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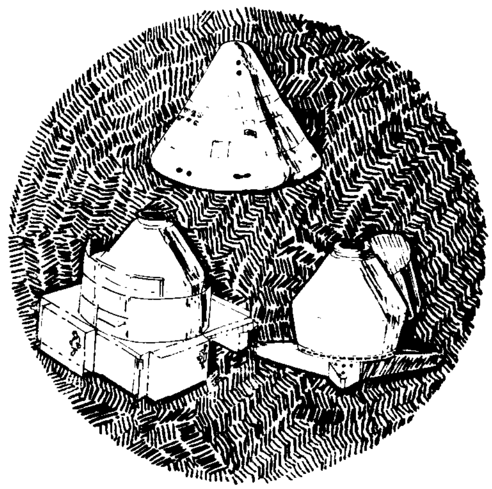

STUDY OF A

Renovated Command Module Laboratory and Renovated Command Module

Contract No. NAS9-6445

FINAL REPORT

Volume II

Mission System Performance and Configuration Analysis

NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION

NASA CR 92156

CONTRACT NAS9-6445

SID 66-1853-2

STUDY OF A RENOVATED COMMAND MODULE LABORATORY AND RENOVATED COMMAND MODULE

Final Report

VOLUME II - MISSION SYSTEM PERFORMANCE AND CONFIGURATION ANALYSIS

15 DECEMBER 1966

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FOREWORD

This document is submitted by the Space and Information Systems Division of North American Aviation, Inc., to the National Aeronautics and Space Administration Manned Spacecraft Center in partial fulfillment of the final reporting requirements of Contract NAS 9-6445, "Study of a Renovated Command Module Laboratory and Renovated Command Module."

The final report has been prepared in a series of five volumes as listed below.

Volume I	Summary	SID 66-1853-1
Volume II	Mission System Performance and Configuration Analysis	SID 66-1853-2
Volume III	Subsystems Analysis	SID 66-1853-3
Volume IV	Resources Requirements Analysis	SID 66-1853-4
Volume V	Cost Analysis (Limited Access)	SID 66-1853-5

S&ID acknowledges the voluntary technical contributions made to this study by a number of companies. The Avco Corporation contributed ablator data which were used as a basis for determining the feasibility of heat shield renovation. A report covering the data provided by Avco is included as an appendix to Volume III.

A. C. Electronics Division of General Motors Corporation supplied data on technical problems associated with renovating the Apollo G&N system and estimated costs.

The Defense Programs Division of General Electric Company provided characteristic data on G.E.'s active space pointing systems.

Westinghouse Electric Corporation provided data on rendezvous radar and transponder characteristics.

The Aeronautical Division of Honeywell, Inc., provided renovation data on the Apollo Block II stabilization and control system and associated costs.

The Autonetics Division of North American Aviation, Inc., provided data on an alternative guidance and navigation system and estimated costs.

Cost information and general renovation requirements on individual components were also provided by numerous other suppliers.

CONTENTS

ILLUSTRATIONS

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TABLES

I. INTRODUCTION

The Apollo Application Program (AAP) currently being defined by NASA will provide capability for lunar orbital survey missions, and extended earth-orbital missions by utilization of hardware, equipment, and technology developed for the basic Apollo lunar mission. Other NASA studies have been conducted to evaluate the spectrum of AAP mission objectives and establish potential experiment groupings. Also being investigated are preliminary designs of the spacecraft configurations capable of significant contributions to extended manned space flight operation and scientific space exploration beyond present goals. In most of these studies, emphasis has been placed on identification of probable AAP experiment configurations and the assessment of man's capability in conducting useful operations in the space environment in support of the experimental mission objectives.

In contrast to prior studies, this study, "Concept/Feasibility Study of the RCM Laboratory and the RCM Spacecraft" had the specific objective of providing a conceptual definition of a renovated command module (RCM) laboratory and a feasibility analysis of an RCM spacecraft for operational use in the AAP and follow-on programs. The possible AAP missions to be included in the study are represented by both the earth and lunar orbital operations; the physical integration of individual AAP experiments into either the RCM laboratory or RCM spacecraft was not within the scope of this study.

Earlier S&ID independent research and development studies relative to an Apollo multiple-mission laboratory indicated the potential for using the Apollo command module pressure shell and Block II subsystems as the basis for a lunar surface laboratory. Further investigations of this laboratory concept have shown its applicability to earth and lunar orbital missions. The series of experimental missions and flight schedules identified in the Apollo Extension Systems utilization studies formed the basis for these investigations. Early flights have shown little need for independent laboratory systems operation and subsystem requirements are therefore minimal on these flights. The relationship between increased mission experimental support requirements and subsystem capabilities suggested that progressive laboratory system development through addition of subsystem "building blocks" to a basic laboratory can accomplish the desired program objectives. Following this system concept, the AAP missions could be accomplished through progressive development, as have other manned space flight programs, with the degree of complexity increasing in an orderly progression of steps.

This study accomplishes the transition from the early Apollo multi-mission laboratory to the RCM laboratory and RCM spacecraft concepts, which use renovated CM structure and subsystems in their development. Results and data derived in the course of the study include all factors necessary to assist NASA in defining spacecraft system configurations and designs required for development and construction of RCM laboratories and RCM spacecraft from Apollo CM's which have completed their primary missions. They also define system configuration designs for use in the AAP program.

APPROACH

The overall technical approach of the study is directed to accomplish the objectives associated with the conceptual definition of the RCM laboratory and the feasibility analysis of the RCM spacecraft. The overall objective of this study is to conduct sufficient basic analyses to accomplish the following:

1. Establish feasibility of renovating and converting used CM's into laboratories and operational CM's for use in the AAP.

2. Identify the basic renovation tasks required.

3. Determine availability of CM's that have completed their primary mission and the time required for renovation and conversion.

4. Provide cost, schedule, and technical information in "building block" form to permit selection by NASA of CM laboratory configurations for future study.

5. Provide cost, schedule, and technical information on renovation and modification required to convert used CM's into renovated CM's.

These objectives are adapted to the technical approach applicable to the "Spacecraft Engineering" portion of the study and are reflected in the investigation, analyses, and design of the RCM laboratory and RCM spacecraft.

Investigations and analyses were performed to determine those modifications and additions necessary to convert both a Block I and Block II CM's into a basic RCM laboratory. By definition, the basic RCM laboratory represents a dependent laboratory capable of operating in conjunction with a CSM. As such, the basic RCM laboratory does not contain any active subsystems and is therefore dependent upon the CSM to which it is docked for active environmental control, electrical power, attitude control, and communication. This basic RCM laboratory can accept logical addition of subsystem functions and capabilities in incremental steps until it reaches the fully independent RCM laboratory configuration, capable of independent operation and full support of experimental requirements. Sufficient volume for installation of experiments inside the laboratory or on the external mounting structure is available for all RCM laboratory configurations.

The candidate subsystems and subsystem incremental "building blocks" for installation in the basic RCM laboratory consist primarily of renovated CM subsystems, plus SM Block II subsystems, components, or other necessary alternates. As the subsystem incremenal "building blocks" are added and the system capabilities are increased, the resulting effects on mission time and experiment support capability is documented to provide a "shopping list" of incremental subsystem "building blocks." This conceptual study approach is illustrated in Figure 1.

The feasibility analysis of the RCM spacecraft consists of the investigation and analyses necessary to determine the required procedures for renovation and modification of used Block II CM's and Block II CM subsystems into operational CM's. The necessary renovation tasks and modifications of the Apollo Block II CM are identified and described in detail in the "Subsystem Analysis" volume of this report (Vol III). The study approach relative to the RCM spacecraft is illustrated in Figure 2.

The system engineering effort described in this volume reports the study results applicable to establishment of the design and performance requirements for the RCM laboratory, its subsystems, and the definition of laboratory system configuration concepts.

REQUIREMENTS AND CONSTRAINTS

The study requirements and constraints to be followed in investigations and analysis of the RCM laboratory and RCM spacecraft were defined in the contract Statement of Work as follows:

1. Maximum utilization of available spare parts and salvageable test and flight articles from the Apollo and subsequent programs.

2. Utilization of other existing developed and qualified spacecraft hardware.

3. Minimum cost approach for all aspects of program (modifications, salvage, and new items) commensurate with crew safety and mission success.

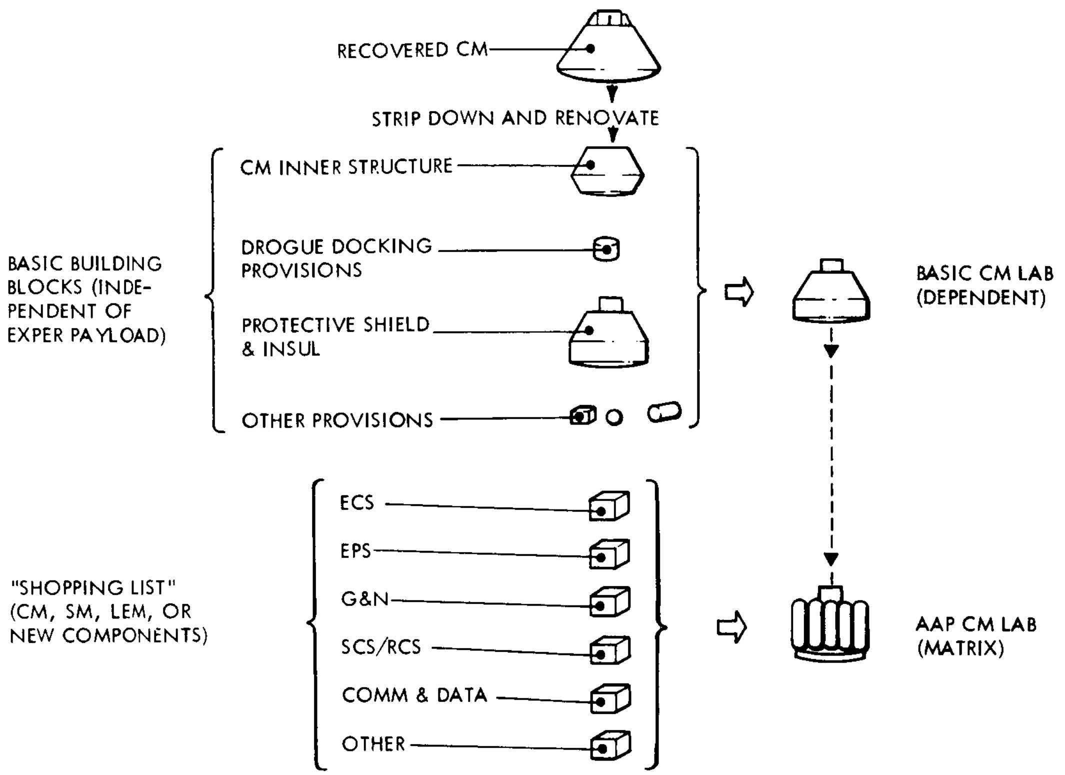

Figure 1. Renovated Command Module Laboratory Concept

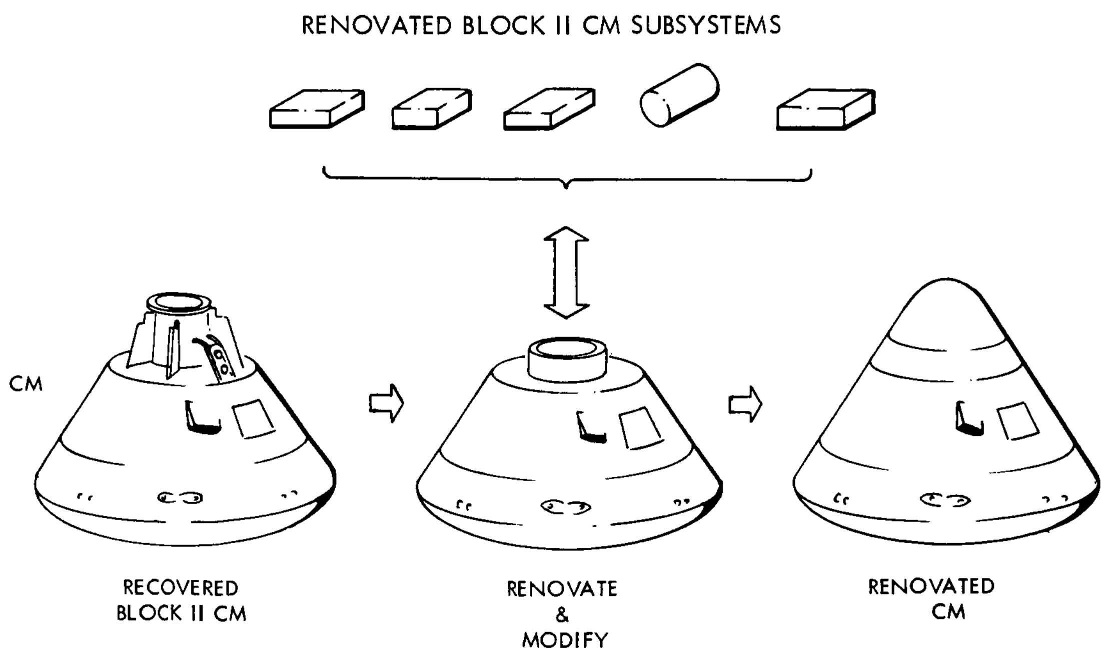

Figure 2. Renovated Command Module Concept

4. Maximum use of and coordination with existing development programs and studies.

5. Minimum modifications to ground support equipment (G&N), Automatic Checkout Equipment (ACE), and the Manned Space Flight Network (MSFN).

6. Time requirements for AAP CM flight articles to be based on availability of CM's from Apollo and other applicable programs and requirements of latest AAP reference flight schedule.

7. Missions: Comparable to current AAP missions.

Subsequent to the foregoing definition, the study requirements, constraints and guidelines were further expanded by a supplemental "Study Plan" summarized below.

RCM spacecraft - This area of investigation consisted of determining the feasibility of reusing major components and/or subsystems for an AAP mission. The studies involved with the RCM spacecraft were only to the depth necessary to determine the feasibility (along with cost and schedule factors) of renovating a CM for a 14-day, low inclination, low altitude, earth orbit mission.

The following ground rules were used:

1. Mission: low inclination, 200 mile, earth orbital logistics flight with a three-man crew.

2. Flight duration: up to 14 days.

3. Crew safety: Apollo criteria and probabilities.

4. Mission success: open, considering Apollo Block II systems.

5. Spacecraft considered for renovation: Apollo Block II vehicles.

6. Reference mission: 213.

7. Number of flight articles: 5.

8. Only relatively undamaged spacecraft to be considered for renovation.

The studies involved with the RCM laboratory were to be more extensive and to the detail level necessary to provide a conceptual definition of the lab. The selection and incorporation of subsystems and components were to be made from the hardware standpoint, rather than experimental requirements. A hardware "building block" approach was to be utilized and the contractor was to determine the effects on mission and payload support capabilities derived by adding each "building block."

RCM laboratory - This investigation was to provide a conceptual definition of the RCM laboratory along with the associated cost and schedule factors, and encompass approximately 40 percent of the study effort. The RCM laboratory was to be mounted upon a structure and attached to the lunar module mounting points in the SLA.

Major emphasis was to be placed on the basic dependent laboratory and the subsystems that might be added to it. The laboratory configuration and arrangement effort were to be minimized.

The basic laboratory includes the pressure vessel, environmental protection (radiation, meteoroid, and passive thermal control), LM docking structure, inner secondary structure, primary mounting structure, and basic instrumentation. The subsystems were to be defined in modular "building blocks" and included at least the following:

1. Stabilization and control
2. Communication and data
3. Inercom (hardline)
4. Thermal control (active glycol loop and radiators)
5. Airlock (two configurations were to be investigated: one on the RCM main hatch, and the other attached to the RCM bottom extending down to the S-IVB dome)
6. Reaction control (the Apollo SM quads were to be used and mounted to the primary structure mounting system)
7. Power (both primary batteries and fuel cells)
8. Oxygen and hydrogen storage
9. Control and display panel

The ground rules to be followed were as follows:

1. Mission - The four reference missions of AAP; polar and low-inclination, low-altitude and equatorial synchronous earth orbits, and a lunar orbit.

2. Mission duration - Baseline 30 days, with modular add-on up to one year or as much over 30 days as practical.

3. Payload - undefined. Primary structure to provide a capability for a maximum RCM laboratory, subsystems and experiment weight of 25,000 pounds. AAP study results were to be used to determine general experiment support requirements (power, thermal control, stabilization, etc.).

4. Number of flight articles - six.

5. Ground test articles and mockups - none unless proven necessary.

6. Spacecraft considered for renovation - Both Apollo Blocks I and II.

7. Only relatively undamaged spacecraft will be considered for renovation.

8. Crew safety - Apollo criteria for lab operation.

9. Mission success - open, considering RCM laboratory subsystems as space qualified hardware.

10. Contractor will be assumed to gain possession of spacecraft at splashdown.

11. Design of the RCM laboratory mount will conform to Apollo Block II CM-to-SM mount.

12. The RCM laboratory will not be optimized for experiments.

Investigations in the areas of mission/payload requirements, crew tasks and timelines; ground, preflight and flight operations, test planning and ground support equipment were to be kept to a minimum consistent with the detail necessary to derive hardware cost and schedule factors. Previous studies had established potential experiment groupings and preliminary designs of the spacecraft configurations.

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II. MISSION ANALYSIS

MISSION OBJECTIVES

The Apollo Applications Program (AAP) will provide NASA with the capability of performing earth orbital and lunar missions. These will extend the scope of the current Apollo program, lead to further definition of requirements and capabilities of the next generation of orbiting laboratories, the development of space technologies, and unmanned and manned planetary exploration missions. The relationship of the AAP program to other national space programs is shown in Figures 3 and 4.

The AAP program is a critical step in the development of technologies for utilization of orbital laboratories, orbital assembly and launch facilities, and for the qualification of subsystems and modules for interplanetary flights. As shown in Figure 4, the AAP overlaps the basic Apollo program, and the extended lunar exploration and earth orbital laboratory programs. Some of the boosters and spacecraft now assigned to the Apollo flights may be reassigned to the AAP by early accomplishment of the Apollo program objectives. It is possible that the AAP can lead directly to spacecraft modules that may be qualified for planetary flyby and manned planetary landing missions.

This program encompasses the following three classes of missions:

Earth Orbital

These missions will exploit the mission capabilities of the first geneations of space laboratories under such conditions as absence of atmosphere, weightlessness, and will provide a comprehensive overview of the earth. Investigations will include those in the physical and biological sciences, astronomy and astrophysics, atmospheric and earth sciences, communications, navigation, and advanced technology. One primary objective will be the development of the operational techniques necessary to demonstrate the broad manned space flight capability inherent in the Apollo system. This will include the performance of long duration missions in high inclination and synchronous orbits. Extended capabilities will be achieved through the use of multiple rendezvous missions and orbital assembly operations. Three basic categories of earth orbit missions have been identified as: low inclination, low altitude earth orbit; low altitude polar orbits; and synchronous orbits.

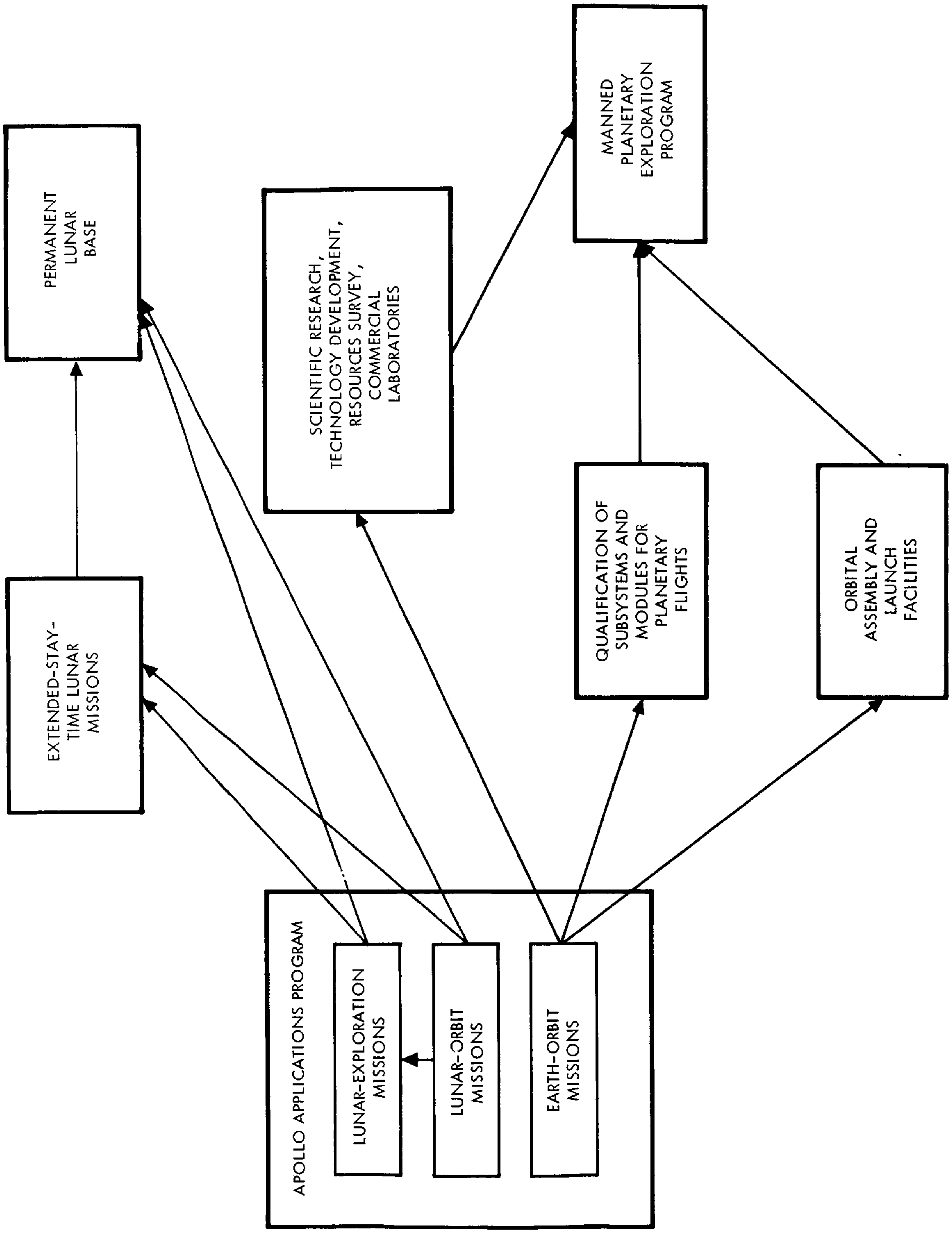


Figure 3. Relationship of AAP and Other National Space Programs

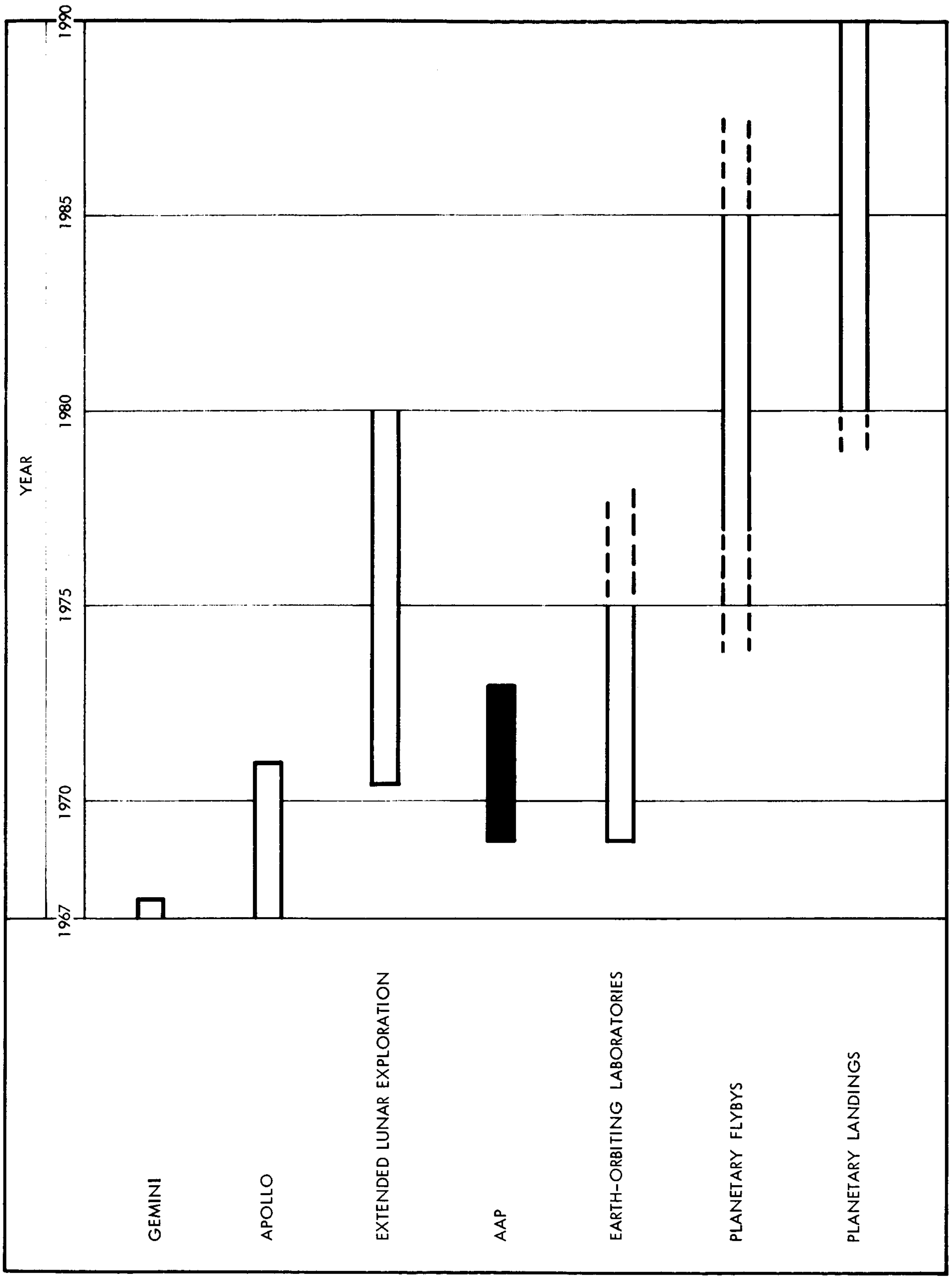


Figure 4. Scheduling Relationships of AAP Missions and Other Manned Space Programs

Lunar Orbit Survey

Lunar orbital missions will be flown, to map and survey the lunar surface to determine potential sites for surface missions, and to establish an overall topographical lunar survey to obtain scientific information concerning regions inaccessible to surface missions. Detailed understanding and exploration of the moon will require a series of flights carrying various remote sensors and orbit-to-surface probes.

Lunar Surface Exploration

Lunar surface missions will provide for exploration of the moon and further investigation and exploitation of its unique characteristics for scientific research. Investigations will be made of stratigraphic and tectonic relationships, ages and composition of lunar soil and rock, determination of the internal structure and composition of the moon, and various scientific surveys.

Typical AAP Missions Catalog and Schedules

Table 1 catalogs AAP missions that include utilization of the S-IVB spent stage, the renovated CM laboratory, and later the basic subsystem module to achieve maximum benefits from the program. The lunar landing missions have not been included as they do not involve utilization of space laboratories for experiment performance. The mission durations may range from 14 days for early missions (that use the unmodified Block II CSM) to as long as a year for later missions involving multiple rendezvous with a laboratory using an AAP type CSM. Missions of this type might emphasize the development of space operations capabilities and qualification of subsystems and modules for manned interplanetary flight.

Typical AAP missions schedules and objectives are shown in Figures 5 and 6.

Illustrated in Figure 6 is an integrated AAP program which would utilize the S-IVB stage, RCM, and basic subsystem module (BSM). The later phases of this program would permit missions lasting from six months to a year to determine man's capabilities, and to qualify the subsystems and modules for interplanetary missions. The relatively short lead times and freedom for selection of specific subsystems associated with the RCM laboratory can impart a high degree of flexibility in the planning of these missions.

Mission duration extensions may be achieved through launches of additional CSM's and support modules as well as through CSM's of increased lifespan. In the case of the independent laboratory, significant increases in

Table 1. Experiment Requirements for AAP Missions

Mission Parameters				Booster		Spacecraft Configuration			
Flight Number	Altitude (nm)	Inclination (deg)	Duration (days)	S-IB	S-V	Modules	Manned	Unmanned	Objectives and Remarks
209)	200	28.5 E	14	X		CSM + RCM Lab	X	X	210 rendezvous with 209 after 1 day
210				X		S-IVB SS			
211)	200	90 E	28	X		CSM	X		Earth surface survey
212				X		CSM	X		
213)	200	28.5 E	28-56	X		CSM	X		S-IVB spent stage opens, artificial "G," PPM and/or biomedical package
214				X		CSM	X		214 rendezvous with 213 after 14-28 days
215	200	28.5 E	28	X		CSM + RCM Lab		X	Unmanned MM
507	200	28.5 E	28		X	CSM + RCM Lab (independent)	X		Astronomy/astrophysics
508	200	90 E	28		X	CSM + RCM Lab (independent)	X		Earth surface and atmosphere survey, MS&S Module
509	19, 350	0	14 min		X	CSM + RCM Lab (independent)	X		Astronomy/astrophysics
510	19, 350	Low E	14 min		X	CSM + RCM Lab	X		Earth-oriented sciences
216)	200	28.5 E	165-180	X		Unmanned MM		X	S-IVB spent stage opens, PPM and/or biomedical package
217)				X		CSM + RCM Lab	X		217 rendezvous with 216 after 30-45 days
218)	200	28.5 E	135	X		CSM	X		218 rendezvous with 217 45 days after 217 launch
219				X		CSM	X		219 rendezvous with 218 45 days after 218 launch
511	44	90 L	14		X	CSM + RCM Lab	X		Remote survey of lunar surface (7 days)
514	19, 350	0	28		X	CSM + RCM Lab		X	Astronomy/astrophysics
515	44	90 L	35		X	CSM + RCM Lab	X		Remote survey of lunar surface (28 days)
220	200	28.5 E		X		RCM Lab		X	Provide laboratory space for 221, 222, 223, 224
221)				X		CSM			Up to one year by resupply of 220 boilerplate mission module
222)	200	28.5 E	360	X		CSM			221 rendezvous with 220 six months after 220 launch
223)				X		CSM			222 rendezvous with 221 three months after 221 three months after 221 launch
224				X		CSM			223 rendezvous with 222 three months after 222 launch 224 rendezvous with 223 three months after 223 launch Biomedical/behavioral and earth oriented schemes
225	200	90 E	28	X		RCM Lab		X	Rendezvous of 226 with 215 earth surface sensing
226				X		CSM	X		
227-232	200	28.5 E	TBD	X (8)		CSM, BSM RCM Lab	X		Follow-on program development support
519) 523)	19, 350 19, 350	TBD 0°	56		X (2)	CSM + RCM Lab + BSM	X		Follow-on program development support 523 CM astronomy Unmanned after return to earth
522 & 524	TBD	TBD	TBD		X (2)	Unmanned MM		X	Follow-on program development support. May be Voyager

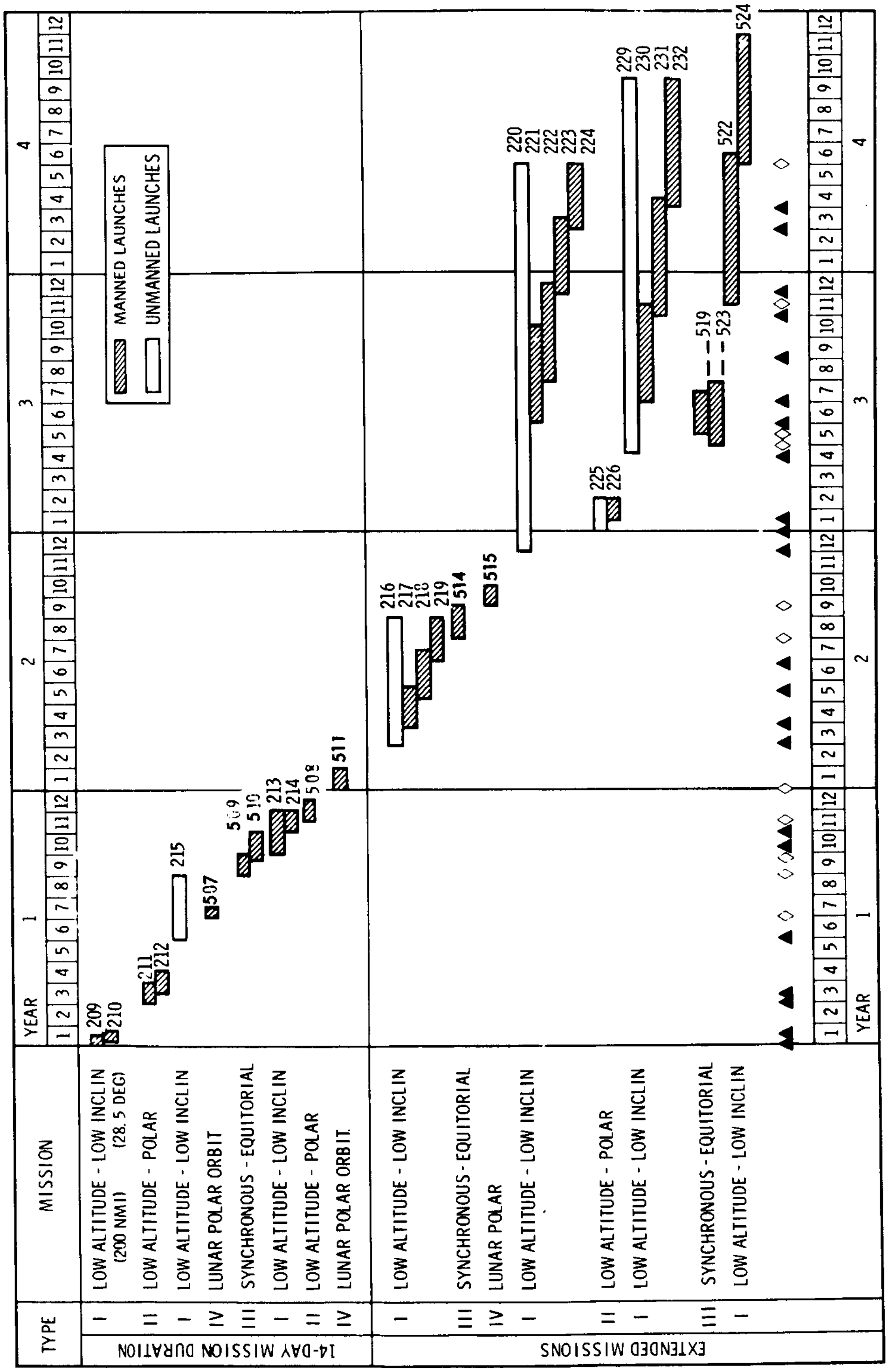


Figure 5. Typical AAP Missions Schedule

PHASE I (14-DAY)

		209	210	211	212	213	214	215	507	508	509	510
	LUNAR-POLAR ORBIT (LPO)								507			
	EARTH-SYNCHRONOUS EQUATORIAL ORBIT (ESO)										509	510
	EARTH-POLAR ORBIT (EPO)			211	212					508		
	EARTH LOW-INCLINATION ORBIT (ELIO)	209	210			213	214	215				
SCIENTIFIC	BIOMEDICAL/BEHAVIORAL	x	x	x	x	x	x	x	x	x	x	x
	PHYSICAL SCIENCES	x	x					x			x	x
	ASTRONOMY/ASTROPHYSICS		x							x	x	
	EARTH-ORIENTED SCIENCES AND APPLICATIONS			x	x	x	x			x		x
	LUNAR SURFACE SENSING SURVEY AND MAPPING								x			
SPACE ENGINEERING AND TESTING	EXTRAVEHICULAR ENGINEERING TASKS			x	x	x	x			x	x	x
	OPERATIONS TECHNIQUES AND ADVANCED-MISSION SPACECRAFT											x
	ADVANCED TECHNOLOGY AND SUPPORTING RESEARCH	x	x	x			x		x			x
MISSION		209	210	211	212	213	214	215	507	508	509	510
VEHICLE		△	▲	△	▲	△	▲	◆	◆	◆	◆	◆
	S-IVB SS	●		●		●		●				
	RCM LABORATORY		●	●	●	●		●	●	●	●	●
	BASIC SUBSYSTEM MODULE											
	SUBSYSTEM SUPPORT MODULE						●					
	COMMAND SERVICE MODULE	●	●	●	●	●	●	●	●	●	●	●

EXTENDED MISSIONS

		216	217	218	219	511	514	515	220	221	222	223	224
	LUNAR-POLAR ORBIT (LPO)					511		515					
	EARTH-SYNCHRONOUS EQUATORIAL ORBIT (ESO)						514						
	EARTH-POLAR ORBIT (EPO)												
	EARTH LOW-INCLINATION ORBIT (ELIO)	216	217	218	219				220	221	222	223	224
SCIENTIFIC	BIOMEDICAL/BEHAVIORAL	x	x	x	x	x	x	x	x	x	x	x	x
	PHYSICAL SCIENCES	x	x	x	x								
	ASTRONOMY/ASTROPHYSICS						x						
	EARTH-ORIENTED SCIENCES AND APPLICATIONS	x	x	x	x				x	x	x	x	x
	LUNAR SURFACE SENSING SURVEY AND MAPPING					x		x					
SPACE ENGINEERING AND TESTING	EXTRAVEHICULAR ENGINEERING TASKS		x								x		x
	OPERATIONS TECHNIQUES AND ADVANCED-MISSION SPACECRAFT					x		x	x	x	x	x	x
	ADVANCED TECHNOLOGY AND SUPPORTING RESEARCH	x	x	x	x	x		x	x	x	x	x	x
MISSION		216	217	218	219	511	514	515	220	221	222	223	224
VEHICLE		△	▲	▲	▲	◆	◆	◆	△	▲	▲	▲	▲
	S-IVB SS	●							●				
	RCM LABORATORY	●		●	●	●	●	●		●		●	
	BASIC SUBSYSTEM MODULE								●		●		
	SUBSYSTEM SUPPORT MODULE		●		●								●
	COMMAND SERVICE MODULE		●	●	●	●	●	●		●	●	●	●

		225	226	227	228	229	230	231	232	519	523	522	524
	LUNAR-POLAR ORBIT (LPO)									519	523		
	EARTH-SYNCHRONOUS EQUATORIAL ORBIT (ESO)											522	524
	EARTH-POLAR ORBIT (EPO)	225	226	227									
	EARTH LOW-INCLINATION ORBIT (ELIO)				228	229	230	231	232				
SCIENTIFIC	BIOMEDICAL/BEHAVIORAL	x	x	x	x	x	x	x	x	x	x	x	x
	PHYSICAL SCIENCES				x	x	x	x	x	x	x		
	ASTRONOMY/ASTROPHYSICS				x	x	x	x	x			x	x
	EARTH-ORIENTED SCIENCES AND APPLICATIONS	x	x	x								x	x
	LUNAR SURFACE SENSING SURVEY AND MAPPING									x	x		
SPACE ENGINEERING AND TESTING	EXTRAVEHICULAR ENGINEERING TASKS					x	x	x	x				
	OPERATIONS TECHNIQUES AND ADVANCED-MISSION SPACECRAFT					x	x	x	x			x	x
	ADVANCED TECHNOLOGY AND SUPPORTING RESEARCH	x	x	x	x					x	x		
MISSION		225	226	227	228	229	230	231	232	519	523	522	524
VEHICLE		△	▲	▲	△	▲	▲	▲	▲	△	▲	△	▲
	S-IVB SS	●			●								
	RCM LABORATORY	●		●		●		●		●	●	●	●
	BASIC SUBSYSTEM MODULE				●		●		●	●		●	
	SUBSYSTEM SUPPORT MODULE	●		●		●		●			●		●
	COMMAND SERVICE MODULE		●	●	●	●	●	●	●	●	●	●	●

● COMPONENTS
◆ SINGLE-LAUNCH VEHICLE
△ PRIMARY VEHICLE
▲ RENDEZVOUS VEHICLE
| DOCKING AND TRANSFER BY CSM
— RENDEZVOUS DOCKING

Figure 6. AAP Mission Objectives

reliability can be achieved through the additional redundancy provided by the laboratory subsystems. The subsystems support required for the experiments can be rotated between the CSM and the laboratory to maximize the total mission life consistent with crew safety. Several examples of extended missions concepts are given in Table 1.

REFERENCE MISSIONS

Four reference missions have been defined to typify the requirements of the basic AAP earth and lunar orbital missions, and to provide baselines against which to evaluate the capabilities of the RCM laboratories and the optional laboratory subsystems. Typical experiment packages are also identified for each of these missions, and the experiment requirements defined. Typical mission profiles are illustrated to provide a basis for determining crew and subsystems housekeeping requirements.

Shown in the preceding section on AAP mission objectives are the reference missions and mission characteristics assumed for the system flight configuration evaluation summarized in Table 2. With the exception of the planned mission duration, the earth polar orbit, earth synchronous altitude orbit and lunar polar orbit missions are assumed to be the same as those studied during the S&ID Apollo Extension Systems (AES) study performed for NASA during 1965. The 45-day mission duration assumed for the 1965 study has been reduced to 30 days, and a low inclination, low altitude earth orbit has been added. Since no Phase II low inclination earth orbit was considered,the AES Flight 229 program is assumed as the basis for obtaining typical experiment requirements. The S-IB boost vehicle will be used for the low altitude, low inclination orbit missions and the S-V for the low altitude earth polar orbit, earth synchronous altitude orbit, and lunar polar orbit.

Reference Mission I

Reference Mission I is a S-IB-boosted, 200-nautical-mile, 28.5-degree inclination orbit mission. The S-IB booster launches a manned Apollo CSM and RCM laboratory into a 100-nautical-mile parking orbit. The RCM laboratory is then docked to the CSM, and the spacecraft is placed into a 200-nautical-mile circular orbit using the service module propulsion system. After mission orbit operations, the CSM separates from the laboratory and initiates the earth entry sequence. The mission flight profile and other characteristics are essentially the same as those of the 14-day Flight 215 mission (SID 65-1727) studied under Contract NAS9-5017, except that the duration was extended to 30 days. The objectives of this mission are to obtain comprehensive biomedical and behavioral data, obtain radar scattering, cross section measurements of terrain, and temperature soundings of the earth atmosphere, ultraviolet mapping of the celestial sphere, X-ray astronomy, and a variety of space physics experiments.

Table 2. Reference Missions

	Low Altitude Earth Orbit				
	ELIO (RM-I)	EPO (RM-II, Rendezvous)	EPO (RM-II, SV)	ESO (RM-III)	LPO (RM-IX)
Orbit Altitude (nm)	200	200	200	19,500	80
Orbit Inclination (deg)	28.5	90	90	-0-	90
Mission Duration (days)	30	30	30	30	30
Configuration	Single launch S-IB + CSM + RCM	Rendezvous S-IB + CSM S-IB + RCM	Single launch SV + CSM + RCM	Single launch SV + CSM + RCM	Single launch SV + CSM + RCM
Flight profile Launch and Ascent	Planar steering	Direct launch 180° launch azimuth	140° launch azimuth with boost steering		S-IVB injection into parking orbit
Parking Orbit	80 x 200 nm	100 x 100 nm (rendezvous)	100 x 200 nm		100 x 100 nm circular
Transfer to Operational Orbit	SPS injection	SPS injection	SPS injection	S-IVB and SPS injection	S-IVB translunar injection SPS lunar orbit insertion
Gross payload in operational orbit (lbs)	33,100	27,000/18,000	96,300	62,000	71,000
Objectives	Biomedical and behavioral Temperature sounding of atmosphere UV mapping of celestial sphere Zero gravity studies of physical properties Nuclear emulsion	Biomedical and behavioral Radiation monitor Test solar cell Atmospheric refraction of stellar images IR data for cloud mapping IR and microwave radiation data for selected earth points Multiwave-length earth mapping	Biomedical and behavioral Radiation monitor Test solar cell Atmospheric refraction of stellar images IR data for cloud mapping IR and microwave radiation data for selected earth points Multiwave-length earth mapping	Biomedical and behavioral Radiation environment monitoring Physical science (magnetic field lines, comet-like particles, micrometeoroid collection) Conjugate Aurora Launch OGO Subsystem development (fluid management for LSS, radioisotope, optical technology)	Multiwave-length mapping of lunar surface Radar altimetry Gravity surveying Geochemical sensing

Reference Mission II

Reference Mission II includes either a dual S-IB launch with rendezvous or a single S-V launch, and will be performed in a 200-nautical mile polar earth orbit. The principal objectives of this mission relate to the further development and use of earth sensing systems and the accumulation of additional data on the earth's surface and atmosphere. A multifrequency radar-imaging capability will be installed in the vehicle. Most of the earth will be mapped during the mission. Other objectives are to obtain biomedical and behavioral data for extended-mission durations; deploy and test a solar-cell array producing about 10 kilowatts of power, and to utilize the power for radar mapping; determine the refraction of stellar images as stars occult through the visible atmospheric fringe (star tracker); obtain infrared emission data for correlation with photographic data to reconstitute cloud images; obtain spectral distributions of radiation in selected frequency bands for various points on the earth surface; and map the earth's surface. Mapping data will include stereo-cartographic photographs for topographic mapping, multicolor photographs to reconstitute full-color and modified false-color photographs, radar mapping data for correlation with these photographic data, broadband VHF data on surface moisture conditions, and earth microwave data for correlation with the VHF data. Knowledge of the spacecraft altitude is needed during some of the experiments and will be provided by equipment used in the experimental program.

Approximately one day is required for spacecraft systems check, initial experiment equipment setup, and equipment shutdown prior to deorbit. Orbital maintenance is not required for this mission, as the exact orbital altitude is not critical to the experiments.

Reference Mission III

Reference Mission III will be performed in a synchronous, equatorial earth orbit, with a planned duration of 30 days. The principal objectives are to perform high-resolution astronomical photography and the associated calibration photometry, to study the nature of the conjugate auroras, and to evaluate solar sailing. Seven extravehicular excursions are planned for micrometeoroid collection. Table 2 summarizes the experiment requirements for this mission.

Reference Mission IV

Reference Mission IV has a 80-nautical-mile, lunar polar orbit. The planned mission duration for the lunar orbit phase is 24 days; 30 days for the total mission. The principal objective of this mission is to map the lunar surface using cameras on the daylight side and radar on the dark side. Ultraviolet spectroscopy, passive microwave and radar altimetry data will also be obtained during both day and night operations; multispectral photography, gravity surveying, and remote chemical sensing will be performed during day operations; and VHF reflectivity will be performed during night operations. Three hours of daylight mapping and three hours of night mapping will be performed during each 24-hour period. Table 2 summarizes the mission experiment requirements.

Reference Mission Profiles

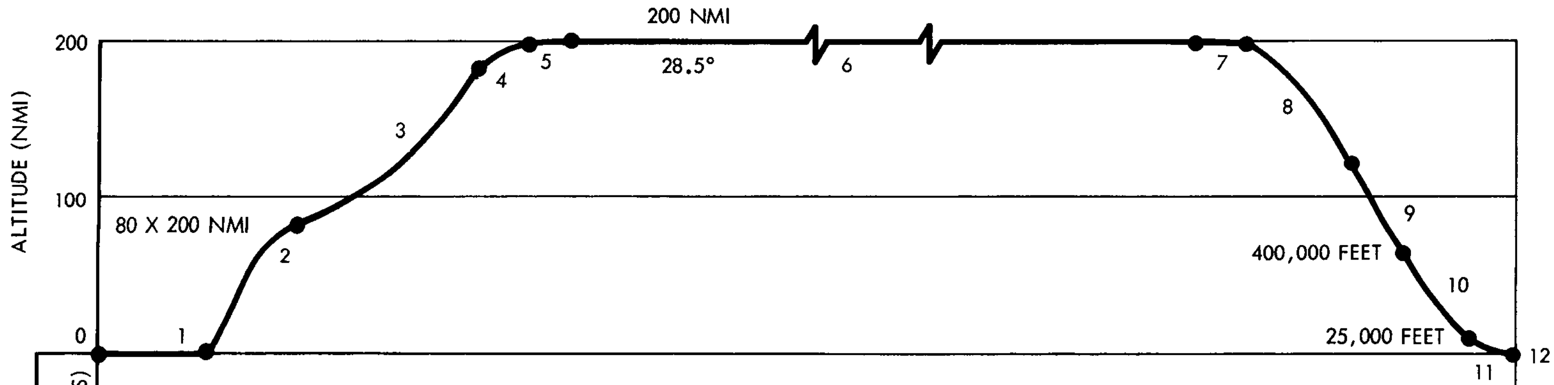
The applicable reference mission profiles are illustrated in Figures 7 through 10, as selected for this study to establish a suitable baseline for the determination of general flight plans, flight performance, payload weights, mission effectiveness and other significant parameters reflecting the reference mission flight support requirements to accomplish stated experimental objectives.

The operational flight profiles for Reference Missions II, III and IV (Tables 3, 4, and 5, respectively) are based on the ground rules and the mission and subsystem analyses defined in SID 65-1534-1, "System Analysis Summary." Since no low inclination earth orbit reference mission was considered in the Phase II AES Study, typical early time period mission requirements are arbitrarily assumed which involve a single Saturn IB launch with a dependent laboratory and experiment payload.

An alternate Reference Mission I would involve a dual launch of Saturn IB, one for the fully independent RCM laboratory with experiments launched unmanned, and the second S-IB to launch a manned CSM to rendezvous with the laboratory in operational orbit. A typical reference mission profile for this alternate is represented in Report SID 65-1727, "AES Flights 214 and 215." Consumables, times and other requirements, however, are to be adjusted to be consistent with the 30-day mission operational baseline established for nominal mission duration.

MISSION ENVIRONMENT

The mission environment analysis portion of the RCM study is to define the aerothermodynamic environment experienced during the phases of the mission that occur within the earth's atmosphere. This environment has an important effect on the condition of the recovered spacecraft components and systems, and also establishes several of the design requirements for the renovated laboratory and spacecraft configurations, such as the ablative material thicknesses needed for heat protection during reentry and SLA thermal control coating and insulation requirements to protect the RCM laboratory from aerodynamic heating during boost.



PHASE	TIME (HOURS)
1. PRELAUNCH	2.0
2. LIFTOFF AND ASCENT	0.2
3. PARKING ORBIT	1.5
4. TRANSPORTATION AND DOCKING	0.5
5. TRANSFER TO OPS ORBIT	0.5
6. EXPERIMENTS IN OPS ORBIT	700.0
7. PREPARE FOR RETURN	9.0
8. DEORBIT	0.01
9. PREPARE FOR ENTRY	0.4
10. ENTRY	0.2
11. LANDING	0.2
12. RECOVERY	

Figure 7. Reference Mission I, Low-Inclination, Low-Altitude Earth Orbit, Thirty-Day Saturn IB Mission

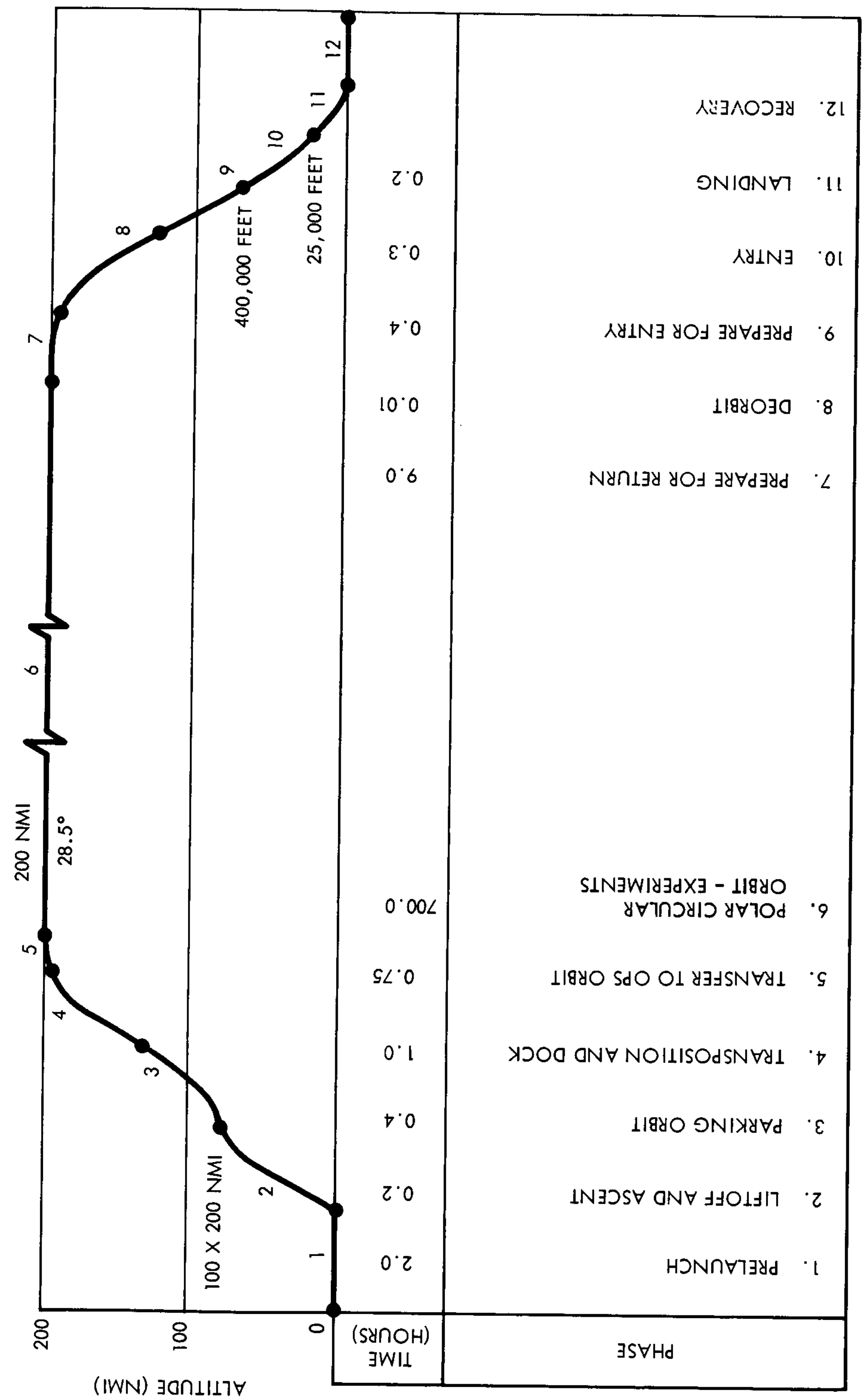


Figure 8. Reference Mission II, Low-Altitude Earth-Polar Orbit, Thirty-Day Saturn V Mission

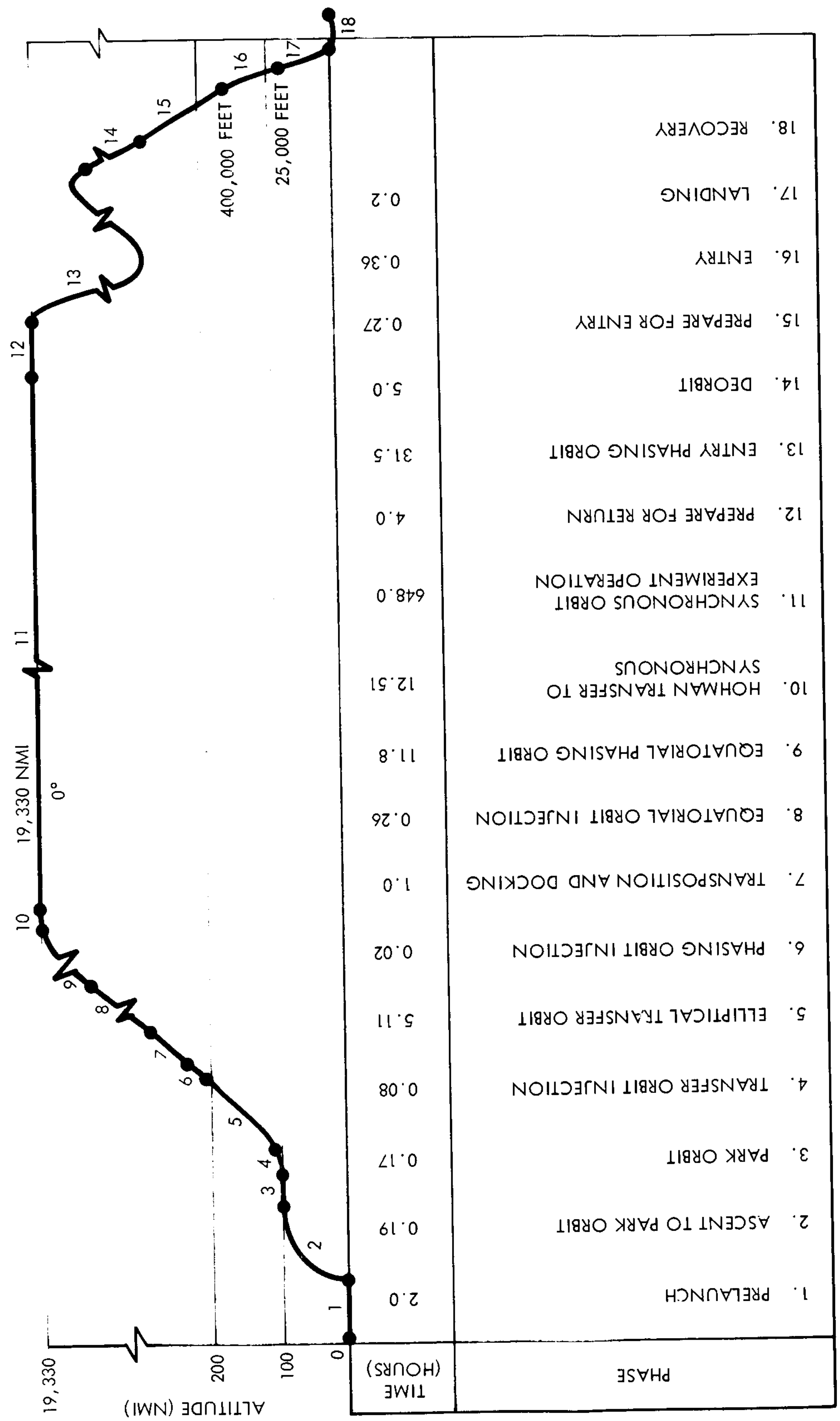


Figure 9. Reference Mission III, Earth-Synchronous Equatorial Orbit, Thirty-Day Mission

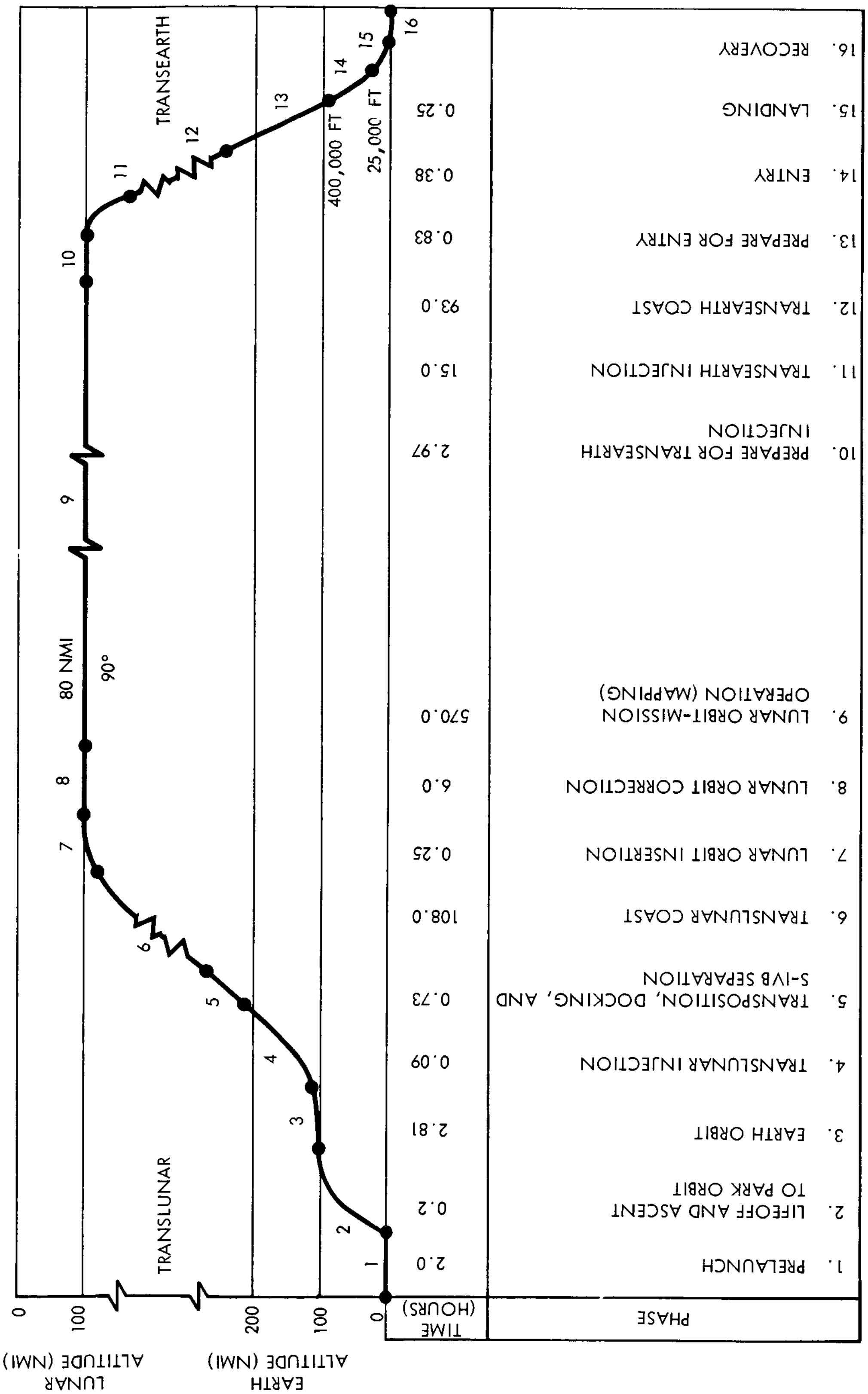


Figure 10. Reference Mission IV, Lunar Polar Orbit, Thirty-Day Mission

Table 3. Phase Times and Durations, Reference Mission II

Phase Description	End Phase Time (Hours)	Phase Duration (Hours)
Prelaunch	T = 0.0	2.00
Ascent to Circular Orbit	0.18	0.18
Parking Orbit Coast	0.58	0.40
Transfer Orbit Injection	0.68	0.10
Transposition and Docking	1.43	0.75
Polar Circular Orbit Insertion and Confirmation	1.48	0.41
Experimentation	710.84	709.00
Prepare for Return to Earth	719.84	9.00
Deorbit	719.84	0.004
Prepare for Entry	720.26	0.42
Entry (0.05 g to Touchdown)	720.54	0.32

Table 4. Phase Times and Durations, Reference Mission III

Phase Description	End Phase Time (Hours)	Phase Duration (Hours)
Prelaunch	T = 0.0	2.00
Ascent to Circular Orbit	0.19	0.19
Parking Orbit (100 nautical miles, 28.4 degrees)	0.36	0.17
Elliptical Transfer Orbit Injection	0.43	0.08
Elliptical Orbit Coast	5.55	5.11
Phasing Orbit Insertion	5.56	0.02
Transposition and Docking	6.56	1.00
Equatorial Orbit Insertion	6.83	0.26
Equatorial Phasing Orbit Coast	18.62	11.80
Hohmann Transfer to Synchronous Orbit and Circularization	31.13	12.51
Synchronous Orbit Experimentation	679.13	648.00
Prepare for Return to Earth	683.13	4.00
Entry Phasing Orbit Insertion	683.15	0.02
Entry Phasing Orbit Coast	714.65	31.50
Deorbit	719.66	5.01
Prepare for Entry	719.94	0.27
Entry (0.05 g to Touchdown)	720.30	0.36

Table 5. Phase Times and Durations, Reference Mission IV

Phase Description	End Phase Time (Hours)	Phase Duration (Hours)
Prelaunch	T = 0.0	2.00
Ascent	0.20	0.20
Earth Orbit	3.01	2.81
Translunar Injection	3.10	0.09
Transposition and Docking and S-IVB Separation	3.83	0.73
Translunar Coast	111.83	108.00
Lunar Orbit Insertion	112.08	0.25
Lunar Orbit Correction	118.08	6.00
Lunar Mapping	6.88.08	570.00
Prepare for Transearth Injection	691.05	2.97
Transearth Injection	706.05	15.00
Transearth Coast	799.05	93.00
Prepare for Entry	799.88	0.83
Entry (0.05 g to Touchdown)	800.26	0.38

A summary of the aerothermodynamic environment during the launch and reentry phases of the mission is given in Figure 11, where peak heating rates and maximum load factors are tabulated to indicate the relative severity of the various trajectory profiles. These values are derived from the Apollo design trajectories, which are shown plotted in terms of altitude and velocity.

The aerodynamic heating of the RCM laboratory and spacecraft during boost have been determined for the Saturn IB and Saturn V design trajectories shown in Figure 11. Heating rate histories have been compiled for the CM, SM, and SLA, based on Apollo heating calculations.

The heating environment that is experienced during command module reentry can best be described in terms of a flight envelope that defines the limits of steep and shallow reentry profiles. The steep reentry high heat rate and shallow reentry high heat load boundaries are shown in Figure 11 for both earth orbital and lunar return missions. The corresponding stagnation point heating profiles are illustrated in Figure 12. The heating histories are based on the correlation of theoretical calculations and full-scale flight test results from the Apollo development program. These nominal heating rates are considerably less than the initial conservative estimates that were used to design the Apollo heatshield, and reflect the increase in knowledge of convective and radiative heating that has taken place as a result of the extensive Apollo experimental and flight test programs.

The performance characteristics of the Block I and Block II heat shields when exposed to their respective earth orbital and lunar return nominal heating environments are shown in Figure 12. Stagnation point ablator thickness requirements for reentry of renovated command module spacecraft from low inclination, low altitude, earth orbits were also calculated. In contrast to the conservative design approach used to define the original heat shield, the renovated command module spacecraft heat protection system is based on the refined state of technology, resulting in a much thinner and lighter heat shield.

The heat protection sytem analysis results show that sufficient virgin material remains after reentry from earth orbit, and possibly after return from lunar missions, to provide thermal protection for an additional reentry from a low inclination, low altitude, earth orbit mission. It is feasible from an aerothermodynamic standpoint to renovate a spacecraft heat shield by removing the charred material down to the level of the virgin material beneath. It is recommended that further study of renovated heat shield requirements be conducted to substantiate the results of this feasibility analysis. Additional study of ablator thickness requirements should be performed for other points on the body, and in areas where protuberances

	MAX HEATING (BTU/FT^2 - SEC)	MAX LOAD FACTOR (g's)	REENTRY VELOCITY (FPS)	REENTRY ANGLE (DEGREES)
SATURN-IB BOOST	1.18*	4.1		
SATURN-V BOOST	1.86*	4.6		
EARTH-ORBITAL HIGH-HEAT-RATE	172.8	20.0	26,803	-10.14
EARTH-ORBITAL HIGH-HEAT-LOAD	56.6	3.85	27,448	- 3.50
LUNAR-RETURN HIGH-HEAT-RATE	506.3	20.0	35,177	- 9.45
LUNAR-RETURN HIGH-HEAT-LOAD	196.8	5.45	35,165	- 5.20

(*)HEATING TO SPACECRAFT LM ADAPTER (SLA)

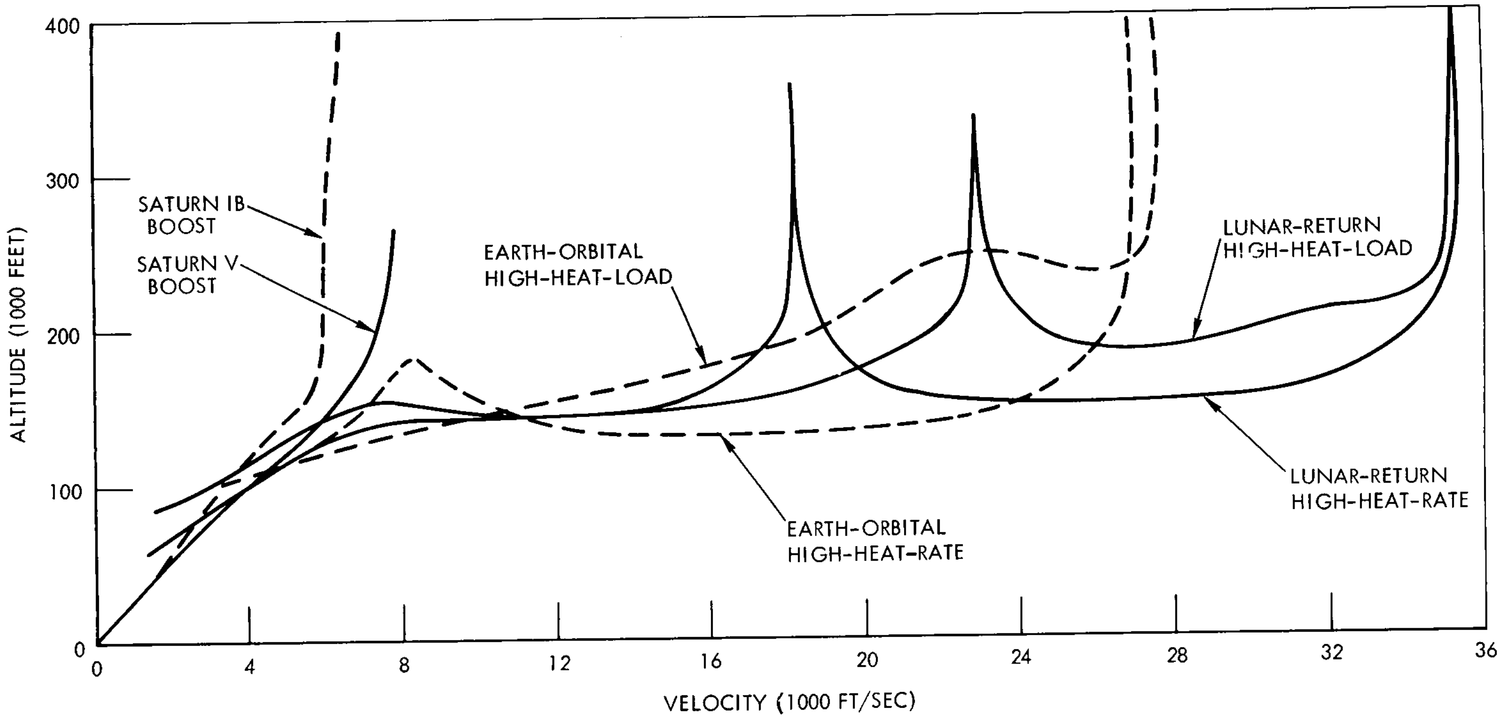


Figure 11. Mission Environment Summary

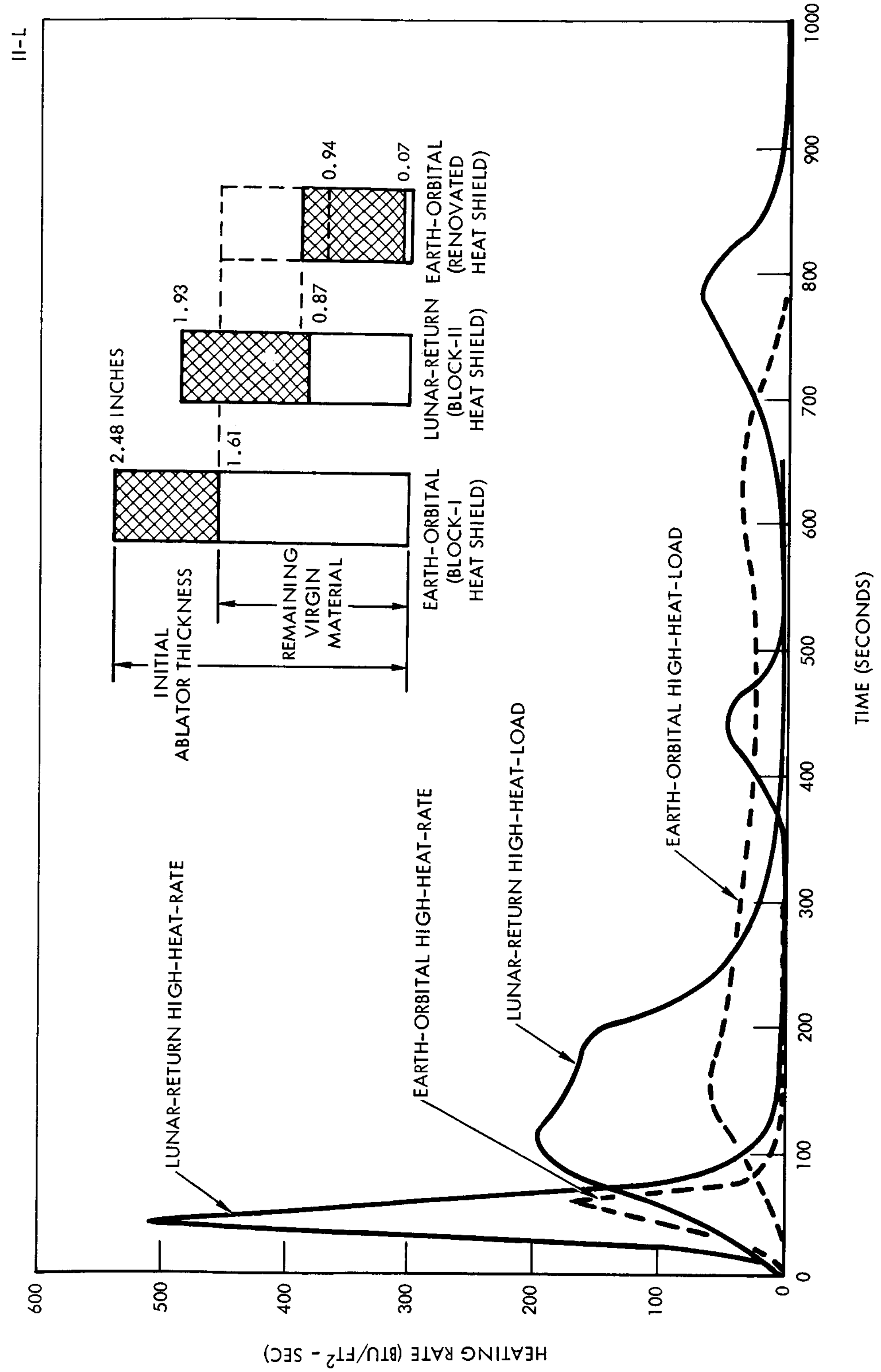


Figure 12. Reentry Environment and Heat-Protection System Performance

require local fairing and contouring of the heat shield. It is also recommended that a test program be formulated to measure the response of ablator samples to repeated heating environments, simulating the reuse of a renovated heat shield.

Launch Environment

In this section, the aerodynamic heating of the RCM laboratory and RCM spacecraft during the boost phase of flight is defined for the Saturn IB and Saturn V design trajectories. These trajectories assume the worst combination of booster dispersions and result in the most severe heating environment to be experienced during ascent. The illustration of the launch configuration in Figure 13 shows the longitudinal stations along the CM, SM, and SLA components at which heating rates are given. Boost-trajectory heating-rate histories are presented in Figures 14 through 16 for the Saturn IB and in Figures 17 through 20 for the Saturn V. The heating rates are valid for a vehicle angle of attack of up to five degrees.

Reentry Environment

The aerothermodynamic environment that is experienced during command module reentry can best be described in terms of a flight envelope that defines the limits of steep and shallow reentry profiles. An altitude versus velocity envelope for earth orbital and lunar return reentry trajectories is shown in Figure 21, based on the Apollo heat shield design trajectories. The lower high heat rate boundary is defined by the crew safety limit of 20 g's. The upper high heat load boundary represents the maximum longitudinal ranging capability that must be provided to satisfy mission performance requirements. The actual trajectory profile for a specific spacecraft will fall somewhere between these limits.

The heating rate histories for the various limiting trajectories are given in Figure 22. The results define the heating environment at the maximum heating point, which is located in the command module pitch plane at a non-dimensionalized distance of S/R = 6.192, measured along the aft heat shield surface from the vehicle centerline. The heating analysis evaluates the contributions of heat transfer due to convection, equilibrium radiation, and non-equilibrium radiation to the total heating rate.

The acceleration load factor histories for the high heat rate and high heat load trajectories are illustrated in Figures 23 and 24, respectively. The load factor profiles are characterized by several sharp peaks corresponding to the initial penetration and terminal descent phases of the reentry.

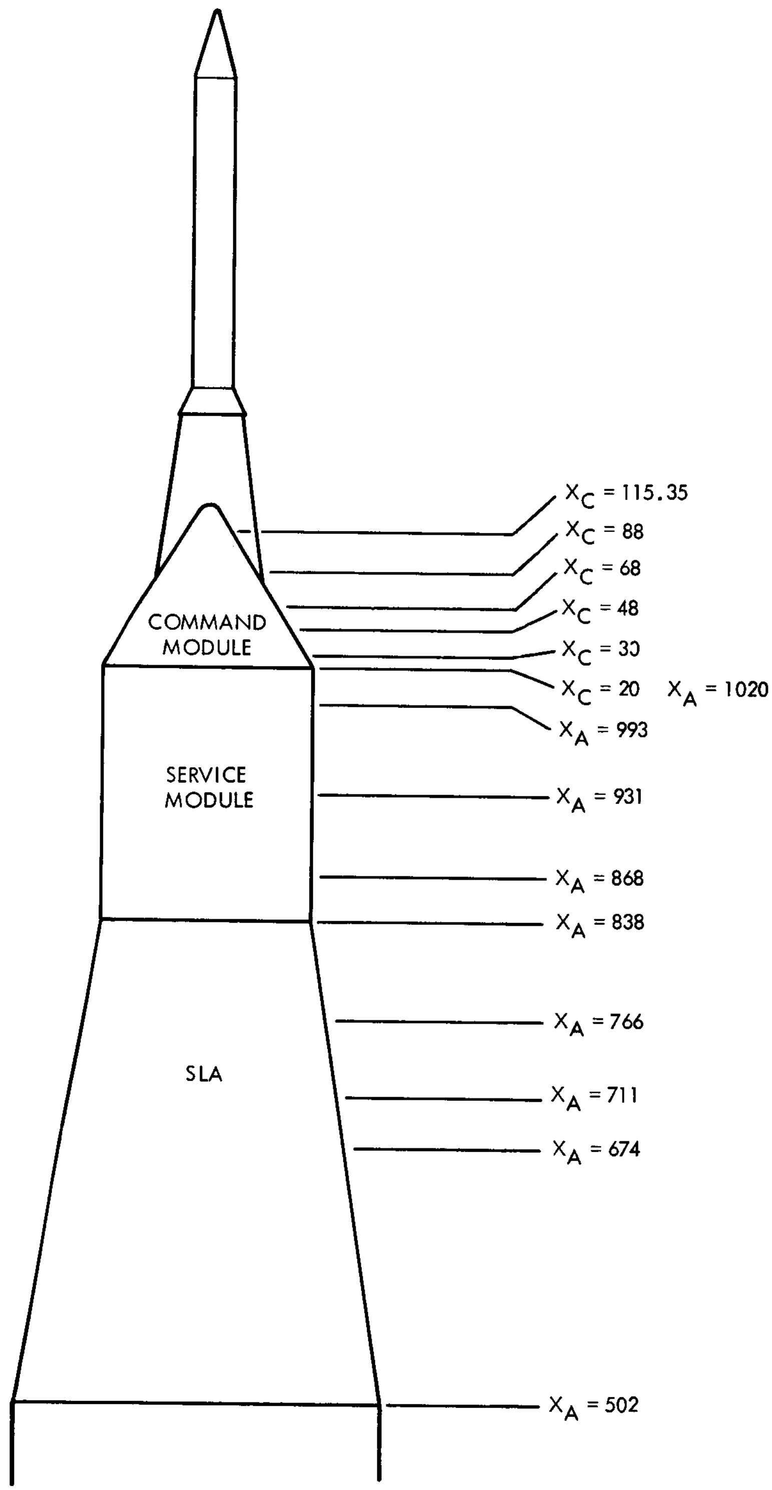


Figure 13. Saturn IB and Saturn V Launch Configuration

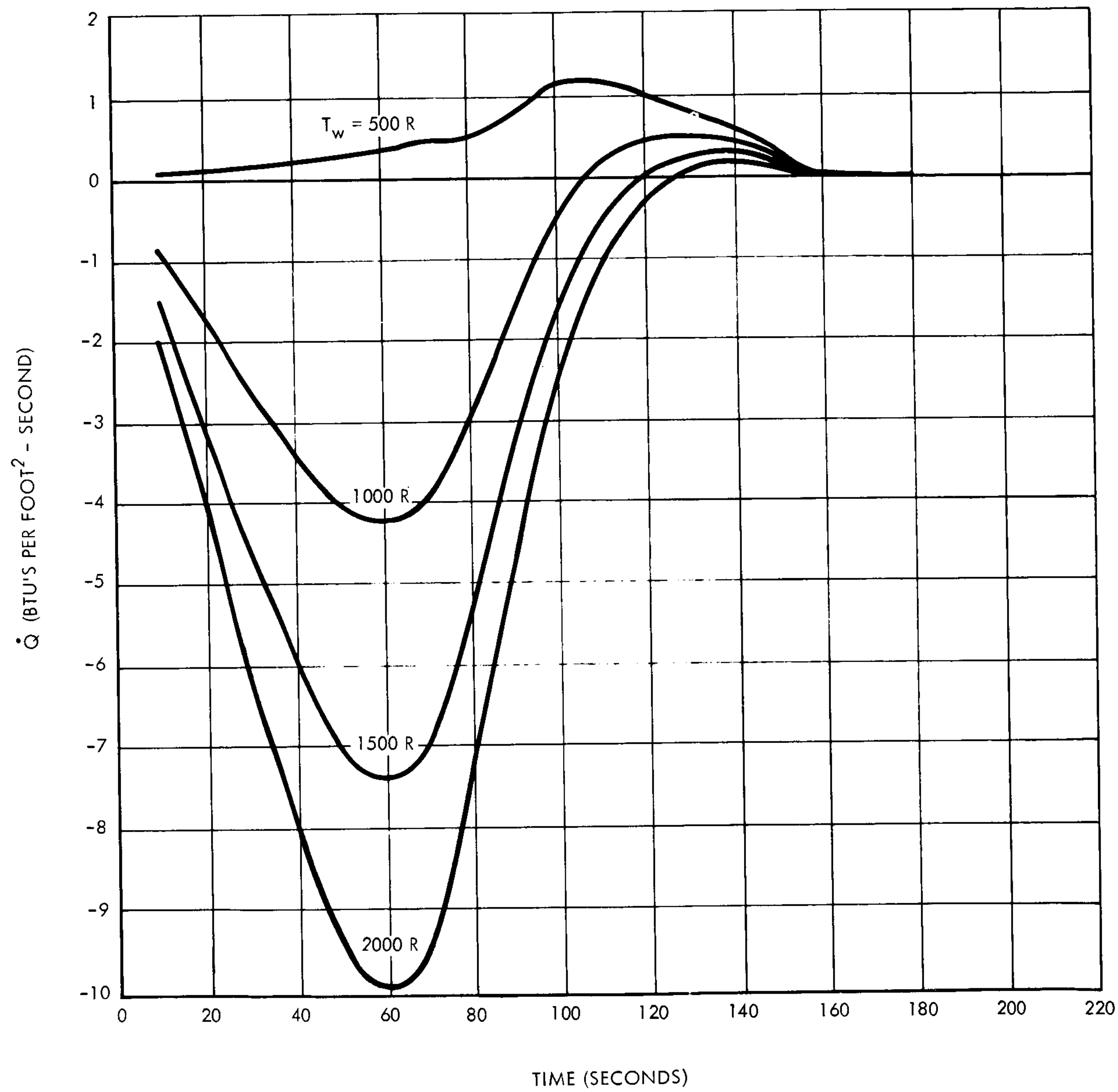


Figure 14. S-IB/S-IVB Adapter, Aerodynamic Heating, X_a = 838

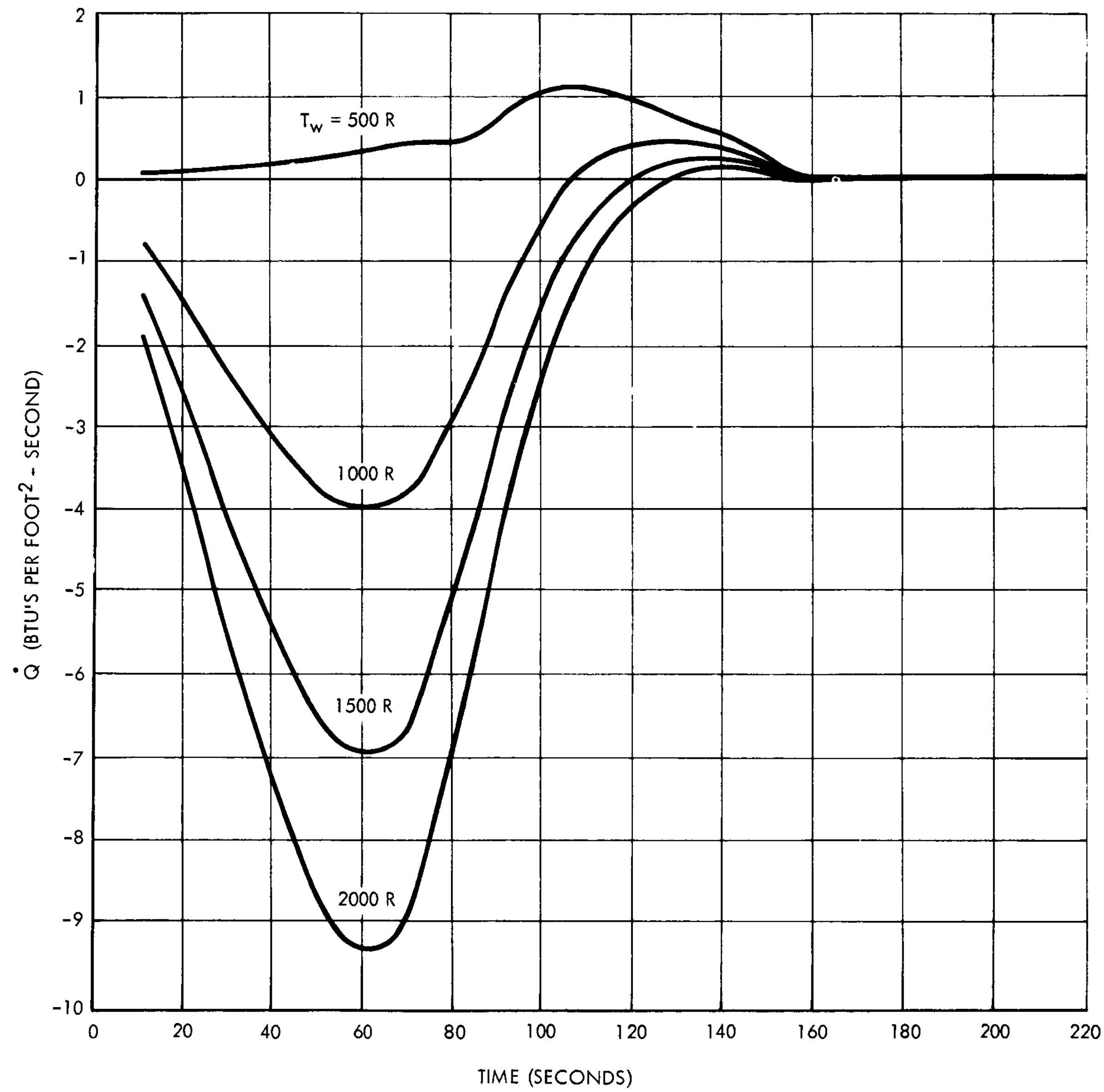


Figure 15. S-IB/S-IVB Adapter, Aerodynamic Heating, X_a = 674

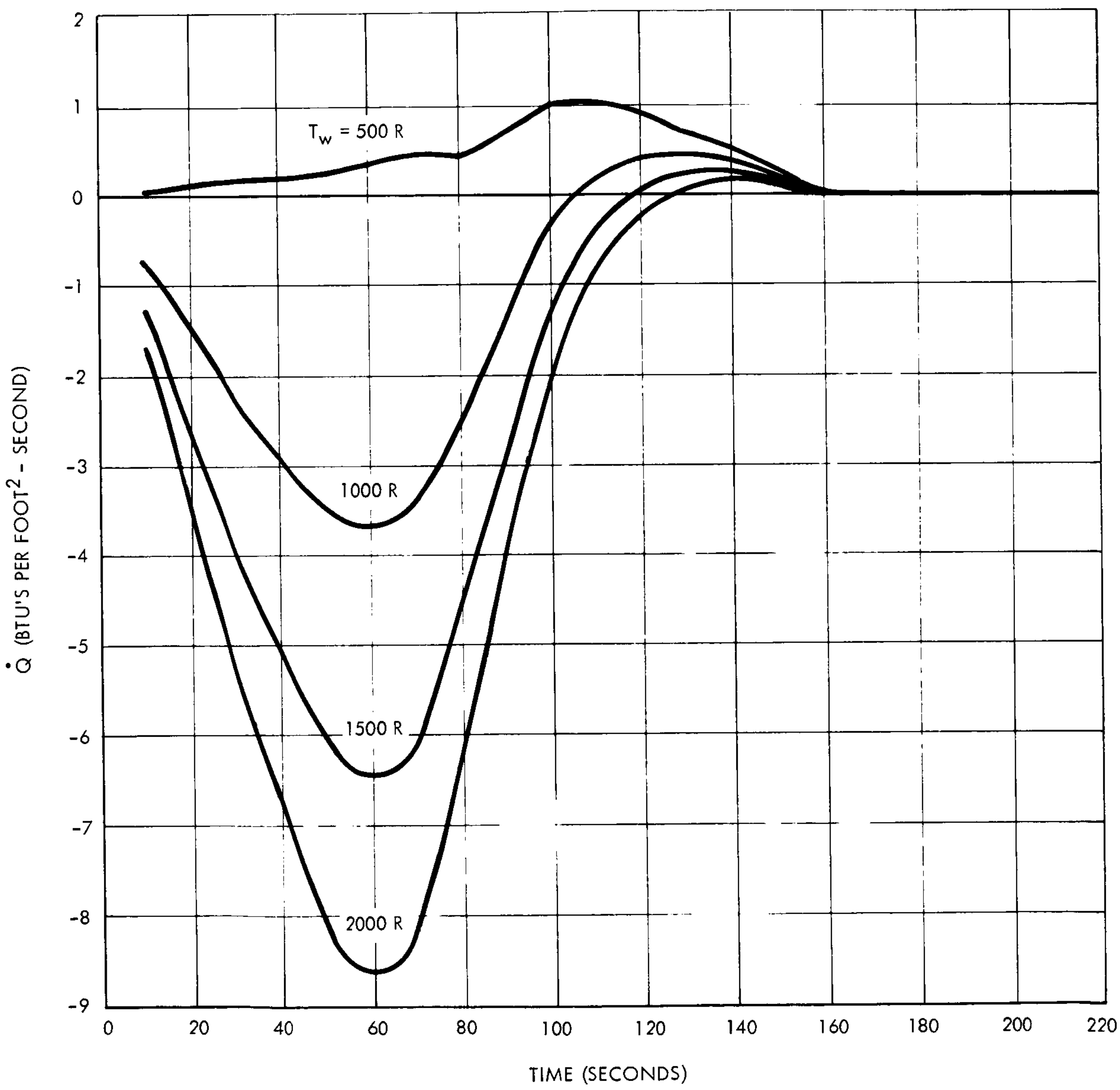


Figure 16. S-IB/S-IVB Adapter, Aerodynamic Heating, X_a = 502

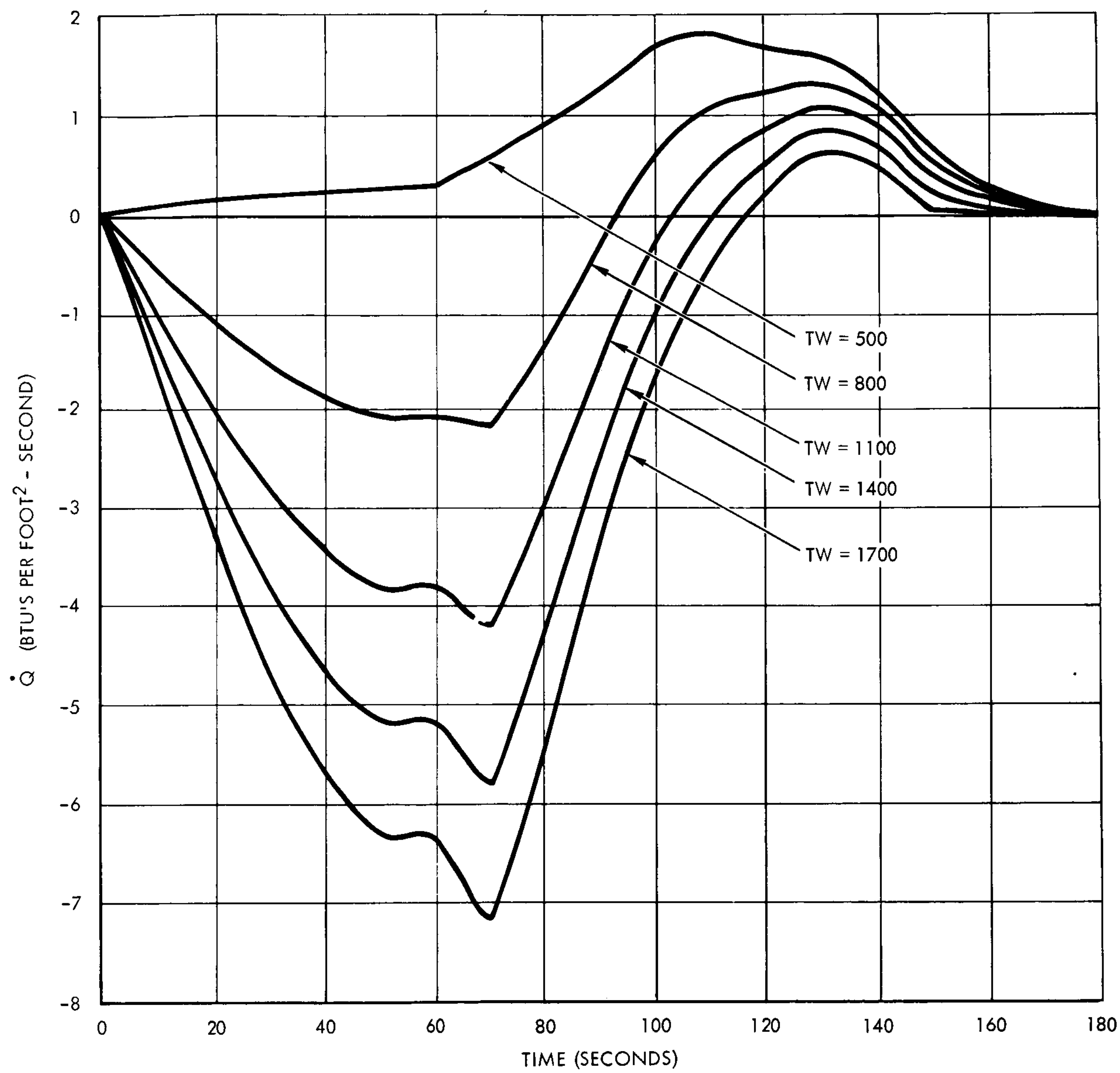


Figure 17. SLA Aerodynamic Heating, Saturn V, Block I (Design Trajectory) $X_a = 838$

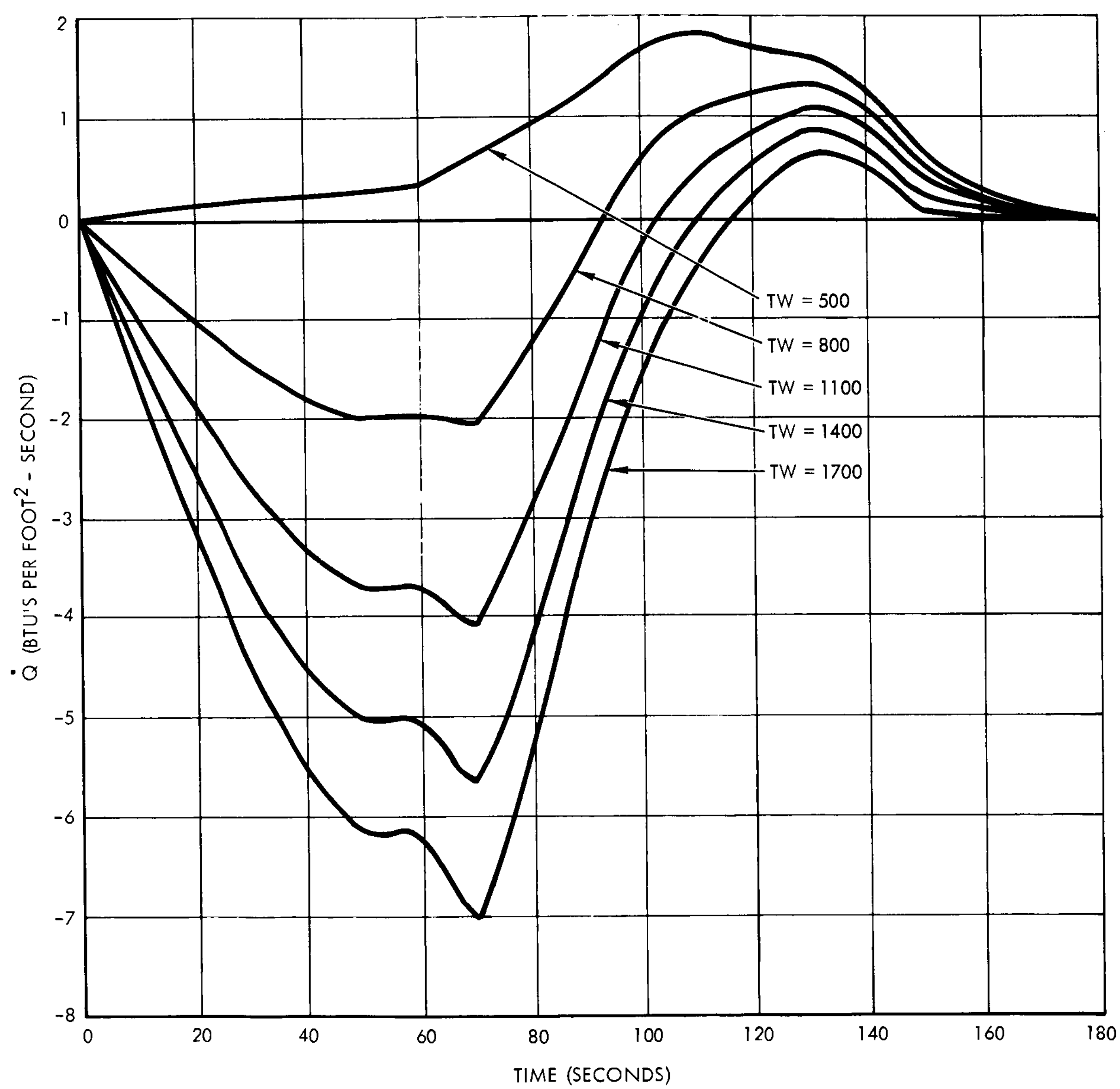


Figure 18. SLA Aerodynamic Heating, Saturn V, Block I (Design Trajectory) $X_a = 766$

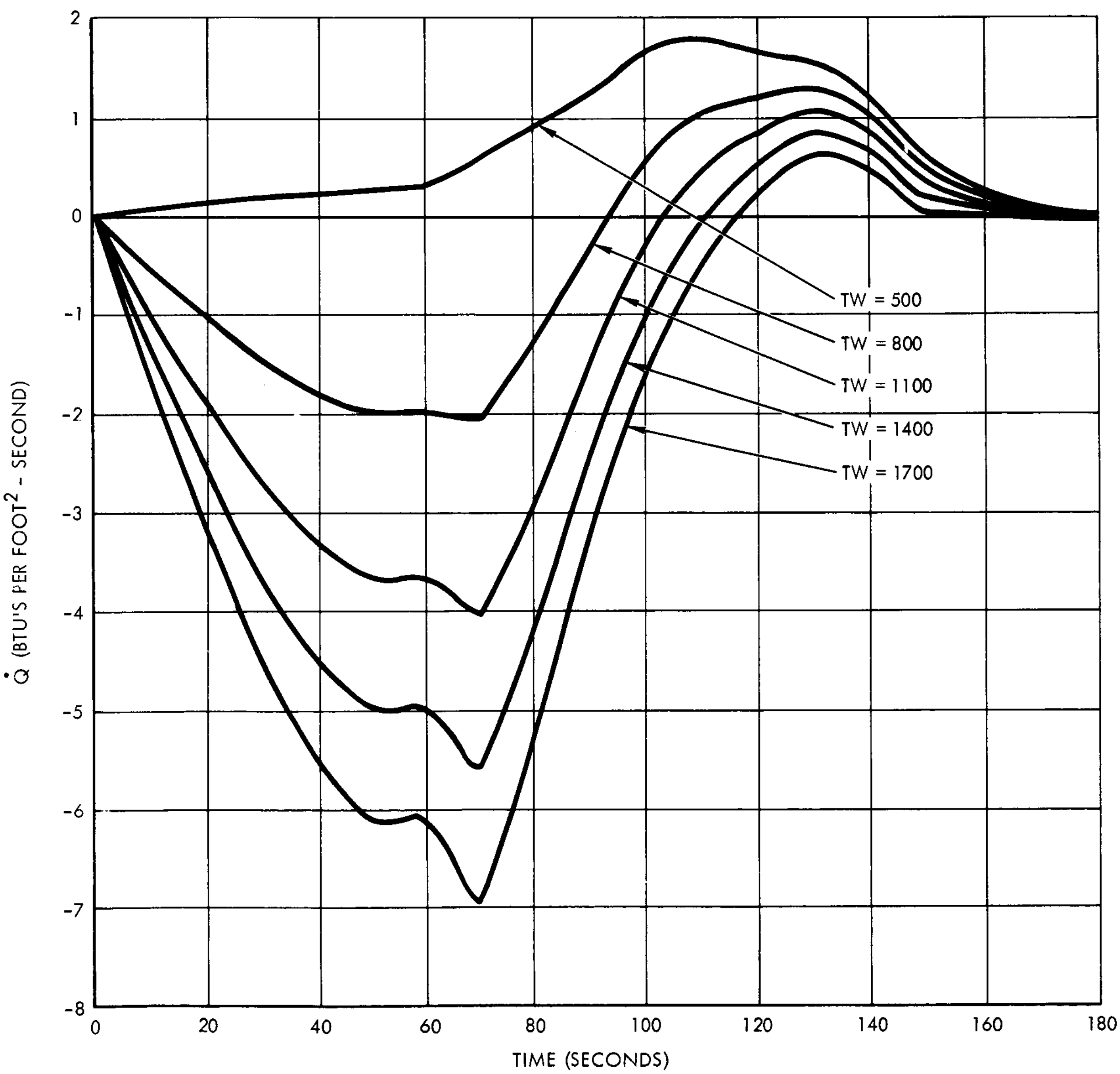


Figure 19. SLA Aerodynamic Heating, Saturn V, Block I (Design Trajectory) $X_a = 711$

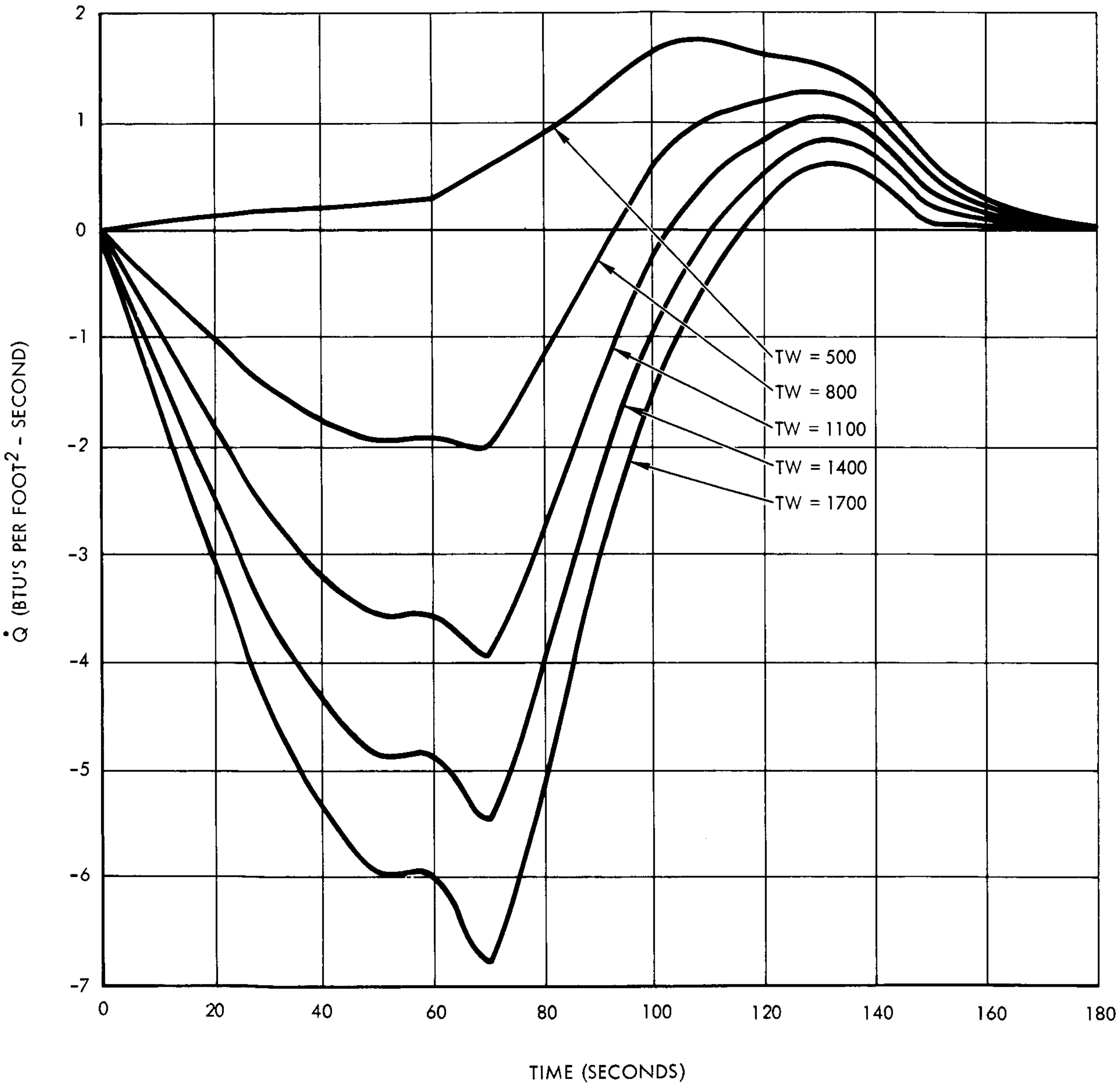


Figure 20. SLA Aerodynamic Heating, Saturn V, Block I (Design Trajectory)
$X_a = 502$

	REENTRY VELOCITY	REENTRY ANGLE
EARTH-ORBITAL HIGH-HEAT-RATE	26,803	-10.14
EARTH-ORBITAL HIGH-HEAT-LOAD	27,448	- 3.50
LUNAR-RETURN HIGH-HEAT RATE	35,177	- 9.45
LUNAR-RETURN HIGH-HEAT-LOAD	35,165	- 5.20

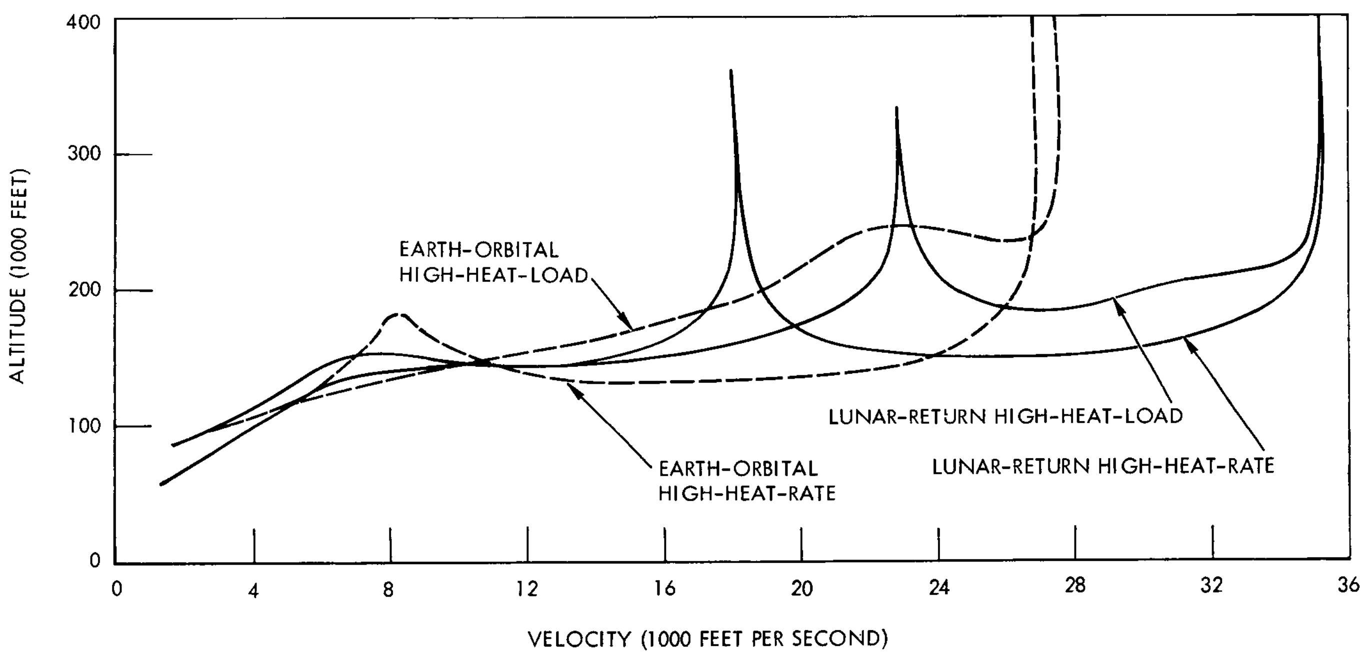


Figure 21. Altitude-Velocity Envelope for Command Module Reentry

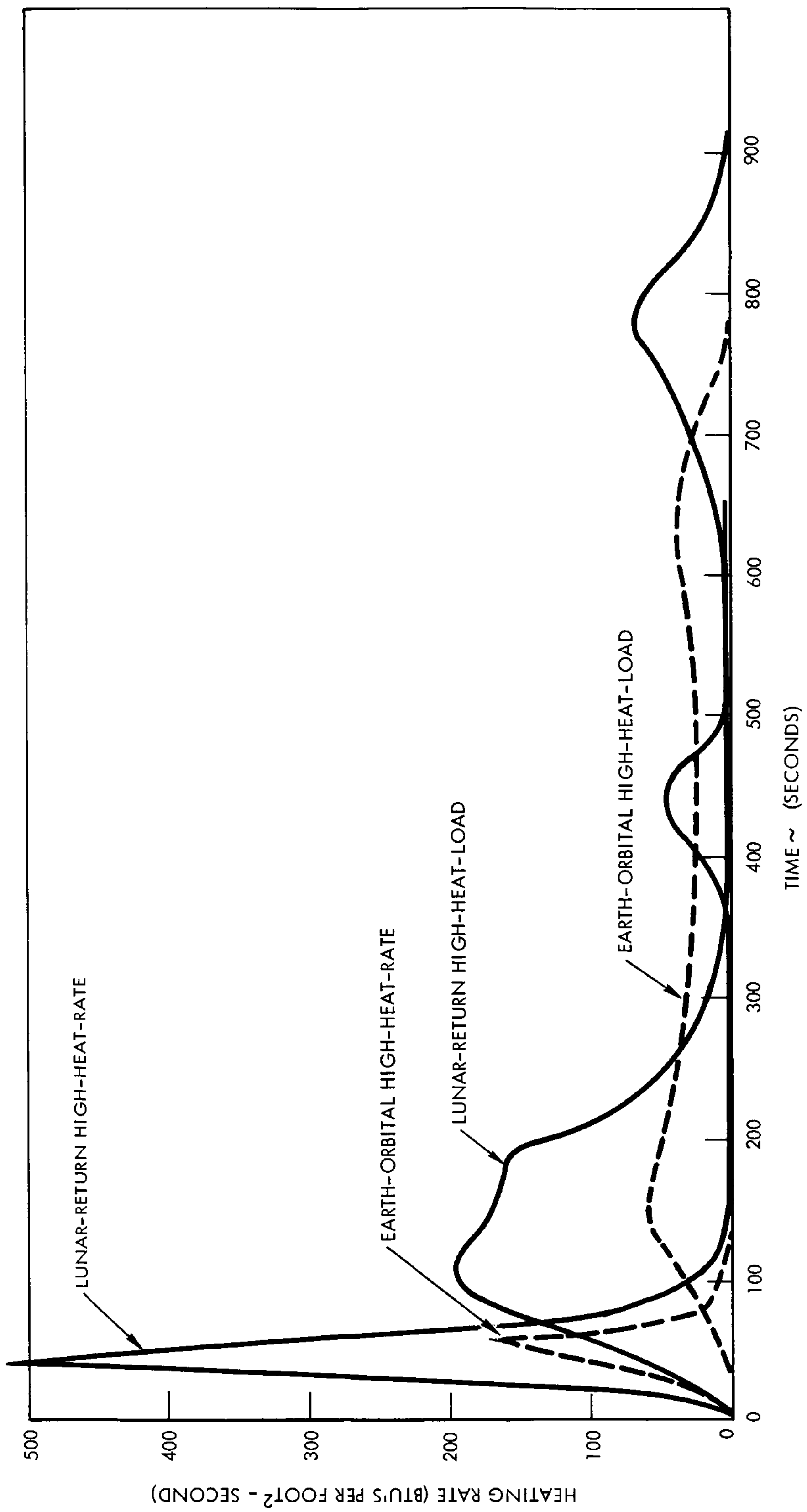


Figure 22. Heating-Rate Time Histories

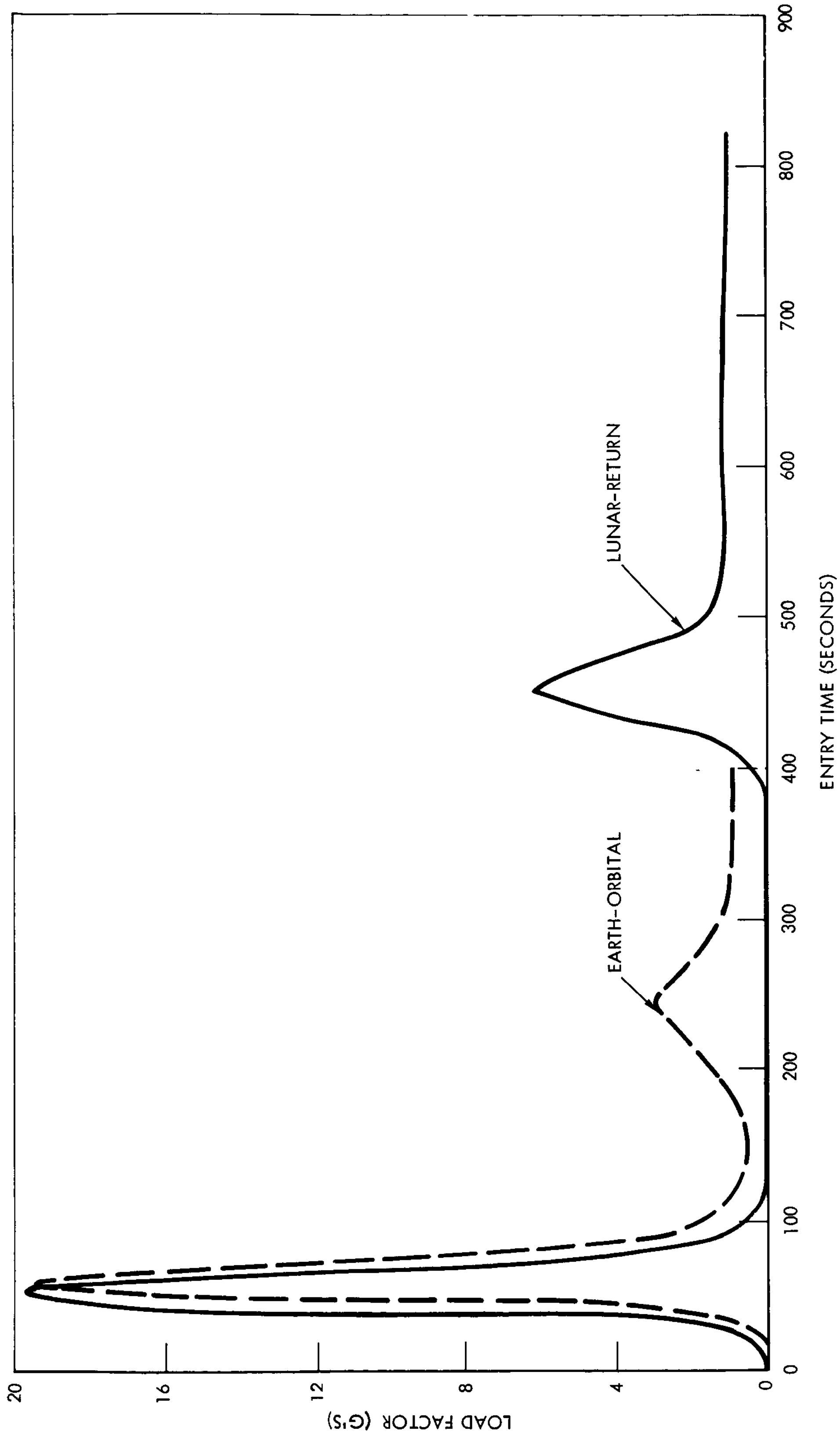


Figure 23. Load-Factor Time Histories for Maximum-Heat-Rate Trajectories
Maximum-Heat-Rate Trajectories

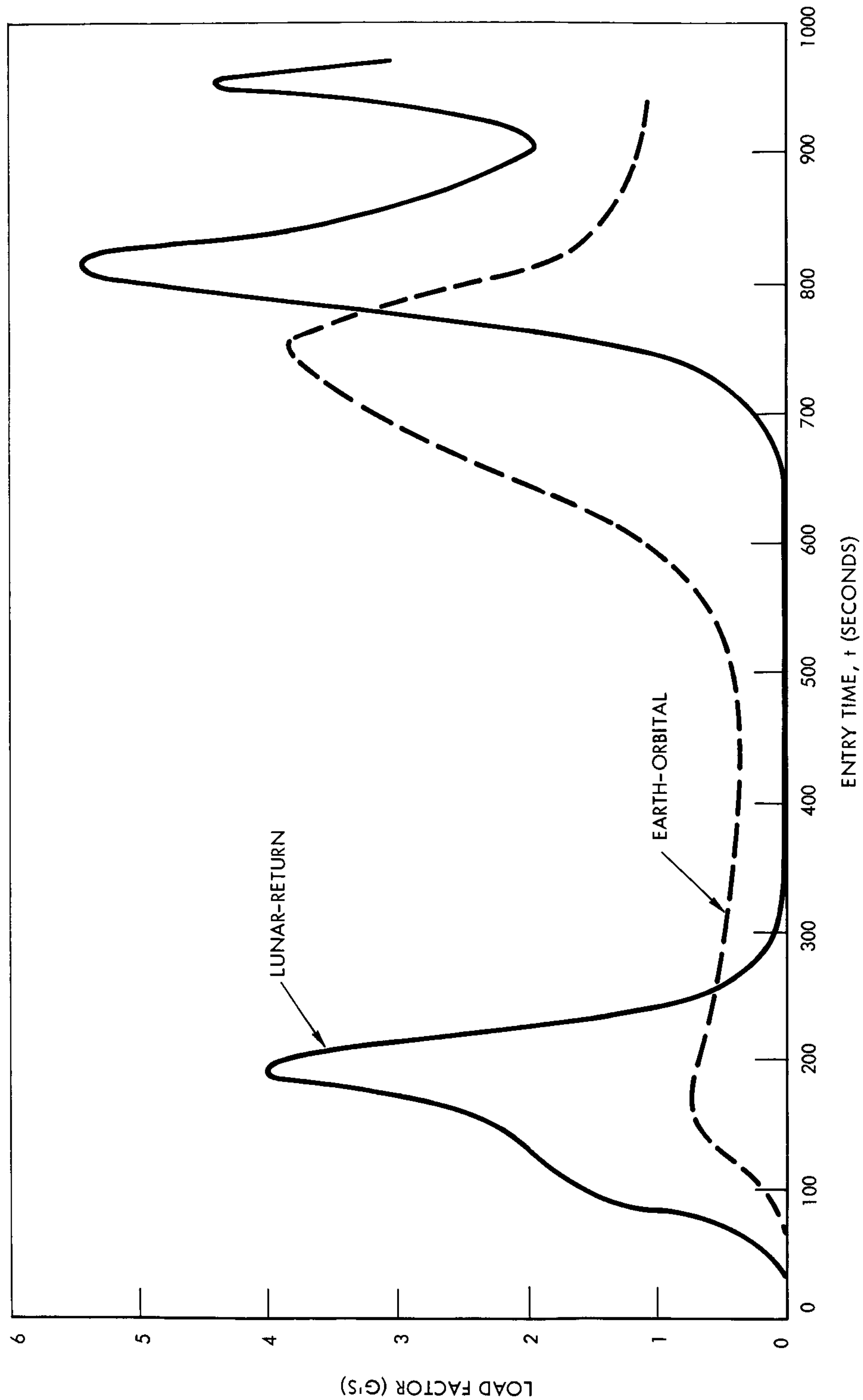


Figure 24. Load-Factor Time Histories for Maximum-Heat-Load Trajectories

Thermal Protection System Performance

A cursory heat shield ablation analysis was performed to support the heat shield refurbishment feasibility study. Results indicate that orbital entry spacecraft are potentially refurbishable for reuse as earth-orbital spacecraft. The furbishment of lunar entry spacecraft is, however, less attractive due to greater ablation response. This analysis considered clean body ablator response/requirements only. The effects of perturbed heating on refurbishment feasibility are pertinent to the conclusions but were not covered in this cursory evaluation.

