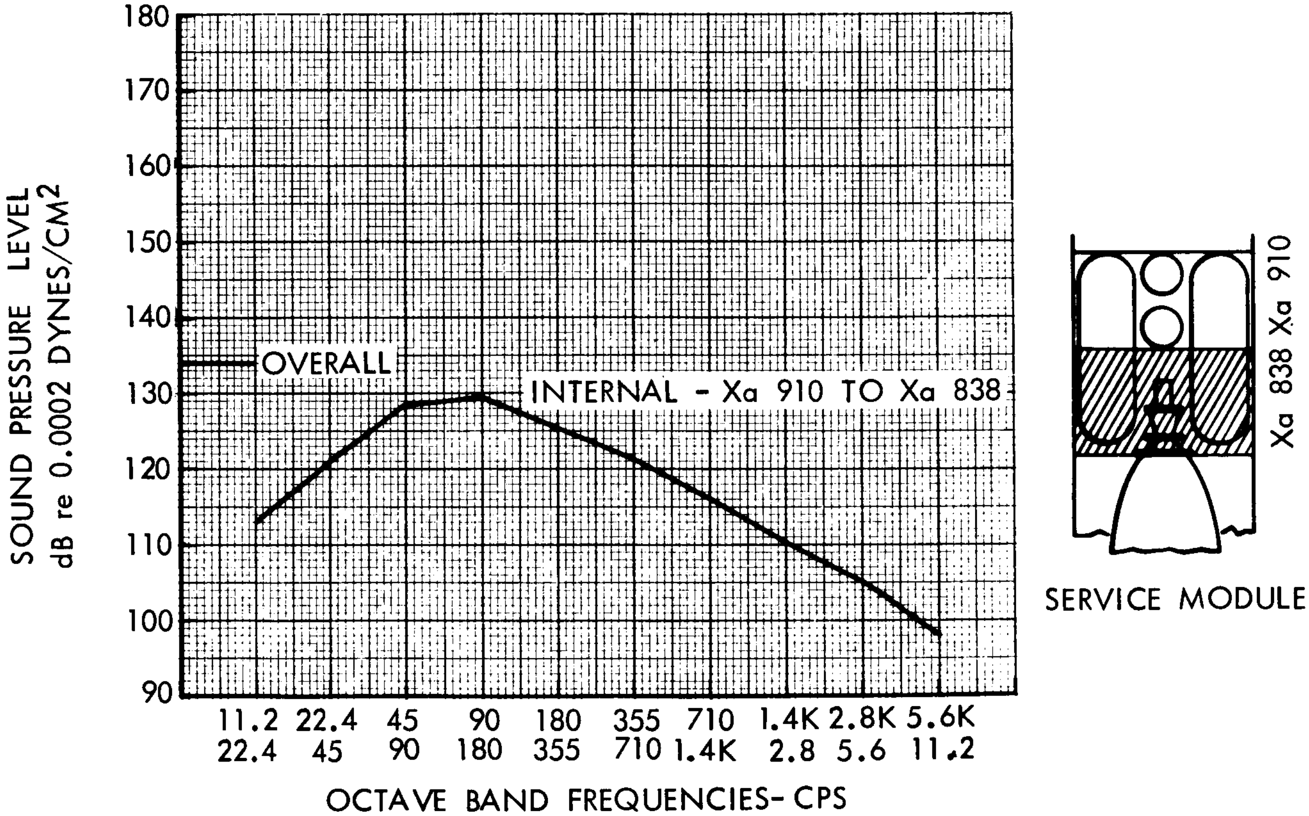


Figure 18. Acoustics SM - Atmospheric Flight - Internal XA910 to Xa838

CONFIDENTIAL

ACOUSTICS
SERVICE MODULE
ATMOSPHERIC FLIGHT

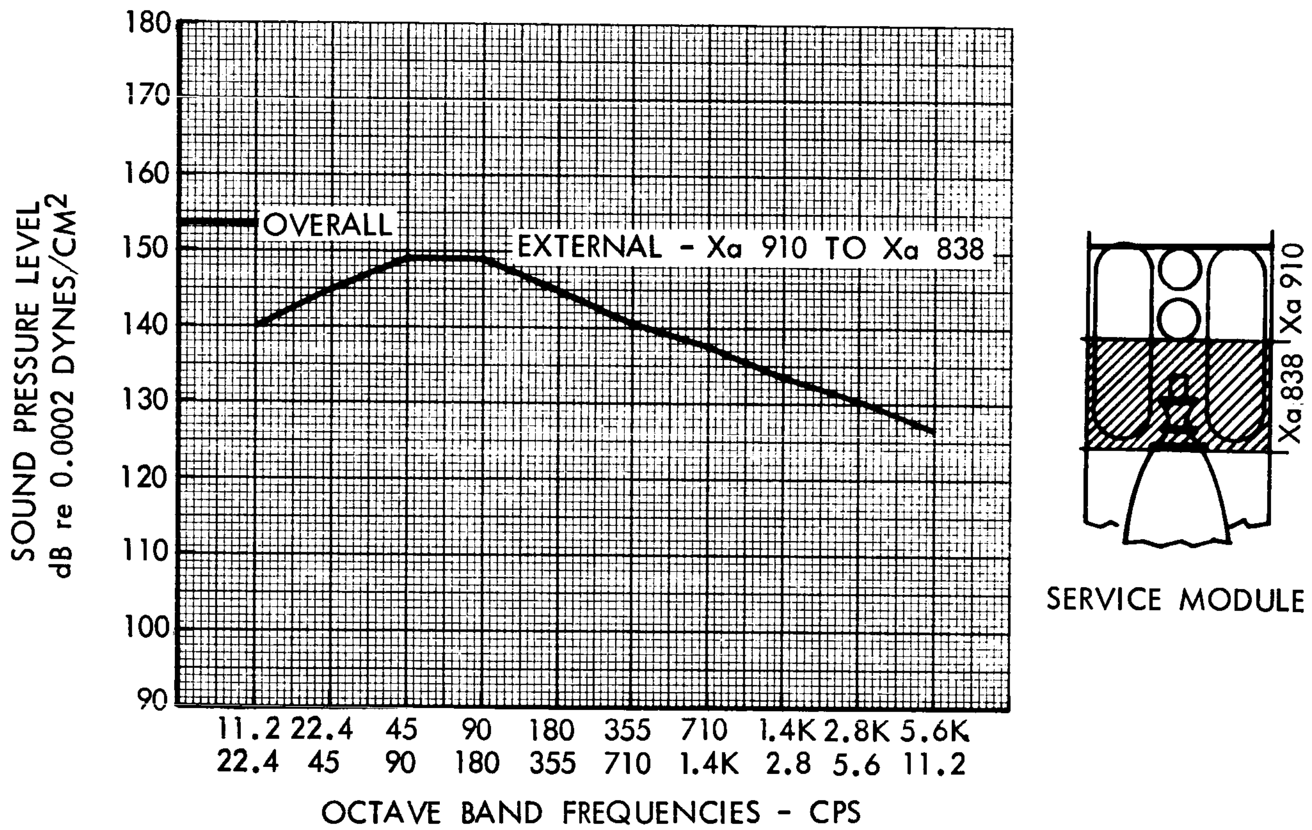


Figure 19. Acoustics SM - Atmospheric Flight - External Xa910 to Xa838

ACOUSTICS
ADAPTER
ATMOSPHERIC FLIGHT

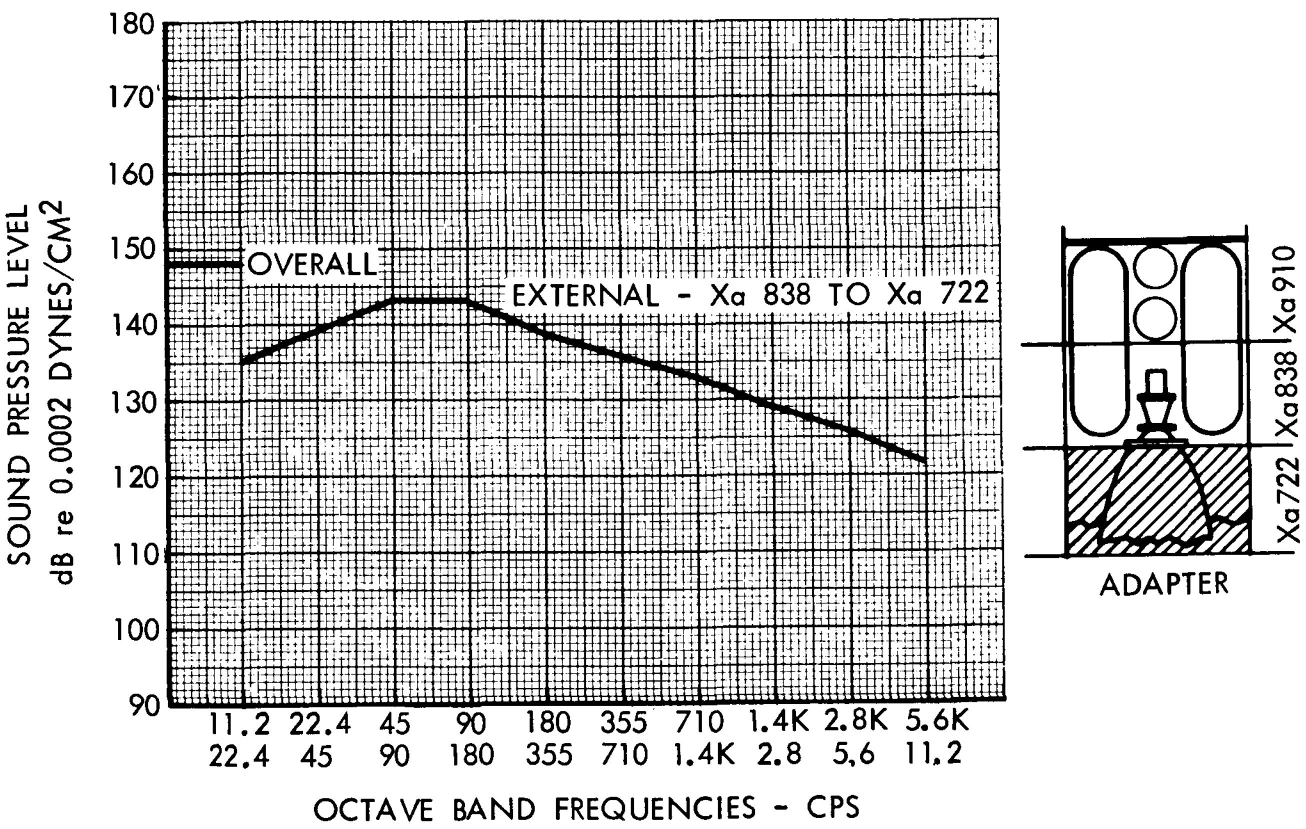


Figure 20. Acoustics Adapter - Atmospheric Flight - External Xa838 to Xa722

CO[redacted]

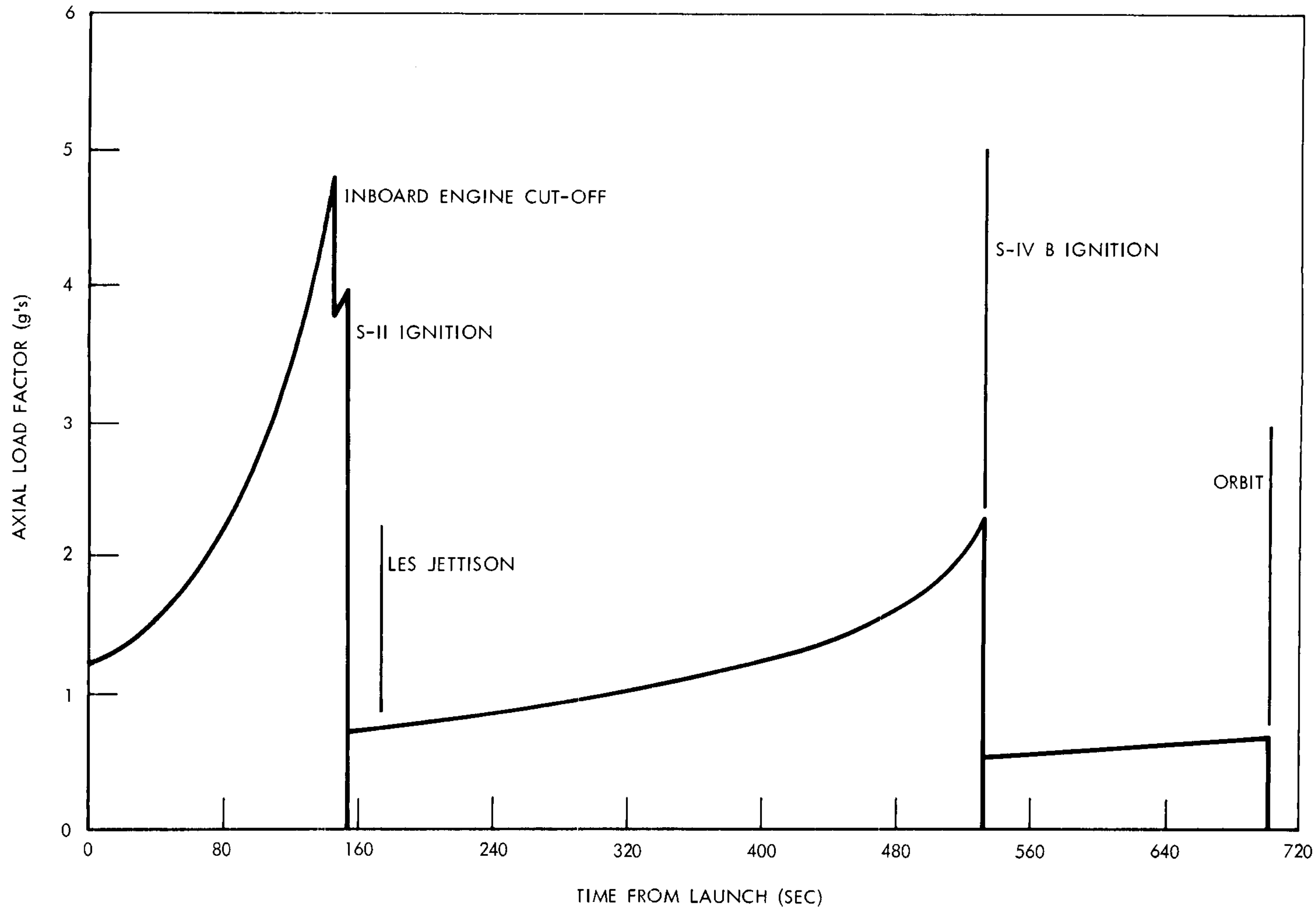


Figure 21. Axial Acceleration - Nominal Saturn V Boost

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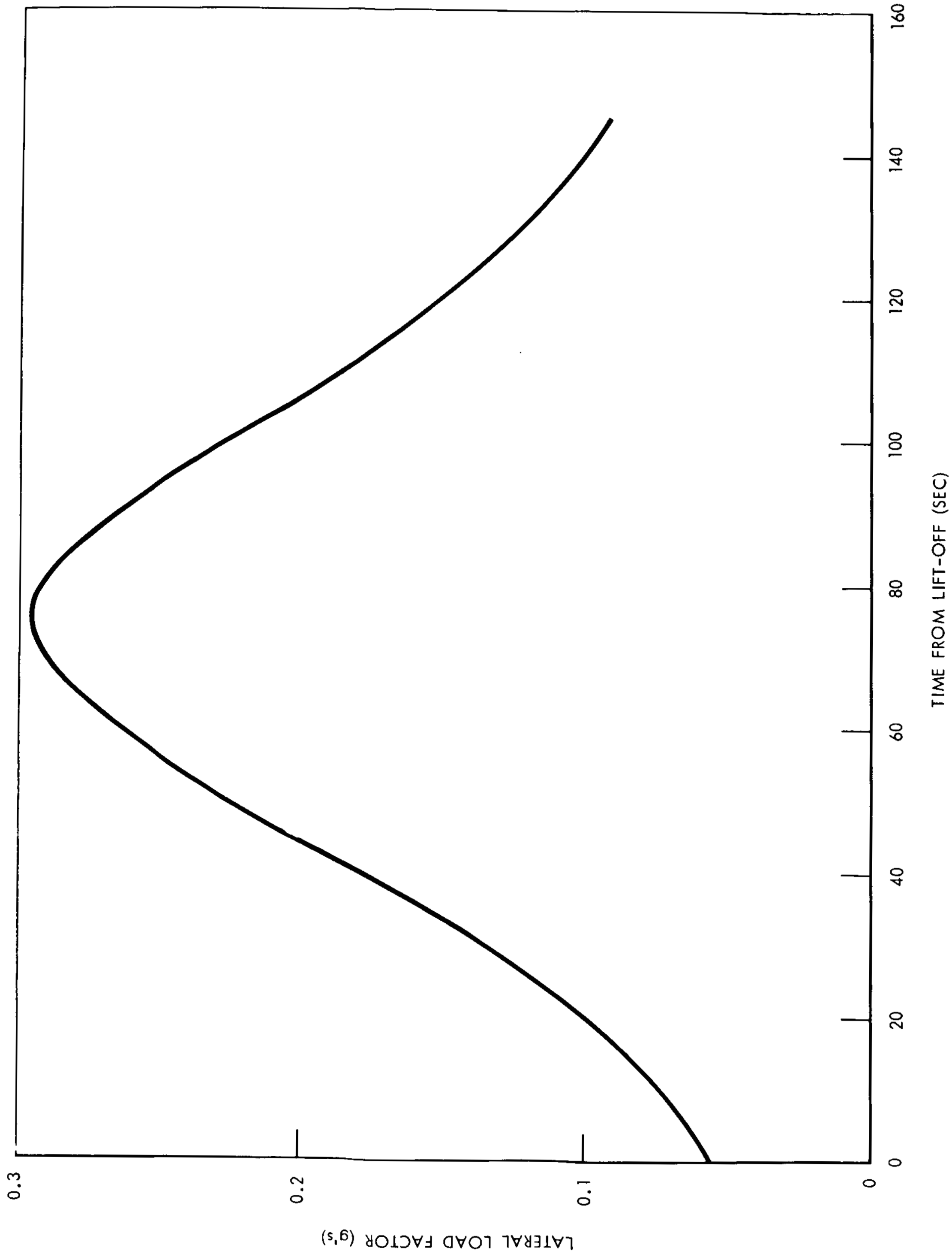

Figure 22. Lateral Acceleration During First Stage Boost

CON[redacted]

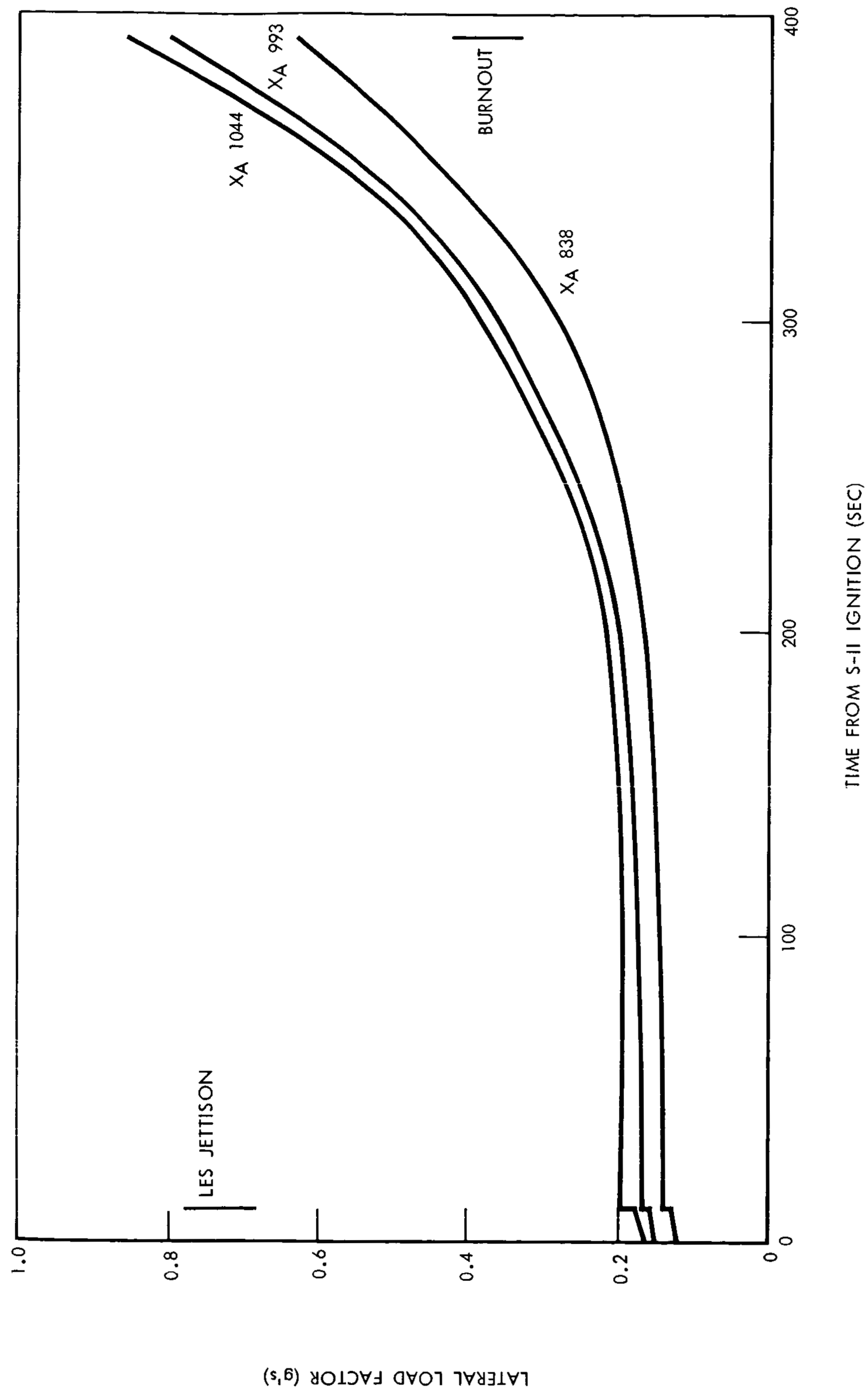

Figure 23. Lateral Acceleration - S-II Flight at Maximum Gimbal Deflection

[redacted]

