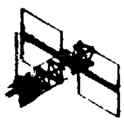
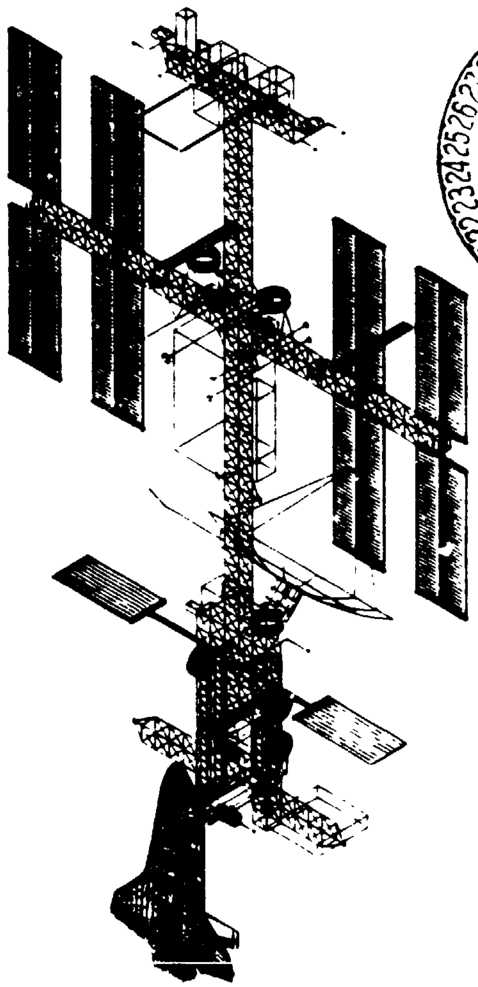
JSC-19989

Space Station Reference Configuration Description

(NASA-TM-87493) SPACE STATION REFERENCE
CONFIGURATION DESCRIPTION (NASA) 798 p
HC A99/MF E03 CSCL 22B
N85-31146
Unclas
G3/18 21826

Systems Engineering and Integration
Space Station Program Office





August 1984

National Aeronautics and
Space Administration

Lyndon B. Johnson Space Center
Houston, Texas

J8400076

JSC-19989

SPACE STATION REFERENCE CONFIGURATION DESCRIPTION

SYSTEMS ENGINEERING AND INTEGRATION
SPACE STATION PROGRAM OFFICE

AUGUST 1984

NATIONAL AERONAUTICS AND
SPACE ADMINISTRATION
LYNDON B. JOHNSON SPACE CENTER
HOUSTON, TEXAS

List of Acronyms and Abbreviations

ACA	Attitude Control Assembly
ACS	Attitude Control System
ARF	Animal Research Facility
ASTP	Apollo Soyuz Test Project
ATCS	Active Thermal Control Subsystem
CER	Cost Estimate Relationships
CMG	Control Moment Gyro
C&T	Communications and Tracking
DDT&E	Design, Develop, Test, and Evaluation
DMS	Data Management System
ECLSS	Environmental Control/Life Support System
ECS	Energy Conversion Subsystem
EMU	Extravehicular Mobility Unit
ESS	Energy Storage Subsystem
EVA	Extravehicular Activity
FCA	Flow Control Assembly
FF	Free Flyer
FOV	Field of View
GN&C	Guidance, Navigation, and Control
GPS	Global Position Satellite
GSE	Ground Support Equipment
HDR	High Data Rate
HM	Habitation Module
HMF	Health Maintenance Facility
HRF	Human Research Facility
Hz	Hertz (Cycles Per Second)
IF	Intermediate Frequency
IDMS	Information and Data Management Subsystem
IOC	Initial Operational Capability
ISA	Inertial Sensor Assemblies
LAB	Laboratory Module
LM	Logistics Module
LDR	Low Data Rate

LVLH	Local Vertical, Local Horizontal
m	Meter
MA	Multiple Access
MC	Manned Core
MLI	Multilayer Insulation
MMPF	Microgravity and Materials Processing Facility
MMU	Manned Maneuvering Unit
MRMS	Mobile Remote Manipulator System
MRWG	Mission Requirement Working Group
nm	Nautical Miles
OCZ	Operational Control Zone
ORU	Orbital Replacement Unit
OTV	Orbital Transfer Vehicle
OMV	Orbital Maneuvering Vehicle
PA	Power Amplifier
PAC	Pointing and Control System
PCU	Power Conversion Unit
POP	Perpendicular to Orbit Plane
PF	Platform
PID	Proportional, Integral, and Differential Gain
PMAD	Power Management and Distribution
RCG	Reference Concept Group
RFP	Request for Proposal
RMS	Remote Manipulator System
RCS	Reaction Control System
SAA	Science and Applications
SE&I	System Engineering and Integration
SS	Space Station
SSPE	Space Station Program Elements
STS	Space Transportation System
TBD	To Be Determined
TEA	Torque Equilibrium Attitude
TDM	Technology Development Missions
TDRS	Tracking and Data Relay Satellite
WCS	Waste Control System

CONTENTS

Page

1.0 INTRODUCTION

This report documents the data generated by the Space Station Program "Skunk Works" over a period of 4 months which supports the definition of a Space Station reference configuration. The data were generated to meet these objectives:

a. Provide a focal point for the definition and assessment of program requirements

b. Establish a basis for estimating program cost

c. Define a reference configuration in sufficient detail to allow its inclusion in the definition phase Request for Proposal (RFP)

Although the reference configuration has been specified in the RFP, the specific data defining that configuration in this document should be viewed as preliminary, subject to change, and presented for information only. It was NASA's intent in providing this information that it be considered as a potential point of departure for the definition phase, not a set of approved design solutions. Baselining of reference configuration characteristics will occur prior to definition phase Contract Start Date (CSD).

This report addresses the IOC and growth of the Manned Station and Unmanned Platforms. Section 3 presents a summary of the configuration description, subsystems, and key evaluation results for the Space Station and Platforms. Sections 4 and 5 contain key design requirements and descriptions of the configurations of the Manned Core Station and Platform, respectively. A functional evaluation is made of the configuration's ability to accommodate payloads, support crew and logistics operations, accommodate integrated systems requirements, and to be assembled and grown efficiently. Detailed subsystem descriptions with options considered and rationale used to select the specific subsystems, are also presented.

In the early phase of this effort, 14 technical areas were identified which served as "targets" for system/subsystem alternative evaluation and analysis. The data developed in these areas was considered too detailed for inclusion in this document. The study results of these technical areas are published as "Space Station Subsystem White Papers" (JSC-20054) and the titles of each paper are identified under the listing of reference documents in the back of this report.

2.0 BACKGROUND

The Space Station Program Office "Skunk Works" activity was initiated to support RFP preparation, to develop a reference configuration, and to provide supporting data bases. The Reference Configuration Group (RCG), a subgroup of the "Skunk Works" SE&I team, was tasked to develop configuration concepts and supporting data. As a point of departure and continuity from previous studies, five concepts were selected for evaluation from those identified by the NASA Headquarters Concept Development Group (CDG). They included the "CDG Planar", the "Delta-Truss", the "Big T-Truss", a cross-type structure referred to as the "Power Tower", and a configuration with a rotating solar array called the "Spinner".

Early in the study, the five configuration concepts were narrowed to three based on the finding that the salient features of all were incorporated in the remaining three. The three concepts selected for further study were the "CDG Planar", the "Delta-Truss", and the "Power Tower". For these configurations, mission functional requirements were converted to design requirements and detailed Space Station and Platform configurations were developed. Systems and subsystems were sized accordingly. Cost sensitivity studies were performed and the integration of subsystems, crew, and customer requirements into the Station configurations were investigated.

The "Power Tower" was finally selected as the reference configuration because it was seen as maximizing the accommodation of current user and growth requirements while demonstrating acceptable design and operations characteristics. It was also recognized that the "Planar" and "Power Tower" configurations are members of the same family which differ basically in their placement of the manned modules and experiment bases with respect to the articulated solar collection devices.

The RCG was comprised of approximately 80 technical specialists from various NASA centers, including JSC, MSFC, GSFC, KSC, LaRC, LeRC, JPL, and Hqs. A listing of personnel involved in the study is contained in Section 6.0 of the report.

3.0 SUMMARY

3.1 CONFIGURATION DESCRIPTION

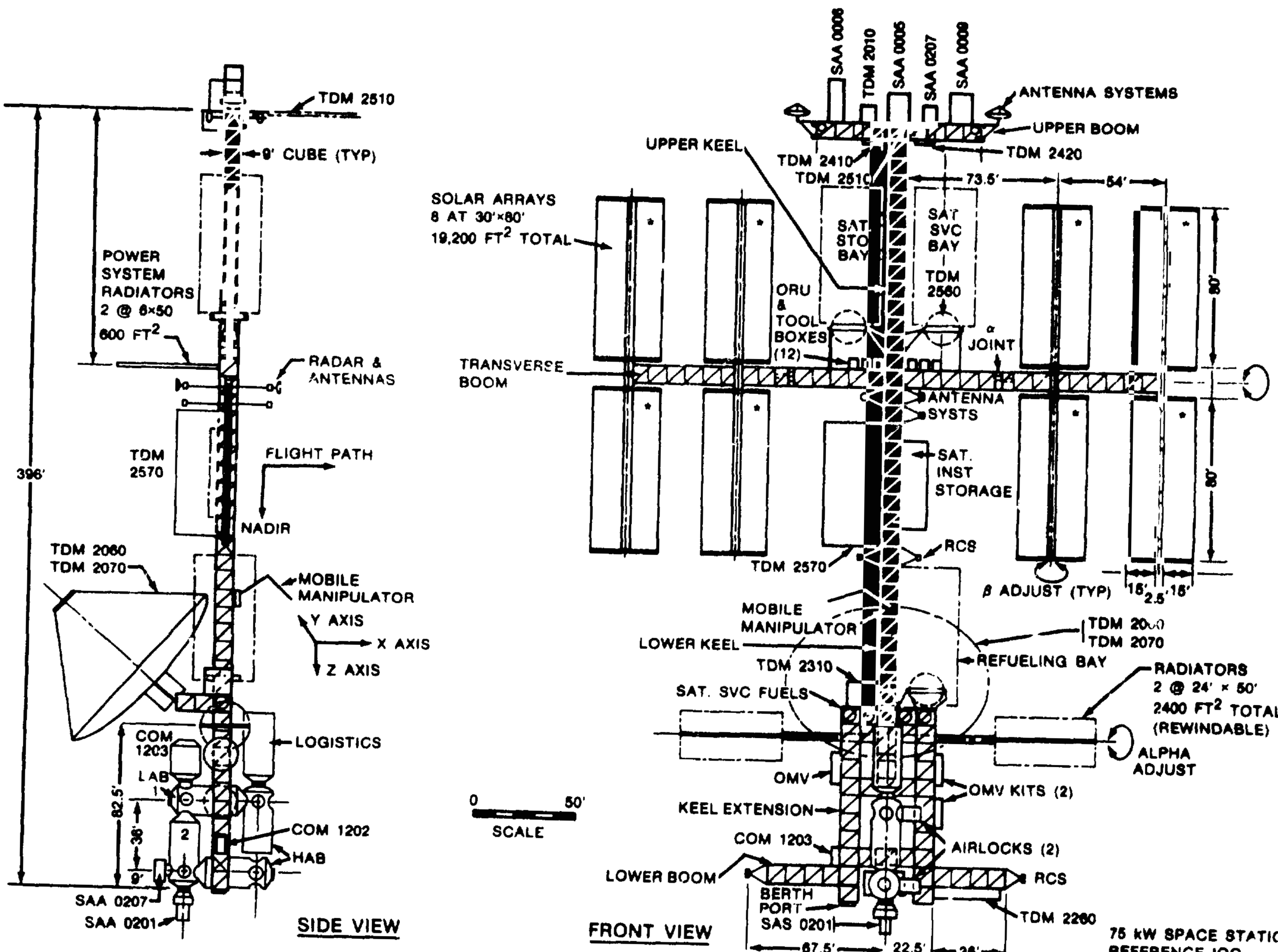
A brief overview of the Space Station reference configuration is provided in this section. More details are provided in subsequent paragraphs.

The reference IOC Space Station configuration is shown in Figure 3.1-1. The Space Station operates in a local vertical-local horizontal (LVLH) orientation, with its keel along the local vertical direction and the solar array boom perpendicular to the orbit plane (POP). The earth-pointed end of the Space Station contains earth-looking payloads. The zenith-pointed end contains solar, stellar, and anti-earth viewing payloads and communications antennas. Non-viewing payloads are located at various places on the Space Station and the pressurized modules are located near the bottom of the keel. Servicing equipment is located along the keel on either side, with the front and back surfaces of the keel kept free for traverse of the Mobile Remote Manipulator System (MRMS). The servicing and refueling facilities, OMV and OTV technology demonstration equipment, and satellite storage and equipment areas are located at various places along the structure.

Gimbaled solar array wings provide full power at any relative alignment of the Space Station and sunline. The solar voltaic power generation system was used in this study and is shown on the configuration to demonstrate a design option rather than to advocate the ultimate system selection. Heat rejection is provided by a combination of body-mounted radiators on the modules, deployed non-rotating radiators on the transverse boom, and deployed rotating radiators near the bottom of the keel.

The long, thin shape of the Space Station gives it a natural proclivity towards gravity-gradient attitude control. Aerodynamic forces and any unbalanced side-to-side masses tend to cause a tilt of the keel off the nadir-zenith line, but the impact on the Attitude Control System can be minimized by maintaining reasonable side-to-side balance in mass distributions, and by allowing the keel to tilt slightly in the orbit plane, if required. Locations for attachment of large transient masses, such as the Orbiter and large space construction elements, have been selected to minimize impacts on the Attitude Control System.

One of the principal advantages of this configuration is the good viewing afforded to all payloads, both externally-mounted and internally-mounted. The configuration also allows good accommodation of tether payloads and good accommodation of communication antennas. Good clearances are provided for Orbiter rendezvous and berthing, and for construction, servicing, and other operations activities. The deployable truss-mounted subsystems and distribution equipment are mostly pre-integrated (prior to launch) to minimize on-orbit time, complexity, risk, and especially EVA activities, during buildup and assembly. The module-mounted subsystems and distribution equipment are



*NOTE: SOLAR DYNAMICS SYSTEM A PRIME OPTION (NOT SHOWN)

Figure 3.1-1

ORIGINAL PAGE IS OF POOR QUALITY

pre-integrated also. Launch and assembly of the IOC Space Station requires from six to eight launches, including some but not all payloads. Assembly is accomplished using the Orbiter RMS and the Space Station MRMS after it is installed. Permanently manned operation can begin after launch 5.

The payload complement shown on Figure 3.1-1 is no doubt more extensive than that which will exist on the early Station, but it was utilized here as a test of accommodation capability. Although physical locations are provided for this quantity of payloads, Space Station resources such as power, data and crew time would have to be time-shared among the payloads.

Some options for the truss structure on the Station are shown in Figure 3.1-2. Some of these options are deployable, some are erectable, some are pre-integrated with subsystems, and some have subsystems installed on orbit after deployment of the structure. These are discussed in more detail in the structures section.

A weight summary of the Station is provided in Table 3.1-1. It should be noted that these weight numbers do not include growth. The payload weight is probably the softest of the numbers shown, since the payload complement used in the study was optimistic.

Five pressurized modules are utilized in the on-orbit configuration of the reference Station: two Habitation Modules, two Laboratory Modules, and a Logistics Module. In addition, a second Logistics Module must be provided in the program, for on-orbit exchange with the first one, to accomplish Station resupply every 90 days. Crew rotation or partial crew rotation will occur during resupply visits.

One Laboratory Module is characterized as primarily a Life Sciences research module and the other as primarily a Materials Sciences research module, for purposes of this study. However, either module can be modified on-orbit to support other activities.

The module arrangement is discussed in paragraph 3.3. Cost analyses have shown that considerable cost avoidance can accrue from having commonality among the pressurized modules. Cursory systems assessments have indicated that a fairly high degree of commonality should be possible. More details are provided later in the report.

A man-tended Station which is operated and maintained by a crew living in the Orbiter and which operates unmanned at other times may offer some cost deferral if pressurized modules on the Station are eliminated from the IOC version. Assuming that the Station will ultimately grow to a fully manned version, the overall configuration will not differ significantly from the reference concept described in this document (Fig. 3.1-3).

Platform concepts can be derived from the Station subsystems. Table 3.1-2 identifies some implications of trying to achieve different levels

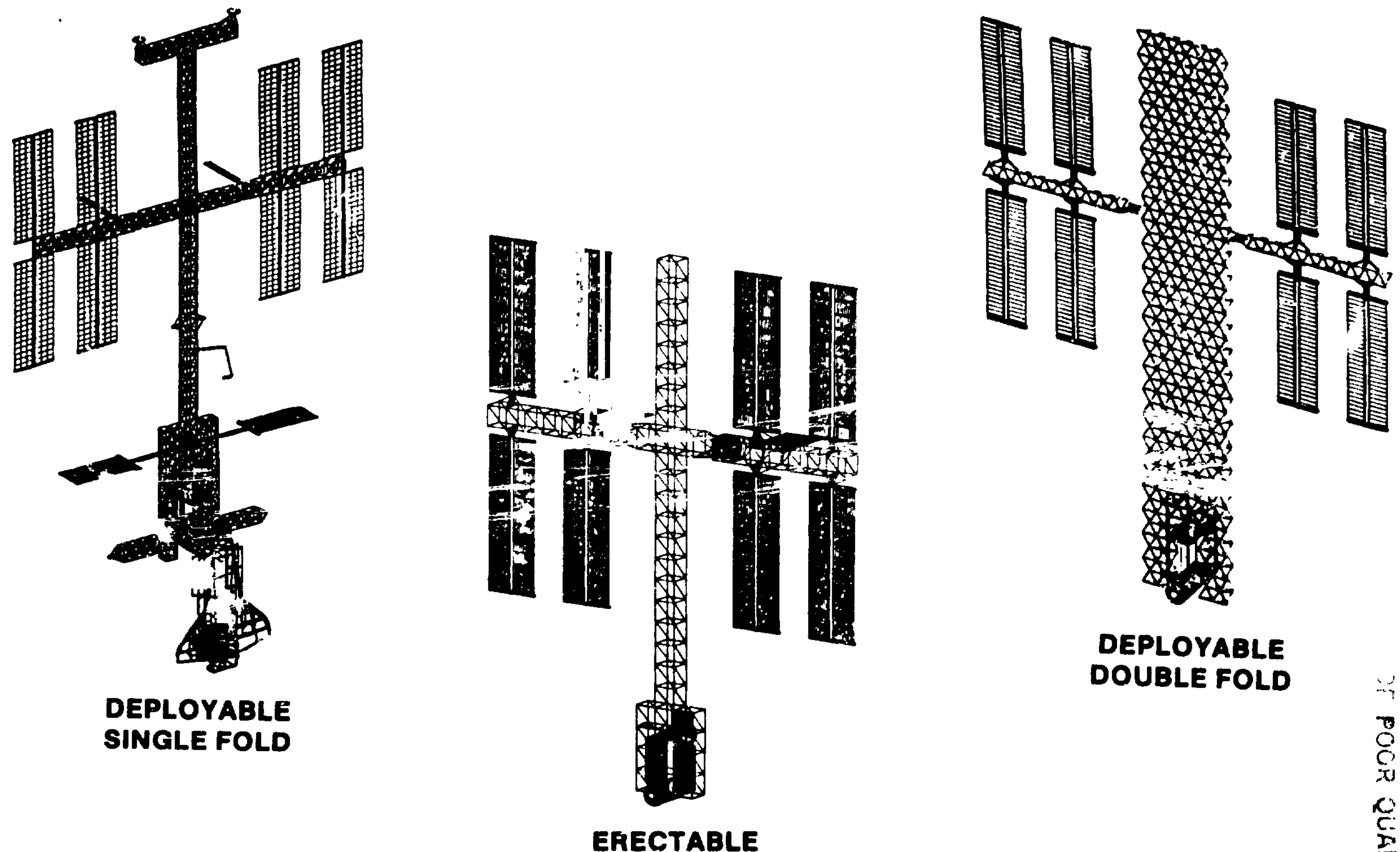


Figure 3.1-2 - REFERENCE STATION STRUCTURAL OPTIONS

TABLE 3.1-1 - IOC SPACE STATION WET WEIGHT (LBS)

SUBSYSTEM	HM1	HM2	LM	LAB1	LAB2	TRUSS STRUC	ATTACH COMPNTS	PAYLOADS	TOTAL
ELECT PWR	643	462	208	505	505	7,867	12,554	-	22,744
GN&C	-	-	-	-	-	3,734	-	-	3,734
COMM & TRK	834	2,106	65	564	564	658	2,304	130	7,225
DATA HANDL	1,005	765	120	805	605	-	1,250	-	4,550
PROPULSION	-	-	2.768	-	-	-	6,951	-	9,719
ECLSS	6,312	6,312	929	929	869	-	-	-	15,351
TCS	474	474	211	1,215	640	2,408	5,396		10,818
STR & MECH	12,964	12,964	7,310	12,964	12,964	15,316	3,224	-	77,706
CREW SYS	15,710	11,080	9,125	3,883	3,883	-	-		43,681
PAYLOADS	-	-	13,148	18,630	35,275	-	-	184,365	251,418
TOTAL	37,942	34,163	33,884	39,495	55,305	29,983	31,679	184,495	446,946

TABLE 3.1-2.- PLATFORM CONCEPT SPECTRUM
DEGREE OF COMMONALITY WITH SS

	HIGH	MEDIUM	LOW
Subsystem	-SS resource Module struct -SS subsystems	-Part of existing SS resource module structure	-New structure -Non-SS subsystems
SE&I	-Mostly existing	-Partially existing	-Partially existing-to-mostly new
Logistics/ Spares	-Common	-Partially common	-Non-common
Maintainability/ Maintenance	-Common	-Partially common	-Non-common
Growth	-High degree; already scarred	-High degree; partially scarred	-Probably limited; scars must be added
Size/Weight	-Largest & heaviest	-Medium size & weight	-Probably smallest and lightest
Cost	-Low D&D but manned CER's; cost TBD	-Higher D&D, mixed CER's; cost TBD	-Higher-to highest D&D, but unmanned CER's; cost TBD

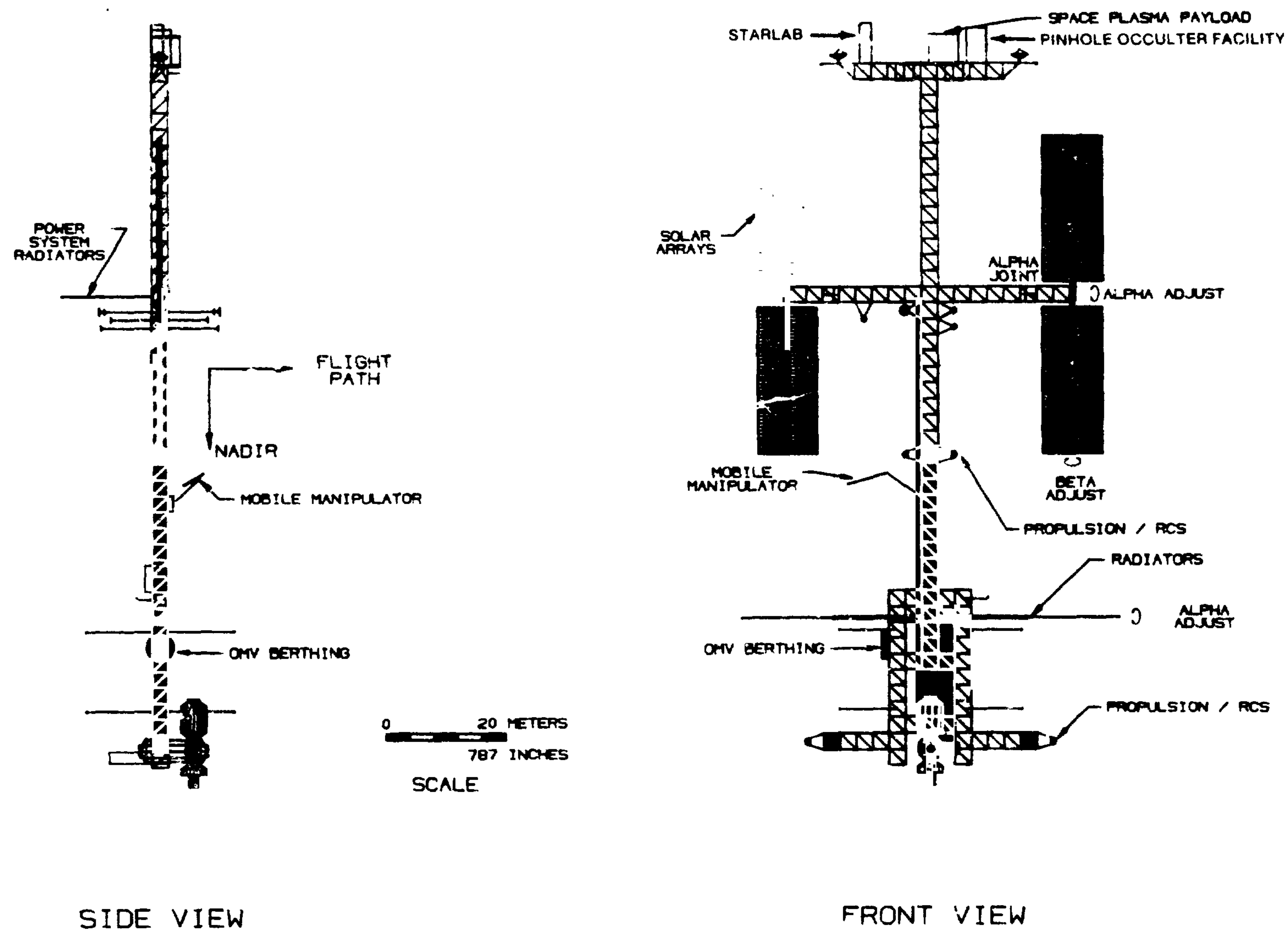


Figure 3.1-3 - MAN-TENDED CONFIGURATION

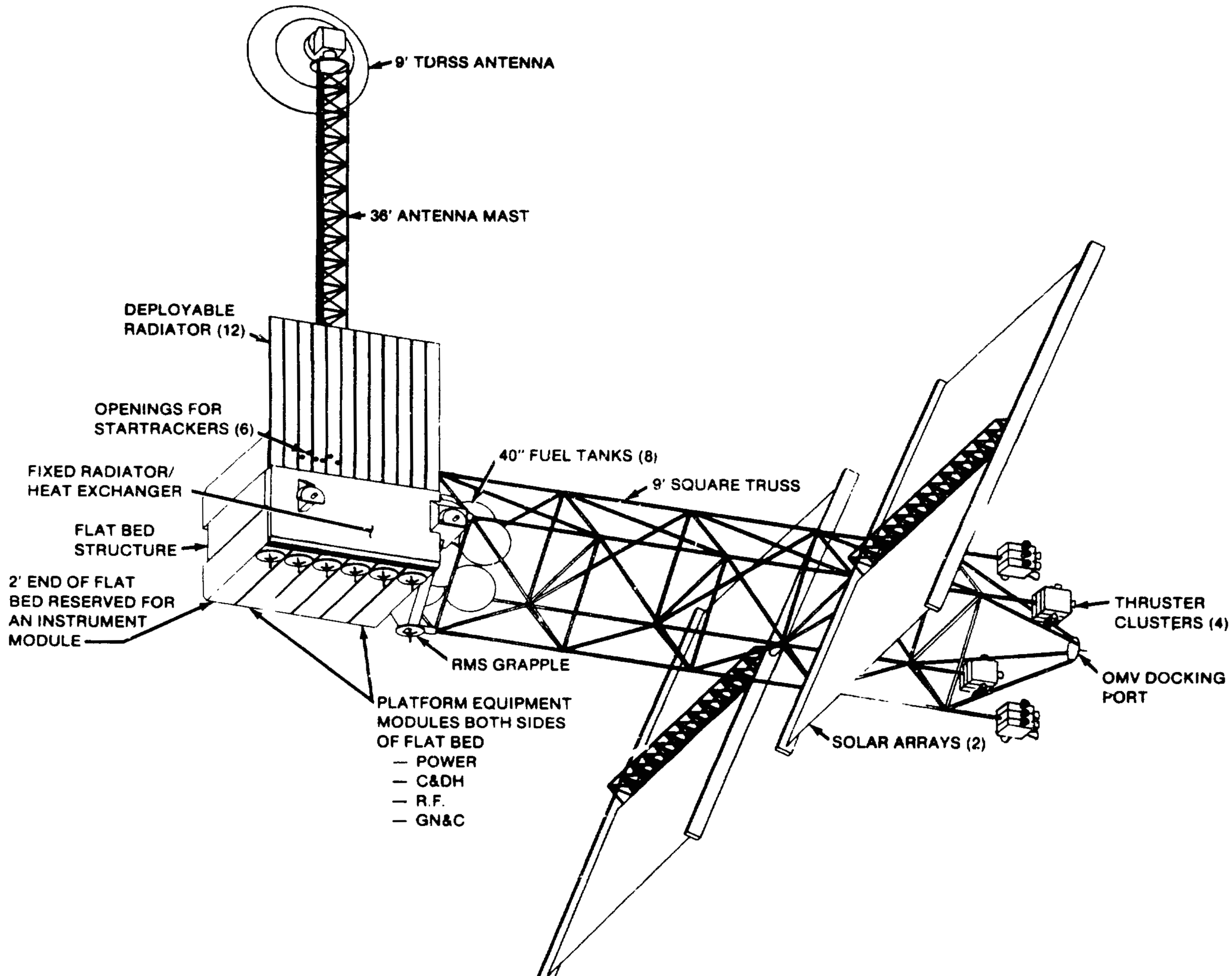


Figure 3.1-4.- CORE PLATFORM (WITHOUT PAYLOAD)

of commonality. Figure 3.1-4 shows a generic platform derived from the reference Station. Platforms that are sized to best meet the user needs in a cost-effective manner should be studied.

There is no "safe haven" per se in the Space Station. Rather, the Station meets the basic safety requirements of 1) operability and safety after the loss of any one module, 2) survivability of the crew for 22 days, and 3) rescuability by providing safe exit from and isolation of any one module from the others and sufficient life support, food, waste management, control/communications, and rescuability within the remaining three-module cluster.

Assessments of berthing/docking appear in the appropriate areas of the report (dynamics, mechanisms, operations, etc.). The growth version of the Space Station is discussed in later portions of the report.

3.2 SUBSYSTEM DESCRIPTION AND PERFORMANCE

A summary listing of some of the key features of the reference subsystems is provided in Table 3.2-1. Each of the subsystems is described in more detail in appropriate sections of the report.

The Station reference subsystems consist mostly of existing and state-of-the- art equipment/designs. However, in some areas, it was deemed to be more cost-effective to provide some advance in the state-of-the-art.

The Guidance, Navigation, and Control (GN&C) and Propulsion subsystems are probably the ones requiring fewest advances in the state-of-the-art. The Crew, Information and Data Management, Communications and Tracking (C&T), and Structures/Mechanisms subsystems require a moderate degree of technology/advanced development. The Electrical Power, Thermal Control, and Environmental Control and Life Support (ECLS) subsystems probably require more advancement than the other subsystems. Further study may indicate that changes should be made in the subsystem concepts. One subsystem concept which should be given significant attention in further studies is the solar dynamic concept of the Electrical Power subsystem. All presently identified technology advancement areas should be studied more thoroughly, to verify their expected costs and performance.

A top-level summary of the IOC SS performance in key subsystems areas is provided as Table 3.2-2. The flexible body frequency is low, but analyses have indicated that there should be no interaction with the GN&C or other subsystems. The number of Control Moment Gyros (CMG's) is very moderate. Housekeeping power is fairly sizable, but possibly can be reduced. Some payloads should be able to utilize the pointing accuracy and stability of the Station directly, without requiring pointing mounts, but others will require pointing mounts.

TABLE 3.2-1 - KEY FEATURES OF SS REFERENCE SUBSYSTEMS

ELECTRICAL POWER

- o PLANAR SOLAR ARRAY (DEPLOYABLE)
- o REGENERATIVE FUEL CELLS
- o GIMBALED ARRAY WINGS (2 AXES)
- o HIGH VOLTAGE AC DISTR.

GN&C

- o DOUBLE-GIMBALED CMG'S
- o MAGNETIC TORQUERS
- o RATE GYROS, STAR-TRACKERS ACCELEROMETERS
- o CONTROL SIGNALS TO RCS (REBOOST & BACKUP ACS)

PROPULSION

- o REBOOST & ACS (BACKUP THRUSTERS
- o PROPELLANT STORAGE & FEED EQUIPT.
- o BLOWDOWN SYSTEM

ECLS

- o REGENERABLE CO_2 CONCENTRATION REDUCTION
- o N_2 STORAGE & RESUPPLY
- o CONDENSATE, WASH WATER, & URINE/FLUSH WATER RECOVERY
- o WATER ELECTROLYSIS
- o WASTE MGT. PROCESSING & STORAGE

THERMAL CONTROL

- o CENTRALIZED/DECENTRALIZED HEAT REJECTION
- o FIXED/ORIENTABLE/DEPLOYED/NON-DEPLOYED RADIATORS (ALL HEAT PIPES), SOME CONSTRUCTABLE
- o 2-PHASE HEAT TRANSPORT
- o THERMAL CAPACITOR

STRUCTURES/MECHANISMS

- o 30' (CYL, PORTION) PRESSURIZED MODULES
- o DEPLOYABLE TRUSSES
- o ERECTABLE STRUCTURE (MINIMIZED)
- o BERTHING
- o MOBILE RMS
- o ROTARY JOINTS (ELECTRICAL & FLUID)

INFORMATION AND DATA MANAGEMSOT

- o HOUSEKEEPING & PAYLOAD DATA BUSES
- o HIERARCHICAL SYSTEM
- o FIBER-OPTIC NETWORK
- o FIXED AND PORTABLE CONTROL CONSOLES
- o SOME FLAT-SCREEN DISPLAYS

COMM & TRACKING

- o KU-BAND & S-BAND (MULTI-FREQUENCY)
- o DIGITAL COMMUNICATIONS
- o MULTIPLE-STEERED-BEAM ANTENNAS
- o WIRELESS CREW COMM
- o RADAR TRACKING & OPTICAL BERTHING
- o ENCRYPTION/DECRYPTION

CREW SYSTEMS

- o PRIVATE CREW QUARTERS
- o WARD ROOM & GALLEY (INCLUDING FOOD FREEZER & DISHWASHER)
- o HEALTH MAINTENANCE FACILITY
- o AIRLOCK(S) WITH HYPERBARIC CHAMBER
- o EVA SUITS, WITH STOWAGE/CLEANING/ MAINTENANCE
- o WASTE MANAGEMENT COLLECTION
- o SHOWERS & HAND WASHERS
- o CLOTHES WASHER & DRYER
- o TRASH MANAGEMENT SYSTEM
- o EQUIPMENT MAINTENANCE FACILITY
- o VEHICLE MANAGEMENT
- o SAFE HAVEN

PAYLOAD & SERVICING ACCOMMODATIONS

- o BASIC LAB VOLUME
- o ATTACHED UNPRESSURIZED PAYLOADS
- o ATTACHED PRESSURIZED PAYLOADS
- o OMV (IOC), MMU (IOC),& OTV (GROWTH)
- o SERVICING OF FREE-FLYING SATELLITES & PLATFORMS, OMV, MMU, & OTV
- o CONSTRUCTION OF LARGE STRUCTURES

TABLE 3.2-2 - IOC SS PERFORMANCE SUMMARY

- FLEX BODY FUNDAMENTAL FREQUENCY: 0.123 Hz
- NUMBER OF SKYLAB CMG'S REQUIRED FOR CONTROL:
  - WITHOUT ORBITER ATTACHED: 3
  - WITH ORBITER ATTACHED: 4
- IOC POWER AVAILABLE
  - TO PAYLOADS: 50 kW
  - HOUSEKEEPING: 25 kW
- IOC HEAT REJECTION AVAILABLE
  - FOR PAYLOADS: 50 kW
  - HOUSEKEEPING: 50 kW
- POINTING
  - ACCURACY: $\pm 1°$
  - STABILITY RATE: $\pm$ .02 DEG/SEC
- REBOOST
  - THRUST LEVEL: 100-300 LB
  - REBOOST FREQUENCY: 90 DAYS
- COMMUNICATIONS
  - DOWNLINK: 300 MBPS (LESS ENCODING)
  - UPLINK: 25 MBPS (LESS ENCODING)
- DATA STORAGE: HOUSEKEEPING ONLY
- CREW TIME
  - FOR PAYLOADS: 5 PEOPLE
  - FOR HOUSEKEEPING: 1 PERSON

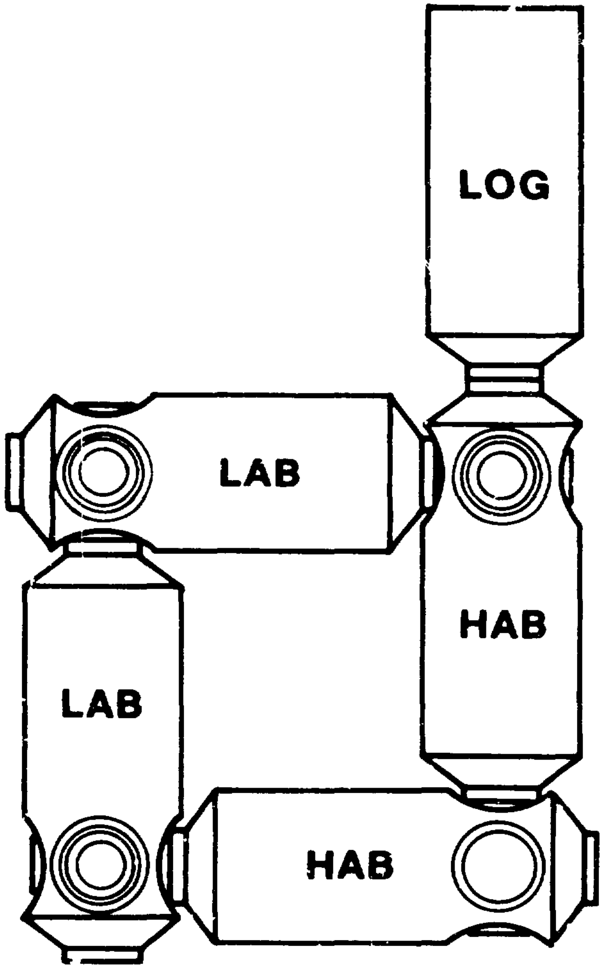
The thrust level during reboost varies from 300 pounds with full tanks to 100 pounds later, since a blow-down system is utilized. Reboost occurs every 90 days. The communications capability of the SS is that of the TDRSS link. Nominal crew time expected to be available for payloads is four equivalent people out of a six-person crew, with the fifth and sixth person used for SS housekeeping and miscellaneous activities.

3.3 MODULE ARRANGEMENT

Figure 3.3-1 depicts the module arrangement used for the reference configuration. This arrangement provides a "racetrack" configuration, i.e., each module (except the Logistics Module) has two exits. Although not a requirement, this probably enhances safety in the event of an emergency evacuation of any module. There is a high degree of module commonality, particularly among the four modules in the racetrack. This results in the fewest number of module types being required. This arrangement also provides a minimum total number of elements and a minimum number of interfaces between elements. Traffic through the Laboratory Modules is low, with the majority of traffic being in the two Habitation Modules. Traffic considerations and interface/integration considerations seem to make it preferable to have the Logistics Module and Orbiter berthed to the Habitation Modules, and to have the pressurized payload modules berthed to the Laboratory Modules.

The above-described arrangement was selected after a fairly brief assessment early in the study to provide a reference module arrangement which could be utilized on all Space Station concepts being assessed. Some of the other module arrangements which were investigated during this assessment are shown in Figure 3.3-2. The first option utilizes "corner cubes" to join modules together, which increases the total number of elements, the number of element types, and the number of interfaces. Total module volume in this option is expected to be inadequate unless the modules are lengthened, and cost is expected to be higher. The second option has two different types of modules, which increases costs. This option also requires traffic to pass through the shorter (Laboratory) modules to get from one longer module (Habitation) to the other (Habitation). This provides a very undesirable traffic flow. The third option shown in Figure 3.3-2 is the one selected as the reference, for the reasons already discussed.

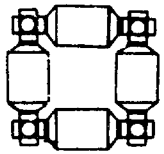
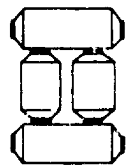
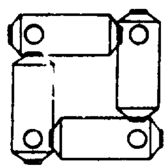
Figure 3.3-3 shows several other module pattern concepts arranged on a large truss, such as the reference Station keel. Some of these utilize interconnecting tunnels or other elements, which may provide some advantages but probably would increase costs. None of these has been assessed in this study, but these and other potential arrangements should be studied further in order to arrive at the best module arrangement possible within the programmatic and technical constraints.



KEY FEATURES

- RACETRACK
- HIGH MODULE COMMONALITY
- MINIMUM TOTAL NUMBER OF ELEMENTS
- MINIMUM NUMBER OF INTERFACES
- TRAFFIC THROUGH LABS MINIMIZED
- USABLE VOLUME IN MODULES IMPACTED BY PORTS

Figure 3.3-1.- REFERENCE MODULE PATTERN

OPTION		CRITERIA					
		ADEQUATE SPACE	CONFIGURA-TION COMPATIBLE	RACE-TRACK	**MIN MODULE DESIGNS	**MIN MODULES IOC	LOW TRAFFIC IN HAB, LAB
I.	 4 SHORT MODULES (20 FT) 4 INTERCONNECT MODULES/ AIRLOCKS		X	X*			X
II.	 2 SHORT MODULES (20 FT) 2 LONG MODULES (40 FT) INTERNAL AIRLOCKS	X	X	X		X	
III.	 4 MEDIUM MODULES (30 FT) INTERNAL AIRLOCKS	X	X	.	X	X	X

*RACETRACK INTERRUPTED BY AIRLOCK OPERATION
**STRONGEST DRIVERS

Figure 3.3-2 - MODULE/MODULE PATTERN SELECTION

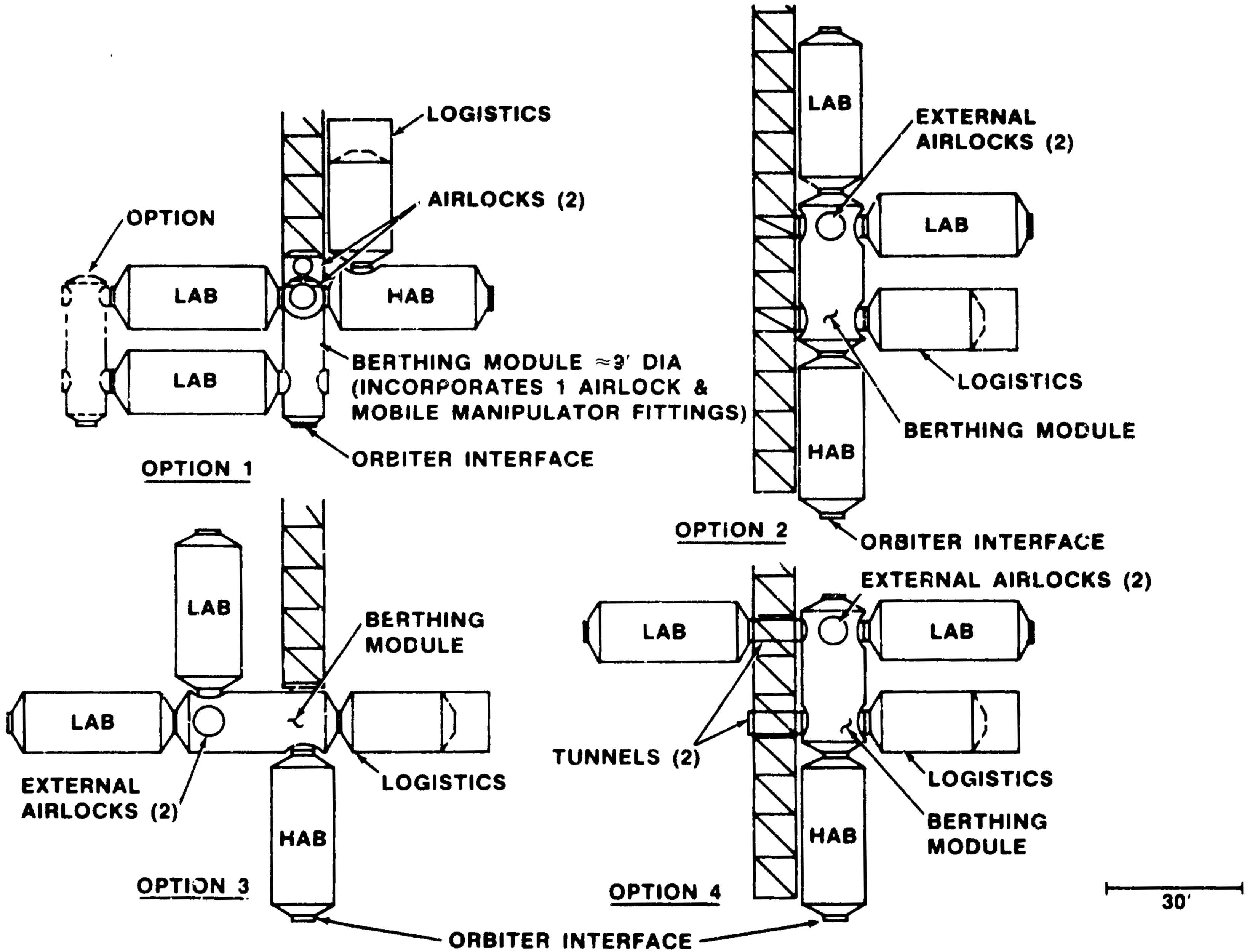


Figure 3.3-3 - TYPICAL MODULE PATTERN OPTIONS (CONT)

The number of berthing ports utilized per module and the function of each port are defined in Table 3.3-1. Two radial and two axial ports per module would satisfy most existing requirements and provide for slight growth. The module internal layouts provided later in the report allow for four radial ports per module, one of which is blocked off to increase usable internal volume. This allows for more growth, but provides a slightly more pessimistic picture of available internal volume, since a good amount of volume has been allowed for each radial port.

FUNCTION	HAB #1		HAB #2		LAB #1		LAB #2	
	AXIAL	RADIAL	AXIAL	RADIAL	AXIAL	RADIAL	AXIAL	RADIAL
• RACETRACK	1	1	1	1	1	1	1	1
• EXTERNAL AIRLOCKS		1		1				
• ORBITER	1			1*				
• LOGISTICS			1					
• 2 PRESSURIZED PAYLOADS (PER REQ'MTS.)					1		1	
SUBTOTAL	2	2	2	3	2	1	2	1

TO MAXIMIZE INTERNAL VOLUME AVAILABLE, AND TO MAXIMIZE MODULE COMMONALITY ASSUME 2 AXIAL AND 2 RADIAL PORTS PER MODULE

* BACKUP PORT FOR ORBITER AND/OR LOG MODULE

Table 3.3-1 - NO. OF PORTS REQUIRED

4.0 SPACE STATION MANNED CORE

4.1 DESIGN REQUIREMENTS

The primary design requirement for the Space Station is to provide a versatile, growing, permanent, manned facility in space with the capability to enable the efficient accomplishment of significant advances in space science, technology, commerce, and transportation. Specifically, the Space Station is required to implement the following objectives.

a. Establish the means for a permanent and productive presence of people in space.

b. Establish routine, continuous, and efficient utilization of space for science, applications, technology development, and operations (including servicing and refueling of space assets).

c. Develop further the commercial utilization of space.

d. Develop and exploit the synergism effects of the man/machine combination in space.

e. Provide essential system elements and operational practices for an integrated national space capability.

f. Stimulate the mutual benefits traditionally derived from cooperation in space with our allies and friends.

g. Reduce the cost and complexity of living in and using space.

h. Ensure leadership in space for the United States in the decade of the 1990's and beyond.

Design requirements have been derived from these objectives, both in terms of an Initial Operational Capability (IOC) and a growth capability. Prime consideration has been given to customer requirements and the operations requirements, and system requirements have been derived from these. At the basic level, the mission, or customer requirements, have been interpreted as necessitating three separate spacecraft: a permanently-manned Space Station in a 28-1/2° inclination, 270 nmi circular orbit; an unmanned co-orbiting platform, which is in a rendezvous-compatible orbit with the Space Station and a second platform in a 98°, 430 nmi, Sun synchronous (2 p.m.) orbit. The polar orbiting platform configuration characterized in this study contains primarily Earth observing payloads, but is intended to be sufficiently versatile to support other payloads and be operable in other near-polar orbits. The platform which is in the rendezvous-compatible orbit with the Space Station is characterized in this study as accommodating payloads for celestial and solar observations and demanding microgravity experiments with a minimum of mechanical disturbance and contamination. The platform is versatile enough to accommodate other types of payloads, if desired.

The manned Space Station element of the program must be designed to meet all the remaining program objectives, as well as providing for the development of sensors and systems for Earth observations and a base for some celestial, solar, and Earth observations which require a manned presence. To meet these objectives, the configuration design must provide for (1) mounting, pointing, servicing instruments which continuously view the Sun, the entire Earth's disk, and the celestial sphere, and provide this capability simultaneously except when prohibited by occultation; (2) significant continuous power for commercial and scientific functions; (3) significant pressurized volume for laboratories and the crew required to service all users; (4) the capability to act as a "Space Port" for the Shuttle, various free-flying satellites, the co-orbiting platform, and space transportation vehicles such as the OTV and OMV; (5) the ability to store, maintain, assemble, and reconfigure vehicles and payloads of all types; (6) the ability to service and refuel free-flying spacecraft, platforms, and attached payloads; and (7) attachment and services to various unpressurized scientific and commercial payloads. Specifically, at IOC, the manned Space Station element is required to provide at least 50kW of continuous power, a minimum of four crew persons to service customers, at least 45 m^3 of laboratory volume, an OMV, ports for attachment of two pressurized customer payloads, attachments and service interfaces for at least seven unpressurized instrument payloads, and provisions for large space construction and OTV development.

The work set forth in this report is directed toward a manned Space Station, and implicitly assumes that early manning is desirable. As a result, the possibility of a man-tended Station has not been explored in this study. "Man-tended" is taken to signify a mode in which the Station is visited at intervals by the Orbiter and operates unmanned the rest of the time. This mode offers some potential cost deferral if the pressurized modules are deleted from the IOC Station altogether. This deletion would require the crew to live entirely out of the Orbiter during their stay on orbit. If pressurized modules are included in the man-tended Station to enhance on-board operations and maintenance, there is little or no cost saving associated with it.

Inherent in the program objectives is that the manned Space Station have the capability to grow in its ability to provide basic services to all types of customers, and to accommodate additional users of foreseen and unforeseen classes. For design purposes, this requirement has been defined as approximately twice the power level, crew size, and number of pressurized payloads of that expected at IOC; increases in the number of unpressurized attached or serviced payloads have not been specifically defined, but the major areas of growth in the types of attached payloads are expected to be in the size and complexity of satellites to be serviced, in the size of construction and assembly projects, and in the addition of increasingly capable OMV-and OTV-type vehicles.

The requirement that the Space Station become a permanent facility implies that, in addition to growth capability, it must also possess the characteristic of being able to incorporate advances in technology as they occur. This compatibility with "modernization", or technology transparency, is then considered a significant design driver. Although this requirement basically applies to all systems and subsystems of the manned core Space Station, its influence on configuration is reflected primarily in the power system area, and its influence on the module design primarily in the ECLSS, since it is anticipated that these areas will be particularly subject to post-IOC development, and because provision must be made in the basic design for significant changes in the systems.

To facilitate the definition of specific configurations for the manned Space Station and platforms, a set of payloads for IOC has been identified to serve as a "pathfinder" in design. This set is shown in Table 4.1-1. This set is not taken as an actual expected manifest at IOC, so much as it is intended to encompass the types of near-term payloads expected to appear on the SSPE's. It will be noted that these "pathfinders" represent specific cases for all the generic user requirements. The Starlab (SAA0006) is representative of a celestial-viewing instrument package, the pinhole facility (SAA0009) of a solar observation instrument, and the plasma payload (SAA0207) of a simultaneous viewing requirement, while the earth observation technology experiment (TDM2260) demands a broad, continuous view of Earth. Microgravity requirements are represented by the ECG production unit (COM1203), the material processing laboratory (COM1201), and the EOS production unit (COM1202). The group of users for which the Space Station provides a base for maintenance, construction, repair, and servicing is represented by the Deployment/Assembly/Construction Technology project (TDM2060) and the servicing technology projects (TDM's 2560 and 2570).

In addition to these technology experiments, a definition of satellite servicing requirements has been developed for use in preliminary definition of a reference configuration for the manned Space Station. This definition includes storage and servicing capabilities and is intended to provide for support of the specific satellites shown in Table 4.1-2. These satellites are intended to represent the range and types of systems and vehicles to be accommodated at IOC; they are not considered as either exclusive or inclusive of the actual satellites which will be serviced during the IOC period.

Closely associated with the customer requirements are operational requirements, some of which also affect the overall configuration of the Space Station and the unmanned platforms. The primary requirement is that the SSPE's must all be launched, and supported by the Space Shuttle. This requirement has a significant effect on the assembly sequence and the modularization of the structure and subsystems. Support by the Shuttle implies that the design of the SSPE's must provide convenient means for rendezvous, berthing, and loading/unloading of the Orbiter. In addition to the Orbiter, the Space Station must also be configured to perform many of the same operations with free-flying satellites, the OMV, and the OTV (in the growth phase). The communications requirements are built around the customer data requirements for near-continuous data transmission through TDRS and for communication and tracking of the Orbiter and free-flying unmanned satellites, vehicles, and EVA crewmen.

The majority of the system requirements have more effect on the detail of subsystem design than on the overall configuration concept. However, the requirement for safe operation with the loss of one pressurized module affects the arrangement of the modules. The general requirements for commonality affect the modularization of the structure and subsystems, and the requirements for maintainability affect the location of components and EVA and manipulator provisions.

TABLE 4.1-1 - SPACE STATION MANNED CORE IOC PAYLOADS

ATTACHED/EXTERNAL PAYLOADS

SAA0005	Transition Radiation & Ion Calorimeter (TRIC)
SAA0006	STARLAB
SAA0009	Pinhole Occulter Facility
SAA0201	LIDAR Facility
SAA0207	Space Plasma Payload (SSP)
COM1203	ECG Production Unit
TDM2010	Materials Performance
TDM2060	Deployment/Assembly/Construction Technology
TDM2070	Structural Dynamics Technology
TDM2260	Earth Observation Instrument Technology
TDM2310	Fluid Management Technology
TDM2410	Attitude Control Technology
TDM2420	Figure Control Technology
TDM2510	Environmental Effects
TMD2560	Satellite Servicing Technology
TDM2570	OTV Servicing Technology

PRESSURIZED PAYLOADS

SAA0307	Life Science Laboratory
SAA0401	Microgravity Research Laboratory
COM1201	MPS Lab #1 (Materials Processing)
COM1202	EOS Production Unit
TDM2020	Materials Processing Technology
TDM2520	Habitation Technology
TDM2530	Medical Technology

TABLE 4.1-2 REPRESENTATIVE IOC SATELLITES REQUIRING SERVICING

CODE	MISSIONS
SAA0012	Space Telescope (ST)
SAA0013	Gamma Ray Observatory (GRO)
SA00014	X-Ray Timing (XTE)
SAA0016	Solar Maximum Mission (SMM)
SAA0019	Far Ultraviolet Spectroscopy Explorer (FUSE)
SAA0017	Advanced X-Ray Astrophysics Facility (AXAF)
	Leasecraft 1 and Payloads
	Leasecraft 2 and Payloads
	Leasecraft 3 and Payloads
	Leasecraft 4 and Payloads
	Leasecraft 5 and Payloads

Taken as a whole, the design requirements dictate that the manned core station be an assembly of significant overall dimensions, with large areas for energy collection and heat rejection, multiple pressurized modules, areas of unpressurized structure, and means for connecting elements for orientation of the various elements and also providing access and clearance for viewing and operations. The requirements for the polar and co-orbiting platforms demand small areas for energy collection and heat rejection and no pressurized volumes. But a high priority is placed on the viewing needs of observational payloads and must be compatible with support by the STS and/or the OMV.

4.2 CONFIGURATION DESCRIPTION

4.2.1 General Arrangement.

The Space Station reference configuration is a set of deployed linear trusses to which pressurized modules, subsystems, and user equipment are attached. The principal structural components are a keel and three booms at right angles to the keel. The IOC configuration is illustrated in Figure 4.2.1-1. The coordinate system is Z parallel to the keel, positive toward nadir; X perpendicular to keel and booms, positive in the direction of flight; and Y parallel to the booms, positive to starboard. The origin is at the center of the intersection of the keel and transverse boom.

Four of the pressurized common modules (two habitation and two laboratory) are arranged in a quadrangle to permit IVA crew movement in case any one module becomes unusable. The keel is divided at the bottom to allow installation of the pressurized modules on the centerline of the keel. This maintains the principal axis in the orbit plane while the Orbiter is berthed, avoiding excessive yaw and roll excursions.

The materials processing module (Lab 2) is located at the top of the quadrangle to keep it as close as possible to the mass center of the station. The control station is at the front of HM2 for a direct view of berthing operations. The HM's are adjacent to each other to eliminate the need for excessive crew movement through the lab modules. The logistics module (LM) is berthed to HM2 to permit unloading with minimum disturbance to laboratory operations.

Primary Orbiter structural interface is at the end of HM1 as shown in Figure 4.2.1-1. An alternate position at the end of Lab 1 is provided for emergency access in case the primary port is inoperable. Both positions allow payload removal without interference.

External airlocks are berthed to the port side hatches of HM1 and HM2. This location leaves the entire forward face of the starboard keel extension open for manipulator travel.

The configuration in Figure 4.2.1-1 employs a photovoltaic power generation system. A solar dynamic system has also been considered as an option for IOC. Such a system is illustrated in Figure 4.2.1-2. The arrangement is identical to the photovoltaic version except for the power system. Discussions in the following sections, with that exception, should be understood to apply to both IOC configurations.

A sample growth configuration is shown in Figure 4.2.1-3. The principal differences are a solar dynamic power system providing 300 kW average power, two additional habitation modules and four additional lab modules. The solar dynamic power system is shown to demonstrate a design option and is not intended to advocate a particular power system growth path. The lower keel is also expanded from one bay wide to three bays for improved structural redundancy and stiffness. Additional lower booms are placed at the ends of the keel extensions to preserve the field of view of the earth sensors. Propulsion system relocation to these booms may be necessary to maintain the Station's mass center between the thrusters.

4.2.1.1 Subsystem Installation

The electrical power system is installed on the transverse boom (see Fig. 4.2.1-1) outboard of the alpha gimbal joints, including power generation, storage, and conditioning. This permits power transfer across the alpha joint to be limited to fully-conditioned AC power. Each array wing is gimbaled individually for beta adjustment to minimize principal axis shifts and to simplify assembly and deployment.

Primary heat rejection is provided by radiators on booms mounted on the lower keel. The radiators are rotated to maintain an edge toward the Sun. They are rewound during the dark portion of the orbit to avoid a continuously rotating fluid joint.

TDRSS and GPS antennas are mounted at the ends of the upper boom for maximum upward view. Tracking and rendezvous antennas are mounted on the lower keel under the transverse boom for a clear view forward and aft along the flight path.

The Attitude Control Assembly (ACA) is a rigid 9-foot cube at the intersection of the keel and the transverse boom. It contains the control moment gyros, star trackers, and other guidance, navigation and control components needed for control of the Station from the initial launch.

Identical sets of propulsion thrusters are mounted at four locations at the ends of the lower boom and on the lower keel, capable of firing forward, aft, and outboard. The lower thrusters (on the lower boom) normally provide all backup attitude control. The upper thrusters normally fire only aft, for orbit maintenance. Orbit maintenance maneuvers are performed at approximately 5 a.m. and 5 p.m. (orbit time) to minimize plume impingement from the upper thrusters on the solar arrays.

A Mobile Remote Manipulator System (MRMS) is capable of moving on the nodes of the truss to any location on the forward faces of the keel and booms. By moving outboard of the alpha joint and rotating the transverse boom, the MRMS can also move along the aft face of the structure.

4.2.1.2 Payload Accommodations

Two primary locations are provided for observational users (see Fig. 4.2.1-4). Solar and stellar sensors are mounted along the upper boom to maximize their view of that part of the sky not occulted by the Earth and to minimize contamination from activities at the inhabited end of the Station. Some mutual blockage of one stellar or solar instrument by another will occur in the region of the orbit poles. Any such lost area is relatively small and can be made up within a week or two as the Station orbit regresses in longitude. Horizon-to-horizon viewing of objects away from the orbit poles is possible for all instruments without blockage. Solar viewing is possible on every orbit because the Sun is always within 52° of the orbit plane. Earth sensors are mounted on the lower boom; local vertical orientation allows continuous viewing limb-to-limb with no image rotation.

Some solar sensors may also be mounted on the transverse boom outboard of the alpha gimbal to simplify pointing if their mass characteristics do not adversely affect the Station's dynamic properties.

Satellite servicing provisions are located along the keel. Two storage/servicing bays are situated parallel to the upper keel. This location minimizes contamination of optical surfaces and other sensitive components. Related ORU and tool storage is provided nearby on the transverse boom. A refueling bay is located on the lower keel. Satellite propellants and cryogens are stored at the top of the keel extension near the refueling bay.

The OMV and OMV kits are berthed alongside the port and starboard keel extensions.

Large structure construction is carried out on the aft face of the lower keel. Ample space for storage of materials and equipment is available on the keel extensions.

Several berthing ports are available on the laboratory and habitation modules for attachment dedicated user modules requiring IVA access.

4.2.2 Flight Modes

In normal operation, the Station is oriented with the keel approximately aligned with local vertical and the Y principal axis (approximately parallel to the transverse boom) held perpendicular to the orbit plane (POP) as shown in Figure 4.2.2-1. A constant pitch attitude is maintained to balance gravity gradient and average aerodynamic torques. CMG's are used for momentum management to compensate for aerodynamic torque variations around the orbit. Long-term (more than a few orbits)

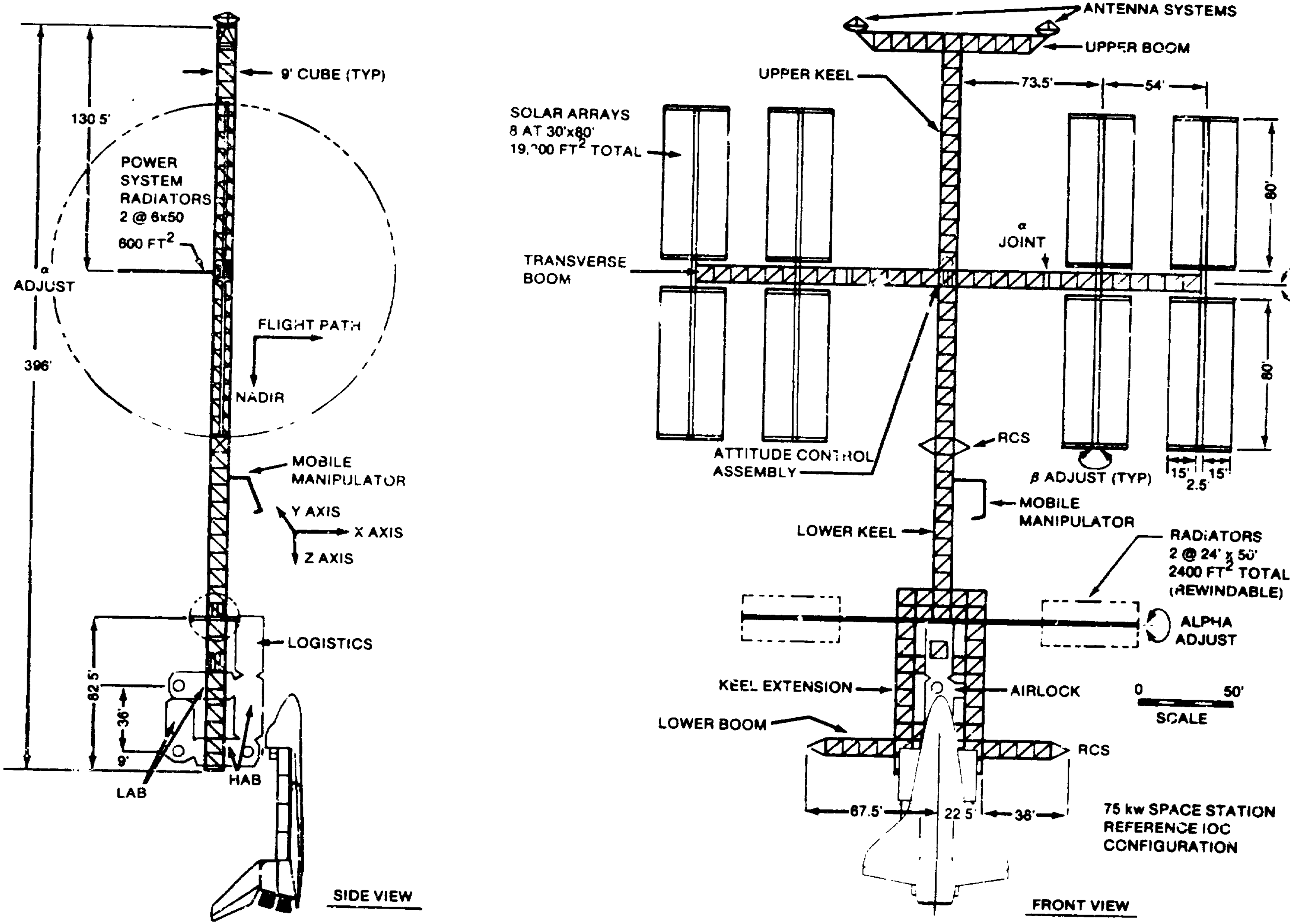


Figure 4.2.1-1 - IOC REFERENCE CONFIGURATION

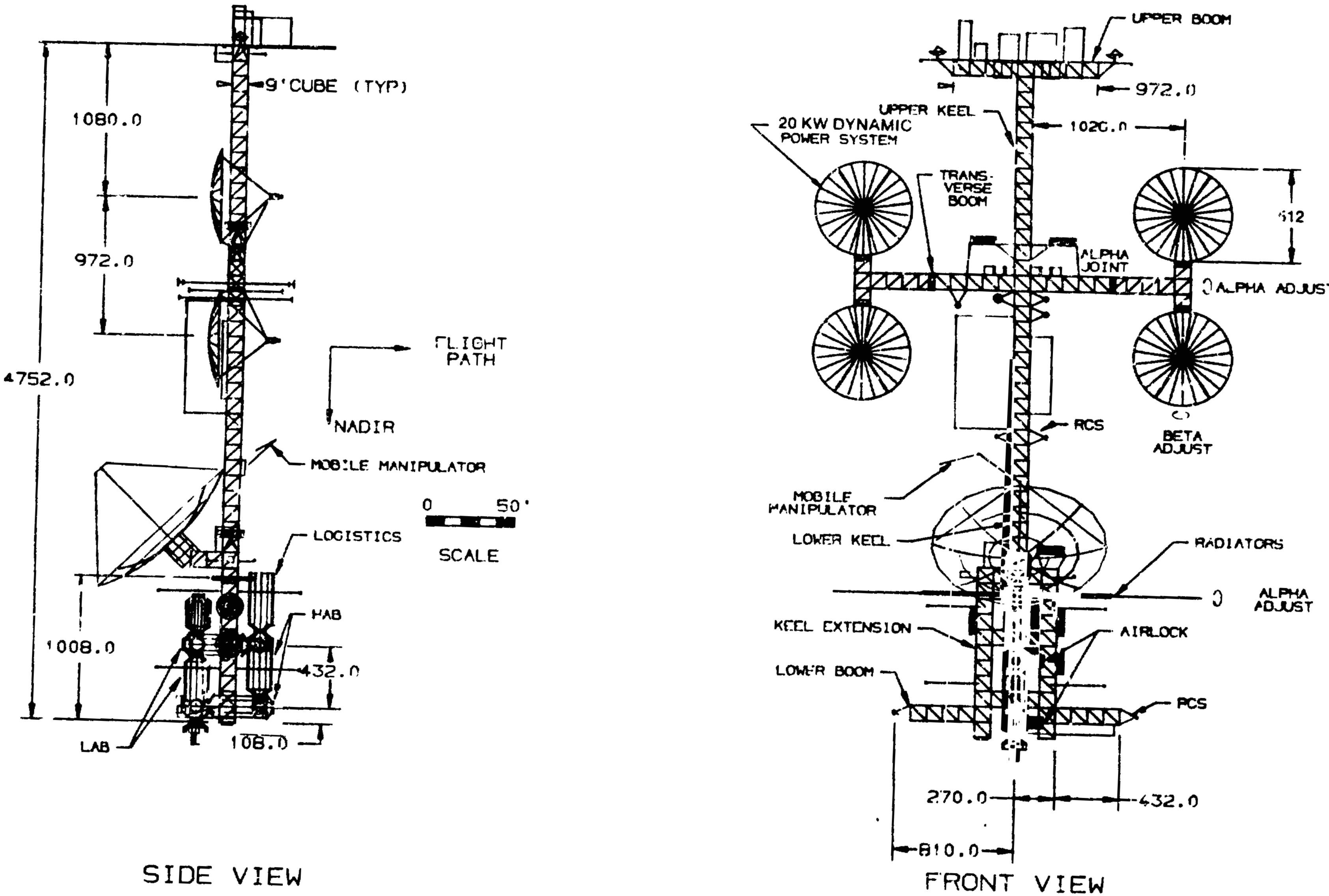


Figure 4.2.1-2 - IOC REFERENCE CONFIGURATION (SOLAR DYNAMIC OPTION)

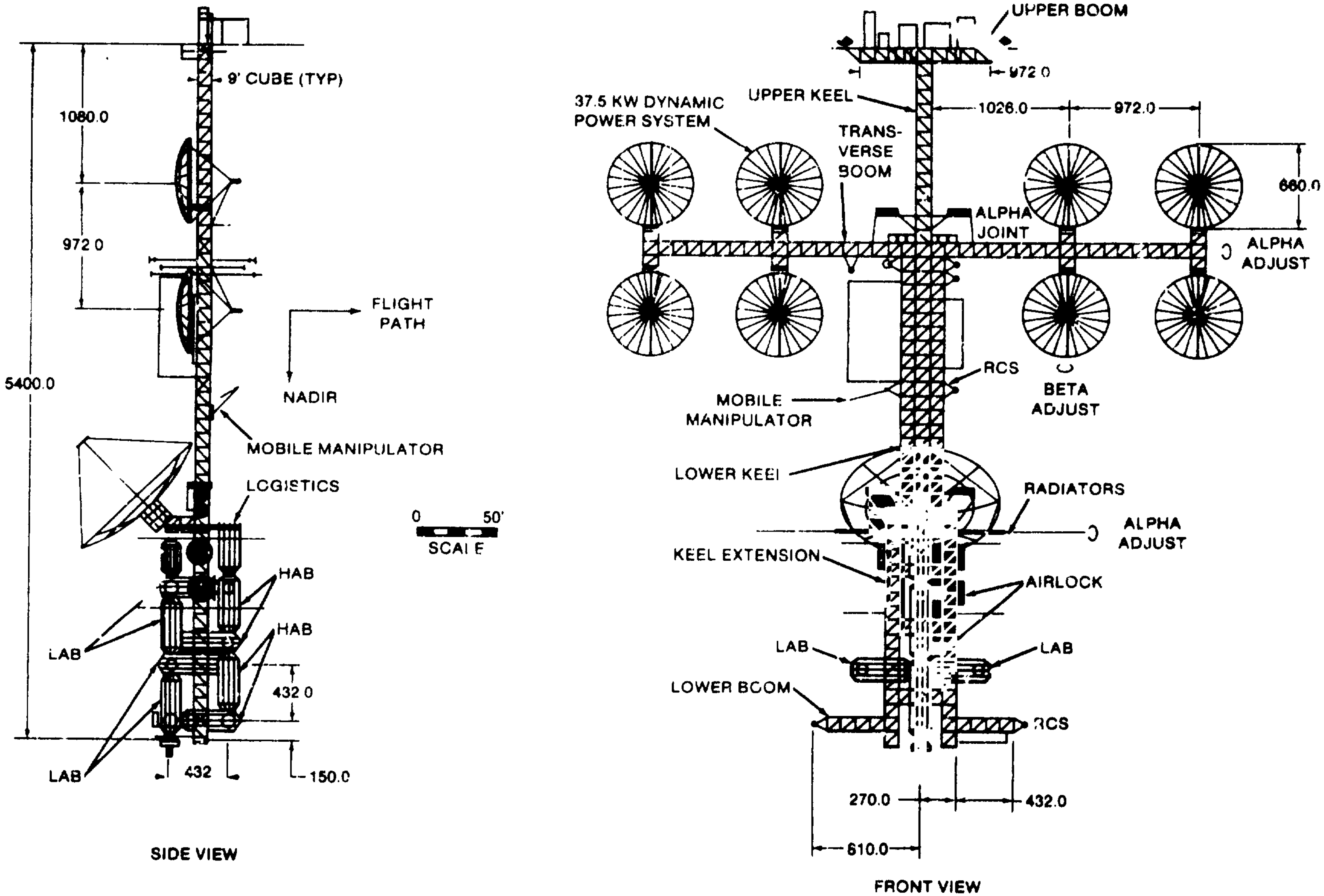
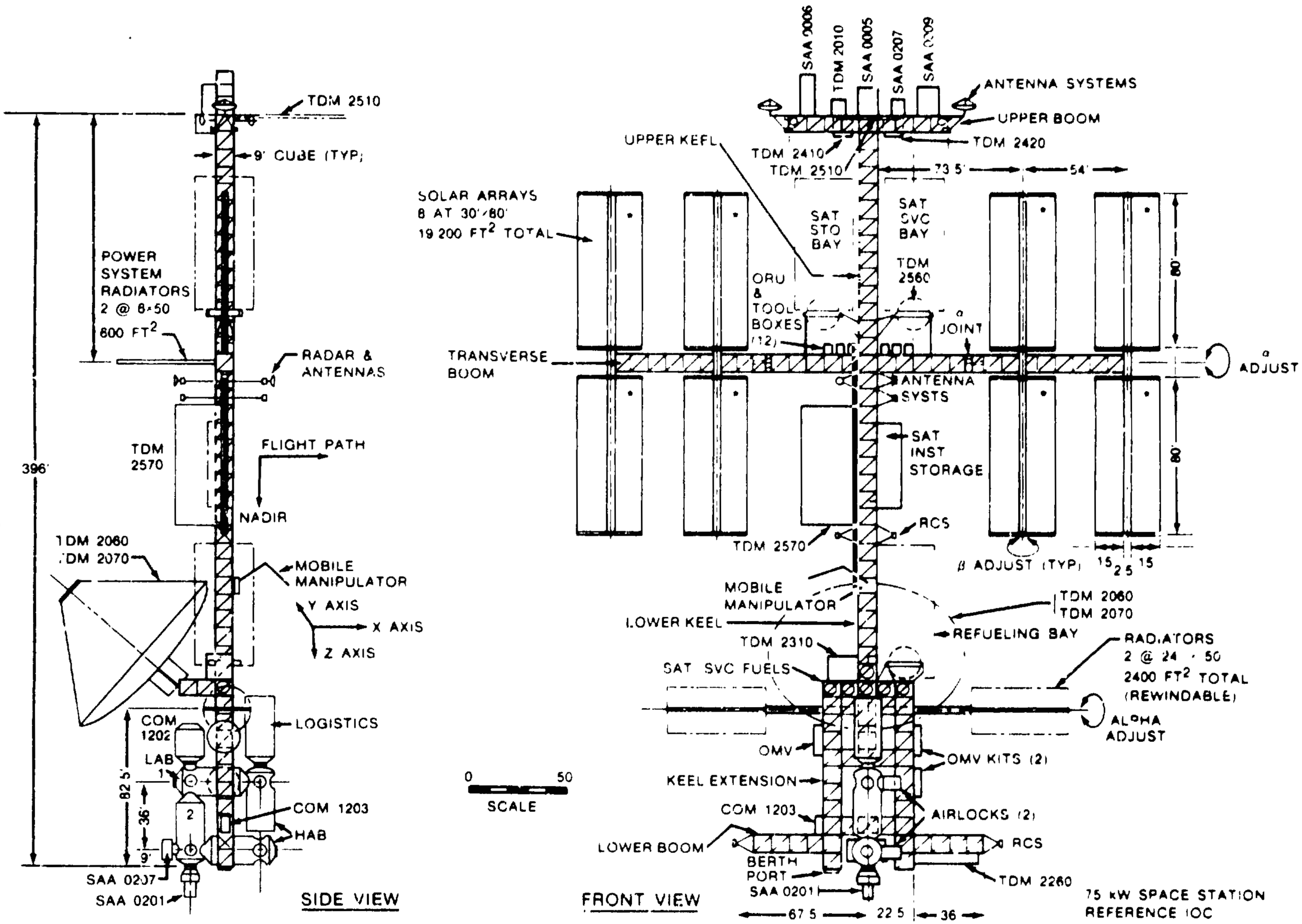


Figure 4.2.1-3 - GROWTH CONFIGURATION

ORIGINAL PAGE IS OF POOR QUALITY



*NOTE SOLAR DYNAMICS SYSTEM A PRIME OPTION (NOT SHOWN)

Figure 4.2.1-4 - USER ACCOMMODATIONS

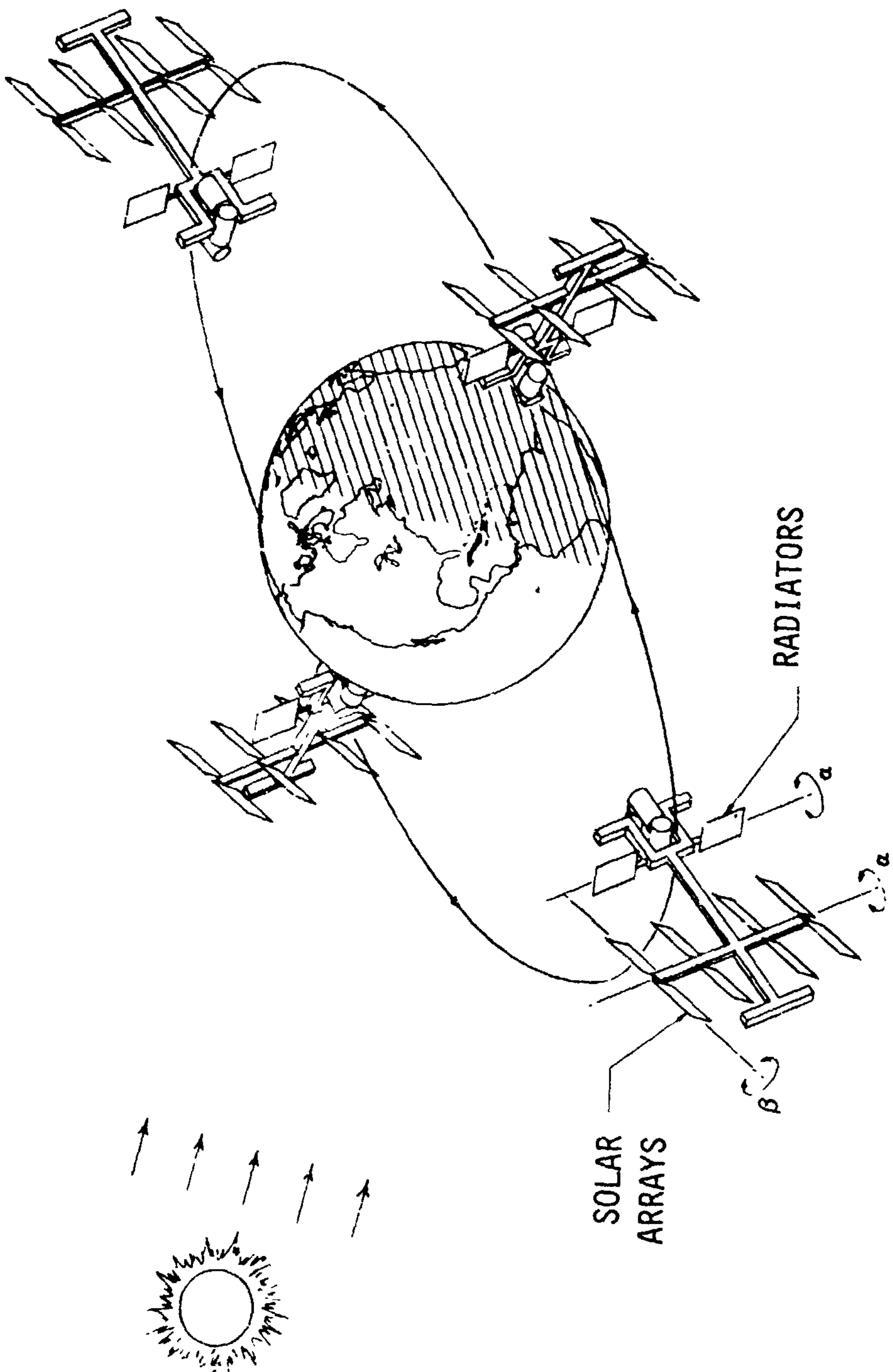


Figure 4.2.2-1 - NORMAL ORIENTATION.

variations in mass properties or atmospheric density are accommodated by small adjustments in the flight attitude of the Station.

The solar array is maintained normal to the Sun vector by a continuous rotation about the alpha gimbal at orbital rate and a slow adjustment about the beta gimbal to track the Sun's motion relative to the orbit plane. The radiators are also rotated at orbital rate, but are "rewound" during the dark part of the orbit to avoid continuously rotating fluid joints.

During the early stages of buildup, a "streamlined" orientation is flown (see Fig. 4.2.2-2). The transverse boom is in the orbit plane and the gimbals are locked with the array masts POP and the array wings at local horizontal. This attitude minimizes aerodynamic drag and enhances gravity gradient stability. The power system produces nearly half of full-rated power in this orientation, which is adequate until the Station is permanently manned.

The streamlined orientation can also be used in the unlikely event of total control system loss. Should this occur, gravity gradient torque will force the transverse boom into the orbit plane. The power system can maintain housekeeping requirements under these conditions, but a reduction in user operations would be necessary until the control problem can be corrected.

4.2.3 Elements

The principal elements of the reference Space Station are: Habitation Modules, Laboratory Modules, Logistics Modules, and Utilities Elements. Figure 4.2.3-1 provides a list of the pressurized module functions by element, and each is briefly discussed below. As mentioned previously, the modules have a high degree of commonality, both at the overall configuration level and at the subsystems level. The basic pressure shell and berthing port arrangement is identical in all Habitation and Laboratory Modules. The Logistics Module is the same as the others, except that the radial port segment has been replaced by an externally-mounted tankage segment, and the pressurized portion of the Logistics Module is therefore only two-thirds as long as that of the other modules. All the Habitation and Laboratory Modules have the same floor and ceiling arrangements except for the portion of the Habitation Module which contains the sleep quarters. Equipment racks have a high degree of commonality across the modules, as do the utilities.

Each Habitation and Laboratory Module has a 22-day supply of food and medical supplies for two crewmen. In the reference Space Station concept, most of the ECLSS equipment is located in the two Habitation Modules. A more distributed ECLSS system may have a greater weight and cost, but this may be offset by cost savings due to module commonality. The EVA airlocks are external modules berthed to the Habitation Modules.

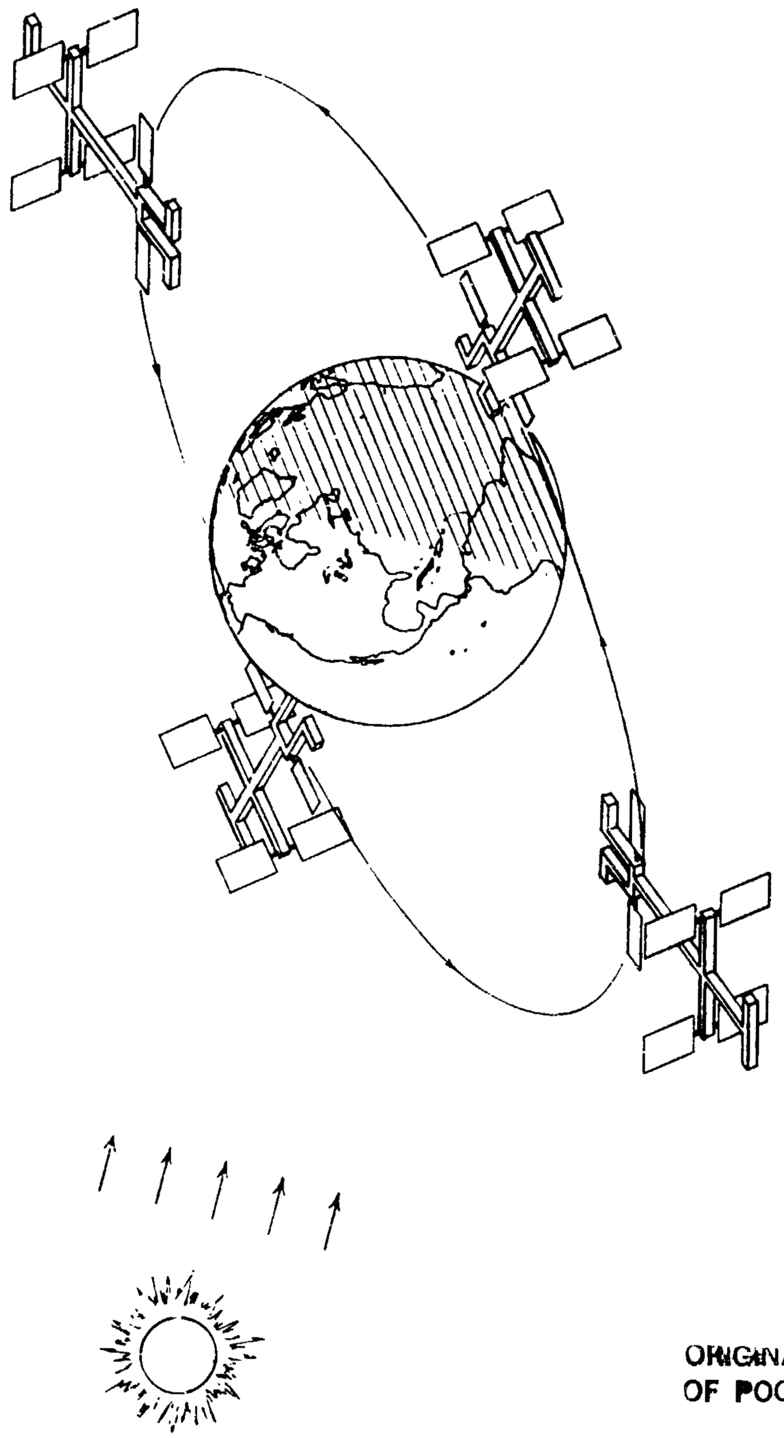


Figure 4.2.2-2 - STREAMLINED ORIENTATION.

PRESSURIZED MODULE FUNCTION ALLOCATION

LAB #1

- COMMON FUNCTIONS*

HAB #1

- COMMON/FUNCTIONS*
- SHOWER/URINAL
- WASTE MANAGEMENT
- PRIMARY ECLSS
- HEALTH MAINTENANCE
- WARDROOM
- GALLEY
- VEGETABLE CHAMBER
- COMMAND STATION

LOGISTICS

- COMMON FUNCTIONS*
  EXCEPT:
  - REDUCED UTILITIES
  - NO HANDWASHER
- EXPENDABLES
- SPARES
- DIRTY WATER STOWAGE
- TRASH STOWAGE

LAB #2

- COMMON FUNCTIONS*

HAB #2

- COMMON FUNCTIONS
- SHOWER/URINAL
- WASTE MANAGEMENT
- PRIMARY ECLSS
- SLEEP QUARTERS
- WASHER/DRYER
- MAINTENANCE/REPAIR STATION
- COMMAND STATION

***COMMON FUNCTIONS (ALL MODULES)**

- PRIMARY & SECONDARY STRUCTURE
- UTILITIES
- STORAGE
- SAFE HAVEN SUPPLIES
- PARTIAL CONTROL STATION
- HANDWASHER

FIGURE 4.2.3-1

Each of the Habitation and Laboratory Modules incorporates the safe haven features mentioned previously. The Orbiter is structurally compatible with any of the berthing ports on the modules for rescue, although the subsystems interfaces may be different than those between modules.

One of the overall guidelines used in the internal layout activity was to try to keep the Laboratory Modules as free as possible of habitability and housekeeping functions. Control stations, utilities, storage, and handwashers are included in the Laboratory Modules, and more equipment may have to be shifted from the Habitation Modules to the Laboratory Modules due to volume limitations in those modules.

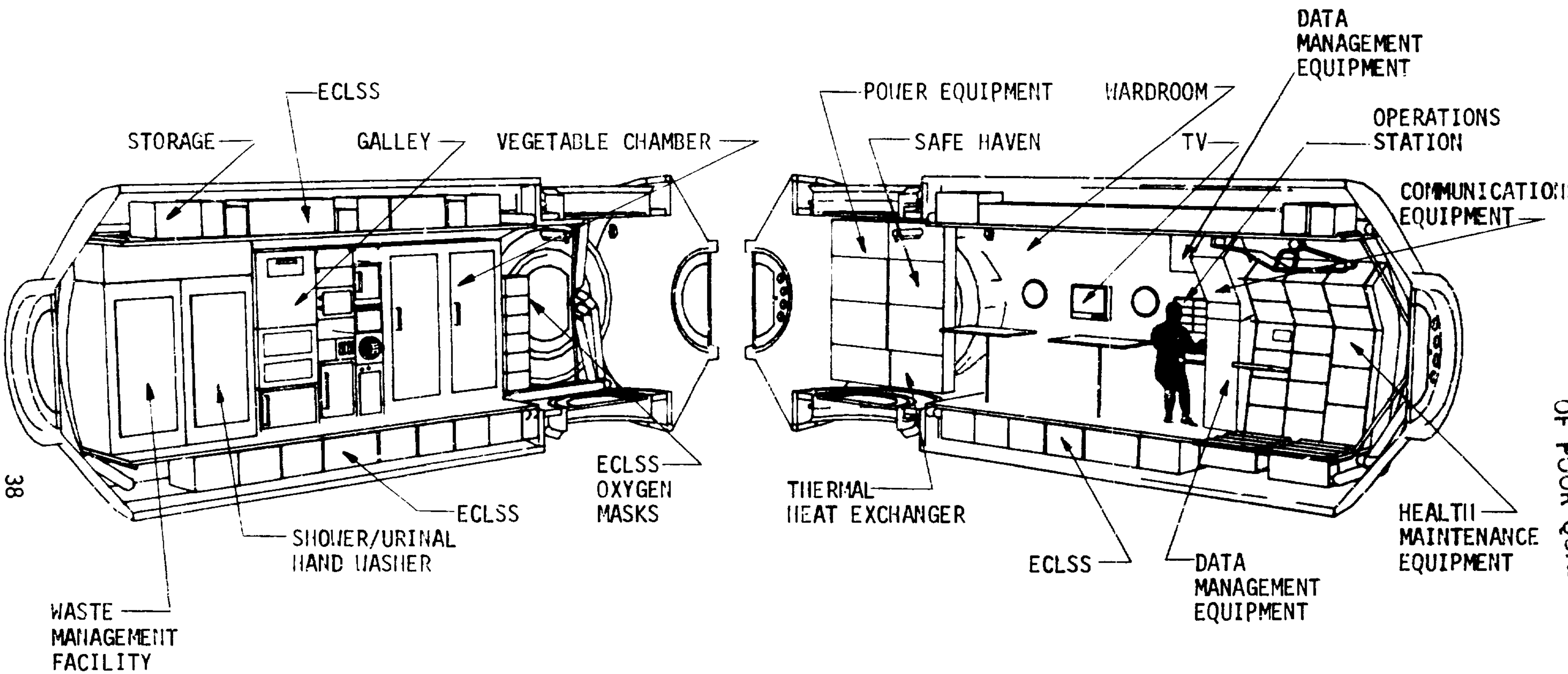
Another key guideline was to provide minimum disturbance in the sleep quarters area (ideally to have a "quiet" module and a living area module). This was not completely achieved due to other constraints. However, the wardroom and health maintenance facility (HMF), which are noisy areas, are located in a separate module from the sleep quarters.

Packaging schemes which allow more dense packaging have been assessed briefly and offer some improvement in volume constraints. There is some equipment within each module to which access is not required frequently, and this equipment would lend itself well to more dense packaging. Access to walls is needed in all concepts for detection, isolation, and repair of pressure shell leaks. This requires that the equipment be movable or that some other equivalent approach must be taken in the design.

A brief assessment was made of the on-orbit spares requirements. It is desired that these be kept to a minimum because of the premium placed on internal volume. It appears that the quantity of spares can be kept low, because of the modularity/redundancy design concepts used for critical subsystems.

The Logistics Module can be more densely packaged than the other modules, since it serves the "closet" function. Care must be taken not to exceed the STS down-weight limit (32,000 lb). This limit is expected to be increased by the timeframe of the Space Station. The nonfluid expendable resupply items and spares are packaged inside the Logistics Module, and the fluids are stored in externally-mounted tanks.

As presented in the reference configuration, the habitation volume is comprised of two habitation modules which contain all those facilities and equipment required for crew sustenance and Station operation. The following areas and categories of equipment are contained in the habitation volume: crew quarters, galley/wardroom, health maintenance facility/exercise area, personal hygiene area, workstations, stowage areas, waste management facilities, and laundry facility. These areas contain equipment, the facilities which "house" this equipment, and access volume which facilitates its use and maintenance. The internal configuration of these modules is shown in Figure 4.2.3-2 (Hab 1) and Figure 4.2.3-3 (Hab 2).



ORIGINAL ... OF POOR QUALITY

Figure 4.2.3-2 - HAB 1 MODULE

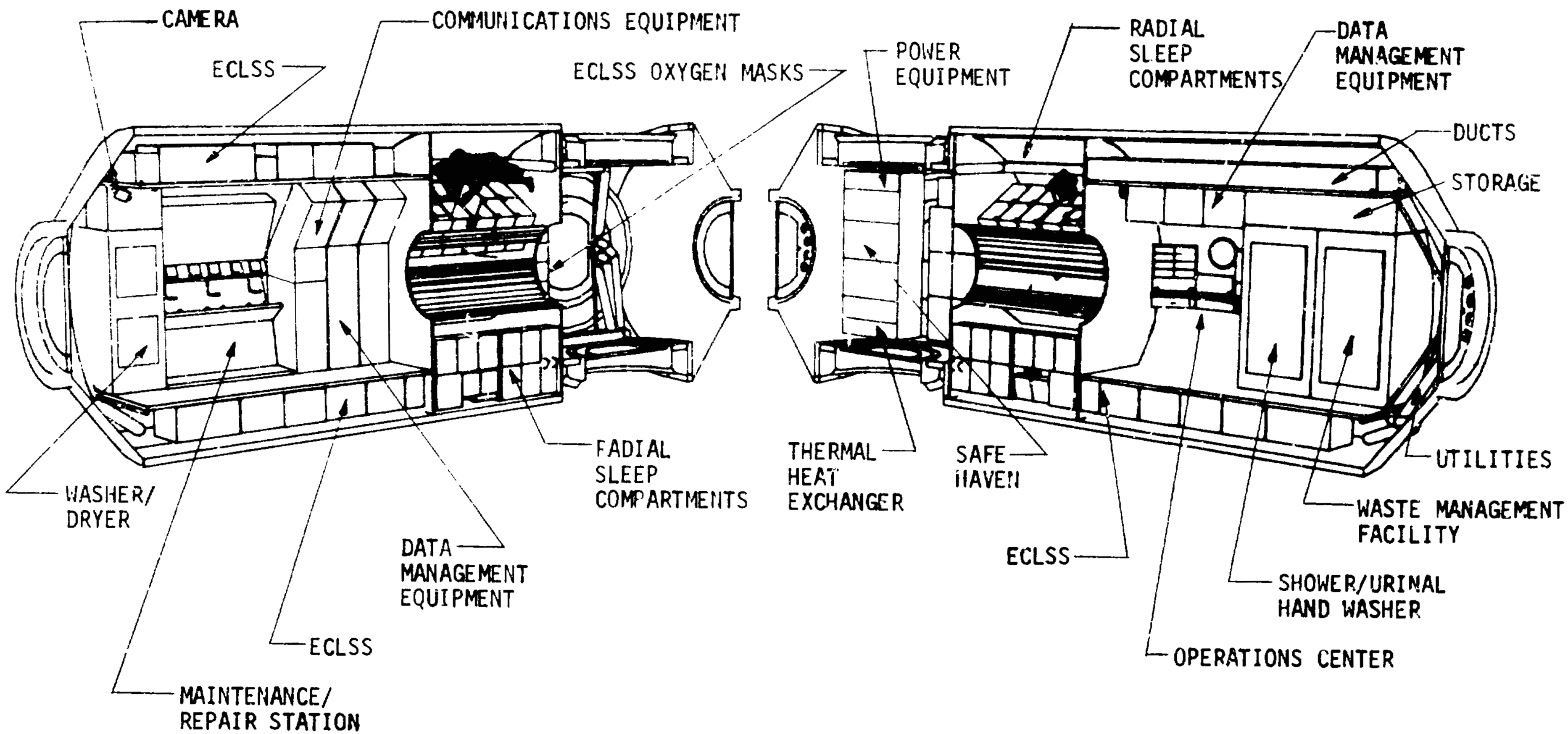


Figure 4.2.3-3 - HAB 2 MODULE

Crew accommodations in nonhabitation areas include the laboratory modules, the airlocks, the logistics module and safe haven provisions. Crew accommodations in the laboratory volumes aid in crew health, safety and well-being and they support the crew in customer services. As presented in the reference configuration, the internal layout of these modules is shown in Figure 4.2.3-4 (Life Sciences Lab) and Figure 4.2.3-5 (Materials Processing Lab).

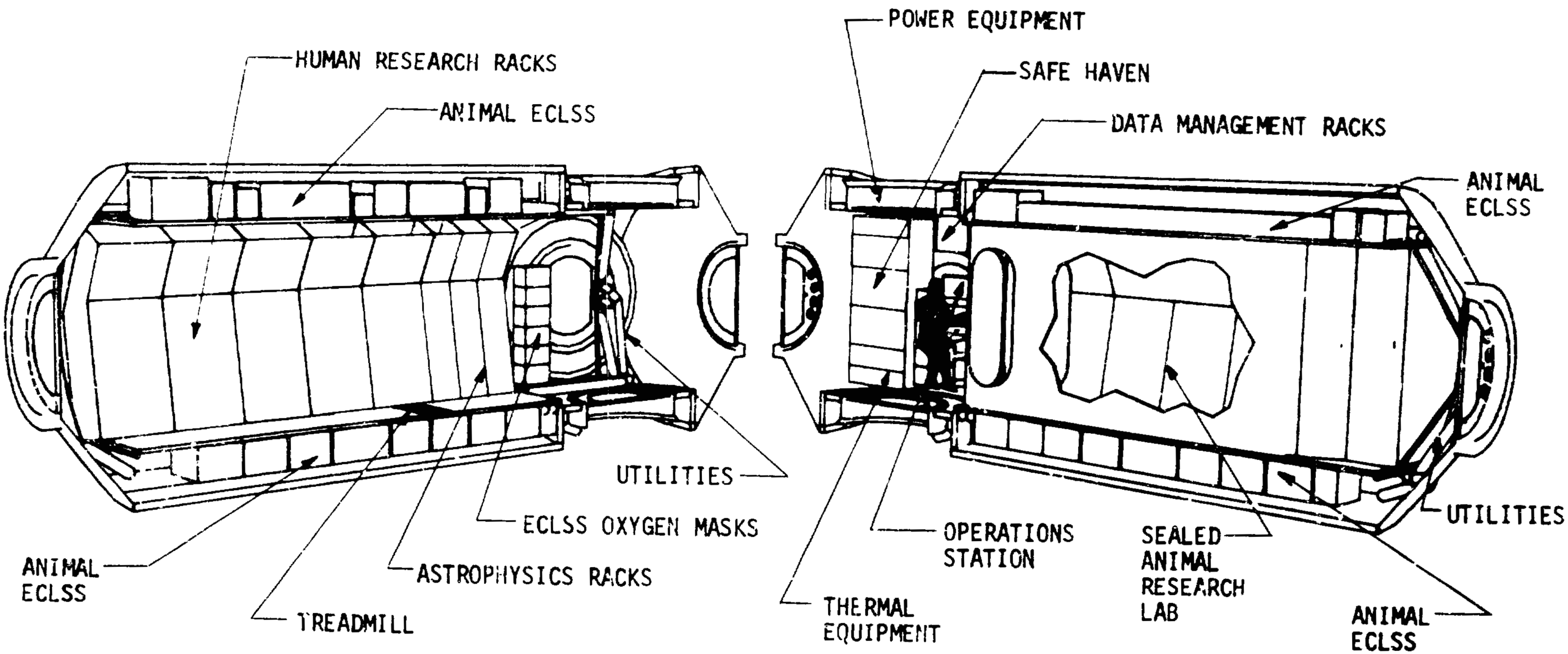


Figure 4.2.3-4 - LIFE SCIENCES LAB

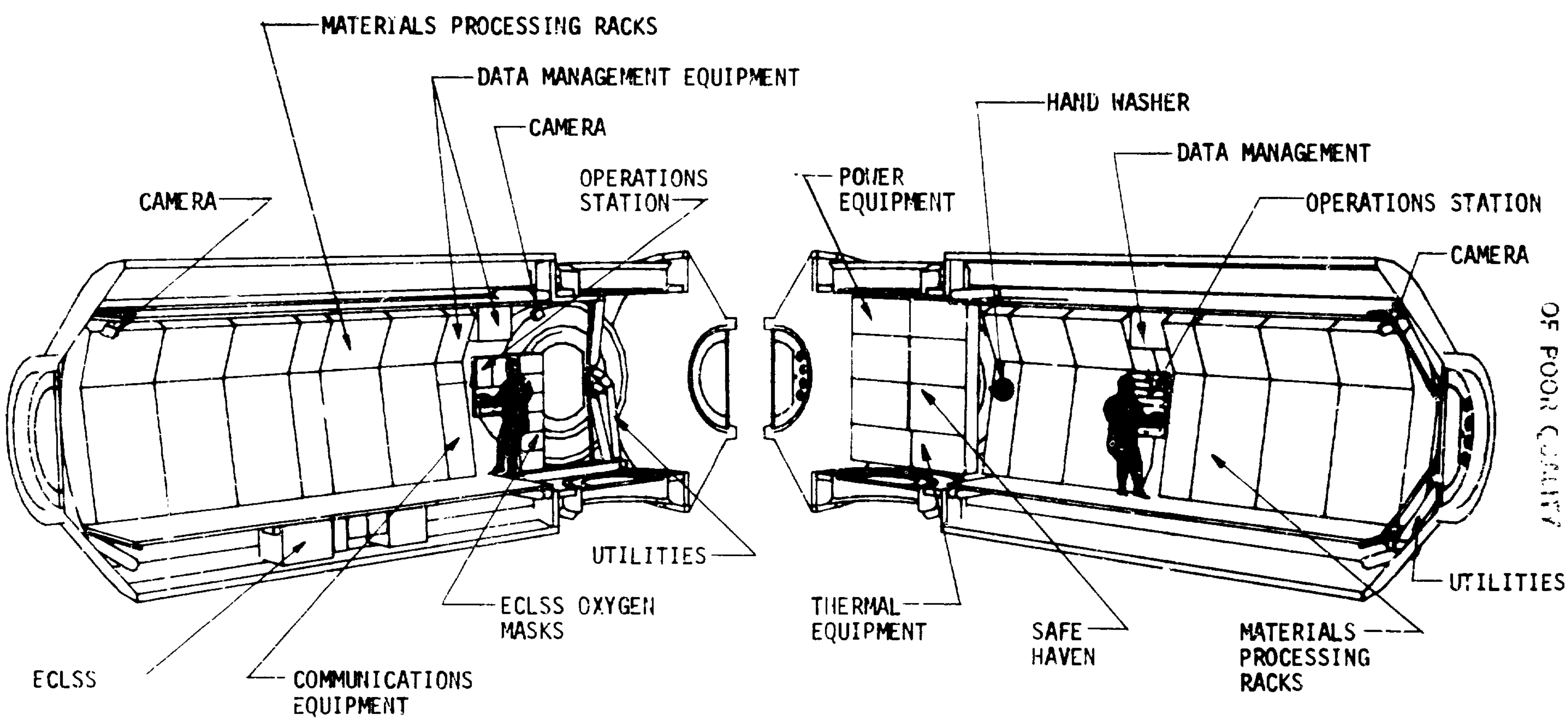


Figure 4.2.3-5 - MATERIALS PROCESSING LAB

4.2.4 Mass Properties

The weights and volumes for the ten subsystems were assembled for the reference configuration.

The ten subsystems were:

a. electrical power

b. guidance, navigation and control

c. communications and tracking

d. data handling

e. propulsion

f. environmental control and life support systems

g. active thermal control system

h. structures and mechanisms

i. crew systems

j. payloads

There were five iterations of weights for the reference configurations. All five iterations showed a steady increase of weights. The increases were due mainly to components that were overlooked in a previous iteration.

Because of the early decision to go with one module arrangement, the volume was only assembled for the one module arrangement. There were three iterations of volume for the common race-track-module arrangement of the Space Station.

The estimated weights and volume of the Space Station are summarized in the following tables. Although these estimates do not include an allowance for growth, growth can be expected to occur.

The subsystem weights for each element of the Space Systems are presented in Table 4.2.4-1. These are the total wet and dry weights in pounds. These weig' after each iteration were distributed to all subsystems for their use, well as for updating and correcting errors. The subsystem data were utilized for the cost estimation of the Space Station. The weight data for the cost estimates were double checked for accuracy.

Table 4.2.4-2 shows the Space Station volume in cubic feet for applicable subsystems that were in the pressurized modules, as well as components that were attached to the Space Station truss structures.

Table 4.2.4-3 presents the total weight for the Space Station by elements and subsystems.

Table 4.2.4-4 presents the total volume for the Space Station by elements and subsystems.

Table 4.2.4-5 presents the 90-day resupply by subsystem for the Space Station.

Table 4.2.4-6 presents the 10-year lifetime replacement cycle for the Space Station

Tables 4.2.4-7 through 4.2.4-15 list by module all the components of a subsystem that are common across the pressurized modules. The units are either number of components, number of men, or nunber of comparison quantity. Example: one unit might be 200 feet of wire.

TABLE 4.2.4-1 - SPACE STATION ELEMENT WEIGHTS - POUNDS

SUBSYSTEM	HM1	HM2	LM	LAB1	LAB2	TRUSS STRUCT	ATTCH COMPTS	PAYLOADS
ELECTRICAL POWER	643	462	208	505	505	7,867	12,554	-
GN&C		-	-	-	-	3,734	-	-
COMMUNICATIONS AND TRACKING	834	2,106	65	564	564	658	2,304	130
DATA HANDLING	1,005	765	120	805	605	-	1,250	-
PROPULSION	-	-	2,768	-	-	-	6,951	-
ECLSS	6,312	6,312	929	929	869	-	-	-
ATCS	474	474	211	1,215	640	2,408	5,396	-
STRUCTURES AND MECH.	12,964	12,964	7,310	12,964	12,964	15,316	3,224	-
CREW SYS.	15,710	11,080	9,125	3,883	3,883	-	-	-
PAYLOADS	-	-	13,148	18,630	32,275	-	-	184,365
TOTALS	37,942	34,163	33,884	39,495	55,305	29,983	31,679	184,495

HM1 = HAB, MOD, 1
HM2 = HAB, MOD, 2
LM = LOGISTICS MOD.
LAB1 = LAB, MOD, 1
LAB2 = LAB, MOD, 2

TABLE 4.2.4-2 - SPACE STATION ELEMENT VOLUME - FT^3

SUBSYSTEM	HM1	HM2	LM	LAB1	LAB2
ELECTRICAL POWER	12.5	8.7	3.6	8.6	8.6
GN&C	-	-	-	-	-
COMMUNICATIONS & TRACKING	35.9	78.2	0.5	23.1	23.1
DATA HANDLING	60.0	39.7	5.2	42.6	33.0
PROPULSION	-	-	57.0	-	-
ECLS	334.8	334.8	40.6	33.6	29.1
TCS	57.1	57.1	25.3	145.5	66.3
CREW SYSTEMS	2,608.0	5,897.0	1,040.0	651.0	639.0
PAYLOADS	-	-	3,748.0	2,800.0	2,800.0
TOTALS	3,108.3	6,415.5	4,920.2	3,704.4	3,599.1

TABLE 4.2.4-2 - SPACE STATION ELEMENT VOLUME - FT^3 (CONTINUED)

SUBSYSTEM	TRUSS STRUC	ATTCH COMPNTS	PAYLOADS
ELECTRICAL POWER	302.0	-	-
GN&C	161.0	-	-
COMMUNICATIONS	9.9	1055.3	2.5
DATA HANDLING	-	7.6	-
PROPULSION	-	130.0	-
ECLSS	-	-	-
ATCS	310.3	427.1	44.6
CREW SYSTEMS	-	-	-
PAYLOADS	-	-	(A) 8,898 (B) 124 X 10^6
TOTALS	783.2	1620.0	(A) 8945.1 (B) 124 X 10^6

TABLE 4.2.4-3 - SPACE STATION WEIGHT - POUNDS

ELEMENT	WEIGHT	SUBSYSTEM	WEIGHT
HM1	37,942	ELECTRICAL POWER	22,744
HM2	34,163	GN&C	3,734
LM	33,884	C&T	7,225
LAB1	39,495	DATA HANDLING	4,550
LAB2	55,305	PROPULSION	9,719
TRUSS STRUCTURE	29,983	ECLSS	15,351
		ATCS	10,818
		STRUCTURES & MECHANICS	77,706
ATTACHED COMPONENTS	31,679	CREW SYSTEMS	43,681
PAYLOADS	184,495	PAYLOADS	251,418
TOTALS	446,946	TOTALS	446,946

TABLE 4.2.4-4 - SPACE STATION TOTAL VOLUME - FT^3

ELEMENT	VOLUME	SUBSYSTEM	VOLUME
HM1	3,108.3	ELECTRICAL POWER	344.0
HM2	6,415.5	GN&C	161.0
LM	4,920.2	C&T	1,228.5
LAB1	3,704.4	DATA HANDLING	188.1
LAB2	3,599.1	PROPULSION	187.0
TRUSS STRUC	783.2	ECLSS	772.9
ATTACHED COMP.	1,620.0	ATCS	1,133.3
PAYLOADS	(A) 8,945.1 (B) $124X10^6$	CREW SYSTEMS	10,835.0
		PAYLOADS	(A) 18,246 (B) $124X10^6$

TABLE 4.2.4-5 - SPACE STATION 90-DAY RESUPPLY

SUBSYSTEM	WEIGHT (LBS)	VOLUME (FT^3)
ELECTRICAL POWER	0	0
GN&C	0	0
COMMUNICATIONS & TRACKING	0	0
DATA HANDLING	0	0
PROPULSION	2,402	57
ECLSS	3,341	240
ATCS	0	0
STRUCTURES & MECHANICS	0	0
CREW SYSTEMS	4,252	278
PAYLOADS	13,148	3,748
TOTALS	23,143	4,323

TABLE 4.2.4-6 - SPACE STATION TEN YEAR LIFETIME SPARE REPLACEMENT CYCLE

SUBSYSTEM	WEIGHT (LBS)	VOLUME (FT^3)	CYCLE (YRS)
ELECTRICAL POWER	15,848	302	5-10
GN&C	0	0	0
COMMUNICATIONS & TRACKING	0	0	0
DATA HANDLING	790	96	5-6
PROPULSION	146	35	5-10
ECLSS	976	72	5
ATCS	722	45	5
MECHANISMS	2,500	0	5-9
CREW SYSTEMS	0	0	0
PAYLOADS	3,305	942	1/4-5
TOTALS	24,207	1,492	

TABLE 4.2.4-7 - SPACE STATION SUBSYSTEM COMMON EQUIPMENT BY MODULE

ELECTRICAL POWER	HM1	HM2	LAB1	LAB2	LM	TRUSS STRUC
BDM						24
MBCU	4	4	4	4	2	8
BDC						24
UPC	60	48	50	50	24	80
0 GA. WIRE						1
8 GA. WIRE	3/10	3/10	3/10	3/10	1/10	1
12 GA. WIRE	1/3	2/10	1/3	1/3	1/12	1
CIF	28	12	8	8	2	
SSS	4	4	4	4	2	
CYCLO (60 X 1)	2	2			2	
CYCLO (50 X 1)	2	2			2	
CYCLO (60 X 3)			1	1		
CYCLO (50 X 3)			1	1		
CYCLO (40 X 3)			2	2		
APMS	1	1	1	1		
OPS	2					

TABLE 4.2.4-8 - SPACE STATION SUBSYSTEM COMMON EQUIPMENT BY MODULE

GN&C	HM1	HM2	LAB1	LAB2	LM	TRUSS STRUC
CMG ASSEMBLY						6
MAGNETIC BAR						6
STAR TRACKER SENSOR						2
HEXAD STRAPDOWN SENSOR						2
MAGNETIC TORQUERS						6
G&C PROCESSOR						3
NAV/TRAFFIC PROCESSOR		3				
RCS CONT. SYS.						3
INTERFACE DEVICES						18

TABLE 4.2.4-9 - SPACE STATION SUBSYSTEM COMMON EQUIPMENT BY MODULE

COMMUNICATIONS & TRACKING	HM1	HM2	LAB1	LAB2	LM
AUDIO TERMINAL		5	4	4	1
SPEAKER MAKE		2	1	1	1
MODULE CONTROLLER		1	1	1	1
EMERGENCY COMM. UNIT		2	4	1	2
WIRELESS COMM.		2	6	2	
WIRELESS COMM. W. UNIT		4	4	4	
LASER DOCKING		1		1	
FIBER OPTIC TRANS-CEIVER		8	8	8	
HDR 4-CHANNEL RCVR.			4		
FAR RANGE MA LDR MOD/DEMOD			2		
PROX. OPS MA LDR MOD/DEMOD			2		
TDRS KU-BAND MOD/DEMOD			2		
PROX. OPS MA LDR MOD/DEMOD			2		
TDRS KU-BAND MOD/DEMOD			2		
TM PROCESSOR & CONT			2		
P/L DATA INTERLEAVER			2		
FF SIG PROC. (LDR)			10		
FF SIG PROC (HDR)			8		
TDRS SIG PROC.(KU)			2		
DIGITAL TV PROC.			4		
REND. RADAR SIG PROC.			2		
VIDEO LINE DRIVER			35		
RFI INTERFACE			15		
EXTERNAL INTERFACE UNIT			1		
COMM & TRACK PROCESSOR COMPUTER			2		
DIGITAL STORAGE				1	1

TABLE 4.2.4-10 - SPACE STATION SUBSYSTEM COMMON EQUIPMENT BY MODULE

COMMUNICATIONS & TRACKING	HM1	HM2	LAB1	LAB2	LM	TRUSS STRUC
GPS LOW GAIN						2
S-BAND LOW GAIN						2
RADAR XPNDR LOW GAIN						2
MRMS S-BAND LOW GAIN						1
S-BAND LOW GAIN MRMS TV						1
S-BAND XPNDR (TDRS)	1					2
S-BAND XPNDR (ORBITER)						2
S-BAND PA/REAMP (T/O)						2
XCVR (MRMS)						1
RCVR (MRMS)						1
XMTR (ORBITER)						1
LOW RATE MUX	3	3	2	2		2
TDRS SIG PROC.	1					2
ENCRYPT/DECRYPT		22				2
ORBITER SIG PROC						2
GPS REC/PROC						2
RADAR XPNDR						2
CABLE	2					1
MED RATE MUX	1	1	1	1		
TV CAMERA	2	4	3	3		
TV MONITOR	8	8	2	2		
STEREO DISPLAY	1					
VIDEO RECORDER	1		2	2		
LASER PLAYER	1		2	2		
PAN/TILT UNIT	1	3	3	3		
AUDIO INTERFACE	1	1	1	1		
CONTROL STATION	1	1	1	1		
LARGE DISPLAY	1					
HIGH RATE MUX		2				

TABLE 4.2.4-11 - SPACE STATION SUBSYSTEM COMMON EQUIPMENT BY MODULE

DATA HANDLING	HM1	HM2	LAB1	LAB2	LM
DATA STORAGE	1	1	1		
TIME + FREQ.	1	1	1		
FAC/MGT CON	1	1	1		
FIXED WORK STATIONS	1	1	1	1	
PORTABLE WORK STATIONS	2	2	2	2	
CBLNG/OPT. NET	1	1	1	1	1
INTERIOR LIGHT	45	20	22	22	6
SUPPORT HARDWARE	1	1	1	1	
IDMS ID'S	6	6	6	3	
SUBSYSTEM ID'S	15	17	15	9	2
PAYLOAD ID'S			6	6	
IDMS SDP'S	6	6	6	3	

TABLE 4.2.4-12 - SPACE STATION SUBSYSTEM COMMON EQUIPMENT BY MODULE

ECLS	HM1	HM2	LAB1	LAB2	LM
ORBITER P/C CONTROL	1	1			
FIRE DETECTION/ SUPPRESSION	1	1	1	1	1
DUMP + RELIEF	1	1	1	1	1
FANS, DUCTING, MUFFLERS	1	1			
CO_2 REMOVAL SUB.	2	2			
CO_2 REDUCTION	2	2			
TRACE CONTAMINANT	1	1			
O_2 GENERATION	2	2			
HUMIDITY/TEMPERATURE CONTROL	1	1	1	1	1
MONITORING-ATMOSPHERIC REVITALIZATION	2	2	1	1	1
PLUMBING, VALVES, FITTINGS INSTALL SUPPORT	1	1			
POTABLE WATER TREATMENT	2	2			
HYGIENE WATER PROCESSING	2	2			
WASH WATER PROCESSING	2	2			
MONITORING-WATER MANAGEMENT	6	6			
URINALS	2	2			
O_2 GENERATION - EVA SERV.	1	1			
CO_2 REDUCTION = EVA SERV.	1	1			
WATER MGT EVA SERVICING	1	1			
AIRLOCK SUPPORT - ULLAGE SAVE PUMPS ACCUL SYS	1		1		
SAFE HAVEN - POS + MASKS	6	6	6	3	4
MODULE TANKAGE - POTABLE WATER	6	6			
MODULE TANKAGE - HYGIENE WATER	6	6			

TABLE 4.2.4-13 - SPACE STATION SUBSYSTEM COMMON EQUIPMENT BY MODULE

TCS	HM1	HM2	LAB1	LAB2	LM	TRUSS STRUC
COLD PLATES	19	6	32	31	1	
HEAT EXCHANGERS	18	6	32	31	1	31
PUMP/ACCUMULATORS	7	7	7	7	7	2
PLUMBING	5.5	4	9	8.7	2	1/4
VALVES/DISCONNECTS	4.5	4	7	6.5	2	1
CONTROLS/INSTRUM	1	1	1	1	1	1
BODY-MOUNT RADIATORS	1	1.4	1.4	1	1.4	
INTERFACE HX'S	1	1.03	1.03	1	1.03	11.2
THERMAL STORAGE						1

TABLE 4.2.4-14 - SPACE STATION SUBSYSTEM COMMON EQUIPMENT BY MODULE

ECLSS	HM1	HM2	LAB1	LAB2	LM
CABIN FANS, DEBRIS TRAP MUFFLEF			1	1	1
DUCTING, FITTINGS			1	1	1
PLUMBING, VALVES FITTINGS			1	1	1
MICROB. CHECK VALVES			1	1	1
PUMP PACKAGES			1	1	1
WATER ACCUMMULATION			1	1	1
N_2 CRYO	2	2			1
WASTE STORAGE	1	1			4
N_2 CRYO - MOD REPRESS TANK	2	2			1
O_2 CRYO - MOD REPRESS TANK	1	1			1
COLLECTOR/COMMODE	1	1			

TABLE 4.2.4-15 - SPACE STATION SUBSYSTEM COMMON EQUIPMENT BY MODULE

CREW SYSTEMS	HM1	HM2	LAB1	LAB2	LM
CLOTHING	3	3	2	2	6
PERSONAL HYGIENE KIT	1	1			6
EQPT CONTAINERS/RESTRAINTS	1	1			1
HOUSEKEEPING SUPPLIES	1				1
ANCILLORY PROVISIONS	1	1			
WARD ROOM	1				
GALLERY	1				1/10
WINDOWS	6	6	4	4	
FOOD CONTAINERS	3	2	2	2	8
HANDWASHER	1	1	1	1	1
CREW QUARTERS		6			
HMF, TOXICOLOGY, RADIATION NOISE	1				1/6
CAMERA EQPT, ETC.					2
STOWAGE CONTAINERS	158	106	56	56	224
SHOWER	1	1			
MAINTENANCE WORK STATION	1				1/3
DRY AND WET WIPES	2	2	2	2	
SLEEP RESTRAINTS	2	2	2	2	
WASTE TRASH STOWAGE	2	2	2	2	
WASHER/DRYER FACILITIES		1			
FREEZER					1
REFRIGERATOR					1
EMU SERVICE STATION	1	1			
MMU SERVICE STATION	1				
EMU	2	2			
HATCHES	4	4	4	4	1
MMU	2				
EVA TOOLS & MISC EVA EQPT	1				
AIRLOCK	1	1			
AURLOCK EQUIP & HYPERBARIC EQPT	1				

4.3 SPACE STATION REFERENCE CONFIGURATION EVALUATION SUMMARY

The reference configuration has been evaluated, with all the driving IOC requirements previously identified and the effects on each subsystem area, and met the requirements without significantly affecting the cost or threatening the feasibility of any of the major subsystems. The configuration provides excellent viewing opportunities for both Earth and celestial instruments, although wide angle gimbal systems are required for the latter type of instruments. The local vertical, local horizontal flight mode permits a straight-forward approach to communications and the use of established rendezvous and docking/berthing techniques. The buildup and assembly sequence appears feasible using the expected Space Shuttle Orbiter capability. The configuration is assessed as being free of structure/control system interaction, requires a moderate level of control authority, and tends to have a nontumbling attitude in an uncontrolled mode. The gimbaled solar arrays and radiators are efficient, since they operate continuously in the preferred solar orientation. Satisfactory locations have been identified for all the "straw man" external and internal IOC payloads and servicing/construction functions. The crew accommodations are judged to be adequate for a 6-person crew, and for the facilities required for control, maintenance, and EVA support activities.

In addition to the assessment against the IOC customer, operations, and systems requirements, the reference configuration has been evaluated against additional criteria. The more qualitative criteria deal with the programmatic aspect of cost and risk, the ease of accomplishment in systems engineering, area of integration, and growth, as well as transparency to evolving subsystems and payload technology.

A cost assessment of the IOC Reference Configuration has revealed no significant cost drivers associated with configuration unique features, nor have significant risk drivers been identified.

The user accommodations have been located on the configuration based on viewing, contamination, configuration, and functional requirements, and are considered to meet all requirements and to be near-optimum in meeting qualitative criteria.

The growth configuration has not been studied to the same depth as the IOC configuration. However, at least one means of accommodating growth in any direction it might take, as summarized in the requirements, has been identified. One scenario discussed for growth in the power area encompasses technological transparency in that solar dynamic primary power sources would be substituted for the photovoltaic arrays envisioned at IOC. This can be accomplished either by replacement of the entire power boom assembly, including mechanisms, or the initial structure and mechanical systems can be "scarred" to include the capability of accommodating the growth and system type change. The addition of modules for increased crew size and laboratory volume can be accommodated by the addition of structure to the keel extensions and the relocation or replacement of the lower boom assemblies with the propulsion and Earth viewing payload provisions. Growth in the construction and satellite and OMV servicing areas is accommodated by the erection or deployment of additional structure to the keel, and relocation of IOC servicing equipment, and perhaps TCS radiators, as required. Growth in the viewing instrument attachment area is accomplished by addition of structure and utilities to the upper and lower booms, with relocation of propulsion and communications equipment if necessary.

4.3.1 Payload Accommodations

A large, representative set of payloads for the IOC Space Station manned core has been identified. The purpose of this set of payloads is to bring out the wide range of customer requirements which will be imposed on the Station. It is not suggested that the entire representative payload set be present on the Station at IOC. Rather, the myriad of requirements brought forth by this representative set of payloads are used to guide the design of the Space Station. The representative payload set provides a measure for assessing the capability and versatility of the Space Station configuration to accommodate its customers.

4.3.1.1 Representative Payload Set

The representative payload set is not an officially accepted IOC payload set; it has no official status. The component payloads were chosen on the basis of the requirements which each imposed on Space Station design and the proposed initial time period (1992) of their operation. Descriptions of the individual payloads are provided within the Mission Requirements Working Group (MRWG) Langley Data Base Documents and other MRWG reports.

The representative set of payloads may be divided into the three broad categories of (1) science and applications (SAA), (2) commercial (COM), and (3) technology development missions (TDM). These payloads and their MRWG identification codes are presented in Table 4.3.1-1.

Servicing and refueling of free-flying spacecraft and platforms will also be accommodated by the reference Space Station. A representative set of free-flying payloads which require servicing and/or refueling has been developed; this set of payloads is presented in Section 4.3.1.4.

4.3.1.2 Attached Payloads

Seventeen payloads in the representative payload set are to be attached to the Space Station manned core. These payloads are identified in Table 4.3.1-2. Requirements imposed by these payloads include:

a. mass distribution and payload location

b. simultaneity of viewing in inertial, solar, Earth, and anti-Earth directions

c. precise pointing fields-of-view, accuracy, and stability

d. contamination-free environment

e. 10^{-5} microgravity

TABLE 4.3.1-1 - REFERENCE IOC PAYLOAD SET

IDENTIFICATION CODE	PAYLOAD NAME
SAA0005	Transition Radiation & Ion Calorimeter (TRIC)
SAA0006	STARLAB
SAA0009	Pinhole Occulter Facility
SAA0201	LIDAR Facility
SAA0207	Space Plasma Payload (SPP)
SAA0307	Life Science Laboratory
SAA0401	Microgravity Research Laboratory
COM1201	MPS Lab #1 (Materials Processing)
COM1202	EOS Production Unit
COM1203	ECG Production Unit
TDM2010	Materials Performance
TDM2020	Materials Processing Technology
TDM2060	Deployment/Assembly/Construction Technology
TDM2070	Structural Dynamics Technology
TDM2260	Earth Observation Instrument Technology
TDM2310	Fluid Management Technology
TDM2410	Attitude Control Technology
TDM2420	Figure Control Technology
TDM2510	Environmental Effects
TDM2520	Habitation Technology
TDM2530	Medical Technology
TDM2560	Satellite Servicing Technology
TDM2570	OTV Servicing Technology

f. construction of large structures

g. servicing

h. attachment of tethers

i. control consoles within pressurized module

4.3.1.2.1 Mass distribution and payload location

The 17 attached payloads represent a significant mass component of the Station. The weight of each attached payload is presented in Table 4.3.1-2. The sum of the weights is approximately 110,000 pounds.

Placement of these attached payloads will influence the mass distribution characteristics of the Space Station. The payloads are distributed on the reference configuration such that the inertial, solar, and anti-Earth pointing instruments are placed on the upper boom. Earth viewing instruments are placed on the lower boom. The technology payloads which have no pointing requirements are situated along the keel. The commercial production units are attached to the Microgravity and Materials Processing Facility module.

4.3.1.2.2 Simultaneity of viewing

Attached payloads listed in Table 4.3.1-3 have specific pointing requirements; attached payloads not listed in this Table have no pointing requirements.

The requirement for different payloads to simultaneously point in different directions is accommodated onboard the reference configuration by having the various instruments, or possibly groups of instruments, independently gimbaled.

In general, instruments requiring similar pointing directions are grouped together. Instruments which require inertial, solar, or anti-Earth viewing directions are situated on the upper boom. (The option, however, is held open to place some solar pointing instruments on the transverse boom.) Instruments requiring pointing in the Earth direction are placed on the lower boom.

4.3.1.2.3 Pointing fields-of-view, accuracy, and stability

Payload requirements for pointing fields-of-view, accuracy, and stability are identified in Table 4.3.1-3.

Placement of the pointing instruments on either the upper boom (for "outward"-looking instruments) or the lower boom (for "downward"-looking instruments) will maximize the possible field of view by reducing that area potentially subtended by the Space Station structure.

TABLE 4.3.1-2 - ATTACHED PAYLOAD WEIGHTS

CODE	PAYLOAD	WEIGHT
SAA0005	Transition Radiation & Ion Calorimeter (TRIC)	12,675 lbs.
SAA0006	STARLAB	7,055
SAA0009	Pinhole Occulter Facility	7,940
SAA0201	LIDAR Facility	4,190
SAA0207	Space Plasma Payload (SPP)	7,055
COM1202	EOS Production Unit	9,920
COM1203	ECG Production Unit	11,025
TDM2010	Materials Performance	1,545
TDM2060	Deployment/Assembly/Construction Technology	8,820
TDM2070	Structural Dynamics (mass requirements are covered by TDM2060)	
TDM2260	Earth Observation Instrument	660
TDM2310	Fluid Management	5,510
TDM2410	Attitude Control	1,100
TDM2420	Figure Control	1,100
TDM2510	Environmental Effects	6,175
TDM2560	Satellite Servicing	7,055
TDM2570	OTV Servicing	17,640

TABLE 4.3.1-3 - POINTING REQUIREMENTS OF IOC PAYLOAD SET

POINTING DIRECTION	IDENTIFICATION CODE	PAYLOAD NAME	FIELD OF VIEW (DEGREES)	POINTING ACCURACY (ARC SEC)	POINTING STABILITY (ARC SEC)
EARTH	SAA0201	LIDAR Facility	60	3600	3600
EARTH	SAA0207	Space Plasma Payload (SPP)	360	3600	3600
EARTH	TDM2260	Earth Observation Instrument Tech.	-	360	-
EARTH	TDM2510	Environmental Effects	20	7200	-
ANTI-EARTH	SAA0005	Transition Radiation and Ion Calorimeter (TRIC)	120	36000	36000
SOLAR	SAA0009	Pinhole Occulter Facility	3	10	1
SOLAR	SAA0207	Space Plasma Payload (SPP)	360	3600	3600
SOLAR	TDM2010	Materials Performance	-	7200	-
SOLAR	TDM2510	Environmental Effects	20	7200	-
INERTIAL	SAA0006	STARLAB	180	2	0.02
INERTIAL	TDM2410	Attitude Control Technology	-	-	-
INERTIAL	TDM2420	Figure Control Technology	-	-	-

Payloads listed under more than one "POINTING DIRECTION" category have component parts with different pointing direction requirements.

Pointing accuracy and stability requirements for payloads SAA0006 (STARLAB), SAA0009 (Pinhole Occulter Facility), and TDM2260 (Earth Observation Instrument Technology) cannot be met without some type of pointing augmentation system for the individual instruments. The nature of this pointing system has not yet been determined.

4.3.1.2.4 Contamination-free environment

Contamination is a serious concern for sensitive-viewing instruments. Placement of those viewing instruments on the upper boom and the lower boom remove them from the vicinity of the satellite refueling area and the Shuttle and OMV berthing areas on the lower keel.

4.3.1.2.5 10^{-5} microgravity

Payloads COM1202 (EOS Production Unit) and COM1203 (ECG Production Unit) have the requirement for a 10^{-5} g environment. These two payloads are customer-supplied commercial production facilities.

(Payload COM1202 requires attachment to the manned core modules through a pressurized port to permit IVA servicing. Payload COM1203 is unpressurized, serviced by EVA, and therefore does not need to be attached at a pressurized port.)

Since the Microgravity and Materials Processing Facility (MMPF) module also has a requirement for 10^{-5} g, payload COM1202 is attached to that MMPF module.

4.3.1.2.6 Construction of large structures

Payload TDM2060 (Deployment/Assembly/Construction Technology) requires the construction of a 30-meter-diameter dish antenna. This antenna is positioned on the reference configuration lower keel for reasons of station mass properties and ease of EVA access.

4.3.1.2.7 Servicing of attached payloads

Servicing of the attached payloads will be by EVA operation. (The exception to this EVA servicing is payload COM1202, which is pressurized and will have IVA servicing.) Attached payloads are situated at various locations on the structure: the upper boom, the lower boom, the lower keel, the modules, and possibly also the power boom. These attached payloads will require regular servicing. In addition, unanticipated repairs or trouble-shooting may be necessary. Mobile RMS (MRMS) and EVA operations must be designed to work synergistically to allow servicing of these payloads.

4.3.1.2.8 Attachment of tethers

Payload SAA0207 (Space Plasma Payload) will require both payload tethering "up" (up 30 km in length from Space Station along a conducting tether) and tethering "down" (down 30 to 300 km from Space Station along a nonconducting tether). Tethering "up" can be accomplished from the upper boom; tethering "down" can be done from the lower boom.

Payload SAA0207 further requires the deployment (and retraction) of a 300-meter dipole antenna. This long dipole antenna needs to have the capability to be rotated plus-or-minus 60° from nadir in order to fly either aligned parallel or perpendicular to the Earth magnetic field lines.

4.3.1.2.9 Control consoles within pressurized module

Many of the externally attached payloads will require control consoles within one of the pressurized modules from which those payloads may be operated routinely.

4.3.1.3 Pressurized Laboratory Modules

Six of the payloads identified in the representative set of payloads (Table 4.3.1-1) are conducted within pressurized laboratory modules. Specifically, these payloads are: SAA0307, Life Science Laboratory; SAA0401, Microgravity Research Laboratory; COM1201, MPS Laboratory #1; TDM2020, Materials Processing Technology; TDM2520, Habitation Technology; and TDM2530, Medical Technology.

At IOC, there are two habitable Space Station modules to conduct laboratory work and to provide operational support. For the purposes of this report, one module is identified to be a Microgravity and Materials Processing Facility module; it will require the maintenance of a microgravity level less than or equal to 10^{-5} g. The other is identified to be a Life Sciences Laboratory module; it is designed to conduct research into the problems and henomenological effects of long-term exposure of humans, animals, and plants to near weightlessness. The Life Sciences Laboratory module is required to maintain a microgravity level of at most 10^{-4} g. Additionally, the Life Sciences Laboratory will have equipment and work space in it for monitoring and controlling the various payloads attached externally to the Station.

Both laboratory modules are intended to have sufficient versatility to support other science and application missions. To that end, both laboratories will be capable of having equipment reconfigured on-orbit.

Science airlocks and optical-quality windows will be provided to facilitate viewing by internally-mounted payloads. Parameters for viewing direction will constrain module orientations and internal arrangements of equipment.

4.3.1.3.1 Microgravity and Materials Processing Facility module

The Microgravity and Materials Processing Facility (MMPF) module will support the development of unique materials and processes. An acceleration environment of 10^{-5} g, or less, is required for periods of days to months. The MMPF module will support the advancement of the knowledge base; the development of processes and process controls; the scale-up to pilot plant operations; and the operation of pre-production and commercial production facilities.

The module is required to be a sectionalized facility composed of equipment stations that deliver the utilities of power, heat rejection, vacuum, control gases, data handling, communications, and command. The principal equipment within the module will include low and high temperature furnaces, crystal growth apparatus, containerless (levitation) furnaces, biological materials separation facilities, a scientific airlock station with equipment attachments exterior to the module, a sample preparation and characterization work station, and instrument and materials stowage. Attachment directly to the interior wall of the module will be necessary for some equipment.

The generic distribution of utility outlets and attachment provisions will permit self-contained, integral equipment to be quickly "plugged" into an equipment station by the crew. A high degree of equipment changeout and reconfiguration will be essential to conduct required research, development, and engineering. Specifically, utility control to permit change of the utility distribution to different equipment as a function of tasks is needed. This utility control function will accommodate utility distribution to new processing equipment that is exchanged on-orbit with obsolete equipment.

A schematic drawing of the Microgravity and Materials Processing Facility module is shown in Figure 4.3.1-1.

4.3.1.3.2 Life sciences laboratory module

The available volume within the Life Sciences Laboratory will be divided between a human research facility, an animal and plant research facility, and operations/control facilities for payloads externally attached to the Station. Few firm requirements exist at present for the volumes of these individual facilities within the Life Sciences Laboratory. For the purposes of sizing the utility requirements of this module within the context of the overall Station configuration, some assumptions of equipment mass and volume have been made. These assumptions are intended only to support the viability of a multidiscipline Life Sciences Laboratory and do not necessarily represent a consensus within the life science community.

A schematic drawing of the Life Sciences Laboratory module is shown in Figure 4.3.1-2.

4.3.1.3.2.1 Human research facility

The human research facility will conduct investigations of the physiological changes which human beings undergo due to exposure to

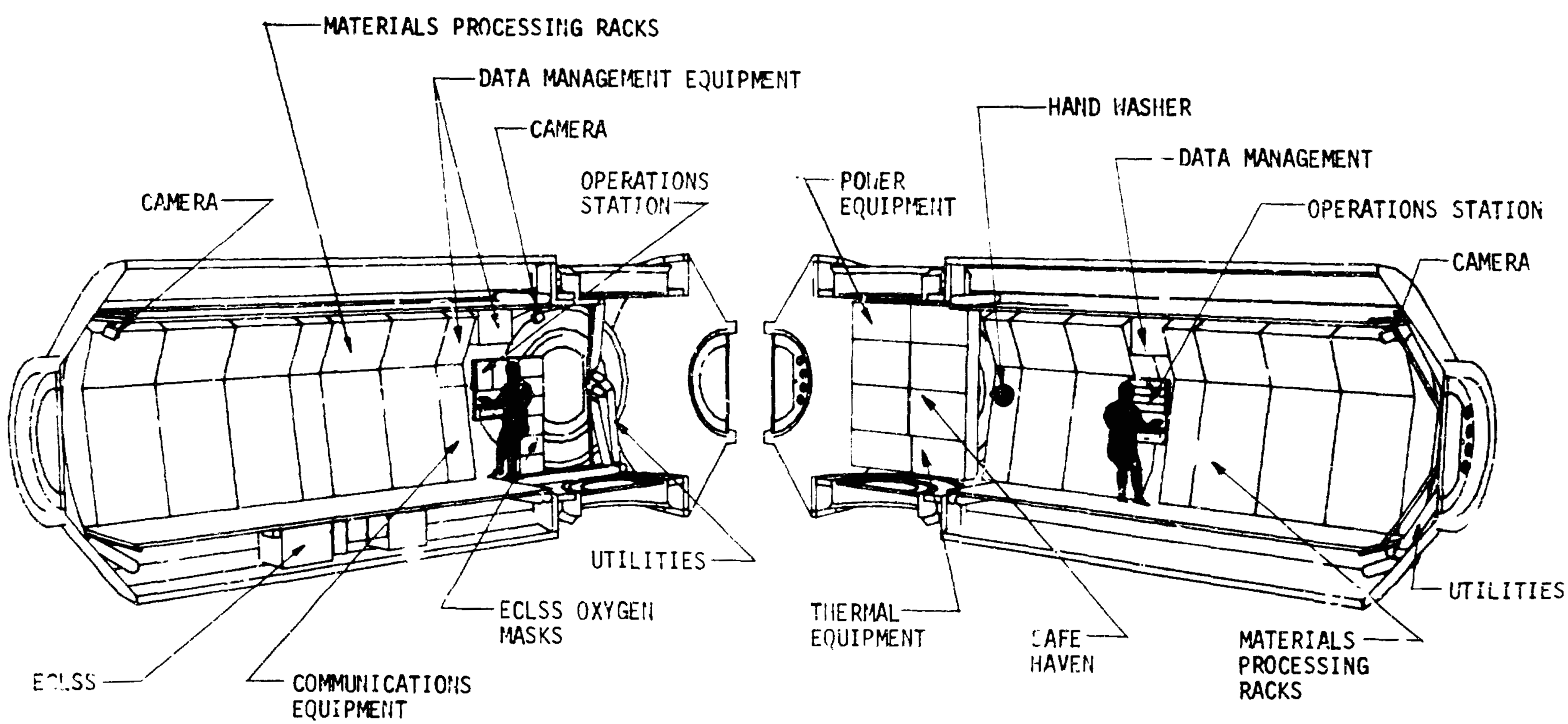


Figure 4.3.1-1 - MICROGRAVITY AND MATERIALS PROCESSING FACILITY

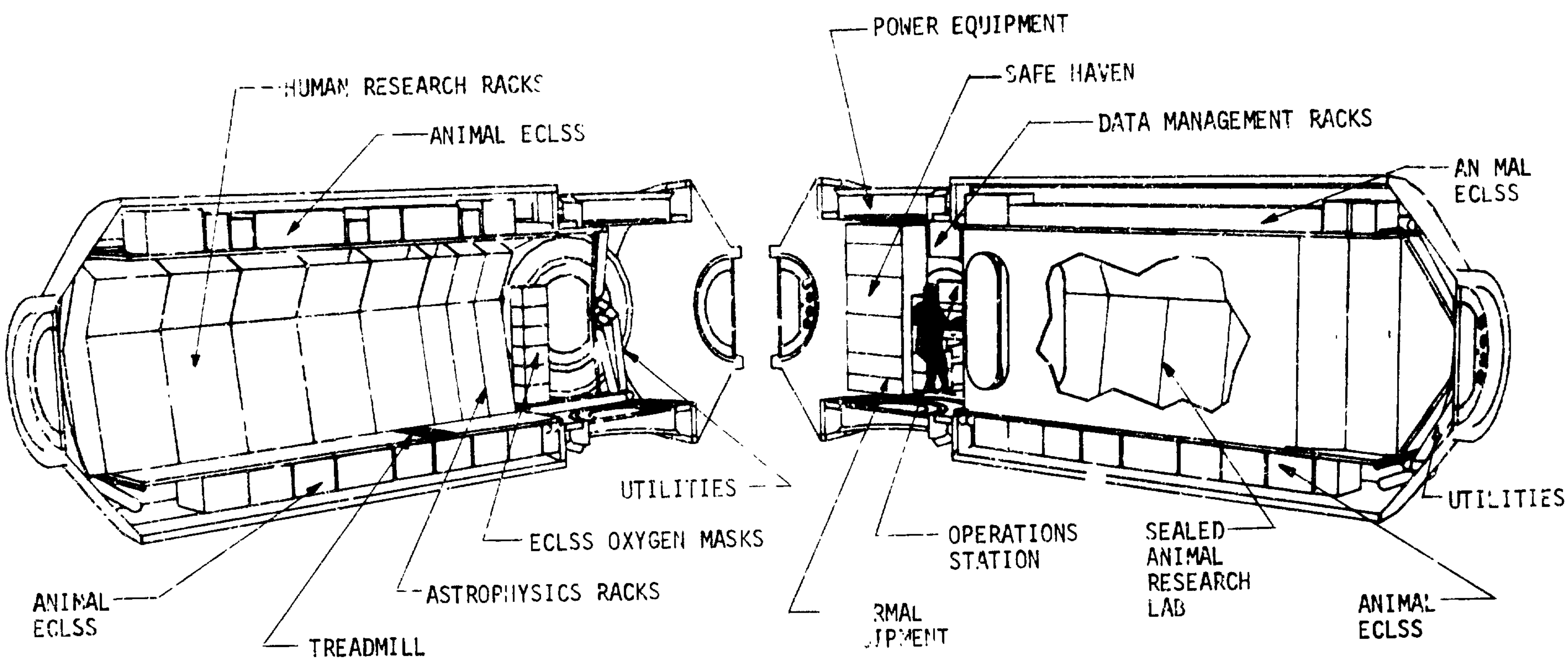

HUMAN RESEARCH RACKS
ANIMAL ECLSS
ANIMAL ECLSS
TREADMILL
ASTROPHYSICS RACKS
ECLSS OXYGEN MASKS
UTILITIES
POWER EQUIPMENT
SAFE HAVEN
DATA MANAGEMENT RACKS
ANIMAL ECLSS
OPERATIONS STATION
SEALED ANIMAL RESEARCH LAB
UTILITIES
ANIMAL ECLSS

Figure 4.3.1-2

microgravity for extended periods of time. (Much of the early research will concern NASA's need to develop countermeasures to protect the health and safety of the Space Station crew.)

Equipment for the human research facility is assumed to require eight 19-inch racks and two 38-inch racks. The contents of these racks are core laboratory equipment, discipline specific equipment, data management equipment, storage space, and supporting hardware. A treadmill, ergometer, and body mass measurement device will be situated in the center aisle. In addition, there will be floor and ceiling storage space and mounting provisions for experiment unique hardware.

Total volume for the equipment is approximately 280 ft^3; total equipment mass is nearly 3175 lbm (1440 kg).

4.3.1.3.2.2 Animal/plant research facility

The animal/plant research facility will conduct long-term studies on microgravity-induced physiological changes in animals; on the mechanisms of gravity sensing and gravity responses in animals and plants; on the effects of microgravity on fundamental biological systems, including reproduction; and on biological and chemical systems to close the ecological cycle and convert waste into food.

Volume for equipment within the animal/plant research facility is approximately 460 ft^3. The equipment mass is nearly 4400 lbm (2000 kg). (Although this equipment volume is greater than that for the human research facility, the overall volume of the animal/plant research facility is approximately the same as that of the human research facility.) Required racks will contain core laboratory equipment, discipline specific equipment, general purpose workbench, advanced rodent holding facilities, refrigerator/freezer, plant growth units, closed ecology life support equipment, and data management equipment. Ceiling storage will also be provided.

The animal/plant research facility is required to be enclosed in a separate compartment within the module. This compartment will have its own independent ECLSS system (provided by the user). The ECLSS will prevent cross-contamination between the rest of the Space Station and the animal and plant specimens. The ECLSS will exclude particles larger than 0.5 microns on the air inlet and particles larger than 0.3 microns on the air outlet. Charcoal filters will eliminate odor transfer.

The animal and plant holding facilities should be as isolated as practical (e.g., by movable barriers) from the laboratory work area. This will provide better control of the light/dark cycle for specimens, more accurate temperature and humidity control, and will isolate the animals from disturbances caused by crew activity.

Again, it must be stated that the particular choices of volumes and equipment for the human research facility and the animal/plant research facility were selected somewhat arbitrarily for reference configuration purposes. These volumes and equipment choices do not represent a concensus decision of the life sciences community.

4.3.1.3.2.3 Operations/Control Facilities

Many of the externally attached payloads and servicing operations will require consoles within the pressurized modules from which those payloads and operations may be monitored and controlled. Equipment which fills three 16-inch racks is provided inside the Life Sciences Laboratory module for those control functions (labeled as "Astrophysics Racks" in Figure 3.3.1-2).

4.3.1.4 Servicing Facility

The Servicing Facility onboard the IOC Space Station manned core will have the capabilities to service and refuel free-flying satellites (which have been brought to the Station), co-orbiting platforms (interpreted to be multi-payload spacecraft which can be berthed to the Station), payloads attached to the Station, the OMV, and the OMV kits. The Servicing Facility will also provide for the storage of satellites, the OMV, two OMV kits, ORU's, instruments, and tools.

A list of representative free-flying satellites which will require on-orbit servicing or refueling at the Space Station is presented in Table 4.3.1-4. Again, this list of representative free-flying payloads has no official status. The set was developed to reflect the type and quantity of free-fliers proposed for the 1991-1993 time period. Servicing and refueling requirements for this representative payload set, along with serious consideration for eventual expansion to growth capabilities, were used to develop the Servicing Facility for IOC.

Two dedicated work sites, or "bays" are required: one bay is needed to perform servicing operations and the other to perform refueling operations. Several of the spacecraft which will be serviced or repaired contain optical instruments which are highly sensitive to molecular and/or particulate contamination. Separate facilities for servicing and refueling operations are necessary to prevent possible contamination of optics which are exposed during servicing to unexpected leakage of propellants or other contaminants possible in the refueling area.

This concern with the sensitivity of payload instruments to various contaminants dictates that the servicing bay be separated and/or "upstream" from the refueling and fluid storage areas, from the Orbiter berthing area, and from any pressurized modules which may vent contaminants (e.g., laboratory or commercial modules).

TABLE 4.3.1-4 - REPRESENTATIVE SET OF IOC FREE-FLYING SATELLITES REQUIRING SERVICING

IDENTIFICATION CODE	FREE-FLYING SATELLITE SERVICING PAYLOADS
SAA0012	Space Telescope (ST)
SAA0013	Gamma Ray Observatory (GRO)
SAA0014	X-Ray Timing (XTE)
SAA0016	Solar Maximum Mission (SMM)
SAA0017	Advanced X-Ray Astrophysics Facility (AXAF)
SAA0019	Far Ultraviolet Spectroscopy Explorer (FUSE)
-	Leasecraft 1 and Payloads
-	Leasecraft 2 and Payloads
-	Leasecraft 3 and Payloads
-	Leasecraft 4 and Payloads
-	Leasecraft 5 and Payloads

The refueling bay and fluid storage area should be located so as to reduce any hazard potential to satellites being serviced, instruments/payloads externally attached to the Station, or Station systems such as the solar arrays or radiators.

Some satellites may require solar protection while in the servicing and storage areas. If needed, thermal protection may be provided by some type of shield or enclosure.

An access corridor with sufficient clearance must be available for the OMV with attached payload to move close enough to the Station so that the Mobile RMS (MRMS) can grapple and berth the OMV and payload.

MRMS access to Servicing Facility elements is required so that payloads may be moved between the servicing, refueling, and storage areas. Also, ORU's must be moved between the Orbiter and the ORU storage area.

A clear translation path is needed for the movement of EVA crew between the core modules and the Servicing Facility elements.

The elements of the Servicing Facility will need to be provided with utilities including power, lighting, CCTV, liquid lines, and data/communication.

The elements which make up a Servicing Facility that accommodates IOC mission servicing are the following:

a. Servicing Bay: a cylindrical volume (not necessarily enclosed) which is 30 feet in diameter and 70 feet in length. This volume allows for the berthing of a 15-ft-diameter-by-60-ft-long satellite with clearances all around for movement of EVA crew and the placement of workstations. The servicing area will have provisions for berthing payloads either by a Flight Support Structure (FSS), which has tilt and rotation capabilities, or by trunnion latches. Moveable or reattachable berthing assemblies would permit the berthing of more than one payload in this area.

The servicing bay is attached to, and parallel with, the upper keel above the transverse boom.

b. Refueling Bay: A cylindrical volume with the same approximate dimensions as the servicing are and similar berthing mechanisms. The refueling bay is situated on the lower keel just above the radiators.

c. Satellite Storage Area: A cylindrical volume with the same dimensions as the servicing area (i.e., 30-ft-diameter-by-70-ft length) and with the same berthing mechanisms. (This volume is in excess of the approximate 15-ft-diameter-by-60-ft-long volume which is actually required for storage purposes. However, allocation of the additional volume would permit this area to evolve into another servicing area for the growth Station.) The satellite storage area is located across the upper keel from the servicing bay.

d. Fluid Storage Area: An area which will provide facilities for storage of propellants, pressurants, and coolants for the payloads. It is located just beneath the refueling bay at the top of the keel extensions.

e. OMV Storage Area: A cylindrical volume approximately 15 feet in diameter and 4 feet in length. The OMV storage area is situated on the keel extension just beneath the radiators.

f. OMV Kits Storage Area: Two cylindrical volumes approximately 15 feet in diameter and 4 feet in length. They are located on the keel extensions opposite to the OMV storage area.

g. ORU Storage Lockers: Each enclosed rectangular locker is 3 x 5 x 5 feet. Ten lockers will be available for ORU storage. They are placed on the power boom inboard of the alpha joints for convenient access from the servicing bay.

h. Payload Instrument Storage: An enclosed rectangular compartment which is 10 x 20 x 30 feet. It is situated on the lower keel opposite the refueling bay.

i. Tool Storage Lockers: Each enclosed rectangular compartment is 3 x 5 x 5 feet. Four lockers will be available for tool storage. They are located with the ORU storage lockers.

j. Monitoring, Control, and Checkout Equipment: Monitoring, control, and checkout is provided for spacecraft and ORU's in storage as well as spacecraft undergoing servicing and refueling operations. These functions will require volume for display consoles within the pressurized modules.

4.3.1.5 Growth

Customer utilization on the growth Space Station will require an expansion of capabilities present on the initial Station. However, it is not clear at this time just which capabilities will grow and to what degree, or how that growth will drive the Station evolution.

An attribute of the reference Space Station configuration is that it can support growth in any or all of its initial capability areas: servicing and refueling; construction of large space structures; materials processing; life science research; astrophysics and solar physics; Earth remote sensing; or sensor development. Growth of some of these capabilities would require increased crew size (e.g., servicing, construction, life science research). Growth of other capabilities would require significantly increased power (e.g., materials processing). Whichever capabilities eventually come forward as growth requirements, the reference configuration can gracefully evolve to meet them.

4.3.2 Crew Accommodations

Crew accommodations refers to all areas which involve crew systems and crew support. The major portion of these accommodations is in the habitation volume. The laboratory volume contains crew-related items if they aid in crew health, safety and well-being, or if they support the crew in providing customer services. Crew accommodations in all volumes are designed and arranged to the zero-g neutral body posture, traffic patterns, congestion avoidance, cleaning and ease of maintenance. Functional group interrelationships were a prime consideration in the basic arrangement. Other considerations in the design and arrangement of facilities and equipment included the ability to support reconfiguration, growth and update; the ability to access facilities and equipment; standardization of crew interfaces and associated equipment; and accommodation of anthropometric strength and size measurements.

4.3.2.1 Habitation Volume

In the reference configuration, the habitation volume consists of two habitation modules which contain all those facilities and equipment required for crew sustenance and station operation. The following areas and categories of equipment are contained in the two modules: crew quarters, galley/wardroom (including housekeeping supplies), health maintenance facility/exercise area, personal hygiene area, workstation areas, stowage areas, waste management facilities, and laundry facility. These areas contain equipment, the facilities which "house" this equipment and access volume which facilitates its use. The internal configuration of these modules is shown in Figure 4.3.2.1-1, 4.3.2.1-1a, and 4.3.2.1-1b (Hab 1) and Figures 4.3.2.1-2, 4.3.2.1-2a and 4.3.2.1-2b (Hab 2).

4.3.2.1.1 Crew Quarters

Private quarters for each crewmember are located in Hab Module 2. Each crewmember has been provided with 150 cu ft of volume which contains a sleepstation with bedding, a communications unit, a desk, a CCTV/CRT, a portable workstation with certain command/control functions, stowage volume (20 cu ft including a "dresser" and area for personal items), a bulletin board, audio/video entertainment and a mirror. These accommodations are shown in Figure 4.3.2.1-3 (Sleep Compartments) and Figure 4.3.2.1-4 (Sleep Compartment Furnishing in Working Position). The volume which has been allocated is below desirable standards, but has been expanded beyond those minimum levels included in previous investigations. Private crew quarters utilize 88 inches of the length of the module.

4.3.2.1.2 Galley/wardroom

The galley/wardroom is located in Hab Module 1. The galley, as depicted in Figure 4.3.2.1-5, occupies 747 cu ft of which 225 cu ft is equipment

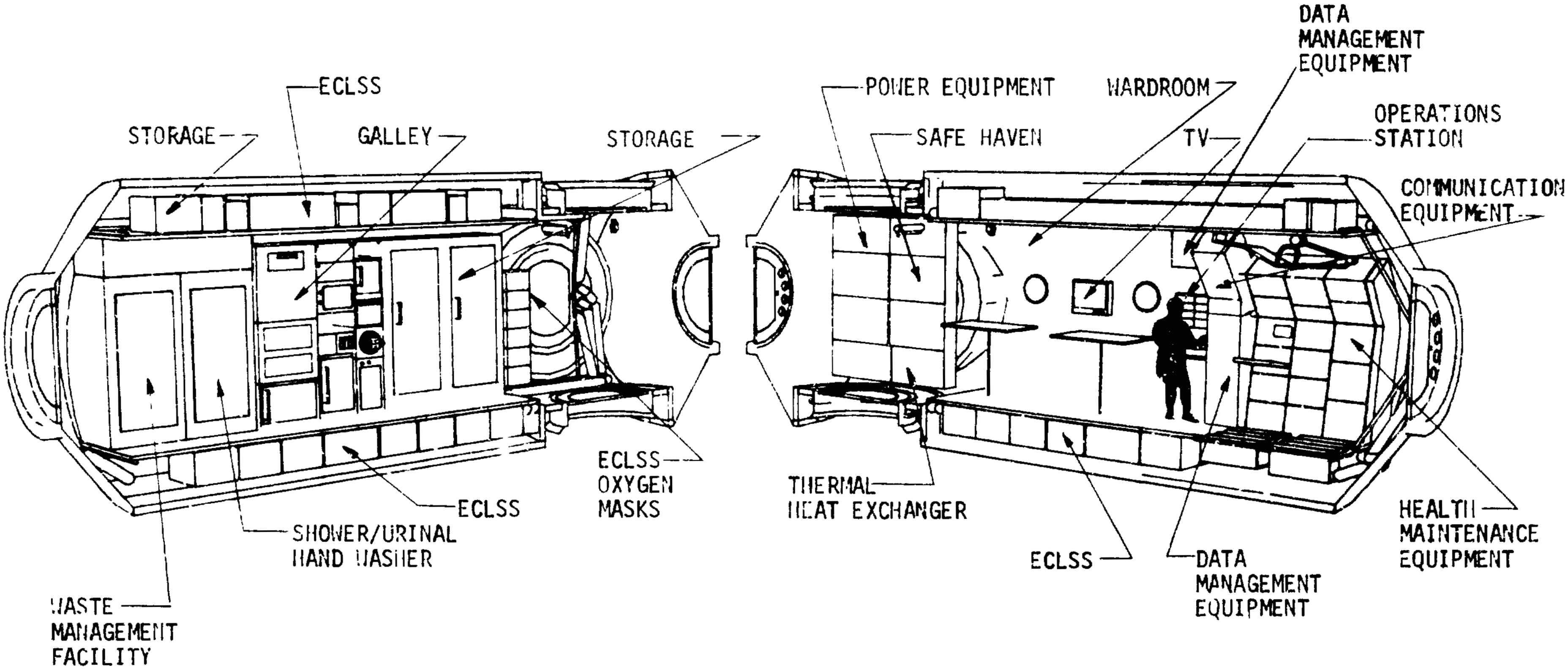


Figure 4.3.2.1-1 - HAB MODULE 1.

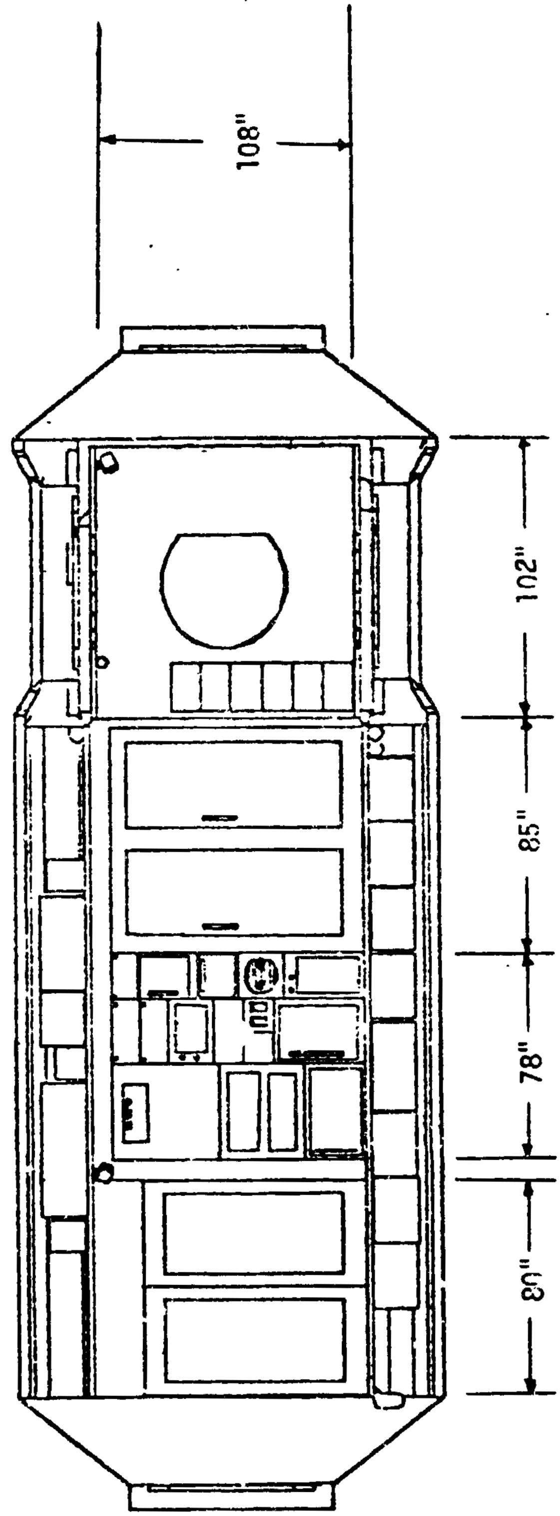


Figure 4.3.2.1-1a - HAB MODULE 1.

PLAID

ORIGINAL PAGE IS
OF POOR QUALITY

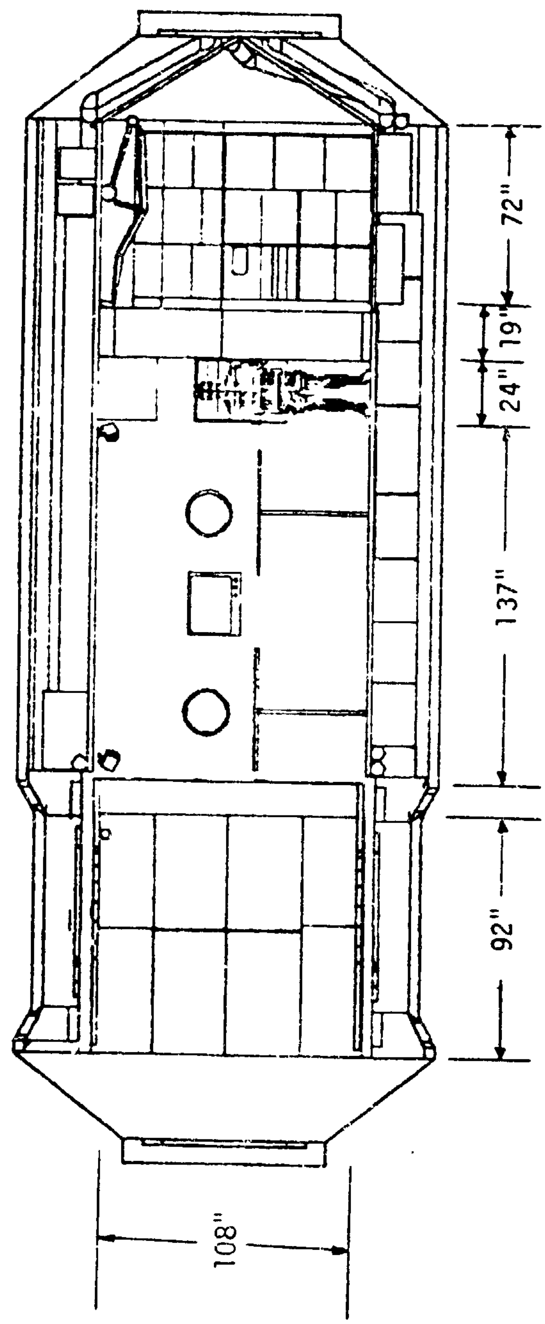


Figure 4.3.2.1-1b - HAB MODULE 1.

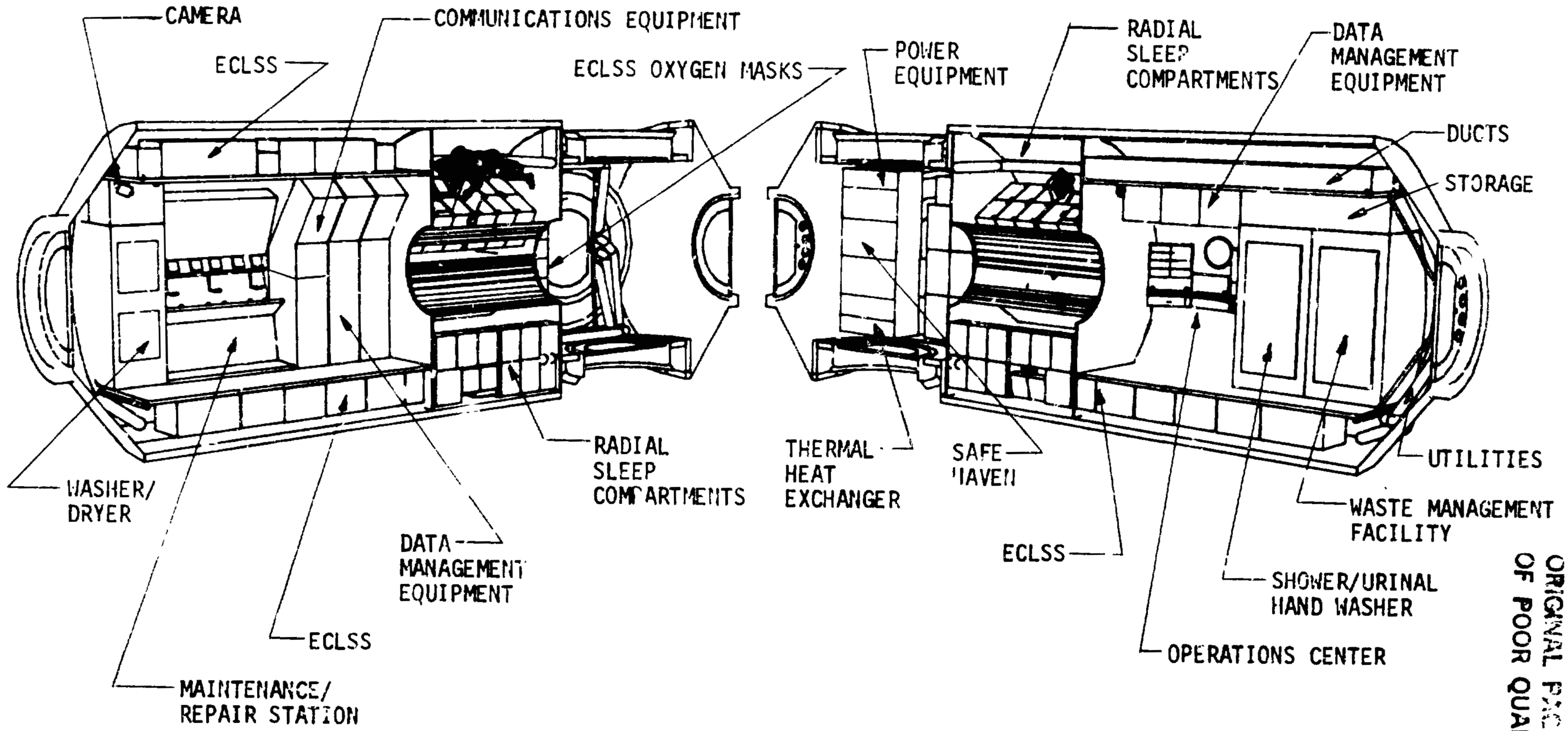


Figure 4.3.2.1-2 - HAB MODULE 2.

PLAID

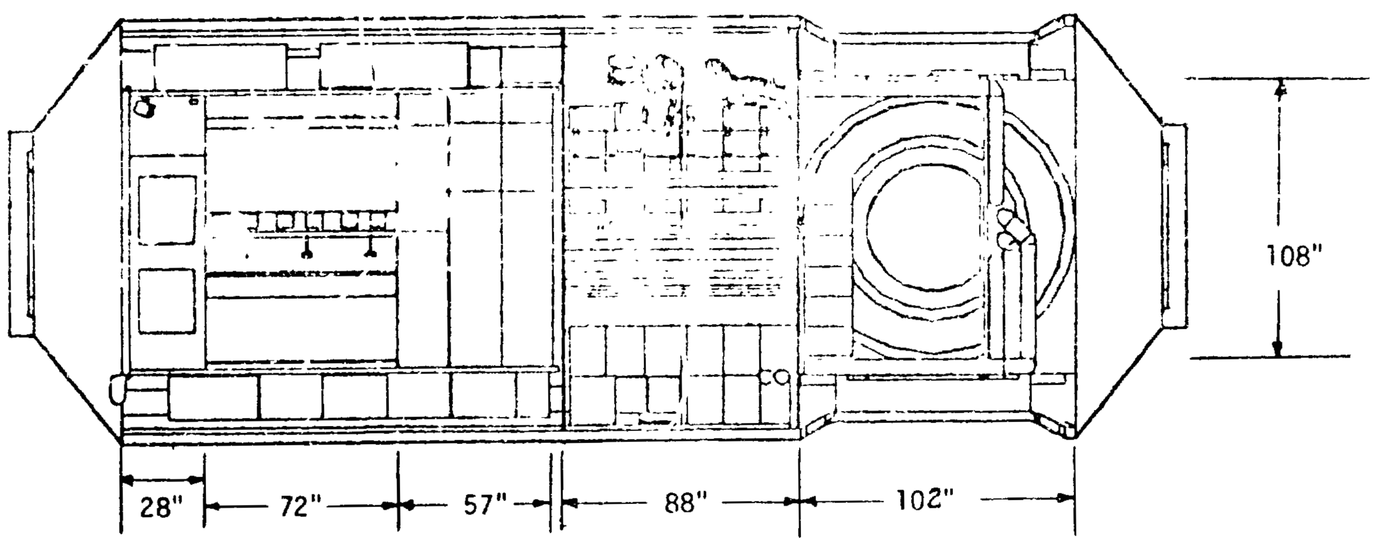


Figure 4.3.2.1-2a - HAB MODULE 2.

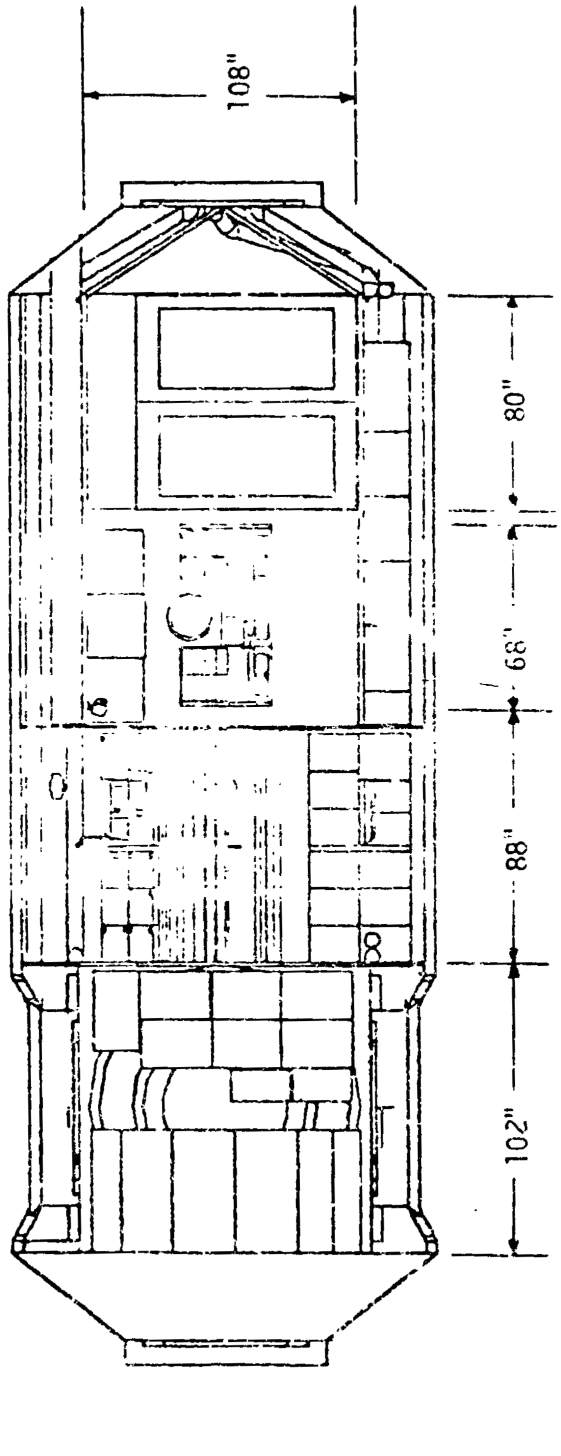


Figure 4.3.2.1-2D - HAB MODULE 2.

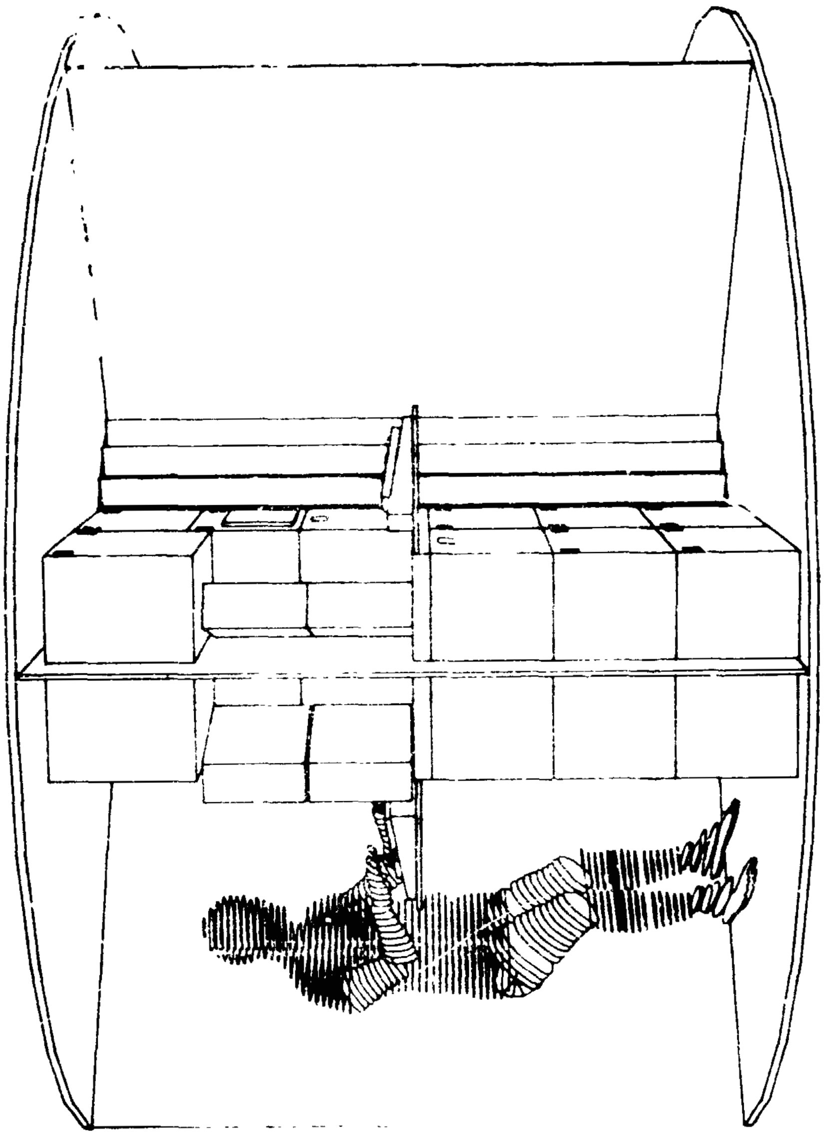


Figure 4.3.2.1-3 - SLEEP COMPARTMENTS.

C-2

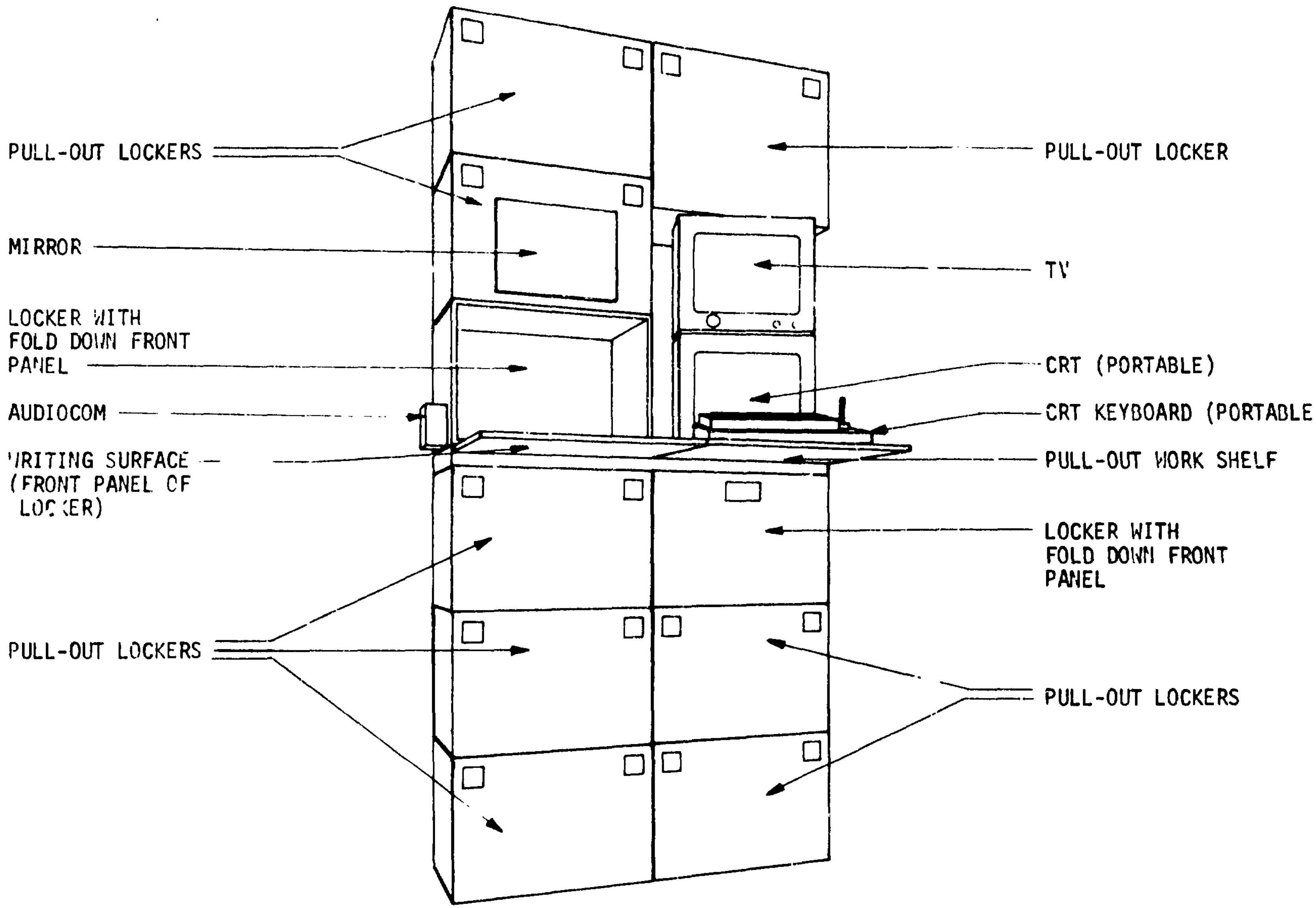


Figure 4.3.2.1-4 - SLEEP COMPARTMENT FURNISHINGS IN WORKING POSITION.

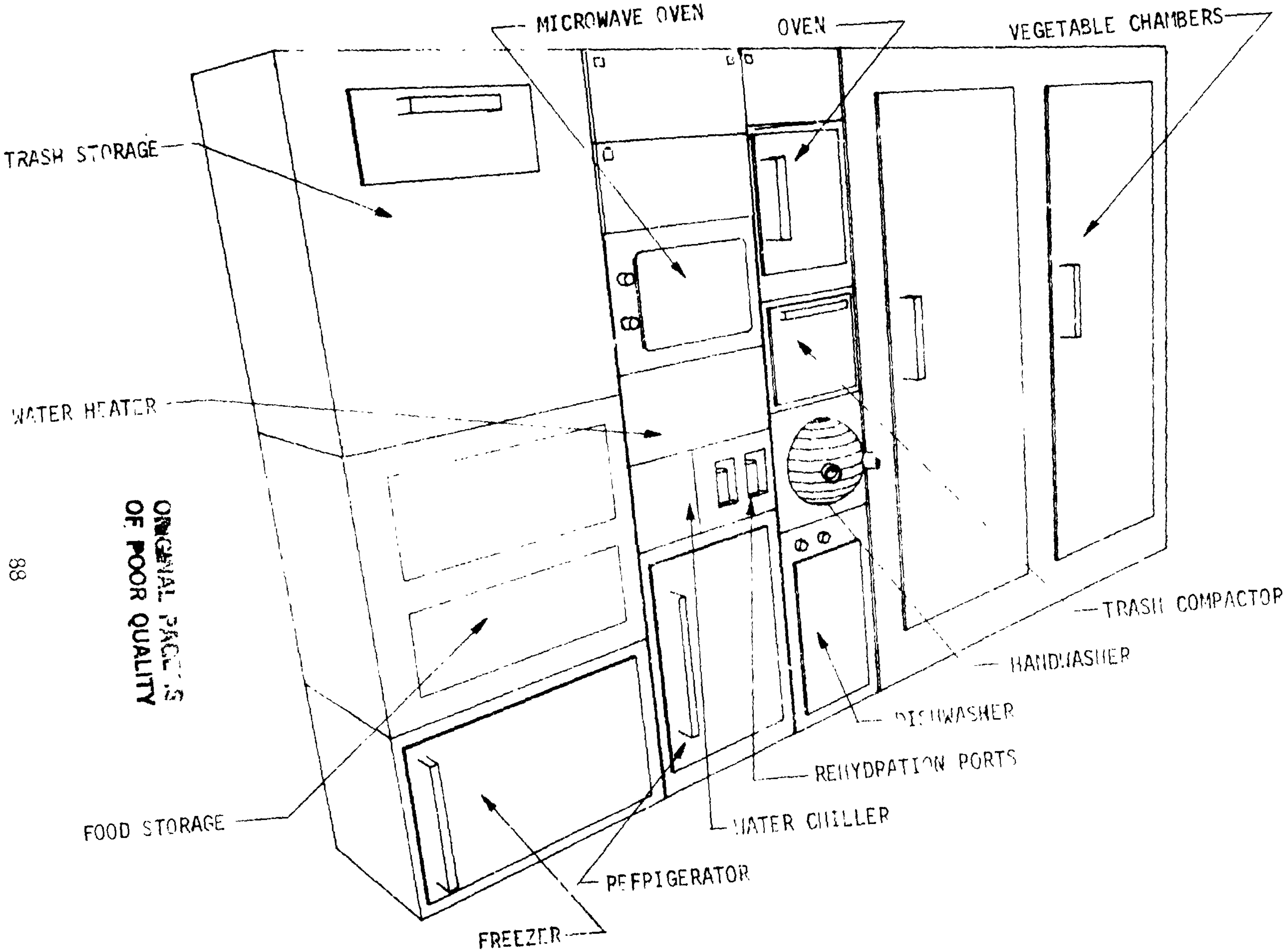


Figure 4.3.2.1-5 - HAB MODULE 1 GALLEY.

ORIGINAL PAGE IS
OF POOR QUALITY

volume, utilizes 163 inches of the length of the module and provides the following accommodations: equipment and supplies necessary for the preparation and heating of food and drink; clean-up and stowage for 14 days is provided. Food and drink stowage is provided by the refrigerator, freezer and ambient food stowage lockers. Cooking accommodations are provided by the oven and microwave oven. There is additional stowage for utensils/appliances, housekeeping supplies, and trash. Galley trash collection and compaction for wet and bulk trash is provided. There is also a handwasher and a zero-g dishwasher. Galley equipment and food are GFE.

The wardroom is an area, across from the galley, occupying 576 cu ft, 137 inches in length, designed to accommodate the entire crew simultaneously as a dining area as well as a meeting area, lounge, viewing and recreation area. This area contains tables (4 sq ft per person), audio/video entertainment equipment, game kits, a window, and IVA communications.

The accommodations provided were chosen to be above minimum levels in an effort to increase acceptability for long-duration flight and to contribute to the optimization of crew performance.

4.3.2.1.3 Health maintenance facility/exercise area

The Health Maintenance Facility/Exercise Area is located in Hab Module 1, utilizing 72 inches of the length of the module, some 65 cu ft of equipment volume and occupying a total of 320 cu ft. The equipment contained in this volume provides for inflight preventive, diagnostic and therapeutic medical and dental capabilities. The health maintenance facility (HMF) is principally for the application of countermeasures including exercise and the treatment of health problems. The HMF and exercise area as depicted in Figure 4.3.2.1-6, share the same volume as well as some of the same equipment. The approach to the design of the HMF is as follows. Its capabilities are tailored to the degree of mission complexity and defined "acceptable medical risks." The requirements are derived from inflight medical experience, mission related activities and hazards, and projected low probability medical scenarios which could be adequately treated with minimal equipment.

The exercise area contains the equipment necessary to enable the crew to retain the requisite physical body tone and also provides a means for recreation. Provisions also exist in the exercise area for monitoring body mass and dimensional changes.

4.3.2.1.4 Personal hygiene areas

Personal hygiene areas which provide facilities for body waste collection and disposal, personal cleanliness, and bathing are located in Hab Modules 1 and 2. Each area is 80 inches in length. There are two waste management systems (for both fecal and urine collection); two additional urine collection systems; two personal hygiene stations for face and handwashing and oral hygiene; and one additional handwashing station in the galley and in each of the remaining habitable modules.

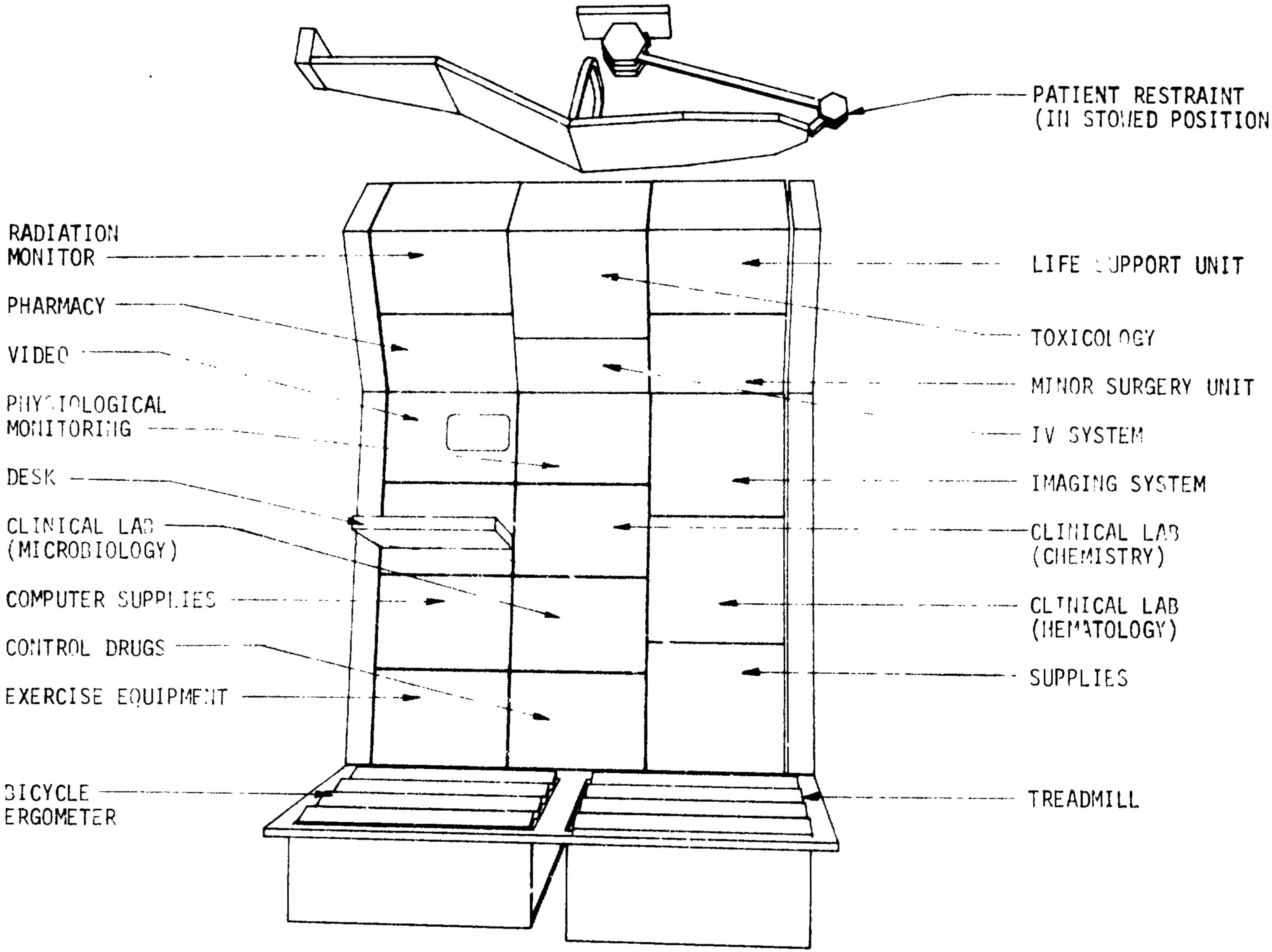


Figure 4.3.2.1-6 - HEALTH MAINTENANCE FACILITY.

The facilities for collecting and disposing of vomitus, fecal matter and urine are sized to anthropometric specifications. Commode compartments are large enough to permit donning, doffing, and temporary stowage of clothing.

The facilities for shaving, hair grooming, teeth cleaning and expectoration are designed for easy cleaning and maintenance on orbit. Handwash facilities have hot, cold, and mixed water controls.

A full body shower facility is provided in each of the Hab modules. Each has hot, cold, and mixed water controls, permits hair and scalp washing, and provides a temperature controlled (heated) private dressing area. There are restraints to stabilize the crew while bathing. Means to facilitate drying and cleanup, waste water collection, and transfer to a storage tank for processing are provided.

4.3.2.1.5 Workstations

Workstations are located throughout the Space Station in all the habitable modules. They are conceived as generic stations capable of supporting Space Station operations, customer services, crew health and equipment maintenance. Hab module 1 contains the following workstations: the work/control station for Space Station operations (Figures.2.1-7) and portions of the health maintenance facility (Figure 4.3.2.1-6). Hab module 2 contains a larger command/control station (Figure 4.3.2.1-8) and the maintenance workstation for equipment maintenance and repair. Each of the Lab modules contain workstations. There is one operations station in Lab 1 (Figure 4.3.2.1-9, Life Sciences Lab) and two operation stations in Lab 2 (Figures 4.3.2.1-10, 4.3.2.1-11; Materials Processing Lab). All workstations are conceived to have the necessary and appropriate level of automation, tools, equipment, and communications required for the specific tasks and activities to be accomplished.

4.3.2.1.6 Crew stowage

Twelve hundred cu ft of stowage has been provided for a crew of six. Stowage volume has been allocated to each module as indicated in Table 4.3.2.1-1. This allocation is the result of the stowage requirements presented in Table 4.3.2.1-2. The logistics module contains the greatest portion of the stowage volume including stowage for clothing for 14 days (this assumes a laundry facility on-board), for a 90-day supply of food, for bedding, and for food trash stowage. The two Hab modules have approximately equal amounts of stowage volume including stowage in the galley, crew quarters, personal hygiene areas, and at the various workstations. Labs 1 and 2 each has a small amount of stowage volume provided. Each stowage container is 2 cu ft and weighs 5 lbs/cu ft (10 lbs) empty.

These stowage requirements have been derived from previous inflight experience as well as extrapolation of consumables for six people over a 90-day period.

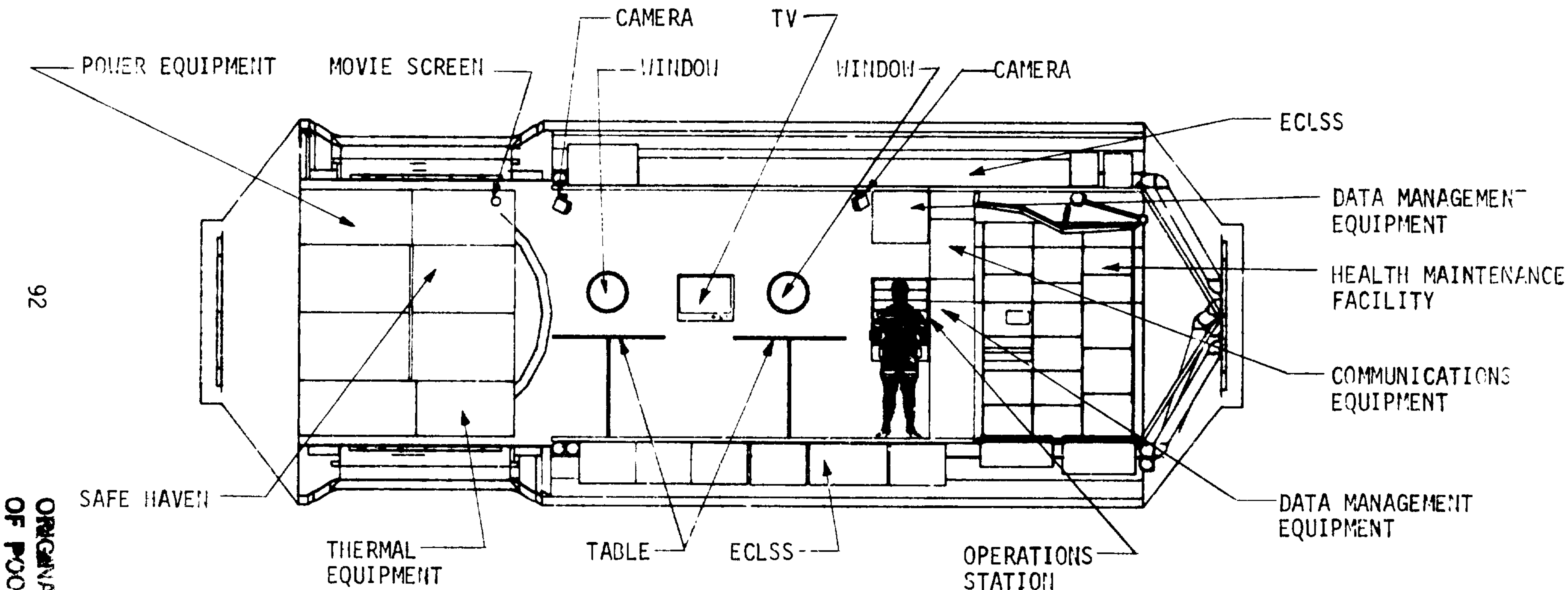


Figure 4.3.2.1-7 - HAB MODULE 1 WORK/CONTROL STATION.

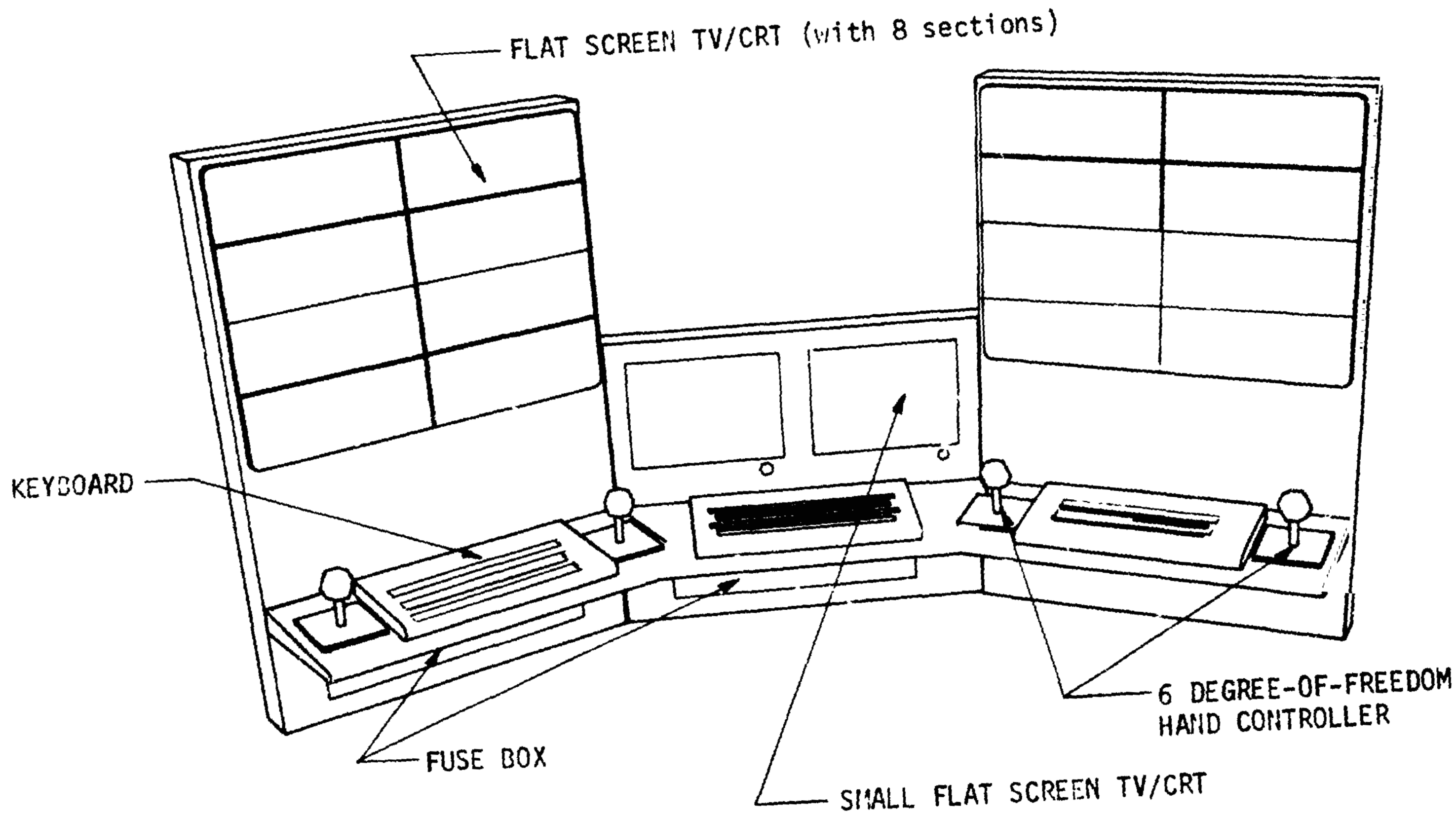


Figure 4.3.2.1-8 - FRONT VIEW OF OPERATIONS CENTER HAB MODULE 2.

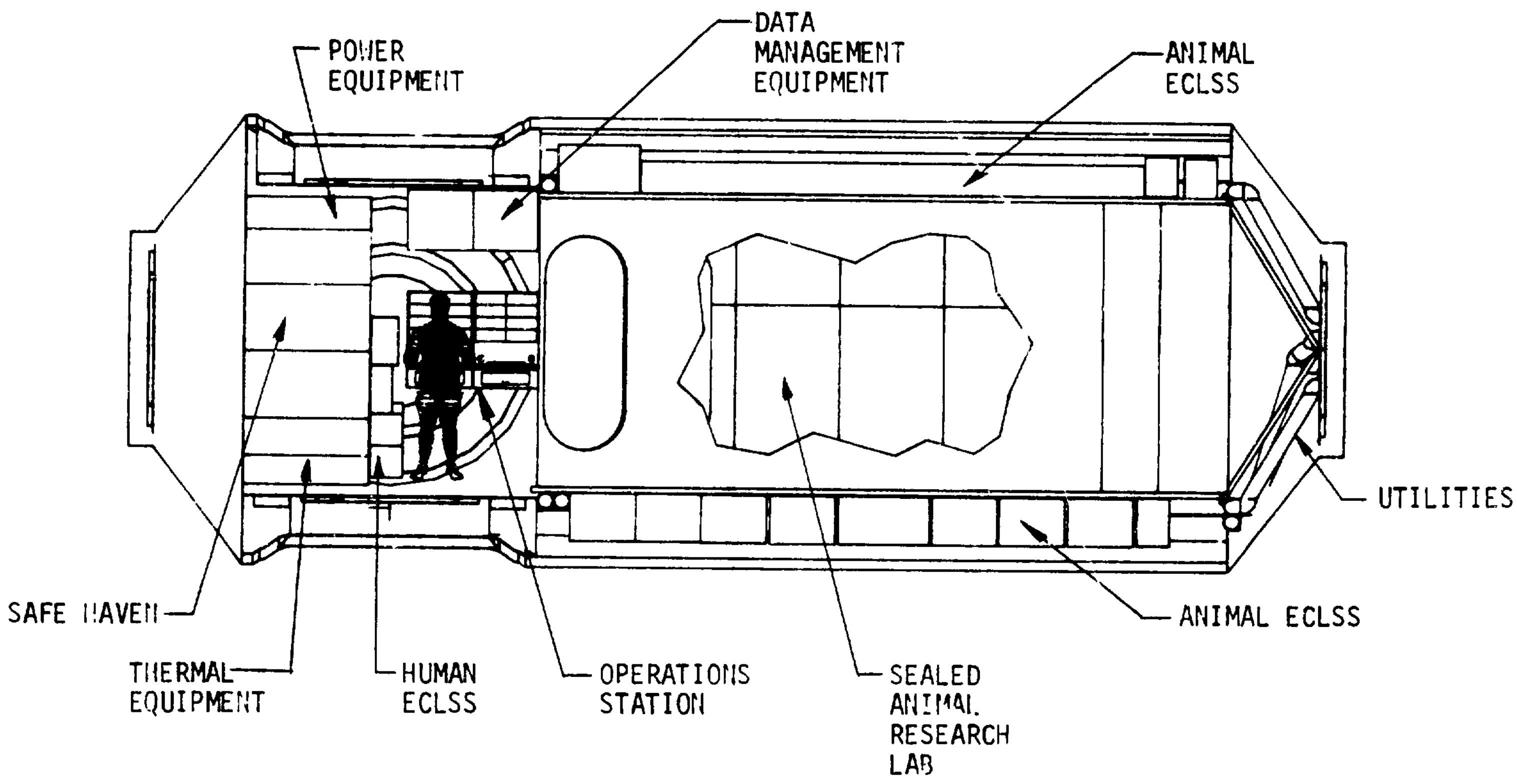


Figure 4.3.2.1-9 - LIFE SCIENCES LAB OPERATIONS STATION.

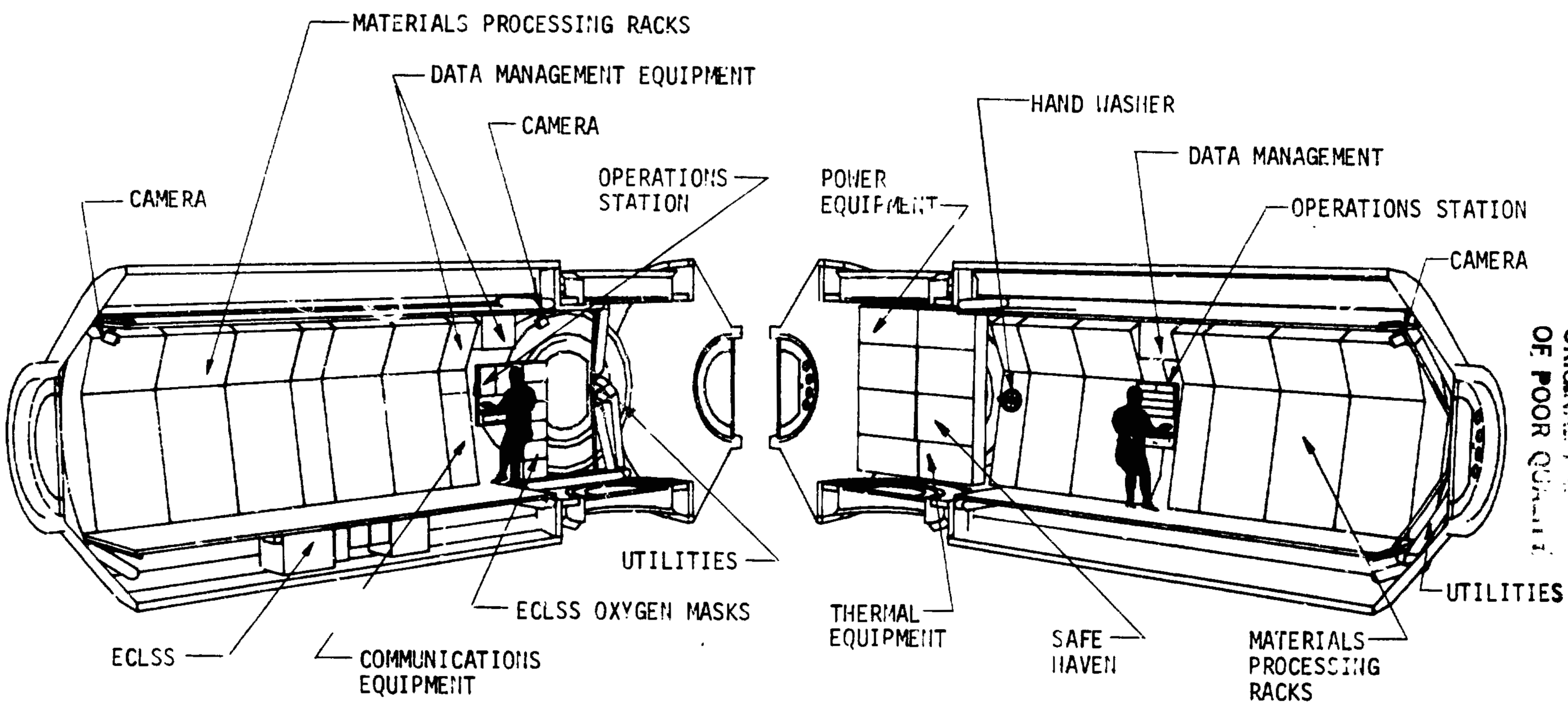


Figure 4.3.2.1-10 - MATERIALS PROCESSING LAB.

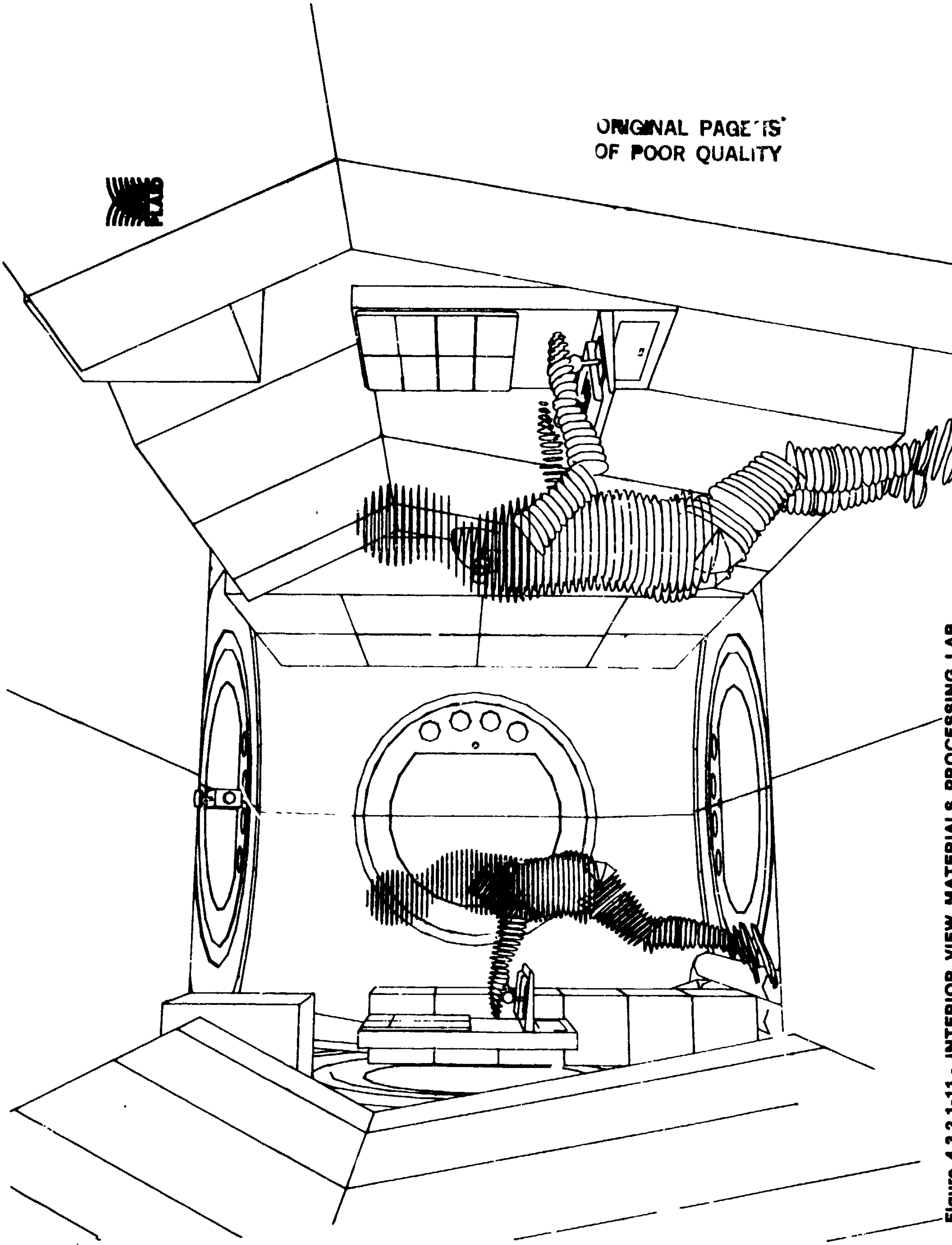


Figure 4.3.2.1-11 - INTERIOR VIEW MATERIALS PROCESSING LAB.

TABLE 4.3.2.1-1 - STOWAGE ALLOCATION FOR CREW OF 6 (PER MISSION)

		NO. OF CONTAINERS	STOWAGE VOLUME (CU FT)
HAB 1	TOTAL	118.00	236.00
GALLEY		40.00	80.00
WARDROOM		10.00	20.00
MAINTENANCE WORKBENCH		10.00	20.00
EQUIPMENT CONTAINERS/RESTRAINTS		3.00	6.00
WORK/CONTROL STATION		6.00	12.00
PERSONAL HYGIENE AREA		7.00	14.00
ADDITIONAL MISCELLANEOUS STOWAGE		42.00	84.00
HAB 2	TOTAL	106.00	212.00
CREW QUARTERS		60.00	120.00
COMM/CONTROL STATION		6.00	12.00
PERSONAL HYGIENE AREA		7.00	14.00
EQUIPMENT CONTAINERS/RESTRAINTS		3.00	6.00
ADDITIONAL MISCELLANEOUS STOWAGE		30.00	60.00
LOGISTICS	TOTAL	264.00	528.00
CLOTHING (14 DAYS)		32.00	64.00
FOOD (90 DAYS)		94.00	188.00
BEDDING (90 DAYS)		35.00	70.00
FOOD TRASH STOWAGE (EMPTY)		50.00	100.00
ADDITIONAL MISCELLANEOUS STOWAGE		53.00	106.00
LAB 1	TOTAL	56.00	112.00
LAB 2	TOTAL	56.00	112.00
	STATION TOTAL	600.00	1200.00

TABLE 4.3.2.1-2 - STOWAGE REQUIREMENTS FOR CREW OF 6.00 PER MISSION

SUBSYSTEM	TOTAL VOLUME (CU FT)	
CREW QUARTERS (INCLUDES CLOTHING, PART OF PERSONAL HYGIENE PROVISIONS AND PART OF ANCILLARY PROVISIONS)	120.00	(20 CU FT/ CREWMEMBER)
GALLEY	74.30	
REFRIGERATOR (14 DAYS)	12.00	
FREEZER (14 DAYS)	12.00	
AMBIENT FOOD (14 DAYS)	16.80	
UTENSIL/APPLIANCE STOW	2.50	
TRASH STOW (EMPTY)	20.00	
HOUSEKEEPING SUPPLIES	11.00	
WORKSTATIONS		
ANCILLARY PROVISIONS (e.g., WRITING EQUIPMENT, FILM, CAMERA EQUIPMENT)	22.66	
SHOWER/HANDWASH/WMF (PERSONAL HYGIENE)	28.50	
TISSUE DISPENSER	9.75	
PERSONAL WETWIPES	4.50	
TOWELS	12.75	
WASHCLOTHS	1.50	
LOGISTICS MODULE	528.00	
CLOTHING (14 DAYS)	63.00	
FOOD (90 DAYS)	188.00	
BEDDING (90 DAYS)	69.00	
FOOD TRASH STOWAGE (EMPTY)	100.00	
ADDITIONAL STOWAGE	108.00	
STOWAGE CONTAINERS (ADDITIONAL)	426.54	
EQUIPMENT CONTAINERS/RESTRAINTS	11.00	
MAINTENANCE WORKSTATION	20.00	(PART OF WORKBENCH)
WARDROOM	20.00	
ADDITIONAL HAB 1	90.00	
ADDITIONAL HAB 2	60.00	
ADDITIONAL LAB 1	112.00	
ADDITIONAL LAB 2	112.00	
TOTAL	1200.00 (2.222 CU FT/MAN/DAY) (5 LBS/CU FT - EMPTY)	

4.3.2.1.7 Laundry facility

A laundry facility has been provided in Hab module 2. The reference configuration concept provides for a washer and dryer. The presence of a laundry facility onboard Space Station substantially reduces the amount of clothing necessary for a 90-day mission. With a laundry facility, only a 14-day supply of clothing is necessary. The clothing supply weighs 185.4 lbs for a crew of 6 and occupies 63 cu ft of volume, while the washer/dryer weighs TBD and occupies TBD ft^3 of volume. Without a laundry facility, a 90-day supply of clothing is necessary. This weighs 1140 lbs for a crew of 6 and occupies 388.35 cu ft.

4.3.2.2 Nonhabitation Volume

There is additional volume for crew accommodations in the non-habitaiton areas. This volume includes the two laboratory modules, the airlocks, the logistics module, and safe haven.

4.3.2.2.1 Crew accommodations in laboratory volume

The crew accommodations in the laboratory volume aid in crew health, safety, and well-being (e.g., lighting, windows, handwashers), and they support the crew in providing customer services (e.g., workstations, stowage volume). Figure 4.3.2.2-1 depicts the internal configuration of Lab 1 (Life Sciences Lab), and Figure 4.3.2.2-2 depicts the internal configuration of Lab 2 (Materials Processing Lab). Most of the internal volume of Lab 2 is comprised of racks for materials processing. The allocation of racks for specific types of processes is depicted in Figures 4.3.2.2-3 and 4.3.2.2-4. Crew accommodations in this Lab consist of a handwasher (as described in 4.3.2.1.4, Personal Hygiene Areas), communications equipment, and two operations stations (as described in 4.3.2.1.5, WorkStations).

Lab 1 contains facilities and accommodations primarily for Life Sciences experiments. However, there are some racks allocated for Astrophysics experiments. That portion of this module dedicated to Life Sciences experiments consists of the Human Research Facility (HRF) and the Animal/Plant Research Facility (APRF). Cohabiting animals and humans in the same module requires special accommodations since each group has micr organisms that could adversely affect the other. Therefore, the ECLSS for the animals habitat must be separate from that provided for the crew. The environment in which animal research is done should also be separate from the environment of the Space Station. Therefore, a "biological lock" isolating the APRF from the rest of the Space Station is required. A room for changing clothing and/or donning cap, gown, mask, and gloves is provided at the entrance of the APRF, as is a shower or handwasher (TBD). The ARPF occupies 670 cu ft. A floor plan is shown in Figure 4.3.2.2-5. Equipment includes racks, a surgical table, and biolock/shower. The nonhuman Life Science community has listed a 1G centrifuge as part of their experimental equipment. The inclusion of such a centrifuge alters the internal configuration of this lab module as indicated in Figure 4.3.2.2-6. The ECLSS to support the ARPF is sized

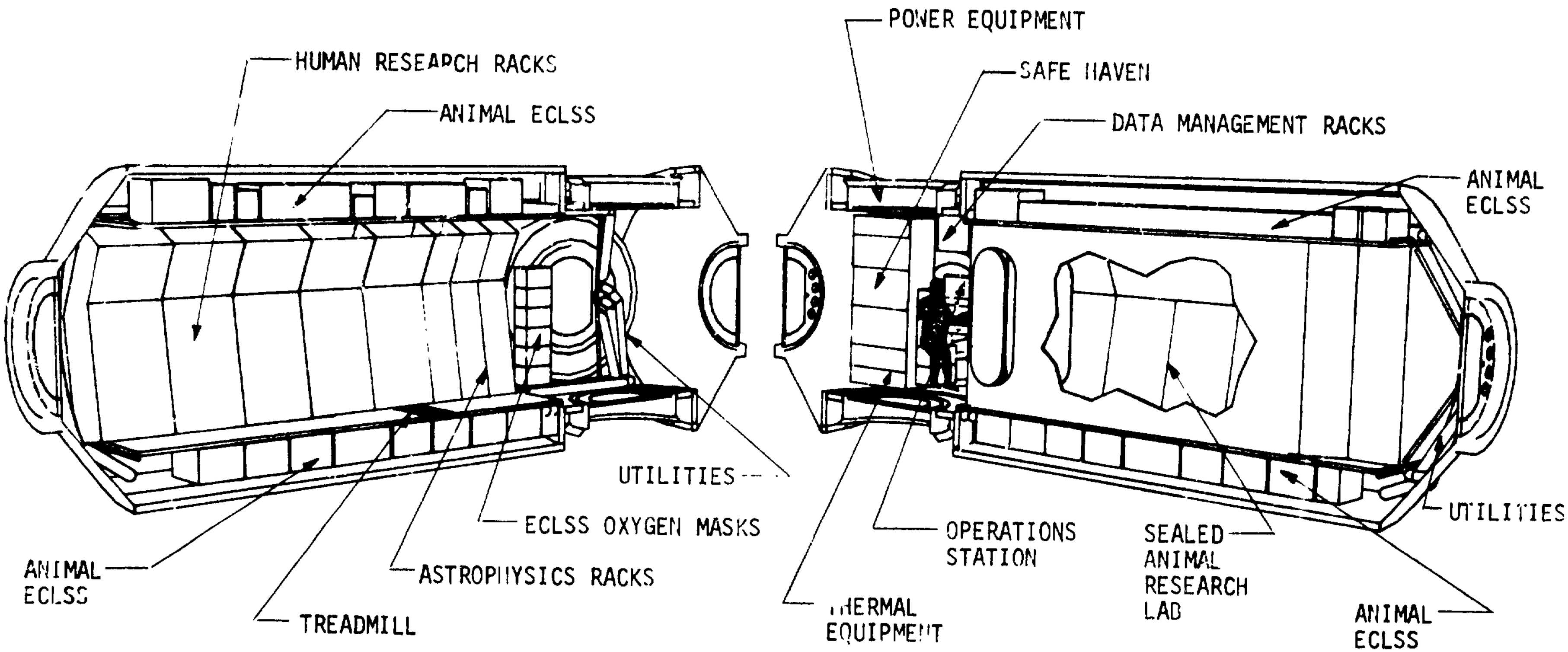


Figure 4.3.2.2-1 - LIFE SCIENCES LAB.

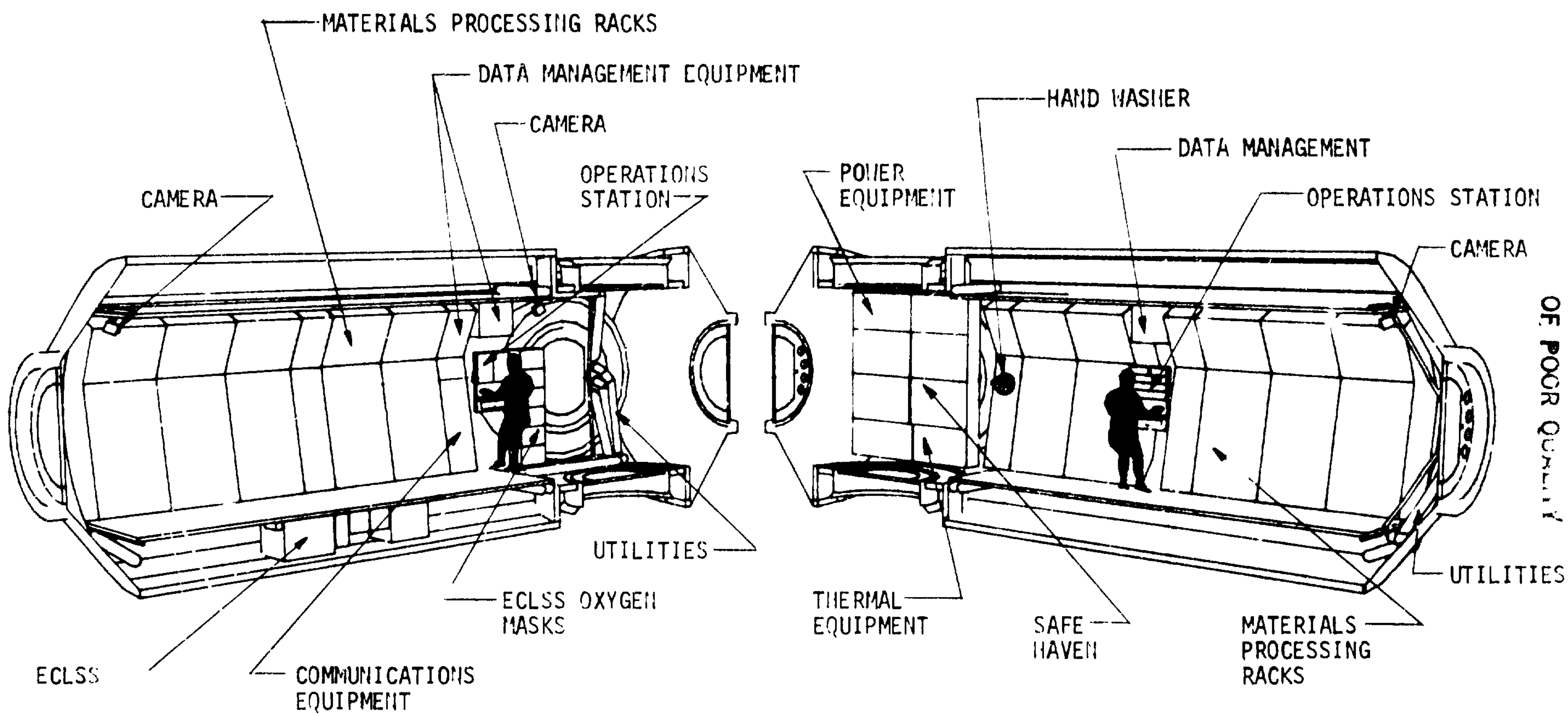


Figure 4.3.2.2-2 - MATERIALS PROCESSING LAB.

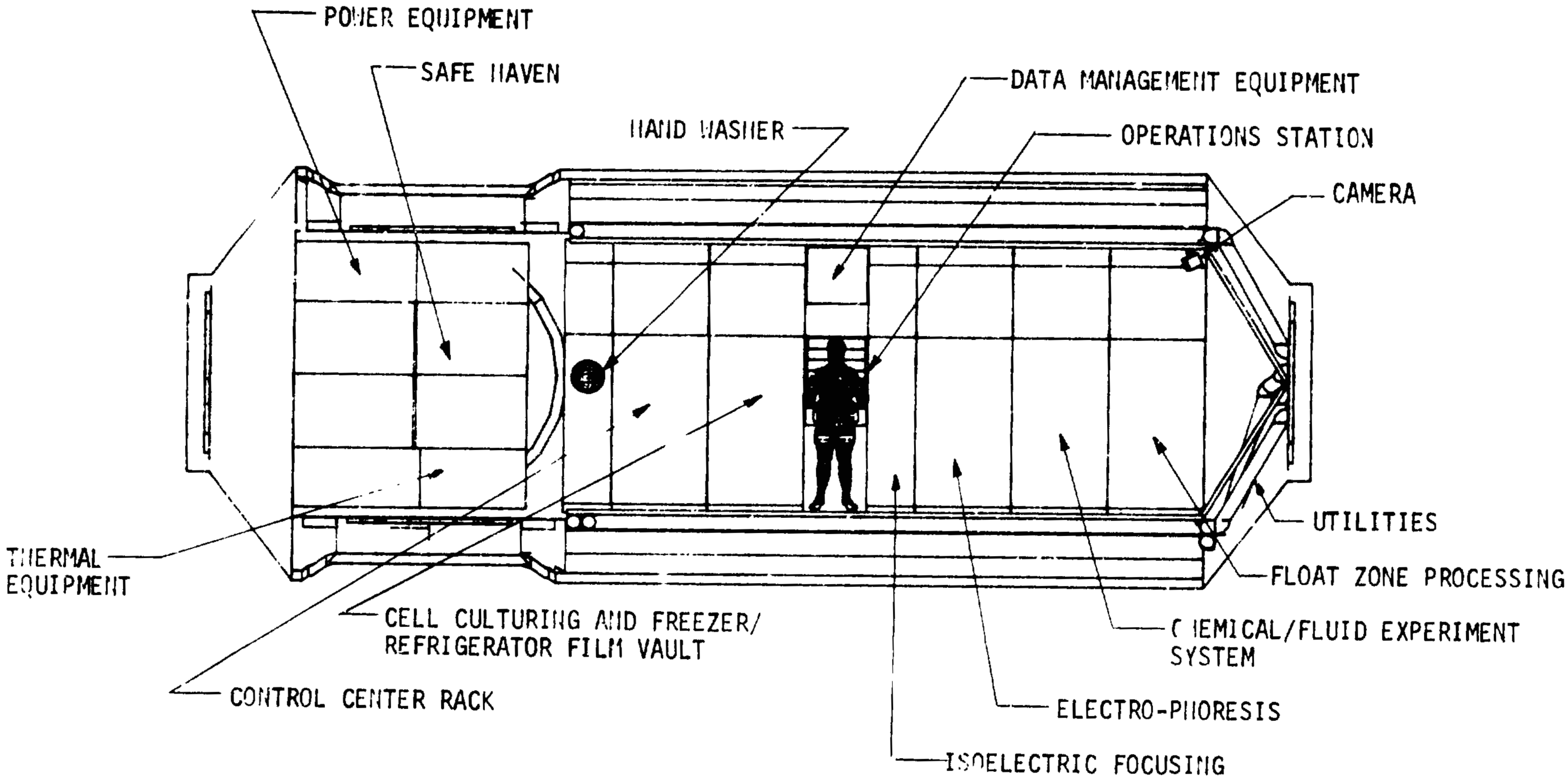


Figure 4.3.2.2-3 - MATERIALS PROCESSING LAB.

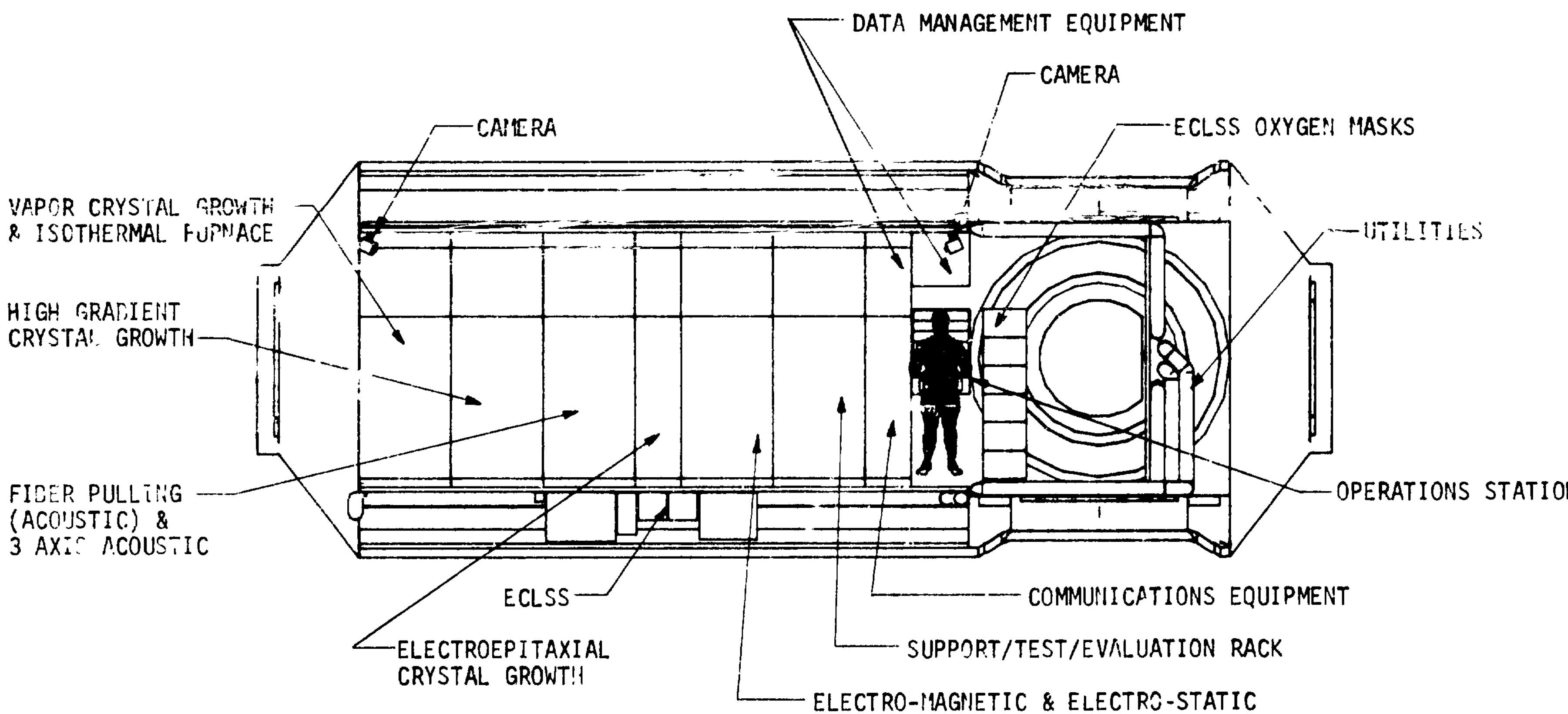


Figure 4.3.2.2-4 - MATERIALS PROCESSING LAB.

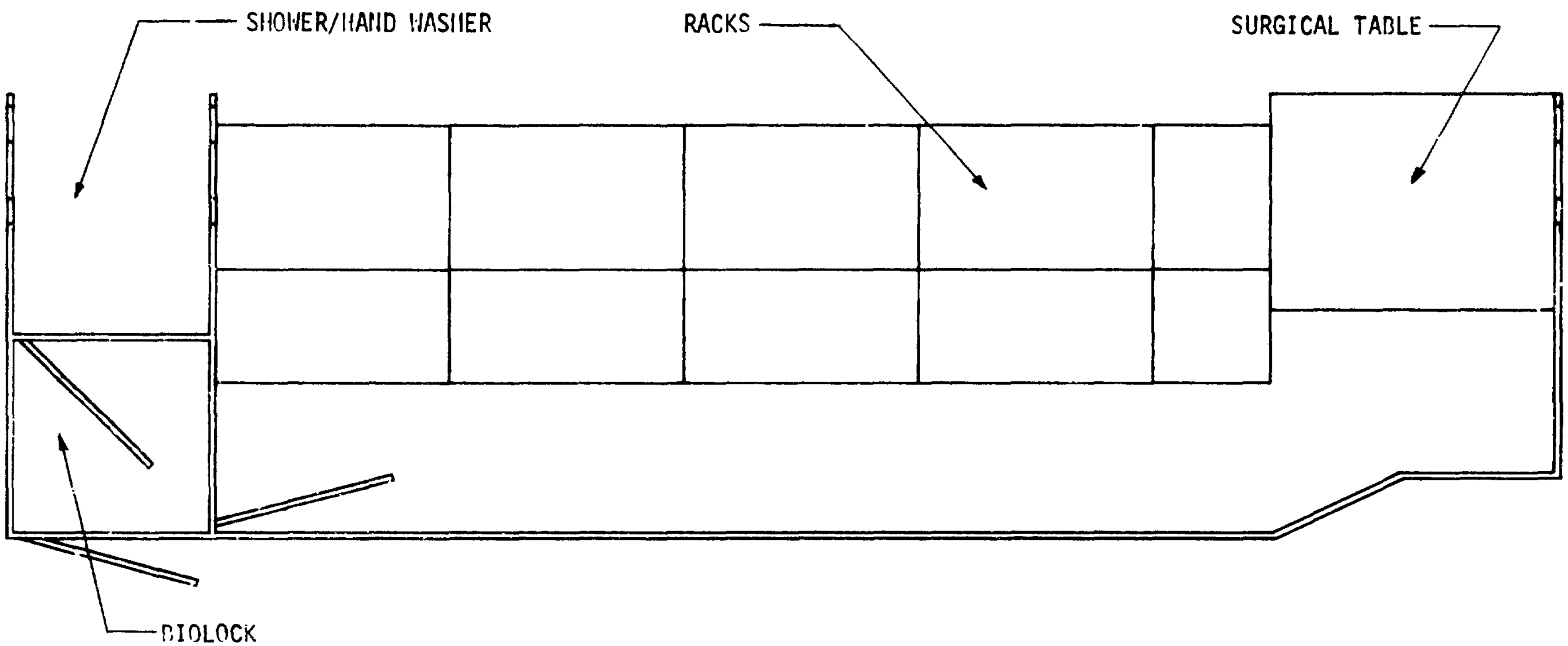


Figure 4.3.2.2-5 - ANIMAL/PLANT RESEARCH LAB.

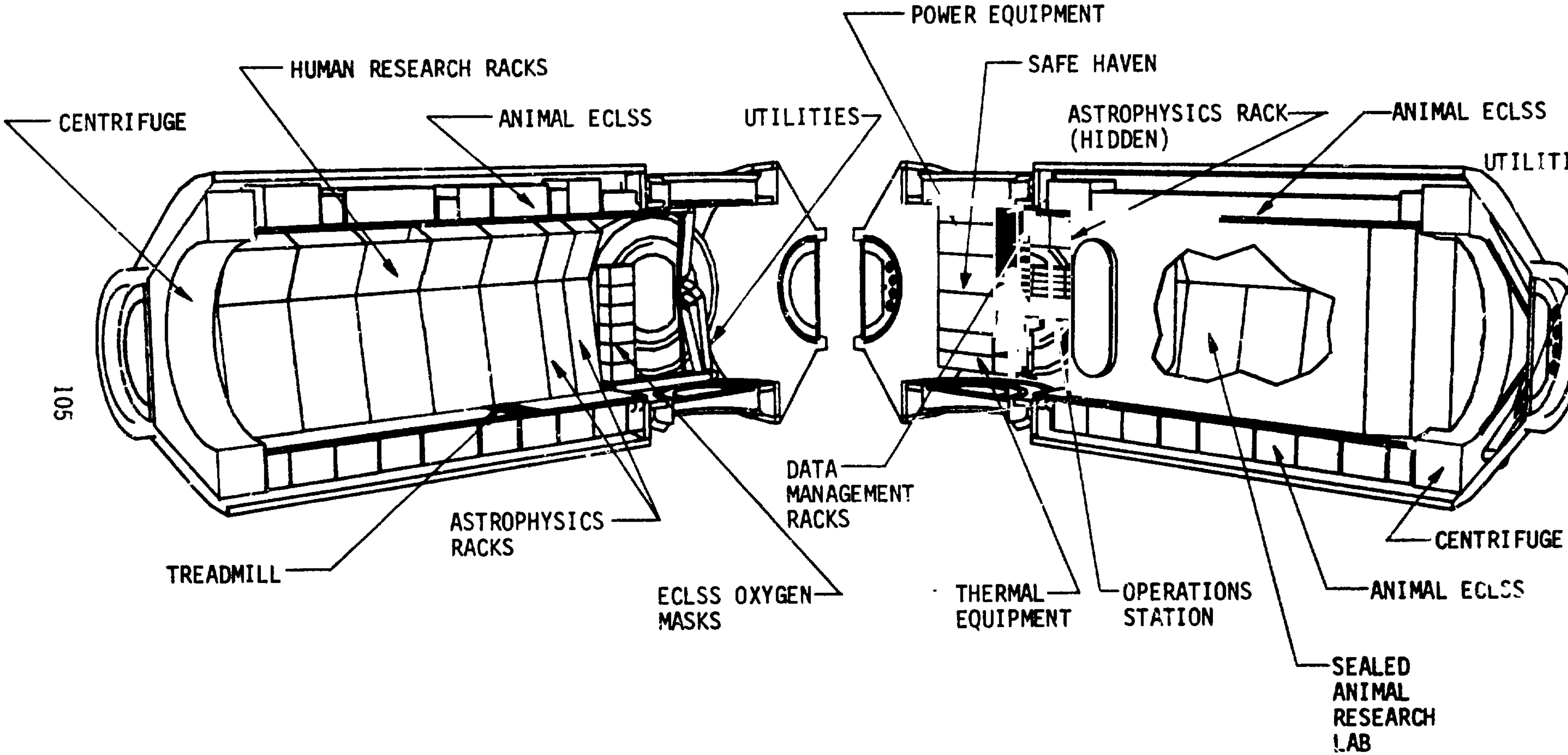


Figure 4.3.2.2-6 - LIFE SCIENCES LAB WITH CENTRIFUGE.

for four crew capability; accommodating two mission specialists and TBD animals (rats, mice, small primates). It requires 375 cu ft of volume of which approximately 245 cu ft is equipment volume with the remainder being access volume to this equipment. The HRF consists of racks to support experimental equipment (Fig. 4.3.2.2-7) and a treadmill for research purposes. In total, it occupies 462 cu ft of volume. In addition, there is an operations station with racks to accommodate data management.

4.3.2.2.2 Airlocks/hyperbaric chamber

Two EVA airlocks, one with hyperbaric chamber capability, are planned for the Space Station. Details are presented in paragraph 4.4.9.2.3 EVA Airlock.

4.3.2.2.3 Safe haven provisions

Safe haven provisions are located in each of the four habitable volumes (both Hab modules and both Lab modules). It consists of a 22-day supply of food, dry and wet wipes, clothing, sleep restraints, waste, and trash stowage. Such a supply, sized for two crewmembers, is stowed in the module segment containing hatches and mechanisms and occupies 50 cu ft of each of these segments. Thus, if any one of the four modules must be evacuated and isolated, the remaining three modules contain safe haven supplies for six crewmembers.

4.3.2.2.4 Crew accommodations in the logistics module

The logistics module contains a 20-cu ft refrigerator, a 60-cu ft freezer and 528 cu ft of stowage containers to support the crew for 90 days. There will be a 90-day supply of food and additional galley provisions, housekeeping supplies, clothing, bedding, personal hygiene items, equipment containers and restraints, ancillary provisions (batteries, writing supplies, paper, etc.), and resupply items for the HMF and maintenance workstation. All of these accommodations occupy 1,675 cu ft of volume (equipment, supplies and access volume included). A breakdown of these accommodations is presented in Table 4.3.2.2-1.

4.3.2.3 Crew Facility Equipment

The family of items required to outfit modules for habitation is defined herein. Such facility equipment includes secondary structure, hatches, mechanisms, utilities (e.g., power, data/comm., thermal, and ECLSS), lighting, windows, and maintenance facilities. Crew Consumables and habitability provisions are not included in this paragraph.

4.3.2.3.1 Secondary structure

Secondary structure in the modules consists of walls, floors, crew transfer tunnels, and framework for retaining and positioning stowage containers, habitability subsystems, utilities, crew workstations, and

TV CAMERA CONN PANEL VTR SWITCH STORAGE VIDEO RECORDER	STORAGE	VACUUM I/F STORAGE DRAWER VIDEO MONITOR LSLE MICRO ELECTRONICS	DISC STORAGE VIDEO DISPLAY STORAGE HARD COPY UNIT	ICRS INTERCOM BSTA TV CAMERA CONN VIDEO ANALOG			
DISPLAY PROPORTIONAL COUNTER ELECTRONICS COMPUTER LSLE CENTRIFUGE LSLE ECHO ECHO SCANNER SCANNER CONVERTER EPSP ACCESS EPSP	WORKBENCH REFRIGERATOR FREEZER	CV/CP I/F PANEL CONTROL UNIT LSLE STRIP CHART GAS TANK ASSEMBLY BAG-IN-BOX LSLE GAMS GAMS ACCESS GAS CYLINDER EPSP ACCESS EPSP	FILES DROP LEAF DESK/ TERMINAL SWING OUT CHAIR P/L FLIGHT DATA COMPUTER DISC SYSTEM EPSP	SMMI NECK CHAMBER: ELECTRONICS PRESSURE SYSTEM ALBNP CONTROL COP STORAGE CONTAINERS LOW-G CENTRIFUGE EPSP			

CARDIOVASCULAR/METABOLICS

CARDIOPULMONARY

PAY LOAD FLIGHT DATA ANALYSIS

FLUIDS/ MUSCULOSKELETAL

ASTROPHYSICS

Figure 4.3.2.2-7 - HUMAN RESEARCH AREA AND ASTROPHYSICS AREA.

TABLE 4.3.2.2-1 - LOGISTICS MODULE

MAN-SYSTEMS - CREW SUPPORT AND ACCOMMODATIONS EQUIPMENT

SUBSYSTEM	NO. ITEMS	POWER (WATTS)	SIZE LxWxH (FT)	EQUIP TOTAL WEIGHT (LBS)	EQUIP TOTAL VOLUME (CU FT)	C F	CFxWT	COSTING WEIGHT	TOTAL VOLUME (INCL ACCESS) (CU FT)
UTILITIES					69.00				69.00
SEC. STRUCTURE				1873.30		0.20	374.66	1873.30	N/A
HATCHES	1.00		50 IN	N/A	10.00				20.00
LIGHTING		320.00		60.00	7.50	0.20	12.00	60.00	
ECLSS					147.60				147.60
FREEZER (EMPTY)		400.00	3x4x5	450.00	60.00	0.20	90.00	450.00	113.40
REFRIGERATOR (EMPTY)		200.00	2x2x5	250.00	20.00	0.20	50.00	250.00	37.80
STOWAGE CONTAINERS	264		2 CU FT	2640.00	528.00	0.20	528.00	2640.00	997.92
RESUPPLY FOR 90 DAYS	TOTAL*			4251.87	278.09	0.20		N/A	525.59
*ITEMIZED RESUPPLY (ALL GFE)									
CLOTHING			185.40	63.00	0.20				
PERSONAL HYGIENE			216.51	33.25	0.20				
EQPT. CONTAINERS/RESTRAINTS			78.00	4.00	0.20				
HOUSEKEEPING SUPPLIES			142.21	11.00	0.20				
ANCILLARY PROVISIONS			4.20	0.15	0.20				
GALLEY			25.00	1.50	0.20				
FOOD			3512.70	153.09	0.20				
HMF			58.85	7.80	0.20				
MAINTENANCE WORKSTATION			29.00	4.30	0.20				
TOTAL LOGISTICS			9525.17	1120.19				5273.30	1911.31

payload facilities. Secondary structure materials include metal, fiberglass, fabric, and screen devices.

Secondary structure within the launched module will be sufficient to withstand launch and crash loads. Once on orbit, such structure may be modified as required for support of new payloads and interior modification and/or growth. Should the module be returned to the ground, the secondary structural design must withstand landing loads. All secondary structure is designed to permit access to equipment for maintenance considerations.

Secondary structure total weights for each module are 3854.3 lb (Hab Module 1), 4756.0 lb (Hab Module 2), 2937.6 lb (Lab Module 1), 2937.6 lb (Lab Module 2), and 1873.3 lb (Logistics Module). Secondary structure volumes for each module are 79.22 cu ft (Hab Module 1) and 126.42 cu ft (Hab Module 2), respectively. Other volumes are either considered as part of encompassing volumes or are volumes yet to be defined.

4.3.2.3.2 Hatches

Hatches are provided on all operational ports and crew airlocks. They are hinged and stowed such that crew and equipment can traverse through the open ports as required. However, during such times, each may be closed to serve as a pressure bulkhead between modules.

When the radial hatches are opened, it is envisioned that they will move inward toward the center approximately 1 foot, then move back to the end cap and then are restrained. End ports or axial hatches rotate 90° down and are restrained due to operational volume constraints caused by utility line penetration.

Hatch size is defined by maintenance and changeout requirements, including the requirement to transfer a single (e.g., Spacelab-sized) rack from the Orbiter to its use location in the Space Station through a port, passageway, tunnel, etc. Hatches may be larger if design indicates the need. Sufficient volume around the hatch operational envelope is provided to prevent unusual body contortions or major reorientations of crewmembers when traversing through passages (other than turning the body long axis perpendicular to the plane of the hatch opening). Hatches are conceived to be capable of supporting either an open-or-closed hatch mode with manual control requiring a maximum of 30 seconds.

Hatch weight has not been determined. However, the minimum diameter of a typical hatch is 50 inches, and volume (assuming 50 in. dia, 12 in. thick hatch) is 10 cu ft. There are six (four radial and two end hatches) each identified in all modules except the logistics module, which has only two end hatches.

4.3.2.3.3 Mechanisms

Crew mechanisms defined in this paragraph are limited to those which release/lock the hatch to the port. Weights are undefined. However, 362.7 cu ft of volume is allocated, which incorporates the hatch and crew operational envelope.

Other crew-related mechanisms will be defined as Station definition evolves.

4.3.2.3.4 Utilities

All modules have selected utilities passing through them (Figs. 4.3.2.3-1, 4.3.2.3-2 and 4.3.2.3-3). Penetrations around a radial port and the opposite axial port permit entry/exit of the majority of the utilities. Additionally, twin penetration panels on the side bulkhead of each module suffice as other passages for utilities.

Systems which require module penetrations include environmental control (ECLSS), power, thermal, housekeeping data, payloads data, and crew water (drinking, waste, wash). Line definition for the ECLSS includes two 4-in.-dia lines penetrating through the bulkheads, and expanding to 6-in.-dia ducts. Air flow on one line provides supply to the module, while the other line is used for collecting exhaust air. The housekeeping bundle, an optical data network, consists of six 1/2-in.-dia lines accommodating data acquisition/telemetry, thermal data, control and navigation status, audio control and signal data bus, power distribution control bus (25Kw, 200 VAC), and a miscellaneous data bus. Internal utilities entering/exiting through the two bulkhead panels include dual 1-1/2 in.-dia coolant supply and return lines, dual 1-in.-dia lines for drinking water, for waste liquid water, condensate water, and wash water. Also included are dual 3/8-in.-dia O_2 supply and 1/2-in.-dia N_2 supply lines. The thermal system has dual liquid water loops and a partial vapor loop passing through a heat exchange and a heat source. Volume calculations include 7.3 cu ft for ECLS, 35.3 cu ft for the optical data network, 17.6 cu ft for internal utilities, and various volumes for thermal (Hab 1-10.8 cu ft, Hab 2-10.8 cu ft, Lab 1-30.0 cu ft, Lab 2-19.1 cu ft, Logistics module -7.3 cu ft) insulation. Weight calculations is a function of materials selected for lines.

4.3.2.3.5 Lighting

The lighting system provides both interior and exterior illumination based upon appropriate human factors specifications. Consideration has been given to luminaires location, brightness ratio and contrast with the surrounding area, glare, and shadows. Care has been given in locating switches to facilitate convenient operation. Night light route locators and switch illumination are placed in frequently darkened areas. Internal lighting as defined is given in Table 4.3.2.3-1.