The analysis consisted of determining the thermal response at two locations on the aft heat shield for two trajectories; both having heat loads equal or greater than earth-orbital entry loads. A 28.5 kfps entry and a maximum heat load lunar return trajectory were utilized.

The ablator thickness requirements were established (sized) for the 28.5 kfps entry by determining the ablator thickness required to protect the bondline to a maximum temperature of 600 F before or at earth impact.

By comparison, in the table of the virgin material remaining after flight with the sized thickness determined for the 28.5 kfps entry, it can be concluded that vehicles which have been subjected to entry from earth orbit are potentially refurbishable from thermal considerations. If this study were extended to include other types of lunar return entry trajectories, such as maximum heating rate trajectories as opposed to the maximum heating load trajectory which was considered, it may be determined that some Block II vehicles may be refurbishable based on individual mission considerations.

The analysis is of a preliminary nature and areas of perturbed flow remain to be considered.

MISSION PERFORMANCE REQUIREMENTS

Mission objectives and experimental requirements imposed on the RCM laboratory system by the projected scope of the AAP operational space experiments have been reviewed on the basis of available data from related studies to the maximum possible extent. Specifically, experiments that have been selected for the Apollo Extension System (AES) flights were considered, where a total of approximately 400 earth-orbiting space station experiment applications were described and assessed as to the requirements imposed upon the candidate space laboratories and space stations. These requirements are summarized by individual application to the corresponding reference mission profile and corresponding AAP mission experimental objectives.

Scheduling and integration of the various experiments and the detail application thereof was not the purpose or part of this RCM laboratory system conceptual study. However, it was found desirable to take advantage of existing and readily available data as well as results from completed computer solutions of other experiment configuration studies to assist in the approach towards accomplishing desired efficiencies of RCM laboratory utilization, crew manpower, electrical energy, and other significant system performance parameters affecting the support mission experimental objectives.

Laboratory Experiment Support Requirements

In earlier AES studies, NASA has selected a group of space experiments which are to have assignment priority for the proposed AAP flight schedules and plans. A schedule of the requirements associated with the individual AAP flights is shown in Table 6.

Table 6 summarizes the experiment requirements for AAP missions that typify groupings of experiments and equipment and are thus indicative of needs for equipment weights, pressurized and unpressurized values, power, and astronaut times allowable for experiments. The data are taken mainly from results of the AES studies, and are consistent with the 1965 NASA Phase II AAP flight programs and the 1965 estimates of booster, CSM, and laboratory capabilities for AAP missions. The experiment weights and values are NASA estimates, and are assumed to be the same for the 30- as well as the 45-day missions. Volume is the total volume of experiment equipment; since most of the equipment does not need to be mounted within the RCM laboratory but can be mounted externally; pressurized volume is not a constraint. The power requirements are NAA estimates. The astronaut hours required for experiment performance exceed the 45-day mission capabilities for three of the missions, and would exceed a 30-day capability for four of the missions. However, the experiment programs implied for these missions could be rescheduled so as to achieve the mission objectives without removing equipment. "Power" refers to the maximum watts required for experiment performance. It is assumed that experiments requiring high peak powers will only be performed when other power-consuming experiments are not being conducted.

Mission objectives and the requirements imposed on the spacecraft for each of the reference missions are described below. The consumables and times are based on the referenced preliminary operating profiles and timelines. Basic characteristics of the four reference missions are summarized in Table 7.

Table 6. Experiment Requirements for AAP Missions

Flight Number	Planned Duration (days)	Orbit		Objectives	Total Weight (lbs)	Pressurized Volume (cu ft)	Unpressurized Volume (cu ft)	Power (watts)	Astronaut Time (man-hours)
		Inclination	Altitude						
209	14	28.5	200	Capillarity investigations, spacesuit evaluation, and EVA	1,754	42.1	15.5	800	400
211	30	28.5	200	Maneuverable sub-satellite	876	17.3	25.5	1400	340
507	14	90	200	Maneuvering and docking, subsystem development and test	10,622	34	343	2435	157
509	14	EO	SYN	Syncom III recapture, large antennas, performance of extendable members	11,322	32.5	122	2200	263
215	14	50	200	Earth survey	4,273	48	284	2885	420
218	45	28.5	200	Study of living organisms, liquid/gas and solids behavior, and space environment	3,716	96	34	1900	767
219	45	28.5	200	EV operations, launch of unmanned satellite	2,516		160	380	386
221	45	28.5	200	Earth survey, atmosphere sensing	3,788	108	166	800	1200
513	14	81.5	200/ 700	EV operations, Echo II observation, meteorological techniques	6,447	58.5	235	500	754
516	45	EO	SYN	Living organisms, astronomical observations and techniques and earth atmosphere sensing	6,447	58.5	235	1050	754
518	45	97	200	Earth surface and atmosphere sensing, data capsule	7,193	72	278 + 6-ft diam sphere	1160	1062
521	45	EO	SYN	Radio-isotope systems, optical technology, micrometeoroid technology	8,254	33	338	3800	1217
523	45	28.5	200	Living organisms; solids/liquid/gas behavior, astronomical observations and techniques	18,149	136	165	1900	1334
229	45	28.5	200	Space structures, subsystems development, launch of unmanned satellites	3,704	5	253	765	150
230	45	28.5	200	EV operations	1,820		135	260	225
RM-II	45	90	200	Earth surface and atmosphere sensing	7,637			520	
RM-III	45	-0-	SYN	Space physics, fluid management, and optical technology	5,622			800	
RM-IV	34	90	80 (2)	Lunar mapping	3,274			2230	

Table 7. Summary of Reference Mission Characteristics

Reference Mission	Mission Trajectory	Duration (Days)	Objectives	Configuration
I & II	200-nautical-mile, circular, low-inclination earth orbit 200-nautical-mile circular, polar earth orbit	30	a. Biomedical and behavioral b. Radiation monitor c. Test solar cell d. Atmospheric refraction of stellar images e. IR data for cloud-cover mapping f. IR and microwave radiation data for selected earth points g. Multiwave-length earth mapping	S-IB, S-V, CSM RCM laboratory
III	Equatorial synchronous earth orbit	30	a. Biomedical and behavioral b. Radiation environment monitoring c. Physical Science (magnetic field lines, comet-like particles, micrometeoroid collection) d. Conjugate Aurora e. Launch OGO f. Subsystem development (fluid management for LSS, radioisotope, optical technology)	S-V CSM RCM laboratory
IV	80-nautical-mile, lunar polar orbit	34 total 28 in orbit	a. Multiwave-length mapping of lunar surface b. Radar altimetry c. Gravity surveying d. Geochemical sensing	S-V CSM RCM laboratory

Reference Missions I and II

The requirements for Reference Missions I and II are summarized in Table 8. Approximately one day is provided for spacecraft systems check, initial experiment equipment setup, and equipment shutdown prior to deorbit. Orbital maintenance is not required for this mission as the exact orbital altitude is not critical to the experiments. However, knowledge of the spacecraft altitude during some of the experiments is important and will be provided by equipment used in the experimental program.

Reference Mission II

Table 9 is a summary of the experiment requirements for this mission.

Reference Mission IV

Table 10 summarizes the main mission experiment requirements.

Reference Missions Experiments Summary

The total ranges of experiment support requirements anticipated for all of the AAP flights and the distributed requirements allocated to the selected Reference Mission Profiles for the RCM laboratory spacecraft flights are summarized in Table 11.

This data may be used to evaluate the effectiveness of various RCM laboratory system configurations and selection of subsystem "building blocks" to maintain adequate laboratory experiment support capability.

Mission Payload Performance

The parametric curves in Figures 25 through 38 present the orbital weight tradeoffs for possible RCM missions in the weight ranges of interest. The missions and associated launch vehicles are summarized in Table 12. This discussion is to describe how to apply the parametric curves, to describe the assumptions on which the curves are based, and to discuss the degree of similarity between the AAP data and RCM data regarding flight profiles and orbital weights.

Use of Parametric Curves

The parametric curves show "CSM return" weight and "laboratory" weight versus "CSM propellant weight" with "launch vehicle injection weight" as a parameter. The meanings of the terms in quotes, as defined for purposes of this study, must be defined because they may be misconstrued.

Table 8. Spacecraft Requirements Summary, Reference Mission I and II

	Electrical Power (kwh)	Spacecraft Attitude Hold (hr)	Experiment Minimum Attitude Dead Band (deg)	SPS Duration (min)	SM RCS Propellant (lb)	Number of EVA's	Data Requirements (hr)
Experiments*							
Biomedical/behavioral package	67.2						14.0
Radiation monitor	5.7						
Large solar cell system		787	±15		1275		1.4
Remote earth atmosphere sensing (0802A, B, C, and D)	9.2	138	0.15		302		4.0
Remote earth surface sensing (0901, 2, 3)	11.6	100	0.1		320		101.2
Housekeeping and Transit*							
Probable non-experiment requirements	1550	2	(0.5)	4.8	250	None	200.0
Total (Approximate)	1650	Not cumulative	±0.1	4.8	2150		325.0

*Thermal control requirements have not been included

Table 9. Spacecraft Requirements Summary, Reference Mission III

	Electrical Energy (kwh)	Spacecraft Attitude Hold (hr)	Experiment Minimum Attitude Dead Band (deg)	SPS Duration (min)	SM RCS Propellant (lb)	Number of EVA's
Experiments*						
Biomedical/behavioral package	67.2					
Radiation monitor	5.7					
Physical science (0502, 3, and 4)	9.2	33.5	±5		15	7
Conjugate aurora (0801)	2.3	32.3	±0.1		75	
Launch OGO (1101)	0.9	(0.3)	±0.05			None
Subsystem development (1403, 4, 6)	77.4	43.0	±0.1		105	
Experiment support	45.0					
Housekeeping and Transit*						
Probable nonexperiment, requirements	1550.0	2.2	(0.5)	10.3	294 (Orbital maintenance not included)	None
Total (Approximate)	1760.0	110.0	±0.05	10.3	490	7

*Thermal control requirements have not been included

Table 10. Experiment Power Requirements, Reference Mission IV

Item	Power Watts	Power kwh
Mapping Group 1 Day Side		
Photography	100	
Multispectral photography	50	
UV spectrometry	150	
IR surveying	10	
Gamma ray surveying	15	
Gravity surveying	25	
X-ray spectroscopy	5	
Remote geochemical sensing	20	
Passive microwave	200	
Radar altimeter	150	
Total	725	
Mapping Group 2 Night Side		
Radar mapping	1500	
UV spectrometry	150	
IR surveying	10	
Gamma ray surveying	15	
X-ray spectroscopy	5	
Passive microwave	200	
VHF reflectivity	200	
Radar altimeter	150	
Total	2230	
Experiment Group 3 Continuous Operation		
Meteoritic dust	5	
Radiation monitoring	5	
Total	10	
Electrical Energy for Experiments		
SCS electronics	120	25.2
Mapping Group 1	725	60.9
Mapping Group 2	2230	187.3
Experiment Group 3	10	8.6
Tape recorder	31	5.2
Transmission	80	2.7
Photography standby	30	15.1
Total		305.0
Spacecraft orientation and stabilization during mapping: Orientation—Local vertical; Attitude deadband—±1/2° in roll, pitch, and yaw; Attitude rates—<0.01°/second		

Table 11. Mission Experiment Support Requirement Summary

Parameter	Mission Requirements					
	Range	ELIO	EPO (S-IB)	EPO (S-V)	ESO	LPO
Mission duration - days	15 - 35	30	30	30	30	30
Crew size	2 - 3	3	3	3	3	3
Experiments payload (pounds)	1000 - 11,000	3700	7600	7600	6540(1)	1190(2)
Experiments equipment volume (cubic feet)						
Total	20 - 720	130	475	475	294	61(2)
Pressurized	20 - 300	96	72	72	60	30(2)
Number of attitude cycles						
Coarse (±2°)	50 - 1000	360	665	665	64	
Fine (±0.1°)	0 - 400	90	180	180	58	
Attitude hold (hours)						
Coarse (±2°)	30 - 750		524	524	24	360
Fine (±0.1°)	0 - 300		160	160	50	288
Electrical power (kwh)	50 - 2000		590	590	640	1067
ECS/EVA (days/number)	15-45/0-30	30/6	30/0	30/0	30/6	30/0
G&C hours active	3 - 20	6	6	6	4	14
Communications usage - cycles		5330	5330	5330	4410	3460
Data rate (bits/sec)		<Block II	<Block II	<Block II	<Block II	<Block II
Thermal control						
Experiments man-hours	150 - 1080	550	550	550	500	450
Mission success	Open					
Crew safety		0.999	0.999	0.999	0.999	

(1) Flight 516
(2) SID 65-1088
(3) 75 percent utilization (LPO hours = 576)

Table 12. Missions Analyzed for Performance

Orbit	Vehicle	
Earth 200 N.M. 28.5°	Saturn IB	
Earth 200 N.M. 90°	Saturn IB	
Earth 200 N.M. 90°	Saturn IB Rend.	
Earth 200 N.M. 90°	Saturn V	
Earth 19,350 N.M. 0°	Saturn V	
Lunar 80 N.M. Low Incl.	Saturn V	Free Return No Stay Penalty
Lunar 80 N.M. Polar	Saturn V	Free Return No Stay Penalty
With all combinations of Free Return - Non-Free Return Maximum Stay Penalty - No Stay Penalty		

The CSM return weight is the total burnout weight of the vehicle after application of the impulse necessary to return to earth. In the case of earth orbital missions, this is the deorbit impulse; in the case of lunar orbit missions, it is a combination of all impulses necessary to return to earth from orbit around the moon (departure from lunar orbit and mid-course corrections). The laboratory weight consists of all weight either consumed in the operational orbit or jettisoned before return. The CSM propellant weight is the total amount of propellant required in the CSM to achieve the operational orbit and to establish the return trajectory. The launch vehicle injection weight is the total initial weight to which the first service module impulse is applied. This weight does not include adapters jettisoned before SPS ignition. The nominal case indicated in the figures by dotted lines, assumes that a 3800-pound SLA has been jettisoned.

Unused reserve propellant should be included as a component of the CSM return weight, as should any unused consumables. Thus, if the option of returning before depletion of the reserve propellant and the consumables is required, the resulting penalty (in terms of allowable laboratory and equipment weight) can be seen on the graph by tracing the effect of an appropriate increase in CSM weight.

Mathematical Model

The data for the graphs were calculated according to the following procedure (See Figure 39). The arrival weight in the operational orbit was determined for the given values of injection weight and transfer ΔV (from the S-IVB burnout orbit). The final weight in the operational orbit (just before return to earth) was calculated for the given values of return weight and return ΔV. The difference between initial and final weight in orbit is the laboratory weight. In all launches involving a CSM, the SPS was employed to complete the task of achieving the operational orbit after S-IVB burnout. The assumed S-IVB burnout trajectories were those employed in the AAP program, where the mission and launch vehicle for the RCM were the same as those in the AAP study. For polar orbit rendezvous, the flight profile was assumed similar to that of the low inclination rendezvous mission of AAP.

Discussion

For all of the near-earth orbital missions, the CSM propellant requirement is far below the CSM tank capacity. If small tanks could be considered for these cases, a smaller CSM weight could result, and increased laboratory weight could be accommodated. Reduced tank weight data was not generated for use in this analysis. Consequently, the advantages of this approach were not pursued.

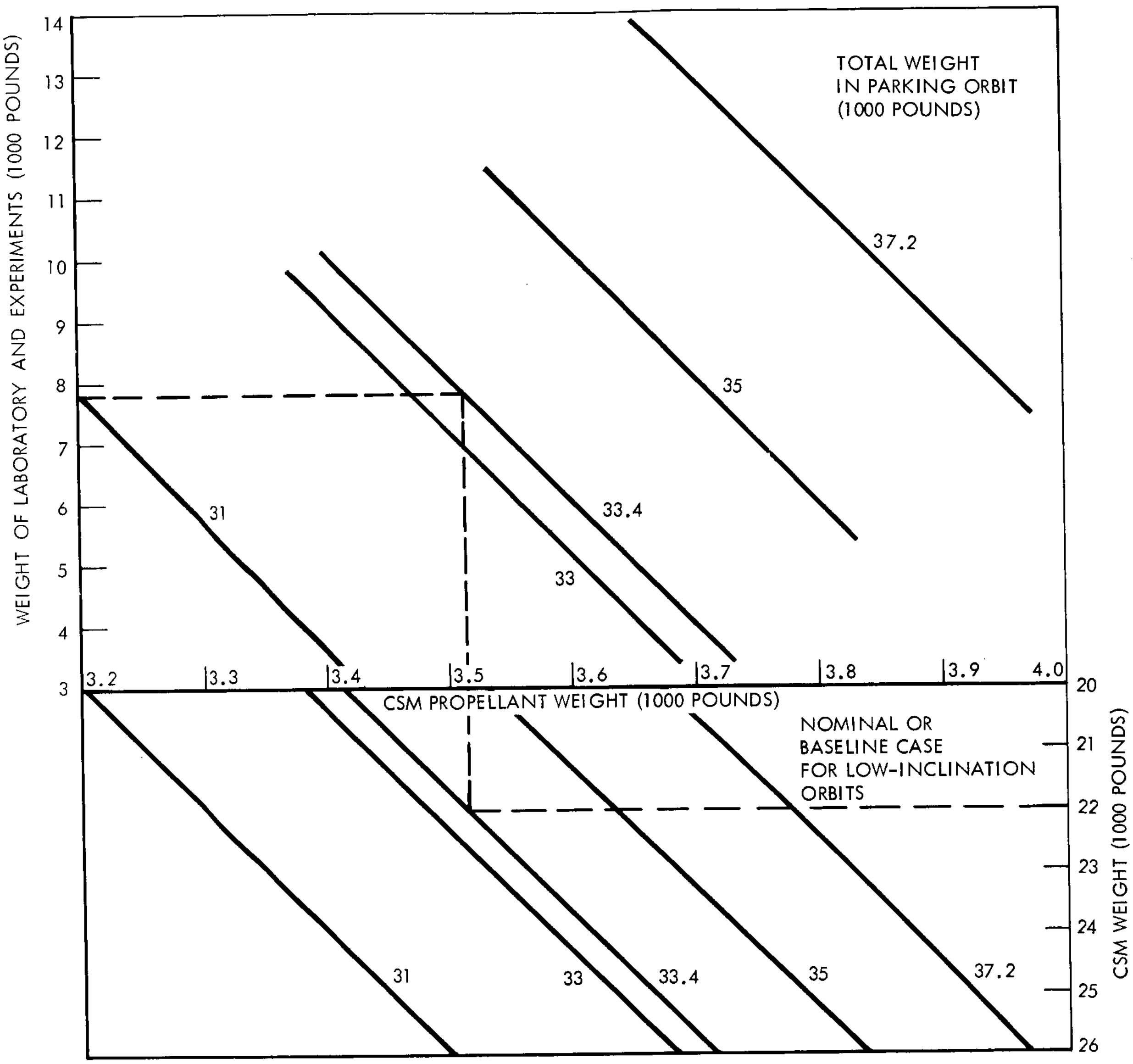
In some of the lunar missions, such as those with free-return capability and freedom to return any time of the month, the CSM propellant requirement exceeds the CSM tank capacity.

The Earth polar orbit launch was found to impose such a penalty as to make use of a single Saturn IB impractical. Approximate calculations show a payload capability (over and above the 22, 000 pounds CSM) of less than 5, 000 pounds and possibly as low as 2, 000 pounds. It was therefore concluded that the polar near-earth missions would have to be accomplished using a Saturn IB rendezvous or a Saturn V.

The Saturn V was found to provide a reasonable laboratory weight in a 24-hour equatorial orbit, providing the CSM is used as a fourth stage. The payload weight thus obtained for the 24-hour satellite mission is conservative in view of the possibility that an inclined 28. 5-degree 24-hour orbit might actually be employed.

Launch Vehicle Capabilities Assumed

In each of the graphs previously described, a nominal case was indicated by a dashed line. The nominal case was selected on the basis of an



NOTES:

1. LAB WEIGHT INCLUDES EXPENDABLES AND ALL PAYLOAD LEFT IN ORBIT.
2. ASSUMPTIONS
   ΔV EXPENDED BY CSM FOR -
   TRANSFER FROM PARKING ORBIT TO 200-NMI CIRCULAR ORBIT = 750 FPS
   REENTRY = 500 FPS
   I_{SP} OF CSM = 320 SECONDS
3. ---NOMINAL CASE FOR THIS STUDY
4. S-IB PAYLOAD INCLUDING SLA CAPABILITIES = 37,200 POUNDS IN 28.5-DEGREE ORBIT -100 NMI
   31,000 POUNDS IN 90-DEGREE ORBIT -100 NMI

Figure 25. Weight of Laboratory and CSM Versus Weight of CSM Propellant for Transfer From Parking Orbit to 200-Nautical-Mile Earth Orbit

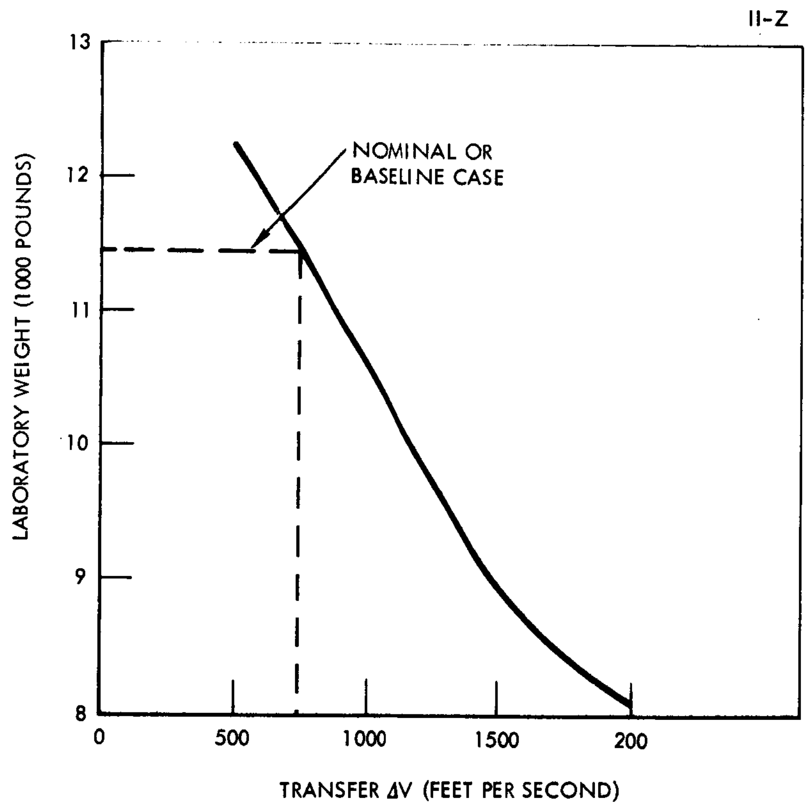


Figure 26. Sensitivity of Laboratory Weight to ΔV Required for CSM Transfer

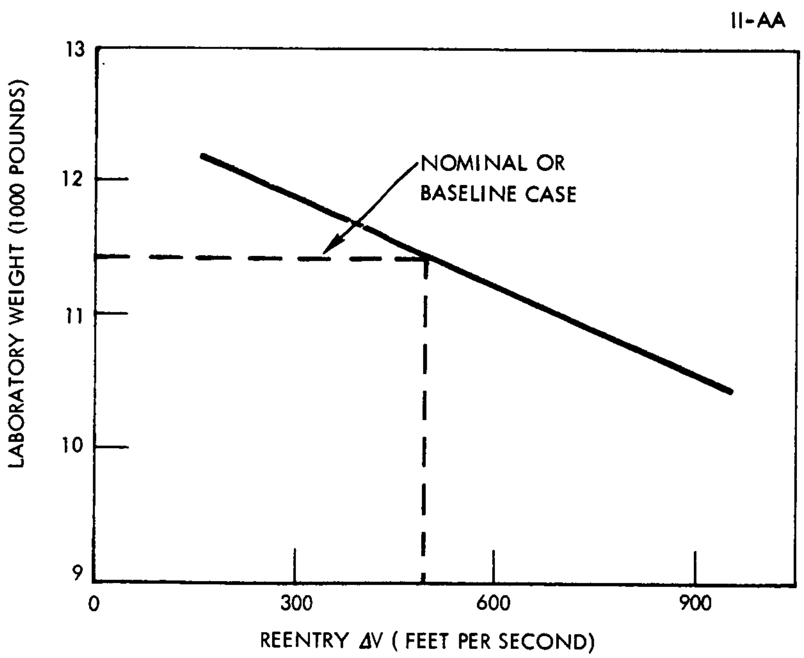


Figure 27. Sensitivity of Laboratory Weight to ΔV Required for CSM Reentry

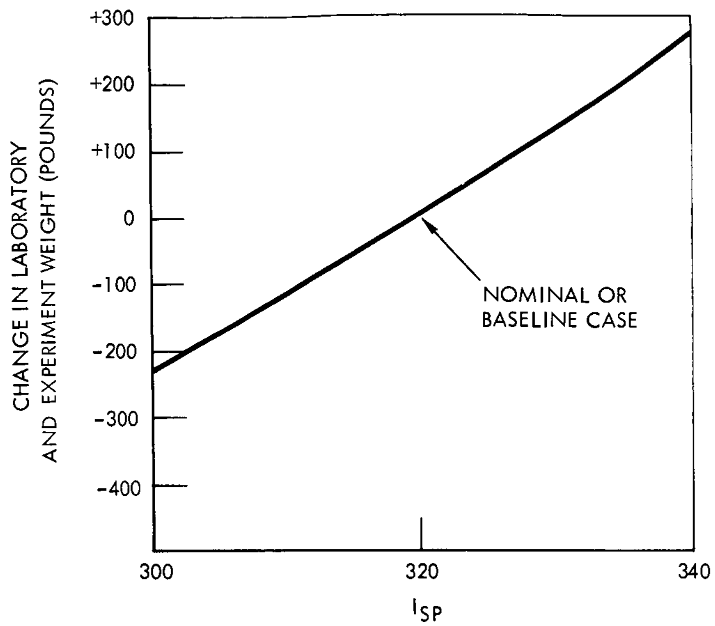


Figure 28. Sensitivity of Laboratory Weight to Service Module I_{SP}

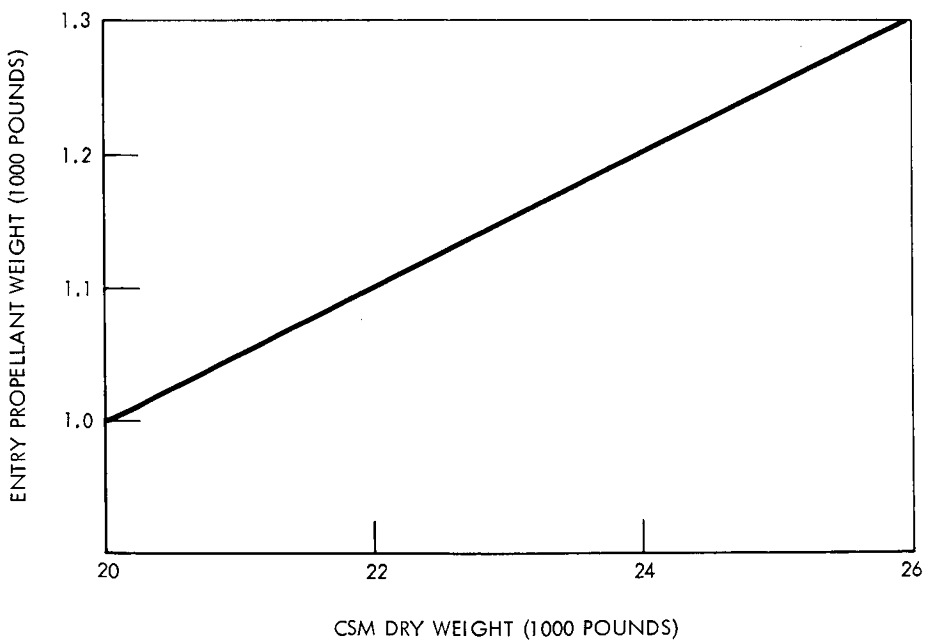


Figure 29. Entry Propellant Requirements for 200-Nautical-Mile Earth Orbital Missions

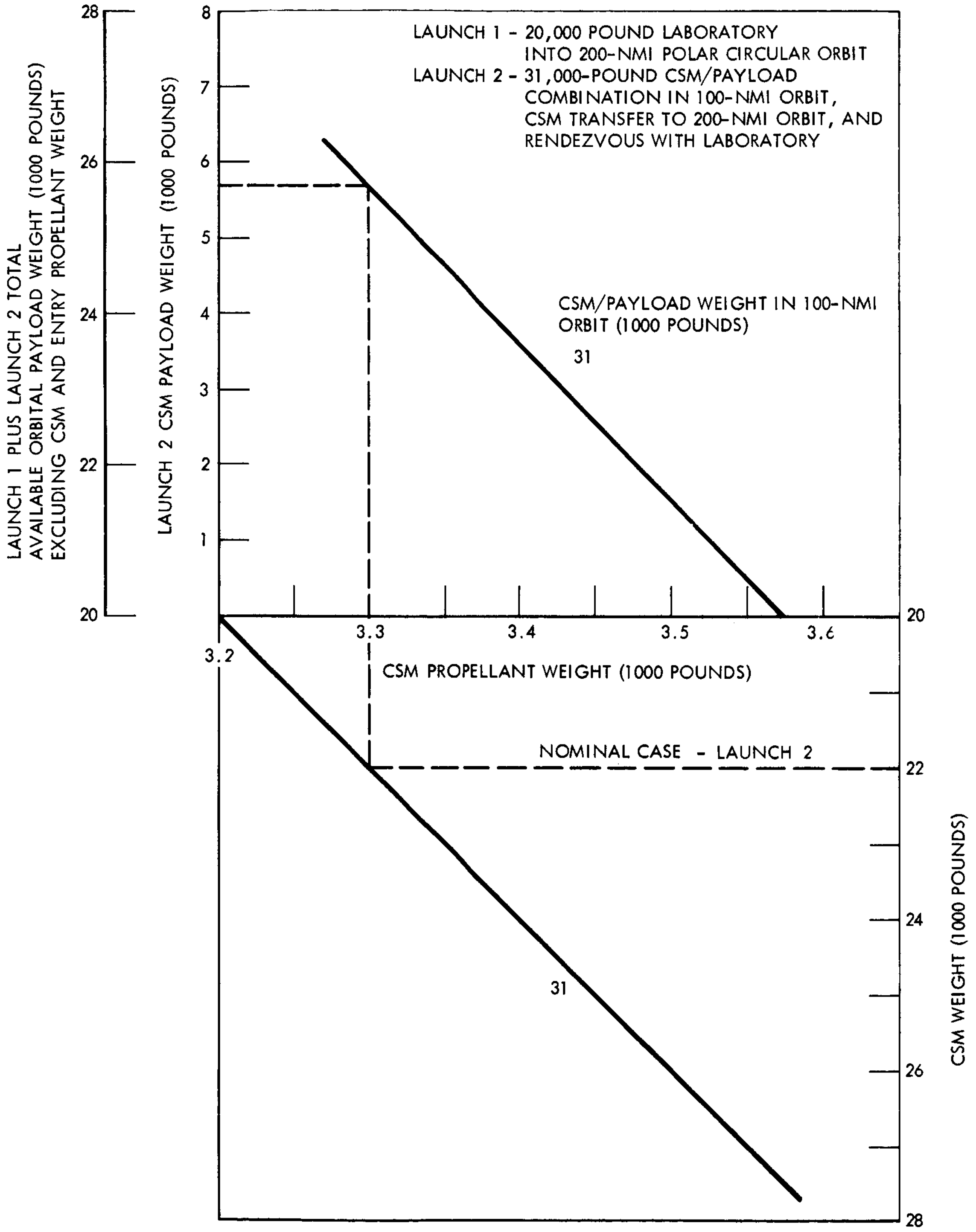


Figure 30. Orbital Weights for S-IB Polar Orbit Rendezvous Mission

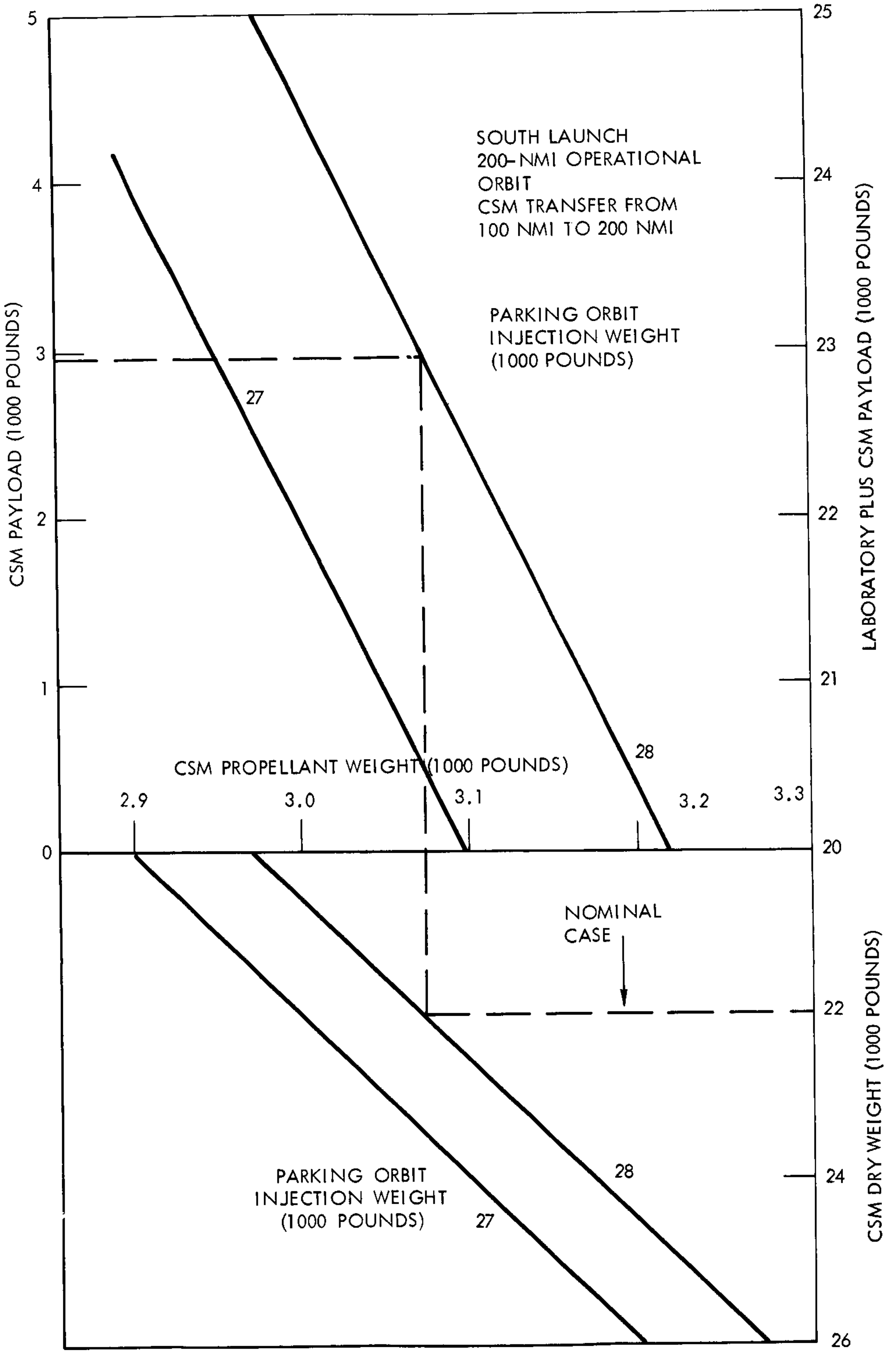


Figure 31. Orbital Weights for S-IB Polar Orbit Rendezvous Mission

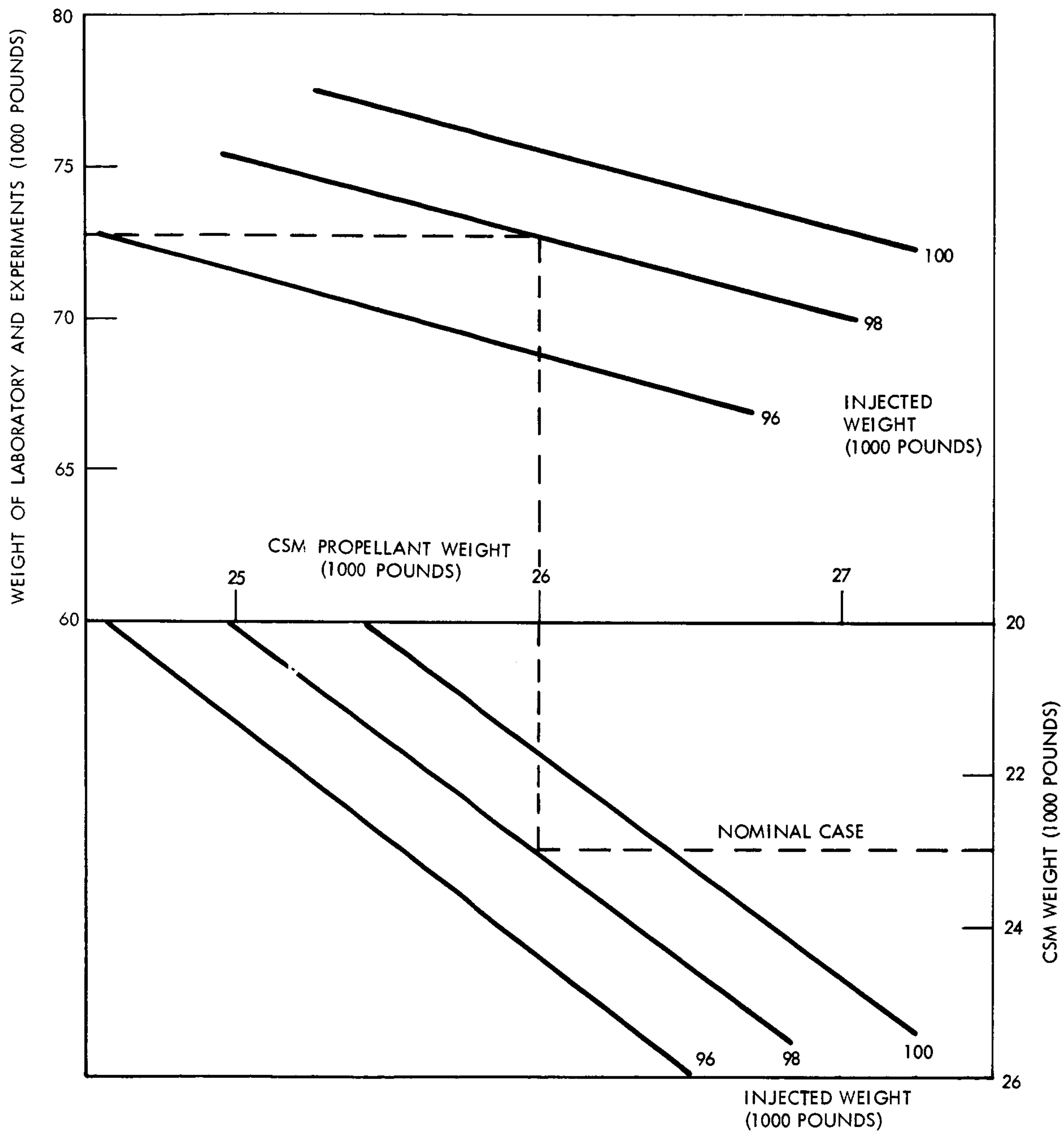


Figure 32. Orbital Weights for Saturn V Polar Orbits

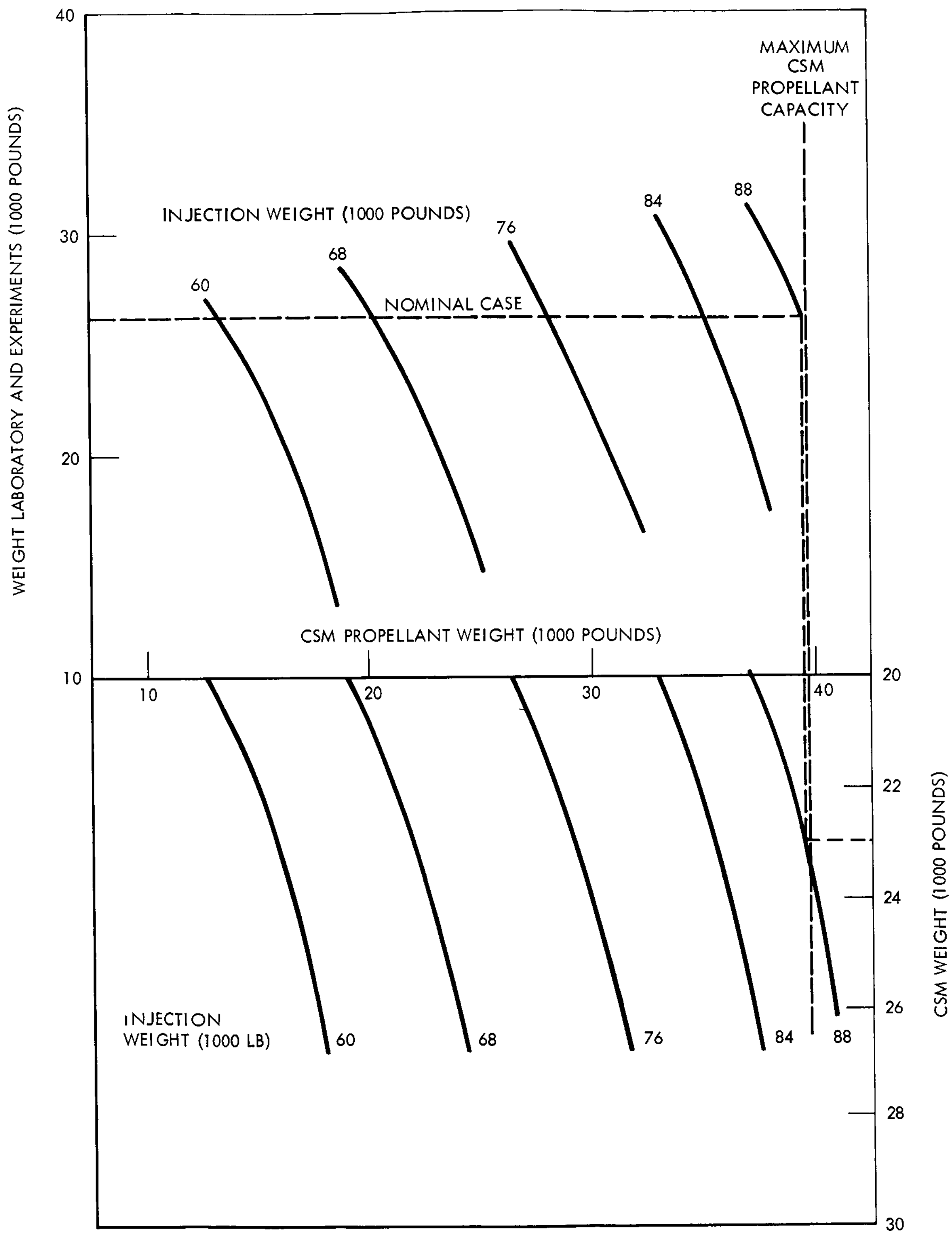
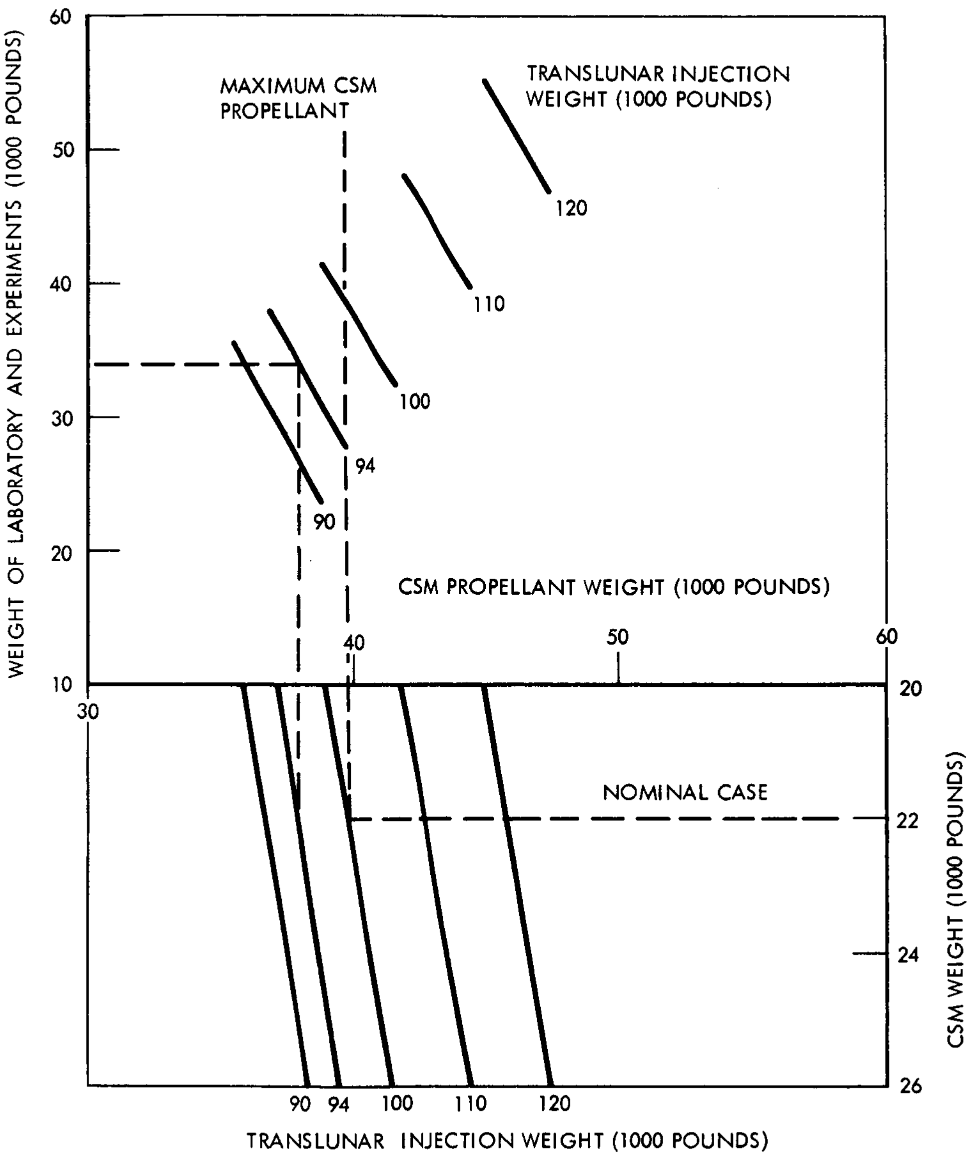


Figure 33. Weight of Laboratory and CSM Versus Weight of CSM Propellant for 24-Hour Synchronous Orbit Mission



TRAN'SLUNAR ΔV = 3700 FT/SEC (FREE RETURN)
TRANSEARTH ΔV = 3700 FT/SEC (NO STAY PENALTY)
STAY PENALTY BASED ON POLAR LUNAR ORBIT
NOMINAL CASE: CSM WEIGHT = 22,000 POUNDS, TLI WEIGHT = 94,000 POUNDS

Figure 34. Weight of Laboratory and CSM Versus Weight of CSM Propellant for Lunar-Orbit Mission

TRANSLUNAR ΔV = 3700 FT/SEC (FREE RETURN)
TRANSEARTH ΔV = 6400 FT/SEC (MAX STAY PENALTY)
STAY PENALTY BASED ON POLAR LUNAR ORBIT
NOMINAL CASE: CSM WEIGHT = 22,000 POUNDS TLI WEIGHT = 94,000 POUNDS

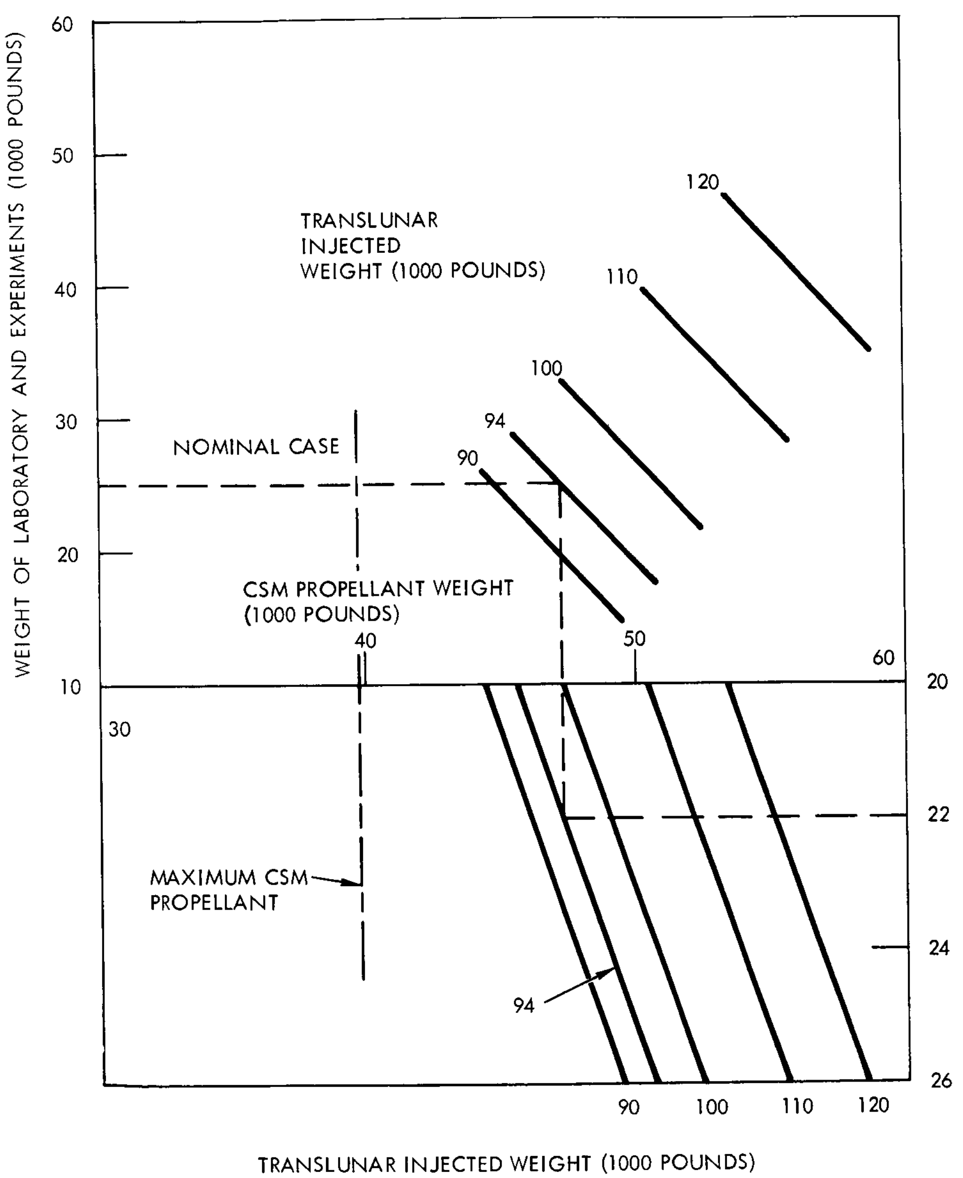


Figure 35. Weight of Laboratory and CSM Versus Weight of CSM Propellant for Lunar-Orbit Mission

TRANSLUNAR ΔV = 2900 FT/SEC (NON-FREE-RETURN)
TRANSEARTH ΔV = 3700 FT/SEC (NO STAY PENALTY)
STAY PENALTY BASED ON POLAR LUNAR ORBIT
NOMINAL CASE: CSM WEIGHT = 22,000 POUNDS, TL1 WEIGHT = 94,000 POUNDS

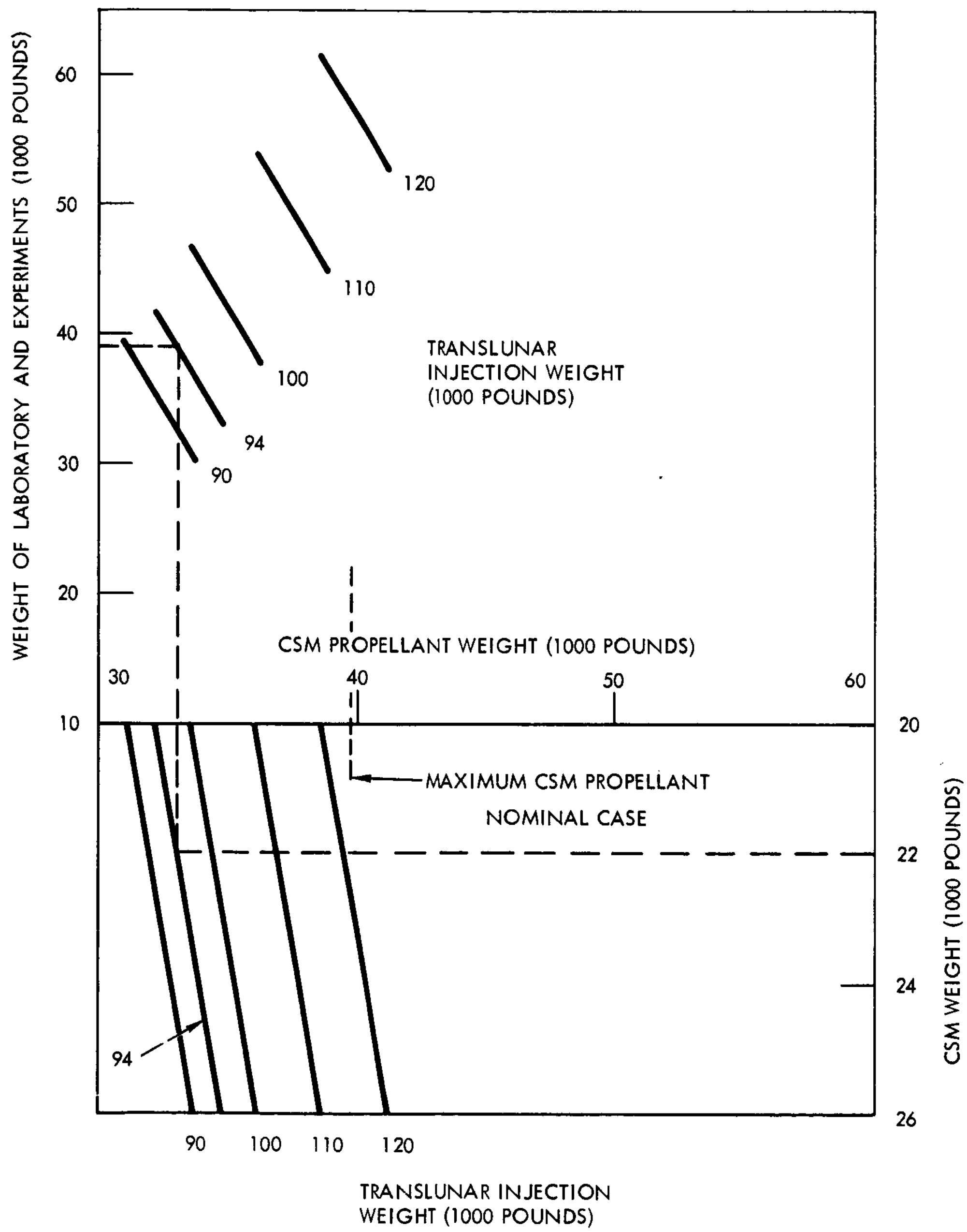
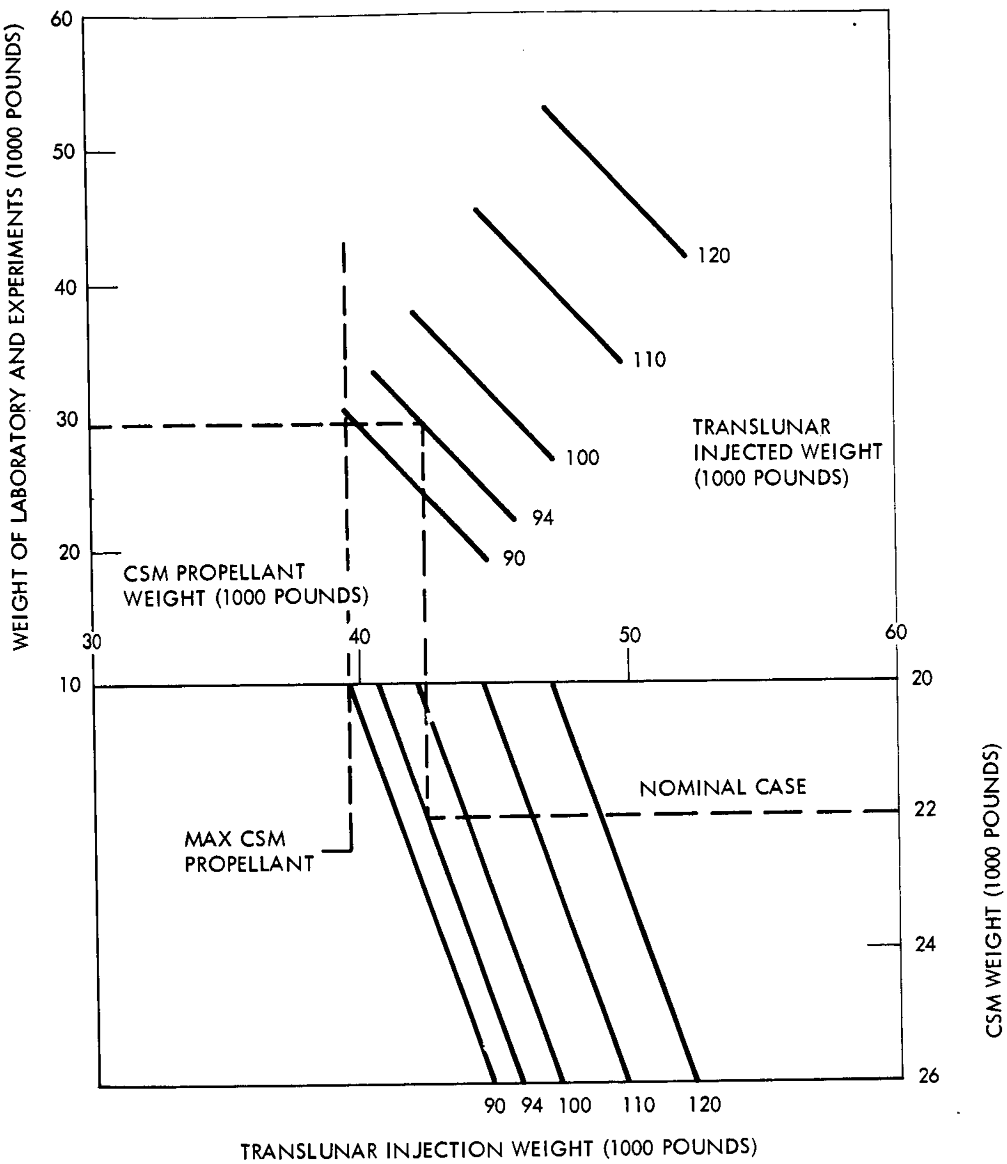


Figure 36. Weight of Laboratory and CSM Versus Weight of CSM Propellant for Lunar-Orbit Mission



TRANSLUNAR ΔV = 2900 FT/SEC (NON-FREE RETURN)

TRANSEARTH ΔV = 6400 FT/SEC (MAX STAY PENALTY)

STAY PENALTY BASED ON POLAR LUNAR ORBIT

NOMINAL CASE: CSM WEIGHT = 22,000 POUNDS, TLI WEIGHT = 94,000 POUNDS

Figure 37. Weight of Laboratory and CSM Versus Weight of CSM Propellant for Lunar-Orbit Mission

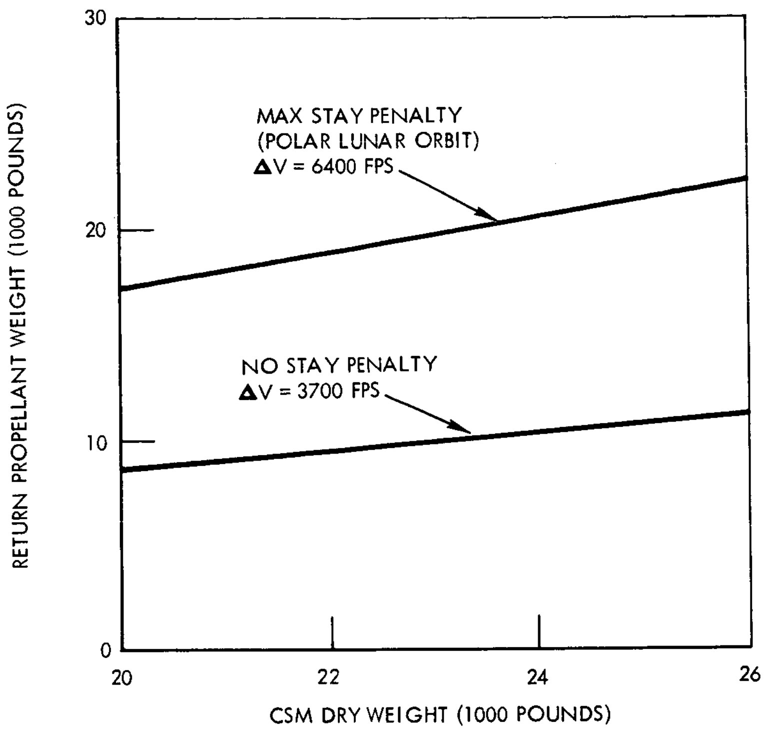
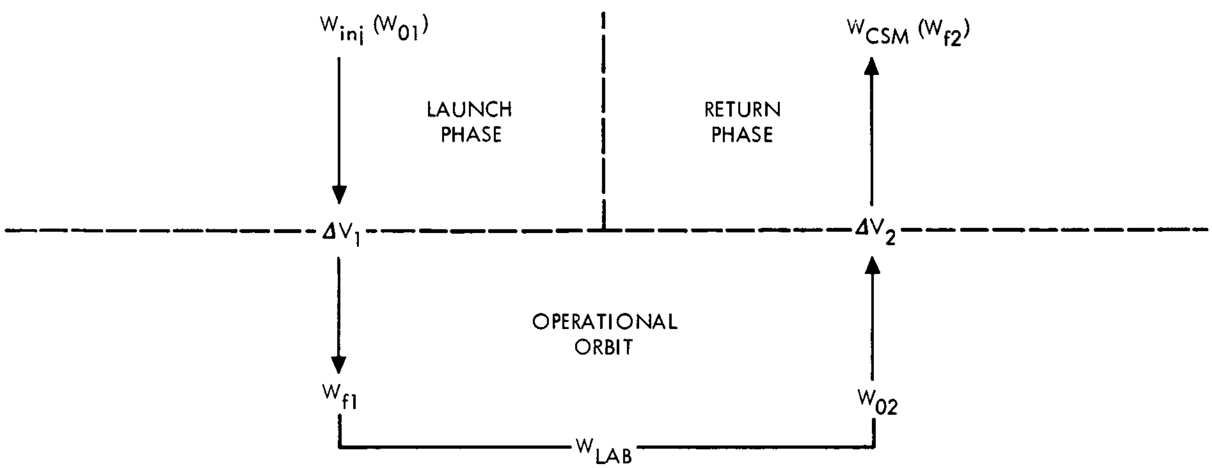


Figure 38. Return Transearth Propellant Requirements for Lunar-Orbit Missions



$$W_{f1} = f(W_{inj}, \Delta V_1, I_{SP})$$
$$W_{02} = f(W_{CSM}, \Delta V_2, I_{SP})$$
$$W_{LAB} = W_{f1} - W_{02}$$

ΔV_1 IS THE SM IMPULSE REQUIRED TO ACHIEVE THE OPERATIONAL ORBIT.

ΔV_2 IS THE SM IMPULSE REQUIRED TO RETURN TO EARTH.

Figure 39. Model for Calculating Allowable RCM Laboratory Weights

estimated launch vehicle capability and an unloaded CSM (containing no propellant, no expendables, but containing three astronauts and the equipment necessary to return them safely to earth).

A CSM weight of 22,000 pounds was selected as an optimistic nominal value. The nominal injection weight capability of the launch vehicle for each mission was determined as described below.

The S-IB injection capability in a 100-nautical-mile circular orbit, taken from Flight 214/215 Mission Description (SID 65-1727), was used as a basis for the tradeoff curves shown in Figure 25 for 200-nautical-mile earth low-inclination orbits.

S-V capability for injection into an intermediate orbit as in the Flight 509 Mission Description (SID 65-1725) was assumed. A laboratory weight tradeoff curve was extracted, without change, from this document and cross-plotted for convenience for a 24-hour earth-synchronous orbit.

The Saturn V vehicle was assumed to perform the injection of the CSM laboratory into a trajectory toward the moon. An injection weight range, including the Apollo value and reasonable upratings, was assumed (90,000 to 120,000 pounds). The injection weight of 94,000 pounds was taken as a nominal value (SID 65-1547, Performance Analysis, Phase II Flights).

The 200-nautical-mile earth-polar-orbit Mission was analyzed as a single Saturn IB launch, a single Saturn V launch, and a rendezvous (dual launch) using Saturn IB's. With the AAP value of Saturn IB injection weight capability for a low-inclination, low-altitude orbit as a reference, a direct South launch capability for a single Saturn IB launch was determined by making an adjustment indicated in the Douglas Saturn IB Users Handbook (Figure 40).

The rendezvous curve presented has two scales: one shows the CSM payload alone; the other shows the total of CSM payload and laboratory payload after rendezvous.

The Saturn V mission curves represent an extension of data from AAP mission 507 (SID 65-1724, page 76).

Comparison of RCM and AAP Performance Studies

The AAP data contained a CSM weight tradeoff curve for only the 24-hour synchronous mission. Therefore, these curves had to be generated for the other missions. The AAP data had been obtained by computer calculation, and the extension of this information by means of additional computer runs seemed an unnecessary expense. Therefore, analytical calculations, based on the simplified model described previously, were employed.

The RCM missions also differed sufficiently from the AAP missions, in some cases, to cause a significant departure from the payload weights that were included in the AAP reports. Following are some of the mission differences:

The payload capabilities associated with a 180-degree launch azimuth from Cape Kennedy were required in this study but had not been determined in the AAP program. (These are shown in Figures 40 and 41 for Saturn IB and Saturn V, respectively.) Also, a polar orbit rendezvous mode was required by the RCM project when the Saturn IB payload in a polar earth orbit was found to be so small (less than 5000 pounds). The AAP program had not included a polar orbit rendezvous mission. Another deviation from the AAP approach was necessitated by the fact that the AAP polar mission (Flight 211) with a single Saturn IB did not include a laboratory, and involved the ignition of the SPS before the attainment of a parking orbit. This mode of operation precludes a transposition and docking operation and requires that the CSM take over the task of ascent guidance from the S-IVB. These conditions were not acceptable for the RCM mission, and a flight mode similar to that of the low-inclination missions was adopted.

Explanation of Terms

The lunar mission graphs relate to both of the two basic types of transfer trajectory for lunar orbit missions, free return (or circumlunar) and non-free return. In the former, the trajectory toward the moon is such that the vehicle eventually returns to earth if retrofiring at the moon is not performed. In the latter, the vehicle may be captured by the moon. Generally the free return trajectories are faster and therefore require more energy for injection and for retrofire than the non-free return trajectories require.

An orbit about the moon remains fixed in space, except for perturbations. Therefore, the angle between the orbit plane and the optimum plane for return to earth is constantly changing. The energy required for returning is strongly affected by the magnitude of this angle. Consequently, one must accept the maximum energy penalty for the privilege of returning at any time during the stay in lunar orbit. The magnitude of the maximum stay penalty depends upon the inclination of the orbit, and is worst for a polar orbit. Performance analyses in the AAP and RCM studies have included the extreme conditions of zero-stay penalty associated with rigidly scheduled departures from lunar orbits and the maximum penalty associated with complete flexibility of departure from a polar lunar orbit. The stay time penalty for the lunar polar orbit goes through zero every 14 days after achievement of the lunar orbit.

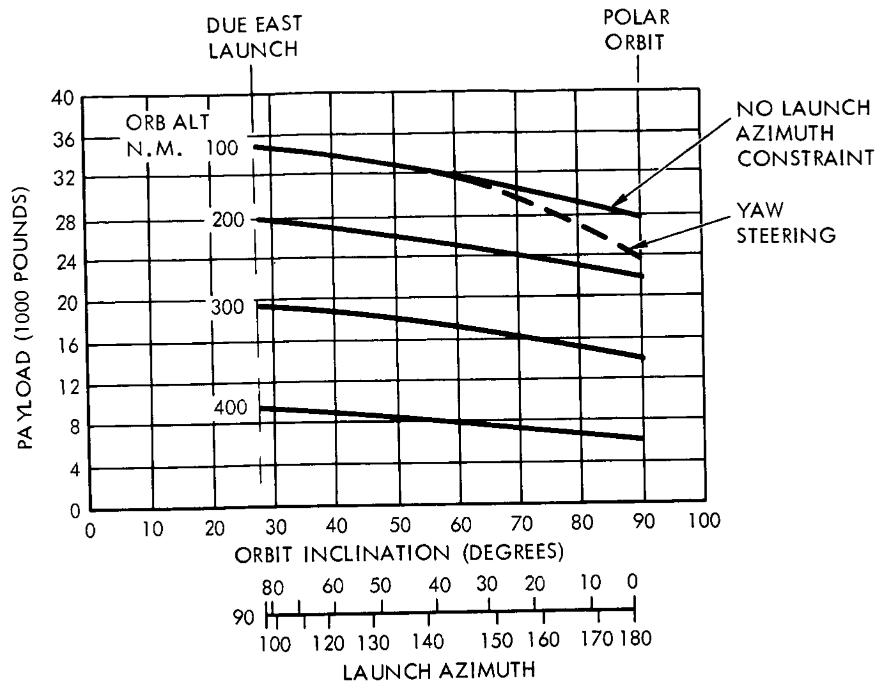


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Figure 40. Saturn IB Payload Versus Orbit Inclination

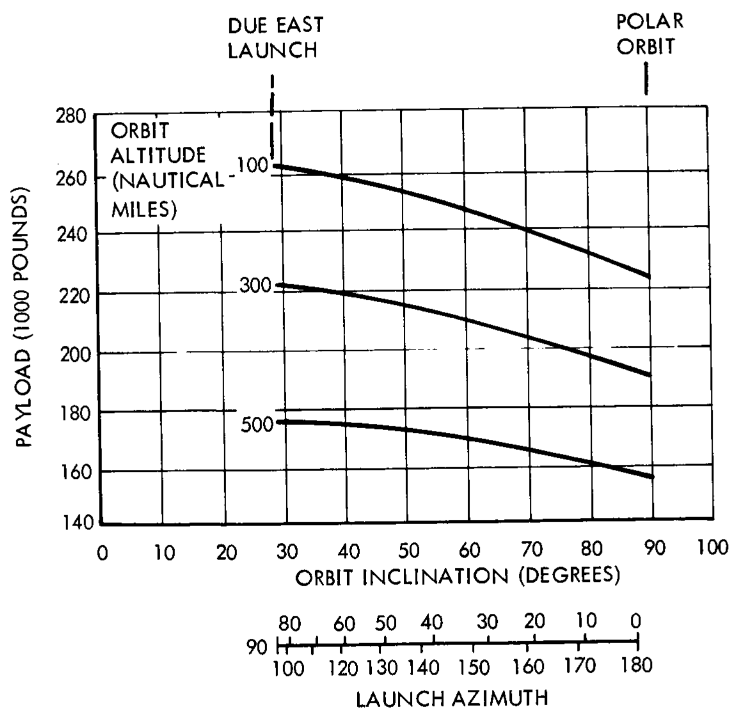

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Figure 41. Saturn V Payload Versus Orbit Inclination

Summary of RCM Laboratory Reference Missions Capabilities

The RCM laboratory AAP mission capabilities have have been examined and analyzed during this study on the basis of four postulated AAP Reference Mission Profiles and various RCM Lab Experiment Payload configurations. The four Reference Mission Flight Profiles selected are -

1. Low-altitude, low-inclination earth orbit
   Launch vehicle, Saturn IB
   Parking orbit - 80 x 200 nautical miles
   Operational orbit - 200 nautical miles altitude, 28.5 degree inclination
   Orbit spacecraft configuration - CSM + RCM laboratory + experiments

2. Low-altitude, polar earth orbit
   Dual launch, Saturn IB
   Launch site - KSC
   First launch - RCM laboratory + experiments, unmanned
   Operational orbit - 200 nauticalmiles direct injection, with yaw steering during ascent
   Second launch - CSM + experiments
   Parking orbit - 100 nautical miles, yaw steering + SM assist
   Operational orbit - 200 nautical miles rendezvous with RCM laboratory
   Alternate launch, Saturn V
   Parking orbit - 80 x 200 nautical miles yaw steering
   Operational orbit - 200 nautical miles

3. Synchronous equatorial earth orbit
   Launch vehicle - Saturn V
   Parking orbit - 100 nautical miles
   Operational orbit - 19, 300 nautical miles altitude, 0 degree inclination
   Orbit spacecraft configuration - CSM + RCM laboratory and experiments

4. Lunar, polar orbit
   Launch vehicle - Saturn V
   Profile similar to Apollo

The RCM laboratory system AAP mission performance capabilities are illustrated in Table 13 and Figure 42.

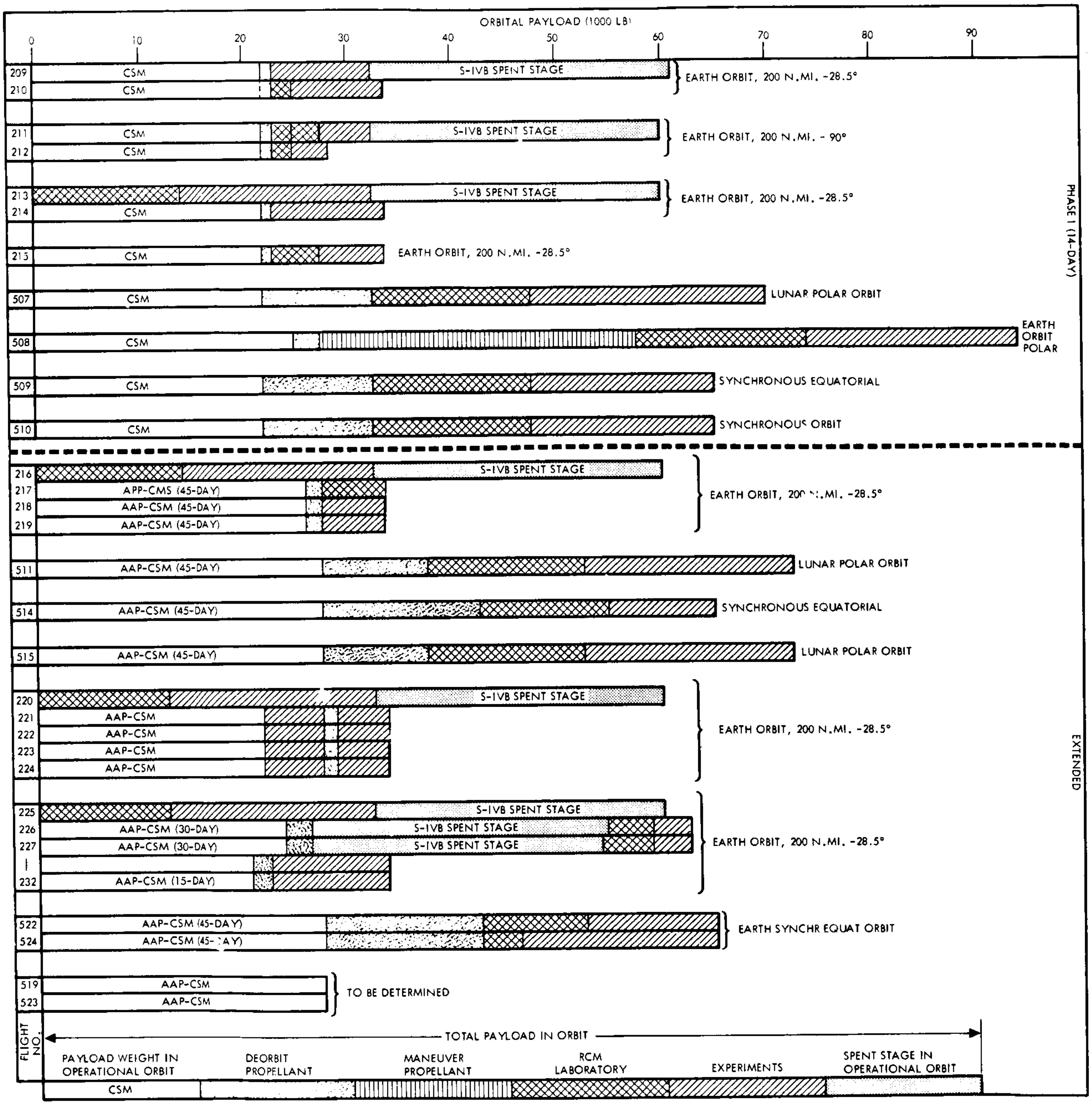

Figure 42. Apollo/Saturn Flights, RCML System Mission Performance Capabilities

Table 13. RCM Laboratory Reference-Mission Capabilities

		30-Day Earth Orbit 200 NMI 28.5 Deg	30-Day Earth Orbit 200 NMI 90 Deg		30-Day Earth Orbit 200 NMI 90 Deg	30-Day Earth Orbit 19,300 NMI 0 Deg	30-Day Lunar Orbit 90 Deg
Launch Vehicle		S-IB	Dual Launch S-IB		S-V	S-V	S-V
Spacecraft Flight Configuration	CSM (30-day) and deorbit propellants	25,661	25,661	—	25,661	40,661	35,661
Spacecraft Flight Configuration	RCM laboratory (dependent)	2,696	—	2,696	2,696	2,696	2,696
Spacecraft Flight Configuration	RCM laboratory (30-day) fully independent	13,691	—	13,691	13,691	13,691	13,691
Spacecraft Flight Configuration	Injected payload to operational orbit	33,100	27,000	18,000	96,300	62,000	71,000
Spacecraft Flight Configuration	Available experiment payload max	4,743	1,339	15,304	67,943	18,643	32,643
Spacecraft Flight Configuration	Available experiment payload min	—	—	4,309	56,948	7,648	31,648
AAP range of experiment payloads		1,000 to 4,500	1,000 to 4,500		4,500 to 15,000	5,000 to 20,000	4,500 to 32,000
Remarks		Independent laboratory plus experiment may be launched unmanned	Launch from KSC requires yaw steering		Maneuvering SPS propellant may be substituted for excess experiment payload		Based on non-free return, no stay penalty flight trajectory

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III. SYSTEM ANALYSIS

Because of the diversified mission requirements, a range of system requirements and operational configurations is necessary. The baseline RCM study configurations are presented in Table 14.

Initially, the system analysis identifies these modifications and additions necessary to convert both Block I and Block II CM's into a basic RCM laboratory. The dependent laboratory operates in conjunction with a Command and Service Module (CSM) which supplies the desired active environmental control, electrical power, attitude control, communication, etc. while docked to the laboratory in the orbital configuration.

The second laboratory system baseline selected represents the fully independent RCM laboratory capable of independent operation and experiment support. The independency is derived from the addition of subsystems to the basic RCM laboratory by selection of the subsystems incremental buildup blocks from the "shopping list" of the subsystem capability deltas. This approach to the selection of a particular RCM laboratory configuration— that may range from the basic to the fully independent system configuration— is shown in Figure 43.

The determination and definition of the subsystems building blocks and the shopping list of capability deltas (incremental steps) forms the essential central part in the analysis to identify logical RCM laboratory configurations that appear most suitable or mandatory to satisfy the requirements of the various AAP missions.

Since the AAP experiments were not specifically identified for inclusion in this study, not all of the suitable RCM laboratory system configurations could be considered. Consequently the system analysis effort was directed toward the construction of the methodological model, workable concepts of system integration, and analysis of the fundamental RCM laboratory configuration.

RCM LABORATORY SYSTEM ANALYSIS

The system analysis is based on the following definition of selected major RCM laboratory system configurations:

Table 14. RCM Baseline Systems - Configurations

	Basic RCM Lab Dependent	RCM Lab (30-Day) Fully Independent	RCM Spacecraft (14 Day)
CM inner structure	X	X	X
CM secondary structure	X	X	X
CM heatshield			X
Airlock	X	X	
Micrometeroid shield	X	X	
Thermal insulation	X	X	
Docking drogue	X	X	
Docking probe			X
Laboratory support structure	X	X	
CM environmental control system		X	X
CM electrical power system		X	X
CM reaction control system			X
G&N			X
Stabilization and control system		X	X
Crew systems			X
Earth landing system			X
Life support system		X	X
Data and communication		X	X
Contr and displays		X	X
SM reaction control system		X	
SM electrical power system		X	
SM environmental control system		X	
SM cryogenics		X	
SM service propulsion system			
SM data and communication		X	
Total weight (lb)	2,696	13,691	11,233

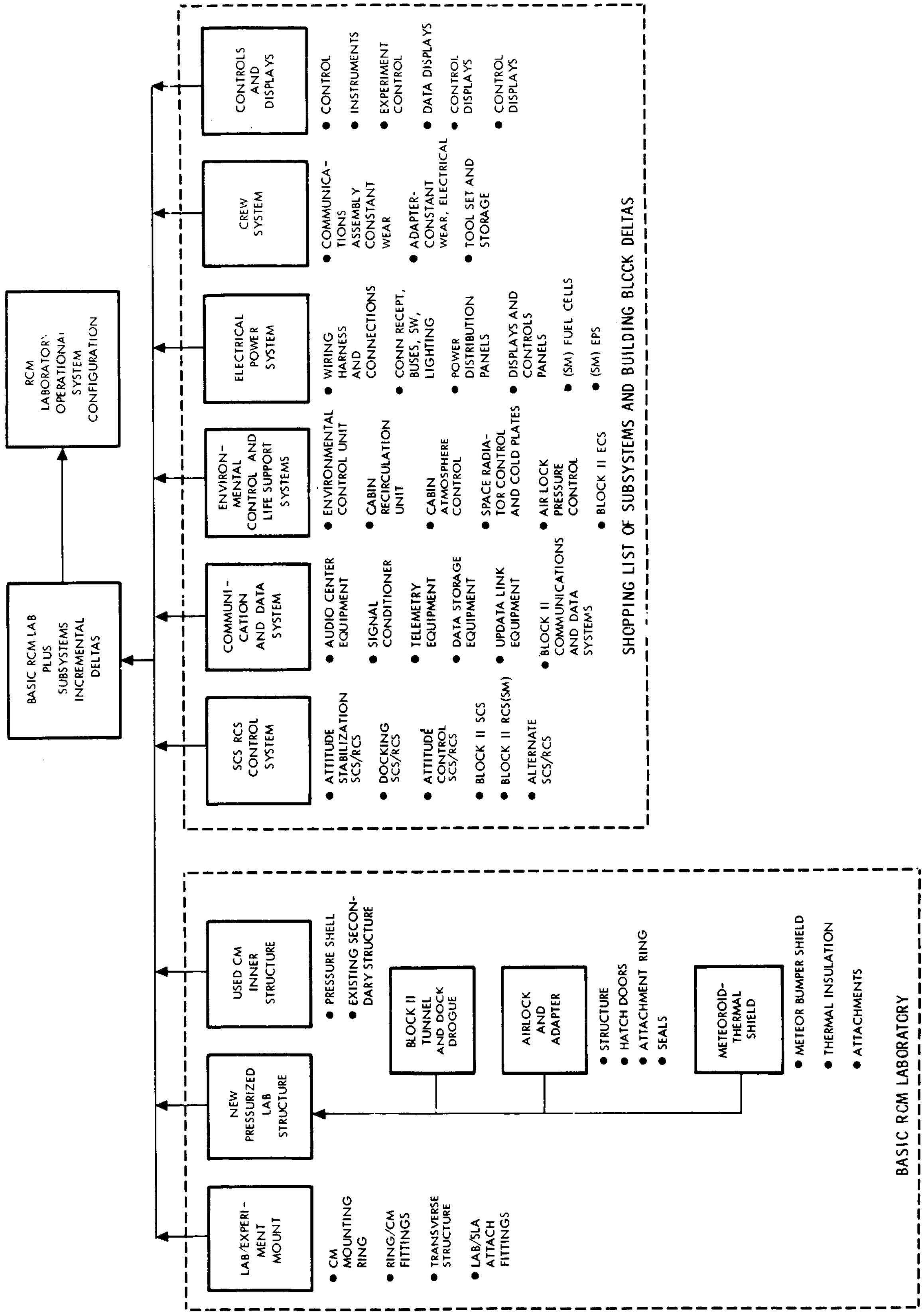


Figure 43. RCM Laboratory System Configuration

Basic RCM Laboratory

The basic (dependent) laboratory consists of the pressure vessel (inner structure), environmental protection (radiation, meteoroid and passive thermal control), a LM docking structure, airlock, basic instrumentation, portions of inner secondary structure and the laboratory support mounting structure.

Fully Independent RCM Laboratory

Major emphasis is placed on the fully independent RCM laboratory and the subsystems that may be added to its configuration. The subsystems are defined in modular building blocks and include at least the following:

Renovated Apollo CM Subsystems Mounted in the Interior of the RCM Laboratory

1. Stabilization and Control
2. Communication and Data
3. Intercom (hardline-LM interface)
4. ECS/LSS
5. EPS
6. Controls and Displays

New Apollo Block II or SM Components Mounted on the Exterior of the RCM Laboratory

1. RCS
2. EPS
3. ECS
4. Cryogenics
5. Consumables
6. Communication and Data

In this RCM laboratory configuration, the shopping list of subsystem building blocks is comprised of the renovated Apollo CM subsystems and new AAP or Block II SM parts and components. Other alternate subsystems, such as LEM and different space-qualified components, are included in the subsystems shopping list when appropriate. These are not being used for evaluation of the laboratory configurations listed.

The system analysis of the different system configurations has a double purpose, first to convert the mission-oriented system requirements into functional system performance requirements, and to determine the specific system configuration performance capability to support the functional requirements to accomplish the mission experimental objectives. Based on these results, the laboratory system configuration mission effectiveness was determined using the an lytic model illustrated in Figure 44.

The conversion of the mission-oriented system requirements into functional system performance requirements has been essentially performed by the analysis of mission performance requirements discussed in Section . The resulting ranges of the functional performance requirements and the requirements necessary to support the experimental objectives of the selected AAP reference missions are summarized in Table 7.

The major task remaining for the analysis effort is the determination and evaluation of the functional performance capabilities of the corresponding laboratory system configurations. The evaluation of the effect that the system capabilities have on support and performance of the mission and experiment payload requirements is the subject of a separate mission effectiveness analysis discussed in a later section of this report.

The RCM laboratory system configuration capabilities evaluation and analysis was conducted by the method outlined in Figure 45, which consists of the basic elements representing the combined system operation of the basic RCM laboratory in conjunction with the CSM and building block deltas selected from the subsystems shopping list. The three selected baseline RCM laboratory configurations are included as representative examples.

Figure 45 includes a postulated First (minimum) Delta RCM laboratory configuration. This contains the minimum possible subsystem building blocks added to the basic configuration, satisfying minimum housekeeping requirements (distribution and control of electrical power, communication and data, controls and displays, and some portions of the environmental

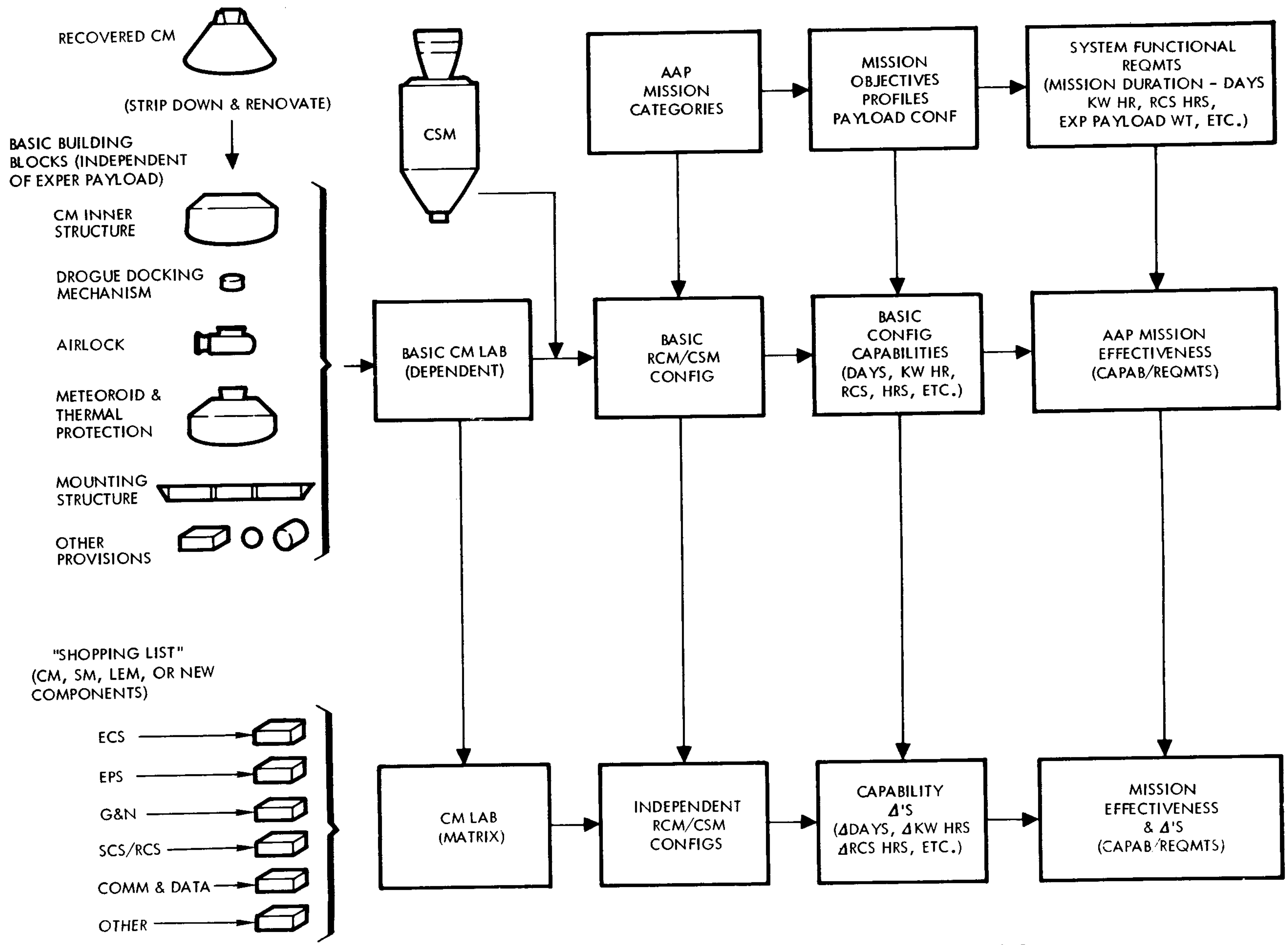


Figure 44. RCM Laboratory System Effectiveness Model

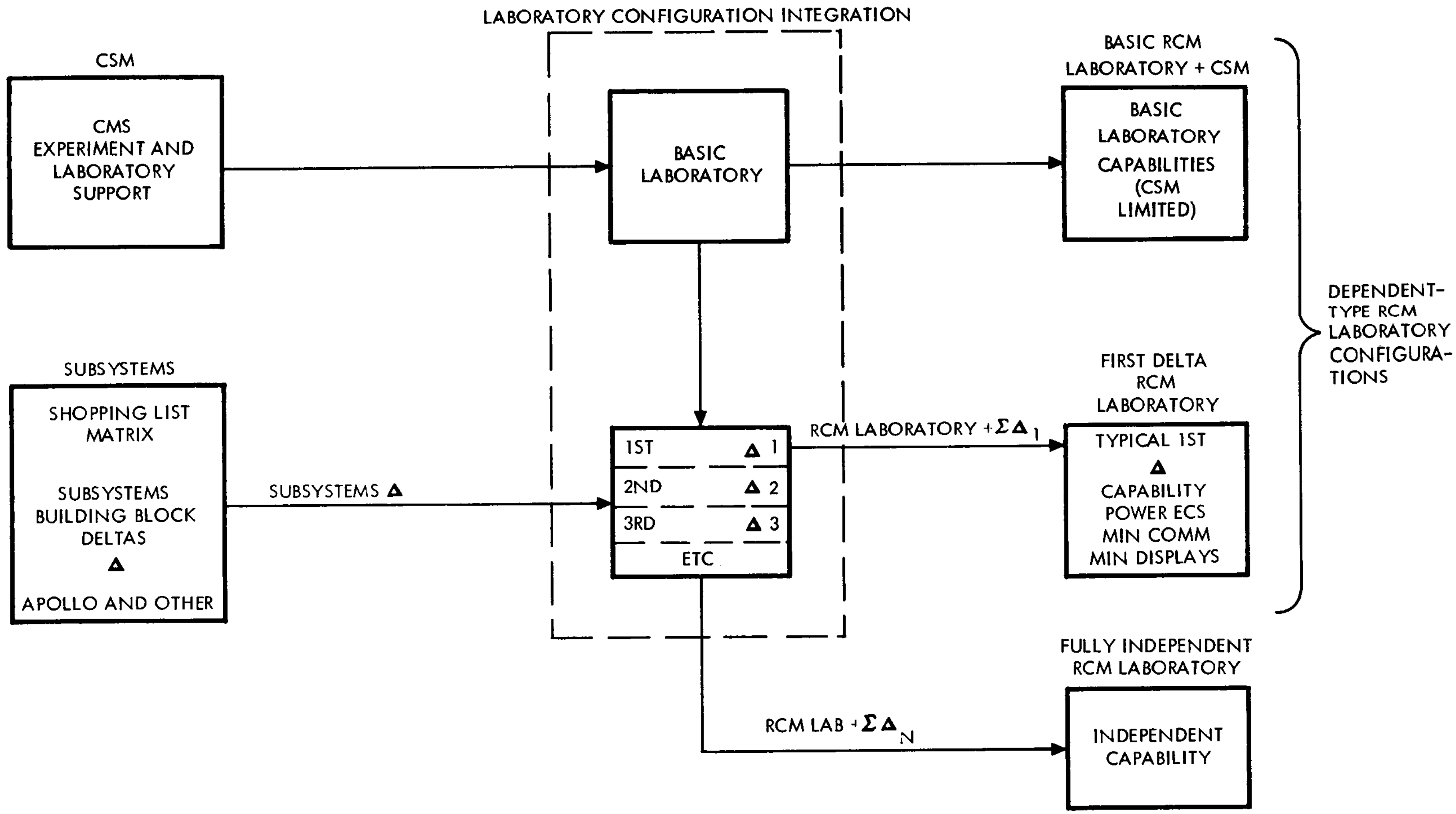


Figure 45. System Configuration Capability

control system required to maintain crew safety). This configuration, however, remains fully dependent upon the CSM for support, similar to the basic RCM laboratory configuration. The First Delta RCM laboratory system configuration illustrates, by example, the approach that is utilized in integrating the possible RCM laboratory configurations from the basic RCM laboratory by addition of subsystem building blocks.