[redacted]

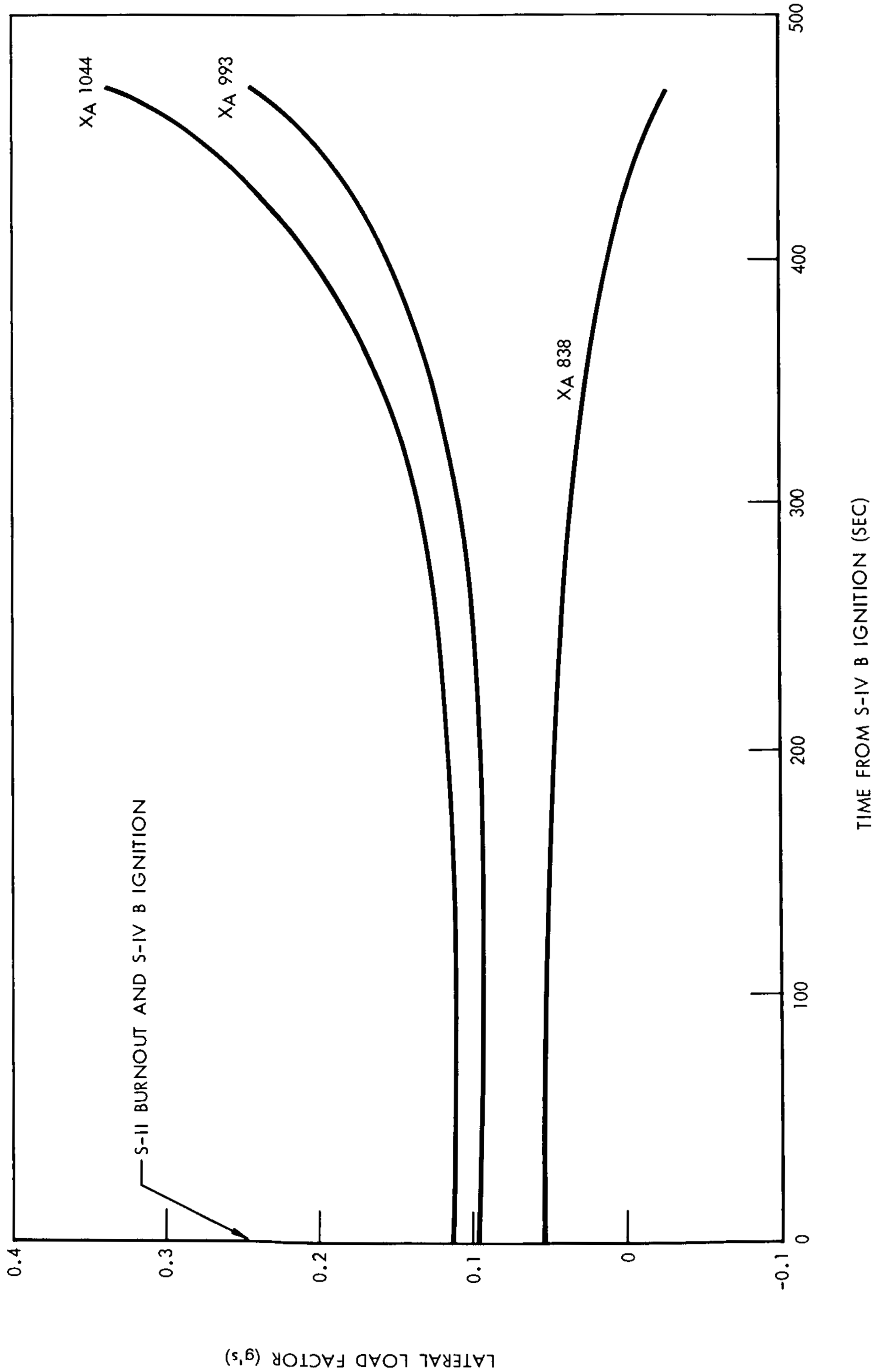

Figure 24. Lateral Acceleration - S-IVB Flight at Maximum Gimbal Deflection

CON[redacted]

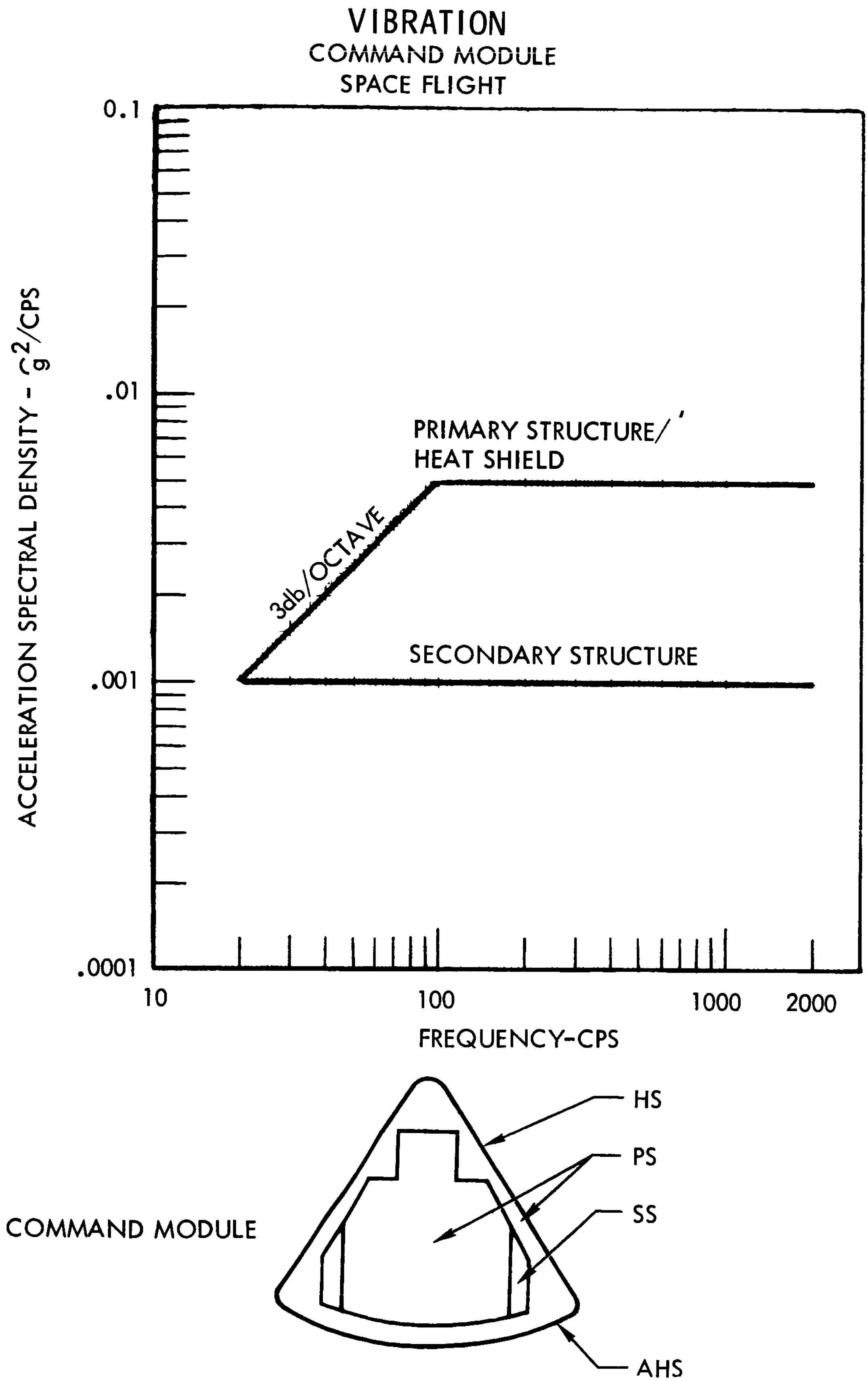

Figure 25. Vibration CM - Space Flight

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VIBRATION
SERVICE MODULE
SPACE FLIGHT

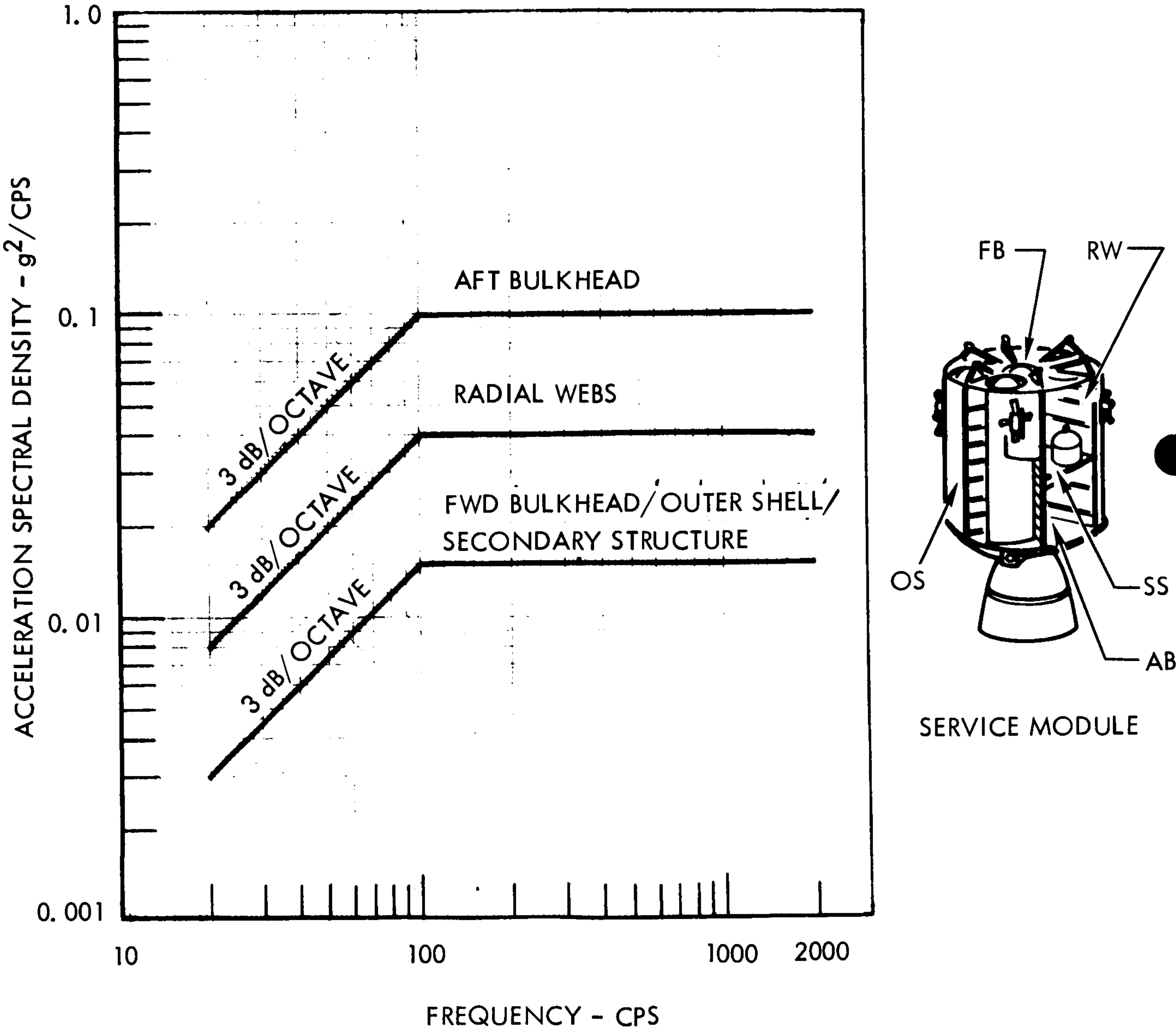

Figure 26. Vibration SM - Space Flight

[redacted]TIAL

CO[redacted]

NOTE: Referenced axis is at the equipment and also parallel to spacecraft's major axis

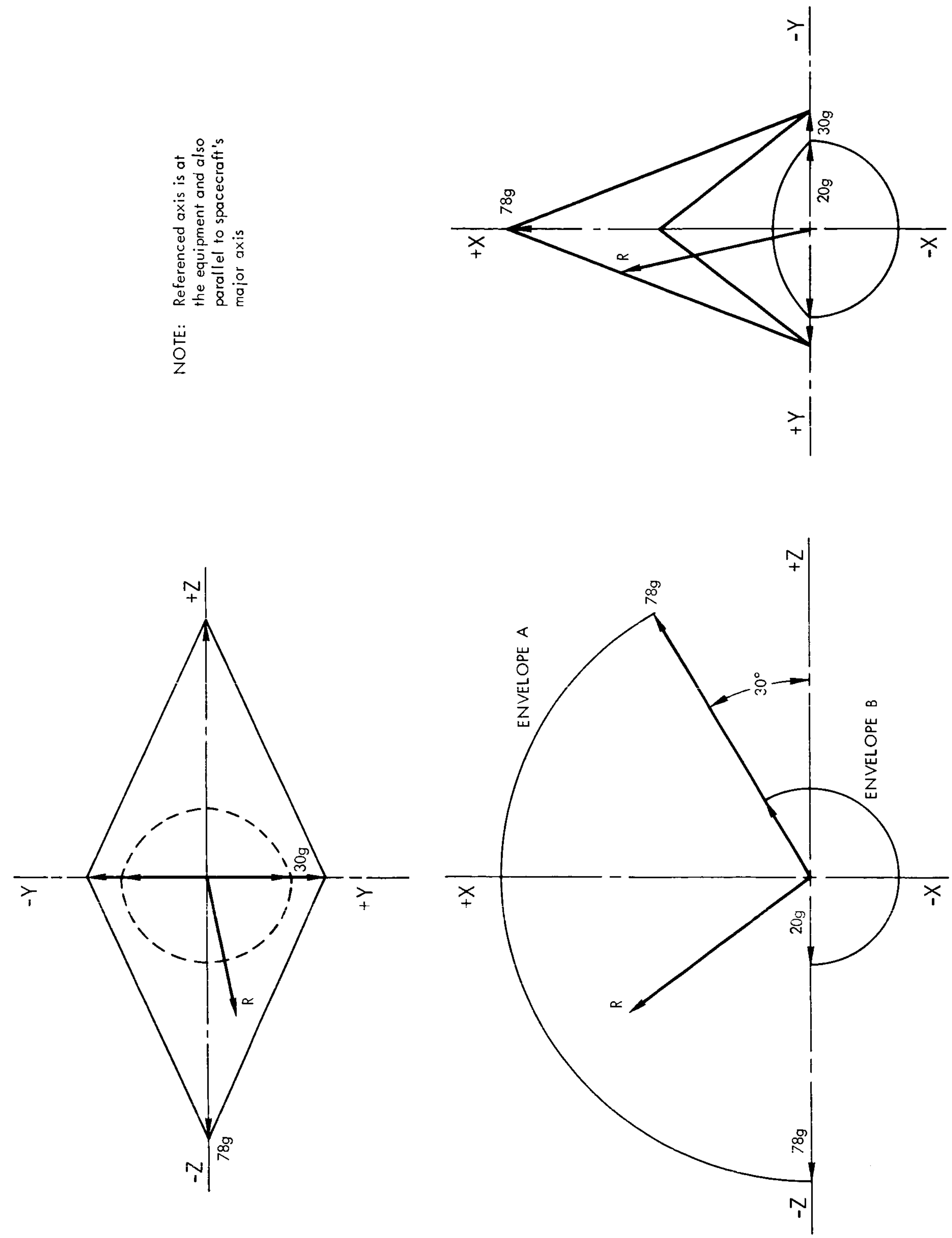

Figure 27. Internal Equipment Ultimate Design Load Diagram I

[redacted]

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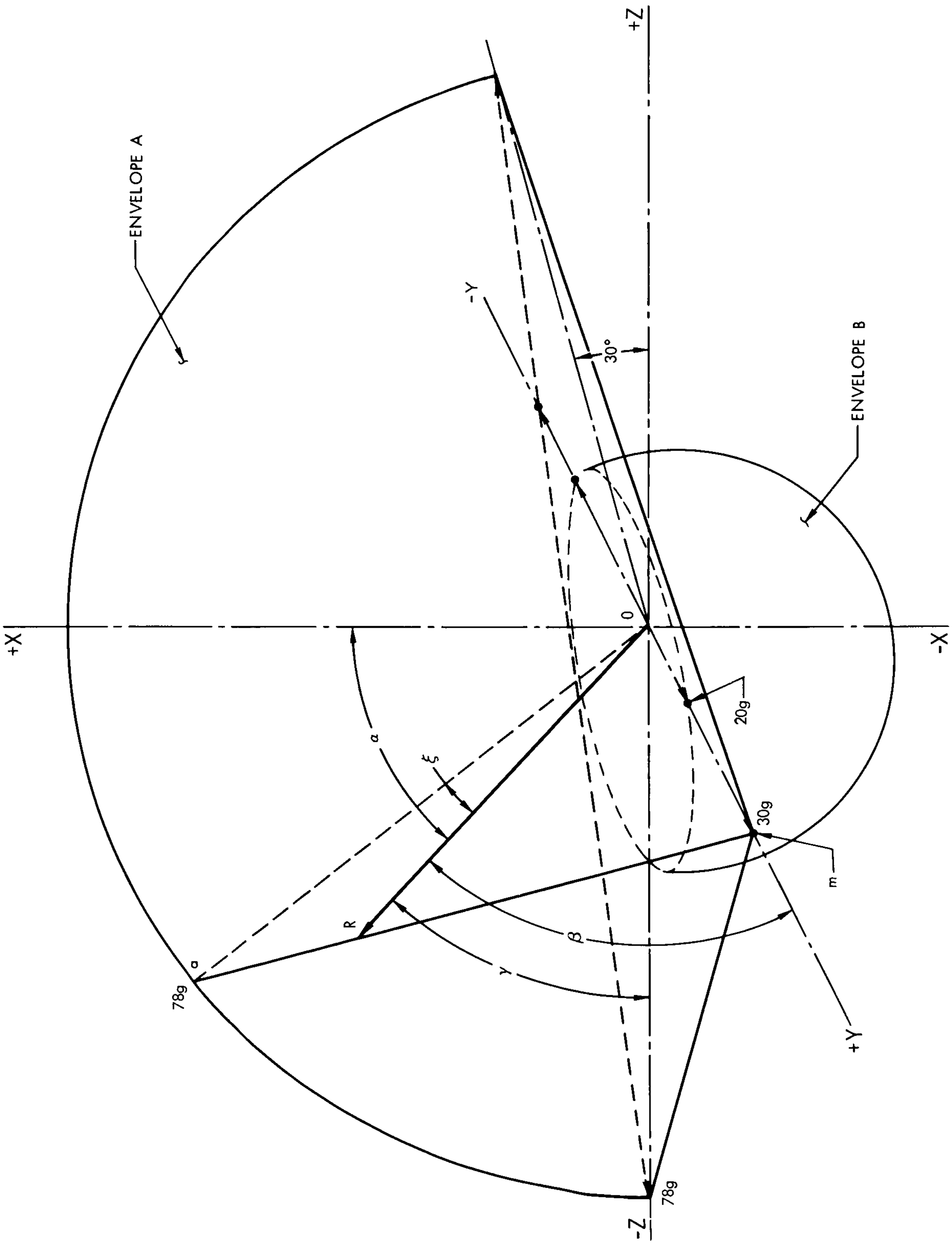

