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SID 64-1344

# CSM TECHNICAL SPECIFICATION (BLOCK II)

(U)

1 December 1964

NAS 9-150

Approved by

Dale D. Myers

Dale D. Myers
Vice President
Apollo Program Manager

[redacted]

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NORTH AMERICAN AVIATION, INC.
SPACE and INFORMATION SYSTEMS DIVISION

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# TECHNICAL REPORT INDEX ABSTRACT

N65-8885

| ACCESSION NUMBER | DOCUMENT SECURITY CLASSIFICATION |
|---|---|
| | |

TITLE OF DOCUMENT

SID 64-1344 CSM Technical Specification Block II

LIBRARY USE ONLY

AUTHOR(S)

W. L. Wheeler/R. L. Cook

| CODE | ORIGINATING AGENCY AND OTHER SOURCES | DOCUMENT NUMBER |
|---|---|---|
| | NAA - S&ID Apollo Engineering<br>Downey, California | SID 64-1344 |

| PUBLICATION DATE | CONTRACT NUMBER |
|---|---|
| 1 December 1964 | NAS9-150 |

DESCRIPTIVE TERMS

**This specification defines the performance and design requirements for the Apollo (Block II) Command and Service Modules (CSM) System. This specification provides the base-line requirements for the CSM (Block II) and its supporting and associated systems.**

ABSTRACT

The CSM System as discussed includes the Launch Escape System (LES), a Command Module (CM), a Service Module (SM), a Spacecraft Lunar Excursion Module Adapter (SLA), the associated Ground Support Equipment (GSE), and the requisite trainers.

Performance characteristics of the Launch Vehicle and other items of Government Furnished Equipment (GFE) with which the design of the CSM is compatible, are also defined.

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| | |
|---|---|
| AFMTCP 80-2<br>1 Oct. 1963 | General Range Safety Plan, Prelaunch Safety Procedure (Volume I) |
| AFMTCP 80-2 | General Range Safety Plan (Volume I) |
| AFSG No. 153 | AF Interim Supplemental Atmosphere to 90 Km. |

National Aeronatuics and Space Administration (NASA)

| | |
|---|---|
| MSFC Memo.<br>R-AERO-Y-12-63<br>9 Dec 1963 | Availability of Computer Subroutine for the 1963 Patrick Reference Atmosphere |
| MSFC Memo.<br>M-P&VE-PP-96-63<br>11 March 1963 | Estimated F-1 Engine Altitude Thrust Decay for S-IC Stage of Saturn V vehicle |
| MSFC Memo.<br>R-P&VE-VAD-64-24<br>17 April 1964 | Saturn V Design Ground Rules |
| NPC 200-2<br>20 April 1962 | Quality-Program Provisions for Space System Contractors |
| NPC 200-3<br>20 April 1962 | Inspection System Provisions for Suppliers of Space Materials, Parts, Components, and Services |
| NPC 250-1<br>July 1963 | Reliability Program Provision for Space System Contractors |

Commercial

| | |
|---|---|
| PTOOSE-24-61S | Wall Mounting Receptacle (Bendix) |

2.2 Precedence. - The order of precedence in the instance of conflicting requirements shall be as follows:

a. The Contract, NAS9-150

b. This Specification

c. CSM Master End Item Specification (Block II), SID 64-1345

d. Other Documents referenced herein

2.3 Effectivity. - Effectivity of contract changes shall be as of 2 November 1964, and subsequent approved specification amendments.

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## 1.0 SCOPE

1.1 Scope. - This specification defines the performance and design requirements for the Apollo (Block II) Command and Service Module (CSM) System and establishes requirements for the design and development and, in conjunction with Specification, SID 64-1345, forms the documentation stating the CSM system technical and testing requirements and configuration description. The CSM System, as discussed in this specification, is composed of a Launch Escape Subsystem (LES), a Command Module (CM), a Service Module (SM), a Spacecraft LEM Adapter (SLA), the associated Ground Support Equipment, and the requisite Trainers. The general configuration of the Block II CSM System, the Lunar Excursion Module (LEM), and the Launch Vehicle (LV) is shown in Figures 1 and 2.

Performance characteristics of the LV and other items of Government Furnished Equipment (GFE) with which the design of the CSM system is compatible, are also specified.

1.2 Objective. - The objective of this specification is to provide base line requirements for the Block II CSM and its supporting and associated systems.

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## 2.0 APPLICABLE DOCUMENTS

The following documents, of exact issue shown, form a part of this specification to the extent specified herein.

2.1 Project Documents. - The asterisk (*) adjacent to a document number indicates that further review and mutual agreement is required prior to incorporation of the document into this specification.

### SPECIFICATIONS

#### Military

| | |
|---|---|
| MIL-E-6051C<br>17 June 1960 | Electrical-Electronic System Compatibility and Interface Control Requirements for Aeronautical Weapon Systems |

#### National Aeronautics and Space Administration (NASA)

| | |
|---|---|
| MSC-GSE-1B<br>23 June 1964 | Apollo Ground Support Equipment General Environmental Criteria and Test Specification |
| MSFC 10M01071<br>6 March 1961 | Environmental Protection When Using Electrical Equipment Within the Areas of Saturn Complexes Where Hazardous Areas Exist, Procedure for |
| MSFC-PROC-158A<br>12 April 1962 | Soldering Electrical Connectors (High Reliability) Procedure for (As amended by MSC-ASPO-S-5B, dated 16 February 1964) |
| *(To be supplied by NASA) | GFE Guidance and Navigation Performance and Interface Specification - Block II |
| *(To be supplied by NASA) | Exhibits for the Apollo Block II Space Suit Assembly Procurement Package |



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

| | |
|---|---|
| MC999-0002B<br>3 January 1963 | Electromagnetic Interference Control for the Apollo Space System |
| ME 414-0095G-0062<br>16 Oct. 1964 | Connector Plug, Shell Size 14, High Temperature, Specification Control Drawing for |
| *SID 62-203 | NAA/S&ID Reliability Program Plan |
| SID 62-590<br>10 June 1962 | Saturn II Separation Data Manual |
| *SID 64-690 | GFE Rendezvous Radar Performance and Interface Specification |
| *SID 62-1244 | GFE Lunar Excursion Module Performance and Interface Specification |
| SID 64-1345<br>1 Dec. 1964 | CSM Master End Item Specification - Block II |
| *SID 64-1388 | GFE Scientific Instrumentation Performance and Interface Specification - Block II |
| *SID 64-1389 | GFE NASA-Furnished Crew Equipment Performance and Interface Specification Block II |
| *SID 64-1390 | GFE Apollo Checkout Equipment Performance and Interface Specification - Block II |
| *SID 64-1613 | GFE CSM-MFSN Communication Performance and Interface Specification - Block II |
| *SID 64-1707 | Apollo Spacecraft Development Test Plan |
| *SID 64-1807 | Apollo Training Equipment Specification |
| *SID (TBD) | Master Ground Operations Specification - Block II |

## DRAWINGS

### NAA/S&ID

| | |
|---|---|
| ICD 13M20108<br>20 July 1964 | Instrument Unit to Spacecraft Physical Requirements (Saturn IB) |
| ICD 13M20109<br>24 July 1964 | Spacecraft/Q-Ball Physical Requirements (Saturn IB) |
| ICD 13M50103<br>28 May 1964 | Instrument Unit to Spacecraft Physical Requirements (Saturn V) |
| ICD 13M50112<br>22 July 1964 | Spacecraft/Q-Ball Physical Requirements (Saturn V) |
| ICD 13M50123<br>29 July 1964 | Envelope LEM/SIVB/IU Clearance, Physical |
| ICD 40M37500<br>(Series) | Electrical Requirements, Instrument Unit/Spacecraft Interface |

## STANDARDS

### Federal

U.S. Standard Atmosphere, 1962

### Military

| | |
|---|---|
| MS-33586A<br>16 December 1958 | Metals; Definitions of Dissimilar |
| MIL-STD-130B<br>24 September 1962 | Identification Marking of U.S. Military Property |

## OTHER PUBLICATIONS

### Military

| | |
|---|---|
| WADC-TR-52-321<br>September 1954 | Anthropometry of Flying Personnel - 1950 |
| ARDC Model<br>Atmosphere, 1959 | |

| | |
|---|---|
| AFMTCP 80-2<br>1 Oct. 1963 | General Range Safety Plan, Prelaunch Safety Procedure (Volume I) |
| AFMTCP 80-2 | General Range Safety Plan (Volume I) |
| AFSG No. 153 | AF Interim Supplemental Atmosphere to 90 Km. |

National Aeronatuics and Space Administration (NASA)

| | |
|---|---|
| MSFC Memo.<br>R-AERO-Y-12-63<br>9 Dec 1963 | Availability of Computer Subroutine for the 1963 Patrick Reference Atmosphere |
| MSFC Memo.<br>M-P&VE-PP-96-63<br>11 March 1963 | Estimated F-1 Engine Altitude Thrust Decay for S-IC Stage of Saturn V vehicle |
| MSFC Memo.<br>R-P&VE-VAD-64-24<br>17 April 1964 | Saturn V Design Ground Rules |
| NPC 200-2<br>20 April 1962 | Quality-Program Provisions for Space System Contractors |
| NPC 200-3<br>20 April 1962 | Inspection System Provisions for Suppliers of Space Materials, Parts, Components, and Services |
| NPC 250-1<br>July 1963 | Reliability Program Provision for Space System Contractors |

Commercial

| | |
|---|---|
| PTOOSE-24-61S | Wall Mounting Receptacle (Bendix) |

2.2 Precedence. - The order of precedence in the instance of conflicting requirements shall be as follows:

a. The Contract, NAS9-150

b. This Specification

c. CSM Master End Item Specification (Block II), SID 64-1345

d. Other Documents referenced herein

2.3 Effectivity. - Effectivity of contract changes shall be as of 2 November 1964, and subsequent approved specification amendments.

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## 3.0 REQUIREMENTS

The basis for design of the CSM System (Block II) shall be the lunar landing mission.

This section 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 principal functional interfaces.

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

3.1 Performance. - The following are the principles to which the basic technical approach of the CSM subsystem 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 Operational Requirements.

3.1.1.1 Mission.

3.1.1.1.1 Manning of Flight. - The CSM shall be designed for manned operation with full utilization of human crew capabilities. Automatic subsystems shall be used only where they will enhance the performance of the mission. Where possible, automatic systems shall be manually operable by override.

3.1.1.1.2 Onboard Command. - The spacecraft will normally utilize inputs from earth-based tracking and computing facilities in conjunction with onboard computations to perform mission requirements. However, the spacecraft shall have the capability of performing any phase of the mission independent of ground facilities or of aborting the mission in its entirety.

3.1.1.1.3 Flight Time Capabilities.

3.1.1.1.3.1 Flight. - The Apollo CSM System shall be designed to accomplish the Lunar Orbit Mission. The CSM System consumable subsystems shall be designed for a nominal mission time of 10.6 days with 3 of these days in lunar orbit using the $\Delta$ V allocations shown in 3.1.1.2.7.3. By judicious system management of duty cycles, alternate missions, such as Earth orbital, may be performed within the resultant capabilities of the CSM system. A 14-day mission capability is to be achieved by proper selection of missions and related duty cycles for use of the total electric power available since the electrical power cryogenics is sized to permit abort from the most critical point in the lunar mission with one cyro tank inoperative. In addition, provisions are available for on-landing of additional lithium hydroxide canisters and the oxygen supply is sized to provide CM leakage of 0.2 pounds per hour plus metabolic oxygen, as required for a 14-day mission.

3.1.1.1.3.2 Post Landing. - The CM shall be habitable for 48 hours and retrievable for 7 days following a water landing.

3.1.1.1.4 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 shall be selected for normal mission landing. Emergency land landing capability shall be provided for early launch aborts.

3.1.1.1.5 Flight Plan. - The Apollo mission flight plan for which the CSM is sized shall be as specified in 3.1.1.1.5.1 through 3.1.1.1.5.4.

3.1.1.1.5.1 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 used in overall CSM and subsystem design. The characteristics of the lunar missions are described in the following subparagraphs:

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 within 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 2.5 hr period on the launch date. Launches may be made provided visual reference conditions sufficient for orientation during high altitude abort exist. A launch window shall be provided, either by maneuvering the space vehicle 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 shall be provided by a Government-furnished launch vehicle from a Government-furnished launch complex.

c. Earth Parking Orbit. - The Earth parking orbit phase shall begin with S-IVB cutoff in orbit and end 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 105 nautical miles. Multiple parking orbits are acceptable, but they shall be compatible with booster performance and lifetime limitations. The duration of this phase shall not exceed 4-1/2 hours.

d. Translunar Injection. - The translunar injection phase shall begin with S-IVB ignition in Earth parking orbit and end with S-IVB cutoff. Final injection into the translunar trajectory shall be located such that the trajectory can be determined by the Manned Space Flight Network (MSFN) within 15 minutes of translunar injection burnout.

e. Translunar Coast. - The translunar coast phase shall begin with S-IVB cutoff and end 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 normally be from 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. Non-free return trajectories shall also be considered. The duration of the non-free return translunar coast phase shall not exceed 110 hours. The translunar trajectories for lunar orbit missions shall have a nominal pericynthion of 80 nautical miles. The Command and Service Module shall include provisions for performing transposition and docking following translunar injection. The Command and Service Module shall be the active vehicle and the LEM-S-IVB, the passive vehicle for this maneuver. With the S-IVB in an attitude-hold mode, the Command and Service Module shall separate from the LEM/S-IVB, perform a turn-around, and fly a closing maneuver to effect initial contact conditions within the capture and impact attenuation capabilities of the docking mechanisms. Upon completion of the docking the Command and Service Module shall separate the CSM/LEM from the S-IVB. The SLA shall remain attached to the S-IVB. Detailed operational characteristics shall be as described in SID 64-1244.

Because of the radiation hazard, the crew shall not enter the tunnel during high intensity region of the trapped radiation belts after translunar injection. The transposition docking and separation from S-IVB shall be completed with two hours after translunar injection. The capability shall exist to perform docking in the mission natural lighting environments.

The Command and Service Module shall be capable of providing for performing translunar midcourse correction maneuvers in the docked configuration. As an emergency operation, the Command and Service Module shall be capable of making use of LEM propulsion as a backup to the SM propulsion.

f. Lunar Orbit Insertion. - Lunar orbit insertion shall begin with Service Propulsion Subsystem (SPS) ignition just prior to pericynthion and end with SPS cutoff in lunar orbit. Insertion shall occur over the nonvisible portion of the Moon. The CSM shall arrive in a circumlunar trajectory which has a nominal pericynthion altitude of 80 nautical miles. A 5-degree plane change capability shall be provided with 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 shall begin with (SPS) cutoff in lunar orbit and end with SPS ignition for transearth injection. The nominal lunar orbit altitude shall be 80 nautical miles. The Command and Service Module shall be stabilized in an orientation suitable for LEM active rendezvous and docking maneuvers. The Command and Service Module shall be capable of taking corrective action as a backup to the LEM for rendezvous and/or shall be capable of serving as the active spacecraft for docking. As an emergency operation, the Command and Service Module shall have the capability of one descent to an altitude of 50,000 feet and effecting a rendezvous and docking with the LEM. After completion of crew and equipment transfer from the LEM to the Command and Service Module, the Command and Service Module shall separate from the LEM.

h. Transearth Injection. - Transearth injection shall begin with SPS ignition in lunar orbit and end 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 parking orbit at the time of injection, the time spend 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 shall 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 85 and 110 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 posigrade motion with respect to the Earth to reduce the entry heating and widen the entry corridor.

j. Entry. - The entry phase begins at the entry interface (nominally 400,000 feet) and ends at drogue parachute deployment. The CM shall be capable of entry with a normal operational corridor with a maximum deceleration of 9 g during the initial pull-out with a minimum L/D=0.30 throughout entry 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 2,500 nautical miles. The maximum emergency 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. Post Landing. - The postlanding phase covers the time from CM touchdown to CM retrieval.

3.1.1.1.5.2 Design Mission Definitions.

3.1.1.1.5.2.1 Design Reference Mission. - The 8.3 day NASA Design Reference Mission shown below shall be the basis for establishing consumable weights in control weights, weight and electrical power reporting, reliability modeling, crew task analyses, engineering simulations, training and training equipment design, trade-off studies, operational procedures, meteroid and radiation shielding analysis, and interface control.

Design Reference Mission (8.3 day)

| | |
|---|---|
| Prelaunch | 10.0 hrs |
| Ascent | .20 |
| Earth Parking Orbit | 2.8 |
| Translunar Injection | .09 |
| Translunar Coast | 61.15 |
| Lunar Orbit Insertion | .10 |
| Lunar Orbit Coast | 44.8 |
| Transearth Injection | .03 |
| Transearth Coast | 89.10 |
| Entry and Landing | .40 |

3.1.1.1.5.2.2 Consumables Design Mission. - The following 10.6-day lunar orbit mission timeline shall serve as a basis for integrated subsystem design for consumables:

| Mission Phase | Duration Hours |
|---|---|
| Prelaunch | 10.00 |
| Ascent phase | 0.19 |
| Earth Parking orbit | 4.40 |
| Translunar injection | 0.09 |

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| Mission Phase | Duration Hours |
|---|---|
| Translunar coat | 77.00 |
| Lunar orbit injection | 0.02 |
| Lunar orbit coast | 88.00 |
| Transearth injection | 0.04 |
| Transearth coast | 84.00 |
| Pre-entry | 0.08 |
| Entry | 0.50 |
| Recovery | 0.17 |

3.1.1.1.5.3 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. The spacecraft shall be designed such that a single crewman can perform all functions required to accomplish a safe return to earth from any point in the mission in case of an emergency.

3.1.1.1.5.3.1 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

3.1.1.1.5.3.2 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



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c. Adequate fuel margin

d. Minimum response-time criticality

e. Primary landing area

f. Adequate margin for consumables

g. Manned Space Flight Control Center (MSCC) and MSFN assistance

h. Hardware reliability

i. Minimum number of abort trajectories

j. Minimum flight-plan complexities

k. Performance flexibility

3.1.1.1.5.3.3 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.

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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.

3.1.1.1.5.3.4 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 $\Delta$ V requirements for discrete abort trajectories shall be readily available on board. Sufficient conversion data shall be available on board to convert propellant readings to $\Delta$ V 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.



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3.1.1.2 Command and Service Modules.

3.1.1.2.1 Command Module (CM). - The CSM shall include a recoverable CM. This module shall contain the communication, navigation, guidance, control, computing, display equipment, and other equipment 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 scientific equipment as specified in SID 64-1388.

a. Housing - The CM shall house three crew members during the launch, translunar, lunar orbit, transearth entry and recovery and post landing 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 and for extravehicular activity during space flight. The upper hatch shall be used for preparation of the Command Module/LEM interface for translunar operations following docking, for intravehicular crew transfer between the Command Module and LEM, and may be used in the postlanding recovery phase. During intravehicular crew transfer or interface operations, the spacesuit may be either vented or pressurized. A portable life support system and pressurized spacesuit will be used for extravehicular activity.

3.1.1.2.2 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.

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3.1.1.1.5.4 Command and Service Module Attitude Management.

a. CSM functions, which are sensitive to CSM attitude, are as follows:

(1) Thrust and lift vector control.

(2) Thermal control of the CSM.

(3) Communications including radio navigation.

(4) Optical navigation.

(5) Reaction control propellant conservation.

(6) Energetic particle radiation.

b. Mission flexibility, particularly the any-day launch capability, shall not be compromised by factors related to attitude management.

c. Nominal or "standard" attitudes, to which the CSM returns upon completion of an activity requiring specific orientation, shall not be employed. When no specific pointing requirements for thermal cycling or other reasons exist, unconstrained angular drift shall be accepted.

d. The capability of communication utilizing the high gain directive antenna shall be preserved except when occulsion by the moon interferes. During normal operations, periodic losses in hi-gain directional antenna communications are permitted provided the MSFN may contact the crew by voice at any time.

e. Optical sighting schedules and choice of observable, celestial bodies shall consider the requirements of thermal cycling.

f. The number and rate of attitude changes shall be minimized to the greatest extent practicable in consideration of reaction control propellant economy.

3.1.1.2.3 Spacecraft-LEM Adapter (SLA). - The SLA shall structurally and functionally adapt the SM to the LV, and provide for in-flight separation of CSM from the LV and cover and support the LEM.

3.1.1.2.4 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.

3.1.1.2.5 Command and Service Module Subsystems. - The CSM subsystems requirements and subsystem descriptions are contained in SID 64-1345.

3.1.1.2.6 Command and Service Module Performance. - The following subparagraphs summarize the nominal performance capabilities of the CM, SM, and SLA.

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

3.1.1.2.6.2 Trajectories. - The general CSM trajectories shall follow the general requirements described in 3.1. After translunar injection, the primary measured CSM positional accuracy shall be provided by the MSFN with the CSM Integrated Guidance and Control Subsystem (G&C) serving as a back-up system in accordance with SID 64-1613, NASA Manned Space Flight Network, and G&C Performance and Interface Specification MSC-(TBD).

3.1.1.2.6.3 Command and Service Module Propulsion Increments After SIVB Separation. - After separation of the SIVB, propulsion increments of

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the CSM shall be supplied by the SPS. For mission comparison purposes, weight report etc., the SPS characteristics 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 | 3,570 |
| b. Transearth | |
| (1) Lunar orbit maneuvers | 455 |
| (2) Transearth injection | 3,160 |
| (3) Transearth midcourse | 300 |

3.1.1.2.7 Lunar Excursion Module (LEM).- The spacecraft shall include a NASA furnished LEM which will serve as a vehicle for carrying two of the crew members and payloads from the spacecraft in a lunar orbit to the lunar surface and back. The LEM will have the capability of performing the separation, lunar descent, landing, ascent rendezvous, and docking independent of the CSM except that the CSM must be stabilized. The interface and performance requirements of the LEM as part of the space vehicle and spacecraft shall be as established in SID 62-1244.

3.1.2 Operability.

3.1.2.1 Reliability.- The mission success reliability objective for Apollo shall be for a lunar orbital rendezvous (LOR) mission followed by the return to earth of the CSM without exceeding the emergency crew limits, given in the design criteria.

3.1.2.1.1 Crew Safety Reliability.- The crew safety reliability objective for the Apollo LOR mission 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.

3.1.2.1.2 Reliability Apportionment.- The reliability objectives for the major Apollo-Saturn systems shall be as delineated on the following page.

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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 |
| CSM | 0.9638 | 0.99958 |
| LEM (defined by the NASA) | 0.984 | 0.9995 |
| APOLLO-SATURN (Total) | 0.90 | 0.999 |

Reliability apportionment shall be based on the design reference mission.

3.1.2.1.3 Micrometeorid and Radiation Probabilities. - Micrometeoroid and radiation environments and protection requirements specified in this document and in Specification SID 64-1345 are for the purpose of providing design bases. The requirements are not to be included in the overall reliability apportionment and evaluation for mission success or crew safety. The CSM micrometeoroid protection shall be designed to give a probability of 0.995 of not requiring abort during the standard design reference lunar landing mission of 8.3 days, using the following criteria.

a. The sporadic meteoroid environment defined is as 3.1.2.4.2.2(c).

b. The mechanics of penetration as given by the modified Summers equation:

$$\frac{\bar{t}}{d_m} = 3.42\left(\frac{\rho_m}{\rho_t}\frac{V_m}{C_t}\right)^{2/3}$$

Where:

$\bar{t}$ = equivalent single sheet thickeners penetrated
$d_m$ = diameter of meteoroid
$\rho_m$ = density of meteoroid
$t$ = density of target
$V_m$ = velocity of meteoroid
$C_t$ = speed of sound in target material



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3.1.2.2 Maintainability.

3.1.2.2.1 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. In-flight maintenance shall not be performed on CSM subsystem.

3.1.2.2.2 Maintenance Concept. - Field maintenance of CSM subsystems shall be performed as follows:

a. For airframe electrical/electronic equipment (either installed or on the bench), checkout and replacement shall be at the integral package (black box) level. A "black box" is defined as a combination of factory replaceable units which are contained within a physical package, and which is removable from the CSM as an integral unit.

b. For non-electrical/electronic equipment (either installed or on the bench), checkout and replacement shall be at the lowest replaceable serialized unit level, which includes only those parts which are removable as integral units from the CSM.

Bench test equipment may be used for malfunction verification of packages (units) removed from the CSM because of suspected failure. The malfunctioned package (unit) shall be returned to the supplier. Bench test equipment may also be used for spares certification before installation.

3.1.2.3 Useful Life.

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

3.1.2.4 Natural Environment.- These requirements define the natural environmental criteria to which the CSM equipment and associated Ground Support Equipment (GSE) shall be designed.

3.1.2.4.1 CSM and GSE Ground Environments.

3.1.2.4.1.1 Transportation, Ground Handling, and Storage. - The following represent the natural environmental extremes which may be encountered by CSM equipment and GSE in a non-operating condition during transportation, ground handling and storage. Handling GSE shall be capable of operating during exposure to the environments. Other GSE and CSM equipment may be protected by suitable packaging for transportation and storage if these environments exceed the equipment design operation requirements. The equipment shall be capable of meeting the operating requirements of the applicable performance specification after exposure, while protected by its normal packaging, to these environments.

(a) Temperature (Air)

| | |
|---|---|
| Air transportation | -45 to +140 F for 8 hours |
| Ground transportation | -20 to +145 F for 2 weeks |
| Storage | +25 to +105 F for 3 years |

(b) Pressure

| | |
|---|---|
| Air transportation | Minimum of 3.47 psia for 8 hours (35,000 ft. altitude). |
| Ground transportation and storage | Minimum of 11.78 psia for 3 years (6,000 ft. altitude) |

(c) Humidity — 0 to 100 percent relative humidity, including conditions wherein condensation takes place in the form of water or frost for at least 30 days.

| | |
|---|---|
| (d) Sunshine | Solar radiation of 360 BTU per square foot per hour for 6 hours per day for 2 weeks. |
| (e) Rain | Up to 0.6 inches per hour 12 hours, 2.5 inch per hour for 1 hour. |
| (f) Sand and dust | As encountered in desert and ocean beach areas, equivalent to 140-mesh silica flour with particle velocity up to 500 feet per minute and a particle density of 0.25 grams per cubic foot. |
| (g) Fungus | As experienced in Florida climate. Materials will not be used which will support or be damaged by fungi. |
| (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 per million concentration or 3 months at 0.25 ppm or 72 hours at 0.5 ppm. |
| (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 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, build-up of 2 seconds and 2 seconds for decay to steady state wind speed.

3.1.2.4.1.2 Sheltered environment areas. - These requirements represent the natural environmental design criteria for CSM equipment and GSE both in operating and nonoperating conditions as determined by normal operational flow sequences. The equipment shall be capable of meeting the operating requirements of the applicable performance specification during and after exposure to these environments. The level of environmental control at each Apollo site shall be as indicated in MSC-GSE-1B.

3.1.2.4.1.2.1 Interior uncontrolled. - An environment in which the temperature, sand, salt spray, etc., are only partially controlled.

| | |
|---|---|
| (a) Temperature | +15 to +105 F for up to 3 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 30 days. |

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| | |
|---|---|
| (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 to 500 feet per minute and a particle density of 0.25 grams per cubic foot. |
| (e) 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. |

3.1.2.4.1.2.2 Other environment areas. - Natural environments to which certain GSE are exposed, such as the launch umbilical tower, shall be as indicated in MSC-GSE-1B.

3.1.2.4.2 CSM Flight Environments. - These requirements represent the natural environmental design criteria for the CSM equipment in an operating condition as experienced during the various flight mission phases. The exposure time for each mission phase environment shall be the maximum as defined in paragraphs 3.1.1.1.5.1 and 3.1.1.1.5.2 for a 14-day design limit mission, unless otherwise defined in this section. The equipment shall be capable of meeting the operating requirements of the applicable performance specification during and after exposure to these environments.

3.1.2.4.2.1 Ascent phase. -

| | |
|---|---|
| (a) Reference Atmosphere | The reference Earth Atmosphere at Cape Kennedy, Florida, shall be in accordance with MSFC Memorandum R-AERO-Y-12-63. For aerodynamic heating estimates above 98,430 ft. (30Km), the reference atmosphere shall be ARDC 1959. |



[redacted]

(b) Ground winds

The design wind speed for launch of the CSM is presented below. Wind speed occurring during the strongest wind month at Cape Kennedy, Florida, are less than those presented 95.0 percent of the time.

| Height (ft) | Steady State Wind (knots) | Peak Wind (knots)* |
|---|---|---|
| 10 | 14.0 | 19.6 |
| 30 | 17.4 | 24.4 |
| 60 | 20.0 | 28.0 |
| 100 | 22.2 | 31.1 |
| 200 | 25.5 | 35.7 |
| 300 | 27.6 | 38.6 |
| 400 | 29.3 | 41.0 |
| 500 | 30.6 | 42.8 |

* 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 - CSM and SLA - The design shall consider flight through synthetic wind profiles based on the following criteria:

(1) Idealized Wind Profile Envelope: Figure 3 presents the 95 percentile wind speed envelope with respect to altitude for the worst wind month for the Atlantic Missile Range. These winds are to be applied without regard to direction.

(2) Figure 4 presents the wind speed changes associated with differentials of altitude from 100 meters to 5000 meters over altitude ranges from 1 km to 80 km. These values are 85% of the 99 percentile wind speed changes. These wind speed changes are to be used in constructing the synthetic wind profiles.

(3) Gusts: Figure 5 presents the quasi-square wave gust that will be superimposed on the constructed wind profile such that the maximum wind speed in 7.65 m/sec greater than the wind speed envelope. This is 85% of the 99 percentile gust. The wave length of the applied gust will be a variable ranging from 50 m to 300 m.

(4) For synthetic profiles not incorporating the quasi-square wave gust, the peak profile speed may be maintained for various altitude thicknesses before decay is begun. Figure 6 presents the allowable range of peak wind thicknesses.

3.1.2.4.2.2 Earth parking orbit, translunar injection, translunar coast, lunar orbit insertion, lunar operations, transearth coast, and pre-entry phases

| | |
|---|---|
| (a) Pressure | Less than $10^{-13}$ mm Hg |
| (b) Electromagnetic Radiation | The sources of electromagnetic radiation presented below impinge on the exterior of the CSM in logical combination for a total time up to 336 hours. |
| Solar flux (all wavelengths). | 442 Btu/$ft^2$-hour |
| Earth emission (excluding reflection) | 73 Btu/$ft^2$-hour |
| Lunar subsolar point emission | 419 Btu/$ft^2$-hour |
| Lunar antisolar point emission | 2.2 Btu/$ft^2$-hour |
| Lunar average albedo (visual range) | 0.073 Btu/$ft^2$-hour |
| Lunar average reflectance for total solar spectrum | 0.047 ± .006 |

| | |
|---|---|
| Earth average albedo (Visual range) | 0.40 |
| Earth average albedo (total solar spectrum) | 0.35 |

(Thermal energy distribution to be interpreted according to cosine law.)

| | |
|---|---|
| Space sink temperature | Zero °Rankin |

(c) Meteoroid — The flux-mass sporadic meteoroid model for CSM exterior exposure in near-earth, cislunar, and near lunar space shall be as presented below. Spacecraft design shall be based on exposure during an 8.3 day LOR mission.

$$\log_{10} N = -1.34 \log_{10} m - 10.423 + \log_{10} A$$

where N = number of impacts per square foot per day
m = particle mass in grams
A = 0.5 in near earth and near lunar space (viewing loss)
A = 1.0 in cislunar space (viewing loss)

Particle density = 0.5 gm/cc for all particle sizes
Average geocentric velocity = 30 km/sec for all particle sizes

The flux relation given above is an average of the monthly variations. The above sporadic criteria are applied to the surface area of the vehicle.

(d) Nuclear Radiation - The neclear radiation environments for near-earth, cislunar, and near-lunar space shall be as presented below. Spacecraft design shall be based on exposure during a 14 day LOR mission. Crew safety assessment shall be based on exposure during an 8.3 day LOR mission.

(1) Trapped Radiation - Radiation levels in the Van Allen and artificial belts will use proton and electron fluxes obtained with the Goddard Orbital Flux Code.

(2) Galactic Cosmic Rays - Galactic cosmic ray doses range from 0.1 rad per week for solar activity maximum to 0.3 rad per week for solar activity minimum.

(3) Solar Particle Events - The solar particle events presented below are applicable to the length of time the spacecraft is at altitudes greater than those shown in Figure (TBD). Solar particles events will be considered to contain solar produced alphas and protons with equal rigidity spectra.

CON[redacted]

a. Time Integrated Spectra. - The time integrated spectra for both alphas and protons with rigidities from 1 to 10 MEV will be (TBD), and with those with rigidities greater than 0.137 BV (10 MEV) will be considered to be of the form

$$N(>P) = N_o \exp(-P/P_o), \quad P \; 0.137 \text{ Bv}$$

where

$N(>P)$ = time integrated flux with rigidities greater than P, particles/cm$^2$

$N_o$ = normalization constant, particles/cm$^2$

$P$ = particles, magnetic rigidity, volts

$P_o$ = characteristic rigidity, volts = constant.

The relativistic expression for magnetic rigidity is:

$$P = -\frac{1}{Z_e}\sqrt{T(T + 2 m_o c^2)}$$

where $Z_e$ = particle's charge in units of electron charge e, i.e., $Z_e = -1$ for protons and $Z_e = -2$ for alphas

$T$ = particle's kinetic energy, eV

$m_o c^2$ = particle's rest mass energy, eV

The constants $P_o$ and $N_o$ are:

$P_o = 10^8$ volts for both alphas and protons

For this value of $P_o$ the normalization constant is given by:

$$N_o = 10.9 \, N(>0.239\text{Bv})$$

where $N(>0.239\text{Bv})$ is the number of particles/cm$^2$ with rigidities greater than 0.239Bv (30 Mev) encountered during the mission. Figure 7 shows the probability of encountering greater than N(>0.239Bv) particles/$cm^2$ in a two week mission plotted against N(>0.239Bv). The values obtained for $N_o$ shall be considered to hold for both alphas and protons.

b. Time Dependent Spectrum - The model time dependent integral spectrum is shown in Figure 8 for several rigidities. The spectrum will be considered to hold for both alphas and protons. Note that the spectrum is normalized to one particle/$cm^2$ with rigidity greater than 0.239 Bv for the entire event.

3.1.2.4.2.3 Entry Phase.

| | |
|---|---|
| Reference Atmosphere | The reference Earth atmosphere for primary and contingency landing sites shall be in accordance with U.S. Standard Atmosphere, 1962 as supplemented by AFSG No. 153. |

3.1.2.4.2.4 Recovery Phase.

| | |
|---|---|
| (a) Reference atmosphere | The reference Earth atmosphere for primary and contingency landing sites shall be in accordance with U.S. Standard Atmosphere, 1962 as supplemented by AFSG no. 153. |
| (b) Sea state | These conditions are based on 95 cumulative percent frequency winds and wave conditions at the primary landing sites, and 90 cumulative percent frequency at the contingency landing sites, for the worst month of the year. |

| | |
|---|---|
| Wind velocity | 3 to 28.5 knots |
| Wave height | 0.5 to 8.5 feet |
| Wave slope | zero to 8.4 degrees |

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3.1.2.4.2.5 Launch Aborts. - The requirements of paragraphs 3.1.2.4.2.1, 3.1.2.4.2.3, and 3.1.2.4.2.4 apply to launch aborts consistent with each abort regime except as noted below.

(a) Sea state

These conditions are based on 95 cumulative percent frequency winds and wave conditions at launch abort landing sites for worst month of the year.

| | |
|---|---|
| Wind velocity | 25.0 knots |
| Wave height (crest to trough) | 11.0 feet |
| Wave slope | 8.7 degrees |

(b) Land

These conditions are for launch abort landing sites at Cape Kennedy, Florida

| | |
|---|---|
| Wind velocity | 23.7 knots |
| Soils (Static bearing strength and slope) | |
| Loose sand | 6000 lb/ft$^2$<br>15 degree maximum |
| Hard sand | 25,000 lb/ft$^2$<br>5 degrees maximum |

3.1.2.4.3 Command Module Post Landing Environments. - These requirements represent the natural 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 for 48 hours during exposure to these environments.

(a) Temperature

| | |
|---|---|
| air | 85 degrees F maximum up to 48 hours. |
| sea | 85 degrees F maximum up to 48 hours. |



[redacted]

| | |
|---|---|
| (b) Altitude | Sea level |
| (c) Humidity | 85 percent relative humidity maximum at 85 degrees F. air temperature, up to 48 hours. |
| (d) Solar radiation | 306 BTU/ft.$^2$ - hr. maximum for a maximum of 6 hrs. with linear increase and decrease to zero BTU/ft.$^2$-hr. in 5 hrs. |
| (e) Rain | Up to 0.6 inch per hour for 12 hours, 2.5 inch per hour for 1 hour |
| (f) Sand and dust | As encountered in desert and ocean beach areas, equivalent to 140-mesh silica flour with particle velocity up to 500 feet per minute and a particle density of 0.25 grams per cubic foot. |
| (g) 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. |

(h) Sea state

| | First 48 hours | Next 5 days |
|---|---|---|
| Wind velocity | 3-28.5 knots | 3-40 knots |
| Wave height (crest to trough) | .5-8.5 feet | .5-18 feet |
| Wave slope | 0-8.4 degrees | -------- |

3.1.2.5 Transportability.

3.1.2.5.1 Ground Handling and Transportability. - Full design recognition shall be given to the durability requirements of CSM equipment and subsystems during preflight preparation. Wherever possible, equipment and modules shall be designed to be transported by common carrier with a minimum of protection. Special packaging and transportation methods shall be as required to prevent system penalties.

3.1.2.6 Human Performance.

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

3.1.2.6.1.1 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.

3.1.2.6.1.2 Abort Initiation. - Provisions shall be made for crew initiation of all abort modes. Initiation of abort modes by automatic subsystems shall be provided only when necessary to insure crew safety.

3.1.2.7 Safety.

3.1.2.7.1 Hazard Proofing. - The design of the Spacecraft (S/C) subsystems and support equipment shall minimize the hazard of fire, explosion and toxicity to the crew, launch area personnel and facilities. The hazards to be avoided include accumulation of leakage of combustible gases, the hazard of spark on ignition sources including static electricity discharge, and toxicity due to inhalation or spillage of certain expendables.

3.1.2.7.2 Equipment. - Design of equipment shall be in accordance with MSFC 10M01071, during any part of the mission operation. Where practicable, the various components shall be hermetically sealed or of explosion-proof construction. The rocket motor igniter cartridges shall be capable of withstanding an electrical impulse of 1 ampere or 1 watt dc for 5 minutes without detonating.

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

3.1.2.8 Induced Environment. - These requirements define the induced environmental criteria to which the CSM equipment and associated GSE shall be designed.

3.1.2.8.1 CSM and GSE Ground Environments.

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3.1.2.8.1.1 Transportation, Ground Handling and Storage. - The following represent the induced environmental extremes which may be encountered by CSM equipment and GSE in a non-operating condition during transportation, ground handling and storage. Handling GSE shall be capable of operating during exposure to the environments. Other GSE and CSM equipment may be protected by suitable packaging for transportation and storage if these environments exceed the equipment design operation requirements. The equipment shall be capable of meeting the operating requirements of the applicable performance specification after exposure, while protected by its normal packaging, to these environments.

(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 - Sinusoidal 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 | ±1.56 g | 0.043 | ±6.0 g |
| 50 to 300 | ±1.30 g | 0.036 | ±5.0 g |
| 300 to 1,000 | ±1.30 g | 0.036 | |
| Over 1,000 | ±1.04 g | 0.029 | |

** NOTE: Weight of equipment and package or containers, if any.

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3.1.2.8.1.2 Sheltered environment areas. - These requirements represent the induced environmental design criteria for CSM equipment and GSE both in operating and non operating conditions as determined by normal operational flow sequences. The equipment shall be capable of meeting the operating requirements of the applicable performance specification during and after exposure to these environments. The level of environmental control at each Apollo site shall be as indicated in MSC-GSE-1B.

3.1.2.8.1.2.1 Interior controlled. - An environment in which the temperature, humidity, sand, salt spray, etc., are controlled.

| | |
|---|---|
| (a) Temperature | +60 to +80 F for up to 3 years.<br><br>+52 to +105 F for 1 hour maximum with environmental equipment out of commission |
| (b) Oxygen Atmosphere | The following conditions apply to the CM interior:<br><br>95 ± 5 percent by weight oxygen at total pressures up to 14.7 psia for up to 24 hours.<br><br>Oxygen partial pressure up to 14.7 psia coincident with total pressure up to 21.0 psia for 2 hours. |
| (c) Humidity | 30 to 70 percent for up to 3 years. |
| (d) Sand and Dust<br>(1) Site | 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) CSM | The following conditions apply to the CM interior, and during open fluid systems activity: Particle count not to |



[redacted]

exceed Level 100,000 of Federal Standard 209: No more than 700 particles per cubic foot larger than 5 microns. No more than 35 of these larger than 20 microns.

3.1.2.8.1.2.2 Other environment areas. - Environments to which certain GSE are exposed, such as the environmental chamber, shall be as indicated in MSC-GSE-1B.

3.1.2.8.2 CSM Flight Environments. - These requirements represent the induced environmental design criteria for the CSM equipment in an operating condition as experienced during the various flight mission phases. The exposure time for each mission phase environment shall be the maximum as defined in paragraphs 3.1.1.1.5.1 and 3.1.1.1.5.2 for a 14-day design limit mission, unless otherwise defined in this section. The equipment shall be capable of meeting the operating requirements of the applicable performance specification during and after exposure to these environments. The following are induced environments which are present for all mission phases.

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

(b) Oxygen atmosphere — The following conditions apply to the CM interior: 95 ± 5 percent by weight of oxygen for 336 hours. Nominal CM interior atmospheric composition is presented below:

| 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 |

| | |
|---|---|
| (c) Humidity | The following conditions apply to the CM interior: zero to 100 percent relative humidity for 336 hours, including conditions where condensation takes place in the form of water or frost. |
| (d) 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 1 percent salt solution by weight for 48 hours. |

3.1.2.8.2.1 Ascent Phase.

| | |
|---|---|
| (a) Temperature | The following condition applies to the CM interior atmosphere: 55 F increasing to 90 F maximum. |
| (b) Pressure | The following condition applies to the CM interior: 14.7 psia nominal decreasing to 6.0 psia. |
| (c) 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 9 through 13. |
| (d) Acoustics | Acoustic noise resulting from ground reflection and aerodynamic turbulence. The design acoustics levels for various zones of the CSM are presented in Figures 14 through 26. |
| (e) Acceleration | The design sustained acceleration levels for the CSM are presented in Figures 27 through 30. |
| (f) Aerodynamic heating | The design shall utilize the trajectory shown in Figure 31 plus LES plume impingement where appropriate. |

3.1.2.8.2.2 Earth parking orbit, translunar injection, translunar coast, lunar orbit insertion, lunar operations, transearth coast, and pre-entry phases.

(a) Temperature — The following condition applies to the CM interior atmosphere: 60F minimum to 90F maximum.

(b) Pressure

| Location | Pressure | Max. Exposure Time |
|---|---|---|
| CSM interior (SM and CM forward and aft compartments) | $1.0 \times 10^{-6}$ mmHg | 336 hours |
| CM interior | 6.0 psia decreasing to 5.0 psia | (parking orbit only) |
| | 5.0 ± 0.2 psia (normal) | 336 hours |
| | $1.0 \times 10^{-4}$ mmHg (emergency) | 100 hours |

(c) Vibration — The design levels for the CSM are presented in Figures 32 and 33.

3.1.2.8.2.3 Entry Phase.

(a) Temperature — The following condition applies to the CM interior atmosphere: 60F increasing to 90F.

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

(c) Vibration — The design vibration levels for the CM are presented in Figure 11 when uniformly reduced by 10 db.

[redacted]

| | |
|---|---|
| (d) Acceleration | The design sustained acceleration level is 20 g. |
| (e) Aerodynamic heating | The design shall utilize the trajectories described in 3.4.1.1.2.1. |

3.1.2.8.2.4 Recovery phase.

| | |
|---|---|
| (a) Temperature | The following condition applies to the CM interior atmosphere: 90F increasing to 110F. |
| (b) Pressure | The following condition applies to the CM interior:<br>5.5 psia (nominal) increasing to<br>14.7 psia (nominal) |
| (c) 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 34 and 35 define the accelerations. |

3.1.2.8.2.5 Launch Aborts. - The requirements of paragraphs 3.1.2.8.2.1, 3.1.2.8.2.3, and 3.1.2.8.2.4 apply to launch aborts consistent with each abort regime except as noted below.

| | |
|---|---|
| (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 36 and 37. |
| (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 38 through 43. |
| (c) LES Acceleration | The design sustained acceleration level is 20 g. |



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3.1.2.8.3 Command Module Post Landing Environments. - These requirements represent the induced 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 for 48 hours during exposure to these environments.

| | |
|---|---|
| (a) Temperature | The following condition applies to the CM interior atmosphere: 110F decreasing maximum in 0.5 hours. 50F minimum to 95F maximum for 48 hours. |
| (b) Humidity | The following condition applies to the CM interior: relative humidity up to 100 percent, for 48 hours, including conditions where condensation takes place in the form of water. |
| (c) 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 1 percent salt solution by weight for 48 hours. |

3.2 Interface Requirements.

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

3.2.1.1 Launch Vehicle Attitude Control. - The limit cycle or dead band for the attitude control subsystem of the LV during transposition and docking in pitch, roll and yaw will be as follows:

| | Excursion | Rate |
|---|---|---|
| S-IVB/LEM after CSM separation | ±1.0 deg | ±0.05 deg/sec |
| S-IVB/LEM/CSM prior to CSM separation | ±1.0 deg | ±0.5 deg/sec |
| S-IVB/LEM/CSM prior and during withdrawal of CSM/LEM | ±1.0 deg | ±0.25 deg/sec |



[redacted]

During on-board navigational sightings in earth orbit the limit cycle for the launch vehicle attitude control subsystemwill also be $\pm 1°$ (limited by the $2°$ field of view of the sextant). The pitch, yaw and roll rate will not exceed $\pm 0.1$ degree/second.

3.2.1.2 Propellant Venting. - The S-IVB propellant venting shall be continuous in earth orbit and the thrust generated shall not cause any moment that cannot be corrected within the attitude control subsystem dead band. After trans-lunar injection, the propellant tanks will be vented to a low pressure and sealed such that no vents will occur again until two hours after translunar injection. The $\Delta V$ from this "blow down" vent will be less than 3 ft/sec.

3.2.1.3 Loads Criteria for SLA and Instrument Unit (IU). - The following maximum flight parameters shall not be exceeded on Block II missions.

3.2.1.3.1 Trajectory. - Loads evaluated for the max q condition shall be based on the trajectory given in Figure 31.

3.2.1.3.2 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 44, and the normal force distribution, for the booster only, as given in Figure 45.

3.2.1.3.3 Booster Control System Criteria. - The booster control system simulation for the purpose of loads analysis shall be approximated by the block diagram in Figure 46. 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. The gains listed on Figure 46 are representative for a period of time corresponding to maximum dynamic pressure, 70 to 75 seconds.

3.2.1.3.4 Booster Stiffness. - Distributions of EI and KAG for the Saturn V booster are given in Figures 47 thru 50.

3.2.1.3.5 Booster Weight Distribution. - Weight distributions for the Saturn V booster structure and propellant are given in Figures 51 thru 55.

3.2.1.3.6 SLA/IU Interface Loads. - The limit loads at the SLA/IU interface resulting from the above criteria, corresponding to a dynamic pressure of 690 psf and an angle of attack 10 deg. ($q\alpha$ = 6900 psf deg) are presented on the following page.

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| | S Shear (1000 lbs) | P Axial Load (1000 lbs) | $M_A$ Moment (1000 in. lb.) | $\Delta M_z$ Moment (1000 in.lbs) | $\Delta My$ Moment (1000 in lbs) |
|---|---|---|---|---|---|
| Adapter/IU Interface (STA502) | 83.4 | -270 | 29,500 | 20 | -127 |

Note: $M_A$ is the moment due to the trajectory and trajectory dispersions. $\Delta M_z$ and $\Delta My$ are fixed direction moments are due to the physical design of the vehicle (CG locations, asymmetry, etc.)

3.2.1.3.7 Booster Staging Data. - First stage engine thrust decay data shall be taken from MSFC Memo M-P&VE-PP-96-63, "Estimated F-1 Engine Altitude Thrust Decay for S-IC Stage of Saturn V Vehicle" dated 11 March, 1963. First stage engine thrust misalignment shall be taken from MSFC Memo. F-P&VE-VAD-64-24, "Saturn V Design Ground Rules" dated 17 April 1964. Separation and sequencing data shall be taken from NAA report SID 62-590, "S-II Separation Data Manual" dated 10 June 1962.

3.2.1.3.8 Spacecraft Geometry. - The S/C geometry is given in Figure 56.

3.2.1.3.9 Spacecraft Aero Data. - Loads evaluated for the max q condition shall use the S/C normal force coefficient and center of pressure data given in Figure 57.

3.2.1.3.10 Spacecraft Stiffness. - Distributions of EI, EA and KAG for the S/C are given in Figures 58 through 64.

3.2.1.3.11 Spacecraft Weight Distribution. - Weight distribution for the S/C is given in Figures 65 through 69. This distribution reflects Sector I of the SM empty, a 29,500 lb. LEM. The sum of distributed weight equals the control weight of 21,200 lbs plus 41,210 lbs. of SPS propellant. This corresponds to a total spacecraft injected weight of 95,710 lbs.

3.2.1.3.12 Spacecraft Bending Moment Corrections for Offset Center of Pressure. - S/C bending moment corrections due to the offset center of pressure are given in Figures 70 through 73.

3.2.1.4 Saturn IB Performance Requirements.

3.2.1.4.1 Payload Capability. - For the Saturn IB missions the LV shall be capable of injecting 35,500 pounds, based on a 8,200-pound LES, into a nominal 105-nautical mile Earth orbit.

3.2.1.5 Saturn V Performance Requirements.

3.2.1.5.1 Payload Capability. - For the Saturn V missions the LV shall be capable of injecting 95,000 pounds, based on an 8,200-pound LES and a 29,500-pound LEM into a translunar trajectory of the free return type having a nominal vacuum perigee altitude of 21 nautical miles with no midcourse corrections required to accomplish the trajectory.

3.2.1.6 Launch Vehicle Mechanical Interfaces.

3.2.1.6.1 Physical Interfaces

a. SLA/IU Interface - The SLA shall structurally and functionally adapt the SM to the LV. In the area of interface with the LV, design of the Adapter and the design of the Instrument Unit (IU) shall meet the requirements of ICD's 13M20108 (Saturn IB) "Instrument Unit to Spacecraft Physical Requirements", 13M50103 (Saturn V) "Instrument Unit to Spacecraft Physical Requirements". Requirements established by ICD 13M50123 "Envelope, LEM/SIVB/IU Clearance, Physical" 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.

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" and 13M50112 (Saturn V) "Spacecraft/"Q"-Ball Physical Requirements".

Note: While the effectivity of these documents is limited to missions A201 and A501, the design requirements established therein provide a baseline reference.

3.2.1.6.2 SLA/IU Venting, Purging and Access Requirements.

a. Boost phase venting - During the boost phase the SM, SLA, 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 SLA/IU interface. Total vent cross-sectional area shall be 200 square 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 SM, SLA 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 Marshall Space Flight Center (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.

3.2.1.7 Launch Vehicle Electrical Interfaces.

3.2.1.7.1 Adapter/IU Interface Provision.- Three type PTOOSE-24-61S electrical connectors shall be provided in the Adapter for electrical mating of CSM and launch vehicle. The connectors shall be mounted to the adapter approximately 25 inches above the SLA/IU interface, and approximately 45 degrees from the -Z axis toward the + Y axis (CSM axes, Table I). The applicable document of ICD series 40M37500, Electrical Requirements, Instrument Unit/Spacecraft Interface, shall define the detail electrical requirements of this interface.

3.2.1.7.2 "Q" Ball Interface Provisions. - Wiring shall be provided from the MSFC-furnished "Q" ball to the Adapter/IU interface. Wiring shall be terminated with an 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 3.2.1.7.5.1, a, (3), (f).

3.2.1.7.3 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.

3.2.1.7.4 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. The signal interface shall not produce additional ground connections to the CSM Vehicle Ground Point (VGP); neither shall interface circuitry provide for commonality of signal and power return circuits prior to convergence at the CSM VGP.

3.2.1.7.5 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.

3.2.1.7.5 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 CSM Signals

(1) Automatic abort circuitry - Loss of power in two out of three 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.

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(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 nonthrusting 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 one signal path for SIVB stage burn. On Saturn V missions, five 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 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 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



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"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. CSM to LV signals

(1) LV 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.



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(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.

3.2.1.7.5.2 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 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 MSC (TBD) G&N Performance and Interface Specification.

a. Attitude error - A continuous readout of vehicle attitude error shall be provided during first stage burn.

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.

3.2.2 Guidance and Navigation. - The interface requirements shall be in accordance with Specification MSC (TBD).

3.2.3 Crew Equipments. - The interface requirements shall be in accordance with Specification SID 64-1389.

3.2.4 Scientific Equipment. - The interface requirements shall be in accordance with Specification SID 64-1388.

3.2.5 GFE-ACE. - The interface requirements shall be in accordance with Specification SID 64-1390.

3.2.6 Launch Facilities.

3.2.6.1 Location. - Saturn IB and V launch vehicles with the Apollo spacecraft payload shall be launched from Complexes 37 and 39 at the Eastern Test Range at Cape Kennedy, Florida.

3.2.6.2 Launch Umbilical Tower. - The Launch Umbilical Tower (LUT) supplies prelaunch monitoring and environmental control functions to the SM and LEM through umbilical swing arms and to the Command Module through an access arm.

3.2.6.3 Apollo Access Arm. - A swing arm from the LUT provides a chamber for environmental control of access and egress through the CM side hatch on the launch pad. The access arm is also used for crew loading for launch and as an emergency egress path. An alignment device on the access arm latches to the LES tower legs. Vertical stops provided on the two tower legs facing in the -Z direction shall be each capable of supporting a 500 pound down load. A bellows from the chamber seals against the side of the CM or against the boost protective cover around the side hatch.

3.2.6.4 SM-Launch Umbilical Tower. - The nominal location of the umbilical attachment from the Launch Umbilical Tower SM swing arm to the SM is at station $X_s$ 366.37, 53° off -Z toward the +Y axis. The umbilical plate contains fluid and electrical connectors. The umbilical is pneumatically released shortly before lift-off and utilizes a lanyard back-up. In the event of a jammed release mechanism, separation can occur with a disconnect pull force of 1,000 pounds maximum.

3.2.6.5 SLA-Launch Umbilical Tower Umbilical for LEM. - The nominal location of the LEM umbilical attachment from the Launch Umbilical Tower is on the SLA at $X_A$ 608.25, 73° off -Z toward the +Y axis. The umbilical plate contains fluid and electrical connectors. On Complex 34 and 37 the umbilical is serviced from a separate LEM swing arm. On Complex 39 the umbilical is serviced from the S-IVB Instrumentation Unit swing arm. The umbilical is pneumatically released shortly before lift-off and utilizes a lanyard backup. In the event of a jammed release mechanism, separation can occur with a disconnect pull force of 500 pounds maximum.

3.2.7 Manned Space Flight Control Center (MSCC) and Manned Space Flight Network (MSFN). - The design configuration of the MSCC and MSFN shall conform with the CSM-MSFN data flow requirements as specified in SID 64-1613.

3.2.8 R&D Instrumentation. - (Not Applicable)

3.2.9 LEM. - The interface requirements shall be in accordance with Specification SID 62-1244.

3.2.10 Rendezvous Radar. - The interface requirements shall be in accordance with Specification SID 64-690.

3.3 Design and Construction.

3.3.1 General Design Features.

3.3.1.1 General Arrangement. - The general arrangement of the Block II CSM and Launch Vehicle is shown in Figures 1 and 2.

3.3.1.2 Design Criteria.

3.3.1.2.1 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:

3.3.1.2.1.1 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.

3.3.1.2.1.1.1 Ultimate Factor of Safety. - 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. The following deviations shall apply:

a. The SM structure, exclusive of the CM to SM interface fittings and radial beam truss members, shall be designed to an ultimate factor of safety of 1.4.

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b. The SLA structure shall be designed to an ultimate factor of safety of 1.4.

c. The ERS parachutes, inclusive of risers, end fittings, and the couch shock attenuators (X-X) shall be designed to an ultimate factor of safety of 1.35.

3.3.1.2.1.1.2 <u>Pressure Vessel.</u> - Pressure vessels shall be designed with the following considerations:

a. Pressure vessel limit loads - Limit loads shall be combined 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.

3.3.1.2.1.2 <u>Performance Margins.</u> - 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.

3.3.1.2.1.2.1 <u>Multiple Failure.</u> - 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.

3.3.1.2.1.2.2 <u>Design Margins.</u> - All CSM subsystems shall be designed to zero or positive margins of safety.

3.3.1.2.1.2.3 <u>Attitude Constraints.</u> - Attitude control is permissible to eliminate system constraints which would impose excessive subsystem requirements.



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3.3.1.2.1.3 Performance Criteria.

3.3.1.2.1.3.1 CM Pressurization. - The repressurization subsystem shall be designed for two complete cabin repressurizations and a continuous leak rate as high as 0.2 pounds per hour.