SYSTEM PARAMETERS

The purpose of the system analysis is to evaluate the capability of the baseline configurations to meet the reference mission performance requirements, and to provide guides to assist in determining configuration or mission operation modifications that will assure an optimum AAP program. The principal facets to be considered are the configuration performance capability (its ability to accommodate needed equipments and consumables, to return payloads, and to meet other direct-support requirements for attitude holds, power, etc.) and the probability aspects of mission performance (the ability of the subsystems and total system to perform reliably for the required mission duration and to provide safe crew return.)

Configuration Performance Capability

The reference mission requirements are defined in terms of expendables required for mission accomplishment; weights and space for experimental equipments and return payloads; astronaut time required for accomplishing experiments and tests; requirements for spacecraft pointing for communications, mapping and other operations requiring sensor pointing; spacecraft thermal control; navigation, guidance, and trajectory requirements.

Mission Reliability

Mission reliability defines the expected probability that the mission can continue for the planned duration or for some period less than the planned duration. The factors considered include malfunctions or failures of CSM subsystems that require abort or alternative mission. The principal factor that may shorten the duration of the mission is crew safety. The crew safety requirement for AAP missions is the same as for Apollo Block II lunar missions (0.999).

Achievement of Mission Objectives

The ability to accomplish the mission objectives is primarily an appraisal of the total system capability for accomplishing the mission experimental and test objectives. The detailed experimental requirements, the state of the art and performance reliability of the experimental and test equipments, and the ability of the subsystems to meet support requirements are considered when this appraisal is made. This appraisal also includes such factors as the effects of excessive levels of radiation, meteoroid activity that can terminate or temporarily disrupt the mission, and astronaut sickness or other constraints.

In the case of the Apollo Block II lunar missions, the basic objectives are to go to the moon and return safely to earth. Even though many detailed test and scientific objectives may be defined, given the accomplishment of the basic objectives the mission will be considered a success. In the case of the AAP missions, the basic objectives are to perform specific experiments and tests in various earth and lunar orbits.

Mission Planning Flexibility

Mission flexibility is a less tangible factor in regard to both preflight and in-flight mission planning. The high costs of the missions require careful planning and replanning of each flight to achieve maximum useful information from each flight and from the AAP as a whole. Preflight flexibility— the ability to modify planning factors such as consumables, flight trajectories, and mission duration, and to change experimental equipments— is important in achieving the most effective overall program.

DEPENDENT RCM LABORATORY SYSTEM CAPABILITY

The dependent RCM laboratory system configuration consists of the basic laboratory illustrated in Figure 46, which is capable of operation only in conjunction with the docked CSM. The dependent laboratory provides a safe 366 cubic foot volume for performance of manned experimental tasks in the space environment. The laboratory is equipped with micrometeoroid radiation protection, thermal insulation with provisions for passive thermal control, airlock, docking provisions with both internal and external volume available for installation of experimental payloads with related equipment.

Active subsystems are not installed in the dependent laboratory, since the various subsystem capabilities are derived from the functional systems of the CSM. Transfer of electrical power into the laboratory is accomplished through the existing LEM interface connectors.

To use the support capabilities of the CSM to the fullest extent possible, additional components and controls for distribution must be installed in the basic RCM laboratory. This normally represents the minimum building block

INNER STRUCTURE	
FORWARD SECTION	281.0
CENTER SECTION	757.9
AFT SECTION	244.8
SECONDARY STRUCTURE	
R.H. EQUIP. BAY	67.0
L.H. EQUIP. COMP	25.6
LOWER EQUIP. BAY	53.8
FORWARD COMPT AREA	5.6
CENTER SECTION	95.2
DOCKING & DROGUE INSTALL.	
DOCKING LATCH, CABLE ETC.	49.2
DROGUE INSTALL.	53.8
AIRLOCK INSTALL.	172.9
METEOROID & THERMAL PROTECT.	
LABORATORY SHELL	239.0
AIRLOCK PROTECT.	72.4
LABORATORY/EXPERIMENT MOUNT	575.0
ELECTRICAL PROVISIONS	2.8
RCM LABORATORY-DEPENDENT	2,696.0

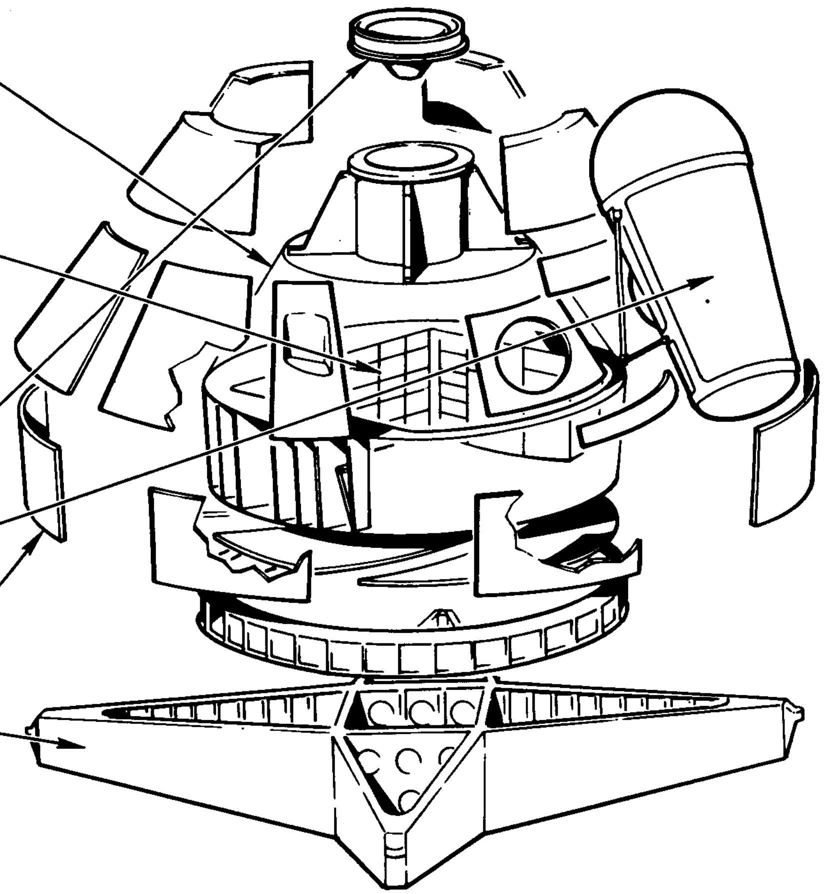


Figure 46. RCM Laboratory, Dependent Configuration

delta not directly related to the basic subsystems shopping list. Capabilities of the dependent RCM laboratory are therefore dependent to a large degree upon the capabilities of the CSM.

The basic AAP CSM configuration in support of RCM laboratory systems consists of Block II subsystems, modified as needed to incorporate changes for extended life and experimental mission performance. Estimated performance assumes achievement of Block II capabilities and reliability goals. A single baseline configuration is assumed for all missions. Assumptions for the CSM weight are as follows:

1. Maximum useful loads (full condition) for RCS, EPS, and ECS (The main propulsion useful load reflects the residual and contingency propellant for the maximum loading condition)

2. Meteoroid shield weight additions to increase the probability of no penetration to 0.995

3. 4.5-day supply of food in the CM; the additional supplies needed are placed in the RCM laboratory.

4. 1.5-day supply of lithium hydroxide for emergency use in the CM with the remainder as necessary in the RCM laboratory.

Subsystems characteristics pertinent to the evaluation are described in the analysis below.

A review of the capabilities of the AAP CSM and of certain of its more critical subsystems to support the accomplishment of objectives of the AAP reference missions follows for the propulsive subsystems, power, attitude, environmental control, data handling, and other factors as required:

Service Module Propulsion

Full service propulsion tanks are assumed for all missions. If the required weight for laboratory and equipment does not require full SPS tanks, the excess propellant can be used to add flexibility to the mission trajectories and return capabilities.

Electrical Power

Table 15 summarizes the total electrical power requirements for trajectory maneuvers, housekeeping (except for thermal control), and experiments. The average power in kilowatts represents the requirement during the orbital phase of the mission. The average power requirements for some of the nonorbital mission segments peak to about 5.2 kilowatts for Reference Mission IV and about 4.0 kilowatts for the other reference missions. These requirements exist for short durations, and where they exceed limits of the fuel-cell capabilities, the CSM batteries are used to make up the deficiencies (the CM has three 40-ampere-hour and the SM two 70-ampere-hour batteries).

The total energy available is 2700 kilowatt hours, permitting about a 2500-watt average load for 45 day missions and proportionately higher average loads for shorter missions. The excess energy is about 1000 kilowatt hours for the Reference Missions.

Table 15. Electrical Power Requirements and Capability

Reference Mission	Average Power (kw)	Experimental Energy (kwh)	Total Energy (kwh)	Available Energy (kwh)	Excess (kwh)
I	1.8	94	1644	2700	1056
II	2.0	210	1760	2700	940
III	2.2	270	1532	2700	1168
IV	2.4	219	1774	2700	926

Attitude Hold Hours and Accuracy

Table 16 summarizes the requirements for spacecraft attitude control. These are based on the experimental requirements and the resulting mission operating profiles. For the lunar-polar-orbit mapping mission, precision attitude holds are required for 6 hours during each 24-hour period. The resulting service module reaction control propellant requirements are given in Table 17. The capabilities and excess capabilities also appear in this table. Both the earth and lunar polar orbit missions have operations requiring attitude-rate constraints of ±0.01 degrees per second that are within the capabilities of the CSM.

Data Handling

The RCM laboratory and experiments are assumed to have their own data storage and management equipments, but transmit through the communication and data subsystem of the CSM. The excess telecommunications

Table 16. Spacecraft Orientation Requirements

Reference Mission	Hours of Attitude Hold		Number of Attitude Holds		
	Coarse (±5°)	Fine (0.1—0.5°)	Coarse	Fine	Total
I	524 (15°)	160	360	90	450
II	524 (5°)	160	665	180	845
III	24	50	64	58	122
IV	576	345			

capabilities include the availability on a time saving basis, of voice communications, data recording, and updata reception; full availability of three low-frequency channels and one video channel, except for certain transmission time constraints; and four timing signals from central timing. The telemetry transmission excess capability, is 21, 312 bits per second, based on a Block II total capability of 51,200 bits per second, and AAP operational requirements of 28,888 bits per second. Up to 213 analog and digital inputs are available in the PCM telemetry equipment for utilization of excess capability. Additional equipment, operating time, or cycling is not required, as experiment or other data can be coordinated with acquisition and storage of the CSM housekeeping data.

Table 18 summarizes communications contact requirements and the mission requirements for data storage and transmission.

Table 19 summarizes by mission the number of cycles and total operating time for the data handling equipments. The pulse modulation (PM) equipment provides for relay of real time, while the FM equipment provides for relay of stored data. The FM time is the total time required to transmit all recorded telemetry data. The earth-synchronous altitude and lunar-orbit missions have the most stringent requirements for real-time data. The total operating time required for S-band ranging for the lunar-polar-orbit mission is about seven times that for the earth-orbit missions.

For Reference Mission III, it is not necessary to record and store housekeeping, telemetry, and voice, as these can be transmitted in real time. Thus, the FM transmitter and data storage equipment could be used for experiment data handling and backup, or, if desirable, it could be removed from the spacecraft.

Table 17. Service Module Reaction Control System Requirements and Capabilities

Reference Mission	Operation	Propellant Required (pounds)		Propellant Available (pounds)		Excess (pounds)		
		Fuel	Oxidizer	Fuel	Oxidizer	Fuel	Oxidizer	Total
I & II	Experimental	591	1112					
	Other	162	260					
	Total	753	1372	972	1507	219	135	354
III	Experimental	391	656					
	Other	422	769					
	Total	813	1425	972	1507	330	307	637
IV	Experimental	282	486					
	Other	673	1168					
	Total	955	1654	972	1507	272	213	485

Table 18. Minimum Telecommunications Functional Requirements Summary

Reference Mission	Telemetering	Down Voice	Tracking and Ranging	Emergency Mode
I	2 minutes of real-time plus as much as possible (up to 8 minutes) of stored data per two hours	With real-time telemetry	Every fourth telemetry transmission (not to exceed 8 hours)	Continuous up-data and voice availability
II	2 minutes of real-time plus as much as possible (up to 8 minutes) of stored data per two hours	With real-time telemetry	Every fourth telemetry transmission (not to exceed 8 hours)	Continuous up-data and voice availability
III	2 minutes each 1/2-hour	With real-time telemetry	Once each 8 hours	Continuous up-data and voice availability
IV	2 minutes real-time each 1/2-hour, plus 6 minutes stored data each 2 hours	With real-time telemetry	Continuous each fourth orbit when in sight	Continuous up-data and voice availability

Table 19. Data Handling Equipment Cycling and Utilization

Equipment \ Reference Mission	I		II		III		IV	
	Number of Cycles	On Hours	Number of Cycles	On Hours	Number of Cycles	On Hours	Number of Cycles	On Hours
S-Band PM Equipment (real-time)	679	32.56	679	32.56	2160	78.75	336	92.4
S-Band FM Equipment (data storage)	673	52.90	673	52.90			336	28.0
S-Band Ranging	158	12.93	135	11.25	135	11.25	84	84
PCM Equipment	2199	73.30	2160	72.0	2160	72.0	1344	44.8
Data Storage Equipment	3539	114	3500	110			1596	63

Reference Mission II creates the most stringent requirements, in terms of on-hours and on-off cycles, for data storage and delay transmission of the stored data. An analysis of MSFN station availability for readout of PM and FM data was made for this mission. The communications contact requirements (Table 62) with the following 13 stations in support of the AAP missions were assumed:

Cape Kennedy
Carnarvon
Madrid
Antigua
Guam
Canberra
Bermuda
Kauai
Goldstone
Grand Canary
Guaymas
Ascension
Corpus Christi

The cumulative-excess-communications time (i.e., the total time during which telemetry communications were scheduled but no data were available for readout) was 82 hours for recorded data and 98 hours for real-time data. The cumulative deficient time (i.e., the total time for which data were lost because the station contact times or durations requirements were not met) was only 4 hours for recorded data and 0.2 hours for real-time data.

Preliminary estimates of on-board data-storage requirements were also computed. The data-storage capabilities shown in Table 20 allow satisfaction of the worst-case requirements.

Table 20. Data-Storage Requirements

Reference Mission	Hours of Data Storage of Delay Data Transmission
I	12
II	12
III	6
IV	6

The studies performed to date have shown that the Block II telecommunication equipments meet the functional requirements of the AAP missions. The tentative communication system reliability requirement (for mission success) for the lunar polar orbit mission is 0.9946. If it is assumed that the equipment must operate continuously, the estimated probability of successful operation is 0.9899. Cycling of the equipment reduces the operating time, but the net cycle effect on overall reliability is not known.

Experiment Hours

Table 21 shows the total time required for experiment performance. Preliminary crew-task-schedule analyses, in which a computer program was used, were performed for Reference Missions II and III. The computer program provided an integrated schedule of experiments with missions operations tasks: those involving meals, exercise, hygiene, recreation, and housekeeping items. The analyses indicated scheduling of 98 to 99 percent of all required activities and an overall crew-time utilization factor of 75-80 percent. Minor modifications of task priorities would allow accomplishment of all required tasks. It thus appears that between 20 and 25 percent of excess available time exists. Some excess time is needed, as it will not be possible to adhere strictly to a preplanned schedule that achieves maximum utilization of all the time available to the astronauts for performing experimental tasks.

Table 21 Experiment Hours

Reference Mission	Experiment Hours
I	798
II	853
III	720
IV	720

Experiment Weight and Volume

Dimensional constraints of the experimental equipment have not been determined. Most of the experiments will be placed in the RCM laboratory, but in some cases it may be desirable to place equipment in the command or service module if space is available. If the service module Sector I contains the fuel cell and other equipment required for maximum-duration missions, there is virtually no capability for placing equipment there.

If the requirements for mission duration and/or mission success reliability were relaxed, the fourth fuel cell and other equipments now located in Sector I could be removed, and about 150 cubic feet could thus be made available for experiments.

The weight allowable for laboratory and experiment equipment is deficient for some missions. The payload deficiencies might be offset in part by allowing modifications to the trajectories and opportunities for abort, provided these limits do not excessively degrade mission success and crew safety.

Return Payload Weight and Volume

The command module must provide space and weight for the return of film, tape, and specimens. The total available volume for return payload is about 11.5 cubic feet, and the weight is about 370 pounds. This volume is outlined in Table 22.

Table 22. Breakdown of Total Return-Payload Volume

Location	Volume (cubic feet)
Low equipment bay—food compartment	0.8
Left-hand equipment bay—food and hygiene storage	1.7
Right-hand equipment bay—extra food and hygiene storage	0.9
Aft storage—an enlargement of Block II storage areas for lithium hydroxide canisters that can be disposed of at data-retrieval time	5.5
Aft storage—portable life support system that could be placed in RCM laboratory	2.6
	Total 11.5

The return payload requirements have not been determined; however, an estimate for return payload indicates that the available volume and weight are adequate. Approximately 150 pounds of data, requiring less than 3 cubic feet of space, is to be returned in the command module.

Environment Control and Life Support

The CSM will provide the laboratory requirements for metabolic and leakage requirements for oxygen and nitrogen. Table 23 summarizes the gas-storage requirements for the 45-day, synchronous earth-orbit mission. The requirements will be less for the missions that have shorter durations or less extravehicular activity and in all cases are amply met by the CSM capabilities.

Table 23. Gas Storage Capability

Item	Oxygen (pounds)	Nitrogen (pounds)
Metabolic oxygen	270	
Leakage, CM and RCM Laboratory	403	118
EVA (seven repressurizations of CM)	42	12
Emergency repressurizations of CM and RCM Laboratory	32	9
	Total weight—886 lb	

Flexibility

The CSM design has several features that provide flexibility in mission planning and permit changing missions to meet new requirements. Among these features are the following:

1. For low-altitude earth orbit or other missions requiring less than 21,000 pounds of SPS propellant, the fuel and oxidizer storage unit tanks in Sections III and VI may be removed and propellant storage will be provided only by the sump tanks in Sections II and V. Also, the Block II SPS pressurization tanks may be removed when propellant tanks are removed from Sections III and VI.

2. For limited duration missions, or other missions for which the electrical power capability of the AAP configuration is not required, the fuel cell and cryogenic tanks that are located in Section I of the service module can be removed. These are easily removable at the launch pad.

3. The launch-site removal of two of the LM RCS tanks in each quadrant will be permitted for weight-critical missions that do not require large RCS propellant quantities.

4. The ECS has the capability to operate alternatively at either 70 percent oxygen and 30 percent nitrogen or 100 percent oxygen.

Trajectory flexibility, alternate-mission capability, and abort capability, whether considered in planning storage or required during the mission, depend in part on the availability of excess SPS and RCS propellant and on the needed life support consumables. The reference mission having maximum flexibility in these respects is the lunar-taxi mission.

Since lighting and other requirements for normal-mission returns, lunar landings, and abort requirements have not been established, estimates of the available launch and return windows cannot be made at this time.

INDEPENDENT RCM LABORATORY SYSTEM CAPABILITY

The mission performance capability of the independent laboratory system is an identifiable function of the subsystem building block deltas selected from the shopping list matrix of possible subsystems alternatives. As previously noted, the basic (dependent) RCM laboratory configuration contains no subsystems and is passively dependent upon the performance capabilities of the AAP CSM and its subsystems. It is possible to add the necessary provisions for electrical energy distribution and control in the laboratory and experiments supplied by the AAP CSM through the docking tunnel LM interface connectors. This, however represents an increment in the capabilities of the basic RCM laboratory, obtained by incorporating the minimum (first) delta. This delta is essentially independent of subsystems location, whether they are installed in the CSM or the RCM laboratory itself.

Consistent with the study approach outlined previously, the desired output of the analyses is the identification and definition of the subsystems and the partial subsystem building block incremental deltas added to the RCM laboratory configuration to enhance its mission and experiment support capabilities. The subsystems evaluation baseline in the RCM laboratory system is a 30-day mission configuration, with nominal housekeeping requirements and experiment support capabilities as identified for the selected representative AAP reference missions.

The initial list of subsystems selected for addition to the Basic RCM laboratory consists of Apollo CSM subsystems, parts, and components. It has been assumed that the recovered CM subsystems are renovated to their original performance condition for use either in the RCM spacecraft or the RCM laboratory. The Apollo SM subsystems and components proposed for use in the RCM Laboratory are new since none of these are recovered after flight.

Summarized in this section are estimates of the subsystem capabilities contained in the baseline configuration of the fully independent RCM laboratory containing complete Apollo CSM subsystems as shown in Table 22.

The baseline used for evaluation was a 30-day mission configuration, with nominal housekeeping and experiment support capabilities for a typical AAP reference mission profile and orbit mission experiment program. Missions or experiment programs imposing requirements in excess of those provided for the 30-day baseline dependent configuration are met by adding subsystems or reducing mission duration. The subsystems shopping list was established to provide a range of mission capabilities lying within the limits of the support requirements summarized in Table 7. This selection allows determining weight, volume, and performance characteristics of the modular additions to the dependent laboratory needed to meet mission requirements.

The subsystems comprise a shopping list, allowing a selection by NASA of systems needed for accomplishing a specified mission. Thus, the subsystems alternatives can be selected consistent with the levels of capabilities required: power or attitude hold hours—or increasing mission duration.

Subsystem building blocks and characteristics are identified as follows: ECS (Figure 47), SCS (Table 24), EPS (Figure 48), RCS (Figure 49), and communications and data (Table 25). These data were obtained from the subsystems engineering analyses. Not included are the Apollo Block II G&N and the SM SPS. The laboratory will be manned only when the CSM is attached and the CM G&N system will provide any required guidance functions. It is also assumed that the 40,000-pound fuel capacity of the service module will be adequate to meet any of the AAP mission requirements. For most low altitude earth orbit missions, only fuel for de-orbit will be required, about 1200 pounds. Typical installation of subsystems in the laboratory are illustrated in Figures 50, 51, and 52.

The oxidizer and fuel storage tanks in Sections III and VI, respectively, of the SM can be removed for low-altitude earth orbital missions requiring less than 21,000 pounds of propellant. Sump tanks in Sections II and V will provide the required propellant storage. The two Block II SPS helium pressurization tanks will be retained when the two propellant storage tanks are removed, but helium may be off-loaded to effect weight saving.

BLOCK II

Δ	IDENTIFICATION	CAPABILITY	WT (LBS)	VOL (CU FT)	POWER (WATTS)	MTBF
1	MINIMUM BLOCK II ECS COMPONENTS	MODERATE EXPER HEAT LOADS ALLOWABLE	175	EQUIVALENT TO APOLLO CSM	REFER TO VOLUME III "SUBSYSTEM ENGINEERING"	SAME AS FOR APOLLO CSM
2	INTERMEDIATE BLOCK II ECS COMPONENTS	HIGH EXPER HEAT LOADS ALLOWABLE	425			
3	BLOCK II ECS	FULL BLOCK II - 30 DAYS FOR 2 MEN	848			

ALTERNATES

1	AAP ECS	DUAL GAS, MOLECULAR SIEVE	948		NA	NA

NOTE: FOR DETAILED TECHNICAL DATA REFER TO VOLUME III "SUBSYSTEM ANALYSIS," RCM LABORATORY SUBSYSTEM BUILDING BLOCKS SECTION XVI.

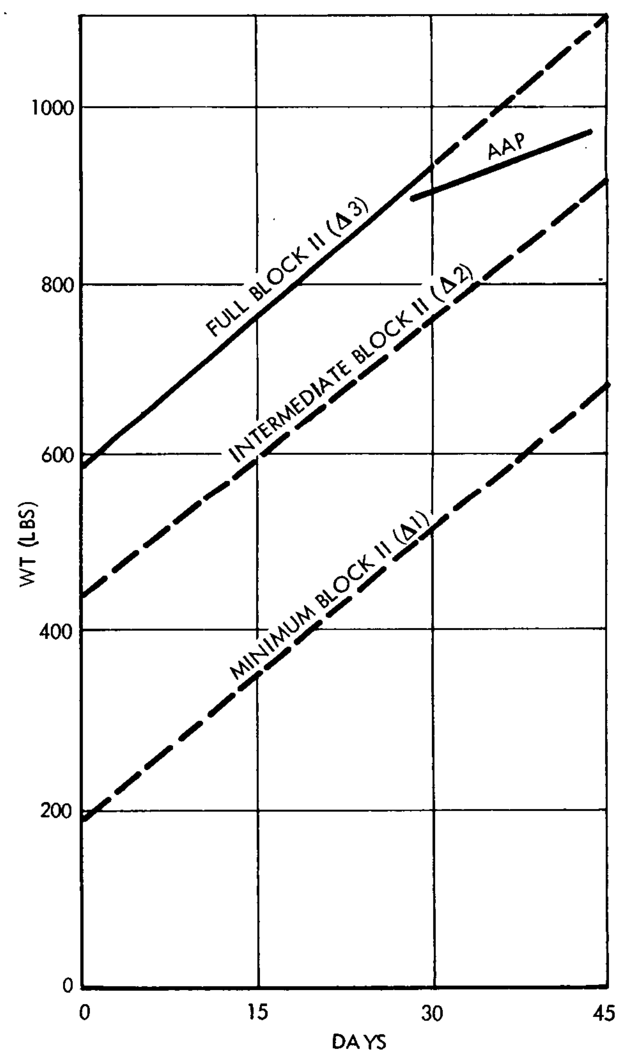


Figure 47. Environmental Control System

Table 24. Stabilization and Control System

BLOCK II

Δ	Identification	Capability	Weight (Lb)	Volume (Cu Ft)	Power (Watts)	MTBF (Hr)
1	Block II less thrust vector components	±0.2°, ±5° deadbands (Provides attitude control in nonthrust modes)	137	2.0 (U)	250	2,500
2	Block II	±0.2°, ±5° deadbands (Provides attitude control in thrust and nonthrust modes)	161	2.0 (U) 0.9 (P)	250	2,500

ALTERNATES

1	Advanced inertial reference and control electronic assembly (Autonetics)	±0.5° deadboard	17	0.4	35	140,000
2	Agena IRD	14-day				

U - Unpressurized
P - Pressurized

BLK II

Δ	IDENT	CAPABILITY	WT	VOL	MTBF
1	WIRING & CONTROLS (LEM INTERFACE)	USE CSM EPS 200 WATTS	316		CSM
2	BLK II 3 FC BATTERIES		3050		*1200

ALTERNATES

	NONE IDENTIFIED				

*EACH FUEL CELL

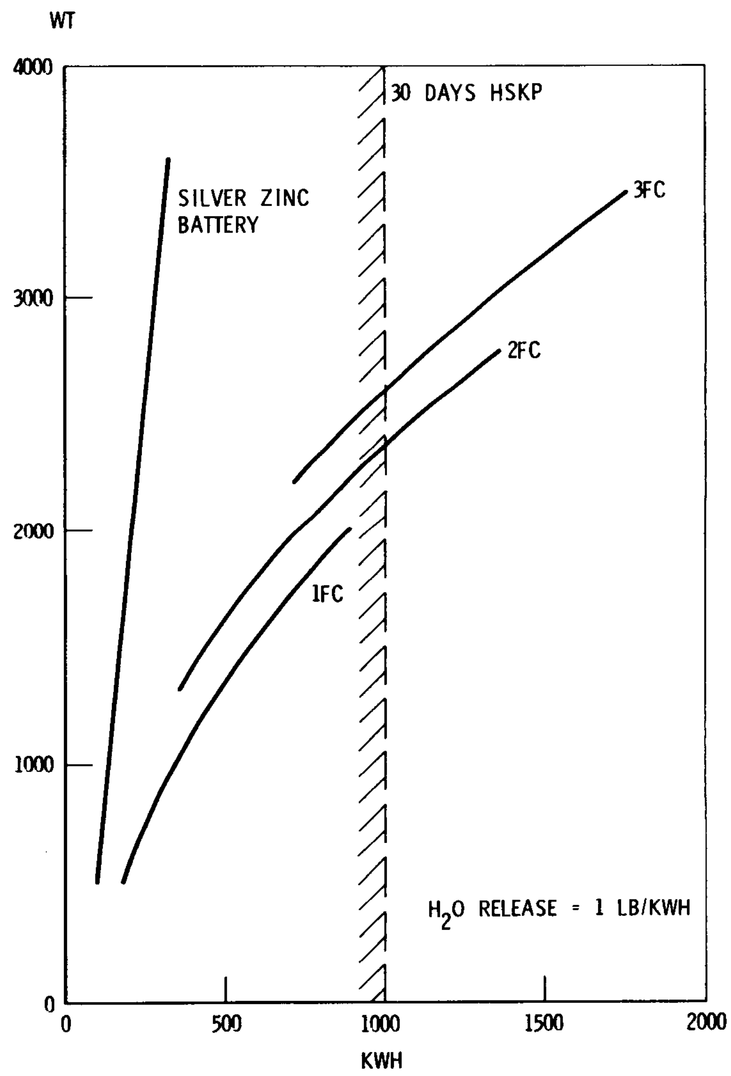


Figure 48. Electrical Power System

BLOCK II[1]

Δ	IDENTIFICATION	CAPABILITY	WT (LBS)	VOL (CU FT)	POWER (WATTS)	LIFE (CYC)
1	BLK II CM	LIMITED	302 + CONS	APOLLO BLK II EQUIVALENT	REFER TO "SUBSYSTEM ANALYSIS" VOLUME III	5000
2	BLK II SM	DEADBANDS - ±0.2°, ±5°. MANUAL AND G&C CONTROL	959 + CONS			10,000

ALTERNATES

1	AGENA GTV SPS	SEE "SUBSYSTEM ANALYSIS" VOLUME III	280			

[1] BLOCK II SYSTEMS CURRENTLY QUALIFIED FOR 400 HOURS PRESSURIZATION

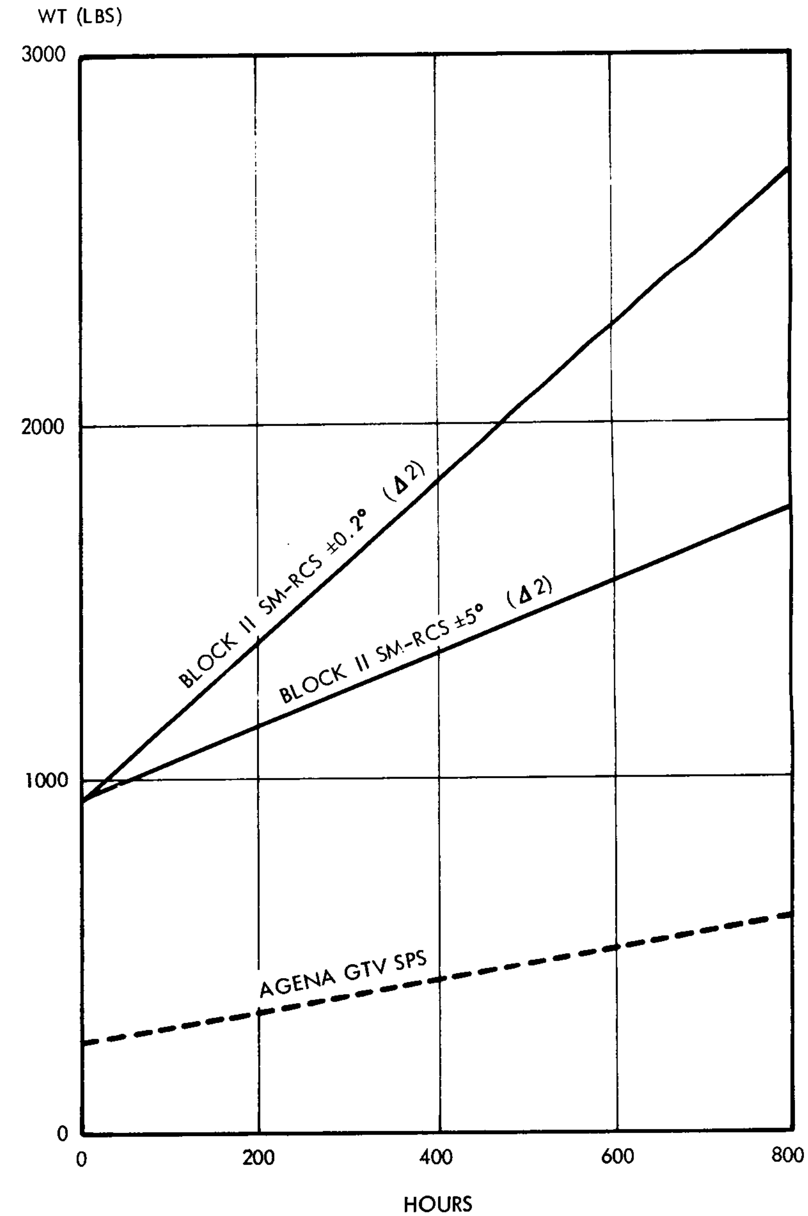


Figure 49. Reaction Control System

Table 25. Communications and Data

BLOCK II

	Identification	Capability	Weight (Lb)	Volume (Cu Ft)	Power (Watts)	MTBF
1	Hardline + CM Audio Center	Voice Within lab Lab and CSM Ground to Lab	10	Same as Apollo CM	20	Same as Apollo Block II
2	CM audio center Transceiver VHF	VHF voice: Lab and supporting CSM Lab and ground Lab and EVA crew	36		20(A) 40(T)	
3	CM audio center Unified S-Band Updata link Central timing	System 1 hardline + S-band voice, timing, ranging, updata commands	118*		120	
4	Complete Block II	Tracking/command/TM Voice 50K b/s	307			

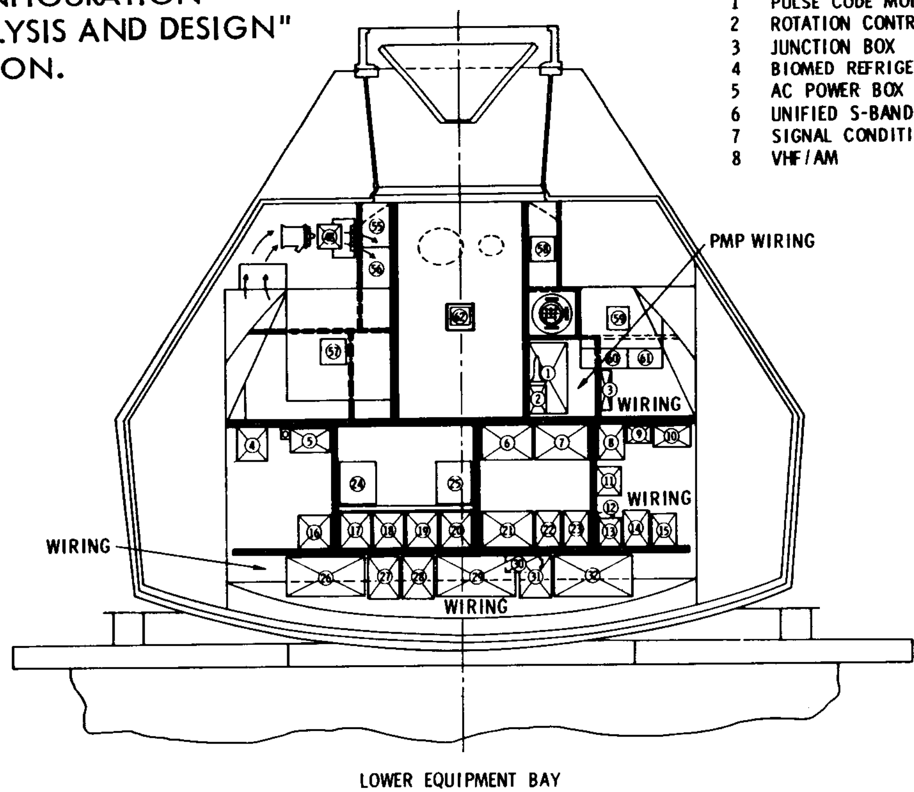
ALTERNATES

	Identification	Capability	Weight (Lb)	Volume (Cu Ft)	Power (Watts)	MTBF
1	SGLS (TRW)	Tracking/command/TM Voice	25			550 days

A - Audio equipment
T - Transceiver

*Cold plate cooling required data storage for experiments included with experimental equipment weight

DETAILED SUBSYSTEM ARRANGEMENT IS GIVEN IN DRAWINGS IN "CONFIGURATION ANALYSIS AND DESIGN" SECTION.



1 PULSE CODE MODULATOR
2 ROTATION CONTROL
3 JUNCTION BOX
4 BIOMED REFRIGERATOR
5 AC POWER BOX
6 UNIFIED S-BAND
7 SIGNAL CONDITIONER
8 VHF/AM
9 TRIPLEXER
10 VHF BEACON
11 MEDICAL SUPPLIES
12 S-BAND RF SWITCH
13 HI FREQ TRANSCEIVER
14 PREMODULATION PROCESSOR
15 AUDIO CENTER
16 CONTROL ELECTRONICS
17 GYRO DISPLAY COUPLER
18 DISPLAY ELECTRONICS
19 REACTION JET DRIVER
20 S-BAND POWER AMPLIFIER
21 DATA STORAGE EQUIPMENT
22 UP DATA LINK
23 CENTRAL TIMING
24 GYRO PACKAGE
25 GYRO PACKAGE
26 INVERTER
27 BATTERY
28 BATTERY
29 INVERTER
30 BATTERY CHARGER
31 BATTERY
32 INVERTER
48 CABIN AIR RECIRCULATION UNIT
55 TIMING CONTROLS AND DISPLAYS
56 LIGHTING CONTROLS AND DISPLAYS
57 COMMUNICATION AND ANTENNA CONTROLS AND DISPLAYS
58 SCS CONTROLS AND DISPLAYS
59 RCS CONTROLS AND DISPLAYS
60 CAUTION/WARNING
61 DOCKING CONTROLS AND DISPLAYS
62 RCM EXPERIMENTS CONTROLS AND DISPLAYS

Figure 50. Fully Independent RCM Laboratory, Block II, Lower Equipment Bay

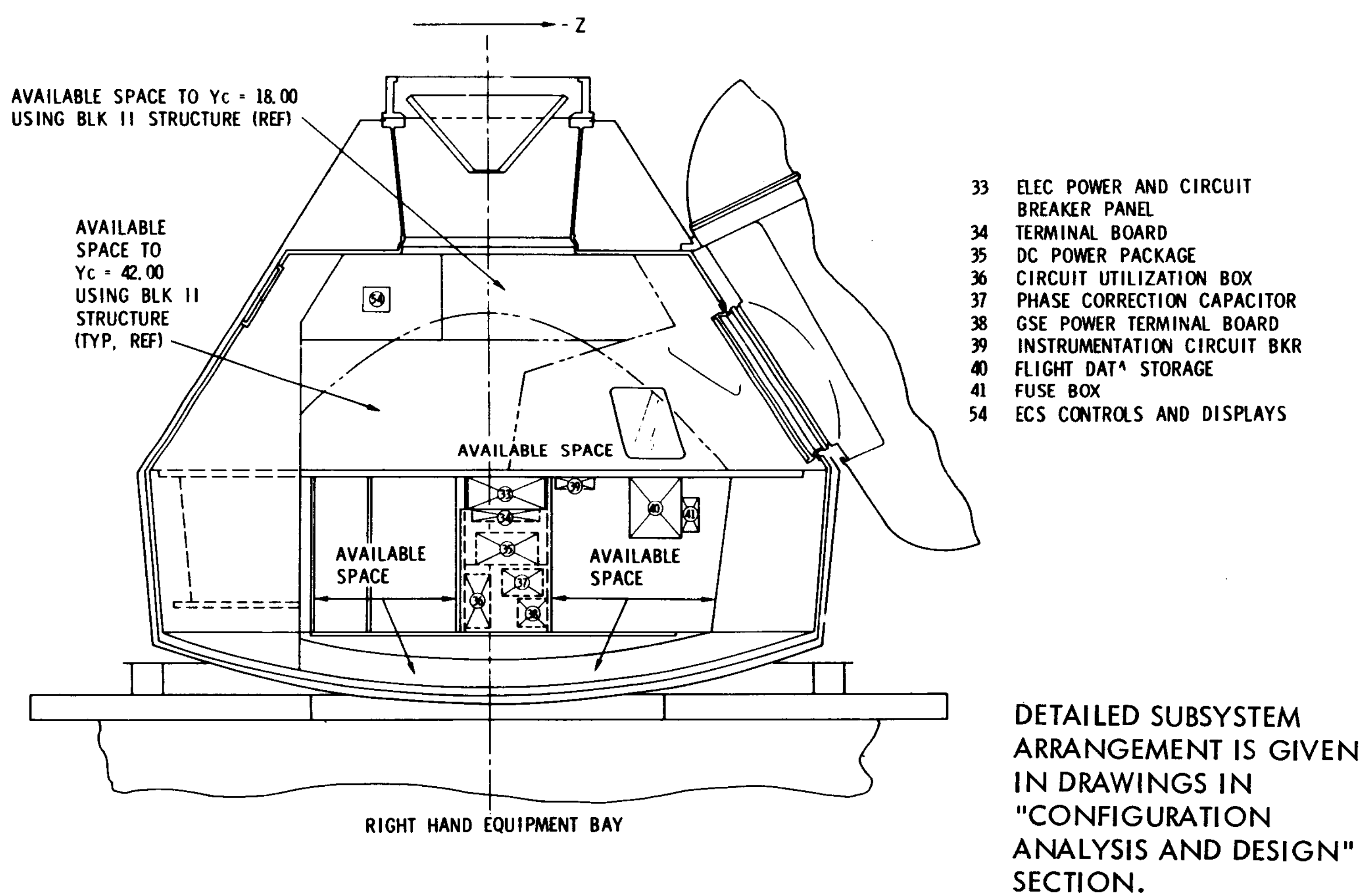
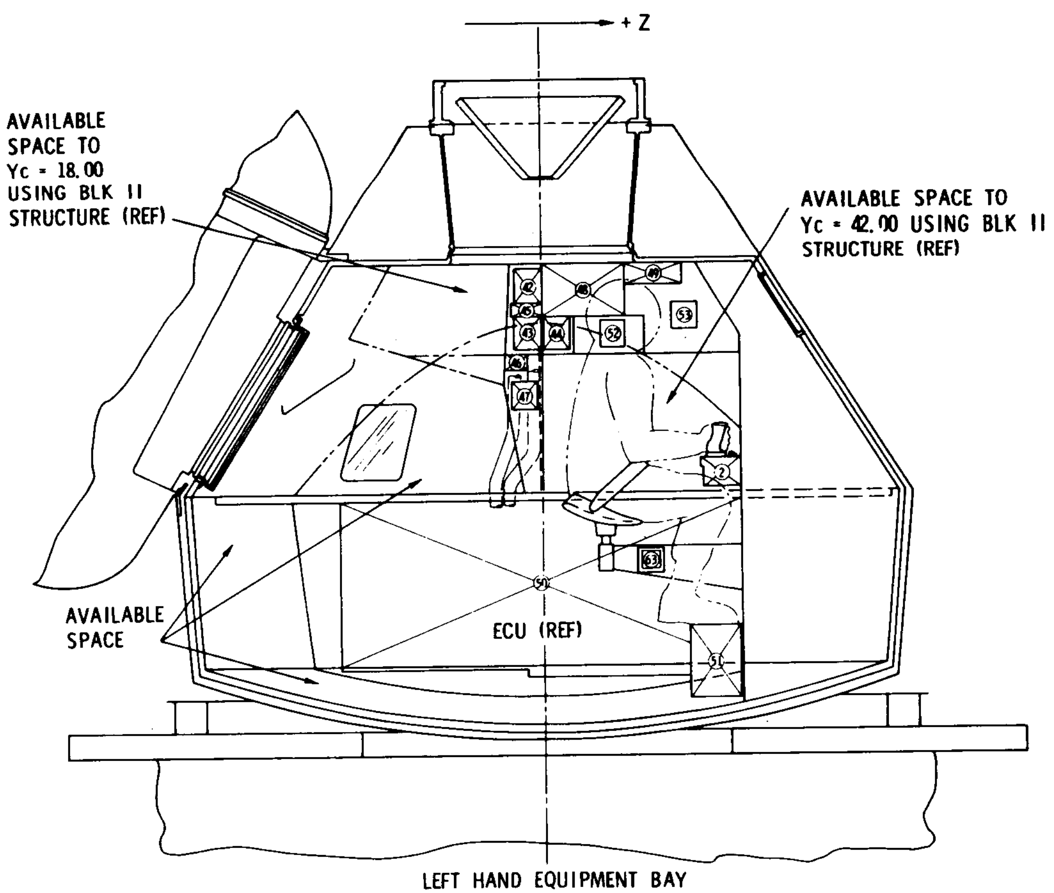


Figure 51. Fully Independent RCM Laboratory, Block II, Right Equipment Bay



DETAILED SUBSYSTEM ARRANGEMENT IS GIVEN IN DRAWINGS IN "CONFIGURATION ANALYSIS AND DESIGN" SECTION.

Figure 52. Fully Independent RCM Laboratory, Block II, Left Equipment Bay

Environmental Control System

In Figure 47, three Block II alternates are provided. The differences are based primarily on the allowable experimental heat loads. All of the alternates provide environmental life support for the laboratory only, and assume two men are in the laboratory and one is in the command module. While this assumption is correct for the independent laboratory configurations normally only one man will be in the dependent laboratory at any one time, as one man will be sleeping in the CM and another will be monitoring the CM subsystems.

The Block II ECLSS provides pure oxygen only; and includes LiOH canisters for CO_2 removal. For missions longer than about 30 days a dual gas system with molecular sieve for CO_2 removal is required.

Stabilization and Control System

An SCS is required whenever the laboratory operates in an independent mode. Two Block II options are identified in Table 24. The first will provide the full Block II SCS capability except during thrust. The second also includes thrust vector control which is required only for the operation of the SPS in the SM.

Electrical Power System

Two alternates have been identified in Figure 48. The first obtains power from the service module, through the LEM interface. About 200 watts can be provided across the interface. The complete Block II system is provided for the independent laboratory. One, two or three fuel cells are available with the number selected depending primarily on the kwh and reliability requirements. It is anticipated that the fuel cell life can be uprated. Advance to the independent laboratory system is also provided by the CSM EPS. No acceptable alternatives to the Block II subsystem were identified in the subsystems analyses.

Reaction Control System

The dependent laboratory is attitude controlled by the CSM. For the independent laboratory, three RCS alternatives were identified as shown in Figure 49. The delta selected will be dependent on the experiment and other mission requirements. Detailed descriptions of the alternates are given in Volume III of this report.

Communications and Data

Varying levels of capabilities for the communications and data subsystem are given in Table 25. The TRW space guidance link subsystems is also a possible alternate. Detailed descriptions of these subsystems alternates are given in Volume III of this report.

INTEGRATED SYSTEM CAPABILITY SUMMARY

The subsystems shopping list established on the basis of Apollo Block II CSM equipment can provide a basic range of laboratory/experiment support capabilities for the baseline 30-day reference missions system requirements as summarized in Table 26. The principal facets of the RCM laboratory subsystem configuration performance capability are the ability of the laboratory to accommodate needed equipment and consumables to meet other direct support requirements for attitude control, electric power, communication, thermal control, and the ability of the subsystems and total laboratory to perform reliably for the required mission duration within allowable crew safety limits and provide safe return to earth.

The estimates presented of comparative subsystem capabilities of the fully independent RCM laboratory system configuration to accomplish the selected reference missions are considered to be representative of typical advanced AAP mission experimental objectives and support requirements. Shown in Table 27 are the significant ranges of AAP experiment support requirements and the corresponding performance capabilities associated with the several RCM laboratory subsystems of the Block II configuration adjusted for the nominal 30-day mission operation. Table 27 lists the subsystem performance capabilities of the fully independent RCM laboratory available for experiment support during a mission of nominal 30-day duration, compared to the experiment support requirements associated with the experiment configuration and experiment objectives of the four reference mission profiles selected as analysis baseline for this study.

Table 27 shows that the RCM laboratory system capabilities may be separated into three categories: those associated with the laboratory/experiment physical configuration (basic RCM laboratory configuration), those associated with the laboratory subsystems used in support of the experiments, and those associated with the capabilities of the spacecraft and laboratory crew to support mission objectives and experiments.

Table 26. Fully Independent RCM Laboratory System Functional Performance

Parameter	Fully Independent RCM Laboratory Configuration Capability	System Requirements				
		ELIO	EPO (S-IB)	EPO (S-V)	ESO	LPO
Mission duration (days)	30	30	30	30	30	30
Crew size	3	3	3	3	3	3
Experiment payloads (pounds)	Mission dependent	3700	7600	7600	6450 1/	11902/
Experiment equipment volume (cu ft)						
External	SLA limited (approx 3000)	130	475	475	294	61 2/
Pressurized	150 + work space	96	72	72	60	30 2/
Number of attitude cycles						
Coarse (±2°)	770	360	665	665	64	
Fine (±0.1°)	210	90	180	180	58	
Attitude hold time						
Coarse, ±2° (Hours)	610 5/		524	524	24	576
Fine (±0.1°) (Hours)	186		160	160	50	345
Electrical power (kwhr)	766 4/		590	590	640	1067
ECS/EVA (days/no.)	30/10	30/6	30/0	30/0	30/6	30/0
G&C hrs. - active		6	6	6	4	14
Communication useage (cycles)		5330	5330	5330	4410	3460
Data rate (bits/sec)	Block II	<Blk II	<Blk II	<Blk II	<Blk II	<Blk II
Thermal control	Block II ECS					
Experiment man-hrs.	720 3/	~550	~550	~550	~550	~450
Mission success	0.9					
Crew safety	0.999	0.999	0.999	0.999	0.999	

1/ Flight 516
2/ SID 65-1088 Lunar Orbital Survey Missions Study, August 1965
3/ 75% utilization (LPO hr = 576)
4/ For experimental support
5/ Trades between coarse/fine/cycles requires definition.

Table 27. AAP Experimental Requirements and RCM Laboratory Capability

Lab Physical Configuration	AAP Requirement Min - Max	RCM Laboratory Dependent	RCM Laboratory Fully Independent
Mission duration (days)	15 - 45	30	30
Crew size (laboratory maximum occupancy)	1 - 3	1 - 2	2 - 3
RCM laboratory weight (lb)	2,700 - 13,690	2,700	13,690
Experiment payload (lb)	1,000 - 12,000	4,740	
Laboratory volume (cu ft)	20 - 300	366	366
Laboratory floor area (sq ft)	5 - 40	40	40
Experiment volume (total) (cu ft)	20 - 720	3,000	3,000
Pressurized volume (cu ft)	20 - 300	366	366
Subsystems			
Attitude hold time		capability is derived from CSM; none in laboratory	
Coarse, ±2° (hours)	30 - 800		610
Fine (±0.1°) (hours)	50 - 250		186
Number of attitude holds			
Coarse, ±2°	0 - 1,000		770
Fine ±0.1°	50 - 300		210
RCS Fuel (lb)	300 - 1,000		650
RCS oxidizer (lb)	500 - 1,800		1,000
Electrical energy (kwh)	500 - 2,000		850
Crew			
Astronaut hours	150 - 1,080	720	1,450
Number of EVA's	0 - 30	10	30
Crew safety	0.999	0.999	0.999
Meteoroid protection	0.995	0.995	0.995
Mission success	Open	Open	Open

IV. CREW OPERATIONS

RCM LABORATORY CREW OPERATIONS

During the RCM study, all of the AAP mission-related studies were examined and evaluated to determine deltas in crew operations associated with each configurational concept and/or reference mission. This program, which evolved from the engineering investigation conducted to determine Apollo spacecraft modifications required to extend mission capabilities, currently calls for approximately twenty-five flights. Most of these flights require some form of experimental appendage.

Beginning with the Extended Mission Apollo Study (XMAS), numerous CSM configurations, external devices, and subsystem concepts were conceived, developed, evaluated, and either rejected or integrated in the extended utilization of the Apollo spacecraft. The evolutionary development of the Apollo Applications Program operational capability is illustrated in Table 28. The AAP mission related documentation (identified in Column 1 of the table) was used to establish the RCM laboratory operational commonalities.

The identification of operations and activities that must be performed by the crew to support mission objectives is predicated on the assumption that the RCM laboratory module can be used to replace the external devices discussed in the various AAP mission related studies. Using the AAP reference missions and objectives as guides, the replacement characteristics of the RCM laboratory module have been tentatively identified for various levels of laboratory development and are presented in Table 29.

DEPENDENT RCM LABORATORY CREW FUNCTIONS

In the various AAP mission-related studies examined and evaluated, the crew performance requirements inherent to the operational support of the spacecraft and its appendages are classed as "crew housekeeping functions," and include all the functions accomplished by the crew in operating the spacecraft and in maintaining themselves. The functions required to provide operational support of a laboratory module or experiment appendage have a similarity for all missions. In some instances these functions may differ to the point of becoming unique to a specific mission phase and/or spacecraft configuration, these differences in terms of time are relatively inconsequential. Functions concerned with crew maintenance or life support are even more constant for the various missions and spacecraft configurations.

Table 28. Apollo Applications Program Evaluotion

AAP Mission Related Study	Identification Concept or Mission	Configuration Description	External Devices	Orbital Lifetime	Crew
XMAS Extended Mission Apollo NAS 9-1963 SID 63-1370	Concept I	CSM with advanced subsystems (Solar cell-battery and molecular sieve)	None	120	2
	Concept II	CSM with advanced subsystems (Solar cell-battery and molecular sieve)	Lab module (dependent) housed in LEM adapter section	120	3
	Concept III	CSM with subsystems quiescent during orbital operations	Lab module (independent) housed in LEM adapter section	360	3
Addendum II SID 64-457	Concept 0	CSM with systems extended by spares and redundancies		90	2
Apollo X Extended Apollo Systems Utilization NAS 9-3140 SID 64-1860	Apollo X (Concept 0)	CSM with systems extended by spares and redundancies	CSM only	45	2
			Lab module in LEM A/S with airlock	45	3
			with platform	45	3
			with telescope	45	3
			with mapping	45	3
			Lunar survey module	28	3
Prolonged Missions SID 65-968	Mission Support	Concept II CSM	Lab (dependent)	45 to 600	3 to 6
	Logistics Support	Concept III CSM	Lab (independent)		
Addendum I SID 65-500	Configuration 1	Block II CSM with pallet in SM Sector I	Rack or lab module with airlock (docked)	14	3
	C	XCSM with expendables, spares, redundancies	Rack or lab module with airlock (docked)	45	3
	D	Block II CSM with modified subsystems SM Sector I empty or pallet	Rack and/or lab module with subsystem dependent 14 days; independent thereafter	45	3
	D[1]	Block II CSM with "stretched" subsystems LMS in SM Sector I	Rack with subsystems	30	3
AES-PDP Apollo Extension System Preliminary Definition Phase NAS 9-5017 SID 65-1534	Phase I Mission 211	Block II CSM (unmodified) with pallet in SM Sector I	None	14	3
	507	Block II CSM (unmodified) with MSF-1 in SM Sector I	Phase I LEM Laboratory	14	3
	509	Block II CSM (unmodified) with pallet in SM Sector I	Phase I LEM Laboratory	14	3
	511	Block II CSM (unmodified) with MSF-1 in SM Sector I	Phase I LEM Laboratory, A/S and D/S	14	3
	Reference Mission 1	XCSM	LEM Laboratory	45	3
	2	XCSM	LEM Laboratory	45	3
	3	XCSM with MSF-1 in SM Sector	LEM Laboratory	28	3
	4	XCSM	LEM Taxi	20	3

Table 29. RCM Laboratory Module Replacement Characteristics

Apollo Applications Program Reference Missions	Objective	Renovated Command Module Replaces
Reference Mission II Earth Polar Orbit 3-Man, 45-Days	Remote sensing of the earth's surface and atmosphere	SM Sector I for Pallet SM Sector I for MSF-1 Experiment rack with airlock Dependent lab module Independent lab module
Reference Mission III Earth Synchronous Orbit 3-Man, 45-Days	Space physics and subsystem development OGO launch	SM Sector I for Pallet Storage for OGO return Experiment rack with airlock Dependent lab module Independent lab module
Reference Mission IV Lunar Polar Orbit 3-Man, 28-Days	Mapping of the lunar surface using cameras and radar	SM Sector I for MSF-1 Experiment rack with airlock Dependent lab module Independent lab module

Table 30. Crew Functions Scheduling Ground Rules

Parameters	Constraints		
	Rigid	Semi-Rigid	Flexible or Not Identified
Duration (time allowed at scheduled times: not total time)	Sleep - 6 hr and 1.5 hr Eat - hot meal 45 minutes, cold snack (or light hot meal) 30 minutes Personal hygiene and defecation - 30 minutes Telecommunications* - 8 minutes (maximum) except for synchronous and lunar orbits Initiate thermal cycling* - 5 minutes Terminate thermal cycling* - 5 minutes Attitude orientation: With IMU coarse and fine alignment - 25 minutes With IMU fine alignment - 15 minutes With neither - 5 minutes	Systems management and checks* - any combination of the following that can be scheduled. Fuel cell purge, each cell (H_2 & O_2)—3 min EPS status checks—6 min Battery check, each battery—1 min Battery recharge, each battery—20 min ECS status check—4 min SM RCS status check—3 min SPS status check—1 min CM RCS status check—1 min Caution and warning light check—1 min DSKY lights check—2 min Telecommunications status check—3 min Lithium hydroxide filter maintenance—5 min Exercise—30 min, 45 min, or 1 hr Work period—not more than 4 hr Scheduled recreation—not less than 1 hr	Nonscheduled recreation—as available Scheduled maintenance / Nonscheduled maintenance — not identified Urination - as required Safety package— Physiological and Performance Monitoring (PPM)
Schedule (frequency and location in time)	Telecommunications - as indicated on schedules Cold snack (or light hot meal) - after 6-hr sleep Personal hygiene and defecation - after sleep periods	Sleep - between 6-hr and 1.5-hr period: 8 hr ±2 hr Eat hot meal - 2 per day between work/exercise periods of at least 2-hr length each Exercise - any combination of 30-min, 45-min, or 1-hr periods totaling 2 hr Not less than 1 hr before a sleep period Not less than 1 hr after a 30-min exercise period or 2 hr after 45-min or 1-hr period Each system management and check shown above, except battery recharge and lithium hydroxide filter maintenance—every 7 hr ±30 min Battery recharge—as required Lithium hydroxide filter maintenance—every 12 hr ±30 min	Cold snack - 2 as desired Scheduled recreation - two 1-hr periods or one 2-hr period as convenient but preferably a 1-hr period immediately before a 6-hr sleep Work period - preferably 30 min between contiguous 2-hr work periods and 1 hr between work periods longer than 2 hr Urination - as required Drinking - as required Overlap of sleep periods - determined primarily by experiment requirements and availability of auditory warning system for out-of-tolerance system developments Battery recharge - as required Attitude orientation - primarily determined by experiment requirements

*Not per crew member

Crew housekeeping function requirements have been analyzed and evaluated during each AAP mission-related study. Tables of crew housekeeping tasks for the various areas of activity, detailed functional analyses, and delta design analyses covering subsystem changes required to extend systems life are available as the result of study and analysis during the preliminary definition phases of the AES. Examination of these data indicates that only small differences in crew housekeeping functions requirements occur when the LEM, LEM lab, and/or other experiment appendage(s) are replaced by the RCM laboratory module.

Crew functions scheduling ground rules and housekeeping requirements are presented in Table 30. These ground rules were used for computer scheduling of AAP mission operations which integrates detailed housekeeping activities, physiological and performance monitoring (PPM), and experiment scheduling requirements for each mission.

Identifiable crew function times per man per day during orbital operations are summarized in Table 31. The times shown are primarily maximum times, and may vary as a function of mission objectives. These times also do not include command, control, and systems management tasks performed during launch, reentry, and recovery phases.

Table 31. Crew Housekeeping Function Times (per man per day)

Activity	Time (Hours)
Sleep	7.5
Eat	3.0
Personal hygiene and defecation	1.0
Exercise	2.0
Recreation	—(2.0) if experiments permit
Safety package (maximum)	1.6
Systems check and management	0.7
Total	15.8

Representative schedules illustrating the application of the crew time characteristics in scheduling activities for the various reference missions are shown in Figures 53, 54, 55, and 56.

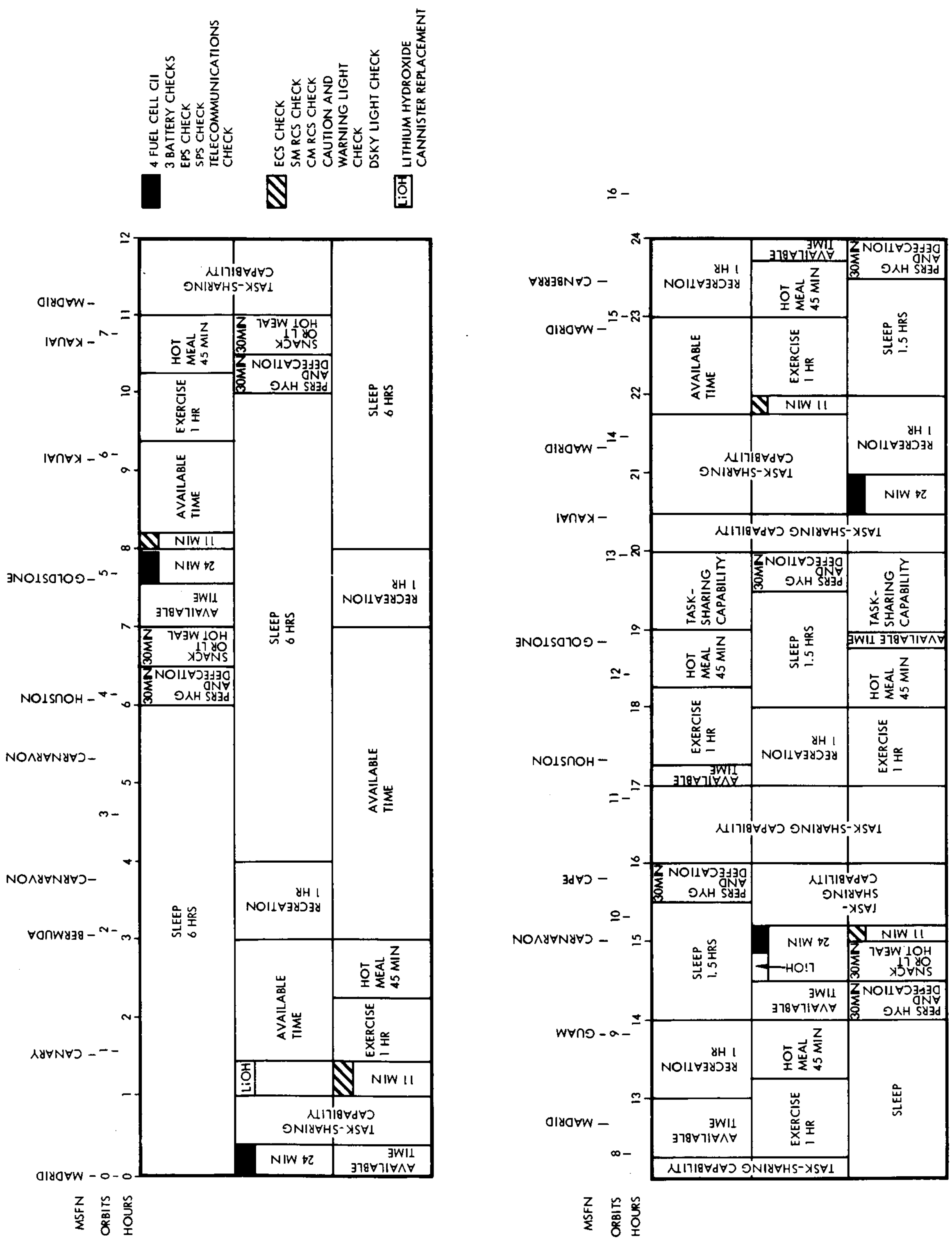


Figure 53. Crew Work-Rest Schedule, Nominal 24 Hours, Reference Mission I

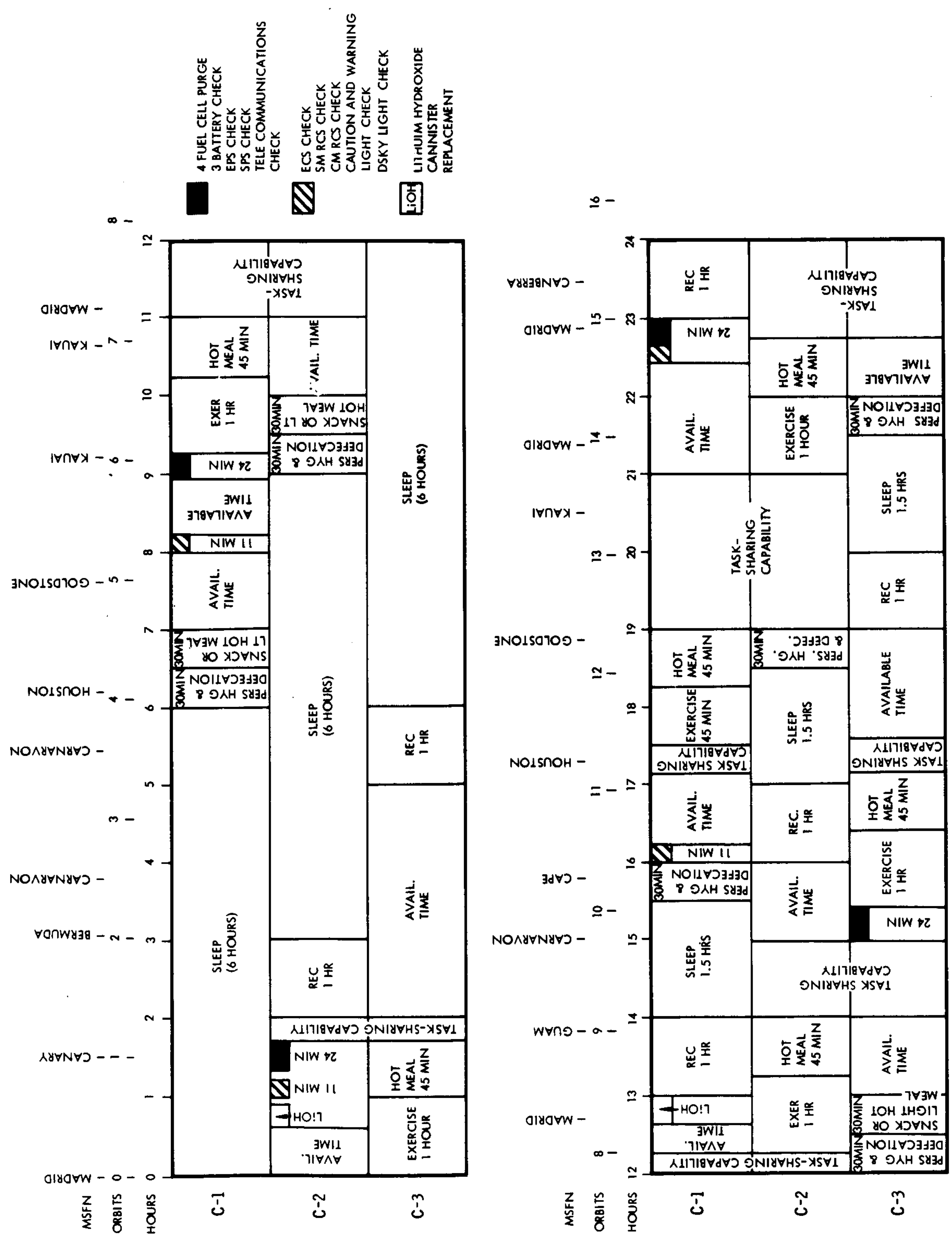


Figure 54. Crew Work-Rest Schedule, Nominal 24 Hours, Reference Mission II

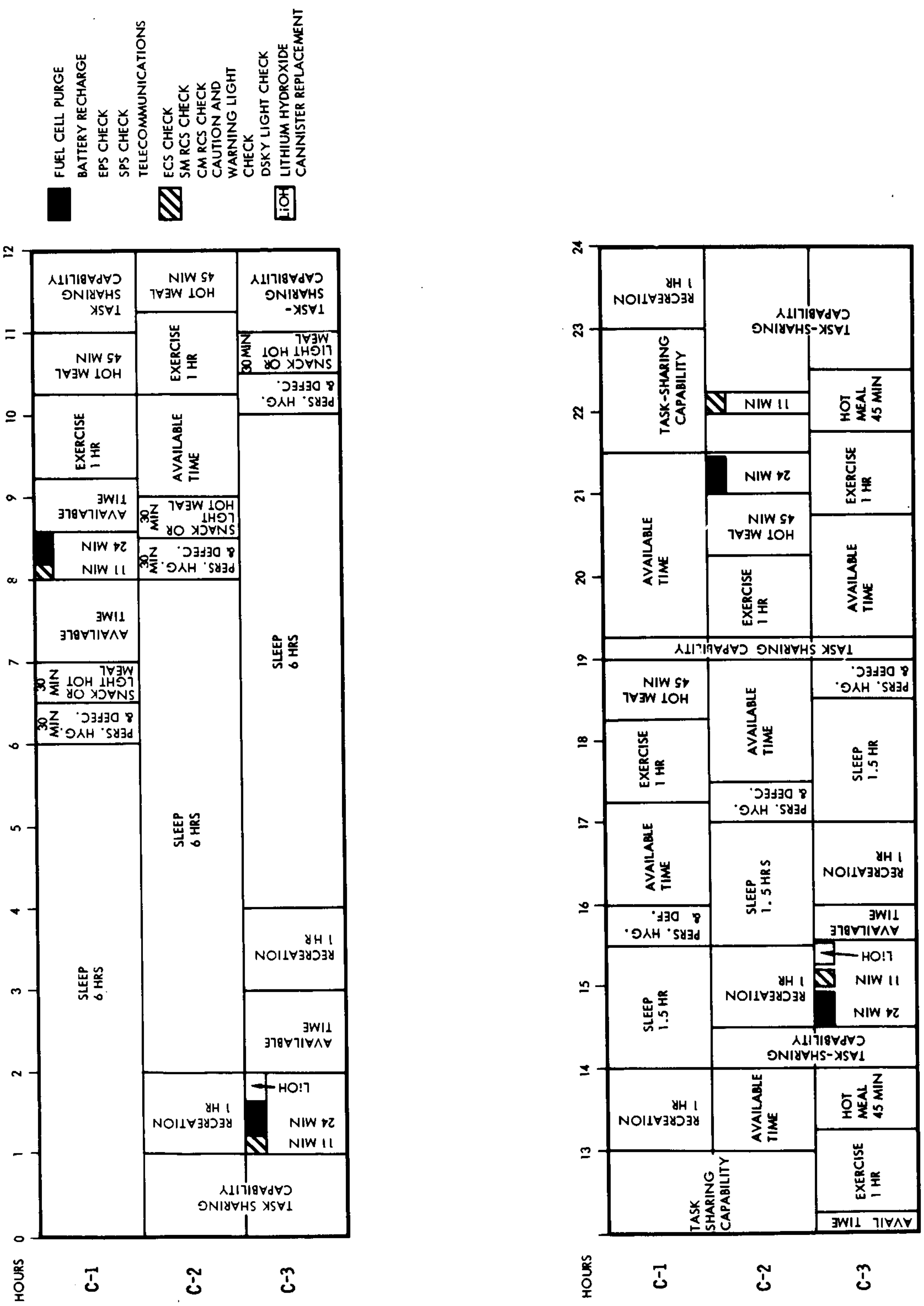


Figure 55. Crew Work-Rest Schedule, Nominal 24 Hours, Reference Mission III

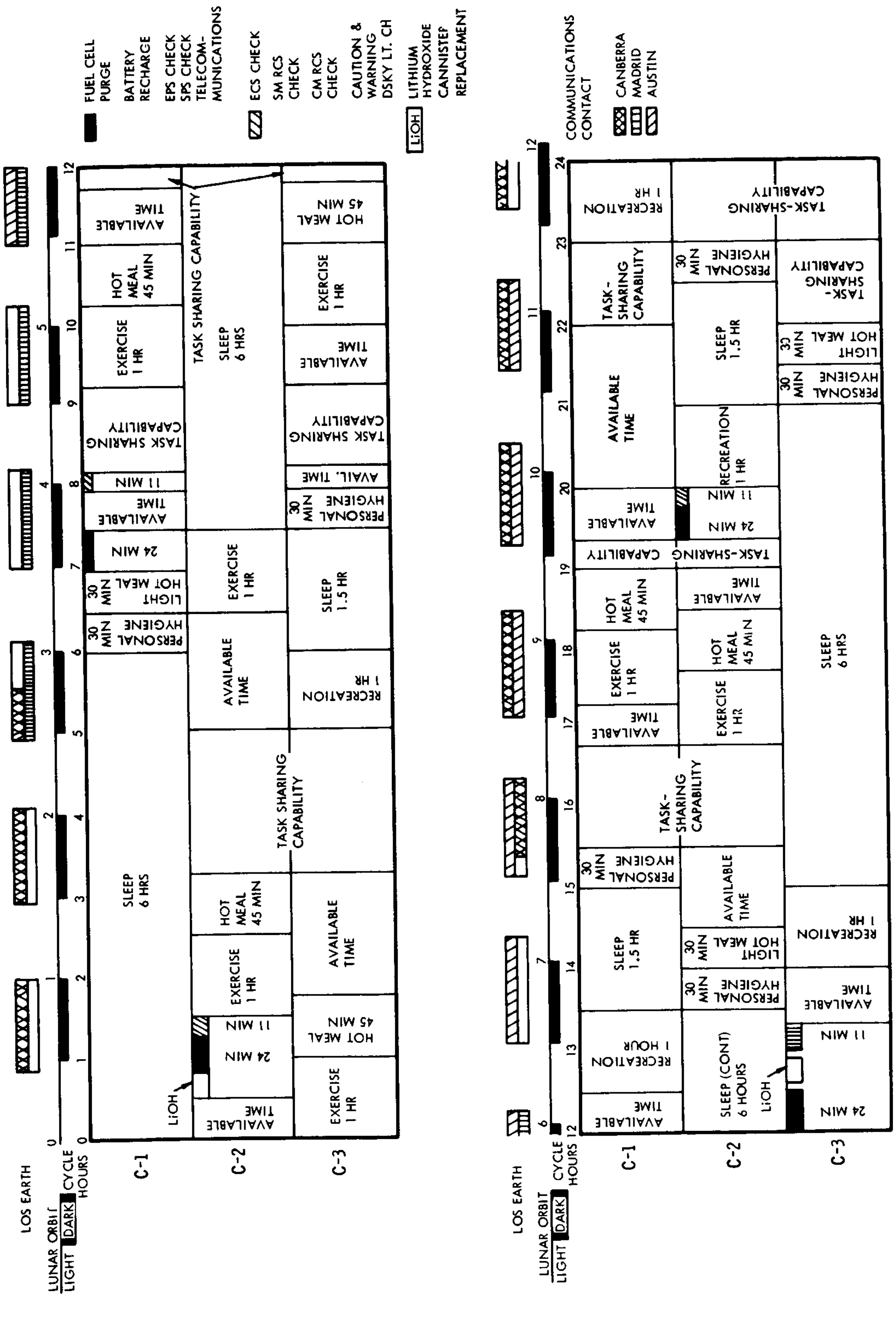


Figure 56. Crew Work-Rest Schedule, Nominal 24 Hours, Reference Mission IV

INDEPENDENT RCM LABORATORY CREW FUNCTIONS

While the actual crew functions to support the RCM laboratory module are essentially the same for either the dependent or independent concept, there are some differences in terms of work locations and the total amount of work to be done. In the dependent laboratory module, all station housekeeping functions are performed in the CSM with only one set of systems operable. However, in the independent laboratory concept, some of the station housekeeping functions are performed in the CSM, some are performed in the RCM laboratory, and some, because of duplicate systems in operation, are performed in both.

A listing of typical crew functions and the primary location where these functions are performed for the RCM independent laboratory concept has been compiled and is summarized in Table 32.

The characteristics of the RCM independent laboratory concept considered in determining the related crew functions are as follows:

1. Spacecraft propulsion, guidance and navigation systems will be removed during the renovation of the RCM laboratory module.

2. A new display and control panel will be developed for the RCM laboratory module.

3. The RCM laboratory module will be unmanned whenever it is not docked to a Command Module.

4. An airlock will be provided in the RCM laboratory module for extra-vehicular activity.

5. The docked station of the CM-RCM will provide for shirt sleeve ingress-egress between the two modules.

6. Command module systems operational status will be maintained at the necessary level to support personnel activities of the crewmen, safety monitoring, radiation shelter, patient care, and emergency escape.

7. Crew schedules shall provide for at least one crewman to be in the command module at all times as a safety precaution.

In addition to these RCM laboratory module characteristics and requirements, certain mission-related characteristics were considered in defining crew functions and scheduling requirements. Certain periodic tasks

Table 32. RCM Independent Laboratory Typical Crew Functions Performance Location

Crew Function	Task Description	Performance Location
Flight Mechanics	All	CM
Systems Operation	Status check	CM and RCM
	Navigation	CM
	Communications	CM and RCM
	Data mangement	CM and RCM
	Fuel cell purge	RCM
	Battery charge	CM
	LiOH Filter change	CM and RCM
Personal Activities	Sleep	CM
	Hygiene	CM
	Eating	CM
	Recreation	CM
	Personal time	CM
	Exercise	RCM
Safety	Radiation protection	CM
	Monitoring	CM and RCM
	Patient care	CM
	Emergency escape	CM
Logistics	Transfer	CM, RCM, EVA
	Service and supply	CM, RCM, EVA
	Repair/replace	CM, RCM, EVA

are scheduled more frequently during lunar orbit than during earth orbit to enhance the margin of safety by allowing more time for repairs and/or earth return. By scheduling system status checks six to twelve times per day during lunar orbit, as compared with three times per day in earth orbit, four to six hours more time would be available for emergency actions.

A summary of time, frequency, and location characteristics of system status monitoring is shown in Table 33 for Earth Orbit and Table 34 for Lunar orbits. Spacecraft systems management tasks, which are constant for both earth and lunar orbits, are summarized in Table 35. In each of the tables, the task characteristics have been summarized for one 20-hour day. To determine total mission requirements, it is necessary to multiply by the number of mission days for tasks located in the command module and by the number of orbital operation days for tasks located in the RCM laboratory module. To determine the number of minutes (hours) per man, it is necessary to divide by the number of crewmen.

A summary of time, frequency, and location of personal activities is shown in Table 36 for one crewman for one day. To determine total mission requirements, it is necessary to multiply these requirements by the number of crewmen and the number of mission days.

Exercise Activities

The primary location of exercise activity will be the RCM laboratory module. However, during periods when the RCM is unmanned, (pre- and post-orbital mission phases) or when not docked to the command module, exercise activities will be performed in the command module.

Personal Hygiene Activity

The primary location for these activities is in the command module. However, emergency facilities for urination, defecation, and hand cleansing will be provided in the RCM.

Eating

The primary location of meal preparation and eating activities will be the command module. However, emergency rations and light snacks will be available in the RCM laboratory module.

Crew mobility time characteristics presented in Table 34 through 37 were determined on the basis of analysis conducted as a part of the Extended Mission Apollo Studies. From these studies, travel rates were determined

Table 33. RCM Independent Laboratory Concept, Summary of Time, Frequency, and Location Characteristics Systems Status Monitoring Tasks, Earth Orbit

Task Description	CM			RCM		
	Task Duration (hours)	Task Frequency (No. /day)	Total Task Time (hours)	Task Duration (hours)	Task Frequency (No. /day)	Total Task Time (hours)
System Status Checks			0000			
ECS	0.067	3	0.20	0.067	3	0.20
EPS and Battery	0.150	3	0.45	0.167	1	0.17
Caution, warning, DSKY	0.067	3	0.20	0.017	3	0.05
Communications	0.050	3	0.15	0.017	3	0.05
Life Support	0.100	3	0.30	0.100	3	0.30
RCS	0.067	3	0.20	0.025	3	0.08
SCS	0.003	3	0.01	0.002	3	0.01
SPS	0.017	3	0.05	-	-	-
Travel						
Aisle-Couch	0.012	6	0.07			
Aisle-Chair				0.012	6	0.07
RCM-CM				0.002	6	0.01
Total Hours			1.63			0.94

Table 34. RCM Independent Laboratory Concept, Summary of Time, Frequency, and Location Characteristics Systems Status Monitoring Tasks, Lunar Orbit

Task Description	CM Task Duration (hours)	CM Task Frequency (No./day)	CM Total Task Time (hours)	RCM Task Duration (hours)	RCM Task Frequency (No./day)	RCM Total Task Time (hours)
Systems Status Check						
ECS	0.042	12	0.50	0.042	12	0.50
EPS and battery	0.084	12	1.01	0.167	1	0.17
Caution, warning, DSKY	0.067	6	0.40	0.017	6	0.10
Communications	0.002	6	0.01	0.002	6	0.01
Life Support	0.100	6	0.60	0.100	6	0.60
RCS	0.042	6	0.25	0.025	6	0.15
SCS	0.002	2	0.01	0.002	12	0.02
SPS	0.025	6	0.15	-	-	-
Travel						
Aisle-couch	0.012	24	0.29			
Aisle-chair				0.012	24	0.29
RCM-CM				0.002	24	0.05
Total hours			3.22			1.89

Table 35. RCM Independent Laboratory Concept, Summary of Time, Frequency, and Location Characteristics, Systems Management Tasks

Task Description	CM			RCM		
	Task Duration (Hours)	Task Frequency (No./Day)	Total Task Time (Hours)	Task Duration (Hours)	Task Frequency (No./Day)	Total Task Time (Hours)
LiOH filter change	0.084	2	0.17	0.084	2	0.17
Fuel cell purge H_2				0.050	1	0.05
Fuel cell purge O_2				0.050	1	0.05
Battery charge	0.334	1	0.33			
Space suit system	0.042	1	0.04	0.042	1	0.04
Travel						
Aisle - couch						
Aisle - chair				0.012	24	0.29
RCM - CM				0.002	7	0.01
Total Hour			0.54			0.61

Table 36. RCM Independent Laboratory Concept, Summary of Time, Frequency, and Location Characteristics, Personal Activity Tasks for One Crewman

Task Description	CM			RCM		
	Task Duration (Hours)	Task Frequency (No./Day)	Total Task Time (Hours)	Task Duration (Hours)	Task Frequency (No./Day)	Total Task Time (Hours)
Personal						
Sleep	6.00	1	6.00			
Nap	1.50	1	1.50			
Hygiene	0.50	2	1.00			
Safety	0.60	1	0.60	1.00	1	1.00
Hot meal	0.75	2	1.50			
Light meal	0.50	1	0.50			
Recreation	1.00	1	1.00			
Exercise				1.00	2	2.00
Travel						
Aisle - couch	0.012	14	0.17			
Aisle - chair				0.012	2	0.02
RCM - CM				0.002	6	0.01
Total Hours			12.27			3.03

to be approximately 10 feet per second during zero "g" and 5 feet per second during artificial "g". For the worst case, travel time from the farthest point in the RCM module to the command module tunnel would be approximately 2.5 seconds in zero "g" and 5.0 seconds in artificial "g". Command module entry requires approximately 2.0 seconds, with aisle/couch or aisle/chair seating and securing requiring approximately 40 seconds. On the basis of these figures the following travel times were developed:

Aisle - couch	0.012 hour
Aisle - chair	0.012 hour
RCM - CM	0.002 hour (zero "g")

RCM- command module travel time includes ingress/egress through the command module tunnel.

A summary of the crew time characteristics for each reference mission for the RCM independent laboratory concept has been compiled and is presented in Table 37. Representative work-rest schedules were compiled using the data summarized in the foregoing tables. These schedules illustrate the application of the crew time and performance location criteria in the development of the most efficient utilization of crew time and location as a function of mission requirements. Typical twenty-four cycles for three crewmen are presented in Figure 57 for earth orbit, and in Figure 58 for lunar orbit.

To effect the most efficient utilization of crew time and location, certain systems status monitoring and systems management tasks were grouped together. To establish tentative schedules, these activities were grouped into 30-minute packages as follows:

1. Earth orbit - command module - Systems Package 1 and 2: a complete systems status check plus one LiOH filter change. Systems Package 3: a complete systems status check plus the space suit system check.

2. Earth orbit - RCM - Systems Package 1 and 2: a complete systems status check plus one LiOH filter change, and one fuel cell purge. System Package 3: a complete systems status check plus the daily EPS check.

3. Lunar orbit - CM - Systems Package 1 for Crew 1 and 2: EPS, ECS, life support systems status check, and one LiOH filter change. Systems Package 1 for Crew 3: the systems status check listed above plus the space suit systems check. Systems Package 2 for all crewmen: EPS, ECS, Communication, DSKY, SPS, and RCS system status checks.

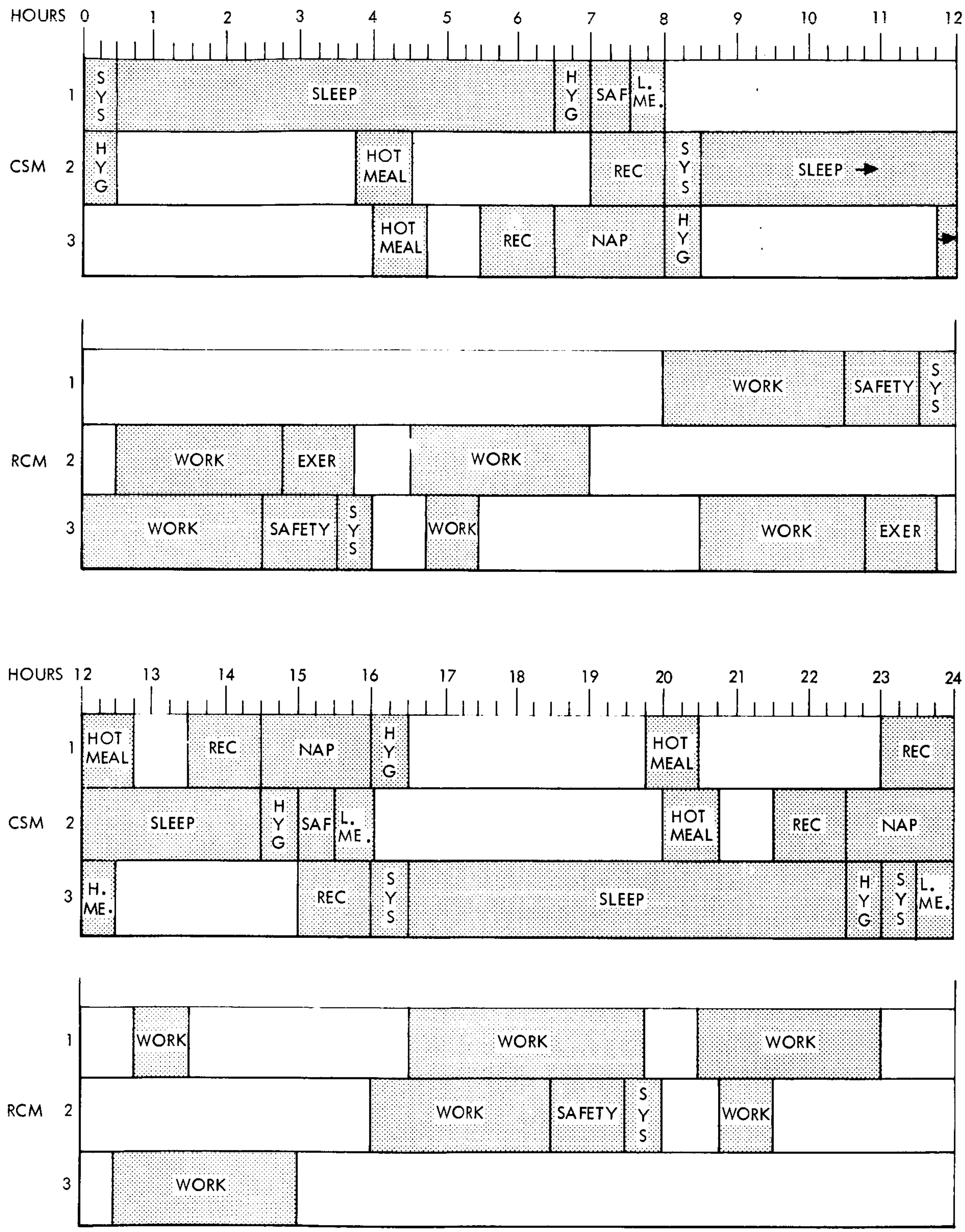


Figure 57. Work-Rest Cycle, Earth

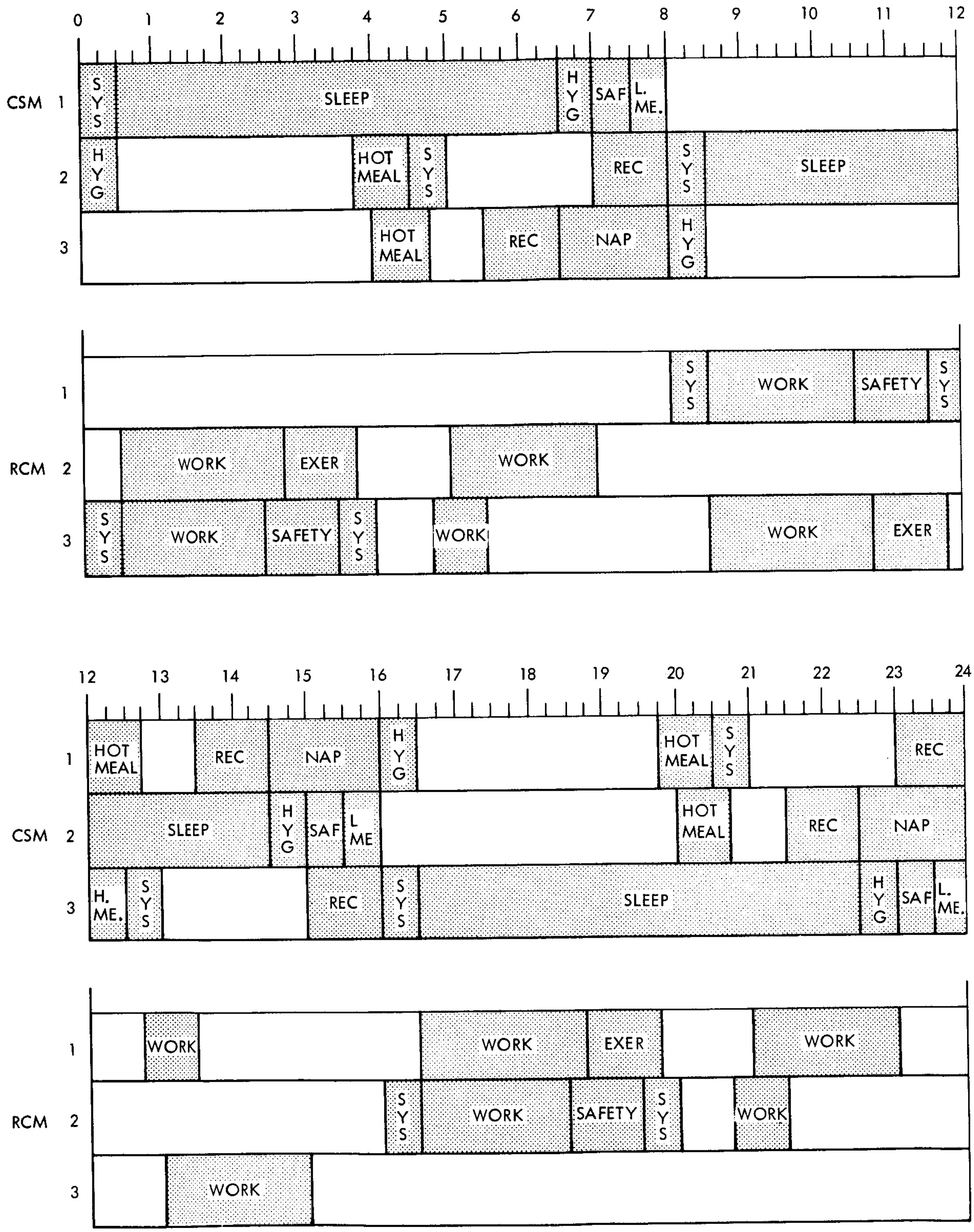


Figure 58. Work-Rest Cycle, Lunar

4. Lunar Orbit - RCM - Systems Package 1 for Crew 1 and 2: ECS, SCS life support, and communications systems check plus one LiOH filter change. Systems Package 1 for crew 3: the systems status checks listed above plus the space suit systems check. Systems Package 2 for Crew 1 and 2: ECS, SCS, caution and warning, and DSKY systems status check plus one fuel cell purge. Systems Package 2 for Crew 3: the systems status checks listed above plus the daily EPS check. All packages include travel time between modules, ingress-egress, and seating.

Since these packages have been closely coordinated with the personal activities and their locations, travel time has been subtracted from the personal activities time requirements. The revised crew time characteristics for the RCM-Independent Laboratory module concept for each of the reference missions is presented in Table 38.

Table 38. Revised Crew Time Characteristics, RCM Independent Laboratory Concept Per Man Per Day

Activity	Reference Mission				
	A ELIO	B-1 EPO (SIB)	B-2 EPO (SV)	C ESO	D LPO
Housekeeping					
Personal	15.0	15.0	15.0	15.0	15.0
Systems Packages	1.0	1.0	1.0	1.0	2.0
Total	16.0	16.0	16.0	16.0	17.0
Experiments					
Available	8.0	8.0	8.0	8.0	7.0
Total	24.0	24.0	24.0	24.0	24.0

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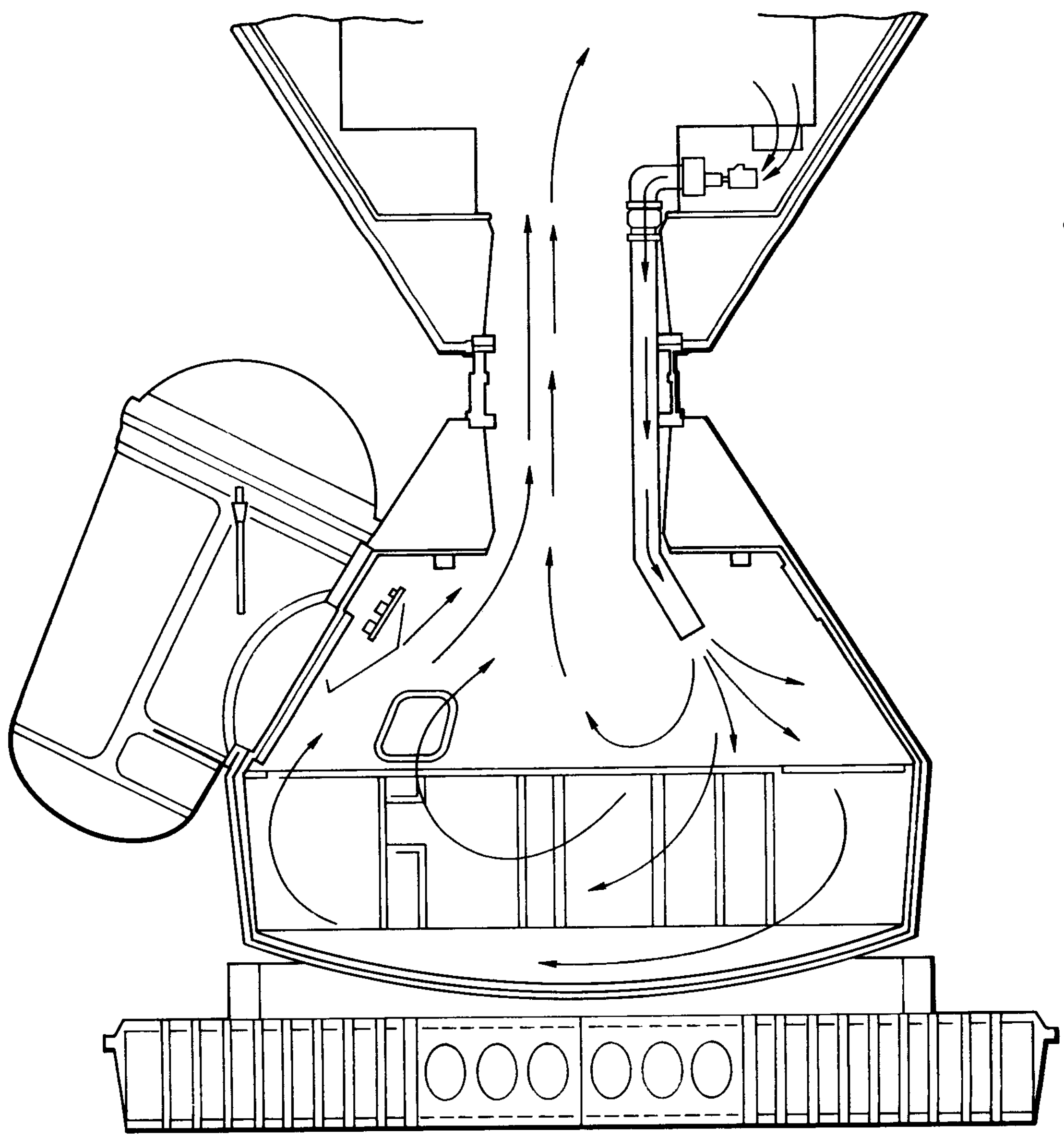
V. RCML THERMAL ANALYSIS

Thermal analyses were performed to establish an insulation and thermal-control coating scheme for passive thermal control of the dependent RCML for AAP reference missions. Optical properties of an external surface coating may be used to passively control heat loss or gain from the laboratory relative to a maintained interior air temperature of 75 F. Heat loads to the laboratory interior which have been generaged show the effects of orbital conditions as a function of optical properties of the surface coating. No one coating will provide a fixed net heat loss or gain for all orbital conditions. Selection of a coating for a fully dependent laboratory must be based on CM ECS capabilities, internal loads, and mission requirements. External surface temperatures are also dependent upon the mission and upon surface optical properties. Temperatures on the internal structure walls, which vary from a high of 90 F to a low of 58 F (with localized excursions beyond these values), indicate adequate performance of the insulation in general. Additional analyses are necessary to isolate and correct local heat shorts and temperature problems.

CONFIGURATION

The configuration which was analyzed is shown in Figure 59, and is a completely dependent lab with ECS maintaining the internal atmosphere. Air temperature was assumed to be regulated at 75 F, with the equipment and metabolic heat loads being rejected through the CM ECS. The primary concern of these analyses was the heat loading resulting from the natural space environment; structural wall temperatures were also of interest.

The laboratory configuration represents a considerable departure from that of the Apollo command module with respect to exposure to the thermal environment encountered in space and orbital flight. The external surface of the CM is a cone, while the pressure shell of the laboratory is a combination of a conical surface above the girth and a nearly cylindrical surface, below the girth. About one third of the total external surface of the laboratory is contained in each of these two surface areas. In addition, the base or aft bulkhead of the CM and the three tension ties are contained within the insulated CSM adapter and have negligible effect on the heat balance of the crew compartment; in contrast, the base of the laboratory is exposed and results in an increase of approximately 30 percent in external



- DEPENDENT LAB CONFIG
- SHIRT SLEEVE ENVIRONMENT
- HEATING & COOLING PROVIDED BY CM ECS ONLY
- CONSTANT AIR TEMPERATURE 75 ± 5°F IN LABORATORY
- PASSIVE TEMPERATURE CONTROL FOR MINIMUM EFFECT FROM SPACE ENVIRONMENT

Figure 59. Passive Thermal Control of the Dependent Laboratory

surface area. Further, the six bolts which connect the base of the laboratory to the SLA support structure supplanting the three tension ties of the CSM configuration provide direct conduction paths from the internal walls to the external environment. These paths make a significant contribution to the heat losses and also provide a noticeable percentage of the heat gained during conditions that result in external heating.

MISSION ASPECTS

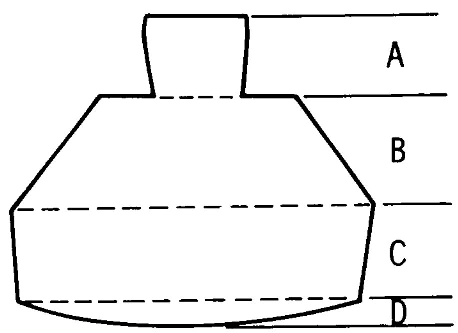
AAP study results indicate that these three orbits provide the highest heating rates: Earth polar-subsolar with +X-axis perpendicular to sun; Earth terminator with +X-axis perpendicular to sun; and Lunar polar-subsolar with +X-axis perpendicular to sun. In each case the heating loads to CM ECS are greatly influenced by sunlight transmittal through the windows. Maximum heat losses result from the passive-temperature-control maneuver of Apollo cislunar flight which requires one revolution per hour about the X-axis normal to the sun line with interruptions up to three hours of inertial hold were considered. These missions were used as the basis of the heat-transfer studies.

INSULATION

Surface temperatures of the RCML 0.016-inch-thick aluminum meteoroid shield were calculated for the selected orbital conditions and vehicle orientations. The optical properties of the shield were varied to determine a specific value for emissivity and the optimum value for the ratio of solar absorptivity to emissivity. The surface temperatures were then used to calculate the heat balance on the RCML. The heat balance on the laboratory was based on an insulation design consisting of 40 layers of crinkled NRC-2 aluminized mylar loosely confined within a space of 1/4-inch to 1/2-inch between the meteoroid shield and laboratory structure (Figure 60). The insulation should not contact the laboratory structure except through supporting fiberglass brackets, preferably of phenolic-nylon composition and minimum structural size. Nylon or teflon bolts fasten the brackets, in a manner to provide minimum contact area between bracket and laboratory structure. There were assumed to be eight brackets circumferentially at four axial positions. The external surface of the laboratory structure was assumed to be cleaned and coated or polished to provide a surface emissivity of 0.1 or less.

For an attitude of X axis normal to the sun, heat loads through the windows of 334 Btu per hour were calculated, based on a total effective transmissivity of 0.45. This heat load can occur in deep space flight during a three-hour inertial hold, synchronous orbit mission, or polar orbit mission, and may be large with respect to the totally dependent laboratory. Provisions for covering the windows are recommended.

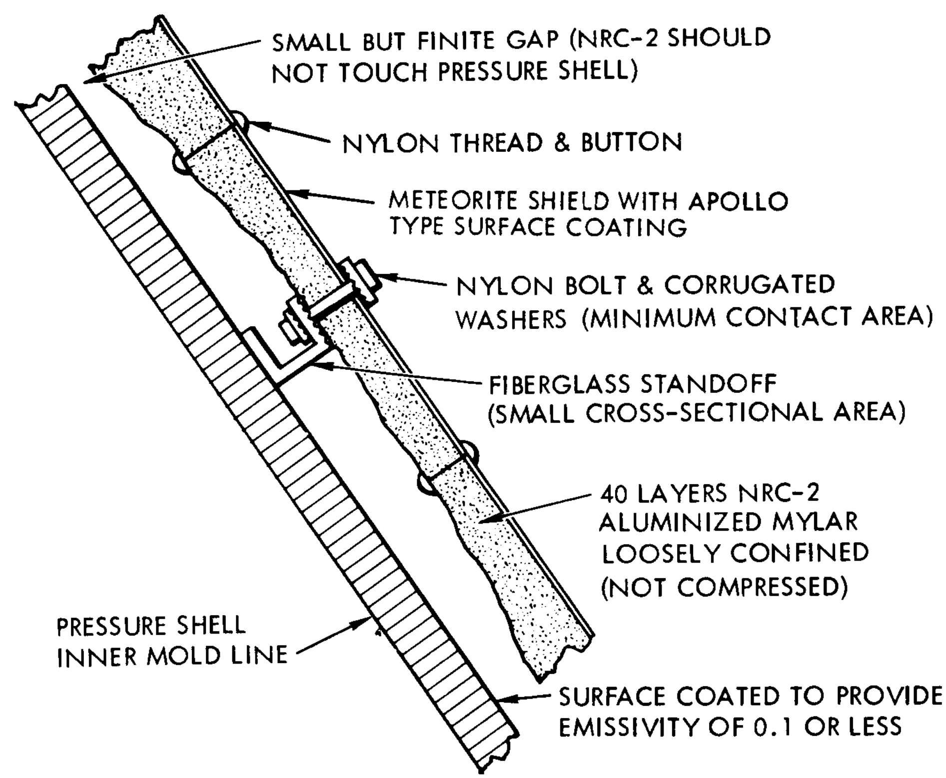
AVERAGE INNER WALL
SURFACE TEMPERATURES



ALL ORBIT CONDITIONS

	INNER SURFACE AVG TEMP °F MAX	INNER SURFACE AVG TEMP °F MIN
A	90	58
B	85	73
C	88	72
D	82	68

TYPICAL INSTALLATION OF THERMAL
INSULATION AROUND RCML



UPPER TEMPERATURE LIMIT OF NCR-2 : +250°F

Figure 60. RCML Insulation and Temperature Distribution

At temperatures above 250 F, the mylar of the insulation begins to soften and the layers fuse together providing direct conduction paths rather than multiple radiation shields. To protect the NRC-2 from exposure to temperatures above 250 F, a thin layer of silica-fiber insulation such as Q-felt or TG-15000 might be installed between the meteoroid shield and the NRC-2. The maximum external surface temperature is controlled by the surface coating ratio of solar absorptivity (α) to emissivity (ϵ). When this ratio is greater than unity, temperatures exceeding 250 F will result when the surface is directly in the sunlight. Therefore, the requirement for a layer of insulation to protect the NRC-2 will rest with the selection of a coating and the associated α / ϵ ratio.

HEAT LOADS AND SURFACE TEMPERATURES

Figure 61 shows the variation of net heat load to the laboratory interior as affected by orbital and space flight conditions, RCML orientations, and the ratio of α to ϵ; positive values are heat gains, and negative values are heat losses. All results are based on maintaining an RCML air temperature of 75 F. This implies that heat gains are removed by an ECS and heat losses are compensated either by an ECS or by heat dissipated by electronic equipment. Heat loads shown on Figure 61 include the maximum amount of heat transmitted through the windows for each flight condition considered.

Figure 61 indicates that no one design criteria, such as a fixed net loss or gain for all orbital conditions, can be satisfied with one value of the α / ϵ ratio. If all the curves in Figure 61 intersected each other at the same point, this one condition could be satisfied in all orbits with one α / ϵ value; however, it would be only coincidence if this were the desired net heat loss or gain.

Figure 62 presents a consideration of the same information shown in Figure 61 except that the heat load transmitted through the windows, which can be as high as 36 percent of the total heat load is eliminated from consideration. The important effect of the windows on the heat balance of the laboratory is easily identified by comparing Figures 61 and 62. For example, in Figure 61 for the ratio of α to ϵ of 0.5, the approximate range of net heat loss is -550 Btu to -1900 Btu, depending on flight conditions and specific values of ϵ. In Figure 62 for the same value of α / ϵ, the range is approximately -750 Btu to -2850 Btu. It is obvious from this comparison that a window louver system or adjustable shade would provide an additional means of controlling the laboratory heat balance in a passive manner.