Figure 28. Internal Equipment Ultimate Design Load Diagram II

VIBRATION
LAUNCH ESCAPE SYSTEM
HIGH "Q" ABORT

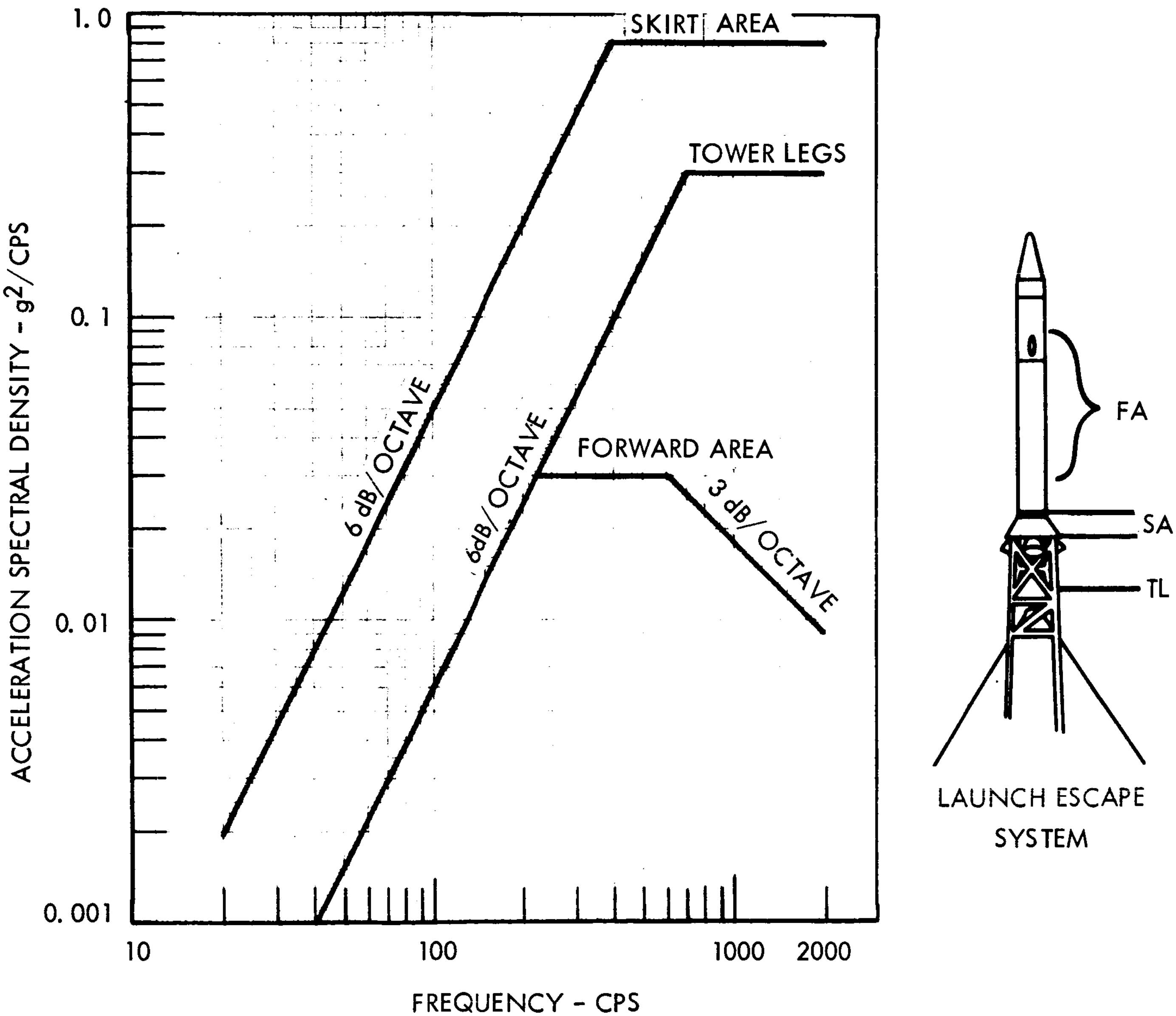

Figure 29. Vibration LES - High "Q" Abort

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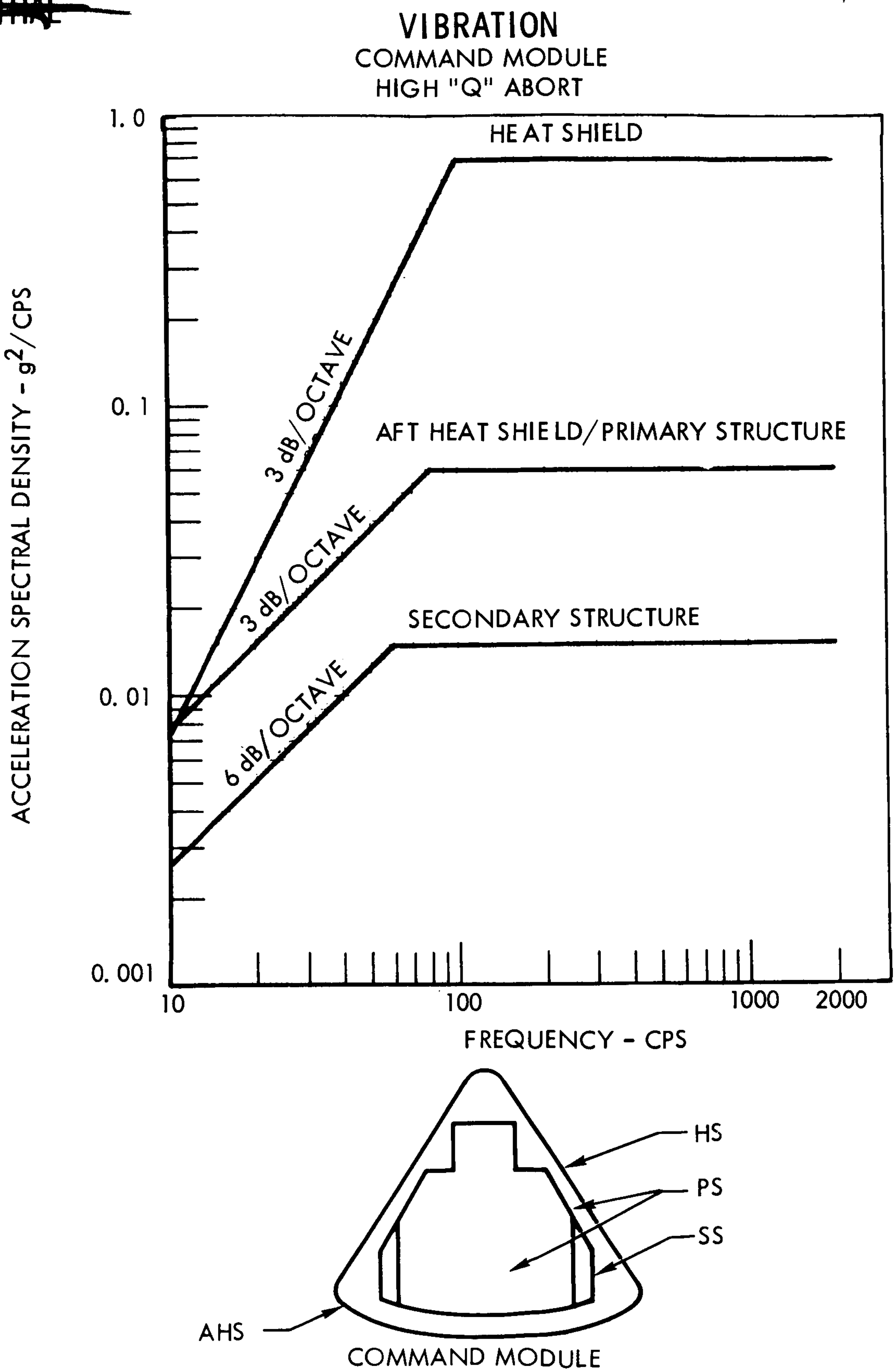

Figure 30. Vibration CM - High "Q" Abort

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ACOUSTICS
LAUNCH ESCAPE SYSTEM
HIGH "Q" ABORT

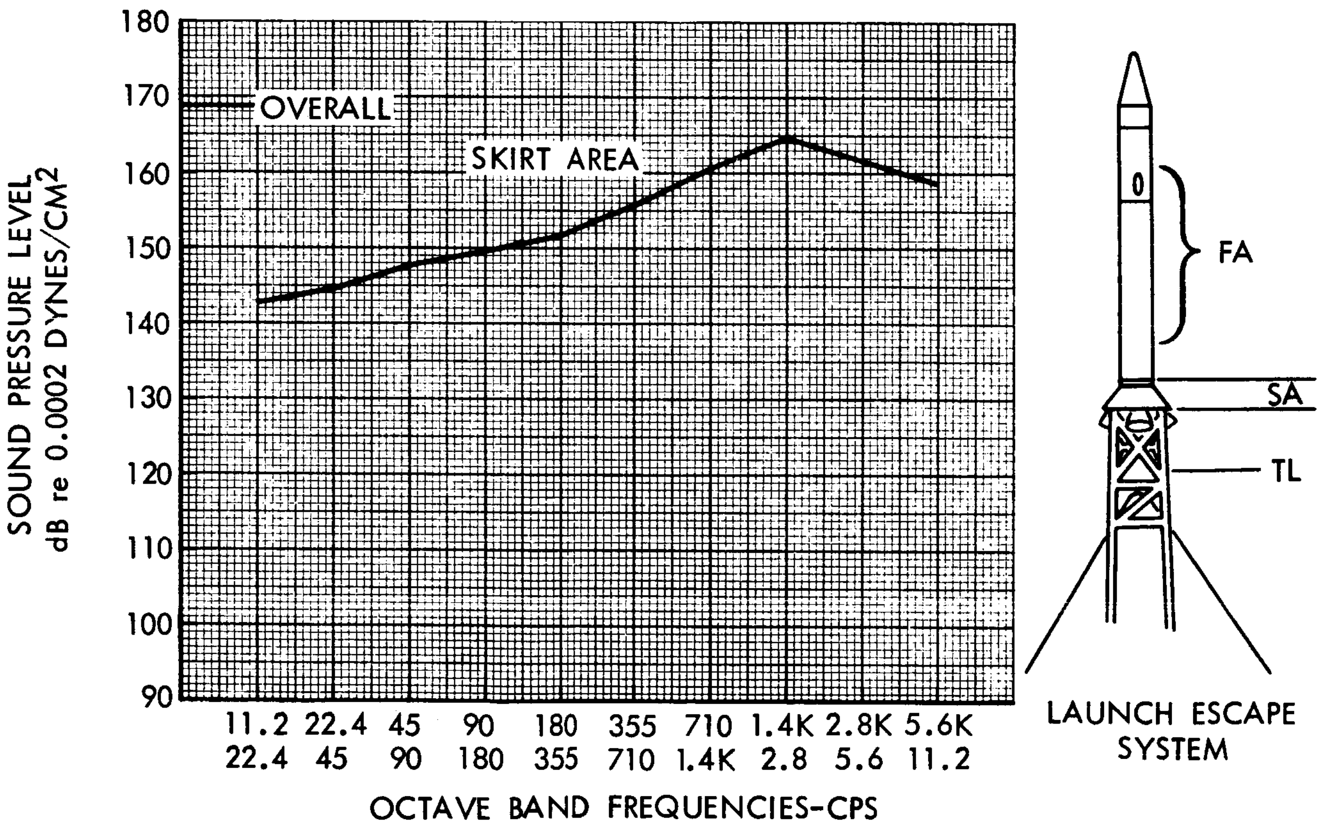


Figure 31. Acoustics LES - High "Q" Abort

~~CONFIDENTIAL~~

ACOUSTICS
COMMAND MODULE
HIGH "Q" ABORT

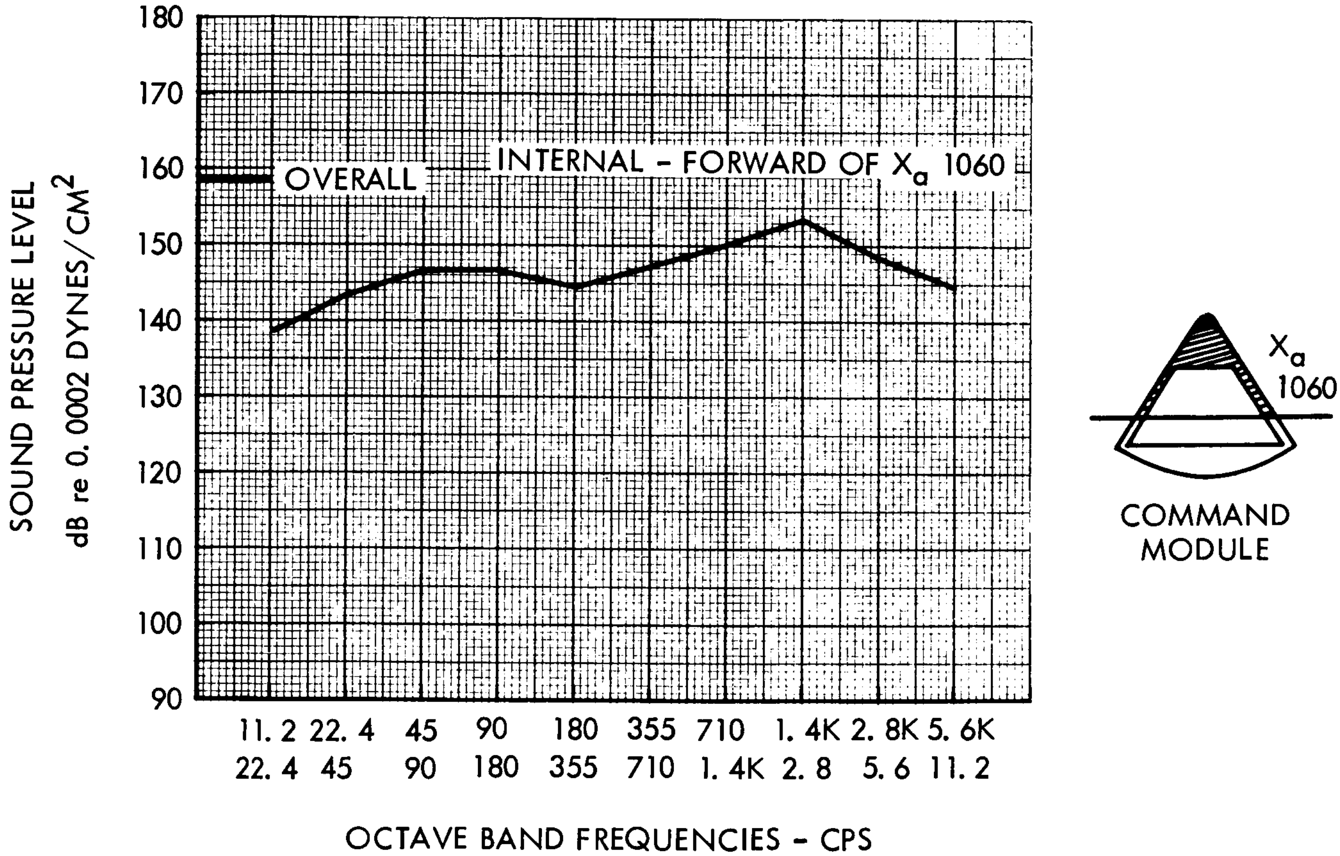
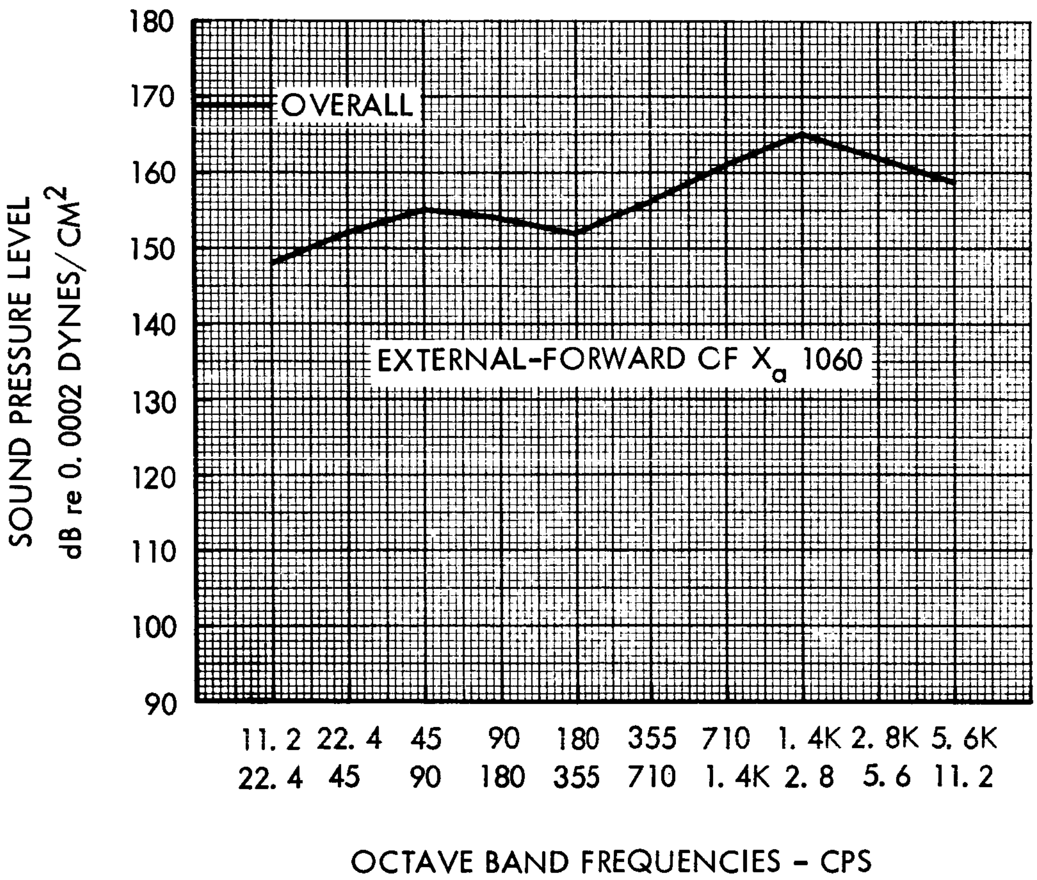


Figure 32. Acoustics CM - High "Q" Abort - Internal - Forward Xa1060

CON[redacted]

ACOUSTICS

COMMAND MODULE
HIGH "Q" ABORT



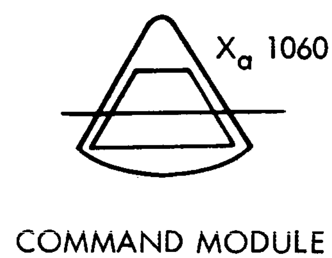
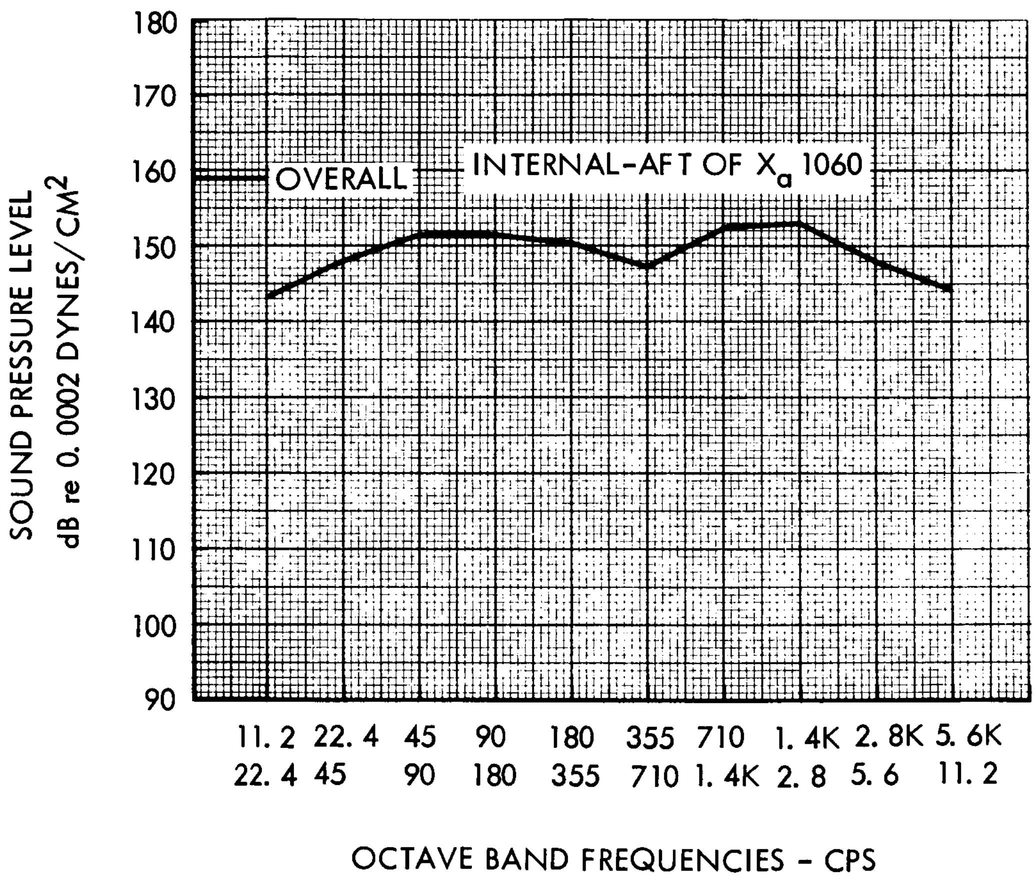