3.3.1.2.1.3.2 Thermal Control. - Thermal design of the Spacecraft modules shall normally use passive means of thermal control, such as insulation, coatings, and control of thermal resistances. Full cognizance shall be taken of thermodynamic considerations in establishing conceptual design and selection of propellants, working fluids, materials for all subsystems such that the maximum allowable temperature range consistent with other design considerations shall be provided. Thermal design shall be based on the following attitude ground rules:

a. Earth Orbit - Spacecraft X-axis is oriented parallel to the velocity vector within ±30°. Duration of this phase will be 1-1/2 to 4-1/2 hours. The spacecraft will be rolled 180° two hours after initiation of this mission phase. Within these limits the worst case roll orientation will be utilized.

b. Translunar Injection Through Transposition - Spacecraft attitude will be prescribed by the required thrust vector for a period of 0.5-2.0 hours and shall be considered random.

c. Translunar Coast - The spacecraft X-axis shall be oriented perpendicular to incident sun with ±20° and shall roll at a rate of 1.5 ± 0.5 rev/hr. This roll shall continue for the entire mission phase of duration 60 to 110 hours except for interruption to accommodate midcourse corrections or contingencies. Thermal design shall allow maintenance of an arbitrary attitude for a maximum period of three hours at any time during this phase to provide for such contingencies. Thermal stabilization shall be determined by reverting to the roll mode and shall be considered as the initial condition for any such contingent operational modes.

d. Lunar Orbit - The spacecraft attitude shall be fixed such that the plus X-axis is apex down within ±20° of the local vertical except for LEM ascent and descent phases during which the attitude shall be arbitrary. In addition, contingency modes shall be accommodated by design for arbitrary orientations during three consecutive orbits, total number of such orbits of arbitrary orientation shall not exceed eight. Thermal stabilization in the lunar orbit mode shall be the assumed initial condition for such contingent operations.

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e. Transearth - The criteria for this phase shall be the same as that for the translunar phase.

f. Instrumentation - Instrumentation shall be provided to indicate the general thermal status of the vehicle during all mission phases.

3.3.1.2.1.3.3 Meteoroid Penetration Mechanics. - The defining equation for penetration mechanics limits thickness sheets is:

$$\frac{\Sigma ti}{d_m} = 3.42K \left( \frac{e_m \cdot V_m}{e_t \cdot C_t} \right)^{2/3}$$

where

$\Sigma ti = t_1 + t_2 = t_3 \text{ --- } t_n$ - total thickness of shielding materials

$K$ = an empirically determined coefficient prescribed for each "target" configuration; e.g., single sheet, multi-wall, pressurized tank, ablative heat shield materials.

$d_m$ = diameter of meteoroid

$e_m$ = meteoroid density

$V_m$ = meteoroid velocity

$e_t$ = shield density

$C_t$ = sonic velocity in shield material

3.3.1.3 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.

3.3.1.3.1 Control Weights (Lbs).

| | |
|---|---|
| Command Module | 11,000 |
| Service Module (1) | 10,200 |
| Total Command and Service Module (1) | 21,200 |
| SLA | 3,800 |
| Launch Escape System (2) | 8,200 |

NOTES: (1) Excluding usable SPS propellant.

(2) Includes boost protective cover and ballast.

Individual CM and SM weights may be varied as required within the limiting Command and Service Module Control Weights shown and as further constrained by any other weight critical component (s) or subsystem (s) subject to NASA concurrence.

3.3.1.3.2 Government-Furnished Equipment. - The following GFE items and associated weights are those used in the above control weights.

| | |
|---|---|
| GFE Total | 31,010 |
| Lunar Excursion Module | 29,500 |
| **Command Module** | |
| Guidance and Navigation | 400 |
| Crew | 528 |
| Crew Equipment | 371 |
| Pressure Garment Assy (3) (including communications) | |
| Constant Wear Garments (7) | |

| | |
|---|---|
| External Thermal Garment (1) | |
| Portable Life Support System (2) (less water) | |
| Emergency Oxygen (2) | |
| Spare Parts and Repair Kit (1) | |
| Biomedical Instrumentation | |
| Radiation Dosimeters | |
| Medical Kit | |
| Instrument Set - Physiological Monitoring | |
| Food Set - 8. 3 day mission | |
| Water Delivery Probe | |
| Survival Equipment Kit (contents only) | |
| Liquid Cooled Garment (2) | |
| Scientific Equipment | 80 |
| Television Camera | 8. 8 |
| <u>Service Module</u> | |
| Rendezvous Radar (including transponder and transmitter) | |
| <u>Launch Escape System</u> | 106 |
| "Q" Ball | 25 |

3.3.2 Selection of 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.

3.3.3 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 kind and quality widely available in supply channels.

d. When practicable, a choice among equally suitable materials and parts shall be provided.

e. Whenever possible, single source items shall be avoided.

f. When practicable, circuits shall be designed with a minimum of adjustable components.

3.3.3.1 Flammable Materials. - Materials that may support combustion or are capable of producing flammable gases (which in addition to other additives to the environment, may reach a flammable concentration) will not be used in areas where the environment or conditions are such that combustion would take place.

3.3.3.2 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.

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

3.3.4 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 1962, Soldering electrical connectors (high reliability) - The soldering of electrical connectors shall be in accordance with specification MSFC-PROC-158A, as amended by MSC-ASPO-S-5B.

3.3.4.1 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.

3.3.4.1.1 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.

3.3.5 Moisture and Fungus Resistance. - 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.

3.3.6 Corrosion of Metal Parts. - 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.

3.3.6.1 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. Protective and coating materials used to satisfy dissimilar metals requirements shall be readily conductive to DC, AC and RF currents.

3.3.6.2 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.

3.3.7 Interchangeability and Replaceability. - 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.

3.3.7.1 Identification and Traceability.- Apollo identification and traceability shall be in accordance with the contractor's approved quality control plans.

3.3.8 Workmanship. (Intentional Blank)

3.3.9 Electromagnetic Interference. - 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 electromagnetic 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, nearby radio frequency sources in the operational environment, and electrostatic potential.

3.3.9.1 CSM and GSE Equipments. - CSM and GSE equipments shall be designed and tested in accordance with specification MC 999-0002B.

3.3.9.2 CSM and GSE Subsystems. - The subsystem shall be designed in accordance with specification MIL-E-6051C.

3.3.10 Identification and Marking. - The CSM and all assemblies, components, and parts shall be marked for identification in accordance with Standard MIL-STD-130.

3.3.11 Storage. - Specific requirements for storage of the CSM are stated in 3.1.2.4.1.

3.3.12 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.

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

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3.3.14 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.

3.3.15 Special Tools. - The functional components of the CSM and component attachments shall be designed so that the use of special tools for assembly, installation, and service shall be kept to a minimum.

3.3.16 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.

3.4 Requirements of Sub-Areas.

3.4.1 Command and Service Modules.

3.4.1.1 CSM Structural Subsystem. - The structural subsystem shall be comprised of the fundamental load-carrying structures. Radiation protection shall be that inherent in the structure designed to carry the loads. Meteoroid shielding shall be provided for critical components only as required to meet reliability apportionments for mission success.

a. Structural Loads - Primary structures are to be designed for the loading conditions as specified below. The loading conditions shall be based on the parameters given in 3.2.1.3.1 through 3.2.1.3.12. These loading conditions shall be considered in the definition of the LEM-SLA interface loads.

(1) Free Standing - The design shall consider the wind criteria given in 3.1.2.4.1.1 (j) (1). This condition may be exceeded during severe thunderstorms or hurricanes 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 percentile winds shall not be experienced by the spacecraft.



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(2) Lift-off - The design shall consider the wind criteria given in 3.1.2.4.2.1 (b).

(3) Atmospheric Flight - The design shall consider the maximum q loads as a result of flight through the synthetic wind profiles given in 3.1.2.4.2.1 (c).

(4) End First Stange Boost and Staging - The design shall consider the loads that result from the cut-off of the booster first stage engines and from separation of the first stage.

(5) End Second Stage Boost - The design shall consider the loads that result from the second stage engines going hard-over at end of boost.

(6) End Third Stage Boost - The design shall consider the loads that result from the third stage engine going hard-over at end of boost.

(7) Spaceflight - The CM docking structure shall be designed for the loads due to docking, RCS firings in the docked configuration and course correction using the SPS.

(8) Entry phase - Primary CM structures are to be designed for a limit load of 20 g during entry.

(9) Noise - The design shall accommodate sound pressure levels in the respective frequency ranges shown in Figure 14 through 26 and 38 through 43.

(10) 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 9 through 17 and 29, 33, 36, and 37.

(11) Dynamic Loads - The calculation of Dynamic Loads shall include the effects of engine start, rebound on the pad, lift off transients including ground winds, gusts, wind shears, boost, engine shutdown, and separation.

(12) Abort - The Launch Escape Vehicle shall be designed to withstand oscillating and tumbling abort conditions except that LES jet impingement pressures will be limited to the structural capability compatible with other loading conditions.

(13) Land and Water Landing - Structural deformation is allowable within the limits of crew safety. Water leakage into the crew compartment of the CM after impact shall be such that there is no impairment of post loading functions (uprighting recovery communications, post landing ventilation, etc.) during a 48-hour period with no crew corrective action. The contractor shall take corrective action to eliminate failure modes resulting in leakage that is identified in the development and qualification program.

3.4.1.1.1 Launch Escape Tower. - 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.

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

Geometric characteristics - The basic external geometry of the CM is shown in Figure (TBD). 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 Figure (TBD).

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

(2) Crew cabin interior - All equipment and structures within the crew cabin shall be free of sharp protrusions which constitute a hazard to the crew or crew equipment.

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 head of the crew in the launch condition. Three 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 unaided egress, within 90 seconds, of the crew on the pad with the boost protective cover installed. In addition, it shall be possible to use this hatch for ingress and egress during space flight for extravehicular operations.

(2) There shall be another inward opening hatch at the forward end of the crew compartment for access to the CM-LEM interface, intravehicular crew transfer between the CM and LEM, and may be used for postlanding egress.

f. Inner structure - The pressure cabin shall be separate from the thermal protection subsystem. The space between the pressure cabin and thermal protection shell shall be vented to limit the collapsing pressures. The venting system shall maintain the cavity pressure within ±1 psi of the altitude ambient pressure. No provision shall be made for over-pressures due to LV explosion except that due to existing structural capability.

g. Docking Provisions - The CM shall include provisions for structurally joining and separating the CSM and LEM during space flight as required for the lunar landing mission design. This capability shall permit the separate spacecraft (CSM and LEM or LEM/S-IVB) to be flown together by flight crew control and be structurally joined to permit intravehicular and extravehicular crew transfer between the CM and LEM and other mission operations. It shall be possible for an unaided crewman to effect docking and crew transfer (either direction). For emergency operations free space crew transfer from the LEM to the CM with no structural tie between the CSM and LEM shall be provided. Initial contact and docked configuration alignments and velocities and interface oxygen leakage shall be as specified in Specification SID 64-1345.

3.4.1.1.2.1 Heat Protection. - The CM shall be designed with a thermal protection shell which will insure that the internal environment of the entry vehicle will not exceed the design limits of the structure and its enclosed subsystems while the CM is entering the Earth's atmosphere. The heat shield shall be designed for the heating rates and heat loads resulting from the NASA furnished entry flight phase design trajectories (see Figure 74). Two heat shield design trajectories will be defined based upon (1) maximum heating rate corresponding to a 20g maneuver load, and (2) maximum heating load corresponding to a 3500 NM entry range. The TPS design shall reflect trim L/D variations between 0.30 and 0.40.

The characteristics of the two trajectories will be as follows:

| | Heat Rate | Heat Load |
|---|---|---|
| L/D | 0.4 | 0.4 |
| Load Limit | 20g | Not applicable |
| Initial entry velocity (Inertial) | 36,333 ft/sec | 36,333 ft/sec |
| Flight path angle (Std. Model) | | |
| Weight | 11,000 | 11,000 |

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| | Heat Rate | Heat Load |
|---|---|---|
| Orbital Inclination | 40° | 40° |
| Atmosphere | 1962 U.S. Standard | Same |
| Atmospheric variations | None | Supplement of 1962 |
| Control mode | ∅ (bank angle) = 0 | MIT Guidance |
| Range | Not applicable | 3,500 nm |
| Entry Interface | 400,000 ft | 400,000 ft |

Allowance for non-standard atmospheric conditions, guidance and control systems deviations and design modifications, and backup trajectory control mode considerations shall be provided for in the shaping of the maximum heating load design trajectory.

3.4.1.1.2.2 CM Flotation and 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.

3.4.1.1.2.3 Portable Life Support System (PLSS). - Facilities shall be provided for resupply of PLSS expendables. Expendables for resupply of the PLSS expendables shall not be provided.

3.4.1.1.3 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 the Electrical Power Subsystem (EPS) and Environmental Control Subsystem (ECS) reactants, and the attachment for the high gain S-Band Antenna and Rendezvous Radar. 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 75 and 76). Equipment contained in each sector shall be as follows.

| | |
|---|---|
| (1) Sector I | Empty |
| (2) Sector II | OPS oxidizer tank |



[redacted]

| | | |
|---|---|---|
| (3) | Sector III | SPS fuel tank |
| (4) | Sector IV | Fuel cells, SPS pressurization package, cryogenic tanks and smaller items of other equipment. |
| (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 27,333 pounds of oxidizer and 13,667 pounds of fuel.**

3.4.1.1.4 Spacecraft LEM Adapter (SLA). - The SLA shall structurally and functionally adapt the SM to the launch vehicle IU and provide housing and support for the LEM. The SLA shall provide for explosive charge separation of the CSM from the adapter in the high altitude SM abort mode, and for CSM separation, docking and LEM withdrawal in the translunar transposition docking mode. At separation, the upper portion of the SLA shall be cut into four panels hinged at the aft end, rotated away from the SM, approximately 170° for normal mission, and secured in position to provide clearance for separation, docking, and LEM withdrawal. The lower fixed portion of the adapter shall provide LEM support and explosive separation devices to permit LEM withdrawal after turnaround docking. The lower fixed portion and upper hinged panels of the adapter remain attached to the S-IVB/IU after spacecraft separation.

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3.4.1.1.5 Docking Subsystem.

3.4.1.1.5.1 Subsystem Performance. - The primary docking subsystem shall allow the CSM and LEM to be safely brought together and structurally joined to permit LEM checkout, intermodular crew transfer through the interconnecting passage, and provide a structural tie of sufficient rigidity to withstand the induced interface loads and subsequently separated. The secondary docking subsystem shall enable the free-space crew transfer with no structure tie between the CSM and LEM.

3.4.1.1.5.1.1 Primary Performance Characteristics. - The following performance characteristics are for the primary docking subsystem.

3.4.1.1.5.1.1.1 Precontact Conditions. - The design of the primary docking subsystem shall ensure a successful docking operation when encountering any combination of the following precontact conditions between the CSM and LEM:

a. Alignment

Radial ± 12 inches
Angular 10° maximum

b. Velocities

| | |
|---|---|
| Axial | 0.1 to 1.0 feet per second |
| Radial | 0.0 to 0.5 feet per second |
| Angular | 0.0 to 1.0 degree per second |

3.4.1.1.5.1.1.2 Final Contact Conditions. - The primary docking subsystem shall, upon engagement of the initial sealing latches, have reduced the precontact velocities to the following:

| | |
|---|---|
| a. Axial (Transposition) | 0.15 fps (max) |
| Axial (Lunar Rendezvous) | 0.35 fps (max) |

3.4.1.1.5.1.1.3 Docked Conditions. - The primary docking subsystem shall, upon completion of the docking operation, have aligned the spacecraft modules to within the following limits:

a. Radial 0.2 inch (max)

b. Angular 0.2 (max)



[redacted]

The primary docking subsystem mechanism shall be capable of engagement at any rotational attitude. Rotational alignment is controlled by pilot operation of the RCS using the visual alignment provisions. Deviations from prescribed relative rotational attitudes, of the CSM and the LEM, will be those associated with proper operation of the visual alignment provisions.

3.4.1.1.5.1.1.4 Docking Weights. - The docking subsystem shall be capable of docking the module weights of the CSM and LEM/S-IVB after translunar injection and of the CSM and the LEM ascent stage after lunar landing and any combination of weights within the above limits which do not exceed the mass and inertia ratios of the lunar orbiting docking case. The weights for the LEM/S-IVB and LEM ascent stage shall be as specified in SID 62-1244.

3.4.1.1.5.1.1.5 Operations. - The primary docking subsystem and spacecraft subsystems will be designed for two docking operations, and the subsequent separations required by the lunar landing mission design.

3.4.1.1.5.1.1.6 Crew Utilization. - In the docked configuration, as the primary mode, an unaided crewman shall be capable of performing all of the functions necessary to accomplish intermodular crew transfer, in either direction, without being exposed to the space environment. During this operation, the crewman shall be clothed in a vented spacesuit. A second unaided crewman shall be able to follow in a similar manner. In addition, as a degraded mode, crewmen shall be capable of accomplishing a similar crew transfer, in either direction, while clothed in pressurized spacesuits.

3.4.1.1.5.1.1.7 Electrical Potential. - Electrostatic compatibility during docking shall be as specified in 3.3.9.

3.4.1.1.5.1.1.8 Leak Rates. - The interface seal and primary docking subsystem shall function so that the initial loss of oxygen during the docking operation will not exceed 1.0 pound. After this initial loss, the interface seal will have a maximum leak rate of 0.1 pound per hour in the space environment.

3.4.1.1.5.1.2 Secondary Performance Characteristics. - The basic performance requirement for the secondary subsystem is the ability of the subsystem to make possible the transfer of the two crewmen and the scientific payload in the event that normal intravehicular transfer cannot be accomplished.

3.4.1.1.5.1.2.1 Crew Transfer Modes. - There are two basic modes of secondary subsystem operation that can be utilized depending on the nature of the Apollo system failures. These modes are:

a. Docked configuration transfer

b. Free space transfer

3.4.1.1.5.1.2.1.1 Docked Configuration Transfer. - The design shall be such that for transfer operations between CSM and LEM in the docked configuration (defined as a minimum, capture latches holding only) an unassisted crewman can perform all of the functions necessary to accomplish extravehicular transfer of himself, a second crewman, and the scientific payload in either direction. During this operation, the transferring crewmen shall be clothed in pressurized spacesuits with portable life support systems (PLSS).

3.4.1.1.5.1.2.1.2 Free Space Transfer. - The design shall be such that transfer operations can be performed from the LEM to the CSM, with no structural tie between the modules. An unassisted crewman shall be capable of performing all of the functions necessary to accomplish the transfer of himself, a second crewman, and the scientific payload. During this operation, the two crewmen shall be clothed in pressurized spacesuits with portable life support systems (PLSS).

3.4.1.1.5.1.2.2 Performance Characteristics. - The secondary docking subsystem shall enable transfer with a maximum relative translational drift rate of 0.1 feet per second, and rotational rate of 0.2 degrees per second at a maximum distance of 50 feet between the CSM and LEM.

3.4.1.2 Mission Support.

3.4.1.2.1 Electrical Power Subsystem (EPS).

3.4.1.2.1.1 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 postlanding recovery, but excluding the PLSS batteries.

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3.4.1.2.1.1.1 Power Output. - The EPS shall be capable of generating 575 kwh of electrical energy from three fuel cells at a minimum rate of 563 watts and a maximum rate of 1,420 watts per fuel cell module. In addition, 3,480 watt hours from storage batteries shall be available.

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

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

3.4.1.2.1.4 Load Grouping. - All electrical loads supplied by the distribution system shall be classified as essential, nonessential, pyrotechnic, or recovery. Essential loads are defined as those loads (except pyrotechnic circuits) that are mandatory for safe return of the 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 postlanding period shall be supplied by the flight and postlanding bus and provisions made for manually disconnecting this bus from the main buses following landing. Redundant busses shall be provided for pyrotechnic circuits and used to supply only that type load.

3.4.1.2.1.5 Cryogenic Gas Storage. - Hydrogen and oxygen reactants for the fuel cells shall be stored in the supercritical state in insulated tanks maintained at constant working pressure.

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

a. Energy sources

Cryogenic gas storage section
storage batteries

b. Power generation equipment

Fuel cell section



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c. Power conversion equipment

Inverters
Battery chargers

d. Power distribution equipment,

Power buses, a-c and d-c,

Associated controls

3.4.1.2.1.7 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. Major component locations shall be as follows:

| | Location |
|---|---|
| Fuel cell modules, radiators, and controls | SM |
| Cryogenic tanks (empty), piping, valves | SM |
| Total reactants, plus reserves | SM |
| Entry batteries | CM |
| Pyrotechnic batteries | CM |
| Battery charger | CM |
| Static inverters | CM |
| EPS display and control panel | CM |

3.4.1.2.1.8 Operating Modes.

3.4.1.2.1.8.1 Normal Operation. - During all mission phases, from launch until reentry, the primary electrical power requirements of the CSM shall be supplied by three fuel cell modules operating in parallel. The storage batteries described in 3.4.1.2.1.1.1 may be utilized to supply required power above the normal capacity of the three fuel cell modules for short duration peaks. Batteries shall be fully charged prior to reentry.

In the event a failure occurs to one of the fuel cell modules, the failed unit shall be capable of being electrically and mechanically isolated from the subsystem and the electrical load required to continue the mission shall be assumed by the two fuel cell modules remaining in operation. The storage batteries may be utilized to supply required power above the normal capacity of two fuel cell modules for short duration peaks. Batteries shall be fully charged prior to reentry.

3.4.1.2.1.8.2 Emergency Operation. - In the event of a failure to two fuel cell modules, the failed units shall be capable of being electrically and mechanically isolated from the subsystem. Spacecraft electrical loads shall be immediately reduced by the crew and manually programmed to remain within the generating capabilities of the one remaining operable fuel cell module. The storage batteries may be utilized to supply required power above the normal capacity of one fuel cell module for short duration peaks. Batteries shall be fully charged prior to reentry.

3.4.1.2.1.8.3 Entry and Recovery. - The fuel cell modules and SM accessories will be jettisoned with the SM. All subsequent electrical power requirements shall be provided by the CM storage batteries.

3.4.1.2.1.9 Mission Abort. - Performance conditions shall be as follows:

a. Fuel Cells - The loss of one fuel cell due to any cause during flight will not constitute grounds for mission abort unless prior to making decision to inject the CSM into the translunar coast phase. The loss of two fuel cells will be a criteria for abort and for deletion of all loads not essential for the safe return of the crew.

b. Cryogenic Storage - The loss of any portion of the cryogenic storage system which makes any one of the four separate reactant sources inoperable is cause for mission abort. This will be reason for deletion of all loads not essential for safe return of the crew.

c. Batteries - Loss of any one of the three main CM batteries will be cause of mission abort. The loss of any one of the redundant EPS pyrotechnic batteries will be cause for mission abort.

d. Inverters - The loss of one inverter shall not be cause for a mission abort. Loss of any two of the three inverters shall be cause for mission abort.

e. Bus Structure - Complete loss of the "Battery Relay Bus" shall be cause for mission abort.

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

3.4.1.2.2.1 Cabin Pressure. - The cabin pressure nominal operating limits shall be 5 ± 0.2 psia. The subsystem shall be capable of maintaining a cabin pressure of at least 3.5 psia for at least 5 minutes following a single 1/2-inch diameter puncture in the pressure compartment.

3.4.1.2.2.2 Oxygen Partial Pressure. - The oxygen partial pressure nominal limits shall be 233 millimeters Hg and emergency conditions 160 mm Hg minimum.

3.4.1.2.2.3 Carbon Dioxide Partial Pressure. - The $CO_2$ partial pressure nominal limit shall be 7.6 mm Hg maximum. In an emergency the limits shall not exceed that given in Figure 77. In the post-landing phase, a maximum of 16 mm Hg $CO_2$ concentration shall be allowable.

3.4.1.2.2.4 Metabolic Action Requirements. - The ECS shall be designed in accordance with the requirements of Table II with the exception of the post-landing phase. For post-landing the following criteria shall be used:

a. Metabolic rate - 800 Btu/hr. per man maximum.

b. Sweat rate - As specified in Figure 78.

c. Drinking water - 12.4 lb/day-man.

d. $CO_2$ production - 3.6 lb $CO_2$/man-day.

e. Allowable effective temperatures - specified in Figure 79.

3.4.1.2.2.5 Temperature Limits.

a. CM temperature - The cabin air temperature nonstressed limits shall be 75 ± 5 degrees F. The nominal and emergency limits shall be as presented in Figure 80 and 81 respectively. For the post-landing phase the effective temperature shall be in accordance with Figure 79.

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

3.4.1.2.2.6 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 80 and 81 respectively.

3.4.1.2.2.7 Equipment Cooling. - The subsystem shall provide thermal control for equipment. No critical equipment shall depend upon the cabin atmosphere for cooling or pressurization.

3.4.1.2.2.8 Subsystem Description. - Environmental control shall be accomplished with a pressure suit circuit, coolant circuit, pressure and temperature control section, oxygen supply section, water management section and a waste management section.

3.4.1.2.2.8.1 Pressure Suit Circuit. - This loop supplies the conditioned atmosphere to the cabin and spacesuit and shall provide removal of debris and noxious gases and for carbon dioxide absorption. Ventilation flow at 3.5 psia and 50°F shall be 12 cfm through each pressure suit. A maximum flow resistance in each suit will be 5 inches of water with flow of 12 cfm of 3.5 psia and 50°F. Provisions should be made for transfer of crewmen between CM and LEM during pressurized and unpressurized conditions.

3.4.1.2.2.8.2 CM Pressure & Temperature Control Section. - This section shall provide cabin ventilation, pressurization and thermal control during all phases of the mission.

3.4.1.2.2.8.3 Oxygen 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. Oxygen shall be provided for LEM pressurization, LEM leakage, docking tunnel leakage and LEM metabolic requirements during transposition docking, LEM checkout in lunar orbit and docking following LEM return.

3.4.1.2.2.8.4 Coolant Circuit. - Dissipation of the thermal load of the CSM shall be accomplished as required by absorbing heat with a circulating coolant and rejecting heat from space radiators and water boilers. The space radiators will be sized such that water boiling will not be required in earth orbit (except for earth launch transients) and cislunar mission phases. Water boiling in lunar orbit shall be limited to rates not exceeding net water production. A completely redundant coolant circuit shall be provided for cooling of critical equipment in the event of a failure of the primary loop.

3.4.1.2.2.8.5 Water Management. - Water shall be collected from the pressure suit circuit 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. Water shall be provided at liftoff to satisfy the crew post-landing metabolic needs and provide for evaporative cooling during exit and re-entry following an immediate abort. A water management program shall be encompassed in the design to provide water requirements for all other phases of the mission. Water shall be supplied to the LEM continuously during the translunar mission phase.

3.4.1.2.2.8.6 Waste Management. - Waste management shall provide ventilation for the refuse storage compartments, and the control of gaseous, solid and liquid wastes from within the CM.

3.4.1.2.2.8.7 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 prevent over-pressurization of low pressure components. Flow limiting devices shall be provided to prevent excessive use of gas supplies and subsequent depletion of such supplies.

3.4.1.2.2.8.8 Preflight Checkout - Accessible fittings shall be provided for performing pressure checks, component performance tests, etc., during pre-flight checkout. These requirements shall prevent the breaking of system integrity for component tests. Provisions shall be made for testing and calibrating all environmental sensors.

3.4.1.2.3 Crew Subsystem.

3.4.1.2.3.1 Subsystem Requirements. - Design and operational procedures shall be in accordance with the crew requirements specified herein.

3.4.1.2.3.1.1 Crew Size and Number. - The CSM design parameters shall accommodate three 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 by appropriate and anthropometric methods.

3.4.1.2.3.1.2 Division of Duties. - Division of duties shall be as follows:

a. Crew duty requirements shall be based on cross-training so that each crew member is able to perform tasks performed by other crew members.

b. Tasks shall be apportioned to make efficient utilization of all crew members.

c. There shall be an established order of command within the crew.

d. Spacecraft design shall recognize the principal distinction in crew duties and designations.

3.4.1.2.3.1.3 Metabolic Parameters. - The average daily metabolic parameters for each crewman are assumed to be as shown in Table II.

3.4.1.2.3.1.4 Environmental Requirements. - The CM interior environment shall be as specified in 3.4.1.2.2.

3.4.1.2.3.1.5 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 5 minutes or less without assistance. At least one crew member shall wear the pressure garment assembly at all times.

3.4.1.2.3.1.6 Food and Water.

a. Food - Provisions for storage of food and associated equipment (GFE), shall be provided within the CM sufficient for a 14-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 provided. One pint of fresh water shall be obtained from each 0.16 pounds of chemical.

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

3.4.1.2.3.1.8 Portable Light. - A portable light shall be provided for illumination of the CM interior.

3.4.1.2.3.2 Subsystem Description.

3.4.1.2.3.2.1 GFE Crew Equipment. - Provisions for the GFE crew equipment are delineated in SID 64-1389, NASA Furnished Crew Equipment Performance and Interface Specification and the Exhibits for the Apollo Block II space suit assembly procurement package, MSC (TBD).

3.4.1.2.3.2.2 Couches. - Couches shall be designed to provide comfortable support during all mission phases. All three crew couch seat pans shall fold to the extent required, to provide necessary work space and adequate access by the crew to all regions of the CM as required.

3.4.1.2.3.2.3 Restraint Subsystem. - A subsystem of restraints shall be provided for crew support and restraint during normal and emergency mission conditions. The restraints shall be similar to that shown in Figure 82, and shall be designed to the acceleration limits of Figures 83, 84, 85, 86 and 88.

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3.4.1.2.3.2.4 <u>Crew Accessories.</u> - Crew accessories shall be provided to assist the crewmen in the performance of tasks under anticipated mission conditions and activities. Specification SID 64-1389 specifies the NASA furnished crew equipment.

3.4.1.2.3.2.5 <u>Window Filter Assemblies.</u> - A window shade assembly shall be provided for all CM windows to attenuate solar heat and visible radiation.

3.4.1.2.3.2.6 <u>Crew Equipment and Suit Interface.</u> - 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. Provisions shall be made for stowage of two PGA helmets and glove pairs allowing rapid access for emergency donning.

3.4.1.2.3.2.7 <u>Personal Hygiene.</u> - Personal hygiene equipment shall be provided to enable crewmen to perform necessary body cleansing during the mission.

3.4.1.2.3.3 <u>NASA-Furnished Crew Equipment.</u>

3.4.1.2.3.3.1 <u>Survival Provisions.</u> - Survival equipment is to be provided which will support and aid in the location and rescue of the crewmen during the post landing situation of 48 hours maximum duration. Climatic conditions do not include extreme cold. Medical requirements shall be satisfied by retrieval of the S/C medical kit. Contents of the survival kit shall be GFE and shall be packaged in the contractor furnished survival package in the CM.

3.4.1.2.3.3.2 <u>Space Suit Assembly.</u> - The NASA furnished Space Suit Assembly components to be stored and/or used in the CM are:

a. Pressure Garment Assembly

b. Constant Wear Garment

c. External Thermal Garment

d. Portable Life Support System

e. Emergency Oxygen System



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f. Space Suit Assembly Spare Parts

g. Liquid Cooled Garment

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

a. (Intentional Blank)

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

Items a through d of the NASA-furnished medical equipment shall be stored in the contractor-furnished medical compartments aboard the CM.

3.4.1.2.3.3.4 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.

3.4.1.2.3.3.5 Radiation Dosimeter. - One radiation dosimeter per crewman will be supplied. The dosimeter will be carried on the crewman's person at all times.

3.4.1.2.3.3.6 Radiation Survey Meter. - One GFE portable, hand-held radiation survey meter will be supplied.

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3.4.1.3 Navigation, Guidance, Control and Propulsion.

3.4.1.3.1 Guidance and Control.

3.4.1.3.1.1 Integrated Guidance and Control Subsystem. - The Integrated Guidance and Control Subsystem (G&C) is comprised of contractor furnished equipment (CFE) designated SCS and Government furnished equipment (GFE) designated GN&C. The contractor portion of the G&C is described in SID 64-1345. The MSC performance and interface document, (TBD), describe the GN&C equipment.

The major interfacing CFE's are the RCS, SPS, EMS, EPS and ECS. The major interfacing GFE is the Rendezvous Radar.

The Guidance, Navigation and Control equipment operation in the overall operation of the G&C Subsystems is shown to aid the understanding of the Guidance, Navigation and Control capabilities and the overall G&C capabilities.

3.4.1.3.1.1.1 G&C Mission Functions. - The integrated Guidance and Control Subsystems shall provide the following functions to guarantee both Mission Success and Crew Survival.

3.4.1.3.1.1.2 Navigation. - The CSM G&C shall provide the capability for navigation during all mission phases including all powered flight phases.

3.4.1.3.1.1.2.1 Primary Inertial Navigation. - The CSM G&C shall be capable of performing the primary inertial navigation functions during all powered flight phases after separation from the SIVB (including abort) and during entry. The powered flight phases include the velocity correction maneuvers during both translunar and transearth injection, midcourse phases, lunar orbit injection, transearth injection, and abort maneuvers. During all Saturn phases the primary inertial navigation function will be provided by the Saturn I. U.

3.4.1.3.1.1.2.2 Primary Navigation. - During Saturn coast phases, during translunar and transearth coast phases and during certain aborts, primary navigation will be provided by the Manned Space Flight Network (MSFN). The CSM G&C shall perform the primary coast navigation functions while in lunar orbit and during all other aborts.

3.4.1.3.1.1.2.3 Secondary Inertial Navigation. - The CSM G& C shall be capable of performing the secondary inertial navigation functions during all Saturn powered flight phases including boost and ascent into orbit and trans-lunar injection.

3.4.1.3.1.1.2.4 Secondary Coast Navigation. - The CSM G&C shall be capable of performing the secondary coast navigation functions during Earth Orbit periods and during translunar and transearth coast. The capability for satisfactory navigation in the presence of SIVB fuel venting shall be provided while in Earth Orbit.

3.4.1.3.1.1.2.5 Powered Flight. - Navigation during the powered flight phases shall be provided by an inertial navigator consisting of a stabilized platform and a digital computer.

3.4.1.3.1.1.2.6 Non-Powered Flight. - Navigation during the non-powered flight or coast phases shall be provided optically (using star sightings and landmark sightings on the Earth and the Moon) in conjunction with the digital computer.

3.4.1.3.1.1.2.7 Data Provision. - The navigation system shall provide the data necessary for the determination of the mission trajectory during all phases and the data necessary for determination of the requirements for the next powered flight phase.

3.4.1.3.1.1.3 Alignment. - The CSM G&C shall be capable of being aligned to any desired coordinate system chosen for mission operations.

3.4.1.3.1.1.3.1 Star Alignment. - The CSM G&C shall be capable of utilizing stars to provide initial fine alignment before every powered flight maneuver, and before earth entry.

3.4.1.3.1.1.3.2 Inertial Alignment. - The CSM G&C shall be capable of IMU alignments with respect to inertial coordinates prior to orbital landmark sightings.

3.4.1.3.1.1.4 Guidance. - The CSM G&C shall generate steering commands during all CSM powered flight maneuvers, during atmospheric entry of the CSM and during either Apollo or Saturn powered flight maneuvers in the event of abort.

3.4.1.3.1.1.4.1 Primary. - The CSM G&C shall be capable of performing the primary guidance functions during all powered flight phases after separation from the SIVB including abort and during entry. The powered flight phases include the velocity correction maneuvers during both translunar and transearth midcourse phases, lunar orbit injection, transearth injection, and abort maneuvers. During all Saturn phases the primary guidance functions will be provided by the Saturn I. U.

3.4.1.3.1.1.4.2 Secondary. - The CSM G&C shall be capable of performing secondary guidance functions during all Saturn powered flight phases including abort into orbit and translunar injection with the SIVB. Secondary guidance functions shall be defined as the generation of steering signals for use in the event of CSM guidance take-over of the Saturn I. U. functions or emergency separation from the SIVB during Saturn powered flight phases. Proper interface with the SIVB Flight Control Sequencer shall be provided to insure satisfactory steering of the SIVB by the CSM G&C.

3.4.1.3.1.1.4.3 Rendezvous. - In the event that active CSM rendezvous with the LEM is required, steering commands shall be provided for CSM guidance during the powered flight maneuvers associated with rescue. These include insertion of the CSM into the desired transfer orbit, the midcourse velocity corrections, the homing maneuvers, and the terminal guidance maneuvers leading to docking.

3.4.1.3.1.1.5 Control. - The CSM G&C shall be capable of providing control signals to the RCS and SPS for control of spacecraft heading during all phases of the mission.

3.4.1.3.1.1.5.1 Attitude Command. - The CSM G&C shall provide spacecraft pointing signals prior to any powered flight maneuver after SIVB separation including entry.

3.4.1.3.1.1.5.2 Powered Flight. - The AGC shall be mechanized to compute steering signals for control of the SPS gimbal system during all powered flight phases after SIVB separation and during entry. The AGC shall also have the capability to perform midcourse corrections with the RCS systems.

3.4.1.3.1.1.5.3 Attitude Hold. - The AGC shall be mechanized to compute rate stabilized signals during attitude maneuvers and during attitude hold operations.

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3.4.1.3.1.1.5.4 BMAG Utilization. - The CSM G &C shall be capable of providing spacecraft pointing maneuver control and attitude hold utilizing the Body Mounted Attitude Gyros (BMAG's) operating in both attitude and attitude rate modes, in the event the AGC is inoperative or turned off.

3.4.1.3.1.1.5.5 SCS Attitude Reference - IMU Alignment. - The SCS attitude reference shall be capable of alignment manually with respect to the IMU reference.

3.4.1.3.1.1.5.6 SCS Attitude Reference - Stars Alignment. - The SCS attitude reference shall be capable of alignment manually with respect to the stars by using the Scanning Telescope.

3.4.1.3.1.1.5.7 Manual Control. - The CSM G &C shall be provided with manual control of steering during all powered and reentry flight phases after SIVB separation.

The CSM G&C manual steering control shall be capable of using AGC steering display data for steering purposes and SCS attitude display for spacecraft pointing control during all powered and reentry flight phases after SIVB separation. The manual steering and pointing signals shall control both the SPS engine gimbals and the RCS engine-on pattern.

3.4.1.3.1.1.5.8 Docking and Rendezvous. - The CSM G& C shall provide attitude control to the CSM during transposition and docking with the LEM/SIVB and during rendezvous and docking with the LEM in lunar orbit.

3.4.1.3.1.1.6 Back-Up. - The CSM G &C shall provide back-up guidance and attitude reference capabilities during all LOR mission phases, and back-up navigation during those phases when MSFN guidance is primary.

3.4.1.3.1.1.6.1 Saturn Powered Flight Phases. - During Saturn powered flight phases the CSM G& C shall provide back-up guidance and attitude reference using the IMU and AGC combination.

3.4.1.3.1.1.6.2 Saturn Coast Phases. - During Saturn coast phases in earth orbit and after translunar injection prior to SIVB separation the CSM G &C shall provide backup attitude reference.

3.4.1.3.1.1.6.3 Earth Orbit. - During earth orbit the CSM G &C shall be capable of providing backup navigation utilizing earth landmarks.



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3.4.1.3.1.1.6.4 Midcourse Navigation. - During translunar and transearth midcourse phases the CSM G&C shall be capable of providing backup navigation during the primary MSFN navigation if needed.

3.4.1.3.1.1.6.5 Rendezvous Radar. - Backup to the LEM guidance and control function after the LEM has been separated from the CSM shall be provided through use of a rendezvous radar. The backup function is in the modes: (1) provide data for active CSM rendezvous, (2) monitor the LEM operation and provide data for the LEM.

3.4.1.3.1.1.6.6 CSM Powered Flight Phases. - During all CSM powered flight phases the CSM G&C shall provide backup attitude reference and guidance using the SCS.

3.4.1.3.1.1.6.7 CSM Attitude Control. - During normal CSM G&C attitude control, backup attitude reference and control shall be provided using the SCS BMAG's.

3.4.1.3.1.1.6.8 Reentry. - During entry the CM G&C shall provide backup attitude reference and roll steering using the entry monitor system and the SCS BMAG's.

3.4.1.3.1.1.6.8.1 Entry Monitor System (EMS). - The EMS shall provide display of the CM reentry conditions for monitoring and/or controlling within acceptable reentry corridor limits. During entry capability for manual steering shall be provided as backup to the primary G&C entry steering.

3.4.1.3.1.1.7 Computation. - The CSM G&C shall be provided with digital and analog computation and data processing facilities suitable to the requirements of all mission phases.

3.4.1.3.1.1.7.1 Steering. - The CSM G&C Apollo Guidance Computer (AGC) shall be mechanized to provide the desired steering signals during all mission powered flight phases and during entry.

3.4.1.3.1.1.7.2 Navigation. - The CSM G&C AGC shall be mechanized to provide navigation capabilities during all mission coast phases.

3.4.1.3.1.1.7.3 Attitude. - The CSM G&C AGC shall be mechanized to provide the necessary attitude angle, attitude rate stabilization, and attitude error signals for spacecraft control during all mission phases.

3.4.1.3.1.1.7.4 <u>AGC Capabilities</u>. - The CSM G&C AGC shall have storage, computation and input-output facilities and program capabilities adequate to all mission needs.

The CSM G&C AGC shall be capable of accepting angle and acceleration data from the IMU, angle information from the G&N optics, inputs from the MSFN, and input commands from the astronaut, and of telemetering required information back to the Earth.

3.4.1.3.1.1.7.5 <u>SCS Capabilities</u>. - The CSM G&C SCS shall be mechanized to provide attitude angle, attitude rate, and attitude error signals for guidance and attitude control during all Apollo mission phases including entry. The CSM G&C SCS shall be capable of accepting angle information from the IMU/AGC from the astronaut.

3.4.1.3.1.1.8 <u>Monitoring</u>

3.4.1.3.1.1.8.1 <u>Saturn Phases</u>. - The CSM G&C shall provide monitoring capabilities during all Saturn phases of the mission.

During all Saturn powered flight phases the CSM G&C shall monitor attitude, attitude rate and attitude error.

3.4.1.3.1.1.8.2 <u>Earth Orbit</u>. - During earth orbit the CSM G&C shall be utilized for monitoring the accelerations due to SIVB tank venting if these accelerations are of sufficient magnitudes to be measured.

3.4.1.3.1.1.8.3 <u>CSM Powered Flight</u>. - During CSM powered flight phases the CSM G&C SCS shall monitor the primary guidance and control system.

3.4.1.3.1.1.9 <u>Data Display</u>. - The CSM G&C shall provide sufficient data display facilities for monitoring, manual steering, and system status indication during all mission phases.

3.4.1.3.1.1.9.1 <u>Saturn Phases</u>. - During Saturn operations the CSM G&C G&N information shall be displayed for monitoring purposes.

3.4.1.3.1.1.10 <u>Abort</u>. - The CSM G&C shall be capable of providing all guidance, navigation, and attitude reference information necessary for successful abort from any phase of the LOR mission after LES jettison.

3.4.1.3.1.1.11 Powered Flight Phases. - The CSM G&C shall provide back-up operation during all mission powered flight phases such that takeover of control, in the event of a failure in the primary G&C during a powered flight phase, will lead to a successful abort.

3.4.1.3.1.1.12 From Translunar Midcourse or Lunar Orbit. - In the event of abort into orbit or abort from translunar midcourse or lunar orbit the CSM G&C shall provide the navigation capabilities, the abort maneuver guidance, and the necessary midcourse velocity corrections defined for crew survival.

3.4.1.3.1.2 G&C Performance Requirements. - During the various mission phases the IGC (Integrated Guidance and Control System) shall function within the mission-dictated limits specified in the following paragraphs.

3.4.1.3.1.2.1 Boost. - The boost phase is defined as existing from pad liftoff to the end of the powered phase for earth orbit insertion. The IGC shall function as specified if any of the following contingencies occur.

3.4.1.3.1.2.1.1 Atmospheric Abort. - After LES separation the IGC shall provide commands to the SPS and RCS such that the CM can be returned safely to earth.

3.4.1.3.1.2.1.2 Extra-Atmospheric Abort. - The IGC shall provide commands to the SPS and RCS such that the CM can assume a safe trajectory and orientation for reentry.

3.4.1.3.1.2.1.3 Abort Into Orbit. - The IGC shall provide commands to the SPS and RCS such that an earth orbit with a perigee above 90 n.m. and an apogee below 450 n.m. can be achieved.

3.4.1.3.1.2.1.4 IGC Takeover of Saturn Guidance. - The IGC must have the capability to provide guidance commands to the Saturn Vehicle in order to achieve an earth orbit suitable for completing the translunar injection after a minimum stay time of 3 orbits.

3.4.1.3.1.2.2 Earth Orbit and Translunar Phases. - The initial condition errors and the translunar injection errors will be such that a total midcourse correction shall not exceed 300 fps. This shall be accomplished in not more than 3 corrections. The lunar trajectories shall have transit time between 60 and 110 hours. The RCS shall provide not more than two vernier corrections with a total velocity correction of more than 5 fps.

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3.4.1.3.1.2.3 Lunar Orbit. - During lunar orbit operations the IGC will be aligned no more than 10 times and a maximum of 20 landmark sightings will be made. During lunar orbit insertion, the IGC shall not use more than 70 feet per second over the nominal fuel requirement.

3.4.1.3.1.2.4 Rendezvous. - During rendezvous the IGC serves as backup guidance and navigation to the LEM G&N system. If the CSM must perform an active rendezvous (i.e., track the LEM and execute maneuvers). The maximum velocity change will be 455 feet per second in a maximum of 10 maneuvers. The navigation and guidance system performance shall be such that at the end of the terminal rendezvous, the relative range and range rate at the docking interface will be 500 ft ± 300 ft and 5 fps ± 3 fps.

3.4.1.3.1.2.5 Transearth Phases. - The initial condition errors and the transearth injection errors will be such that a total midcourse correction shall not exceed 300 fps. This shall be accomplished in not more than 3 corrections. The earth return trajectories shall have transit time between 60 and 110 hours. The RCS shall provide 3 vernier corrections with a total velocity requirement of not more than 15 fps. During transearth injection, the primary guidance system shall not use more than 60 feet per second over the nominal fuel requirement.

In transearth back-up mode where optical navigation is used, the IGS shall have a guidance and navigation performance such that no more than 50 navigation measurements are required to satisfy the entry corridor requirements. The 50 navigation measurements will require not more than 33 specific spacecraft attitude changes. In this mode, the midcourse correction requirements will be the same as for the primary navigation techniques.

In event of failure of the primary guidance system, the SCS system is used for backup transearth injection. It shall have a total pointing accuracy (including initial alignment) of better than one degree rms and a $\Delta V$ magnitude error including the SPS thrust tailoff uncertainty of less than 15 feet per second or 0.5 percent rms of the total injection velocity, whichever is the least.

3.4.1.3.1.2.6 Entry. - The 3 sigma entry corridor (vacuum perigee) depth due to navigation accuracy and midcourse execution errors will be no greater than 20 n.m. In the event of failure of the primary guidance system, the SCS backup will not have a range control capability.



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

3.4.1.3.2.1 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 minimum impulse bit of not more than 2 pound-seconds shall be provided. The subsystem shall have the capability to deplete unused propellant and pressurant prior to CM touchdown. The subsystem shall also be capable of providing three-axis control and/or rate damping as required for CM stabilization during aborts.

3.4.1.3.2.2 Subsystem Description. - The CM/RCS shall be pulse-modulated, pressure-fed, and utilize earth storable hypergolic propellant. Two separate subsystems shall be provided which are capable of independent or simultaneous operation, and each shall be capable of meeting the total torque and impulse requirements. Each subsystem shall consist of pressurization and propellant storage/distribution sections and six thrusters installed to provide three-axis rotational control. Propellant tanks shall be positive expulsion type. Each subsystem shall have the capability for manual isolation of the pressurant and propellant sections.

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

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

c. Pressurant - The pressurant shall be helium (He).

3.4.1.3.3 Service Module Reaction Control Subsystem (SM/RCS). - The SM/RCS shall be used only after CSM separation from the launch vehicle.

3.4.1.3.3.1 Subsystem Requirements. - The SM/RCS shall provide translational and rotational control of the CSM during all unpowered phases and roll control during powered phases. The subsystem shall be disabled on-board for EVA phases. The subsystem shall be capable of supplying a minimum impulse bit of not more than 0.6 pound-seconds. The subsystem shall also be capable of providing the impulse required for the following maneuvers:

a. Spacecraft separation from the launch vehicle.

b. Ullage settling of SPS propellant.

c. Minor velocity corrections of less than SPS capability.

d. SM separation from the CM.

e. Emergency LEM rendezvous and docking.

3.4.1.3.3.2 Subsystem Description. - The SM/RCS shall be pulse-modulated, pressure-fed, and utilize earth-storable hypergolic propellants. The subsystem shall consist of pressurization and propellant storage/distribution sections and 16 thrusters installed to provide 3-axis rotational and translational control. Propellant tanks shall be positive expulsion type. A propellant quantity indicating section shall be provided which is functional in a zero gravity environment. The subsystem shall have a capability for manual isolation of the pressurant and propellant sections.

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

b. Fuel - The fuel shall be a mixture of 50 percent hydrazine ($N_2H_4$) and 50 percent unsymmetrical dimethylhydrazine (UDMH).

c. Pressurant - The pressurant shall be helium (He).

Thermal control of subsystem components in lunar orbit and contingency situations shall be provided by on-off electrical heaters.

3.4.1.3.4 Service Propulsion Subsystem (SPS).

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

a. All major velocity increments required for translunar midcourse velocity corrections, inserting the CSM and LEM into a lunar orbit, for rendezvous of the CSM with the LEM on a backup mode, for lunar orbit maneuvers, for injection of the CSM from lunar orbit into the transearth trajectory, and transearth midcourse velocity correction.

b. Abort propulsion after jettison of the LES.

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3.4.1.3.4.1.1 Propellants and Pressurant. - The SPS shall utilize the following fluids:

a. Nitrogen tetroxide ($N_2O_4$) as the oxidizer.

b. A mixture of 50 percent hydrazine ($N_2H_4$) and 50 percent unsymmetrical dimethylhydrazine (UDMH) as the fuel.

c. Helium (He) as the pressurant.

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

a. Thrust = 21,500 pounds nominal in a vacuum.

b. Minimum specific impulse: Isp = 313 seconds (-3 sigma value) at end of 750 seconds of operation.

c. Operating life = 750 seconds minimum.

d. Minimum impulse bit = 5000 ± 200 pound-second. (1 sigma value).

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

3.4.1.3.4.2.1 Rocket Engine Subsystem. - The SPS engine shall be a single unit, liquid-fueled, pressure-fed, non-throttleable rocket engine, 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.

3.4.1.3.4.2.2 Propellant Subsystem. - The propellant subsystem shall consist of an oxidizer and a fuel supply, each with a storage and sump tank in series, and a distribution system.

3.4.1.3.4.2.3 Pressurization Subsystem. - The pressurization subsystem shall consist of a high-pressure helium supply, contained within two spherical tanks, and associated pressure regulators, isolation valves, check valves, and pressure relief valves.

3.4.1.3.4.2.4 Propellant Quantity Gaging. - A propellant quantity gaging assembly shall be provided. This gaging assembly in conjunction with displays shall provide quantity remaining data to the crew.

3.4.1.3.4.2.5 Propellant Utilization. - A manually operated propellant utilization control shall be supplied.

3.4.1.4 Sequencing, Sensing and Recovery.

3.4.1.4.1 Sequential Events Control Subsystem.

3.4.1.4.1.2 Subsystem Requirements. - 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.

3.4.1.4.1.3 Subsystem Description. - The sequencing subsystem description is delineated in detail in SID 64-1345. This subsystem shall be capable of performing the proper sequencing of events during ascent, LES abort, adapter separation and SPS abort, transposition and docking, LEM abandonment, entry, and providing monitor capabilities.

3.4.1.4.1.4 Pyrotechnic Subsystem and Devices.

3.4.1.4.1.4.1 Subsystem Requirements. - All Pyrotechnic Subsystem electrical circuitry shall provide for redundant design through the electro-explosive interface.

3.4.1.4.1.4.1.1 Standard Electro-Explosive Device. - All electrically-actuated pyrotechnic devices shall be fired by the Apollo standard initiator (ASI).

3.4.1.4.1.4.1.2 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.

3.4.1.4.1.4.1.3 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.

3.4.1.4.1.4.1.4 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.

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3.4.1.4.1.4.1.5 Secondary Effects. - Possible adverse effects of the operation of explosive devices, such as fragmentation or sharp edges, shall be minimized.

3.4.1.4.1.4.1.6 Circuit Isolation. - The firing circuits including power sources for pyrotechnic subsystems shall be separate and independent from all other CM circuitry.

3.4.1.4.1.4.1.7 R.F. Radiation Hazards. - All ordnance devices used at AMR shall comply with design, certification, and validation requirements with regard to R.F. Radiation Hazards as specified in Appendix A of AFMTCP 80-2, Vol. I, General Range Safety Plan, dated October 1, 1963.

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

3.4.1.4.2.1 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.

a. Jettison Capability - Propulsion and trajectory shaping for LES jettison 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 recovery operation. The minimum no-wind range requirement for aborts from a nominal launch vehicle (zero attitude change and zero degree per second attitude rate) shall be 3,000 ft at apogee. The plane of the abort trajectory shall be nominally in a down range direction.

(2) The abort capability shall provide for critical launch vehicle malfunctions which occur at lift-off. The subsystem shall provide recovery at, or above, ground level for the following booster malfunction conditions and associated parameters for abort initiation.



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| | Average Booster Divergence Rate Deg/Sec | Attitude Divergence at Abort Initiation Deg. |
|---|---|---|
| Condition I | (TBD) | (TBD) |
| Condition II | (TBD) | (TBD) |

(3) During approximately the first 40 seconds following lift-off, range safety considerations preclude thrust termination.

(4) Approximately 40 seconds after lift-off, the LV thrust will be terminated automatically at abort initiation.

(5) A minimum separation of the CM from the LV at maximum dynamic pressure shall be 350 feet in 3 seconds following CM separation from the SM. For all aborts, a minimum "miss distance" of 300 feet shall be provided for the case of abort at zero degrees angle of attack and stable flight of the LV. The term "miss distance" is defined as the distance between the LES vehicle and the launch vehicle/SM components at a time when the LES vehicle crosses a plane which contains the launch vehicle and is perpendicular to the launch vehicle flight path. For combined worst off-nominal conditions of launch vehicle trajectory and LES performance, the LES shall be capable of achieving separation and avoiding recontact with the launch vehicle.

(6) The LES shall be capable of performing its function at the maximum dynamic pressure incurred during the boost (see Figure 31), with abort initiated prior to structural break-up of the LV or spacecraft configuration. A minimum capability shall be provided for abort at conditions described as follows.

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

(7) The maximum altitude for LES abort shall exceed the altitude for:

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

(b) Achieving a dynamic pressure condition permitting utilization of a SM abort.

3.4.1.4.2.1.1 Performance Criteria. - The LES shall provide for crew escape from a critically malfunctioning LV from the time of access arm retraction until shortly after the second booster stage separation. Crew accelerations incurred during LES abort and entry following abort shall not exceed the emergency limits of Figures 58 through 88. The following conditions reflect the LEV capability only, and there is no inference that the launch vehicle or spacecraft can achieve these initial abort conditions.

3.4.1.4.2.1.2 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 nominal space vehicle. For combined worst off-nominal conditions of the space vehicle and the LES, the LES shall be capable of achieving separation and avoiding recontact with the space vehicle.

3.4.1.4.2.2 Subsystem Description.

3.4.1.4.2.2.1 Launch Escape Subsystem (LES). - The LES shall include the following components:

a. Q-Ball

b. Launch escape tower canard system

c. Pitch control motor

d. Launch escape motor

e. Structural skirt

f. Tower structure

g. Tower/CM separation system

h. Tower jettison motor

i. Forward heat shield separation and retention system (not included in control weight for LES.)

j. Boost protective cover

k. Tension ties (not included in control weight for LES)

3.4.1.4.2.2.2 Abort Initiation and Control. - 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.

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

3.4.1.4.3.1 Subsystem Requirements. - The subsystem shall satisfy the following requirements after normal entry, and pad escape or launch abort.

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3.4.1.4.3.1.1 Loads. - The ERS parachutes and ELS shall be designed for loads resulting from CM gross launch weight of 11,500 pounds.

3.4.1.4.3.1.2 Recovery Stabilization. - Stabilize the CM during recovery.

3.4.1.4.3.1.3 Velocity Control. - Reduce the vertical touchdown velocity to not more than 35.0 feet per second at sea level.

3.4.1.4.3.1.4 Impact Attenuation. - Reduce impact acceleration such that the CM flotation is not impaired. Any further attenuation required to prevent exceeding the crew acceleration limits delineated in Figures 82 through 84 shall be provided by crewman shock attenuation devices. The CM pitch attitude at impact is nominally negative 30 degrees.

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

a. Provisions for recovery antenna deployment.

b. Provisions for visual location aids.

c. Provision for CM flotation attitude control.

d. Exterior recovery party communications umbilical connection.

e. CM pick up sling.

3.4.1.4.3.2 Subsystem Description. - The ERS shall consist of two actively reefed ribbon type drogue parachutes deployed by mortar and a cluster of three simultaneously deployed, actively reefed Ringsail landing parachutes, crushable honeycomb ribs in the CM impact area, and crushable honeycomb shock structs for crew couch impact attenuation.

Drogue parachutes shall be sized so that any one shall stabilize the vehicle descent, and place the CM in the proper attitude for landing parachute deployment.

Main parachutes shall be sized so that satisfactory operation of any two of the three will satisfy the vertical velocity requirement. Pyrotechnic devices to disconnect the drogue and landing parachutes shall be provided.

3.4.1.4.4 Launch Vehicle - EDS. - The LV-EDS contained in the CSM system will provide the following capabilities.

3.4.1.4.4.1 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.

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

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

b. Manually enable the automatic abort circuitry in the event of failure of automatic enabling at lift-off.

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

d. Manually deactivate the entire automatic abort mode by a single switch.

e. Manually deactivate the automatic abort signal for angular overrates for all three planes simultaneously with one switch.

f. Manually deactivate the first stage 2-engines-out automatic abort signal.

3.4.1.4.4.3 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."

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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.

3.4.1.4.5 Displays and Controls Subsystem (D&C).

3.4.1.4.5.1 Subsystem Requirements. - The location arrangement, and illumination of the displays and controls shall be compatible with mission and crew requirements. Sufficient depth of information and command access to the CSM subsystems shall be provided to enable the three-man crew to accomplish the following operations:

a. Effect manual CSM system management and/or control

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 required.

e. Monitor LEM subsystems as required while docked.

f. Monitor and control functions related to the SIVB as required.