The highest heating conditions for the laboratory are encountered in those earth and lunar orbits that pass over the subsolar point in a local vertical orientation with the X-axis along the velocity vector. This orientation exposes the base to solar heating, which results in higher mean temperatures on the external surface. A comparison of Figures 63 and 64, which present the external-surface mean temperatures in earth orbit for the three geometrical divisions of the structure, illustrates this point. Figure 65 shows the difference in surface mean temperatures in lunar orbit with respect to the side of the spacecraft exposed to the sun and the side facing the lunar surface. Figure 66 presents the external surface mean temperatures that will occur in high-altitude orbits and deep space. Figure 67 shows the effect of α/ϵ and specific values of ϵ on minimum mean temperatures reached during flight in lunar shade. Figures 63 through 67 also illustrate the effect of geometrical shape on the resulting mean temperatures. Figure 68 presents the maximum temperatures that may occur as a function of α/ϵ, and these maximum temperatures shown will occur in all phases of space flight. Figure 69 illustrates the minimum temperatures that may result as a function of surface emissivity (ϵ). The minimum temperature in lunar shade from Figure 67 is -215 F, however, lower temperatures, as shown in Figure 68, may occur during a three-hour hold in deep space or high altitude orbits, the absolute minimum depending on a specific coating an and its associated value of ϵ.

Figures 70 and 71 show the transient temperatures of the external surface sectors of the RCM laboratory in earth orbit. Comparison of the two figures illustrates the effect of utilizing surface coatings with different thermal characteristics. The effectiveness of the insulation is also affected in a particular way by the coating on the external surface. If the insulation provides only a resistance to heat conduction, the rate at which heat is transferred in or out will be a linear function of the external-surface mean temperature. The effectiveness of this type of insulation is far surpassed, at least at moderate and low temperatures, by essentially eliminating heat conduction and allowing heat to be transferred by radiation only. The rate at which heat is transferred in this manner is a function of the external surface absolute temperature raised to the fourth power. As a result, the laboratory interior will heat more rapidly than it will cool. Therefore, to minimize this effect due to environmental heating, it would be desirable to select a coating that would limit the maximum external surface temperature to approximately 75 F or lower, since the insulation becomes more effective as the temperature goes down.

Heat shorts, which are direct conduction paths, constitute linear heat transfer mechanisms. In addition to affecting the heat balance by this direct conduction, they cause localized hot or cold regions that may cause structural and component distortions, condensation, or hot spots hazardous to the crew. The six support bolts can, in such orientations as rolling about the X-axis normal to the sun, make the major contribution to the total heat loss. To help control this loss, the laboratory SLA support structure should be coated to provide a minimal emissivity value with the ratio of α to ϵ less than unity.

INTERNAL WALL TEMPERATURES

The laboratory internal walls are divided into four general regions: forward bulkhead and tunnel, conical section above the girth, cylindrical section below the girth, and the base. The maximum and minimum average temperatures that may occur are shown in Figure 60. There will be localized temperature excursions above and below these values; however, a detailed transient analysis would be necessary to provide these data.

CONCLUSIONS AND RECOMMENDATIONS

The insulation scheme which has been described will provide moderate wall temperatures. Heat loss or gain consistent with the CM ECS capability (Volume III, Section XVI) can be controlled by the selection of proper coatings. However, no one coating will satisfy all orbital and spaceflight conditions. It is recommended that consideration be given to selecting the external coating to match particular mission and laboratory-performance requirements. It is also recommended that a coating be selected that will limit the maximum external surface temperature to 250 F to maintain proper insulation performance without requiring additional high-temperature insulation. In addition, it is recommended that consideration be given to providing a controllable shade over each RCM laboratory hatch window.

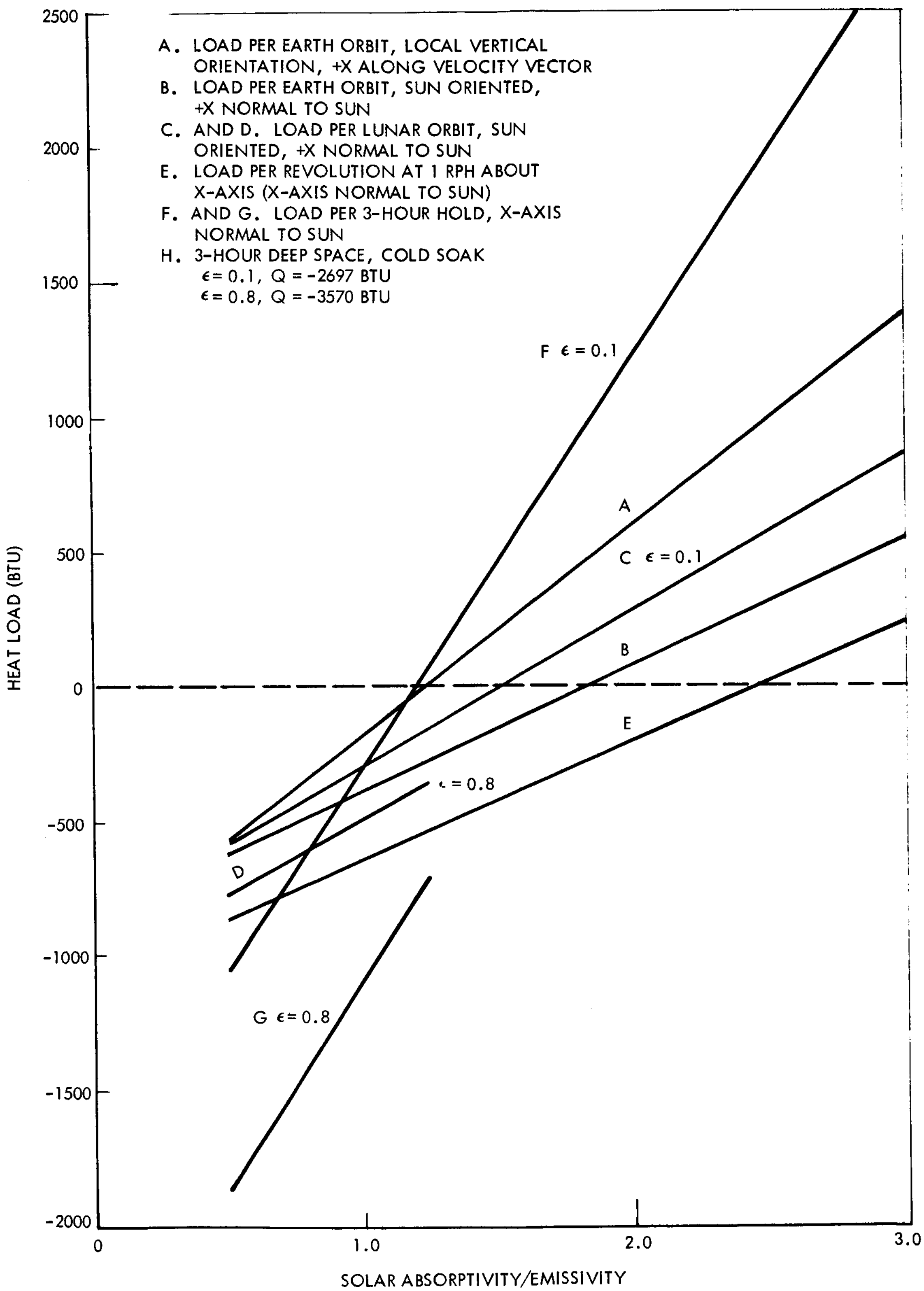


Figure 61. Total Heat Load

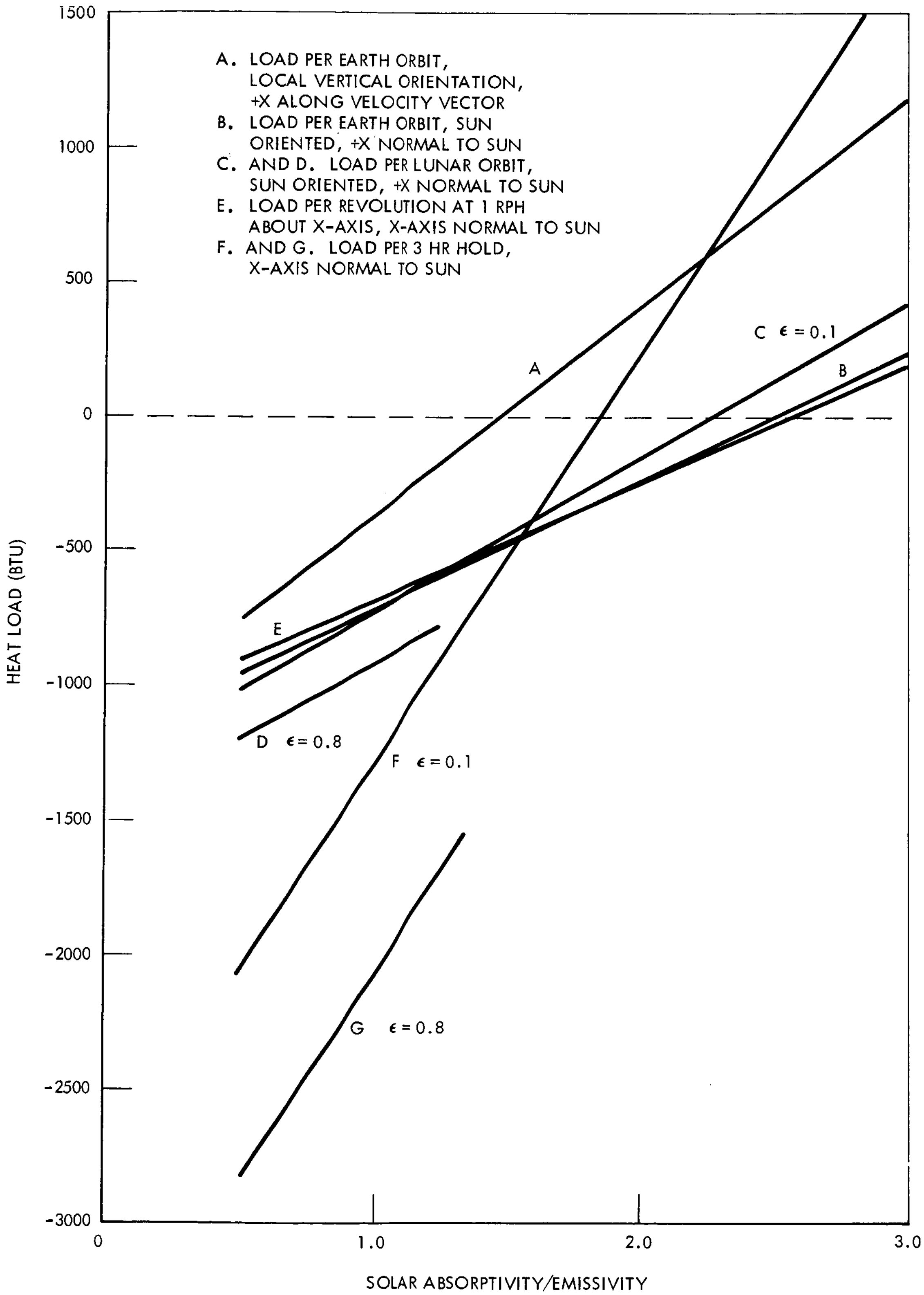


Figure 62. Net Heat Load Minus Window Load

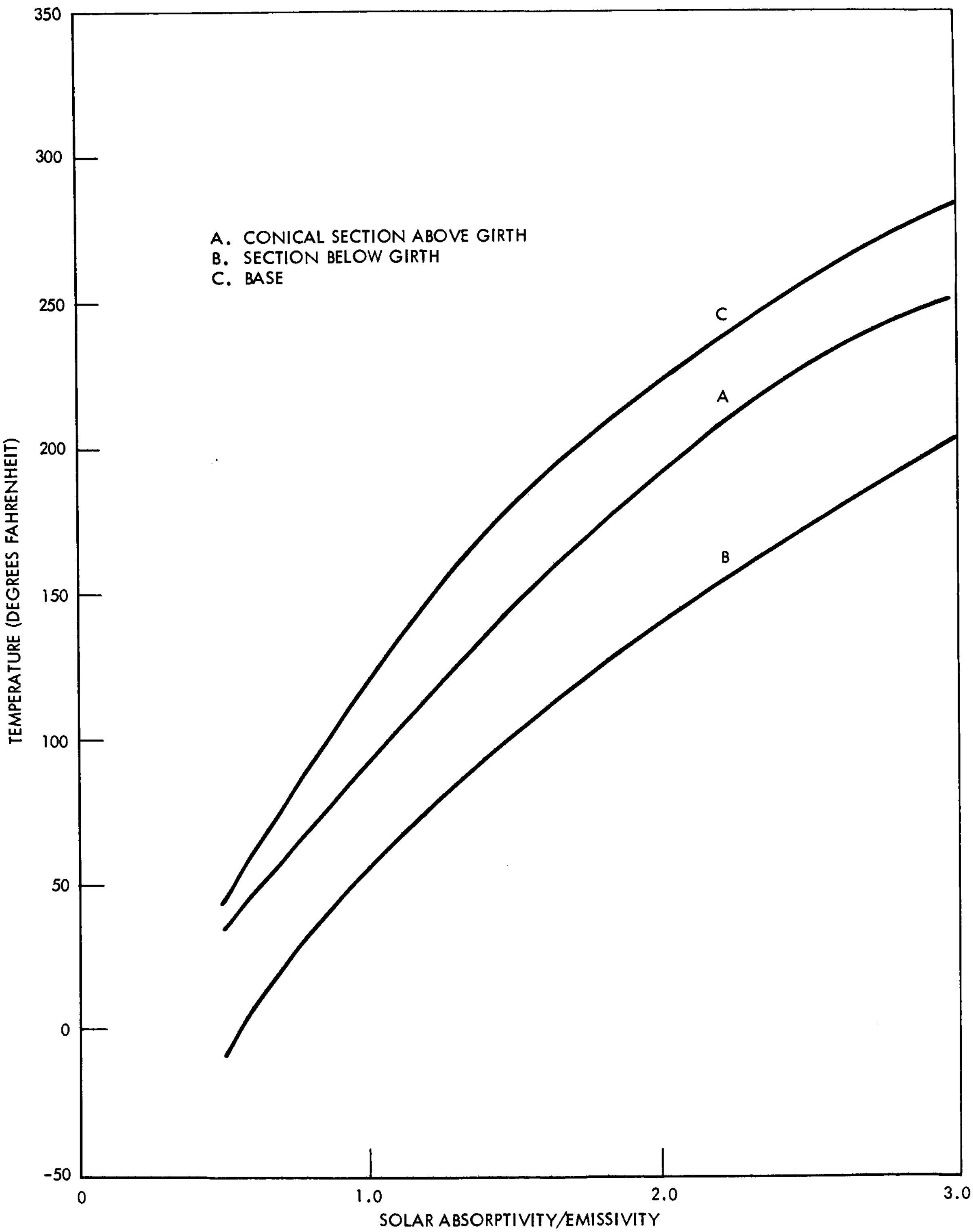


Figure 63. Mean Temperatures of Exterior Surface, Earth Orbit, Local Vertical Orientation, +X Axis Along Velocity Vector

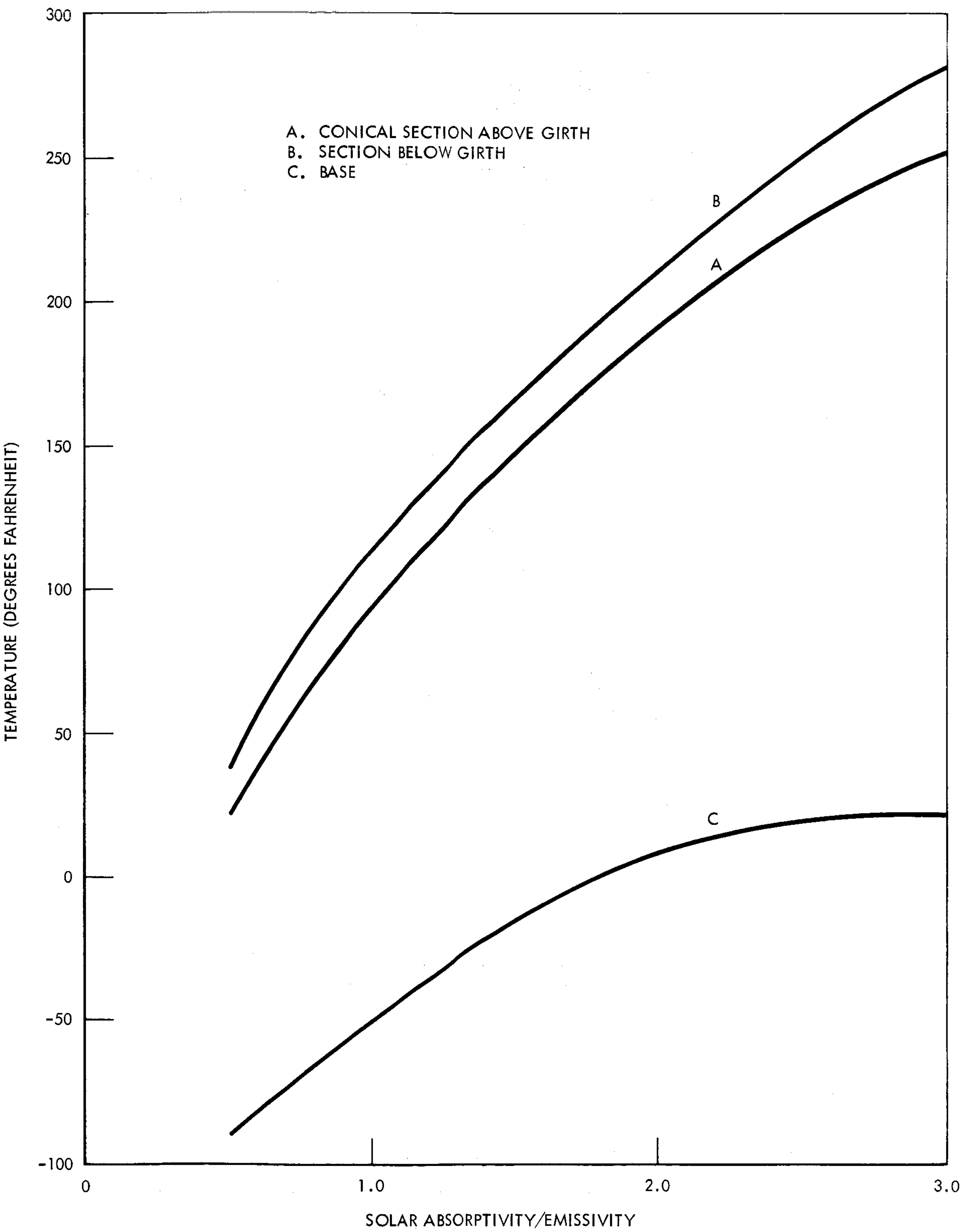


Figure 64. Mean Temperatures of Exterior Surface, Earth Orbit, Sun Oriented, +X Axis Normal to Sun

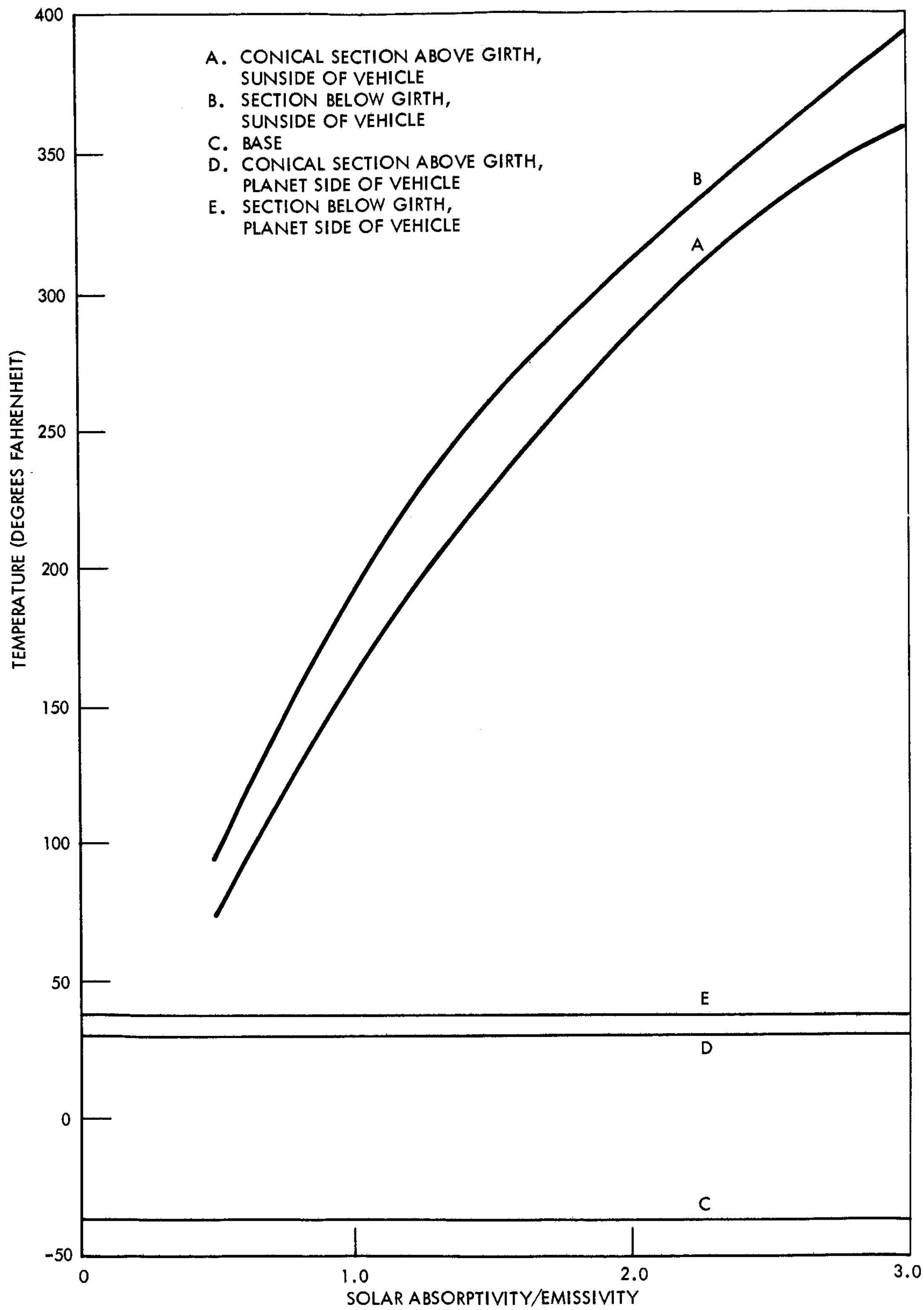


Figure 65. Mean Temperatures of Exterior Surface, Lunar Orbit, Sun Oriented

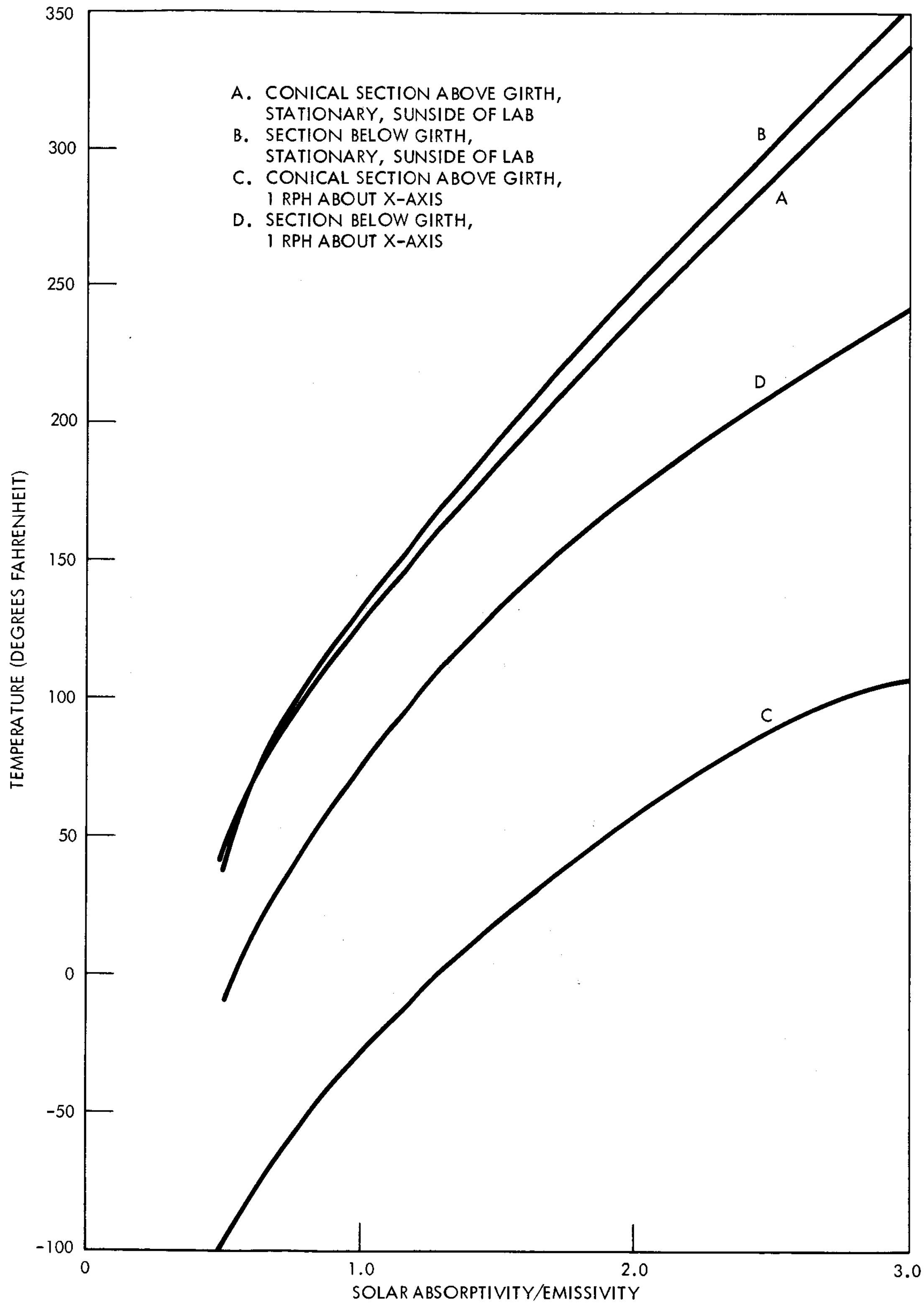


Figure 66. Mean Temperatures of Exterior Surface, Cislunar Flight, Terminator, and Synchronous Orbits, +X Axis Normal to Sun

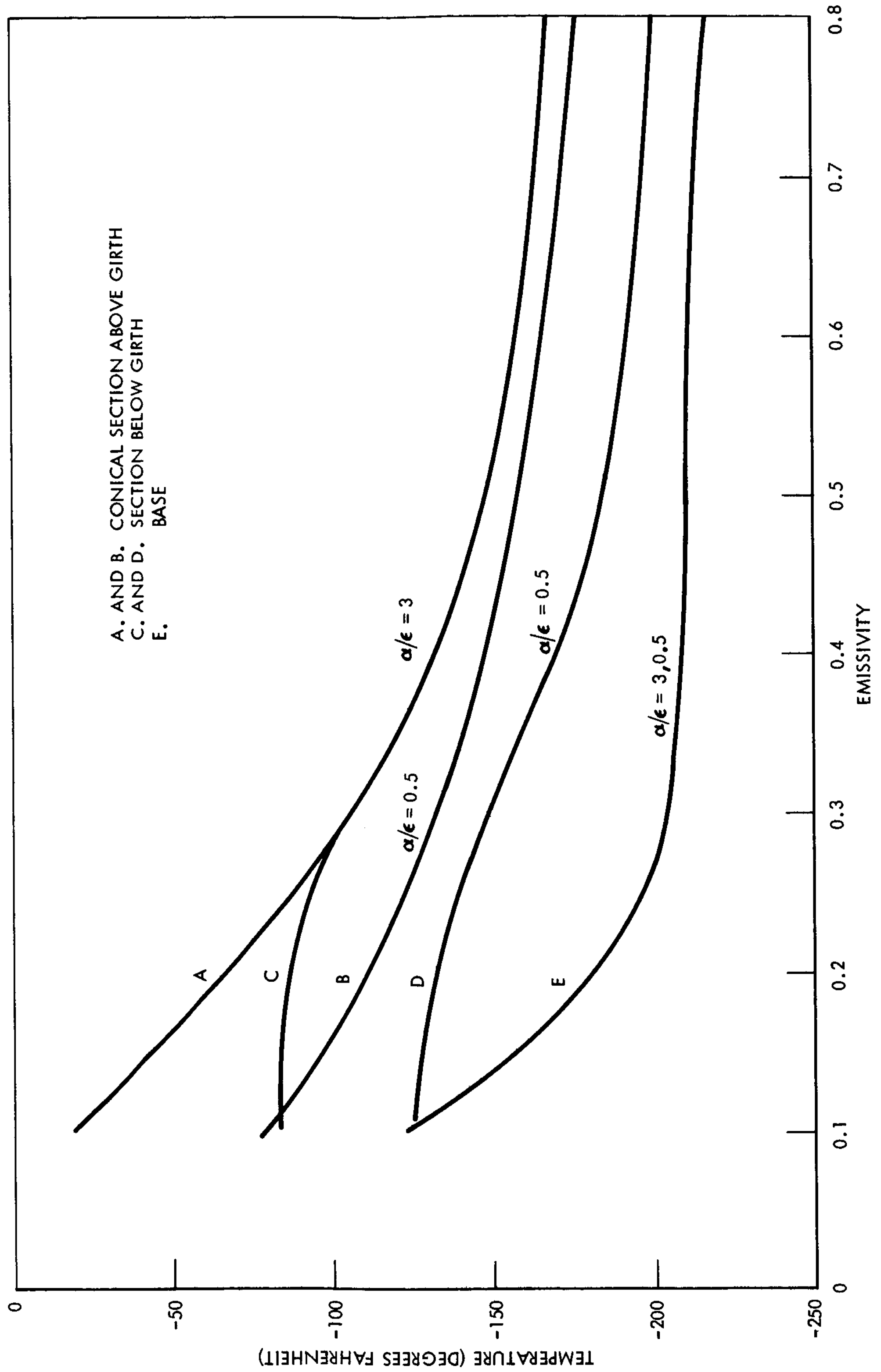


Figure 67. Mean Temperatures of Exterior Surface, Time Averaged Through Lunar

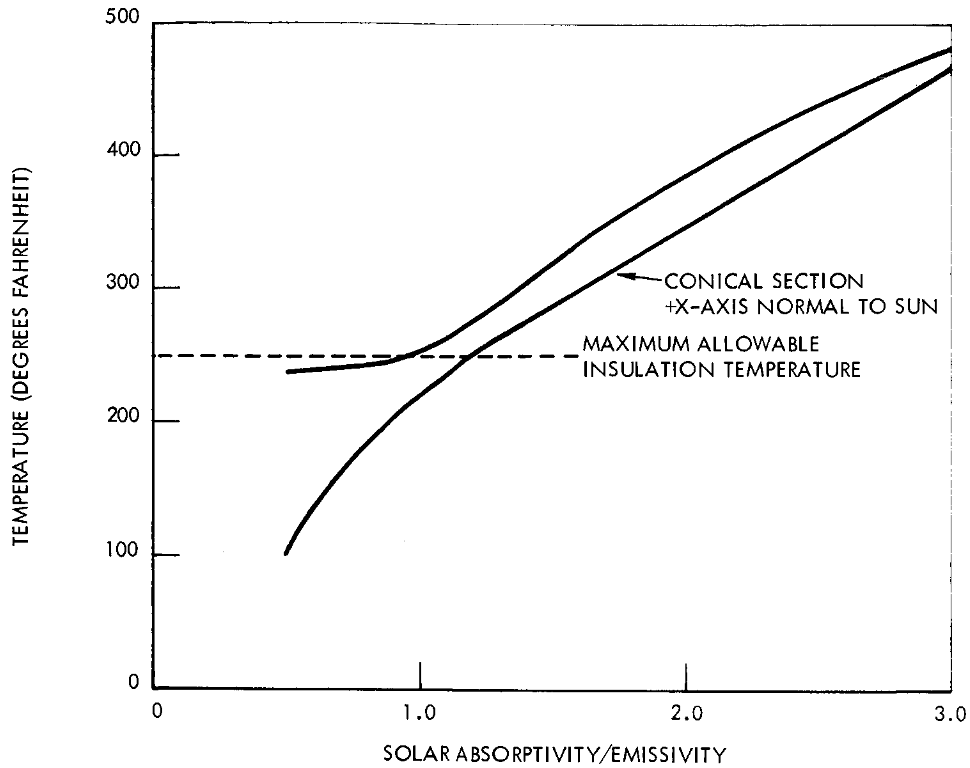


Figure 68. Maximum External Surface Temperatures

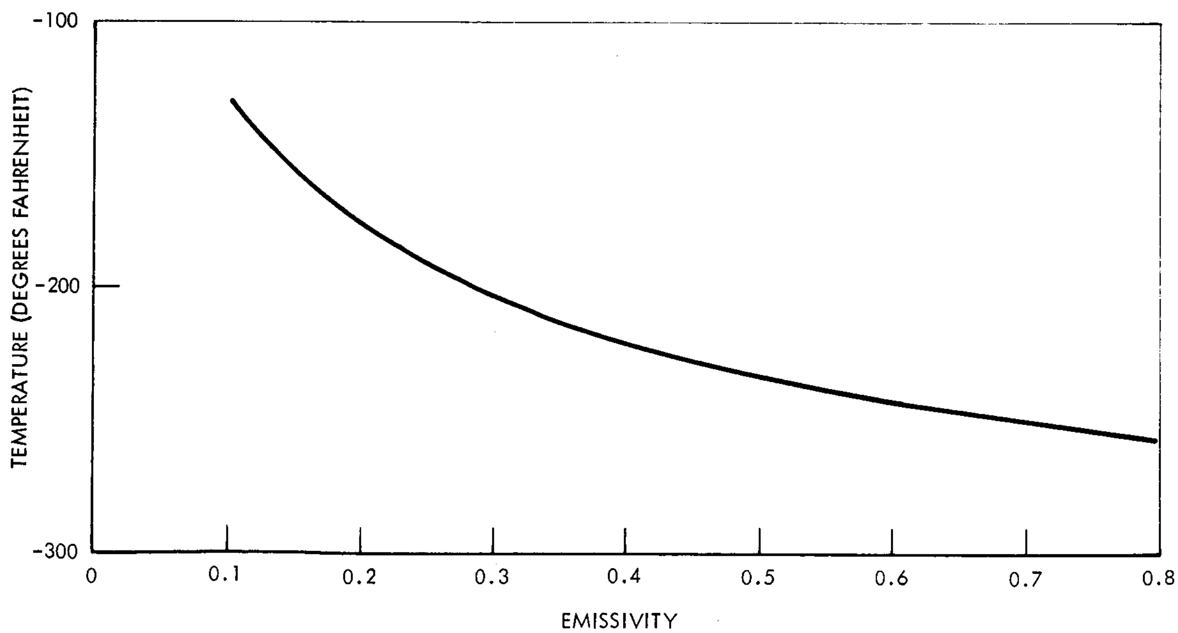


Figure 69. Minimum External Surface Temperatures

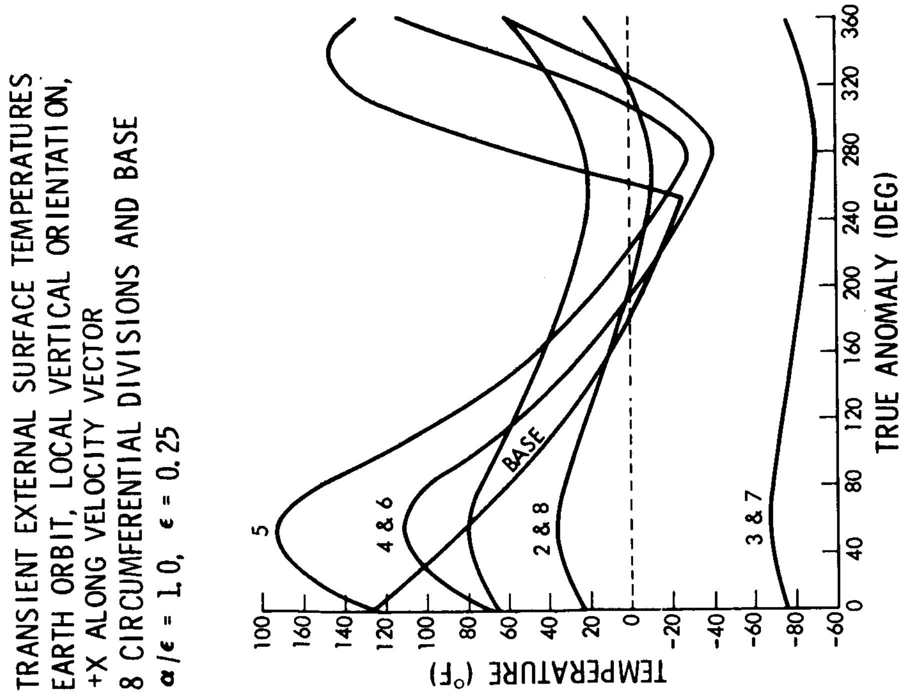


Figure 70. RCM Laboratory External Surface Transient Temperatures for $\alpha/\epsilon = 1.0$

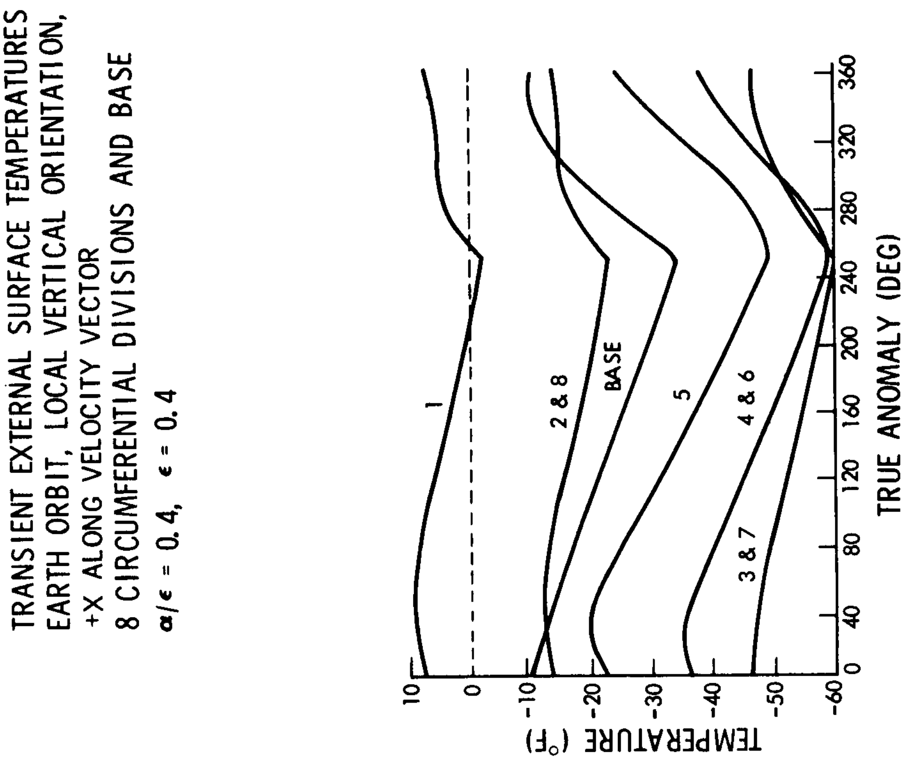


Figure 71. RCM Laboratory External Surface Transient Temperatures for $\alpha/\epsilon = 0.4$

VI. SPACE PHYSICS

MICROMETEOROID SAFETY ANALYSIS

Computation of the micrometeoroid shielding required for the dependent RCM laboratory flight configuration (including the CSM) was performed to meet the Apollo requirement of no micrometeoroid penetration probability, $P_o = 0.995$.

Two NASA-MSC meteoroid environments were considered: the EC-1 environment and the Revised EC-1 environment. In both cases, the meteoroid density and velocity were the same. The fluxes, however, were different in that the EC-1 flux is essentially the 1963 Whipple flux model. The Revised EC-1 environment is considered less severe for the smaller mass meteoroids.

The methods used in this analysis are those developed for Apollo. Penetration mechanics are the most recent developed at NAA-S&ID and are based on extensive hypervelocity impact testing.

Analysis Criteria and Guidelines

The following criteria and guidelines were used as the basis for the shielding analysis:

1. Consider the study's three basic missions: low altitude earth orbit, synchronous earth orbit, lunar orbit (all of 15-, 30-, and 45-day duration).

2. Study range of applicable environmental conditions.

3. Define the meteoroid shielding requirements

4. Define pertinent shielding design constraints in regard to RCM laboratory components and component installation.

5. Consistent with 2 above, use both NASA EC-1 flux,

$$\log N = -1.34 \log M - 10.423 + \log A$$

as used on Apollo, and use the revised flux NASA is proposing for Apollo

$$\log \frac{N}{1.4} = -\log M - 9.69 + \log A$$

In both cases use

$P_m = 0.5$ gram/cm^3

$V_m = 30$ km/sec.

$A = 1.0$ for cislunar space

$A = 0.5$ for near Earth.

6. Define meteoroid shielding to give 0.995 probability of no mission abort.

7. Use NAA/SID penetration equations and finite sheet factor

   a. Metal $\begin{cases} p = 1.38\, d_m^{1.1}\, P_m^{1/2}\, V_m^{2/3} \big/ P_t^{1/6}\, H_t^{1/4}, \text{ cm} \\ t = 1.8\, P \end{cases}$

   b. Glass $p = .64\, d_m^{1.2}\, P_m^{1/2}\, V_m^{2/3}$, cm

   c. Ablator $p = 2.52\, d_m^{1.3}\, P_m^{1/2}\, V_m^{2/3}$, cm

8. Use the following failure modes for system components:

   a. SPS tanks - penetration greater than one quarter of the wall = cracking and propellant leakage and mission abort.

   b. CM heat shield - any full depth ablator penetration = failure on entry.

   c. CM laboratory - any perforation = mission abort.

   d. ECS and EDS radiator tubes - puncture of any two of four circuits = mission abort.

   e. CM heat shield windows - any penetration over 0.80-inch into structural window = failure of structural window and crew loss.

f. CM laboratory windows - any penetration over 0.001-inch in structural window = failure of outer structural window and loss of thermal protection.

g. Airlock - numerous perforations result in O_2 loss and mission abort.

h. External O_2 tank - any perforation allows O_2 loss and mission abort.

Meteoroid Environment

MSC meteoroid environments considered. The EC-1 environment and a revised EC-1 environment, which are shown in Figure 72. In both cases, meteoroid density and velocity were the same. The fluxes, however, were different. The EC-1 flux is essentially the 1963 Whipple flux and has been widely used for shield analysis, being employed until recently for Apollo.

The flux for the revised environment was obtained unofficially from NASA-MSC, and represents a forthcoming revision to the EC-1 flux. A modified version of the revised environment has already been adopted for Apollo. The revised environment is considerably less severe than EC-1 for the smaller mass meteoroids.

Analytic Methods

The analytic methods employed were those developed for Apollo, and are summarized as follows:

1. penetration mechanics for quasi-infinite and single-sheet structures

$$\left.\begin{array}{l} p = 1.38\, d^{1.1}\, P_p^{1/2}\, V_p^{2/3} \big/ H_t^{1/4}\, P_t^{1/6} \\ \bar{t} = 1.8p \end{array}\right\} \text{Metal}$$

$$p = 0.64\, d_p^{1.2}\, P_p^{1/2}\, V_p^{2/3} \qquad \text{Glass}$$

$$p = 2.51\, d_p^{1.3}\, P_n^{1/2}\, V_m^{2/3} \qquad \text{Ablator}$$

2. Penetration mechanics for multisheet structures

$$\bar{t} = t_1 + \sum_{i=1}^{n} \frac{t_i}{K_2}$$

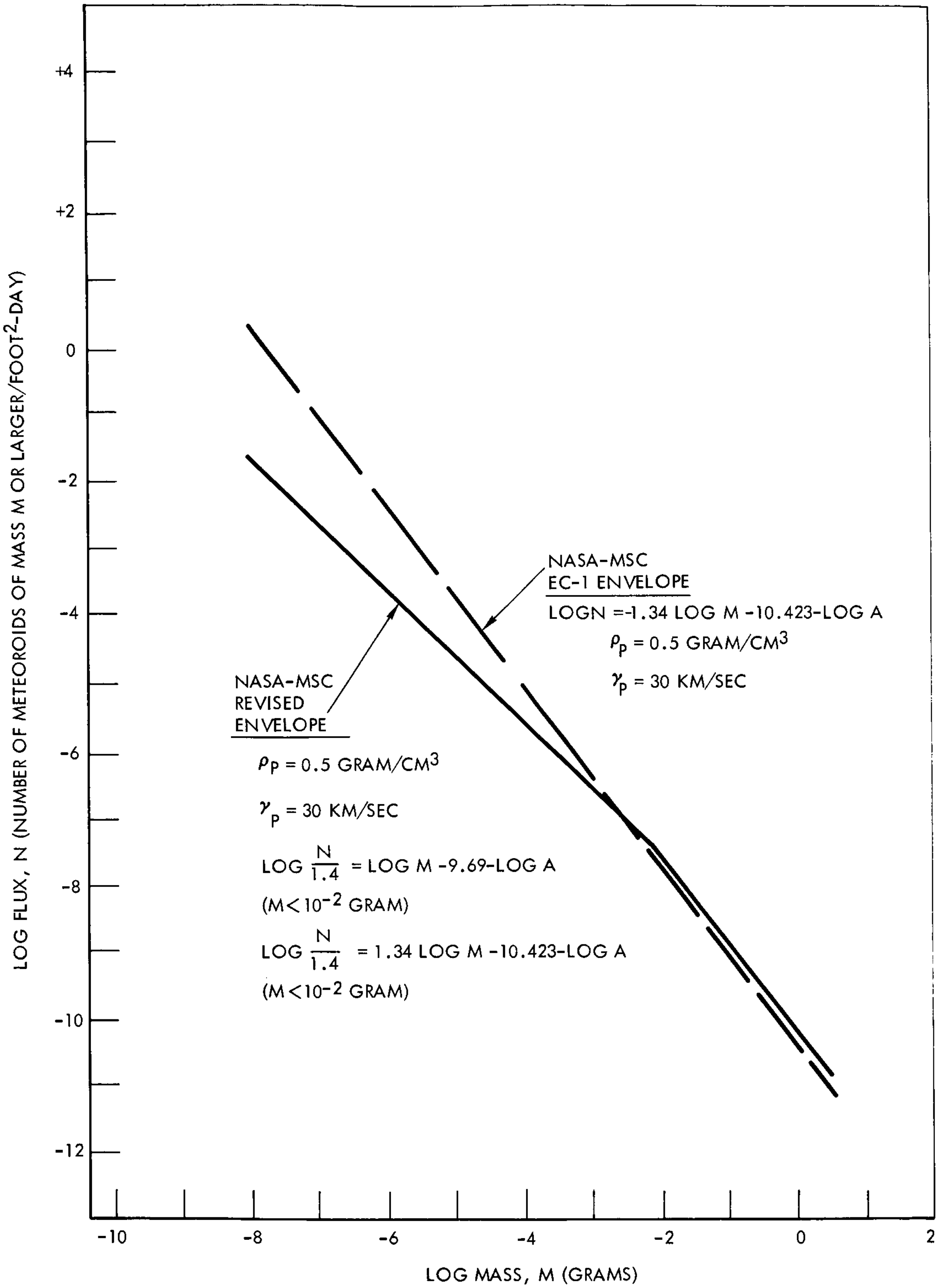


Figure 72. Meteoroid Environments

3. Allocation of shielding by minimum-weight shield method and computer program

$$\frac{\partial W_{st}}{\partial P_{oi}} + \frac{\partial (P_{oT} - P_{01} \cdot P_{02} \cdots P_{on})}{\partial P_{oi}} = 0$$

4. Symbols

p = penetration

σ = density

V = velocity

H = hardness

$\bar{t}$ = minimum thickness to resist perforation

K = efficiency factor

t_i = thickness of sheet (i)

W_{st} = total shield wt

P_{oi} = probable no-failure of component i

P_{oT} = overall probability of no-failure

Penetration mechanics are the most recent developed at NAA-SID, and are based on extensive hypervelocity impact testing. Similarly, the minimum weight shielding method was developed for Apollo by NAA-SID. It is based on LaGrange's variational method and allocates shielding to spacecraft components to minimize total shield weight.

Shielding Requirements and Location

Shielding requirements and location are defined in Figure 73. Except for the LM cabin, these are the same as used for Apollo. In all cases shield locations were selected to give efficient meteoroid protection, yet meet various other constraints. Allowable damage, except for the LM cabin, are supported by test data and/or analysis. Allowable damage for the cabin was selected as being adequatedly conservative. Future studies and development could investigate more favorable criteria which allow limited size perforations and if found acceptable, might reduce shield requirements for this component.

Allocation of Shielding

Shield calculations were made for the dependent RCM laboratory configuration for both environments and several missions. Results for the 30-day synchronous earth orbit mission are summarized in Figure 74. A total of 12 major components were considered, and shielding allocated to

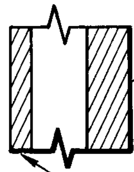
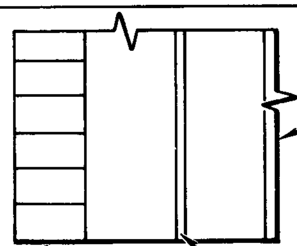
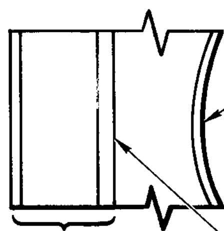
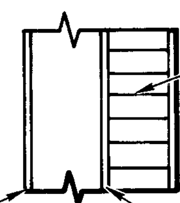
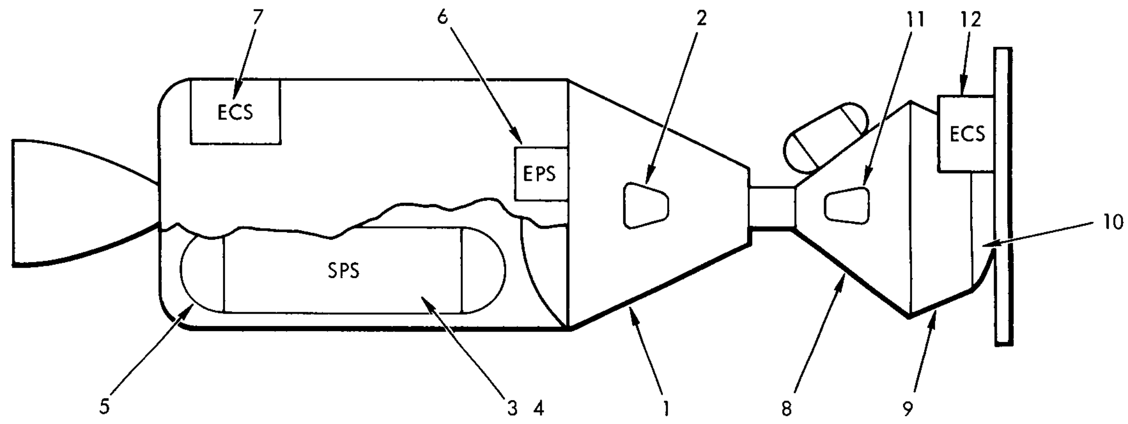
SHIELDING ARRANGEMT	MAJOR COMPONENT	ALLOWABLE DAMAGE	LOCATION OF SHIELD MATERIAL
A	CM AND LABORATORY WINDOWS	LIMIT PENETRATION OF STRUCTURAL WINDOW TO LESS THAN 0.001 INCH	
B	CM HEAT SHIELD	LIMIT PENETRATION INTO ABLATOR TO LESS THAN FULL DEPTH	ABL SHIELD NOT REQUIRED
C	SM SPS TANKS, CYLINDRICAL SECTION	LIMIT PENETRATION INTO TANK WALL TO LESS THAN 1/4 OF WALL	
D	SM SPS TANKS, LOWER DOMES	SM SAME AS ABOVE	
E	SM AND LABORATORY RADIATORS	PREVENT PERFORMATION OF TUBE WALL CONSIDER REDUNDANCY	
F	LABORATORY CABIN	PREVENT ANY PERFORMATION OF THE INNER FACE SHEET	

Figure 73. Meteoroid Shielding Requirements and Locations



CALCULATION FOR: REVISED MSC ENVIRONMENT, 30-DAY SYNCHRONOUS EARTH ORBIT MISSION, PROBABILITY OF NO MISSION ABORT DUE TO METEOROID IMPACT = 0.995 WITH SHIELDING PRESCRIBED, DEPENDENT SPACECRAFT.

COMPONENT	SHIELD PLUS MOUNTING WEIGHT (EST) FOR COMPONENT (W_{SHi}) (POUNDS)	RELIABILITY WITH SHIELDING (PROBABILITY OF NOT EXCEEDING ALLOWABLE DAMAGE)	EFFECT OF SUSTAINING MORE THAN ALLOWABLE DAMAGE *	SHIELDING ARRANGEMENT AND ALLOWABLE DAMAGE
(1) COMMAND MODULE	0.0	0.99908	CL	B
(2) CM WINDOW	12.5	0.9998	CL	A
(3) SPS TANK SIDES	75.0	0.9989	CL	C
(4) SPS TANK SIDES	73.2	0.9991	CL	C
(5) SPS TANK ENDS	5.0	0.9995	CL	D
(6) SM EPS	5.0	0.99999	MA	E
(7) SM ECS	12.8	0.99999	MA	E
(8) RCM	18.2	0.9995	MA	F
(9) RCM	23.0	0.9995	MA	F
(10) RCM	23.3	0.9995	MA	F
BUMPER FOR 8, 9, AND 10	111.4			
(11) RCM WINDOW	12.5	0.9998	MA	A
(12) RCM ECS	12.8	0.99999	MA	E
	387.0			

(*) MA = MISSION ABORT; CL = CREW LOSS.

Figure 74. Typical Minimum-Weight Reliability and Shielding Allocation to SC Components, 30-Day Synchronous Earth-Orbit Mission

each. Being preliminary, the analysis omits perhaps 20 additional components which are less vulnerable. Significant findings are as follows:

1. Some shielding is required on each module.

2. The shield weight is almost equally divided between the SM and RCM laboratory with little shield weight required for the CM.

3. No modification to the CM ablator would be required for the missions studied. However, as pointed out previously, the 45-day synchronous mission pushes the Block II Ablator near its reliability limit.

Overall Shielding Weight

The total shield weights for the dependent spacecraft were computed for several missions and two environments. The results are shown in Figure 75. The lower graph compares shield weights obtained with the two different environments. There is distinctly less shield weight associated with the revised environment. Note that the lines are converging. This is due to the fact that for longer missions, larger meteoroid masses must be designed for. And, for larger masses the two environments are the same (Figure 72).

The shield weight associated with the different missions is compared in the upper graph. The polar earth orbit and lunar orbit missions show lower shield weight required due to the shielding offered by the earth and the moon while in low altitude orbit. The rather large weight associated with short missions is due to approximately 100 pounds for the RCM bumper.

It is estimated that total shield weight for the independent spacecraft would be about 25 percent higher than for the dependent spacecraft due to a net increase in vulnerable area by the added systems.

If it is necessary to reduce shield weight, the following areas might prove profitable:

Evaluate the possibility of designing for limited puncture of the laboratory cabin and in-flight repair.

Evaluate the possibility of designing for puncture of the CM heat-shield and in-flight repair.

Shield Thickness and Weights - Programmed Components

Table 39 shows the required shielding goals for the programmed components, and Table 40 summarizes the calculated shield values for a

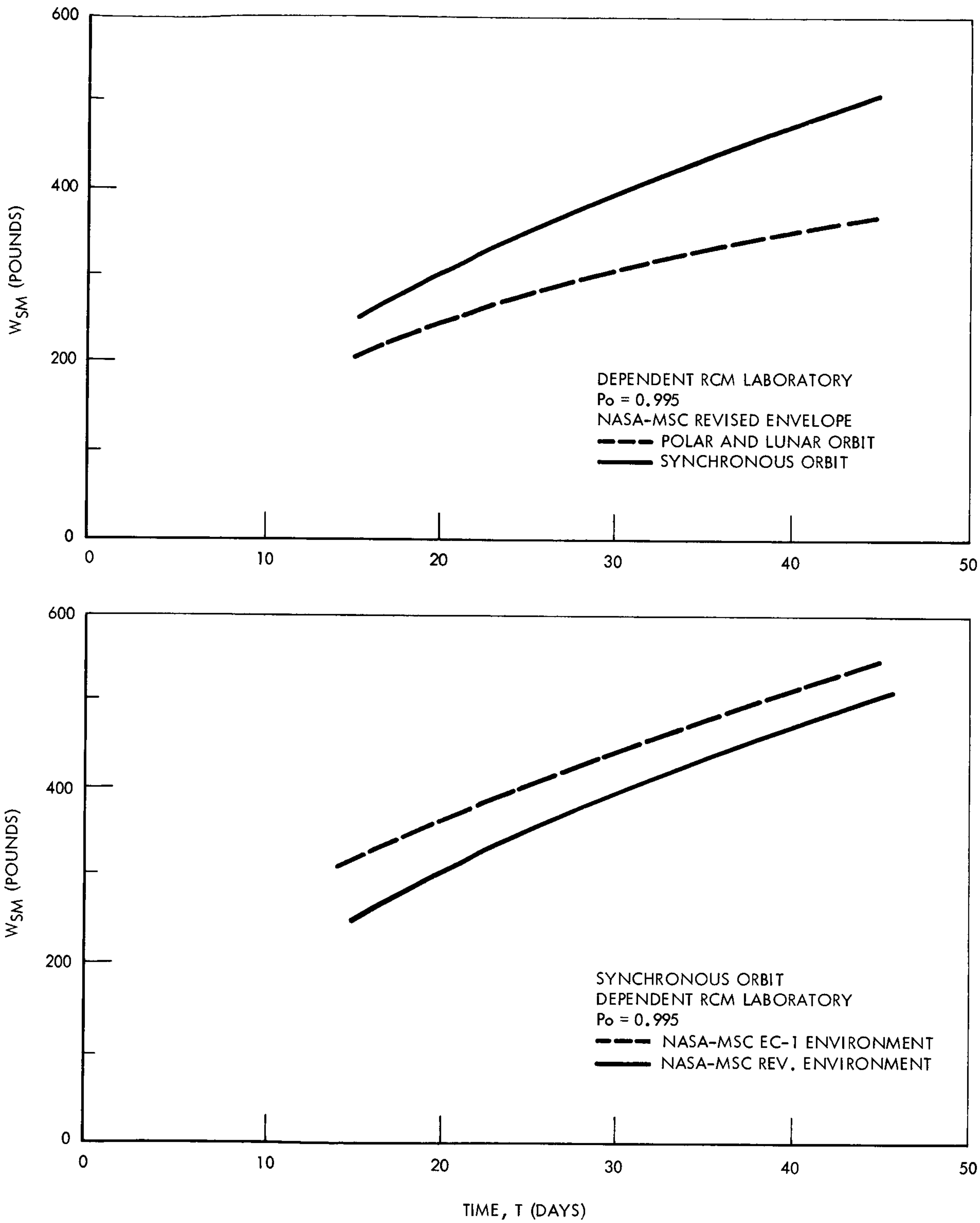


Figure 75. Total Spacecraft Shield Weight Plus Mounting Weight Versus Mission Duration

Table 39. Required Goal for Components in Computer Program

	45-day Synchronous		30-day Polar Orbit	
	EC-1 Flux	Revised Apollo Flux	EC-1 Flux	Revised Apollo Flux
CM ablator	0.99851	0.99862	0.999504	0.999540
CM windows	(0.9998)	(0.9998)	(0.9998)	(0.9998)
ECS radiator	(0.99999)	(0.99999)	(0.99999)	(0.99999)
EPS radiator	(0.99999)	(0.99999)	(0.99999)	(0.99999)
Laboratory windows	(0.9998)	(0.9998)	(0.9998)	(0.9998)
Laboratory radiator panel	(0.99999)[1]	(0.99999)[1]	(0.99999)[1]	(0.99999)*
CM suit loop supply	(0.9999)	(0.9999)	(0.9999)	(0.9999)
Overall goal	0.995	0.995	0.995	0.995
Program goal	0.99701	0.99697	0.99602	0.99598

*Preset value

Table 40. Summary of Shield Weights and Thicknesses for Components in Shield Weight Program, T = 30 Days, P_o = Variable

Computer Entry No.	Flux	T/1 - P_o = 3,000			T/1 - P_o = 7,500			T/1 - P_o = 30,000		
		t_s	W_s	W_{sm}	t_s	W_s	W_{sm}	t_s	W_s	W_{sm}
1. SM SPS tanks	EC-1	0.012	29.4	44.1	0.023	56.6	85.0	0.045	111.2	167.0
2. SM SPS tanks		0.013	30.8	46.3	0.024	55.0	82.5	0.046	103.4	155.0
3. SM SPS tanks		0	0	0	0.012	8.5	8.5	0.050	34.8	34.8
4. Laboratory		0.007	8.0	8.0	0.018	21.5	21.5	0.040	48.5	48.5
5. Laboratory		0.012	13.6	13.6	0.023	26.1	26.1	0.045	51.2	51.2
6. Laboratory	EC-1	0.012	13.7	13.7	0.023	26.4	26.4	0.045	51.8	51.8
TOTALS			96.2	125.7		194.8	250.0		401.6	508.3
1. SM SPS tanks	Revised Apollo Flux	0.006	16.1	24.1	0.020	49.9	75.0	0.050	123.9	186.0
2. SM SPS tanks		0.008	18.9	28.4	0.022	48.8	73.2	0.051	114.2	171.0
3. SM SPS tanks		0	0	0	0.007	5.0	5.0	0.057	40.5	40.5
4. Laboratory		0.001	1.5	1.5	0.015	18.2	18.2	0.045	54.8	54.8
5. Laboratory		0.006	7.5	7.5	0.020	25.0	23.0	0.050	57.0	57.0
6. Laboraotyr	Revised Apollo Flux	0.006	7.5	7.5	0.020	23.3	23.0	0.050	57.7	57.7
TOTALS			52.2	67.0		168.9	217.7		448.8	567.0

30-day mission in an undisturbed flux for a range of overall shielding goals. The totals are plotted in Figure 76 in a manner which allows interpolation of results to other $T/1-P_o$ values associated with the various missions.

The $T/1-P_o$ values to shield for are as follows:

1. 15-, 30-, 45-day polar earth orbit

   Effective time in flux is one half the elapsed time due to earth shielding.

   $T/1-P_o = 15/1 - 0.99576^* = 1770;\ 3550;\ 5620$

   *Obtain from Table 39.

2. 15-, 30-, 45-day synchronous earth orbit

   $T/1-P_o = 15/1 - 0.99576 = 3550;\ 7,500;\ 15,000$

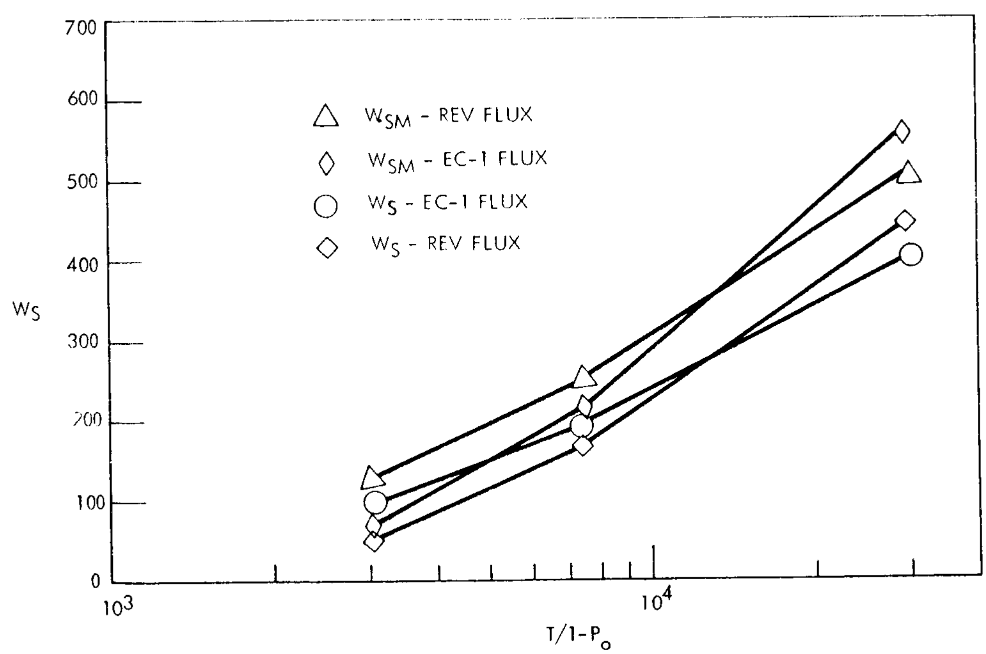


Figure 76. Total Shield Weight of Components Considered in Shield Weight Program Versus Vulnerability Parameters

3. 15-, 30-, 45-day lunar orbit

Effective time in undisturbed flux is approximately one half the elapsed time in space because of lunar shielding; therefore, use same T/1-Po values as polar earth oribt mission.

The appropriate total shield weight values were obtained from Figure 76 and listed in Table 41 for the synchronous earth orbit missions. The shielding for the polar earth orbital and lunar orbital missions can be obtained similarly and would result in approximately one half the shield weight. A similar interpolation can be applied to determine the individual shield thicknesses when required.

Discussion

The micrometeoroid shielding analysis described utilizes the minimum weight shielding program used on Apollo shielding analysis (SID 65-1135). All modules of the RCM laboratory configuration were treated, as each requires substantial shield additions. The preliminary analysis omits some 20 components less vulnerable but which would be treated in a detailed analysis.

Locations selected for shield material are defined in Figure 73 . These are the same shield locations found appropriate in Apollo. The amount of shielding to be added is summarized in Table 41, for the synchronous orbit missions. As indicated shield weight plus estimated mounting weights are between 200 and 600 pounds. Figure 77 shows the shield configuration, and shield weight sensitivity to mission duration.

Shielding calculations were not completed for the polar orbit and lunar missions. Shielding requirements would be about the same for these two missions for the same mission duration. It is estimated that the shielding weight would be about half that required for the synchronous orbit missions due to the shielding of the earth on the moon. (See Figure 77.)

RADIATION SAFETY ANALYSIS

In the four Reference Mission Profiles of this study, the RCM laboratory and its contents will be subjected to a high-energy space radiation environment. The environment has been divided into the categories of trapped particles and solar particles (disregarding galactic particles as relatively unimportant for present purposes).

The resulting crew radiation safety is summarized in Figure 78 for the dependent RCM laboratory and the Apollo CM on the respective Reference

Table 41. Shield Thicknesses and Weights, Synchronous Earth Orbit

No.	Component	Flux	T = 15 Days			T = 30			T = 45		
			t_s in.	W_s lb	W_{sm} lb	t_s	W_s	W_{sm}	t_s	W_s	W_{sm}
1	HS ablator	EC-1	0	0	0	0	0	0	0	0	0
2	HS windows		0.29	9.0	18.0	0.36	11.0	22.0	0.59	18.5	37.0
3-5 and 8-10	Shielding for SPS Tanks and Laboratory wall		-	110.0	145.0	-	200.0	250.0	-	300.0	380.0
	Laboratory bumper		0.016	55.0	111.4	0.016	55.0	111.4	0.016	55.0	111.4
6	CM EPS radiator*		0.025	3.4	3.4	0.037	5.0	5.0	0.044	5.9	5.9
7	CM ECS radiator*		0.043	9.5	9.5	0.058	12.8	12.8	0.069	15.1	15.1
11	Lab windows		0.29	9.0	18.0	0.36	11.0	22.0	0.59	18.5	37.0
12	Lab ECS radiator*	EC-1	0.043	9.5	9.5	0.058	12.8	12.8	0.069	15.1	15.1
	TOTAL			205.4	314.8		307.6	436.0		428.1	601.5
1	HS ablator	Revised Apollo Flux	0	0	0	0	0	0	0	0	0
2	HS windows		0.20	12.5		0.20	12.5		0.20	12.5	
3-5 and 8-10	Shielding for SPS Tanks and Laboratory wall		--	65.0	90.0	--	170.0	220.0	--	310.0	390.0
	Laboratory bumper		0.016	55.0	111.4	0.016	55.0	111.4	0.016	55.0	111.4
6	CM EPS radiator*		0.025	3.4	3.4	0.037	5.0	5.0	0.044	5.9	5.9
7	CM ECS radiator*		0.043	9.5	9.5	0.058	12.8	12.3	0.069	15.1	15.1
11	Lab windows		0.20	12.5	25.0	0.20	12.5	25.0	0.20	18.5	25.0
12	Lab ECS radiator*	Revised Apollo Flux	0.043	9.5	9.5	0.058	12.8	12.8	0.059	15.1	15.1
	TOTAL			167.4	248.8		280.6	387.0		426.4	562.5

*Add 0.030-inch for total thickness

T = Mission length, days.
t_s = Shield thickness, in.
W_s = Shield weight, lb
W_{sm} = Shield and mounting weight (estimated), lb

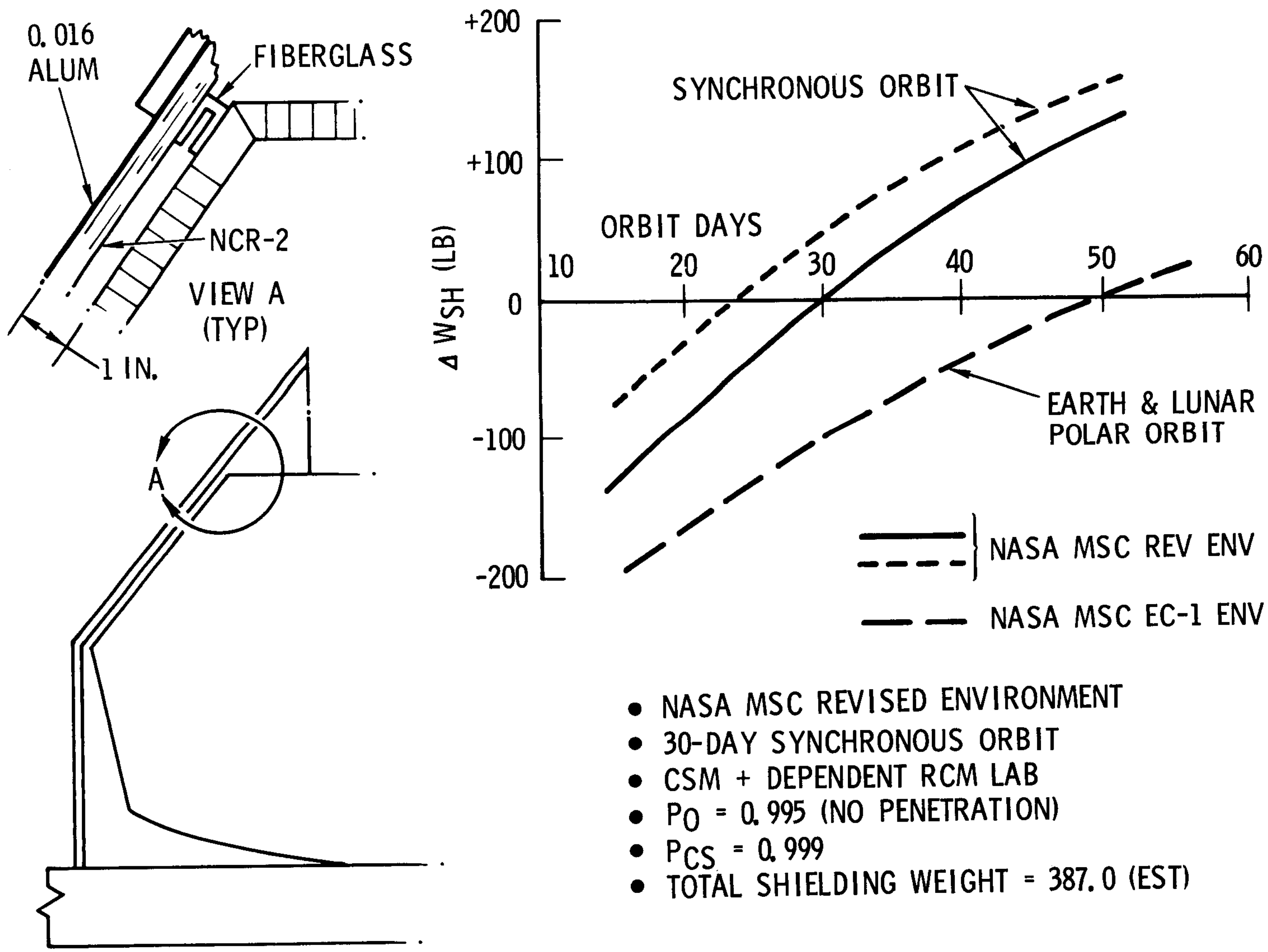
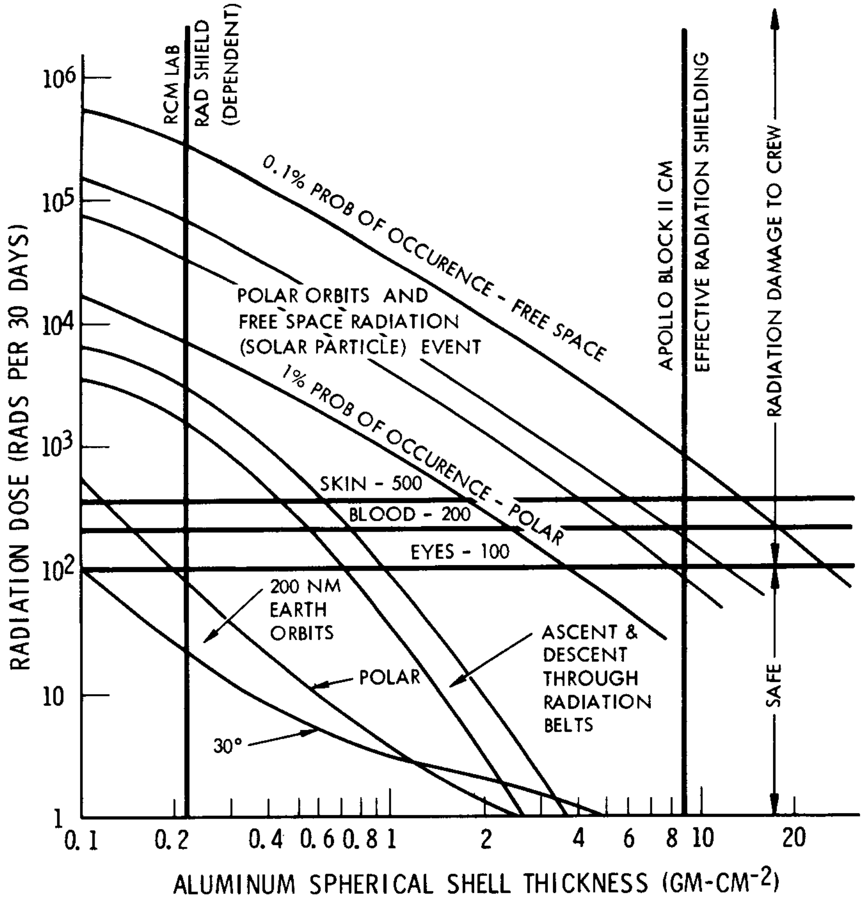


Figure 77. RCM Laboratory Micrometeoroid Effectiveness



CREW RADIATION SAFETY

- RCM LABORATORY (DEPENDENT)
- RADIATION ENVIRONMENTS:

  LOW ALTITUDE EARTH ORBITS - TRAPPED PROTONS, ELECTRONS & BREMSSTRAHLUNG

  SYNCHRONOUS & LUNAR ORBITS - EXPOSURE TO HIGH ENERGY PARTICLES INDUCED BY SOLAR EVENTS

RCM LABORATORY RADIATION SAFETY

- LABORATORY IS SAFE FOR LOW ALTITUDE EARTH ORBITS
- LABORATORY IS SAFE IN DEEP SPACE - CREW RETURN TO COMMAND MODULE FOR SOLAR EVENT RADIATION PROTECTION
- CREW IN COMMAND MODULE DURING TRANSIT THROUGH RADIATION BELTS

Figure 78. RCM Laboratory Radiation Shielding, Thirty-Day Mission

Mission Profiles. The total radiation doses per 30-day mission are conservatively based on a continuous crew occupancy of the laboratory for the full 30-day mission duration. Because any single crew member is not continuously occupying the laboratory, he will receive proportionally smaller total radiation doses.

Trapped Protons and Electrons

The missions involving circular earth orbits, including a 200-nautical mile altitude with 30-degree and 39-degree inclinations and a 19,340-nautical mile synchronous altitude with 0-degree (equatorial) inclination, are all appreciably removed in space from the inner proton belt peak intensity region, which has approximately 2×10^4 protons-cm^{-2}-s^{-1} above 40 Mev at 1700 nautical miles above the geomagnetic equator. These earth orbits are also appreciably distant from the electron belt peak intensity region, which has approximately 10^3 electrons-cm^{-2}-s^{-1} above 40 kev at 5200 nautical miles above the geomagnetic equator.

The bases for the trapped particle doses presented here are the daily energy-integrated orbital fluxes interpolated from the most recent available data collected and processed by J.I. Vette, Aerospace Corporation (References 1 and 2). These data are also the current bases for Apollo and AAP trapped radiation calculations. The increase in proton flux with altitude in the neighborhood of the 200-nautical mile mission baseline altitude is shown in Figure 79. A comparison of the integral fluxes versus energy indicates that the spectrum for the polar orbit is significantly less penetrating than that for the low inclination orbit and therefore would be expected to produce smaller doses.

Daily trapped proton doses to the eye of an astronaut are shown in Figure 80 as a function of circular orbital altitude for a low inclination (30 degree) earth orbit. The dose values were cross-plotted from those computed by Hill et al (Reference 3) from the Vette proton map designated AP3 (Reference 1), which gives a trapped proton spectrum above 5 Mev with a recommended most reliable region above 60 Mev.

Figure 81 shows the variation with shield thickness of the trapped proton dose to the eye and to the abdomen (at an effective depth corresponding to that for blood forming organs of the body) for both 30-degree and 90-degree (polar) orbital inclinations. The machine-computed dose values of Hill et al (Reference 3) were available between 1 and 30 g-cm^{-2}. The doses for shield thicknesses below 1 g-cm^{-2} were estimated by extrapolating a power law in shield thickness from the dose values at 1 and 5 g-cm^{-2}.

The electron doses shown in Figure 82 are proportional to daily orbital electron fluxes projected by Vette (Reference 2) to December 1968. The

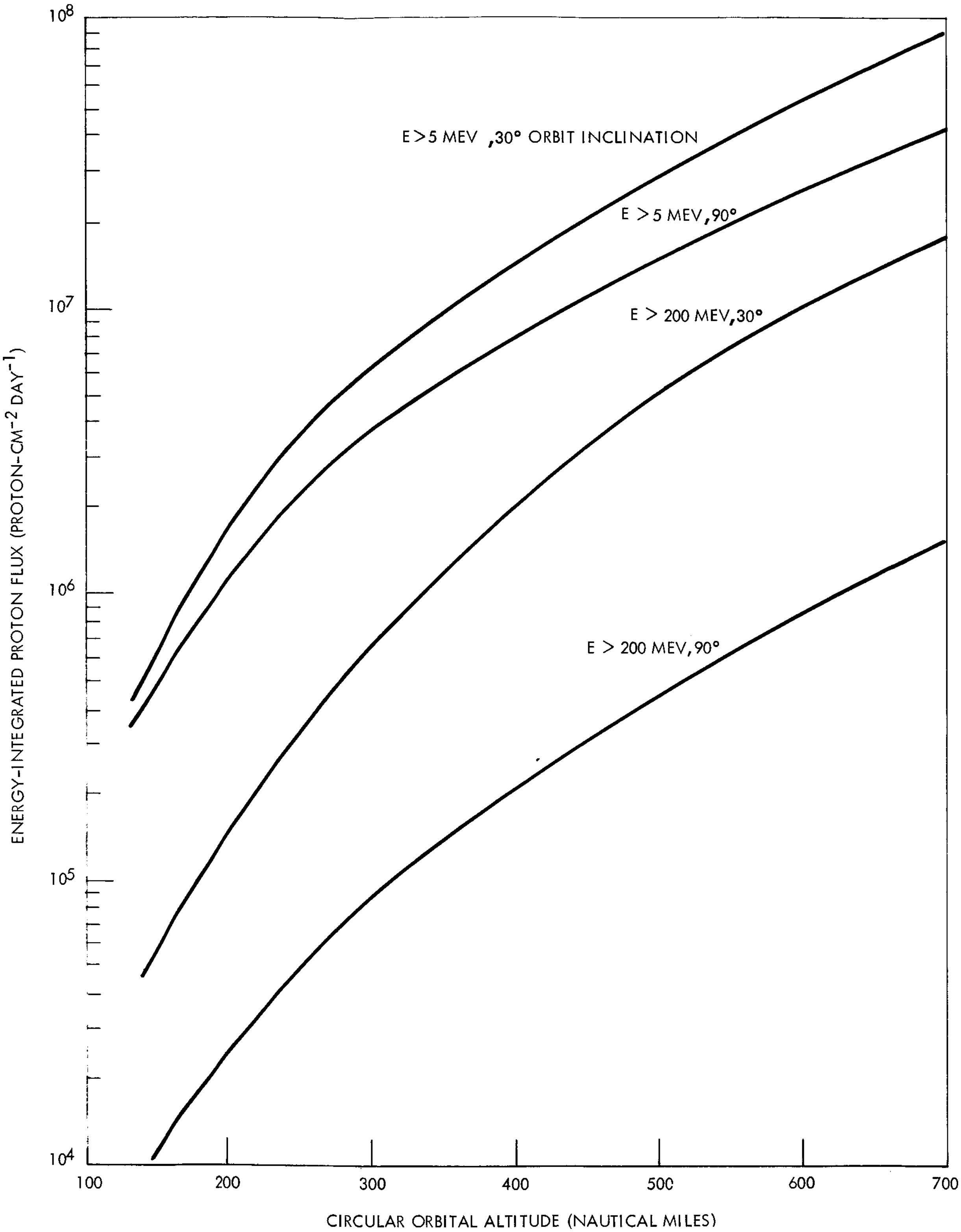


Figure 79. Trapped-Proton Flux Versus Altitude

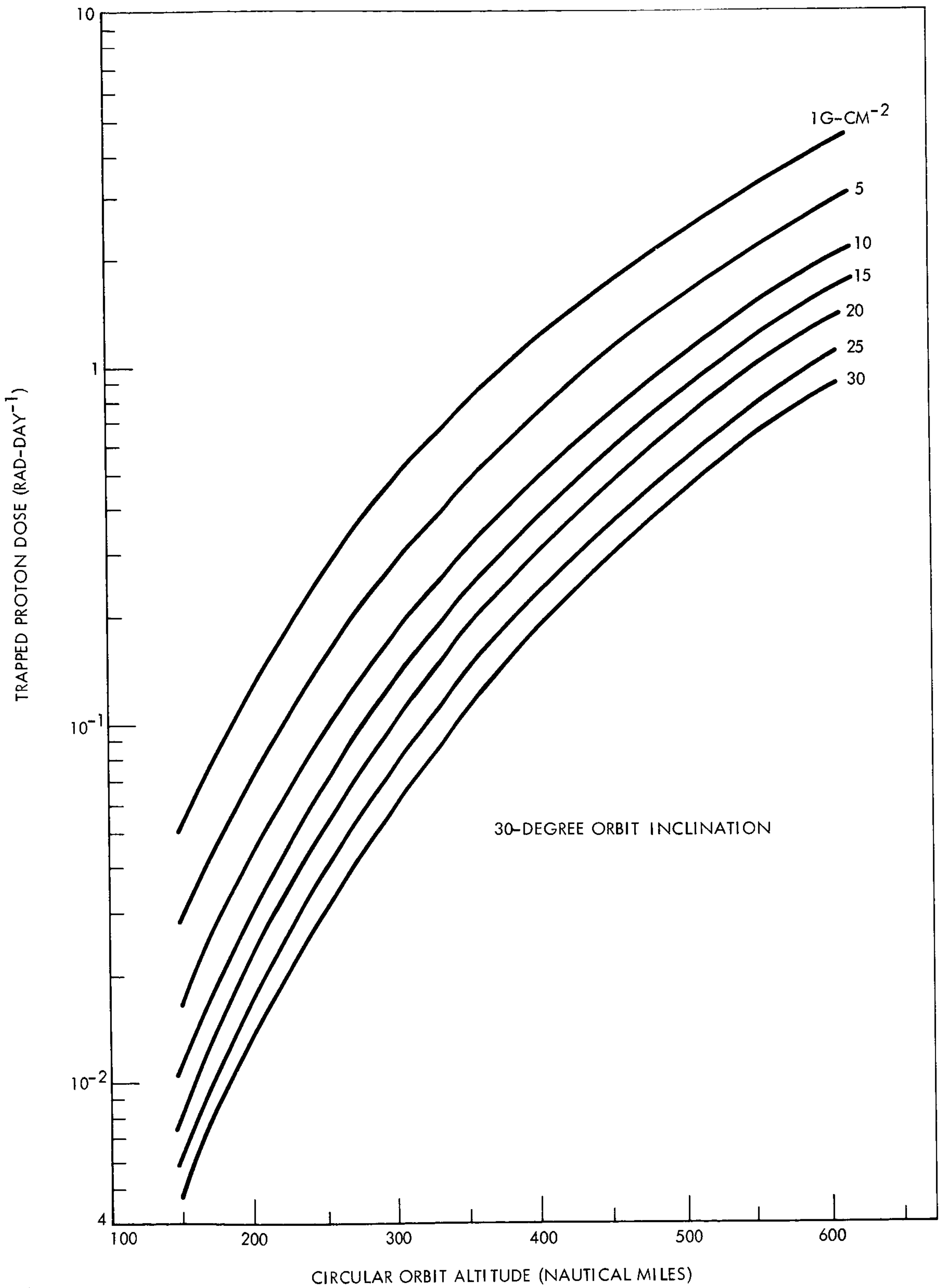


Figure 80. Daily Trapped-Proton Dose

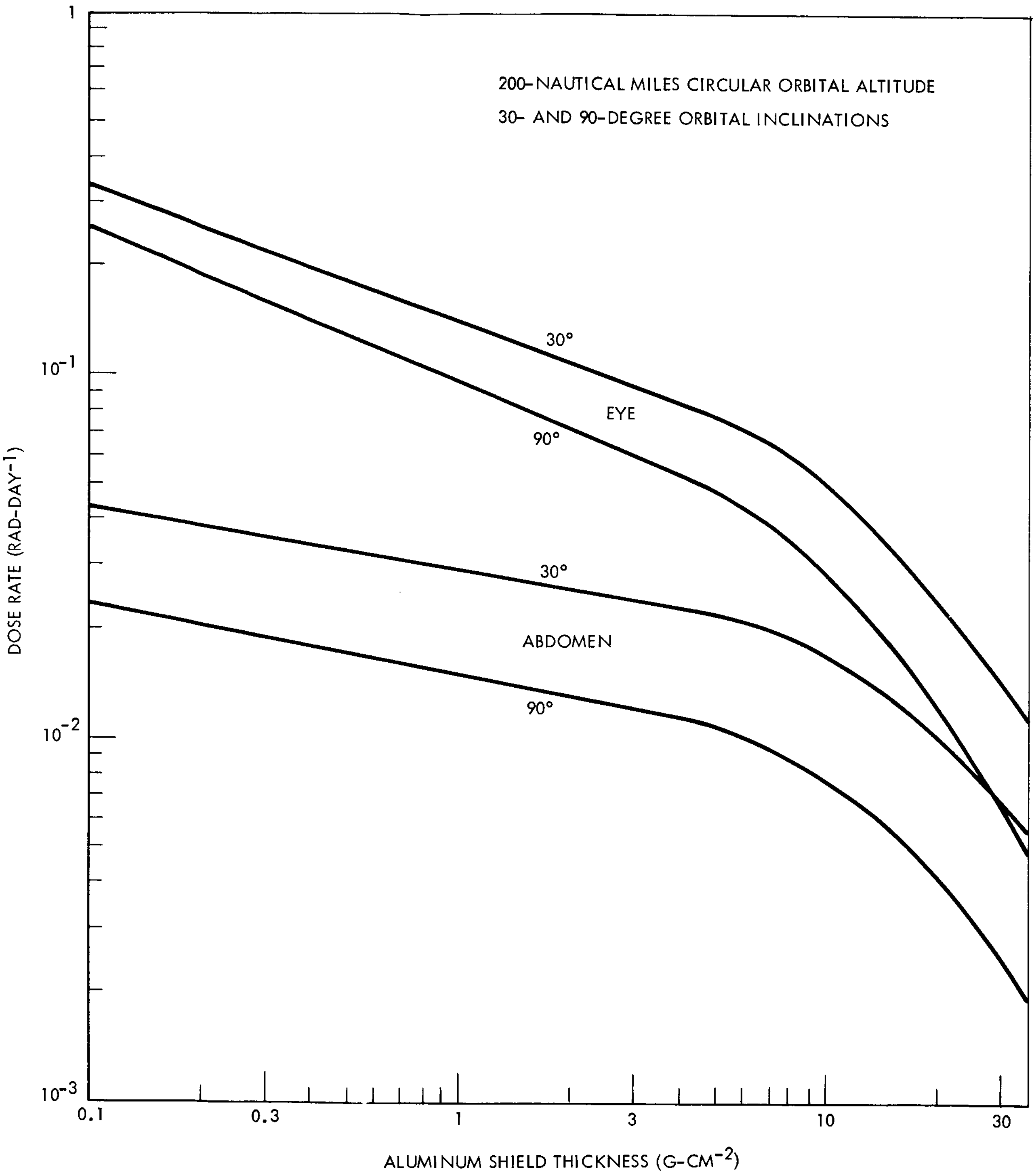


Figure 81. Trapped-Proton Dose Rates

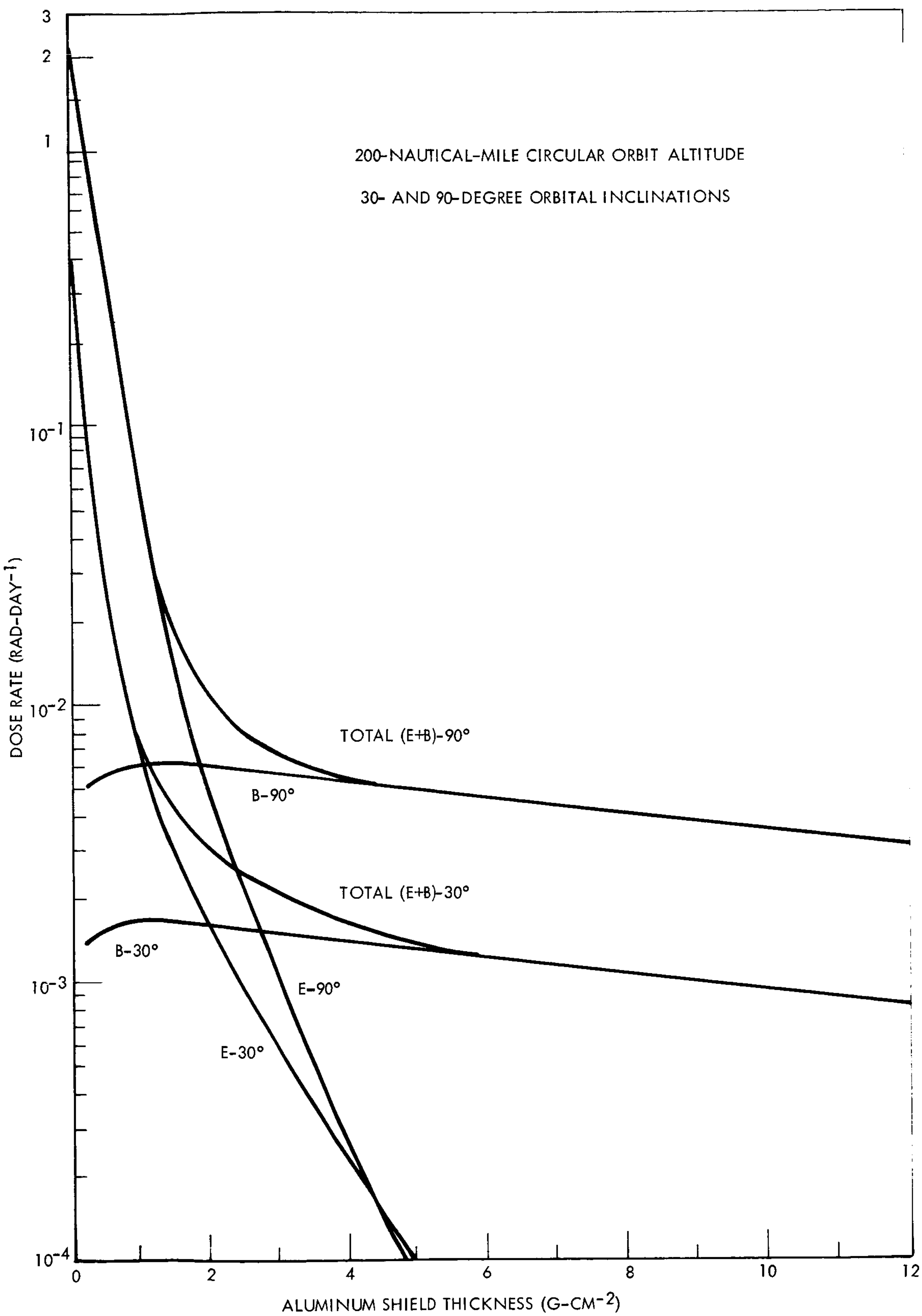


Figure 82. Trapped-Electron (E) and Premsstrahlung (B) Dose Rates at Eye

projection includes the anticipated additional decay of the Starfish nuclear burst artificial electron belt and the build-up of the natural electron belt with increasing solar activity to that date. For shields thicker than 2 and 3 g-cm^{-2} the bremsstrahlung (or secondary X-ray) dose becomes comparable and exceeds the electron dose and persists to considerably greater shield thicknesses than do electrons, but still produces doses appreciably less than those produced by trapped protons.

Due to the anticipated thinner walls of the RCM laboratory as compared to other spacecraft, the evaluation of the electron dose was given further consideration. Trapped electron doses as evaluated previously (References 4 and 5) are observed to decrease approximately exponentially with shield thickness up to thicknesses at which bremsstrahlung doses rise to comparable values. Therefore, the machine-calculated electron doses from Hill et al (Reference 3) were curve fitted at 1 and 4 g-cm^{-2} (the only values for which doses were given) and extrapolated to other shield thicknesses. An extrapolation of zero shield thickness gave unreasonably low surface doses. Therefore, a second curve fit for the electron dose was used to yield the expected surface doses and match the values from the first plot at 0.1 g-cm^{-2}.

The electron surface doses were estimated from

$$D = 1.6 \times 10^{-8} \left(\frac{dE}{dX}\right) \sigma_e(E > 0)$$

$\left(\frac{dE}{dX}\right)$ = average 1.85 Mev-cm^2-g^{-1} energy loss of electrons (Reference 6), and

$\sigma_e(E > 0)$ = energy-integrated electron flux

= 5.4×10^9 electrons-cm^{-2}-day^{-1} above zero energy at 200 nautical miles for 30-degree orbital inclination and 3.2×10^{10} for 90-degree inclination.

Unshielded electron doses calculated as above are 4.8×10^3 rad-day^{-1} for 30-degree orbital inclination and 2.88×10^4 rad-day^{-1} for 90-degree inclination at 200 nautical mile orbital altitude.

The results of the trapped proton and electron doses versus shield thickness derived in the above manner are shown with totals for 30-day missions in Figure 83. Since over 40 percent of the surface area of the RCM laboratory wall structure has an effective solid aluminum thickness of only 32 mils (0.22 g-cm^{-2}), electron doses of 10 to 100 rads may be

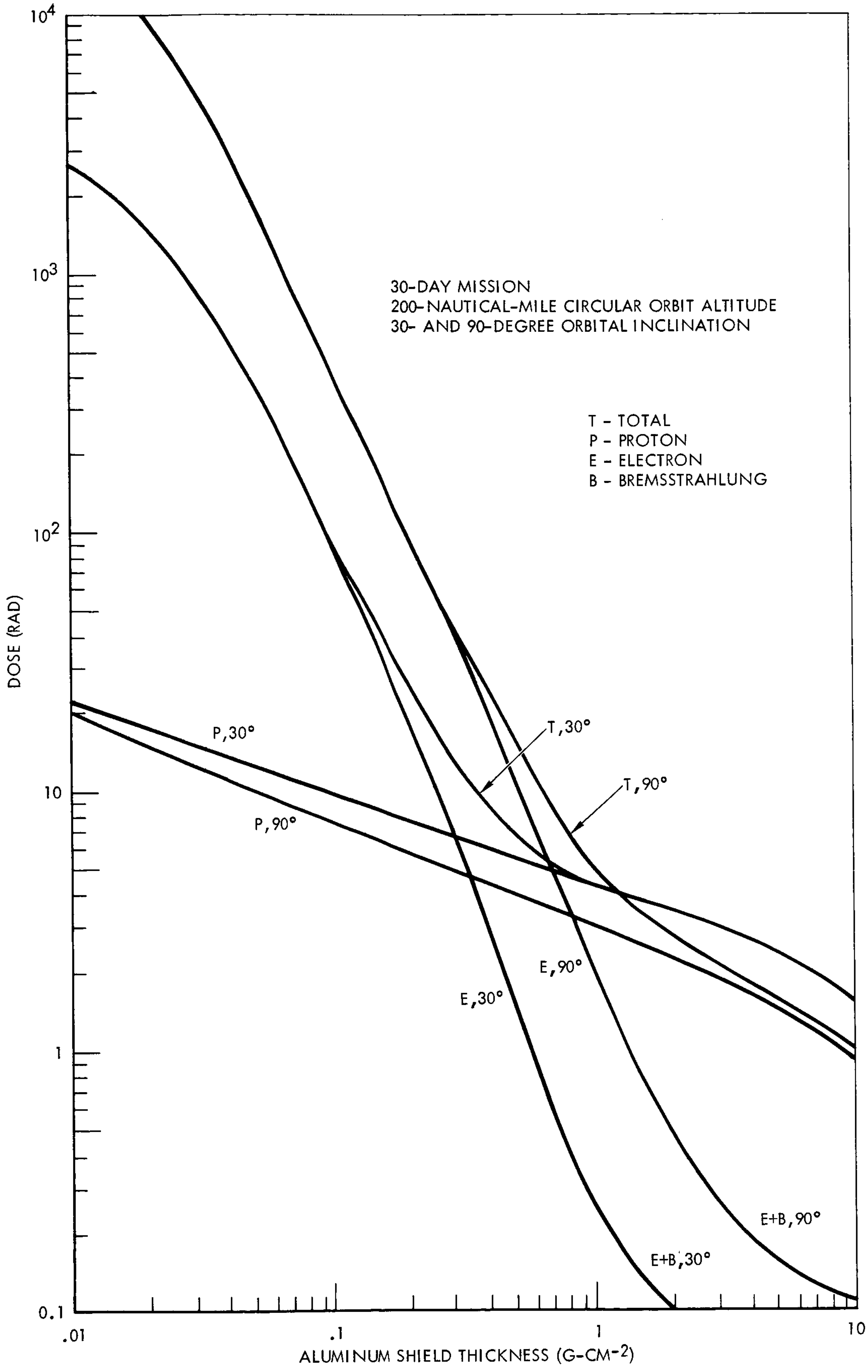


Figure 83. Total Trapped Radiation and Component Doses

encountered. To verify these results, a more detailed electron dose calculation was performed using a computer code which included the effects of electron straggling. Although the previously extrapolated results were based upon digital computer programs which included the effect of straggling, the available published doses were not available for such thin shields but only for those of 1 g-cm^{-2} or greater. The more detailed calculation was based on piecewise-continuous fits to Vette's electron energy-integrated fluxes (Reference 2), interpolated for a 200 nautical mile altitude from 150- and 300-nautical mile data. The resulting piecewise-discontinuous differential energy spectra were of the form

$$\frac{d\delta}{dE} = A_o e^{-E/E_o} \text{ electrons-cm}^{-2}\text{-Mev}^{-1}\text{-day}^{-1}$$

with the constants given as follows:

Orbital inclination (deg)	Energy interval (Mev)	A_o	E_o (Mev)
30	0 to 1	3.86×10^{10}	0.140
30	1 to 2	4.23×10^{7}	0.572
30	2 to	2.98×10^{5}	4.03
90	0 to 0.75	1.30×10^{11}	0.246
90	0.75 to 4	1.40×10^{10}	0.492
90	4 to 5.5	2.84×10^{7}	1.354
90	5.5 to 7	6.91×10^{5}	3.905

The results of the dose calculations based on the above spectra oscillate about the smoothed electron dose curves in Figure 83 and agree well within an order of magnitude. A more exact evaluation of the dose would depend on the ultimate interior contents of the RCM laboratory, the positions of the dose points, and the exact geometric disposition of the wall structures of differing thicknesses. There is a high probability that the ultimate effective shielding thickness may be more than 0.5 to 1 g-cm^{-2} after all interior components are established. The result would be a transition from electrons to trapped protons as the dominate component with acceptable doses from 1 to 10 rads total for a 30-day mission, excluding the occurrence of solar particle events, which will be considered later.

Other baseline missions for which the trapped radiation doses have been evaluated are the 14-day lunar mission and the synchronous orbital mission. The significant contributions from the trapped radiation to both of these missions is only during ascent and descent. The doses are given in Figure 84. Occupancy of the bare RCM laboratory with only 0.2 to 0.5 g-cm^{-2} equivalent aluminum walls is seen to be undesirable during ascent and decent without additional shielding. However, the RCM spacecraft would provide sufficient protection with 2 to 8 g-cm^{-2} of shielding.

To obtain an approximate value of the electron dose accumulated during a 30-day synchronous orbit, the integral flux data of Vette (Reference 2) were extrapolated from tabulated values at 17,000 and 18,000 nautical miles to the synchronous altitude of 19,340 nautical miles to obtain 1.65×10^8 electrons-cm^{-2}-day^{-1} with energies above 0 and a resultant unshielded surface dose of 146 rad for a 30-day equatorial mission. Crude extrapolations by an exponential in the shield thickness gives electron doses of about 1 to 2 orders-of-magnitude less than the doses at corresponding shield thicknesses for the 200-nautical-mile orbits. Therefore, the trapped electron dose acquired during synchronous orbit of 30 days duration appears to be negligible.

Solar Particle Events

The more or less steady-state solar proton output of low-energy protons (i.e., solar wind) from 0.2 to 10 kev with very large number fluxes of 10^{15} to 10^{16} protons-cm^{-2} is not normally expected to penetrate the geomagnetosphere and would only be of consequence to optical and thermal control surfaces, not RCM laboratory interiors on lunar missions.

The geomagnetic field also screens out large numbers of energetic (1 to 100 Mev) particles associated with solar particle events on a disturbed sun. These particles can only descend to the lower (i.e., 200 nautical mile) orbital operational altitudes within polar cones above the auroral latitudes of approximately 60 degrees. For an unperturbed field, there is a characteristic cut-off energy associated with each latitude and altitude. For single events sufficiently spaced such in time that the fast particles from a given event do not encounter the perturbed field produced by the solar plasma from a previous event, the accumulated solar particle mission dose may be estimated from Figure 85. The model solar event and probabilities of encounter are the same as those used in recent AES studies (Reference 7).

The free space dose levels in Figure are applicable to 30-day lunar and synchronous orbit missions. The free space doses have been reduced by a factor of 4, which is approximately the fraction (i.e., 0.25) of the free space integral flux above 30 Mev penetrating to a 200-nautical mile polar orbit with geomagnetic cut-off latitude of 68 degrees. The solar particle

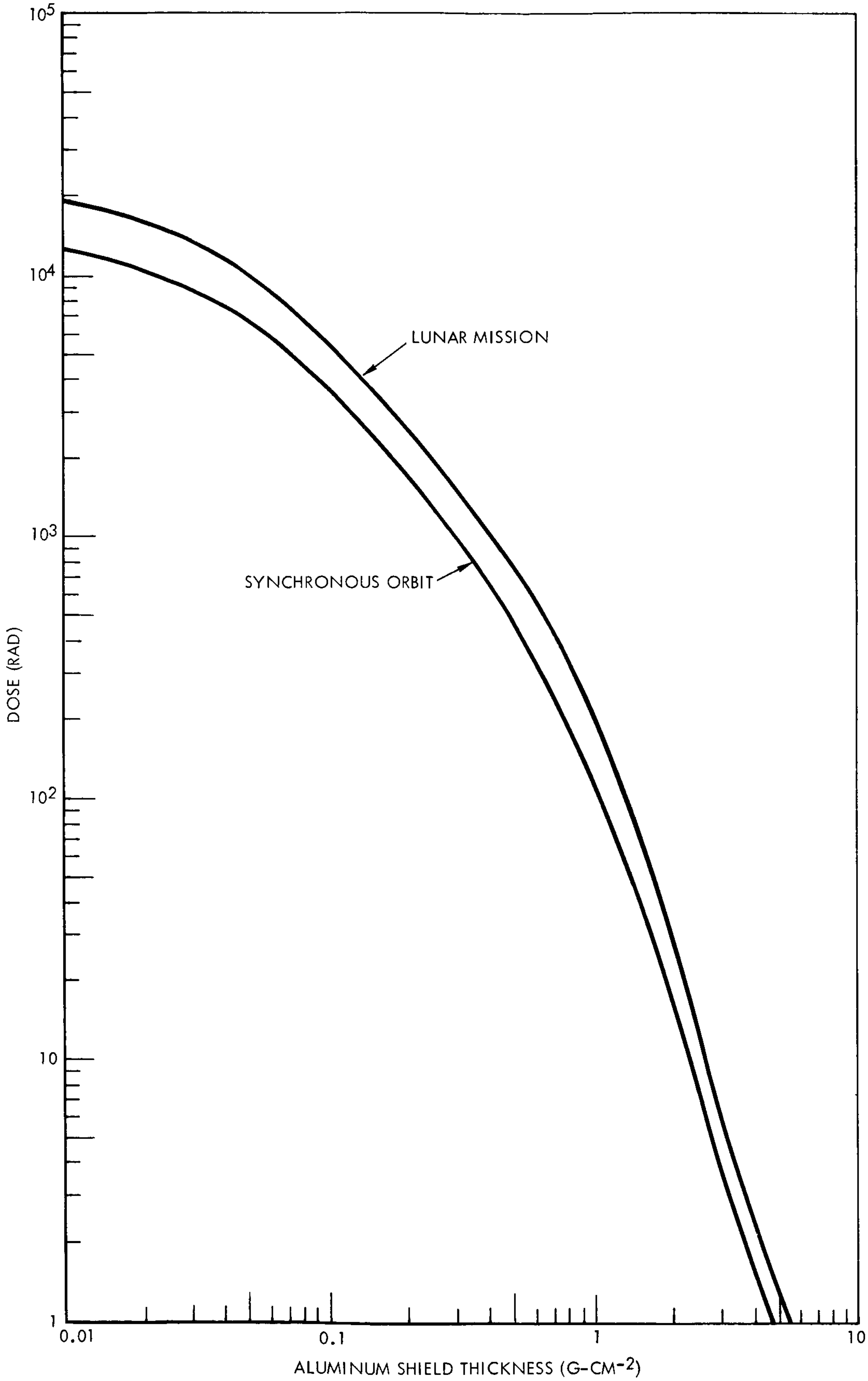


Figure 84. Total Trapped Radiation From Electrons and Protons During Ascent and Descent

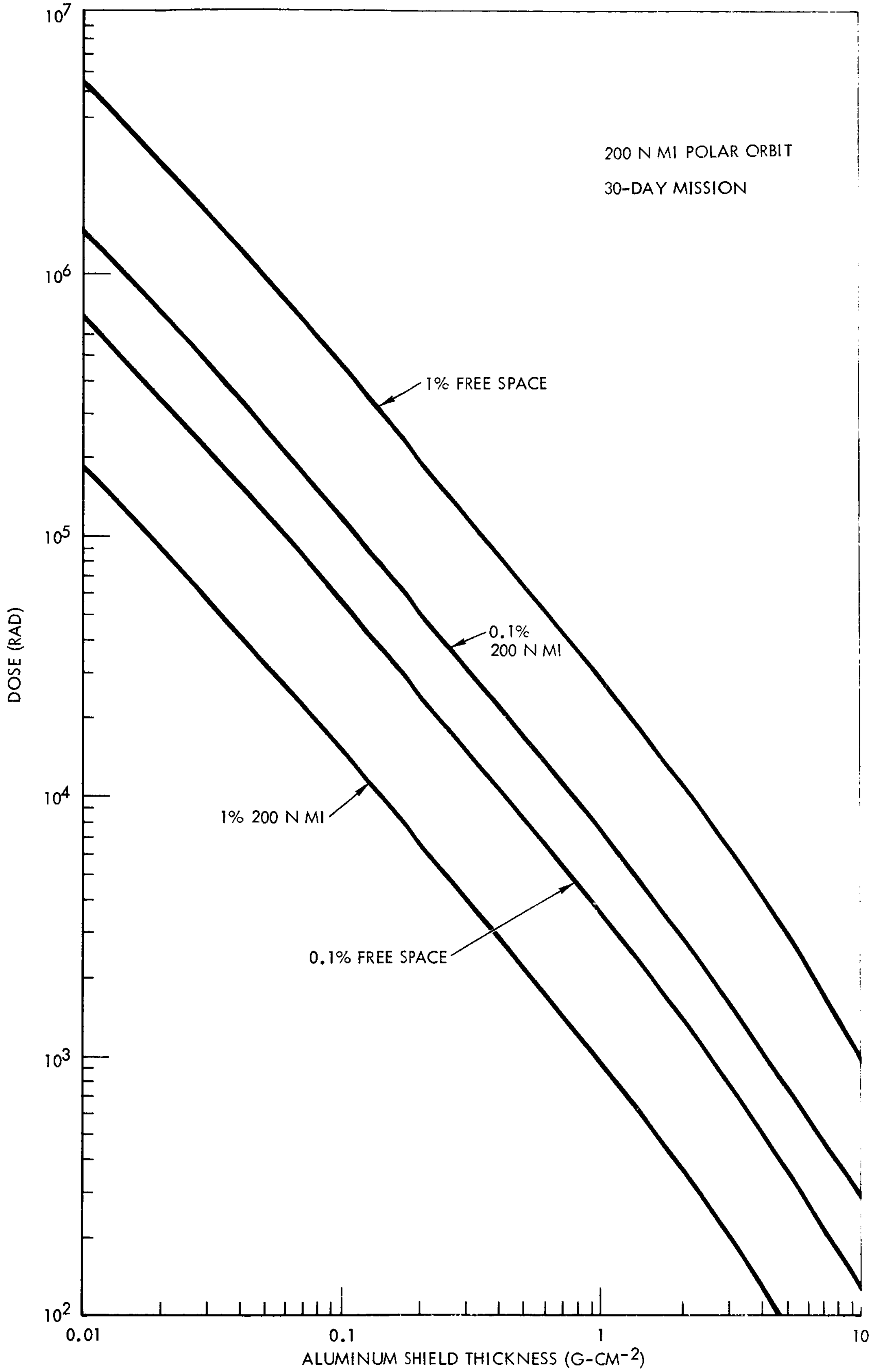


Figure 85. Solar Particle Dose (Protons and Alphas) in Free Space in 200-Nautical-Mile Polar Orbit Versus Probability of Encounter and Shield Thickness

event contribution to the low inclination (30 degrees) is expected to be negligible since the cutoff energy for vertically incident protons is above 6 Bev at 200 nautical miles.