4.3.2.3.6 Windows

Windows are located throughout the module cluster to facilitate work and recreational tasks. Each window is sized and located to permit viewing by two or more persons at a time. Window workstations permit, as required, shadowed viewing and restraint for crewmembers and equipment. Specific equipment which will be accommodated at each window workstation

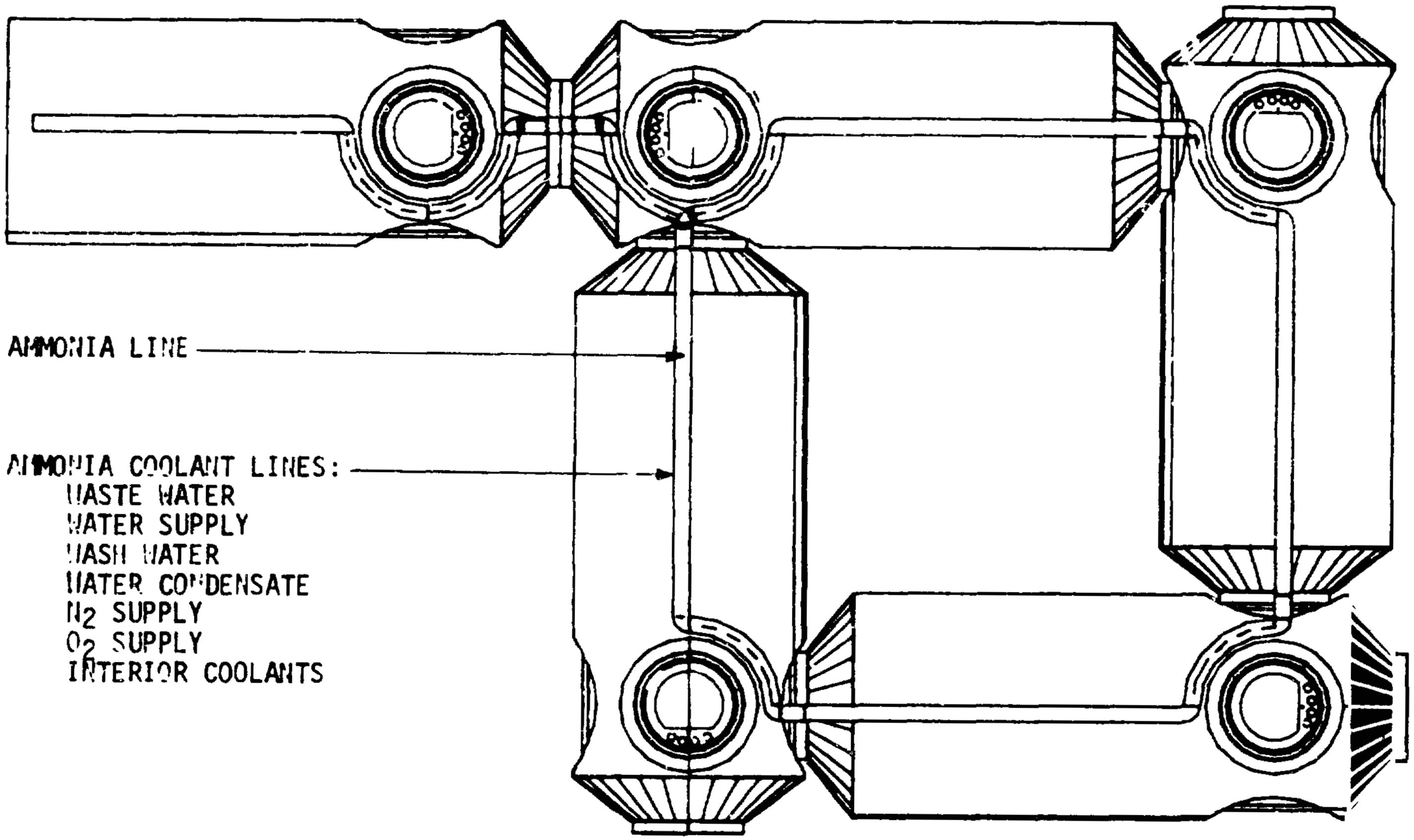


Figure 4.3.2.3-1 - VIEW OF MODULES AND EXTERNAL UTILITY LINES.

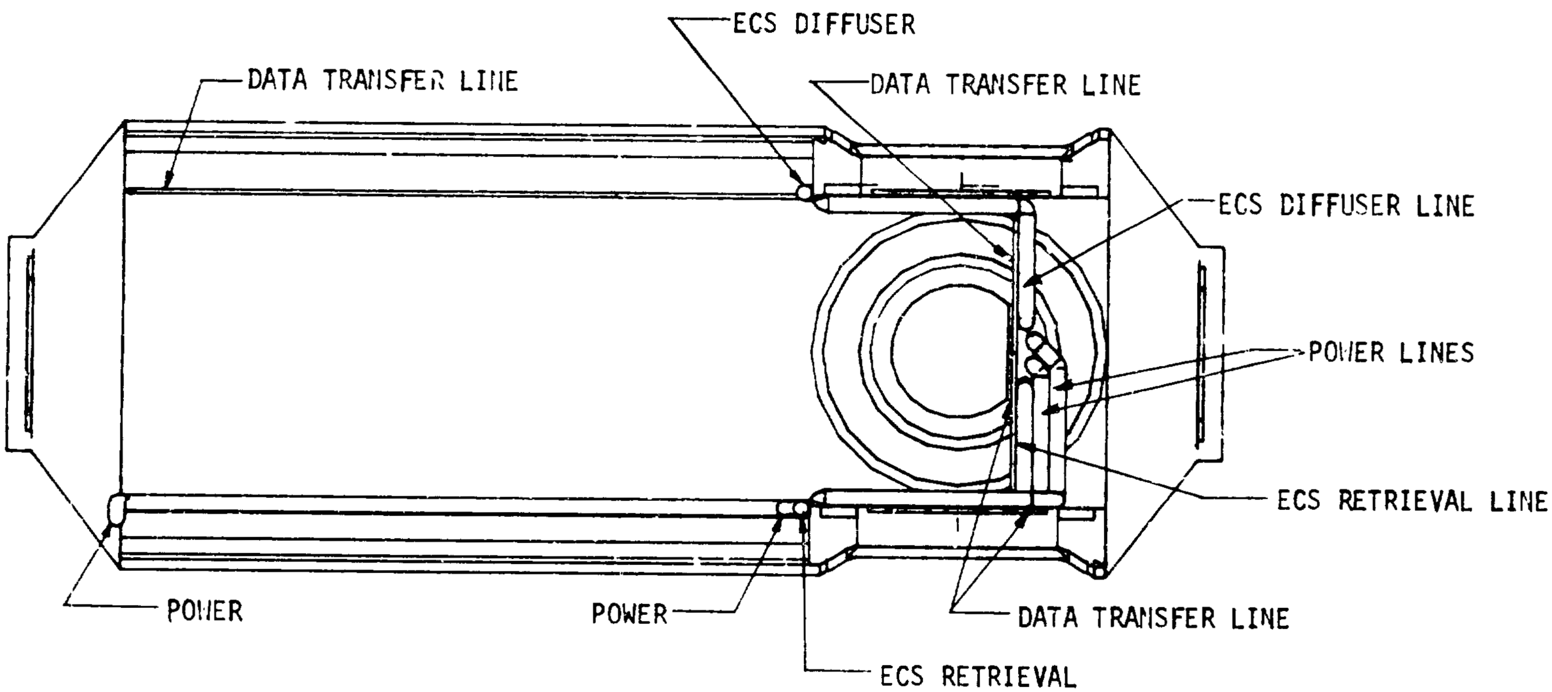


Figure 4.3.2.3-2 - CROSS SECTION VIEW #1 OF MODULE AND INTERIOR UTILITIES.

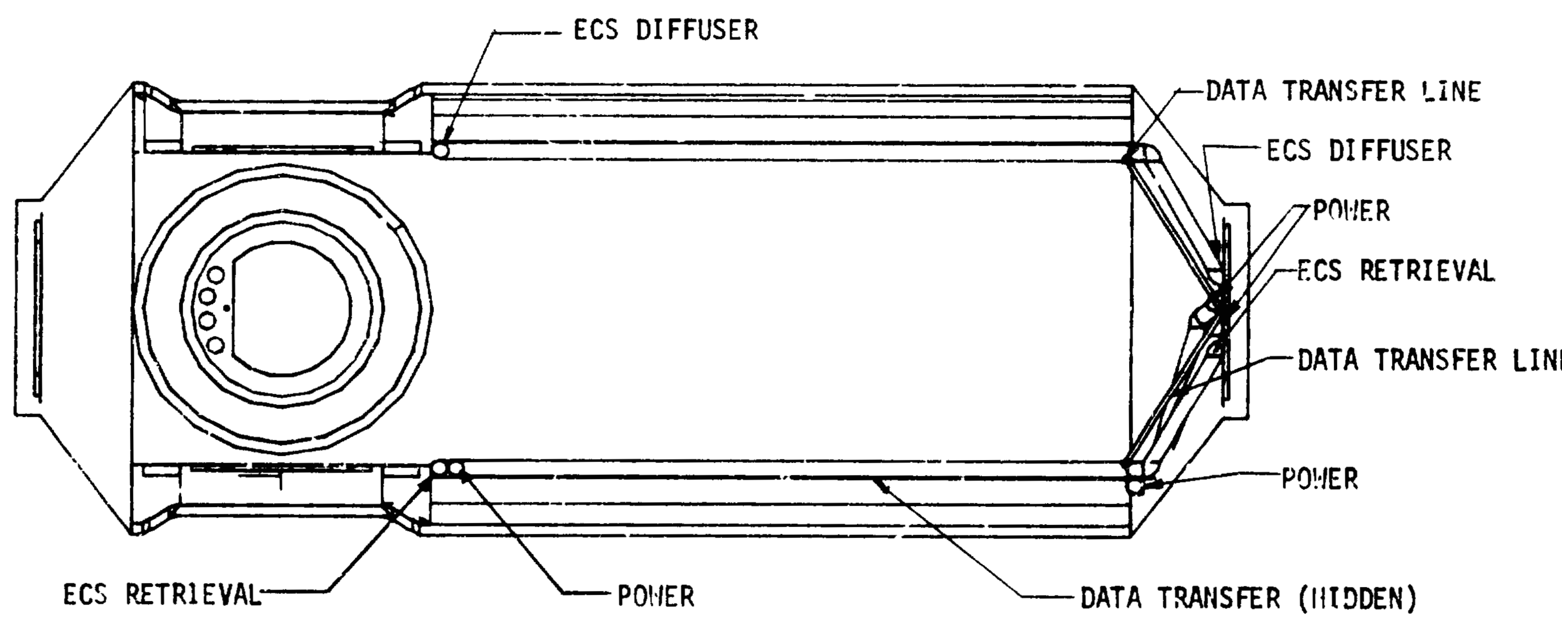


Figure 4.3.2.3-3 - CROSS SECTION VIEW #2 OF MODULE AND INTERIOR UTILITIES.

TABLE 4.3.2.3-1 - INTERNAL LIGHTING

	USAGE RATE	DIMENSION (LxWxH) INCHES	WEIGHT (LB) 90 DAYS	VOLUME (CU FT)	POWER (WATTS)	POWER DUTY CYCLE
<u>LIGHTING</u>	(10 LBS/LIGHT; 1.25 CU FT/LIGHT)		569.99	70.00		
HABITAT 1		TOTAL	150.00	18.75	680.00	
INTERIOR LIGHTS	3.00 REQUIRED/40 WATTS EACH		30.00	3.75	120.00	60%
NIGHT LIGHTS	2.00 REQUIRED/10 WATTS EACH				80.00	100%
PERSONAL HYGIENE FAC.	4.00		40.00	5.00	160.00	40%
GALLEY/WARDROOM	8.00		80.00	10.00	320.00	50%
HABITAT 2		TOTAL	230.00	28.75	840.00	
INTERIOR LIGHTS	3.00 REQUIRED		30.00	3.75	120.00	60%
6 CREW COMPARTMENT	2.00 EACH		120.00	15.00	480.00	40%
PERSONAL HYGIENE FAC.	4.00 EACH		80.00	10.00	160.00	40%
NIGHT LIGHTS	2.00 REQUIRED				80.00	100%
LAB 1		TOTAL	60.00	7.50	320.00	50%
INTERIOR LIGHTS	6.00 REQUIRED		60.00	7.50	240.00	50%
NIGHT LIGHTS	2.00 REQUIRED				80.00	100%
LAB 2		TOTAL	60.00	7.50	320.00	50%
INTERIOR LIGHTS	6.00 REQUIRED		60.00	7.50	240.00	50%
NIGHT LIGHTS	2.00 REQUIRED				80.00	100%
LOGISTICS		TOTAL	60.00	7.50	320.00	
INTERIOR LIGHTS	6.30 REQUIRED		60.00	7.50	240.00	50%
NIGHT LIGHTS	2.00 REQUIRED				80 00	100%

include mounted voice tape recorder, event timer, means to mount cameras, means to secure handheld cameras, small light, method to secure paper and checklists, writing station, body restraints, CRT monitor and keyboard, maps, moving map display with an optical driver to view the flight path, optical maps to identify future flight paths, method of measuring angles and the horizon if appropriate, control of adjacent lighting, and easily deployed hood or curtain to block interior light. Baseline windows are not planned to be of optical quality or to be UV transmissive. However, some techniques will be used to verify that window transparency will be maintained. The current configuration identifies two 20-in.-dia and four 10-in.-dia windows in each of the two habitat modules. Four 10-in.-dia windows are located in the two laboratory modules.

4.3.2.3.7 Maintenance Facilities

The crew will serve as the main capability for accomplishing hardware and software maintenance. This capability will be enhanced via the maintenance workstation. Included in the workstation will be equipment permitting simple diagnosis of circuitry, repair of hardware, and/or replacement of elements of a subsystem. Although the primary maintenance technique will be the changeout of Orbital Replaceable Unit (ORU's), the maintenance workstation will provide supplemental capability to diagnose and repair to the elemental level as appropriate. Additional items may be fabricated from components of failed items or from raw materials. Controlled lighting and magnification capability will be provided. A dedicated microprocessor/screen/printer will serve as a diagnostic tool, supplemented by diagnostics with handheld instruments. Electrical test equipment, maintenance tool kit (powered and manual), and a soldering kit will be available. Support equipment include crew equipment and small item restraints, a particle control system, measurement kit (tape, micrometer, thread gages, etc.), fastener kit (nuts, bolts, washers, etc.), and an adhesive kit (tape, cement, etc.).

A special feature of the workstation will be a portion that is removable and which converts into an EV compatible portable maintenance workstation. This device can be transported, along with attached diagnostic and repair equipment and spares, to any location on the Station requiring maintenance, diagnosis, or repair. It will be capable of being positioned and rigidly restrained at the worksite. Tools, diagnostic and repair equipment, and a computer-enhanced, audio-visual checklist will be easily accessible to the crewmember.

Estimated dimensions for the basi,smaintenances workbench are 19in. by 57 in. by 84 in.) (92.3 cu ft); weight is estimated to be 238 lb.

4.3.2.4 Volumetric Analysis

4.3.2.4.1 Introduction

A comprehensive account of volumetric utilization was undertaken for the reference configuration. Assessments for all systems and subsystems been included to the extent that current definition is available. Table 4.3.2.4-1 shows volumes within the cylinder as they will be discussed in subsequent sections.

4.3.2.4.2 Allocations

4.3.2.4.2.1 End cone volumes

For the purposes of this analysis, end cone volumes are included only to show they are primarily occupied by utilities and hatches. The available end cone volume not occupied by this equipment is considered access volume for hatch operation and utility line servicing and maintenance.

4.3.2.4.2.2 Connecting end volume

A 9-ft long section of the 30 ft cylindrical axis length is utilized for module interconnection where four radial ports are located 90 degrees apart. Because of the complexity of the connecting end volume, the 1257-cu-ft volume contained therein is summarized separately (Table 4.3.2.4-2) from the remaining cylindrical volume (21 ft of the 30 ft cylindrical axis).

Figure 4.3.2.4-1 shows a cut-away view of the protrusion of radial port mechanisms into the connecting end volume. If a rectangular box 8.5 ft by 8.5 ft by 9 ft (650 cu ft, Fig. 4.3.2.4-2) is conceptualized in this end, space along one side (152 cu ft of volume) is available for equipment installation where one radial port is blanked off.

The volume outside the rectangular walls and inside the inner cylindrical surface is occupied by the radial port module connection mechanisms. Remaining volume inside this end "cube" is utilized as hallway access for traverse between modules, and as access volume to the equipment (mentioned previously) installed against the blanked off port. Some residual volume (128 cu ft) is available around the radial port mechanisms (inside the inner cylindrical wall and outside the conceptual cube wall) but does not provide a form factor compatible with conventional packaging. While it is certain this 128-cubic-foot volume will be eventually utilized, for purposes of this analysis, it is treated as nonfunctional.

4.3.2.4.2.3 Habitable volume

The remaining 21 ft of the 30 ft cylindrical axis is considered habitable volume (2932 cu ft). A conventional layout (in terms of floors and ceilings) is utilized in the reference configuration, but the arrangement and layout methodology is not relevant to volumetric accounting, in that equipment volume requirements are basically constant (given that packaging form factors could alter them somewhat). While installation methods and alternative layouts may provide more or less access, maintenance, and operational volume, the percentage gain or loss will not significantly change the fact that volume in HM1 and HM2 is minimally sufficient to inadequate.

TABLE 4.3.2.4-1 - VOLUMETRIC SUMMARY BY LOCATION

	CYLINDRICAL AXIS (LENGTH IN FEET) (INNER DIAMETER = 13 ft 4 in.)	VOLUME (CU FT)
TOTAL VOLUME	30	4189
HABITABLE VOLUME	(21)	2932
CONNECTING END	(9)	1257
END CUBE (VOLUME WITHIN CONNECTING END)		(650)
CONNECTING END LESS END CUBE (9 ft LONG CYL. LESS CUBE)		(607)

TABLE 4.3.2.4-2 - CONNECTING END VOLUME ALLOCATION SUMMARY

(COMMON TO BOTH HM1 AND HM2)

FUNCTION	VOLUME (CU FT)
EQUIPMENT (POWER, SAFE HAVEN, THERMAL, ECLSS (PARTIAL)	152
ACCESS	156
HALLWAY TRAVERSE (OUTSIDE END CUBE)	494
MODULE CONNECT MECHANISMS	327
NOT UTILIZED	128
TOTAL	1257

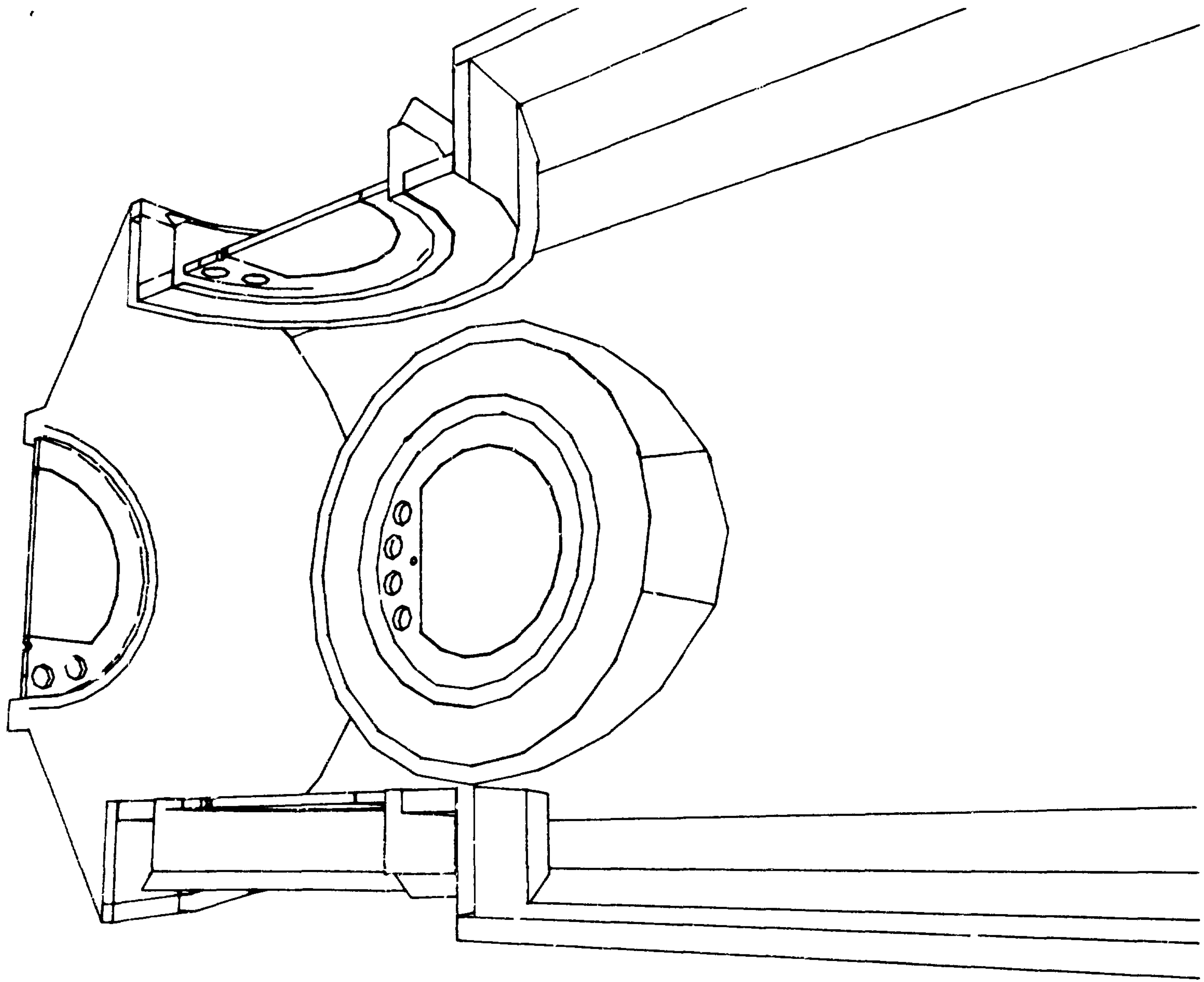


Figure 4.3.2.4-1 - VIEW OF RADIAL HATCHES

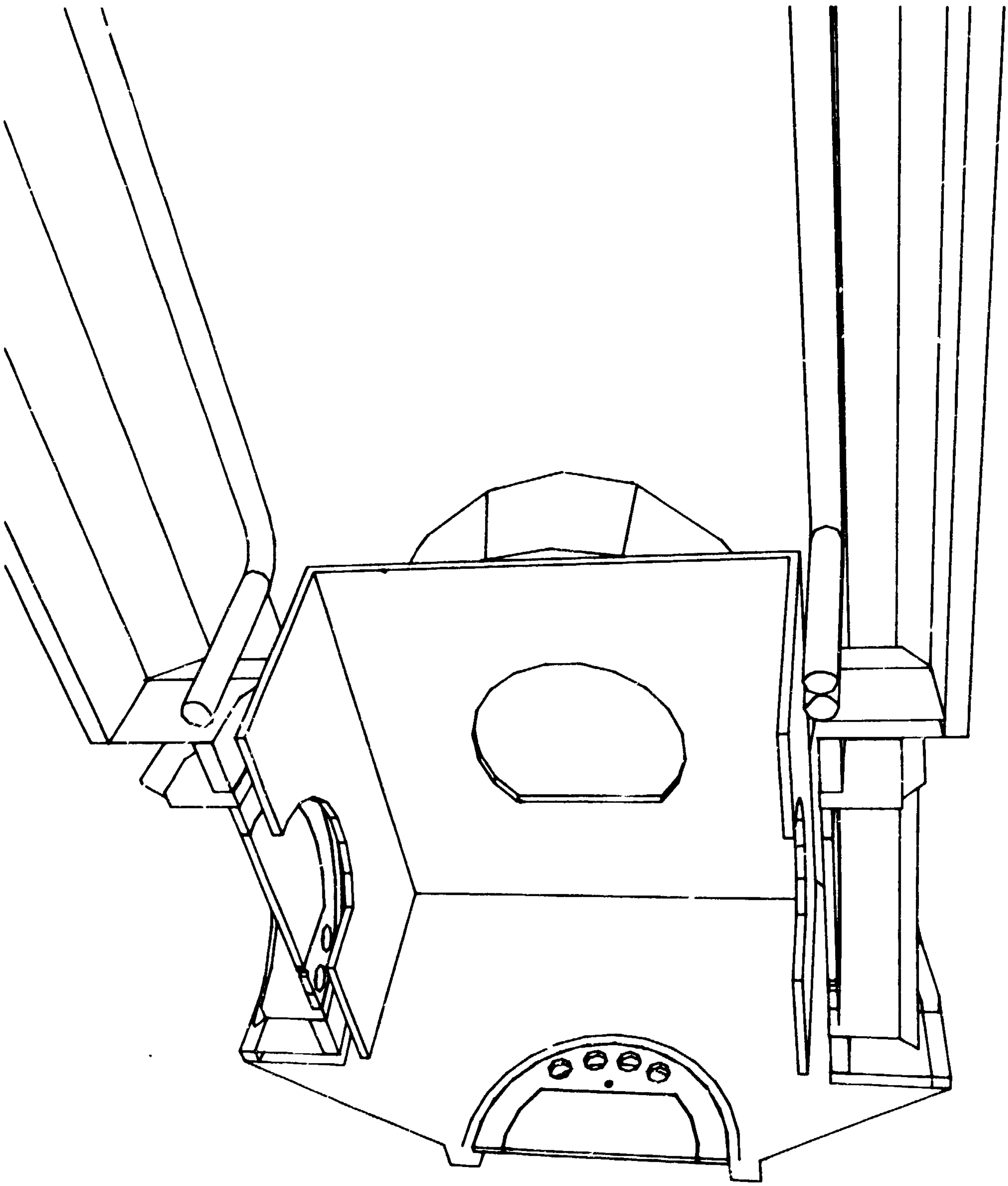


Figure 4.3.2.4-2 - VIEW OF RADIAL HATCH AREA

Table 4.3.2.4-3 and Table 4.3.2.4-4 show equipment and access volume allocations by subsystem for the two hab modules. Access volumes are shown adjacent to equipment volumes and are in addition to equipment volumes.

4.3.2.4.4 Access volume

Access volume requirements vary according to the equipment being utilized. For example, a drawer, when pulled out and removed from its cabinet, requires slightly more than its own original volume plus enough room for someone to stand and perform the removal. An operable rule of thumb for this condition requires 2.7 times the equipment volume for equipment plus access volume. However, access volume can sometimes be shared by equipment on opposite walls sharing a common aisle provided simultaneous functions are not required in the same space. In other cases, a factor greater than 2.7 is required if both vertical and horizontal manipulation are required. In the case of the reference configuration for the Space Station, insufficient detail exists to ascertain whether simultaneous operations will be performed in the lab modules in opposing equipment racks. For the habitation and logistics modules it is generally assumed that this will not be the case, and the same assumption is generalized to the lab modules. Given these assumptions, a factor of 3.0 (or 1.5 for each side) times the equipment rack volume along one side of a module is considered minimally adequate to provide space for an individual to operate or maintain the equipment and to allow passage during operations. This translates to half the depth of the equipment being available for a person to operate in front of the equipment and 1.5 times the equipment depth being available as aisle space for others to pass by with no room left to operate on the opposite wall. Alternatively, two people could operate back to back with no room between them to allow passage.

4.3.2.4.5 Conclusions

The access volume for crew quarters requires that volume is needed for movement in and out of the area, and is satisfied by the 50 in.diameter tunnel common to the six entrances and the 150 cu ft per crew quarters (which includes both equipment and access volume).

The galley/wardroom share a common aisle and are conceptually similar to the crew quarters in terms of volumetric requirements. In addition to providing shared volume for meal preparation, the wardroom provides a common meeting location for recreation, station management meetings, dining, etc. If galley equipment must be removed, the tables can be removed to increase the maintenance access.

Waste management facilities, like the crew quarters, utilize access volume internally, and require only enough room externally for the door to operate.

The respective external access factors of .52 and 1.6 for the opposing volumes of hygiene and medical areas permit removal of the medical racks for maintenance, if required.

TABLE 4.3.2.4-3 - HABITAT MODULE 1 SUBSYSTEM EQUIPMENT AND ACCESS VOLUME ALLOCATIONS

(INSIDE CEILING AND FLOOR)		EQPT (IN CU FT)	ACCESS
WORKSTATIONS		39	62
EQUIPMENT RACKS		32	44
STOWAGE		192	92
CREW	(236)		
SPARES	(48)		
HYGIENE		228	118
GALLEY/WARDROOM		737	458
MEDICAL		88	141
SECONDARY STRUCTURE		79	(COMMON)
	SUBTOTAL	1395	915
	TOTAL	2310	

(OUTSIDE CEILING AND FLOOR)		EQPT	ACCESS
ECLSS		162	(COMMON)
LIGHTING		19	(COMMON)
UTILITIES		73	(COMMON)
STOWAGE (SPARES)		331	37
	SUBTOTAL	585	37
	TOTAL	622	
(CONNECTING END)		1257	N/A
	GRAND TOTAL	4189	

TABLE 4.3.2.4-4- HABITAT MODULE 2 SUBSYSTEM EQUIPMENT AND ACCESS VOLUME ALLOCATIONS

(INSIDE CEILING AND FLOOR)		EQPT (IN CU FT)	ACCESS
WASHER/DRYER		39	67
MAINTENANCE		96	222
DATA MGT/COMM		97	135
CREW QUARTERS (INCL. 120 CU FT STOWAGE)		900	100
HYGIENE		194	110
WORK STATION		20	203
SECONDARY STRUCTURE		127	(COMMON)
	SUBTOTAL	1473	837
	TOTAL	2310	

(OUTSIDE CEILING AND FLOOR)		EQPT	ACCESS
ECLSS		162	(COMMON)
LIGHTING		29	(COMMON)
UTILITIES		73	(COMMON)
STOWAGE		321	37
	SUBTOTAL	585	37
	TOTAL	622	
(CONNECTING END)		1257	N/A
	GRAND TOTAL	4189	

It can be seen from Table 4.3.2.4-3, Table 4.3.2.4-4, and the internal configuration drawings that, while the desirable equipment-to-access volume ratio of 3.0 is not achieved, there is sufficient volume for operational activities to occur. In some cases, operations plus passage by another individual can be accomplished. In all cases, the total of access volume factors for the opposing pieces of equipment across the aisle will permit removal/replacement and maintenance.

Utilizing the packaging factors from HM1 and HM2, it should be possible to realize 30 cubic meters (1059 cu ft) to 45 cubic meters (1589 cu ft) of equipment volume per laboratory module, or 60 cubic meters (2119 cu ft) to 90 cubic meters (3178 cu ft) total.

The volumetric allocations of Labs 1 and 2 in the reference configuration are not discussed in this section, as the functions and allocations are discussed elsewhere. However, the Materials Processing (Lab 2) module is configured with approximately 18 cubic meters (640 cu ft) of equipment volume with the capability to add an approximately equal amount for a total of 35 cubic meters (1235 cu ft) to 40 cubic meters (1412 cu ft) of equipment volume. Equipment volumetric overhead (ECLSS, utilities, workstations, data management, power, thermal, safe haven, communications, secondary structure, lighting, hatches, etc.) to the laboratory modules is approximately 21 percent. Access volume requirements added to this equipment overhead increases the total overhead to approximately 42 percent.

4.3.3 Space Station dynamics and control analysis

Analyses of the rigid body control and orbital altitude maintenance requirements, flexible body control, and flexible body response due to typical disturbances are documented in Sections 4.3.3.1 through 4.3.3.6. The results should be considered preliminary since details of many of the structural and control aspects of the configuration were not available during the analysis cycle.

Sections 4.3.3.1 through 4.3.3.4 deal with input data for the analysis while 4.3.3.5 presents analyses results in an attempt to demonstrate the performance of the station in the natural and induced dynamic environments. A summary of the results is presented in Section 4.3.3.6.

4.3.3.1 Applied Forces and Torques

This subsection contains a description of models which have been used to generate loading events for preliminary analyses of the reference configuration of the Space Station.

Aerodynamic and gravity-gradient models generate forces and torques acting upon the Space Station as a function of orbital position and vehicle attitude which are used for the purposes of control system sizing and configuration trade studies. Orbiter berthing, crew motion, and RCS firings models generate significant loading events which are used for control and response analyses and trade studies.

Due to time limitations and availability of data, the following additional disturbance sources were identified but not included in this study.

- Manipulator Dynamics
- Construction Dynamics
- Manufacturing Dynamics
- Rotating Machinery Dynamics
- Shuttle RCS Plume Impingement
- Crew Motions During Nominal Operation
- Momentum Management Schemes[1]
- Solar Pressure

These loading events are not considered to be an exhaustive list. However, taken together with the ones which were studied, this set is believed to be representative of the set of disturbances which are significant for dynamics and control systems studies.

---

[1]See White Papers "Space Station Fluidic Momentum Controller," and "Space Station Large Area Magnetic Torquer."

4.3.3.1.1 Gravity-gradient torques

Gravity gradient torques act upon any satellite with a mass distribution which results in unequal principal axis inertias.

Two of the principal axis inertias for the reference configuration are approximately equal in magnitude. The minimum axis is half an order of magnitude smaller and the configuration will fly with the minimum axis aligned within a few degrees of nadir.

To compute the gravity-gradient torques, the mass moments of inertia in body axes are calculated from the geometric definition of the reference configuration (see mass properties in Tables 4.3.3.4-1 through 4.3.3.4-4) Assuming a uniform gravity-field, the orientation of the mass moments of inertia in the gravity field is used to calculate the resultant torque acting upon the Space Station.

The detail for the model of gravity-gradient torques is contained in References 3 and 4.

4.3.3.1.2 Aerodynamic drag and torques

The reference configuration utilizes large-area solar arrays to generate power and large-area radiators to dump excess heat energy. These arrays and radiators are controlled with alpha and beta rotary joints to track the Sun. As a result of the solar-tracking, the drag area of the station varies as the orbit is traversed.

The reference configuration which is not geometrically symmetrical about its center of mass and is flown with its x- axis in the orbit plane (see Fig. 4.2.1-1) with the power booms perpendicular to the orbit plane. The aerodynamic torques about the x- and z-axes would be cyclic with respect to inertial space if the atmospheric density was constant and nonrotating. However, since the atmosphere rotates with the Earth and 3 is not constant (Figure 4.3.3.5-11), secular torques occur about these axes. The aerodynamic torque about the y-axis is due to center of mass to center of pressure offset and will result in secular torque if it is not nulled with gravity-gradient torque.

Drag area of the reference configuration is calculated by a solid area projection which includes the effects of orbital position and shading.

Aerodynamic forces are calculated as a function of the area, the local atmospheric density and the center-of-mass velocity with respect to the atmosphere. Center of pressure offset with respect to the center of mass is then used to calculate the aerodynamic torques.

The detail for the model of aerodynamic drag force and torques is contained in References 3 and 4.

4.3.3.1.3 Atmosphere variation model

The nominal altitude range of the Space Station is from 250 to 270 nmi. Control system sizing studies were performed at altitudes of 220, 250 and 270 nmi. At altitudes in the range of 100 to 300 nmi, the atmospheric density depends upon the degree of solar activity and the diurnal effect as well as the altitude. The atmospheric density model is a complex function of Space Station altitude and location relative to the incoming Sun-line.

The nominal altitude range of the Space Station is from 250 to 270 nmi. Control system sizing studies were performed at altitudes of 220, 250 and 270 nmi. At altitudes in the range of 100 to 300 nmi, the atmospheric density depends upon the degree of solar activity and the diurnal effect as well as the altitude. The atmospheric density model is a complex function of Space Station altitude and location relative to the incoming Sun-line.

Recent empirical models of the upper atmosphere density, based on drag data from various satellites (reference 5) and direct mass-spectrometric measurements of upper atmosphere (reference 6) have been included in this analysis.

These atmospheric models reflect the observational evidence for both long and short-term variations in upper atmosphere density, composition, and temperature. Long-term variations are dominated by the 11 year solar cycle (Figure 4.3.3.1-1) related variations in the extreme ultraviolet (EUV) radiation from the Sun and semiannual and seasonal variations. Short-term variations are dominated by the diurnal variation caused by the daylight side heating of the Sun and variations associated with energy inputs related to geomagnetic activity. The statistically predicted solar flux indicators, Figure 4.3.3.1-1, as well as the geomagnetic index, Ap, were obtained from MSFC Mission Analysis Division. The 2 sigma peak F10.7 values of 242 and corresponding 2 sigma Ap value of 22 was chosen for the analysis to be used to size the CMG requirements.

On rare occasions, extreme solar magnetic activity results in short-term (less than 12 hours) atmospheric densities three times as large as the 2 sigma case. Present plans calls for controlling the Station with the RCS during these occasions if necessary.

4.3.3.1.4 Docking/berthing

The following assumptions were used to derive a model for the disturbance to the Space Station control and stabilization system due to berthing of the Shuttle Orbiter vehicle:

- Significant forces along the x berthing axis only
- Station initially at zero rate state

- Orbiter closing at 0.1 ft/sec rate along berthing axis
- Constant force berthing mechanism
- 0.5-ft displacement of berthing mechanism

The berthing disturbance is modeled as a 500-lb force acting along the berthing axis for a period of 1 second. This is considered to be a conservative model for design purposes. Although docking forces experienced on SKYLAB were an order of magnitude higher, it is anticipated that the berthing operation may actually produce forces which are an order of magnitude below the design model level.

The time history for the berthing force model is given in Figure 4.3.3.1-2.

The z-axis moment arms for the reference configuration in body coordinates are 78 ft without payloads and 120 ft with payloads.

4.3.3.1.5 Shuttle launch and landing load factors

Load factors for Shuttle launch and landing are given in Table 4.3.3.1-1. Load factors for intermediate flight events are given in Reference 7. These load factors were specified for preliminary structural design purposes and are included here for completeness, although they were not required or used for dynamics and control analyses.

4.3.3.1.6 Crew motion (kick-off/stopping)

The following assumptions were used to derive models for the disturbance to the Space Station control system due to on-board crew movement:

- Station initially at zero rate state
- Free-flight distances of 30 ft and 10 ft
- 25-lb peak force

The first crew motion model is an end to end traversal of a habitat module. The force generated by the crew member increases linearly to a maximum of 25 pounds over an interval of 1 second and then decreases to zero as wall contact is lost. At the opposing wall, the force increases instantly to 25 lbs as the wall is contacted and decreases linearly to zero over a 1 second interval. The time between force pulses of 13 seconds is calculated from a free-flight distance of 30 feet.

The time history for this crew motion model is given in Figure 4.3.3.1-3.

The force is applied along the x-axis, and the moment arms for the reference configuration are 78 feet without payloads and 120 feet with payloads. When the Orbiter is berthed to the station, the z-axis moment arms become 25 feet without payloads and 59 feet with payloads.

The time history for this crew motion model is given in Figure 4.3.3.1-3.

The force is applied along the x-axis, and the moment arms for the reference configuration are 78 ft without payloads and 120 ft with payloads. When the Orbiter is berthed to the station, the z-axis moment arms become 25 ft without payloads and 59 ft with payloads.

4.3.3.1.7 RCS re-boost firing disturbance

The following assumptions were used to derive a model for the disturbance to the Space Station control and stabilization system during a reboost firing of the RCS thruster.

The Station is initially at zero angular rate state. The RCS control system will off-modulate four jets to hold the attitude of the Station within a $\pm$ 1.0° deadband while applying the desired Delta-V along the flight path. Eventually, the Station will reach a limit condition about a 1.0 degree bias or attitude error. The time history for the RCS firing disturbance model is given in Figure 4.3.3.1-4 for the initial 500 seconds of reboost.

TABLE 4.3.3.1-1 - SHUTTLE LIFT-OFF AND LANDING LOAD FACTORS FOR PRELIMINARY DESIGN

	LIFT-OFF	LANDING (9.6 FT/SEC SINK RATE)	LANDING (6 FT/SEC SINK RATE)
Nx	-3.7 -0.2	±3.0	±2.6
Ny	±1.8	±2.25	±1.2
Nz	±3.5	+6.3 -1.5	+4.5 -0.3
x	±3.7 rad/Sec^2	±4.8	±4.0
y	±7.7	±11.3	±5.4
z	±3.1	±4.7	±3.0

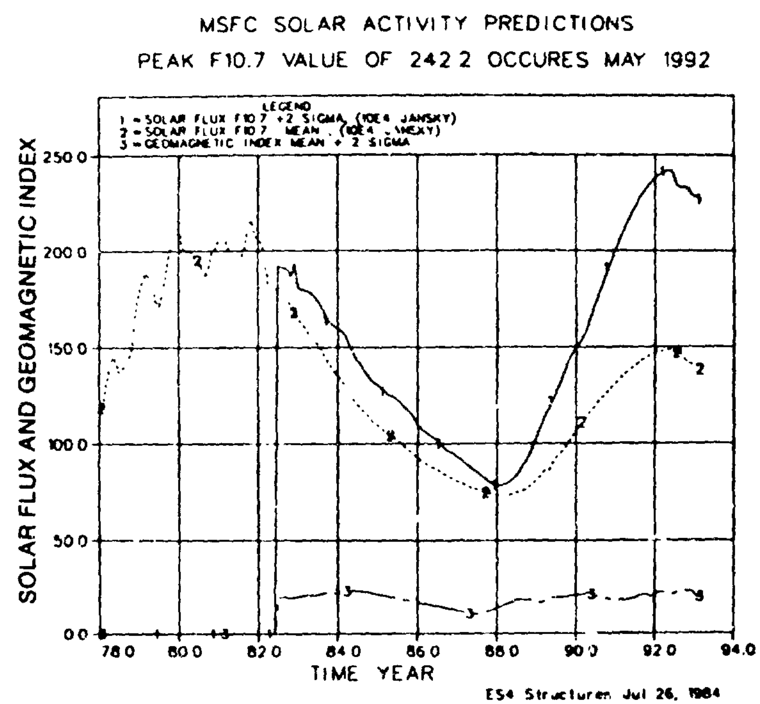


Figure 4.3.3.1-1 - LONG TERM SOLAR FLUX VARIATIONS

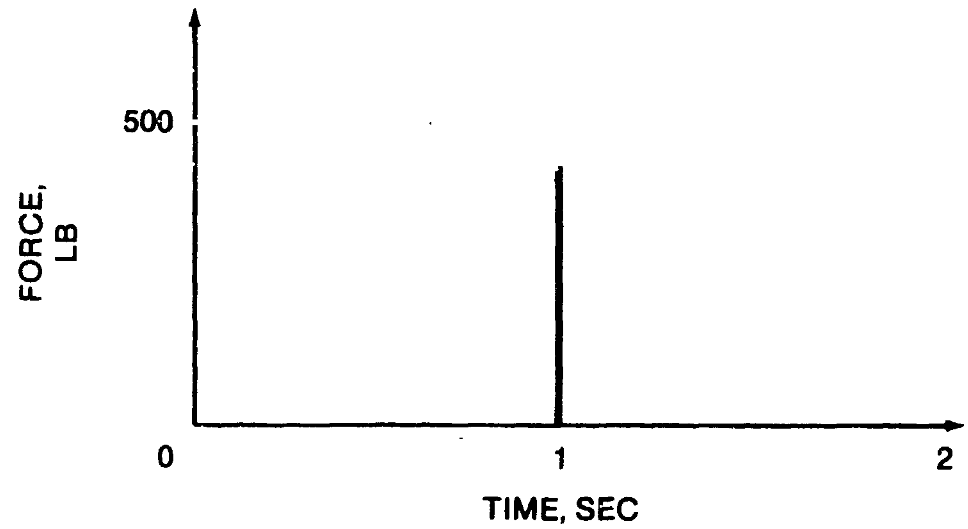


Figure 4.3.3.1-2 - ORBITER BERTHING FORCE

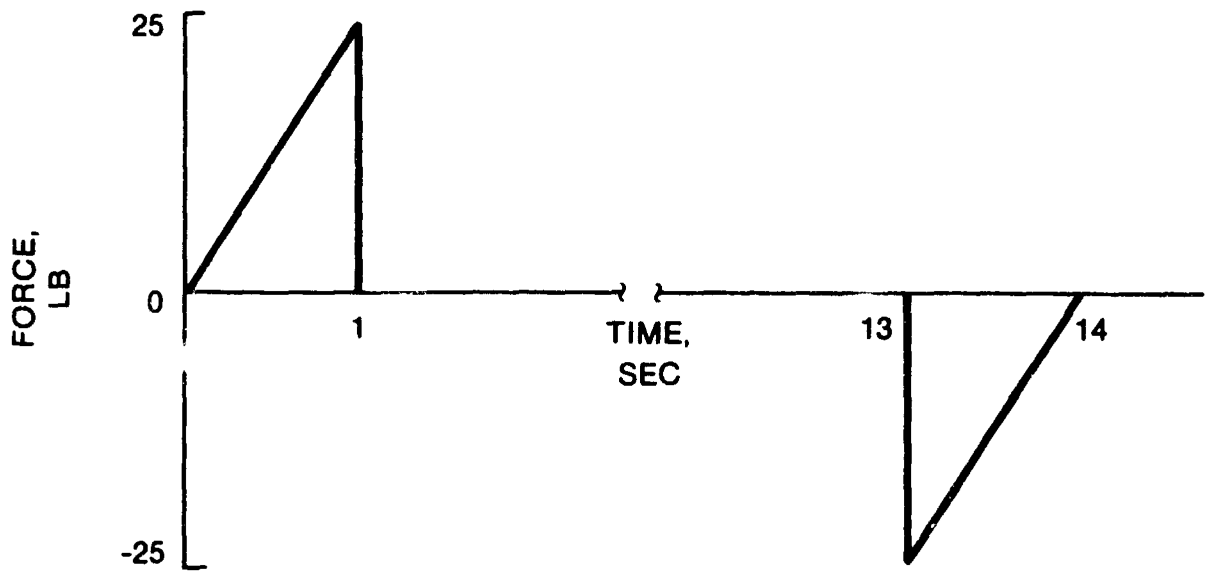


Figure 4.3.3.1-3 - CREW MOTION DISTURBANCE

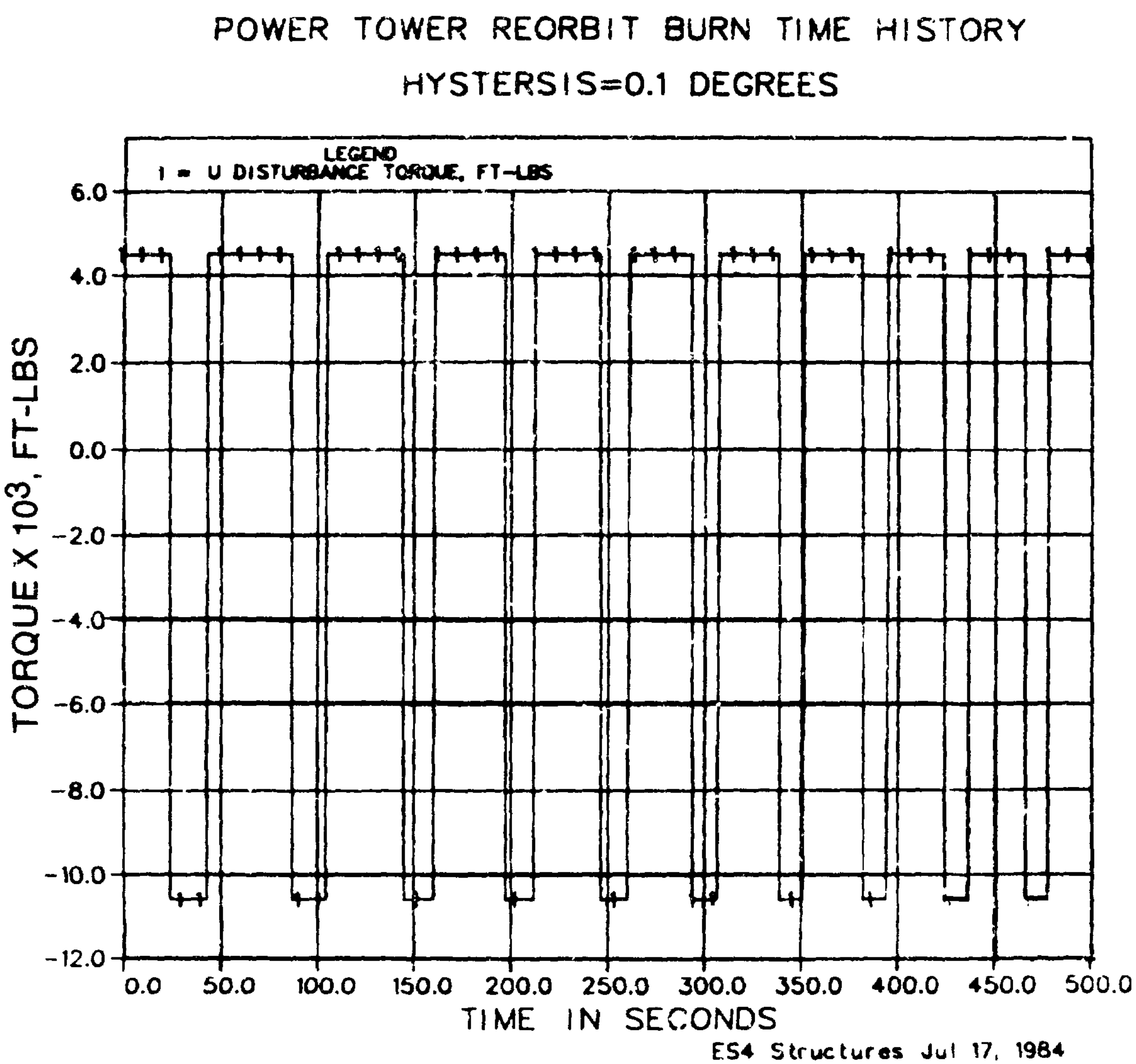


Figure 4.3.3.1-4.- RCS TORQUE HISTORY

4.3.3.2 Control System Definition

4.3.3.2.1 Control moment gyro

A proportional, integral, and differential gain (PID) type control method was chosen for the Space Station attitude control system. This method of control is relatively simple and is well within the state of the art. The integral gain is especially useful in nulling very small pointing errors.

There are several methods for calculating PID gains. However, only one will be discussed here. Figure 4.3.3.2-1 represents the Attitude Control System (ACS) rigid and flexible body dynamics for a single body fixed control axis. The following definitions apply:

T_D = disturbance torque (N.m)

T_{com} = torque applied by control actuators (N.m)

T_R = resultant torque as seen by the vehicle (N.m)

I = vehicle inertia about the body axis ($Kg.m^2$)

K_{bi} = model gain of the i'th flexible body mode (unitless)

ω_i = frequency of the i'th flexible body mode (Rad/s)

ζ_i = damping ration of the i'th flexible body mode (unitless)

G_P = proportional control gain (N-m)

G_I = integral control gain (N-m/s)

G_D = differential control gain (N.m.s)

S = La place operator (1/s)

T_c = commanded torque (N-m)

m = total number of flexible body modes (unitless)

Some simplifying assumptions have been made to aid in analysis:

a. The range of bending mode resonant frequencies was considered to be low enough to allow the sensor and control actuator dynamics to be neglected.

b. The body axes are assumed to coincide with the principal axes.

This assumption decouples the body axes and justifies a planar model.

c. The sample rate was high enough to allow the system to be considered as continuous.

d. The control sensor and control actuators have been colocated.

This final assumption allows the model gain to be written as

$$K_{bi} = \frac{\phi_i^2 \, I}{M g_i}$$

Where ϕ_i is the normalized slope of the i'th bending mode at the sensor location and Mg_i is the corresponding generalized mass.

Temporarily ignoring the flexible body dynamics the rigid body transfer function from disturbance to command can be written. After simplification the transfer function is

$$\frac{T_C}{T_D} = \frac{G_D s^2 + G_p s + G_I}{I s^3 + G_D s^2 + G_p s + G_I}$$

Setting the characteristic equation to zero and dividing by the vehicle inertia gives

$$s^3 + \frac{G_D s^2}{I} + \frac{G_p s}{I} + \frac{G_I}{I} = 0$$

Gain calculations are made by setting the above equation equal to the product of a second order equation dependent only upon the control parameters and a first order equation determined by the integral gain.

Thus

$$s^3 + \frac{G_D s^2}{I} + \frac{G_p s}{I} + \frac{G_I}{I} = (s^2 + 2\zeta_C \omega_C s + \omega_C^2)(s + \omega_I)$$

Where

ω_C = control frequency (rad/sec)
ζ_C = damping ratio (unitless)
ω_I = integral loop frequency (rad/sec)

After multiplying and equating coefficients of like powers, the following relationships can be derived:

$$G_D = I(\omega_I + 2\zeta_c\omega_c)$$

$$G_p = I(\omega_c^2 + 2\zeta_c\omega_c\omega_I)$$

$$G_I = I\omega_c^2\omega_I$$

In practice, it is common to assume the control and integral frequences to be related by:

$$\omega_I = K\omega_c$$

where K is a constant

If this substitution is made, the gain equations become

$$G_D = I\omega_c(2\zeta_c + K)$$

$$G_p = I\omega_c^2(1 + 2K\zeta_c)$$

$$G_I = I\omega_c^3 K$$

Generally the control frequency of the rigid body should be kept sufficiently low so that there is a wide separation with frequencies of other degrees of freedom. For the purpose of this analysis, the first mode elastic body frequency will establish the upper limit on the control frequency of the rigid body. The control system parameters must be chosen such that the transient response to a disturbance does not impose excessive torques on the spacecraft (as would be the case for a very fast responding system) and yet not so "sluggish" that large attitude errors accumulate.

The rigid body control frequency was selected to be one decade below the first elastic body mode. This is sufficiently below the first mode so that strong coupling does not exist and unreasonable settling times do not result. Further, the values of K and ζ_c for each configuration were selected to be 0.1 and 0.707 respectively and the elastic body damping ratio was chosen to be 0.005.

4.3.3.2.2 Reaction control system

The RCS is used primarily to maintain attitude control during reboost maneuvers, but also serves as a backup to the CMG system. The RCS must counteract disturbance torques due to thruster misalignment, center of gravity offset and thrust mismatch between pairs of thrusters used to impart velocity changes during these maneuvers. Since the pitch thrusters are used in pairs for the reboost maneuver, they are off-modulated for attitude control. Figure 4.3.3.2-2 shows a block diagram of the RCS. The following nomenclature is used:

E	=	Attitude Error
K_D	=	Compensator Gain
DB	=	Control System Deadband
h	=	Hysteresis
T_L	=	Control Torque
T_D	=	Disturbance Torque
I	=	Space Station Inertia

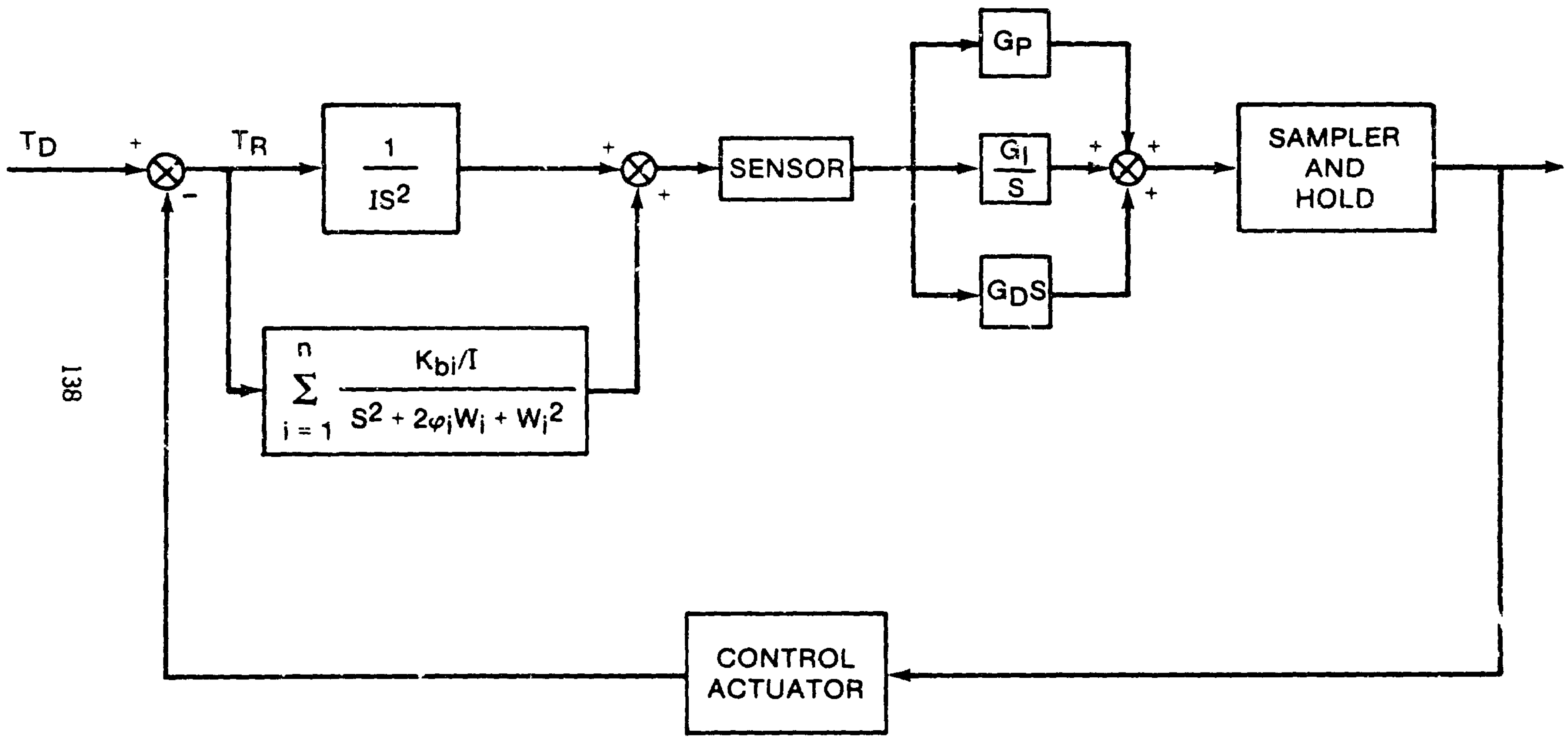


Figure 4.3.3.2-1 ATTITUDE CONTROL SYSTEM

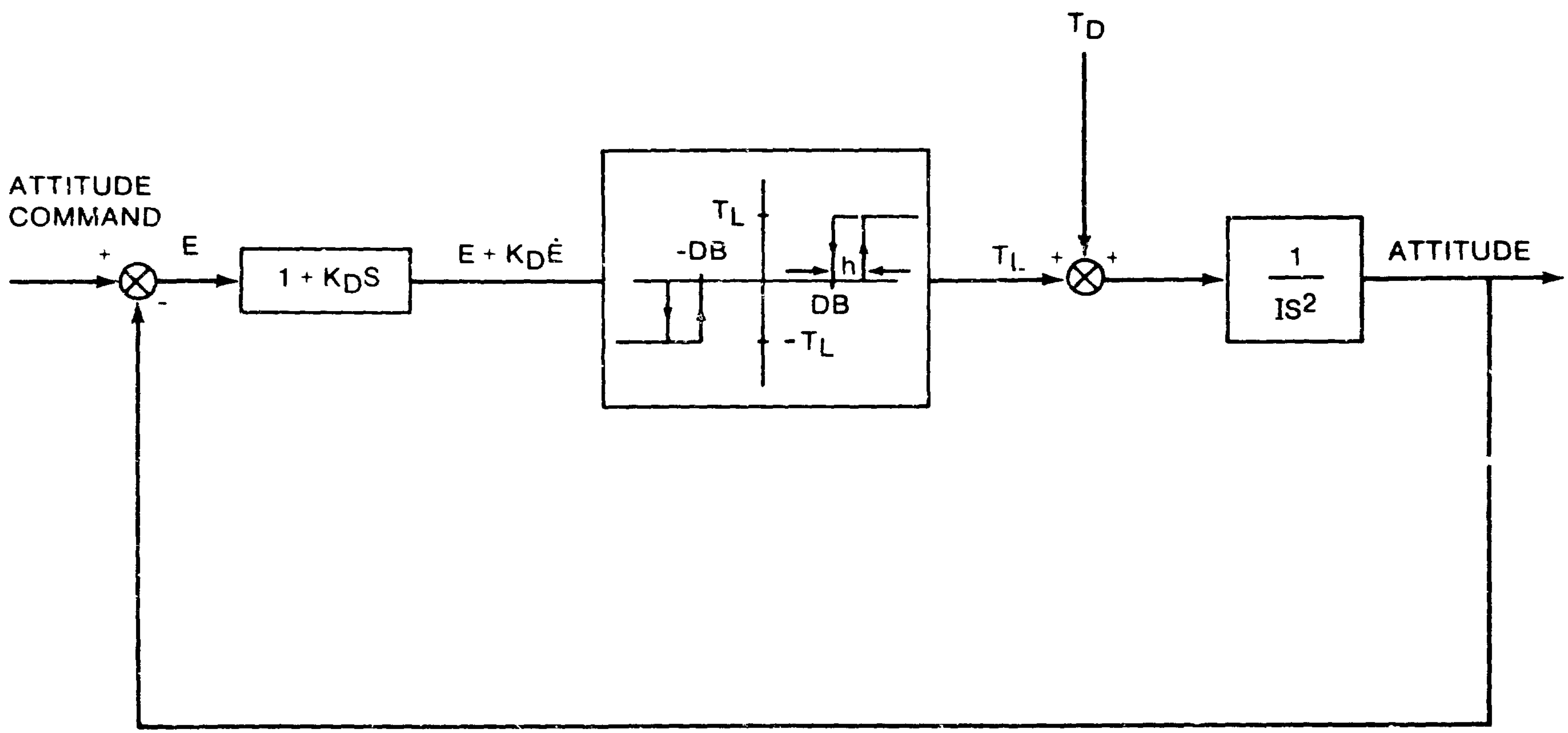


Figure 4.3.3.2-2 - RCS CONTROL SYSTEM

4.3.3.3 Dynamic Model and Free Vibration Characteristics

The purpose of this section is to present a preliminary analysis of the flexible body dynamics of the reference Space Station. Since the Station must accommodate growth, both IOC and growth configurations are addressed. Primary consideration is given to the IOC Station with the results thereof providing a reference for considering other stages of construction and growth. Furthermore, the IOC Station is considered with and without payloads and/or attached orbiter and with the solar arrays oriented either normal to the flight path or normal to the Nadir.

For both the IOC and growth configurations, free-free vibration frequencies and modal shapes are presented. Transient responses due to Orbiter berthing, crew motion, RCS firing and certain MRMS functions are also presented in Section 4.3.3.5 . All analytical results were derived using the MSC NASTRAN structural analysis code of Reference 8.

4.3.3.3.1 Analytical model of IOC station

The IOC Station is shown in Figure 4.3.3.3-1. Nine-foot bay square trusses as discussed in Reference 9 are employed for the lower, transverse and upper booms, keel extension, and lower and upper keels. An analytical finite element model using MSC NASTRAN was developed employing beam type members to produce a continuous beam representation of the discrete member box truss. Considerable literature presently exists for such continuous beam representations (e.g. References 10 and 11). However, for the particular truss construction of the reference configuration, equivalent continuous beam stiffness properties do not exist in the literature. Nevertheless, some similar trusses have been examined in the literature. These stiffness properties take the form,

$$EI = (CEI)\ b^2\ (EA)$$

$$GJ = (CGJ)\ b^2\ (EA)$$

where b is the dimension of the cubic repeating truss bay and EA is the axial stiffness of every truss member. Upper bounds for the coefficients CEI and CGJ may be derived by assuming that the cross section battens are infinitely stiff. These yield values of CEI and CGS of 1.18 and 0.35, respectively.

Recently, through a personal correspondence with the authors of Reference 10, formulas for the stiffness properties of the box truss beam with flexible battens were developed. This led to values of CEI and CGJ of 1.07 and 0. [illegible] respectively. Unless otherwise stated, these latter values were used for the NASTRAN finite element beam model. A summary of properties used for the truss beam representation is provided in Table 4.3.3.3-1.

The continuous beam representation for this truss also revealed some torsional bending coupling as well as some higher order beam effects. For the preliminary nature of this study it was considered appropriate to neglect these effects. They may be included in future studies.

The solar array blankets of the IOC Station were not modeled in detail except for their mass which was distributed along the array mast which supports each array. However, approximate frequencies for the blankets are provided herein. The solar arrays are assumed to be supported by 30-inch diameter double laced continuous longeron coilable masts whose properties are given in Table 4.3.3.3-1. The radiators are assumed to be supported by 2-foot diameter aluminum tubes of 0.1 inch thickness.

All payloads are modeled as lumped masses with appropriate center-of-gravity offsets. The Station modules are treated as rigid beam members rigidly connected to one another in the appropriate race track arrangement. Both payload and module weights are provided in Section 4.3.3.[illegible]. Inertial properties of the analytical model are given in Table 4.3.3.3-2 and the undeformed geometry of the model is shown in Figure 4.3.3.3-2.

It is believed that the rotary alpha and beta joints of the station as well as joint connections in the lattice trusses will involve some degree of nonlinearity. This is an important subject for future investigation, but is beyond the scope of the present study. It is also assumed that the alpha rotation rate of one cycle per 94 minutes is slow enough that the arrays may be considered stationary.

Since the Station is constructed in orbit, vibration modes of the Station during construction should also be addressed, however, this is beyond the scope of the present study.

4.3.3.3.1.1 Solar arrays normal to flight path

Tables 4.3.3.3-3 through 4.3.3.3-6 provide the free vibration frequencies and a brief verbal description of their corresponding mode shape when the arrays are normal to the flight path. In Tables 4.3.3.3-3 through 4.3.3.3-6 the mode shape descriptions are provided for the four cases of IOC Station without attachments, with payloads, with Orbiter and with both payloads and Orbiter. Each table contains the results for all four cases and a brief description of the mode shape associated with one of the four cases whose values are given in the first column of results in each table. Thus the order of the columns for each table is changed so that mode shape descriptions are provided for all four cases and a glance across the columns of any of these tables indicates changes in frequency as attachments are added or subtracted.

Selected mode shape deformations of the IOC Station are presented in Figures 4.3.3.3-3 through 4.3.3.3-7 for the Station without attached Orbiter and payloads; in Figures 4.3.3.3-8 through 4.3.3.3-12 for the Station with attached payloads only; in Figures 4.3.3.3-13 through 4.3.3.3-16 for the Station with attached Orbiter only; and in Figures 4.3.3.3-17 through 4.3.3.3-20 for the Station with both payloads and Orbiter attached.

It is not always obvious how to characterize individual modes. However, in Tables 4.3.3.3-3 through 4.3.3.3-6 an attempt is made to characterize the mode shapes using symbols defined in the table. Further, it is attempted to reflect the relative contribution of different parts of the Station to each mode shape by the order in which the symbols are given. In general, however, mode shapes generally fall into the following three categories:

(a) Array mast dominated modes
(b) Primary structure (keel deformation) dominated modes
(c) Radiator mast dominated modes

It is evident from Tables 4.3.3.3-3 through 4.3.3.3-6 that the station has many closely spaced modes partially due to the repetitive nature of its construction, (e.g.,eight solar arrays). Modes which involve the array masts by themselves have frequencies near 0.16 Hz. (See Figures 4.3.3.3-4, 4.3.3.3-9, 4.3.3.3-14 and 4.3.3.3-18 as well as Tables 4.3.3.3-3 through 4.3.3.3-6.) These modes are effectively the cantilever modes of the individual masts, (with the solar blanket mass distributed over their length), and as such are insensitive to the presence of payloads and/or Orbiter.

Below the cantilever mast modes are the lowest modes of the Station. In general, the lowest station modes involve coupling of array mast bending and keel twist or bending, with the radiators and upper and lower booms moving as rigid bodies. (See Figures 4.3.3.3-3, 4.3.3.3-8, 4.3.3.3-13 and 4.3.3.3-17.) Depending on attachments, these modes occur at frequencies of 0.096 to 0.15 Hz. A stiffer keel would raise these frequencies, but never higher than the array mast cantilever frequency of about 0.15 Hz. As shown in Tables 4.3.3.3.3-3 through 4.3.3.3-6, when payloads and/or Orbiter are attached to the Station, these frequencies decrease and keel deformation becomes more pronounced because the keel is dynamically softened due to the presence of added mass. Consequently, when these modes have frequencies close to 0.16 they are denoted as array mast dominated modes and otherwise as primary structure dominated modes.

Above the cantilever array mast modes are additional coupled modes. In the absence of attachments there are two modes involving coupling of array mast bending and rigid body keel pitch. (See Figure 4.3.3.3-5.) These modes resemble those occurring in a free-free beam having a heavy mass at its center and as such are slightly higher than the cantilever modes. When payloads and/or Orbiter are attacned the frequency of these modes decreases slightly, (they can never go below the cantilever modes), and become indistinguishable in mode shape from the cantilever modes and are hence listed as such in the Tables as array mast dominated modes.

Above these array dominated modes are those dominated by deformation of primary structure; that is, keel twist, keel roll bending, and keel pitch bending. (See Figs. 4.3.3.3-6, 4.3.3.3-11, 4.3.3.3-15, and 4.3.3.3-19.) Primary structure dominated modes start at a frequency of 0.2094 Hz in the case of no attachments and as low as .197 Hz when both payloads and Orbiter are attached. Bending of the transverse boom occurs at a frequency of about 0.526 Hz in the absence of attachments, at 0.486 Hz when payloads only are attached, at 0.509 Hz when Orbiter alone is attached and at 0.475 Hz when both payloads and Orbiter are attached. Thus the transvese bending mode is nearly insensitive to attachments.

Finally, the lowest radiator mast dominated mode occurs at a frequency of 0.572 Hz in the absences of attachments. (See Figures 4.3.3.3-7, 4.3.3.3-12, 4.3.3.3-16 and 4.3.3.3-20.) The frequency of these modes change little when attachments are added to the Station.

4.3.3.3.1.2 Solar arrays normal to the Nadir

Vibration modes for the case of the solar arrays oriented normal to the Nadir are quite similar to those for solar arrays oriented normal to the flight path and hence are not shown in this document. Nevertheless, transient response results for array oriented normal to the Nadir are given in section 4.3.3.5.

4.3.3.3.1.3 Solar array modes

As mentioned earlier, the NASTRAN finite element model does not allow for solar array blanket modes inasmuch as the arrays are only taken into account as nonstructural mass distributed along the array support mast. In order to produce a first approximation to the array blanket modes it is assumed that each blanket is attached to a mast near its root and at its end, and that the array behaves as a membrane grounded at these locations. As a consequence, the lowest blanket frequencies are given approximately by the formula,

frequency in Hz = (1/2)[square root (P/L/M)]

Where P is the blanket tension, L is the blanket length, and M is the blanket mass. For the IOC station, the blanket length is 80 feet and weighs 1200 pounds. Tension in the blanket must be reacted by compression in the array mast. It is assumed that P is 50 percent of the mast axial strength. For the 30-inch diameter array mast, the axial compressive strength is about 2000 pounds. Then, a first approximation to the lowest blanket frequency would be 0.29 Hz. In addition, the axial compression would reduce the array mast frequency from about 0.16 to 0.11 Hz. Presence of the axial compression will also reduce the bending strength of the mast. This is an important consideration since it is shown in Section 4.3.3.5, Tables 4.3.3.5-11 and 4.3.3.5-16, that the array mast root has the smallest margin of safety on the station.

4.3.3.3.2 Growth configuration

The growth configuration is depicted in Figure 4.3.3.3-21 and the associated analytical model is depicted in Figure 4.3.3.3-22. The growth configuration differs from the IOC Configuration in size, number of modules, and the use of a solar dynamic system for power generation rather than a solar array based system. A single bay upper keel is used in this study and the radiator mast is assumed to be an aluminum tube of 10.5-inch diameter with 0.10-inch wall thickness.

Growth Station modes are similar in character to IOC modes and fall into the following general categories:

a. keel twist
b. coupled keel roll bending and keel twist
c. keel pitch bending
d. array mast modes
e. radiator mast modes

Selected mode shapes for each of these categories are displayed in Figures 4.3.3.3-23 through 4.3.3.3-27 for the Growth Station without attachments and for the dynamic solar collectors oriented normal to the flight path. Each figure shows a front and side view of the mode shape. The lowest mode involves keel twist and occurs at a frequency of 0.108 Hz which is about 8.8 percent lower than the IOC Station in a similar condition.

4.3.3.3.3 Concluding remarks

A finite element dynamics model and free vibration characteristics of the reference gravity-gradient Space Station have been presented. Vibration modes for both the IOC and the growth configurations have been presented for the the following four cases: without attachments, with payloads attached, with Orbiter attached, and with both payloads and Orbiter attached.

For the IOC Configuration, modes were characterized into three basic types, modes dominated by array mast deformation, modes dominated by primary structure (keel) deformation, and modes dominated by radiator mast deformations.

Cantilevered array mast modes occur at about 0.16 Hz. Modes of the solar array blankets were not included in the finite element analysis but the lowest frequency of these modes was approximated as being 0.29 Hz. Modes with somewhat lower frequencies involve coupled array mast and primary structure (keel) deformation. They may be either array mast dominated or primary structure dominated. Stiffening the keel would increase the frequencies of these modes, but never above the frequency of the cantilever array mast modes. Modes with frequencies sligthly higher than the array mast cantilevered frequencies are also array mast dominated, but involve some primary structural motion. Increasing the mass of the primary structure would tend to lower these frequencies, but never below the cantilevered array mast frequency. Finally, in the absence of Station attachments, modes having a frequency of about 0.21 Hz and higher are dominated by primary structural deformation.

Growth configuration frequencies were also provided, but detailed characterization was not given. The lowest growth configuration frequency was 0.108 Hz in the absence of attachments. This is only 8.8 percent lower than the lowest IOC configuration frequency in the absence of attachments. Consequently, it appeared that the Growth Station would not respond radically different than the IOC Station.

TABLE 4.3.3.3-1 - PROPERTIES OF ANALYTICAL REFERENCE CONFIGURATION SPACE STATION MODEL

COMPONENT	BENDING STIFFNESS (FT-LBS-FT)	TORSIONAL STIFFNESS (FT-LBS-FT)	MASS LENGTH (SLUGS/FT)	BENDING STRENGTH (FT-LBS)	TORSIONAL STRENGTH (FT-LBS)
BOOMS & KEELS	1.31 E+9	3.18 E+8	0.25	35,000	15,000
30-INCH ASTRO MAST	3.13 E+6	2.08 E+5	2.3	3,480	208

TABLE 4.3.3.3-2 - INERTIAL PROPERTIES OF ANALYTICAL MODEL

CASE	WEIGHT (LBS)	C. G. COORDINATES (FT)	MOMENTS OF INERTIA ($LB\text{-}FT\text{-}SEC^2$)
WITHOUT PAYLOADS AND ORBITER	269.000	(1.1,0,84.2)	(8.63 E+7, 7.81 E+7, 1.20 E+7)
WITH PAYLOADS ONLY	373,200	(-1.1,0,128.2)	(2.06 E+8, 1.98 E+8, 1.45 E+7)
WITH ORBITER ONLY	508,800	(5.7,0,34.4)	(1.45 E+8, 2.37 E+8, 1.35 E+7)
WITH PAYLOADS AND ORBITER	608,600	(3.59,0,68.3)	(3.21 E+8, 3.14 E+8, 1.62 E+7)

TABLE 4.3.3.3-3 IOC REFERENCE CONFIGURATION FREQUENCIES IN Hz WITH MODE SHAPE DESCRIPTIONS FOR THE CASE WITHOUT PAYLOADS AND ORBITER ATTACHED

MODE SHAPE DESCRIPTION*	WITHOUT PAYLOADS AND ORBITER	WITH PAYLOADS ONLY	WITH ORBITER ONLY	WITH PAYLOADS AND ORBITER
AM/KRB/KT/RRR/TBRR/MRR/LBRR	0.138	0.123	0.120	0.0963
AM/KPB	0.148	0.130	0.135	0.102
AM/KPB	0.1502	0.132	0.138	0.1139
	0.1525	0.1540	0.153	0.1515
	0.161	0.1544	0.158	0.153
	0.1630	0.161	0.1613	0.1613
	0.1636	0.162	0.1632	0.1626
	0.1642	0.16310	0.16325	0.1628
AM	0.16426	0.16315	0.1643	0.1631
	0.1643	0.1633	0.16435	0.16315
	0.1644	0.1642	0.1644	0.1638
	0.1646	0.16439	0.1646	0.1642
	0.1648	0.16445	0.1647	0.1645
	0.16485	0.1648	0.16475	0.1647
AM/KRP	0.1752	0.16485	0.1648	0.16475
AM/KRP	0.1770	0.16486	0.172	0.1648
KT/KRB/AM/RRR/TBRR/LBRR	0.2094	0.173	0.1840	0.16485
KT/KRB/AM/TBRR/RRR/LBRR	0.2126	0.178	0.188	0.1695
KPB/AM/RM	0.2640	0.198	0.209	0.1970
KRB/AM/RM/LBRR	0.5218	0.299	0.403	0.2627
TB/AM/RM	0.526	0.319	0.501	0.2983
RM/KPB	0.572	0.320	0.565	0.3168
RM/LBRR	0.587	0.430	0.568	0.3713
RM/TB	0.592	0.475	0.588	0.4280
KPB/RM/TB/AM	0.675	0.486	0.589	0.4745
KPB/AM	0.720	0.570	0.709	0.5572
KPB/AM	0.879	0.579	0.873	0.5671
AM	0.901	0.590	0.901	0.5874
KPB	0.921	0.624	0.918	0.5885
AM/KRB	0.948	0.707	0.948	0.6889

KEY:

AM - ARRAY MAST BENDING
LB - LOWER BOOM BENDING
KT - KEEL TWIST
KRB - KEEL ROLL BENDING
RRR - RADIATOR RIGID BODY ROLL
LBRR - LOWER BOOM RIGID BODY ROLL
RM - RADIATOR MAST BENDING
TB - TRANSVERSE BOOM BENDING
KPB - KEEL PITCH BENDING
KRP - KEEL RIGID BODY PITCH
MRR - MODULE RIGID BODY ROLL
TBRR - TRANSVERSE BOOM RIGID BODY ROLL

* MODE SHAPE DESCRIPTIONS CORRESPOND TO FIRST COLUMN OF VALUES ONLY

TABLE 4.3.3.3-4 IOC REFERENCE CONFIGURATION FREQUENCIES IN Hz WITH MODE SHAPE DESCRIPTIONS FOR THE CASE OF PAYLOADS ATTACHED

MODE SHAPE DESCRIPTION*	WITH PAYLOADS ONLY	WITHOUT PAYLOADS AND ORBITER	WITH ORBITER ONLY	WITH PAYLOADS AND ORBITER
KT/AM/TBRR/RRR/LBRR	0.123	0.138	0.120	0.0965
AM/KT/KRB/RRR/MRR/LBRR	0.130	0.148	0.135	0.1020
AM/KPB	0.132	0.1502	0.138	0.1139
AM/KPB	0.1542	0.1525	0.153	0.1515
	0.1544	0.161	0.158	0.1530
	0.1613	0.1630	0.1613	0.1613
	0.1623	0.1636	0.1632	0.1626
	0.1631	0.1642	0.16325	0.16285
	0.16315	0.16426	0.1643	0.1631
AM	0.1633	0.1643	0.16435	0.16315
	0.1642	0.1644	0.1644	0.1638
	0.1644	0.1646	0.1646	0.1642
	0.16445	0.1648	0.1647	0.1645
	0.1648	0.16486	0.16475	0.1647
	0.16485	0.1752	0.1648	0.16475
	0.16486	0.1770	0.172	0.1648
AM/KPB	0.173	0.2094	0.1840	0.16485
AM/KRB/RRR/LBRR	0.178	0.2126	0.188	0.1695
KT/AM/RRR/LBRR	0.198	0.2640	0.209	0.1970
KT/AM/RRR/LBRR	0.299	0.5218	0.403	0.2627
KT/RKB/AM/RRR	0.319	0.526	0.501	0.2983
KPB/AM/RM/MRP	0.320	0.572	0.565	0.3168
AM	0.430	0.587	0.568	0.3713
KT/KRB/RRR/LBRR	0.475	0.592	0.588	0.4280
TB/KPB/RM	0.486	0.675	0.589	0.4745
RM	0.570	0.720	0.709	0.5572
RM	0.579	0.879	0.873	0.5671
RM	0.590	0.901	0.901	0.5874
RM/KRB/KT	0.624	0.921	0.918	0.5885
KPB/AM/RM	0.707	0.948	0.948	0.6889

KEY:

AM - ARRAY MAST BENDING
LB - LOWER BOOM BENDING
KT - KEEL TWIST
KRB - KEEL ROLL BENDING
RRR - RADIATOR RIGID BODY ROLL
LBRR - LOWER BOOM RIGID BODY ROLL
RM - RADIATOR MAST BENDING
TB - TRANSVERSE BOOM BENDING
KPB - KEEL PITCH BENDING
KRP - KEEL RIGID BODY PITCH
MRR - MODULE RIGID BODY ROLL
TBRR - TRANSVERSE BOOM RIGID BODY ROLL

* MODE SHAPE DESCRIPTIONS CORRESPOND TO FIRST COLUMN OF VALUES ONLY

TABLE 4.3.3.3-5 IOC REFERENCE CONFIGURATION FREQUENCIES IN Hz WITH MODE SHAPE DESCRIPTIONS FOR THE CASE OF ORBITER ATTACHED

MODE SHAPE DESCRIPTION*	WITH ORBITER ONLY	WITH PAYLOADS ONLY	WITHOUT PAYLOADS AND ORBITER	WITH PAYLOADS AND ORBITER
KT/AM/TBRR/RRR/LBRR	0.120	0.123	0.138	0.0963
AM/KPB	0.135	0.130	0.148	0.1020
AM/KRB/RRR/LBRR	0.138	0.132	0.1502	0.1139
AM	0.153	0.1542	0.1525	0.1515
AM/KPB	0.158	0.1544	0.161	0.1530
	0.1613	0.1613	0.1630	0.1613
	0.1632	0.1623	0.1636	0.1626
	0.16325	0.1631	0.1642	0.16285
	0.1643	0.16315	0.16426	0.1631
AM	0.16435	0.1633	0.1643	0.16315
	0.1644	0.1642	0.1644	0.1638
	0.1646	0.1644	0.1646	0.1642
	0.1647	0.16445	0.1648	0.1645
	0.16475	0.1648	0.16486	0.1647
	0.1648	0.16485	0.1752	0.16475
AM/KRB/RRR/LBRR/MRR	0.172	0.16486	0.1770	0.1648
AM/KRB/RRR/LBRR/MRRR	0.1840	0.173	0.2094	0.16485
KPB/AM	0.188	0.178	0.2126	0.1695
KT/AM/RRR/LBRR/MRR	0.209	0.198	0.2640	0.1970
KRB/AM/RRR/LBRR/MRR	0.403	0.299	0.5218	0.2627
TB/AM/RM	0.501	0.319	0.526	0.2983
RM	0.565	0.320	0.572	0.3168
RM	0.568	0.430	0.587	0.3713
RM	0.588	0.475	0.592	0.4280
RM	0.589	0.486	0.675	0.4745
KPB/TB/AM	0.709	0.570	0.720	0.5572
KPB/TB/AM	0.873	0.579	0.879	0.5671
AM	0.901	0.590	0.901	0.5874
KPB/AM	0.918	0.624	0.921	0.5885
AM	0.948	0.707	0.948	0.6889

KEY:

AM - ARRAY MAST BENDING
LB - LOWER BOOM BENDING
KT - KEEL TWIST
KRB - KEEL ROLL BENDING
RRR - RADIATOR RIGID BODY ROLL
LBRR - LOWER BOOM RIGID BODY ROLL
RM - RADIATOR MAST BENDING
TB - TRANSVERSE BOOM BENDING
KPB - KEEL PITCH BENDING
KRP - KEEL RIGID BODY PITCH
MRR - MODULE RIGID BODY ROLL
TBRR - TRANSVERSE BOOM RIGID BODY ROLL

* MODE SHAPE DESCRIPTIONS CORRESPOND TO FIRST COLUMN OF VALUES ONLY

TABLE 4.3.3.3-6 IOC REFERENCE CONFIGURATION FREQUENCIES IN Hz WITH MODE SHAPE DESCRIPTIONS FOR THE CASE OF ORBITER ATTACHED

MODE SHAPE DESCRIPTION*	WITH PAYLOADS AND ORBITER	WITH ORBITER ONLY	WITH PAYLOADS ONLY	WITHOUT PAYLOADS AND ORBITER
KPB/AM	0.0963	0.120	0.123	0.138
KPB/AM	0.102	0.135	0.130	0.148
KRB/KT/AM/RRR/LBRR	0.1139	0.138	0.132	0.1502
	0.1515	0.153	0.1542	0.1525
	0.153	0.158	0.1544	0.161
	0.1613	0.1613	0.1613	0.1630
	0.1626	0.1632	0.1623	0.1636
	0.1628	0.16325	0.1631	0.1642
	0.1631	0.1643	0.16315	0.16426
AM	0.16315	0.16435	0.1633	0.1643
	0.1638	0.1644	0.1642	0.1644
	0.1642	0.1646	0.1644	0.1646
	0.1645	0.1647	0.16445	0.1648
	0.1647	0.16475	0.1648	0.16486
	0.16475	0.1648	0.16485	0.1752
	0.1648	0.172	0.16486	0.1770
AM/KRB	0.16485	0.1840	0.173	0.2094
AM/KRB	0.1695	0.188	0.178	0.2126
KT/AM/RRR/LBRR	0.1970	0.209	0.198	0.2640
KPB/AM	0.2627	0.403	0.299	0.5218
KT/AM	0.2983	0.501	0.319	0.526
KRB/AM	0.3169	0.565	0.320	0.572
KRB/AM/LBRR/RRR	0.3713	0.568	0.430	0.587
AM/TB	0.4280	0.588	0.475	0.592
TB	0.4745	0.589	0.486	0.675
RM	0.5572	0.709	0.570	0.720
RM	0.5671	0.873	0.579	0.879
RM	0.5874	0.901	0.590	0.901
RM	0.5885	0.918	0.624	0.921
TB/TPB/AM/RM	0.6889	0.948	0.707	0.948

KEY:

AM - ARRAY MAST BENDING
LB - LOWER BOOM BENDING
KT - KEEL TWIST
KRB - KEEL ROLL BENDING
RRR - RADIATOR RIGID BODY ROLL
LBRR - LOWER BOOM RIGID BODY ROLL
RM - RADIATOR MAST BENDING
TB - TRANSVERSE BOOM BENDING
KPB - KEEL PITCH BENDING
KRP - KEEL RIGID BODY PITCH
MRR - MODULE RIGID BODY ROLL
TBRR - TRANSVERSE BOOM RIGID BODY ROLL

* MODE SHAPE DESCRIPTIONS CORRESPOND TO FIRST COLUMN OF VALUES ONLY

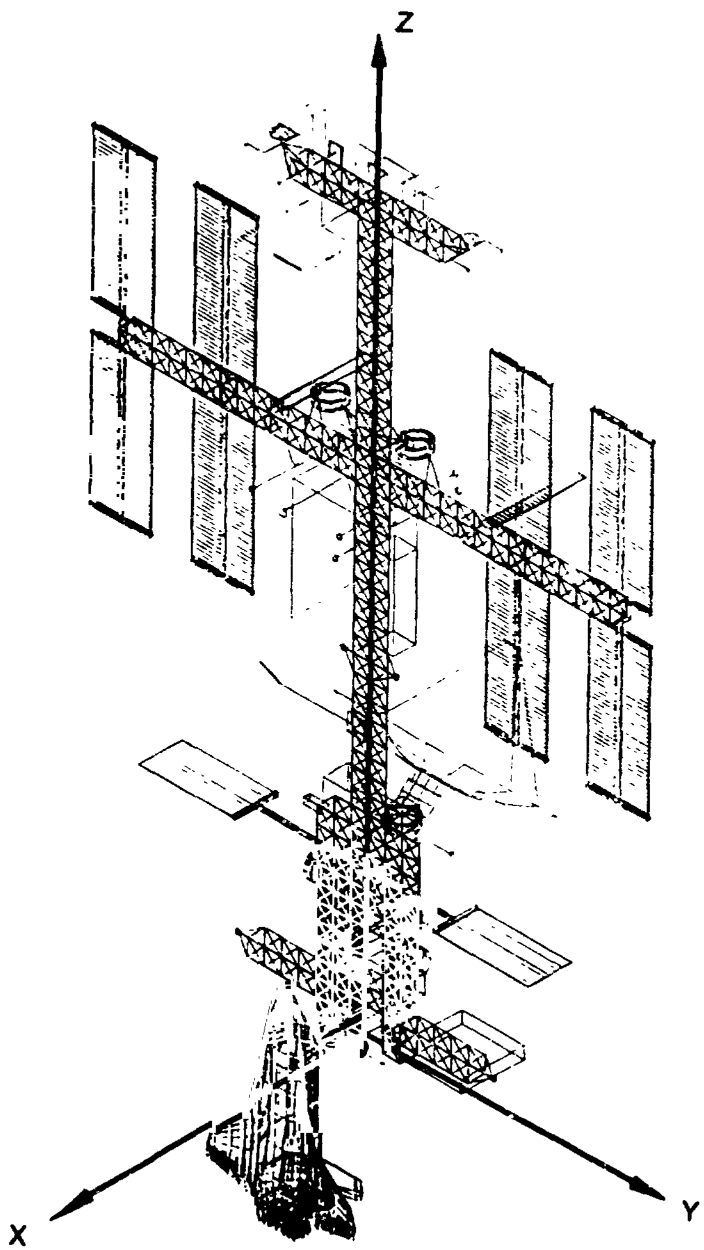


Figure 4.3.3.3-1.- IOC REFERENCE SPACE STATION

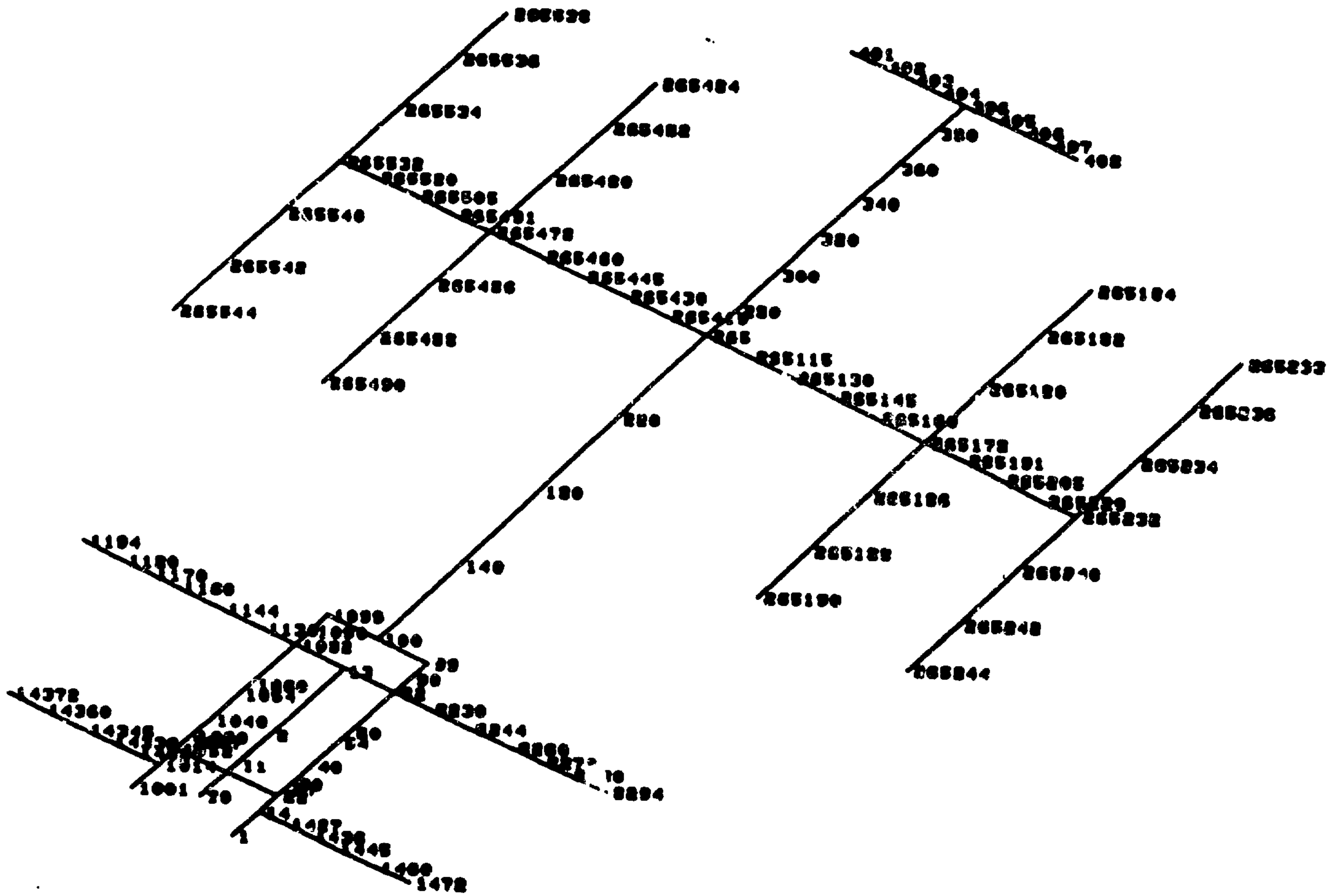


Figure 4.3.3.3-2 - UNDEFORMED ANALYTICAL MODEL

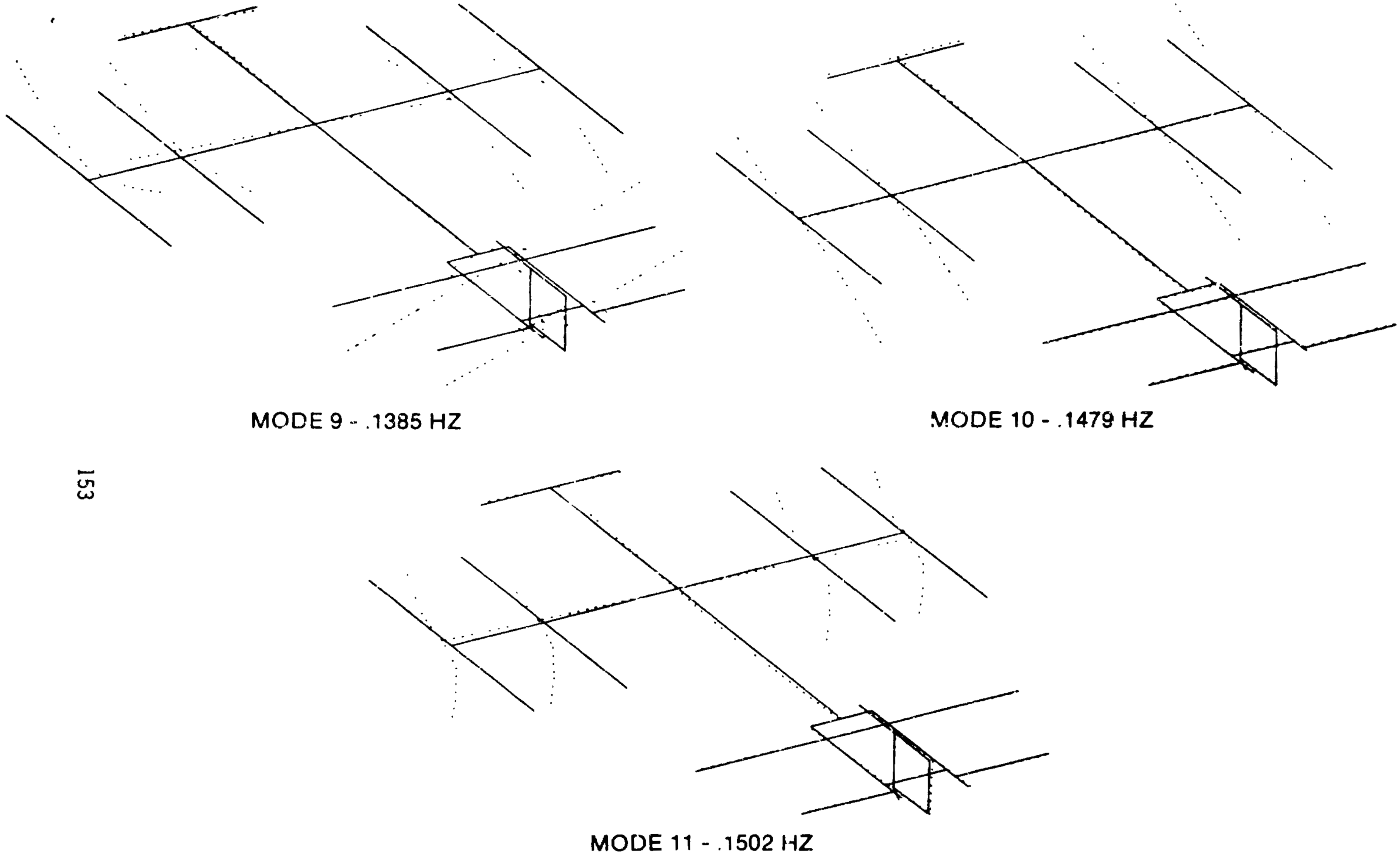
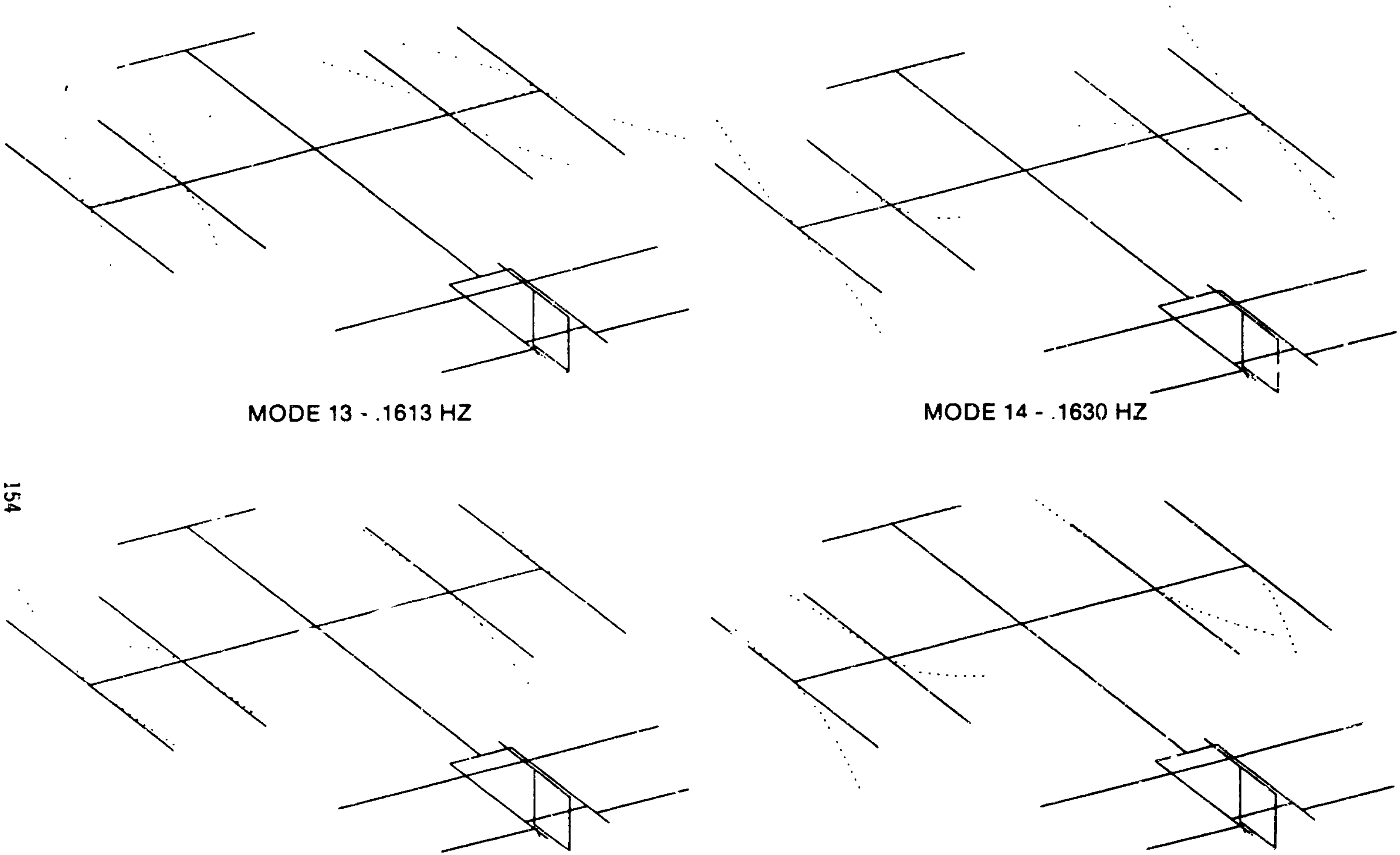


Figure 4.3.3.3-3 - LOWEST ARRAY MAST DOMINATED MODES WITH SOME PRIMARY STRUCTURE DEFORMATION IN THE ABSENCE OF PAYLOADS AND ORBITER



MODE 13 - .1613 HZ

MODE 14 - .1630 HZ

MODE 15 - .1636 HZ

MODE 16 - .1642 HZ

Figure 4.3.3.3-4 - ARRAY MAST CANTILEVER MODES IN THE ABSENCE OF PAYLOADS AND ORBITER

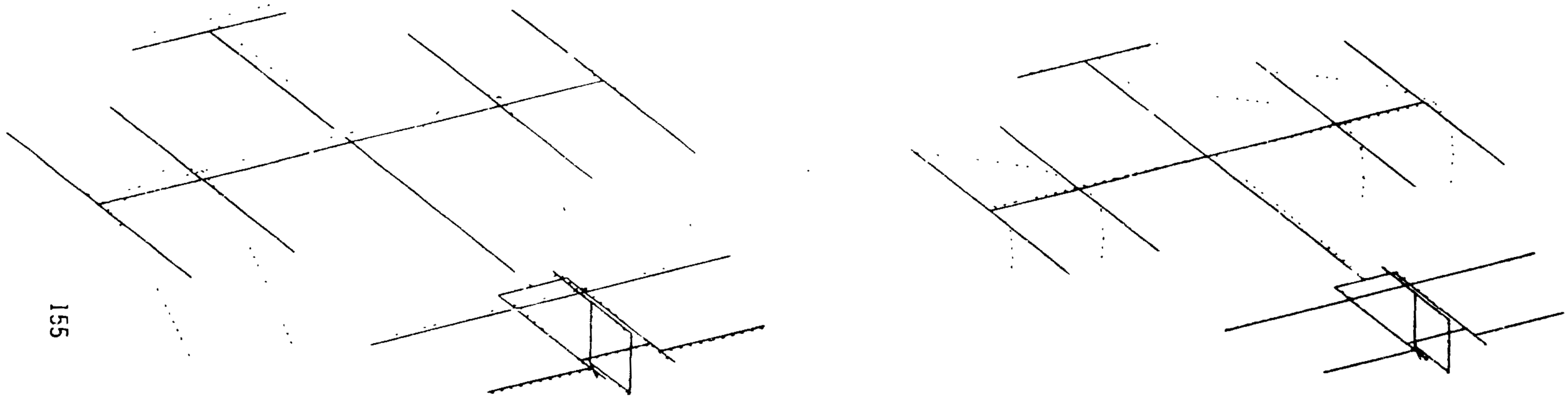


Figure 4.3.3.3-5 - ARRAY MAST DOMINATED MODES WITH SOME PRIMARY STRUCTURE DEFORMATION IN THE ABSENCE OF PAYLOADS AND ORBITER

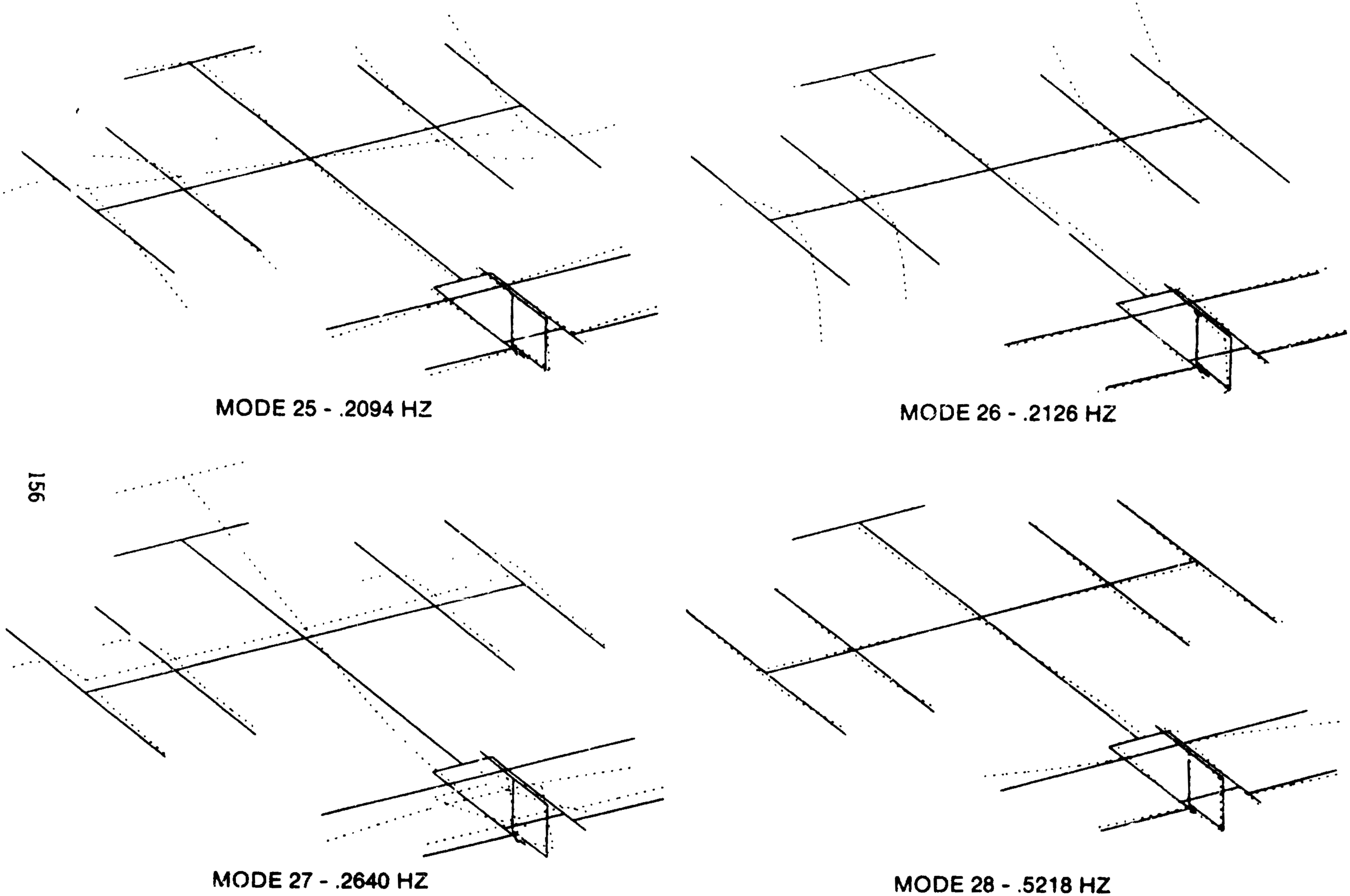


Figure 4.3.3.3-6 - PRIMARY STRUCTURE MODES IN THE ABSENCE OF PAYLOADS AND ORBITER

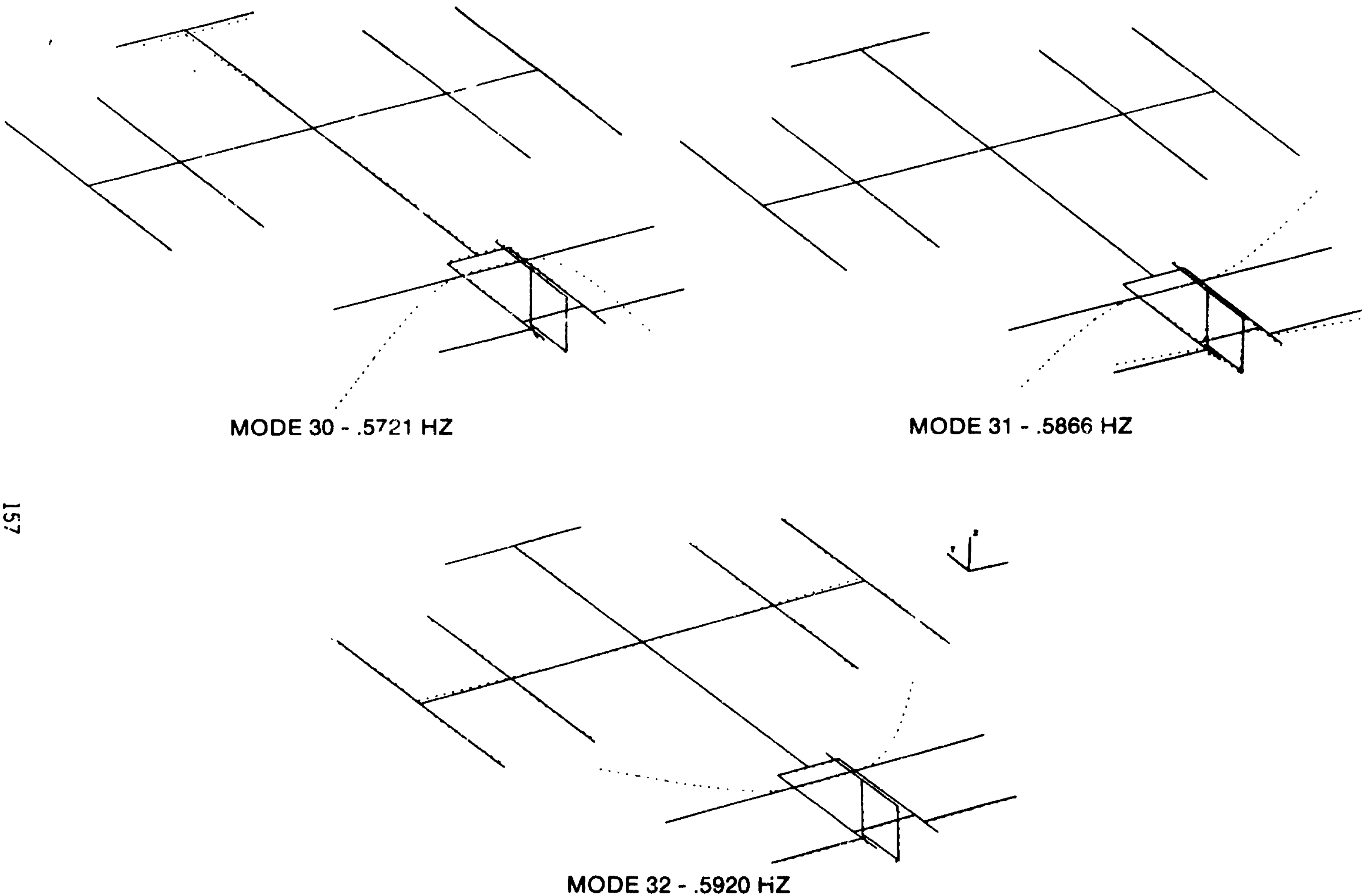
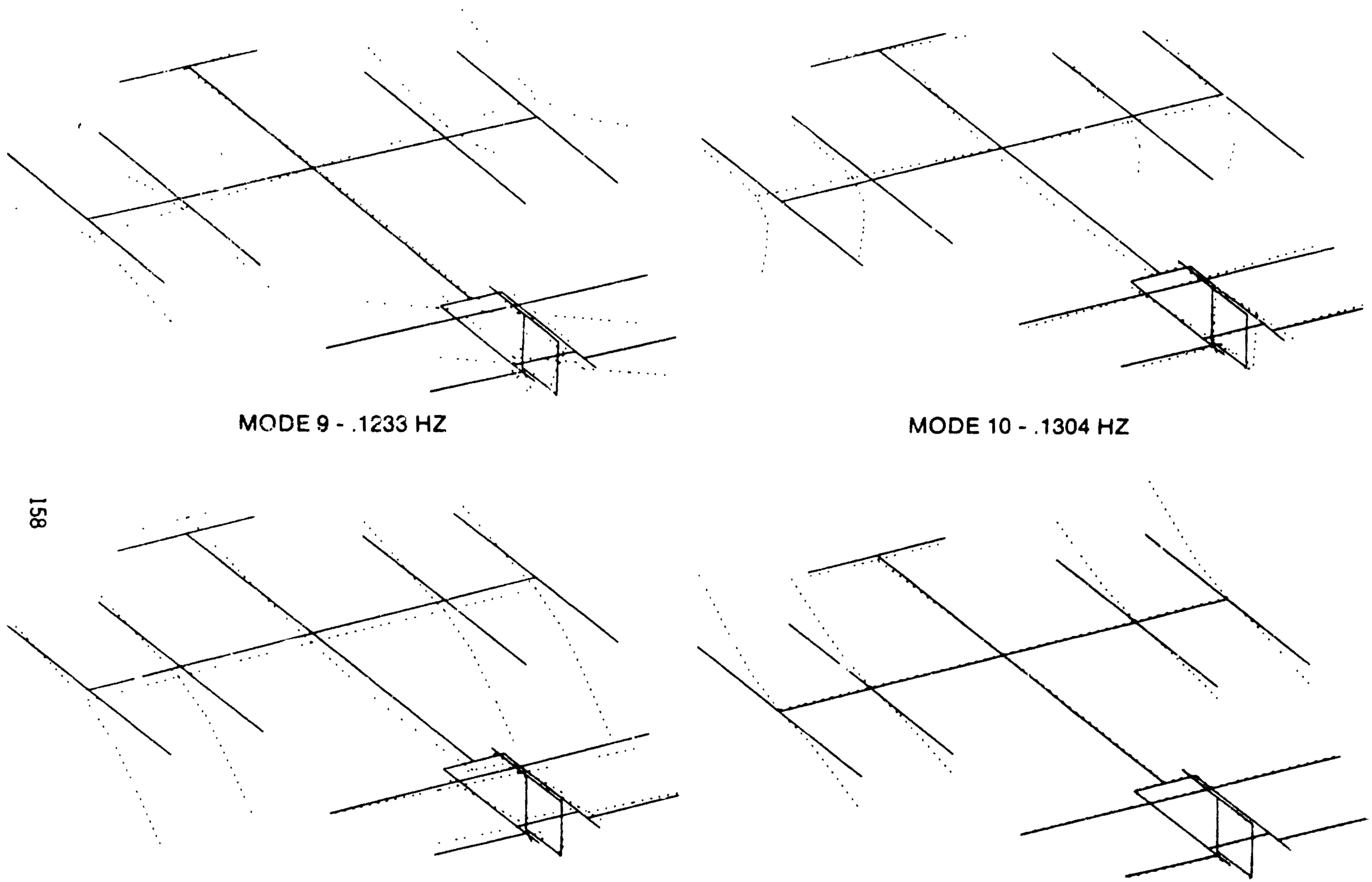


Figure 4.3.3.3-7 - RADIATOR MAST DOMINATED MODES IN THE ABSENCE OF PAYLOADS AND ORBITER



MODE 9 - .1233 HZ

MODE 10 - .1304 HZ

MODE 11 - .1317 HZ

MODE 12 - .1542 HZ

Figure 4.3.3.3-8 - LOWEST ARRAY MAST DOMINATED MODES WITH SOME PRIMARY STRUCTURE DEFORMATION WITH PAYLOADS ATTACHED

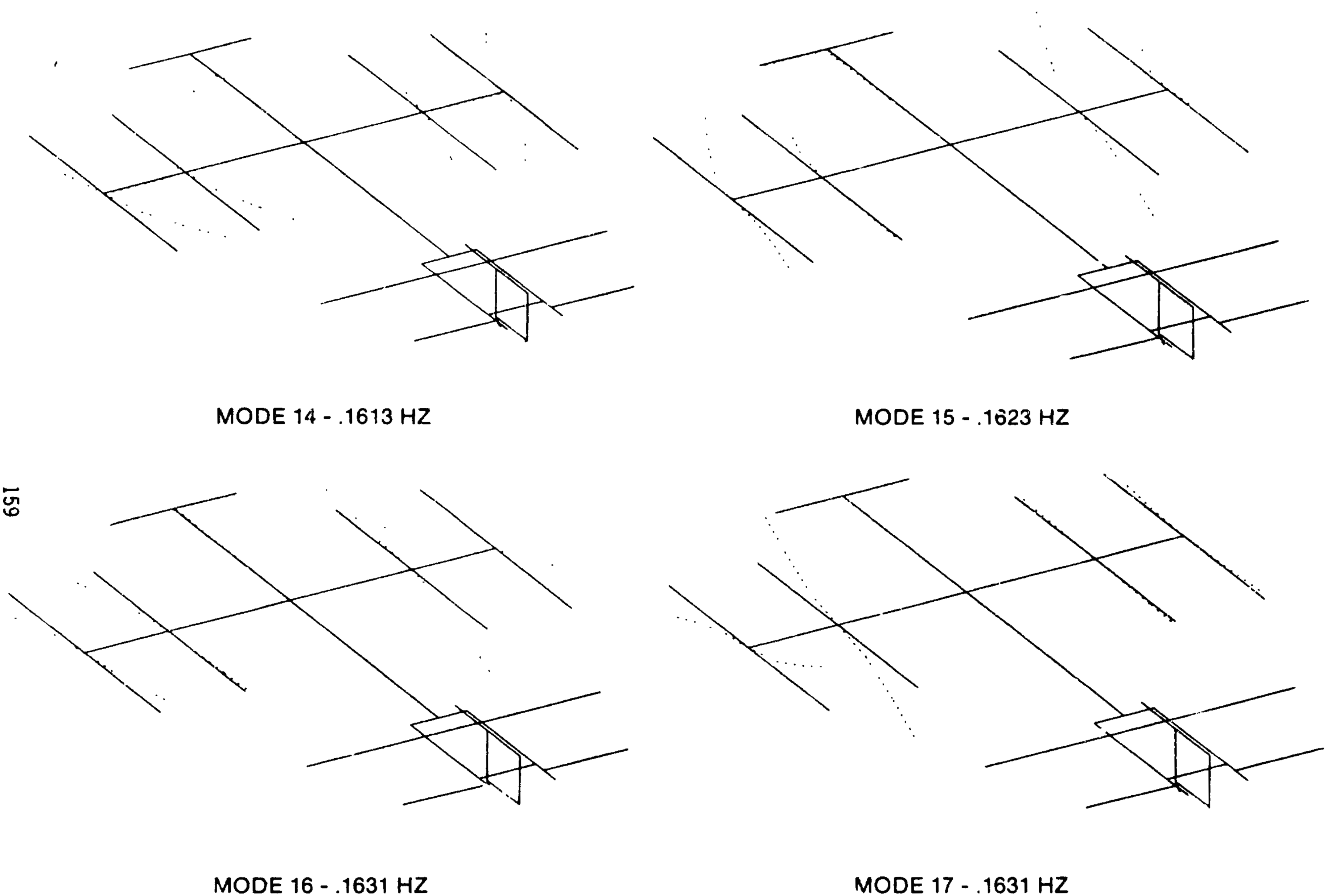
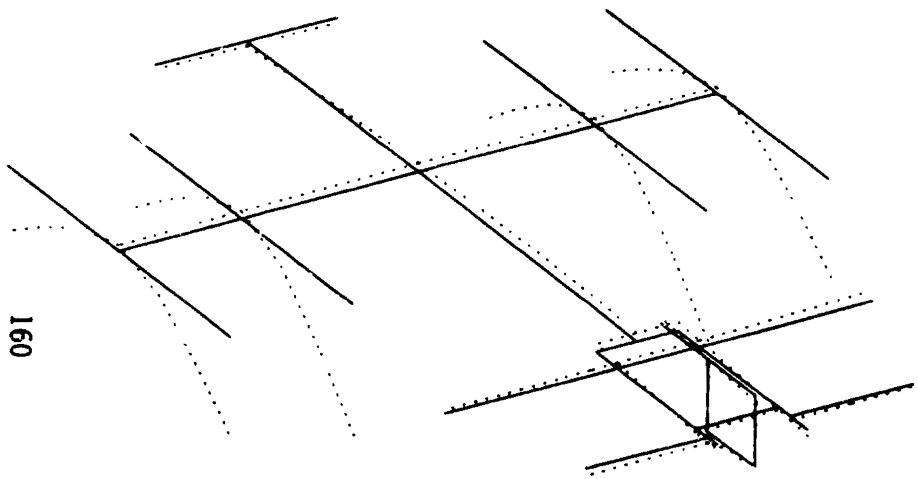


Figure 4.3.3.3-9 - ARRAY MAST CANTILEVER MODES WITH PAYLOADS ATTACHED


MODE 25 - .1733 HZ

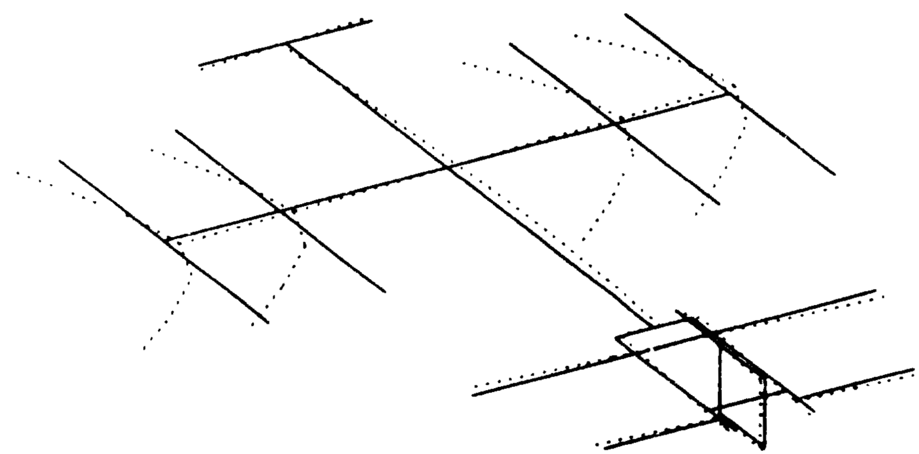
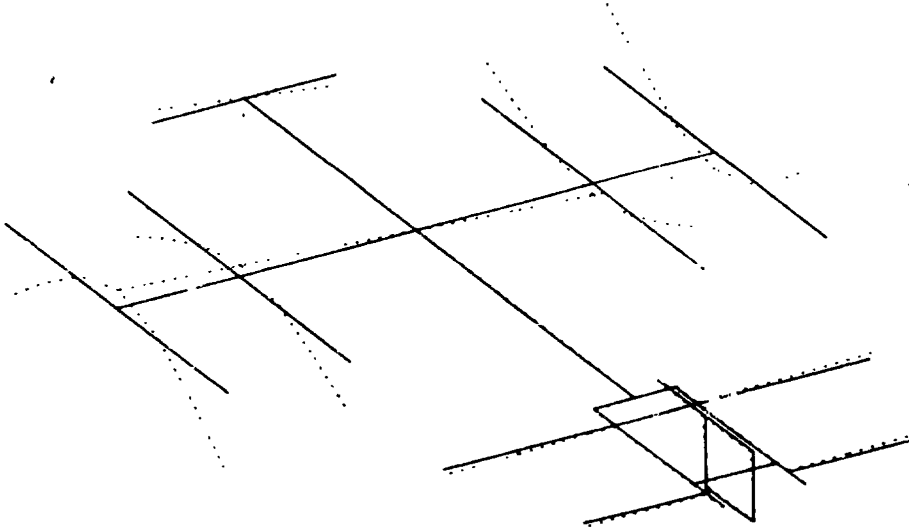
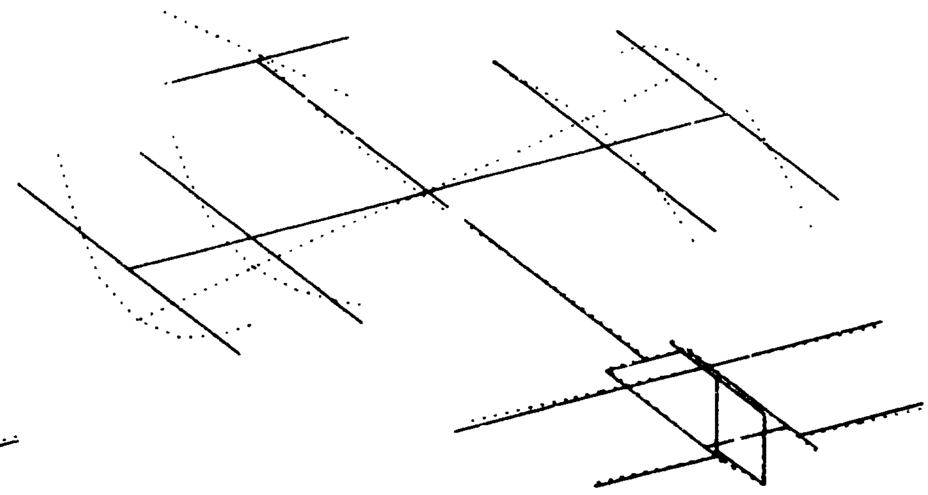

MODE 26 - .1778 HZ

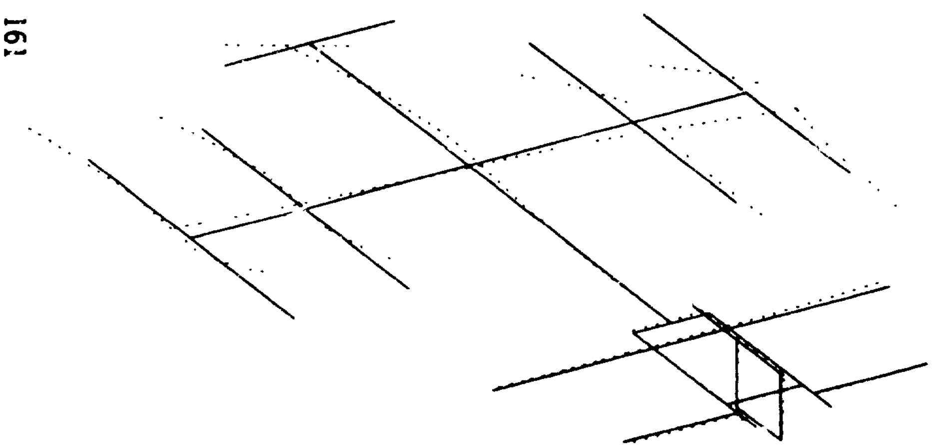
Figure 4.3.3.3-10 - ARRAY MAST DOMINATED MODES WITH SOME PRIMARY STRUCTURE DEFORMATION WITH PAYLOADS ATTACHED



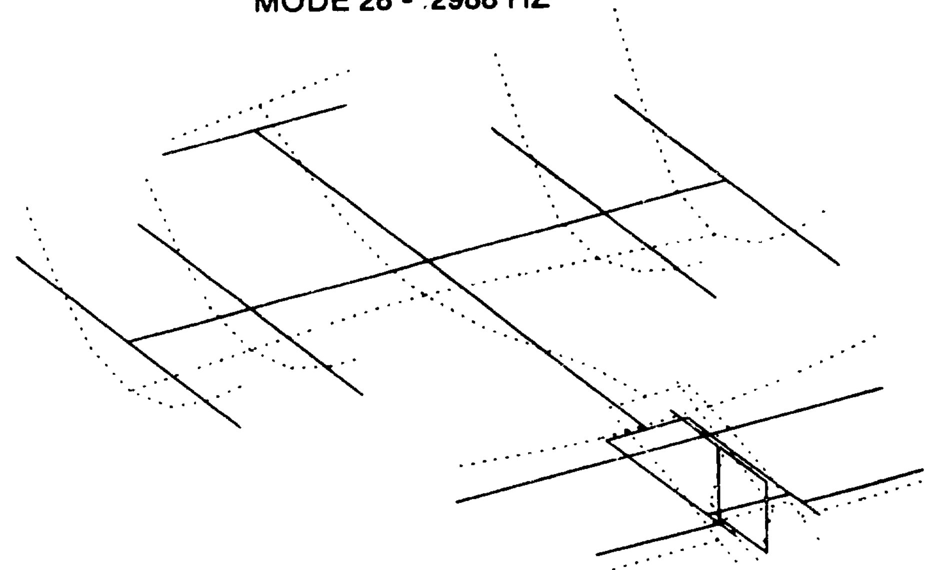
MODE 27 - .1983 HZ



MODE 28 - .2988 HZ

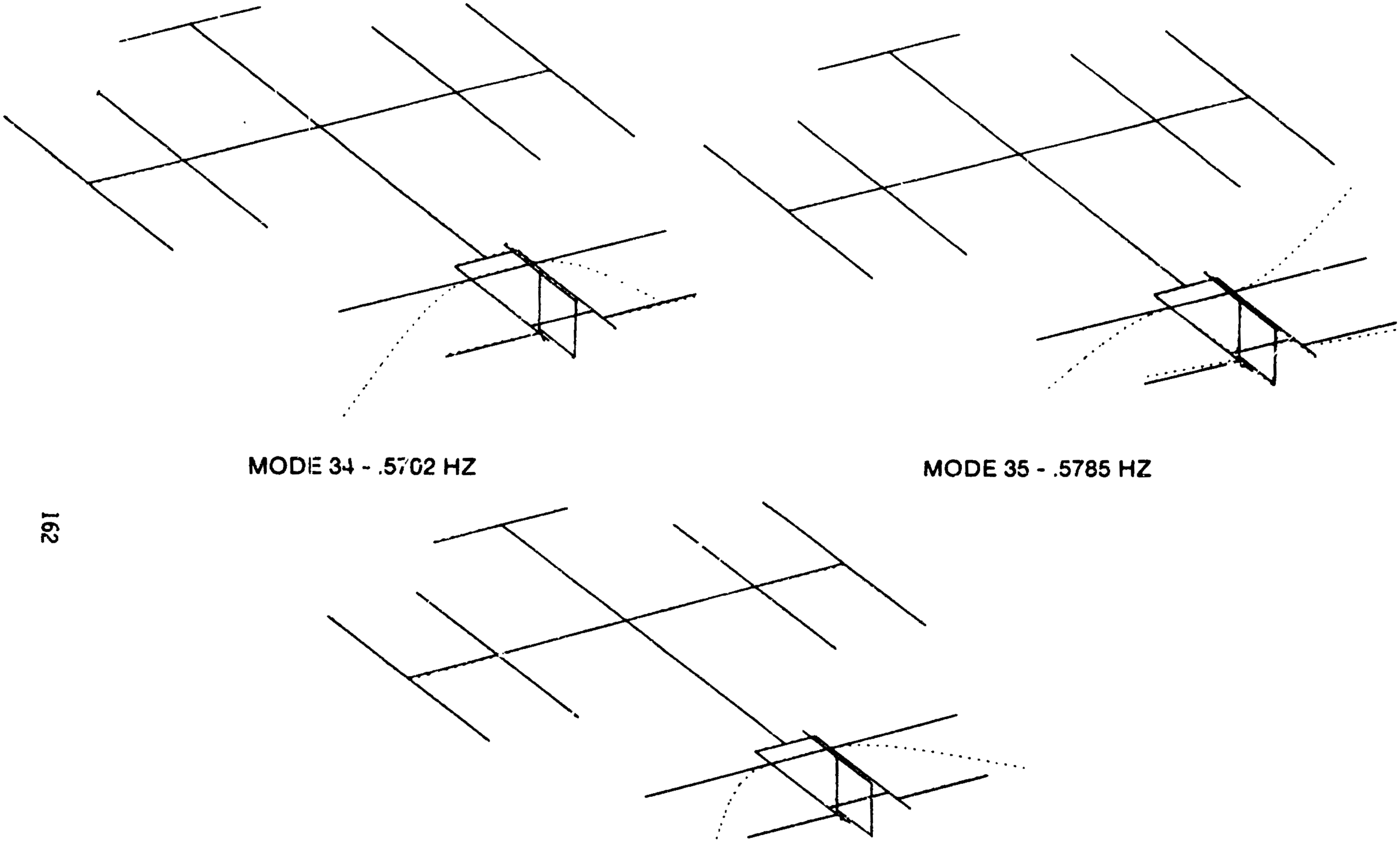


MODE 29 - .3190 HZ



MODE 30 - .3195 HZ

Figure 4.3.3.3-11 - PRIMARY STRUCTURE MODES WITH PAYLOADS ATTACHED

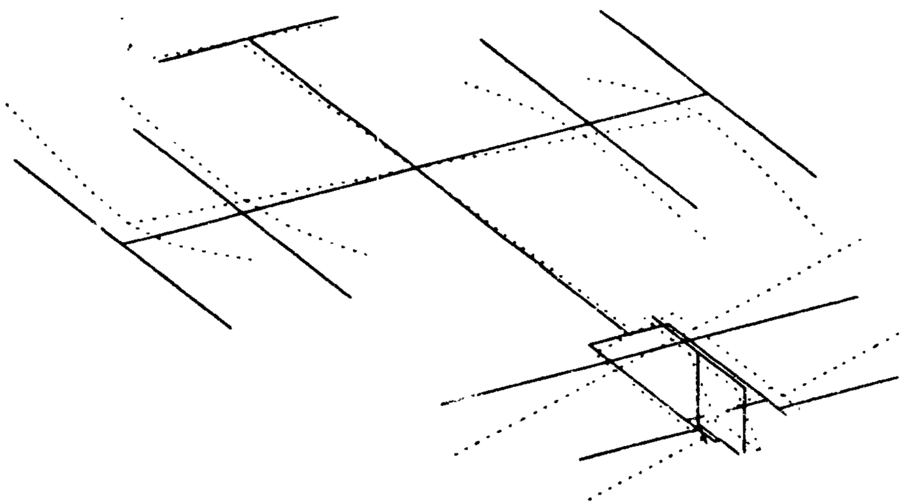


MODE 34 - .5702 HZ

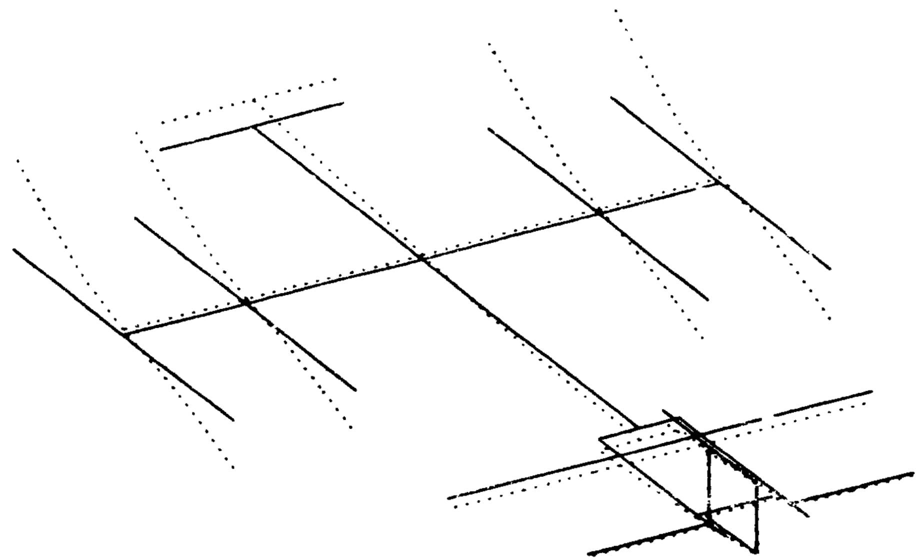
MODE 35 - .5785 HZ

MODE 36 - .5897 HZ

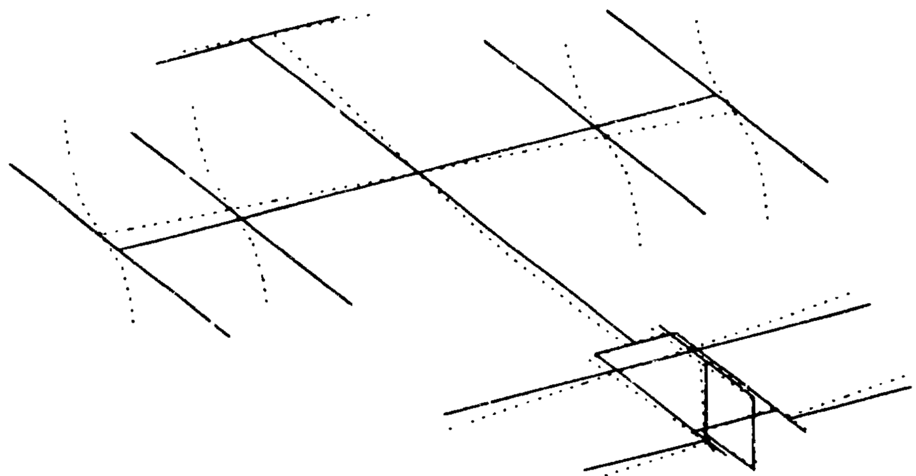
Figure 4.3.3.3-12 - RADIATOR MAST DOMINATED MODES WITH PAYLOADS ATTACHED



MODE 9 - .1200 HZ

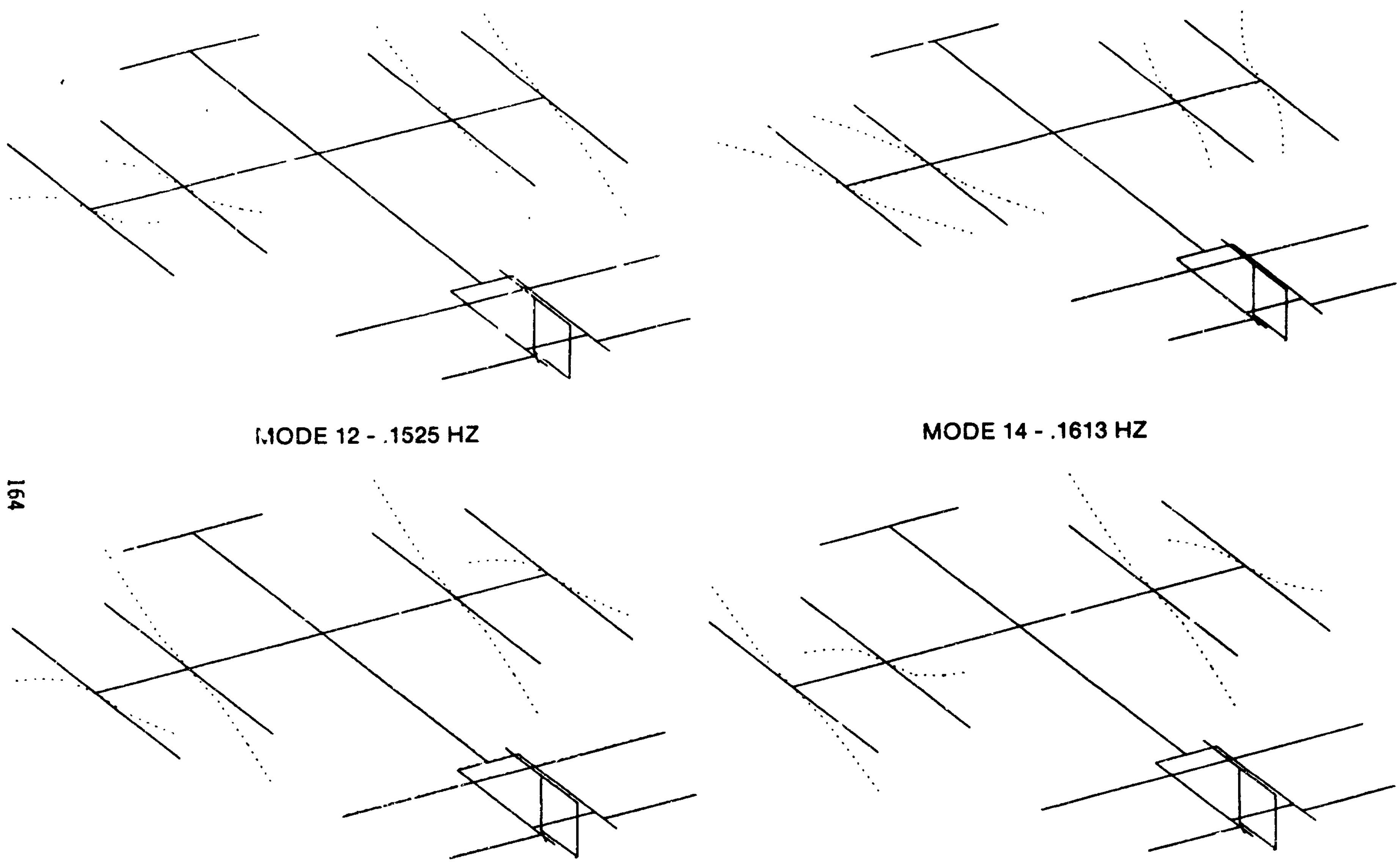


MODE 10 - .1345 HZ



MODE 11 - .1376 HZ

Figure 4.3.3.3-13 - LOWEST ARRAY MODES DOMINATED BY PRIMARY STRUCTURE AND ARRAY MAST DEFORMATION WHEN ORBITER IS ATTACHED



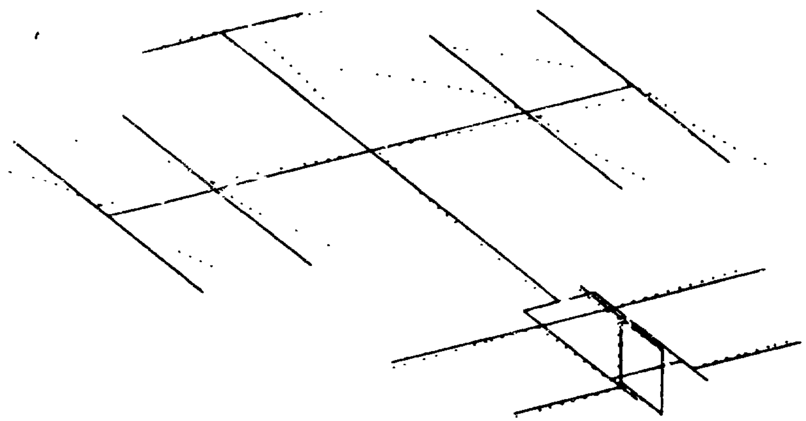
MODE 12 - .1525 HZ

MODE 14 - .1613 HZ

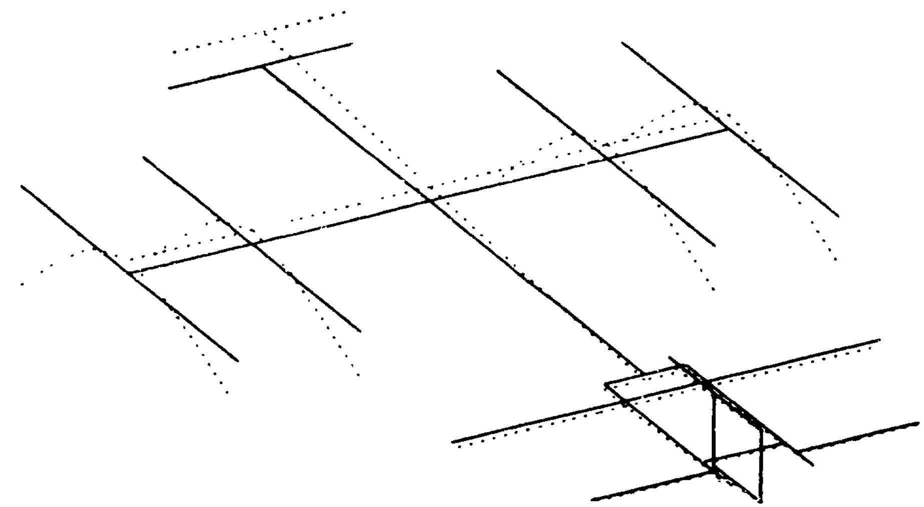
MODE 15 - .1632 HZ

MODE 16 - .1632 HZ

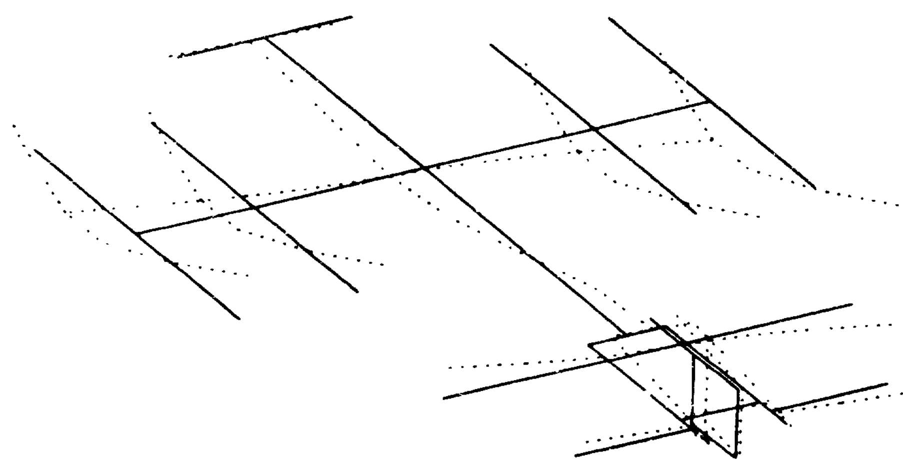
Figure 4.3.3.3-14 - ARRAY MAST CANTILEVER MODES WHEN ORBITER IS ATTACHED



MODE 25 - .1840 HZ

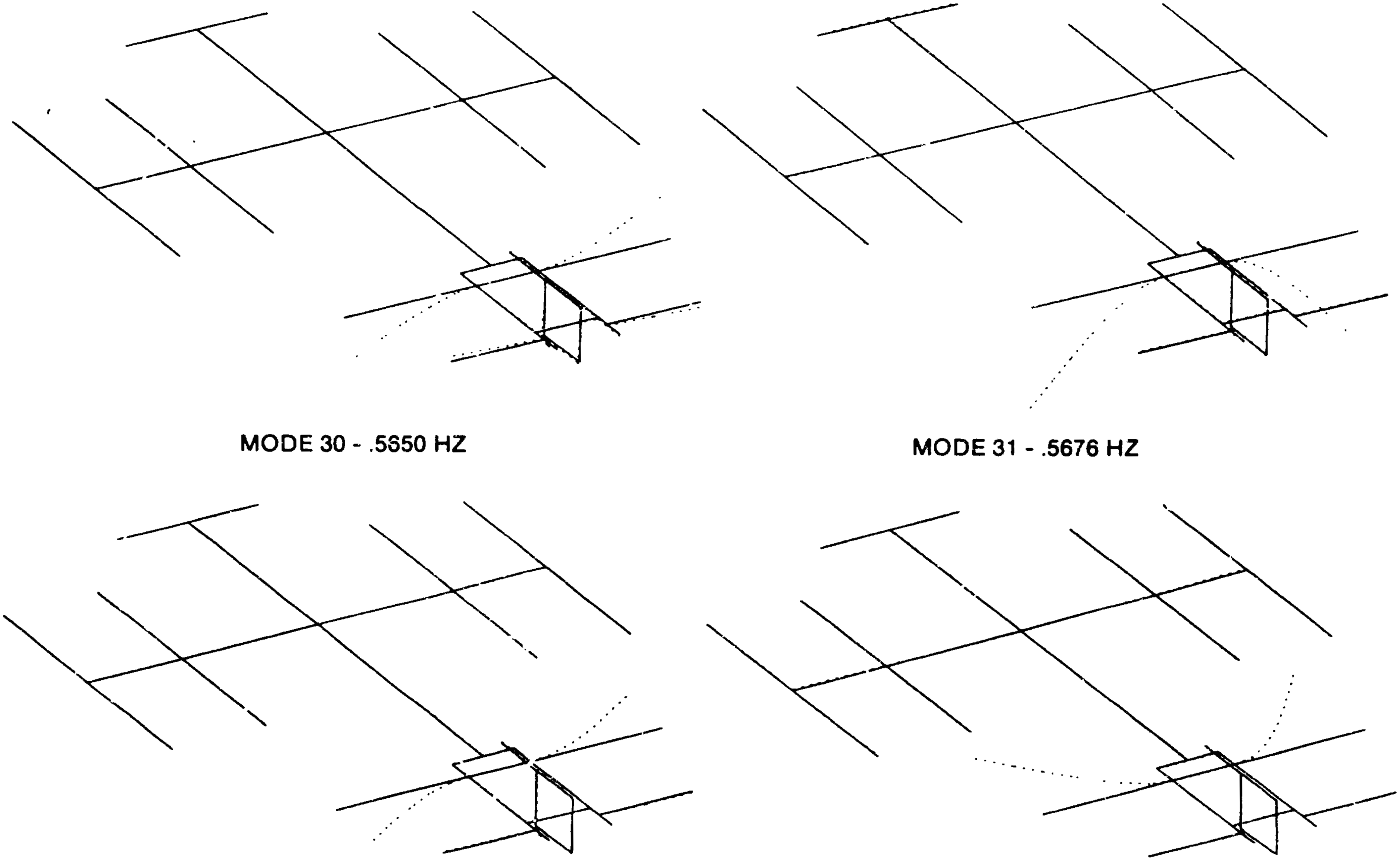


MODE 26 - .1877 HZ



MODE 28 - .4030 HZ

Figure 4.3.3.3-15 - PRIMARY STRUCTURE MODES WHEN ORBITER IS ATTACHED

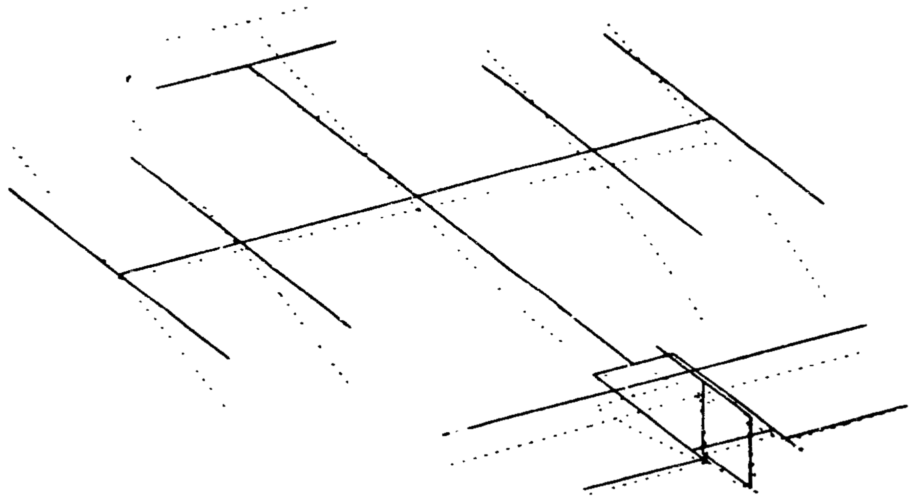

MODE 30 - .5650 HZ

MODE 31 - .5676 HZ

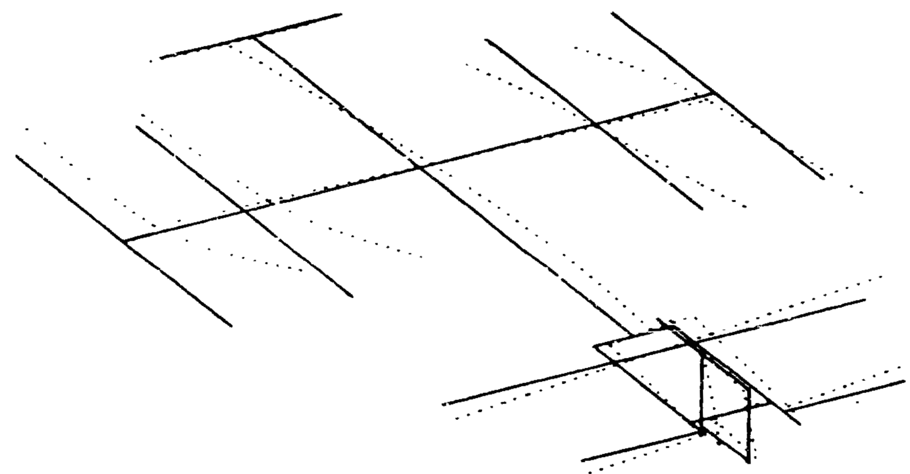
MODE 32 - .5881 HZ

MODE 33 - .5890 HZ

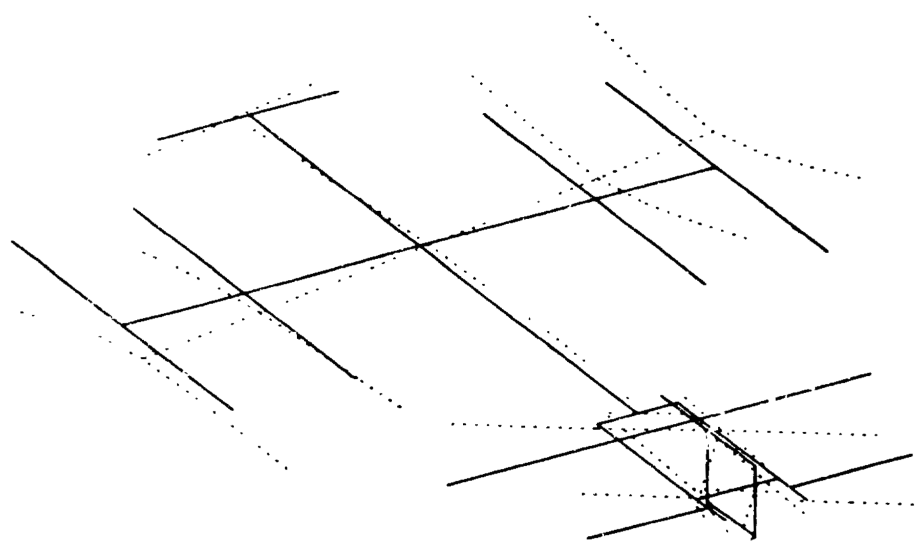
Figure 4.3.3.3-16 - RADIATOR MAST MODES WHEN ORBITER IS ATTACHED



MODE 9 - .0965 HZ



MODE 10 - .1020 HZ



MODE 11 - .1139 HZ

Figure 4.3.3.3-17 - LOWEST MODES DOMINATED BY PRIMARY STRUCTURE DEFORMATION WITH BOTH PAYLOADS AND ORBITER ATTACHED

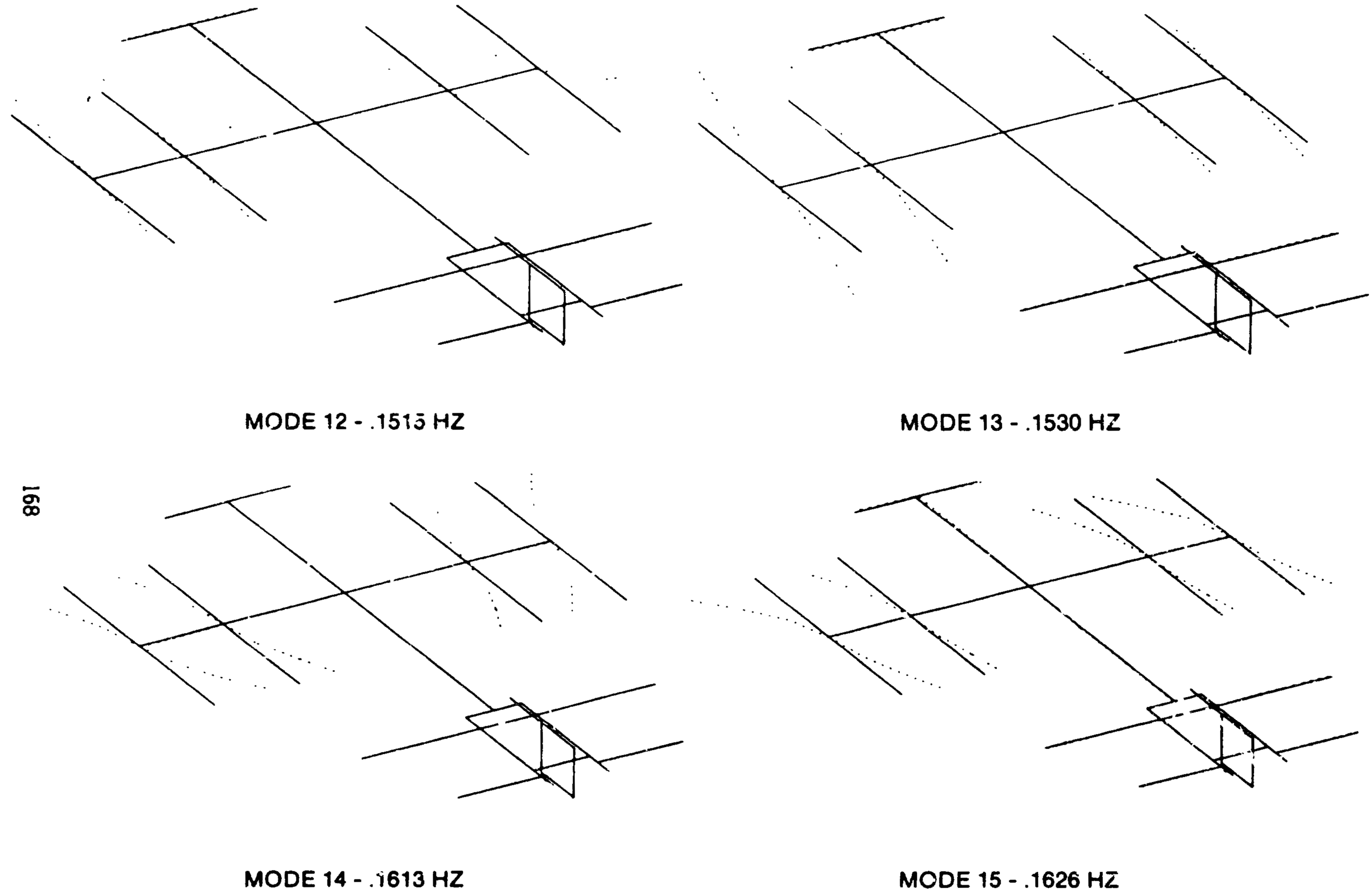
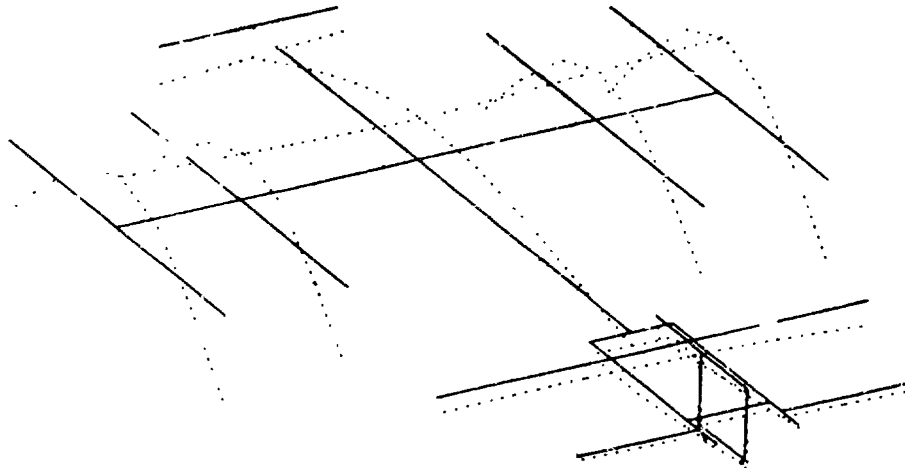
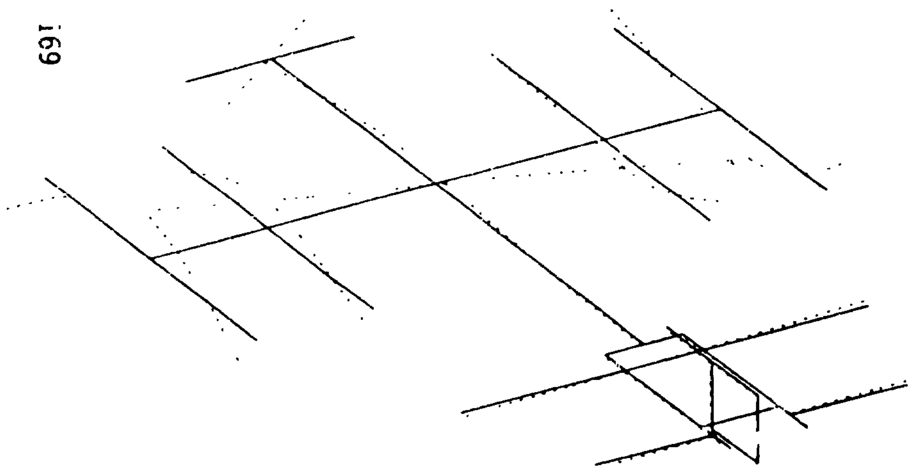


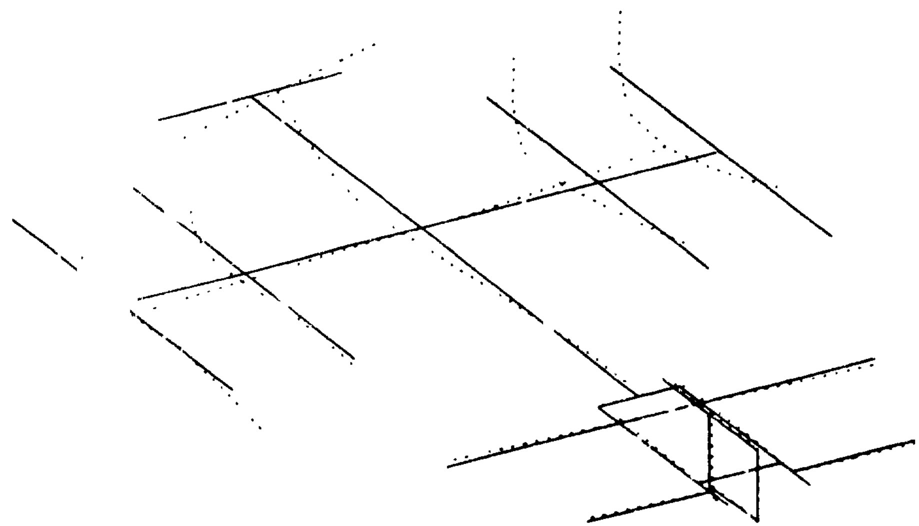
Figure 4.3.3.3-18 - ARRAY MAST CANTILEVER MODES WITH BOTH PAYLOADS AND ORBITER ATTACHED



MODE 28 - .2627 HZ

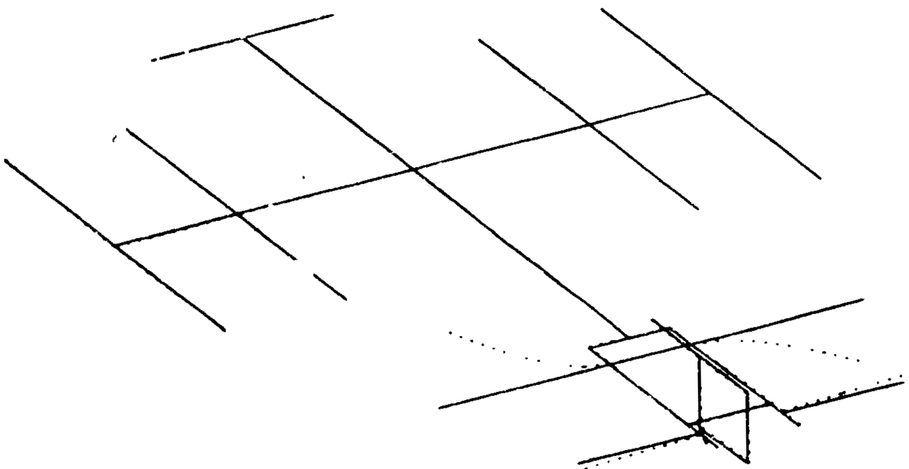


MODE 29 - .2983 HZ

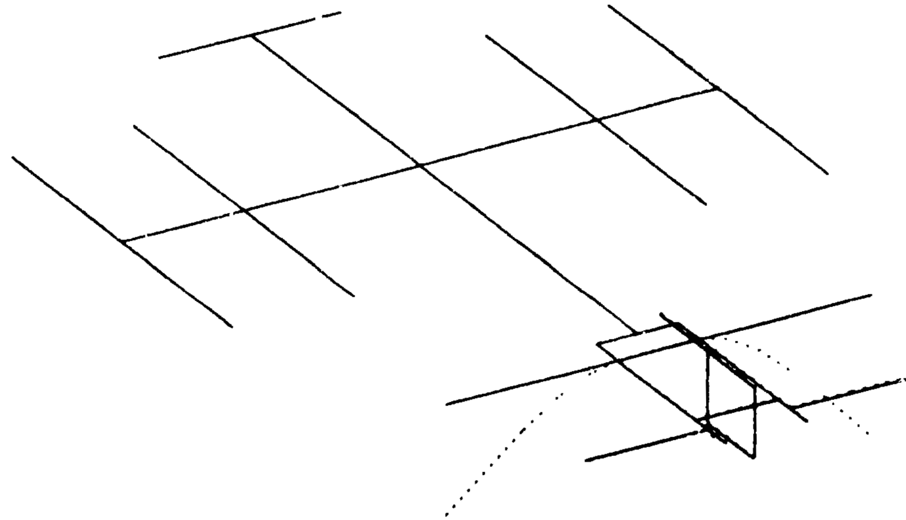


MODE 30 - .3168 HZ

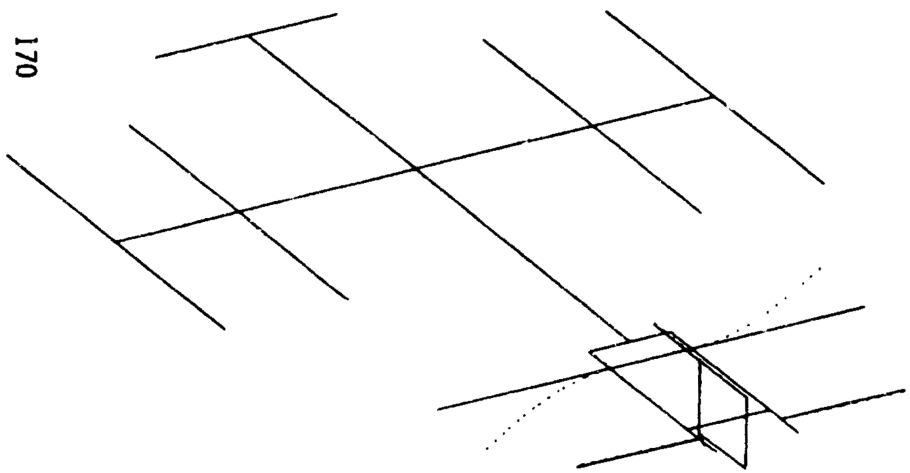
Figure 4.3.3.3-19 - PRIMARY STRUCTURE MODES WITH BOTH PAYLOADS AND ORBITER ATTACHED



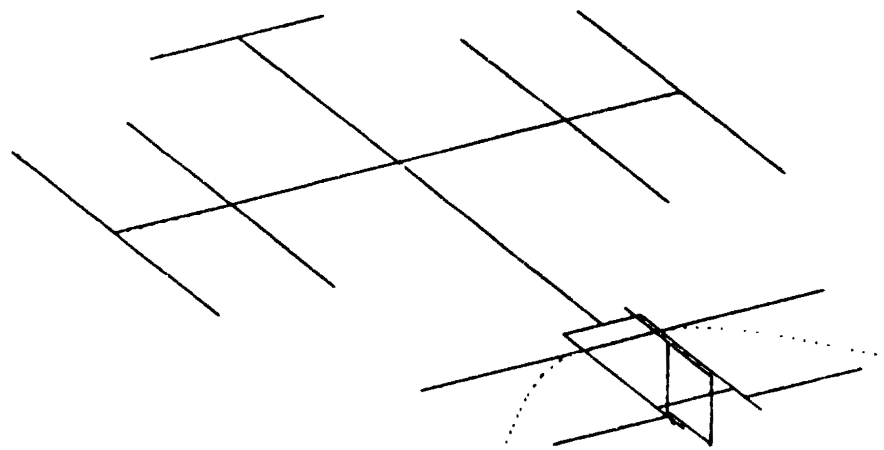
MODE 34 - .5572 HZ



MODE 35 - .5671 HZ



MODE 36 - .5874 HZ



MODE 37 - .5885 HZ

Figure 4.3.3.3-20.- RADIATOR MAST MODES WITH BOTH PAYLOADS AND ORBITER ATTACHED

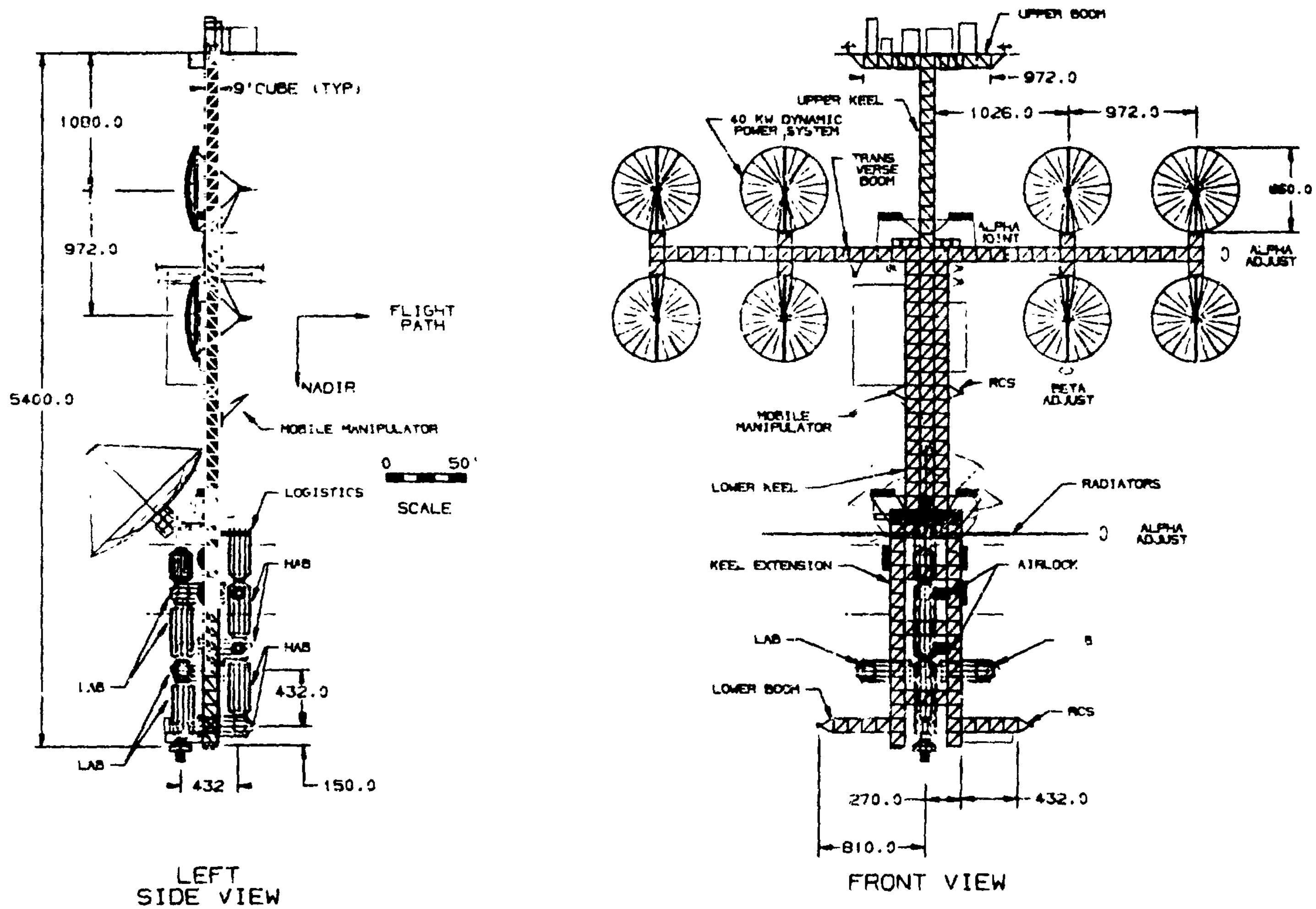


Figure 4.3.3.3-21 - GROWTH CONFIGURATION USED IN DYNAMIC STUDY

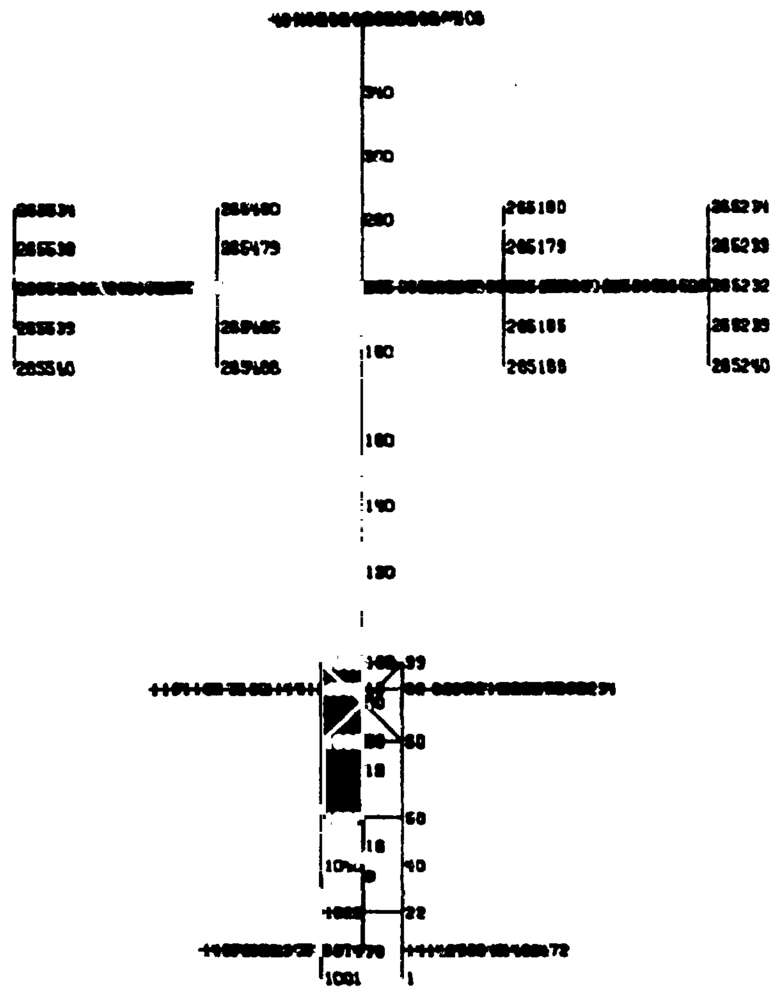


Figure 4.3.3.3-22 - ANALYTICAL MODEL FOR GROWTH CONFIGURATION

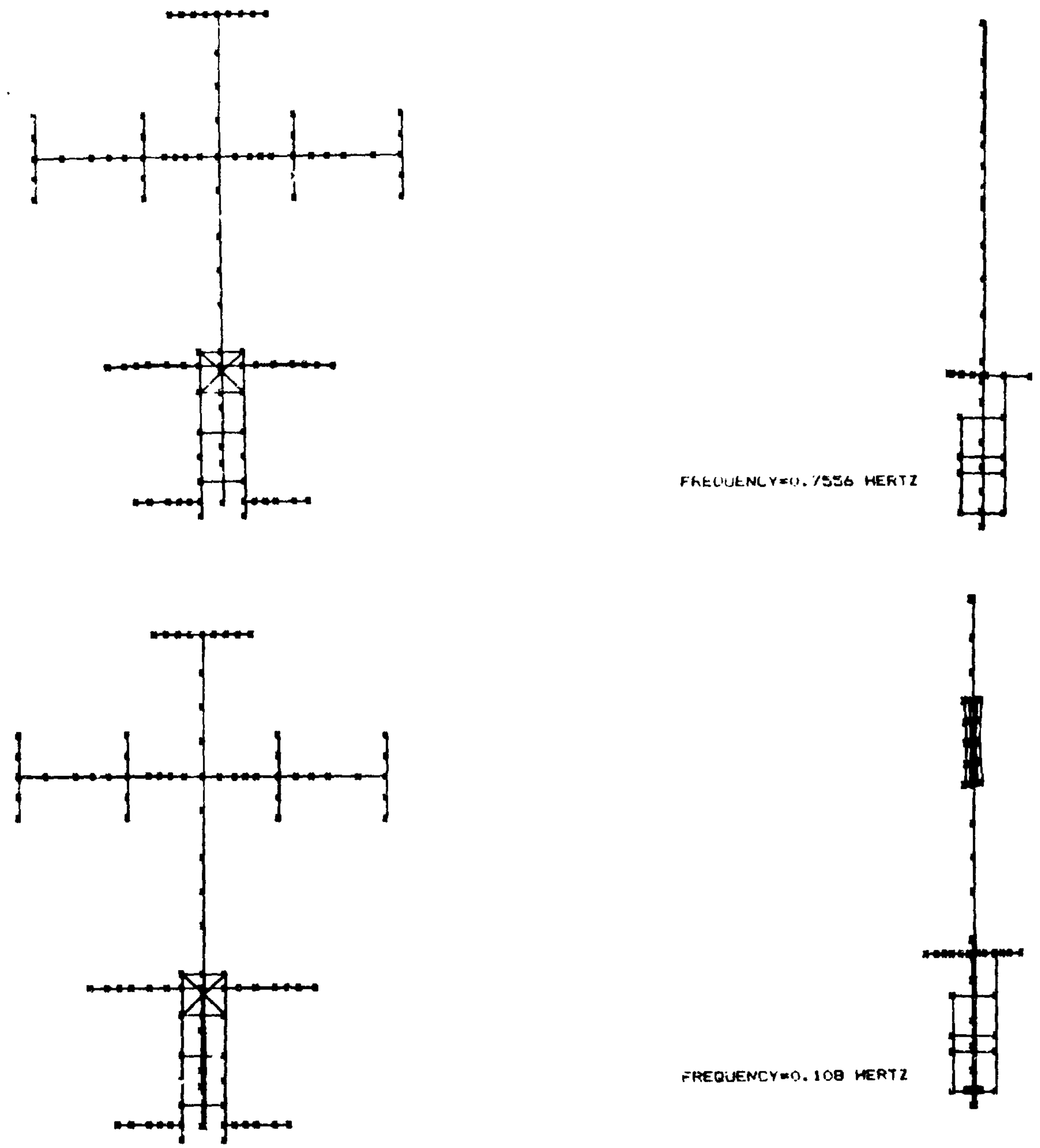


Figure 4.3.3.3-23 - SELECTED GROWTH STATION KEEL TWIST MODES IN THE ABSENCE OF ATTACHMENTS

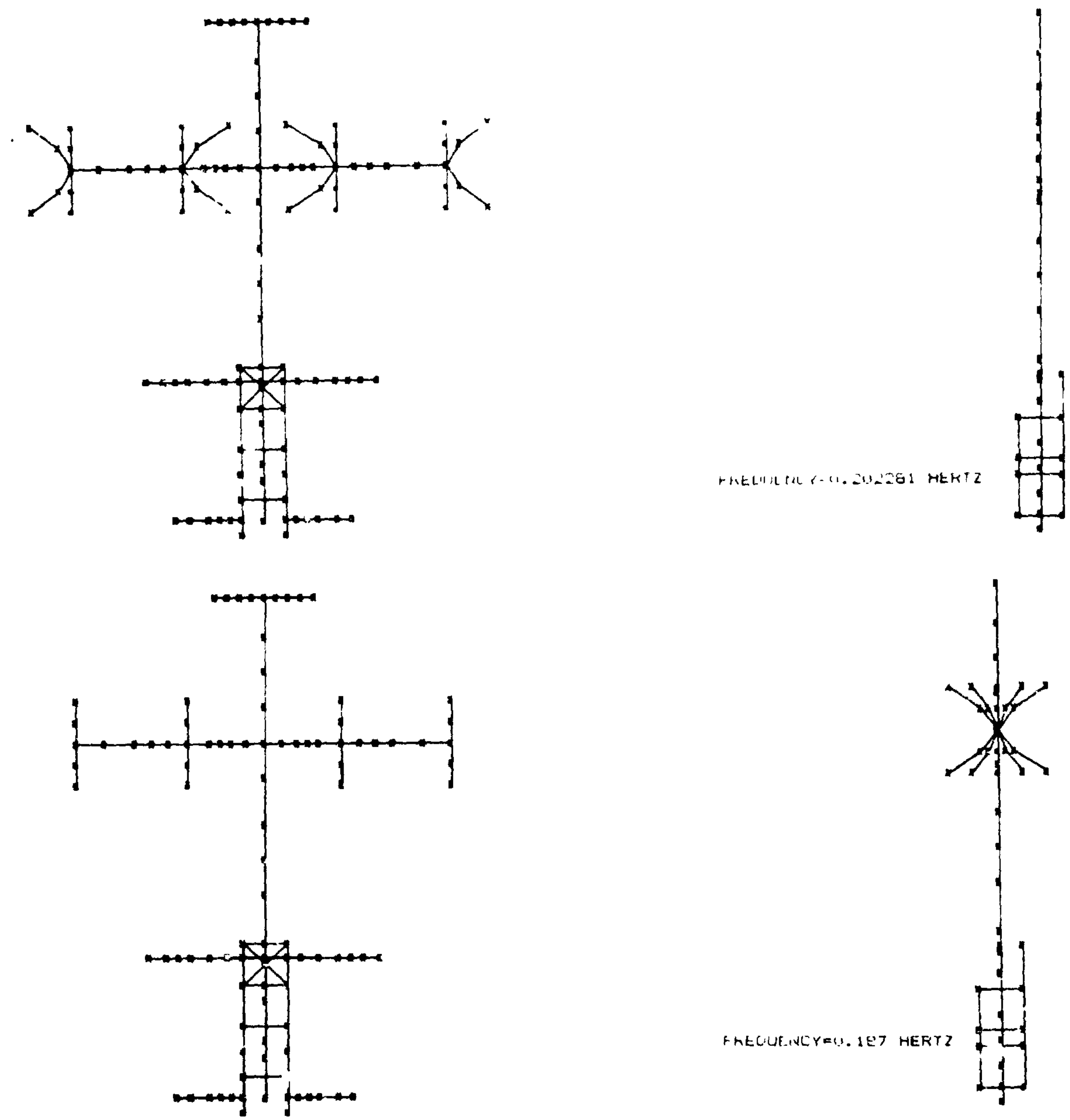


Figure 4.3.3.3- 24 - SELECTED GROWTH STATION ARRAY MAST MODES

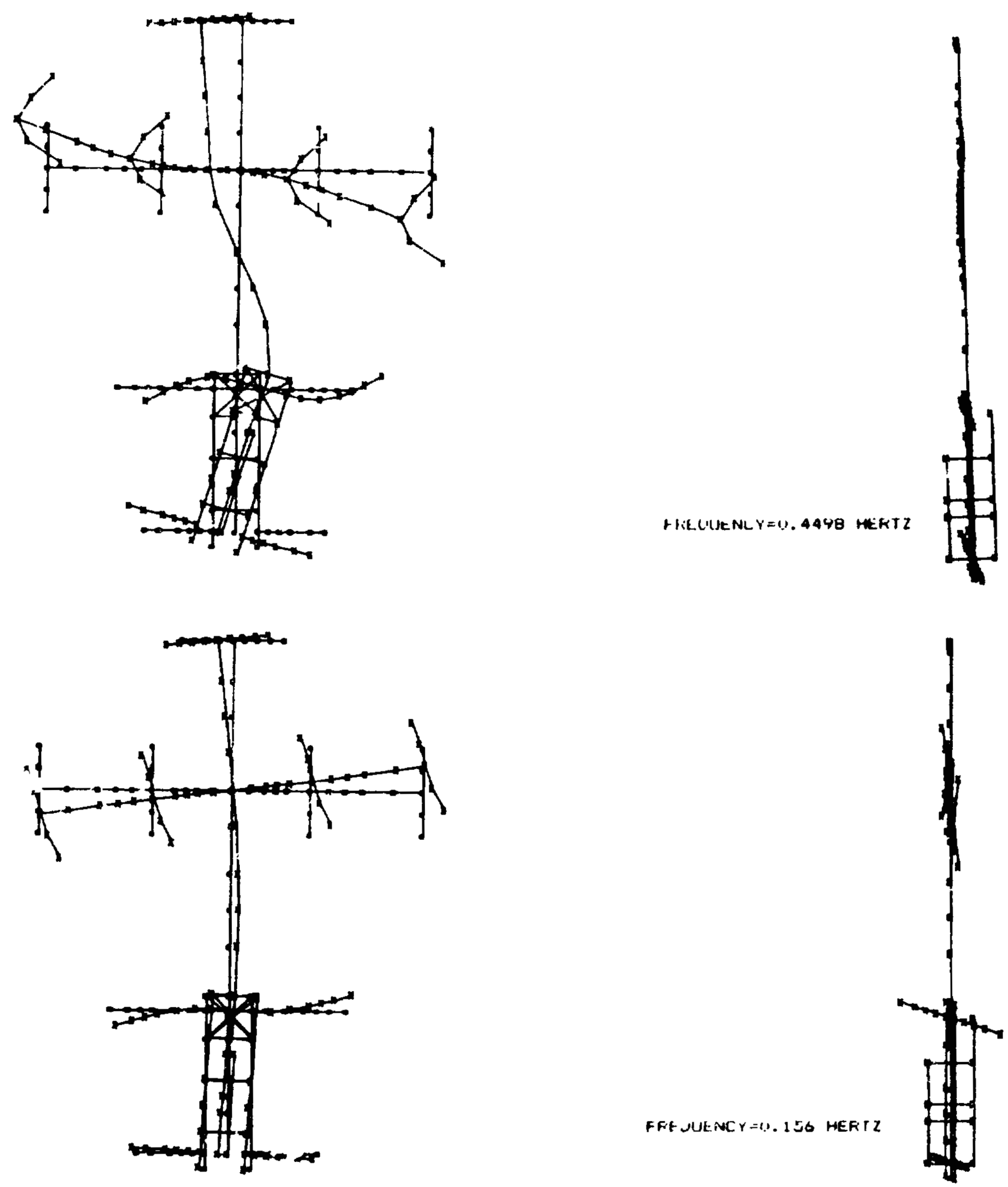


Figure 4.3.3.3-25 - SELECTED GROWTH STATION COUPLED KEEL ROLL BENDING AND KEEL TWIST MODES IN THE ABSENCE OF ATTACHEMENTS

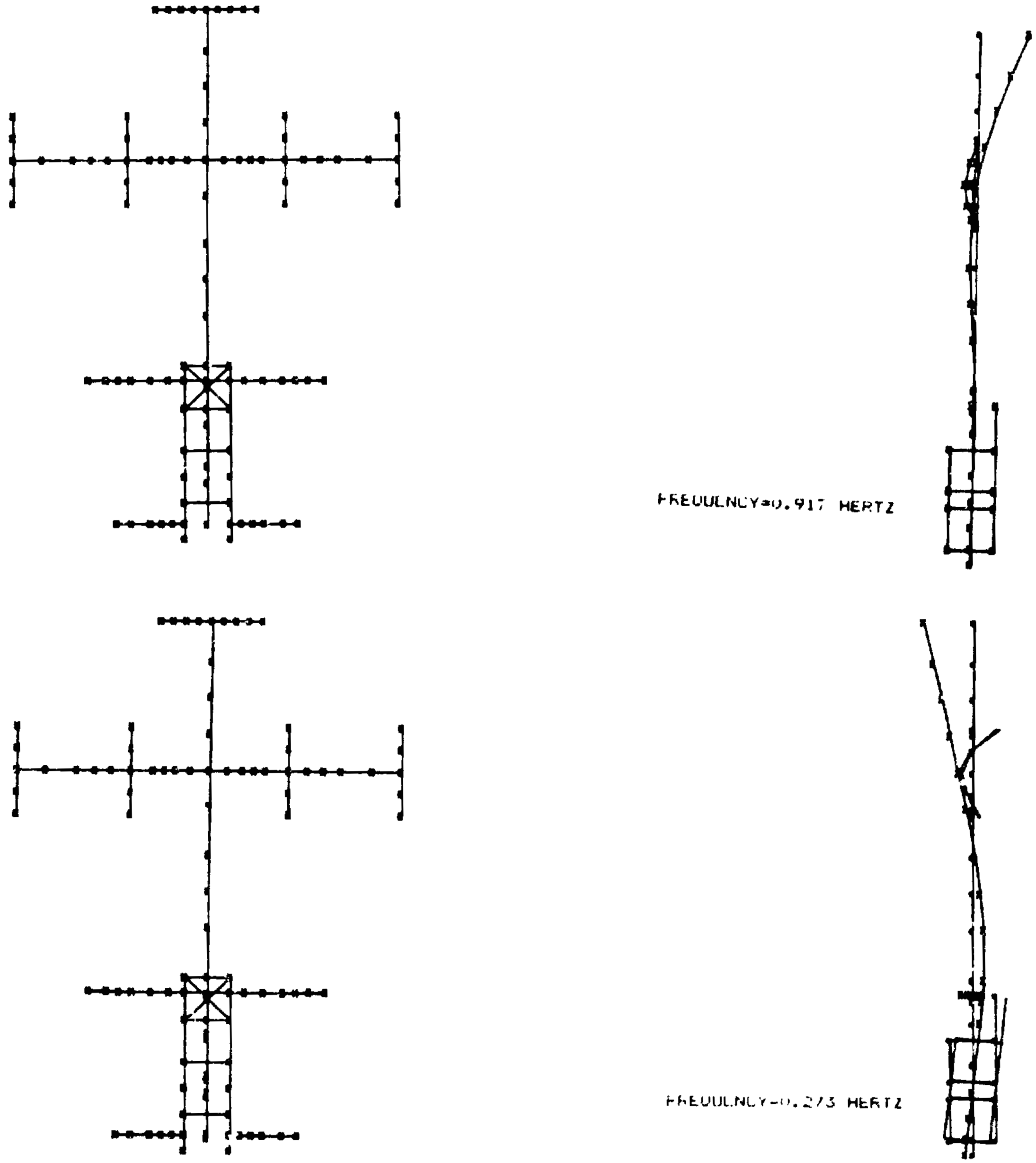


Figure 4.3.3.3-26 - SELECTED GROWTH STATION KEEL PITCH BENDING MODES IN THE ABSENCE OF ATTACHMENTS

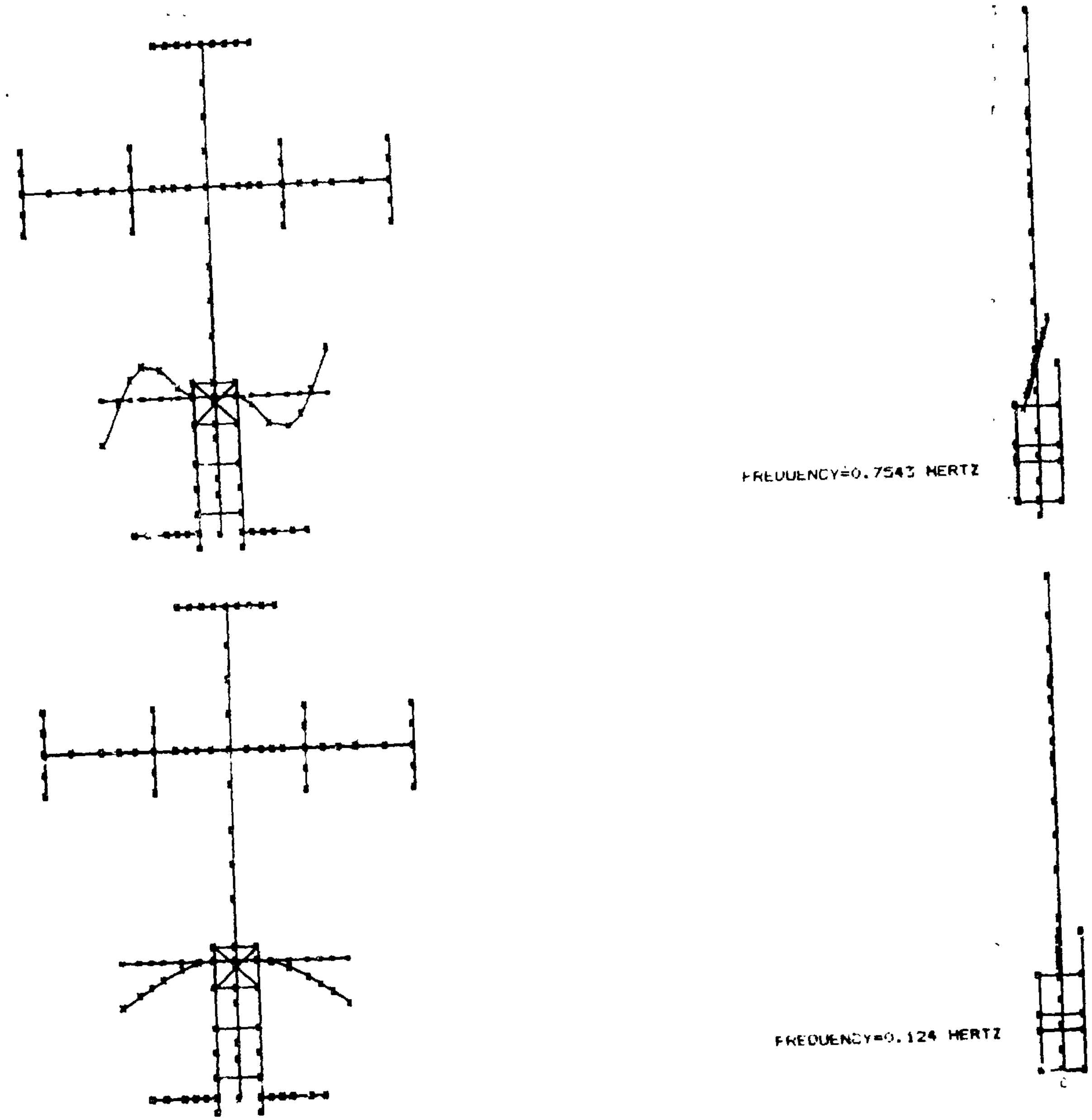


Figure 4.3.3.3-27 - SELECTED GROWTH STATION RADIATOR MAST MODES

4.3.3.4 Mass Properties

Mass properties for the IOC Station used in the rigid body analyses are shown in Table 4.3.3.4-1. Six configurations are characterized in this table. The minimum mass configuration, 124,321 kg, represents the basic Station without the Orbiter, payloads, or servicing items attached. The maximum mass configuration, 312,028 kg, has all these elements attached. The Station element location, masses, and projected areas are shown in Table 4.3.3.4-2. Payload and servicing item locations and masses are shown in Tables 4.3.3.4-3 and 4.3.3.4-4, respectively. The mass properties used for the flexible response and controls analyses are noted in the appropriate paragraphs. As is usually the case, these differ slightly from the values listed in the tables.

TABLE 4.3.3.4-1 - STATION MASS PROPERTIES

	UNITS	WITHOUT ORBITER BARE	WITHOUT ORBITER PAYLOADS	WITHOUT ORBITER PAYLOADS SERVICING	WITH ORBITER BARE	WITH ORBITER PAYLOADS	WITH ORBITER PAYLOADS SERVICING	ORBITER ALONE
WEIGHT								
INERTIAS	KG	124321	157964	205027	231238	264900	306394	106759
IXX	10** KG-M^2	98.72	193.93	256.25	177.57	317.35	417.00	1.36
IYY	10** KG-M^2	94.71	190.10	251.13	181.81	323.50	422.52	10.16
IZZ	10** KG-M^2	7.549	10.10	11.398	18.70	22.99	25.10	10.63
IXY	10** KG-M^2	0	0.11	0.094725	SMALL	0.36	0.38	SMALL
IXZ	10** KG-M^2	-0.7073	-1.09	-1.18063	-22.14	-30.73	37.14	0.35
IYZ	10** KG-M^2	0	-1.47	-0.54212	SMALL	-0.60	0.45	SMALL
C.G. LOCATION								
XCG	METERS	0.1338	-.31979	-0.27027	5.48	4.43	3.32	11.66
YCG	METERS	0	-0.32	-0.34699	0	0.19	0.23	0
ZCG	METERS	55.78	48.22	43.739	71.63	65.10	59.80	90.16

NOTE: THE REFERENCE COORDINATE SYSTEM IS SHOWN ON FIGURE 4.3.3.5-2.

POWER TOWER REFERENCE CONFIGURATION

ITEM	CENTER OF MASS COORDINATES METERS X	Y	Z	WEIGHT KGS	PROJECTED AREAS M**2 X	Y	Z
SOLAR ARRAYS							
POWER BOOM	0.00	0.00	0.00	1065.91	3.81	0.00	3.81
UPPER +Y OUTBOARD WING	0.00	40.23	-14.78	544.30	222.97	0.00	0.00
UPPER +Y INBOARD WING	0.00	23.77	-14.78	544.30	222.97	0.00	0.00
LOWER +Y OUTBOARD WING	0.00	40.23	14.78	544.30	222.97	0.00	0.00
LOWER +Y INBOARD WING	0.00	23.77	14.78	544.30	222.97	0.00	0.00
UPPER -Y OUTBOARD WING	0.00	-40.23	-14.78	544.30	222.97	0.00	0.00
UPPER -Y INBOARD WING	0.00	-23.77	-14.78	544.30	222.97	0.00	0.00
LOWER -Y OUTBOARD WING	0.00	-40.23	14.78	544.30	222.97	0.00	0.00
LOWER -Y INBOARD WING	0.00	-23.77	14.78	544.30	222.97	0.00	0.00
RESOURCES MODULE							
+Y	0.00	2.74	0.00	7333.64	17.84	5.20	17.84
-Y	0.00	-2.74	0.00	7333.64	17.84	5.20	17.84
UPPER BOOM	0.00	0.00	-38.10	150.32	1.35	0.00	1.35
LOWER KEEL	0.00	0.00	26.06	660.91	0.54	0.54	0.15
UPPER KEEL	0.00	0.00	-19.20	477.27	8.13	0.54	0.45
FIVE BAY SQUARE PLATFORM	0.00	0.00	57.61	637.73	3.76	0.75	0.75
POS. Y KEEL EXT.	0.00	5.49	72.69	220.45	0.90	0.90	0.15
NEG. Y KEEL EXT.	0.00	-5.49	72.69	220.45	0.90	0.90	0.15
POS. Y LOWER BOOM	0.00	12.34	76.81	165.45	0.60	0.15	0.60
NEG. Y LOWER BOOM	0.00	-12.34	76.81	165.45	0.60	0.15	0.60
HORIZONTAL HAB	2.29	0.00	78.18	14562.73	ERR	14.31	48.12
VERTICAL HAB	5.49	0.00	78.41	12303.18	ERR	48.12	14.31
LOGISTICS	5.49	0.00	59.13	17192.27	ERR	48.12	14.31
HORIZONTAL LAB	-2.29	0.00	67.21	24679.55	ERR	14.31	48.12
VERTICAL LAB	-5.49	0.00	74.98	21685.91	ERR	48.12	14.31
POWER SYSTEM RADIATORS							
POS Y	-8.99	16.46	0.00	170.45	0.00	0.00	27.87
NEG Y	-8.99	-16.46	0.00	170.45	0.00	0.00	27.87
MODULE RADIATORS							
POS Y	0.00	23.62	55.78	681.82	0.00	0.00	111.48
NEG Y	0.00	-23.62	55.78	681.82	0.00	0.00	111.48
FUEL TANK	0.00	0.00	52.12	8505.00	0.00	0.00	0.00
REMOTE MANIPULATOR	0.00	0.00	31.55	909.09	0.00	0.00	0.00

Table 4.3.3.4-2 - STATION ELEMENT LOCATIONS, MASSES AND AREAS

POWER TOWER REFERENCE CONFIGURATION

ITEM	CENTER OF MASS COORDINATES METERS X	Y	Z	WEIGHT kGS
PAYLOADS				
TDM 2010	2.74	-19.20	0.00	700.00
SAA 0005	0.00	0.00	-39.70	1100.00
SAA 0006	0.00	-10.97	-39.70	661.36
SAA 0009	0.00	10.97	-39.70	743.64
SAA 0207	0.00	4.57	-39.70	1603.64
TDM 2420	-2.74	3.66	-38.10	500.91
TDM 2410	-2.74	-3.66	-38.10	500.91
TDM 2510	7.62	0.00	-39.70	2800.00
TDM 2560	0.00	9.14	-9.14	3206.82
TDM 2060	-15.24	0.00	51.82	4000.18
TDM 2070	-15.24	0.00	51.82	0.00
TDM 2310	0.00	-3.05	51.82	2505.45
COM 1203	0.00	-6.86	75.44	5000.00
TDM 2260	0.00	10.67	78.18	300.45
SAA 0201	-5.49	0.00	80.92	392.27
SAA 0207	-7.62	0.00	78.18	1603.64
TDM 2570	-4.57	-4.57	16.76	8016.82

Table 4.3.3.4-3 - PAYLOAD LOCATIONS AND MASSES

POWER TOWER REFERENCE CONFIGURATION

ITEM	CENTER OF MASS COORDINATES METERS X	Y	Z	WEIGHT kGS
SATELLITE SERVICING				
SERVICING ATTACH(ESS-TYPE)	0.00	6.71	-7.62	1704.55
STORAGE ATTACHMENT	0.00	0.00	0.00	0.00
REFUELING ATTACHMENT	[illegible]	6.10	48.16	1704.55
OMV STORAGE ATTACHMENT	[illegible]	-7.62	60.35	1136.36
OMV	0.00	-7.62	60.35	5681.82
ORU STORAGE BOXES	0.00	0.00	-2.13	1980.64
O[illegible] STORAGE BOXES	0.00	0.00	-2.13	39[illegible].73
2 TOOL BOXES(REFUEL. AREA)	0.00	0.00	47.12	284.09
INSTRUMENT STORAGE SHELTER	-0.91	3.35	16.15	4545.45
FUEL STORAGE TANKS	0.00	0.00	52.12	2840.91
FUEL STORAGE TANK	0.00	0.00	49.38	113.64
SATELLITE CHECKOUT EQUIP	0.00	0.00	0.00	0.00
OMV CHECKOUT EQUIP	0.00	0.00	0.00	0.00
SET ORU'S	0.00	0.00	-2.13	1818.18
SET OF TOOLS	0.00	0.00	-2.13	272.73
OMV KITS	0.00	7.92	62.79	3977.27
SET INSTRUMENTS	-0.91	3.35	16.15	909.09
FUELS/GAS[illegible]	0.00	0.00	52.12	10454.55
SATELLITE	0.00	7.01	-17.98	1818.18
SATELLITE	0.00	-7.01	-17.98	2272.73

Table 4.3.3.4-4 - SERVICING ITEM LOCATIONS AND MASSES

4.3.3.5 Performance Assessment

4.3.3.5.1 Rigid body control dynamics

The objective of this analysis was to determine the control requirements for Space Station operations including attitude control and orbit maintenance as a function of the natural on-orbit dynamic environment. The dynamic environments simulated included gravity gradient torques, aerodynamic drag, and aerodynamic torques. System requirements for the baseline configuration were determined for parametric variations of altitude and mass properties.

4.3.3.5.1.1 Simulation capabilities

Two independently developed programs with similar capabilities were used to predict the on-orbit dynamics of the Space Station, the Space Station Dynamics (SSDYNAMICS) and the Rigid Body Control Dynamics (RCD) program (References 12 through 16).

The SSDYNAMICS program, developed at JSC, initializes with the Space Station on a user prescribed orbit and calculates the time histories of altitude and attitude as a function of the dynamic environments encountered. A simple, feed back control system is modeled. The associated control torque time histories and momentum storage requirements for maintaining the Station at a prescribed orientation or maneuver, e.g., solar inertial or earth fixed (LVLH), are computed. The control torques are computed as a linear function of the attitude and rate errors of the Space Station (Figure 4.3.3.2-1). CMG torque limits are imposed where applicable. Motion of the solar arrays and radiators relative to the main body is included in the analysis. Mass and aerodynamic properties for the Space Station are computed as a function of the Space Station's individual components. The atmosphere density is computed as a function of the time of year, altitude, longitude, latitude, F10.7 solar flux index, and Ap geomagnetic index. The atmosphere is simulated as rotating with the Earth's surface. The equations of motion which are based on Newton's second law of motion and Euler's moment equations are solved using a variable step Runge-Kutta integration routine.

The Rigid Body Control Dynamics (RCD) program is a major software program within the Interactive Design and Evaluation of Advanced Spacecraft (IDEAS) computer-aided design and analysis system developed at Langley Research Center (Reference 16). RCD uses equation of motion essentially the same as those employed in SSDYNAMICS to compute on-orbit environmental and maneuver forces and torques at user-specified circular orbital altitudes and spacecraft orientation. Mass and aerodynamic properties are automatically computed and input by other IDEAS modules in RCD. Momentum storage and desaturation requirements, control system predesign control, and maneuvers are calculated for one orbit and for the spacecraft lifetime. Principal features of RCD are shown in Figure 4.3.3.5-1. Results from both the SSDYNAMICS and the RCD programs are summarized in Section 4.3.3.5.1.3.

4.3.3.5.1.2 Flight mode selection

The reference Space Station flies with its keel stabilized by CMG'S to a quasi local vertical local horizontal (LVLH) mode with its solar arrays rotating at orbital rate using the alpha joint drives, to point at the Sun. The Sun beta condition is handled with beta joint drives. The power booms are maintained perpendicular to the orbit plane to simplify the alpha and beta joint drive system. The configuration is pitched in the orbit plane to an average torque equilibrium attitude (TEA) to preclude momentum buildup in the pitch plane. The pitch plane gravity gradient torque varies over an orbit due to rotation of the solar arrays (Figure 4.3.3.5-8). The aerodynamic torque also varies over an orbit due to solar array rotation and due to the diurnal density variation. A gravity gradient drift flight mode was analyzed and discarded after several simulations since the atmospheric variations and solar array rotations caused a close to resonance pendulum motion of the keel body with peak to peak oscillations of approximate 13° at an altitude of 270 nmi and 19.5° at 250 nmi for the IOC mass condition.

4.3.3.5.1.3 Control requirements for the natural environment

Simulations of the six different mass properties conditions of the Station were conducted with parametric variations of the altitude to determine the minimum CMG control requirements for attitude maintainance. The simulation used a 2 sigma atmosphere condition and included short-term atmospheric density variations. Cyclic momentum, secular momentum, trim angle, number of CMG'S, and back-up RCS momentum storage were summarized per orbit in Table 4.3.3.5-1.

Four skylab equivalent CMG'S will suffice for the nominal altitude range of 250 to 270 nmi. However, if the altitude is reduced to 220 nmi, the IOC + Orbiter condition will require slightly over six skylab CMG'S. The secular momentum buildup per orbit shows the need to frequently desaturate the CMG'S. Magnetic torquers can be used in lieu of RCS to prevent structural response accelerations and contamination. Also another method of nulling the secular torque buildup about the Station x- and z-axes involves rotating the Station about an axes such that the solar array boom axis prescribes a tilted cone with respect to the orbit plane. The resulting secular gravity gradient torques can be made to null the secular aerodynamic torques. However this method was not used since it violates the requirement that the solar array booms remain perpendicular to the orbit plane. Future trade studies may be conducted to determine the cost trade-off between the two before-mentioned methods of secular momentum alleviation.

Data for the preceding table was derived from detailed flight simultations. Time histories from one such simulation of the IOC configuration including payloads at 250 nmi are presented in Figures 4.3.3.5-3 through 4.3.3.5-11. The coordinate system used with Figures 4.3.3.5-3 through 4.3.3.5-11 is shown in Figure 4.3.3.5-2.

C-3

This case was chosen since its mass properties resulted in large out of plane torque requirements. The simulation starts at the sub-solar point in the orbit at an inclination of 28.5° and with an Earth longitude of 0.0°. Thus, the local time is 12:00 noon, and the latitude is 28.5° (Fig. 4.3.3.5- 3). The station is pitched in the orbit plane to an angle of - 0.59° such that the pitch axes momentum requirements at the end of an orbit are zero (Fig. 4.3.3.5-4).

Figure 4.3.3.5-4 shows the inertial torque impulse vector with respect to an inertial coordinate system fixed in the plane of the orbit. It was concluded from this figure that 3.3 Skylab CMG's would be required for attitude hold.

This number was arrived at by vectorally adding the peak-to-peak y-axes momentum requirement of 12000 ft-lb-sec to the concurrent x and z resultant momentum requirement of 9000 ft-lb-secs resulting in a total momentum storage of 15000 ft-lb-sec. By properly initilizing the CMG's, they need only to store half of the peak-to-peak cylic momentum. Thus half of 15000 ft-lb-secs require 3.3 equivalent Skylab CMG's. Note that, during this same orbit, a slight secular momentum of 1200 ft-lb-sec accumulated about the orbit z-axes. As mentioned above, this secular momentum will require some form of desaturation.

The resultant drag force over an orbit is shown in Figure 4.3.3.5-5. The peak drag force is seen to be slightly over 0.2 pounds and occurs at a time of 0.16 orbits. Emergency RCS propellent requirements for the case of full CMG failure are calculated from the integration of the absolute value of the control torque vector with respect to the space station body axes. These torque impulse requirements are shown in Figure 4.3.3.5-6. It is seen that at the end of an orbit, the torque impulse vector is 30000,24000,1600 ft-lb-secs.

Figure 4.3.3.5-7 shows the control torque vector required to hold the station LVLH over the orbit. The oscilations about the y-axes are caused by the solar array articulations and variations in aerodrag.

The gravity gradient torques are shown in Figure 4.3.3.5-8. The y-axes component varies due to the solar array articulation. The aerodynamic torques are shown in Figure 4.3.3.5-9. The diurnal variation is apparent in the y-axes component of torque. The slight variation about the x-axis is caused by the aerodrag variations. The aerodynamic drag impulse is shown in Figure 4.3.3.5-10. The peak impulse at the end of an orbit is 600 lb-sec. The aerodensity occuring during the orbit is shown in Figure 4.3.3.5-11. The density varies by approximately a factor of 3 over the orbit

4.3.3.5.1.4 Control requirements during system failures

A series of simulations were conducted for the IOC configuration to understand the control requirements and resulting dynamics in the event that a Space Station system either failed or was temporarily shut down for service. Three types of cases were investigated. The first type being a case in which the CMG system fails to provide any control torque. The second type being a case in which the CMG system fails to provide control torque and a back-up system provides attitude hold about the z-axes. The third type is a series of cases where the alpha joint/joints were failed or shut down with the solar array either parallel or perpendicular to the main keel.

The motion of the IOC Space Station with respect to a rotating orbit coordinate system for the first failure mode case shown in Figure 4.3.3.5-12 and Figure 4.3.3.5-13 for five orbits. The IOC is seen to rotate about its keel in Figure 4.3.3.5-12 and to swing in a pendulum manner in Figure 4.3.3.5-13.

The motion of the IOC with respect to a rotating orbit coordinate system for the second failure mode case are shown in Figure 4.3.3.5-14 and Figure 4.3.3.5-15 for a 270 nmi and 250 nmi altitudes respectively. The aerodynamic orbital variation is seen to cause a near resonance pendulum motion with peak-to-peak oscillations of 13 and 19.5° for the 270 and 250 nmi altitudes, respectively.

The control requirements related to the third type of failure are summarized in Table 4.3.3.5-2. The CMG requirements for these cases include the angular impulse required to bring the solar arrays up to the Station's angular rate and the angular impulse caused by the natural environments. A singular alpha failure requires 0.25 Skylab CMG's to accelerate the arrays to station angular rates and 0.5 Skylab CMG's for a double alpha joint failure. These failure cases are only slightly more severe than for nominal operations. However the peak secular momentum (3000 ft- lb-sec) about the z-axes has increased greatly over the nominal case (1000 ft-lb-sec). This will greatly impact the desaturation requirements.

4.3.3.5.1.5 Control requirements for the induced environment

A few simple rigid body calculations were made in an attempt to develop insight into the effects of manipulator motion and crew motion on the momentum storage and torque requirements for the Station. For example, crew kick-off in the lower habitation module results in the following:

Approximate distance to CM = 90 ft
Peak torque = 25 lbs x 90 ft = 2,250 ft-lbs
Angular momentum = 0.5 X 2250 = 1125 ft-lbs
Peak torque output of 4 CMG's = 4 x 160 = 640 ft-lbs.

These results suggest that the CMG's will torque saturate during the later part of the assumed crew kick impulse. This is not necessarily a problem but it does indicate how easily crew motion can overpower the CMG's. Also the corresponding momentum storage requirement is equivalent to about one-half the capacity of a reference CMG. Now consider the case where a 32,000 pound payload is being removed from the Orbiter bay by the Station manipulator while the Orbiter is docked to the Station.

Assume:

Payload/manipulator tip velocity = 0.2 ft/sec
arm extended so that distance to CM = 100 ft

angular momentum to start motion:

H = mass x V x moment arm

H = 32000/32.2 X .2 X 100 = 20000 ft-lb-sec

Peak torque - depends on start-up transient

Equivalent CMG's - 20000/ 2300 = 8.6

The later calculations show that if it is assumed that the Station is to be controlled by CMG's during manipulator operations, then the number of CMG's required can be significant relative to those required of control due to the natural environment. Obviously additional work must be done in this area.

Other similar calculation were made such as moving payloads on the upper boom with the manipulator. These results were essentially the same as those reported above.

4.3.3.5.1.6 Controllability assessment during assembly

The on-orbit configurations of the reference configuration after the first two launches are shown in Figure 4.3.3.5.-16. The mass properties for each configuration are shown in Table 4.3.3.5-3. Simulation of the mass properties conditions for each of the configurations were conducted in order to determine the flight orientation and minimum CMG requirements for attitude maintenance. The simulation was performed at 270 nmi, using a constant density representative of the MSFC J-70 atmosphere, with +2 sigma solar flux predictions. The momentum storage requirements are shown in Table 4.3.3.5-4. This table shows that the momentum storage requirements for attitude control are well within momentum storage capabilities of 1 Skylab-equivalent CMG. The corresponding force, torque and momentum time histories for each configuration are shown in Figure 4.3.3.5-17.

4.3.3.5.1.6 Controllability assessment during assembly

The on-orbit configurations of the reference configuration after the first two launches are shown in Figure 4.3.3.5.-16. The mass properties for each configuration are shown in Table 4.3.3.5-3. Simulation of the mass properties conditions for each of the configurations were conducted in order to determine the flight orientation and minimum CMG requirements for attitude maintenance. The simulation was performed at 270 nmi, using a constant density representative of the MSFC J-70 atmosphere, with +2 sigma solar flux predictions. The momentum storage requirements are shown in Table 4.3.3.5-4. This table shows that the momentum storage requirements for attitude control are well within momentum storage capabilities of 1 Skylab-equivalent CMG. The corresponding force, torque and momentum time histories for each configuration are shown in Figure 4.3.3.5-17.

4.3.3.5.1.7 Control requirement sensitivities to payload placement

The configuration of the station was designed to maintain symmetry with respect to the x-z plane causing the z principal axis to lie in the x-z plane. However, the addition of payloads to the configuration causes the z principle axis to shift out of the orbit plane. A study was conducted to determine the control sensitivity of the IOC+PAYLOADS configuration to variations in the number of attached payloads. Mass properties were computed for each variation by removing a payload, computing the mass properties, replacing the payload, and then removing the next payload until all the payloads had been examined. The removal of the OTV Servicing Technology TDM 2570 caused the largest out-of-orbit plane shift of the z principal axis. The corresponding CMG control requirements were increased to 4.8 over the nominal case's 3.3 for an altitude of 250 nmi. These results indicate that a strategy should be developed for payload placement which minimizes the impact to the CMG control requirements.

4.3.3.5.1.8 Orbit maintenance

An investigation of the orbit altitude loss due to aerodynamic drag and of the impulse required to reboost was performed. Computations were made for predicted maximum and minimum atmospheric densities and station masses for the IOC timeframe. The following parameter values were used to characterize the system:

Drag coefficient = 2.3
Minimum mass 124,321 kg
maximum mass 204,892 kg
Reference area for minimum mass = 1475.8 m^2
Reference area for maximum mass = 1976 m^2
Atmospheric model - MSFC J-70 with + 2 sigma solar flux predictions. From NASA SP 8021, March 1983 revision.

Plots of decay time histories for initial altitudes ranging from 250 to 270 nmi are given in Figures 4.3.3.5-18 and 4.3.3.5-19. Detailed data from which the figure were derived is summarized in Table 4.3.3.5-5. The data on the table indicates that for an initial altitude of 270 nmi, the minimum mass Station will decay to an altitude of 250 nmi in 90 days when the 2 sigma high density atmosphere is assumed. These values, which are intended to be representative of the most severe condition, satisfy the present system design requirements.

Decay rates for the high mass Station (payloads and servicing items added) are lower than those for the low mass configuration.

Estimates of the reboost requirements are also shown on Table 4.3.3.5-5. These calculations were based on a Hohman transfer from the lower to the higher circular orbit. Reboost from an orbit of 250 nmi to an orbit of 270 nmi requires 2600 pounds of propellant having a specific impulse of 220 pound-second/pounds. The corresponding number for reboost following a 90-day decay from 270 nmi and maximum station mass is 3,564 pounds.

4.3.3.5.2 Flexible body attitude control dynamics

Both time and frequency response cases were run to study control system/flex body interaction. Bending mode data obtained from NASTRAN were run with a PID controller to obtain open and closed loop responses. The band width of the control system was set a decade below the lowest bending mode frequency. Sensor and actuator dynamics were not modeled. Configuration data are presented in Table 4.3.3.5-6.

4.3.3.5.2.1 Pointing accuracy

a. Crew disturbance torques Disturbance torques due to crew motion were simulated in all three axes of the Space Station The shape of the disturbance which is shown in Section 4.3.3.1 simulates an astronaut pushing off a module wall and then contacting the opposite wall.

Disturbance torque results are summarized in Table 4.3.3.5.7. Time histories of pointing error are shown in Figures 4.3.3.5-20, 4.3.3.5-21, and 4.3.3.5-22.

b. Berthing torques A disturbance torque resulting from the Orbiter berthing with the station is shown in Section 4.3.3.1.

The magnitude of the torque and the resulting pointing errors and settling times are summarized in Table 4.3.5-8. Time histories of pointing error are presented in Figures 4.3.3.5-23, 4.3.3.5-24, and 4.3.3.5-25.

c. Torque Limited Actuators The crew disturbance and berthing cases were repeated with the control actuator torque limited to

640 foot-pounds. This simulates the condition of using four CMG's where the maximum CMG output torque is 160 foot-pounds. Results for both crew and berthing disturbances are shown in Figure 4.3.3.5-26 to 4.3.3.5-31. Comparing these results with the nonlimited cases shows essentially no difference for crew disturbances. Berthing disturbances, on the other hand, produce attitude error excursions well in excess of 1° in the torque limited case.

d. Time Response Results Analysis of the pointing error plots indicates a stable control system. Damping is good and the control system is able to hold the pointing error in each axis to within the one degree pointing requirement for crew disturbance torques. Pointing errors greater than one degree are seen during berthing when the control torque is limited to 640 foot-pounds. The large settling times observed in all the runs are indicative of the low bandwidth controller being used to avoid control/flex body interaction.

4.3.3.5.2.2 Control stability

a. Frequency response Open loop frequency response results in the form of Bode Plots and Nichols charts are shown in Figure 4.3.3.5-32 - 4.3.3.5-37.

b. Frequency results Analysis of the frequency response plots shows acceptable rigid body gain margins and phase margins on the order of 60°. It is anticipated that the inclusion of actuator and sensor dynamics in the system model will lower the phase margin somewhat. The important point, however, is that the results show no control system/flex body interaction. Those bending modes which do show greater than unity open loop gain are at frequencies well above the control system bandwidth and are phase stable. Of course, it may be desirable to adopt some type of filtering to insure attenuation of these higher frequency modes.

4.3.3.5.2.3 RCS results

A set of RCS cases were run to simulate a center of gravity offset during the reboost maneuver. Figure 4.3.3.5-38 illustrates the assumed conditions.

The figure shows that during a four jet reboost, center of gravity offset results in a 4500 ft-lb pitch disturbance torque. The control system will off modulate jets to hold the Station within a $\pm$ 1.0° deadband. Eventually, the Station will reach a limit cycle condition about a 1.0° bias or attitude error. The frequency of these periodic firings can obviously excite structural bending modes. Using the configuration and autopilot data given below in Table 4.3.3.5-9, the autopilot hysteresis (h) was parameterized to determine its effect on limit cycle frequency. Using a value of .05° for hysteresis, Figure 4.3.3.5-39 shows torque acting on the vehicle as a function of time. Figure 4.3.3.5-40 illustrates in the phase plane the tendency of the station to limit cycle about a 1.0° error. The limit cycle frequency is approximately 0.05 Hz which is below the first bending mode frequency. Torque histories for several different values of hysteresis were used to drive an open loop flexible model of the station. These results appear in Section 4.3.3.5.3 of this report.

4.3.3.5.3 Transient response of IOC Space Station

The purpose of this section is to present and discuss preliminary results for the transient response of the reference IOC Station to the dynamic load cases of section 4.3.3.1. This includes responses to berthing, crew kick-off, RCS firing sequence for Station reboost and MRMS operations. The analytical model used to derive these results is discussed in Section 4.3.3.3. Section 4.3.3.3 also summarizes the modal character of the IOC and growth Stations. All the results of this section are derived from the MSC NASTRAN finite element structural analysis program of Reference 8.

4.3.3.5.3.1 Orbiter berthing

Berthing of the Orbiter to the Station is simulated by a 500-pound pulse of one second duration as described in Section 4.3.3.1. The pulse acts in the negative x direction (opposite to the flight path). To simulate misalignment of the Station berthing port to the Orbiter berthing port, the effect of an additional torque pulse about the Nadir axis is also examined. The duration of this torque is also 1 second and has a magnitude of 2100 foot-pounds. Station responses of particular interest due to berthing are the peak acceleration levels at the modules, center of the upper boom, tip of the transverse boom, and tip of an array mast. In addition, the response of a 5000-pound experimental payload on a soft spring support is also examined in order to assess the practicality of isolating payloads which have low acceleration level requirements. Accelerations were obtained for both zero and one-half of 1 percent of critical damping. Peak accelerations were tabulated for the conservative assumption of no damping.

It is observed that the longest period of vibration, of a mode whose shape involves motion of the berthing port, is, from Table 4.3.3.3-3, 7.2 to 10.4 seconds, depending on attachments. Thus, it takes from 3.6 to 5.2 seconds for locations furthest from the berthing port to experience the affect of the berthing load and 7.2 to 10.4 seconds for the berthing port to experience the wave reflected from these remote locations. This is considerably longer than the puls duration itself, so that the berthing loading specified herein approximates an impulse.

4.3.3.5.3.1.1 Solar arrays normal to the flight path

Time histories of acceleration response to berthing at the selected station locations when the solar arrays are oriented normal to the flight path are given in Figures 4.3.3.5-41 through 4.3.3.5-44. In these figures, one-half of one percent of critical damping is assumed in each mode. Table 4.3.3.5-10 summarizes the peak acceleration magnitudes at these locations for the conservative case of zero damping. Results are expressed in multiples of g, (i.e., 32.2 ft./sec./sec.), and as ratios to the acceleration of each location were the entire Station rigid. This ratio, denoted as the amplification factor, "a", provides a measure of flexibility effects.

Also of interest are the peak bending moments and torques occurring at the intersection of the keel extension and lower keel, transverse boom root, alpha-rotary joint, top of upper keel and array mast root. Table 4.3.3.4-11 contains peak moments and torques at these locations and their margins of safety (M.S.) using a factor of safety of 1.5. Strength values used in calculating the margins of safety are given in Table 4.3.3.3-1.

Peak acceleration magnitudes at the selected Station locations range from about 0.0012 to 0.016 g's. With the exception of the center of the upper boom, peak acceleration magnitudes change little with the attachment of payloads. Further, the amplification factor at the tip of the transverse boom is very large when payloads are present. This occurs, not because the flexible body acceleration is very large, but because in an entirely rigid station with payloads attached, the instantaneous axis of rotation due to the berthing load lies nearly along the transverse boom. Hence the rigid body rotation of this location is near zero.

The effect of damping on Station response can be appreciated by comparing the peak values in Table 4.3.3.5-10 where no damping is assumed with the peaks occuring in Figures 4.3.3.5-42 through 4.3.3.5-48 where 0.5 percent damping is assumed. For example, the peak response at the center of the upper boom is about 39 percent lower in the presence of damping. The large damping effect is basically attributed to the fact that the highest accelerations in the absence of damping occur long after the applied load is removed. Thus damping significantly reduces these later occurring peaks and earlier occurring peaks become the maximums experienced by the station. It is believed that the conservatism introduced by using the peaks in the absence of damping will compensate for the uncertainties presently existing in load definition. In future studies, when load definition is better, it would be desirable to use peak values in the presence of damping.

Inasmuch as no acceleration requirements during a berthing operation exist at the present time, little can be said about the acceleration levels themselves. However, the internal structural force resultants require consideration relative to allowable Station strength.

Table 4.3.3.4-11 contains peak bending moment and torque magnitudes at selected locations on the station. The attachment of payloads has little effect on the root bending moment at the upper end of the keel extension, but more than doubles the root bending moment of the transverse boom. Moreover, since the addition of payloads to the Station makes the

Station unsymmetric about the keel axis, torques exist in certain locations of the structure where they did not exist in the absence of the payloads.

Margins of safety assuming a factor of safety of 1.5 are also shown in the Table. All margins of safety exceed a value of 4.7 with the lowest values occurring at the root of the array mast. Thus, this location should be given careful scrutiny in future studies when the loading environment is better defined. Finally, the results indicate that the flexible deformation of the array mast, from root to tip, does not exceed 4 inches.

As discussed previously in this section, the Orbiter berthing load is essentially impulsive for this low frequency station. As shown in Reference 17, but for a different Station configuration, internal force resultants can be considerably higher when load duration is longer. To more accurately account for load duration as well as berthing load level, future studies should consider simulation of the berthing operation. This could be accomplished by using a simplified berthing mechanism model between the Station berthing port and an Orbiter with an initial berthing velocity. Presently, initial berthing velocities are believed to be about 0.1 to 0.3 feet per second.

4.3.3.5.3.1.2 Solar arrays normal to the Nadir axis

Response results when the solar arrays are oriented normal to the Nadir are summarized in Table 4.3.3.5-12. Results are similar to those of Table 4.3.3.3-10 for arrays oriented normal to the flight path. The primary difference between results for the two array orientations occurs at the tip of the array mast. The peak acceleration for arrays oriented normal to the flight path is up to 5 times larger than that for arrays oriented normal to the Nadir. This in turn indicates larger bending moments at the array mast root when arrays are oriented normal to the flight path.

4.3.3.5.3.2 Response to crew-kick-off motions

As described in the Section 4.3.3.1 of this document, two different crew-motion cases are examined, one based on crew kick-off across the lower habitation module, denoted as transverse crew motion and the other based on crew kick-off along the length of the habitation module, denoted as axial crew motion.

4.3.3.5.3.2.1 Solar arrays oriented normal to the flight path

Figures 4.3.3.5-45 thru 4.3.3.5-48 illustrate the time history of the acceleration response due to axial crew motion at the same selected Station locations as that used for berthing when the solar arrays are normal to the flight path. For the results presented in the figures, one-half of one percent critical damping is assumed in each mode.

Responses of the modules, as shown in Figure 4.3.3.5-45 clearly indicates the occurrence of the crew-kick-off. Table 4.3.3.5-13 summarizes the peak acceleration magnitudes of the responses for the conservative assumption of no damping. Acceleration values are given in multiples of g and amplification factors, are also provided. Internal loads are not provided for crew motion since they are not expected to be significant from a strength point of view.

Table 4.3.3.5-13 indicates that attaching payloads to the Station increases some accelerations and decreases others. In general, attaching the Orbiter to the Station decreases acceleration levels. Again, as explained under the berthing discussion, amplification factors when only payloads are attached to the Station are high along the transverse boom since it lies along the instantaneous axis for rigid body response to crew motion. When both payloads and Orbiter are attached to the Station, the instantaneous axis of rigid body rotation for these crew motions lies along the upper end of the vertically oriented solar array masts, so that amplification factors are very large for this location. Moreover, when payloads and/or orbiter are attached to the Station, there is generally little difference between the peak acceleration responses for transverse and axial crew motions.

4.3.3.5.3.2.2 Solar arrays oriented normal to the Nadir Axis

Response results when the solar arrays are oriented normal to the Nadir are summarized in Table 4.3.3.5-14. A comparison with the results shown in Table 4.3.3.5-13 indicates that, in general, the peak accelerations are lower when the arrays are normal to the Nadir for all configurations except when the payloads and Orbiter are attached to the Station.

Certain payloads require accelerations on the order of 10^{-5} g's. Table 4.3.3.4-13 indicates that this requirement is only met when the Orbiter is attached. Hence it will be necessary for payloads to have an isolation system which allows them to function at very low acceleration levels. To ascertain the possibility of doing this, the design of a soft spring isolator and its analytical verification is discussed in the next subsection.

4.3.3.5.3.3 Isolation of experimental payloads

In this section isolation from axial crew-kick-off is examined for a 5000-pound material science payload (COM 1203), located on the laboratory module. Isolation is provided by the use of a soft spring support for the experiment. Design of the isolation spring stiffness was determined by considering a simple two-mass problem. Response of a two-mass system to a step load applied to one-mass is well known. On the basis of this system and the requirement to have accelerations on the order of 10^{-5} g's, the spring stiffness is approximately 5 pounds per foot. Verifying this isolation design through NASTRAN analysis of the IOC Station with the isolated experiment included, but no attached payloads and Orbiter, reduced the acceleration level before isolation of 0.00015 g's to an acceptable level of 0.000012 g's .

4.3.3.5.3.4 Response to RCS reboost firing sequence

The RCS reboost firing sequence used for the transient analysis presented herein is presented in Section 4.3.3.1. Each RCS is assumed to deliver 75 pounds of thrust. Two thrusters are located on the lower boom and two are located 175.5 feet up the keel. The RCS thrusters are assumed to fire in pairs in a staggered sequence thus rocking the Station in the pitch plane.

Acceleration response time histories at selected Station locations are provided in Figures 4.3.3.5-49 through 4.3.3.5-52. Peak accelerations in g's are summarized in Table 4.3.3.5-15. These results assume one-half of one percent (.5%) of critical damping in each mode. The RCS firing sequence for reboost can occur over one or more orbits. However, in the present study only 500 seconds of this sequence is simulated. The results indicate that acceleration levels are not increasing in an unbounded fashion during the first 500 seconds of response history.

After about 500 seconds, the thrusters are turned off and the responses decay to zero. Furthermore, at Station locations far from the Station center of gravity, acceleration levels are generally higher due to reboost than due to other load cases. The peak acceleration at the upper boom, which accommodates several payloads, is nearly 0.025 g's. However, low-g requirements for experimental payloads are relaxed during reboost operations.

Table 4.3.3.5-16 displays peak bending moments and torques at selected locations on the Station. This table also displays margins of safety assuming a safety factor of 1.5 and strength values used to perform these calculations are given in Table 4.3.3.3-1. All margins of safety are positive. As in the case of berthing, the lowest margins occur at the roots of the array masts. They are somewhat lower for reboost than for berthing. (Compare Tables 4.3.3.5-11 with 4.3.3.5-16.) Since there is considerable freedom in designing the RCS firing sequence and in adjusting the magnitude of the RCS thrust, it is important in future studies to give special attention to margins of safety at the roots of the array masts during reboost as these may become dangerously low.

4.3.3.5.3.5 Response to MRMS operations

The Mobile Remote Manipulator System (MRMS) can produce loads during its operations when accelerating and decelerating payloads. In order to assess the Station dynamic response to MRMS operations, it is assumed that a worst case occurs when the MRMS is located at one end of the upper boom and the arm of the MRMS is decelerating a payload weighing 10,000 pounds. Deceleration loads for the Shuttle RMS arm as given in Reference 18 are used after being scaled for a 10,000-pound payload. This results in applied loads on the Station of 17.2 pounds in the flight path direction and 877 foot-pounds of twist torque about the negative Nadir axis. These loads are assumed to occur as step loads with a 1 second duration.

Peak acceleration magnitudes in g's and amplification factors (acceleration of flexible station at specified location ratioed to acceleration of rigid station at the same location) due to the MRMS operation, are summarized in Table 4.3.3.5-17. Acceleration responses at a habitation module, center of the upper boom, tip of transverse boom, and tip of an array mast are provided. All of these are quite high. Peak acceleration response at the end of the upper boom where the MRMS is located is found to be 0.0484 g's. This leads to an amplification factor of 33.9. The implication of this on the station performance is a potential control problem in performing MRMS activities on flexible portions of the Station when payloads are not attached to the Station. It is probable that the presence of payloads in the vicinity of the MRMS operation is what plays the major role in reducing acceleration responses at the MRMS.

4.3.3.5.3.6 Effect of bay size on peak accelerations

It is of interest to examine the sensitivity of the station responses to the choice of various bay dimensions. In Figure 4.3.3.5-53 this is accomplished by considering peak acceleration responses at four station locations, namely, application point of applied load, center of upper boom, tip of transverse boom, and tip of an array mast. Peak acceleration responses are presented for cubic bay dimensions from 1 to 15 feet as amplification factors ratioing the actual acceleration at some location to the corresponding acceleration at the same location were the entire Station fully rigid. These results were compiled using earlier values for CEI and CGJ which are now known to be too high. Nevertheless, these results provide trends and insight into the effect of variations of stiffness about the nominal 9-foot bay IOC reference configuration on Station response levels.

As the bay dimension is varied the Station truss structural stiffnesses change in proportion to the square of the bay dimension. Only the array mast and radiator mast stiffnesses remain unchanged.

In the limit as the bay dimension goes to zero, all parts of the Station become completely isolated from the rigid modules and are thus unaffected by any loads applied at the modules such as berthing and crew motion. Indeed the curves of Figure 4.3.3.5-53, for all but the berthing port, confirm this trend down to the bay dimension of 1 foot. The dashed continuation of the curves to the origin represents the consequence of this postulate. The curve for the berthing port, which is on one of the habitation modules, limits to a finite value as the bay dimension goes to zero. This value is the response of the rigid modules due to the applied load ratioed to the response of an entirely rigid Station due to the same applied load.

As the bay dimension is increased from zero, the response of the berthing port decreases because more of the mass of the Station is allowed to participate in the response. The decrease in this response is not dramatic, going from a ratio of about 1.55 at a 1 foot bay dimension to about 1.2 at a 15-foot bay dimension. Thus acceleration levels at the modules are not very sensitive to truss stiffness. On the other hand, the response of all other locations removed from the modules increases rapidly as the bay dimension is increased.

As the bay dimension is increased the modal frequencies of the Station undergo continuous change, while the frequency content of the applied pulse remains unchanged. For certain bay dimensions the Station modal frequencies will better lineup with the frequency content of the applied loads and hence larger responses will result. This helps to explain the rapid variations in response peaks as the bay dimension is varied.

For very large bay dimensions, the truss stiffness goes to infinity, and the curves should asymptote to limiting values. These limiting values will not be unity as though the Station were entirely rigid, since the array and radiator masts are always flexible. Interestingly, a 15- foot bay dimension does not in this sense closely approximate the rigid limiting case. In order to ascertain the limiting rigid structure values, results are shown for a 1000-foot bay dimension with the understanding that this is entirely an academic situation with no practical application. Figure 4.3.3.5-53 indicates that amplification ratios limit to values between 1 and 1.2 depending on Station location.

The fact that very small bay dimensions leads to essential isolation of parts of the Station has benefits and drawbacks. It is beneficial that the acceleration levels and hence also internal loads are kept low in certain parts of the Station removed from the module location. On the other hand, too small a bay dimension can have the drawback of rendering the same parts of the Station uncontollable. At present, no requirement to precisely control portions of the Station far removed from the modules has been identified. However, if in the future this became a requirement, control of remote locations using actuators at the modules would be essentially impossible when the bay dimension is too low and consequently an expensive distributed control system would probably be required. Thus it appears wise to avoid too low a bay dimension. Figure 4.3.3.4-53 indicates that the minimum bay dimension desirable in order to avoid possible future control problems on remote Station locations depends on the location itself. The worst location is the upper boom. Responses at this location fall off rapidly when the bay dimension is less than 8 feet. Consequently, it is wise to keep the bay dimension above 8 feet to avoid possible future control problems at locations remote from the modules.

4.3.3.5.3.7 Concluding remarks

Preliminary analytical results for the dynamic response of the reference Space Station configuration have been presented and discussed. Disturbances due to Orbiter berthing, crew motion, RCS reboost firing sequence and MRMS operations have been examined. The Station with and without payloads and/or Orbiter was considered and arrays were oriented either normal to the flight path or normal to the Nadir. Time histories and summary tables of peak accelerations, amplification factors, peak bending moments and torques and margins of safety using a safety factor of 1.5 were presented and discussed.

Considering all four load cases, maximum peak accelerations occur at the tip of the array mast due to RCS reboost firing with a magnitude of 0.028 g's. This led to relatively high internal bending moments at the root of an array mast with a margin of safety of 4.13. Peak accelerations also at an array mast root due to berthing were not much lower and the corresponding margin of safety was 4.71. Since neither the RCS firing sequence for reboost nor the berthing loads are very well defined at present, future studies with better defined applied loads should give special attention to margins of safety at the root of the array masts. Furthermore, array mast flexible deformation from root to tip did not exceed about 4 inches.

It was found that crew motion leads to exceeding the 10^{-5} g order of magnitude acceleration requirement for the laboratory module. However, a soft spring lator was designed for a 5000-pound experimental payload and its performance was verified in the full finite element analysis. This led to an acceptable acceleration level of 0.000012 g's for the isolated experiment.

Operations of the MRMS at one end of the upper boom produced relatively large accelerations at the MRMS location itself. This needs future attention as it could lead to problems in controlling the MRMS operations.

An examination of the effect of bay dimension on Station response to Orbiter berthing indicated that the bay dimension would have to be extremely large to provide dynamic responses approximating the limiting case of a rigid keel. Even the use of erectable 15-foot bays would fall far short of achieving this. The reference configuration with its 9-foot bays is therefore a very flexible structure. It appears to be able to meet dynamic performance requirements as they are presently defined. However, it is noted herein, that if additional performance requirements, such as on payloads located some distance from the station modules are later imposed on the Station, an additional control system would become necessary. Dynamic results indicate that a bay dimension much less than 9 feet would probably require a distributed control system.

4.3.3.6 Summary

Rigid body control requirements for si operational configurations of the IOC Station were determined. The configurations differed according to the number and type of items attached to the basic Station. These items were the Orbiter, payloads and servicing equipment. In establishing the requirements, the orbital altitude range was varied between 220 and

270 nmi and the corresponding atmospheres were characterized by 2 sigma high densities which varied over an orbit and rotated with the Earth. The analysis results indicate that four equivalent Skylab CMG's will provide sufficient momentum storage and torque to control the station in the natural environment (gravity-gradient and aerodrag) for the prescribed altitude range of 250 to 270 nmi. Some secular momentum buildup occurs on the roll and yaw axes which must be nulled. Three different methods for doing this were discussed. Secular momentum buildup is precluded on the pitch axis by flying with a small pitch angle (less than 10°) such that the angular momentum accumulated over an orbit due to gravity-gradient is the negative of that due to aerodynamic torque.

A limited number of system failure cases were investigated. Under the rather severe assumption that all control systems failed, the Station rotated about the keel and started to swing in a pendulum manner. A subset of this case assumed that all control was lost except for the z-axis (yaw). In this case, the Station oscillated in the orbit (pitch) plane. The case of a failed alpha joint was also investigated. The number of CMG's required to provide angular impulse to bring the solar arrays to the Station (keel) rotational rate varied from two to three depending on the assumed failure conditions. The associated secular momentum buildup was about three times the nominal value.

A cursory investigation of the control requirements due to induced environments was also performed. Torque generated by crew motion exceed the torque capability of four CMG's and requires about one-half the momentum storage capability. Torques and momentum due to docking/berthing and moving of payloads with the MRMS were estimated to be two to three times the capability of four CMG's. This would seem to indicate that control of the Station during berthing/docking and subsequent MRMS activity should be accomplished with the RCS or possibly the control requirements should be reduced.

Orbit decay and reboost calculations for the maximum and minimum mass IOC configuration were performed. For the highest atmosphere densities and most severe drag-to-mass ratios, the IOC Station decays from a 270 nmi orbit to 250 nmi in 90 days which meets the system requirement. The reboost propellant required for a 90-day decay period and highest mass configuration is about 3600 lbs. for a propellant having an ISP of 220 seconds. Flexible body attitude control dynamics studies were made for the IOC configuration without the Orbiter, payloads, and servicing items added.

The control frequency was set one decade below the lowest bending frequency. Vibration modes in the analyses were included to approximately one decade above the lowest mode. The peak pointing error due to crew-kick-off was less than 0.02° at the navigation base. The maximum settling time was 150 seconds. Similar disturbance responses for docking/berthing were less than 2° and 400 seconds, respectively.

Associated frequency response investigations show adequate gain and phase margins and no significant control system/flex body interaction was observed. A simple RCS control system was designed and torque histories which simulated a four RCS jet orbital reboost were also generated. These were used in an uncontrolled response analysis to investigate structural loading.

Structural dynamic models of both the IOC and Growth Station configurations were developed and utilized for the flexible control and transient response analyses. Continuous beam representations were employed for the lower, transverse and upper beams, keel extension, and lower and upper keels. Payloads were modeled as lumped masses and the station modules were treated as rigid beam members rigidly connected to one another in the racetrack arrangement. No attempt was made to model nonlinear effects such as those potentially associated with the large rotating joints. The frequency of the fundamental vibration mode of both the IOC and Growth Station is approximately 0.1 Hz. The lower frequency modes are associated with coupled motion of the solar arrays and keel for the IOC configuration and with keel twist for the growth configuration. Transient response results for disturbances from Orbiter berthing, crew-kick-off , RCS reboost and MRMS operations were developed. Maximum bending moments at the base of the solar array mast were well below the allowable. Array tip maximum deflections were on the order of 4 inches. Acceleration responses due to crew-kick-off exceeded the 1.0 E-5 g requirement in the laboratory module. Consequently, a soft spring isolator was designed for a 5000-pound experiment located in the laboratory module. This led to an acceptable acceleration level of 1.2 E-5 g for the isolated experiment.

An examination of the effect of bay dimension on Station response to Orbiter berthing indicates that the keel is far from rigid. Increasing the bay width to 15 feet did not significantly change the response levels.