Figure 33. Acoustics CM - High "Q" Abort - External - Forward of X1060

ACOUSTICS

COMMAND MODULE
HIGH "Q" ABORT



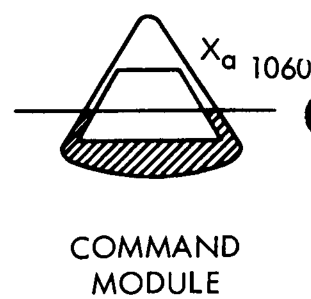


Figure 34. Acoustics CM - High "Q" Abort - Internal - Aft of Xa1060

ACOUSTICS
COMMAND MODULE
HIGH "Q" ABORT

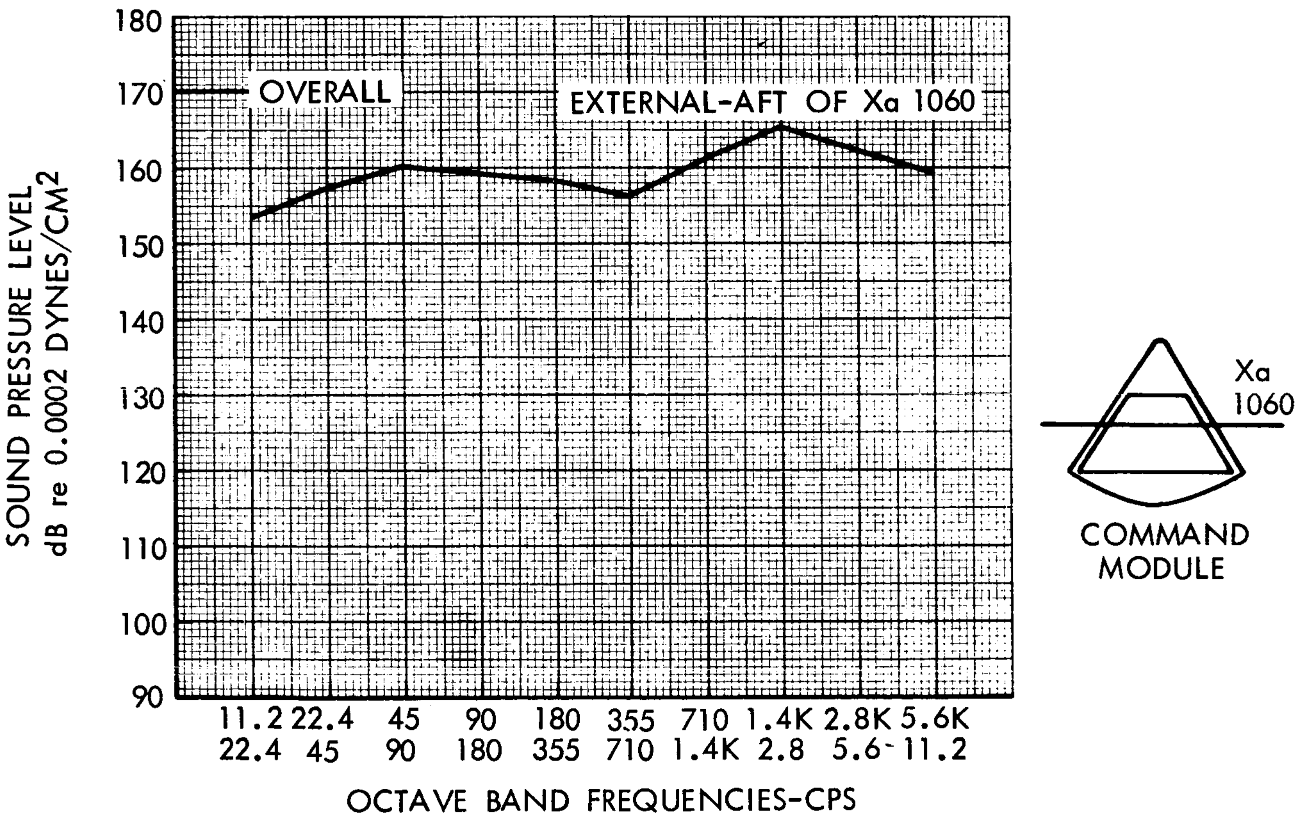


Figure 35. Acoustics CM - High "Q" Abort - External - Aft of Xa1060

~~CONFIDENTIAL~~

ACOUSTICS
COMMAND MODULE
HIGH "Q" ABORT

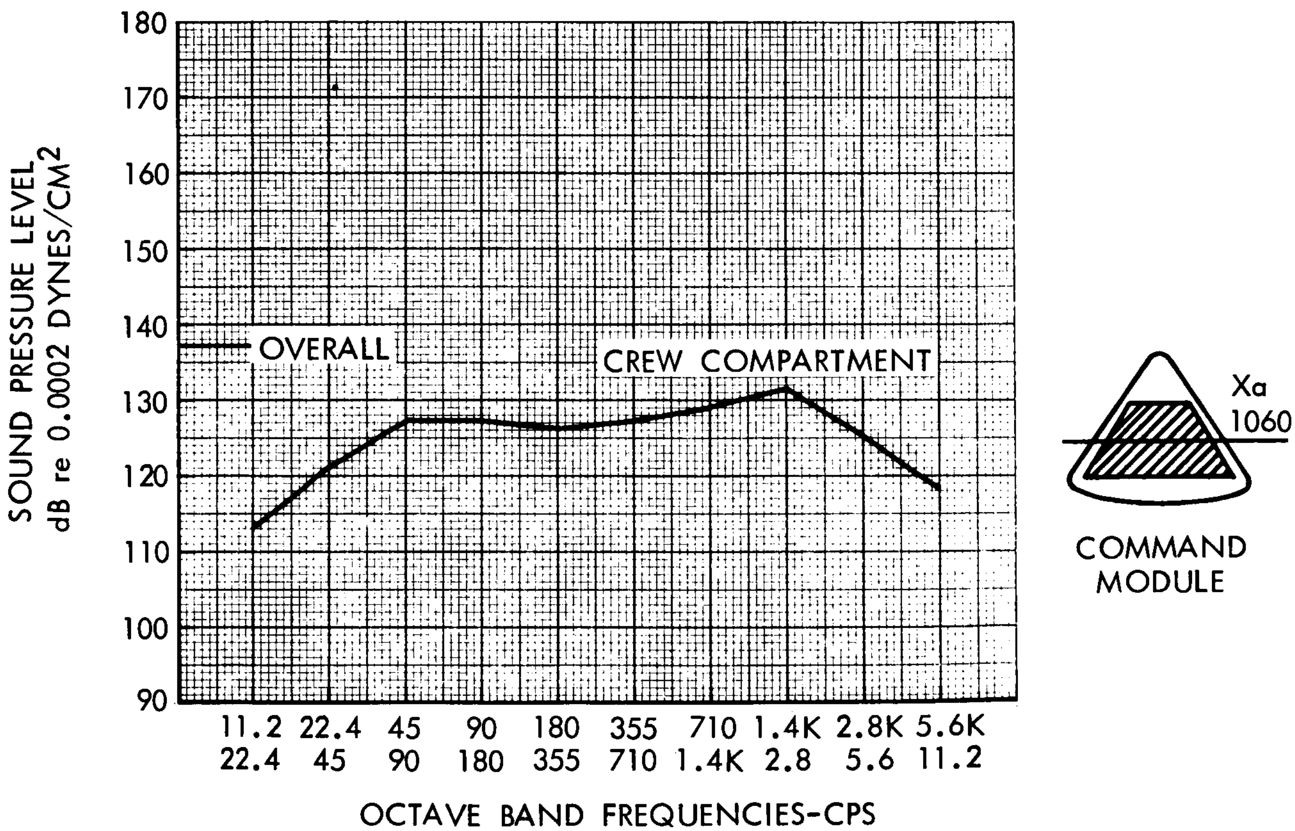


Figure 36. Acoustics CM - High "Q" Abort - Crew Compartment

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CON[redacted]

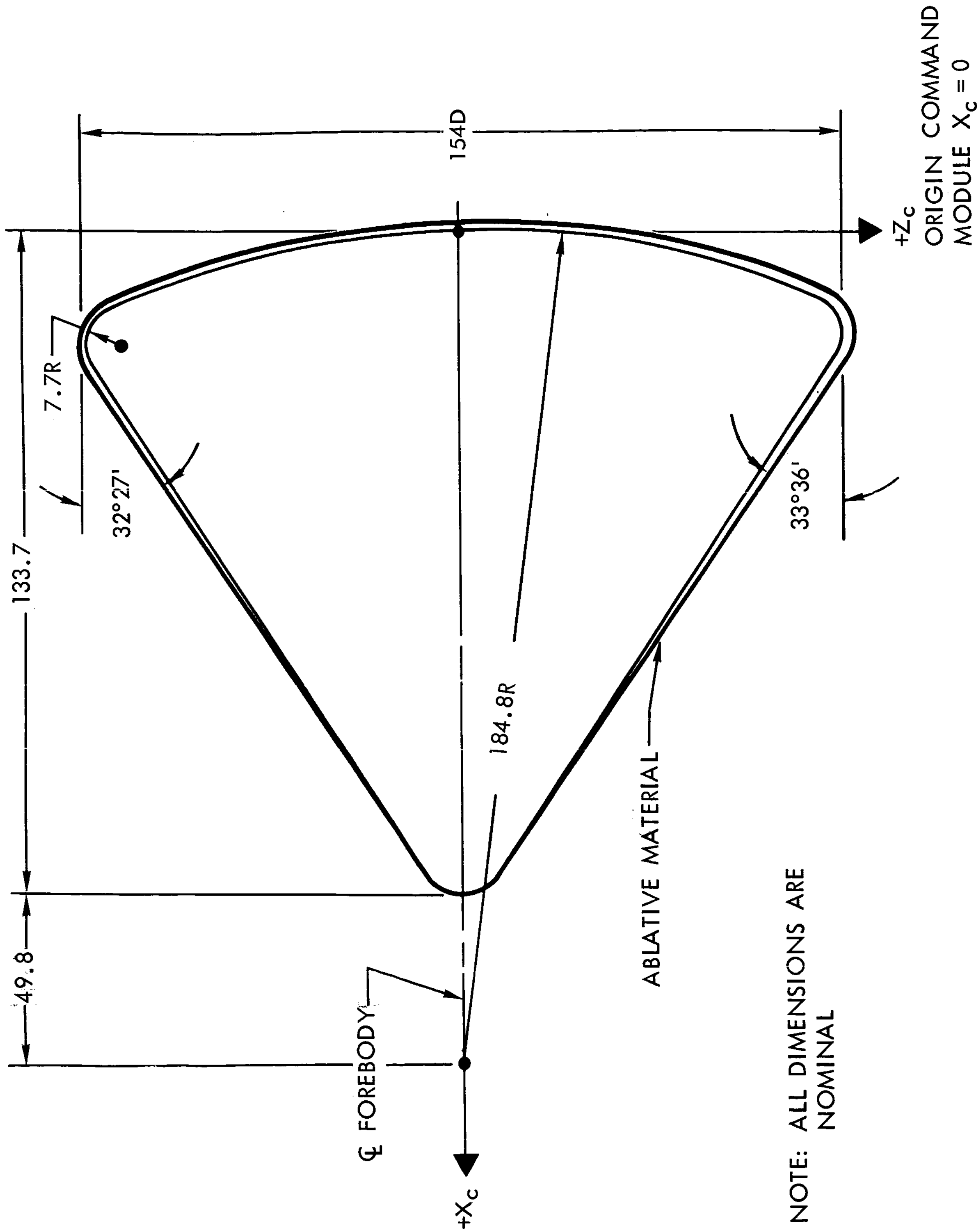

Figure 37. Command Module External Dimensions Including Ablative Material

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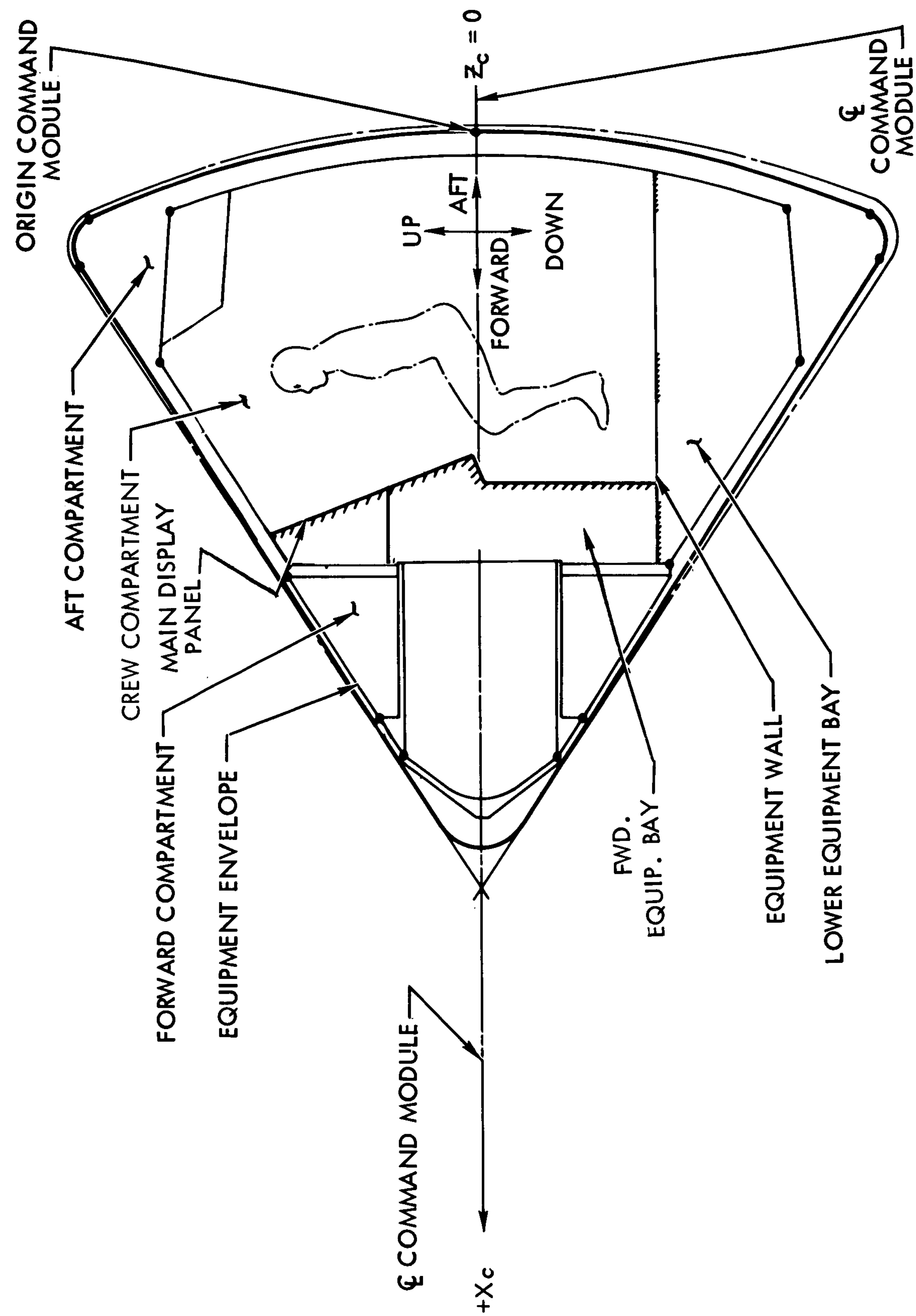


Figure 38. Area Designations - Side View

~~CONFIDENTIAL~~

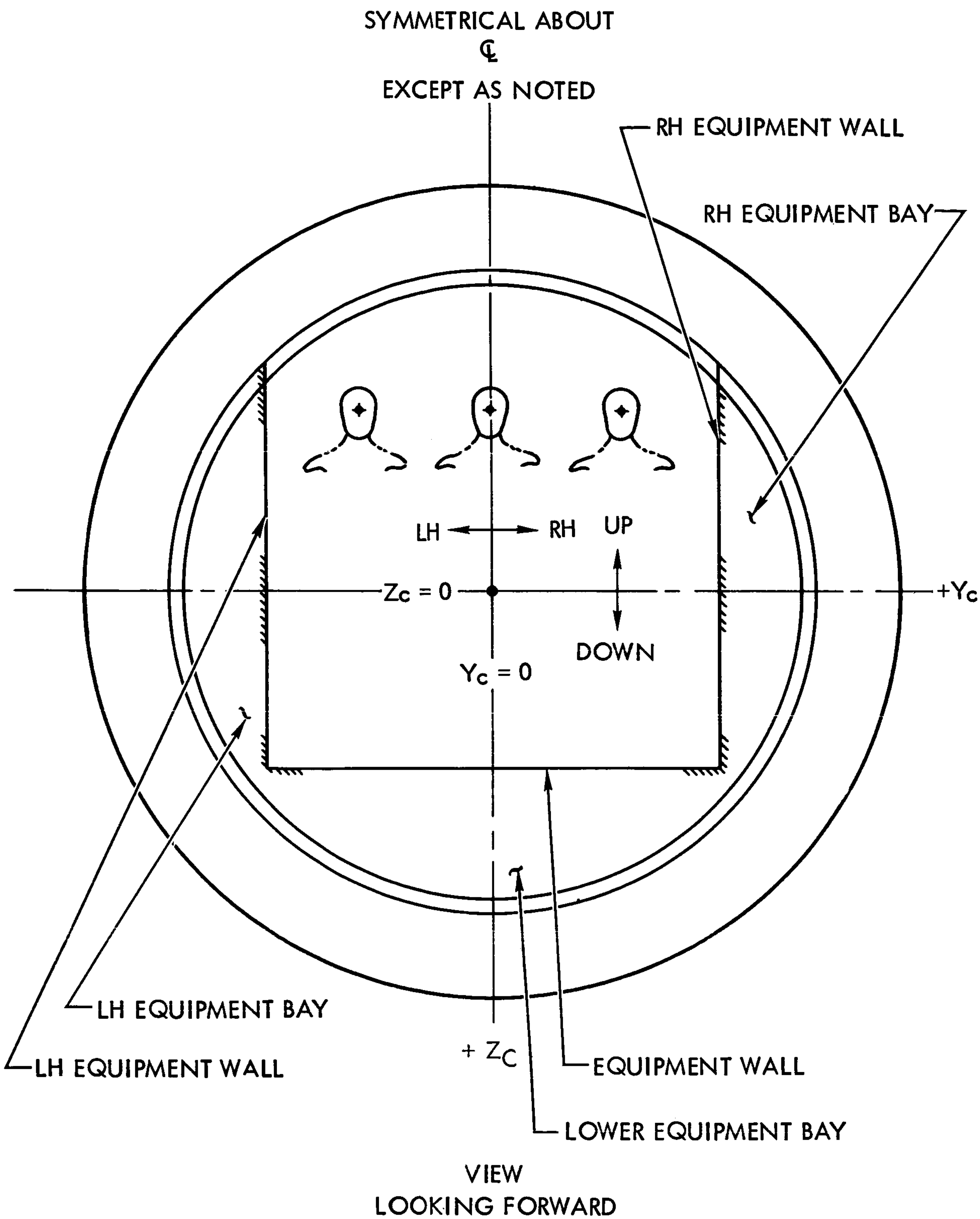
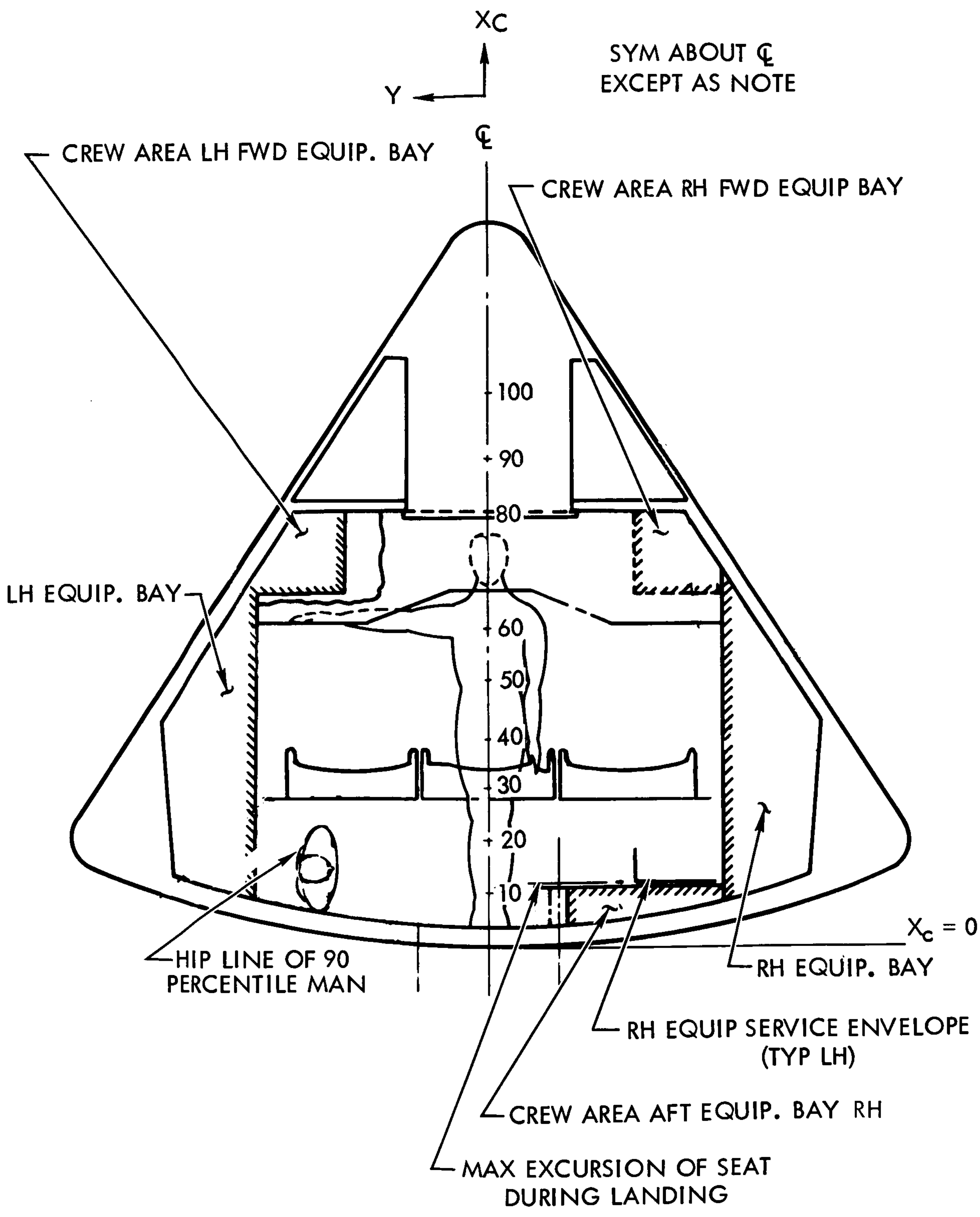