From Figure 85 the 1 percent and 0.1 percent probable solar particle events for all baseline missions—except the low inclination earth orbital mission—are seen to give excessive doses behind 32- to 120-mil walls (0.22 to 0.82 g-cm^{-2}), which values bracket the approximate range of anticipated inner CM structural wall solid aluminum equivalent thicknesses. Allowable doses are 500 rad for skin of whole body, 200 rad for blood-forming organs and 100 rad for the eyes (Reference 8). In the event of a large solar event, the laboratory crew would usually find adequate protection by entering the RCM spacecraft, which with its several, 2 to 8 g-cm^{-2}, would provide adequate protection.

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VII. SYSTEM RELIABILITY AND CREW SAFETY

RCM SPACECRAFT

The renovated Block II command module (RCM spacecraft) was analyzed to determine the probabilities of mission continuation and mission accomplishment for the 14-day, low-inclination, low-altitude, earth-orbit baseline mission. Methodology and basic data contained in SID 66-872, Effects on Crew Safety, prepared in support of the Apollo Applications Program, were utilized. These data were supplemented, where necessary, to meet the specific requirements of the RCM missions.

Mission continuation and mission accomplishment are defined as follows:

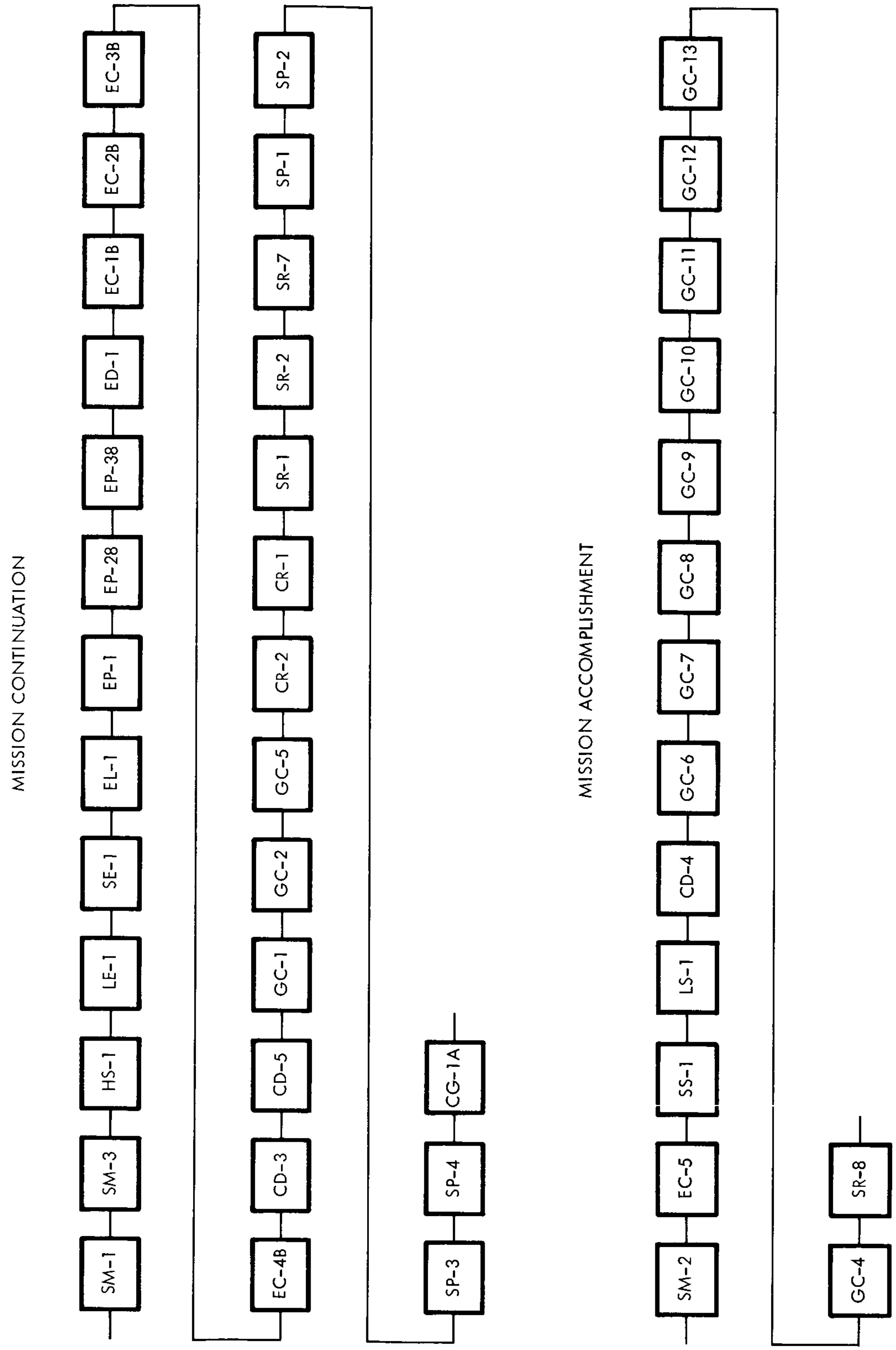
Mission continuation refers to probability of having had, at any point in the mission, no prior failure or combination of failures which would have caused an aborted (foreshortened) mission. This parameter includes only those reliability degradations associated with necessary spacecraft operations, independent of experiments.

Mission Accomplishment includes mission continuation plus additional reliability degradation resulting from imposition of experiment support requirements. This also relates to the probabilities of no prior abortive failures, but it may be further expanded, by applying weighting factors to the individual experiments, to optimize their scheduling to obtain the best possible "accomplishment" value.

Equipment failure rates were assumed to be the same as an unused Block II spacecraft. Figure 86 shows the subsystem logic employed in the analysis. Tables relating the code designations in the logic diagrams to equipment functions are contained in SID 66-872, Definition of the Reliability Evaluation Index (REI) Concepts in Support of AAP Task 4. 1. 1.

Table 42 is a compilation of the results and includes the effects of the launch vehicle, GOSS, and GSE. Specific reliability numerics for the experiments equipment, (Column 6) are not available; however, the degradation of this equipment is not expected to influence the values in Column 8 appreciably.

Figure 87 is a plot of mission success probability and excludes the effect of other Apollo systems.



NOTE: SR-8 EXCLUDED FROM CALCULATIONS. RATES NOT AVAILABLE; PROBABLE SMALL DEGRADATION ONLY.

Figure 86. RCM-CM Subsystem Logic, Earth-Orbital Mission

Table 42. Reliability Estimates for Major System Elements
(RCM Spacecraft, 14-Day Low-Inclination Earth Orbit)

Mission Phase No.	Cumulative Mission Time (Hours)	Mission Continuation					Mission Accomplishment		
		(1) Boosters S-1B and S-IV	(2) GOSS	(3) GSE	(4) CSM	(5) System (1)X(2)X(3)X(4)	*(6) Experiment	(7) CSM	(8) System (5)X(6)X(7)
1	0.05	0.971911	0.96525	0.971	0.999737	0.971655	To be determined	0.999997	0.971652
2	0.25	0.966387	0.9576	0.9655	0.999333	0.965740		0.9999869	0.965727
3	2.50	0.966387	0.9476	0.9555	0.996196	0.962683		0.9999752	0.962660
4	24.0	0.966387	0.93772	0.9455	0.982397	0.949117		0.994248	0.943658
5	72.0	0.966387	0.93316	0.9387	0.959994	0.926971		0.987712	0.915580
6	120.0	0.966387	0.92886	0.93784	0.937736	0.904987		0.975428	0.882750
7	168.0	0.966387	0.92840	0.93698	0.907834	0.875651		0.96696	0.846720
8	216.0	0.966387	0.92795	0.93611	0.885766	0.853906		0.962339	0.821747
9	264.0	0.966387	0.92749	0.93525	0.861516	0.830074		0.958697	0.795789
10	312.0	0.966387	0.92715	0.93450	0.838119	0.807195		0.957078	0.772549
11	336.0	0.966387	0.92695	0.93410	0.829941	0.799123		0.956672	0.764499
12	338.9	0.966387	0.92693	0.92406	0.827321	0.796584		0.956602	0.762014
13	341.37	0.966387	0.92690	0.93402	0.823288	0.793043		0.956602	0.758627
14	342.03	0.966387	0.92689	0.93400	0.822677	0.792076		0.956602	0.757701
15	342.15	0.966387	0.92689	0.93400	0.822536	0.791941		0.956602	0.757572

*Values for experiment equipment not known, but would cause some slight degradation.

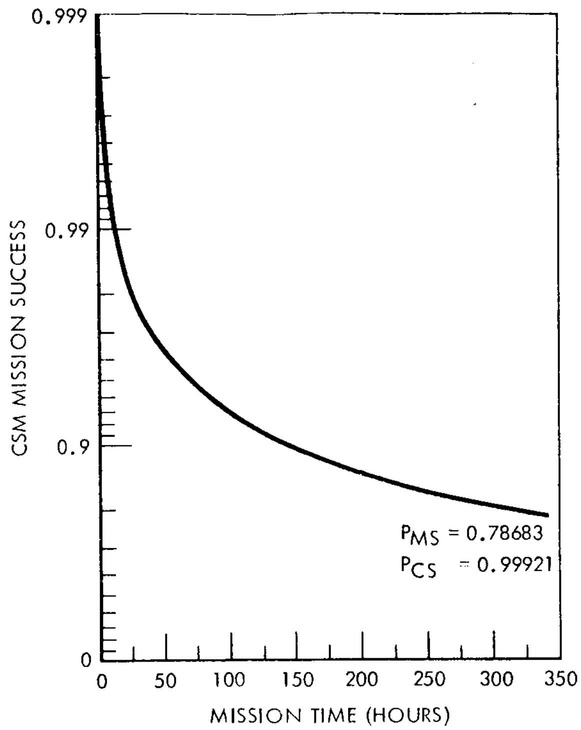


Figure 87. CSM Mission Success, RCM Spacecraft, Fourteen-Day Earth Orbit (Excluding Boost, GOSS, and GSE)

The following ground rules were used in the analysis:

1. 2 stage booster reliability: R = 0.9664 (NASA defined goal).

2. GOSS: R = 0.999 for 105.4 hours (NASA defined goal).

3. GSE: R = 0.9999 for 105.4 hours (NASA defined goal). However, a reliability of 0.9998 was utilized due to experiments.

4. Data were based on a detailed analysis of seven major subsystems (EPS, ECS, CGSS, IES, CMRCS, SMRCS, and SPS). Extrapolation and/or interpolation of Block II apportionments were used for the other subsystems.

5. The effect of nuclear radiation, micrometeoroids, and human factors was not considered.

The REI concept of the AAP studies describes the methodology and contains functional curves and supporting failure rate data.

Table 43 contains the assumed usage rates of equipment/functions for other than fulltime operating equipment or "one use" items such as the launch

Table 43. Assumed Usage Rated for RCM Missions

Mission / Code	14-day EO		30-day LI and PEO		30-day ESO and LO		Function
	Cycle/Hr	Reliability	Cycle/Hr	Reliability	Cycle/Hr	Reliability	
SM-2	3 EVA's	0.937	5 EVA's	0.935	5 EVA's	0.935	Structures and mechanics, EVA's
EC-5	3 EVA's	0.93795	5 EVA's	0.9365	5 EVA's	0.9365	Repressurization
SS-1	3 EVA's	0.945	5 EVA's	0.9432	5 EVA's	0.9432	Space suits
LS-1	3 cycles	0.937	5 cycles	0.935	5 cycles	0.935	Portable life support
CD-4	9 hrs	0.9346	19.26	0.999	19.26	0.999	Data storage
GC-1	5.5 hrs	0.9943	5.5 hrs	0.9943	13 hrs	0.9865	G&C transit and housekeeping
GC-4	9 cycles	0.9349	20 cycles	0.9988	25 cycles	0.99855	SCS warmup
GC-5	10 cycles	0.9329	22 cycles	0.9985			Landmark navigational sightings
GC-6	1 cycle	0.9442	1 cycle	0.9442	35 cycles	0.9981	SCS alignments
GC-7	15 cycles	0.9968	32 cycles	0.9935	10 cycles	0.9978	G&N coarse alignments
GC-8	40 cycles	0.9951	85 cycles	0.990	65 cycles	0.9924	G&N fine alignments
GC-9	72 cycles	0.996	154 cycles	0.9918	150 cycles	0.9918	SCS alignment to G&N
GC-10	75 cycles	0.9872	161 cycles	0.972	160 cycles	0.973	Automatic maneuvers
GC-11	25 cycles	0.9978	54 cycles	0.9953	50 cycles	0.9956	Fine manual maneuvers
GC-12	20 hrs	0.866	42.8 hrs	0.735	110 hrs	0.928	G&N inertial hold
GC-13	8 hrs	0.9983	17.1 hrs	0.9963	25 hrs	0.9946	SCS inertial hold
GC-14					60 cycles	0.994	Star navigational sightings
CD-5	12	0.9386	22 cycles	0.99976	30 cycles	0.99969	Updata link
SM-RCS	500 Seconds, 10,000 Cycles						

escape system. Assumed usage rates for the AAP spacecraft supporting the laboratory for the 30 day baseline missions also are tabulated.

Table 44 contains estimated equipment usage for various AAP missions for which experiments have been defined. It is noted that usage generally is restricted to a relatively small number of items which vary for each mission, depending on experiment requirements. The values contained in Table 44 include most of the functions and therefore represent a worst case condition.

AAP CSM SUPPORTING LABORATORY MISSIONS

Similar analyses to those described above were conducted on the Block II spacecraft which would support the laboratories for the various missions. The configuration reflects the Block II CSM except for the following assumptions:

1. Sufficient consumables, including cryogenics and RCS propellant, were assumed available for the longer duration missions.

2. Wearout was not considered to be a factor within the mission duration.

Since wearout is not considered and all failures are assumed to be random, the analysis is subject to conditional reliabilities. Given that a certain point in the mission (t_o - t_1) has been reached with no failures, the reliability of specific equipments for the subsequent and equal time interval (t_1 - t_2) is the same as for the period t_o - t_2) is the same as for the period t_o - t_1, assuming the same rate of operation. Moreover, since the end reliabilities are cumulative, the unreliability of equipment items whose operation occurs early in the mission (e.g., launch escape) is carried throughout the analysis.

Table 45 contains cumulative and point mission success probabilities for the various RCM baseline missions. Figures 88, 89, and 90 contain plots of mission success versus mission time. These values were obtained by combining mission continuation and mission accomplishment probabilities (e.g., columns 4 and 7 of Table 42) and exclude the other Apollo systems. Assumed equipment usage rates are contained in Table 43. Some of the major contributors to unreliability include EP-1, the basic electrical power system which operates full time, CG-1, cryogenic storage, and the integrated electronics in general. The same usage rates and subsystem logic were used for the 30-day lunar and 30-day earth synchronous orbit with the exception of the electrical power system and service module reaction control system. Here it was assumed that all fuel cells, inverters, and SM RCS quads would be required until attainment of lunar orbit and synchronous orbit respectively; subsequently, at least 2 of three fuel cells and inverters, and 3 of 4 quads would be required.

Table 44. Estimated Integrated Electronics Usage for Typical AAP Missions (Experimental Operations)

Mission / Function	507 - 14-day ESO			509 - 14-day ESO			511 - 14-day L. O.			214/215 earth rendezvous			216/217 earth rendezvous		
	Cycle	Hrs	Reliability	Cycle	Hrs	Reliability	Cycle	Hrs	Reliability	Cycle	Hrs	Reliability	Cycle	Hrs	Reliability
G&N inertial hold								110.0	0.9290		15.85	0.9896		0.25	0.9998
G&N alignment (fine)	44	6.6					65	9.75	0.9922	5	0.75	0.9994	2	0.30	0.9997
G&N alignment (coarse)	26	10.87	0.9911				7	1.75	0.9983	5	1.33	0.9988	2	0.53	0.9995
Auto maneuver	14	3.5	0.9977				16	4.0	0.9977	15	3.75	0.9978	1	0.25	0.9999
Navigation sighting (landmark)										100	8.33	0.9932			
BMAG alignment				35	5.83	0.9995									
Manual maneuver				80	20	0.9950				5	3.25	0.9988			
SCS inertial hold					26.52	0.9951									
SCS warmup				32	32	0.9980				8	8.0	0.9995	1	1	0.9999
G&N local vertical hold		219.9	0.8610												
Translation maneuver					0.34	0.9999					4.8	0.9999			
Navigation sighting (star)							28	2.33	0.9981	100	10.0	0.9906	16	1.60	0.9984
SPS firings							6	0.945	0.9993				2	0.315	0.9997
G&N transit/house-keeping - ascent		*	0.9978		*	0.9865		*	0.9844		*	0.9917		*	0.9979
G&N transit/house-keeping - orbit		263.5	0.9713		274.08	0.9702		*	0.9752		429.0	0.9537		712.66	0.9243

*Reliability is a compilation of various equipment and operating time so no single time may be stated.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 45. CSM Reliability Summary

Subsystem	14-day EO	30-day LIEO	30-day Synchronous ER	30-day LR
Structures and Mechanical	0.9370	0.9350	0.9350	0.9350
SM-1	(0.966)	(0.966)	(0.966)	(0.966)
SM-2	(0.937)	(0.935)	(0.935)	(0.935)
Heat shield HS-1	0.945	0.945)	0.945	0.945
Launch escape LE-1	0.9446	0.9446	0.9446	0.9446
Separation SE-1	0.9387	0.9387	0.9387	0.9387
Earth landing EL-1	0.9434	0.9434	0.9434	0.9435
Electrical power	0.93733	0.86183	0.84690	0.82665
EP-1	(0.94100)	(0.87500)	(0.87500)	(0.87500)
EP-2	(0.99860)	(0.9942)	0.98435)	(0.96527)
EP-3	(0.99750)	(0.9907)	(0.98327)	(0.97874)
Emergency detection ED-1	0.955	0.955	0.955	0.955
Environmental control	0.99532	0.99026	0.90877	0.90877
EC-1	(0.9311)	(0.99804)	(0.97300)	(0.97300)
EC-2	(0.9965)	(0.99270)	(0.99140)	(0.99140)
EC-3	(0.9455)	(0.9403)	(0.94300)	(0.94300)
EC-4	(0.9453)	(0.9453)	(0.9373)	(0.9373)
EC-5	(0.93795)	(0.9365)	(0.933)	(0.933)
Cryogenic storage CG-1	0.93500	0.87500	0.87500	0.87500
Portable life support LS-1	0.9370	0.9350	0.935	0.935
Space suits SS-1	0.945	0.9432	0.9432	0.9432
Integrated electronics	0.91304	0.83110	0.78064	0.78064
CD-2	—	—	(0.98900)	(0.98900)
CD-3	(0.9979)	(0.9955)	—	—
CD-4	(0.9346)	(0.99900)	(0.99900)	(0.99900)
CD-5	(0.9386)	(0.9376)	(0.9369)	(0.9369)
GC-1	(0.99430)	(0.9943)	(0.98650)	(0.98650)
GC-2	(0.96200)	(0.921)	(0.92100)	(0.92100)
GC-3	—	—	(0.9467)	(0.9467)
GC-4	(0.9349)	(0.99880)	(0.92855)	(0.92855)
GC-5	(0.9329)	(0.9985)	—	—
GC-6	(0.9442)	(0.9442)	(0.99810)	(0.99810)
GC-7	(0.99680)	(0.9935)	(0.99780)	(0.99780)
GC-8	(0.99510)	(0.99000)	(0.99240)	(0.99240)
GC-9	(0.99600)	(0.99180)	(0.99180)	(0.99180)
GC-10	(0.98720)	(0.97200)	(0.97300)	(0.97300)
GC-11	(0.99775)	(0.99530)	(0.99560)	(0.99560)
GC-12	(0.98660)	(0.97350)	(0.92800)	(0.92800)
GC-13	(0.99830)	(0.99630)	(0.99460)	(0.99460)
GC-14	—	—	(0.99400)	(0.99400)
CM-RCS	0.99137	0.98147	0.98147	0.98147
CR-1	(0.99140)	(0.98150)	(0.98150)	(0.98150)
CR-2	(0.947)	(0.947)	(0.947)	(0.947)
SM-RCS	0.99948	0.99892	0.99764	0.99368
SR-1	(0.9575)	(0.9548)	(0.9548)	(0.9548)
SR-2	(0.9372)	(0.9421)	(0.9421)	(0.9421)
SR-7/8	(0.93763)	(0.99900)	(0.99772)	(0.99376)
SPS	0.99799	0.99624	0.99608	0.99608
SP-1	(0.9983)	(0.99655)	0.99670	(0.9967)
SP-2			0.9341	(0.9341)
SP-3	0.93693	93693	0.9542	(0.9542)
SP-4			0.94786	(0.94786)
CSM	0.78683	0.60535	0.51203	0.49780

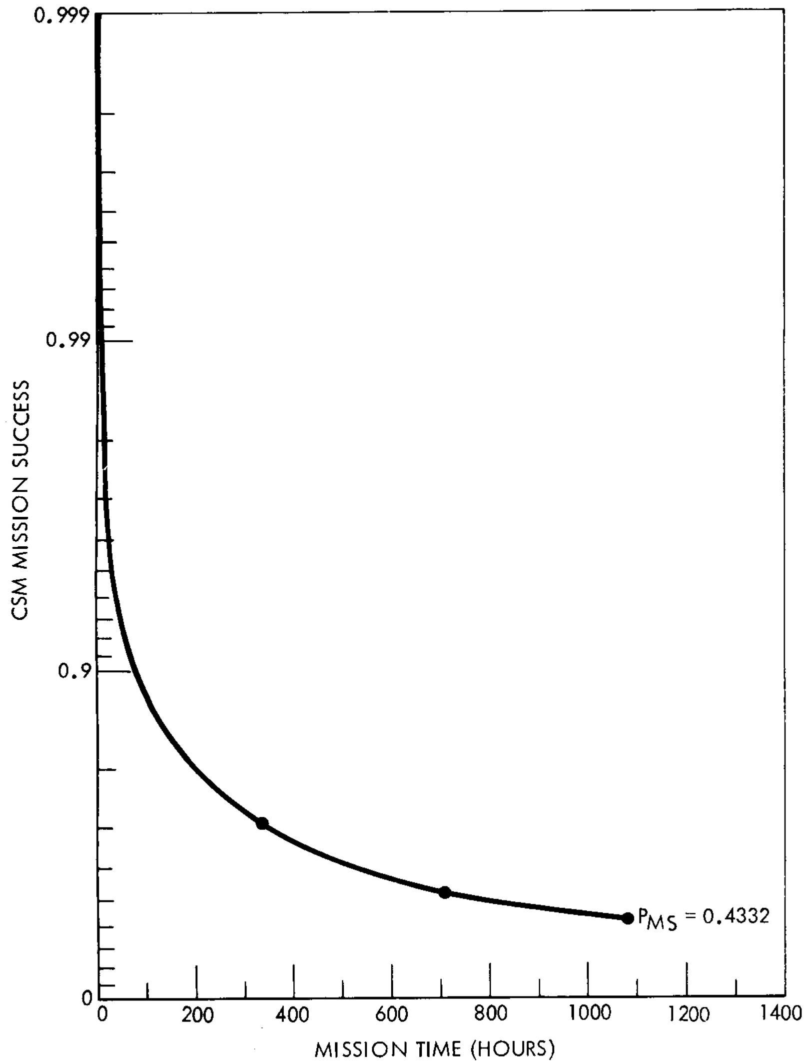


Figure 88. CSM Mission Success, Block II Apollo CM, Earth-Synchronous Orbit (Excluding Boost, GOSS, and GSE)

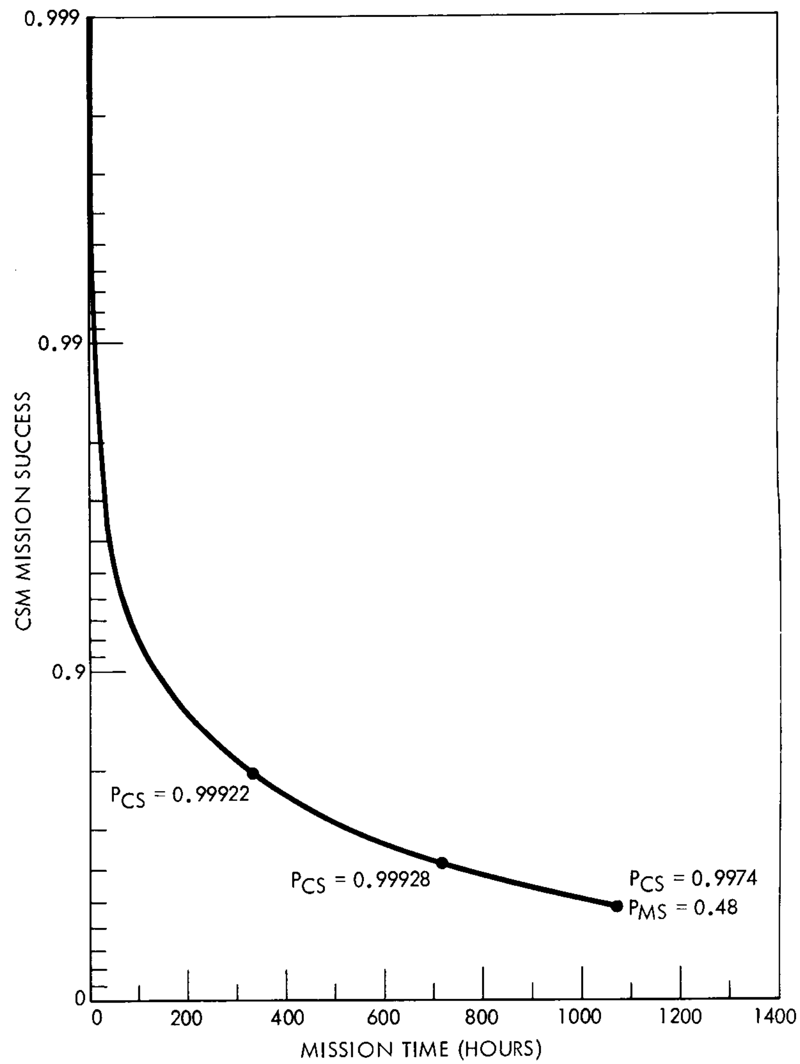


Figure 89. CSM Mission Success, Block II Apollo, Low-Altitude Earth Orbit (Excluding Boost, GOSS, and GSE)

RCM LABORATORY

The fully dependent laboratory essentially is an empty structure and, as such, presents no reliability problems with the exception of the requirement for structural integrity. The independent baseline laboratory contains basic Apollo subsystems except for equipment associated with launch and reentry. In addition, the guidance and navigation equipment has been deleted. For purposes of estimating mission success it was assumed that equipment mounted outside the laboratory (e.g., reaction control) would be installed and protected in such a manner that reliability would not be degraded. Crew safety logic was used for the environmental control, cryogenic storage, and electrical power (except for fuel cells, where a requirement of at least 2 of 3 was assumed) since abandonment of the laboratory need not be effected until services essential to experiments are lost. Table 46 lists mission success estimates for a 45-day independent baseline laboratory. Assumed duty cycles are listed for electronics and reaction control. The reliability values for these functions may be adjusted to reflect different usage rates. Figure 91 contains a plot of laboratory mission success probability versus time, and does not include any effects of other Apollo systems. The independent laboratory represents the worst case as regards reliability because of the equipment installed. The reliability of intermediate laboratory configurations will vary upwards, depending on the specified equipment, approaching the basic structural and airlock reliability as an upper limit.

CREW SAFETY

Crew safety probabilities for the supporting Apollo spacecraft are tabulated in Table 47 for the 30-day lunar orbit and indicated in Figures for 14, 30, and 45 days. Crew safety is defined as the system reliability plus the system unreliability times the probability of safe abort.

$$R_{CS} = R_{MS} + Q_{MS} R_{SA}$$

where

R_{CS} = probability of system crew safety

R_{CS} = probability of no abortive failures in system

Q_{MS} R_{SA} is a complex term representing the summation of the joint probabilities of all possible abort states and subsequent safe aborts.

Use of this "classic" approach would have required development of an extensive computer program and considerable computer time. Therefore the procedure used reflects the approach described and data contained in SID 66-872, Effects on Crew Safety, prepared for the Apollo Applications Program.

Table 46. Independent Laboratory Baseline (45 Days)

Subsystem	Mission Success Probability
Structure	0.99999+
Airlock	0.99970
Docking	0.99998
Electrical power (less cells)	0.99988
Fuel cells (2 of 3)	0.98000
Environmental control	0.99710
Cryogenic storage	0.99800
SCS warmup (20 cycles)	0.99883
SCS inertial hold (40 Hours)	0.99150
Manual maneuvers (50 cycles)	0.99560
SCS alignments (20)	0.99890
Data storage (20 Hours)	0.99890
Communication and data	0.99320
Reaction control (1000 sec burn 10,000 cycles)	0.99008
Separation	0.99998
	0.94301

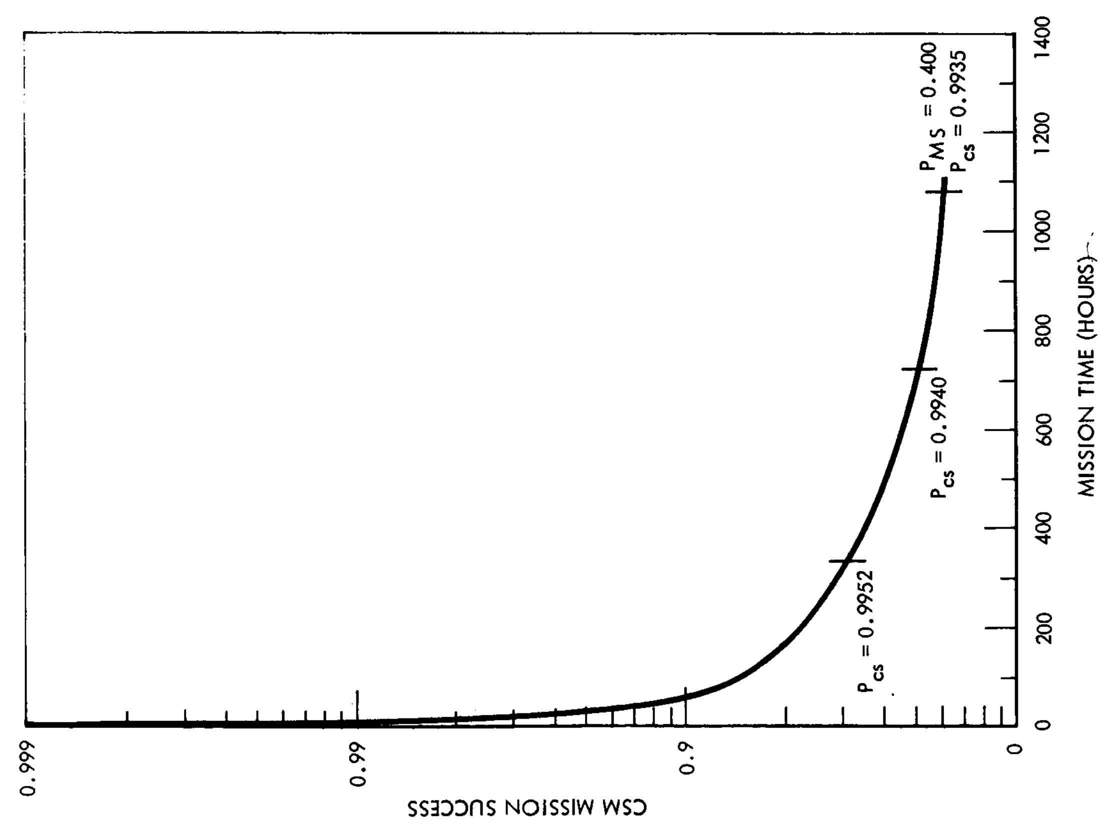


Figure 90. CSM Mission Success, Block II Apollo, Lunar Orbit (Excluding Boost, GOSS, and GSE)

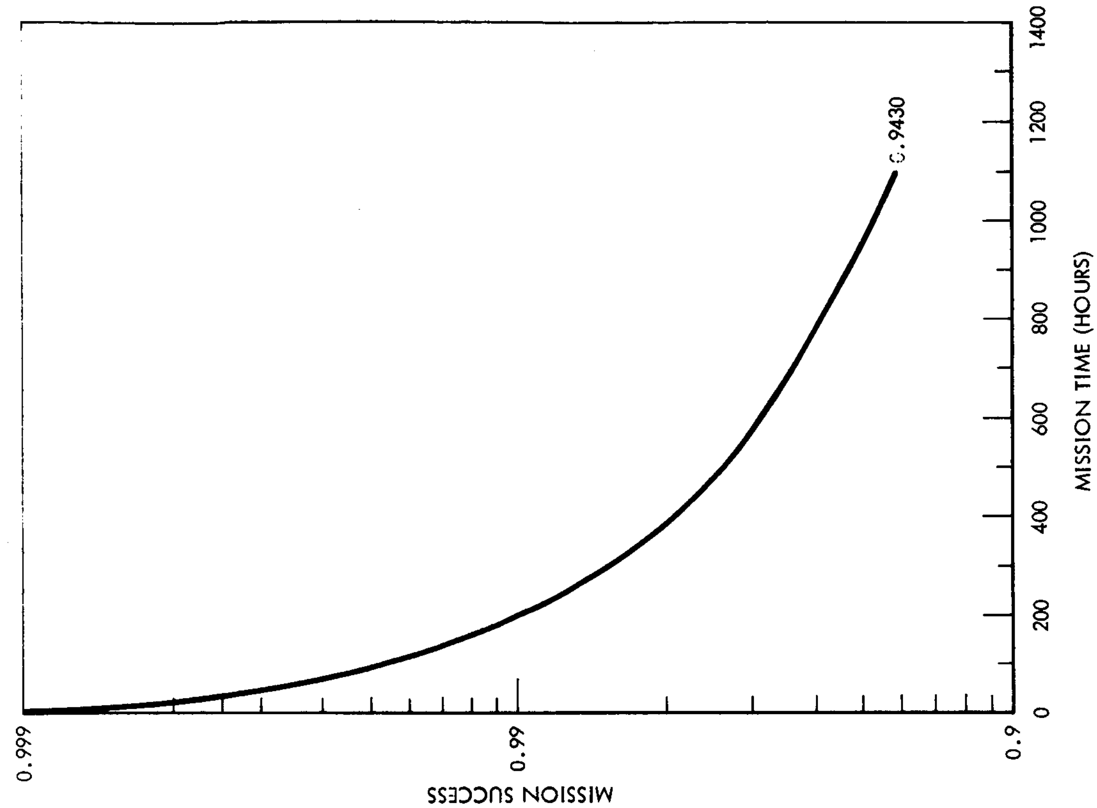


Figure 91. Mission Success, RCM Independent Laboratory (Baseline)

Table 47. CSM Crew Safety, 30-Day Lunar Orbit

System	Crew Safety
Structures	0.99999+
Heat shield	0.99996
Launch escape	0.99996
Separation system	0.99998
Earth landing	
Parachute system	0.99995
Impact and flotation	0.99998
Docking mechanisms	0.99999+
Electrical power	0.99925
Emergency detection	0.99999+
Environmental control	0.99799
Cryogenic storage	0.99856
Space suits	0.99999+
Portable life support	0.99999+
Integrated electronics	0.99950
(Stabilization control) (Guidance and navigation) (Communications and data)	
CM reaction control	0.99992
SM reaction control	0.99997
Service propulsion	0.99928
CSM	0.99432

It is noted that the effect of the laboratory on crew safety is not degrading. The probability of an anomaly requiring abort occurring in the laboratory simultaneously with a similar anomaly in the CSM is extremely remote. Therefore, if an anomaly requiring abort occurs in the laboratory, the abort is initiated in a spacecraft in which the condition of all subsystems is above the point requiring abort, resulting in a higher probability of safe abort.

An analysis of subsystems proposed for the fully independent laboratory indicates that while subsystem failures could occur which would impair or prevent experiment completion, few of these would require abort for crew safety reasons. Failure of any of the subsystems (electrical power regulation and distribution, environmental control, or electronics) reduces the capacity of the laboratory to support experiments or to provide a benign crew environment. Failure of equipment elements mounted outside the laboratory pressure shell essentially would have the same result. However, there are several events, each having a low probability of occurrence, which could jeopardize crew safety under certain conditions. A substantial meteroid penetration of the laboratory pressure shell during an EVA experiment would require closing the access hatch to the command module. The crew member monitoring the EVA from the laboratory should, therefore, be in a pressure suit. This also would expedite rescue operations should the crew member conducting the EVA become incapacitated for any reason. Presumably the command module would be depressurized when the two crew members returned from the laboratory; however, repressurization is within the capability of the ECS. Another potential hazard would be an explosive decompression of one of the exterior-mounted pressure vessels which could cause a shrapnel effect both on the laboratory and the command module unless suitable precautions are taken. Precautions could include wrapping the pressure vessels with material designed to prevent such an occurrence, or to provide shielding for additional micrometeroid impact protection.

Substantial penetration of the laboratory cannot be totally eliminated by design. However, the probability of penetrations greater than 1/2-inch in diameter is extremely low. Precautions against potential catastrophic results of pressure vessel explosions can be incorporated in design. This area has been subjected to detailed study as reported in prior section on micrometeroid safety analysis.

With the exception of a catastrophic type failure as mentioned above, in the remainder of possible equipment failure the crew will have the capability of retreating to the command module, sealing it off if necessary, and then making a decision as to the proper course of action. As indicated earlier, an abort caused by the inability of the laboratory to support further experiments would be accomplished in a CSM possessing more than the minimum amount of required operating equipment, or the mission already would have been aborted.

VIII. CONFIGURATION ANALYSIS AND DESIGN

Using the renovated command module as either a spacecraft or laboratory is a concept based on earlier studies which utilized the command module inner structure as the basis for a multi-mission module (the COMLAB). With a basepoint configuration thus established, it was necessary to consider the affects of utilizing a renovated command module as a refurbished spacecraft and as a laboratory with various stages of dependency on other vehicles such as the AAP CSM. These current studies were conducted utilizing a laboratory configuration which incorporated the same general arrangement as the previous studies. Additional laboratory configuration concepts were investigated in parallel with these studies to determine the effects of possible configuration variables and to determine if the baseline configuration should be revised. Configuration concepts were also established to identify the compatibility of the RCM laboratory with other major system elements for various orbital applications. The renovated spacecraft required no configuration analysis since its arrangement, both exterior and interior, would be identical to the existing Apollo spacecraft, although the proposed mission is low earth orbit only.

RCM LABORATORY ALTERNATE CONFIGURATIONS

The laboratory configuration was reviewed from a number of aspects: revised location of airlock for easier ingress/egress, revised location of laboratory in regard to laboratory mount to improve volume for experiments, etc.

Layouts were made of various laboratory arrangements which utilized a renovated inner structure from a Block I Apollo command module. The design ground rules were established to use only the inner pressure shell structure with all internal structural modifications minimized. Figure 92 shows seven alternate concepts that were studied, in addition to the baseline as defined in the previous NAA RCM Study proposal, SID 66-1135. This baseline concept is shown, with an experiment package, in Figure 93. Each concept described has at least one docking port capable of accepting an Apollo CSM. Concepts 1 and 3 have two docking ports to illustrate the capability of accommodating two Apollo CSM vehicles simultaneously. It was assumed that the experiments, undefined in this study, would be mounted to the laboratory mount structure attaching to the four LEM fittings on the SLA.

Concept 1

Concept 1 presents an arrangement of an RCM laboratory composed of a Block I CM inner structure modified by the installation of a Block II forward tunnel and pressure hatch assembly, with a LEM docking drogue replacing the original tunnel assembly on the forward bulkhead. A new 36-inch O.D. airlock assembly, with Block II pressure hatches and a LEM docking ring and drogue, are installed in the center of the aft pressure bulkhead. The airlock provides ready access to space when the drogue is removed unless there is an Apollo CSM docked to the adapter end of the airlock. This aftmost docking adapter is used only as an alternate port with the forward docking adapter serving as the primary docking position for an Apollo CSM. The laboratory experiment mounting structure is located beneath the laboratory, similar to that of the COMLAB basepoint configuration.

Advantages

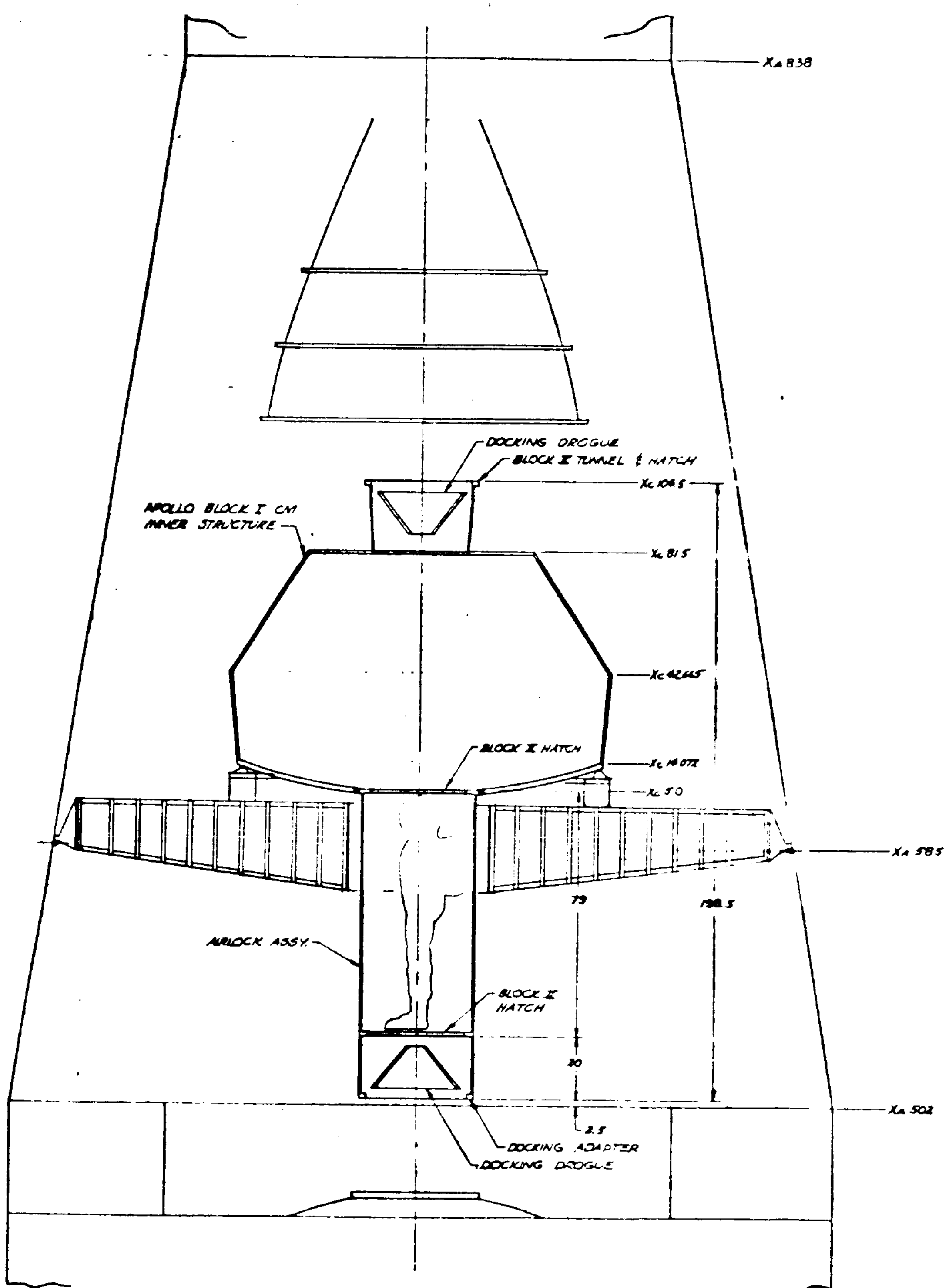
The lab and docked Apollo CSM would make a compact spacecraft combination. The experiments could be mounted conveniently under the support spider-beam assembly with access to them through the airlock. The docking port at the aft end of the airlock would permit a second Apollo CSM to dock and still keep all vehicles concentric. The overall length is compatible with the space available in the SLA and the spider-beam assembly could be located under the laboratory and at SLA Station Xa585 in the same manner as COMLAB.

Disadvantages

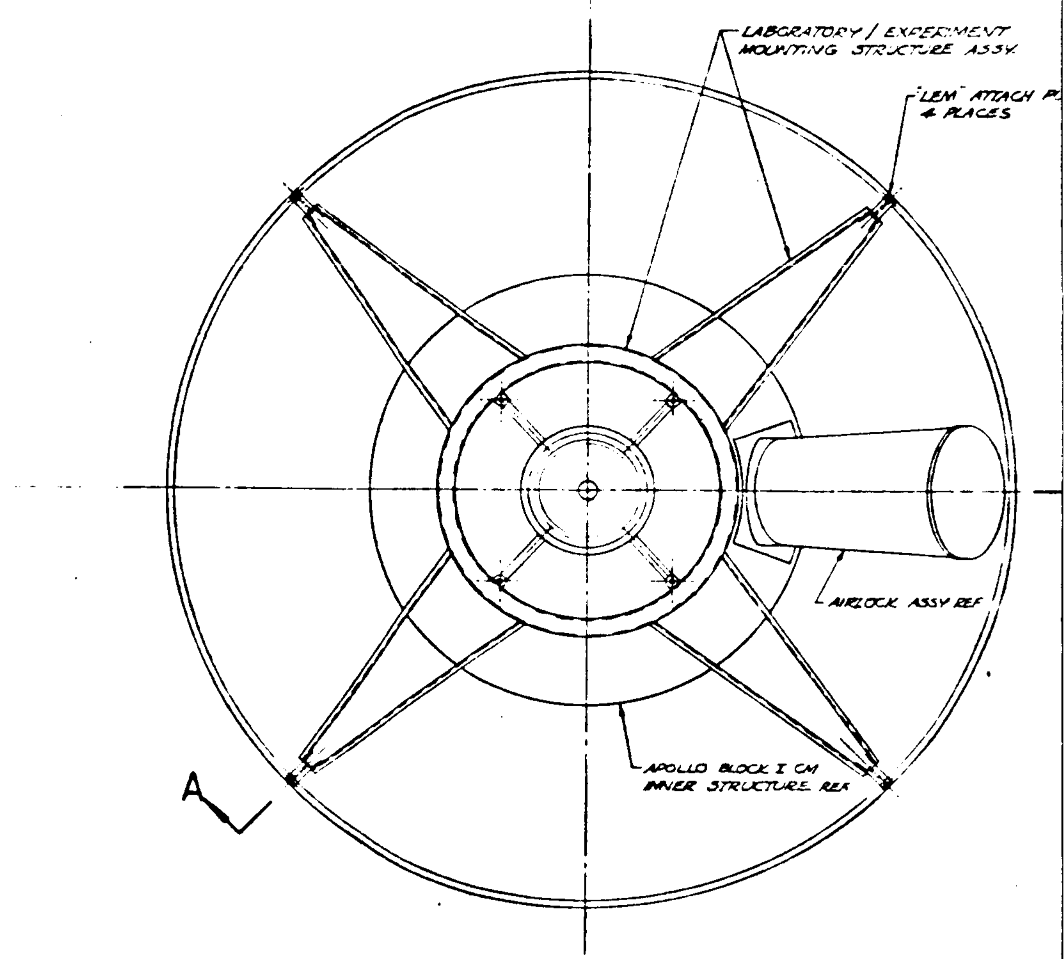
This concept requires a major modification of the inner structure involving both the forward and aft bulkheads. The airlock location in the center of the "floor" is undesirable and wasteful of prime floorspace in the lab. The straight-through airlock configuration requires two men for operation because the crewman in the airlock cannot operate the pressure hatch located at his feet; a second man is required to open and close this hatch. The in-line location of the alternate docking port on the aft end of the airlock places the LEM docking drogue in the path of the exit from the airlock and requires special storage provisions for it when the airlock is in use. If an Apollo CSM is docked at the alternate port, it prevents the use of the airlock for EVA activities.

Concept 2

The Concept 2 arrangement consists of a Block I CM inner structure, requiring a minimum of changes to conform to the basic laboratory configuration requirements. The present forward tunnel has the end fitting



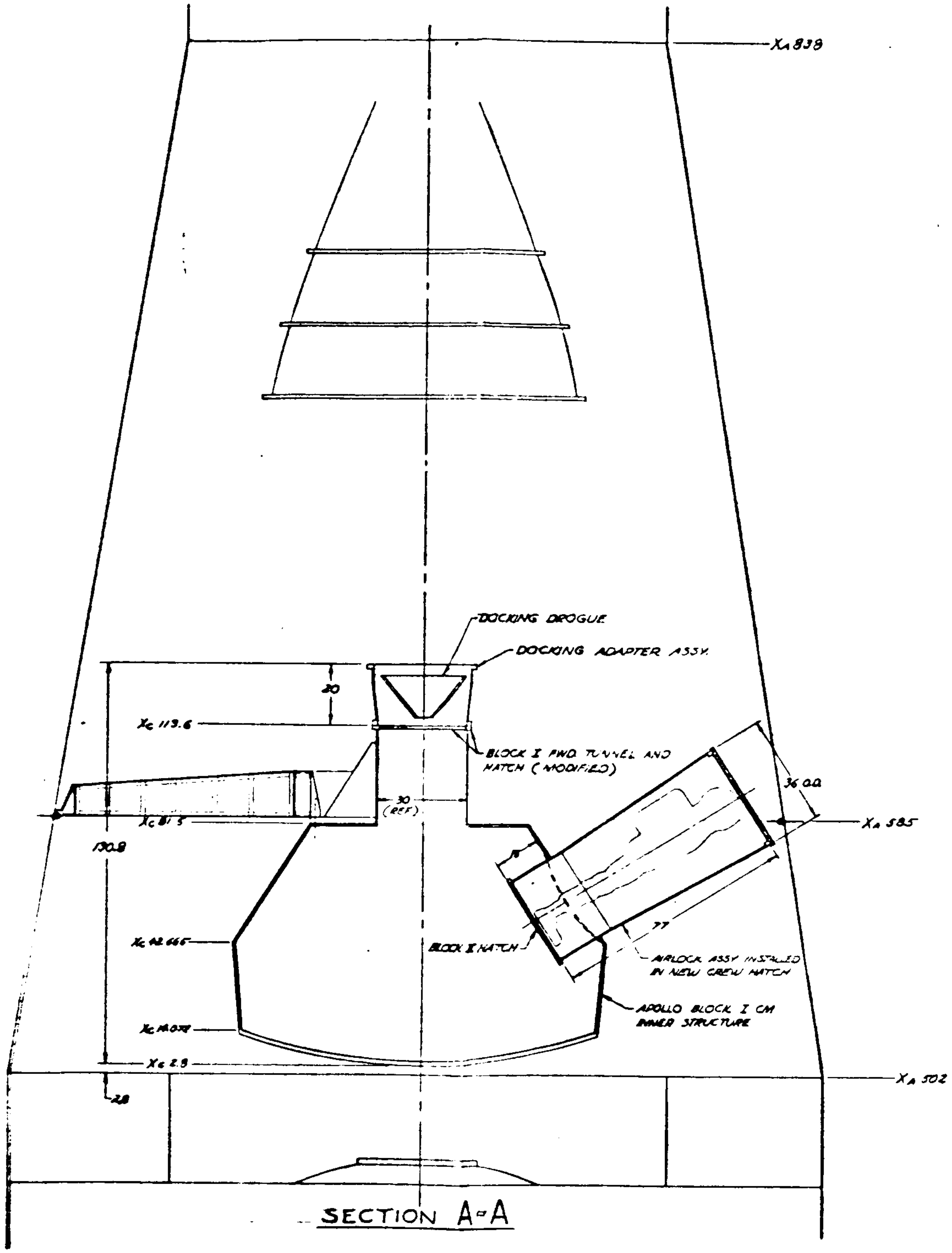
CONCEPT 1



VIEW LOOKING AFT

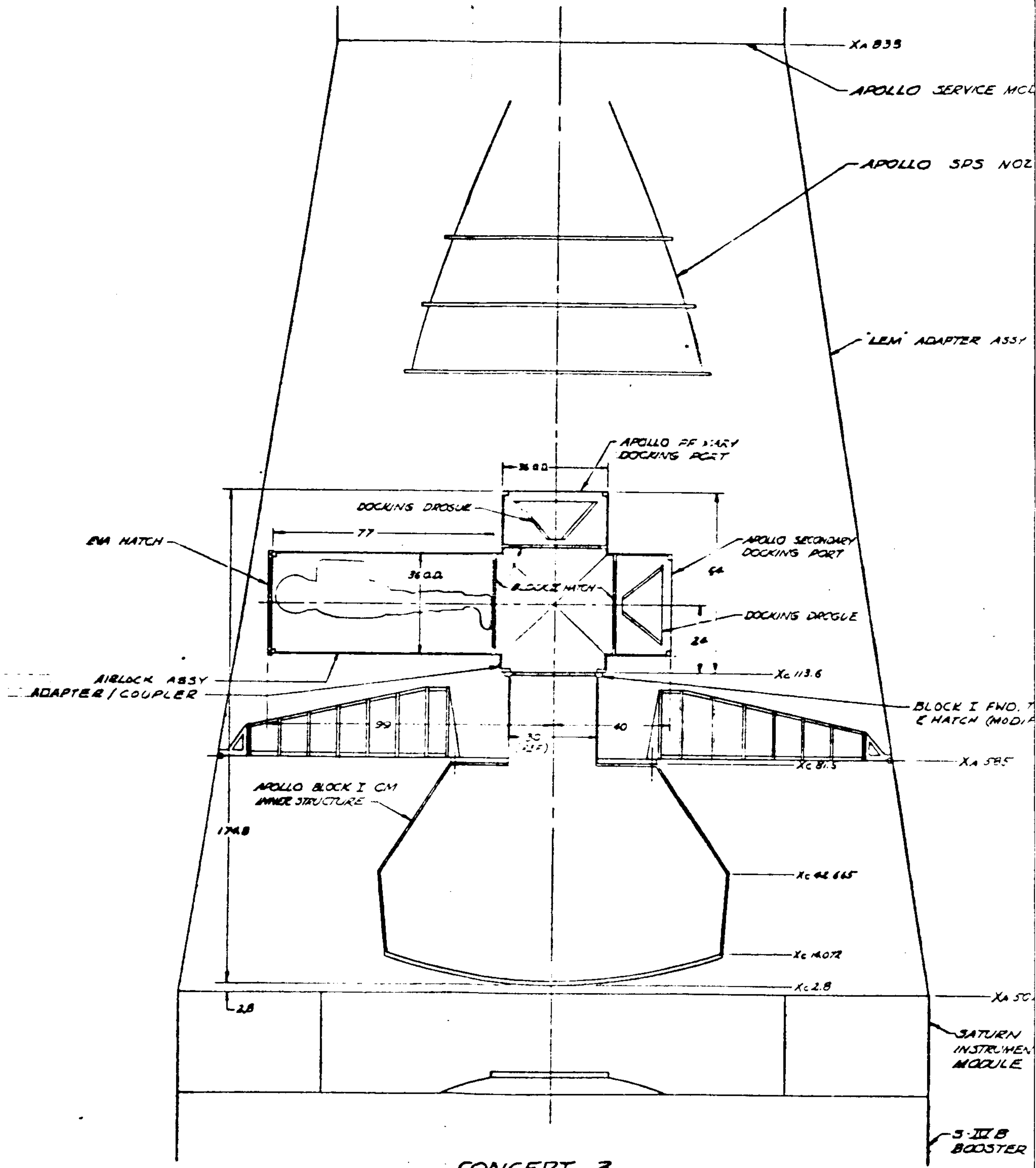
FOLDOUT FRAME 1

FOLDOUT FRAME 2



SECTION A-A

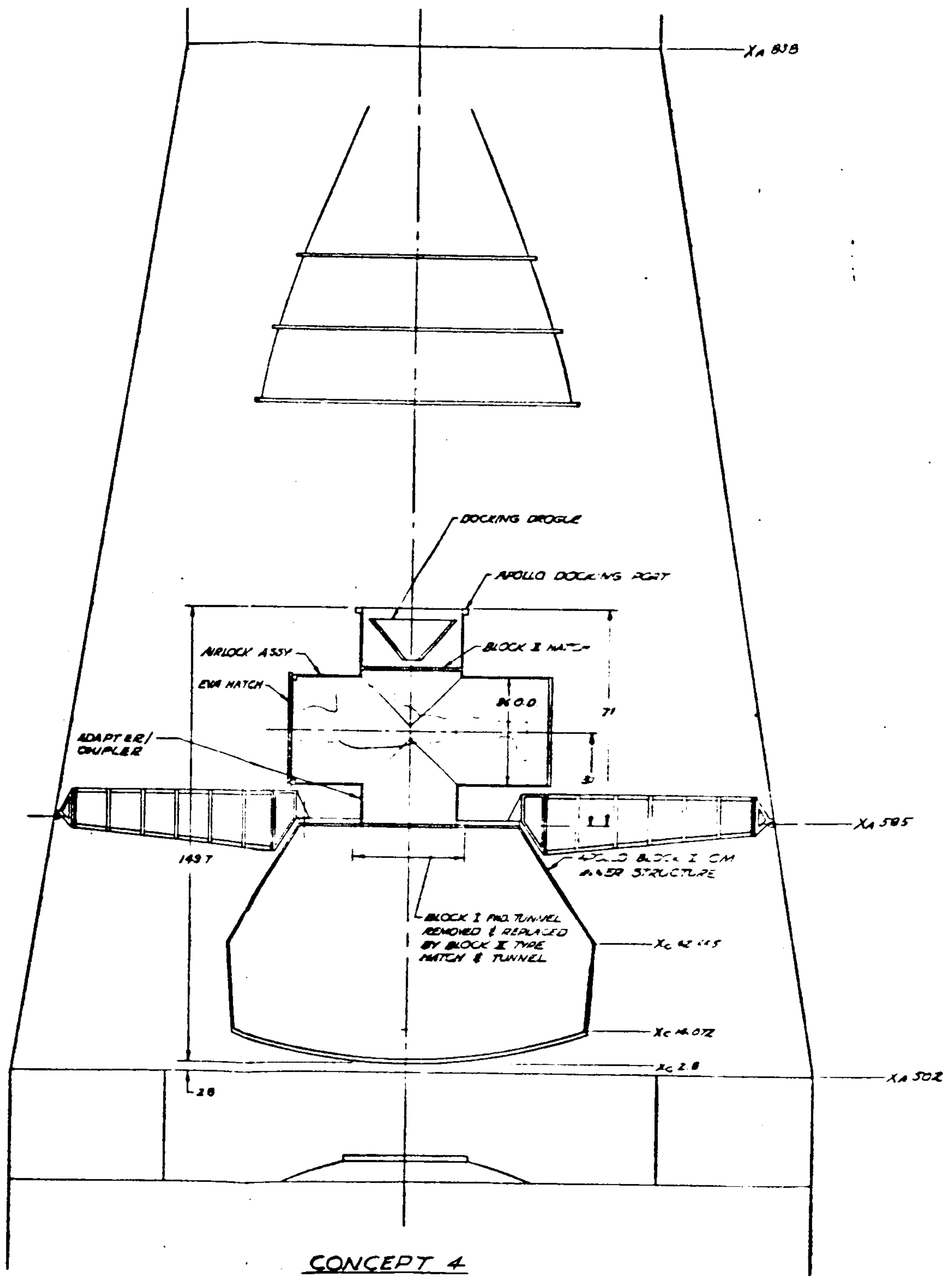
CONCEPT 2



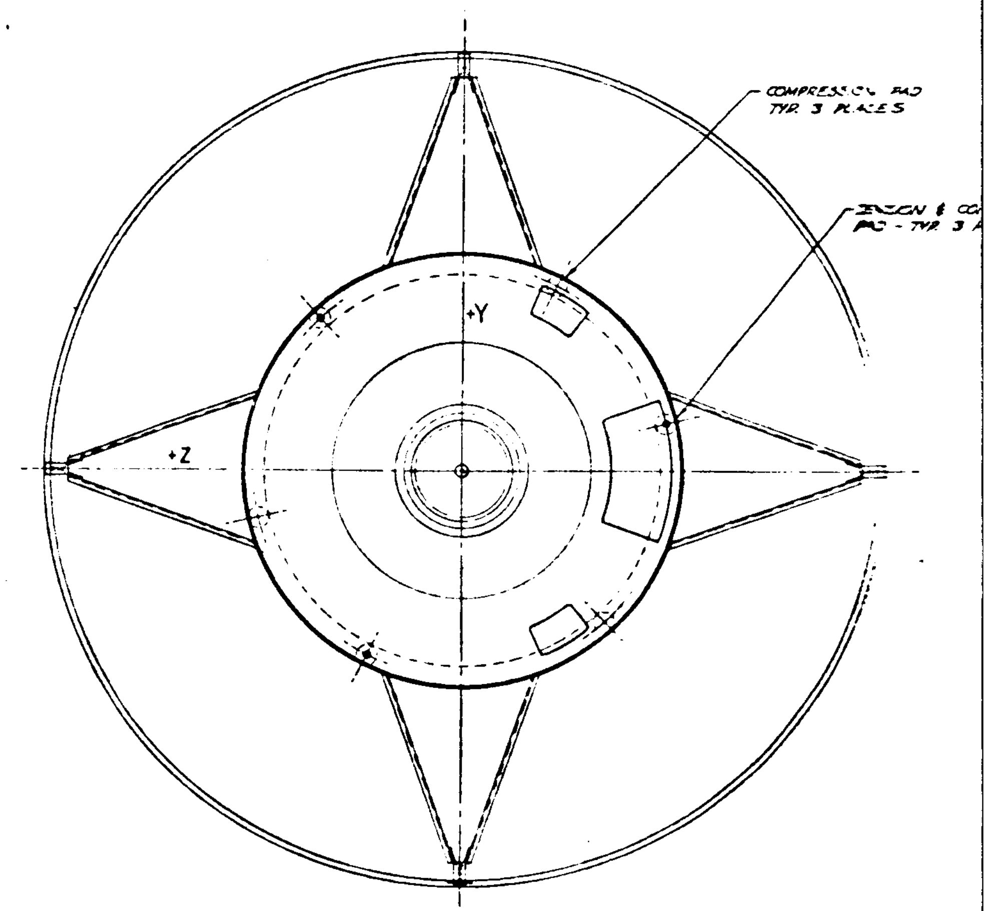
CONCEPT 3

FOLDOUT FRAME 3

FOLDOUT FRAME 4



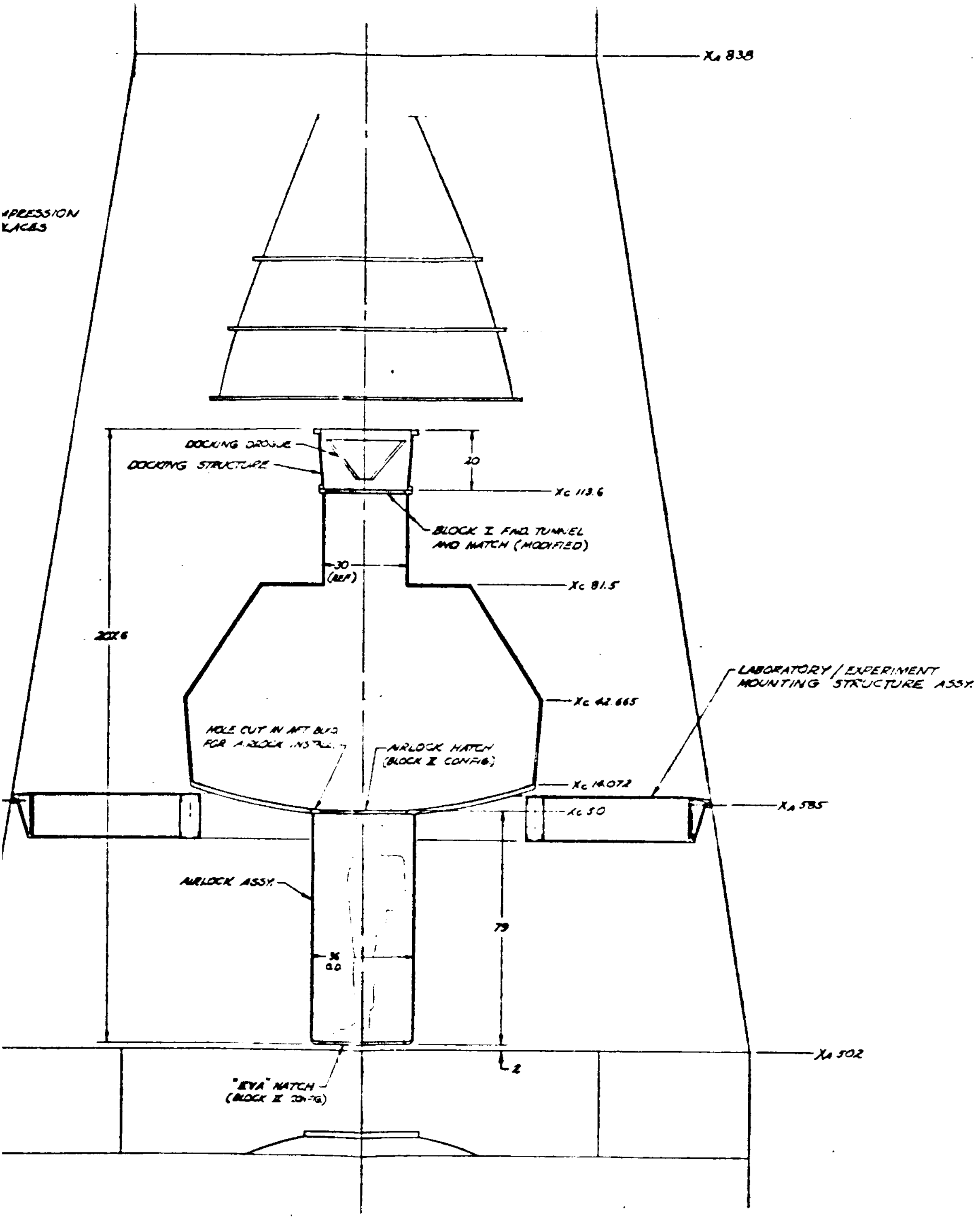
CONCEPT 4



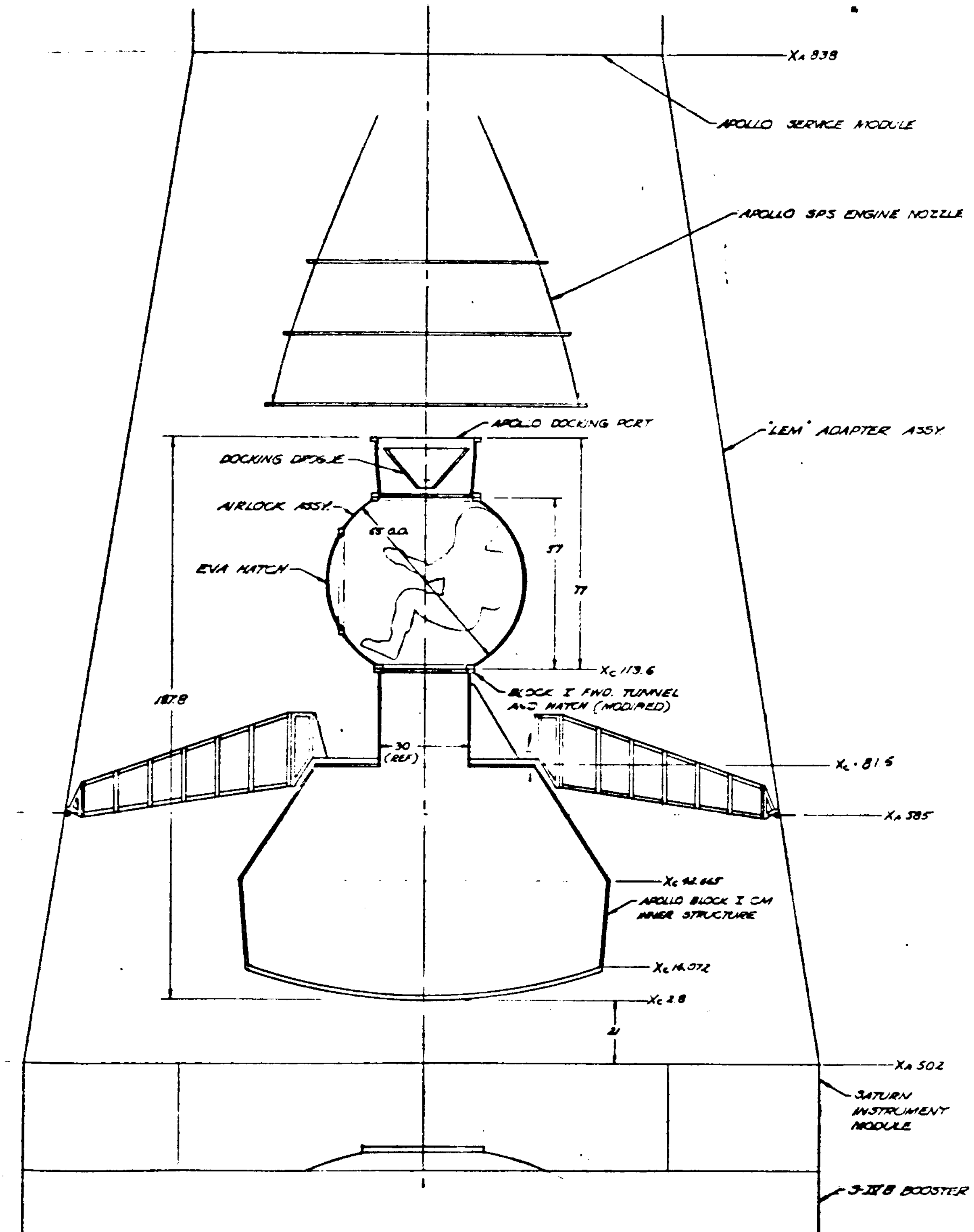
VIEW LOOKING AFT
CONCEPT 5

FOLDOUT FRAME

FOLDOUT FRAME 5



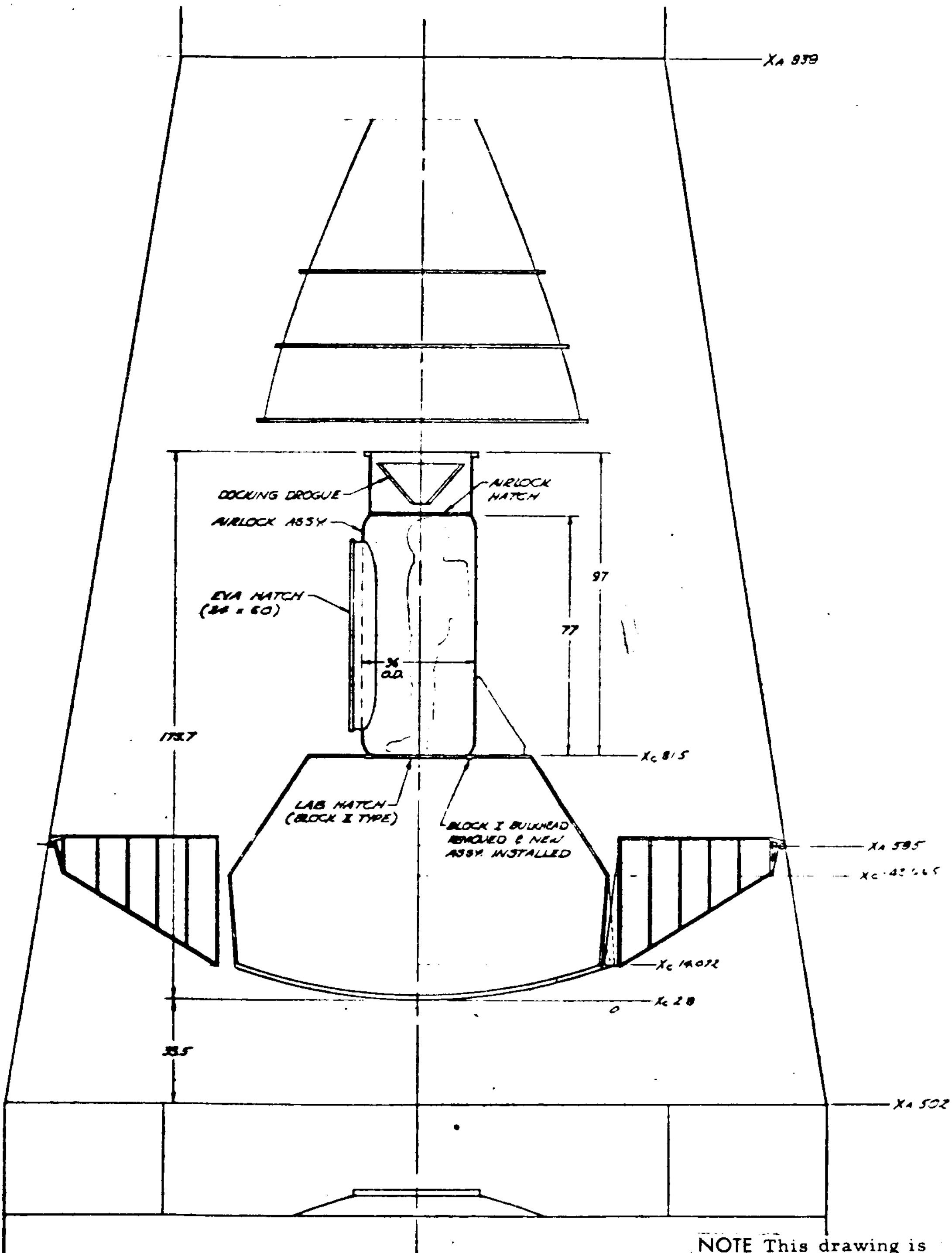
CONCEPT 5



CONCEPT 6

FOLDOUT FRAME 6

FOLDOUT FRAME 7



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CONCEPT 7

Figure 92. Candidate Laboratory Concepts

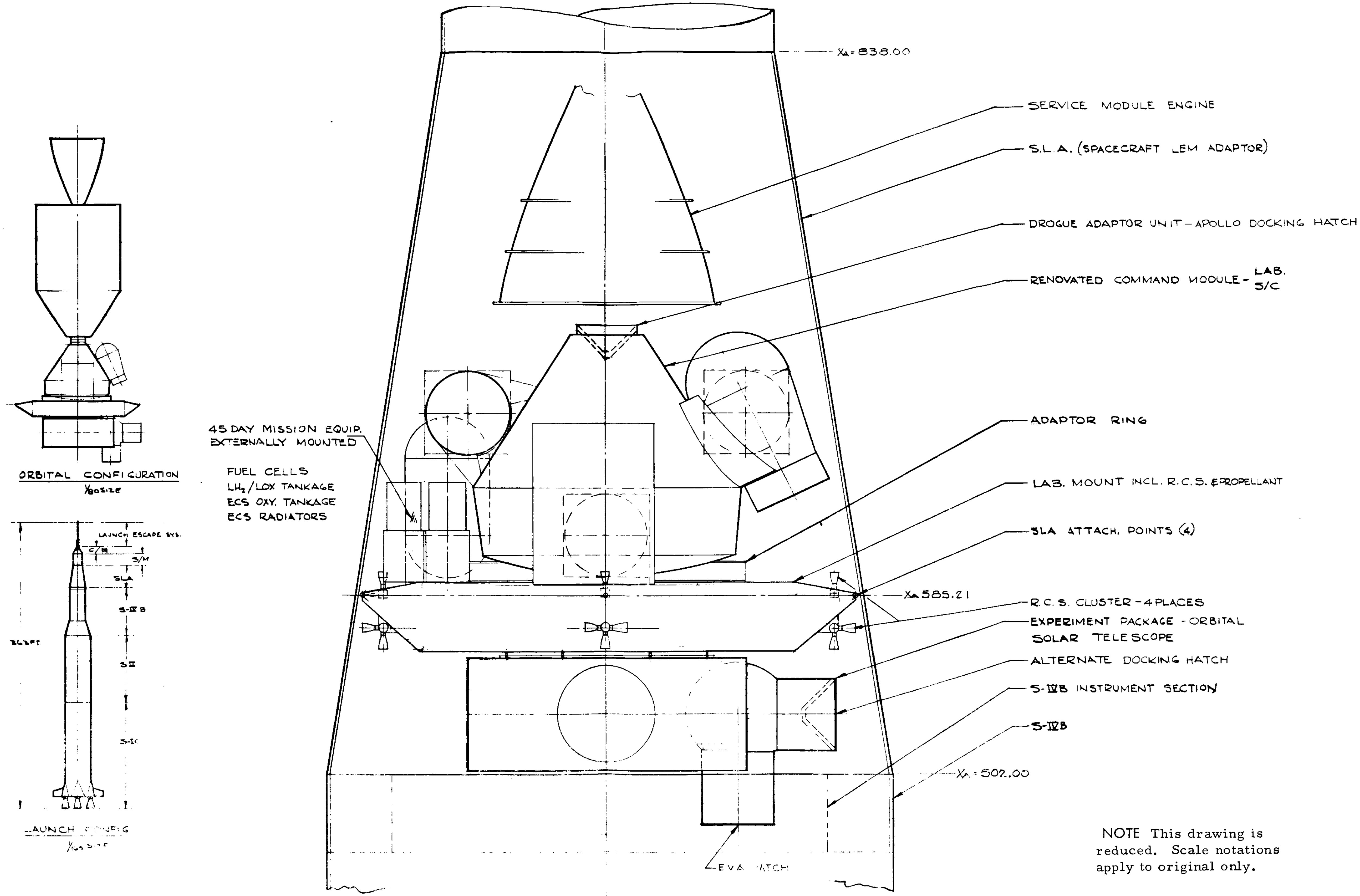

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Figure 93. Design Concept, RCM Laboratory Experiment Package

at Station Xc113.6 modified to accept a Block II pressure hatch and a 20-inch long docking adapter containing a LM drogue installed on the forward end. The side crew hatch is replaced by a new hatch design containing a one-man airlock of a straight-through design. The complete lab is suspended on the four Apollo LES tower leg attachment bolts from underneath the structural spider-beam assembly that attaches to the LM fittings in the SLA.

Advantages

Concept 2 offers interesting possibilities in the manner of structural support during launch because it employs the four main longerons of the inner structure in tension, and could result in a lighter overall weight for the laboratory and the experiment mounting structure. The only structural modification required would be at Station Xc113.6, where the forward end fittings are changed to use a Block II pressure hatch and mount an Apollo docking adapter containing a LM drogue. The docked laboratory and Apollo CSM would form a compact assembly in orbit. The simple straight-through airlock mounted within the confines of the crew side hatch permits a spaceman to egress directly into the area occupied by the experiments on the spider-beam assembly and still stay within fields of view from the laboratory and the Apollo CSM. The location of the airlock on the side permits egress to free space at any time without disturbing the pressurized crew tunnel connecting the laboratory and the docked Apollo CSM. The forward tunnel can conveniently be used for access to the interior of the laboratory when it is mounted inside the SLA.

Disadvantages

The location of the structural spider-beam assembly on the forward bulkhead of the laboratory breaks up the volume available for experiments. It would be difficult to mount large experiment antennas or telescopes on the forward surface of the spider-beam assembly because of the confined space available between the docked Apollo CSM and the laboratory. The side airlock cannot be used for access to the interior of the laboratory when it is mounted for launch within the SLA. The straight-through airlock configuration requires an additional crewman to operate the hatches when it is cycled in flight.

Concept 3

Concept 3 represents the minimum modification to a Block I CM inner structure required to make a laboratory. The forward tunnel fitting at the Station Xc113.6 is modified to accept a Block II pressure hatch and the attachment of a special adapter/airlock/coupler (AAC) assembly. A 36-inch O.D. airlock together with two docking adapters comprise the AAC assembly.

Each docking adapter contains a Block II pressure hatch and a LM drogue suitable for receiving an Apollo CSM. The laboratory is supported by a structural ring and four beams in the same manner as Concept 2.

Advantages

This arrangement requires a minimum of structural modifications confined to the forward end of the tunnel at Station Xc113.6 of the laboratory. It provides a true airlock that can be used for egress to space without disturbing the pressurized tunnel between the laboratory and a docked Apollo CSM. It also provides an alternate docking port for a second Apollo CSM. The location of the structural spider-beam on the forward bulkhead of the laboratory could be used to accept the loads from the AAC and therefore keep them out of the laboratory itself. Most experiments and the airlock would be located in the area between the laboratory and the CSM where visual observations can be easily made.

Disadvantages

The AAC is a very complex structure and contains four pressure hatches, each of which is a source of pressure-leaks during flight. The straight-through airlock requires a crewman inside to help open the hatches located at the feet of the man in the airlock. The physical arrangement of this assembly breaks up the available space within the SLA and prohibits the installation of large experiment antennas and telescopes. The location of the alternate docking port at right angles to the laboratory CSM centerline may present problems in stability and environmental control during flight.

Concept 4

This concept employs a minimum length AAC mounted to a renovated Apollo Block I CM inner structure to form a laboratory. The forward tunnel assembly is removed and replaced by an AAC structure similar to a Block II tunnel and pressure hatch in the attachment area at the forward bulkhead of the CM inner structure. The airlock is located in the middle of the AAC assembly and the forwardmost portion consists of an Apollo docking port with a removable LM drogue. The laboratory assembly is supported by tension bolts at the four Apollo LES tower leg attachment fittings from a structural spider-beam assembly mounted into the four LM attach points of the SLA.

Advantages

The modifications to the inner structure are confined to the center portion of the forward bulkhead. The laboratory is supported from the four LES tower attachments on the forward bulkheads by the structural spider-beam that also supports the AAC in an efficient manner. Experiments would be mounted on the spider-beam between the laboratory and the Apollo CSM where they are readily accessible from the airlock and most may be seen from the interior of the docked vehicles. This configuration is a compact arrangement when docked to an Apollo CSM in flight.

Disadvantages

When the airlock of the AAC is being used, it isolated the laboratory from the environment of the Apollo CSM and prevents crew passage from one to the other. The airlock should be about 44 inches in diameter to ensure use by a spaceman wearing the present spacesuit design. The installation of the AAC assembly requires major rework of the forward bulkhead of the laboratory structure.

Concept 5

Concept 5 at first glance appears to be the same as Concept 1, but is actually very different. The existing Block I forward tunnel is modified at Station Xc113.6 to accept a Block II hatch and a 20-inch-long docking adapter containing a LM docking drogue. A 36-inch-diameter airlock with Block II hatches on each end is installed in the center of the aft pressure bulkhead. The laboratory assembly is supported on the existing six bearing pads on the underside by a spider-beam structure similar to the basepoint COMLAB configuration.

Advantages

The modification of the forward tunnel of Station Xc113.6 to employ a Block II pressure hatch and an Apollo docking port is a fairly simple change. The location of the airlock assembly in the aft bulkhead permits free use of the airlock without interfering with the connection between the laboratory and the CSM. The location of the structural spider-beam provides a clean mounting platform for the laboratory and the experiments. Access to the interior of the laboratory on the launch pad would be in the conventional, way through the crew side hatch.

Disadvantages

The overall length of the assembly is excessive and wasteful of space in the SLA. The location of the airlock in the center of the "floor" of the laboratory is undesirable and the straight-through airlock requires a second crewman to aid the spaceman in the airlock in the operation of the hatch at his feet. Experiments located on the lower surface of the spider-beam could not be seen from the laboratory, although some could be seen from the docked Apollo CSM.

Concept 6

This arrangement also represents a minimum modification to the Block I CM inner structure in the same manner as Concept 3 to make a laboratory. The concept requires the modification of the forward tunnel at Station Xc113.6 to accept a Block II pressure hatch and the attachment of a special airlock/docking assembly. A 65-inch O.D. spherical airlock is attached to the laboratory at Station Xc113.6 and a 20-inch-long docking adapter and Block II pressure hatch mounted diametrically opposite for Apollo CSM docking. An EVA hatch is provided in the side of the airlock for egress to free space. The laboratory assembly is supported by four tension bolts mounted on the forward bulkhead in the four Apollo LES tower leg fittings and underneath a structural spider-beam assembly that picks up the four LEM/attach points of the SLA.

Advantages

The location of the airlock between the laboratory and the docked Apollo CSM permits the crewman using the airlock to be seen during EVA activities. The arrangement of the airlock and docking port requires a minimum of modifications to the forward tunnel at Station Xc113.6.

Disadvantages

The arrangement of the airlock requires the spaceman to do considerable maneuvering to get in and out of the hatch for EVA. The airlock cannot be conveniently supported from the structural spider-beam assembly. In the same measure, the spider-beam assembly would be more complex than for most other concepts. The general arrangement wastes available space within the SLA.

Concept 7

Concept 7 has the forward tunnel of the laboratory inner structure removed and replaced by a 36-inch-diameter airlock assembly mounted directly in the forward pressure bulkhead with a Block II pressure hatch located at Station Xc81.5. A 20-inch-long docking adapter containing a LEM drogue is mounted on the forward end of the airlock structure to be used by docking with an Apollo CSM. A 24-by-60-inch pressure door is located in the side of the airlock structure for EVA use. The laboratory may be supported from the forward or aft bulkheads, and the selection must be based on reasons other than those presently available to the design group.

Advantages

The conventional door in the side of the airlock is by far the most convenient of any studied. The existing side crew hatch would remain unchanged and could be used for access to the interior of the laboratory when it is within the SLA. All structural modifications to the assembly, for construction of a laboratory, would be confined to the forward pressure bulkhead.

Disadvantages

The airlock installation shown requires considerable rework of the forward bulkhead of the laboratory, and is the most complex of any of the concepts studied. The 24-inch-wide-by-60-inch-long airlock door would present a fastening-and-sealing problem more difficult than the circular hatches of the preceding concepts. When in use, the airlock would isolate the laboratory from the CSM. This configuration, although different from the straight-through type, still requires a second crewman to operate the hatch at the feet of the spaceman. Neither the upper nor lower bulkhead of the laboratory lies in the plane of the LM support points in the SLA; therefore, the spider-beam structure will be heavier and more complex than if straight-across structure could be used. The docked Apollo CSM and laboratory assembly would be long and possibly heavier than most other concepts.

Conclusions

The conclusions presented here represent an attempt to find a configuration that could serve the requirements for a renovated CM laboratory as well as the COMLAB concept. Again, it should be noted that the COMLAB is considered as the basepoint design, and this study serves only as a backup effort.

Concept 2 appears to be the only design offering the desirable features of the COMLAB. The straight-through airlock design would be a decided improvement over the airlock design envisioned for COMLAB when used in a zero-g environment. The mounting of the structural spider assembly to the forward instead of the aft bulkhead of the laboratory appears satisfactory but would require further investigation. Otherwise, Concept 2 and the COMLAB appear similar in problem areas, and no significant advantage would be gained by changing the design basepoint.

LABORATORY ORBITAL CONFIGURATIONS

The basepoint configurations for the laboratory was used in establishing several representative orbital configurations where the laboratory was used in conjunction with a solar telescope and the NASA basic subsystem module. These concepts are illustrated in Figures 94, 95, and 96, and identify representative orbital configurations for various mission applications. The layouts are intended to convey the versatility of the laboratory concept as it may be used with experiments only or combined with other modules to provide more capability for orbital missions.

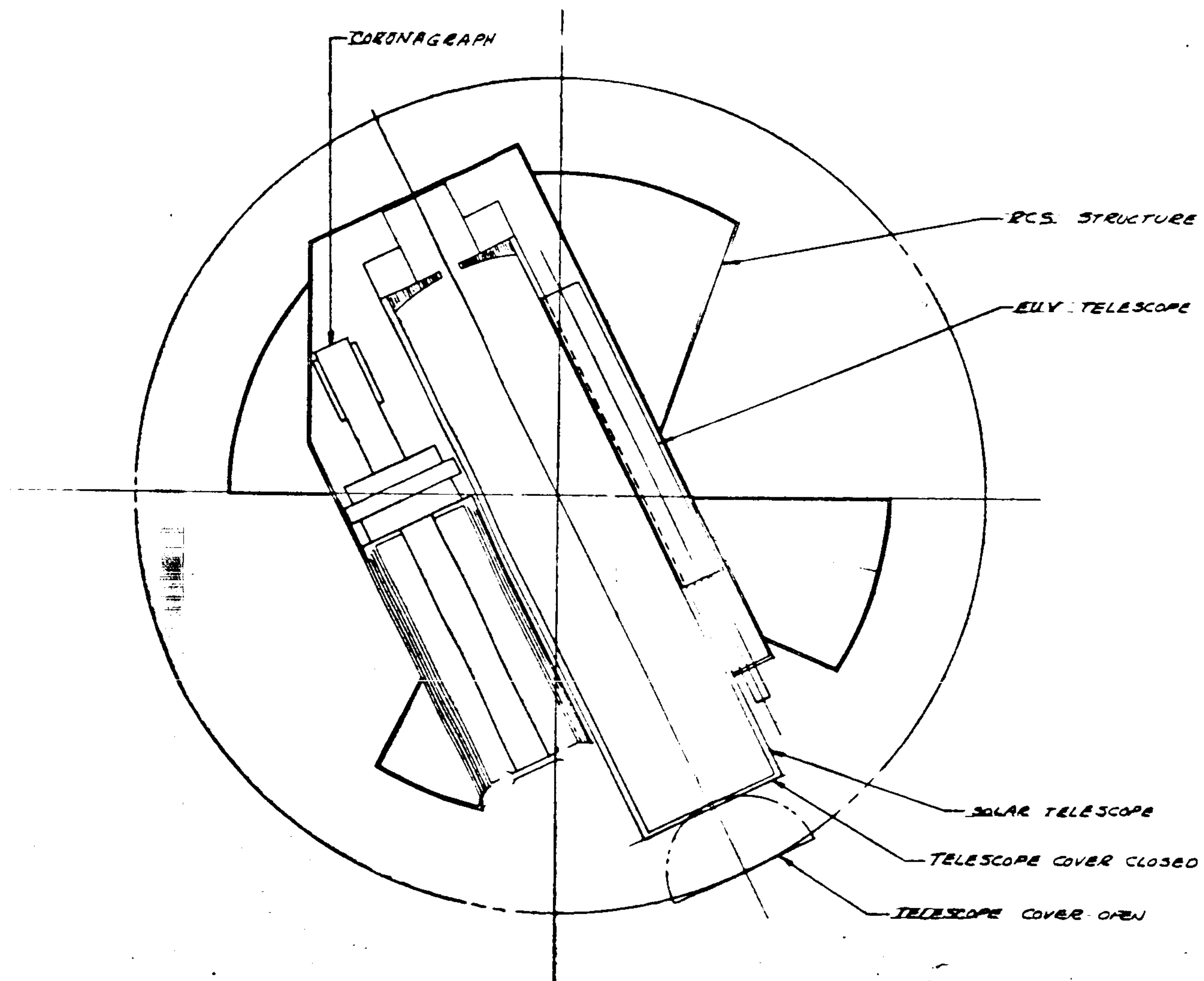
Figure 94 shows an independent laboratory used for a solar telescope mission. The solar telescope experiment equipment is mounted below the laboratory mount and oriented to permit extension of the coronagraph between the RCS panels of the laboratory. Ample volume is available below the laboratory for the experiment installation so that no problems are apparent in the boost configuration. The orbital configuration is shown with an Apollo CSM docked to the laboratory; however, it is possible to operate the laboratory/experiment because the laboratory can operate independently.

In recognition of the existence of other concepts similar to the RCM laboratory, Figure 95 shows an arrangement utilizing one of those concepts, the NASA basic subsystem module (BSM). In this arrangement, the RCM laboratory is suspended from the laboratory mount and the BSM is mounted on the upper surface of the mount. An adapter ring connects the docking tunnel of the RCM laboratory to the center docking port on the BSM. It will be noted that the 183-inch-diameter BSM may be accommodated in the Apollo SLA by removing the existing tubular mounts and replacing them with mount fittings to mate with the laboratory mount. The orbital configuration indicates the capability of accommodating several Apollo-type vehicles simultaneously.

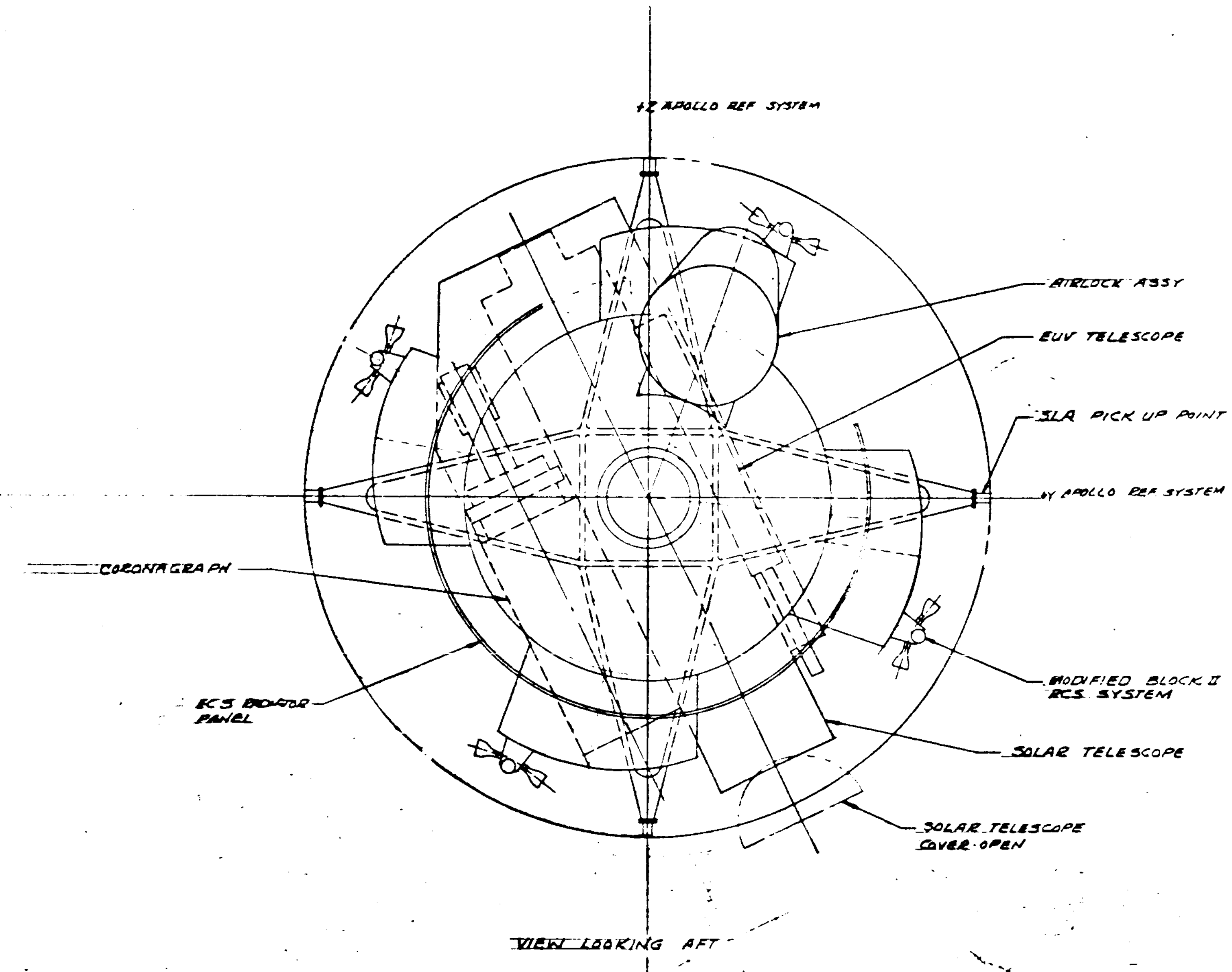
Figure 96 is similar in the use of the BSM, except that the NASA module is used virtually unchanged and the RCM laboratory is installed on the upper surface of the BSM. This concept eliminates the laboratory mount structure but retains the laboratory adapter ring for supply to the BSM. An opening is required in the laboratory floor for access to the BSM, thus reducing the amount of prime floor space available for laboratory operations. The orbital configurations show how several Apollo-type vehicles may be docked to the BSM and how the laboratory/BSM concept may be utilized in conjunction with the S-IVB spent-stage concept.

STRUCTURES AND DYNAMICS ANALYSIS

The structures and dynamics analysis has considered three flight configurations in which the Apollo CM inner structure, functioning as a laboratory, is mounted on a cruciform structure in the SLA. The four LM pick-up points are used in all configurations. Consideration has been given to both a truss and a beam design that can support the laboratory from the LES attachments at CM Station 81.5 or the SM attachments at CM Station 14. No structural problems were encountered within the limits of the SLA structural capability.



VIEW A-A

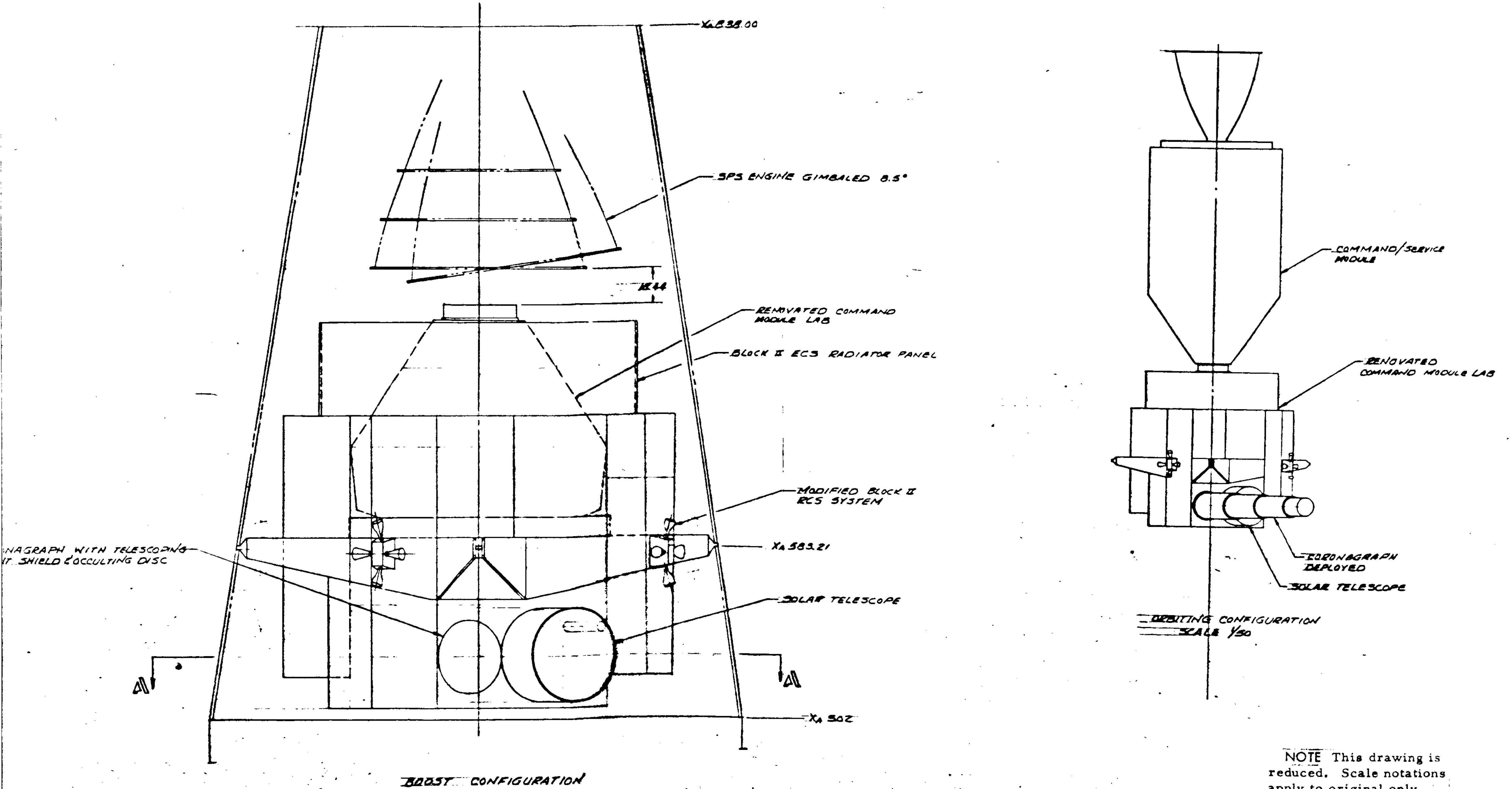


VIEW LOOKING AFT

FOLDOUT FRAME

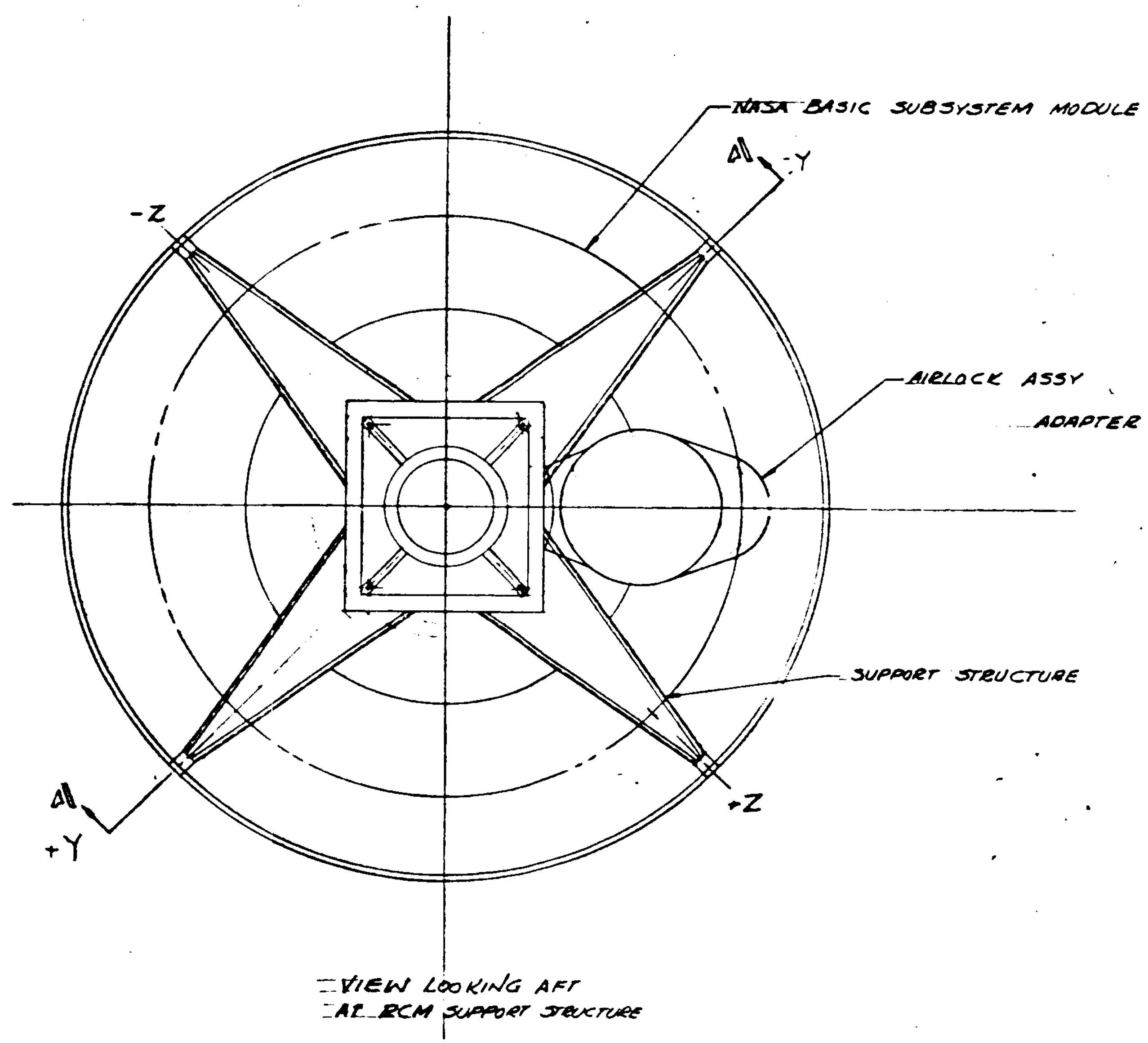
FOLDOUT FRAME 1

FOLDOUT FRAME 2

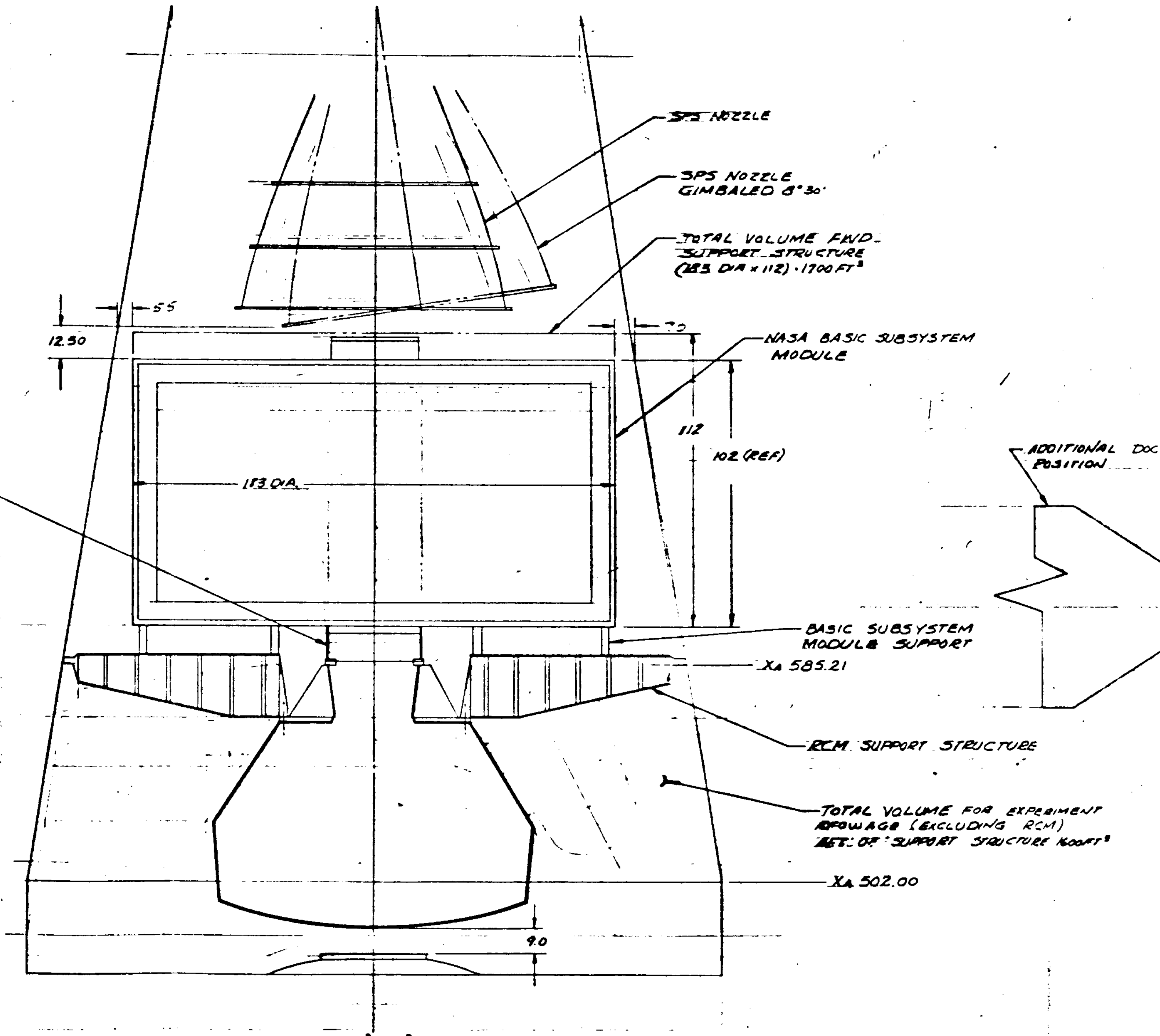


NOTE This drawing is reduced. Scale notations apply to original only.