TABLE 4.3.3.5-1 - SUMMARIES FOR IOC SPACE STATION PER ORBIT

ALT (NMI)	CYCLIC MOMENTUM VECTOR (ft-lb-s) x1000	SECULAR MOMENTUM VECTOR (ft-lb-s) x1000	TRIM ANGLE degree	CMG'S	RCS MOMENTUM VECTOR (ft-lb-s) x1000
IOC:					
270	0,9.5,.9	0,0,.9	-2.2	2.3	1,21,0
250	.2,13,1.3	.2,0,1.3	-3.2	2.8	2,30,0
220	1,23,2.7	1,0,2.7	-6.1	5.1	3,57,0
IOC +Orbiter:					
270	0,9.2,1	1,0,1	5.66	2.3	1.8,23,0
250	4,12.7,16	.2,0,1.6	4.88	3.4	2,35,0
220	1,22.5,3	0,0,3	2.53	6.3	4,69,0
IOC +Payloads:					
270	5,7.7,9	0,0,.9	-1.27	2.7	30,17,1
250	5,11,9	0,0,1.2	-1.59	3.3	30,24,1
220	5,17,8	0,0,2.5	-2.70	4.9	30,48,1
IOC +Payloads +Orbiter:					
270	3, 9,4.2	0,0,1	4.64	2.2	15,22,1
250	3,12,4	0,0,1.6	4.25	3.4	15,33,1
220	3.21,4	0 0,3	3.13	5.7	16,65,1
IOC +Payloads +Service:					
270	1.6,6.5,4	0,0,.7	-.893	2.2	13.5,15,.5
250	2.2, 9,3.8	0,0,1	-1.12	2.5	15.5,22,.5
220	2.5,15,3	0,0,2.2	-1.79	4.0	14,43,.5
IOC + Payloads +Service +Orbiter:					
270	2,8,2,3.8	0,0,1,0	4.97	2.3	10,20,2.5
250	2, 11,4.0	0,0,1.6	4.74	3.0	10,31,2.5
220	2.4,17.5,5	0,0,3.0	4.05	5.3	11,60,2.5

TABLE 4.3.3.5-2 - JOINT FAILURE SUMMARY RESULTS

ALT (NMI)	CYCLIE MOMENTUM VECTOR (FT-LB-S) x1000	SECULAR MOMENTUM VECTOR (FT-LB-S) x1000	TRIM ANGLE DEGREE	CMG'S	RCS MOMENTUM VECTOR (FT-LB-S) x1000
IOC - SINGLE ALPHA JOINT FAILURE PERPENDICULAR TO KEEL:					
270	5,5,5	2.5,0,1.7	-	1.7	1,20,1
IOC - SINGLE ALPHA JOINT FAILURE PARALLEL TO KEEL:					
270	1,9.5,3	0,0,3	-	2.45	2,20,7
IOC - DOUBLE ALPHA JOINT FAILURE PARALLEL TO KEEL:					
270	.4,9.2,1	.1,0,1	-	2.5	1.5,24,0

TABLE 4.3.3.5-3 - MASS PROPERTIES SUMMARY

	FLIGHT 1	FLIGHT 2 INERTIAS ($kg\text{-}m^2$ xE6)
IXX	1.85	11.2
IYY	.50	9.09
IZZ	1.41	2.17
IXY	1.00	0.00
IXZ	0.00	5.00
IYZ	0.00	0.00

TABLE 4.3.3.5-4 - MOMENTUM STORAGE REQUIREMENTS FOR REFERENCE CONFIGURATION AFTER FIRST TWO LAUNCHES

	CYCLIC MOMENTUM VECTOR (nmi-sec)	SECULAR MOMENTUM VECTOR (nmi-sec)	TRIM ANGLE (DEG)	CMG'S
FLIGHT 1	252,10,155	45,0,10	0	1
FLIGHT 2	29,0,34	4,0,6	5.01	1

TABLE 4.3.3.5-5 - ESTIMATED ORBIT DECAY AND REBOOST CHARACTERISTICS

Inital Altitude (nmi)	Final Altitude (nmi) Min/Max Mass	Reboost Propellant (Kg) Min/Max Mass	Decay Time (Days) Min/Max Mass
Maximum density - 8/1991 launch			
300	291/292	540/780	90/90
270	250/253.5	1190/1620	90/90
250	210/211	2420/2880	90/90
300	220/220	4840/7970	386/478
270	220/220	3000/4950	154/190
250	220/220	1790/2960	78/91
Minimum density - 4/1994 launch			
300	296/296.3	250/340	90/90
270	260/262	610/790	90/90
250	232/236	1070/1380	90/90
300	220/220	4840/7970	1965/2169
270	220/220	3000/4950	360/507
250	220/220	1790/2960	151/188

Reboost I_{sp} = 220 lb-sec/lb

TABLE 4.3.3.5-6 - CONFIGURATION DATA

Ixx -	47510000	Kg m^2
Iyy -	38570000	Kg m^2
Izz -	13160000	Kg m^2

No. of Modes:

X Axis	19
Y Axis	21
Z Axis	21

Frequency Range: 0 .09 - 1.07 Hz

Control Frequency: 0 .0094 Hz

TABLE 4.3.3.5-7 - DISTURBANCE TORQUE DUE TO CREW MOTION

DISTURBANCE TORQUE	X	Y	Z
T_D (NT - M)	2813	2813	508
POINTING ERROR	X	Y	Z
(DEGREES)	.014	.016	.008
SETTLING TIME	X	Y	Z
(SECONDS)	150	145	140

TABLE 4.3.3.5-8 - DISTURBANCE TORQUES DUE TO BERTHING

DISTURBANCE TORQUE	X	Y	Z
T_D	49506	49506	22373
POINTING ERROR	X	Y	Z
(DEGREE)	.42	.5	.69
SETTLING TIME (SECONDS)	400	400	400

TABLE 4.3.3.5-9 - AUTOPILOT AND CONFIGURATION DATA

K_D = 1.0 deg/(deg/s)

DB = 1.0 deg

h = 0 to 0.1 deg

T_L = 10,575 ft-lbs

T_D = 4,500 ft-lbs

I = 70,837,100 slug ft2

TABLE 4.3.3.5-10 - PEAK ACCELERATION RESPONSE IN G's AT SELECTED STATION LOCATIONS DUE TO ORBITER BERTHING FOR SOLAR ARRAYS ORIENTED NORMAL TO THE FLIGHT PATH

LOCATION / CASE	BERTHING PORT	CENTER OF UPPER BOOM	TIP OF TRANSVERSE BOOM	TIP OF ARRAY MAST
WITHOUT PAYLOADS	0.00476 a=1.70	0.0162 a=5.21	0.00460 a=4.48	0.00123 a=5.05
WITH PAYLOADS	0.00432 a=1.77	0.00280 a=2.44	0.00596 a=92.3	0.00126 a=17.5

KEY: a = amplification factor=(accel. of flexible station)/(accel. of rigid station)

TABLE 4.3.3.5-11 - PEAK BENDING MOMENTS AND TORQUES IN FOOT-LBS AT SELECTED STATION LOCATIONS DUE TO ORBITER BERTHING FOR SOLAR ARRAYS ORIENTED NORMAL TO THE FLIGHT PATH

LOCATION / CASE	KEEL EXT. ROOT	TOP OF UPPER KEEL	TRANSVERSE BOOM ROOT	ALPHA JOINT	ARRAY MAST ROOT
WITHOUT PAYLOADS	Moment: 6930. M.S.=7.15	Moment: 0.004 M.S.=*	Moment: 2890. M.S.=18.5	Moment: 1172. M.S.=#	Moment: 406. M.S.=4.71
	Torque: 350 . M.S.=67.6	Torque: 0.0 M.S.=*	Torque: 1009. M.S.=22.8	Torque: 1008. M.S.=#	Torque: 0.16 M.S.=865
WITH PAYLOADS	Moment: 6990. M.S.=7.08	Moment: 248. M.S.=227.	Moment: 7120. M.S.=6.93	Moment: 3170. M.S.=#	Moment: 381. M.S.=5.09
	Torque: 510. M.S.=46.1	Torque: 728. M.S.=32.	Torque: 2500. M.S.=8.6	Torque: 2400. M.S.=#	Torque: 0.0 M.S.=*

KEY: M.S.=Margin of Safety=(strength)/(1.5 x moment or torque) - 1

*: Margin of Safety exceeds one million

#: Allowable strength for alpha joint presently unavailable

TABLE 4.3.3.5-12 - PEAK ACCELERATION RESPONSE IN g's AT SELECTED STATION LOCATIONS DUE TO ORBITER BERTHING FOR SOLAR ARRAYS ORIENTED NORMAL TO THE NADIR

CASE \ LOCATION	BERTHING PORT	CENTER OF UPPER BOOM	TIP OF TRANSVERSE BOOM	TIP OF ARRAY MAST
WITHOUT PAYLOADS	0.0044 a=1.52	0.0125 a=4.46	0.00275 a=3.09	0.00275 a=3.09
WITH PAYLOADS	0.0041 a=1.71	0.0027 a=1.14	0.0056 a=18.1	0.0056 a=18.1

KEY: a=amplification factor=(flexible response)/(rigid station response)

TABLE 4.3.3.5-13 PEAK ACCELERATION RESPONSE IN g's AT SELECTED STATION LOCATIONS DUE TO AXIAL AND TRANSVERSE CREW KICK-OFF FOR SOLAR ARRAYS ORIENTED NORMAL TO THE FLIGHT PATH

	LOCATION	HABITATION MODULE	CENTER OF UPPER BOOM	TIP OF TRANSVERSE BOOM	TIP OF ARRAY MAST
WITHOUT PAYLOADS AND ORBITER	AXIAL KICK	1.49×10^{-4} a=1.23	5.37×10^{-4} a=1.0	1.5×10^{-4} a=2.9[illegible]	5.47×10^{-4} a=4.46
	TRANS. KICK	2.35×10^{-4} a=1.54	8.07×10^{-4} a=5.2	1.51×10^{-4} a=2.94	5.16×10^{-4} a=4.23
WITH PAYLOADS	AXIAL KICK	1.70×10^{-4} a=1.39	1.25×10^{-4} a=2.18	2.68 x [illegible] a=82.7	5.18×10^{-4} a=17.2 ONLY
	TRANS. KICK	1.86×10^{-4} a=1.52	1.42×10^{-4} a=2.47	3.10×10^{-4} a=95.4	6.89×10^{-4} a=19.1
WITH ORBITER	AXIAL KICK	5.68×10^{-5} a=1.08	1.88×10^{-4} a=63.1	6.77×10^{-5} a=4.27	2.03×10^{-4} a=55.9 ONLY
	TRANS. KICK	5.34×10^{-5} a=1.01	1.92×10^{-4} a=64.3	7.42×10^{-5} a=4.69	2.54×10^{-4} a=69.8
WITH PAYLOADS AND ORBITER	AXIAL KICK	4.91×10^{-5} a=1.0	2.27×10^{-5} a=3 26	9.78×10^{-5} a=8.08	2.01×10^{-4} a=94.6
	TRANS. KICK	4.66×10^{-5} a=0.94	1.16×10^{-5} a=1.67	5.50×10^{-5} a=4.5	1.62×10^{-5} a=76.2

KEY: a=amplification factor=(accel. of flexible station)/(accel. of rigid station)

TABLE 4.3.3.5-14 PEAK ACCELERATION RESPONSE IN g's AT SELECTED STATION LOCATIONS DUE TO AXIAL CREW KICK-OFF FOR SOLAR ARRAYS ORIENTED NORMAL TO THE NADIR

	LOCATION	HABITATION MODULE	CENTER OF UPPER BOOM	TIP OF TRANSVERSE BOOM	TIP OF ARRAY MAST
WITHOUT PAYLOADS AND ORBITER	AXIAL KICK	1.45×10^{-4} a=1.58	2.6×10^{-4} a=2.89	4.0×10^{-4} a=4.44	3.5×10^{-4} a=3.4
WITH PAYLOADS	AXIAL KICK	1.35×10^{-4} a=1.96	1.17×10^{-4} a=1.77	2.1×10^{-4} a=3.09	2.5×10^{-4} a=3.42 ONLY
WITH ORBITER	AXIAL KICK	5.3×10^{-5} a=1.06	1.25×10^{-4} a=2.91	2.1×10^{-5} a=4.17	1.6×10^{-4} a=2.60 ONLY
WITH PAYLOADS AND ORBITER	AXIAL KICK	5.4×10^{-5} a=1.32	8.8×10^{-5} a=4.1	1.3×10^{-4} a=4.1	1.4×10^{-4} a=3.18

KEY: a=amplification factor=(accel. of flexible station)/(accel. of rigid station)

TABLE 4.3.3.5-15 - PEAK ACCELERATION RESPONSE IN g's AT SELECTED STATION LOCATIONS DUE TO RCS REBOOST FIRING SEQUENCE

CASE \ LOCATION	HABITATION MODULE	CENTER OF UPPER BOOM	TIP OF TRANSVERSE BOOM	TIP OF ARRAY MAST
WITHOUT PAYLOADS	0.0034	0.024	0.012	0.028
WITH PAYLOADS	0.0024	0.0062	0.0030	0.0037

TABLE 4.3.3.5-16 - PEAK BENDING MOMENTS AND TORQUES IN FOOT-LBS AT SELECTED STATION LOCATIONS DUE TO RCS REBOOST FIRING SEQUENCE

LOCATION / CASE	LOWER KEEL ROOT	ROOT OF LOWER BOOM	TRANSVERSE BOOM ROOT	ALPHA JOINT	ARRAY MAST ROOT
WITHOUT PAYLOADS	Moment: 9,134 M.S.=8.62	Moment: 12.6 M.S.=*	Moment: 3,460 M.S.=15.3	Moment: 2,800 M.S.=#	Moment: 452.3 M.S.=4.10
	Torque: 0 M.S.=*	Torque: 0 M.S.=*	Torque: 942.0 M.S.=24.7	Torque: 856.0 M.S.=#	Torque: 0 M.S.=*
WITH PAYLOADS	Moment: 15,820 M.S.=8.36	Moment: 20.8 M.S.=2700.	Moment: 5,260 M.S.=9.70	Moment: 2,750 M.S.=#	Moment: 357.0 M.S.=3.65
	Torque: 1,925 M.S.=9.26	Torque: 0 M.S.=0	Torque: 1,660 M.S.=13.6	Torque: 1,660 M.S.=#	Torque: 0 M.S.=*

KEY: M.S.=Margin of Safety=(strength)/(1.5 x moment or torque) - 1

*: Margin of Safety exceeds one million

#: Allowable strength for alpha joint presently unavailable

TABLE 4.3.5-17 - PEAK ACCELERATION RESPONSE IN G'S AT SELECTED STATION LOCATIONS DUE TO MRMS OPERATIONS ON THE UPPER BOOM

CASE \ LOCATION	HABITATION MODULE	AT MRMS	TIP OF TRANSVERSE BOOM	TIP OF ATTAY MAST
WITHOUT PAYLOADS	0.000550 a=5.7	0.0248 a=33.9	0.0102 a=22.0	0.00870 a=13.8
WITH PAYLOADS	0.00015 a=1.15	0.00030 a=1.25	0.0026 a=17.6	0.0050 a=24.2

KEY: a=AMPLIFICATION FACTOR=(ACCEL. RESPONSE OF FLEXIBLE STATION)/(ACCEL. RESPONSE OF RIGID STATION)

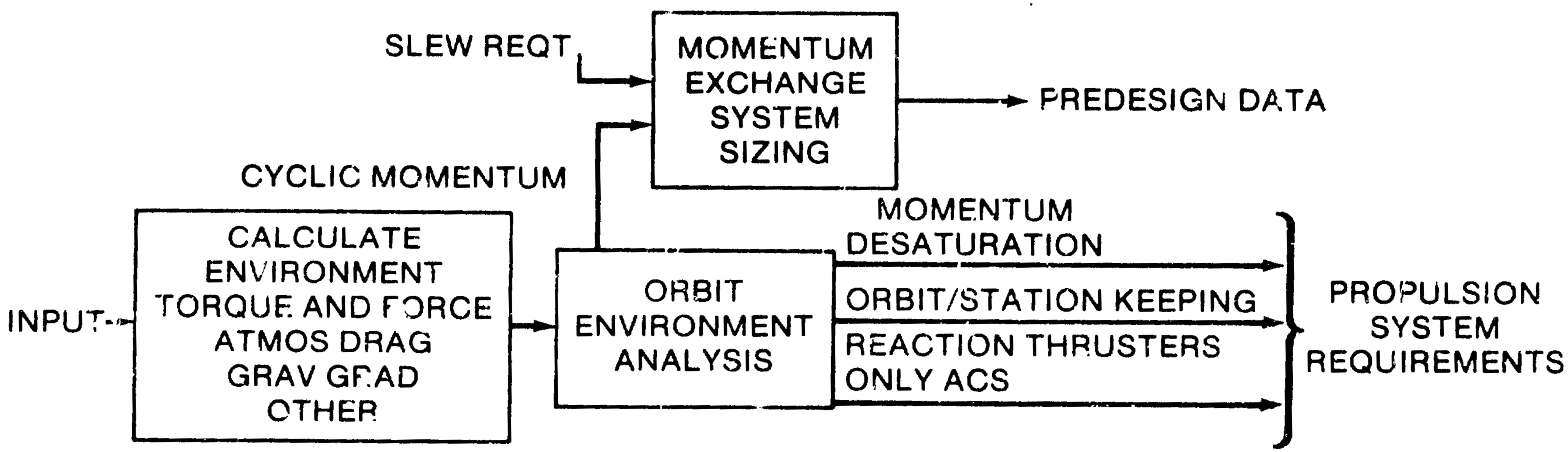


Figure 4.3.3.5-1 - RIGID-BODY CONTROL DYNAMICS MODULE (RCD)

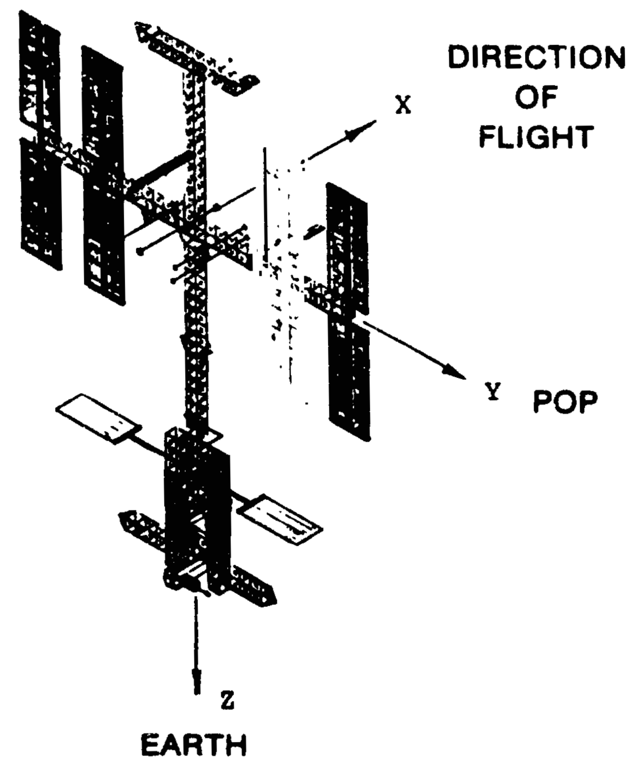


Figure 4.3.3.5-2 - COORDINATE SYSTEM FOR RIGID BODY SIMULATION

ORIGINAL PAGE IS
OF POOR QUALITY

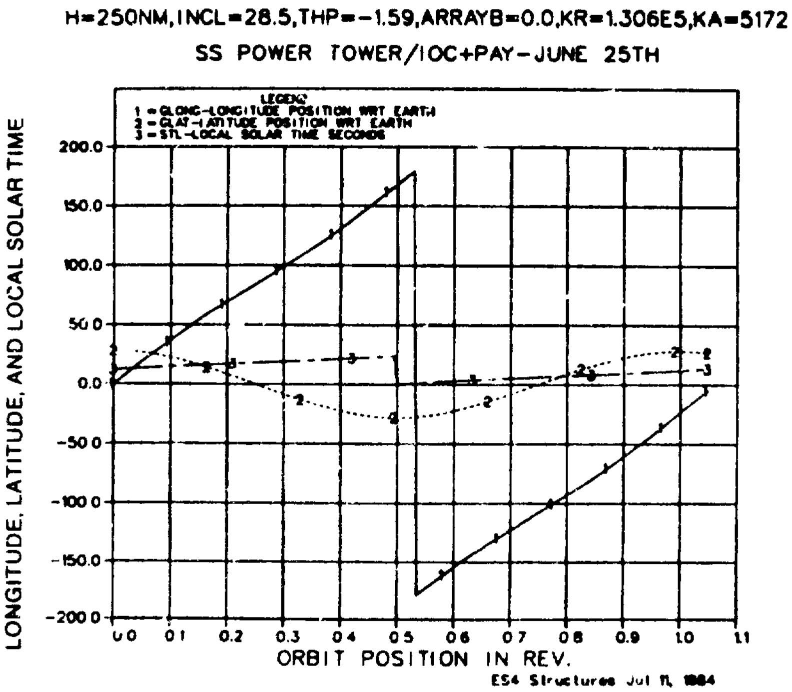


Figure 4.3.3.5-3 - SPACE STATION LONGITUDE, LATITUDE, AND LOCAL SUN TIME HISTORY

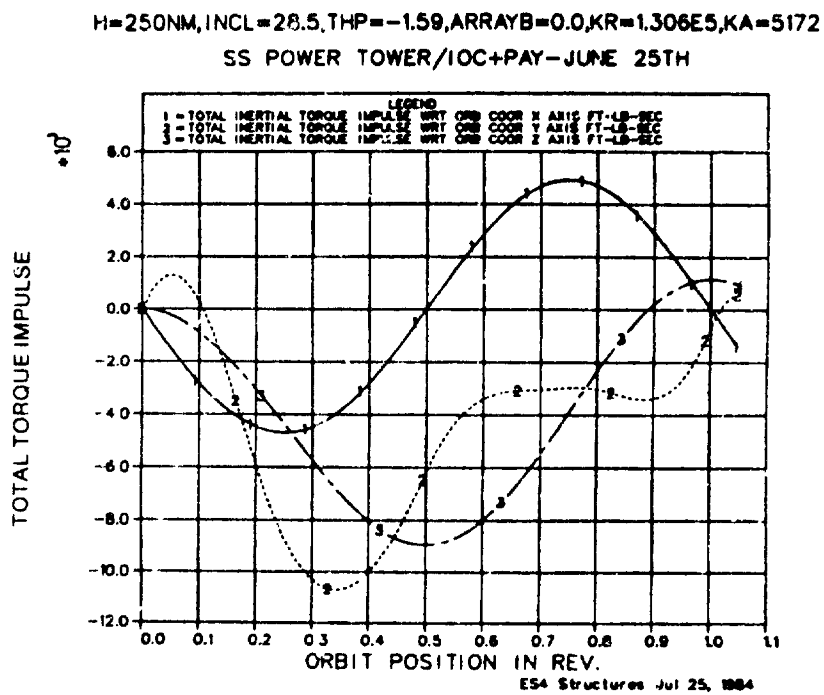


Figure 4.3.3.5-4 - TOTAL INERTIAL TORQUE IMPULSE HISTORY WRT AN INERTIALLY FIXED ORBIT COORDINATE SYSTEM

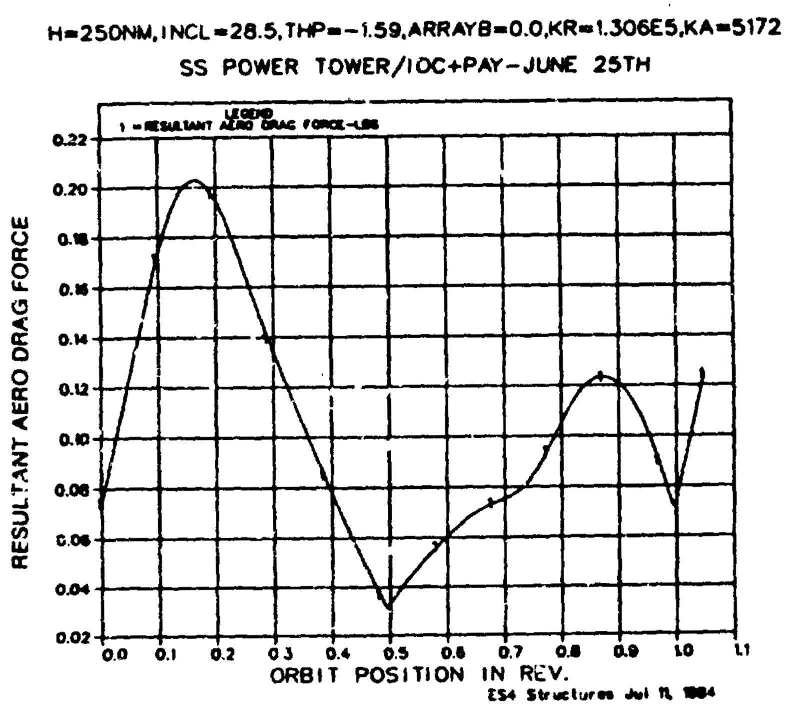


Figure 4.3.3.5-5 - RESULTANT AERODYNAMIC DRAG FORCE HISTORY WRT BODY COORDINATE SYSTEM

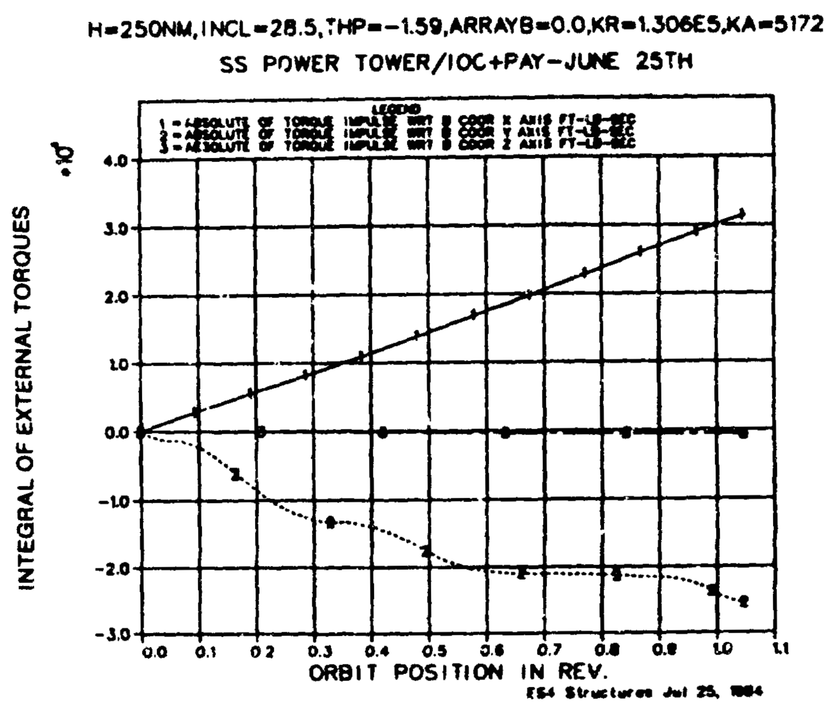


Figure 4.3.3.5-6 - ABSOLUTE INTEGRAL OF EXTERNAL TORQUES WRT BODY COORDINATE SYSTEM

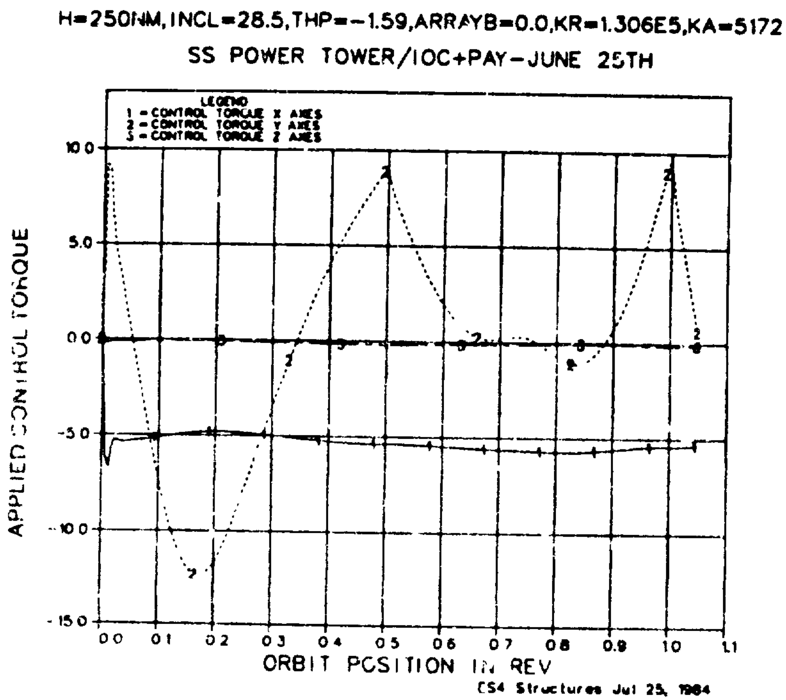


Figure 4.3.3.5-7 **APPLIED CONTROL TORQUE HISTORIES WRT BODY COORDINATE SYSTEM**

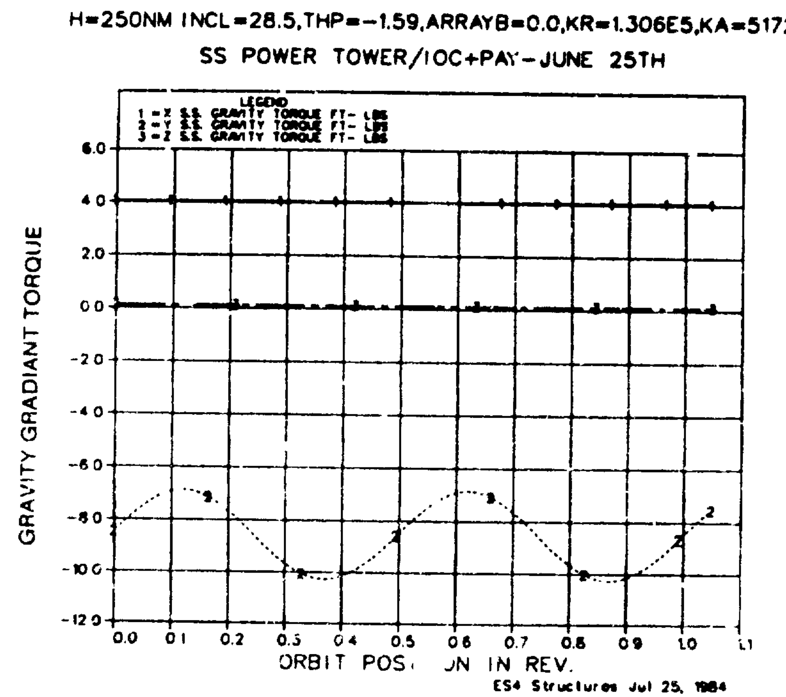


Figure 4.3.3.5-8 **GRAVITY GRADIENT TORQUE HISTORIES WRT BODY COORDINATE SYSTEM**

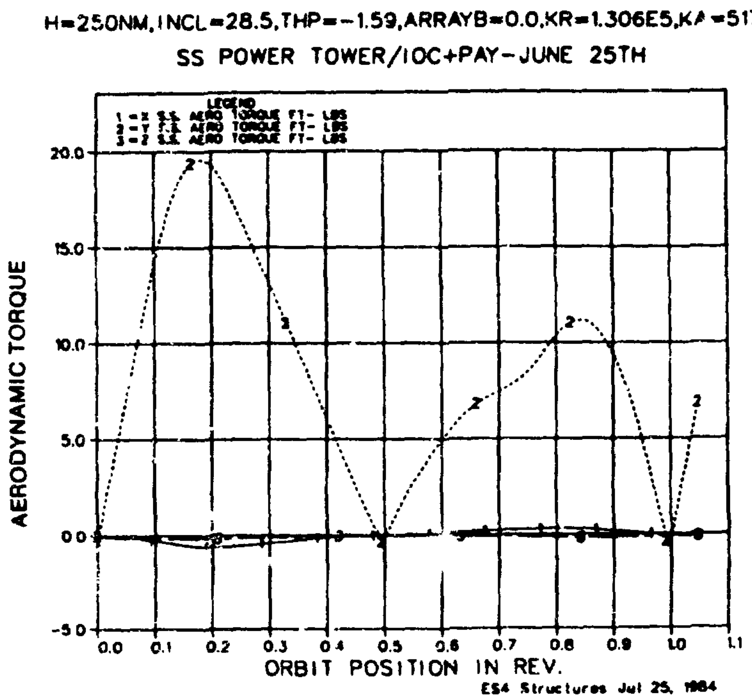


Figure 4.3.3.5-9 - AERODYNAMIC TORQUE HISTORIES WRT BODY COORDINATE SYSTEM

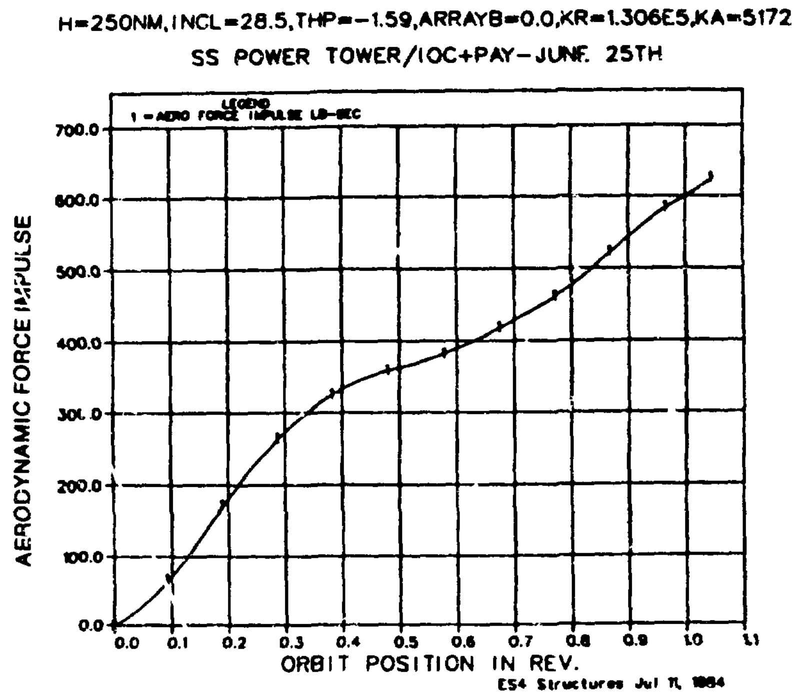


Figure 4.3.3.5-10 - TOTAL AERODYNAMIC DRAG IMPULSE HISTORY WRT BODY COORDINATE SYSTEM

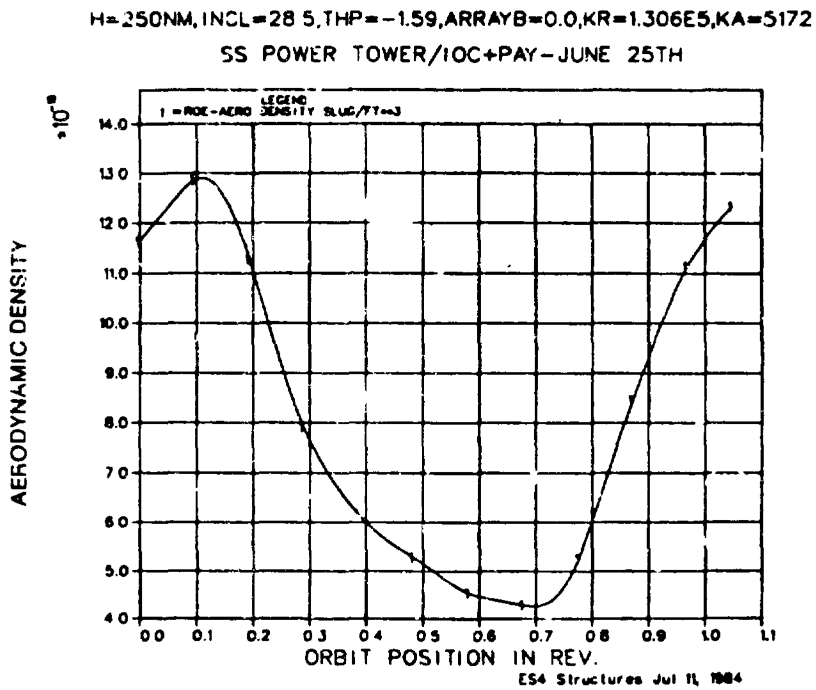


Figure 4.3.3.5-11 - AERODYNAMIC DENSITY AS A FUNCTION OF ORBIT POSITION

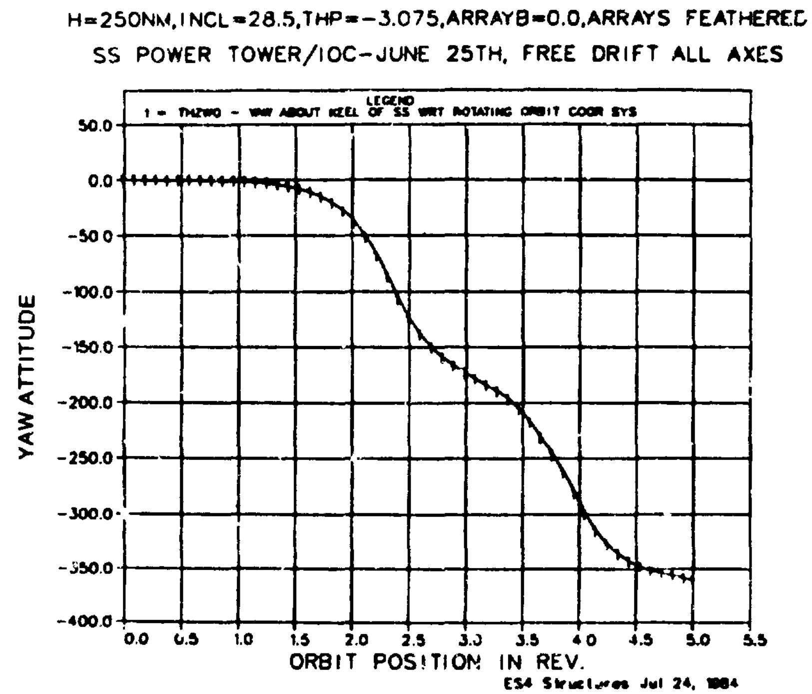


Figure 4.3.3.5-12 - YAW ATTITUDE HISTORY OF ROTATION WRT A ROTATING ORBIT COORDINATE SYSTEM

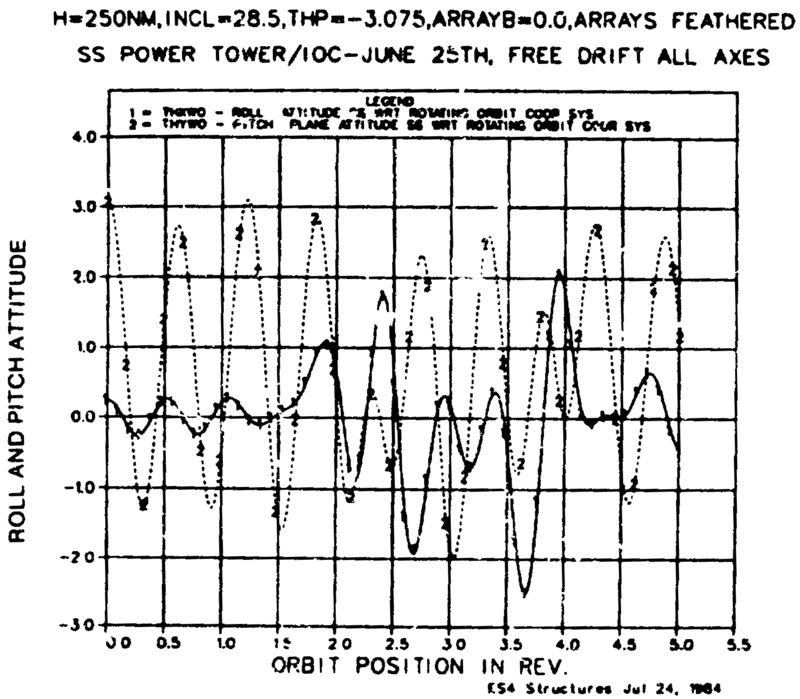


Figure 4.3.3.5-13 - ROLL AND PITCH HISTORY OF KEEL WRT A ROTATING ORBIT COORDINATE SYSTEM

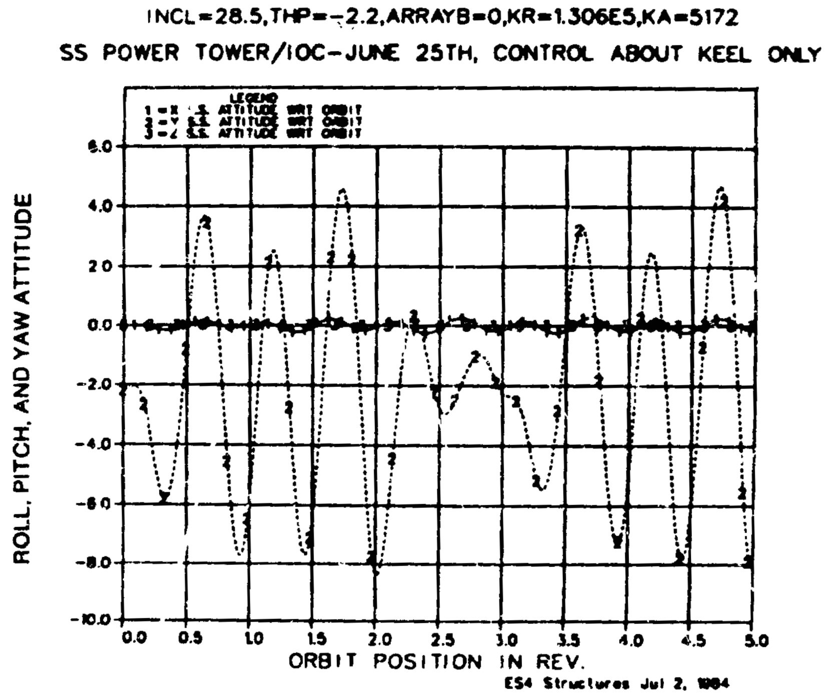


Figure 4.3.3.5-14.- ROLL, PITCH, AND YAW ATTITUDE HISTORY WRT A ROTATING ORBIT COORDINATE SYSTEM

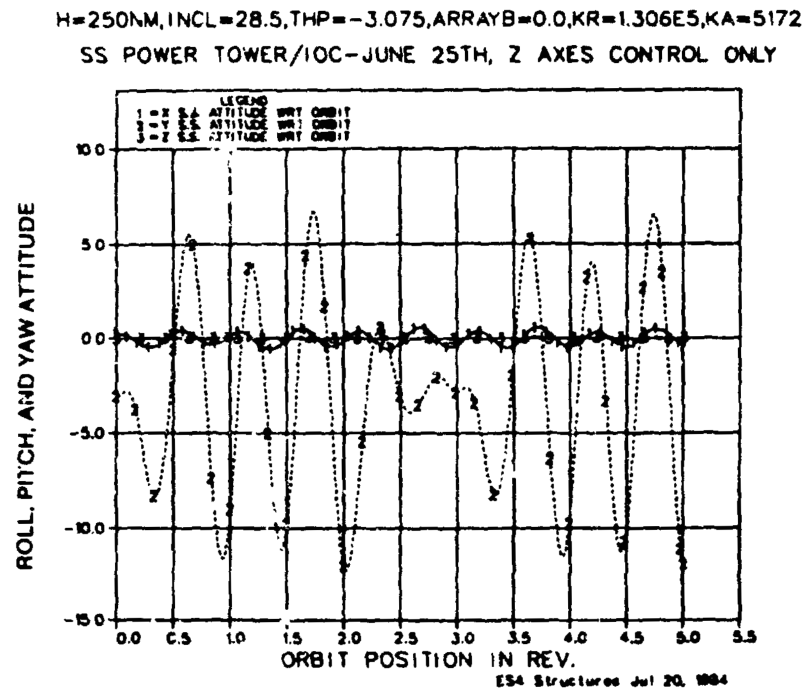
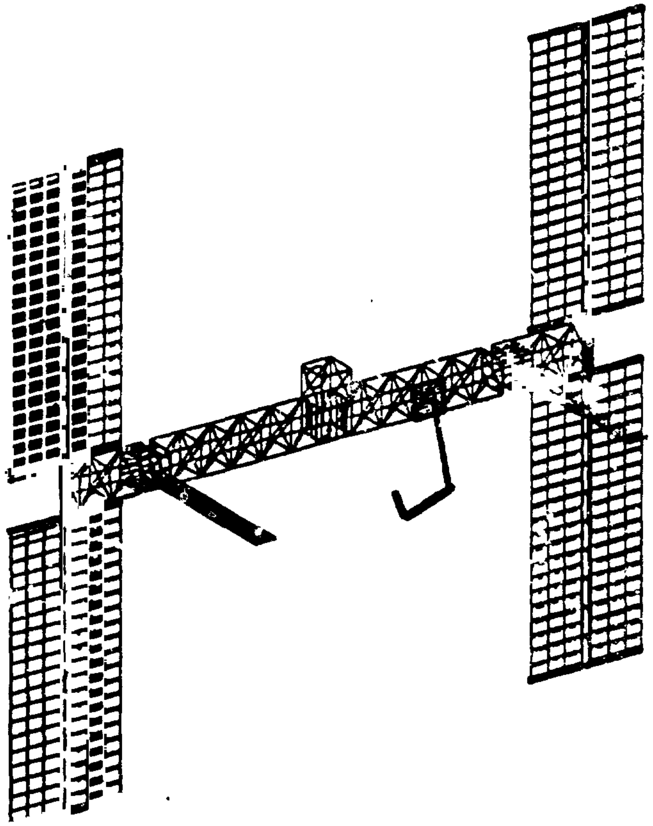
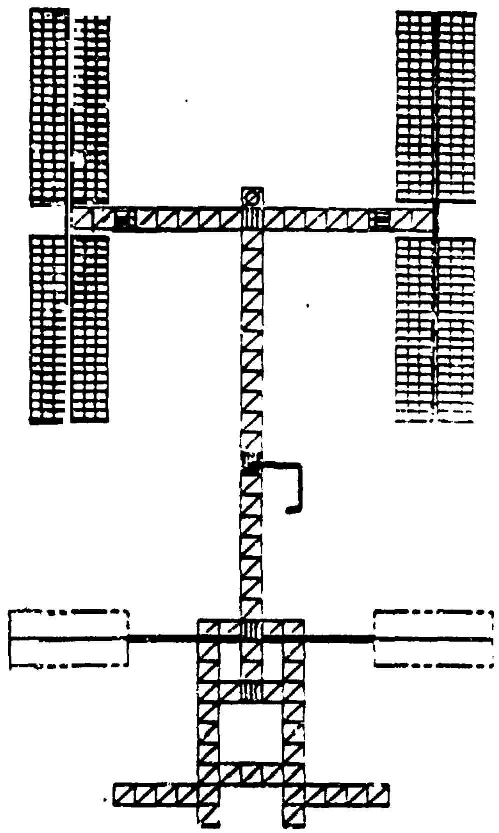


Figure 4.3.3.5-15 - ROLL, PITCH, AND YAW ATTITUDE HISTORY WRT A ROTATING ORBIT COORDINATE SYSTEM

OF POOR QUALITY



CONFIGURATION AFTER
FIRST FLIGHT



CONFIGURATION AFTER
SECOND FLIGHT

Figure 4.3.3.5-16 - BUILD-UP CONFIGURATIONS

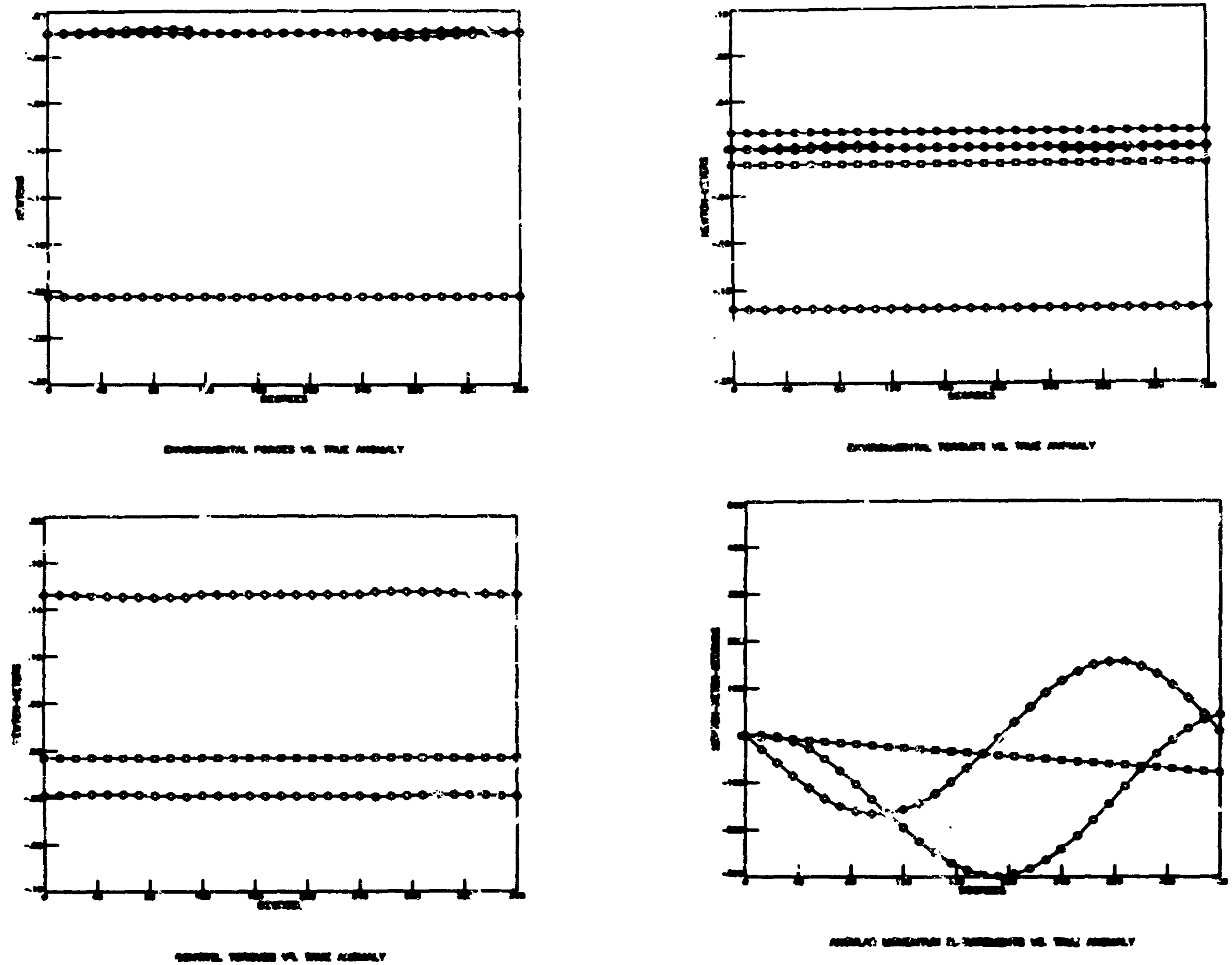


Figure 4.3.3.5-17a - FLIGHT 1 FORCE/TORQUE/MOMENTUM TIME HISTORIES

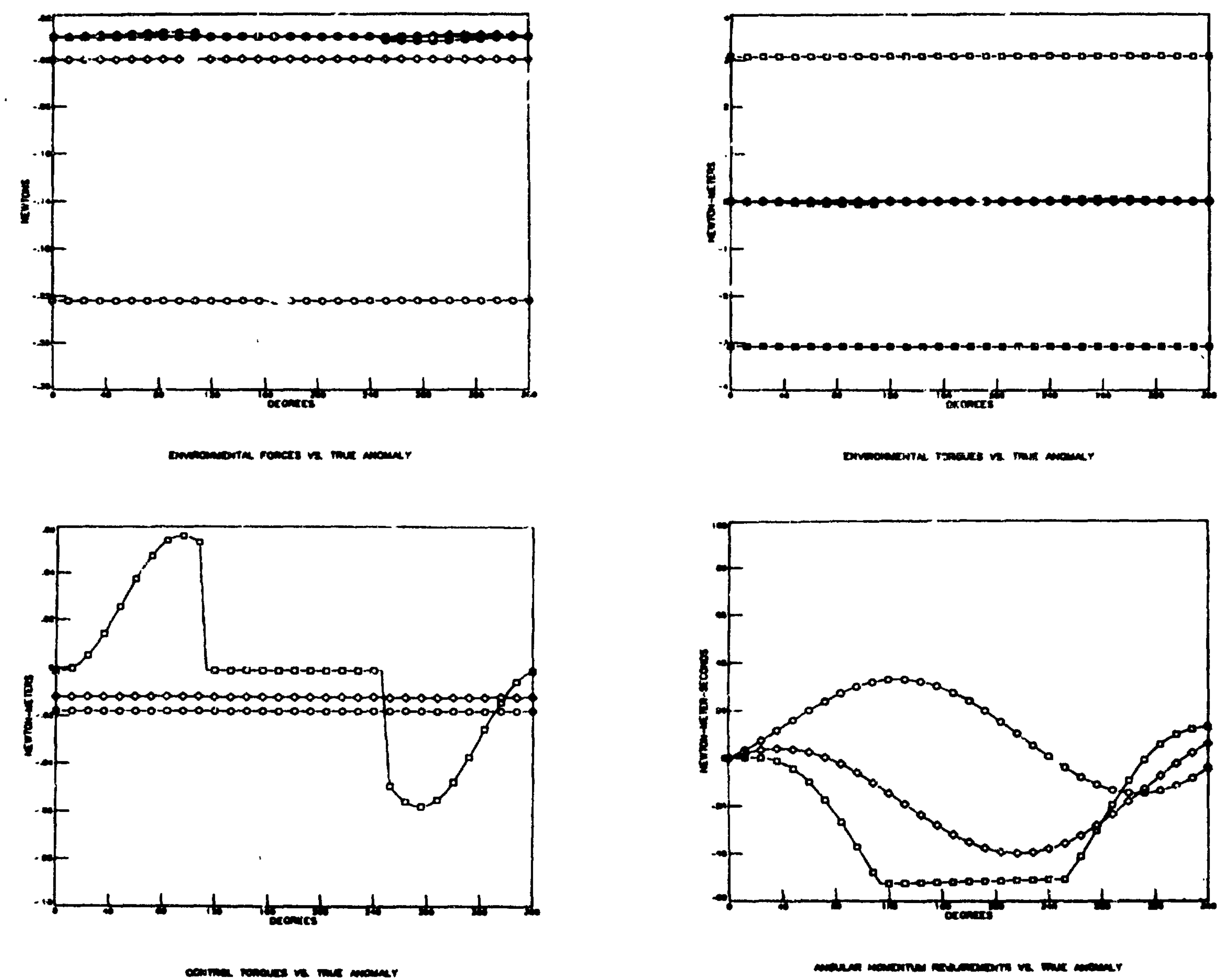


Figure 4.3.3.5-17b - FLIGHT 2 FORCE/TORQUE/MOMENTUM TIME HISTORIES.

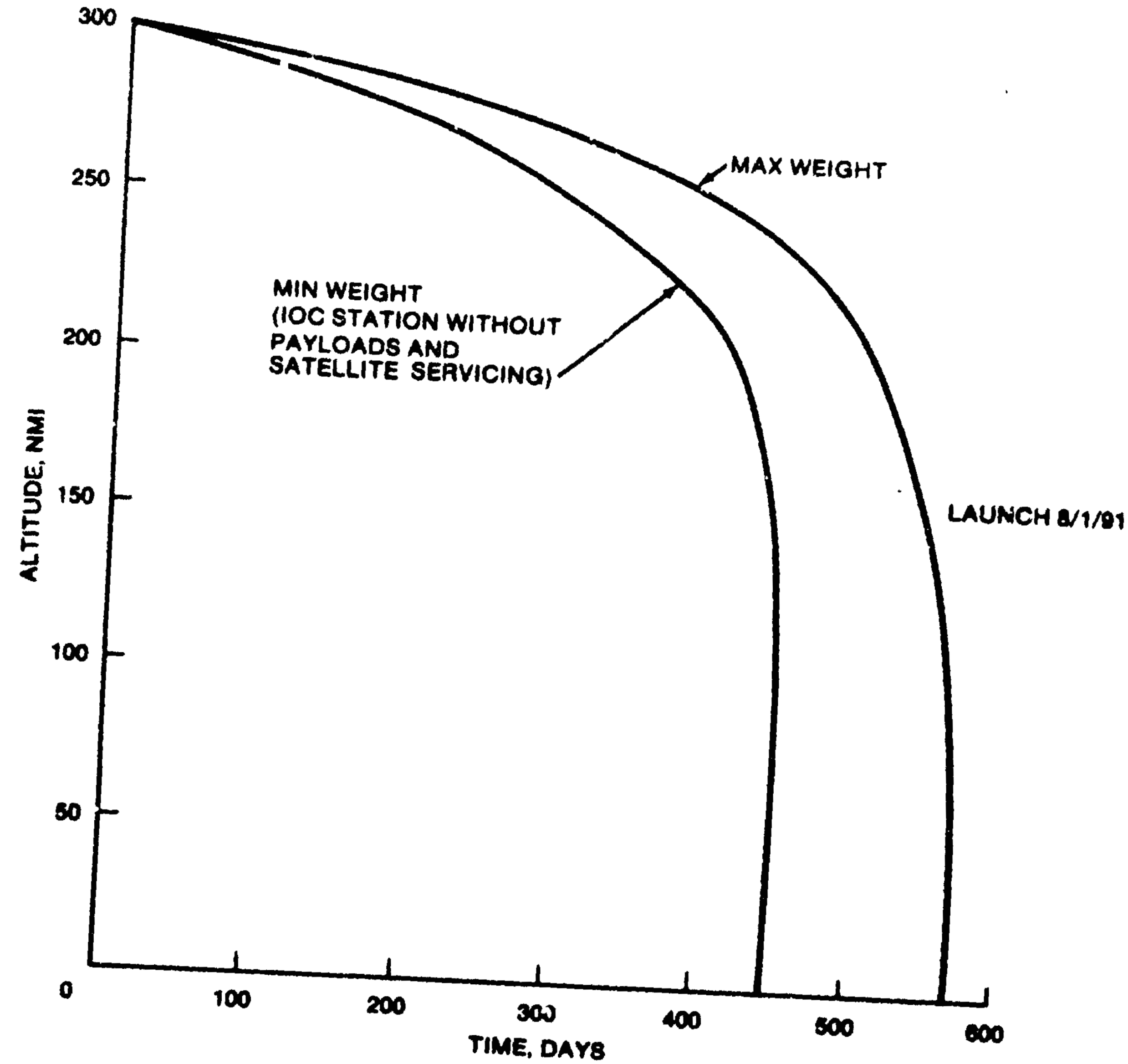


Figure 4.3.3.5-18 - ORBIT DECAY FOR MAXIMUM DENSITY ATMOSPHERE

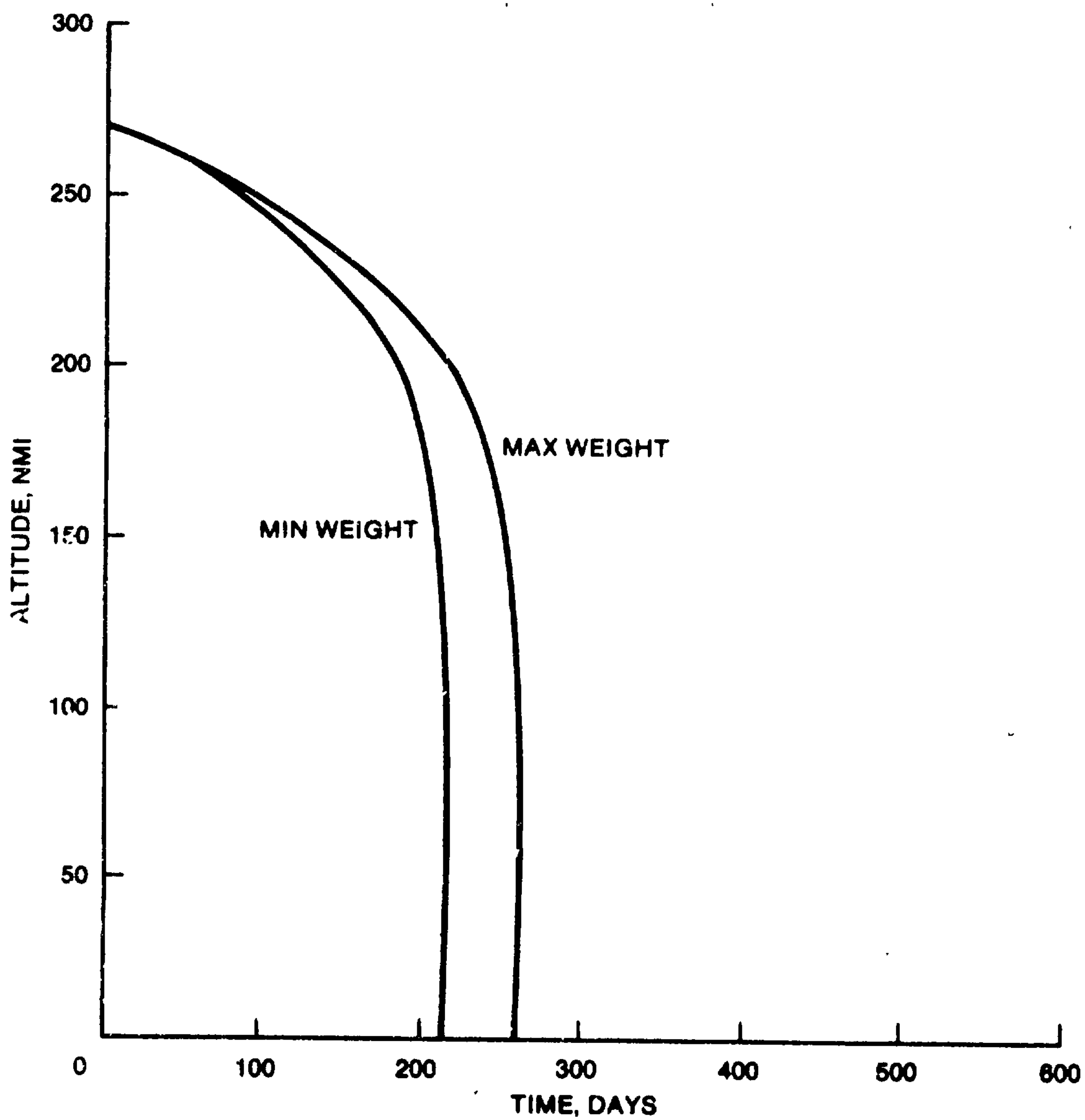


Figure 4.3.3.5-18 - CONTINUED

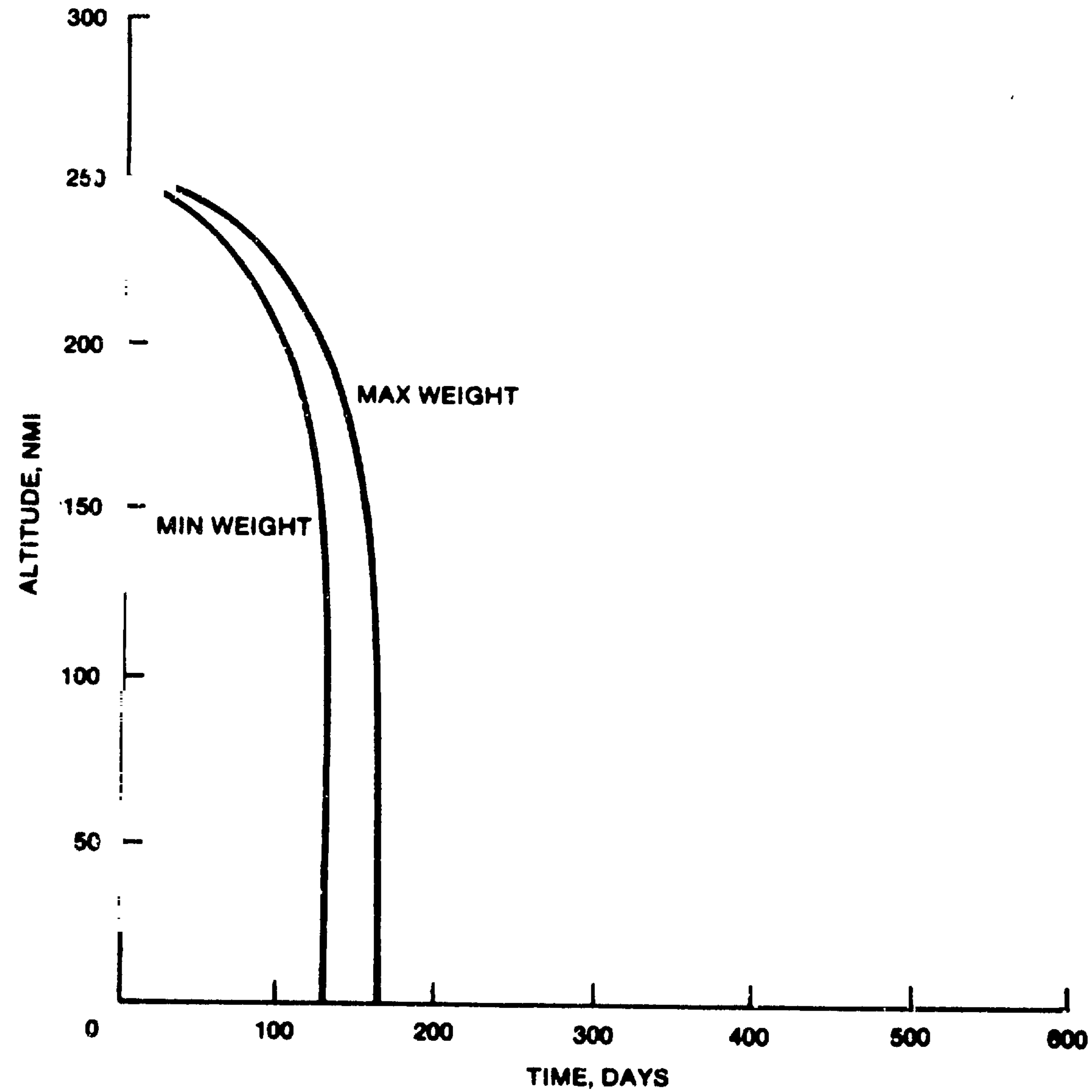


Figure 4.3.3.5-18 - CONTINUED

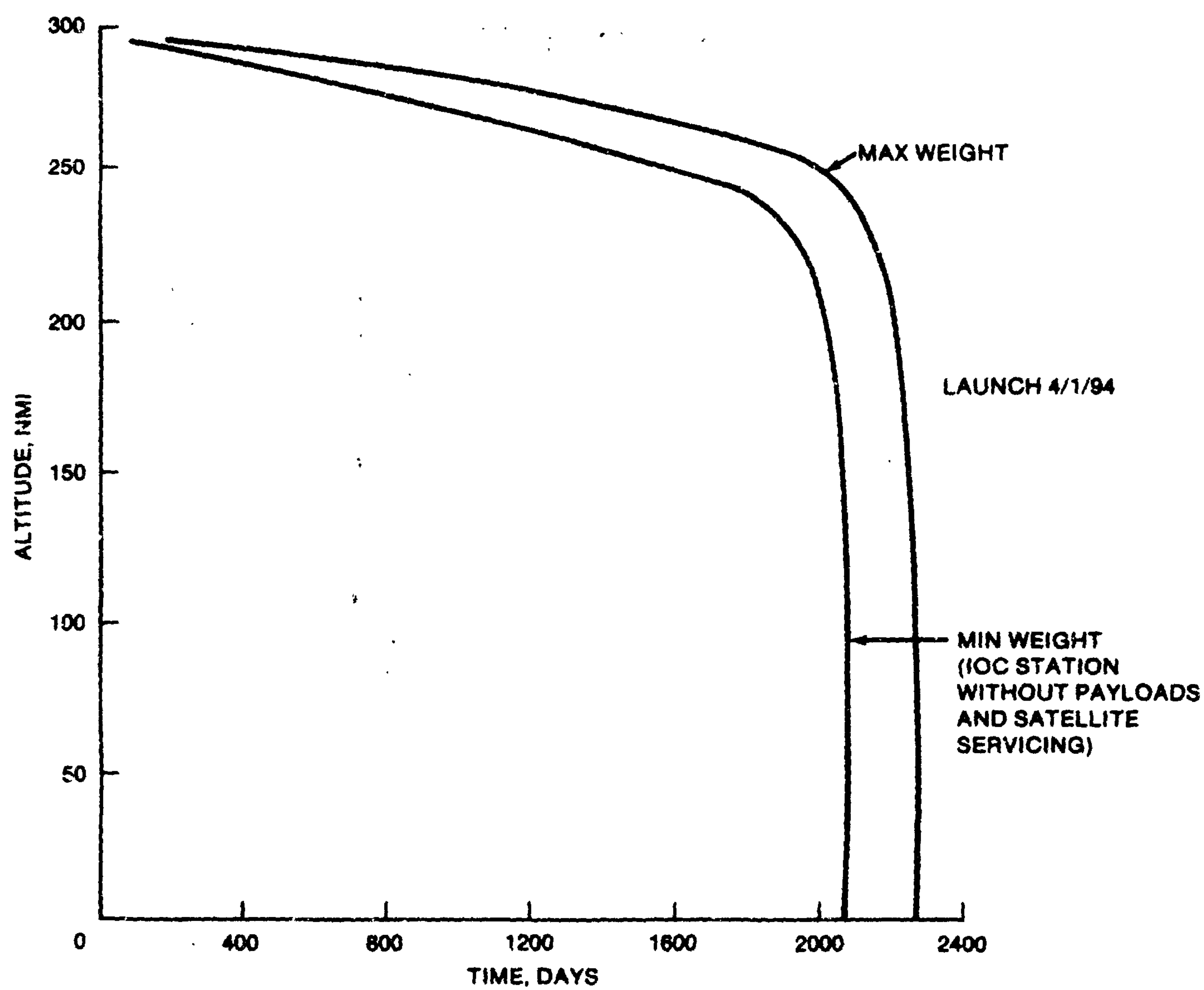


Figure 4.3.3.5-19 - ALTITUDE DECAY FOR MINIMUM DENSITY ATMOSPHERE

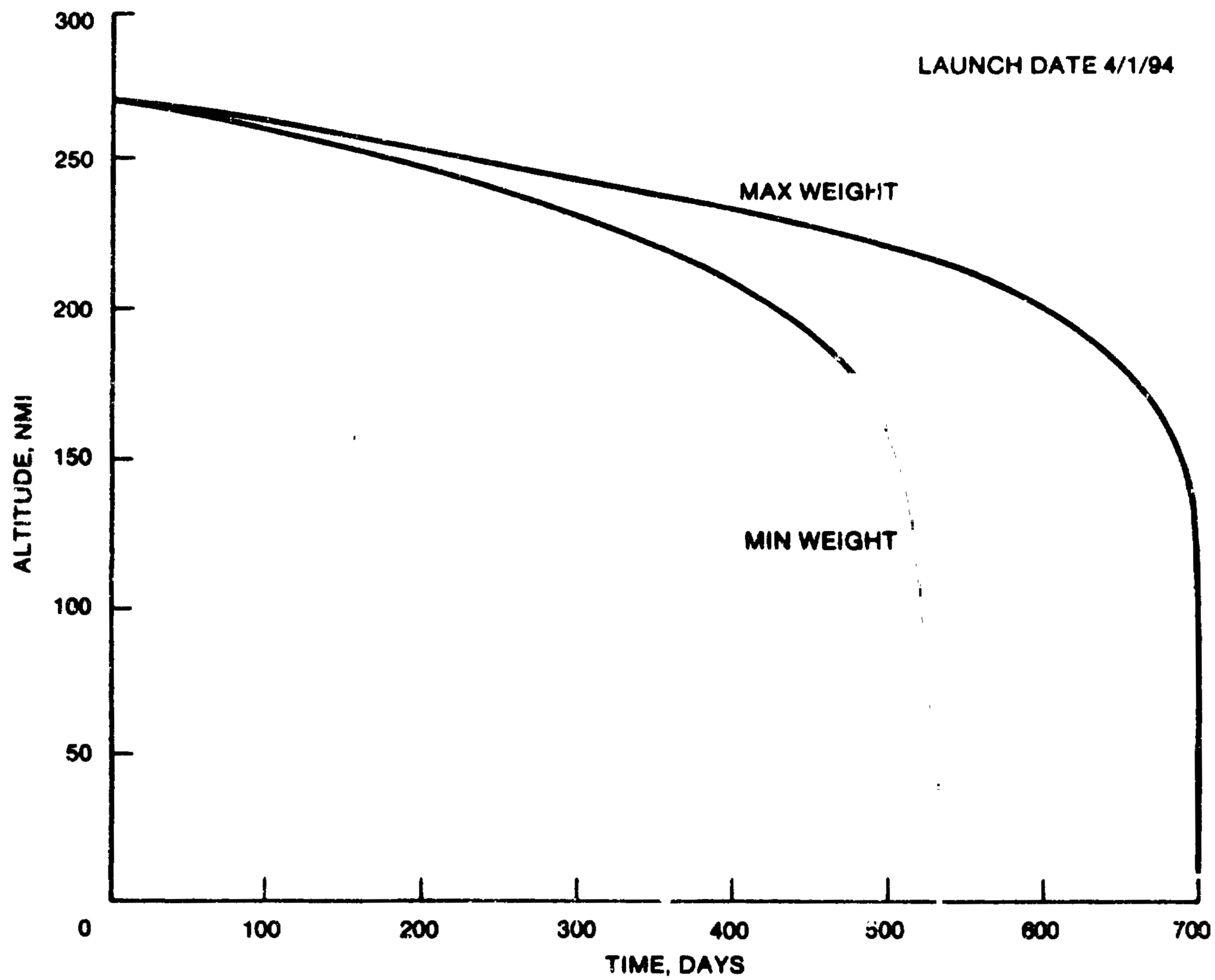


Figure 4.3.3.5-19 - CONTINUED

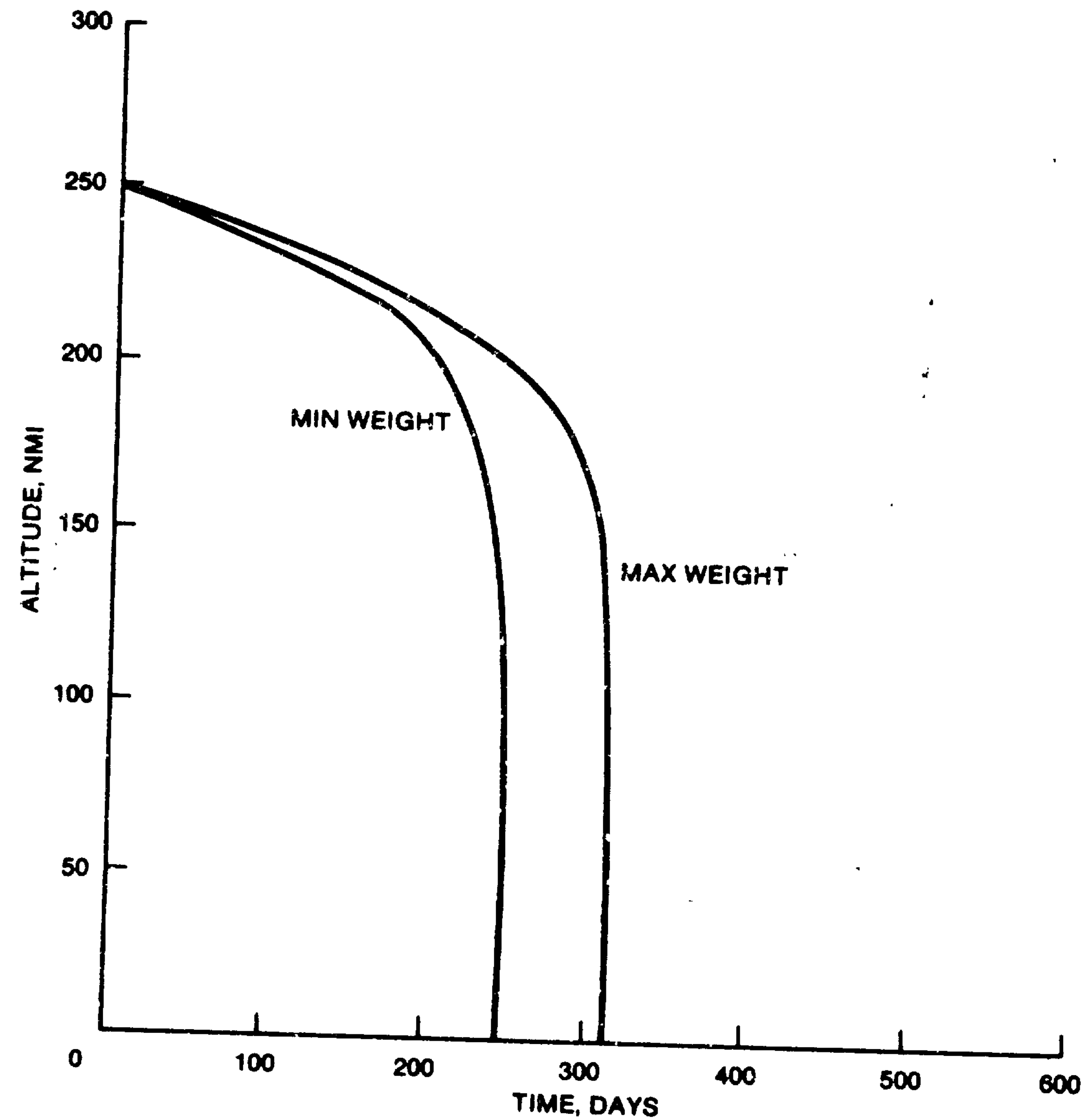


Figure 4.3.3.5-19 - CONTINLED

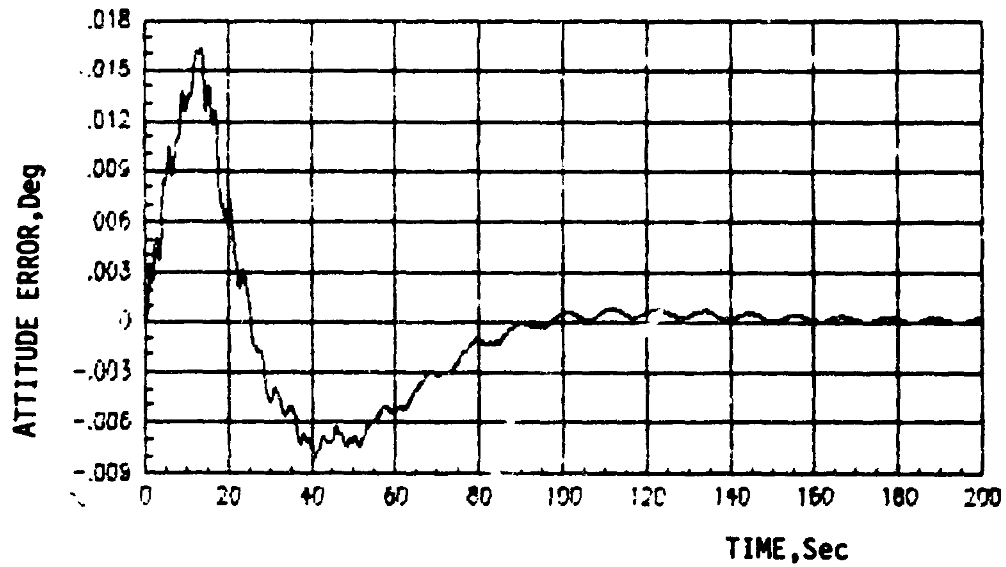


Figure 4.3.3.5-20.- X AXIS CREW DISTURBANCE

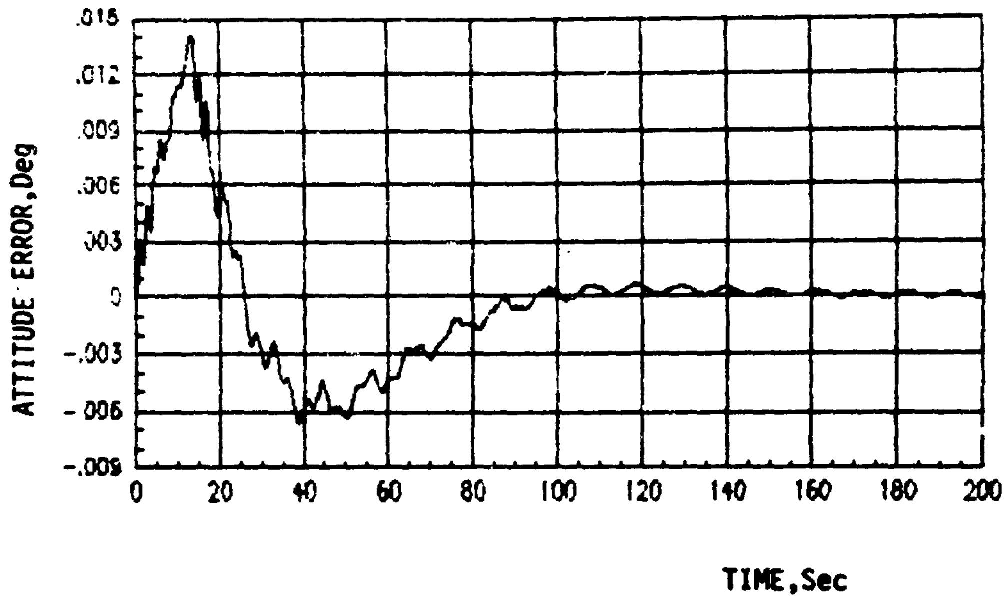


Figure 4.3.3.5-21. - Y AXIS CREW DISTURBANCE

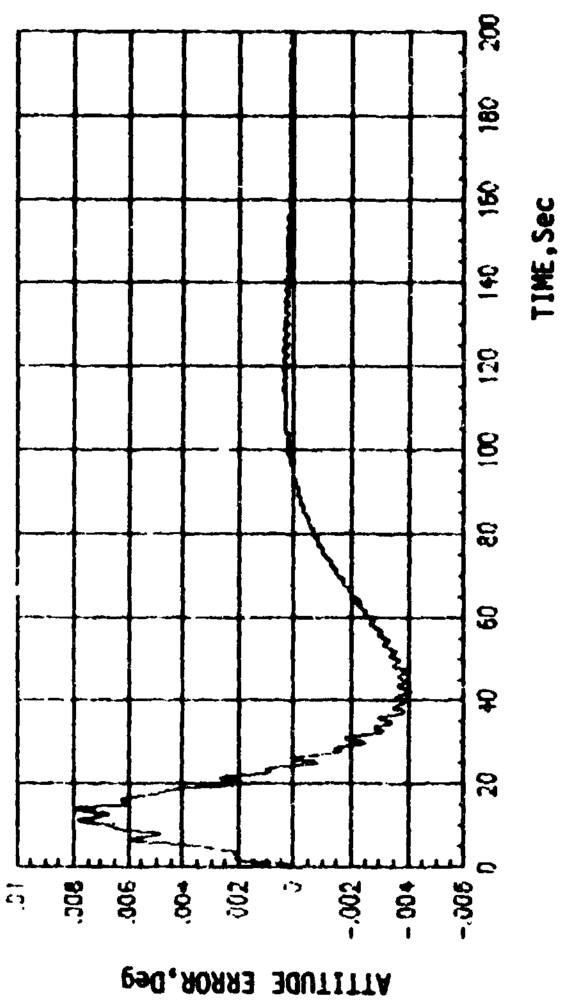


Figure 4.3.3.5-22 - Z AXIS CREW DISTURBANCE

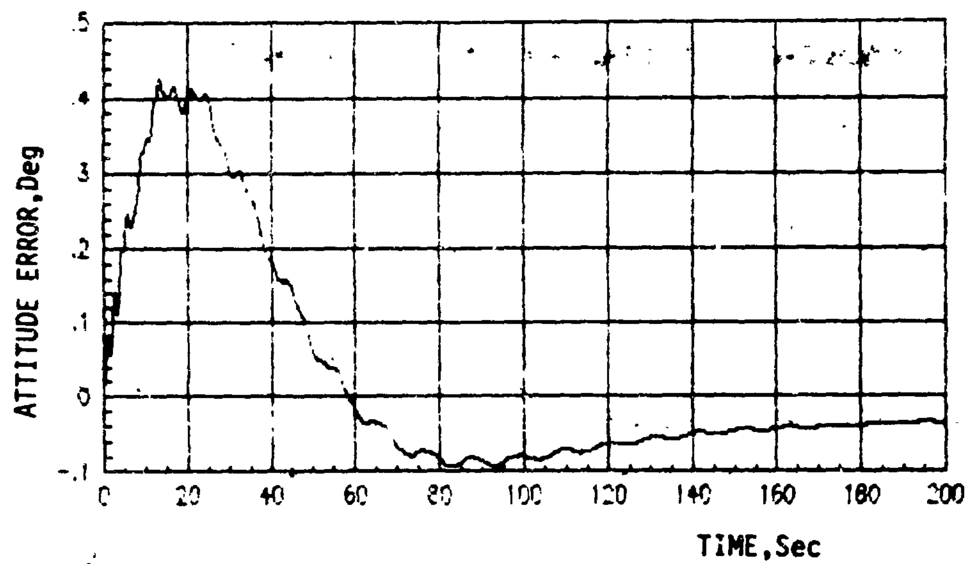


Figure 4.3.3.5-23 - X AXIS BERTHING DISTURBANCE

ORIGINAL PAGE IS
OF POOR QUALITY

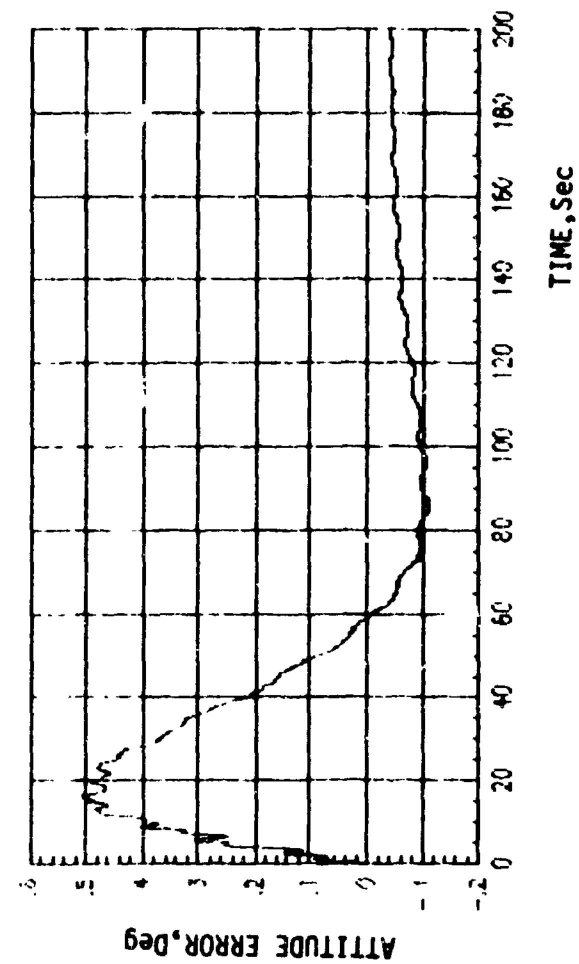


Figure 4.3.3.5-24 - Y AXIS BERTHING DISTURBANCE

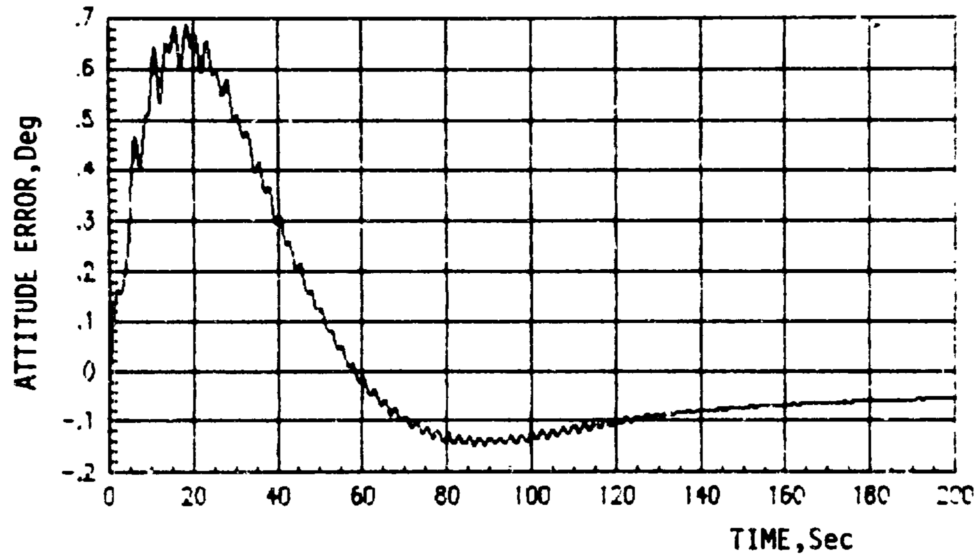


Figure 4.3.3.5-25 - Z AXIS BERTHING DISTURBANCE

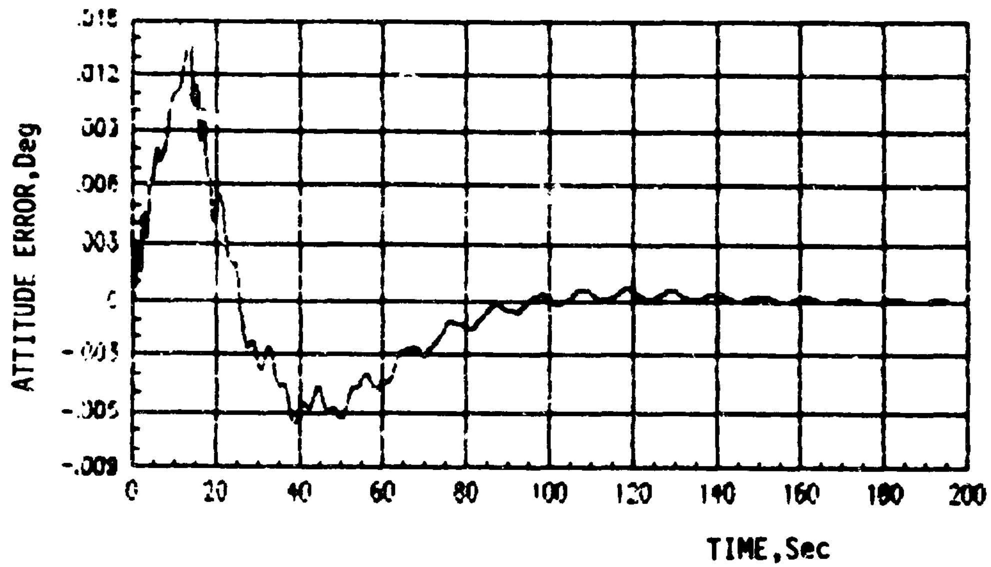


Figure 4.3.3.5-26 - X AXIS CREW DISTURBANCE TORQUE LIMITED

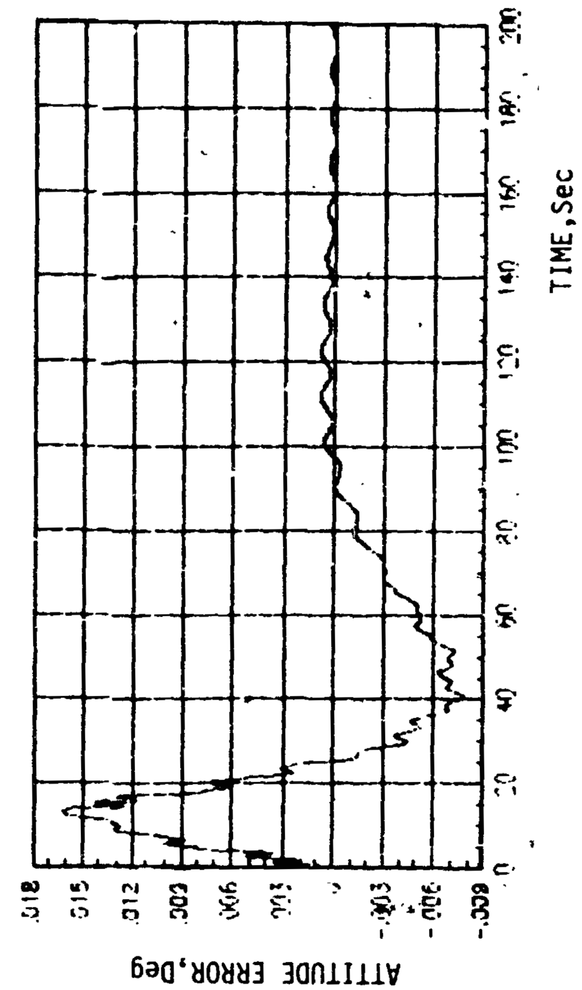


Figure 4.3.3.5-27 - Y AXIS CREW DISTURBANCE TORQUE LIMITED

ORIGINAL PAGE IS
OF POOR QUALITY

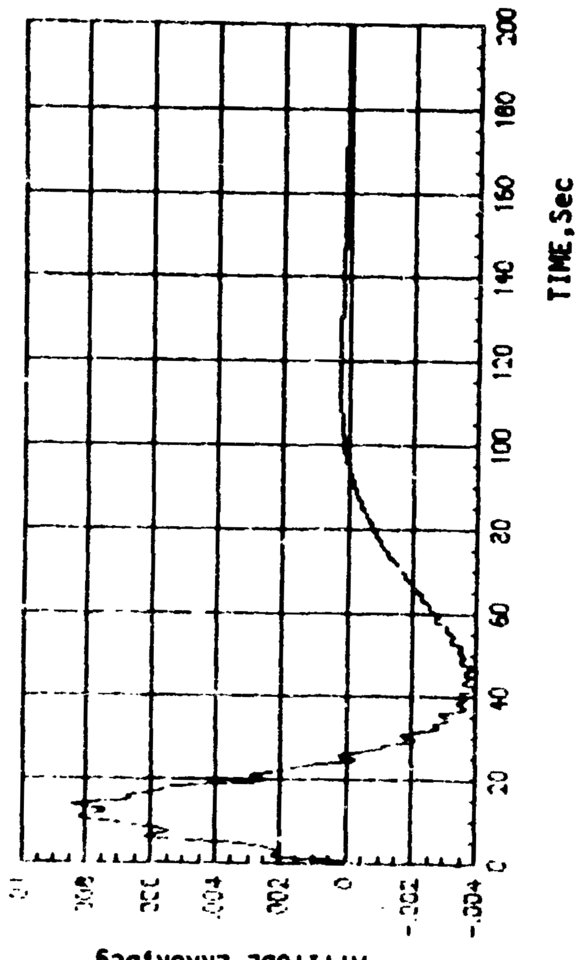


Figure 4.3.3.5-28 - Z AXIS CREW DISTURBANCE TORQUE LIMITED

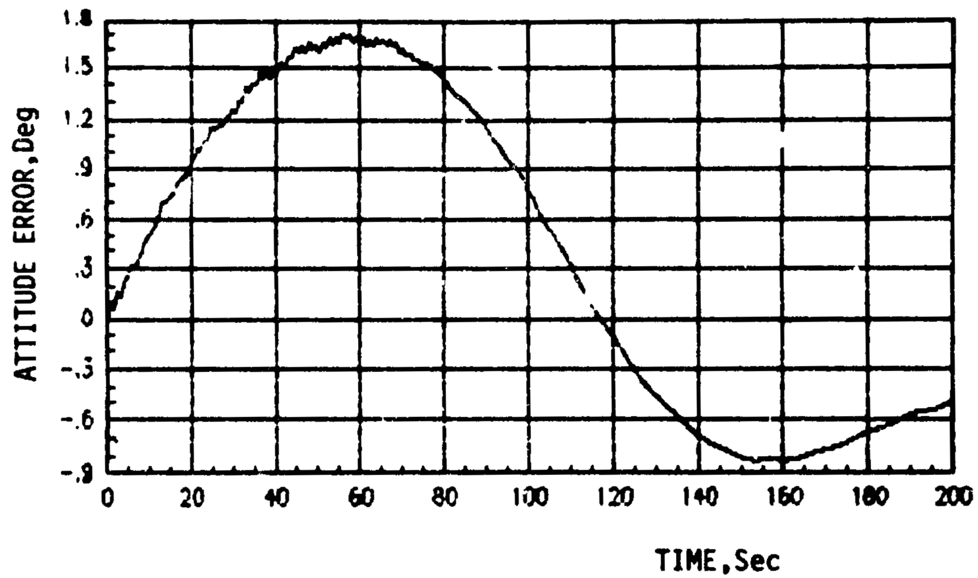


Figure 4.3.3.5-29 - X AXIS BERTHING DISTURBANCE TORQUE LIMITED

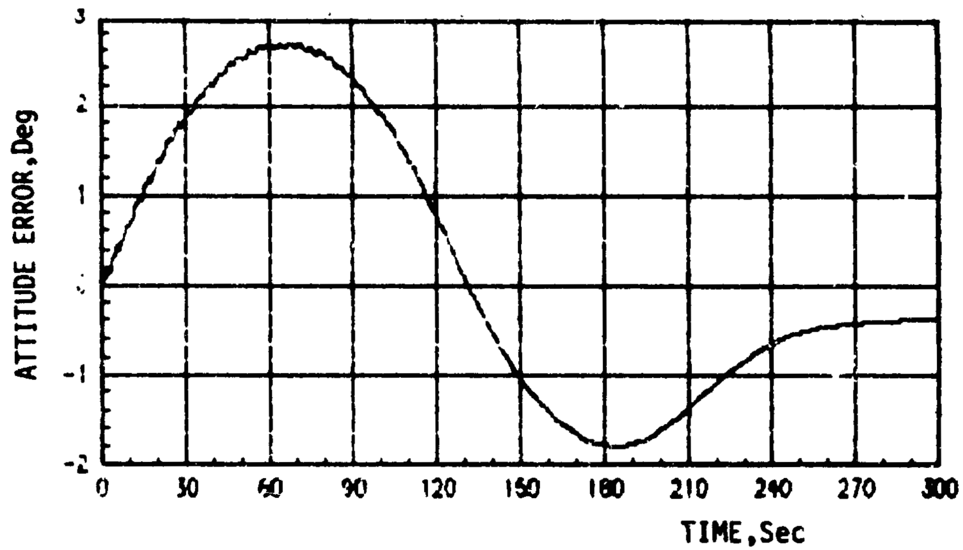


Figure 4.3.3.5-30 - Y AXIS BERTHING DISTURBANCE TORQUE LIMITED

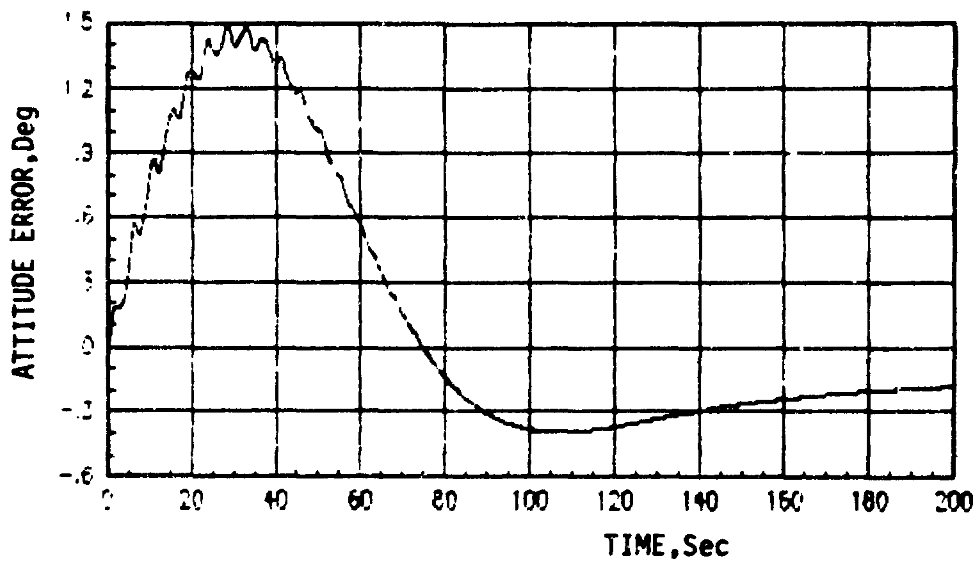


Figure 4.3.3.5-31 - Z AXIS BERTHING DISTURBANCE TORQUE LIMITED

ORIGINAL ... OF POOR QUALITY

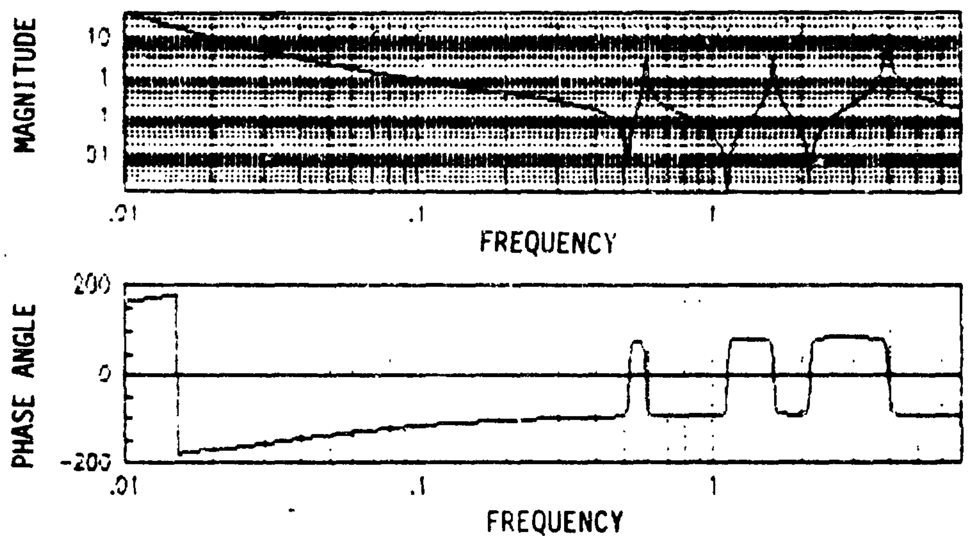


Figure 4.3.3.5-32 - X AXIS BODE PLOT

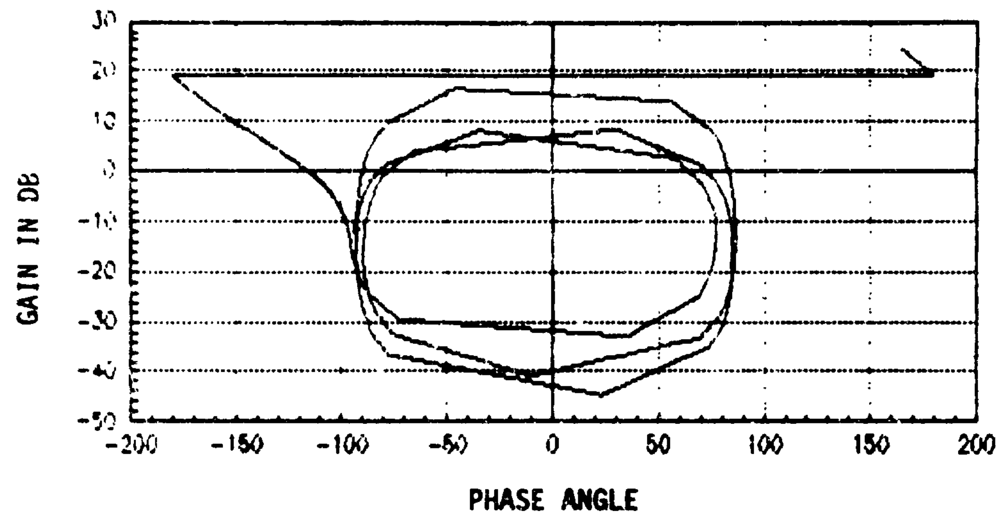


Figure 4.3.3.5-33 - X AXIS NICHOLS CHART

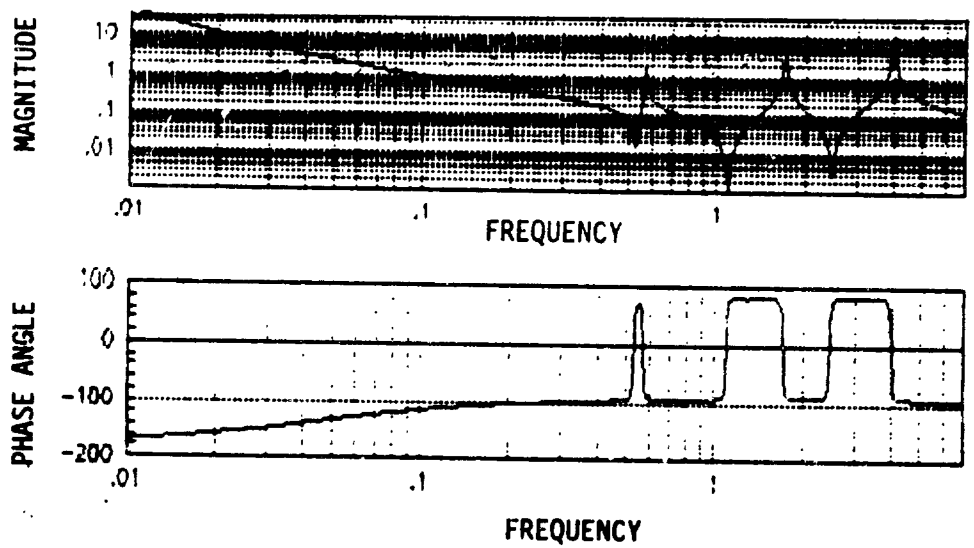


Figure 4.3.3.5-34 - Y AXIS BODE PLOT

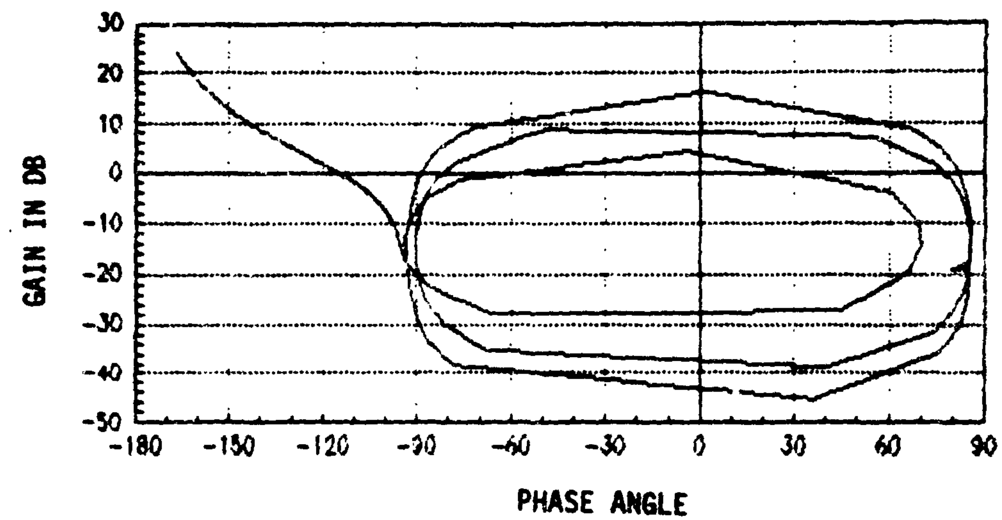


Figure 4.3.3.5-35 - Y AXIS NICHOLS CHART

ORIGINAL PAGE IS
OF POOR QUALITY

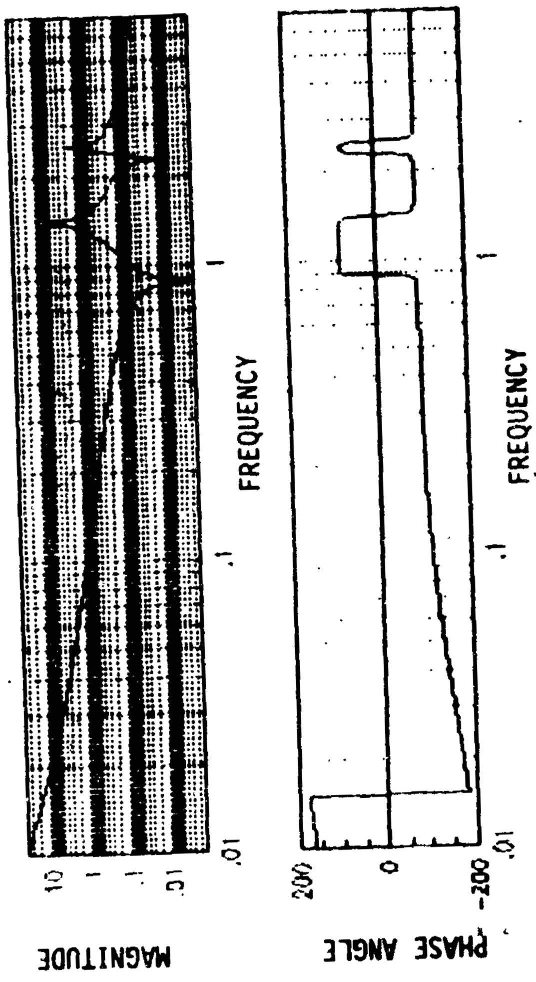


Figure 4.3.3.5-36 - Z AXIS BODE PLOT

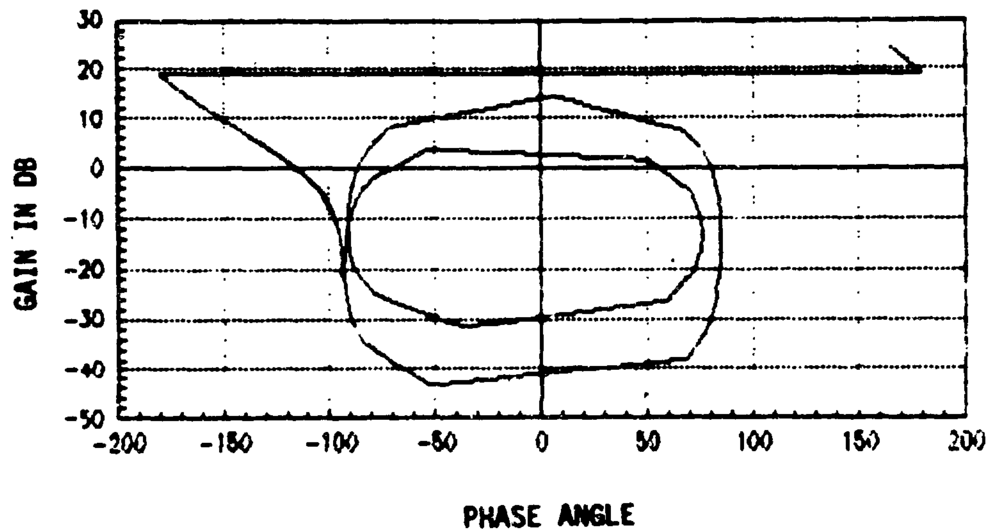


Figure 4.3.3.5-37 - Z AXIS NICHOLS CHART

ORIGINAL PAGE IS
OF POOR QUALITY

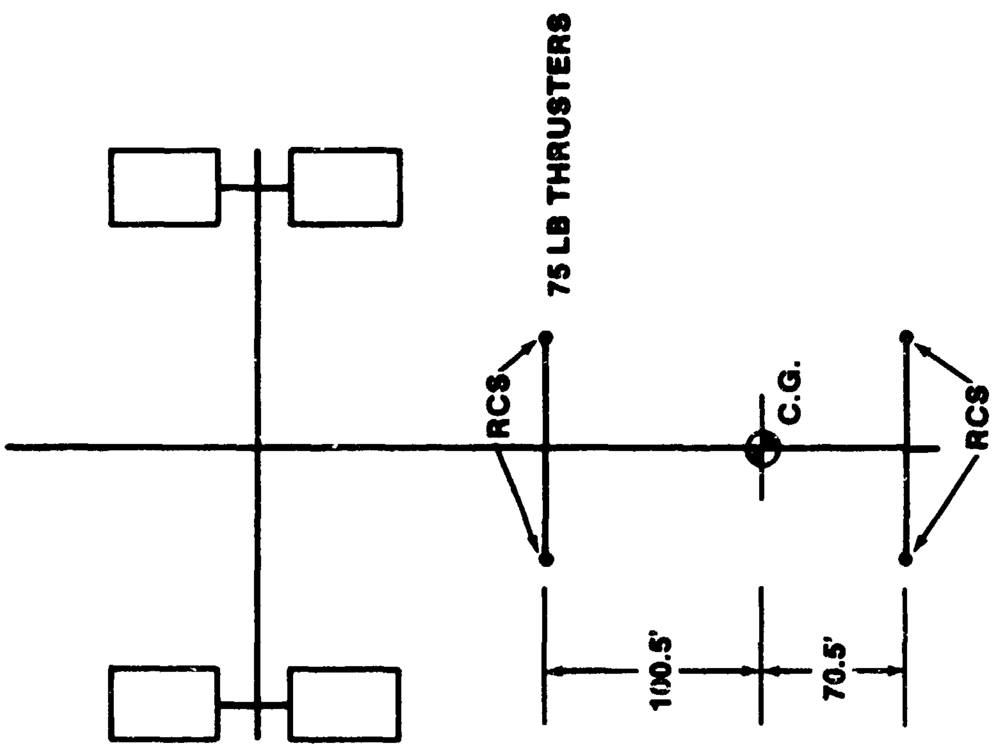


Figure 4.3.3.5-38 - RCS CONFIGURATION

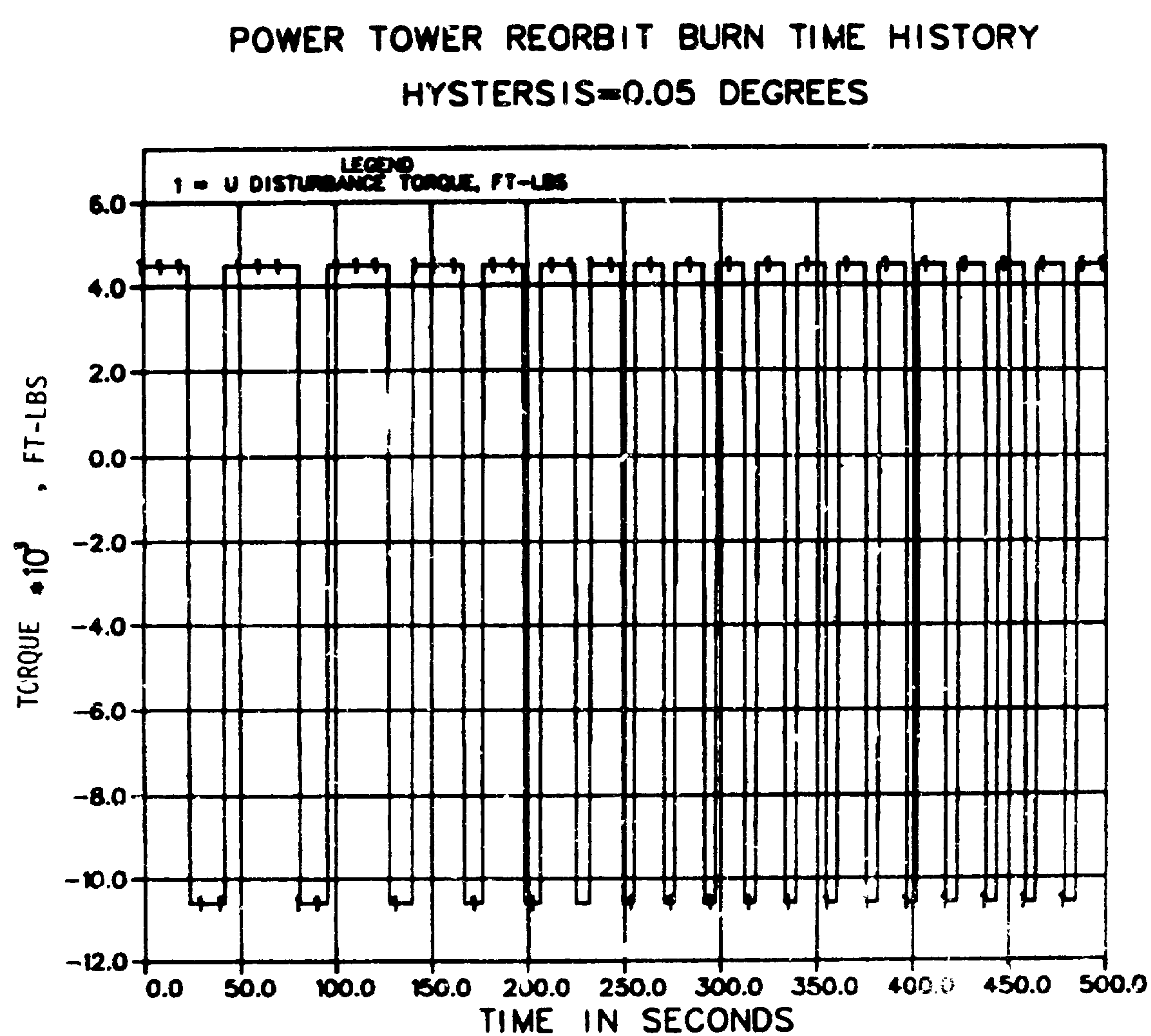


Figure 4.3.3.5-39 - . RQUE HISTORY

ORIGINAL PAGE IS OF POOR QUALITY

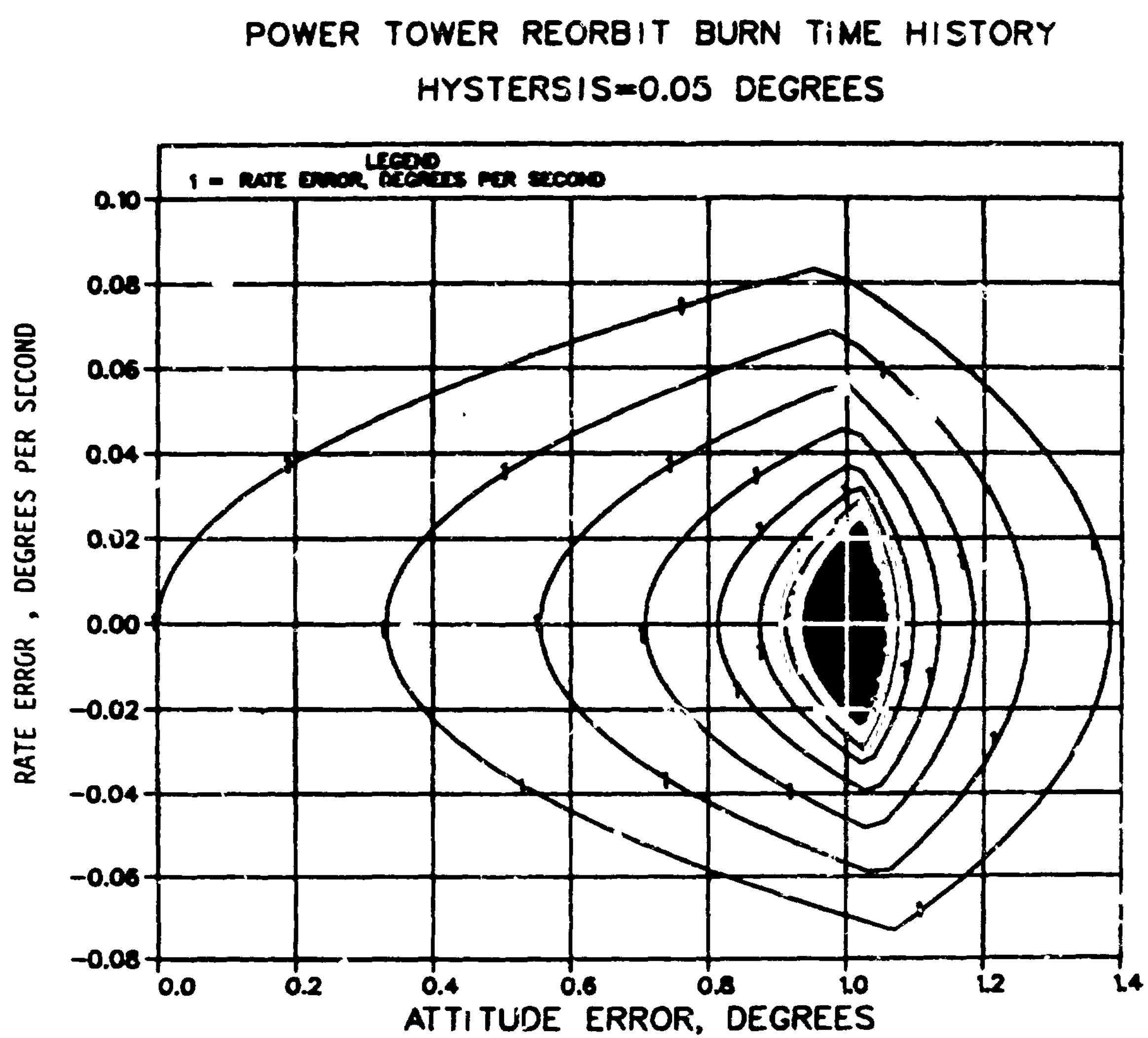


Figure 4.3.3.5-40 - PHASE PLANE

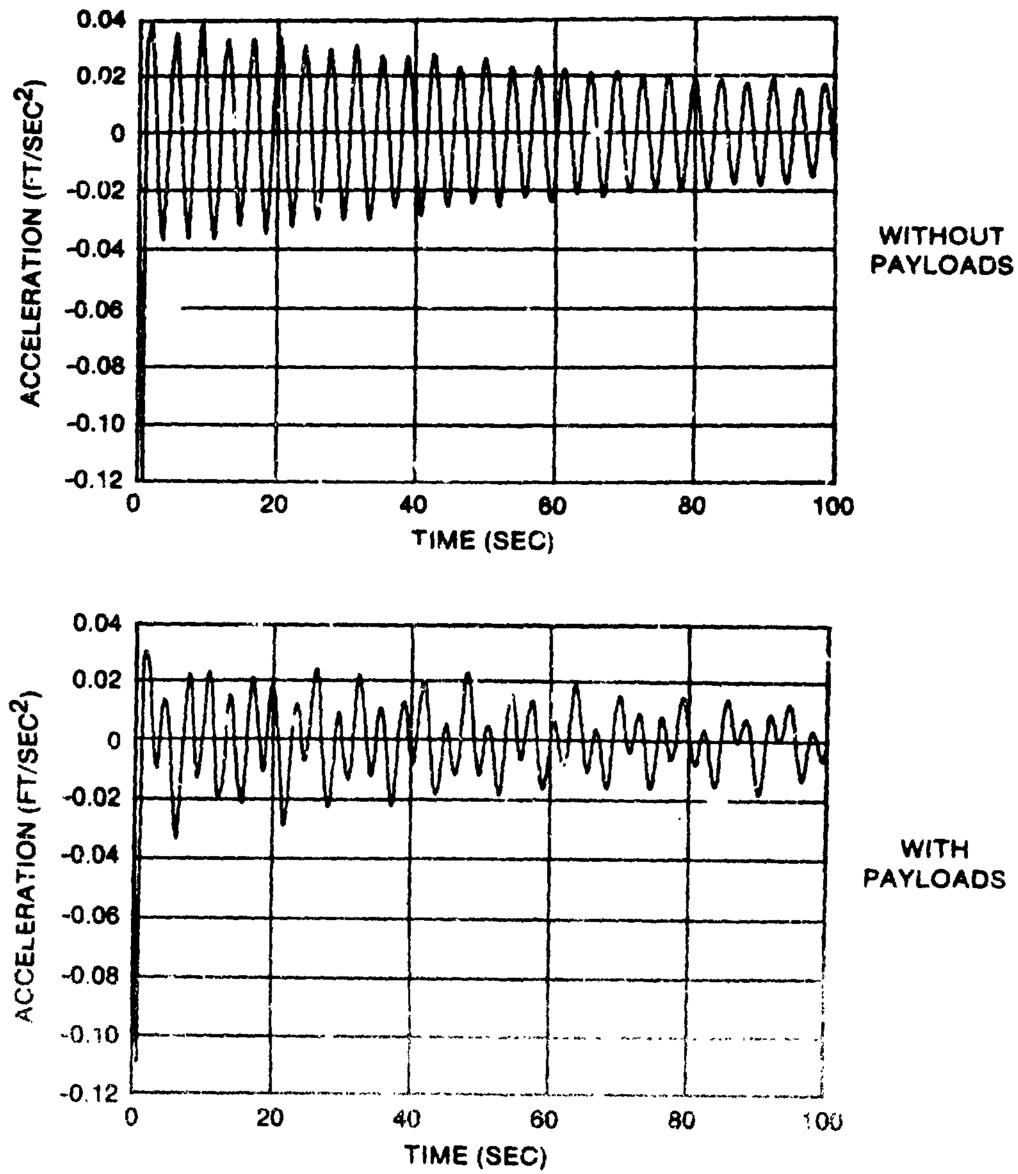


Figure 4.3.3.5-41 **TRANSIENT RESPONSE AT BERTHING PORT IN X DIRECTION DUE TO ORBITER BERTHING**

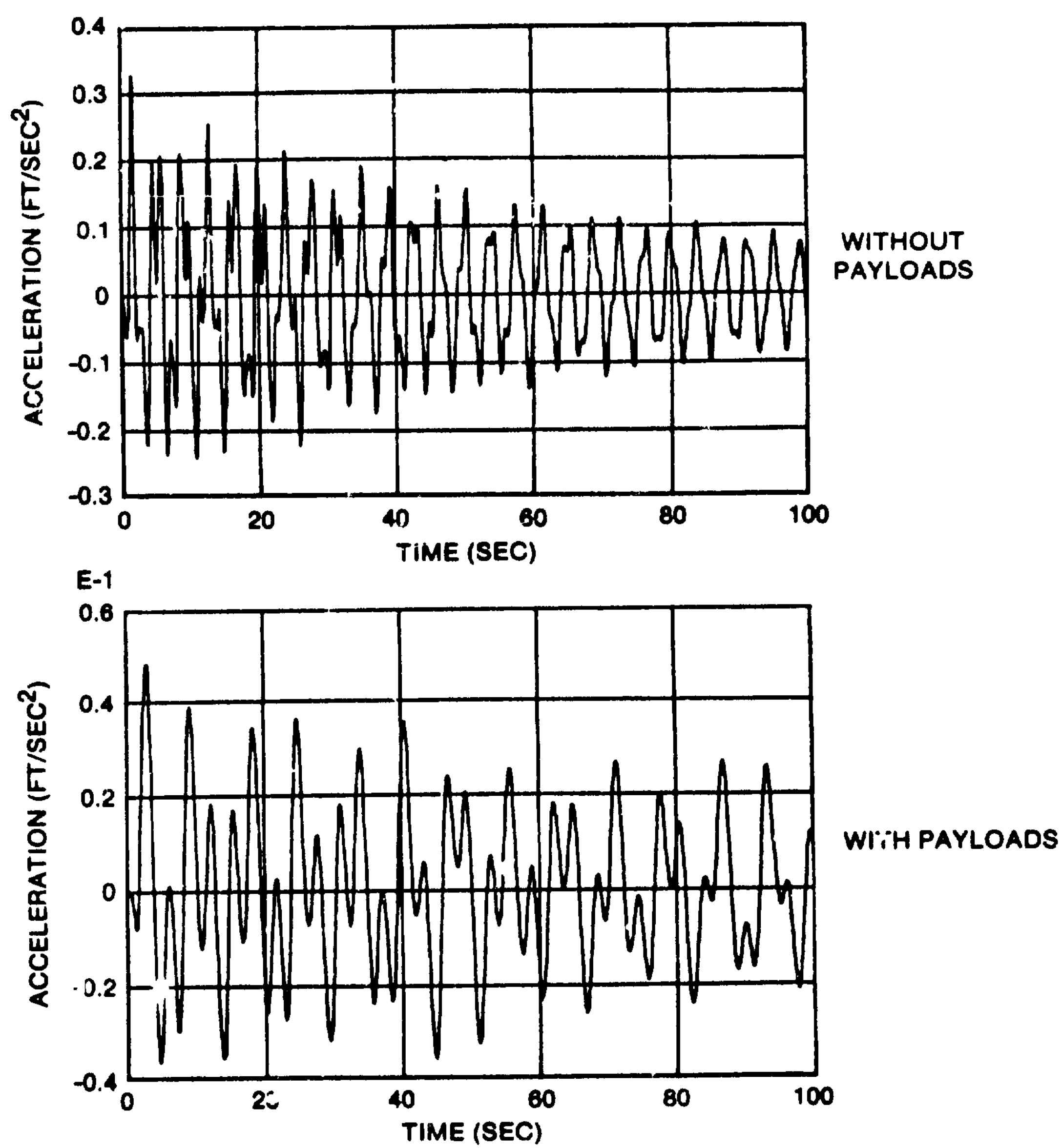


Figure 4.3.3.5-42 **TRANSIENT RESPONSE AT CENTER OF UPPER BOOM IN X DIRECTION DUE TO ORBITER BERTHING**

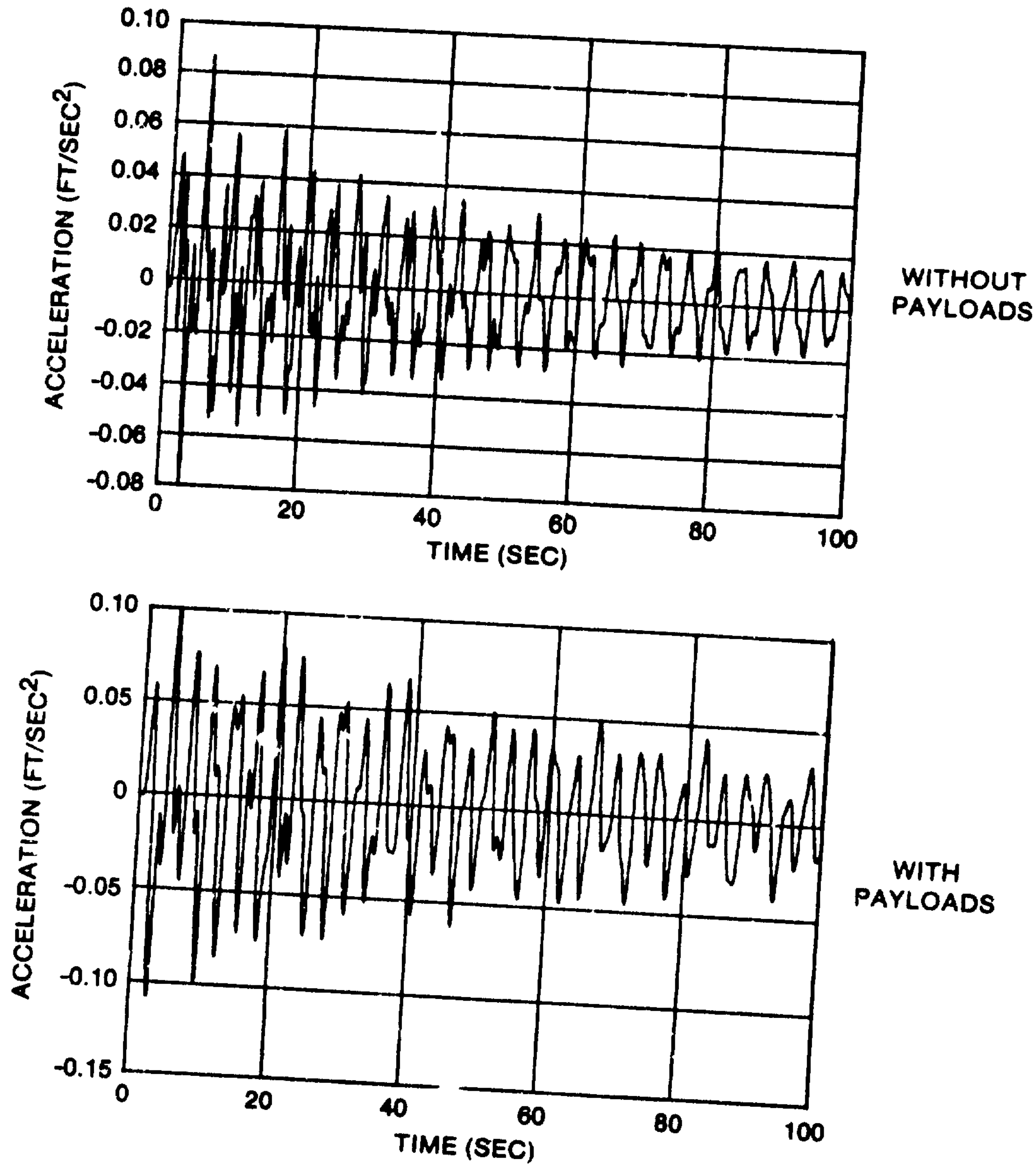


Figure 4.3.3.5-43 TRANSIENT RESPONSE AT TIP OF TRANSVERSE BOOM IN X DIRECTION DUE TO ORBITER BERTHING

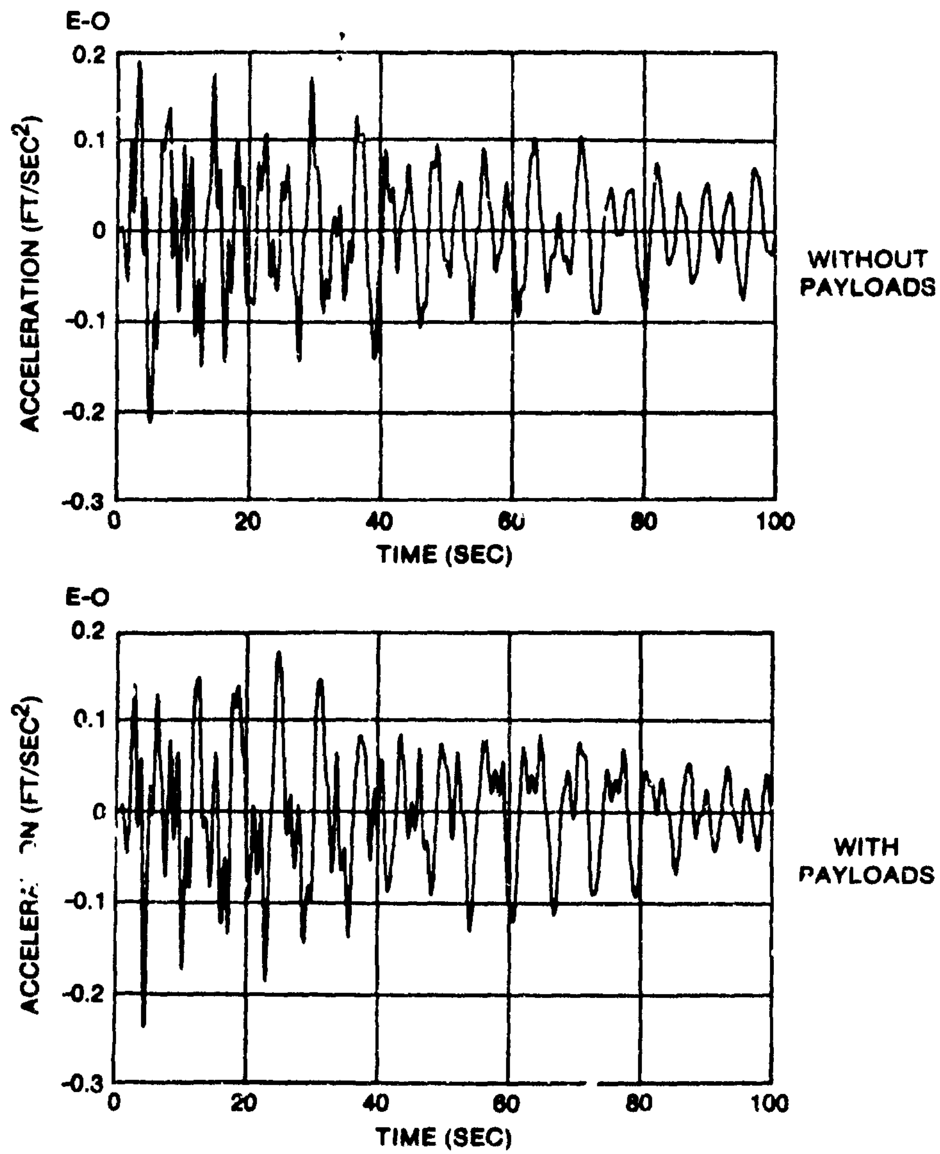


Figure 4.3.3.5-44 TRANSIENT RESPONSE AT TIP OF ARRAY MAST IN X DIRECTION DUE TO ORBITER BERTHING

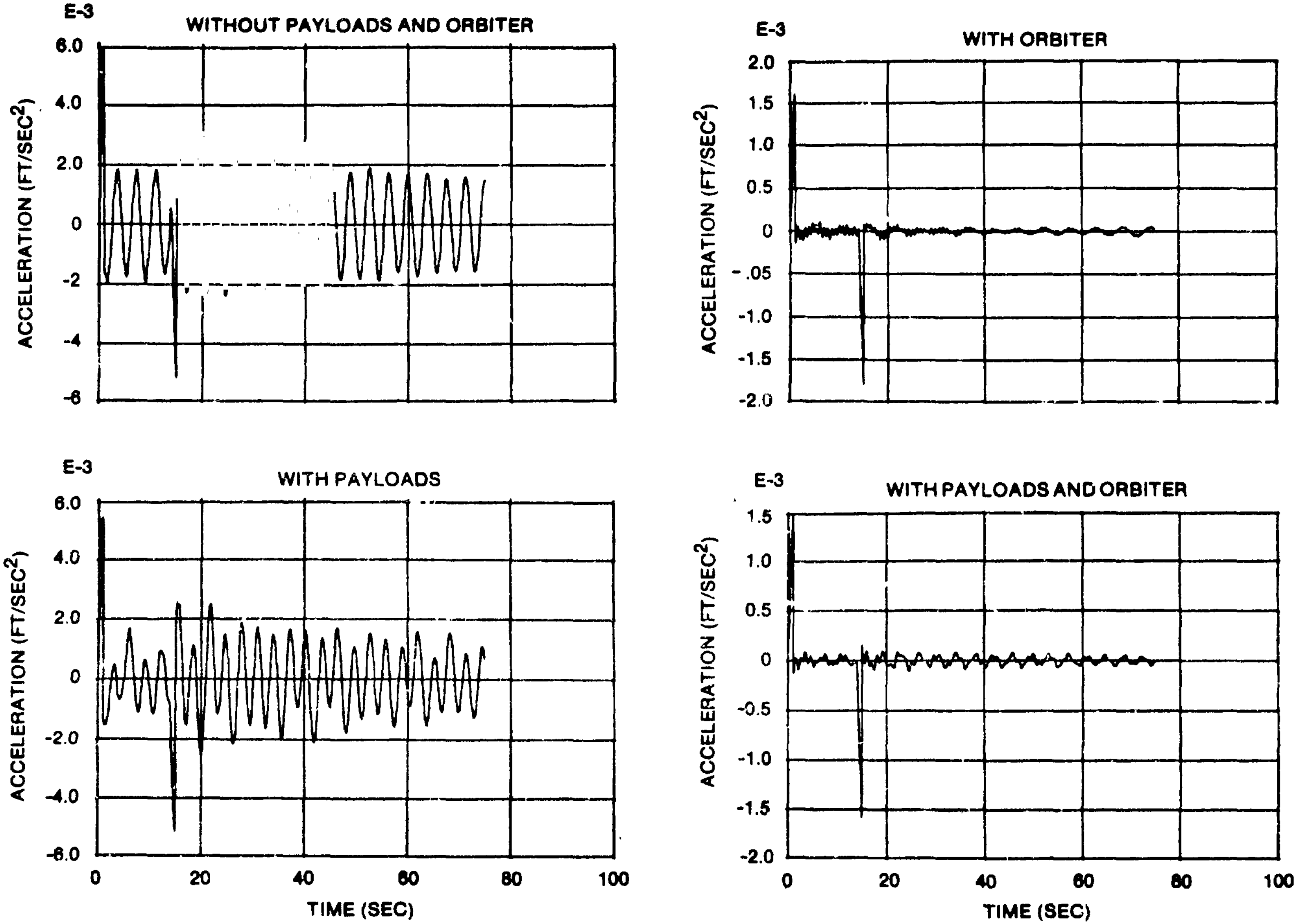


Figure 4.3.3.5-45. TRANSIENT RESPONSE AT HABITATION MODULE IN X DIRECTION DUE TO AXIAL CREW-KICK-OFF

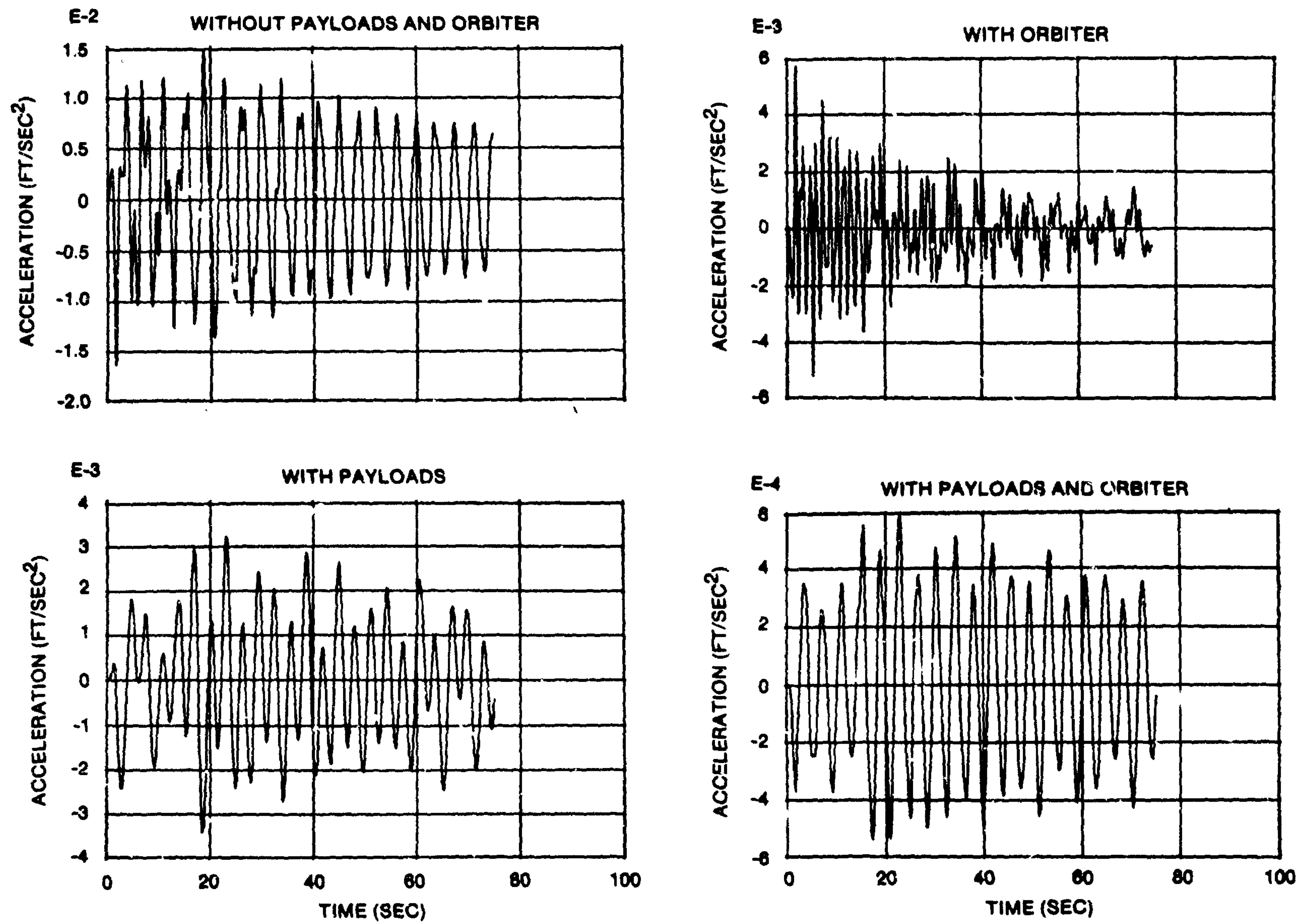


Figure 4.3.3.5-46 TRANSIENT RESPONSE AT CENTER OF UPPER BOOM IN X DIRECTION DUE TO AXIAL CREW-KICK-OFF

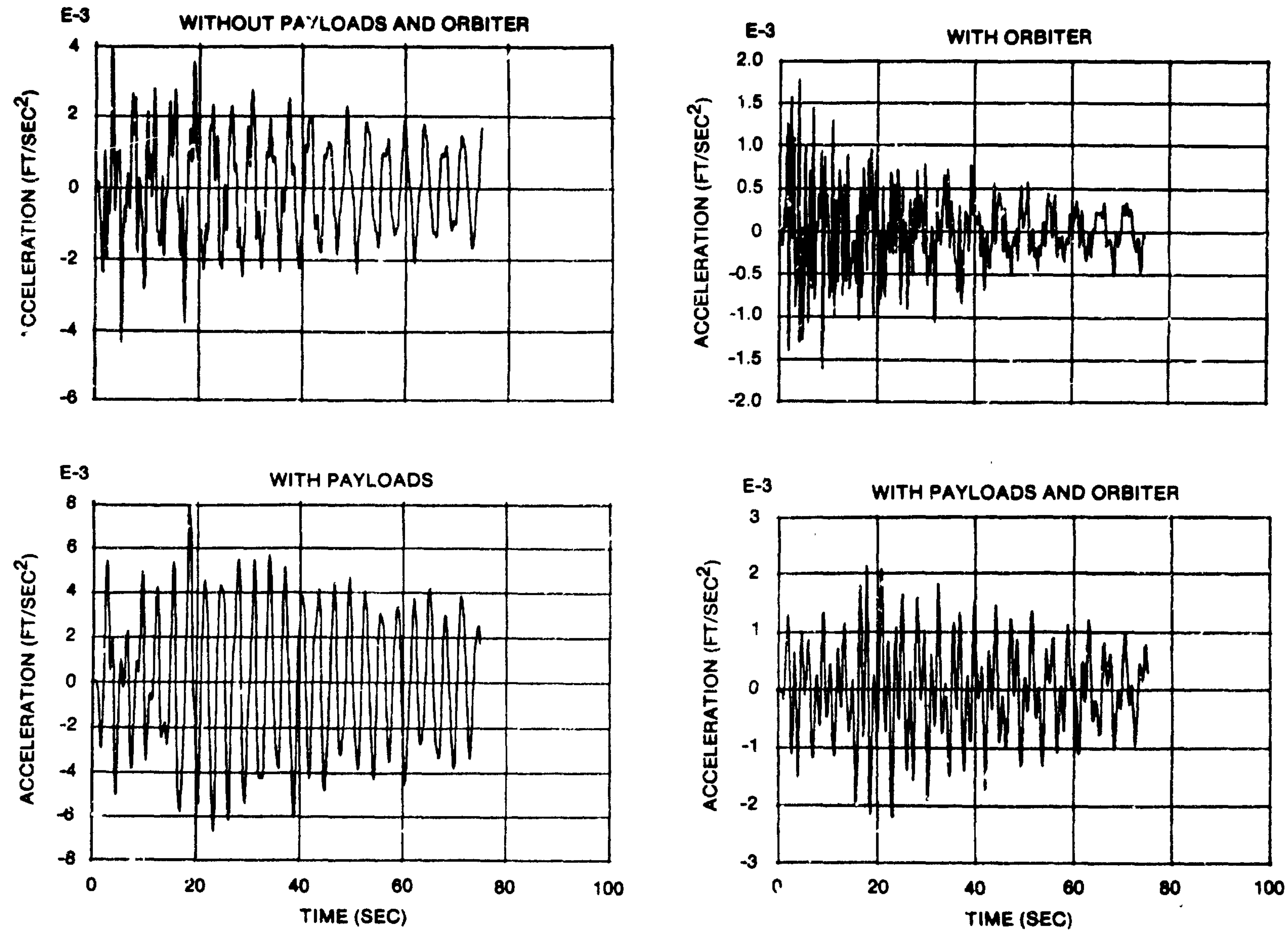


Figure 4.3.3.5-47 TRANSIENT RESPONSE AT TIP OF TRANSVERSE BOOM IN X DIRECTION DUE TO AXIAL CREW-KICK-OFF

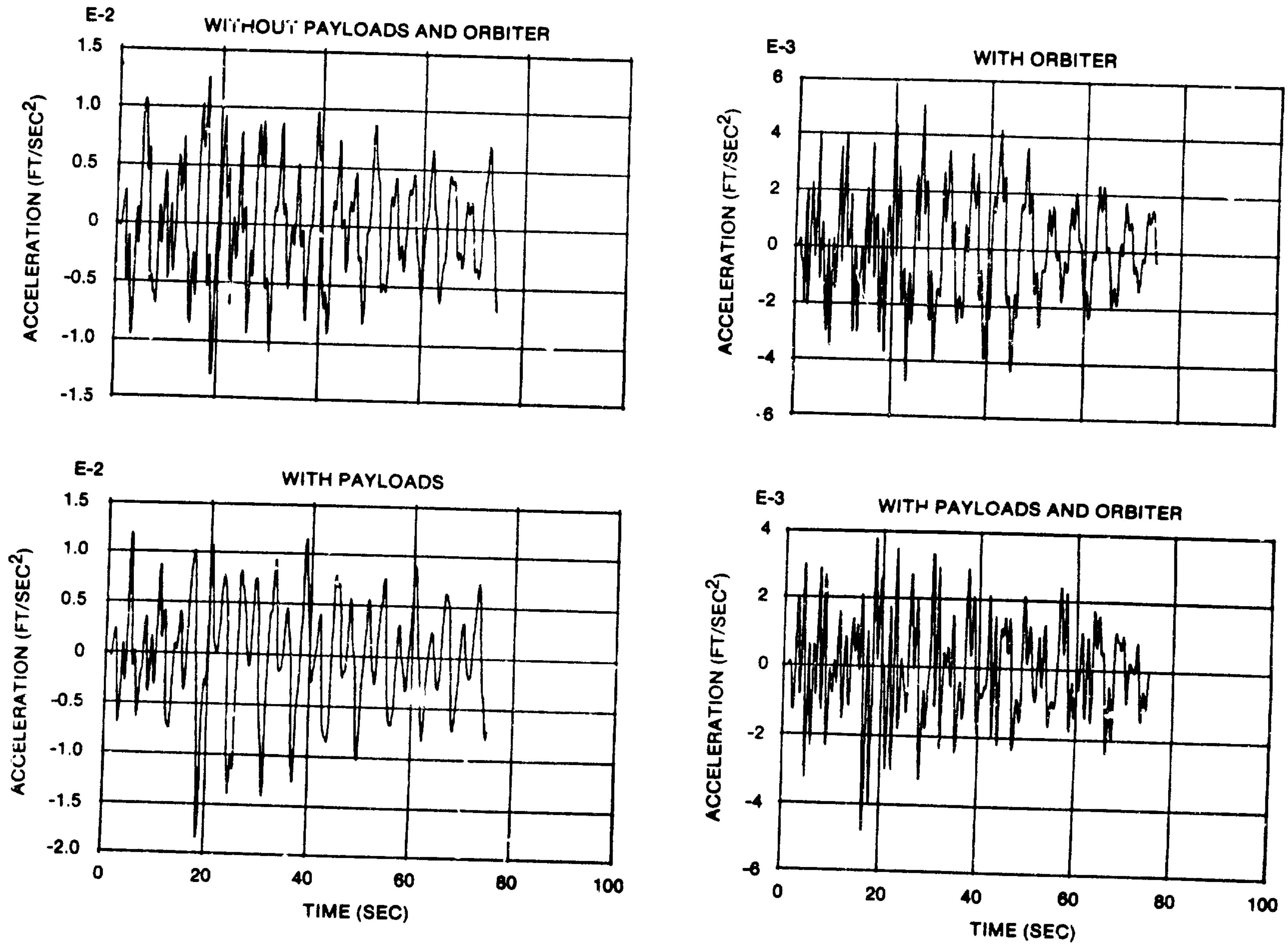


Figure 4.3.3.5-48. TRANSIENT RESPONSE AT TIP OF ARRAY MAST IN X DIRECTION DUE TO AXIAL CREW-KICK-OFF

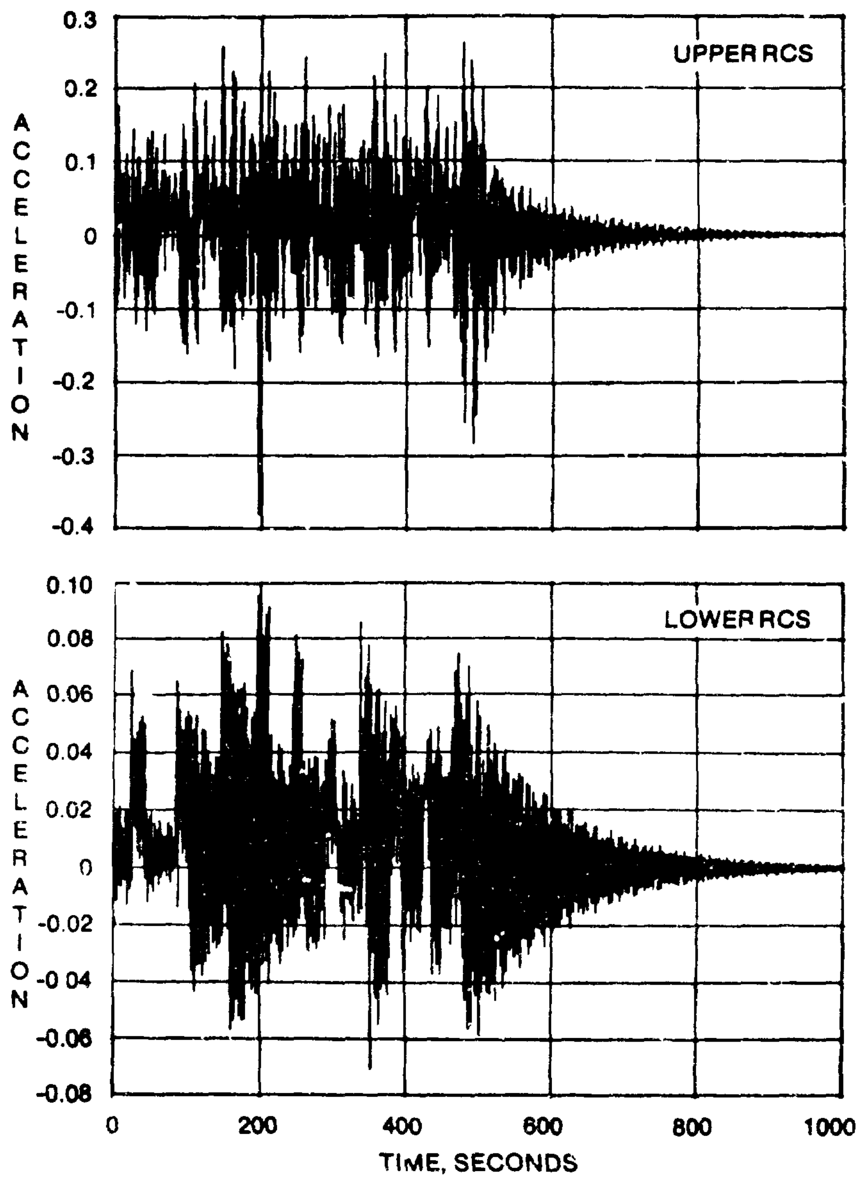


Figure 4.3.3.5-49 ACCELERATION TIME HISTORIES IN FEET/SEC^2 AT THE UPPER AND LOWER RCS LOCATIONS DUE TO RCS REBOOST FIRING FOR IOC STATION WITHOUT ATTACHMENTS

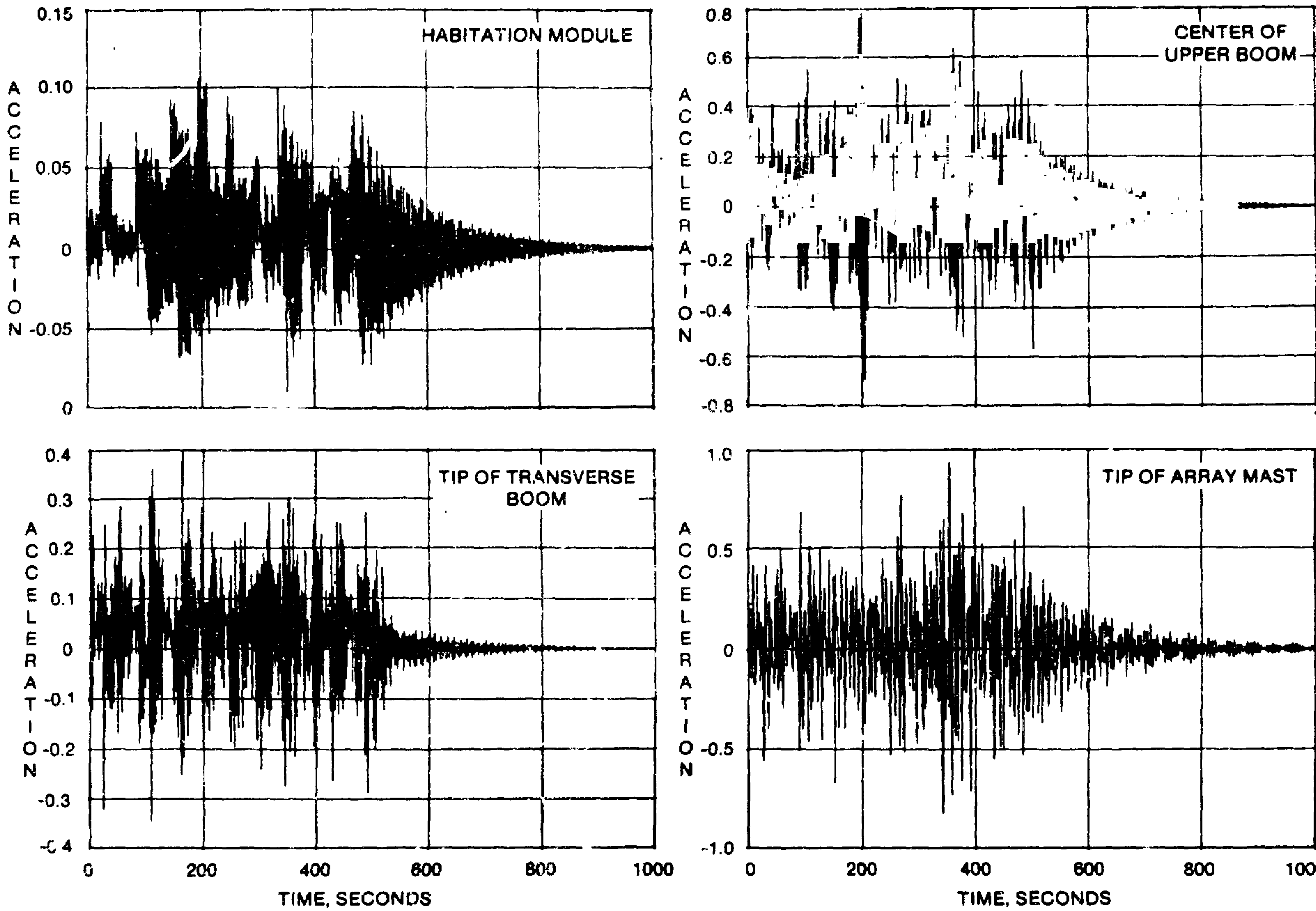


Figure 4.3.3.5-50 ACCELERATION TIME HISTORIES IN FEET/SEC^2 AT SELECTED STATION LOCATIONS DUE TO RCS REBOOST FIRING FOR IOC STATION WITHOUT ATTACHMENTS

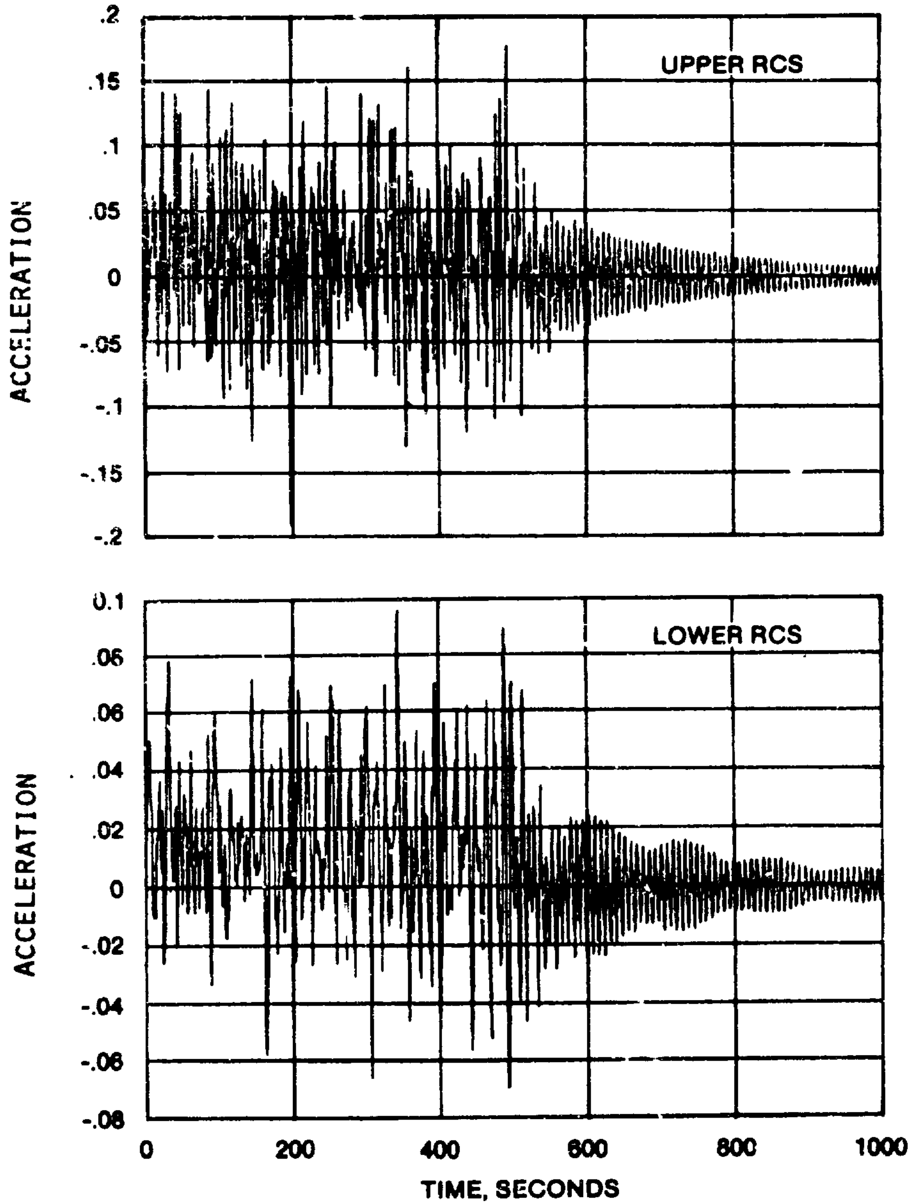


Figure 4.3.?.?-51 - ACCELERATION TIME HISTORIES IN FEET/SEC^2 AT THE UPPER AND LOWER RCS LOCATIONS DUE TO RCS REBOOST FIRING FOR IOC STATION WITH ATTACHED PAYLOADS

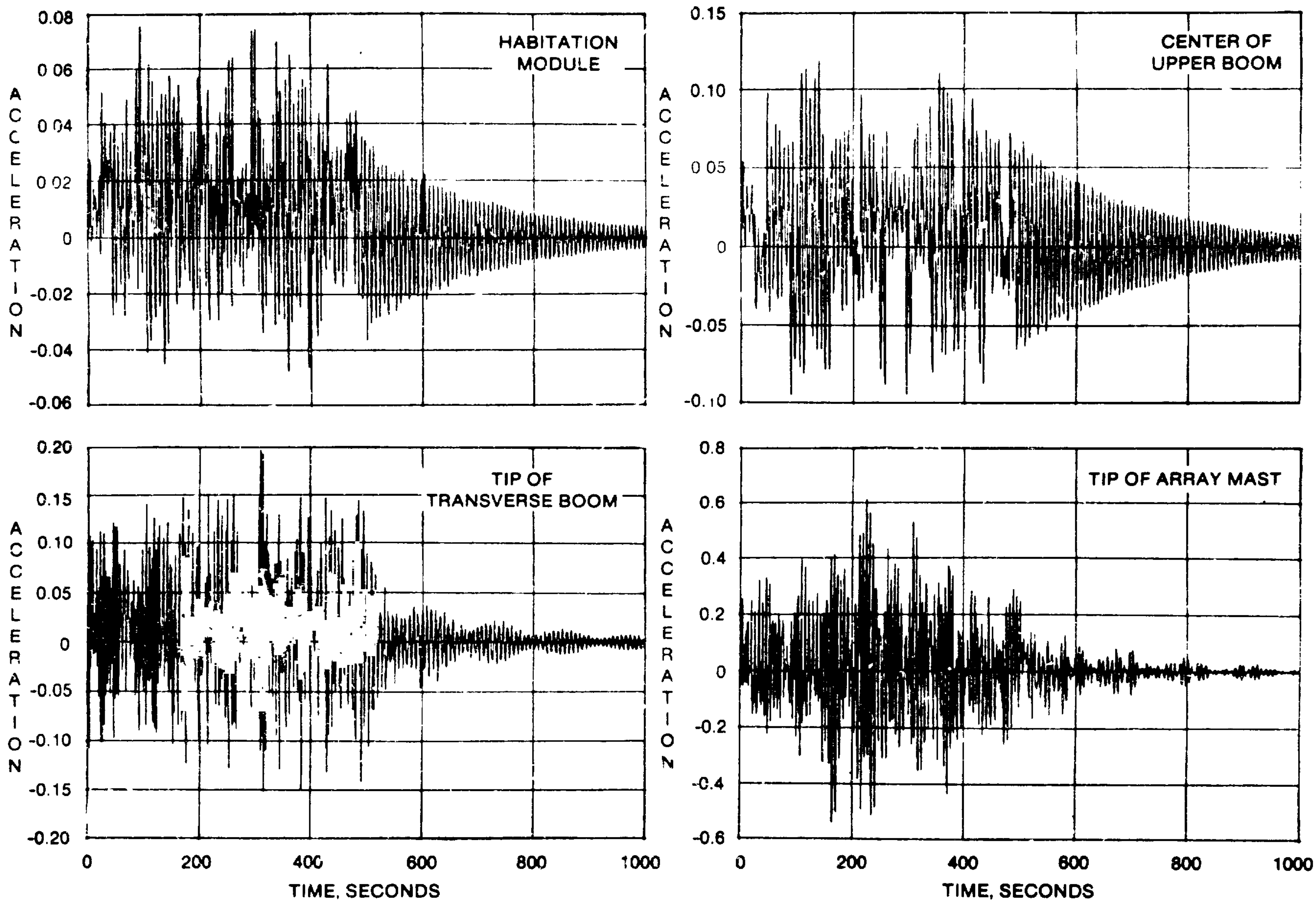


Figure 4.3.3.5-52 ACCELERATION TIME HISTORIES IN FEET/SEC^2 AT THE SELECTED STATION LOCATIONS DUE TO RCS REBOOST FIRING FOR IOC STATION WITH ATTACHED PAYLOADS

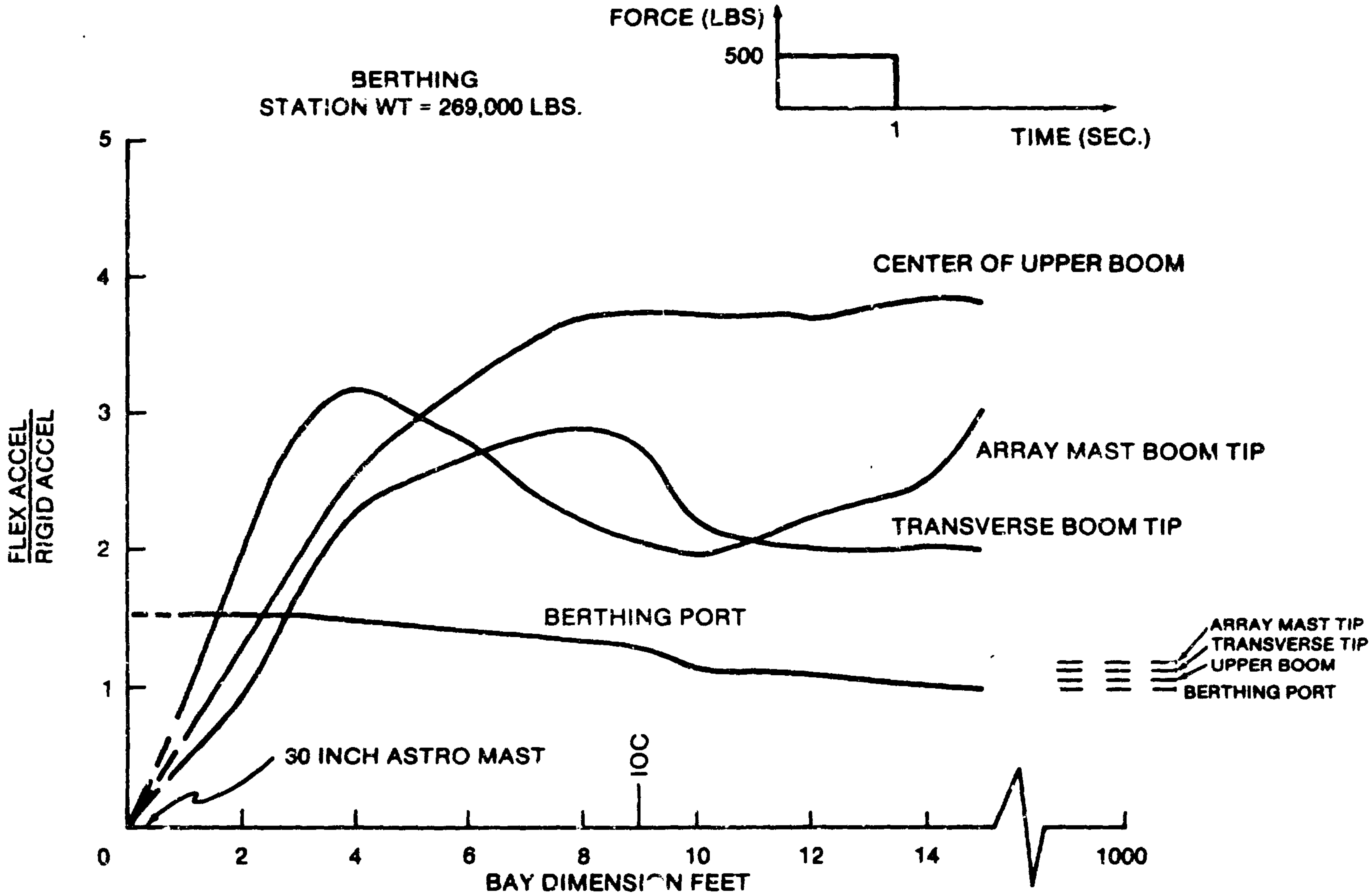


Figure 4.3.3.5-53 - VARIATION OF RESPONSE WITH BAY DIMENSION DUE TO ORBITER BERTHING

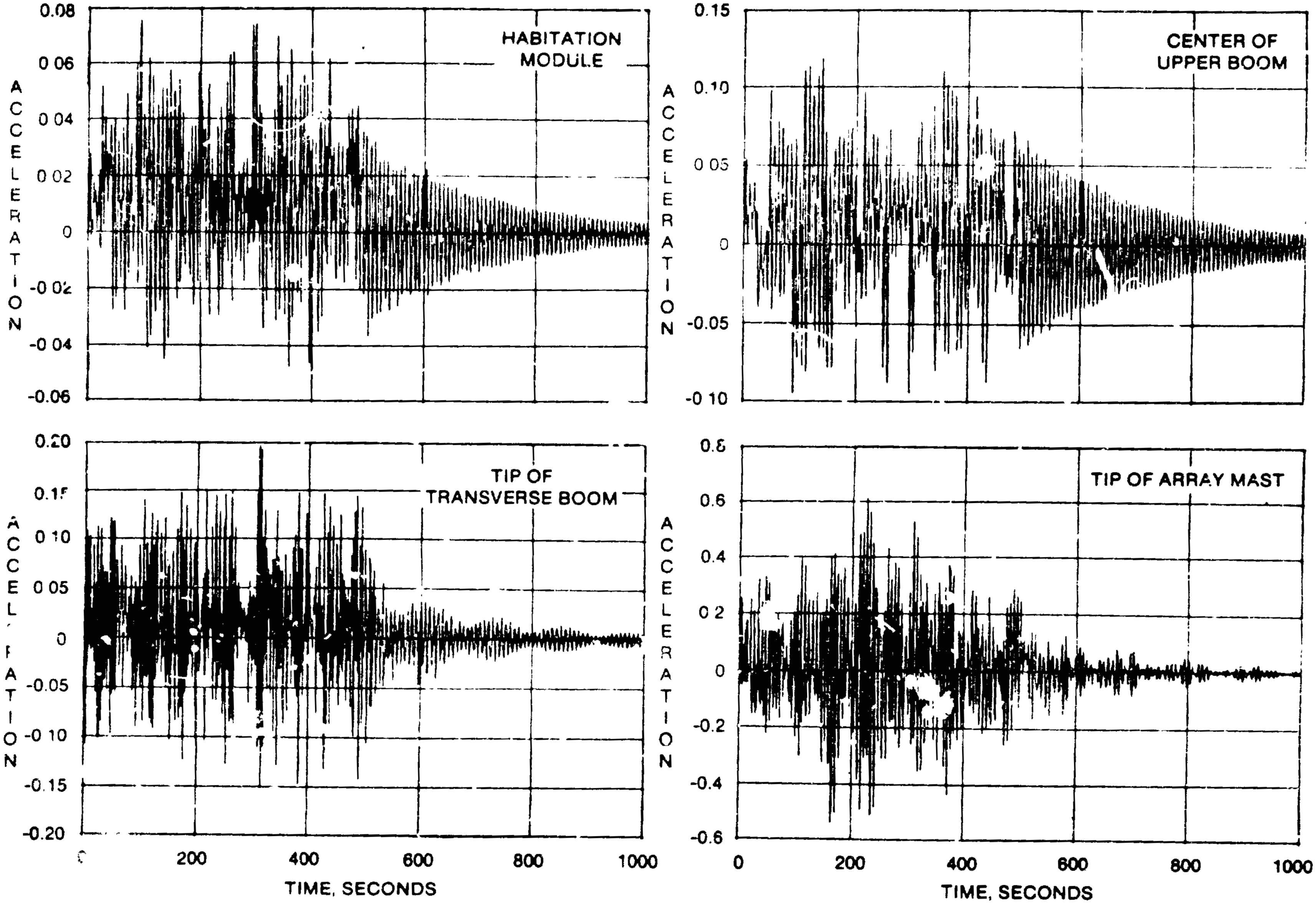


Figure 4.3.3.5-52 ACCELERATION TIME HISTORIES IN FEET/SEC^2 AT THE SELECTED STATION LOCATIONS DUE TO RCS REBOOST FIRING FOR IOC STATION WITH ATTACHED PAYLOADS

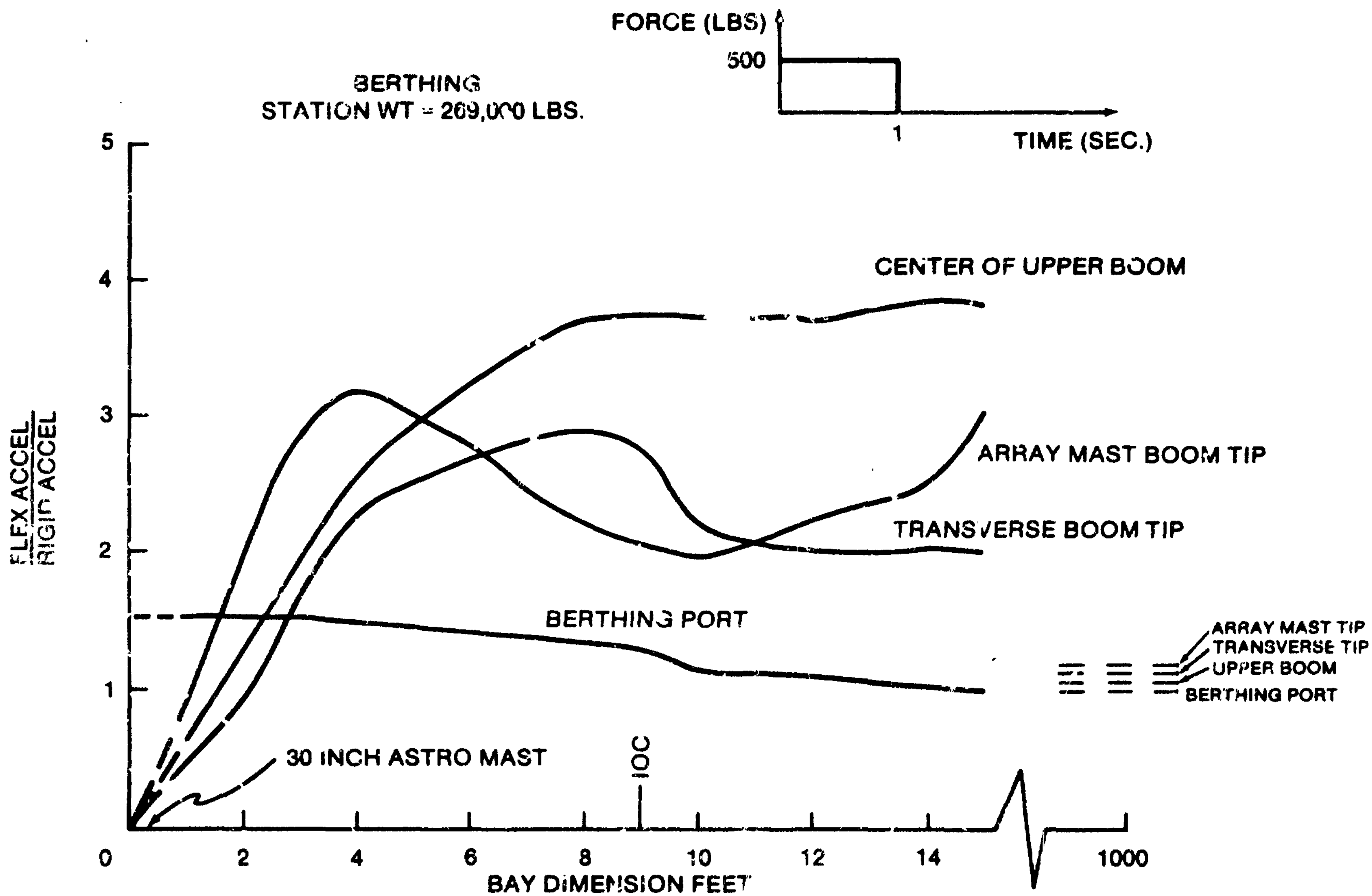


Figure 4.3.3.5-53 - VARIATION OF RESPONSE WITH BAY DIMENSION DUE TO ORBITER BERTHING

4.3.4 Power Capability and Utilization

4.3.4.1 Power capability

The electrical power system consists of a generation subsystem, storage subsystem, and management and distribution subsystem. The subsystems are of modular design to facilitate repair/replacement and growth. The initial station power system may be either solar dynamic or photovoltaic based. The growth station will be solar dynamic.

The power system provides electrical power to support station activities and is capable of accepting new technology as it becomes available, or as it is needed. The power system must accomplish these objectives while simultaneously minimizing adverse effects on station altitude and attitude maintenance and interference with Space Station customers.

The power system integration with other users of the Space Station is primarily through electrical interfaces. The identification of these interfaces helps determine the requirements and responsibilities of both the power system and the Space Station users. Electrical power produced by the solar arrays or the solar dynamic power system is transferred across beta joints and distributed to the solar photovoltaic energy storage subsystem and management and distribution elements outboard of the alpha joints. The DC power from the generation and storage subsystems (or AC from the solar dynamic power system) is converted to high frequency AC power and transferred across the rotary alpha joints. Power is then distributed via power lines along the upper and lower keel to various Space Station users. This power is distributed to attached payloads and communication antennas along the upper boom. Likewise, power is distributed to the reaction control system modules, mobile manipulator system, thermal control system, Space Station modules, and other attached payloads on the lower keel. The management and distribution subsystem controls the power and provides individual protection for each user interface, power source, and subsystem hardware and wiring. A properly designed management and distribution subsystem can function with several different generation subsystems, thus accommodating Space Station growth.

The area requirements and configuration of the power generation subsystem have a major effect on power system integration with the station. The drag force experienced by the station is directly related to the area requirements, and the drag force affects the orbit altitude of the station and the amount of propellants required for altitude maintenance. As the power system grows in size and power requirement, the impact of drag forces will become more severe. Also, in the Space Station reference configuration most of the station mass is near the bottom of the keel and most of the drag force is near the top. Smaller area requirements will thus allow the station to experience smaller aerodynamic torque and to fly more nearly vertical. The generation subsystem configuration was selected to minimize gravity gradient roll torques at high beta angles. The spacing of the generation subsystem elements from the main boom and from each other was based primarily on shadowing and viewing considerations. This spacing also affects the Space Station keel length, because interference by the volume swept by the generation subsystem elements must be avoided.

4.3.4.2 Power utilization

4.3.4.2.1 Introduction

The issue of power system size and capacity is fundamental to size and operational capability of the Space Station. There is a natural desire to be "power-rich" at the inception; however, cost considerations dictate a compromise which will require close modeling of power use. The power utilization analysis is an attempt to provide realistic time histories of power consumption for each subsystem.

The technique requires an electrical equipment list which defines the power draws for various operating modes for each piece of electrical equipment. In addition to a crew timeline, a definition of equipment operation is required in the form of activity blocks. These blocks correspond to a time period for which a particular activity is performed. Another set of input data defines the link between the power levels that exist for equipment operation and activity blocks and timelines. The resulting analysis produces a variety of specific information which describes the power configuration of the Space Station at any point in time and the associated statistics for the system.

This technique has been developed and utilized on the Shuttle. The technique has been applied previously to a Space Station design study and a study for an eight-man, 120kW station which preceded the current study.

An initial output was a set of four guidelines (Table 4.3.4.2-1) for the use in the development of the subsystems.

4.3.4.2.2 Reference crew activity

The two 12-hour shifts provide a nominal operation of 24 hours a day for the crew of six with a 6-day workweek for each crewmember. The crew complement is two station specialists and four mission specialists. The Station specialists work 9 hours on operations, 8 hours on overhead activities such as eating, exercise, planning, and personal time. The mission specialist split of activities is 9 hours and 7 hours. All crewmembers have 8 hours for sleep.

TABLE 4.3.4.2-1 GUIDELINES FORMULATED BY THE POWER UTILIZATION GROUP

1. DON'T LEAVE REDUNDANT SYSTEMS POWERED, ROTATE THE USE OF REDUNDANT STRINGS WHERE POSSIBLE

2. USE LOW WATTAGE AREA LIGHTING AND APPROPRIATE TASK LIGHTING FOR WORK AREAS

3. DON'T USE ELECTRICITY TO MAKE HEAT IF THERE IS A COST EFFECTIVE WAY TO USE EXISTING HEAT SOURCES

4. THE STATION WILL SUPPLY CONDITIONED UTILITY GRADE POWER. DO NOT SPEND POWER AND WEIGHT TO RE-CONDITION THE POWER AS ON ORBITER (12% OF POWER)

4.3.4.2.3 Analysis considerations

The power analysis has certain assumptions which affect the results:

a. The power is provided to the subsystems and payloads at the bus, which means that the power distribution and control subsystem accounts for the inefficiencies associated with the delivery and conditioning of the power.

b. Interior low-level area lighting is provided for normal transit through modules. Dedicated task lighting is provided for work places at 125 watts per person per task.

c. One active work station is provided for each crewman.

d. A significant portion of payload activities would be onboard computer or ground controlled.

4.3.4.2.4 Preliminary Results

The preliminary power profile for the total station is illustrated in figure 4.3.4.2-1. The power level exceeds the 75 kW system capability primarily due to the size of the reference payload complement (Table 4.3.1-1) and a current lack of understanding of payload operation phasing. The profile for the facility averages is shown in figure 4.3.4.2-2. These results should be taken only as indicative of the general operating range of the facility due to the preliminary nature of the subsystems definition. The distribution of power utilization among the subsystems is given in table 4.3.4.2-2. The profile for the payloads only is shown in figure 4.3.4.2-3.

The quality of this analysis is consistent with the maturity of the subsystem and payload data. The housekeeping power requirements should be interpreted as an indication that the Station can potentially operate below the 30 kW housekeeping level.

Prior experience and analysis performed to support the current study tend to indicate that station power consumption is not strongly affected by the number of crewmembers onboard or the level of crew activity. The power consumption is more strongly a function of the design size of the subsystems.

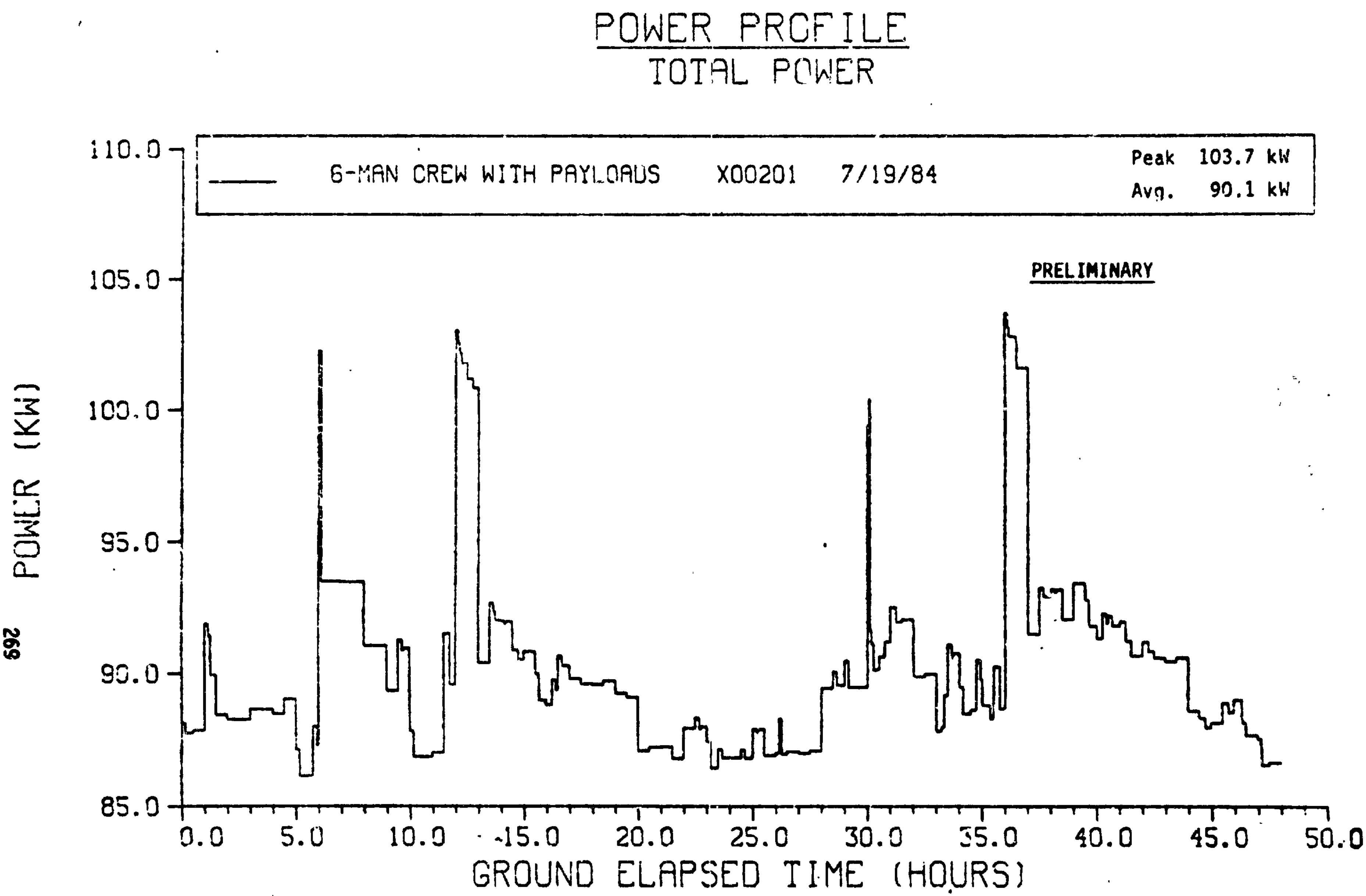


FIGURE 4.3.4.2-1 - TOTAL POWER PROFILE FOR 6-MAN CREW WITH PAYLOADS

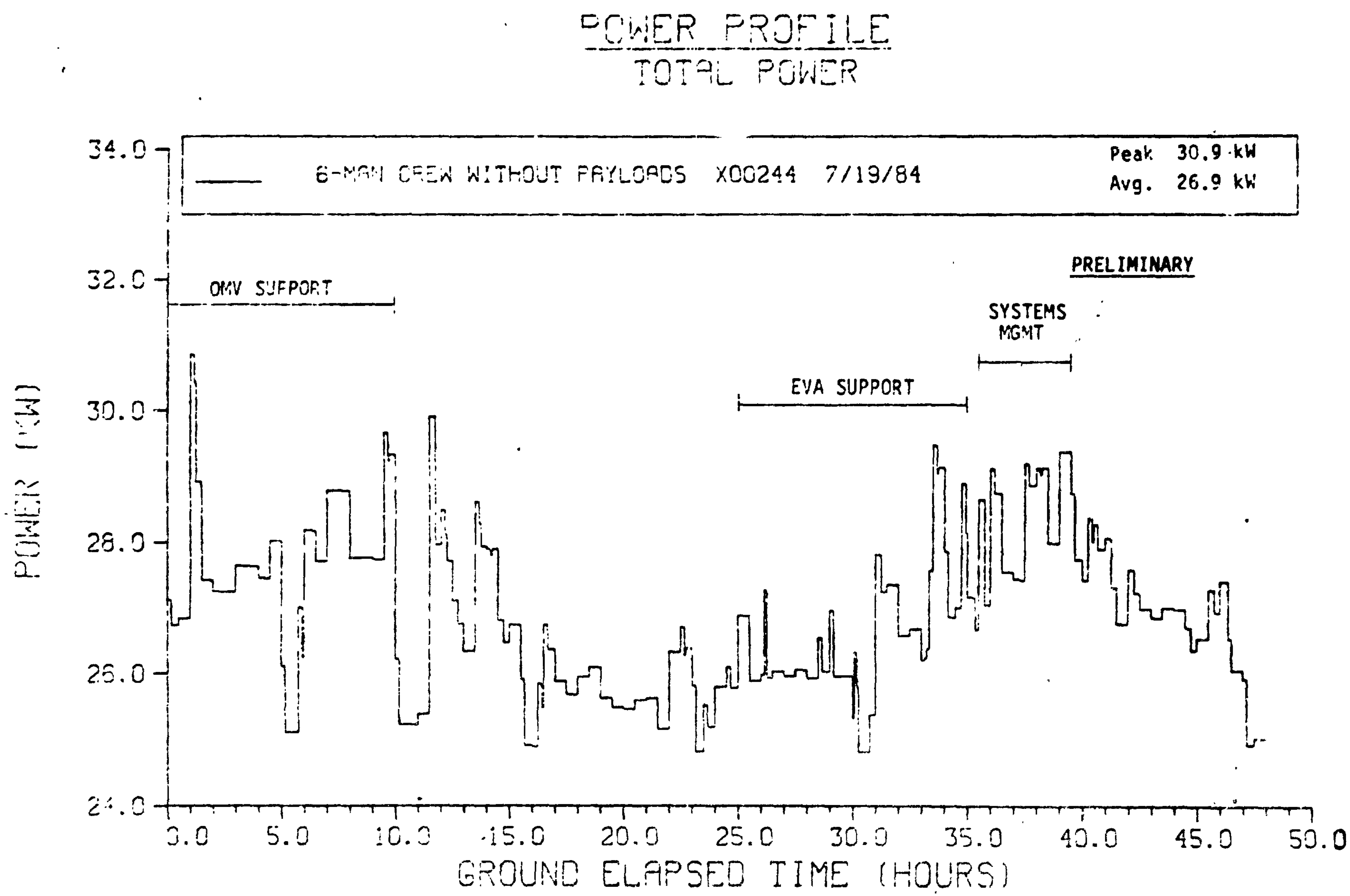


FIGURE 4.3.4.2-2 - TOTAL POWER PROFILE FOR 6-MAN CREW WITHOUT PAYLOADS

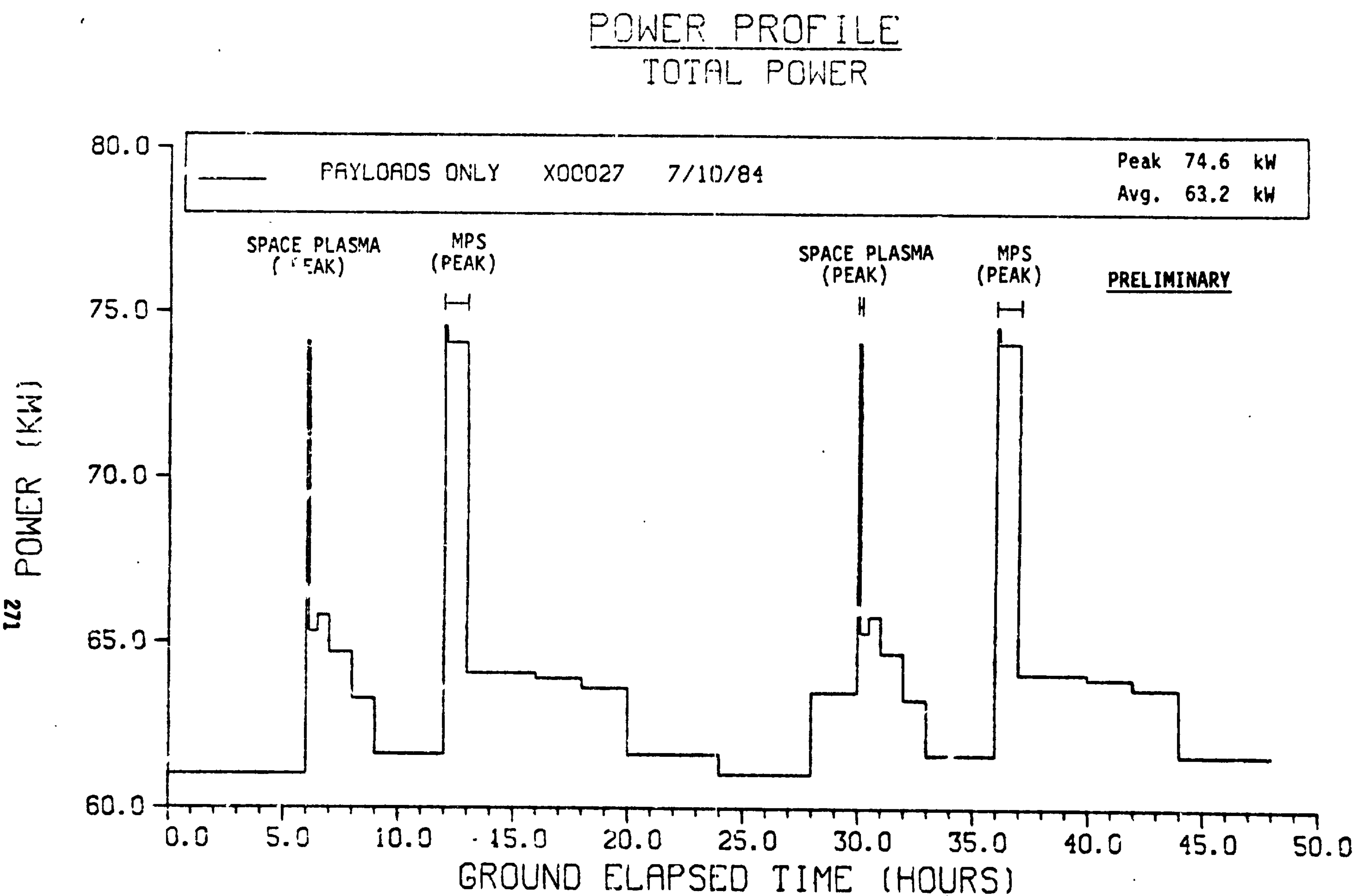


FIGURE 4.3.4.2-3 - TOTAL POWER PROFILE FOR PAYLOADS ONLY

TABLE 4.3.4.2-2 - PRELIMINARY DISTRIBUTION OF POWER UTILIZATION

	SUBSYSTEM	TOTAL POWER ALL COMPONENTS ON-KW	AVERAGE POWER KW OVER 48-HOUR MISSION	KWH CONSUMED	PERCENT W/O PAYLOADS
1.	POWER GENERATION	0.00	0.00	0.0	0.0
2.	POWER DISTRIBUTION	2.65	2.65	127.2	10.0
3.	GUIDANCE CONTROL	5.99	5.99	227.5	22.3
4.	NAVIGATION	0.00	0.00	0.0	0.0
5.	TRAFFIC CONTROL	0.00	0.00	[illegible]	0.0
6.	RF SYSTEMS	3.07	1.59	[illegible]	5.9
7.	ANTENNAS	0.89	0.24	[illegible]	0.9
8.	AUDIO	0.78	0.52	[illegible]	2.0
9.	VIDEO	4.65	2.53	[illegible]	9.4
10.	SIGNAL PROCESSING	1.46	0.55	[illegible]	2.0
11.	DATA ACQUISITION/TA/CDS	0.89	0.39	[illegible]	1.7
12.	TRACKING	2.63	0.12	[illegible]	0.5
13.	SYSTEM MONITORING/CONTROL	0.10	0.07	[illegible]	0.2
14.	DATA MANAGEMENT	2.68	1.50	[illegible]	5.5
15.	INTEGRATED D & C	5.70	0.85	[illegible]	3.1
16.	FACILTIES MANAGEMENT	0.00	0.00	[illegible]	0.0
17.	OPERATIONS/PLANNING/SUPPORT	0.00	0.00	0.0	0.0
18.	ORBIT PROPULSION	0.00	0.00	[illegible]	1.8
19.	FLUIDS MANAGEMENT	0.00	0.00	0.0	0.0
20.	PROPULSION STAGE MANAGEMENT	0.22	0.04	[illegible]	0.1
21.	ECLS	10.97	7.07	339.4	26.3
22.	EVA	6.21	0.35	[illegible]	1.3
23.	ACTIVE THERMAL CONTROL	0.26	0.26	12.2	0.9
24.	STRUCTURES	0.00	0.00	0.0	0.0
25.	MECHANICS	0.00	0.00	0.0	0.0
26.	PASSIVE THERMAL CONTROL	0.00	0.00	0.0	0.0
27.	CREW ACCOMMODATIONS	8.97	0.85	40.5	3.1
28.	HEALTH MAINTENANCE	3.82	0.53	[illegible]	2.0
30.	PAYLOAD SUPPORT	0.00	0.00	0.0	0.0
35.	RMS	2.77	0.27	[illegible]	1.0
	SUBTOTAL	65.21	26.89	[illegible]	100.0
29.	PAYLOAD OPERATIONS	118.04	63.16	[illegible]	-
	TOTAL	183.25	90.05	4321.7	-

4.3.5 Thermal capability

4.3.5.1 Introduction

The thermal design of a Space Station is highly dependent on the configuration and is therefore sensitive to seemingly slight changes in the configuration. The following thermal evaluation criteria were considered for this study:

a. Integrated station environment effects on thermal design requirements

b. Radiator area requirements

c. Maximum use of waste heat

d. Minimize sensitivity of thermal design to surface coatings contamination and degradation

e. Design complexity and maintainability

f. System power

g. Hardware commonality

h. Safe haven requirements

i. Technology status

4.3.5.2 Thermal integration

A totally integrated Thermal Control System which efficiently uses all heat sinks and sources is ideal. However, this idealism must be moderated by practicality. With this in mind, the Space Station configuration was reviewed to identify functions and systems requiring thermal control. Considering the evaluation criteria and the requirements specified in the RFP, levels of integration and rationale were formulated.

Table 4.3.5-1 presents the major identifiable functions/systems and the design approach and rationale selected. Where specific thermal requirements were not defined, such as OMV, OTV, and satellite servicing, assumptions were made as to the thermal design approach.

FUNCTION/SYSTEM	DESIGN APPROACH	RATIONALE
O LOGISTICS MODULE	O AUTONOMOUS ATCS (BODY MOUNTED RADIATORS)	O SIMPLIFIES MAKE AND BREAK INTERFACES
		O LOW HEAT REJECTION
	O THERMOS BOTTLE PASSIVE DESIGN INSENSITIVE TO COATING DEGRADATION	O MINIMIZES THERMAL LOAD EXCURSIONS ON ATCS AND LIMITS ENVIRONMENT OF UNPRESSURIZED VOLUME
O HABITAT AND LAB MODULES	O INTEGRATED ATCS FOR MODULE HEAT REJECTION THROUGH CENTRAL BUS RADIATORS	O EFFICIENT USE OF HEAT SOURCES AND SINKS
	O BODY MOUNTED RADIATORS	O SAFE HAVEN AND INITIAL BUILDUP CONSIDERATIONS
	O PASSIVE DESIGN - THERMOS BOTTLE INSULATION AND NON-RADIATOR SURFACE THERMAL OPTICAL PROPERTIES α/ε = 1, .8/.8	O MINIMAL THERMAL LOAD EXCURSION ON ATCS, NO SURFACE PROPERTY MAINTENANCE
O POWER SYSTEM	O AUTONOMOUS ATCS	O STATION/PLATFORM DESIGN COMMONALITY
		O MINIMIZE MAKE AND BREAK INTERFACES
	O PASSIVE TCS INSULATION AND COATINGS TO LIMIT COMPONENT ENVIRONMENTS	O LIMIT COMPONENT TEMPERATURES AND THERMAL LOAD EXCURSIONS ON ATCS

TABLE 4.3.5-1 TCS LEVELS OF INTEGRATION

FUNCTION SYSTEM	DESIGN APPROACH	RATIONALE
0 OMV STORAGE	0 NO THERMAL HARDWARE INTERFACE REQUIRED	0 OMV PROVIDES OWN THERMAL CONTROL USING STATION POWER
	0 ANALYTICAL DEFINITION OF STORAGE AREA INTEGRATED THERMAL ENVIRONMENT	
0 OTV HANGAR	0 BASIC DESIGN IS PASSIVE WITH INSULATION AND COATINGS TAILORED TO MEET OTV LIMITS	0 REQMTS. NOT IDENTIFIED 0 OTV DESIGN SHOULD CONSIDER SSP THERMAL ACCOMMODATION CAPABILITY
	0 ATCS CAN BE ADDED AS UNIQUE REQUIREMENTS ARE IDENTIFIED	
0 PRESSURIZED PAYLOADS	0 HEAT SINK/SOURCE PROVIDED BY ATCS INTEGRATED WITH CENTRAL BUS AND RADIATORS	0 ACTIVE THERMAL CONTROL REQUIRED UNLESS PAYLOAD PASSIVELY DESIGNED WITH LOW POWER CONSUMPTION
0 ATTACHED PAYLOADS	0 SAME AS ABOVE	0 SAME AS ABOVE EXCEPT PAYLOAD PASSIVE THERMAL DESIGNS CAN BE ACCOMPLISHED FOR LARGER POWER LOADS

TABLE 4.3.5-1 - TCS LEVELS OF INTEGRATION (CONTINUED)

O SATELLITE SERVICING/STORAGE/ REFUELING BAYS	O DEPLOYABLE THERMAL PROTECTION SHROUD TO LIMIT ENVIRONMENT FOR SOME SATELLITES	O REQMT. FOR PASSIVELY CONTROLLED ENCLOSURE IS INFERRED BY CONSIDERING POSSIBLE THERMAL DESIGNS OF VARIOUS SATELLITES
	O ANALYTICAL DEFINITION OF INTEGRATED THERMAL ENVIRONMENT	
O PROPULSION	O PASSIVE TCS DESIGN/ MAKE UP HEAT FOR TANKS AND FLUID DISTRIBUTION SYSTEM	O TCS DESIGN CAN BE ACCOMPLISHED FOR MINIMAL OR NO MAKE UP HEAT
	O PASSIVE TCS DESIGN FOR ENGINES AND HEAT SOAK BACK PROTECTION	O MINIMIZE MAKE AND BREAK INTERFACES AND PROPULSION DESIGN COMPLEXITY
O ANTENNAS	O PASSIVE TCS DESIGN UNIQUE TO EACH TYPE	O GIMBAL MOTORS/MECH. REQUIRES SIMPLE HEATING APPROACH (IF REQUIRED)
		O ATCS INCREASES MAKE AND BREAK INTERFACES, ANTENNAS DESIGN COMPLEXITY AND FLUID LINE RUNS
		O CONTINUOUS OPERATION OR STANDBY OF ANTENNAS PLANNED. THERMAL CONDITIONING REQUIRED ONLY FOR CONTINGENCIES

TABLE 4.3.5-1 TCS LEVELS OF INTEGRATION (CONTINUED)

4.3.5.3 System Overview and Capability

The candidate Active Thermal Control Subsystem (ATCS) functionally illustrated by Figure 4.3 5-1 is a hybrid design concept that maximizes the use of local thermal control for individual Station modules and satisfies the remaining thermal control requirements with a centralized system. The heat acquisition and transport function is accomplished with a thermal bus approach using a two-phase working fluid to transfer heat by evaporation and condensation rather than by sensible heat changes of a single phase coolant. As a result, the thermal bus essentially operates at a constant temperature over the entire length of the flow circuit with pumping power requirements that are very small compared to a single phase fluid system. Each manned Station module contains multiple two-phase water heat transport circuits to accommodate Station isothermality and heat load capacity requirements. With the exception of the Logistics Module, the manned module water circuits interface with appropriate central two-phase ammonia thermal bus circuits outside the pressurized cabin areas operating at about 35°F, 70°F, and 90°F. These central flow circuits in turn interface with an erectable, centralized heat pipe radiator system. This radiator assembly includes a gimbal system to orient the radiators toward a minimum thermal environment thus minimizing radiator size.

In addition, each manned Station module has heat pipe radiators (operating at about 35°F) integrated with the module meteroid protection shield. Although hardware commonality is a goal, the size of these body-mounted radiators may vary from module to module depending on surface area availability considering berthing ports, windows, thermal blockage, etc. To a limited degree, these module radiators augment the central radiators during normal operations. However, with the exception of the Logistics Module, the principal function of these radiators is to provide an autonomous safe haven capability and to possibly assist during the Station buildup and assembly. To avoid "make and break" fluid line operations for the logistics module, body-mounted radiators are used exclusively to reject waste heat.

A separate two-phase ammonia thermal bus (operating at about 115°F) is provided for electrical power system waste heat rejection from the regenerative fuel cells, electrolysis units, and power conditioning equipment. This flow circuit also includes a thermal storage unit to minimize the radiator size since the night/day orbital power system waste heat load varies by a factor of about 3.5. Heat pipe radiators are provided for rejection of the power system waste heat.

The IOC radiator areas for the reference ATCS concept are summarized below:

	Radiator Area (ft^2)*
Body-Mt	960 **
Central	4627
Power System	1226
Log Module	122

* Total radiating area
** Excludes Logistics Module

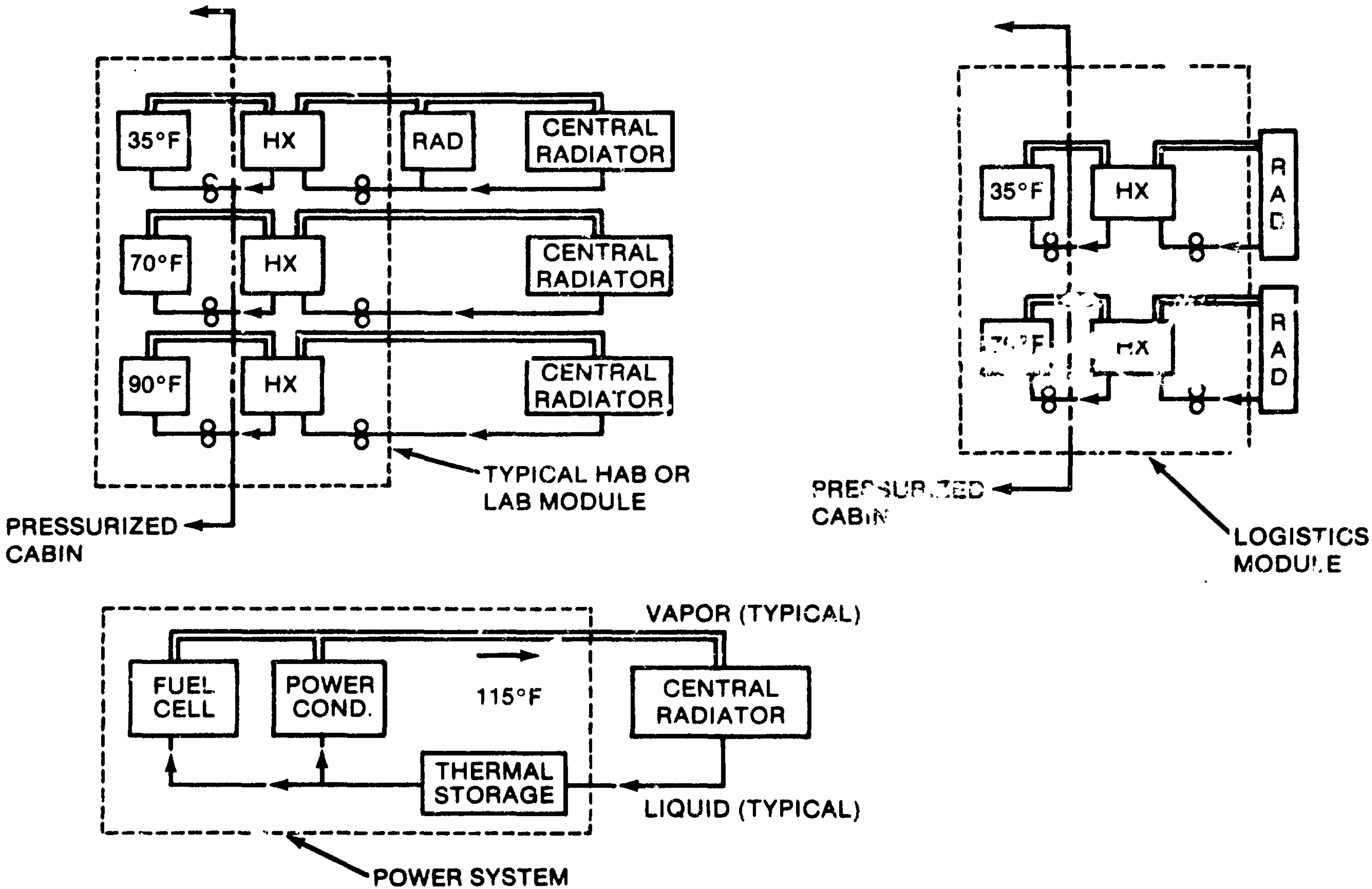


Figure 4.3.5-1 - ATCS DESIGN APPROACH

C-4

Virtually all elements of the reference configuration receive direct solar radiation in the local vertical flight mode. Ideally the passive thermal design of habitable modules for these conditions would use radiator-type coatings (i.e., low solar absorptivity to emittance, α/ϵ, ratios) in conjunction with an insulation design to minimize heat leak into the modules for those surfaces receiving sun while maximizing heat leak out for other surfaces. However, considering the desire to minimize coating refurbishment and the fact that the module total passive heat loss or gain is small in comparison to the total internal heat load (approximately 1 to 6 percent of the Habitat Module load), a thermos bottle insulation design was selected. This desensitizes the design to environmental variations and coating degradation and allows the selection of surface treatments such as anodized aluminum with an α/ϵ approaching 1.0 for the habitable modules and external airlocks.

The OMV servicing and storage areas do not require thermal protection since the OMV must be designed for the extremes of thermal environment to accomplish its mission. Also, no active thermal control interface is provided since the thermal design of the OMV will probably be completely passive.

Satellite storage, service, and refueling area requirements are varied in order to accommodate a large spectrum of satellite thermal design environments from stationary attitudes which represent the extremes of temperature capability to spinning modes which tend to a moderate range of temperatures. This implies a capability to provide thermal protection for some payloads in order to limit the range of environments to which the satellite is subjected. Temperature extremes for unprotected satellites with various α/ϵ thermal properties are presented in Section 4.4.7.4.3 as well as internal temperatures for a deployable 15-by-60-feet-long cylindrical thermal protection shroud as a function of internal power generation. A diameter greater than 15 feet will probably be required to meet access requirements. For those satellites requiring the thermal protection shroud the allowable satellite and equipment heat generation/power will be limited by the external α/ϵ ratio of the shroud and the allowable maximum temperature environment of the payload. Therefore, the ability to design the shroud to eliminate surface property refurbishment may be limited. The shroud thermal design must be based on a review of payload and supporting equipment power and temperature requirements and servicing/refueling operations.

The OTV hangar, 33 by 33 by 66 feet, is considered to have a completely passive thermal design composed of multilayer high performance insulation and appropriate hangar external surface thermal optical properties. Actual performance requirements must be selected based on detailed OTV design requirements and servicing operations. Active thermal control interfaces with the central bus can be added as these requirements are identified.

For an α/ϵ of 1.0, an internal heat load of approximately 2kW can be accommodated without exceeding an average internal hangar temperature of 100°F.

Propellants for satellite, OMV, and OTV refueling and the station contingency propellant are located within the truss work just below the lower keel in the keel extension. Insulation closeouts are provided to form an enclosure around the tanks. Since the design approach for the cryogenic fluids (hydrogen, helium, nitrogen) has not been selected, the thermal design is left as an open issue. However, it is expected that an approach similar to that of the mono- and bi-propellants can be applied. Applying an α/ε = 1.0 (0.8/0.8) external surface treatment in conjunction with high performance multilayer insulation results in minimum makeup heat to maintain the mono- and bi-propellants between a 40°F and 100°F operating range. For each 9-foot truss cube containing the tanks, 54 watts of makeup heat are required. The makeup heat is provided by electrical heaters. Where practical, propellant lines will be run parallel to the central thermal bus with a common insulation cocoon, thereby obtaining required makeup heat from the thermal bus transfer lines. Those lines not wrapped with the thermal bus lines require approximately 0.27 watts of makeup heat per foot. Makeup heat for operation of the catalytic beds for the station RCS engines was not established since this is highly dependent on the detailed engine design and structural integration approach.

The 90-day station propellants are located in the unpressurized section of the logistics module. The design of the logistics module is similar to that of the habitation and laboratory modules. In addition, the body-mounted radiators are mounted to the greatest extent possible on the unpressurized portion of the module to provide an acceptable boundary temperature for hardware located in the unpressurized compartment. A thermal curtain (multi-layer insulation) closes out the open end of the compartment.

The number and type of antennas for the reference configuration are varied and their particular thermal requirements vary depending on location, internal power generation and duty cycle. Since the antennas and associated avionics are expected to be either in an operational or standby mode at all times, a passive design similar to that of the Space Shuttle Orbiter Ku-band is envisioned rather than integration into the Active Thermal Control Subsystem.

Passive thermal control of the solar arrays can be obtained to a small degree by optimizing the back face thermal optical properties. For an α/ε of 0.5 and 0.7, the maximum and minimum temperatures during an orbit are 166°F and -135°F, and for an α/ε of 0.2 and 0.9 they become 138°F and -142°F, respectively. More complete results are presented in Section 4.4.7.4.7. Final selection of surface properties and materials treatment must consider the solar array detailed design and evaluation of the contamination environment to which the array will be subjected.

4.3.6 Communications and Tracking Antenna Systems

4.3.6.1 Required Antenna Coverage (Field-of-View) and Operational Zones

4.3.6.1.1 Operational zones

The operational zones specified for the Space Station Program are depicted in Figure 4.3.6-1. These consist of nine zonal areas: (1) proximity operations, (2) control, (3) departure, (4) rendezvous, (5 & 6) co-orbital, (7 & 8) non-co-orbital, (9) parking (see Section 4.3.8.2.1). Only zones 1 through 6 will be the responsibility of the Space Station for communications and tracking.

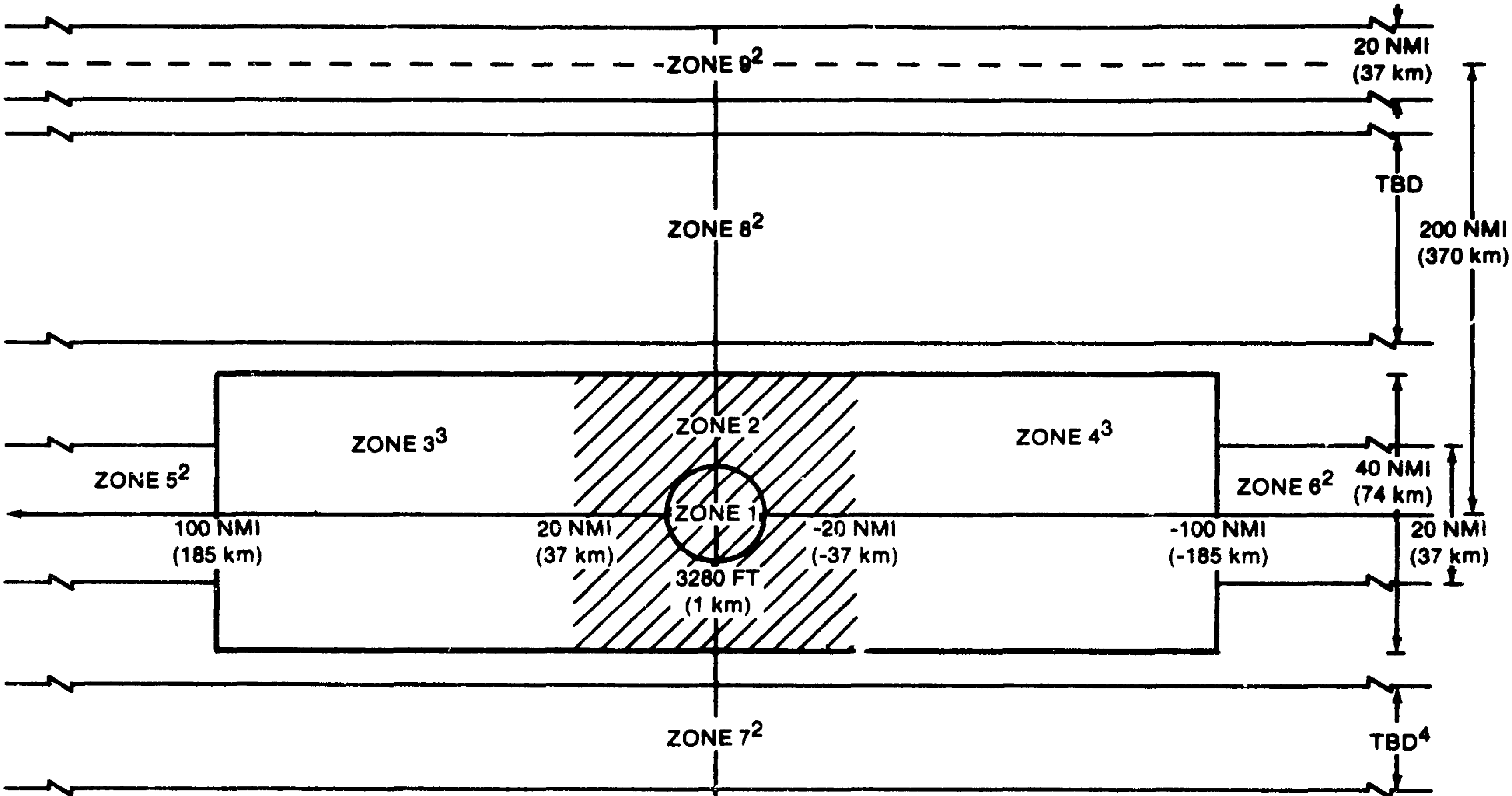
Figure 4.3.6-2 depicts communications and tracking system coverage of these operational zones. The approach taken was to cover zone 1 and the control zone with an antenna system which can achieve spherical coverage out to 20 nmi since Space Station control of operations is limited to this area. Zones 5 and 6 are covered by antenna systems which produce a conical beam of $\pm 8.5°$ out to 1080 nmi (line of sight grazing the atmosphere).

These antenna systems will be described in Section 4.3.6.2.

4.3.6.1.2 Coverage required for an earth-oriented Space Station

a. Antenna coordinate system: Figure 4.3.6-3 shows the coordinate system used to define the cone and clock angle direction (theta and phi respectively) from the Space Station antennas to any other vehicle. For an earth-fixed orientation, the z-axis is pointed toward zenith (opposite the nadir direction), the x-axis is pointed along the Space Station velocity vector, and the y-axis is perpendicular to the orbit plane (POP). The boresight or peak gain direction of the antenna is usually assumed pointed along the z-axis, so that the theta/phi plots will show the angular variation relative to the antenna boresight.

b. TDRS: Two Tracking and Data Relay Satellites (TDRS) will be located at west longitudes of 41° and 171°, with a spare at 83°. Since the TDRS is in synchronous orbit, the look angle from the Space Station to TDRS will vary very slowly over a short period of time. However, in a 1-day period the cone and clock angles vary over a wide range. This can be seen in figure 4.3.6-4 where the theta and phi angle variation is plotted for one 24-hour period. One orbit pass corresponds to one horseshoe shape set of points around the y-axis. Each point is separated in time by two minutes. For the whole year the cone and clock angles will fall within the shaded area shown in the figure. Note that the theta angle is never greater than 100°. The earth's horizon is assumed to be at 80° corresponding to about the top of the ionosphere. Note the large angular area over which the TDRS antenna does not have to look. This allows greater flexibility in the location of the antenna on the Space Station since such a large area of obscuration is available without interfering with the TDRS look angle directions.



ZONE ASSIGNMENTS:
1. PROXIMITY OPERATIONS ZONE
2. CONTROL ZONE
3. DEPARTURE ACTIVITY ZONE
4. RENDEZVOUS ACTIVITY ZONE
5. LEADING COORBITING SATELLITE ZONE
6. TRAILING COORBITING SATELLITE ZONE
7. LOWER NON-COORBITING SATELLITE ZONE
8. UPPER NON-COORBITING SATELLITE ZONE
9. PARKING ORBIT ZONE

NOTES:
1. THIS DRAWING IS CURVILLINEAR AND IS NOT DRAWN TO SCALE.
2. THIS ZONE IS CONTINUOUS ABOUT THE EARTH
3. THIS ZONE BEGINS AT THE SPACE STATION
4. THE MINIMUM OPERATING ALTITUDE FOR THIS ZONE IS ~ **100 NMI (~185 km)**

Figure 4.3.6-1 - OPERATIONAL CONTROL ZONES.[1]

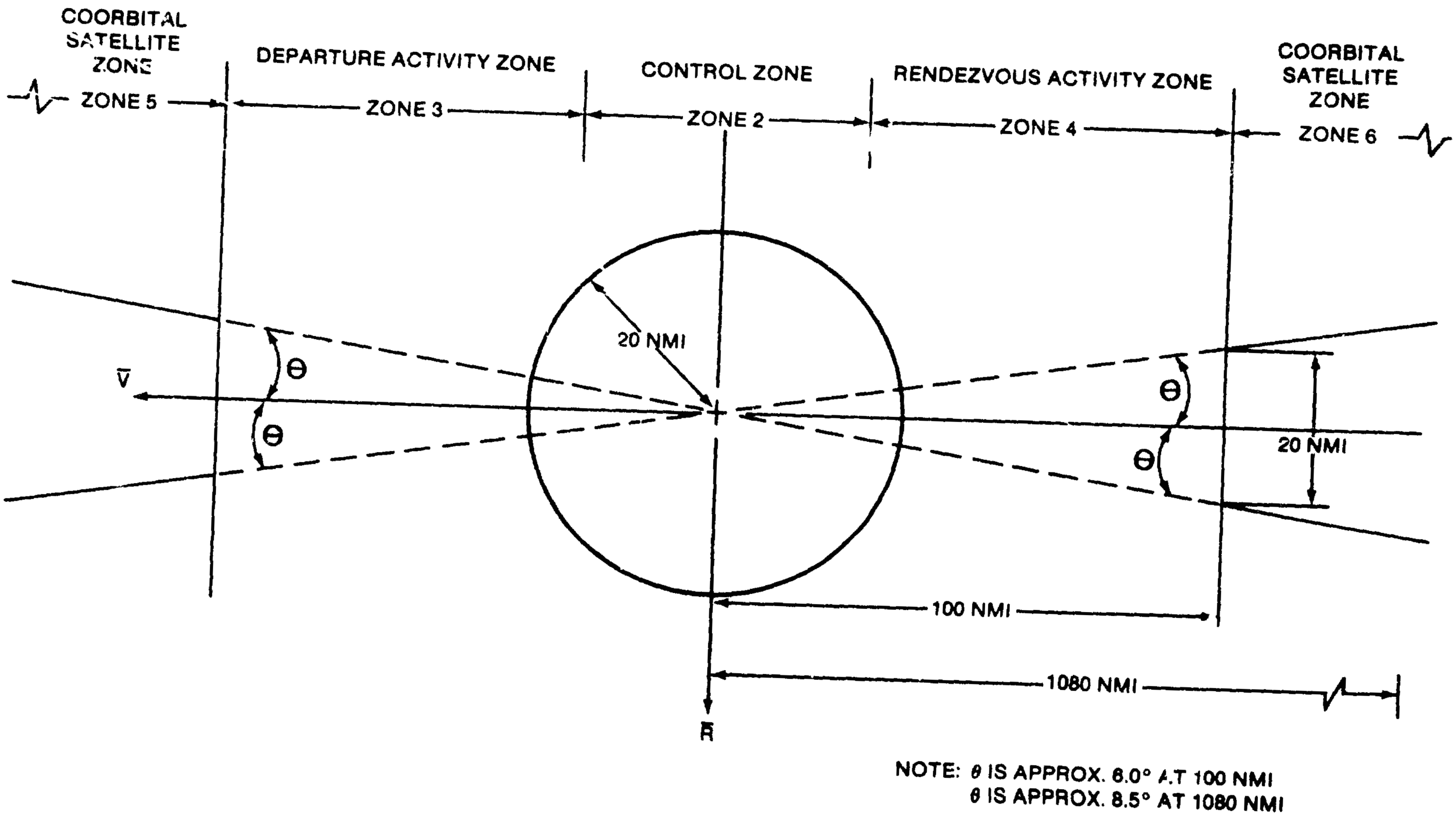
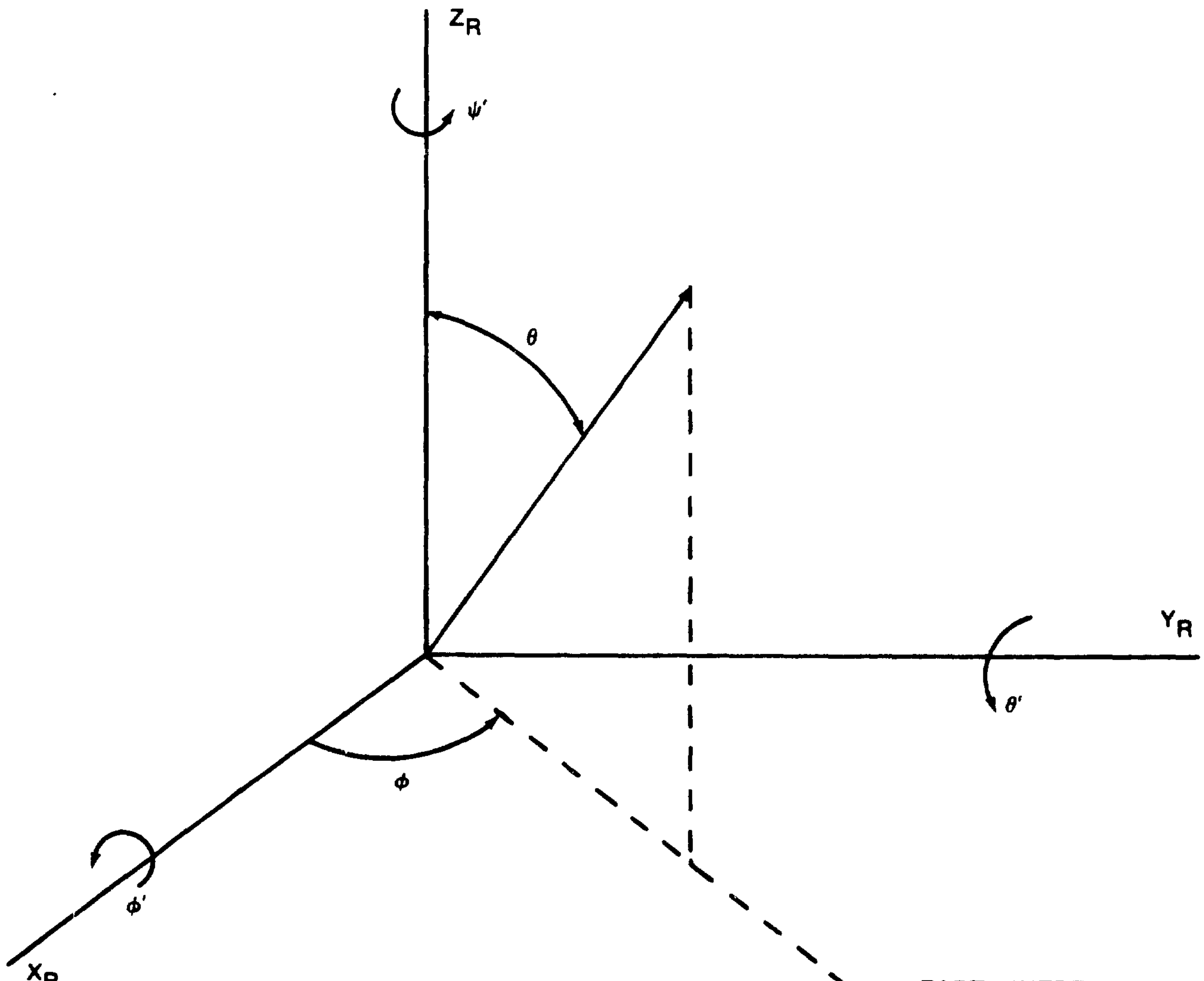


Figure 4.3.6 -2 ANTENNA COVERAGE OF OPERATIONAL ZONES.



ANTENNA CO-ORDINATE SYSTEM

EARTH INERTIAL

Z_R → ZENITH (OPPOSITE NADIR)
X_R → ALONG ORBIT VELOCITY VECTOR
Y_R → POP

Figure 4.3.6-3

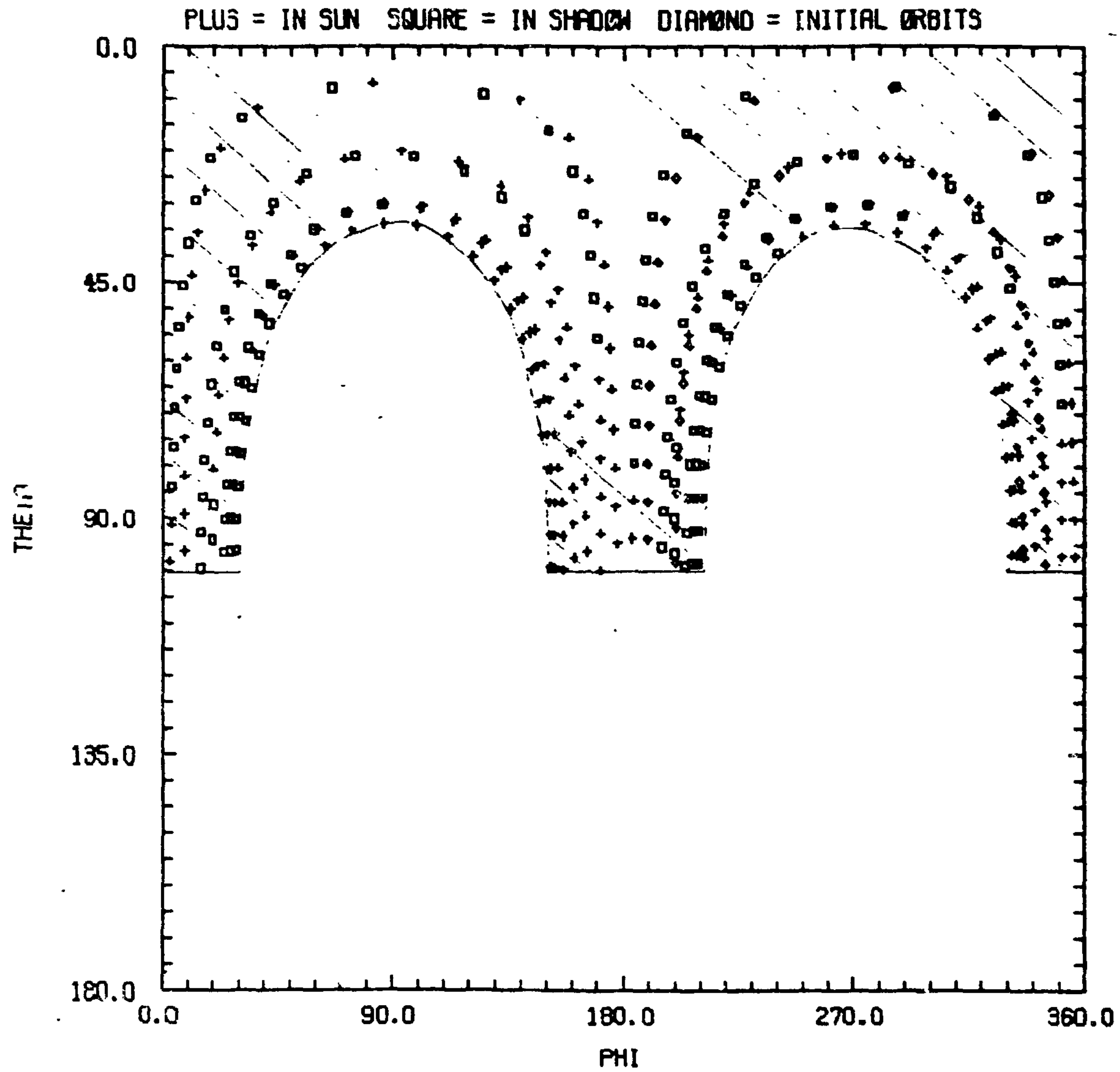


FIGURE 4.3.6 -4 ANGULAR COVERAGE TO TDRS FOR AN EARTH ORIENTED SPACE STATION. ANTENNA Z-AXIS POINTED 180^{0} FROM NADIR, X-AXIS POINTED ALONG FLIGHT PATH.

c. GPS: The GPS look angle requirements are similar to those of the TDRS. However, since the GPS satellites are at a highly inclined orbit (55°), there are periods when the cone and clock angle directions are all near the positive and negative y-axes. Thus for GPS, unobscured view of the upper hemisphere (theta = 0° to 90°, phi = 0° to 360°) will be required so that at least four GPS are always in view.

d. Co-orbiting S/C: With the Space Station in an earth-fixed nadir pointed orientation the $\pm 8.5°$ cone coverage required around the forward and aft tracks stays fixed throughout the orbit. However, small roll, pitch, and yaw excursions of about 5° can occur. Thus, with one antenna pointed forward and one pointed aft an angular cone of at least $\pm 14°$ about both the positive and negative x-axes must remain unobscured. If larger angular rotations of the Space Station about the roll, pitch and yaw axes are expected, then larger unobscured fields of views will be necessary.

4.3.6.2 Antenna System Description

4.3.6.2.1 Communication links

4.3.6.2.1.1 TDRS

Two 9-foot diameter parabolic antennas will be used to communicate to the ground via the TDRS. Two antennas are necessary in order to minimize link outages during handover between the two TDRS satellites. For a 10- to 15-minute period each orbit both TDRS satellites are in simultaneous view. During this period the communication link must be transferred from the TDRS satellite that is setting to the one that has just risen above the horizon. If one antenna is used to track both satellites, considerable data outage will occur during the period required to slew the antenna 130° from one TDRS to the other and to re-acquire the TDRS signal. To minimize this outage period high slew rates would be necessary, requiring much higher power to drive the gimbals. Low power is required to drive the gimbals during the normal tracking periods since the look angles change very slowly. Each antenna will contain a dual feed system, one for S-band and one for Ku-band. A tracking feed will also be incorporated to maintain antenna pointing initial acquisition. The RF transmit and receiving electronics will be located near the feed to minimize cable losses.

During the initial build up of the Space Station, two hemispherical coverage low gain (0-3db) S-band antennas will be used to provide a low data rate S-band link to both the TDRS and the Orbiter. Small microstrip antennas can be used for this application. These antennas are used for the TDRS link when the Orbiter is not present. When the Orbiter is in the vicinity of the Space Station these antennas are switched to the Orbiter link. The TDRS link from the Space Station is turned off. Communication to the ground is now accomplished via the Orbiter TDRS link. When the Space Station Assembly is completed, these antennas, after relocation, will be used to provide a low data rate S-band link to

TDRS as a backup in the event of loss of tracking by the 9-foot high gain antennas. Separate RF transmit and receive electronics are located near each of these antennas.

4.3.6.2.1.2 Orbiter

As stated in the previous paragraph, low gain near hemispherical coverage S-band antennas will be used to communicate with the Orbiter in the near proximity of the Space Station during the assembly phase. After the reference configuration is completed, two additional low gain S-band antennas will be added to provide coverage to the Orbiter during proximity operations.

These antennas can provide coverage out to a range of about 5 n. mi. Beyond this range a low gain (10 db) horn antenna will be used to provide communication to the maximum range of 20 n. mi. specified in the requirements. Two steerable horn antennas will be required to provide coverage along the forward and aft zones. Since these antennas have a beamwidth of about 50 degrees, a single axis gimbal will be required to provide hemispherical ($\pm 90^{\circ}$) coverage in the plane of the orbit.

4.3.6.2.1.3 EVA, OMV, FF, Platform

The communications link to these various users will operate at K-band (25 to 27 GHz). Section 4.3.2.1 and Figure 4.3.6-2 discuss the required operational coverage zones. As pointed out in that section, two basic coverage zones will be provided. These are the near range control zone and the far range co-orbital zone. Two antennas systems will be used to cover these zones. In the near range control zone (20 nmi sphere) a multiple access link will be used to communicate with the EVA's and OMV. This zone will be covered by a set of switchable antennas with overlapping beams to provide hemispherical coverage. Figure 4.3.6-5 shows the mounting configuration of seven horn antennas, and their transmit/receive electronics, which can provide hemispherical coverage. The peak gain of each antenna is approximately 9 db with a crossover gain between antennas of 5 db.

A separate transmitter and receiver are connected to each antenna so that multiple users can be served. The signal received from each antenna is down converted to a different IF frequency, corresponding to a different beam. Each IF frequency is multiplexed and transmitted by cable to the control center, where the signals are again separated and converted to a common IF frequency. A switch unit connects the mixer output to the demodulator corresponding to a particular beam. Several user signals can be present in the same beam. In this situation more than one demodulator would be connected to the same mixer. The demodulator selects only the signal that matches the identifying code fed to it. CDMA has been assumed for this design. TDMA could be used with a simpler design. The switching unit would be located at the antenna, and the multiplex units eliminated using a single IF frequency. Using TDMA requires an increase

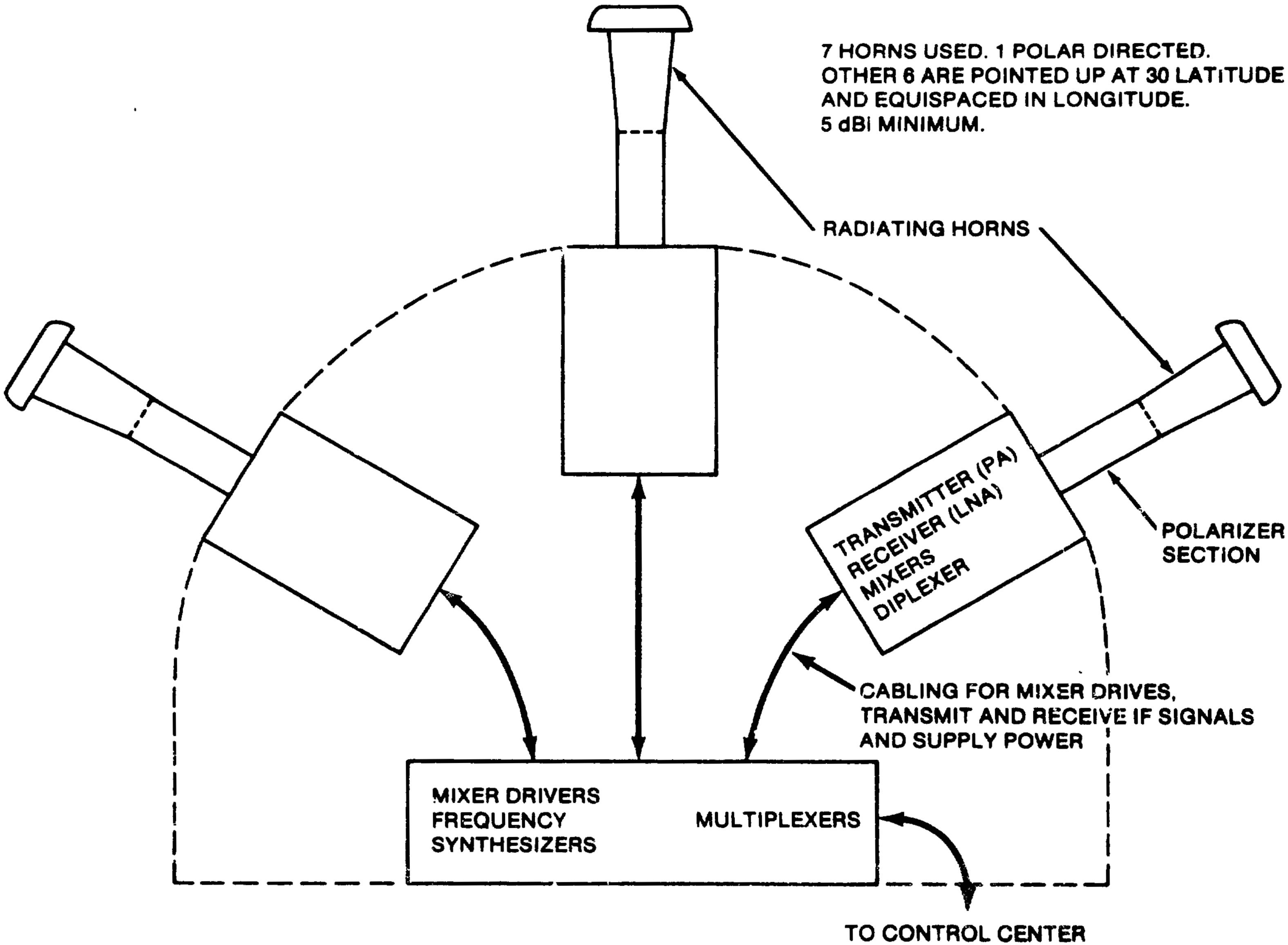


Figure 4.3.6 -5 NEAR RANGE ANTENNA SYSTEM

in the transmitter power in a direct ratio to the number of users being served. Higher power microwave amplifiers may not be available at 25 GHz. For this reason and other considerations the use of TDMA may be precluded. A more detailed trade-off analysis will be necessary before a multiple access system design is chosen. In any event this antenna design will work with any MA system.

An antenna switching arrangement, similar to that used for the received signal, is incorporated for the transmitted signal. The transmit and receive signals are connected to the same antenna using a deplexer, and RF filters to provide isolation. Additional single element broad beam antennas with 0-3 db gain will be used to provide extra coverage to EVA activity in and around the immediate Space Station structure.

For the initial Space Station configuration a single access communication link will be used to communicate with the platform and FF in the far range co-orbital zone (100-1080 nmi). To provide this coverage a 28-inch diameter K-band steerable dish is used. The peak gain of this antenna is 43 db. This is the gain required for the maximum range of 1080 nmi. At this range a FF located at the -10 nmi edge of the coverage zone will be at an angle of -8.9^{0} from the Space Station velocity vector. At +10 nmi this angle is 7.8^{0}, for a total cone angle coverage of 16.7^{0} ($\pm 8.5^{0}$). To cover this angular zone a two axis gimbal system is required for pointing the antenna toward the communicating user. The antenna must be realigned each time the link is switched to a new user. Two of these antennas are required; one pointed forward and one pointed aft. A separate set of RF transmit and receive electronics will be mounted near the feed of each of these antennas.

In the Space Station growth configuration a multiple access capability will be incorporated for the far range co-orbital zone. To provide this capability the 28 inch diameter dish will be replaced with a multiple beam offset reflector antenna. Figure 4.3.6-6 shows the design of a multibeam offset reflector antenna which can provide the required coverage. This antenna contains about 100 switchable feeds, which can provide beams anywhere in the $\pm 8.5^{0}$ cone. The peak gain of this antenna is also 43 db.

The antenna boresight must be pointed about $.5^{0}$ below the orbit track so that the $+8.5^{0}$ cone coverage of the antenna can reach the lower scan angle of 8.9^{0}. A two axis gimbal system is required for proper alignment of the antenna system boresight. This gimbaling system will allow for realignment of the boresight which may be needed to compensate for any small pitch, roll or yaw excursions. Two antennas are required, one pointed forward of the Space Station, the other pointed aft.

Each feed of the new multibeam far range antenna system contains both a transmit and a receive module, each consisting of a filter, mixer, buffer amplifier, and either a power amplifier (PA) or low noise amplifier (LNA). The transmit and receive modules are diplexed to the feed. The signals to and from the transmit and receive modules and the control center are at IF frequencies. A mixer frequency is also provided for

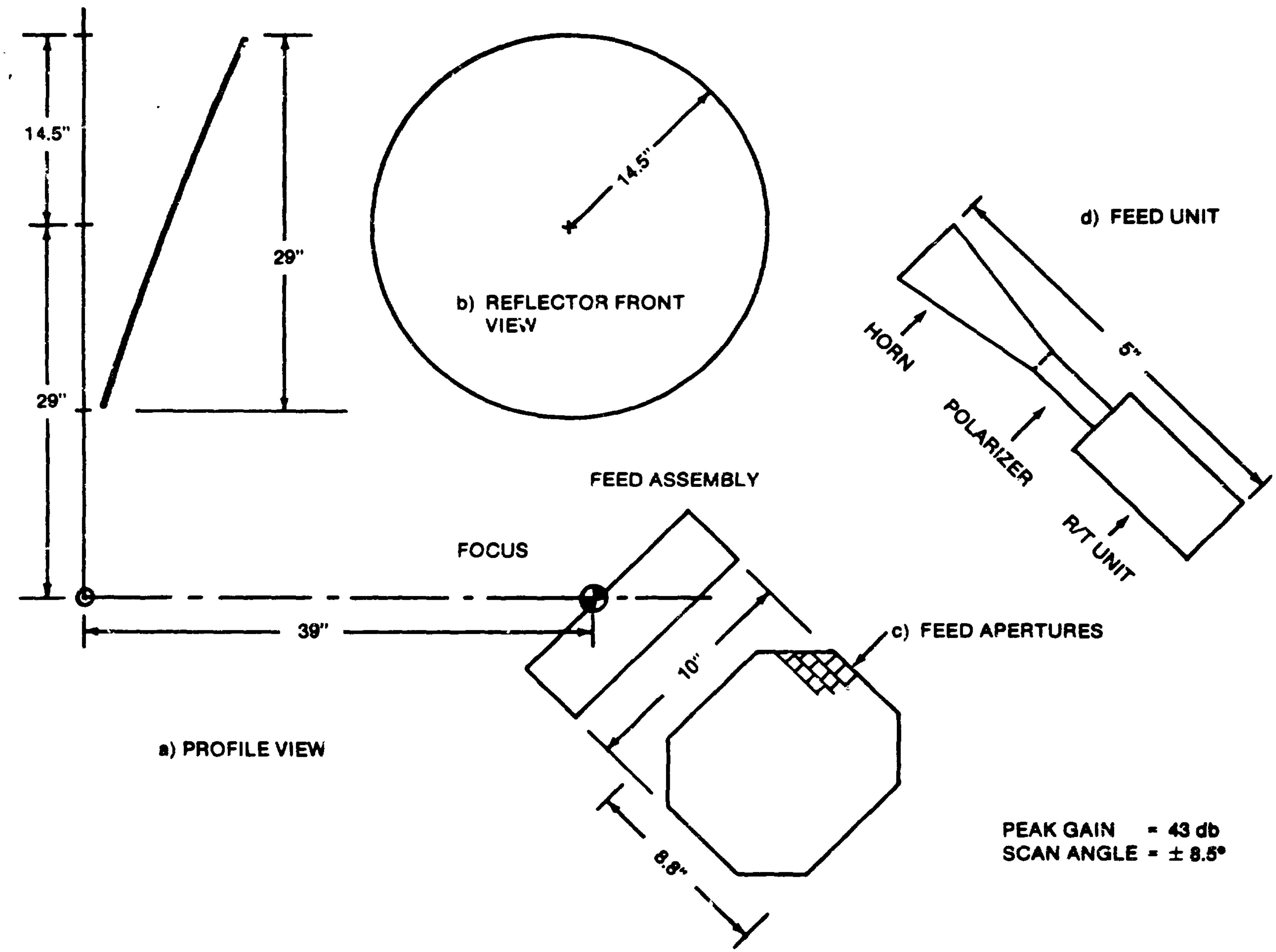


Figure 4.3.6 -6 - ANTENNA SYSTEM FOR FAR RANGE.

the up and down heterodyne conversion. A beam switching matrix is incorporated at the antenna to switch the IF signal to the correct beam. Up to four beams can be operated simultaneously. A gate signal is provided to enable the LNA and PA of the operating beams, and to disable the corresponding amplifiers on all the non-operating beams.

In the growth configuration a higher gain (50 db) antenna will also be incorporated to provide high data rate (25 mbps) capability out to the maximum range (See Table 4.4.3-1) . This antenna is a 5-foot diameter steerable dish with a single beam. Only one user can be serviced at a time. Two of these antennas are also needed to cover the forward and aft directions. Since the beamwidth of this antenna is about .5°, a two axis gimbal will be required to realign the antenna to a different user in the $\pm 8.5^{\circ}$ ($\pm 14^{\circ}$ for small attitude angle excursions) coverage zone.

4.3.6.2.2 Tracking and navigation

Two antenna systems are used to provide the Space Station tracking and navigation functions. One is a rendezvous radar antenna using a steerable 3-foot diameter dish operating at Ku-band. Two of these antennas are required to provide tracking of the Orbiter or OMV anywhere in the required 20 nmi sphere. The other is a low gain (0-3 db), dual switched, hemispherical coverage antenna, which can receive position location information from the GPS satellites. Small microstrip antennas can be used for this application. Two of these antennas will provide the hemispherical coverage required.

An additional pair of low gain (0-5 db) Ku-band microstrip antennas will also be included for use with a rendezvous radar transponder which operates with the Orbiter rendezvous radar. The radar transponder is included on the Space Station in order to improve the tracking capability of the Orbiter rendezvous radar, especially during the assembly phase. It also aids in distinguishing returns from the various parts of the Space Station structure, and thus avoids parallax errors when approaching the docking port.

4.3.6.2.3 Mobile Remote Manipulator System (MRMS)

To transmit commands and receive TV and TM from the MRMS, an S-band link will be used from the Space Station to the MRMS. One low gain near hemispherical coverage microstrip antenna will be located on each end of the MRMS. Two onmidirectional antennas with a dipole type pattern will be mounted on the Space Station. A biconical antenna or an array of microstrip antennas mounted around a cylinder could provide the beam pattern coverage required. The antenna will have to be mounted away from the Space Station structure and the pattern shaped to minimize the effects of reflections from the structure. One of these antennas will be mounted on the transverse boom for use during the assembly phase. As the Space Station is completed these antennas are moved to their permanent locations.

4.3.6.3 Antenna Locations for the Reference Configuration

Table 4.3.6-1 lists the total antennas to be used on the reference configuration. Included in the table are the antenna type, frequency of operation, and the elements served by each antenna. In the initial configuration antennas Nos. 2 and 3 will not be incorporated. As pointed out in Section 4.3.6.2.1.3, a 28-inch diameter K-band steerable dish antenna will be used in the initial configuration instead of the multibeam antenna listed for antenna No. 2. The antenna location discussion, however, will include both antennas Nos. 2 and 3. In general two antennas will be needed for each link. For proximity operations four additional low gain antennas (No. 5 in the table), with near hemispherical coverage, are included to provide extra coverage to EVA activity in and around the immediate Space Station structure. Four low gain S-band antennas (No. 7) are shown. Two of these antennas and the GPS (No. 8) antennas and their RF electronics will be located on the transverse boom for use during the early Space Station assembly phase. When the Space Station is complete these antennas will have significant obscuration from the solar panels and upper and lower keel structure. To alleviate this problem these antennas and the RF electronics associated with them must be moved to a less obscured location. However, if the moving of these antennas is not feasible, two additional antennas and their electronics will be required. The two systems left on the transverse boom could be used for limited redundancy capability. Assuming these antennas can be moved, the two S-band antennas will be located on the upper boom to provide a low data rate backup link to TDRS. The two GPS antennas will also be located on the upper boom. The two additional low gain S-band antennas (No. 7) will be located on the lower boom to provide additional coverage to the orbiter in the proximity region. The S-band horn antennas (No. 6), which provide coverage to the orbiter out to 20 nmi, will be located on the upper boom. These antennas provide the primary coverage for the S-band link to the orbiter.

Figure 4.3.6-7 is a plan view of the reference configuration showing antenna locations. The reference configuration is flown earth-oriented with the bottom end of the Space Station always pointed toward nadir as indicated in the figure. For this situation the look angle coverage requirements are as discussed in Section 4.3.6.1.2. Both the TDRS and GPS antennas must have unobscured views in the upper hemisphere looking toward space. To provide this coverage these antennas are located on the upper boom, as shown in figure 4.3.6-7. The GPS antennas are located on each end of the boom with the boresight looking along the z-axis toward space. Although one hemispherical antenna could provide the coverage, two are included since the placement of payload experiments on the boom may cause obscuration. Figure 4.3.6-8 shows an obscuration plot for the TDRS antenna located on the starboard side of the boom. The antenna boresight reference is aligned as described in Section 4.3.6.1.2. The shaded areas in the plot correspond to the angular volume where obscuration occurs. (Note: This obscuration plot and the ones to follow were made for the configuration which is identical to that shown in Figure 4.3.6-7 except that it included three lower keel sections, and did not include the large antenna experiment TDM 2060. The results, however,

ANTENNAS FOR REF. CONFIGURATION.

COMM. AND TRACK. TRACK		ANTENNA TYPE	FREQ. GHz	NO.	ELEMENTS SERVED
1.	TDRS S/KU BAND	9 ft dish	2/14	2	TDRS
2.	MED. GAIN (43db) CO-ORBIT M.A.	2.5 ft multi-feed offset Ref.	25	2	FF, PLAT-FORM, L&HDR
3.	HIGH GAIN (50db) CO-ORBIT S.A.	5 ft dish	25	2	FF, PLAT-FORM, HDR
4.	5-9db GAIN PROX. OPS M.A.	7 element switched array, hemisph. cov.	25	2	EMU, OMV, L&HDR
5.	0-3 db Gain PROX. OPS M.A.	Single element, near hemisph. cov.	25	4	EMU, OMV, L&HDR
6.	S-BAND LOW GAIN (10db)	Horn	2	2	ORBITER
7.	S-BAND LOW GAIN (0-3db)	Dual switched, hemisphere cov.	2	4	ORBITER, TDRS
8.	GPS LOW GAIN (0-3db)	Dual switched, hemisphere cov.	1.5	2	GPS
9.	RENDEZVOUS RADAR MED. GAIN (40db)	3 ft dish	14	2	ORBITER, OMV
10.	REND. RADAR XPNDR LOW GAIN (0-3db)	Near hemisphere coverage	14	2	ORBITER
11.	MRMS LOW GAIN (0-3db)	See Note 2	2	4	Mobile Rem. Manip. Sys.
		TOTAL		28	

TABLE 4.3.6 -1

NOTES: 1. Antenna #5 required to provide extra coverage to EVA activity in and around the immediate Space Station structure.

2. Two antennas of #11 are mounted on the MRMS. They are single element ant. with near hemispherical coverage. The other two are mounted on each side of the keel structure. These ant. will produce omni-directional dipole type patterns.

3. The Radar XPNDR. (#10) is included on the Space Station to enhance the capability of the Orbiter's rend. radar to see the Space Station, especially during assembly.

4. The S-band low gain ant. (#7) are used to provide coverage to the Orbiter and to TDRS during the assembly phase, and to provide a low data rate S-band link to TDRS as a backup to the 9-ft dish (#1). Two of these antennas will provide extra coverage to the Orbiter when it is stationkeeping near the Space Station.

5. In the initial configuration antenna #2 will be a single beam 28-in dish. Antenna #3 will be added in the growth configuration.