3.4.1.4.5.2 Subsystem Description. - The D&C shall present information to and accommodate control action inputs from the CSM flight crew during the mission as described in 3.1.1.1.5.

The primary location of the D&C equipment shall be the main display console, which is located above the crew couches in the CM. Secondary locations of the 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 so as to establish a D&C subsystem. The master caution and warning subsystem and crew compartment lighting equipment are part of the D&C equipment and shall be provided as defined in SID 64-1345.

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3.4.1.4.5.3 Entry Monitor System. - Entry monitor system displays and controls shall be provided.

3.4.1.5 Information Acquisition.

3.4.1.5.1 Communications Subsystem.

3.4.1.5.1.1 Subsystem Requirements. - Communications capability shall be provided between the CSM and the MSFN, LEM, EVA, launch complex, and the recovery forces. The communications equipment shall be compatible with the equipments with which it interfaces, as illustrated in Figure 89 and described in the following Performance and Interface Specifications:

| | |
|---|---|
| SID 64-1389 | NASA Furnished Crew Equipment |
| SID 64-1613 | CSM-MSFN |
| SID 64-1244 | LEM |
| MSC(TBD) | G&C |

The following types of communications shall be provided:

Two-way voice and voice conferences

Tracking and ranging aids

Telemetry (real-time and stored) transmission

Spacecraft timing

Television transmission

Up-data reception and utilization

Keyed transmission

Recovery aids

3.4.1.5.1.1.1 Voice and Voice Conference.-

a. CSM-MSFN. - Voice communications capability between the CSM and the MSFN to a minimum slant range of 2,000 nautical miles shall be provided by a VHF/AM and an S-Band communication link. Voice communications beyond 2,000 n. mi. shall be accomplished by the S-Band communications link. During extended earth orbital missions, the capability to establish a minimum of one contact per orbit using an HF communications link shall be provided. The orbital HF capability will not be required on lunar missions.

b. CM-MSFN. - Voice communications capability between the CM and the MSFN during the mission phase from CM-SM separation to main parachute deployment, with the exception of entry blackout, shall be provided by the S-Band link.

c. CM-Recovery Forces. - Voice communications capability between the CM and the recovery forces shall be provided from main parachute deployment through the recovery period by a VHF/AM communications link, and for long-distance communications after landing by an HF communications link. The capability shall be provided for use of the crew life raft VHF communications equipment inside the CM after landing through a connection capability with the VHF voice recovery antenna. A recovery hardline and connector shall be provided for deployment after landing to enable voice communications between the crew and recovery personnel outside the CM.

d. CM Intercom. - An intercommunications capability between the crew members inside the CM shall be provided.

e. CSM-EVA. - Voice communications capability between the crew members inside the CSM and an EVA immediately outside the CSM shall be provided by a VHF/AM communications link.

f. CSM-LEM. - Voice communications capability between the CSM and the LEM shall be provided during all line of sight phases of the lunar excursion.

g. MSFN/CSM/LEM and MSFN/CSM/EVA Voice Conferences. - Voice conference capability between the MSFN, CSM, and LEM or EVA shall be provided by use of the CSM to relay voice from the LEM or EVA to the MSFN via the CSM-MSFN duplex S-Band voice link, and to relay voice from the MSFN to the LEM or EVA via the CSM-LEM simplex VHF or CSM-EVA duplex VHF voice links, respectively.

3.4.1.5.1.1.2 Tracking and Ranging Aids. - The CM shall be equipped to permit the MSFN to track the CM at any time during which the CM is in line of sight of an appropriately equipped MSFN station except during entry blackout. During deep space phases of flight, this tracking shall be done by means of the S-Band communications link; during the near-earth phases of flight, either the S-Band or C-Band communications links may be utilized.

a. S-Band Tracking and Ranging. - The CM S-Band communications equipment shall enable the MSFN to determine spacecraft velocity by transmitting a phase modulated carrier to the CM, where the carrier is received and a carrier coherently related at a 240/221 ratio to the frequency of the received carrier is generated and transmitted from the CM to the MSFN. The CM shall enable the MSFN to determine accurately the spacecraft range through the reception, demodulation, and retransmission of a pseudo-random-noise ranging (PRN) signal generated by the MSFN. This ranging capability shall be possible at any time when the MSFN and CM are communicating via the S-Band link, without interrupting any other data transmission under way at that time.

b. C-Band Tracking. - The CM shall be equipped to enable the MSFN to track the CM to a minimum range of 4,000 nautical miles with C-Band radar interrogations and responses. The CM C-Band equipment will only be flown on sufficient missions to verify the adequacy of the Unified S-Band tracking system, and it will not be required for lunar missions.

3.4.1.5.1.1.3 Telemetry. - The CSM shall be capable of gathering, encoding, storing, and transmitting the data required by the MCC.

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a. PCM Telemetry. - The CSM shall have the capability of providing the status of the spacecraft to the MSFN via a PCM telemeter. Two modes of telemetry for a low bit rate output of 1600 bits/second and a high bit rate output of 51,200 bits/second. The capability shall be provided to transmit the PCM telemetry via the S-Band communications link or to record it for later transmission.

b. Scientific Analog Data. - The capability shall exist for transmitting three channels of scientific analog simultaneously with real-time voice, PCM telemetry, doppler tracking, and PRN ranging. This data may be time-shared with tape playback or television transmission.

c. LEM PCM. - The capability shall exist to receive, record, and play back at a later time low bit rate PCM data which shall be transmitted from the LEM to the CSM via a VHF communications link.

d. EVA Telemetry Relay. - During extra-vehicular operations immediately outside the CSM, the capability shall exist for the CSM to relay transmissions from the Extra-Vehicular Suit telemeter to the MSFN via the S-Band communications link. The capability shall exist to check out the EVS telemeter prior to the egressing of the EVA from the CSM, by relaying the EVS telemeter composite waveform to the MSFN for analysis via the S-Band link.

e. Data Storage and Playback. - The capability to store CSM and LEM PCM telemetry and voice and three channels of scientific data shall be provided. The capability shall exist to playback at a 32 to 1 ratio, and transmit via the S-Band link, recorded voice and PCM low bit rate data, or to playback at a 1 to 1 ratio recorded voice and high bit rate PCM for transmission via the S-Band link. The capability shall exist for playback and transmission of recorded PCM and scientific data or PCM and voice simultaneously with real-time PCM, voice, doppler tracking, and PRN ranging. Time correlation for recorded voice shall be provided through simultaneous transmission of recorded voice and recorded PCM, which contains spacecraft time in the data formats. The capability shall exist to record all low bit rate data and voice when real-time transmission is not possible.

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f. Launch Complex Hardlines. - Hardline connections shall be provided between the CSM and the launch complex prior to launch to furnish PCM and analog data to the launch complex and the GSE.

3.4.1.5.1.1.4 Spacecraft Time. - A time reference shall be provided for those CSM functions which require an accurate timing source.

3.4.1.5.1.1.5 Television. - The CSM shall contain a capability of transmitting to the MSFN high resolution pictures of near commercial quality in real time. An analog TV output of 320 lines per frame at a rate of 10 frames per second for transmission to earth via the S-Band link shall be provided.

3.4.1.5.1.1.6 Up-Data Reception. - The CSM shall incorporate decoding and control equipment for use with the S-Band communications link for reception and control of real-time commands, computer and timing information, and for display of information to the crew members. The equipment shall be open for commands during all flight phases in which S-Band communications are possible. The capability shall be provided to enable the crew members to override certain received commands.

3.4.1.5.1.1.7 Key Transmission. - An S-Band transmission mode shall exist that enables the crew members to send keyed information to the MSFN in the event of a contingency condition wherein insufficient RF power is transmitted to enable voice transmission.

3.4.1.5.1.1.8 Recovery Aids. - In addition to the recovery voice functions a VHF beacon and associated recovery antenna shall be provided to assist the recovery forces in locating the CM. The antenna shall be deployed and the beacon automatically actuated at main parachute deployment. The crew shall have the capability to manually override the automatic actuation function.

3.4.1.5.1.2 Subsystem Description. - The major components of the communications subsystem shall include the following items:

3.4.1.5.1.2.1 Contractor Furnished Equipment. -

a. VHF/AM transmitter-receiver

b. Unified S-Band equipment

c. S-Band power amplifier

d. C-Band transponder equipment (early Block II missions)

e. VHF recovery beacon

f. HF transceiver

g. Audio center

h. C-Band Antenna

i. PCM telemetry

j. Premodulation processor

k. Up-data link

l. Teleprinter

m. Signal conditioning equipment

n. Central timing equipment

o. Data storage equipment

p. VHF omni antennas

q. S-Band omni antennas

r. VHF recovery voice antenna

s. VHF recovery beacon antenna

t. HF orbital antenna (non-lunar missions only)

u. HF recovery antenna

v. S-Band high gain antenna

CON[redacted]

3.4.1.5.1.3 <u>Government Furnished Equipment</u>. - The following GFE shall be provided:

a. VHF survival beacon per SID (TBD)

b. Flight qualification recorder calibrator - timer per SID (TBD).

3.4.1.5.1.4 <u>Functions</u>. - The communication subsystem functions and detailed description are contained in SID 64-1345.

3.4.1.5.1.5 <u>Functional Interfaces.</u>

3.4.1.5.1.5.1 <u>Passive Thermal Control</u>. - Attainment of passive thermal control shall have priority over S-Band communications functions requiring use of the S-Band high-gain antenna.

3.4.1.5.1.5.2 <u>Rendezvous</u>. - During the rendezvous and docking maneuvers the utilization of the high gain antenna will not be constrained due to CSM attitude.

3.4.1.5.2 <u>**Instrumentation Subsystem.**</u>

3.4.1.5.2.1 <u>Subsystem Requirements</u>. - A capability to condition and transfer information as to CSM subsystem status during the mission shall be provided on-board the CSM. The following parameters are to be measured:

a. Pressure

b. Temperature

c. Flow

d. Acceleration

e. Rates

f. Quantity

g. Angular Position

h. Current

i. Events

j. Attitude

k. Voltage

l. Frequency

m. $p^H$ factor

n. Reference junction

o. Data distribution

p. Special instrumentation.

3.4.1.5.2.2 Subsystem Description.

3.4.1.5.2.2.1 Operational. - The operational instrumentation shall be physically and functionally compatible with the other subsystems specified in SID 64-1345.

Special instrumentation shall be provided as follows:

a. Nuclear Radiation Detection. - Operational radiation detection instrumentation shall monitor energy, flux rate, dose rate and the accumulated dosage of various types of nuclear radiation encountered during the Apollo mission. The instrumentation shall provide nuclear data outputs to the communications subsystem and CM displays in accordance with the following requirements with the exception of personal dosimeters.

   (1) Indication of external nuclear radiation environment throughout the Apollo mission.

   (2) Indication of directionality of anisotropic proton particles for CSM reorientation to provide maximum radiation shielding.

   (3) Total accumulated radiation dosage indication and high dose rate alarm for each astronaut throughout the Apollo mission.

b. Dosimeter. - Each astronaut shall be supplied with one self-contained NASA/GFE whole body tissue equivalent dosimeter. (Ref. SID 64-1389)

c. 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.

d. Biomedical Instrumentation. - Instrumentation shall be provided at NASA-GSE to monitor biomedical parameters, physiological behavior and reactions to stimuli and environment of each of the astronauts.

3.4.1.5.2.2.2. R and D Instrumentation. - Not applicable.

3.4.1.5.2.2.3 Scientific Equipment. - Space, weight, power and data handling provisions for scientific equipment will be provided. The space allocation shall be 2.7 cubic feet as delineated in Specification SID 64-1388.

3.4.1.5.2.2.4 LEM Information. - There shall be no provisions to acquire information as to the status of LEM subsystems through use of the CSM instrumentation.

3.4.2 Training Equipment. - A program plan shall be provided for training the flight crew, ground operations personnel, and other personnel in the skills and knowledge required for operation of the Apollo system. The program shall be supported with trainers as defined hereunder and in SID 64-1807, Apollo Model Specification for Apollo Training Equipment. These Block II trainers shall be derived by successive modifications of the Block I trainers.

3.4.2.1 Apollo Mission Simulator.

3.4.2.1.1 Trainer Concept. - The Apollo mission simulator (AMS) shall provide for the simulation of the S/C with sufficient realism for the training of the flight crew, MSFN and MSCC personnel, and integrated flight and ground crews in all phases of the total mission. The trainer shall be a fixed based device operating in a controlled environment and capable of providing training in all tasks associated with continuous mission phases as follows: launch countdown, launch, earth orbit, earth orbit rendezvous and docking, translunar including transposition and docking, lunar orbit, lunar rendezvous and docking, transearth, entry and landing. The phases of the mission shall be presented in a continuous fashion without apparent re-programming or switching transients. The trainer shall be capable of providing integrated training and operation with the Manned Spaceflight Control Center (MSCC) and the LEM Mission Simulator. The crew and ground operations functions to be trained shall be those encountered in the operational systems being simulated. The design configuration of the AMS shall be successfully modified to simulate each Block II flight vehicle and its associated booster and mission in turn.

The AMS 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 these alternate procedures. At least one malfunction shall be provided for each alternate procedure. Additional malfunctions will be provided to supply a library of malfunctions related to crew actions and alternatives. Malfunctions for which no crew alternative exists will not be employed.

3.4.2.1.2 Major Equipment Groups. - The trainer shall consist of the following major equipment groups:

a. Simulated CM

b. Instructor-operator station complex

c. Computer complex

d. Simulated SC subsystems equipment

e. Visual simulation equipment

f. Aural simulation equipment

g. Interface equipment

h. Closed circuit television

i. Recording equipment

j. Air conditioning equipment

3.4.2.1.2.1 Simulated CM. - The simulated CM shall be an authentic replica of the CM internally, with respect to size, shape, and equipment location. The simulated CM shall be stationary with the X axis vertical. Controls and displays shall be authentic replicas of the S/C controls and displays.

3.4.2.1.2.1.1 Spacecraft Hardware. - S/C controls, displays and panel designs may be used for training equipment provided these items are modified as required for trainer requirements. Consideration shall be given to the use of less expensive materials than those required for the S/C and to the less stringent environmental requirement for trainers.

3.4.2.1.2.2 Instructor-Operator Station Complex. - The instructor-operator console, which consists of three stations, shall have an arrangement similar to the crew stations of the S/C. The station corresponding to S/C commander shall be master instructor control station. Each station shall have repeater instruments similar to those at the corresponding station in the S/C and other instruments and indicators as required to fulfill the training requirements. Communication facilities shall be provided to permit communications between instructor stations and the crew stations in the CM.

3.4.2.1.2.3 Computer Complex. - Digital computers, analog-to-digital converters and digital-to-analog converters shall be supplied as required.

3.4.2.1.2.4 Simulated S/C Subsystems. - The simulated systems shall be realistic representations of actual S/C systems. Characteristics of components of the systems, such as motors, valves, regulators, shall be included, as required, to produce the static and dynamic performance of the systems under simulated normal and malfunction conditions.

3.4.2.1.2.5 Visual Simulation Equipment. - Simulated external visual stimuli shall be presented to the three crew members through the CM windows, except the hatch window, and the telescope and sextant. Objects viewed through the windows and optical instruments which are pertinent to crew training shall be simulated to a degree of accuracy consistent with the training requirements. The images presented to the windows and optical instruments shall be of such resolution, brightness, distortion level, and accommodation as to be realistic and subjectively acceptable.

3.4.2.1.2.6 Aural Simulation Equipment. - Aural simulation equipment shall provide background noise and specific event aural cues through loud speakers in the CM and through crew members' headsets.

3.4.2.1.2.7 Interface Equipment. - Interface equipment shall be provided as required to permit the trainer to operate independently, integrated with the MSCC, integrated with the LEM Mission Simulator (LMS), or integrated with both the LMS and the MSCC.

3.4.2.1.2.8 Closed Circuit Television. - Television cameras shall be installed in the simulated CM to permit visual monitoring of crews, and the interior of the CM. Camera location shall be out of the crew's normal field of view so far as practicable. Sensitivity of the cameras shall be sufficient to present a clear picture to the instructor-operator station complex monitoring scopes, using normal CM interior illumination.

3.4.2.1.2.9 Recording Equipment. - Magnetic tape recording equipment shall be provided for communication and trainer data playback after training sessions. Analog parameter recording equipment shall be provided for recording of selected dynamic time histories. Provisions shall be made to permit the selection and recording of those trainer variables indicative of significant characteristics of the training exercise.

3.4.2.1.2.10 Air Conditioning Equipment. - Air conditioning equipment shall provide ambient temperature control in the simulated CM and the pressure suits, and pressure suit pressurization. Plenum cooling air will be supplied to simulator components by the using facility.

3.4.3 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 during tests at factory subsequent to manufacturing completion; prelaunch and post launch operations at the test site; and major developmental tests such as house CSM tests, propulsion tests and environmental tests. The Master Ground Operations Specification, (Block II) - (number to be defined) further defines GSE requirements.

3.4.3.1 GSE Concept.

3.4.3.1.1 Design Concept. - The GSE design concept delineates four general categories of equipment for supporting servicing, handling, checkout 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.

3.4.3.1.2 Operations Support. - 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.

3.4.3.1.3 CSM Checkout Concept. - CSM checkout shall consist of:

a. Local/manual operations

b. Remote/semi-automatic operation

c. Remote/manual operation

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The local/manual and remote/semi-automatic operations shall be performed with the acceptance checkout equipment. Adequate flexibility shall be incorporated to accommodate frequent changes. The equipment required for hazardous operations shall be designed for remote/manual operation from protected areas. NAA shall provide all carry-on equipment for ACE usage, all flight hardware required for checkout of the CSM with ACE, and all carry-carry-on equipment for checkout of the Guidance and Navigation subsystem with ACE. Refer to GFE-ACE Specification SID 64 - 1390.

3.4.3.1.4 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 subsystems not installed in the vehicle may be accomplished by bench test equipment (BTE). BTE shall be limited to local manual operation.

3.4.3.2 Support Requirements. - The level to which CSM GSE shall support the vehicles, operations, and sites is as follows.

3.4.3.2.1 Test Preparation and Acceptance Area. - Equipment shall be provided in the test preparation and acceptance area to functionally check-out S/C 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 cell and cryogenic subsystems and associated servicing equipment will not be conducted in this area.

3.4.3.2.2 House Spacecraft. - GSE for house S/C shall perform tests for the following purposes:

a. Engineering development

b. Field operations support (ACE-S/C 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 -S/C capabilities are developed. Servicing, handling, checkout, and auxiliary equipment shall be provided as required.

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

3.4.3.2.4 Las Cruces Propulsion System Development Facility. - Support of test preparation and firing preparation shall be in the local/manual mode with equipment having diagnostic capability as specified in 3.1.2.2.2. 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.

3.4.3.2.5 Space Environmental Simulation Laboratory. - Support checkout of the CSM shall be accomplished with ACE-S/C 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.

3.4.3.2.6 Atlantic Missile Range. - Equipment shall be provided for the complete functional checkout of the S/C 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 subsystems, and operation and verification of the fuel cell, cryogenic and the environmental control subsystems. Individual subsystem and integrated systems tests shall be conducted in the operations and checkout building. Tests and checkout in the operations and checkout building and at the launch complex shall be designed for use of ACE-S/C equipment.

3.4.3.2.7 Marshall Space Flight Center (MSFC). - Handling and auxiliary GSE will be utilized at MSFC to support dynamic and umbilical tests.

3.4.4 Other Equipment. (intentional blank)

# 4.0 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 and NPC 200-3. 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 1 to the Apollo Contract, NAS9-150. NAA/S&ID shall prepare and submit to NASA a Quality Assurance Program Plan per the requirements of Exhibit I.

4.2 Reliability Program. - NAA/S&ID shall establish a Reliability Program in accordance with NASA Publication NPC 250-1. Implementation of this document shall be as specified in the NAA/S&ID Reliability Program Plan (SID 62-203).

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 definitions and ground rules for establishing this program are as follows:

4.3.1 Definitions.

a. Qualification tests - Functional tests performed on production hardware at and above mission levels of all critical environments to assure that the hardware will meet the design requirements and will perform its function for its use cycle.

b. Criticality - Criticality describes the impact of failure of equipment (part, component, subsystem) on crew safety or mission success. Criticality is non-numerical and is classified as follows:

Criticality I - Those items whose failure may result in loss of crew.

Criticality II - Those items whose failure may result in loss of mission.

Criticality III - Those items whose failure does not affect mission success or crew safety.

[redacted]

c. Production hardware - Hardware that is manufactured with the same tooling, processes, quality control procedures, and to the same design as to that which will be used in manned flight.

d. Failure - The inability of a system, subsystem, component, or part to perform its required function with specified limits under specified conditions for a specified duration.

e. Operational cycle - The period of time extending from the beginning of acceptance tests to the end of a mission that a system, subsystem, component or part is expected to operate under sequential application of environments, induced or natural; including, but not limited to, end-item test time, acceptance time, checkout time, transportation and handling time, the planned mission time, and any critical abort or emergency conditions. The planning mission for qualification shall be 14 days duration. Any deviation as to mission duration shall be submitted to NASA for approval.

4.3.2 Ground Rules.

a. The qualification program is limited to tests conducted on individual parts, components, subassemblies, assemblies, and subsystems. The qualification program shall consist of a series of tests at any or all assembly levels listed above. Generally, these shall occur only at the highest practical level of assembly. If tests are required at several levels, those at lower levels shall be initiated prior to those at higher levels of assembly.

b. Production hardware shall be used throughout.

c. Acceptance tests shall precede all qualification tests.

d. No refurbished equipment shall be used without specific NASA approval.

e. Functional operation is required. During all qualification tests all interfaces shall be present or simulated.

f. Adjustments will be permitted during an operational cycle only if they are part of a normal procedure.



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g. Limited life items and single shot devices may be replaced at the conclusion of satisfactory operation to meet their life time requirement.

h. Any failure shall be cause for positive correction action. The degree of retest in event of failure shall be agreed upon between the NASA and the contractor after evaluation of the failure. In event of failure, the contractor shall immediately advise NASA.

i. Requalification shall be performed when:

(1) Design or manufacturing processes are changed to the extent that the original tests are invalidated.

(2) Inspection, test, or other data indicate that a more severe environment or operational condition exists than that to which the equipment was originally qualified.

(3) Manufacturing source is changed.

j. Qualification by similarity may be accepted provided:

(1) The item was qualified to the Apollo environmental levels, and,

(2) The item was fabricated by the same manufacturer with the same processes and quality control.

(3) The item was designed to equivalent specifications as required in the Apollo designs.

k. Qualification tests shall include [illegible] environments [illegible] as required, the anticipated environments during the operational cycles in level, range, and sequence. Combined environments shall be used when necessary and practical.

l. Where redundancy in design exists, the qualification test program will assure that each redundant component will be included in the test program.

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m. Qualification test specifications shall be written for each item and the qualification test program will fully encompass the design specification requirements.

n. As a general rule, it is not economically practical or feasible to conduct qualification tests on complete subsystems. Accordingly, most of the qualification tests should be conducted on lower levels of assemblies to the degree necessary to provide confidence on a subsystem basis. This will be done by conducting tests at each hardware level such that when the total qualification program on a subsystem is completed all items of hardware and all operational modes will, as a minimum, be tested an amount equivalent to a subsystem qualification test. This is commonly called an "equivalent" subsystem.

o. The qualification program shall be established in two phases, namely, that required to support Block I vehicle missions and secondly, Block II LOR missions. The program should be established so that hardware required for qualification is selected on a time phase basis throughout the production program as an objective. The number of units required prior to the first manned Block I flight should be reduced to a minimum. In determining the number of units required for qualification, all prior development tests including integrated ground tests should be considered in determining the number of units required. Portions of the development tests may be used to reduce the qualification test program provided all qualification requirements are met and prior NASA approval is obtained.

p. Qualification tests supporting a particular vehicle shall be completed prior to that vehicle being delivered from the contractor's plant. The minimum qualification will include one set of equipments subjected to sequential, singly applied environments at design limit conditions, and another set subjected to one operational cycle and one subsequent mission cycle at nominal mission conditions. For the unmanned Saturn IB flights, that portion of the above qualification tests supporting these vehicles shall be completed prior to launch and shall comprise one set of equipment subjected to sequential, singly applied environments consistent with the particular mission exposure of the spacecraft, and another set subjected to one cycle of the ground portion of the operational

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cycle and the applicable portion of one mission cycle to levels consistent with the particular mission exposure. Successful completion of the first cycle shall constitute interim qualification with final qualification contingent on completion of the second cycle. Limited life items may be refurbished before start of the second cycle.

q. Subsequent to the completion of the qualification test program further tests shall be conducted at conditions more severe than design-limit. The purpose of these tests shall be to determine failure modes actual design margins.

4.3.3 Spacecraft Development Test. - The S/C development tests shall be in accordance with the Development Test Plan, SID 64-1707.

## 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.

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## 5.0 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.

# 6.0 NOTES

## 6.1 Definitions.

6.1.1 Reference Axes. - The reference axes of the SCM shall be orthogonal and shall be identified as shown in Table I.

6.1.2 System. - The CSM System is composed of a Launch Escape Subsystem (LES), a Command Module (CM), a Service Module (SM), a Spacecraft LEM Adapter (SLA), the associated Ground Support Equipment, and the requisite Trainers.

6.1.3 Spacecraft. - The spacecraft consists of the airborne portion of the CSM system and includes the LES, CM, SM, SLA, and associated airborne GFE, as applicable to mission phases.

6.1.4 Subsystem. - A subsystem is a combination of equipment designed to perform a specific function such as communications, environmental control, launch escape, and earth recovery.

6.1.5 Acronyms. - The acronyms peculiar to the Apollo Program referenced in this specification are listed alphabetically as follows:

(ACE-S/C) Acceptance Checkout Equipment Spacecraft

(CM) Command Module

(CM/RCS) Command Module Reaction Control Subsystem

(CSM) Command and Service Module

D & C) Displays and Controls Subsystem

(ERS) Earth Recovery Subsystem

(EPS) Electrical Power Subsystem

(EDS) Emergency Detection System

(ECS) Environmental Control Subsystem

(GFE) Government Furnished Equipment

(GSE) Ground Support Equipment

(IU) Instrumentation Unit

(G & C) Integrated Guidance and Control Subsystem

(LES) Launch Escape Subsystem

(LV) Launch Vehicle

(LEM) Lunar Excursion Module

(LOR) Lunar Orbit Rendezvous

(MSCC) Manned Space Flight Control Center

C[redacted]L

(MSFN) Manned Space Flight Network

(MSFC) Marshall Space Flight Center

(PLSS) Portable Life Support System

(RCS) Reaction Control Subsystem

(SM) Service Module

(SLA) Service Module LEM Adapter

(SM/RCS) Service Module Reaction Control Subsystem

(SPS) Service Propulsion Subsystem

(S/C) Spacecraft

(SCS) Stabilization and Control Subsystem

(WSMR) White Sands Missile Range



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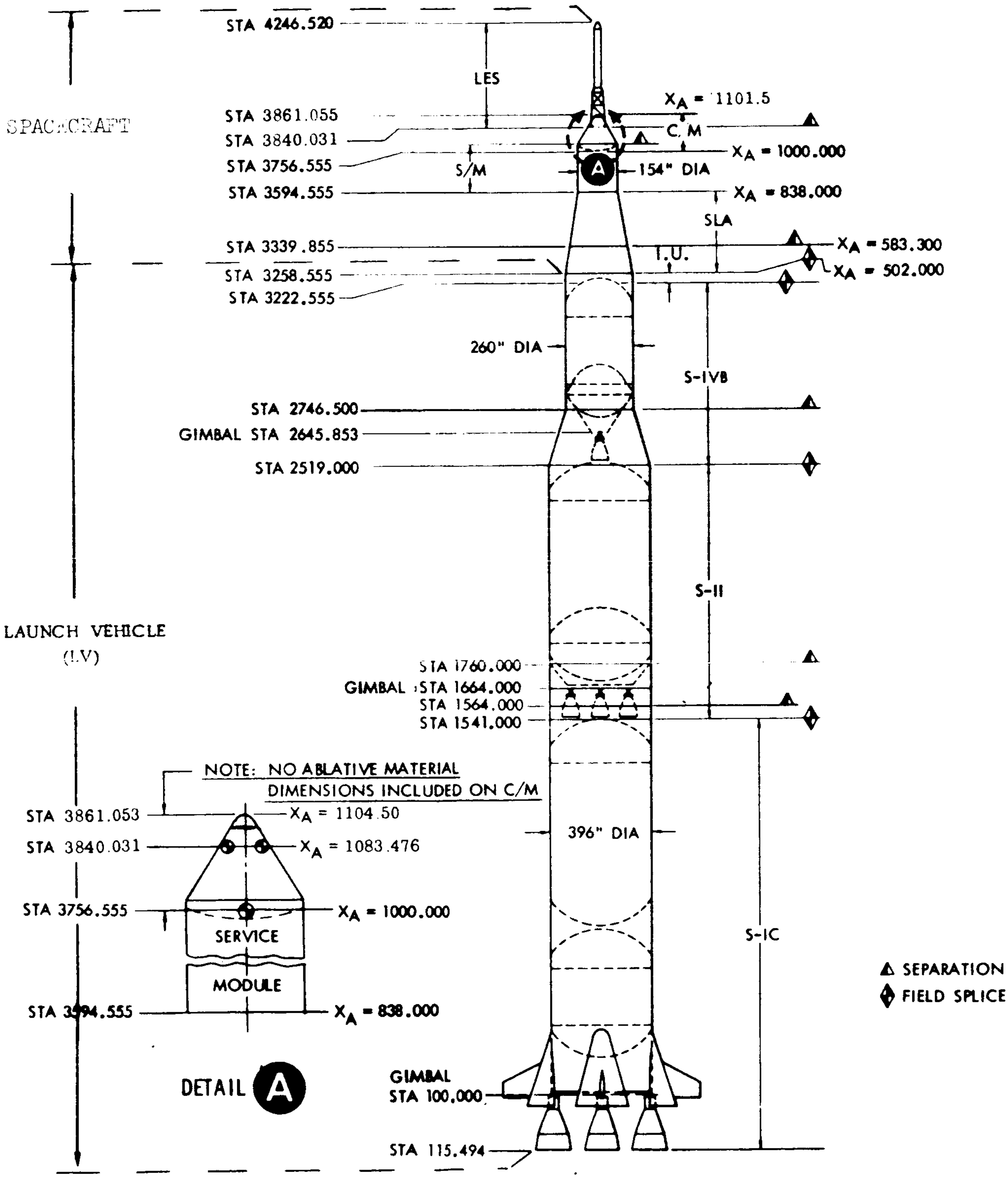


Figure 1. Saturn V LOR Configuration



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SID

NOTE: Art to be updated.

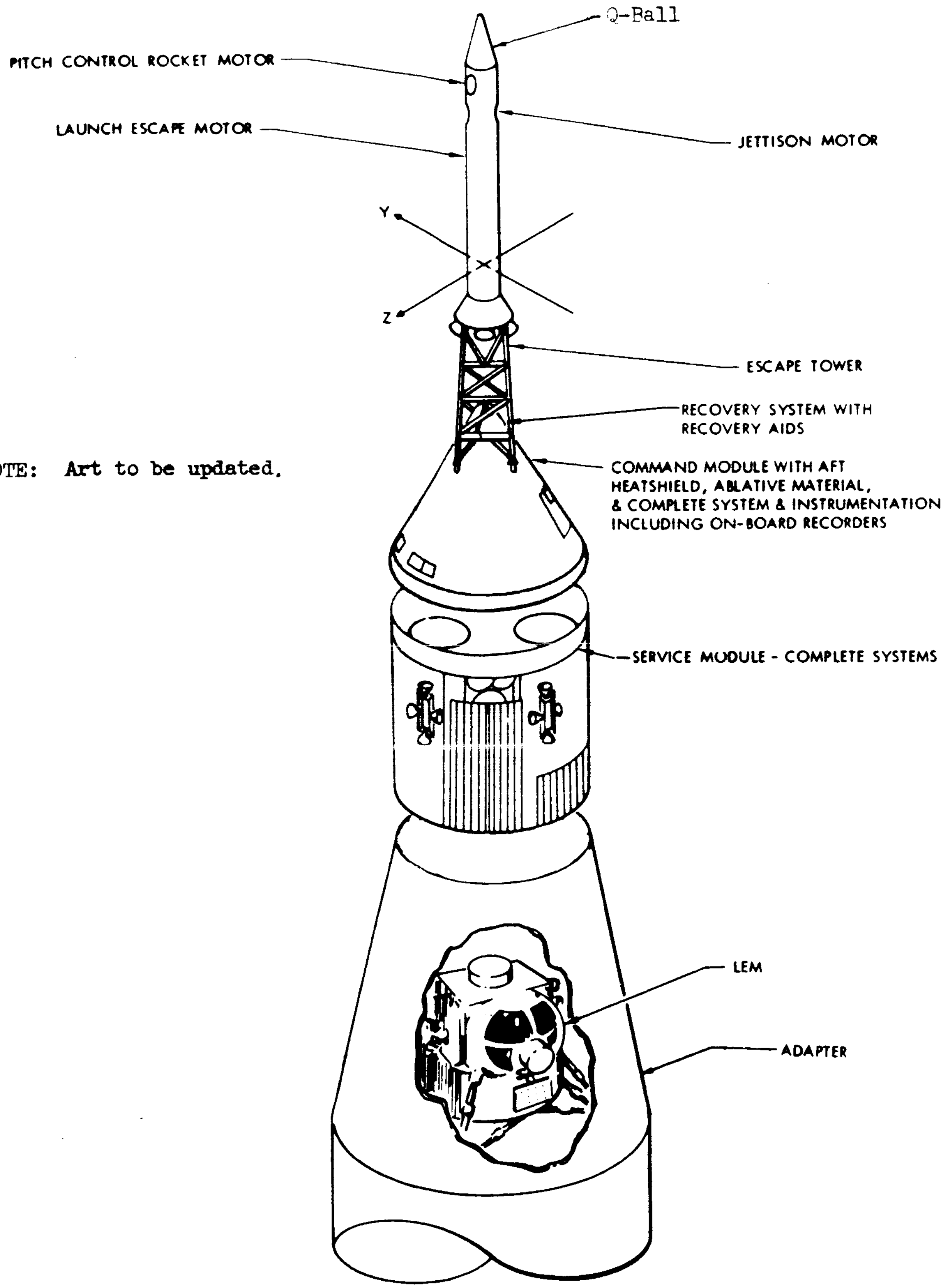


**Figure 2. Saturn V,** Lunar Landing Mission - Manned Flight

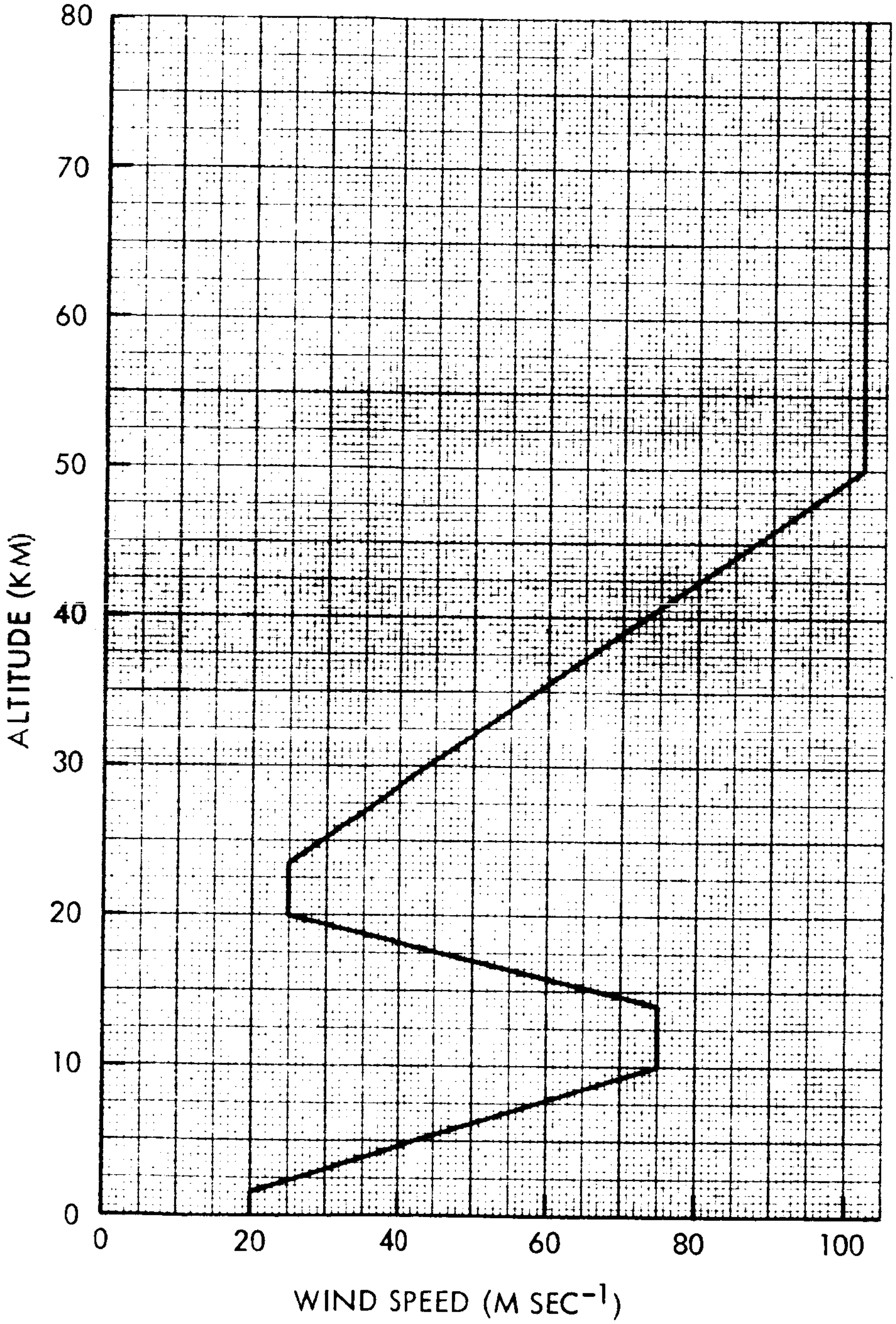


Figure 3. Scalar Wind Speed Profile Envelope (95 Percentile) for Atlantic Missile Range

[redacted]

| WIND SPEED IN THE 7 TO 15 KM ALTITUDE REGION | SCALE-OF DISTANCE | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| ($M\ SEC^{-1}$) | 5000 | 4000 | 3000 | 2000 | 1000 | 800 | 600 | 400 | 200 | 100 |
| | WIND SPEED CHANGES | | | | | | | | | |
| ≥ 70 | 52.3 | 50.3 | 46.9 | 40.8 | 30.0 | 26.9 | 23.3 | 18.7 | 11.9 | 7.6 |
| 60 | 42.5 | 40.9 | 38.2 | 32.9 | 23.8 | 21.1 | 18.1 | 14.4 | 10.4 | 7.6 |
| 50 | 34.8 | 33.5 | 32.3 | 27.8 | 20.3 | 18.2 | 15.7 | 12.8 | 9.4 | 7.6 |
| 40 | 28.0 | 27.2 | 26.4 | 23.0 | 17.6 | 15.9 | 14.0 | 11.6 | 9.0 | 7.6 |
| 30 | 22.1 | 21.4 | 20.6 | 18.4 | 14.4 | 13.3 | 11.8 | 10.2 | 8.4 | 7.6 |
| 20 | 16.2 | 15.8 | 15.6 | 14.4 | 11.9 | 11.2 | 10.2 | 9.3 | 8.1 | 7.6 |
| WIND SPEED IN THE 50 TO 80 KM ALTITUDE REGION | SCALE-OF-DISTANCE | | | | | | | | | |
| ($M\ SEC^{-1}$) | 5000 | 4000 | 3000 | 2000 | 1000 | 800 | 600 | 400 | 200 | 100 |
| | WIND SPEED CHANGES | | | | | | | | | |
| ≥100 | 68.0 | 64.6 | 59.8 | 52.7 | 38.2 | 33.2 | 27.5 | 20.7 | 12.8 | 7.6 |

**Figure 4.** Envelopes of Wind Speed Change For Various Scale-of-Distance Corresponding to Wind Speeds in the 7 to 15 KM and 50 to 80 KM Altitude Region (85% of the 99 Percentile Wind Shear)

[redacted]

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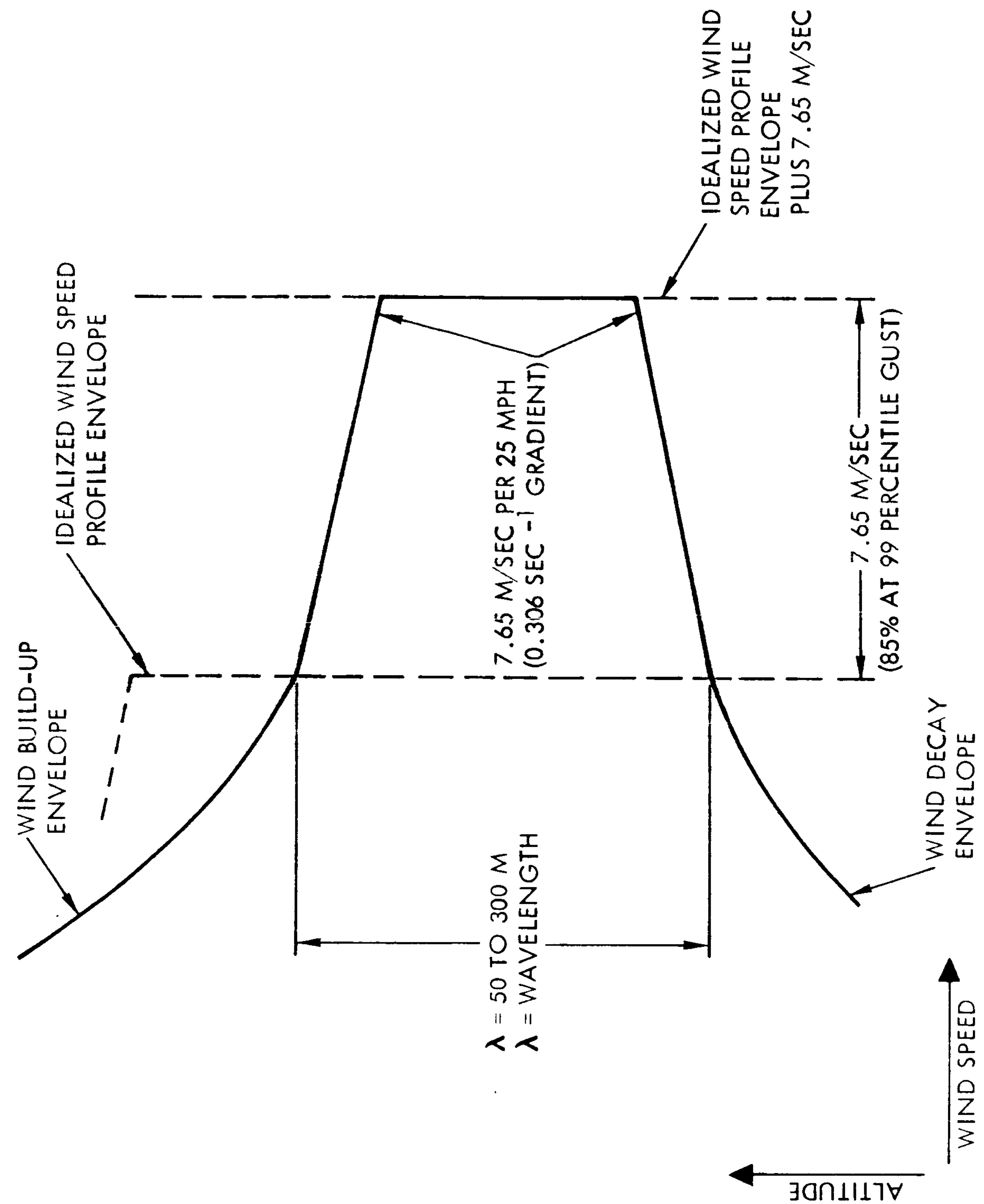


Figure 5. Relationship Between the Quasi-Square Gust and the Wind Speed Profile Envelope

| QUASI-STEADY STATE WIND SPEED ($\pm$ 5 M $SEC^{-1}$) | MINIMUM THICKNESS (KM) | MAXIMUM THICKNESS (KM) | ALTITUDE RANGE (KM) |
|---|---|---|---|
| 50 | 0 | 5 | 8.5 TO 16.5 |
| 75 | 0 | 3 | 10.5 TO 15.5 |

Figure 6. Allowable Altitude Thickness of Synthetic Profile Peak Wind

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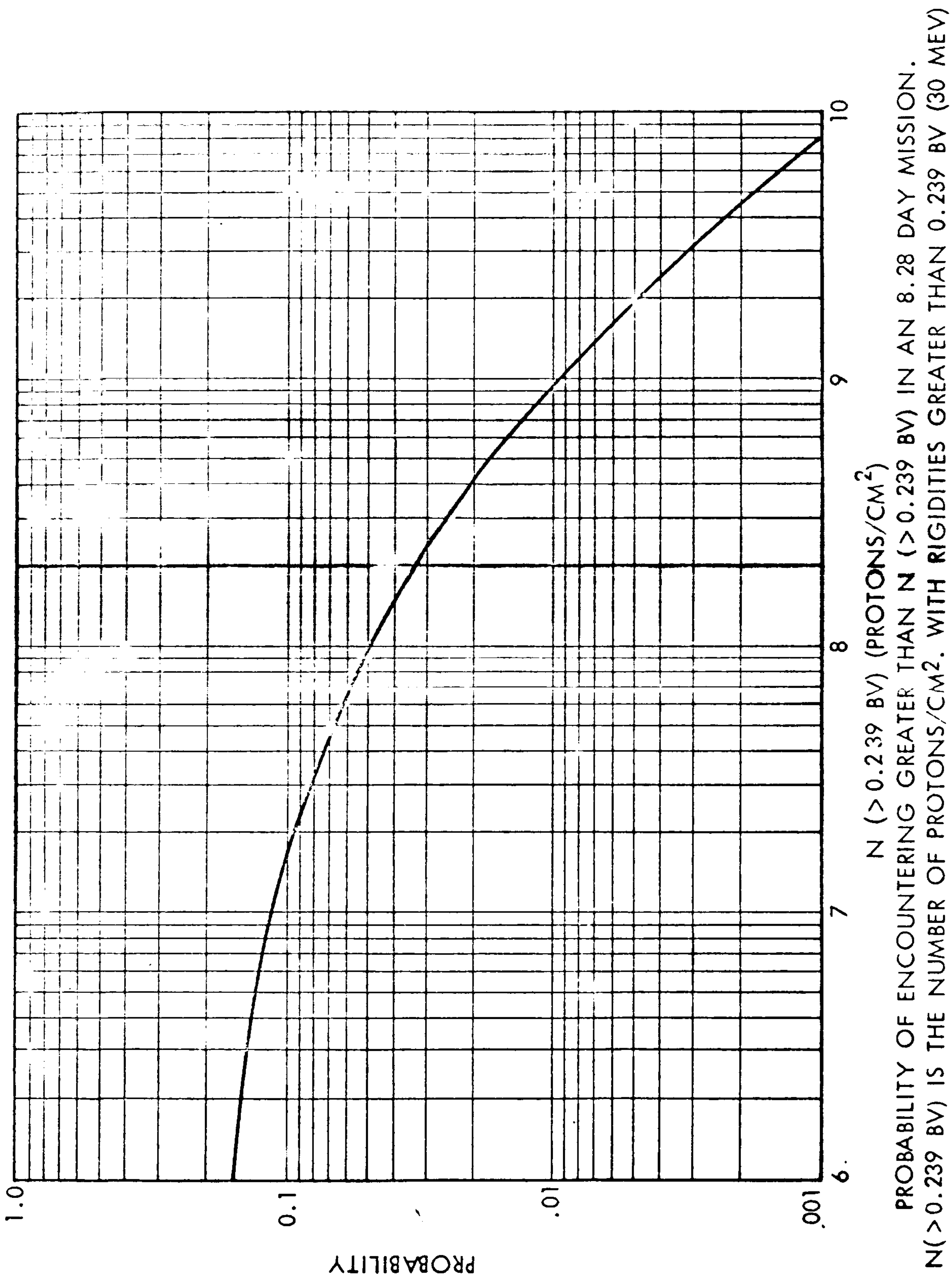


Figure 7. Probability - Solar Particle Events



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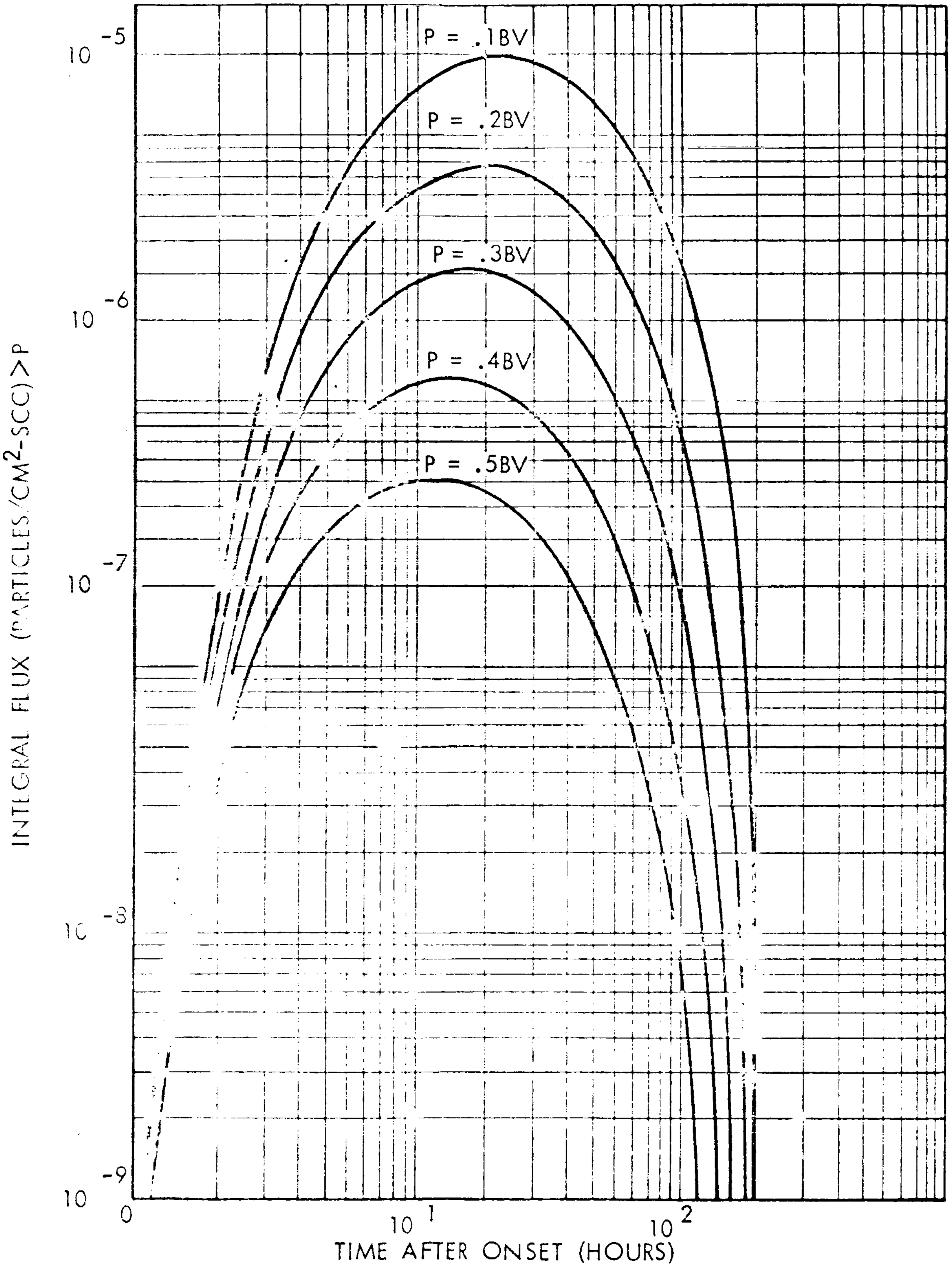


Figure 8. Normalized Model Time Dependent Integral Spectrum

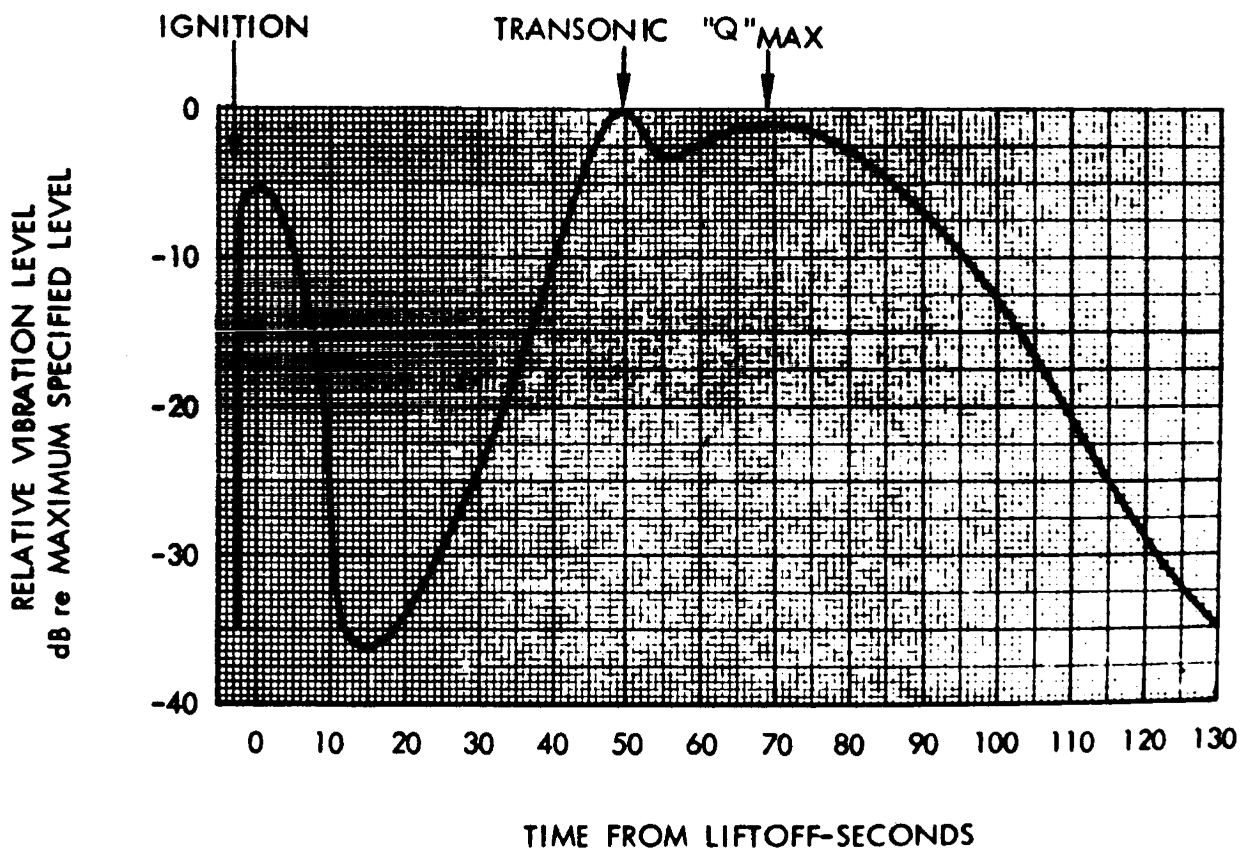


Figure 9. Vibration Time History - Atmospheric Flight

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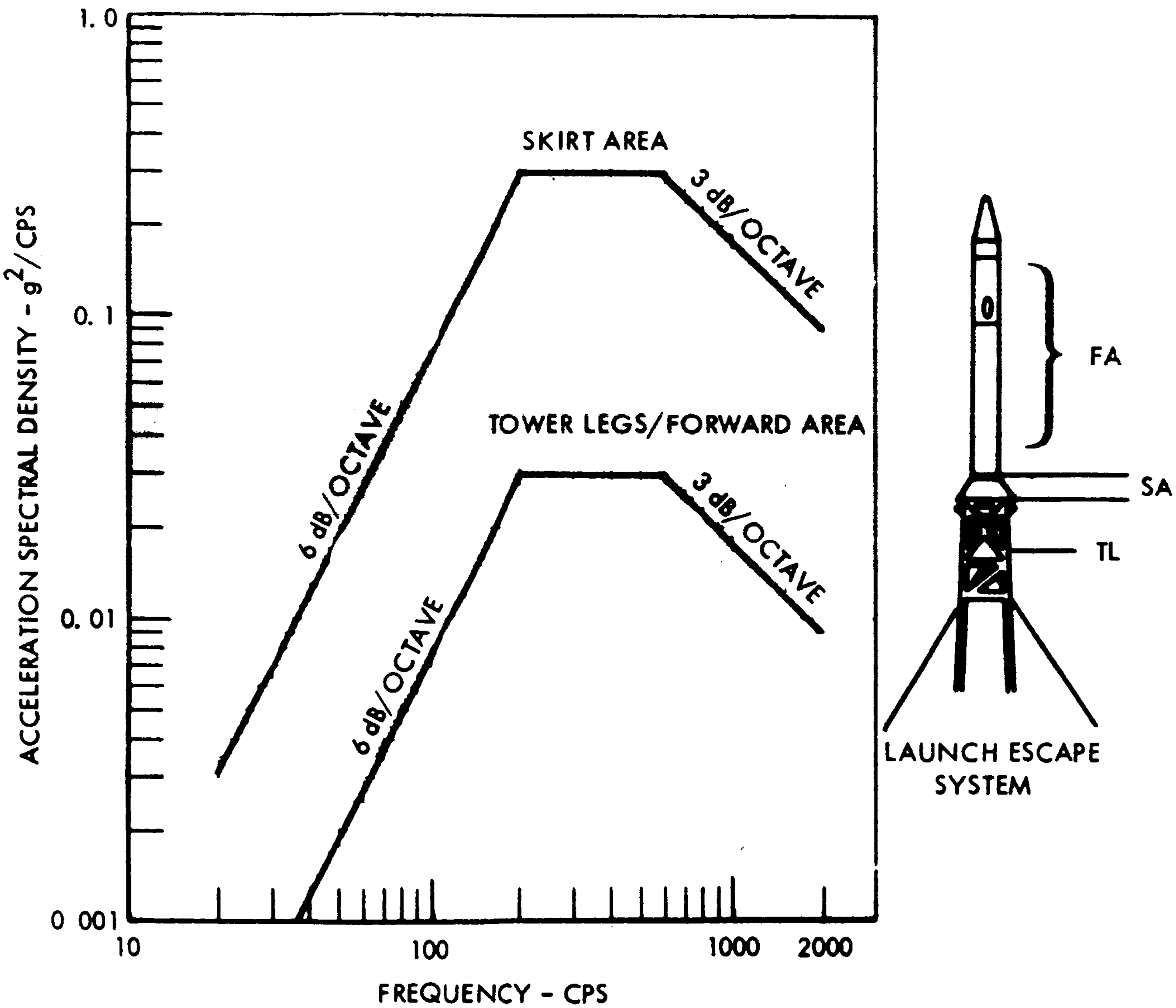


