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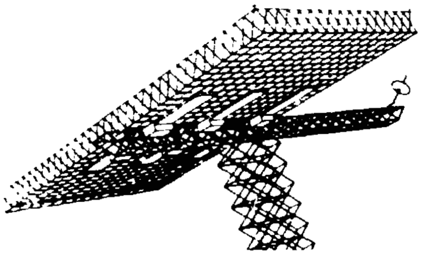
Conceptual Design and Evaluation of Selected Space Station Concepts

VOLUME 2

December 1983



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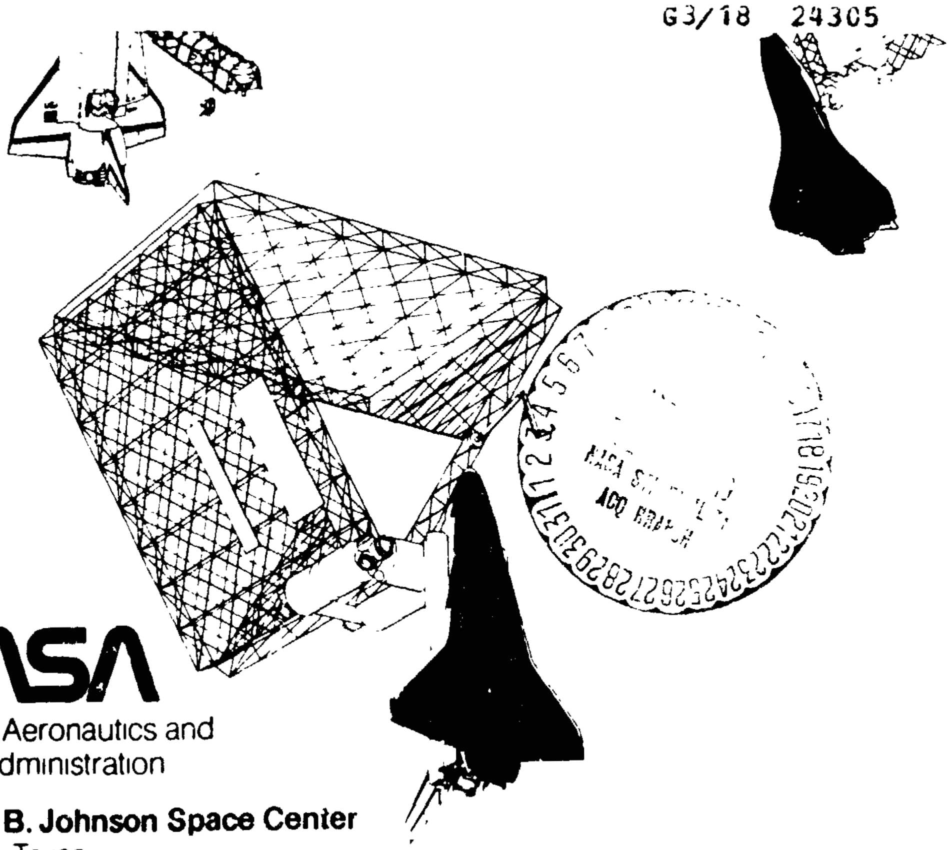

National Aeronautics and Space Administration

Lyndon B. Johnson Space Center
Houston, Texas

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LIST OF ACRONYMS AND ABBREVIATIONS

BB	Building Block
C/CM	Command/Control Module
CDG	Concept Development Group
CMG	Control Moment Gyro
DDT&E	Design, Develop, Test, and Evaluation
ESS	Energy Storage Subsystem
ET	External Tank
EVA	Extra Vehicular Activity
FF	Free Flyer
HM	Habitation Module
HZ	Hertz (Cycles per second)
IM	Interconnect Module
IOC	Initial Operational Capability
LM	Laboratory Module
LVLH	Local Vertical, Local Horizontal
MAS	Mission Analysis Study
OCZ	Operational Control Zone
OTV	Orbital Transfer Vehicle
OMV	Orbital Maneuvering Vehicle
PCM	Power Conditioning Module
PMAD	Power Management and Distribution
RMS	Remote Manipulator System
RCS	Reaction Control System
SE&I	System Engineering and Integration
SSTF	Space Station Task Force
SOC	Space Operations Center
SOW	Statement of Work
SSCM	Space Station Cost Model
TEA	Torque Equilibrium Attitude
TDRS	Tracking and Data Relay Satellite
WCS	Waste Control System