ORIGINAL PAGE IS
OF POOR QUALITY

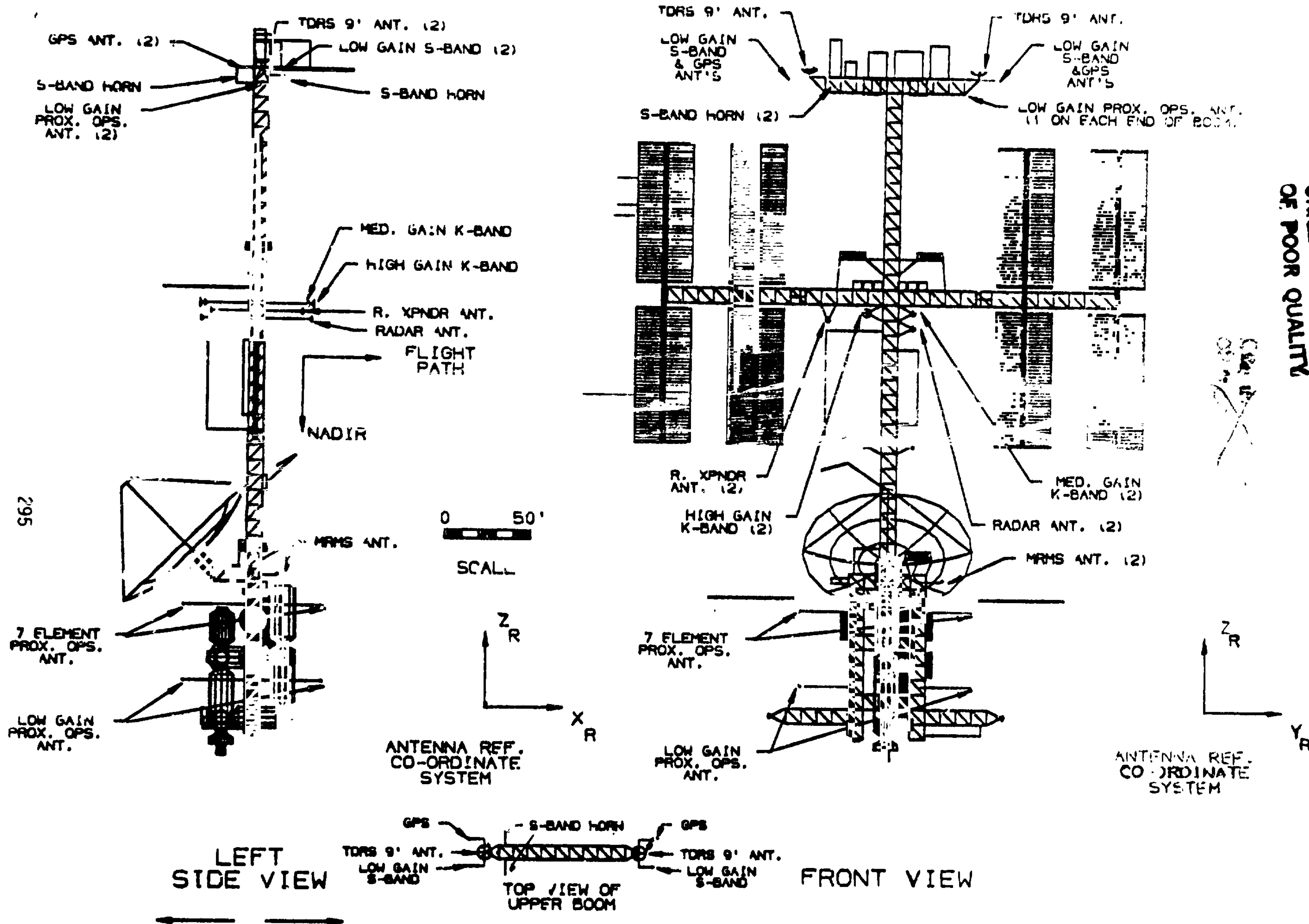


FIGURE 4.3.6-7 Antenna Locations

would not be much different for a single keel configuration.) Figure 4.3.6-8 shows that there is a clear field of view (FOV) over the required coverage angles indicated by the hatched area. Figure 4.3.6-4 indicates the required angular coverage. A similar clear upper hemisphere FOV will also be available for each of the GPS antennas except for slight obscurations that occur from payloads. However, the two GPS antenna coverages complement each other so that payload obscurations are avoided. The low gain TDRS S-band antennas located near the GPS antennas will have the same coverage capability as the GPS antennas.

The medium and high gain antennas (No. 2 and No. 3) used for the far range co-orbit communications need only a $\pm 15^{\circ}$ FOV around the forward and aft orbit track. This is easily obtained by placing these antennas on the upper portion of the lower keel, as seen in Figure 4.3.6-7. One each of these antennas is pointed along each track direction. Figures 4.3.6-9 & -10 show respectively the obscuration plots for the forward and aft looking 5-ft diameter K-band antennas (No. 3), located on the starboard side of the lower keel. For this plot the solar arrays were rotated from their position shown in Figure 4.3.6-7 by 90° in alpha and 52° in beta, placing the arrays in the position that can cause the greatest obscuration. The hatched in areas on each plot correspond to the $\pm 15^{\circ}$ FOV requirement, showing that a sufficiently clear FOV exists. A similar set of plots would result for the medium gain antenna, and for the rendezvous radar antenna (No. 9) located on the port side of the lower keel. For the radar, however, a full hemispherical ($\pm 90^{\circ}$) FOV is needed around the positive and negative x-axes. For the forward looking direction shown in Figure 4.3.6-9, this FOV (theta=0-180, & phi=0-90 & 270-360) is mostly clear except for some obscuration caused by the solar arrays. In the aft looking direction, where the required FOV is theta=0-180 & phi=90-270 degrees, a similar situation exists. This is shown in figure 4.3.6-10. These areas of obscuration will reduce when the solar arrays rotate away from this orientation. The radar transpower antenna, which has the same FOV requirement as the rendezvous radar antenna, will have obscuration similar to that shown in figures 4.3.6-9 & -10.

The S-band horn antennas (No. 6), which are located on the upper boom pointing along the forward and aft direction, will have a clear FOV over the forward and aft hemisphere. The obscuration plots for the forward looking antenna is shown in Figure 4.3.6-11. In this particular plot the z-axis of the antenna has been rotated so that it points along the x-axis. In this orientation the required FOV is the angular volume of theta=0-90° & phi=0-360° as indicated by the hatched area shown on the figure. Two of the low gain S-band antennas (No. 7) that are used to provide additional coverage to the Orbiter will be located out on either end of the lower boom structure, pointed along the positive and negative Y-axis respectively (not shown on the figure).

The 7-element prox. ops. antennas (No. 4) are located near the top of the keel extension. They are placed away from the structure in both the x and y direction and pointed along the positive and negative x-axis, in order to

ORIGINAL PAGE IS
OF POOR QUALITY

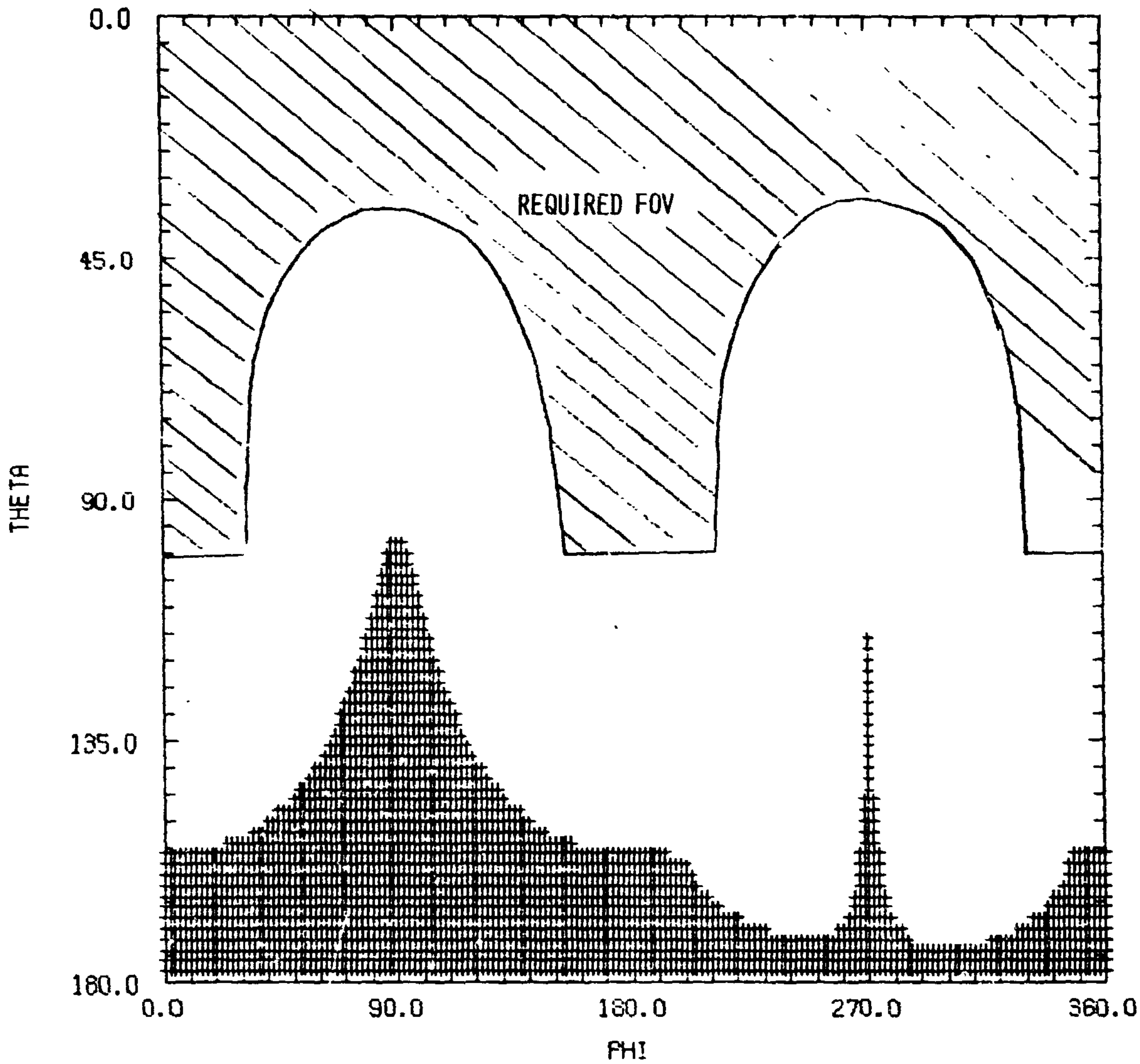


FIGURE 4.3.6 -8 OBSCURATION PLOT FOR TDRS ANTENNA LOCATED ON UPPER BOOM. ANTENNA Z-AXIS POINTED 180° FROM NADIR, X-AXIS POINTED ALONG FLIGHT PATH.

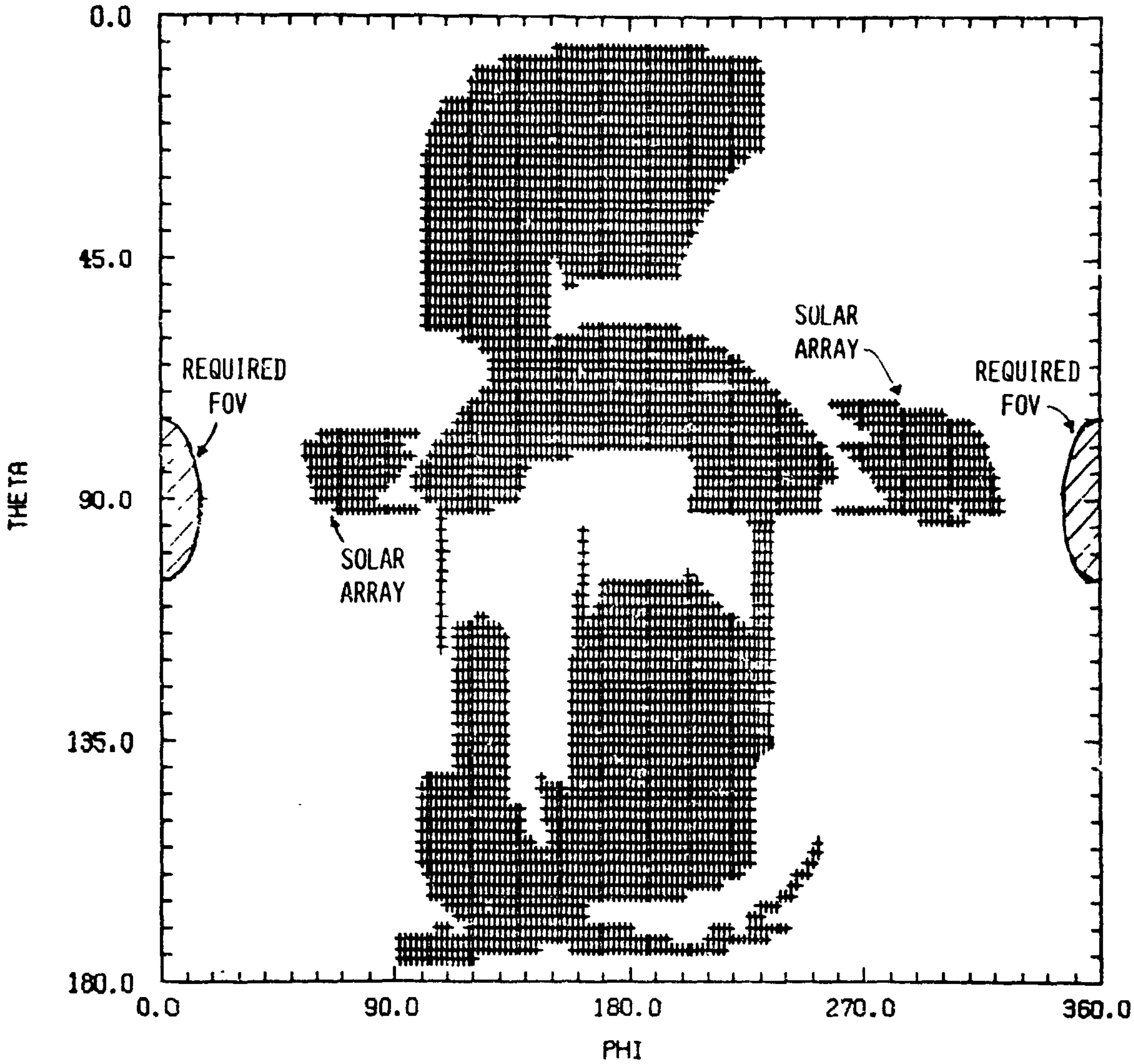


FIGURE 4.3.6 -9 OBSCURATION UT FOR MEDIUM AND HIGH GAIN FAR RANGE ANTENNAS LOCATED ON FORWARD SIDE OF UPPER SEC ION O LOWER KEEL. SOLAR ARRAYS ROTATED 90^{0} IN α AND 52^{0} IN β. ANTENNA Z-AXIS POINTED 0° FROM NADIR, X-AXIS POINTED ALONG FLIGHT PATH.

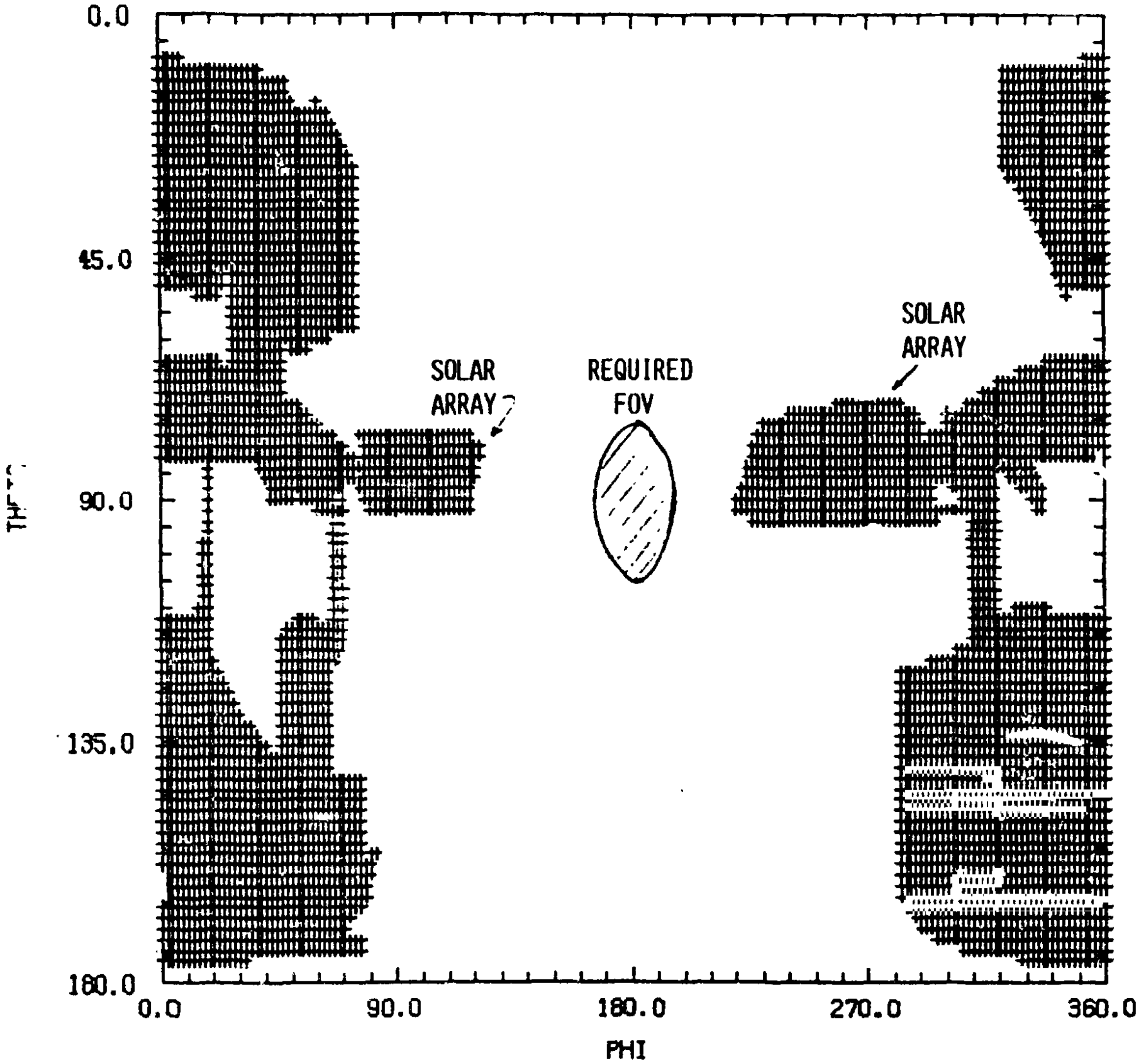


FIGURE 4.3.6 -10 OBSCURATION PLOT FOR MEDIUM AND HIGH GAIN FAR RANGE ANTENNAS LOCATED ON AFT SIDE OF UPPER SECTION OF LOWER KEEL. SOLAR ARRAYS ROTATED 90° IN α AND 52° IN β. ANTENNA Z-AXIS POINTED 180° FROM NADIR, X-AXIS POINTED ALONG VELOCITY VECTOR.

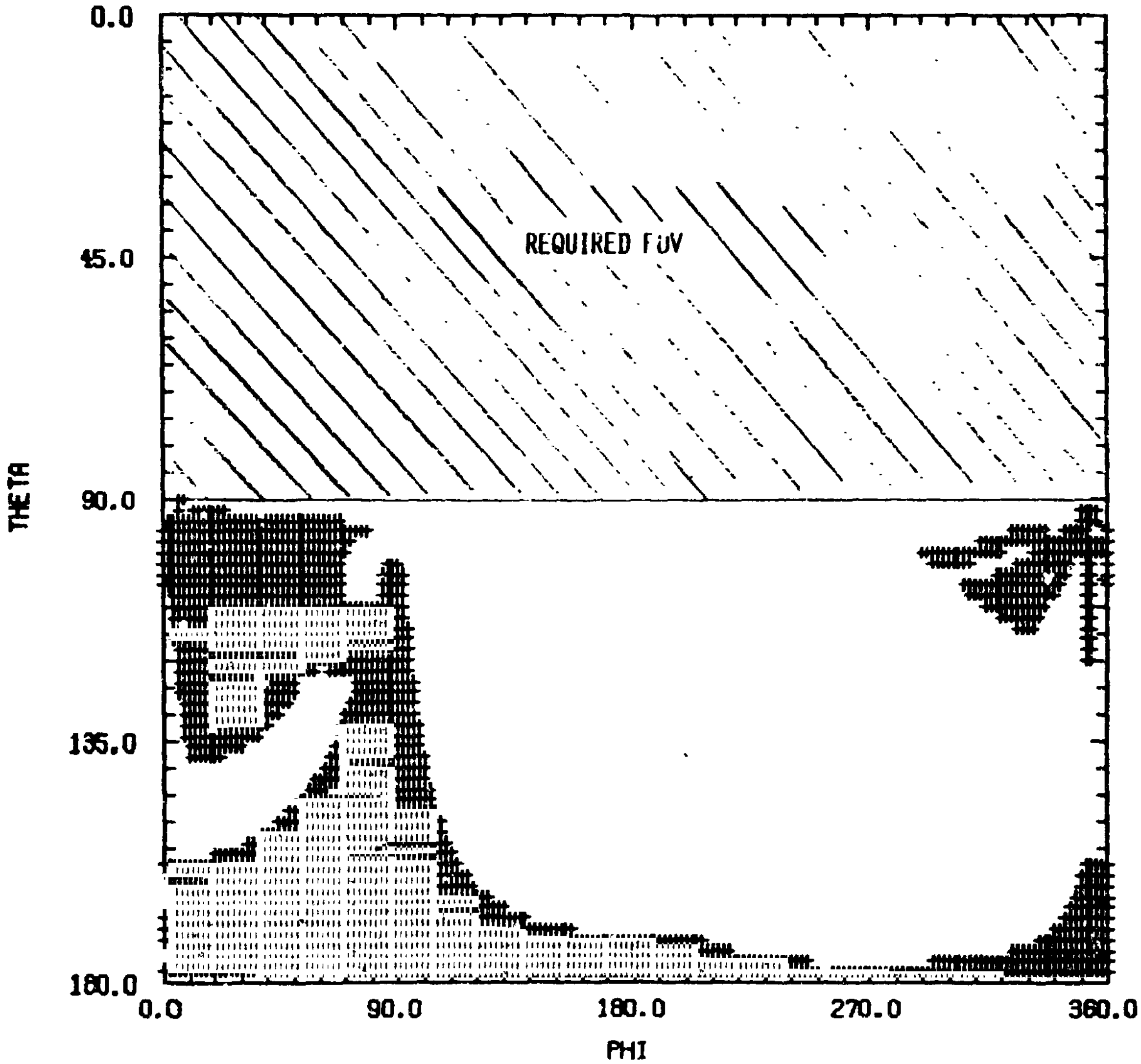


FIGURE 4.3.6.-11 OBSCURATION PLOT FOR ORBITER HORN ANTENNA LOCATED ON FORWARD SIDE OF THE UPPER BOOM. ANTENNA Z-AXIS POINTED ALONG FLIGHT PATH. X-AXIS POINTED TOWARD NADIR.

provide unobscured hemispherical coverage in the forward and aft direction. Figures 4.3.6-12 & -13 show obscuration plots for these two antennas, for the z-axis of each antenna rotated so they point along the positive and negative x-axis, respectively, as in Figure 4.3.6-11.

Two of the low gain prox. ops antennas (No. 5) will be located on each end of the upper boom pointed along the positive and negative y-axis, respectively. Figure 4.3.6-14 shows an obscuration plot for the antenna on the starboard end of the upper boom, with the z-axis of the antenna pointed along the negative y-axis. The solar panels for this figure were rotated 90° in alpha and 52° in beta. The required hemispherical FOV is the angular volume of theta=0-90° & phi=0-360°. This area is essentially clear except for the small area of blockage caused by the solar panels around phi=270°. The prox. op. antenna on the other end of the upper boom will have a similar obscuration plot, except the shawdows from the solar arrays are located around phi=90 degrees. The hemispherical FOV will lie in the positive y-axis direction. The other two No. 5 antennas are located on the keel extension section with their axes pointed along the positive and negative y-axis similar to the other antennas on the upper boom.

Two of the MRMS low gain antennas (No. 11) will be located on the top of keel extension, one on the forward side and one on the aft side. The omnidirectional patterns of these antennas are aligned to provide coverage to the MRMS up and down each side of the keel. The other two antennas as stated earlier will be mounted on either end of the MRMS.

Locations of the low gain broad beam antennas (Nos. 5, 7, 8, 10 & 11), are only suggested to provide good FOV coverage. The obscuration plots shown for the various antenna locations give line of sight blockage information only, and do not necessary represent the RF blockage which may occur. The actual antenna patterns from these antennas or any of the broad beam low gain antennas will be affected by reflections and refractions from nearby str tures. A more detailed volumetric pattern computer analysis, of these antennas mounted on the structure, will be needed before true RF blockage can be determined and suitable locations chosen. This also applies to the high gain pencil beam antennas, in which case the problem is from multipath reflections from structures entering the antenna side lobes.

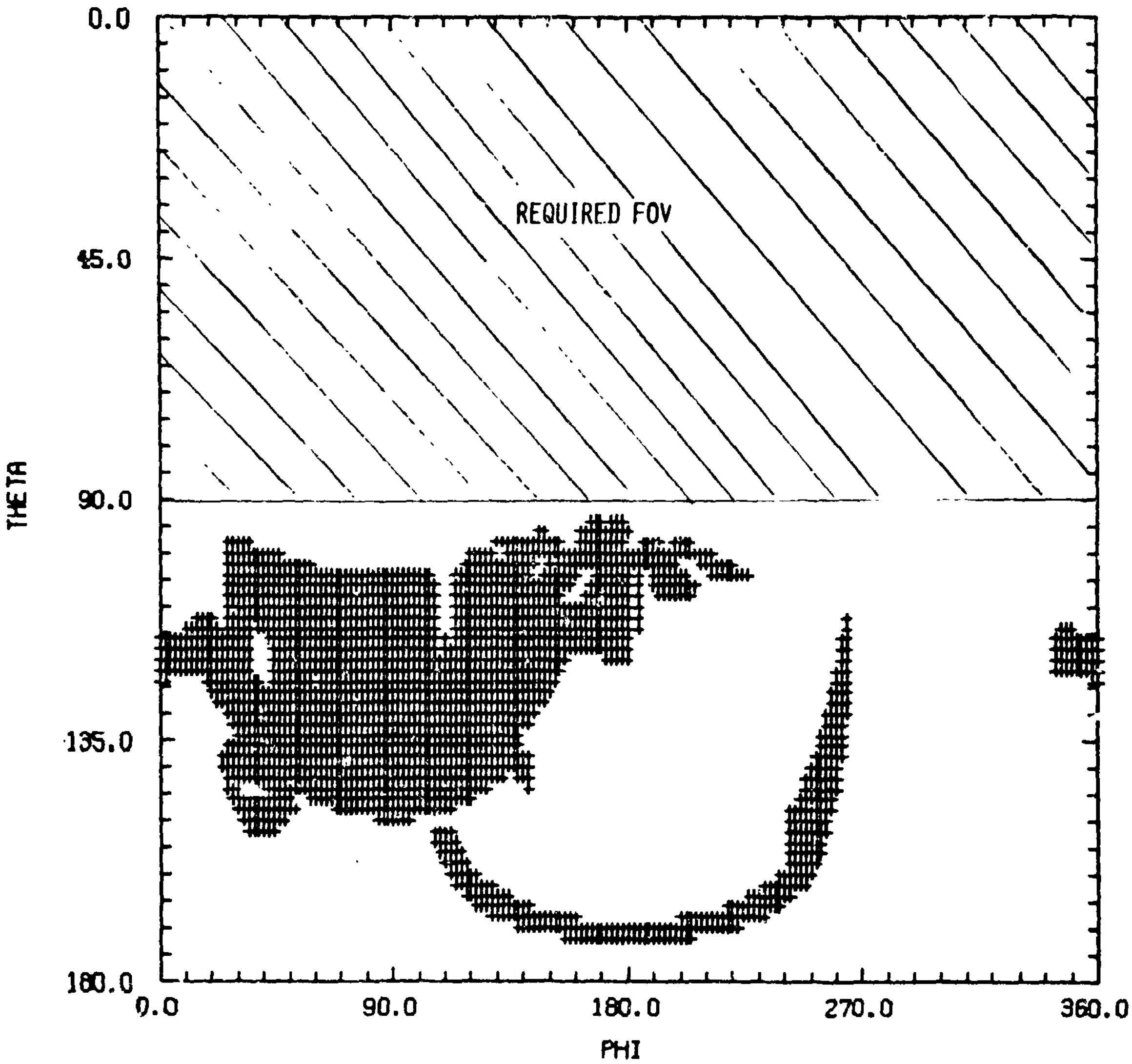


FIGURE 4.3.6 -12 OBSCURATION PLOT FOR 7 ELEMENT PROX. OPS. ANTENNA LOCATED ON THE STARBOARD SIDE OF THE KEEL EXTENSION (NEAR TOP). ANTENNA Z-AXIS POINTED ALONG FLIGHT PATH, X-AXIS POINTED TOWARD NADIR.

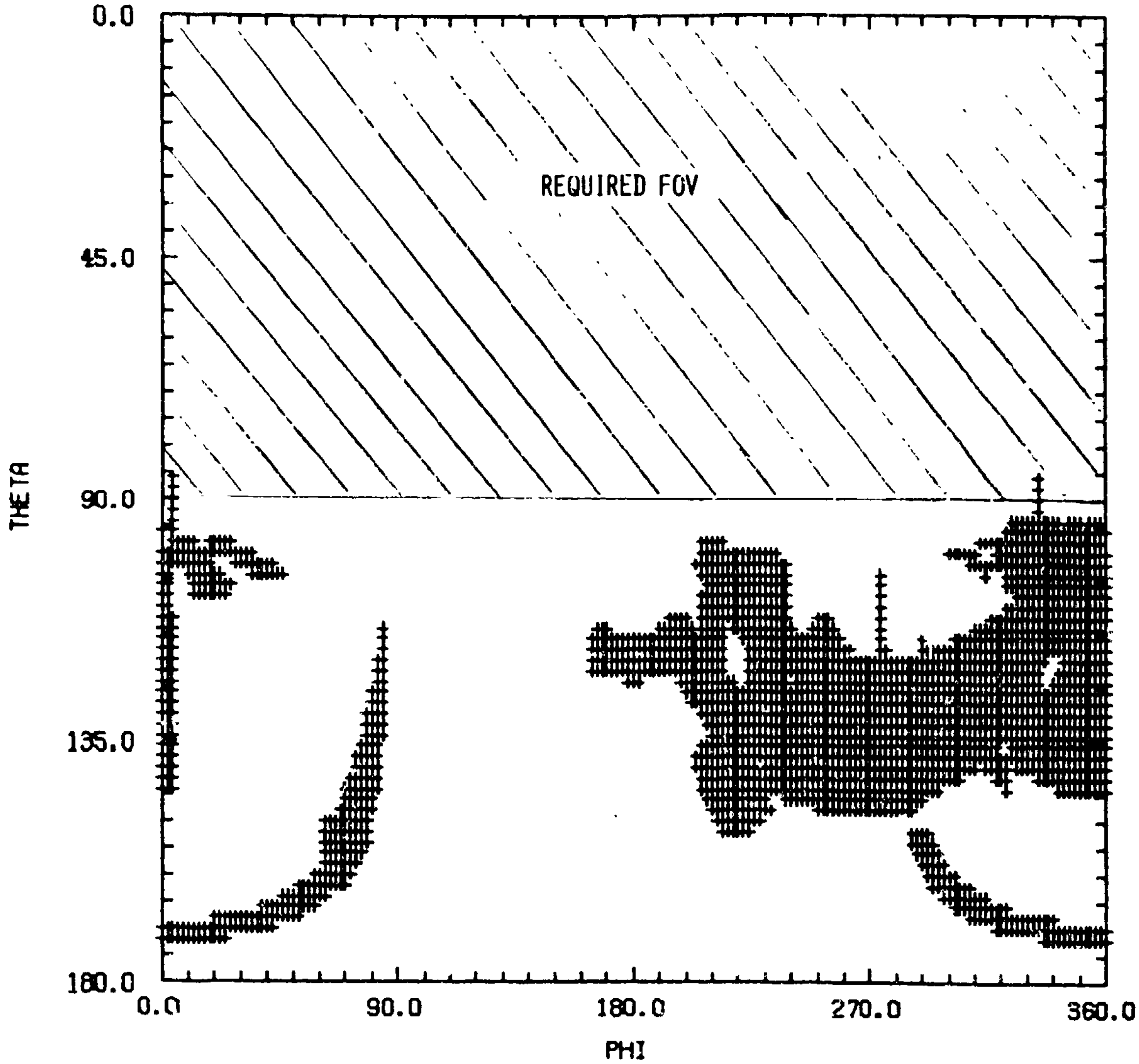


FIGURE 4.3.6 -13 OBSCURATION PLOT FOR 7 ELEMENT PROX. OPS. ANTENNA LOCATED ON THE PORT SIDE OF THE KEEL EXTENSION (NEAR TOP). ANTENNA Z-AXIS POINTED 180° FROM THE FLIGHT PATH, X-A IS POINTED 180° FROM NADIR.

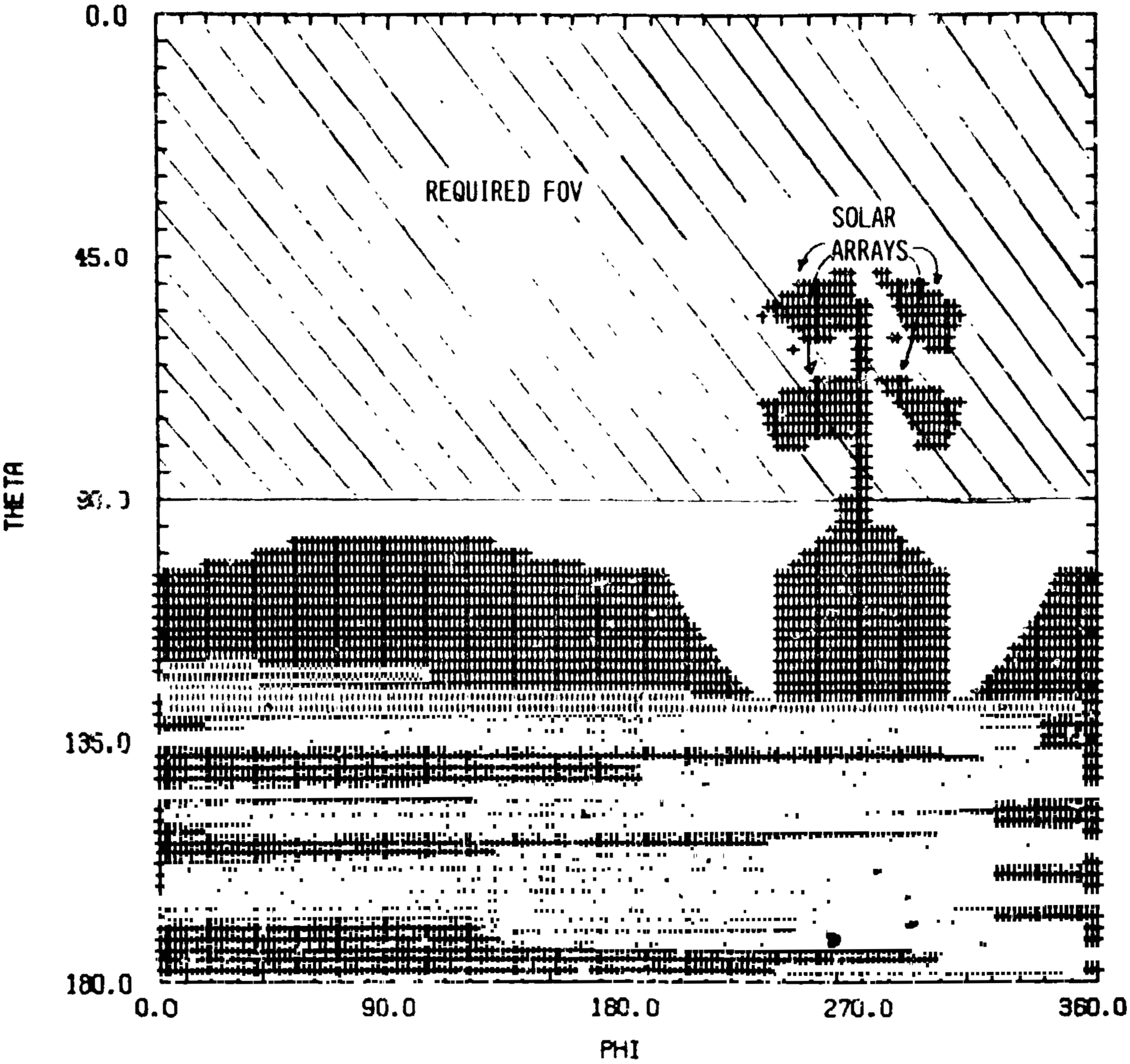


FIGURE 4.3.6 -14 OBSCURATION PLOT OF LOW GAIN PROX. OPS. ANTENNA LOCATED ON THE STARBOARD END OF THE UPPER BOOM. SOLAR ARRAYS ROTATED 90° IN α AND 52° IN β. ANTENNA Z-AXIS POINTED IN STARBOARD DIRECTION, X-AXIS POINTED ALONG FLIGHT PATH.

4.3.7 Launch and Assembly

4.3.7.1 Introduction

At present, no spacecraft has ever been assembled on-orbit. The construction of a Space Station therefore presents a broad spectrum of design and development challenges that must be addressed early in a configuration analysis.

A Space Station reference configuration launch and assembly scenario was developed for use in determining Space Station design implications and requirements on the STS. It was recognized that there were many variations and options applicable to the launch and assembly of such a system as the Space Station; however, the following scenario is considered to be an achievable one based on the available inputs. The effort described herein includes the objectives that were established and an explanation of the assumptions that were made. Also included are descriptions of each payload package and the assembly operations leading to IOC. Finally, estimates of the required crew timelines for the assembly operations are included.

The scope of this scenario is limited to the launch and assembly/installation of the major Station elements leading to IOC and does not address station payload related items. The intent is to convey a starting point for more extensive studies leading to a well defined launch and assembly scenario.

Only the assembly of the IOC Station is described. It is felt that the launch and assembly operations associated with the growth of the station would be quite similar to those presented here.

4.3.7.2 Launch/Assembly Description

Early in the study three general objectives were selected and were incorporated into the reference scenario. The three objectives are:

1. Minimized STS-supported EVA. EVA that is conducted from and supported by the Orbiter is a limited resource. On a typical seven day mission, there is currently a limit of two, two-man EVA's of approximately six hours duration each. In addition, should an emergency arise, the Orbiter may be required to return to Earth before all mission objectives have been met. For this reason an objective was to minimize the amount of EVA required in the early assembly phases of the Space Station. Some EVA will of necessity be required during the early phases of the buildup, however, the objective was to keep the durations and complexities to a minimum. After the station can be manned and the necessary consumables are on board, EVA could be conducted without the presence of an orbiter. Therefore, many operations required to achieve the final station configuration but not required to reach habitability were reserved for station-supported EVA missions.

2. Self-powered and controllable spacecraft from Flight 1. Another important objective for launch and assembly planning was to leave on-orbit from the first flight a spacecraft element capable of generating some amount of power for attitude control, thermal control, and communications functions. This goal dictates the order in which major station elements are launched through the first two flights.

3. Early habitability. A final important objective was to man the station as early in the buildup phase as is practical without making major subsystem changes. Having the station manned allows the assembly operations such as systems checkout, interface verification, and utilities installation to continue between STS flights. This objective dictates the order in which station elements are launched beginning with the third flight.

In establishing the launch and assembly scenario, certain assumptions concerning the STS were made. These include the following:

1. Only one Orbiter is required for each launch/assembly phase.

2. Only one Logistics Module is required to reach IOC (no changeout during buildup).

3. Flights 1 through 5 are capable of supporting EVA.

4. Shuttle Remote Manipulator System (SRMS) is available on Flights 1 through 7.

5. The Orbiter would not be used in a hovering mode from point to point mode.

It was also assumed that the OMV would not be used to assist in the assembly operations.

The launch/assembly scenario is composed of seven STS flights and associated operations. Figure 4.3.7.2-1 illustrates the major components of the Space Station and indicates the flights on which they are manifested.

The basic structure of the Space Station is composed of linear, sequentially deployed, single fold, box truss type beam segments as described in the White Paper entitled, "Space Station Truss Structures and Construction Considerations." The reference structure has strut members that are two inches in outer diameter. A single unit of this beam is a 9- by 9-ft cube formed by 18 struts and eight connecting nodal joints. Utilities are pre-integrated into the structure, thereby reducing the task of routing cables, hoses, etc., to one of making relatively simple connections at the major structural interfaces.

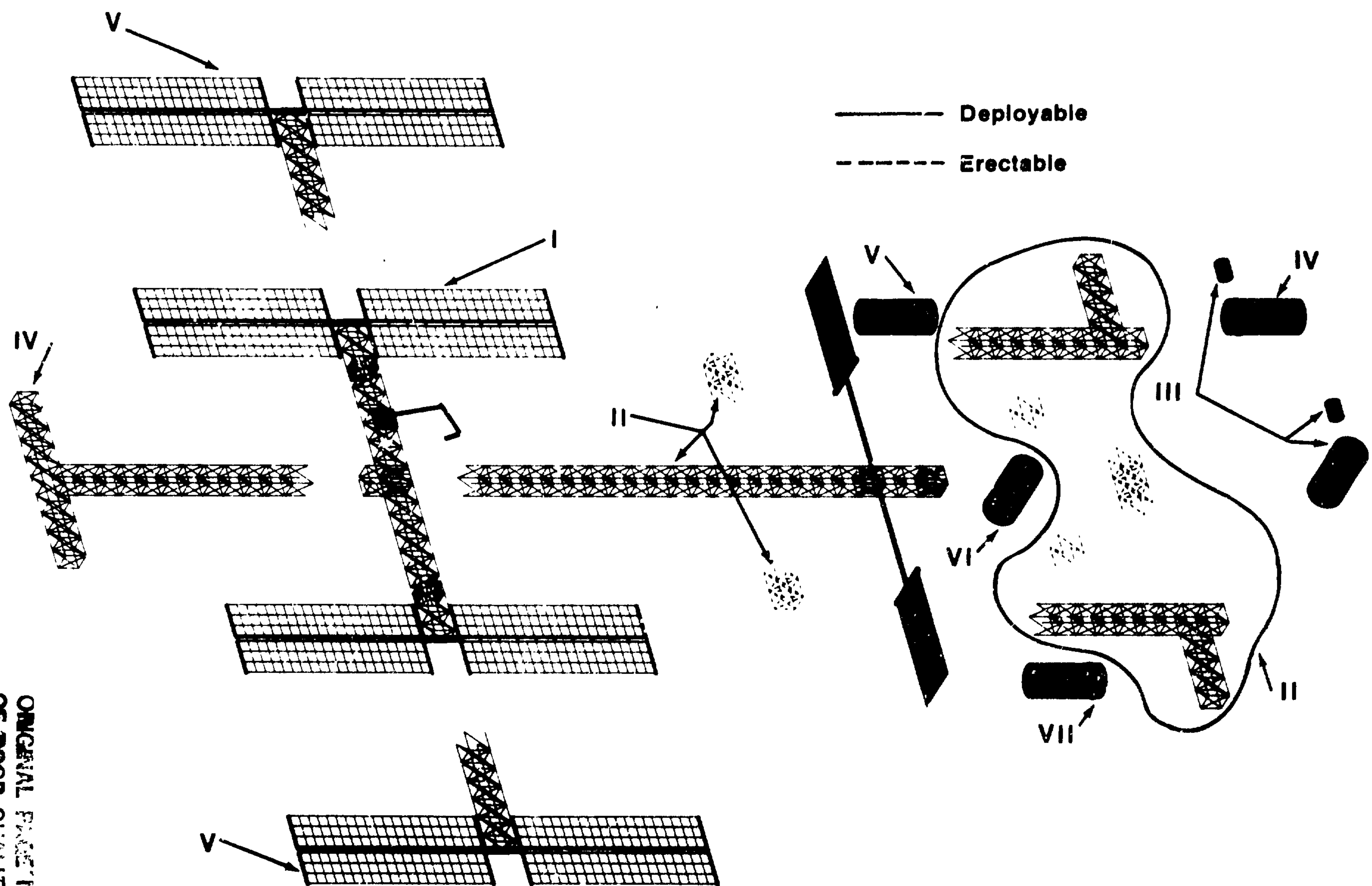


Figure 4.3.7.2-1

Certain segments of the station structure, it is felt, could be efficiently assembled in a strut-by-strut manner by the crew. This is especially true for short connecting segments which could become quite difficult to integrate, package, and deploy with longer segments.

The listing that follows is a summary of the payload components making up each of the seven flights.

FLIGHT	MAJOR SPACE STATION ELEMENTS
I	- inboard solar array wing pairs
	- rotating power joints
	- power conditioning radiator arrays
	- inboard transverse boom structure
	- power conditioning equipment
	- control equipment
	- communication equipment
	- berthing structure
	- MRMS
II	- lower keel structure
	- port keel extension structure
	- starboard keel extension structure
	- lower boom structure
	- main radiator panels
	- closeout structure
	- main radiator booms
	- RCS
	- berthing structure
III	- HM1 (habitation module 1)
	- AL1 (airlock 1)
	- AL2 (airlock 2)
IV	- HM2 (habitation module 2)
	- upper keel structure
	- upper boom structure
	- antennas
V	- LOG1 (logistic module 1)
	- port solar array wing pair
	- starboard solar array wing pair
	- port outboard transverse boom structure
	- starboard outboard transverse boom structure
VI	- LAB1 (laboratory module 1)
	- equipment spares
	- external experiments
VII	- LAB2 (laboratory 2)
	- equipment spares
	- external experiments

Such items as the OMV, OMV hangar, and satellite servicing equipment have not been included in this scenario. Further study is needed in this area to determine whether these items could be delivered on Flights VI or VII, on subsequent flights, or possibly by resupply flights.

4.3.7.3 Shuttle Payload Bay Packaging

A major design constraint on the principal Space Station elements is compatibility with the STS for launch. This constraint includes such critical payload parameters as weight, volume, and center of gravity. For each of the seven STS flights in the reference scenario, an analysis was carried out to define a payload package that was within the limits set by geometry and launch capability of the Orbiter. Figures 4.3.7.3-1 through 4.3.7.3-7 illustrate the launch package for each flight.

4.3.7.4 Component Installation Description

Following is a description of the assembly procedures using illustrations as appropriate. The approach taken was to develop only the details of the operations necessary to understand the total assembly task.

FLIGHT I

TASK/OPERATION	FIGURE	METHOD
1. Install radiator panels	I-1	RMS/EVA
2. Remove payload package from launch restraints; rotate 90° and insert into deployment restraints	I-2	RMS
3. Deploy port half of deployable structure; deploy starboard half of deployable structure	I-3	Mechanisms
4. Deploy solar array blanket boxes; deploy solar array blankets	I-4 I-5	Mechanisms
5. Erect berthing structure support bay; attach berthing port		EVA
6. Install MRMS (Mobile Remote Manipulator System)		SRMS/EVA
7. Perform systems checkout; depart	I-6	Orbiter Systems

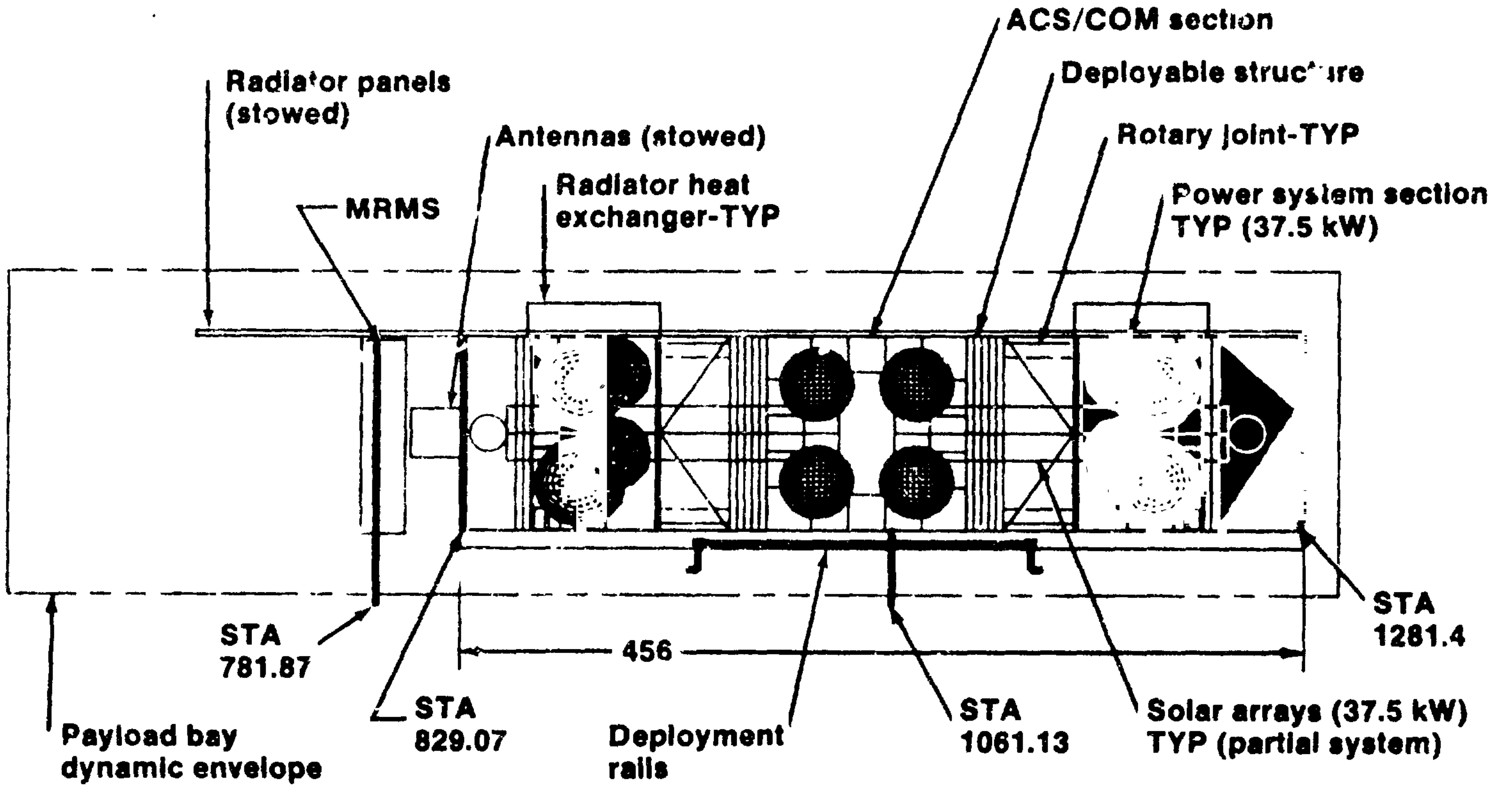


Figure 4.3.7.3-1 Launch package I.

ORIGINAL PAGE IS OF POOR QUALITY

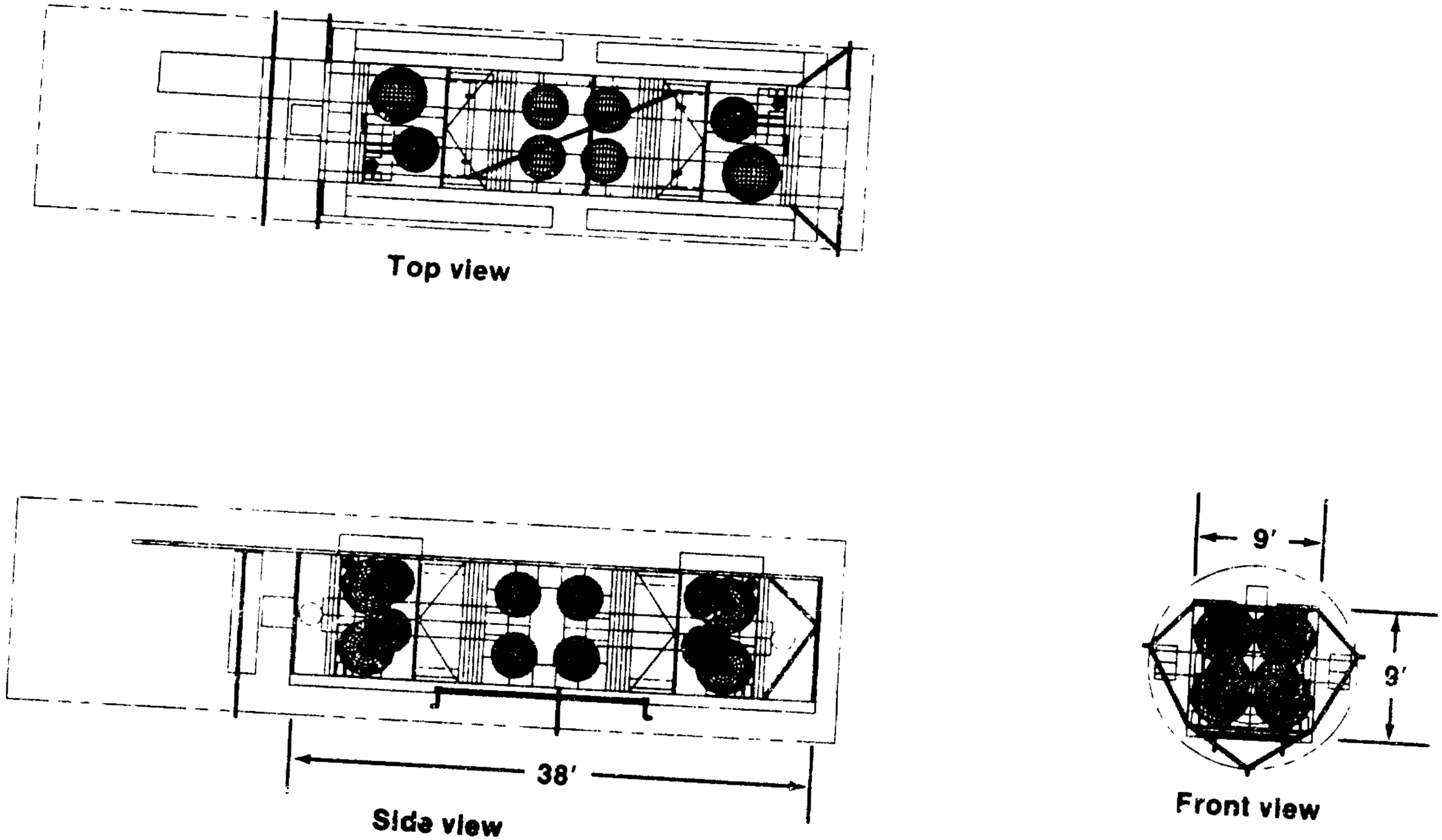


Figure 4.3.7.3-1 (cont.). Three views of launch package I.

Lower keel and boom launch configuration

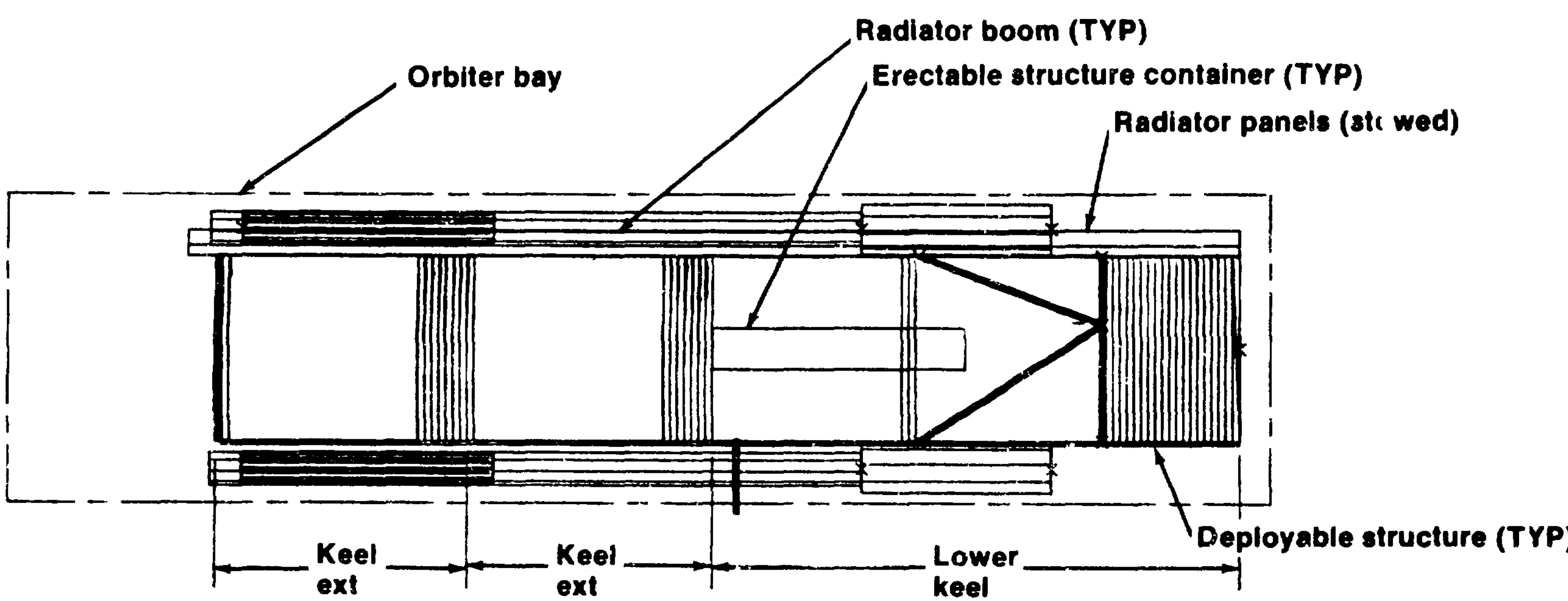


Figure 4.3.7.3-2 Launch package II.

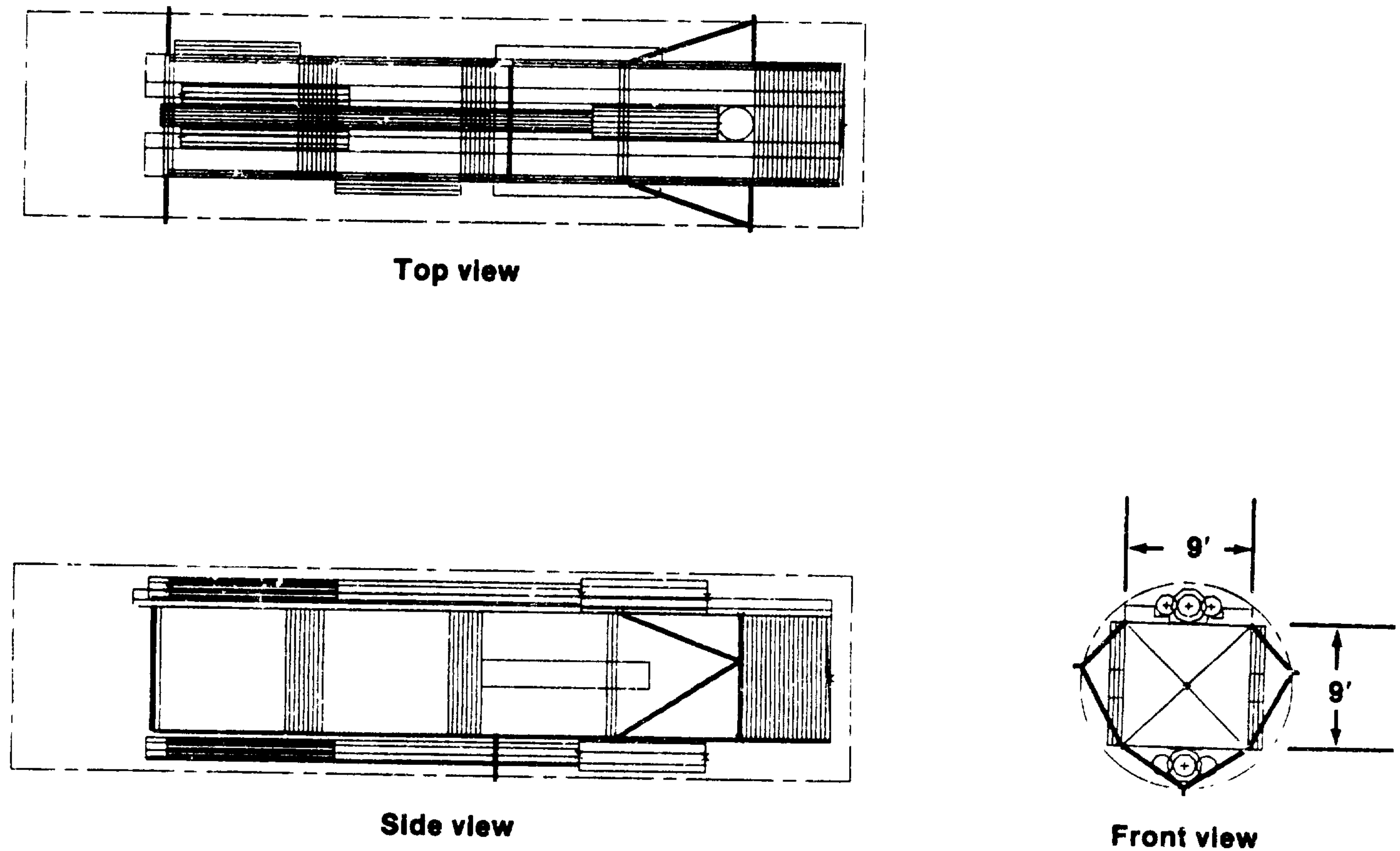


Figure 4.3.7.3-2 (cont.). Th:ee views of launch package II.

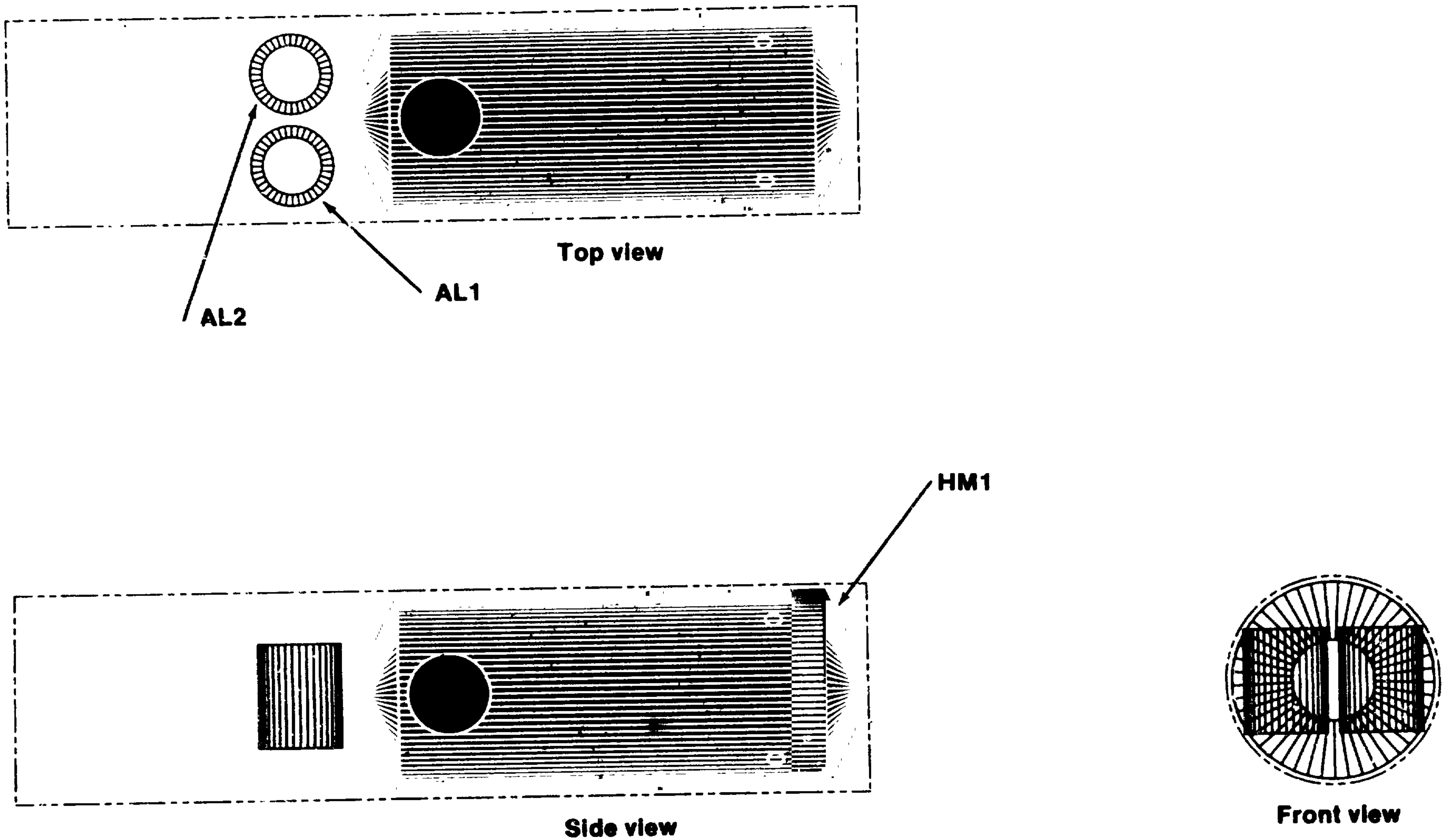


Figure 4.3.7.3-3 Launch Package III.

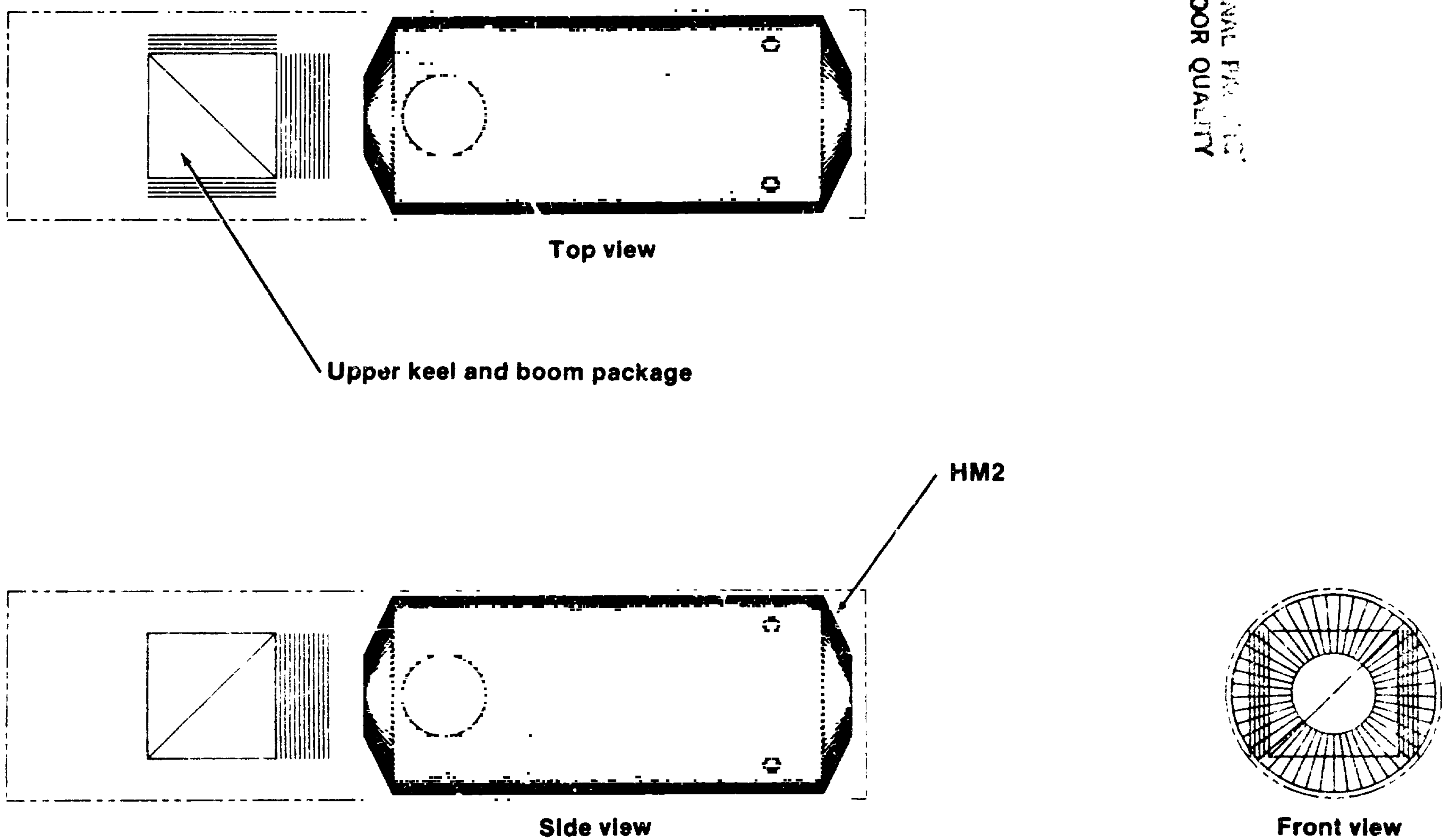


Figure 4.3.7.3-4 Launch Package IV.

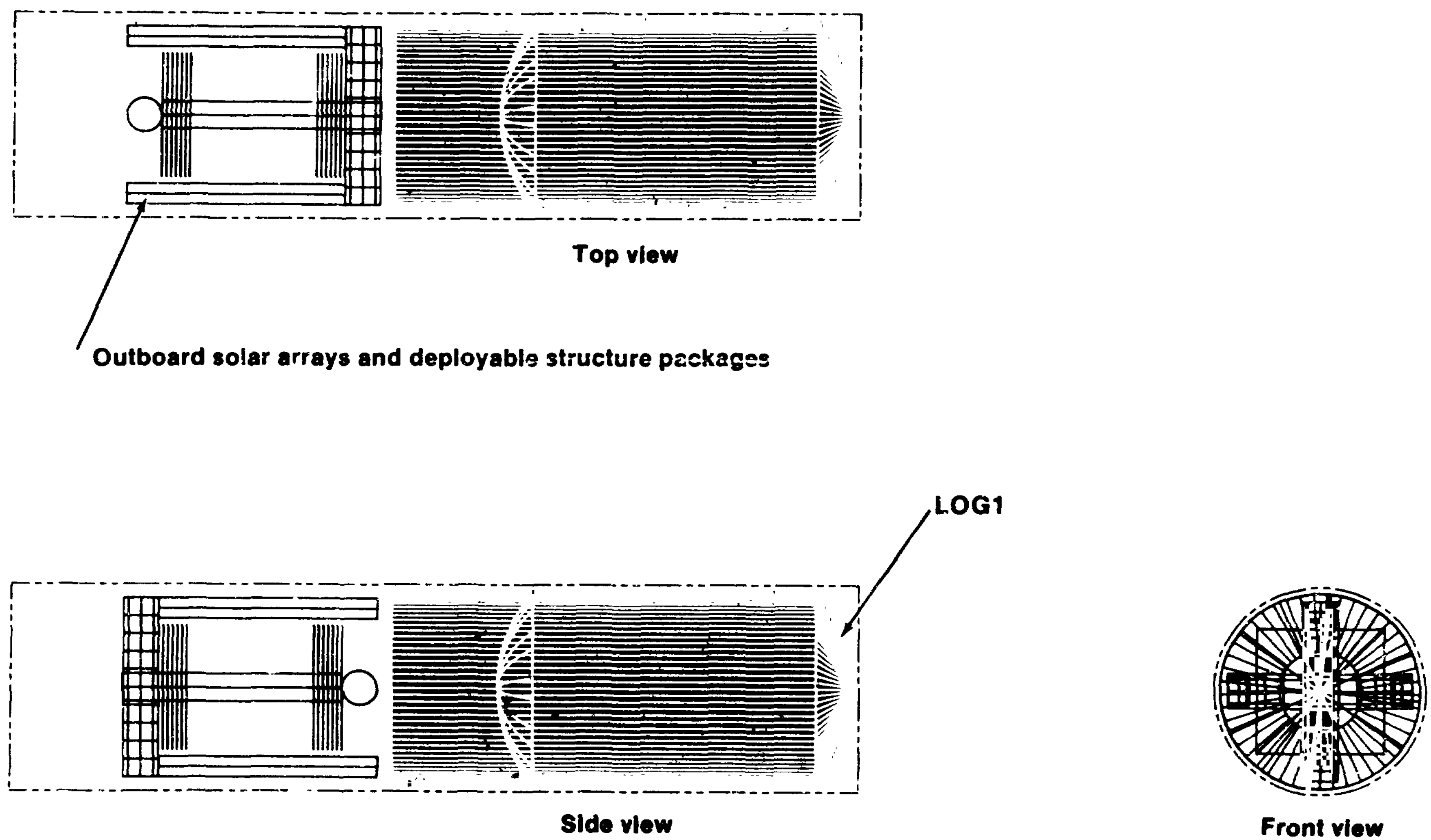


Figure 4.3.1.3-5 Launch Package V.

ORIGINAL PAGE IS
OF POOR QUALITY

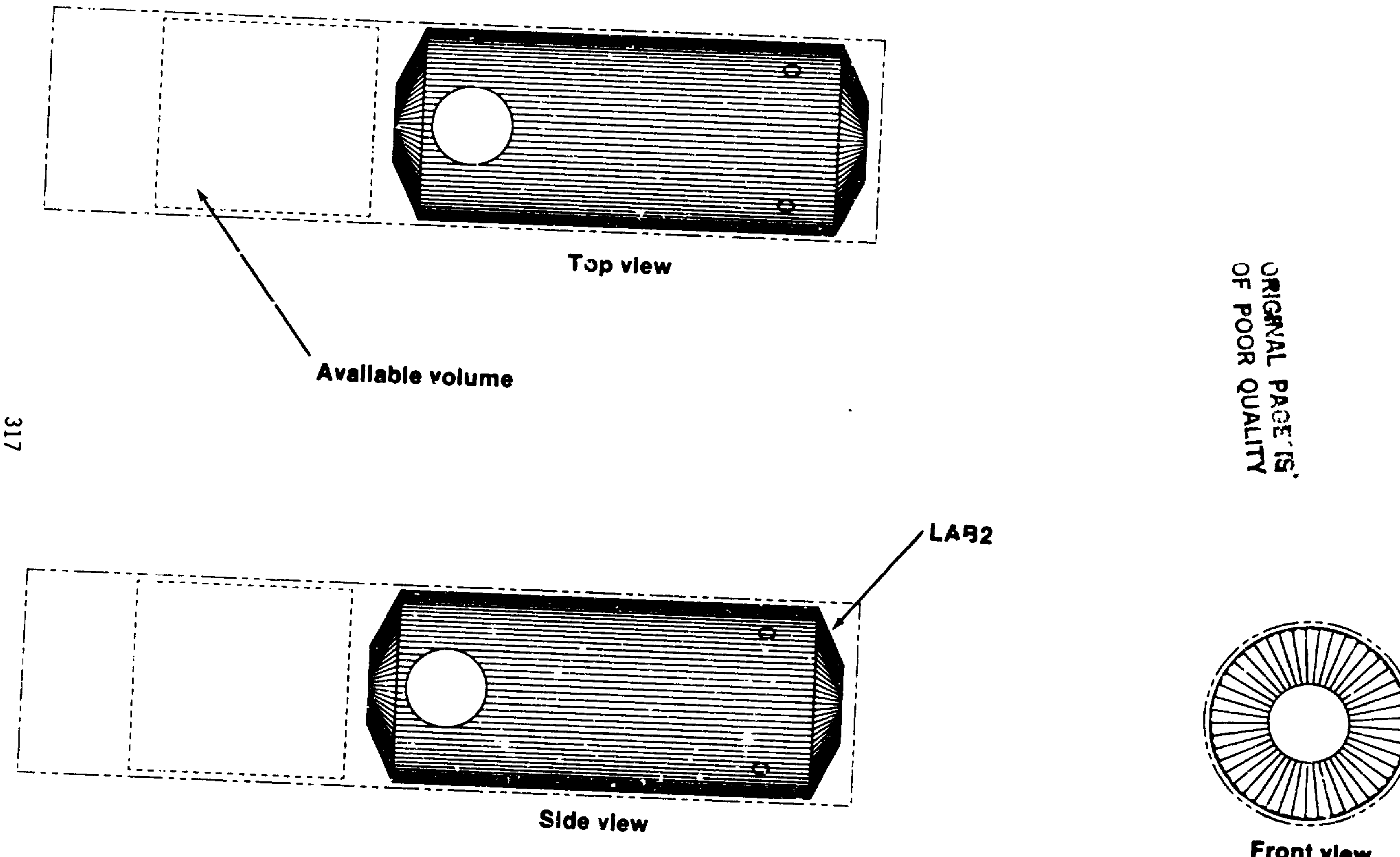


Figure 4.3.7.3-6 Launch Package VI.

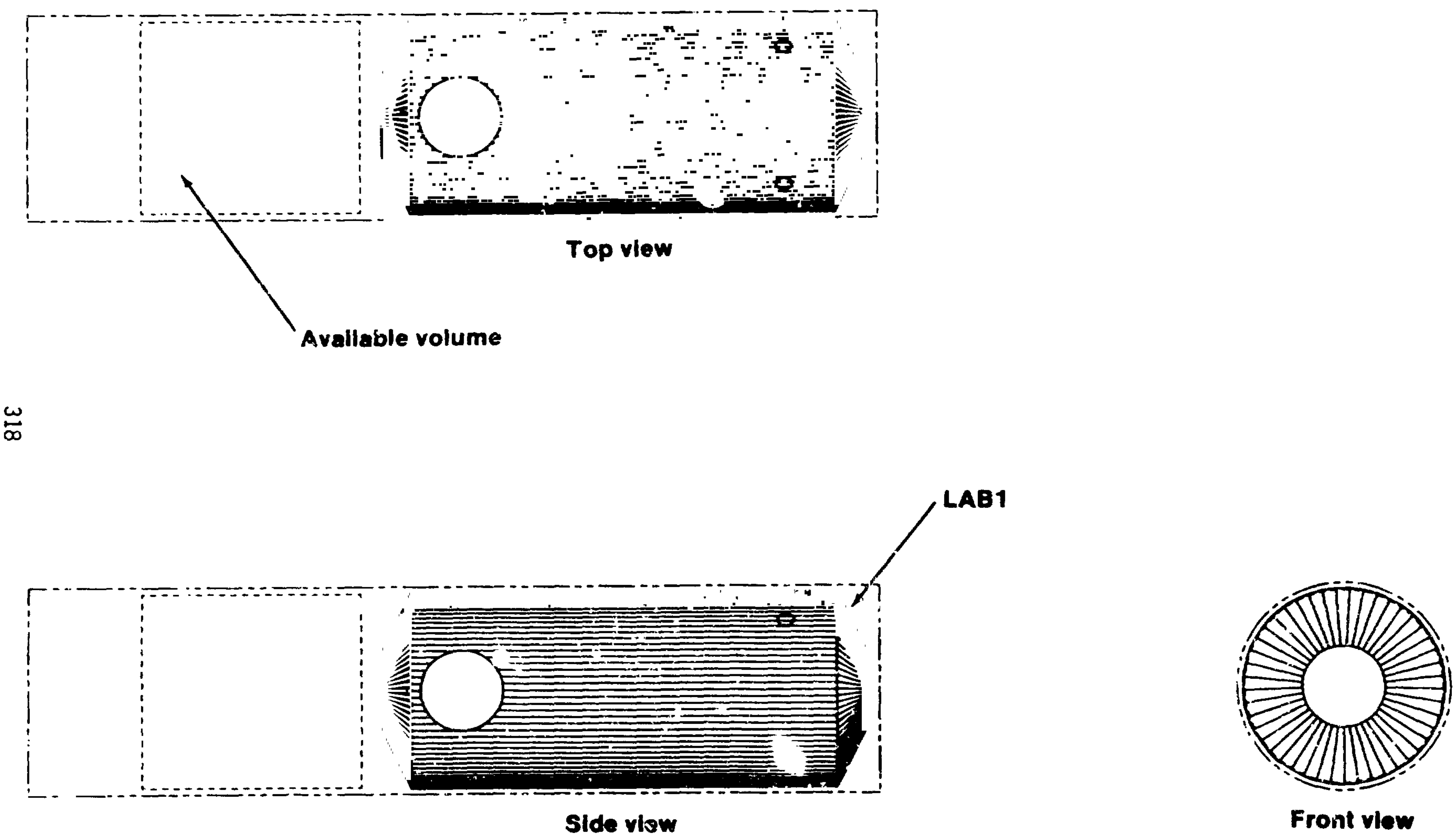


Figure 4.3.7.3-7 Launch Package VII.

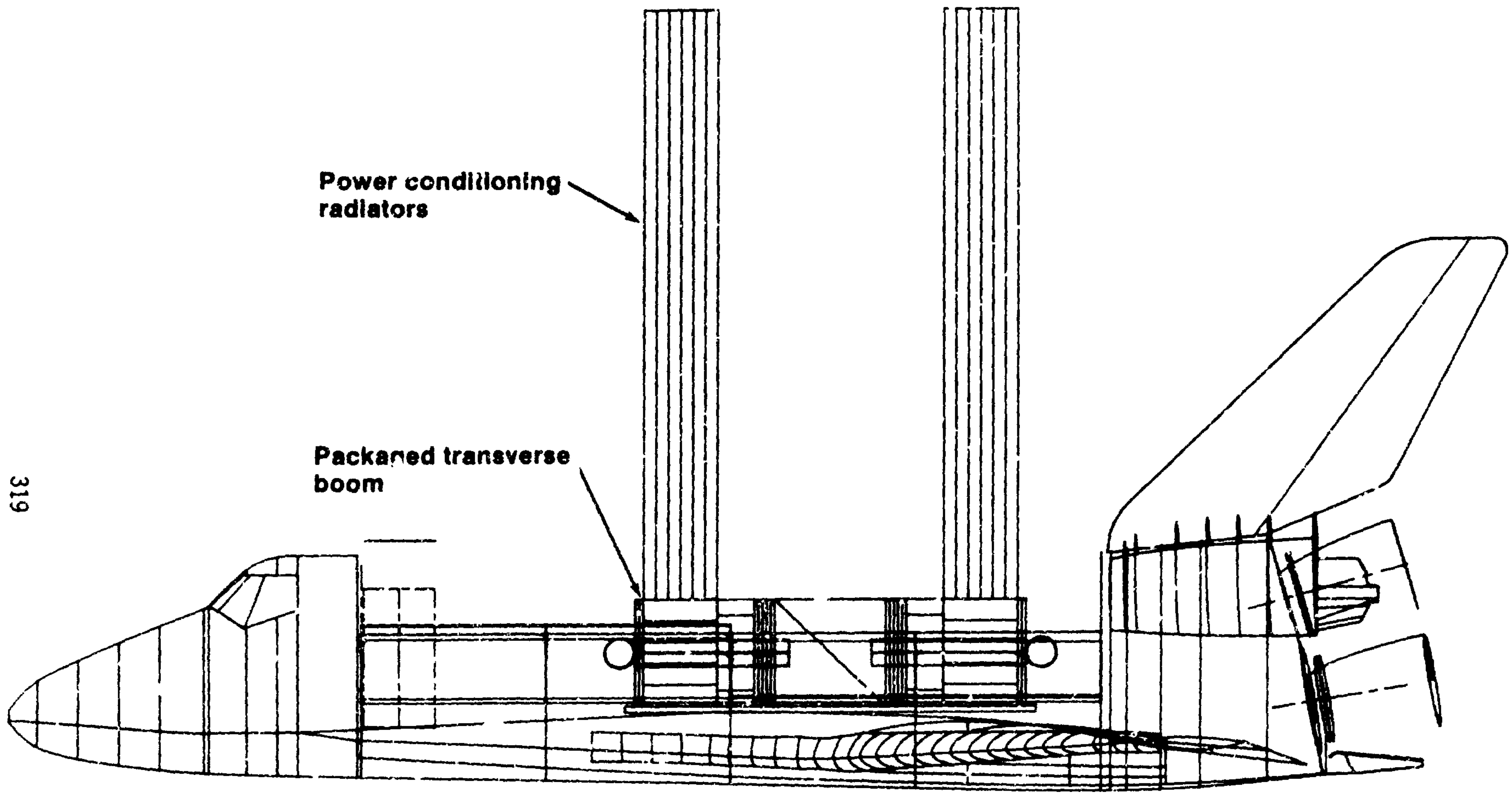


Figure I-1 Power conditioning radiators shown installed on first launch package while still in cargo bay.

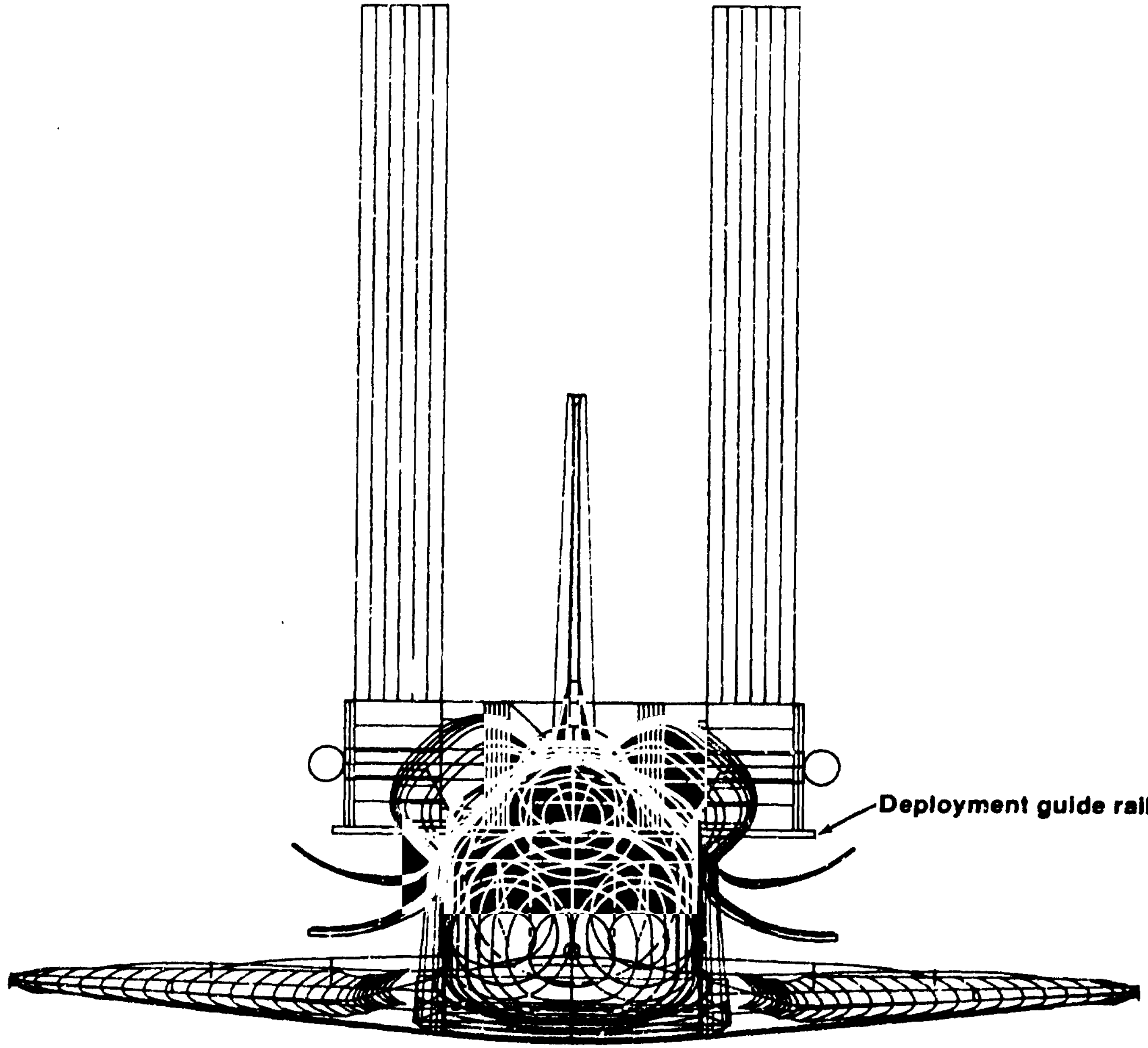


Figure I-2 Launch package removed from cargo bay using SRMS, rotated 90° and attached to cargo bay sides.

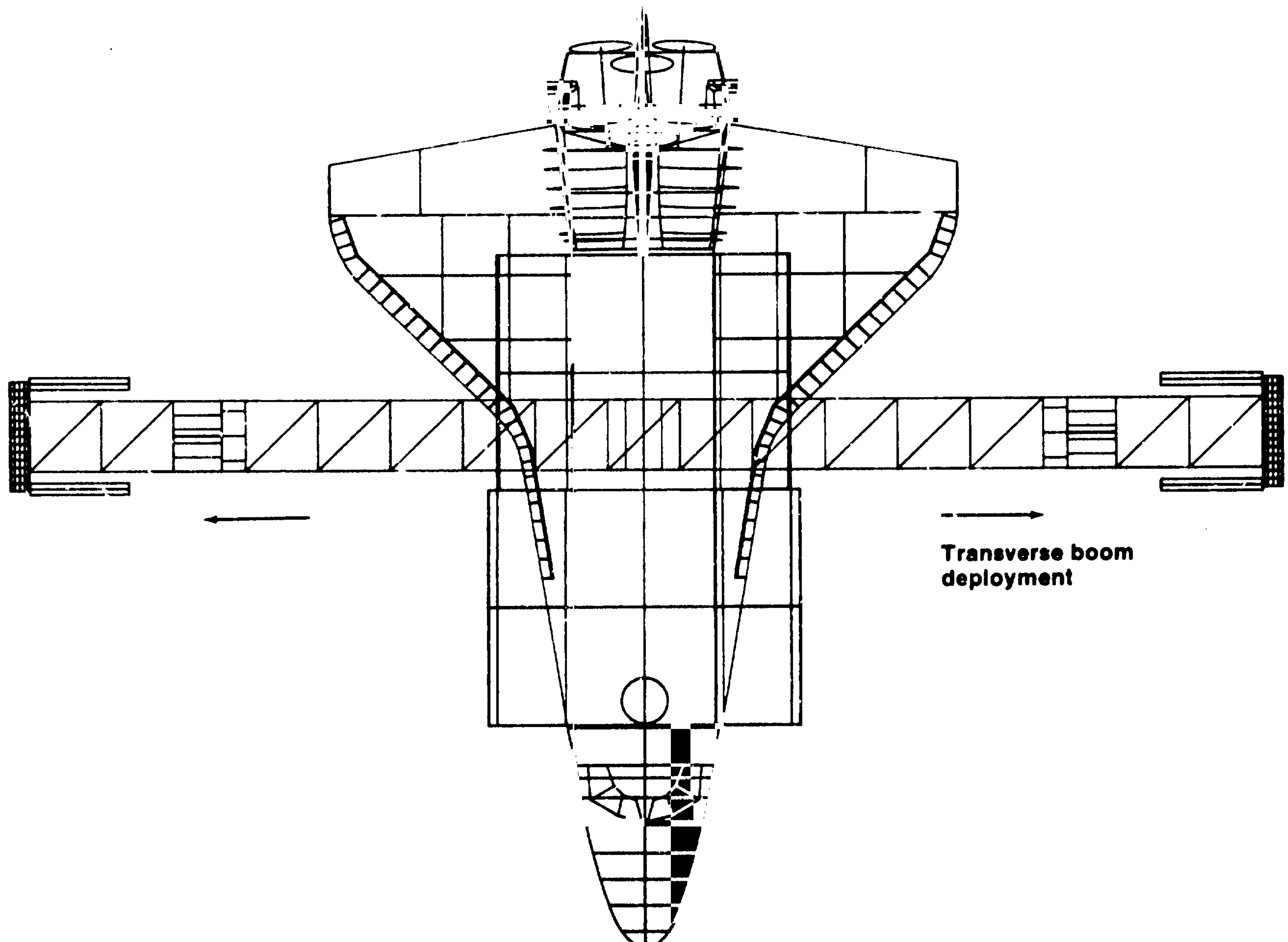


Figure I-3 Transverse boom deployed from guide rails.

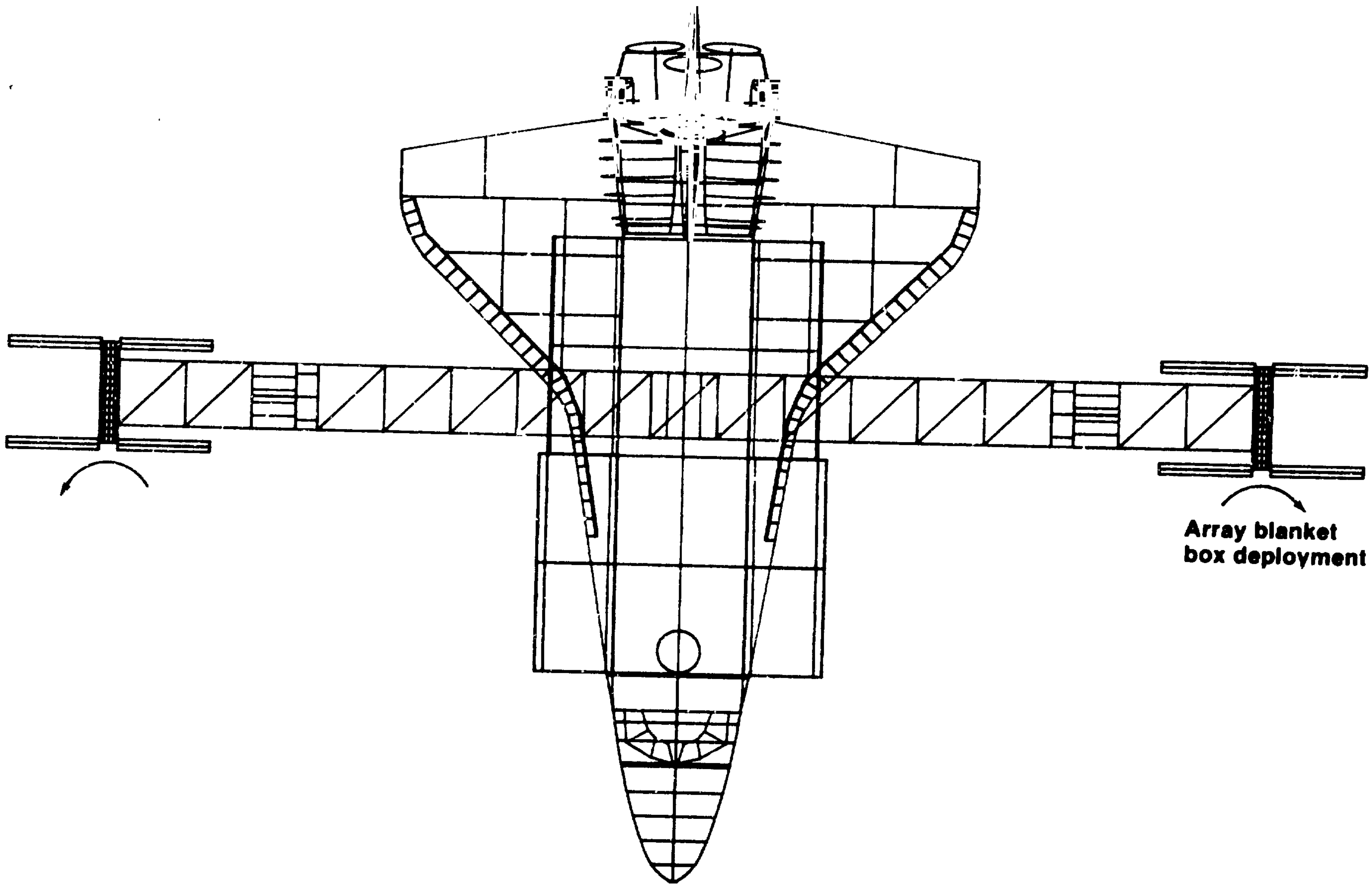


Figure I-4 Top half of solar array blanket box rotated into position and ready for blanket deployment.

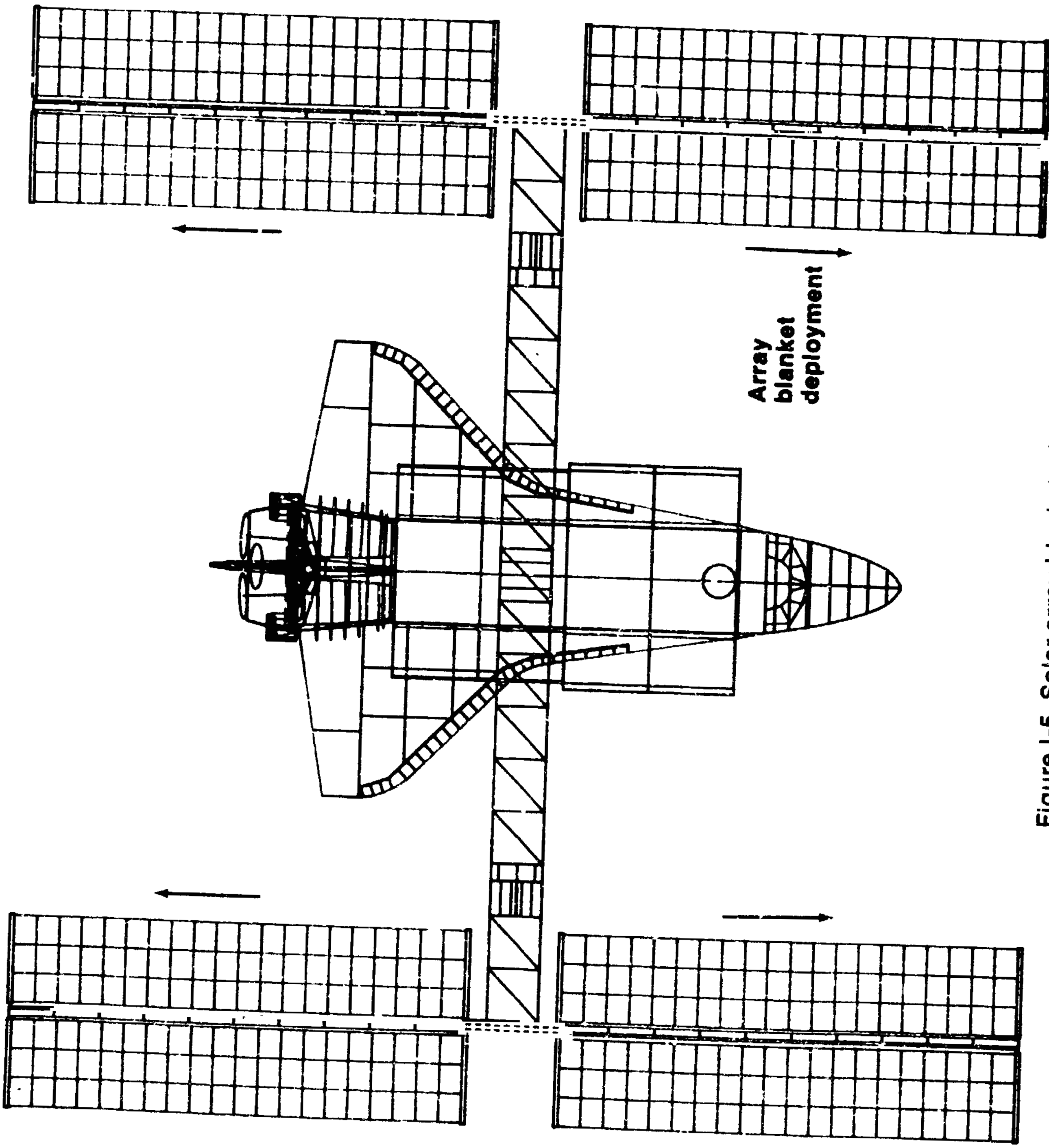


Figure I-5 Solar array blankets deployed.

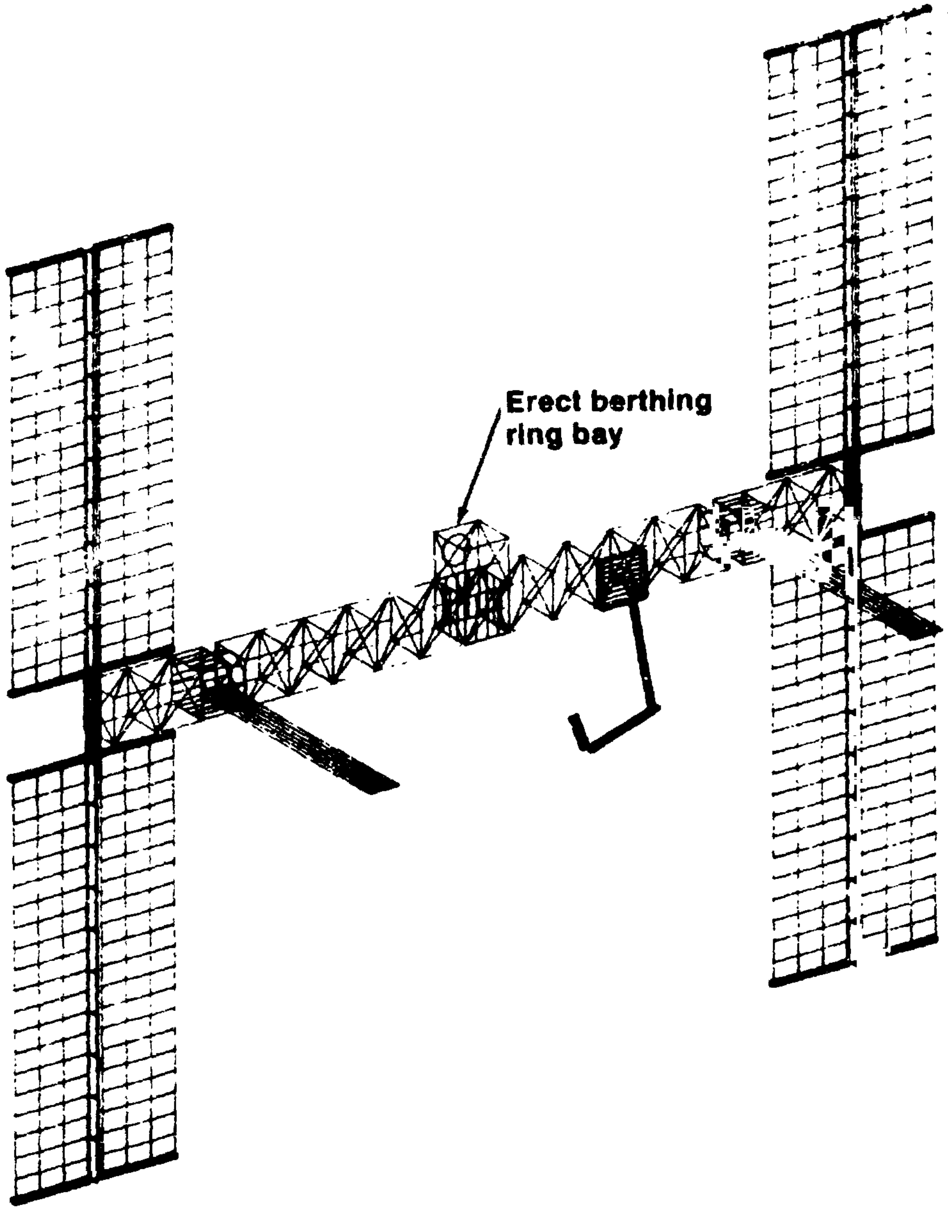


Figure I-6 Side bay erected on transverse boom to accommodate a berthing ring for Flight II.

FLIGHT II

	TASK/OPERATION	FIGURE	METHOD
1.	Remove lower keel package from payload bay; unfold deployment rails; attach package to transverse boom	II-1	MRMS/EVA
2.	Deploy lower keel deployable structure; unfold radiator support booms	II-2	Mechanism
3.	Unfold radiator arms; erect two keel extension bays on the port and starboard sides of the lower keel boom	II-3	MRMS/EVA
4.	Install radiator panels in the port and starboard heat exchanger booms	II-3	MRMS/EVA
5.	Remove the port keel extension boom package from the cargo bay; transport to attachment site; unfold deployment rails; attach package to erected keel extension bay on port side	II-4	MRMS/EVA
6.	Deploy port keel extension structure	II-4	Mechanism
7.	Remove the starboard keel extension boom package from the cargo bay; transport to attachment site; unfold deployment rails; attach package to erected keel extension bay on starboard side	II-5	MRMS/EVA
8.	Deploy starboard keel extension structure	II-6	Mechanism
9.	Erect keel extension interconnect bays		MRMS/EVA
10.	Perform systems checkout; depart	II-6	Orbiter Systems

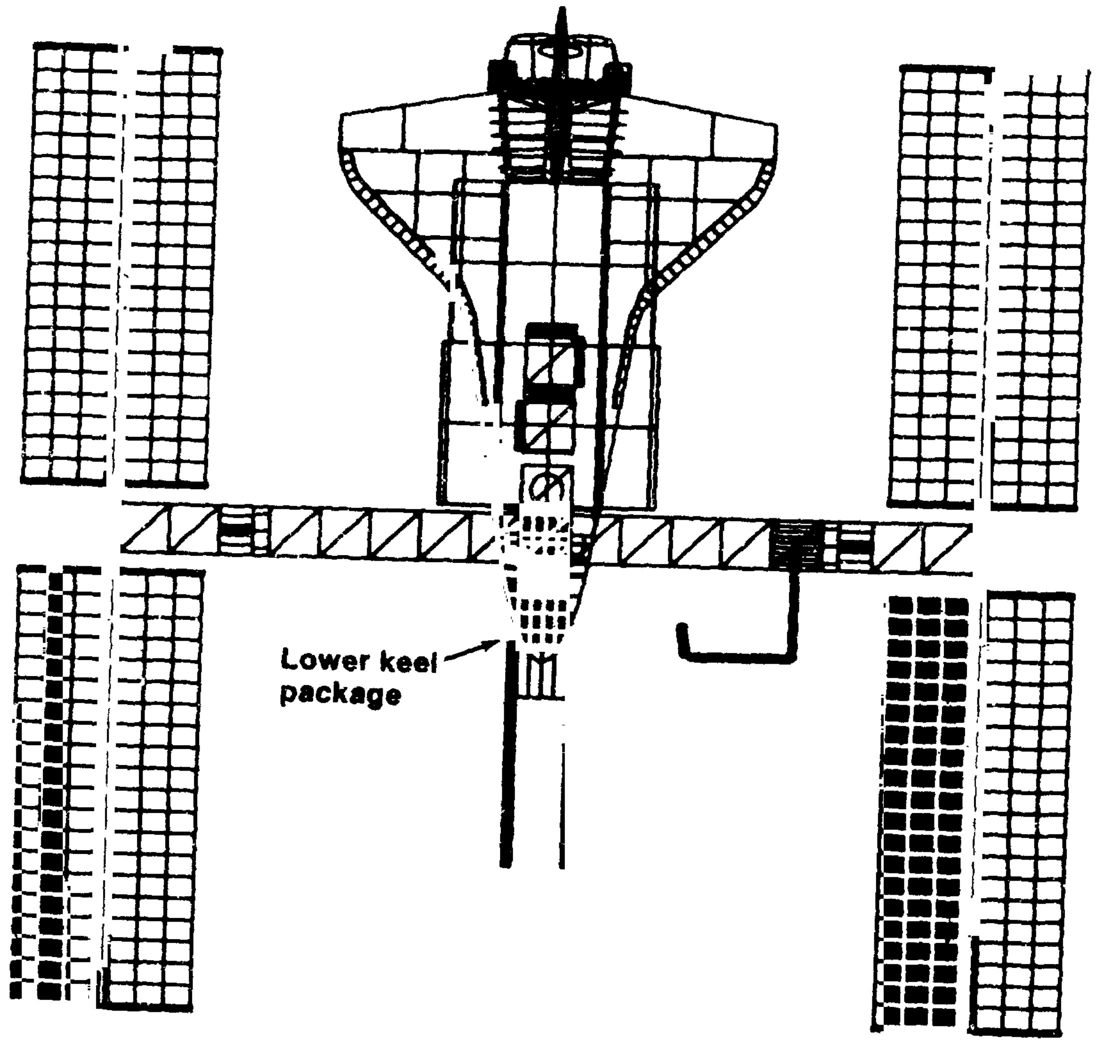


Figure II-1 Beginning of second flight build-up. Shuttle shown attached to berthing ring on first flight package. Lower keel package shown removed from cargo bay and attached ready for deployment.

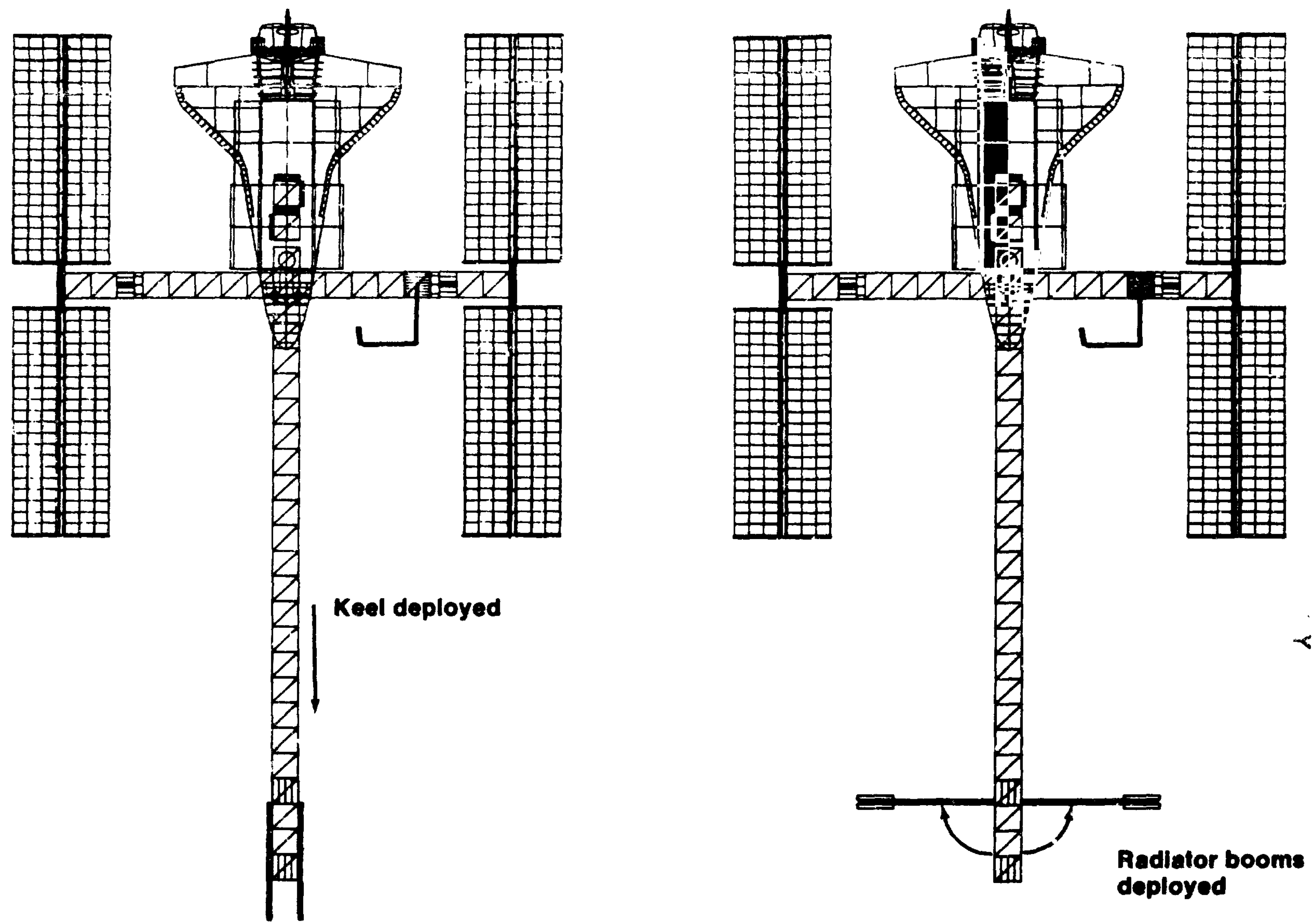


Figure II-2 Lower keel structure and radiator boom deployed.

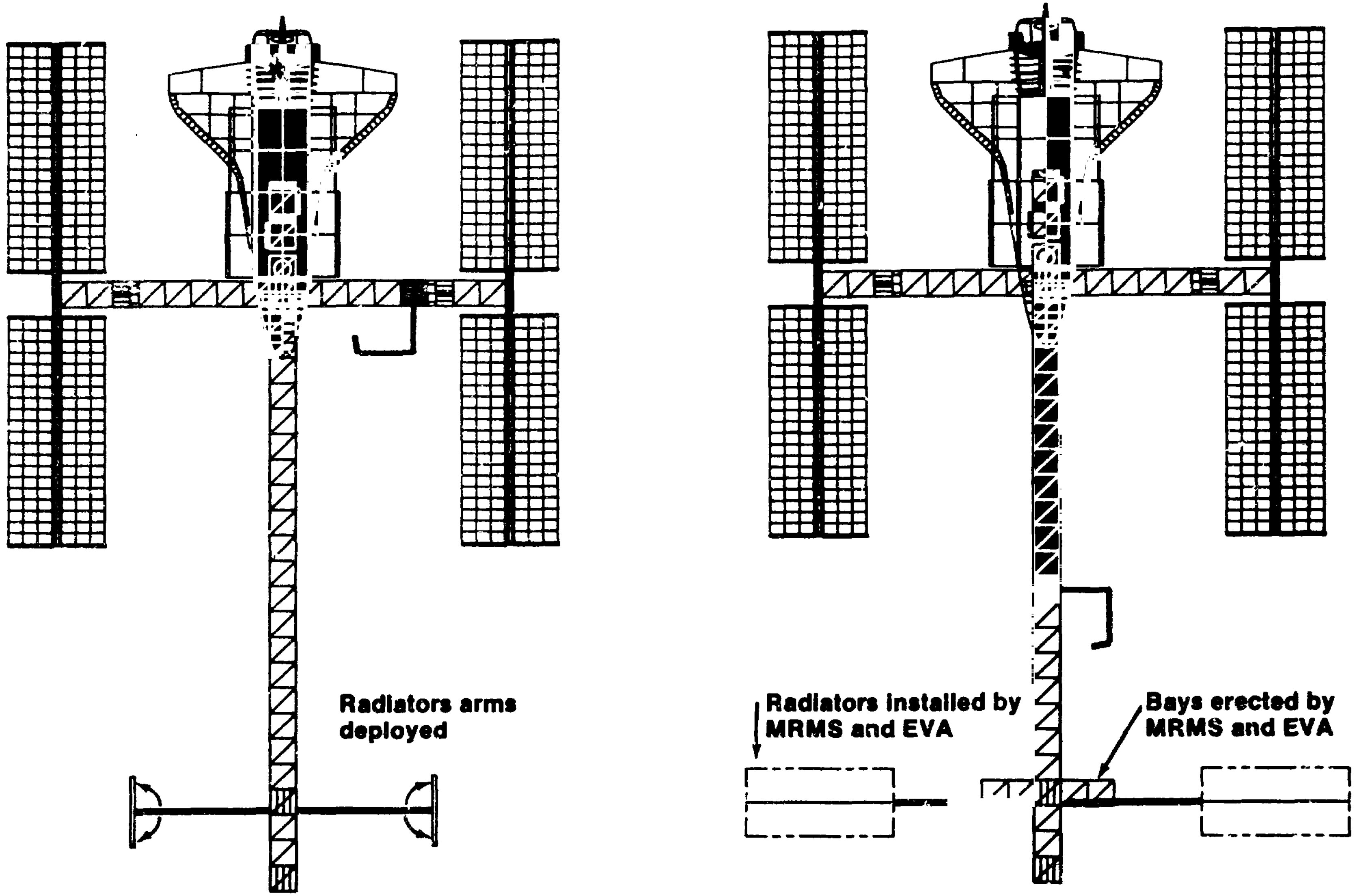


Figure II-3 Radiator arms deployed and radiators installed.

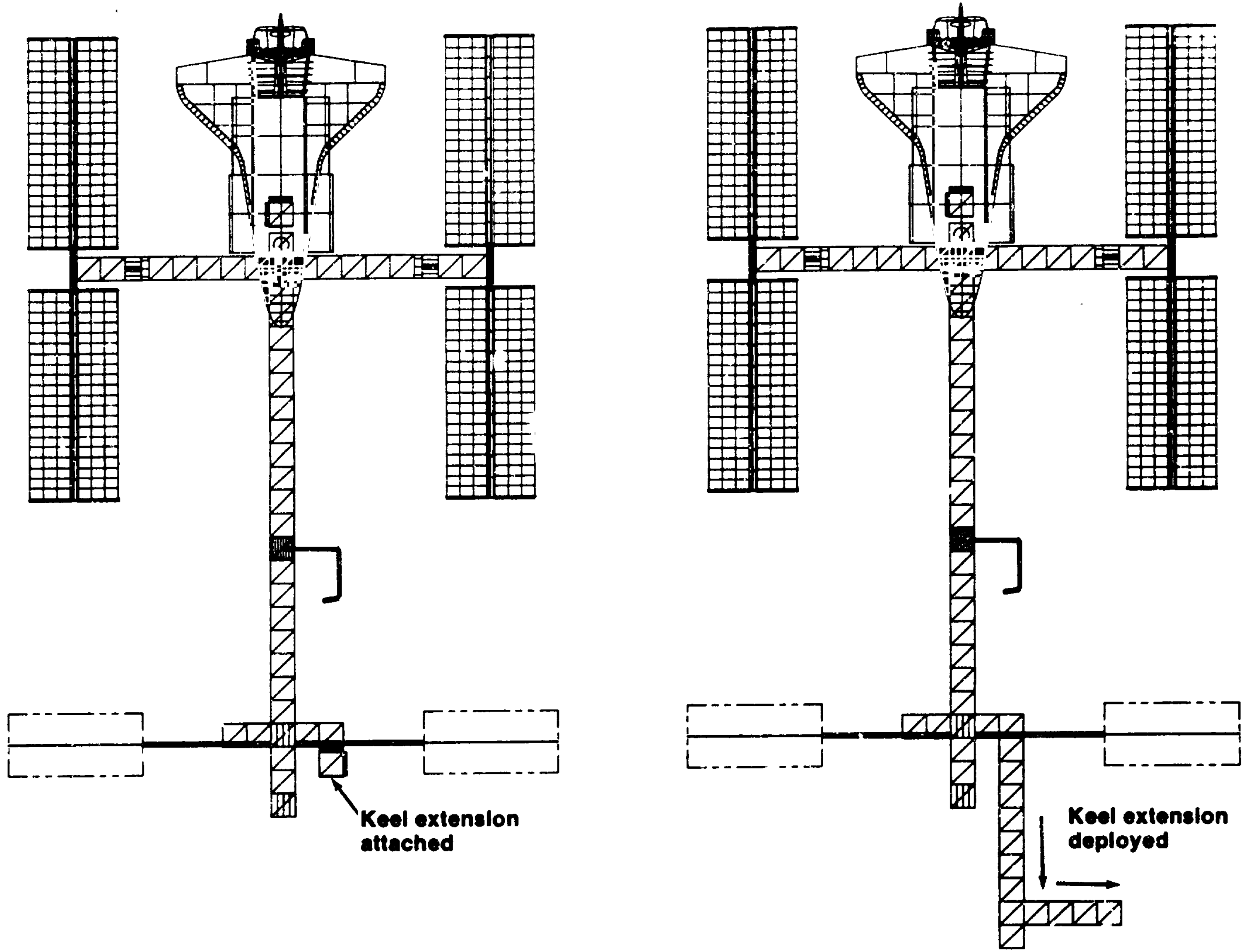


Figure II-4 Port keel extension attached and deployed.

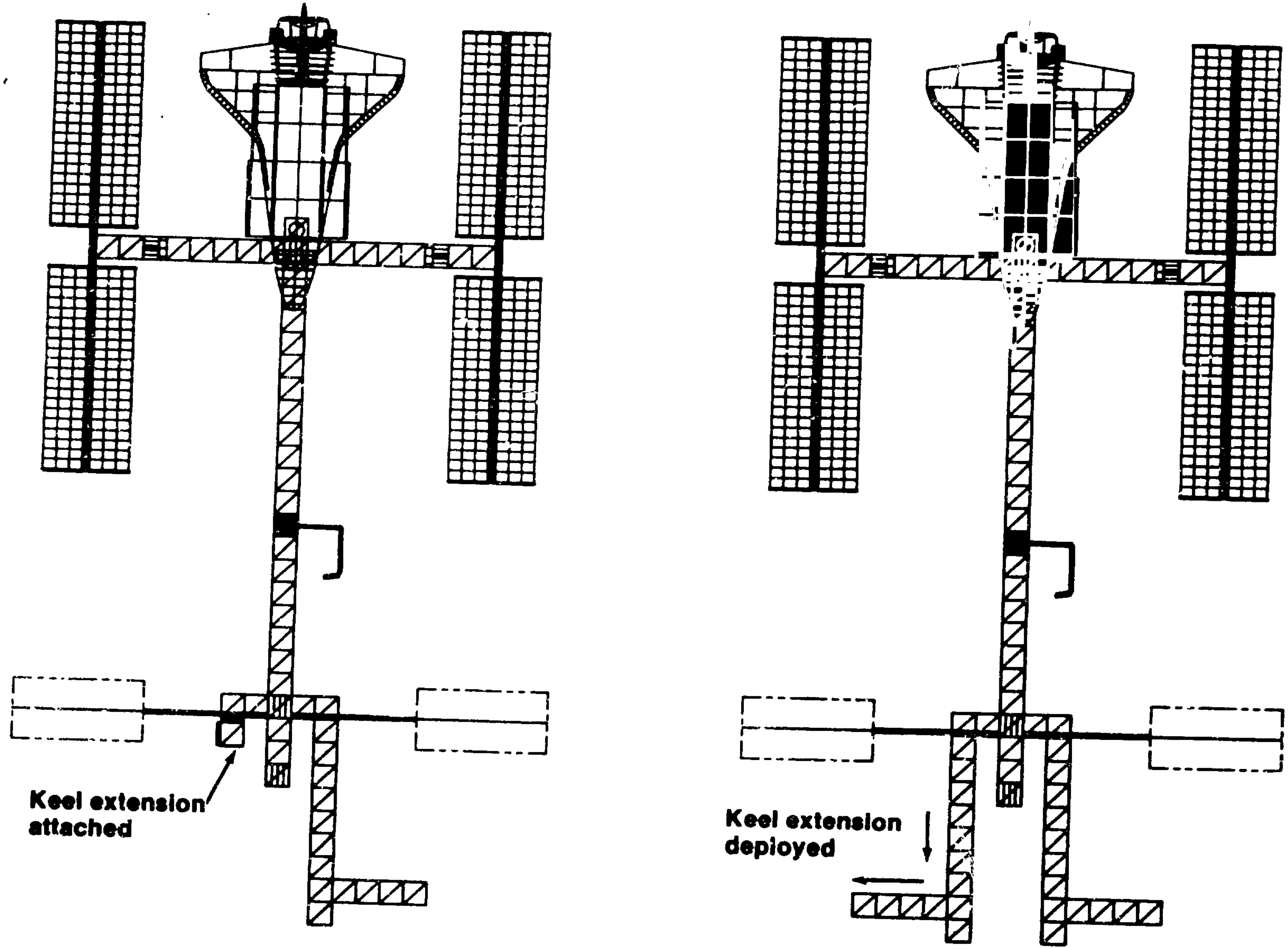


Figure II-5 Starboard keel extension attached and deployed.

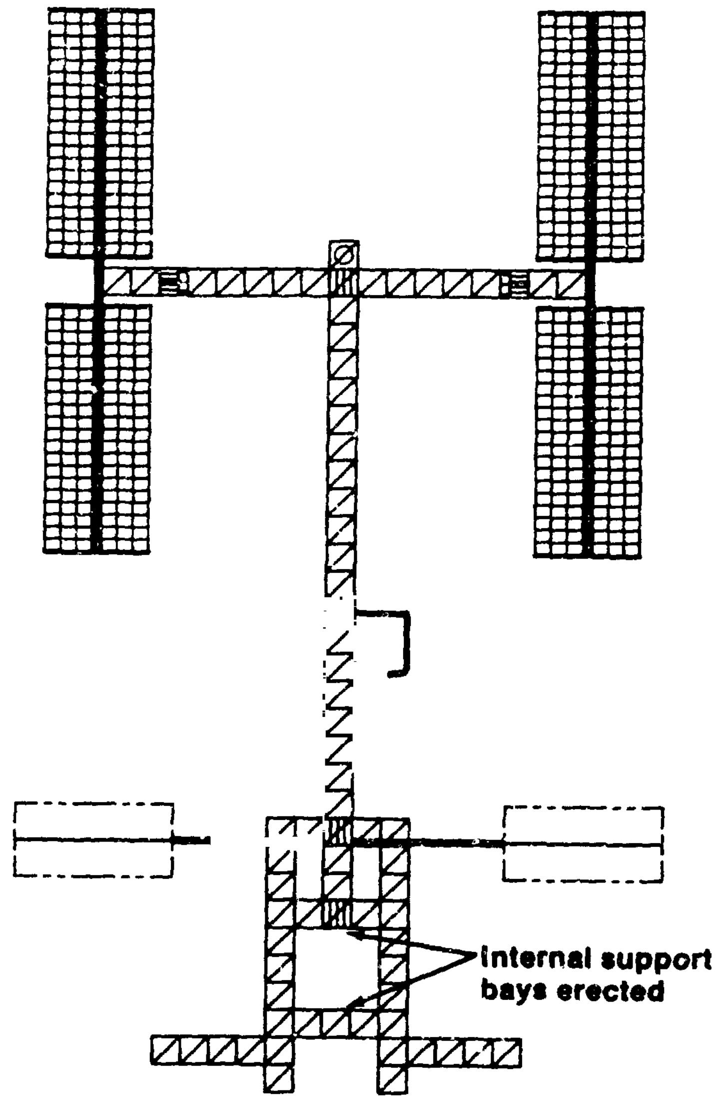


Figure II-6 Completed lower keel structure showing support bays erected between the keel extensions.

FLIGHT III

	TASK/OPERATION	FIGURE	METHOD
1.	Remove HM1 from payload bay; attach to keel extension structure	III-1	MRMS/EVA
2.	Remove AL2 from payload bay; attach to HM1	III-2	MRMS/EVA
3.	Remove AL1 from payload bay; attach to HM1		MRMS/EVA
4.	Perform systems checkout; depart	III-3	Orbiter Systems

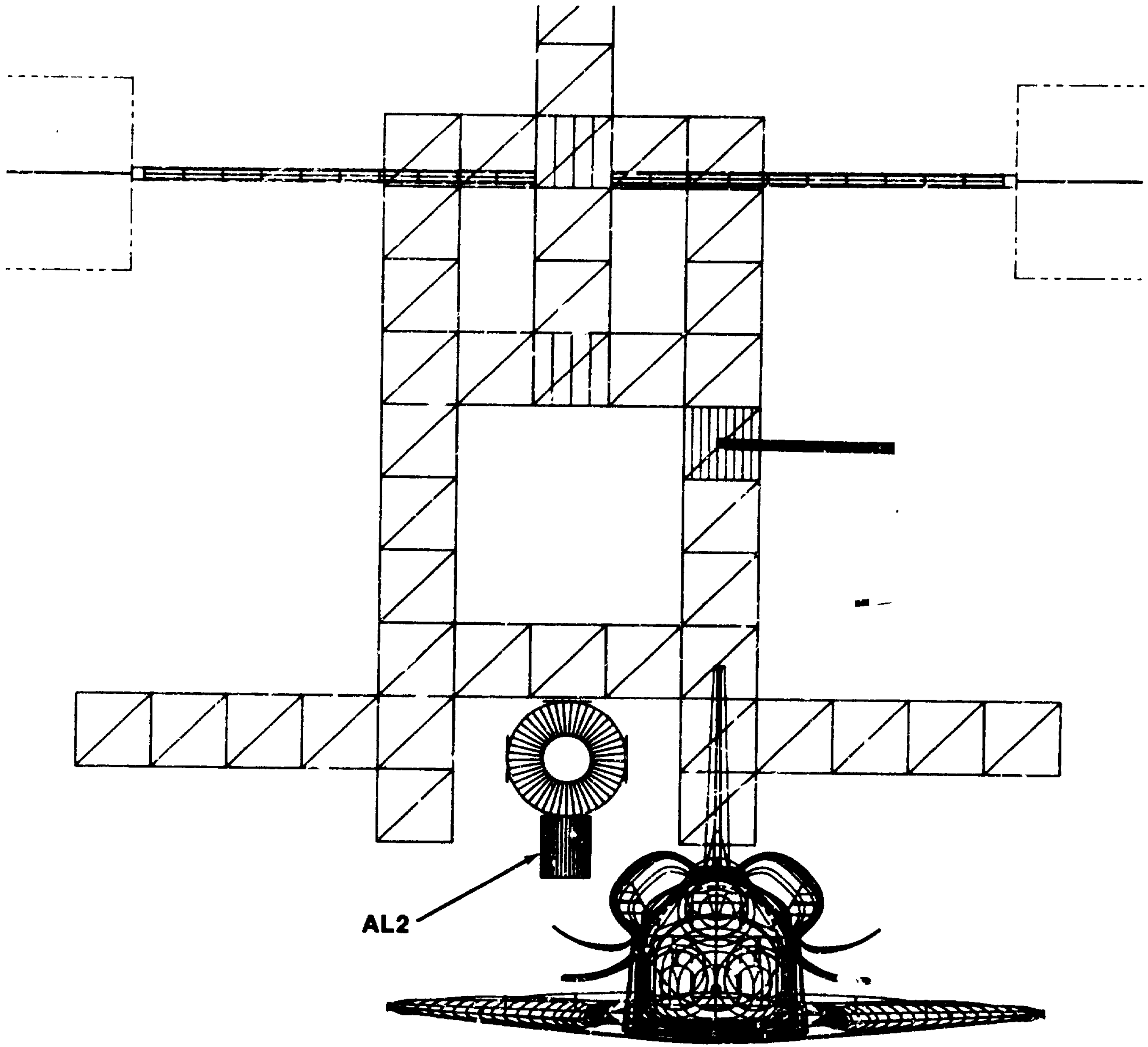


Figure III-2 AL2 temporarily attached to HM1.

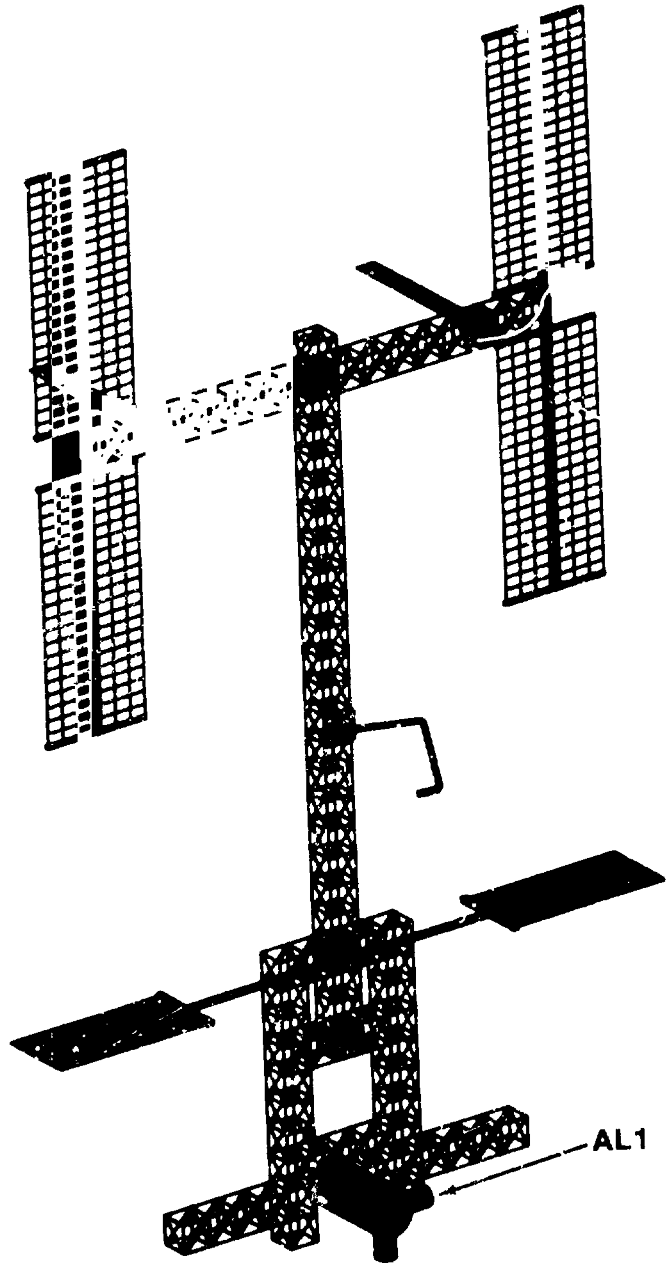


Figure III-3 Configuration at end of Flight III.

FLIGHT IV

	TASK/OPERATION	FIGURE	METHOD
1.	Remove HM2 from payload bay; attach HM1	IV-1	MRMS/EVA
2.	Attach AL2 to HM2		MRMS
3.	Remove upper keel structure package from payload bay; transport to attachment site; unfold deployment rails; attach package to transverse beam	IV-2	MRMS/EVA
4.	Deploy upper keel and uppe' boom structure		Mechanism
5.	Perform systems checkout; depart	IV-3	Orbiter Systems

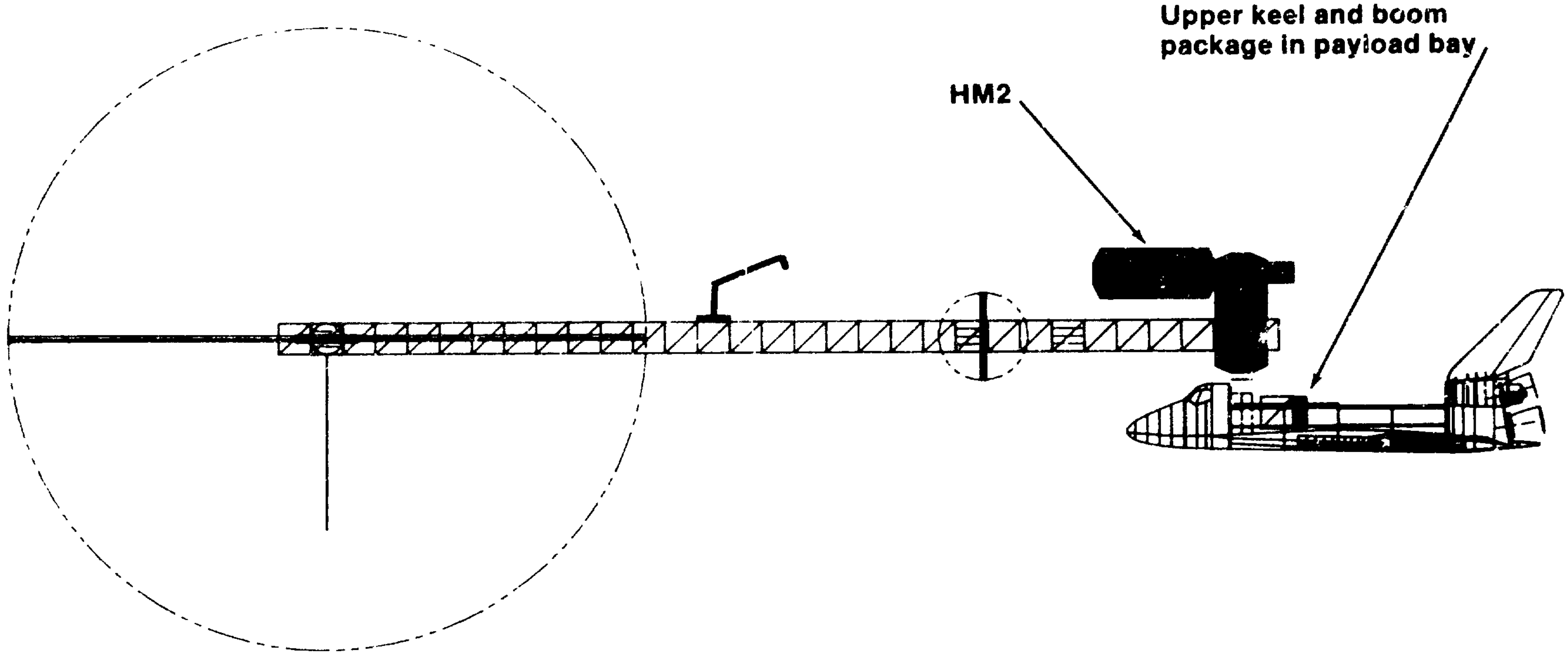


Figure IV-1 HM2 attached to HM1.

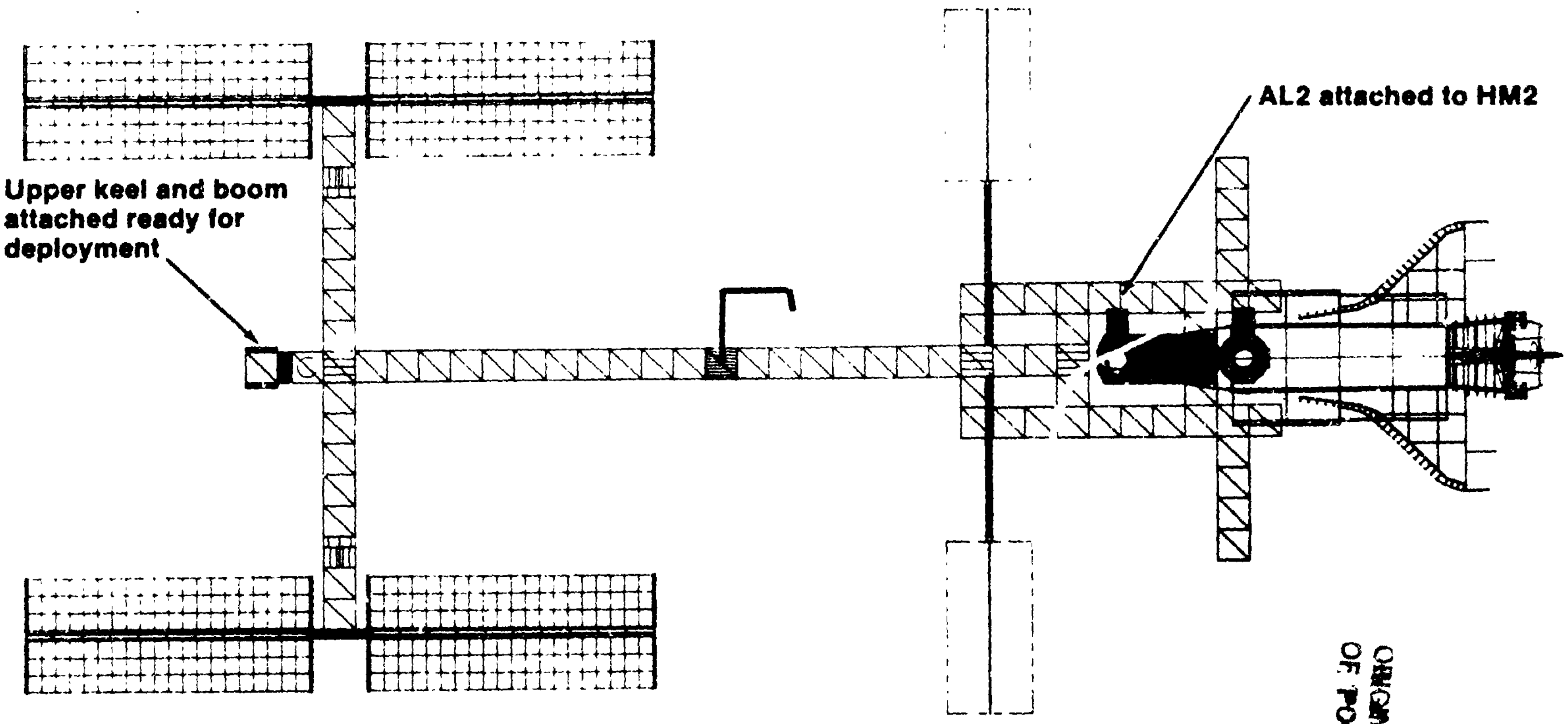
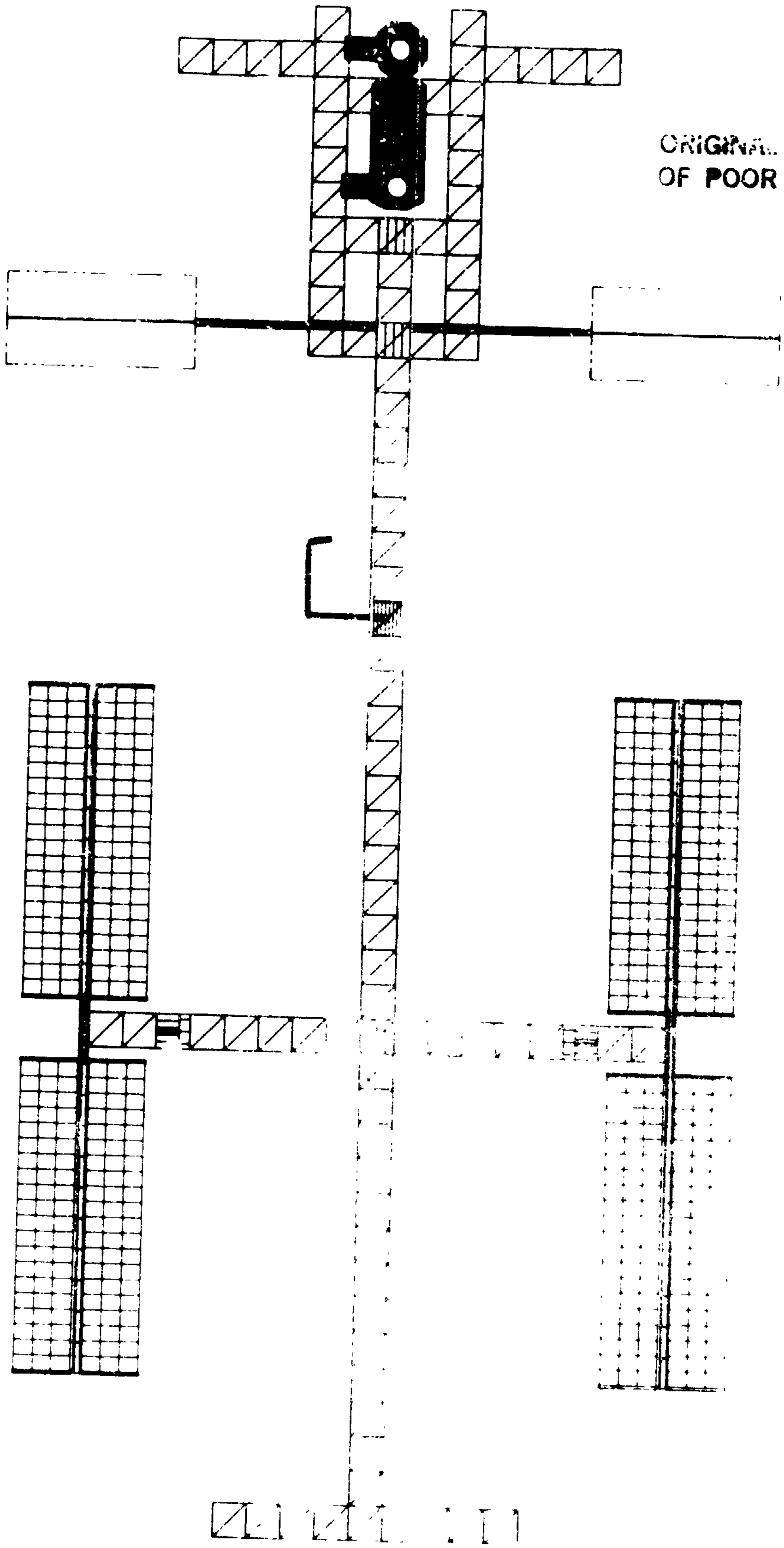


Figure IV-2 Upper keel and boom package attached prior to deployment.



ORIGINAL PAGE IS
OF POOR QUALITY

[illegible] keel and boom structure deployed.

FLIGHT V

	TASK/OPERATION	FIGURE	METHOD
1.	Remove LOG1 from payload bay; attach to HM2	V-1	SRMS/EVA MRMS
2.	Remove port solar array addition package from payload bay; transport to attachment site; attach to transverse beam		SRMS/EVA MRMS
3.	Deploy port outboard transverse beam structure	V-2	Mechanism
4.	Remove starboard solar array addition package from payload bay; transport to attachment site; attach to transverse beam		SRMS/EVA MRMS
5.	Deploy starboard outboard transverse beam structure	V-3	Mechanism
6.	Deploy solar array blanket boxes; deploy solar array blankets		Mechanism
7.	Perform systems checkout; depart	V-4	Orbiter Systems

It was assumed for scenario development purposes that the Station was permanently manned at this point in the assembly sequence. Many operations could be conducted at this point including performing EVA assembly tasks such as verification of attachments, installation of additional radiator panels, and installation of permanent hard lines as necessary. This will greatly reduce the amount of time that an orbiter has to be on-orbit to support the assembly operations.

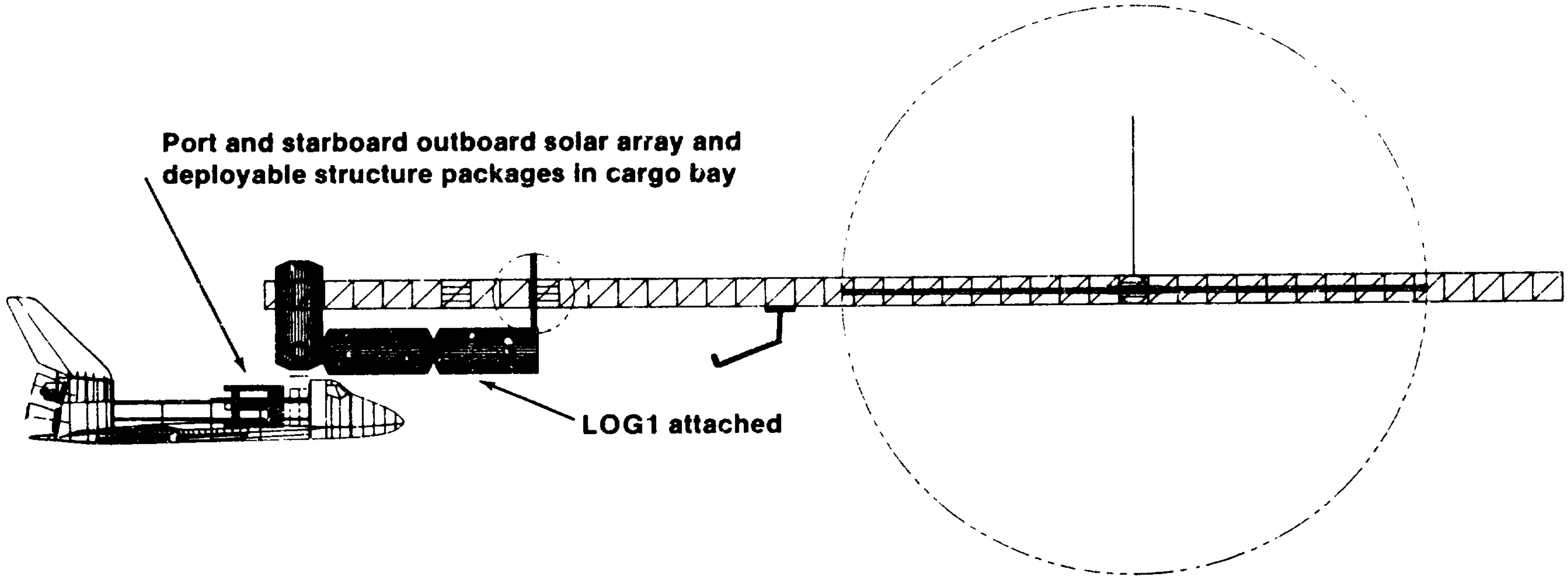


Figure V-1 Logistics module attached.

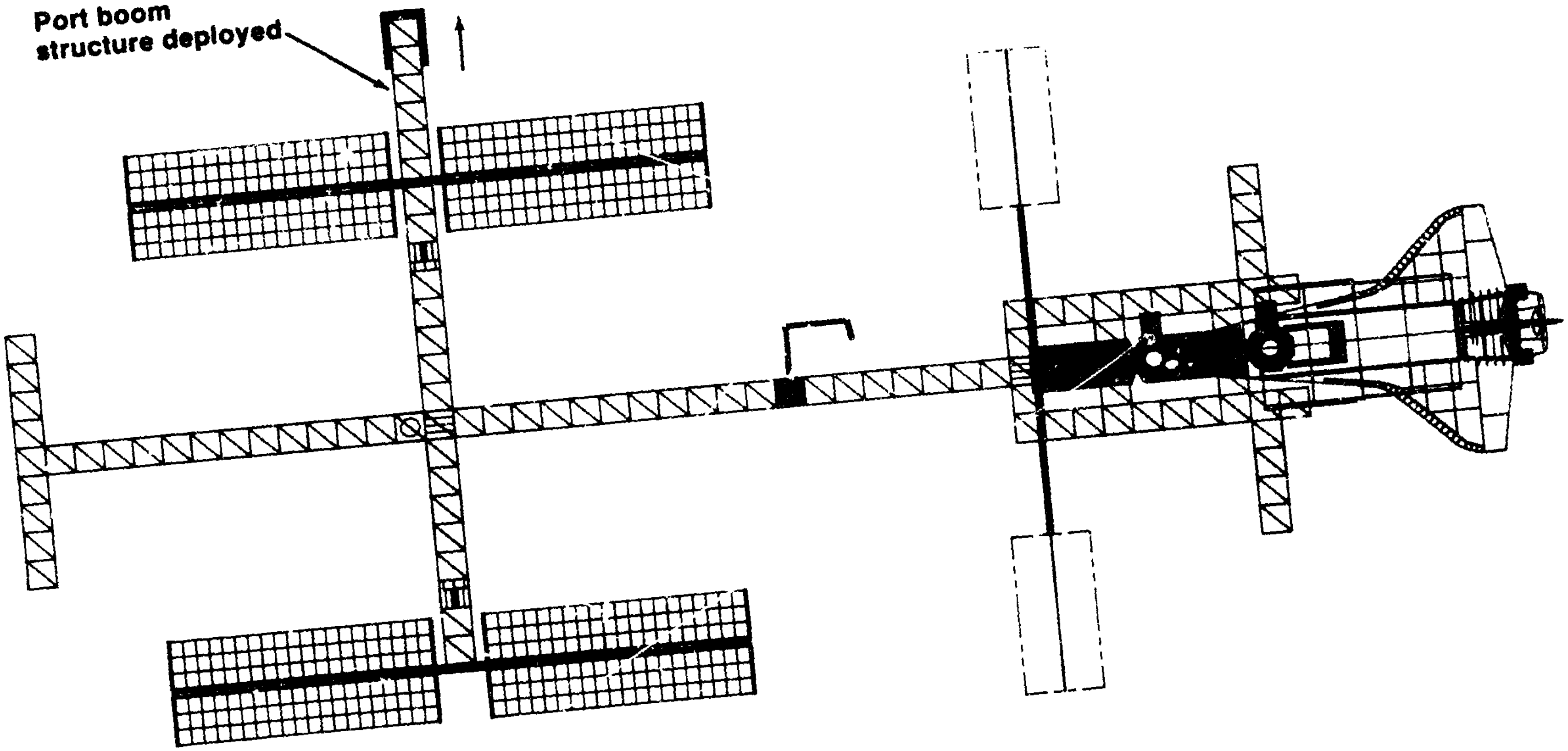


Figure V-2 Port outboard transverse boom addition deployed.

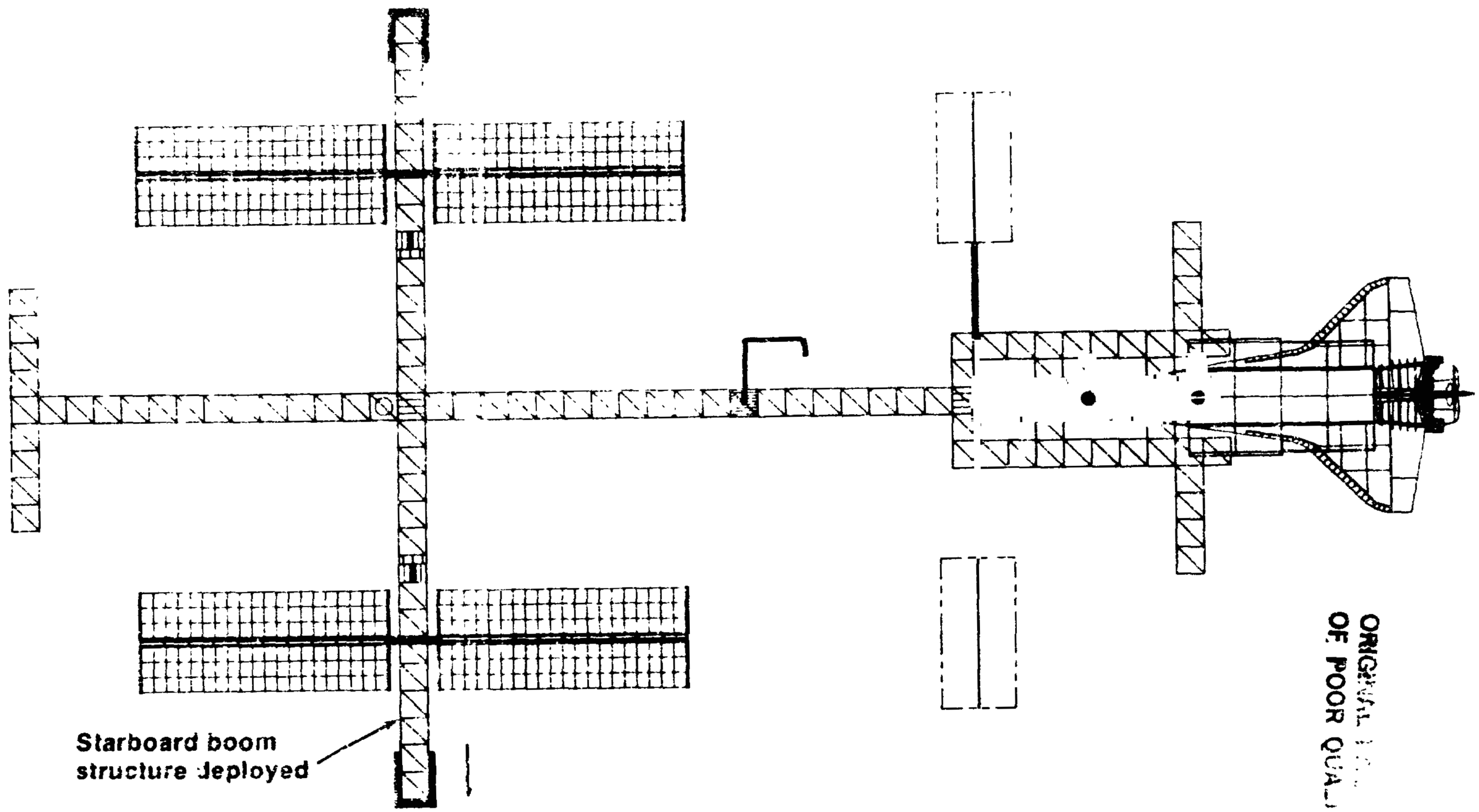


Figure V-3 Starboard outboard transverse boom addition deployed.

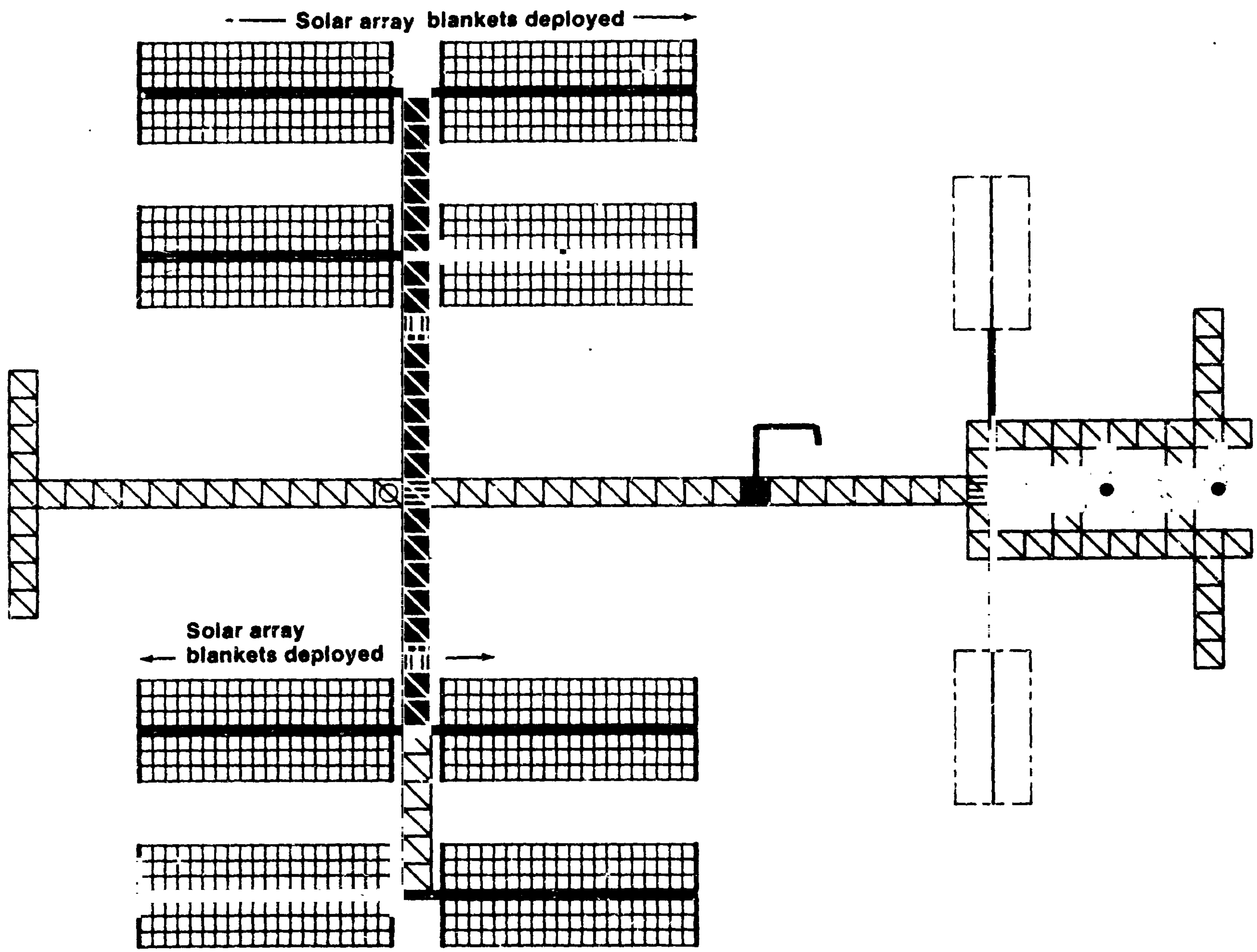


Figure V-4 Configuration at end of Flight V showing outboard structure and solar arrays deployed.

FLIGHT VI

TASK/OPERATION	FIGURE	METHOD
1. Remove LAB2 from payload bay; attach to HM2 and to keel extension structure		SRMS/EVA MRMS
2. Perform systems checkout, depart	VI-1	Orbiter Systems

No major Station element other than LAB2 has been designated for launcn on Flight VI. Such items as Station systems spares and external payloads could be delivered on this flight as suggested in the payload summary given in 4.3.7.2. It is felt that as further studies are conducted many items will be identified as being required early in the build-up sequence. Hence, the available volume on this flight and on Flight VII was intentionally illustrated in the launch packaging figures in 4.3.7.3.

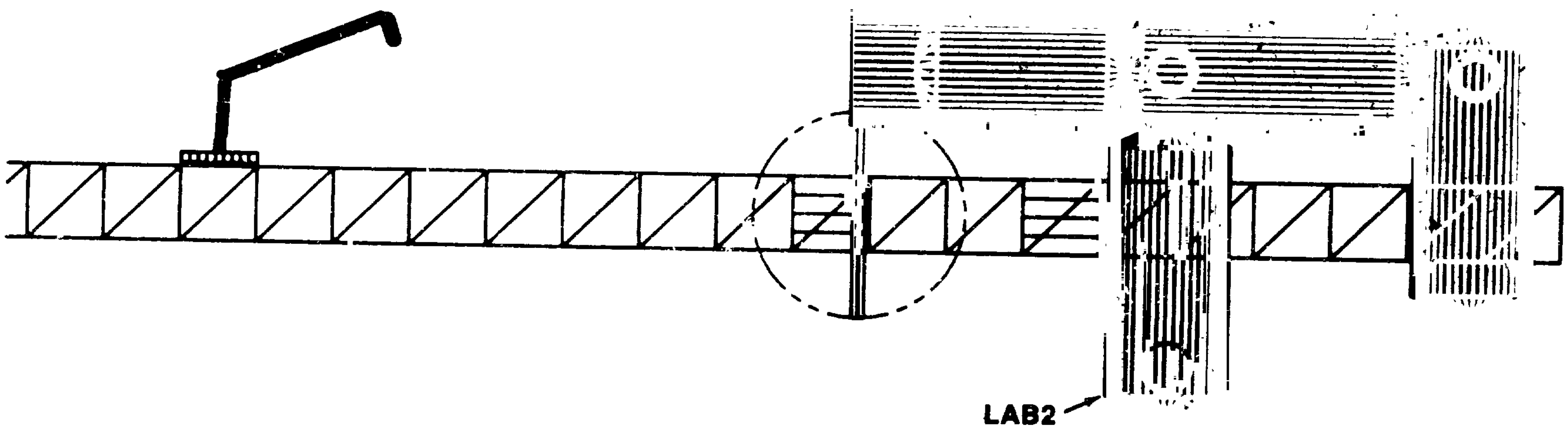


Figure VI-1 Configuration at end of Flight VI showing LAB2 attached.

FLIGHT VII

	TASK/OPERATION	FIGURE	METHOD
1.	Remove LAB1 from payload bay; attach to HM1, and LAB2		SRMS/EVA MRMS
2.	Perform systems checkout; depart	VII-1	Orbiter Systems

As on the previous flight, there exists a volume for miscellaneous items or payloads in the launch package.

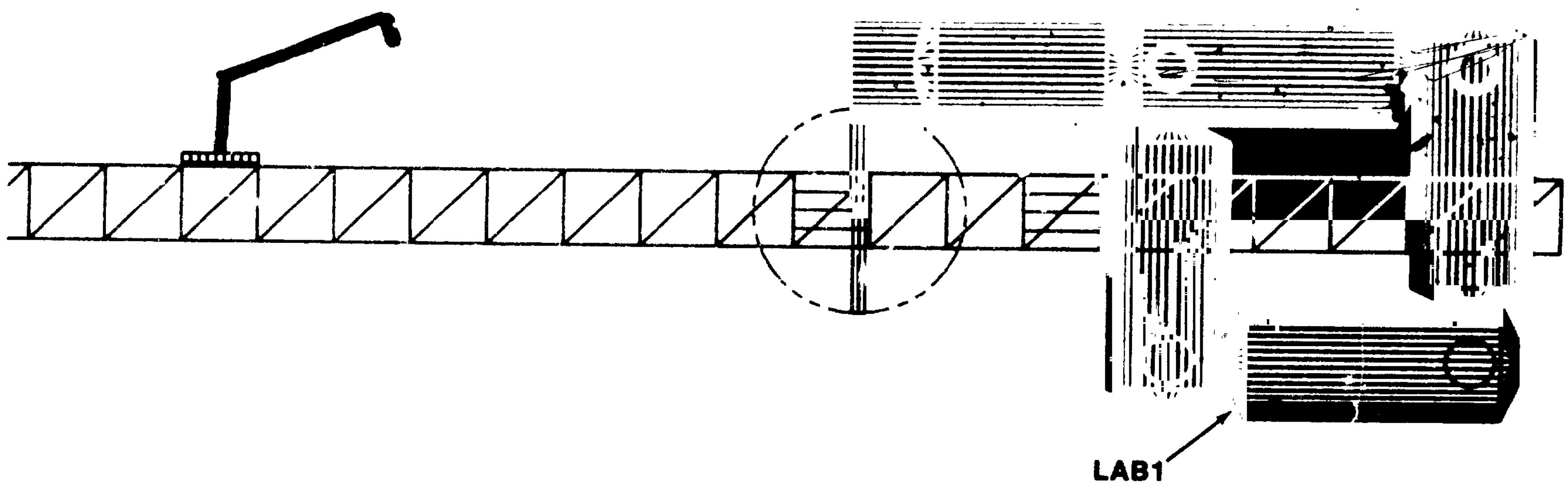


Figure VII-1 Configuration at end of Flight VII showing LAB1 attached and the "racetrack" completed.

Figure 4.3.7.4-1 illustrates the final configuration of the IOC Reference Space Station. Assembly techniques described in the reference scenario would be applicable to the envisioned growth configuration of the Station.

4.3.7.5 Assembly Operations Crew Timelines

As a part of the study estimates were made of the crew timelines required to perform the on-orbit assembly operations. These estimates were based on NASA Contractor Report 3751, "Analysis of Large Space Structures Assembly", and on SRMS, EVA, and neutral buoyancy testing experience.

Table 4.3.7.5 1 lists the estimated number of hours of two-man EVA's required to perform the assembly tasks associated with each flight in the reference scenario.

The results of this analysis, including a breakdown of the EVA, SRMS, and MRMS operations required to perform the assembly tasks described in 4.3.7.4, are detailed in Appendix A.

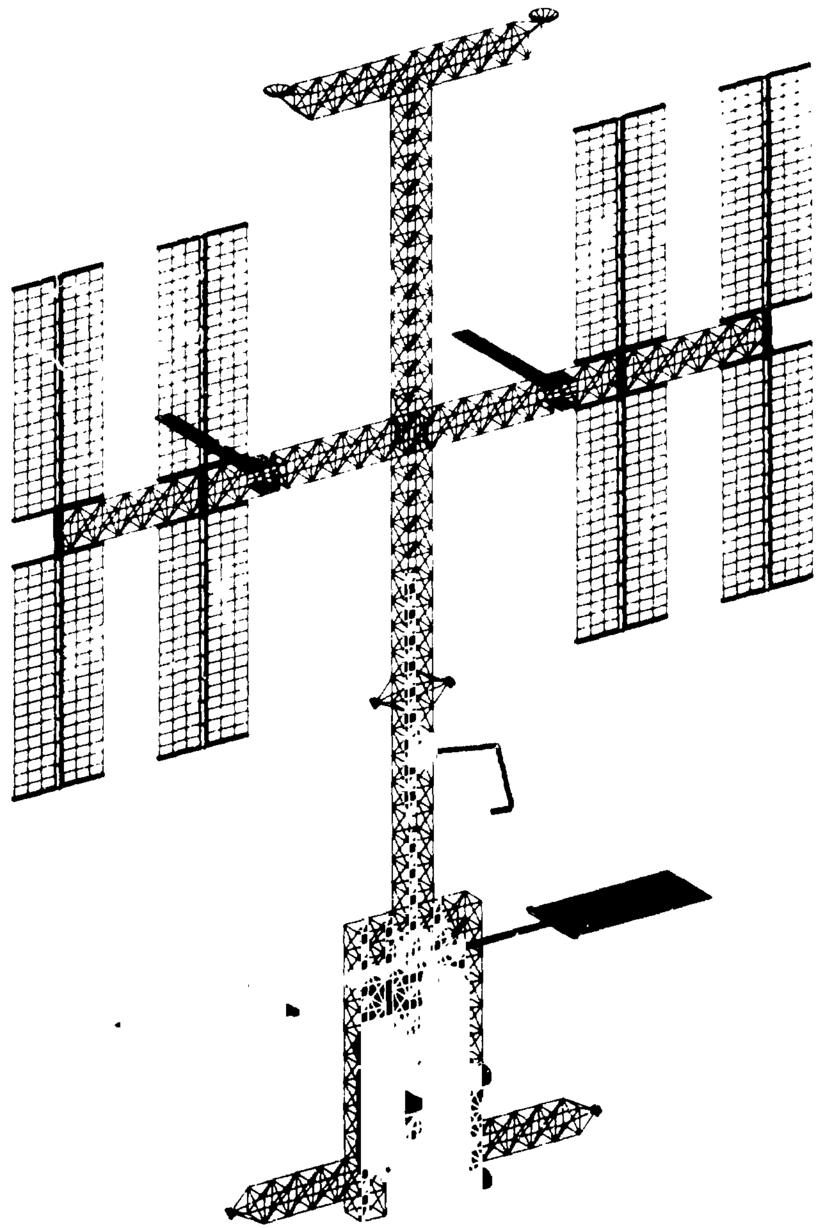


Figure 4.3.7.4-1 IOC reference configuration space station.

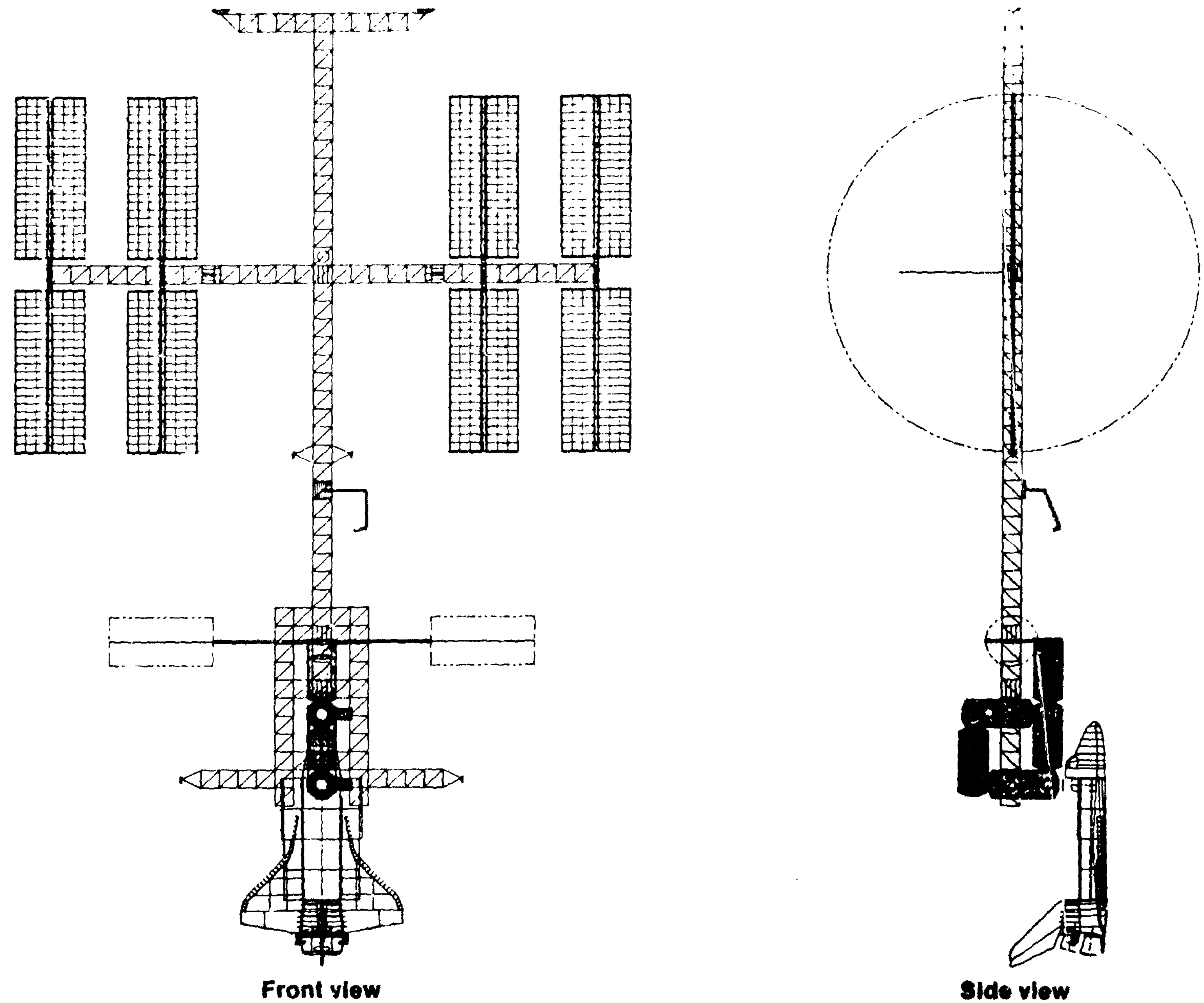


Figure 4.3.7.4-1 (cont.).

4.3.8 Operations

4.3.8.1 Manipulator Operations

Assembly of the Space Station, as well as the placement and servicing of externally mounted payloads and maintenance of station ORU's all require a Mobile Remote Manipulator System (MRMS). Some of the more critical requirements of the MRMS will be station assembly, module removal, OMV/OTV berthing in the hangar area, deployment of the OMV/OTV from the hangar area and as an aid to OMV, OTV and satellite servicing. The analysis conducted emphasizes the use of the current Shuttle RMS to the maximum extent possible for assembly. The MRMS was considered only for those operations which exceeded the reach capability of the Shuttle RMS. The MRMS analysis was performed using the RMS Desk Top Planning Simulation (RPS) developed for RMS mission planning activities and used to define RMS payload handling capabilities and procedures for STS missions. The program was updated and modified to include the reference configuration, as well as a generalized manipulator in the sense that the length of the manipulator booms can be varied to accommodate larger reach envelopes than the current RMS. The number of active joints can be reduced and the booms shortened so that a Handling and Position Aid (HPA) type of mechanism can also be accurately simulated.

The manipulator analysis included herein is based on a kinematic model of the RMS in that no rigid or flexible body dynamics are included. This limitation, however, does not invalidate the feasibility of using the RMS for station assembly since all modules handled are within the weight and inertia limits verified for standard RMS operations. The study results are based on the current RMS control algorithms and software and verify the reach capability as well as the maneuver path for both the RMS and the MRMS. The simulation also checks for singularities and joint reach limits. In summary, all maneuvers studied for the reference configuration assembly sequence should be valid with the exception of possible crew visibility constraints. RMS operator eye-point and CCTV views can also be generated using the RPS simulation, and these results will be considered in future analysis. The study focuses on assembly of the major structural components and modules of the Space Station only and was performed using the following assumptions and guidelines:

a. Orbiter RMS use was restricted to the current port RMS.

b. All construction was performed with the Orbiter firmly attached to the Space Station structure.

c. The MRMS dimensions and control algorithms were identical to those of the current RMS.

d. The MRMS could move on either the front or back side of the upper and lower keels, the keel extension, and the transverse boom.

e. Crew visibility of maneuvers was not considered.

f. Grapple fixture locations and orientations considered only RMS/MRMS reach capability and did not consider module structural design at the attach point.

The RMS/MRMS maneuvers are not necessarily the most optimum maneuvers, but do verify reach capability, adequate clearances along the entire maneuver path, and freedom from encountering reach limits or singularities.

4.3.8.1.1 Station assembly

a. Flight 1 - The Orbiter port RMS is required to remove the initial package of truss structure components from the Orbiter payload bay and place it on the guide rails which permit the transverse boom to be deployed (Figure 4.3.8.1-1). The Orbiter RMS will also be required to assist in positioning the MRMS onto the truss structure once it is fully deployed.

b. Flight 2 - Once the Orbiter is docked to the structure following Flight 1, the MRMS can be used to remove the new structure packages from the Orbiter and position them for attachment to the existing structure. Figure 4.3.8.1-2a shows the MRMS removing the package that makes up the lower keel (step 1) and placing it in position for deployment (step 2). Figure 4.3.8.1-2b shows the same maneuver using the Orbiter port RMS.

c. Flight 3 - Flight 3 requires the Orbiter to dock to the auxiliary berthing ring at the module end of the lower keel as shown in Figure 4.3.8.1-3. The MRMS is then used to remove the first Habitation Module (HM1) from the Orbiter payload bay (step 1) and position it for attachment to the keel extension (step 2).

d. Flight 4 - The Orbiter is docked to the HM1 as shown in Figure 4.3.8.1-4. The rationale for docking the Orbiter to the HM1 as shown allows the module to be entered directly from the Orbiter without an EVA. This berthing port is not the primary or auxiliary, but it is required to facilitate transfer of the second Habitation Module (HM2) and still allow access to the interior of the HM1. As on Flight 3, the MRMS is used to remove the HM2 from the Orbiter payload bay (step 1) and position it as shown (step 2).

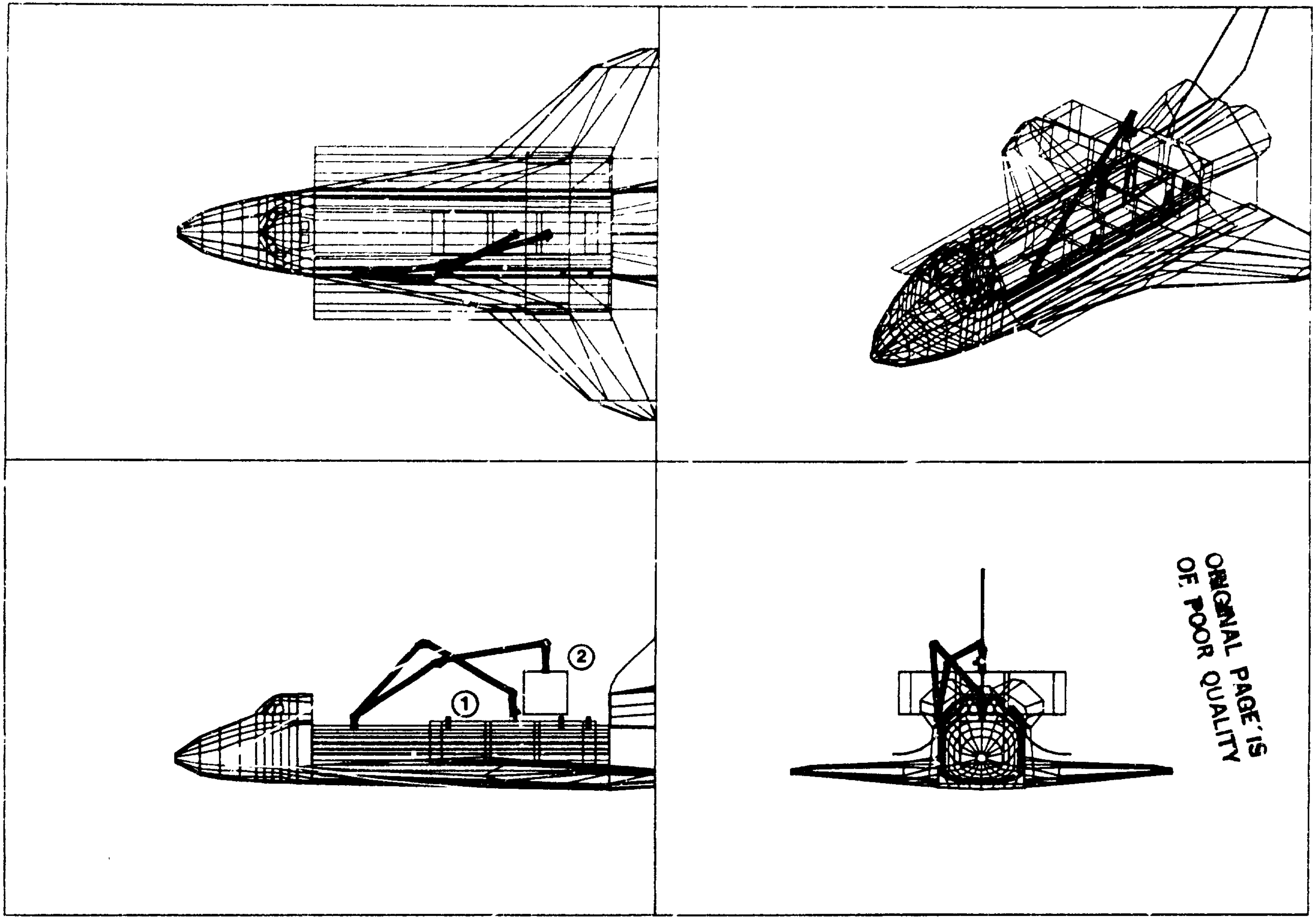


Figure 4.3.8.1-1 - FLIGHT 1 PORT RMS OPERATIONS

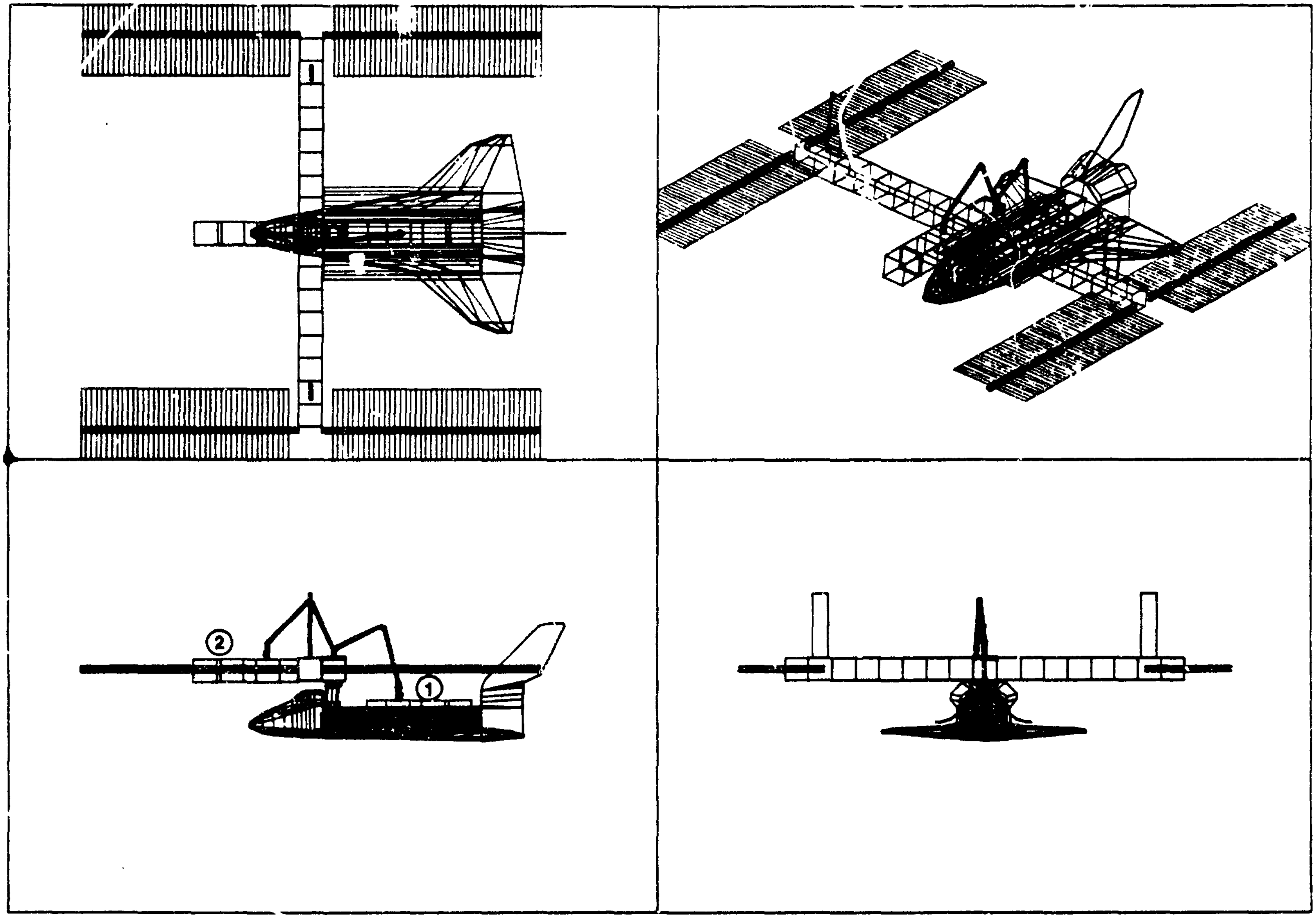
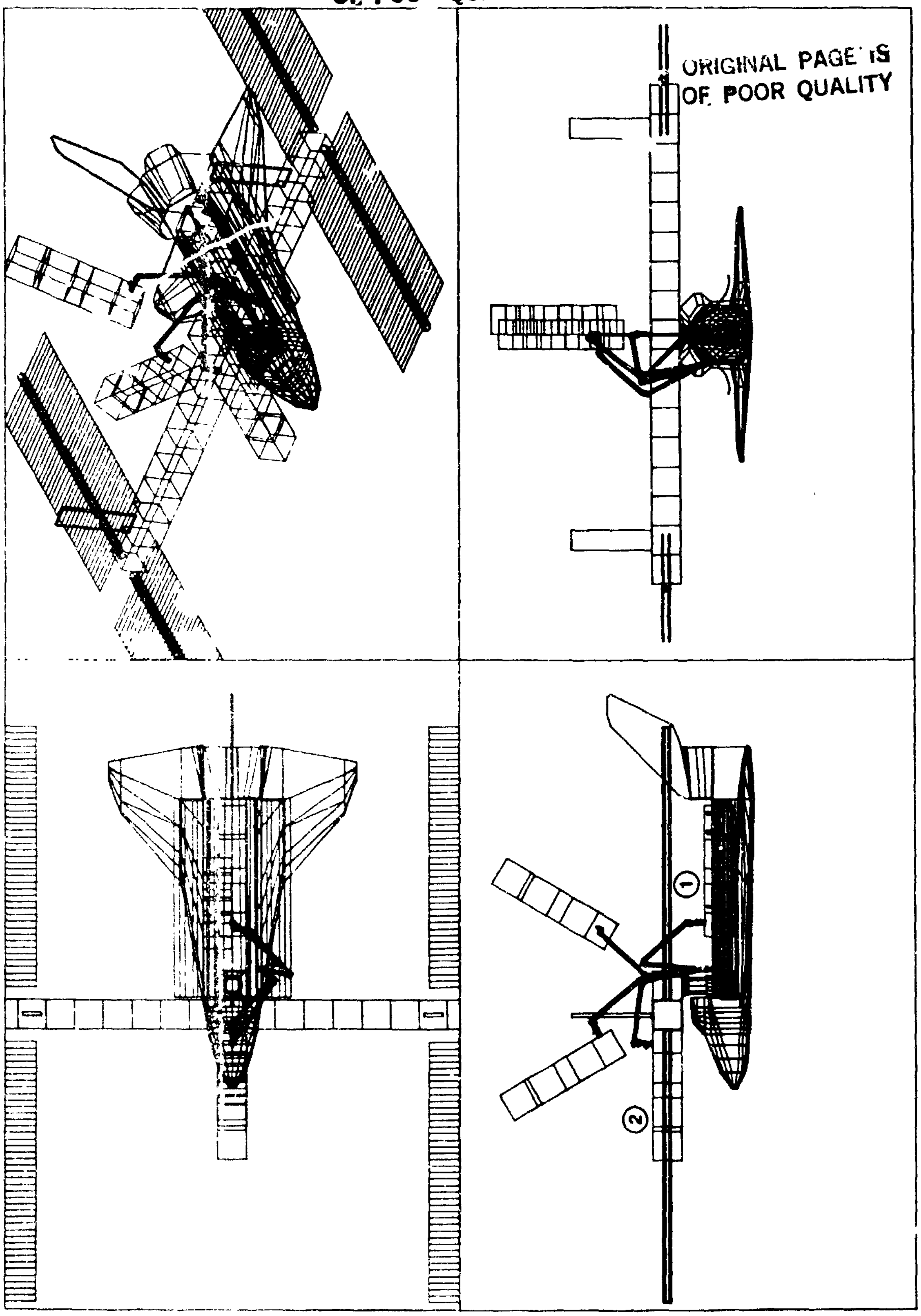


Figure 4.3.8.1-2a- FLIGHT 2 MRMS OPERATIONS



ORIGINAL PAGE IS
OF POOR QUALITY

Figure 4.3.8.1-2b - FLIGHT 2 ORBITER RMS OPERATIONS

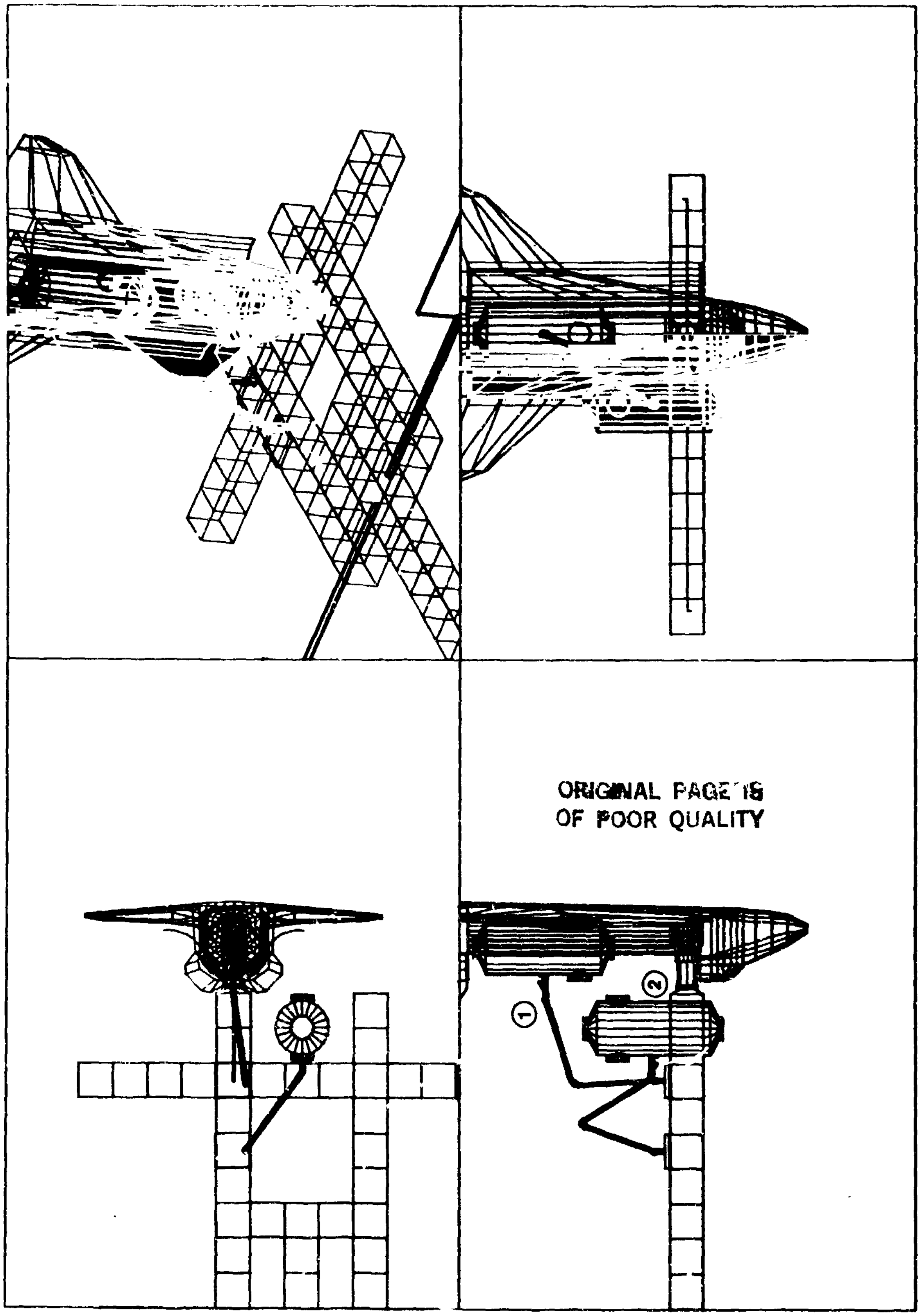


Figure 4.3.8.1-3 - FLIGHT 3 PLACEMENT OF HM1 USING MRMS

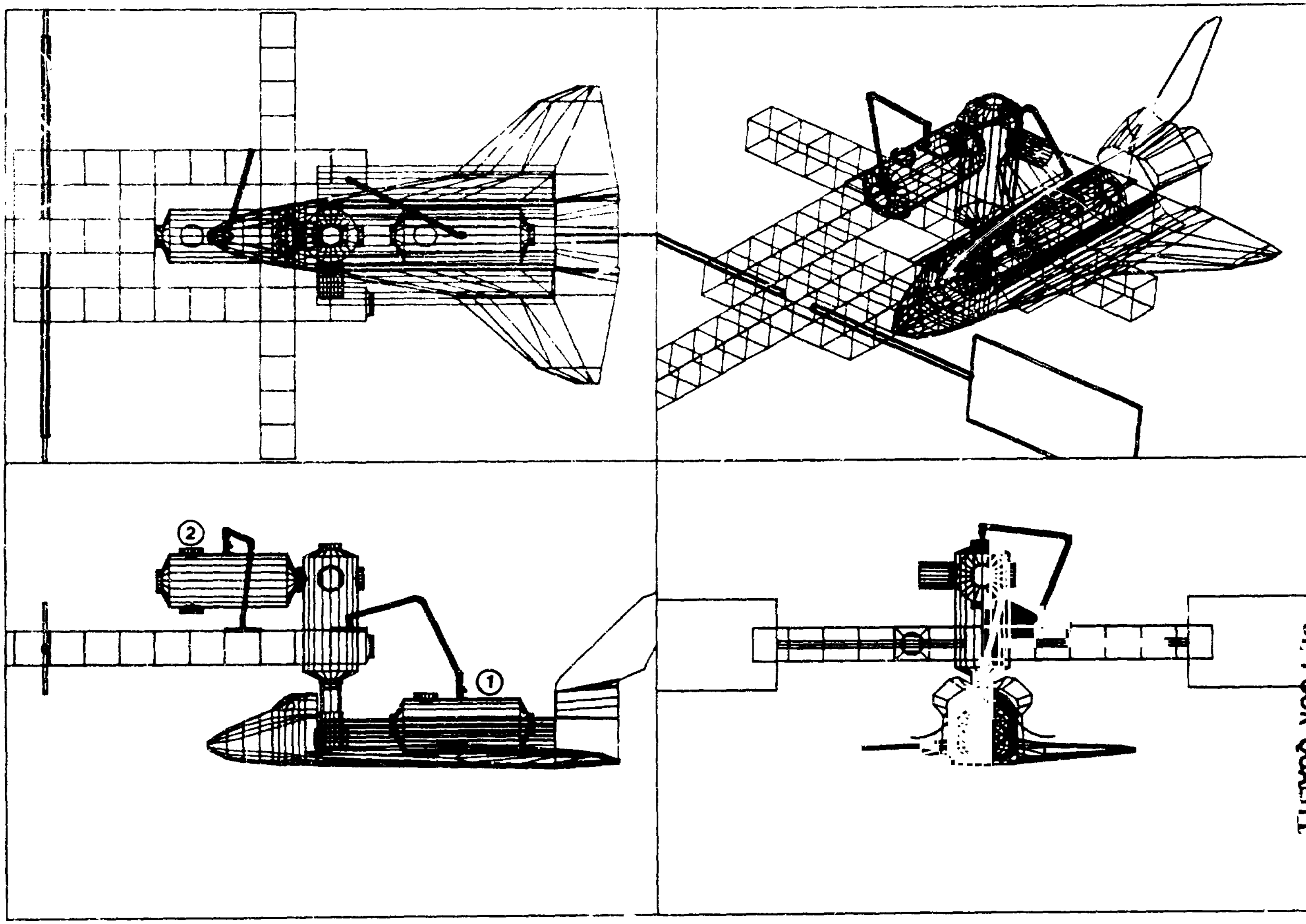
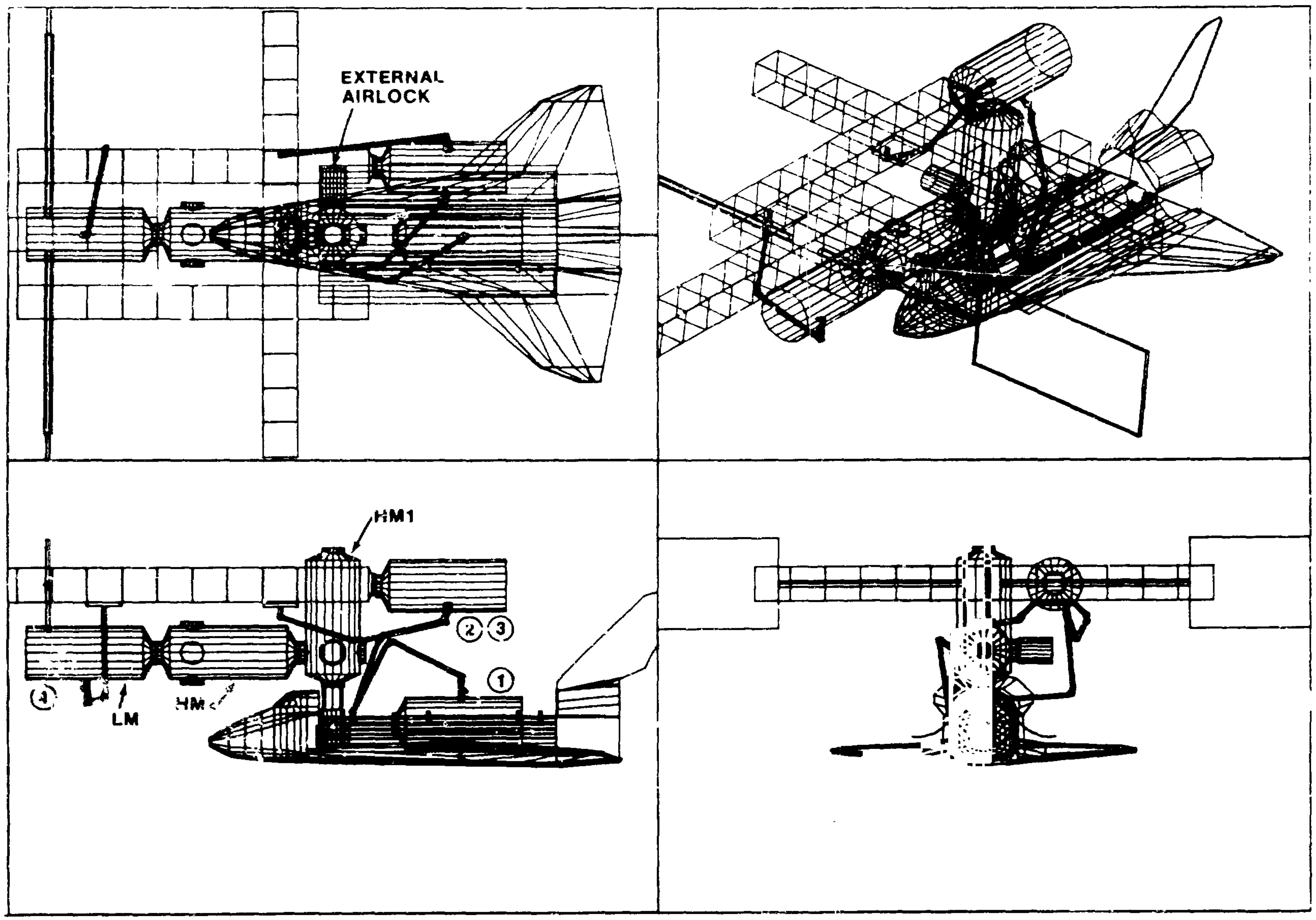


Figure 4.3.8.1-4 - FLIGHT 4 PLACEMENT OF HM2 USING MRMS

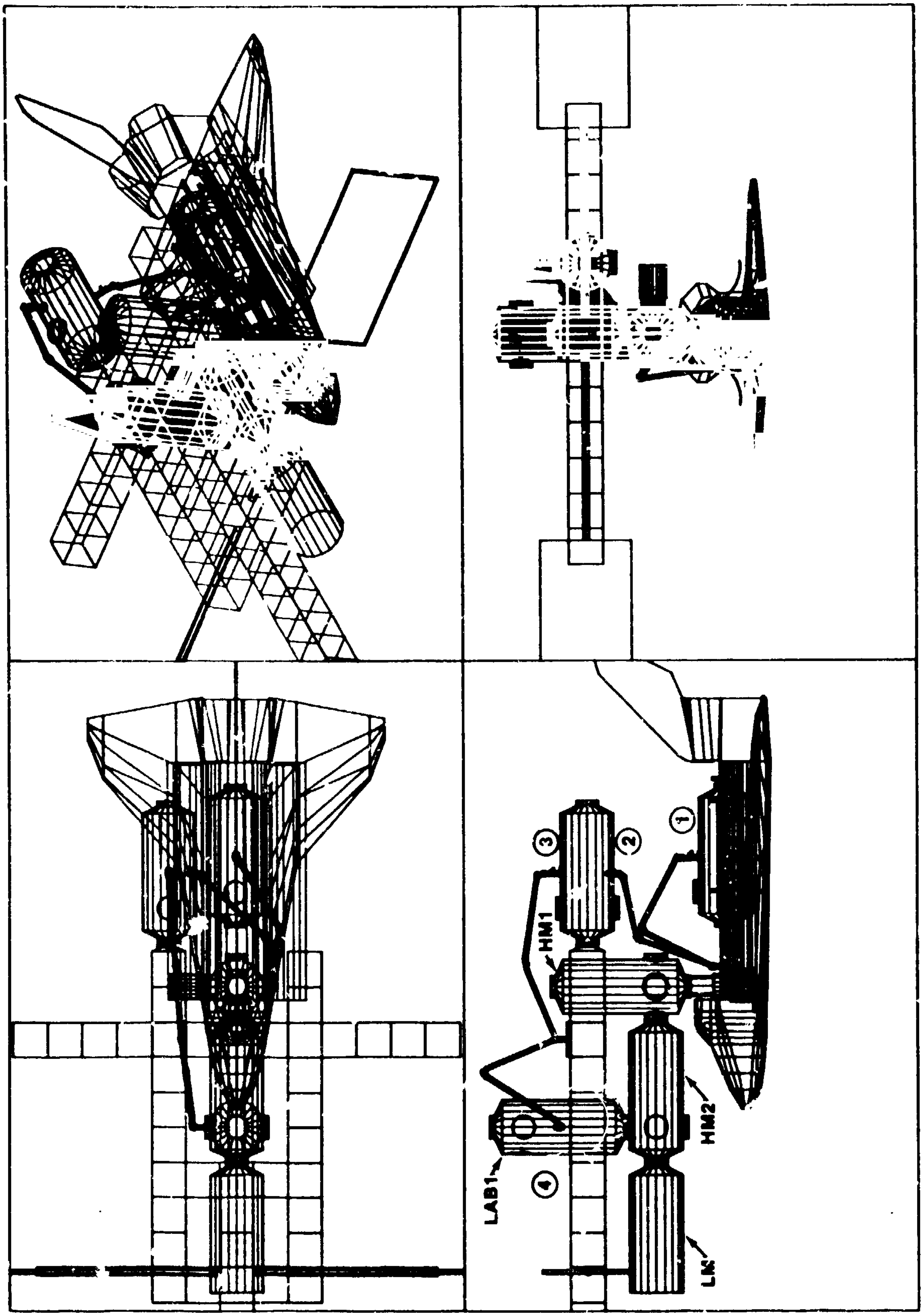
e. Flight 5 - With the Orbiter docked to the primary berthing port on the HM1 in the orientation shown in Figure 4.3.8.1-5, the Orbiter port RMS is used to remove the logistics module (LM) from the Orbiter payload bay (step 1) and attach it to the auxiliary berthing ring located on the truss structure (step 2). The MRMS is then used to remove the LM from the auxiliary berthing ring (step 3) and attach it to the HM2 (step 4). Care must be taken when maneuvering the MRMS around the external airlock attached to the HM1. For this and all remaining assembly flights, the Orbiter is attached to the primary berthing port on the HM1.

f. Flight 6 - A scenario similar to that of Flight 5 is used to transfer the LAB module (LAB1) from the Orbiter to the MRMS for attachment to the existing modules; however, the MRMS is required to operate on the opposite side of the lower keel from all previous flights (Figure 4.3.8.1-6).

g. Flight 7 - The transfer of the second laboratory module (LAB2) from the Orbiter bay to the MRMS (steps 1, 2 and 3) of Figure 4.3.8.1-7 is identical to that of Flight 6. The MRMS then positions LAB2 (step 4), completing the racetrack configuration (Figure 4.3.8.1-7).



Firgure 4.3.8.1-5 - FLIGHT 5 PLACEMENT OF LOGISTICS MODULE REQUIRING BOTH THE RMS AND MRMS



Firgure 4.3.6.1-3 - FLIGHT 6 PLACEMENT OF LAB1 USING BOTH THE RMS AND MRMS.

ORIGINAL PAGE IS
OF POOR QUALITY

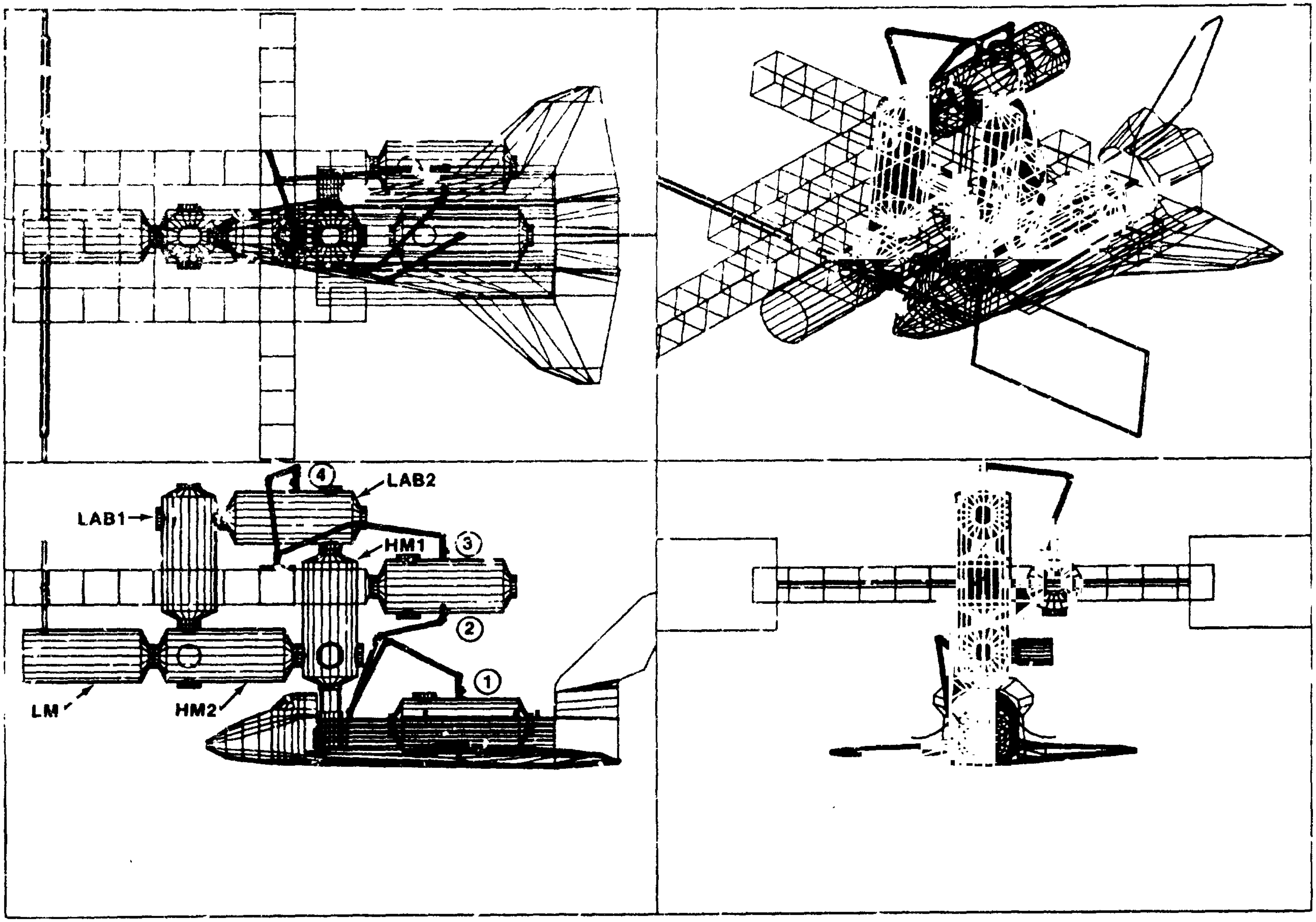


Figure 4.3.8.1-7 - FLIGHT 7 PLACEMENT OF LAB2 USING BOTH THE RMS AND MRMS

4.3.8.1.2 Station maintenance/payload placement

Figure 4.3.8.1-8 shows representative locations and sizes of the externally mounted payloads and servicing structures. The Space Station structure is removed from this figure for clarity. In Figure 4.3.8.1-9, the Space Station structure has been inserted. Also shown in this figure is the reach envelope of the MRMS as defined by the location of the wrist pitch joint. The envelopes are shown for various locations along the truss structure where MRMS assistance in servicing the station and externally mounted payloads could be required. With the MRMS able to travel along both the front and back sides of both the upper and lower keels and the transverse boom, it is apparent that all areas of concern can be reached with an MRMS having the same dimensions as the current RMS.

4.3.8.2 Orbital Flight Operations

The Space Station System consists of a manned orbiting Space Station, space platforms, free-flying satellites, orbital transfer vehicles, and orbital maneuvering vehicles that interact with the Space Station in orbit. In order to utilize all of these elements efficiently, many detached operations will interact directly with the station. Missions involving two or more spacecraft will frequently require rendezvous and/or Proximity Operations (PROX OPS). Rendezvous is the act of bringing two vehicles, a target and a chaser, together. The amount of time required to rendezvous can range from hours to days. This time may influence the launch/departure time. In general, the rendezvous maneuvers will leave the chaser vehicle at a specified offset point ahead of the target if the chaser is manned or behind the target if the chaser is unmanned. At this point, the rendezvous is completed. The chaser may then stationkeep prior to initiation of PROX OPS involved with transition to the target and berthing. PROX OPS is concerned with that portion of a mission when two vehicles are within approximately 1000 feet of each other and one is performing maneuvers relative to the other. These maneuvers may or may not be performed manually by the active vehicle controller; however, the results of these maneuvers will be monitored closely through direct visual or electronic means. PROX OPS begins at that final state resulting from the rendezvous activities (normally on the target's velocity vector). PROX OPS maneuvers may include transitions to various offset points, stationkeeping, approaches, flyarounds, and the separation to a "stand-off" position. Other activities such as departures, monitoring and servicing co-orbiting satellites (satellites traveling in the same orbit as the Space Station) are also included.

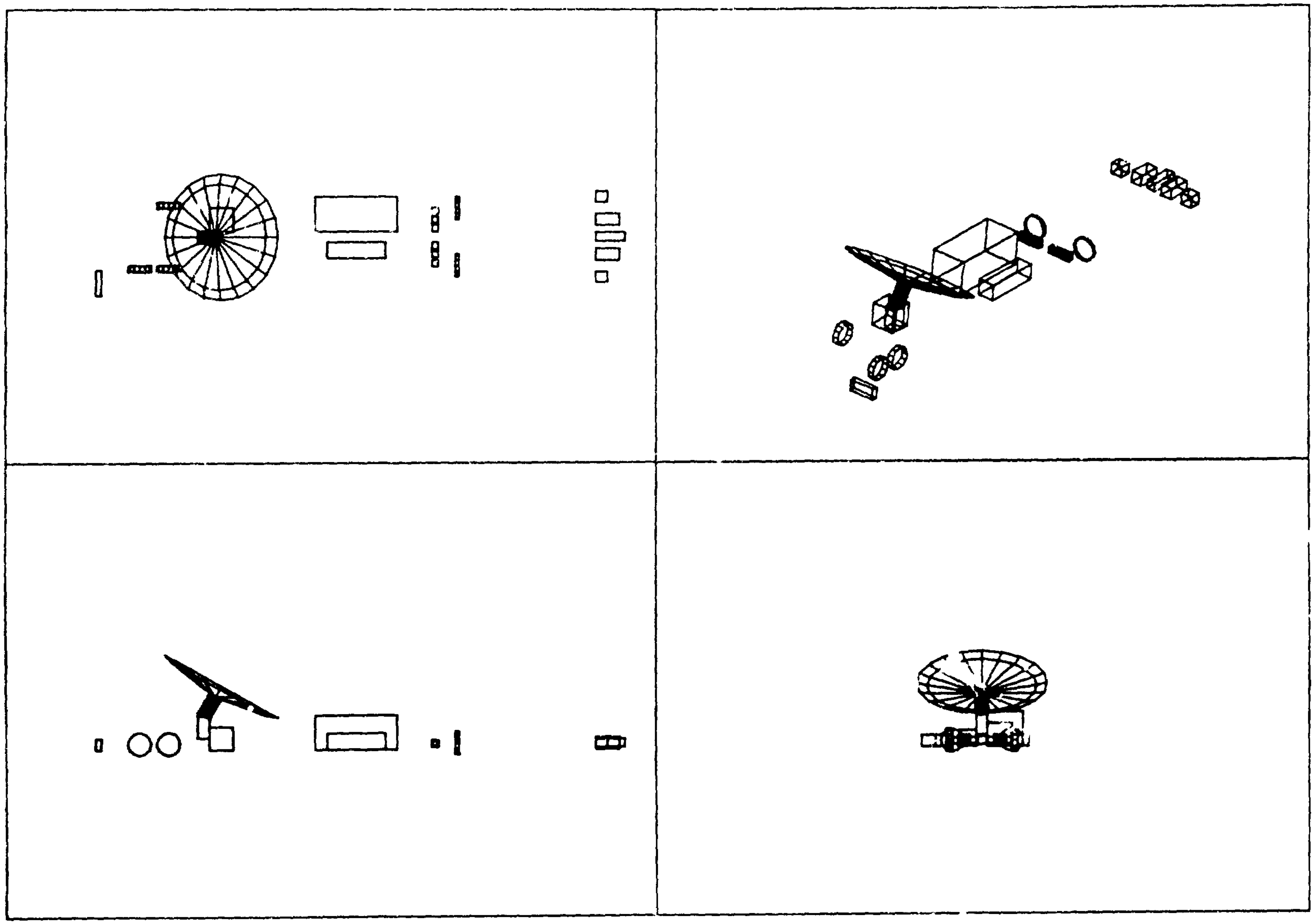
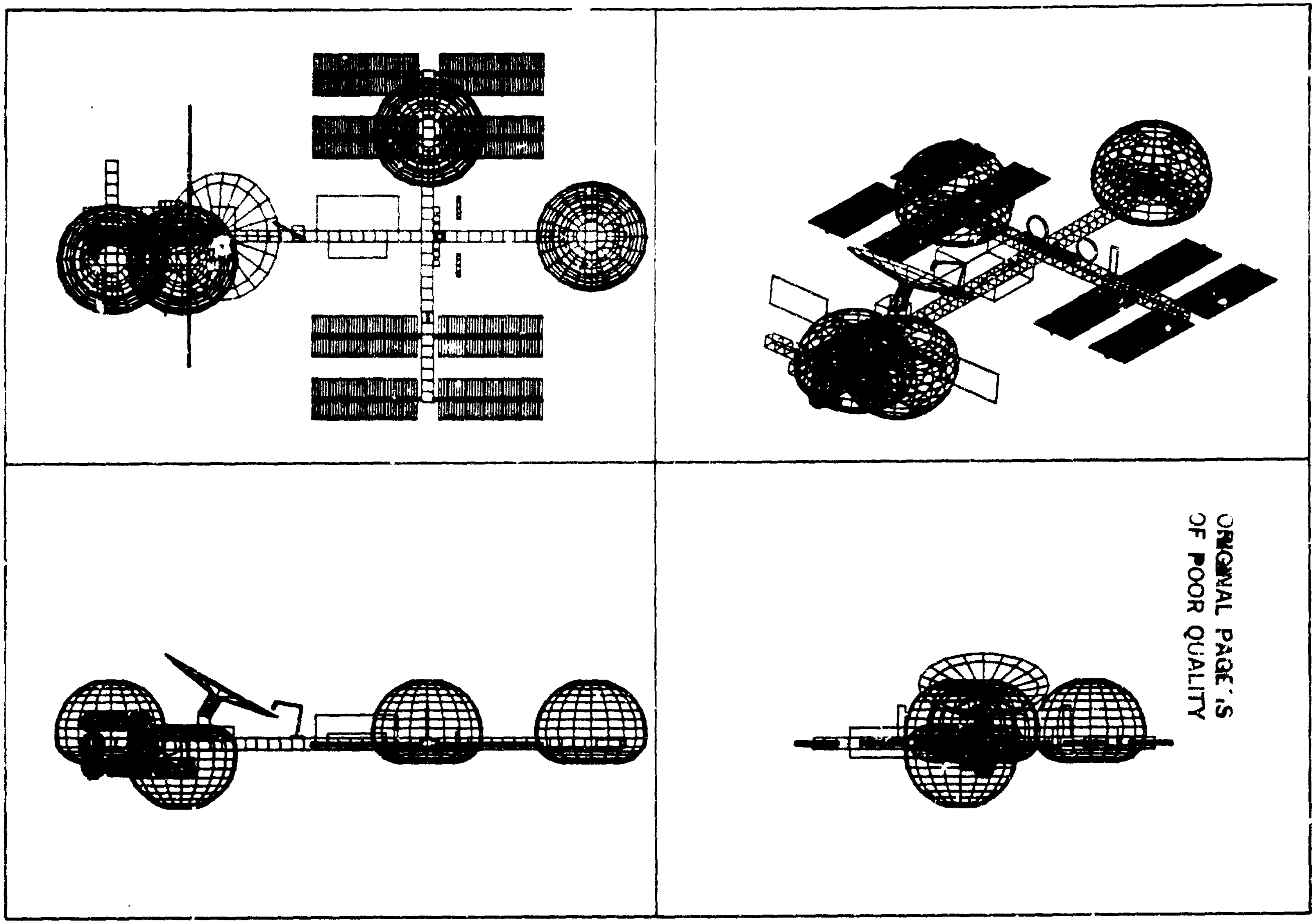


Figure 4.3.8.1-8 - EXTERNALLY MOU^'TED PAYLCADS



Firgure 4.3.8.1-9 - KRMS REACH CAPABILITY FOR SERVICING EXTERNALLY MOUNTED PAYLOADS

4.3.8.2.1 Operational Control Zones for the Space Station

The concept of Operational Control Zones has been developed in an effort to provide an operational foundation for the Space Station System. Any concept for multiple detached operations must address the following operational requirements. First, the concept must allow for standardized flight planning and operations. This means that the routine need for long lead-time preparation prior to execution of multiple detached operations must be avoided. Second, the concept must permit standardized crew planning and operations. This will simplify crew training and day-to-day crew activity planning. Third, the concept should allow early definition of requirements. These include Communication and Telemetry requirements, as well as Command/Control/Tracking requirements. Knowing the requirements early will help to minimize over-design. Finally, any plan for conducting multiple detached operations should provide for collision avoidance between spacecraft and hold Space Station disturbance and contamination from thruster firings to a reasonable level.

In order to satisfy these operational requirements, the current concept earmarks nine specific operational regions or zones in the orbital area of the Space Station. Each of these zones is dedicated to a specific type(s) of detached operation(s).

The following operational considerations and guidelines were taken into account in defining the location and size of the Operational Control Zones:

a. The amount of work on the Space Station will be held at a minimum by having the ground responsible for all flight planning, tracking, and control operations where a high level of crew involvement would otherwise be required for final rendezvous and PROX OPS phases.

b. For all flight operations in which a vehicle is approaching, stationkeeping, performing flyarounds, and/or berthing with the Space Station, a crew member will be required to actively monitor and if necessary control the flight operations. Thus, if the active vehicle is unmanned, the capability to actively take over Command/Control must exist. For most spaceflight operations performed to date, this control usually takes over at a separation distance of approximately 37 km (20 nmi) between the two vehicles.

c. Co-orbiting Satellites will be maintained in a desired orbit position relative to the Space Station and will be actively controlled from their ground control centers. The definition of this relative position will be a function of the payload's requirements on the Space Station. If it requires no Space Station support other than routine servicing, its relative control position can be allowed to be very large, possibly hundreds of kilometers along the Space Station's flight path. However, all co-orbiting satellites will be maintained approximately coplanar with the Space Station.

d. Non-co-orbiting satellites will be positioned to avoid any possible recontact with the Space Station and its co-orbiting satellites. The Command/Control of the non-co-orbiting satellites will be maintained by their ground control centers. No attempt will be made to control the relative position of these vehicles, except in instances of collision avoidance. However, some control may be exercised upon the satellite's relative altitude in order to control the magnitude of the relative plane at the time of satellite servicing, especially if the satellite is to be serviced from the Space Station.

e. Payloads returning from geosynchronous or other high energy orbits, and requiring Space Station support, will be initially targeted to a parking orbit above the Space Station. This will allow for relative phasing and planar adjustments between the returning vehicle's orbit and that of the Space Station. This parking orbit will also act as a hand-over interface for flight planning activities, allowing the descent from the high energy orbit to be decoupled from those activities occurring in low earth orbit such as final rendezvous with or servicing by the Space Station.

A detailed discussion of each of these operational zones is outlined below. Refer to Figures 4.3.8.2.1-1 through -8.

Zone 1: Proximity operations zone

Zone 1 consists of the region of space enclosed by a 1-km (approximately 3280 ft.) sphere centered on the Space Station. Within this zone, all nominal proximity operations including stationkeeping, flyarounds, final approaches and berthing will take place. In addition, nominal MMU/EVA activity will be performed within this zone.

Zone 2: Control zone

This concept assumes that the Space Station will have the capability to acquire active Command/Control/Track of any unmanned vehicle that enters this zone.

This zone contains a "slab-like" volume of space that is curved and centered about the Space Station orbit track. (Note that this shape is common to all zones centered around the Station's orbit.) In the horizontal direction (curvilinear "x"), Zone 2 begins 37 km (20 nmi) behind the Space Station and ends 37 km ahead of it. In the vertical direction (curvilinear "z"), the zone starts 37 km below the Space Station's orbit and ends 37 km above it. Finally, Zone 2 extends $\pm$ 9 km (5 nmi) out of the Space Station's orbital plane (curvilinear "y").

In order to support its Command/Control responsibility, the Space Station is assumed to have several additional capabilities which will be used only when an unmanned vehicle enters Zone 2. The functions are:

a. Monitoring: The Space Station will be capable of actively monitoring the "system health" portion of any unmanned vehicle's telemetry. This will support any safety questions such as system malfunctions that must be considered while maneuvering the active vehicle within near proximity of the Space Station.

b. Tracking: The Space Station will be capable of tracking/commanding/controlling all unmanned vehicles within the Control Zone. Limiting Space Station traffic control authority to Zone 2 provides two obvious advantages. The first is that it reduces and helps to define antenna requirements. The second advantage is the freeing of valuable crew time that is better spent on things other than routine monitoring and flight planning activities.

Zone 3: Departure zone

Most nominal departures from the Space Station will take place within this zone, after initial deployment and separation maneuvers are performed in Zone 1. Zone 3 is centered upon the Space Station's flight path and begins at the Space Station, extending forward to approximately 185 km (100 nmi). In the vertical dimension it begins approximately 37 km (20 nmi) below the Station's orbit and ends approximately 37 km above the Space Station's orbit track. The zone extends approximately $\pm$ 9 km (5 nmi) out of the Space Station's orbit plane.

Zone 4: Rendezvous zone

All nominal rendezvous with the Space Station will be targeted to arrive within this zone. Zone 4 is centered upon the Space Station's orbit track, and extends rearward from the Space Station to approximately 185 km (100 nmi). It is approximately $\pm$ 37 km (20 nmi) in the vertical dimension, and approximately $\pm$ 9 km (5 nmi) in the out-of-plane dimension. This zone's location and size have been designed to be consistent with the standard Stable Orbit Rendezvous Technique In Stable Orbit Rendezvous, the chaser arrives at an offset point some distance behind the target and performs its closing maneuvers from this point.

Zone 5 & 6: Co-orbiting zones

These two zones are dedicated to co-orbiting satellite operations. Zone 5 is the leading co-orbiting satellite zone. It begins approximately 185 km (100 nmi) ahead of the Space Station. It is centered around the Space Station's flight path, and extends forward to the opposite side of the orbit; i.e., 180° away from the Space Station (approximately

21,609 km or 11,668 nmi for an assumed Space Station orbit of 500 km (270 n. mi.). Zone 6 is the Trailing Coorbiting Satellite Zone. It begins approximately 185 km (100 n. mi.) behind the Space Station and follows the Space Station orbit track until it contacts Zone 5. Hence, the co-orbiting satellite zones are continuous around the Earth.

Zone 7 & 8: Non-co-orbiting zones

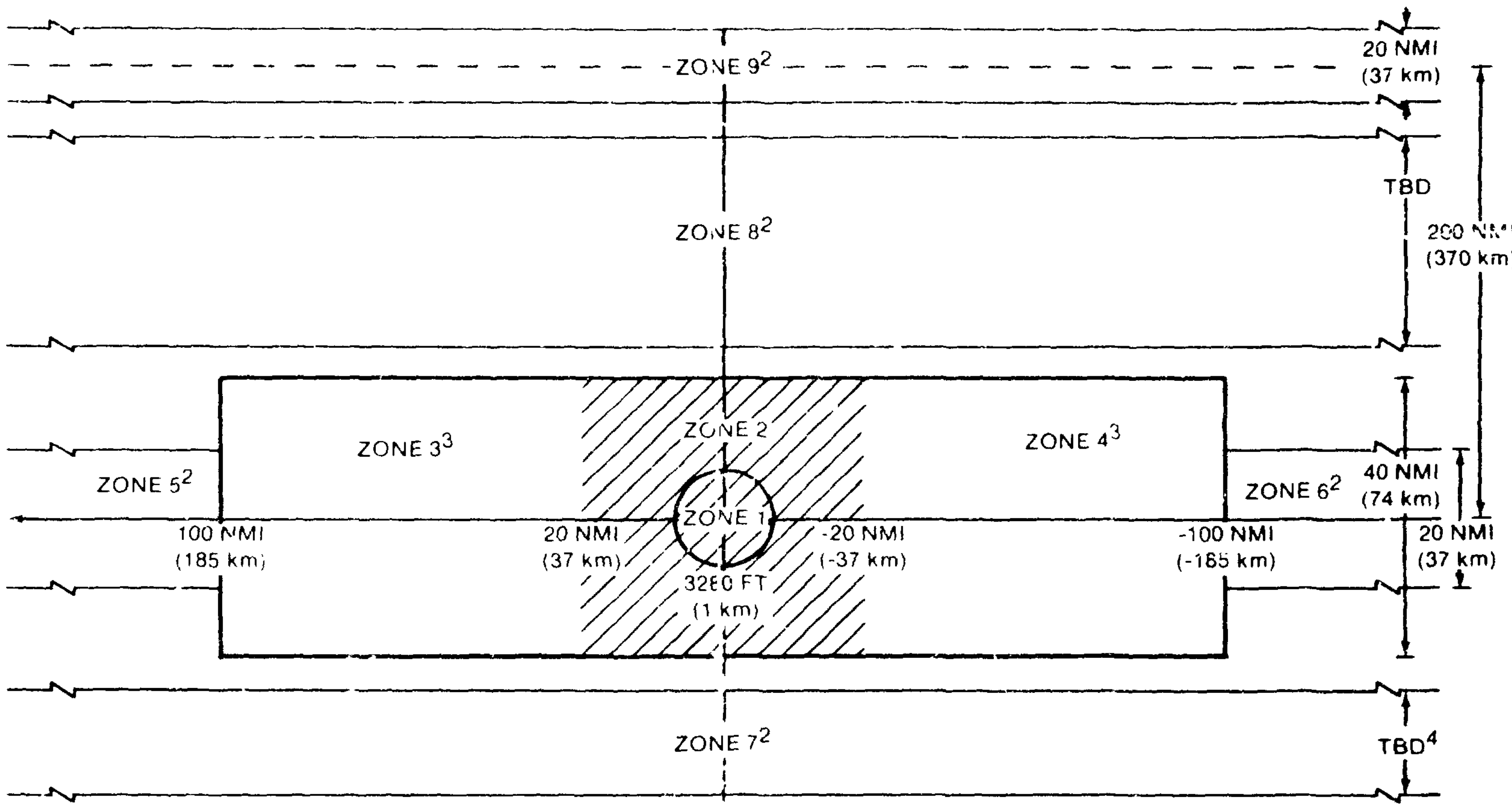
These zones contain the non-co-orbiting satellites. Both Zones 7 and 8 are concentric spherical shells centered about the Earth. Zone 7 begins approximately 37 km (20 nmi) below the Space Station. It extends downward to approximately 185 km (100 nmi) altitude above the Earth. Zone 8 begins approximately 37 km (20 nmi) above the Space Station. It extends to an altitude of approximately 352 km (190 nmi) above the Space Station's orbit.

Zone 9: Parking orbit zone

This zone surrounds the parking orbit used by orbital transfer vehicles (OTV's) returning from geosynchronous or other high energy orbits. It may also be used by spacecraft returning from lunar or planetary missions. Zone 9 is a spherical shell, centered upon the Earth, that encloses the Space Station and Zones 1-8. This shell begins approximately 352 km (190 nmi) above the Space Station orbit track, and extends upward to an altitude approximately 389 km (210 nmi) above the Space Station.

4.3.8.2.2 Rendezvous operations

Future orbital missions involving two or more spacecraft will frequently require rendezvous. A simplistic definition of rendezvous is the act of bringing two vehicles, a target and a chaser, together. The target vehicle is assumed to be in a known, stable orbit. The chaser performs a series of predetermined maneuvers designed to ensure that the rendezvous with the target will occur at a specified position in the target's orbit and within a prescribed amount of time. The magnitude of these maneuvers is dependent upon many factors, but it may be significantly influenced by the time of launch (or departure, in the case of a Space Station based "launch"). The target position at rendezvous completion depends mainly on lighting requirements, while the amount of time that is prescribed for the rendezvous can range from hours to days. The prescribed rendezvous time may influence the launch/departure time, which in turn influences the lighting in the final phases of the mission. In general, the rendezvous maneuvers will leave the chaser vehicle at a specified offset point (typically 1000 ft) ahead of the target if the chaser is manned or behind the target if the chaser is unmanned. At this point, the rendezvous completed. A stable orbit rendezvous is assumed which takes the chaser to a close offset range (between 5 nmi and 20 nmi). Manned vehicles will continue to approximately 100 feet in a



ZONE ASSIGNMENTS
1 PROXIMITY OPERATIONS ZONE
2 CONTROL ZONE
3 DEPARTURE ACTIVITY ZONE
4 RENDEZVOUS ACTIVITY ZONE
5 LEADING COORBITING SATELLITE ZONE
6 TRAILING COORBITING SATELLITE ZONE
7 LOWER NON-COORBITING SATELLITE ZONE
8 UPPER NON-COORBITING SATELLITE ZONE
9 PARKING ORBIT ZONE

NOTES.
1 THIS DRAWING IS CURVILLINEAR AND IS NOT DRAWN TO SCALE
2 THIS ZONE IS CONTINUOUS ABOUT THE EARTH
3 THIS ZONE BEGINS AT THE SPACE STATION
4 THE MINIMUM OPERATING ALTITUDE FOR THIS ZONE IS ~ **100 NMI (~185 km)**
5 ZONES 2,3,4,5, & 6 EXTEND ~ ±9 km(±5 NMI) OUT-OF-PLANE
6 ZONES 7,8 AND 9 ARE SPHERICAL SHELLS CENTERED ABOUT THE EARTH

Figure 4.3.8.2.1-1 - OPERATIONAL CONTROL ZONES.[1]

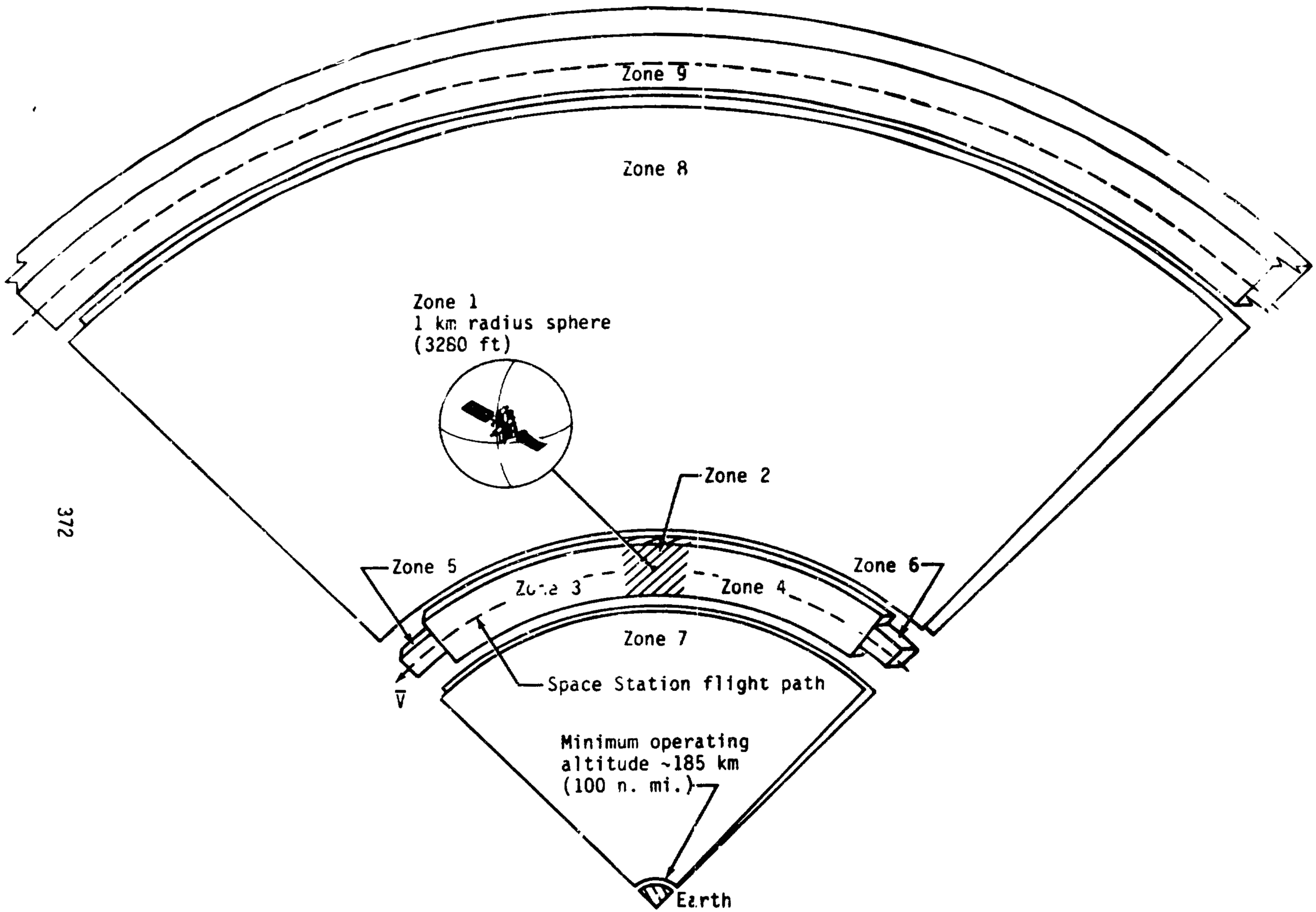


Figure 4.3.8.2.1-2 - SCALE DRAWING OF OPERATIONAL CONTROL ZONES IN 3 DIMENSIONS

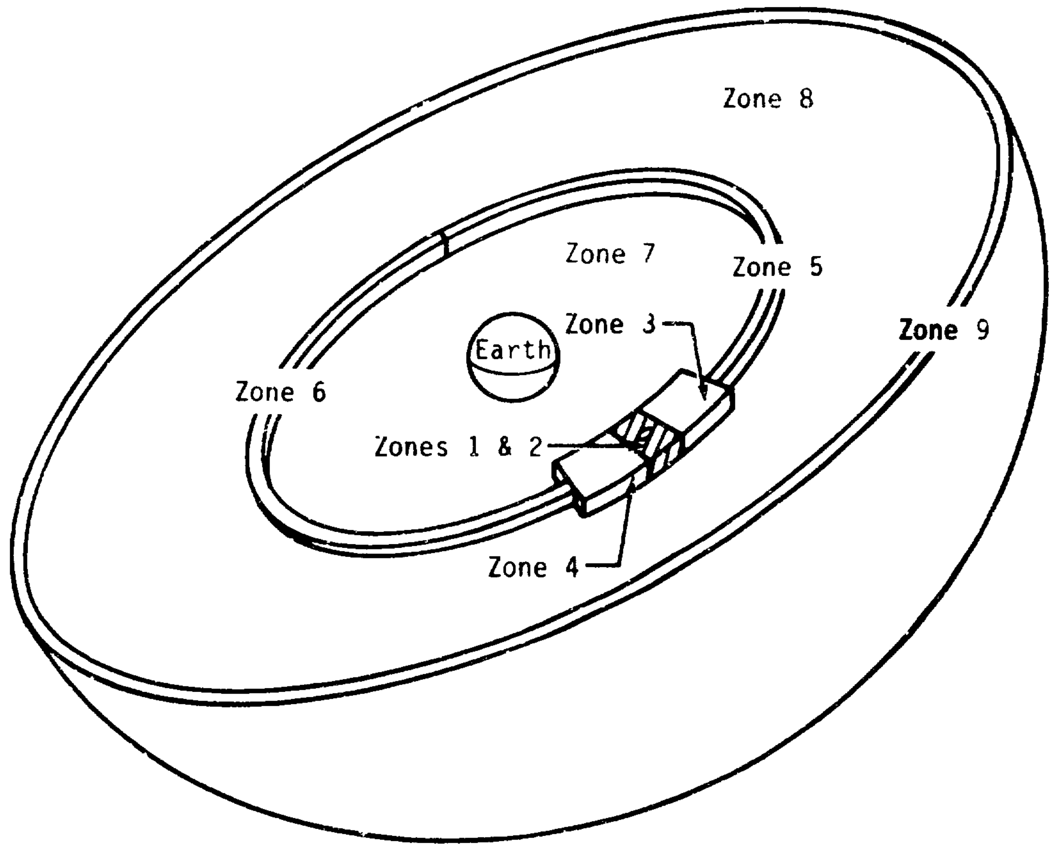


Figure 4.3.8.2.1-3 - CUTAWAY VIEW OF OPERATIONAL CONTROL ZONES (HEMISPHERICAL CUTAWAY)

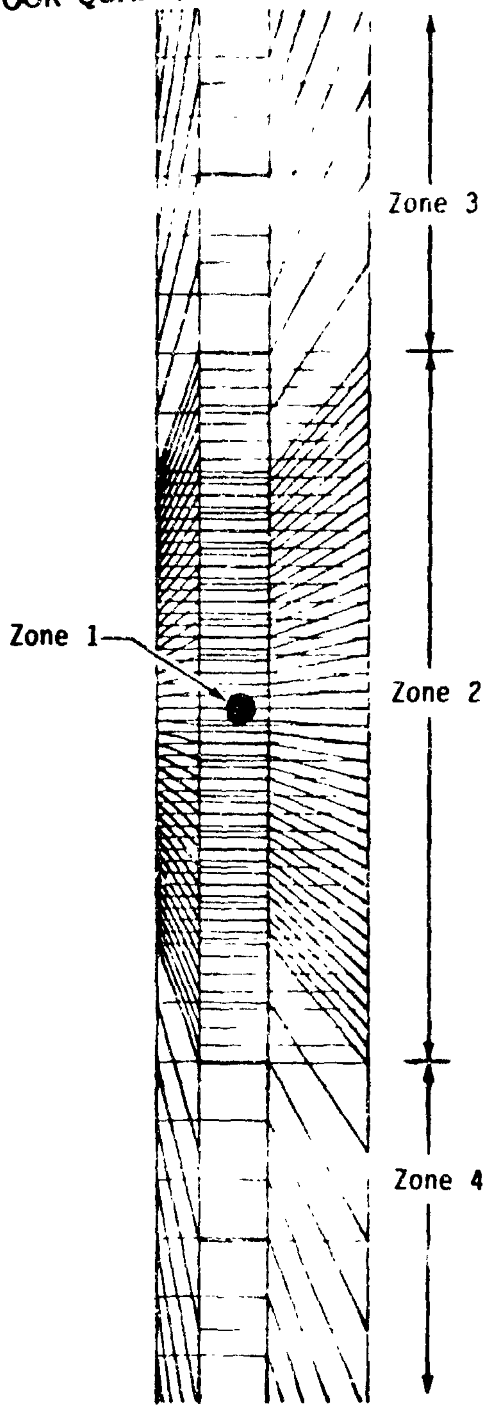


Figure 4.3.8.2.1-4 - OPERATIONAL CONTROL ZONES: NEARFIELD VIEW, SLIGHTLY OUT-OF-PLANE

ORIGINAL PAGE IS
OF POOR QUALITY

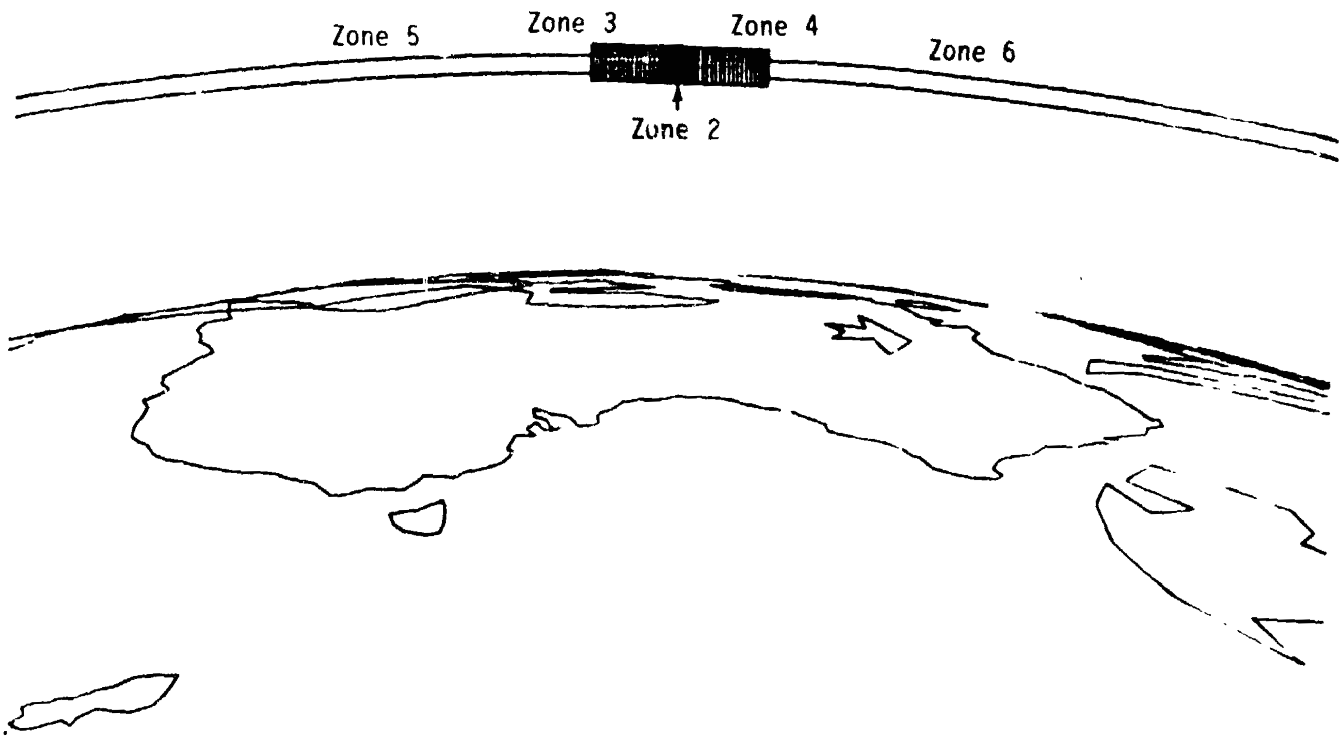


Figure 4.3.8.2.1-5 - OPERATIONAL CONTROL ZONES: NEARFIELD VIEW, OUT-OF-PLANE (DRAWING TO SCALE)

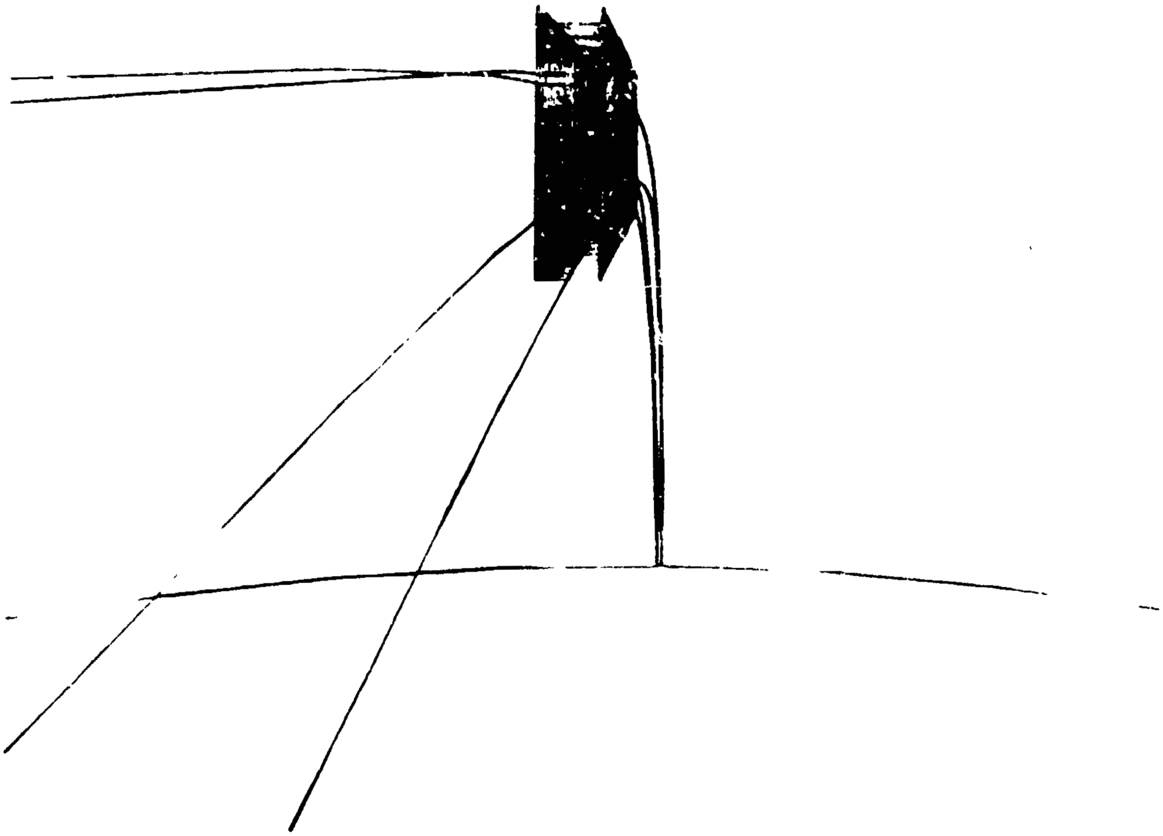


Figure 4.3.8.2.1-6 - OPERATIONAL CONTROL ZONES: NEARFIELD VIEW; EYEPOINT SLIGHTLY OUT-OF-PLANE (VIEW A) (DRAWING TO SCALE)

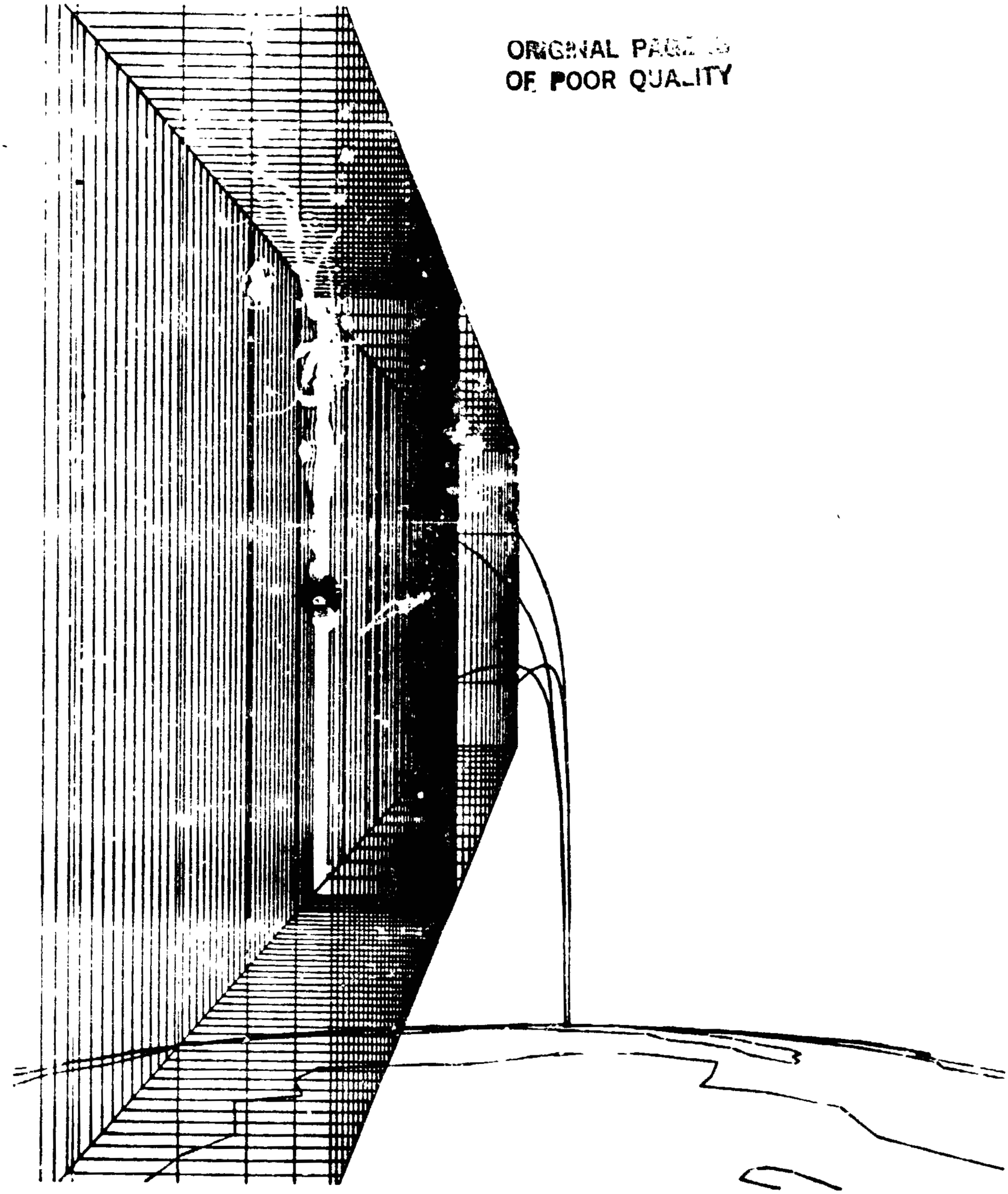


Figure 4.3.8.2.1-7 - OPERATIONAL CONTROL ZONES: VIEW OF ZONE 1 FROM WITHIN ZONE 2.