Figure 10. Vibration LES - Atmospheric Flight

[redacted]

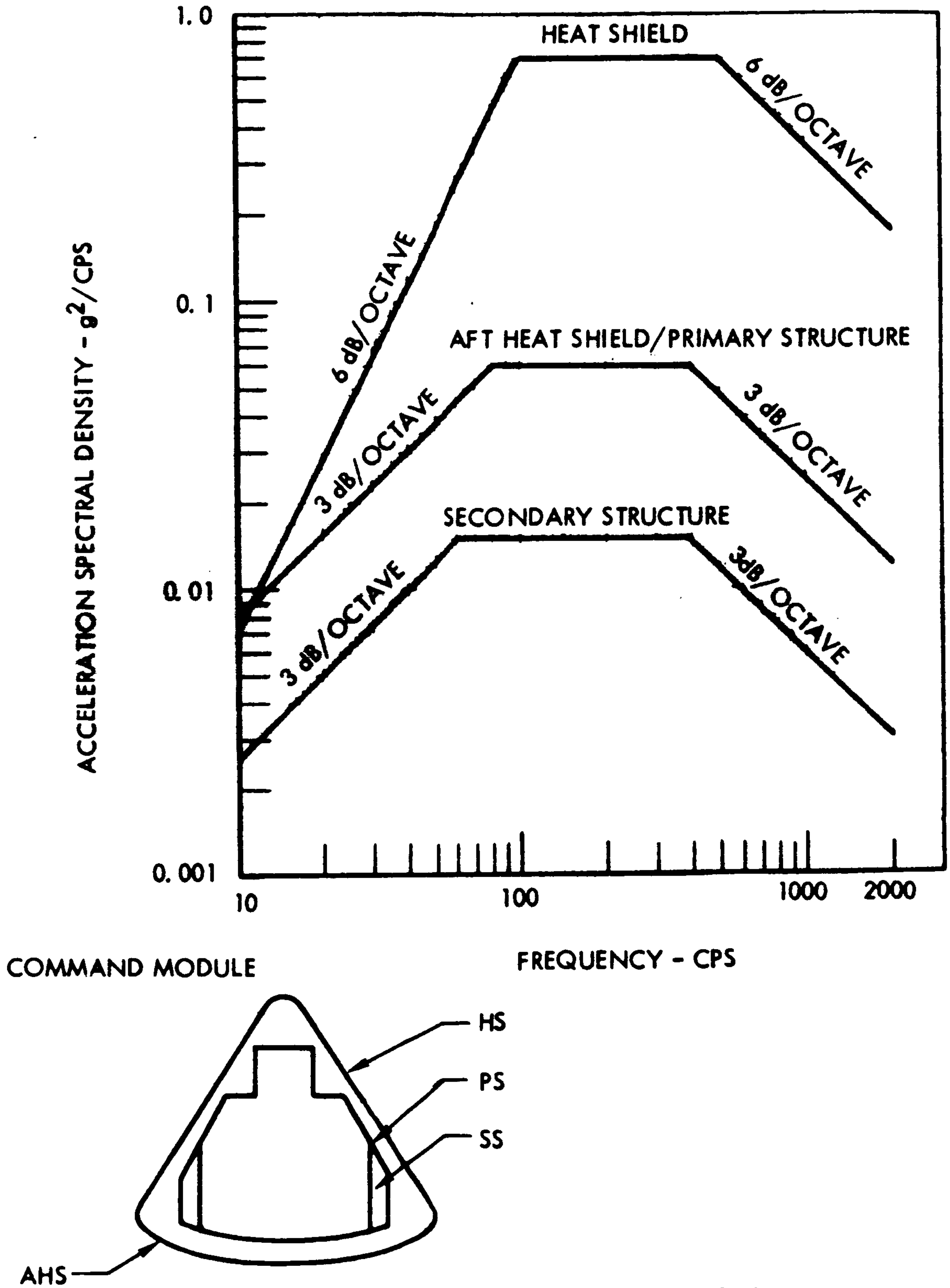


Figure 11. Vibration CM - Atmospheric Flight

[redacted]

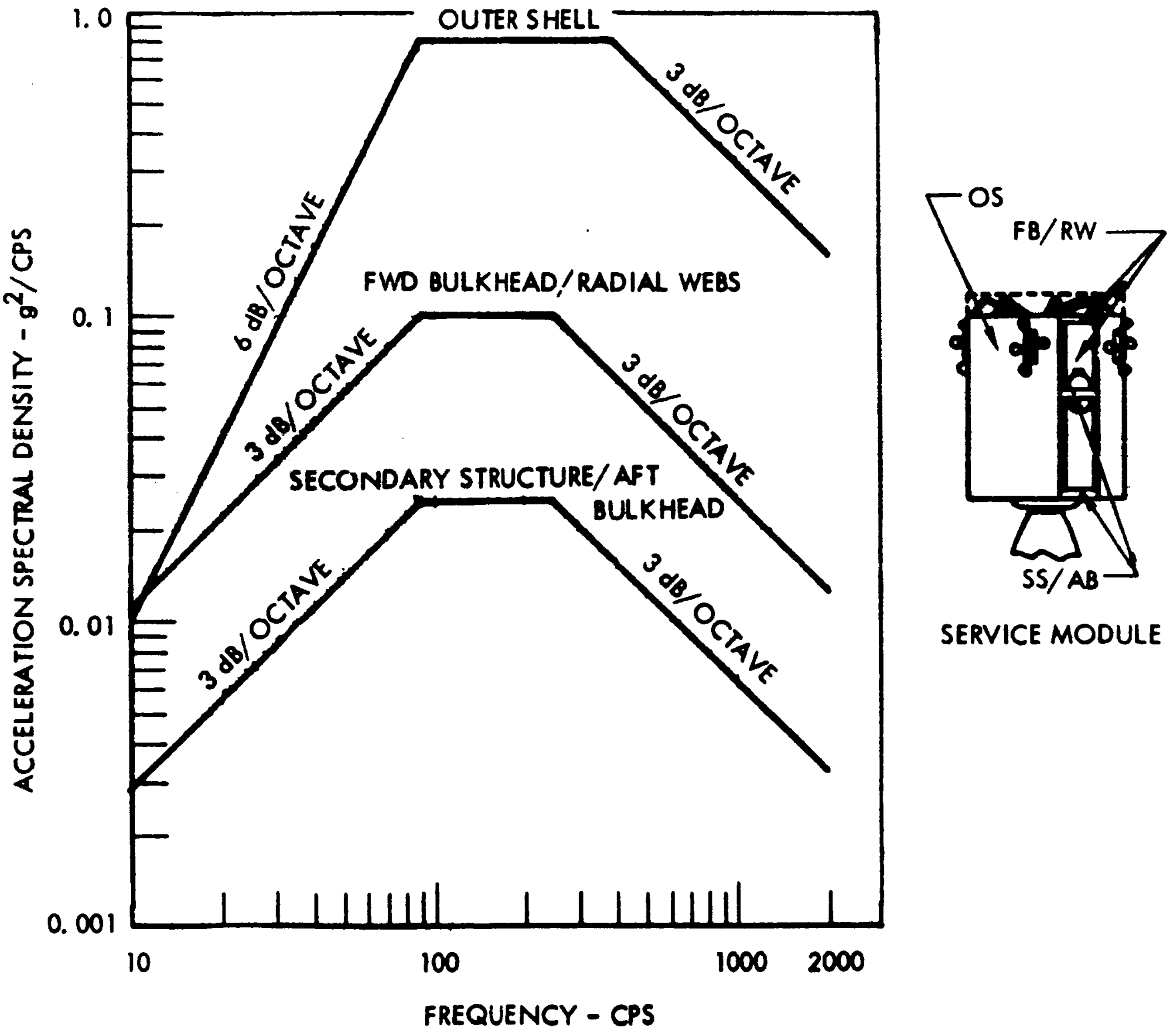


Figure 12 . **Vibration SM - Atmospheric Flight**

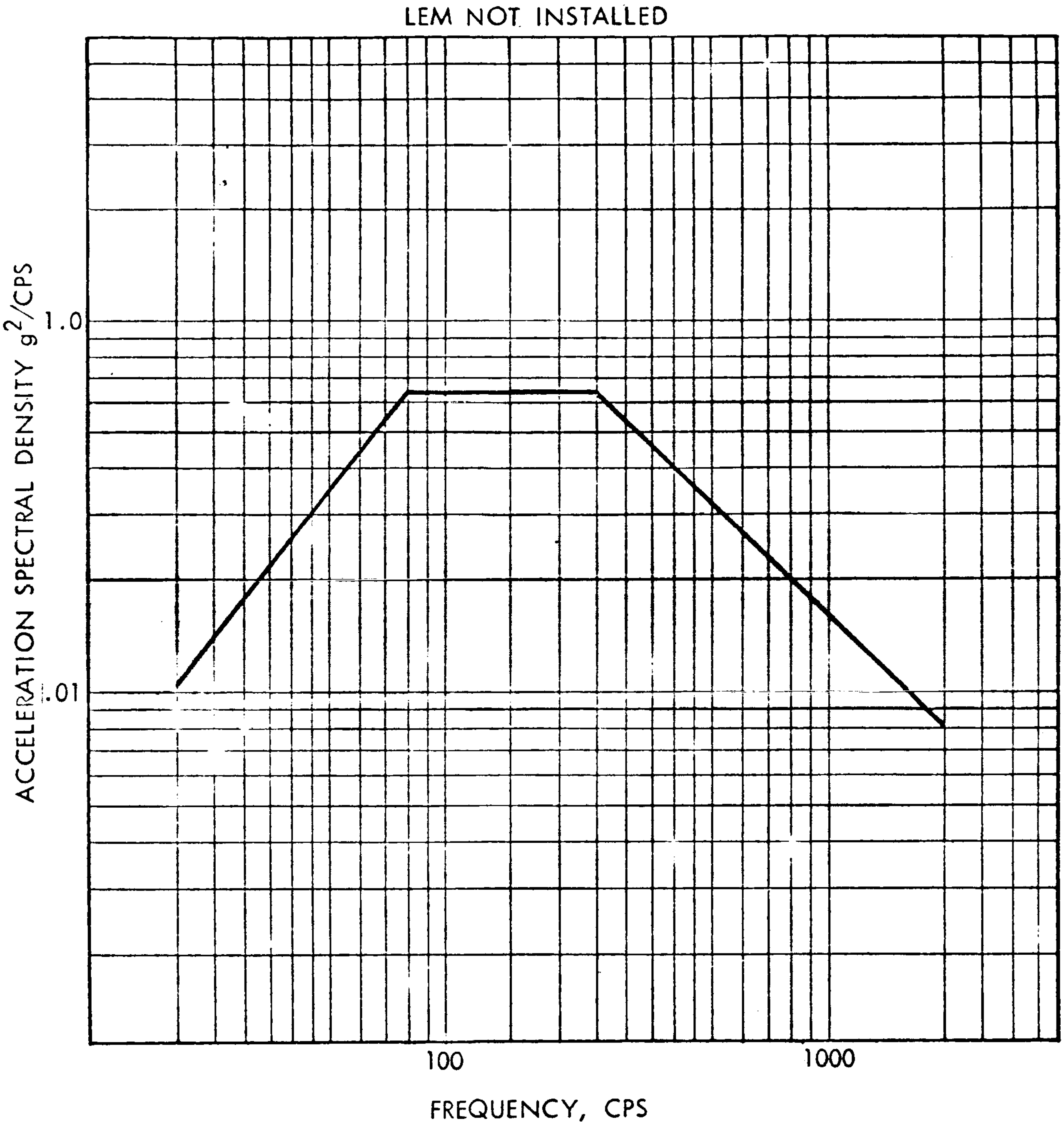


Figure 13 Vibration SLA Atmospheric Flight

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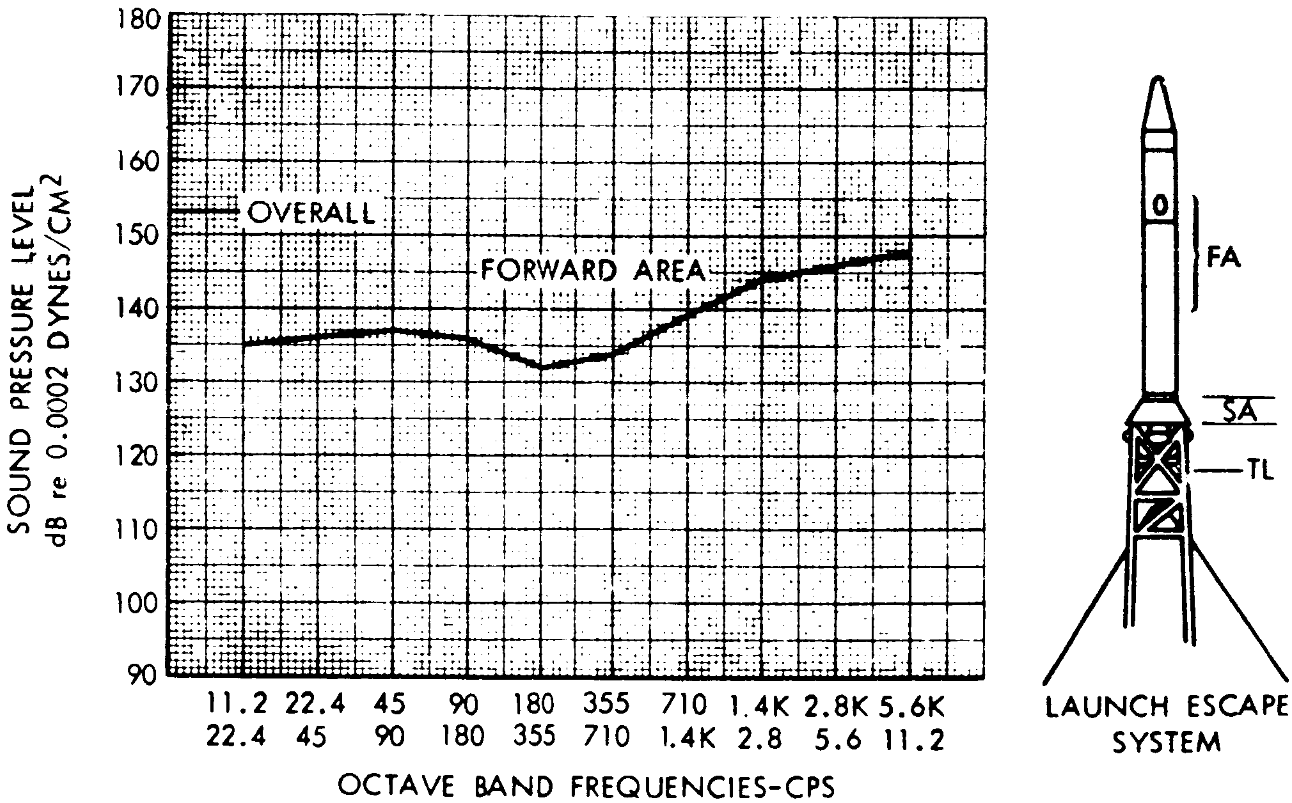


Figure 14 Acoustics LES - Atmospheric Flight - Forward Area

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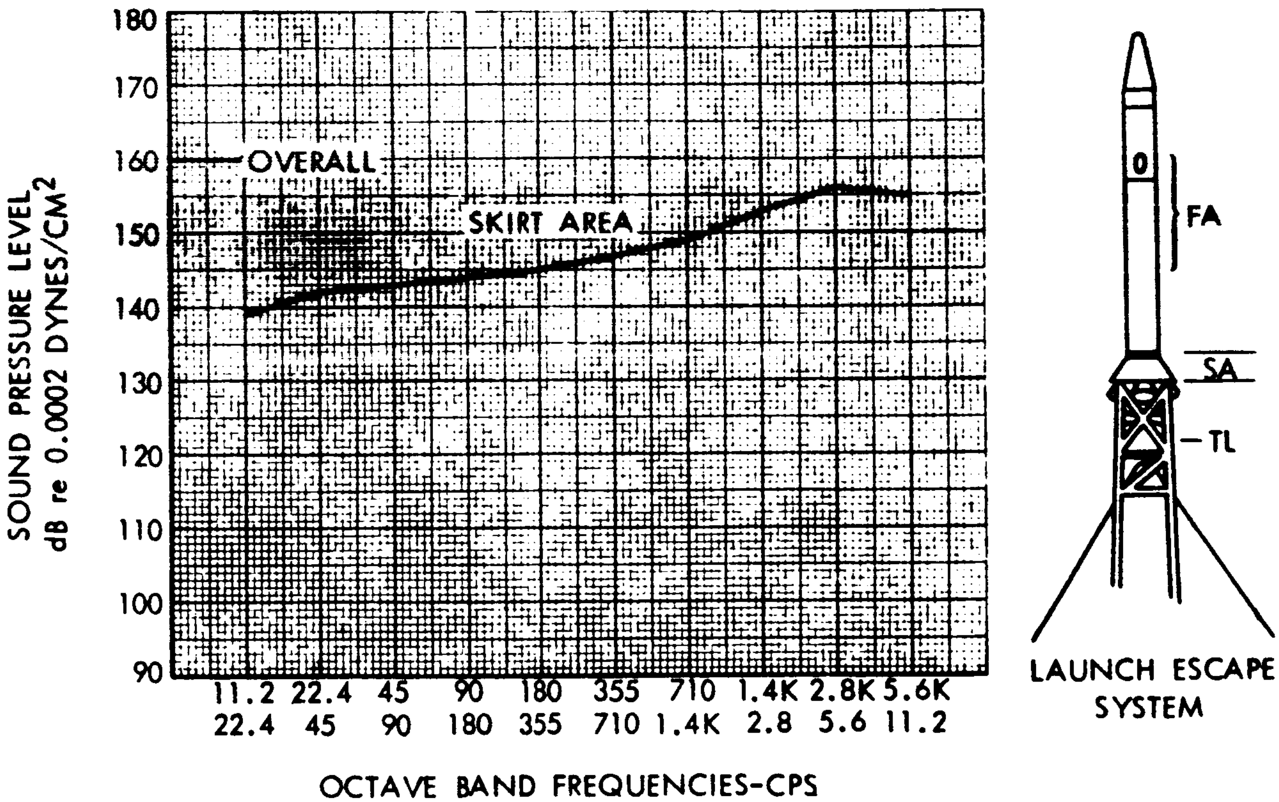


Figure 15 Acoustics LES - Atmospheric Flight - Skirt Area

[redacted]

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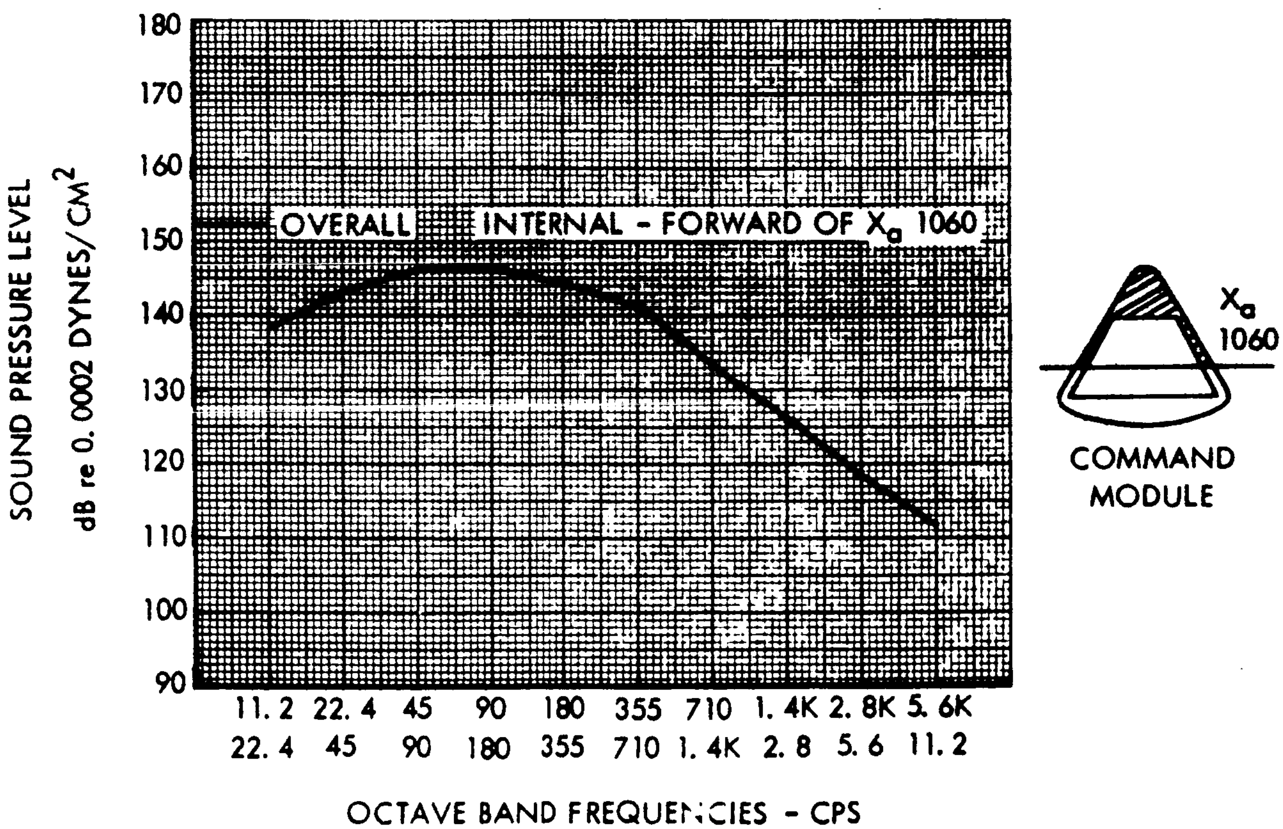


Figure 16. Acoustics CM - Atmospheric Flight - Internal - Forward Xa1060

[redacted]

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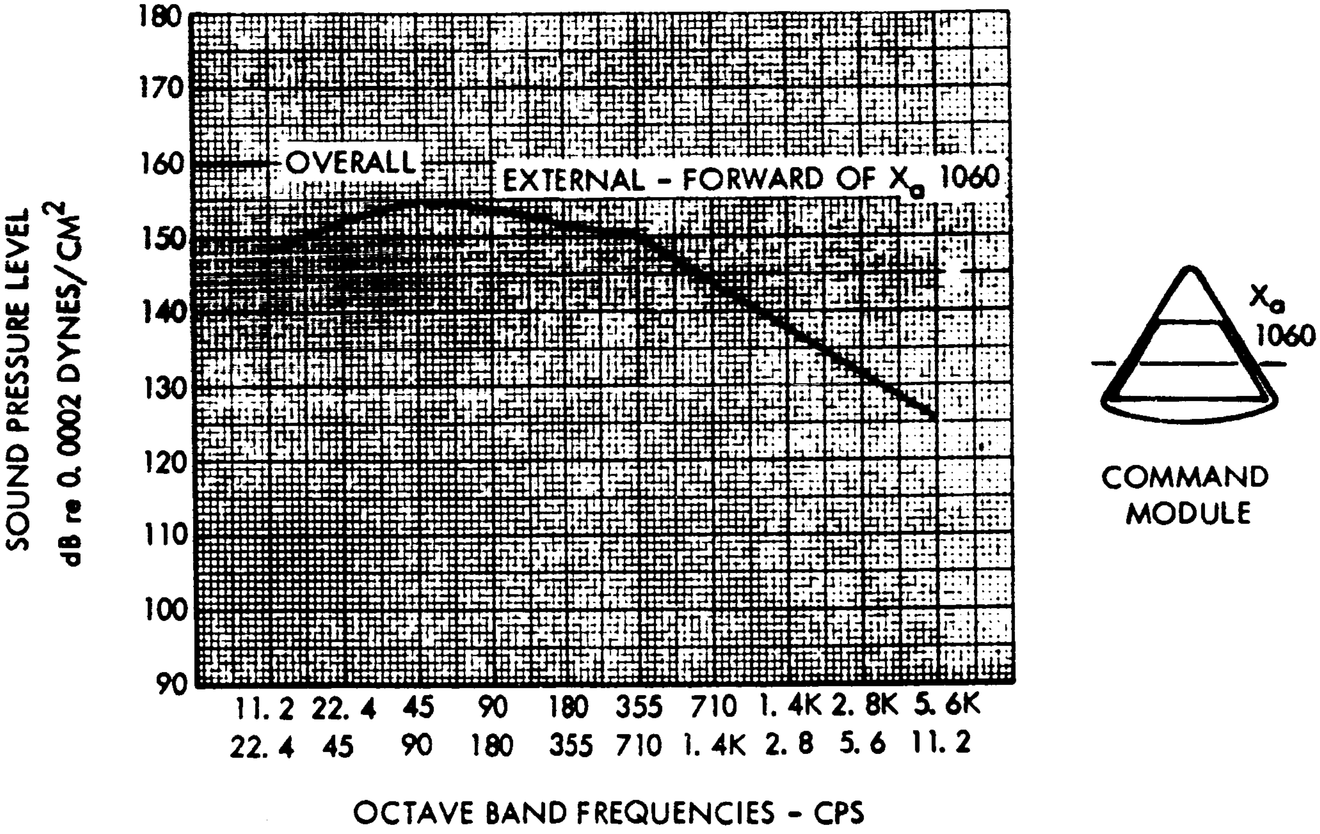


Figure 17 **Acoustics CM - Atmospheric Flight - External - Forward Xa1060**

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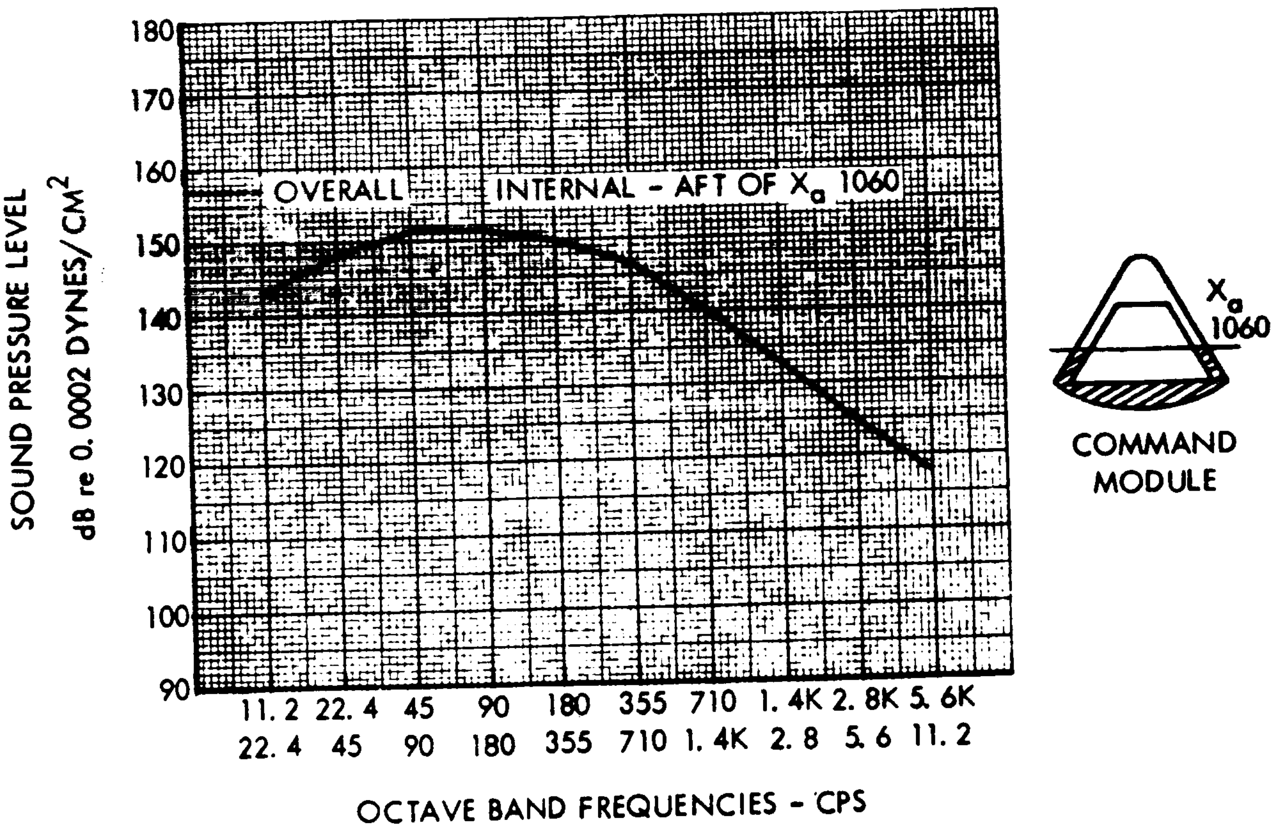


Figure 18. Acoustics CM - Atmospheric Flight - Internal - Aft Xa1060

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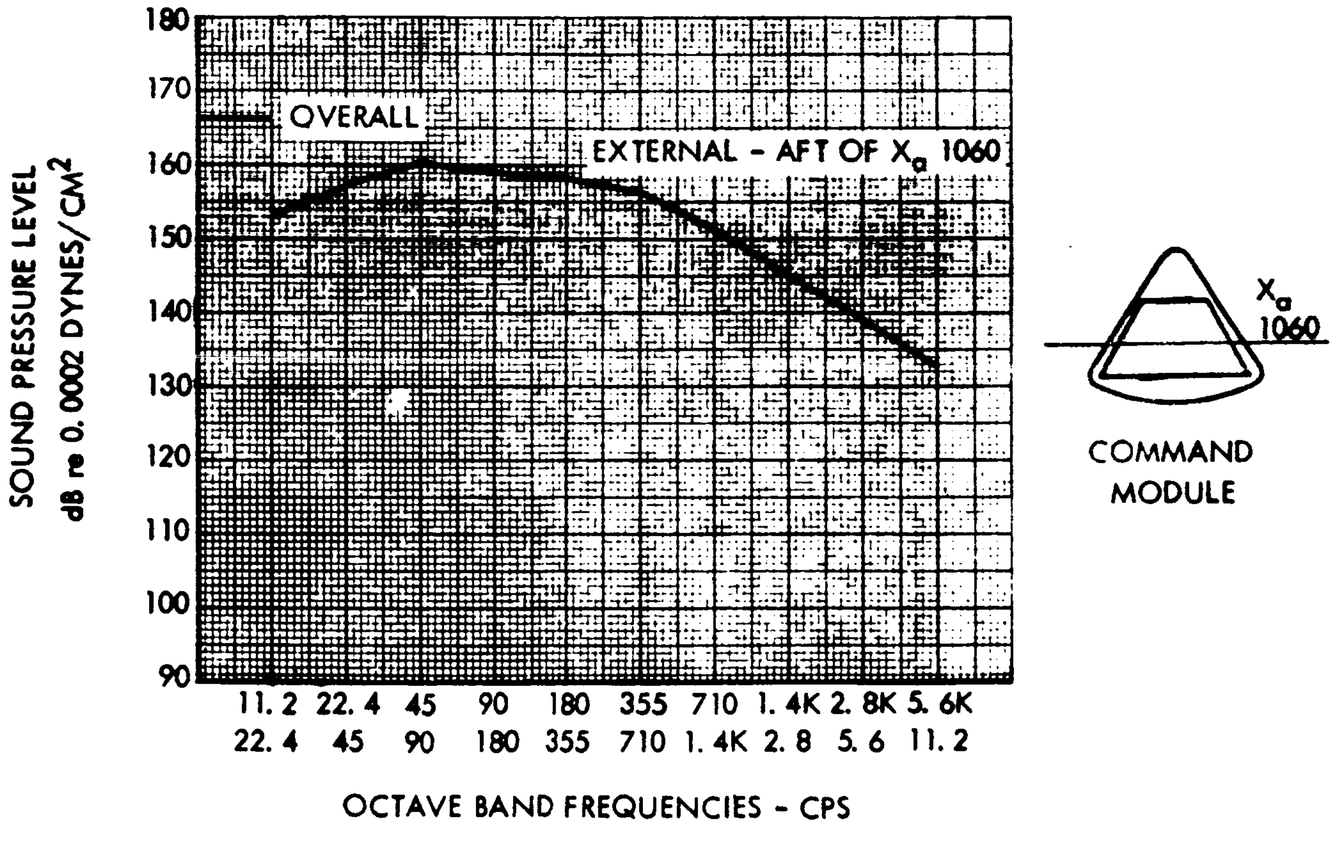


Figure 19 Acoustics CM - Atmospheric Flight - External - Aft Xa1060

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CON[redacted]

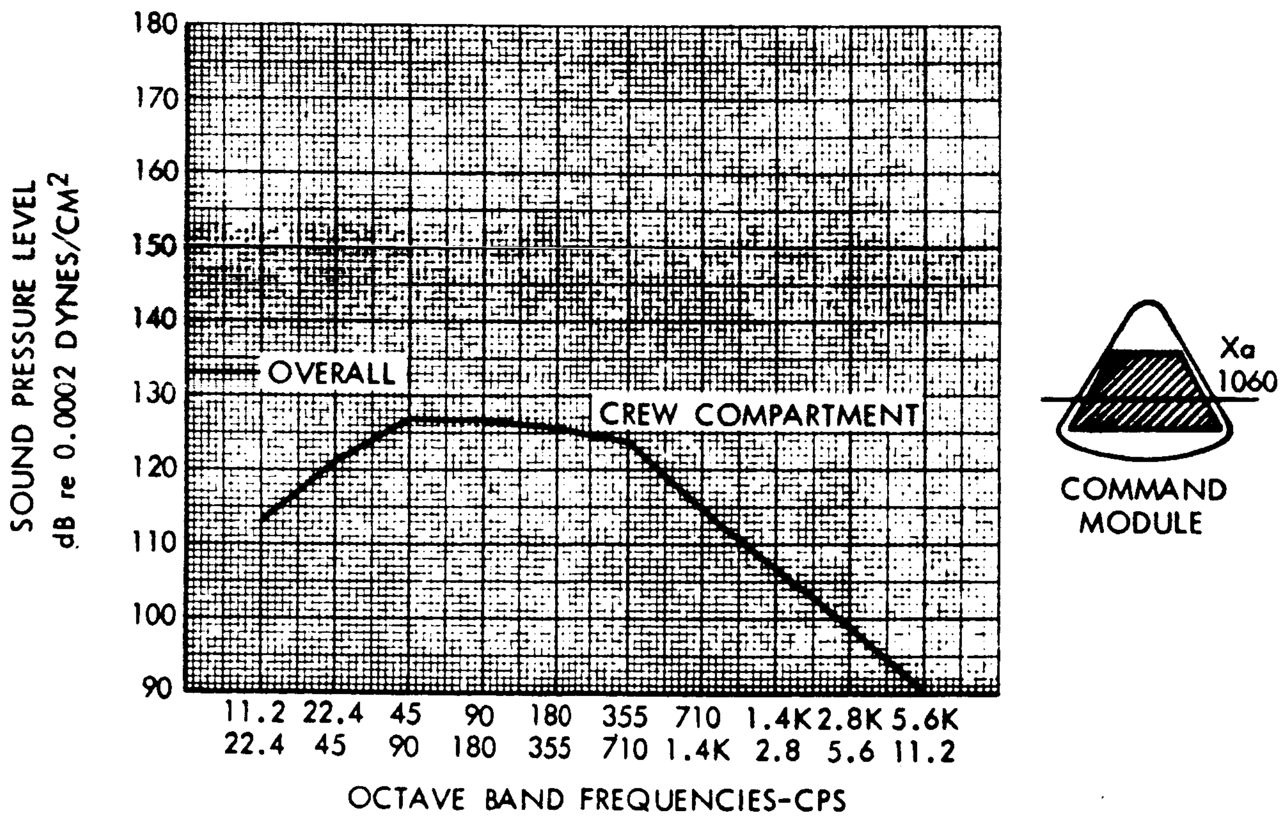


Figure 20. Acoustics CM - Atmospheric Flight - Crew Compartment

[redacted]

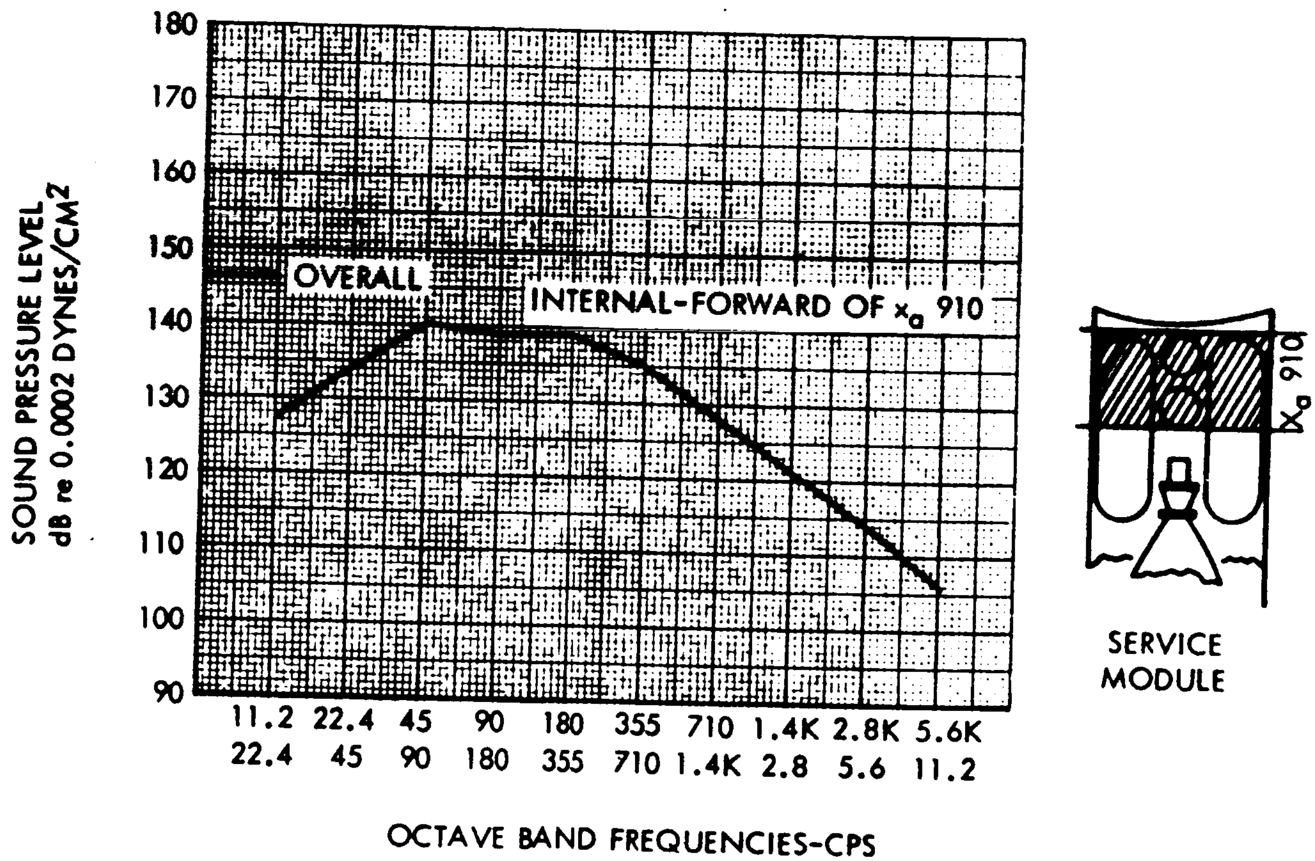


Figure 21 . Acoustics SM - Atmospheric Flight - Internal - Forward Xa910

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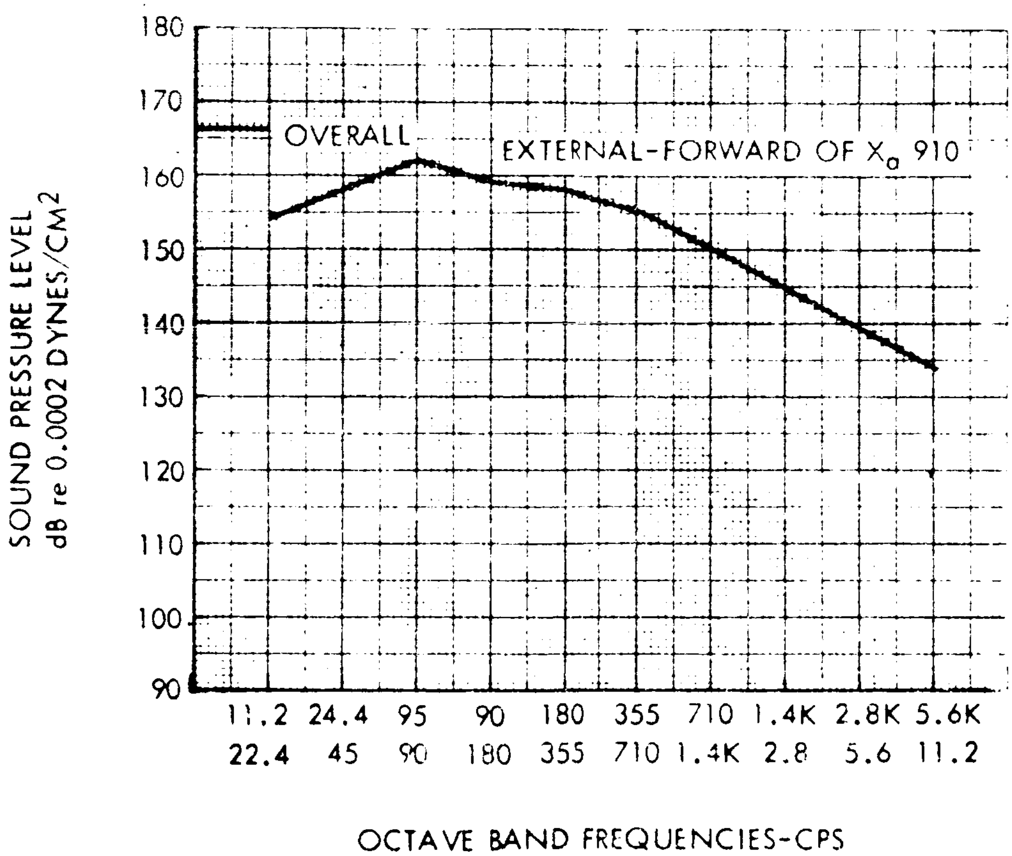


Figure 22. Acoustics SM - Atmospheric Flight - External - Forward Xa910

[redacted]

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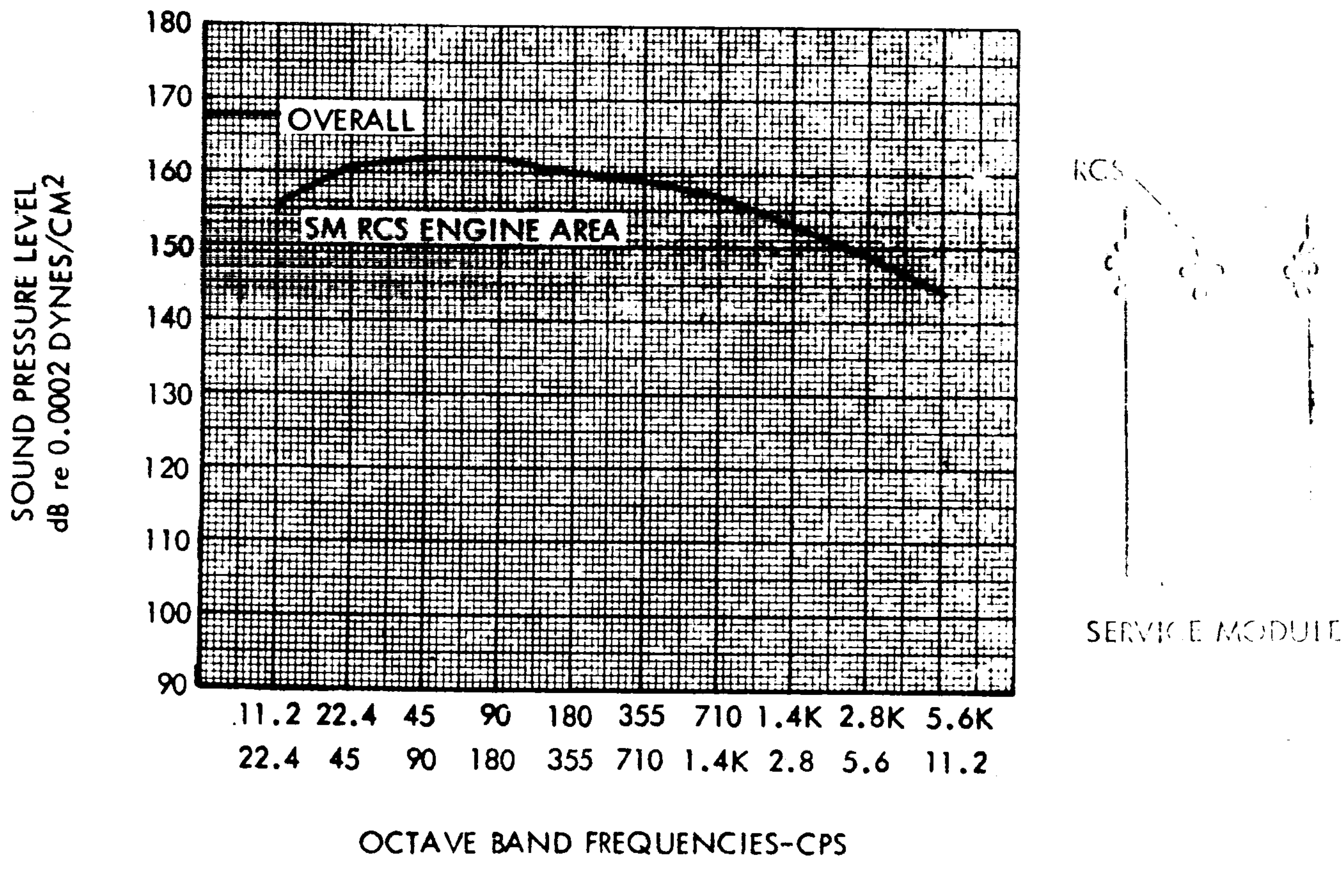


Figure 23 Acoustics SM/RCS Engine & Panel - Atmospheric Flight

[redacted]

CON[redacted]

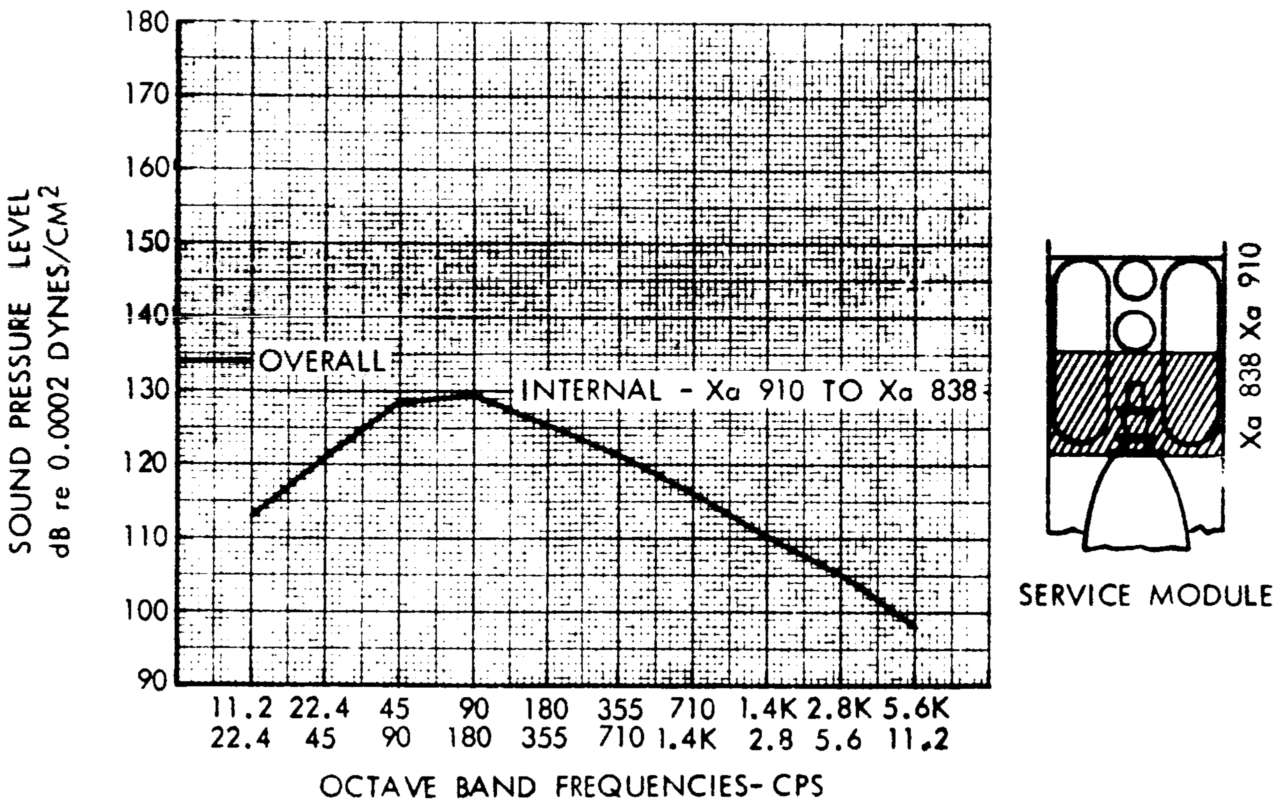


Figure 24 Acoustics SM - Atmospheric Flight - Internal XA910 to Xa838

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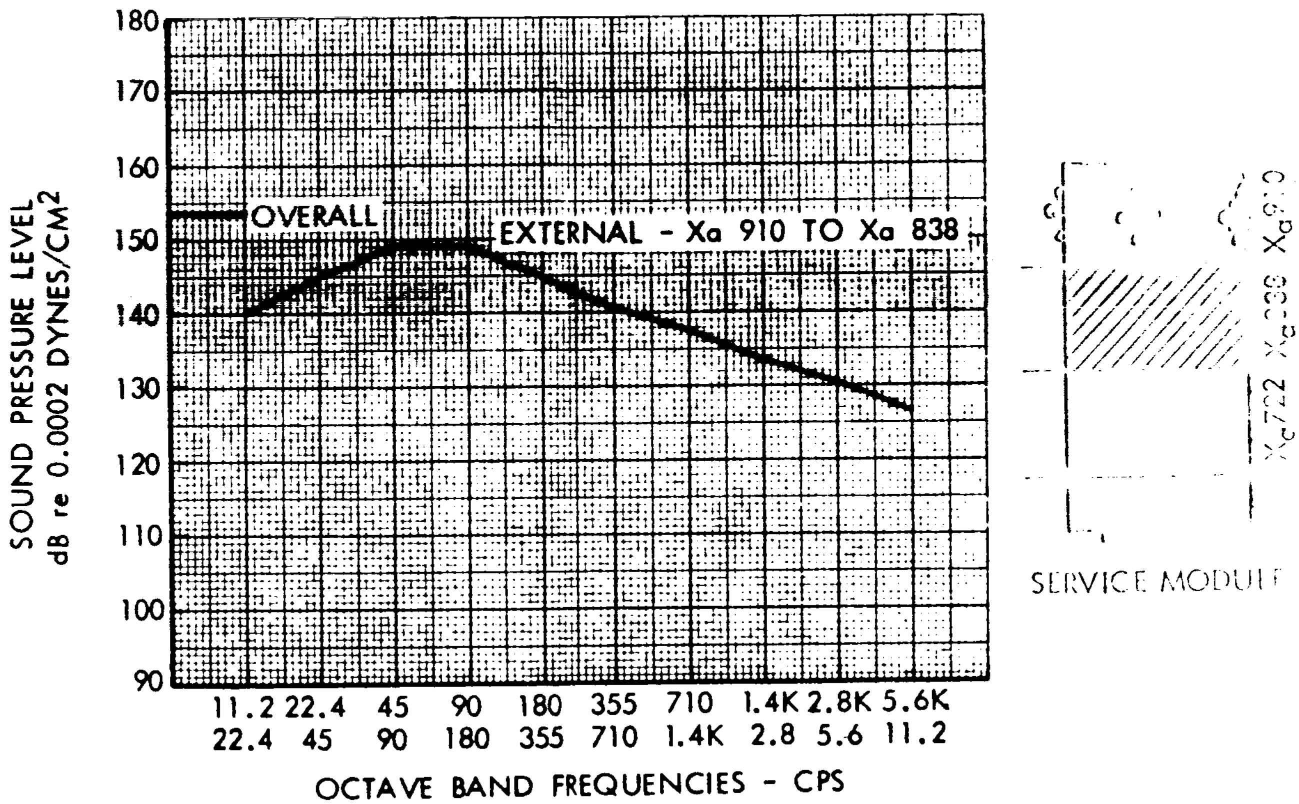


Figure 2: · Acoustics SM - Atmospheric Flight - External Xa910 to Xa838

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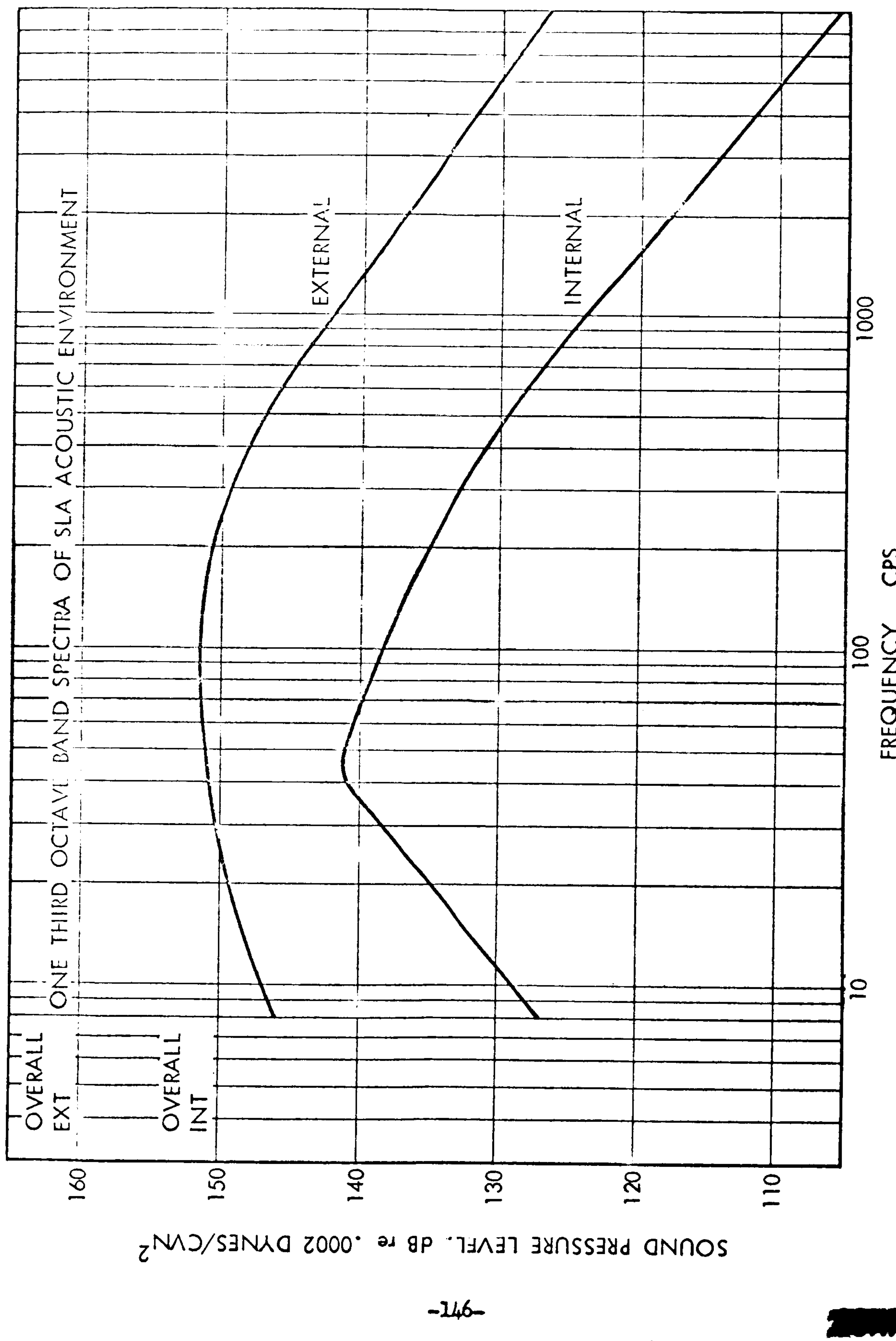