Figure 39. Area Designations - View Looking Forward

~~CONFIDENTIAL~~



VIEW LOOKING TOWARD LOWER END OF CREW AREA

Figure 40. Area Designations - View Looking Toward Lower End of Crew Area

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CON[redacted]

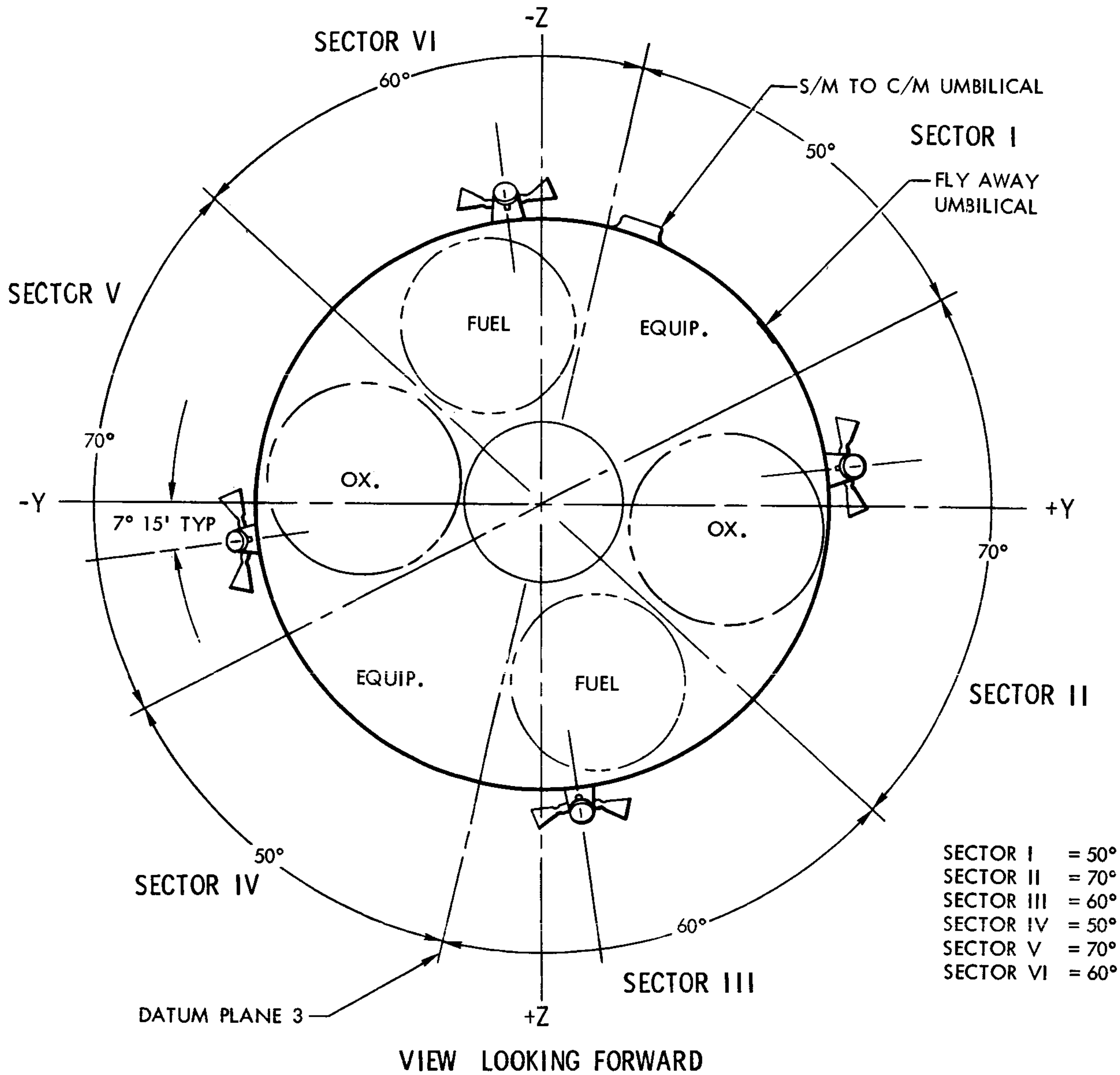

Figure 41. Service Module Inboard Profile - View Looking Forward

[redacted]

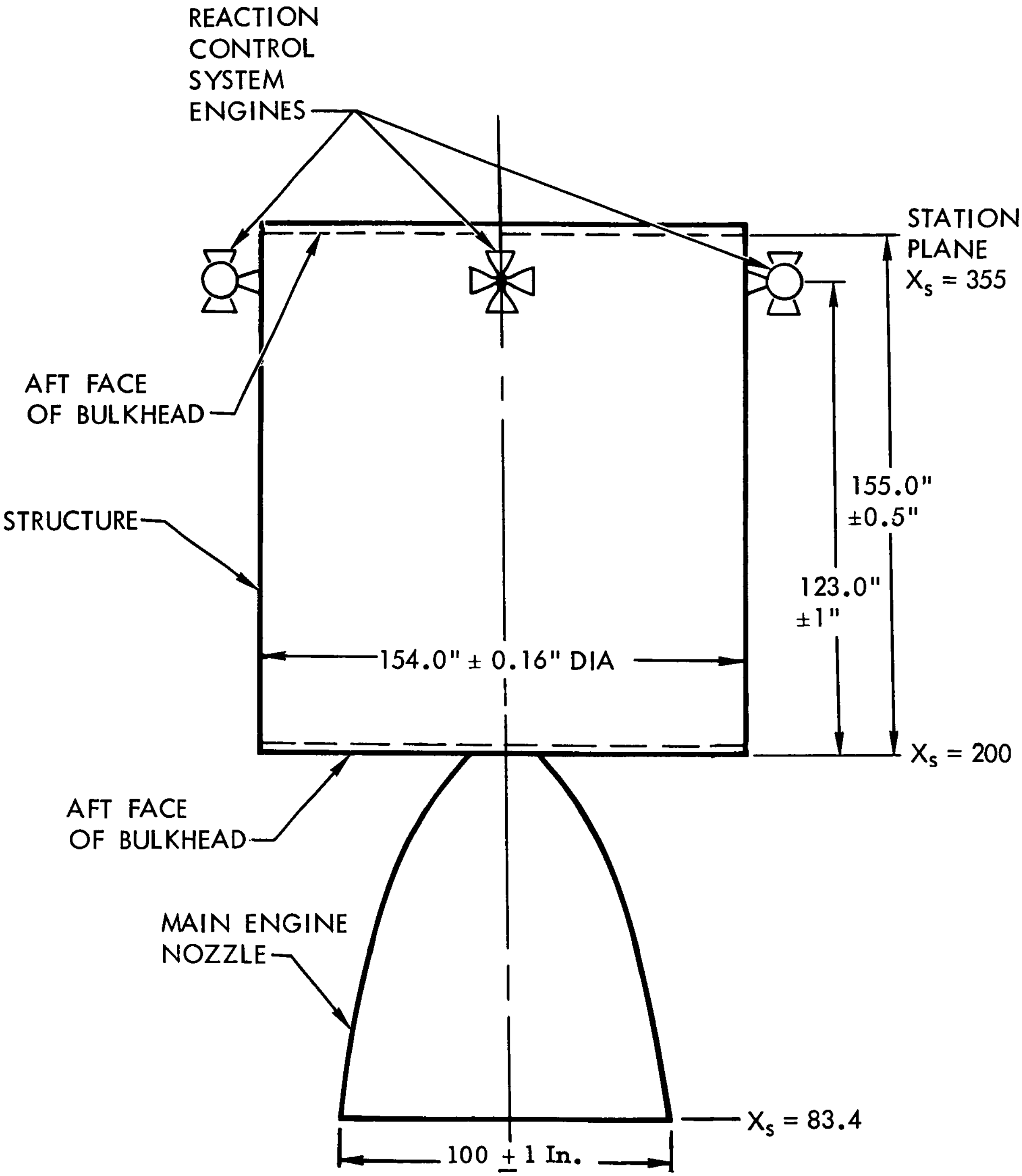

Figure 42. Service Module Inboard Profile

CON[redacted]

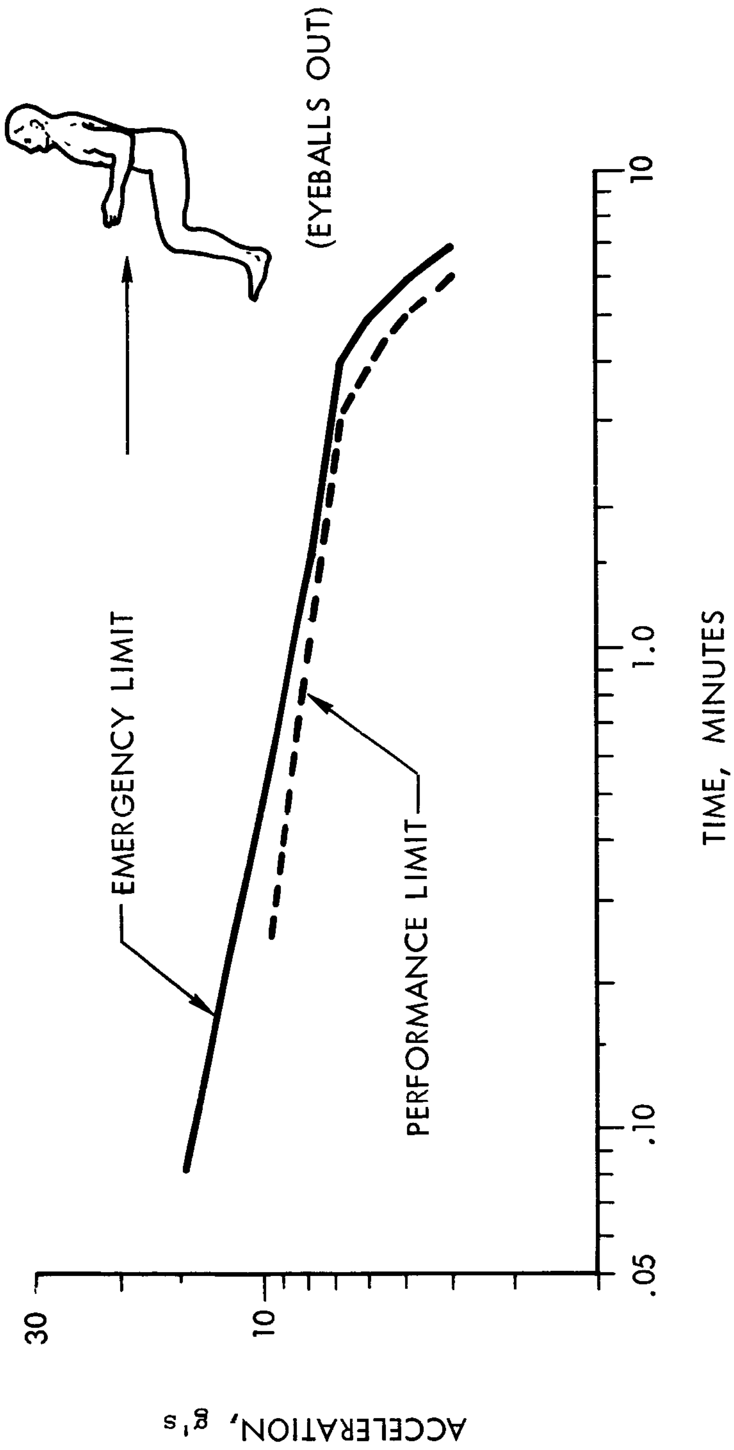

Figure 43. Sustained Acceleration - Eyeballs Out

[redacted]

[redacted]

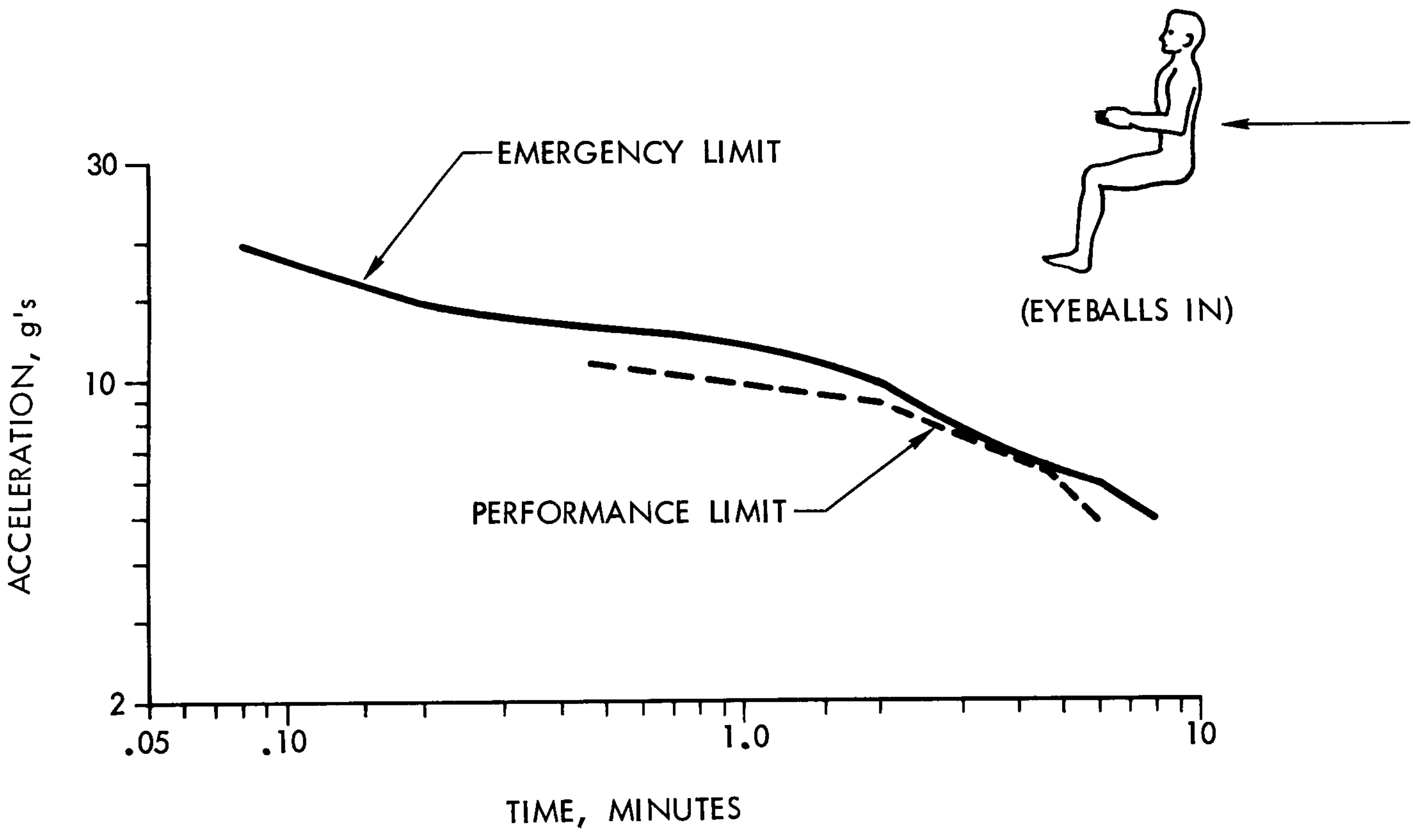

Figure 44. Sustained Acceleration - Eyeballs In

[redacted]AL

CON[redacted]

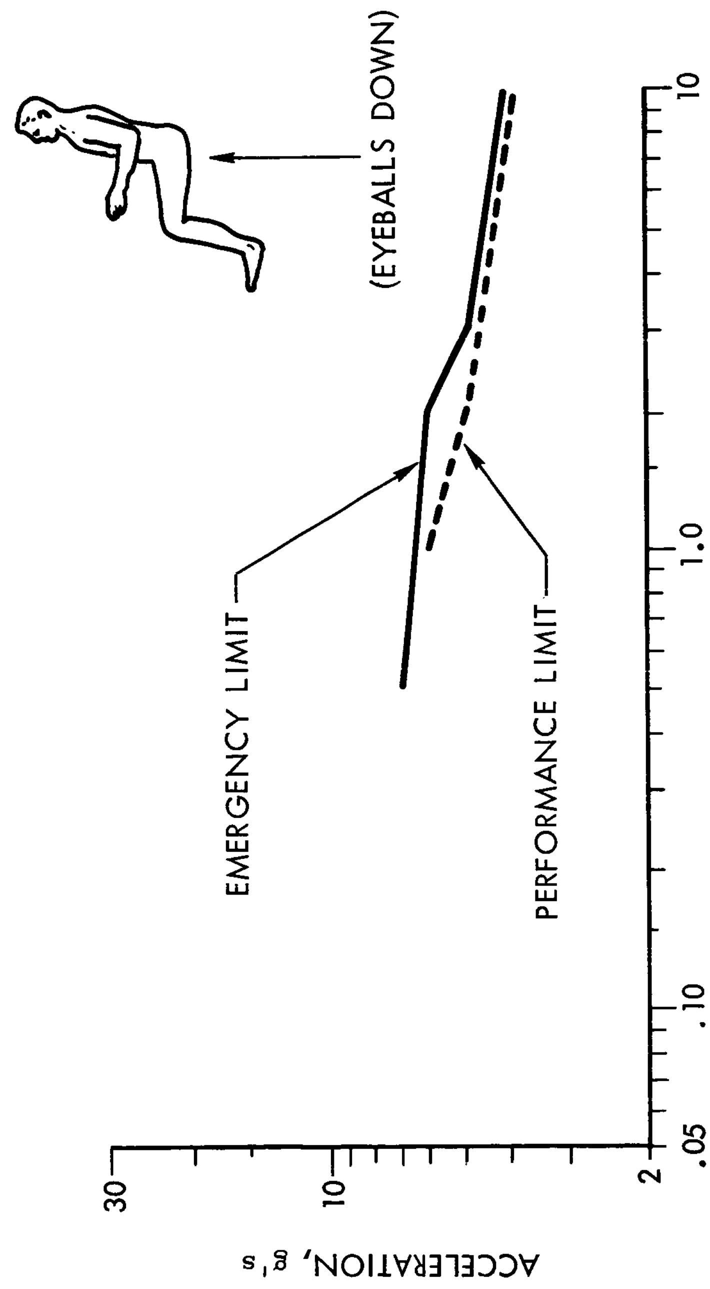

Figure 45. Sustained Acceleration - Eyeballs Down

[redacted]

[redacted]