ORIGINAL PAGE IS
OF POOR QUALITY

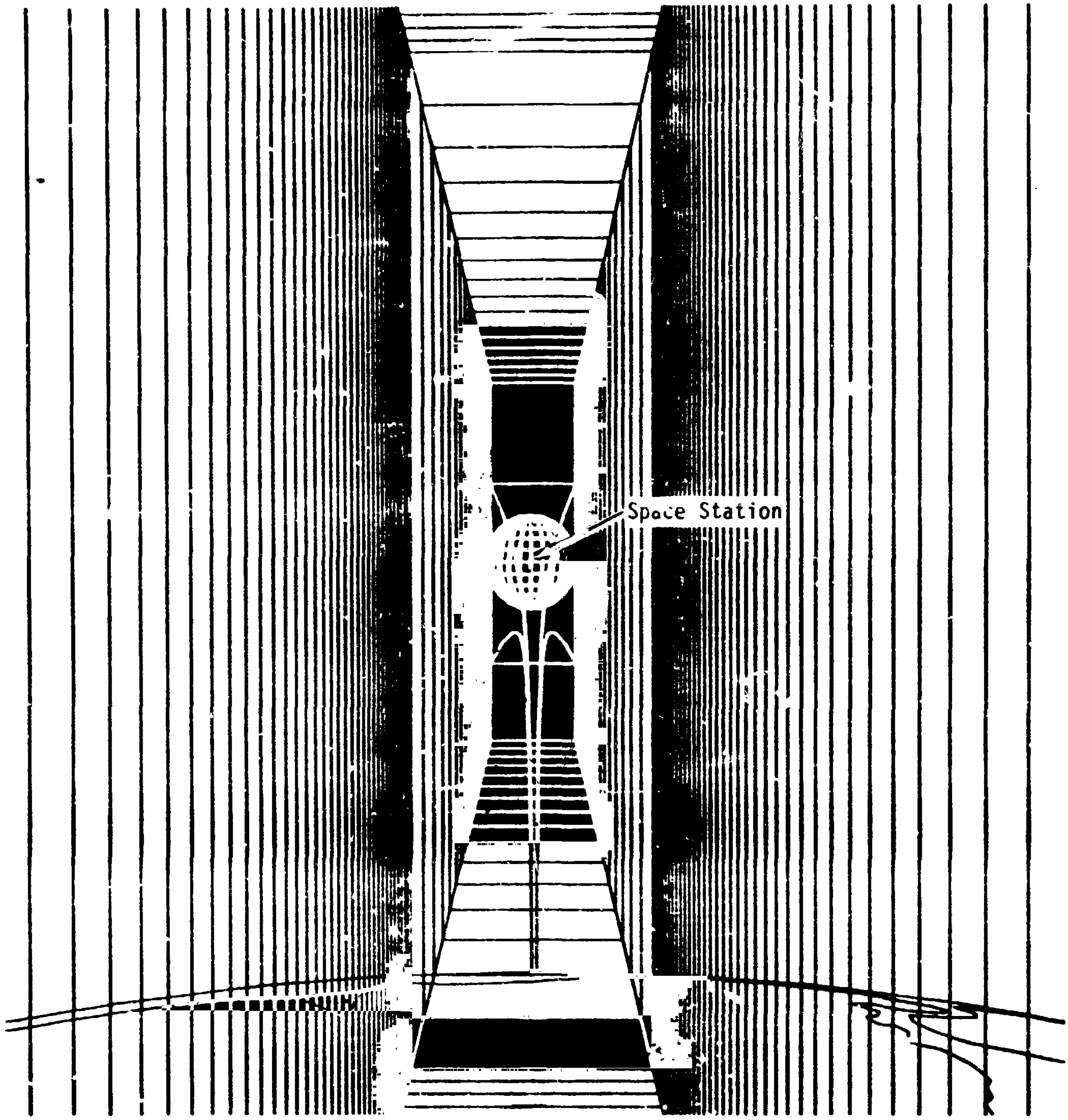


Figure 4.3.8.2.1-8 - OPERATIONAL CONTROL ZONES: APPROACHING ZONE 1 FROM AHEAD OF THE SPACE STATION

man-in-the-loop mode while unmanned free-flyers will continue with an automated mode. All planning, targeting, and navigation for the longer ranges of the rendezvous will be external to the Space Station and therefore will impose no requirement on the Space Station. Once a vehicle is in the final rendezvous phase the Space Station will monitor all vehicles. Monitoring includes communication, tracking, and back-up targeting for manned vehicles and communication, tracking, targeting, and back-up take over of unmanned vehicles. These activities imply requirements on the Space Station for various hardware and software systems.

4.3.8.2.3 Proximity operations

In order to develop an efficient set of PROX OPS procedures, numerous operational considerations must be addressed regarding the final position achieved by the active vehicle and the PROX OPS techniques used to maneuver this vehicle to the desired position. A major concern is the effect of the RCS plume on the target. This plume impingement creates problems of contamination and disturbance (over-pressure) experienced by the target. In addition, the active vehicle's RCS propellant usage is critical. Propellant consumption influences the choice of braking maneuvers and stationkeeping techniques. Another concern for Orbiter operations is whether it is more efficient to compute required maneuvers using onboard targeting software or to have the crew execute the maneuvers manually using out-the-window data such as Crew Optical Alignment Sight (COAS) or Rendezvous Radar (RR) data. In any case, crew visibility and procedural simplicity have considerable impact on the selected maneuvers. A final major consideration is that of lighting during PROX OPS procedures.

4.3.8.2.3.1 Lighting impact on proximity operations

The PROX OPS procedures discussed here are extremely dependent on the lighting available for the crewman to view the activities taking place. The crewman who is monitoring activities or actively involved in performing the PROX OPS maneuvers requires that lighting (real or artificial) must be of the proper intensity as well as being projected in the correct direction. The primary source of natural light is the sun, and its position relative to a vehicle is a function of time. Hence, operations which impose a constraint on lighting direction also impose a restriction on the place in orbit when the activity can be performed or the attitude of the viewing vehicle. With this consideration, the direction of lighting must be in favor of the viewer having the final authority in performing the activities. Therefore, lighting direction is a function of the active vehicle being manned or unmanned.

If the active vehicle is manned, the lighting should be in a direction favorable to the manned active vehicle. For the situation of natural lighting, the lighting direction can be controlled by scheduling the activities at the appropriate time of day. It is for this reason that manned active approaches are initiated on the positive velocity vector (+ V-bar) of the target and designed to take place as soon after orbital sunrise as possible and be completed before orbital sunset. Conversely, unmanned approaches to a manned Space Station will be initiated from behind the Space Station; i.e., on its negative velocity vector (-V-bar). This provides the manned Space Station with the proper lighting needed to monitor the vehicle's approach.

4.3.8.2.3.2 Approach and berthing to the Space Station

A typical return mission for an active vehicle which has already executed a successful return rendezvous profile placing it at an 8 nmi offset point behind the Station begins with an auto-return maneuver sequence that brings the vehicle to a 1000-foot offset point. This point will be behind the Station for unmanned vehicles and ahead of the Station for manned vehicles, consistent with favorable lighting. The final PROX OPS returns are considered to begin at this point.

The final PROX OPS begin with a period of stationkeeping at 1000 feet. For returns to any port, a maneuver is performed to establish a closing rate until a position TBD feet from the Space Station is reached. At this range stationkeeping may be performed, if required. For V-bar ports, a final closing rate is established along the V-bar until the berthing range is reached. Returns along other axes involve a similar approach, if required, then utilize a constant range flyaround to align the approaching vehicle on the appropriate axis for final closure to the berthing port.

4.3.8.2.3.3 Separation from the Space Station

A typical separation scenario for a vehicle docked to the Space Station begins with small separation rate (approximately 0.2 fps using either a mechanical system or a vehicle maneuver) to begin the vehicle movement away from the port. The direction of the separation will be directly away from the berthing port to maximize an opening rate. A small intermediate maneuver may be required to ensure a favorable geometry between the separating vehicle and the station for the large separation burn to follow. After a coast of approximately 15 minutes the separation vehicle will perform the large separation maneuver of approximately 3 fps. This will ensure a separation range of approximately 10 nmi at orbital transfer maneuver ignition. This range is needed so that the station is outside the assumed explosion range of the separating vehicle. It is at this point that the separation has been completed.

4.3.8.2.3.4 Plume impingement/contamination

Keeping plume impingement on all orbiting elements at the lowest possible levels is a very important aspect of PROX OPS procedures both in Shuttle and Space Station operations. Currently, the Orbiter uses its RCS jets (primary and vernier) to control its attitude and execute various maneuvers. Subsequently, vehicles in the near vicinity of the Orbiter will experience both disturbance and contamination on their surfaces. Disturbance is in the form of external torques and forces caused by pressure differences on vehicle surfaces while contamination results depend on the toxic qualities of the impinging plume.

The plume flowfield contours of the Orbiter in the Low Z and Normal Z modes and for the Orbital Maneuvering System (OMS) are shown in Figures 4.3.8.2.3-1 and -2. These plume flowfield contours illustrate the expected amount of dynamic pressure to be experienced by a vehicle in the near vicinity of the Orbiter. Figures 4.3.8.2.3-3 and -4 illustrate the Orbiter plume flux contours or the expected amount of plume particles that will be incident on a vehicle's surfaces when it is near the Orbiter.

In order to demonstrate the effects of plume disturbances on the Space Station, a series of Orbiter 1-second jet firings have been made using the power tower configuration shown in Figure 4.3.8.2.3-5. Two PRCS jet firings modes were selected for this demonstration. The first mode was a Normal Z mode and the second was a Low Z jet mode where the + X-jets are fired to provide a +Z Orbiter body translation. These 1-second jet firings were made starting at berthing and at 50-foot intervals out to 200 feet (Figure 4.3.8.2.3-6) along the Space Station plus x-axis which would represent a final berthing approach direction or initial separation direction. The result of these 1-second jet firings can be seen in table 4.3.8.2.3-1. The resulting disturbances indicate that the Orbiter thruster configuration set up in the Digital Auto Pilot has a very pronounced effect on the resulting impingement forces and torques. More detailed analysis of the effects of the various vehicles' plume impingement on the Space Station during PROX OPS must be considered. These 1-second jet firings give an initial example of some of the expected disturbance forces from these operations. However, some PROX OPS activities may result in many jet firings lasting longer than one second. The duration and relative range of these firings will be the result of future PROX OPS activity requirements.

Figures 4.3.8.2.3-7 through -9 illustrate a typical Orbiter primary reaction control system (PRCS) thruster time history for a normal Z V-bar approach to the Power Tower Space Station configuration. The figures are plotted for the +X, +Y, and +Z thrusters respectively, with each set's total ontime being plotted as a function of the Orbiter's range from the station. The approach begins at approximately 850 feet with a 3 minute period of stationkeeping. The approach is initiated via two initial burns at about 810 feet; a 1.0 fps -Z and a .17 fps + X burn. This combination places the Orbiter on a closing trajectory toward the Station. When the range equals approximately 575 feet, the pilot initiates a series of braking maneuvers to control the approach velocity until all rates are nulled at approximately 50 feet. For this case, the plots clearly illustrate the increased firings necessary in the near proximity of the station in order to null the Orbiter's closing rate. It should be understood that these results simulate a typical Orbiter V-bar approach and by no means represent the total actual Space Station PROX OPS environment.

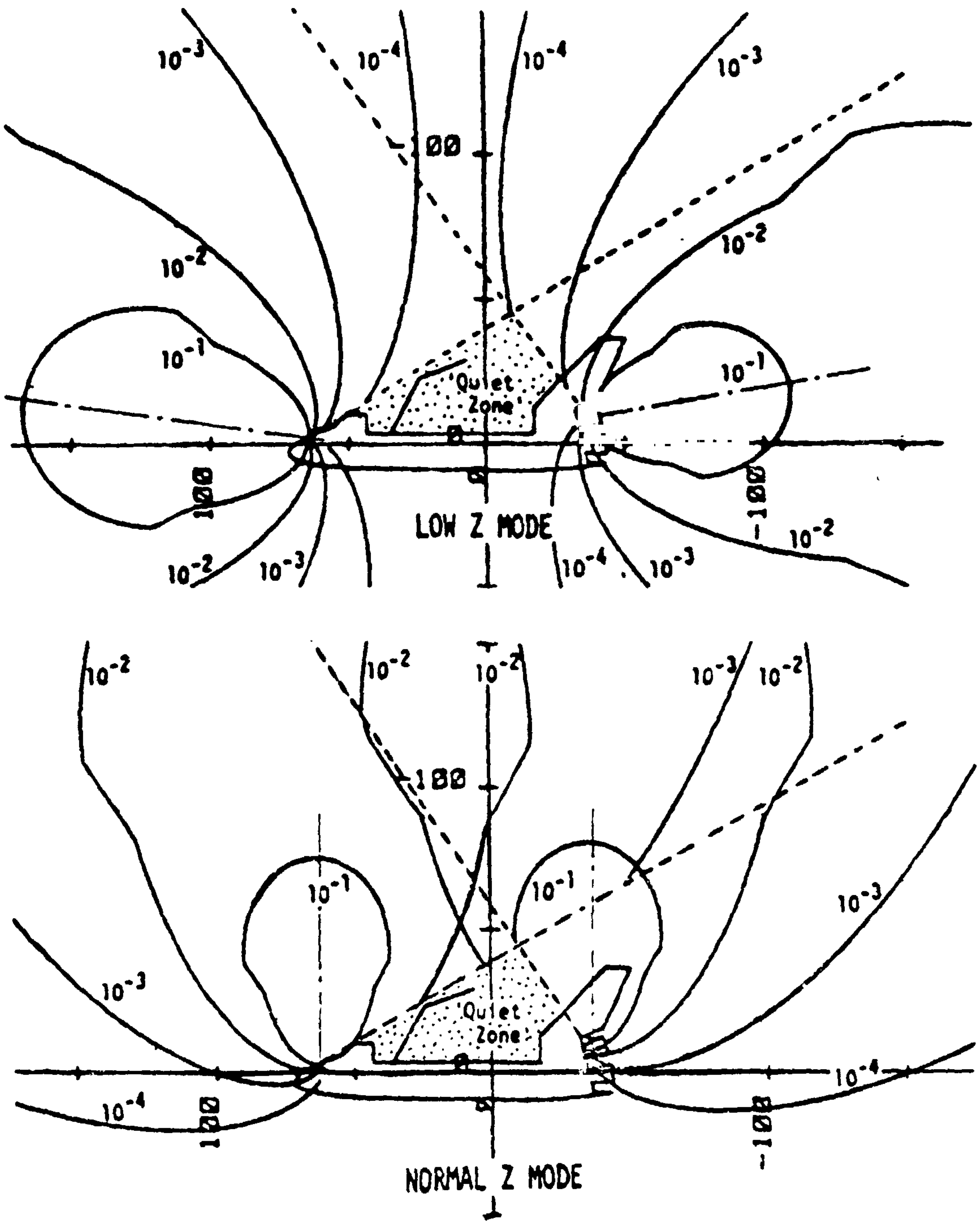


Figure 4.3.8.2.3-1 - ORBITER LOW Z AND NORMAL Z DYNAMIC PRESSURE PLUME CONTOURS.

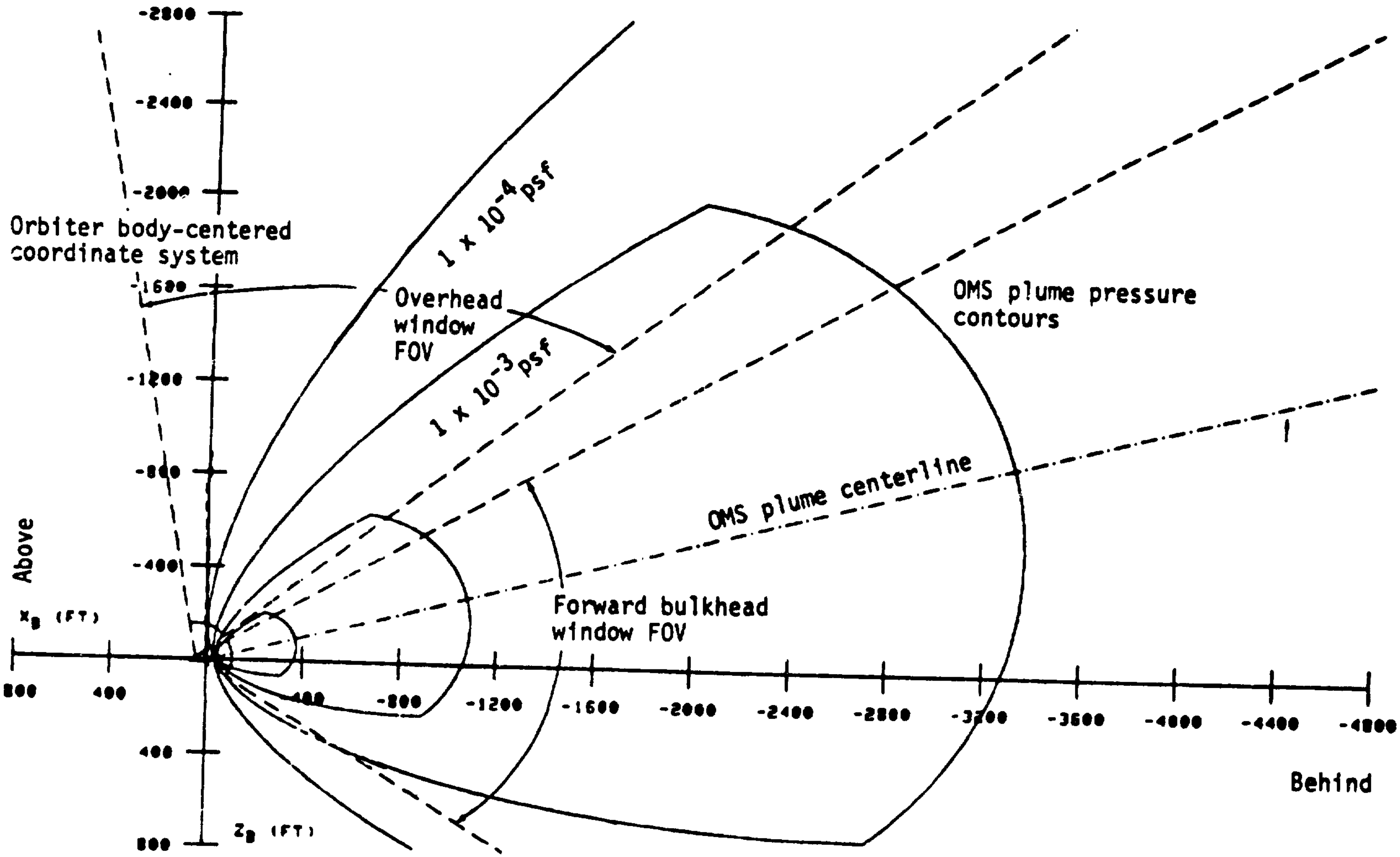


Figure 4.3.8.2.3-2 - ORBITER OMS ENGINE DYNAMIC PRESSURE PLUME CONTOURS.

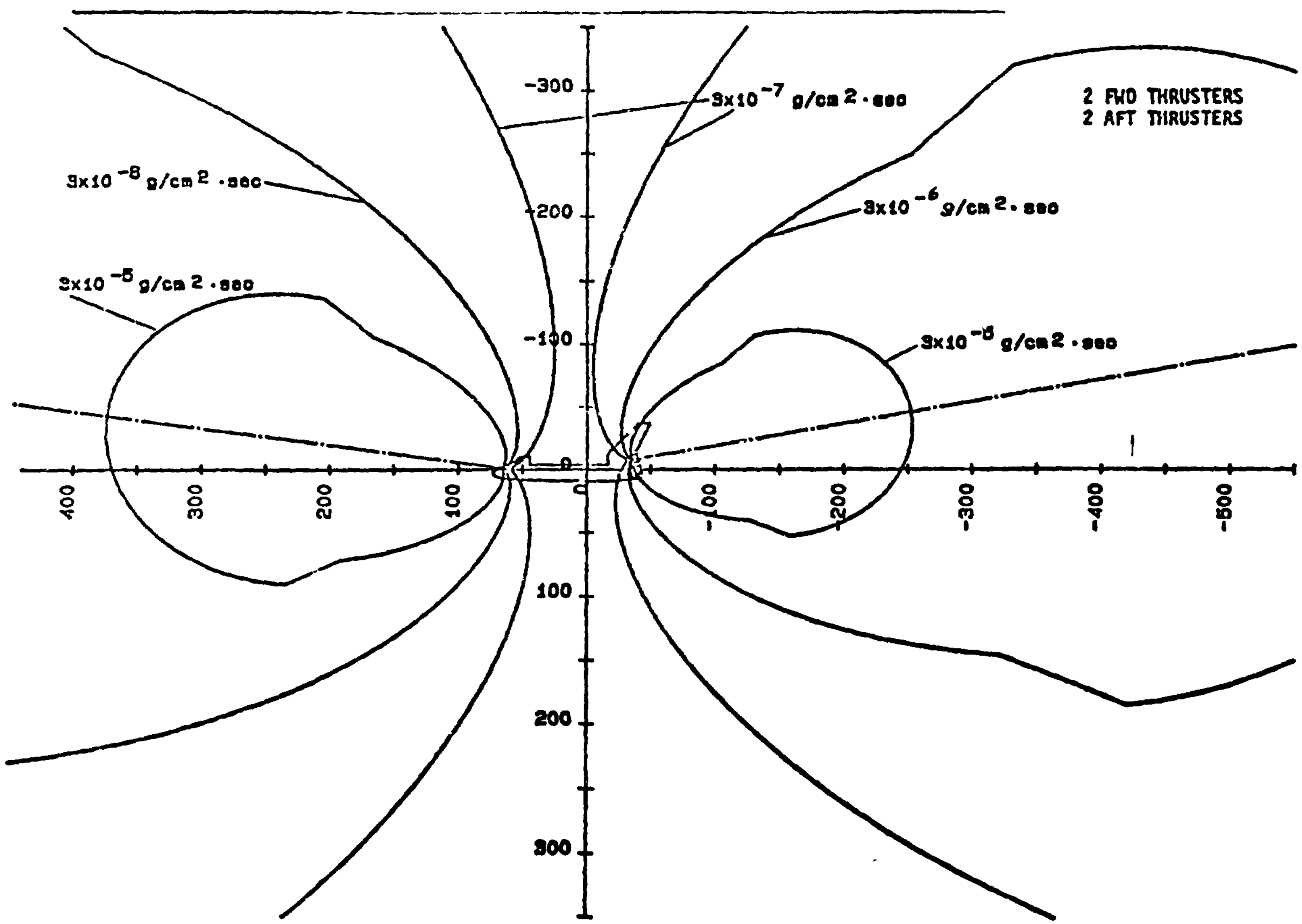


Figure 4.3.8.2.3-3 - ORBITER LOW Z PLUME FLUX CONTOURS.

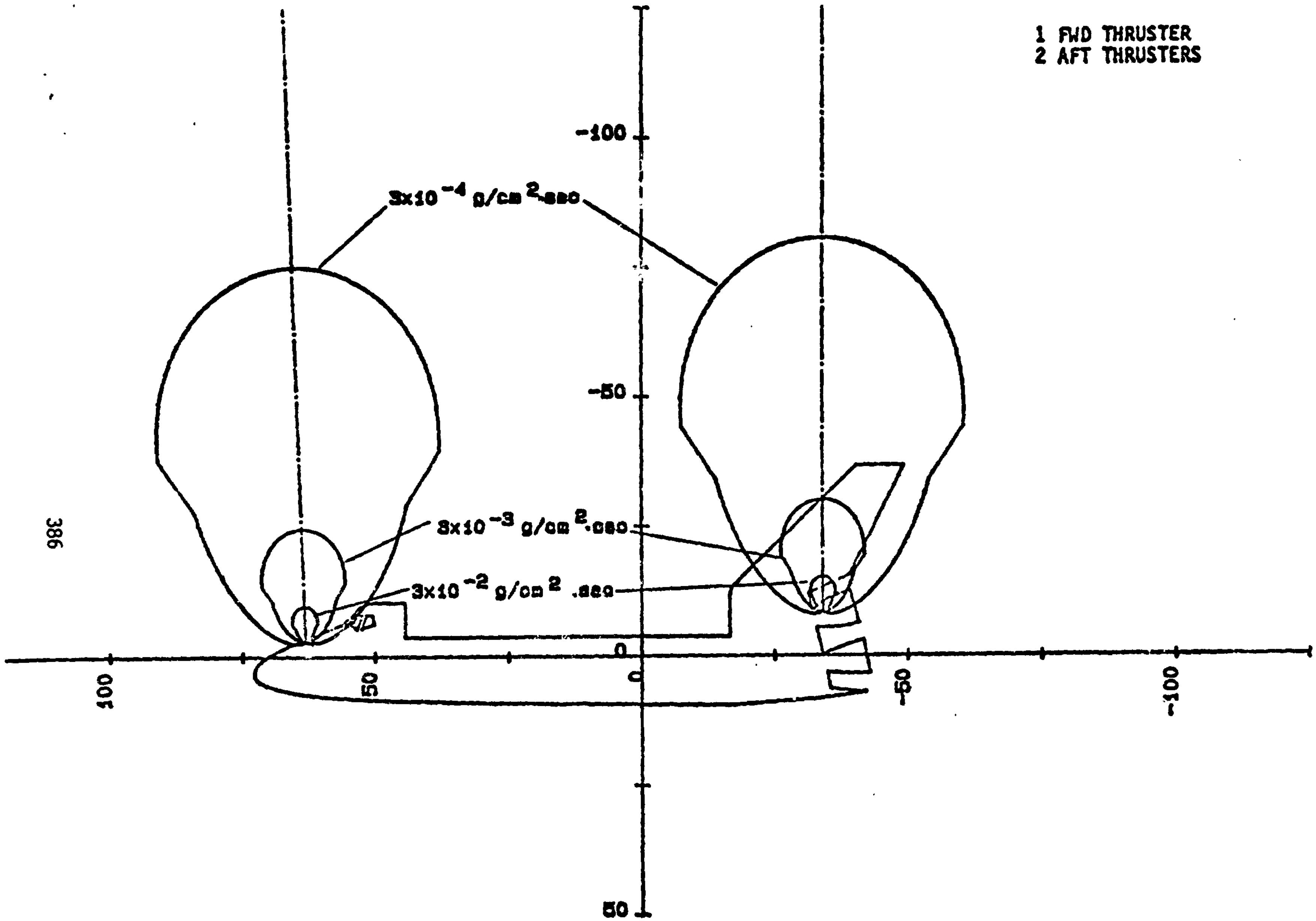


Figure 4.3.6.2.3-4 - ORBITER NORM Z PLUME FLUX CONTOURS.

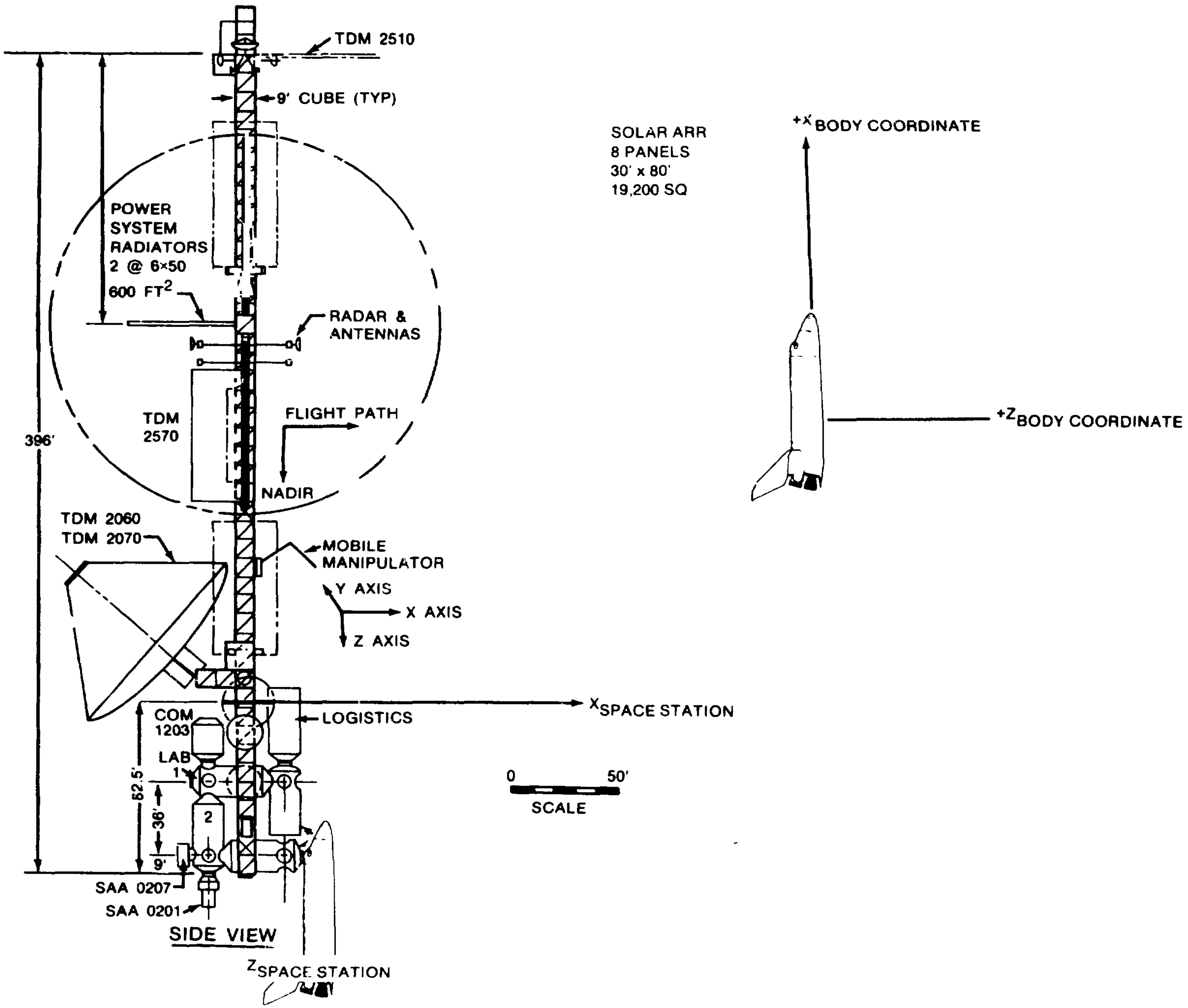


Figure 4.3.8.2.3-5.- POWER TOWER SPACE STATION CONFIGURATION

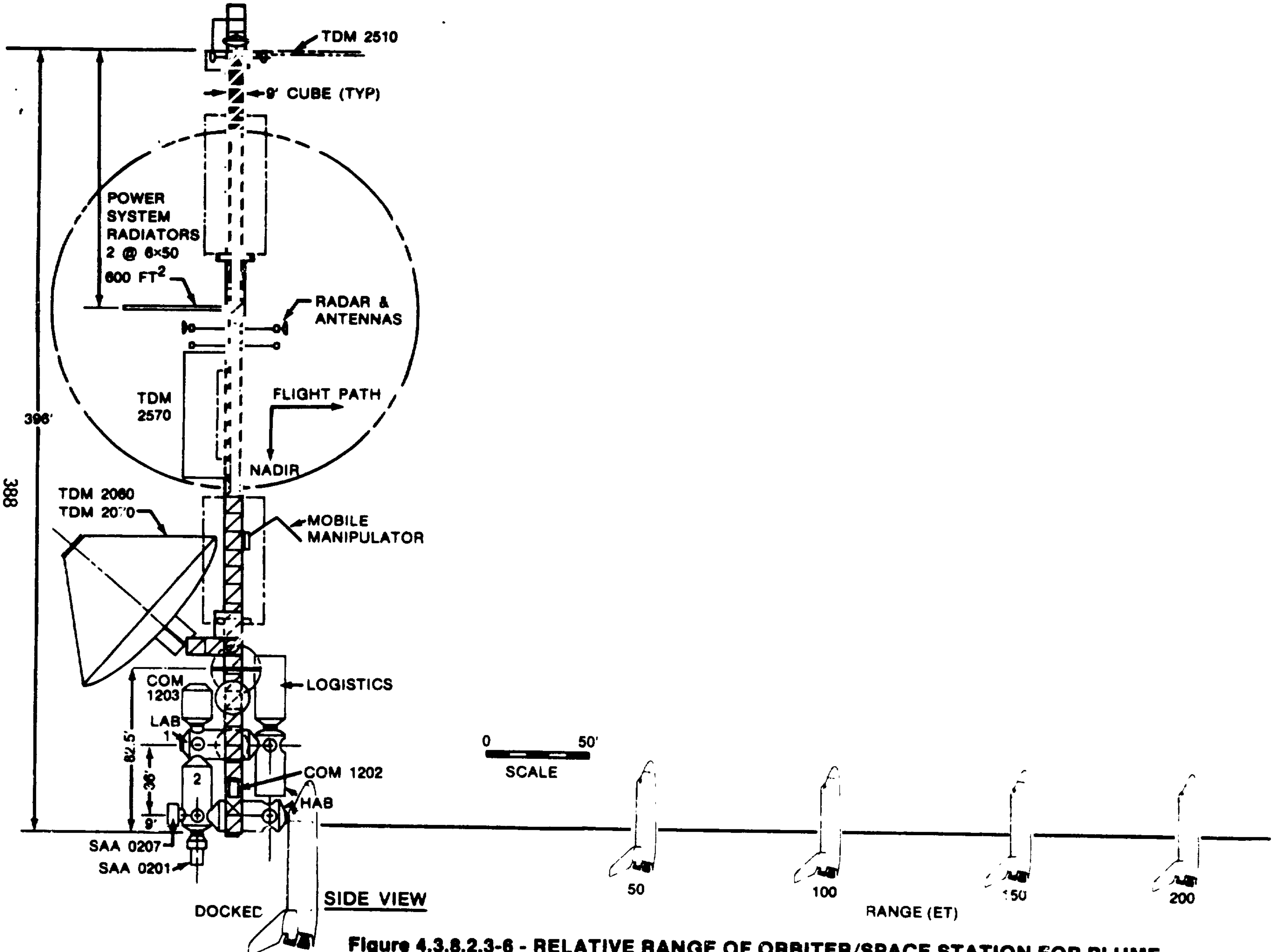


Figure 4.3.8.2.3-6 - RELATIVE RANGE OF ORBITER/SPACE STATION FOR PLUME IMPINGEMENT ANALYSIS

-Z 1 SEC FIRING

RANGE (FT)	TOTAL PLUME FORCE (lbf)	TRANSLATIONAL ΔV (FPS)	PLUME TORQUE IMPLUSE (ft-lbf-sec)			RESULTING DELTA OMEGA (deg/sec)		
			X	Y	Z	X	Y	Z
0 (DOCKED)	154.72	5.60E-4	14.20	-.391E + 3	-101.	1.90E-5	-3.70E-4	-8.20E-4
50	63.83	2.31E-4	10.40	.66E + 4	-38.5	1.11E-5	5.40E-3	-3.12E-4
100	50.5	1.83E-4	9.09	.74E + 4	-32.0	9.60E-6	5.98E-3	-2.60E-4
150	35.40	1.30E-4	8.16	.55E + 4	-18.5	7.83E-6	4.50E-3	-1.50E-4
200	22.80	8.30E-5	5.52	.36E + 4	-8.18	5.01E-6	2.93E-3	-7.15E-5

NOR Z (1 SEC FIRING)

RANGE (FT)	TOTAL PLUME FORCE (lbf)	TRANSLATIONAL ΔV (FPS)	PLUME TORQUE IMPLUSE X	Y	Z	RESULTING DELTA OMEGA X	Y	Z
0 (DOCKED)	741.01	2.68E-03	0.13E-1	-36300	-0.45E-2	1.00E-8	2.94E-2	-4.00E-8
50	230.22	8.34E-04	0.43E-3	-9000	0.97E-4	0.0	-7.30E-3	0.0
100	175.59	6.36E-04	0.15E-4	-4700	0.26E-3	0.0	-3.80E-3	0.0
150	167.32	6.06E-04	-0.10E-3	-2950	0.25E-3	0.0	-2.40E-3	0.0
200	156.14	5.65E-04	-0.14E-3	-551	0.24E-3	0.0	-4.50E-4	0.0

TABLE 4.3.8.2.3-1 ORBITER/SPACE STATION
ESTIMATED PLUME IMPINGEMENT DISTURBANCES

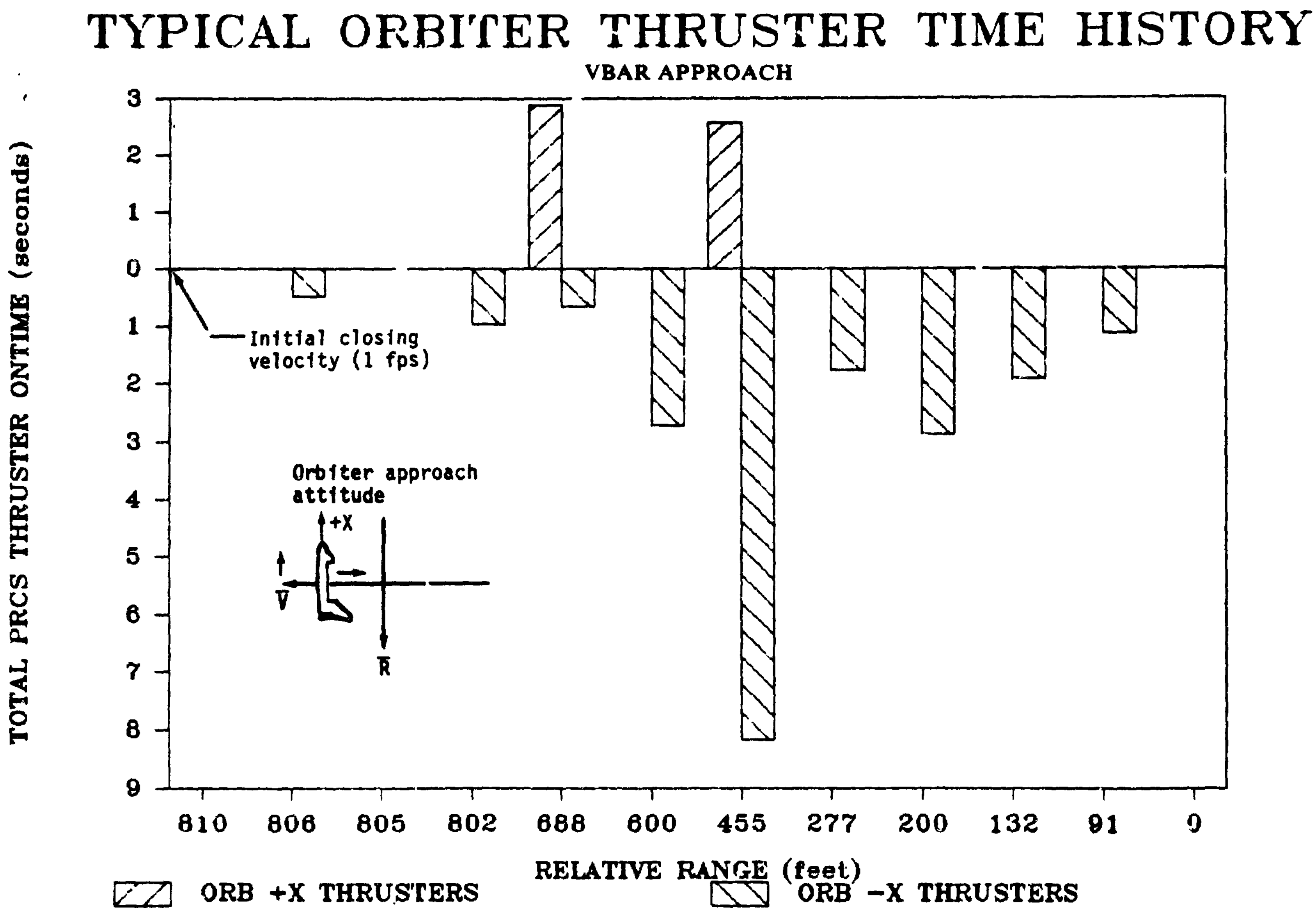


Figure 4.:.8.2.3-7 - Typical Orbiter ± X PRCS thruster time history for a V-bar approach.

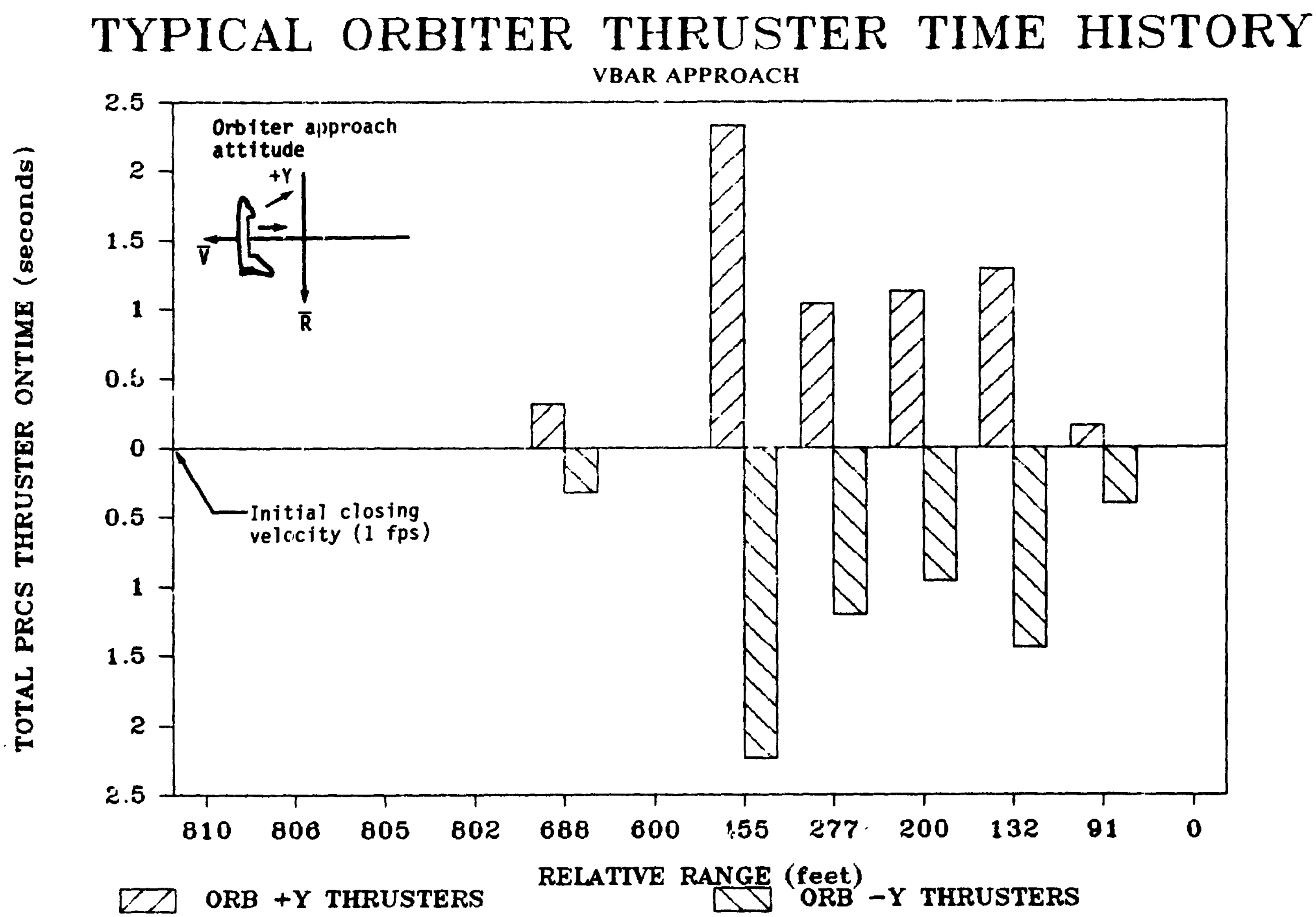


Figure 4.3.8.2.3-8 - Typical Orbiter ± Y PRCS thruster time history for a V-bar approach.

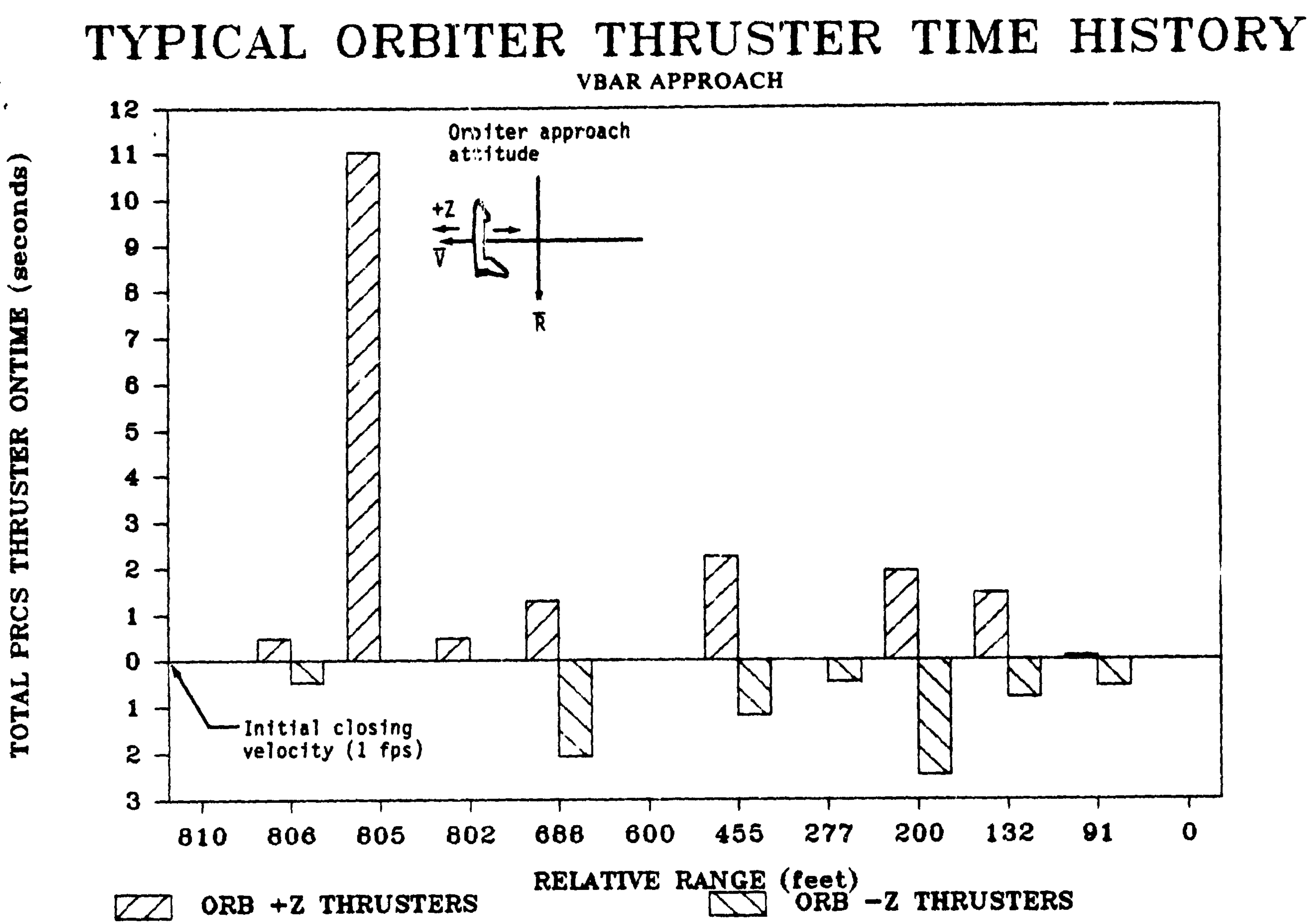


Figure 4.3.8.2.3-9 - Typical Orbiter ± Z PRCS thruster time history for a V-bar approach.

4.3.8.2.4 Station reference design suitability for PROX OPS flight operations

The Station, which maintains an LVLH stabilized attitude profile, has one docking port located on its positive V-bar, with an alternate port located on the + R-bar. Two large booms project from the central truss structure in the out-of-plane direction and each contains a set of four solar arrays. The area of these arrays is approximately 2400 square feet each. Figure 4.3.8.2.4-1 shows the Station from four different orientations. Clockwise from the upper left, the Station is shown looking along its positive angular momentum vector; along the positive V-bar; up the positive R-bar; and finally in an arbitrary orientation essentially viewed from below (+ R-bar). The solar array panels, radiators, and various modules (habitation, laboratory, etc.) are clearly visible in the figure.

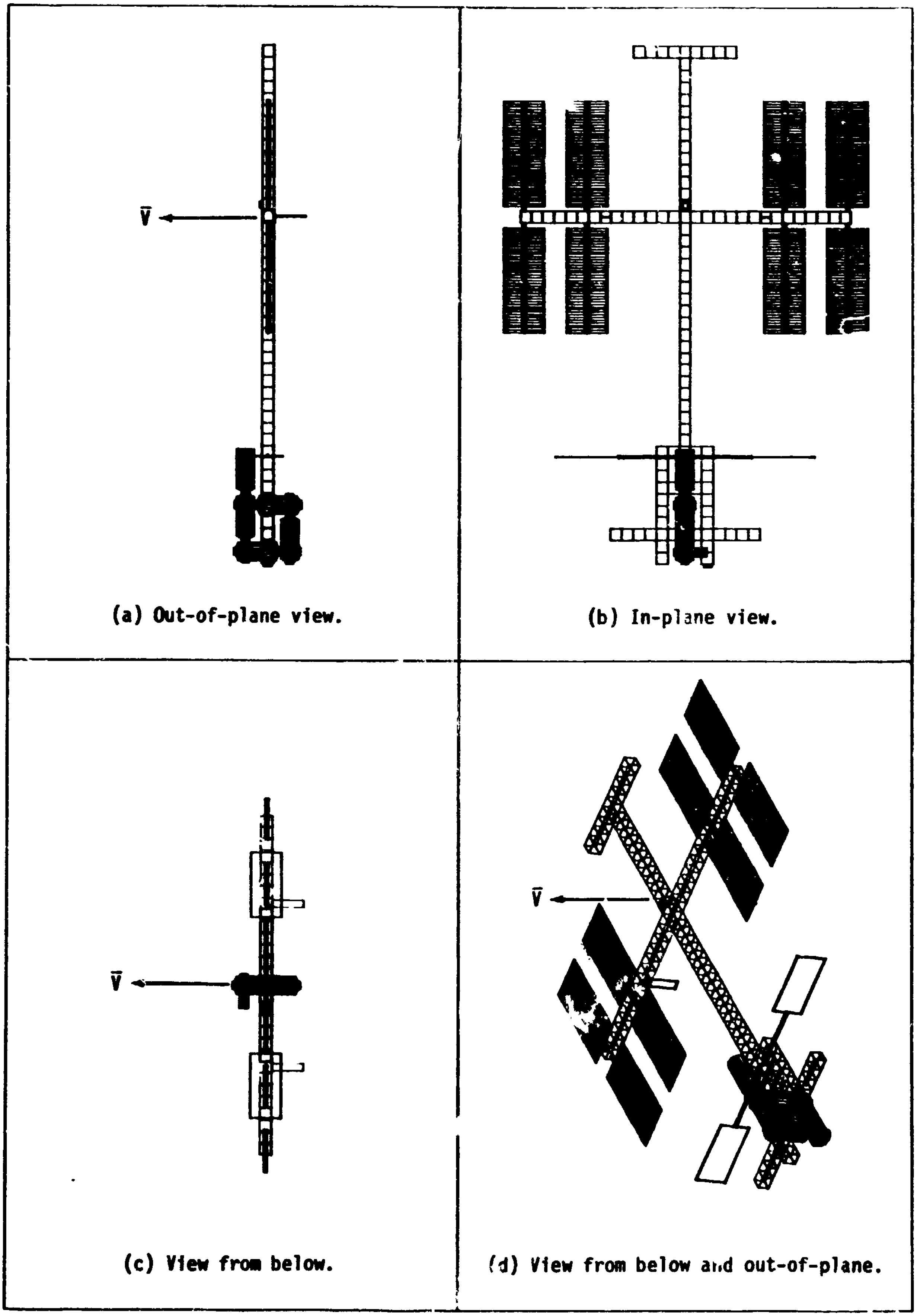
This specific configuration has been reviewed and evaluated in terms of the following evaluation criteria:

a. Procedure design/implementation
b. Propellant usage
c. Mission timeline impact
d. Lighting
e. Clearances
f. Plume impingement

The results of this general review are discussed in the following sections:

4.3.8.2.4.1 Procedure design/implementation ease

With the accessibility of a port on the positive V-bar, the Space Station configuration provides for procedural design of the simplest of approach scenarios. This is ideal from the standpoint of a manned vehicle approach to the Station. The alternate port, located on the positive radius vector, offers a reasonable approach for an unmanned vehicle. After an initial approach to the V-bar, the unmanned vehicle can initiate a flyaround (aided by the effects of orbital mechanics) allowing it to reach + R-bar. This flyaround and the final approach to berthing (along the + R-bar) may all be executed with the aid of ideal lighting conditions. If the flyaround is initiated near orbital sunrise the Space Station controllers should have "over-the-shoulder" lighting for the entire scenario. This is not the case for a manned approach along the R-bar. Manned approaches are targeted to the + V-bar. To flyaround to the + R-bar would be more expensive in terms of propellant cost (flying against effects of orbital mechanics). Second, the R-bar approach would have to be timed to occur near orbital sunrise or orbital sunset in order to insure that the Sun would not be in the crews' eyes during the approach (not within $\pm$ 20° of the LCS).



(a) Out-of-plane view.

(b) In-plane view.

(c) View from below.

(d) View from below and out-of-plane.

Figure 4.3.8.2.4-1 - VARIOUS VIEWS OF SPACE STATION CONFIGURATION

An alternative, if the + V-bar port should not be available, would be to target the rendezvous directly for the + R-bar. This would require timing such that the approach to the port could be made with proper lighting (depending on whether the vehicle is manned or unmanned). If stationkeeping were required in order to achieve this lighting, propellant usage would increase.

In order to facilitate ideal operations it is recommended that the plus R-bar port be relocated or an additional port be added to the minus V-bar axis. This would be especially beneficial in the case of unmanned vehicles as it would aid in the reduction of propellant consumption by eliminating the need for a flyaround. Lighting would be essentially the same as those operations occurring near orbital sunset. Also, better vehicle berthing clearances would be expected using ports on the + and - V-bar rather than the + R-bar. Adequate clearance could be a problem with the current configuration.

4.3.8.2.4.2 Propellant usage

Propellant usage for approaches to a V-bar port is less than for the other locations. This is expected since all approaches are along the V-bar up to about 200 feet. At this point, a manned + V-bar approach to the port would simply continue along the V-bar to berthing. However, a flyaround would be necessary to reach the + R-bar port unless the rendezvous were targeted to the + R-bar, as mentioned above. Also, an approach along the V-bar can be stopped at anytime and a minimum propellant stationkeeping position could be established. Establishing stationkeeping during a flyaround (especially on the + R-bar) can be expensive in terms of propellant usage. The most practical method would be to stationkeep on the V-bar.

4.3.8.2.4.3 Mission time impact

Generally, approaches to the V-bar require the least amount of time. Approaches to R-bar ports require alignment with the port prior to closure. The main tradeoff between a flyaround from the V-bar to the R-bar and direct rendezvous to the R-bar would be the resulting propellant usage. However, the V-bar port locations allow for more mission flexibility since they nominally allow for a stationkeeping phase prior to starting the approach at a small cost, both in terms of complexity and propellant usage.

4.3.8.2.4.4 Lighting

Activities during PROX OPS are very dependent upon lighting conditions since it is mandatory that the crew observe all activities by visual means using the eye (preferably), television, or other acceptable electronic techniques. Therefore, the location of the sun will impact the activity timeline for PROX OPS. The basic ground rule for daylight approaches is that the angle between the line-of-sight to the sun and the line-of-sight to the vehicle be greater than 20 degrees. Proximity operations, especially final approach and berthing, are also preferred to be done in daylight.

The reference configurations's plus V-bar port provides the best port location for manned vehicle approaches such as the Orbiter berthing with the Space Station. The + R-bar port is a much more difficult port for a manned vehicle to accomplish final approach and berthing because the Sun may be near the line-of-sight to the Space Station during these final activities.

The + V-bar port is undesirable for approaches of unmanned vehicles to the Space Station since the crew on the Space Station would also have a possible Sun lighting problem during daylight operations. However, the + R-bar port provides an ideal port location for sunlight operations with only one basic problem: a flyaround would nominally be required to reach this port. If the stationkeeping position prior to the flyaround was on the - V-bar, the impact would be minimized. Therefore, from a lighting standpoint the reference configuration is excellent for manned approaches to the + V-bar port and undesirable for manned approaches to the R-bar port. For unmanned approaches, the + R-bar port provides a very good situation for final approaches and berthing whereas the + V-bar port is an undesirable location. Adding a - V-bar port would make this an excellent configuration for both manned and unmanned active vehicle activities.

Lighting considerations for separations from the Space Station are much the same as those discussed for approaches. Manned separations are preferred along the + V-bar and should be initiated early in the orbital day for lighting. Similarly, unmanned vehicle operations are preferred from the V-bar early in the orbital day. The station has a + V-bar port; hence it is very satisfactory from the standpoint of manned vehicle separations.

The Station also has a + R-bar port which allows both manned and unmanned separations from the preferred direction for each type vehicle. Therefore, it will experience minimum impact with regards to lighting during separations.

Approaches and separations during darkness are also possibilities. They would require, however, sufficient artificial lighting such as running lights on both vehicles or flood lights. The assessment of the configuration regarding this requirement cannot be made at this point.

4.3.8.2.4.5 Transfer vehicle visibility

Vehicles in the vicinity of the Space Station will require monitoring by the Space Station traffic controllers at all times. This could be accomplished by strategically locating cameras around the Space Station structure. The number and location of cameras on the station is TBD.

4.3.8.2.4.6 Clearances

The reference configuration appears to present no major clearance problems between the station elements and PROX OPS vehicles as large as the Orbiter. However, if an out-of-plane flyaround were ever required, the lower booms emanating from the keel extension and the radiator panels might present some clearance problems. Also, active vehicle-to-vehicle-clearances may be required when trying to dock two vehicles at once to the Space Station, especially two Orbiters. Adding a minus V-bar port or moving the R-bar port to the minus V-bar position would basically solve this problem.

4.3.8.2.4.7 Plume impingement

Vehicles located in the near vicinity of another vehicle actively using thrusters to perform maneuvers (translation, rotation, or stationkeekping) are subject to impingement of thruster exhaust. Impingement on vehicle surfaces can result in two undesirable consequences--disturbance and contamination. Each of these is a function of the location of the impinged vehicle relative to the thruster centerline and the relative range. Contamination is also dependent on the type of exhaust products emitted; for instance, combustion reactions produce a greater volume of contaminants than a compressed gas system.

It should be realized that plume impingement cannot be totally eliminated during the PROX OPS period, but its effects can be reduced through vehicle and procedure design considerations. During separation, jets must be fired to move the vehicle directly away from the Space Station; this results in the jet plume being directed towards the Space Station. For approaches, the relative motion must be nulled at some time prior to berthing which again results in jets being fired toward the Space Station. Design considerations could place Space Station elements in a position or attitude relative to approach lanes to reduce impingement.

The reference configuration has several areas quite susceptible to plume impingement for both separations and returns. For PROX OPS activities to and from the + V-bar port, the radiator panels are essentially normal to direction of thrusting, although offset. Also, one of the modules is directly in line with the thrust direction.

The rotation feature of the solar arrays may allow for a reduction in plume impingement. Through a possible combination of timing the V-bar approach or separation to occur slightly before orbital noon and orienting the Orbiter in the most advantageous attitude, the arrays may have rotated to a position essentially "edge-on" with respect to the plume. Problems could arise, however, during approach/separation operations if orientation is such that the plume flow is essentially normal to the arrays.

4.3.8.3 Construction of Large Structures

The Space Station will serve as a base for construction of large structures. The Station was assessed relative to its capability to support construction of a large submillimeter astronomical observatory depicted in Figure 4.3.8.3-1. This structure is essentially a 100-by-100-foot cylinder, with a 30-foot extension at the base, weighing 20,000 lbs. to 60,000 lbs. The following areas relating to the construction of this observatory were considered as part of this preliminary evaluation: offloading and storage of component parts, work stations, construction methods, and impacts on the Station.

The high structure to Station mass ratio from 1:10 to 1:5 must be considered with respect to attitude maintenance during offloading and subsequent related operations. The MRMS will be positioned on one of the keel extensions when accessing the Orbiter cargo bay. Preliminary calculations indicate that an offloaded mass of approximately 50,000 lbs. would shift the Station center of gravity three to six feet out of the orbital plane, which should be considered when sizing the Station attitude control system. The observatory packaging, mass, and number of packages are unknown; however, a gross estimate of the volume of storage required is 20,000 ft^3 or more. The storage area choice must consider the location of the construction area, MRMS translation and reach capability,and the OMV/payloads servicing bays, all of which occupy or require space along the truss. Another important consideration is CG/CP offset, necessitating that the storage/construction area be within the orbital plane, and located on the "back" of the lower keel as indicated by TDM2060 in the station configuration reference drawing.

Several methods of construction were examined before being discarded as impractical or unfeasible. The method appearing to have more advantages utilizes the support structure of the instrument unit itself during construction, and requires a turntable with the added ability to tilt towards the lower keel (see the station configuration reference drawing). The pivotal point must be approximately the radius of the instrument unit away from the lower keel. The bottom of the instrument unit would be mounted on the turntable and tilted so that the top of the unit is as close as possible to the MRMS for an adequate reach envelope. At this point the MRMS can begin construction and the turntable can present the correct arc and tilt away from the lower keel as the cylindrical base grows.

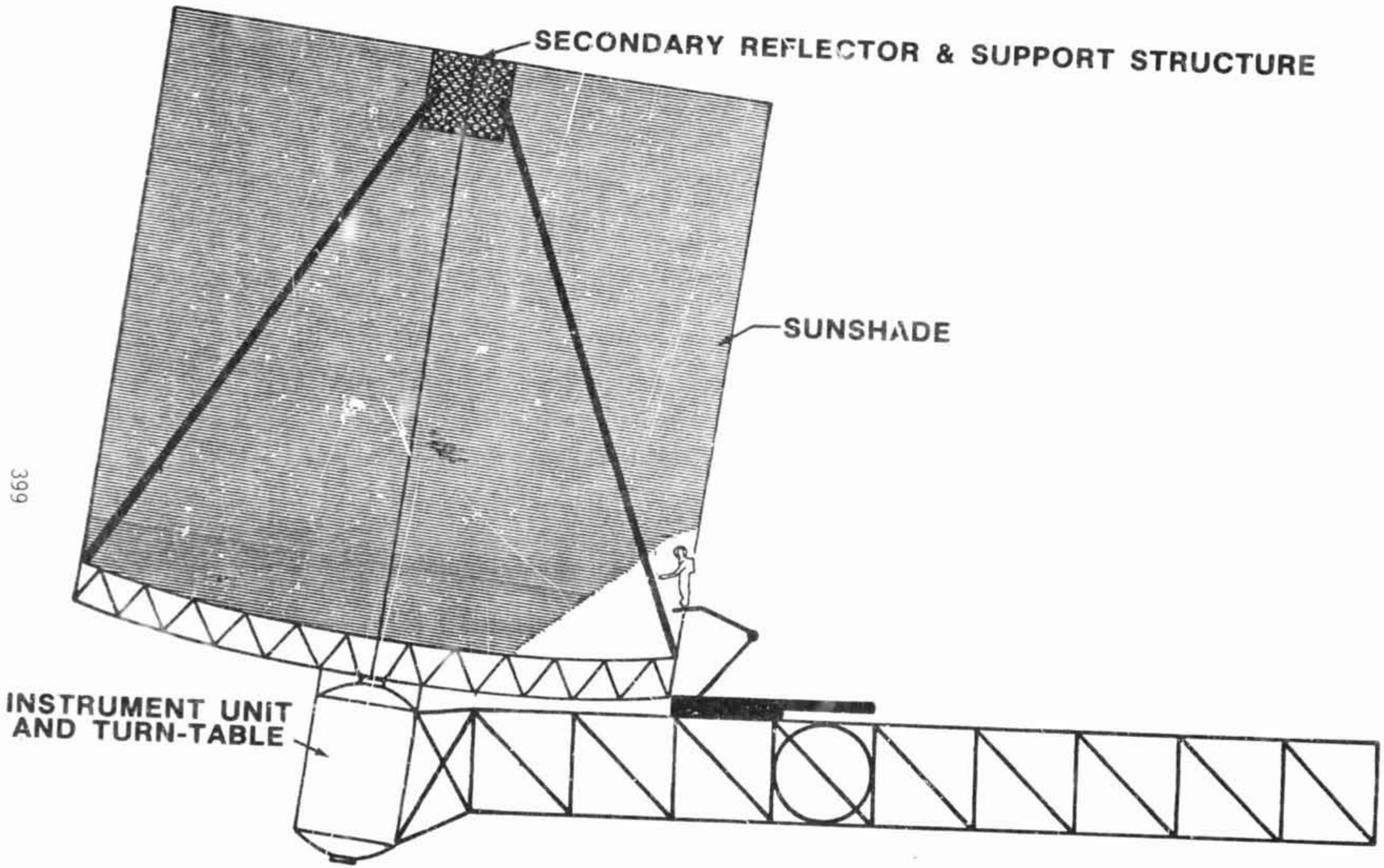


Figure 4.3.8.3-1 - LARGE SUBMILLIMETER ASTRONOMICAL OBSERVATORY

The scenario for the construction of the sunshade can take two forms. Construct two or three panels of the shade, mounting each as it is constructed; construct the secondary reflector and mount it; then construct and mount the remainder of the sunshade panels. The alternative method is to first construct the secondary reflector, mount it, then one by one construct and mount each panel of the sunshade. In either case a construction area would be required for assembling the sunshade panels.

Construction itself of the panels could be a problem. They have not been designed and little is known other than their dimensions of approximately 40 by 100 ft. The Station affords no construction platform this size, but it cannot be stated categorically that only this size is required.

Following are the main areas requiring further study in order to accommodate construction of this large observatory at the Space Station. The effect of long and short duration exposure of the covered facets to the Sun is unknown. The effect on the Station of a delay in deployment of the observatory should be assessed. Attitude control consumables and CMG sizing may be affected if the observatory remains attached for a prolonged period of time. Plume impingement modeling should be done.

Station RCS thrusters are in proximity to the construction site and the Orbiter will probably return at least twice during construction. Finally, payload view factors during the period of observatory construction may be adversely affected and will need a detailed assessment.

4.3.8.4 Logistics/Resupply Operations

Logistics for the orbital operation of the Space Station system will consist of the orderly planning and execution of the resupply of propellants and consumables, delivery of spare/repair parts, delivery/return of payloads, and any new or damaged ORU, return of waste and processed material, and delivery of consumables to support crew rotation. Logistics support includes the packaging, handling, storage, and transportation of Station consumables and replacement equipment. Long-term activity planning will provide the integration of requirements and schedules for the various logistics tasks. The STS will provide the means for delivery to the Station or return to the ground on a 90-day resupply schedule.

The Space Station has unique features relative to logistics and resupply. The STS has a short turnaround (1 week) when mated to the Station. The Logistics Module offers limited stowage capability for equipment and gases/fluids on any single flight, and may have to be supplemented by a pallet for additional tankage and supplies. Yet the manned mission is very extensive. Crew turnover will primarily be on the same 90-day schedule as resupply, requiring volume/weight allocations to support unique crew requirements that might otherwise be used for Station or payloads, and, due to the small number of crewmembers, there will be no dedicated on-board maintenance/logistics crewmember to perform maintenance on an on-call basis. This means that scheduled maintenance and logistics functions must be minimized. In order to minimize the logistics tasks, consideration should be given to the level of system redundancy of Station elements, quantities of consumables on-board, and maintainability requirements to reduce the frequency of required resupply or repair missions.

Logistics/resupply operations at the Station consists, in a broad context, primarily of changing modules, transferring crew, and properly employing the crew maintenance capabilities between flights. The scenario for changing modules consists of removing a fresh Logistics Module (LM) from the Orbiter, berthing with the Station, connecting fluids and power/data, disconnecting the used LM, and berthing the used LM in the Orbiter cargo bay for return to earth. Any other ORU's not part of the LM or payloads would generally follow the same operational scenario. Following are general requirements derived from the above scenario:

a. Rigid Orbiter attachment to the Station

b. Adequate RMS reach and adequate staging area for all items either for removal from Orbiter or for replacement in the Orbiter

c. Good visibility for the RMS/MRMS operator

Station operations during crew transfer must consider the fact that normal Station operations will be interrupted and there may be a maximum of twelve to sixteen people on the Station for short periods. Consumables such as food, clothing, spares, payloads, and payload materials must be unloaded and taken to use areas. Trash and waste must be processed and stowed, and dirty clothing, processed material, experiment samples, and ORU's must be stowed. These considerations lead to requirements for good module traffic patterns and both internal and external storage space. Section 4.3.8.1 assessed the MRMS capability to reach payloads externally mounted, and the reach envelopes of the MRMS demonstrate adequate capability to off-load the Orbiter and provide for external storage in the vicinity of the module or on the lower or upper keel.

4.3.9 Safety

4.3.9.1 General

The reference configuration as presently conceived is intended to meet the safety design requirements defined in the proposal (RFP). These requirements are to be applicable to all Space Station systems, subsystems, and operations. These requirements apply under worst case natural and induced environments. The configuration design is to either eliminate by design or control all hazards to the maximum extent and in the most cost-effective manner possible.

4.3.9.2 Habitable Module Egress Capability

The Habitation Modules and Lab Modules assembled in the manner described in the reference configuration (i.e., the "race track") allows for two egress paths from any one of these modules. The Logistics Module is generally considered to be exempt from the dual egress path capability implied by the safe haven requirements stated in the RFP. Although the Logistics Module has only one egress path, the risk to a crewmember occupying this volume during the occurrence of an accident forcing evacuation of the volume can probably be reduced to an acceptable level by the proper location of equipment, adequate materials control, elimination of potential ignition sources, and maintenance of adequate traverse clearance.

The previously mentioned capabilities are described as a part of the IOC phase of the reference configuration. Habitation and Lab Modules added to form additional "race tracks" must meet the egress requirements. Additional Lab (for international and commercial users) and Logistics Modules, not included in the "race track," will have to be assessed on a case-by-case basis according to their hazard potential.

4.3.9.3 Protection/Control of High-Pressure or Hazardous Fluid Tanks

It is required that potentially explosive containers be located outside of habitable areas, isolated and protected such that the failure of one will not propagate to others, and designed to "leak-before-rupture." The reference configuration allows for potentially explosive containers (e.g., high-pressure tanks, tanks containing hypergolics, tanks containing flammable and/or toxic fluids) to be located outside of and away from habitable areas. The reference configuration provides ample areas on the keel of the Station structure for high-pressure and explosive containers to be located away from the inhabited regions of the Space Station. In addition, potentially explosive containers located in a group or cluster of similar containers are required to be protected such that the failure of one does not propagate to the others.

4.3.9.4 Isolation of Modules After an Accident

Individual modules within the module assembly of the reference configuration (i.e., the "race track") are capable of being isolated from other modules in the case of an accident or potentially hazardous event such as loss of pressure or fire. In addition, those modules which contain confined hazardous or toxic materials and which are not a part of the "race track" will be capable of being isolated from the remainder of the Space Station habitable areas during emergencies and/or accidents (e.g., loss of pressure, fire, release of toxic materials and/or fluids, materials offgassing).

4.3.9.5 Reaction Time After the Occurrence of a Leak

The hatches of each individual module will have the capability of being closed rapidly in the case of a module leak. In addition, the Environmental Control and Life Support System (ECLSS) is required to have the capability to accommodate atmosphere leakage up to 2.3 kg/day (5 lb/day) for the total Space Station pressurized volume.

4.3.9.6 Safe Haven Capabilities

Although not required, the module arrangement (i.e., the "race track") of the reference configuration allows the Space Station to tolerate any single credible failure, including the complete functional loss of a module. The "race track" modular arrangement allows any crewmember dual access to habitable conditions within the Space Station without EVA. The reference configuration's safe haven concept provides for habitable conditions for the crew in the remaining modules including atmosphere, food, water, waste management, health maintenance, personal hygiene, sleeping provisions, communications, command/control, and fire suppression. The reference configuration provides for redundant command and control by allocating locations where a portable control "console" can be installed as needed. All critical systems/subsystems necessary for implementing the safe haven concept (e.g., ECLSS) will provide emergency capabilities for a period of up to 22 days.

4.3.9.7 EVA Operations

The reference configuration provides capabilities to facilitate EVA operations. A singular concern for EVA operations with regard to the reference configuration is the large distances from the Station airlock to potential EVA work sites. Due to the large dimensions of the Space Station structure, a means of transporting and supporting the EVA crew from the airlock to a potential work site may have to be developed. This may be accomplished by the projected traveling RMS or extensive use of the Manned Maneuvering Unit (MMU). The antenna locations are apparently such that radiation hazards to EVA crewmembers are minimal. At present, the current locations of the Reaction Control System (RCS) packages do not seem to constitute a hazard to EVA crewmembers.

4.3.9.8 Repair and Reactivation of Modules after an Accident

A major issue to be resolved for the reference configuration is the capability to repair and reactivate a module or modules after an accident. Detailed conceptual means of implementing this goal has not been developed at this time. As a general design requirement, all critical systems/subsystems are to be fail-operational/fail-safe/ restorable. In addition, there is a design goal for the inner surfaces of the module walls to be accessible for inspection and repair by the shirt-sleeved crew.

4.3.9.9 Orbiter Docking Ports

The ability to docks with and access the Orbiter from the various volumes of the Space Station is accommodated by the reference configuration.

4.3.9.10 Other Safety Considerations

Several areas of safety concern with regard to the reference configuration will require additional study. These areas include but are not limited to: (1) servicing of satellites and/or free-flyers either on or in proximity of the Space Station, (2) servicing of solid and/or liquid upper stages (PAM's, OMV's, OTV's, etc.) either on or in proximity of the Space Station, (3) flight operations of upper stages in transit to and from the vicinity of the Space Station and, (4) assembly operations associated with the change-out of a module either for return to Earth for repair or refurbishment or for relocation in the growth phase of the reference configuration.

4.4 SUBSYSTEM DESCRIPTION

4.4.1 Electrical power

4.4.1.1 Introduction

The Space Station is a much larger orbiting vehicle than NASA has ever flown in the past. Design criteria different from those of previous programs must be met. These considerations are reflected in the power system design in several ways. The power system must support Station activities which are largely undefined and which will change as activities on the Station evolve. The power system must be flexible enough to meet changing demands and growth over the anticipated 30-year station life. Initial power demands of 75 kW are expected to grow to 300 kW by the year 2000. The power system must be modular with well defined interfaces to accommodate repair-by-replacement and growth without disruption. It should also be able to accept new technology as it becomes available or as it is needed.

The power system for the Space Station consists of three elements: energy conversion subsystem (ECS), energy storage subsystem (ESS), and power management and distribution (PMAD). Within each of these elements are several candidate technologies for consideration as options. Possible technologies are listed in Table 4.4.1.1-1. Both deployable and erectable structures are considered.

A primary goal of the power system design is the development of a utility-type system as nearly independent as possible of outside support from the crew or another subsystem.

For this study effort, two reference power system configurations were considered for the initial Space Station. One was a silicon photovoltaic power system with regenerative fuel cell energy storage and high-voltage ac distribution. A second used a solar dynamic power system with thermal storage and high-voltage ac distribution. Each of these power system configurations has merit for the initial Station. Only the initial Station photovoltaic power system was subject to analysis and integration with other subsystems. A sketch of the initial photovoltaic power system is shown in Figure 4.4.1.1-1. A sketch of the initial solar dynamic power system is shown in Figure 4.4.1.1-2. These selections were representative of many applicable power system options. Each component of the power system configurations will be described in the paragraphs below.

TABLE 4.4.1.1-1 - ELECTRICAL POWER SYSTEM OPTIONS

ENERGY CONVERSION	ENERGY STORAGE	POWER MANAGEMENT AND DISTRIBUTION
DYNAMIC	Thermal	High-voltage ac
Brayton		
Rankine	Regenerative Fuel Cells	High-voltage dc
Stirling		
	NiCd Batteries	High-Voltage Hybrid
PHOTOVOLTAIC		
Silicon planar	Flywheels	
GaAs concentrator		
	NiH_2 Batteries	

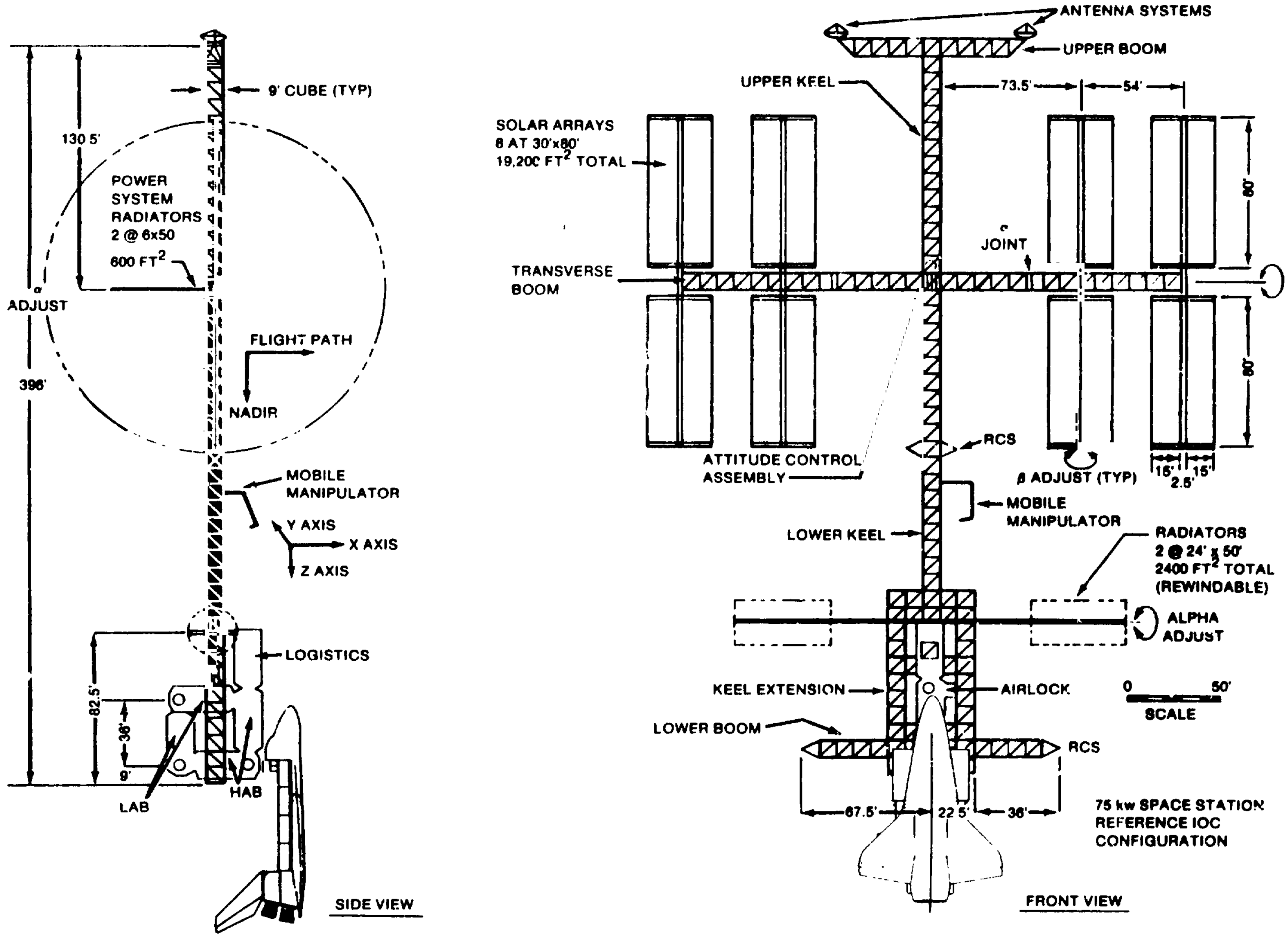


Figure 4.4.1.1-1 - IOC REFERENCE CONFIGURATION

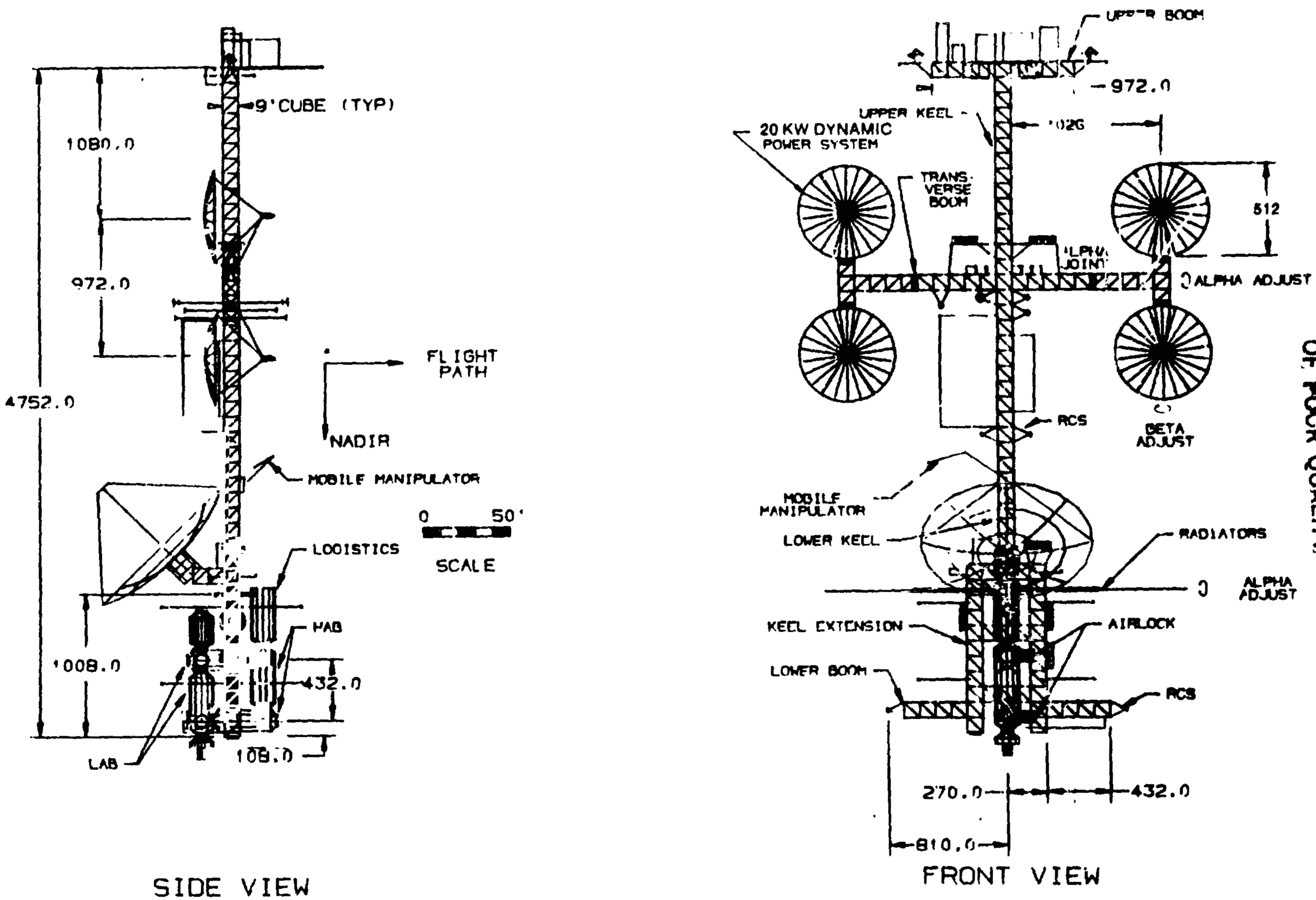


Figure 4.4.1.1-2 - IOC REFERENCE CONFIGURATION (SOLAR DYNAMIC OPTION)

4.4.1.2 Photovoltaic System Generation and Storage

Assumptions used in this study are outlined in Table 4.4.1.2-1.

4.4.1.2.1 Photovoltaic generation

The photovoltaic system chosen for this study is the flexible, planar array utilizing the large area (5.9 cm x 5.9 cm) silicon cells. This reduces the total number of cells and the manufacturing and handling steps required in array fabrication. The cells would be attached to a flexible kapton (or similar material) substrate and welded to the attached circuitry in order to provide better resistance to the thermal cycling that will be experienced in LEO. Since the blanket will have little or no structural stiffness, a means of structural support must be provided. For this study, a unit weight for the solar array blanket and support structure of 2.5 kg/m^2 was assumed. Whether this lightweight structure is adequate is subject to further study.

A design life goal of 10 years was chosen for the solar array. Since the Station will last longer than the array life goal, provision must be made to change out solar array components. Roughly a 10 percent degradation in cell performance is expected to occur over the 10-year life of the array. The rate of degradation is also subject to further study.

The solar array wing used in this study would allow both deployment and retraction so that the entire wing could be replaced. Other concepts might allow replacement of smaller units. Concepts requiring EVA to erect structure and panels were also considered. As the array wings get larger, erectable concepts become more attractive. Growth capability is a function of the specific array design. In this study, growth would require adding 2 or 4 wings. Several packaging concepts to transport the array to orbit were also addressed in the study.

Two-axis gimbals with rotary power transfer devices have been provided to orient the array normal to the sun. Because of the geometry of the reference Space Station configuration, some of the station (structure or experiments) will shadow portions of the solar array during part of the orbit. This must be recognized as part of the design of the array.

The solar array size is a function of the power level, array orientation, storage system efficiency, distribution system efficiency, expected degradation, and the orbital parameters. The initial array area is approximately 19,200 ft^2.

4.4.1.2.2 Energy storage subsystem (ESS)

The ESS defined in this study consisted of four regenerative fuel cell modules with the required supporting hardware. The storage units will be mounted on the array support structure outboard of the alpha gimbal. Each 20 kW module would weigh about 1262 lbs. and occupy about 75 ft^3.

4.4.1.3 Dynamic System Generation and Storage

The dynamic system chosen for this study is a generic representation of the three most applicable technologies: Brayton, Rankine, or Stirling, each using molten salt thermal storage. Basic operation of the solar dynamic system involves the use of a mirror to collect and concentrate the Sun's energy on a heat receiver. A heat engine is then used to extract energy from the receiver and convert it to mechanical energy to drive an electrical alternator. Excess heat is collected and stored in the thermal storage material within the receiver and is used to operate the engine during eclipse periods. Waste heat from the engine is rejected to space through a dedicated radiator. Because the storage element is between the Sun and the engine, the engine operates continuously around the orbit. The major assumptions used to size the dynamic system for this study are shown in Table 4.4.1.3-1.

Specific choice of heat engine cycle will depend on the result of trades concerning receiver temperature, performance, and development risk. The numbers presented here are indicative of a Rankine cycle using Toluene as a working fluid or a low temperature Brayton cycle. Design results are given in Table 4.4.1.3-2 for both the initial and growth stations.

With proper design, the same rotating machinery can be used on the initial and growth station. The radiator required as part of this power system fits behind the primary mirror. This radiator is an integral part of the power system and is not part of the station thermal system.

The collector can be a single parabolic mirror or a two-element Cassegrainian. Either may be deployable or erectable although erectable devices seem to package in smaller volumes. The mirror has high reflectivity and high geometric accuracy to provide the concentration that permits a small receiver aperture to be used, resulting in low reradiation losses.

The heat receiver absorbs, stores, and transfers the solar energy to the engine working fluid. In most concepts, the storage is in the heat of fusion of a salt with a fixed melting point. Therefore the engine peak operating temperature will be chosen in conjunction with the choice of salt and containment material. Mirror accuracy will also be a factor in the choice of operating temperature since high temperatures place more stringent requirements on mirror fabrication.

The unit size was arbitrarily selected and should not be considered optimum. However, the use of four units allows a relatively simple initial configuration which is amenable to growth.

Two axis gimbals, identical to those used in the solar array, have been provided with rotary power transfer devices to maintain alignment of each wing to the Sun. In addition, a small amount of fine pointing capability has been given to each unit to compensate for station pointing and gimbal errors.

TABLE 4.4.1.3-1 - Dynamic Power System Design Assumptions

Collector efficiency	90%
Receiver efficiency	90%
Engine efficiency	20 - 30%
Sun period	59 min.
Eclipse period	35 min.

TABLE 4.4.1.3-2 - Dynamic Power System Design Results

	Initial	Growth
Station power demand	75 kW	300 kW
Unit engine size	20 kW	37.5 kW
Number of installed units	4	8
Mirror/radiator diameter	13 m	18 m
Mirror type	simple parabolic	
Unit weight	2100 kg	3410 kg

4.4.1.4 Power Management and Distribution (PMAD)

The PMAD subsystem transfers electrical power from the source to the user interface. It conditions and controls the power and provides individual protection for each user interface, power source, and PMAD hardware and wiring. The subsystem manipulates sources, loads, and buses as required for most effective utilization of power and provides the mechanization for recharging the energy storage devices. If the PMAD system is properly designed, it can function with several different power sources.

DC power (or ac from a solar dynamic power system) from the generation and storage devices is converted to high frequency three-phase power for transmission via multiple redundant buses throughout the station. Each bus consists of a substation with overrides for selected switches, an automated power management system (APMS) interface, a primary ac bus, secondary ac and dc buses, and user utility outlets. Disconnects and crossties controlled in the substation are in the main bus structure to provide the flexibility to utilize to the maximum that portion of the primary circuitry remaining intact after a fault or during maintenance. Finally, the interface between the PMAD and the user is the utility power controller (UPC), which is controlled by the APMS via a link to the data bus. Although this design utilizes high-frequency ac distribution, the choice between ac or dc for primary power transmission will be determined by weight, volume, cost, and growth potential considerations. The interface with the user will remain sine wave ac. The distribution of PMAD equipment around the station is shown in Table 4.4.1.4-1.

TABLE 4.4.1.4-1 - DISTRIBUTION OF PMAD EQUIPMENT

Central structure	1983 lbs
Logistics module	212 lbs
Habitat module	471 lbs
Laboratory module	514 lbs
Control module	652 lbs

4.4.2 Guidance, Navigation and Control

This system has the responsibility for managing the sensing and acquisition of information, computation, and actuation required to provide position and attitude control for the Space Station and to point its solar arrays, radiators, and payload mounting surfaces. The GN&C system will interface with the Information and Data Management System (IDM), Communication and Tracking System (C&T) and Propulsion System (RCS) to perform these functions. The GN&C system will also manage the traffic control function and proximity operations. GN&C support will be provided to the payloads attached to the Station and to the Station traffic. Satellites and vehicles berthed to the Station will have their control systems inhibited, and the Station will be responsible for attitude control. The GN&C system (in conjunction with the propulsion system) will have responsibility for orbit altitude maintenance and adjustment, collision avoidance and deboost in the event that disposal of the Space Station is required. The GN&C system will use the mass properties, propellants, and structural bending data provided by IDM to provide attitude control and orbit maintenance of the Space Station. The baseline system architecture is shown in figure 4.4.2-1.

4.4.2.1 Functional Description

The GN&C is a major Space Station system which is critical for attitude control, orbital maintenance, solar pointing, traffic control, proximity operations, and experiment stability. These functions are vital to the success of the mission, efficiency of operation, and safety of the Station and crew. For this reason, the highest level of reliability must be considered from the outset, using error detection techniques, redundancy, standby equipment, and backup or alternate means for achieving the most critical functions. In contrast to reliability, the performance requirements such as sensor accuracy, memory capacity, computation speed and effector characteristics are not as critical to this program, and certainly not beyond existing technology. Therefore, new development should be considered primarily to update the specified equipment in terms of efficiency, reliability and lifetime. The equipment and software interfaces must permit the substitution or addition of elements for this purpose throughout the lifetime of the Station.

4.4.2.1.1 Baseline GN&C approach

The baseline control system (Figure 4.4.2-1) is a centralized system of colocated attitude sensors and CMG control effectors which provide inherent stability of the flexible body. The navigation system utilizes the Inertial Sensor Assemblies (ISA) composed of rate gyros and accelerometers to provide attitude, attitude rate, position, and velocity. Star trackers will be used to update attitudes, and GPS satellites will be used to update position and velocity. The navigation processing will be performed in the Navigation and Traffic Control Computer.

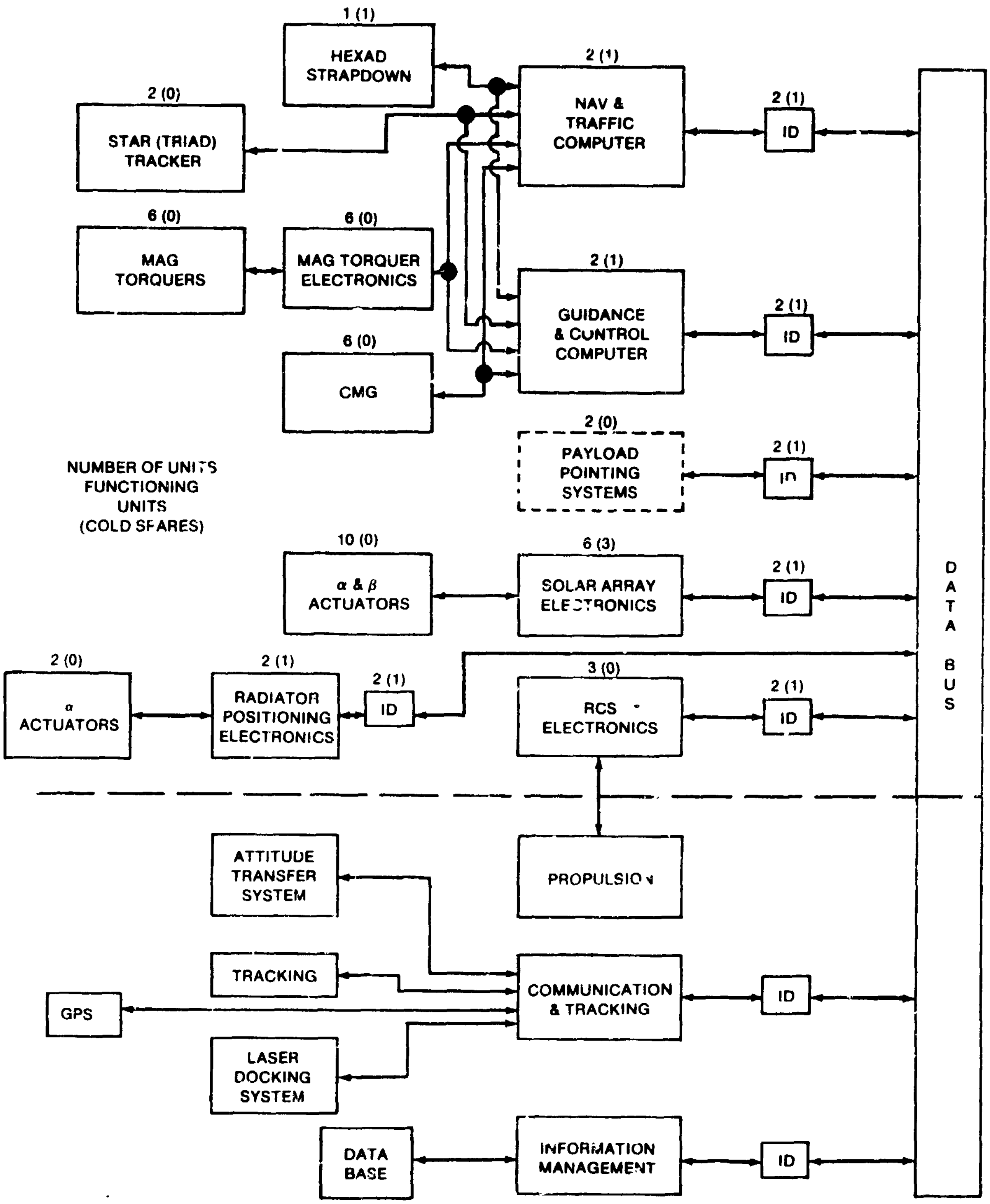


Figure 4.4.2-1 - GN&C SYSTEM ARCHITECTURE

The primary attitude control effectors will be CMG's and may be supplemented by magnetic torquers to perform continuous momentum management. The RCS will be used to perform all translation maneuvers of the Station. RCS will be used to backup the CMG's when the momentum or torque exceeds the capability of the CMG's. Also, the RCS will be used for primary attitude control with reduced pointing in the event that the CMG's become inoperative, and the RCS will be used to perform momentum management in the event that the magnetic torquers fail. All of the Station's control systems (CMG, RCS, magnetic torquer, etc.) will have the capability to operate simultaneously or individually. An attitude transfer system will be used to determine the relative alignment between the navigation base and the mounting surface of the pointing instruments.

4.4.2.1.2 GN&C equipment location

The GN&C equipment locations are as follows:

a. The CMG's and ISA's will be installed in the Attitude Control Assembly (ACA) on the ground prior to Space Station launch. This ACA will, in turn, be installed at the intersection of the transverse boom and the main keel to optimize vehicle mass distribution for control dynamics. The accessibility of this equipment for maintenance will have to be addressed in more detail. However, it should be noted that all CMG's are centrally located on one reference base and that the gyros (ISA's) are colocated with the CMG's.

b. The star trackers have tentatively been located on the ACA. An alternate location would be on the upper boom. A study will have to be made to determine the optimum location.

c. The magnetic torquer locations are TBD at this time, but will likely be on the main keel and/or one of the booms.

d. The solar array and thermal radiator drive electronic units will most likely be placed at the locations of the drives themselves. If this proves impractical, they will be placed in the ACA.

e. The G&C processors and the interface devices (ID's) associated with flight control and its sensors and actuators will be located in the ACA. These processors and ID's are required to provide attitude control during the unmanned buildup phase of IOC.

f. The navigation/traffic control processors and the other miscellaneous GN&C electronics will be located in Habitat Module 2. The location of each of the ID's that support the remote sensors will be at the sensor itself.

g. The philosophy on spares is to have flight critical spares installed in place. It may also be desirable to have the spares for all the remote GN&C sensors installed and in place to alleviate complex installation/alignment tasks.

4.4.2.1.3 Alternate control system approach

An innovative approach is the pointing and control system (PAC) which would colocate the control sensors and effectors with the pointing payloads and dynamically couple this system to the remaining Space Station. This would permit precisely oriented pointing mounts that would be loosely connected to the Station, thereby isolating some of the Station's motions from the pointing instruments. The Station's control torques would be effected through a spring-coupled system between the Station and the PAC.

The Station could typically have multiple PAC elements, each of which is internally redundant and completely independent of the other in terms of sensors, computers, and effectors. Each element would be capable of operating in a slave mode, in which the effectors are commanded by the master element to support the momentum management task of the entire Station. The elements would be connected by an optical link or star trackers that would establish the relative alignment of one to the other. The master element would be the one which has the best viewing direction to the stars, unless there is catastrophic failure of one element, in which case the other assumes command. The master element always establishes the reference coordinate system. Because of the rigid structure and the close proximity of sensors and actuators, the most stable part of the Station would also be the best pointing part of the Station. Therefore, the experiment pointing mounts would be attached directly to PAC elements.

A single PAC element would be configured to fit easily within the Shuttle payload bay for replacement and return of intact elements in case of major malfunctions. For this reason, the attachment of the elements to the Station would be simplified. During operation there would be an option for rigid connection to the Station structure or spring-coupled connection. One advantage of the PAC is that transient disturbances on the Station such as man motion and rotating machinery are filtered and attenuated before reaching the stable element. Another advantage is that the bandwidth of the controller can be increased substantially without interacting with the Station structure. This makes the PAC element a very stable structure, capable of holding the Station in an orientation to minimize momentum accumulation and simultaneously providing a high degree of isolation, stability and pointing knowledge to the experimenters. This translates into much simpler and lower-cost mounts for pointing the various instruments. The mounts can be built with low technology gimbals, since their main function will be instrument orientation rather than a high level of stability. The other two major problems of conventional mounts, namely, the constraint on experiment center of mass relative to the gimbal point and the transfer of electrical and fluid lines across the gimbals, are greatly reduced.

4.4.2.1.4 Traffic control and proximity operations

The GN&C system will have management responsibility for traffic control and proximity operations of station traffic which is operating within the

Station's area of influence, in order to assure that the operational procedures meet safety and mission constraints. The area of influence is defined to be a 30° cone about the plus and minus velocity vectors of the Station out to 2000 km and 100 percent of a sphere with a radius of 8 km around the Station. The GN&C system will also have the capability to monitor and control vehicles which lie outside this area, but whose trajectories will eventually cause them to enter this area of influence.

The responsibilities for traffic control and proximity operations will be shared among the GN&C, IDM, C&T, and ground systems. Assurance of the safety of the Station will be the responsibility of the GN&C. The ground systems have responsibility for monitoring space systems while they are beyond the area of influence of the Station; however, the GN&C system assumes responsibility for the final approach and berthing of Station traffic.

Traffic control includes tracking, monitoring, and controlling of the space systems which are performing operations within the Station's area of influence. Proximity operations include stationkeeping (formation flying), berthing, and departure from the station. Using C&T and IDM system, the GN&C system will determine and maintain the relative state vectors (position and velocity) of station traffic. This information will be used to establish the velocity maneuvers required to maintain safe positions for stationkeeping and for berthing. Tracking of station traffic and detectable debris will be provided, and trajectory extrapolations will be performed to predict the possibility of collision. The GN&C system will have command, control, monitoring and abort capabilities to achieve safe berthing of active vehicles with the Station. An override capability to wave off any rendezvous which is deemed to present a safety hazard to the Station will be provided. Remote piloting of unmanned transportation systems which perform proximity operations with other systems in the area of influence will be supported. This will require capability for continuous command, control, monitoring, communication, and tracking of the transportation system and monitoring and tracking of the target vehicle.

The relative state vectors of tethered satellites will be provided and control of the tether system (e.g. reel control) to meet both safety and operational requirements will also be provided. The capability to retrieve, stow or jettison the tethered satellite will also be provided if the safety of the Station is in jeopardy.

4.4.2.2 Equipment

4.4.2.2.1 Inertial sensor assembly (ISA)

4.4.2.2.1.1 Traditional approach.

The ISA will consist of gyros and accelerometers mounted in a redundant strapdown configuration. The ISA will measure attitude, rate, and accelerations. The ISA outputs will include attitude, rate, velocity, and acceleration in various reference systems that are TBD at this time.

The ISA will also provide subsystem status, i.e., malfunction, configuration (mode), command/control, and data routing. The ISA's will be located with the CMG's in order to avoid structural instabilities. Accessibility, modularity, and maintainability considerations will have to be recognized as significant design drivers. The hexad (6 skewed sensors) configuration was selected because it allows complete fault isolation and recovery for the first two failures and partial for a third failure. Also the hexad is tantamount to having spares in place so that failed components can be replaced later when Orbiter brings up replacements.

Laser and fiber optic gyros will be evaluated as candidate components. Laser gyros which employ dither will be evaluated within NASA to insure that the dither vibrations and/or its related noise do not adversely effect star trackers, flight control or low acceleration experiments. Particular attention will be given to fine pointing, low acceleration and high resolution/low noise rate data. Accelerometers must be able to measure reboost accelerations and monitor low level accelerations in support of experimentation.

4.4.2.2.1.2 Optical approach

An optical ISA approach is the same as the traditional ISA approach except that the rate gyros are eliminated. Currently under development by JSC/ASD is state-of-the-art star tracker technology for the direct and continuous determination of inertial attitude by continuous tracking of two or more stars. Spacecraft located in orbit can use star trackers to measure attitude and angular rates directly. Large angle excursions require that standby gyro assemblies be available to provide attitude information during these events. Each tracker assembly will have three tracker heads with fields of view arranged in a triangle having 10 to 30 degrees between the lines of sight of the star trackers. Any two star trackers are sufficient to determine attitude and attitude rate. If two star trackers become inoperative, stars in only one tracker will define the attitude of two body axes accurately but the third body axis will be very coarsely defined. Built in microprocessors will manage the acquisition, identification, tracking and data conversion functions of the trackers. The design goal is 5 arcsec (1 sigma) per axis. Lab tests show 10 arcsec accuracy is easily achievable. The design goal is to support angular rate precision of .001 deg/sec at 10 Hz update rate.

4.4.2.2.2 Star tracker

Star trackers will be used to update the GN&C attitude reference for the traditional ISA system or for direct attitude reference and rate for the optical ISA system. The FOV is bounded at the upper end by view blockage, attainable accuracy, imaging lens collecting aperture and scattered light shielding and at the lower end by the number of stars with known coordinates, angular rates and practical upper size of image lens. A 2-degree FOV is recommended for the star tracker as the best compromise, since it yields maximum flexibility for location relative to the structure, compact light shades, a reasonably small imaging aperture, sensitivity sufficient to guarantee at least one star per FOV using existing star catalog, and capability to measure angular rates of up to 0.1 deg/sec.

The largest currently available precision star catalog contains approximately 256,000 stars. About 100,000 of the cataloged stars are redundant or not usable, because they are too faint or too close to other brighter stars. The distribution of stars in the catalog is non-uniform and can vary over a 10:1 range, thus requiring that the star catalog for the star trackers contain at least 156,000 stars in order to guarantee at least one star per FOV. A catalog of this size and precision requires about 10 megabytes of ROM storage which is achievable with bubble memory chips or a miniature laser disk drive in the near future. Only one catalog storage device and one backup device would be required for all of the star trackers, since each would use only a small subset of the catalog at any one time.

Light shading is primarily driven by FOV and imaging optics aperture size. If a relatively modest FOV and lens size is selected, a shade length of one foot can provide 10 degrees bright source shading for Earth, Moon, and spacecraft and 20 degrees for the Sun. The sensor head that is proposed is immune to damage by any sources except the Sun. A combination of blocking filters and automatic gain control could be used for protection against the sunlight.

A single triad assembly of three FOV would provide fail safe capability. Adding a second triad assembly provides fail-op, fail-op, fail-op, fail-safe capability provided that the two triads are mechanically tied together. Some additional redundancy capability is lost with tracker head failures due to reducing the number of fields available which increases the probability of occasional blockage by the Moon, Sun or co-orbiting objects.

Installation options are driven primarily by maintenance and accessibility. The only sensor-related driver is that the light shade be in the vacuum environment ahead of the tracker's imaging lens or window. Mounting configurations can range from the entire assembly being in a vacuum to the assembly being inside the modules, viewing through a port. The first option requires EVA for maintenance but offers the maximum flexibility for sensor pointing. The last option requires no EVA but has minimum pointing capability. Also, the sensor head can be separated from the processing electronics with head located outside and the electronics inside or the lens and light shade can be located outside with the head and electronics inside with a viewing port.

4.4.2.2.3 Computers and data interfaces

The overall design will be compatible with the Information and Data Management (IDM) system. The IDM design philosophy is distributed processing. GN&C will utilize the IDM data network to communicate both intra- as well as intersystem data (see Figure 4.4.2-1). The GN&C system will utilize IDM standard/common components as dictated by the results of trade studies to be conducted in support of the IDM architecture and topology. The computational requirements will be partitioned into two processors: (1) navigation and traffic, and (2) guidance and control.

The guidance and control and the navigation and traffic control areas can be considered as individual subsystems within the overall GN&C area of responsibility. Each of these subsystems requires its own standard/common processors and interface devices (ID). The associated redundancy for each of these devices will be determined as a result of the technology level and design capability chosen as a result of trade studies. The system will be designed to meet a minimum of fail operational/fail safe/restorable. The system will be designed to accept technology development so that improvements can be made to the GN&C system with little or no impact to the remaining system. The requirement of three computers for fault isolation will set the minimum. Because fast reaction is not required on the Station, consideration will be given to operating only two computers normally in order to reduce the power requirement. In the event of a computer failure, two computers can recognize that a failure has occurred. The control system can hold last output until the third computer is brought on line and the failed computer is identified. The two banks of computers (NAV & traffic computers and guidance & control computers) will be capable of taking over the other's functions in order to fulfill the redundancy requirements.

4.4.2.2.4 Control moment gyros (CMG)

The CMG's are devices that store momentum by changing spin vector orientation of a wheel which is spinning at a constant rate. The CMG's have a double gimbal which permits momentum storage about any axis. The double gimbal approach provides for redundancy and flexibility of operation. A combination of double gimbal CMG's and magnetic desaturation devices is recommended for the GN&C baseline system. The NASA TM-82390, Steering Law for Parallel Mounted Double Gimbaled CMG'S - Revision A, by Kennel (Reference 34), is recommended as a guide for the implementation of CMG steering laws.

Momentum storage requirements are defined here as the peak cyclic momentum plus secular accumulation over one orbital period. This is equivalent to taking the maximum RSS value of the momentum components about each axis. The full range of the momentum storage device is considered in the sizing, assuming a prior knowledge of the momentum profile or the ability to initialize to a state which takes maximum advantage of momentum capacity. Double gimbal CMG's will be oversized by 50 percent to account for uncertainty of the analysis.

Momentum management is necessary to prevent saturation of the CMG's. A storage device will eventually become saturated by secular (bias) torques which act in one direction. Momentum desaturation can be achieved by applying torques that reset the storage devices to their initial state. These torques can be generated in various ways such as realignment of station principal axes in the gravity field (gravity gradient desaturation), utilization of aerodynamic moments, generation of magnetic fields which react against earth's field, or thrusters which produce reaction control torques. Orienting the station in the gravity gradient field is the preferred momentum management technique; however, magnetic torquers may be required in addition to supplement gravity gradient momentum management. Large area magnetic coils will also be studied.

4.4.2.2.5 Magnetic torquers

If a device is required to desaturate the CMG's, a magnetic torquing device is a low power, reliable, and nonconsumable means of providing torques or momentum into the station. Magnetic torquing coils have flown on numerous spacecraft over the years with excellent results. These systems, along with systems currently under design for use in the late 1980's, provide torque outputs as large as .088 ft-lbs. Depending on design, it is reasonable that magnetic torquing requirements may be greater and several bars may be required. Previous coil applications have been directed heavily toward linear control; however, increased magnetic dipole requirements may make non-linear magnetic torquing a viable option. As a result of the large magnetic dipole being created, significant magnetic field disturbances will be produced in the vicinity of the magnetic torquers; however, the field decreases with the square of the distance, and the earth field predominates at moderate distances from the bars.

An alternative technique for producing control torques using magnetic torquers is the large area coil. Current flowing in a large loop with multiple turns creates an magnetic field which reacts with the Earth's field to produce desired torques. The coil loop may be attached to the perimeter of a large structure or deployed using booms. Although these produce less concentrated magnetic fields than torquer bars, consideration must also be given to not locating magnetically sensitive devices near the large coil loop; however, moved six feet from the coils, the Earth's magnetic field predominates.

While magnetic torquing is considered primarily for desaturation of CMG's, other applications are available. Magnetic torquers can be used to provide a supplement to the primary control in the event CMG control authority is reduced or unavailable. Magnetic torquers provide a simple, low power, nonconsumable torquing alternative which will increase overall space station redundancy and reliability.

4.4.2.2.6 Reaction Control System (RCS)

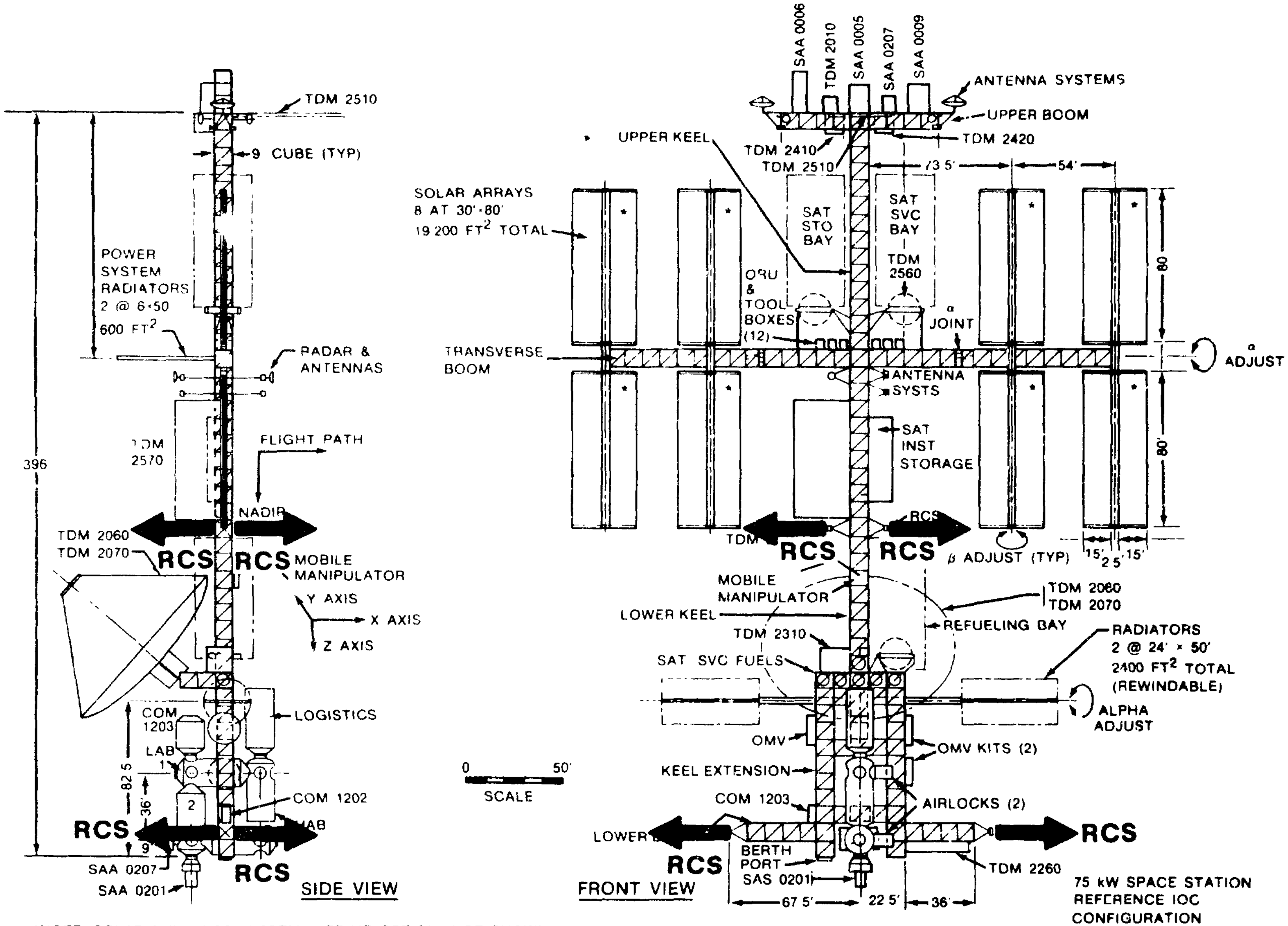
Reaction control will be achieved by means of appropriately placed thrusters which are capable of producing torques about three orthogonal body axes and translational forces. The thrusters will maintain attitude and translational control by time modulated commands as generated by the control algorithm. Placement of the thrusters should be such that impingement and contamination effects will be minimized. In order to keep the control laws simple and to avoid transporting propellent across rotating joints, the thrusters should be placed on a section of the structure which allows 3-axis control in translation and rotation.

The greatest reboost efficiency from the fuel consumption point of view can be achieved by firing only those thrusters which accelerate the Station along the velocity vector. The required thruster geometry would be for the center of mass to be inside the thruster envelope in order to produce the required torques. Attitude and translation control are decoupled by thruster modulation. The thruster envelope includes the effects of unequal thrust and lever arms, thrust misalignment and body flexing.

A force vector arrangement will be used to demonstrate a possible propulsion system geometry. Control redundancy has not been considered; however, the final design should include contingencies which insure that control will be maintained in the event of thruster failure. Thruster redundancy will be addressed in Section 4.4.5 of this report. The thruster arrangement for the Station is shown in Figure 4.4.2-2. The thrusters are represented by heavy arrows which indicate the force direction. Time modulation of those thrusters shown in the side view will provide for translation. Torque about all body axes is provided while the translational forces are parallel to the x-axis only. A total of 12 thrusters will be needed to satisfy the torque and force requirements. Since the flight path is always along the x body axis, no reorientation will be required for reboost.

The RCS will be used to transfer the Station to a circular orbit 20 nmi higher or lower than its present orbit in addition to a 90-day reboost as required due to atmospheric drag. The Station has a minimum operating altitude of 250 nmi and maximum operating altitude of 300 nmi with 270 nmi as nominal. The reboost and orbit altitude transfers will be performed immediately after Orbiter departs in order to maintain .00001 g's or less until the Orbiter returns. Because the 20 nmi transfer will not be utilized after every Orbiter visit, this propellent will be used as reserve propellent for other contingencies that will be discussed later. Reboost propellent will be based on an altitude transfer at Orbiter departure which will be sufficient for 90 days. The Orbiter is expected to rendezvous with the Station at the same altitude. The propellent is sized on the greater reboost at 250 nmi with nominal atmospheres or 270 nmi with two sigma atmospheres. Analysis has shown that 270 nmi with two sigma atmospheres is greater.

The RCS will be used to avoid collision with space debris, space station traffic, or natural space objects. Collision avoidance will not size the system; however, a propellent budget sufficient to impart a 5 ft/s velocity change in the station has been provided. If additional propellent is required, it may be obtained from the 20 nmi orbit transfer propellent, provided it has not been used. No propellent will be provided for station disposal (deboost), because it is assumed that the entire propellent load can be made available.



*NOTE SOLAR DYNAMICS SYSTEM A PRIME OPTION (NOT SHOWN)

Figure 4.4.2-2 - RCS THRUSTER LOCATION

The RCS will be the backup system for attitude control in the event that the CMG system becomes inoperative and will provide momentum management in the event that the magnetic torquers become inoperative. Failure of both the CMG system and the magnetic torquer system such that both systems are in the malfunctioned status at the same time is considered to be too remote to size propellent for both. Consequently the propellent reserve will be based on whichever requires the most propellent. Failure of the entire CMG system is considered an emergency, and the Orbiter is expected to bring up the necessary parts/personnel for repairs in 22 days. Consequently, the propellent will be sized for reduced pointing for 20 days awaiting Orbiter arrival, 2 days of nominal pointing for Orbiter berthing and 2 days after Orbiter arrival (with reduced pointing) for repairs. Sufficient propellent is provided for CMG spin up.

The magnetic torquer failures will not be considered an emergency situation, and the RCS will be sized to perform momentum management for 90 days realizing that the Orbiter may return early since the .00001-g environment will be violated with RCS in use. The RCS will also be used to perform attitude control augmentation in the event that the torque or momentum exceeds the capability of the CMG-magnetic torquer systems. No propellent has been provided for excesses except in the cases listed below; however, it is assumed that this propellent would come from the 20 nmi transfer budget providing that it has not already been used. All available control effectors (magnetic torquers and functioning CMGs) will be used to conserve propellent.

Some of the events that are expected to exceed the torquer/momentum capability of the CMG-magnetic torquer control systems are the Orbiter capture and subsequent berthing/docking and module berthing and/or change out; consequently, propellent has been budgeted for these events. The velocity difference between the Orbiter and the Station was assumed to be .1 ft/s in three orthogonal directions and .2 deg/s about three orthogonal axes. Two of the velocities were assumed to act on a 75-foot moment arm, and the third was assumed to be zero. The manipulator arm on the Station was assumed to require 5 accelerations and 5 decelerations along and about two axes for berthing the orbiter and each module with each acting on a 75-foot moment arm.

The required thruster force is sized by the maximum disturbance torque. The manipulator arm has been assumed to provide the highest torque. Since the Station arm has not been defined yet, it was assumed that the arm would be capable of a 25-lb force which would be 100 feet from the Station's center of gravity. Using two thrusters, each 50 feet from the center of gravity, a 25-lb thrust would be required.

4.4.2.2.7 Solar array actuator interfaces

There are two types of solar array actuators. The Alpha (α) actuator is required to maintain the solar panel longitudinal axis perpendicular to the sun line. Motion in this axis is due to the orbit of the Space Station about the earth. The α actuator turns at one revolution per orbit. The beta (β) actuator maintains the solar panel transverse axis perpendicular to the Sun line. Motion in this axis is due to both the Earth's rotation about the Sun and the nodal regression of the orbit. This actuator cycles approximately eight times per year.

The apparent load on these actuators is assumed to be largely an inertia load. In addition to inertia, these actuators must overcome stiction and rolling friction. Torques due to aerodynamic loads, plume impingement, dynamic motions of the Space Station, and the solar panels themselves are assumed to be negligible.

The α actuators will also incorporate four slip rings or roller rings to pass the electrical power generated by four solar panels, three to provide for data links to the β actuator drive electronics and three to provide for conditioned power to the β actuator drive electronics. The β actuators, in addition to their inertia loads, must overcome torques due to power and coolant lines.

The prime mover for both actuators will be brushless dc motors. The motor stator will be divided into three segments. Each segment will contain a complete three-phase winding to provide electromagnetic torque summing. The motors will be sized such that any two segments can drive the load. This technique provides a match between the three channel input from the GN&C system and the nonredundant output of the actuator. Alternate approaches include velocity summation of three independent motors using differential gears or other unique motor designs not presented herein.

4.4.2.2.8 Core radiator actuator interface

The alpha α actuator(s) for the core radiators will be approximately one foot in diameter. These actuators will maintain the radiator transverse axis normal to the Sun line. The radiators are unwound on the dark side of the orbit to eliminate the need for continuous rotation of a fluid joint.

The design of this actuator will be similar to the actuators described in paragraph 4.4.2.2.7.

4.4.2.2.9 Payload pointing systems

Table 4.4.2-1 lists selected payloads which have pointing requirements and shows the suggested location for the payloads. Two payloads (Pinhole Occulter, SAA0009, and Starlab, SAA006) have been recommended for the 28.5^{o} platform rather than the Space Station because of their severe pointing requirements. Also, the Pinhole Occulter Facility (50 meters in length) would prefer to be located on an inertially-oriented platform.

If the Station were pointed to within 1 degree, only the Earth Observation Instrument Technology Payload (TDM2260) would require a pointing system to meet the pointing requirement of 0.1 degrees. An inexpensive pointing system could be used to meet this requirement. It is also in the realm of possibility that the Station may be able to point that accurately, in which case, no pointing system would be required.

Two solar experiments (Space Plasma Payload, SAA0207, and Material Performance, TDM2010) mounted on the transverse boom require a single axis gimbal for angle correction. With the β angle gimbal, the Station pointing accuracy is sufficient to meet their pointing requirements of 1 and 2 degrees respectively.

With the present payload list (Table 4.4.2-1), only inexpensive pointing systems are required if two payloads (SAA0006, SAA0009) are relocated to the 28.5° platform. It is possible that future payloads which can be accommodated by the Station will require a payload pointing system in which case the Dornier IPS will be a good candidate. It appears that a new development for a pointing system for the station is unwarranted at this time.

The payload pointing mount will be interfaced to the GN&C system using a payload pointing mount interface electronics unit. The function of this unit will be to detect faults in the incoming commands from the GN&C system, isolate the fault, and reconfigure such that the payload pointing mount receives a valid coarse pointing command.

4.4.2.2.10 System sizing

The sensor and processors have been chosen such that at least fail operational/fail safe/restorable redundancy exists for all components. Cold components will be used where practical to reduce the power requirements. Control effectors have been sized 50 percent greater than the number of units estimated to be necessary while making certain that sufficient redundancy exits.

a. IOC station. Table 4.4.2-2 contains the quantity, size and weight of the components selected for the IOC Station. Table 4.4.2-3 contains the electrical power requirements.

b. Growth station. The growth Station has not been sufficiently designed and analyzed to determine the system size of the control effectors. Depending on the design of the growth Station relative to the IOC Station, the number of control effectors could either be reduced or increased. It is anticipated that the growth Station will require about the same or a slightly greater number of control effectors. The sensors and processors are anticipated to be the same as the IOC Station.

TABLE 4.4.2-1 - PAYLOAD ACCOMMODATIONS LOCATION

Orientation	ID Code	Payload	FOV	Pointing Accuracy Required
Station Zenith				
Anti-Earth	SAA00005	Transition Radiation & Ion Calorimeter (TRIC)	120°	10°
Station Nadir				
Earth	SAA0201	LIDAR	60°	1°
Earth	SAA0207	Space Plasma Payload (Tether)	360°	1°
Earth	TDM2260	Earth Obs. Instr. Tech.	-	0.1°
Earth	TDM2510	Environmental Effects	20°	2°
Transfer Boom (Solar Array)				
Solar	SAA0207	Space Plasma Payload	-	1°
Solar	TDM2010	Material Performance	-	2°
Inertial	TDM2410	Attitude Control Tech.	-	-
Inertial	TDM2420	Figure Control Tech.	-	-
28.5° Platform				
Solar	SAA0009	Pinhole Occulter Fac.	3°	10 arcsec
Inertial	SAA0006	Starlab	180°	2 arcsec

TABLE 4.4.2-2 - SPACE STATION GN&C EQUIPMENT LIST

Unit Description	Dimensions Per Unit (In.)	No. Units	Vol. Unit (Cu.Ft.)	Vol. Total (Cu.Ft.)	Weight Per Unit (Lbs.)	Weight Total (Lbs.)
Actuators						
CMG Assembly	42 Sphere	6	22.45	134.70	420.00	2520.00
Magnetic Bar	2.25 Dia.x98	6	0.23	1.35	109.00	654.00
Total Actuators				136.05		3174.00
Sensors						
Star Tracker Triad	13x13x12	2	1.17	2.35	20.00	40.00
Hexad Strapdown	12x12x12	1	1.00	1.00	50.00	50.00
Spares for above	12x12x12	1	1.00	1.00	50.00	50.00
Total Sensors				4.35		140.00
Electronic Support						
Magnetic Torquers	9x9x9	6	0.56	3.38	20.00	120.00
RCS Cont System	12x18x8	3	1.00	3.00	35.00	105.00
G&C Processors	7.5x10.5x15	3	0.68	2.05	20.00	60.00
NAV/Traffic Proc	7.5x10.5x15	3	0.68	2.05	20.00	60.00
Interface Devices	7.5x10.5x15	18	0.68	12.30	5.00	90.00
Solar Array Elec.	8x9x6	6	0.25	1.50	12.00	72.00
Spares for above	8x9x6	3	0.25	0.75	12.00	36.00
Radiator Elec.	8x9x6	2	0.25	0.50	12.00	24.00
Spares for above	8x9x6	1	0.25	0.25	12.00	12.00
Payload Elec.	2x5x6	2	0.04	0.08	3.00	6.00
Total Electronics				25.86		585.00
Grand Totals				166.26		3899.00

TABLE 4.4.2-3 - SPACE STATION GN&C POWER REQUIREMENTS

Unit Description	No. Units	Avg Pwr Per Unit (Watts)	Avg Pwr Total (Watts)	Peak Pwr Per Unit (Watts)	Peak Pwr Total (Watts)
Actuators					
CMG Assembly	6	100.00	600.00	125.00	750.00
Magnetic Bar	6	3.50	21.00	7.00	42.00
Total Actuators			621.00		792.00
Sensors					
Star Tracker Triad	2	10.00	20.00	12.50	25.00
Hexad Strapdown	1	100.00	100.00	125.00	125.00
Spares for above	1	0.00	0.00	0.00	0.00
Total Sensors			120.00		150.00
Electronic Support					
Magnetic Torquer	6	2.50	15.00	5.00	30.00
RCS Cont System	3	25.00	75.00	35.00	105.00
G&C Processors	3	50.00	150.00	65.00	195.00
NAV/Traffic Proc	3	50.00	150.00	65.00	195.00
Interface Devices	18	20.00	360.00	25.00	450.00
Solar Array Elec.	6	70.00	420.00	90.00	540.00
Spares for above	3	0.00	0.00	0.00	0.00
Radiator Elec.	2	70.00	140.00	90.00	180.00
Spares for above	1	0.00	0.00	0.00	0.00
Payload Elec.	2	25.00	50.00	35.00	70.00
Total Electronics			1360.00		1765.00
Grand Totals			2101.00		2707.00

4.4.3 Communications and Tracking System

4.4.3.1 Communications and Tracking System Definition

The Communications and Tracking System will be designed to provide communications and tracking services between the Space Station and ground. This is done via a relay satellite system, various space vehicles interoperating with the Space Station, as well as internal communications services. Interoperating vehicles for the initial phase will include the Space Shuttle Orbiter, EVA crewmembers, co-orbiting free-flyers, orbital maneuvering vehicles (OMV's), and a space platform. For the growth phase an orbital transfer vehicle (OTV) will be included.

In translating the general communications and tracking system requirements into a Space Station System Concept, it has been necessary to formulate some basic assumptions. These are summarized below:

a. Free-flyers will rendezvous and dock with the Space Station under control of an OMV. It is assumed that the Space Station will communicate with free-flyers out to 1080 nmi.

b. The co-orbiting space platform will be treated as a free-flyer for communications to/from the Space Station.

c. Space Station/OMV communications and tracking are required only out to 20 nmi.

d. Co-orbital vehicles located beyond 20 nmi. of the Space Station along the velocity vector will provide position data to the Space Station.

e. Tracking data will be updated on a near continuous basis.

f. Low data rate users will be required to provide at least 10 db of EIRP at 1080 nmi.

g. High data rate users will be required to provide at least 23 db of EIRP at 1080 nmi.

h. In addition to some duplicated hardware, redundancy requirements will be met by either electronic design, functional considerations and/or operational considerations at some reduced performance capabilities.

i. Selectively locating communications and tracking hardware throughout the Space Station will satisfy the requirement of safe haven.

Each link will provide a variety of services and will entail unique operational requirements. Multiple simultaneous links will also be required. Figure 4.4.3-1 shows an overall Space Station communications and tracking system block diagram.

The Space Station/ground uplink and downlink channels will operate through a relay satellite at S-band and Ku-band. The communication links between the Space Station and Orbiter will operate at S-band frequencies. The links between the Space Station and space platforms, free-flyers, EVA, OMV, and/or manned/unmanned OTV's will be at K-band.

The communication system will be capable of transmission, reception, and processing of voice, telemetry, commands, wideband data, television (TV), and text and graphics. The system will include the capability for private communications including Communications Security (COMSEC) requirements. Relay of interoperating vehicle data to/from the ground will be provided through the Space Station via a synchronous satellite.

The internal C&T system will provide for Space Station intermodule and intramodule C&T. Services provided by the internal C&T system include video, audio, commands, telemetry, data text and graphics, and C&T management/control/distribution. The C&T System hardware will have embedded processors that are software controlled for system configuration.

The C&T system will be developed and implemented incrementally to support assembly, initial and growth phases. The phases will incorporate a logical progression in C&T system capability matching the growth in Space Station C&T requirements.

A more detailed description of the communications and tracking functional areas (RF communications, audio system, video system, tracking systems, signal processing, data acquisition/telemetry/commands, systems monitoring and control, and antenna systems) is contained in the following sections.

Communication link characteristics are shown in Table 4.4.3-1 and tracking link characteristics are shown in Table 4.4.3-2. Performance requirements both external and internal are shown in Tables 4.4.3-3 and 4.4.3-4.

4.4.3.2 RF Communications

The Space Station Program RF system consists of dedicated RF links to TDRS, Orbiter, MRMS, and RF links with free-flyers, co-orbiting platform, EVA's and OMV.

An initial S- low data rate TDRS, GPS, and SS/Orbiter S-band capability wi. e provided during the assembly phase when the Space Station is unmanned. The S-band RF systems hardware may require relocation if after assembly there is significant antenna blockage or it is necessary to optimize performance by minimizing cable runs.

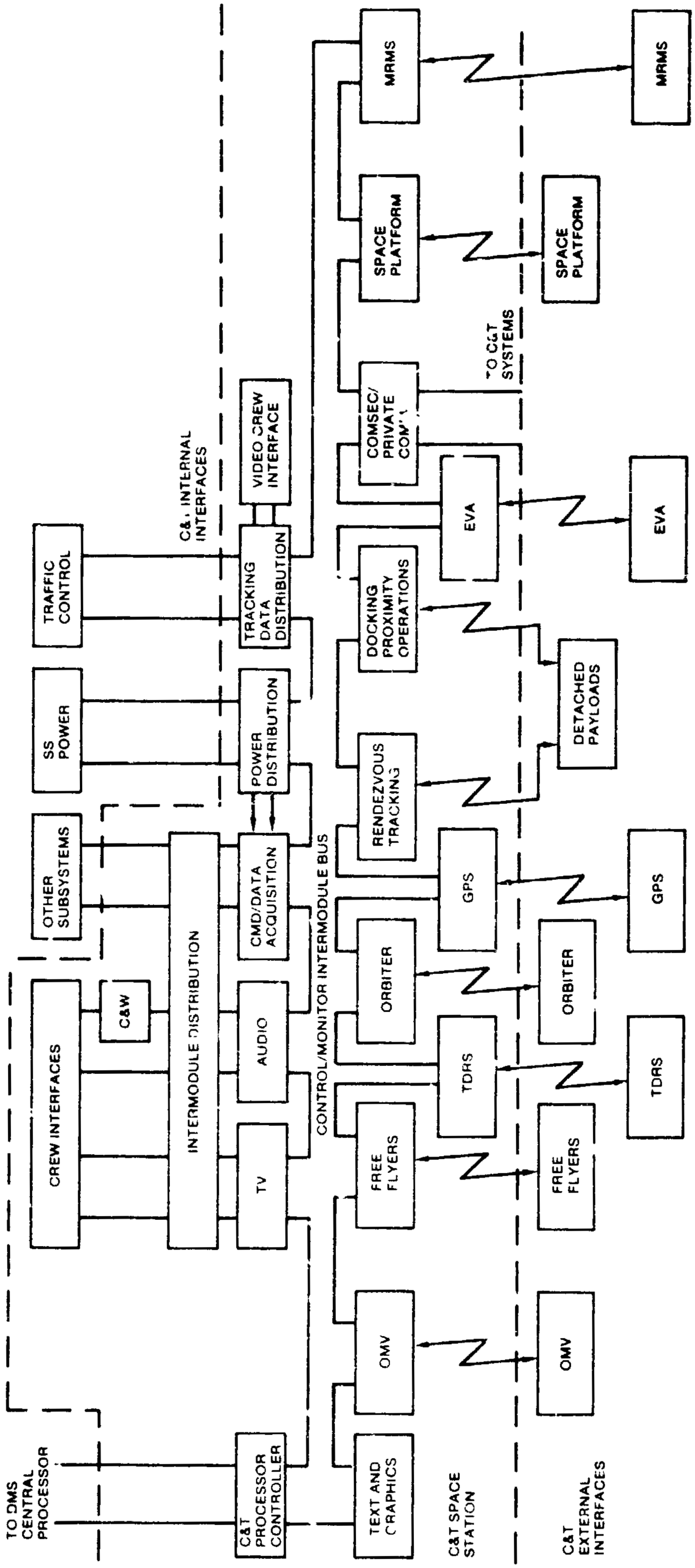


Figure 4.4.3-1 - SPACE STATION COMMUNICATIONS AND TRACKING SYSTEM

TABLE 4.4.3-1 - COMMUNICATION LINK CHARACTERISTICS

VEHICLE LINKS	FREQUENCY		CAPABILITIES		DATA RATE (MAX)		NO. OF VEHICLES (SIMULTANEOUS)	RANGE (MAX) (NMI)	COMMENTS
	XMIT*	RVC*	XMIT*	RCV*	XMIT*	RCV*			
SS-TDRS	KuBAND	KuBAND	VOICE, TLM TV, DATA	VOICE, CMD TV, DATA TEXT & GRAPICS	300-MBPS	25 MBPS	ONE	23,000	SINGLE ACCESS - TDRS COMPATIBLE
	S-BAND	S-BAND	VOICE, TLM	VOICE, CMD	3 MBPS	300 KBPS	ONE	23,000	SINGLE ACCESS - TDRS COMPATIBLE
	S-BAND	S-BAND	TLM	CMD	96 KBPS	32 KBPS	ONE	23,000	REDUCED DATA RATE DURING ASSEMBLY PHASE
SS-ORBITER	S-BAND	S-BAND	TLM	CMD	16 KBPS	2 KBPS	ONE	20	ORBITER COMPATIBLE
	K-BAND	K-BAND	VOICE	VOICE	16 KBPS	16 KBPS	ONE	20	UPGRADED ORBITER EVA SYS
SS-EVA/MMU	K-BAND	K-BAND	VOICE, CMD	VOICE, TLM	48 KBPS	64 KBPS	TWO	0.54	PROX. OPS. MULTI-ACCESS SYS.
			TV		400 KBPS				FREEZE-FRAME TV
	-	K-BAND	-	TV		25 MBPS	TWO	0.54	PROX. OPS HI-DATA-RATE SYS STANDARD TV
SS-FREE FLYER PLATFORM	K-BAND	K-BAND	CMD	TLM	48 KBPS	64 KBPS	ONE (EIGHT)**	1080	FAR-RANGE LOW-DATA-RATE SYS
	-	K-BAND	-	TV OR DATA	-	5 MBPS (25 MBPS)**	ONE (EIGHT)**	1080	FAR-RANGE HI-DATA-SYS CO-ORBITAL

TABLE 4.4.3-1 - COMMUNICATION LINK CHARACTERISTICS (CONTINUED)

VEHICLE LINKS	FREQUENCY		CAPABILITIES		DATA RATE (MAX)		NO. OF VEHICLES (SIMULTANEOUS)	RANGE (MAX) (NMI)	COMMENTS
	XMIT*	RCV*	XMIT*	RCV*	XMIT*	RCV*			
SS-OMV	K-BAND	K-BAND	CMD	TLM	48 KBPS	64 KBPS	ONE	20	PROX. OPS MULTI-ACCESS STS.
	-	K-BAND	-	TV	-	25 MBPS	ONE	0.54	PROX. OPS. HI-DATA-RATE SYS. STANDARD TV
SS(MRMS)-ORBITER	S-BAND	S-BAND	TLM	CMD	16 KBPS	2 KBPS	ONE	0.05	ORBITER COMPATIBLE
			TV		(4.5 MHZ)	-			
SS-MRMS	S-BAND	S-BAND	CMD	TLM	2 KBPS	16 KBPS	ONE	0.05	MRMS DEDICATED RF SYS.
				TV	(4.5 MHZ)				

* REFERENCED TO SPACE STATION.

** A TOTAL OF EIGHT VEHICLES WILL BE SERVED BY THIS LINK AT HIGH OR LOW DATA RATES OR A COMBINATION OF THE TWO RATES IN THE GROWTH CONFIGURATION

TABLE 4.4.3-2 - TRACKING LINK CHARACTERISTICS

VEHICLE LINKS	CAPABILITIES	FREQ.	DATA RATE	# OF VEH.	RANGE	COMMENTS
SS-ORB SS-OMV	RANGE RANGE-RATE AZIMUTH ELEVATION ANGLE RATE	Ku BAND	1/SEC DURING TRACK	ONE	20 NMI.	
SS-PLATFORM SS- FREE-FLYER	GPS NAV POSITION DATA ON COMM LINK	K BAND	1/10 SEC	EIGHT	1080 NMI.	GPS RCVR REQ'D ON EACH VEH.
SS-EVA	GPS POSITION ON COMM LINK	K BAND	1/SEC	TWO	3280 FT	GPS RCVR ON EACH MMU
SS-GPS	NAV POSITION DATA	L BAND	1/10 SEC	FOUR NAV-STAR	N/A	
SS-DOCKING VEHICLES	RANGE RANGE-RATE AZIMUTH ELEVATION ATTITUDE	OPTICAL	1/SEC	ONE	1000 FT	ONE FOR EACH DOCKING PORT

TABLE 4.4.3-3 - EXTERNAL COMMUNICATIONS AND TRACKING PERFORMANCE REQUIREMENTS

FUNCTION	PERFORMANCE REQUIREMENT	REMARKS
TELEMETRY WIDEBAND AND COMMAND DATA	BER $\leq 10^{-5}$	BER MEASURED AT EARLIEST BIT SYNCHRONIZER OUTPUT COMMAND AUTHENTICATION REQUIRED
VOICE DIGITAL	BER $\leq 10^{-5}$ VOICE RECOGNITION REQUIRED, VOICE INTELLIGIBILITY 90 PERCENT	BER MEASURED AT EARLIEST BIT SYNCHRONIZER OUTPUT VOICE INTELLIGIBILITY MEASURED USING HARVARD PHONETICALLY BALANCED WORD LIST
TEXT AND GRAPHICS	BER $\leq 10^{-5}$	BER MEASURED AT EARLIEST BIT SYNCHRONIZER OUTPUT
TELEVISION DIGITAL	BER $\leq 10^{-5}$, NTSC STANDARD DIGITAL TO ANALOG PROCESSOR, Spp/ Nrms 35 Db RESOLUTION = 250 LINES	

TABLE 4.4.3-3 - EXTERNAL COMMUNICATIONS AND TRACKING PERFORMANCE REQUIREMENTS (CONTINUED)

FUNCTION	PERFORMANCE REQUIREMENT	REMARKS
TELEVISION ANALOG	Spp/Nrms $\geq$ 35 Db, NTSC STANDARD RESOLUTION = 250 LINES	MEASURED AT THE TV TERMINAL
COMPUTER DATA	BER $\leq 10^{-9}$	BER MEASURED AT INPUT TO COMPUTER TERMINAL
LONG RANGE TRACKING	MAX RANGE - 1080 nmi COVERAGE - LIMITED TO COMM DATA LINK COVERAGE ACCURACY - (GPS POSITION) +/- 15M (49.2 ft)	
SHORT RANGE TRACKING	MAX RANGE - 20 NMI COVERAGE - 4 PI STERADIANS ACCURACIES: ANGLE - +/- 10 MRAD (0.57 DEG) RANGE - +/- 100 M (328 FT) OR 1% VELOCITY - .3 M/SEC (1 FPS) OR 1%	

TABLE 4.4.3-3 - EXTERNAL COMMUNICATIONS AND TRACKING PERFORMANCE REQUIREMENTS (CONTINUED)

FUNCTION	PERFORMANCE REQUIREMENT	REMARKS
PROXIMITY OPERATIONS TRACKING	MAX RANGE - 3280 Ft COVERAGE - LIMITED TO COMM DATA COVERAGE ACCURACY: GPS POSITION- +/-1M (3.3 FT)	
DOCKING SENSORS	MAX RANGE - 1000 Ft COVERAGE - 20 DEG CONE ACCURACIES: RANGE - +/- 0.5CM (.02 FT) ANGLE - +/- 2 MRAD (0.1 DEG) VELOCITY - 1.0 CM/SEC (0.03 FPS) ATTITUDE - +/- 10 MRAD (0.57 DEG)	

TABLE 4.4.3-4 - INTRA-SS COMMUNICATIONS
PERFORMANCE REQUIREMENTS

FUNCTION	PERFORMANCE REQUIREMENT	REMARKS
TELEMETRY COMMAND AND WIDEBAND DATA	BER $\leq 10^{-9}$	BER MEASURED END-TO END OVER THE INTENDED COMMUNICATION PATH
VIDEO HIGH-RESOLUTION DIGITAL	BER = 10^{-7} AT OUTPUT OF TV DIGITAL TO ANALOG PROCESSOR, Spp/Nrms = 48 dB RESOLUTION = 700 LINES	
NTSC DIGITAL	BER < 10^{-7} AT OUTPUT OF TV DIGITAL TO ANALOG PROCESSOR SPP/Nrms > 40 Db RESOLUTION = 350 LINES	
HIGH RESOLUTION ANALOG	Spp/Nrms > 48 dB RESOLUTION = 700 LINES	
NTSC ANALOG	Spp/Nrms >40 dB RESOLUTION = 350 LINES	
VOICE DIGITAL	BER < 10^{-5} VOICE INTELLIGIBILITY > 90 PERCENT. VOICE RECOGNITION REQUIRED 99%	WORD INTELLIGIBILITY MEASURED USING HARVARD PHONETICALLY BALANCED WORD LISTS
CONTROL/ MONITORING/ STATUS DIGITAL DATA	BER $\leq 10^{-9}$	BER MEASURED END-TO-END OVER THE INTENDED COMMUNICATIONS PATH

The Space Station RF system will provide spherical communications coverage out to 20 nmi. for near range operations using low gain hemisphere antennas. Figure 4.4.3.2 describes this multiaccess RF communication concept. For proximity operations out to 3280 feet a low data rate (LDR) multiaccess system will transmit simultaneously to two EVA's, commands, voice and freeze frame TV and receive voice and telemetry from both. To receive TV from an EVA a four channel high data rate (HDR) receiver is provided.

The system also provides LDR communications to/from an OMV transmitting commands and receiving telemetry out to 20 nmi. In addition, the OMV can transmit TV to the Space Station over the HDR channel within 3280 feet during docking maneuvers.

The Space Station will have two-way communications out to 20 nmi. with the orbiter. The system will be compatible with the present Orbiter S-band payload interrogator system providing telemetry and command capability. A voice capability will be provided utilizing the Space Station EVA link in conjunction with the upgraded Orbiter EVA system.

An RF system on the Space Station that provides commands to and telemetry/TV from a mobile remote manipulator system (MRMS) is shown in Figure 4.4.3-3. The system design is compatible with the Orbiter for operations during assembly and also will provide control capability when the Space Station is manned. Return TV to the Orbiter will use an FM link which will be compatible with the Orbiter's EVA TV S-band FM system.

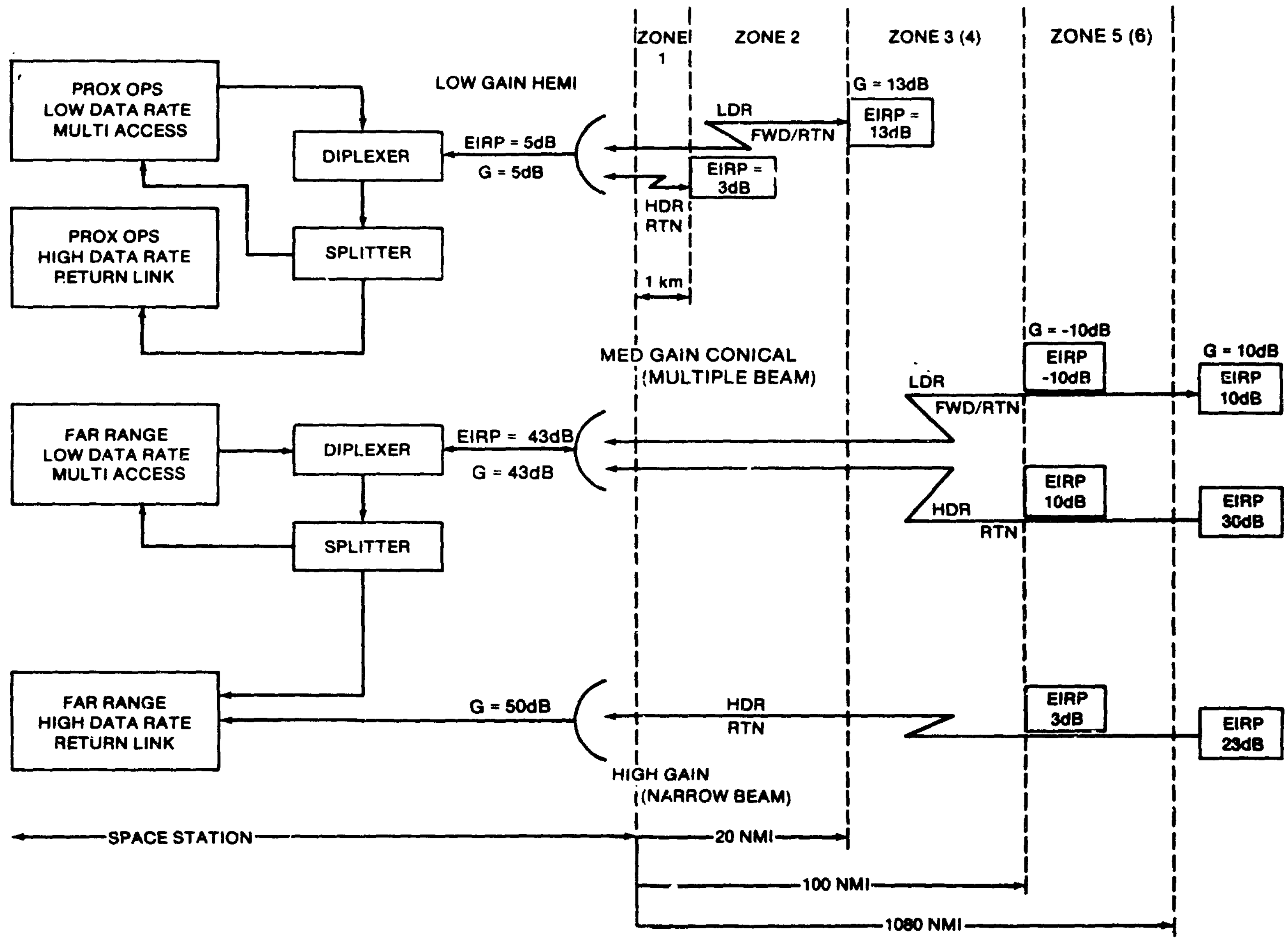
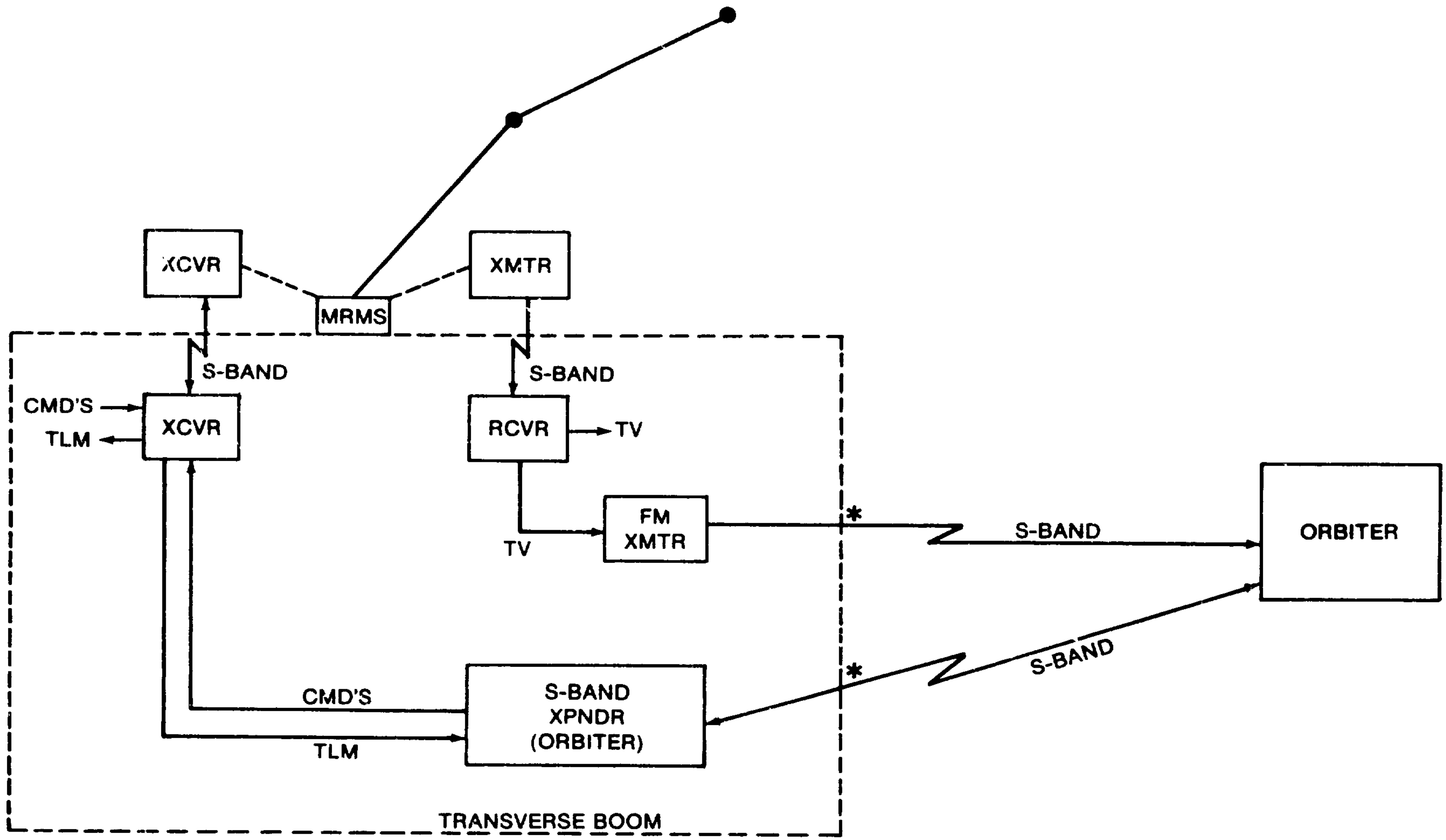


Figure 4.4.3.-2 - MULTI ACCESS RF COMMUNICATIONS FOR CO-ORBITAL ZONE COVERAGE.



*ORBITER COMPATIBLE LINKS

Figure 4.4.3-3 - MRMS RF SYSTEM CONCEPT

For the initial Space Station configuration a far range, single access, system will be used for coverage out to 1080 nmi. This system will transmit commands to co-orbiting free-flyers and platforms and receive telementry and data or slow scan TV from these co-orbiting free-flyers and platforms. Since this system utilizes a single beam steerable parabolic dish antenna, approximately 2.5 feet in diameter, only one user at a time is serviced.

This single access system will require the HDR users to provide an EIRP of + 23 db for coverage at 1080 nmi. for data rates of 5 mbps. The LDR users will be required to provide a minimum of +10 db for coverage at 1080 nmi. for a data rate of 64K bps.

The growth RF systems far range configuration as shown in Figure 4.4.3-2, will utilize a medium gain, multibeam, offset reflector antenna which will provide four beams out to 1080 nmi. This will allow for the simultaneous communications with eight free-flyers (four forward/four aft), one of which may be a co-orbiting platform. This multiaccess system will have a low data rate capability for transmitting commands and receiving telemetry and a high data rate capability to receive wideband scientific data or digital TV. HDR users will be required to provide a minimum of + 30 db of EIRP for data rates of 25 mbps at 1080 nmi.

This will be a four channel system. The LDR portion of the multiaccess system will require the users to provide + 10 db of EIRP at 1080 nmi. for a 64K bps data rate capability similiar to that of the single access system in the initial configuration.

The HDR system will be used in combination with the medium gain multibeam antenna to provide up to eight high data rate channels simultaneously (four-Fwd/four-Aft).

The respective EIRP's and receive antenna gains for the LDR multiaccess system were sized for a time division modulation scheme which requires a large received signal power. However, this design will support other modulation schemes such as frequency hopping, direct sequence, or frequency division. This could be more desirable if some other performance characteristics were considered primary. The HDR link, because of the wide bandwidths required, may be limited to a frequency division modulation technique.

The Space Station TDRS link will include both S-band and Ku-band single access links. The S-band and Ku-band links will have the capability to use maximum TDRS data rate capacity. A low data rate capability at S-Band is provided as a backup.

Power amplifiers, low noise amplifiers, and up/down converters will be located at the antennas to obtain optimum performance. All electronics mounted on or near the antennas will be redundant because of the difficulty of repairing and/or replacing components. IF signals from a module will be routed from their respective modulation sources to the antenna electronics for up conversion to the specified RF frequency. The down converted signals will be routed to their respective demodulators located in a module. Distributing the C&T hardware in such a manner will permit the system to be improved, reconfigured or expanded without modifying the entire RF subsystem.

4.4.3.3 Audio System

The audio system processes and distributes various sources of crew voice and other audio signals throughout the Space Station modules, berthing ports and airlocks. Intercom and paging channels are provided along with duplex voice channels to the ground, and to EVA/MMU's and Shuttle Orbiter(s). Speech recognition (voice controlled commands) and speech synthesis is provided along with the capability for recording, for CCTV usage, for recreational purposes and private lines. Distribution of caution and warning tones is also furnished.

Each Space Station module will contain a largely self-contained all digital system for audio generation, processing, switching, and distribution. Duplex, multiaccess TDM channels are used to accommodate a crew size of six for the initial phase and 16 crewmembers in the growth configuration.

Bandwidth compression techniques will be used to limit external transmissions to 16 Kbps per voice channel (Orbiter channel to remain at 32 Kbps) while still retaining good "toll quality" voice fidelity. Capability for private voice transmission (localized encryption) will be provided for medical, family, or proprietary conversations (compatible decryption devices must be provided by each listener on the ground).

A crew-mounted wireless communicator will be used to transmit crew voice and to receive onboard and ground audio signals. Wall-mounted speaker/mic units will also be supplied for hands-free operation.

An emergency communications unit provides for intermodule voice communications (short-range RF system) during an audio circuit loss via the primary system.

Local audio terminals in each module contain voice recognition units which can decipher voice commands and distribute encoded digital words to the display and control (D&C) system. A recognition accuracy of at least 99 percent will be met. Voice synthesis units will also be provided such that digital command words from the D&C system can initiate a good quality synthesized voice output to the crew.

Since the audio system will be frequently reconfigured by the crew (channel selection, transmit/receive modes, volume levels, privacy control, etc.) various switching functions will be localized at the particular local audio terminals in the imme iate vicinity of the working crewmember.

Due to on-orbit permanency of the Space Station, the audio system is designed to maximize flexibility in hardware and system capabilities for distributed control and processing. Realtime adaptation and online reprogramming capabilities will be provided.

Interfaces with the D&C system for caution and warning tones and voice command/synthesis signals, with the onboard recording system, with the CCTV system, and principally with the C&T network signal processors and associated RF transmission/reception systems are handled through interface hardware external to the audio system.

4.4.3.4 Video System

Video service will be provided in the appropriate Space Station elements for such operational tasks as docking, tracking, construction, and other monitoring tasks; to provide recreational/entertainment/leisure, to record and store video, and for training.

The configuration, safety, and functional requirements of the Station call for control stations in each habitable module so that different crewmembers can perform their required tasks with minimal or no interruption to or from others. The system will be simple to operate since there will be a large number of users specialized in many different fields, and special training for Space Station equipment is kept to a minimum.

These requirements drive the design of the television system to a distributed control system where camera controls, video switching, and other system functions will be controlled from any workstation or monitoring location. This distributed control station concept will allow continuous operation even if parts of the Station become uninhabitable. These workstations will incorporate user-friendly input devices such as touch screen sensors, joysticks, and voice control inputs used in conjunction with color graphics generated menus, and displays. The capability to move TV monitors from one location to another will be incorporated.

Video signals, both black and white and NTSC color will be distributed aboard the Station and for transmission to/from the ground. The format for video signals will be both digital and analog. Large video bandwidth capability will be available onboard for high definition television.

Solid state video cameras will be located in each module; some with pan/tilt capability. External cameras will be located in appropriate locations for viewing and visual tracking of external activities. Two cameras will be provided at each docking port to aid the laser docking system.

4.4.3.5 Tracking and Berthing/Docking System

The tracking and berthing/docking system hardware will primarily be based on function by operational zones in accordance with the table below:

Far Range	0-1080 nmi.
Near Range	0-20 nmi.
Proximity Range	0-3280 ft.
Docking/Berthing Range	0-1000 ft.

The far range position information will be obtained by providing a GPS receiver/processor on each vehicle to be monitored, and transmitting the navigational data to the Space Station utilizing the communications return link. The C&T processor will combine this data with the onboard GPS information to provide relative position with the Space Station.

The near range zone tracking will be accomplished with a radar consisting of a transmitter/receiver RF assembly, a signal processor, and a 3-foot parabolic dish antenna. Two complete systems will be required to provide adequate coverage. The primary purpose of this radar is to track approaching vehicles as they enter the near range zone, through the proximity operations zone, to a position within the acquisition region of the docking system. It will also monitor departing vehicles in a similar manner.

The proximity operations will involve simultaneous tracking of two EVA/EMU/MMU's. A GPS receiver/processor on the MMU will supply position information to the Space Station via the communications data link. A small ranging radar will also be a part of the MMU to give the astronaut range and range-rate information to any object in front of him.

Each docking/berthing port on the Space Station will include a laser docking system that will provide range, range-rate, angular position, and attitude of the docking vehicle. Small passive retro-reflectors will be required on each vehicle in a standardized pattern to facilitate the attitude data.

A multichannel GPS receiver/processor will be included on the Space Station to provide positional data to the Space Station navigation system.

A Ku-band radar transponder will be included to provide augmented tracking for the Orbiter Rendezvous Radar.

Structural stability monitor sensors, using laser ranging techniques, will be placed at appropriate locations on the Space Station structure with corresponding retro-reflectors. This will provide flexure and bending data for structural integrity monitoring, relative structure alignment with the Nav-base, and for fine-tuning of antenna pointing, as required.

4.4.3.6 Signal Processing

Signal processors to interface with TV, text and graphics, digitized audio, command, telemetry, and data acquisition systems for the processing and protocols necessary for optimum operation of these systems will be provided. These processors will provide the equipment necessary for synchronization, encoding and decoding, error detection and correction, routing and distribution of signals throughout the Space Station and Station elements.

Command authentication will be processed for command channels and encryption/decryption and key distribution for secure links will be provided.

The capability to receive and transmit text and graphic hardcopy will be available.

Dedicated system processors for the various elements (free-flyers, OMV, TDRSS, etc.) will be provided for efficient handling of these links. The processor hardware will be distributed throughout the communication and tracking system to optimize overall performance.

4.4.3.7 Data Acquisition, Telemetry, and Command

The command/data acquisition system processes and validates uplink commands, processes commands to detached vehicles, and processes Space Station, detached vehicle, and experiment data for onboard distribution, down-link telemetry, or onboard recording. These functions will be accomplished by the use of local data bus networks to interconnect the elements and transport information to various users.

Uplink operational commands authentication processing and formatting for use in autonomous control and management when the station is unmanned and for use by the facility management system when manned, is performed.

Capability to encode and process onboard originated commands and wide band data for transmission to detached vehicles, including the Shuttle orbiter shall be provided.

Uplink science commands will be processed for transmission to onboard and detached vehicle experiment packages. This information may be time-shared with other uplink data if advantageous.

Operational, engineering, and science data from various onboard and detached vehicle sources will be acquired, processed, formatted, and interleaved as necessary for distribution onboard and down-link transmission via TDRSS.

Provisions for packetization of data sources (either at the source or at the point of multiplexing) shall be provided to optimize end-to-end data handling and distribution to experiment data users at widely dispersed locations.

To minimize hardware and software impacts from requirements changes and equipment upgrading distributed processing and control shall be emphasized. Maximum flexibility in hardware and system capability should be a major goal in the design of the system. Realtime adaptation range changes capabilities, reprogramming of data formats including deletions and addition of data sources will be required for the system. The ability to accommodate different redundancy requirements for numerous systems by the simple addition or deletion of hardware/software should be included in the design.

4.4.3.8 Systems Monitoring and Control

The basic monitoring and control concept is for hardware (internal and external) to be controlled and monitored by a dedicated C&T processor/controller. The C&T processor will interface with the Space Station regional data bus. The C&T processor wild perform system level tascs via a local system data bus dedicated to the C&T system. The local data bus will be routed throughout the Space Station because of the wide distribution of C&T hardware. The processor tasks include LRU configuration control and verification, LRU status, fault monitoring and distributing control information to the audio and video distribution control centers. The C&T processor will format information to be distributed to the data management system, such as LRU operational status, system configuration, etc.

Local processing within each LRU will be required to decode and process commands from the C&T processor. Status/fault detection will be performed at the LRU level.

4.4.3.9 Antenna Systems (see Sections 4.3.6.2 and 4.3.6.3)

4.4.4 Information and Data Management

4.4.4.1 Introduction

4.4.4.1.1 Overall concept

4.4.4.1.1.1 Distributed processing approach

The Space Station Information and Data Management Subsystem (IDMS) will employ a distributed architecture that allocates processing services resident in each subsystem. The IDMS will consist of two elements: a core (housekeeping) services element providing data distribution, display and control, time and frequency reference, and short-term core data storage services to subsystem and payload users in the space and ground environment; a user services element to receive and transmit user data, without preprocessing of this data, in the space and ground environment.

The IDMS concept will center around an Optical Data Distribution Network (ODDNet) through which each subsystem or payload transfers information to/from its own effectors and sensors and to other subsystems (e.g. Communications and Tracking) (see Fig. 4.4.4.1-1). Each subsystem will use standard Subsystem Data Processors (SDP's) running applications programs developed by each subsystem. The SDP's will interface to the ODDNet through Interface Devices (ID's) which will provide all inter-and intradata transfers between and within the modules. The Connection Device (CD) allows the ODDNet to be distributed between modules. Subsystem data may be input into the IDMS through the ID if a SDP is not required.

4.4.4.1.1.2 Growth

The distributed processing approach allows for the evolutionary growth of the IDMS to support the incremental expansion of the Space Station. The addition of new modules should easily be accommodated by adding more SDP's and ID's to the ODDNet. As new technology increases the capability of the IDMS. elements may be replaced with higher technology elements which will offer greater capability with decreased size, weight, and power requirements.

4.4.4.1.1.3 Commonality

The IDMS concept will employ common elements in the form of ID's, SDP's, and executive software overhead distributed through all subsystems and modules. This level of commonality extends to both the co-orbiting and polar platforms, as well as the Space Station and the ground.

4.4.4.1.2 Environmental considerations

The IDMS distributed concept is based on the assumption that cosmic rays and charged particles will affect the performance and longevity of the various IDMS related data handling devices. This, together with the Earth-based, and Space Station operations related, induced EMI environment drives the redundancy, maintainability, and processing concepts. Specifically, error protection, measures to protect against error and fault propagation, and diversity of location will be fundamental parts of the distributed IDMS concept to protect against the anticipated environmental effects.

4.4.4.1.3 Data protection

The IDMS concept will be to protect data by partitioning subsystem processing to individual, subsystem resident devices. The ODDNet will prevent unintentional radiation of Space Station data and preclude interference by intentional and unintentional sources. Relying then, on encryption/decryption techniques for protection of the RF link input/output interface with the IDMS, the IDMS will be assured of operating in a hostile environment without upset. Those users requiring protection of proprietary or classified data/functions will be required to provide the essential barriers that preclude unauthorized data access.

4.4.4.2 IDMS Architecture

4.4.4.2.1 Common element

4.4.4.2.1.1 Optical data distribution network (ODDNet)

All information and data transfer between and within Space Station modules will be made via the ODDNet, a network of optical fiber cables convieniently accessable everywhere in the Space Station. The architecture of the ODDNet will be designed to include sufficient redundancy, or fault tolerant qualities, to be consistent with Space Station requirements. This high capacity data transfer medium will handle all the information needs of the Space Station Initial and growth configurations.

ODDNet compatibility and connection will be accomplished by ID's and SDP's.

4.4.4.2.1.1.1 Interface device (ID)

ID's will be used to make all connections to the ODDNet providing all inter- and intrasubsystem/payload data transfer between Space Station modules and within modules where short cable runs are obviated. IDs may also be used where the ODDNet is available, to make subsystem connections to remotely located sensors and effectors. Connection to the ID will be by electrical connector from the SDP's. Users with special high speed data requirements may be provided with optical connections to the network. Such users would then be provided with an ID to be located within their equipment enclosure.

Each ID will contain an optical transceiver and the necessary processing electronics to provide coupling to the ODDNet consistent with its fault tolerant design. IDs are commonality items, each designed for a single subsystem/payload user. Internal fault tolerance within the ID will facilitate redundant system connections.

4.4.4.2.1.1.2 Subsystem data processor (SDP)

Each subsystem/payload user will be responsible for its own computational, logic and storage needs using a SDP composed of standardized modules or slices commonality items). A standard slice (to be a part of each subsystem element with direct connection to the ODDNet) will format, code and/or sequence data for transfer compatibility with the network. Additional common element slices will be devoted to subsystem peculiar processing, computation, storage etc. as necessary. Specialized processing outside SDP capabilities will be the responsibilty of the user.

4.4.4.2.1.2 Multi-purpose applications consoles (MPAC's)

The Multi-Purpose Applications Console (MPAC) will provide the man/machine interface onboard the Space Station. These MPAC's shall contain multifunctional display screens and controls to assist the crew in their duties and to alert the crew of any catastrophic failures.

There will be two types of MPAC's, fixed, and portable (see Fig. 4.4.4.2-1). The difference between these two types is that fixed MPACs will be used for routine operations while the portable MPAC's will be designed to handle operations located away from fixed units, such as maintenance or operations where direct outside viewing through a window is desired. Both MPAC types will have multifunctional display screens and programmable controls. There should be no dedicated displays on the Space Station MPAC.

Resident in the fixed MPAC will be the capabilty to print data and graphics from the display screen. The crew will have the capability to plot timed events data which will be selected from the MPAC. The operator will also be able to choose between raw and processed data. In addition, a method for recording video images will be provided.

The design of the Space Station MPAC must take into account the zero-g environmental effects and astronaut positions. Granted that a local vertical is desired, a one-g rigidity in the design may not be desired. For example, the display screen may be positioned to any astronaut orientation.

4.4.4.2.1.3 Software

4.4.4.2.1.3.1 Network operating system (NOS)

The IDMS NOS will consist of a set of common software provided to all subsystems to perform the distributed protocol, formatting, data handling, control, and display functions common to all subsystems as determined by the IDMS architecture. The NOS basis will depend on a common High Order Language (HOL) and common processor (slices) to be successful. The approach will be to base the NOS on a highly structured concept that partitions functions based on communication between functions at a high level.

4.4.4.2.1.3.2 Operational

Software for operations, conceptually, will consist of a subset of the NOS where routine data acquisition, formatting, health and status reporting, control I/O, and maintenance functions occur as common elements of all subsystem software.

4.4.4.2.1.3.3 Applications

Application software commonality will exist through the IDMS concept of a HOL interpreter and program writer. Each subsystem will be able to write application software in the standardized HOL. The interpreter will convert the application software to the structured HOL to be compatible with the NOS.

4.4.4.2.1.4 Standards

The use of standards will be consistent with the established commonality requirements. Standards will minimize the types of spares used on Space Station as well as the types of interfaces. This will allow the use of standard test equipment as well as a "remove and replace" philosophy. At all possible times, standards will avoid multiple, functionally equivalent types of hardware.

For example, one area of standardization will be MPAC's. The same hardware will be used on the fixed MPAC's for displays and controls, even though the functions of the displays and controls will vary. In addition, the symbology of the displays should be standardized, such as a flashing red number would indicate a sensor out of limits. Another area to benefit from standards will be software. Reducing the number of languages will allow consolidation of training, compiler development, and compiler maintenance.

4.4.4.2.1.5 Government-furnished equipment (GFE)

Subsystem and Payload users will be provided a portable MPAC with the necessary ID's, SDP's and a GFE test set. The purpose of this GFE will be to allow system level access and control to the Subsystem/Payload at the Vendor's plant and to assure ODDNet and IDMS interface compatibility. The test set will provide only those services necessary to functionally simulate the IDMS. These simulations will include the ODDNet, Time and Frequency functions, Data Storage and the NOS protocol.

4.4.4.2.2 Core subsystems (housekeeping)

4.4.4.2.2.1 Subsystem partitioning

The objective of subsystem partitioning is to minimize the number of interfaces by grouping (partitioning) like, interrelated functions into subsystems. IDMS partitioning will be consistent with the distributed, stand alone processing approach described in section 4.4.4.1.1.1. Subsystem users will interface with the ODDNet by way of ID's and SDP's (sections 4.4.4.2.1.1.1 and 4.4.4.2.1.1.2). Access and principal operation will be accomplished using MPAC's.

4.4.4.2.2.2 Core services

4.4.4.2.2.2.1 Data storage

Adequate mass storage capability will be provided by the IDMS for the collection of Station level status concerning subsystems, inventory, and proximity activities. General Space Station capacities, margins and limits for the purpose of records, trend analyses and distibution will be stored as needed. Periodically, stored data will be downlinked to the ground for archival purposes.

4.4.4.2.2.2.2 Time and frequency reference

The IDMS will provide time and frequency references, distributed to all subsystems simultaneously. The NOS time tags all subsystem I/O transactions using the distributed time and frequency reference to assure time correlation of all data. The distributed frequency reference provides a stable source for all subsystems and payloads.

4.4.4.2.2.2.3 Facilities management

Facilities Management is an IDMS service that will be available to all subsystems and payloads. This service will be responsible for the manangement and sequencing of IDMS data storage and access and Station level data analysis and ODDNet management. In addition, this service will be responsible for Station level configuration management including logistics, resource allocation, and scheduling data for the crew's use. Tutorials for refreshing the crew on procedures will also be provided from this service.

4.4.4.2.2.2.4 Data Base Management

A Data Base Management System (DBMS) will be provided to support the convenient and effective storage, exchange, manipulation, and retreival of data by all appropriate Station subsystems and users. This DBMS will operate within the IDMS NOS architecture so as to provide multiple, concurrent access to onboard data with bandwidth and response time characteristics sufficient to support Station requirements. The system will require minimal manual administrative support, thereby providing such automatic features as recovery from machine and user errors; up-load and down-load of data with appropriate ground facilities; elimination of redundant and insignificant data; and data partitioning and integrity protections. Additional features to be provided by the DBMS include multiple views of data; temporary workspaces for data manipulation; a natural language mechanism for online query from the MPAC's; and a HOL interface for operations and applications usage. The system will incorporate advanced data base management techniques, such as the use of the relational data model, which support data element types necessary to scientific, engineering, process control, Station housekeeping, and other tasks which must be supported by the IDMS. In particular, the system must support a full set of geometric entities for use in the MPAC display of three-dimensional objects, animation, CAD, and two-dimensional graphical data representations.

4.4.4.2.2.3 ODDNet management

A distributed processing concept will be used for the management of the ODDNet. Protocols and priorities may be preprogrammed either in the ID or a common module of the SDP. Override of selected functions may be commanded by the Facility Management unit or manually via a MPAC.

4.4.4.2.2.4 User interfaces

As in subsystem interfaces, user systems will interface with the ODDNet by way of ID's and SDP's. Access and principal operation will be accomplished using MPAC's.

4.4.4.2.3 User accommodations

The IDMS will provide a set of standard terminals and displays (MPAC's) and access devices (ID's and SDP's) which will satisfy the requirements of most users. However, some users will have special requirements beyond that provided in the standard system, e.g., unique, dedicated processors and sensors, encryption devices, research instrumentation and preprocessors, payloads/experiments, special-purpose controls, etc. In cases such as these, the user is expected to provide for his specialized needs, and he is encouraged to use the common module and ID's wherever possible in developing his system.

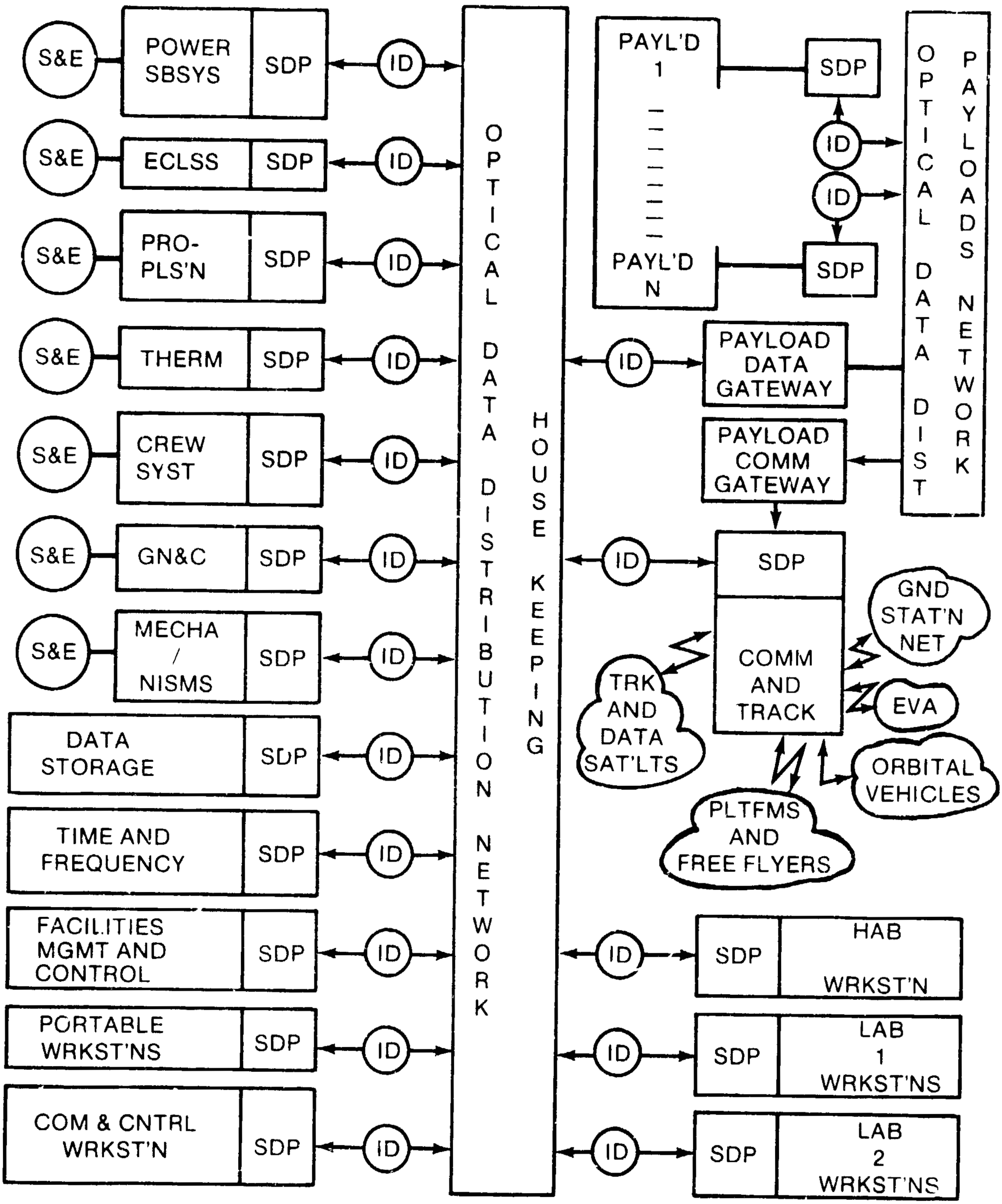


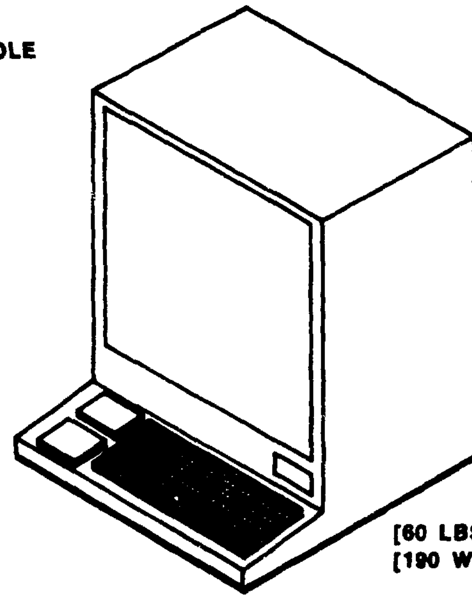
Figure 4.4.4.1-1 - INFORMATION AND MANAGEMENT SUBSYSTEM SPACE STATION

MULTI PURPOSE APPLICATIONS CONSOLE
(WORK STATION)

BASIC TERMINAL

24" FLAT PANEL SCREEN

MULTI WINDOW

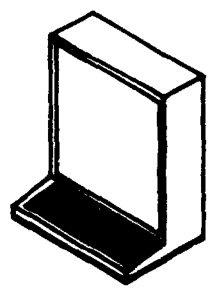


WORK STATION

- MULTI WINDOWS
- REPLICATION FOR EXPANSION
- VIDEO/DIGITAL/TEXT/GRAPHICS CAPABILITY
- FLAT PANEL SCREEN
  - LOW POWER
  - HIGH RESOLUTION

- MAIN CREW INTERFACE
- BUILDING BLOCK
- CAN BE USED BY CORE OR USERS
- VOICE I/O
- DETACHABLE KEYBOARD

MULTI PURPOSE APPLICATIONS TERMINAL
(PORTABLE TERMINAL)



PORTABLE TERMINAL

- MULTI WINDOWS
- SAFE HAVEN
- MAINTENANCE
- CREW TRAINING
- ENTERTAINMENT
- FLEXIBILITY IN LOCATING TO SUPPORT OPERATIONS/USERS
- GFE TO SUBSYSTEM CONTRACTORS FOR USE IN DDT&E, SOFTWARE DEVELOPMENT ALSO USERS/PAYLOAD DEVELOPERS

Figure 4.4.4.2-1.- WORK STATIONS

4.4.5 Propulsion

4.4.5.1 Introduction

The propulsion subsystem concept selected for Space Station configuration evaluation was monopropellant hydrazine, which represents state-of-art design, low initial cost, and modest performance. This concept is described herein for the reference configuration IOC Space Station.

The impact on several Space Station configuration alternatives for this baseline propulsion subsystem was examined. The propulsion subsystem was developed only to the extent necessary to estimate physical characteristics (weights, volumes, etc.) and to identify significant design and operational features that would bear on the relative merit of Space Station configurations. Configuration impact comparisons focused on Space Station buildup and operational scenarios, propellant resupply strategy, hardware commonality and maintainability.

Propulsion subsystem impulse requirements for the Space Station configuration were provided by the GN&C group. For these comparative studies it was implicitly sought to minimize the use of propulsion for control. Propulsion was assumed required only for altitude maintenance/reboost, relief of torques that lie outside the capability of the momentum management system, collision avoidance, and backup to loss of primary attitude control (CMG's). Primary attitude control was assumed to be maintained with control moment gyros and magnetic torque devices. No detailed evaluation was performed at this time on propulsion evolvability for growth station, propulsion strategies other than those provided by the GN&C group, or propulsion subsystems other than blowdown hydrazine thrusters (75 to 25 lbf).

Other propulsion concepts may be more beneficial to the overall Space Station program by offering reduced resupply costs, integration with other subsystems, the ability to efficiently dispose of unneeded consumables, the enabling of more efficient strategies for formation flying with platforms, reduced contamination, and reduced overall life-cycle costs. Representative of such subsystems are hydrogen-oxygen, warm hydrogen, and multifuel, low-thrust resistojets. Additional comments are provided in paragraph 4.4.5.5.

4.4.5.2 Subsystem Requirements

The propulsion subsystem specific requirements are intimately related to numerous other Space Station subsystems and functions. Typical of these requirements are those arising from the GN&C, operations, resupply, commonality, reliability, maintainability, payload accommodations, contamination, and safety. These requirements appear throughout the RFP and are collected here for convenience.

4.4.5.2.1 Propulsion requirements

The propulsion subsystem is required to perform the altitude maintenance and attitude control maneuvers. However, for the reference configuration, the CMG's and magnetic torquers are assumed to be the primary control effectors for performing the attitude control function. Therefore, the propulsion requirements were assumed to fall into two major categories: those that can be done only by the propulsion subsystem, and those in which the propulsion subsystem operates in the event that another subsystem fails and a backup capability is required. The primary requirements are

(1) reboost and orbit adjustment
(2) collision avoidance
(3) accommodation of disturbances resulting from docking, berthing, and movement of objects by the manipulator
(4) maintenance of an adequate propellant reserve margin

The backup requirements are

(5) three-axis stabilization in the event of failure of momentum exchange devices (in this event, control is provided at a reduced level)
(6) desaturation of the momentum exchange devices

4.4.5.2.2 Guidance, Navigation and Control requirements

The GN&C requirements and strategy dictate the propulsion subsystem design and operation. The input for the GN&C comes from the operational requirements and the payload accommodation needs. Requirements applicable to propulsion are

(7) all credible failures of critical systems shall have fail operational/fail safe/restorable levels of redundancy except for pressure vessels

(8) selective or total inhibits shall be available for EVA operations

(9) propellant usage data base shall be provided

4.4.5.2.3 Operational requirements

The operational requirements pertain to restrictions and needs based on the overall Space Station. Those requirements pertinent to propulsion are

(10) the Space Station shall not perform any maneuvers during docking/berthing operations

(11) maintenance operations including resupply shall consider EVA a limited resource

(12) maintenance operations shall minimize the need for special skills such as soldering and welding or time consuming procedures

(13) removal, repair, or replacement of equipment shall be at the ORU level without the need for special fixtures

(14) subsystems shall be as functionally independent as possible to facilitate maintenance

(15) subsystems shall be automated to the fullest extent possible

(16) the program shall support the rapid assimilation of new technology without requiring major redesign or revalidation

(17) the station will be resupplied by the Orbiter on a 90-day basis

4.4.5.2.4 Customer accommodation requirements

One critical customer accommodation requirement that impacts the propulsion subsystem is

(18) the steady state gravity level of the station shall be designed for .00001 g's or less

4.4.5.2.5 System requirements

Requirements placed on all subsystems are

(19) SSPE's shall have the ability to remain operational indefinitely through periodic maintenance

(20) onboard spares shall be provided

(21) maintenance of ORU's shall not introduce hazardous conditions

(22) contaminants from external sources shall be limited

4.4.5.2.6 Safety requirements

Safety requirements that have a particular impact on the propulsion subsystem are

(23) materials in the habitable volumes shall not outgas toxic constituents

(24) toxic fluid storage or lines shall be external to the pressurized volumes

4.4.5.3 Propulsion Subsystem Physical and Functional Description

The propulsion subsystem consists of four thruster clusters, multiple propellant storage tanks and a propellant supply system. A schematic of the propulsion subsystem is shown in Figure 4.4.5-1.

The propellant tanks utilize a 3 to 1 blowdown pressurization scheme. The nitrogen pressurant gas is separated from the hydrazine by a positive expulsion diaphragm. The four thruster clusters are located in a pattern that accommodates the varying locations of the Space Station center of gravity. Thruster cluster locations are shown in Figure 4.4.5-1. Thruster firing directions, shown in Figure 4.4.2-2, provide torques about the three axes of the Space Station and translational capability along the x-axis.

The fail-operational/fail-safe redundancy requirement dictates three thrusters in each firing direction on each of the four thruster cluster modules. This results in a total of 36 thrusters. The line routing and isolation capability for these multiply redundant thrusters and a method of stacking three identical cluster modules for triple redundancy are shown in Figure 4.4.5-2.

In addition to installation of multiple thrusters in each cl[illegible] for redundancy, consideration must be given to thruster life and [illegible]sion for spares. The x-axis thrusters used for periodic reboost will likely reach end of life conditions first. One way to accommodate this circumstance would be to provide in-place spares, that is, additional thrusters in the x-direction at each location.

Impulse requirements and propellant quantities are shown in Table 4.4.5-1. The impulse requirements for reboost and for Orbiter docking disturbances must be met at 90-day intervals; thus propellants are resupplied at each Orbiter visit. This quantity is approximately 2,300 pounds. The contingency impulse requirements for altitude transfer, collision avoidance and backup attitude control must be met only if the situation arises. Thus propellants must be maintained onboard for that purpose. This quantity is approximately 4,700 pounds. The 90-day resupply propellant quantity sized the resupply tank volume; the contingency propellant quantity sized the onboard tank volume, as shown in Table 4.4.5-2.

Propulsion subsystem performance is also shown in Table 4.4.5-2 along with other significant system features and characteristics. A single thrust level of 25 lbf minimum was selected based on control. This provides simplicity of the control system design, minimum hardware inventory and appropriate performance for the varied requirements. The 3 to 1 propellant pressurization blowdown ratio results in a maximum thrust of 75 lbf per thruster.

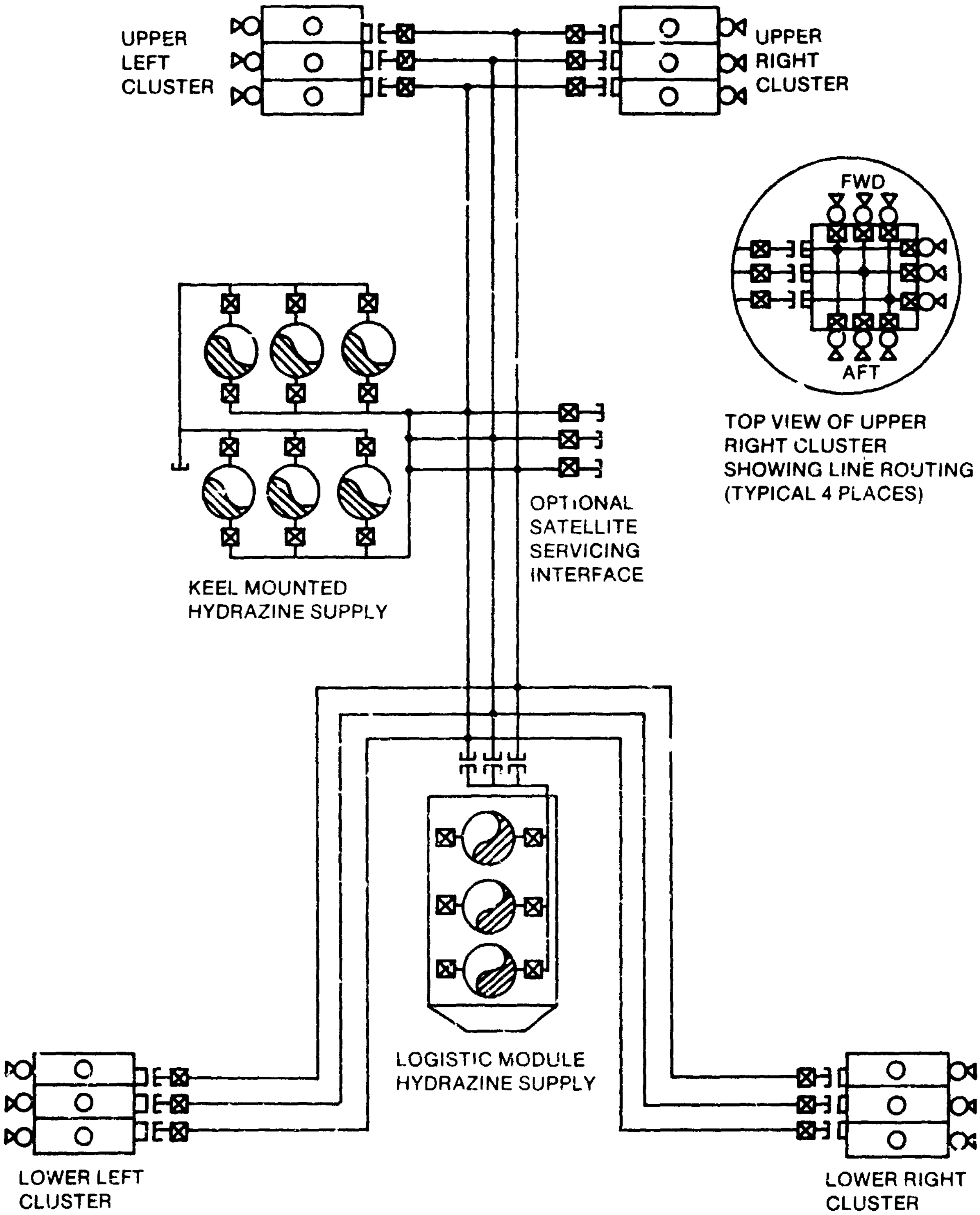


Figure 4.4.5-1 - REFERENCE CONFIGURATION PROPULSION SYSTEM WITH APPROXIMATE EQUIPMENT LOCATIONS

FEATURES SHOWN:

- TRIPLE REDUNDANT INDEPENDENTLY REPLACABLE MODULES
- PLUMBING ARRANGEMENTS
- DISTRIBUTION SYSTEM DISCONNECTS
- DISTRIBUTION SYSTEM ISOLATION VALVES

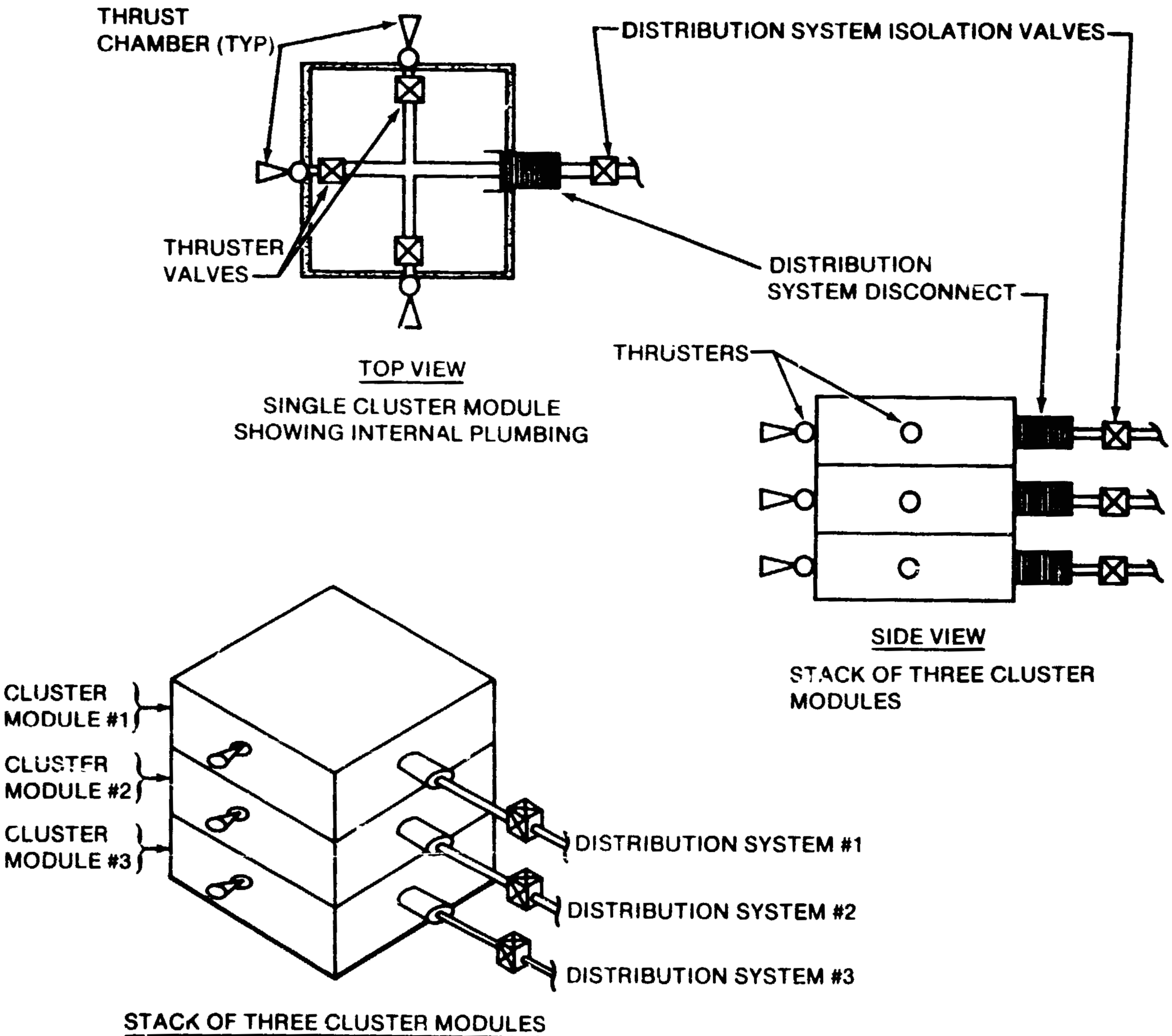


Figure 4.4.5-2 - TYPIC[illegible] THRUSTER CL[illegible]ER

A detailed weight summary is shown in Table 4.4.5-3. These data are presented separately to show the propulsion subsystem component weights which are located on the station lower keel and those located in the Logistics Module (LM). Also on this table, total dry system weights and wet weights are shown for both the station and the LM.

A summary of volume and average power usage for the propulsion subsystem is shown in Table 4.4.5-4 for those system components which occupy significant volume and/or demand quantifiable amounts of average power. A shrouded cluster of six propellant tanks, located on the lower keel, was estimated to require an average power of 35 watts for thermal stability. The same value was used as an estimate of the thermal requirement for the cluster of three LM tanks. An average power of 50 watts was estimated as required for the catalyst beds on each of the four thruster clusters. Distribution line heating requirements were estimated, based on Orbiter experience, to be 640 watts average power for the 1,600 foot Station run, and 20 watts for the 50 foot run on the LM.

4.4.5.4 Propellant Storage, Distribution, Management and Resupply

4.4.5.4.1 Propellant storage

The total quantity of propellant required for the Space Station propulsion subsystem reference configuration is 7,050 lb of hydrazine (N_2H_4). The propellant requirement is split between contingency propellant, 4,730 lb, and 90-day resupply propellant, 2320 lb. The propulsion subsystem will also contain a total of 57 cubic feet (190 lb) of gaseous nitrogen (GN_2) pressurant.

The tanks selected for the reference configuration are spherical elastomeric diaphragm tanks 40 inches in diameter. The selected tank shell material is 6A-4V titanium. Six of these tanks contain the contingency propellant and three additional tanks contain the 90-day resupply propellant.

The contingency tankage (six tanks) will remain with the Space Station. These tanks will be located near the station center of gravity envelope, at the interface between the lower keel and the keel extension. The contingency tankage will be packaged within the envelope of one of the 9-foot cubes of the lower keel truss structure, near the X-axis centerline. The tankage is packaged such that the diagonal brace struts of the Space Station truss structure will not preclude removal and replacement of individual tanks or tankage modules for maintenance or contingency purposes. Tanks are also arranged as a pair of three-tank modules to facilitate maintenance, as required. These modules should incorporate Space Station Mobile Manipulator/Orbiter RMS grapple fittings to facilitate handling and assembly. Individual tanks within a three-tank module incorporate propellant/pressurant quick-disconnect fittings. Structural/mechanical connections should also minimize connection/disconnection time. As an alternate, propellant tanks could be located closer to thrusters to facilitate Space Station launch/assembly.

TRANSLATION REQUIREMENTS	MOMENT ARM (FEET)	ANGULAR MOMENTUM (FT-LB-SEC)	90-DAY RE-SUPPLY IMPULSE (LB-SEC)	90-DAY RE-SUPPLY N_2H_4 (LBS)	CONTINGENCY IMPULSE (LB-SEC)	CONTINGENCY N_2H_4 (LBS)
• COLLISION AVOIDANCE $\Delta V = 5$ FT/SEC	—	—	—	—	61,500	280
• ALTITUDE TRANSFER 20 N. MILES	—	—	—	—	831,000	3,780
• RE BOOST 90 DAYS AT 270 NM (MIN)	—	—	483,000	2,200	—	—
ROTATIONAL REQUIREMENTS						
• ATTITUDE CONTROL 24 DAYS - Z AXIS ONLY	68	10,000,000	—	—	147,000	670
• ORBITER CAPTURE	88	150,000			—	—
• ORBITER BERTHING	88	1,500,000	26,000	120	—	—
• MODULE BERTHING	88	600,000			—	—
N_2H_4 RE SUPPLIED EACH 90 DAYS (LOCATED IN LOGISTICS MODULE)	—	—	509,000	2,320	—	—
N_2H_4 CONTINGENCY (LOCATED ON STATION)	—	—	—	—	1,039,500	4,730
TOTAL AVAILABLE	1,548,500 LB-SEC - 7,050 LBS-N_2H_4					

Table 4.4.5-1 REFERENCE CONFIGURATION IMPULSE REQUIREMENTS

	LOCATION	
	STATION	LOG MODULE
PROPELLANT	N_2H_4	N_2H_4
SPEFIC IMPULSE (LB -SEC/LB)	220	220
TOTAL IMPULSE (LB -SEC)	1,039,500	509,000
N_2H_4 (LBS)	4,730	2,320
N_2H_4 VOLUME (FT ^{3})	76	38
PRESSURANT VOL (FT ^{3}) GN_2	38	19
TOTAL TANK VOLUME (FT ^{3})	114	57
NUMBER OF 40" DIA TANKS REQ'D.	6	3
VOLUME OF ONE 40" DIA TANK (FT ^{3})	19	19
NUMBER OF THRUSTER CLUSTERS	4	0
NUMBER OF THRUSTERS PER CLUSTER	9	0
TOTAL NUMBER OF THRUSTERS	36	0
THRUST LEVEL (MIN-MAX) (LBS)	25-75	—
TANK/SUPPLY PRESSURE (MAX-MIN) (PSI)	300-100	—
PRESSURIZATION SCHEME	BLOW DOWN	—
THRUST MOMENT ARM-Z AXIS (FT)	68	—
THRUST MOMENT ARM-X AXIS (FT) AVG.	162/2	—
THRUST MOMENT ARM-Y AXIS (FT) AVG.	162/2	—

Table 4.4.5-2 - REFERENCE CONFIGURATION PROPULSION SYSTEM FEATURES

	LOWER KEEL	LOG MODULE	TOTAL
PROPELLANT	4,730	2,320	7,050
PROPELLANT TANKS	504	252	756
PRESSURIZATION GAS	126	64	190
PROPELLANT DISTRIBUTION SYSTEM	400	25	425
THRUSTER CLUSTERS	216	0	216
THERMAL CONDITIONING SYSTEM	200	5	205
INSTRUMENTATION	50	15	65
ISOLATION VALVES	81	18	99
DISCONNECTS	90	18	108
SECONDARY STRUCTURE	154	33	187
CLUSTER BOOMS	120	0	120
PROPELLANT RESIDUALS	280	18	298
DRY WT. TOTAL	1,815	366	2,181
WET WT. TOTAL	6,951	2,768	9,719

Table 4.4.5-3 - REFERENCE CONFIGURATION PROPULSION SYSTEM WEIGHT SUMMARY

	VOLUME (FT^3)		POWER (kW)	
	LOWER KEEL	LOGISTIC MODULE	LOWER KEEL	LOGISTIC MODULE
PROPELLANT TANKAGE	114	57	.035*	.035*
THRUSTER CLUSTER	16	-0-	.200*	-0-
DISTRIBUTION LINES			.640*	.020*
DISCONNECTS	9	1	—	—
THERMAL CONTROL			*SEE ABOVE	*SEE ABOVE
ISOLATION VALVES	3	1	NEGLIGIBLE	NEGLIGIBLE
INSTRUMENTATION	1	.2	NEGLIGIBLE	NEGLIGIBLE
SECONDARY/STRUCTURE	50	2.5	—	—
VOLUME TOTALS	193	84.2		
POWER TOTALS			.875	.055

Table 4.4.5· PEFERENCE CONFIGURATION PROPULSION SYSTEM VOLUME AND POWER SUMMARY

The 90-day resupply tankage (three tanks) will be located on the LM, which is relatively close to the contingency tankage. The LM will remain connected to the Station between Orbiter resupply missions. The nitrogen purge gas, if required, will be located within the LM.

The 90-day resupply tankage will be packaged within the end section (aft skirt) of the LM, which is the end opposite from the docking adapter. Ideally, this resupply three-tank module will maximize the number of components which are common with the contingency tankage three-tank modules. The resupply tankage module also incorporates propellant/pressurant resupply quick-disconnect interfaces. These interfaces allow connection to the propulsion subsystem propellant distribution system.

Propellant Storage Interfaces: The propellant storage & distribution systems will interface with the following SS subsystems: Structural/Mechanical; Propellant/Pressurant Manifolds; Power; Guidance, Navigation and Control (GN&C); Data Management System (DMS); and the Thermal Control Subsystem. This reference configuration (Figure 4.4.3-1) provides the option for interconnection with other compatible hydrazine propellant tankage, such as the OMV or satellite reservicing propellant tankage.

4.4.5.4.2 Propellant distribution

The propellant distribution system of the Space Station reference configuration propulsion subsystem is designed to provide and control the flow of propellant from the propellant storage tanks to the thrusters. It is also required to provide the fluid system for resupply of the propulsion subsystem storage tanks. The propellant distribution system will also provide propellant interconnect capability with other hydrazine storage systems onboard the Space Station. Propellant feed lines are routed from the centralized propellant storage tanks to the four thruster clusters at the end of each cluster boom. A resupply propellant fill line system connects the Space Station central tankage to the LM propellant interface via the central Space Station propellant manifold.

Triple redundant lines (fail operational/fail safe) have been selected for this reference configuration. The three line runs will be separated by an optimum distance to preclude simultaneous damage to lines. The reference configuration has been baselined to utilize hardlines for propellant distribution. An alternative would be to use convoluted stainless steel flex-hoses. However, it is felt that the additional pressure drop and additional weight would be unacceptable, considering the relatively long line runs. Teflon-lined (smooth-bore) flex-hoses could be utilized, but long term propellant permeation could be a problem.

The selected line is 3/4 inch O.D. stainless steel tubing. The total propellant distribution system line length is approximately 1600 feet, including redundancy. The longest thruster feed line run is approximately 150 feet. The selected system pressure range is 300-100 psia for the three-to-one blowdown system. Propellant distribution system pressures will be slightly higher to account for pressure drops. Also, reloading of the diaphragm propellant tanks will require pressures in the 600-700 psia range. The line safety factor is in excess of 4.

The propulsion subsystem reference configuration propellant distribution system has been baselined to have heaters and insulation wrap applied to the propellant lines to preclude hydrazine freezing. An alternative would be to utilize waste heat from other subsystems as a heat source. A trade study is required to determine the optimum method.

The reference configuration has a total of 36 propellant disconnects. The LM has 6 and the Space Station has 30. The disconnects are of the quick-action type where applicable. They are designed to incorporate features to preclude misconnection. The reference configuration incorporates a total of 33 propellant isolation valves. The LM has six and the Station has 27. The valves are of the electromechanical solenoid type.

4.4.5.4.3 Propellant utilization and management

The propellant utilization and management system is designed to gage propellant quantities, control intertank propellant usage, and directly or indirectly control and monitor propellant resupply operations. The operational strategy for the propulsion subsystem reference configuration involves feeding propellant to the thrusters from only one propellant tank at a time, with the rest of the propellant tank isolation valves closed. When the first tank is depleted, its isolation valve will be closed and the second propellant tank's isolation valve will be opened. In this manner, some of the disadvantages of the blowdown system will be minimized. The propellant utilization and management system has temperature and pressure transducers located at key points within the propulsion subsystem. Since propellant quantity gaging is accomplished by the pressure/volume/temperature technique, only one tank at a time will suffer the inherent propellant gaging inaccuracies. The other tanks will have accurately known propellant quantities either empty or full. This in turn will facilitate resupply operations. Components of the Space Station propulsion subsystem will interface directly with the IDMS.

4.4.5.4.4 Resupply

The 90-day resupply three-tank module will be installed on the LM aft skirt, and the propellant will be loaded at ground facilities. Since the 90-day resupply tankage is based on the LM, on-orbit propellant servicing operations are not required for this tankage. Three-tank modules for reloading of Station-based contingency tankage will be installed, on an as-required basis, in the LM aft skirt on the ground and loaded consistent with the amount of contingency propellant used by the Station between Orbiter 90-day resupply missions. Thus, a LM could contain as many as nine tanks for a worst case resupply mission. Six of these tanks are for Station-based contingency tankage reloading, and three tanks fullfill the 90-day resupply tankage requirement. The purge gas, if required, will be installed in the LM aft skirt and loaded at ground facilities.

4.4.5.5 Concerns and Issues

4.4.5.5.1 Commonality

Space Station: If only the reference Space Station hydrazine subsystem is considered, the components that may be exactly replicated include tankage, thrusters, valves, and instrumentation. It is more likely that most of these components would be part of a module-like subassembly that could be replicated except for mounting. The reference propulsion subsystem shown earlier includes replication of tanks and thruster modules, with some exceptions for mounting left and right modules.

Platforms: Component commonality between the Space Station and Platform propulsion subsystems is desirable from an initial and operational cost perspective. In the level of description available on the two platforms, no unique propulsion subsystem requirements were identified that violate the commonality possibilities discussed. However, requirements are not sufficiently defined to assess thoroughly the potential impacts on propulsion subsystem requirements.

OMV: The Orbital Maneuvering Vehicle (OMV) is anticipated to use hydrazine and/or monomethylhydrazine/nitrogen-tetroxide propulsion subsystems. The OMV may provide an alternative option for Space Station reboost and other translational functions. The propellant storage facilities provided at the Station might also serve as a common depot for the station propulsion subsystem needs. And component commonality might be available between the OMV and the station.

OTV: The Orbital Transfer Vehicle (OTV) and its propellant storage facilities may also provide a component and propellant storage commonality, depending on the propulsion subsystems selected.

Payload: Platform and satellite consumables include hydrazine, liquid hydrogen and liquid helium. The quantities of these fluids, which are anticipated to be stored on the Station for resupply of payloads, significantly exceed the Station propellant requirements. Use of these fluids for Station propulsion might enhance resupply logistics.

4.4.5.5.2 Hardware life/replacement

Very long life monopropellant hydrazine thrusters must be designed to meet the requirements for maintainability, limited crew attention, and overall Space Station life. The reboost requirement will result in a very large throughput of propellant for the catalyst bed of those thrusters when compared to current flight propulsion subsystems.

Several alternatives are available to meet the life requirements. These include the installation of in place spares that could be functionally switched into the system, or the development of improved designs of catalyst beds through early technology programs.

4.4.5.5.3 Station buildup

The mass properties (magnitude and distribution) of the Space Station will undergo significant change throughout the life of the station, first during the assembly of the IOC station and then with the addition of attached payloads and servicing facilities. Propulsion subsystem accommodation of these changes is an issue. Component selection and packaging, station scarring, and installation and resupply techniques will be affected.

4.4.5.5.4 Growth station

The growth version of the Space Station will be significantly larger, notably as a result of increased power capacity and the expansion of servicing and other facilities. There will be an increased demand for accommodation of payload and platform servicing, of OTV and OMV activities, and of onboard operations. These may lead to control, propulsion and propellant management subsystem requirements for greater capability and flexibility than those anticipated for the initial station.

4.4.5.5.5 Advanced and alternate propulsion options

Other propulsion subsystem options must be considered for both the initial and growth station to obtain the most effective balance of capability and cost. Consideration must be given to:

(a) ability to respond to variations and uncertainty of required impulse with station growth and expanded mission definitions

(b) simultaneous altitude maintenance and CMG desaturation

(c) reduced resupply requirements and manifesting variations

(d) increased operations flexibility, particularly in coordinating activities with platforms

(e) reduced operational complexity

(f) increased component life and maintainability

(g) integrated fluid systems management

(h) disposal of unwanted Station fluids

(i) initial and life cycle costs.

Options, in addition to the reference blowdown hydrazine subsystem, include Earth-storable bipropellant thrusters, hydrogen-oxygen thrusters, multifuel resistojets, and thermally-augmented subsystems. The bipropellant subsystem offers the opportunity to make use of OMV and Orbiter propellants, and possibly of common components. The H-O subsystem offers clean exhaust and high specific impulse, and could use cryogen H-O storage boiloff, external tank scavenged-fluids, or electrolysis of water with potential interface with the ECLS or other subsystems. The resistojet with a thrust level of less than 1 lbf could accommodate altitude maintenance and CMG desaturation for the Station and platforms while maintaining acceleration continuously below .00001 g. Platform ephemeris could be maintained in spite of the highly variable atmospheric perturbations. The resistojet would operate at low temperature to achieve long life and could operate on a broad range of fluid, including hydrazine, water, hydrogen, carbon dioxide.

4.4.6 Environmental Control and Life Support System

4.4.6.1 Introduction

The ECLSS described herein provides for partial system closure for the initial Space Station. Partial closure of the ECLSS is based on regenerative processes for closing both the metabolic oxygen and water cycles. The only resupply materials are nitrogen for module and airlock leakage makeup and extravehicular activity (EVA) equipment servicing, and food for crew consumption. A completely closed ECLSS is not viable at this time because of the high cost and risk of developing food regeneration technology. Nitrogen generation via hydrazine dissociation could be considered for the Growth Station. The partially closed ECLSS is selected because it offers the optimum balance between development costs, technology risk and resupply penalties for Space Station application.

The partially closed cycle ECLSS shall contain sufficient modularity and flexibility in design to accommodate on-orbit maintainability, repair, and evolutionary growth capability. These characteristics are important in that the initial Space Station ECLSS must allow for future incorporation of additional capability (increase in crew size and habitat area) without incurring additional development costs. The ECLSS design must also be able to adapt to advances in state-of-the-art concepts that offer increased efficiency, process yield, reliability and/or maintainability without requiring major alterations to existing ECLSS flight hardware.

4.4.6.2 System Functional Requirements

The functions of the partially closed ECLSS include atmosphere pressure and composition control, module temperature and humidity control, atmospheric revitalization, water management, metabolic waste management, and EVA equipment servicing. Detailed expansion of these requirements is as follows:

a. Atmosphere Pressure and Composition Control

(1) Control and monitor module total pressure within specified limits

(2) Control and monitor oxygen (O_2) and nitrogen (N_2) partial pressures

(3) Provide fire detection and suppression

b. Module Temperature and Humidity Control

(4) Control ambient temperature

(5) Control humidity

(6) Provide air ventilation

c. Atmospheric Revitalization

(1) Remove and concentrate metabolic carbon dioxide (CO_2)

(2) Reduce CO_2 for water recovery

(3) Generate O_2

(4) Provide makeup N_2 and O_2 to compensate for module leakage and airlock losses

(5) Control and monitor trace contaminants and microorganisms

d. Water Management

(1) Collect, pretreat and process waste water

- Hygiene water (urine/flush, shower, and handwash)

- Potable water (humidity control and CO_2 reduction condensate)

- Wash water (dish washer and clothes washer)

(2) Posttreat recovered water

(3) Monitor water quality

(4) Store and deliver recovered water to water use interfaces

e. Waste Management

(1) Collect urine for water recovery

(2) Collect and store fecal matter

f. EVA Equipment Servicing

(1) Provide O_2, water (H_2O), and heat sink regeneration services to Extravehicular Mobility Unit (EMU)

(2) Provide N_2 recharge capability for Manned Maneuvering Unit (MMU)

(3) Provide life support services to airlock/hyperbaric facility

In fulfilling these requirements for the initial Station, the ECLSS will use a regenerative CO_2 removal process for collection of metabolic CO_2 and subsequent delivery to the CO_2 reduction process. The CO_2 reduction process will reduce CO_2 to water and carbonaceous products. Overboard venting of these carbonaceous products will not be permitted. After appropriate processing, the CO_2 reduction effluent water and

C - 6

condensate from the Station temperature and humidity control process will be used as potable water for food preparation and drinking purposes. Additional water for reprocessing will be obtained from hygiene (handwash, shower, urine/flush) and housekeeping water (dish/clothes washing) sources. A phase change process and pre- and posttreatment processes will be used to recover usable water from these hygiene water sources. Effluent waste water from dish and clothes washing will be reclaimed by a filtration process. Oxygen will be provided by electrolysis of the processed hygiene water. Nitrogen for the initial Space Station will be supplied from either cryogenic or high pressure gas storage. A contaminant control process will be included in the ECLSS to remove airborne trace contaminants. In addition, a solid waste management process will be included for handling and treatment of human fecal matter.

4.4.6.3 Reference Configuration

4.4.6.3.1 <u>System sizing criteria</u>

The following criteria were used in sizing and developing the reference ECLSS for Space Station:

a. Crew Size
   Initial: 6 persons
   Growth: 18 persons

b. Space Station Leakage: 5 lb/day, maximum

c. Resupply Interval: 90 days

d. Safe Haven Emergency Capability
   Duration: 22 days
   Location: 2 modules, minimum

e. EVA equipment servicing
   EVA duration: 16 person-hours/EVA; 5 EVA's/week
   MMU Recharge: 115 lb N_2/week

f. No dumping of gases (goal)

g. Fail Operational/Fail Safe (Restorable)

4.4.6.3.2 <u>Basic ECLSS definition</u>

A conceptual design of the partially closed ECLSS satisfying the requirements identified in Section 4.4.6.2 for the initial and growth versions of the Space Station is depicted in Figure 4.4.6.1. The mass balance for specific regenerative process technologies is depicted in Figure 4.4.6.2 The mass balance numbers are for a 6-person crew and all flow rates are in lb/day. <u>It should be emphasized that specific ECLSS</u>

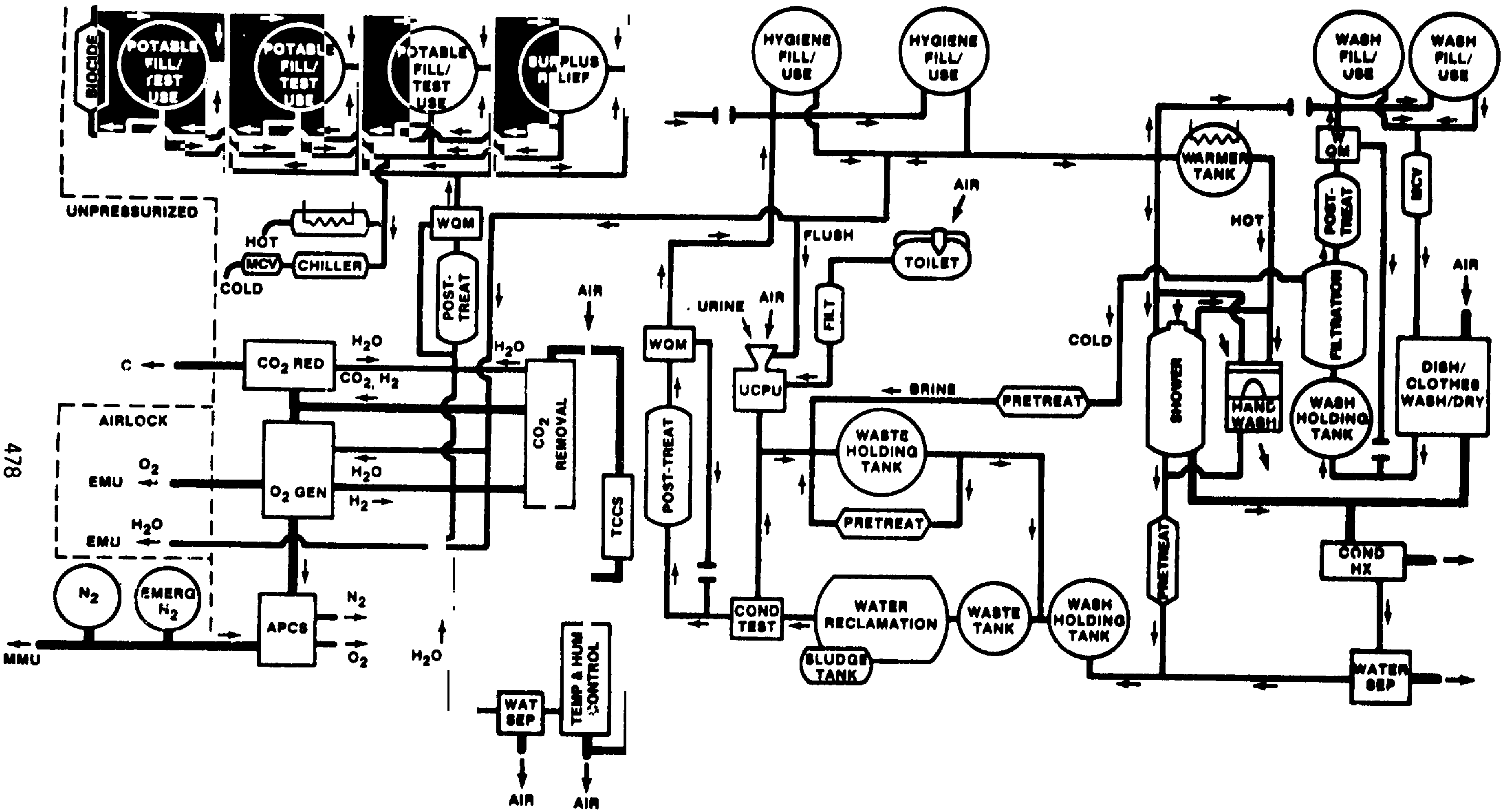


Figure 4.4.6-1 Partially Closed ECLSS for Space Station

UNITS—LB/DAY

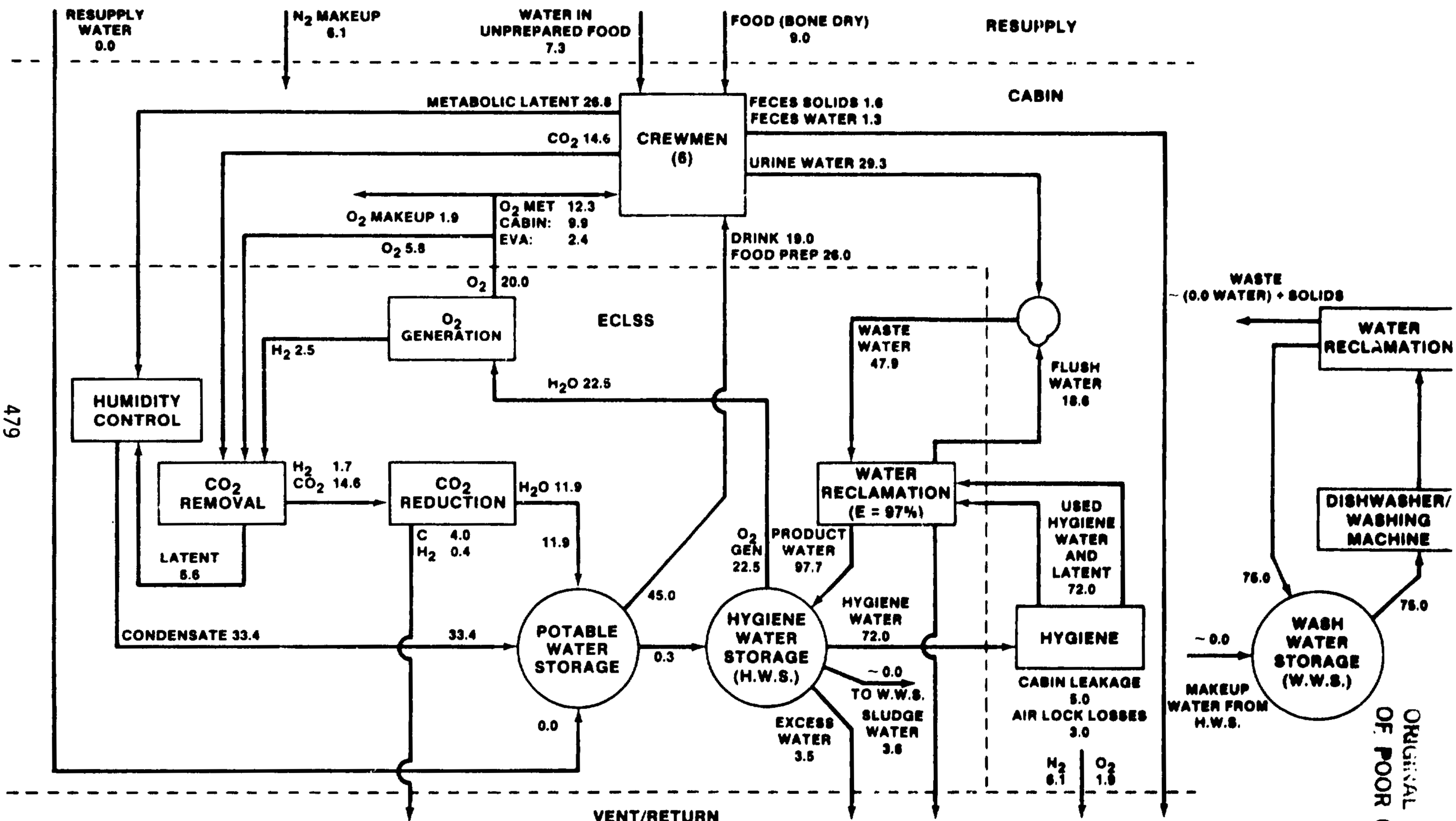
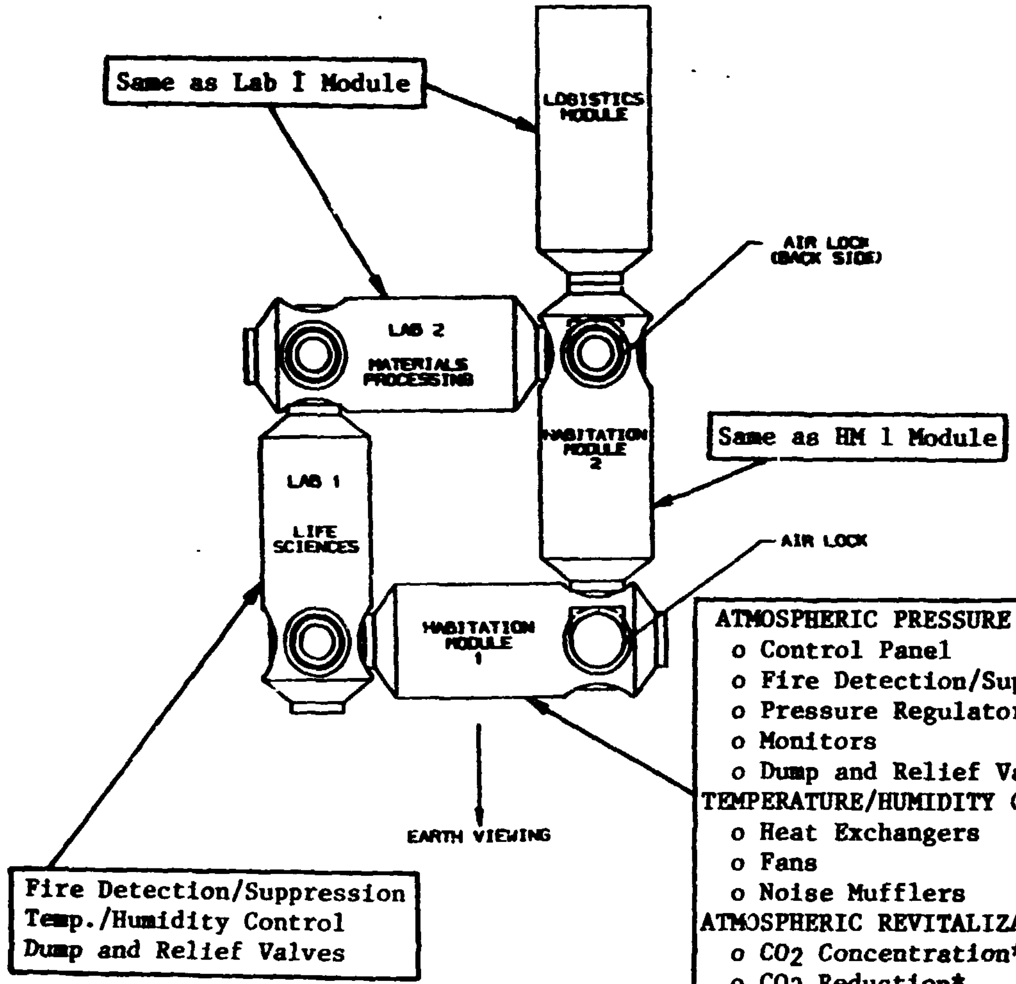


Figure 4.4.6-2 Reference Partially Closed ECLSS Mass Balance



ATMOSPHERIC PRESSURE & COMPOSITION
- o Control Panel
- o Fire Detection/Suppression
- o Pressure Regulators
- o Monitors
- o Dump and Relief Valves

TEMPERATURE/HUMIDITY CONTROL
- o Heat Exchangers
- o Fans
- o Noise Mufflers

ATMOSPHERIC REVITALIZATION
- o CO_2 Concentration*
- o CO_2 Reduction*
- o O_2 Generation**
- o Catalytic Oxidizer/Filters
- o Monitor

WATER MANAGEMENT*
- o Hygiene Water Recovery
- o Potable Water Recovery
- o Storage and Dispensing
- o Monitoring and Collection

WASTE MANAGEMENT
- o 1 Commode
- o 2 Urinals

EVA SUPPORT
- o EMU O_2, H_2O, Heat Sink Recovery
- o MMU - N_2 Supply
- o Suit - Servicing, Cleaning
- o Airlock/Hyperbaric Facility Support

SAFE HAVEN
- o 6 Portable Oxygen Supply (POS)
- o O_2 and H_2O Recovery
- o Ventilation
- o Temperature/Humidity Control
- o CO_2 and Contamination Control
- o Waste Collection and Storage

* 2 - Three Person Units
** 1 - Three Person Unit (Module)
1 - Three Person Unit (EVA)

Figure 4.4.6-3 PARTIALLY CLOSED ECLSS EQUIPMENT/SPACE STATION LAYOUT processes were chosen only for the purpose of illustrating ECLSS mass balances. The baseline processes for CO_2 removal, CO_2 reduction, O_2 generation and waste water reclamation have not been selected.

The Space Station habitable module configuration used in this reference ECLSS is depicted in Figure 4.4.6-3, and the recommended centralized/distributed functional allocation of the ECLSS is given in Table 4.4.6-1. Although a totally distributed ECLSS (i.e., independent subsystems in each Space Station module) initially appears more supportive of the "common module" philosophy, replication of all ECLSS functions in each module involves excessive hardware. Furthermore, such extensive replication of hardware does not use the available module volumes effectively. As an example, it is impractical to locate waste management and waste water generation and recovery equipment in each module. It is also prudent to co-locate the waste water generation and recovery equipment to minimize interface plumbing and to enhance reliability and maintenance. Therefore, the centralized/distributed approach was selected for the reference ECLSS configuration. This approach also supports the "common module" by providing the following basic ECLSS equipment in each module: ventilation fans and ducting, temperature and humidity control, module atmosphere dump and pressure relief valves, fluid transport plumbing, and fire detection/suppression. Table 4.4.6-2 contains a similar functional allocation for the Growth Station.

Distribution of the ECLSS equipment for each module is defined in the Crew Accommodations section, Figures 4.3.2.1-1 through 4.3.2.2-6. Two 3-person capacity ECLSS units that provide the functions of CO_2 removal, CO_2 reduction, trace contaminant control and waste water reclamation will be centrally located in each of the two Habitation Modules. These two modules, which interface with the EVA airlock/hyperbaric facilities, were selected for the centralized location of the ECLSS processing equipment to consolidate equipment and minimize replication and interfaces. One of the ECLSS units in each of these modules will be in an "unpowered" standby mode and the other unit will be "operational." Each 3-person capacity ECLSS unit will be capable of operating at a 3-person capacity in normal mode and at a 6-person capacity off-nominal mode. The O_2 generation function in each of these two modules will consist of a 3-person capacity (low pressure) unit for module and metabolic usage and a 3-person capacity (high pressure) unit for EMU recharging. Each of these electrolysis units will be capable of generating O_2 for 6 crewmen (at a higher current density) for off-nominal mode operation.

Air revitalization in the remaining modules will be accomplished with air distribution ducts. Humidity and temperature control will be localized in each habitable module. The ducting and plumbing in each module will also include provisions for bypassing each module and be capable of isolation from each module.

The ECLSS weight, volume, and power requirements for the initial Space Station are presented in Table 4.4.6-3. The power levels specified in

TABLE 4.4.6-1 INITIAL SPACE STATION ECLSS FUNCTIONAL ALLOCATION

Function	HM1	HM2	LAB1	LAB2	LM
Fire. Det. & Supp.	X	X	X	X	X
CO_2 Removal	X	X			
Cabin Thermal Control	X	X	X	X	X
CO_2 Reduction	X	X			
O_2 Supply	X	X			
N_2 Supply					X
Atmospheric Control	X	X	X	X	X
Potable Water Reclamation	X	X			
Potable Water Treatment/Storage	X	X			
Waste Management	X	X			
Hygiene/Wash Water Reclamation & Storage	X	X			
Safe Haven Provisions	X	X			
EVA Equipment Servicing	X	X			

TABLE 4.4.6-2 GROWTH (18 PERSON) SPACE STATION ECLSS FUNCTIONAL ALLOCATION

Function	HM1	HM2	LAB1	LAB2	LM1	HM3	HM4	LAB3	LAB4	LM2	LM3
Fire. Det. & Supp.	X	X	X	X	X	X	X	X	X	X	X
CO_2 Removal	X	X				X	X				
Cabin Thermal Control	X	X	X	X	X	X	X	X	X	X	X
CO_2 Reduction	X	X				X	X				
O_2 Supply	X	X				X	X				
N_2 Supply					X					X	X
Atmospheric Control	X	X	X	X	X	X	X	X	X	X	X
Potable Water Reclamation	X	X				X	X				
Potable Water Treatment/Storage	X	X				X	X				
Waste Management	X	X				X	X				
Hygiene/Wash Water Reclamation & Storage	X	X				X	X				
Safe Haven Provisions	X	X				X	X				
EVA Equipment Servicing	X	X				X	X				

TABLE 4.4.6-3 INITIAL SPACE STATION ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEM SUMMARY

Subsystem Function	Weight (LBS)	Volume (FT^3)	Power (Watts)
Atmospheric Pressure/ Composition Control	685	45	450
Atmospheric Revitalization, Ventilation	2506	211	5260
Humidity/Temperature Control Ventilation	500	25	400
Water Management	2526	117	650
Waste Management	500	40	900*
EVA Servicing	84	4	270
Safe Haven	420	31	-
Tankage (Dry)	2050	300	-
Fluids	6000		
Total Dry / Wet	9351 / 15,351	773	7030

*Urinals/commodes (intermittent) if all used at once.

Table 4.4.6-3 are for continuous ECLSS operation throughout the orbital period. However, cyclic light side/dark side operation of the ECLSS may be a more effective approach in power management. Cyclic operation could take advantage of the availability of the "inexpensive" direct solar power during the sunlit (light side) phase of the orbital period.

4.4.6.3.3 EVA equipment servicing

EMU and MMU servicing will be an integral function of the basic Space Station ECLSS. Replenishment of EMU and MMU expendables and the recovery of waste products will be provided. The ECLSS will supply oxygen and water for the EMU's and reprocess the collected waste water and CO_2. The ECLSS also will provide nitrogen for MMU propellant system recharging, and generate make up gases for airlock/hyperbaric facility leakage and pressurization as well as atmospheric revitalization and cooling for pre-EVA activities. A detailed description of the EVA systems is contained in Section 4.4.9.

4.4.6.4 Conclusions

Partial closure of the ECLSS using regenerative processes offers the most cost effective approach and enhances evolutionary growth flexibility for Space Station. Additional development costs should be offset by the substantial reduction in resupply requirements. Furthermore, locating the ECLSS equipment in the centralized/distributed manner described above minimizes the amount of hardware while accommodating the "common module" philosophy, and fulfills the required redundancy and "safe haven" requirements without additional backup equipment. For example, the two 3-man CO_2 removal subsystems in the two separate modules eliminate the need for a backup lithium hydroxide CO_2 removal system. Similarly, the redundant electrolysis units eliminate the need for backup emergency O_2 tanks. Only additional emergency N_2 for module leakage make up and portable O_2 supplies and masks for use in module evacuation are required in achieving a "safe haven". Centralizing the ECLSS also localizes noisy process equipment and minimizes waste fluid line lengths. The dual centralized configuration also enhances system maintenance operations and readily accommodates temporary crew overload as during crew change out.

4.4.7 Thermal Control

4.4.7.1 Introduction

The following requirements were used for this evaluation:

a. Base station electrical power characteristics (IOC and growth).

(1) Useful electrical power at bus: 75 kW_E (IOC).
300 kW_E (growth).

(2) Orbital average waste heat load: 95.8 kW_T (IOC).
393.9 kW_T (growth).

(3) Sunside waste heat load:

(a) Customer: 50 kW_T (IOC) and 250 kW_T (growth).

(b) Housekeeping: 25 kW_T (IOC) and 50 kW_T (growth).

(c) Power subsystem: 13 kW_T (IOC) and 53 kW_T (growth).

(4) Darkside waste heat load:

(a) Customer: 50 kW_T (IOC) and 250 kW_T (growth).

(b) Housekeeping: 25 kW_T (IOC) and 50 kW_T (growth).

(c) Power subsystem: 46 kW_T (IOC) and 182 kW_T (growth).

b. Manned core waste heat distribution and allowable heat source interface temperatures for IOC (Table 4.4.7-1).

c. Manned core waste heat distribution and allowable heat source interface temperatures for growth (Table 4.4.7-2).

The major design assumptions used for this sizing study are as follows:

a. Heat acquisition and transport by a pumped two-phase "thermal-bus" approach with water as the transport fluid in manned modules and ammonia outside.

b. Heat rejection by high capacity heat pipe radiators.

c. No heat load sharing between Station modules (i.e., a module with excess heat rejection capability from its body-mounted radiators does not reject heat from other modules).

d. IOC housekeeping power distribution by Station module.

(1) HM1 - 27%.

(2) HM2 - 27%.

(3) LAB1 - 29%.

(4) LAB2 - 12%.

(5) LM - 5%.

TABLE 4.4.7-1 - MANNED CORE WASTE HEAT DISTRIBUTION AND ALLOWABLE HEAT SOURCE INTERFACE TEMPERATURES FOR IOC

	SPACE STATION TOTALS*	HAB 1	HAB 2	LAB 1	LAB 2	LOG	POWER SYS.	ATTACH P/L
CUSTOMER	10.5 kW @ 35°F			10.0				0.5
	16.5 kW @ 68 - 77°F				8.5			8.0
	23.0 kW @ 68 - 122°F			13.5				9.5
HOUSEKEEPING	9.6 kW @ 35°F	2.24	2.24	3.96	0.61	0.56		
	11.8 kW @ 68 - 77°F	3.45	3.45	2.4	2.0	0.5		
	+[3.6]**		[3.6]**					
POWER SYSTEM	24.4 kW @ 115°F ±5°F						24.4	
TALS	95.8 kW +[3.6]**	5.69	5.69 +[3.6]**	29.86	11.11	1.06	24.4	18.0

*ORBITAL AVERAGE LOADS BASED ON AN ELECTRICAL BUS LOAD OF 75 kW_E
**VALUE IN [ ] IS PASSIVELY RADIATED (COMMUNICATIONS) AND IS NOT PART OF ACTIVE WASTE HEAT LOAD.

TABLE 4.4.7-2 - MANNED CORE WASTE HEAT DISTRIBUTION AND ALLOWABLE HEAT SOURCE INTERFACE TEMPERATURES FOR GROWTH

	SPACE STATION TOTALS*	HAB 1	HAB 2	LAB 1	LAB 2	LOG	HAB 3	HAB 4	LAB 3	LAB 4	LAB 5	LAB 6	POW SYS.	ATTACH P/L
	20.0 kW @ 35°F			20.0										
CUSTOMER	110.0 kW @ 68 - 77°F				12.7				2.0	20.5	11.7	13.1		50.0
	120.0 kW @ 68 - 122°F			20.3					41.0		21.2	10.0		27.5
	16.0 kW @ 35°F	1.9	1.9	3.95	0.7	0.95	1.9	1.9	0.7	0.7	0.7	0.7		
HOUSEKEEPING	30.4 kW @ 68 - 77°F	6.3	6.3	3.8	3.0	0.89	1.7	1.7	1.7	1.7	1.7	1.6		
	+[3.6]**		[3.6]**											
POWER SYSTEM	97.5 kW @ 115°F ± 5°F												97.5	
TOTALS	393.9 kW +[3.6]**	8.2	8.2 +[3.6]**	48.1	16.4	1.8	3.6	3.6	45.4	22.9	35.3	25.4	97.5	77.5

*ORBITAL AVERAGE LOADS BASED ON AN ELECTRICAL BUS LOAD OF 300 kW
**VALUE IN [ ] IS PASSIVELY RADIATED (COMMUNICATIONS) AND IS NOT PART OF ACTIVE WASTE HEAT LOAD.

4.4.7.2 Heat Acquisition and Transport

4.4.7.2.1 Sizing approach

The isothermal requirements for customer thermal control and the capability to adapt to highly variable thermal loads have led to more advanced thermal management system concepts than have previously been utilized on spacecraft. These concepts take advantage of the properties of two-phase fluids which include an enhanced thermal capacity (latent heat of vaporization), improved heat transfer coefficients, decreased pumping power requirements, and reduced fluid inventory. The reference configuration Active Thermal Control Subsystem (ATCS) is composed of (a) two-phase ammonia external thermal transport loops, or "thermal buses", which transport the heat load from the station modules, power subsystem, and attached payloads to the central radiators for heat rejection and (b) two-phase water loops internal to the manned modules which transport heat from the equipment and experiment coldplates to the module/external thermal bus interface heat exchangers.

The external and internal thermal transport loops for the manned modules are segmented into three separate temperature levels (35°F, 70°F, and 90°F) to reduce required central radiator area and to enhance the isothermal characteristics of the thermal management system. The 35°F loop services ECLSS and some customer heat loads; the 70°F loop services equipment for "housekeeping" and customer heat loads; and the 90°F loop services customer heat loads. Thermal control of power subsystem equipment (regenerative fuel cells and power conditioning hardware) is accomplished by an autonomous 115°F loop.

The analysis sizing the thermal acquisition/transport system for the reference configuration was accomplished utilizing a computer code for a parallel flow thermal bus system. The program was configured to size single-phase as well as two-phase transport systems and to incorporate a variety of fluids with temperature dependent properties.

The program optimizes the sum of the fluid weight, line weight, pump package weight, pump power penalty weight, and valve weight for a parallel flow system. Line wall thickness is calculated from stress and micrometeoriod considerations and is then compared to standard pipe sizes. Assumptions used in the analysis are as follows:

a. Pump efficiency: 25 percent
b. Pumping power penalty: 350 lbs/kW
c. Coldplate weight: 20 lbs/kW
d. Heat exchanger weight:
   3.2 lbs/kW for two-phase to single-phase
   6.4 lbs/kW for two-phase to two-phase
e. Valve weight: 6 lbs each
f. Minimum pipe ID: 0.188 in.
g. Module internal bus plumbing: stainless steel
h. External bus plumbing: aluminum
i. Heat loads and loop temperature per Table 4.4.7-3

TABLE 4.4.7-3
IOC SPACE STATION
THERMAL LOAD DISTRIBUTION

ELEMENT	35F(2C)	70F(21C)	90F(32C)	115F(46C)	TOTAL
		HEAT LOADS (KW)			
HAB 1	2.24	3.45	0.00	0.00	5.69
HAB 2	2.24	3.45	0.00	0.00	5.69
LAB 1	13.96	2.40	13.50	0.00	29.86
LAB 2	0.61	10.50	0.00	0.00	11.11
PAYLOAD	0.50	8.00	9.50	0.00	18.00
LOGISTICS	0.56	0.50	0.00	0.00	1.06
POWER MODULE	0.00	0.00	0.00	24.40	24.40
IOC TOTAL	20.11	28.30	23.00	24.40	95.81

Further evaluation of the thermal acquisition/heat exchanger components (i.e., coldplates and heat exchangers) was not undertaken because of the greater level of detail required and the time constraints present.

4.4.7.2.2 External centralized thermal bus sizing

Fluid transport lines for the centralized system are external to the modules, requiring the tube wall thicknesses to be sized to reduce the probability of micrometeroid penetration. The external thermal loops will interface with the modules across a heat exchanger. However, the heat exchanger weights are not included in the central bus but are instead included in the module heat transport weights. There is assumed to be a distance of 30 feet between the Station modules which are connected in parallel, and 50 feet between the first Station module and the central radiator condenser. A schematic of the external thermal bus (excluding the power subsystem) is given in Figure 4.4.7-1. Although the logistics module is shown to require some thermal rejection capability, it is assumed to be self-contained and independent of the central transport system.

Results for a two-phase ammonia external bus capable of transporting 70.35 kW from the modules and attached payloads (excluding the power subsystem) to the central radiator are shown in Table 4.4.7-4. The vapor lines are sized to minimize pressure drop in order to maintain a $\pm 5°F$ temperature band for isothermality across the heat loads. It should be noted that the central transport lines will ultimately be sized by the expected growth potential of the Space Station. This is especially true of those lines which cross the radiator gimbal mechanism. These lines will either be significantly oversized for IOC or the plumbing and fluid joints will be changed out at the time the thermal load surpasses the system capability. The results presented for the IOC external bus represent minimum capabilities.

4.4.7.2.3 Module/element internal thermal transport sizing

The individual module/element thermal loads, as given previously in Table 4.4.7-3, were further subdivided into several thermal quantities to represent coldplate locations within the modules. Each module heat load at a particular temperature level was divided into three to six individual loads, with the larger module loads being divided into the larger number of individual loads, all of which are connected in parallel. As is apparent, this is a rough representation of what the actual heat load distribution would be, but is considered sufficient for determining approximations of system line sizes and weight. A schematic of the internal thermal transport system for each module/element is shown in Figures 4.4.7-2(a) and 4.4.7-2(b).

Results for each of the station module elements, using two-phase water with stainless steel plumbing for thermal transport,are given for IOC in Tables 4.4.7-5 through 4.4.7-7. The attached payload and power subsystem transport sizing results are summarized by Table 4.4.7-8. They utilize the same fluid as the central thermal bus, i.e., two-phase ammonia, not water.