Figure 26 Acoustics-SLA-Atmospheric Flight



[redacted]

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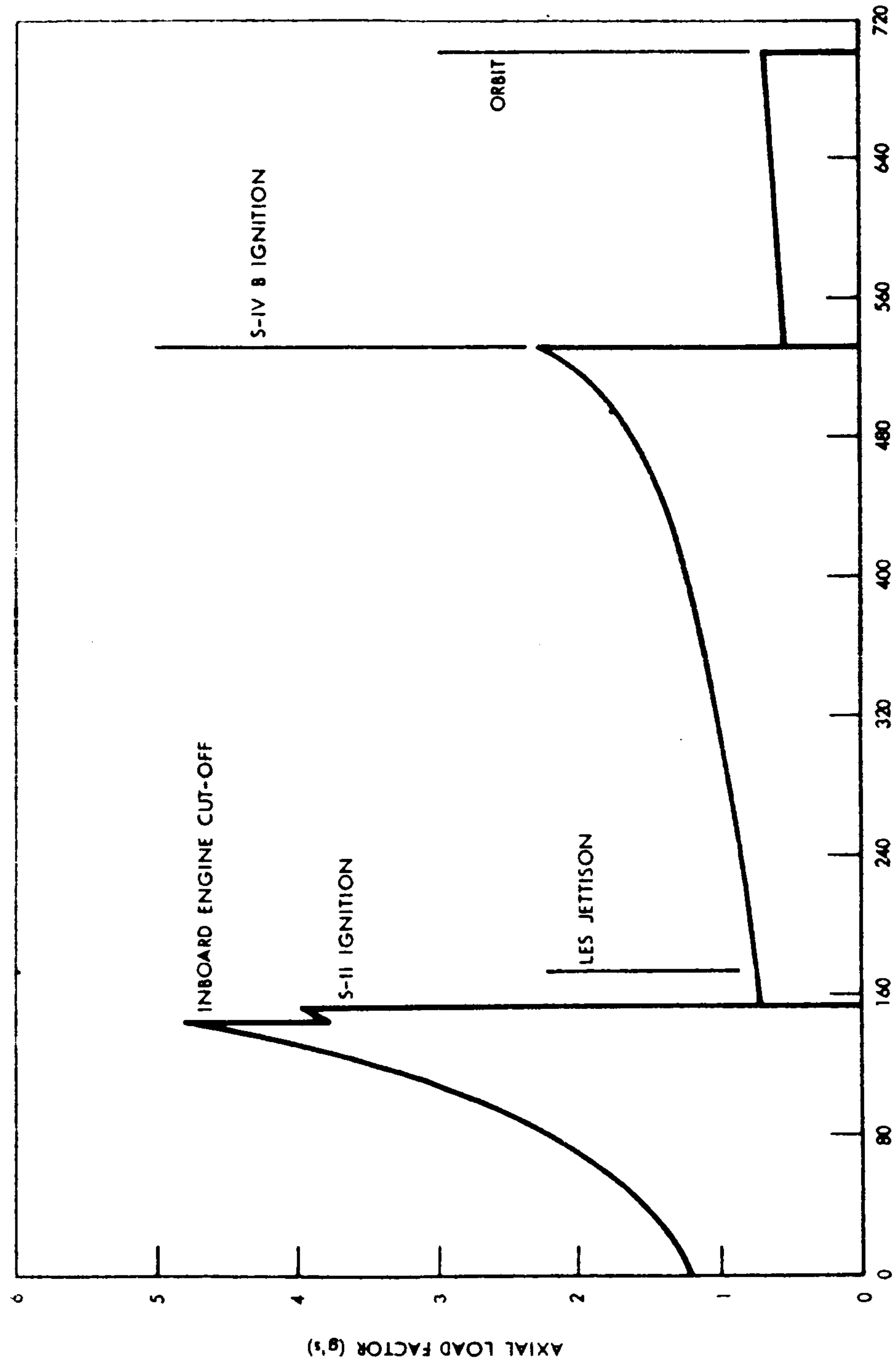


Figure 27 Axial Acceleration - Nominal Saturn V Boost (Preliminary)

CON[redacted]

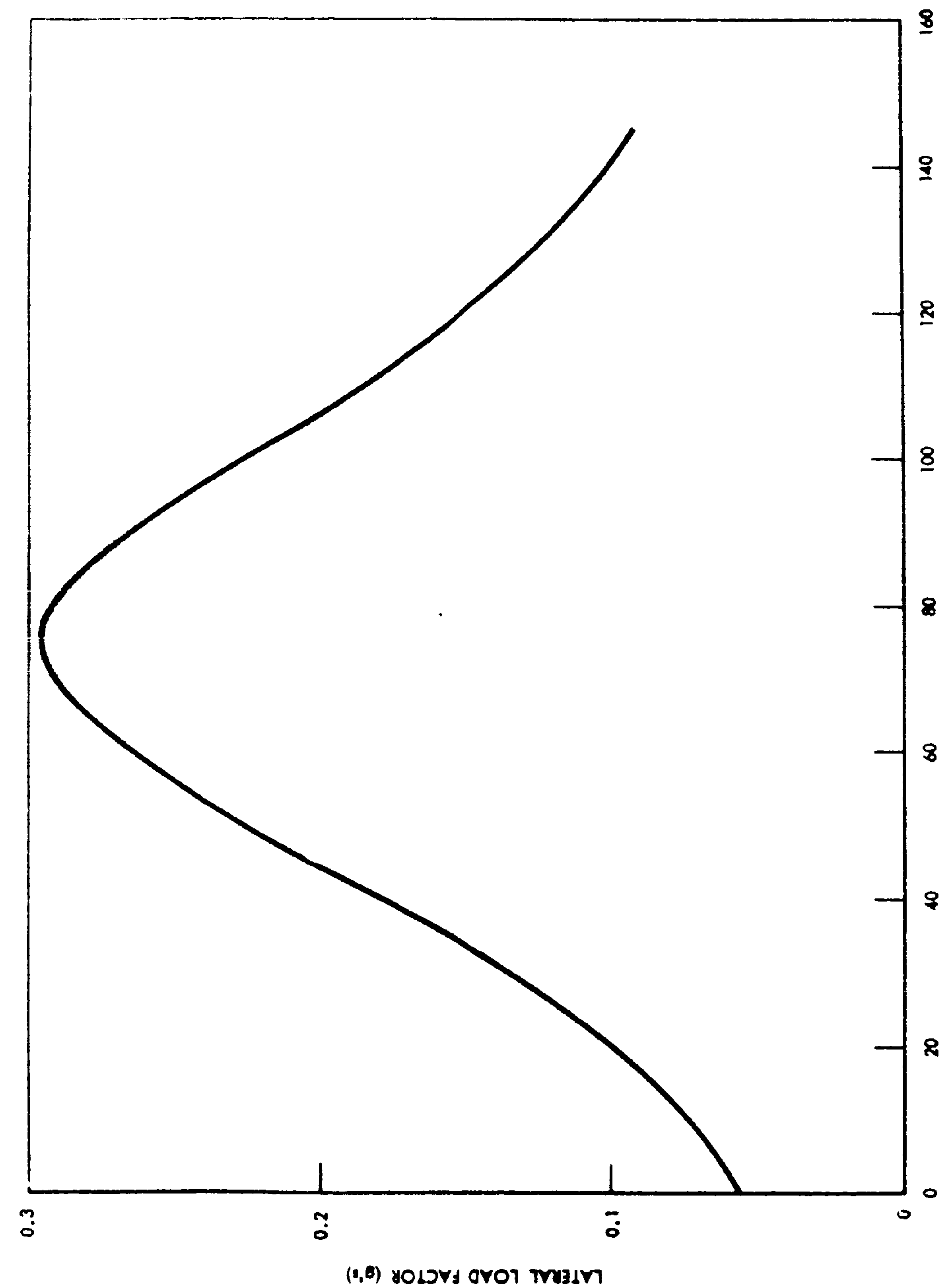


Figure 28 Lateral Acceleration During First Stage Boost (Preliminary)



[redacted]

CON[redacted]

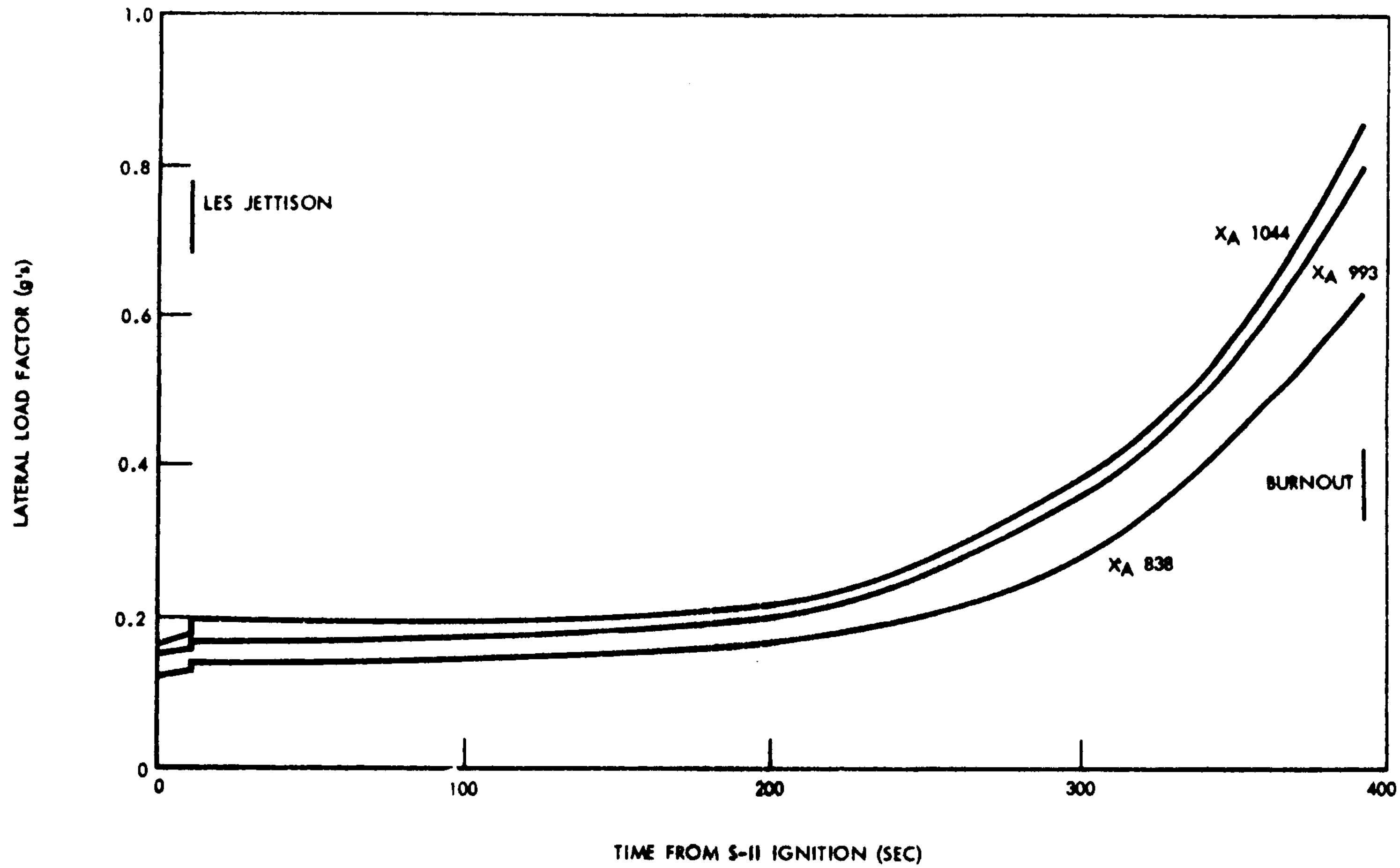


Figure 29 **Lateral Acceleration - S-II Flight at Maximum Gimbal Deflection** (Preliminary)



[redacted]

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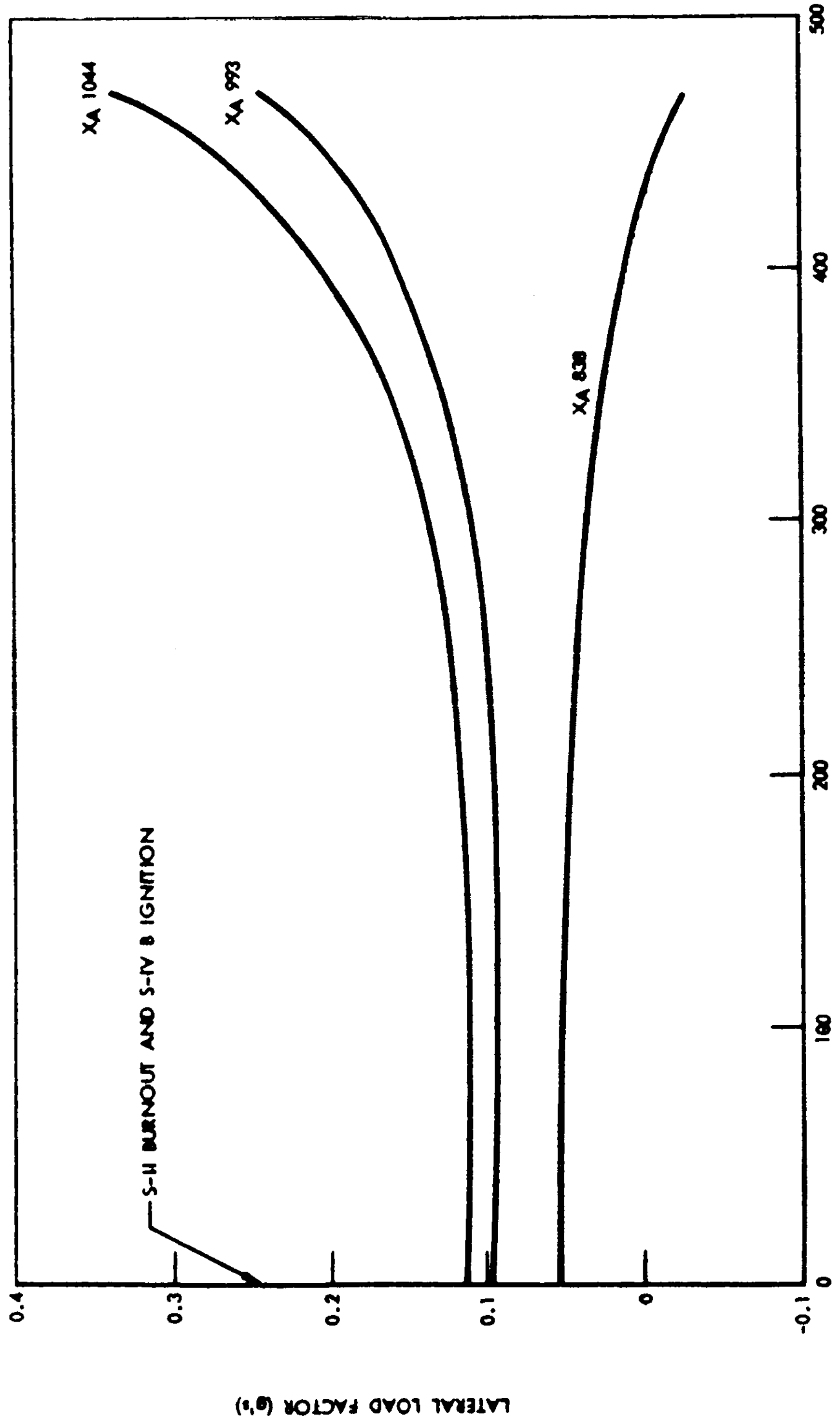


Figure 30 **Lateral Acceleration - S-IVB Flight at Maximum Gimbal Deflection** (Preliminary)

| Time (Sec.) | | Dyn. Press. (P.S.F.) | Mach No. | Weight x 10-3 (lbs.) | Thrust x 10-3 (lbs.) | Long. Acc. (g) | Alt ft. |
|---|---|---|---|---|---|---|---|
| .000 | | .0 | .000 | 6348 | 7610 | 1.200 | 0 |
| 8.000 | | 3.7 | .050 | 6117 | 7619 | 1.240 | 220 |
| 16.000 | | 17.9 | .112 | 5886 | 7650 | 1.297 | 945 |
| 24.000 | | 47.7 | .186 | 5655 | 7698 | 1.353 | 2880 |
| 32.000 | | 98.0 | .277 | 5424 | 7772 | 1.422 | 4350 |
| 40.000 | | 173.0 | .388 | 5193 | 7870 | 1.505 | 7300 |
| 48.000 | | 273.5 | .525 | 4962 | 7889 | 1.595 | 11250 |
| 56.000 | | 394.5 | .695 | 4731 | 8122 | 1.690 | 16380 |
| 64.000 | | 522.5 | .908 | 4500 | 8261 | 1.770 | 22750 |
| 72.000 | | 634.0 | 1.174 | 4269 | 8394 | 1.865 | 30400 |
| 80.000 | | 703.0 | 1.516 | 4039 | 8512 | 2.015 | 39600 |
| 83.000 | Max q | 709.0 | 1.664 | 3952 | 8550 | 2.075 | 43100 |
| 88.000 | | 683.0 | 1.927 | 3808 | 8605 | 2.180 | 49700 |
| 96.000 | | 554.0 | 2.335 | 3577 | 8668 | 2.360 | 61300 |
| 104.000 | | 408.0 | 2.756 | 3346 | 8704 | 2.560 | 74500 |
| 112.000 | | 287.0 | 3.241 | 3115 | 8725 | 2.770 | 88900 |
| 120.000 | | 190.0 | 3.772 | 2884 | 8735 | 3.010 | 104600 |
| 128.000 | | 118.5 | 4.292 | 2653 | 8740 | 3.280 | 122200 |
| 136.000 | | 72.4 | 4.852 | 2422 | 8743 | 3.600 | 140700 |
| 144.000 | | 44.5 | 5.515 | 2191 | 8744 | 3.990 | 161000 |
| 152.000 | | 28.0 | 6.513 | 1960 | 8745 | 4.460 | 183000 |
| 154.567 | | 24.1 | 6.956 | 1886 | 8745 | 4.630 | 190000 |
| 154.567 | 1 ENG CO | 24.1 | 6.956 | 1886 | 6996 | 3.710 | 190000 |
| 158.567 | Cutoff | 18.0 | 7.602 | 1793 | 6996 | 3.900 | 202000 |
| 158.567 | Separation | 18.0 | 7.602 | 1412 | 0 | -.004 | 202000 |
| 160.000 | | 15.3 | 7.695 | 1412 | 0 | -.003 | 206000 |
| 162.367 | 2nd stage ignition | 11.7 | 7.798 | 1412 | 0 | -.002 | 213000 |
| 426.924 | 1st stage impact | 85700.0 | 7.593 | 382 | 0 | .000 | 0 |

Figure 31 - Saturn V Three-Stage to Orbit Boost Trajectory - 100 NM Orbit

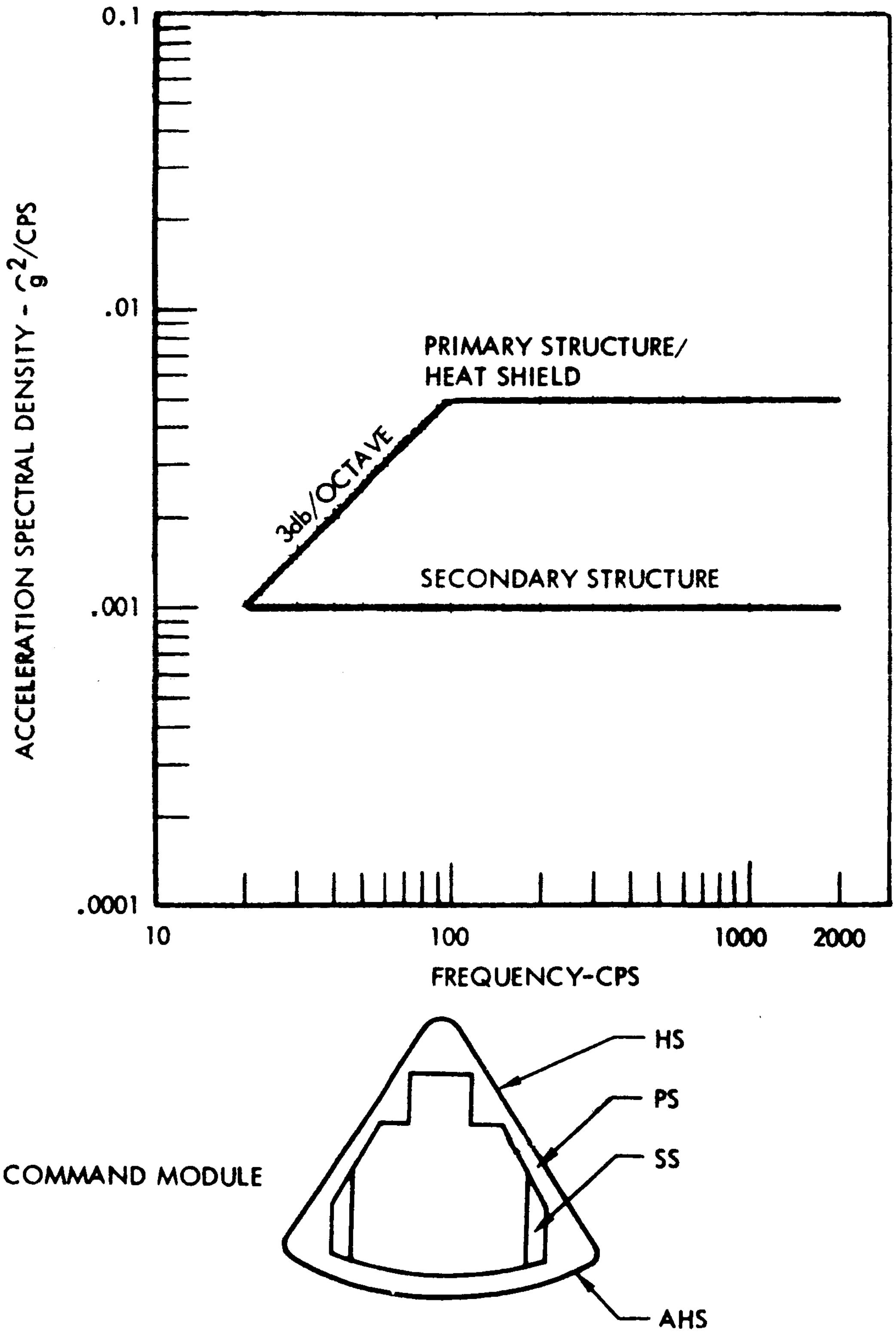


Figure 32 Vibration CM - Space Flight

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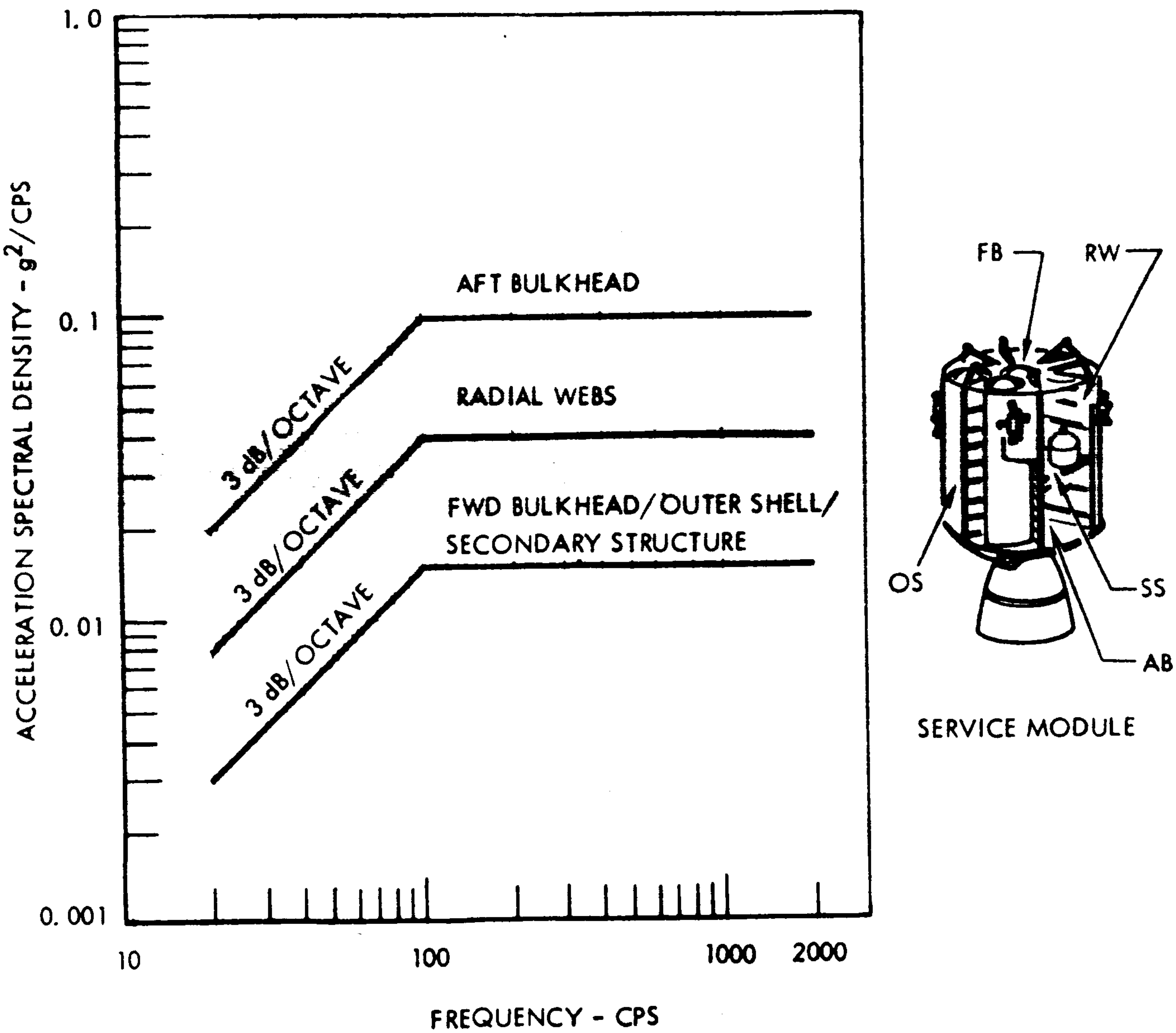


Figure 33 Vibration SM - Space Flight



[redacted]

[redacted]

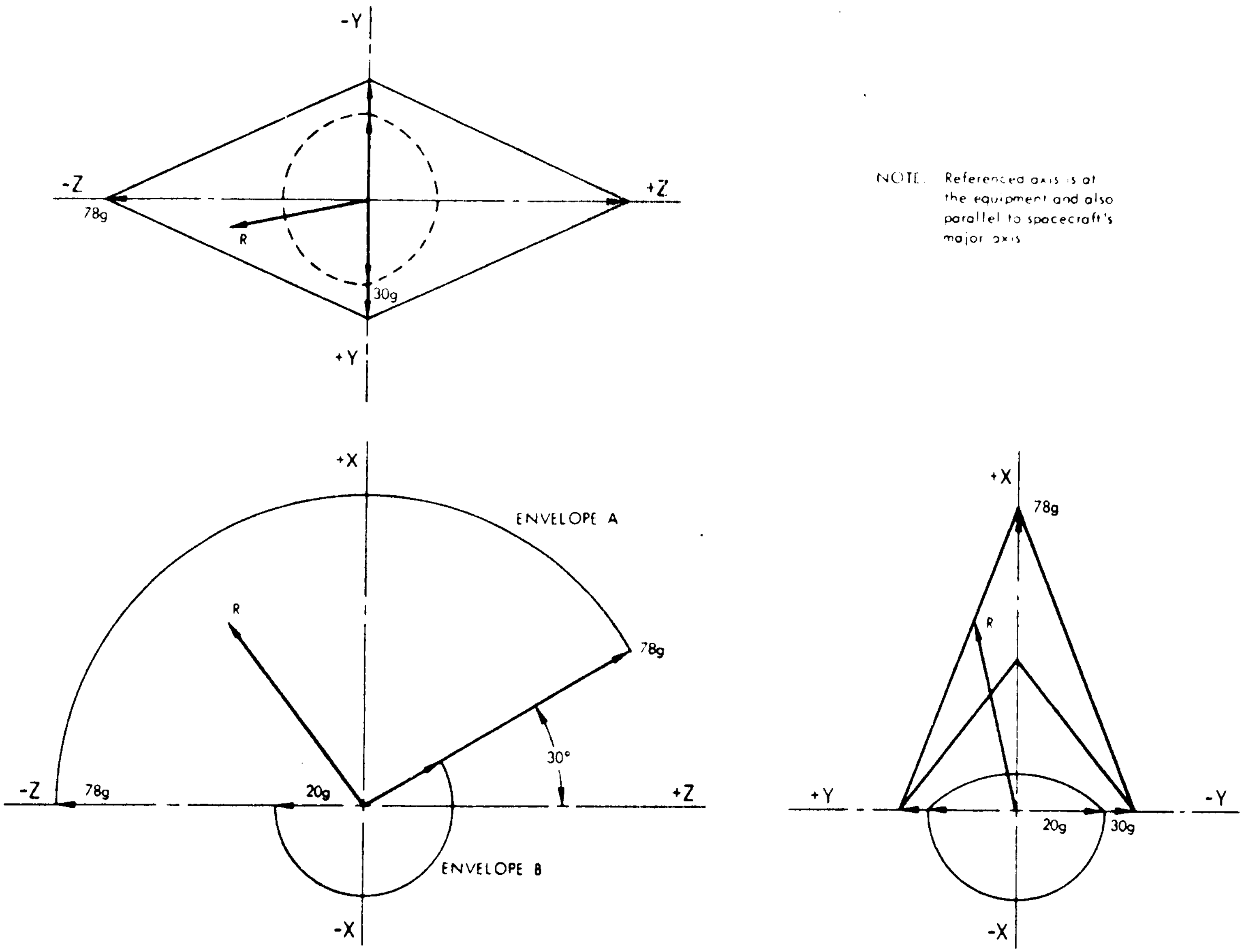


Figure 34 Internal Equipment Ultimate Design Accelerations Diagram I



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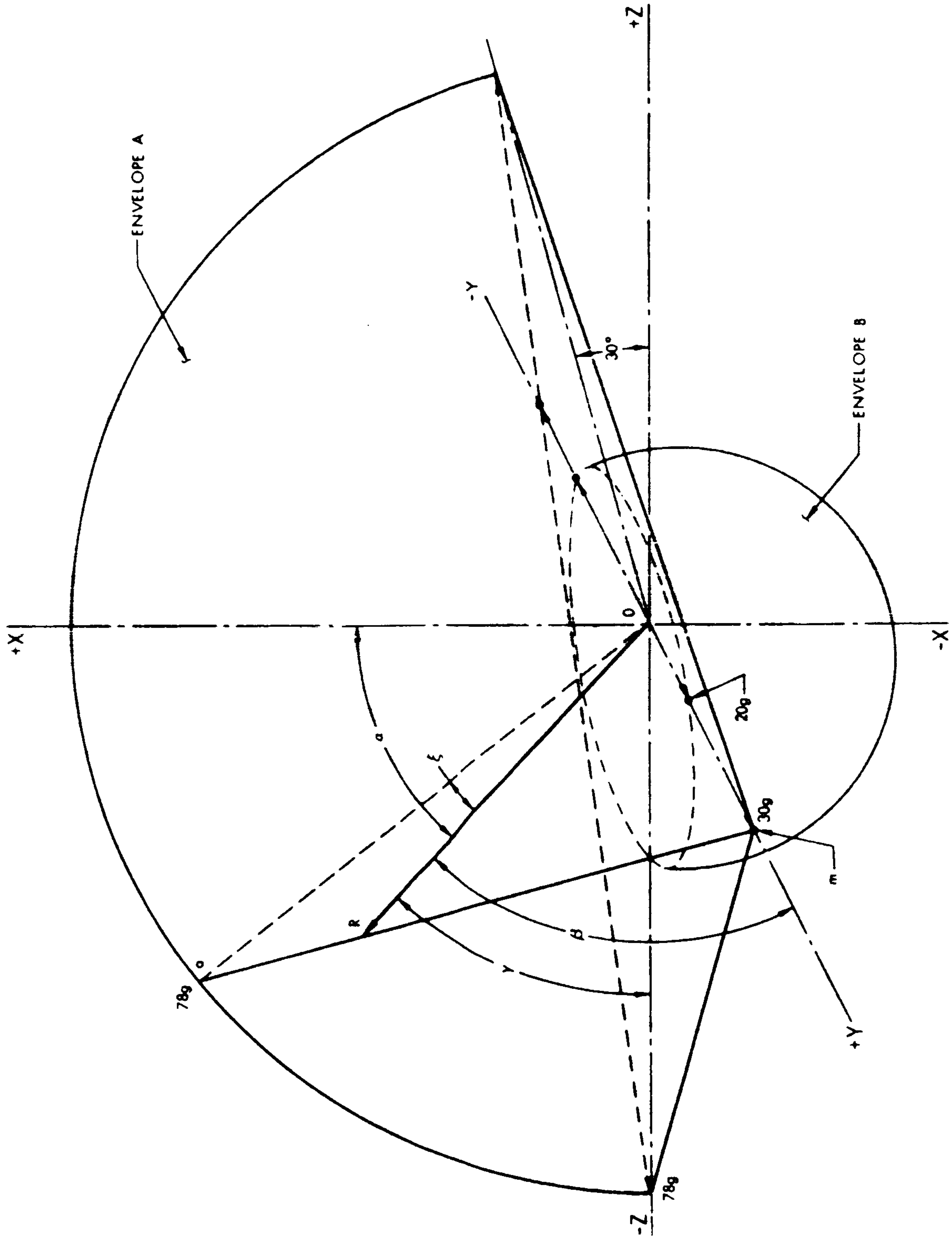


Figure 35 Internal Equipment Ultimate Design Accelerations Diagram II



[redacted]

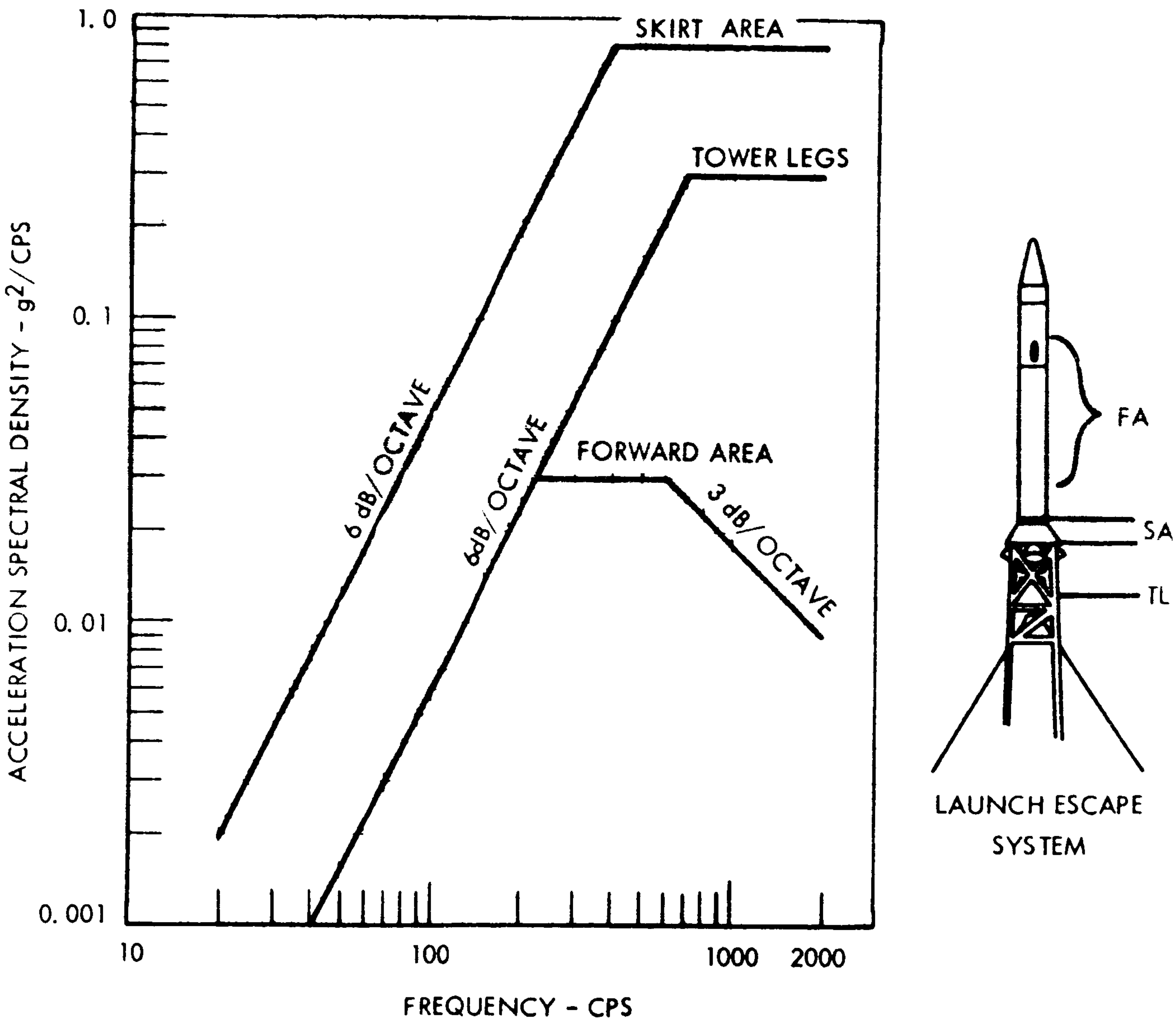


Figure 36 Vibration LES - High "Q" Abort

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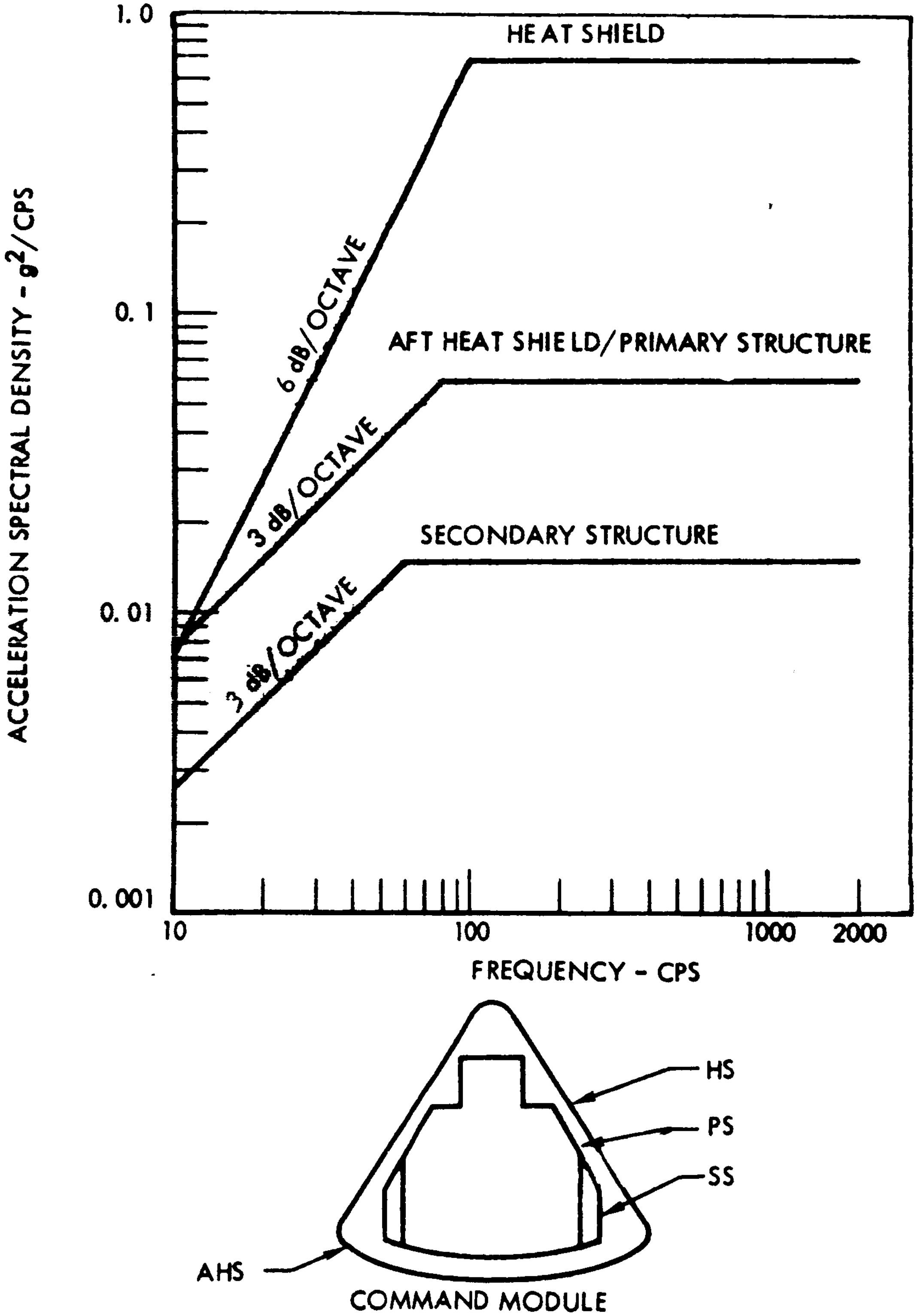


Figure 37 **Vibration CM - High "Q" Abort**

CON[redacted]

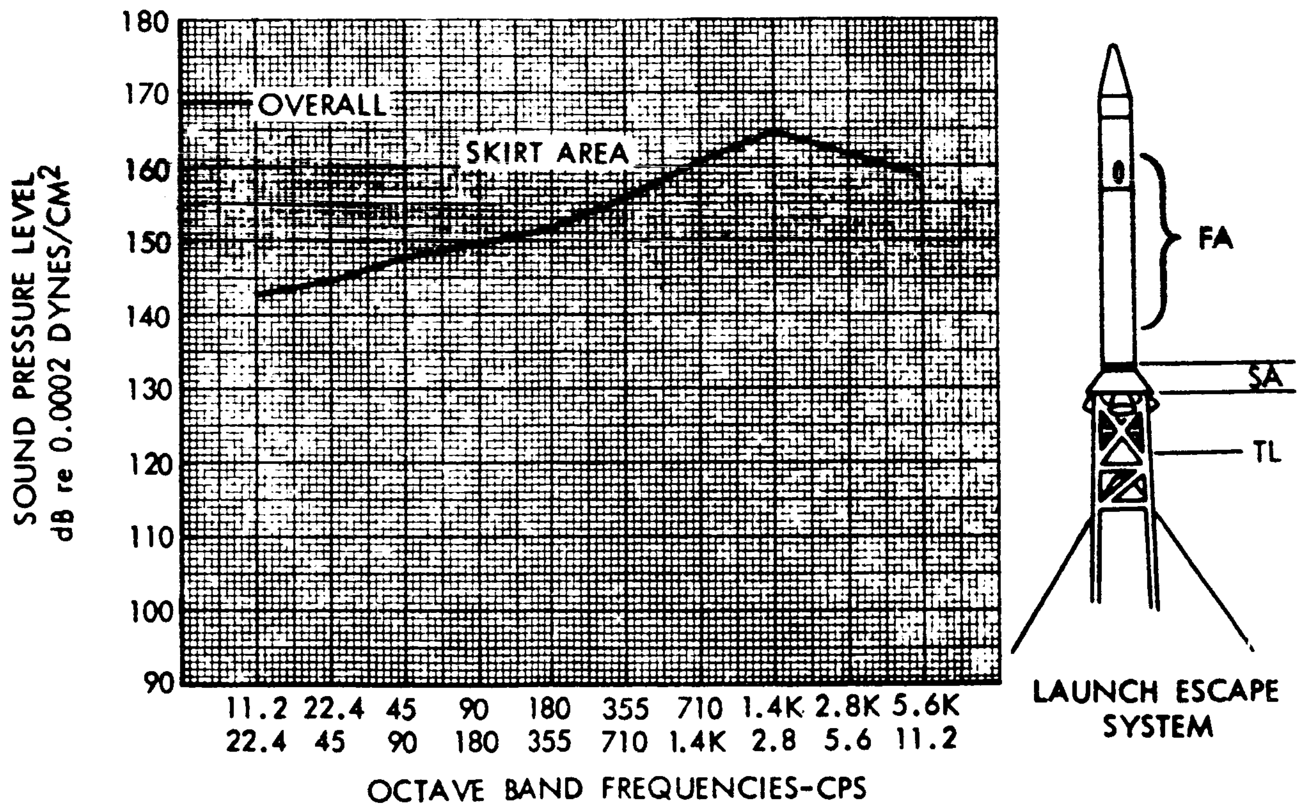


Figure 38 Acoustics LES - High "Q" Abort

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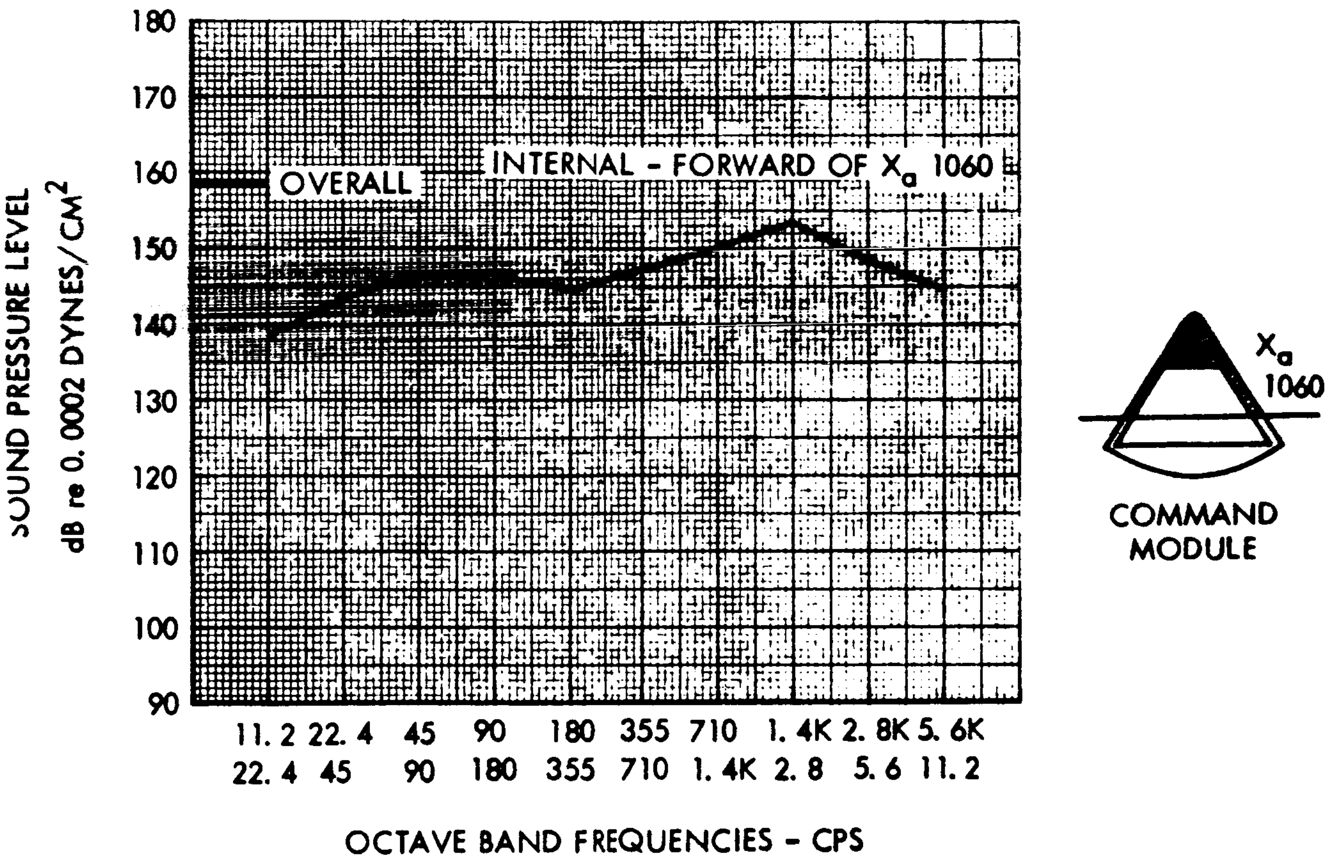


Figure 39. Acoustics CM - High "Q" Abort - Internal - Forward Xa1060



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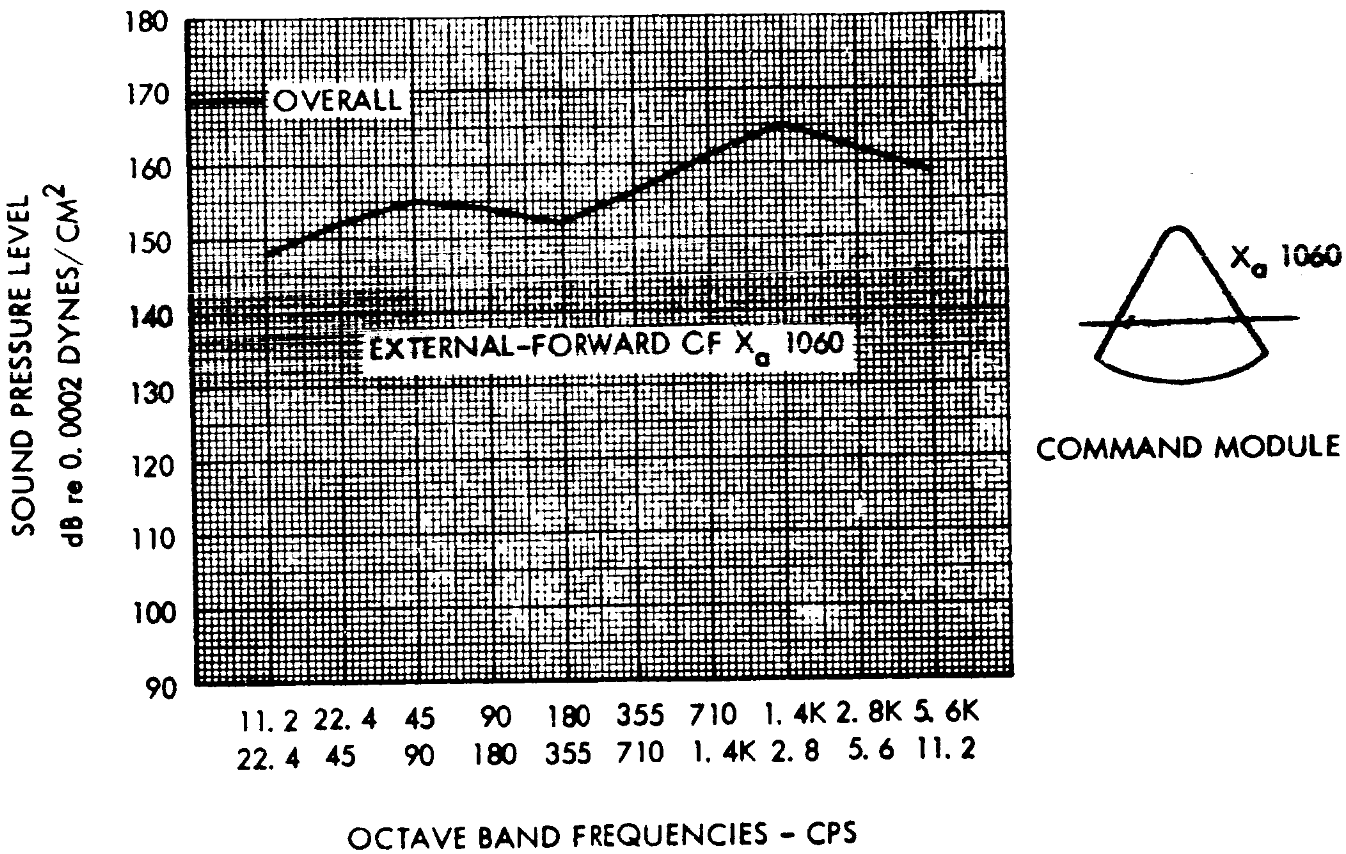


Figure 40 Acoustics CM - High "Q" Abort - External - Forward of X1060

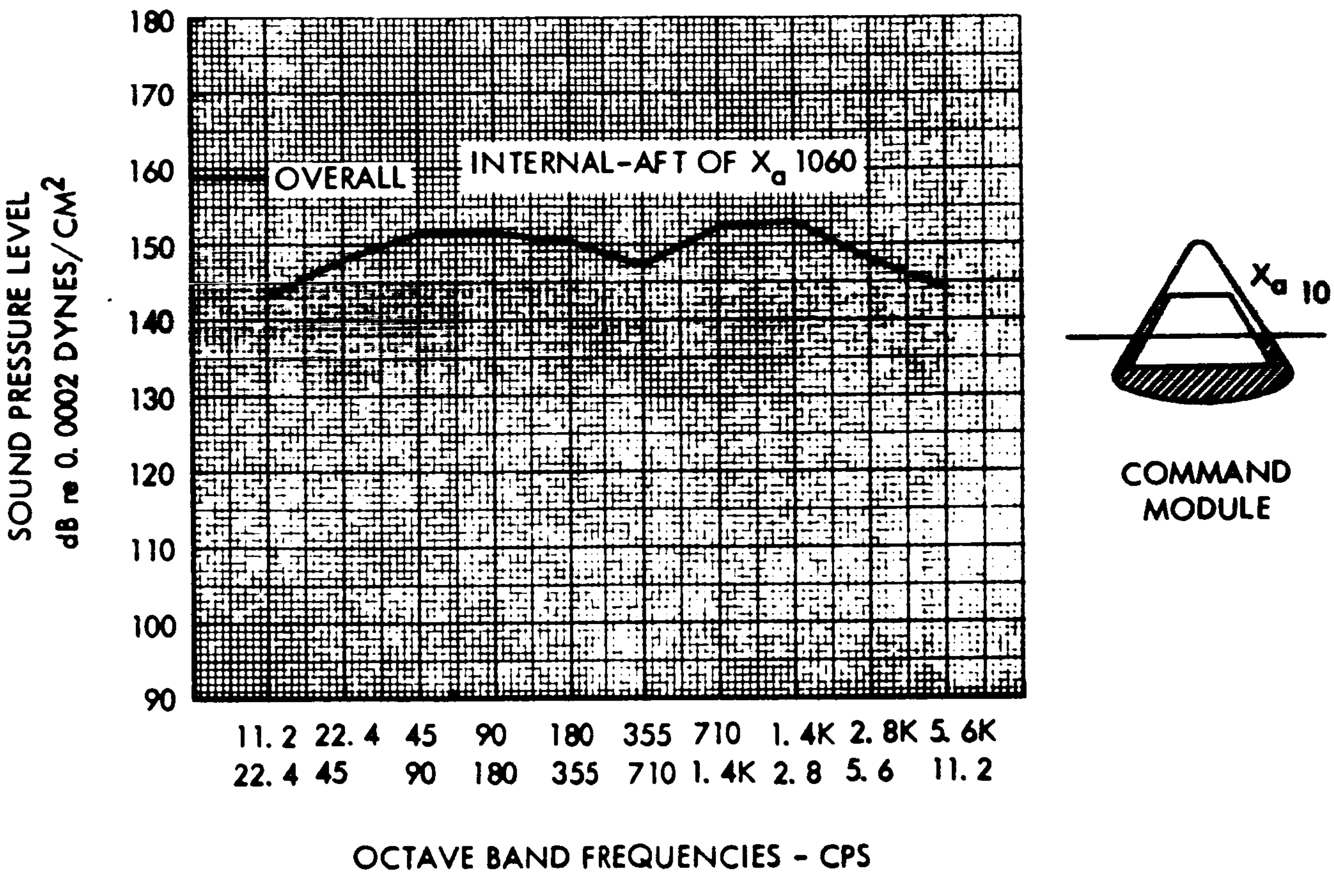


Figure 41. Acoustics CM - High "Q" Abort - Internal - Aft of Xa1060

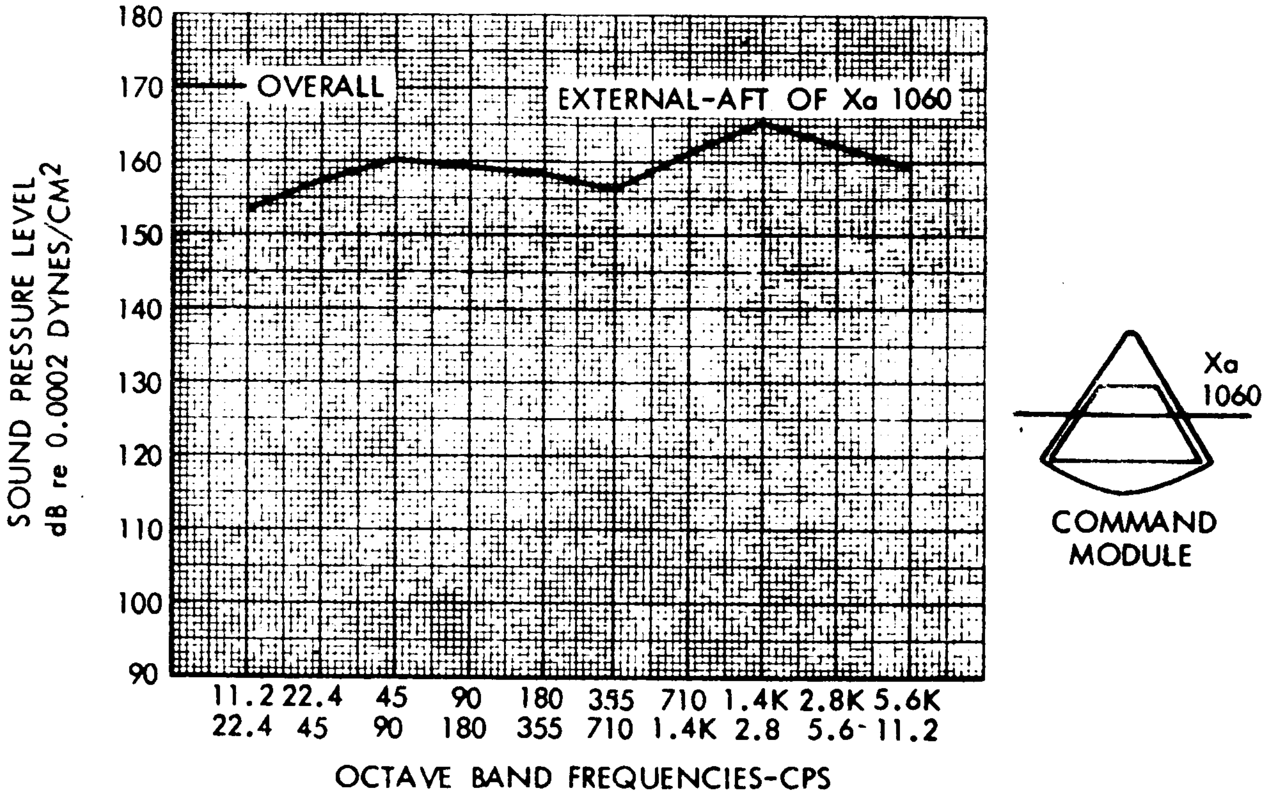


Figure 42 Acoustics CM - High "Q" Abort - External - Aft of Xa1060

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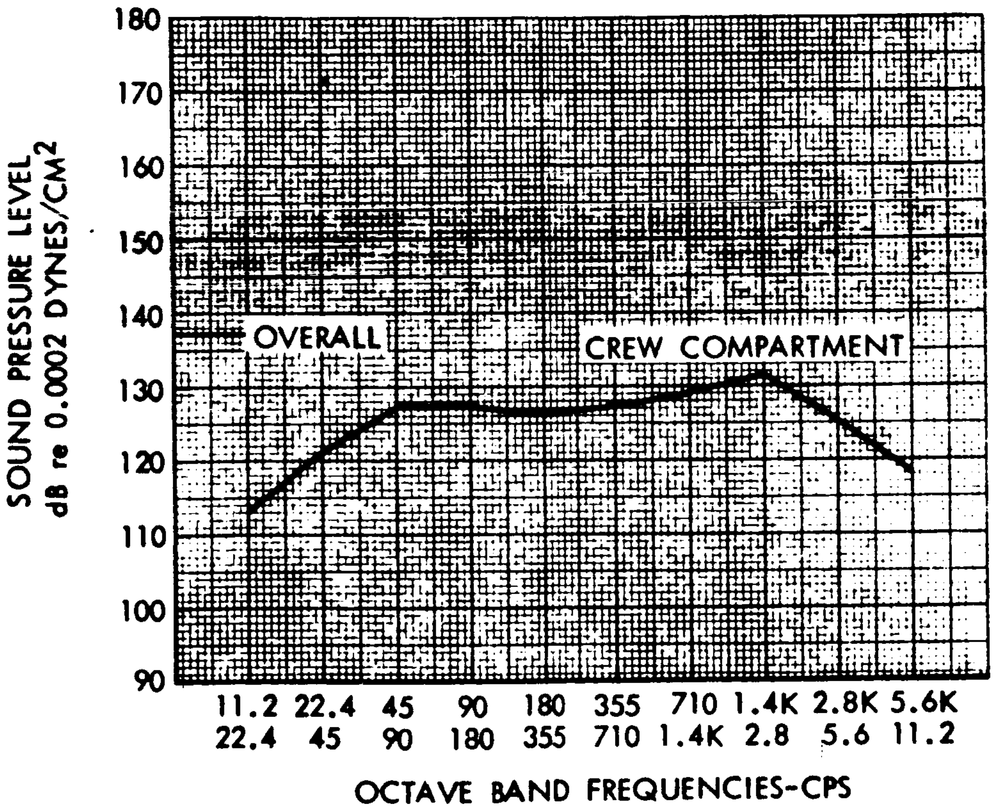