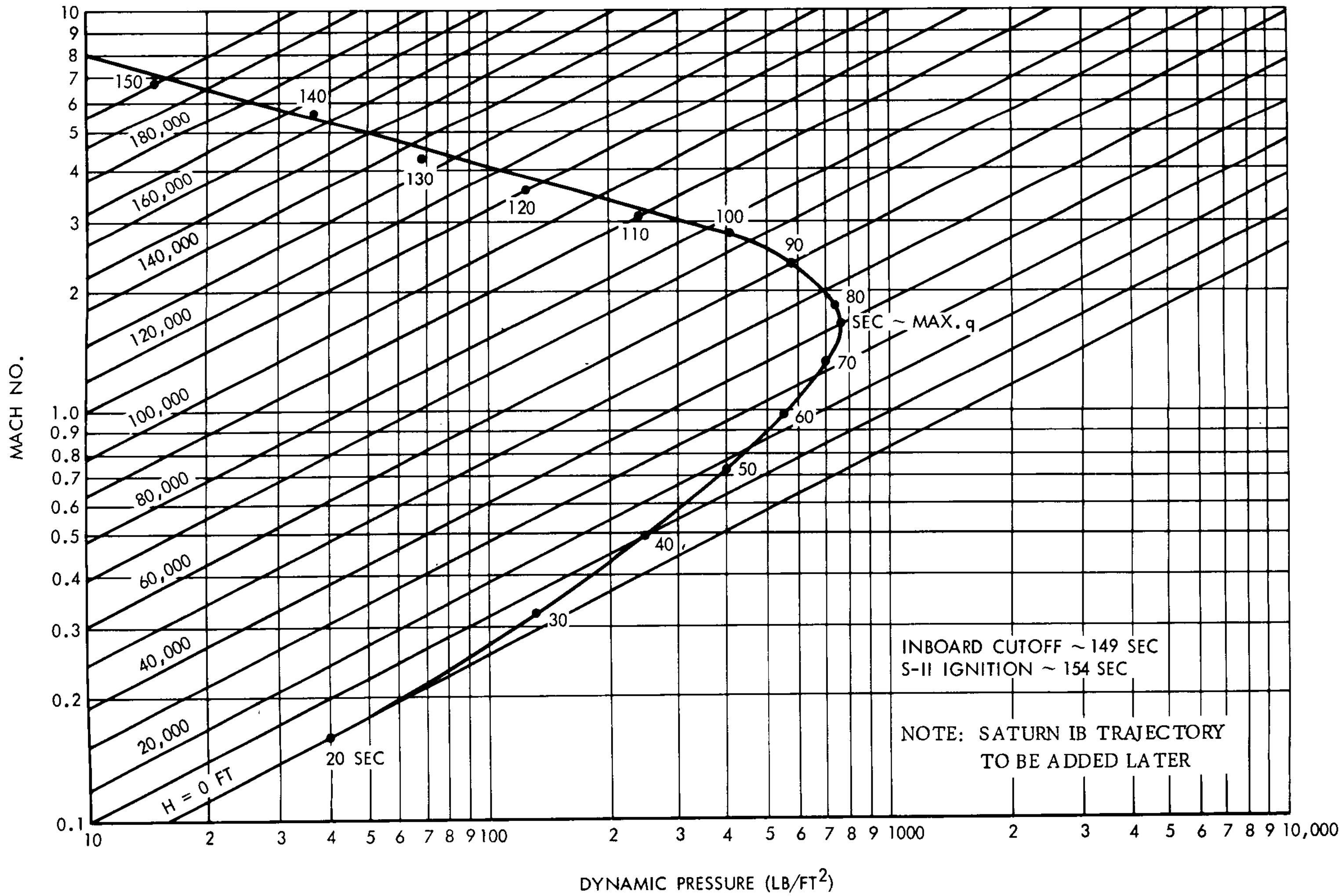

Figure 46. Saturn V Two-Stage Boost Trajectory - 100 NM Orbit

[redacted]

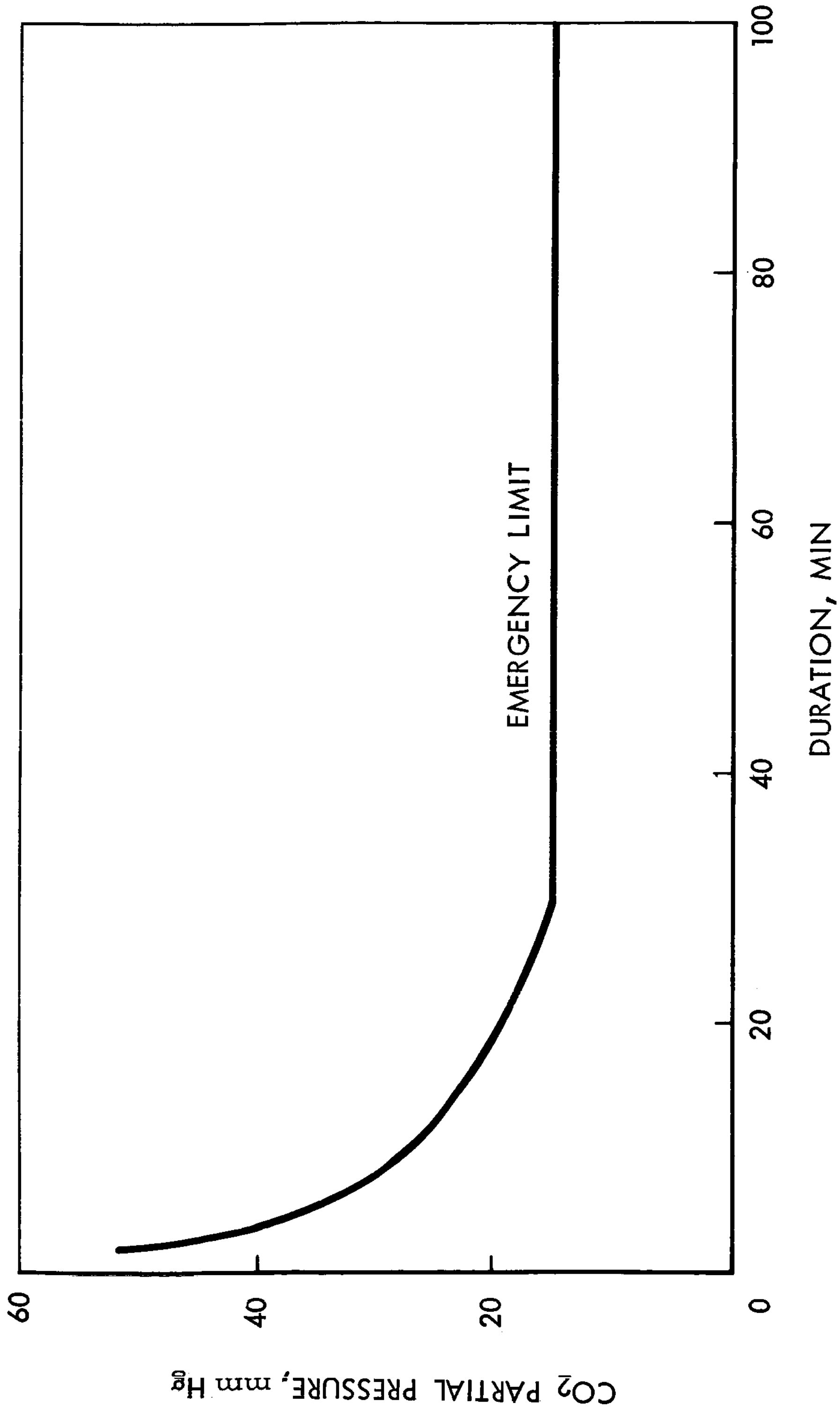

Figure 47. Emergency Carbon Dioxide Limit

[redacted]

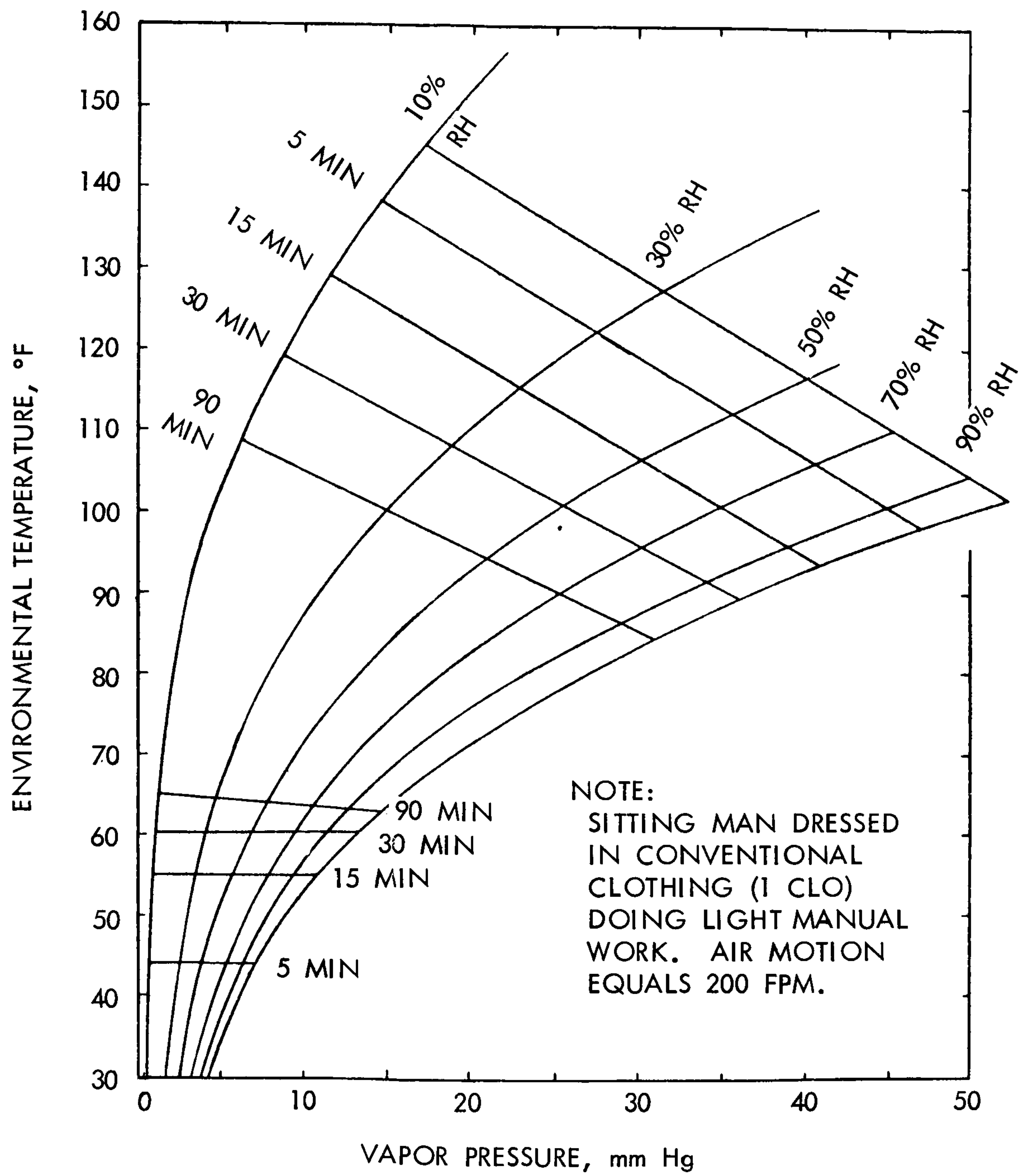

Figure 48. Temperature and Humidity Nominal Limit

C[redacted]

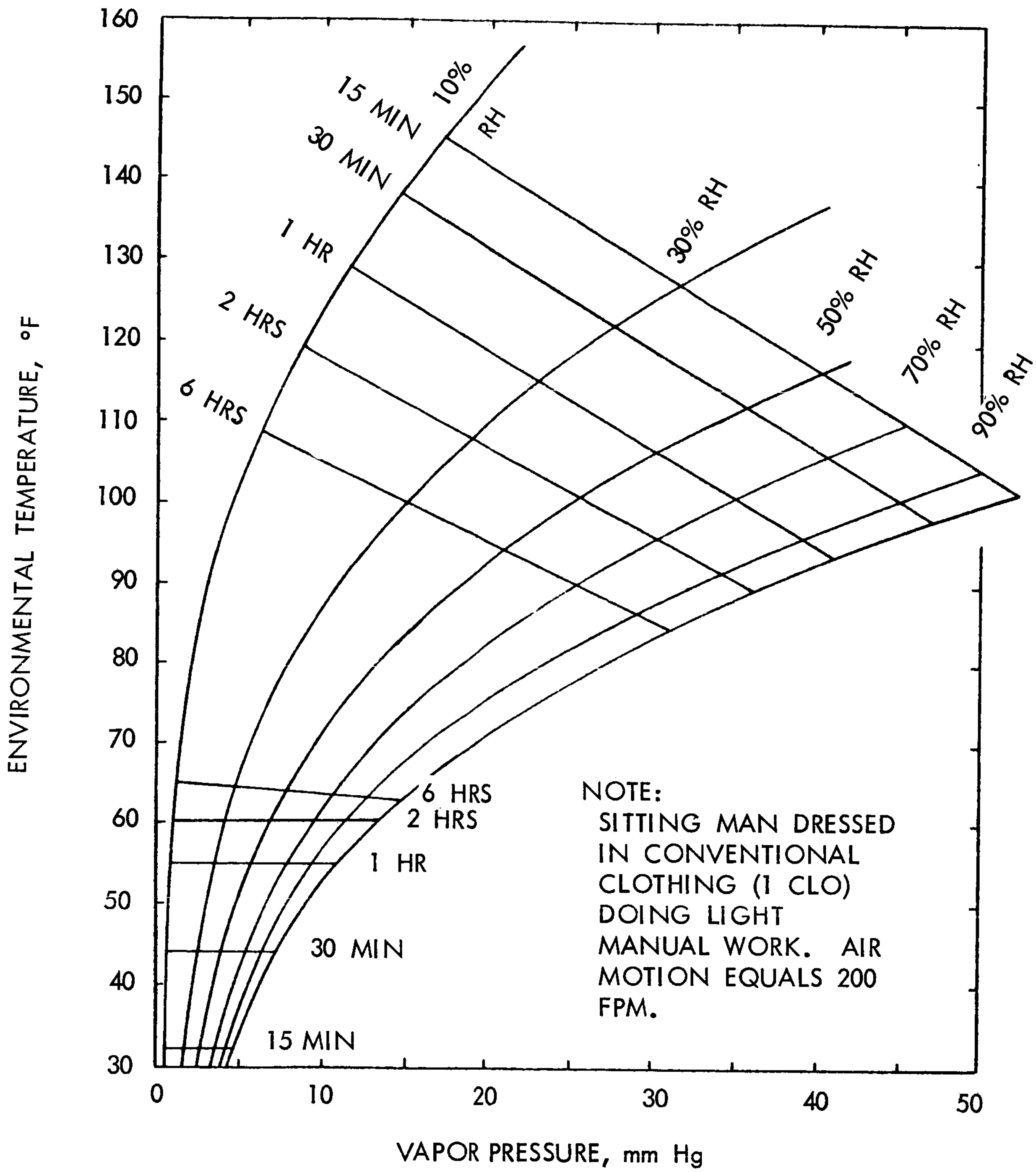

Figure 49. Temperature and Humidity Emergency Limit

[redacted]

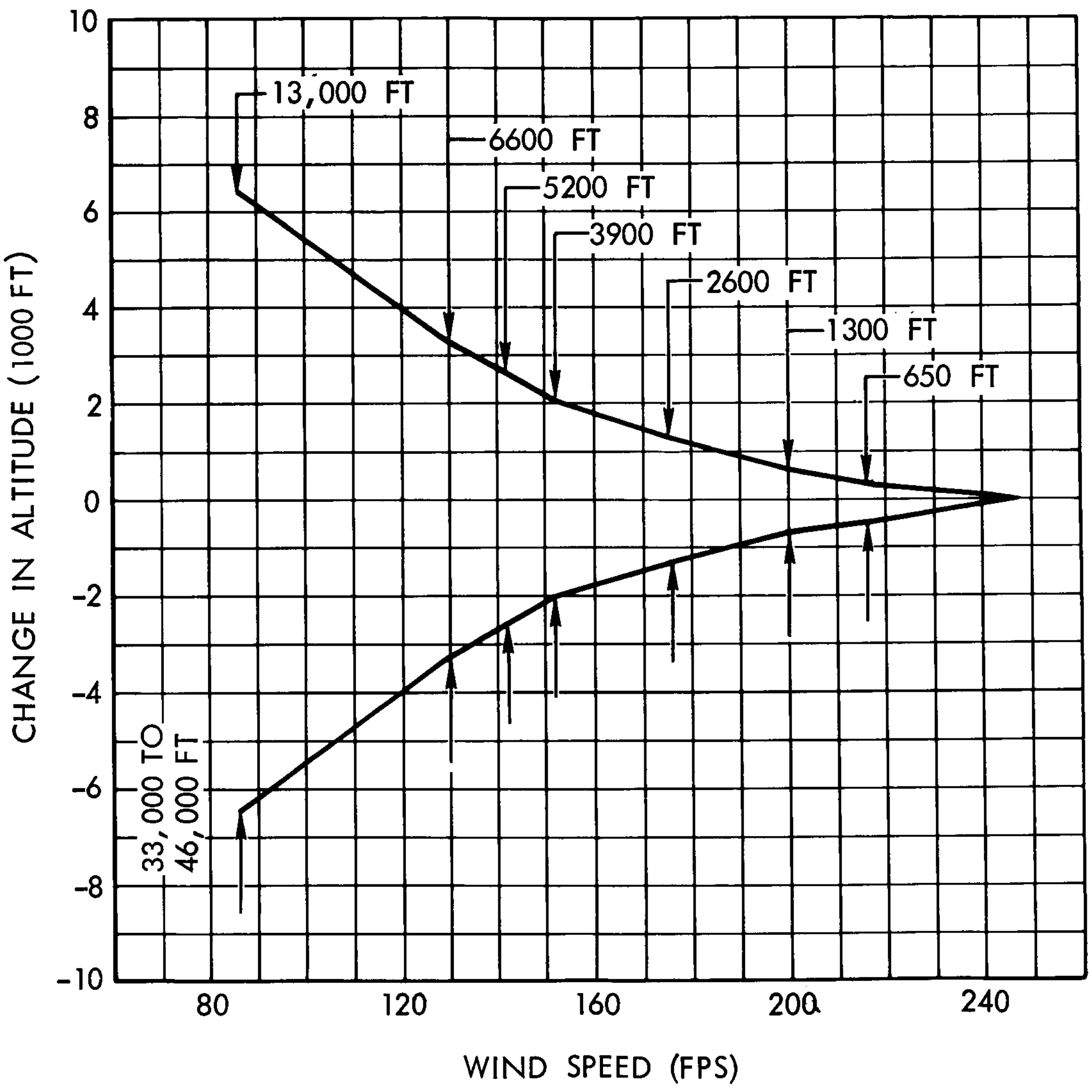

Figure 50. Wind Shear Profile

CON[redacted]

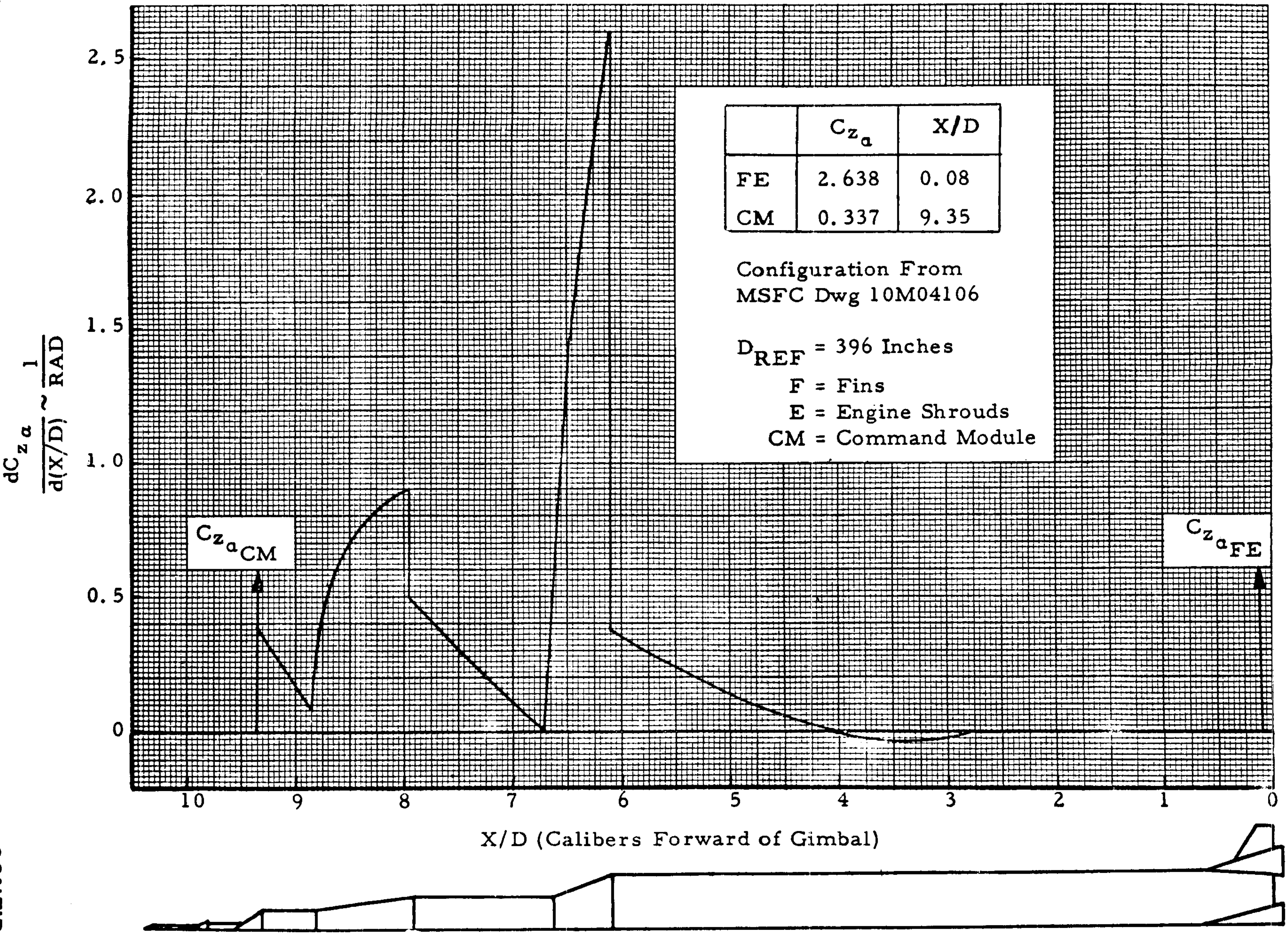

Figure 51. Linear Load Distribution for Mach Numbers 1.35

[redacted]