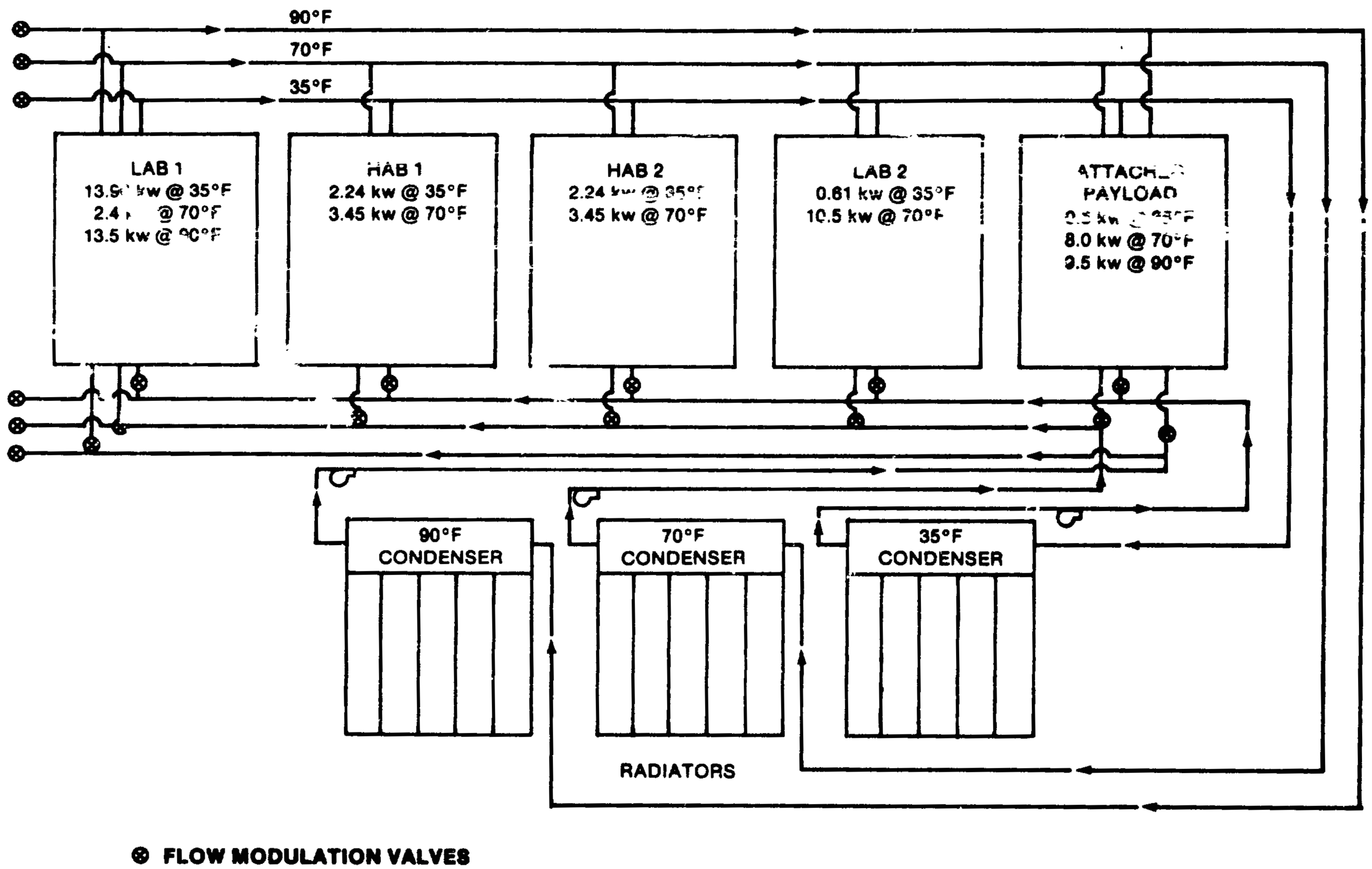


Figure 4.4.7-1 - EXTERNAL THERMAL BUS SCHEMATIC

TABLE 4.4.7-4
TWO PHASE AMMONIA EXTERNAL BUS
THERMAL TRANSPORT CHARACTERISTICS

EXTERNAL BUS-IOC	35F(2C)	70F(21C)	90F(32C)	115F(46C)	TOTAL
TOTAL Q	19.55	27.80	23.00	0.00	70.35
MASS FLOW (LB/HR)	123.46	186.79	160.96	0.00	
LIQ LINE ID (IN.)	0.19	0.22	0.23	0.00	
VAP LINE ID (IN.)	0.63	0.54	0.42	0.00	
PUMP POWER (WATTS)	25.00	35.30	23.50	0.00	83.80
PMP PWR PENALTY (LBS)	8.80	12.40	8.20	0.00	29.40
PUMP PACK WT (LBS)	19.40	23.30	21.00	0.00	63.70
LIQ LINE WT (LBS)	28.10	32.00	10.90	0.00	71.00
VAP LINE WT (LBS)	86.20	72.90	19.60	0.00	178.70
FLUID WT (LBS)	1.40	1.80	0.90	0.00	4.10
VALVE WT (LBS)	30.00	30.00	12.00	0.00	72.00
COLDPLATE WT (LBS)	0.00	0.00	0.00	0.00	0.00
HEAT EX WT (LBS)	0.00	0.00	0.00	0.00	0.00
TOT TRANS WT (LBS)	173.90	172.40	72.60	0.00	418.90
WET WEIGHT (LBS)	165.10	160.00	64.40	0.00	389.50
TOTAL DRY WT (LBS)	163.70	158.20	63.50	0.00	385.40

TABLE 4.4.7-5
TWO PHASE WATER
THERMAL TRANSPORT CHARACTERISTICS

HAB 1 MODULE	35F(2C)	70F(21C)	90F(32C)	115F(46C)	TOTAL
TOTAL Q	2.24	3.45	0.00	0.00	5.69
MASS FLOW (LB/HR)	7.12	11.16	0.00	0.00	
LIQ LINE ID (IN.)	0.19	0.19	0.00	0.00	
VAP LINE ID (IN.)	1.85	1.46	0.00	0.00	
PUMP POWER (WATTS)	0.10	0.20	0.00	0.00	0.30
PMP PWR PENALTY (LBS)	0.00	0.10	0.00	0.00	0.10
PUMP PACK WT (LBS)	11.10	11.70	0.00	0.00	22.80
LIQ LINE WT (LBS)	0.60	0.60	0.00	0.00	1.20
VAP LINE WT (LBS)	11.90	9.40	0.00	0.00	21.30
FLUID WT (LBS)	0.40	0.40	0.00	0.00	0.80
VALVE WT (LBS)	18.00	24.00	0.00	0.00	42.00
COLDPLATE WT (LBS)	44.80	69.00	0.00	0.00	113.80
HEAT EX WT (LBS)	14.30	22.10	0.00	0.00	36.40
TOT TRANS WT (LBS)	101.10	137.30	0.00	0.00	238.40
WET WEIGHT (LBS)	101.10	137.20	0.00	0.00	238.30
TOTAL DRY WT (LBS)	100.70	136.80	0.00	0.00	237.50

TWO PHASE WATER
THERMAL TRANSPORT CHARACTERISTICS

HAB 2 MODULE	35F(2C)	70F(21C)	90F(32C)	115F(46C)	TOTAL
TOTAL Q	2.24	3.45	0.00	0.00	5.69
MASS FLOW (LB/HR)	7.12	11.16	0.00	0.00	
LIQ LINE ID (IN.)	0.19	0.19	0.00	0.00	
VAP LINE ID (IN.)	1.85	1.46	0.00	0.00	
PUMP POWER (WATTS)	0.10	0.20	0.00	0.00	0.30
PMP PWR PENALTY (LBS)	0.00	0.10	0.00	0.00	0.10
PUMP PACK WT (LBS)	11.10	11.70	0.00	0.00	22.80
LIQ LINE WT (LBS)	0.60	0.60	0.00	0.00	1.20
VAP LINE WT (LBS)	11.90	9.40	0.00	0.00	21.30
FLUID WT (LBS)	0.40	0.40	0.00	0.00	0.80
VALVE WT (LBS)	18.00	24.00	0.00	0.00	42.00
COLDPLATE WT (LBS)	44.80	69.00	0.00	0.00	113.80
HEAT EX WT (LBS)	14.30	22.10	0.00	0.00	36.40
TOT TRANS WT (LBS)	101.10	137.30	0.00	0.00	238.40
WET WEIGHT (LBS)	101.10	137.20	0.00	0.00	238.30
TOTAL DRY WT (LBS)	100.70	136.80	0.00	0.00	237.50

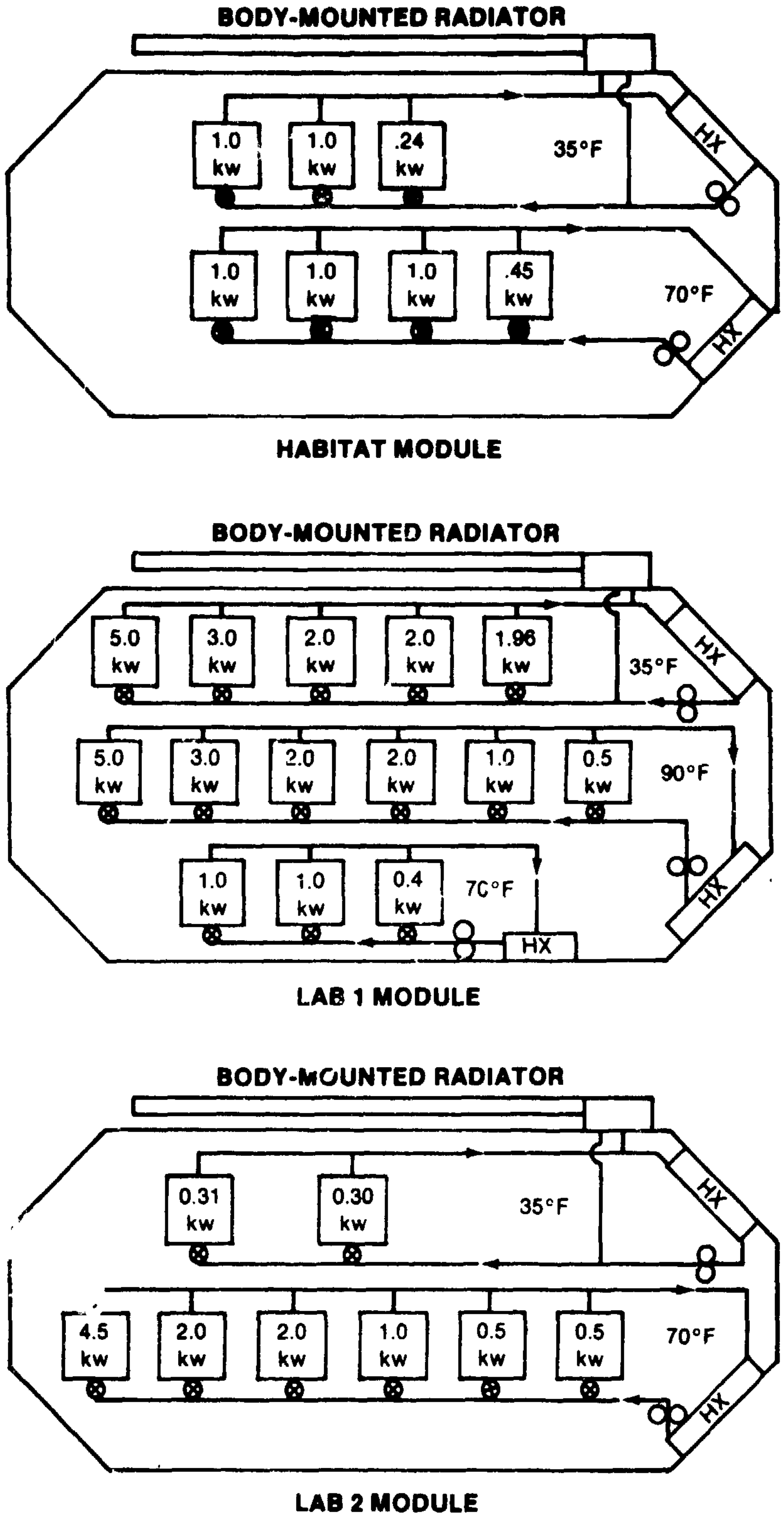


Figure 4.4.7-2 (a) - INTERNAL THERMAL TRANSPORT SCHEMATICS

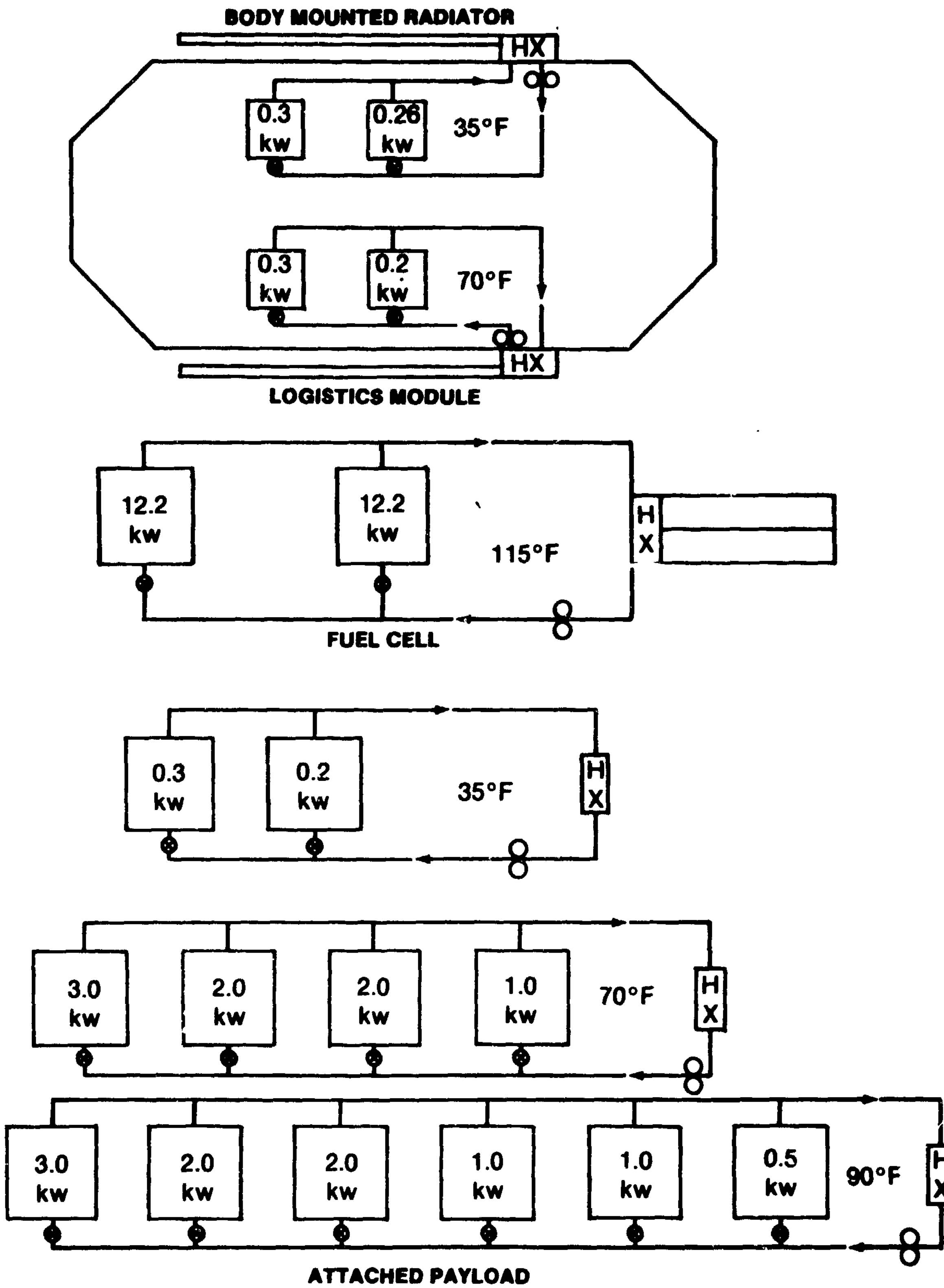


Figure 4.4.7.-2 (b) - INTERNAL THERMAL TRANSPORT SCHEMATICS

TABLE 4.4.7-6
TWO PHASE WATER
THERMAL TRANSPORT CHARACTERISTICS

LAB 1 MODULE	35F(2C)	70F(21C)	90F(32C)	115F(46C)	TOTAL
TOTAL Q	13.96	2.40	13.50	0.00	29.86
MASS FLOW (LB/HR)	44.35	7.76	44.12	0.00	
LIQ LINE ID (IN.)	0.19	0.19	0.19	0.00	
VAP LINE ID (IN.)	3.88	1.31	1.94	0.00	
PUMP POWER (WATTS)	0.80	0.10	0.80	0.00	1.70
PMP PWR PENALTY (LBS)	0.30	0.00	0.30	0.00	0.60
PUMP PACK WT (LBS)	16.70	11.20	16.50	0.00	44.40
LIQ LINE WT (LBS)	0.60	0.60	0.60	0.00	1.80
VAP LINE WT (LBS)	40.00	8.10	12.50	0.00	60.60
FLUID WT (LBS)	0.40	0.40	0.40	0.00	1.20
VALVE WT (LBS)	30.00	18.00	36.00	0.00	84.00
COLDPLATE WT (LBS)	279.20	48.00	270.00	0.00	597.20
HEAT EX WT (LBS)	89.30	15.40	86.40	0.00	191.10
TOT TRANS WT (LBS)	456.50	101.70	422.70	0.00	980.90
WET WEIGHT (LBS)	456.20	101.70	422.40	0.00	980.30
TOTAL DRY WT (LBS)	455.80	101.30	422.00	0.00	979.10

TWO PHASE WATER
THERMAL TRANSPORT CHARACTERISTICS

LAB 2 MODULE	35F(2C)	70F(21C)	90F(32C)	115F(46C)	TOTAL
TOTAL Q	0.61	10.50	0.00	0.00	11.11
MASS FLOW (LB/HR)	1.94	33.95	0.00	0.00	
LIQ LINE ID (IN.)	0.19	0.19	0.00	0.00	
VAP LINE ID (IN.)	1.33	2.28	0.00	0.00	
PUMP POWER (WATTS)	0.10	0.60	0.00	0.00	0.70
PMP PWR PENALTY (LBS)	0.00	0.20	0.00	0.00	0.20
PUMP PACK WT (LBS)	10.30	15.00	0.00	0.00	25.30
LIQ LINE WT (LBS)	0.60	0.60	0.00	0.00	1.20
VAP LINE WT (LBS)	8.20	16.20	0.00	0.00	24.40
FLUID WT (LBS)	0.40	0.40	0.00	0.00	0.80
VALVE WT (LBS)	12.00	36.00	0.00	0.00	48.00
COLDPLATE WT (LBS)	12.20	210.00	0.00	0.00	222.20
HEAT EX WT (LBS)	3.90	67.20	0.00	0.00	71.10
TOT TRANS WT (LBS)	47.60	345.60	0.00	0.00	393.20
WET WEIGHT (LBS)	47.60	345.40	0.00	0.00	393.00
TOTAL DRY WT (LBS)	47.20	345.00	0.00	0.00	392.20

The internal thermal transport systems were sized for an IOC capability. However, some excess capability will ultimately be designed into the systems. This growth potential will be a function of the growth projections for the Space Station, i.e., expansion of existing module capabilities or addition of supplemental modules. The results presented for the IOC thermal system represent minimum capabilities.

A summary of significant results for the reference IOC configuration are given in Table 4.4.7-8(a). As is evident, pumping power requirements are quite low and the fluid inventory relatively small. It should be noted, however, that fluid accumulator quantities are not included in the analysis.

4.4.7.2.4 Design sensitivities

In order to evaluate the sensitivity of the thermal acquisition/transport system weight and pumping power to different design parameters, alternate working fluids were incorporated into the analysis. Single versus two-phase transport systems were also evaluated. Subsequently, integrated transport systems were composed of the various fluids and fluid phases.

4.4.7.2.4.1 External centralized thermal bus alternatives

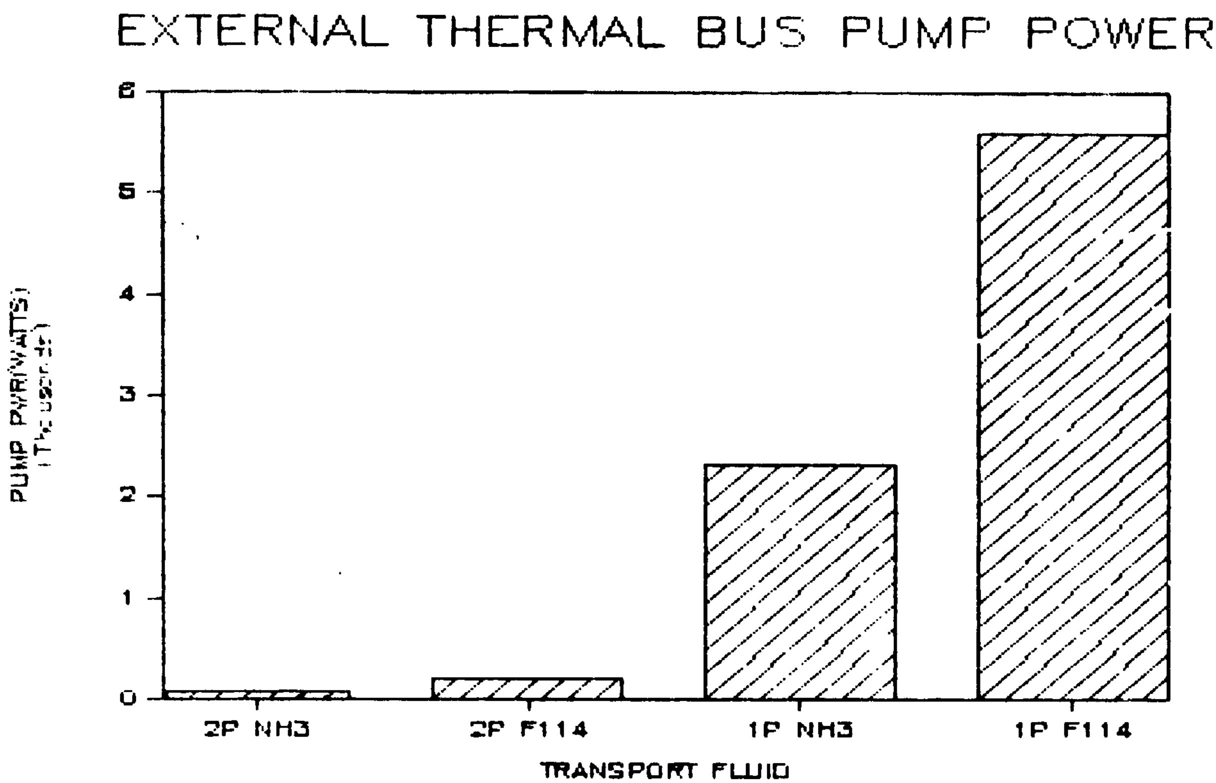
Two-phase ammonia was chosen as the working fluid for the centralized thermal bus because of its attractive thermal characteristics such as its large heat of vaporization, and low freezing temperature. Ammonia operating strictly as a liquid would also be a possible consideration for thermal transport because of its high specific heat, so this was evaluated as well. Additionally, Freon-114 was assessed in a one- and two-phase operation for the external bus. Figures 4.4.7-3(a) and 4.4.7-3(b) illustrate the comparability between the fluids and their working phases for pump power, system weight, and maximum line size at IOC. In all cases, an isothermality of +5°F was maintained within the transport loop. As is shown, the two-phase ammonia requires significantly less pumping power, lower total weight, and smaller line sizes.

4.4.7.2.4.2 Internal thermal transport alternatives

Two-phase water was selected as the working fluid for the module internal thermal transport systems because of its nontoxicity, high specific heat, and extremely large heat of vaporization. However, the disadvantages of water are its high freezing point and low vapor pressure. Consequently, if toxicity requirements were to be relaxed or redefined, there is a possibility that another fluid such as one in the Freon class could be considered as an alternative. To illustrate how this would affect module thermal transport system pump power, system weight, and line sizes, an analysis was accomplished for a single and two-phase Freon system as well as for a single phase water system.

TABLE 4.4.7-8(A)
SUMMARY OF IOC SPACE STATION
THERMAL TRANSPORT CHARACTERISTICS

ELEMENT	PUMP PWR (WATTS)	FLUID WT (LBS)	DRY WT (LBS)	TOT TRANS WT (LBS)
CENTRAL BUS	83.80	4.10	385.40	389.50
HAB 1	0.30	0.80	237.50	238.30
HAB 2	0.30	0.80	237.50	238.30
LAB 1	1.70	1.20	979.10	980.30
LAB 2	0.70	0.80	392.20	393.00
PAYLOAD	5.40	0.60	589.30	589.90
LOGISTICS	0.20	0.80	84.60	85.40
POWER MODULES	24.10	0.90	708.10	709.00
IOC TOTAL	116.50	10.00	3613.70	3623.70



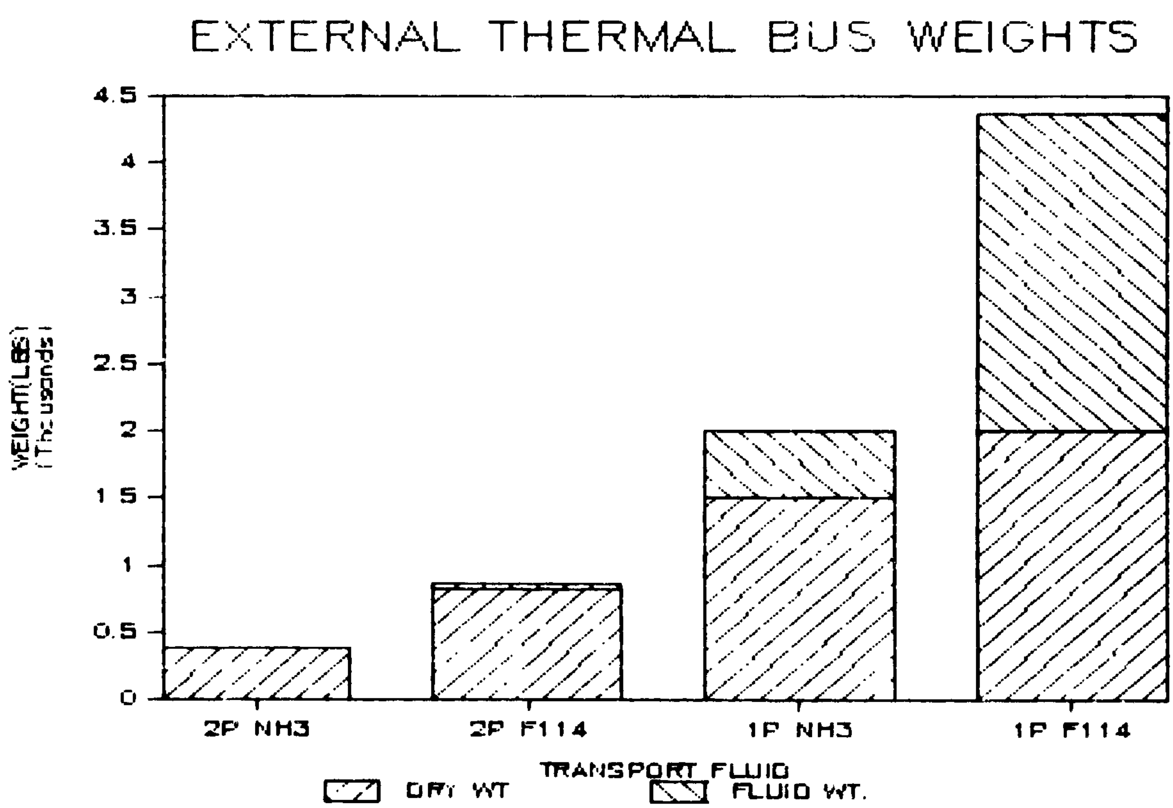


FIGURE 4.4.7-3(a)

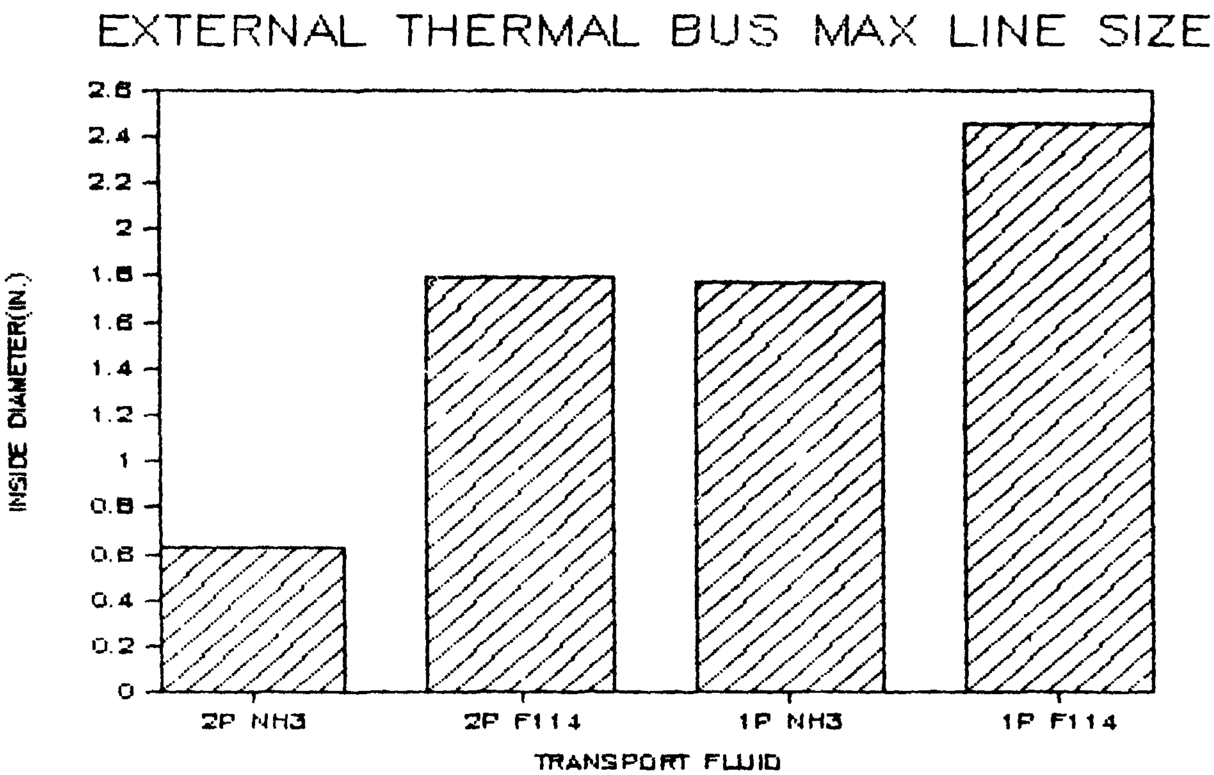


FIGURE 4.4.7-3(b)

Figures 4.4.7-4(a) and 4.4.7-4(b) present a comparison between the one- and two-phase water and the one- and two-phase Freon-114 systems. As shown, two-phase Freon-114 thermal transport lines would be considerably smaller than the two-phase water lines. Also, system weights would be somewhat lighter with Freon due to aluminum rather than stainless steel being used in the plumbing system. Pump power would be higher with the Freon system, however. The single phase water and Freon thermal transport system require much higher pump power with higher system weight because of the increased fluid inventory and flow rates.

4.4.7.2.4.3 Integrated thermal transport alternatives

Various combinations of the external and internal thermal transport schemes were composed to assess their viability for optimizing the thermal control system for the reference configuration. Hybrid systems which were created for analysis are as follows:

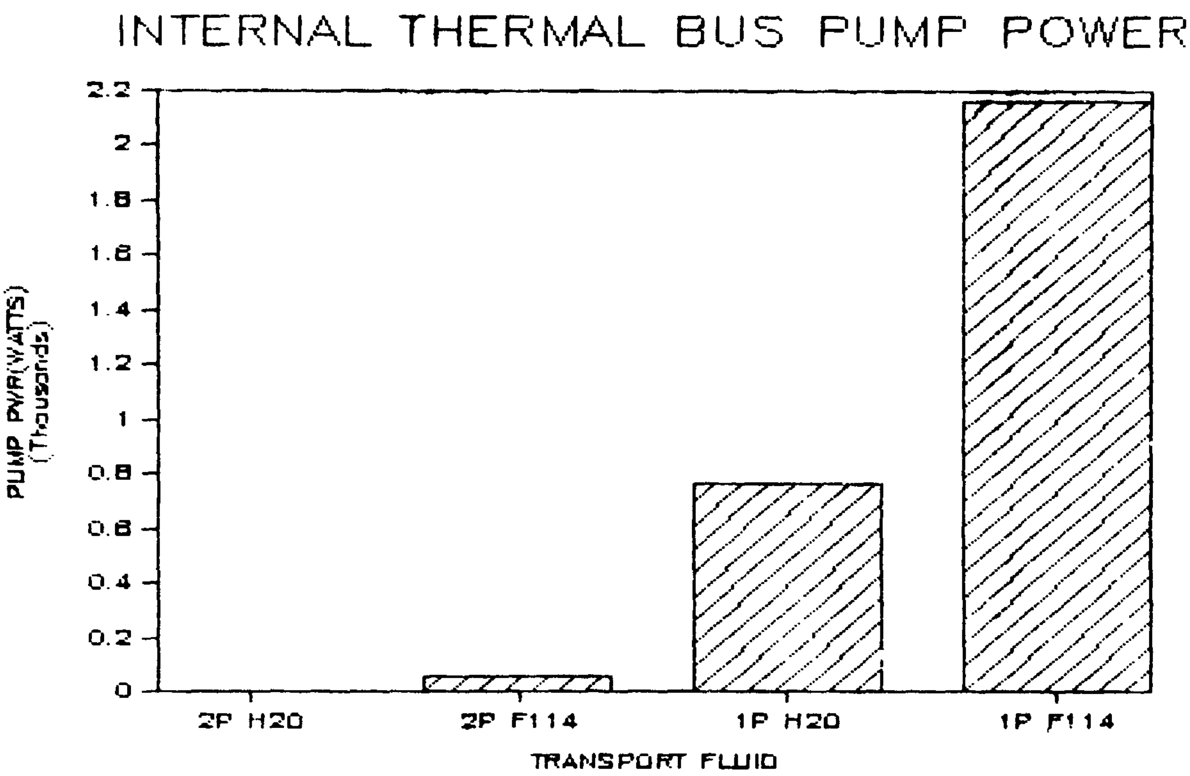
Case No.	External Thermal Bus/Internal Thermal Transport
1	Single-phase Freon-114/Single-phase Freon-114
2	Single-phase ammonia/Single-phase water
3	Two-phase ammonia/Single-phase water
4	Two-phase ammonia/Two-phase water
5	Two-phase ammonia/Two-phase water (70°F & 90°F loads combined)
6	Two-phase Freon-114/Two-phase water
7	Two-phase ammonia/Two-phase Freon-114
8	Two-phase Freon-114/Two-phase Freon-114

Figures 4.4.7-5(a) and 4.4.7-5(b) compare these thermal transport systems in terms of total system weight, maximum line diameter, and pumping power. All of the two-phase systems are comparable in terms of weight, although the total Freon-114 system is the lightest because interface heat exchangers between the central loop and module loop would not be required. The all single-phase systems would be considerably heavier with higher pumping power requirements.

4.4.7.2.5 External thermal bus/module thermal transport system interconnects

Module heat loads will be collected by the central thermal bus to be rejected by the central radiators. Conceptually, the station modules should be connected in parallel to the central bus in order to minimize pressure drop, to minimize two-phase flow in a single conduit, and to minimize system complexity for accommodating load sharing. There are, however, a variety of concepts for connecting these modules to facilitate system growth and to minimize single-point failures. A minimization of fluid disconnects and reduction of system complexity is also required. Additionally, EVA activity for connection of the module/payload thermal loads into the central system should be minimal.

In consonance with the previous requirements, a number of interconnection concepts have been proposed. Functional diagrams of these, with positive



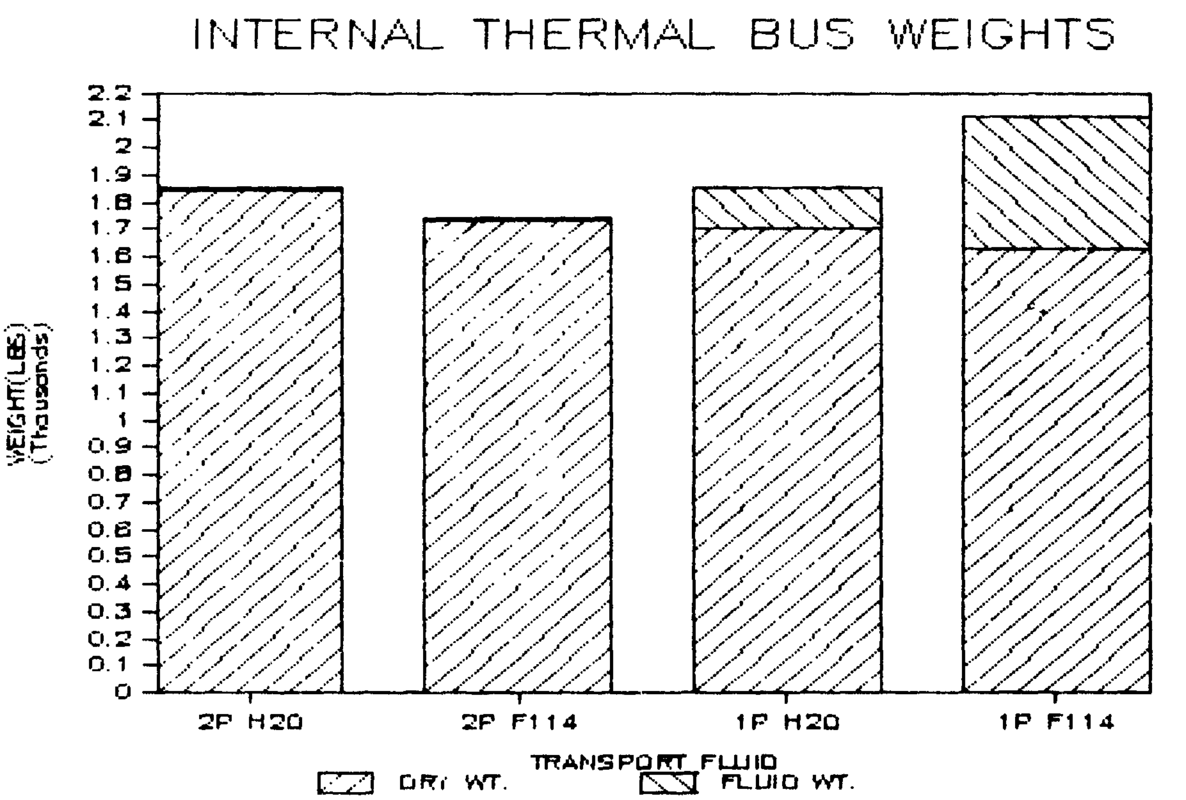


FIGURE 4.4.7-4(a)

FIGURE 4.4.7-4(b)

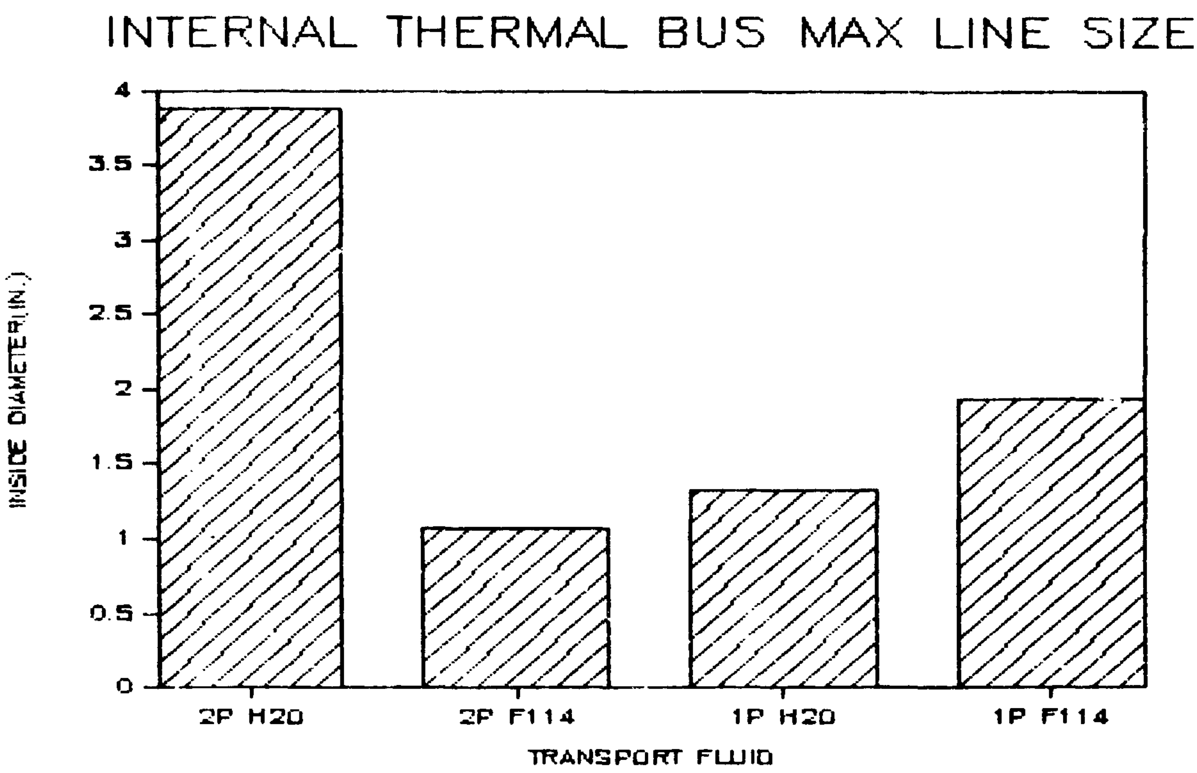
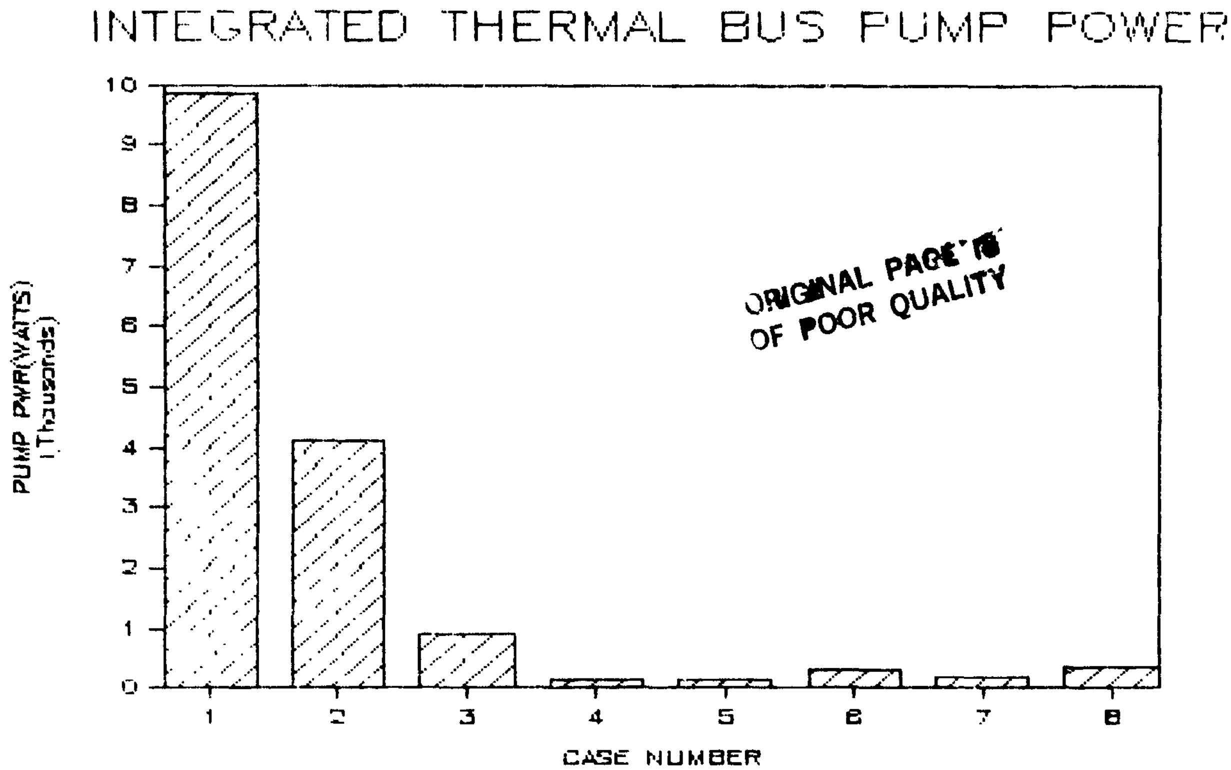


FIGURE 4.4.7-4(b)



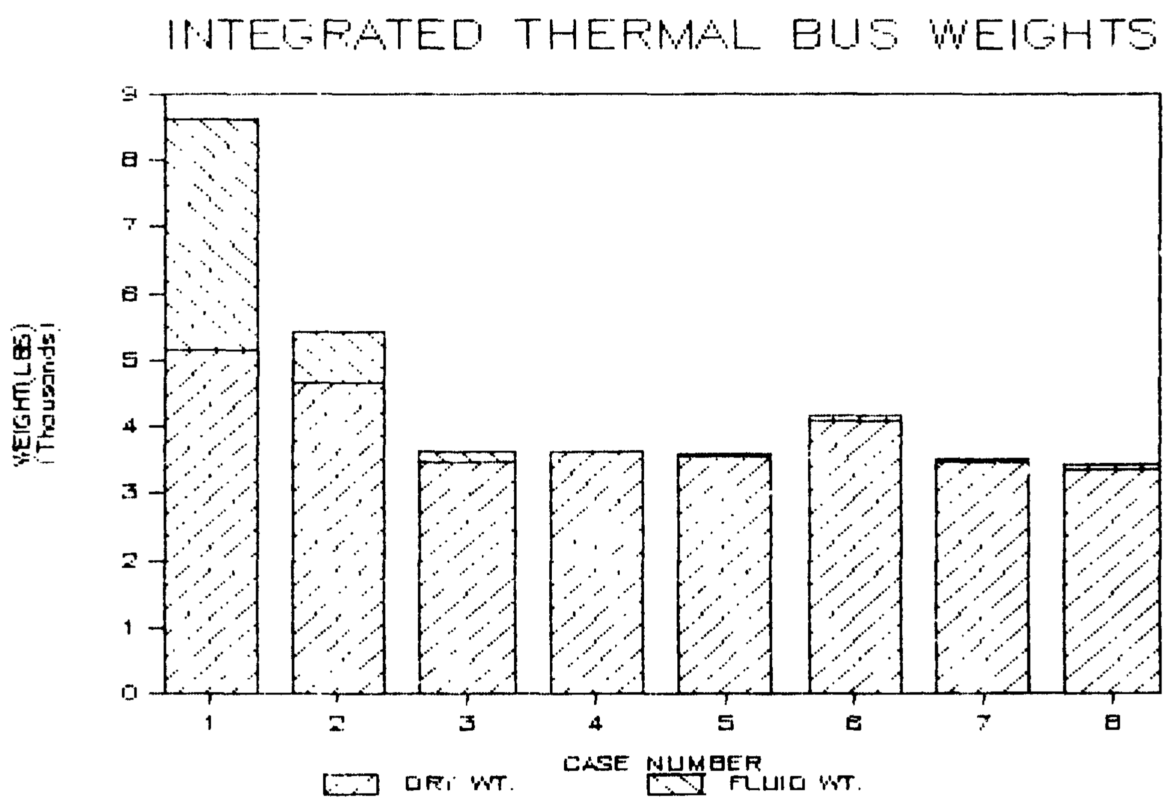


FIGURE 4.4.7-5(a)

FIGURE 4.4.7-5(b)

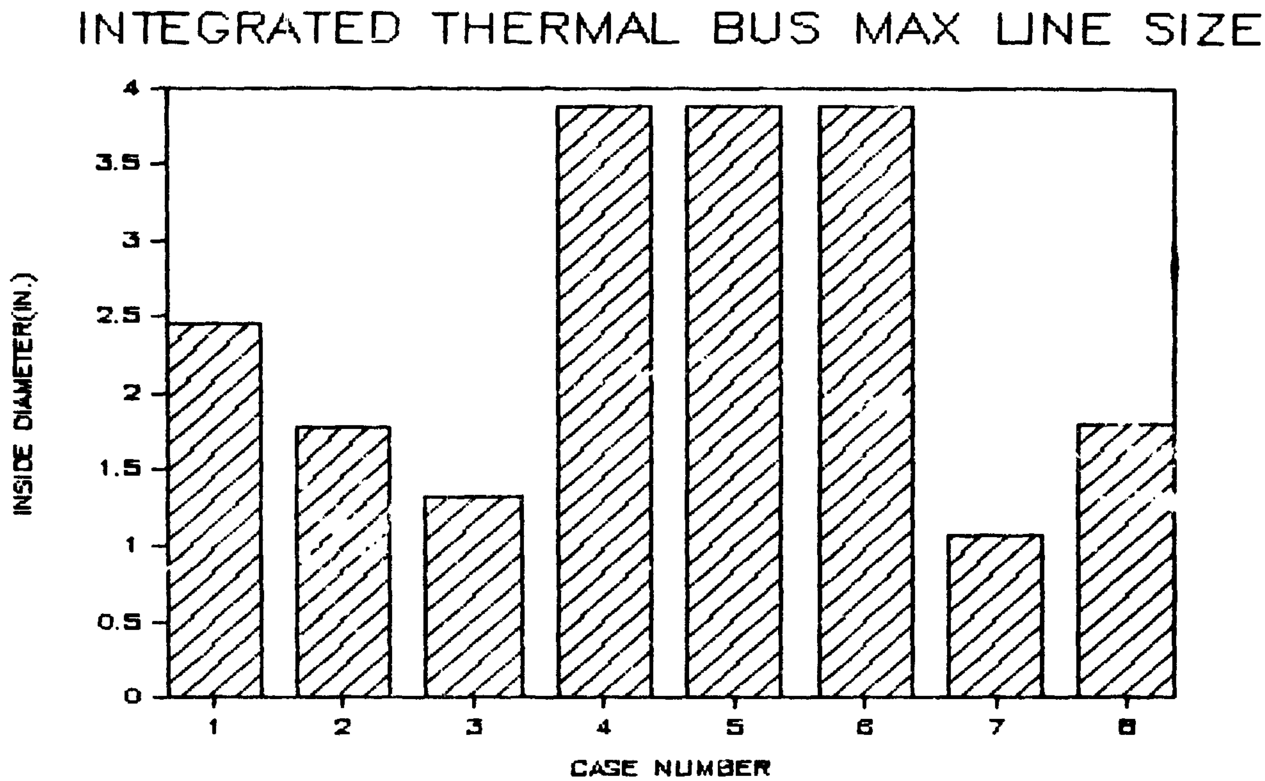


FIGURE 4.4.7-5(b)

and negative attributes denoted, are shown in Figure 4.4.7-6(a) and (b). As shown, a flow control assembly (FCA) consisting of a pump, accumulator, and regenerative heat exchanger is included in each concept. This can either take the form of an autonomous, modular package or as module-integrated hardware. Module interconnects to the central bus in each concept can be accomplished with fluid couplings external to the habitable volume, with contact heat exchangers (also external to the habitable volume), or with an intermediate nontoxic loop using fluid couplings internal to the habitable volume. The lightest approach would utilize external fluid couplings and the heaviest would use contact heat exchangers. For this study, a fluid coupling approach was assumed for sizing purposes. However, although weight is an important criterion, factors such as EVA support and reliability prevent fluid couplings from being the most desirable approach.

Therefore, an indepth review of the proposed connection schemes is required with a trades analysis being accomplished to establish the optimum candidate. This was not accomplished during this brief study, though the criticality and complexity of this task was identified.

4.4.7.3 Heat Rejection

4.4.7.3.1 Sizing approach

The reference configuration heat rejection system consists of high capacity, monogroove, heat pipe radiator panels, as shown in Figure 4.4.7-7. In this approach, the radiators will be erected in space with an MRMS as illustrated by Figure 4.4.7-8. An individual radiator element can be removed and replaced if damaged. The heat pipe radiator elements are "plugged in" to contact heat exchangers as shown by Figure 4.4.7-9 which illustrates one concept under development. These heat exchangers provide a loose fit for the heat pipes when they are initially plugged in. A clamping action is then provided by the contact heat exchanger, thus giving the contact force needed for good heat transfer contact conductance.

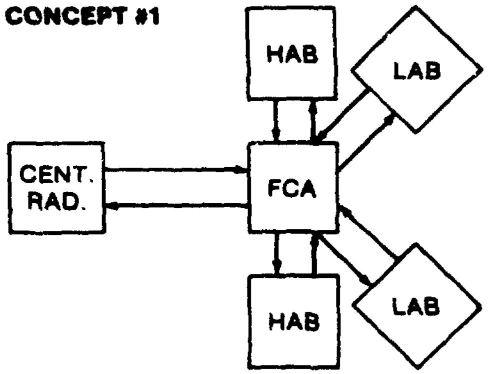
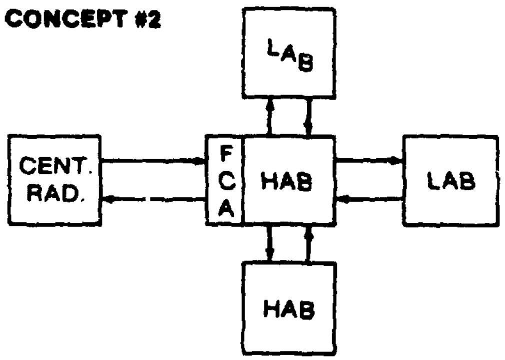
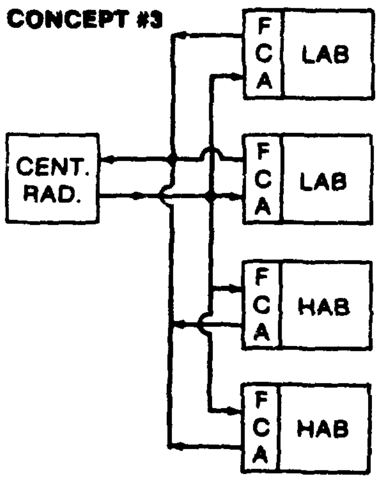
The heat rejection system illustrated by Figure 4.3.5-1 was divided into four functional segments:

a. A centralized system which rejects the majority of the station heat load except for power system, logistics module, and passively rejected heat loads

b. A power system radiator which rejects the fuel cell and power conditioning heat loads with a 115°F heat load interface temperature

c. A body-mounted logistics module radiator system which rejects the entire logistics module heat load

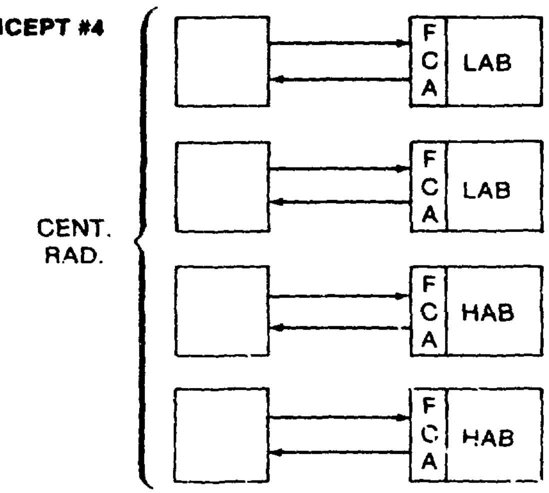
d. A body-mounted radiator on each of the remaining modules which interfaces with the 35°F central transport loop to provide autonomous safe haven thermal control for each module

	PROS	CONS
CONCEPT #1 	• TOTAL CENT. RADIATOR AVAILABLE IF STATION MODULE FAILS	• VULNERABLE TO SINGLE POINT FAILURE (i.e., FCA) • REQUIRES ANOTHER MODULE (i.e., FCA) TO ACCOMMODATE DURING LAUNCH & ASSEMBLY • MAY REQUIRE ANOTHER FCA
CONCEPT #2 	• TOTAL CENT. RADIATOR AVAILABLE IF STATION MODULE FAILS	• VULNERABLE TO SINGLE POINT FAILURE (i.e., FCA) • REQUIRES FCA IN ANOTHER MODULE FOR SAFE HAVEN
CONCEPT #3 	• TOTAL CENT. RADIATOR AVAILABLE IF STATION MODULE FAILS	• VULNERABLE TO SINGLE POINT FAILURE AT EXTERNAL PLUMBING MANIFOLD • REQUIRES ADDITIONAL FCA's • COMPLICATES EXTERNAL PLUMBING HOOKUP

NOTE: FCA - "FLOW CONTROL ASSEMBLY" CONSISTS OF PUMP/ACCUM. AND REGENERATIVE HX.

Figure 4.4.7-6 (a) - CENTRAL BUS PLUMBING CONCEPTS

CONCEPT #4



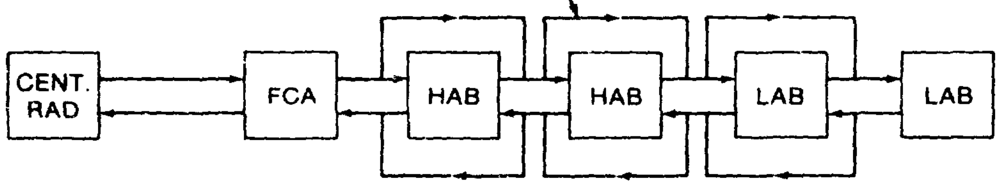
PROS

- PROVIDES DECENTRALIZED CAPABILITY WITHOUT BODY-MT. RADIATORS
- ACCOMMODATES GROWTH EASILY
- TOTAL CENT. RADIATOR AVAILABLE IF STATION MODULE FAILS

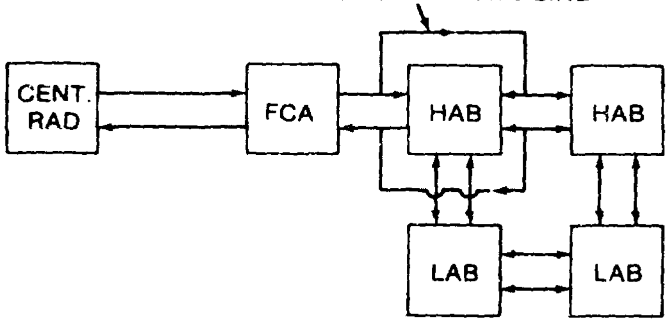
CONS

- LARGE NO. OF PLUMBING LINES THRU GIMBAL
- TOTAL CENT. RADIATOR NOT AVAILABLE (WITHOUT ADDITIONAL MANIFOLDING & VALVES) IF STATION MODULE FAILS
- REQUIRES ADDITIONAL FCA's
- VULNERABLE TO SINGLE POINT FAILURE (i.e., FCA)
- MAY REQUIRE ANOTHER FCA
- REQUIRES ADDITIONAL PLUMBING & COMPLICATES HOOKUP

CONCEPT #5



CONCEPT #6



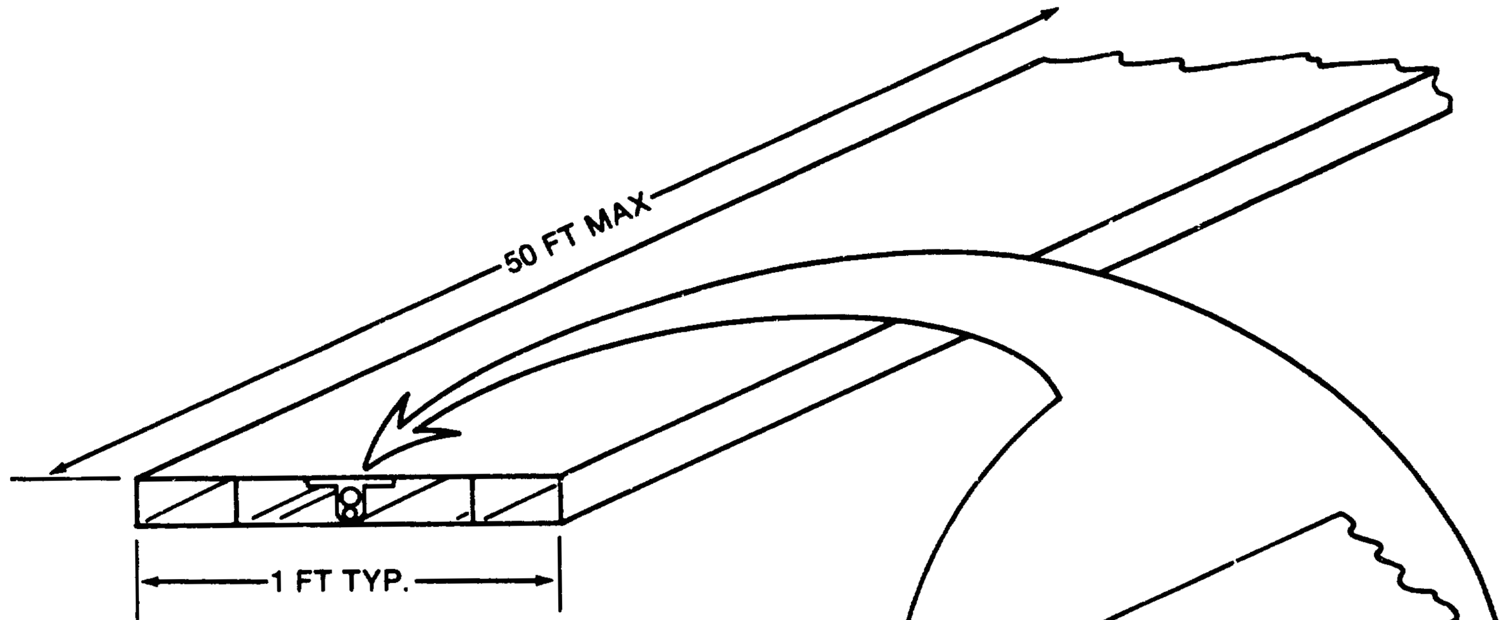
PROS

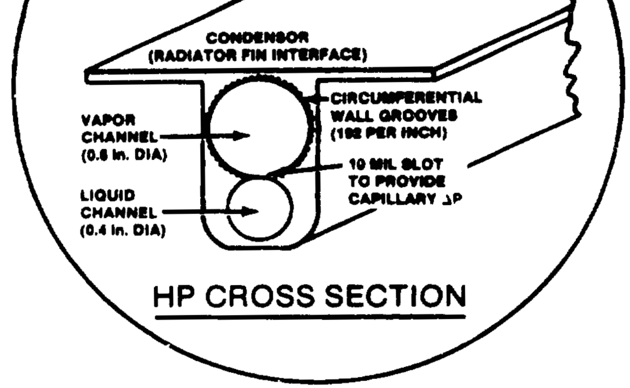
- TOTAL CENT. RADIATOR AVAILABLE IF STATION MODULE FAILS

CONS

- COMPLICATED PLUMBING HOOKUP
- VULNERABLE TO SINGLE POINT FAILURE (i.e., FCA)
- MAY REQUIRE ANOTHER FCA
- PROPER FLOW DISTRIBUTION MAY BE PROBLEM FOR FLOW IN EITHER DIRECTION OPERATION

Figure 4.4.7-6 (b) - CENTRAL BUS PLUMBING CONCEPTS





- **VERY HIGH CAPACITY - DUE TO SEPARATE, LOW PRESSURE DROP LIQUID FLOW CHANNEL**
- **VERY LIGHTWEIGHT - ESSENTIALLY AN AL TUBE**
- **VERY SIMPLE CONCEPT - NO WICKS, NO VARIABLE CONDUCTANCE CONTROLS, NO ION PUMPS**
- **RELATIVELY EASY TO MANUFACTURE - PRIMARILY AN EXTRUSION**
- **CAPABLE OF BEING TESTED AT NEAR FULL CAPACITY IN ONE-G**
- **RELATIVELY EASY TO ANALYZE - DUE TO SIMPLICITY OF DESIGN**

Figure 4.4.7-7 - MONOGROOVE HEAT PIPE RADIATOR PANEL

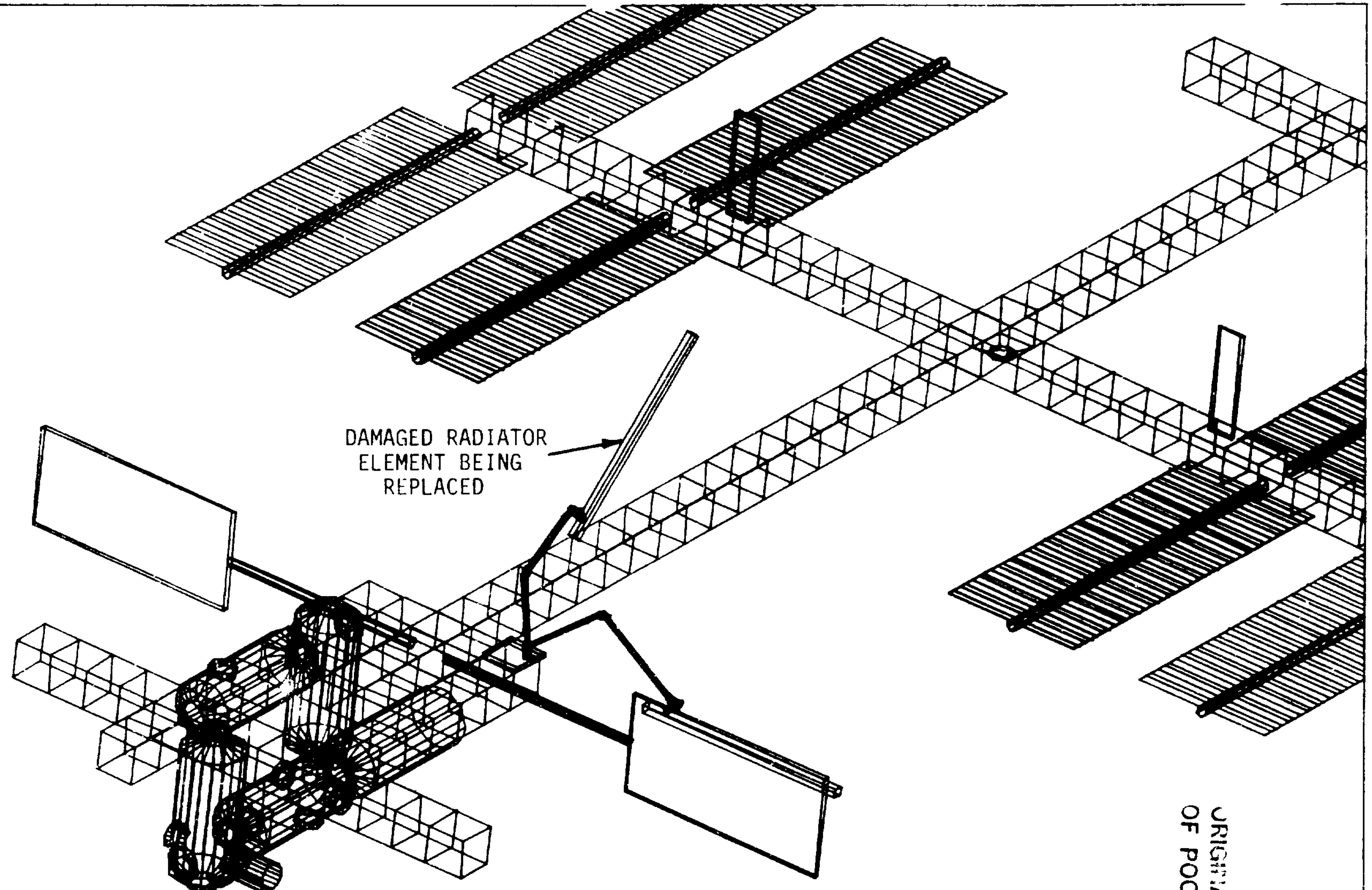


FIGURE 4.4.7-8 - SPACE CONSTRUCTABLE RADIATOR

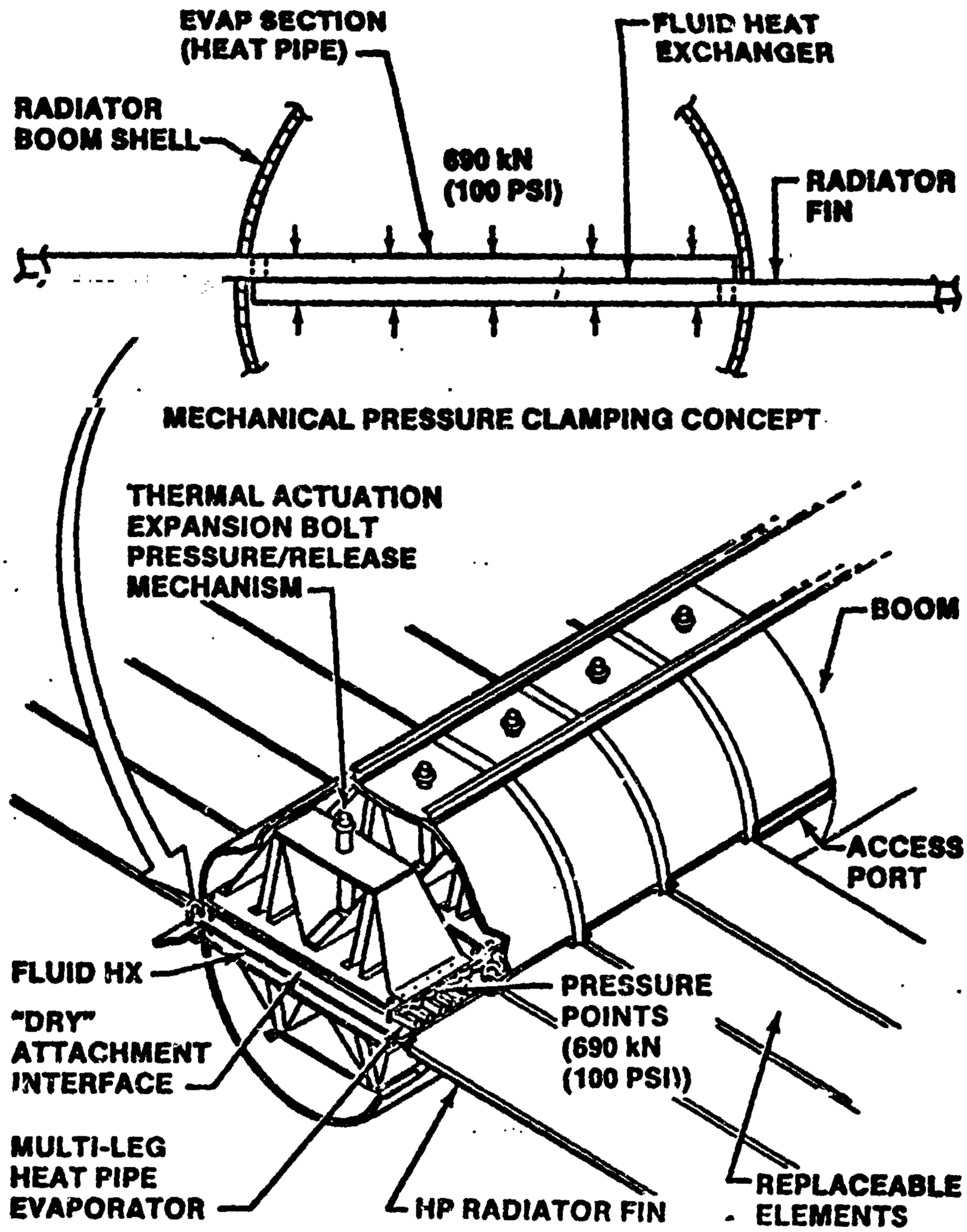


FIGURE 4.4.7-9
MECHANICAL CONTACT HEAT EXCHANGER

4.4.7.3.2 Central radiator system sizing

A central radiator system using a radiator panel fin thickness of .020 inches for the articulated radiators was chosen giving a weight per unit projected area of 1.4 lb/ft^2. The panels were assumed to be 50 feet long and 1 foot wide. Other panel lengths and widths could be used to meet launch packaging and assembly or deployment requirements without significantly degrading thermal performances.

Each panel has approximately 2 feet of additional length for an evaporator section. However, this section's weight was lumped with the contact heat exchanger. A detailed sizing analysis of the contact heat exchanger was not performed. Based on prototype designs, the heat exchanger was assumed to weigh 20 pound per panel and have an overall conductance of 461 BTU/HR-°F-PANEL.

To provide maximum hardware commonality and to simplify the analysis, the same radiator panel and contact heat exchanger design was used for all operating temperatures. Future studies should include optimizations for each temperature to allow an assessment of the reduced thermal performance associated with common hardware.

The central radiators were sized with orbit-averaged incident heat fluxes and view factors to space at $\beta = 0°$. The incident heat fluxes and view factors were obtained from a TRASYS computer program analysis which included the orbital varying radiator/station geometry and multiple diffuse reflections. Station surfaces were assumed to be at the radiator temperature. The radiator view factor to space was found to vary from 0.9 to 0.98 twice an orbit. Additional analysis must be performed to determine if a thermal capacitor or larger radiator is required.

The solar absorptivity was assumed to have an end of life value of 0.2. While the UV degradation rate is expected to be small for the central radiators, the effects of atomic oxygen and contamination have not been quantified and require further analysis.

4.4.7.3.3 Power system radiator sizing

The power system radiator and heat exchanger construction was assumed identical to the central system. These radiators are located on the transverse boom and were found to have a view factor orbital variation of 0.75 to 0.91. The radiators were sized with an orbit average view factor of 0.80

Due to the different efficiencies between fuel cell operation and electrolysis, the IOC power system waste heat load varies between 13 kW on the Sun side to 46 kW on the dark side. The IOC power system radiator was sized for the orbit average heat load of 24.4 kW. An 11.9 kWhr thermal capacitor was used to average the varying heat load, ignoring any inherent thermal capacitance in the system. A detailed sizing analysis of the thermal capacitor was not performed. The capacitor was assumed to weigh 50 pound/kWhr. Of this 50 pound/kWhr, approximately 34 pound/kWhr is required by the phase change wax material. Since the equivalent radiator weight is about 68 pound/kWhr, and the equivalent radiating area is about 76 feet^2/kWhr, the cost of developing such a capacitor may not be worth the 214-pound system weight reduction.

4.4.7.3.4 Logistics module radiator sizing

To eliminate a thermal interface between the station and the logistics module, the logistics module radiators were sized to continuously reject the entire module heat load.

The logistics module body-mounted radiators were assumed to be constructed with an .010-inch aluminum fin. These radiators are not space constructable and were assumed to use a brazed heat exchanger weighing 3 pound/panel. Since these radiators also serve as a meteoroid bumper shield, a weight credit for the meteoroid protection was taken in the radiator weight.

The radiators were located on the windward side of the logistics module and assumed to be 11 feet long. The required width was determined based on orbit-average β =52° incident heat fluxes. A 180° segment of the module circumference was used to reject the module heat load. Due to the incident heat flux orbital variation, a thermal capacitor may be required in this system, but has not been included in this study.

4.4.7.3.5 Safe-haven body radiators

Since safe-haven heat rejection requirements have not been defined, the radiators were assumed to cover a 180° segment of the module circumference. The heat rejection capability of the radiators was then determined with orbit-average incident heat fluxes.

4.4.7.3.6 Heat rejection system results

The radiator area and radiator and heat exchanger weights are summarized in Table 4.4.7-9. The power system capacitor weight of 594 lb is not included in the weight shown in Table 4.4.7-9. These results are presented for the IOC configuration and represent minimum capabilities; however some excess capability should ultimately be designed into the system.

4.4.7.3.7 Design sensitivities

a. Heat pipe radiator fin thickness Previous heat pipe radiator fin thickness optimizations minimized only the radiator panel weight. The heat rejection system weight also includes the heat exchanger weight, which is dependent on the fin thickness through the radiating area required. In addition, launch packaging, on-orbit operations and reboost propellant requirements favor reduced radiator area at the expense of weight. To assess the effects of the fin thickness on the heat rejection weight and area, a system was sized for a range of thicknesses from .010 inch to .030 inch for two-sided and single-sided radiators as summarized by Figure 4.4.7-10(a) and 4.4.7-10(b).

TABLE 4.4.7-9 - HEAT REJECTION SYSTEM RESULTS FOR IOC SPACE STATION

	BUS TEMP DEG F	SOLAR Q B/FT2-HR	IR Q B/FT2-HR	ALPHA	EPS	F TO SPACE	FIN EFF	CONDUCT B/HR-DEG F -PANEL	EFF SINK T DEG F	RAD TEMP DEG F	HEAT LOAD KW	RADIATING AREA, FT2	PANELS REQ'D	RADIATOR WEIGHT,LB	HX WEIGHT,LB	TOTAL WEIGHT,LB
BODY RADS																
LOG	35.00	99.00	21.00	0.20	0.80	0.95	0.79	1000.00	-24.84	34.73	0.56	78.03	7.09	62.43	21.28	83.71
	70.00	99.00	21.00	0.20	0.80	0.95	0.75	1000.00	-19.15	69.57	0.50	43.27	3.93	34.61	11.80	46.41
TOTAL BODY											1.06	121.3	11.0	97.0	33.1	130.1
CENT RADS																
LOW BUS	35.00	12.00	26.00	0.20	0.80	0.93	0.88	461.00	-80.03	26.51	19.54	1704.38	17.04	1193.07	340.88	1533.94
MID BUS	70.00	12.00	26.00	0.20	0.80	0.93	0.86	461.00	-77.84	58.04	27.80	1722.00	17.22	1205.40	344.40	1549.80
HI BUS	90.00	12.00	26.00	0.20	0.80	0.93	0.85	461.00	-76.72	75.81	23.00	1200.86	12.01	840.60	240.17	1080.78
TOTAL CENT											70.34	4627.24	46.27	3239.07	925.45	4164.52
POWER RAD	115.00	13.00	23.00	0.20	0.80	0.80	0.83	461.00	-69.49	100.26	24.40	1225.93	12.26	858.15	245.19	1103.34
TOTAL STA											95.80	5974.48			1203.72	5397.98
SAFE HAVEN BODY RADS																
HAB2	35.00	99.00	21.00	0.20	0.80	0.96	0.79	1000.00	-25.98	34.72	1.77	240.00	21.82	192.00	65.45	257.45
LAB1	35.00	99.00	21.00	0.20	0.80	0.97	0.79	1000.00	-27.10	34.72	1.81	240.00	21.82	192.00	65.45	257.45
LAB2	35.00	105.00	5.40	0.20	0.80	0.64	0.79	1000.00	-21.94	34.83	1.12	240.00	21.82	192.00	65.45	257.45
HAB1	35.00	105.00	5.30	0.20	0.80	0.64	0.79	1000.00	-22.29	34.82	1.12	240.00	21.82	192.00	65.45	257.45
TOTAL SH											5.82	960.00	37.27	768.00	261.82	1029.82

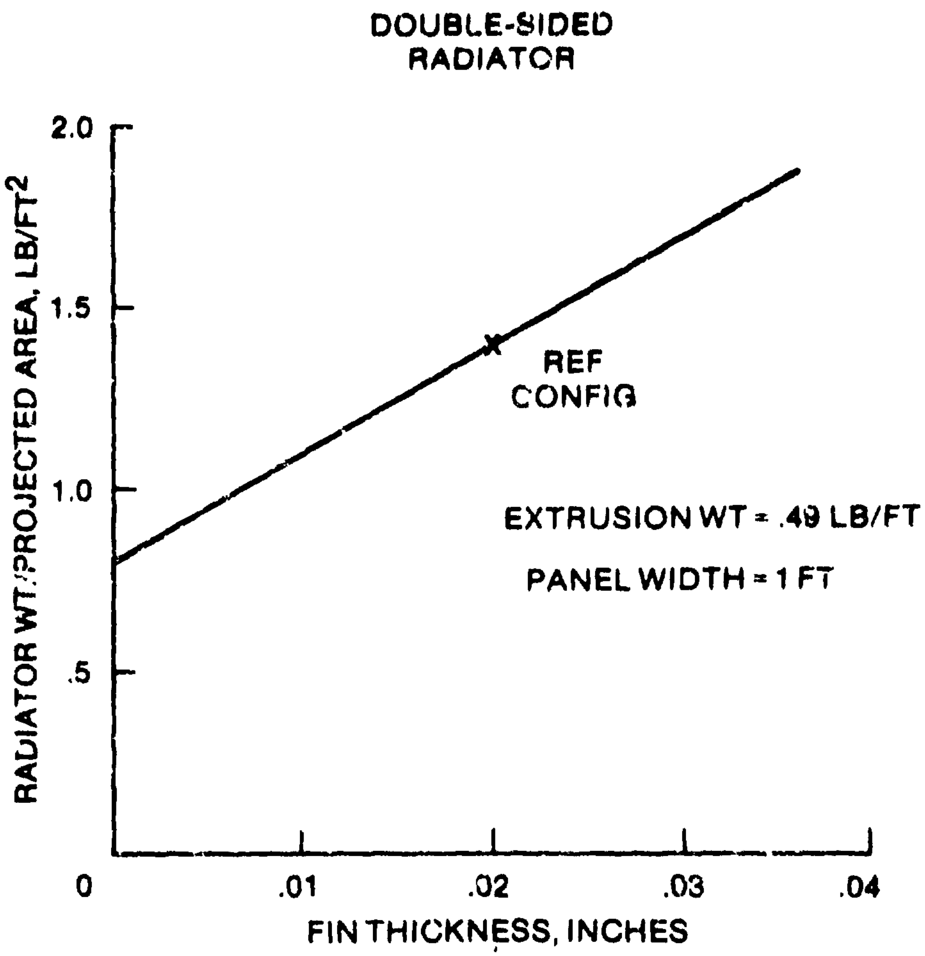


Figure 4.4.7-10 (a) - HEAT PIPE RADIATOR WEIGHT

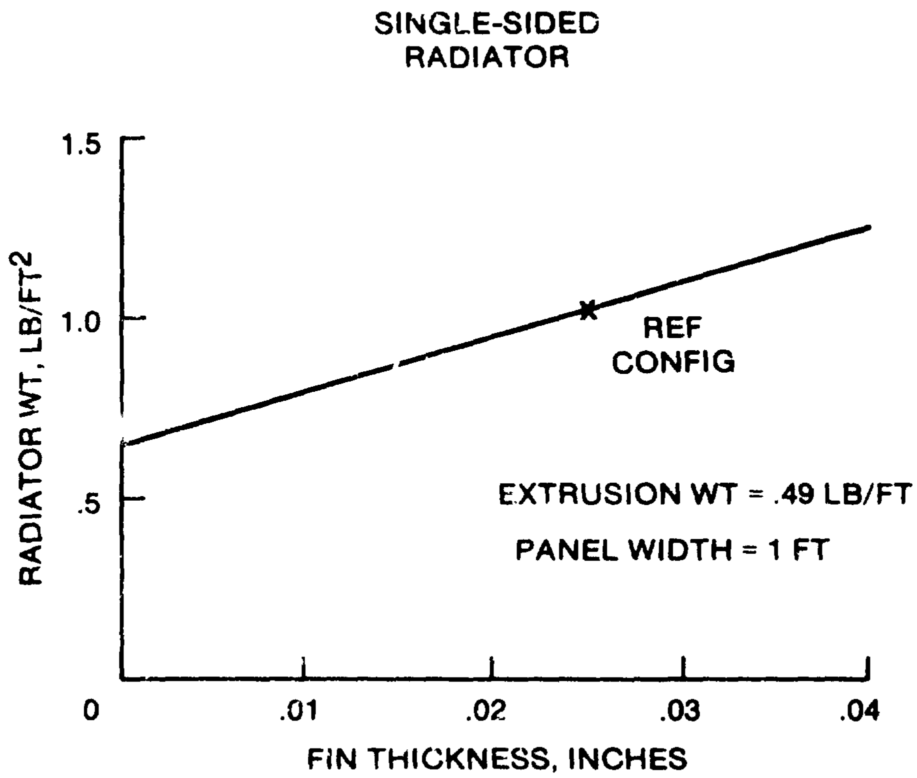


Figure 4.4.7.-10 (b) - HEAT PIPE RADIATOR WEIGHT

The minimum heat rejection system weight of the combined central and power systems was found to occur at a fin thickness of about .015 inch as illustrated by Figure 4.4.7-11(a). Since the sensitivity of the launch packaging, on-orbit operations and reboost costs to the radiator area are unknown, the reference configuration fin thickness was arbitrarily taken to .020 inch rather than the minimum. This thicker fin results in a 3 percent system weight increase while providing a 6 percent area reduction. Figure 4.4.7-11(b) illustrates similar information for the single sided body-mounted radiators. Future studies should attempt to quantify the station cost sensitivities to radiator area.

b. Fixed vs articulated radiators Since there is some risk and development cost associated with the rotating joint required by the articulated radiator concept, a fixed central radiator system was sized for the reference configuration. The fixed central radiator was assumed to be edge to the earth and perpendicular to the orbit plane. Both systems had the same heat load distribution, fin effectiveness, and radiator/heat exchanger weight. Since the fixed radiator will be affected to a greater degree by solar absorptance (α) degradation rate, α values of 0.2 and 0.3 were used.

The results, normalized by the articulated system, are shown below.

Total Central Radiator System Weight Factors

	$\alpha = 0.2$	$\alpha = 0.3$
$\beta = 0°$	1.39	1.9
$\beta = 52°$	1.23	1.45

c. Battery vs. fuel cell Of the 24.4 kW orbit average fuel cell heat rejection, about 16.3 kW are required by the fuel cell itself. One alternative to the fuel cell is a rechargeable battery. During power generation on the dark side, the battery was assumed to produce 13.6 kW of waste heat. During recharge, only a very small waste heat is produced, resulting in an 5.2 kW orbit average heat rejection requirement. In addition, the battery requires a 35°F cooling interface temperature, while the fuel cell system was assumed to use 115°F. The results, excluding the heat rejection requirements for the power conditioning equipment common to the two systems are:

	FUEL CELL	BATTERY
Radiating Area, Ft^2	819	584
Radiator plus H/X Weight, LB	737	526
Capacitor Weight, LB	594	235
Total Heat Rejection Weight, LB	1331	761

d. Solar absorptivity sensitivity Since some uncertainty exists in the solar absorptivity degradation rate, the heat loads and environments shown in Table 4.4.7-9 were applied with a range of absorptivities from the reference configuration value of 0.2 to the fully degraded value of 0.8. The required radiator area relative to the

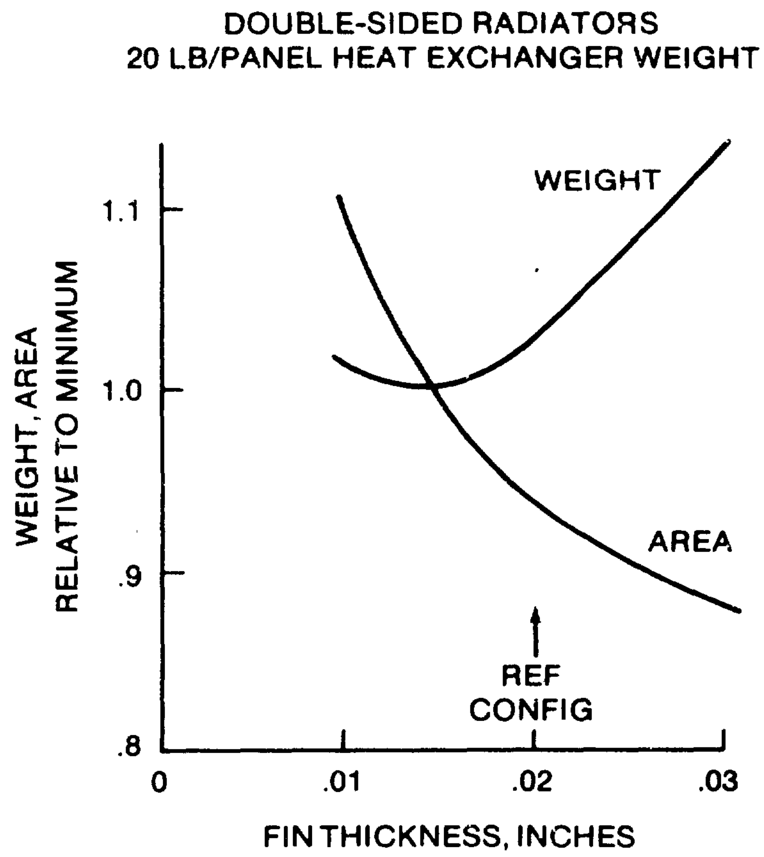


Figure 4.4.7-11 (a) - HEAT REJECTION SYSTEM SENSITIVITY TO FIN THICKNESS

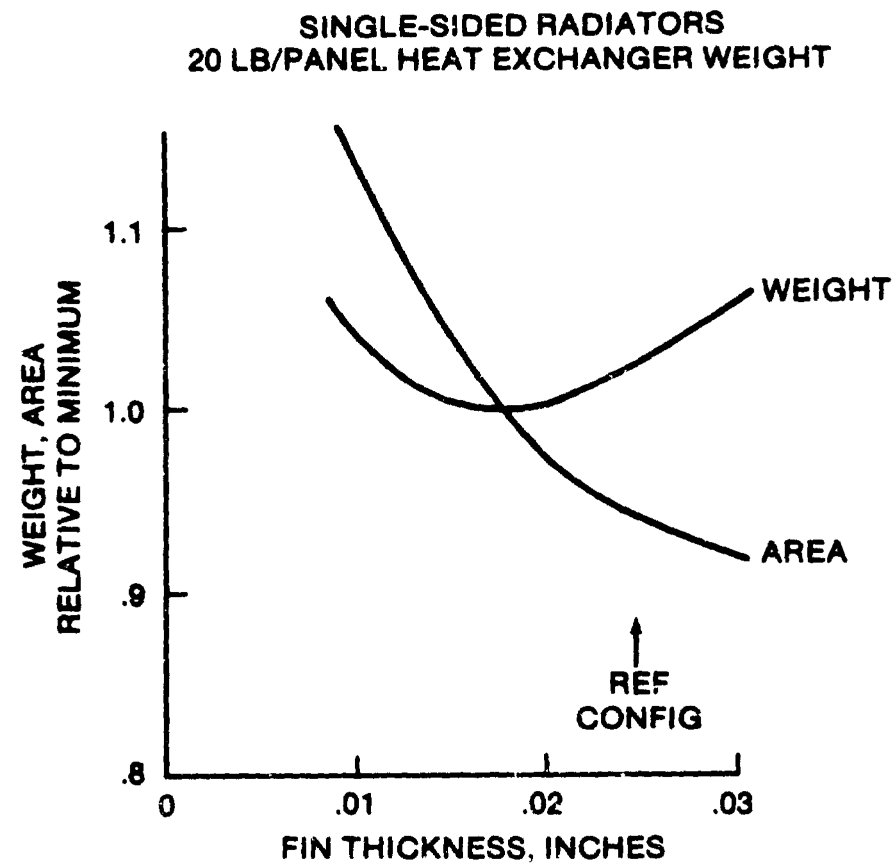


Figure 4.4.7-11 (b) - HEAT REJECTION SYSTEM SENSITIVITY TO FIN THICKNESS

reference configuration is shown in Figure 4.4.7-12 for both the logistics module radiators and the combined central and power system radiators. Also shown is the total body-mounted safe-haven radiator heat rejection capacity for a constant surface area.

Due to the direct solar flux, the body-mounted radiators cannot tolerate an absorptivity degradation much greater than 0.2.

e. Elimination of 90°F thermal loop Since the heat loads rejected by the 90°F loop have allowable lower limits of 68°F (see Table 4.4.7-1), they could satisfactorily be rejected by the 70°F loop. Rejecting these heat loads at a lower temperature would impose a penalty in net increased central radiating areas and weights, but would eliminate extra transport hardware and reduce interfaces for another loop. For the reference configuration, the central radiating area and weight requirements increased by 5 percent when the 90°F loop was eliminated. These small rejection system increases should be traded off against the decreases in transport system complexity and weights.

4.4.7.4 Passive thermal control/protection

The basic approach to the passive thermal design of the Space Station is to desensitize, as far as practical, the station elements and subsystems to the external environment and surface thermal optical property degradation while minimizing makeup heat requirements and thermal load excursions on the Active Thermal Control System.

The thermal surface treatments and insulation performance are an integral part of the overall active and passive design and must be selected to obtain an acceptable heat balance that will not result in local condensation on internal surfaces of pressurized volumes, excessive heat leaks, or violate structural and subsystem temperature limits.

All pressurized modules will be provided with a high performance multilayer insulation (MLI) with an effective emittance of approximately .01 and external surface (with the exception of body mounted radiators) solar absorptance, α , to emittance, ϵ , ratio approaching 1.0. The MLI will consist of approximately 20 layers of organically coated aluminized film with dacron mesh separaters weighing approximately 0.25 pounds per square foot. The outer package layers are porolated to allow for adequate viscous and molecular venting. The selection of an α/ϵ of 1.0 with individual magnitudes of approximately 0.8 should negate surface refurbishment requirements. A similar approach appears appropriate for the power system and external airlocks.

The following summarizes the major station elements insulation weights:

a. Habitation, Laboratory, Logistics modules - 400 pounds each,

b. OTV hangar - 2340 pounds

c. Power system - 130 pounds

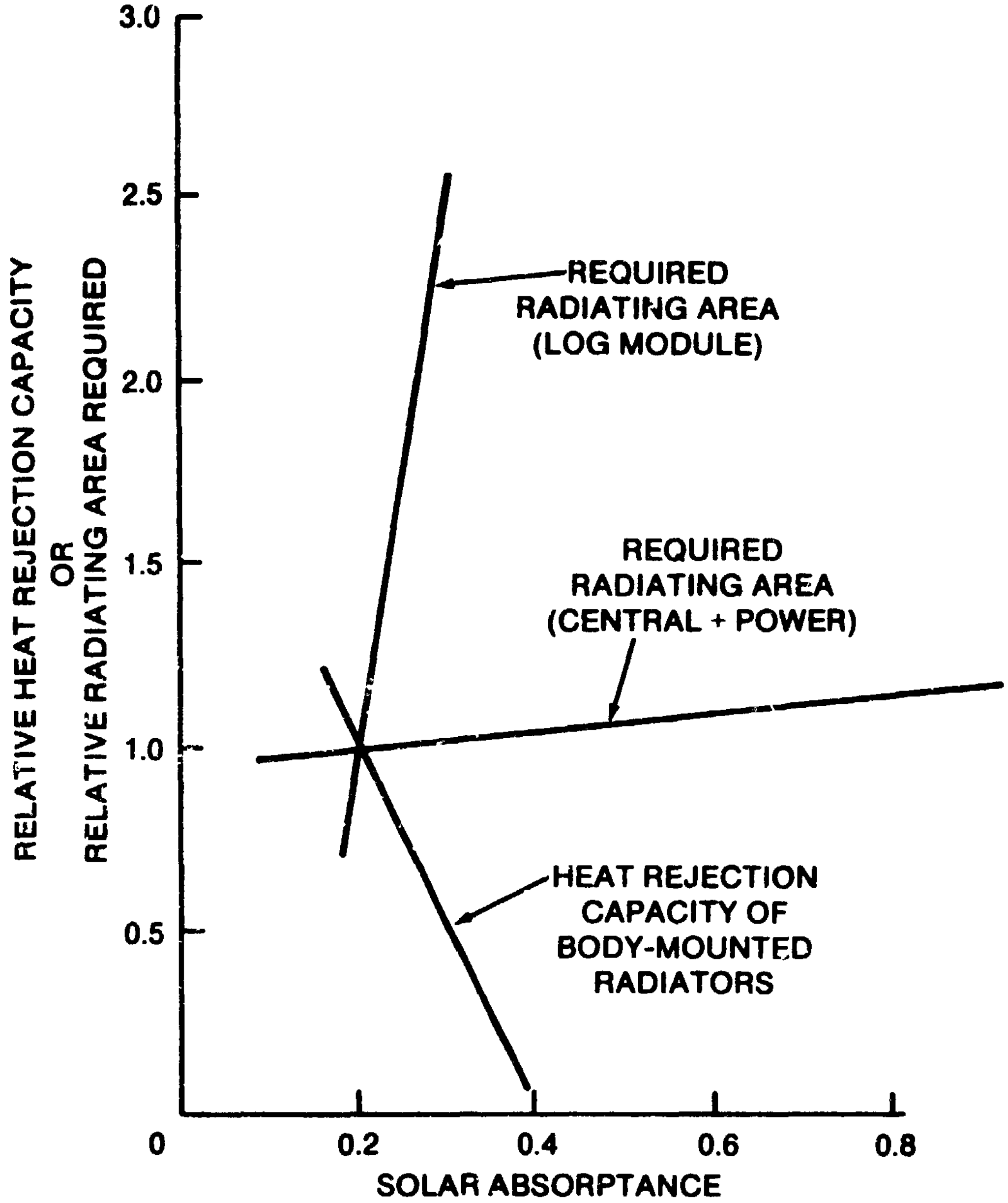


Figure 4.4.7-12 - EFFECTS OF SOLAR ABSORPTANCE ON RADIATING AREAS REQUIRED AND HEAT REJECTION CAPACITY

d. Propulsion (station only)

- station contingency tanks - 120 pounds
- propellant lines - 15 pounds
- engine modules - 15 pounds

e. External airlocks - 50 pounds each,

f. Satellite thermal protection shrouds - 795 pounds each.

4.4.7.4.1 Pressurized modules

Typical module heat leaks are presented for Habitation modules 1 and 2 in Table 4.4.7-10 for a range of surface thermal optical properties and insulation performance characteristics. The maximum heat leak out (1191 BTU/HR or 345 watts) of the modules occurs for an insulation effective emittance of 0.02 and an α/ϵ ratio of 0.3/0.8 while the maximum heat gain (-363 BTU/HR or 106 watts) occurs for an insulation effective emittance of 0.02 and an α/ϵ ratio of 0.8/0.8. These represent a small portion of the total internal heat load and do not effect the performance requirements of the Active Thermal Control Subsystem. Therefore, an external module surface treatment such as anodizing can be selected in lieu of a painted surface thereby eliminating the need for surface finish refurbishment. The module heat leaks were determined from a simple cylindrical math model with four external circumferential nodes (micrometeoriod shield) separated from the module wall by a gap and insulation on the module wall. The internal air node was held at a constant 70°F.

4.4.7.4.2 OMV servicing and storage

Since no requirement is identified to provide an active or passive thermal interface for the OMV, the power requirements for thermal control of the OMV are dependent on the final thermal design of the OMV. Therefore, the OMV becomes a thermal integration consideration for definition of the OMV thermal environment and resulting OMV thermal system power usage.

4.4.7.4.3 Satellite thermal considerations

For satellites not requiring a thermal protection shroud, a cylindrical model divided into four circumferential nodes and top and bottom nodes (ends), assumed adiabatic on the backside, was used to define maximum and minimum sink temperatures that would be experienced by a satellite for various thermal optical properties. In addition, maximum and minimum transient temperatures for a 0.05-inch aluminum wall were also determined. These are presented in Table 4.4.7-11. Figure 4.4.7-13(a) and (b) show the Orbital variation of sink tempeature for a satellite located in the -y service bay with an α/ϵ of .8/.8.

TABLE 4.4.7-10

HABITABLE MODULE PASSIVE HEAT BALANCE

(ORBITAL AVERAGE)

MODULE	INSULATION ϵ_{EFF}	HEAT GAIN/LOSS BTU/HR		
		α/ϵ = .3/.8	α/ϵ = .7/.86	α/ϵ = .8/.8
HM1	.01	607	44	-216
HM2	.01	710	20	-106
HM1	.02	1020	74	-363
HM2	.02	1191	257	-176

NOTE: HEAT GAIN DENOTED BY NEGATIVE VALUE.

TABLE 4.4.7-11 - UNPROTECTED SATELLITE TEMPERATURES

TEMPERATURES ON SATELLITE SURFACES - +Y SATELLITE BAY, BETA = 52°

	α/ϵ = .8/.8		α/ϵ = .69/86		α/ϵ = .3/.8		α/ϵ = .1/.8	
	SINK TEMPS	TRANSIENT .05" WALL	SINK TEMPS	TRANSIENT .05" WALL	SINK TEMPS	TRANSIENT .05" WALL	SINK TEMPS	TRANSIENT .05" WALL
TOP	200 -460	162 -118	165 -460	125 -125	60 -460	-10 -140	-70 -460	-140 -190
BOTTOM	145 5	102 25	125 5	85 25	70 5	45 15	30 5	20 10
CYLINDER	270 -140	255 -75	230 -140	220 -80	120 -140	95 -100	0 -140	-30 -120

TEMPERATURES ON SATELLITE SURFACES - -Y SATELLITE BAY, BETA = 52°

	α/ϵ = .8/.8		α/ϵ = .69/86		α/ϵ = .3/.8		α/ϵ = .1/.8	
	SINK TEMPS	TRANSIENT .05" WALL	SINK TEMPS	TRANSIENT .05" WALL	SINK TEMPS	TRANSIENT .05" WALL	SINK TEMPS	TRANSIENT .05" WALL
TOP	160 -460	125 -120	130 -460	85 -130	25 -460	-40 -150	-90 -460	-160 -200
BOTTOM	165 5	110 35	140 5	90 30	80 5	50 20	35 5	20 12
CYLINDER	285 -145	270 -85	245 -145	225 -95	130 -145	95 -110	10 -145	-35 -125

TEMPERATURES ON SATELLITE SURFACES OF REFUELING BAY - BETA = -52°

	α/ϵ = .8/.8		α/ϵ = .69/86		α/ϵ = .3/.8		α/ϵ = .1/.8	
	SINK TEMPS	TRANSIENT .05" WALL	SINK TEMPS	TRANSIENT .05" WALL	SINK TEMPS	TRANSIENT .05" WALL	SINK TEMPS	TRANSIENT .05" WALL
TOP	270 -250	225 -50	230 -250	180 -70	115 -250	40 -90	-20 -250	-95 -145
BOTTOM	75 -40	50 -10	60 -40	35 -10	15 -40	0 -25	-20 -40	-28 -35
CYLINDER	250 -145	245 -95	215 -145	210 -100	110 -145	100 -110	-5 -145	-20 -120

TEMPERATURES ON SATELLITE SURFACES OF REFUELING BAY - BETA = +52°

	α/ϵ = .8/.8		α/ϵ = .69/86		α/ϵ = .3/.8		α/ϵ = .1/.8	
	SINK TEMPS	TRANSIENT .05" WALL	SINK TEMPS	TRANSIENT .05" WALL	SINK TEMPS	TRANSIENT .05" WALL	SINK TEMPS	TRANSIENT .05" WALL
TOP	235 -220	75 -60	200 -220	40 -75	90 -220	-65 -115	-35 -220	-150 -165
BOTTOM	165 -45	70 0	135 -45	50 -10	65 -45	0 -25	0 -45	-30 -35
CYLINDER	280 -135	270 -100	245 -135	230 -100	130 -135	100 -115	10 -135	-25 -120

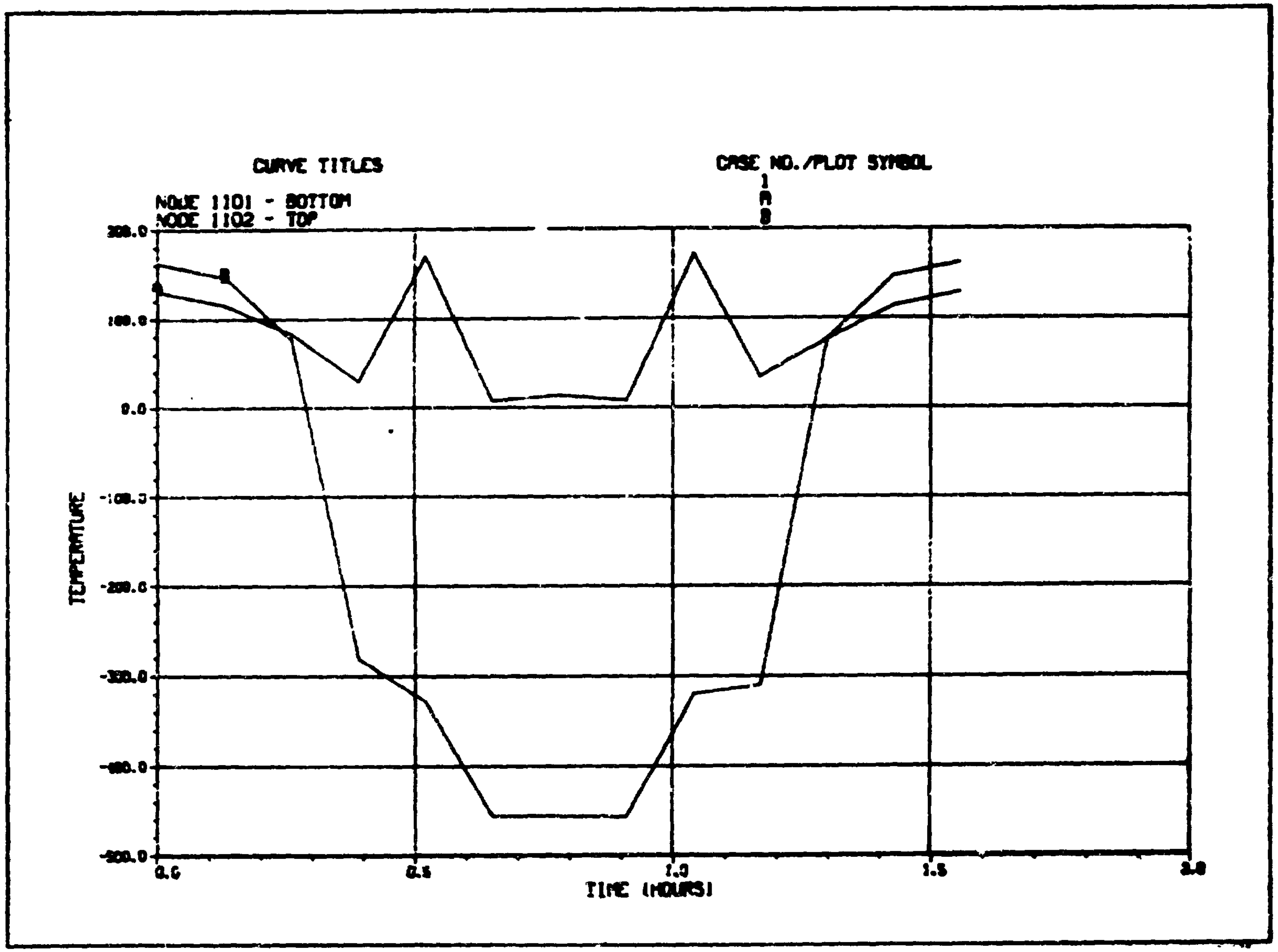


FIGURE 4.4.7-13a -Y SATELLITE SERVICE BAY SINK TEMPS. α/ϵ = .8/.8

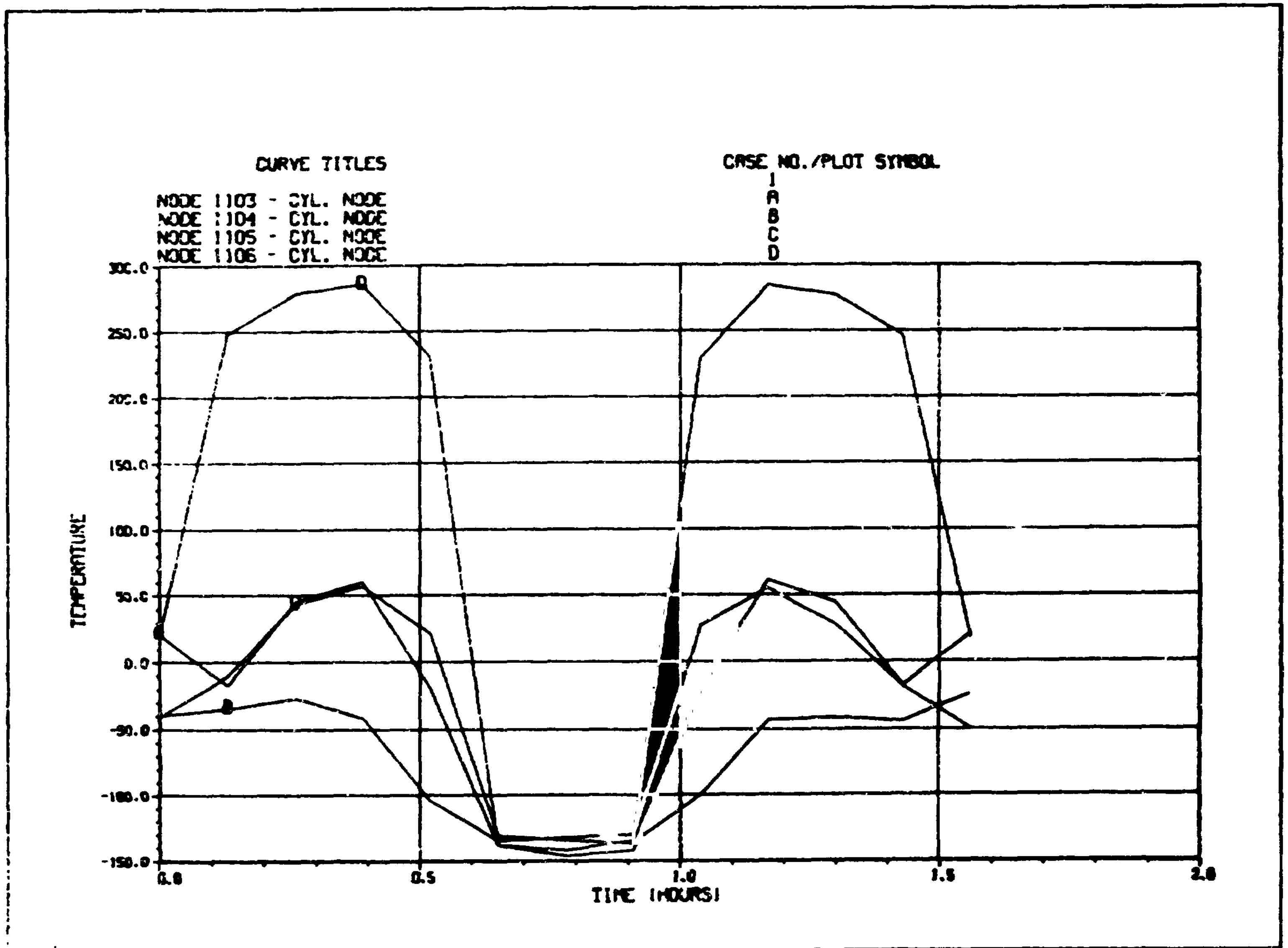


FIGURE 4.4.7-13b -Y SATELLITE SERVICE BAY SINK TEMPS. α/ϵ = .8/.8

For satellites requiring a thermal protection shroud, a 15 feet diameter by 60-feet long deployable multilayer insulation ($\epsilon_{EFF.}$ = .01) shroud was assumed with thermal closeouts at each end. The average Orbital internal temperatures for various external surface α/ϵ ratios were determined as a function of internal heat generation to represent payload and service operations. The +Y storage bay and -Y service bay temperatures are shown in Figure 4.4.7-14. Note that a small variation in temperature exists between the two locations. For an α/ϵ of 0.8/0.8 and no internal heat generation, an acceptable temperature range between 50 and 64°F can be maintained. However, internal heat generation would be limited to approximately 475 watts in order not to exceed 100°F. On the other hand, an α/ϵ of 0.1/0.8 would result in an approximate temperature of -83°F with no internal heat generation and would allow approximately 1.25 kW of internal heat generation before exceeding 100°F. These values are presented to point out the sensitivity of the internal shroud temperatures to α/ϵ and internal power generation and the necessity to undertake a detailed review of satellite requirements and servicing operations in order to define the thermal design envelope for the thermal protection shrouds.

Similar data for the refueling bay is presented in Figure 4.4.7-15 for its extreme environments of +52° beta angle and -52° beta angle. For high α/ϵ ratios, this location runs 10° to 20°F warmer than the storage and service bays. These results indicate that low α/ϵ thermal coatings will have to be selected which might require periodic refurbishment.

4.4.7.4.4 OTV Facility

The OTV facility, TDM 2570, was assumed to have multilayer insulation on the inside surface of the hangar walls with an effective emittance of 0.1. Figure 4.4.7-16 presents the Orbital average internal hangar temperatures for various α/ϵ ratios as a function of internal power generation. Because of the large surface areas of the hangar, an α/ϵ of 1.0 can be applied and still maintain an acceptable temperature range, 25°F to 100°F, for 0 to approximately 2 kW internal heat generation. Hard anodized aluminum (α = .69, ϵ = .86) can maintain temperatures above 0°F and could be allowed to degrade from contamination sources thereby deleting any refurbishment requirements. Active thermal interface requirements have not been identified. However, support equipment pose the potential for active thermal control. These requirements must be identified during the definition phase.

4.4.7.4.5 Propulsion System Thermal Control.

The OMV, OTV, satellite refueling and station contingency propellants are located and thermally protected as described in Section 4.3.5.3. Since the exact location of each type of propellant has not been identified, the hottest and coldest locations for a beta angle of 52° were considered for this study. These locations are the +Y and -Y corners of the upper keel extension. The analysis is based on a fully

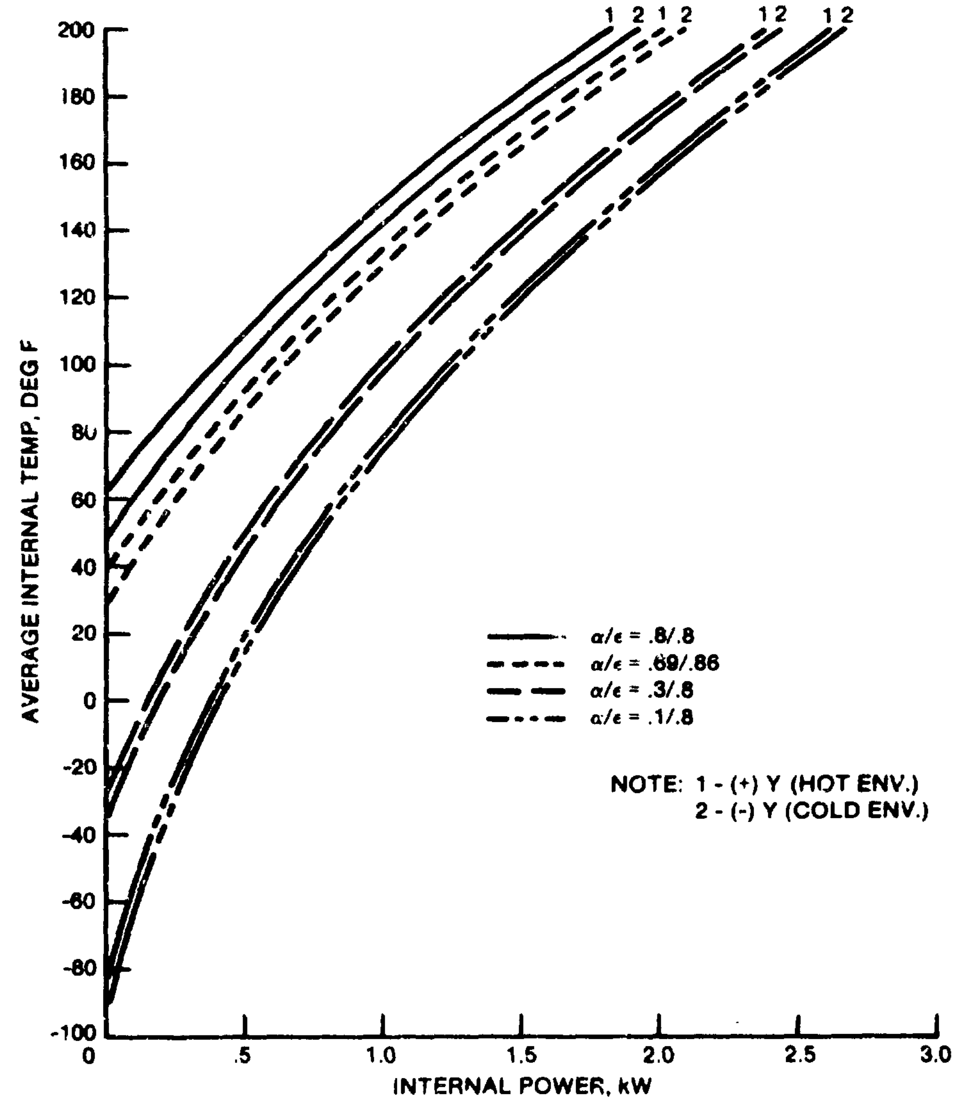


Figure 4.4.7-14 - PAYLOAD SERVICE BAY SHROUD INTERNAL TEMPERATURE

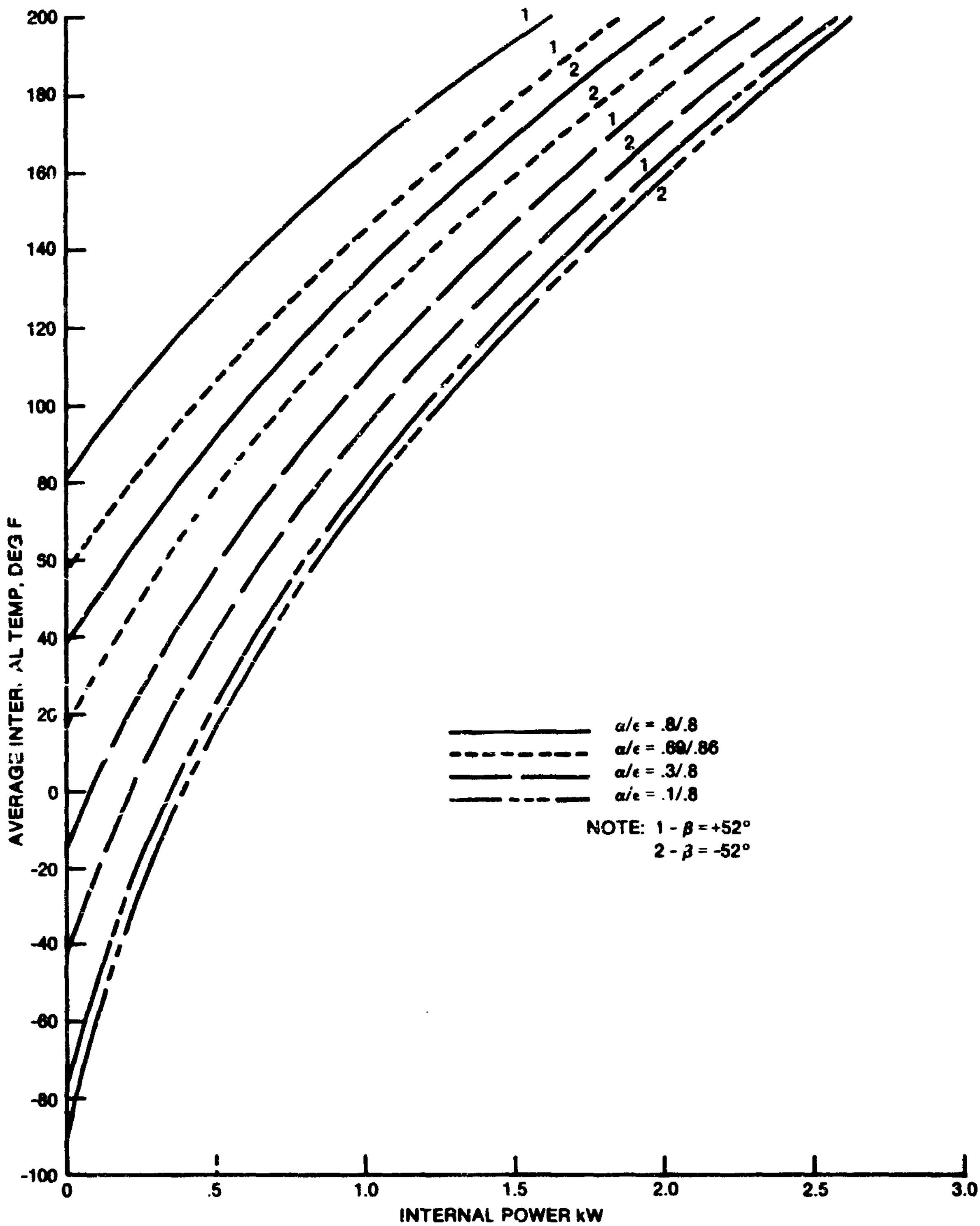


Figure 4.4.7-15 - REFUELING BAY INTERNAL SHROUD TEMPERATURE

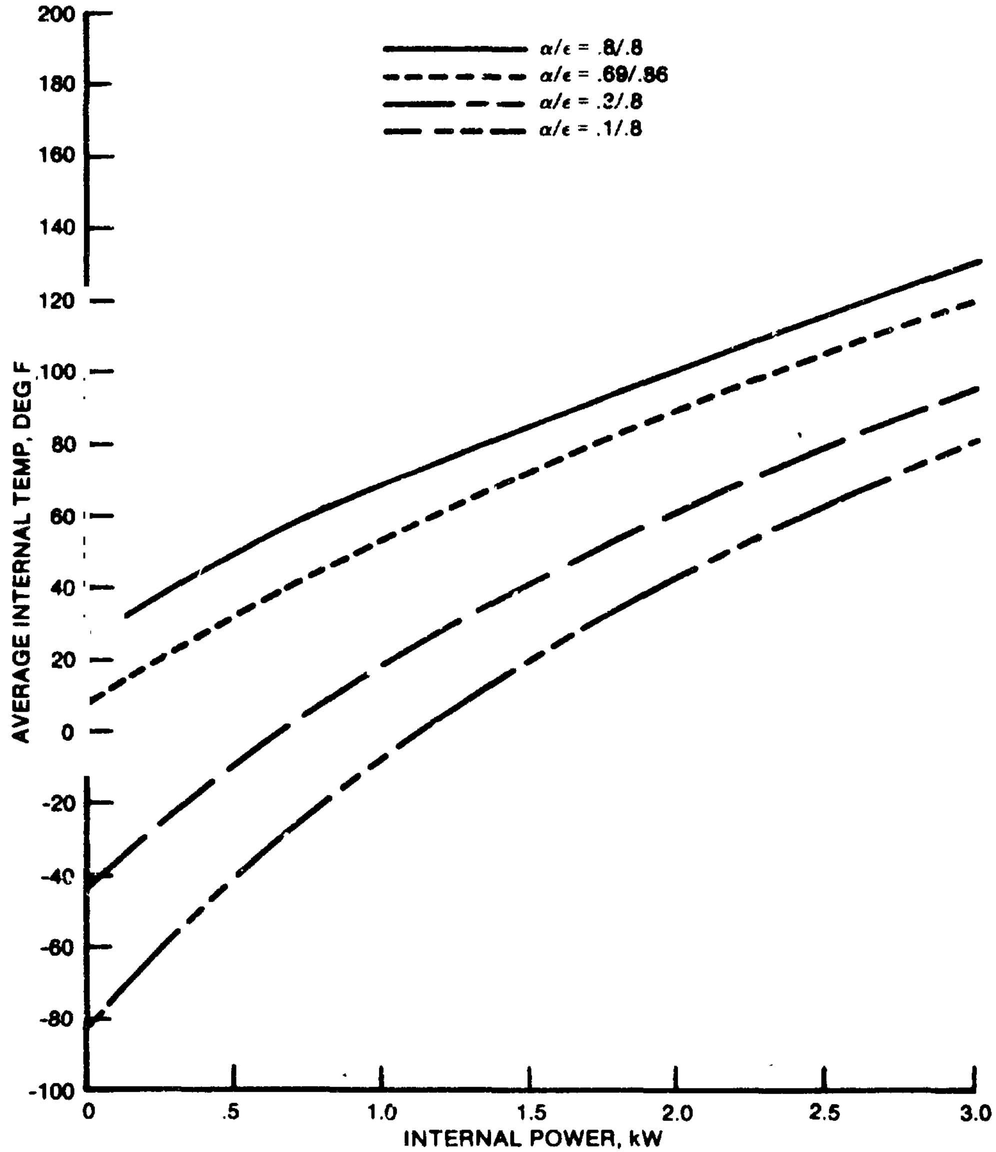


Figure 4.4.7-16 - OTV HANGER INTERNAL TEMPERATURE

insulated 9-foot cube assumed to be adiabatic at the wall adjacent to the next 9-foot cube. Average Orbital tank temperatures and makeup heat required for various α/ϵ ratios and an insulation effective emittance of 0.01 are as follows:

α/ϵ	+Y LOCATION		-Y LOCATION	
	TEMP (°F)	MAKE UP (WATTS)	TEMP (°F)	MAKEUP (WATTS)
.3/.8	-10	59	-50	87
.69/.86	38	18	-20	64
.8/.8	77	--	-4	54

The above analysis applies only to the bipropellants and monopropellants with an assumed operation temperature range of 40 to 100°F. As can be seen, this range can be accommodated with minimum makeup heat requirements by applying an α/ϵ of .8/.8 external surface treatment. The makeup heat is supplied by electric heaters controlled by a software thermostat logic thereby avoiding the use of failure-prone thermostats. The station contingency monopropellants are located within one truss cube. Therefore, the maximum makeup heat is 54 watts. Other bipropellant and monopropellants will require approximately the same amount of makeup heat per truss cube. However, cubes located in the center portion of the keel extension should require less heat since two sides of these cubes are adjacent to other propellant tanks. Design of these areas and cryogenic propellants are left open for the definition phase.

Approximately 1600 feet of propellant transfer and distribution lines are required in both sides of the lower keel, keel extension, and lower boom. The lower keel and both sides of the keel extension were analyzed for a beta angle of -52° to determine acceptable thermal optical properties and makeup heat to maintain the propellant between 40°F and 100°F (Table 4.4.7-12). Since the lines are small, 0.75-inch diameter, a multilayer insulation effective emittance of 0.02 was assumed to account for performance degradation which results from wrapping. The two sides of the keel extension represent the extremes of environment encountered. An α/ϵ of 0.8/0.8 results in minimum makeup heat requirements, 0.1 watts per foot of line for the cold condition, but slightly violates the upper limit of 100°F. Assuming that propulsion limit can be increased, a propellant line makeup heat of 160 watts is required. The maximum makeup heat of 0.27 watts per foot or a total of 432 watts occurs for an α/ϵ of 0.1/0.8 which is representative of aluminized Teflon film. The film could be allowed to degrade over a period of time (represented by the α/ϵ = .3/.8 in Table 4.4.7-12). However, the allowable degradation might be limited if the makeup heat required for the initial α/ϵ values result in over heating as a result of local degradation. The insulation

TABLE 4.4.7-12 - PROPELLANT LINE TEMPS. AND MAKEUP HEAT

LOCATION	α/ϵ = .1/.8		α/ϵ = .3/.8		α/ϵ = .8/.8	
	LINE TEMP °F	MAKEUP HEAT, WATTS/FT.	LINE TEMP °F	MAKEUP HEAT, WATTS/FT.	LINE TEMP °F	MAKEUP HEAT, WATTS/FT.
UPPER KEEL	55	0.255	55	0.175	63	-
+Y EXTENSION	55	0.235	55	0.12	105	-
-Y EXTENSION	55	0.255	55	0.22	55	0.10

NOTE: 55°F CONTROL TEMP. WHEN HEAT REQUIRED.

and external surface material must be flexible to accommodate packaging and deployment of the flexible lines. This will be the prime driver in selecting a material that best meets the overall design requirements. This is an open issue for the definition phase. In addition, every effort should be made to co-locate the central thermal bus transfer lines with the propellant lines to facilitate use of a common insulation wrap thereby reducing the makeup heat required from electrical heaters.

4.4.7.4.6 Communication system

Antenna thermal design is completely passive. Each electronics box will be of minimal thickness with the electronics conductively coupled to a radiator plate on the surface of the box for heat dissipation during standby and operational modes. Electrical heaters, controlled by software logic, can be applied for "off" or contingency conditions. Operation of these heaters is minimal and does not add to the power requirements of the operational and standby modes. The electronics box is insulated and a radiator-type coating, requiring periodic refurbishment, is applied to the exterior surfaces. Standby heaters and appropriate thermal surface treatments are applied to the antenna gyros and gimbals.

Since the actual thermal requirements of each antenna are unique to the particular design, only an estimate of electronics package radiator areas and "off" or contingency makeup heat requirements was determined. Radiator areas were determined by ratioing the Shuttle Orbiter Ku-band antenna power and radiator area. Contingency makeup or standby power was estimated by assuming that the Shuttle Orbiter Ku-band electronics box was maintained at 100°F at the radiator in a cold condition when the box was on. The standby power required to maintain a minimum temperature of 0°F was estimated assuming that the ratio of the standby power to the operational power is equivalent to the ratio of the fourth power of the two temperature levels. Radiator area and standby or makeup power estimates are presented in Table 4.4.7-13.

4.4.7.4.7 Solar array temperatures

Figure 4.4.7-17 shows the variation in solar array temperatures for a beta angle of 0° and various back face thermal optical properties. The data indicates that the temperature (138°F to 166°F) does not vary appreciably for the common range of materials (crossed hatched area) generally applied to solar arrays.

Since the solar cell efficiency is inversely proportional to the cell temperature, it is desirable to apply a back face material or surface treatment with the lowest α/ϵ ratio that is practical, in order to minimize the array temperature. The sensitivity of the solar cell efficiency to temperature is depicted in Figure 4.4.7-18. For the common back face materials, the efficiency can range for 0.112 to 0.121. Obviously, degradation of the thermal optical properties can result in large impacts to the array power producing capability. Therefore, materials or surface treatments must be selected to minimize

TABLE 4.4.7-13 - COMMUNICATIONS AND TRACKING

EQUIPMENT	QUANTITY	TOTAL PWR, WATTS	TOTAL RADIATOR AREA, FT^2	TOTAL MAKEUP HEAT, WATTS
ANTENNA:				
TDRS S/Ku-band Ant.	2	200	-	91
Co-Orbit Med. Gain Ant.	2	160	-	73
Co-orbit High Gain Ant.	2	150	-	68
Low Gain Multi-beam Prox. Ops Ant.	2	-	-	-*
Near Hemisph. cov. Prox. Ops Ant.	4	-	-	-*
Orbiter Low Gain (10db) S-band Ant.	2	15	-	7
Orbiter Low Gain (0-3db) S-band Ant.	2	-	-	-*
GPS Low Gain Ant.	2	-	-	-*
Med. Gain Rendezvous Radar Ant.	2	200	-	91
MRMS Low Gain S-band Ant.	1	-	-	-*
GPS low gain	2	-	-	-*
S-BAND low gain	2	-	-	-*
Radar Xpndr low gain	2	-	-	-*
MRMS S-band low gain	1	-	-	-*
S-band low gain MRMS-TV	1	-	-	-*
R.F. SYSTEMS: (1)				
TDRS S/Ku-band Xmit/Rec.	4	400	6	182
RF Assy for Co-orbit Med. Gain Ant.	4	120	1.8**	55
RF Assy for Co-orbit High Gain Ant.	4	150	2.3**	68
RF Assy for Low Gain Prox. Ops. Ant.	4	140	2.0**	64
RF Assy for Near Hemi. Cov. Prox. Ops. Ant.	8	40	-**	18
RF Assy for Orbiter S-band Antennas	8	80	-**	36
S-BAND xpndr (TDRS)	2	82	1.3	37
S-BAND xpndr (Orbiter)	2	82	1.3	37
S-BAND pa/preamp (T/O)	2	800	12	364
XCVR (MRMS)	1	20	-**	9
RCVR (MRMS)	1	10	-**	4
XMTR (ORBITER)	1	13	-**	6
TRACKING:				
RF Assy for Rendezvous Radar Antenna	2	450	7	204
Laser Docking System	1	20	-**	9
Structural Stab. Monit.	4	400	6	182
GPS rec./proc.	2	180	2.7	82
Radar XPNDR	2	20	-**	9

* MAKEUP HEAT DEPENDS ON MATERIAL LIMITS, ANTENNA CONFIGURATION AND LOCATION AND LOCAL STRUCTURE.
** POTENTIAL FOR DESIGN WITHOUT DISCREET RADIATOR AREA.

degradation from ultraviolet radiation and atomic oxygen and they must be protected from contamination sources to minimize or negate the sizeable task of refurbishment.

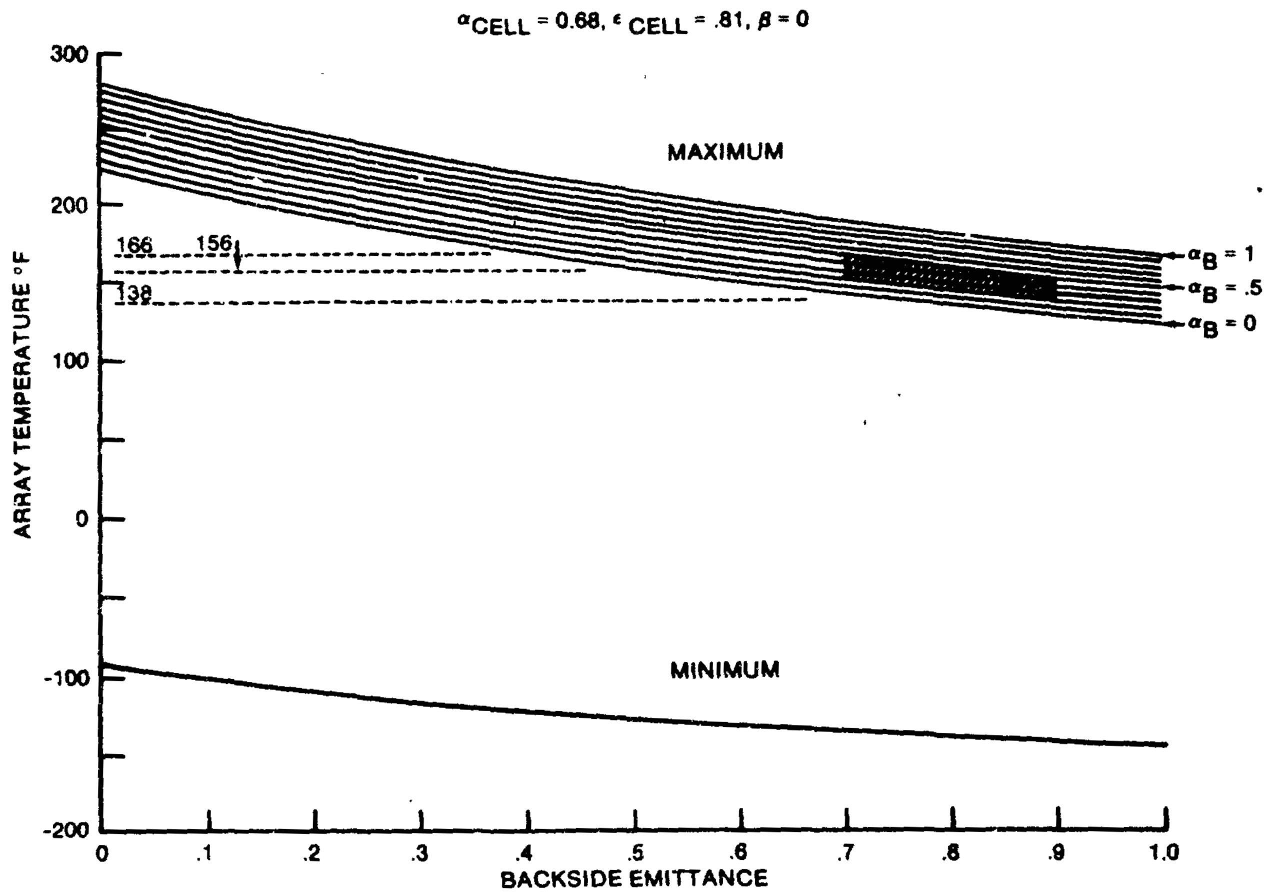


Figure 4.4.7-17 - SOLAR ARRAY TEMPERATURES

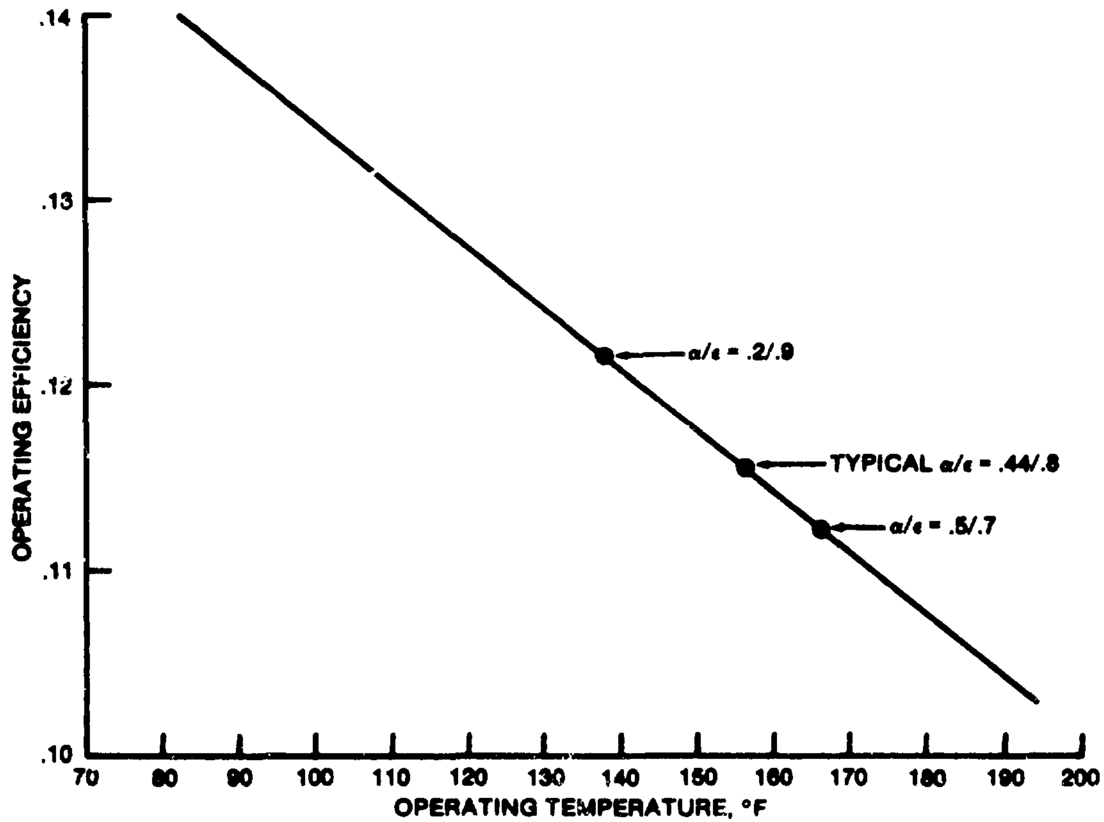


Figure 4.4.7-18 - SOLAR ARRAY EFFICIENCY

4.4.8 Structures and Mechanisms

4.4.8.1 Introduction

This section of the report deals with various aspects of the main structural truss, the pressurized modules, and the mechanisms of the operational IOC Space Station. It is not meant to be a mathematical paper on structural design or analysis, or a presentation of novel structures or mechanical devices. It is, however, meant only to transmit concepts, thoughts, and ideas as well as preliminary analysis. Each subsection will be a condensation of the more extensive White Papers (published separately). The topics to be covered are as follows:

a. Space Station truss structures and construction considerations

b. Preliminary structural design and analysis of a Shuttle launched Space Station manned habitable module

c. Module and payload truss attachments

d. Module - module and berthing interface

e. Solar panel deployment mechanisms

f. Conceptual rotary joint design for Space Station application

g. Structural deployment verification

h. Conceptual design of a Mobile Remote Manipulator System

4.4.8.2 Space Station Truss Structures and Construction Considerations

4.4.8.2.1 Introduction

Over the past several years, a considerable effort has been expended on structural design for large space systems. Most of the effort has been generic in nature and directed towards large reflector applications. The concept of the Space Station represents a new set of structural requirements that were not considered for the previous application. However, the experience gained on these past structural studies provides a wealth of knowledge and insight that can guide the selection process toward an optimum or near optimum Space Station structural foundation.

Prior studies for the construction of large space systems indicate that there are three major approaches for packaging, transporting, and construction of those structures in orbit. These approaches involve structural members that are classified as deployable-single fold, erectable, and deployable-double fold. An overview of these approaches and the identified structural requirements for the Space Station will be presented herein. A more detailed study is presented in the White Paper entitled "Space Station Truss Structures and Construction Consideration."

For a comparison study of the three approaches, overall dimensions of a reference Space Station were used. These dimensions are shown in Figure 4.4.8.2-1 which depicts the deployable-single fold structural truss arrangement. The overall size is representative of a gravity gradient stabilized station with a solar array that will deliver 75 kW continuous power.

4.4.8.2.2 Truss requirements

The following list contains some of the requirements that have been identified for the Space Station truss system:

a. Provide a structural foundation for construction of the Space Station

b. Provide a surface area for the attachment of payloads and utility lines

c. Provide a structural stiffness that will minimize Space Station control problems

d. Provide a road bed for a track system that will allow the use of a Mobile Remote Manipulator System (MRMS) to aid in Space Station construction and transportation of payloads

e. Provide a redundant structure that will offer alternate load paths if a member of the truss is damaged so that strength integrity of the Space Station is not impaired

f. Provide structural repair capability without the loss of structural integrity

4.4.8.2.3 Deployable-single fold beams

Due to the need for redundancy in the structural arrangement, a box beam with four longerons and diagonals between the longerons was considered. In determining the size of the beam, it was considered desirable to make the beam cross-section as large as possible to provide bending stiffness, but confined to the dimensions of the Shuttle cargo bay. In addition, a large cross-section will provide a wide track for the MRMS and the payloads it must carry, and also provide a large open area inside the packaged beam for incorporation of utility lines.

The Shuttle cargo bay can accept a 14-foot diameter payload to transport to orbit. This diameter establishes a maximum upper limit on a square cross-section of 10 feet. For the current study, a maximum cross-section size of 9 feet has been established which will allow additional space in the cargo bay for external beam fitting attachments for installing solar arrays, radiators, and etc. The beam chosen for this study is an orthogonal tetrahedral design, having longerons that fold inward and diagonals that telescope to effect a compact packaging configuration. In addition, a provision was made to incorporate quick-attach joints at the side of each bay to permit addition of additional beam structures. This configuration is shown in Figure 4.4.8.2-2.

The packaging characteristics of this beam are such that each bay compacts to a dimension equal to 2-longeron diameters. For this particular example, a 216-foot long beam can be deployed from an 8-foot long package using 2-inch diameter longerons.

Deployment of the beam is a sequential and controlled operation in which one bay unfolds at a time so that instabilities normal to the deployment direction may be avoided. A deployment canister can accomplish this task but will have to be discarded after the deployment because it would interfere with the MRMS movement. Energy schemes to provide deployment can include precompressed springs in the member joints of each bay, electric motors with lead screw or chain drive mechanisms, or use of the MRMS.

Space Station assembly analysis indicates that it will take a minimum of two Shuttle flights to provide the structural foundation required for the reference station configuration condidered in this study. The basic limitations are the size of the beam required, the folding characteristics of the beam, and the size of the Shuttle cargo bay.

4.4.8.2.4 Erectable beam

The erectable beam concept has the advantage over the deployable beam just discussed that the maximum Shuttle cargo bay diameter constraint is eliminated. Erectable beam members do not rely on folding techniques and can be packaged as individual pieces which will allow a greater packaging density. The only limitations placed on this type of construction is the individual member length which should be compatible with the equipment storage and function of the Space Station. In this section, two different sizes of erectable beams were considered. The first has a 9-foot cross-section identical to the deployable beam of the previous section. The second has a 15-foot cross-section which will not only increase the beam structural stiffness but will also be compatible with the attachments for a 14-foot maximum diameter Shuttle payload. In both cases, the bays should be square so that the MRMS can move in orthogonal directions as required.

The principal difference between the erectable and deployable beam of the previous section is that the beam has to be constructed while in orbit. This task will require a large amount of EVA for the construction personnel. In addition, the utility lines that could be incorporated on the deployable beam will have to be added to the erectable beam. However, the erectable beam has the advantage of requiring only one Shuttle flight to provide the structural foundation for the reference Space Station configuration.

4.4.8.2.5 Deployable-double fold truss

It is generally understood that a double fold structure is a more efficient means of packaging truss structures for transport in the Shuttle cargo bay than the single fold concept. Because of the high packaging efficiency of the double fold concept, it would be considered feasible to deploy a larger than required truss in orbit to minimize subsequent add-on structure for future Space Station growth.

The double fold truss considered for this study was made of a tetrahedral configuration. Figure 4.4.8.2-3 shows the reference Space Station using the tetrahedral truss as the foundation structure. Figure 4.4.8.2-4 shows a single tetrahedral element of this truss configuration. The members of this element were considered to be 10-feet long for this study. It should be noted that the inherent characteristics of the tetrahedral element requires that all members have the same length thereby reducing manufacturing costs and spares requirements. In addition, this truss system is highly redundant which will allow alternate structural load paths and preserve strength integrity in case some are the members are damaged.

The upper and lower surface members hinge at their midpoints for compact folding. As a result, the packaged truss becomes a function of the truss node fitting dimensions (or member diameter) and the length of the individual truss member. It was shown in this study that this packaging concept will allow the entire Space Station truss system to be installed in the Shuttle cargo bay leaving room for additional packages such as solar arrays, radiators, etc. This implies that the complete Space Station structure can be delivered to orbit in one flight.

Two methods of deploying the truss were considered. In both methods the deployment energy was considered to be prestressed springs in the hinged joints of the top and bottom surface members. The first method is a free deployment in space that required the package to be removed from the cargo bay using the RMS. While supporting the package with the RMS, activate the controlled deployment mechanism that allows the truss to expand at a controlled rate. The second method utilizes a set of rails that can be attached to the Shuttle cargo bay longerons. The packaged truss is then inserted in the rail system by the RMS. The rail system can then serve as the controlled deployment device; and also stabilize the package, freeing the RMS for other duties.

4.4.8.2.6 Comparison study

The three approaches studied are believed to generally represent the major techniques for constructing the Space Station foundation.

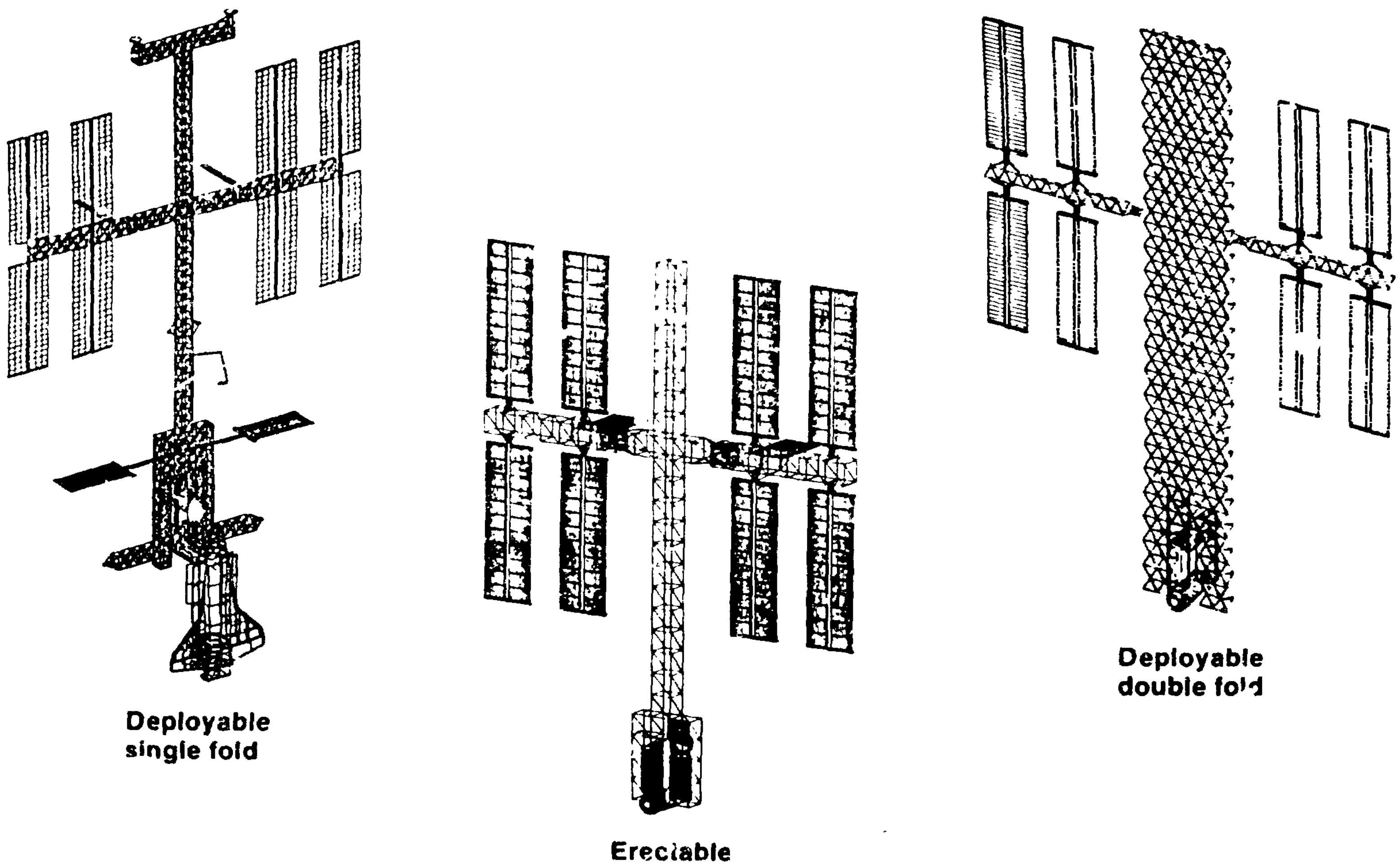


Figure 4.4.8.2-1 - REFERENCE STATION STRUCTURAL OPTIONS

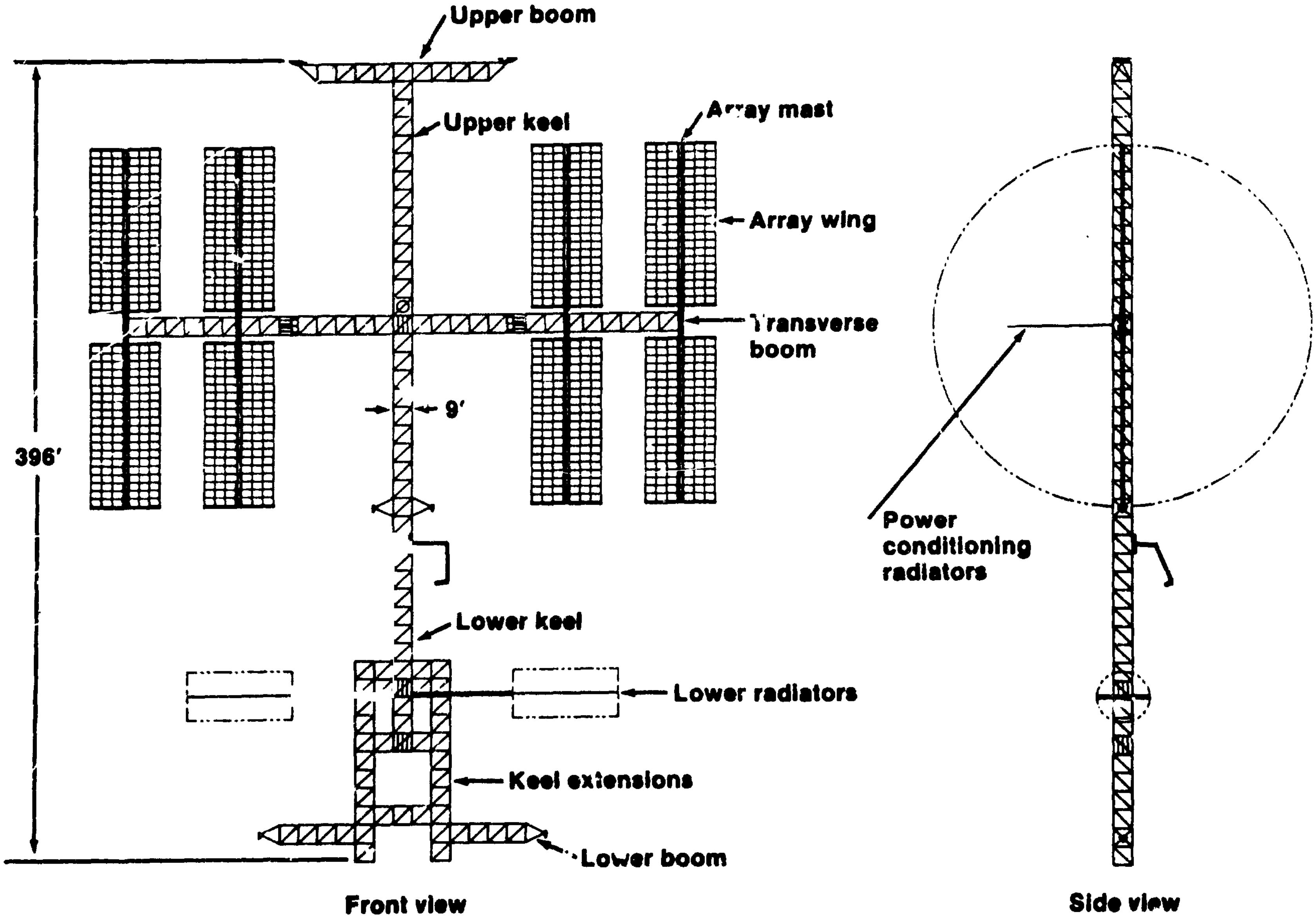


Figure 4.4.8.2-2 - OVERALL SPACE STATION DIMENSIONS AND NOMENCLATURE

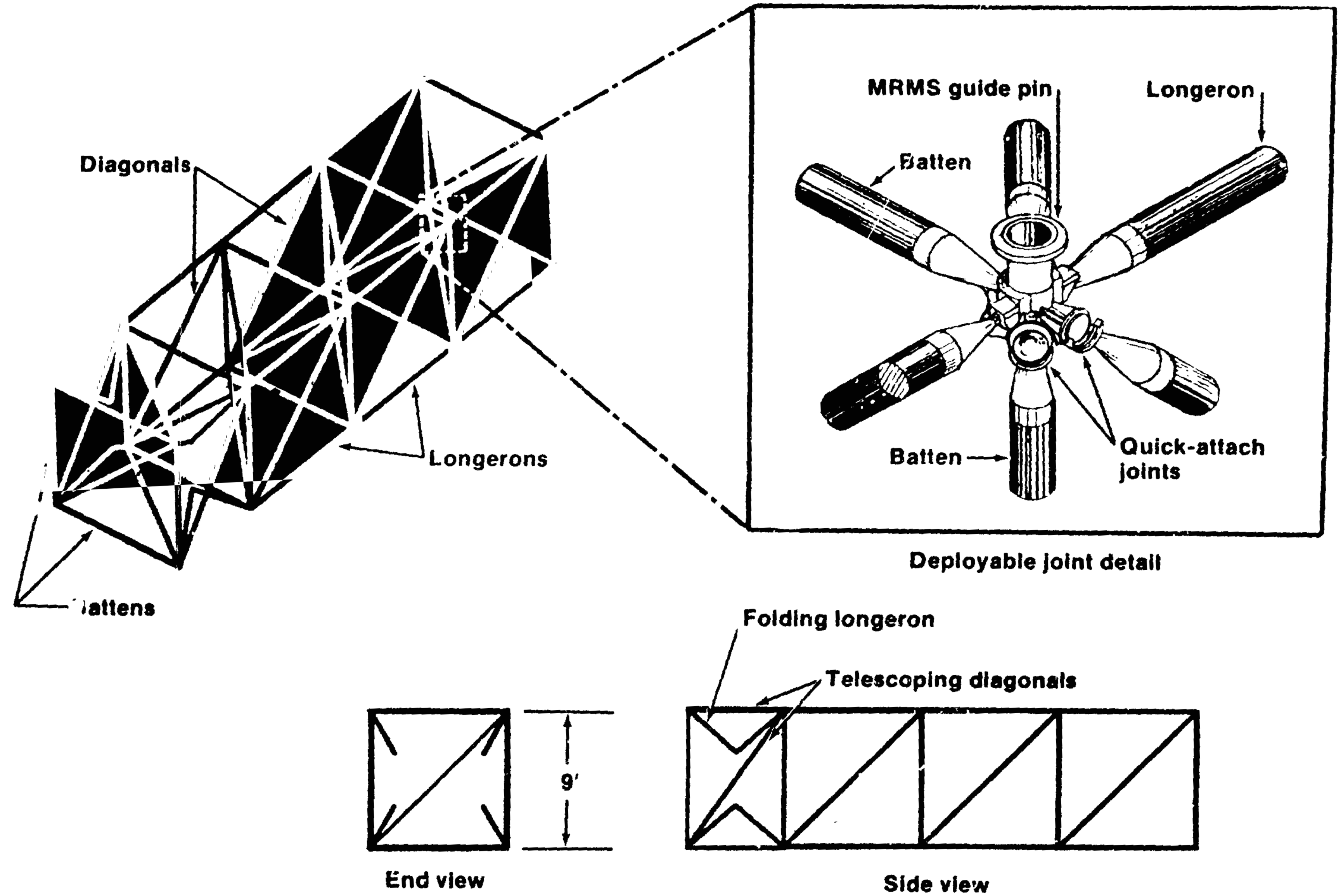


Figure 4.4.8.2-3 - SCHEMATIC OF DEPLOYABLE BEAM SHOWING ONE BAY BEING DEPLOYED AND DETAIL OF JOINT

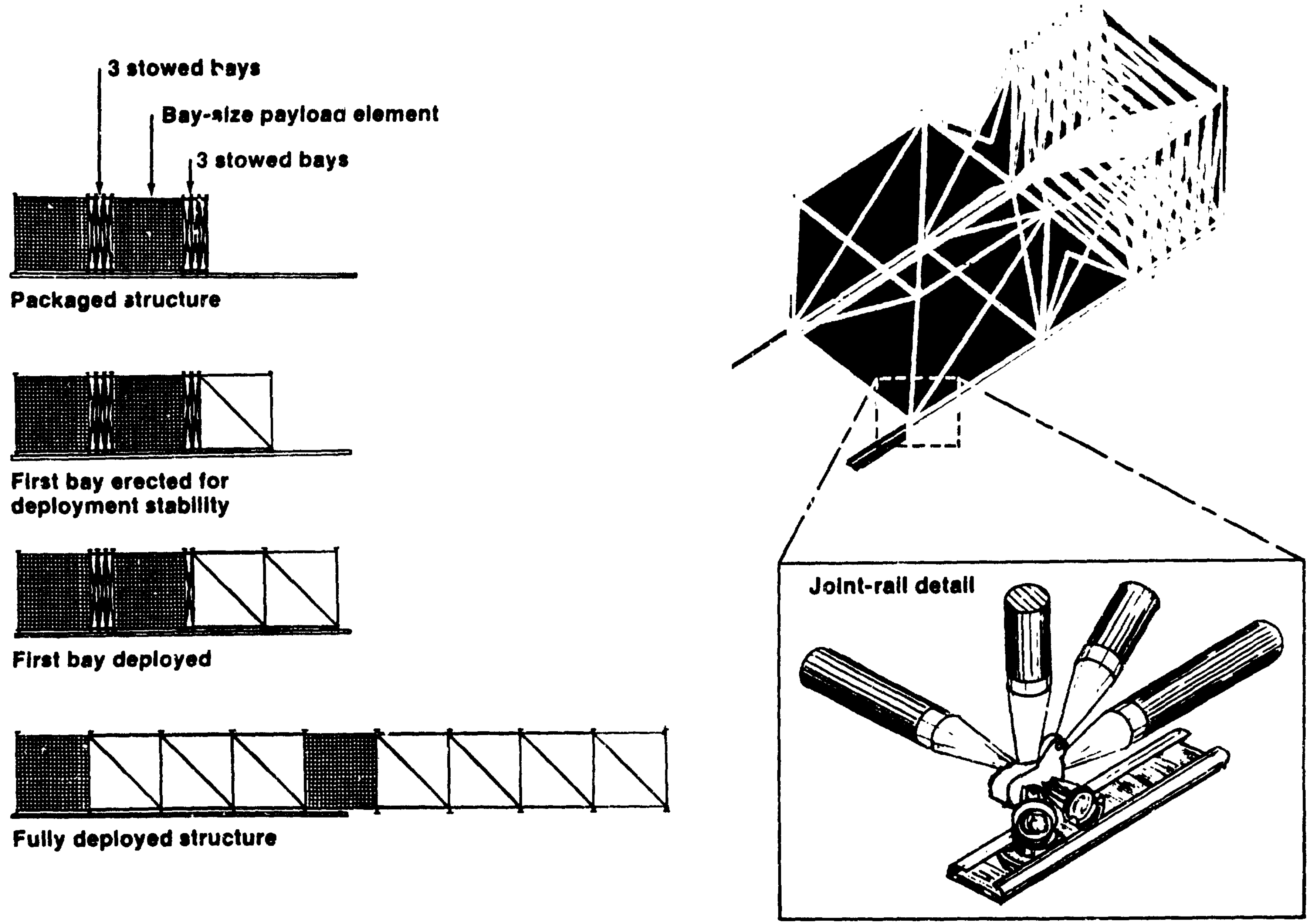


Figure 4.4.8.2-4 - TRUSS DEPLOYMENT SEQUENCE AND RAIL DETAIL

Although the three different constructions are not exactly comparable, they are similar enough that general observations can be made. The parameters selected for characterizing each construction approach are part count, structural weight, and structural stiffness because of their relationship to the structural design, fabrication costs, and performance of the Space Station. In an attempt to achieve commonality between constructions, one- and three-bay wide versions of the 9-foot deployable beam, a 15-foot erectable beam, and a six-bay wide deployable tetrahedral truss were examined.

The results showed that the 15-foot bay erectable beam resulted in a structure which has about half as many parts and weighs half as much as the 9-foot deployable beam, yet possesses 3 times the stiffness. The deployable tetrahedral truss is seen to have 50 percent or more parts than the 9-foot deployable beam but has only a slightly higher weight. In addition, the tetrahedral truss is over twice as stiff as the 9-foot deployable beam but only slightly stiffer than the three-bay version. Some appropriate papers can be found in Reference 9 through 33.

4.4.8.3 Preliminary Structural Design and Analysis of a Shuttle-Launched Space Station Manned Habitable Module

4.4.8.3.1 Introduction

Previous studies have shown that generally the structural sizing of a Shuttle-launched Space Station module is governed by the loads experienced for the Shuttle flight environments rather than the activity that the module is to perform while attached to the Space Station. As a result, it could be considered feasible to design a single structural module shell for Shuttle flight environments that would serve a general-purpose function for all manned and unmanned activities planned for the Space Station. This concept would permit an assembly line production of module shells requiring only one set of tooling. Interior equipment installation at the end of the assembly line is all that would be required to transform the structural shell into a special identity as a habitatation, service, or laboratory module.

At the time this Space Station module study was initiated, the preliminary definition of the module was outlined as a cylindrical pressure vessel 34 feet long and 15 feet in diameter. The general structural configuration chosen for this study intended to optimize the structural components so that they could perform a multifunction purpose.

The following sections summarize the structural design requirements that were defined at the time the study began and the preliminary design work that was done to establish the module integrity to meet those requirements. Detailed information can be found in the White Paper of the same title as this section.

4.4.8.3.2 Module design requirements

The following list specifies general requirements that were identified for this study.

a. The module should be made of materials that will provide a service life of 10 years or more without intermediate refurbishment

b. Module gross weight and overall dimensions will not violate shuttle and payload bay constraints

c. Provide strength and life integrity to sustain a manned shirt sleeve-environment of 14.7 psia

d. Provide adequate internal attachment structure for module function configuration

e. Provide meteoroid/debris protection at a 95 percent probability of not having a penetration for 10 years

f. Provide docking/berthing capability to other modules and to the Space Shuttle

g. Provide dual ingress and egress capability

h. Provide windows for observation

i. Impose structural ultimate factors of safety for structural design and analysis

1. Factor of safety of 2.0 for pressure loading

2. Factor of safety 1.5 for mechanical and thermal loading.

4.4.8.3.3 Proposed module construction

Implied in the requirement for a module service life of 10 years or better is the selection of a structural material that will not suffer erosion from the space environment as well as maintain sufficient strength to endure multiple repetitions of pressure and thermal cycling. Also implied in the requirement to maintain an internal pressure of 14.7 psia is an acceptable leak rate in the pressure vessel. It is expected that it would be difficult to maintain an acceptable leak rate for a module made of conventional riveted and mechanically fastened skin-stringer construction for 10-years tenure; therefore, to satisfy both the life and material requirements, it is proposed that the module be constructed of all welded integrally machined skin-stringer panels using 2219-T851 aluminum plate. This material has good strength, a high fracture toughness, a good resistance to stress corrosion, good weldability, and good machineability. It would be expected that structural certification could be minimized by devising a simple proof pressure test on each module to verify the 10-year strength and life integrity requirement.

Shuttle cargo dimensions dictate a maximum 15-foot diameter module. However, there are restrictions to this dimension due to deflections of the payload in the cargo bay. These restrictions are defined in the Space Shuttle System Payload Accommodations Specification and usually restrict the maximum payload diameter to a 3-inch clearance between the payload and the Shuttle structure. The maximum diameter chosen for the module design was 14.5 feet. Also, maximum gross weight for the Shuttle payload is 65,000 pounds for ascent but restricted to 32,000 pounds for landing. However, the 32,000-pound landing weight may be improved by applying suitable restrictions to reentry maneuvers and landing sink speed. A representative 43,000-pound gross weight payload was chosen for this design study. Figure 4.4.8.3-1 shows the overall module geometry used in this study.

Internal attachment structure for equipment and experiments will be provided by the ring frames inside the module. With the exception of the berthing segment, the module cylinder will have 5 equal bays of 55.6 inches that contain heavy ring frames for the attachment of equipment, experiments, floors, and other hardware required for the specific module configuration. The ring frames should be designed for in-plane loading only since out-of-plane loads will cause twisting of the frames and produce a heavy weight design. Any out of plane loads should be sheared to the skin-stringer panels of the module.

Meteoroid/debris protection of the module must be designed to provided a 95 percent probability of not having a pressure skin penetrated in 10 years. The concept chosen for this design was the double wall bumper which has a .045 inch thick aluminum bumper located 2.0 inches away from the module pressure skin. The 2-inch gap is filled with multilayer insulation to provide thermal control and may also improve the meteoroid/debris protection slightly. Since there is a requirement to prevent crushing of the multilayer insulation, there will be nonthermal conducting stand-offs between the two aluminum sheets that will keep the bumper from deflecting into the insulation.

Docking/berthing capability will be designed into the module conical end domes as well as a cylindrical segment in the side of the module. These various berthing positions will allow greater flexibility in generating several Space Station configurations from a common module design. It is expected that the module will be securely anchored to the Space Station by means other than the docking/berthing ports. Therefore, these ports should be designed for light berthing loads only and not be required to support the Space Station integrity. The current port design includes a berthing mechanism support structure that has a 70-inch clear diameter. It has been estimated that the actual hatch diameter is 50 inches with the remainder of the area being dedicated to disconnects for plumbing, electrical, air conditioning, etc. The cylindrical segment in the side of the module contains four berthing ports located 90° apart for the reference Space Station configuration. This would allow the module to be rigidly attached to the truss structure by the same trunnions used to support the module in the Shuttle cargo bay and permit a variety of choices for berthing with other modules or the Shuttle.

The cylindrical portion of the module has been designed with four 16-inch diameter windows that can be used for observation. A double pane concept allows a redundant pressure pane to save module pressure integrity if the primary pressure pane should break.

4.4.8.3.4 Module-to-Shuttle attach points

Design loads for the module are strongly dependent on the module center of gravity location within the Shuttle cargo bay. For this study, the module will use the Shuttle attach fittings for deployable payloads and an active keel fitting. Available fitting locations for the module (as a payload) are given in the Space Shuttle System Payload Accommodations Interface Control document. A five-point attachment system (four Shuttle longeron attachments and one-keel attachment) will be statically indeterminate. However, it is possible to make certain simplifying assumptions that will render the problem statically determinate.

It will be advantageous to locate the forward and aft longeron attach points as far apart as possible to reduce the module reaction loads entering the Shuttle longeron. For this study, the trunnions will be located at the forward and aft ring frames of the cylindrical module section approximately 360 inches apart. The location of the module within the cargo bay was determined by the load capability of the Orbiter longerons. The large liftoff and landing loads that were determined in the next section will require that splitter beams be used at both the forward and aft trunnion locations to divide the module reactions between adjacent Shuttle frames. Figure 4.4.8.3-2 shows the module position in the cargo bay that maximizes the Orbiter load carrying capability. A forward splitter beam is needed to distribute the high z-direction loads while the aft splitter beam is needed for x-direction reactions.

4.4.8.3.5 Module design loads

Shuttle ascent and descent load factors were obtained for the module which were based on prior Shuttle payload experience. It was noted that the angular accelerations appeared to be rather high particularly for the landing condition. However, these accelerations could not be confirmed until a math model of the module could be constructed and processed by the computer program. This work is still in progress. Using the load factors as given, ultimate module-to-Shuttle attach reaction loads were generated assuming the module to be a simple beam and making certain assumptions to render the problem statically determinant. The resulting reaction loads were used to size the attach ring frames at stations X_0 862.5., X_0 1055.5, and X_0 1222.5. Ultimate shear and bending moment diagrams about the module centerline were also generated for the critical load conditions and used to size the integrally-stiffened module skin.

An ultimate factor of safety of 1.4 was used for the module strength integrity and a factor of 2.0 was used for the module pressure vessel. For manned habitation, the module was considered to contain a "shirt sleeve" environment of 14.7 psia. Pressure conditioning for astronauts will be reserved for airlock modules and are not a part of this study. The maximum thermal environment for the module was considered to occur while the module is in orbit and was determined to varying between +150°F and -20°F; however, this structural loading was considered negligible due to the anticipation of applying thermal insulation and coatings over the module surface.

On-Orbit module loads considered combined internal pressure and berthing loads. Berthing loads considered a worse case condition which was based on the assumption that the Shuttle would be berthed to the Space Station and cantilevered from one module. Using a Shuttle (plus payload) mass of 269,000 pounds and a berthing acceleration of 3.8 X 10^{-4} g's at the Shuttle and a 40-foot moment arm, the bending and axial load on the module berthing mechanism is 49,100 inch-pounds limit. The axial load is 102 pounds limit.

4.4.8.3.6 Meteoroid/debris protection

The meteoroid/debris protection criterion for the manned habitable module has been established as 95 percent probability of not having a penetration of the module pressure skin for a 10-year life in the space environment. Using this criterion and the meteoroid/debris flux, the estimated bumper thickness should be .045 inches for a 2-inch spacing between the bumper and the module skin. There is some question that the .070-inch thick module skin will be thick enough to resist the impact loading caused by the limited 2-inch spacing between skins. However, this criterion will be used until the flux can be better defined.

Due to the large unsupported areas of the bumper, nonconductive standoffs must be used between the bumper and the module skin to maintain the 2-inch spacing and not crush the multilayer insulation. Crushing the multilayer insulation will cause degrading of its thermal characteristics.

4.4.8.3.7 Primary ring frame design

The module reaction loads are transmitted to the Orbiter structure by the trunnion and keel fittings which are attached to the primary ring frames of the module. Three primary rings are proposed for the module: the forward ring at station Xo 862.5, the midmodule keel ring at Xo 1055.5, and the aft ring at station Xo 1222.5. Since the trunnions are offset from the ring section neutral axis, the reaction loads will result in a combination of thrust and shear loads as well as bending moments that must be used to size the rings.

A lightweight ring design would certainly adhere to a varying load distribution around the periphery of the ring and dictate thin webs and beam flanges in areas where the loads are low. However, for this exercise it was assumed that the ring would maintain a constant cross-section that would allow additional stiffness for attachment of equipment inside the module. These ring sections will have to be checked as soon as the mass of the installed equipment becomes known. Preliminary sizing of the ring frames assumed that they would appear as a simple I-beam section.

4.4.8.3.8 Intermediate ring frame design

The intermediate ring frames within the module will be required to have a multipurpose function of providing a solid support for the module skin panels and providing a solid support structure for the equipment mounted inside the module. Experience has shown that the ring frame stiffness required to insure skin panel stability is much less than the stiffness required to support the in-plane forces generated by the mass items mounted the frame. Therefore, the sizing of the intermediate ring frames should concentrate on the forces generated by the equipment mass items. However, since these forces have not yet been defined, the preliminary sizing will focus on the forces applied by the window and berthing ports and the results assessed when the equipment loads and their locations become available.

Preliminary sizing of the ring frames will consider that all intermediate rings are identical and that the sectional properties will remain constant around the circumference of the ring. This will add material where it is not needed for the present design load conditions, but this additional material may be needed for the equipment loads to be determined later.

Maximum intermediate ring loads were determined from the combined internal pressure and berthing loads that would occur at one of the cylindrical segment berthing ports. These loads are transmitted to the ring frame by means of the berthing support structure which spans the primary forward ring and the first intermediate ring at station X_0 944.5.

4.4.8.3.9 Cylinder design

In this study, the primary load carrying structure is considered to be an integrally stiffened skin with ring frames. The skin will resist the pressure loads for habitability and also be stiffened by stringers equally space around the inner circumference to resist body bending and axial loads from the Shuttle flight environment.

Placement of the stringers on the inner surface of the cylinder skin will provide a clear area between the meteroid shield and the skin for the insulation installation. Ring frames designed in the preceeding sections will be used internally to stiffen the thin-walled shell and also provide material for the attachment of equipment and bulkheads for the module function. Additional structure will be required at the intersection of the module and aft trunnions to distribute the Nx shear loads and prevent large torsional moments from being induced in the aft primary ring frame.

4.4.8.3.10 Conical end dome design

A truncated conical shell has been proposed as the end closure configuration for the pressurized module. The primary load carrying structure for the domes is considered to be an integrally stiffened skin with ring frames. The skin will be required to resist the internal pressure loads of the module and the integrally machined stringers will be required to resist the bending and axial loads from the Shuttle flight environment and Space Station berthing environment.

Loads analysis indicated that the critical design condition for the cone occurs for combined internal pressure and berthing. Structural arrangement for the truncated cone resulted in placement of 20 integrally machined stiffeners equally spaced around the inner surface of the cone skin to react the axial and bending loads. Skin thickness to resist internal pressure loads was variable depending on the location of welds due to the reduced allowables for an as-welded material. However, a minimum thickness of .07 inches was retained in the pure membrane areas due to the meteoroid/debris requirement.

4.4.8.3.11 Window segment design

The aft cylindrical segment of the module will contain four 16-inch diameter double pane windows located 90° degrees apart around the perimeter of the cylinder. The structural requirements for these windows are to contain the module pressure and provide meteoroid/debris protection for the crew and equipment. It is also desirable to be able to replace these windows while in orbit if they should become damaged.

The outer window pane will be required to act as the redundant pane and also the meteoroid/debris shield. This pane will be considered to be unloaded as long as the inner pane is in place. However, it will be required to carry all the pressure load if the inner pane becomes broken. The air gap between the two panes should be vented to the space environment in order to keep moisture from forming on the glass surfaces. The venting should be controlled such that it will produce only a slow leak if the inner pressure pane should break and can be sealed off completely while the broken pane is being replaced.

The design of the pressure pane should contain a preloading mechanism that will keep the glass in a compression state of stress through its thickness. This compressed state of stress will keep any flaws in the glass from growing when pressure loads are applied. Feasibility of such a preloading mechanism has been studied and approved by structural engineers at JSC. It is also recommended that both panes of window be made identical to reduce manufacturing costs and replacement complexity.

The structural arrangement of the module cylinder segment containing the four windows will be identical to the other stiffened cylinder segments presented in Section 4.4.8.3.9 with the exception of the additional stiffening required for the window cutouts. Module bending and axial loads will not be reacted by the windows.

4.4.8.3.12 Berthing segment design

Four berthing ports are located in the cylindrical body of the module to provide flexibility in Space Station construction and configuration. It was found that due to the large cut-outs required to fit the berthing mechanisms, the berthing segment should be located at the forward end of the module to preclude interference with the module-to-Shuttle attach structure. In addition, the module design loads are much lower in the forward end for the flight environment. Therefore, the critical design loads for the berthing segment emerged as the on-orbit conditions of combined internal pressure and berthing loads.

4.4.8.3.13 Module weight summary

Primary structure weight for the module containing the four berthing ports is summarized below. It should be noted that the primary load carrying structure weight is approximately 25 percent of the total assumed gross module weight which is in keeping with earlier studies performed on modules of different configurations. It should also be noted that the weights calculated for the meteoroid/debris bumper system and the thermal protection system are not charged to the primary structure system.

Cylindrical Sidewall Assembly:

(1)	Skin/Stringers	2561 lbs
(2)	Primary Ring Frames	1508 lbs
(3)	Intermediate Ring Frames	957 lbs
(4)	Berthing Segment	3489 lbs
(5)	Window Segment	1101 lbs
(6)	Trunnion Longerons	667 lbs
	TOTAL	10,283 lbs

Primary Structure

Cylindrical Sidewall Assy	10283 lbs
Conical End Domes	630 lbs
Total Primary Structure	10913 lbs

Subsystems

Meteoroid/Debris Shield	946 lbs
Thermal Insulation	180 lbs
Remaining Subsystems	28044 lbs
Total Subsystems	29170 lbs

Secondary Structure

10% of Subsystems	2917 lbs
Total Secondary Structure	2917 lbs
Total Launch Gross Weight	43000 lbs

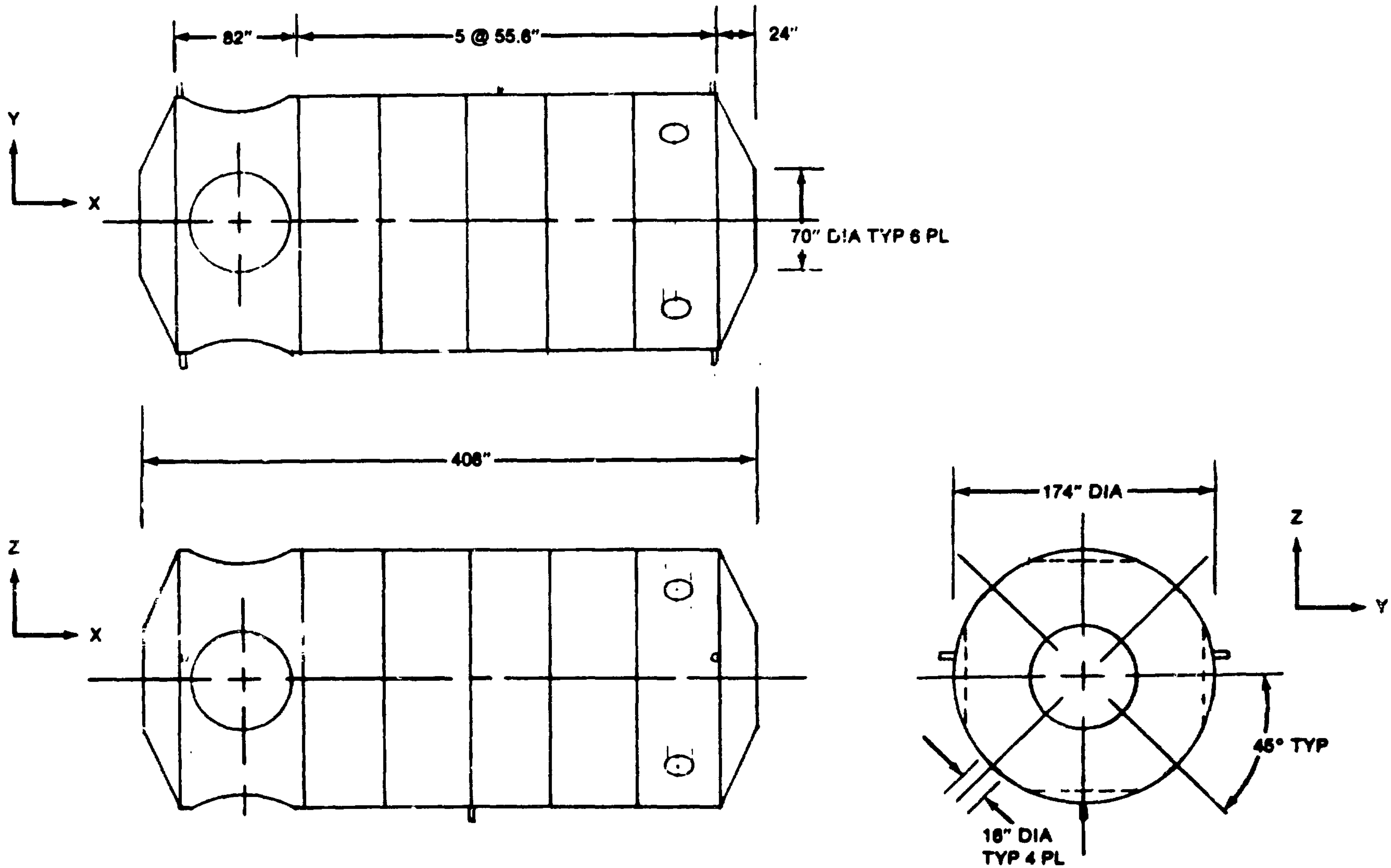


Figure 4.4.8.3-1 - OVERALL MODULE GEOMETRY

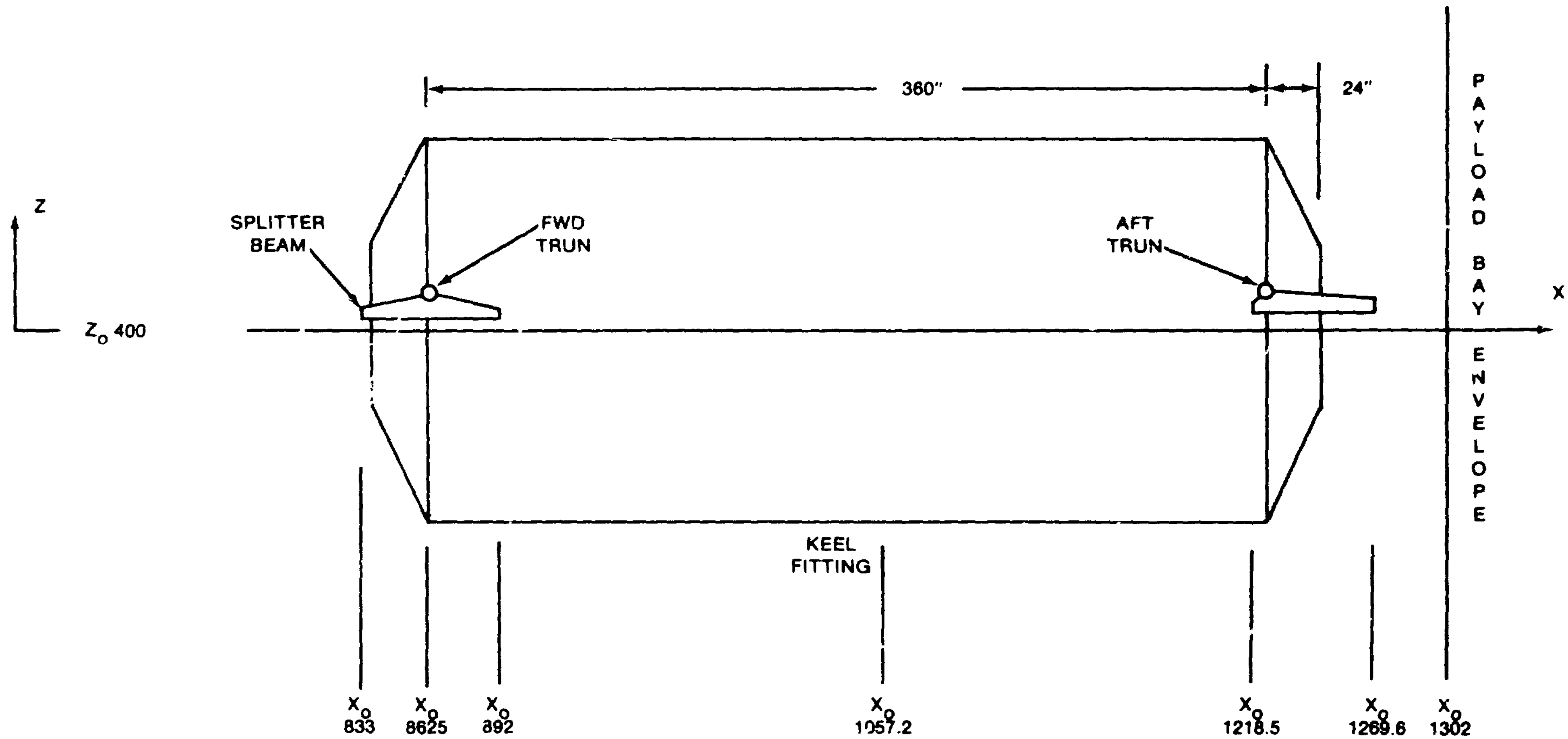


Figure 4.4.8.3-2 - MODULE-TO-SHUTTLE ATTACH POINT LOCATIONS

4.4.8.4 Module and Payload Attachment

4.4.8.4.1 Introduction

Successful construction and operation of the manned orbiting Space Station is dependent on the methods that are used for the attachment of modules, payloads, and other equipment to the truss structure. In this section, the issues associated with attaching these components to the truss structure are discussed.

4.4.8.4.2 Basic concepts

Two scenarios for attaching modules and payloads to the truss have been studied. Both methods use the Orbiter cargo bay trunnion hardware which will allow this hardware to serve a multipurpose function. The purpose of these scenarios was to investigate methods for attachment that can be easily manipulated by personnel in the space environment and also provide adequate structural load paths.

Figure 4.4.8.4-1 shows the first scenario in which a module (or payload) has been located in its approximate position on a planar truss by the Orbiter RMS. Next, structural tripods and bipods are installed on the trunnions and attached to appropriate truss nodes. Finally, the tripod and bipod strut assemblies are adjusted by a lock/unlock telescoping feature designed into the struts. After ma final adjustments, the struts are locked in place and the RMS is removed.

Figure 4.4.8.4-2 shows the second scenario which uses four tripod strut assemblies which are mounted to appropriate truss node points to form a pseudo-Orbiter cargo bay. The module (or payload) is then removed from the Orbiter cargo bay by the RMS and placed into the trunnion fittings of the tripod fixtures. Final adjustment is made for any misalignment, the trunnions are locked in place, and the RMS is removed.

Both these scenarios indicate that attachment struts making up the bipod and tripod assemblies should be designed for adjustment in length to allow for unplanned distortions due to manufacturing tolerances and/or thermal expansion and contraction. Both also advocate using the same trunnion attachments provided for payload storage in the Orbiter cargo bay. Finally, both require attachment to truss nodes rather than attachment to a truss member. The following sections will elaborate on these strut/attachment requirements.

4.4.8.4.3 Strut adjustment

Studies have shown that there will be unplanned misalignments between the payload and the truss node geometry which will require adjustment by Space Station personnel prior to locking the payload to the Space Station structure. Basically, small misalignments will be caused by manufacturing tolerances in the hardware and thermal distortions due to thermal gradients and dissimilar materials. However, commonality of hardware will require that the attaching struts have large length adjustments to accommodate various payload sizes and shapes.

Figure 4.4.3.4-3 shows an adjustable strut configuration developed for this study. The telescoping mechanism in the middle of the strut, allows for large adjustments in length. The turnbuckle at the top of the strut, allows for fine adjustments. Details of the telescoping mechanism are shown in Figures 4.4.3.4-4 and 4.4.3.4-5. The screw-jack turnbuckle mechanism is shown in Figure 4.4.8.4-6. Also shown in this figure is a proposed trunnion fitting that will cradle and lock the payload trunnion pin to the strut mounting structure and a lanyard attachment that will allow quick release at the trunnion fitting/strut interface.

4.4.8.4.4 Trunnion attachments

Payloads transported in the Orbiter cargo bay contain attachment points that have been designed for the Orbiter flight environment. Previous studies have shown that the flight environments are more critical for the design of these fittings than the Space Station environment. Therefore, these fittings are more than adequate for Space Station attachments.

4.4.8.4.5 Truss node attachments

Due to the inherent characteristics of truss design, all loading should be introduced at the truss nodes where there is a collection of structural members to share the load. In addition, configuration of trusses studied in this report dictate that all node joints are essentially pin-ended which will react concentrated loading but have no resistance to bending moments. For this reason, all truss loading should be introduced at the node points in such a manner that there will be no induced bending.

Figure 4.4.8.4-7 shows a design concept for a tetratruss node fitting. The payload attachment strut is pinned to a swivel in the center of the fitting that eliminates bending in two planes. In addition, the fitting was designed so that the strut load path would pass through the intersection of the load paths of the truss members. This arrangement will eliminate induced bending in the remaining plane.

4.4.8.4.6 Attachment to a square truss

Proposed concepts for mounting modules to 9-foot and 15-foot square trusses are shown in Figures 4.4.8.4-8 through 4.4.8.4-11. The strut arrangements shown in these figures are somewhat different from the arrangements depicted in the scenarios due to the variation in truss configurations. However, these attachments still form a statically stable system.

4.4.8.4.7 Concluding remarks

The truss structure being proposed for the Space Station presents a unique design feature of presenting a "peg board" to which various payloads and modules can be attached. This section presented preliminary design concepts that take advantage of this inherent truss capability. However, these concepts do not represent a closed set of attachment designs but rather present a starting point for future designs. Versatility, rigidity, stability, structural load paths, commonality of hardware, and ease of crew installation are of prime importance in design of this system.

4.4.8.5 Module to Module Interface, Docking/Berthing

In the interest of commonality of hardware and maximum operational flexibility, the module-to-module interfaces and docking/berthing interfaces are geometrically identical. That is, any interface will mate and latch to any other interface, including completion of a pressurized transfer interface. The performance requirements for the module-to-module interface are based on (1) its use as an assembly interface in an essentially static alignment, adjustment, interconnect operation and (2) its use as a passive port for docking or berthing of the Orbiter, which will carry the active docking/berthing mechanism. Alternately, the station interfaces identified as berthing ports could also include active docking/berthing mechanisms but that is not the case for the reference configuration. The Orbiter must be given the capability for total active control of the rendezvous, approach and docking/berthing task routinely because only the Orbiter has significant translation capability and for contingency cases because the Station systems may not be working or control of the Station systems may be inaccessible to the crew.

The module-to-module interfaces will be identical throughout the Station in the reference configuration. The interfaces feature a 6-inch extension capability to facilitate the module assembly and module change out processes. In the IOC module pattern, side-mounted interfaces are mated only to end-mounted interfaces, so a clear option exists to place all the extension capability in the end ports. This would significantly increase the internal useable volume in the vicinity of the side ports.

4.4.8.5.1 Module-to-module interface

The pressurized modules will be joined to each other using either the Orbiter RMS or the Station Manipulator in a highly controlled, low velocity berthing process. There should be no need for active shock attenuation within the interface mechanisms. Currently developed assembly scenarios involve mounting of the modules-to-truss structure prior to connection of the pressurized interfaces, so the latter operation will be basically static. Mounting of the modules-to-truss structure will be a berthing process quite similar to berthing a payload in the Orbiter payload bay. The design drivers in terms of loads

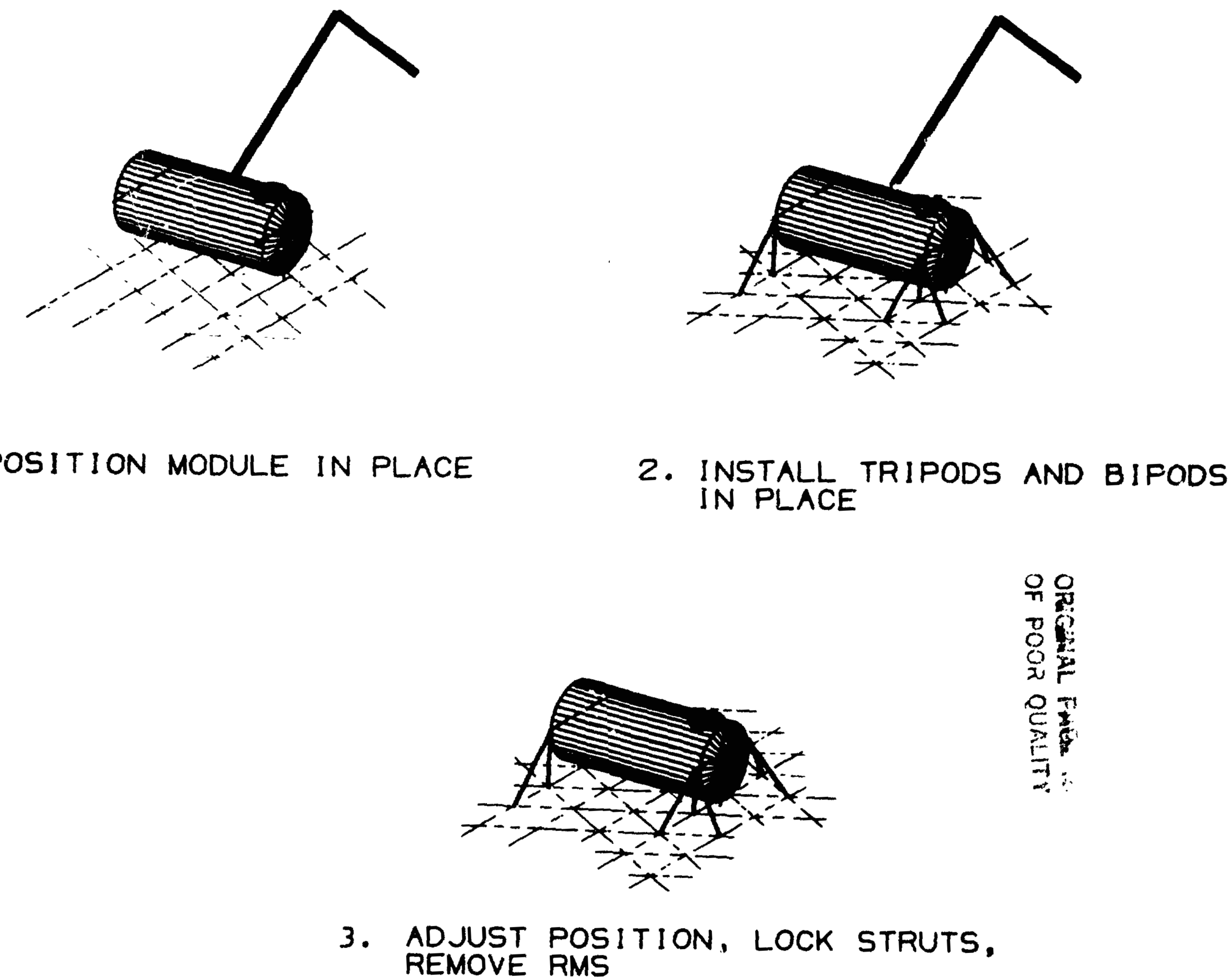


Figure 4.4.8.4 -1 - SEQUENCE OF INSTALLING MODULE ON TRUSS-METHOD 1

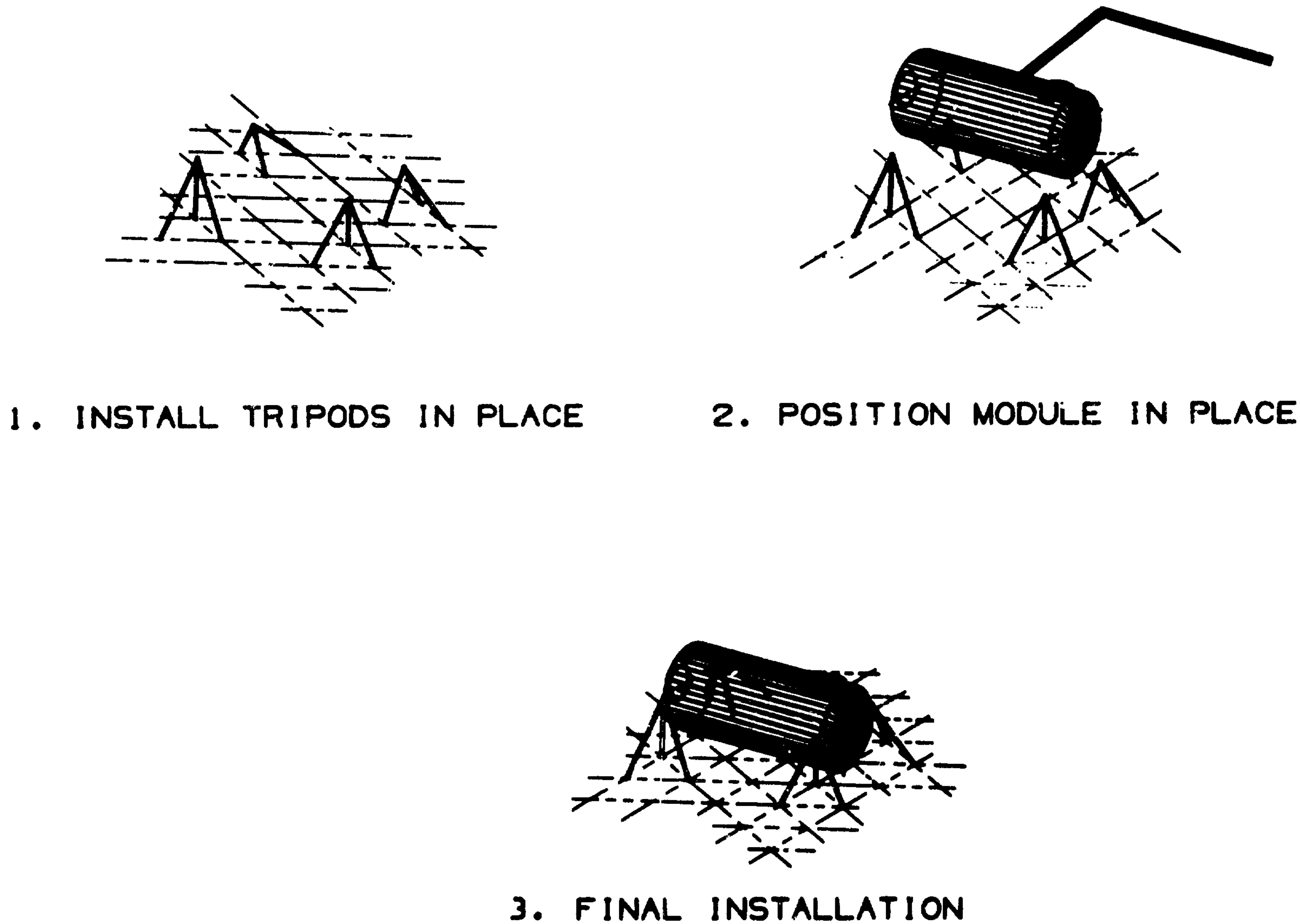


Figure 4.4.8.4 -2 - SEQUENCE OF INSTALLING MODULE IN PLACE-METHOD 2

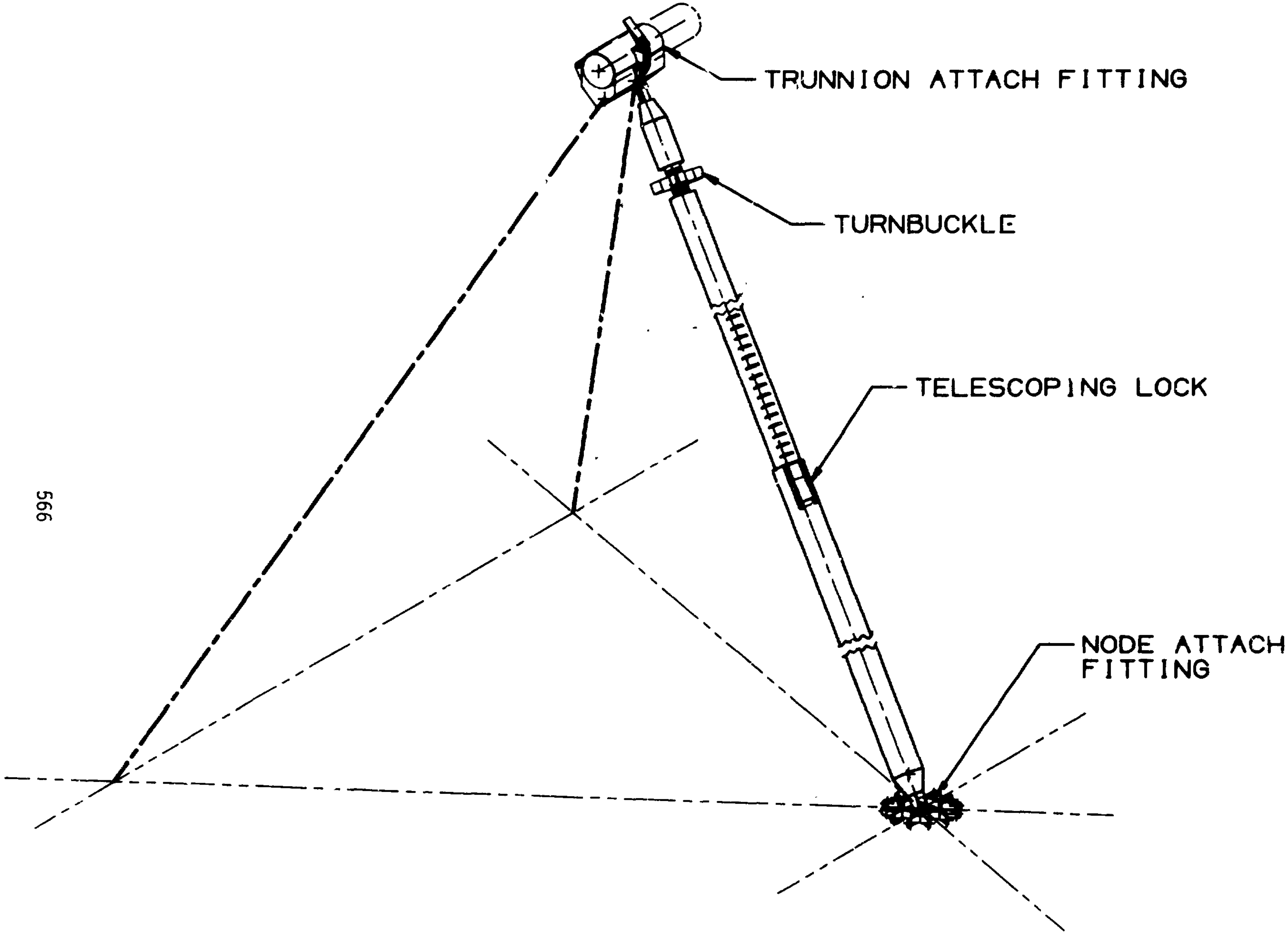


Figure 4.4.8.4 -3 - ADJUSTABLE STRUT ASSEMBLY

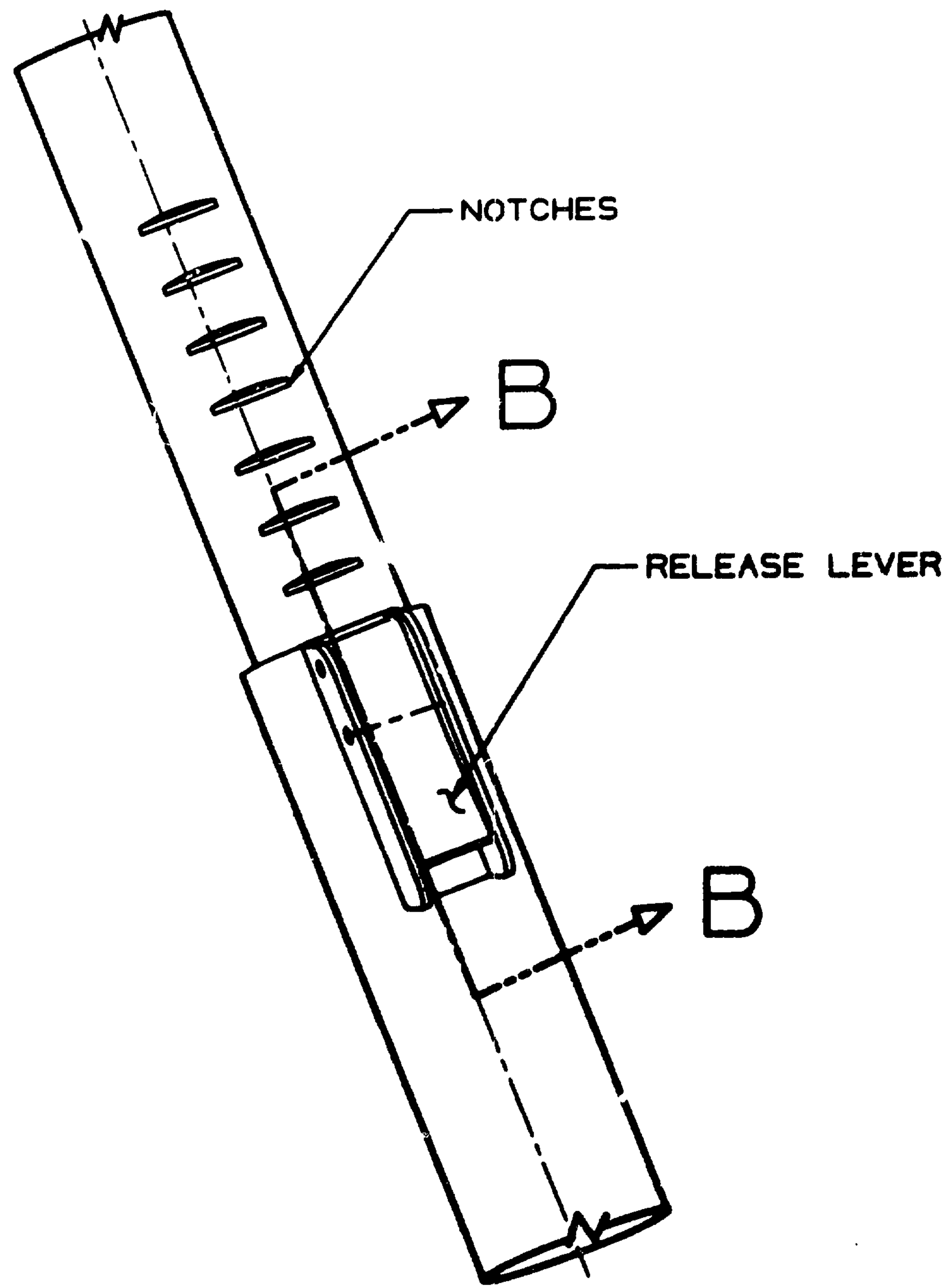
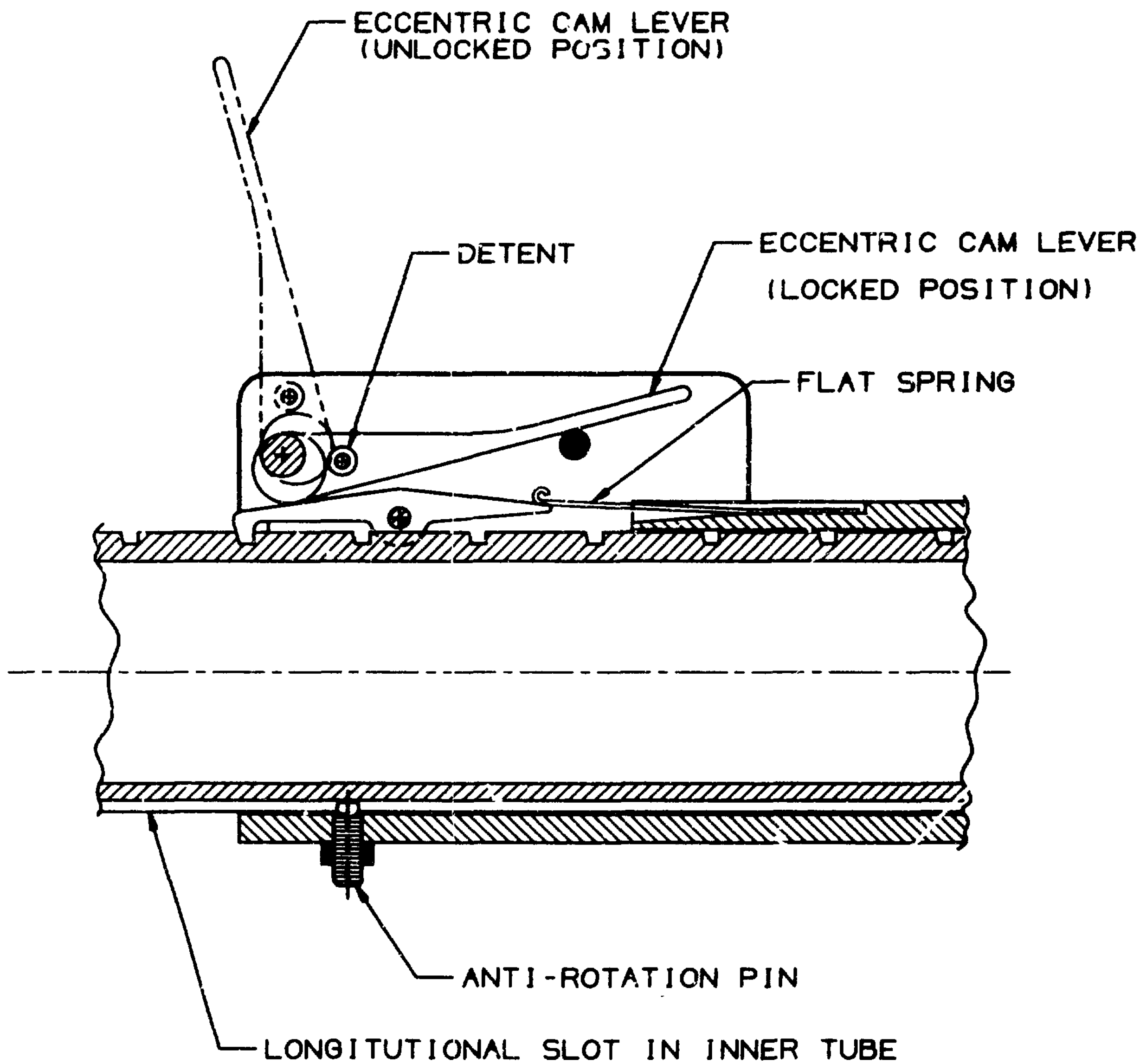


Figure 4.4.8.4-4 - TELESCOPING LOCK



SECTION B-B

Figure 4.4.8.4-5 - CROSS-SECTION THROUGH TELESCOPING LOCK

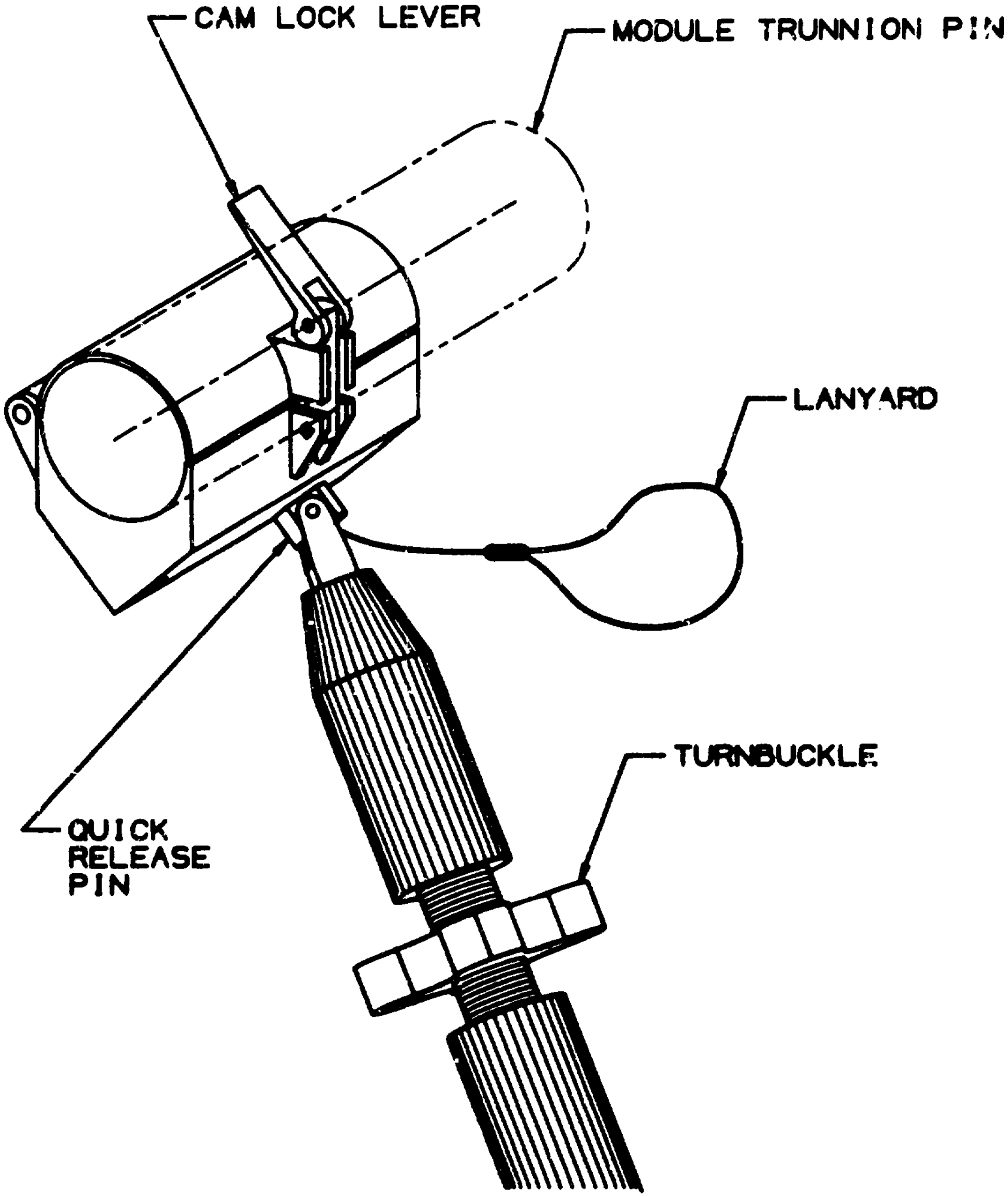


Figure 4.4.8.4 -6 - TRUNNION ATTACH FITTING

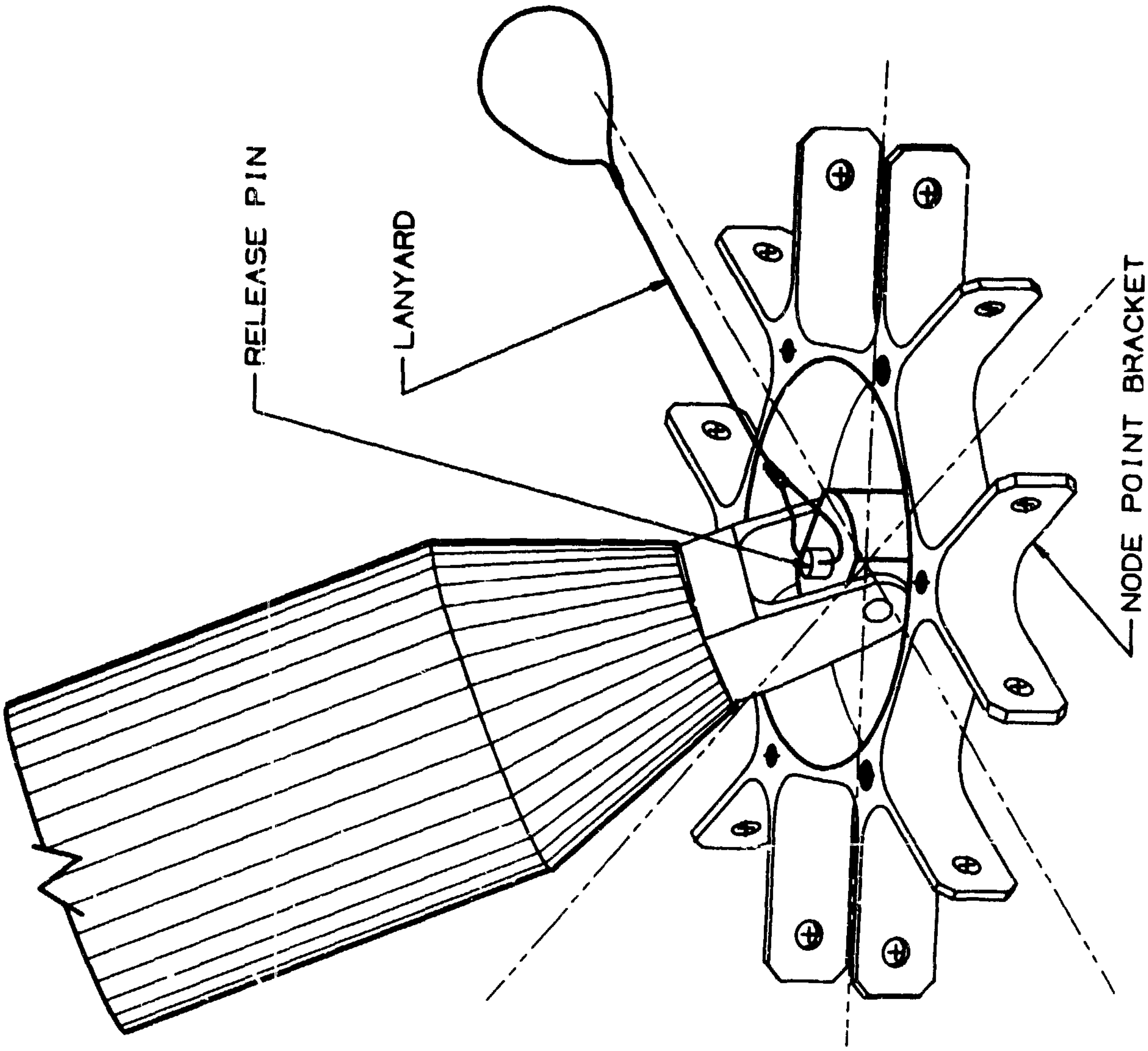


Figure 4.4.8.4-7 - NODE ATTACH FITTING

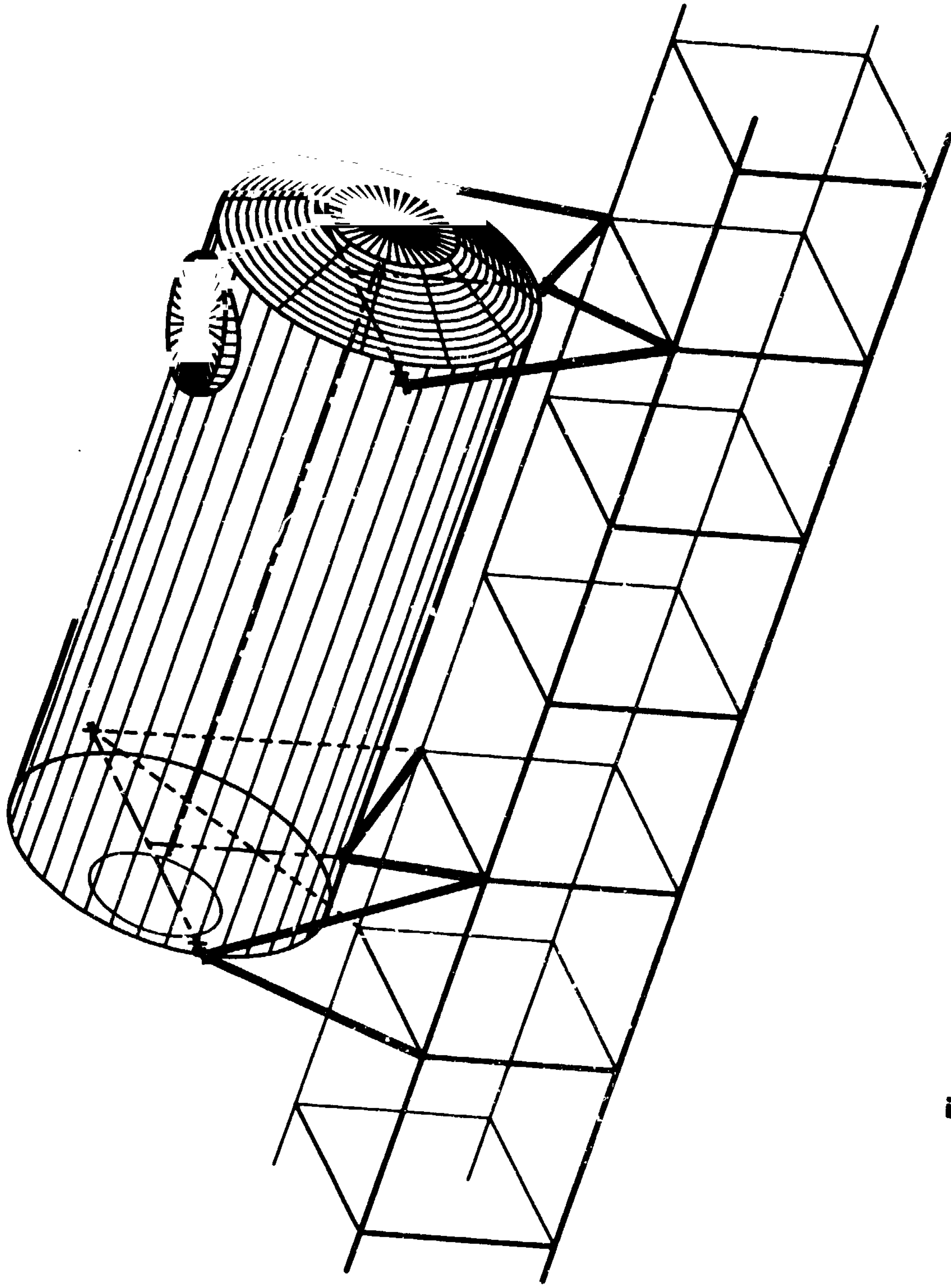


Figure 4.4.3.4-8 - MODULE ATTACHMENT TO 9 FT SQUARE TRUSS

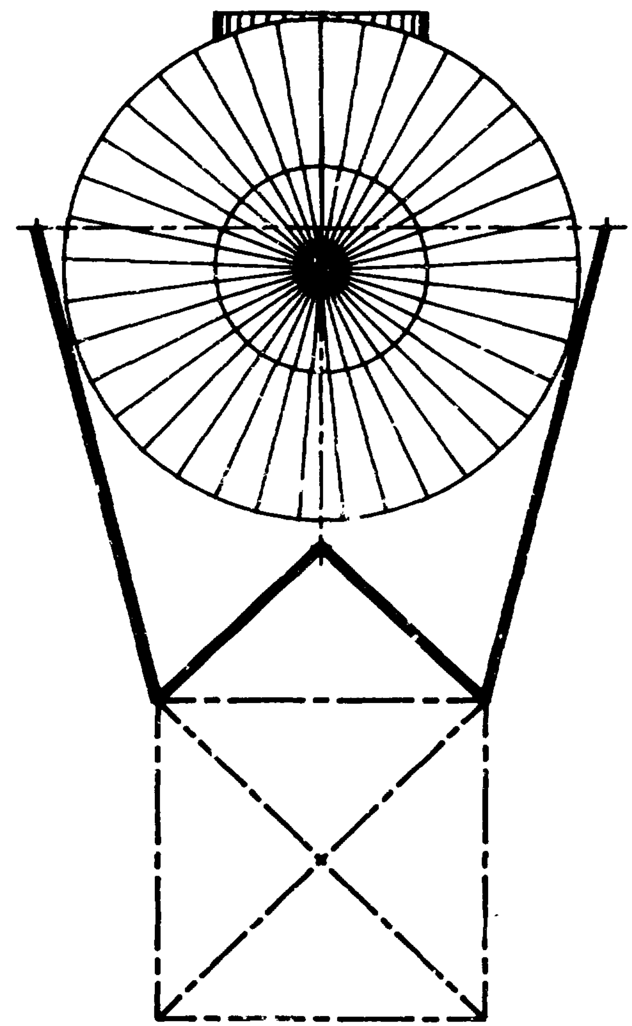


Figure 4.4.8.4 -9 - END VIEW OF MODULE ON 9 FT TRUSS

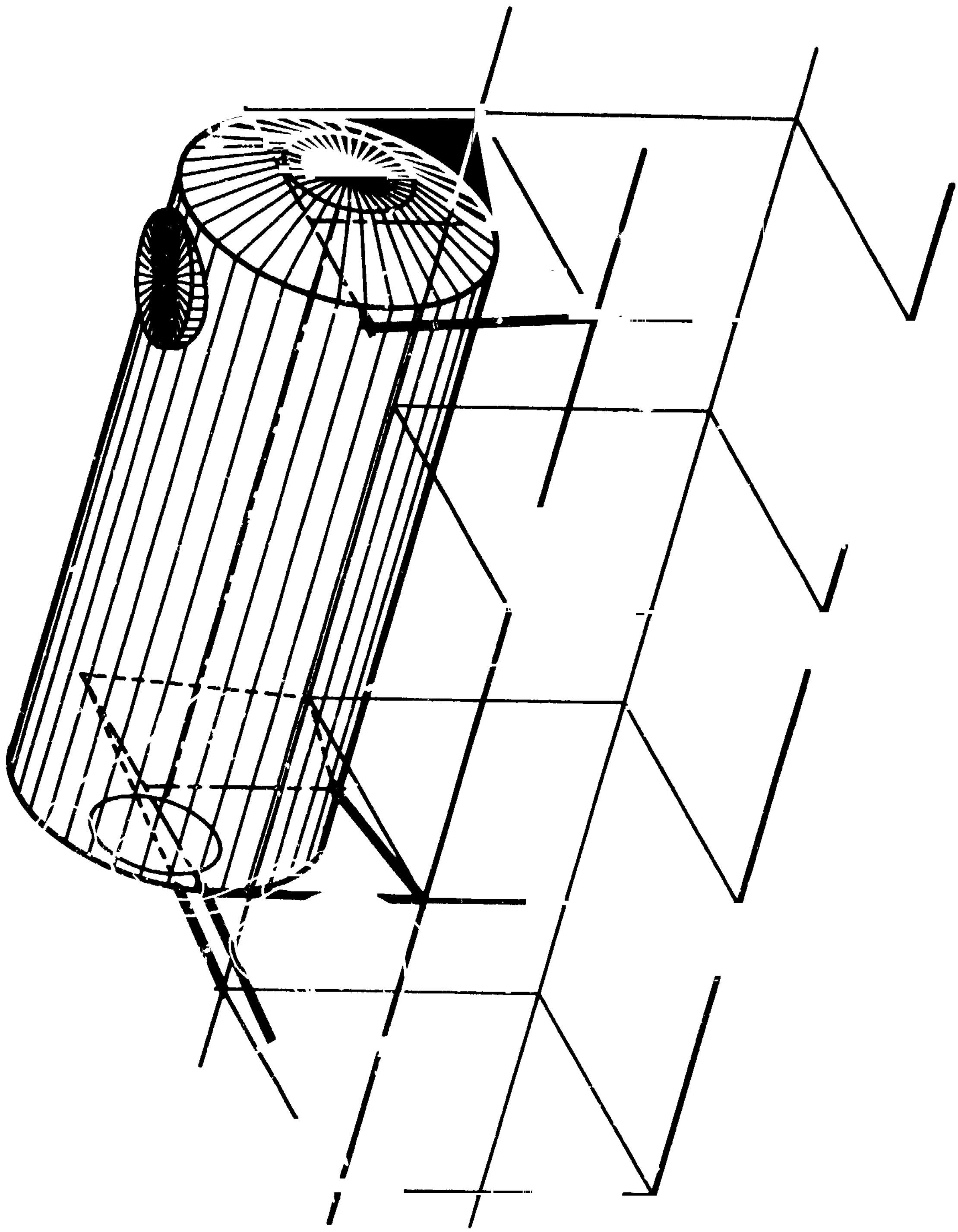


Figure 4.4.8.4-10 - MODULE ATTACHMENT TO 15 FT SQUARE TRUSS

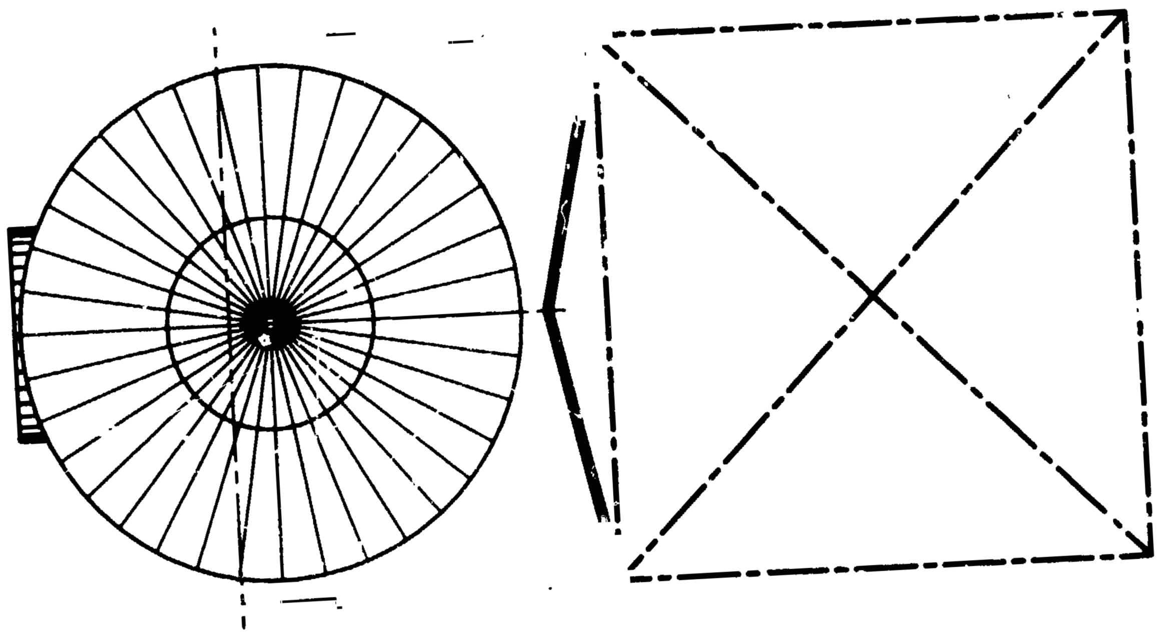


Figure 4.4.8.4-11 - END VIEW OF MODULE ON 15 FT TRUSS

and mechanical flexibility for both the truss-mounting and pressurized interfaces will be those associated with alignment and adjustment both for assembly operations and for control of the load paths for the assembled configuration.

The general design requirements for the module-to-module interface are

a. Physical interface capable of mating with any other module-to-module interface or to any docking/berthing port

b. Alignment and adjustment capability consistent with development of the module pattern specified

c. Capability for retraction of interface to provide clearance for module removal

d. Provision of pressurized clear passage way consistent with a 50-inch diameter hatch opening

e Provision of space for utilities interconnects

f. Provision of a latch system capable of accommodating pressure loads and dynamic loads in the mated configuration.

One design concept which meets these general requirements is shown in Figure 4.4.8.5-1. The physical interface incorporates the ring-finger guide concept as used in the Apollo-Soyuz Test Project system. The guides are only 4 inches high, consistent with the misalignments associated with berthing and assembly operations. Even in the relatively static assembly process, the guides will assist in achieving proper alignment of the modules prior to latching of the interfaces. Outer diameter of the interfacing structural ring is 80 inches.

With a system of four pairs of electromechanical actuators, the structural interface ring is provided with an axial stroke of 6 inches. The primary purpose of this stroke capability is to facilitate the assembly and disassembly process. The modules may be transported with interfaces retracted, which benefits the payload bay installation process.

After latching of the structural interfaces, the actuators will be fully extended. There is considerable room for designer's choice in the latching concept. The reference concept includes the structural ring mounted latch shown in Figure 4.4.8.5-2. Mated loads are carried through the locked actuators.

Once the structural interfaces are in the extended and latched position, separate small actuators extend telescoping pressure tunnel elements from either side of the interface. Tunnel elements are shown in the retracted and extended positions in Figure 4.4.8.5-3. Redundant pressure seals are incorporated throughout the pressure tunnel system, as shown in Figure 4.4.8.5-4. Extended life positive sealing is the driving requirement, so a totally passive seal is preferred. For leak detection, the ability to over-pressurize the cavity between redundant seals is highly desirable. The tunnel system will carry only radial pressure loads. Axial pressure loads and dynamic loads will be carried through the structural latch system. These loads are defined in Paragraph 4.4.8.5.3.3 of this section. Note that the telescoping pressure tunnel offers limited lateral or angular alignment capability. If increased alignment capability should prove to be a requirement, an extendible flexible pressure tunnel may be a more desirable solution.

Utilities interconnects will form a major part of the interface between modules and between individual modules and the truss structure. Major service utilities, (electrical power, thermal control fluid and vapor, and data systems) will be installed on the truss structure in bus fashion. The thermal control fluid and vapor (ammonia) system will be routed to liquid-to-liquid heat exchangers mounted outside the module pressurize volume. Power and data busses will be routed from the truss to umbilical plates on the sides of individual modules. Within the module string, fluid, gas, and air circulation interconnects must be included across the module-to-module interface. In addition, the electrical power and data busses will also be carried through the module-to-module interface. A detailed listing of the utilities supplied to each module is shown in Table 4.4.8.5-1.

The preferred method for completion of utilities interconnects across the module-to-module interface is manual connection within the pressurized tunnel. This reduces the need for sophisticated automatic umbilical mechanisms and renders these interconnects available for IVA inspection and servicing. As an alternative, the electrical power interconnects between modules could be mounted on the berthing interfaces rings and thereby automatically connected as the structural interface is completed.

4.4.8.5.2 Androgyny, indexing

The optimum geometry for the utilities interconnects depends heavily on the requirements for androgyny and indexing of the interfaces, hatch size and shape, and the inner diameter of the pressure tunnel.

With the system of four alignment guides, the modules may be connected in any of four orientations, 90° apart. The location and design of internal elements such as hatches and utilities interconnects will influence the number of possible orientations for which alignment of all elements can be achieved. For example, if a conventional D-shaped hatch is used, the flat part of the hatches will be aligned for only one module/module orientation. The sole criterion for alignment of internal elements is that the elements be symmetrical about an axis that passes through the edges of the alignment guides. See Figure 4.4.8.5-5. By arrangement of the utilities interconnects such that they are symmetrically located about one, two, or four axes, an equal number of module/module orientations will give total alignment of internal elements. Note that the hatch also must be designed with the same degree of symmetry for perfect alignment. Two orientations require a hatch with two flats, or rectangular or elliptical shape. Four orientations require that the hatch be a square or any other polygon divisible by 4.

As mentioned, the favored concept for utilities interconnects is manual completion of the connections within the pressurized environment. For this concept, the intramodule utilities will be terminated on the interface base ring (to which the hatch is mounted). The area around the hatch will be used for manual installation of jumper connections between interface base rings, which in the mated position are about 30 inches apart. With flexible jumper assemblies, it is not mandatory that all utilities interfaces be aligned for all module/module orientations. (Nor has it been established that perfect alignment of hatches is mandatory.)

The use of inductive couplings (cone pairs) for electrical power interconnects offers the option of automatic connection as the structural interface is completed. For the option, the cone pairs are mounted on the berthing interface ring outside the pressure tunnel and are joined as these interfaces are brought together. This creates a special case for interconnect geometry in that the cone pairs comprise male/female sets. Since a polarized connector cannot be located on an axis of symmetry, the minimum number of connector positions is twice the desired number of module/module orientations. That is, eight connectors (four male, four female) are required to accommodate all module orientations provided by the four guide berthing system.

4.4.8.5.3 <u>Docking/berthing interface</u>

The Shuttle Orbiter will join the Space Station at regular intervals for resupply, exchange of crew, and return of manufactured items and waste products. The Orbiter will also be used as the base for assembly of early Station elements and must join with those elements already in orbit as it delivers a new element. The concept of loosely coupling the two craft through a flexible pressure tunnel and accomplishing transfer of materials and elements by RMS to manipulator hand-off (with EVA assist as required) has been considered. The advantage of this concept is that it eliminates the need for a rigid structural coupling and contact forces from berthing or docking that might be an order of magnitude higher than any other operational forces applied to the station. Stay time of the Orbiter in close proximity to the station will range from hours to days.

Attitude control of both the Station and the Orbiter must be maintained during close proximity operations. In addition, the Orbiter must translate as required to maintain the station keeping position without risk of collision with the Station. Orbiter attitude control and translation maneuvering requires the use of RCS thrusters. Periodic firing of RCS thrusters in the vicinity of the Station cannot be permitted because of contamination and plume impingement forces. Therefore, joining the Orbiter to the Station by a structural interface through which the Station can provide attitude control of the Orbiter is considered mandatory.

Concepts for bringing the Orbiter structural interface into contact with the mating station interface must be evaluated against two primary constraints: (1) impingement of Orbiter RCS plumes on station elements must be minimized and (2) interface contact forces must be minimized. Recent studies have typically adopted the berthing concept, wherein the Orbiter RMS grapples the Station and maneuvers the interfaces into contact at extremely low velocities (0.1 fps or less). However, the Station mass will far exceed the design capability of the RMS, which is 32,000 pounds. Considerably higher masses can be handled by limiting the rate of movement. Further gains are possible with minor to moderate changes in RMS hardware and control software. More extensive system modifications will be required to gain full capability for RMS controlled berthing with the Station especially with respect to initial capture of the Orbiter. The RMS contractor is currently examining the effects on dynamic performance of doubling the joint brake torque and equalizing the torque capability of the shoulder and wrist joints.

A practical alternative to RMS controlled berthing is docking, whereby the Orbiter translates directly into interface contact using the RCS thrusters. Recent flight experience and simulations have shown that the current Orbiter systems can place the interfaces into close proximity at rates of 0.1 fps or less, provided that the pilot has adequate visual or sensor coverage from which to determine control input requirements. Recent developments in laser sensor technology can be applied to provide precise final approach control information.

The selected approach for the Station reference configuration is to provide an interface mechanism that is fully compatible with either RMS controlled berthing or docking operations. Detailed evaluation of developing RMS capability, man-in-the-loop simulation of the Orbiter final approach to RMS grapple or docking, and detailed study of interface mechanism design and performance will be required to establish the technique that offers the best capability for achieving low-force mating with minimum RCS plume impingement for both normal operation and for design failure modes.

4.4.8.5.3.1 Design concept

The docking/berthing mechanism will be mounted to a telescoping module mounted in the front of the Orbiter payload bay as shown in Figure 4.4.8.5-6. In the extended position the interface ring is 27 inches above the Orbiter outer moldline. With actuators fully retracted, the interface remains 15 inches outside the mold line. The alignment guides and the interface ring are identical to the module-to-module interface. This is so that the Orbiter can be berthed to any module port, though the end ports are preferred because of the greater clearance provided.

The basic concept for design of all elements of the docking/berthing mechanism is to achieve positive mating of interfaces with minimum forces. The heart of the system is a laser sensor which will provide precise information on the relative positions and rates of the interfaces as they are brought into contact. A system of four pairs of electromechanical attenuator/actuators control the position of the docking interface relative to its base ring. The actuator system provides an overall stroke capability of 12 inches.

The structural latches are identical to those shown in Figure 4.4.8.5-2. An alternate capture latch concept with extended reach capability is being evaluated. The extended reach latch (see Fig. 4.4.8.5-7) can be used to achieve low force control of the approaching interface while the interfaces are still several inches apart. This concept can expand the capture envelope by reducing positioning accuracy requirements, thereby making possible either reduced contact velocity or reduced RCS thruster firings, or both.

The docking/berthing mechanism will incorporate an extendible pressure tunnel which mates with the module-to-module tunnel interface. Manual completion of required utilities interconnects will be accomplished as for the module-to-module assembly process.

4.4.8.5.3.2 Operation

The docking/berthing interface will be brought into close proximity to the station berthing port, either by using a modified Orbiter RMS within the limit of its capabilities or by directly approaching with the Orbiter under RCS control. The laser range/rate/attitude sensor system will be active during the rendezvous and approach phase, providing the data required to minimize approach velocity as the interfaces are brought into proximity.

After latching of the interfaces and stroking of the actuator system to eliminate relative velocities and angular rates, the actuators are slowly driven to the retracted position. The pressure tunnel is then extended from either the Orbiter or station side of the interface. Required utilities interconnects are manually completed within the pressurized interface.

4.4.8.5.3.3 Performance

The alignment guide height of 4 inches can accommodate lateral misalignments somewhat greater then + 3 inches in combination with pitch and yaw misalignments of about + 5 degrees. Roll misalignment of + 5 degrees can also be tolerated. The lateral misalignment capability approaches + 4 inches as the pitch and yaw misalignments are reduced. This capability is consistent with RMS design berthing performance and with projected Orbiter docking capability if adequate information on the relative position of interfaces is available to the pilot. An increase in guide length to gain greater misalignment tolerance would impact the useable internal volume of the modules unless folding guides were provided. An alternative is to provide longer guides on a limited number of identified berthing ports and of course, on the Orbiter docking/berthing mechanism.

Interface contact velocity for RMS berthing operations can be limited to 0.1 fps. The major problem with berthing of very large masses is that tip velocity must be severely restricted because of arm flexibility/control interactions and joint drive torque and braking capability. Current dynamic studies indicate that it may be necessary to limit the velocity for large masses to 0.05 fps or less. This is not a problem for the berthing interface mechanism. The real difficulty lies in the RMS track and capture maneuver. The Orbiter must position itself under RCS control within the working track, capture, and deceleration envelope of the RMS, at a very low velocity relative to the Station. For the current RMS, this working envelope and associated relative velocity may shrink to zero as the Station mass increases. As previously mentioned, significant improvement in RMS capability to handle large masses is possible. However, during evaluation of potential RMS capability it became clear that the requirements for positioning the Orbiter for RMS grapple might be more severe that those being used for design of the berthing interface, so the obvious question is, "Why not simply dock?"

Orbiter flight experience and recent flight simulations show that the Orbiter can fly to a relatively precise position with a residual velocity of about 0.03 - 0.04 fps using the normal Z RCS mode and with a velocity less than 0.1 fps using the low Z RCS mode. Flying in to dock versus flying in to grapple with the RMS will place the Orbiter 20-30 feet closer to the station while the RCS system is still activated and a detected RCS failure could lead to an abort with extensive RCS thrusting away from the station (plumes toward the station). The choice of soft docking versus RMS controlled berthing must be based on detailed study and simulation of

a. Improved RMS capability to track, capture, and maneuver large masses

b. Orbiter approach to RMS grapple and to closure of docking interfaces, specifically with regard to resulting impingement of RCS plumes on the Station

c. Failure modes and effects

d. Operating capability of docking/berthing mechanisms

More detailed discussion of Orbiter proximity operations and the nature and magnitude of the RCS plume impingement problem may be found in Paragraph 4.3.8.2.3.4 of the Operations Section of this document.

Loads applied to the Station during the docking/berthing process must be defined through detailed simulation of the process, including simulation of the mechanism attenuation characteristics. The large offset between the Orbiter center of gravity and the docking/berthing axis renders crude simulations subject to significant errors. Estimates predict that the Orbiter berthing or docking maneuvers can be completed with an applied load to the station of less than 500 pounds.

The loads across the mated module-to-module and docking/berthing interfaces are dominated by the pressure load, which for the reference tunnel diameter is 48,800 pounds. The maximum predicted dynamic environment, at the berthing port with Orbiter mated is an axial force of 102 pounds and a moment of 4,089 foot-pounds. This is based on an acceleration of 3.8×10^{-4}g applied to the Station, an Orbiter weight of 269,000 pounds with the Orbiter c.g. displaced 40 feet from the berthing axis. The reference design includes eight latches which gives a design latch load of 6,100 pounds to accommodate pressures and about 6,800 pounds if the worst case dynamic load is applied.

SUBSYSTEM	NO.	COMMODITY	CONNECTOR SIZE
ELECTRICAL POWER	2	25 KW, 200 VAC	6" DIA.
OPTICAL HOUSEKEEPING DATA NETWORK	6	FIBER OPTICS DATA BUSSES	1/2" DIA.
ENVIRONMENTAL CONTROL	2	AIR INTAKE, EXHAUST	4" DIA.
	2	DRINKING WATER	1-1/2" DIA.
	2	WASTE WATER	1" DIA.
	2	O_2 SUPPLY	1" DIA.
	2	N_2 SUPPLY	1" DIA.
	2	WASTE WATER (CONDENSATE)	1" DIA.
	2	WASH WATER	1" DIA.