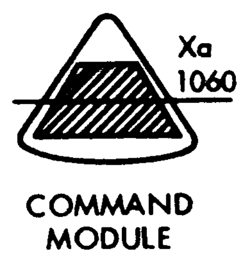


Figure 43 Acoustics CM - High "Q" Abort - Crew Compartment

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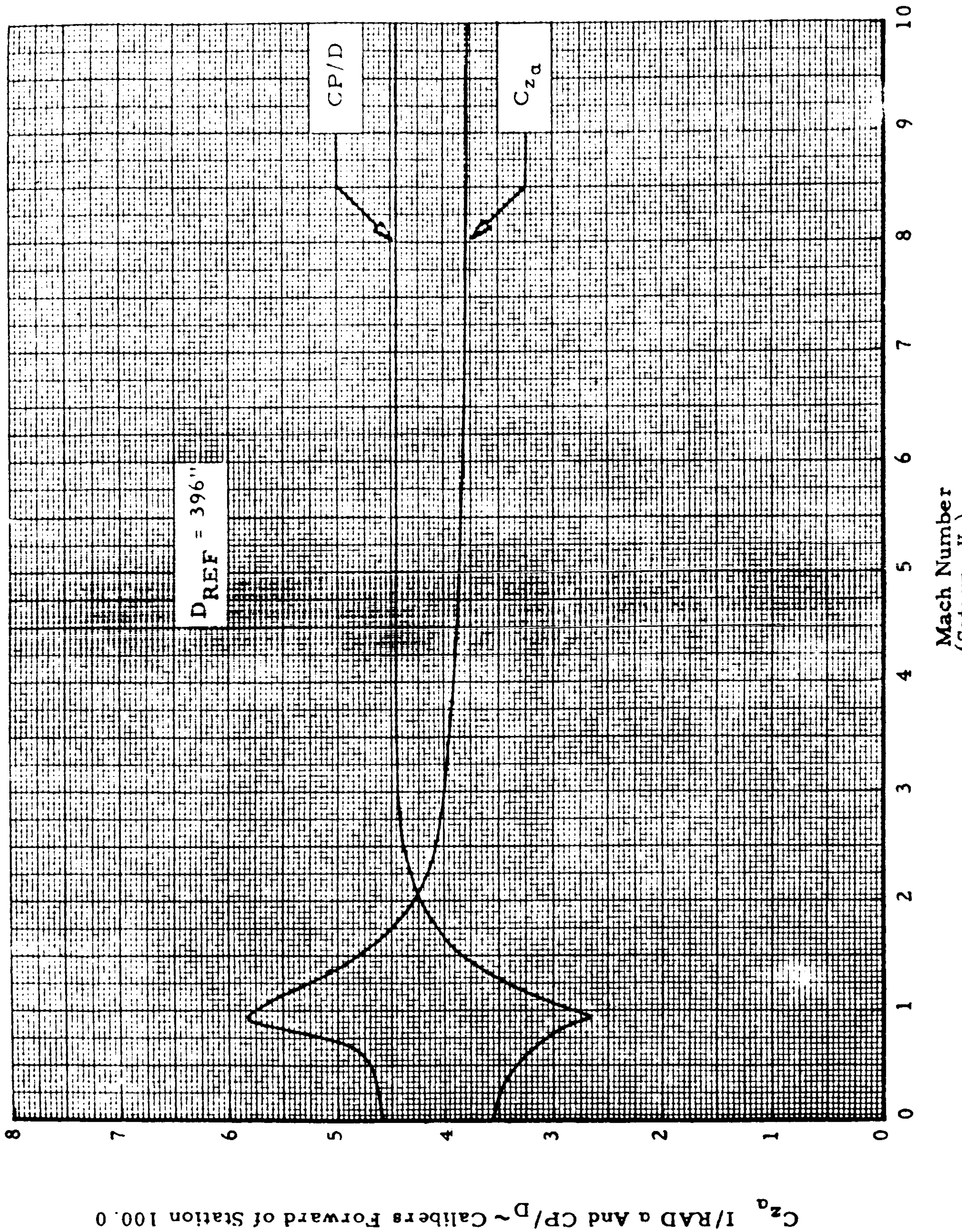


Figure 44. Gradient of Normal Force Coefficient and Center of Pressure Versus Mach Numbers



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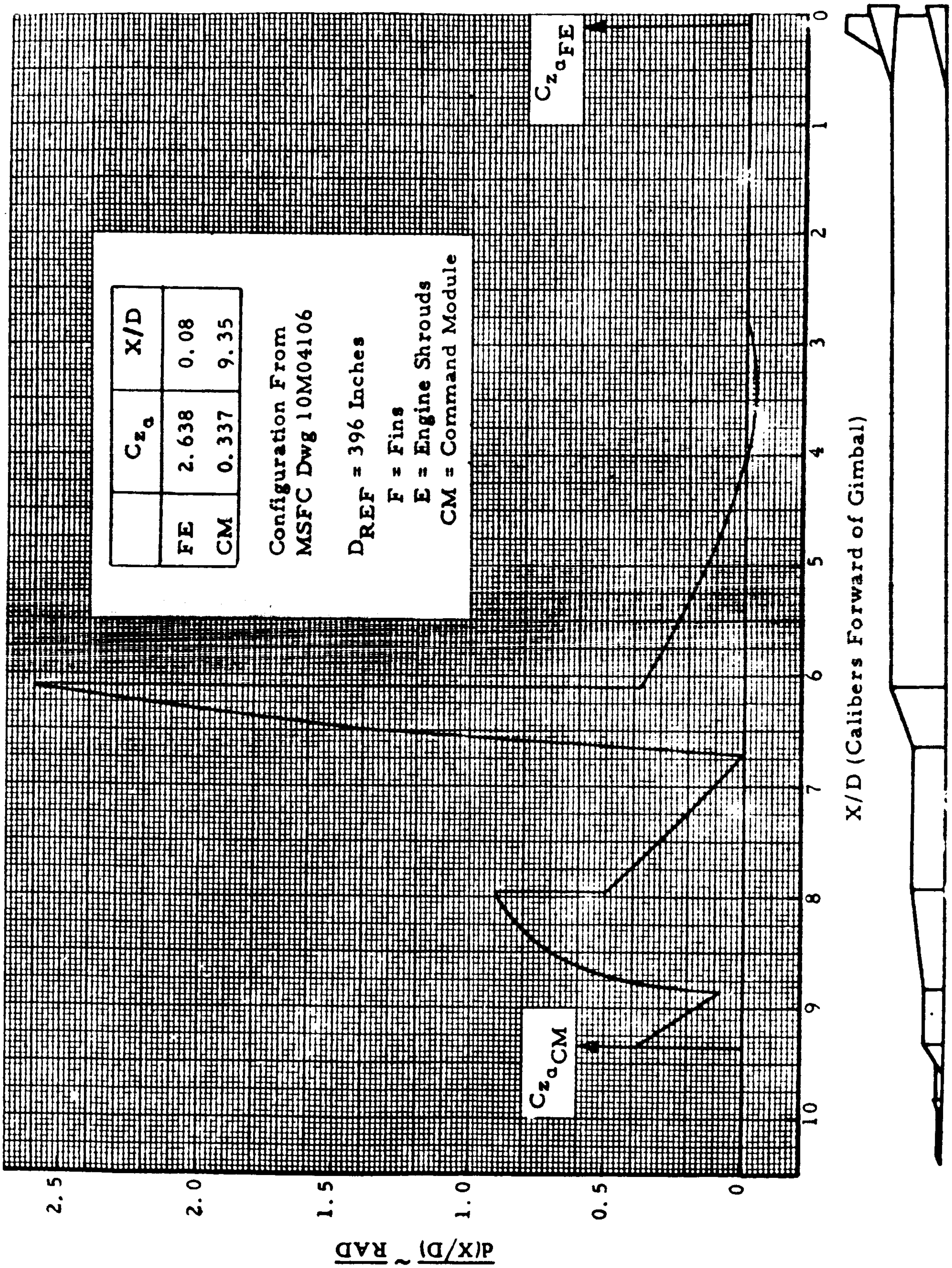


Figure 45. Linear Load Distribution for Mach Numbers 1.35 (Saturn V)

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$a_1$ = RATE GYRO GAIN FACTOR = .97
$a_o$ = POSITION GYRO GAIN FACTOR = .38
$b_o$ = ANGLE OF ATTACK FEEDBACK GAIN FACTOR = .27
v = VEHICLE VELOCITY
s = LAPLACE TRANSFORM

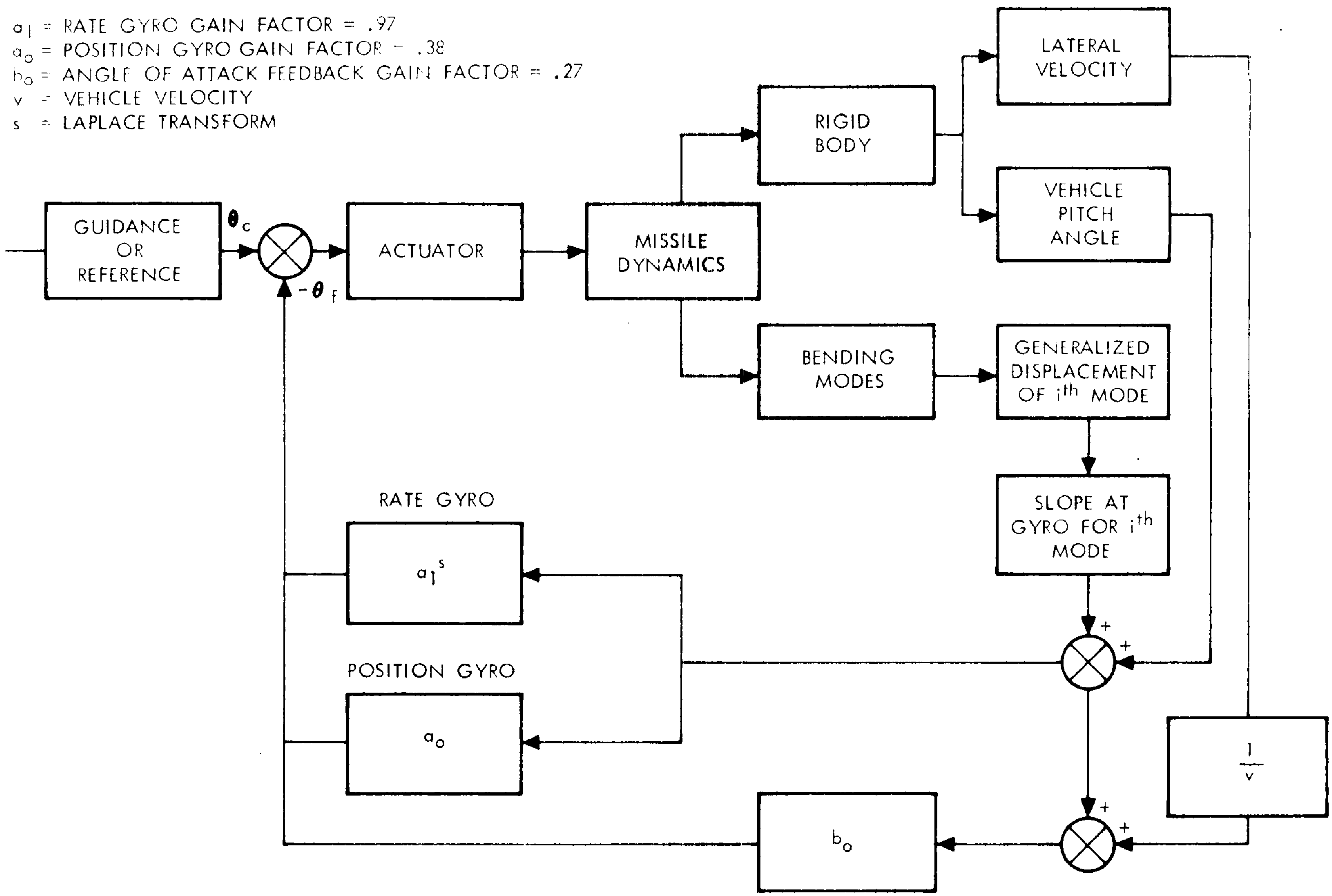


Figure 46 Booster Control System Block Diagram



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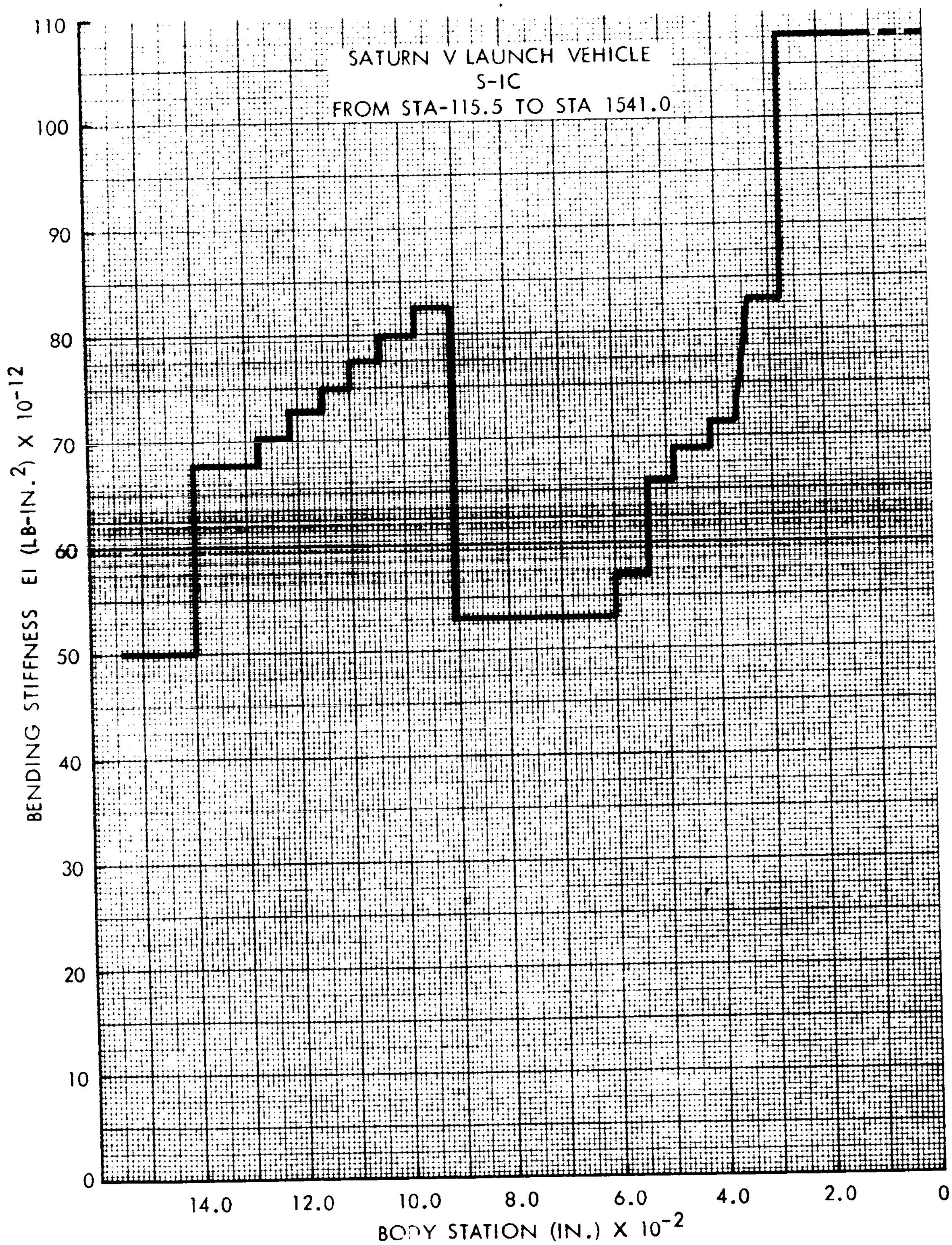


Figure 47 Saturn V, S-IC, EI Distribution

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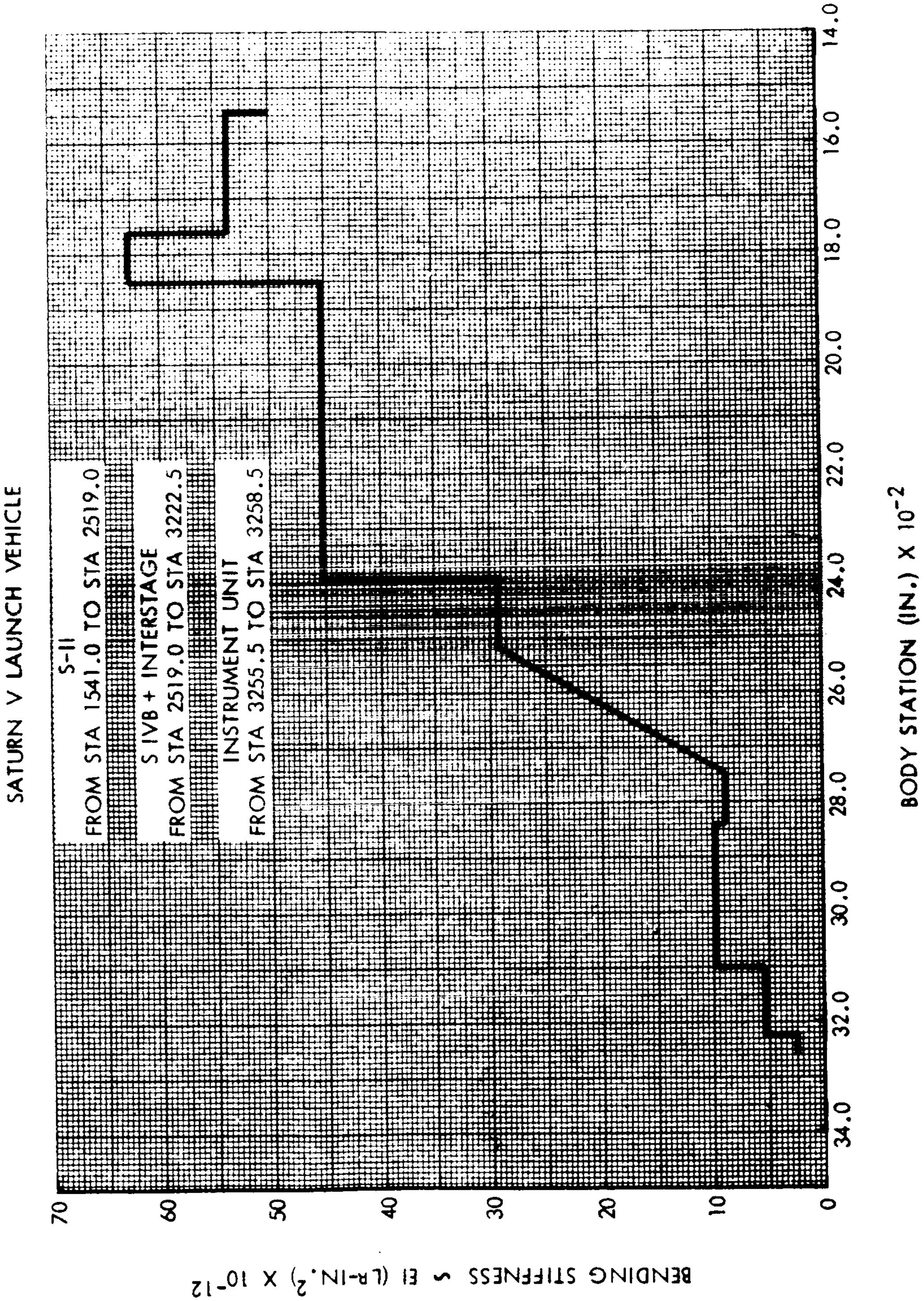


Figure 48. Saturn V, SSI, EI Distribution





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SATURN V LAUNCH VEHICLE

S-IC
FROM STA-115.5 TO STA 1541.0

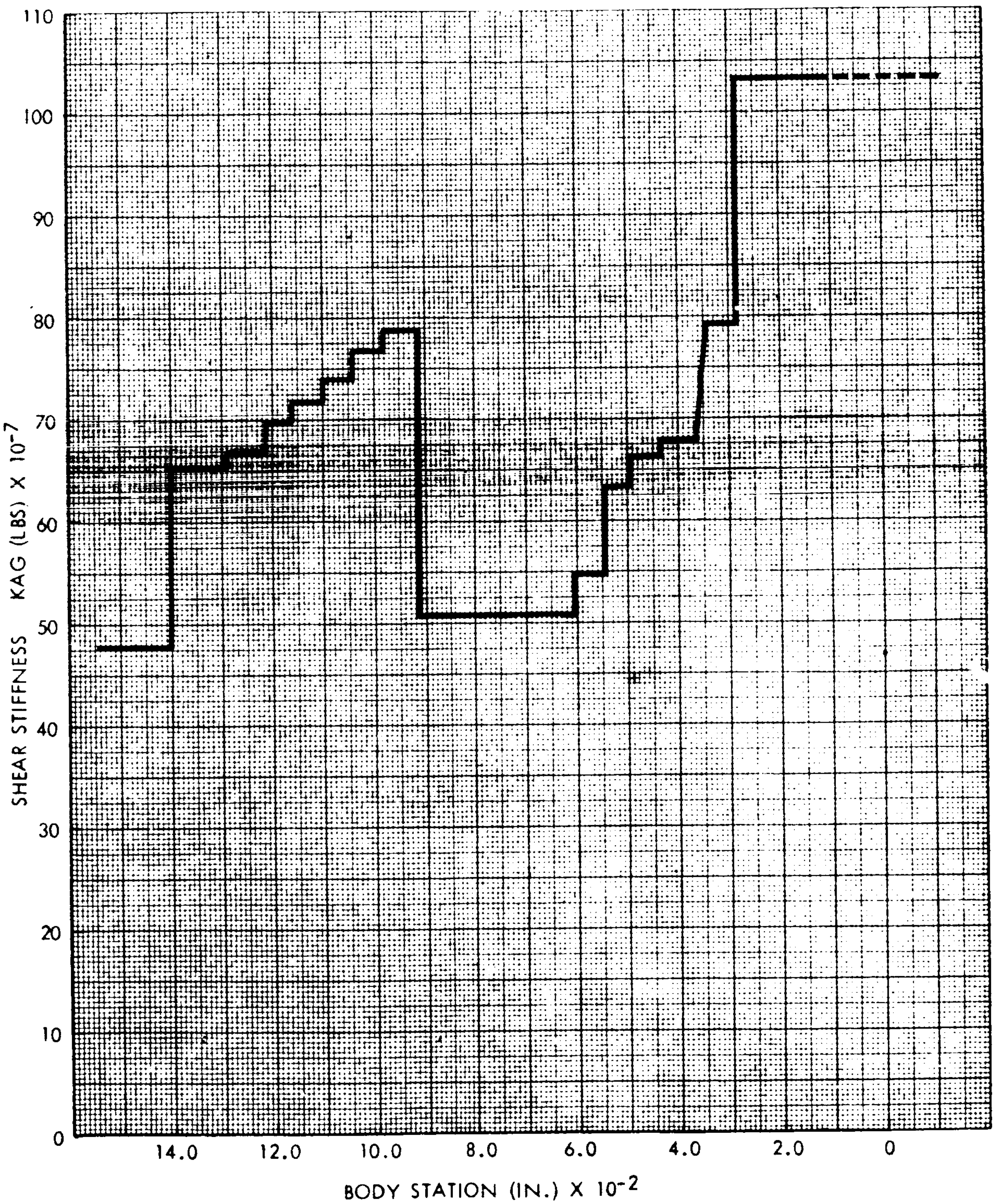


Figure 49. Saturn V, S-IC, KAG Distribution



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CO[redacted]

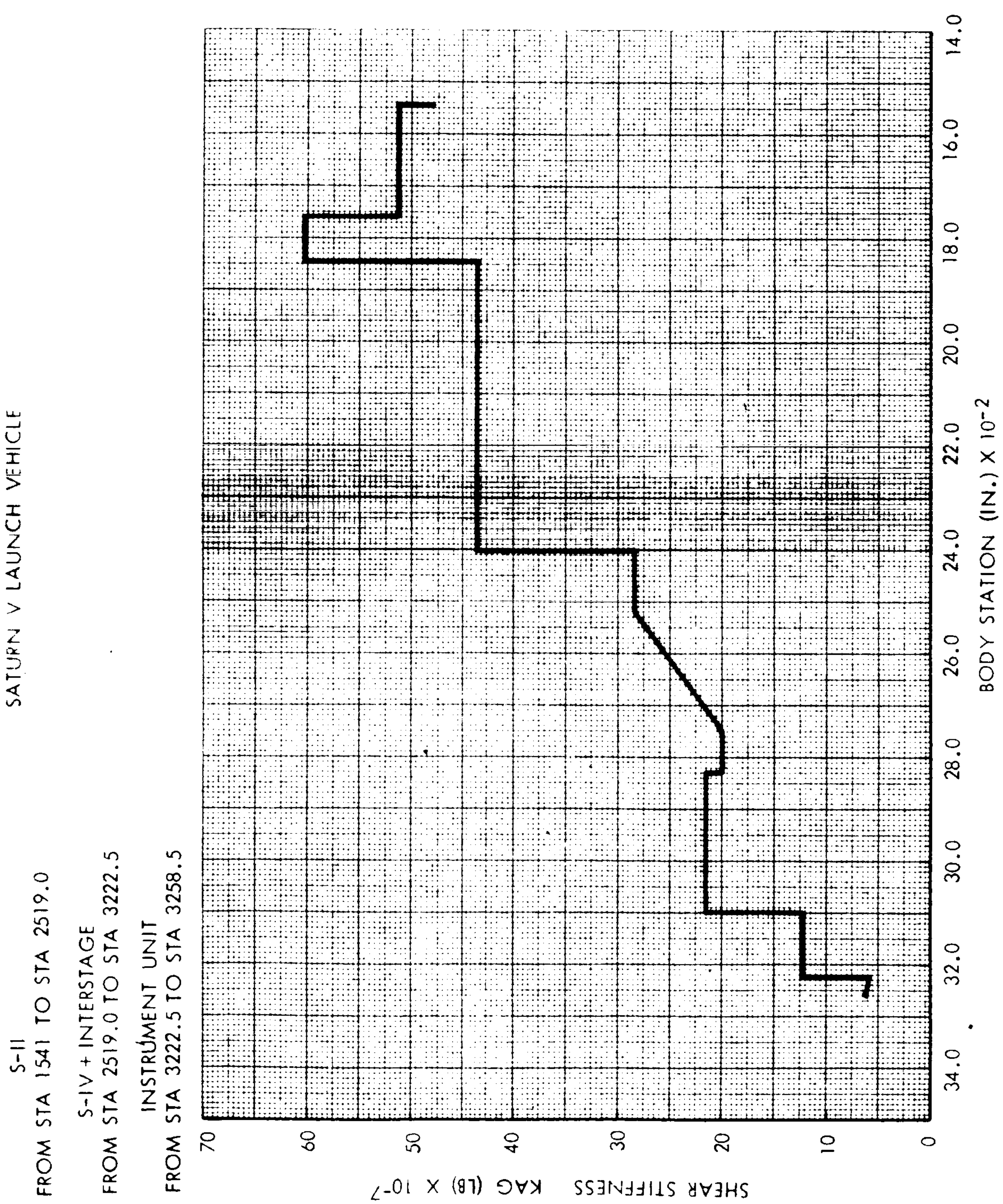


Figure 50 Saturn V, S-II, KAG Distribution



[redacted]

CON[redacted]

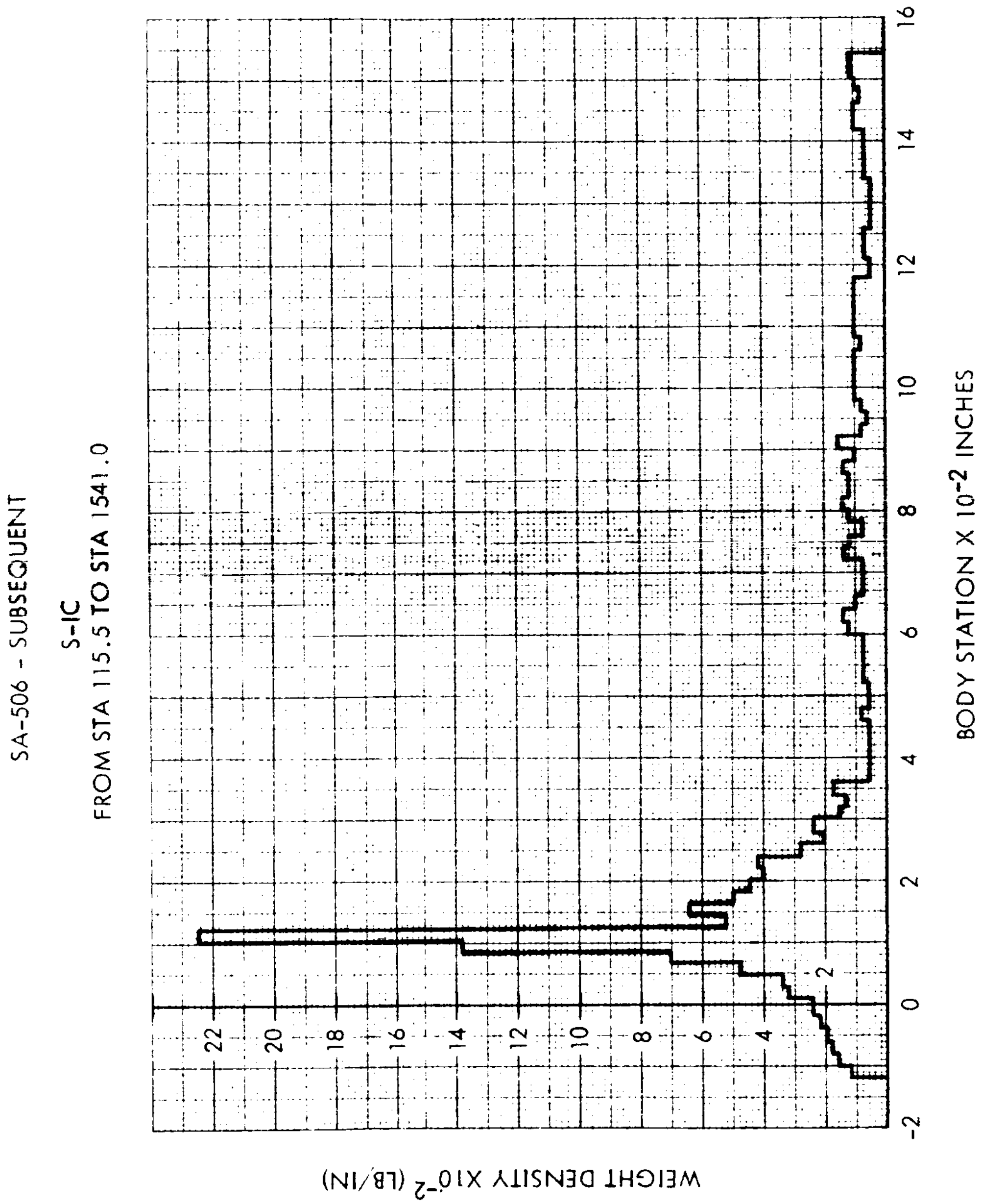


Figure 51 Operational Saturn V Launch Vehicle Dry Weight Distribution Along the Longitudinal Axis



[redacted]

CON[redacted]

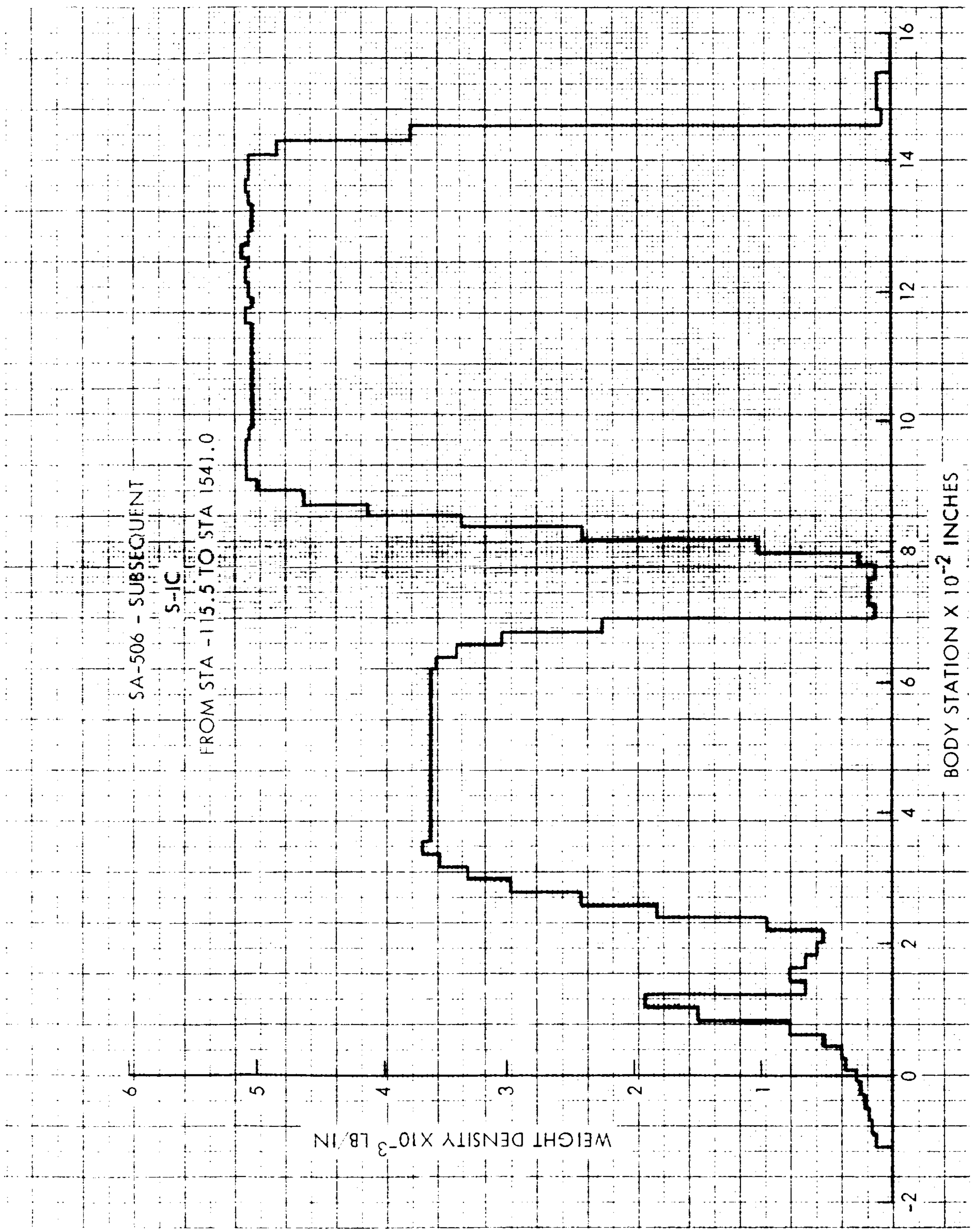


Figure 52 Operational Saturn V Launch Vehicle Wet Weight Distribution Along the Longitudinal Axis



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CON[redacted]

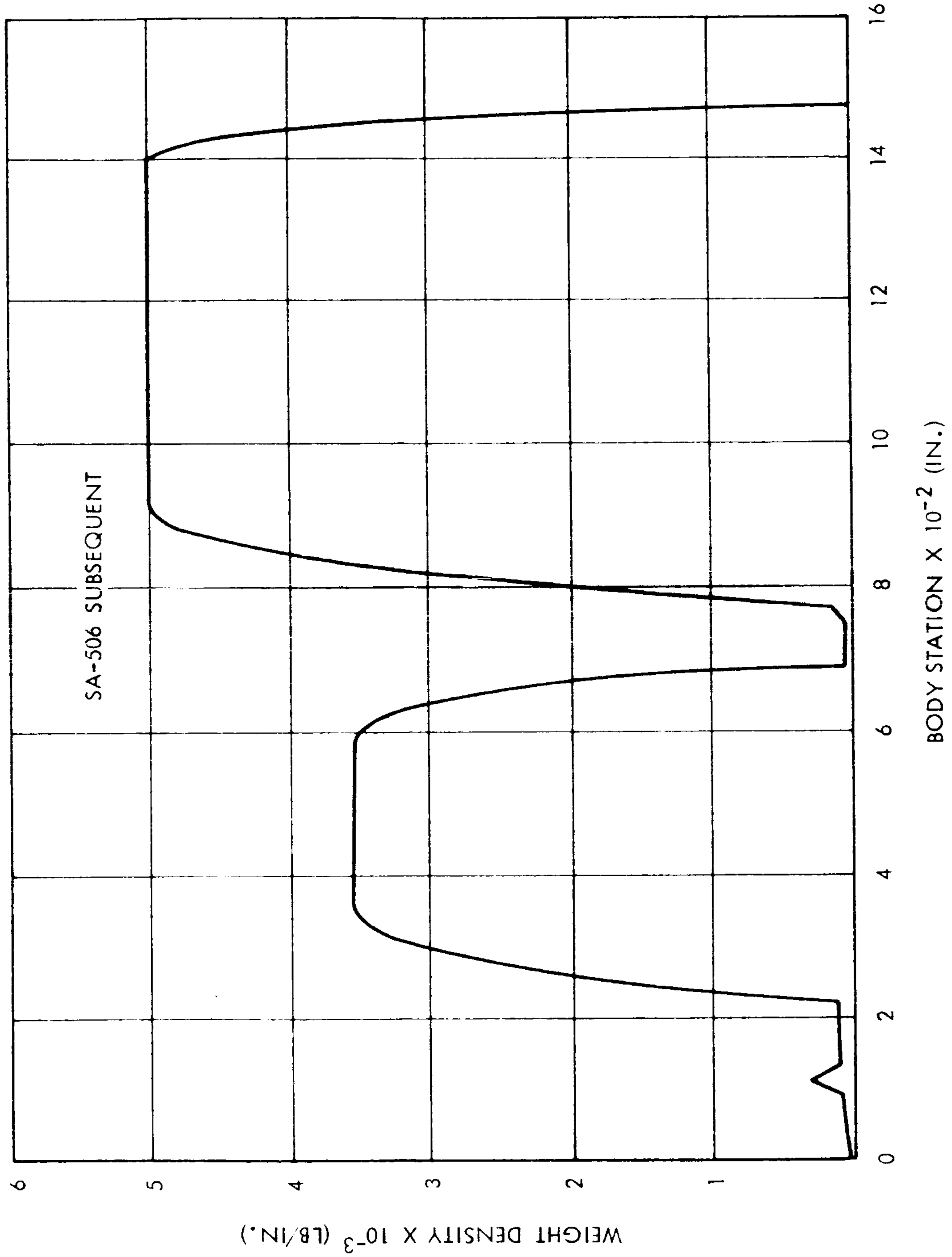


Figure 53 Operational Saturn V Launch Vehicle Propellant Distribution S-IC



[redacted]

CON[redacted]

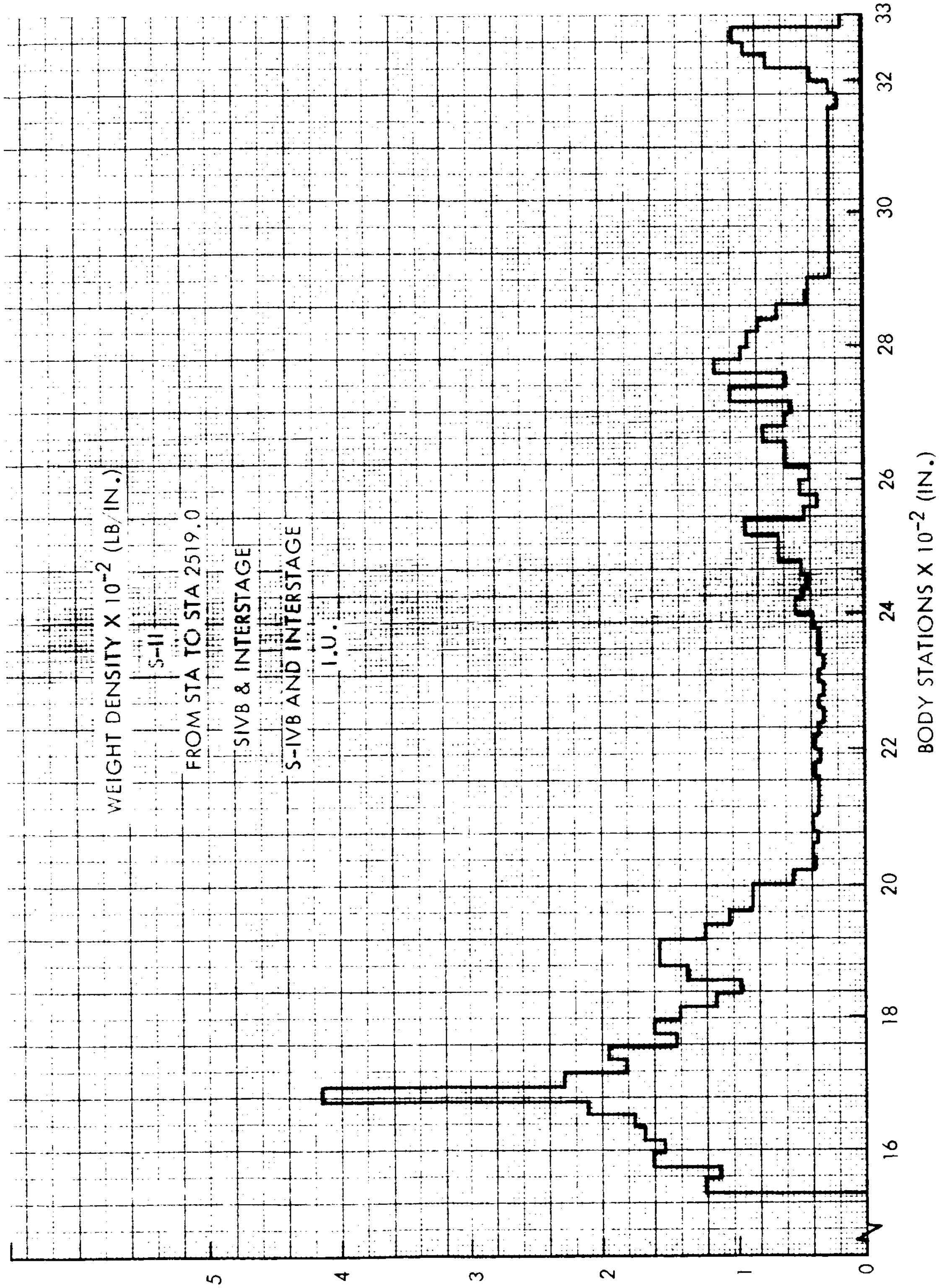


Figure 54 Operational Saturn V Launch Vehicle Dry Weight Distribution Along the Longitudinal Axis



[redacted]

CON[redacted]

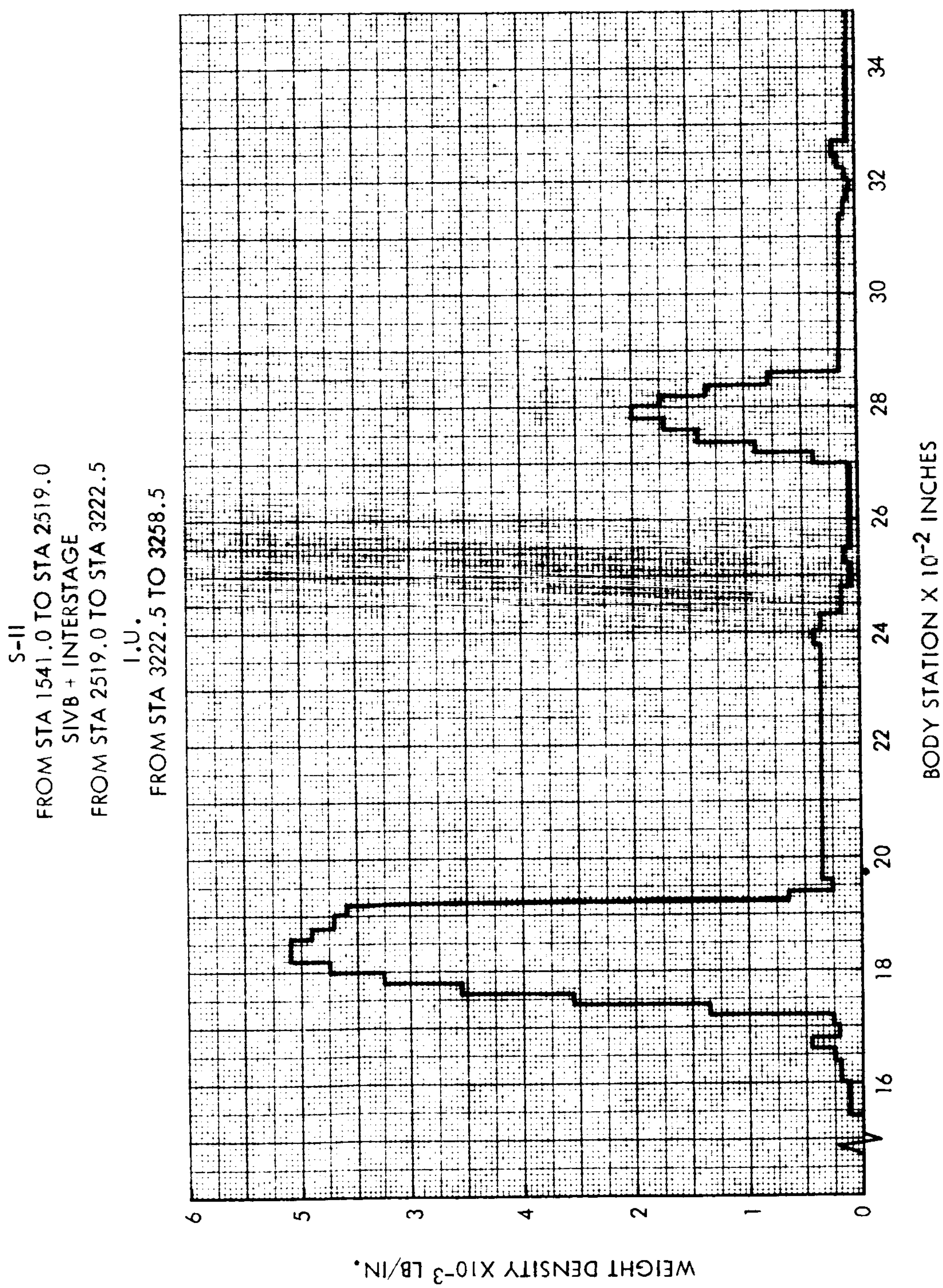


Figure 55 Operational Saturn V Launch Vehicle Wet Weight Distribution Along the Longitudinal Axis



[redacted]

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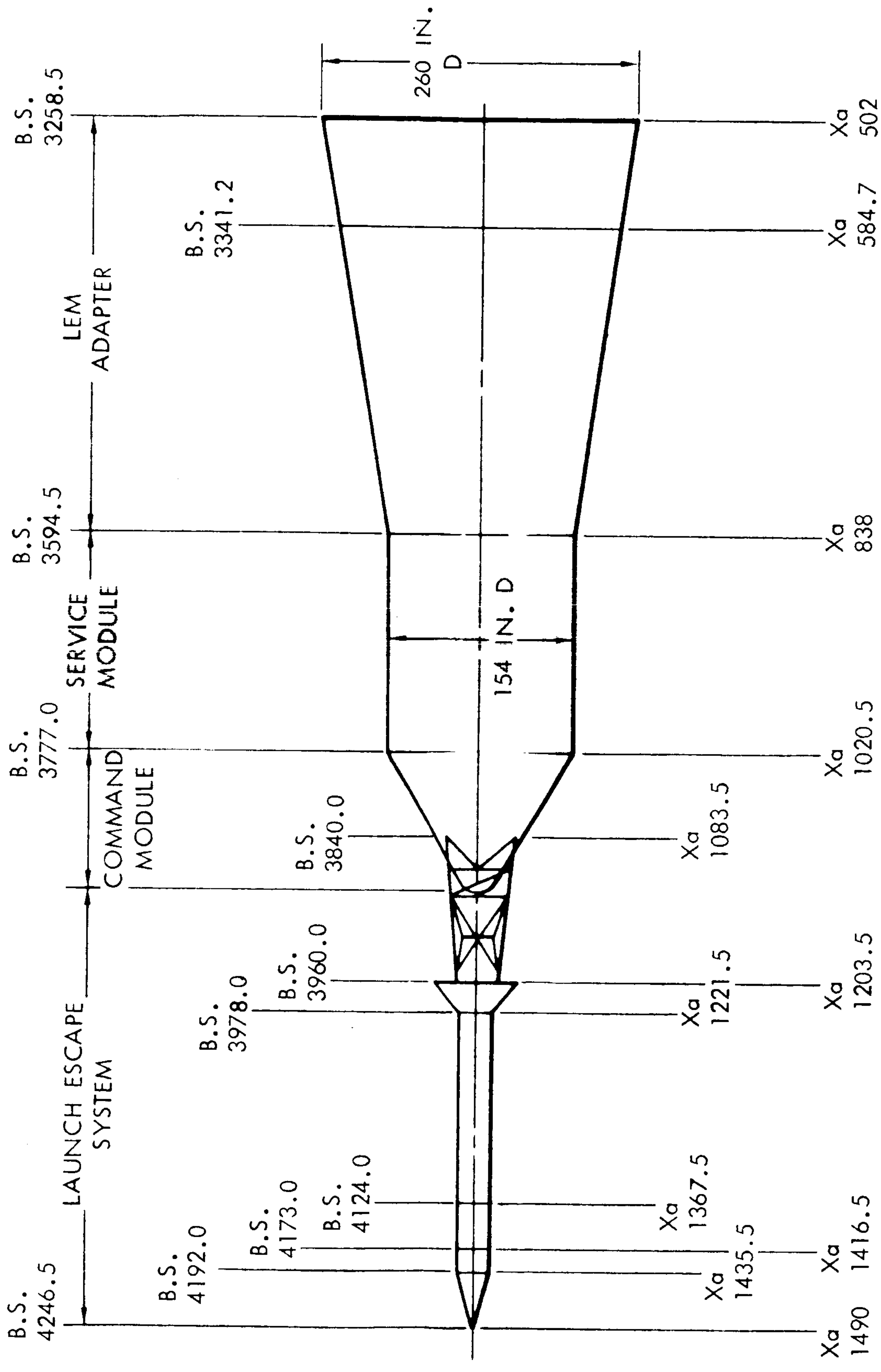


$S_{REF} = 129\ FT^2$ $D_{REF} = 154$ IN.