Figure 94. RCM Laboratory, Manned Orbital Solar Telescope



VIEW LOOKING AFT
AT RCM SUPPORT STRUCTURE



VIEW A-A
BOOST CONFIGURATION

FOLDOUT FRAME 1

FOLDOUT FRAME 2

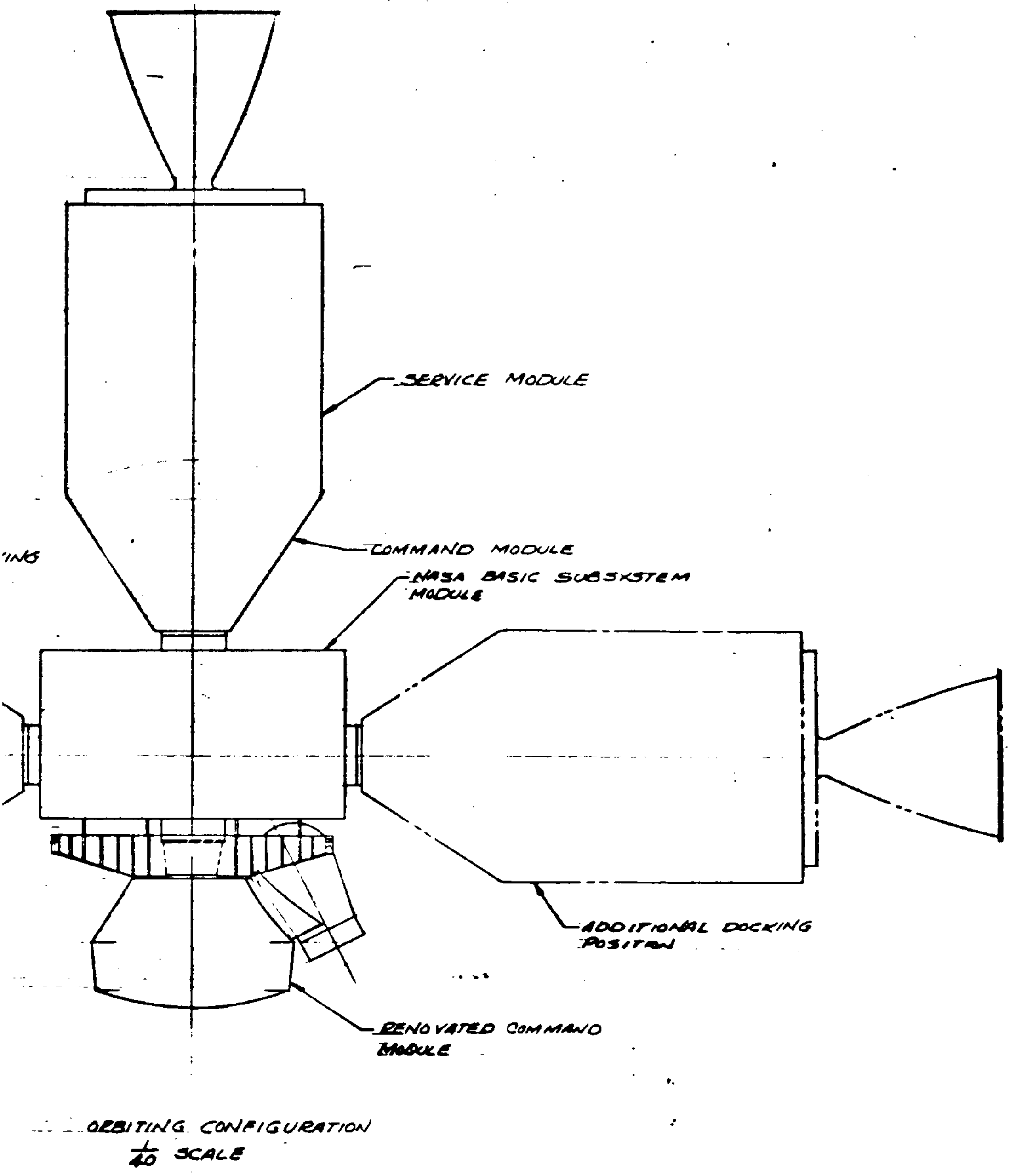


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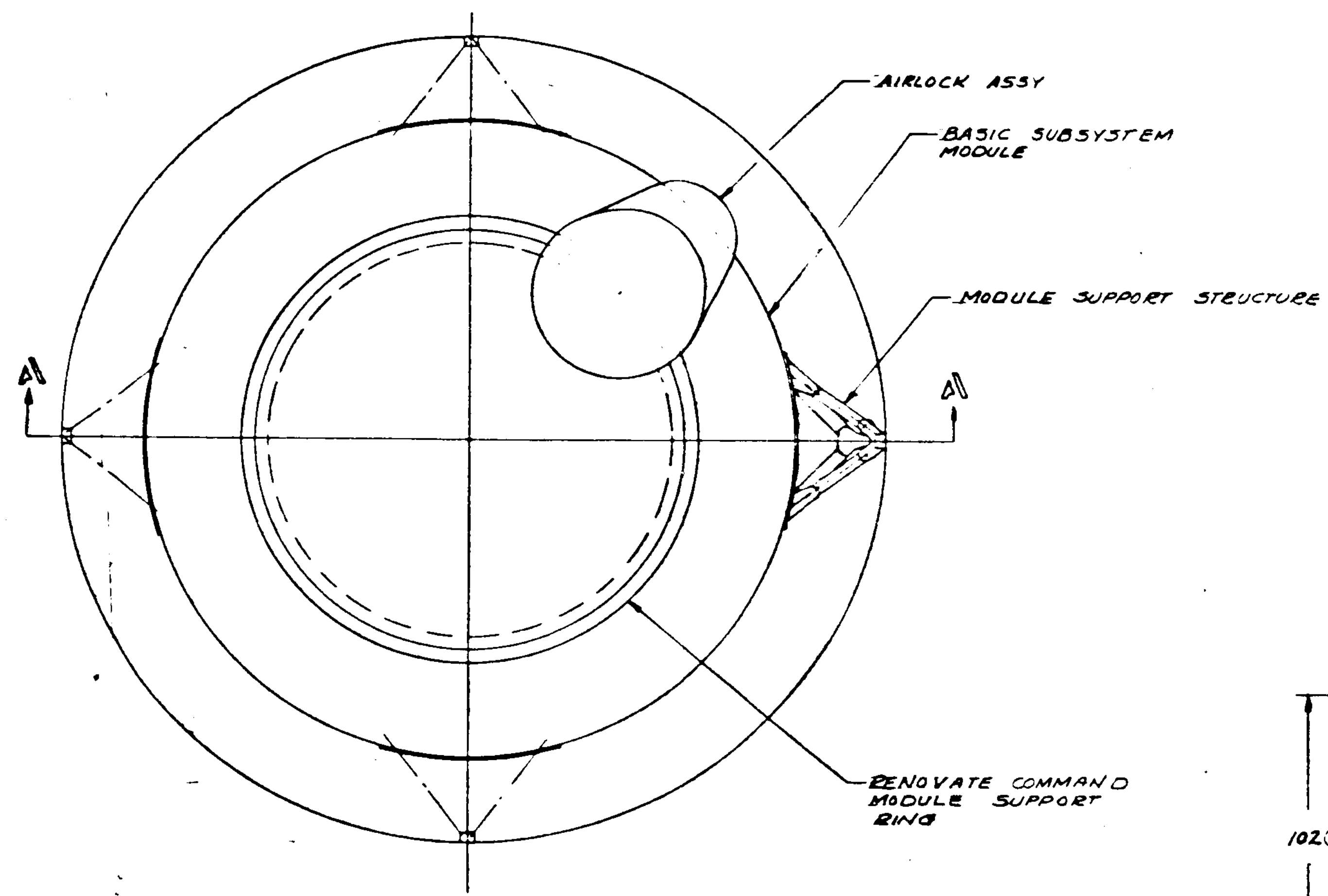
Figure 95. RCM Laboratory With Basic Subsystem Module Located Forward



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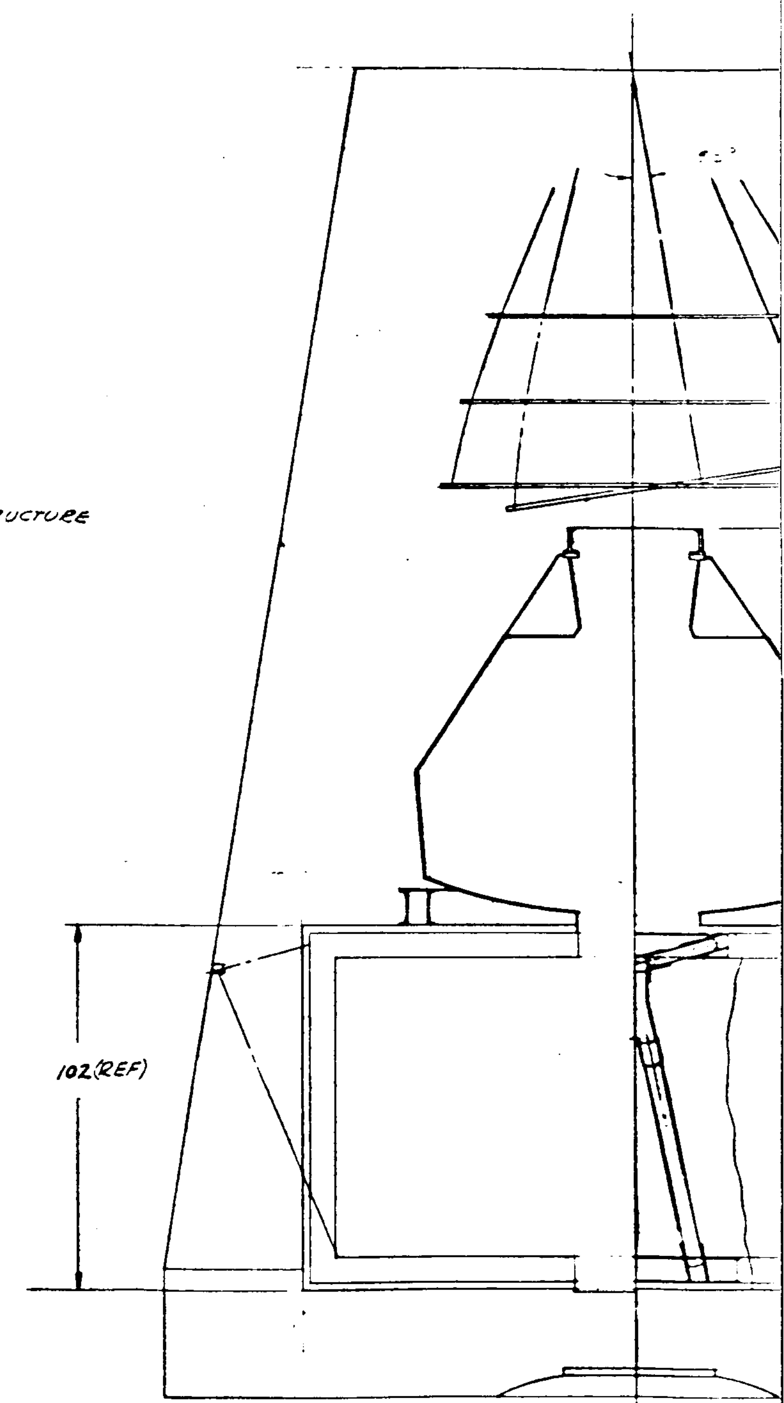
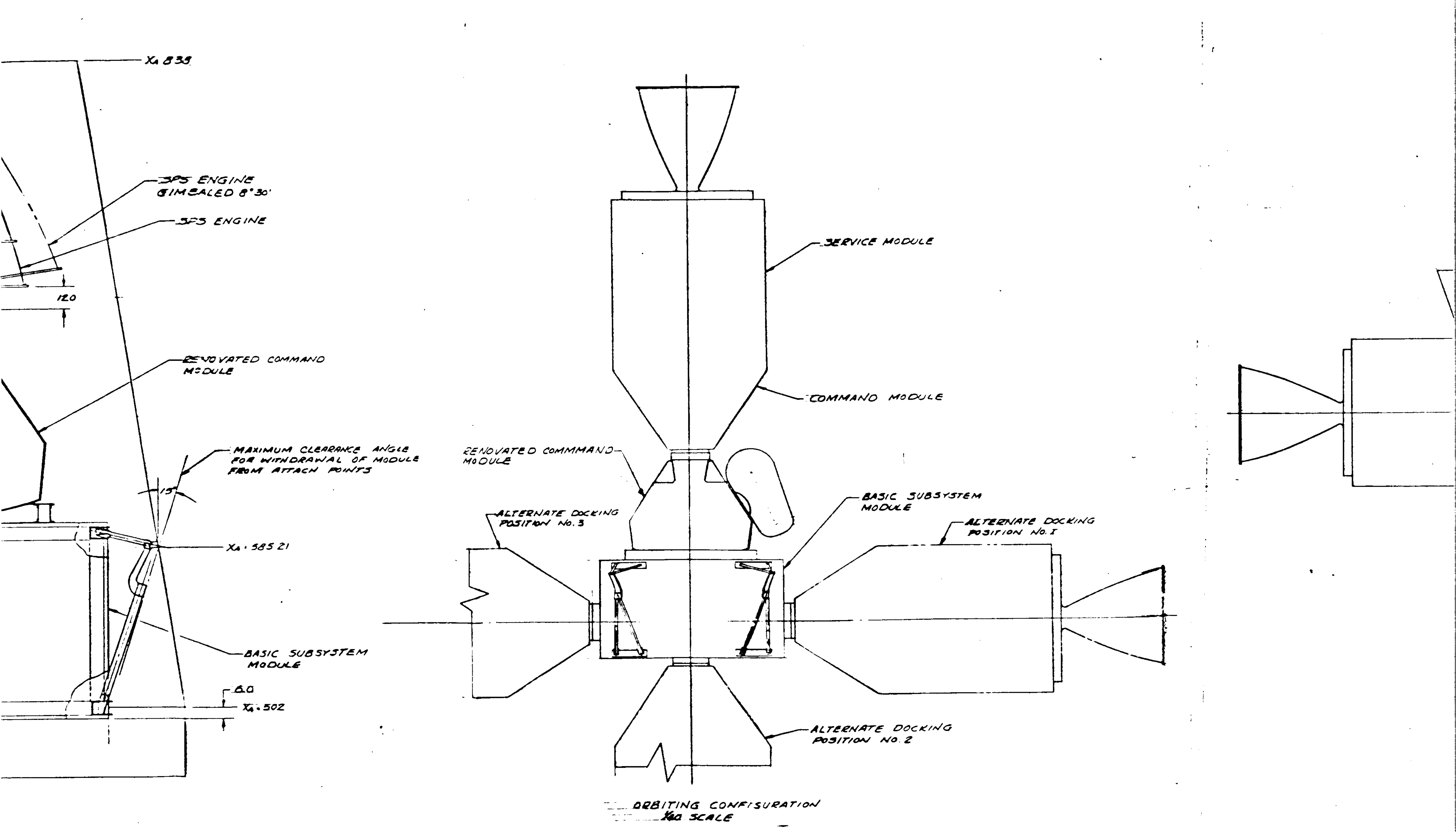


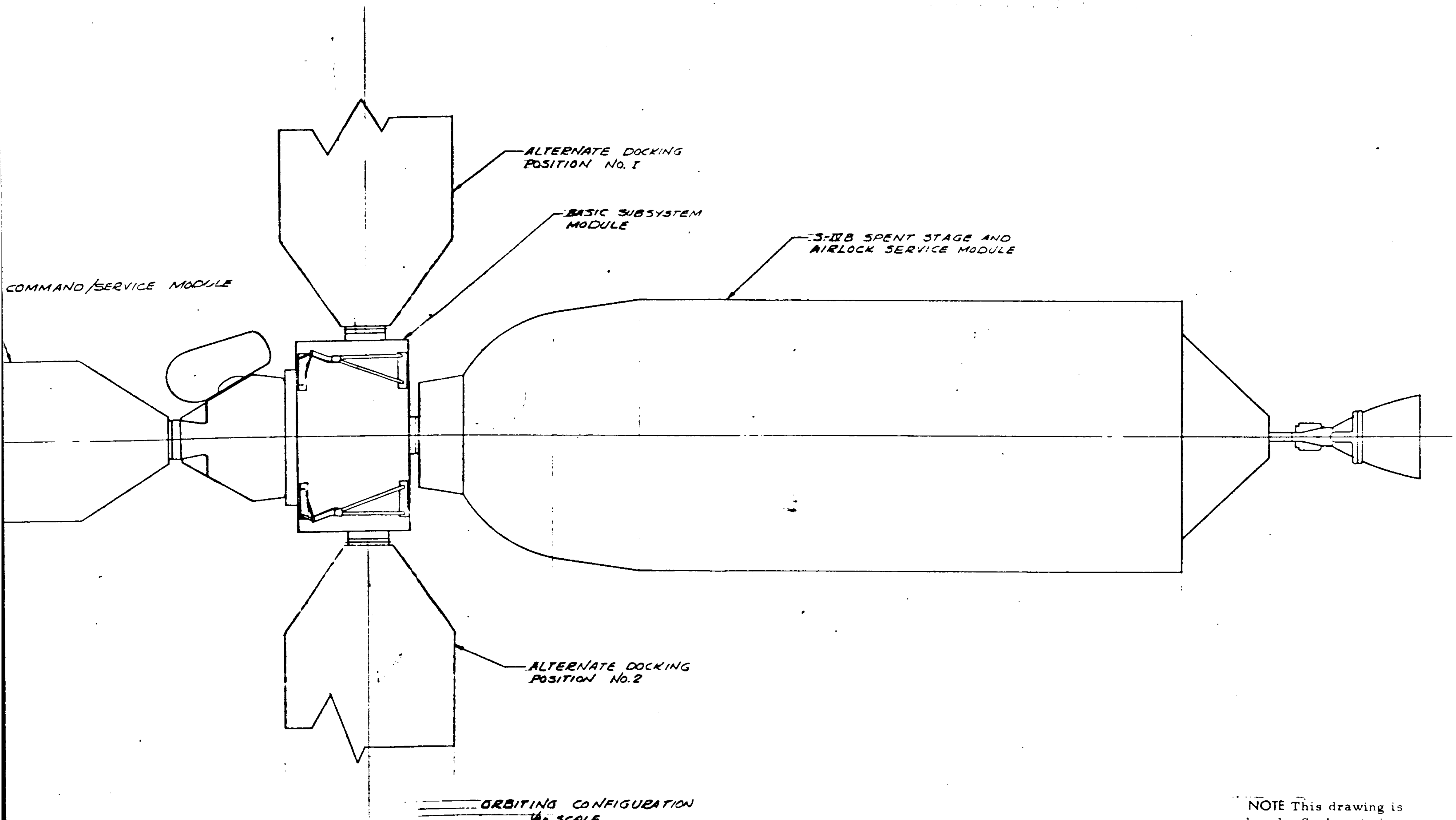
VIEW A-A
BOOST CONFIGURATION

FOLDOUT FRAME 1

FOLDOUT FRAME 2



ORBITING CONFIGURATION
1/40 SCALE



NOTE This drawing is reduced. Scale notations apply to original only.

Figure 96. RCM Laboratory With Basic Subsystem Module Located Aft

The SLA structure has been designed for the external air loads associated with the flight environments of the Saturn V booster; the air loads associated with configurations employing the S-IB booster are of a smaller magnitude. The integrity of the SLA structure depends on some internal support, which is adequately provided by the cruciform.

For preliminary design, a load factor of 45 has been used for the effect of random vibration on all equipment mounted in the laboratory or on the cruciform. This value is based on the Apollo equipment random vibration given in ARM 5, and is considered conservative for a cruciform-mounted laboratory application.

Preliminary Evaluation of Dynamic Load Factors

Figure 97 shows the launch configurations considered.

Configuration 1

The loads used to design the laboratory-supporting beams in the SLA were taken from the LEM/SLA Loads Report MH01-05118-424. If the stiffness and c. g. location of the combination laboratory and supporting structure were made the same as the LM, these loads would be applicable. The laboratory is a smaller payload than the LM, however, and a lighter supporting structure is possible. The resulting reduction in the stiffness of this arrangement will increase the dynamic load factor. For a first attempt at the structural design, an increase of 0.5-g lateral and 1.0-g axial was considered appropriate. The load factors used for the critical conditions were:

Lift-off:

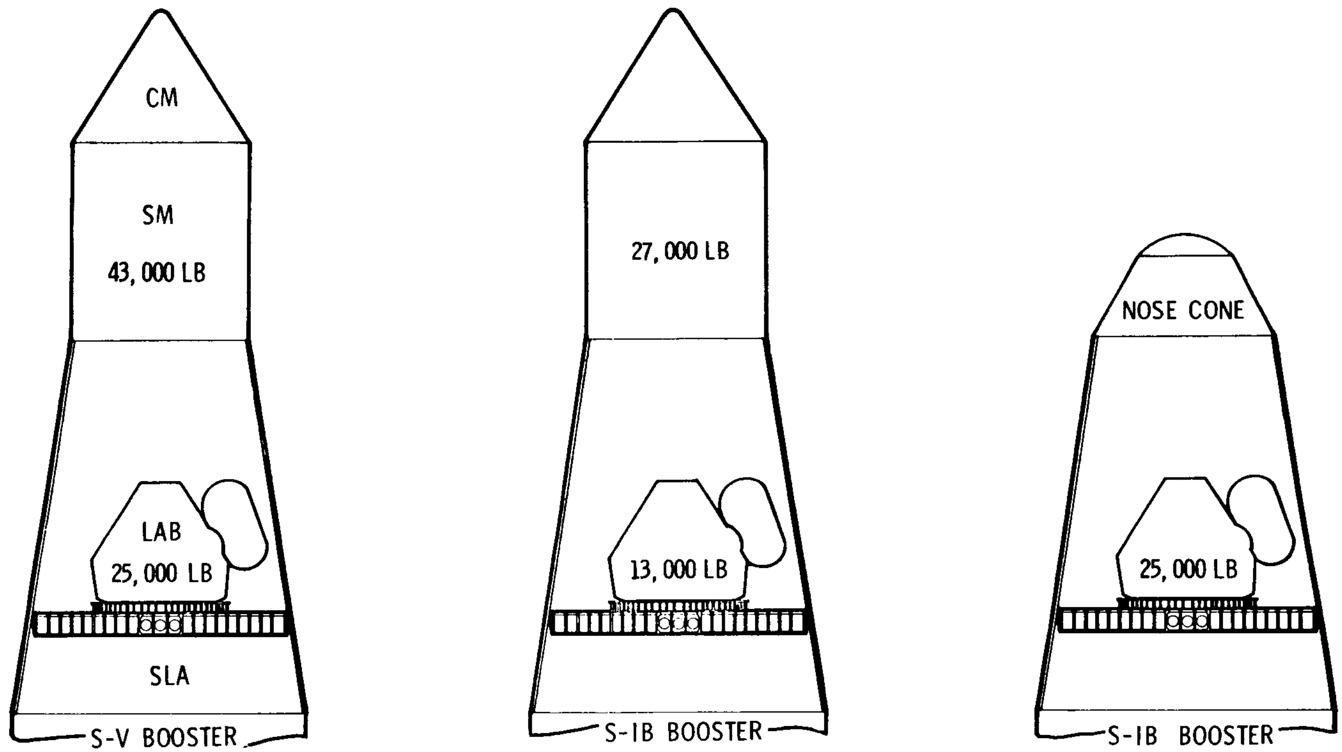
Axial load = (1.6 + 1.000) 1.5 = 3.9 g
Lateral load = (0.65 + 0.5000) 1.5 = 1.73 g

Maximum $q\alpha$:

Axial load = (2.0 + 1.000) 1.5 = 4.5 g
Lateral load = (0.300 + 0.500) 1.5 = 1.20 g

End Boost:

Axial load = (4.9 + 1.000) 1.5 = 8.85 g
Lateral load = (0.100 + 0.500) 1.5 = 0.900 g



CONDITION	LOAD FACTOR X	LOAD FACTOR Y OR Z
LIFT OFF	3.9	1.73
MAX $Q\alpha$	4.5	1.20
END BOOST	8.85	0.90

CONFIGURATION 1

LOAD FACTORS NOT CRITICAL

CONFIGURATION 2

CONDITION	LOAD FACTOR X	LOAD FACTOR Y OR Z
LIFT OFF	4.47	2.4
MAX $Q\alpha$	NOT CRITICAL	
END BOOST	NOT CRITICAL	

CONFIGURATION 3

Figure 97. Launch Configurations, Loads

Separation:

Axial load = (-1.900 - 1.000) 1.5 = -4.350 g
Lateral load = (.18 + 0.500) 1.5 = 1.02 g

Configuration 2

Except for the lift-off condition, all load factors are less than for Configuration 1. Lift-off load factors were:

Axial load = (1.98 + 1.000) 1.5 = 4.47 g
Lateral load = (1.10 + 0.500) 1.5 = 2.400 g

Configuration 3

The load factors for this configuration are not critical.

It should be noted that the aforementioned values are ultimate and are for preliminary design use only; a more complete dynamic loads analysis will be required before the structure can be considered space- or man-rated.

C.G. Location

The LM/SLA attachment points being used for the RCM laboratory mounting structure have an ultimate load capability of 61,800 pounds. To avoid exceeding the capability of these attachments and the SLA structure, the location of the c.g. of the laboratory configuration and experiments must be restricted to the envelope shown in Figure 98.

The critical condition is based on the inertia factors of the Saturn V booster. Load factors considered for end boost were:

Axial load = 8.85 g (ultimate)
Lateral load = 0.90 g (ultimate)

Maximum SLA Structure Capability

With mass c.g. at the center of the supporting structure, weight is 33,800 pounds. Supported laboratory weights of less than 15,000 pounds have no c.g. location limits.

Laboratory/Experiment Support (Cruciform Structure)

Three cruciform designs (Figure 99) have been analyzed. The first can support a payload of 10,000 pounds. This structure, weighing 575 pounds, consists of 20-inch-deep shear web beams with extruded tee caps; web

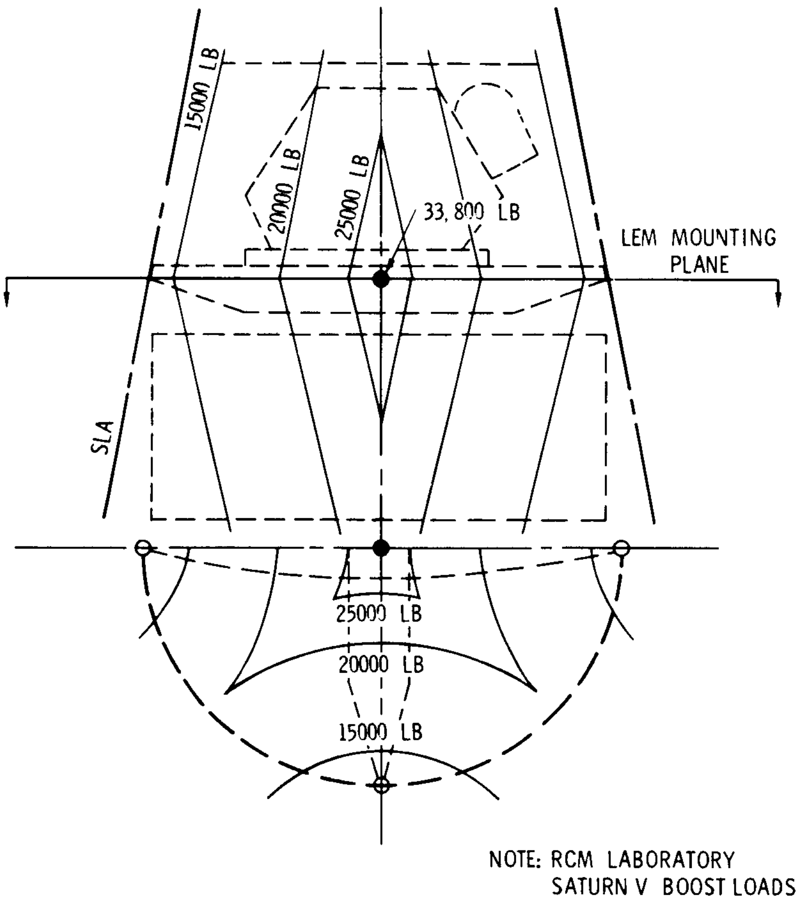
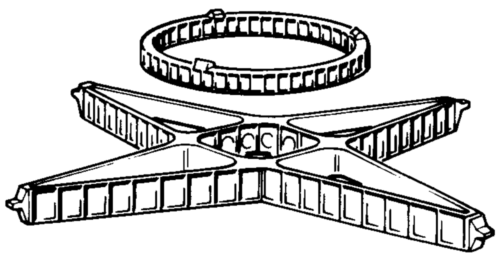


Figure 98. Allowable CG Envelopes as Function of Total Laboratory Weight

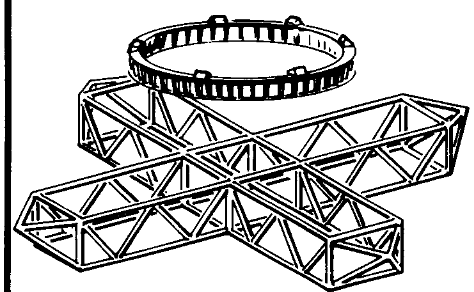
DESIGN NO. 1



WEB-BEAM DESIGN

DESIGN LOAD	-	10,000 LB
MAX LOAD LIMIT	-	15,000 LB
SAFETY FACTOR	-	1.5
TOTAL WEIGHT	-	575 LB

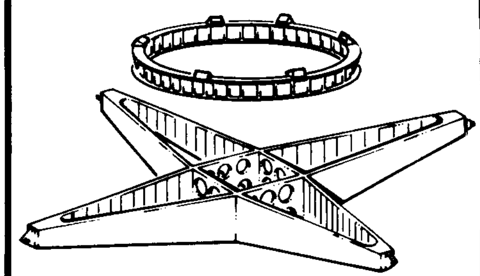
DESIGN NO. 2



TRUSS DESIGN

DESIGN LOAD	-	25,000 LB
MAX LOAD LIMIT	-	33,800 LB
SAFETY FACTOR	-	1.5
TOTAL WEIGHT	-	638 LB

DESIGN NO. 3



WEB-BEAM DESIGN

DESIGN LOAD	-	25,000 LB
MAX LOAD LIMIT	-	33,800 LB
SAFETY FACTOR	-	1.5
TOTAL WEIGHT	-	688 LB

Figure 99. Laboratory/Experiment Support Structure

stiffeners are required in the region of maximum shear. For a payload of 25,000 pounds, two cruciform designs were considered; one was a truss structure, the other a shear web beam 30 inches deep. The truss structure was constructed of tubular members of weldable aluminum alloy to achieve the best weight-stability ratio. It was assumed that the diagonal truss members would be located relative to the supported masses so as to avoid bending of the upper caps. The weight of this truss structure is 638 pounds. The web beam consists of a 30-inch-deep shear web with extruded tee caps; web stiffeners are needed to increase web stability. This arrangement is more adaptable than the truss to variations in the location of the payload mass. This cruciform weighs 688 pounds; however, some weight could be saved by reducing the depth of the beams near the support points.

Laboratory Mounting

To accommodate the alternative laboratory configurations to the best advantage, the mounting of the CM inner structure to the cruciform can be achieved by picking up available attachment points at the forward or aft bulkhead. The four LES attachment points on the forward bulkhead can support the following total loads: axial, 320,000 pounds; shear, 190,000 pounds; moment (x or y axis), $5.8\ (10)^6$ pound-inches.

The three tension ties and six compression pads at the aft bulkhead of the CM inner structure can support the following total loads: axial, 270,000 pounds; shear, 160,000 pounds; moment (x or y axis), $2.5\ (10)^6$ pound-inches.

These structural capabilities are far in excess of the loads that will be imposed by the laboratory on the cruciform structure.

Alternative Airlock Configurations

The significant loading conditions on the airlock structure are internal pressure and handling; both induce low stress levels in the structural materials, and, as a result, design and functional requirements will predicate the material thicknesses.

Seven laboratory configurations, incorporating a number of airlocks, have been considered; airlock configurations are shown in Figure 92.

Concept 1

Concept 1, with an airlock installed in the aft bulkhead along the "X" axis, lends itself more to a newly designed structure than to existing CM's.

Assuming this arrangement is to be installed in an existing structure, the airlock assembly must be bonded in place. Loss of cabin pressure due to leakage at the bulkhead-to-airlock joint is possible. Also, radially disposed floor beams will be required to provide a load path between the airlock and the aft sidewall longerons of the inner structure.

Concept 2

The airlock on Concept 2 is incorporated in the main crew access hatch. This design will require a supporting structure on the airlock to prevent overloading or distorting the hatch mechanism and sealing surfaces.

Concept 3

Concept 3, which incorporates an airlock with a dual docking drogue and the complete assembly mounted on the forward surface of the Block I tunnel, presents no structural problems. The additional loads associated with docking and midcourse maneuvering are small in magnitude and can be accommodated without imposing a structural penalty.

Concept 4

The structural difference between Concepts 4 and 3 is the introduction of a Block II tunnel. To install this arrangement in an existing Block I inner structure will require extensive detail design consideration to maintain the structural integrity and provide a leak-proof cabin.

Concept 5

Structurally, Concept 5 is almost the same as Concept 1; however, the loads on the aft bulkhead will be smaller, because no docking drogue is incorporated in the airlock.

Concept 6

Concept 6 introduces a spherical airlock mounted forward of the forward tunnel. No changes are required on the basic inner structure, and no significant load increase will be experienced. Structurally, this con- configuration is the best.

Concept 7

In Concept 7, the forward tunnel is replaced with an airlock assembly and a docking drogue. As with Concepts 1 and 4, difficulties with the redesign of the forward bulkhead will be encountered. Additional stiffness will be required in the airlock structure to ensure a pressure-tight seal on the large side access hatch.

Test Requirements

The following tests are recommended to qualify the structure as space- or man-rated:

Cabin-Pressure Test

A cabin-pressure test up to 12.9 psi ultimate on all configurations incorporating changes in the basic CM inner structure is included.

Static Test of Cruciform

A static test of the cruciform requires an SLA structure and a CM inner structure, as well as the cruciform structure, to ensure the correct relative stiffness of the assembly. The loads required for this test are 225,000 pounds axial and 22,500 pounds lateral. The lateral load should be applied 67.5 inches above the SLA attachment points to achieve the maximum moment on the cruciform beams and the maximum reaction on the SLA attachments.

Vibration Test

A vibration test requires a cruciform structure, a CM inner structure, all equipment and experimental package attachment structure, and representative masses of all equipment and experimental packages. The test will determine the dynamic response of the cruciform and establish acceleration levels at equipment-attachment points.

Docking Interface

A docking interface test requires a CM inner structure and structure representing the docking interface. The loading condition is a cabin pressure of 7.5 psi and a moment of 174,000 pounds-inches applied at the docking interface. The purpose of the test is to establish the structural integrity of the docked structure during midcourse maneuvers.

Conclusion

The existing SLA structure limits the maximum weight and c.g. location of the laboratory and proposed experimental packages. There are no structural problems associated with the mounting of the laboratory in the SLA. This mounting can consist of either a truss or web beam cruciform structure; of these two structural arrangements, the web beam cruciform structure is more readily adaptable to changes in location of equipment and experimental packages. The loads on the airlock are small and impose no design restrictions.

MASS PROPERTIES

Mass properties of the RCM laboratory and spacecraft were determined throughout the study and monitored closely to assure compatibility with launch vehicle capability.

Designs were reviewed as they progressed, and weight tradeoffs were performed where alternative approaches developed. Mass properties data were generated by utilizing the existing Apollo recording and reporting computer program. The data provided in the master tape for specific Apollo end-item modules were altered by modifications or changes to the basic configuration, as dictated by the RCM laboratory or spacecraft subsystem definition.

Figure 100 shows how the mass properties were generated during the study. The three basic module mass properties were derived by utilizing the Apollo computer program. For specific missions, the ΔW for the mission-dependent subsystems is determined by iterating with the selected reference mission experiments requirements. The micrometeoroid-thermal shield weight is combined with the mission experiments, ΔW for the various subsystems and the basic laboratory weight, to derive the total RCM laboratory mass properties. Similarly, the life support weight is combined with the RCM spacecraft subsystems ΔW and the basic RCM spacecraft weight to derive the total RCM spacecraft mass properties.

The SM RCS and EPS cryogenic system weight is then computed. Mass properties for the total RCM laboratory and spacecraft, the basic SM, the SM RCS, and cryogenic system is combined with approximate values for propulsion tanks, pressurant, pressurant tanks, and SM structure. These accumulated masses and ΔV requirements for a particular mission are utilized to compute the approximate SPS propellant required. By recycling and iterating the propulsion system and SM structure masses with propellant required, the final CSM and laboratory configuration weight is determined. The summation of the final CSM, laboratory, and adapter weights represents the mission spacecraft injected weight, which then can be compared with the launch vehicle capability. The weight allowance of the experiments is derived by deduction, based on an assumed launch vehicle earth-orbital capability. A typical RCM laboratory weight breakdown to the level of major assembly and subsystem is presented, and corresponds to the baseline RCM configurations selected for this study.

Tables 48 through 56 present RCML configurations evolved from the structures and systems applicable to the Block II CSM configuration. The systems and their service fluids reflect this capability. Life support requirements are defined for a thirty-day mission.

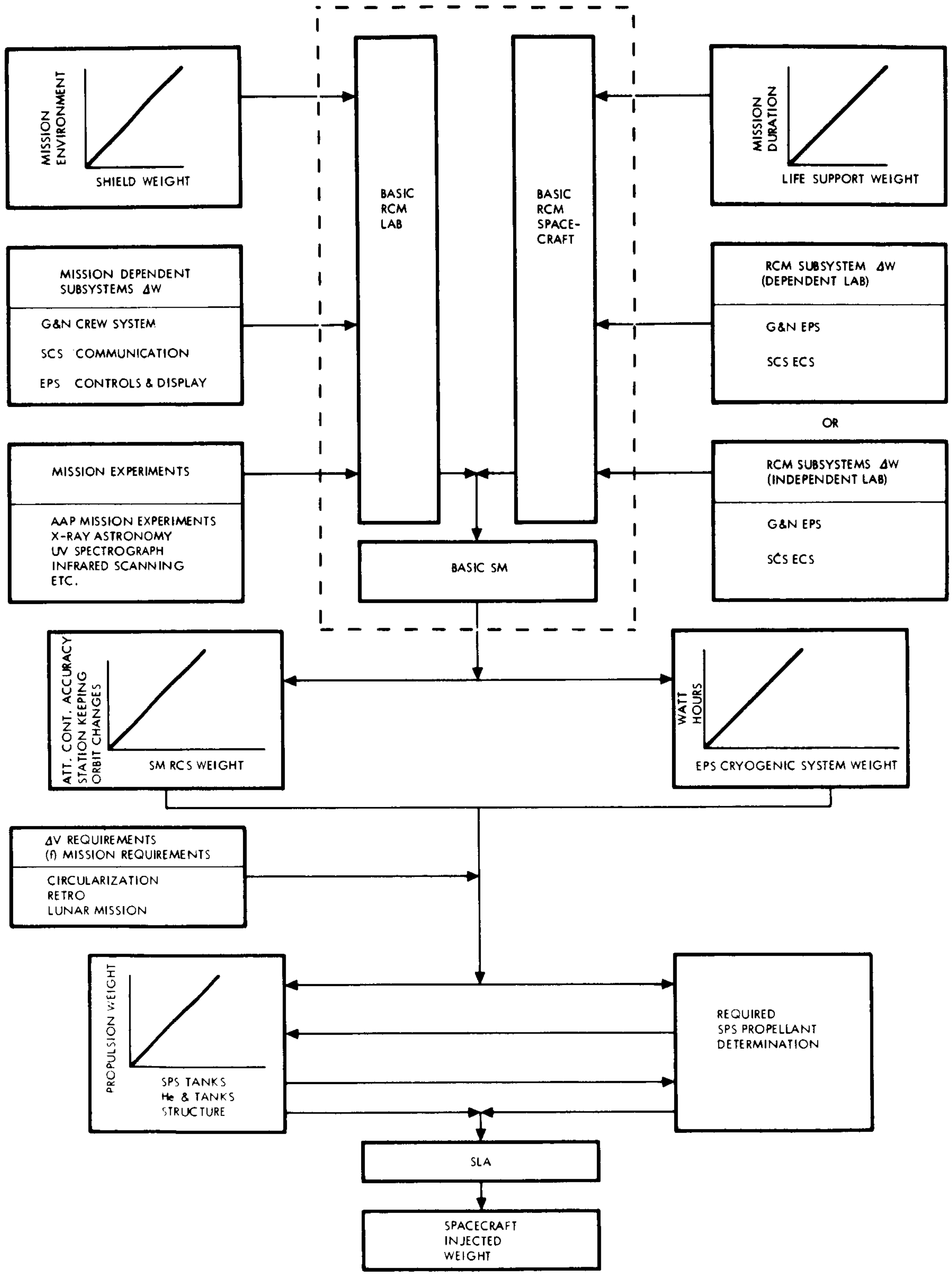


Figure 100. Spacecraft-Payload Mass Properties Synthesis

Table 48. RCML Configurations

Item	Fully Dependent Laboratory*			Fully Independent Laboratory*	
RCM baseline laboratory	2,121.0				2,121.0
Systems, platform, etc.	1,885.0				
Systems, platform, etc.					7,459.0
Dry weight	4,006.0				9,580.0**
Usable fluids					1,861.0
	4,006.0				11,441.0
	X cg	624.5		X cg	612.1
	Y cg	-0.1		Y cg	-0.3
	Z cg	-0.4		Z cg	3.2
	MI Ixx	2572	slug ft^2	MI Ixx	4211
	MI Iyy	1885	slug ft^2	MI Iyy	3444
	MI Izz	2077	slug ft^2	MI Izz	3854
	PI Ixy	36.6	slug ft^2	PI Ixy	-20.6
	PI Ixz	40.8	slug ft^2	PI Ixz	-228.0
	PI Iyz	-23.8	slug ft^2	PI Iyz	8.7

*These configurations do not contain the CSM stack.
**Center of gravity and inertias reflect this weight.

Table 49. Block II Renovated Command Module CSM Configuration for Fully Dependent or Fully Independent Laboratory

Item	Weight	X	Y	Z
Weight Empty				
Structure	5693.0	1041.8	-0.2	2.1
Stabilization and control	191.2	1039.0	-16.2	28.6
Guidance and navigation	379.0	1053.7	-1.9	36.7
Crew systems	83.8	1044.8	-3.6	-10.7
Environmental control	430.1	1035.4	-32.7	7.0
Earth landing system	631.4	1090.3	0.3	-0.1
Instrumentation	17.4	1037.4	16.5	5.1
Electrical power	1427.5	1031.6	12.3	17.0
Reaction control	298.4	1031.9	-1.7	-2.4
Communications	301.4	1034.8	12.1	35.3
Controls and displays	381.0	1061.4	-0.2	-13.4
Ballast	122.0	1017.0	0.0	-67.1
Expendables and Residuals				
Crew systems	907.7	1043.4	-0.2	-11.7
Reaction control	270.0	1022.6	-5.6	56.9
Environmental control	99.8	1018.5	-15.4	18.2
Total	11233.7			

MASS PROPERTY CHARACTERISTICS

Item	Moment of Inertia	Product of Inertia
Weight = 11233.7 X cg = 1042.8 Y cg = -0.1 Z cg = 5.4	Ixx = 5480.7 slug ft^2 Iyy = 5117.7 slug ft^2 Izz = 4636.9 slug ft^2	Ixy = 2.0 slug ft^2 Ixz = -307.0 slug ft^2 Iyz = 29.0 slug ft^2

NOTE:
Eighty pounds of scientific equipment and 408 pounds of aft heatshield ablator were removed from the August 1, 1966 CM 103 status.
Eighteen pounds of controls and displays were added to accommodate the RCS revision.

Table 50. Block II Service Module CSM Configuration for Fully Dependent or Fully Independent Laboratory

Item	Weight	X	Y	Z
Weight Empty				
Structure	4547.1	921.5	-0.7	3.3
Environmental control	167.0	905.5	3.8	-4.1
Instrumentation	50.4	935.5	7.2	3.8
Electrical Power	1743.3	953.2	-24.7	32.4
Main propulsion	1219.6	842.4	-1.1	2.3
Reaction control	426.2	950.7	0.0	0.0
Communication	134.6	855.6	-12.4	23.3
Residuals				
Reaction control	152.0	943.3	0.0	0.0
Electrical power	11.8	911.2	-27.6	40.7
Environmental control	3.2	920.8	-24.1	40.7
Main propulsion	833.8	856.6	9.8	2.1
Expendables				
Reaction control	1224.0	941.1	0.0	0.0
Electrical power	491.2	915.7	-26.0	40.8
Environmental control	146.8	920.8	-24.1	40.7
Total	11151.0			

MASS PROPERTY CHARACTERISTICS

Item	Moment of Inertia	Product of Inertia
Weight = 11151.0		
X cg = 915.6	Ixx = 7541.0 slug ft^2	Ixy = 292.0 slug ft^2
Y cg = -4.9	Iyy = 12342.0 slug ft^2	Ixz = 404.0 slug ft^2
Z cg = 9.4	Izz = 12051.0 slug ft^2	Iyz = 222.0 slug ft^2

Table 51. Detail Weight Statement RCM Baseline Laboratory Applicable to Fully Dependent or Fully Independent Laboratory

BASIC BODY STRUCTURE		
Forward Section		281.0
Honeycomb panels	73.3	
Frames and rings	85.9	
Windows, hatches, etc.	17.5	
Mechanisms	7.8	
Fitting and attachment parts	96.5	
Center Section		757.9
Honeycomb panels	308.2	
Longerons	126.2	
Frames and rings	143.9	
Windows, hatches, etc.	74.1	
Mechanisms	8.9	
Body to heat shield attachment	44.7	
Fittings and attachment parts	51.9	
Aft Section		244.8
Honeycomb panels	148.8	
Frames and rings	89.7	
Body to heat shield attachment	4.4	
Fittings and attachment parts	1.9	
SECONDARY STRUCTURE		
Right-hand Equipment Bay		67.0
Left-hand Equipment Compartment		25.6
Lower Equipment Bay		53.8
Forward Compartment Area		5.6
Heat Shield—Center Section		95.2
Fitting and attachment	2.5	
Windows and hatch covers	86.5	
Umbilical provisions	6.2	
Electrical Provisions		2.8
DOCKING AND DROGUE INSTALLATION		
Docking Latch, Cable, etc.		49.2
Drogue Installation		53.8
Fitting—adapter ring	32.0	
Drogue assembly	17.2	
Support structure	4.6	
Airlock Installation		172.9
Airlock shell	50.0	
Adapter ring	16.0	

Table 51. Detail Weight Statement RCM Baseline Laboratory Applicable to Fully Dependent or Fully Independent Laboratory (Cont)

Airlock support structure	21.8	
Outside hatch	35.3	
Adapter clamp	16.0	
Inside hatch	33.8	
METEOROID AND THERMAL PROTECTION		
Baseline Laboratory		239.0
Meteoroid bumper	101.0	
Insulation	53.0	
Clips and doublers	85.0	
Airlock Installation		72.4
Meteoroid bumper	28.0	
Insulation	14.4	
Clips and doublers	30.0	
Total		2121.0

Table 52. Summary Weight Statement for Baseline RCM Laboratory

Basic body structure		1283.7
Secondary structure		250.0
Docking, drogue, and miscellaneous structure		103.0
Airlock installation		172.9
Meteoroid and thermal protection		311.4
Baseline laboratory	239.0	
Airlock	72.4	
Total		2121.0

MASS PROPERTY CHARACTERISTICS

Item	Moment of Inertia	Product of Inertia
Weight = 2121.0	I_{xx} = 992.5 slug ft^2	I_{xy} = 4.5 slug ft^2
X cg = 636.5	I_{yy} = 959.3 slug ft2	I_{xz} = -75.6 slug ft^2
Y cg = 0.7	I_{zz} = 767.3 slug ft^2	I_{yz} = 6.9 slug ft^2
Z cg = -8.9		

Table 53. Systems and Structure Added to Baseline RCM Laboratory to Create a Fully Dependent Laboratory

Systems and Structure to RCM Laboratory			893.7
Instrumentation and wiring		50.0	
Communications and data		181.5	
EPS provisions and controls and displays		316.2	
Environmental control system		294.0	
ECS components	219.0		
Radiators	75.0		
Additional supporting structure		52.0	
Life Support Requirements for 30 Days			416.4
Food hygiene and waste management		171.0	
LiOH and containers		225.4	
Required storage racks		20.0	
Laboratory/experiments mount			575.0
Total			1885.1

MASS PROPERTY CHARACTERISTICS

Item	Moment of Inertia	Product of Inertia
Weight = 1885.1	Ixx = 1507.4 slug ft^2	Ixy = 39.9 slug ft^2
X cg = 611.1	Iyy = 706.8 slug ft^2	Ixz = 13.2 slug ft^2
Y cg = -1.0	Izz = 1163.1 slug ft^2	Iyz = -30.9 slug ft^2
Z cg = 9.2		

Table 54. Systems and Structure Added to Baseline RCM Laboratory to Create a Fully Independent Laboratory

Systems and structure to RCM laboratory				2938.0
Instrumentation and wiring			50.0	
Stabilization and control (Block II)			191.2	
Communication and data (Block II)			307.4	
Intercomm and hardline			10.0	
ECS (no radiators) (Block II)			415.4	
EPS (no radiators) (Block II)			1059.1	
Controls and displays (most of Block II)			299.0	
Structural mounting for systems (similar to Block II)			189.5	
Life support requirement for 30-day mission			416.4	
Food, hygiene, and waste management		171.0		
LiOh + containers		225.4		
Required storage racks		20.0		
Systems and structure external to RCM laboratory				4521.0
Laboratory/experiment mount			680.0	
System support platform on laboratory/mount			324.0	
Tube assembly for radiators support			91.0	
Systems installations (SM type Block II)			3426.0	
RCS (including thermal shield)		959.0		
EPS (including thermal shield + radiators)		1992.0		
ECS (including radiators)		178.0		
Plumbing and hook-up contingencies		130.0		
Residual fluid and gases		167.0		
RCS	152.0			
EPS	11.8			
ECS	3.2			
Total				7459.0

MASS PROPERTY CHARACTERISTICS

Item	Moment of Inertia	Product of Inertia
Weight - 7459.0 X cg - 605.1 Y cg - -0.6 Z cg - 6.6 Usable Fluids RCS - 1224.0 EPS - 491.0 ECS - 146.0 1861.0	Ixx = 3131.1 slug ft^2 Iyy = 2530.8 slug ft^2 Izz = 2719.9 slug ft^2	Ixy = 2.7 slug ft^2 Ixz = 29.3 slug ft^2 Iyz = 1.7 slug ft^2

Table 55. Detail Weight Statements for Several Systems and Structure Added to the Baseline RCM Laboratory to Create a Fully Independent Laboratory

Systems and Structure Mounted External to the Baseline RCM—Platform Mounted			
Reaction Control System			959.0
RCS panels		234.0	
RCS fuel system support		142.0	
Oxidizer system support		16.0	
Pressure system support		12.0	
Engine support		46.0	
Fuel system		104.5	
Tanks and expulsion	56.4		
Plumbing and fittings	12.2		
Valves and regulators	19.6		
Temperature control	.5		
Supports	15.8		
Oxidizer System		112.3	
Tanks and expulsion	62.4		
Plumbing and fittings	14.0		
Valves and regulators	19.6		
Temperature control	0.5		
Supports	15.8		
Pressurization System		115.0	
Tanks	46.0		
Plumbing and fittings	16.0		
Valves and regulators	50.0		
Supports	3.0		
Engine System		85.2	
Engines	83.8		
Temperature control	0.4		
Supports	1.0		
Tank Thermal Protection		92.0	
Tank covers	92.0		
Electrical Power System			1992.0
Secondary structure		63.4	
H_2 tank support and shelves	17.4		
O_2 tank support and shelves	46.0		
Tank thermal protection		141.4	
Tank covering	141.4		

Table 55. Detail Weight Statements for Several Systems and Structure Added to the Baseline RCM Laboratory to Create a Fully Independent Laboratory

Electrical Power Equipment			1334.4	
H_2 system		219.8		
Subcontractor items	194.8			
Plumbing	7.4			
Valves	11.6			
Supports and shelves	6.0			
O_2 System		198.9		
Subcontractor items	166.4			
Plumbing	6.4			
Valves	18.8			
Supports and shelves	7.3			
Fuel Cell System		915.7		
Subcontractor items	723.0			
Plumbing	24.3			
Supports	8.0			
Water glycol	21.7			
Space radiators	71.2			
KOH	8.4			
Power distribution box	35.3			
Control panel	5.4			
Terminal distribution panel	18.4			
Electrical Installations			452.8	
Electrical harness		418.0		
Support and installation provision		34.8		
Environmental Control System				178.0
Water glycol circuit			93.8	
Valves		8.4		
Space radiators		85.4		
O_2 supply system			3.9	
Plumbing		3.9		
Water supply system			2.3	
Plumbing		1.3		
Supports and attachment parts		1.0		
Heat transfer system			78.0	
Miscellaneous components		24.0		
Heat exchanger		3.0		
Plumbing and fittings		39.0		
Heat transfer fluid		12.0		

Table 56. Detail Weight Statements for Several Systems and Structure Added to the Baseline RCM Laboratory to Create a Fully Independent Laboratory

SYSTEMS AND STRUCTURE TO RCM		
Instrumentation and Wiring		50.0
Instrumentation	15.0	
Panels and mounts	6.0	
Wiring	29.0	
Stabilization and Control		191.2
Gyro package	44.7	
Control electronics	16.2	
Servo amplifier	12.4	
Display electronics	24.7	
Solenoid driver amplifier	19.8	
Gyro display coupler	26.0	
Gyro package mounting plate	4.6	
Displays and controls	42.8	
Communications and Data		307.4
Electrical provisions	3.8	
Unified S-band	31.3	
S-band power amplifier	31.6	
Signal conditioner	34.3	
Recorder	39.6	
Audio center	7.5	
Premodulator processor	11.4	
Central timer	9.2	
Up-data link	18.0	
HF transceiver	6.2	
VHF-AM transmitter-receiver	12.7	
Recovery beacon	2.0	
Triplexer	1.7	
PCM	42.0	
HF/VHF recovery antenna/transmitter lines	11.3	
2KMC hi-gain antenna and transmitter lines	2.3	
VHF OMNI antenna transmitter lines	1.9	
TV equipment	13.4	
2KMC OMNI antenna and transmitter lines	22.6	
Video coaxial + connectors	1.6	
Supports	0.8	
Data display panel-instrumentation		2.2

Table 56. Detail Weight Statements for Several Systems and Structure Added to the Baseline RCM Laboratory to Create a Fully Independent Laboratory (Cont)

Environmental Control System		
Pressure suit circuit		26.1
Controls	0.2	
Suit flow limiter	0.1	
CO_2 sensor	2.7	
Ducting, fittings, etc.	18.0	
Free condensate control	5.1	
Water Glycol Circuit		87.2
Controls	0.2	
Plumbing	16.7	
Water glycol	23.1	
Coldplates	43.4	
Supports, etc.	3.8	
Pressure and Temperature Control		5.8
Valves	1.1	
Ducting	1.8	
Plumbing	0.9	
Supports, etc.	2.0	
Oxygen Supply System		16.3
Oxygen surge tank	8.9	
Plumbing	5.3	
Supports, etc.	2.1	
Water Supply System		14.5
Plumbing	10.6	
Supports and attachment parts	2.8	
Fittings	0.1	
Water metering	1.0	
Common Items		3.6
ECS	0.4	
Cabin pressurization system	3.2	
Waste Management		17.3
H_2O tank installation	2.3	
Valves	1.2	
Lines and fittings	7.5	
Vacuum cleaner head and hose	2.3	
Plumbing installation	4.0	
AiResearch Components		244.6
Environmental control unit	148.3	
Oxygen control panel	8.7	
Water control panel	2.5	
Miscellaneous components	85.1	

Table 56. Detail Weight Statements for Several Systems and Structure Added to the Baseline RCM Laboratory to Create a Fully Independent Laboratory (Cont)

Electrical Power System			1059.1
Electrical power equipment		297.5	
Batteries-energy source	56.1		
Batteries-post landing	28.3		
Batteries-pyrotechnic	5.6		
Plumbing	1.5		
Supports	0.9		
Inverters	144.0		
Battery charger	3.8		
DC power panel	7.1		
AC power box	10.6		
Battery circuit breaker panel	2.3		
Electrical circuit breaker panel	5.2		
Fuse box	1.9		
Miscellaneous requirements	7.7		
Lighting	0.6		
Phase correction capacitor	3.1		
Terminal distribution panel	16.8		
Supports and installation provisions	2.0		
Electrical Installation		646.6	
Electrical harness	509.7		
Lower equipment bay motor switch	5.6		
Circuit interrupter	15.5		
RCS controller	43.5		
Humidity fix	1.0		
Supports and installation provisions	63.2		
Junction box assembly	4.6		
Circuit utilization box	3.5		
Electrical Power System		115.0	
Master event sequence control	90.6		
Pyro continuity box	10.8		
Supports and hardware	3.1		
Internal lighting	10.5		
Controls and Displays			299.0
Main display panel-control station		28.6	
Mode select	2.1		
Event timer	1.1		
Docking provisions	0.1		
Crew safety	2.5		
Mounting panels	22.8		

Table 56. Detail Weight Statements for Several Systems and Structure Added to the Baseline RCM Laboratory to Create a Fully Independent Laboratory (Cont)

Main display panel-center station		74.4
Reaction control	10.1	
GMT readout	2.6	
ECS gauges and control	9.6	
Hi-gain antenna control	4.2	
Cryogenic	5.2	
Caution and warning	4.6	
Mounting panels	37.4	
Switches-miscellaneous	0.7	
Main Display Panel-Management Station		42.9
Communications control	8.1	
Master caution	0.3	
Power distribution	2.5	
Fuel cell controls	9.0	
Mounting panels	23.0	
Main Display Panel-Right-Hand Console		23.3
Bus switches	3.1	
Audio panel	2.7	
Circuit breakers	9.5	
Mounting panels	8.0	
Main Display, Panel-Left-Hand Console		21.1
Lighting controls	3.2	
SCS power controls	5.5	
Circuit breakers	5.5	
Mounting panels	6.9	
Remote Equipment-Lower Equipment Bay		21.6
Lighting controls	3.2	
Transponder controls	0.3	
Timers	4.7	
Audio controls	1.5	
IFTS	2.4	
RCS	0.5	
Panels	9.0	
Remote Equipment-Right Hand Forward Equipment Bay		14.2
Circuit breakers-panel 11	3.7	
Circuit breakers-panel 13	4.9	
Panels	5.6	

Table 56. Detail Weight Statements for Several Systems and Structure Added to the Baseline RCM Laboratory to Create a Fully Independent Laboratory (Cont)

Remote Equipment		13.2	
Detectors	13.2		
Electrical Provisions		58.5	
Communications data distribution panel	1.5		
SCS power junction box	0.9		
Panel wiring and connectors	56.1		
Lighting		1.2	
Window shades	1.2		
Structural Mounting for Systems			189.5
Secondary structure		189.5	
Right-hand equipment bay	67.4		
Left-hand equipment bay	28.2		
Main display panel	34.5		
Lower equipment bay	53.8		
Forward compartment area	5.6		

SYSTEM DESIGN

The design of the renovated CM laboratory was based on a logical development of a previous concept presented in a NAA RCM Study proposal, SID 66-1135, which was considered a basepoint for the subject study. The design approach was considered in two basic tasks: (1) renovation and utilization of Apollo CM subsystems, which are detailed elsewhere in this report, and (2) utilization and integration of those systems into a renovated CM as a reusable spacecraft and as a laboratory. This section of the report will deal only with the RCS laboratory, because it required considerable design study to identify the arrangement of equipment and systems to provide a usable laboratory. The RCM spacecraft is not included in this section, inasmuch as the design is identical to the existing Apollo.

RCM Laboratory Design

The laboratory was considered in three incremental steps of dependency on additional vehicles for operation. These three laboratories are identified as (1) the basic laboratory, consisting only of the pressure shell, laboratory mount, airlock, and minimum, austere systems to operate the airlock and to light the interior of the laboratory; (2) the minimum dependent laboratory, in which some subsystems are added to provide additional capability of the CSM to conduct experiments; and (3) the independent laboratory, with a full complement of systems to permit unmanned missions for an extended time with the capability of resupply by docking to logistics vehicles.

The objective of this design study was to establish a concept where the basic laboratory could be defined in detail, then, using that as a basepoint, to establish the additional requirements for the minimum dependent and the fully independent laboratories.

The basic structural concept remains unchanged for all laboratory arrangements. There are three basic parts to the total structure: (1) the CM inner shell and its thermal protection and micrometeoroid shielding, (2) the airlock with thermal/meteoroid protection, and (3) the laboratory mount.

The basic laboratory and the minimum dependent laboratory are shown with a single concept for each, since they represent the minimal-type laboratory and would have few or no alternatives. The independent laboratory, however, is shown in three concepts: (1) Block I CM and systems with EPS, ECS, and RCS systems using Block II SM systems with no change; (2) Block II CM and systems with EPS, ECS, and RCS systems using Block II SM systems with no change; and (3) Block I CM and systems using modified Block II systems for ECS, EPS, and RCS for improved laboratory arrangement.

The laboratory designs are based on the following ground rules:

1. The inner structure of a recovered Apollo CM (Block I or II) to be renovated and used as the pressurized crew compartment of the laboratory (Figure 101)

2. A one-man airlock to be located in place of the side crew hatch on the CM

3. The completed laboratory to be capable of docking with an Apollo CSM, with crew transfer to be accomplished in the same manner as with the LM

4. The laboratory mount to be a cruciform structure to mate with the LM/SLA fittings and to support the laboratory

5. Existing systems and structures from the renovated CM to be used where possible

6. Block II SM systems to be used for fuel cells, radiators, and RCS

Basic Laboratory

The basic configuration for a renovated Apollo CM is shown in Figure 102. This concept employs a renovated Block I CM inner structure that has been modified by removal of the forward tunnel and installation of a new Block II forward tunnel assembly (Section F-F). The crew side access hatch has been replaced with a new hatch assembly containing a Block II circular hatch and mounting a one-man airlock assembly on the outer surface. The airlock assembly details are shown in Figure 103. Docking this laboratory with a manned Apollo Block II CSM in flight is provided by the use of a new 7.25-inch-long adapter on the end of the forward tunnel. This adapter contains a removable LM docking drogue assembly to receive the probe of the CSM. In flight, the Apollo CSM will dock to the RCM laboratory as it would with a LM vehicle, and crew transfer will be similar.

The basic RCM laboratory structure includes a stand-off covering (Sections H, I, J, K) composed of insulation with a 0.016 aluminum micrometeoroid shield on the outside and fiberglass structural supports that hold this covering in proper relation to the outer surfaces of the laboratory. The airlock is protected in the same manner. The laboratory is attached by three tension bolts and mounted on six compression pads to the support structure beam (Figure 104).

All equipment was removed from the interior of the CM during renovation, leaving only the secondary structures in the left, lower, and right equipment bays. The main display panels are taken out entirely.

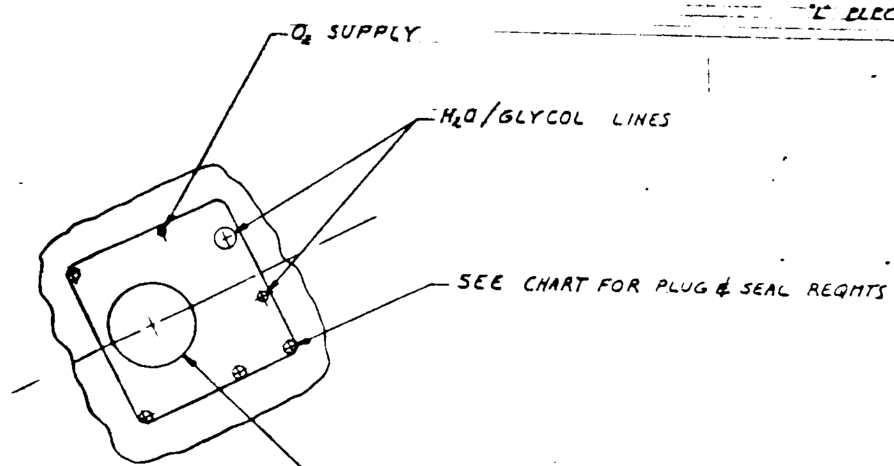
Existing floodlights are relocated on the forward bulkhead to light the interior of the RCM laboratory. A small control panel for activating the airlock is located in the left equipment bay. Electricity is provided to the interior of the RCM laboratory through quick-connect umbilicals in the docking tunnel. Life support environment is provided from a docked Apollo CSM, and the CM postlanding ventilation blower is used to blow oxygen from the Apollo CM cabin air recirculating blower exhaust through a 5-inch-diameter, 6-foot-long tube into the RCM laboratory for use by crewmen there.

This dependent configuration has only the provisions necessary for connection to a manned Apollo Block II CSM, for allowing crew transfer into the laboratory, and for use of the airlock for EVA by one crewman.

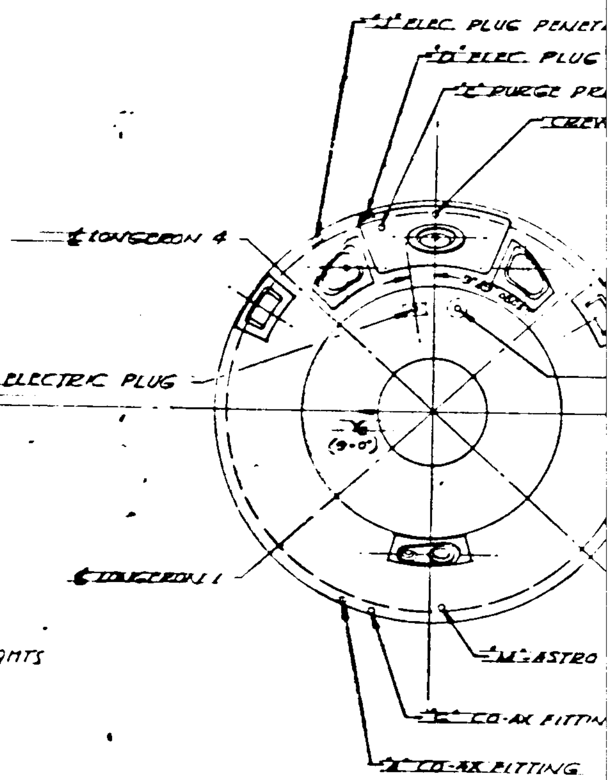
The Z axis of the RCM laboratory is positioned 45 degrees off the Z axis of the SLA. This orientation places the airlock off to the side and not in line with the two access openings in the SLA structure on the Z axis, and allows more room for personnel to enter the interior of the SLA when the vehicle is mated on the launch pad.

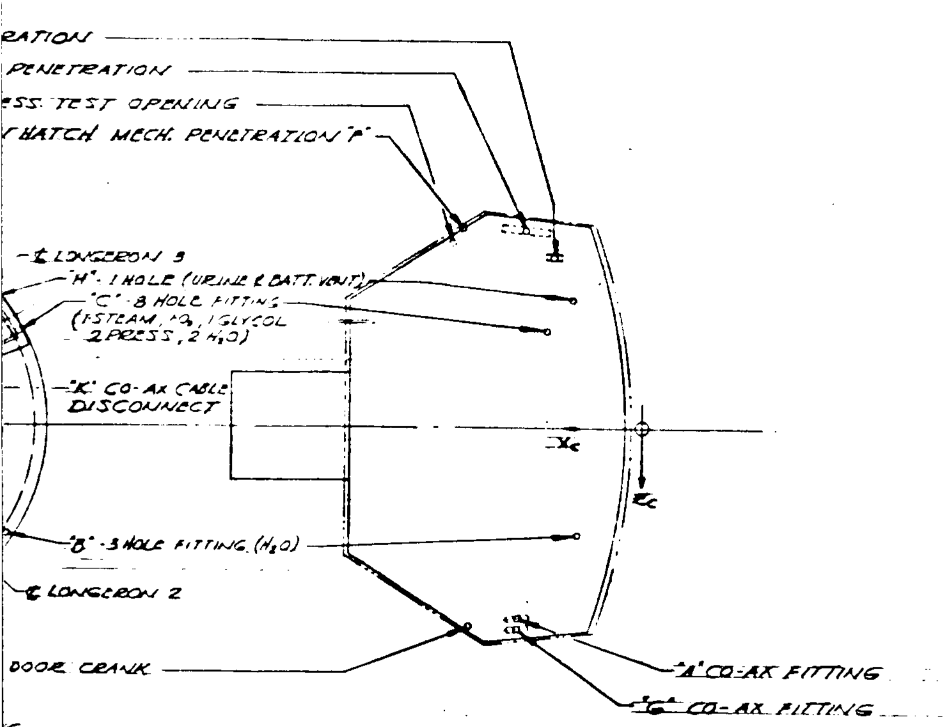
POINT	INNER ML. COORDINATES X_c	Y_c	Z_c	θ_c	COVER DWG. NO.	SEAL DWG. NO.	ATTACHMENT HDWR.	COMLAB USAGE
A	34.1600	24.5000	53.5474	65° 24' 51.48"	Y16-317020 1 REQ	Y16-3[illegible]7017 1 REQ	NAS 1003-3A 18 REQ	
B	18.6856	-48.8043	30.4243	148° 06' 29.44"	Y16-317089-3 4 REQ	ME114-[illegible]-0004 4 REQ	LD153-0003-2403 4 REQ	GN₂ SUPPLY SEE DETAIL A
C	26.0000	-51.5856	-26.9500	207° 35' 02.46"				MAIN ELEC. POWER
D	31.6250	19.8750	-55.2046	289° 48' 00.42"				*NOT APPLICABLE
E	51.5000	14.8752	-51.8759	286° 00' 00.00"				*NOT APPLICABLE
F	48.8898	0	-55.6309	270° 00' 00.00"				
G	34.1600	16.3750	56.5636	73° 51' 15.87"	Y16-317020 1 REQ	Y16-317017 1 REQ	NAS 1003-3A 18 REQ	
H	19.2500	-45.0084	-36.0000	218° 39' 16.62"	Y16-317089-11 1 REQ	ME114-[illegible]-0004 1 REQ	2WIC-14-21-90 1 REQ	
J	24.0000	33.1225	-47.6526	304° 48' 09.74"				SUIT CONNECTOR (2)
M	47.5930	-4.1250	56.3068	34° 11' 23.87"	Y16-311075 1 REQ	Y16-311127 1 REQ	NAS 1003-3A 16 REQ	
FWD. BULKHEAD PENETRATION								
K		-6.1000	-29.3000		Y16-316098 1 REQ	Y16-316053 1 REQ	NAS 1003-3A 16 REQ	
L		5.0144	-28.6199		Y16-316099 1 REQ	Y16-316052 1 REQ	NAS 1003-3A 20 REQ	

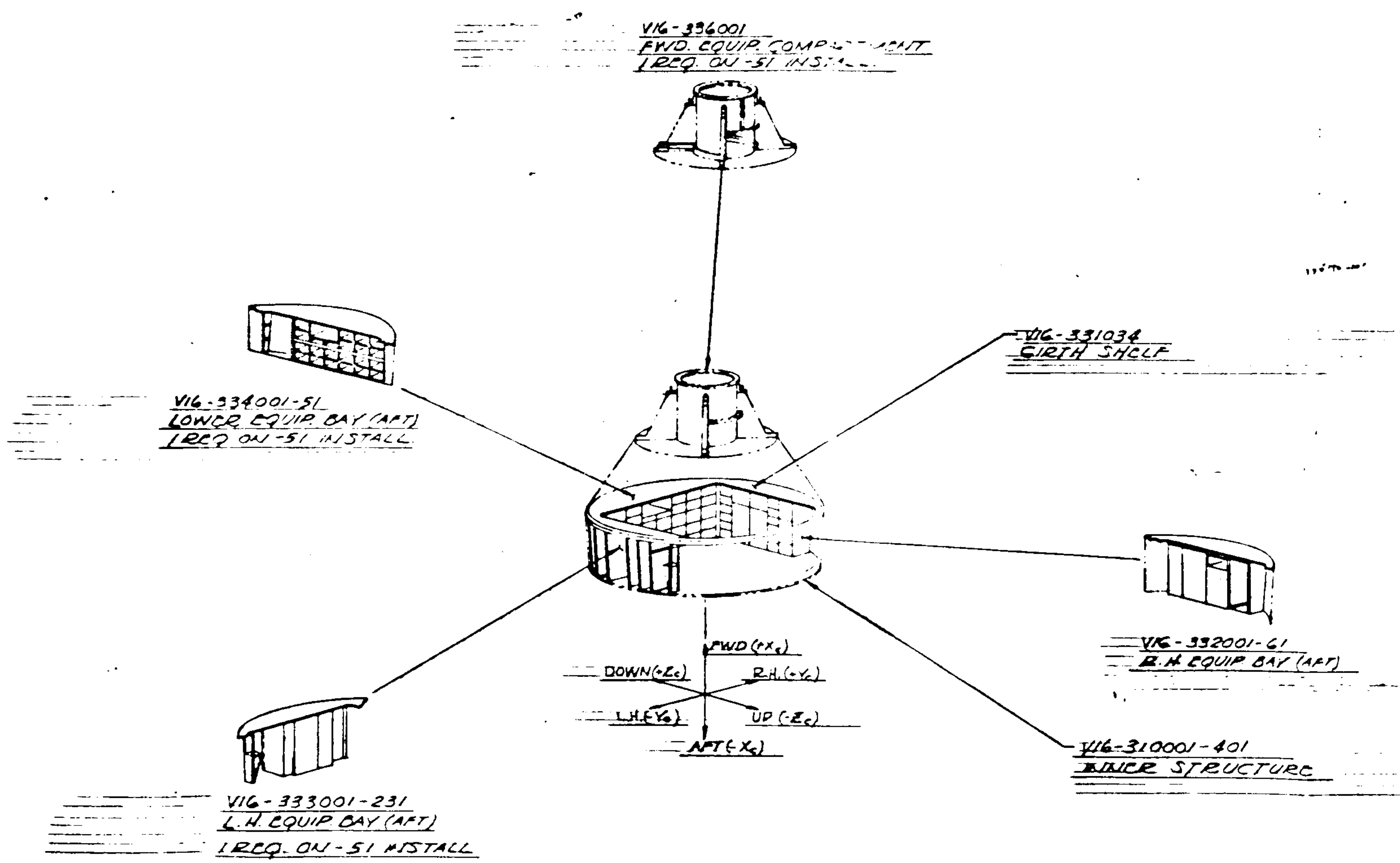
*EXISTING HATCH NOT USED



DETAIL A







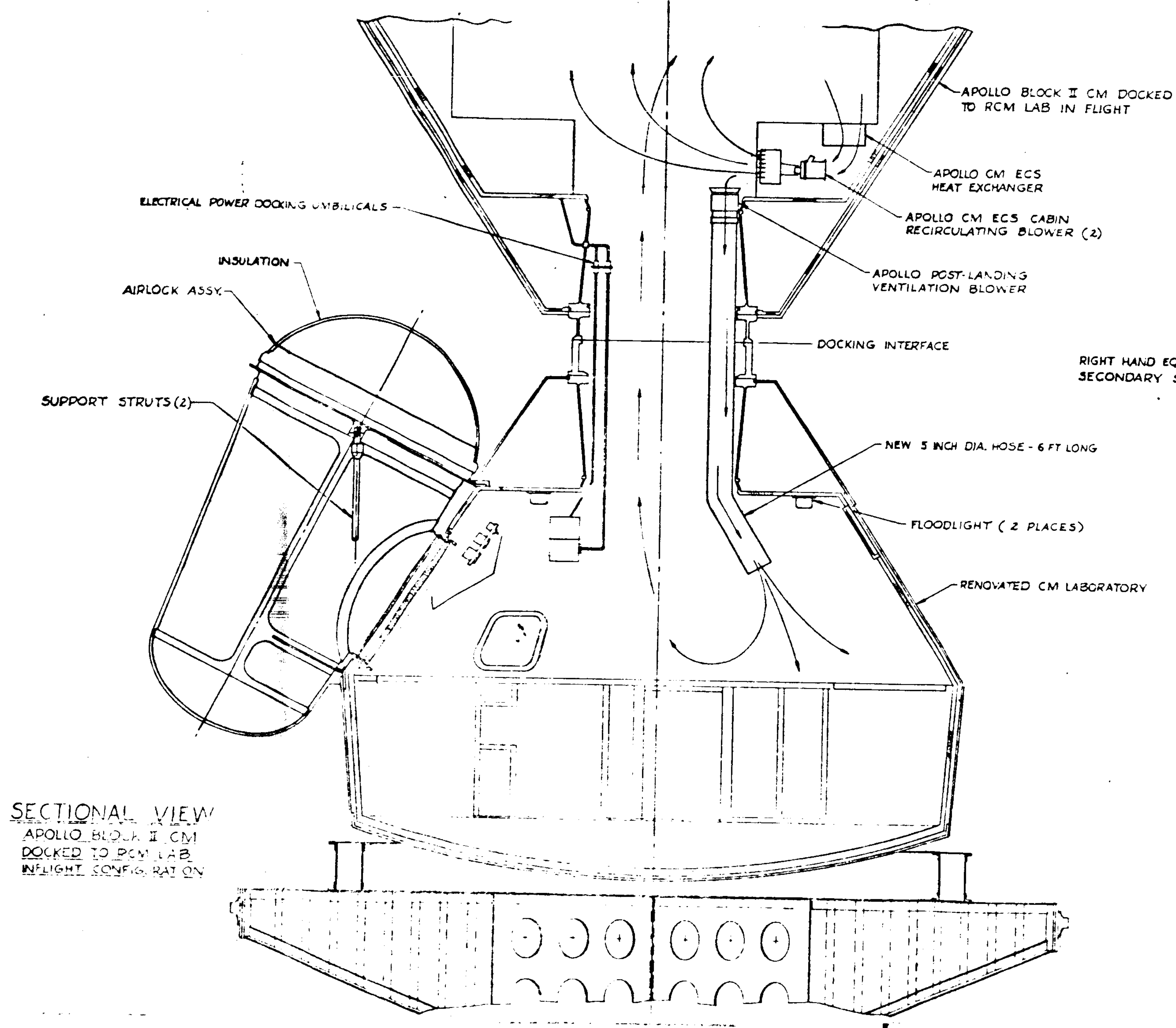
FOLDOUT FRAME 3

FOLDOUT FRAME 4

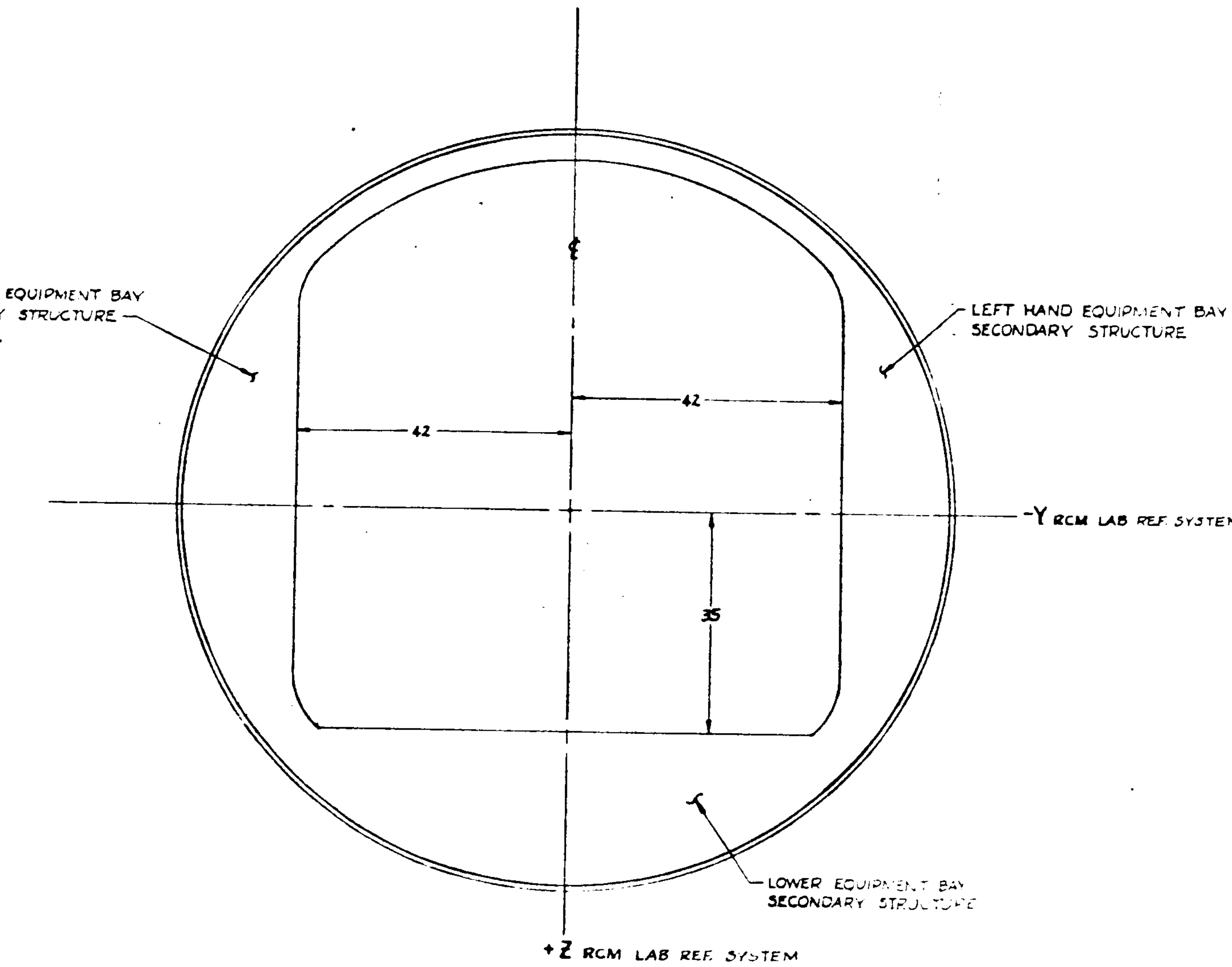
	DWG NUMBER	TITLE	RETAIN	REMOVE
1	V16-310001-401	INNER STRUCTURE	X	
2	V16-331034	GIRTH SHELF	X	
3	V16-332001-61	R.H. EQUIP. BAY (AFT)	X	
4	V16-335130-101	MAIN DISPLAY PANEL		X
5	V16-335132	MAIN DISPLAY PANEL (MODIF.)		X
6	V16-336001	FWD. EQUIP. COMPARTMENT	X	
7	V16-334001-51	LOWER EQUIP. BAY (AFT)	X	
8	V16-335040-801	L.H. FWD. EQUIP. BAY		X
9	V16-334002-141	UPPER EQUIP. BAY		X
10	V16-335060-401	R.H. FWD. EQUIP BAY		X
11	V16-337001-11	AFT COMP. STR. (OUTER)		X
12	V16-332002-311	R.H. EQUIP. BAY (FWD.)		X
13	V16-333001-231	L.H. EQUIP. BAY (AFT.)	X	
14	V16-333030-221	L.H. EQUIP. BAY (FWD.)		X
15	V16-335040-501	L.H. UPPER EQUIP. BAY		X

NOTE This drawing is reduced. Scale notations apply to original only.

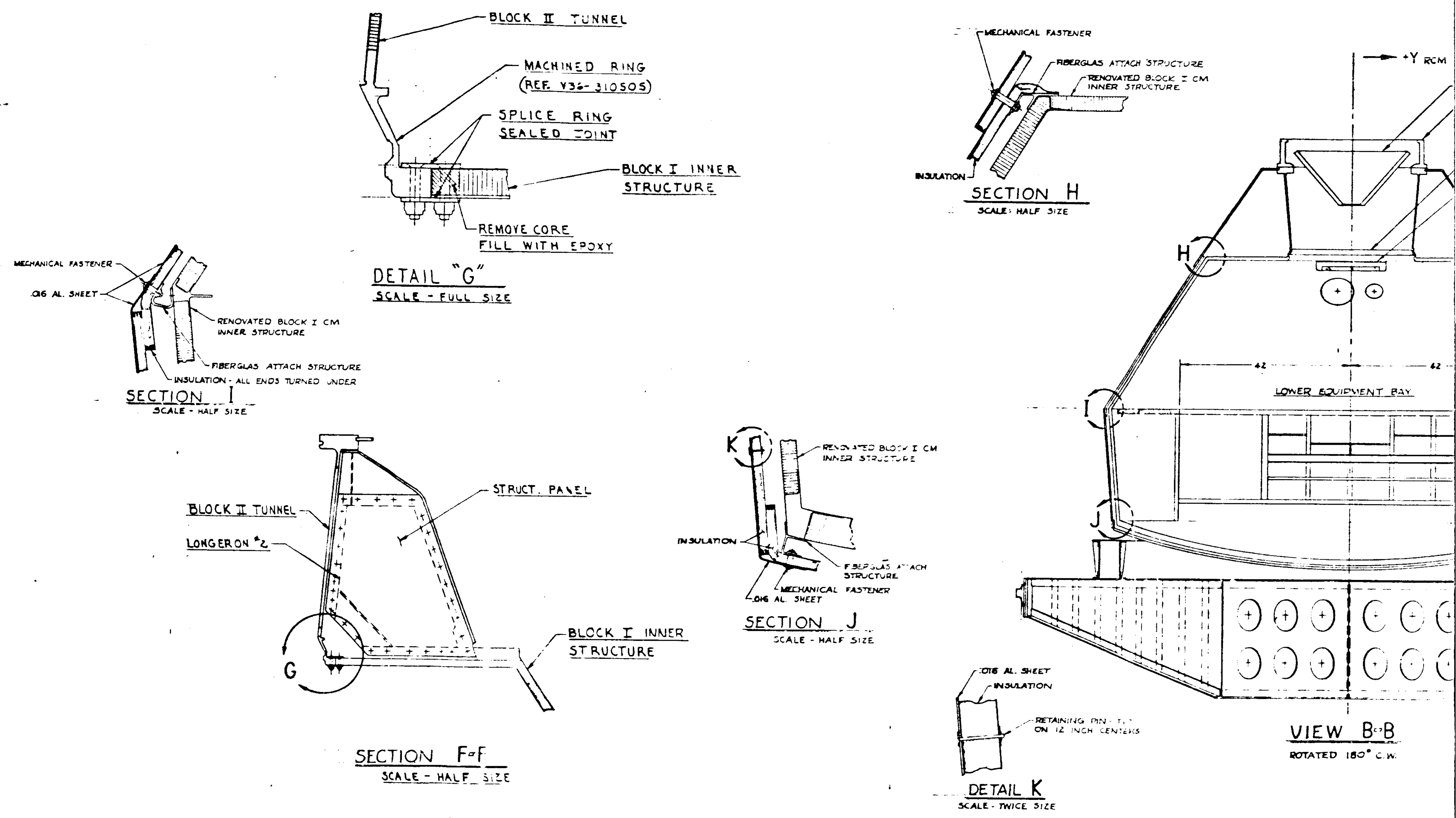
Figure 101. Spacecraft 006 Renovation



SECTIONAL VIEW
APOLLO BLOCK II CM DOCKED TO RCM LAB INFLIGHT CONFIGURATION

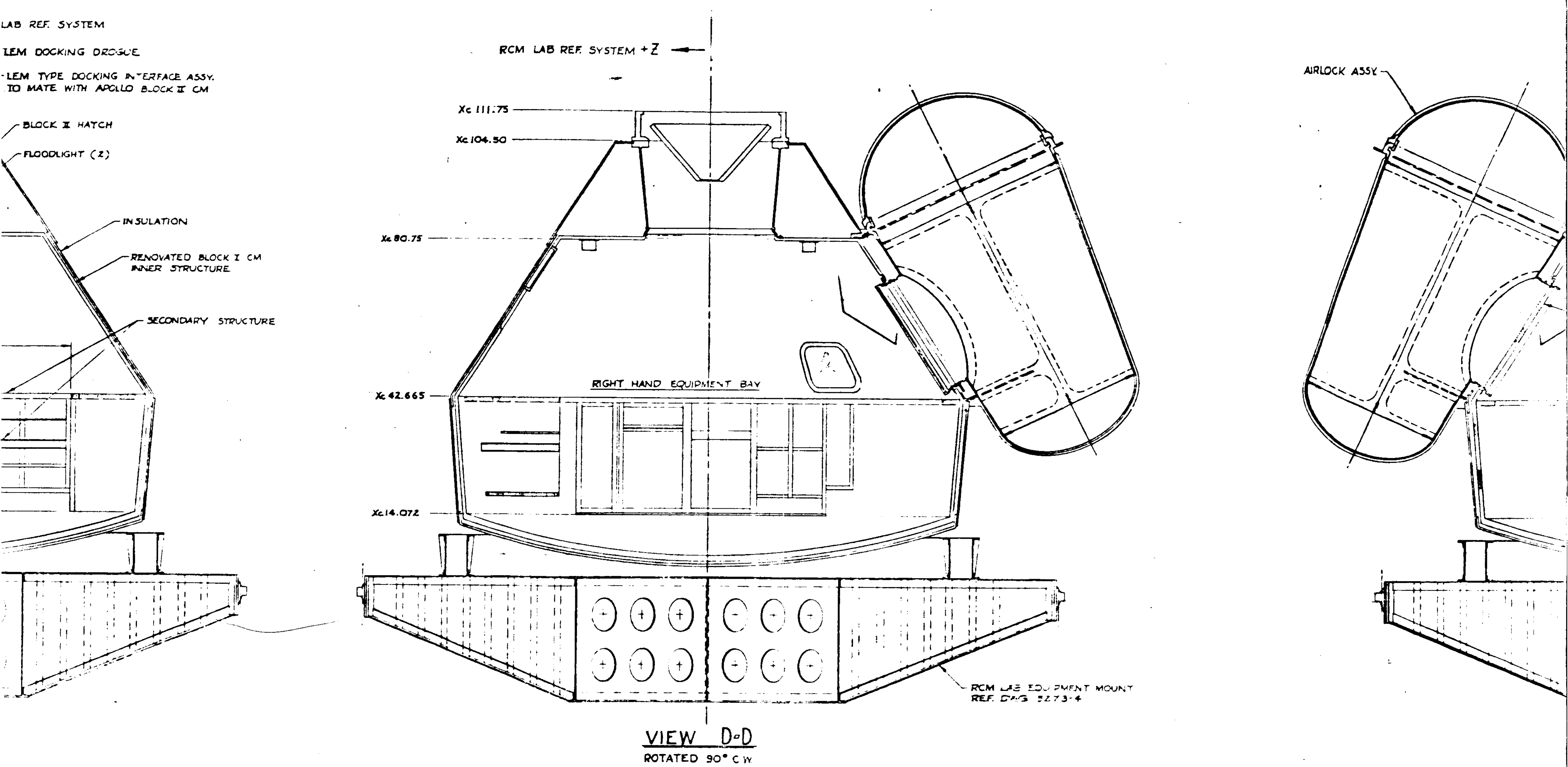


VIEW E-E
ROTATED 90° C.C.W.



FOLDOUT FRAME 3

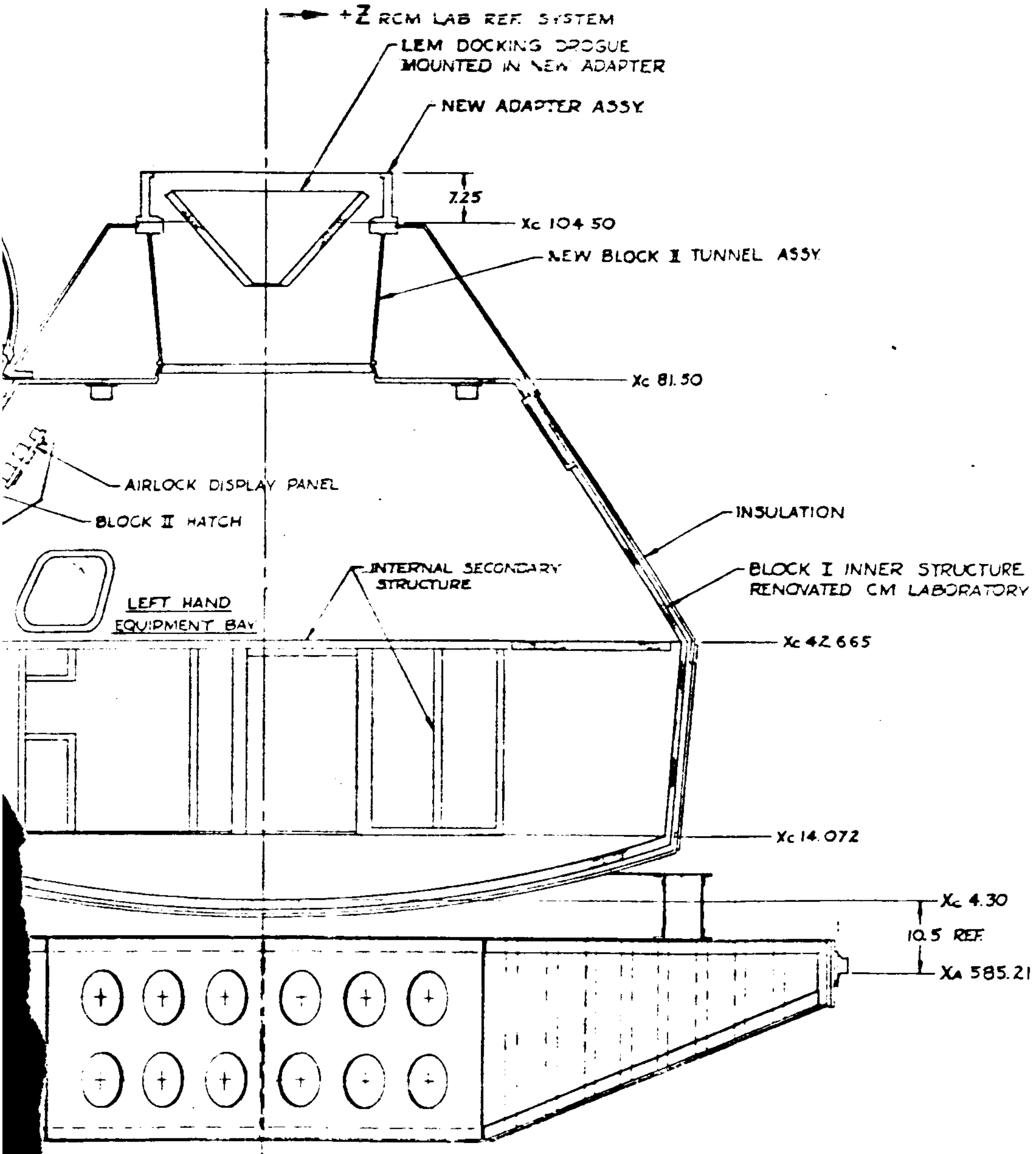
FOLDOUT FRAME 4



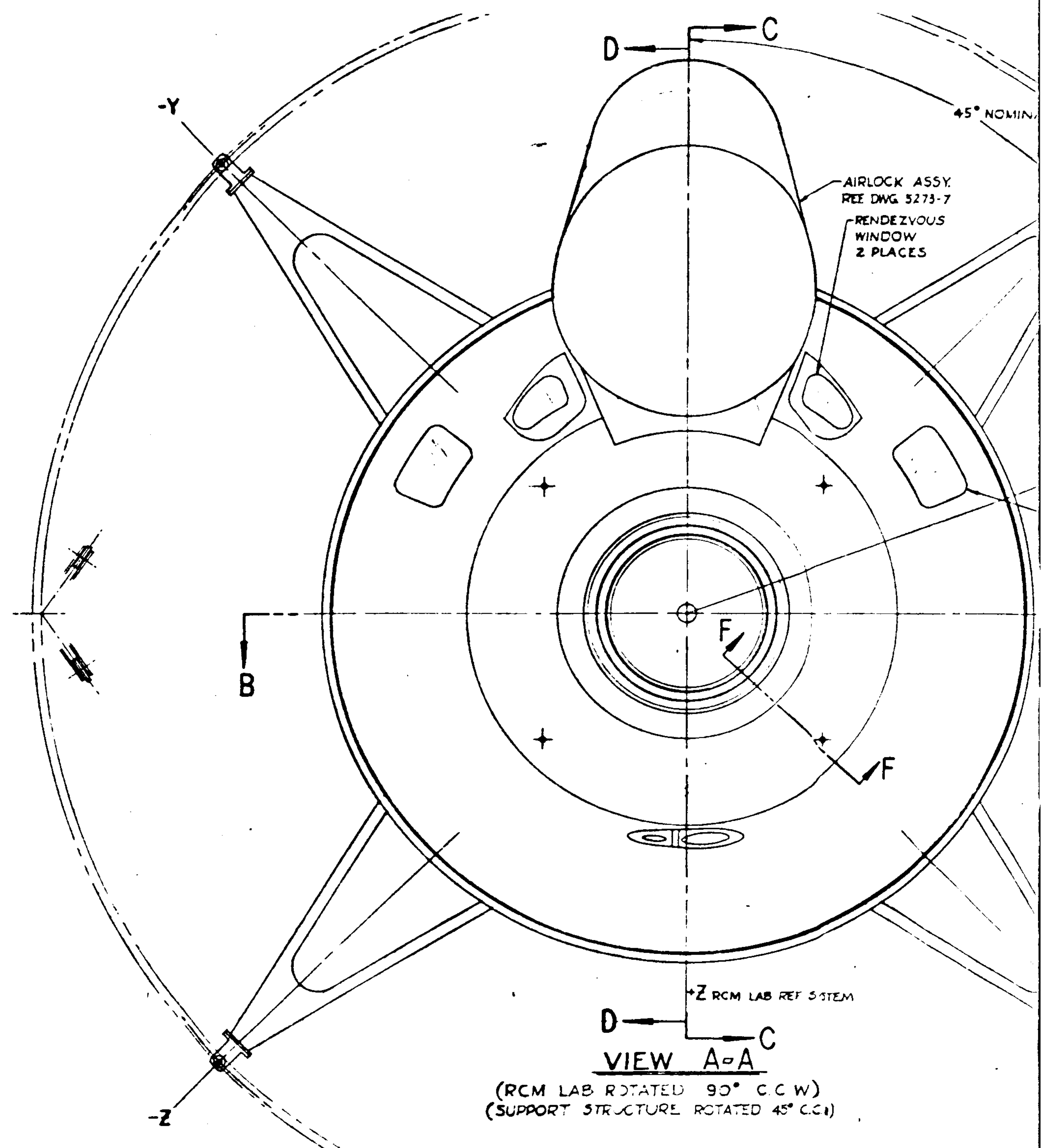
VIEW D-D
ROTATED 90° CW

FOLDOUT FRAME 5

FOLDOUT FRAME 6



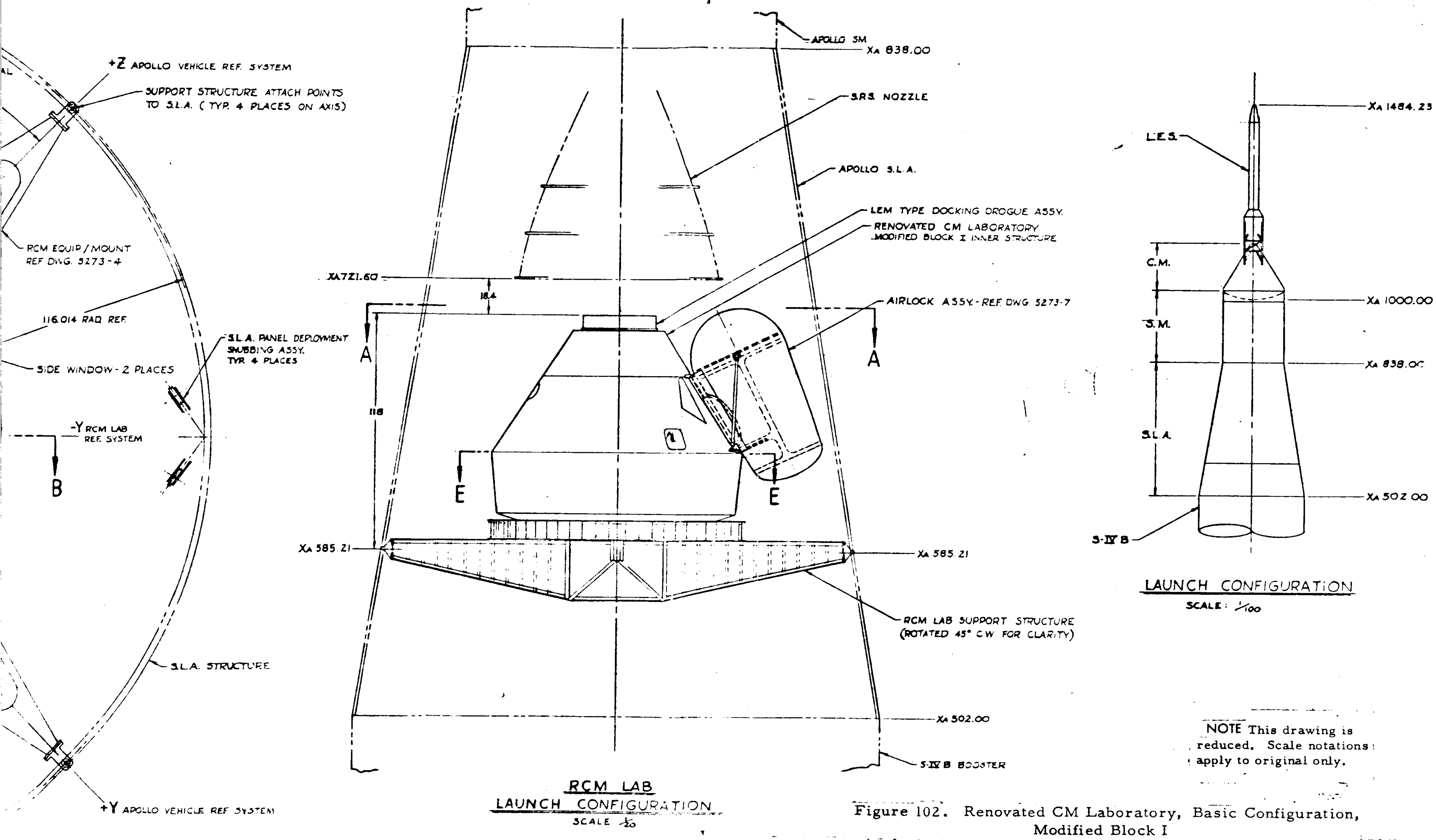
VIEW C-C
ROTATED 90° C.C.W.



VIEW A-A
(RCM LAB ROTATED 90° C.C.W.)
(SUPPORT STRUCTURE ROTATED 45° C.C.W.)

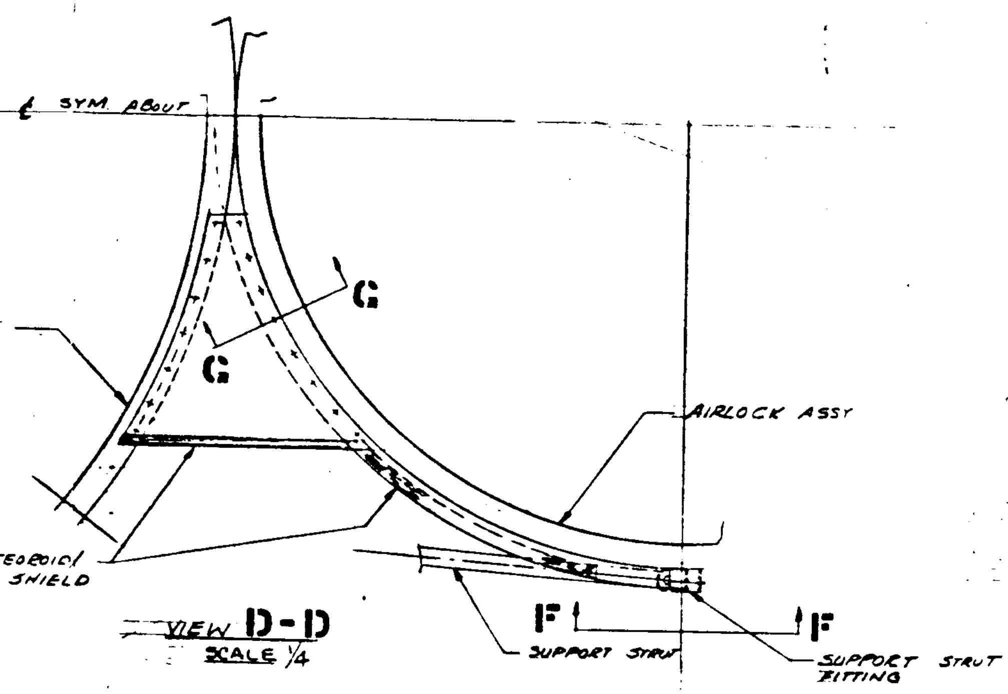
FOLDOUT FRAME 1

FOLDOUT FRAME

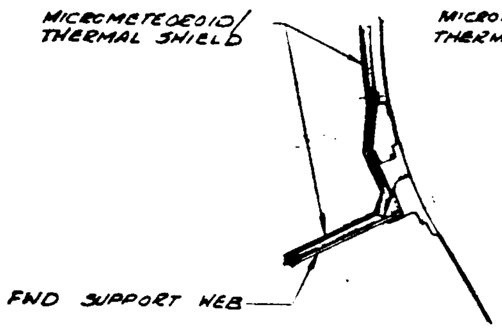


NOTE This drawing is reduced. Scale notations apply to original only.