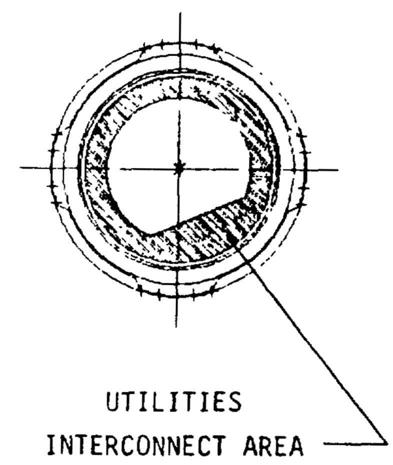


Table 4.4.8.5-1 - MODULE TO MODULE UTILITIES TRANSFER REQUIREMENTS

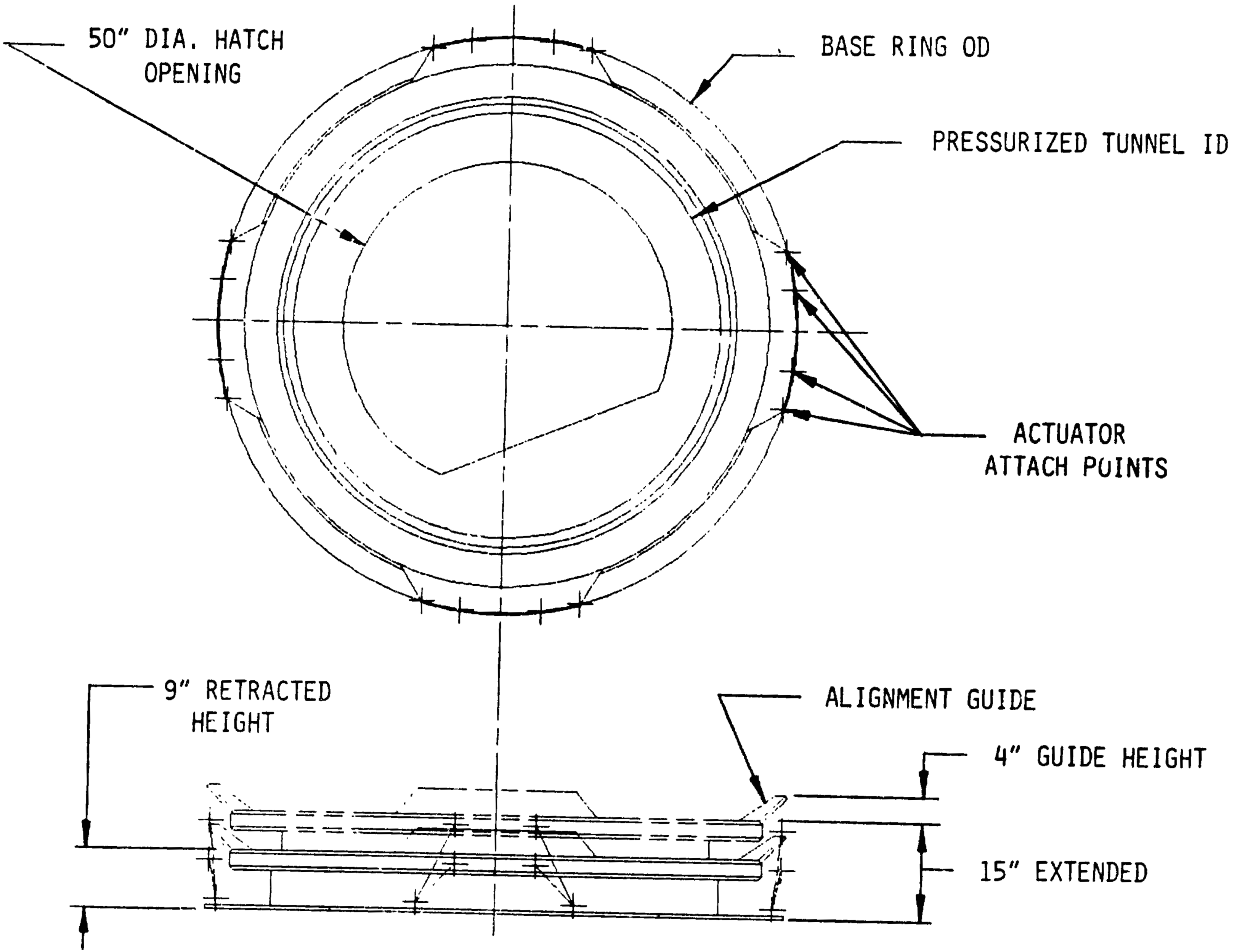


Figure 4.4.8.5-1 - MODULE TO MODULE INTERFACE

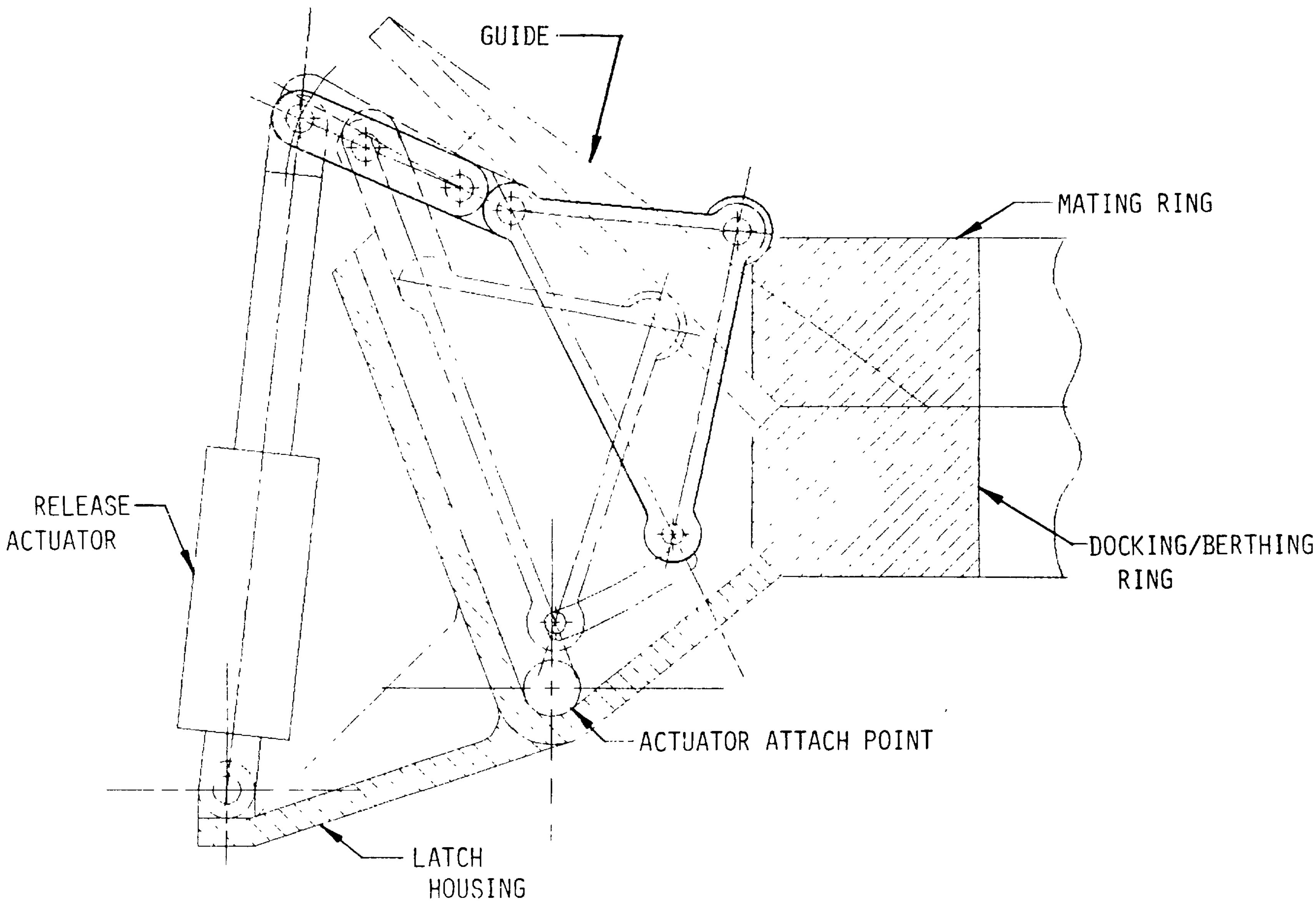


Figure 4.4.8.5-2 - DOCKING/BERTHING CAPTURE/STRUCTURAL LATCH

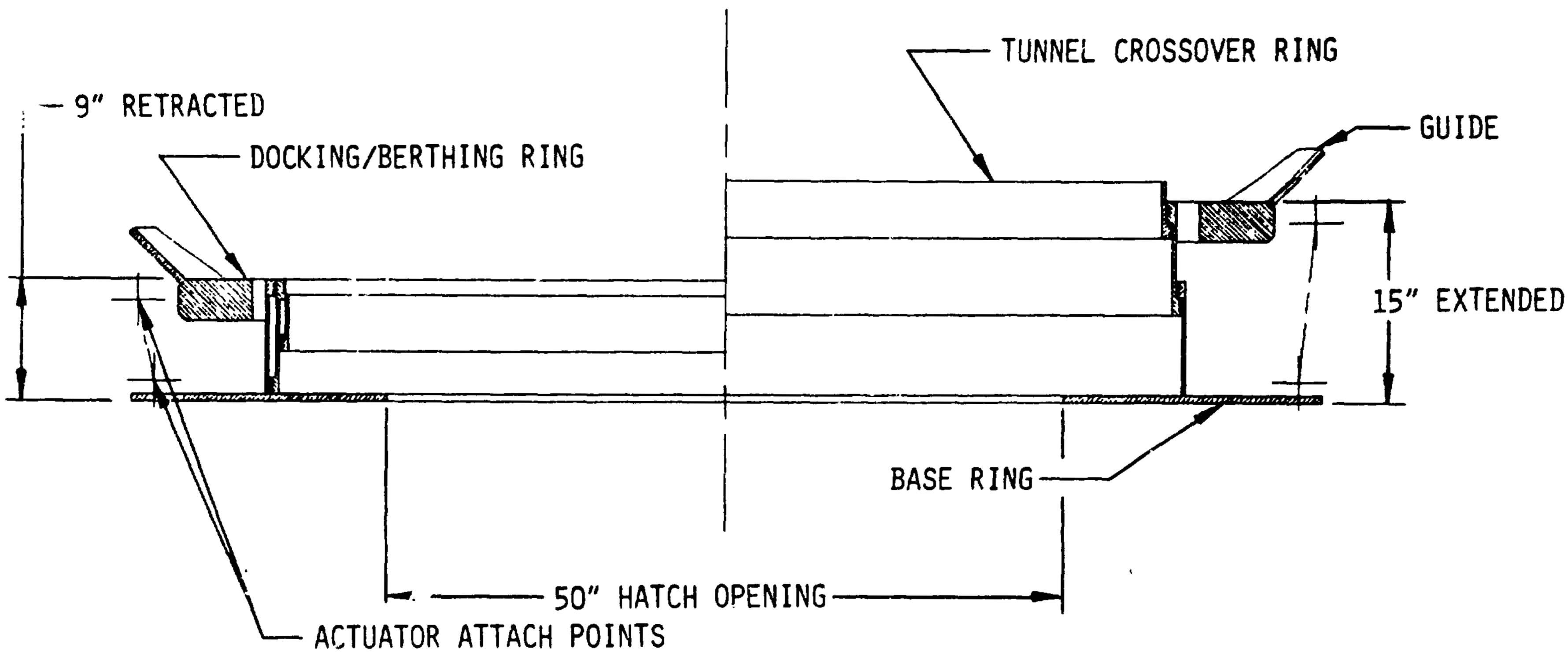


Figure 4.4.8.5-3 - TELESCOPING PRESSURE TUNNEL

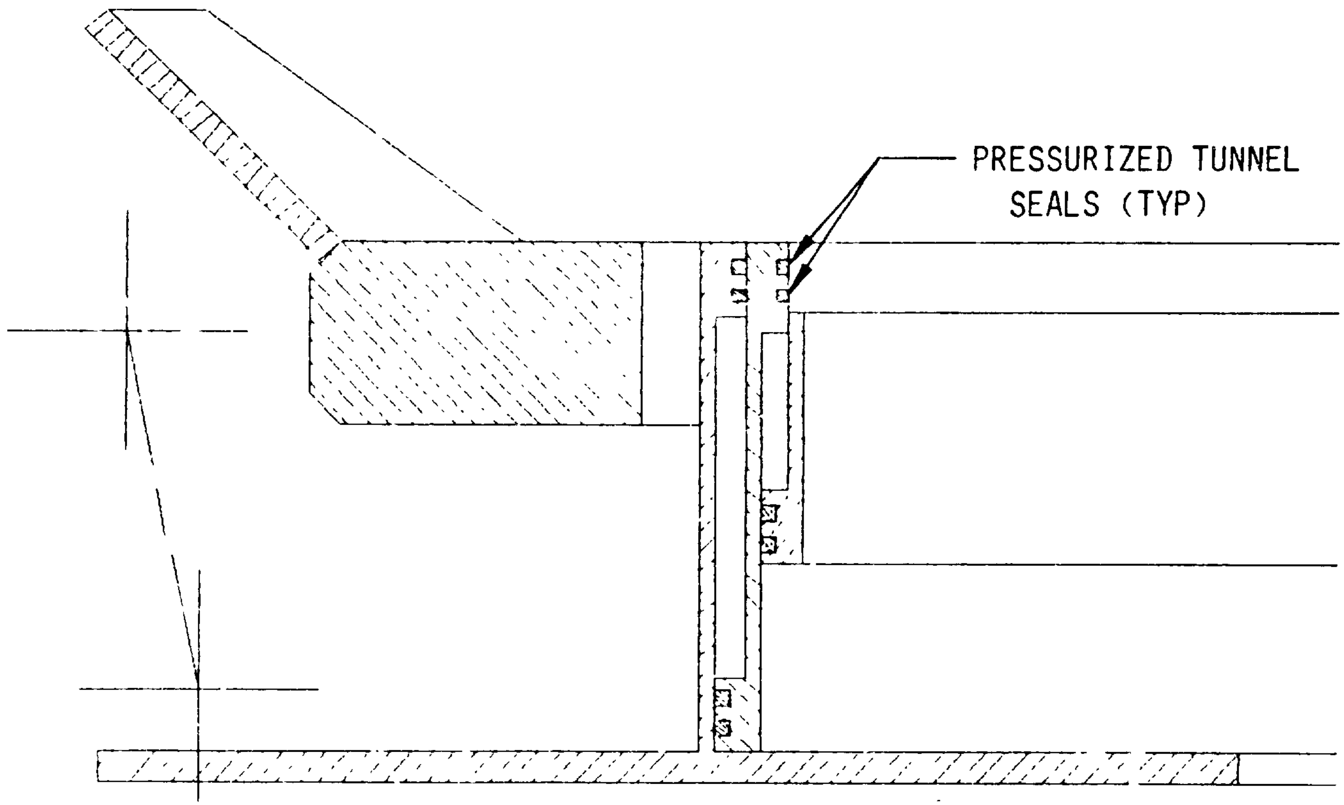


Figure 4.4.8.5-4 - PRESSURE TUNNEL SEALING CONCEPT

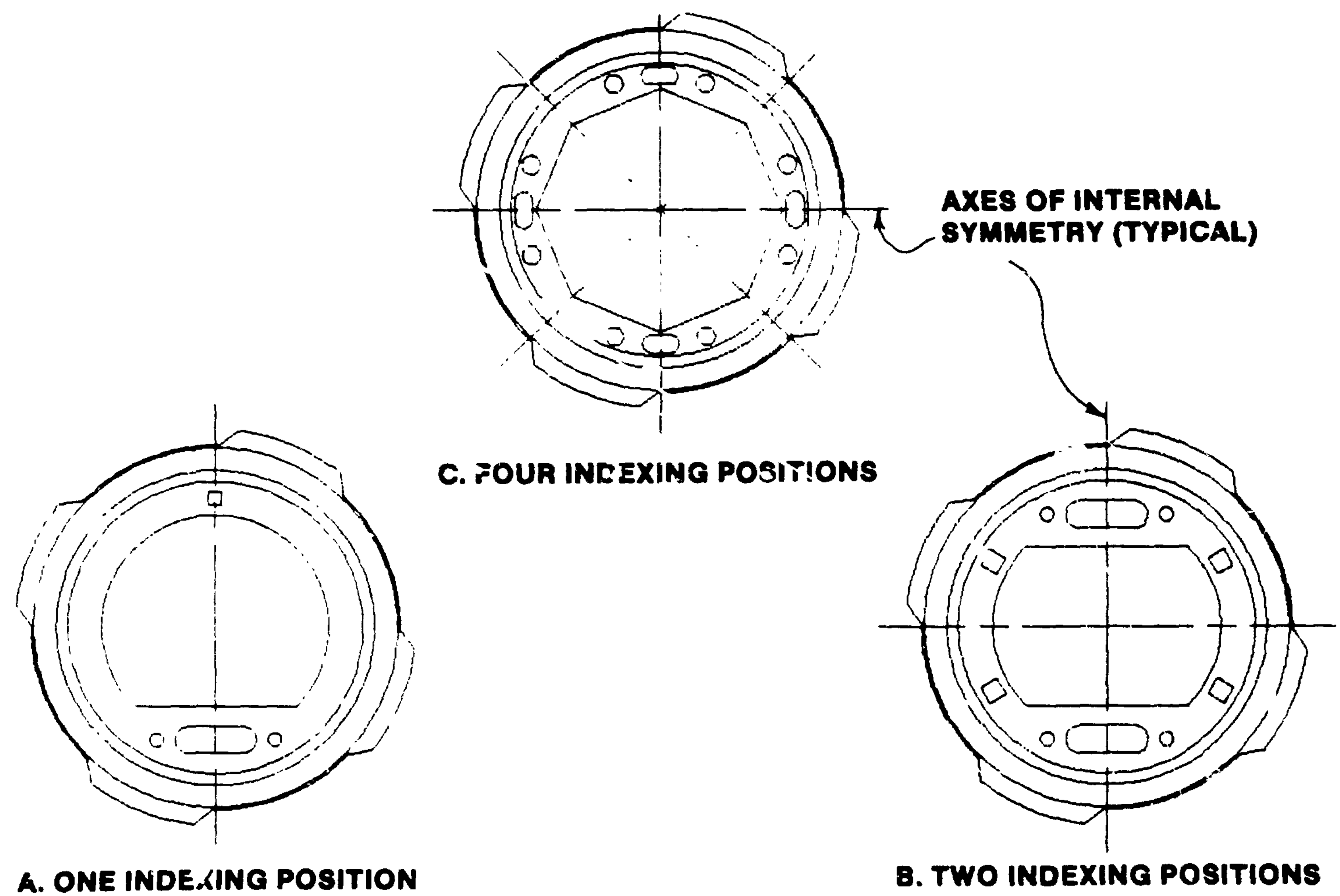


Figure 4.4.8.5-5 - INTERFACE INTERNAL SYMMETRY FOR INDEXING

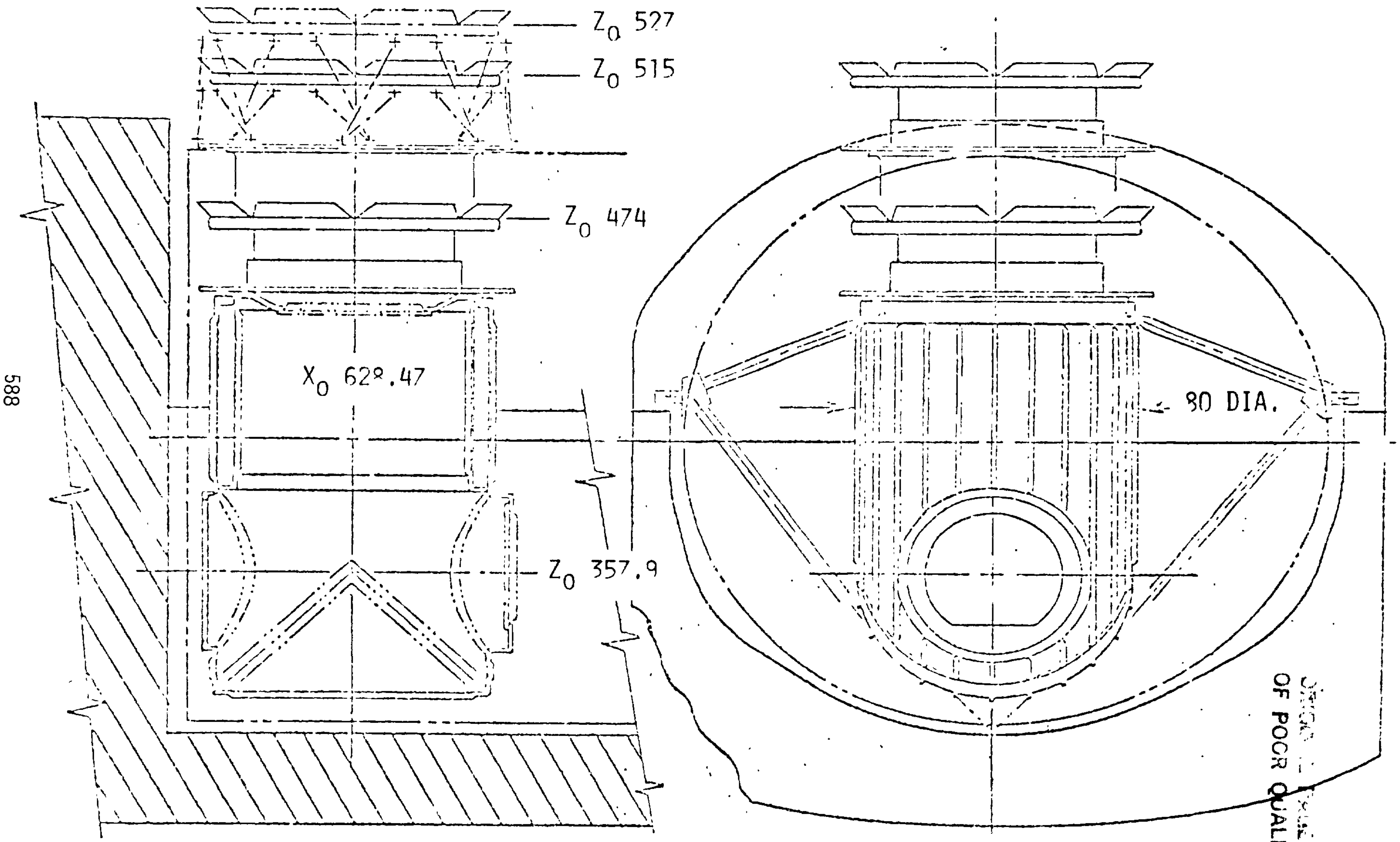


Figure 4.4.8.5-6 - ORBITER DOCKING/BERTHING MODULE

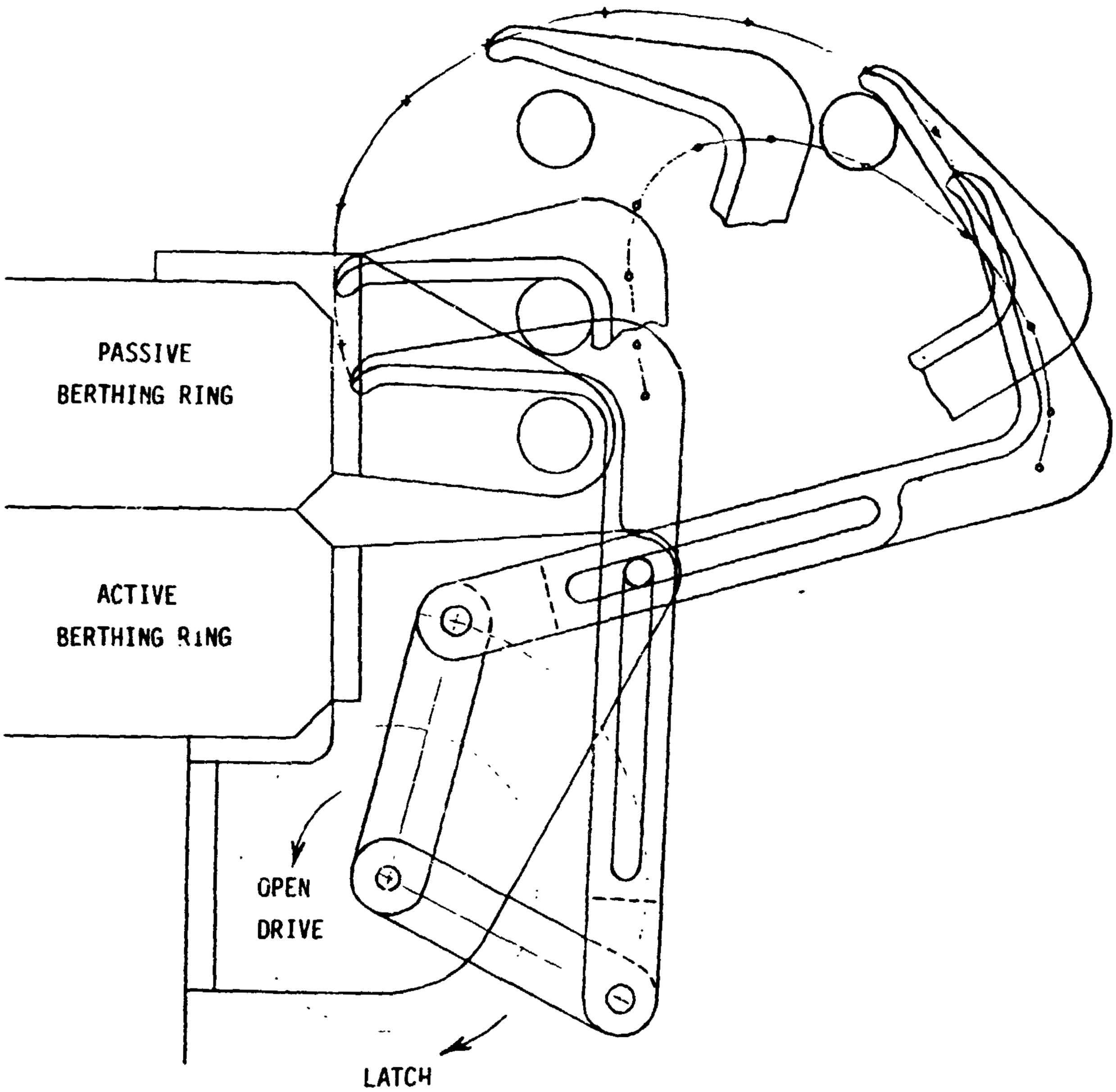


Figure 4.4.8.5-7 - EXTENDED REACH CAPTURE LATCH CONCEPT

4.4.8.6 Solar Panel Deployment Mechanism

The packaging and deployment concepts for the Space Station solar array elements are based on minimum extension of the technology developed over the last few years to support various spacecraft and studies of advanced platforms and the Space Station.

The heart of the solar panel deployment system is an automatically deployable boom structure which forms a central mast that supports array panels on either side. Several deployable boom concepts which meet the general packaging and stiffness requirements have been developed. The reference Space Station system is not based on a specific design. That choice is left to the designer based on a more detailed study.

The automatically deployable booms suitable for this application are not redundant, either structurally or in the deployment mechanism and process. Further, the boom structures have little damage tolerance. Damage to a single element can cause buckling of the total boom assembly. In the reference concept, damage tolerance and a degree of redundant capability is gained through overall design of the solar array system wherein, for the IOC power level, eight independently deployed booms are included. This distribution of power systems will minimize the effect of boom deployment failures or operational damage, and will simplify replacement of a boom assembly, should that be required.

4.4.8.6.1 Packaging

Individual 960" x 180" solar array panels are accordian folded into a 186" x 12" x 9" rectangular box. One is mounted to either side of a cylindrical boom deployment canister (see Fig. 4.4.8.6-1). The boxes are rotated 90° about the boom deployment axis such that the two boxes are adjacent and extend perpendicular to the boom deployment canister. In the reference configuration, the bases of two deployment cannisters are mounted to either end of the beta rotation mechanism, which is centrally attached to elements of the power boom truss. The deployment canisters are 30 inches in diameter. The total length of two deployment canisters, including the beta rotation interface, is 204 inches.

4.4.8.6.2 Deployment

In the favored deployment concept, the power boom is deployed with the array system in the packaged configuration just described, yielding the configuration shown in Figure 4.4.8.6-2. The individual array boxes are then rotated to the deploy position with an array blanket box on either side of the mast deployment canister as shown in Figure 4.4.8.6-3. Lastly, the transverse beam which forms the top of the array panel boxes is released and the mast extends, unfolding the array blankets (Fig. 4.4.8.6-4). Note that the individual array blankets are attached only to the transverse beam at the deployed end, and to the array box on the cannister end. The accordian-folded array blankets are flattened by a negator spring tension system following deployment of the mast.

4.4.8.6.3 Deployable mast design

As previously mentioned, several concepts for automatically deployable booms have been developed or advocated. These include cylindrical booms, which are formed by one or more metallic, cylindrical, thin, shells. Also included are coilable lattice booms, beam structures whose longeron and batten members are usually fiberglass rods and whose diagonals are typically steel cables. Another type is articulated lattice booms, whose longeron and batten members are segments of metallic tubing that are not deformed when the boom is retracted.

A detailed evaluation of existing boom concepts, and others that might develop is needed to evaluate deployment reliability with the consideration of incorporating redundancy in the deployment process.

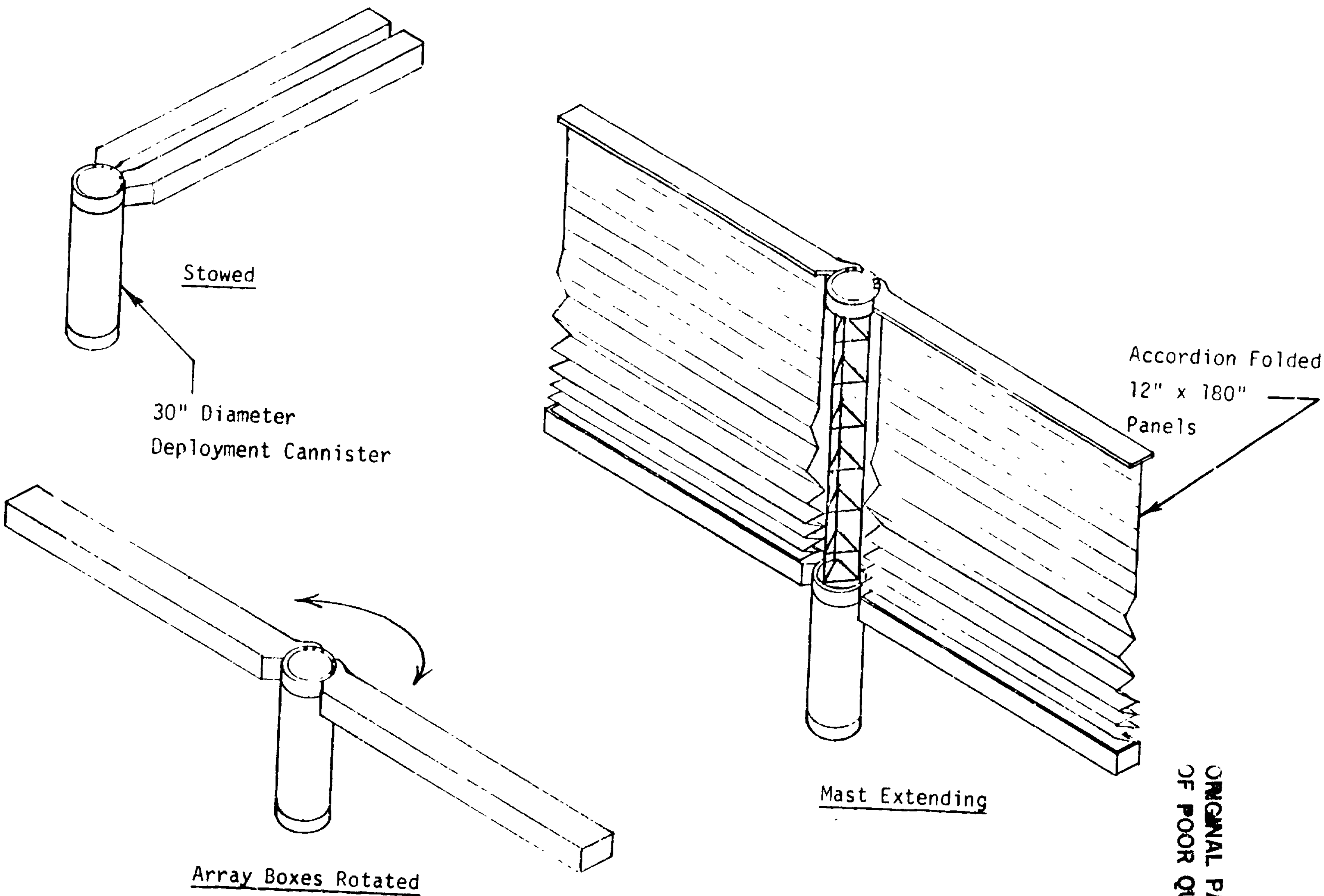


Figure 4.4.8.6-1 - SOLAR PANEL PACKAGING/DEPLOYMENT CONCEPT

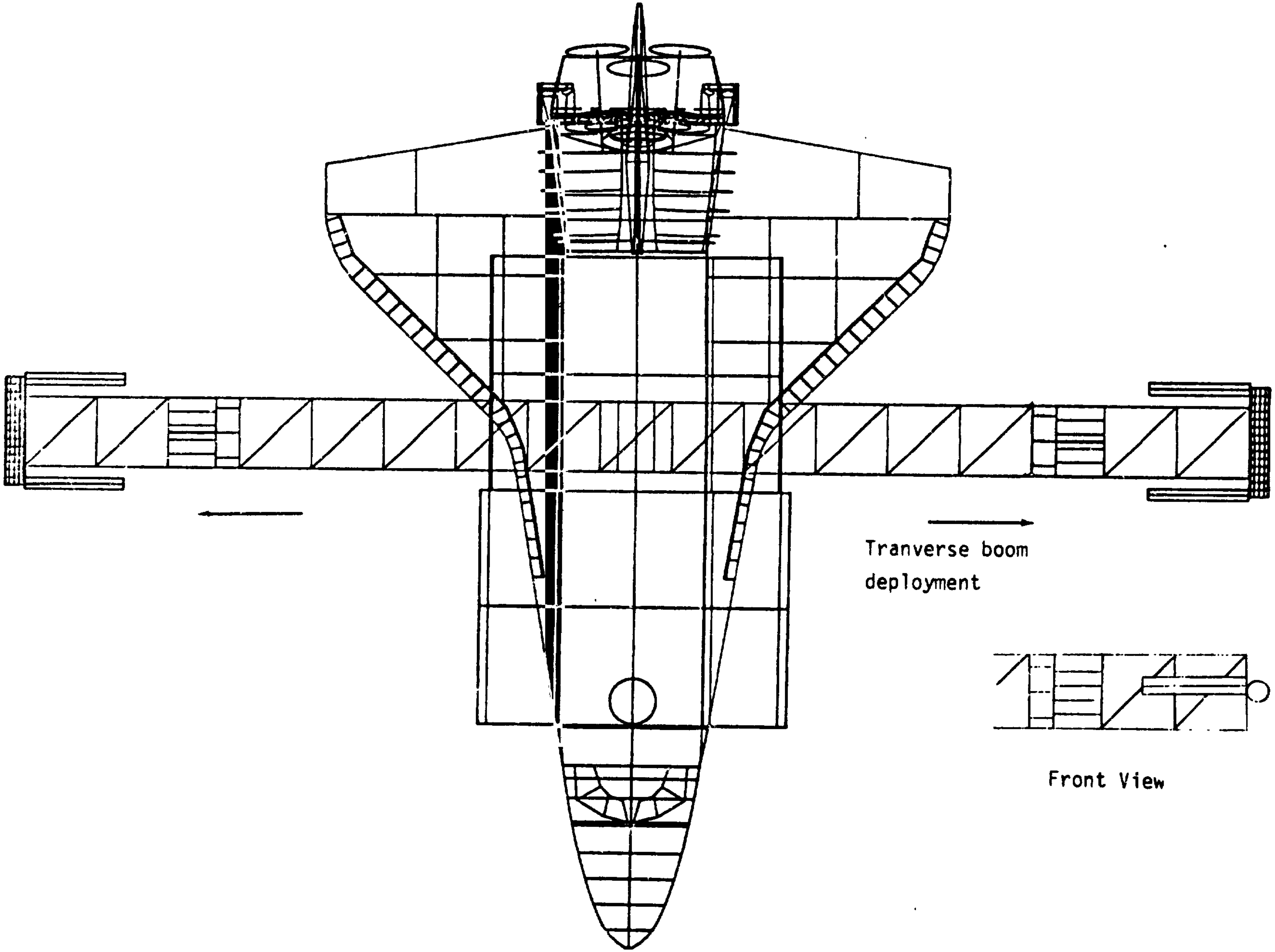


Figure 4.4.8.6-2 - SOLAR PANEL SYSTEM READY FOR DEPLOYMENT

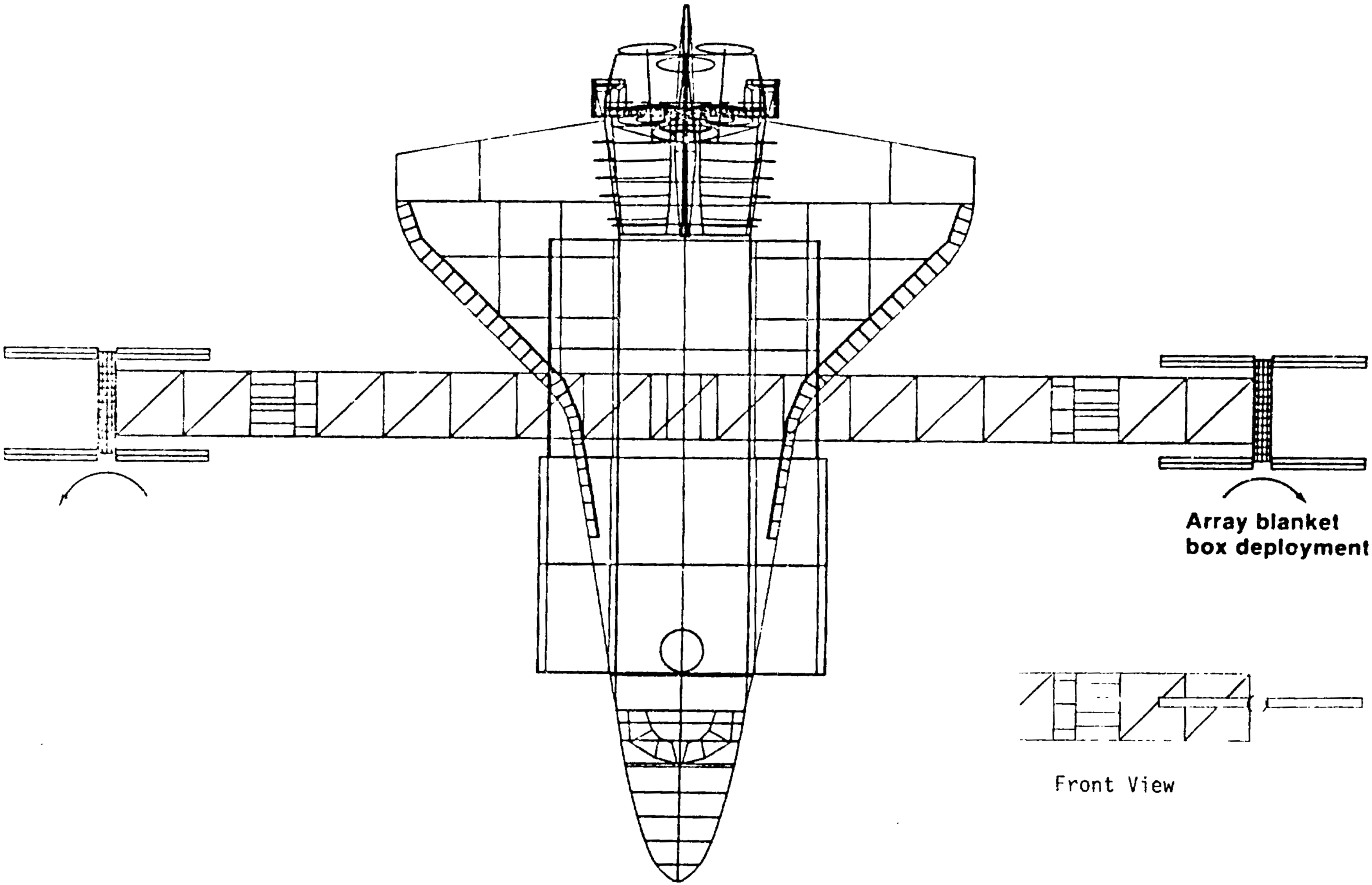


Figure 4.4.8.6-3 - PANEL STOWAGE BOX ROTATION

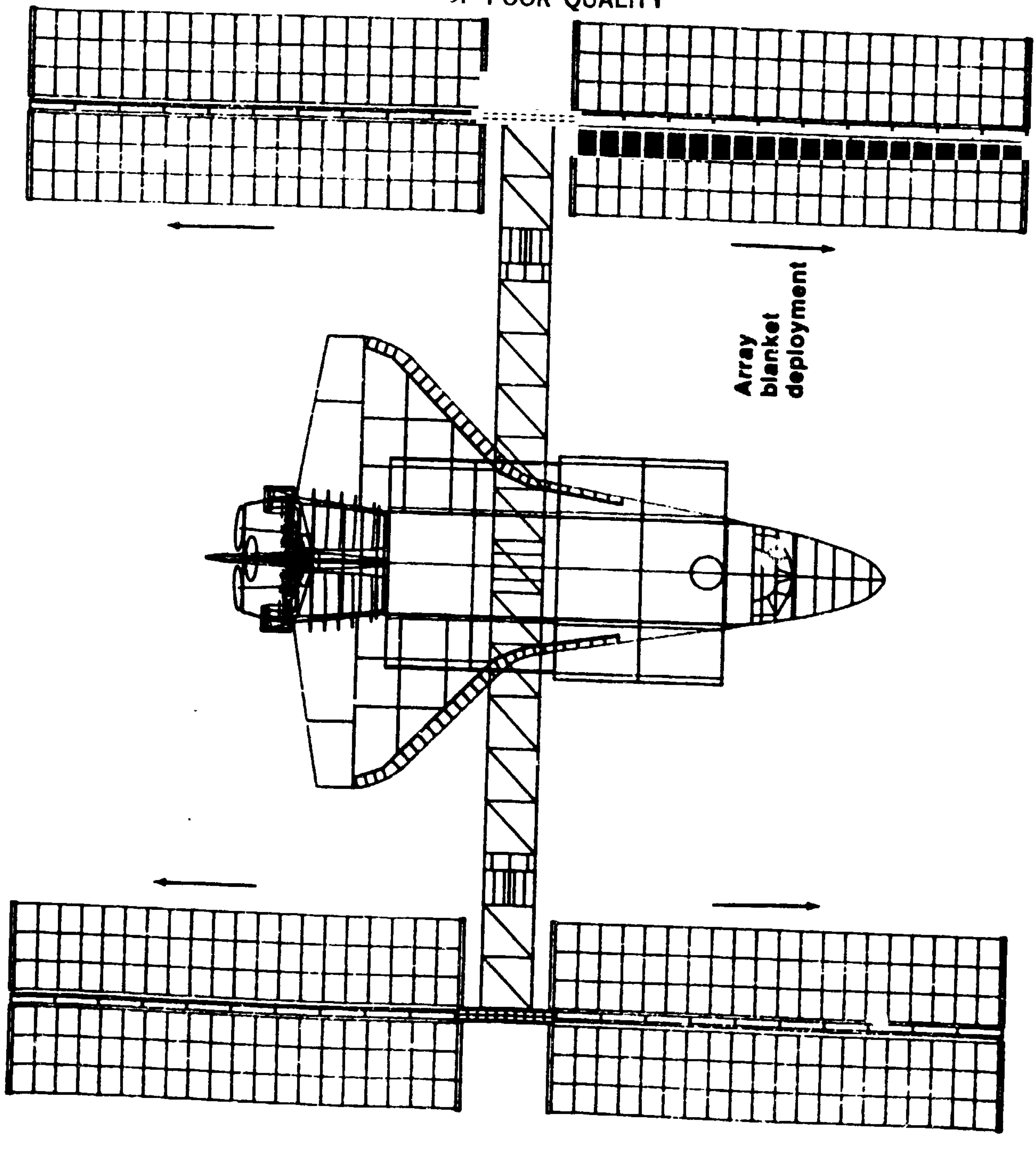


Figure 4.4.8.6-4 - ARRAY DEPLOYMENT MAST EXTENSION

4.4.8.7 Conceptual Rotary Joint Design for Space Station Applications

The reference Space Station configuration has a functional requirement for three sets of rotation devices to provide pointing of the solar arrays (alpha and beta axes) and the thermal control system radiators (alpha axis only). Of the three drives, only the alpha-axis drive on the power boom is required to provide a continuous rotation capability. The beta axis solar array drives (at the base of each of the array masts) 105 degrees at a very low rate and the alpha-axis radiator drives (at the nadir end of the lower keel) drive approximately 260° and then rewind to an initial starting position while the Station is on the dark portion of its orbit.

All of the drives must conduct a significant portion of one of the major Station utilities across the rotating interface. In addition the solar array alpha joints function as primary structural elements on the Power Boom.

A 3-week study was undertaken at the Langley Research Center in support of the Space Station Project Office to develop a "strawman" design concept of a single continuously rotating solar array Alpha joint. The purpose was to point out major system issues in the design of such a joint and provide a guide to the areas requiring further study.

At the time of the study inception, some configurations being considered required both the thermal and electrical utilities to be carried across the rotating interface. Therefore, a "worst case" configuration which transferred electrical power and thermal fluid/vapor across the joint was chosen for this effort. The structural issue was addressed by imposing a design constraint requiring the stiffness distribution of the attached truss structures be maintained across the joint. A more detailed presentation of the design requirements for the study is located in the specific White Paper dealing with this effort.

4.4.8.7.1 Joint description

The rotary joint concept developed consisted of three major parts:

a. A fluid coupling

b. A power coupling

c. A mechanical drive coupling

Figure 4.4.8.7-1a and Figure 4.4.8.7-1b show the joint concept using a slip ring assembly or an inductive power coupling. The fluid coupling and power coupling are located in tandem on the centerline of the joint and are suspended from the mechanical drive housings. The mechanical drive forms the link between sections of the power boom truss elements.

The fluid coupling consists of a ported rotary shaft inside a mating stationary housing. Eight fluid lines pass through the rotary shaft carrying liquid and gaseous ammonia for four separate circuits. A 4-inch diameter cylindrical conduit is provided through the center of the shaft to allow fluid and electrical connections to the power coupling. The shaft extends through the housing and attaches to the power coupling elements which rotate with the shaft.

The power coupling is sized to carry 100 kW of electrical power through either a slip ring assembly or an inductive coupling. A separate cooling loop would have to be provided for the inductive coupling.

4.4.8.7.2 Fluid coupling design overview

The fluid coupling design concept is capable of handling the ammonia flow required for 159 kW of cooling, and of providing for continuous rotation of one side of the joint relative to the other. Figure 4.4.8.7.2-1 provides a cross section view of the coupling.

The coupling has an approximate overall length of 31 inches including the shaft extension to mate with the power coupling. The larger diameter is approximately 14.5 inches, and the small diameter about 9.7 inches. The overall weight is estimated at 500 pounds based on the use of 316 stainless steel as the primary structural material. It is possible that the weight could be reduced significantly, but this would probably be at the expense of additional manufacturing complexity. The torque required to drive the coupling (overcome seal sliding friction) is estimated at 8655 inch-pounds.

The coupling is basically a rotary shaft. Fluid will enter the coupling through the end of the rotating shaft, flow to annular passages on the shaft outside diameter, and then through tubes in the outer housing. Rotary seals provide the isolation between flow paths at the shaft/housing interface. A radiator/scavenger system is used to collect any gas or liquid which may leak past the seals. Fluid flow characteristics and the heat transfer between circuits in the coupling are summarized in Tables 4.4.8.7.2-1 and 4.4.8.7.2-2 respectively.

Materials proposed for the fabrication of the fluid coupling were selected based on the compatibility and known prior use in ammonia systems. Ammonia is corrosive, in one form or another, to many of the commonly used engineering materials. Ammonia manufacturers and suppliers were contacted for recommendations and a limited search was made of the literature. The metals which have been most commonly used with ammonia are carbon steel and 304 or 316 stainless steel. There are a number of nonmetallic materials which are compatible with ammonia including TFE Teflon. The conceptual design utilized these materials since the time available did not permit a more detailed investigation in this area.

The seals which were evaluated for this application are rotary Teflon seals which contain a support spring. The seals are loaded against the sealing surfaces by both the spring and the internal pressure. A number

TABLE 4.4.8.7.2-1 - SUMMARY OF FLOW CHARACTERISTICS

CIRCUIT NUMBER	TOT HEAT REJECTION	FLOWRATE	PRESSURE DROP THROUGH JOINT		
	kW	lb/hr	psi Liquid Side	Gas Side	Total
1	76	509.9	32.4	.15	32.6
2	6	38.2	3.0	.019	3.0
3	30	265.4	18.9	.086	19.0
4	39	283.7	19.8	.093	19.7

TABLE 4.4.8.7.2-2 - SUMMARY OF HEAT TRANSFER ANALYSIS

CIRCUIT NUMBER	AVG JOINT TEMP °F	FLUID TEMP °F	FLUID COND BTU/ HR SQ IN.°F	CONDUCTANCE PER LENGTH BTU/IN.	CIRCUIT HT GAIN WATTS
1	86	70	2.4169	6.814	60
2	86		.2965	2.682	24
3	86		1.4571	-1.144	-10
4	86		1.5865	-9.034	-80

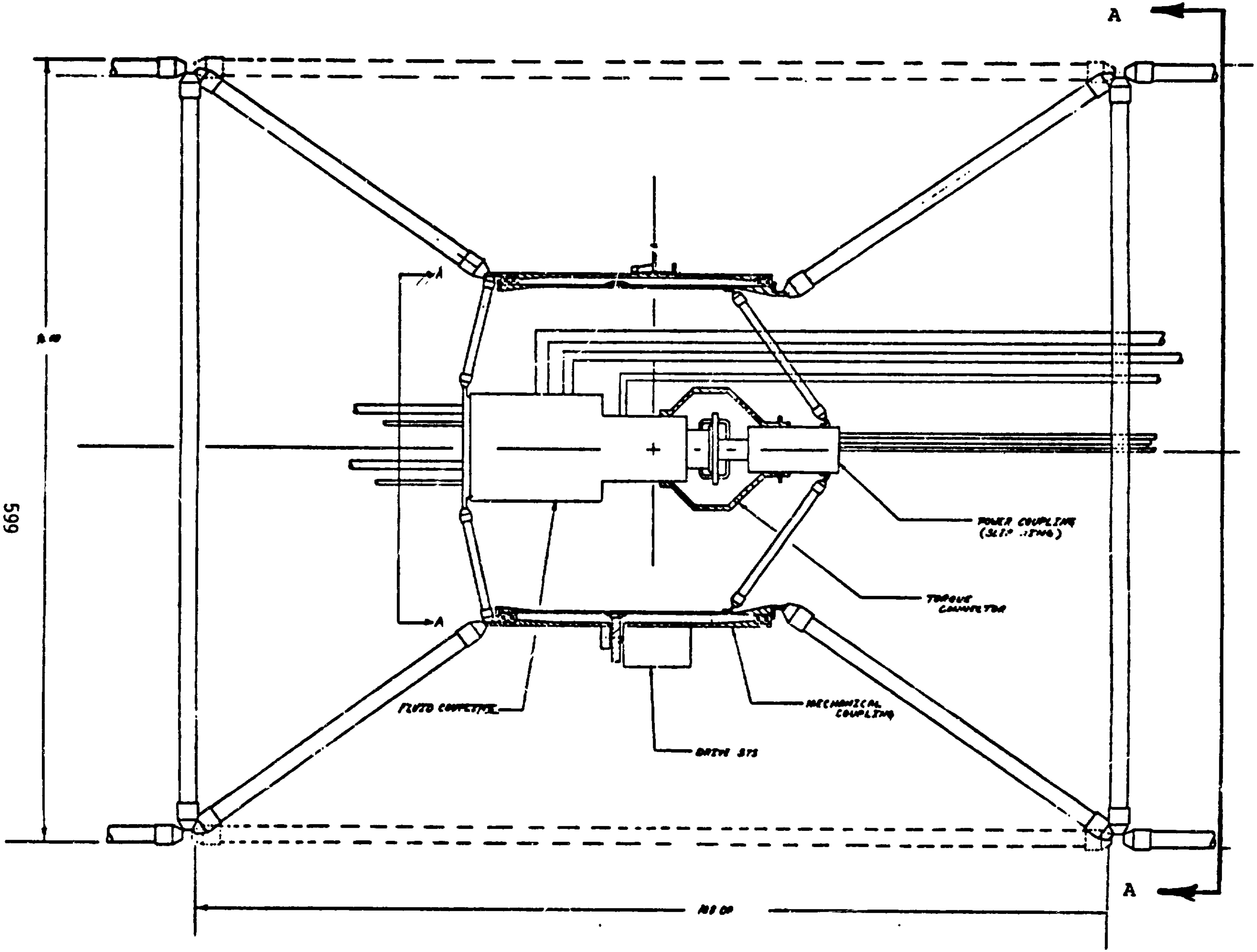


Figure 4.4.8.7 -1a - ROTARY JOINT-SLIP RING CONCEPT

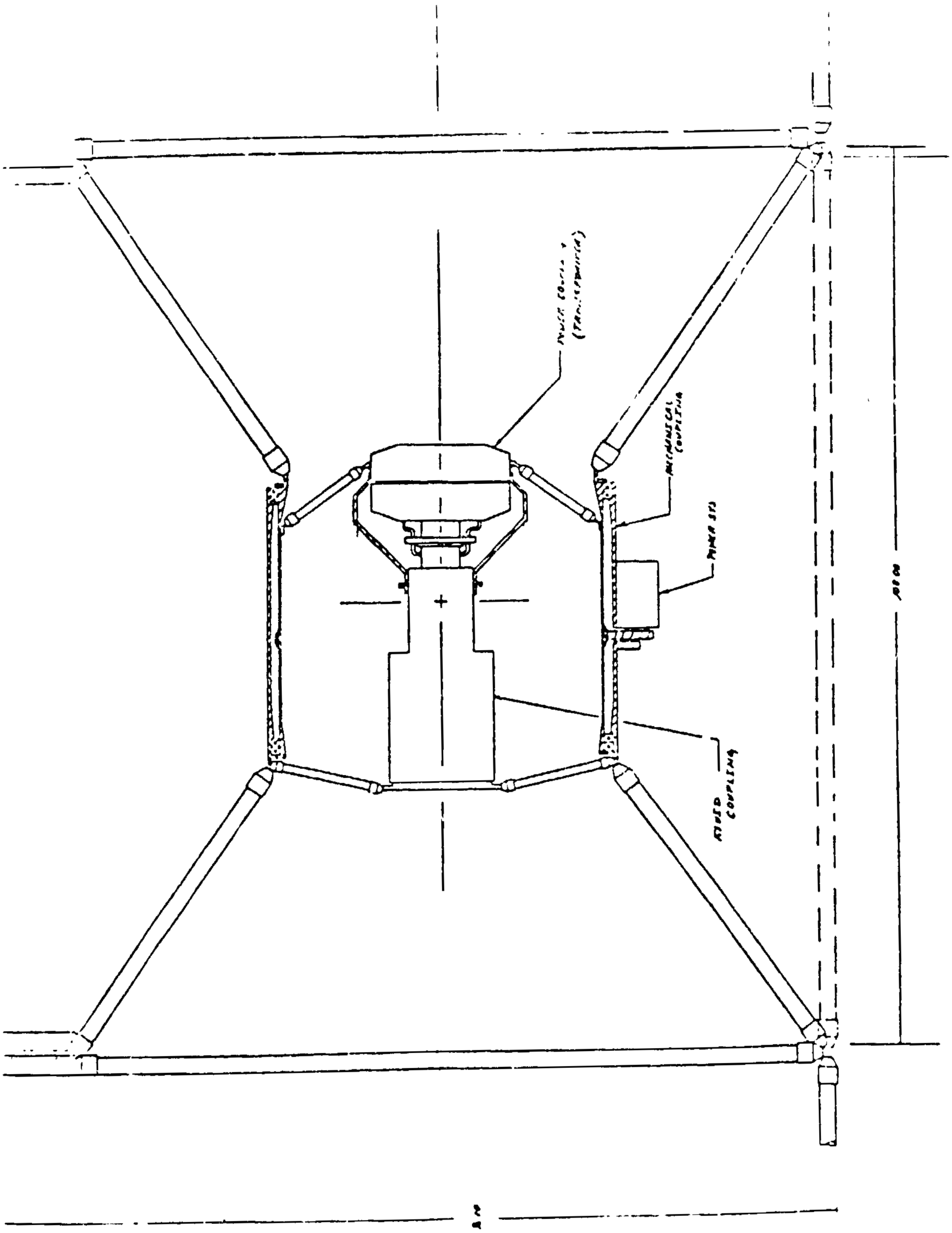


Figure 4.4.8.7 -1b - ROTARY JOINT-INDUCTANCE COUPLING

of seals of this type are available commercially. The following information supplied by the Fluorocarbon Company is typical.

Material	- Virgin TFE Teflon provides compatibility with NH3 and has good sealing characteristics.
Springs	- CRES 316, CRES 304, or Hastelloy springs
Lifetime	- At the rotation rate anticipated, one revolution every 90 minutes, the lifetime of the seals should exceed 10 years.
Leakage	- "Near zero"
Surface Finish	- The surface in contact with the seals should be polished to a 6-8 microinch finish for best results.

The seal configuration which appears most promising has a "C" shaped cross section with a metal spring inside the section. Figure 4.4.8.7.2-1 shows the fluid coupling with 18 seals in place. Each flow passage is isolated with a separate pair of seals. An additional seal is provided at each end of the coupling to reduce the possibility of any ammonia leakage. The radiator/scavenger system will condense any leakage to the outboard seal cavities in the scavenger reservoir.

4.4.8.7.3 Scavenger system

Any dynamic system under pressure will leak some small amount past the seals. A system which would "pump" most of the leakage to a storage tank is relatively simple in concept. A small flow passage would be located between the seals which define the annular flow passages. This passage will be connected to a storage tank outside of the truss supporting structure. The storage tank would be passively maintained at temperature of 0°F or lower by covering the external surfaces with a material such as silvered Teflon. Any ammonia leaking into the area between the seals would be "pumped" to the tank and condensed on the walls. The tank could be sized to accept all expected leakage for the lifetime of the station. Or it could be designed with appropriate valves and a bladder to reclaim the condensed liquid.

4.4.8.7.4 Power coupling

The power transfer techniques used in this design were derived from similar concepts developed by the Space Division of the General Electric Company under contract to NASA Lewis Research Center. The concepts were published as NASA CR-165431 and CR-168021. Since it was beyond the scope of this study to determine the overall configuration of the power distribution system, two basic configurations of the power coupling were used to establish trial layouts for the rotary joint.

4.4.8.7.4.1 Design overview

Other studies have established that power distribution throughout the Space Station can be most economically achieved using (relatively) high voltage AC at 5 to 30 kHz. Transformers for this range of frequencies are lightweight in comparison to those used in the 60 Hz commercial distribution network, and can be made quite efficient by use of modern magnetic materials technology. Solar arrays generate DC power; inverters and transformers are required for an AC system. Integration of rotary transformers into the power distribution system may result in overall power system economies, even though it introduces considerable complication into the design of a rotary joint. Therefore one design examined in this concept development incorporated a rotary transformer as a cylindrical assembly about 17.3 inches in diameter and about 9.2 inches between attachment flanges with an estimated weight of 500 pounds.

The alternative technique which was examined assumed conventional power transmission (insofar as the rotary joint is concerned) via slip rings. Slip rings may transmit either AC or DC, and are not affected (within reasonable limits) by the voltage level being carried or by voltage variations. They may be high maintenance items relative to transformers. Selection of materials and specific design parameters are critical for space applications, particularly with the very slow rotation rates anticipated for this design. They are, however, very efficient, quite light, and have few packaging constraints. The unit examined in this study is adapted from a design referenced in the reports noted above as a cylinder about 5.7 inches in diameter and 11.0 inches between flanges with an estimated weight of 30 pounds.

4.4.8.7.4.2 Configuration

The transformer reported in the CR's noted above, was designed to operate at power levels similar to those required for this study, but at a higher voltage and at 20 kHz. However, other considerations may lead to a frequency of 10 kHz. Since the volume of magnetic material required in a transformer is almost exactly inversely proportional to operating frequency, all other parameters being held constant, the size of the transformer was increased for conservatism from that of the noted study to accommodate the doubled magnetic volume. The transformer of the noted study had redundant windings. Since that was not a requirement of this design, no effort was made to exactly size the winding space. Furthermore it was determined that the excess space could be used to utilize more efficient winding techniques, thereby reducing copper losses.

The exact configuration of the transformer is described in the noted reports and need not be repeated here except in general terms. The unit is of pancake construction, consisting of four independent circular transformer systems, each with a different diameter. These are mounted concentrically on a common axis. The support structure is arranged so that the primary windings and part of the magnetic circuits rotate with one attachment flange. The remainder of the magnetic circuits rotate with the other flange, at the opposite end of the unit major axis. The structure incorporates bearings to locate and maintain alignment of the parts which have relative movement.

The slip ring assembly used in the alternative design was taken directly from the noted study regarding envelope, weight, and form. Details of the internal design were not presented in the original study.

4.4.8.7.4.3 Electrical characteristics

The power transfer device for this study was specified as being capable of passing 100 kW at a frequency of 20 kHz or less, at a voltage of 200 volts or less. The power was to be in four separate circuits. Although not specified, it was assumed that the four circuits would divide the power equally, i.e., each circuit was designed for 25 kW. As stated above, the transformer was sized for the required power at 10 kHz. For constant power, frequency is the only variable which has an appreciable affect on the size and weight of the unit; the winding remains approximately the same size (volume) regardless of the voltage, within the range of probable Space Station distribution voltages. No attempt was made to optimize the windings for a specific voltage or frequency in this design; the allocated volume or the transformer of the noted report was used.

The efficiency of the transformer is expected to be greater than 98 percent. This should be easily achieved, but still will require that 2 kW be dissipated in the transformer. Because of the small volume and radiative surface, supplemental cooling must be provided. In this design, a portion of the cooling capacity being passed through the fluid loop of the rotary joint is used to cool the transformer, versus heatpipe cooling in the transformer of the noted report.

For the design of the alternative slip ring assembly, less than 100 watts will be dissipated in the assembly. Supplemental cooling will probably not be required.

4.4.8.7.5 Mechanical drive coupling

4.4.8.7.5.1 Structural design overview

The structural design of the mechanical drive coupling is basically two concentric structural cylinders maintained in alignment and held together by two sets of preloaded angular contact bearings. Attached to the inner cylinder would be a gear driven by a worm pinion/ gear reducer/ motor drive assembly mounted on the outer cylinder. The fluid coupling and the power coupling are supported from the cylinders using truss elements. The cylinders are, in turn, connected to the Space Station Transverse Power Boom truss using a transition structure which for this study will be assembled with "snap connection" fittings similar to those used for erectable truss members elsewhere on the Space Station structure. The sizing of structural elements was driven by stiffness considerations based on the requirement to maintain the stiffness distribution through the joint.

Table 4.4.8.7.5-1 summarizes the axial, bending, and torsional stiffnesses for trusses transitioning to several diameters and for a 42-inch outside diameter bearing assembly.

The transition truss concept proposed provides a structural tie between the basic 9-foot cubic truss structure and a bearing housing having a nominal diameter of 42 inches. In order to maintain the required stiffness it is necessary to fabricate the 12 members of the truss of a graphite-epoxy composite having an elastic modulus in tension of 40,000,000 psi. The maximum cross-sectional area of each member must be 12.75 square inches. Therefore a tube with a 6 inch OD and a wall thickness of 0.78 inch was selected. This suggests that particular attention must be paid to the design of the joint and the local supporting structure at both ends of these tubes. The weight of a single transition truss assembly is estimated at 680 pounds. The bearing assembly stiffness is governed by bending. The results are based on the use of a preload of 1021 pound-inches on 2 pairs of back to back, Kaydon type A, 42 in OD bearings spaced 32.4 inches apart.

The housing is fabricated of graphite-epoxy composite with its wrapping biased to yield a tensile modulus of 40,000,000 psi and a shear modulus of 10,000,000 psi. The worm gear was mounted with its longitudinal axis perpendicular to the longitudinal axis of the bearing assembly. In doing so, the torsional stiffness of the bearing assembly through the drive mechanism is essentially a function of the EA of the worm gear. Particular attention should be paid to the design and fabrication of the worm gear assembly to minimize backlash and the resulting nonlinear behavior. Weight of the drive housing and bearing assembly is estimated to be 380 pounds.

4.4.8.7.5.2 Motor/drive selection

The internal cylinder of the power couplings is driven relative to the outer cylinder using a worm gear/reducer drive/motor assembly mounted on the outer cylinder (Fig.4.4.8.7.5-1). The worm gear will be mounted between bearing assemblies which are designed to react both thrust and radial loads. The worm gear is driven by a stepper motor though a reducer drive. The total reduction through the system is approximately 2400:1. During rotation, the motor will step about 200 increments per revolution with rotational speed of 26.7 rpm. The torque requirement for the motor is calculated to be 1.357 foot-pounds. The motor sized for this design concept has a capability of 3.33 foot-pounds at the described rates. Motor drive and gear mechanism weight is estimated at 40 pounds.

TABLE 4.4.8.7.5-1 - STRUCTURAL STIFFNESS SUMMARY

TRUSS TRANSITION
2" OD X .1 WALL MEMBERS

CASE	42" OD	48" OD	60" OD	REQUIRED
AXIAL LB/IN.	9.102 E05	9.882 E05	1.162 E06	1.627 E06
BENDING IN.-LB/RAD	1.088 E08	1.636 E08	3.364 E08	3.748 E09
TORSIONAL IN.-LB/RAD	4.212 E08	5.820 E08	9.988 E08	2.039 E07

DRIVE HOUSING

	42 IN. OD BEARING	REQUIRED
AXIAL LB/IN.	5.703 E06	1.952 E06
BENDING IN.-LB/RAD	4.496 E09	4.496 E09
TORSIONAL IN.-LB/RAD	2.447 E06	1.949 E09

4.4.8.7.6 Conclusions

The rotary drive described in this report meets all of the requirements that were defined at the inception of the study. The design of the drives, fluid feedthrough, electrical feedthrough, and structure are all well within the current state of the art and should require only the normal design/development effort for a subsystem of this type.

Several areas were identified that require a more "indepth" design and trade-off study than could be performed in the limited time available for this effort. The most obvious area for additional work is weight reduction. The primary weight offender is the structure which provides the transition from the "square" truss platform to the cylindrical drive housing. The root of the problem is the reduction in truss stiffness as the section is reduced to the 48 inch drive diameter. Three possible areas for study seem evident:

a. Increase the drive housing diameter.

b. Utilize a different type of design for the transition structure.

c. Refine the stiffness requirement used for the joint.

The size of the drive in this design is essentially controlled by the bearing configuration. For this study, the 42-inch diameter bearing was chosen because of its relative "off the shelf" availability and weight. The required stiffness and load carrying capacity are readily met with the size bearings currently in the design. Larger diameter thin section bearings are available but the sections are significantly thicker with correspondingly higher weight, rotating inertia, and torque.

The tabular data presented shows the resulting effective loss in axial and bending stiffness in the transition structure as the diameter is reduced and truss member cross section is kept constant. The final weight values in the design are a result of meeting the stiffness requirements by increasing the transition truss member diameter and wall thickness.

With the truss type of transition design there is also a problem of local deflections at the attachment points between the drive cylinder and the truss. This was recognized as an issue during the study, but not addressed in detail other than making an allowance for additional structure in the weight estimates.

The requirement that the joint maintain the stiffness distribution of the truss was chosen as an initial design limit. The rationale was simply not to reduce the basic stiffness of the overall structure. A trade study examining the effects of such a local stiffness reduction on the affected Space Station systems seems warranted.

Materials for the ammonia transport system are an obvious concern. However, with reasonable care in selection and implementation it should not be a significant problem. Leakage to the outside can be controlled to acceptable levels with a scavenge system. Aside from materials-compatibility, seal life is predominantly a function of rotational speed and abrasion of the seal elements. With the low rotational speed and the use of good finishes (6 to 8 micro-in. rms) on the seal running surface life should be acceptable.

The torque required to overcome system friction is substantial. However, the average power required is small due to the low rotational speed. The friction torque level itself is not of great concern since it is an internal force control system. Of more concern is the issue of rotational smoothness or the tendency of the drive to exhibit a "stick-slip" friction characteristic.

The choice of an electrical feedthrough approach is not driven by the joint design, but lies elsewhere in other Space Station system requirements. Either approach, sliprings or a rotary induction coupling, can be accommodated for power transfer.

4.4.8.8 Deployment Verification

4.4.8.8.1 Introduction

Space Station studies of a decade ago envisioned Space Station as various combinations of modules attached together in clusters. Current requirements for station design are far more extensive and include satellite servicing, large space construction and payload attachments that can only be accommodated by providing large service areas on the station. This is best accomplished by lightweight truss structures. At the present time, virtually every Space Station concept uses large trusses as one of its major components.

These truss structures can be erected on orbit by EVA crew operating in the payload bay vicinity of the Orbiter. The erectable truss is assembled in a strut by strut mode, and is an EVA intensive operation. The truss structure can be a deployment design in lieu of the erectable type. The deployable truss is automatically deployed from the payload bay, with minor EVA assist from crew.

In a deployable truss structure, such as envisioned for the Space Station, verification that the structure can be successfully put in place is of prime importance. In the past, deployable structures, such as were used for the deployment of satellite solar cells, thermal radiators, etc. were designed for high reliability. To obtain the high hardware reliability, historically, it was necessary to overdesign the device and perform an extensive amount of testing and analysis. This led to long development times and high cost. In the construction of the Space Station the trusses are deployed from the payload bay of the Shuttle Orbiter with the assistance of the crew. The presence of the crew during the deployment phase makes this program unlike any previous program in that now we can obtain high Space Station reliability without the need for excessively high hardware reliability. In other words, the crew provides backup procedure if a deployment malfunction were to occur.

4.4.8.8.2 Parameters

4.4.8.8.2.1 Types of structure

Current Space Station concepts include three basic types of deployable truss structure:

a. Tetrahedral, planar truss

b. Box beam

c. Continuous coilable longeron mast (CCLM)

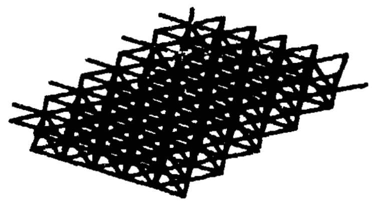
The tetrahedral truss design includes hinged and folding members that enable compact packaging in the payload bay. The box beam design includes hinged, folding, and telescoping members for compact packaging. It may also include a deployment mechanism that is a permanent and load-bearing part of the beam after deployment. The CCLM design includes coiled longerons, hinged struts, and cable diagonals. It also includes a deployment canister that is part of the load path after deployment. See Figure 4.4.8.8-1 for views of truss structures.

The tetrahedral truss is a planar truss. It follows that the deployment process involves two dimensional change in size, in width and length. The box beam and CCLM deployment undergo only a length change only during deployment, while their cross-section remains constant.

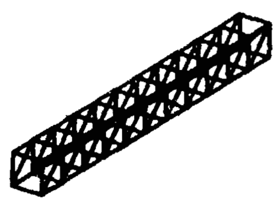
4.4.8.8.2.2 Deployment means

The truss deployment process requires energy for moving the members in position and to operate the mechanisms that lock the truss into the deployed state. The energy source may be from stored energy in springs integral to each joint of the deployable truss. An alternate energy source is an electric motor driven mechanical drive. The tetrahedral truss can be deployed only by means of stored energy within the joints. The box beam deployment can be of the mechanical drive or the stored energy type design. However, the deployment reliability of telescoping members with stored energy joints is not well established. The CCLM can be deployed by means of a mechanical drive type energy source only.

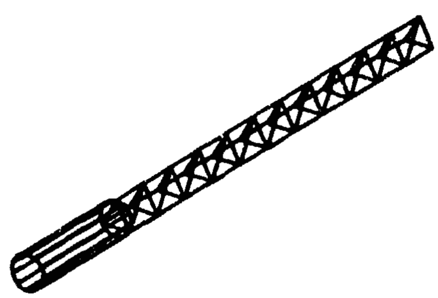
The ideal deployment process would provide sufficient energy to each joint to affect deployment and proper lockout of the joint, with a minimum amount of residual energy left over after deployment. This can be more easily accomplished with a stored energy type system where the required deployment energy is contained within each joint. This affords an even distribution of deployment forces within the structure. The mechanical drive type energy source transfers energy to the deploying



TETRAHEDRAL



BOX BEAM



CONTINUOUS COILABLE LONGERON MAST (CCLM)

FIG. 4.4.8.8.1 VIEWS OF DEPLOYABLE TRUSSES beam at a definite number of points. The dynamic load distribution within the deploying beam is always a function of the stiffness and mass distribution of the partially deployed portion of the structure. The share of the deployment energy that each deploying joint receives is a function of a load path that behaves in a nonlinear mode and varies in stiffness as a function of time. This inevitably results in uneven deployment.

The deployment process for a deployable structure may be controlled in order to slow deployment rate, and thereby minimizing dynamic effects of deployment. In a stored energy system, this can be accomplished with simple tether-type mechanisms that are unreeled at a controlled rate. In a mechanical drive system adequate control of the deployment process is a function of available information about the load distribution within the deploying structure that may be used as feedback information to the control device. A closed-loop control system would be difficult to design due to the inherent limitations of being able to obtain feedback data. An open-loop type deployment control of a mechanical drive is inherently unreliable, because it can overload and fail truss joints without any indication (acquired data or visual observation) that it is doing so until the failure occurs.

A unique problem in truss deployment occurs when testing is attempted under one-g loading on the members. The energy that is required to activate each member varies as a function of orientation of the member with respect to local vertical direction. Members that open "down" are assisted by gravity. Members that open "up" are retarded by the weight of the members. For certain truss orientations a large portion of the truss is lifted, i.e., energy is needed above and beyond that of normal deployment. Thus, ground testing of these structures is a difficult task.

4.4.8.8.2.3 Verification rationale

The goal of the verification activities is to assure with maximum reliability that the truss will successfully deploy on orbit. The process of deployment certification is based on a logical combination of tests, analysis, and deployment backup procedures by the crew.

The requirements for analysis and ground test activities are significantly impacted by the planned use of on orbit EVA backup procedures by the crew. The availab ty of such backup procedures is a function of the design of a particular deployable truss. If the truss members and joints are designed so that crew EVA activity with the use of on-orbit equipment can assure full deployment, then the requirements for certification data from ground testing and analysis are considerably reduced.

4.4.8.8.3 Verification procedures

4.4.8.8.3.1 Deployment analysis

The analysis which must be performed to complement the deployment tests consists of kinematic/dynamics, stress, and deployment failure mode. Kinematic/dynamic deployment analysis studies the relative motion and internal loads of each component as the entire structure deploys. This analysis must consider the various node and joint tolerances as well as the usual bending, torsional, and external stiffness. Inertia load and most importantly, the friction loads need to be included to insure that the structure have a positive deployment margin during all phases. Special emphasis should be placed on the final locking phases of deployment. The subsequent preloading of the joints (where required) should be thoroughly studied to guarantee that all joints are sufficiently preloaded so as to remain in compression (that is, clearances are all in one direction at a joint or node) under the highest expected operational loads.

The stress analysis as usual is used to determine the adequacy of the components to withstand the maximum-induced loads.

The deployment failure analysis is a very detailed analysis aimed at determining all possible modes of failure. These failure analyses should include mechanism jamming due to higher than expected friction loads and the inadvertent presence of foreign objects. The analysis should include the loads/stress included in the members as a result of the mechanism jamming as well as any subsequent failures. A further study should include a loads and stress analysis of the application of contingency devices such as small jacks, strut replacement aids, etc.

The analysis activity will vary in difficulty according to the type of energy source that is used for deployment. In the case of stored energy systems the analytical model will be more amenable to simplifying assumptions since the distribution of deployment energy within the model is known. On the other hand, for the mechanically powered system special consideration will be given to the nonlinear behavior of the deploying truss in the dynamic modeling. Correlation of analysis results with ground test data will be extensive in either case.

4.4.8.8.3.2 Deployment testing

4.4.8.8.3.2.1 Testing in one-g environment

Deployment verification tests for deployable trusses may be performed on Earth. However, the deployment tests performed in a one-G environment offer formidable challenges. The deployment mechanisms for any structure (tetratruss, box beam, etc.) must provide deployment energies far in excess of those required for zero-G deployment on orbit. It is further complicated by the fact, that the required deployment energy for each joint will vary as a function of member orientation with respect to local vertical. Ground tests will have to be designed such that gravity effects on deployment are minimized. Due to the sizes of the deployable trusses under consideration the test activity will require large facilities.

There are several options for ground testing of deployable trusses. One method involves the suspension of the truss from long cables and allowing deployment to take place in the horizonal plane. This method is limited in size by the height of existing facilities. The length of the suspension cables should be long enough to minimize horizontal loads from the suspension system. A second method involves the use of rollers at each node and deploying the truss on a large flat surface, such as a parking lot. The results of this test may not be viable if the casters offer excessive resistance to truss nodal joint translation. A third method involves the flotation of the truss in a sufficiently large body of water. This may be accomplished by varying the effective density of each truss member by filling it with foam plastic to the point where the truss would be neutrally buoyant. The truss then would be submerged to a sufficient depth and the truss deployed. This approach would require the fabrication of a test article that would not corrode or otherwise malfunction in a water environment. A fourth method of testing could be various drop tests. However, this would most likely be limited to smaller test articles and may very well result in the destruction of the test article.

4.4.8.8.3.2.2 On-orbit tests

For any type of deployable structure, consideration should be given to on orbit testing for verification of truss deployment. Testing of this type would offer the exact environment in which the Space Station will be deployed. The trusses are designed for deployment from the payload bay; thus, on-orbit test hardware and deployment aids would be identical to the actual Space Station hardware. A demonstrated successful deployment would provide complete certification of hardware deployment aids and crew procedures. In order to minimize costs, the experiment could be manifested on a space available basis.

4.4.8.8.3.2.3 Box beam deployment

The reference Space Station concept has a 9 feet square deployable box beam for the keel structure of the station. Figure 4.4.8.8-2 shows the station concept and associated box beam structure.

The box beam deployment schematic is shown in Figure 4.4.8.8-3. The beam is deployed a bay at a time in a sequential fashion.

The box beam could be tested in orbit for deployment reliability in the same fashion as the tetrahedral truss. Such test would be conducted under the actual operating environment and would provide an adequate verification of deployment. The box beam is currently envisioned without a retraction feature. For an on-orbit test, either a retraction means or some form of controlled deorbit would be required.

Ground testing of the box beam would have limitations quite similar to the tetrahedral truss. Gravity effects would increase the required deployment energies for the strut joints. To minimize the gravity effects tests would have to be conducted where the beam is suspended from cables or on rollers on a flat surface. Figure 4.4.8.8-4 shows a schematic of such test arrangements. Due to the long length of the keel beam it would be difficult to find a tall enough structure that could serve as the tower to suspend the beam for the cable-supported deployment test. The test option with the rollers would be difficult to accomplish due to the amount of rolling resistance as a larger and larger portion of the beam was deployed.

The box beam could well be tested underwater with neutral buoyancy flotation means. The design of the deployment motors and associated electrical devices for the test would have to accommodate submersion in water.

4.4.8.8.3.3 Deployment backup

It is the incorporation of crew backup into the deployment reliability which can render extensive cost savings. Crew backup to deployment simply means that if during the automatic deployment of the Space Station structural element a malfunction were to occur, a crew member with small tools could fix the malfunctioning component and thus result in totally successful deployed structures. The cost savings mentioned earlier will be realized in a greatly decreased testing and design program. In essence, this philosophy can accept less hardware reliability because of the additional crew supplied reliability.

Since the Space Shuttle Orbiter has repeatedly proven itself as a "spacetruck", the aerospace community must begin to treat the payloads (especially deployable structure) as <u>cargo</u> not as another space vehicle.

The crew, who provides the additional backup in case of a malfunction, must have the tools onboard to fix any deployment problem. As the most probable failure modes are identified and tools would then be designed so that these failures can be fixed.

Figure 4.4.8.8-5 shows a worse case jammed tetratruss joint due to a foreign matter obstruction. Also shown is a crew member performing Manned Maneuvering Unit (MMU) fly around inspection. Figure 4.4.8.8-6 reveals the details of the deployment jack components. The crew member is shown (in Fig. 4.4.8.8-7) jacking down the jammed strut after the tool has been attached to the truss cell. In figure 4.4.8.8-8 the crew member is shown removing the obstruction while the jack tool retains the strut. In Figure 4.4.8.8-9 the crew member is shown jacking the strut to its full deployed position. Tools of this kind with simple modification can be utilized to remove and replace a broken strut, hinge on even a node. Similar tools can be designed to accommodate any failures of a deployable truss as long as the deployable structure had been originally designed to be repaired, such as by providing for removable struts, removable joints, etc.

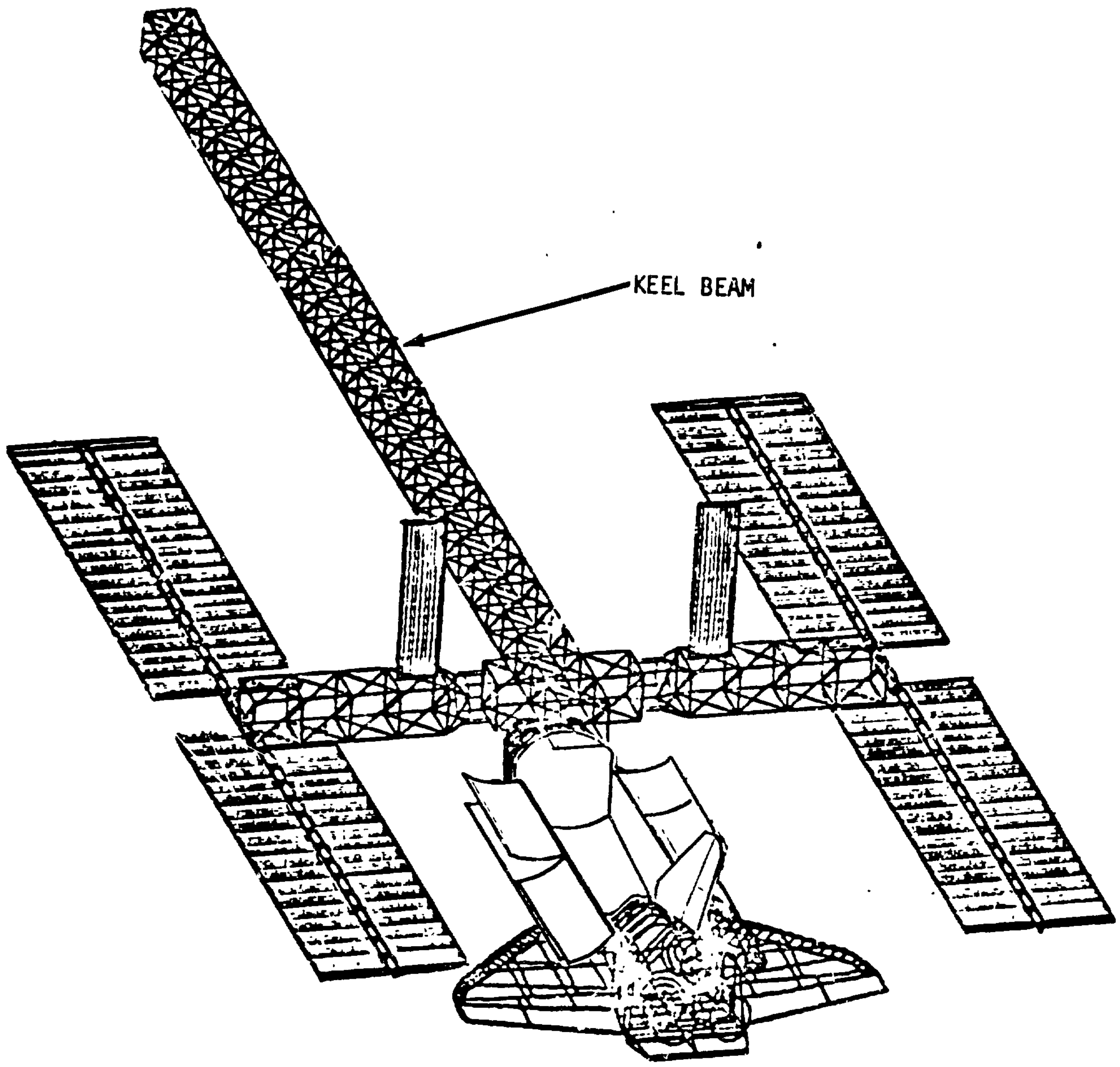


Figure 4.4.8.8-2 - POWER TOWER WITH KEEL BEAM

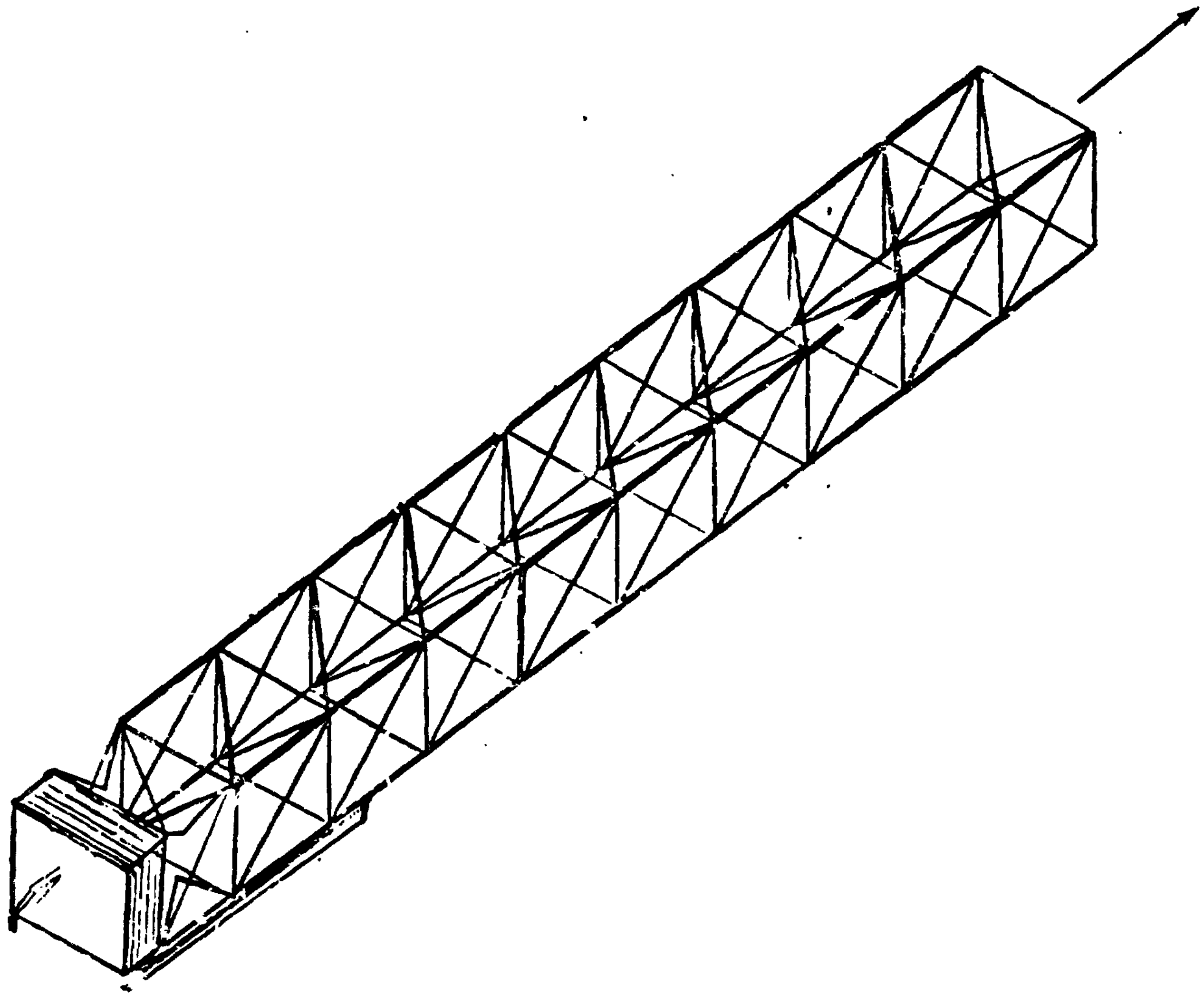


Figure 4.4.8.8-3 - SCHEMATIC OF BOX BEAM BEING DEPLOYED

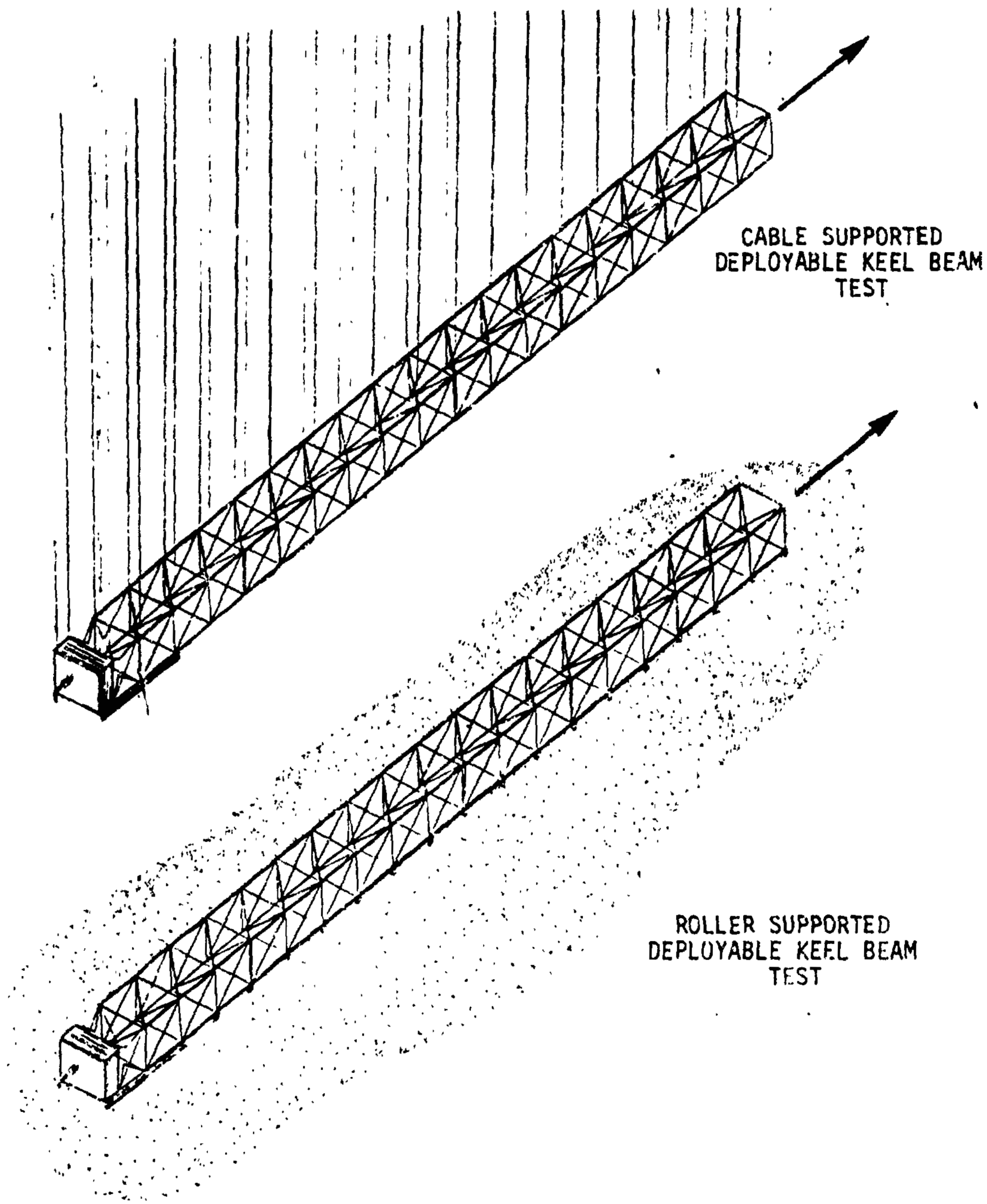


Figure 4.4.8.8-4 - BOX BEAM GROUND TESTS

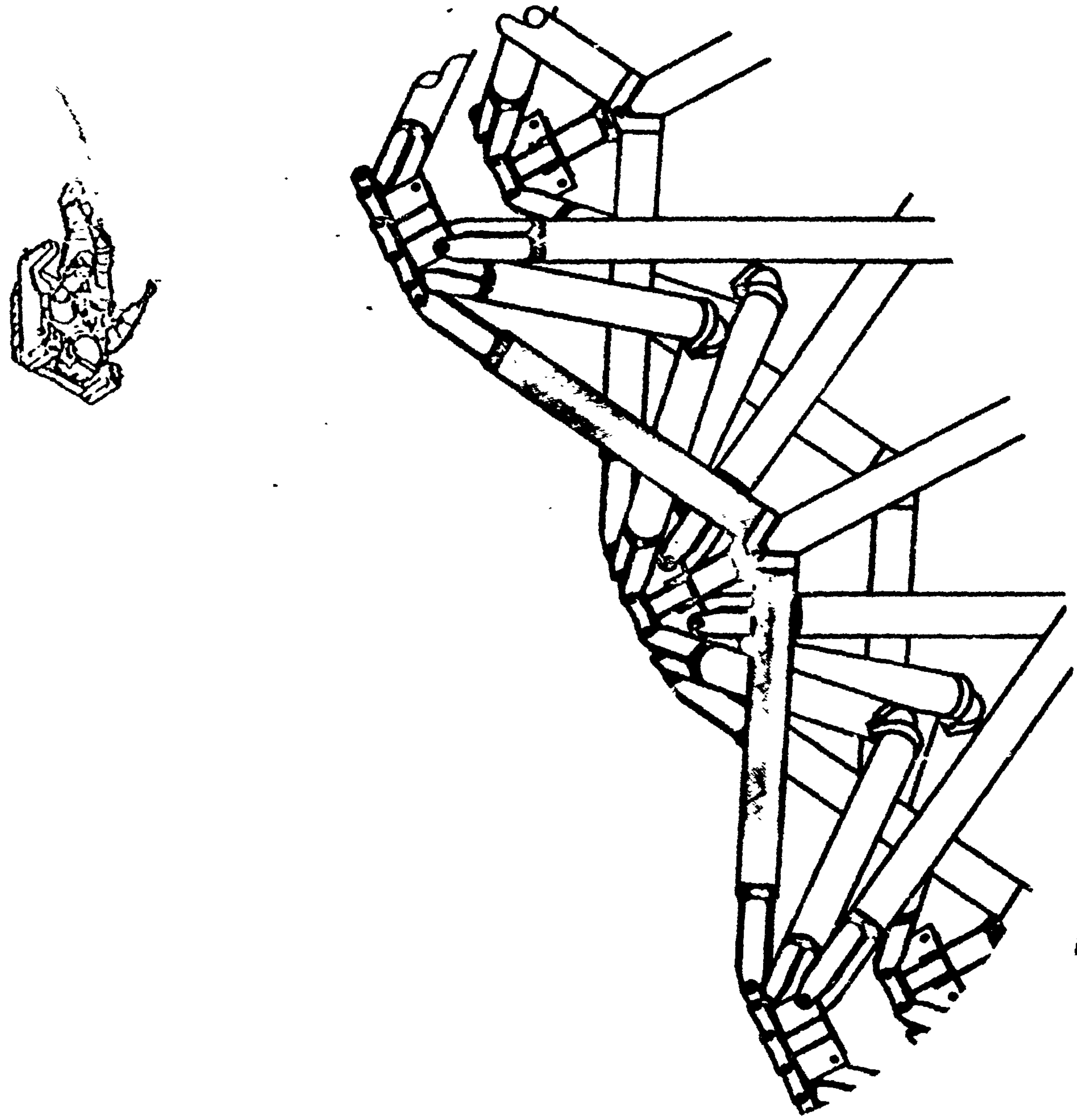


Figure 4.4.8.8-5 - MMU FLY-AROUND INSPECTION

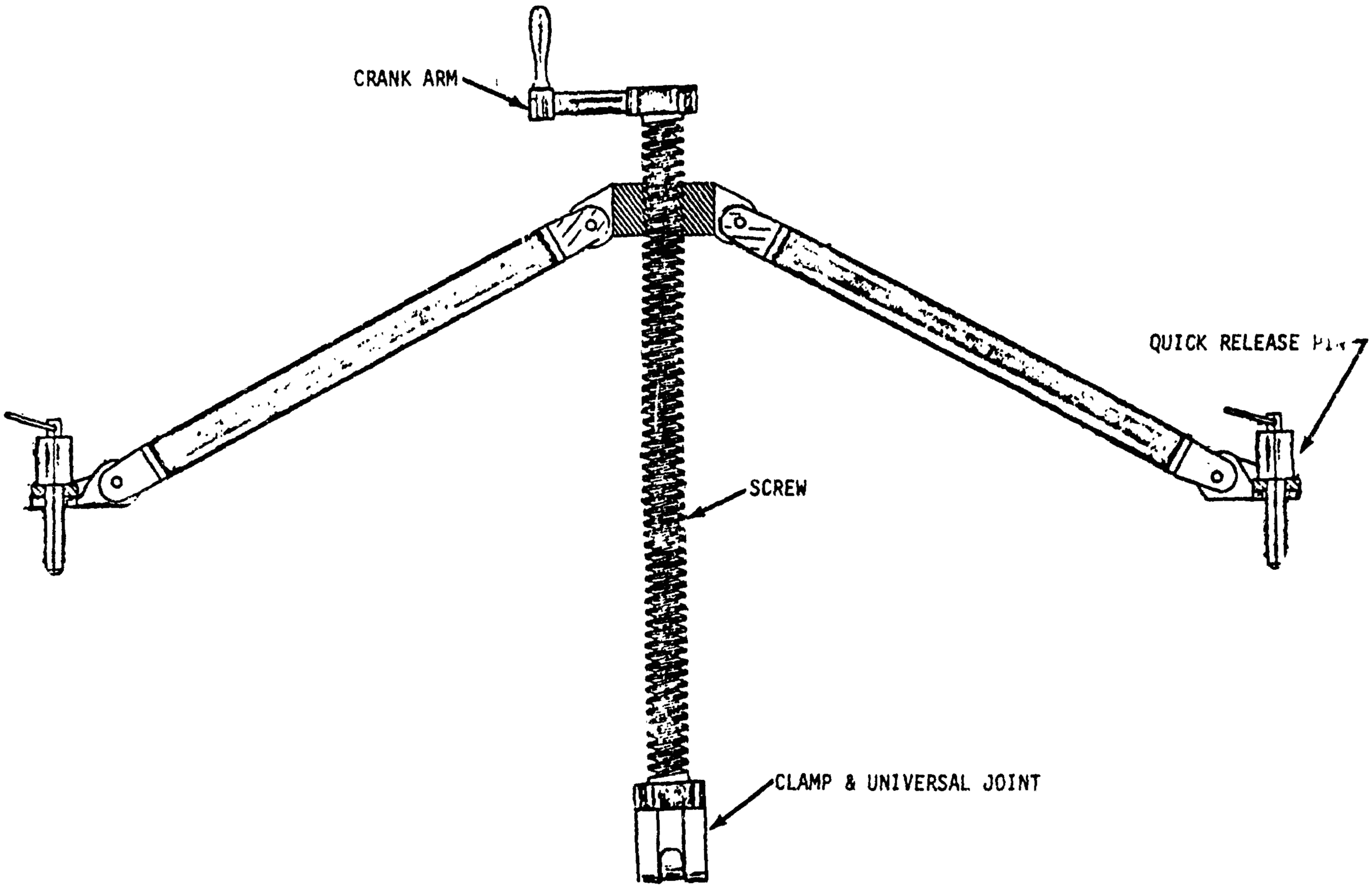


Figure 4.4.8.8-6 - DEPLOYMENT JACK COMPONENTS

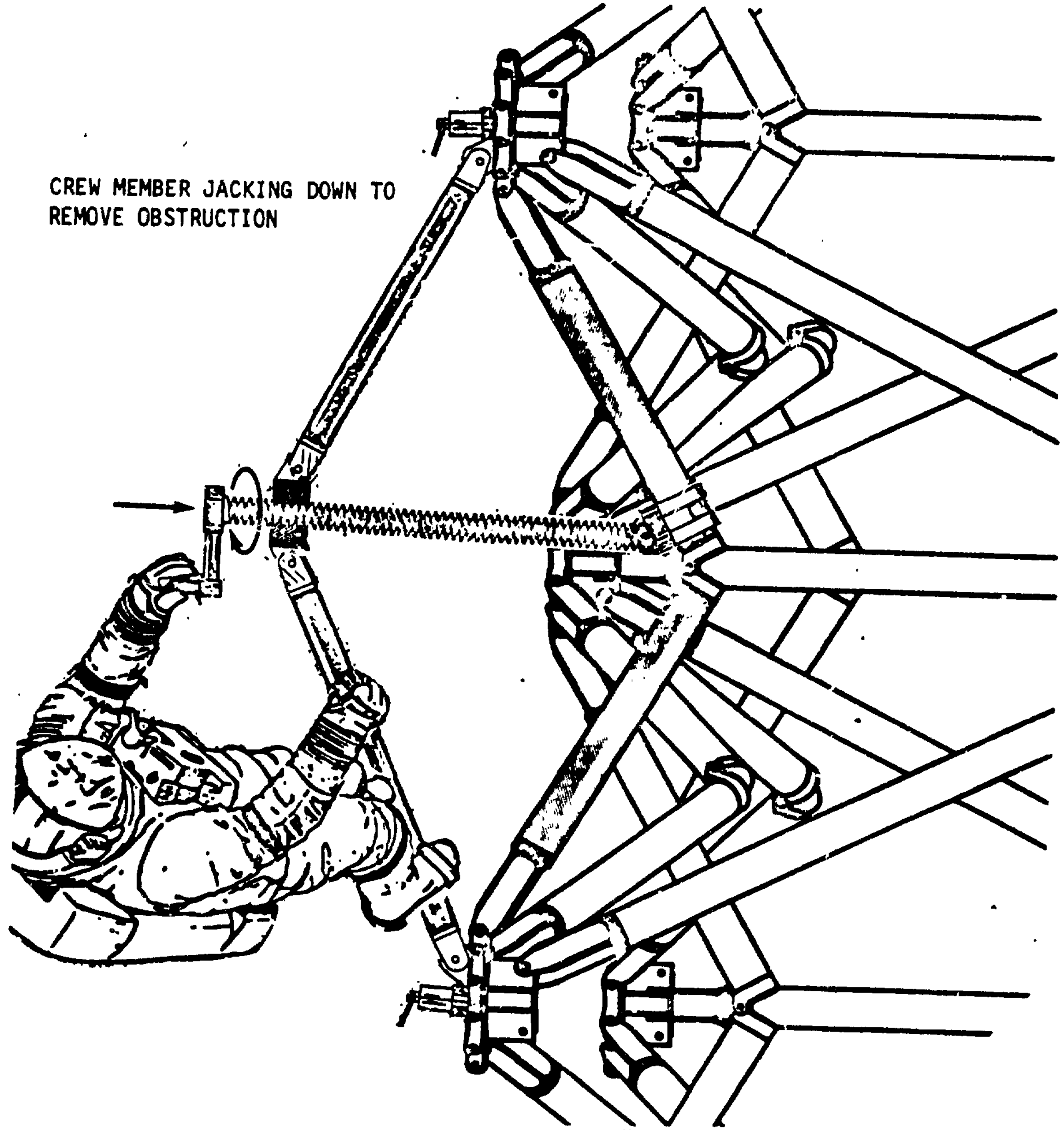


Figure 4.4.8.8-7 - ON ORBIT USE OF JACK

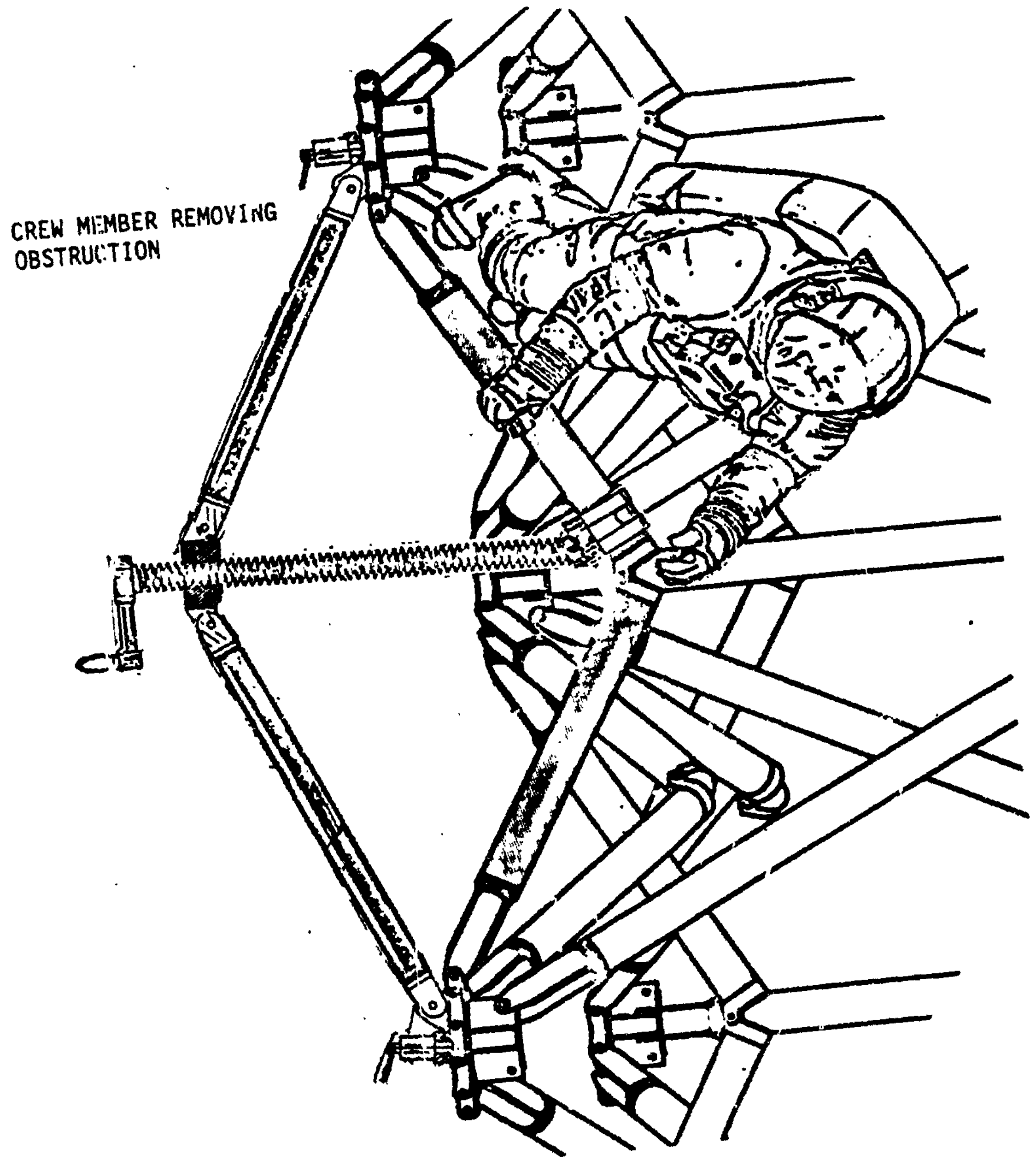


Figure 4.4.8.8-8 - ON ORBIT USE OF JACK

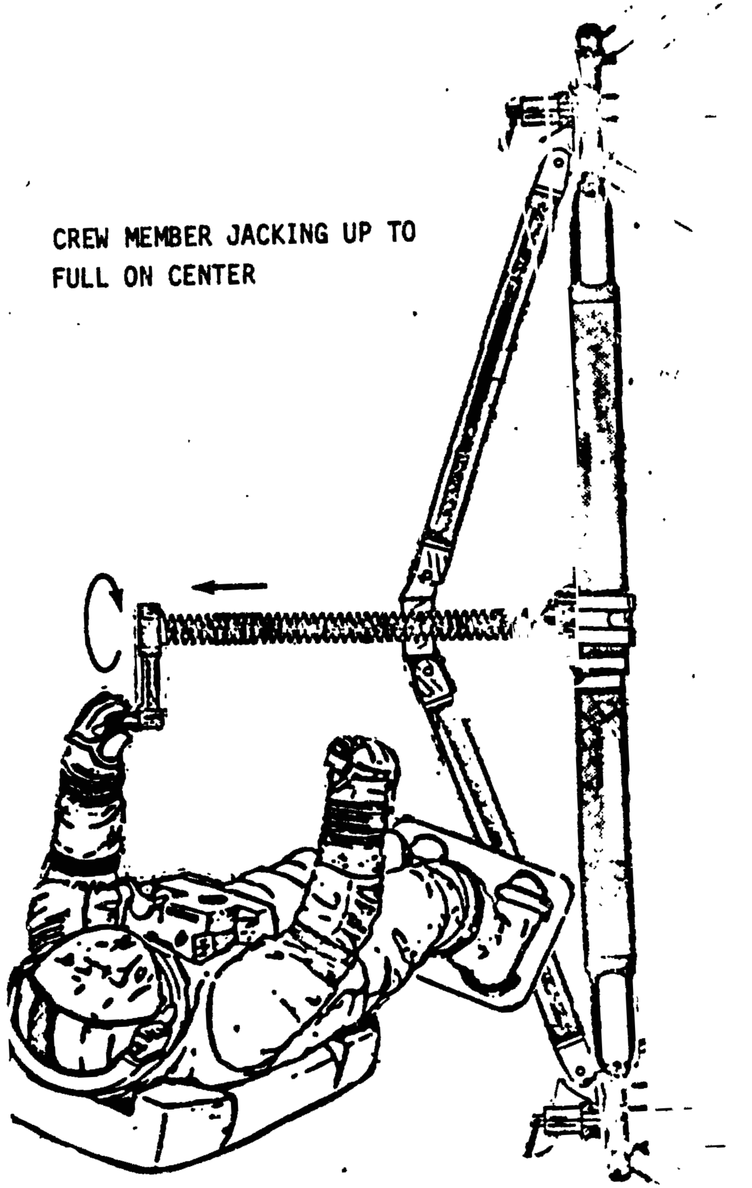


Figure 4.4.8.8-9 - ON ORBIT USE OF JACK

4.4.8.9 Conceptual Design of a Mobile Remote Manipulator System

Space Station studies have identified the need for a Mobile Remote Manipulator System (MRMS). Such a logistics or utility device is envisioned to be outfitted with a spacecrane capability (i.e., Shuttle RMS) and astronaut-foot restraint arms. The system is required during initial Station construction activities, to position astronauts for EVA functions, to transport modules and/or payloads from the Shuttle cargo bay and position them for attachment to the truss structure. Subsequent to the initial Space Station construction activities, a MRMS is considered necessary for maintenance or repair activities and to provide a construction capability for future station growth or large spacecraft assembly and servicing.

4.4.8.9.1 <u>MRMS mobility requirements</u>

The square bay truss structure of the Space Station configurations shown in reference 1 suggests the need for an MRMS which can move in two orthogonal directions. This capability permits movement (1) along the Space Station keel structure between the modules and the solar array support structure, and (2) perpendicularly along the solar array support booms. An MRMS with only undirectional mobility theoretically could be rail-mounted to accomplish this function, but would probably need to be detached and reattached to additional orthogonal rails to move in a perpendicular direction. This is an operational drawback which probably should be avoided. Mounting rails onto the Space Station truss structure introduces additional mass and significant design complexity which also must be considered.

The lower mass alternative of positioning rails on the MRMS (instead of the truss) which "ride" on the truss hard points is possible. However, endless tracks (chains or belts) which provide mobility in this case must completely span two truss bays to ensure stability of the MRMS during motion. Such an arrangement avoids the increased mass and complexity of rails attached to the truss structure and provides for a "smooth", continuous unidirectional motion capability. However, movement in a perpendicular direction is not simply accomplished and the undesirable feature of an MRMS which must be bays in length is introduced. A 2-bay-long MRMS presents Shuttle packaging problems. It also and degrades maneuverability and, therefore, usefulness for maintenance and construction activities, particularly in close proximity to the modular habitats or surface-attached equipment. An MRMS for movement on a triangular gridwork truss is presented in the structures and mechanism White Paper entitled "Space Station Truss Structures and Construction Procedures". A robotic walker "spider" conceptually could serve as an MRMS and accomplish the necessary functions, but would require extensive development of a device which is not considered state-of-the-art.

4.4.8.9.2 Reference MRMS design

A conceptual design for a bidirectional MRMS which is only 1-bay square and avoids a truss-mounted rail system is illustrated in Figure 4.4.8.9-1. This design, which is a modification of a 2-bay long device, consists of three basic elements or layers. The bottom or track layer consists of a square track arrangement which rides on structural guide pins attached to the truss nodes. The four tracks are arranged in a single plane and connected at the corners of "switches" which can be aligned to permit motion over the guide pins in either of two orthogonal directions (see Fig. 4.4.8.9-2). The track layer does not rotate relative to the truss structure. The four corner switches rotate 90°, but only when centered over the guide pins.

The central element consists of a push/pull drive mechanism which also has 360° rotation capability (see Fig. 4.4.8.9-3). This feature permits platform movement in four directions by either a push or pull operation and greatly enhances maneuverability without requiring additional structure for translation. It also permits changing movement direction without rotating the logistics platform or attached payloads. The push/pull motion is envisioned to be powered by an electric stepper motor through a rack and pinion drive. Mounted on the drawbar ends are "drive" rods which are aligned with, and electrically inserted into, the nodal guide pins and locked. The switches are aligned appropriately and the drawbar is actuated to push or pull the MRMS in the desired direction. The drawbar extends to span a complete bay, such that four-point support of the MRMS is maintained at all times. Translation of the MRMS is accomplished by operation of the push/pull drive mechanism to move the platform longitudinally in an "inch-worm" fashion. This sequence of events is illustrated in the upper half of Figure 4.4.8.9-4. Transerse translation involves use of the pivoting as well as push/pull feature of the mechanism, and is illustrated in the lower half of Figure 4.4.8.9-4. Sketch (A) shows the MRMS pivoting 90° from the direction of travel. Sketch (B) shows a translation to construct an adjacent truss cell which is in the next row. Sketch (C) shows the MRMS sliding onto the cell just constructed. Sketch (D) shows a 90° rotation into a position parallel to the original, but on an adjacent row. Sketches (E) and (F) show longitudinal motion and construction of added cells to complete a platform. The corner switch illustrated in Figure 4.4.8.9-5 corner switch illustrated in Figure 4.4.8.9-5 shows the open top mechanism feature which permits the drawbar to lock onto a guide pin which is also occupied by a track switch.

The top element of the MRMS is the logistics platform which is envisioned to rotate with respect to either the track or drive elements (see Fig. 4.4.8.9-5). This platform would serve to transport payloads and cargo over the Space Station surface. A central feature of this element would be the capability to operate a transposed Shuttle RMS, which is shown in Figure 4.4.8.9-6 mounted on a moving carriage. Also shown in Figure 4.4.8.9-6 are Mobile Foot Restraint (MFR) positioning arms. Pressure-suited astronauts attached to the MFR's are positioned within their work envelope by the movable positioning arms. The MFR arms should not be considered to be miniature versions of the Shuttle RMS, with its precision control requirements. Rather, each MFR arm should be controllable by the astronaut who can adjust its position in a manner similar to a utility serviceman operating a "cherry picker" bucket. The degrees of freedom required by the MFR arms are determined by the extend to which EVA is utilized to perform various future Space Station functions.

The MRMS should have a self-contained, rechargeable power supply which does not require umbilicals or power rails. Control of all features of the MRMS should reside with the EVA astronaut(s) to avoid hardline or RF control links to a central station. Transport cradles or similar devices must be provided to support payloads being moved about the Space Station surface by the MRMS.

4.4.8.9.3 MRMS plane change concept

The proposed MRMS can operate over both the "top" or "bottom" surfaces of the reference configuration, if required. It uses the rotary joints and solar array boom rotation as a turntable to translate between the two parallel surfaces. For generality and versatility, however, it is desirable to operate the MRMS in a plane which is perpendicular to these parallel surfaces. Two concepts for rotating the operational plane of the MRMS 90° from its original position are shown in Figures 4.4.8.9-7 and 4.4.8.9-8. The concept shown in Figure 4.4.8.9-7 uses a tilting frame approach to rotate the MRMS 90° and enable translation and operation onto a perpendicular plane. The tilting mechanism is envisioned to be self-contained and installed as a truss cell unit into a beam or along a platform edge. This unit should be capable of rotating 90° in both the "left" and "right" hand directions.

A second approach is illustrated in Figure 4.4.8.9-8 which is operationally more complex, but mechanically simpler and probably more compact than that of Figure 4.4.8.9-7. This concept consists of a planar guide pin frame which is attached to the original structure at the center of an element which replaces a truss strut. A center section of this element contains a "T" fitting which has two perpendicular rotational degrees of freedom. Operationally, the MRMS translates laterally onto the attached guide pin frame. The MRMS and frame are then rotated 180° around the frame centerline attachment as shown in Figure 4.4.8.9-8 into an "upside down position". The MRMS and frame are then rotated 90° around the strut element centerline to a position which permits the MRMS to translate onto the plane which is perpendicular to its original operational plane.

Two devices such as those just discussed or another appropriate design, placed on opposite faces of the truss structure would permit rapid and convenient translation of the MRMS between "top" and "bottom" surfaces of the station without interrupting the rotation of the solar wing for this purpose.

4.4.8.9.4 Concluding remarks

A conceptual design for a Mobile Remote Manipulator System has been presented. This concept does not require continuous rails for mobility (only guide pins at truss hardpoints) and is very compact, being only 1-bay square. The MRMS proposed is highly maneuverable being able to move in any direction along the orthogonal guide pin array under complete control. The proposed concept would greatly enhance the safety and operational capabilities of astronauts performing EVA functions, such as structural assembly, payload transport and attachment, Space Station maintenance, repair or modification, and future spacecraft construction as servicing.

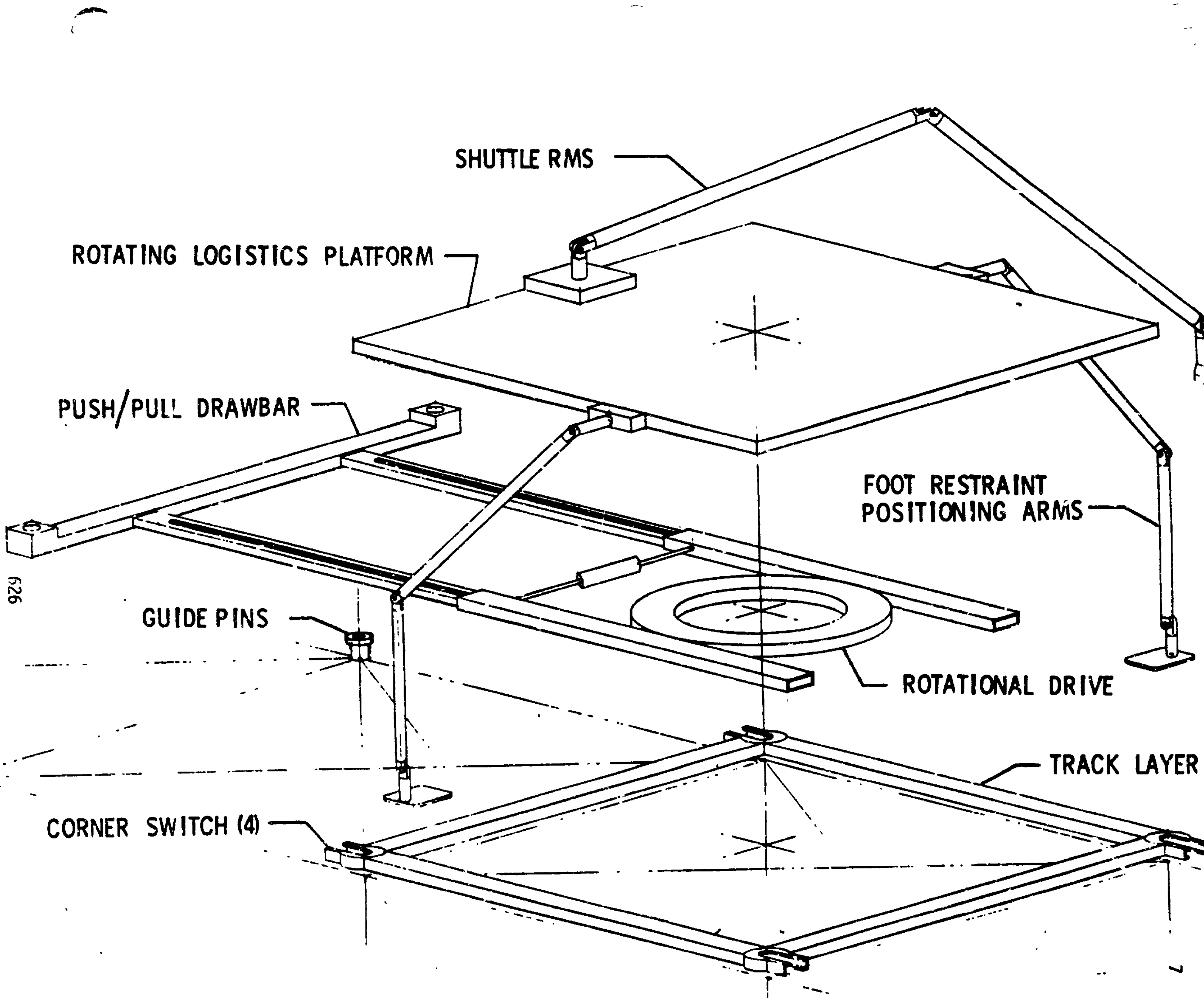


Figure 4.4.8.9-1 - MOBILE REMOTE MANIPULATOR SYSTEM ELEMENTS (EXPLODED VIEW)

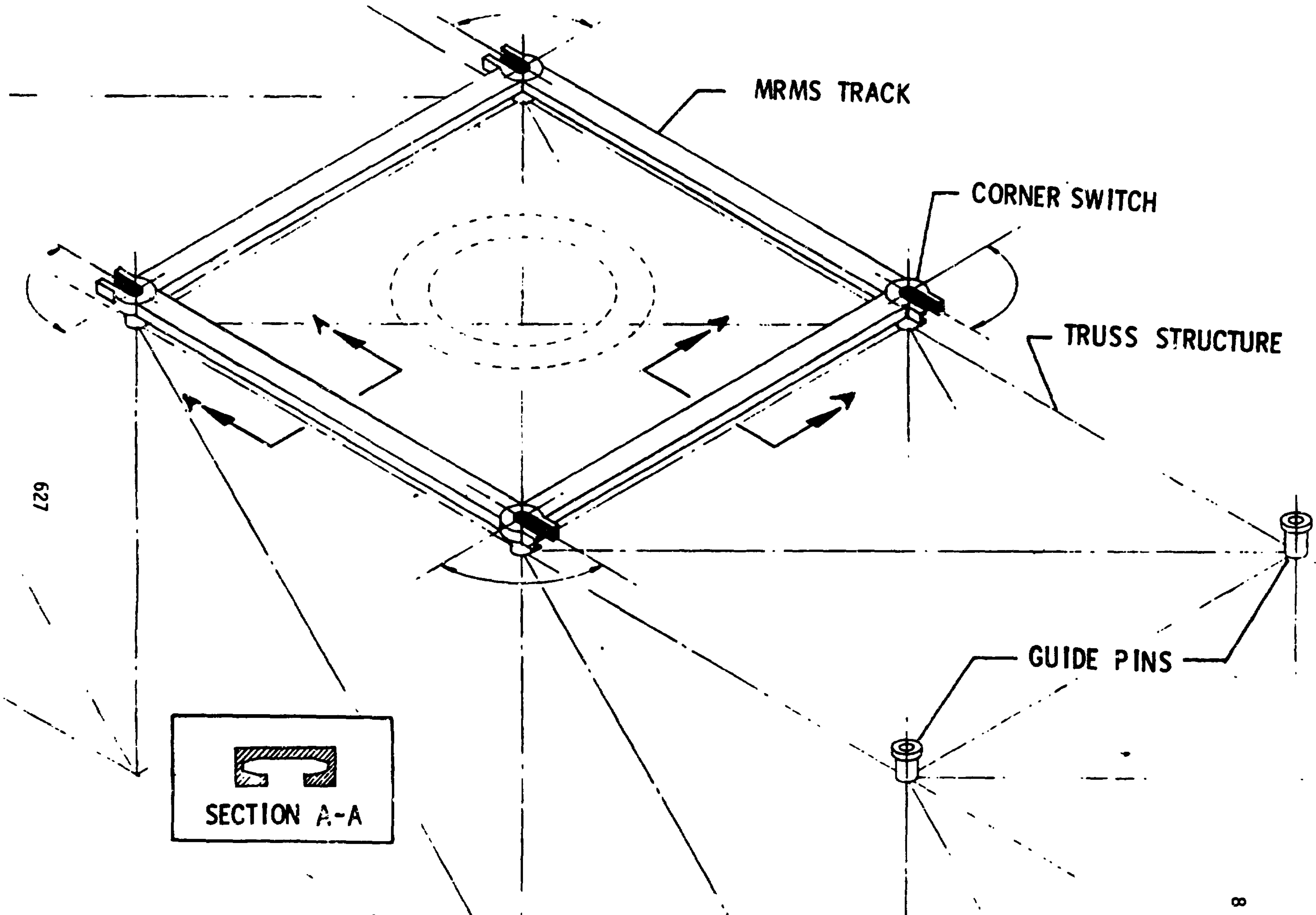


Figure 4.4.8.9-2 - TRACK AND CORNER SWITCH ARRANGEMENT

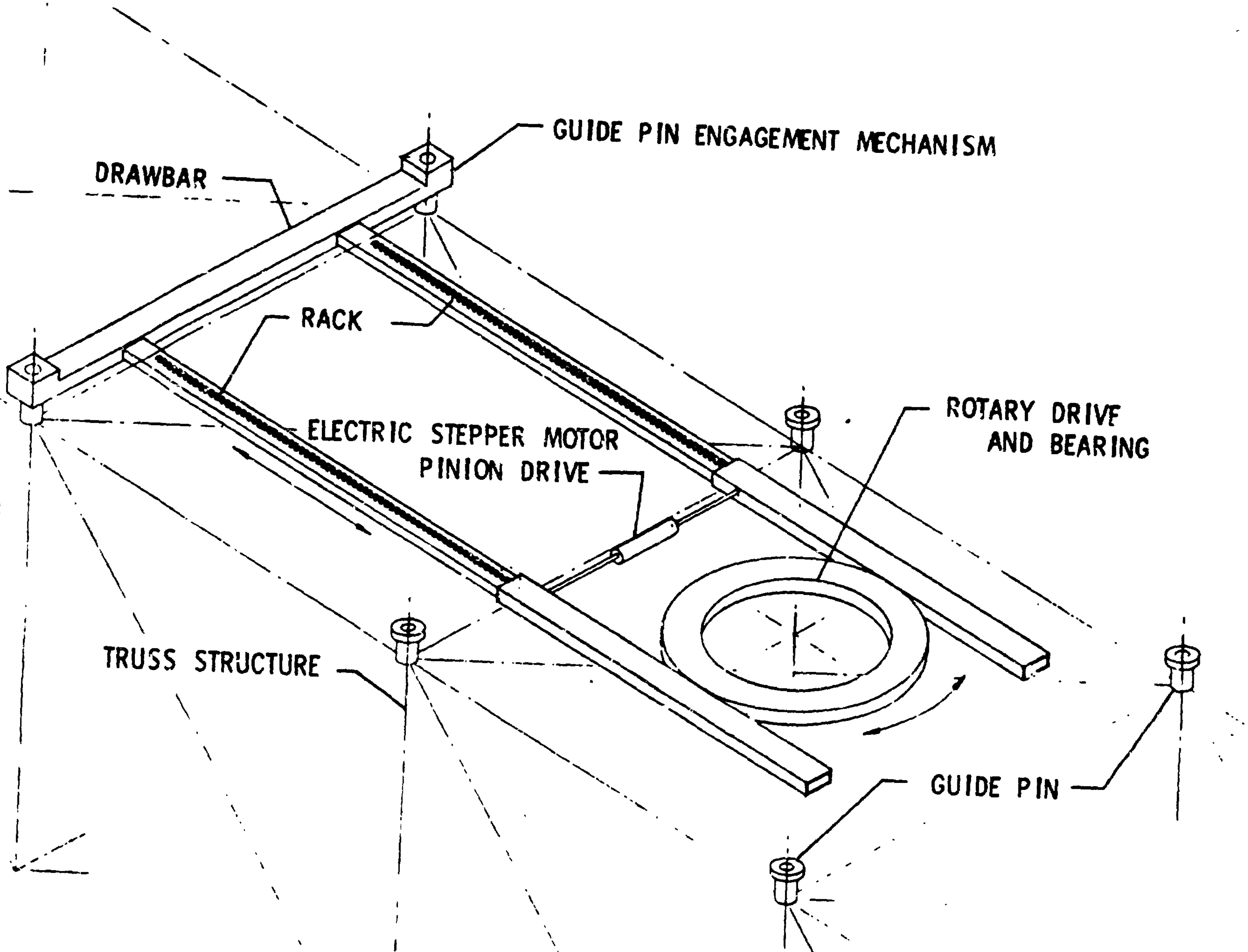


Figure 4.4.8.9-3 - PUSH/PULL DRIVE SYSTEM ARRANGEMENT

9

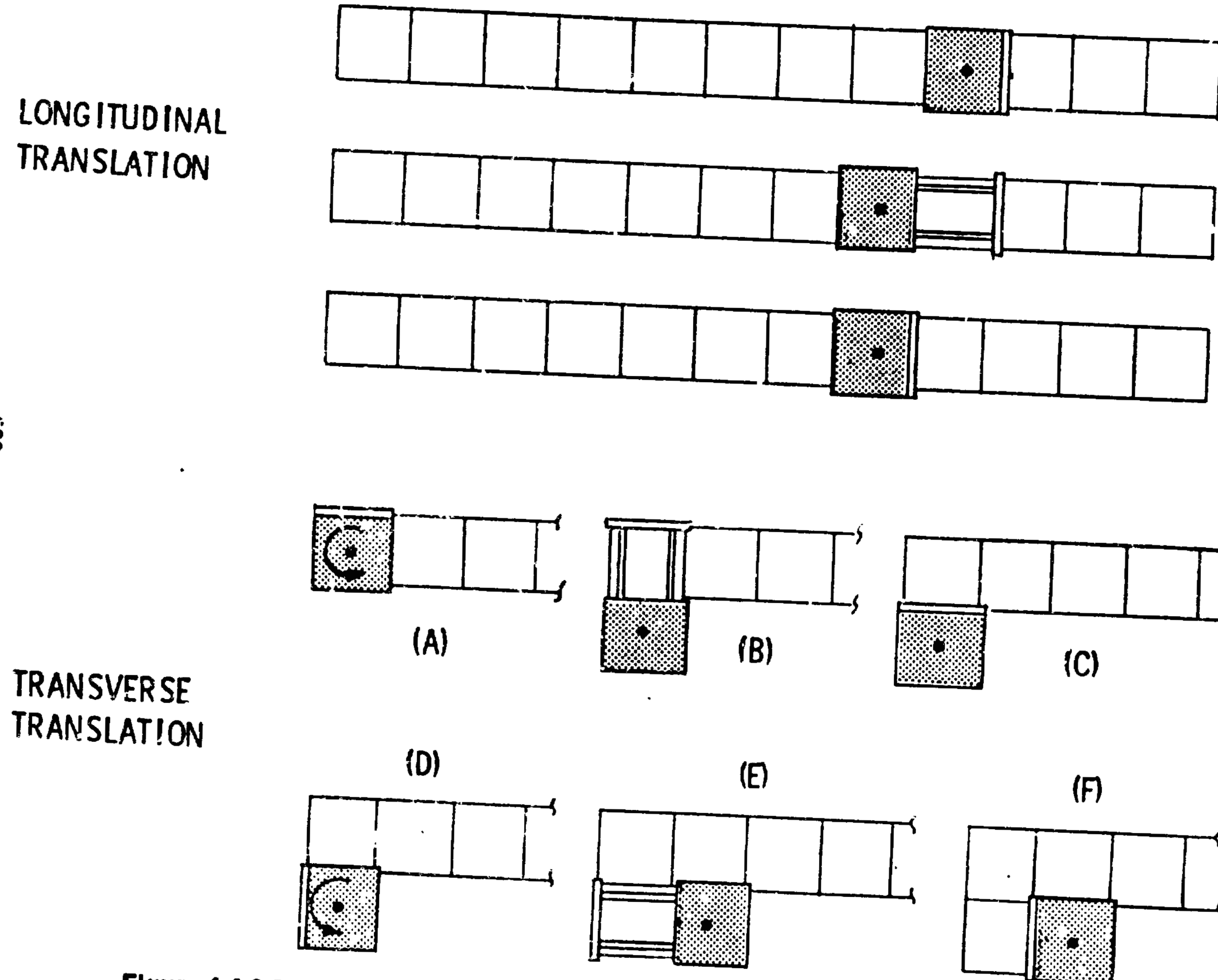


Figure 4.4.8.9-4 - MOBILE REMOTE MANIPULATOR SYSTEM TRANSLATION

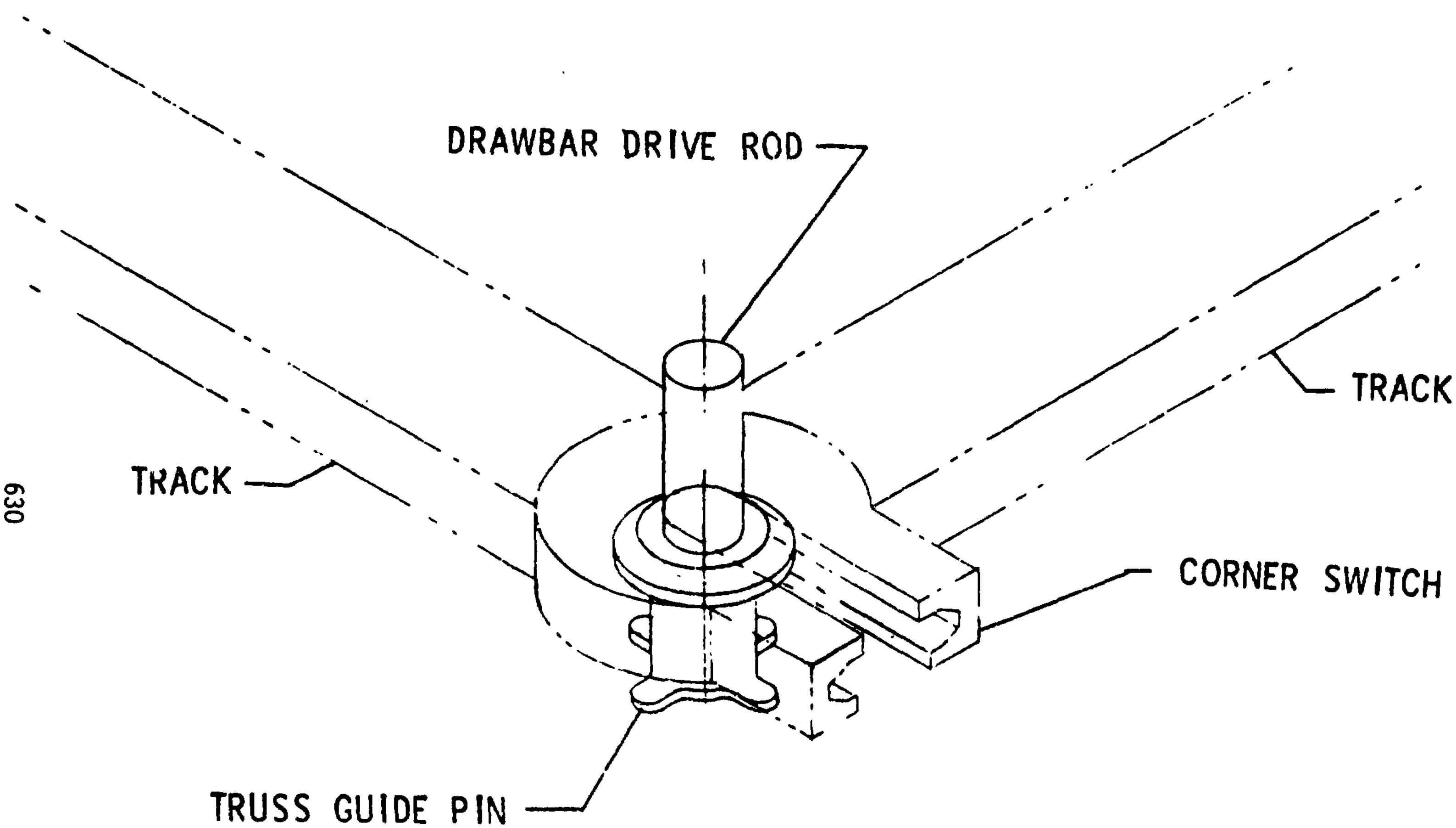


Figure 4.4.8.9-5 - CORNER SWITCH, GUIDE PIN, AND DRIVE ROD ARRANGEMENT

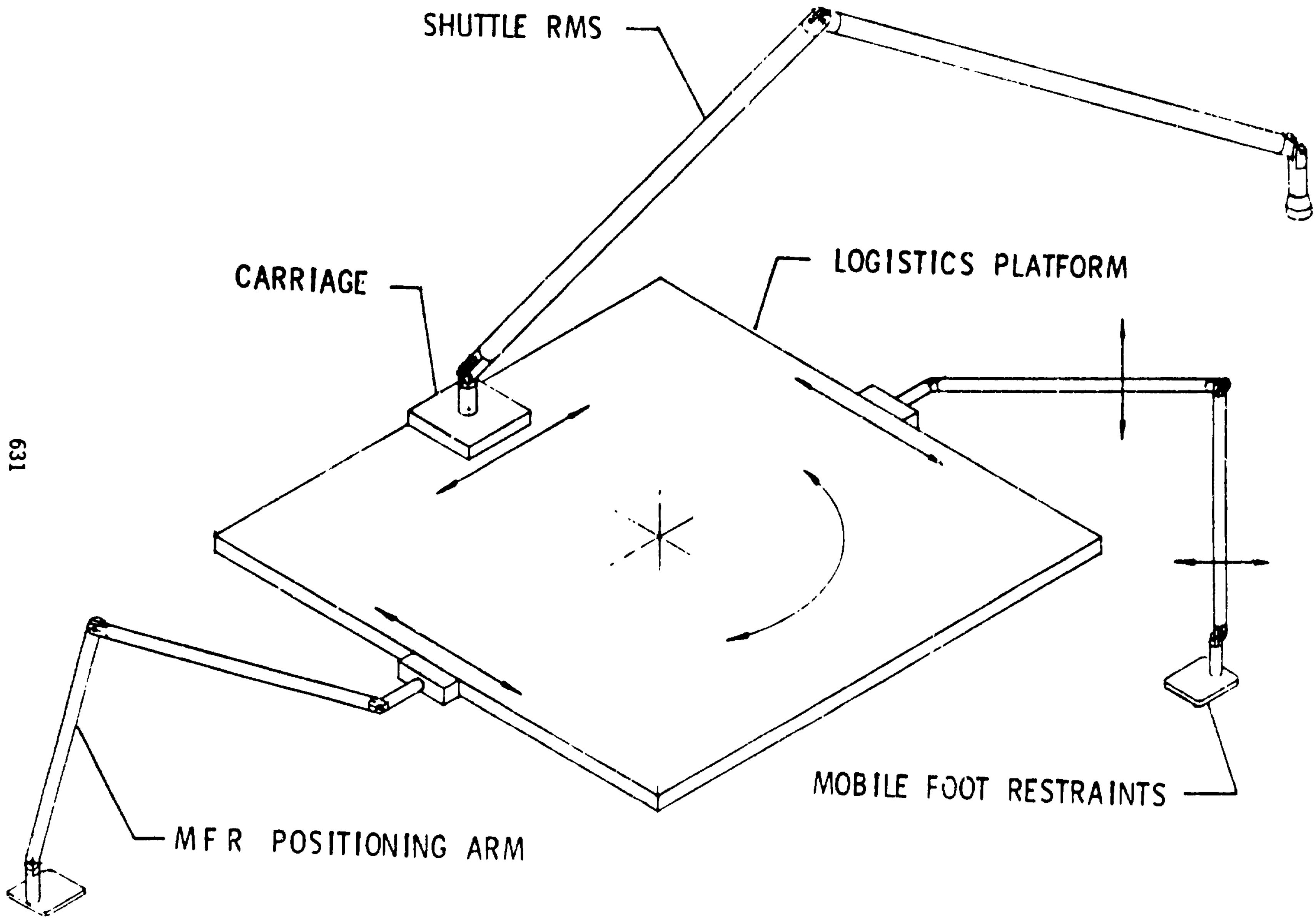


Figure 4.4.8.9-6 - REMOTE MANIPULATOR AND MFR POSITIONING ARM ARRANGEMENT

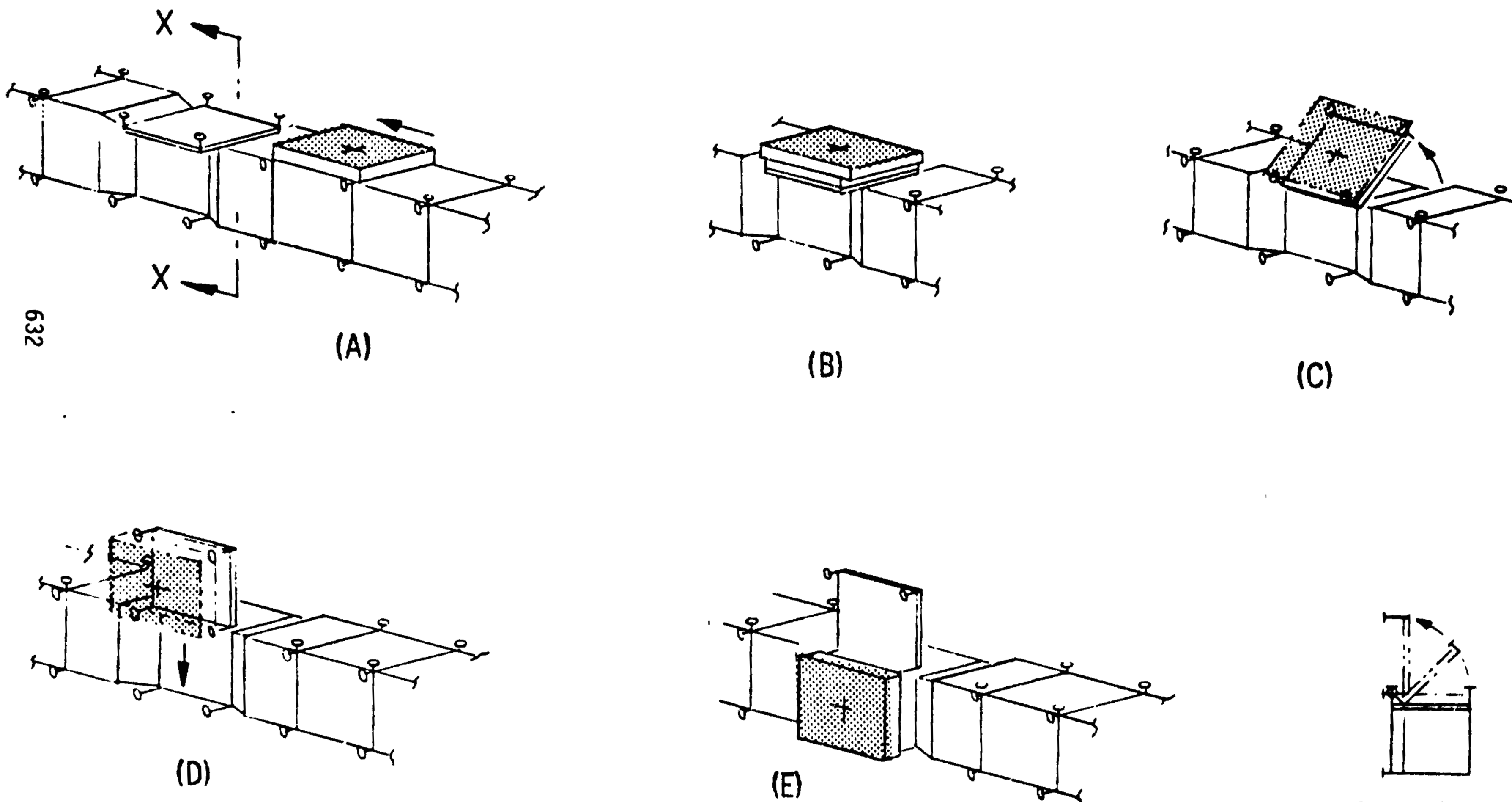


Figure 4.4.8.9-7 - MRMS OPERATIONAL PLANE CHANGE-CONCEPT I

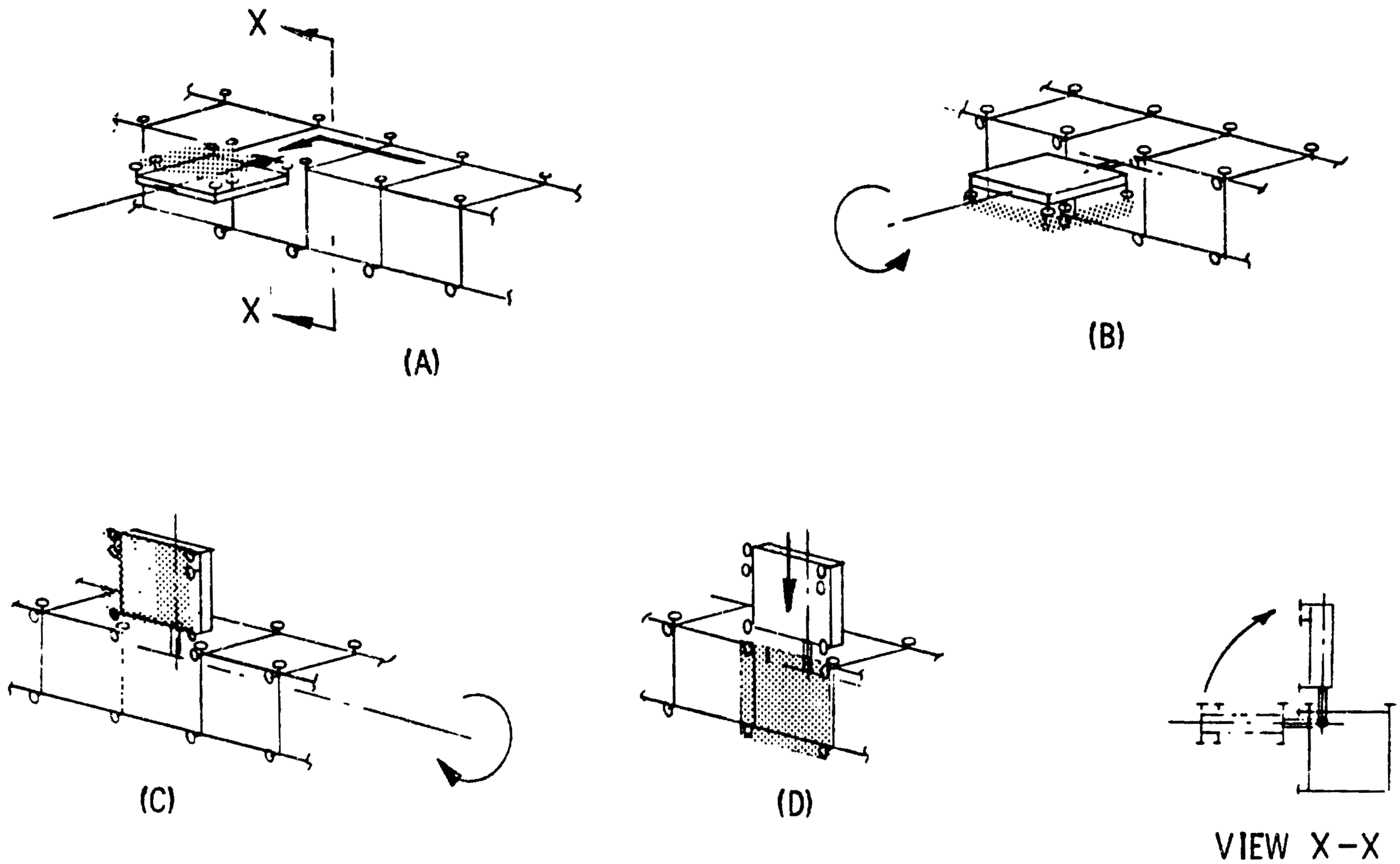


Figure 4.4.8.9-8 - MRMS OPERATIONAL PLANE CHANGE-CONCEPT II

4.4.8.10 Summary

Close inspection of the reference Space Station configuration indicated several areas requiring detailed technical support of the structures and mechanics team.

The basic structural foundation for the Space Station is the truss system. The truss configurations investigated for this station are clearly not a closed set but are only representative of trusses that can be successfully used. The best truss for the prime structure will be determined after trade studies considering deployment reliability, EVA time, cost, structural redundancy after damage, and ease and desirability of integrated utilities.

The heart of the solar panel deployment system is an automatically deployed mast structure. This mast is required not only to support the solar array but also to extract the solar panels from their containers during deployment. The masts presently available for solar array deployment are not redundant, either structurally or in the deployment process. Furthermore, damage to a single element, caused by debris impact or manipulator contact, can cause buckling of the total mast array and possible additional damage to neighboring structure. In the reference concept, eight independently deployed masts provide some degree of redundancy to the power capability.

Numerous module and payload attachment designs have been studied. The design emphasizes versatility, rigidity, and ease of crew installation. Design concepts were discussed for attaching payloads and modules to the 9-foot square truss beams, the 15-foot square truss beam and the planar tetratruss. It appears that the attachment of various items to the truss configurations presents little challenge. The most difficult was attaching the 14-foot modules to the 9-foot deployable keel.

It was concluded that the module to module interface should conform to the following general requirements:

a. Capability of mating with any other module interface or to any docking/berthing port.

b. Capability of alignment adjustment that is consistent with the development of specified module patterns.

c. Provision for retraction capability of the interface to allow clearance for module installation and removal.

d. Provision for a pressurized passage that is consistent with a 50-inch diameter hatch opening.

e. Provision for utilities interconnect (where necessary).

f. Capability of an interface latch to accommodate pressure loads and dynamic loads in the mated configuration.

After the second orbiter buildup flight, the RMS will not have sufficient strength to move the station to the orbiter. This seems to necessitate a docking system be attached to the Station during the Station buildup. Docking, rather than berthing, requires the design of a docking system that induces higher loads into the Station, thereby requiring stronger and stiffer interfaces.

The preliminary structural design and analysis study of a Shuttle-launched common module has been completed. It is clear that the primary structural shell presents no new technology requirement either in the fabrication or in the analysis phases. The largest loads are experienced during launch and landing (assuming the module to be returnable). The weight of the primary structural shell was determined to be approximately ten thousand pounds. Areas requiring additional development are the effects of long-term meteroid/debris protection and the precompressed window design.

Since the reference configuration uses numerous rotating joints, a detailed study of these joints was performed. The results indicate that the rotating joints can be designed to have continuity of stiffness across the joint and also function mechanically and electrically. The remaining concern of the rotating joint is its tolerance to meteoroid and space debris impact. A failure mode effect and replacement study for each part of the joint must be performed in the near future. Further joint stiffness studies considering the wear of the bearings (thus relieving preload) must also be made.

In the past, deployable structures, such as those used for the deployment of satellite solar cells, thermal radiators etc. were designed for high reliability. To obtain this reliability, it was necessary to over-design the device and perform an extensive amount of testing and analysis. In the construction of the Space Station, the crew provides a backup procedure to be incorporated if a deployment malfunction was to occur. Overall Space Station construction reliability can be just as high as in the past, but it comes from a combination of hardware reliability and crew backup procedure reliability.

The MRMS design employs the previously developed Orbiter RMS mounted to a mobile base that moves from truss node to truss node by means of short tracks hand mounted to the base. It utilizes rechargeable batteries that can support full operations for approximately 6 hours. The crew ride on and controls the manipulator during all operations. This concept is workable and requires only concept verification in the area of node-to-node traveling. It should greatly enhance the operational capabilities of the Station crew in performing such tasks as payload transport, Space Station repair/modification and structural assembly.

Two concerns shculd be mentioned. The first is that the primary truss structure, the solar array booms, and the rotary joint strut structure have little damage tolerance to impact by such things as space debris, or the Station manipulator. The second concern is the durability of the rotary joints, as well as procedures in case of joint failure.

4.4.9 EVA System Configuration

4.4.9.1 Introduction

The Space Station provides an EVA capability for assembly, maintenance, servicing, and repair of the Space Station and other large space structures; servicing and repair of satellites; maintenance, servicing and repair of the OMV and OTV; and maintenance, servicing, and repair of payloads and experiments. EVA systems can be broken into two areas: the GFE which includes the EMU's, MMU's and miscellaneous universal EVA equipment and the Space Station dedicated hardware which includes the EMU and MMU automatic service stations, the airlock, and Space Station dedicated EVA support hardware. This section presents a reference configuration for the Space Station dedicated hardware. It also presents weight and volume estimates for preliminary concepts of the GFE. RFP-9BC72-72-4-36P describes a 1-year study effort that will define EVA hardware requirements for the Space Station era.

4.4.9.2 Space Station Dedicated EVA Systems Configuration

The Space Station provides the hardware necessary to support routine EVA aboard the Space Station. This hardware includes the EMU servicing/checkout subsystem and the MMU servicing/checkout subsystem, the airlocks, various EVA support equipment outside the Space Station, and the EVA communications and data management system.

The Space Station provides for simultaneous EVA's (8 hours per day, 5 days per week) of two crew members during IOC and for four crew members during the growth phase. The following is a list of the EVA requirements which impact the Space Station systems configuration. These requirements are organized into the subsystem areas.

4.4.9.2.1 EMU service/checkout subsystem

The EMU reservice/checkout subsystem will be partially located in the Habitability or the Command Module and partially in the EVA airlock. Table 4.4.9-1 lists the functions of the EMU reservice/ checkout subsystems. Figure 4.4.9-1 presents the reference EMU service subsystem configuration. The power, mass, and volume penalties for the EMU service/checkout station are presented in Tables 4.4.9-2a,c,d,g.

The above reference configuration is preliminary. The EVA hardware requirements are a subject of an independent RFP effort. The results of this independent study will affect the EMU reservice/checkout subsystem configuration. The EVA study results will be provided to the Phase B Contractors both prior to and at IRR.

The EMU service station will interface with the ECLSS. The EMU's will contain regenerative, nonventing life support modules. The service station will regenerate the EMU life support modules. The EMU will normally be reserviced as an assembly. The EMU modules which require more than one hour to be reserviced will be serviceable at the modular level to accommodate contingency reservicing requirements by replacement.

The EMU's will support 8 hours of EVA/day (cumulative time at reduced pressure) at an average metabolic rate of 1000 BTU/Hr. The 8 hours EVA will generally be divided by a break in the Space Station. EMU subsystems may be revitalized during this break. The EMU's are regenerated in 12 hours without human intervention to provide next day EVA capability. Rapid emergency EVA egress (within 10 minutes) is possible with minimal EMU functional checkout. The EMU service station provides for automatic servicing, checkout, and IV operations of the EMU while in the airlock.

Automatic servicing and performance checkout of the EMU's includes expendables replenishment such as O_2 and H_2O resupply and the regeneration of time dependent processes such as CO_2 and H_2O removal, heat rejection and power storage. The service station provides for the automatic checkout of the EMU components and system performance. The service data may be retained by the Space Station data system. The EMU software automatically updates to adjust for small EMU instrumentation shifts to reduce the need for frequent instrumentation adjustments. Performance trend data shall be used to define the need for maintenance of the EMU. The service station will automatically dry the suit. EMU servicing capabilities are based on 10 reservices per week for IOC and for 20 reservices per week for the growth station.

4.4.9.2.2 MMU reservice/checkout subsystem

The MMU will be used for free-flying proximity operations around the Space Station.

The MMU uses an expendable cold gas propellant which must be resupplied after each use. The MMU also requires a recharge of its batteries. The MMU will be automatically checked out after each use. The reference MMU service station is presented in figure 4.4.9-2. Power, mass, and volume penalties are presented in Tables 4.4.9-2a,c,d,g.

Automatic servicing and performance checkout of the MMU's includes expendables replenishment such as N_2 resupply and the regeneration of time-dependent processes such as thermal and power storage. The MMU service station provides for the automatic checkout of the MMU components and system. The service data is retained by the data system. Performance trend data is used to define the need for maintenance of the MMU. The entire normal servicing will be accomplished in 12 hours without human intervention. Propellant reservicing will be accomplished in 10 minutes. Servicing capabilities are based on 10 reservices per week for IOC and for 20 reservices per week for the growth station.

4.4.9.2.3 EVA airlock

The EVA airlock provides the means for transferring an EVA crewmember from the Space Station to space, the stowage for the EMU's inside the airlock and for the MMU's outside the airlock, and provides a hyperbaric chamber for treatment of rapid decompression illness.

The EVA airlock provides a controlled rate of depressurization and pressurization. The nominal rate does not exceed .1 psi/sec. The maximum rates are not to exceed 1 psi/sec. Control of depressurization and pressurization is possible from inside the Space Station, inside the airlock and from outside the airlock by a single individual. An airlock gas recovery subsystem shall pump airlock gas back into the Space Station during the initial depressurization to conserve consumables. The airlock design accommodates the transfer of a standard laboratory experiment rack or the return of an incapacitated EVA crewmember. Life support umbilical connectors is available outside the airlock to allow umbilical EVA operations.

Stowage of the EMU's inside the airlock versus in the Space Station is required to allow for automatic checkout of the EMU's during depressurization and for reconnection of life support for contingencies while at vacuum. Stowage of the MMU's outside the airlock is required to centralize the EVA servicing equipment and to localize the EVA hardware. This localization also allows for easier relocation of the EVA equipment for flexibility for growth phases. The power, volume, and mass penalties for the airlock are shown in Tables 4.4.9-2b,d,f,g.. The airlock system consists of the basic airlock structure, the airlock gas recovery compressor, the hyperbaric equipment and the lights. The basic airlock structure is larger than the Shuttle airlock to provide additional space for the servicing hardware and additional space for the crew to maneuver for rapid egress capability. The structure is heavier because of the 5 atmosphere pressure requirements of the hyperbaric chamber and the weights of common docking adapters.

The airlock, the EMU, and the EMU service system have the capability of serving as a hyperbaric chamber for two crewmembers (one assisting). The airlock pressure is raised to as high as 5.0 atmospheres above the ambient cabin pressure (88 psi). The crewmembers breath alternately between O_2 off a mask and airlock gas with the O_2 pressure maintained above 3.0 psi. The O_2 concentration does not exceed 30 percent at any time. A small airlock between the EVA airlock and the Space Station is provided for passing medication, equipment, food, etc. The unmanned EMU's provide the life support for this contingency. Provisions will be provided for decontaminating the EMU after a chemical spill event. Verification of acceptable contamination levels shall be made prior to opening the airlock/(Space Station) hatch.

4.4.9.2.4 Equipment airlock

The equipment airlock serves as an equipment and tools transfer airlock. It is provided to minimize the consumables and time required to pass items between the Space Station and the EVA crew. The equipment airlock can be located at any convenient location on the Space Station. The dimensions, mass, and power penalties are presented for the reference configuration in Table 4.4.9-2b,d,f,g. These dimensions are arbitrary and are to be the subject of an independent EVA system requirements study to be performed to support phase B.

4.4.9.2.5 Miscellaneous EVA support equipment

Miscellaneous EVA support equipment is to be provided by the Space Station. Tools, tethers, foot restraints, hand holds, slide wires, equipment storage boxes, etc. are required for routine EVA. Weight and volume penalties for the reference configuration are presented in Table 4.4.9-2. The final values will be dependent on the Space Station configuration and the nature and amount of EVA required.

The EVA crewmember productivity is enhanced through the use of powered tools. Battery powered tools will be utilized for activities at remote sites. Plug in power tools will be provided for use at locations where an extensive amount of EVA work is expected to be performed (i.e., at the end of a RMS). Electrical outlets and flood lights will be provided at these locations.

Sufficient food and drink for 8 hours (38 oz. H_2O and 750 calories of food) will be available for use by an EVA crewmember while EVA. For EVA's with a scheduled break, only H_2O need be provided. EVA water and food containers will be cleaned and refilled with galley subsystems.

The LCVG (liquid cooling and ventilation garment) will be routinely washed or replaced. Provisions for EVA equipment and spares stowage are provided inside the Space Station and on the outside the EVA airlock. Maintenance of all EVA equipment is performed inside the Space Station.

4.4.9.2.6 Communications, data, and TV

The EVA crew will be provided with data by use of full page freeze frame TV transmission. EVA procedures, satellite servicing procedures, payload servicing procedures, Space Station assembly procedures, etc. will be provided in this manner. An audible instructions channel is provided to give audible instructions corresponding to a particular data page. The page displayed is controlled by voice command of the EVA crewmember eliminating the need for an IV crewmember to monitor procedures. One communications channel per two EVA crewmembers is provided. One data channel will be timeshared between all EVA crewmember and the Space Station. High speed TV reception from a portable EVA TV camera is available.

The description of the communications system used to support EVA is presented in section 4.4.3 communication and tracking subsystem description.

4.4.9.3 Government-Furnished EVA Hardware

The NASA will develop common use EVA hardware for use on Space Station. This hardware includes the EMU's, the MMU's, and miscellaneous EVA equipment such as TV cameras, tool caddies, etc. For reference, this hardware is described below:

4.4.9.3.1 EMU configuration

The requirements for the EMU are the subject of a 1-year study effort issued under RFP-9BC72-72-4-36P. Current concepts indicate that the EMU will be regenerable, automatically reservicable, and maintainable in the Space Station. The EMU performance will be automatically verifiable. The EMU will use a modular configuration to facilitate replacement and repair. There may be more than one technical approach for each modular function (i.e., CO_2 removal) depending on the evolution of technology, EVA requirements, application requirements (i.e., contingency EMU's may allow subsystem venting).

The mass, volume, and power required by the EMU's are presented in Table 4.4.9-2. The consumables and the returnables are also provided. Water and food are provided for an EVA crewmember while on EVA. An approach for the collection of urine will be provided.

4.4.9.3.2 EMU spares and maintenance

The EMU will be fully maintained in the Space Station. There will be at least one spare of each functional module. There will be at least two spares of all crewmember specific hardware such as arms, legs, gloves, and the LCVG.

There will be provided spare piece parts to repair the EMU modules. Repairs will be simple and require minimal tools (i.e., replacement of a computer card). All repairs will be performed inside the Space Station.

4.4.9.3.3 MMU Configuration

The MMU configuration used in this reference configuration uses the same design concept as that used on the current Shuttle program with the exception that the N_2 propellant capacity has been increased by a factor of 2.5 and that the MMU will be augmented with range-rate equipment the same or similar to that used at the docking ports. The MMU design will provide the capability for near-in stationkeeping, short-distance maneuvering, long-distance traverse (to 300 meters or more), and small object maneuvering (i.e., 600 lbs. for 100 meters). Mass and volume penalties are shown in Table 4.4.9-2. A nominal of two MMU usages per week have been planned for this reference configuration.

4.4.9.3.2.4 MMU Spares and Maintenance

The MMU will be fully maintained outside the Space Station. There will be provided spare piece parts to repair the MMU.

TABLE 4.4.9-1 - EVA RESERVICE/CHECKOUT SUBSYSTEMS

Reservicing

Recharge power supplies and power supplies checkou
Refill oxygen
Regenerate heat rejection module
Makeup cooling water leakage
Regenerate CO_2 removal module
Regenerate humidity removal module
EMU drying
Service trend data dump
MMU N_2 refill

Automatic Pre-EVA Checkout

EMU structural and leakage check
Fan performance check
Pump performance check
Computer/controller software check
Caution/warning software check
Suit pressure regulators check
Cooling temperature measurement calibration
Suit pressure transducer check
CO_2 sensor checkout
Cooling loop gas separator performance and water filtration
Primary and secondary O_2 supply transducer calibration
Vent flow sensor performance check and software correction
Metabolic rate calculation check
Positive/negative relief valve checkout

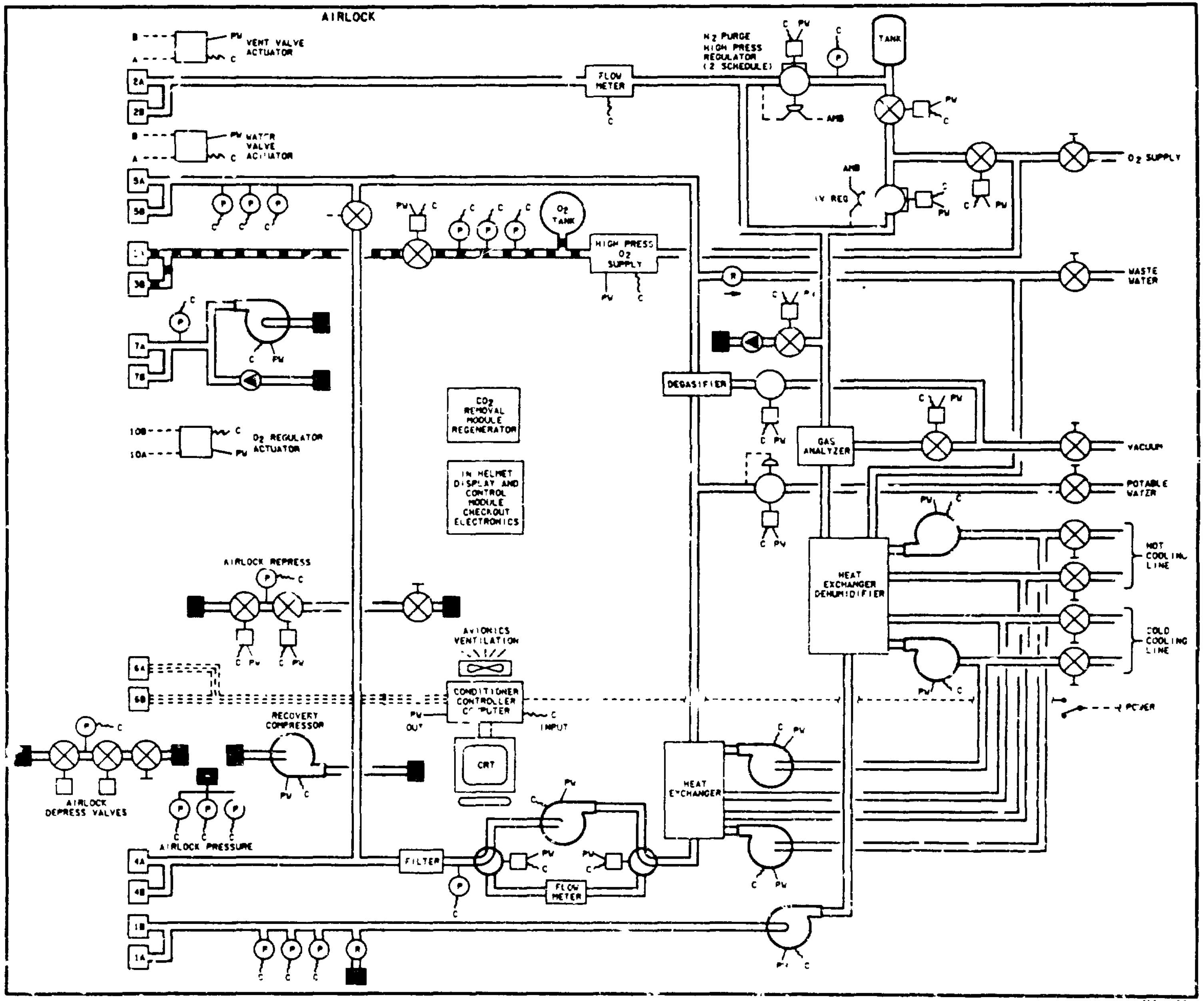
Post-EVA Performance Checkout

Ambient pressure transducer
Cooling loop leakage
Purge valve flow rate performance
LCVG structural and leakage
MMU systems performance check

Automatic EVA prep operations

Nitrogen purge and verification
Airlock depressurization/repressurization
IV pressure regulation
EMU integrity checkout

FIGURE 4.4.9-1
EMU SERVICE STATION



MMU SERVICE STATION

Figure 4.4.9-2

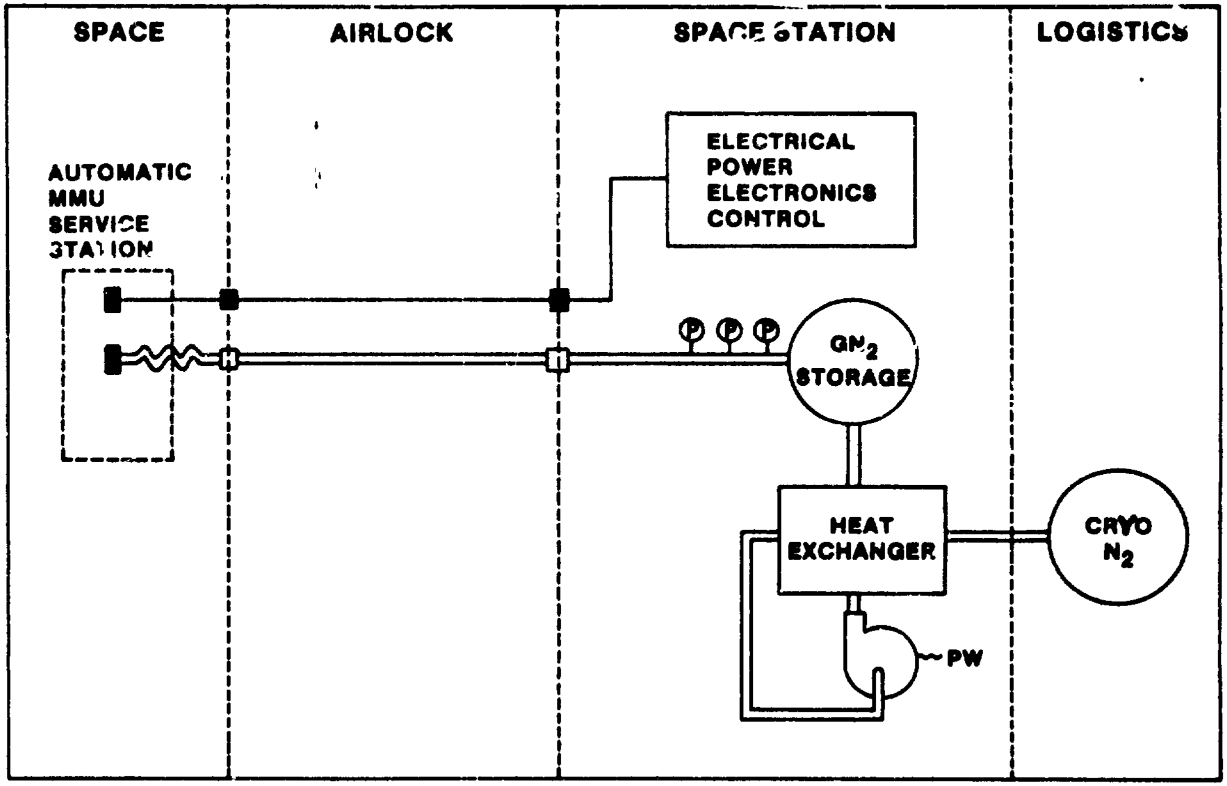


TABLE 4.4.9-2.a - EVA SUBSYSTEM WEIGHT/POWER/DIMENSIONS EXTERNAL AIRLOCK

COMPONENT	POWER WATTS DAY	POWER WATTS DAY	ACT BLOCKS POWERED 84xx	DIMENSIONS FTxFTxFT	WT/LBS	VOLUME CU FT
1 EMU SERVICE STATION (2 EMU'S SERVICED)						
ARLK VENT LOOP	100	100	17,23,24,25		139	22
ARLK COOL LOOP	30	30	17,23,24,25		140	22
HIGH PRESS 02 LOOP	100	0	17,19,22,23 24,25,27		44	1.8
HELMET DIS & CONT MOD	25	25	25		2	1.9
CO_2 REMOVAL REGEN	75	75	24,25		75	6
HUMIDITY REMOVAL & SUIT DRYER	200	200	24,25		102	4.9
POS PRESS RELIEF VLV CHECKOUT	80	80	25		20	2.1
				TOTAL	547	63
2 MMU SERVICE STATION (2 MMU'S SERVICED)						
HIGH PRESS N_2 SUPPLY	270	0	17,19,22,23 24,25,27	1x2x1	40	2
PWR CND/ELECTRONICS	210	210	20,21	1x1x1	50	1
SPARES (10%)	-	-	-	1x1x.2	17	0.2
N_2 STORAGE TANK (CRYO)	-	-	-	3.5	938	71
N_2 CRYOGENICS	-	-	-	-	1840	-
3 EMU						
EMU UNIT 1	170	170	25	.95x.75x6	440.9	17.05
EMU SERVICE STAND 1	20	20	25	4x3x1.5	30	18
EMU UNIT 2	170	170	25	.95x.75x6	440.9	17.05
EMU SERVICE STAND 2	10	10	25	4x3x1.5	30	18
LCVG 1	-	-	-	1.4x1.4x.4	8	1.2
LCVG 2	-	-	-	1.4x1.4x.4	8	1.2
SPARES	-	-	-	-	549.7	19.7
4 MMU						
MMU UNIT 1	-	-	-	3x4x2	451.1	31
MMU SERVICE STAND 1	10	10	21	3.2x4.2x1.0	50	-
MMU UNIT 2	-	-	-	3x4x2	451.1	31
MMU SERVICE STAND 2	10	10	21	3.2x4.2x1.0	50	-
SPARES	-	-	-	2x2x2	100	6
5 EVA TOOLS						
PWR TOOLS (BATTERY)	50	50	24	1x1x1	80	1
PWR TOOLS (UTILITY)	50	50	19	1x1x1	80	1
NON-POWERED TOOLS	-	-	-	1x1x1	40	1
EQUIP REPAIR TOOLS	-	-	-	1x1x.5	70	0.5

TABLE 4.4.9-2.b - EVA SUBSYSTEM WEIGHT/POWER/DIMENSIONS

COMPONENT	POWER WATTS DAY	POWER WATTS DAY	ACT BLOCKS POWERED 84xx	DIMENSIONS FTxFTxFT	WT/LBS	VOLUME CU FT
6 MISC EVA EQUIP						
~~EVA EQUIP~~	-	-	-	-	130	3.25
EQUIP STOWAGE LOCKER	-	-	-	3x3x1	10	9
COM MOD EQUIP FRAME	-	-	-	-	0	
EVA SYSTEM TOTALS	PEAK PWR (WATTS) = 1180 NIGHT/1550 DAY				6914.4	354.78

AIRLOCK WEIGHT/VOLUME/POWER PENALTY

~~AIRLOCK~~						
~~EVA AIR~~LOCK 1	-	-	-	6.7x8.3	1316	200
EQUIP ATTACHMENT	25	25	17,18,19,22			
LIGHTS			23			
GAS RECOVERY	1400	1400	18			
EQUIPMENT AIRLOCK	-	-	-	2x3	90	9.5
LIGHTS	3	3	5x DURING EVA			
GAS RECOVERY	100	100	5x DURING EVA			
HYPERBARIC EQUIP			-	1x1x1	50	1
AIRLOCK TOTALS	PEAK PWR = 1425 WATTS (6 MINUTES)*				1471	210.5

*TIME SEQUENCED WITH EVA
INTEGRATED PEAK PWR = 1425 NIGHT/1550 DAY

TABLE 4.4.9-2.c - SECONDARY EVA SYSTEMS POWER/VOLUME/MASS

COMPONENT	POWER WATTS DAY	POWER WATTS DAY	ACT BLOCKS POWERED 84xx	DIMENSIONS FTxFTxFT	WT/LBS	VOLUME CU FT
1 SECONDARY EMU						
EMU UNIT 1	170	170	-	.95x.75x6	440.9	17.05
EMU SERVICE STAND 1	20	20	-	4x3x1.5	30	30
EMU UNIT 2	170	170	-	.95x.75x6	440.9	17.05
EMU SERVICE STAND 2	10	10	-	4x3x1.5	30	30
2 SECONDARY EMU SERVICE STATION						
ARLK VENT LOOP	100	100	-			
ARLK COOL LOOP	30	30	-			
HIGH PRESS O_2 LOOP	100	0	-			
PWR CND/ELECTRONICS	410	410	-			
HELMET DIS & CONT	25	25	-			
CO_2 REMOVAL REGEN	75	75	-			
HUMIDITY REMOVAL & SUIT DRYER	200	200	-			
POS PRESS RELIEF VLV CHECKOUT	80	80	-			
				TOTAL	547	63
3 SECONDARY AIRLOCK	-	-	-	6.7x8.3	1316	200
EQUIPMENT ATTACHMENT					15	
LIGHTS	25	25	-			
GAS RECOVERY	1400	1400	-			
HYPERBARIC EQUIP	-	-	-	1x1x1	50	1
SECONDARY EVA TOTALS BACKUP SYSTEM POWER = 0					2869.8	358.1

TABLE 4.4.9-2.d - EVA SUBSYSTEM WEIGHT/POWER/DIMENSIONS (EXTERNAL AIRLOCK)

COMPONENT	CONSUMABLES	RETURNABLES	LOCATION
1 EMU SERVICE STATION (2 EMU'S SERVICED)			
ARLK VENT LOOP	.015# H_2O/RCHG/EMU		
ARLK COOL LOOP	1.25# O_2/RCHG/EMU		
HIGH PRESS O_2 LOOP			
PWR CND/ELECTRONICS			
HELMET DIS & CONT MOD			
CO_2 REMOVAL REGEN	.4# N_2/2 WEEKS	1.6# CO_2/EMU MAX	
HUMID REMOVAL & SUIT	.05# H_2/RCHG	3.2# H_2O/EMU MAX	
DRYER	750 CALORIES/MAN/DAY	5# URINE/EMU PEAK	
POS PRESS RELIEF VLV	2.5# H_2O/EMU MAX	2# URINE/EMU NOM	
CHECKOUT	1.3# H_2O/EMU NOM		
SPARES (10%)			ARLK
			IN SS
		FRONTAL AREA = 60 SQ. FT.	
2 MMU SERVICE STATION (2 MMU'S SERVICED)			
HIGH PRESS N_2 SUP	57.5# N_2/RCHS/MMU		ARLK
PWR/CND/ELECTRONICS			ARLK
SPARES (10%)			IN SS
N_2 STOR. TANK (CRYO)	32 MMU RCHGS/90 DAYS		LOG OR
N_2 CRYOGENICS			EXT
3 EMU			
EMU UNIT 1	100#/HR COOLING @ 45		ARLK
EMU SERVICE STAND 1			ARLK
EMU UNIT 2	100#/HR COOLING @ 45		ARLK
EMU SERVICE STAND 2			ARLK
LCVG 1	1 WASH/2 EVA'S		IN SS
LCVG 2			IN SS
SPARES			IN SS
4 MMU			
MMU UNIT 1			OUT ARLK
MMU SERVICE STAND 1			OUT ARLK
MMU UNIT 2			OUT ARLK
MMU SERVICE STAND 2			OUT ARLK
SPARES			IN SS
5 EVA TOOLS			
PWR TOOLS (BATTERY)	BAT/RESUPPLY		OUT ARLK
PWR TOOLS (UTILITY)			OUT ARLK
NON-POWERED TOOLS			OUT ARLK
EQUIP REPAIR TOOLS			IN SS

TABLE 4.4.9-2.e - SECONDARY EVA SYSTEMS POWER/VOLUME/MASS

COMPONENT	CONSUMABLES	RETURNABLES	LOCATION
1 SECONDARY EMU'S			
EMU UNIT 1	100#/HR COOLING @ 45		ARLK
EMU SERVICE STAND 1			ARLK
EMU UNIT 2	100#/HR COOLING @ 45		ARLK
EMU SERVICE STAND 2			ARLK
2 SECONDARY EMU SERVICE			ARLK
ARLK VENT LOOP	.015# H_2O/RCHG		
ARLK COOL LOOP	1.25# O_2/RCHG		
HIGH PRESS O_2 LOOP			
PWR CND/ELECTRONICS			
HELMET DIS & CONT MOD			
CO_2 REMOVAL REGEN	.4# N_2/2 WEEKS	1.6# CO_2/EMU MAX	
HUMID REMOVAL & SUIT DRYER	.05# H_2/RCHG 750 CALORIES/MAN/DAY	3.2 H_2/EMU MAX 5# URINE/EMU PEAK	
POS PRESS RELIEF VLV CHECKOUT	2.5# H_2O/EMU MAX 1.3# H_2O/EMU NOM	2# URINE/EMU NOM	
3 SECONDARY AIRLOCK			HAB1 HAB2
EQUIP ATTACHMENT			
LIGHTS			
GAS RECOVERY			
HYPERBARIC EQUIP			
SECONDARY EVA TOTALS			

TABLE 4.4.9-2.f - EVA SUBSYSTEM WEIGHT/POWER/DIMENSIONS

COMPONENT	CONSUMABLES	RETURNABLES	LOCATION
6 MISC EVA EQUIPMENT			
EVA EQUIP			OUT ARLK
EQUIP STOWAGE LOCKER			OUT ARLK
COM MOD EQUIP FRAME (NEEDED FOR INTERNAL ARLK ONLY)			
EVA SYSTEMS TOTALS			
AIRLOCK			
EVA AIRLOCK 1			HAB 1 HAB 2
EQUIP ATTACHMENT			
LIGHTS			
GAS RECOVERY	3# AIR/DEPRESS		
EQUIPMENT AIRLOCK			TBD
LIGHTS			
GAS RECOVERY	.65# AIR/DEPRESS		
HYPERBARIC EQUIP			ARLK
AIRLOCK TOTALS			

TABLE 4.4.9-2.g - EVA ACTIVITY BLOCK DESCRIPTION

A.B. NO.	A.B. NAME	DESCRIPTION	EVA DAY TIME (HRS)	MMU DAY TIME (HRS)	NON-EVA DAY TIME (HRS)
8417	PREPEV	EVA PREP-IV IN AIRLOCK	0 - .15	0 - .15	-----
8418	DEPRES	EVA-AIRLOCK DEPRESS	.15 - .25	.15 - .25	-----
8419	EVA	EVA	.25 - 8.25	.25 - 8.15	-----
8420	MMUSRV	MMU SERVICE	-----	8.0 - 24.0	-----
8421	MMUCHK	MMU CHECKOUT	-----	7.5 - 8.0	-----
8422	REPRES	EVA-AIRLOCK REPRESS	8.15 - 8.25	8.15 - 8.25	-----
8423	POSTEV	POST EVA-IN AIRLOCK	8.25 - 8.35	8.25 - 8.33	-----
8424	EMUCHG	EMU RECHARGE	8.35 - 21.35	8.35 - 21.35	-----
8425	CHECK	EMU CHECKOUT	8.35 - 10.35	8.35 - 10.35	-----
8426	UTLPWR	EVA UTILITY PWR	.5 - 7.0	-----	-----
8427	READY	EVA EQUIP READY	21.35 - 24.0	21.35 - 24.0	0-24

5.0 UNMANNED PLATFORMS

The unmanned Platforms operate as complementary extensions of the Space Station Manned Core capabilities. They will be of a size appropriate to be useful to most potential customers. The payloads they carry will be comprised of (1) a scientific instrument or a compatible set of instruments that have a similar field of investigation, (2) a technology development mission, or (3) commercial production units.

As scientific and programmatic constraints and requirements have matured, the design of Platforms as a Space Station Program Element has evolved through a number of stages. In its initial form, the unmanned Platform was envisioned as a large structure drawn directly from the Space Station structural configuration which could accommodate a large complement of many varied instruments. Variations on this approach to Platform design included use of a resource module duplicated from the Space Station, a scaled-down resource module from the Space Station, and then finally use of subsystem elements from the Space Station. It is expected that the basic configuration design would be common to both polar and co-orbiting platforms. As the design, cost, the user requirements have matured, the concept has evolved into a relatively small Platform design. This smaller Platform would accommodate different payload packages made up of complementary sets of instruments. One or more of these smaller Platforms could be placed into polar orbit and/or into co-orbit with the Space Station.

A key feature of this most recent concept is the use of a Platform support services core. This core provides a common Platform to which all payloads can be attached. It is replicated as funding and instrument payloads are available, and the capability of the core can be increased to accommodate larger numbers and/or more demanding payloads in the growth period. The basic core is designed to meet the requirements of near-term (IOC) missions. A growth core, which has expanded payload accommodation features beyond those of the IOC module, has also been defined. The growth configuration can be reached either by the fabrication of a new core spacecraft or by expansion of the payload accommodation capabilities by subsystem enhancement while the core remains in orbit.

The Platform will have a near-hemispheric field of view in one direction. The field of view may be either inertial, for solar or astrophysics studies, or Earth oriented. Payload pointing control would be provided by the platform core itself, without payload gimbals, at an accuracy level suitable for most customers. The core will accept fine error signals from the instruments if required to meet their objectives.

The interface to the payload instruments attached to the Platform will be standardized to be compatible with the instrument payload interface on the Space Station. This will allow easy interchange of payloads on the Platform. It will also permit convenient exchange of instrument payloads between the Space Station Manned Core and the Platforms.

The Platform design will use Space Station Manned Core elements to the maximum extent that is cost effective. The basic elements of the Platform Core will be modular and will incorporate standard interfaces to allow in-orbit servicing, repair, and upgrading. This modularity will permit growth of the Platform capabilities while on-orbit to meet the increasing needs of future customers.

The near-term core design is sized to require only one STS launch to place it in orbit with payload. This consideration limits the total Platform launch weight into polar orbit to about 30,300 lbs. The Platform Core will have initial capability to provide 5 kW of electrical power for customers. This is sufficient for most scientific investigations. The growth capability of each IOC-type Platform Core will be to 20 kW which is needed by some post IOC missions.

5.1 DESIGN REQUIREMENTS

The design objectives given in section 5.0 are subject to trade studies to determine the best approach for implementation. For the purpose of defining a reference configuration, the set of design requirements stated below has been derived from these objectives based on the needs of potential users and the requirements for Platforms that are stated in the RFP.

5.1.1 General Design Requirements

The primary design objective for the Platforms is to provide a versatile, growing, permanent, unmanned facility in space with the capability to accomplish significant advances in space science, technology, and commerce. The Platform design features are primarily driven by payload accommodation. The Platform must operate in several mission dependent modes:

a. Nadir orientation.

b. Inertial orientation; instruments Sun pointed.

c. Inertial orientation; instruments celestial pointed.

Design requirements for the reference Platform configuration have been derived from these objectives, both for an initial (near-term) and a growth capability. These are summarized below.

a. The IOC Platform will be capable of being placed in orbit by a single STS launch with a complement of payloads.

b. Servicing interval will typically be 2 years with a 1-year contingency. Hardware elements will be serviceable using Orbital Replaceable Units (ORU's) or an equivalent approach. Servicing will be provided by the STS, the Space Station, or via Orbital Maneuvering Vehicle (OMV) in combination with one of these elements.

c. A single Platform Core will be used to support all Platform missions. Maximum use will be made of Space Station hardware and software elements. Post IOC missions may require growth of the Platform Core.

d. Any single failure will not preclude continued normal operation of the Platform. Any subsequent failure will not preclude operation in a degraded mode until service can be provided.

e. Platform will have a minimum operating life of 10 years, with maintenance, and a goal of 15 years.

f. In polar orbit, the Platform will be capable of a nadir orientation and will provide an unobstructed view of the Earth including the limbs of the atmosphere.

g. In co-orbit the Platform will be capable of either nadir or inertial orientations. (Co-orbit means that the Platform's inclination will be the same as the Space Station Manned Core.)

h. The configuration of the Platform will accommodate selected sets of instruments listed in the current Space Station Mission Data Base. The reference configuration herein uses the data base as of July 1984. However, the Platform Core is also designed to accommodate future undefined missions without basic changes.

i. Capability will be provided both for the enhancement of payload accommodation through growth on-orbit during the lifetime of an initial Platform, as well as for the expansion of payload accommodations for future missions through a growth Platform design.

j. The primary communication link for the Platform will be through the TDRSS.

k. The Platform will be capable of operating autonomously with a minimum of ground control. Telemetry status of all systems will be provided for monitoring and for maintaining an audit trail.

l. On-orbit assembly as well as instrument changes or upgrades may be used as required.

m. For cost estimating purposes, it was assumed that the Space Station Program will provide the Platform Core subsystem hardware and the payload support structure for the Platform. Instruments will be provided by the customer.

n. If greater pointing accuracy is required than that provided by the Platform Core, then fine error signals will be provided by the customer instruments.

o. The Platform will be designed so that normal functions do not interfere with the performance of the instrument sensors or sensitive surfaces. The instruments will be designed and grouped such that they will not contaminate neighboring instruments. Instruments that are especially sensitive to contamination will provide door mechanisms to be closed just prior to a propulsive event or the beginning of another contamination situation.

p. The design must meet instrument viewing requirements and provide an adequate view to cold space for instrument detector active cooler radiators.

q. Solar array area will be determined by overall platform power requirements, but the location, aspect ratio and deployment schemes must meet the following requirements:

(1) Solar and aerodynamic torques kept within the control capability of the GN&C system.

(2) Payload fields of view requirements including reflected light, thermal radiation and radiator view fields.

(3) Capability to fit within the Shuttle cargo bay.

(4) Minimum shadowing of array by platform structure.

5.1.2 Payloads and Missions

A large, representative group of instrument has been identified for both initial and growth Platform concepts. These instruments will require both co-orbiting and polar orbits. It is not suggested that this entire representative group of instruments be present on-orbit at any one time. Rather, through consideration of the myriad of requirements raised by this group of instruments, an understanding of the sizing of both the initial and growth Platform Core modules may best be realized. To this end, the representative group of instruments has been partitioned into a number of mission payload sets on the basis of viewing objectives and power requirements. The following two sections describe these mission payload sets and their accompanying mission objectives.

5.1.2.1 Polar Payloads and Missions

The payloads identified for polar orbit support a strategy for obtaining integrated Earth science measurements from low Earth orbit. Synergistic groupings of instruments, coupled with near-simultaneous measurements, are directed toward a comprehensive multidisciplinary approach to understanding the Earth as a system. This is the basis behind the Earth Observation System (EOS), which may ultimately provide a facility capability for additional users such as NOAA and commercial customers.

The EOS payloads are comprised of a wide range of instruments spanning the spectral region from the UV to the submillimeter microwave. The instruments cover applications in oceanic, atmospheric, solar, and land resources sciences. The instruments have been partitioned into four mission payload sets, and the composition of each Platform payload is addressed below.

The initial mission parameters of a Sun synchronous orbit with a 2 p.m descending node were chosen on the basis of global coverage and consistency of illumination. The 2 p.m. nodal crossing was chosen to avoid the Sun glint over water for orbit crossings near noon and to provide maximum contrast with the 2 a.m. ascending node for the thermal and soil moisture measurements.

The EOS science requires good "two-day" coverage. In considering orbit altitudes for good "two-day" coverage, further trades were made by evaluating the spacecraft environments of radiation, contamination and drag together with atmospheric path length for measurements at the extreme scan positions and the resulting pixel distortion compared to the nadir view for the wide swath instruments.

Based on these considerations, the altitude and inclination required for the polar instruments are 380 nmi and 98.2°, respectively. This orbit has a 16-day repeat cycle.

a. Polar mission payload A is considered an early mission and carries a collection of all the land and ocean instruments which require consistent, global coverage (two-day repeat cycle). The instruments are

(1) The Moderate Resolution Imaging Spectrometer (MODIS)

(2) The High Resolution Multifrequency Microwave Radiometer (HMMR)

(3) The LIDAR Atmospheric Sounder for Aerosol Measurements (LASA-A)

(4) The Radar Altimeter (ALT)

(5) The Scatterometer (SCAT) and

(6) The Automated Data Collection and Location System (ADCLS)

b. Polar mission payload B is considered an early mission and comprises high spatial resolution, small swath-width, target-selective instruments for land and some ocean applications as well as their supportive instrumentation. The payload includes:

(1) The High Resolution Imaging Spectrometer (HIRIS)

(2) The Thermal IR Mapping Spectrometer (TIMS)

(3) The LIDAR Atmospheric Sounder for Aerosol Measurements (LASA-A)

(4) The LIDAR Altimeter (LASA-B)

c. Polar mission payload Synthetic Aperture Radar (SAR) is assigned essentially its own Platform to maximize the available power and data capability. Solar and magnetospheric monitors will also be included in this Platform. This is also considered an early mission.

d. Polar mission payload C is considered to be a later mission set. It includes the following atmospheric instruments to obtain simultaneous coverage of lower and upper atmospheric processes and composition.

(1) Correlation Radiometer (CORAD)

(2) IR Interferometer Spectrometer (MINS)

(3) Differential Absorption LIDAR (DIAL)

(4) Doppler LIDAR (DOPLID)

(5) IR Radiometer (IRAD)

(6) Tilting Etalon Spectrometer (TES)

(7) Submillimeter Spectrometer (SUMIS)

(8) Microwave Limb Scanner (MLS)

(9) UV/Visible Spectrometer (UVVS)

(10) Fabry-Perot Interferometer (FAPIN)

(11) Michelson Interferometer (MIKIN)

Table 5.1.2-1 summarizes the payload characteristics for the polar mission payloads.

Considered within payload requirements for polar missions A, B, and C are additional requirements for growth. These include 800 watts power and 0.2 Mb/s data rate for instruments or experiments. Weight margin is not explicitly shown because with in-orbit integration of new instruments, weight margin is very large.

TABLE 5.1.2-1- POLAR MISSION PAYLOAD CHARACTERISTICS

	WEIGHT LB	AVERAGE ORBITAL OPERATING PWR, W	DATA RATE Mbps	POINTING CONTROL, ARC SEC	ALTITUDE NMI	LAUNCH
POLAR A	5,456	4,500	5.4	360	380	NEAR TERM
POLAR B	8,140	4,000	188.9	360	380	NEAR TERM
POLAR SAR	2,508	2,800	300.0	360	380	NEAR TERM
POLAR C	10,037	7,400	2.4	60	380	LATER

The numbers listed here are representative of these classes of payload instruments. The Space Station Mission Data Base should be consulted for specific values. "Near-term" configurations are candidates for the IOC capability; "later" configurations are growth.

5.1.2.2 Co-Orbiting Mission Payloads

Payloads [illegible] flown as early missions in co-orbit are a collection of solar and astrophysical instruments The large masses, nominal altitude selection, and pointing requirements of the instruments suggest that most of the instruments be considered separate mission payloads. The instruments to fly in co-orbit are as follows:

Co-orbit payload SIRTF	Space Infrared Telescope Facility
Co-orbit payload Starlab	Starlab
Co-orbit payload P/OF	Pinhole/Occulta on Faci.
Co-orbit payload ASO	Advanced Solar Observatory
Co-orbit payload HTP	High Throughput Experiment

The altitude and inclination requirements of these instruments are 270 nmi and 28.5°, respectively, except for SIRTF which is 378 to 540 nmi and 28.5°. The 378-540 nmi altitude for SIRTF was selected to minimize optical contamination. Note that co-orbiting means that the Platform will be flown at the same inclination as the manned Space Station.

Table 5.1.2-2 summarizes the instrument characteristics for each of the co-orbiting Platforms.

An example of a growth mission for co-orbiting Platforms would be a materials processing production unit which could require 20 kW of power.

TABLE 5.1.2-2.- CO-ORBITING PLATFORM PAYLOAD CHARACTERISTICS

	WEIGHT LB	AVERAGE ORBITAL OPERATING PWR, W	DATA RATE Mbps	POINTING CONTROL, ARC SEC	ALTITUDE NMI	LAUNCH
SIRTF	8,820	1,000	1.0	0.15	378-540	NEAR TERM
STARLAB	7,056	2,400	16.0	2	270	NEAR TERM
P/OF	7,938	700	1.4	10	270	NEAR TERM
ASO	19,625	4.500	42.0	1	270	LATER
HTP	22,050	2,400	0.3	3	270	LATER

5.2 CONFIGURATION DESCRIPTION

The following section describes the Platform reference configuration. The reference Platform Core configuration is shown in 3-view planform in Figure 5.2-1 and in perspective in Figure 5.2-2. Three Platform configurations equipped with various instrument payloads are shown in Figures 5.2-3, 5.2-4, and 5.2-5.

The reference description provided here is based on studies which maximized the use of Space Station Core hardware in Platforms. Equal importance was placed on maximizing utility to the users by meeting instrument requirements as defined to date. This reference configuration description is provided only as a departure point for Phase B studies and is not considered to represent an optimized design.

5.2.1 General Arrangement

A Platform consists of a Platform core and an instrument payload. These will normally be launched as an integrated unit. However, some payloads may be better launched independently and integrated with the Platform Core in orbit.

A Platform Core provides support services to the payload. It provides pointing and accepts fine error signals for improved pointing for one instrument. Individual instruments provide finer pointing, if required. Attitude control is provided by momentum exchange with propulsive backup. The Platform Core provides its own thermal control using passive techniques (i.e., no mechanical pumps) but does not provide heat rejection capability for payloads mounted to the end of the Core, such as the Starlab payload. Electrical power is provided by solar arrays with batteries for energy storage.

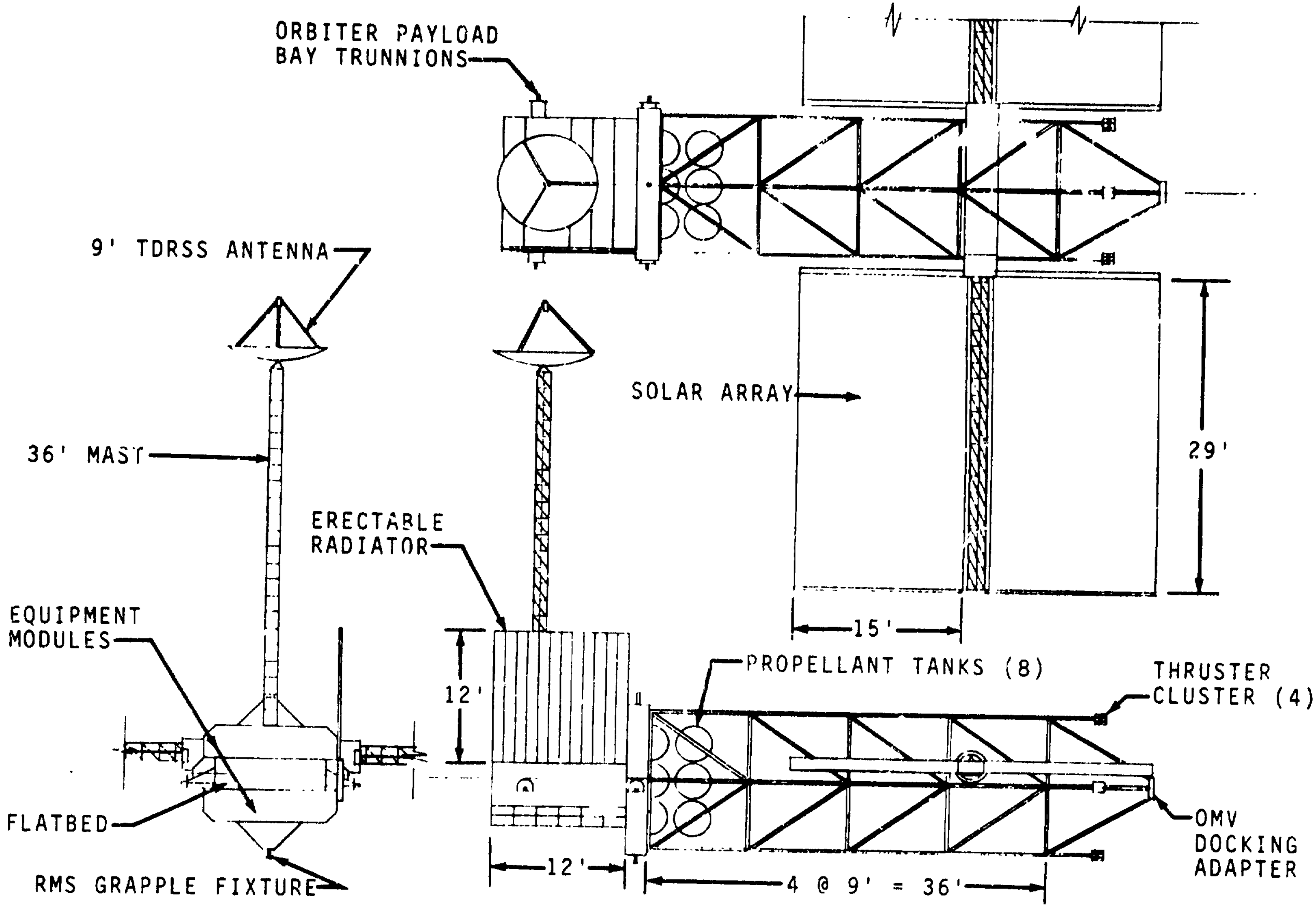


Figure 5.2- CORE PLATFORM

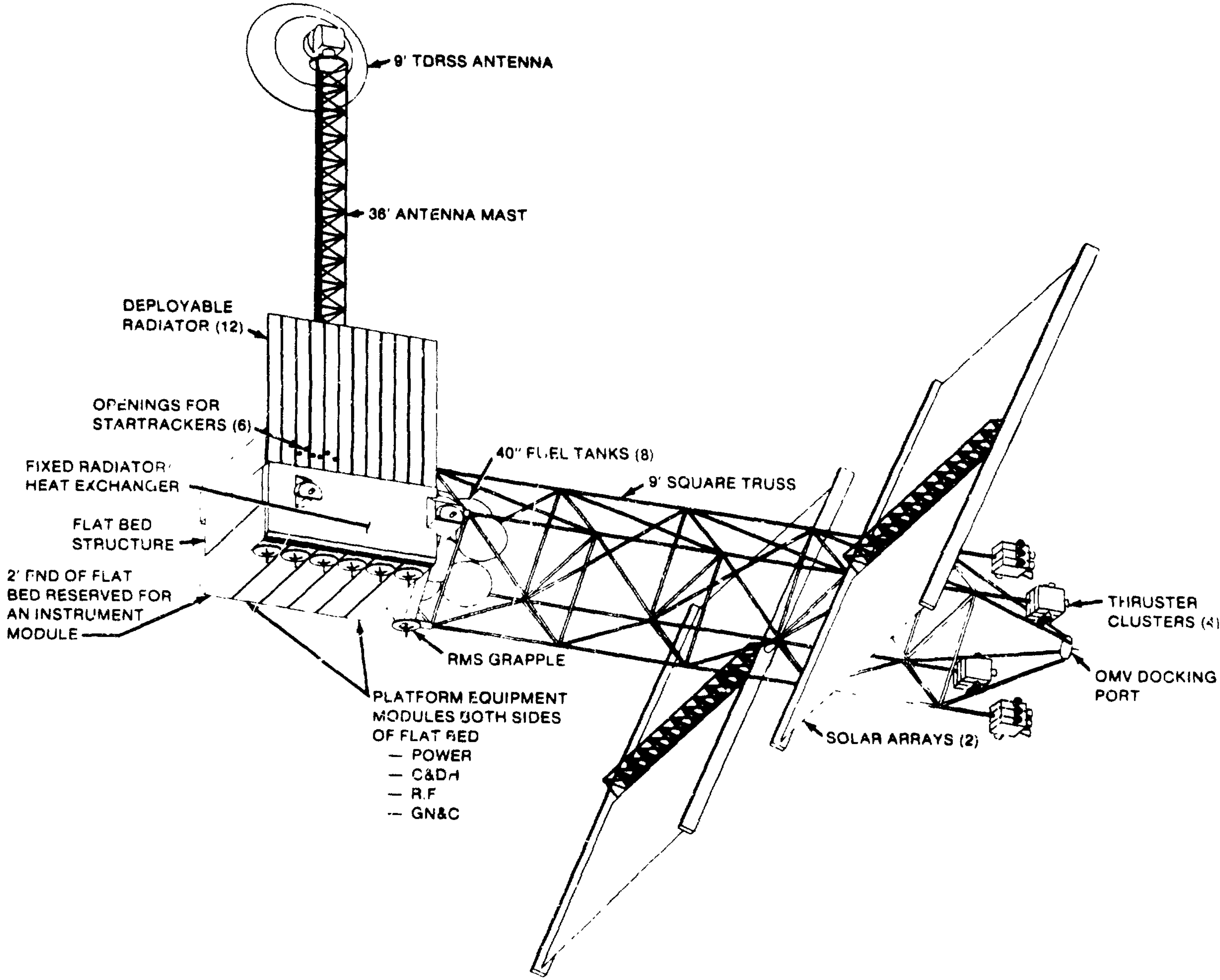


Figure 5.2-2 - CORE PLATFORM

ORIGINAL PAGE IS
OF POOR QUALITY

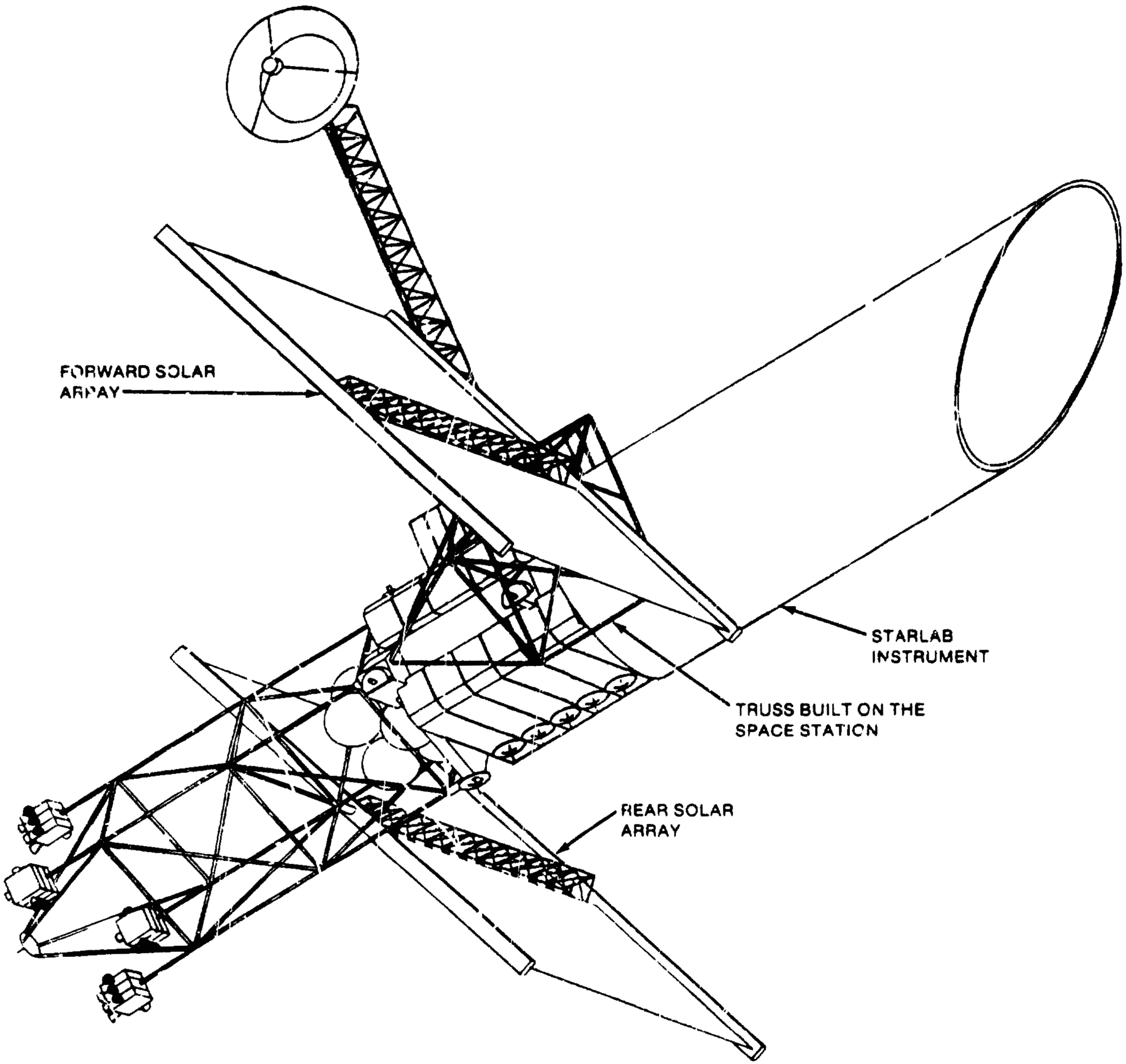


Figure 5.2-3 - STARLAB PAYLOAD PLATFORM

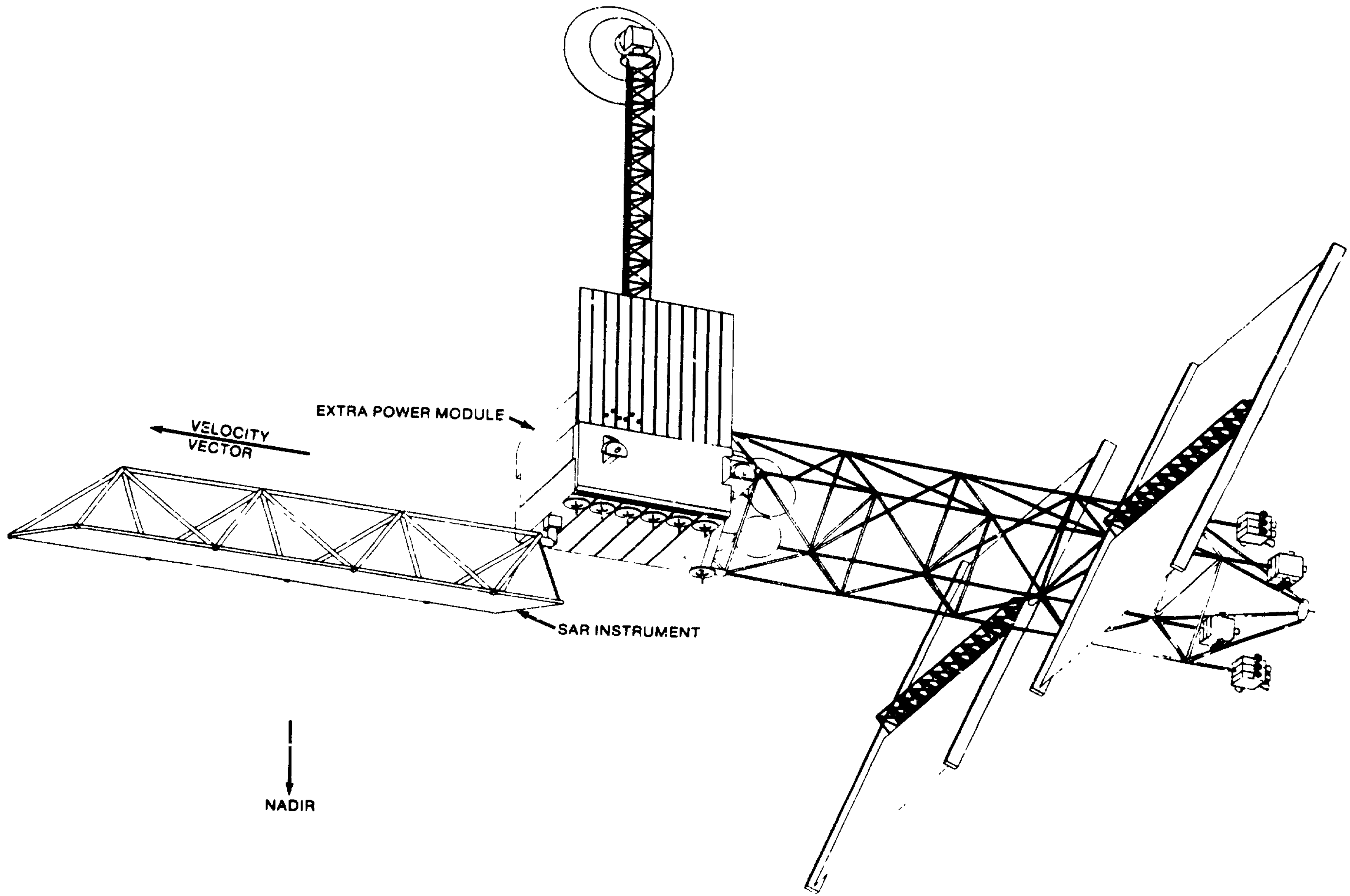


Figure 5.2-4 - SAR PAYLOAD PLATFORM

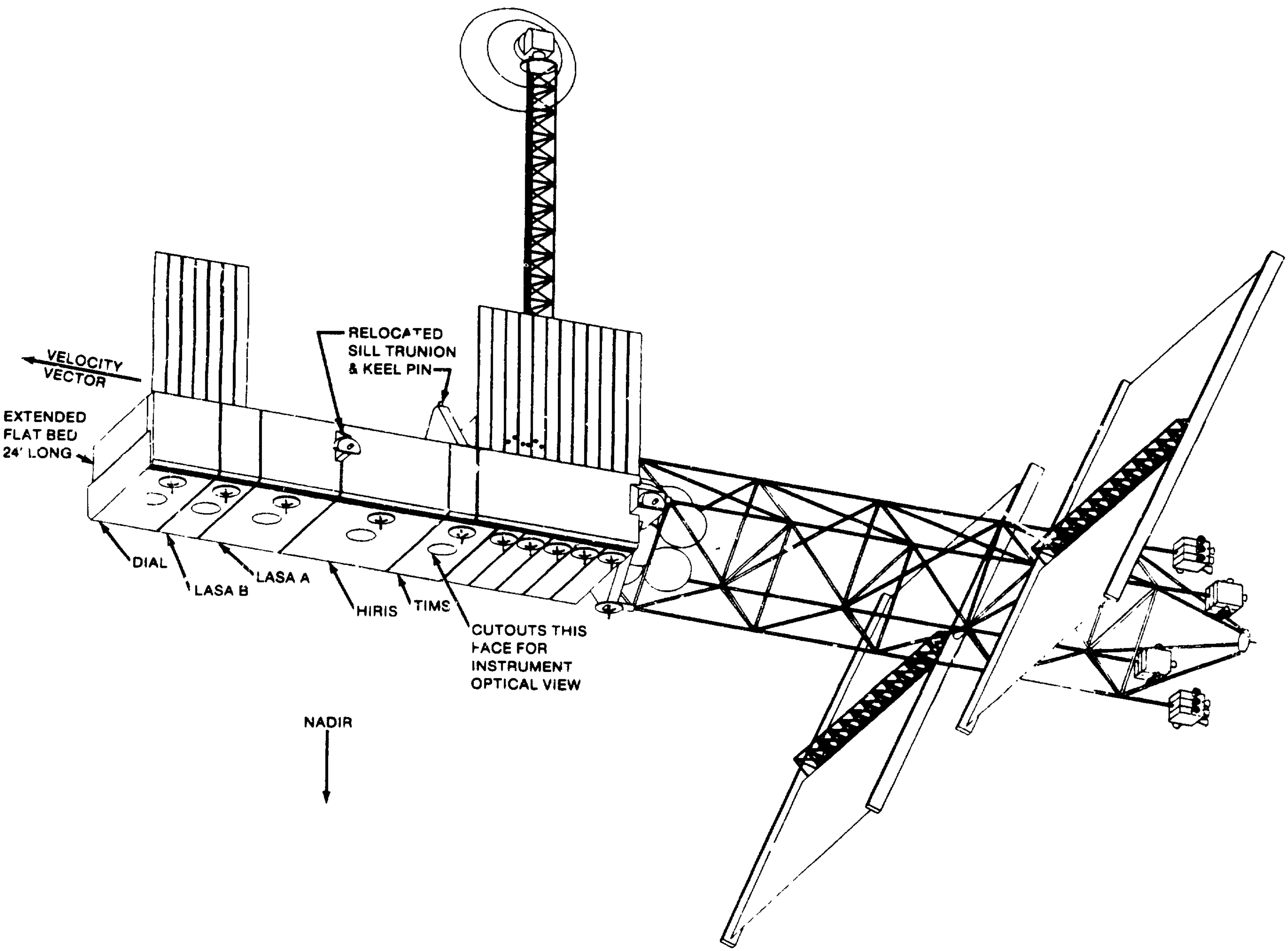


Figure 5.2-5 - POLAR PLATFORM PAYLOAD 6 WITH DIAL INSTRUMENT ADDED

The Platform Core provides data management and data links for all instruments. Engineering data on each subsystem's status will be included in the data stream.

The Platform Core is illustrated in Figures 5.2-1 and 5.2-2. The backbone of the Core is a flat bed structure. It provides the mechanical, electrical, and thermal interfaces to all Core subsystems. These subsystems are contained in orbital replaceable unit (ORU) modules. A bulkhead is mounted to one end of the flat bed. It provides mountings for the propulsion system fuel tanks and also carries the two Shuttle sill trunnions that handle ± x and ± z launch loads. Two additional sill trunnions that carry only ± z launch loads are mounted near the other end of the flat bed. The keel pin that carries all the ± y launch loads is mounted on the bulkhead (not visible in the figure).

On the side of the bulkhead away from the flat bed, a truss structure is deployed in orbit. This truss is built of the same elements as the manned Space Station. This structure carries the solar arrays and the propulsion thrusters.

The solar arrays are identical to the Space Station's arrays except for the initial configuration, where each half would be 29 feet long. A full 80-foot-long array could be used in the later, high power growth configuration Platforms.

The reference configuration uses a single degree-of-freedom for array articulation. This results in up to approximately a 30-degree cosine loss for the polar Platform arrays. The stellar and solar platforms do not suffer this loss.

For the polar Platforms, the array location was chosen to be 27 feet out on the deployed truss. This location gives (1) unobstructed instrument viewing of the Earth's limb, (2) minimum shadowing of the array by the body of the platform, (3) maximum view to cold space by the radiators, and (4) clear field-of-view for the startrackers.

For the stellar and solar viewing Platforms, the array location was selected to provide minimum aerodynamic and gravity gradient torques. This is illustrated in Figure 5.2-3 showing the Starlab Platform. One solar array is located 9 feet out on the deployed truss behind the bulkhead. The other array is deployed on a second truss 9 feet forward of the Platform Core and to one side. This arrangement does not interfere with the field-of-view of the star trackers or of the stellar or solar instruments. Shadowing of the arrays by the Platform is not a problem because the Sun is always normal to the array. The view to space by the radiators is slightly less but this is compensated for by the fact that the Sun never directly shines on any radiator surface. For the stellar and solar missions only, balancing weights and an additional balancing boom are required to maintain gravity gradient torque within allowable limits. The boom is not illustrated in Figure 5.2-3, but it would be located on the rear truss directly opposite the solar array. It would be made of the same deployable mast as used for the solar array and be the same length. The balance weights used fo analysis were: 1000 lbs on the end of the balance boom, 275 lbs added to the end of the solar array opposite the balance boom, and 1262 lbs added to the end of the boom that carries the TDRSS antenna.

The last 2 feet of the Platform Core flat bed is reserved for instrument mounting. This is adequate for several of the single large instrument platforms. Figure 5.2-4 shows the polar SAR instrument mounted in this location.

Multi-instrument payloads will require additional mounting space. This will be provided by utilizing additional extension flat bed structures attached to the end of the Core flat bed structure. All flat bed structures will incorporate standard instrument/ORU mounting provisions so that payloads can be replaced and serviced as well as transferred between Platforms and the Space Station mounting points with minimum difficulty. A maximum growth-sized Platform is envisioned to have three flat bed structures attached to the Core flat bed for a total of four. Figure 5.2.-5 illustrates Polar Platform with payload B including the growth DIAL instrument. This Platform would be launched with two additional flat beds and with instruments in all locations except the last 5 feet of the additional flat beds. For better launch loads distribution, the 2 $\pm$ z sill trunnions and the keel pin would be relocated from their normal positions on the Core flat bed to locations on the instrument flat beds. The DIAL would be added to the Platform on the available 5 feet of flat bed on a later Shuttle servicing mission. Another flat bed structure and more instruments could be added on a later Shuttle flight.

Each instrument module and ORU module will have built-in heat pipes coupled to a flat bed radiator system. It is anticipated that the same standard module design used by the Core ORU modules would also be used by the instruments. In order to standardize the flat bed mounting hard points, these modules would come in 1-foot width increments with a minimum width of 2 feet. All modules will be removable by a Remote Manipulator System. The figures show standard RMS grapple fittings on each module to illustrate this feature. The actual handling fittings will probably be much smaller and lighter.

5.2.2 Weight and Power

The properties of the reference Platforms which can accommodate the nine payloads discussed in Section 5.1.2 are summarized in Tables 5.2.2-1 and 5.2.2-2

Table 5.2.2-1 shows the estimated weight of the reference design Platforms. A common basic Core Platform was assumed for the design of all mission Platforms. The weight differences shown in Table 5.2.2-1 reflect for example, the different truss/solar array configuration for the polar and co-orbiting mission Platforms, the use of balancing weights on the Starlab Platform, and additional power capability required for the Polar SAR and Polar C mission payloads. Each near term mission payload can be accommodated in one launch. The ASO payload may have to be launched as two missions because of instrument size. The Polar C mission payload exceeds the estimated STS capability for polar launch. Both require further study.

Table 5.2.2-2 shows the estimated power requirements and the characteristics of the array and energy storage devices required to provide the required power. The Core Platform orbital average power estimate is 2896 W.

TABLE 5.2.2-1 - REFERENCE PLATFORM WEIGHT

MISSION	CORE PLATFORM LB	PROPELLANT LB	INSTRUMENTS LB	FLAT BED INSTRUMENT STRUCTURE LB	TOTAL PLATFORM LB
NEAR TERM:					
SIRTF	11,902*	5,700	8,820	0	26,422
STARLAB	14,439	2,900	7,056	0	24,395
P/OF	11,902*	2,900	7,938	0	22,740
POLAR A	11,795	5,700	5,456	4,600	27,551
POLAR B	11,795	5,700	8,140	4,600	30,235
POLAR SAR	11,995	5,700	2,508	0	20,203
LATER:					
ASO	11,902*	2,900	19,625	0	34,427
HTP	11,902*	2,900	22,050	0	36,852
POLAR C	12,395	5,700	10,037	4,600	32,732

* Required inertia balancing weights for these missions have not been identified.

TABLE 5.2.2-2 - REFERENCE PLATFORM POWER

MISSION	PEAK PLATFORM BUS POWER, W	ORBITAL AVERAGE POWER, W	MINIMUM ARRAY AREA, SQ FT	MINIMUM BATTERY WEIGHT, LB
NEAR TERM:				
SIRTF	4,895	3,921	882	1,074
STARLAB	6,796	5,308	1,194	1,341
P/OF	4,896	3,347	742	752
POLAR A	7,391	7,391	1,722	1,629
POLAR B	9,946	6,673	1,582	1,788
POLAR SAR	19,596	5,818	1,420	2,079
LATER				
ASO	8,896	5,637	1,205	809
HTP	5,896	5,021	1,130	1.241
POLAR C	10,381	10,292	2,399	2,265

5.2.3 Flight Modes

5.2.3.1 Polar Platform

In normal operations, the polar Platform will be in a Sun-synchronous orbit with a 2 p.m. descending node and a 2 a.m. ascending node. The + z axis will be aligned with the local vertical (i.e., Earth-viewing), the y axis will be normal to the orbit plane (POP), and the + x axis will be aligned along the direction of flight as shown in Figure 5.2.3-1. Reaction wheels are the prime Platform actuators to control all external disturbance torques. Magnetic torquers will be used for reaction wheel desaturation as resulting from the accrual of momentum due to secular disturbance torques.

The solar array is maintained such that the normal is aligned at a nearly constant 30° angle with the Sun vector.

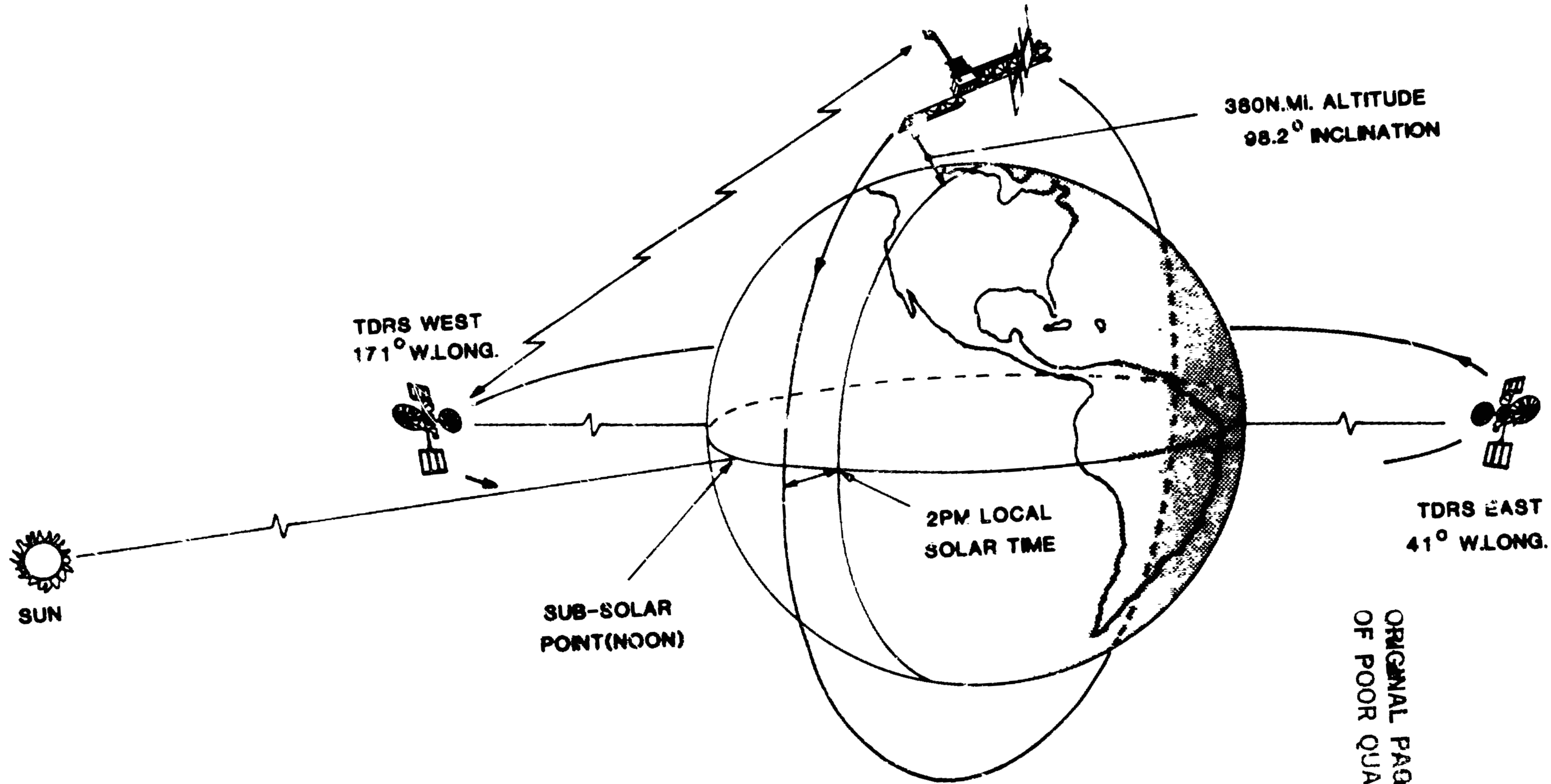


Figure 5.2.3-1 - SYNTHETIC APERATURE RADAR POLAR PAYLOAD PLATFORM

All Platforms will be launched in a fully configured mode (i.e., core with instruments). Therefore, the initial flight mode will remain unchanged throughout the Platform's lifetime, notwithstanding the later enlargement of a Platform by the addition of new instruments. The only occasion for an interruption in the normal flight mode will be during a servicing operation when the polar Platform will be deboosted to a lower altitude for rendezvous with the STS.

5.2.3.2 Co-Orbiting Platform

In normal operation, the co-orbiting Platform will be flown at the same inclination as the manned Space Station. With only a few exceptions (e.g., SIRTF), the flight altitude will be the same as the manned Space Station (i.e., 270 nmi). The orientation will be in one of two inertial modes, stellar or solar (Figures 5.2.3-2 and 5.2.3-3). The + x axis will point inertially to the observation object in space. The y and z axes will be oriented, in conjunction with solar array rotation, such that the Sun vector is normal to the solar array. As a result, the + y axis faces of a co-orbiting Platform, which are radiating directions of the fixed radiators, will have no direct solar input.

As explained for the polar Platform, the initial flight mode will remain unchanged except when interrupted by a servicing operation. For the co-orbiting Platform, one servicing operation mode will entail transfer to, and berthing with, the manned Space Station.

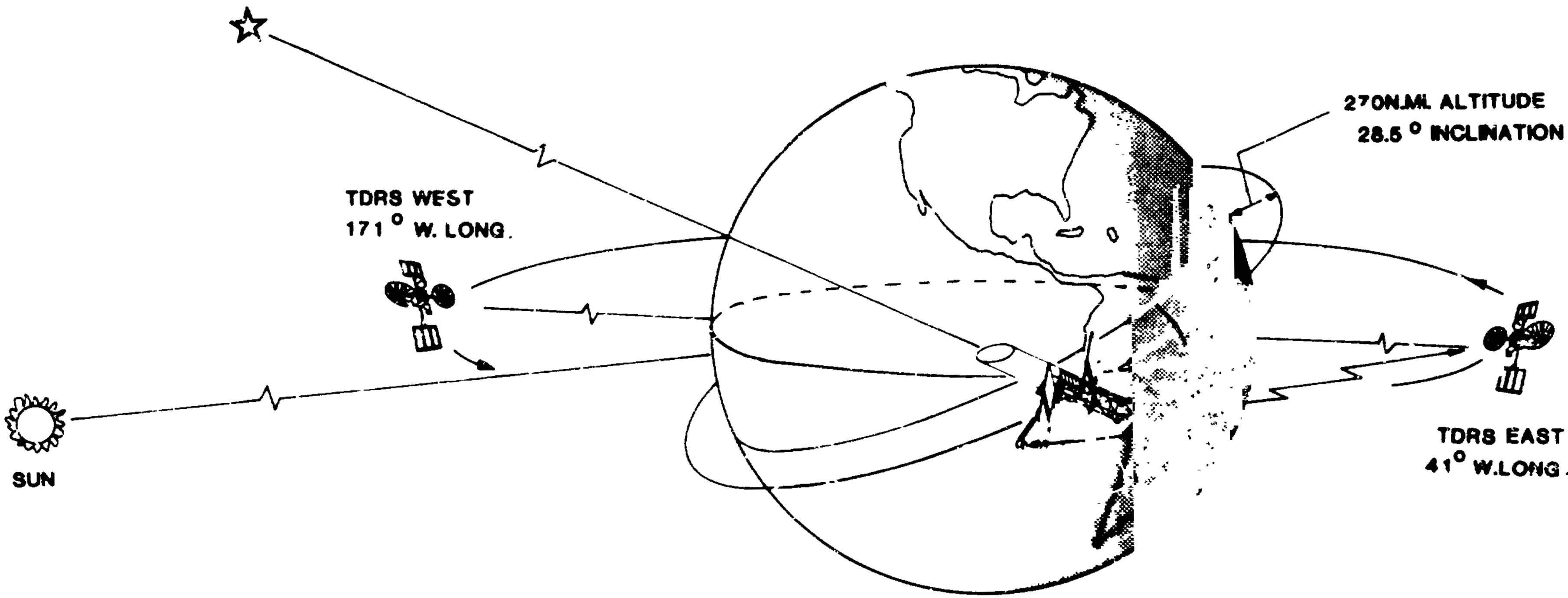


Figure 5.2.3-2 - STARLAB CO-ORBITER PAYLOAD PLATFORM

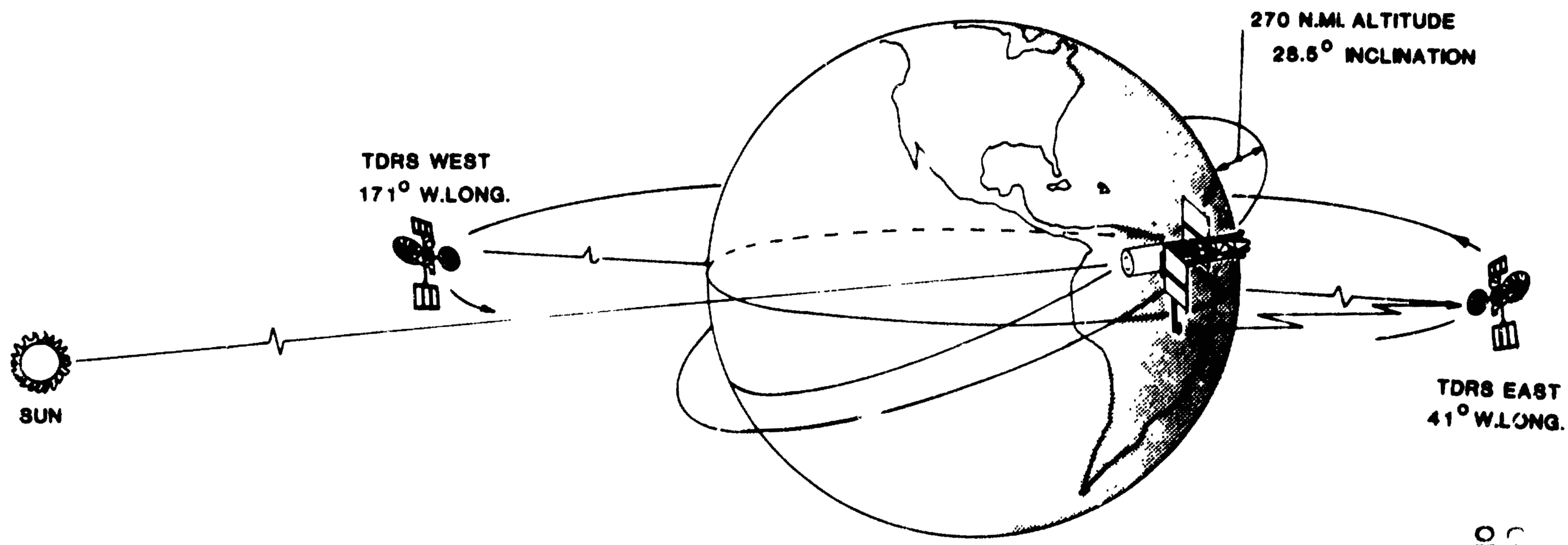


Figure 5.2.3-3 - ADVANCED SOLAR OBSERVATORY CO-ORBITER PAYLOAD PLATFORM

5.2.4 Launch and Serving Operations

5.2.4.1 Launch Operations

Co-orbiting Platforms will be launched from the Eastern Test Range (ETR) and polar Platforms from the Western Test Range (WTR). All reference polar Platforms and the reference co-orbiting Platform at 378 to 540 nmi (SIRTF) require orbit transfer from a lower initial launch altitude. Assumed STS payload launch capability in the early 1990's is

ETR - 28.5°, 270 nmi = 58,400 lb
378 to 540 nmi = Not attainable (payload transfer from low orbit required)
150 nmi = 72,000 lb
WTR - 98.2°, 380 nmi = Not attainable (payload transfer from low orbit required)
150 nmi = 30,300 lb

If the above assumed payload launch capabilities are determined to be too high, the heavier mission payload weights can be reduced by removing selected instruments.

NOTE: The projected STS performance capabilities (launch and landing payload weights) may be in conflict with the above numbers. The baseline STS performance information to be used during the Definition Phase studies will be provided via the associated SSP/STS Payload Integration Plan which will be a RFP applicable document.

A transfer propulsion subystem is included in the reference design, with the OMV providing an alternate approach to orbit transfer. For those Platforms which operate at altitudes exceeding the STS operational capability, the transfer propulsion subsystem will be used to lower the Platform for rendezvous with the STS for servicing. After servicing and refueling, the transfer propulsion subsystem will re-boost the Platform to operational orbit until the next servicing operation is required. The transfer propulsion subsystem will contain orbit adjust propellant for a minimum of 2 years between service intervals plus 1 year reserve (total 3 years).

When a growth Platform has exceeded the total propulsion capability of the Core tanks, several alternatives are possible: (1) an additional propellant tank(s) could be added, (2) the OMU could provide part of the delta velocity, and/or (3) the Shuttle could go to a higher altitude if the servicing weight was less than shown in Table 5.2.2-1.

5.2.4.2 Servicing Operations

There are several options available for Platform servicing. The polar Platforms may be serviced in-situ by the STS or the OMV. The co-orbiting Platforms may be serviced in-situ by the OMV or the STS, on the Station after berthing, or within the proximity operational zone of the Station

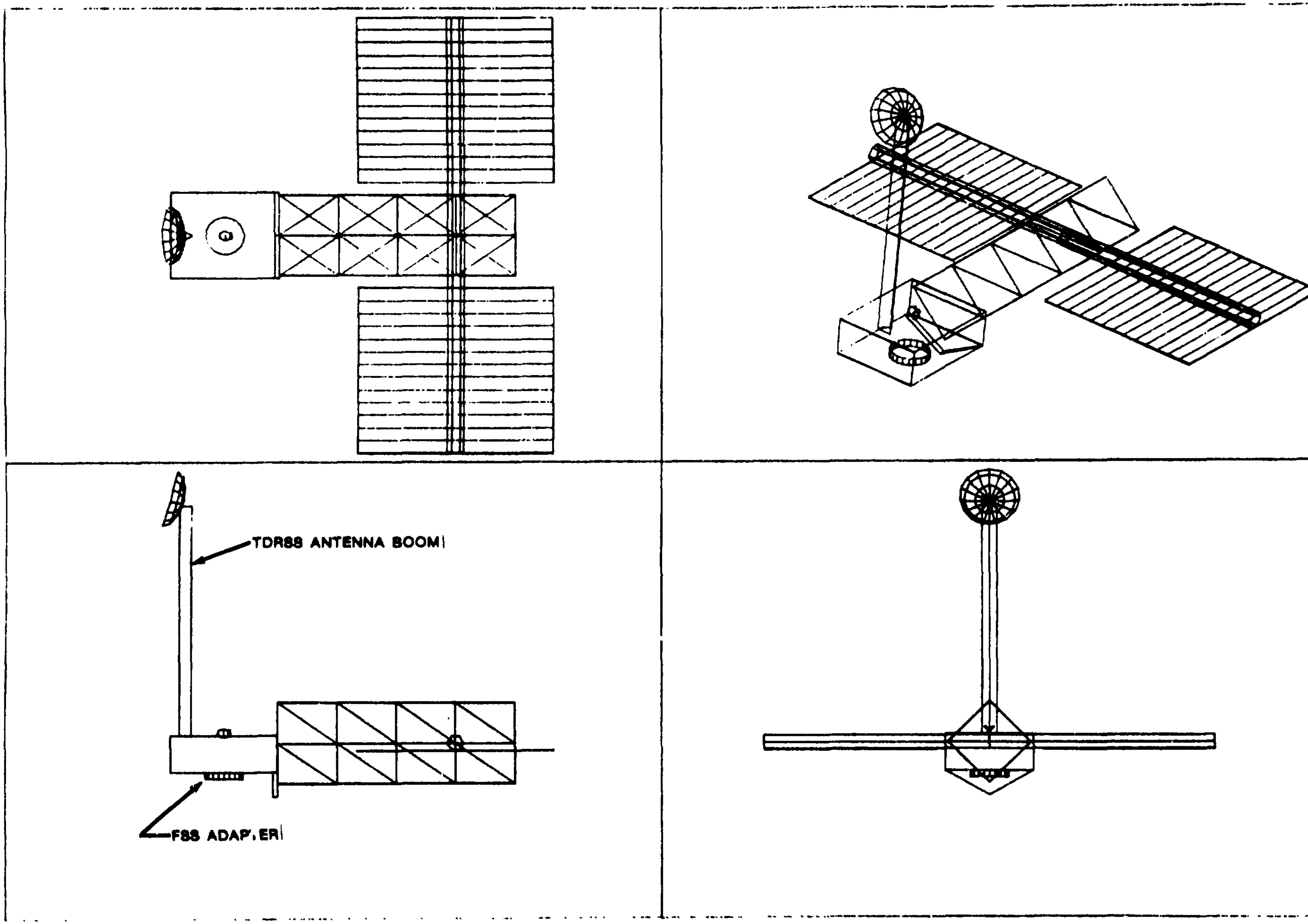


Figure 5.2.4-1 - PLATFORM CONFIGURATION

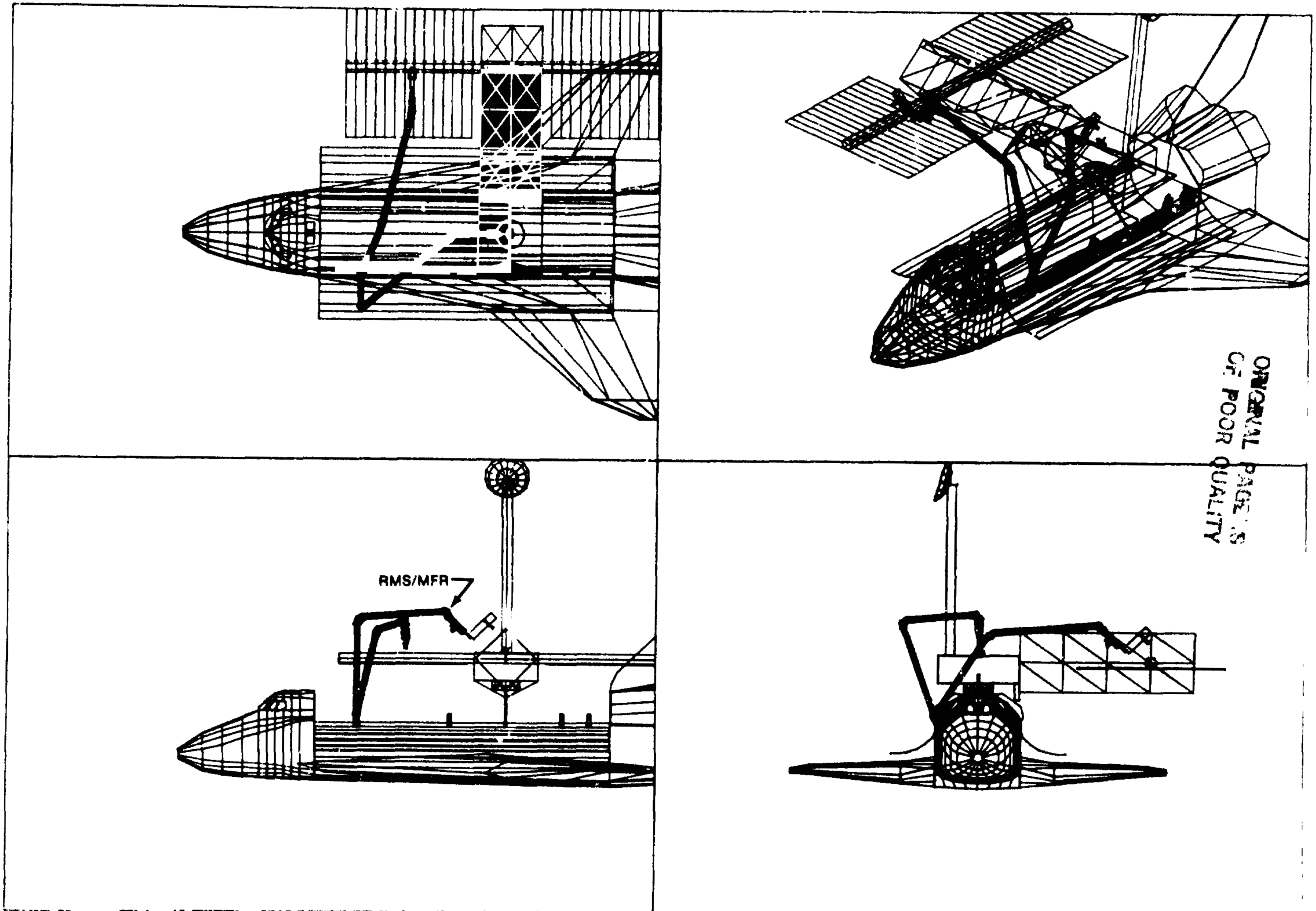


Figure 5.2.4-2 - PLATFORM SERVICING FROM THE ORBITER

by EVA crew. Each option has operational advantages and disadvantages. The Platform configuration evaluated is shown in Figure 5.2.4.-1.

5.2.4.2.1 Servicing by STS/OMV

5.2.4.2.1.1 Polar Platform

In-situ servicing by the STS/OMV will have to be developed to support the polar Platform. Routine servicing of the polar Platform could be performed in its orbit by an OMV brought up by the STS. For this option the OMV must have basic servicing capabilities, such as consumables replenishment, payload changeout, and possibly limited payload repair capability. For service operations beyond the OMV robotics capability, it will be necessary to retrieve the polar Platform and berth it in the STS for EVA/RMS repair/service. The Platform would be deboosted to a lower orbit (either by use of its own propulsion system or by use of the OMV) for an STS rendezvous. In general, the Platform propulsion system will be used to deboost to a lower orbit prior to STS launch, so that the STS may rendezvous with the Platform. If the OMV must be used for deboost, then a double rendezvous is required. The STS would be launched in a compatible orbit, the OMV would rendezvous with and retrieve the Platform, and then the STS would re-rendezvous with the OMV/Platform.

Of major concern in servicing the Platform from the Orbiter is capturing it with the RMS. The deployed appendages on the Platform are a definite hindrance to capture with the RMS. If these appendages, the TDRSS antenna and mast in particular, cannot be retracted, then grapple fixture location and track and capture procedures become major drivers in servicing. This is also true for capturing the Platform at the station with the MRMS. Once the Platform is captured and placed in a surrogate payload bay fixture above the payload bay as shown in Figure 5.2.4-2, the RMS/MFR have adequate access to all parts of the Platform.

5.2.4.2.1.2 Co-Orbiting Platform

Two in-situ servicing options exist for the co-orbiting Platform. The first using the STS, and the second the OMV. Although STS schedule impact is the main disadvantage, using the STS for co-orbiting Platform servicing is a viable option. It is conceivable that a servicing operation could be combined with an STS satellite deployment mission if the orbits are compatible. It is also conceivable that a Platform servicing mission could be combined with a station resupply visit, except that the STS will contain a logistics module on both the up and down legs of the mission, plus the extra crew aboard for the crew rotation.

The second in-situ servicing option for the co-orbiting Platform consists of sending the Space Station based OMV to the Platform location. The OMV will have certain capabilities developed for polar Platform servicing which could be applied for co-orbiting Platform servicing. Even though the Platform is initially co-planar, differential drag and nodal regression between the Station and Platform may require considerable

Platform propulsive capability to maintain the Platform location relative to the Station to meet the servicing interval requirements. Despite these constraints, the Platform should be accessible to the OMV.

If the Platform is placed at 28.5° inclination at altitudes other than the Space Station, then differential nodal regression will force the servicing intervals to be periodic, since the Platform fuel supply is not sufficient for large nodal changes. For instance, a 108 nmi altitude difference will force the servicing intervals to be no more often than 21 months.

5.2.4.2.2 Servicing by Space Station

Utilization of the Space Station for Platform servicing offers two candidate schemes based on either direct on-Station or Station proximity zone operations.

5.2.4.2.2.1 On-Station operations

Several options for Platform servicing on the Space Station have been identified. If the Platform is designed such that its solar arrays and antenna boom are retracted when the Platform arrives at the Station, the existing satellite servicing bays may be used for Platform servicing if no more than two additional flat beds or their equivalent are added to the Core configuration. Retention in the bay can be accomplished either by surrogate payload by a structure grasping the trunnions used to launch in the STS or by an FSS with an adapter attached to the Platform.

If the boom and/or arrays cannot be retracted, the servicing bays can be used only if the Platform is rotated away from the Station keel. Provision for a servicing position on the -x side of the keel would permit boom and arrays to remain deployed.

Satellites to be serviced will be captured by the MRMS at the lower end of the Station and transported to the servicing bay or other location. A surrogate payload bay system is mounted on the MRMS in such a way as not to interfere with MRMS operations, as indicated in Figure 5.2.4-3. Because of the limited area available on the base for such attachments, it is possible that an FSS or equivalent, possibly mounted on stand-offs, will be required. Further study will be needed to evaluate these options.

Several locations on the reference Platform configuration for mounting an FSS adapter have been evaluated. The first location is at the end of the flat bed. This has the advantage of similarity to other satellites to be serviced, so that few special procedures will be required. The manipulator would have ample clearance for mating and demating the Platform. In this option, adding flat bed or other payloads to the Platform is not possible without removal of the adapter from the Platform. However, the servicing bays will have surrogate payload bay structure to handle satellites that do not incorporate an FSS adapter. Use of this capability to retain the Platform in the bay would permit addition of flat beds to the Platform.

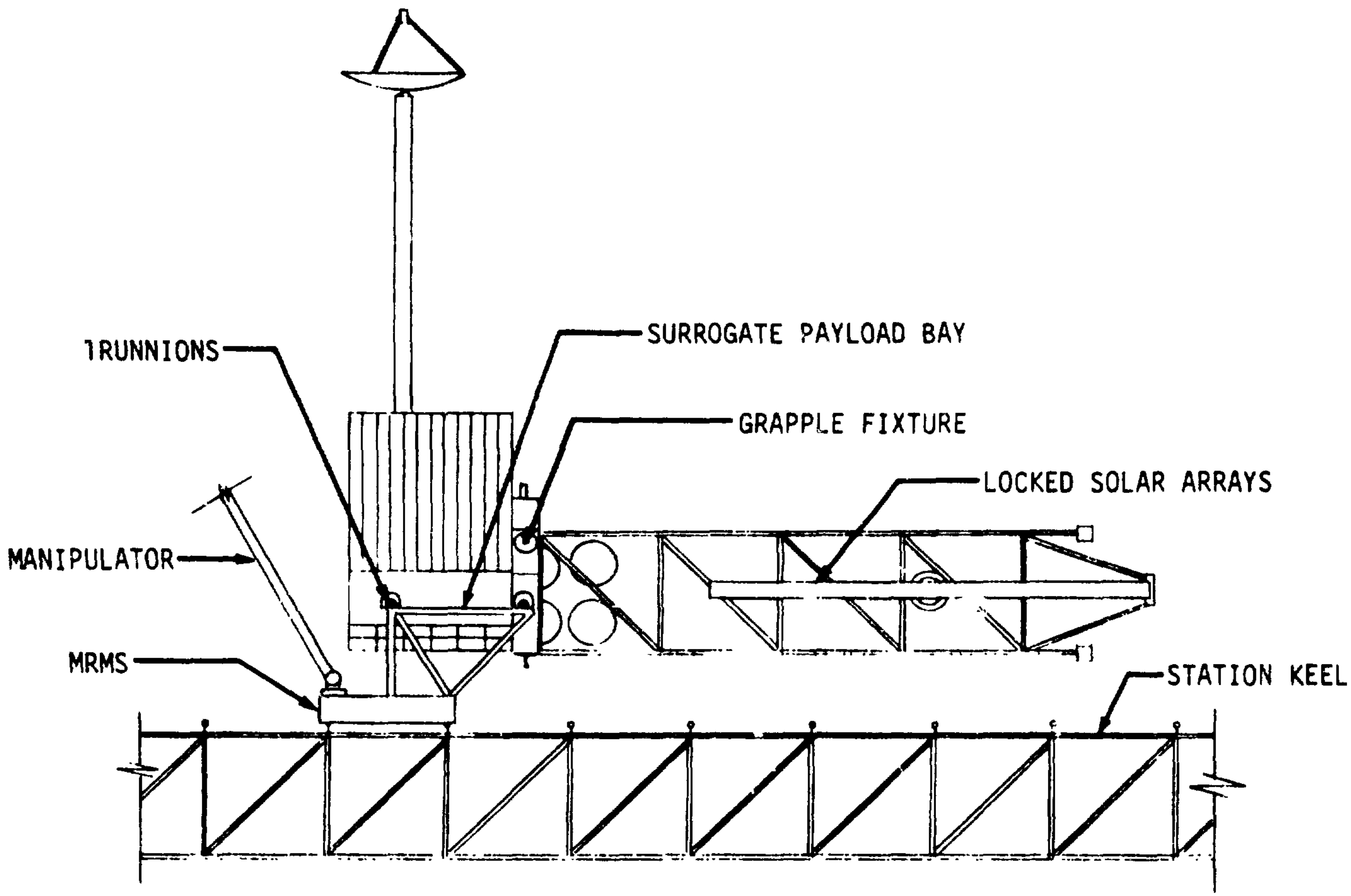


Figure 5.2.4-3 - PLATFORM ON MRMS

The second FSS adapter location studied is on the end of the Platform near the RCS. This location avoids the problem of FSS removal for flat bed extensions and can be designed for compatibility with the FSS.

As the Platform configuration becomes better defined, a more detailed analysis of grapple fixture locations for RMS/MRMS handling will be necessary. Detailed investigation of accessibility to specific areas of the Platform using both RMS/MFR and MRMS/MFR will also be required.

5.2.4.2.2.2. Station proximity zone operations

The option of bringing the Platform into the proximity operations zone of the Station for servicing by EVA crew has been considered. The proximity operations (PROX OPS) zone is a 1 km sphere centered at the Station. The Platform propulsion system or OMV would be used to bring the Platform to the rendezvous zone. For safety, the OMV would be used to finally bring the Platform into the PROX OPS zone, since the OMV will be designed with redundant sytems, possibly auto rendezvous, and man-in-the-loop control capability. The Platform propulsion or the OMV must have the capability to retrieve and bring the Platform to the proximity of the Station, and the Platform must be maintained in an orbital position relative to the Station that meets the servicing interval requirements. The main advantage of this option is the accessibility of EVA crew to the Platform. The OMV capability designed for polar Platform servicing could also be used. The operation would be performed at a stand-off position of several hundred meters because the safety risks associated with close quarter station-keeping (say a few tens of meters), possibly with EVA crew in the vicinity. Even at several hundred meters, the Platform must station-keep with the Station. Platform attitude can be controlled with the onboard reaction wheels, but translation maneuvers still must be made using the Platform propulsion system. EVA crew should not be in the vicinity of any thruster firings. In order to minimize orbital mechanics effects, the Platform should be placed on the V-bar in the PROX OPS zone. This might allow shutting-down the Platform thrusters for periods of free drift during which the EVA crew could perform servicing. The period of free drift is dependent on many variables, and studies would have to be performed to see if it is even feasible. In addition, lighting must be provided for the EVA operations. Finally, EVA time is limited, so that servicing operations must be carefully scheduled. If an operation should take longer than EVA time constraints, the Platform could be moved to a stand-off position of several miles for re-rendezvous later. Considering the safety implications, the flight mechanics problems, and the scheduling complications, proximity servicing does not look promising.