Figure 56 Spacecraft Geometry



~~CONFIDENTIAL~~

| COMPONENT | BREAK STATIONS (IN.) | LINEARIZED FOR 7° | | LINEARIZED FOR 8° | | LINEARIZED FOR 9° | |
|---|---|---|---|---|---|---|---|
| | | CN PER RADIAN | C.P. (IN.) | CN PER RADIAN | C.P. (IN.) | CN PER RADIAN | C.P. (IN.) |
| LES NOSE CONE | 1490.0-1435.5 | .00951 | 1453.0 | .00933 | 1453.0 | .00933 | 1453.0 |
| KICKER MOTOR | 1435.5-1416.5 | .00190 | 1426.0 | .00181 | 1426.0 | .00173 | 1426.0 |
| JETTISON MOTOR | 1416.5-1367.5 | .00268 | 1394.6 | .00251 | 1394.6 | .00242 | 1394.6 |
| LES MOTOR | 1367.5-1221.5 | .00812 | 1269.0 | .00752 | 1269.0 | .00683 | 1269.0 |
| LES MOTOR SKIRT | 1221.5-1203.5 | .01132 | 1211.0 | .01046 | 1211.0 | .00985 | 1211.0 |
| LES TOWER | 1203.5-1083.5 | .01573 | 1120.0 | .01555 | 1120.0 | .01581 | 1120.0 |
| FWD HEATSHIELD & CREW COMP | 1083.5-1020.5 | .26539 | 1056.7 | .25597 | 1058.5 | .24241 | 1060.2 |
| SERVICE MODULE | 1020.5-838.0 | .06793 | 947.0 | .09610 | 945.0 | .12246 | 944.0 |
| FWD LEM ADAPTER | 838.0-584.7 | .26539 | 694.8 | .27110 | 694.0 | .27844 | 694.3 |
| AFT LEM ADAPTER | 584.7-502.0 | .17033 | 542.9 | .16636 | 543.5 | .16230 | 543.8 |

Figure 57 Spacecraft Normal Force Coefficient Derivative (CN Iradian) and Center of Pressure (CP. IN. $S_{Ref}$ = 855 $Ft^2$)

CONFI[redacted]

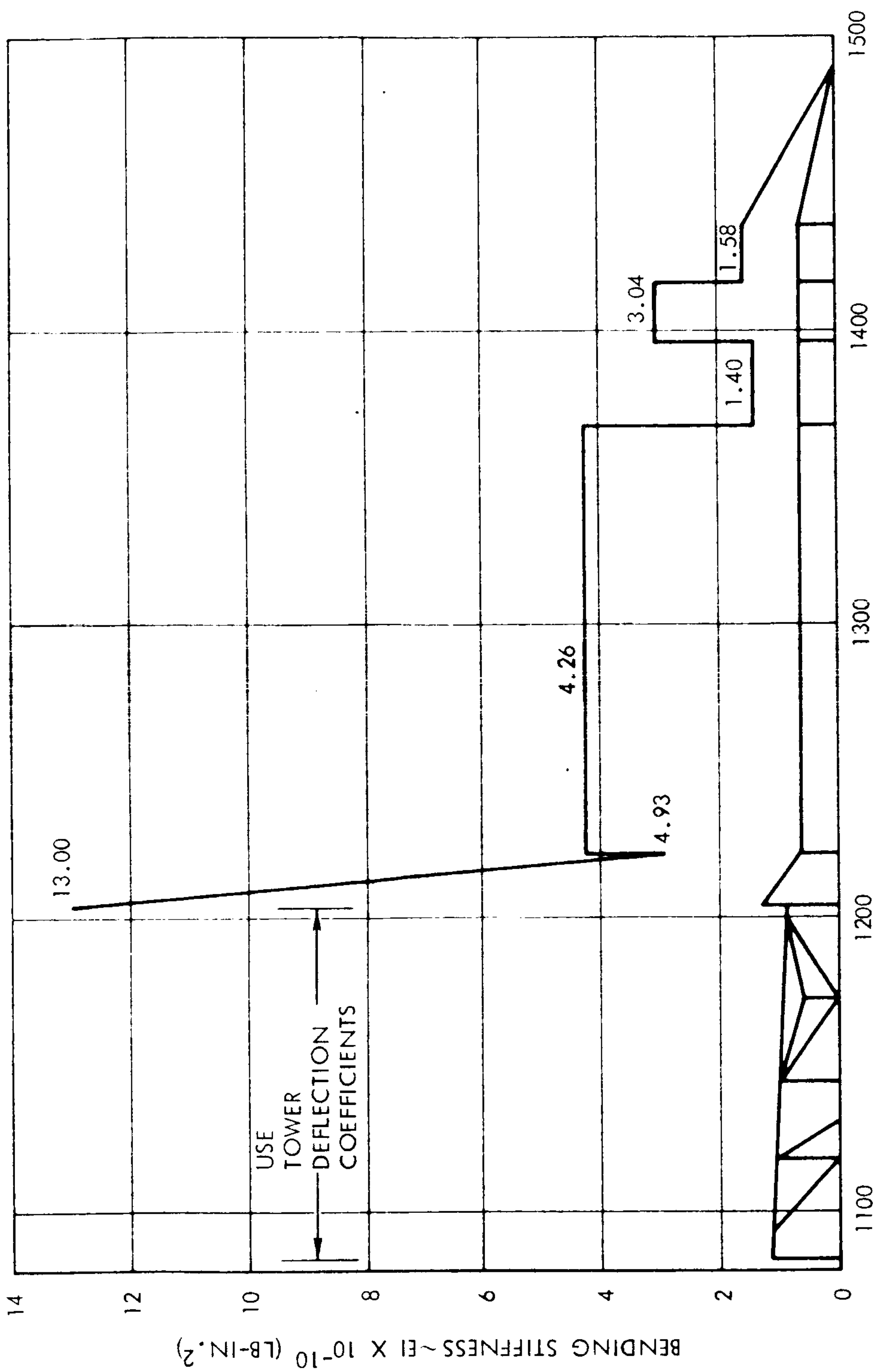


Figure 58 Block II Launch Escape System Bending Stiffness Distribution



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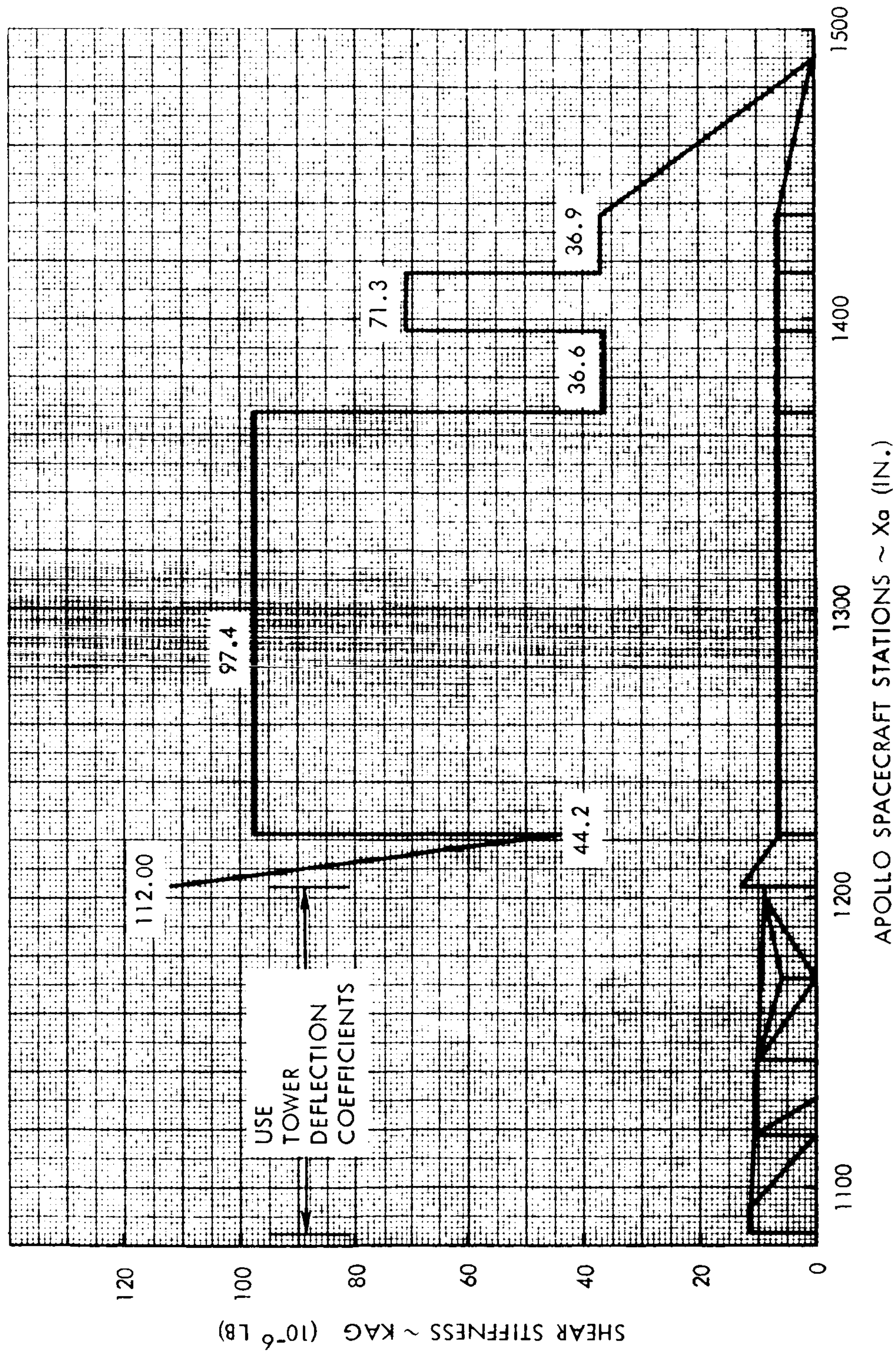


Figure 59 Block II Launch Escape System Shear Stiffness Distribution



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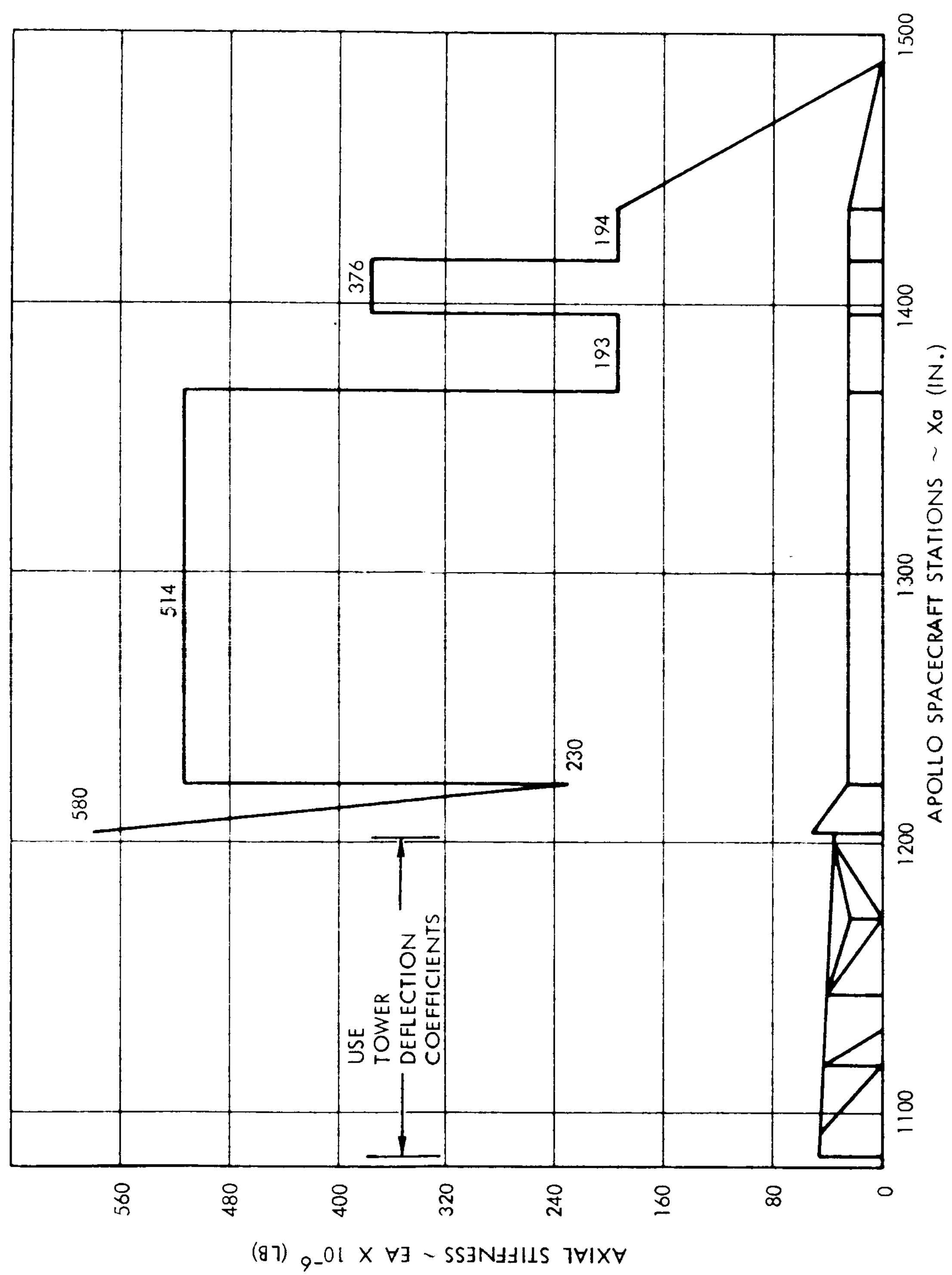
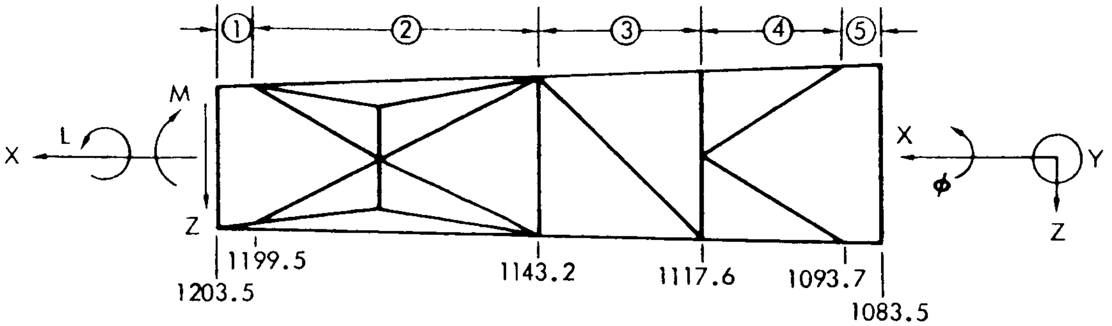



Figure 60 Block II Launch Escape System Axial Stiffness Distribution

CON[redacted]




LATERAL DEFLECTION COEFFICIENTS

SECTION ① $\begin{Bmatrix} \delta_Z \\ \theta \end{Bmatrix} = 10^{-6} \begin{bmatrix} 2.39 & .00773 \\ .00677 & .0000543 \end{bmatrix} \begin{Bmatrix} Z \\ M \end{Bmatrix}$

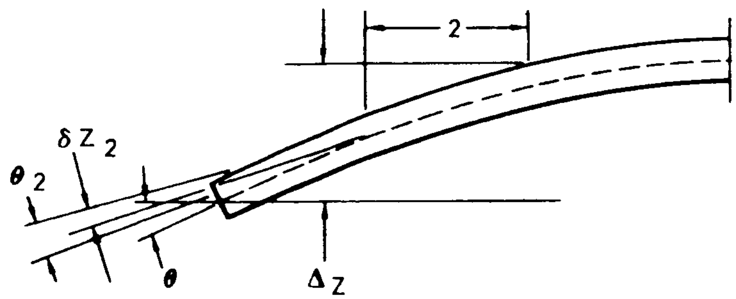
SECTION ② $\begin{Bmatrix} \delta_Z \\ \theta \end{Bmatrix} = 10^{-6} \begin{bmatrix} 9.58 & -.0252 \\ -.0244 & .001769 \end{bmatrix} \begin{Bmatrix} Z \\ M \end{Bmatrix}$

SECTION ③ $\begin{Bmatrix} \delta_Z \\ \theta \end{Bmatrix} = 10^{-6} \begin{bmatrix} 5.04 & .00433 \\ .00364 & .000668 \end{bmatrix} \begin{Bmatrix} Z \\ M \end{Bmatrix}$

SECTION ④ $\begin{Bmatrix} \delta_Z \\ \theta \end{Bmatrix} = 10^{-6} \begin{bmatrix} 4.48 & .00592 \\ .01154 & .000582 \end{bmatrix} \begin{Bmatrix} Z \\ M \end{Bmatrix}$

SECTION ⑤ $\begin{Bmatrix} \delta_Z \\ \theta \end{Bmatrix} = 10^{-6} \begin{bmatrix} 9.71 & .00323 \\ -.001492 & .0001452 \end{bmatrix} \begin{Bmatrix} Z \\ M \end{Bmatrix}$

OVERALL: $\begin{Bmatrix} \Delta_Z \\ \theta \end{Bmatrix} = 10^{-6} \begin{bmatrix} 37.74 & -.1172 \\ -.1172 & .003218 \end{bmatrix} \begin{Bmatrix} Z \\ M \end{Bmatrix}$




AXIAL DEFLECTION COEFFICIENTS

SECTION 1: $\delta X = .0205 \quad 10^{-6} X$
SECTION 2: $\delta X = .581 \quad 10^{-6} X$
SECTION 3: $\delta X = .289 \quad 10^{-6} X$
SECTION 4: $\delta X = .294 \quad 10^{-6} X$
SECTION 5: $\delta X = .0797 \quad 10^{-6} X$
OVERALL: $\Delta X = 1.264 \quad 10^{-6} X$

NOTE:
1. ALL LOADS ARE IN LBS OR IN-LBS.
2. ALL DIMENSIONS IN INCHES
3. ALL ROTATIONS IN RADIANS

Figure 61 Block II Launch Escape System Tower Deflection Coefficients

CON[redacted]

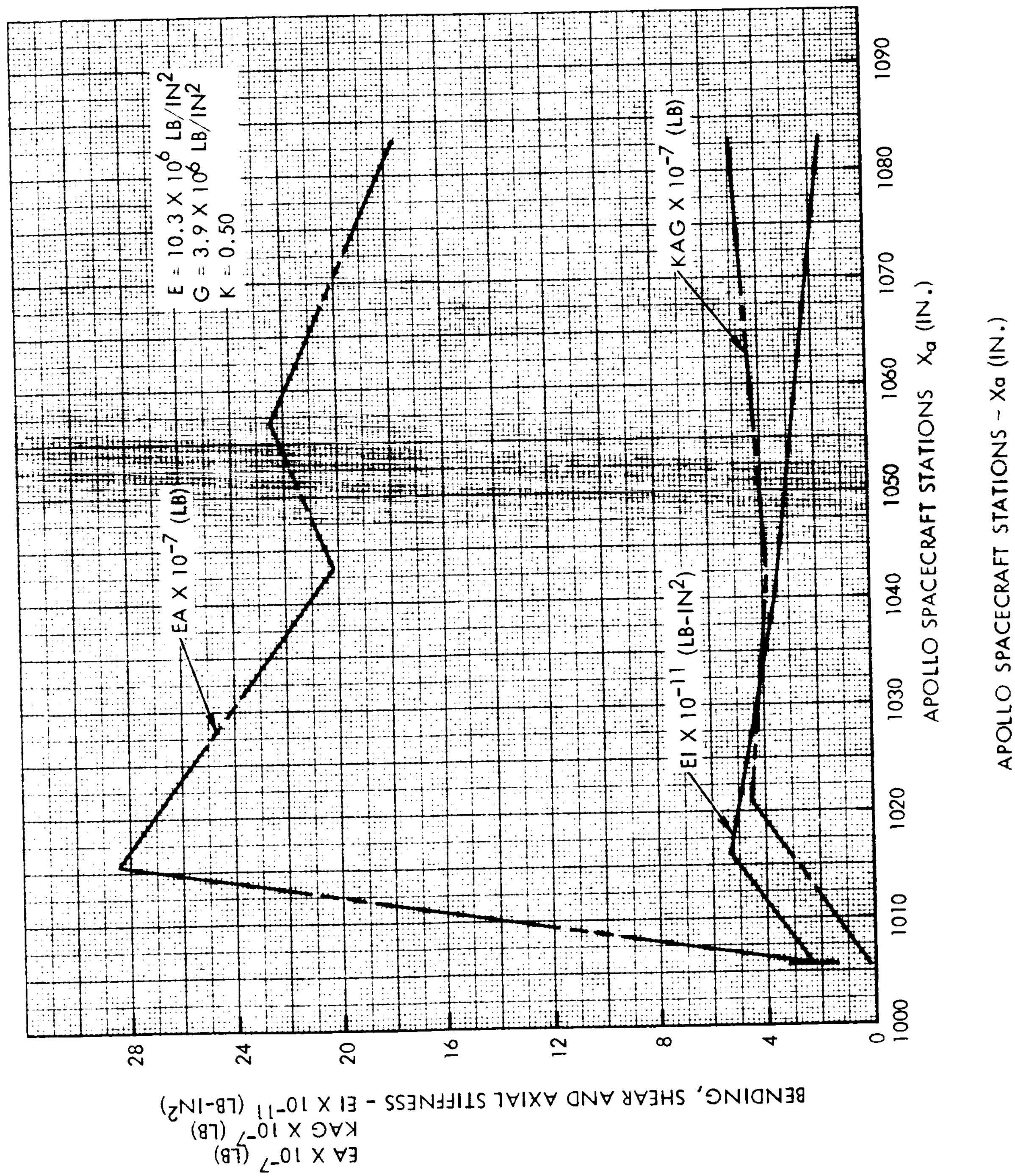



Figure 62 Block II Command Module Bending, Shear and Axial Stiffness Distribution



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[redacted]

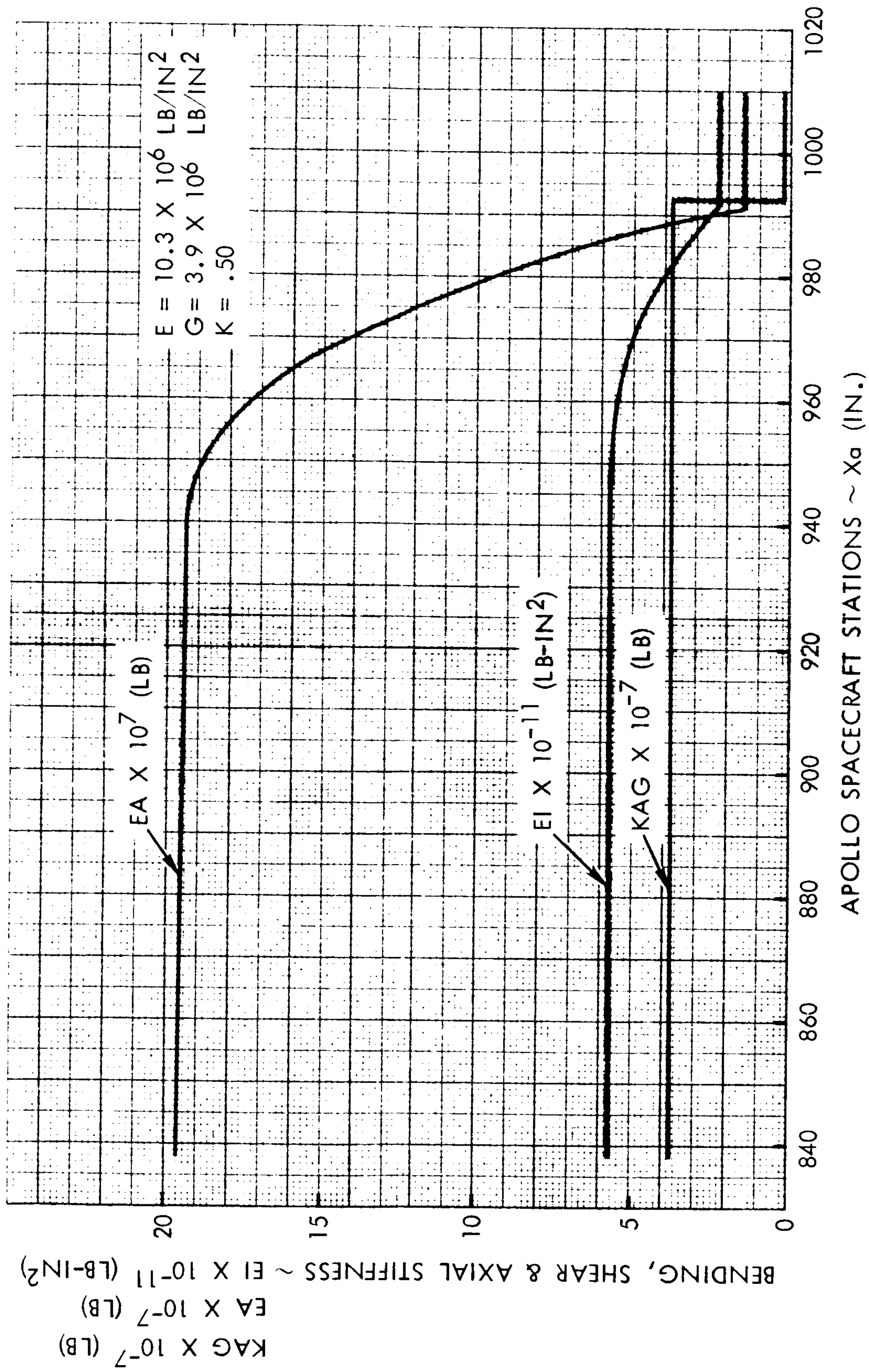


Figure 63 BLOCK II SERVICE MODULE
BENDING, SHEAR & AXIAL STIFFNESS DISTRIBUTION



[redacted]

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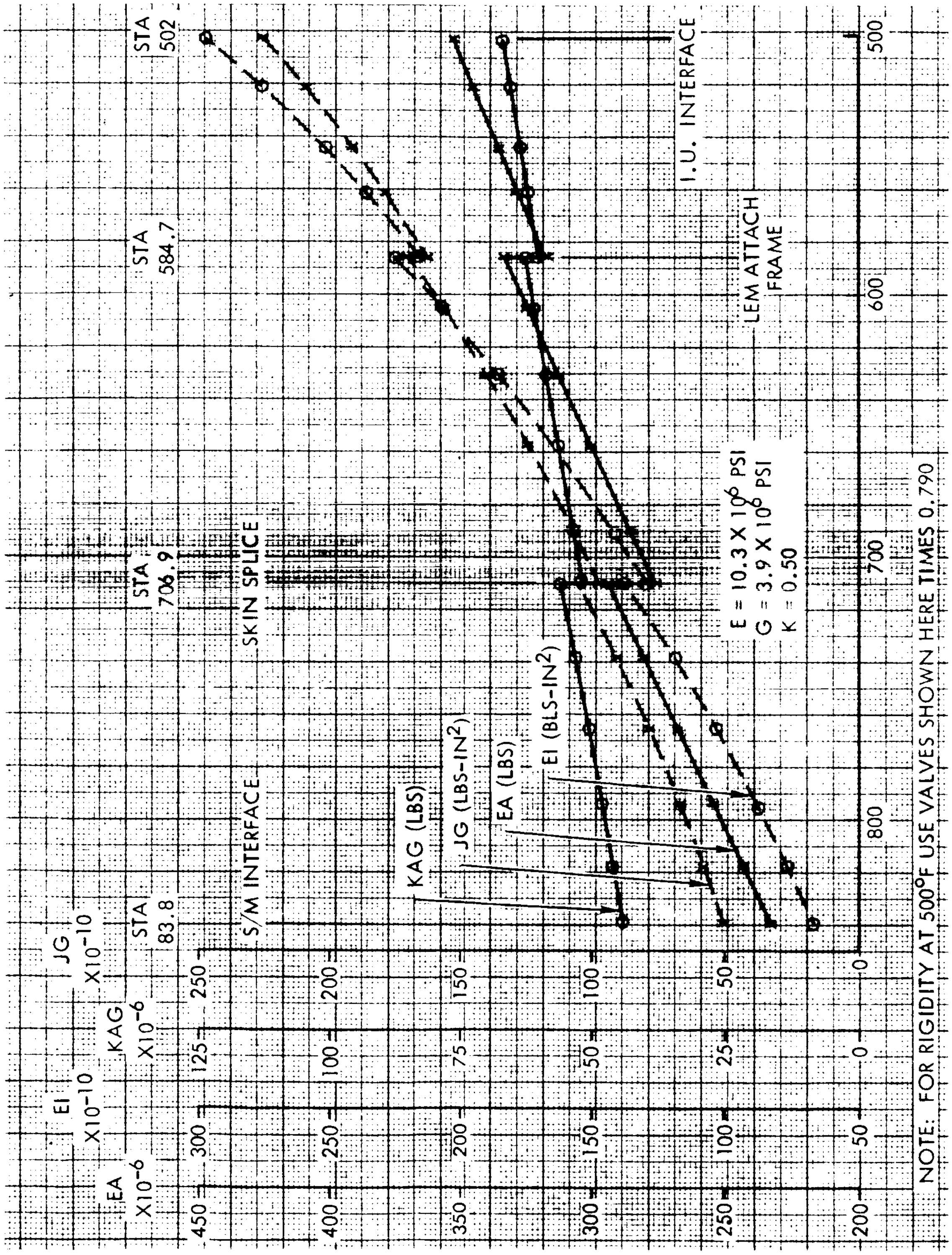


**Figure 64** Spacecraft LEM Adapter Rigidity Curves



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CON[redacted]

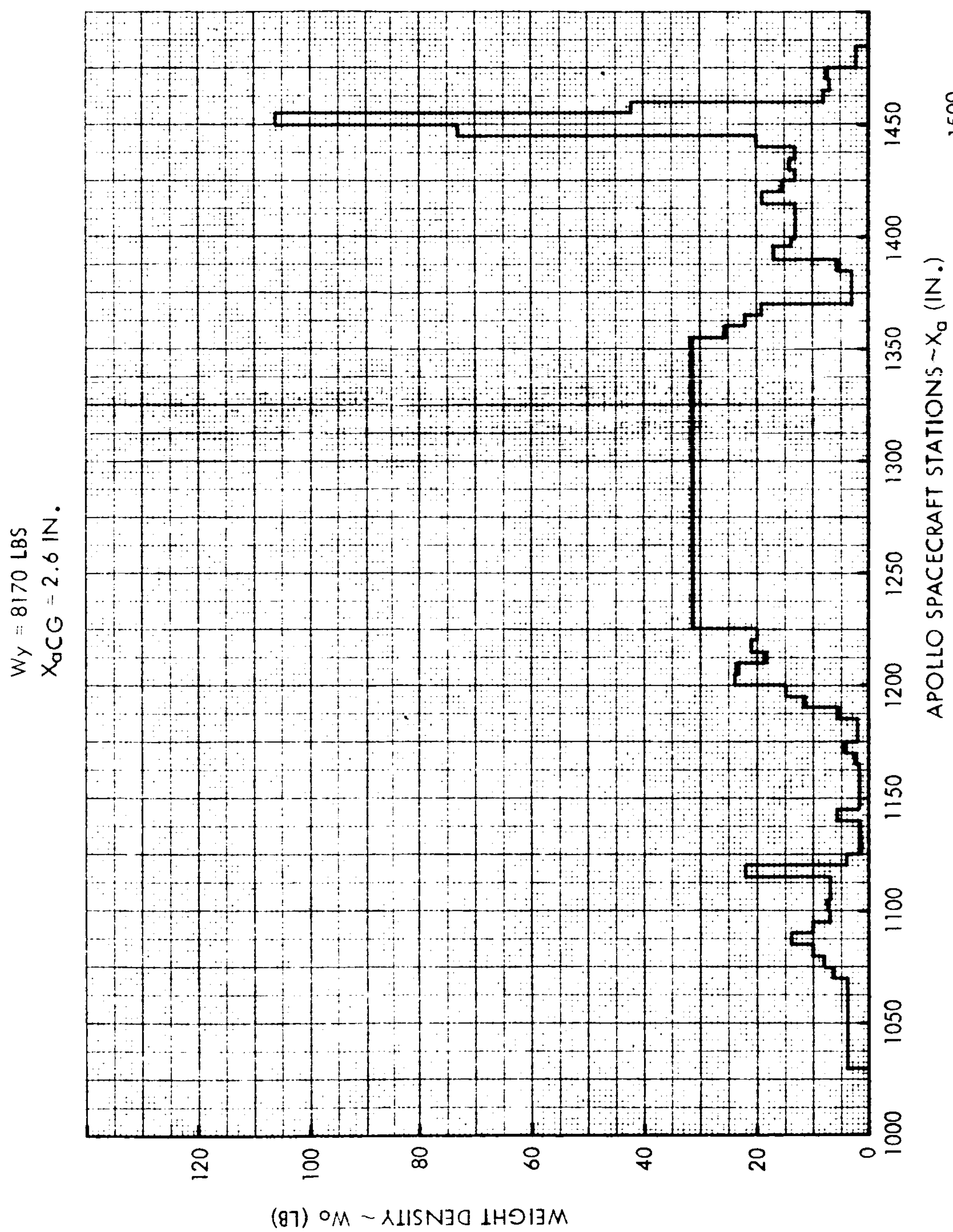


**Figure 65** Block II Launch Escape System





[redacted]IAL

CON[redacted]

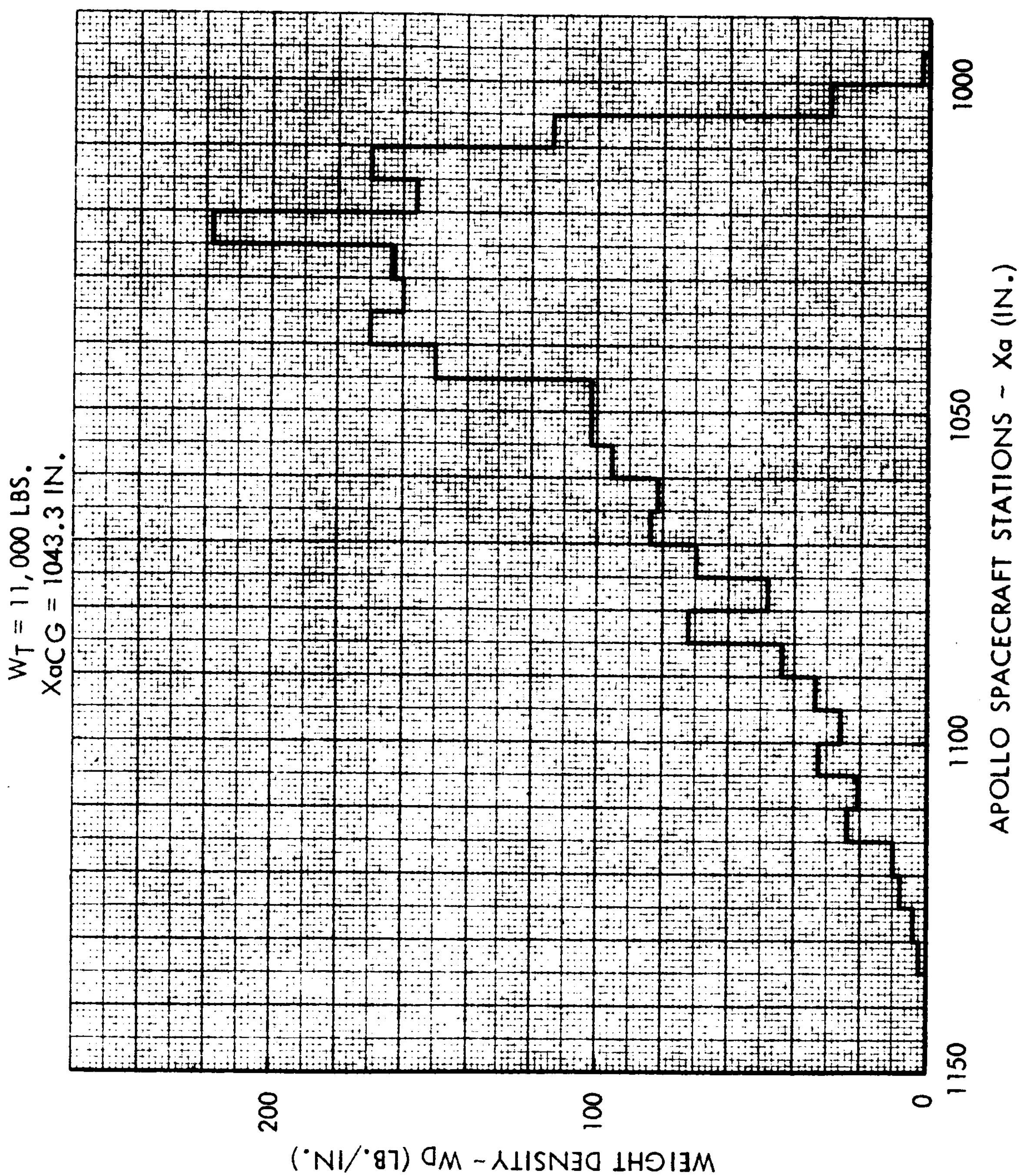


Figure 66 Block II Command Module

CO[redacted]

$W_T$ = 10,200 LBS.

$X_{aCG}$ = 904.2 IN.

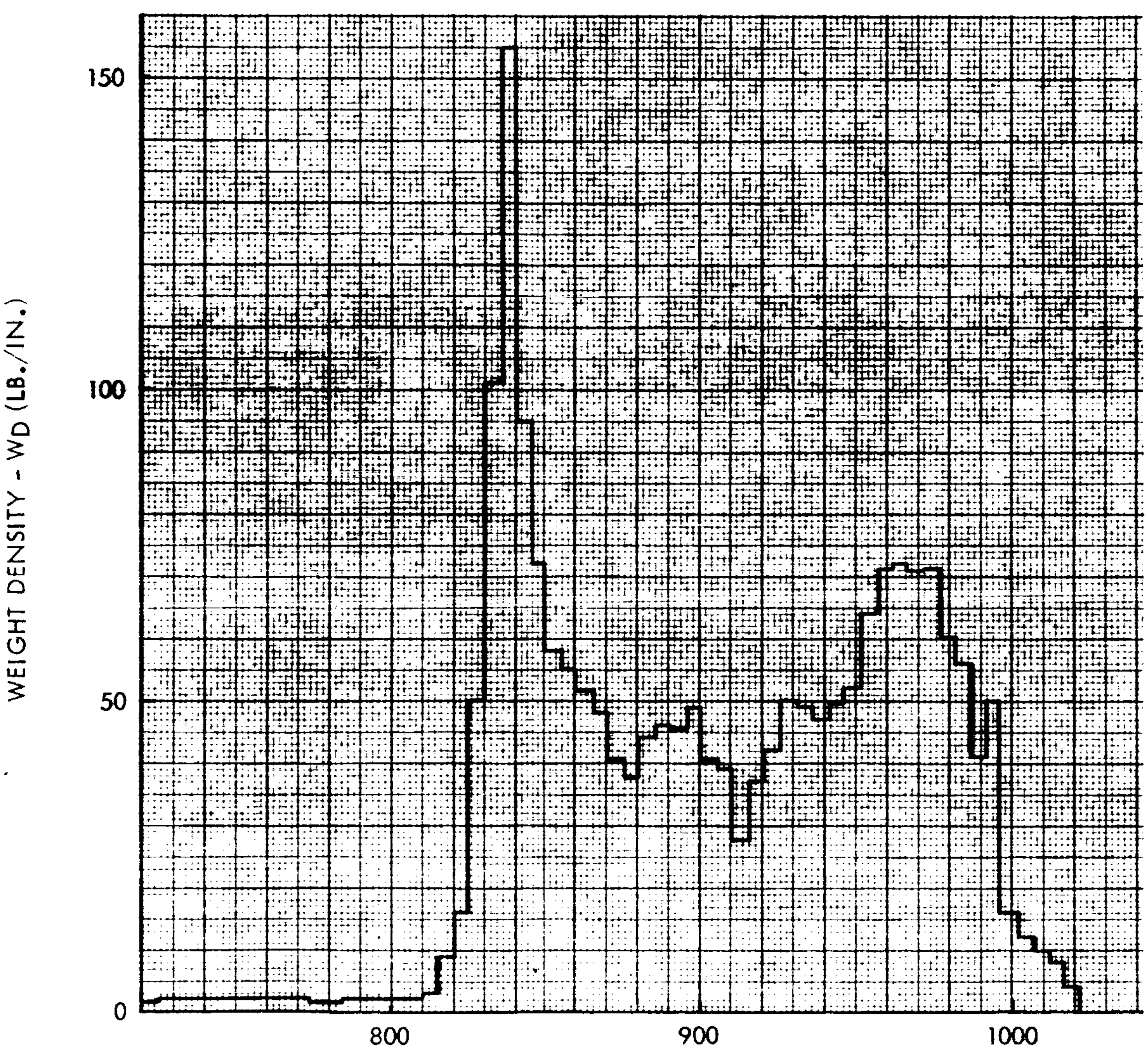


**Figure 67** Block II 155 In. SM (Without SPS Propellant)



[redacted]

CO[redacted]

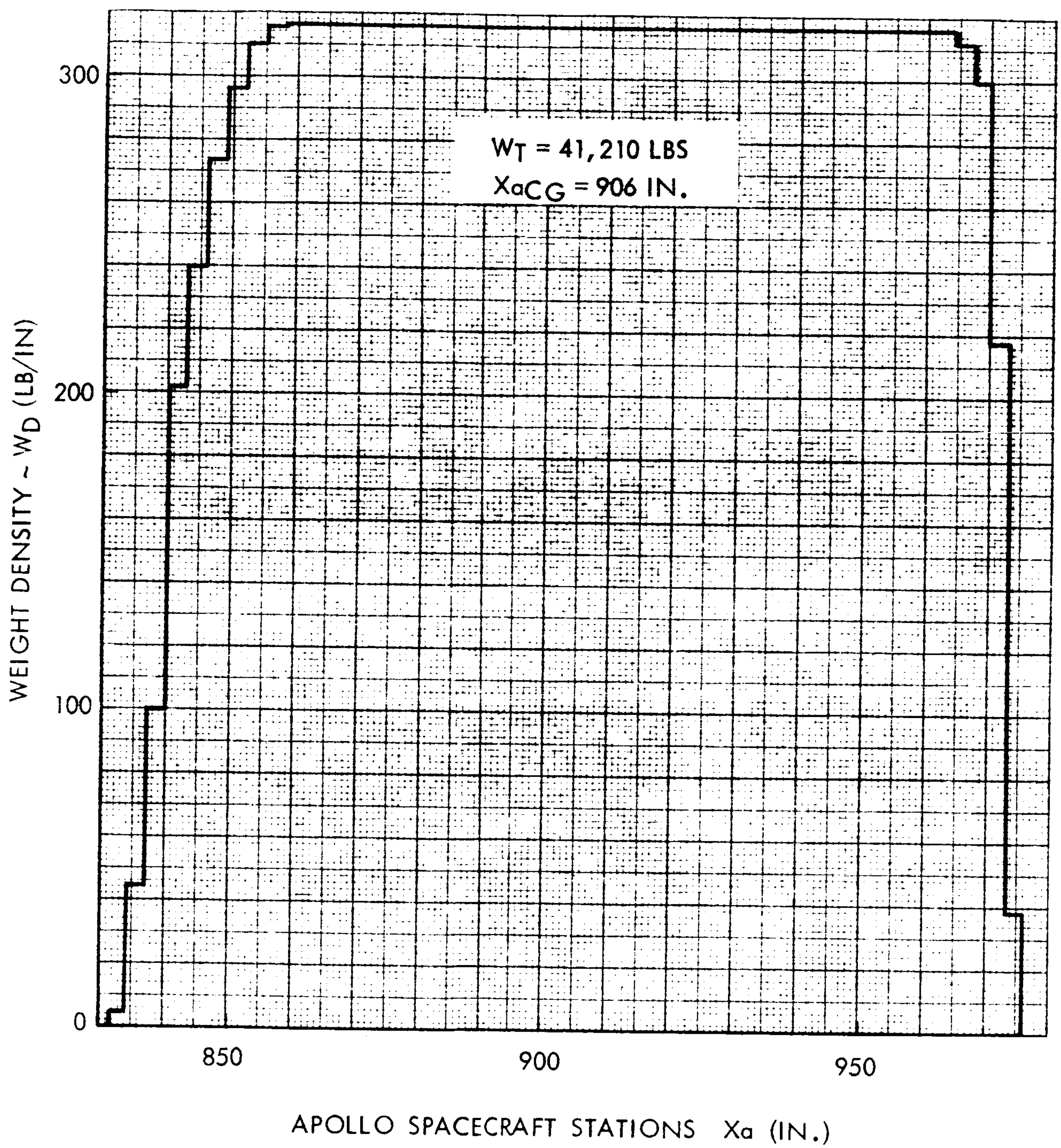


Figure 68 Block II Propellant 155 In. SM

CON[redacted]

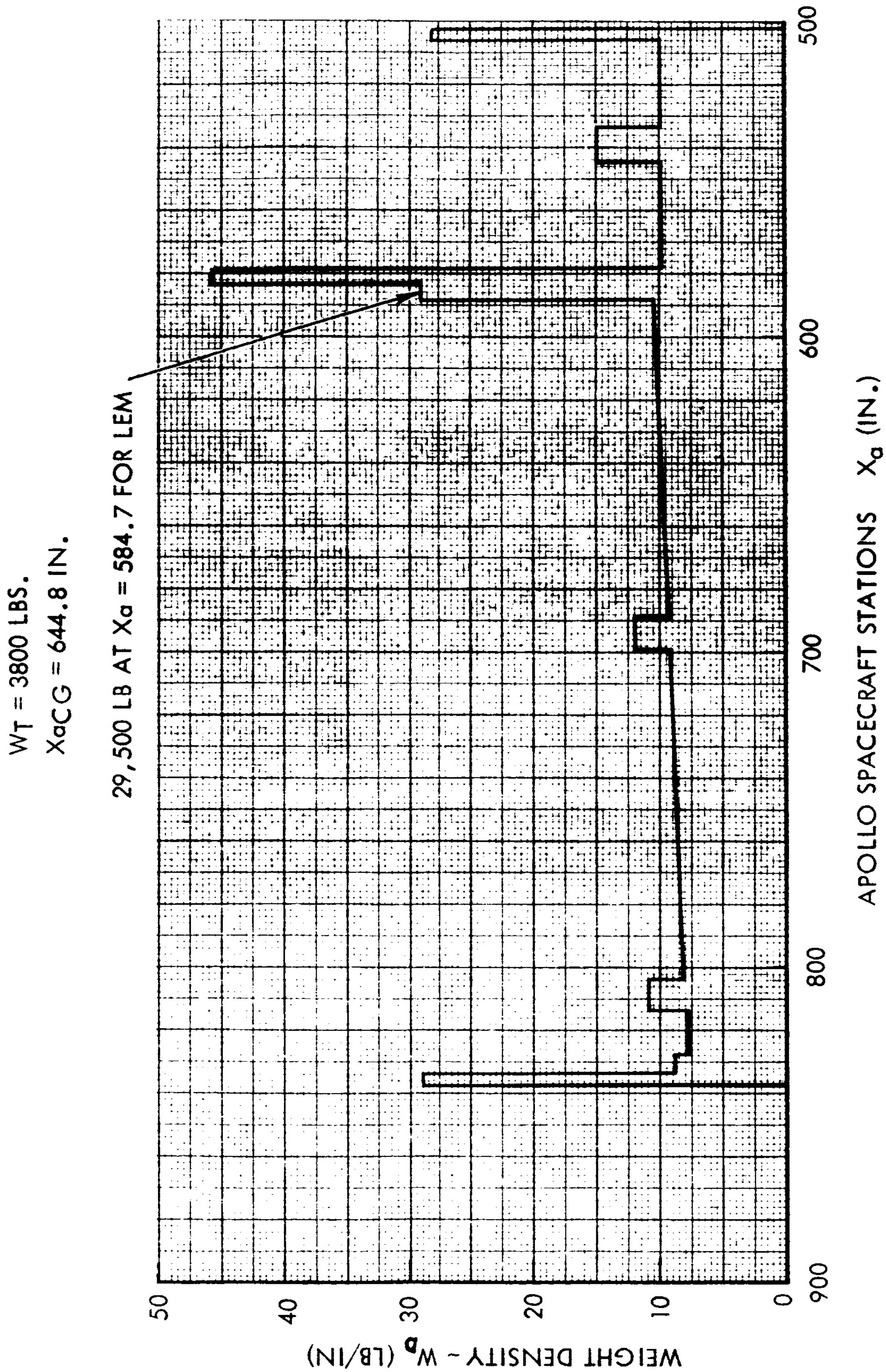


Figure 69 Block II Spacecraft LEM Adapter



[redacted]

CONF[redacted]

| COMPONENT | BREAK STATIONS (IN.) | LINEARIZED FOR 7° | | LINEARIZED FOR 8° | | LINEARIZED FOR 9° | |
|---|---|---|---|---|---|---|---|
| | | CNα PER RADIAN | C.P. (IN.) | CNα PER RADIAN | C.P. (IN.) | CNα PER RADIAN | C.P. (IN.) |
| LES NOSE CONE | 1490.0-1435.5 | .00951 | 1453.0 | .00933 | 1453.0 | .00933 | 1453.0 |
| KICKER MOTOR | 1435.5-1416.5 | .00190 | 1426.0 | .00181 | 1426.0 | .00173 | 1426.0 |
| JETTISON MOTOR | 1416.5-1367.5 | .00268 | 1394.6 | .00251 | 1394.6 | .00242 | 1394.6 |
| LES MOTOR | 1367.5-1221.5 | .00812 | 1269.0 | .00752 | 1269.0 | .00683 | 1269.0 |
| LES MOTOR SKIRT | 1221.5-1203.5 | .01132 | 1211.0 | .01046 | 1211.0 | .00985 | 1211.0 |
| LES TOWER | 1203.5-1083.5 | .01573 | 1120.0 | .01555 | 1120.0 | .01581 | 1120.0 |
| FWD HEATSHIELD & CREW COMP | 1083.5-1020.5 | .26539 | 1056.7 | .25597 | 1058.5 | .24241 | 1060.2 |
| SERVICE MODULE | 1020.5-838.0 | .06793 | 947.0 | .09610 | 945.0 | .12246 | 944.0 |
| FWD LEM ADAPTER | 838.0-584.7 | .26539 | 694.8 | .27110 | 694.0 | .27844 | 694.3 |
| AFT LEM ADAPTER | 584.7-502.0 | .17033 | 542.9 | .16636 | 543.5 | .16230 | 543.8 |

Figure 70 Spacecraft Normal Force Coefficient Derivative (CNα 1 Radian) and Center of Pressure (CP In.) $S_{Ref}$ = 855.3 $Ft^2$



[redacted]

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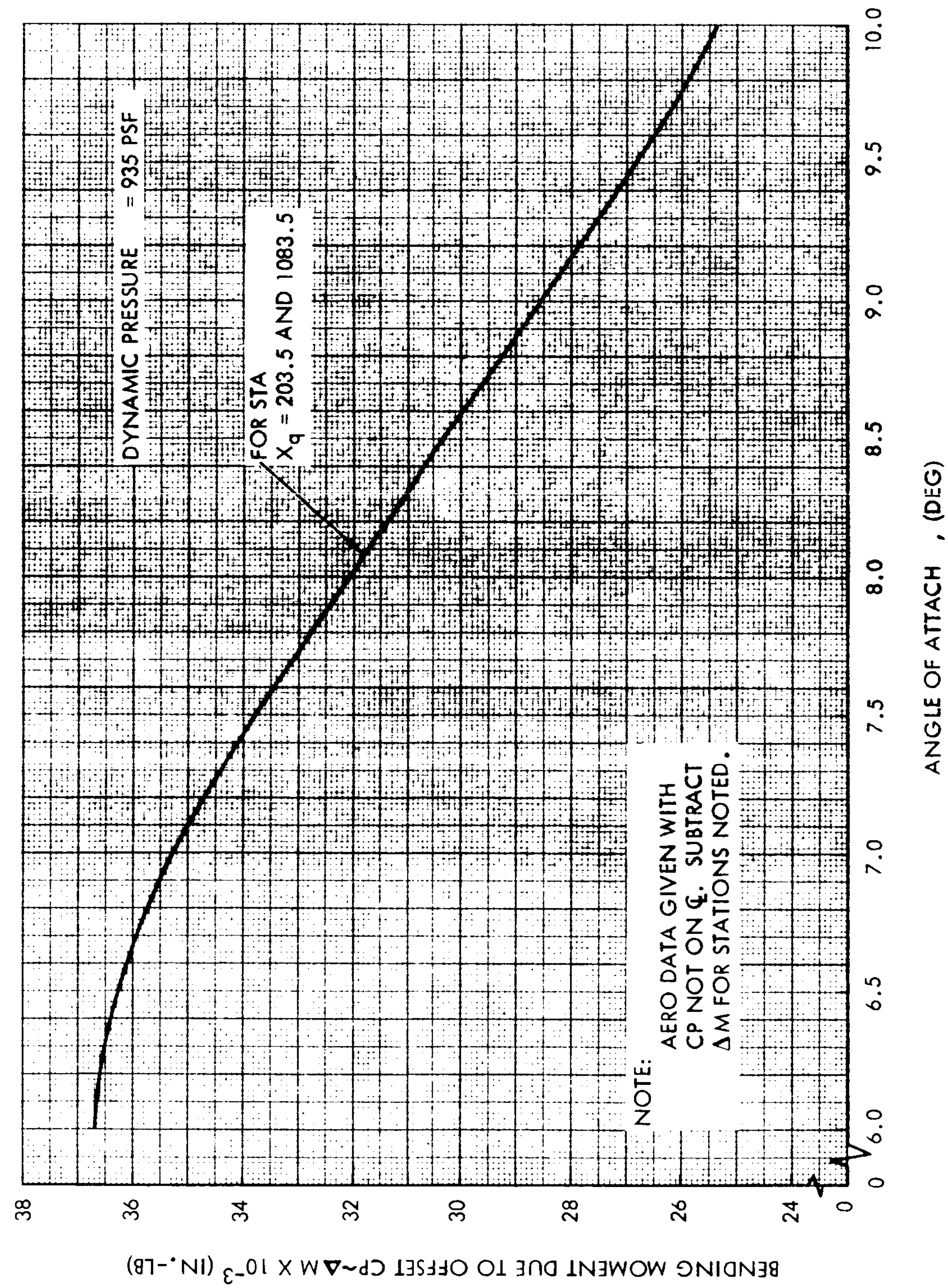


Figure 71 Bending Moment Correction for Offset CP



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CON[redacted]

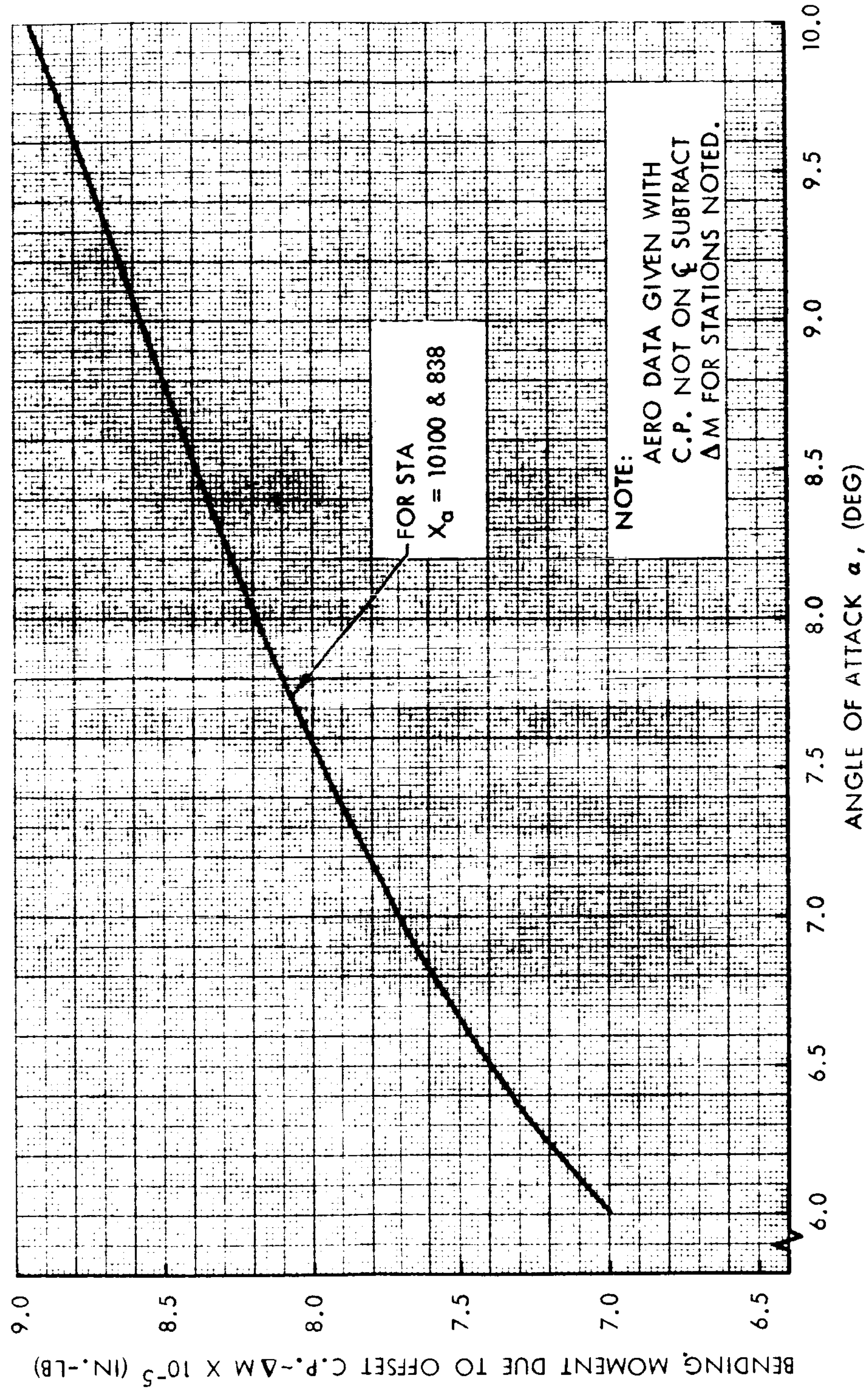


Figure 72 Bending Moment Correction for Offset CP



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C[redacted]

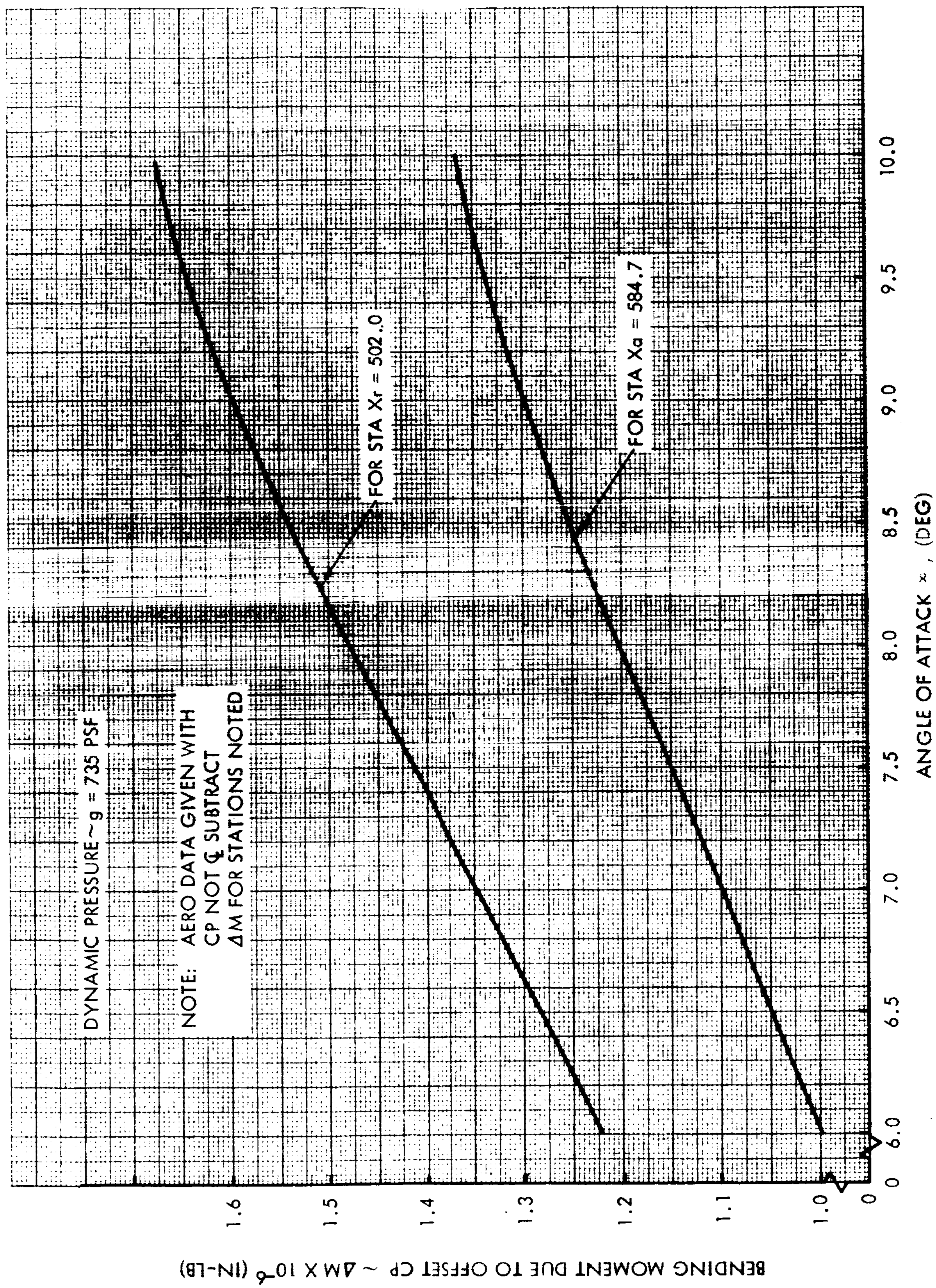


Figure 73 Bending Moment Correction for Offset CP



[redacted]