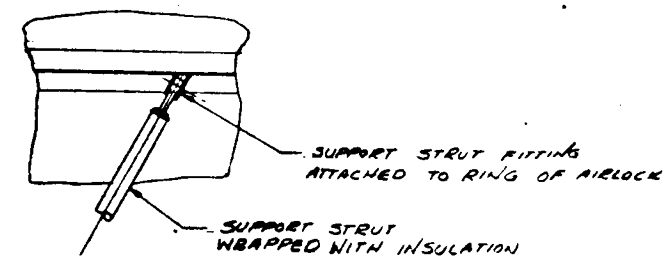
Figure 102. Renovated CM Laboratory, Basic Configuration, Modified Block I



VIEW D-D
SCALE 1/4



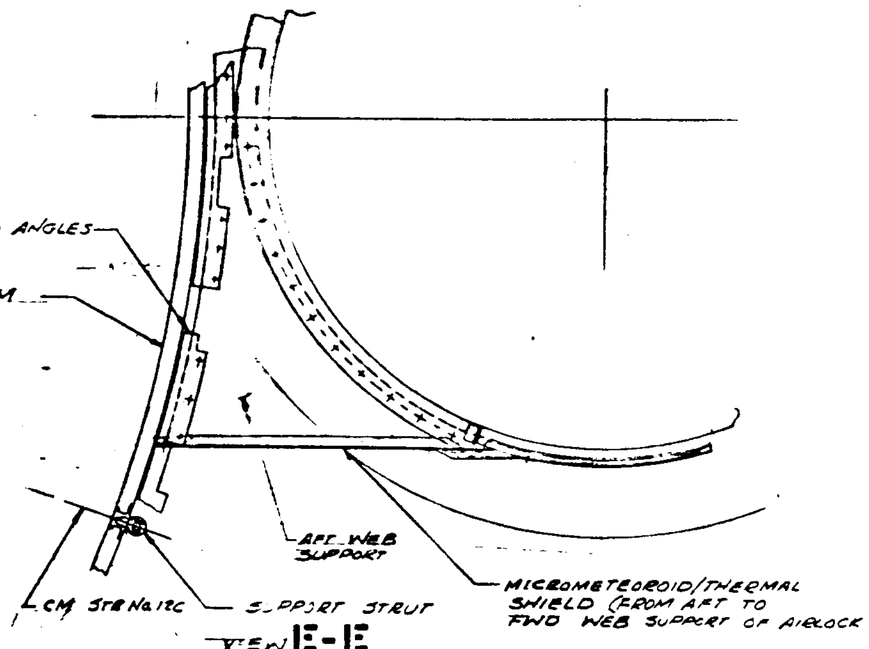
SECTION G-G
SCALE 1/4



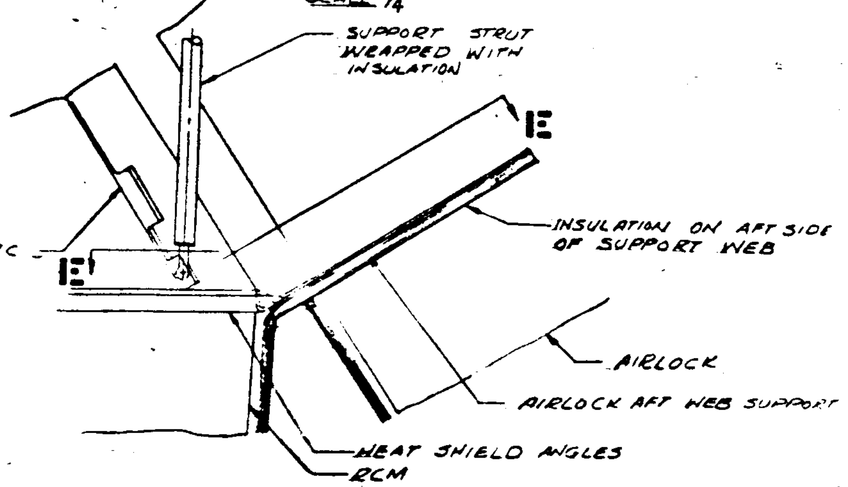
VIEW F-F
SCALE 1/4

FOLDOUT FRAME

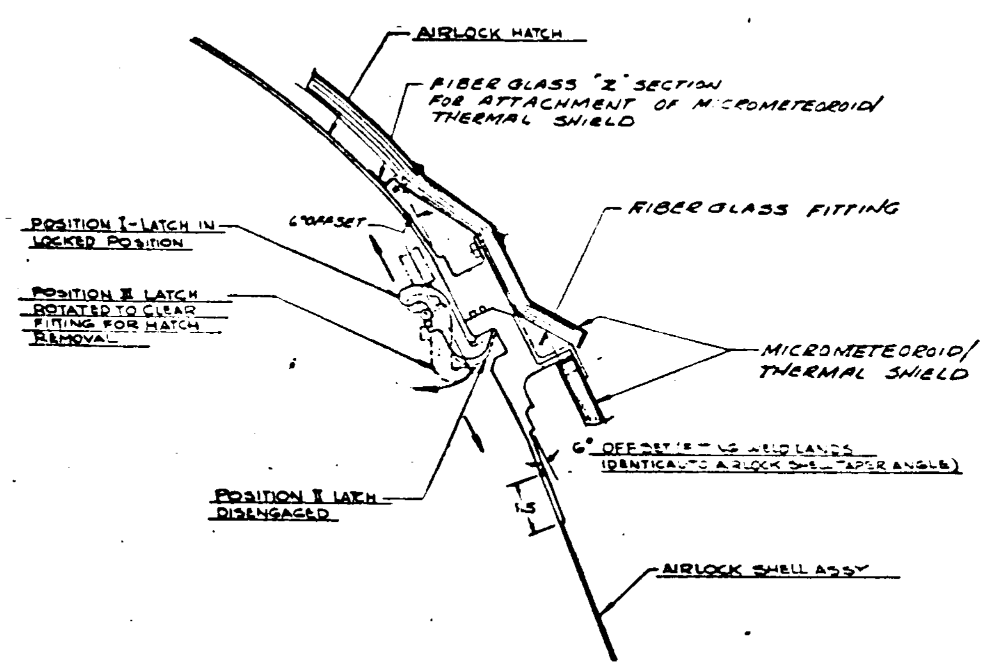
FOLDOUT FRAME



VIEW E-E
SCALE 1/4



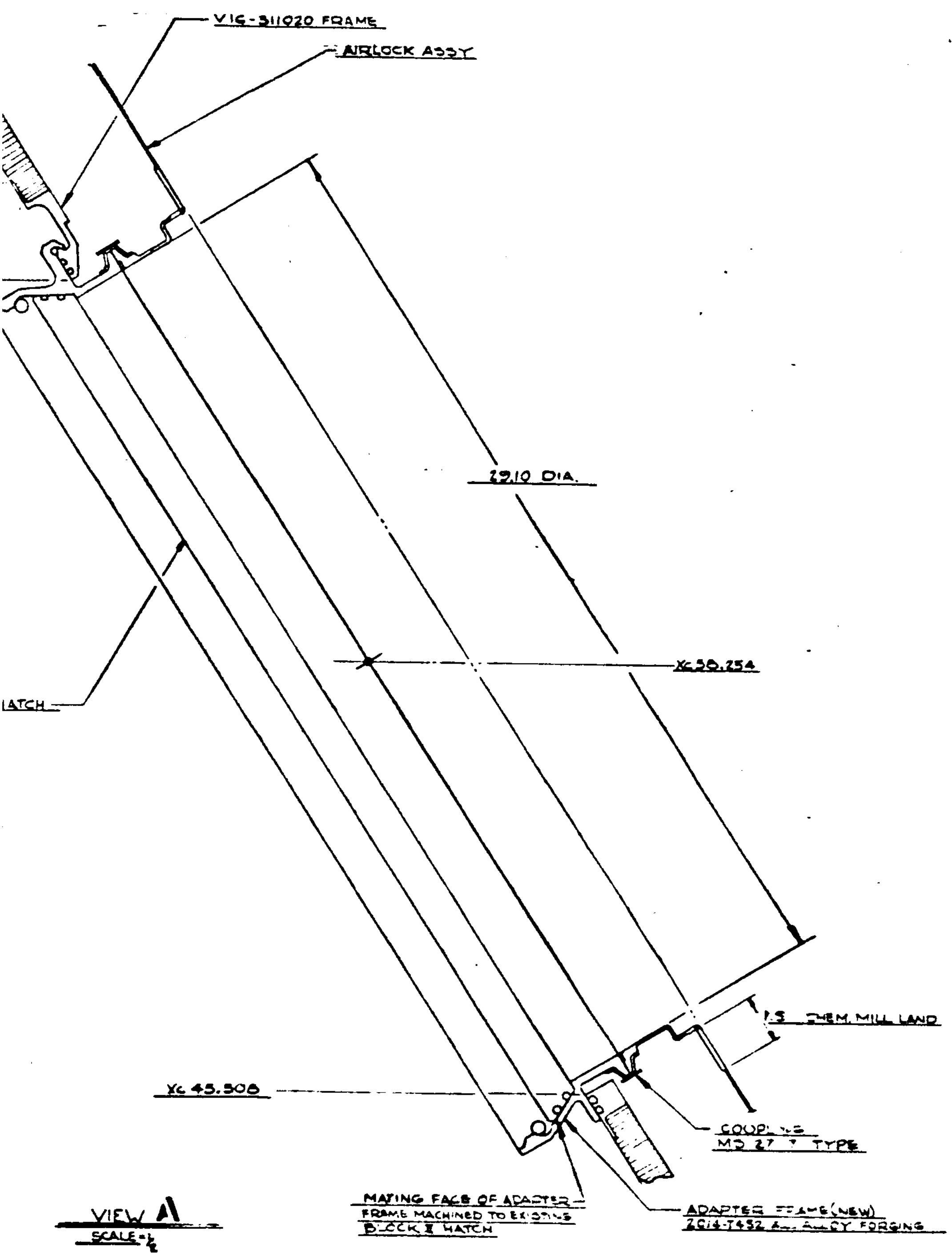
DETAIL C
SCALE 1/4



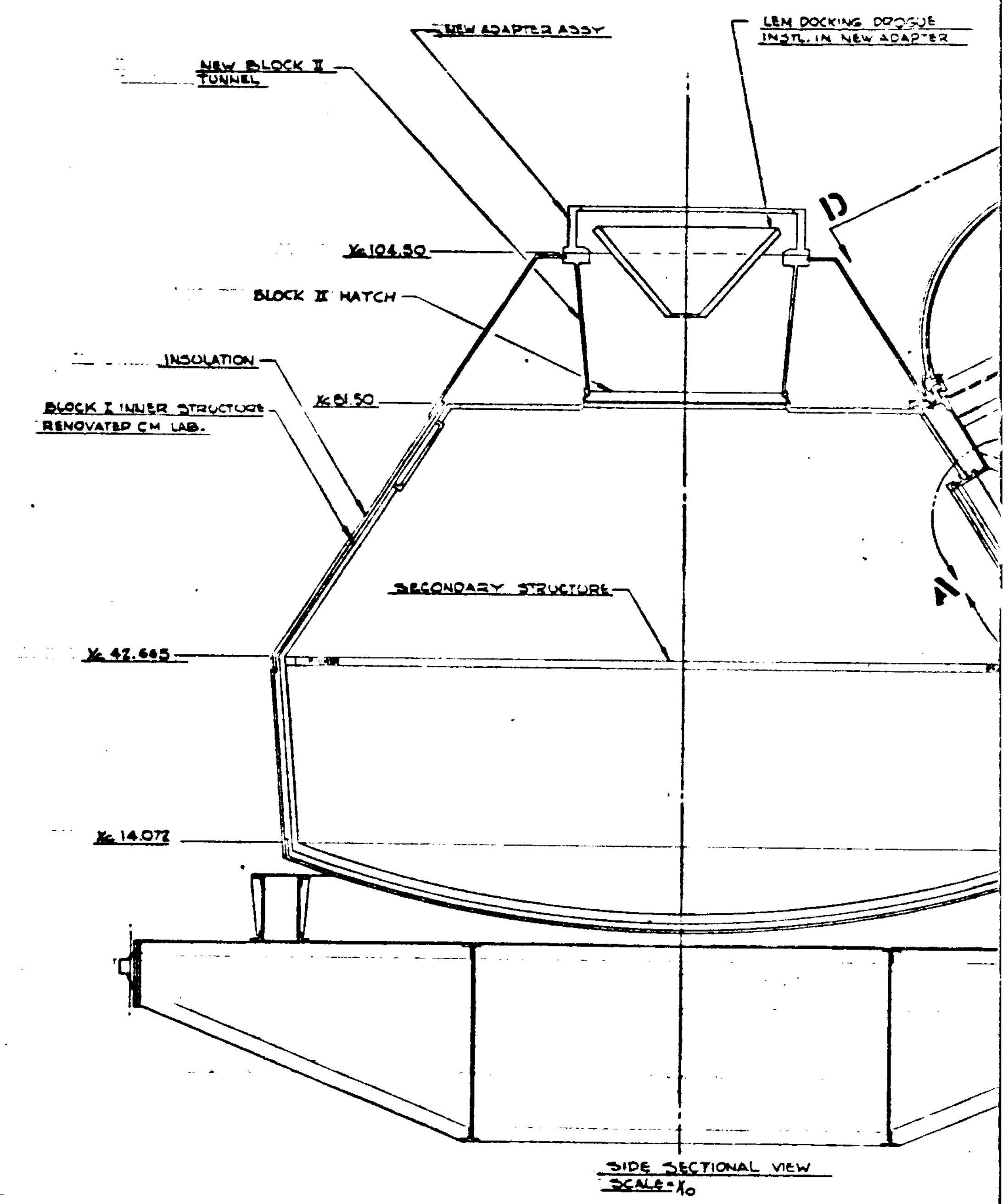
VIEW B
SCALE 1/4

FOLDOUT FRAME 2

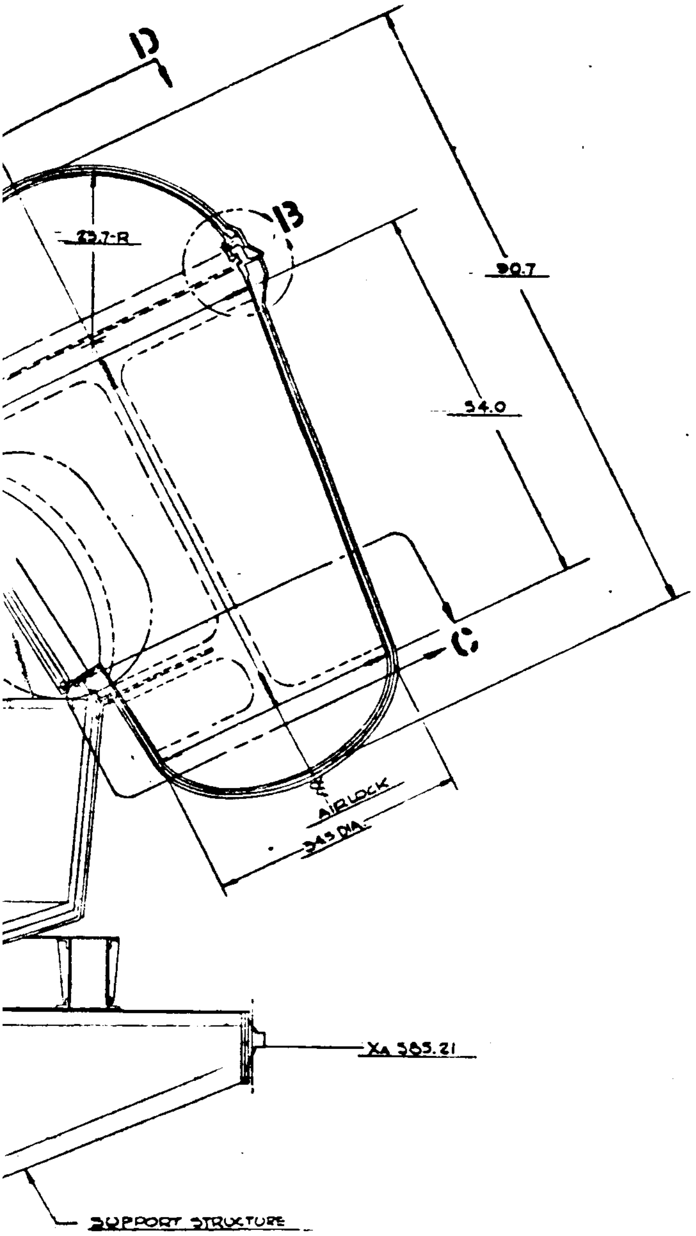
FOLDOUT FRAME 3

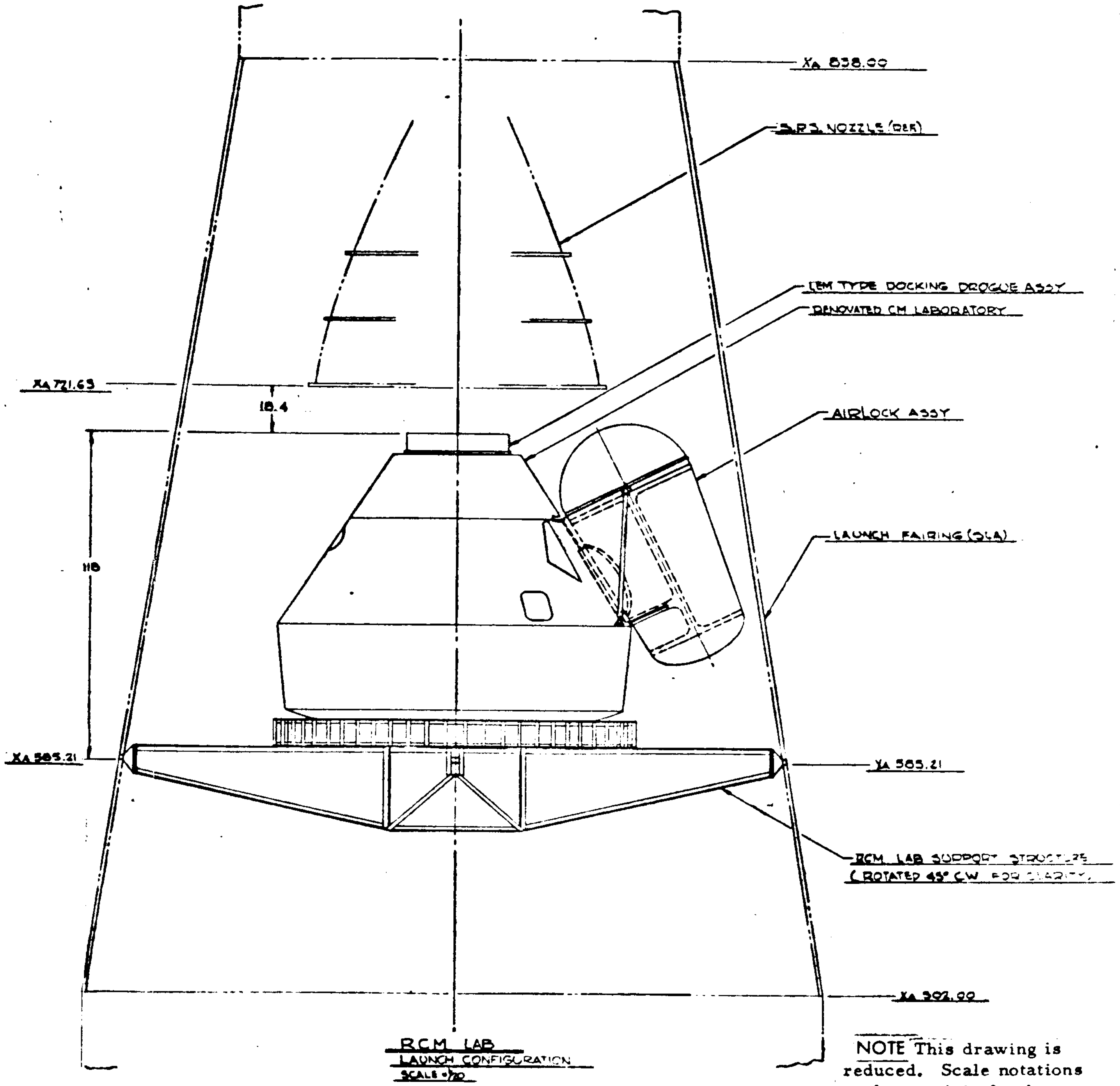


FOLDOUT FRAME 4



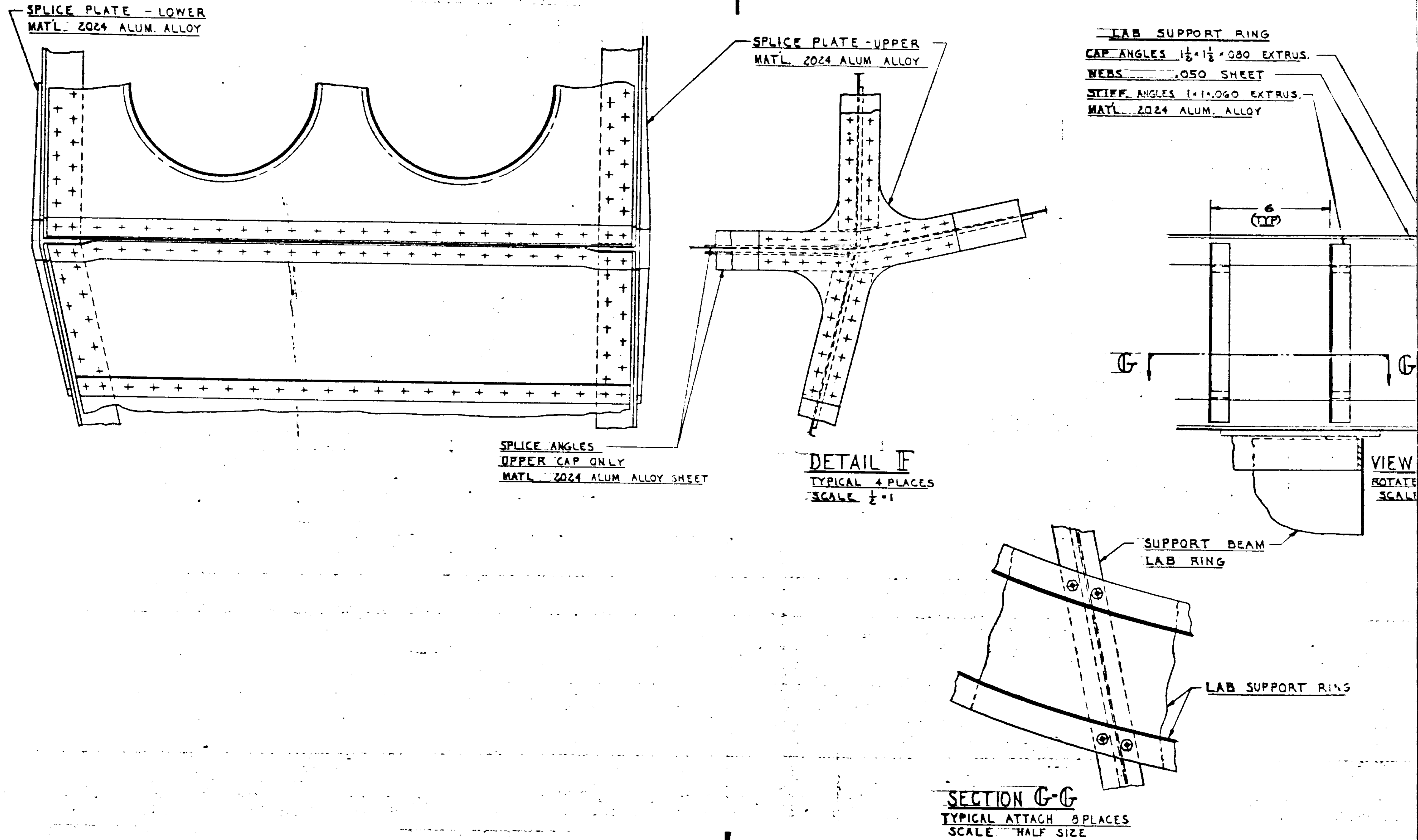
FOLDOUT FRAME 5





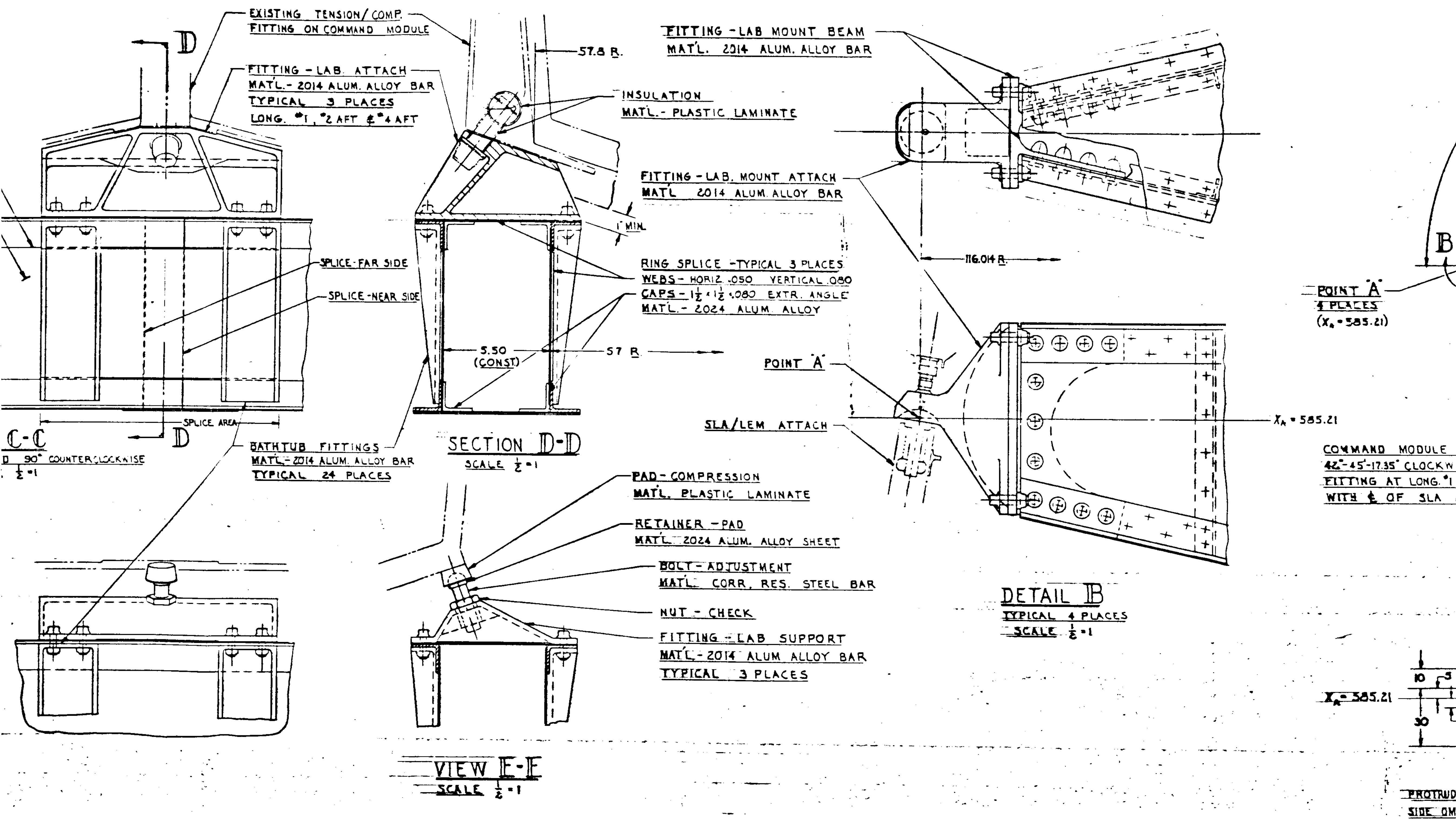
NOTE This drawing is reduced. Scale notations apply to original only.

Figure 103. Airlock Assembly, Renovated Command Module Laboratory



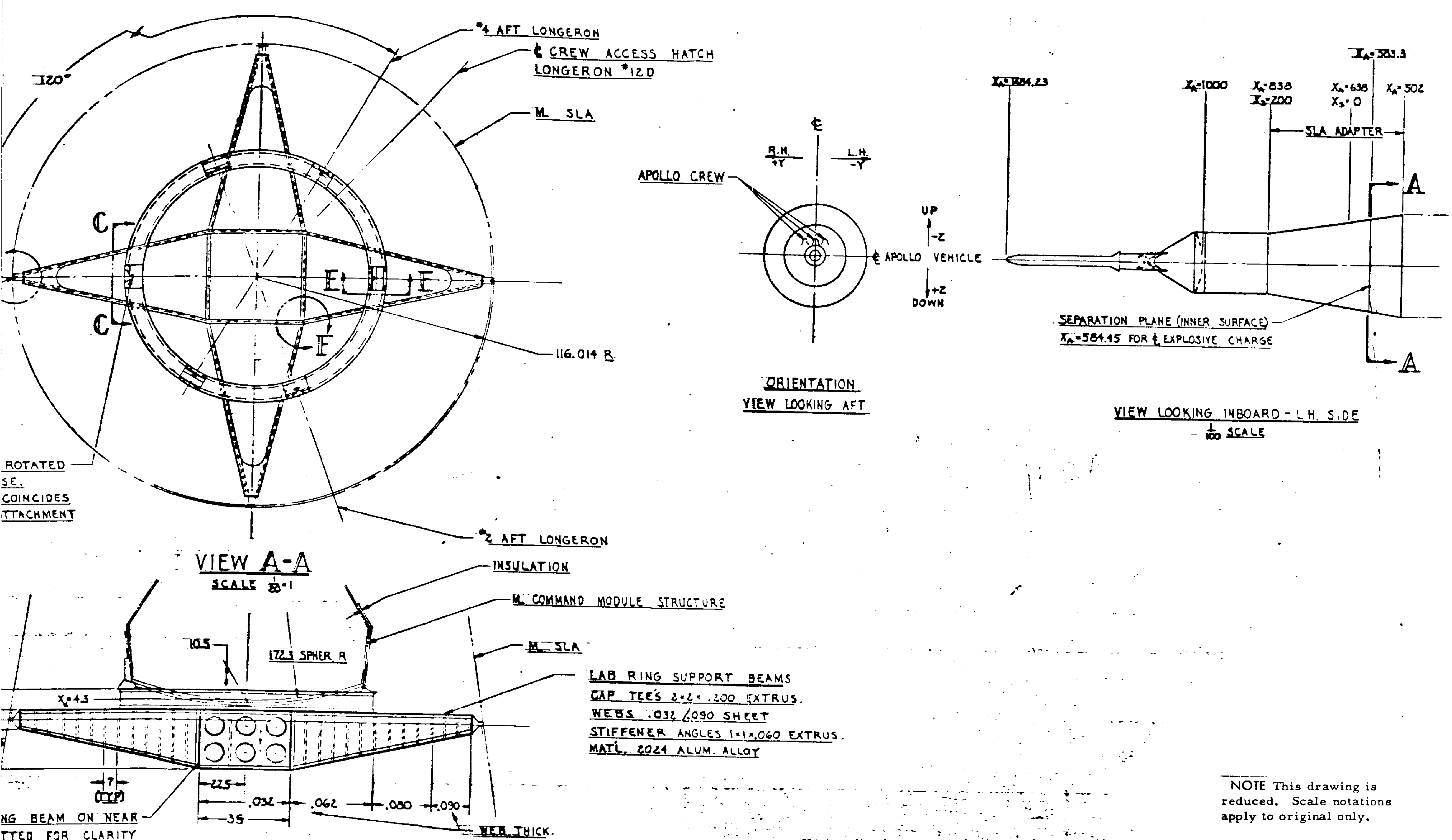
FOLDOUT FRAME 1

FOLDOUT FRAME 2



FOLDOUT FRAME 3

FOLDOUT FRAME 4



NOTE This drawing is reduced. Scale notations apply to original only.

Figure 104. Renovated CM Laboratory/Equipment Mount (Web Beam Concept)

Laboratory Mount. Two laboratory mounts were considered in this study—one designed to a laboratory weight of 15,000 pounds (Figure 105) and the other designed for a laboratory weight of 25,000 pounds (Figure 104). Both of these structures are based on the built-up web/beam concept. An alternative design, capable of accepting the 25,000-pound load, is shown in Figure 106. The alternative design is based on a tubular truss concept.

The structure shown in Figure 105 was designed during a previous company-funded study, and employs the concept of a cruciform support structure and a laboratory adapter ring. This concept permits attachment to the laboratory at the existing compression and tension/compression fittings, and still allows for selected orientation on the simple support structure. Additional features of the adapter ring concept are that the ring assembly allows a convenient method for handling and moving the laboratory by provision of a flat surface for resting, and it provides an adapter when installing the laboratory on structures other than the mount structure.

The adapter ring is designed as a toroidal box beam made up of 2024 aluminum sheets and extruded shapes. The cross-sectional dimensions of the box are 10 inches high by 5 inches wide, with a flat sheet close-out at top and bottom. The sides are flat sheets with angle stiffeners. The corners of the box are extruded angles, arranged with the upper and lower box flanges exposed to facilitate attachment of fittings. There are six machined fittings designed to mate with the compression and tension/compression fittings on the CM inner shell. The three compression-only fittings provide for adjustable compression pads in order to assure proper prestress in the pads so that each fitting accepts its share of the boost loads. The remaining three fittings have bolted attachments to the CM inner structure. Nonmetalic pads are used at all fittings to thermally isolate the CM inner structure from the laboratory mount structure. The entire ring assembly is of riveted construction with bolt attachment at the critical points, such as the laboratory attach fittings.

The mount structure (cruciform) is made up of constant-depth beams 20 inches high, configured to form a square 40 inches on a side at the center of the assembly, with four legs of two beams each extending outward to match the LM/SLA attach points. The beam subassemblies that form the central portion of the structure are made up of sheet webs that have flanged lightening holes and extruded angle stiffeners. The beam caps are extruded "T" members. The entire structure is a riveted assembly. The outboard beams are made of sheet webs stiffened with extruded angles and capped with extruded "T" members. The beams are attached to the central structure with bolted joints to provide for easy disassembly for shipping. The outer ends of each pair of beams are connected by a single machined fitting that is designed to mate with the existing LM/SLA fittings.

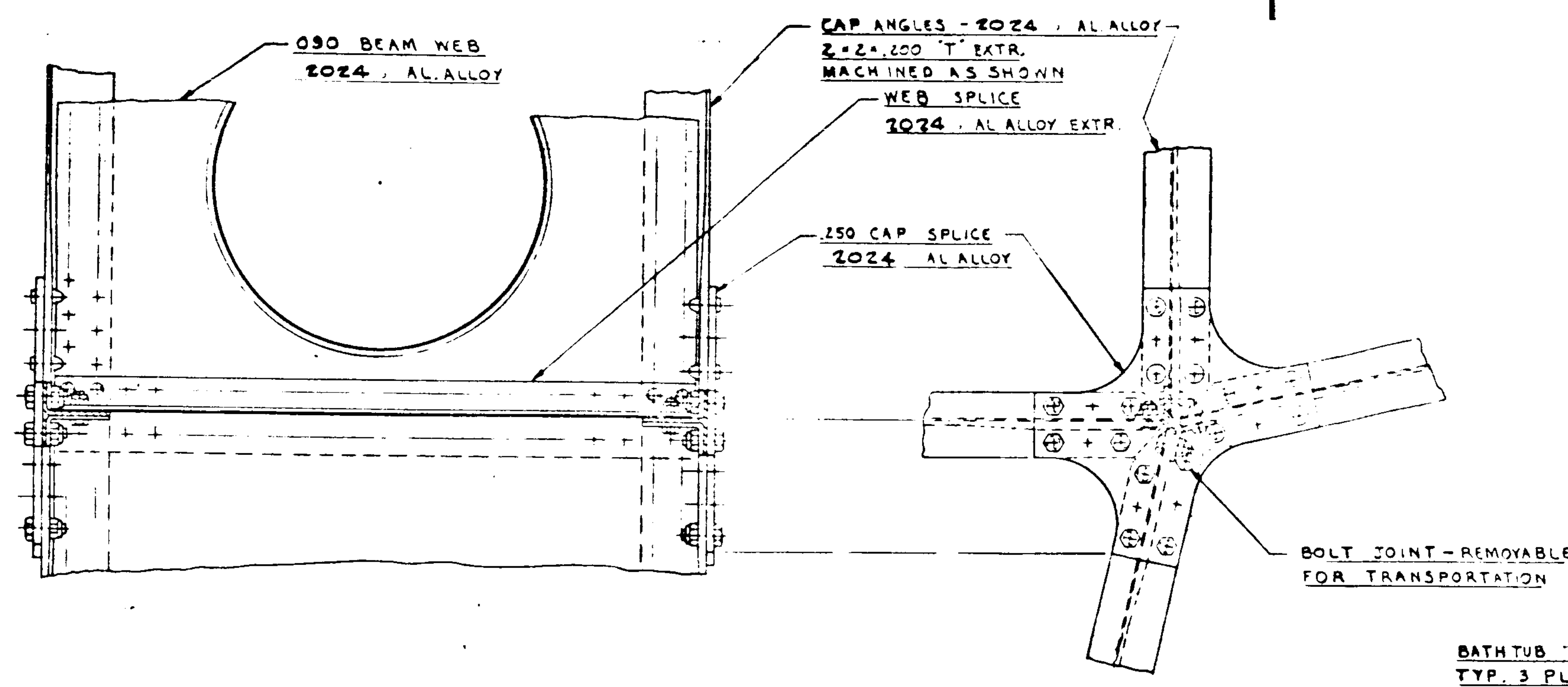
The ring is attached to the mount structure simply by being bolted together through the flanges of each assembly at their intersection.

An alternative to this design may be considered in the area of the bolted connection of inner and outer beams. A completely riveted joint would simplify and lighten the assembly but may impose some shipping restrictions because of the large span of the beam assembly.

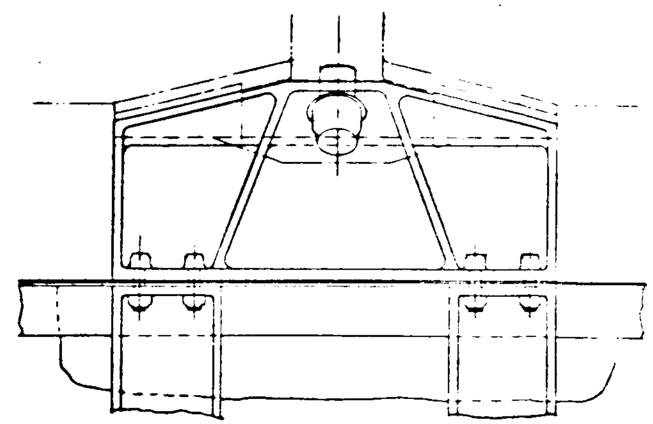
Figure 104 shows a new laboratory mount designed for the increased load, 25,000 pounds. The concept is the same as that of Figure 105, but the support beam height has been increased to 30 inches in the area of the central structure and tapers down to a 10-inch depth at the LM/SLA fittings. The laboratory adapter ring is identical to the previously described ring. The construction of the mount structure is the same as that previously described, except for the joint detail at the juncture of the inner and outer beams, where this design has an all-riveted joint as opposed to the bolted joint shown in Figure 105. The other area of design change is at the outer joint of the paired outboard beams. This design utilizes a two-piece fitting where one fitting is designed to join the outer ends of two beams with a riveted connection. The outer face of this fitting is flat, to provide a good surface for tooling pick-up in the assembly jig. A separate, machined fitting is designed to mate with the LM/SLA fitting and attach to the flat face of the outboard beam fitting. This type of fitting permits the use of shims to assure proper fit up of the laboratory mount to the SLA.

Figure 106 shows the alternative support structure design where a tubular truss structure replaces the web beam structure. In this concept, the laboratory adapter ring remains identical to the previous concepts. The tubular truss structure is 30 inches deep, with the upper and lower members converging to a single fitting at the LM/SLA attachment. Individual fittings are required on the upper tube members to provide attachment of the adapter ring.

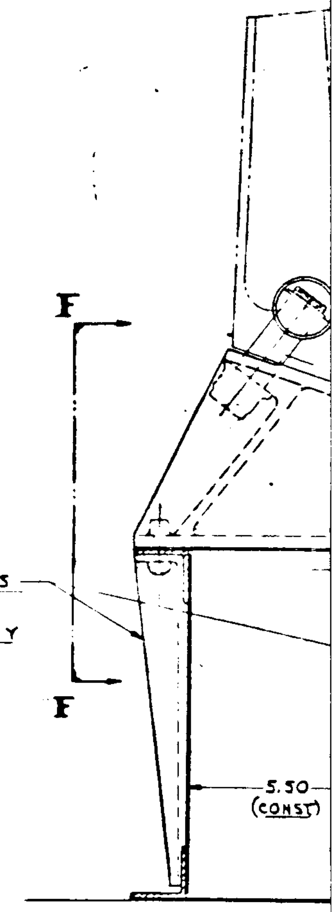
In considering the advantages of one concept compared to the other, it was apparent that there was little to choose from on the basis of weight, inasmuch as the truss structure was estimated to weigh 638 pounds and the web beam weighed 688 pounds. Therefore, other aspects were reviewed, such as flexibility of design to accommodate revisions in attach points for equipment, etc., and ease of fabrication and fit-up. When reviewed as to these criteria, the tubular truss offers some disadvantages in both areas. In order to provide additional support points on a tubular truss, that attach point must be backed by an additional tubular member or a fitting spanning two adjacent tube joints. The fabrication of a large tubular structure requires stress relieving of the welded structure to assure alignment and stability to the assembly. In the case of the web beam structure, the addition of new attach fittings requires the simple addition of a doubler or stiffener, to accept



DETAIL F TYP 4 PLCS
1/2 SCALE



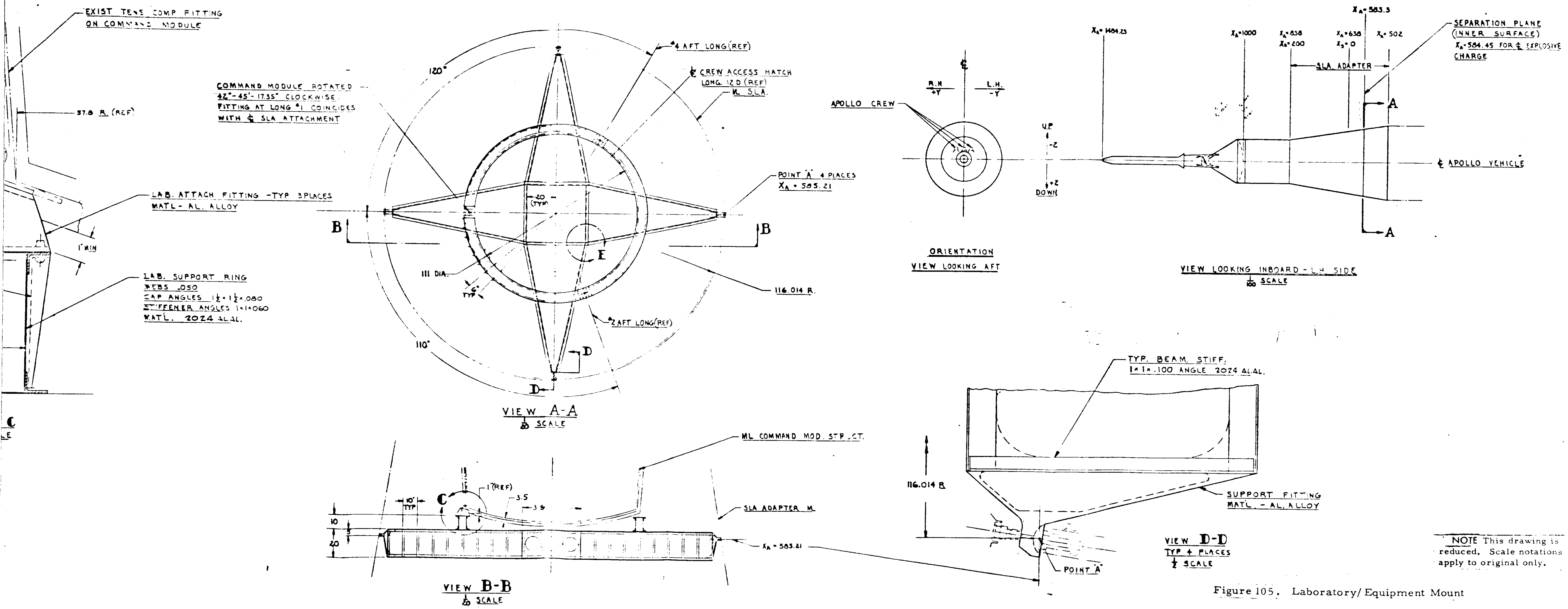
VIEW F-F
1/2 SIZE



DETAIL
1/2 SCA

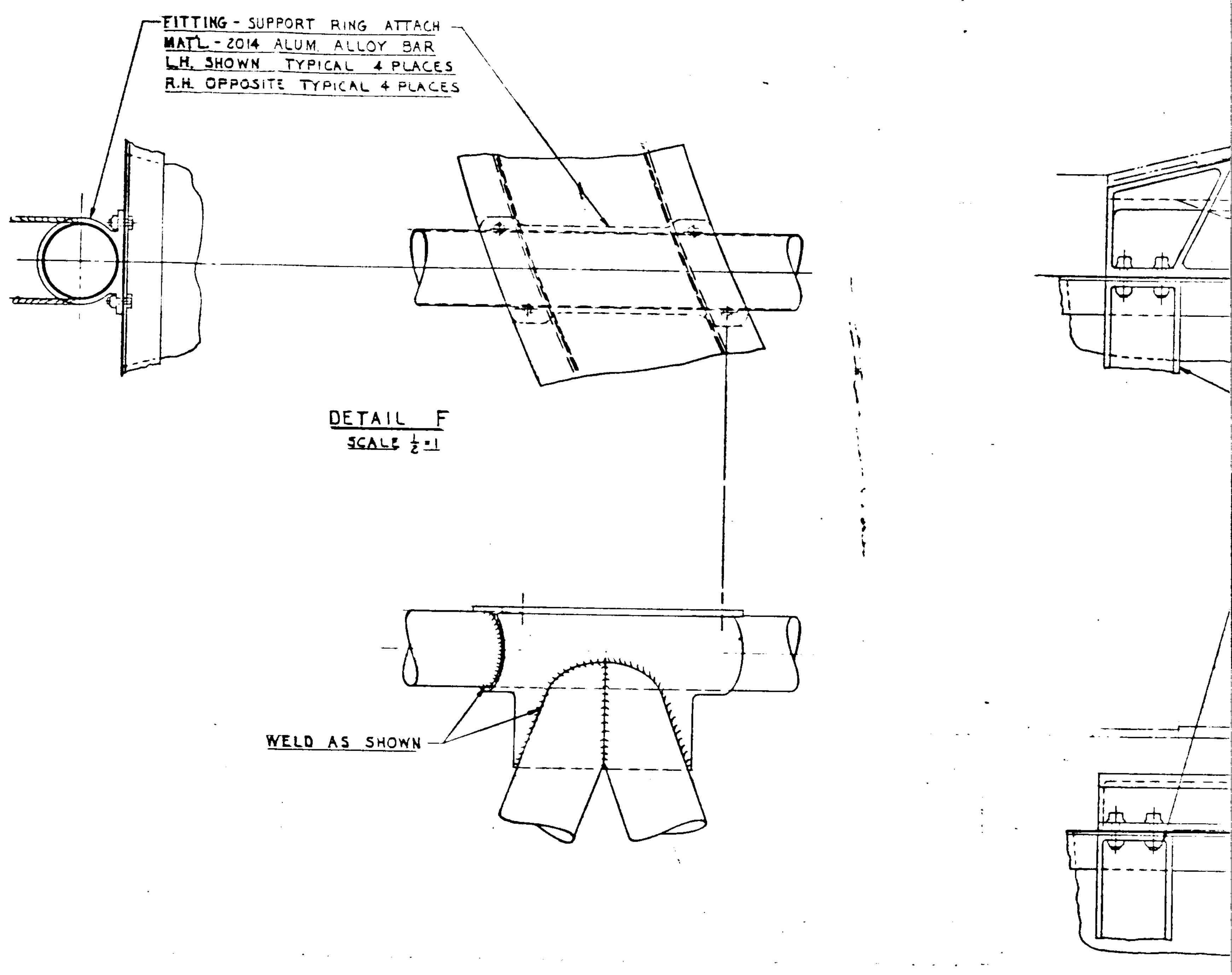
FOLDOUT FRAME 1

FOLDOUT FRAME 2



NOTE This drawing is reduced. Scale notations apply to original only.

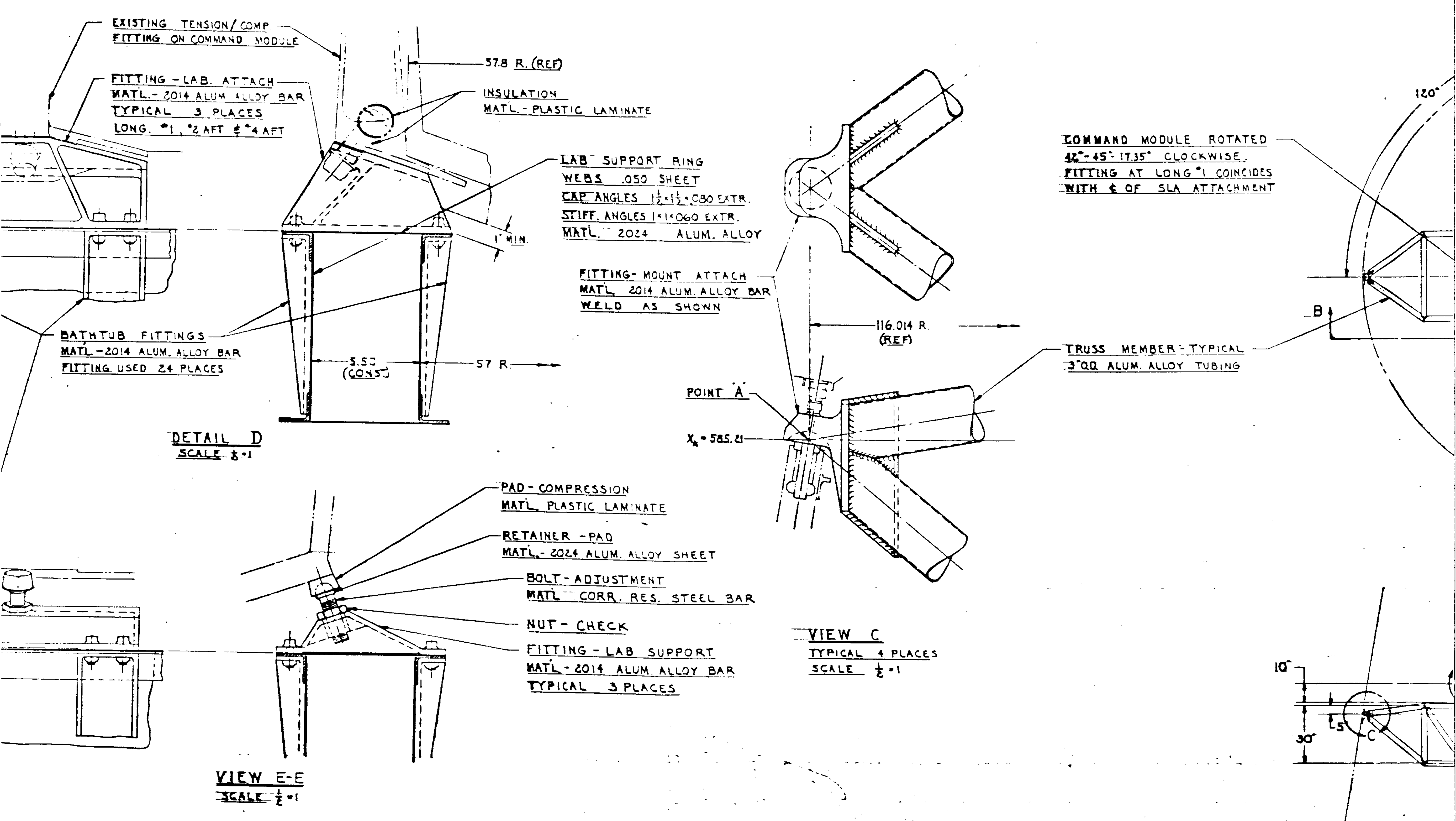
Figure 105. Laboratory/ Equipment Mount



DETAIL F
SCALE $\frac{1}{2}=1$

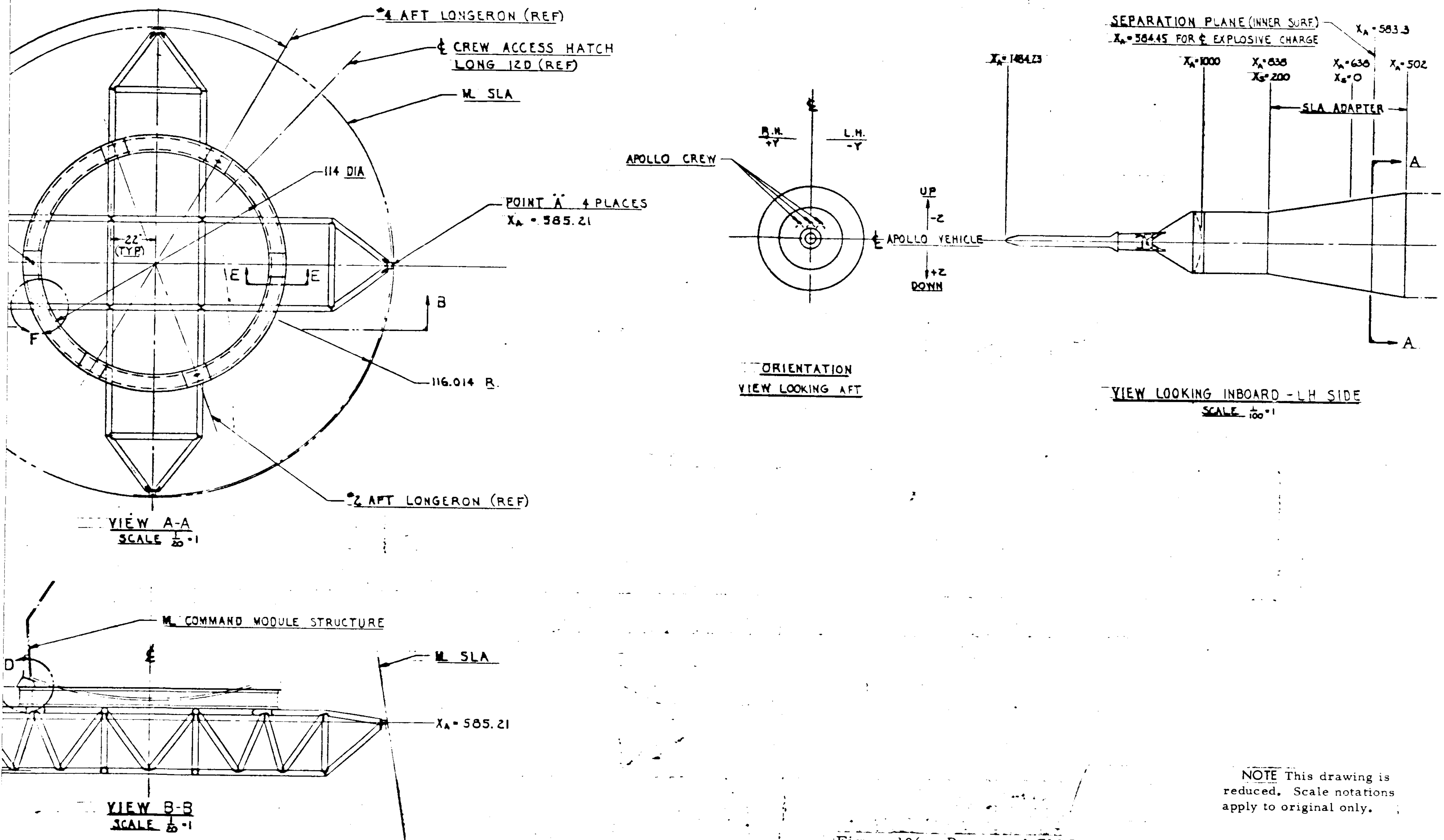
FOLDOUT FRAME 1

FOLDOUT FRAME 2



FOLDOUT FRAME

FOLDOUT FRAME 3



NOTE This drawing is reduced. Scale notations apply to original only.

Figure 106. Renovated CM Laboratory/ Equipment Mount (Truss Concept)

the fitting, without disturbing the basic structure. The fabrication of the web beam can be easily broken down to relatively small subassemblies, with the final assembly limited to the fit-up of the major mating points and the assembly of the components by clips and doublers. Therefore, the web beam structure was selected as the best approach for the laboratory mount in this study. The laboratory mount (Figure 104) is used on all RCM laboratory configurations, although the configuration may not represent the maximum design weight.

Airlock. The airlock design (Figure 103) is similar in concept to the airlock designed for COMLAB. The major differences are in the location of the ingress/egress hatch and the method of supporting the airlock assembly to the CM structure. The previous airlock egress hatch was at the bottom of the airlock and required a sizable mechanism to operate the hatch. In addition, the bottom location of the hatch prevented direct-view surveillance of the emerging astronaut by the spacecraft commander in the CSM.

The new airlock hatch is at the upper end and utilizes the entire hemispherical section as the cover. This permits a larger opening from which the suited astronaut with back pack can emerge. It also permits a simplification of manufacturing tooling, because the tools may be withdrawn easily from the large end opening. The portion of design not yet achieved is the latching and hinging technique. As may be noted in the launch configuration, there appears to be marginal clearance between the airlock and the Apollo SPS engine nozzle if conventional swing-type hinges are used. It is possible to design articulated hinges in which a parabolic opening path may be used to clear the nozzle if there is a requirement for on-the-pad access to the laboratory. The hinging mechanism must also provide a limited linear motion in order to properly engage and pull down the locking latches. The latches and mechanism must be located on the hatch to simplify the latching mechanism and to provide a smooth-edged opening to reduce the hazard of damaging an astronaut's space suit during ingress or egress.

The airlock is simply supported to the CM at the upper and lower ends by aluminum sheet webs, and a simple support strut is used to accommodate the boost-loading condition.

The entire airlock is covered with a thermal/micrometeoroid shield attached in the same manner as on the laboratory shell. The thermal insulation consists of multilayers of aluminized mylar attached to the 0.016 aluminum micrometeoroid shield by nylon support buttons spaced about 12 inches on center. This assembly is then mounted to the airlock with nonmetallic fittings that are adhesively bonded to the outer surface of the airlock. Screws and nylon bushings are used as fasteners to prevent compacting of the mylar at attach points.

Minimum-Delta Dependent Laboratory

A minimum-delta dependent RCM laboratory (Figure 107) differs in concept from the dependent laboratory in the number of internal systems and the self-contained environmental control system. It offers greater flexibility in the choice of missions and the capability of longer flight durations.

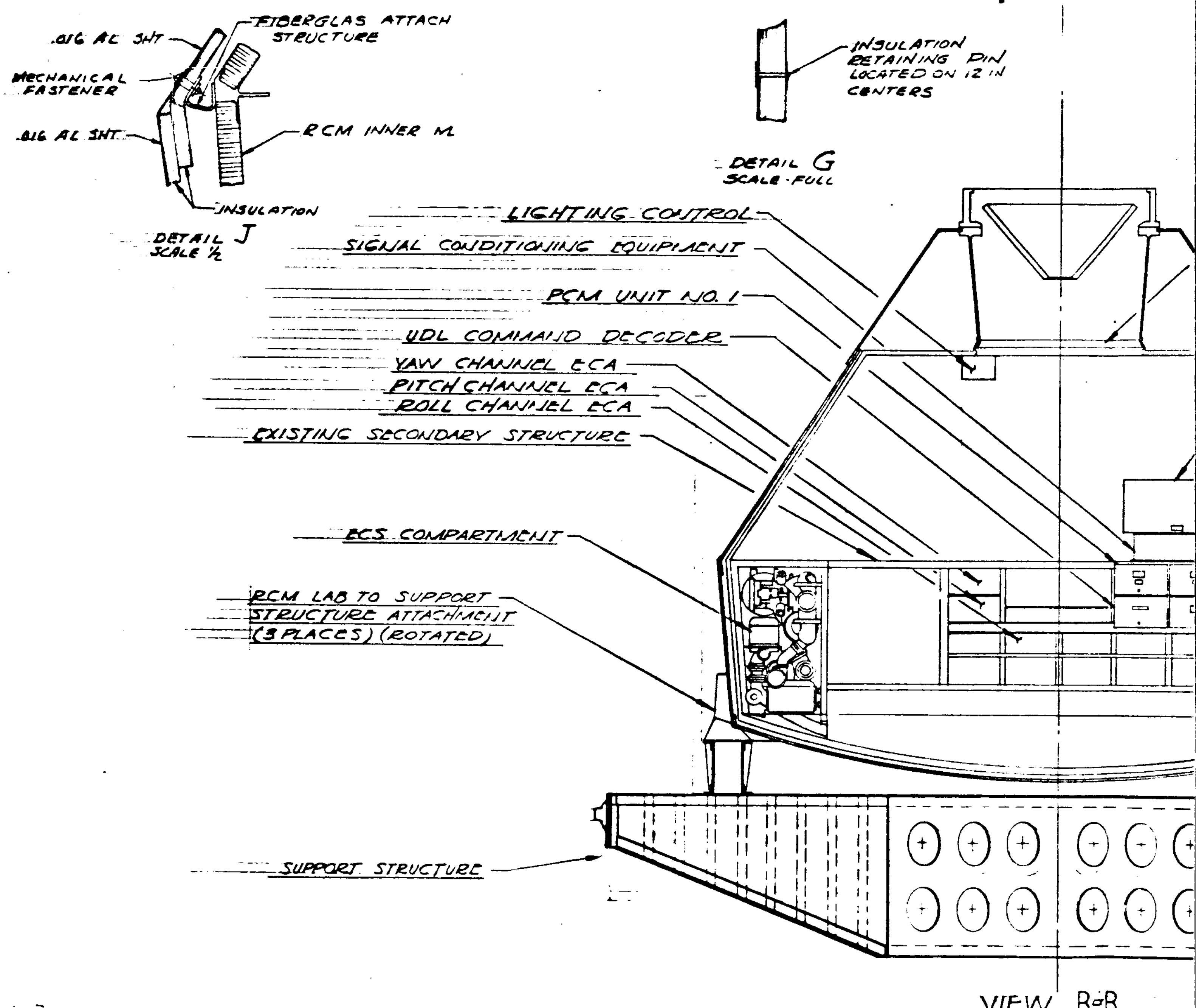
The primary and secondary laboratory structures, airlock assembly, docking provisions, and equipment/support beam are identical with the basic RCM laboratory (Figure 102). This configuration has the standard Block I ECS in the left equipment bay. The lower equipment bay contains selected communication and control equipment. The right equipment bay has electrical subsystems and a new main display panel where all the displays and controls necessary for the operation of the RCM laboratory and experiments are mounted for operation and viewing by a single crewman.

Additional CO_2 absorbers are stowed within the laboratory for use in the ECS of the laboratory, and extra ones could be transferred into the Apollo CSM if the need arose. Oxygen for environmental control and pressurization could be taken from the docked Apollo CSM supply via an external supply line or stowed in a special supply tank on the equipment beam of the RCM laboratory. Electrical power may be received via umbilicals within the docking tunnel, and external lines may be connected externally between the Apollo SM and the laboratory. The existing electrical feed-through for the umbilical from the SM on the Block I configuration is located 16 degrees off the -Z axis of the laboratory and can be used for electrical lines that are required to penetrate the laboratory walls.

Independent Laboratory—Block I CM

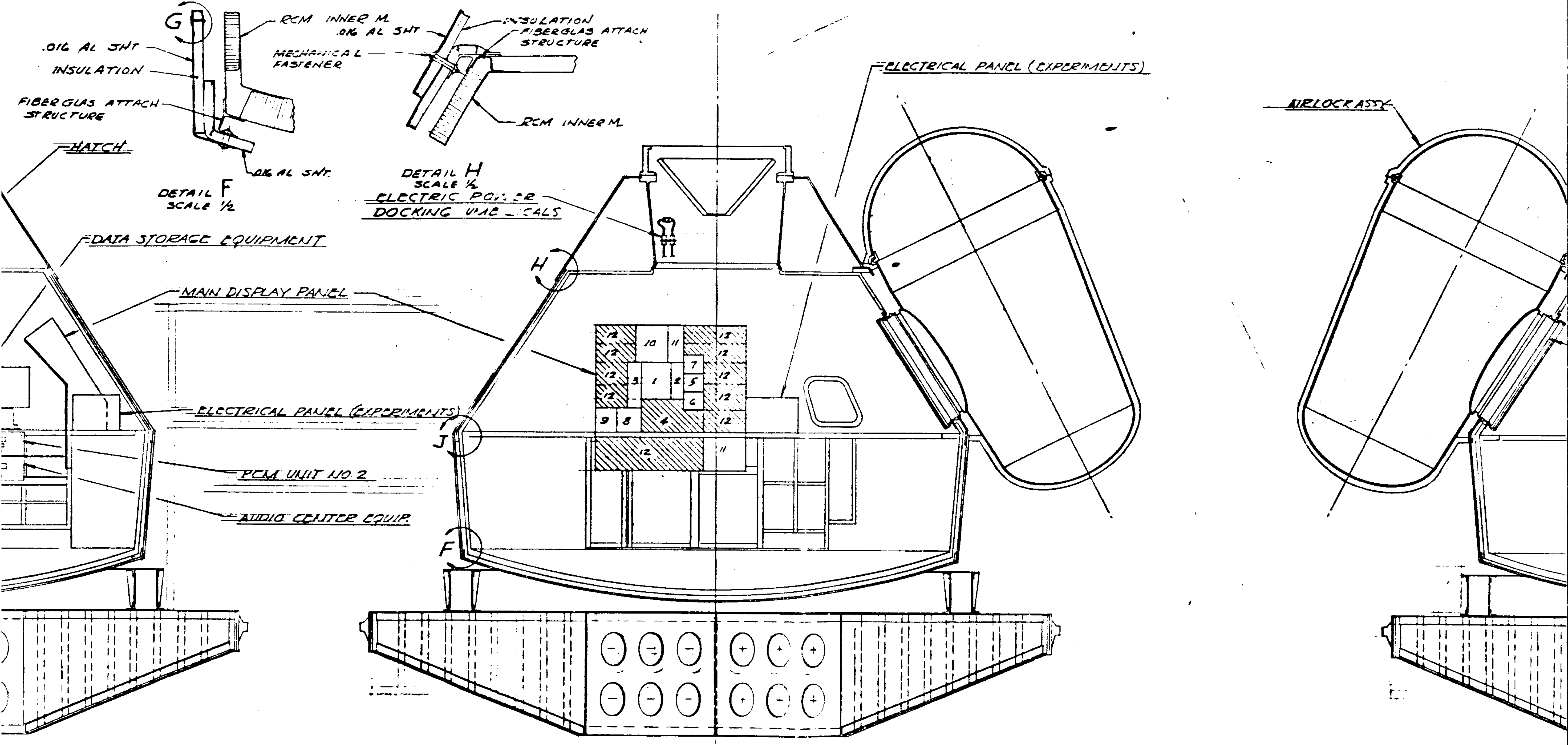
An independent RCM laboratory concept based on a modified Block I inner structure is shown in Figure 108. The basic assembly shown is the same renovated CM inner structure with protective covering, airlock, docking provisions, and equipment/support beam structure used for the dependent concept shown in Figure 102. Also, this design has all the systems and capability to perform a manned or unmanned mission. Three Apollo-type fuel cells are provided as the electrical power supply, with double the propellant capacity of a Block II SM. The four LO_2 and four LH_2 tanks are mounted within the structure of the equipment/support beam and protected by the laboratory above and insulation/micrometeoroid shielding on all other sides. The fuel cells are spaced 90 degrees apart and mounted directly to the equipment/support structure.

Four Apollo Block II SM RCS clusters are provided, equally spaced about the center of the vehicle for attitude stabilization and docking control.



FOLDOUT FRAME

FOLDOUT FRAME



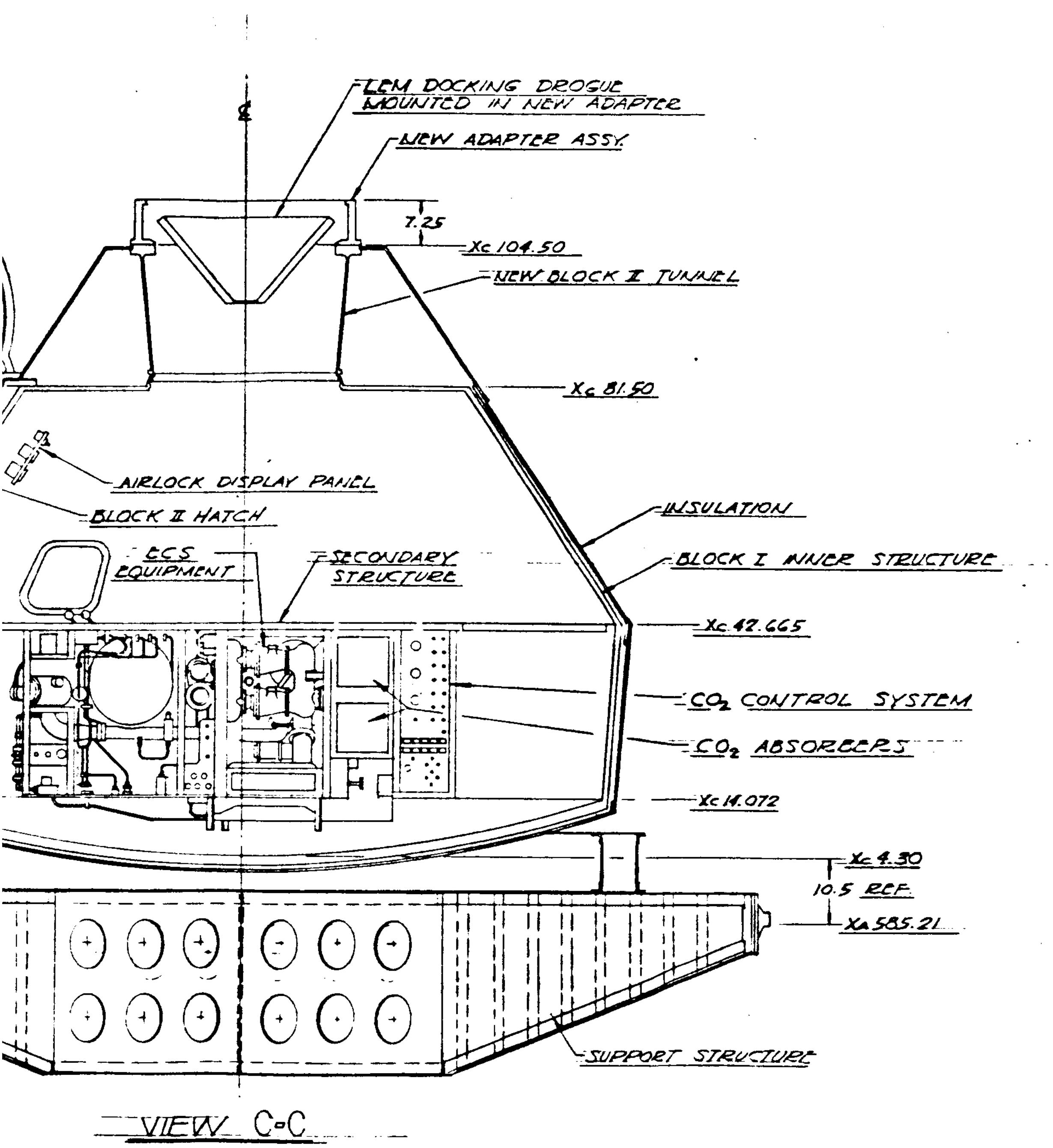
VIEW D-D

MAIN DISPLAY PANEL

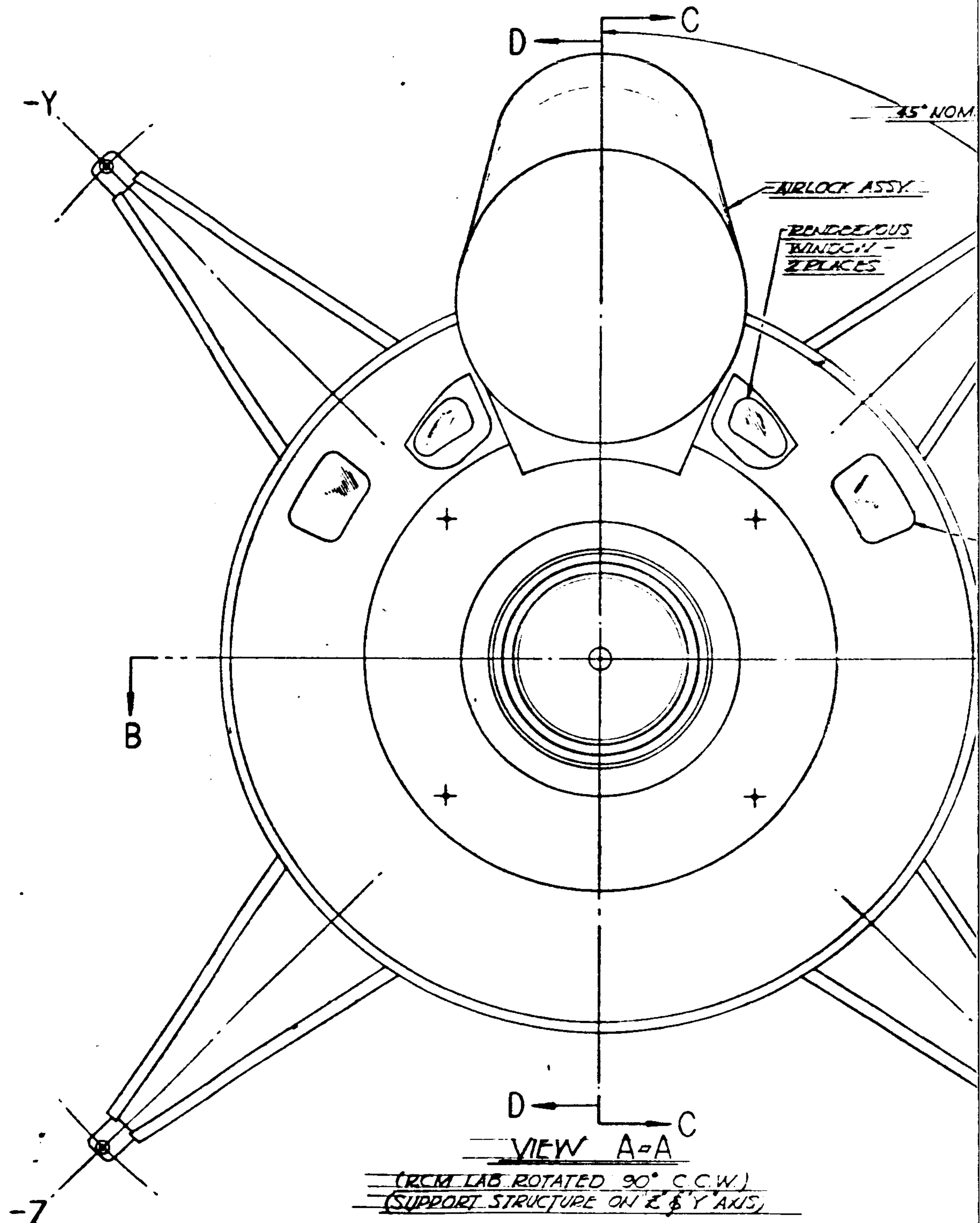
1. CLOSED CIRCUIT T/V DISPLAY
2. CLOSED CIRCUIT T/V CONTROLS
3. DATA SYS. & RECORDER DISPLAYS & CONTROLS
4. ASTRON. TELESCOPE MOUNT DISPLAYS & CONTROLS
5. WARNING & CAUTION SYSTEM
6. COMMUN. DISPLAYS & CONTROLS
7. AIRLOCK CONTROLS
8. POWER DISPLAYS & CONTROLS & LIGHTING CONTROLS
9. CIRCUIT BREAKER PANEL
10. S/C ATTITUDE TIME DISPLAYS
11. ECS DISPLAYS & MONITOR
12. INDIVIDUAL EXPERIMENT D&C

FOLDOUT FRAME 2

FOLDOUT FRAME 3



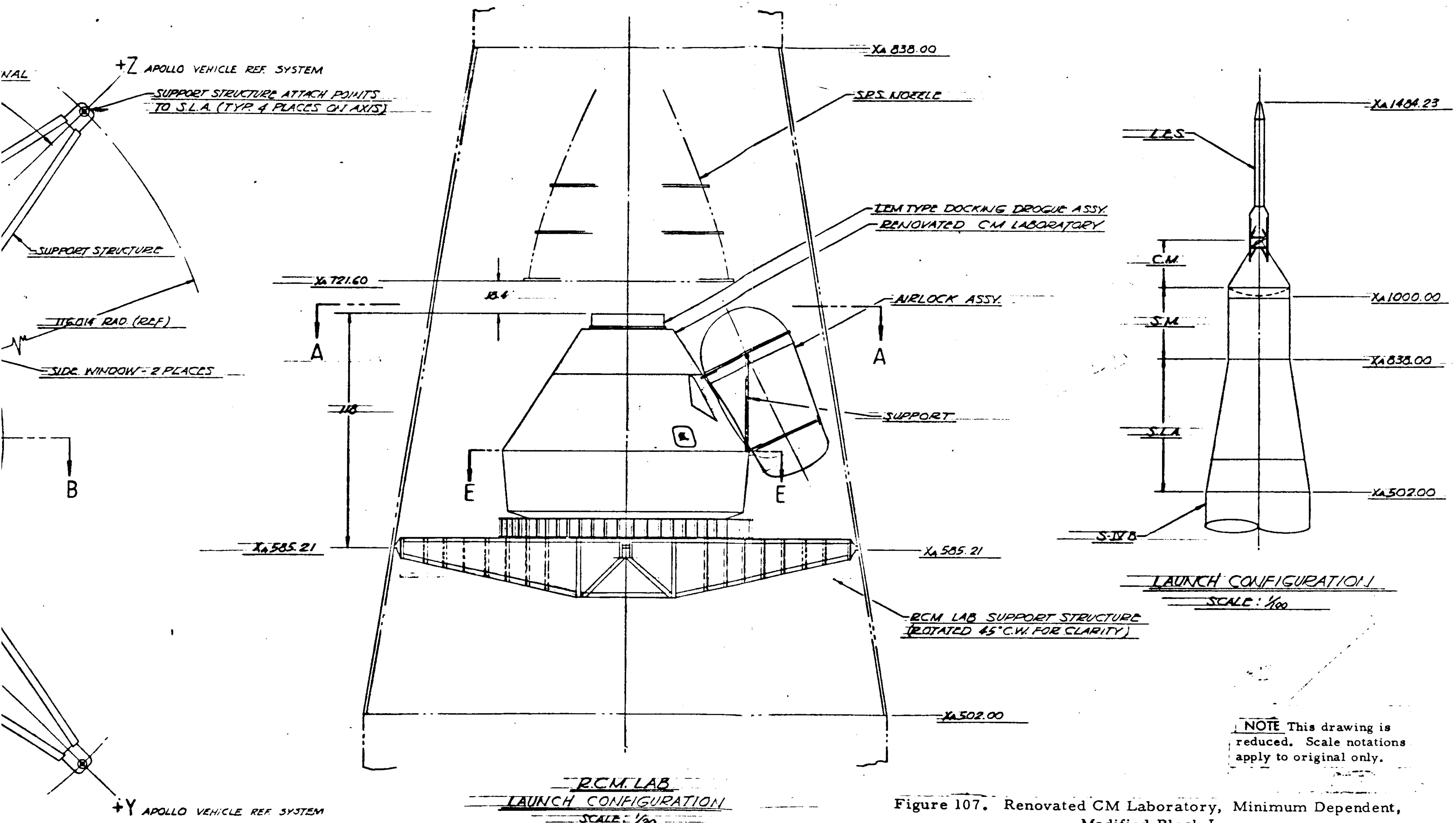
VIEW C-C



VIEW A-A
(RCM LAB ROTATED 90° C.C.W.)
(SUPPORT STRUCTURE ON Z & Y AXIS)

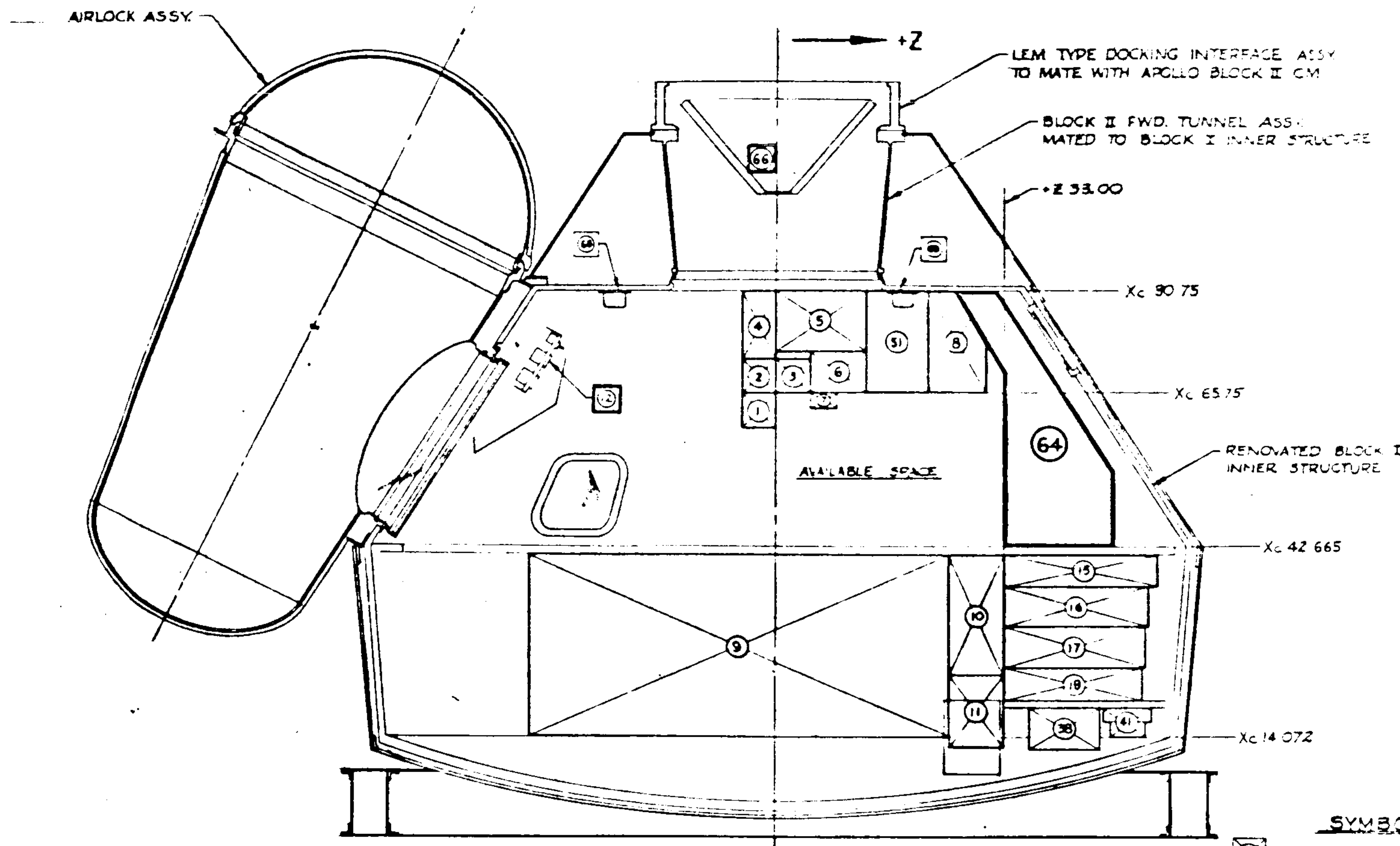
FOLDOUT FRAME 4

FOLDOUT FRAME 5



NOTE This drawing is reduced. Scale notations apply to original only.

Figure 107. Renovated CM Laboratory, Minimum Dependent, Modified Block I



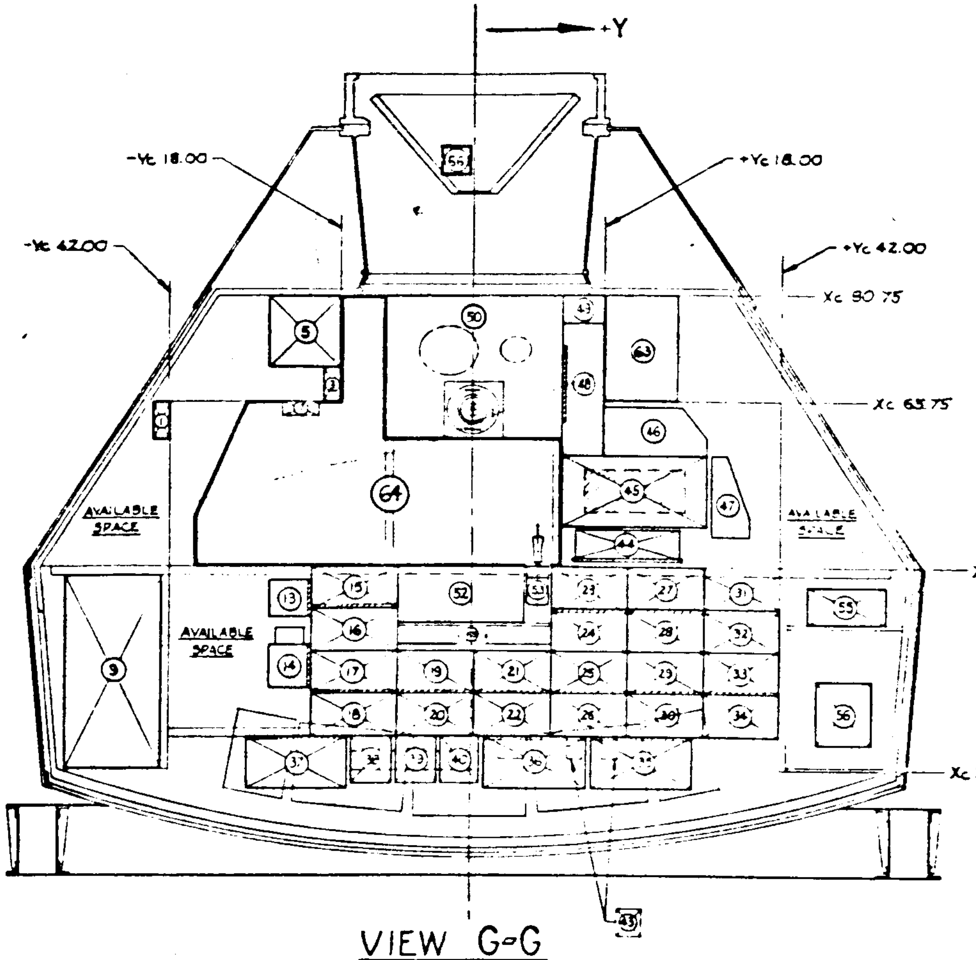
VIEW F-F
LOOKING AT LH EQUIP BAY

SYMBOLS
BLOCK I EQUIPMENT &
BLOCK I EQUIPME
NEW EQUIP
BLOCK

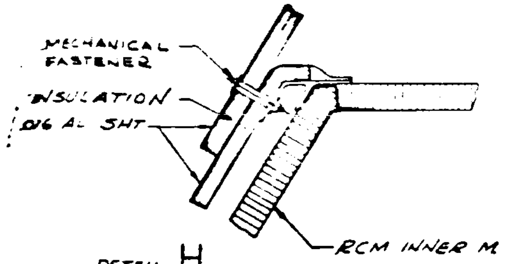
1. PRESSURE SUIT CONNECTOR VALVE
2. " " " "
3. PRESSURE SUIT CONNECTOR VALVE
4. PERSONAL COMMUNICATIONS CONNECTOR
5. CABIN AIR RECIRCULATION & PRESSURIZATION UNIT
6. CABIN TEMP CONTROL
7. WATER DELIVERY ASSY.
8. CLOCK & EVENT TIMER PANEL
9. ENVIRONMENTAL CONTROL SYSTEM
10. OXYGEN CONTROL PANEL
11. WATER VALVE PANEL
12. AIRLOCK CONTROL PANEL
13. RATE GYRO ASSY.
14. ATTITUDE GYRO & ACCELEROMETER ASSY.
15. GYRO/ACCELEROMETER ELECTRONIC CONTROL ASSY
16. YAW CHANNEL ELECTRONIC CONTROL ASSY.
17. AUXILIARY ECA
18. PITCH CHANNEL ECA
19. ROLL CHANNEL ECA
20. SPARE
21. UNIFIED S-BAND EQUIPMENT
22. UP-DATA LINK
23. PCM UNIT I TELEMETRY EQUIPMENT
24. TELEMETRY EQUIPMENT
25. S-BAND POWER AMPLIFIER
26. SPARE
27. PCM UNIT 2 TELEMETRY EQUIPMENT
28. VHF MULTIPLEXER
29. VHF/AM TRANSMITTER - RECEIVER
30. VHF/FM TRANSMITTER & HF TRANSCEIVER
31. DATA WIR
32. PRE-MODU
33. AUDIO CE
34. CENTRAL
35. POWER STA
36. "
37. POWER STA
38. BATTERY
39. BATTERY
40. BATTERY
41. AC POWE-
42. BATTERY
43. POWER TR
44. SIGNAL CO
45. DATA STO

FOLDOUT FRAME 1

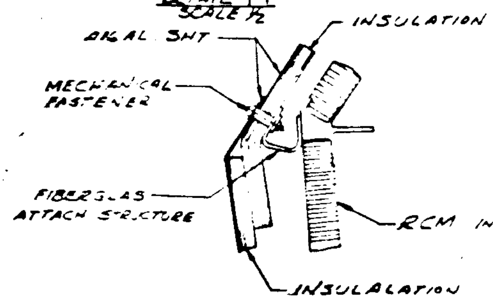
FOLDOUT FRAME 2



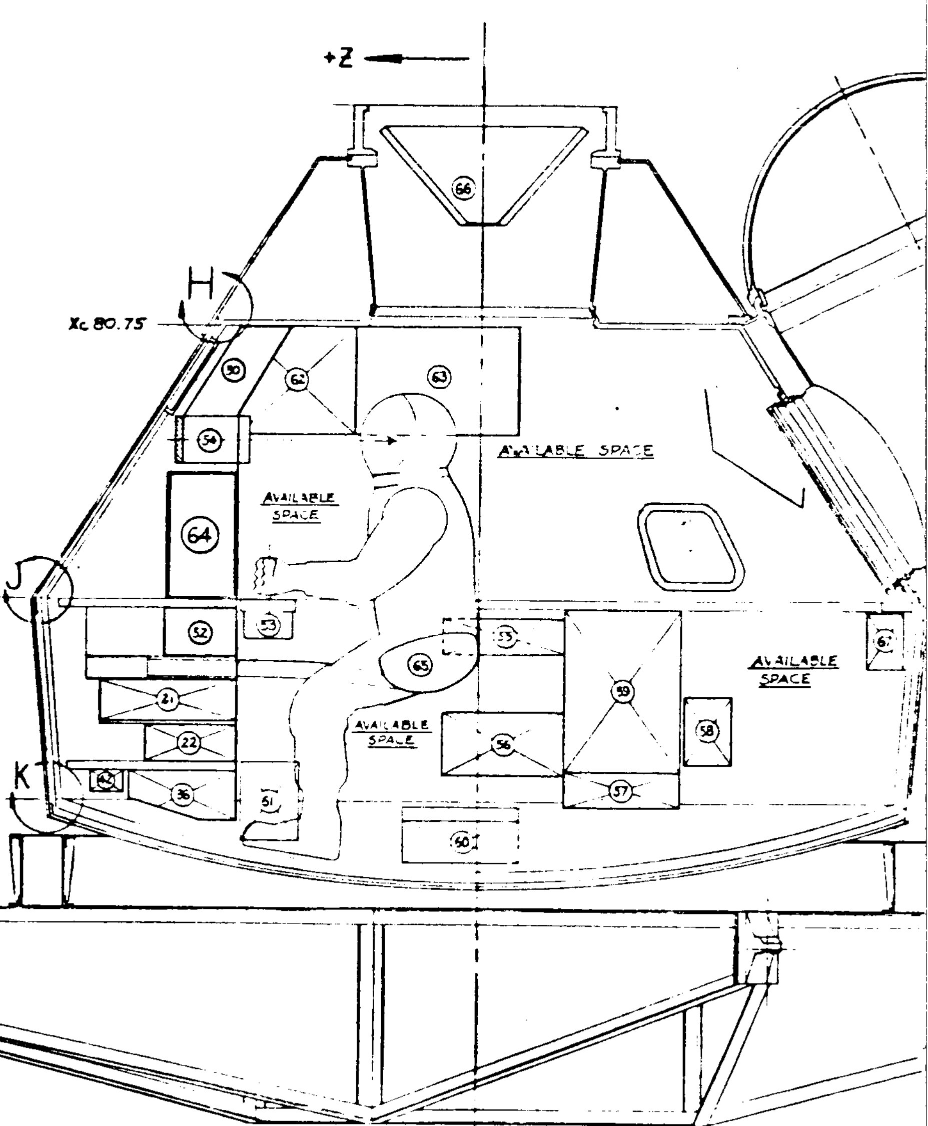
VIEW G-G
LOOKING AT LOWER EQUIP. BAY



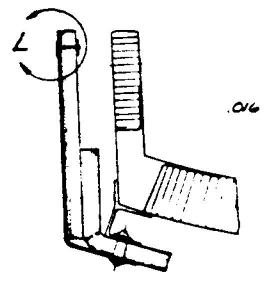
DETAIL H
SCALE ½



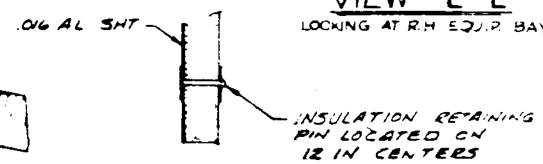
DETAIL J
SCALE ½



VIEW E-E
LOOKING AT R.H. EQUIP. BAY



DETAIL K
SCALE ½



DETAIL L
SCALE FULL

...LOCATION
...NT RELOCATED
...ENT
... I COLDPLATES

...NG
...LATION PROCESSOR
...TER
...TIMING
...TIC INVERTER

...TIC INVERTER

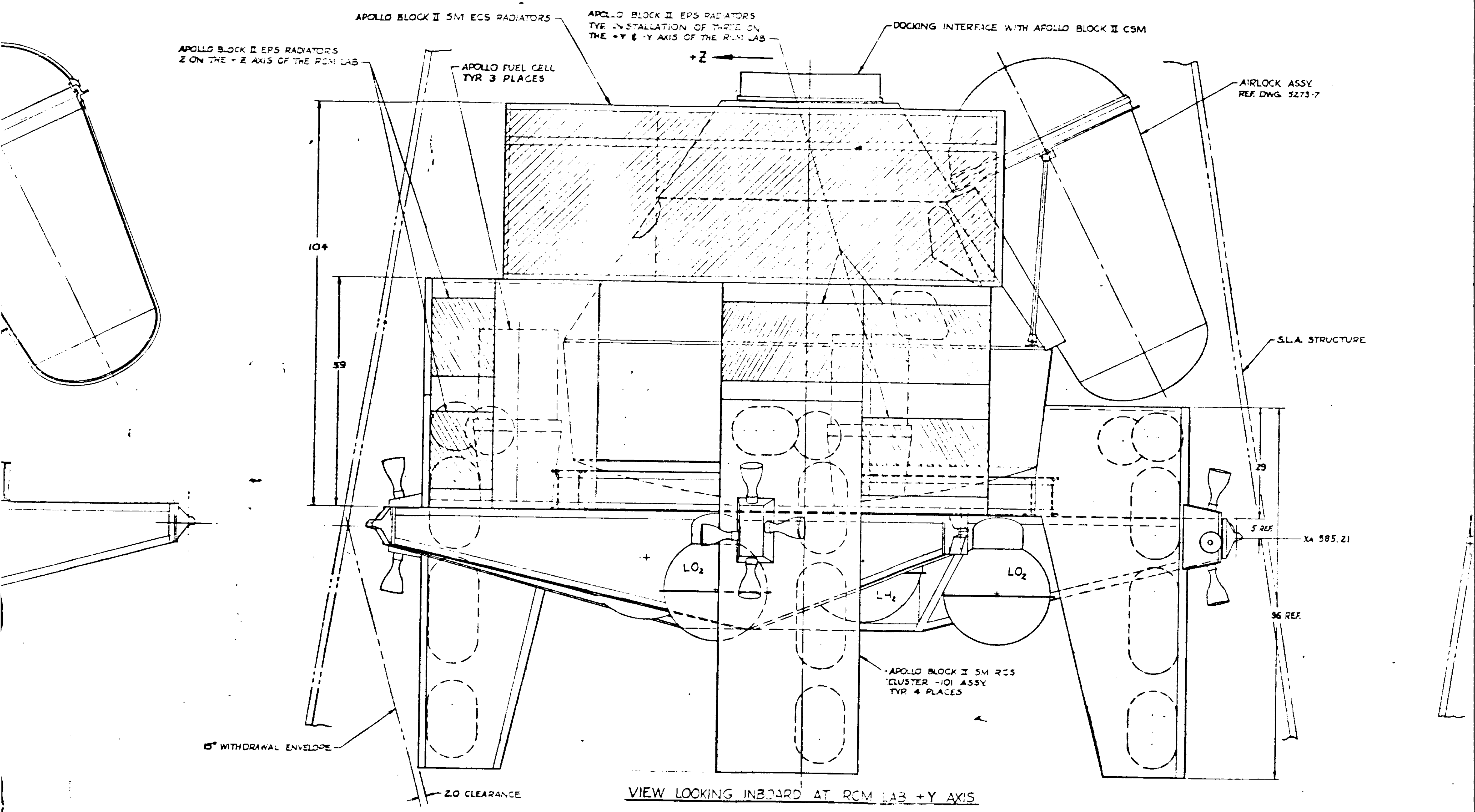
... CONTROL PANEL
...CHARGER
...ANSFER SWITCH
...NDITIONING EQUIPMENT
...AGE EQUIPMENT

46 ECS CONTROLS & DISPLAYS
47 DOCKING CONTROLS & DISPLAYS
48 CAUTION & WARNING
49 LIGHTING CONTROL
50 SCS CONTROLS & DISPLAYS
51 COMMUNICATION / ANTENNA CONTROLS & DISPLAYS
52 RCS CONTROLS & DISPLAYS
53 ROTATION CONTROL ASSY
54 FDAI DISPLAY INSTRUMENT
55 DC POWER PANEL
56 ELECTRICAL POWER CIRCUIT BOX
57 CIRCUIT INTERRUPTER
58 FUSE BOX
59 CO_2 ABSORBER STORAGE
60 CO_2 ABSORBER STORAGE

61 CO_2 ABSORBER STORAGE
62 INFLIGHT TEST PANEL
63 EPS CONTROLS & DISPLAYS
64 RCM EXPERIMENT CONTROLS & DISPLAYS
65 CREWMAN SEAT & RESTRAINT ASSY.
66 LEM TYPE DOCKING DROGUE
67 CM/SM ELECTRICAL UMBILICAL PLATE
68 FLOODLIGHTS

FOLDOUT FRAME

FOLDOUT FRAME 3

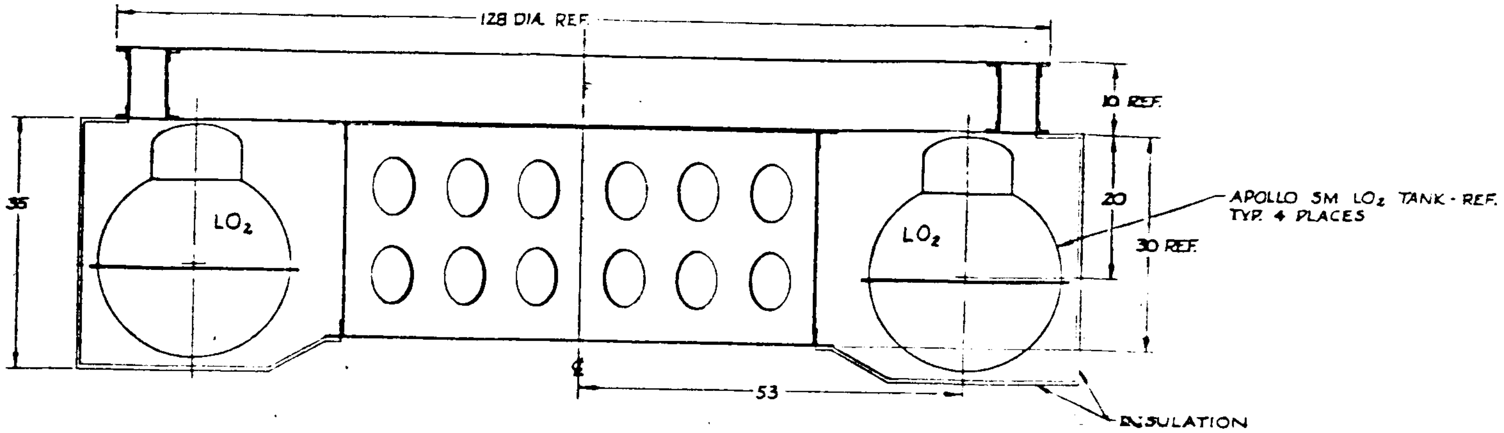


VIEW LOOKING INBOARD AT RCM LAB +Y AXIS

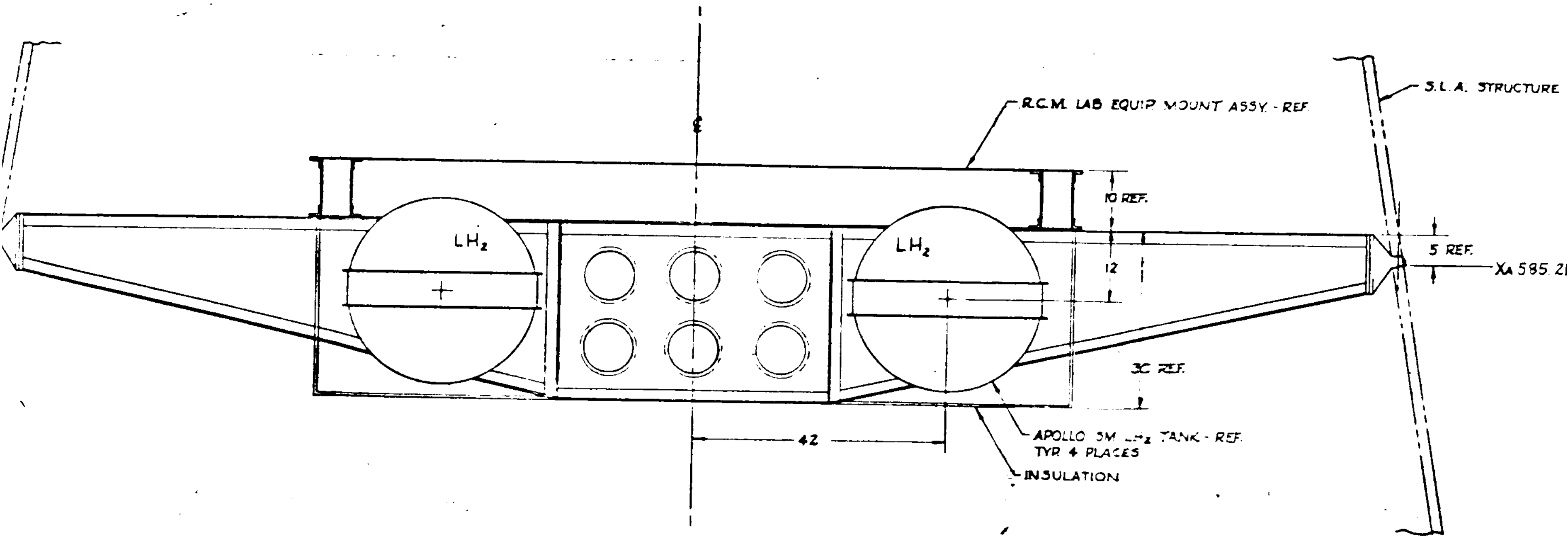
FOLDOUT FRAME 4

FOLDOUT FRAME 5

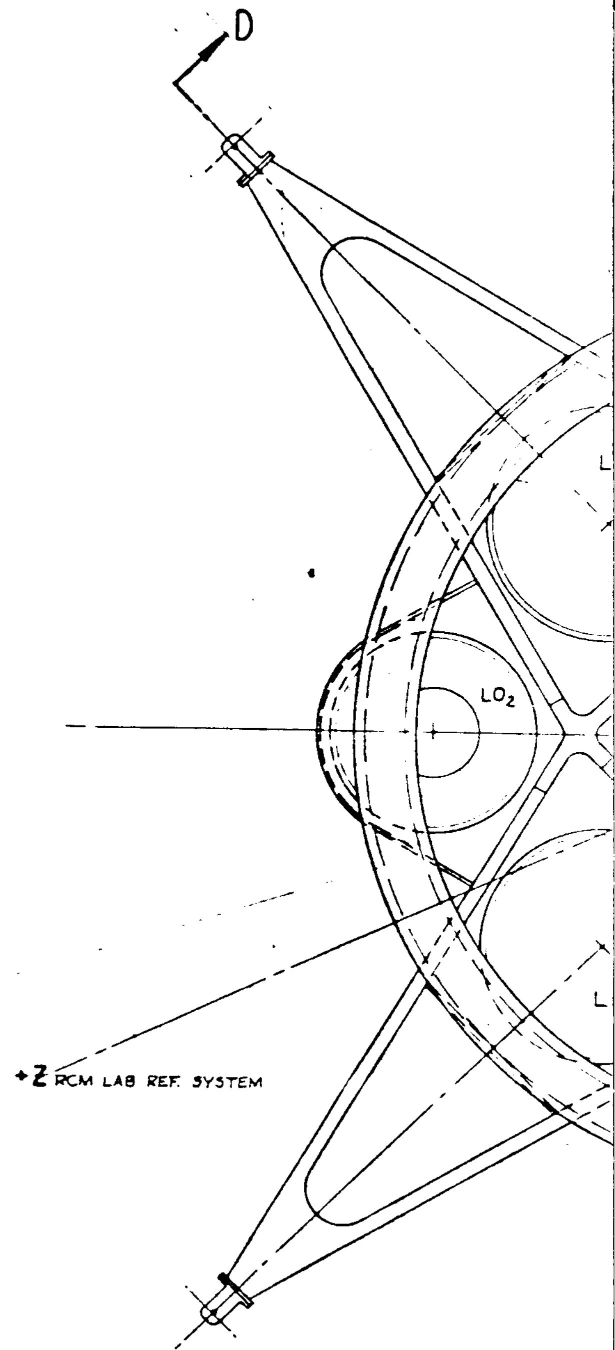
7



SECTION C-C
VIEW ROTATED 90° C.C.W.