5.2.5 Dynamics and Control

A limited amount of analysis of the Platform has been performed. Work has been focused on three areas: rigid body control requirements, reboost control by the RCS, and uncontrolled flexible response. Two combinations of the Platform Core with payloads have been analyzed. The results presented are intended to provide insight as to where further efforts should be concentrated.

5.2.5.1 Disturbance Forces and Torques

A list of applied forces and torques is given in Section 4.3.3.1 for the IOC Manned Core Station. Obviously, not all the items listed are applicable to the Platform. Of the natural environments, gravity gradient torques and aerodynamic forces and torques were considered. Both are characterized in Section 4.3.3.1. Induced environments studied were RCS reboost and berthing which are characterized in the following sections.

5.2.5.2 Reaction Control System

The RCS control system used to maintain attitude control during the reboost maneuver is described in section 4.3.3.2.2. Using the Polar B configuration and the autopilot data given in Table 5.2.5-1, the autopilot hysteresis (h) was parameterized to determine its effect on limit cycle frequency. Using a value of 0.5 degrees for hysteresis, Figure 5.2.5-1 shows the torque acting on the Platform as a function of time. Figure 5.2.5-2 illustrates the Platform limit cycle. The torque history was used to drive a structural dynamics response model of the Platform, reported on and discussed in Section 5.2.5.2.

TABLE 5.2.5-1.- AUTOPILOT DATA VALUES

KD	=	1.0 degrees/ (degrees/sec)
DB	=	1.0 degree
h	=	0.0 to 1.0 degrees
T2	=	583.0 ft-lbs
TL	=	184.0 ft-lbs
I	=	225876 slug-ft^2

5.2.5.3 Structural Dynamics

A NASTRAN finite element model was constructed (Fig. 5.2.5-3) for the Platform configuration of Figure 5.2-5. The Core body was modeled as a common element of the Space Station central keel. This structure is a box beam of 9-foot-square cross-section consisting of graphite/epoxy tube material. The beam properties for this section were extracted directly from the Space Station finite element model. The instrument section of the core body was modeled as a lumped mass and inertia charateristics

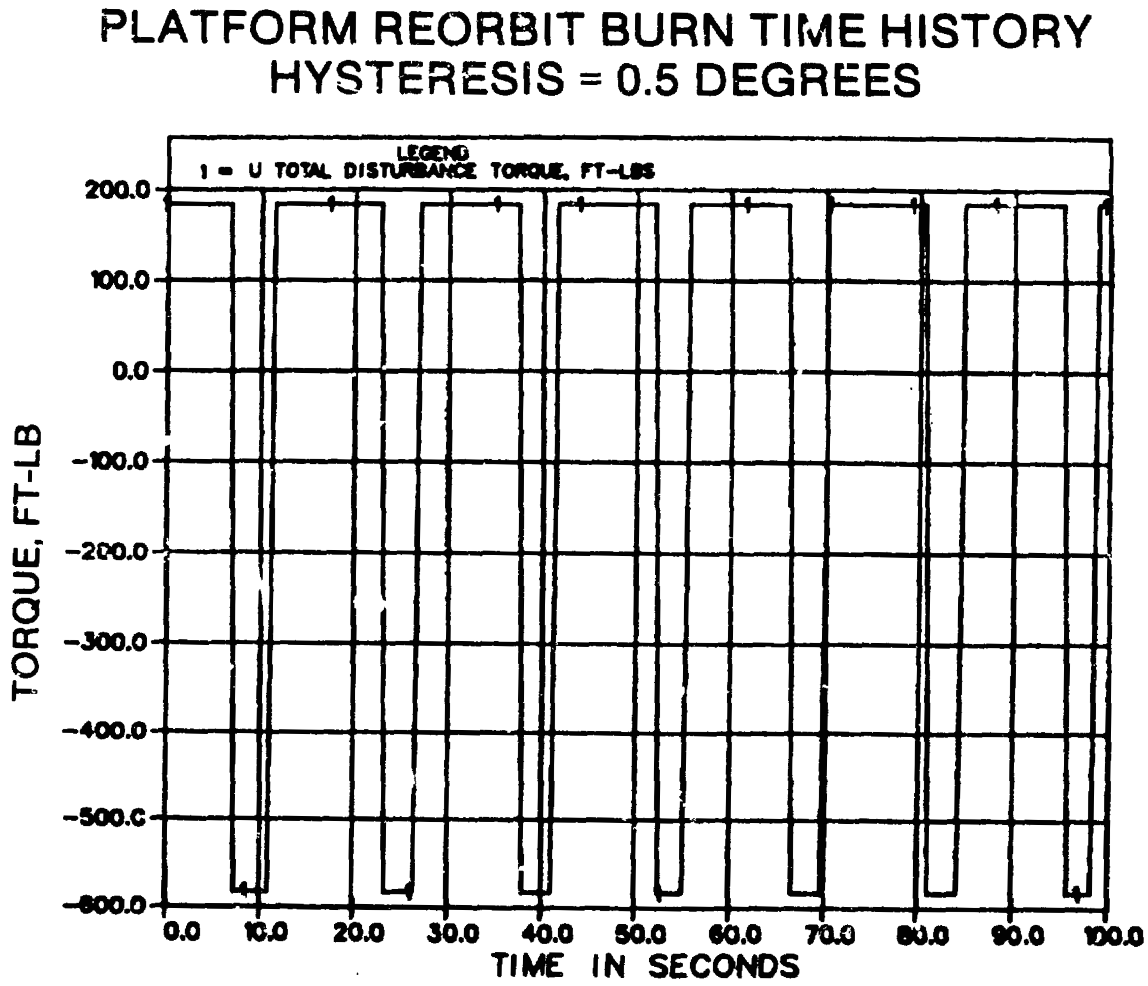


Figure 5.2.5-1 - RCS TORQUES

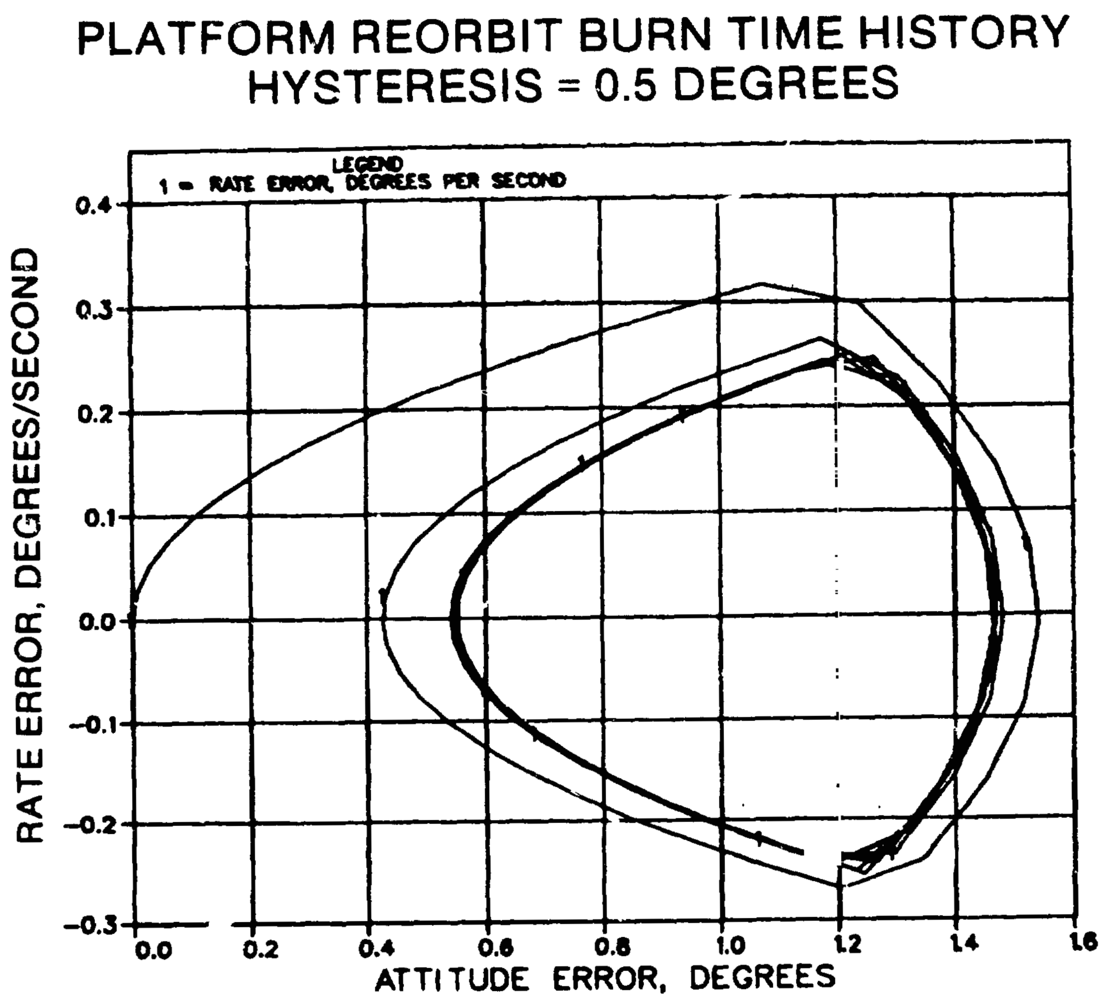


Figure 5.2.5-2 - PLATFORM PHASE PLANE

SPACE STATION PLATFORM MODEL

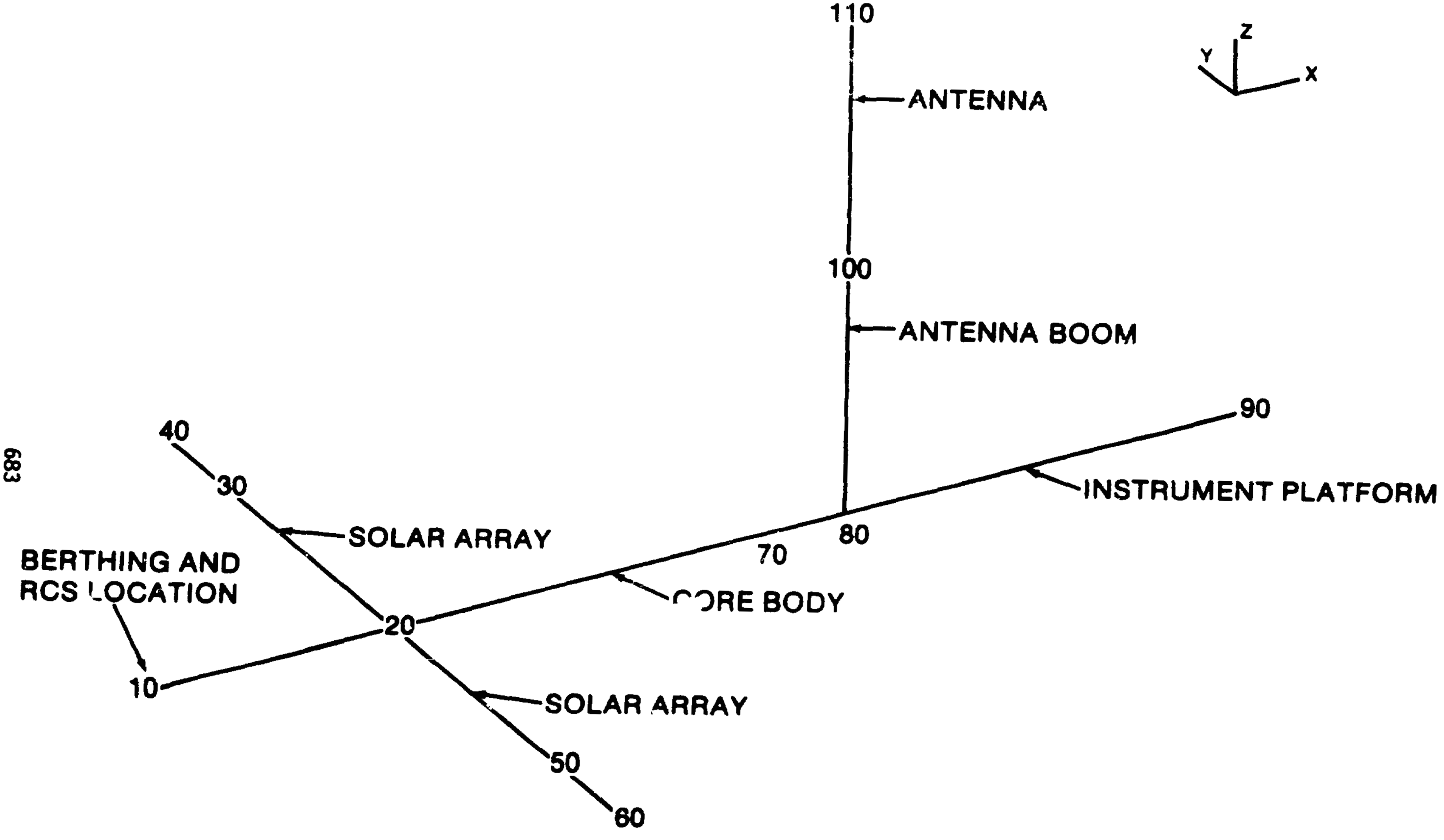


Figure 5.2.5-3 - FINITE ELEMENT MODEL

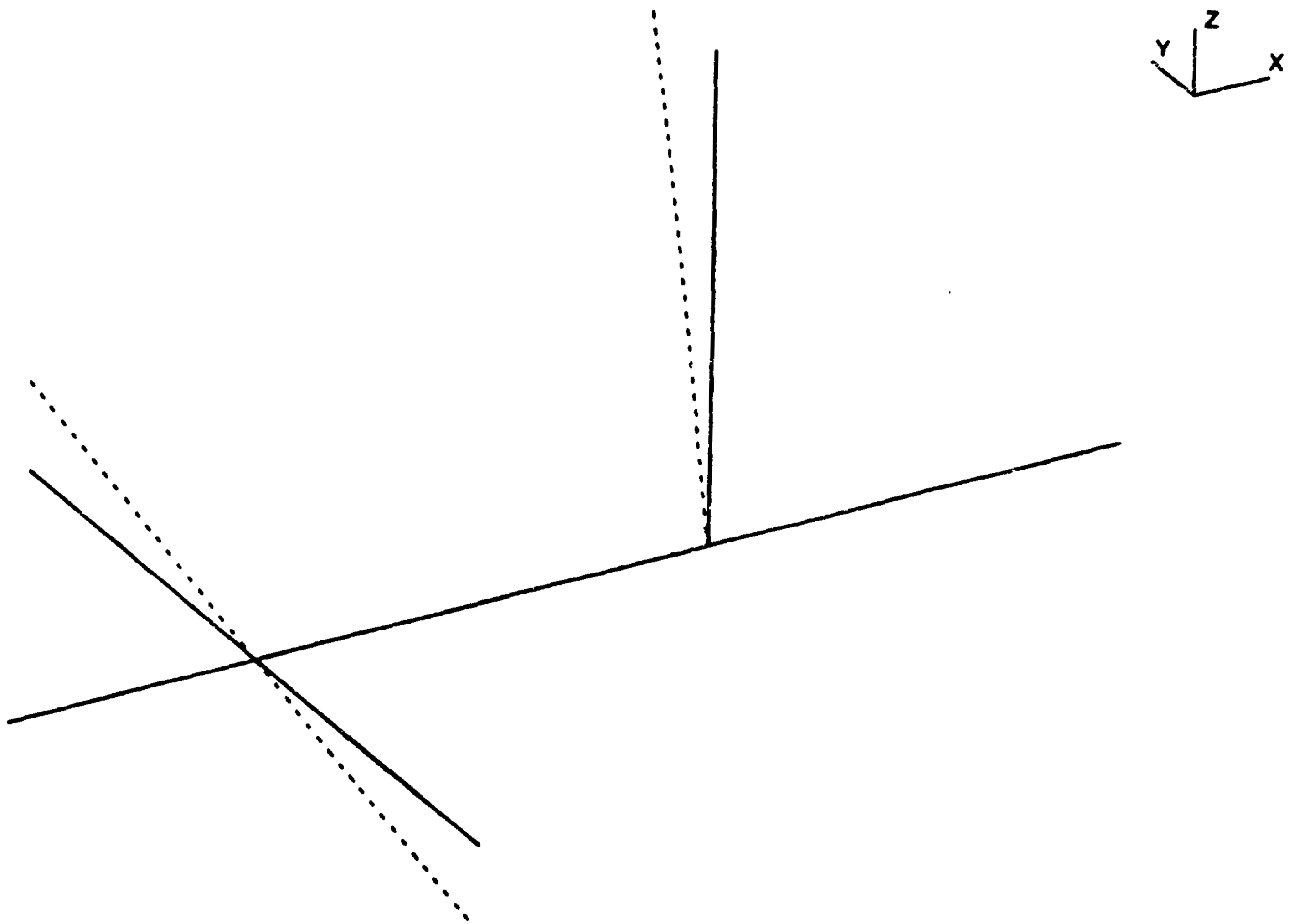
at the appropriate center of mass. The solar arrays were also modeled as common Space Station elements utilizing the structural characteristics from the Space Station finite element model. The array inertia characteristics were lumped at the array center of mass located on the structural support boom. The antenna boom consists of support structure similar to the array support boom with a concentrated mass at the top to simulate the antenna. The Platform structural model does not include a berthed Orbiter condition. This effect is not insignificant and should be included in future studies. Vibration modes were assumed to exhibit 0.5 percent of critical damping. The berthing port and reboost thrusters are located at the end of the Core body close to the solar arrays. This end of the model is the application point for the berthing and reboost forcing functions.

The finite element model was used to characterize the structural dynamics of the Platform in terms of mode shapes and corresponding vibration frequencies (Figs. 5.2.5-4 through 5.2.5-10). Results of this analysis indicate a low frequency torsion mode at 0.28 Hz for the Core body. The bending of the Core body occurs at approximately 0.33 Hz. Significance of these modes comes from the displacement of the solar arrays, the antenna boom, and the instrument Platform which participate in these low frequency modes. Solar array torsion is predicted at slightly above 0.33 Hz; array bending with the Core body occurs at 0.63 Hz, and as a cantilevered beam at approximately 1 Hz. The antenna boom also participates in the 0.63 Hz body mode and also exhibits bending at 0.75 Hz. The tip mass of the antenna appears to have significant impact on the dynamics of the antenna support structure. The effect of payloads on the dynamics of the Platform will depend on the individual mass of the payload, but will serve to reduce the frequencies of the overall system.

Forces acting on the Platform while on-orbit include, but are not restricted to, Orbiter berthing and orbit reboost. These two conditions are the forcing functions utilized in the analysis set. The Orbiter berthing force is assumed to be identical to the force used in Space Station analyses and consists of a 1-second square wave with a magnitude of 500 pounds (Fig. 5.2.5-11). The reboost forcing function was generated by a Platform model that includes on-orbit disturbances and a flight control system. Various control system parameters were investigated to arrive at a control input which seemed most realistic. Varying control deadbands have a significant impact on frequency content of the forcing function. For systems of low frequency, caution must be exercised in the selection of these parameters such that the control input is not resonant with the Platform structural dynamics.

Orbiter berthing transient response analysis (Figs. 5.2.5-12 through 5.2.5-23) indicates acceleration levels of 0.016 g's at the Core body in the direction of the applied force. Dynamic amplification is responsible for acceleration levels of 0.032 g's at the solar array tip. The maximum response of the Platform occurs during the berthing force application and diminishes rapidly in the Core body after force termination. The

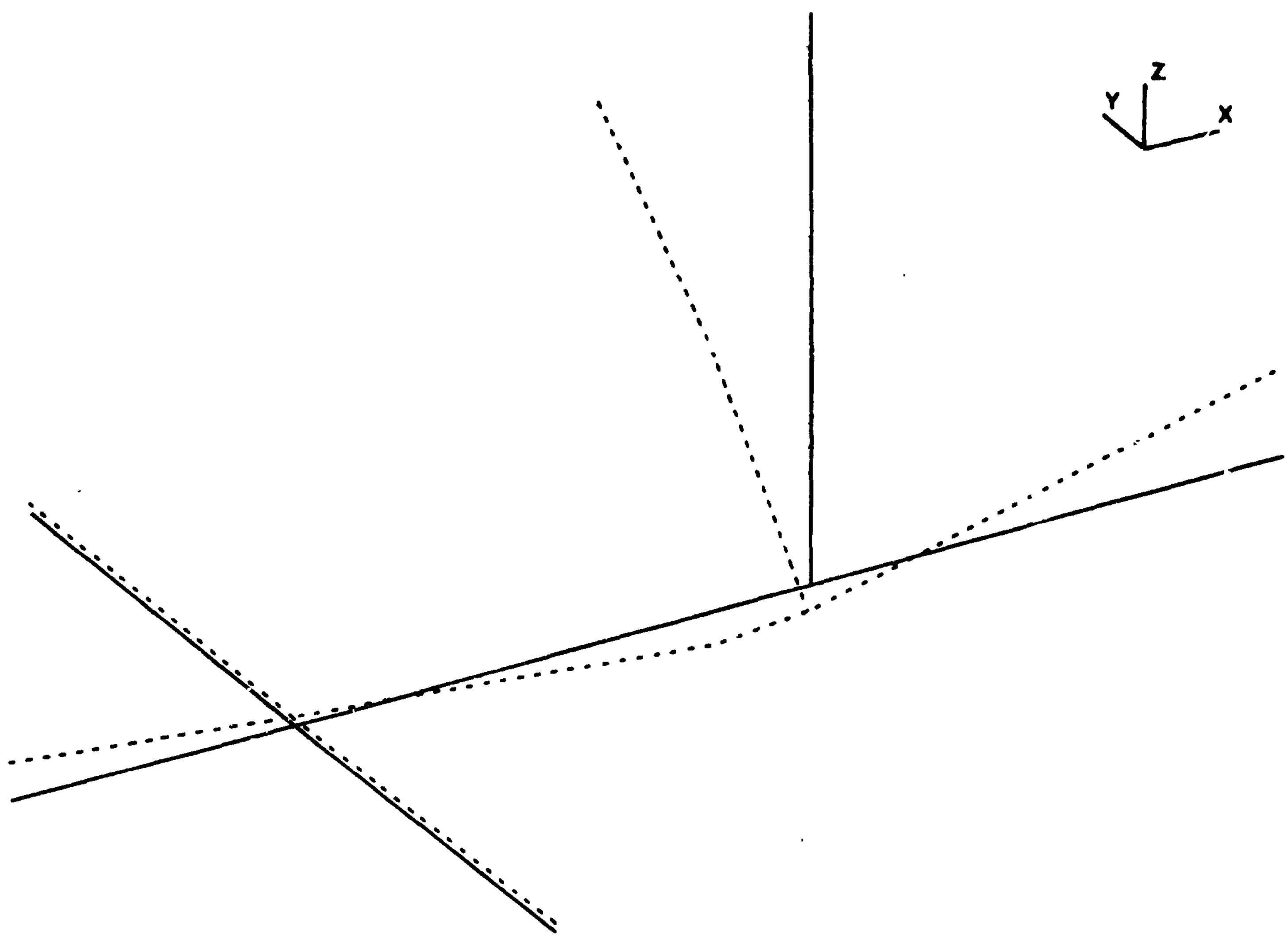
SPACE STATION PLATFORM MODEL



FREQ (1) - .28 HZ

Figure 5.2.5-4 - MODE 7

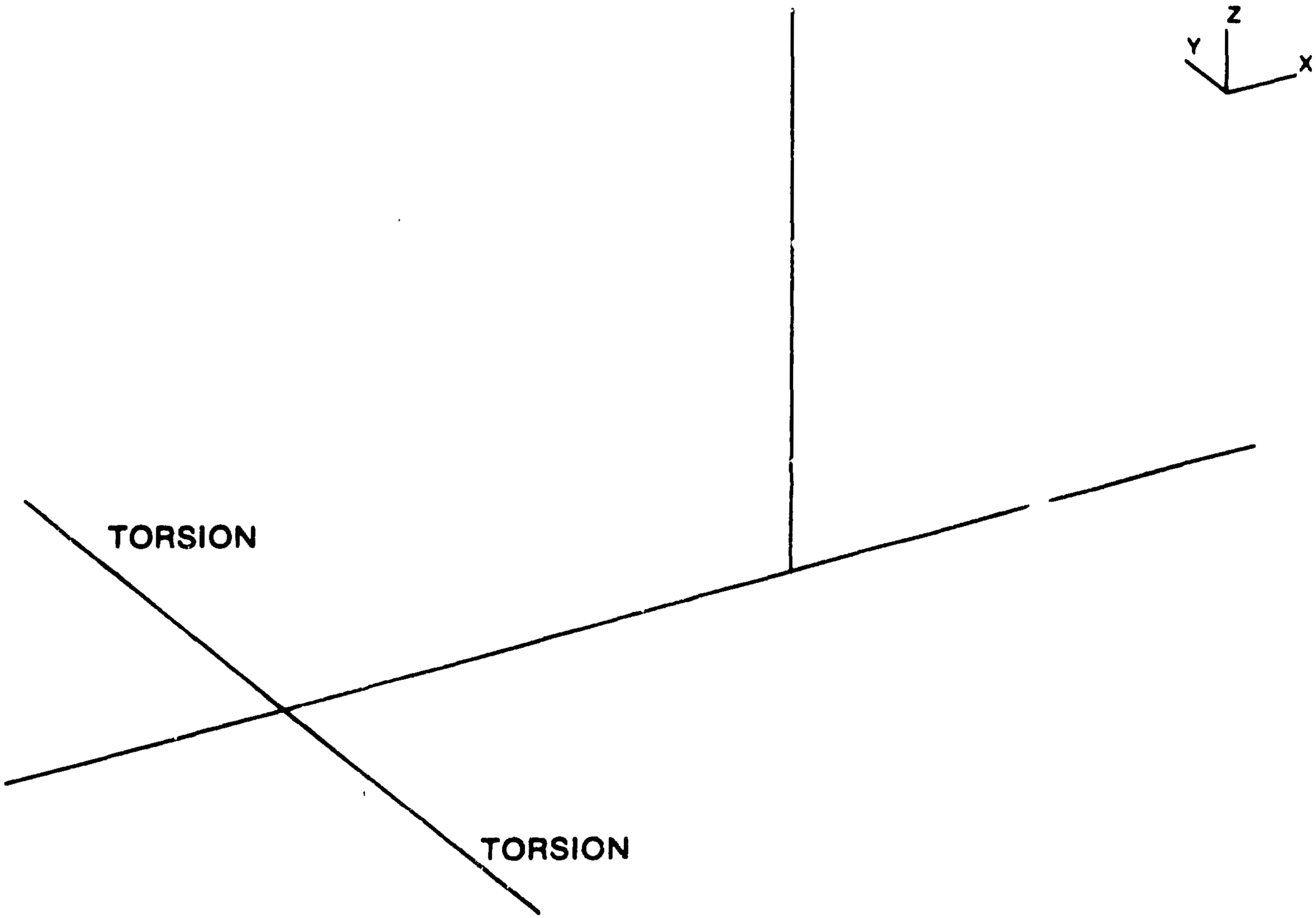
SPACE STATION PLATFORM MODEL



FREQ (2) - .33 HZ

Figure 5-5 - MODE 8

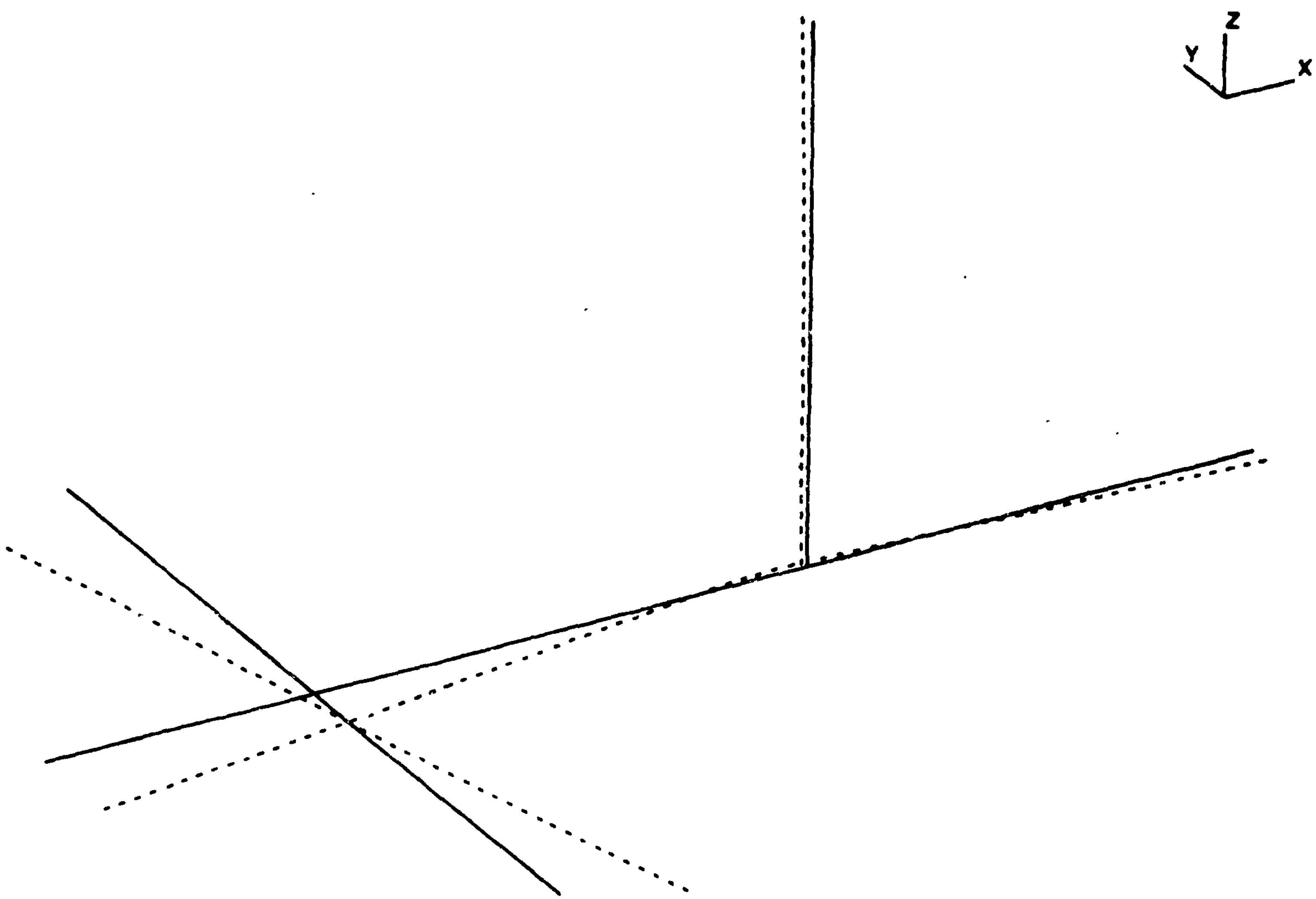
SPACE STATION PLATFORM MODEL



FREQ (3) - .33 HZ

Figure 5.2.5-6 - MODE 9

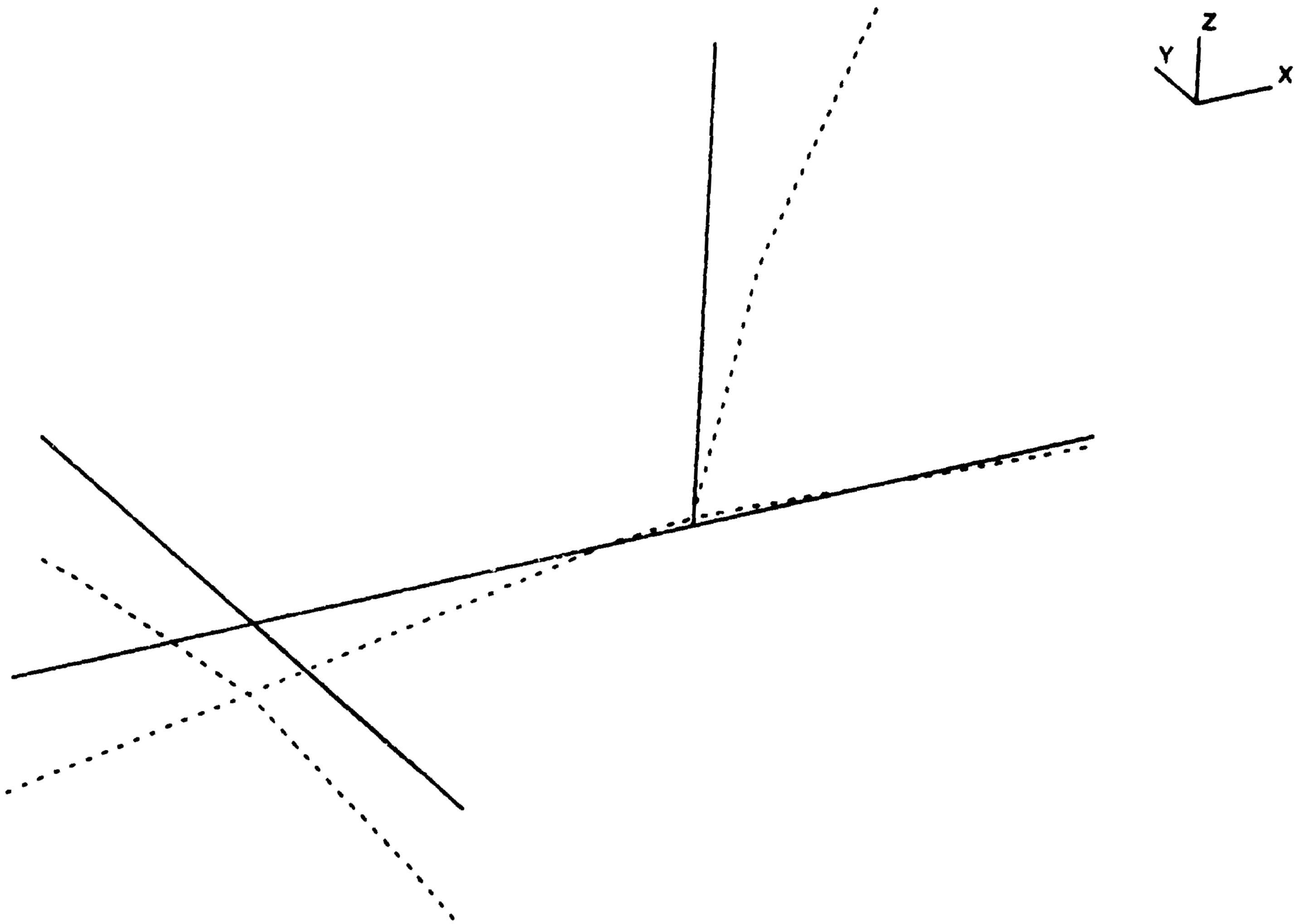
SPACE STATION PLATFORM MODEL



FREQ (4) - .58 HZ

Figure 5.2.5· ʌODE 10

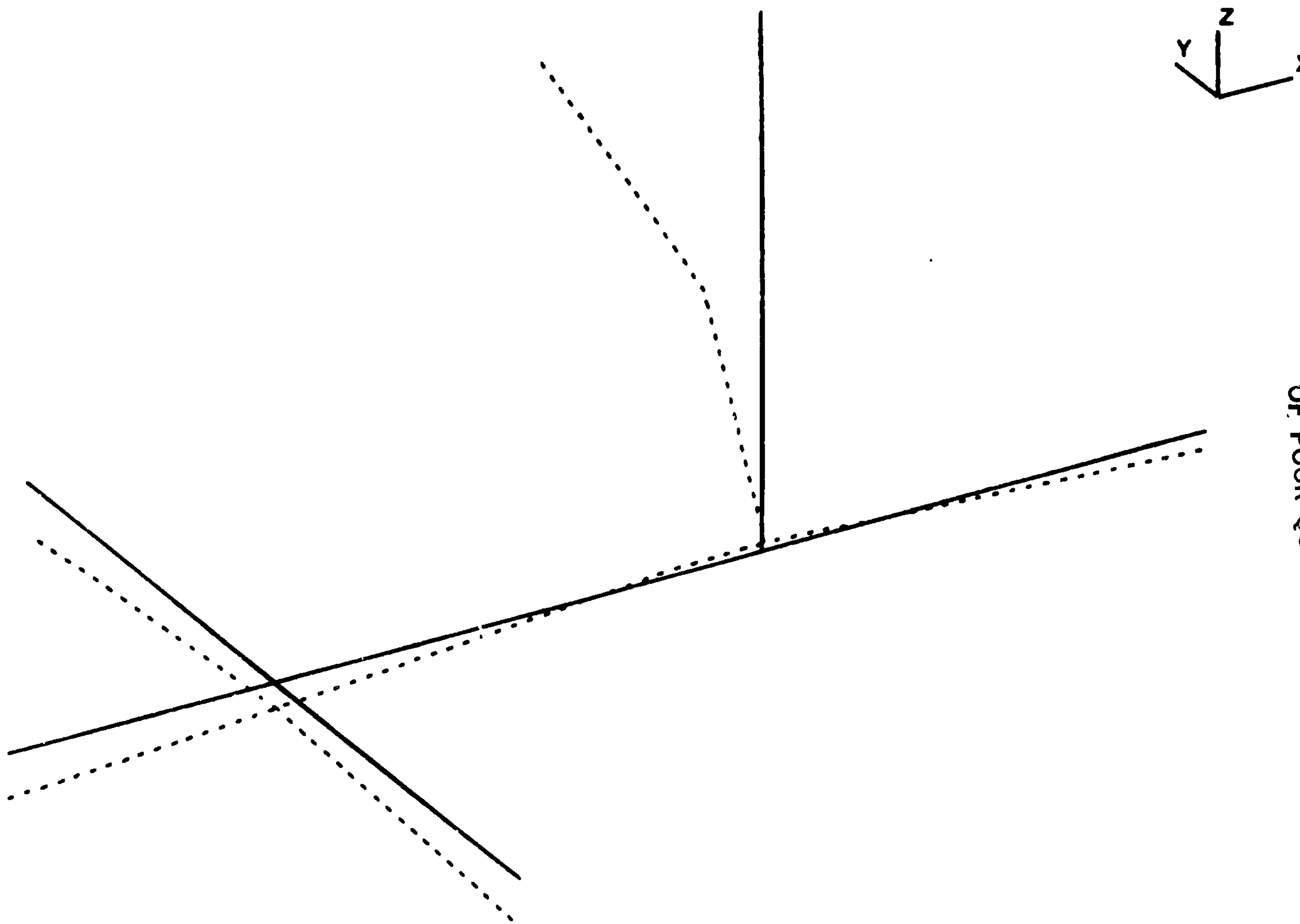
SPACE STATION PLATFORM MODEL



FREQ (5) - .63 HZ

Figure 5.2.5-8 - MODE 11

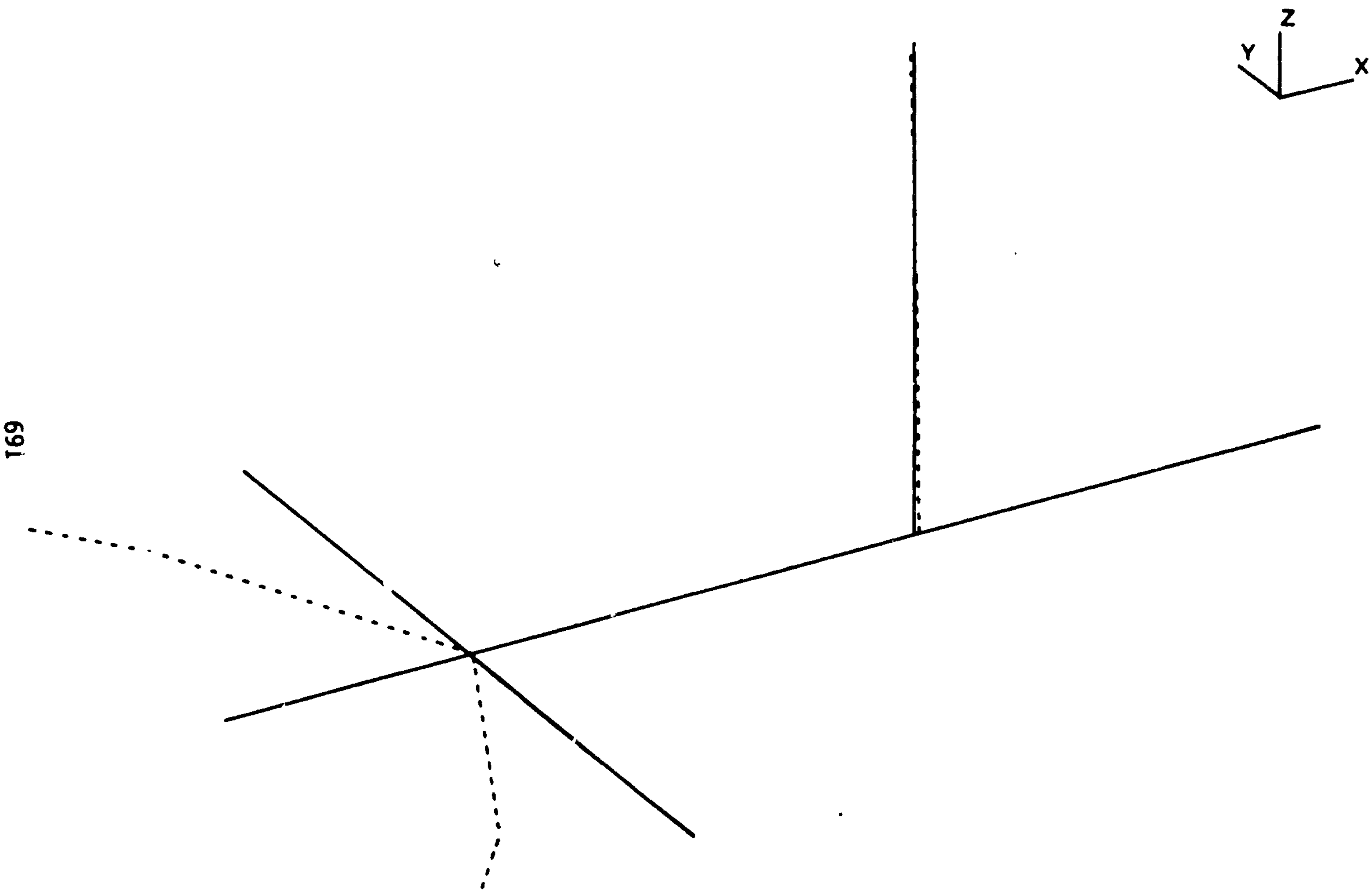
SPACE STATION PLATFORM MODEL



FREQ (6) - .75 HZ

Figure 5.) - MODE 12

SPACE STATION PLATFORM MODEL



FREQ (7) - 1.04 HZ

Figure 5.2.5-10 - MODE 13

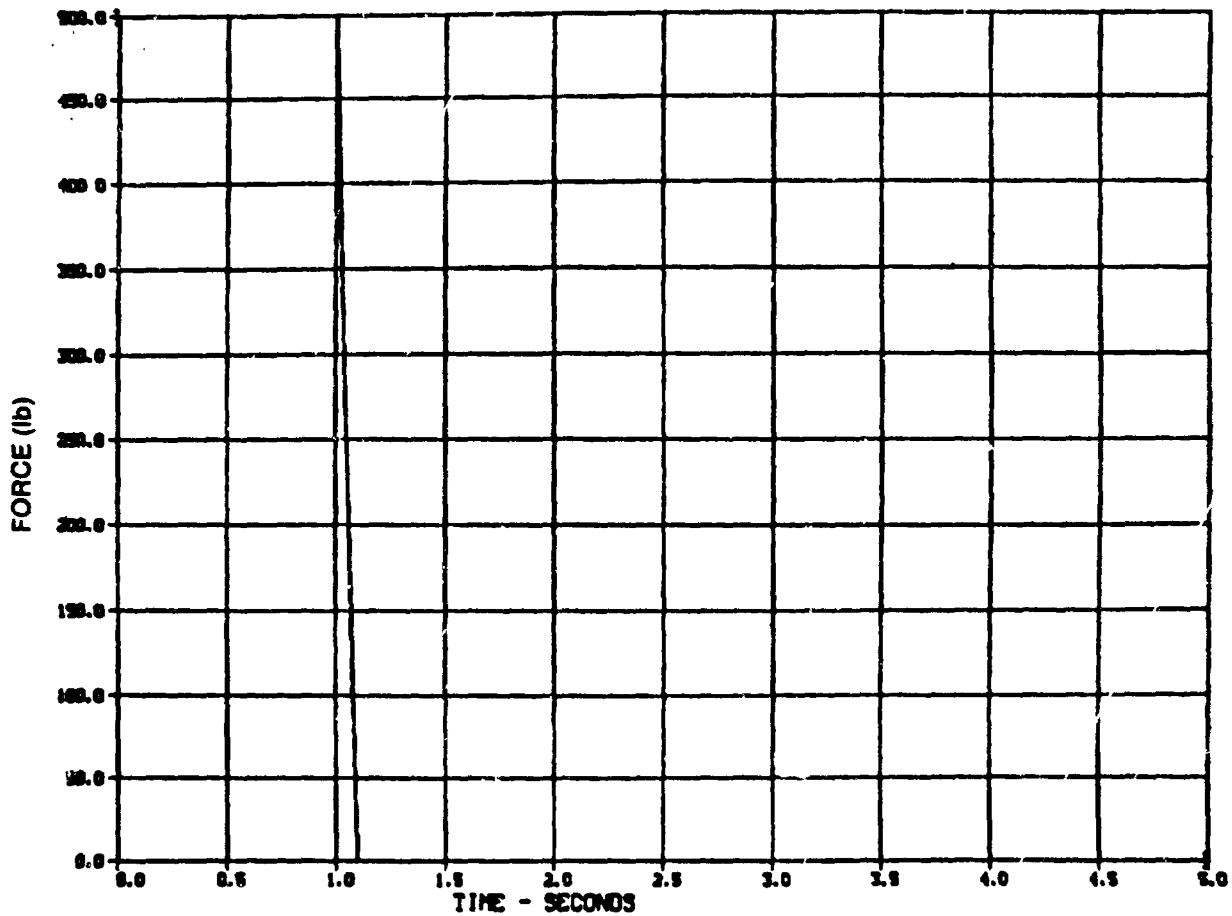


Figure 5.2.5-11 - ORBITER BERTHING UNPUT FORCES

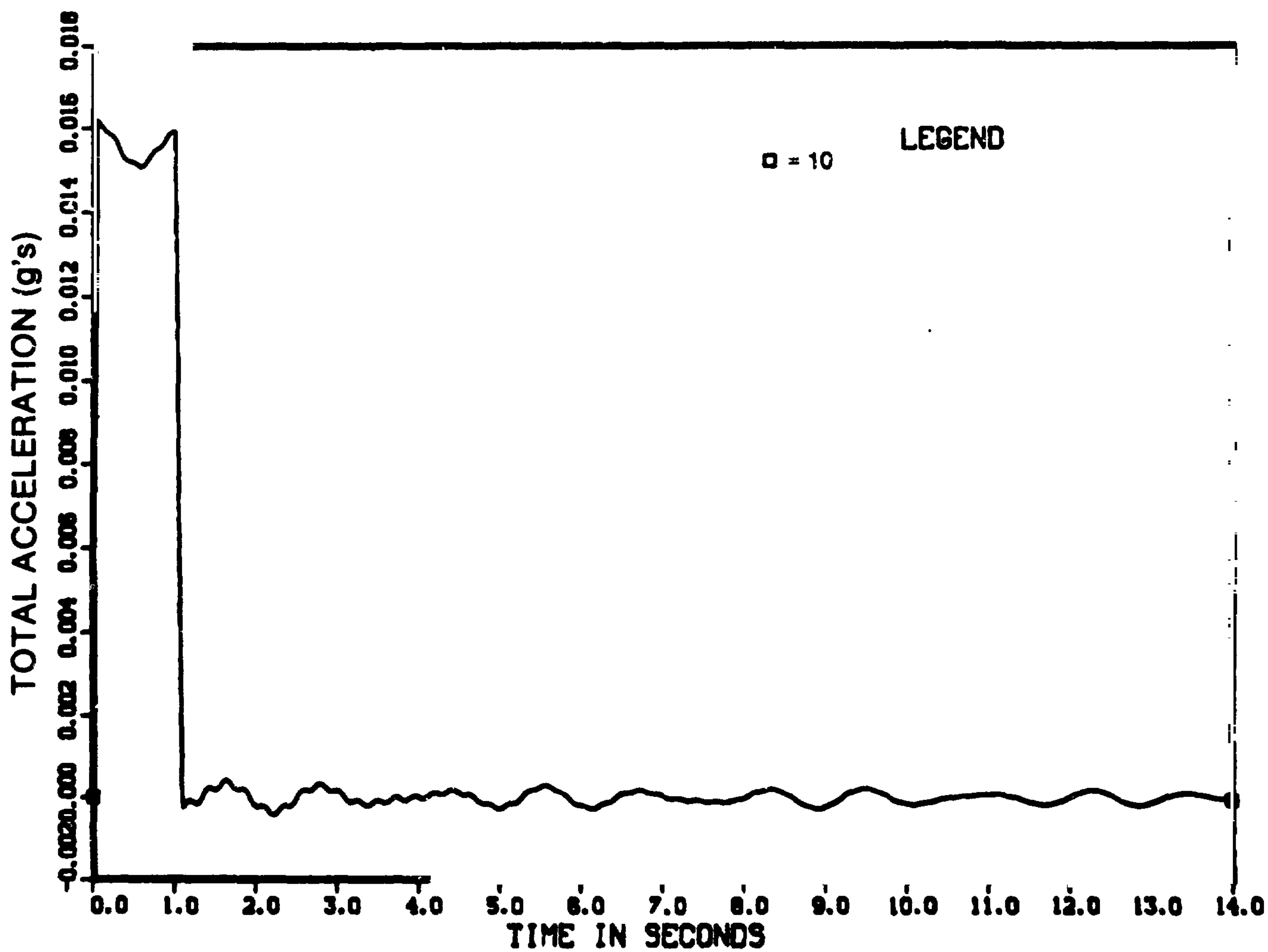


Figure 5.2.5-12.- BERTHING TRANSIENT RESPONSE

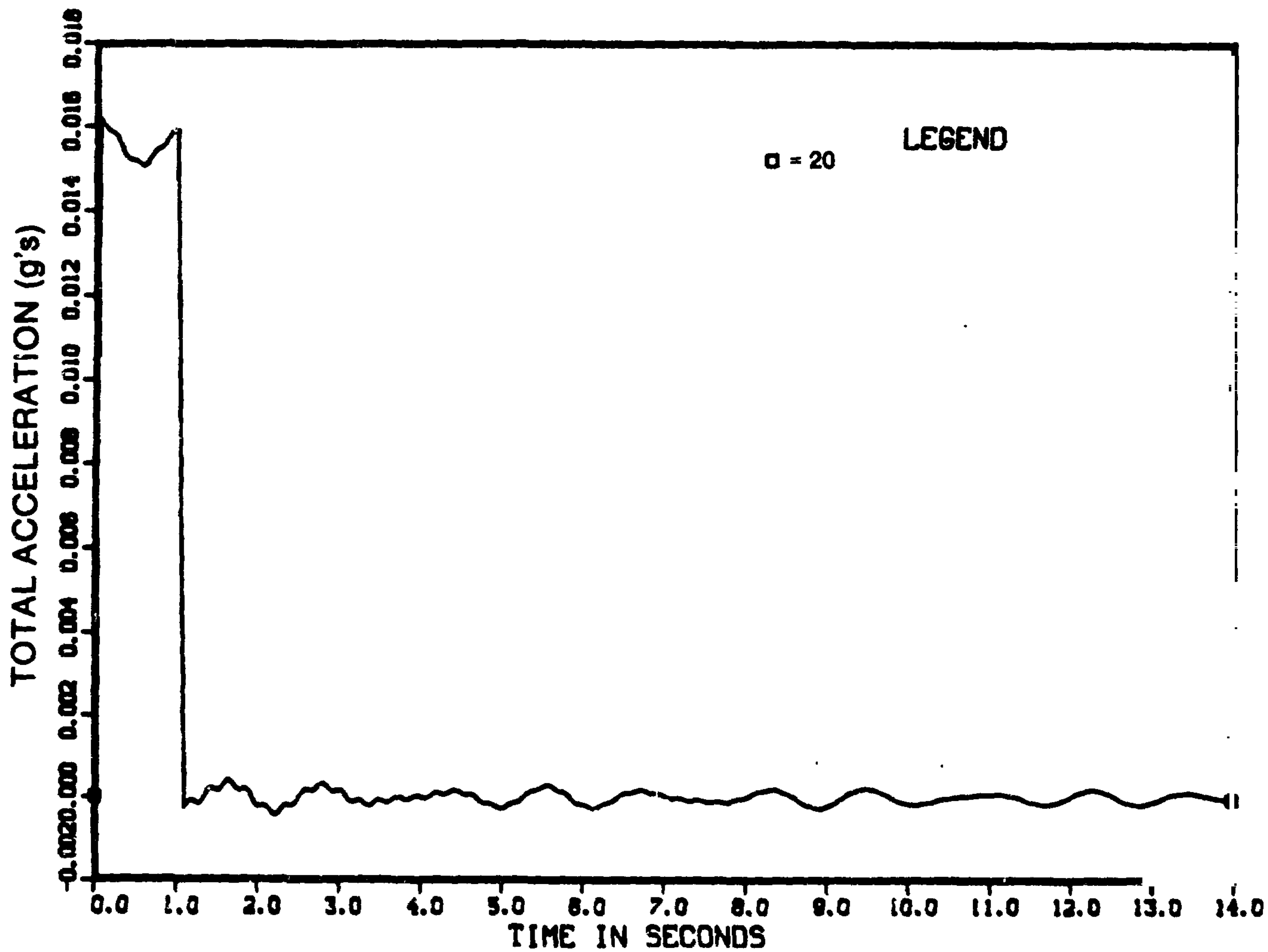


Figure 5.2.5-13 - BERTHING TRANSIENT RESPONSE

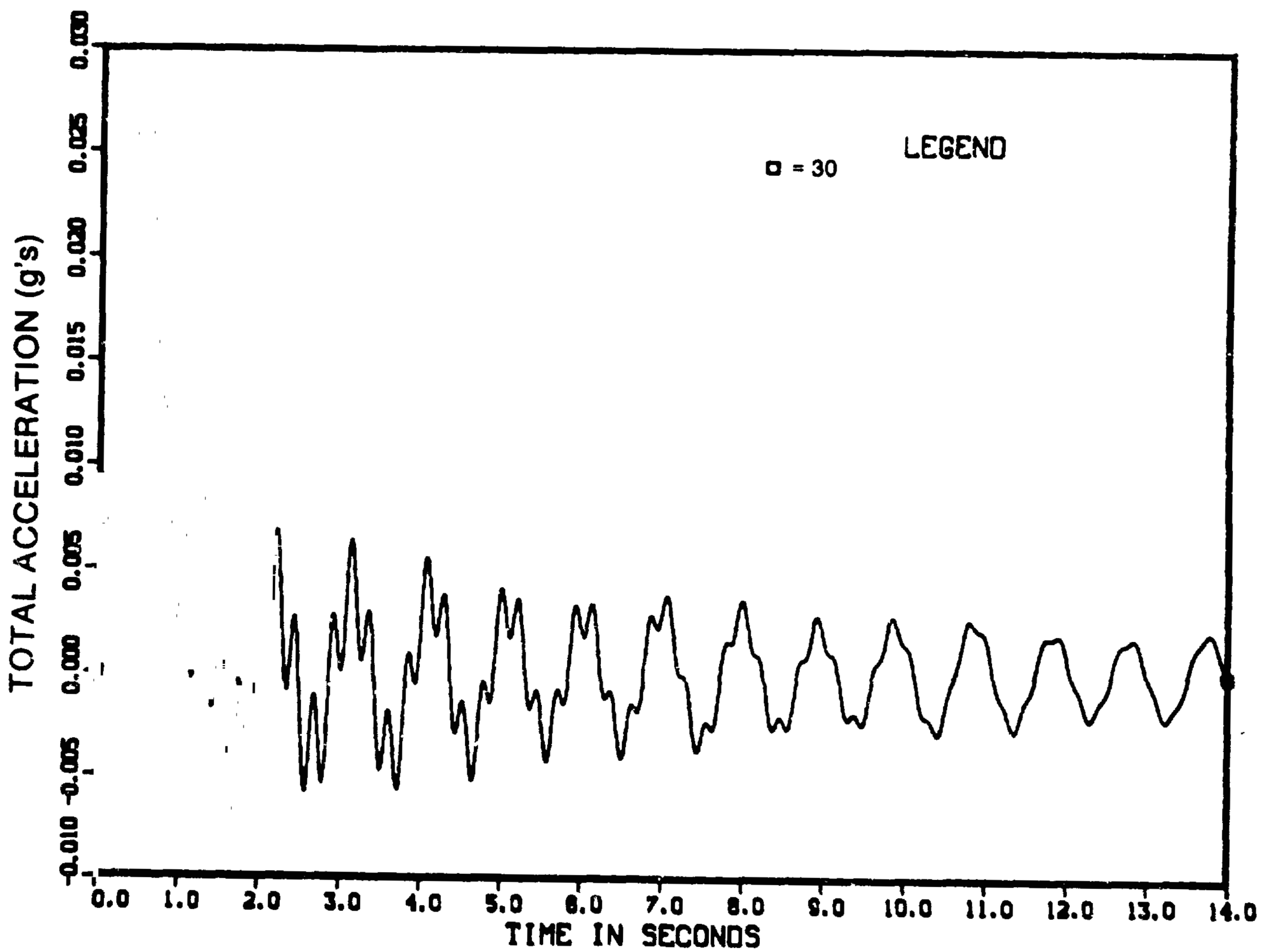


Figure 5.2.5-14 - BERTHING TRANSIENT RESPONSE

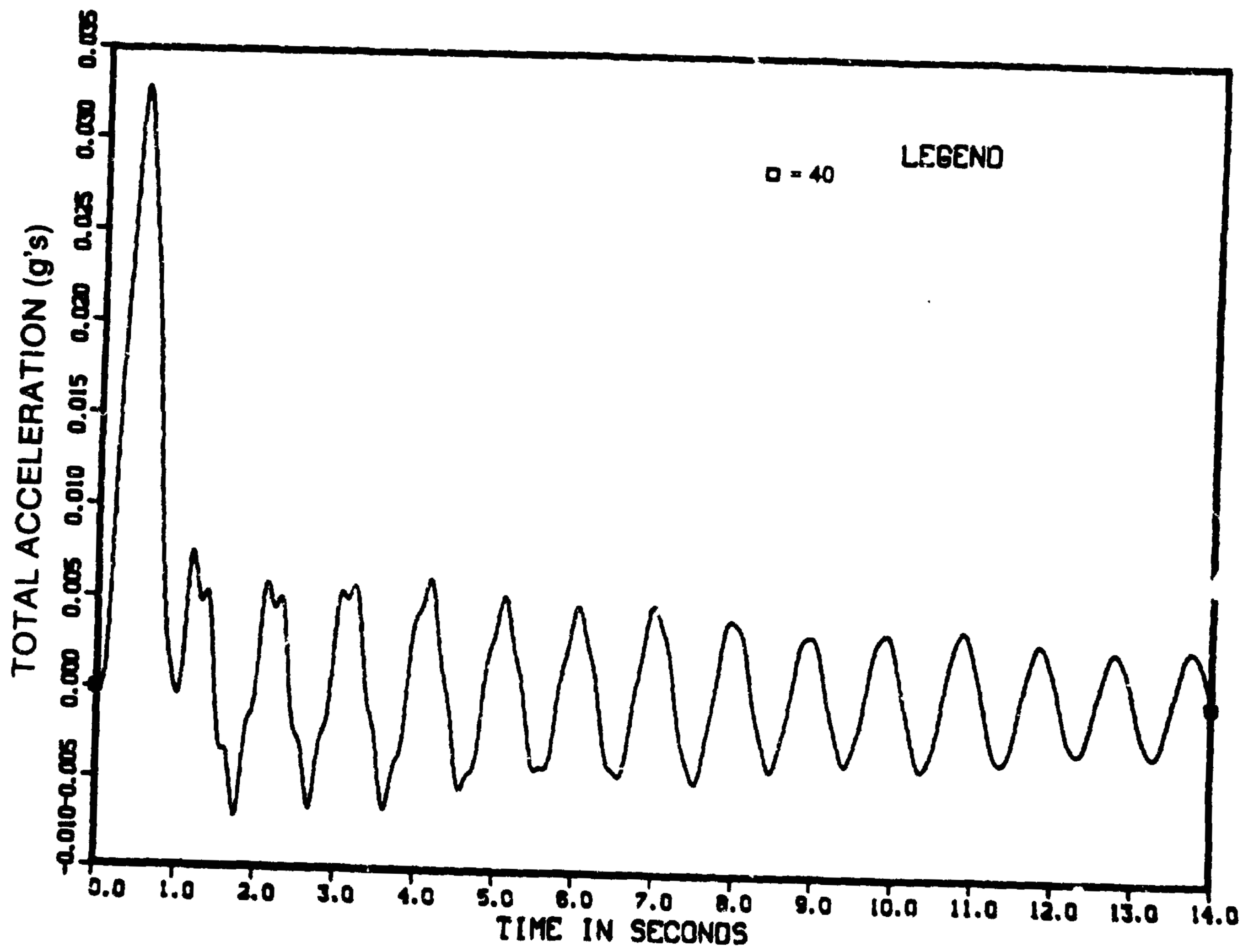


Figure 5.2.5-15 - BERTHING TRANSIENT RESPONSE

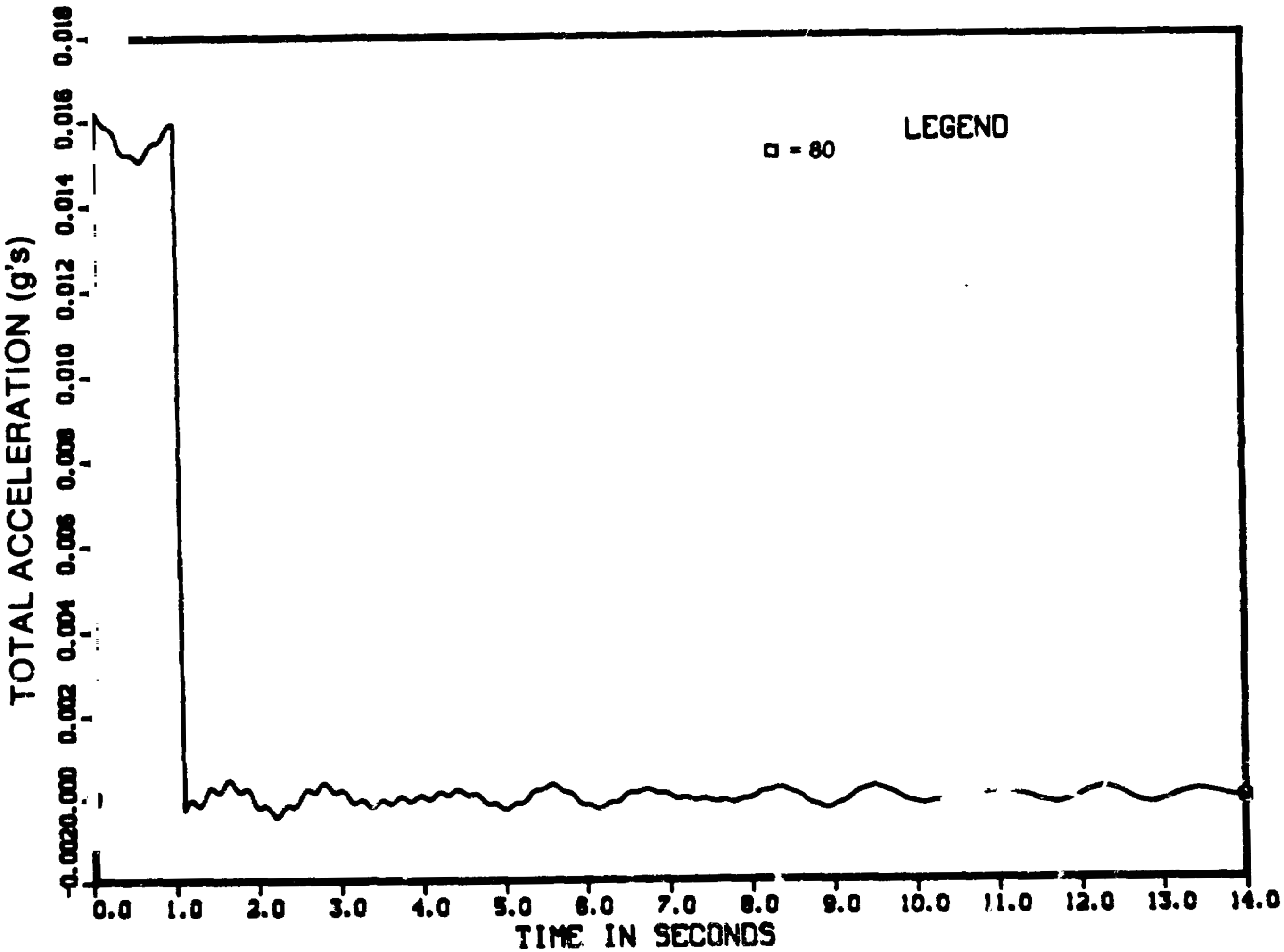


Figure 5.2.5-16 - BERTHING TRANSIENT RESPONSE

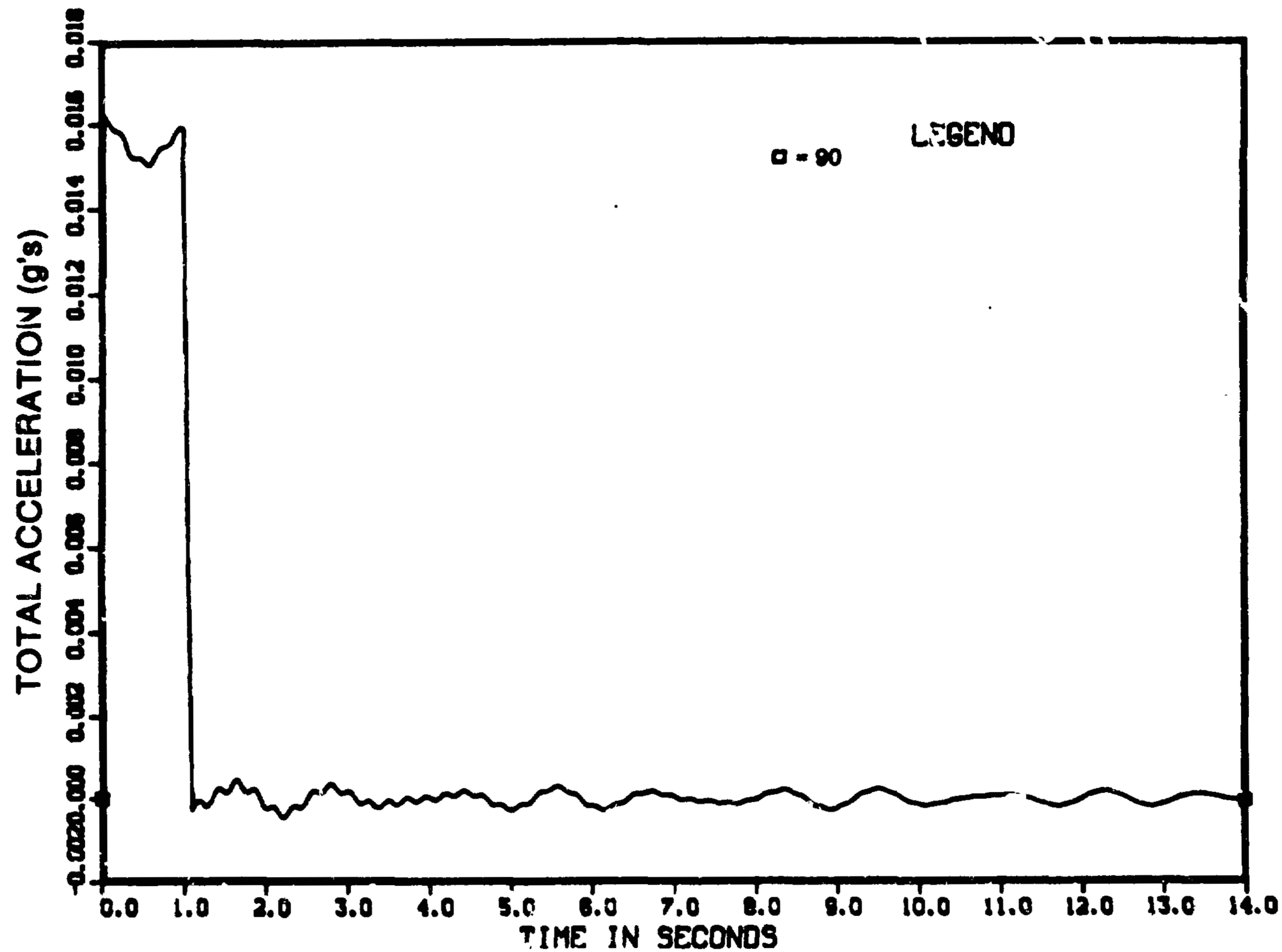


Figure 5.2.5-17 · BERTHING TRANSIENT RESPONSE

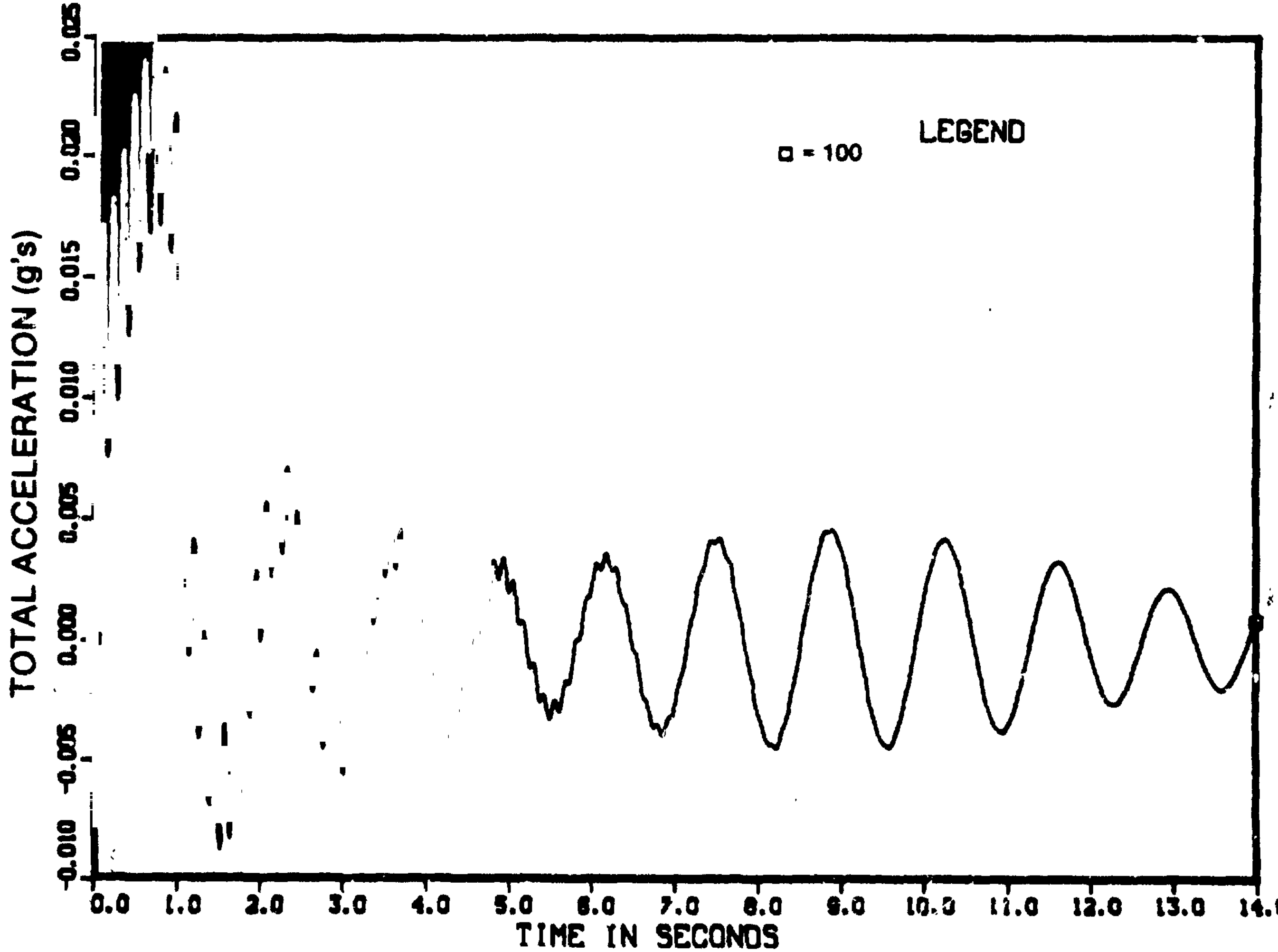


Figure 5.2.5-18 - BERTHING TRANSIENT RESPONSE

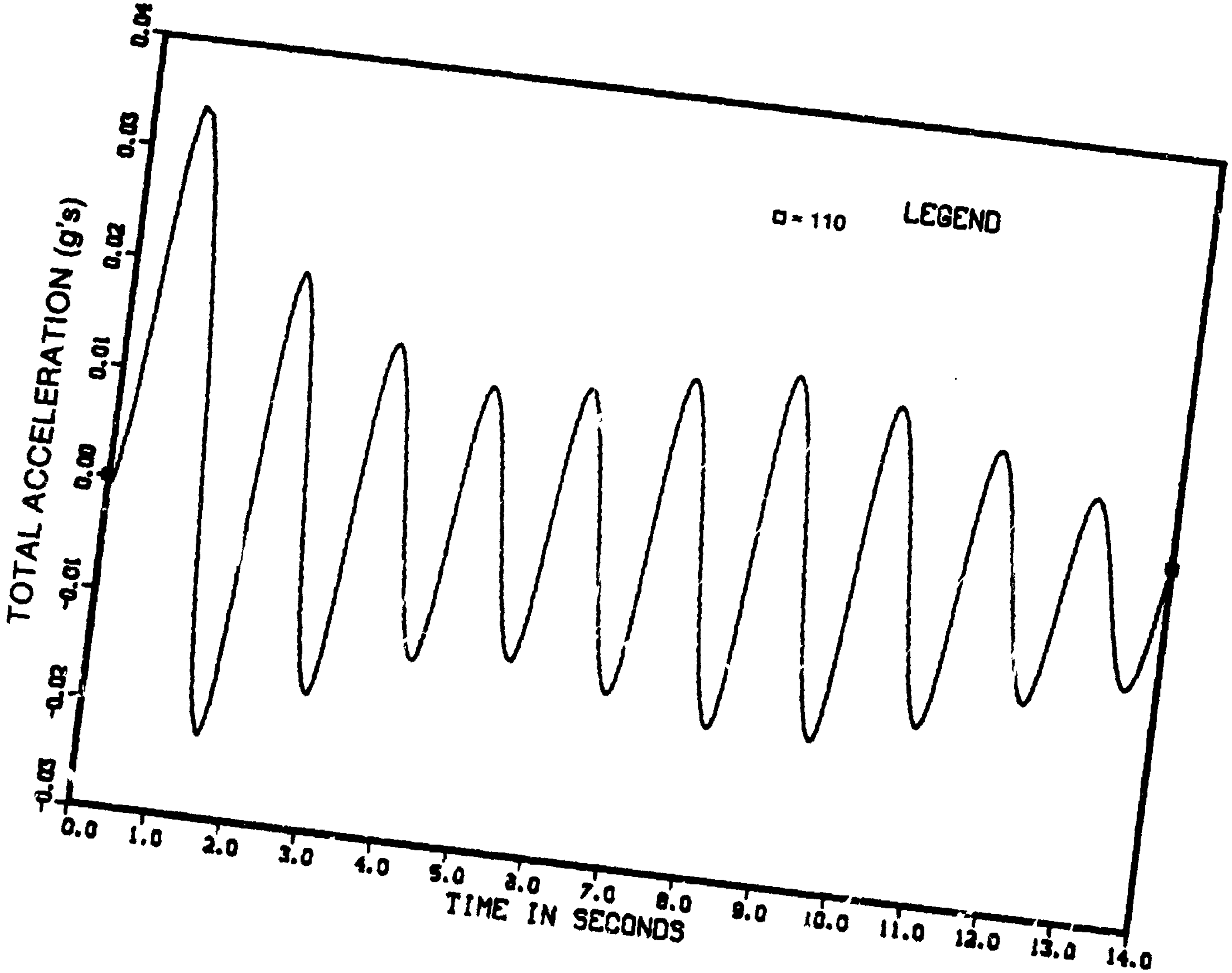


Figure 5.2.5-19 - BERTHING TRANSIENT RESPONSE

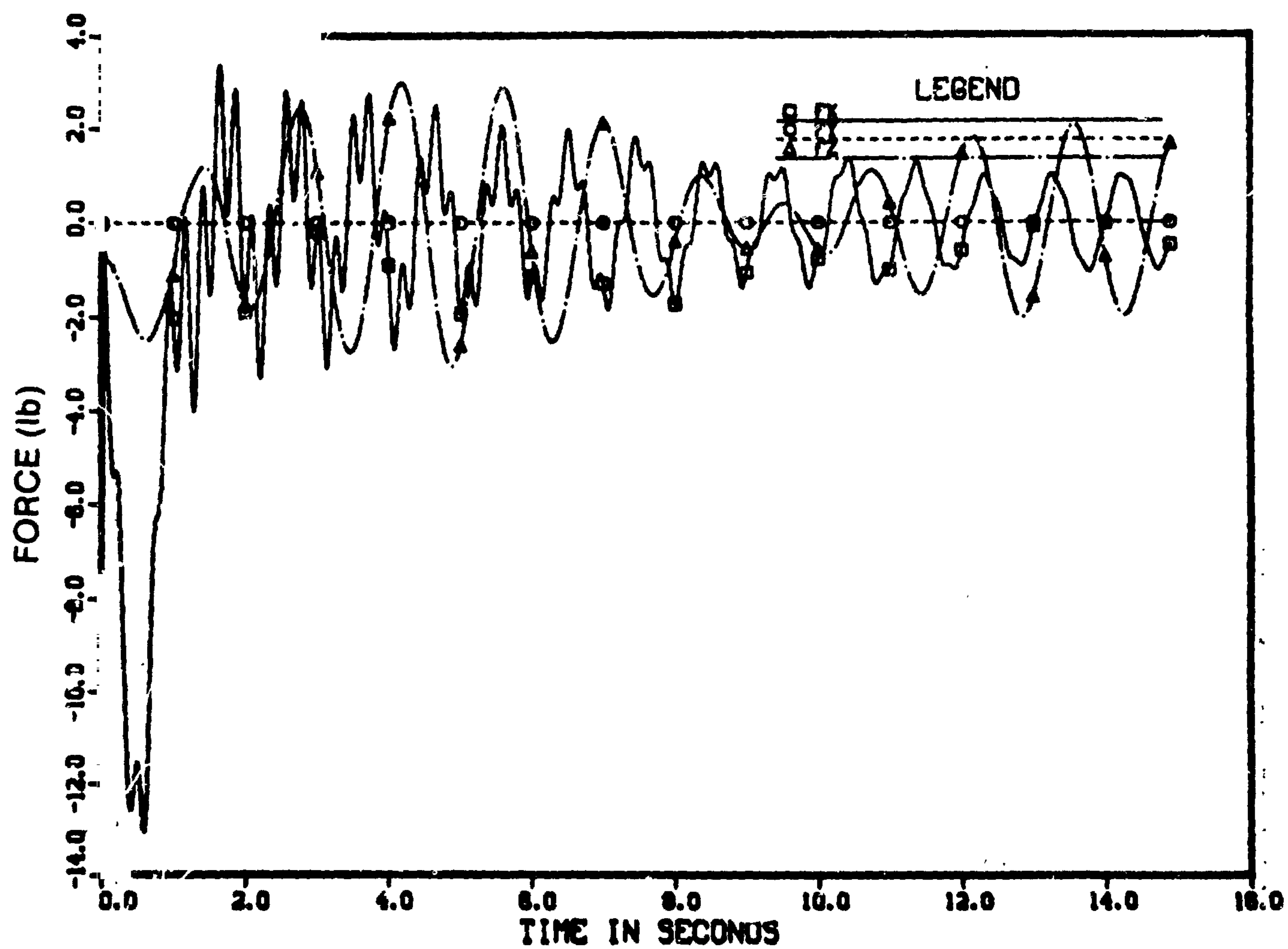


Figure 5.2.5-20 - SOLAR ARRAY BERTHING LOADS

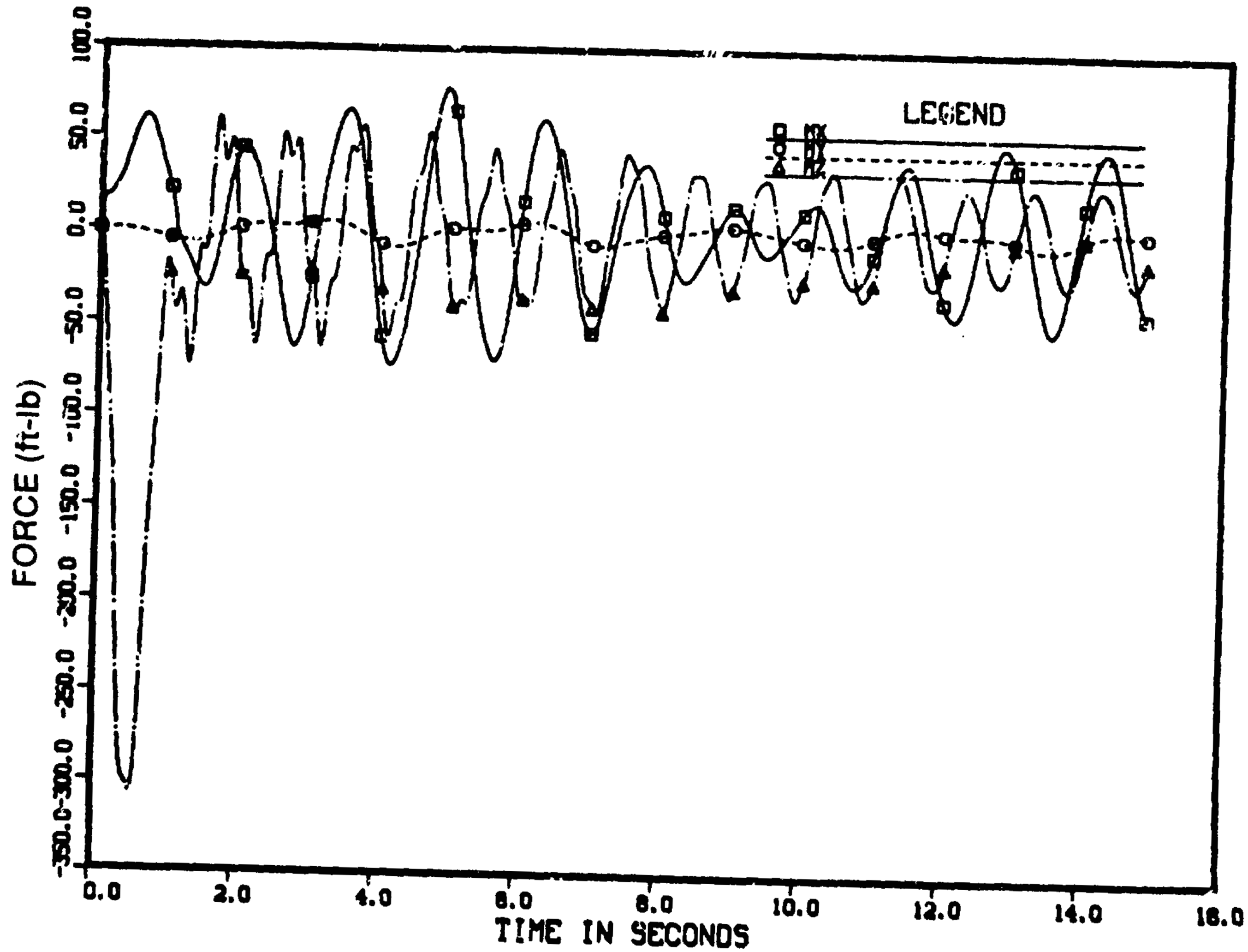


Figure 5.2.5-21 - SOLAR ARRAY BERTHING LOADS

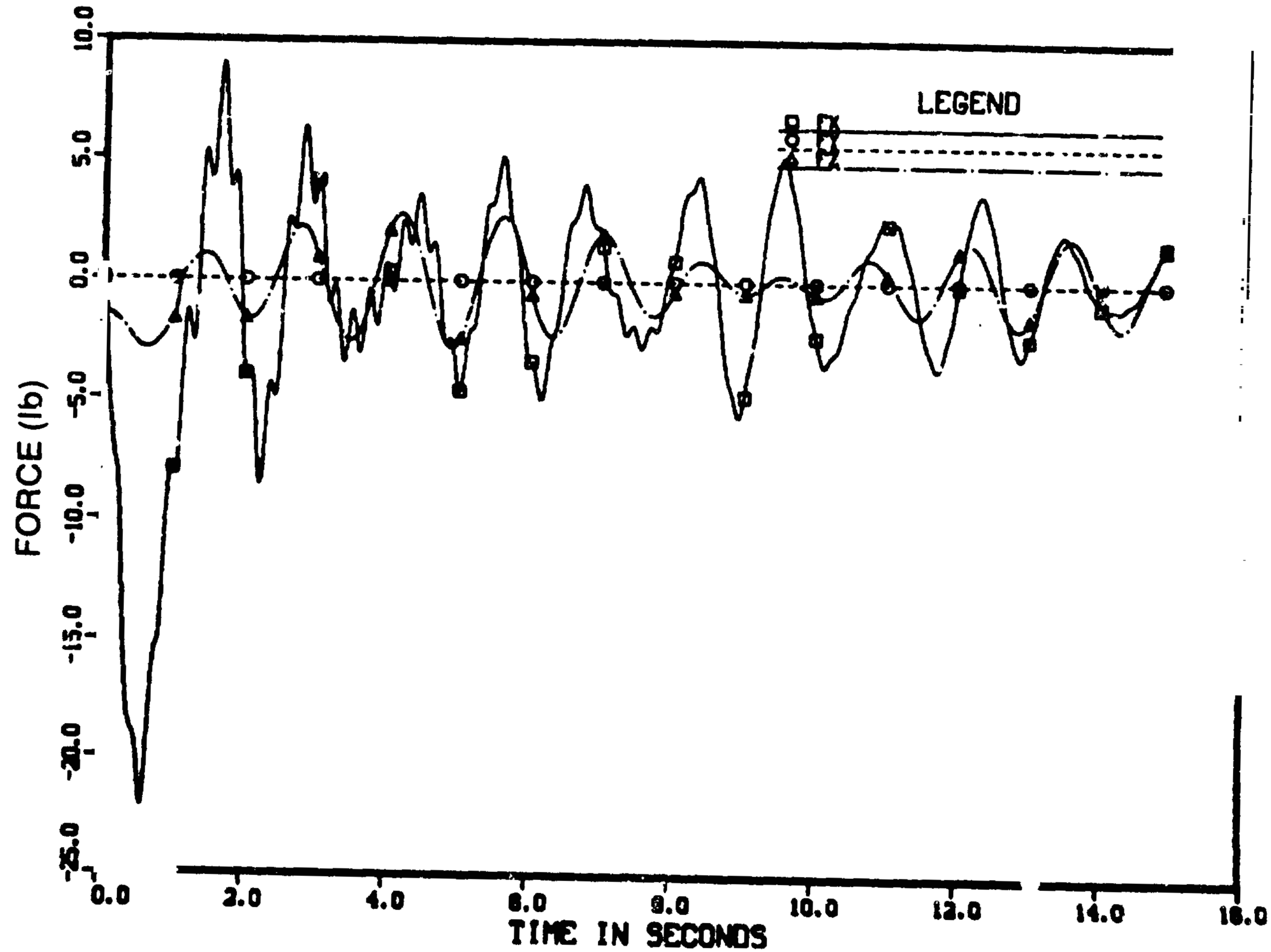


Figure 5.2.5-22 - ANTENNA BOOM BERTHING LOADS

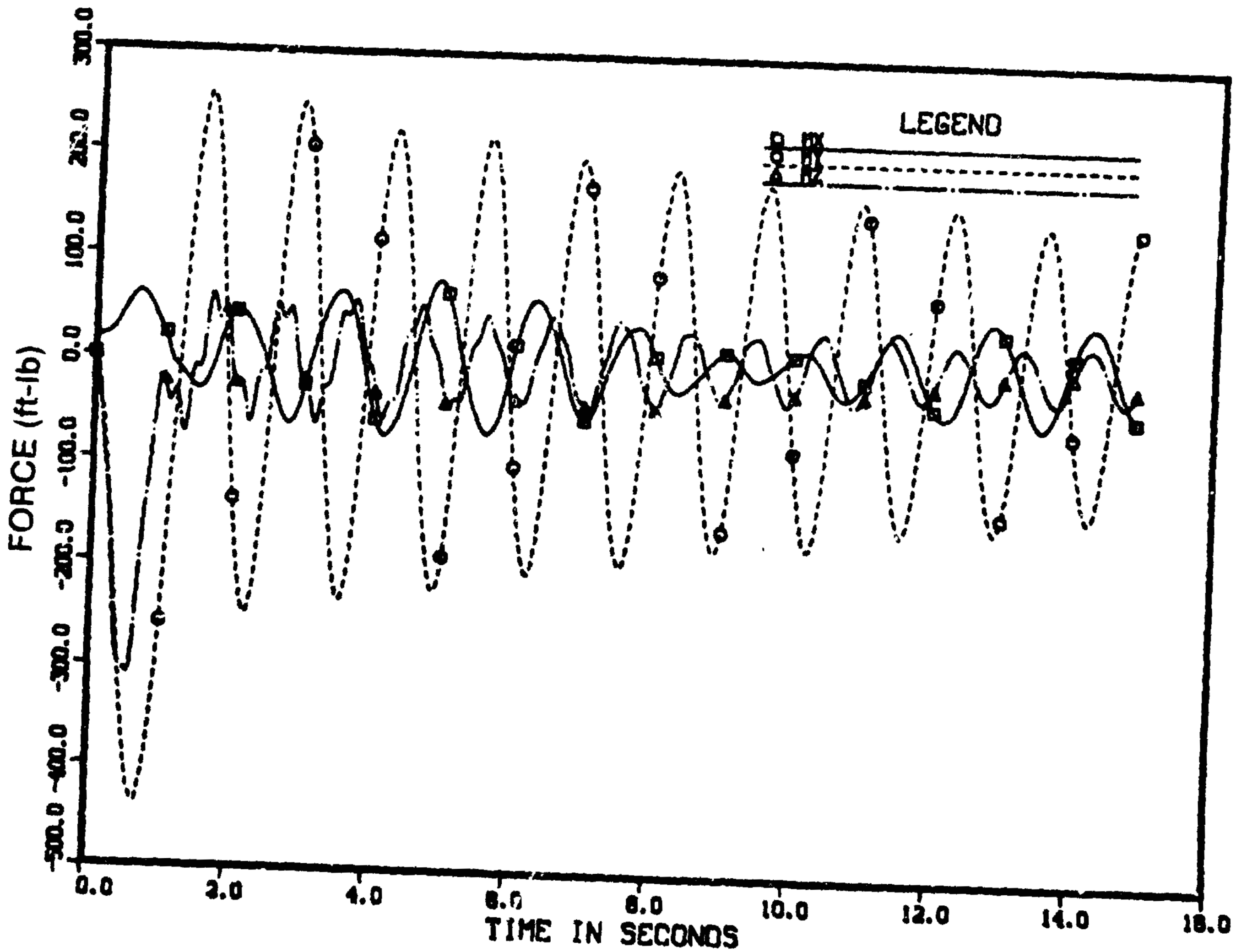


Figure 5.2.5-23 - ANTENNA BOOM BERTHING LOADS

flexible appendages vibrate for a longer time in a damped oscillatory motion. The Core body of the Platform behaves as a rigid body whereas the appendages account for the structural dynamics of the system. Berthing loads produce 450 ft-lb at the root of the antenna support boom concurrent with 22 lb of shear. This represents the highest load in the system for berthing and is well within the 3950 ft-lb moment capability of the beam. The solar array support beam carries 310 ft-lb at the root and is not considered significant from a strength viewpoint.

The reboost forces, necessary to keep the vehicle at the proper orbit, are a train of square pulses generated from off modulated RCS jets driven by the attitude control algorithm (Figs. 5.2.5.-24 and 5.2.5-25). The temporal character of these pulses dictates the amount of dynamic involvement of the flexible components on the Platform. Figures 5.2.5-26 through 5.2.5-37 show the Platform response to the reboost forces. The antenna support boom has a response character similar to the input forces with a maximum antenna acceleration of 0.07 g's and a root moment value of 700 ft-lb. An interesting dynamic feature of this configuration is the vibratory communication between the solar array boom and the antenna support structure. The coupling between these systems allows for vibration energy in the solar arrays to transfer to vibration in the antenna boom. This transfer of energy is responsible for the damped response of the solar arrays; the energy in the arrays is reduced, but the energy is not dissipated, it is simply transferred to the antenna boom. The growth character of the flexible response is of concern because of its similarity to a resonant condition. Reboost forcing functions of slightly different character may lead to a large increase in loads due to resonance and must be analyzed carefully. The analysis does not indicate a specific violation of structural allowables; however, it does indicate sensitivity to the reboost force inputs.

5.2.5.4 Rigid Body Control Dynamics

The objective of this analysis was to determine the control requirements for the unmanned Platform elements as a function of the natural on-orbit dynamic environment. However, only two representative Platform missions were analyzed in detail and will be reported on herein. They are (1) the polar payload B, and (2) the co-orbiting payload Starlab. The dynamic environments simulated included gravity gradient torques, aerodynamic drag, and aerodynamic torques.

5.2.5.4.1 Simulation capabilities

The SSDYNAMICS program, developed at JSC, and described in Section 4.3.3.5.1.1 was used to simulate the on-orbit dynamics of the Platform elements.

5.2.5.4.2 Mass properties

Mass properties used in the SSDYNAMIC simulation for the two Platform studies are shown in Table 5.2.5-2. Two sets of mass properties are

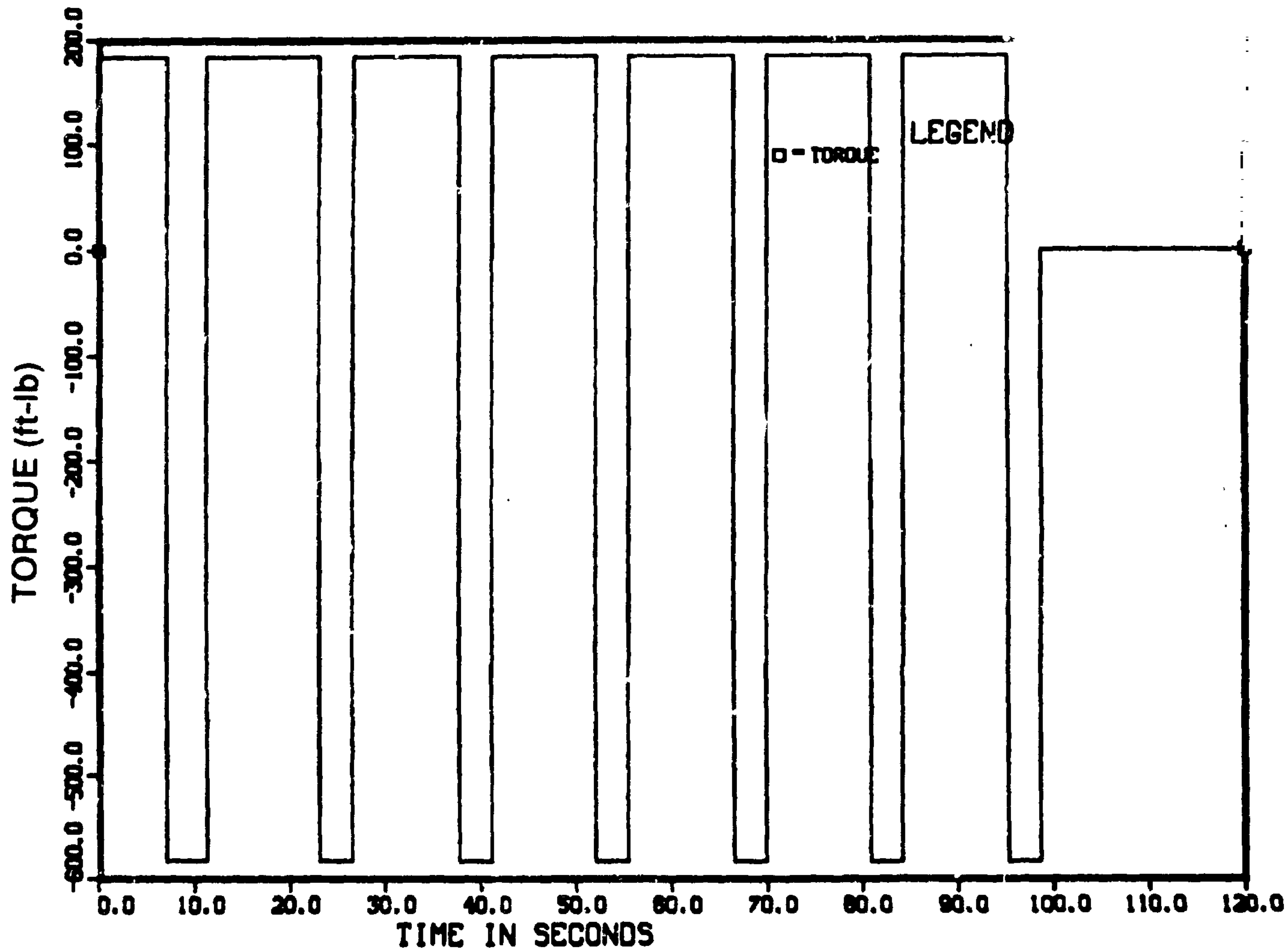


Figure 5.2.5-24 - PLATFORM REBOOST TRANSIENT TORQUE

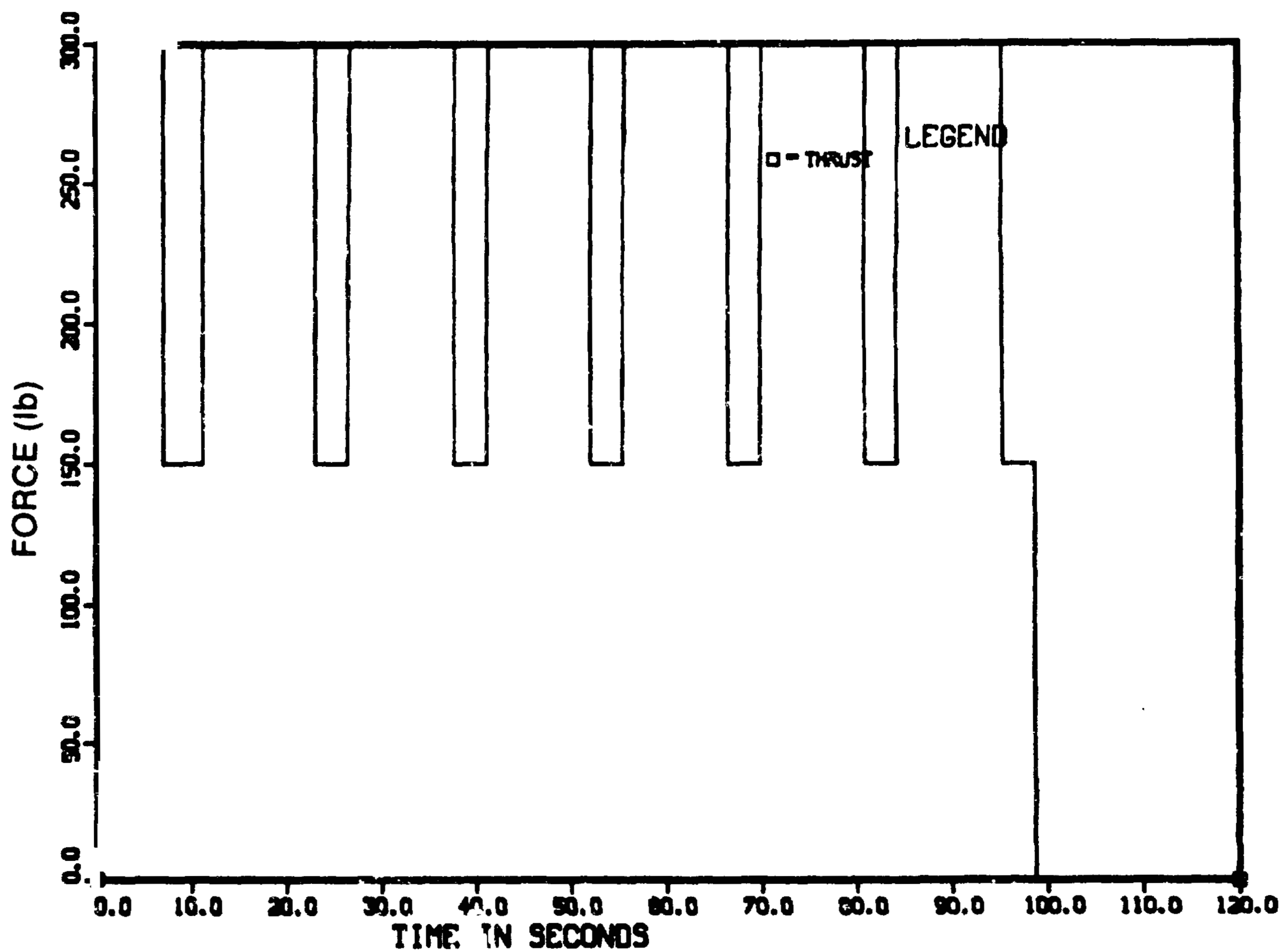


Figure 5.2.5-25 - PLATFORM REBOOST TRANSIENT FORCE

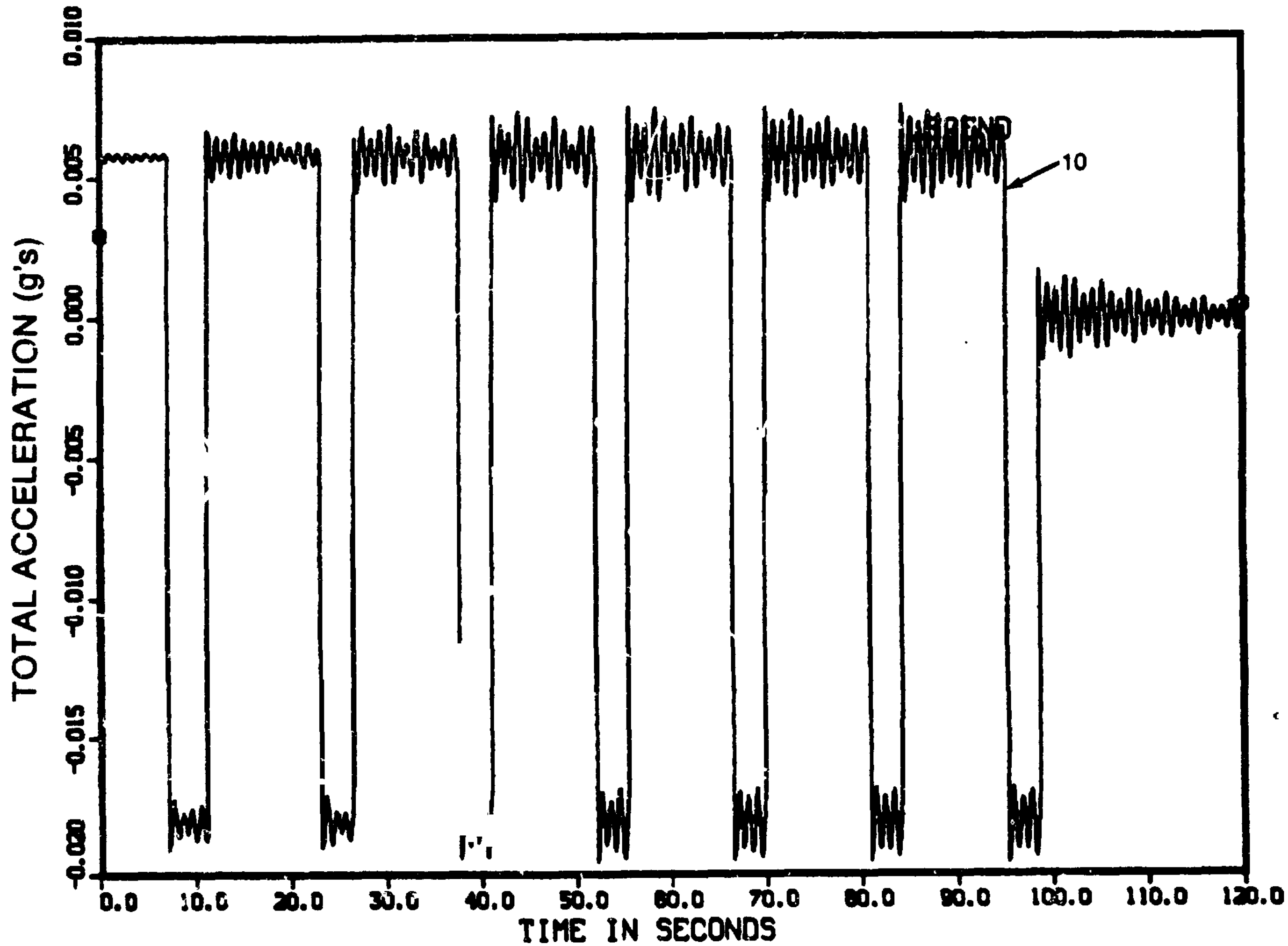


Figure 5.2.5-26 - REBOOST TRANSIENT RESPONSE

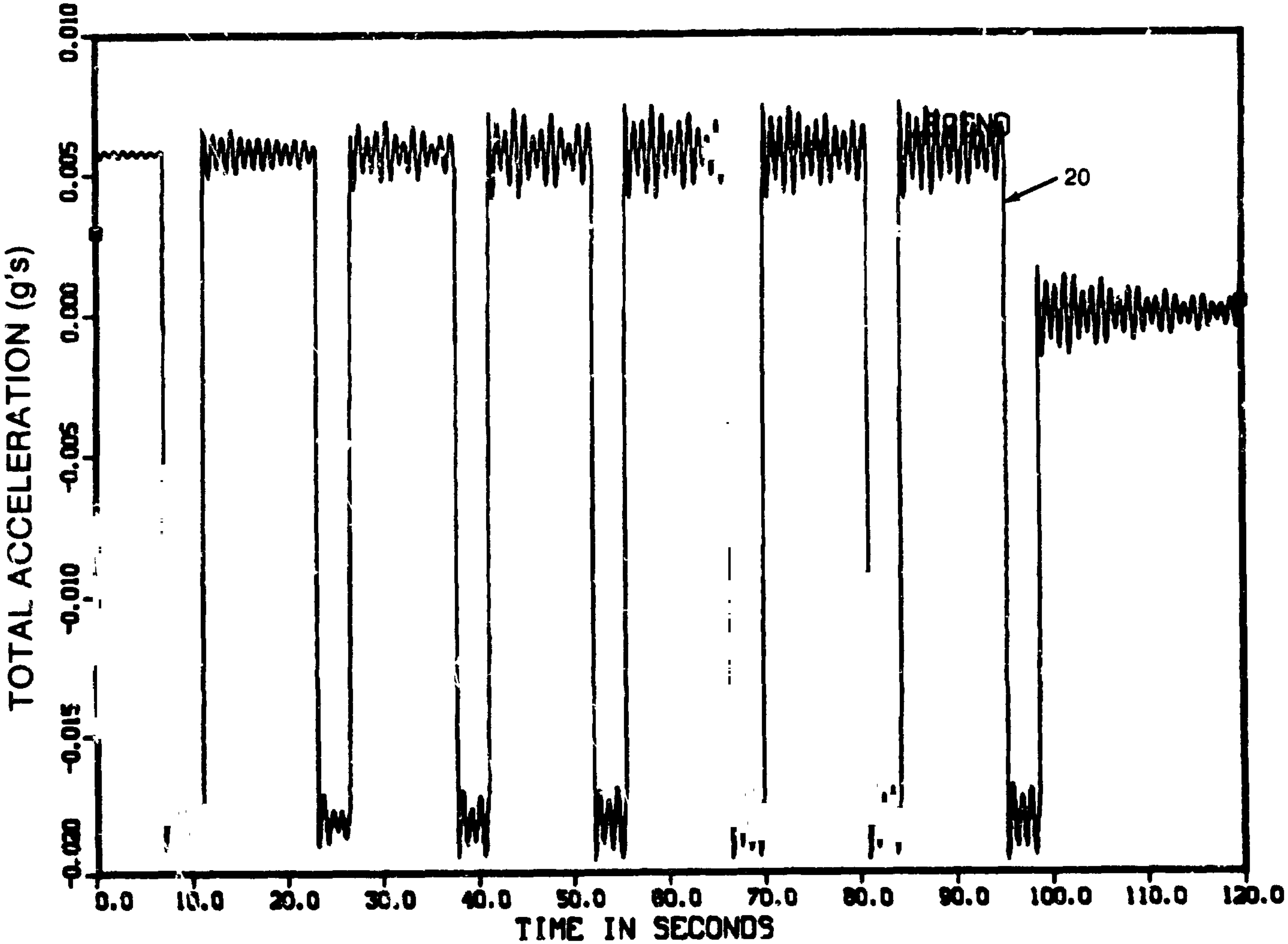


Figure 5.2.5-27– REBOOST TRANSIENT RESPONSE

ORIGINAL PAGE IS
OF POOR QUALITY

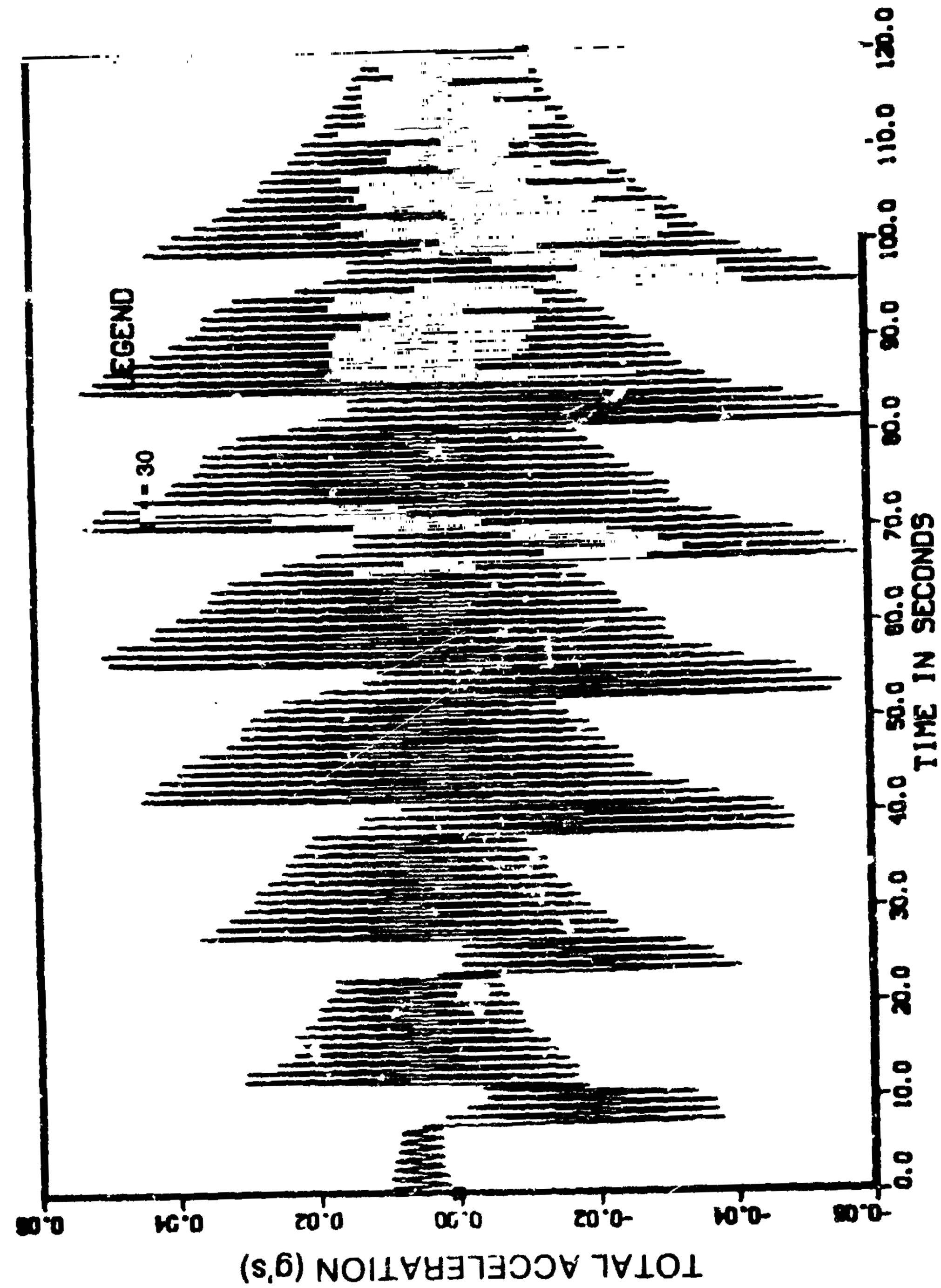


Figure 5.2.5-28 - REBOOST TRANSIENT RESPONSE

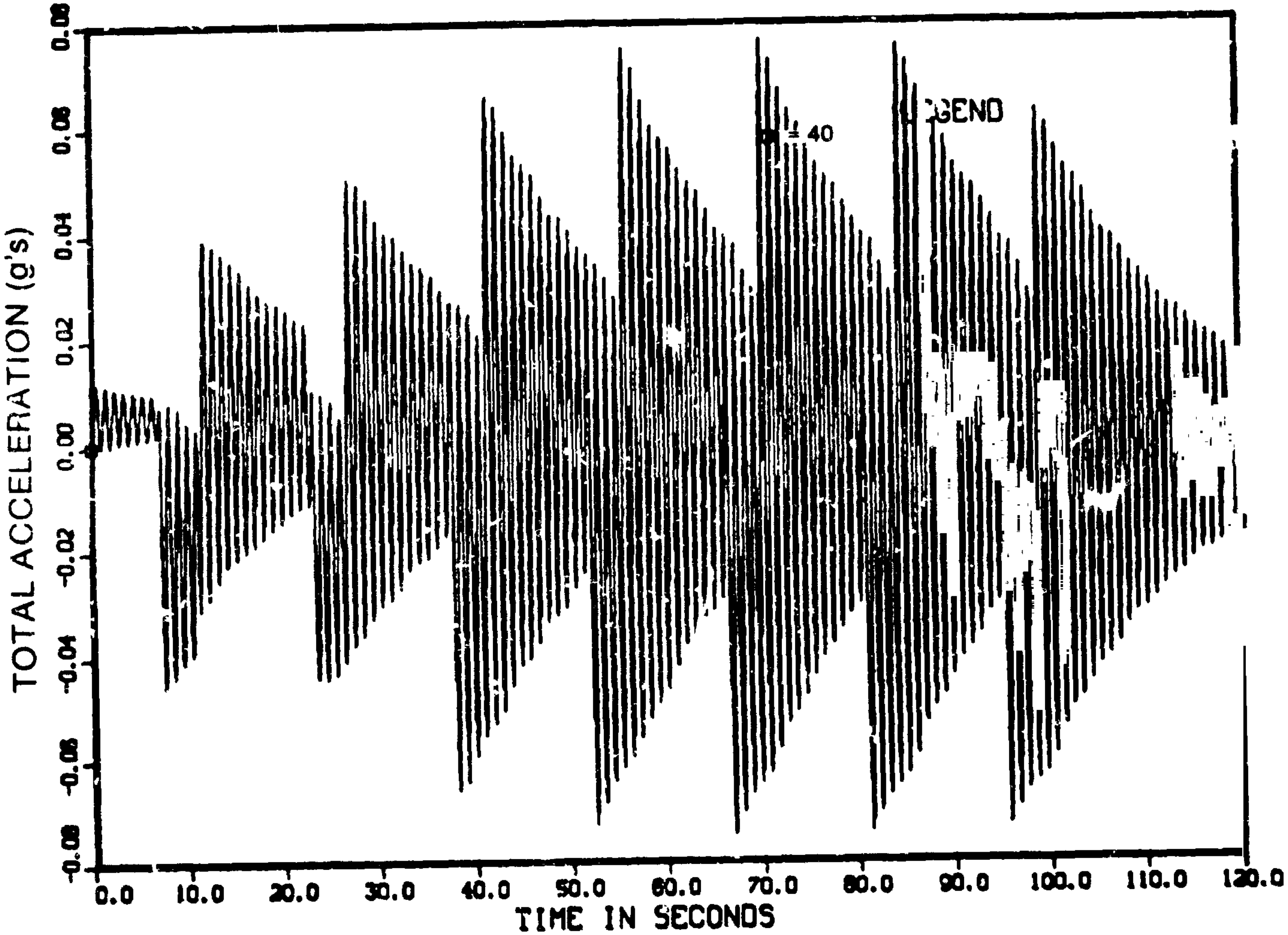


Figure 5.2.5-29 - REBOOST TRANSIENT RESPONSE

ORIGINAL PAGE IS
OF POOR QUALITY

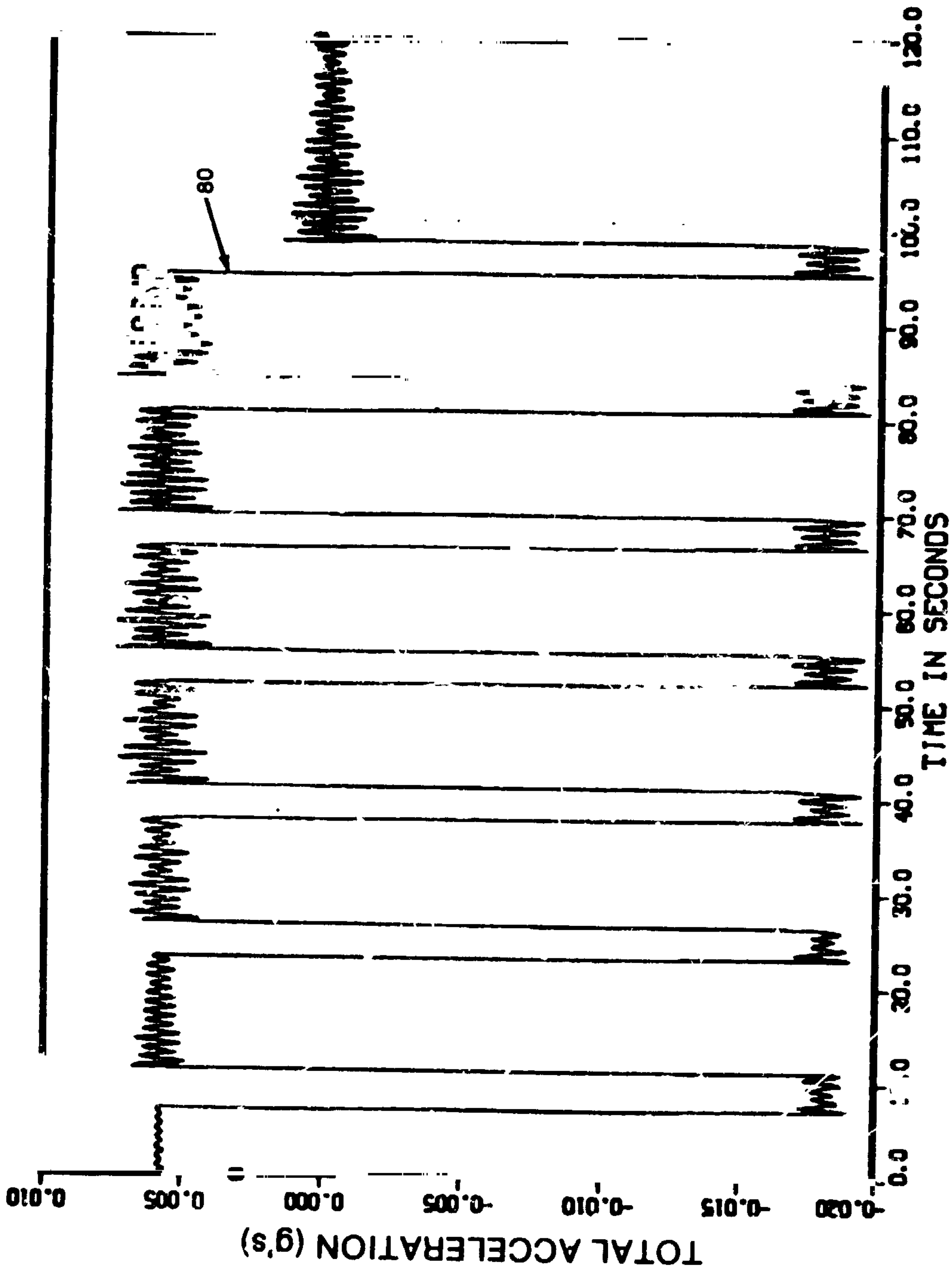


Figure 5.2.5-30 - REBOOST TRANSIENT RESPONSE

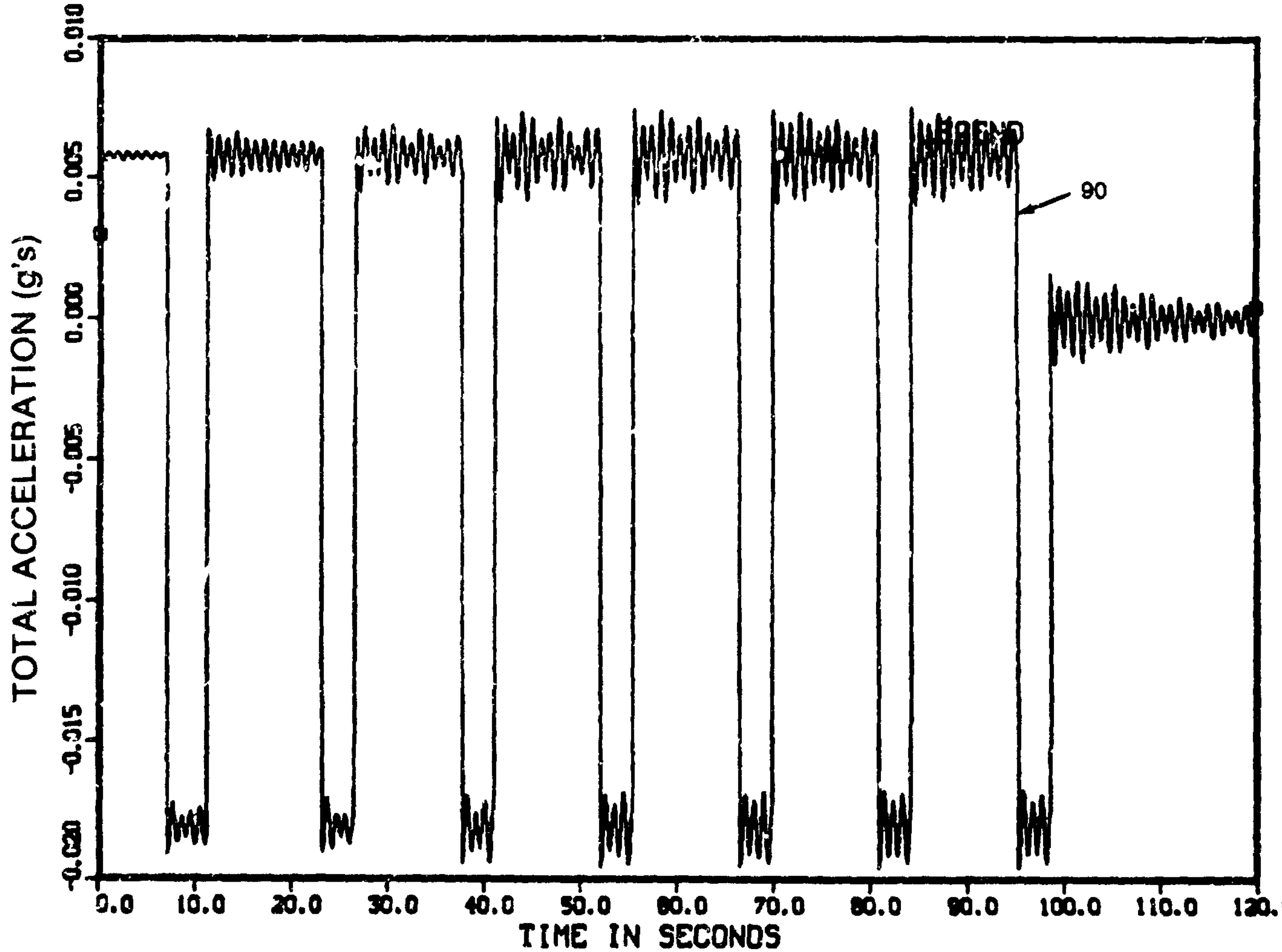


Figure 5.2.5-31 - REBOOST TRANSIENT RESPONSE

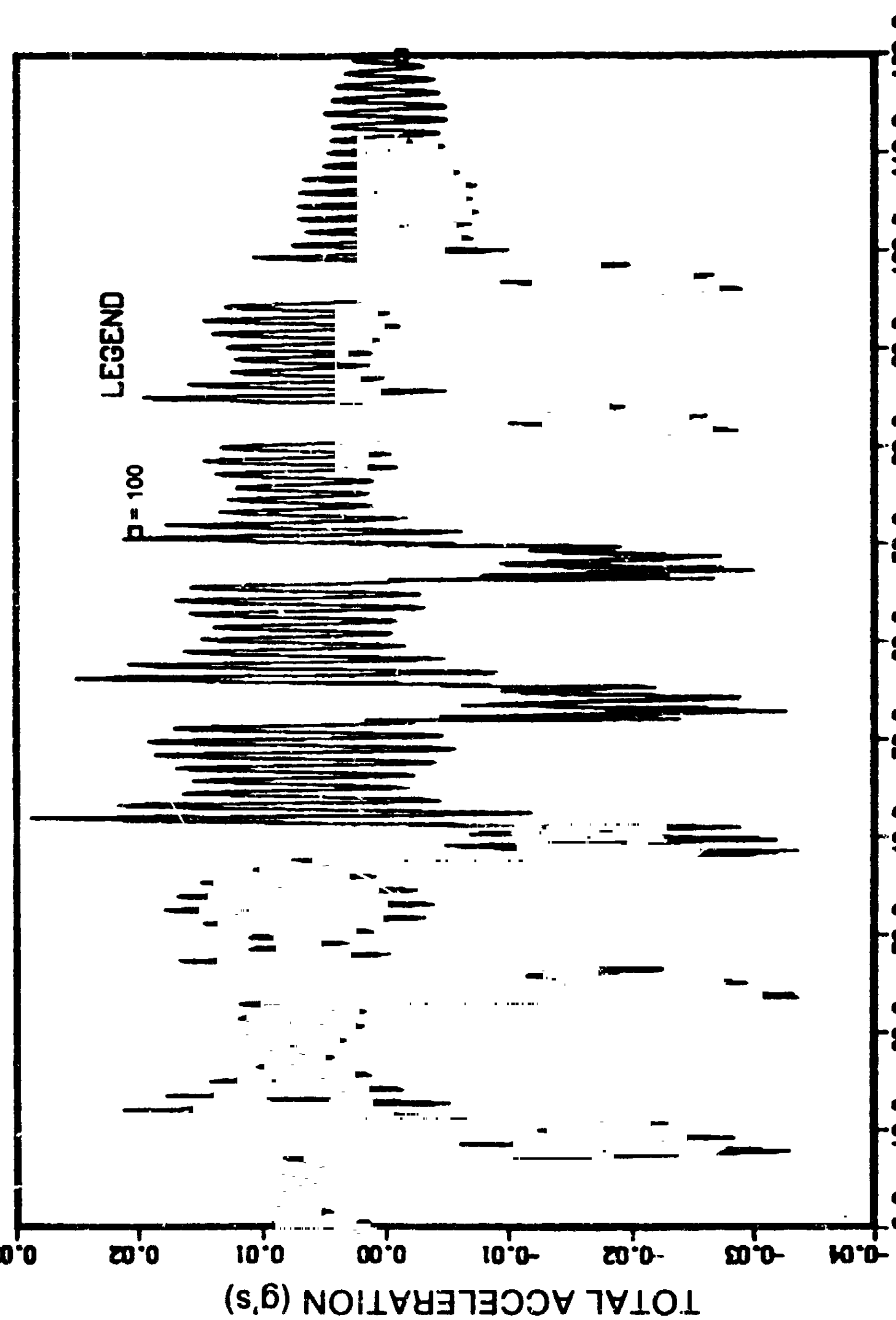


Figure 5.2.5-32 - REBOOST TRANSIENT RESPONSE

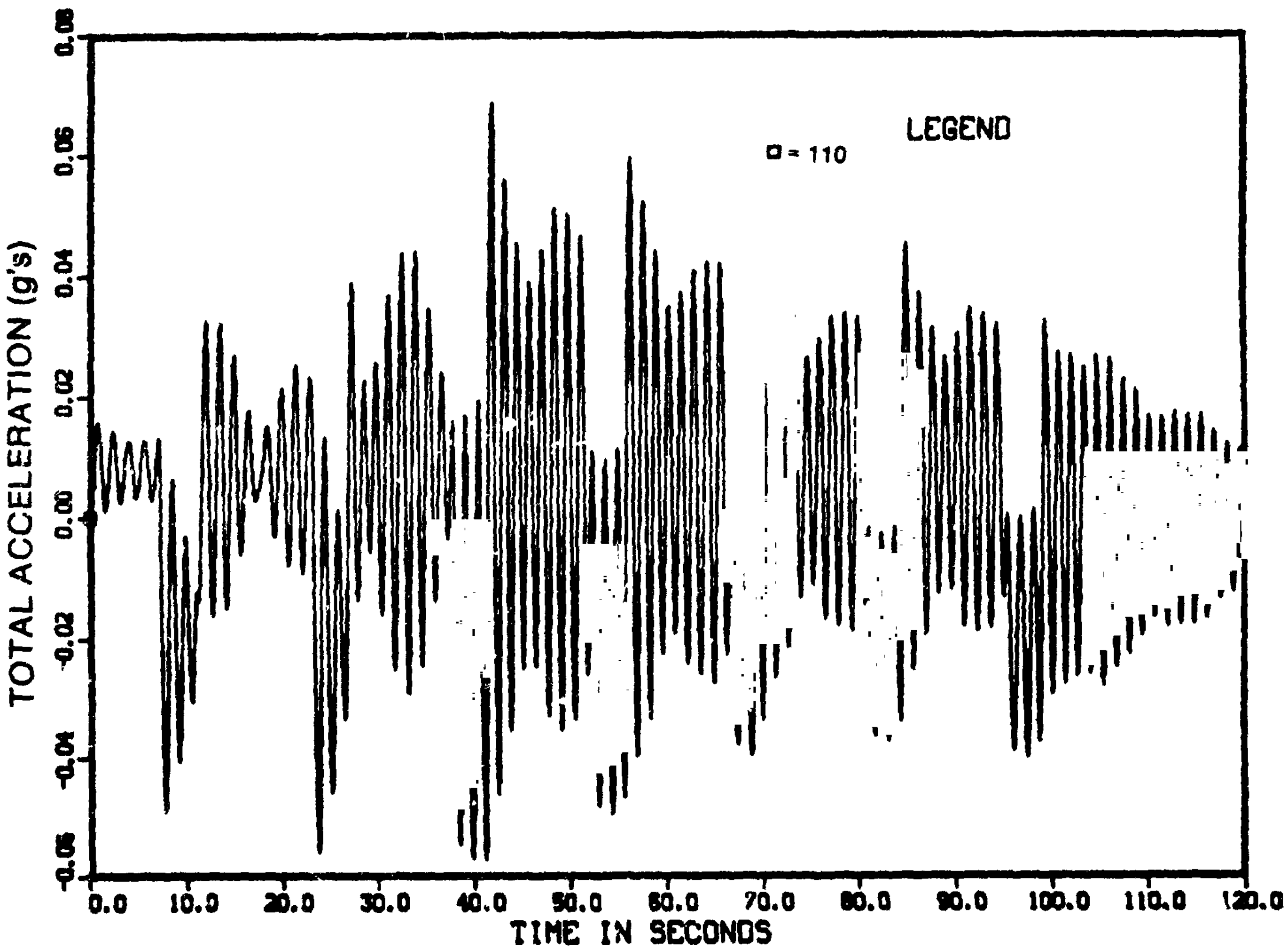


Figure 5.2.5-33 - REBOOST TRANSIENT RESPONSE

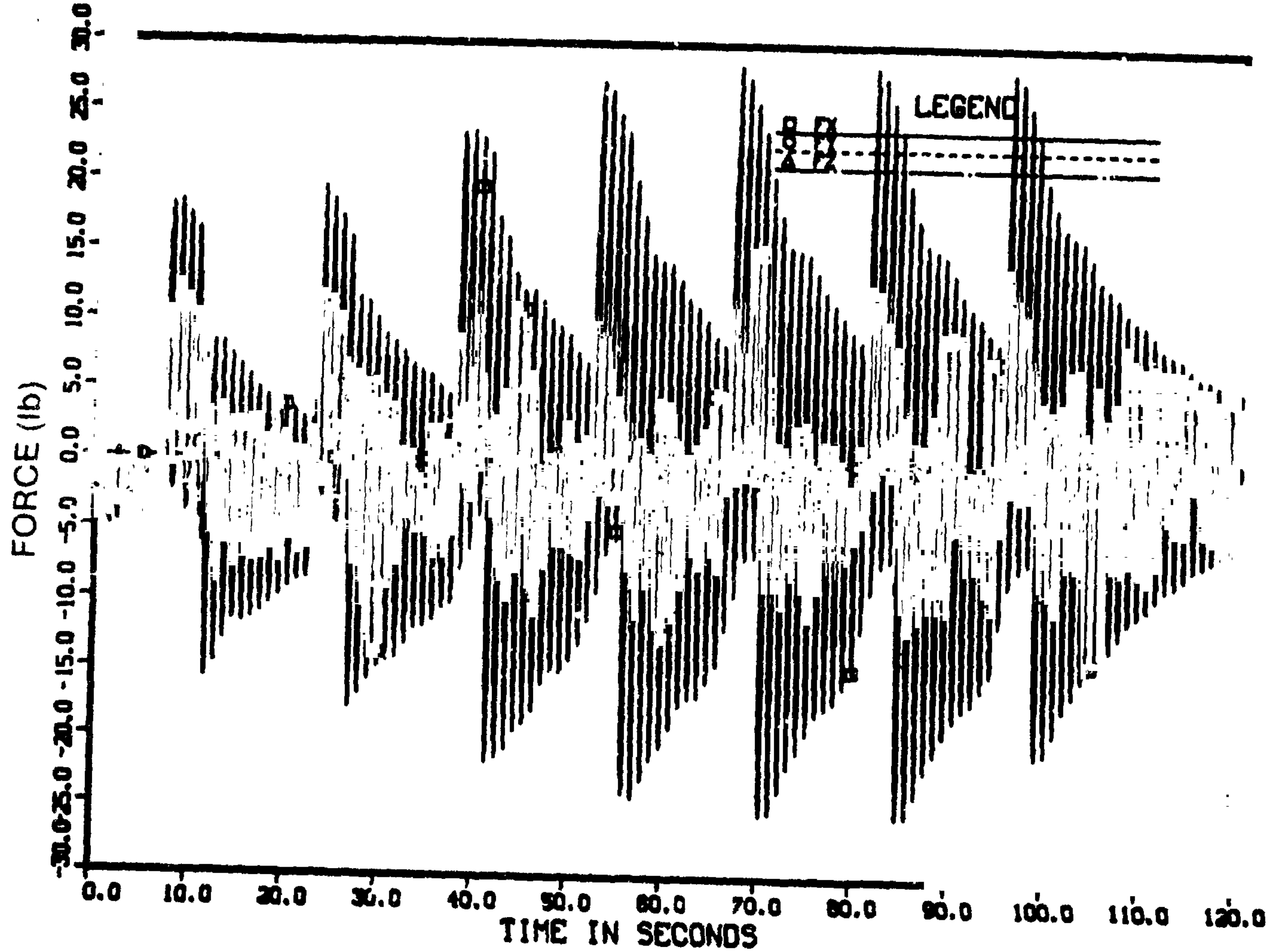


Figure 5.2.5-34 - SOLAR ARRAY REBOOST LOADS

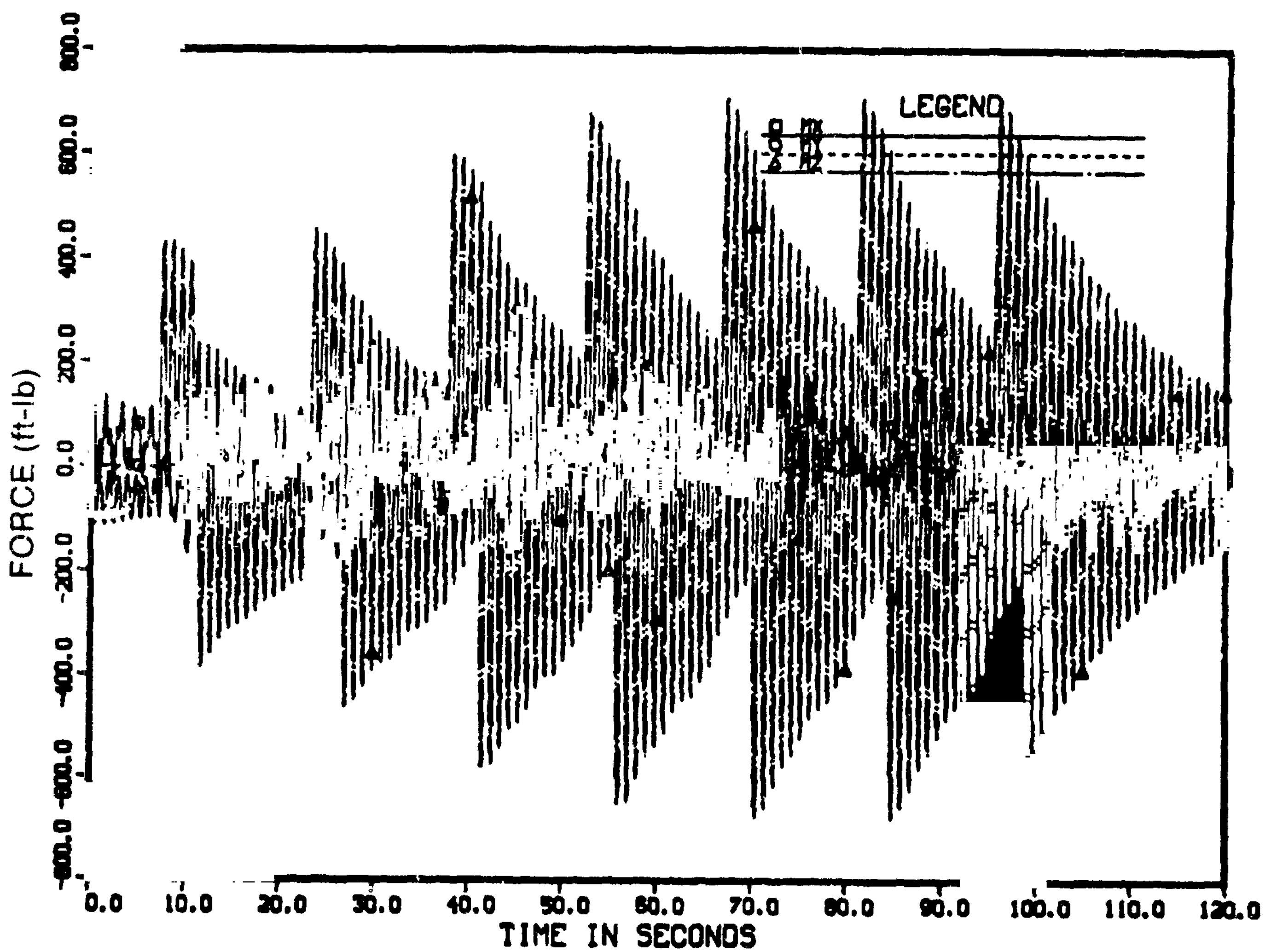


Figure 5.2.5-35 - SOLAR ARRAY REBOOST LOADS

ORIGINAL PAGE IS
OF POOR QUALITY

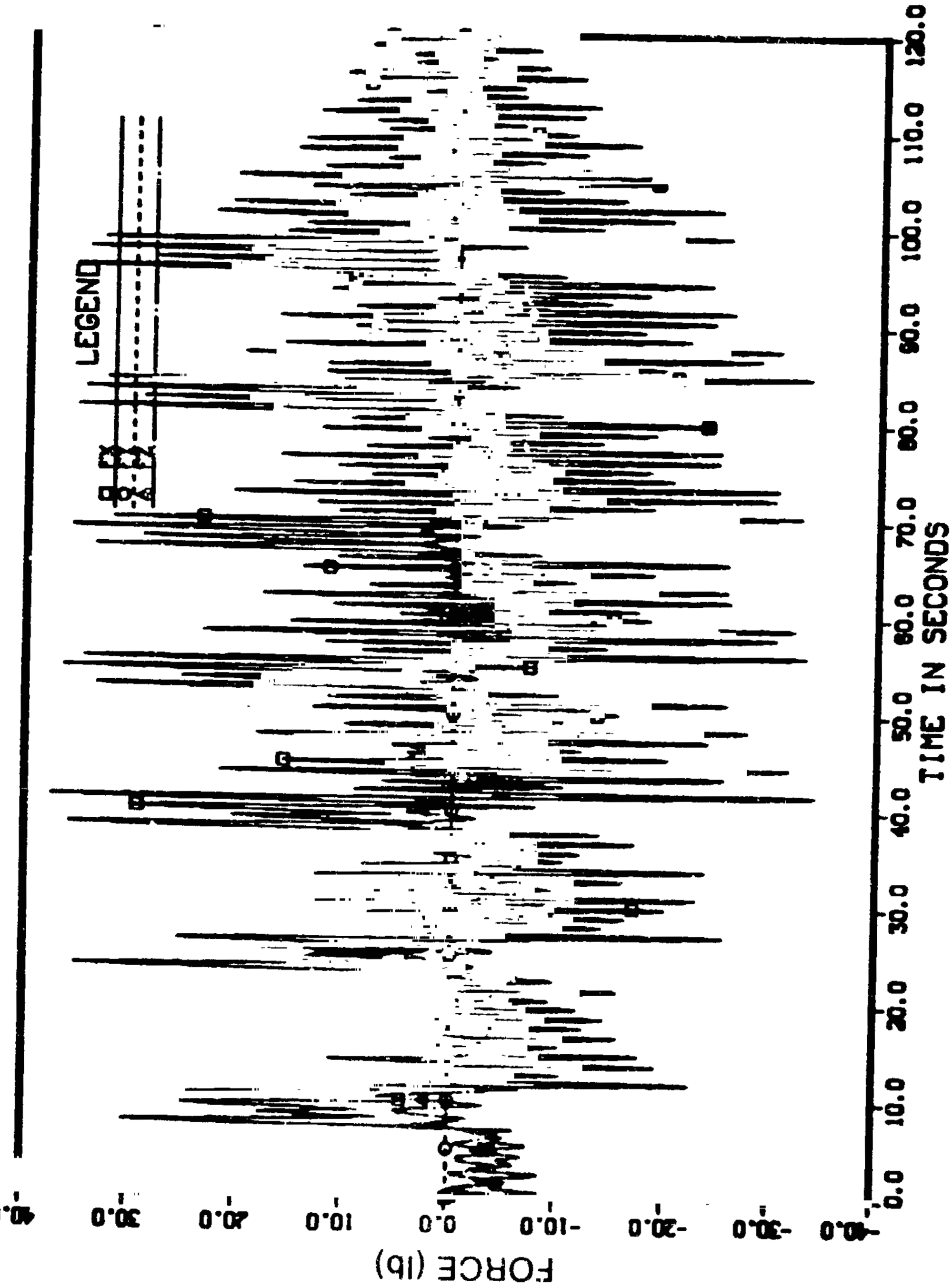


Figure 5.2.5-36 - ANTENNA BOOM REBOOST LOADS

ORIGINAL PAGE IS
OF POOR QUALITY

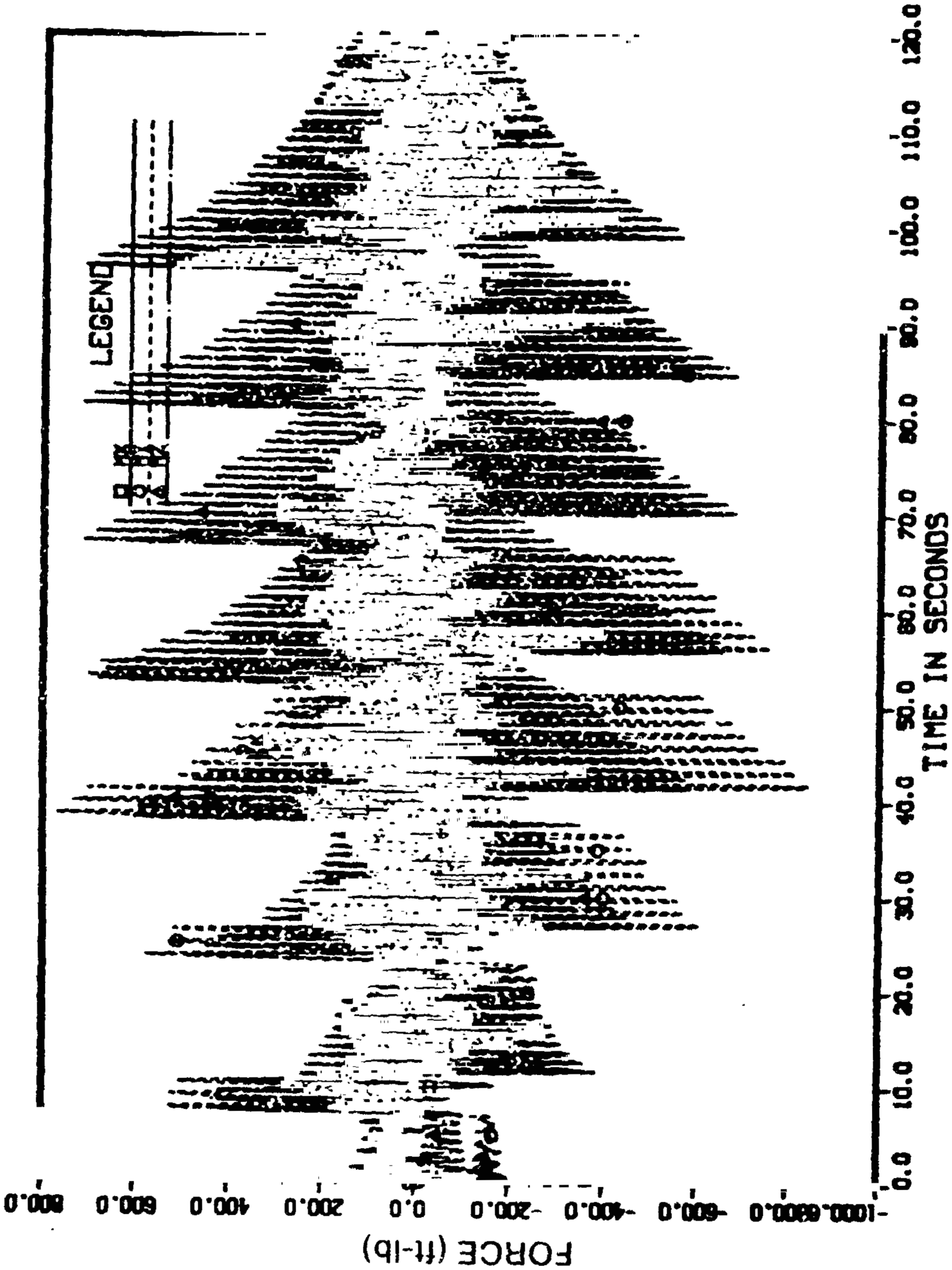


Figure 5.2.5-37 - ANTENNA BOOM REBOOST LOADS

shown for the Starlab Platform representing the initial design (Starlab-1) and the third configuration iteration (Starlab-3) which had added ballast weights and had an added ballast boom opposite the cold solar array. The mass property inertias given in Table 5.2.5 are relative to the Platform's center of mass and use the coordinate system shown in Figures 5.2-3 and 5.2-5 for the Starlab and Polar B, respectively. The products of inertia are computed as the negative integral. The origin of the coordinate system is in the geometric center of the Platform Core.

TABLE 5.2.5-2. - PLATFORM MASS PROPERTIES

	POLAR B	STARLAB-1	STARLAB-3
WEIGHT (LBS)	32,164	24,872	24,394
Inertia (slug-ft^2)			
IXX	40,688	41,710	136,160
IYY	225,876	113,318	147,475
IZZ	230,415	124,908	138,513
IXY	0	-20,814	-1,674
IXZ	479	2,582	642
IYZ	0	-11	-1,534
C.G. Location (ft)			
XCG	-5.11	0.28	0.39
YCG	0.0	0.19	0.93
ZCG	0.61	-0.08	-2.19

5.2.5.4.3 <u>Flight modes simulated</u>

The two Platforms analyzed had different flight modes. The Polar B Platform flies in a Sun synchronous orbit with a LVLH flight mode at an altitude of 380 nmi and an inclination of 98.2 degrees. The repetitive descending node is at 2:00 p.m. The solar arrays and the antenna were held inertially fixed over the orbit.

The Starlab Platform flies in an inertial mode for celestial observation at an inclination of 28.5 degrees. The solar arrays have two degrees of freedom to point at the Sun: (1) an alpha positioning capability about the y-axis and (2) a roll of the Core Platform about its x-axis.

5.2.5.4.4 <u>Control requirements for the disturbance environment</u>

Simulation of the Polar B revealed minimum control requirements over an orbit. The configuration was flown with its solar arrays initially parallel to the x-axis at the subsolar point in the orbit. The peak aerodynamic and gravity gradient torques at the 380 nmi altitude were less than .005 ft-lb. (Figs. 5.2.5-38 and 5.2.5-39, respectively). The aerodyamic density encountered over the orbit is shown in Figure 5.2.5-40. The angular momentum storage requirements are shown in Figure 5.2.5-41.

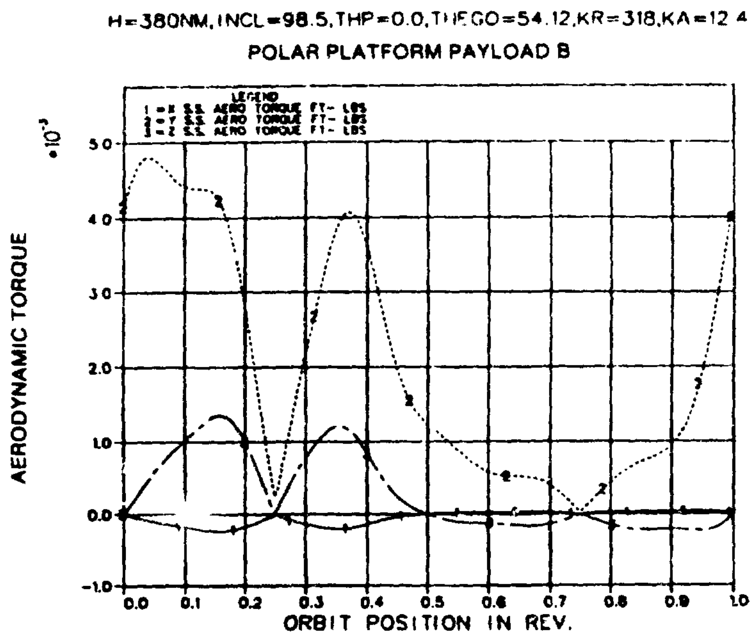


Figure 5.2.5-38 - AERODYNAMIC TORQUE HISTORIES WRT THE BODY COORDINATE SYSTEM

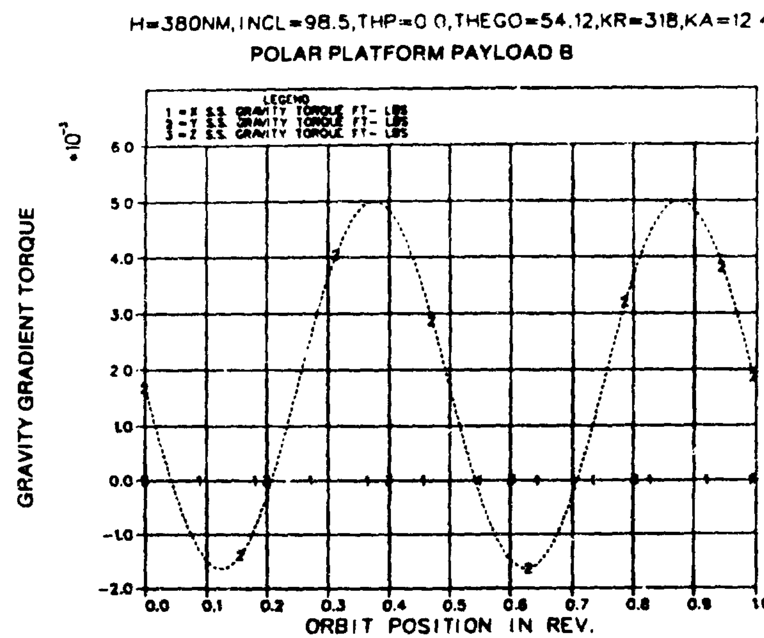


Figure 5.2.5-39.- GRAVITY GRADIENT TORQUE HISTORIES WRT THE BODY COORDINATE SYSTEM

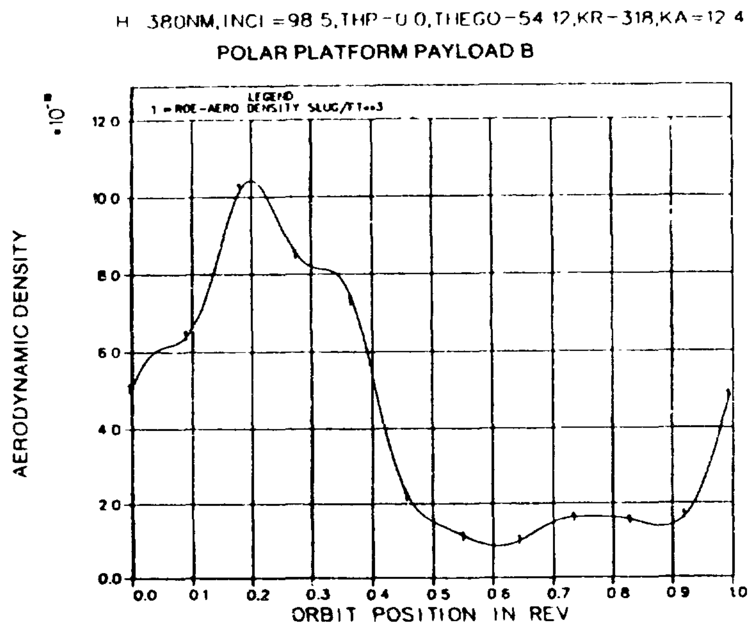


Figure 5.2.5-40 - AERODYNAMIC DENSITY AS A FUNCTION OF ORBIT POSITION

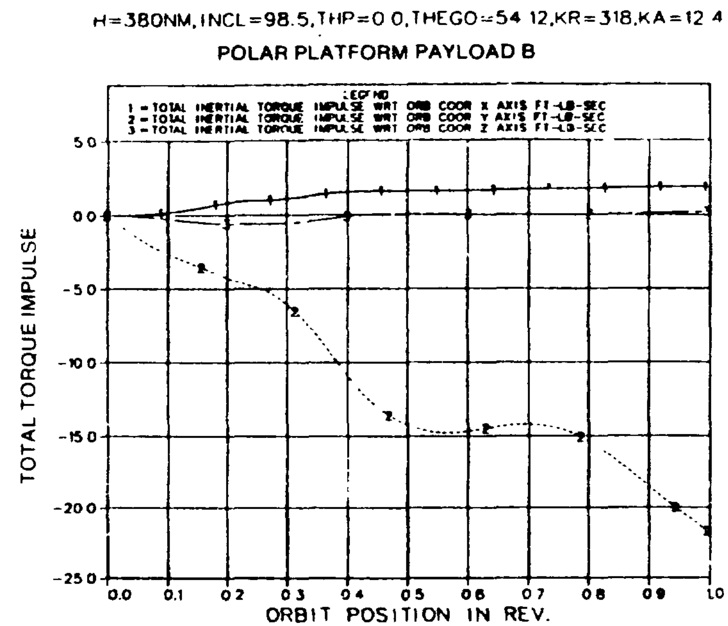


Figure 5.2.5-41 - TOTAL INERTIAL TORQUE IMPULSE HISTORY WRT AN INERTIALLY FIXED ORBIT COORDINATE SYSTEM

The secular y-axis (pitch) momentum storage of 22.0 ft-lb-sec over one orbit can be removed by using magnetic torquers to bias out the average gravity gradient and aerodynamic pitch plane torques.

Initial simulations of the Starlab-1 platform at 270 nmi and at an inclination of 28.5 degrees revealed a large secular momentum buildup over an orbit (see Fig. 5.2.5-42) of 440 ft-lb-sec. This occurs when the Platform is rotated about its z body axis by 30 degrees placing the x principal axis 45 degrees out of the orbit plane. The gravity gradient z axis torque is biased as shown in Figure 5.2.5-43. The 440 ft-lb-sec of secular momentum buildup is large only in the sense that it is desirable to control this axis with a Space Telescope reaction wheel which has a momentum capability of only 200 ft-lb-sec. To remedy this condition, the Starlab Platform underwent several configuration changes involving the addition of ballast weight and a ballast boom. The mass properties for Starlab-3 given in Table 5.2.5 show the inertia properties to be closely balanced. Simulation of the on-orbit flight characteristics of Starlab-3 showed in fact that the momentum requirements were reduced to an acceptable level of 63.0 ft-lb-sec per orbit (Fig. 5.2.5-44).

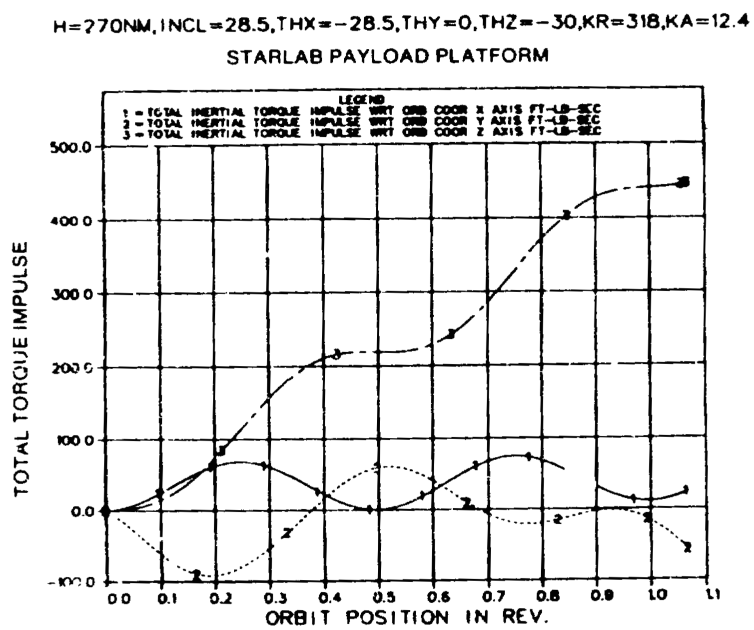


Figure 5.2.5-42 - TOTAL INERTIAL TORQUE IMPULSE HISTORY WRT AN INERTIALLY FIXED ORBIT COORDINATE SYSTEM

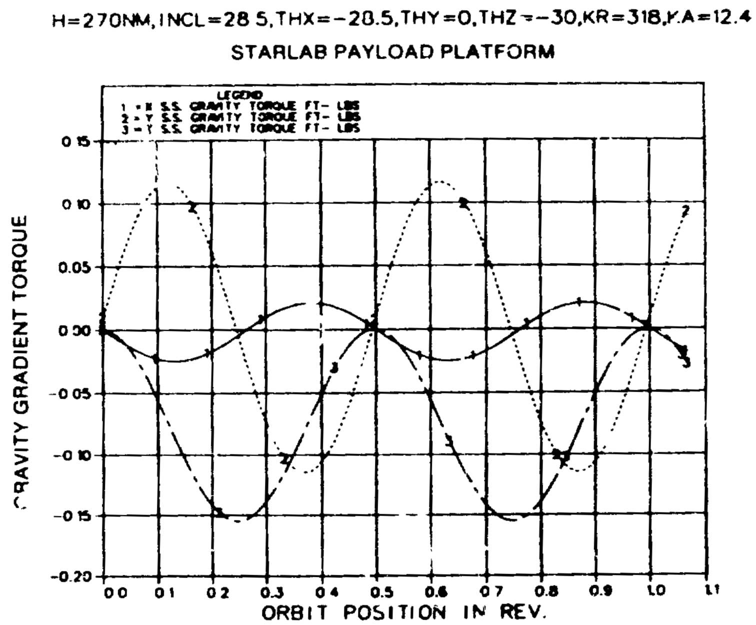


Figure 5.2.5-43 - GRAVITY GRADIENT TORQUE HISTORIES WRT THE BODY COORDINATE SYSTEM

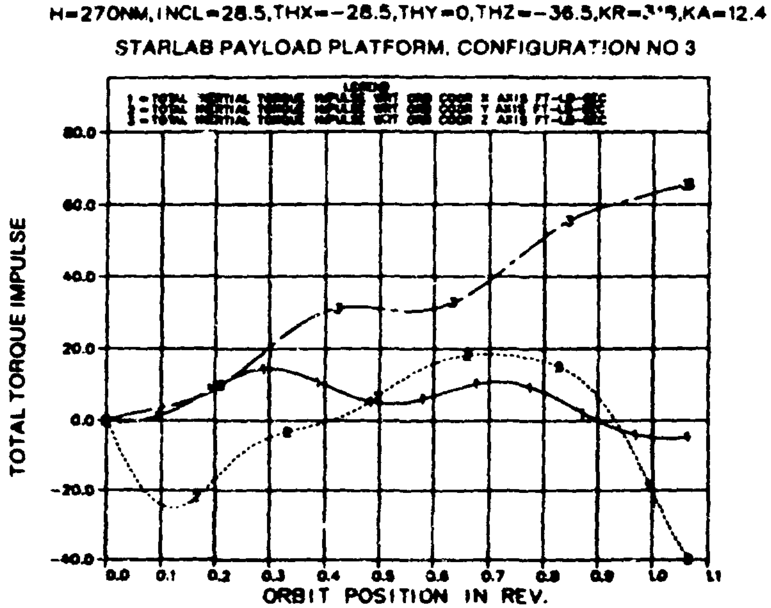


Figure 5.2.5-44 - TOTAL INERTIAL TORQUE IMPULSE HISTORY WRT AN INERTIALLY FIXED ORBIT COORDINATE SYSTEM

5.3 SUBSYSTEMS

The following sections describe the subsystem functions, implementation schemes, common Core Space Station hardware applicability, and their reference design concepts.

5.3.1 Structure

The structure subsystem provides support during launch and operation for all Core and payload module components.

5.3.1.1 Subsystem Functions

a. Provide for ORU mounting.

b. Provide for launch support of deployable appendages.

c. Provide Shuttle launch support.

d. Provide support for payloads.

e. Provide support for solar arrays.

f. Provide grapple for RMS deployment.

g. Provide for OMV docking.

h. Accept launch loads of the ORU and instrument modules.

i. Accept launch loads of the stowed solar arrays and antenna.

j. Transmit launch loads of the payload and Core Platform to the Shuttle cargo bay keel and longerons.

k. Provide support for the propulsion subsystem.

l. Provide the interface with the Shuttle RMS, Space Station and OMV.

5.3.1.2 Potential Hardware Implementation

The Platform Core flat bed structure and instrument flat bed structure (if required) may be a machined longeron/shear plate construction which is capable of accepting distributed or point loading at the payload attachment and ORU module interfaces. While aluminum was assumed in the reference configuration, composites may be used if performance vs. cost is acceptable.

5.3.1.3 Space Station Reference Hardware Applicability

The Platform structure subsystem in a large part will be totally unique from Space Station. Potential commonality exists with modular ORU structure and interfaces. The deployable truss will be of the Space Station design.

5.3.1.4 Platform Reference Design

The co-orbiting and polar Platforms are configured from common hardware elements that can be assembled in a modular fashion to accommodate specific mission requirements. An overall description of the family of IOC Platform configurations is contained in Paragraph 5.2.1. The basic structural elements consist of a Core flat bed structure, instrument flat bed structure (if required), Platform ORU modules and instrument modules. Please refer to Figure 5.3.1-1 for the following discussion.

The Core flat bed will be a rectanglar box structure approximately 12 ft long, 10 ft wide, and 3 ft deep. At one of the 3-by 10-ft ends a bulkhead will be attached. This bulkhead spans the Shuttle's 15-foot diameter and carries the two sill trunnions that react the Shuttle's thrust loads. It also carries the keel pin. Cantilevered from the bulkhead, away from the flat bed structure is the module that carries the fuel tanks. Near the other end of the flat bed structure, two more sill trunnions are provided. These carry only the $\pm$ z Shuttle loads.

Two additional uses for the Core Platform have been identified. In one case, a Core flat bed structure with bulkhead and attached propulsion tanks can be flown in the Shuttle and used to bring fuel, new instruments, and ORU modules to orbit for servicing a Platform. It can also be used to return instruments and ORU modules to Earth.

In addition, a Core flat bed structure could be used to point a cluster of instruments that are attached to the manned Space Station. These instruments might be permanently assigned to the Space Station or they could be "tried out" on the Space Station and later transferred to a Platform as required. This is illustrated in Figure 5.3.1-2. A three-axis gimbal would mount to the top face of the flat bed and the instruments would be attached to the bottom face. The top face could carry any ORU's required such as for the operation of the gimbals. Balancing weights may also be attached to the top face as required so that the center of gravity of the assembly will be near the gimbal axis. Note that each instrument is provided with heat rejection capability as part of the gimballed structure. Power and data handling is provided by the Space Station through the gimbal.

After achieving orbit, a four-bay truss will be deployed from the bulkhead in the same direction as the fuel tanks. Each bay is 9 ft long for a total of 36 ft. This truss uses the same deployable elements as the Space Station truss structure. The truss carries a Space Station type solar array that has been shortened to about 29 ft of active

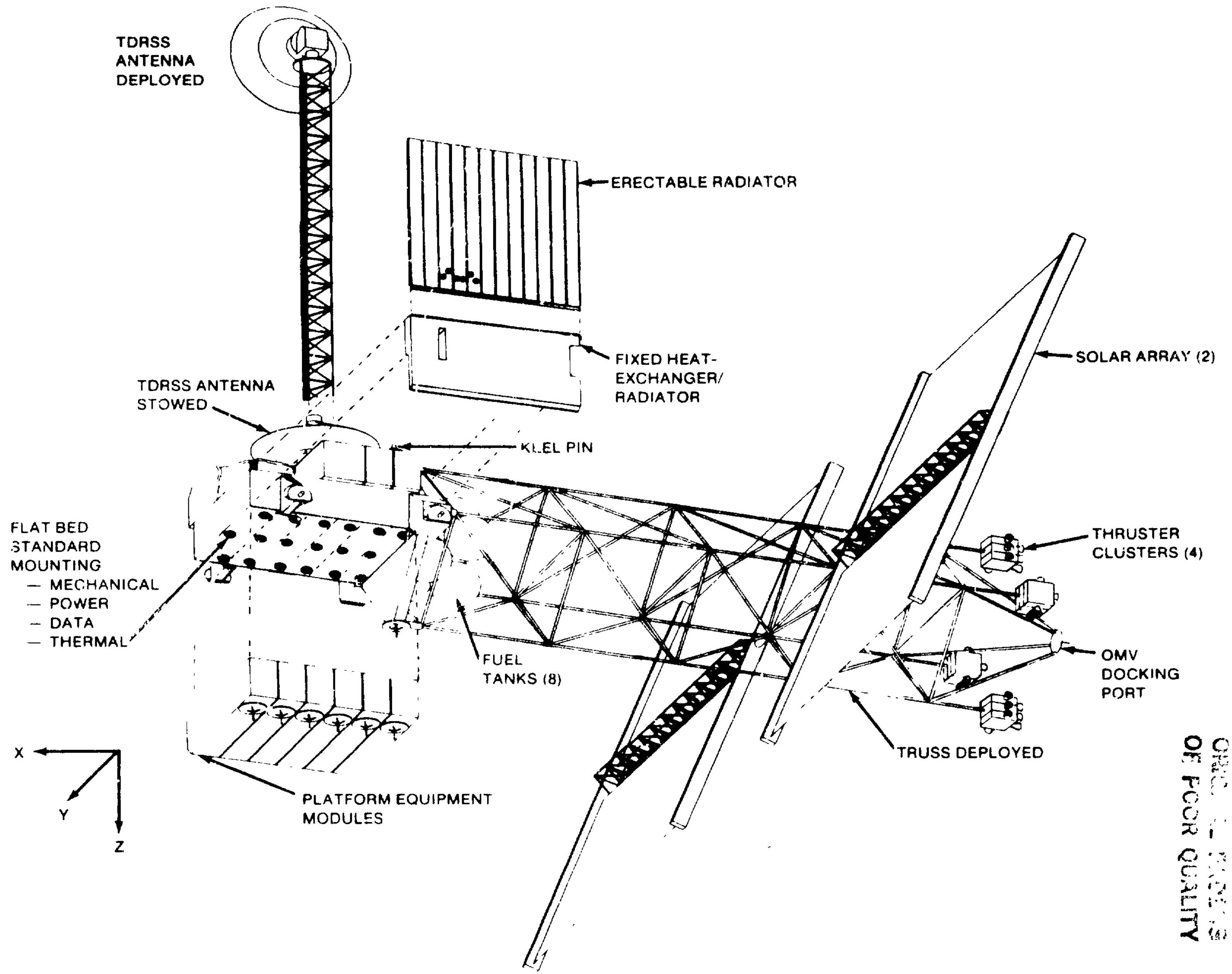


Figure 5.3.1-1 - EXPLODED VIEW OF CORE PLATFORM

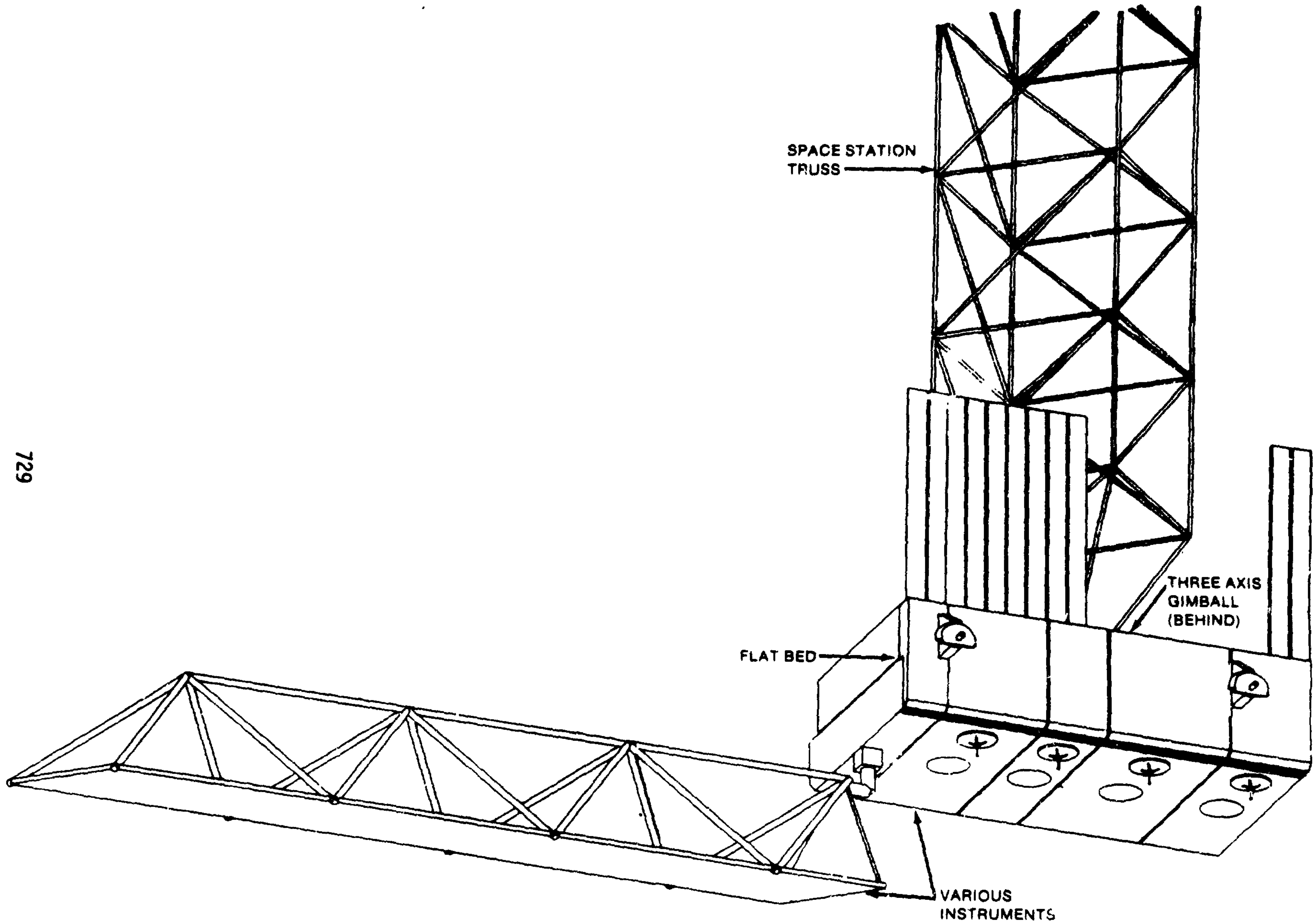


Figure 5.3.1-2 - FLAT BED GIMBALLED FROM SPACE STATION

length. At its far end it also carries four thruster modules and an OMV docking port for maneuvering operations.

All other Platform components are mounted to the Core flat bed structure in modules. These modules mount across the 10-ft width dimension of the flat bed and they are 2 to 4 ft wide. The last 2 feet of the 12-ft-long Core structure is reserved for the attachment of one or more scientific instruments. The large TDRSS dish antenna is deployed 36 feet above the Core structure using the same deployable truss design that is used to extend the Space Station and Platform solar arrays. Heat pipes carry waste heat along the 10-ft dimension of the module to a fixed heat exchanger/radiator mounted at the side of the module that always faces away from the Sun. Erectable radiators derived from the Space Station plug into the fixed heat exchanger/radiator to achieve the required heat rejection for a given mission. The fixed heat exchanger/radiator includes longitudinal heat pipes to evenly share the thermal load among all the radiators.

For those Platform missions requiring additional flat bed structure for scientific instrument mounting, one or more 12-ft-long flat bed structures may be attached to the Core structure. For these missions, the $\pm z$ sill trunnions may be mounted to one of the extension flat bed structures rather than to the Core structure, depending on a loads analysis for that configuration. Similarly, the keel pin may be relocated if a loads analysis so indicates.

Each scientific instrument will use a modular mounting technique like that used for the Platform subsystem. It will span the 10-ft-dimension of the flat bed structure and contain heat pipes and radiators to carry away waste heat. These modules will be a minimum of 2 ft wide and a maximum of about 8 ft wide (in 1-foot increments).

The method of mechanically mounting any module to the flat bed structure so as to minimize errors in instrument pointing will be studied. This will also include electrical and, if required, thermal fluid connections. Since any module may require replacement in orbit, the chosen mounting technique will carefully consider the ease of attaching and detaching a module. The Platforms may be serviced by the STS and/or a teleoperated OMV. The co-orbiting Platforms may also be serviced on the Space Station itself. Any flat bed with instruments that is gimbaled from the Space Station will also be serviced by the Space Station. Since a standard method of servicing all Platforms is desired, all servicing modes will be considered in the module interface design.

5.3.1.5 Weight Table

Table 5.3.1 shows the structure subsystem weights.

TABLE 5.3.1 - STRUCTURE SUBSYSTEM WEIGHT

Item	Weight (lb)
Core Flat Bed	2300
Bulkhead	270
Trunnion Supports	145
Total	2715

5.3.2 Mechanisms

The mechanisms subsystem provides latching, deployment and damping for Platform elements.

5.3.2.1 Subsystem Functions

a. Provide for the deployment of Platform truss structure.

b. Provide for latching, alignment, and unlatching of ORU and instrument modules.

c. Release, deploy, and retract the high gain antenna.

d. Release, deploy, and retract the solar array booms, blanket housings, and solar array blankets.

5.3.2.2 Potential Hardware Implementation

Simple deployments could be accomplished with high speed, high torque motors, stepper motors, or passive stored energy devices. Standard devices such as pinpullers and release nuts may also be utilized.

5.3.2.3 Space Station Reference Hardware Applicability

Solar array deployment mechanisms used on Space Station will be used with shorter blankets for the Platform. Pinpullers and release nuts qualified for use on the Space Station could be used in Platform applications. Commonality may also exist between Space Station and Platform ORU latch design. The deployable truss will be the same design as the Space Station truss.

5.3.2.4 Platform Reference Design

The reference design assumes use of Core Space Station ORU latch mechanisms, solar array stow and deploy mechanisms, and as many miscellaneous devices as feasible. Aerospace standard devices will be employed where Space Station devices are not feasible with the minimum use of special design devices.

5.3.2.5 Weight Table

Table 5.3.2 shows the mechanisms subsystem weights.

TABLE 5.3.2 - MECHANISMS SUBSYSTEM WEIGHT

Item	Weight (lb)
ORU Latch Mechanism	324
Solar Array Stow and Deployment	196
Miscellaneous Deploy Devices/Dampers	60
Total	580

5.3.3 Thermal Control

5.3.3.1 Overview

The Platform reference configuration Thermal Control System (TCS) is composed of two-phase ammonia cold plates and heat pipe radiators which service Platform ORU modules located on the Core flat bed and instrument modules located on additional flat beds as required. Multilayer insulation covers all flat bed surfaces with the exception of radiators and sensor openings.

The propulsion system utilizes electrical heaters, multilayer insulation and appropriate surface coatings common to the Core Station thermal design (ref. section 4.4.7.4.5). Likewise, the communications and tracking hardware not located on the flat bed have an autonomous TCS design common to that of the Core Station (Ref. Section 4.4.7.4.6). Heaters will be controlled by an on-board computer using thermal sensor input data.

The ORU and the flat bed-mounted instrument modules contain cold plates and connections to the fixed radiator heat exchanger. The heat exchanger contains a heat distribution system which distributes the heat to the erectable radiators. This method of control allows highly variable heat loads while maintaining nearly constant or isothermal temperature control. The thermal coupling between an instrument module or an ORU module and the fixed radiator heat exchanger may be through flexible heat pipes if precise alignment is required. Techniques required to make and break these thermal connections to allow servicing will be the subject of a trade-off study. The erectable radiator elements are common to the Core Station but of shorter length. Payloads not requiring standard instrument modules will provide an autonomous thermal control system.

5.3.3.2 Subsystem Functions

The TCS will maintain the Platform within allowable temperature limits for all flight modes from launch to the end of the mission, including all servicing modes. The TCS is required to control Platform and payload interface temperatures for orbital average heat rejection loads varying from 4.5 kW to approximately 9.9 kW. For the 9.9 kW Platforms, the total instrument power is 4.5 kW, housekeeping power is 2.9 kW, and the power system losses are approximately 2.5 kW. The instrument module interface temperatures should be held to within + 5°F of the set point temperatures. Environmental inputs which must be accommodated follow:

a. Those imposed on stellar or solar pointing co-orbiting Platforms at 270 to 540 nmi with a Beta angle of 0 to 52°

b. Those imposed on Earth-oriented polar Platforms where the angle between the orbit plane and the solar vector will be from 40 to 25°.

5.3.3.3 Subsystem Capabilities and Sizing

Both the Core Platform and instrument flat beds have a common thermal control design with a capability of rejecting 5.4 kW (Core Platform heat rejection requirement) of waste heat. Instruments have the option of utilizing the flat bed thermal control capability or providing their own autonomous system. Also, the instrument flat bed radiator area can be reduced for heat rejection level requirements below 5.4 kW by removing the appropriate number of radiator panels from the fixed radiator heat exchanger.

The radiators are mounted off the -y face of the flat bed, Figure 5.2-2, in the xz plane. The total radiator (including the fixed radiator heat exchanger radiator) is 12 ft wide by 18 ft long (216 ft^2). The top 9 feet of the erectable radiator radiates from both sides. The above radiator was sized for a polar Platform and the following parameters:

Coating properties	-	$\epsilon = 0.8$
		$\alpha = 0.2$
Fin efficiency	= .84	
View factor to space	= .95	
Radiator temperature	= 70°F	

Under these conditions, the polar Platform radiators can reject approximately 20 watts/ft^2 in the -y direction and 10 watts/ft^2 in the +y direction. The co-orbiting platform can reject approximately 18 watts/ft^2 from the -y face and 15 watts/ft^2 from the +y face.

Depending on the variations in ORU and instrument module power profiles and the flat bed thermal balance, heaters will be used as required to minimize structural gradients, to provide make-up heat, for propellant tanks, thruster valves and lines, and for non-flat bed communications and tracking hardware which are common to the Core Station (Ref. Sections 4.4.7.4.5 and 4.4.7.4.6).

For the full 9.9 kW heat rejection load, the TCS weight requirement is estimated to be 1000 pounds for the Platform Core and payload; 600 pounds is the estimated TCS weight for the Platform Core alone.

5.3.4 Electrical Power

The electrical power subsystem of the Platforms provides the facilities for the generation, storage, control, and distribution of electrical energy.

5.3.4.1 Subsystem Requirements

a. Energy storage: The energy storage system will be sized for the near-polar orbit described in Section 5.1.2 which has a maximum solar occultation period of 36 minutes and a minimum insolation period of 63 minutes.

b. Distribution: Power will be managed and distributed in a manner which permits the sensing, isolation, and correction of non-catastrophic faults.

c. Redundancy: The power subsystem will contain sufficient redundancy (block, functional, or other) to satisfy the system lifetime requirement and the fault tolerance requirements of Section 5.1.1.

d. The near-term Platforms will provide a total of about 5.0 kW power input for customers.

e. Provision for growth: The power subsystem will be capable of growth to support a continuous payload power level of 20 kW.

5.3.4.2 Potential Hardware Implementation

5.3.4.2.1 Power generation

Roll-out or fold-out solar panels are possible implementations for power generation. There may be occasions when it will be necessary to retract the panels. Total array area will be divided into two panels and located so as to not block instrument and radiator fields of view while not being shadowed by the Platform.

5.3.4.2.2 Energy storage

Rechargeable nickel-cadmium batteries, nickel-hydrogen batteries, or regenerative fuel cells are potential choices for Platform energy storage. The NiCd represents the lowest risk and longest lifetime alternative at this time.

5.3.4.2.3 Power conditioning and distribution

Power conditioning, ac or dc power distribution, and the distribution voltage to the users must be determined as a function of Space Station commonality, control, efficiency, environmental effects, and fault detection and correction. Fine regulation may be left to the users.

5.3.4.3 Space Station Reference Hardware Applicability

There are strong cost incentives for hardware and software commonality between the Platforms and the Space Station. Solar array technology, power conversion equipment, and perhaps power control and distribution equipment can use Space Station counterparts directly or by adaptation.

5.3.4.3.1 Conversion, control, and distribution equipment

The Platform power subsystem must handle high power levels, including growth capability to 20 kW of payload power. The Platform power conditioning module design should replicate those of the Space Station if that design is small enough to accommodate Platform redundancy.

5.3.4.3.2 Solar array

The Platform power subsystem will have the same operating voltage as the Space Station to make the design and the hardware developed for the Space Station solar arrays applicable. The only substantial difference will be in reducing the number of parallel strings to the number required for the shorter array used by the Platform.

5.3.4.3.3 Energy storage

The Space Station Regenerative Fuel Cells (RFC) will be excessively large for Platform use if sized for 20 kW as in the Core Space Station reference design. Use of RFC's for Platforms is therefore dependent on results of Phase B studies.

5.3.4.4 Platform Reference Design

The Platform reference design has a total user and housekeeping leveled continuous power requirement at IOC of 7.4 kW. Some instrument loads draw peak powers which are much larger than their average powers. For this reason the reference design uses the battery-share mode (load leveling) to handle the peak power requirement, thereby decreasing the size of the solar array needed.

Significant features of the major components of the reference design are discussed below.

5.3.4.4.1 Power generation

The reference design uses a conventional technology planar, silicon-cell solar array for power generation. A power density of 20 W/lb is assumed. The area of the reference design array is 1720 ft^2 (at 11.15 W/ft^2 BOL). Array switching is employed for coarse voltage control and array shorting where the full array power is not needed. Capability to increase array size for later missions will be provided.

5.3.4.4.2 Energy storage

The reference design uses conventional nickel-cadmium batteries with an assumed efficiency of 76 percent . If used at an average depth of discharge of 20 percent, a total capacity of 800 ampere-hours will be needed in the Core Platform. Additional battery capacity will be provided for especially demanding payloads (Polar C and SAR) within an ORU module mounted either on the Core Platform or on the instrument flat bed. The Core Platform design includes capability to connect these additional batteries into the subsystem.

5.3.4.4.3 Conversion and distribution equipment

The Platform reference design incorporates distribution of power at the same voltage and frequency as the Space Station (high voltage, possible ac distribution). The overall efficiency of power conditioning is estimated to be 95 percent. The reference design has three 5 kW Space Station conversion modules, one being a spare. Distribution equipment is assumed to be designed to meet Platform requirements and will use Space Station hardware as appropriate.

5.3.4.5 Weight Table

Table 5.3.4 shows estimated weights for items comprising the electrical power subsystem.

TABLE 5.3.4 - WEIGHT TABLE
PLATFORM ELECTRICAL POWER SUBSYSTEM

ITEM	WEIGHT(LB)
Solar Array	964
Batteries 800 Ah	1,764
Conversion Equipment	132
Control & Distribution Equipment	154
	3,014

The distribution equipment was assumed to use 100 W. Other power equipment efficiencies are as stated above.

5.3.5 Communications and Tracking

The communications and tracking subsystem provides communications between the Platform and TDRS, GPS, and ground as well as the Space Station, STS, and OMV as required.

5.3.5.1 Subsystem Functions

a. Transfer to the ground:

(1) Science data

(2) Engineering data

(3) Emergency data for failure evaluation

b. Receive commands from the ground.

c. Support measurement of relative range and range rate with respect to the TDRS/TDAS.

d. Support orbit determination with GPS.

e. Support rendezvous and docking operations with the STS and the OMV.

f. Support transfer of data between co-orbiting Platforms and the Space Station.

5.3.5.2 Potential Hardware Implementation

The communications and tracking system will be essentially the same for all Platforms, except that a co-orbiting Platform will have a direct link to the Space Station. The Platforms will have the capability to communicate through the TDRS/TDAS systems on both S-Band and Ku-Band single access channels, have a GPS link as a navigational aid, be capable of receiving commands from and transmitting telemetry and TV to the Space Station and receive commands from and transmit telemetry and TV to an Orbiter, as well as interfacing with the Orbiter's rendezvous radar system. An S-band Multi-Access (MA) low-rate data TDRS/TDAS link for housekeeping purposes will also be considered.

The communications and tracking subsystem consists of turn-around transponders, modulators/demodulators, power amplifiers, and appropriate switching at S-Band and Ku-Band for command, telemetry, and tracking through the TDRSS. In order to accomplish this throughout the orbit, a steerable, high-gain antenna will be required. This will be a 9-ft antenna for commonality with the Space Station antennas, although a smaller antenna would suffice. For appropriate viewing of the TDRS, the high gain antenna will be on a boom. To reduce the requirement for generation of high RF power, the transponders must be at the end of this boom near the antenna. The antenna will have combined S-Band and Ku-Band feeds. Pointing will be accomplished mechanically during acquisition and electronically during tracking acquisition and electronically during tracking. To handle the various vehicle orientations, the feed must be circularly polarized. A BER of less than 10^{-5} will be achieved without coding.

Low gain S-Band antennas will also be provided for communications through the TDRS at roughly 1-3 kbps. Switching and sequencing is provided through external controls. Low gain antennas may be required for direct ground communications.

Communications are received from the GPS through a low gain antenna. The antenna should be located to have the most advantageous position to view the GPS.

Communications with the STS during rendezvous and docking operations will be provided through a NASA Standard Transponder (S-Band) and low gain antennas. The antennas can be the same low gain ones used for the low data rate TDRS link but consideration should be given to separate ones located in order to support the docking operation. This link will transfer commands and telemetry between the co-orbiting and polar Platforms and the STS and the OMV. In addition, TV is transmitted from the Platforms to the STS.

Communications between the co-orbiting Platforms and the Space Station will be through a separate transponder and antenna. The antenna will be located to view the Space Station during all orbits. A steerable antenna may be required even though the antenna size will be small. The link will support both 5 Mbps and 64 kbps channels from the co-orbiting Platform to the Space Station and will also support 48/64 kbps channels from the Space Station to the co-orbiting Platform. Although equipment for this function is included in the Platform reference design, the incorporation of this capability in the payload module will be studied in Phase B. The use of slow scan video on the platform as a rendezvous aid will also be evaluated.

5.3.5.3 Space Station Reference Hardware Applicability

The Space Station antennas and transmitters/receivers are used in the Platform reference design. Also assumed were the Space Station GPS system and communications processors which are included in the information and data management subsystem.

Feasibility of the use of a smaller TDRS antenna and reliability tradeoffs between one and two antennas should be studied in Phase B.

5.3.5.4 Weight and Power Table

Table 5.3.5 shows communications and tracking weights and power requirements.

TABLE 5.3.5 - WEIGHT AND POWER TABLES
PLATFORM COMMUNICATIONS AND TRACKING SUBSYSTEM

EQUIPMENT	QUANT.	WEIGHT EACH(LB)	TOTAL WGHT(LB)	TOTAL PWR(W)
ANTENNAS:				
TDRS H. Gain Ku/S-Band	1	100	100	200
TDRS Low Gain S-Band	2	2	4	-
Med. Gain Ant.	2	10	20	50
GPS Low Gain Antenna	2	2	4	-
Radar Xpndr Low Gain	2	2	4	-
RF SYSTEMS:				
TDRS S/Ku-Band Xmit/Rec.	2	80	160	400
*SS Link LDR Xmit/Rec.	2	40	80	75
S-Band Xpndr (ORBITER)	2	45	90	80
*VIDEO:				
TV Camera	4	5	20	40
TV Pan/Tilt	4	8	32	80
TRACKING:				
GPS Rec./Proc.	2	44	88	188
Radar Xpndr	2	5	10	20
CABLE:		350		
TOTAL			962	1,133

* To be flown only on co-orbiters.

5.3.6 Information and Data Management Subsystem (IDMS)

The IDMS provides command and data processing and storage and serves as the Platform executive.

5.3.6.1 Subsystem Capabilities

The Platform IDMS will employ a distributed architecture that allocates processing services resident in each subsystem.

a. The combined onboard information rate is from 300 kbps to 300 Mbps and will be made available for transmission through TDRSS at rates from 20 to 300 Mbps.

b. The uplink command rates will be less than 300 kbps over S-Band with a similar rate over Ku-Band.

c. Instrument data storage requirements are from 10 to 600 gigabits.

d. Housekeeping data may be transmitted in real time to the ground via an S-Band link through TDRSS and/or recorded on a special housekeeping recorder for later transmission.

5.3.6.2 Subsystem Functions

The data management system will supply time and frequency standard signals and housekeeping/engineering subsystems data to other spacecraft systems as well as all control and synchronizing signals necessary for commanding, data collection, and system monitoring.

The command process will include reception, validation, expansion, interpretation, storage, distribution, and execution of commands in packet form. The command types will be immediate execute, delayed execute, and stored. Commands may also be generated internal to the Platform in response to on-board algorithms or in the form of command support requests from the subsystems or payload.

Data collection and processing of science and engineering data packets by the IDMS will include adaptive sampling including memory dumps of the payload instruments and engineering subsystems, merging of data streams, formatting of the data (and encoding where required) for storage and/or immediate transmission. The IDMS will provide subsystem data bases for inter- and intra-subsystems access and storage. The IDMS will monitor the Platforms health and status in support of autonomous operation.

Instruments that generate very high data rates or very large masses of data may be required to provide their own separate bit stream. The IDMS would not alter the information content of any bit stream, but would route it to the appropriate transmitter or mass data storage device. Each Platform typically has one (maximum of two) instrument of this class.

5.3.6.3 Potential Hardware Implementation

A distributed data collection and command system implementation is envisioned with a redundant serial fiber-optic bus providing the path for all information (commands and data) exchanges (see Fig. 5.3.6). The science payload instruments and engineering subsystems are assumed to be processor based with sufficient memory to process and compose data packets. They will respond to, decode, and store (if necessary) higher order commands. Payload instruments and engineering subsystems will interface with the fiber-optic bus network via interface devices (ID's). The engineering subsystems and payload instruments may use general purpose subsystem data processors (SDP's) which, under software control, will provide subsystem peculiar processing, computation, and storage.

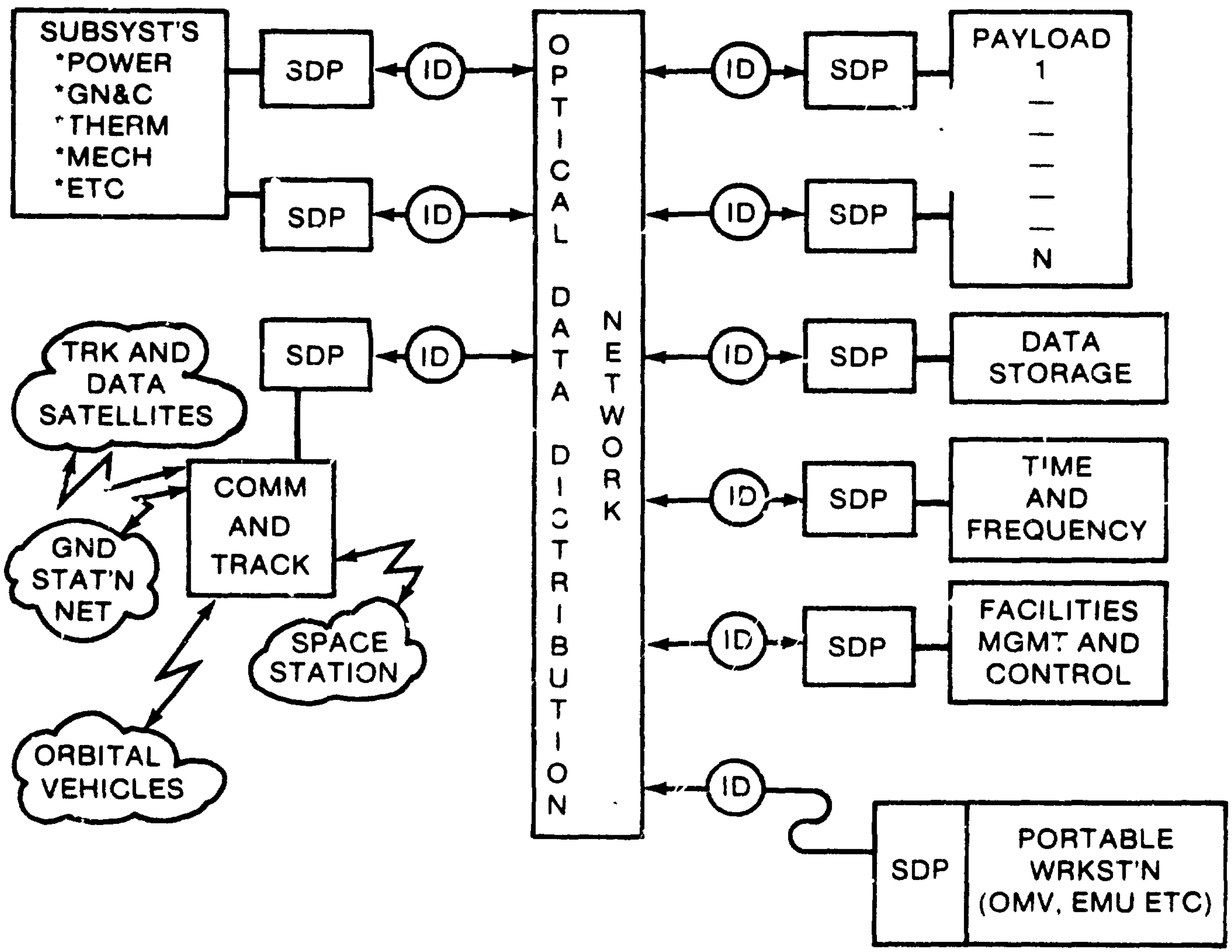


Figure 5.3-6 - INFORMATION AND DATA MANAGEMENT SUBSYSTEM PLATFORM

5.3.6.4 Space Station Reference Hardware Applicability

The Space Station IDMS hardware is appropriate for use on the Platforms. This assumes that those functions related to workstation control (i.e., human control) can be deactivated. The Space Station mass storage devices, which are as yet not well defined, may not be applicable due to the higher data rates and higher storage requirements expected on the Platforms.

5.3.6.5 Platform Reference Design

The reference IDMS configuration is a distributed architecture using functional elements common with the Space Station. The core data distribution network will be fiber-optic. The Platform reference tape recorders are assumed to be larger than the Space Station reference mass storage devices. Two large (new development) recorders are included in the Platform reference design.

5.3.6.6 Weight and Power Table

The IDMS, for weight and power purposes, consists of a core system which will be the same for each Platform. Bookkept separately is the weight and power for two mass storage devices which will be chosen for each Platform according to data requirements. The Platform reference design includes two large digital tape recorders (DTR). This information is contained in Table 5.3.6.

TABLE 5.3.6 - IDMS WEIGHT AND POWER

ELEMENTS	NO.	WEIGHT EACH(LB)	TOTAL WEIGHT(LB)	POWER EACH(W)	POWER TOTAL(W)
Opt Net/Cabl'n	1	50	50	-	-
IDMS SDP's	5	20	100	50	250
Data Storage	3	70	210	50	150
Time & Freq	3	20	60	50	150
Facil Mgmt/Ctl	3	20	60	50	150
IDMS ID's	5	4.5	26	20	100
Subsystem ID's	13	4.5	59	20	260
Payload ID's	12	4.5	54	20	240
DATA ACQ./TM/CMDS:					
TM Processor/Cont.	1	40	40	75	75
Low Rate Mux.	3	10	30	25	75
High Rate Mux.	1	70	70	120	120
SIGNAL PROCESSING:					
TDRS Signal Processor	2	20	40	25	50
SS Link LDR Signal Processor	2	10	20	25	50
SS Link HDR Signal Processor	1	10	10	20	20
Digital TV Processor	2	6	12	50	100
TOTAL			841		1,790

5.3.7 Guidance, Navigation, and Control (GN&C)

The GN&C subsystem provides Platform base-body control, solar array pointing, antenna pointing commands, GPS data processing, and supports instrument pointing functions.

5.3.7.1 Subsystem Characteristics

The GN&C subsystem, in conjunction with the propulsion subsystem, communications and tracking subsystems, and the information and data management subsystem, provides the basic functions of guidance, navigation, attitude control, orbit maintenance, and passive proximity operations. Normal operation of the GN&C will not require interaction or assistance from ground station systems; however, the GN&C will be capable of accepting ground backup commands.

a. The GN&C subsystem will provide the following Platform performance:

Accuracy (3 sigma) without fine error sensor (FES)	.03 deg
Accuracy with FES	FES dependent
Knowledge of pointing (without FES)	0.01 degree
Knowledge of pointing (with FES)	FES dependent
Stability	0.002 deg/10 min
Jitter	0.0003 deg/1 sec

b. The GN&C subsystem will have the capability to store and dump the secular momentum expected to result from atmospheric drag, solar pressure, gravity gradient, and Platform rotation, and must do so for all initially planned configurations.

c. The propulsion subsystem will provide a fail-safe backup attitude control capability for a contingency period of 12 months.

d. Platform position will be derived from the Global Positioning System (GPS) to an accuracy of 330 ft using the coarse code. Platform position will be controlled in polar orbit to maintain a ground track repeat every 16 days (233 orbits) with ground traces within $\pm$ 5.4 nmi (cross track).

e. Solar array pointing control accuracy will be maintained to a suitable level. Array drive rotation rates will consider acquisition time, Earth orbit rate, and instrument slew requirements.

f. Programmed yaw capability will be provided for alignment of instrument ground tracks.

5.3.7.2 Subsystem Functions

The GN&C Subsystem will provide the following functions:

a. Three-axis attitude determination, acquisition, and control for nadir orientations and celestial or Sun-referenced inertial orientations.

b. Solar array pointing and control, and high gain antenna pointing and control commands.

c. Platform position determination and control commands.

d. Delta-V control commands.

e. Relate Platform attitude and attitude rate to that of payloads via an attitude transfer system.

f. Momentum management.

g. Fault detection, isolation, and reconfiguration within the GN&C system.

h. State vector prediction, propagation, and transfer to payloads.

i. Control and safing for proximity operations associated with berthing, deployment, and recovery.

j. Establish and maintain a survival mode which has adequate power collection and control capability to ensure spacecraft survival until the next servicing opportunity.

k. On-board alignment and calibration of sensors.

l. Support for the functioning of instrument pointing systems if present and the ability to accept data from instrument resident sensors to be used to supplement basebody attitude control sensors.

m. Control of reaction control thrusters contained in the propulsion subsystem.

n. Coordinate transformation capability.

o. Payload articulation devices, such as momentum compensated gimbals, as needed to meet payload pointing requirements.

5.3.7.3 Potential Hardware Implementation

The basic Platform hardware complement is assumed to consist of the following components:

a. Actuators:

(1) Momentum exchange devices (MED) such as reaction wheels or small CMG's.

(2) Momentum desaturation control actuators such as magnetic torque rods, or devices for active gravity gradient control.

(3) Solar panel drive actuators and control electronics.

b. Sensors:

(1) Absolute attitude reference devices such as star trackers.

(2) Relative attitude sensors to function in an attitude transfer system.

(3) Acquisition Sun sensors.

(4) Target reference devices such as fine guidance sensors or high accuracy star trackers.

(5) Inertial reference devices such as strapdown gyro assemblies, rate integrating gyros, and accelerometers.

(6) Magnetic field sensors such as fluxgate magnetometers.

c. Processors: GN&C processors suitable for navigation and control functions and supporting interface devices.

The following services are assumed to be provided by other Platform subsystems:

(1) GPS data from the communications and tracking subsystem.

(2) High gain antenna drive assembly as part of the communications and tracking subsystem.

(3) Reaction control, translation, drag makeup, and control torque during maneuvers as part of the propulsion subsystem.

(4) Fine guidance sensing by the instruments as needed.

(5) Rendezvous and docking support from the communications and tracking subsystem.

5.3.7.4 Space Station Reference Hardware Applicability

The following Space Station hardware components should be considered for use in the Platforms:

a. Magnetic torquers.

b. Star tracker triad.

c. Inertial reference units.

d. GN&C processors and associated ID's.

e. Attitude transfer system.

Other Space Station components should be examined for applicability.

5.3.7.5 Platform Reference Design

The Table in Section 5.3.7.6 gives the system components which are anticipated for use in the Platforms. The number of units, their heritage, and the relation to the Space Station are also shown. This section provides reference design rationale regarding the selection of components listed.

a. Reaction wheels - four are chosen for redundancy. Space Telescope heritage is used for reference.

b. Magnetic torque rods - four are chosen for redundancy. Space Telescope heritage is assumed for consistency with the Space Station reference configuration.

c. Star tracker triad - two are assumed for redundancy. Space Station reference configuration component used.

d. Two-axis Sun sensors - two are assumed. Heritage not identified but NASA standard should be considered.

e. Magnetometers - two are assumed for use in the momentum management system. Heritage not assumed.

f. Inertial Sensor Assembly (ISA) - one strapdown hexad unit assumed to have adequate redundancy. Space Station commonality assumed. Ring laser gyros and filament gyros may be used.

g. GN&C processor and associated ID - two assumed for redundancy. Space Station commonality anticipated.

h. Solar panel drive and electronics - representative numbers shown, new design. Must handle inertial loads and power transfer (including potential for growth to 23 kW total continuous power).

i. Attitude transfer system. An attitude transfer or measurement system will be provided on the co-orbiting and polar Platforms. This system will be capable of measuring the instantaneous alignment between the GN&C navigation base and the payload mounting surfaces. Relative alignments shall be determined within 5 arc sec. Utilization of the attitude transfer system outputs in the guidance and control algorithms will be a control option, enabling overall Platform control to be driven by a single instrument.

5.3.7.6 Weight and Power Table

Table 5.3.7 shows GN&C weights and power requirements.

5.3.8 Propulsion

The propulsion subsystem provides translation for orbit transfer and orbit adjust as well as backup control torque capability.

5.3.8.1 Characteristics

The propulsion subsystem will provide the following performance capability for Platforms:

a. Transfer from the baseline orbit of the Space Station or the STS Orbiter to the Platform operational orbit.

TABLE 5.3.7 - REFERENCE PLATFORM GN&C WEIGHT AND POWER ESTIMATES

	QTY	WEIGHT (LB)	POWER (W)
Reaction Wheels (202 ft-lb-sec, Space Telescope)	4	421	220/1600 (260 avg)
Magnetic Torque Rods (39,274A-ft^2, Space Telescope)	4	434	28/100 (70 avg)
Star Tracker Triad	2	40	46
Sun Sensor (two axis)	2	13	4
Magnetometers	2	12	6
Hexad Strapdown Gryo	1	50	125
GN&C Processors	2	40	130
Attitude Transfer System	1	20	25
Processor BIU	2	10	50
Solar Panel Drive & Electronics	2	375	70
GN&C TOTALS:		1,415	786

b. Maintenance of the Platform operational orbit for a period of not less than 24 consecutive months plus 12 months reserve in any set of calendar years.

c. Return from the operational orbit for rendezvous in the proximity of the STS Orbiter or the Space Station.

d. Provide torque capability about three orthogonal body axes for backup reaction control for the GN&C subsystem.

5.3.8.2 Functions

The propulsion subsystem provides the following functions:

a. Translational forces and three axis body torques from multiple, appropriately placed thrusters.

b. Propellant transfer to the thrusters through a system of tanks, fluid lines, and valves.

c. Capability for resupply fluid transfer from the Space Station, the STS Orbiter or, if required, the OMV.

5.3.8.3 Potential Hardware Implementation

Platform propulsion requirements can be met with state-of-the-art technology systems. Earth storable monopropellant or bipropellant hot gas systems are applicable to the basic requirements. In addition, a cold gas system may be required to provide a low temperature, non-contaminating exhaust gas environment for Space Station or STS Orbiter rendezvous.

Propellant quantity requirements are sufficiently high that a multitank supply system will be required. Multiple propellant supply tanks offer the possibility of use of an existing design or tooling, and adaptibility to Platform configuration volume limitations. In addition, there is the possibility of reduced propellant slosh influence on the Platform and the possibility of some control of the propellant center of mass.

5.3.8.4 Subsystem Performance

The propulsion system selected in the reference configuration uses the hydrazine components used in the manned Space Station. It uses eight, 40-inch-diameter spherical tanks and will have a capacity of 6300 pounds of hydrazine. These tanks contain an elastomeric diaphragm to separate the nitrogen pressurant gas from the hydrazine. The blowdown ratio is 3 to 1. Tanks will be insulated and heated as required to prevent freezing. Propulsion system requirements are based on providing for the necessary ΔV for the worst case polar platform (Polar B) but can be offloaded for less energetic missions. The subsystem must be capable of raising any polar Platform which can be launched from WTR by the STS from the STS parking orbit altitude of 148 nmi to the Platform's operating altitude of 380 nmi. It then will do stationkeeping for a period of up to 3 years and return the Platform to Shuttle altitude for refueling and servicing. With the projected STS capability of approximately 30,300 lbs, the maximum dry weight for the polar Platform and payload may be as large as 24,000 lbs.

The thrusters are located at least 50 feet from the payload to minimize payload contamination and are oriented along the velocity vector when the Platform is in the nadir-oriented mode to minimize reorientation operations for drag make-up and orbit transfer burns. The Platform center of gravity, propellant center of gravity , and thrust vector are aligned to minimize the effect of center of gravity shift due to propellant usage. Additionally, the propulsion system is modular for ease of servicing and design flexibility.

5.3.8.5 Thruster Clusters

Four triple redundant thruster clusters developed for the Space Station are used. The 25- to 75-pound size of the longitudinally oriented thrusters are appropriate for Platform propulsion. Smaller thrusters for attitude control may be required for the Platforms. (The Space Station reference configuration uses the same size thruster in all orientations). The thruster clusters will be insulated and heated to prevent hydrazine freezing.

5.3.8.6 Hydrazine Lines

Redundant hydrazine lines will connect the thruster clusters to the propellant supply tanks. This line will either deploy with the deployable truss or will be installed by an astronaut. In either case, the lines will be dry at launch and will not contain fuel until the Platform is at a safe distance from the Shuttle or Space Station. The lines will be insulated and heated as necessary to prevent freezing.

5.3.8.7 Redundancy

All systems are redundant to the point that no failure in a thruster or pair of thrusters will result in the failure of a function.

5.3.8.8 Cold Gas System

The cold gas system size provides for 4.9 ft/sec translational velocity capability or 25,515 ft-lb-sec control momentum capability. The system is assumed to be required only for attitude hold in the final berthing operations. The cold gas system can provide a minimum of 4 days attitude hold mode capability.

5.3.8.9 Weight and Power Tables

Table 5.3.8-1 shows overall propulsion subsystem weights. The weight estimate includes fluids, tankage, fluid distribution system, thrusters, thermal conditioning, instrumentation, and integrating structure.

TABLE 5.3.8-1 - PLATFORM PROPULSION REFERENCE DESIGN WEIGHT

	WEIGHT - LB
N_2H_4 System	
Fully Loaded System	7,500
Maximum Expendable Propellant	6,300
Inert Weight	1,200
GN_2 System	
Fully Loaded System	300
Maximum Expendable Propellant	64
Inert Weight	236

Table 5.3.8-2 shows the power estimate for propulsion maintenance and operations.

TABLE 5.3.8-2 - PLATFORM PROPULSION REFERENCE DESIGN POWER

	POWER - W
N_2H_4	
Maintenance Orbital Average	175
Firing Maximum	760
GN_2 System	
Maintenance Orbital Average	Negligible
Firing Maximum	300

5.4 SAFETY

5.4.1 Introduction

The safety considerations involved in the design and operations of a low-inclination Platform and a polar Platform are approximately the same. In addition, the low-inclination Platform also involves proximity operations with the Space Station, while the polar Platform involves similar operations with the STS Orbiter. Overall safety considerations also include interfaces and operations involved with EVA in conjunction with Platform(s).

5.4.2 Protection/Control of High-Pressure or Hazardous Fluid Tanks

The design of pressure vessels or tanks containing hazardous fluids will comply with the standard design practice of leak-before-rupture implemented through a fracture control program. In addition, tanks/pressure vessels will either be protected in such manner to prevent propagation of failure to other tanks or located away from critical systems/subsystems or docking/berthing areas for the Orbiter and/or Space Station.

5.4.3 Propulsion Systems

Platform propulsion systems will be designed to similar safety considerations as those for both the Orbiter and the Space Station. In addition, the proximity operations of Platforms in the vicinity of the Space Station and during EVA's will be the subject of additional safety studies and analyses.

5.4.4 Proximity Operations

Safety considerations with regard to the operation of Platforms in the proximity of the Space Station will be the subject of additional safety studies. Such studies will develop safety design and operational requirements for the Platforms with regard to proximity operations.

5.4.5 EVA Operations

EVA activities involved in the operations and servicing of the Platform(s) will be the subject of additional safety studies and analyses.

5.4.6 Space Station Berthing Capabilities

Sufficient berthing capability is provided by the Platform(s) and are adequately accommodated by the berthing port arrangement of the reference configuration.

5.4.7 Other Safety Considerations

Other areas of safety concern will be the subject of additional safety studies and analyses. These areas include but are not restricted to: (1) servicing operations involving the low-inclination Platform at or near the Space Station and the polar Platform in conjunction with the Orbiter, and (2) assembly and repair operations associated with the Platform(s).

6.0 ACKNOWLEDGEMENTS.

The Reference Configuration Group (RCG) of the "Skunk Works" SE&I team was comprised of technical specialists from various NASA Centers. The following list acknowledges the individuals with primary responsibility for various report sections and contributing personnel. Al Louviere served as head of the SE&I team with Bass Redd serving as his deputy. Mark Craig led the Reference Configuration Group.

REPORT SECTION	LEAD	CONTRIBUTING PERSONNEL
1.0 Introduction	S. Nassiff (JSC)	
2.0 Background	S. Nassiff (JSC)	
3.0 Summary and Design Reqm'ts	J. Butler (MSFC) D. Greenshields (JSC)	D. Kaplan (JPL) W. Schneider (JSC) B. Holder (JSC) G. Robinson (MSFC) J. Lewis (JSC) P. Nicaise (MSFC) J. Dunning (LeRC) C. Gott (JSC) T. Taylor (JSC) L. Livingston (JSC) D. Tomlin (MSFC)
4.0 Space Station Manned Core	J. Butler (MSFC) D. Greenshields (JSC)	Same as Section 3.0
4.1 Design Reqm'ts	D. Greenshields (JSC)	Same as Section 3.0
4.2 Configuration Description	L. Livingston (JSC)	J. Butler (MSFC) D. Greenshields (JSC)
4.2.1 General Arrangement		O. Meredith (MSFC) S. Avans (MSFC) B. Ondrak (MSFC) S. Croomes (MSFC)
4.2.2 Flight Modes		J. Stokes (MSFC) D. Deryder (LaRC) B. Garrett (LaRC) E. Kempke (LeRC) J. Dunning (LeRC) M. Mikulas (LaRC) M. Dalton (JSC) T. Sasseen (KSC) K. Hinson (JSC) L. Purves (MSFC)

REPORT SECTION	LEAD	CONTRIBUTING PERSONNEL
4.2.3 Elements	D. Greenshields (JSC)	Same as Section 4.2
4.2.4 Mass Properties	D. Howes (JSC) B. Ondrak (MSFC)	Same as Section 4.2
4.3 Configuration Evaluation	D. Greenshields (JSC) J. Butler (MSFC)	Same as Section 3.0
4.3.1 Payload Accommodation	D. Kaplan (JPL)	B. Ondrak (MSFC) J. Butler (MSFC) R. Duncan (MSFC) K. Morrison (JSC) C. Robinson (JSC) D. Suddeth (GSFC) J. Sos (GSFC)
4.3.2 Crew Accommodations	J. Lewis (JSC)	J. Stokes (MSFC) G. Robinson (MSFC) J. Hall (LaRC) C. Robinson (JSC) M. Dalton (JSC) A. Benrend (JSC) M. Myers (JSC) J. Queller (Hqs)
4.3.3 Dynamics & Control	B. Holder (JSC)	C. Moore (MSFC) D. Tomlin (MSFC) P. Nicaise (MSFC) J. Schliesin[illegible] (JSC) B [illegible]eryder [illegible]C) D. Deryder (LaRC) L. Perez (JSC) J. Farmer (MSFC) K. Grady (GSFC) M. Lazaron (JSC) J. Sunkel (JSC) J. Housner (LaRC)
4.3.4 Power Capability & Utilization	L. Crabtree (MSFC) D. Howes (JSC)	J. Graves (MSFC) J. Dunning (LeRC) E. Kempke (LeRC) R. Rice (JSC) W. Simon (JSC) [illegible]. Gott (JSC)
4.3.5 Thermal Capability	T. Taylor (JSC)	D. Seymour (MSFC) G. Robinson (MSFC) J. Hall (LaRC) W. Morris (JSC) A. Fitzkee (GSFC)

REPORT SECTION	LEAD	CONTRIBUTING PERSONNEL
4.3.6 Comm. & Tracking	S. Lenett (JSC)	J. Clubb (MSFC) E. Bracalente (LaRC) M. Kapell (JSC) D. Rhoades (JSC) R. Knight (LeRC) B. Lercy (LeRC)
4.3.7 Launch, Assembly, and Growth	W. Schneider (JSC) O. Meredith (MSFC)	B. Ramage (MSFC) A. Crumbley (MSFC) S. Croomes (MSFC) C. Gott (JSC) R. Cain (JSC) M. Mikulas (LaRC)
4.3.8 Operations	C. Gott (JSC)	B. Ramage (MSFC) A. Crumbley (MSFC) K. Morrison (JSC) S. Croomes (MSFC) R. Becker (JSC) T. White (JSC) T. Saseen (KSC) C. Anderson (JSC) R. Kain (JSC) D. Homan (JSC) L. Hudson (JSC) A. Dupont (JSC) B. Nader (JSC) E. Mitchell (JSC)
4.3.9 Safety	H. Williams (JSC)	D. Baillie (JSC) S. Truelock (JSC) D. Berlad (JSC)
4.4.1 Electrical Power	L. Crabtree (MSFC)	J. Graves (MSFC) R. Rice (JSC) W. Simon (JSC) J. Dunning (LeRC) E. Kempke (LeRC) W. Chandler (JSC) D. Howes (JSC)
4.4.2 Guidance, Navigation, & Control	P. Nicaise (MSFC)	D. Tomlin (MSFC) J. Farmer (MSFC) C. Woolley (LaRC) T. Barry (JSC) K. Grady (GSFC) C. Thomas (JSC)

REPORT SECTION	LEAD	CONTRIBUTING PERSONNEL
4.4.3 Comm. & Tracking	S. Lenett (JSC)	J. Clubb (MSFC) E. Bracalente (LaRC) M. Kapell (JSC) D. Rhoades (JSC) R. Knight (LeRC) B. Leroy (LeRC)
4.4.4 Information and Data Management	M. Holley (JSC)	A. Newcomb (JSC) S. Anderson (JSC) J. Clubb (MSFC) T. Barry (JSC) A. Dula (KSC) C. Herman (JSC)
4.4.5 Propulsion	D. Westrope (MSFC)	C. Aukerman (LeRC) R. Donovan (LeRC) D. Kendrick (JSC) D. Anderson (KSC)
4.4.6 Environmental Control and Life Support Sys.	A. Behrend (JSC)	C. Robinson (JSC) K. Upshaw (MSFC) D. Seymour (MSFC) M. Myers (JSC) D. Deryder (LaRC) J. Hall (LaRC) G. Robinson (MSFC)
4.4.7 Thermal	W. Morris (JSC)	C. Robinson (JSC) K. Upshaw (MSFC) D. Seymour (MSFC) R. Parrish (JSC) A. Fitzkee (GSFC) T. Taylor (JSC) J. Hall (LaRC) G. Robinson (MSFC)
4.4.8 Structures and Mechanisms	W. Schneider (JSC) K. Hinson (JSC)	S. Avans (MSFC) S. Croomes (MSFC) M. Mikulas (LaRC) J. Housner (LaRC) L. Purves (GSFC) K. Hinson (JSC) C. Wesselski (JSC)
4.4.9 EVA Systems	D. Kissinger (JSC)	J. Lewis (JSC) J. Queller (Hqs) G. Robinson (MSFC) J. Stokes (MSFC)

REPORT SECTION	LEAD	CONTRIBUTING PERSONNEL
5.0 Unmanned Platforms	D. Suddeth (GSFC) R. Pompfrey (JPL)	R. Muller (GSFC) J. Graf (JPL)
5.1 Design Reqm'ts	S. Bergeson-Willis (GSFC)	R. Muller (GSFC) J. Graf (JPL) D. Vane (JPL)
5.2 Configuration Description	R. Muller (GSFC) J. Sunkel (JSC) B. Holder (JSC) T. Premack (GSFC)	J. Young (GSFC) J. Bell (JSC) L. Livingston (JSC) J. Schliesing (JSC) R. Berka (JSC)
5.3 Subsystems	D. Greenshields (JSC)	J. Young (GSFC) A. Fitzkee (GSFC) T. Taylor (JSC) S. Lenett (JSC) M. Kapell (JSC) D. Rhoades (JSC) E. Bracalente (LaRC) K. Grady (GSFC) D. Suddeth (GSFC) D. Kendrick (JSC) R. Klementson (JPL) J. Rakewicz (JPL) C. Gilchriest (JPL) C. Hermann (JSC) A. Dula (KSC) S. Anderson (JSC) R. Hartenstrin (GSFC) C. Thomas (JSC) C. Woolley (LaRC) D. Weary (JSC) C. Bell (JPL) M. Mangano (JPL) R. Rowley (JPL) R. Miyake (JPL) W. Wuest (JPL) M. Holley (JSC) T. Berry (JSC) D. Tomlin (MSFC)
5.4 Safety	H. Williams (JSC)	S. Truelock (JSC)

REFERENCES

1. JSC-19521, Conceptional Design and Evaluation of selected Space Station Concepts, December 1983.

2. JSC-19371, Space Station Vol. 4-preliminary power profiles for IOC Configuration-MPAD, February 1984.

3. Leondis, Alex: "Large Advanced Space System Computer Aided Design and Analysis Program." NASA CR 159191-1, July 1980.

4. Space Station Preliminary Design Report, NASA JSC Report 18555, September 1982.

5. Jacchia, L. G.: Thermospheric Temperature, Density, and Composition: New Models. SAO Special Report No. 375, March 15, 1977.

6. Hedin, A. E.: A Revised Thermospheric Model Based on Mass Spectrometer and Incoherent Scatter Data. Journal of Geophysics Research, Vol. 88, December 1983.

7. JSC 07700, Shuttle Payload Accommodations Handbook, Vol. 14, Attachment 1.

8. McCormick, Caleb; editor: MSC NASTRAN Manual.

9. Mikulas, Martin, et al: Space Station Truss Structures and Construction Considerations. Space Station Project Office White Paper, August 1984.

10. Noor, A. K.; Anderson, M. S.; Greene, W. H.: Continuous Models for Beam and Platelike Lattice Structures. AIAA Journal, Vol. 16. No. 12, December 1978.

11. Noor, A. K.; Anderson, C.M.: Analysis of Beam-Like Lattice Trusses. Computer Methods in Applied Mechanics and Engineering, Vol. 20, 1979.

12. Leondis, Alex: Large Advanced Space Systems Computer-Aided Design and Analysis Program. NASA CR-159191, 1980.

13. Garrett, L. Bernard: "Interactive Design and Analysis of future Large Spacecraft Concepts." NASA TP-1937, 1981.

14. Heck, Michael L.: Rigid body Control Dynamics: Test Case Results and Documentation - AMA 4-9 Report No. 83-8, Analytical Mechanics Associates, Inc., April 1983.

15. Heck, Michael L.: Final Report Rigid Body Control Dynamics Part II. AMA Report No. 83-22, Analytical Mechanics Associates, Inc., April 1983.

16. Garrett, L. B. and Ferebee, M. J., Jr.: "Comparative Analysis of

Large Antenna Spacecraft Using the IDEAS System." AIAA Paper 83-0798-CP May 2-4, 1983.

17. Dorsey, J. T.; and Bush, H. G.: Dynamic Characteristics of a Space Station Solar Wing Array. NASA TM-85780, 1984.

18. Load Specification - SRMS Manipulator Arm. SPAR-SG. 409 SPAR Aerospace, Toronto, Ontario, Canada, September 1978.

19. Mikulas, Martin, M., Jr.; Bush, Harold G.; Wallsom, Richard E.; Dorsey, John T.; and Rhodes, Marvin D.: A Manned-Machine Space Station Construction Concept, NASA TM 85762, February 1984.

20. Heard, Walter L.,Jr.; Bush, Harold G.; Wallsom, Richard E.; and Jensen, J. Kermit: A Mobile Work Station Concept for Mechanically Aid Astronaut Assembly of Large Space Trusses. NASA TP 2108, March 1983.

21. Large Space Systems Technology - 1979 Technology Review, Hampton, Virginia. NASA CP 2118, November 1979.

22. Large Space Antenna Systems Technology - 1982, Parts 1 and 2. NASA CP 2269, November 30-December 3, 1982, pp. 257-283.

23. Greenberg, H. S.: "Development of Deployable Structures for Large Space Platforms". NASA CR-170689, December 1982.

24. Cox, R. L.; Nelson, R. A.: "Development of Deployable Structures for large Space Platforms." NASA CR-170690, December 1982.

25. Heard, Walter L., Jr.; Bush, Harold G.; Wallsom, Richard E.; and Jenson, J. Kermit: "A Mobile Work Station Concept for Mechanically Aided Astronaut Assembly of Large Space Trusses." NASA TP 2108 March 1983.

26. Mikulas, Martin M., Jr. and Bush, Harold G.: "Advances in Structural Concepts." Large Space Antenna Systems Technology-1982, Part 1, NASA CP 2269, November 30-December 3, 1982, pp. 257-283.

27. Dorsey, John T.: "Structural Performance of Orthogonal Tetrahedral Truss Space Station Configurations." NASA TM 86260, July 1984.

28. Bush, Harold G.; Mikulas, Martin M., Jr.; Wallsom, Richard E.; and Jensen, J. Kermit: "Conceptual Design of a Mobile Remote Manipulator System." NASA TM 86262, July 1984.

29. Mikulas, Martin M., Jr.; Bush, Harold G.; Wallsom, Richard E.; Dorsey, John T.; and Rhodes, Marvin D.: "A Manned-Machine Space Station Construction Concept," NASA TM 85762, February 1984.

30. Mikulas, Martin M., Jr.; Bush, Harold, G.; and Card, Michael F.: "Structural Stiffness, Strength and Dynamic Characteristics of Large Tetrahedral Space Truss Structures," NASA TM X-74001, March 1977.

31. Armstrong, W. H.; Skoumal, D. E.; and Straayer, J. W.: "Large Space Erectable Structures," Contract NAS9-14914, April 1977.

32. Heard, W. L., Jr.; Bush, H. G. ; Walz, J. E.; and Rehder, J. J.: Structural sizing Considerations for Large Space Platforms. Journal of Spacecraft and Rockets, Vol. 18, No. 6, pp. 556-564, November-December 1981.

33. Bluck, R. M.; Johnson, R. R.: "Fabrication of Slender Struts for Deployable Antennas." NASA CR-172 164, April, 1983.

34. Kennel, NASA TM-82390, Steering Law for Parallel Mounted Double Gimbaled GMG's--Revision A.

SPACE STATION SUBSYSTEM WHITE PAPERS (JSC-20054)

- Space Station Truss Structures and Construction Considerations
- Solar Panel Deployment Mechanisms
- Module and Payload Truss Attachment
- Module to Module Interface, Docking/Berthing
- Preliminary Structural Design and Analysis of a Shuttle Launched Space Station Manned Habitable Module
- Concept for a Space Station Solar Array Rotary Joint
- Deployment Verification
- Conceptual Design of a Mobile Remote Manipulator System
- Implementing Space Station Autonomy/Automation
- Space Station Large Area Magnetic Torquer
- Space Station Fluidic Momentum Controller
- Solar Dynamic Energy Conversion Systems for a Manned Space Station
- A Power Management and Distribution System for a Manned Space Station
- Module Pattern

APPENDIX A

ASSEMBLY OPERATIONS CREW TIMELINES EVALUATION

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY*	COMMENTS
FIRST FLIGHT				See Assumption General
o Install Radiators 12 (50'x1'x1")	260	EVA-MFR	3	See Assumption-B-1
o EV2 ingress MFR & maneuver to position for helping to install radiators on 38' structure for installing radiators	(15)	EVA-MFR	1	
o EV1/EV2 release radiator panel and move it to insertion location, then insert radiator into heat exchanger boom, EV1 tighten pressure mechanism (6 radiators)	(120)	EVA-MFR	3	Difficult task due to size of radiators and installation tolerances
o EV1 move to other insertion location	(5)	EVA	1	
o Same as two steps above (6 radiators)	(120)	EVA-MFR	3	
o Grapple, lift 38' package from cargo bay and rotate 90°	15	RMS	1	
o Deploy solar array blanket boxes (4)	5	AUTO	1	See Assumption-B-2
o Deploy rail extensions (2 on each end)	5	AUTO	1	See Assumption-B-3
o Deploy transverse boom (14 bays)	82	AUTO-MFR	1	
o EV2/MFR in position to verify correct boom deployment (4 different locations)	(10)	MFR	1	
o Deploy boom (14 bays)	72	AUTO	1	See Assumption-B-[illegible]

* 1 - Proven, 2 - Ea[illegible], 3 - Hard, 4 - Questionable, 5 - Not Possible

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
FIRST FLIGHT (continued)				
o Install ATV/manipulator on boom	100	EVA-RMS	4	See Assumption-B-5
o RMS grapple ATV	(5)	RMS	1	
o Release launch latches	(5)	EVA	1	
o Lift ATV from cargo bay and place over boom	(10)	RMS	1	
o EVA crewmembers translate to ATV site and attach ATV to boom	(20)	EVA	4	Task questionable due to lack of data as to how the ATV will attach to the boom
o RMS grapple manipulator	(5)	RMS	1	
o Release launch restraint on base base of manipulator	(10)	EVA	1	
o Release manipulator MPM's and maneuver manipulator to ATV for installation	(10)	RMS-AUTO	2	
o Install manipulator on ATV	(30)	RMS-EVA	3	
o Release manipulator and remove portable grapple fixture	(5)	RMS-EVA	1	
o Checkout MRMS	30	EVA	1	From here out, the ATV/ Manipulator will be referred to as MRMS
o Deploy solar array blankets (4)	60	AUTO	1	See Assumption-B-6
o Erect one bay and attach docking device	40	EVA	2	See Assumption-B-7

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
FIRST FLIGHT (continued)				
o Plug MRMS into recharge panel	5	EVA	1	
o Checkout systems				No data listed due to lack of information
o Release 38' structure				Release retention latches and Orbiter backs away
o Stow RMS	10	RMS	1	
TOTAL	617 MIN 10.28 HR			

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
SECOND FLIGHT				
o Dock with structure				See Assumptions-General, C-1
o Checkout MRMS	30	EVA	1	
o Install lower keel	60	EVA-RMS	3	
o Grapple lower keel	(5)	RMS	1	
o Release lower keel from holding fixture	(5)	EVA	1	
o Maneuver lower keel to installation location	(15)	RMS	2	
o EVA crewmembers get in position for attaching lower keel	(15)	EVA	1	
o Attach lower keel to structure	(20)	EVA-RMS	3	Difficult location for attaching large package. No data on attaching hardware.
o Connect Utilities	10	EVA	1	See Assumption-C-2
o Deploy rail extensions	15	AUTO	1	
o Deploy lower keel	92	AUTO	3	See Assumption-C-3 (18 cells)
o Deploy radiator booms (2)	5	AUTO	1	
o Deploy radiator heat exchanger boom (4)	5	AUTO	1	
o Erect bays (4) next to radiator booms	52	EVA	1	

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
SECOND FLIGHT (continued)				
o Install/stow radiators, 48 (50'x1'x1")	245	MRMS-RMS-EVA	4	
o Grapple radiator package	(5)	RMS	1	
o.Release radiator launch restraint	(5)	EVA	1	
o Maneuver radiator package to MRMS and attach	(30)	EVA-RMS	2	
o MRMS translate to position for radiator stowage	(20)	MRMS	1	
o Remove radiators (42) from MRMS and stow on sides of lower keel	(50)	EVA	2	See Assumption-C-4
o MRMS translate to position for radiator 6) installation	(5)	MRMS	1	
o EV2 ingress MRMS MFR	(5)	EVA	1	
o EV2/MFR retrieve a radiator from the bundle, EV1 maneuvers EV2/MFR to radiator installation position. EV2 installs radiator & tightens it down. (20 min/rad. x 3 rad.)	(60)	MRMS/MFR	4	Questionable task due to size of radiators and tolerances. Also one person installation because one is required to operate the MRMS.
o Move MRMS to other radiator site	(5)	MRMS	1	
o Same as two steps above (3 radiators)	(60)	MRMS/MFR	4	
o Install keel extensions (2)	219	MRMS-EVA-AUTO	2	

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
SECOND FLIGHT (continued)				
o MRMS translate to position for keel extensions (2) installation	(5)	MRMS	1	
o Manipulator grapple keel extension with docking/berthing ring on it (keel extensions located on end of lower keel)	(5)	MRMS	1	See Assumption-C-5
o Release first keel extension from second keel extension	(10)	EVA	1	
o Maneuver keel extension to final installation location	(5)	MRMS	1	
o Install keel extension	([illegible]0)	EVA	2	
o Deploy keel extension	(67)	AUTO	1	
o Repeat above 5 steps for second keel extension	(107)	MRMS-EVA-AUTO	2	
o MRMS translate back to cargo bay area	25	MRMS	1	
o Install lower boom and ancillary structure	115	[illegible]		
o Grapple lower boom pallet	(5)	RMS	1	See Assumption-C-6
o Release pallet launch restraint	(10)	EVA	1	
o Maneuver pallet to MRMS and latch down	(20(	RMS-EVA	1	

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
SECOND FLIGHT (continued)				
o MRMS translate to keel extension	(25)	MRMS	1	
o Erect a bay between the end of the lower keel and each keel extension	(20	EVA	1	
o MRMS translate down keel extension	(5)	MRMS	1	
o Erect three bays between the keel extensions	(30)	EVA	1	
o Install MRMS recharge panel on lower keel	15	EVA	2	
o Plug MRMS into recharge panel	5	EVA	1	
o EV crewmembers translate back to Orbiter	15	EVA	1	
TOTAL	911 MIN 15.2 HR			

9

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
THIRD FLIGHT				
o Dock with structure				See Assumption-D-1
o Retrieve MRMS	35	MRMS-EVA	1	
o Translate to MRMS	(10)	EVA	1	
o Activate MRMS	(15)	EVA	1	
o MRMS translate to lower end of lower keel	(10)	MRMS	1	
o Install module mounting structure	40	MRMS-EVA	3	
o MRMS grapple mounting structure,	(15)	MRMS-EVA	1	
o MRMS MANEUVER MOUNTING STRUCTURE	(15)	MRMS	1	
o Attach mounting structure	(10)	EVA	3	No data on attachment hardware
o Install HAB 1 Module	70	ATV-EVA	3	
o Translate MRMS and grapple HAB 1 module	(10)	MRMS	1	
o Lift HAB 1 module out of cargo bay and position it at its attachment location	(30)	MRMS	1	
o Attach HAB 1 module to mounting structure	(30)	EVA	3	No data on attachment hardware
o Connect utilities	50	EVA	1	

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
THIRD FLIGHT (continued)				
o Retrieve pigtails from keel extension	(15)	EVA	1	
o Connect connectors to two side panels	(20)	EVA	1	
o Connect connectors to end of module	(15)	EVA	1	
o Install Airlock 1	20	MRMS	1	
o Grapple Airlock 1, release	(10)	MRMS	1	
o MRMS maneuver Airlock 1 to HAB 1 berthing location	(5)	MRMS	1	
o Berth Airlock 1 to HAB 1	(5)	MRMS	1	
o Temp stow Airlock 2	20	MRMS	1	See Assumption-D-2
o Grapple Airlock 2, release launch restraints	(10)	MRMS	1	
o MRMS maneuver Airlock 2 to HAB 1(5) berthing location	MRMS	1		
o Berth Airlock 2 to HAB 1	(5)	MRMS	1	
o MRMS translate to recharge panel	10	MRMS	1	
o Plug MRMS into recharge panel	5	EVA	1	
o EV crewmembers translate back to Orbiter	5	EVA	1	

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
THIRD FLIGHT (continued)				
o Checkout HAB 1				No data listed due to lack of information
TOTAL	255 MIN 4.25 HR			

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
FOURTH FLIGHT				
o Dock with HAB 1 module				
o MRMS activation	15	EVA	1	
o Install module mounting structure	59	MRMS-RMS-EVA	3	See Assumption-E-1
o Grapple mounting structure, release launch restraint	(15)	RMS-EVA	1	
o Lift mounting structure out of cargo bay	(10)	RMS	1	
o MRMS grapple mounting structure and RMS release mounting structure	(5)	MRMS-RMS	1	
o MRMS maneuver mounting structure to attachment location	(10)	MRMS	1	
o Attach mounting structure	(10)	EVA	3	No data on attaching hardware
o Berth/attach HAB 2 module to HAB 1 module	75	MRMS-EVA	3	
o MRMS grapple HAB 2 module	(5)	MRMS	1	
o MRMS maneuver HAB 2 module and berth it to the HAB 1 module	(40)	MRMS	1	
o Attach HAB 2 module to mounting structure	(30)	EVA	3	No data on attaching h dware

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
FOURTH FLIGHT (continued)				
o Connect side panel utilities	25	EVA	1	
o Remove Airlock 2 from HAB 1 and berth it to HAB 2	20	MRMS	1	
o Install upper keel	130	MRMS-RMS- EVA-AUTO	3 1	
o Grapple upper keel, release launch restraints and lift out of cargo bay	(15)	RMS-EVA	1	
o MRMS grapple upper keel and RMS release upper keel	(5)	MRMS-RMS	1	
o MRMS translate to upper keel installation location	(30)	1		
o Unfold deployment rails	(5)	EVA	1	
o Attach upper keel to transverse boom	(30)	MRMS-EVA	3	No data on attaching hardware
o Deploy upper keel and upper boom	(110)	AUTO	3	See Assumption-E-2
o MRMS translate to recharge location, plug MRMS in for recharge	35	MRMS-EVA	1	
o EV crewmembers translate back to	5	EVA	1	
o Checkout HAB 2				No data listed due to lack of information
TOTAL	355 MIN 5.9 HR			

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
FIFTH FLIGHT				
o Dock with HAB 1 module				
o MRMS activation	15	EVA	1	
o Berth logistics module to HAB 2 module	60	MRMS-RMS	2	
o RMS grapple logistics module	(5)	RMS	1	
o Lift LOG module out of cargo bay	(10)	RMS	1	
o MRMS grapple LOG module and RMS release LOG module	(5)	MRMS-RMS	1	
o MRMS maneuver LOG module and berth it to the HAB 2 module	(25)	MRMS	2	
o Connect external thermal connector	(15)	EVA	1	
o Install port solar array addition pkg.	172	MRMS-RMS-EVA-AUTO	3	
o Grapple port solar array package and release launch restraints	(5)	RMS	[illegible]	
o Lift array package out of cargo bay	(10)	RMS	1	
o MRMS grapple array package and RMS release array package	(5)	MRMS-RMS	1	
o MRMS translate to port end of transverse boom	(40)	MRMS	1	
o Unfold deployment rails	(5)	EVA	1	

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
FIFTH FLIGHT (continued)				
o Attach port solar array package to end of transverse boom and mate connectors	(35)	MRMS-EVA	3	No data on attaching hardware
o Deploy port outboard transverse boom structure	(32)	AUTO	3	See Assumption-F-1
o MRMS translate back to cargo bay area	(40)	MRMS	1	
o Install starboard solar array addition package	(132	MRMS-RMS-EVA-AUTO		
o Grapple starboard solar array addition package and release launch restraints	(5)	RMS	1	
o Lift array package out of cargo bay	(10)	RMS	1	
o MRMS grapple array package and RMS release array package	(5)	MRMS-RMS	1	
o MRMS translate to starboard end of transverse boom	(40)	MRMS	1	
o Unfold deployment rails	(5)	EVA	1	
o Attach starboard solar array package to end of transverse boom and mate connectors	(35)	MRMS-EVA	3	No data on attaching hardware
o Deploy starboard outboard transverse boom structure	(32)	AUTO	3	See Assumption-F-1

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
FIFTH FLIGHT (continued)				
o Deploy outboard solar arrays (4)	60	AUTO	1	
o MRMS translate to recharge location, plug MRMS in for recharge	45	MRMS-EVA	1	
o EV crewmembers translate back to Orbiter	5	EVA	1	
o Module activation for inhabitance				
<u>TOTAL</u>	489 MIN 8.15 HR			

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
SIXTH FLIGHT				
o Dock with HAB 1 module				
o MRMS activation	15	EVA	1	
o Install module mounting structure	50	ATV-RMS-EVA	3	See Assumption-G-1
o Grapple mounting structure, release launch restraint	(15)	RMS-EVA	1	
o Lift mounting structure out of cargo bay	(10)	RMS	1	
o MRMS grapple mounting structure & RMS release mounting structure	(5)	MRMS-RMS	1	
o MRMS maneuver mounting structure to attachment location	(10)	MRMS	1	
o Attach mounting structure	(10)	EVA	3	No data on attaching hardware
o Berth/attach LAB 2 module	75	MRMS-RMS-EVA	3	
o Grapple LAB 2 module, release launch restraint	(5)	RMS	1	
o Lift LAB 2 out of cargo bay	(10)	RMS	1	
o MRMS grapple LAB 2 module and release LAB 2 module	(5)	MRMS-RMS	1	
o MRMS maneuver LAB 2 and berth it to the HAB 2 module	(25)	MRMS	3	Close tolerances if berth and attachment are both required

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
SIXTH FLIGHT				
o Dock with HAB 1 module				
o MRMS activation	15	EVA	1	
o Install module mounting structure	50	ATV-RMS-EVA	3	See Assumption-G-1
o Grapple mounting structure, release launch restraint	(15)	RMS-EVA	1	
o Lift mounting structure out of cargo bay	(10)	RMS	1	
o MRMS grapple mounting structure & RMS release mounting structure	(5)	MRMS-RMS	1	
o MRMS maneuver mounting structure to attachment location	(10)	MRMS	1	
o Attach mounting structure	(10)	EVA	3	No data on attaching hardware
o Berth/attach LAB 2 module	75	MRMS-RMS-EVA	3	
o Grapple LAB 2 module, release launch restraint	(5)	RMS	1	
o Lift LAB 2 out of cargo bay	(10)	RMS	1	
o MRMS grapple LAB 2 module and release LAB 2 module	(5)	MRMS-RMS	1	
o MRMS maneuver LAB 2 and berth it to the HAB 2 module	(25)	MRMS	3	Close tolerances if berth and attachment are both required

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
SIXTH FLIGHT (continued)				
o Attach LAB 2 module to mounting structure	(30)	EVA	3	No data on attaching hardware
o Connect side panel utilities	25	EVA	1	
o MRMS translate to recharge location, plug MRMS in for recharge	10	MRMS-EVA	1	
o EV crewmembers translate back to Orbiter	5	EVA	1	
TOTAL	180 MIN 3 HR			

ASSEMBLY TASK EVALUATION

FUNCTION/TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
SEVENTH FLIGHT				
o Dock with LAB 1 module				
o MRMS activation	15	EVA	1	
o Install module mounting structure	50	MRMS-RMS-EVA	3	See Assumption-H-1
o Grapple mounting structure, release launch restraint	(15)	RMS-EVA	1	
o Lift mounting structure out of cargo bay	(10)	RMS	1	
o MRMS grapple mounting structure & RMS release mounting structure	(5)	MRMS-RMS	1	
o MRMS maneuver mounting structure to attachment location	(10)	MRMS	1	
o Attach mounting structure	(10)	EVA	3	No data on attaching hardware
o Berth LAB 1 module	80	MRMS-RMS-EVA	3	
o Grapple LAB 1 module, release launch restraint	(5)	RMS	1	
o Lift LAB 1 out of cargo bay	(10)	RMS	1	
o MRMS grapple LAB 1 module and RMS release LAB 1 module	(5)	MRMS-RMS	1	
o MRMS maneuver LAB 1 & berth it to HAB 1 and LAB 2 modules	(30)	MRMS	3	Difficult task due to berthing to two different modules at same time

ASSEMBLY TASK EVALUATION

FUNCTION TASK	TIME (MIN)	METHOD	FEASIBILITY	COMMENTS
SEVENTH FLIGHT (continued)				
o Attach LAB 1 module to mounting structure	(30)	EVA	3	No data on attaching hardware
o Connect side panel utilities	25	EVA	1	
o MRMS translate to recharge location, plug MRMS in for recharge	10	MRMS-EVA	1	
o EV crewmembers translate back to Orbiter	5	EVA	1	
TOTAL	185 MIN 3.1 HR			

SPACE STATION ASSEMBLY ASSUMPTIONS

The following assumptions are made for assembling the Space Station concept.

A. GENERAL

1. Two EV crewmembers are located in the cargo bay, the RMS has been activated and checked out, and one MMU is available for use on each mission.

B. FLIGHT ONE - Basic Structure

1. (Install radiators)

a. Assume that the radiator heater exchanger booms are not covered by the launch stowage of the radiator panels.

b. Assume some type of crew-operated quick release handling aid for the solar array panels. It must leave free access to one radiator edge, in order to permit installation beside the previously installed radiator panel.

c. Assume approximately one inch clearance between installed radiator panels.

d. Assume radiator assembly bars are parallel to 38-foot package long axis.

e. Assume two equipment restraint types required, a launch restraint and separate panel temporary restraint.

2. (Deploy Solar Array blanket boxes)

a. Assume that deployment of solar arrays is automatic.

b. Assume parallel booms are deployed simultaneously.

3. (Deploy rail extensions)

a. Assume automatic extension of two rails at a time.

4. (Deploy booms)

a. Assume automatic boom deployment of 3 minutes/bay, plus a 2-minute inspection of eight joints/bay, plus a preload time of 2 minutes.

5. (Install ATV/Manipulator)

a. Assume ATV/manipulator has proven high reliability. This will represent new technology.

b. Assume a battery recharge/maintenance station for the ATV, located on the Space Station structure.

c. Assume mechanical and electrical connection guides provided for manipulator installation of the ATV.

6. (Deploy solar array blankets)

a. Assume arrays are deployed sequentially.

b. Assume crew involvement is limited to observation only.

7. (Erect bay/attach docking device)

a. Assume two-crewmember operation, with crew in foot restraints.

b. Assume struts and nodes are restrained at assembly worksite.

c. Assume that the berthing device is already attached to four assembly struts.

C. FLIGHT TWO - Additional Structure/Appendages

1. (Dock with structure)

a. Assume there is no plume impingement problem.

b. Assume Orbiter control system is disabled during assembly and control is maintained by the Space Station CMG's.

2. (Connect utilities)

a. Assume utilities are located inside the cells.

b. Assume crew will manually connect electrical, instrumentation, and plumbing lines to panels on side of structure out of MRMS path.

3. (Deploy lower keel)

a. Assumed a deployment rate of 3 minutes/bay, plus a 2-minute inspection of each bay, plus a preload time of 2 minutes for 18 bays.

b. Assume a feasibility level of 3 due to the state-of-the-art level and number of bays.

c. Assume EV crew participation limited to observation for nominal operations and assistance in contingency operations.

4. (Install/stow radiators)

a. Assume two-EVA-crew-attach radiator panel restraint bracketry on both sides of lower keel.

b. Assume these radiator panels will be installed later when the Space Station is inhabited.

c. Assume six installed radiators are sufficient at this point.

5. (Install keel extensions)

a. Assume that docking/berthing ring is already attached to the end of one of the two keel extensions.

b. Assume one crewmember operating MRMS and the other one doing the attachment.

c. Assume a deployment rate of 3 minutes/bay, plus a preload time of 2 minutes, plus 2 minute/bay inspection of eight joints/bay for 10 bays.

6. (Install lower boom and ancillary structure)

a. Assume the two lower boom packages and 48 erectable struts and associated connectors are attached to a pallet.

b. Assume a deployment rate of 3 minutes/bay, plus a preload time of 2 minutes, plus 2 minutes per bay inspection of eight joints/bay.

D. FLIGHT THREE - HAB 1 Module

1. Same assumption as assumption C-1 (Flight two)

2. (Temp stow airlock 2)

Airlock 2 will be stowed temporarily to HAB 1 until HAB 2 is in place, which is where it will normally be located.

E. FLIGHT FOUR - HAB 2 Module

1. (Install module mounting structure)

a. Assume that the package is pre-assembled and crew installed.

2. (Install upper keel)

a. Assume a deployment rate of 3 minutes/bay, plus a preload time of 2 minutes, plus 2 minutes per bay inspection of eight joints/bay.

b. Assume a feasibility level of 3 due to the state-of-the-art level and number of bays.

c. Assume EV crew participation limited to observation for nominal operations and assistance in contingency operations.

F. FLIGHT FIVE - Logistics Module

1. (Install port solar array addition package)

a. Assume a deployment rate of 3 minutes/bay, plus a preload time of 2 minutes, plus 2 minutes per bay inspection of eight joints/bay.

b. Assumed a feasibility level of 3 due to the state-of-the-art-level.

c. Assume EV crew participation limited to observation for nominal operations and assistance in contingency operations.

G. FLIGHT SIX - LAB 2 Module

1. (Install module mounting structure)

a. Assume that the package is pre-assembled and crew installed.

H. FLIGHT SEVEN - LAB 1 Module

1. (Install module mounting structure)

a. Assume that the package is pre-assembled and crew installed.

* U.S. GOVERNMENT PRINTING OFFICE 1984—789-013/888