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4.0 SUBSYSTEM/SYSTEMS DEFINITION

4.1 ECLSS

The ECLSS described herein is of the partially closed cycle option. The option has been selected as the result of a trade study on degree of ECLSS cycle closure. The options on cycle closure considered include:

1. Open cycle: Shuttle-type ECLS

2. Enhanced open cycle: Open cycle plus regenerative CO2 removal

3. Minimum closed cycle I: Regenerative CO2 removal and close the hygiene water and shower water cycle with a multi-filtration process.

4. Minimum closed cycle II: In addition to the regenerative CO2 removal and multi-filtration subsystems, a post-treatment subsystem will process the humidity condensate for the makeup hygiene water and part of the drinking water requirements.

5. Partially closed cycle: Close O2, potable, hygiene and wash water cycles. Only food will be resupplied.

6. Completed closed cycle: Close all cycles, including regeneration of food.

The trade study first rules out the completely close cycle option from further consideration due to the high cost and risk of developing food regeneration technology. For the other options the launch weight/volume decrease with closure of the ECLS cycles, while the power requirements increases with the

cycle closure. The life cycle cost savings realizable by the ECLS of the partially closed cycle option will certainly overwhelm the additional DDT&E costs required in a short period of Space Station operation. The trade study concludes that the partially closed cycle ECLS strikes the optimum balance between developmental costs, technology risk and resupply penalties for station application.

Though the trade study strongly suggests that the partially closed cycle option embodies the "optimum" degree of cycle closure, the concept of an evolutionary Space Station may dictate that varying ECLS configurations be adopted for different versions of the station. For example, during the initial buildup of the station, an ECLS with a lower degree of cycle closure may be used for intermitten manned occupancy of the station by a small crew prior to installation and activation of the habitat module. Only when continued manned occupancy of the station by a large crew begins, will a partially closed cycle ECLS be needed.

To accommodate evolutionary changes in the station, the partially closed cycle ECLS must contain sufficient modularity and flexibility in design. That is, the design shall allow incorporation of additional capabilities without reincurring DDT&E costs, as the crew size and habitable area increase with the incremental buildup of the station to fulfill the mission requirements. The design shall also be able to adapt to, without requiring major alterations, advances in the state-of-the-art ECLS concepts from which processes with greater energy efficiency, process yield, reliability and maintainability will emerge.

A preliminary conceptual design of the partially closed cycle ECLS satisfying the above requirements has been developed for each of the initial and the growth versions of the station. The two versions of ECLS are essentially the

same except in numbers of the subsystems distributed in the station modules. The ECLS is illustrated in the integrated system schematics shown in figure 4.1-1. The ECLS is applicable for all three station configurations, i.e., the SOC, Streamlined and Delta configurations, since they have the same number of modules.

The ECLS uses a regenerative CO2 removal subsystem to collect the metabolically generated CO2. The CO2 collected is delivered to a CO2 reduction subsystem in which the CO2 is converted to water via hydrogenation. The water produced by CO2 reduction and the humidity condensate collected in the heat exchanger for cabin temperature and humidity control are used together as potable water for drink and food preparation after being sterlized through a post-treatment process. The O2 supply is provided by a water electrolysis process which draws water from hygiene water storage/supply. The hygiene, shower and urine water recovery subsystem employs a phase-change process with pre- and post-treatments to produce quality water for hygiene shower electrolysis uses. The N2 supply is provided by a cryogenic or high pressure gas nitrogen storage. A dishwasher and a clothes washer/dryer are installed to provide additional crew comfort and to enable further saving of expendables. The effluent wash water is reclaimed by a filtration wash water recovery subsystem. An example of the mass balance schematic of the initial version ECLS is given in figure 4.1-2. The mass balance is for an eight-man crew and all flow rates are in lb/day. It is noted that SPE, EDC, Sabatier, and VCD are chosen as the O2 generation, CO2 removal, CO2 reduction and hygiene water reclamation processes in the schematic only for the purpose of illustrating the ECLS subsystem mass balance. The baseline subsystems for these functions have not been selected.