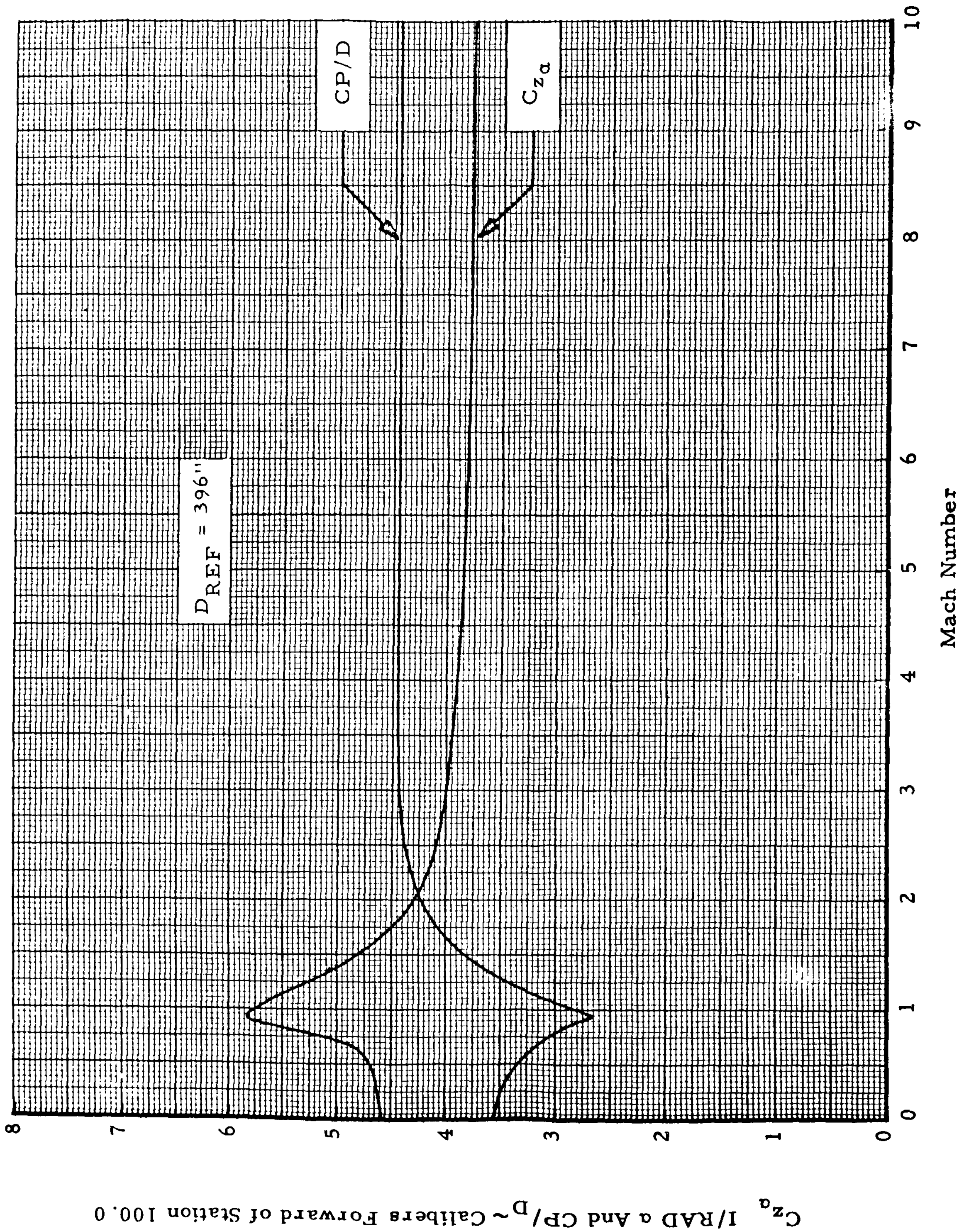

Figure 52. Gradient of Normal Force Coefficient and Center of Pressure Versus Mach Numbers

CO[redacted]

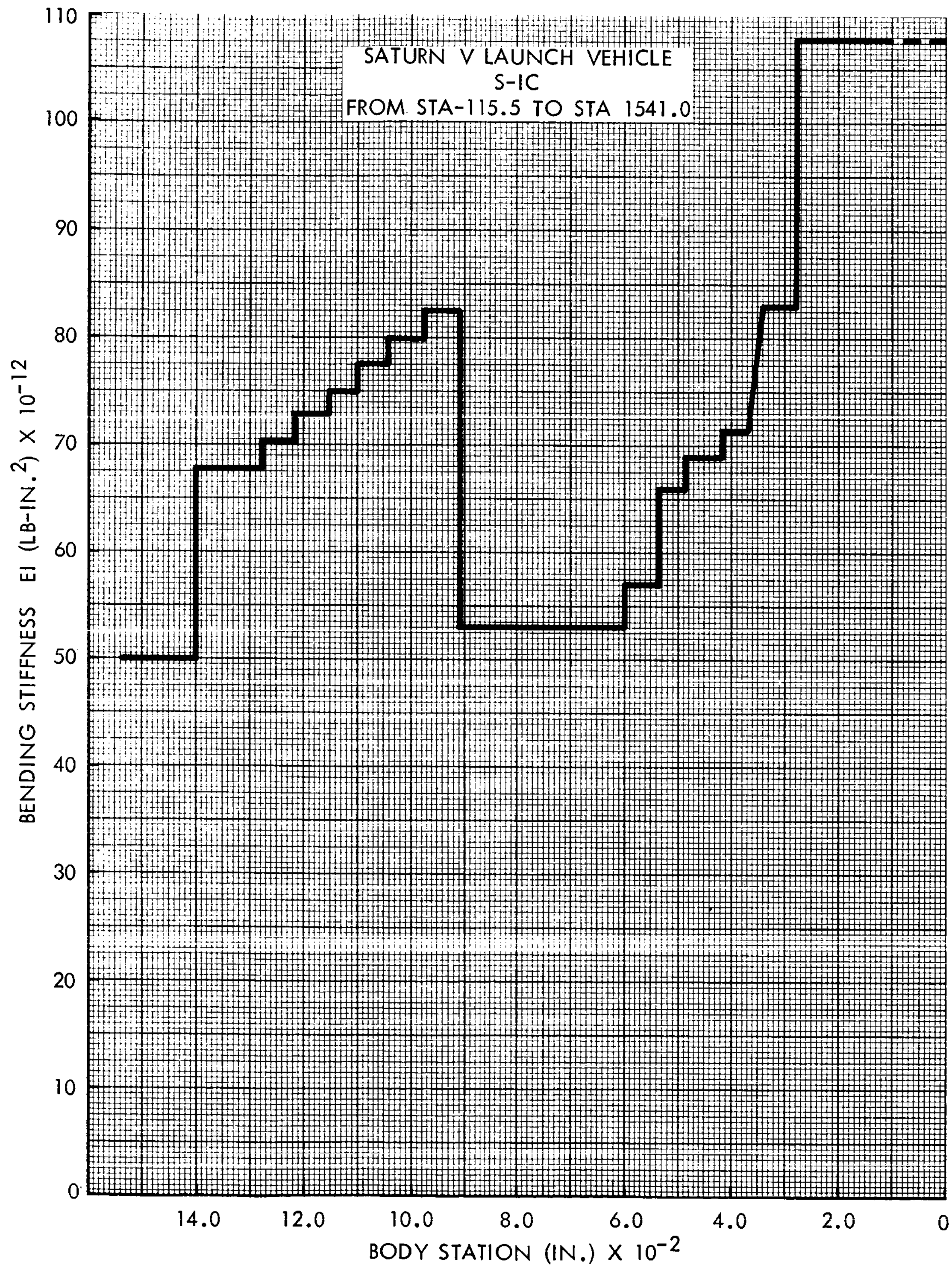

Figure 53. Saturn V, S-IC, EI Distribution

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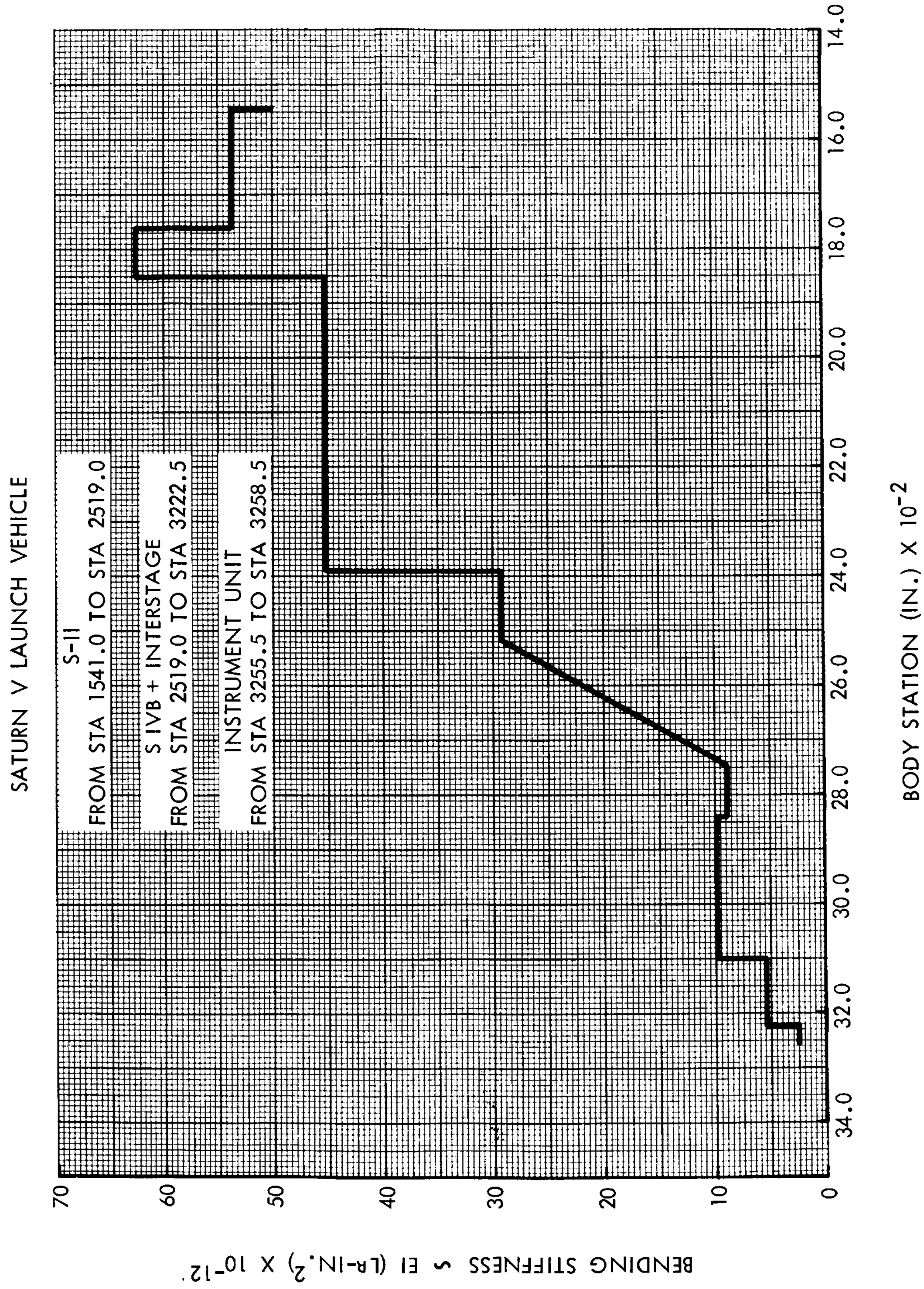


Figure 54. Saturn V, S-II, EI Distribution

Figure 54. Saturn V, SSI, EI Distribution

~~CONFIDENTIAL~~

CO[redacted]

SATURN V LAUNCH VEHICLE

S-IC
FROM STA-115.5 TO STA 1541.0

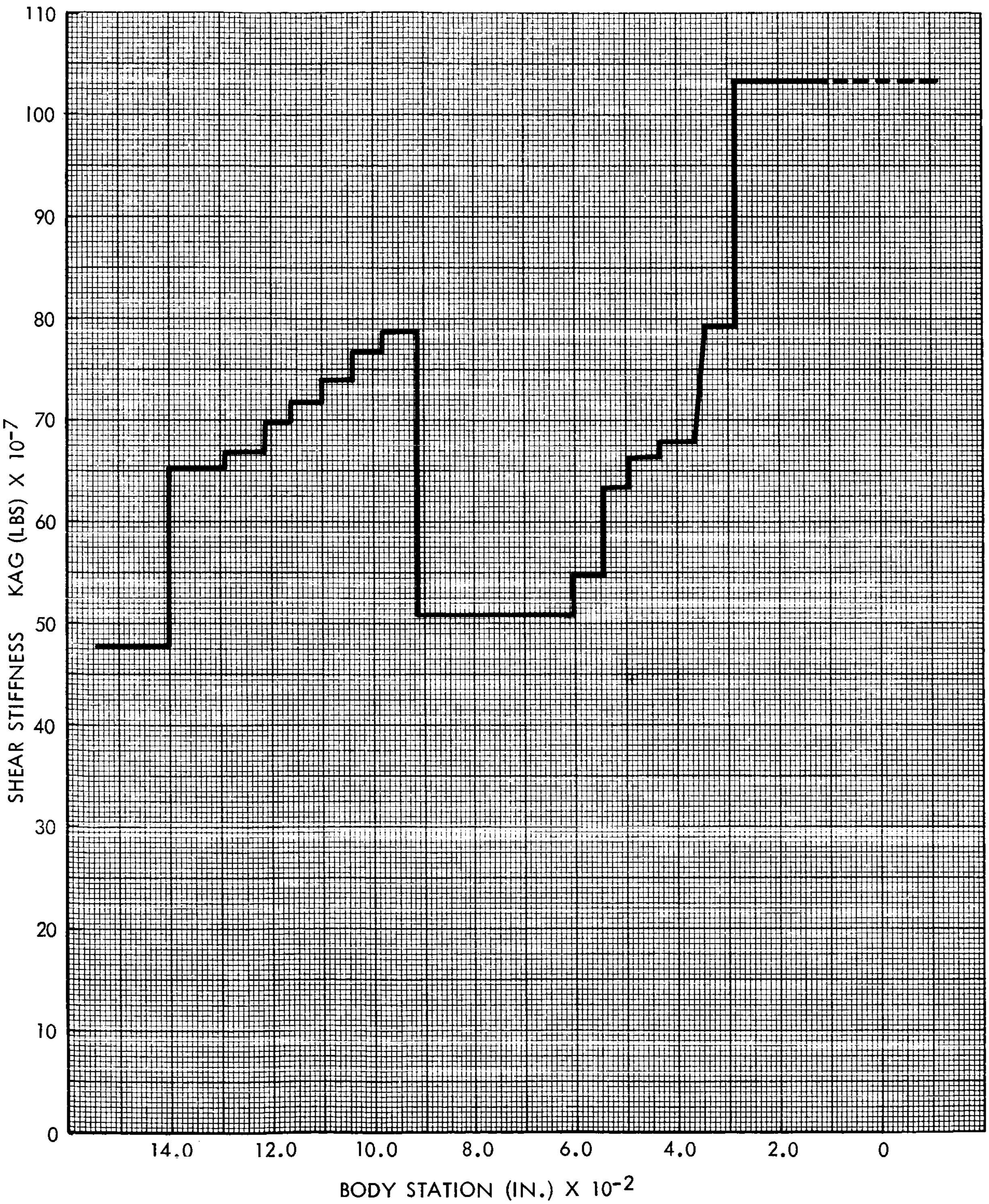


Figure 55. Saturn V, S-IC, KAG Distribution

~~CONFIDENTIAL~~

~~CONFIDENTIAL~~

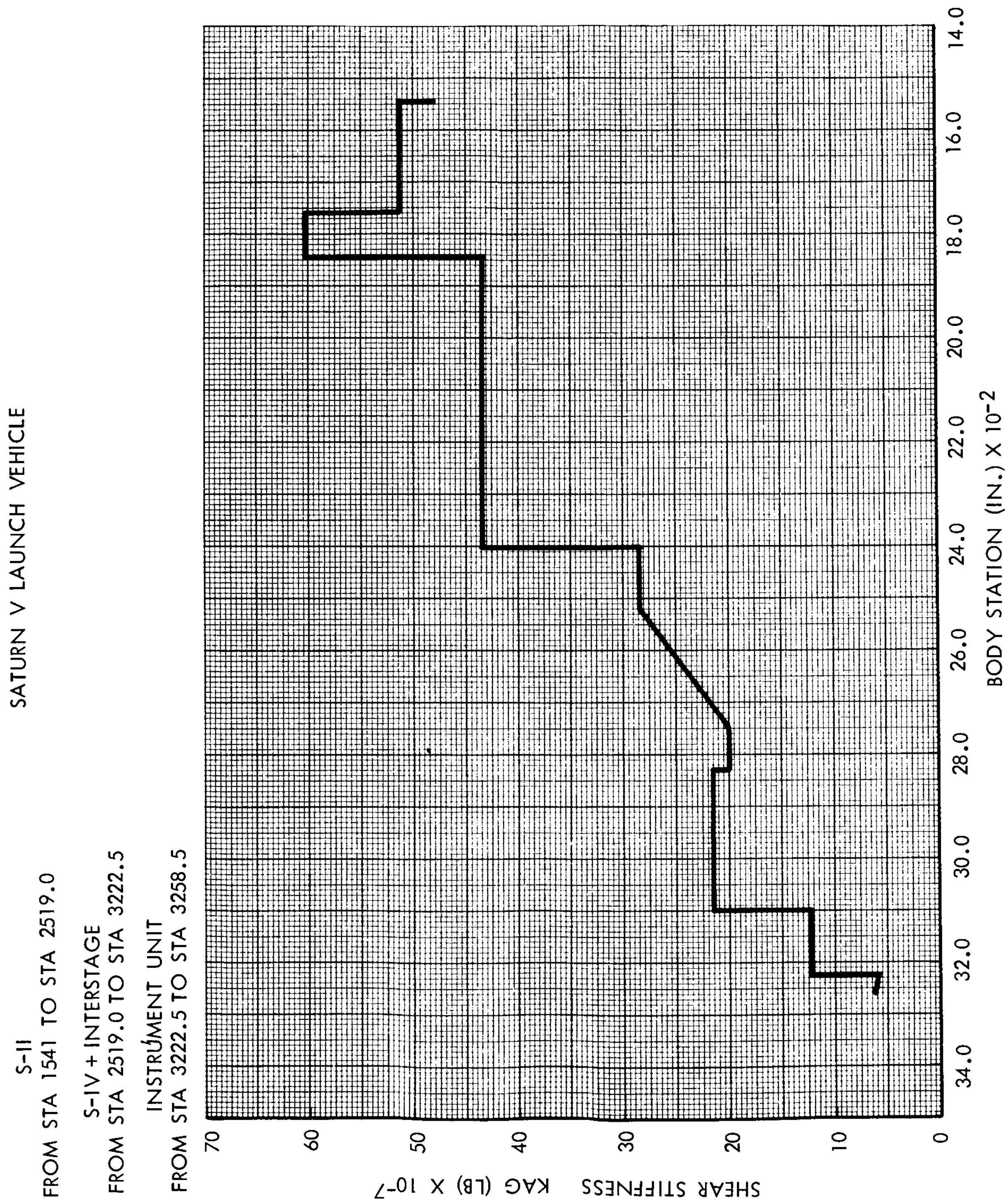


Figure 56. Saturn V, S-II, KAG Distribution

~~CONFIDENTIAL~~

CON[redacted]