CON[redacted]

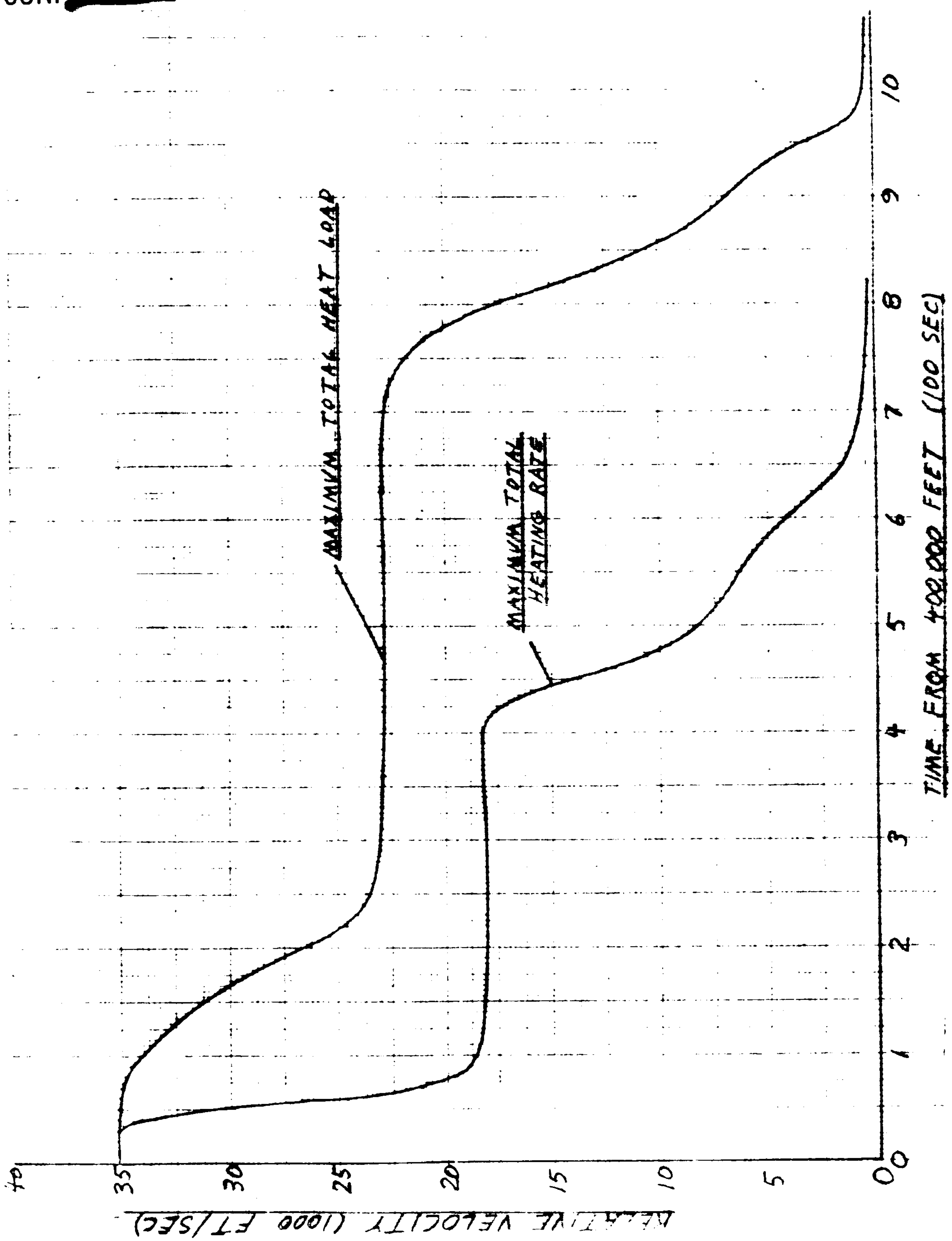


Figure 74. (Sheet 1 of 2) Entry Flight Phase Design Trajectories (NASA Furnished)

CON[redacted]

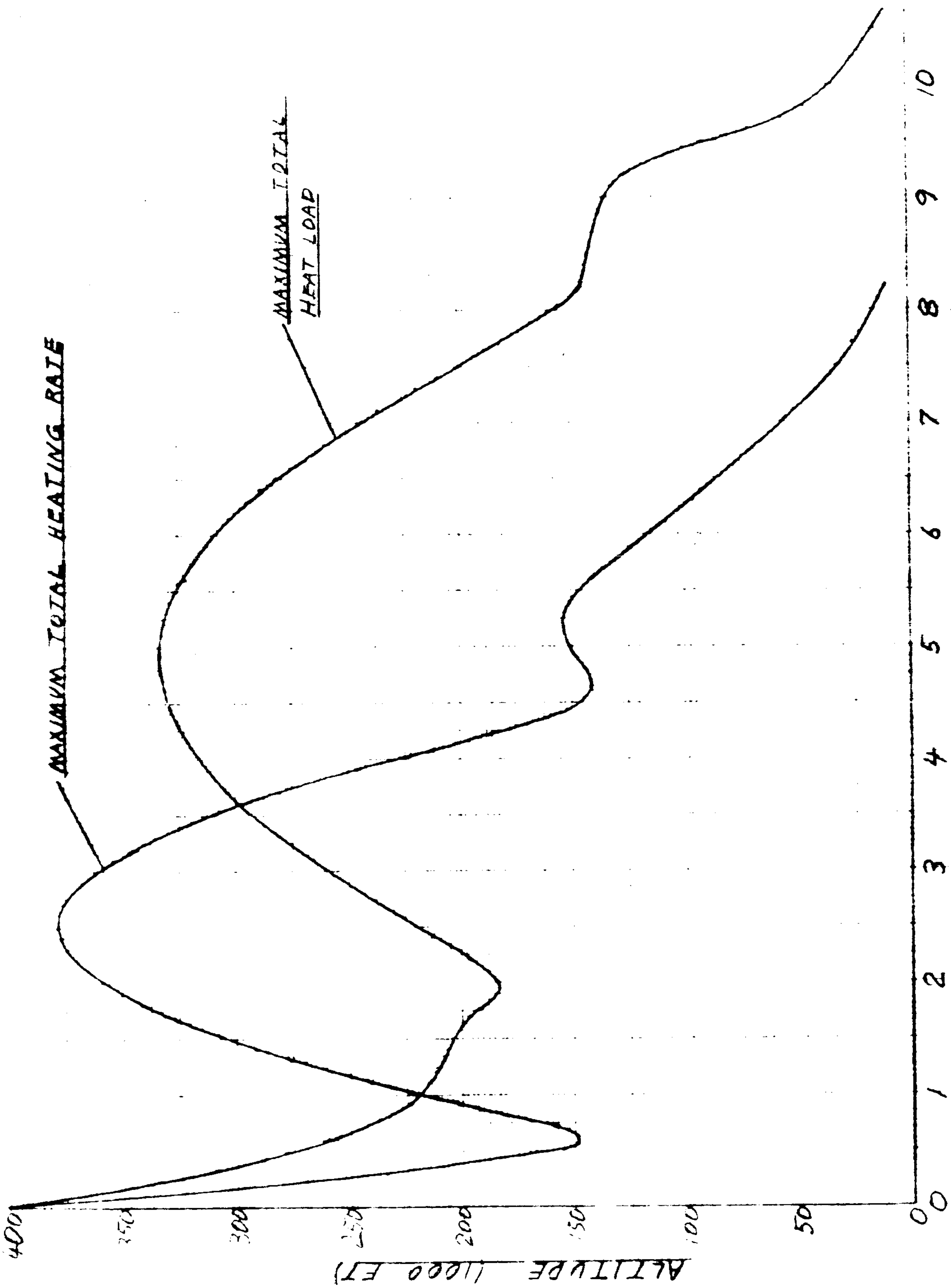


Figure 74. (Sheet 2 of 2) Entry Flight Phase Design Trajectories (NASA Furnished)



[redacted]

CO[redacted]

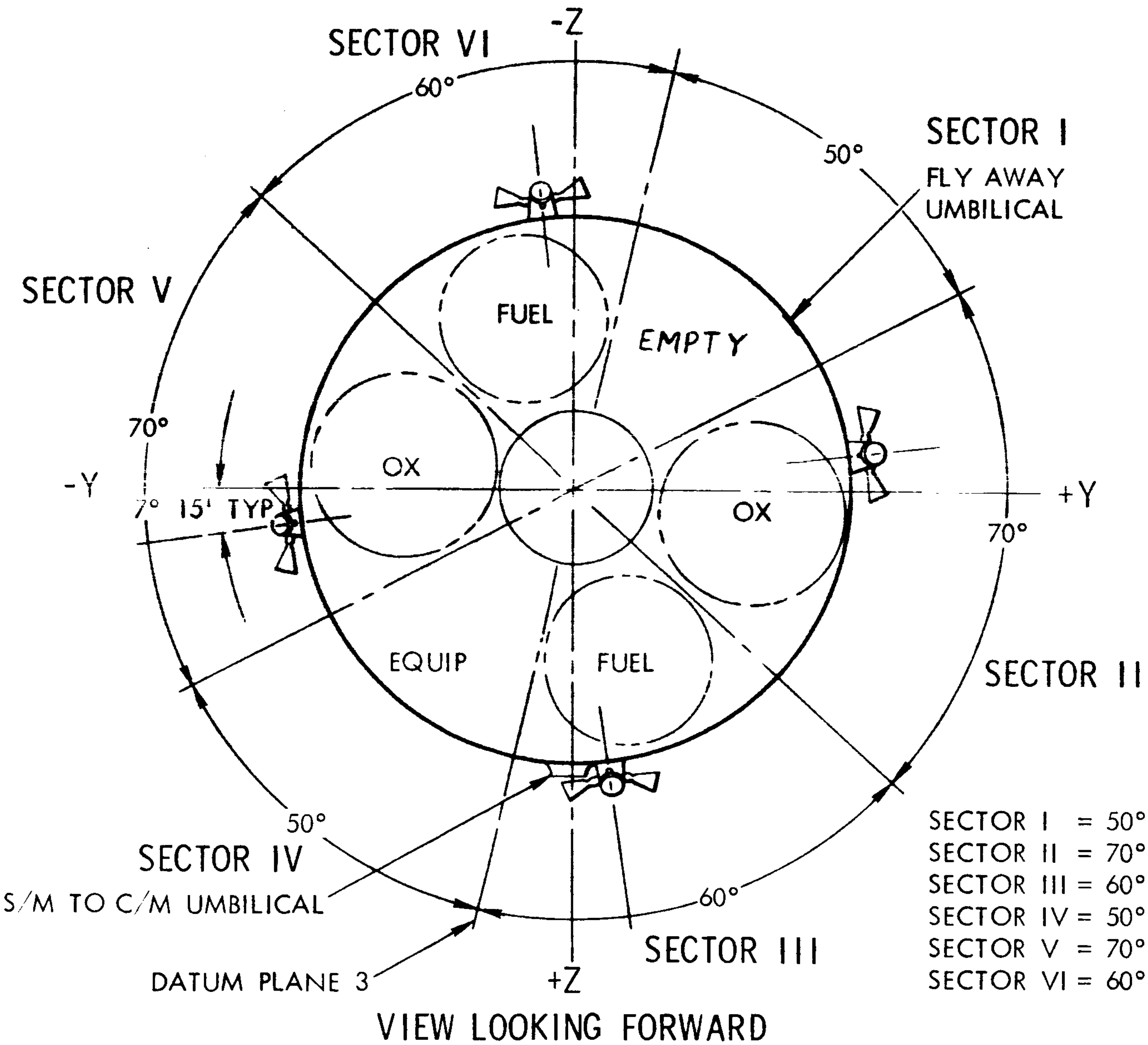


Figure 75 Service Module Inboard Profile-View Looking Forward



[redacted]

CON[redacted]

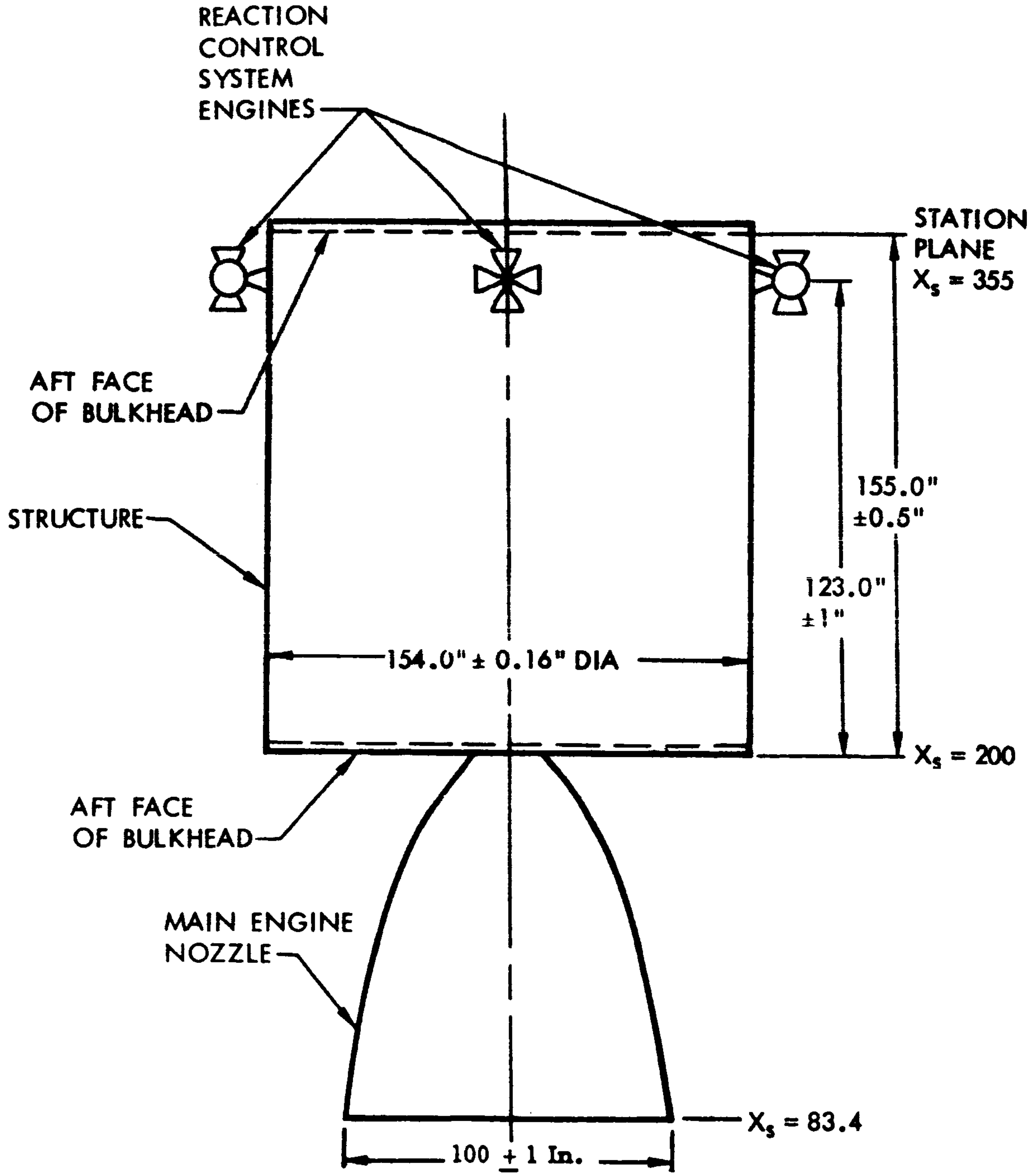


Figure 76   SM External Geometry

[redacted]

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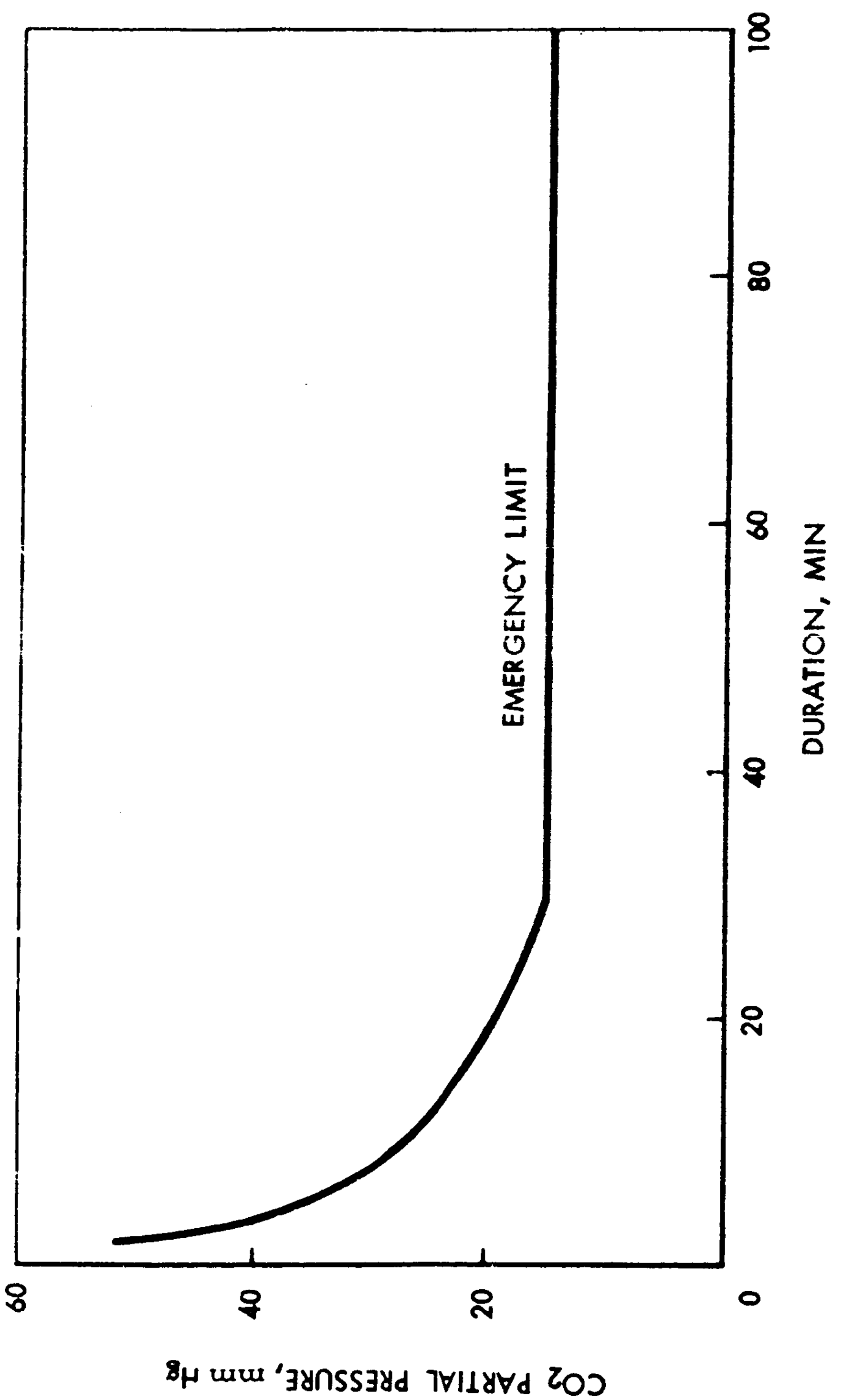


Figure 77 Emergency Carbon Dioxide Limit



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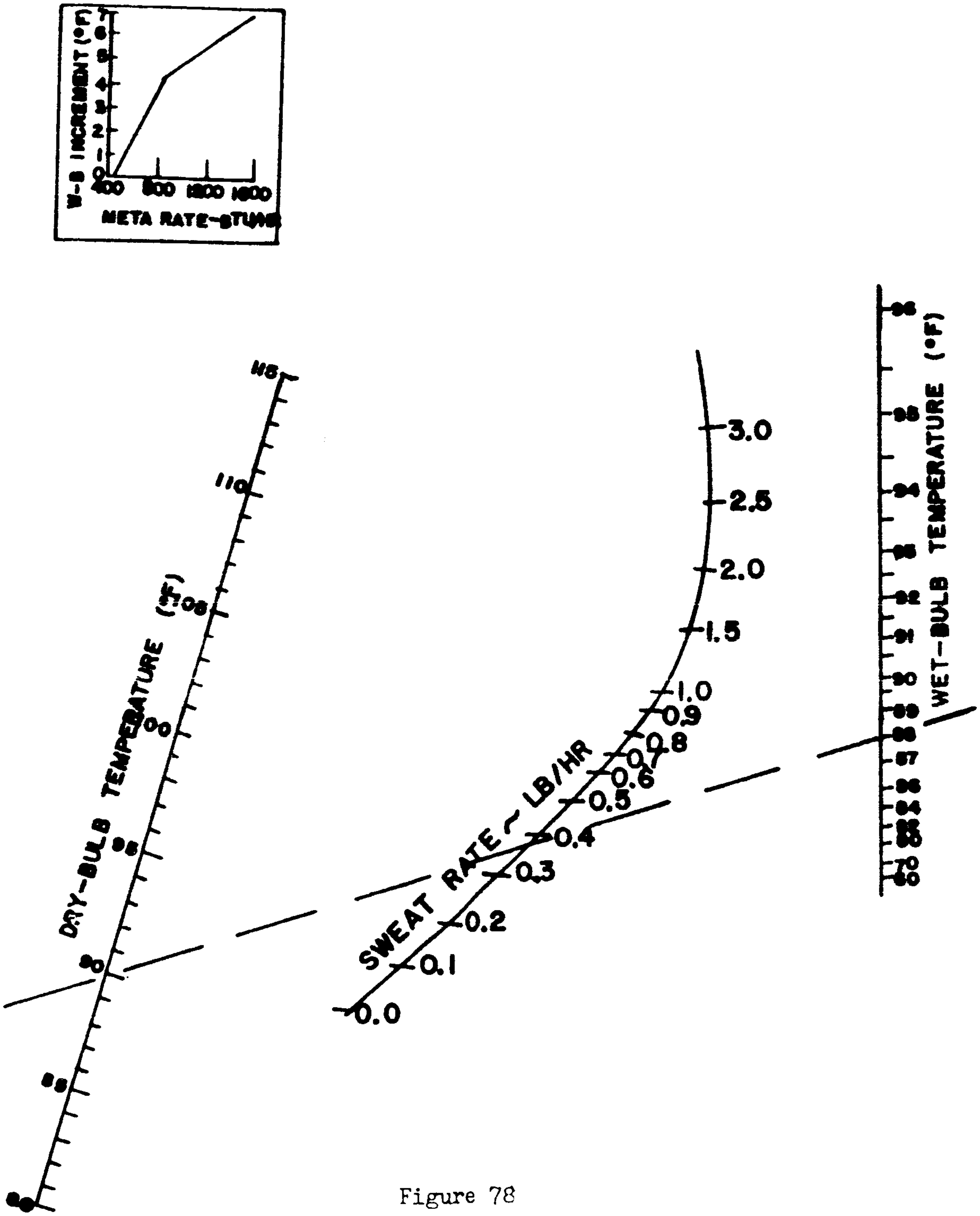


Figure 78

**NOMOGRAM FOR THE CALCULATION OF SWEAT RATE**
**(THE INSET CHART GIVES THE INCREMENT TO BE ADDED TO THE WET-BULB TEMPERATURE FOR METABOLIC RATES BETWEEN 400 AND 1600 BTU/HR.)**

CO[redacted]

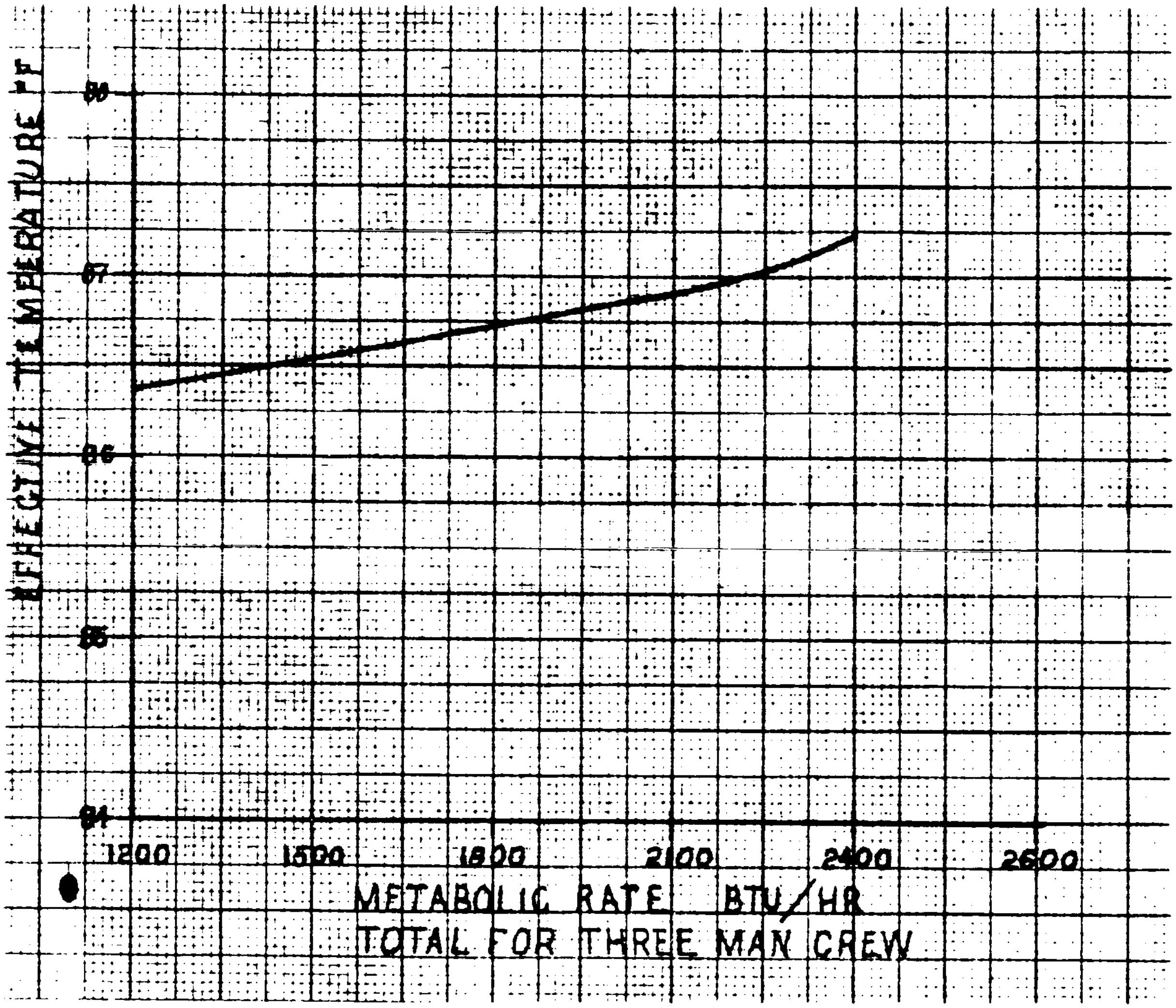


Figure 79 Post Landing ECS, Allowable Effective Temperatures

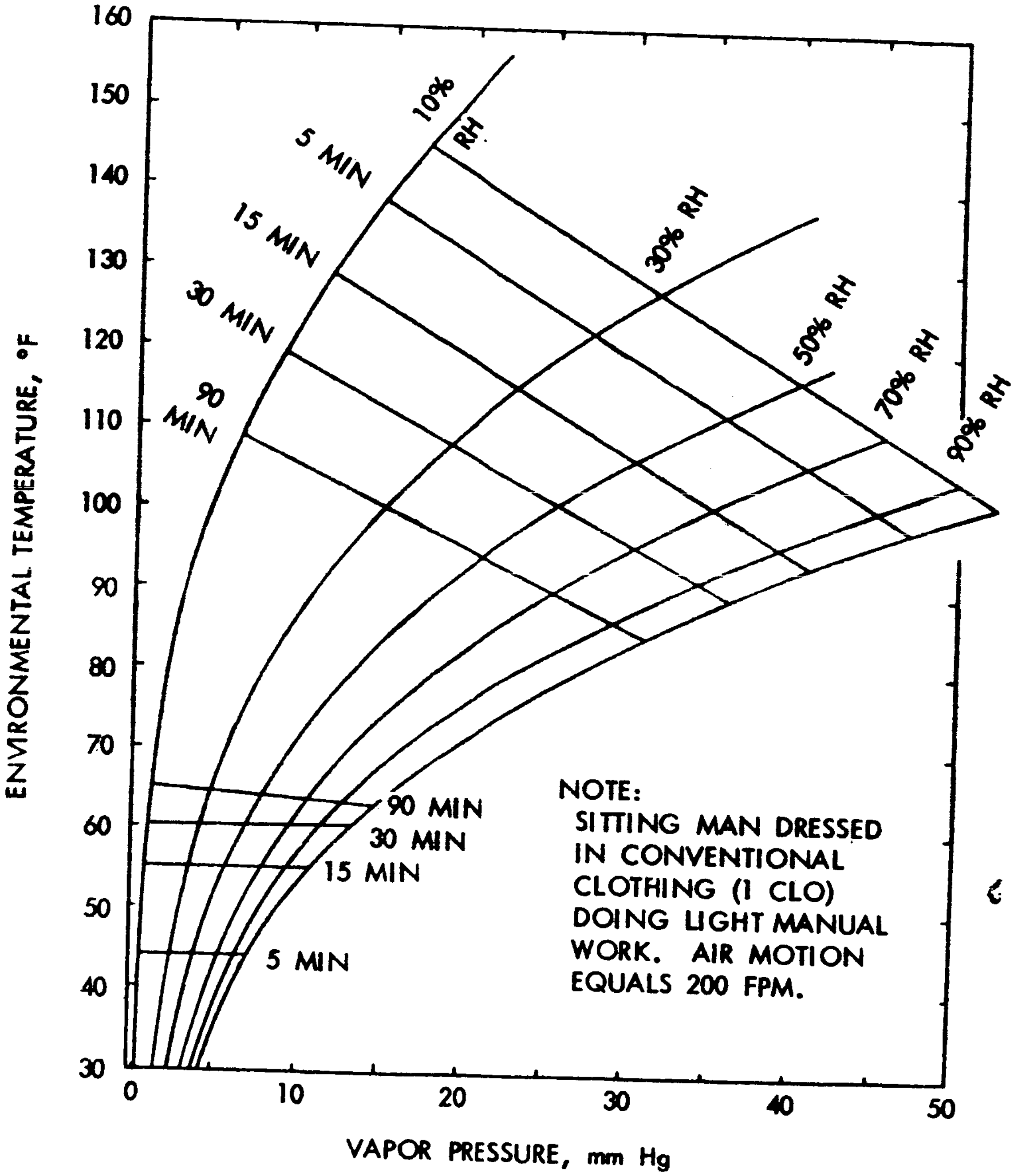


Figure 80 Temperature and Humidity Nominal Limit

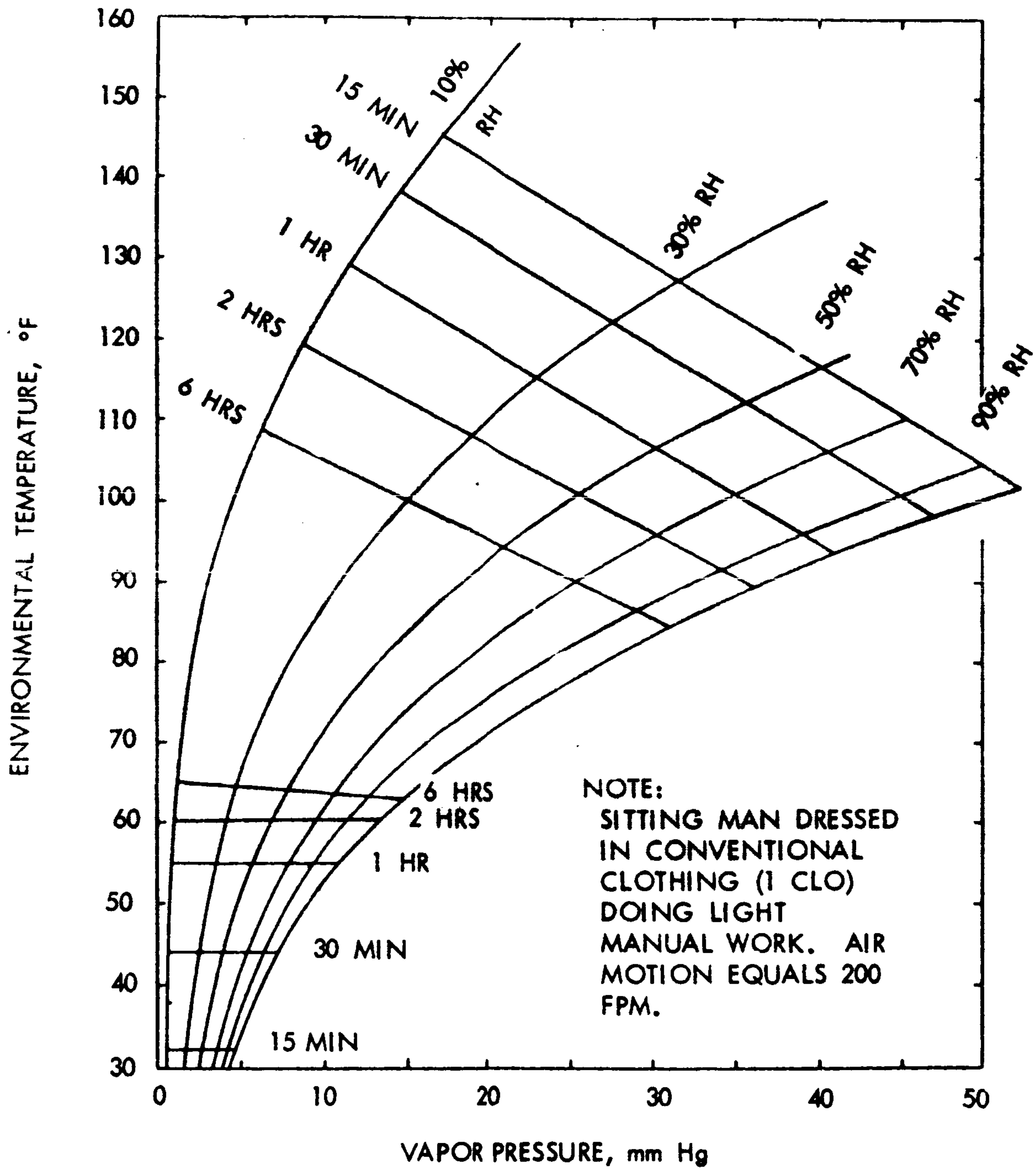


Figure 81 Temperature and Humidity Emergency Limit

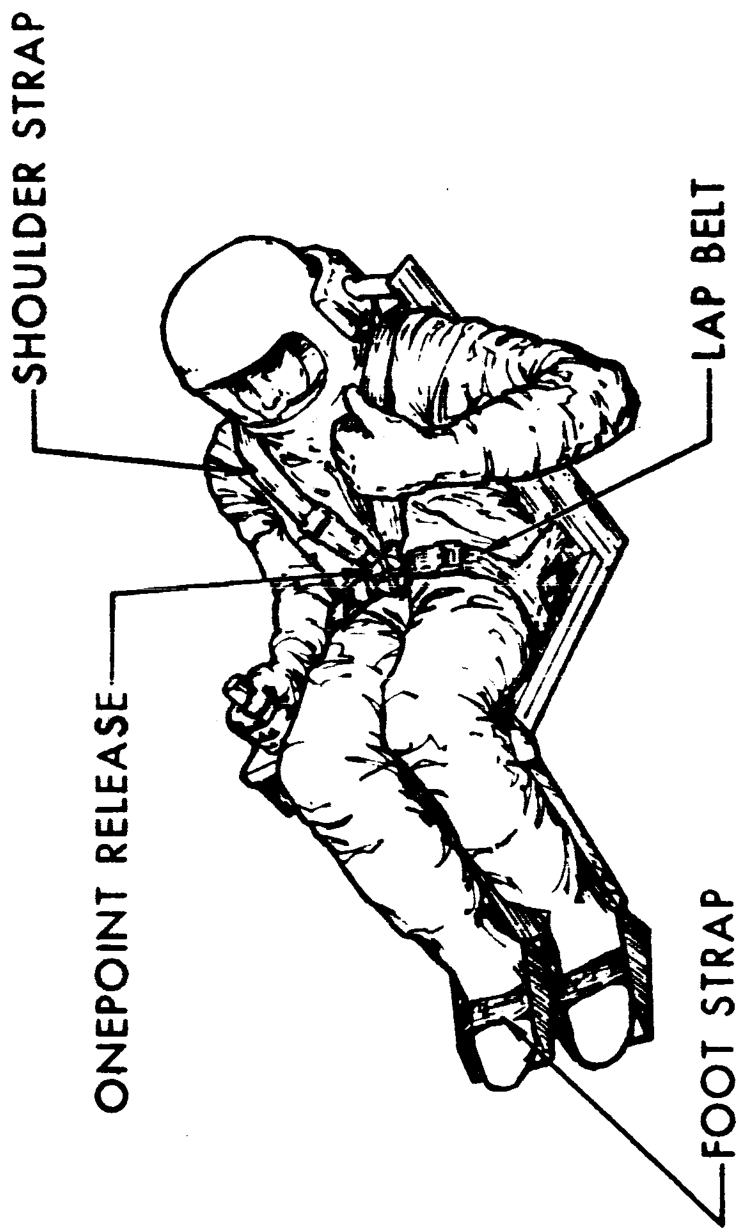


Figure 82. Restraint Harness

[redacted]

Intentional Blank
(Data TBD)

Figure 83 Apollo Normal Mission Impact Limits

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Intentional Blank
(Data TBD)

Figure 84 Apollo Emergency Impace Limits

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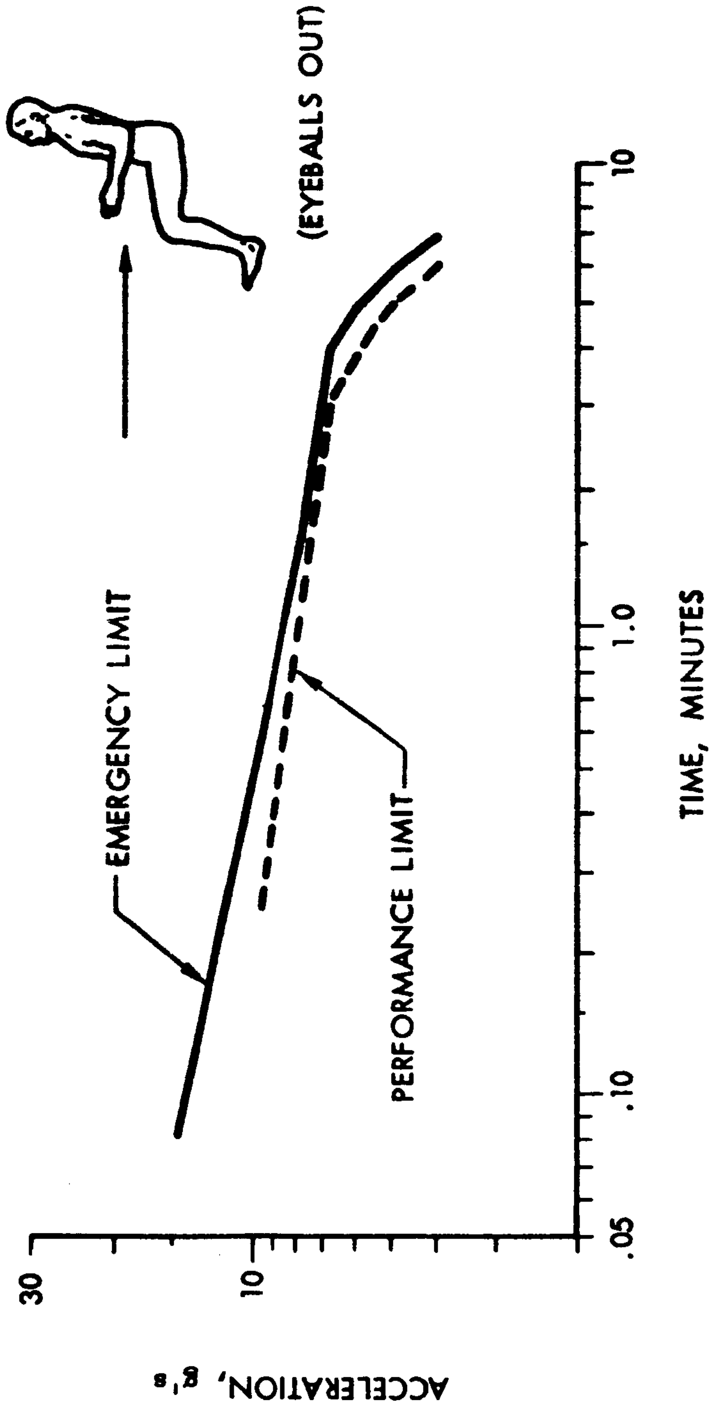


Figure 85 Sustained Acceleration - Eyeballs Out

~~CONFIDENTIAL~~

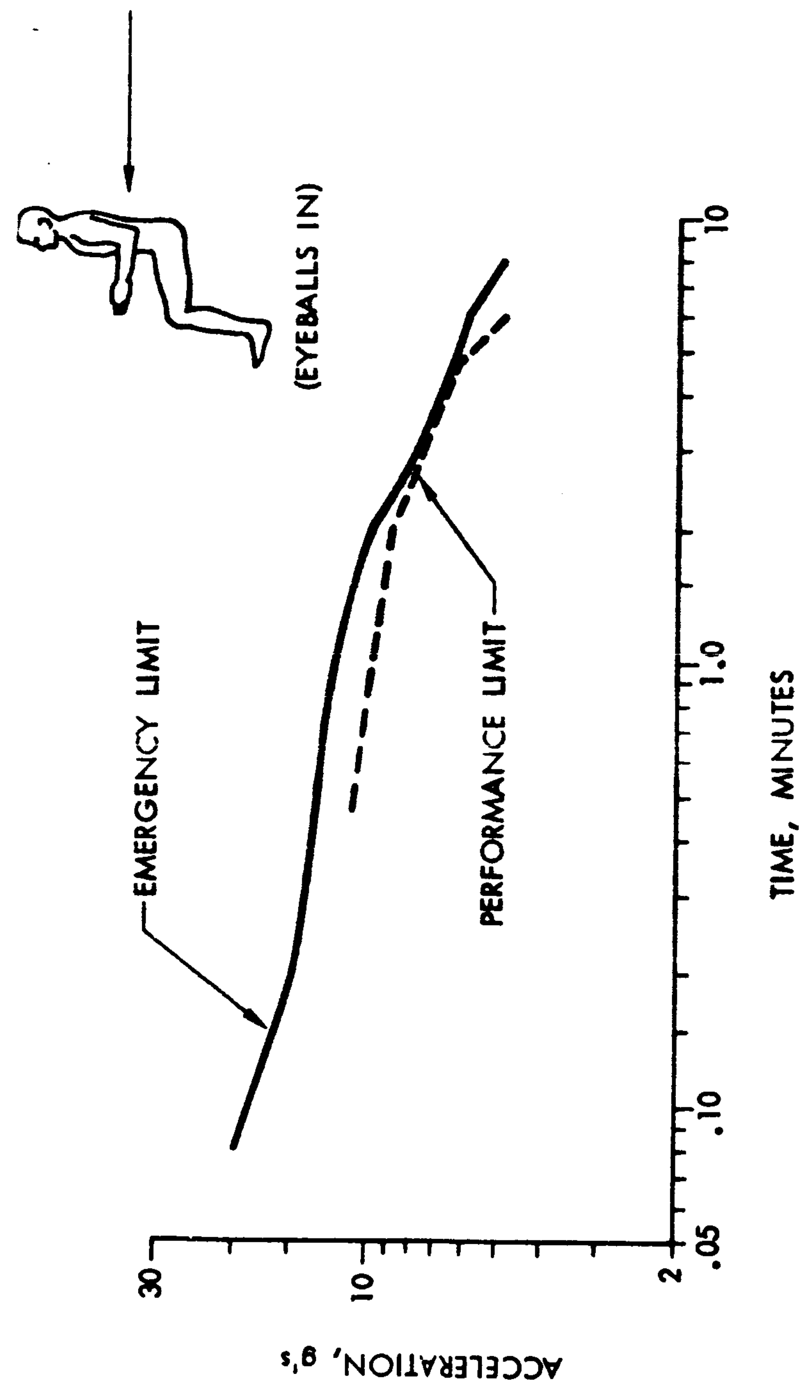


Figure 86 Sustained Acceleration - Eyeballs In

CONFI[redacted]

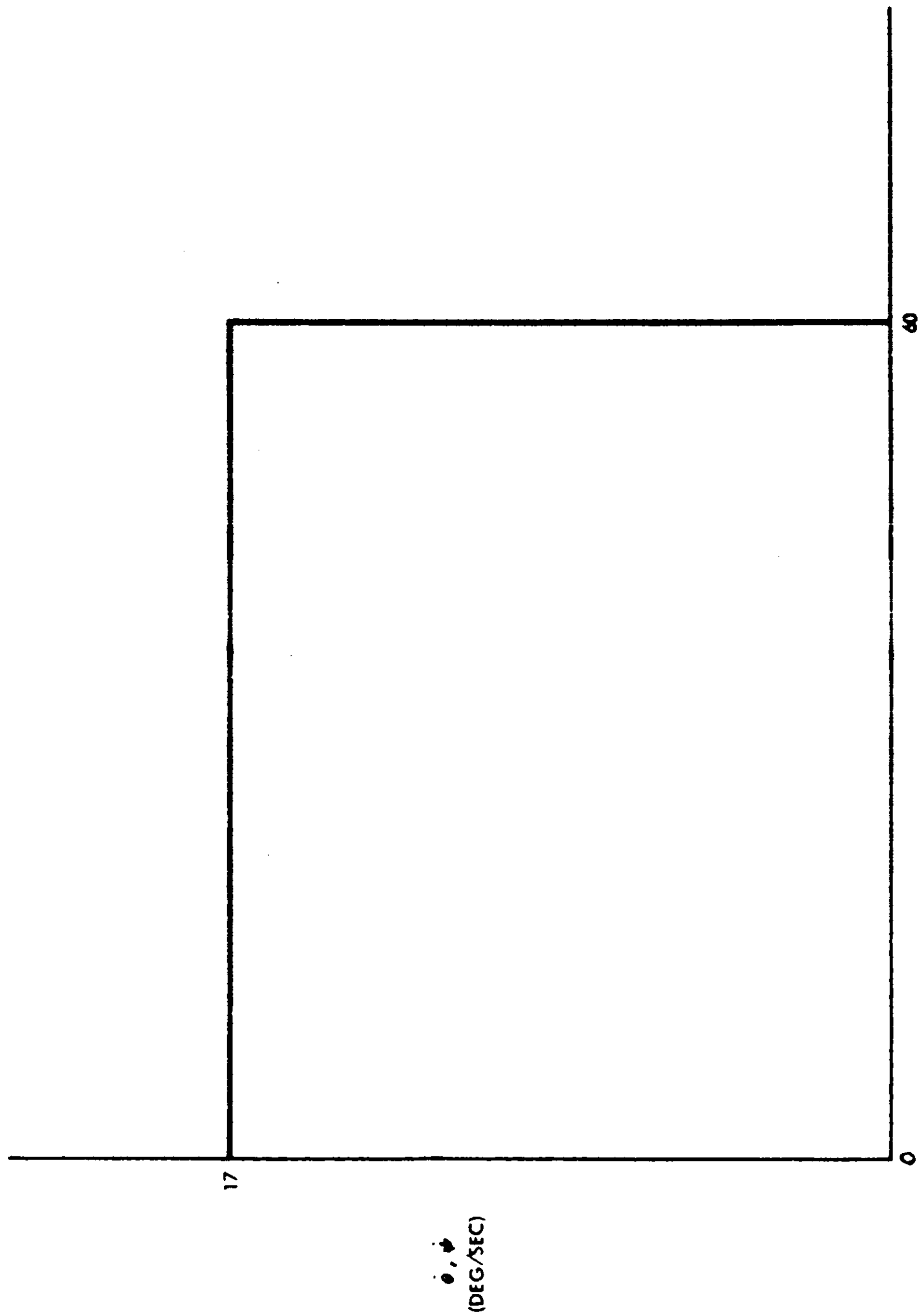


Figure 87 Attitude and Attitude Rate



[redacted]ENTIAL

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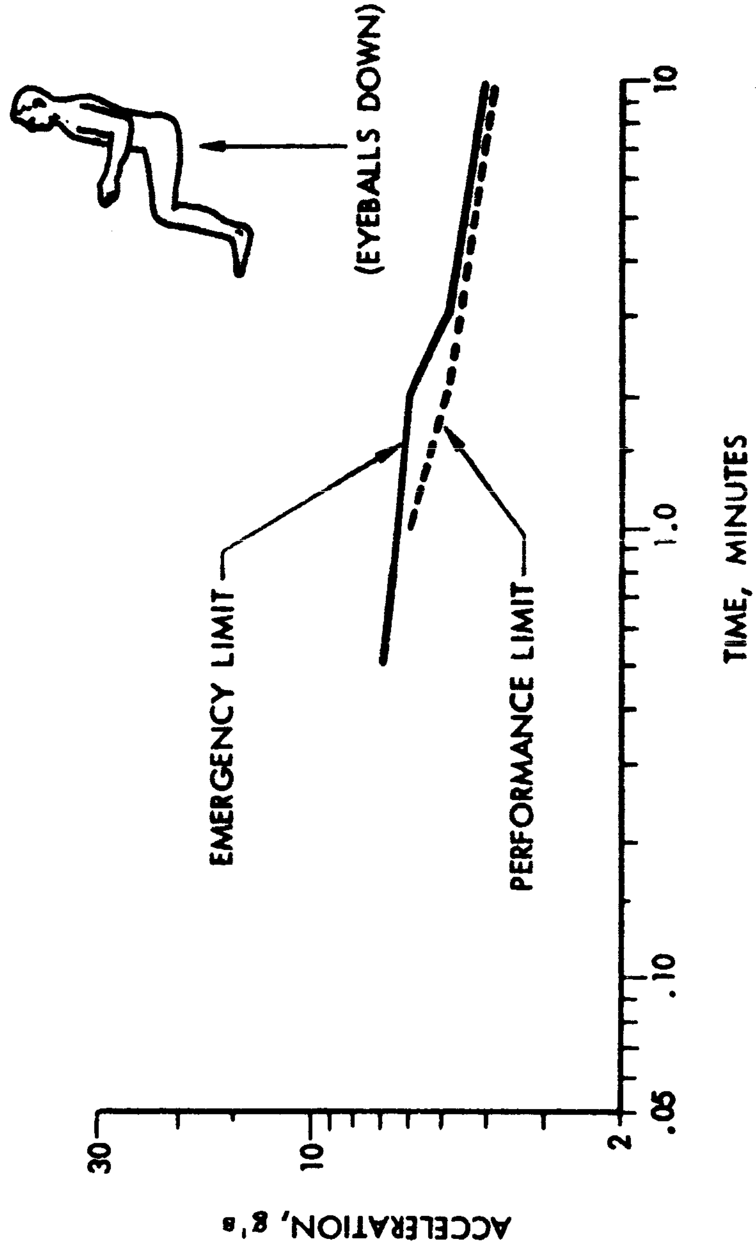


Figure 88 **Sustained Acceleration - Eyeballs Down**

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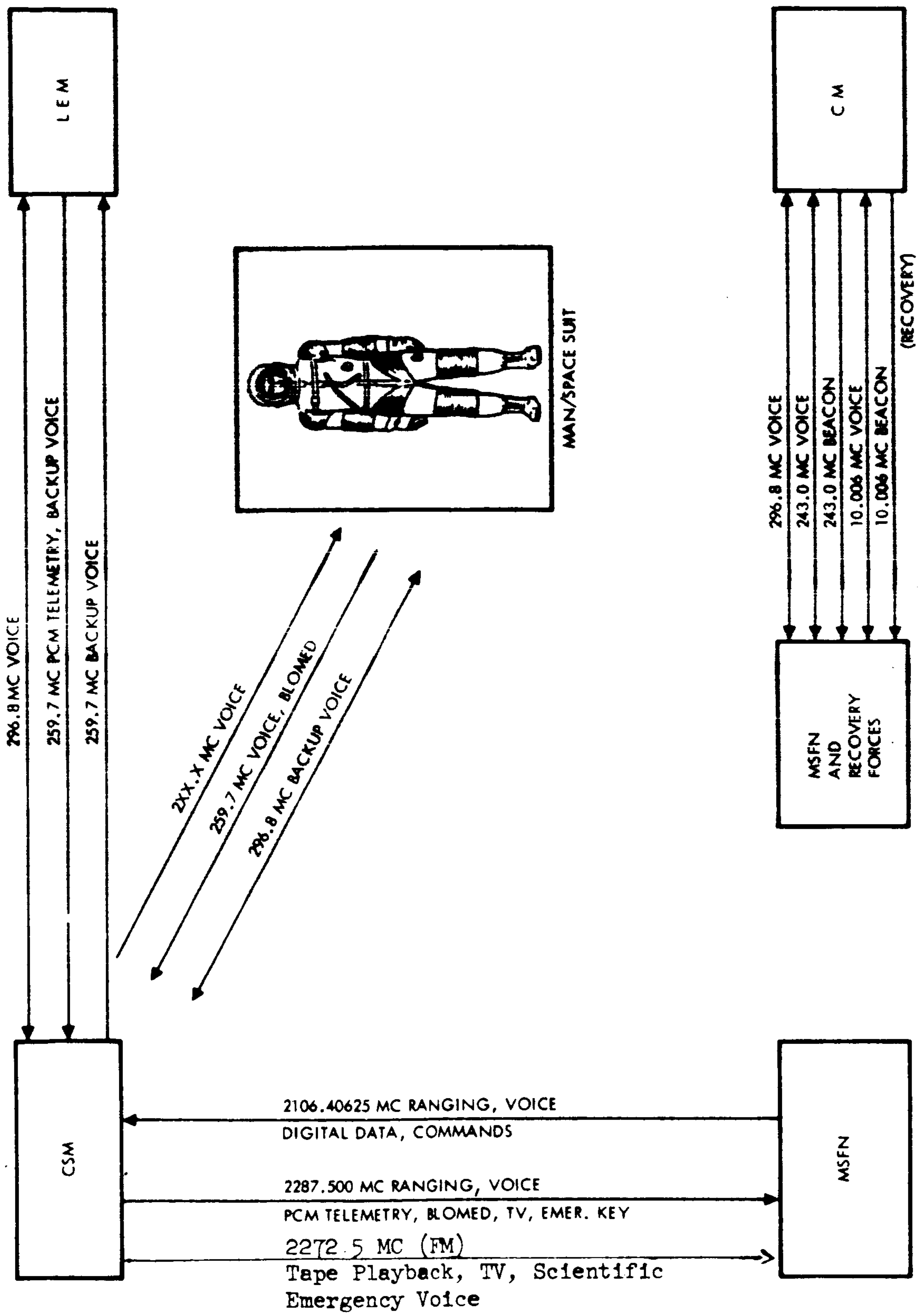


Figure 99. Communication Lines and Frequencies



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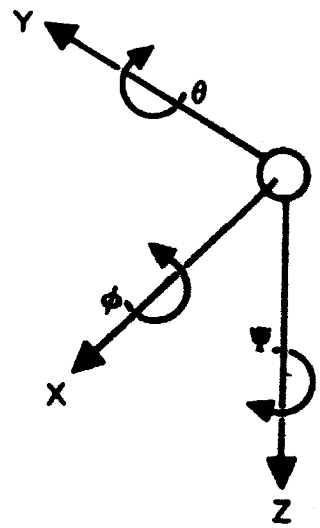
Table 1. Reference Axes

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 | $\phi$ | U | p |
| Y | Pitch | $\theta$ | V | q |
| Z | Yaw | $\psi$ | W | r |

## Table 2. Metabolic Rate, Thermal Balance and Water Requirements

| 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 |

The above values are based on duty cycles for each crewman as follows: Routine flignt - 8 hours per day in pressure suit and balance unsuited. During the 8-hour suited mode 125 Btu's per hour of heat convected to cabin air.

Emergency depression - Each crewman in pressure suit continuously with no heat loss to cabin.