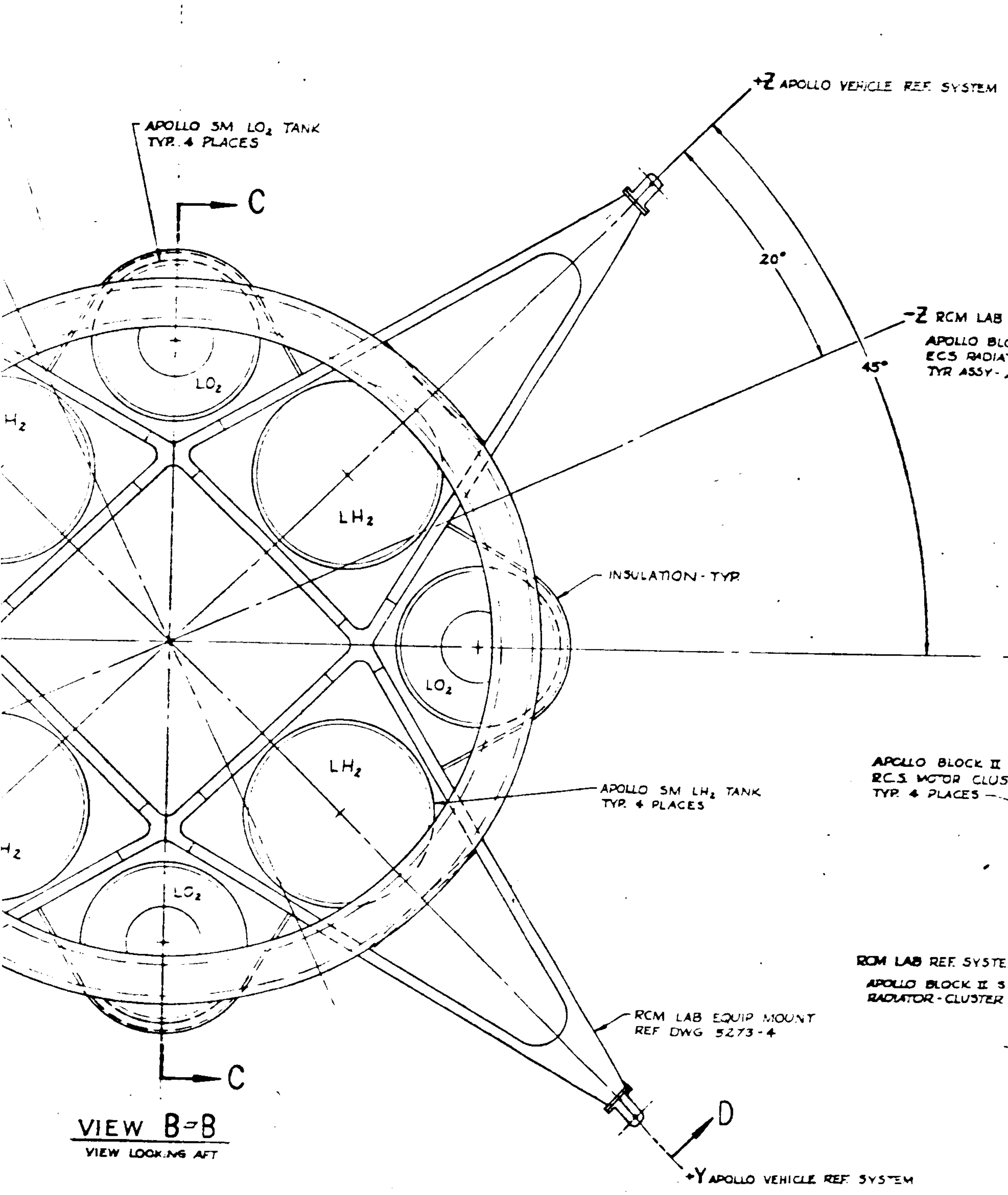


SECTION D-D
VIEW ROTATED 45° C.C.W.

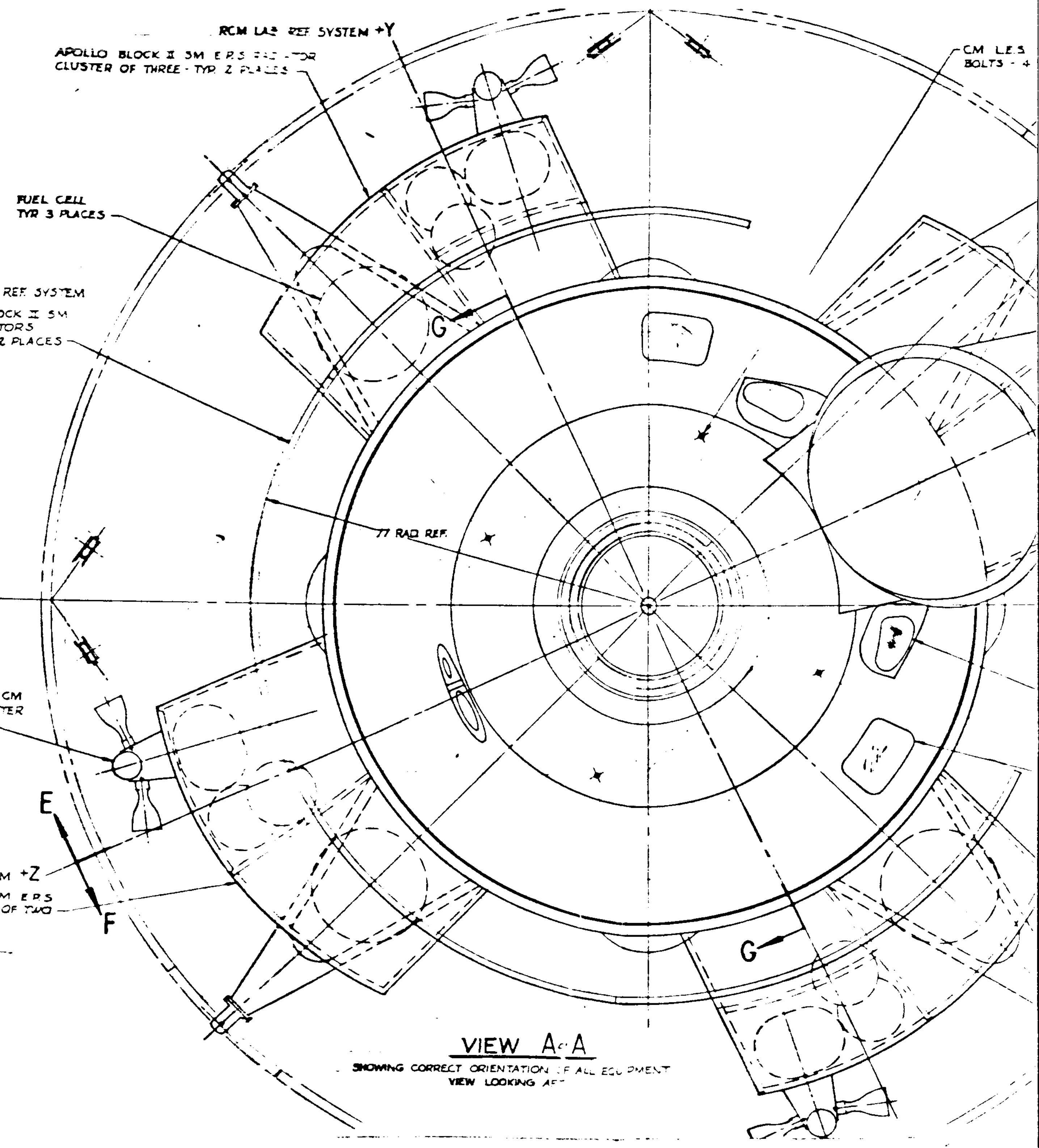


FOLDOUT FRAME 6

FOLDOUT FRAME 7



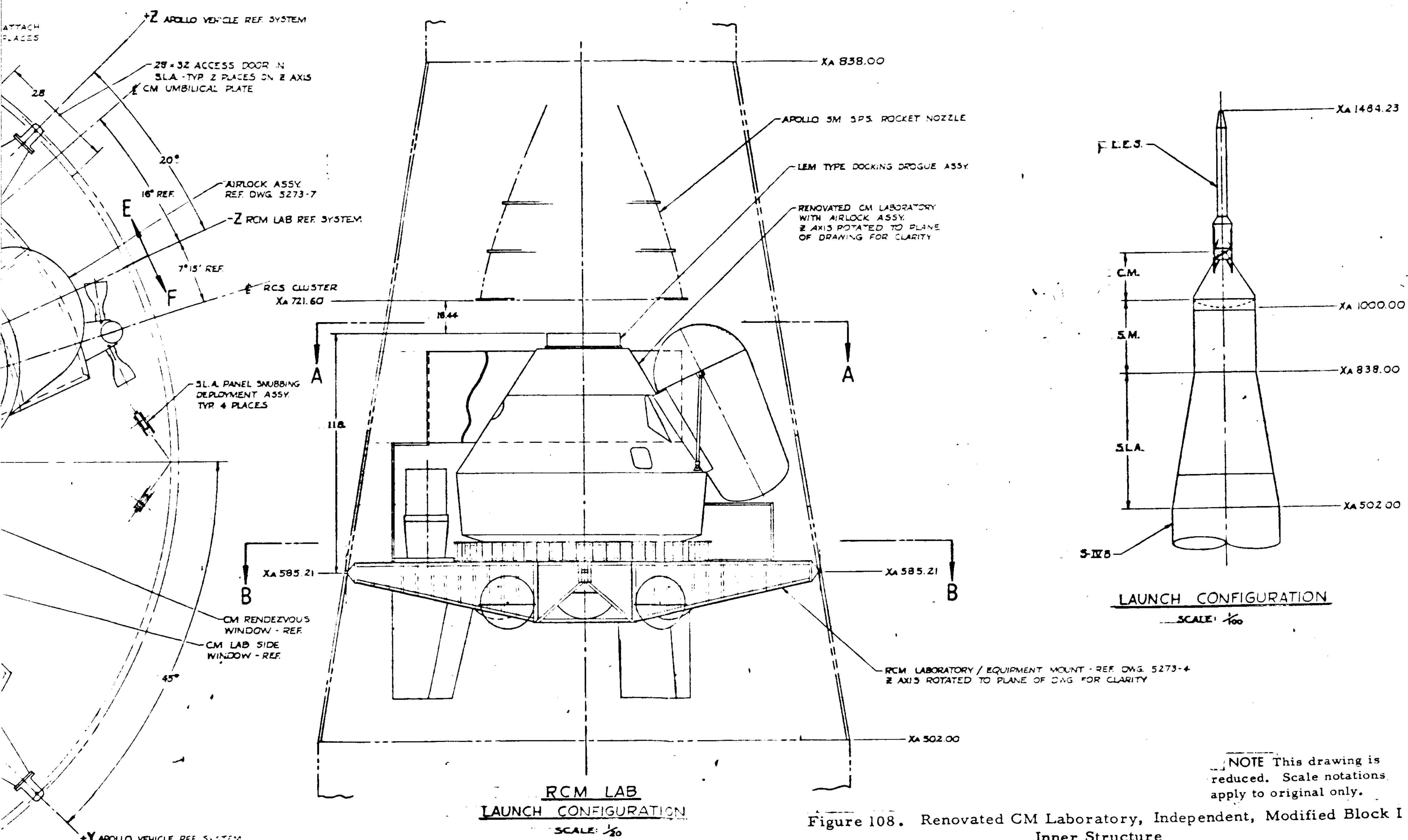
VIEW B-B
VIEW LOOKING AFT



VIEW A-A
SHOWING CORRECT ORIENTATION OF ALL EQUIPMENT
VIEW LOOKING AFT

FOLDOUT FRAME 8

FOLDOUT FRAME



NOTE This drawing is reduced. Scale notations apply to original only.

Figure 108. Renovated CM Laboratory, Independent, Modified Block I Inner Structure

The RCS motor clusters are identical in configuration and constructed from the -101 assembly of the Block II SM. The RCS assemblies project above and below the equipment/support beam and are located so that a 15-degree withdrawal angle is provided to ensure safe separation of the RCM laboratory from the SLA in flight. On the Y axis of the lab are located a total of six Block II SM EPS radiators, three facing outboard in each direction, above the RCS panels. On the +Z axis are positioned two EPS radiators with space for a third should it be required. Above these installations are located two Block II ECS radiators on the Y axis of the lab, one facing outboard in each direction. These radiators are supported by the equipment/support beam and stabilized by additional truss structures attached to the four LES tower leg fittings on the RCM laboratory forward bulkhead. The existing umbilical plate feed-through located 16 degrees off the -Z axis of the laboratory is used for electrical and hard-line penetration of laboratory walls.

This configuration differs in arrangement from the simpler concepts in that the Z axis of the laboratory is only 20 degrees off the Z axis of the Apollo SLA axis. The installation of RCS clusters, fuel cells, and airlock assembly results in selection of this orientation so that existing personnel access openings on the Z axis of the SLA can be used, and there will be adequate clearances provided between the RCM laboratory and the SLA panel snubbing located 45 degrees off the major axis of the SLA.

The interior of this laboratory contains much of the original Block I equipment and systems that were renovated and reinstalled. Where possible, this equipment has been left in a Block I arrangement mounted on existing coldplates. The lower equipment bay was selected as the main display and control area. The equipment removed from the main display panel of the CM has been almost completely relocated in the lower equipment bay together with new equipment needed for experiments and laboratory control in an arrangement that allows one crewman to reach them from the position shown.

Independent Laboratory—Block II CM

The renovated CM laboratory configuration (Figure 109) consists of an Apollo Block II CM inner structure and subsystems arranged in a similar manner to the concept shown in Figure 108. The protective cover is the same as on previous configurations, as are airlock and docking provisions. The equipment/support beam (Figure 104) is modified by the addition of a thick honeycomb floor on top of the cross-beam and under the support ring structure. This vehicle is designed to perform a mission either manned or unmanned as an independent laboratory. The Z-axis of the laboratory is located 45 degrees off the Z-axis of the SLA in the same orientation that the basic dependent laboratory is positioned.

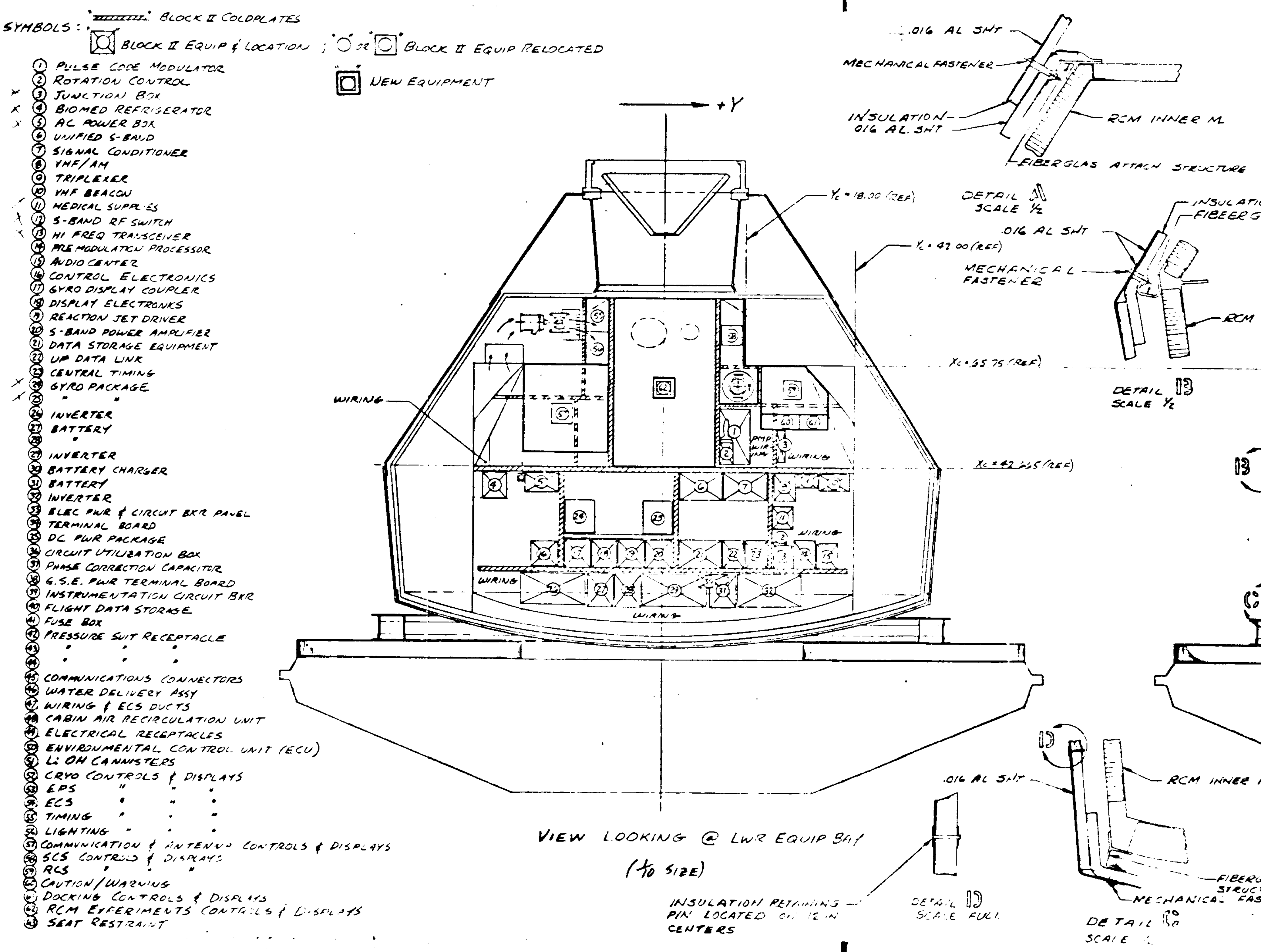
The RCS clusters are Block II SM -102 assemblies that are all the same and have insulation and structure added to protect the otherwise exposed pressurant and propellant tanks. These four clusters are positioned directly on the Z and Y axis of the laboratory instead of being located 7 degrees 15 minutes off those axes as they are on the Apollo SM. The electric power system has three Apollo fuel cells located on the opposite side of the laboratory from the airlock. These are protected and insulated, but are readily accessible for maintenance on the launch pad. The existing umbilical feed-through plate in the wall of the laboratory structure is located 2 degrees off the +Z axis of the laboratory, and conveniently close to the fuel cell installation. This arrangement will permit minimum length electrical leads and an optimum electric distribution system.

Four LH_2 tanks are located between the legs of the equipment/support beam. Two LO_2 tanks are also mounted within the confines of the beam, while the other two LO_2 tanks are mounted above the beam on the honeycomb floor, one on each side of the airlock. All tanks are insulated and shielded from micrometeoroid penetration. The eight Block II EPS radiators are positioned half on each side of the Z-axis of the laboratory. The two Block II ECS radiators, are located directly above the EPS radiators, one on each side of the Z-axis. The ten radiators are mounted on a tubular truss space-frame that is in turn anchored to the equipment/support beam and honeycomb floor.

Some of the secondary structure and much of the internal equipment inside the Block II CM is different in detail from similar equipment and installations in a Block I command module. This design attempts to make maximum utilization of existing coldplates and mounting structure within the renovated Block II CM inner structure and, therefore, has a different internal arrangement from the Block I inner structure shown in Figure 108. Both the existing and the new controls and displays for the experiments have been arranged about the lower equipment bay so they can be operated by a single crewman. This configuration has very limited visibility from the windows of the laboratory and, therefore, a new conceptual arrangement was prepared (Figure 109) as a "maximum visibility" version. The EPS radiators are not in optimum locations on this configuration and the RCS clusters must be redesigned to fit as shown.

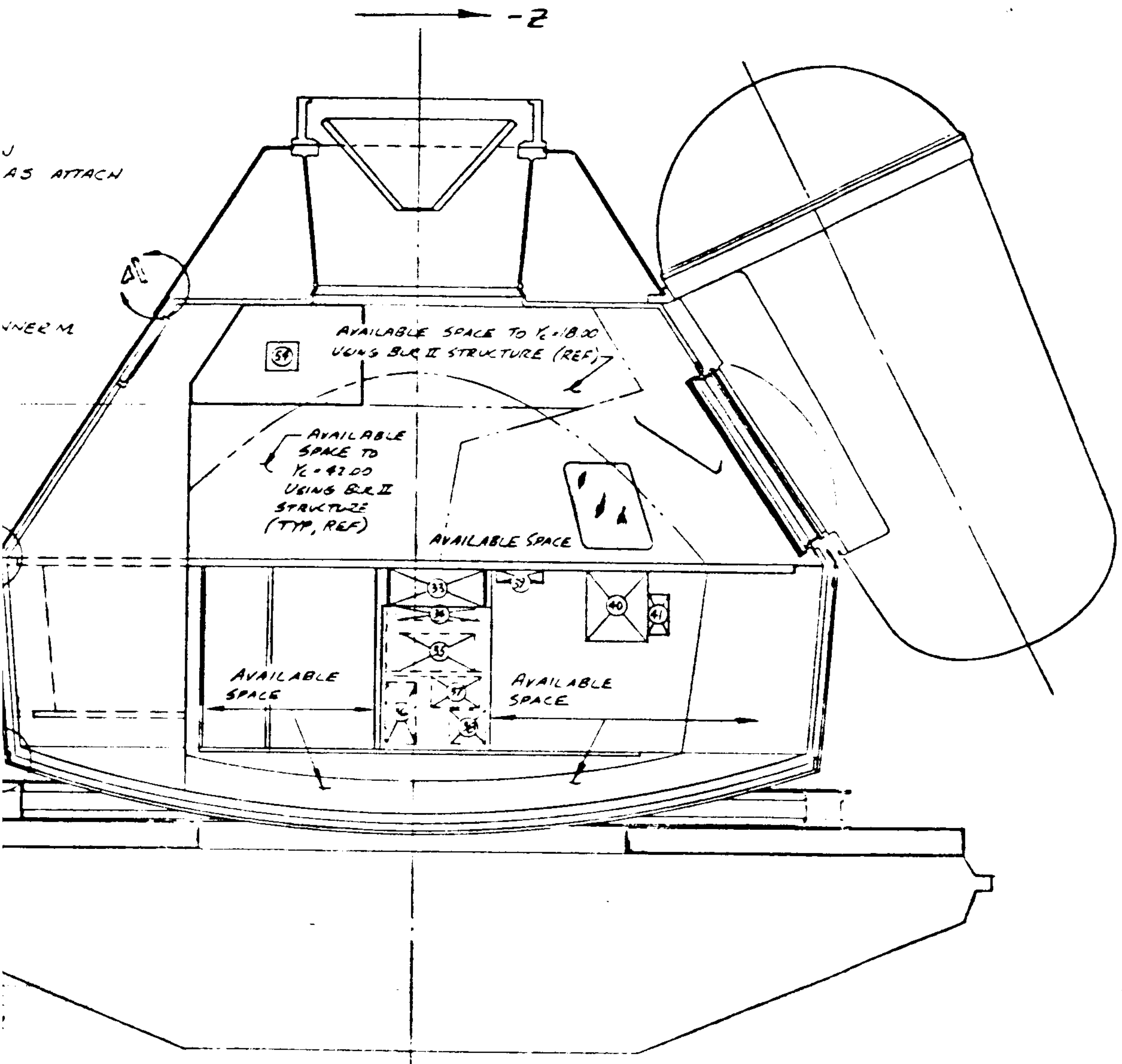
Integrated Independent Laboratory

The RCM laboratory configuration (Figure 110) represents the final effort of this program to design a laboratory using existing equipment that is compact, functional, and has good visibility from all windows. The laboratory inner structure, protective covering, airlock, docking provisions, and equipment/support beam are the same as in earlier configurations. The Z-axis of the laboratory is 20 degrees off the Z-axis of the SLA, and the



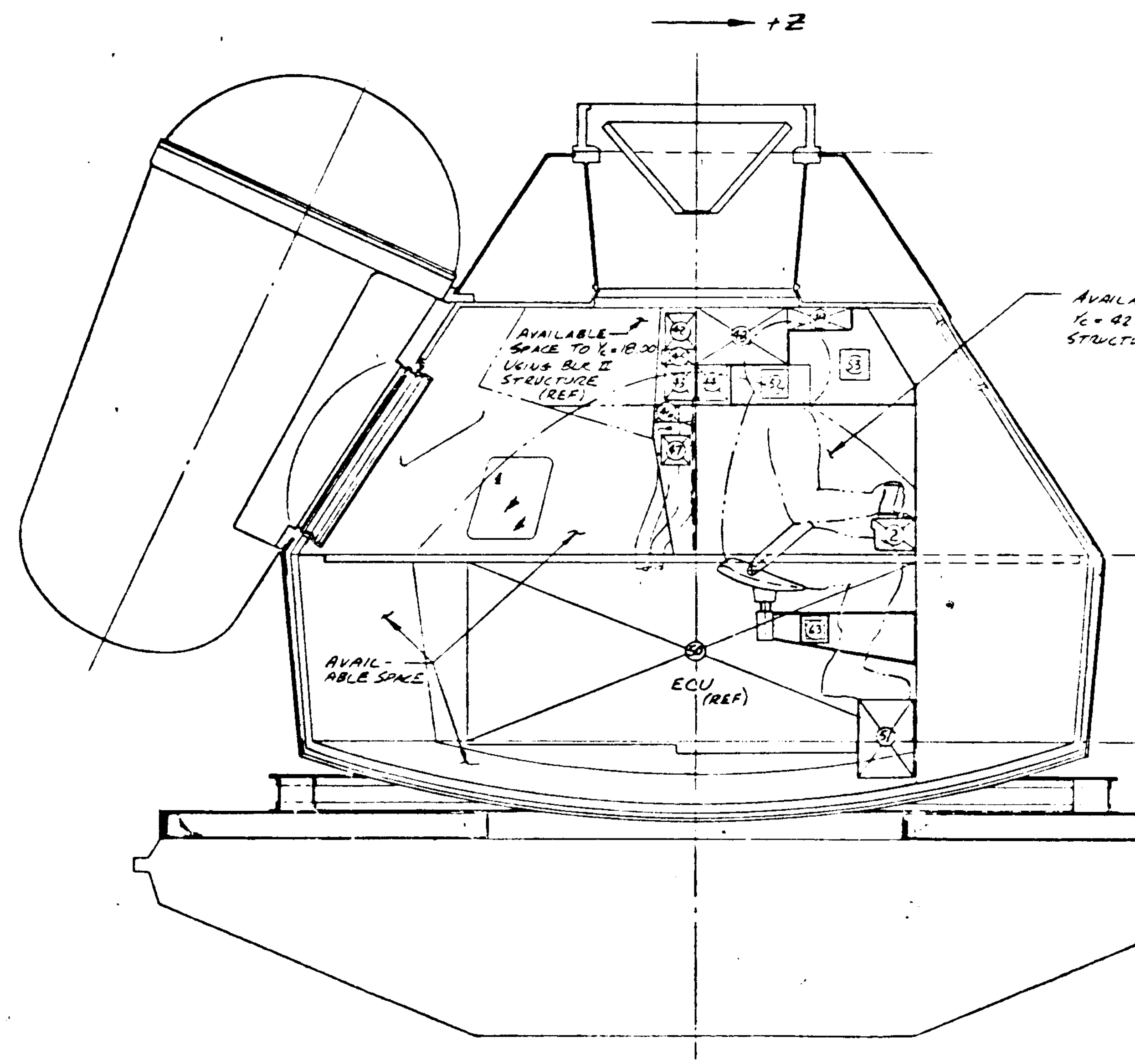
FOLDOUT FRAME 1

FOLDOUT FRAME 2



VIEW LOOKING @ R.H. EQUIP BAY

($\frac{1}{10}$ SIZE)



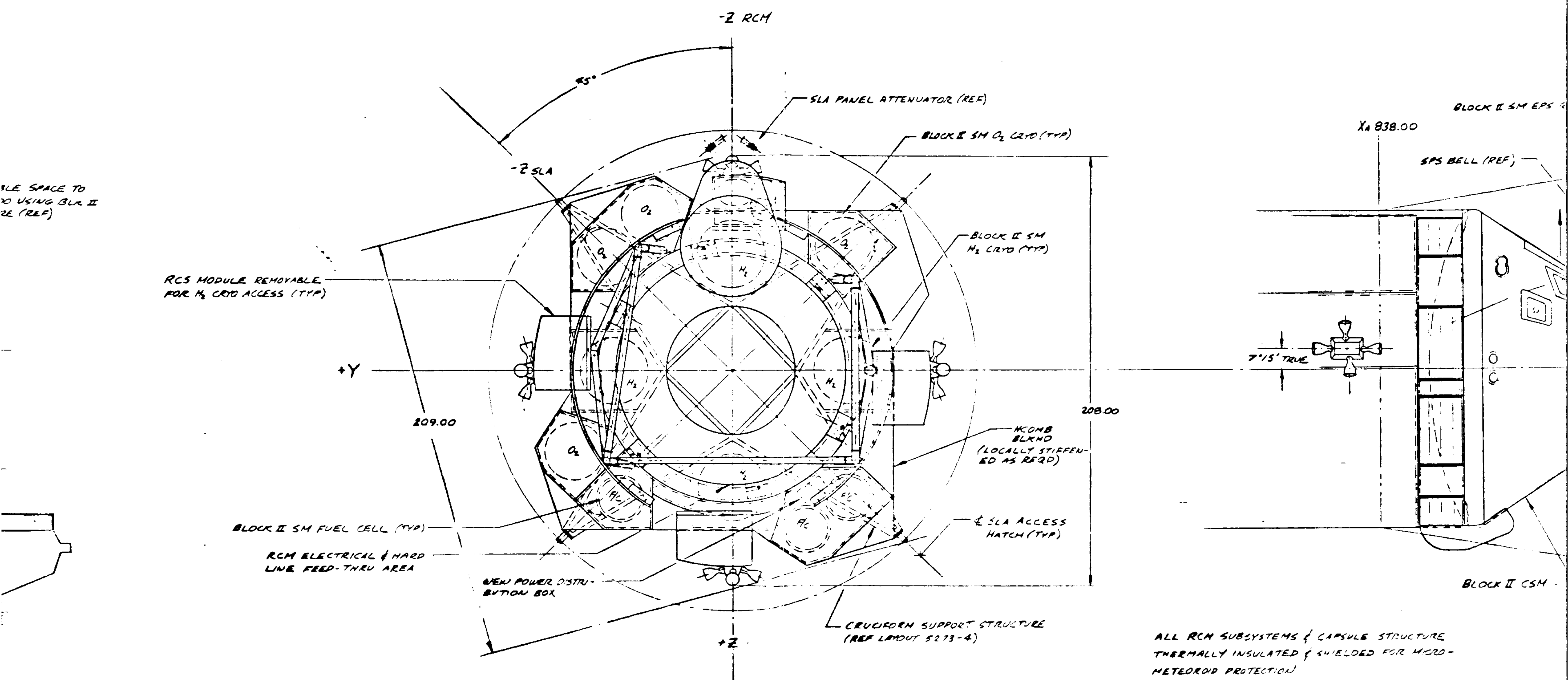
VIEW LOOKING @ L.H. EQUIP BAY

($\frac{1}{10}$ SIZE)

Foldout FRAME 3

FOLDOUT FRAME

FOLDOUT FRAME

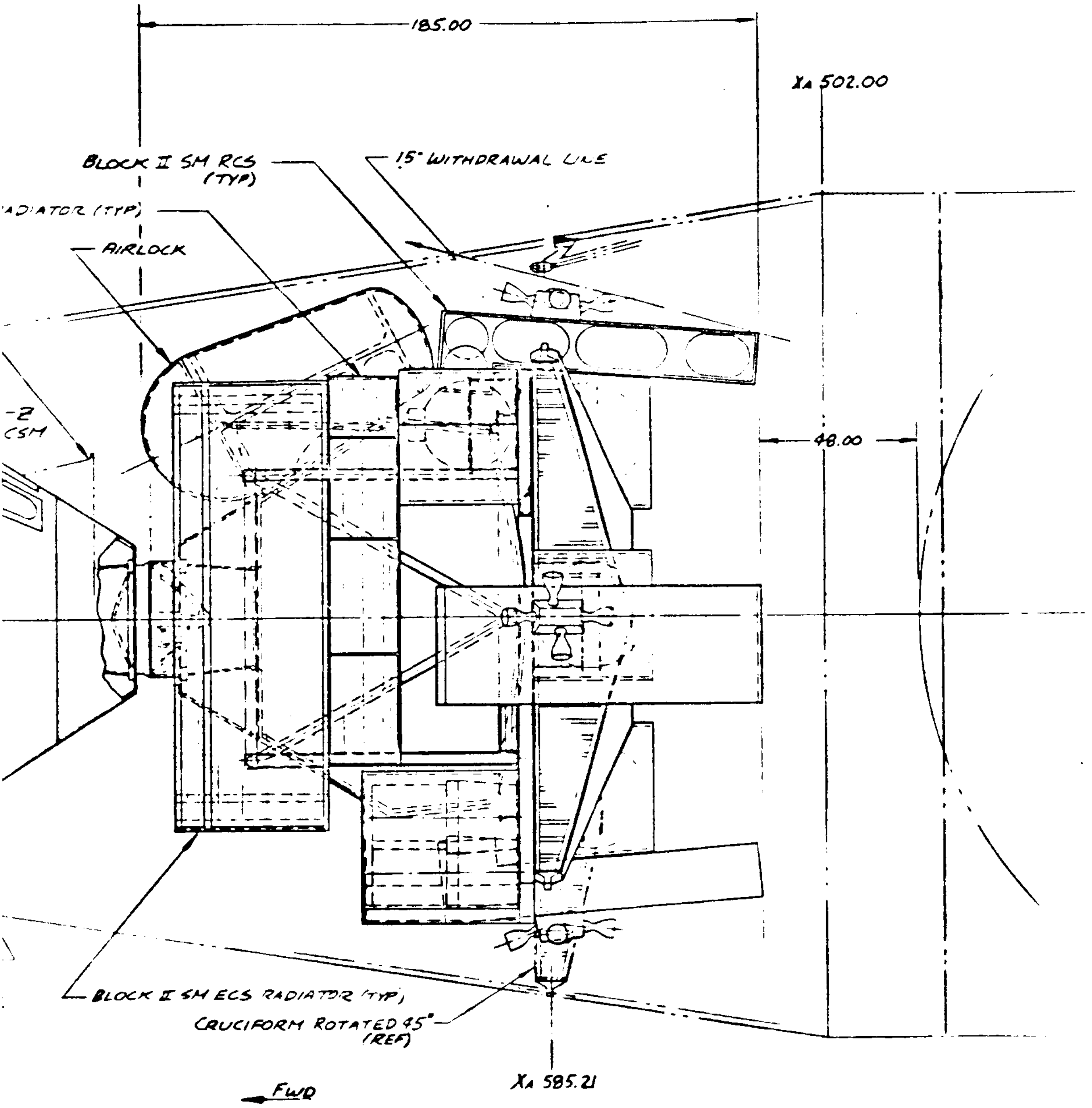


VIEW LOOKING AFT
(CM STRUCTURE OMITTED FOR CLARITY)
(1/20 SIZE)

Fold out FRAME 5

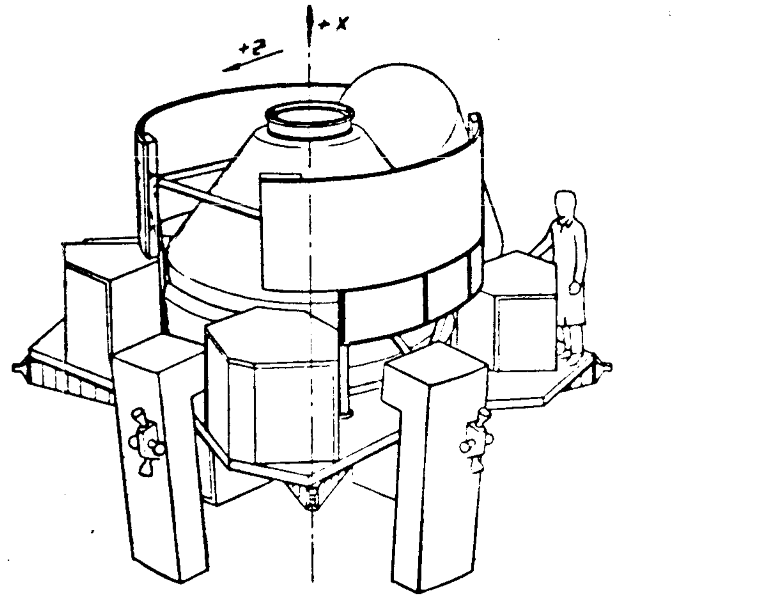
FOLDOUT FRAME 4

FOLDOUT FRAME



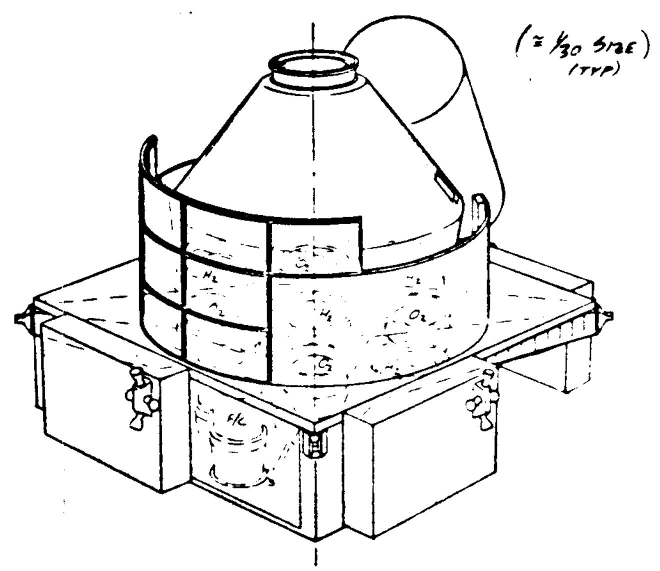
RCM INDEPENDENT LAB - SIDE VIEW

(1/20 SIZE)

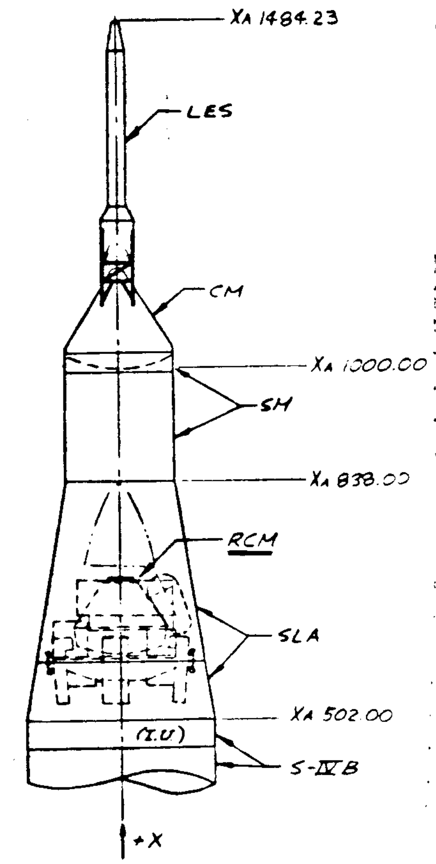


30 DAY INDEPENDENT BLK II RCM (SHOWN IN ADJACENT DETAILS)
- BLK II SUBSYSTEMS UNCHANGED
- NO SUBSYSTEMS BURIED WITHIN CRUCIFORM
- CRYO & FUEL CELLS SHELF MOUNTED

(≈ 1/30 SIZE) (TYP)



PROJECTED "MAX VISIBILITY" VERSION OF ABOVE CONFIGURATION
- RECONFIGURED BLK II SM RCS
- RADIATORS RELOCATED TO BLKHD
- FUEL CELLS SUPPORTED BELOW BLKHD
- (5) CRYO TANKS BURIED WITHIN CRUCIFORM

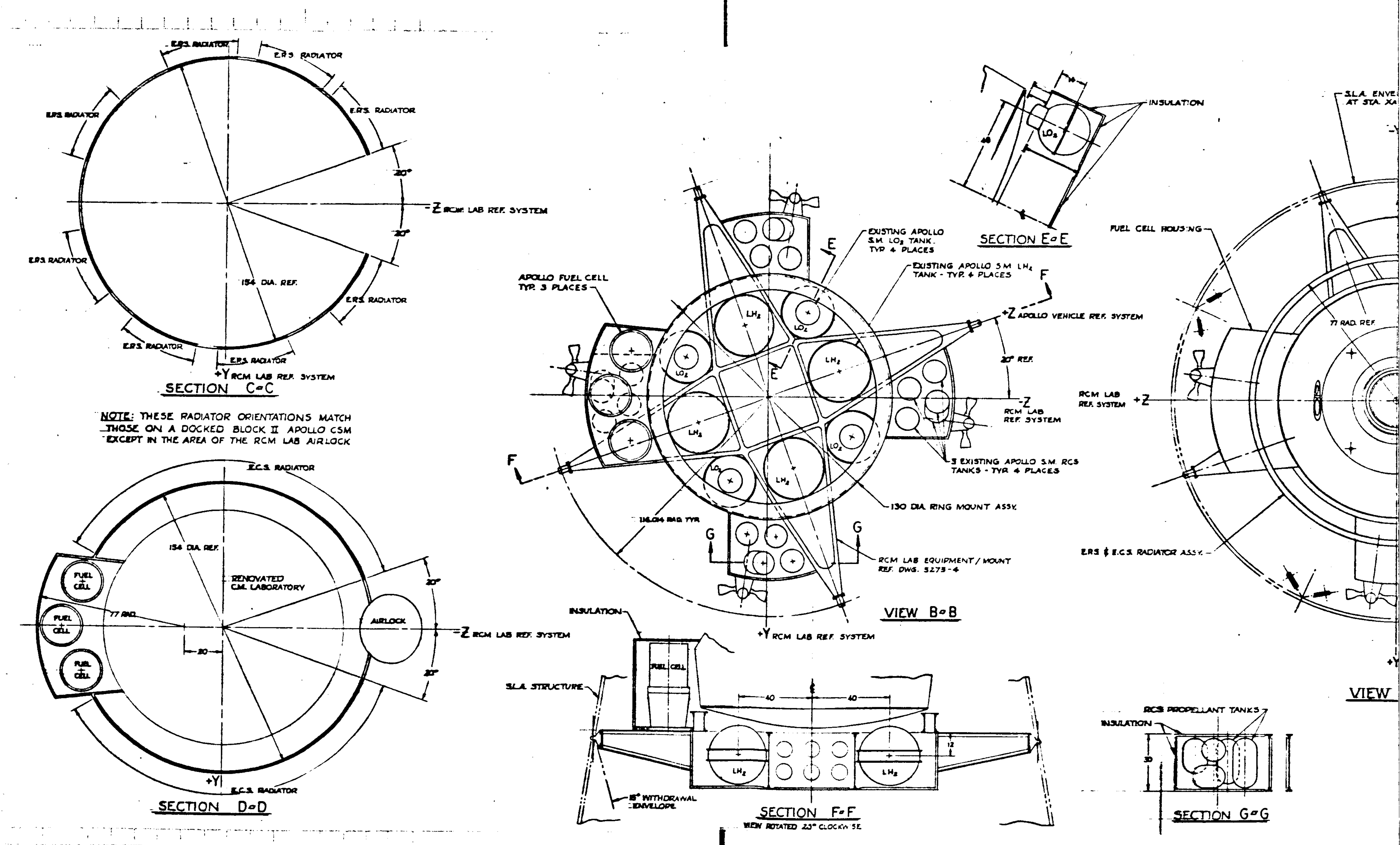


LAUNCH CONFIGURATION

(1/100 SIZE)

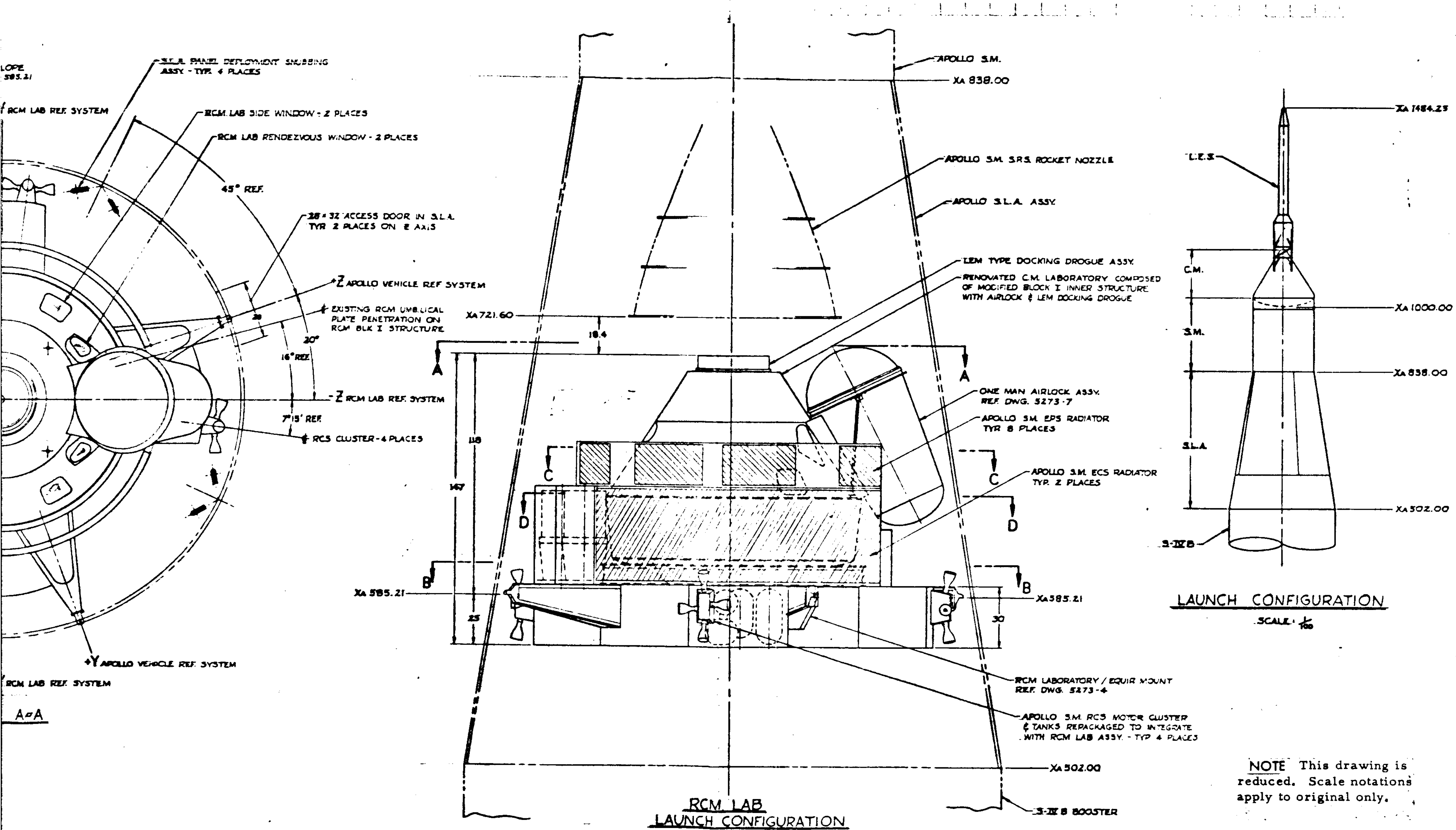
NOTE This drawing is reduced. Scale notations apply to original only.

Figure 109. Renovated CM Laboratory, Independent, Block II Structures and Systems



FOLDOUT FRAME 1

FOLDOUT FRAME 2



NOTE This drawing is reduced. Scale notations apply to original only.

Figure 110. Renovated CM Laboratory, Independent, Modified Block I Inner Structure

RCS is 7 degrees 15 minutes off both the Y and Z axis the same as the Apollo SM. This orientation provides a good compromise in terms of pad access, SLA clearances, docked alignment to an Apollo CSM, control logic, and reaction control moment arms for the RCS motors.

The existing EPS and ECS radiators are located in positions similar to those of the Apollo Block II CSM that would be docked to the RCM laboratory in flight configuration. This assumes that the +Z axis of both vehicles coincide when docked and the commander of the Apollo CSM can see directly into the right rendezvous window of the laboratory. This arrangement of radiators is ideal because it permits similar systems of both vehicles to operate in the same environment regardless of orientation.

The Block II RCS clusters are repackaged as shown to fit within the 30-inch depth of the equipment/support beam. The new assembly has four sides in plain view and attaches to the equipment/support beam on two sides where it is protected. The other surfaces are insulated and have micrometeoroid protection. This new configuration places the RCS assemblies in more appropriate positions and removes the portions that formerly projected down into the volume below the beam that is reserved for experiments.

Three Apollo fuel cells are provided as the electrical power source and are mounted above the beam and RCS package on the +Z axis of the laboratory where they are accessible for on-the-pad maintenance.

This configuration is the most compact of all independent concepts studied in accordance with the basic ground rules. The fact that makes this possible is the repackaging of the RCS tankage in each cluster. Visibility is good, both from the laboratory and from the docked Apollo CM. Placing the radiators low around the laboratory protects the backsides of the radiators and results in an overall lighter-weight installation. The exit hatch of the airlock is free of obstructions and may be clearly seen from the docked Apollo CSM. The radiator installation is functional because of the close proximity of each radiator to its operational system and the unobstructed view of free space from each radiator. The new shape for the RCS clusters permits the RCM laboratory to be shipped without removing the RCS assemblies such as is necessary for the other configurations.

Conclusions

The RCM laboratory configurations discussed herein have shown an orderly evolution from the dependent through the integrated independent concept. However, this does not necessarily represent the optimum solution to the problem of an RCM laboratory.

Figures 108, 109, and 110 show that the ECS and EPS radiators shield the laboratory pressurized compartment, to a large extent. Closing across the upper end of the cylinder formed by the radiators with a bulkhead will protect both the near-surfaces of the radiators and the outer surfaces of the crew compartment from micrometeoroids. This will also provide a protected area where lines and cables can be safely routed. Enclosing the crew compartment in this manner will interfere with visibility from the laboratory, but will otherwise result in a more functional and lighter-weight vehicle.

In summary, the concept presented in Figure 110 appears to be a reasonable design for a renovated and modified Block I command module that conforms to the ground rules previously stated.

IX. SYSTEM EFFECTIVENESS

SUMMARY

Estimates of system capability and mission effectiveness for the dependent and independent laboratories defined in Section VIII are summarized in this section, and estimates of incremental capabilities and effectiveness of the dependent laboratory are provided. These estimates are achieved through adding "shopping list" subsystems to the dependent laboratory configuration.

The purposes of the mission effectiveness analyses are to determine the capability of the baseline configuration to meet the reference mission performance requirements, and to provide guides to assist in determining configuration or mission-operation modifications that will assure an optimum AAP. The principal facets of mission effectiveness are the configuration performance capability (i. e., capability of the configuration to accommodate needed equipment and consumables and to meet other direct-support requirements for attitude holds, power, etc.) and the ability of the subsystems and total system to perform reliably for the required mission duration and to return the crew safely.

Estimates of comparative capabilities of the baseline configurations to perform reference or analysis missions that represent typical advanced (extended duration) AAP mission requirements are provided. These estimates may serve as basepoints for evaluating the effects of changes in the configurations on system capability. For example, if the experiment planned for a specific mission requires a significant increase in electrical energy or in propellant for attitude hold, then the effect of the increased requirements on payload weight and the compromises needed for mission accomplishment can readily be determined. Among the compromises possible might be a change in the planned mission duration, off-loading of some equipment, changing the orbit to a lower altitude, or reducing the mission duration.

APPROACH

Figure 111 illustrates the general approach to the evaluation. Figure 112 is a breakout showing the procedure for identifying and evaluating the subsystem deltas and capabilities. Figure 113 describes the approach used in obtaining mission and cost effectiveness process in terms of the types of information and parameters selected for the effectiveness evaluation.

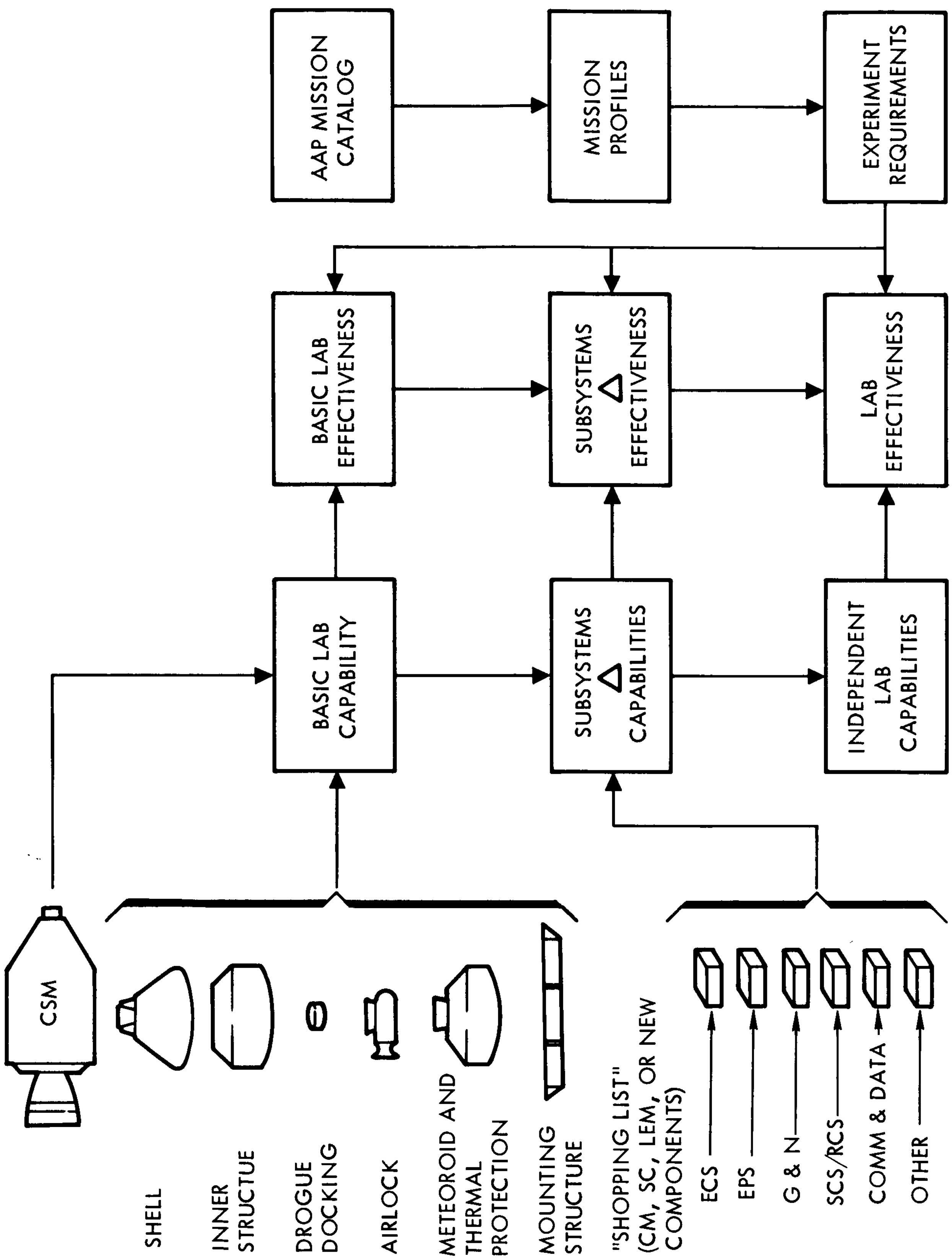


Figure 111. Renovated CM Laboratory Effectiveness Approach

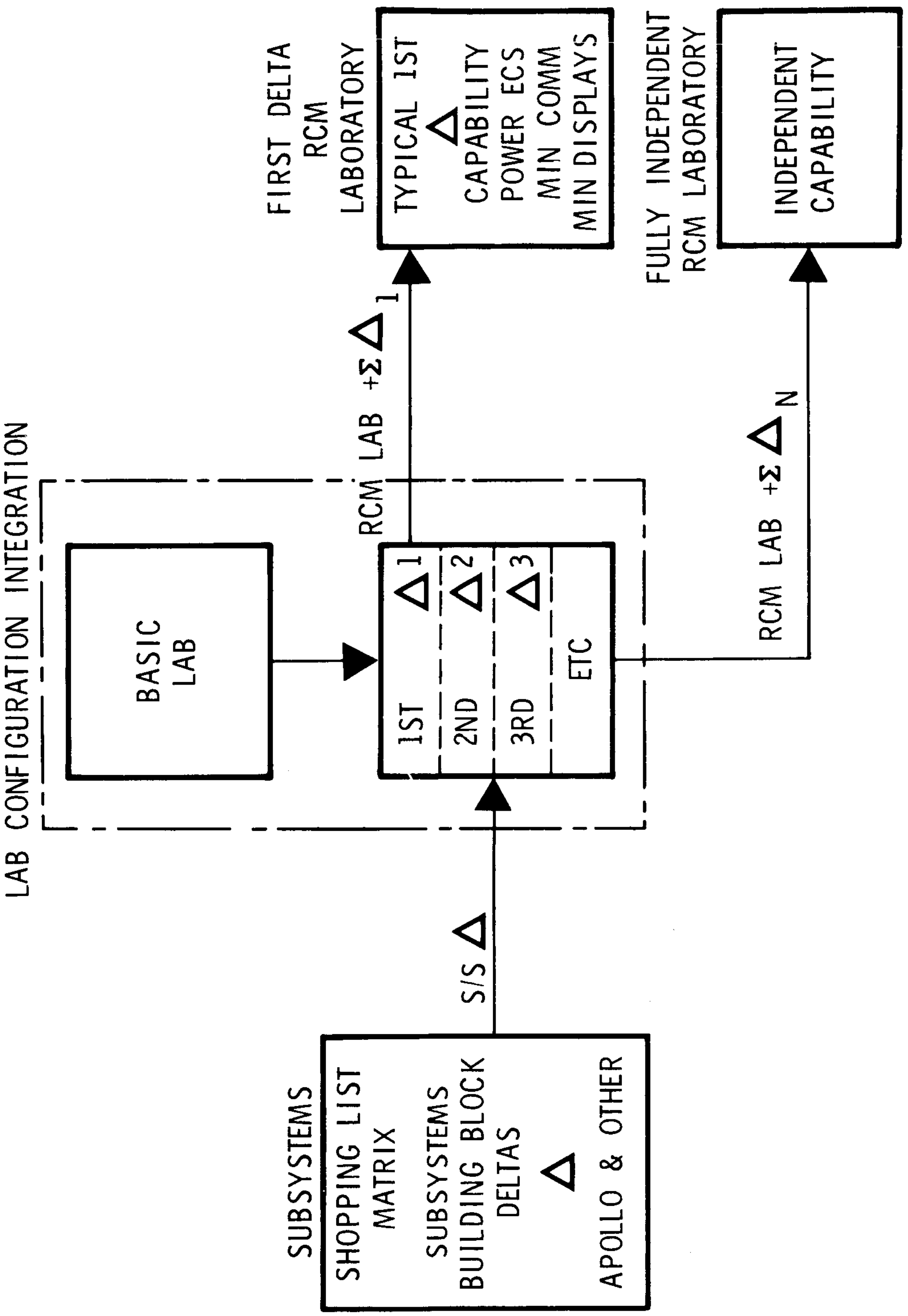


Figure 112. Configuration Capability

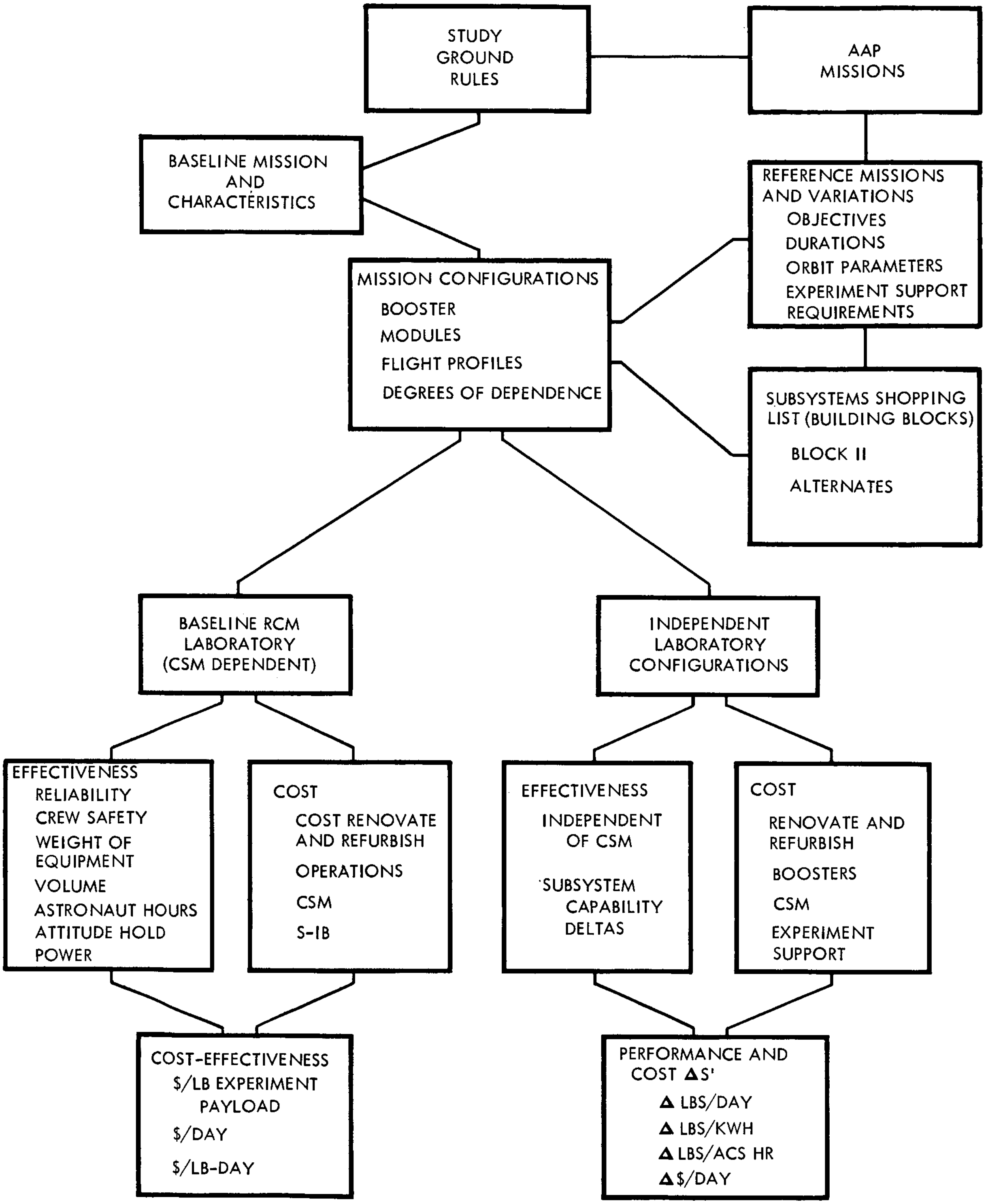


Figure 113. Mission Cost Effectiveness Parameters

Characteristics of the four basic reference AAP mission categories are defined on the basis of analyses of AAP missions and objectives. These missions include a low-inclination, low-altitude earth orbit, a low-altitude earth polar orbit, a synchronous equatorial orbit, and an 80-nautical-mile lunar polar orbit. The reference missions have been selected as having 30-day durations, though feasible and realistic alternative missions lasting from 15 to 45 days are also considered. For the purpose of defining and evaluating the basic renovated laboratory, a single 30-day baseline mission was identified. This is the low-inclination, low-altitude, earth-orbit mission. Based on mission characteristics (that is, the durations, orbit parameters, housekeeping, and experiments orbital requirements), basic mission configurations were defined, including the booster and spacecraft modules, and operating modes. These led to definitions of the baseline CSM/RCM laboratory as well as configurations for the performance of each of the reference missions. These then provide the basis for evaluation of the capabilities of the renovated CM laboratory for the dependent and independent laboratory versions. Finally estimates of effectiveness and cost effectiveness were determined to provide a basis for determining best mission configurations.

The significant evaluation parameters, and the relative importance of the various evaluation parameters, will depend on the objectives of the specific missions. Typical parameters most significant to specific AAP experiment categories are given in Table 57.

The reference-mission requirements are defined in terms of expendables required for mission accomplishment; weights and space for experimental equipment and return payloads; astronaut time required for accomplishing experiments and tests; requirements for spacecraft pointing for communications, mapping, and other operations requiring sensor pointing and spacecraft thermal control; and the navigation, guidance, and trajectory requirements.

Mission Duration

The mission duration is assumed to be 30 days. However, some missions not otherwise possible may be accomplished by allowing a reduction in the duration from the assumed 30 days. The minimum required durations depend on the experiment-time span; for early missions, short time spans are indicated, but subsequent missions may require extended durations as being essential to the economic and efficient development of advanced spaceflights. The modular laboratory concept permits varying the durations to obtain maximum utilization of subsystem capabilities and the allowable payloads that can be placed in orbit.

Table 57. Effectiveness Parameters

AAP Experiment Category	Effectiveness Parameter
Biomedical and behavioral	Mission time span and astronaut drop in orbit
Physical sciences	(Experiment equipment weight) x (Astronaut days in orbit)
Astronomy and astrophysics	(Experiment equipment weight) x (Mission duration)
Earth-oriented sciences and applications	(Experiment equipment weight, mission duration, and experiment equipment weight) x (Mission duration)
Lunar surface sensing, survey, and mapping	(Experiment equipment weight) x (Mission duration)
EVEA extravehicular engineering activity	(Astronaut days in orbit and experiment equipment weight) x (Astronaut days in orbit)
Operations techniques	Astronaut days in orbit
Module and subsystems development and qualification	Experiment equipment weight, mission duration, and mission time span

Astronaut Time Available for Experiments

Because of the limited understanding of man's tolerance to the space environment and his capabilities for task performance, perhaps the most important evaluation parameter is the astronaut time available for performing experiments, particularly those in areas of biomedical and behavioral operations, techniques studies, and extravehicular engineering tasks. For normal missions not requiring EVA, it has been determined that 8 hours out of each 24 can be made available for performing experimental tasks. Since this time is not likely to vary with different missions, excepting for those requiring extensive EVA, and if crew size is fixed, then, time available for experiment performance becomes synonymous with mission duration.

Weight Allowable for Experiment Equipment

The weight allowable for experiment equipment is the total payload weight that can be placed in the indicated earth orbit less the CSM and laboratory weights (Section II). Weight is invariably an important consideration in mission planning. Because of the high cost of getting equipment and astronauts into orbit, it is desirable to take maximum advantage of the payload capability provided by the boost vehicle.

Volume Available for Experiments and Equipment

In general, an adequate volume for experiment performance is needed, but more than the volume required for efficient performance will not necessarily add to mission success. At the same time, the pressurized and unpressurized volumes for any spacecraft configuration must take into account the varied mission requirements that will exist as well as the growth potential for the vehicle. The pressurized volume of the RCM laboratory is adequate to meet all of the AAP flights considered. (The adequacy of the free space work area for astronaut use needs to be determined.)

Crew Safety and Mission Success

Independent Laboratory

When the independent laboratory is providing attitude control and other experiment support, the subsystems performance interferences between the CSM and the laboratory are minimized. This would require that the reliability for the SCS/RCS in the laboratory be the same as for the CM, or that a CM override capability exist when needed. It is assumed that one or more astronauts will be in the CM at all times. If the radiation level becomes excessively high due to solar flare activity, the astronauts can move to the CM and, if needed, abort the mission.

Mission reliability defines the expected probability that the mission can continue for the planned duration or for some period less than the planned duration. The factors considered here include malfunctions or failures of CSM subsystems that require abort or alternative mission. The principal factor that may terminate the mission short of its planned duration is crew safety. The crew safety requirement for the AAP missions is the same as for the Apollo Block II lunar missions (0.999).

Dependent Laboratory

The effectiveness of the dependent laboratory depends on the experiment and housekeeping support capabilities of the CSM. Normally only one man at any one time can remain in the laboratory for experiment performance as one man is needed for monitoring subsystems. However, experiments requiring participation of two astronauts can be performed by modifying the sleep and work cycle to allow two of the astronauts to sleep at the same time. No experiments could be performed during this period unless they could be accomplished by the astronaut assigned to monitoring and controlling the spacecraft subsystems.

Crew safety is taken as being the same as for the CSM. If subsystems malfunctions occur that would affect crew safety, either in the CSM or laboratory, the astronauts may immediately return to the command module and, if necessary, abort the mission.

ACCOMPLISHMENT OF MISSION OBJECTIVES

System Capability

The ability to accomplish mission objectives depends mainly on system capability. The detailed experimental requirements, the state-of-the-art and performance reliability of the experimental and test equipment, and the ability of the CSM subsystems to meet support requirements must be considered. Other factors, such as the effects of excessive levels of radiation, meteoroid activity that can terminate or temporarily disrupt the mission, and astronaut sickness or other constraints to effective performance are also significant but are not given detailed consideration in this section.

Mission-Planning Flexibility

Mission flexibility is another, less tangible, factor with respect to both preflight and inflight mission planning. The high costs of the missions require careful planning of each flight to achieve a maximum amount of useful information from each flight and from the program as a whole.

Preflight flexibility— the ability to modify planning factors such as consumables, flight trajectories, and mission duration, and to change experimental equipment— is important to achieving the most effective overall program. In some respects, the AAP program differs from past missile and space-systems test programs. While the AAP missions include investigation and qualification of systems and performance among their objectives, the nature of the scientific data, and perhaps less reliable performance of some scientific equipment, tends to increase the dependence of subsequent flights upon results of earlier flights. The subsystem building block approach permits maximum flexibility in providing, with minimum lead time, specific laboratory configurations that can best meet changes in the requirements for specific experiments.

Shown in Figure 114 are comparative capabilities of laboratory options for meeting the AAP-experiment requirements. An "as is" system could be provided on short notice, but would contribute to the accomplishment of a relatively small percentage of the AAP experiments and, for the most part, would be inefficient. The basic laboratory, with a free-space volume of about 300 cubic feet, would be able to economically support some of the experiments required during the early phases of the AAP program, and may also be used for the storage and transport of supplies and materials. Its duration is limited to the support capabilities of the CSM.

The first delta is a system providing capabilities for utilizing CSM-supplied power for accomplishing experiment groups, such as those identified for the reference missions. Mission life is limited by the amount of consumables provided by the CSM and the requirements imposed by the mission and experiments.

The independent laboratory can utilize both the CSM subsystems and the subsystems mounted on and within the laboratory; in this way, mission durations of from 30 to 45 days can be achieved when using Apollo Block II-type subsystems. When utilizing the laboratory, power, ECS, and AES subsystems, the CSM subsystems might be placed on a standby basis such that monitoring is not required. Two men could then participate in performing experiments when desired. The independent laboratory in a multiple-docking arrangement can allow the meeting of all basic AAP-experiment requirements, including extended-duration missions with initial crew sizes of six or more astronauts.

A new CM shell, with or without design modifications, could be considered as a growth feature of the RCM concept, and also could be used if recovered CM's are not available. Also modifications to the Apollo subsystems to provide additional life extension, or new alternative subsystems, can be considered for use during later phases of the AAP.

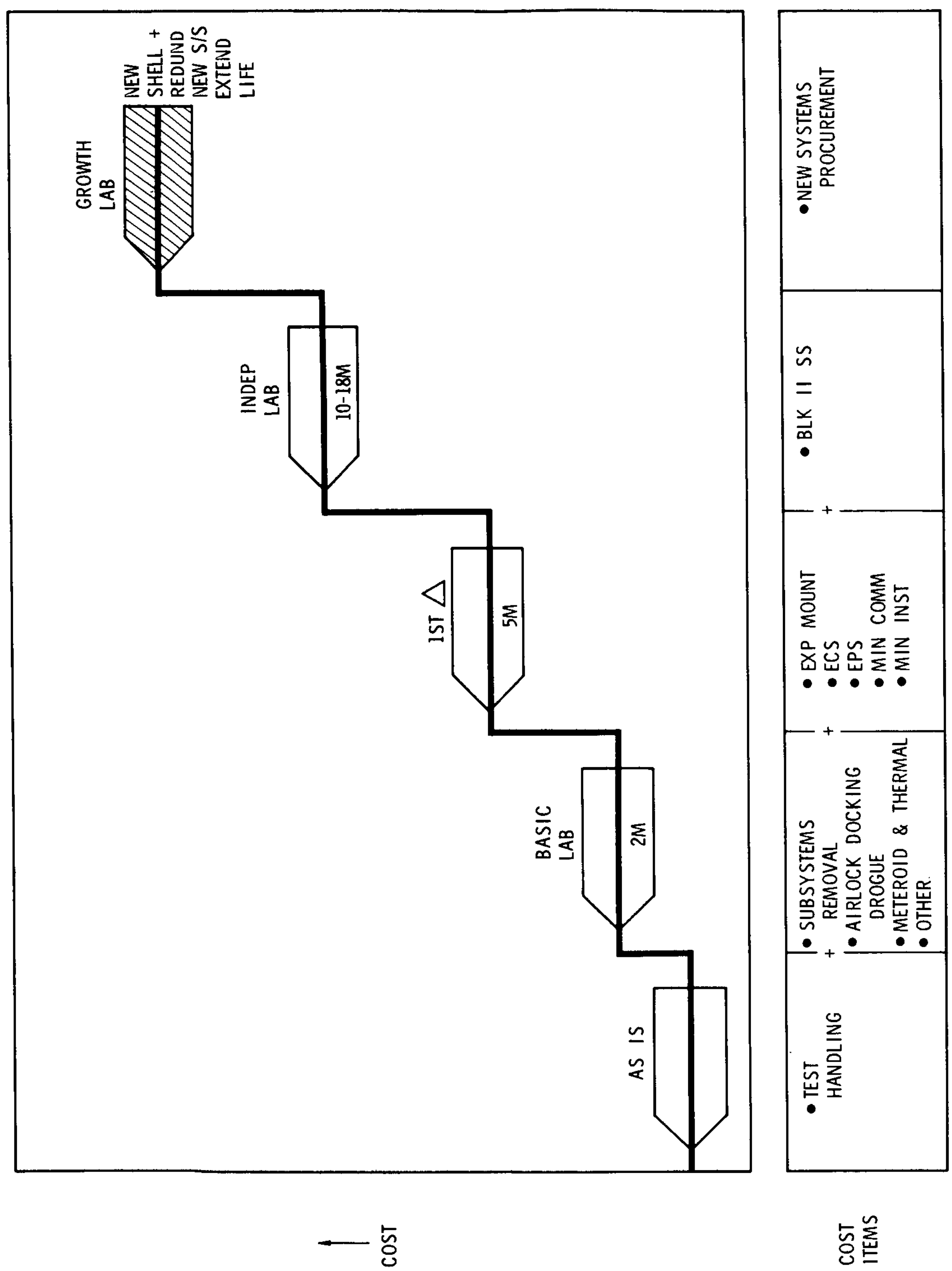


Figure 114. Renovated CM Utilization Options

Laboratory Baseline Capabilities

In this section is presented an evaluation of the capabilities of the dependent and independent laboratories for supporting AAP-type experiments and the costs of CM renovation and refurbishment. The early AAP flights emphasize the testing of equipment, investigations of space operations, and man's capabilities for performing varied experiment and EVA engineering tasks. Except for multispectral investigations, the equipment weights required are modest, and the missions can be accomplished in a low-inclination, low-altitude earth orbit using simple, though weight-limited, S-IB launches. Some of the later, longer-duration missions will require heavy sensing equipment or large-mass structures for extravehicular engineering operations and, thus, much larger payloads. For those missions that can be accomplished in low-inclination earth orbits, either multiple launches using S-IB boosters or single launches using S-V boosters are indicated. Because of the much larger payloads that can be placed in low-altitude, low-inclination, earth orbits and the consequent much lower cost per pound for putting these payloads in orbit, experiments should be scheduled for flights in these orbits wherever possible, and the near-polar or synchronous orbits should be used only when needed to satisfy the requirements of important AAP experiments. Accordingly, the significant evaluation parameters for the early missions are:

1. Astronaut hours available for performing experimental tasks

2. Total mission duration or maximum mission durations consistent with astronaut and subsystems capabilities being desired

3. The weight of experiment support equipment that can be placed in orbit— due to the high costs of putting payload in orbit and the S-IB payload limitations

Other parameters that are sometimes pertinent include the volume available for experiment equipment and experiment operations, and the laboratory subsystems capabilities for experiment support, mission success, and crew safety. Subsystem-support parameters include the capabilities to meet experiment requirements for electrical power, thermal control, attitude control, and environmental control for life support. Nominal capabilities are implied in the baseline configurations, but adequacy of the systems to support specific missions will depend upon the specific mission requirements. Requirements for added subsystem support will ordinarily exact a penalty cost from the payload weight.

Dependent Laboratory

The mission capability of the dependent laboratory is primarily limited by the ability of the CSM to provide attitude control, power, ECS, and other mission support. The principal considerations, therefore, are availability of consumables and the life and reliability of CSM subsystems. Assuming that adjustments can be made to the consumables placed in the CSM, the dependent laboratory could have a capability for performing a variety of early AAP missions. Section II shows the allowable weight of experiment equipment for a fully dependent laboratory versus CSM weight for each of the referenced missions. The feasible S-IB missions involve low-altitude earth orbits. Approximately 4000 pounds of experiment equipment can be launched into a low-inclination, 200-nautical-mile-altitude, earth orbit. A dual launch with orbital rendezvous is required for a polar orbit.

Mission success and crew safety are essentially the same for the dependent laboratory configuration as for the Apollo CSM, except for experiment-equipment failures. Mission success versus mission duration is illustrated in Figure 115. Applicable ground rules are as follows:

1. Reliability estimates are based on defined RCM laboratory reference missions (30 days).

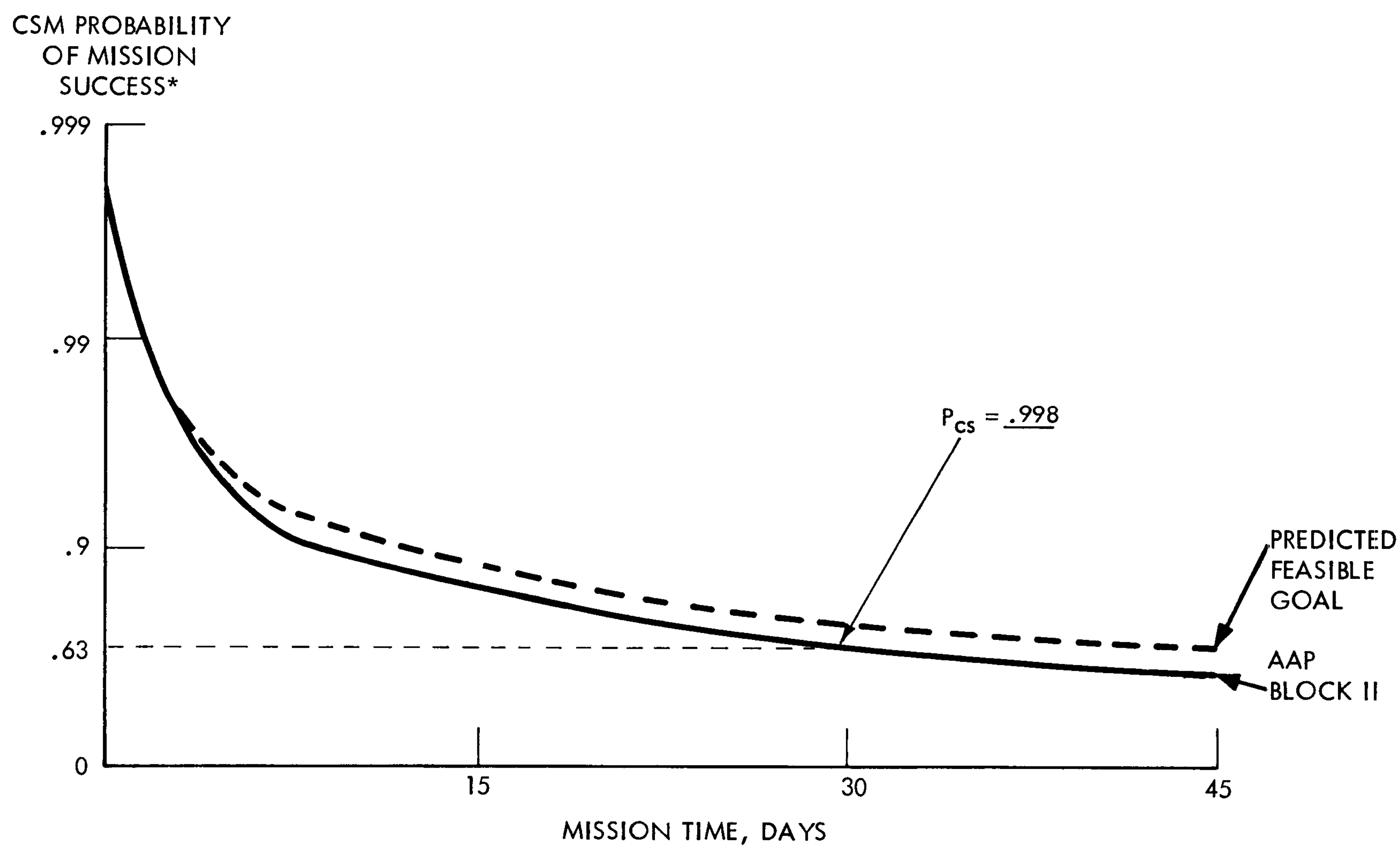
2. The flight configuration is to consist of the Apollo Block II CM, AAP 30-day SM, RCM laboratory, and subsystem building blocks.

3. Abort criteria require abort whenever one additional failure would expose the crew to environments beyond the specified emergency limits.

4. Crew safety is based on having no failure or combination of failures that would result in loss of the crew.

5. Mission success is defined as the probability of having no failure or combination of failures that would require abort or exceed crew safety limits.

6. R(CS) = R(MS + Q(MS) R(SA).



*NEGLECTING BOOST, GOSS, & GSE

Figure 115. Mission Success and Crew Safety, Low-Altitude Earth Orbit

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X. COST EFFECTIVENESS

OBJECTIVES

The objectives of the cost effectiveness evaluation are to establish the overall feasibility of the RCM laboratory concept; to provide a baseline against which to consider the relative merits of the RCM as compared with alternative systems, to assist in defining a best use of the RCM in the AAP to assist in defining laboratory configurations for specific missions, and to establish a baseline to assess the feasibility and value of adding subsystems to the basic dependent laboratory to provide extended mission durations or increased experiment support capabilities (more precise attitude holds, increased attitude hold hours, increased thermal control, or a more adequate thermal balance under more stringent experimental conditions, and increased power).

Cost effectiveness factors are also defined as to the subsystems required for experiment support. These are deltas that can be added to the basic dependent or independent laboratory effectiveness estimates to obtain total capability and cost effectiveness. These deltas can also be used in selecting experiments and planning missions. It may be assumed that an AAP flight program, including overall objectives and the major cost items such as boosters, CSM's, and perhaps laboratories, have been defined and budgeted. It is then necessary to identify specific mission objectives and experiment packages for each of the flights to maximize mission results. The selection and scheduling of experiments from the available alternatives can be considered in terms of capabilities and costs of the laboratory and specific subsystems required. The overall cost effectiveness of the RCM laboratory configurations can then be determined and presented in a form similar to that illustrated in Figure 116. System effectiveness data contained herein are based on preceding sections of this volume.

COST EFFECTIVENESS PARAMETERS

As indicated in Section III, significant evaluation parameters will vary according to the specific mission objectives. Typical parameters significant to specific AAP experiment categories are given in Section II. Using these parameters as a basis, cost effectiveness parameters have been defined for the basic experiments categories, shown in Table 58. The selected cost effectiveness parameters for use in this study are cost per day, cost per

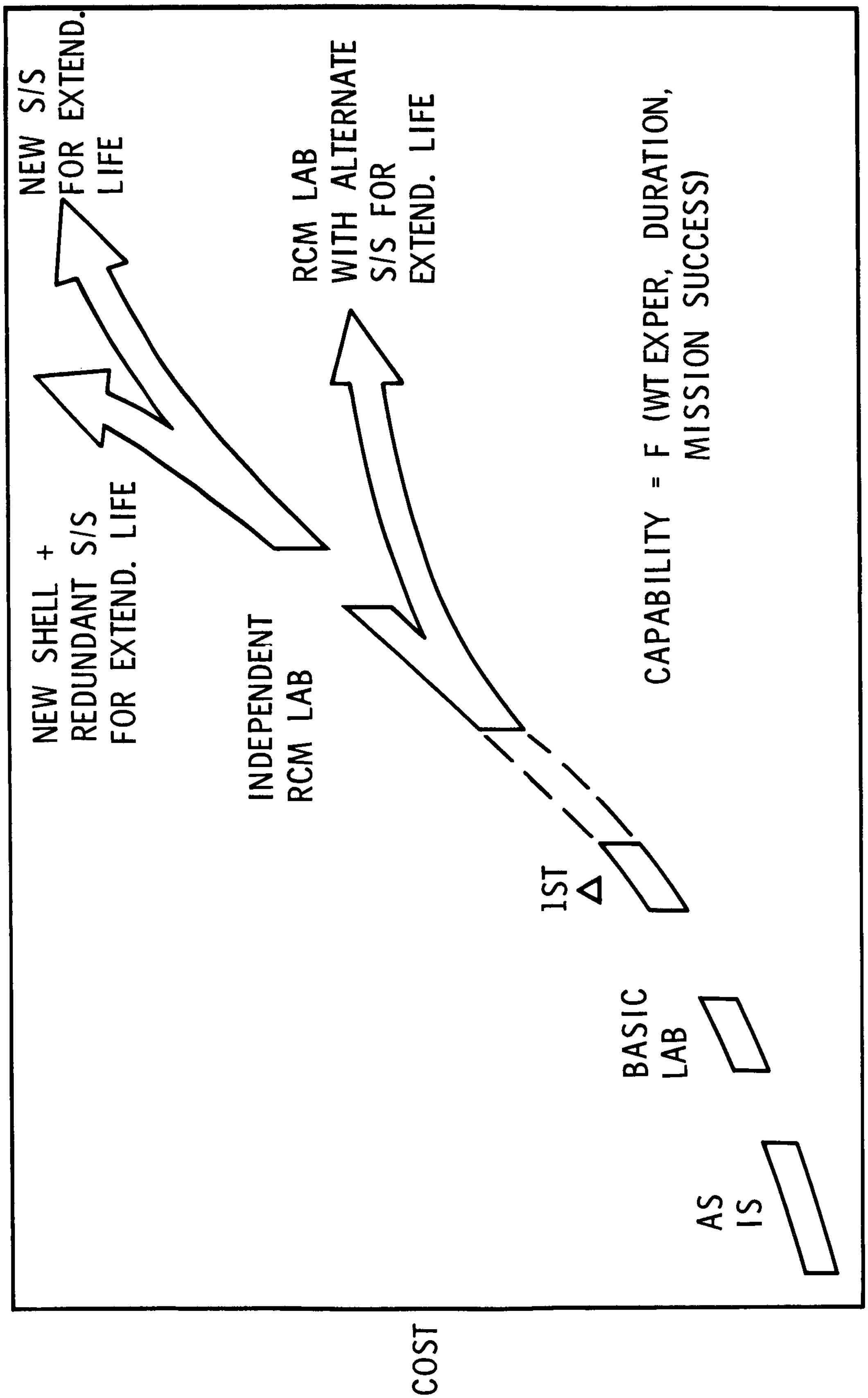

Figure 116. Configuration Cost Effectiveness, RCM Laboratory System

Table 58. Cost-Effectiveness Parameters

AAP Experiment Category	Effectiveness Parameter	Cost Effectiveness Measures
Biomedical/behavioral	Mission time span Astronaut drop in orbit	Dollars per day
Physical sciences	(Experiment equipment weight) x (Astronaut days) (in orbit)	Dollars per pound -day
Astronomy/astrophysics	(Experiment equipment weight) x (Mission duration)	Dollars per pound -day
Earth oriented sciences and applications	(Experiment equipment weight) (Mission duration) (Experiment equipment weight) x (Mission duration)	Dollars per pound Dollars per day Dollars per pound -day
Lunar surface sensing, survey and mapping	(Experiment equipment weight) x (Mission duration)	Dollars per pound -day
EVA	(Astronaut days in orbit) (Astronaut days) (Experiment equipment weight) x (in orbit)	Dollars per day Dollars per pound -day
Operations techniques	Astronaut days in orbit	Dollars per day
Module and subsystems Development and qualification	Experiment equipment weight Mission duration Mission time span	Dollars per pound Dollars per day Dollars per day

pound of experiment payload placed in orbit, and cost per pound-day. Mission time span is also an important consideration. The requirements for mission time span and the utility of specified time spans when measured in regard to the requirements will depend on specific experiment needs. The AAP bridges the gap between the Apollo program and future longer duration earth orbital laboratories and manned interplanetary missions. Generally incremental increases in mission duration are envisioned, the first being from the 14-day Apollo program to a length of about 30 days, then to about 60 days, and then to six months or a year. A time span of two to three years may be desired as a qualification before undertaking manned missions to Mars and Venus. In general, multiple CSM docking arrangements would permit achieving any desired time span. No attempt is made in this study to evaluate configurations in regard to this last parameter.

Cost Effectiveness Estimates

Table 59 lists initial results of the cost effectiveness analysis for the independent and dependent laboratories for each of the reference missions. The principal cost assumptions are summarized in Table 59 .

Table 59. Summary of Mission Cost Items

Cost Item	Dollars in Millions
S-IB (launched)	50 (S-I, S-IVB)
SV (launched)	160 (S-IC, S-II, S-IVB, IU)
CSM (launched)	44
Dependent laboratory (launched)	6
Semidependent laboratory (launched)	1.9
Independent laboratory	17
Facilities	6
30 day operations	10
Development	0
Experiments	Not included

Table 60 summarizes results for the reference mission flights identified in the mission analysis. The experiment weights in orbit are the maximum allowable, based on the assumption of maximum utilization of the potential payload capability of the launch vehicle. In some cases as much as 35,000 pounds of SPS fuel might be carried to permit orbit altitude changes. It is also assumed that separate equipment or expendable packages can be carried to utilize the boost vehicle payload capability in providing an increased mission duration.

Table 60. Cost Effectiveness Summary

Mission	ELIO Single Launch	ELIO Rendezvous	EPO	EPO	EPO	EPO	ESO	ESO	LPO	LPO
Configuration	1 S-IB 1 CSM 1 RCM(SD)	2 S-IB 1 CSM 1 RCM(I)	2 S-IB 1 CSM 1 RCM(SD)	2 S IB 1 CSM 1 RCM(I)	1 S-V 1 CSM 1 RCM(SD)	1 S-V 1 CSM 1 RCM(I)	1 S-V 1 CSM 1 RCM(SD)	1 S-V 1 CSM 1 RCM(I)	1 S-V 1 CSM 1 RCM(SD)	1 S-V 1 CSM 1 RCM(I)
Total* cost (millions)	100	161	150	161	210	221	210	221	210	221
Duration (days)	30 →									→ 30
Experiment weight (1000 lb)	4.7	19.4	16.6	4.3	67.9	56.9	18.6	7.6	32.6	21.6
Cost/lb experiment ($1000/lb)	21.3	8.3	9.0	37.5	3.1	3.9	11.3	29.2	6.5	10.2
Cost/day ($1000/day)	3333	5367	5000	5367	7000	7367	7000	7367	7000	7367
Cost/lb-day ($)	710	277	300	1250	103	130	377	973	217	340

*Excludes experiments and packaging. Includes launch facilities and 30-day operations.

The low-altitude earth orbits are assumed to be at 200 nautical miles altitude. A somewhat lower altitude might be used for the thirty-day mission, and would allow some increase in experiment weight. The costs per pound for providing orbital facilities and support of experiments are seen to range from about $3,000 to $37,000, and the cost per pound-day at from $103 to $1250. Not shown is a S-V low inclination earth orbit which would permit even lower costs per pound and per pound-day.

Experiment Weight Cost-Effectiveness

Figure 117 shows the effect of variations in the experiment weight on the cost per pound for providing necessary laboratory equipment and supplemental subsystems support for manned flights. These results are based on costs and allowable weights given in the Table 60 . The dotted and solid lines are for S-IB and S-V launches, respectively.

Since it is assumed that the full booster payload potential is effectively used, inefficiency in use of the payload capabilities would result in an effective shift of the points shown along the curves upwards and to the left.

Cost Effectiveness of Experiment Pound-Day

Figures 118 and 119 give costs per experiment pound-day for the dependent and independent laboratory configurations, respectively. The circles are for a thirty-day mission.

The effect on the overall cost-effectiveness of extending the mission duration from 30 to 60 days is shown for two cases as follows:

1. No payload penalty. This case assumes that the increase in duration can be obtained through more efficient utilization of the existing subsystems capabilities and perhaps increases in the expected MTBF resulting from increased experiments and debugging of the subsystems. For this case the costs would drop to the lower sets of points that are shown in the figures.

2. An overall payload penalty of 75 pounds per day. This assumption is made arbitrarily to account for consumables and other requirements associated with increased mission durations. Different spacecraft masses and equipment can impose different requirements for attitude control fuel and power requirements. However, for the smaller laboratory configurations the errors will remain small. For the large laboratories shown, the percentage errors will be small, though the actual errors may be large and the results shown in the figures should not change greatly.

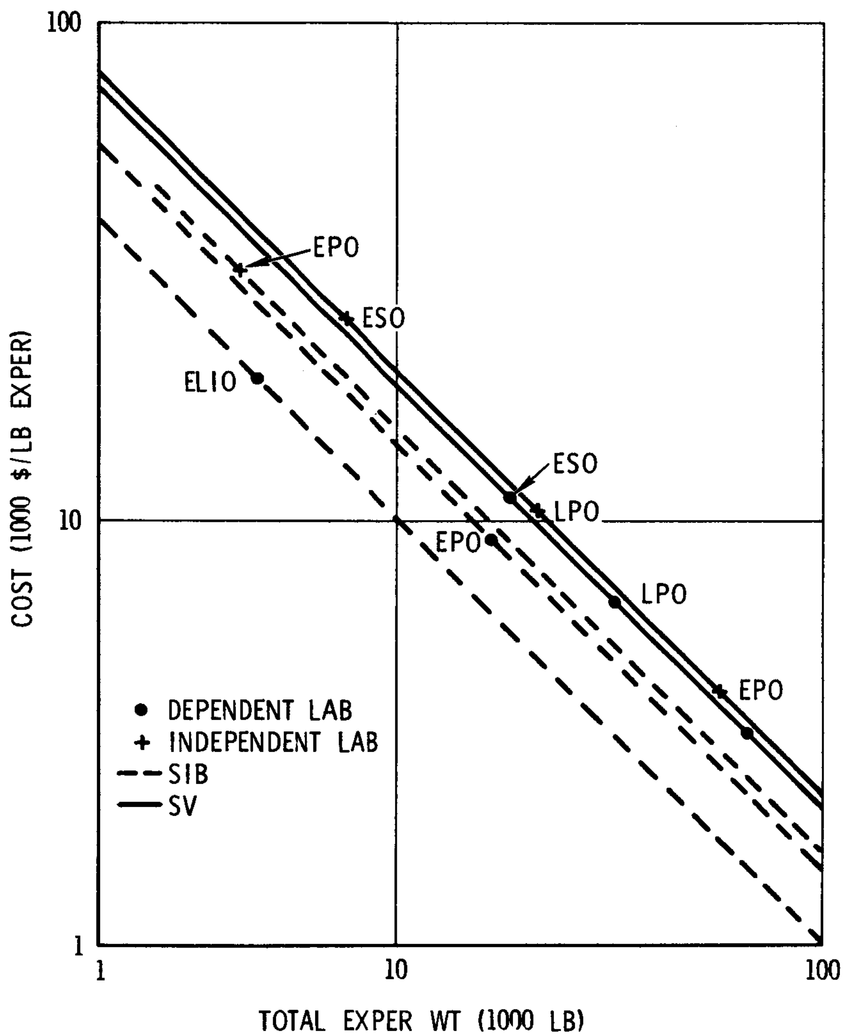

Figure 117. Cost Per Pound of Experiments Versus Experiment Payload Weight

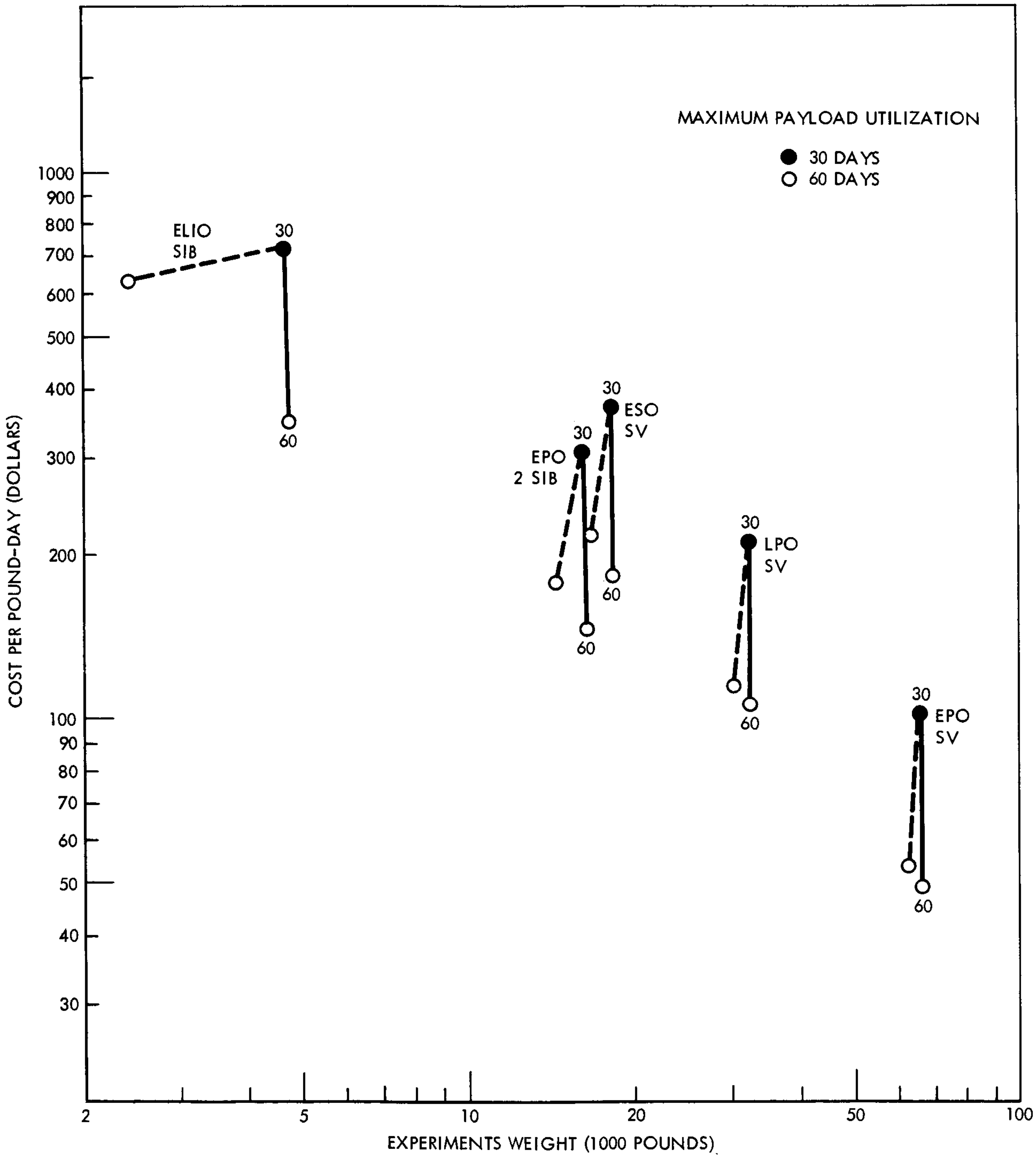

Figure 118. Cost Per Pound-Day, Dependent Laboratory

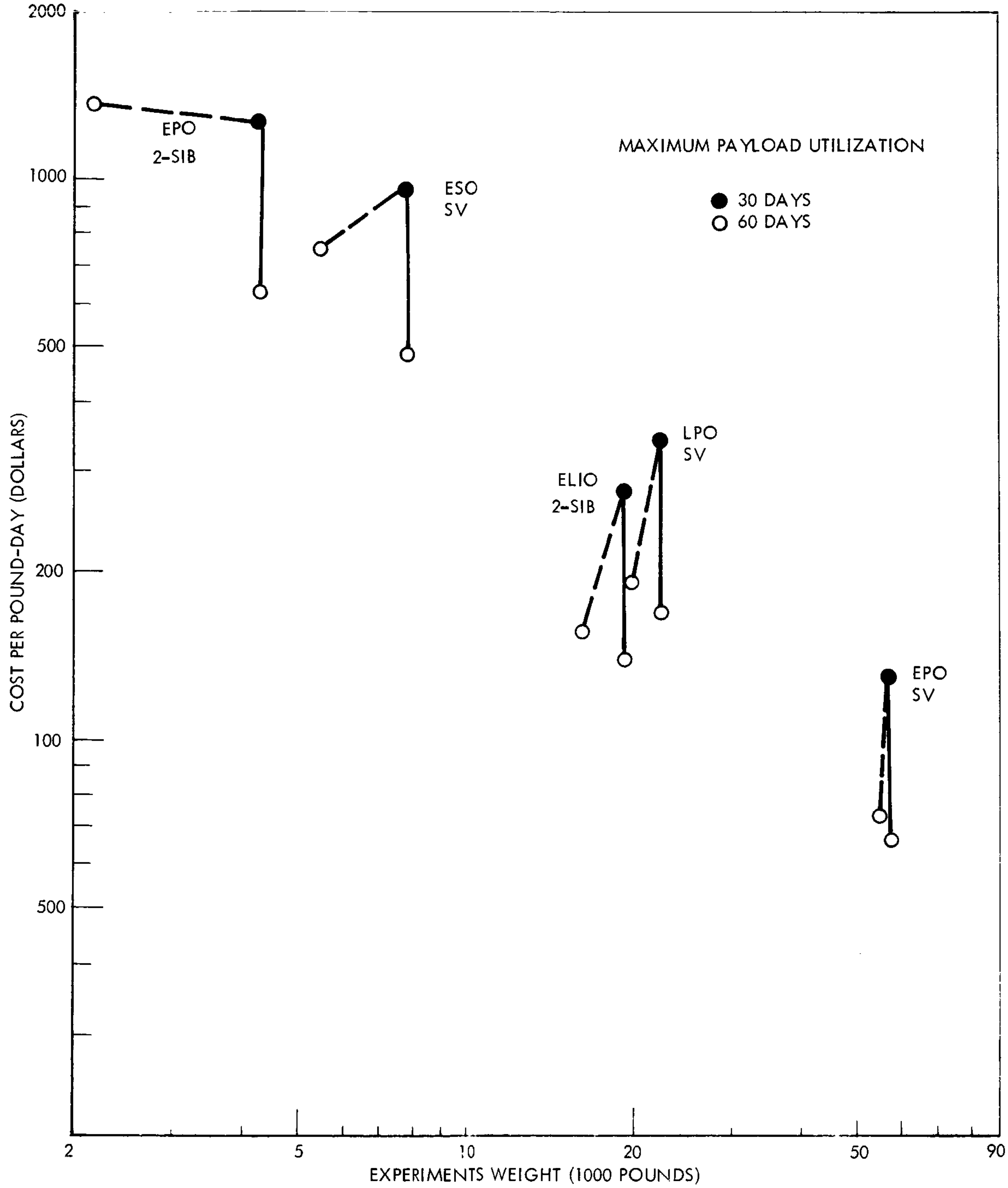

Figure 119. Cost Per Pound-Day, Independent Laboratory

Cost Effectiveness of Mission Duration

The mission duration and total hours available for performing experiments are two significant measures of effectiveness for many categories of AAP missions. As shown in Figure 120 for these categories, the preferred spacecraft/laboratory configurations will minimize cost for a given mission duration or maximize time in orbit per unit cost.

Past studies indicate that for three-man AAP configurations, each astronaut can devote about eight hours each day to performing experiments. For these configurations, the mission duration is equivalent to the time available for mission performance.

In many cases, extensions in mission duration can be achieved at little or no weight cost. The Apollo flight demonstrations will provide increased assurance of the extended life capabilities of the Apollo subsystems.

While the S-V missions appear most costly assuming a fixed mission duration, the large payloads allowed can allow increases in consumables, and thus permit maximum utilization of the life capabilities of the Apollo subsystems. The importance of maximizing experiment yields through careful advance planning and integration of experiment programs, however, cannot be overemphasized.

SUGGESTED COST-EFFECTIVENESS MODEL

The preceding cost effectiveness analysis was performed on a general basis for the dependent RCM laboratory and the independent RCM laboratory system configuration, without any reference to specific experiments, their configurations, or objectives. After specific experiments and objectives are defined for specific missions, detailed cost effectiveness analysis can be performed by a method suggested in this example.

Example applications of the shopping list approach to the RCM laboratory cost effectiveness analysis can be: (1) determine the cost and cost effectiveness of a laboratory configuration for a synchronous orbit mission, (2) determine the delta cost of added performance (kwh, or mission duration), and (3) assist in defining a specific mission/configuration.

Application 1 - Determine Cost and Cost Effectiveness, Laboratory Configuration for Synchronous Orbit Mission

Assumptions:

CSM capabilities: defined
Duration: 30 days
Weight allowable for experiments and support: lbs

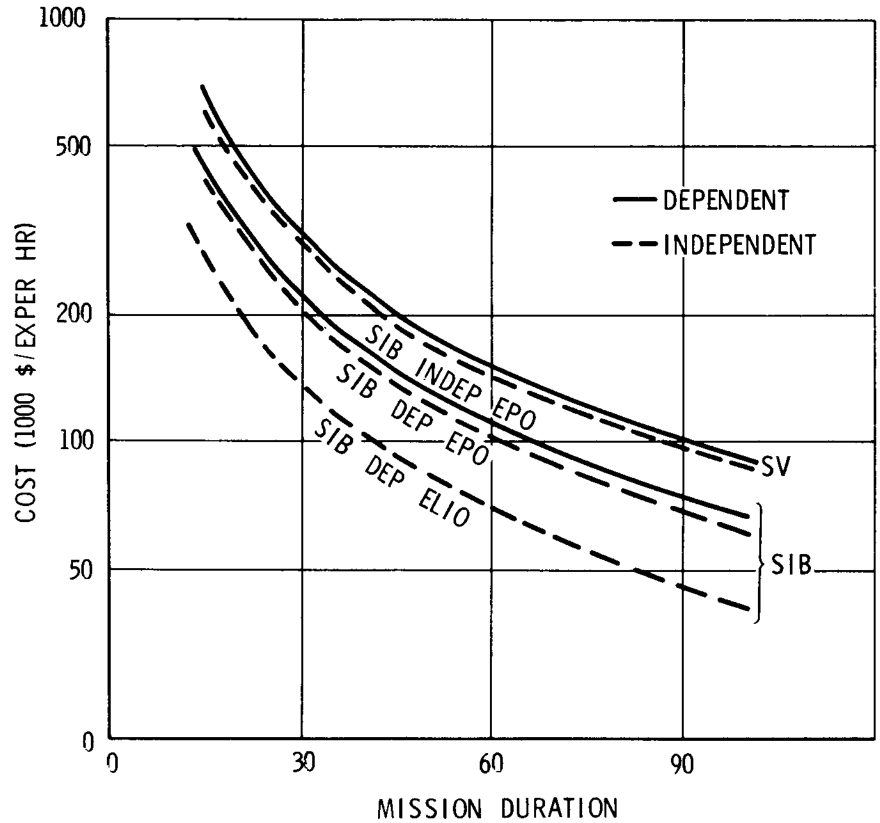

Figure 120. Cost Per Hour Versus Mission Duration

Mission Requirements:

EPS: 1500 kwh (experiments)
ECS: 30 days, two men
ACS: 50 hours fine, 150 hours coarse

Thermal recovery requirements: none

Communications and Data: bits

Subsystems Additions	Δ No.	Weight	Power	Cost
Life support ECS EPS ACS Communications				

Total Cost = Basic mission cost + cost of subsystems additions
Cost per day = Basic system cost/day + subsystems additions cost per day
Weight penalty = Experiment equipment

Application 2 - Determine Delta Cost for Added Performance

Requirements: 1000 additional kwh power required

Mission duration: No change

Subsystems Additions	Δ No.	Weight	Power	Cost
EPS Heat exchange				

Total cost = basic mission cost plus cost of subsystems additions
Cost per day = basic mission cost per day, plus cost per day for added subsystems
Payload penalty = total weight of subsystems additions

Application 3 - Define Specific Mission/Configuration

This is a first iteration for selection of an experiment group leading to selection of laboratory subsystems for experiment support.

Mission duration, flight plan, CSM support and weight allowable for experiments and support are defined. The laboratory includes the life support, ECS, and communications and data capabilities required, but not the EPS, ACS and SCS.

Weight and Support Requirements Summary

	Power for Experiments (kwh)	ACS (hours)	Astronaut (hours)	Experiment (weight)	Support (weight)	Total* (weight)
Total allowable	NA	NA	240	NA	NA	7000
Experiments A B C D etc.						

*Can be cumulative total remaining weight. Based on the totals presented, EPS, SCS, and ACS subsystems might be selected as a first iteration, and the weight and performance capability then determined of the laboratory with subsystems.

SUMMARY

The cost effectiveness analyses presented are general in nature since it was the desire of NASA not to include individual specific experiments or missions in this study. The cost effectiveness considerations have been limited to the following analyses:

1. Economic feasibility of renovating the command module for reuse as a command module for low-altitude earth orbit use and as a laboratory. For the laboratory, this analysis includes consideration of what is removed, what remains in the command module, what needs to be replaced in the basic laboratory, what choices these are, and associated costs

2. Identification of the options for replacements of the renovated subsystems and the costs associated with each of the options

3. Feasibility of the building-block or "shopping-list" approach, including problems related to removals and reinstallation of the "shopping list" items, the subsystems interfaces, and items requiring quantitative definition

4. Methods for using the "shopping-list" approach and study results to identify example mission configuration for each of the AAP mission categories