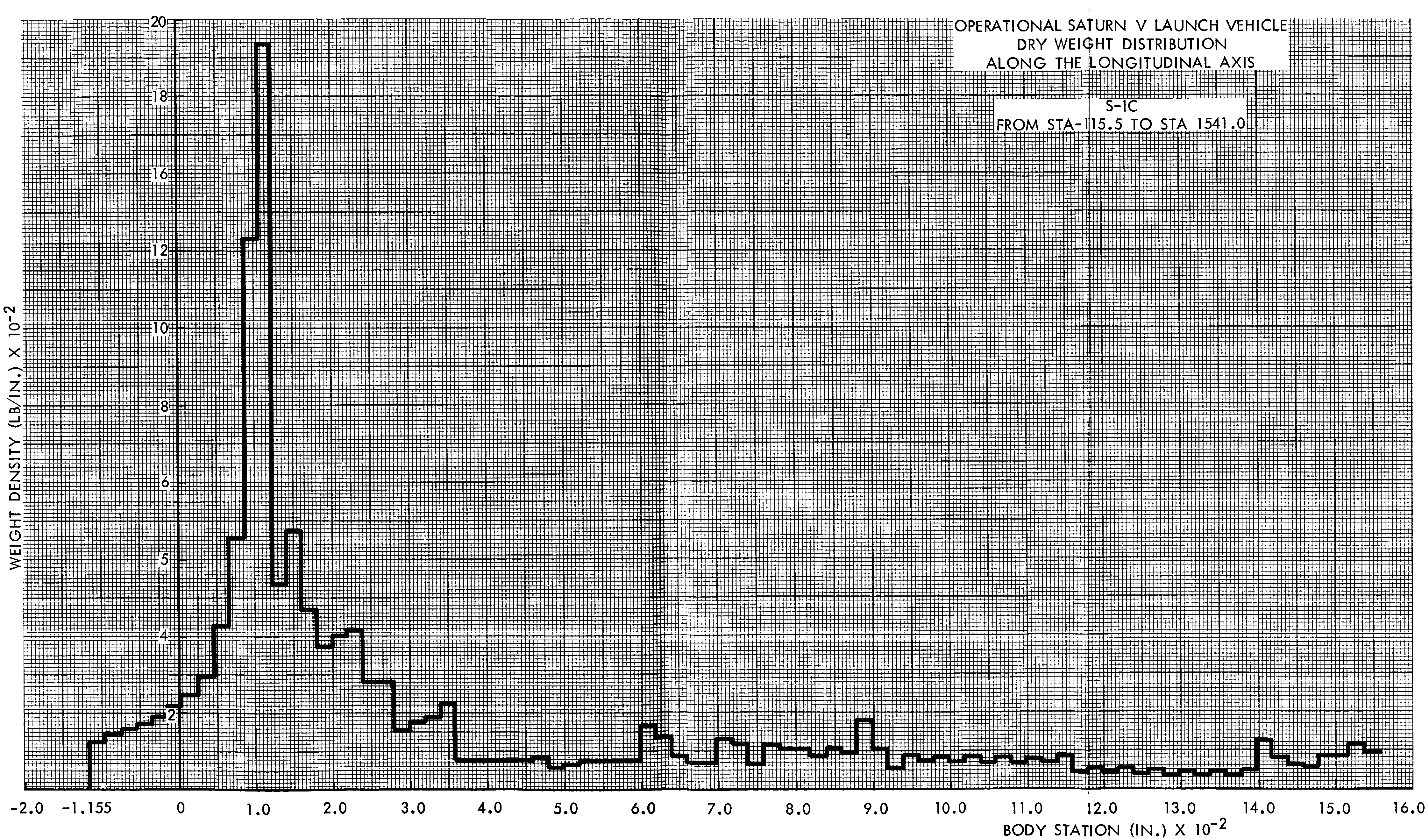


Figure 57. Saturn V, S-IC, Dry Weight Distribution

~~CONFIDENTIAL~~

OPERATIONAL SATURN V LAUNCH VEHICLE
WET WEIGHT DISTRIBUTION
ALONG THE LONGITUDINAL AXIS
FROM STA-115.5 TO STA 1541

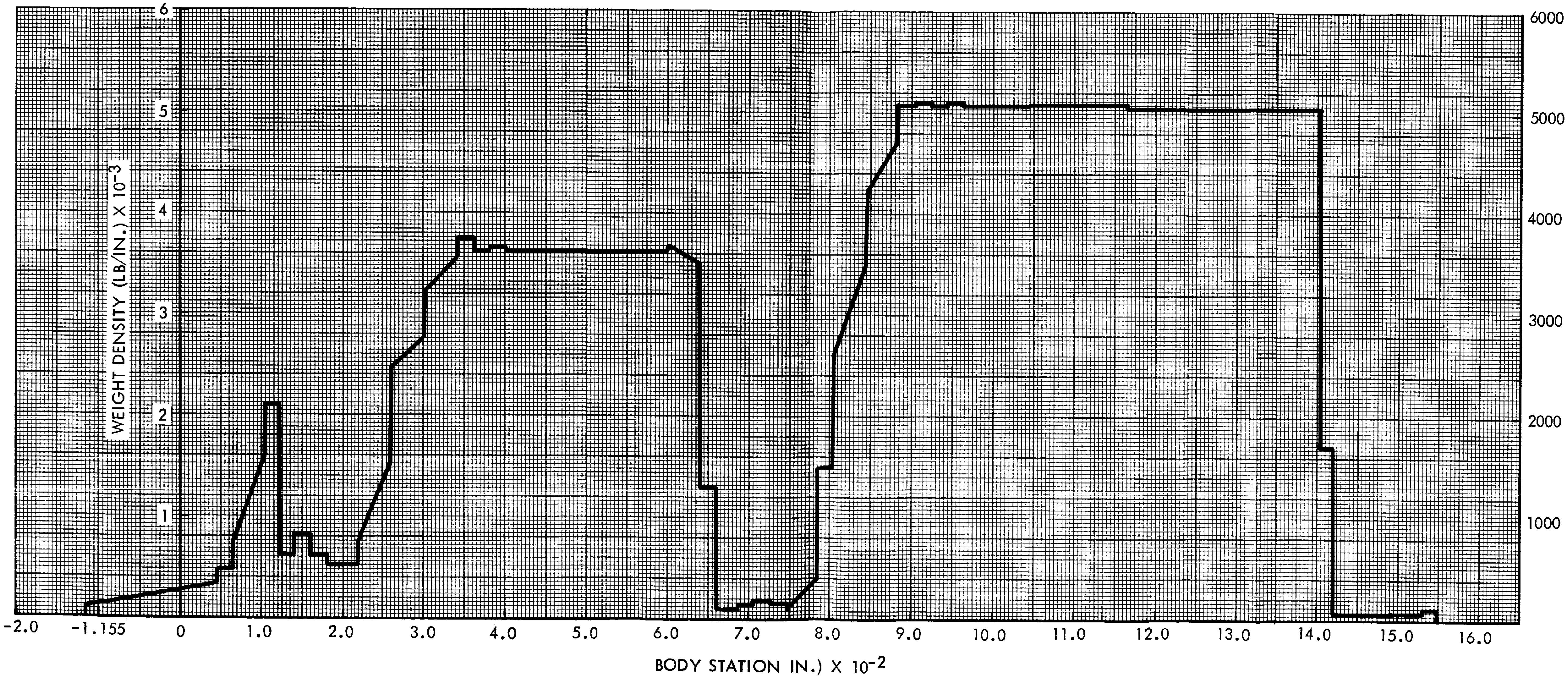


Figure 58. Saturn V, S-IC, Wet Weight Distribution

~~CONFIDENTIAL~~

OPERATIONAL SATURN V LAUNCH VEHICLE
FUEL WEIGHT DISTRIBUTION
ALONG THE LONGITUDINAL AXIS

S-IC
FROM STA-115.5 TO STA 1541.0

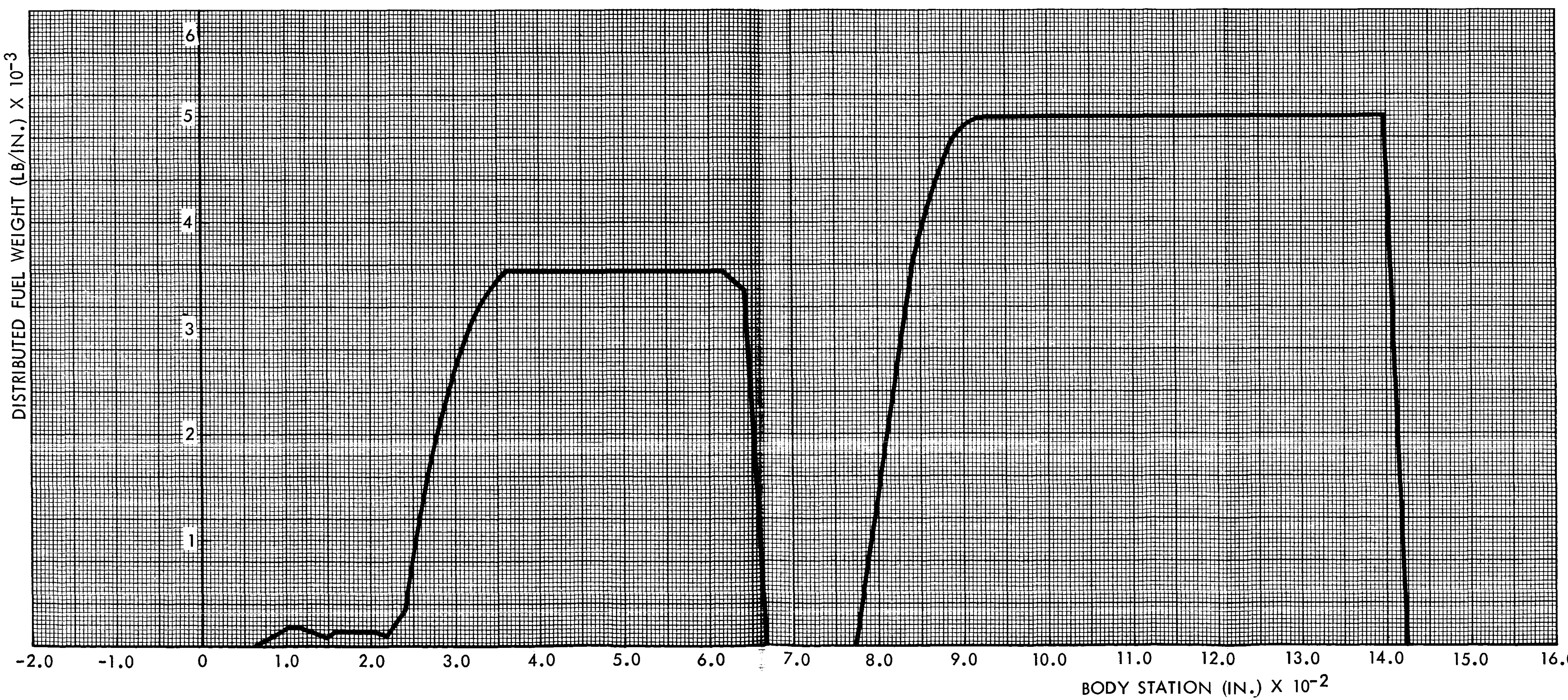


Figure 59. Saturn V, S-IC, Full Weight Distribution

~~CONFIDENTIAL~~

CON[redacted]

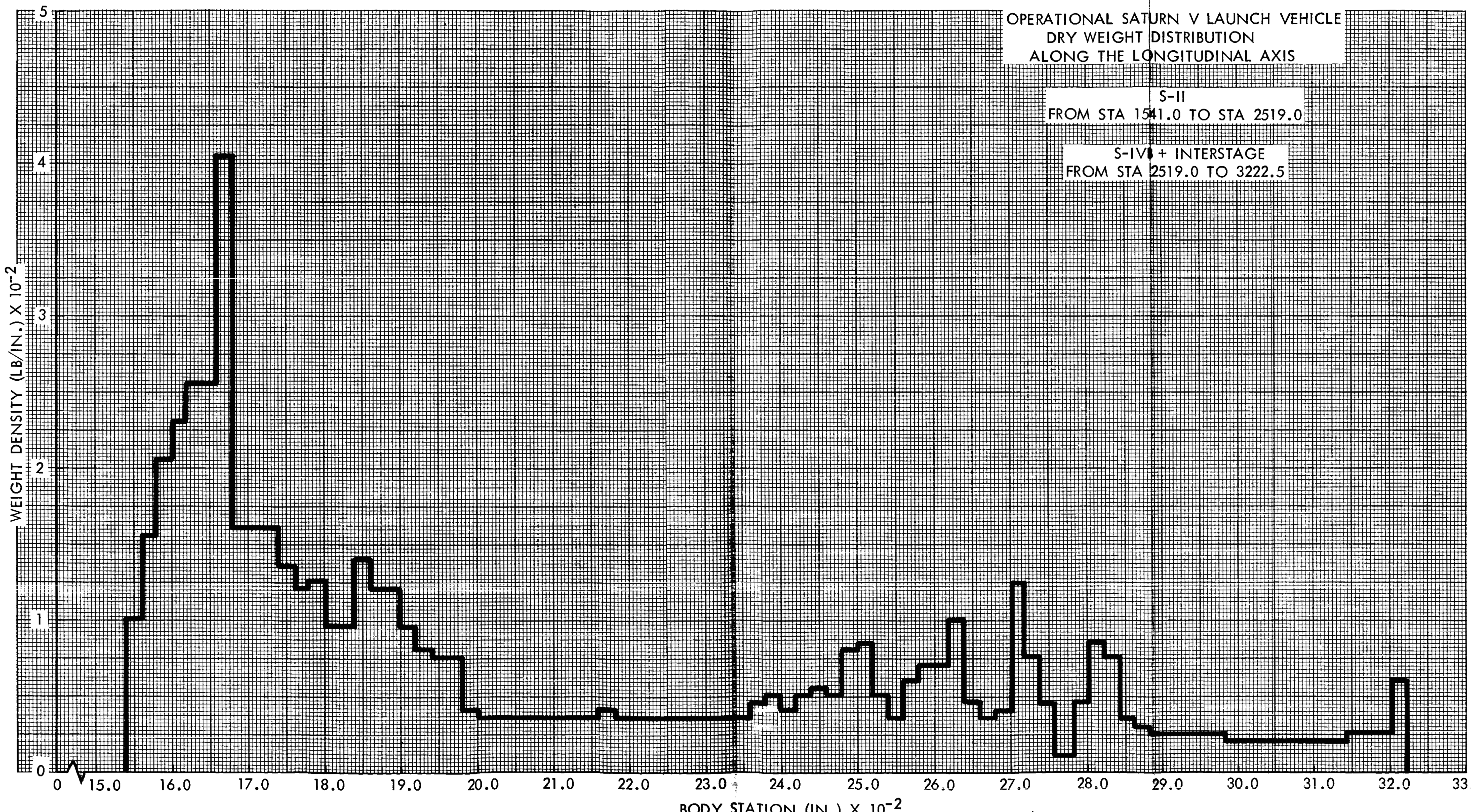


Figure 60. Saturn V, S-II, Dry Weight Distribution

~~CONFIDENTIAL~~

OPERATIONAL SATURN V LAUNCH VEHICLE
WET WEIGHT DISTRIBUTION
ALONG THE LONGITUDINAL AXIS

S-II
FROM STA 1541.0 TO STA 2519.0

S-IVB + INTERSTAGE
FROM STA 2519.0 TO STA 3222.5

INSTRUMENT UNIT
FROM STA 3222.5 TO STA 3258.5

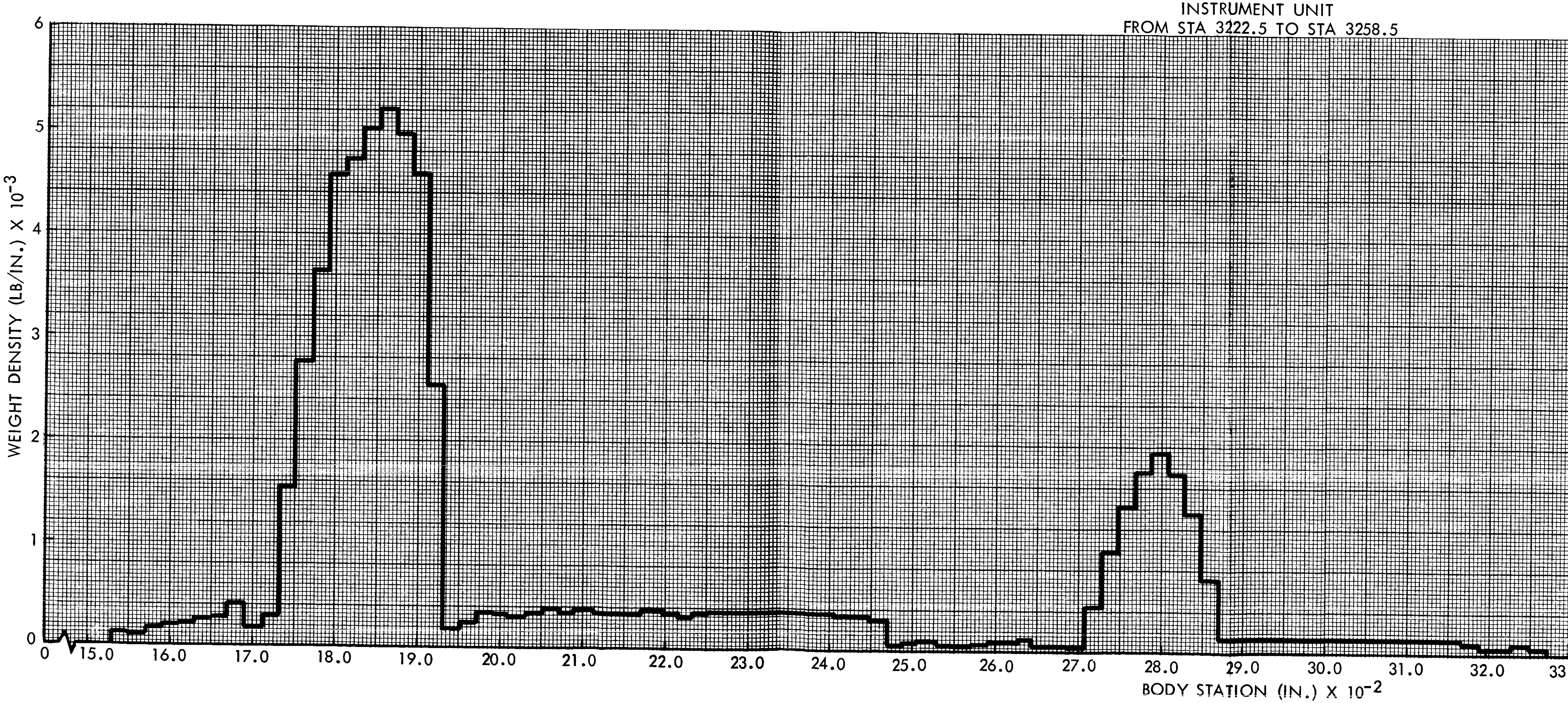
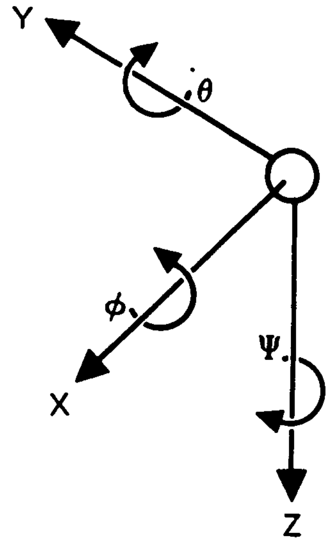


Figure 61. Saturn V, S-II, Wet Weight Distribution

CON[redacted]

Table 1. Reference Axes



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

~~CONFIDENTIAL~~

[redacted]

Table 2. Estimates of Metabolic Rate, Thermal Balance and Water

General Requirements for Apollo Crew Member

Per Man		Command Module Routine Flight Per Day	Command Module Emergency Decompression Per Day
Heat Output	Btu	11,200	12,000
Oxygen	lb	1.84	1.97
Carbon Dioxide	lb	2.12	2.27
Latent Heat (lungs)	Btu	2,800	3,000
Latent Heat (sweat)	Btu	1,310	7,230
Sensible Heat	Btu	7,090	1,870
Urinary Loss	g	1,200	1,200
Sweat Loss	g	600	3,140
Lung Loss	g	1,200	1,300
Total Water Requirement	g	3,000	5,640
Total Water Requirement	lb	6.6	12.4

Post Landing Requirements

Metabolic Heat Load
- 8.5 ft wave height - 600 Btu/hr.
- 0.5 ft wave height - 400 Btu/hr.

Allowable Effective Temperature
- 600 Btu/hr. heat load - 86.5 F
- 400 Btu/hr. heat load - 88 F

(Allowable effective temperature may be exceeded for four consecutive hours.)

[redacted]