The ECLS also contains other components which are common to ECLS of both

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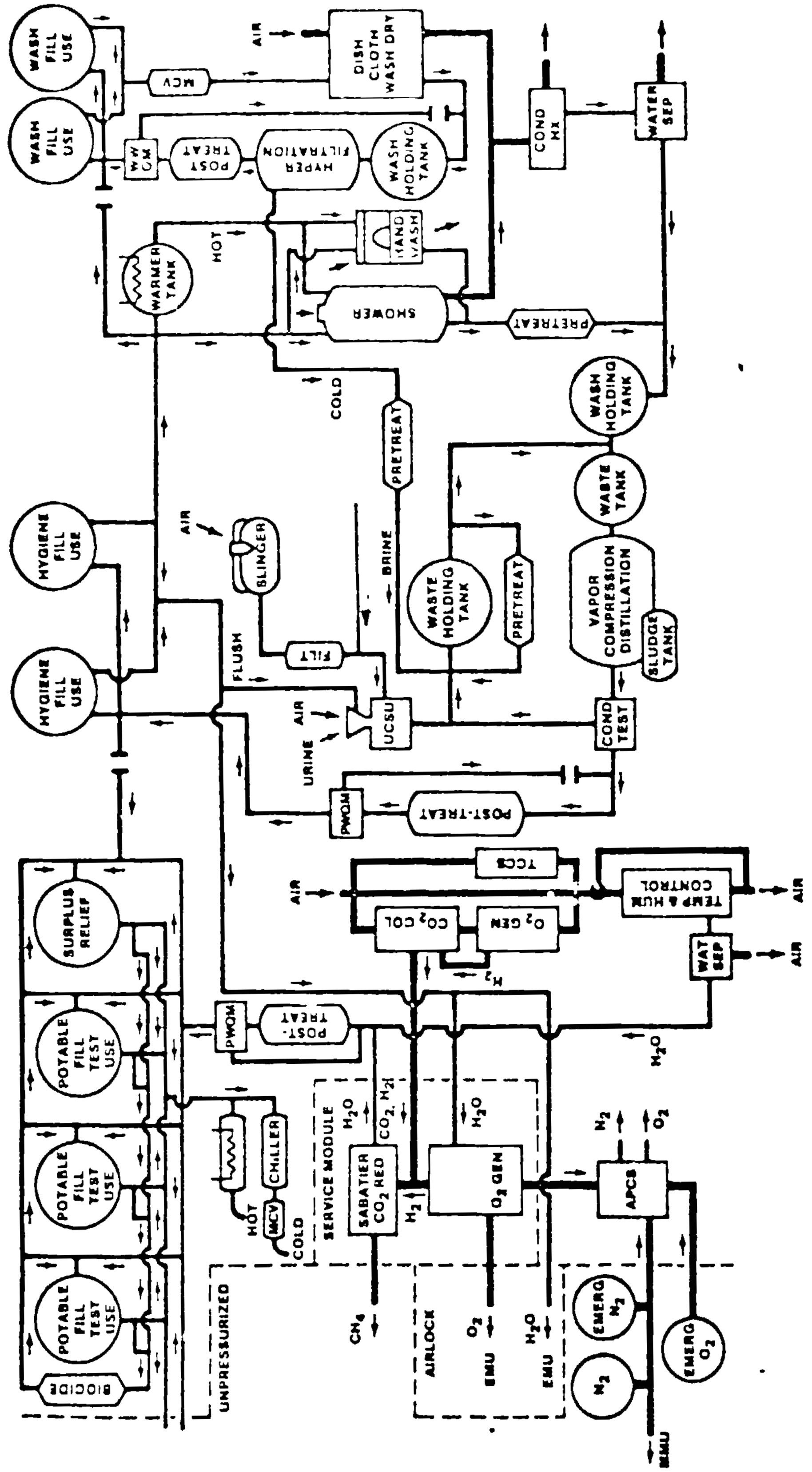

FIGURE 4.1.1 ECLS SUBSYSTEM FOR CDG VERSION SPACE STATION

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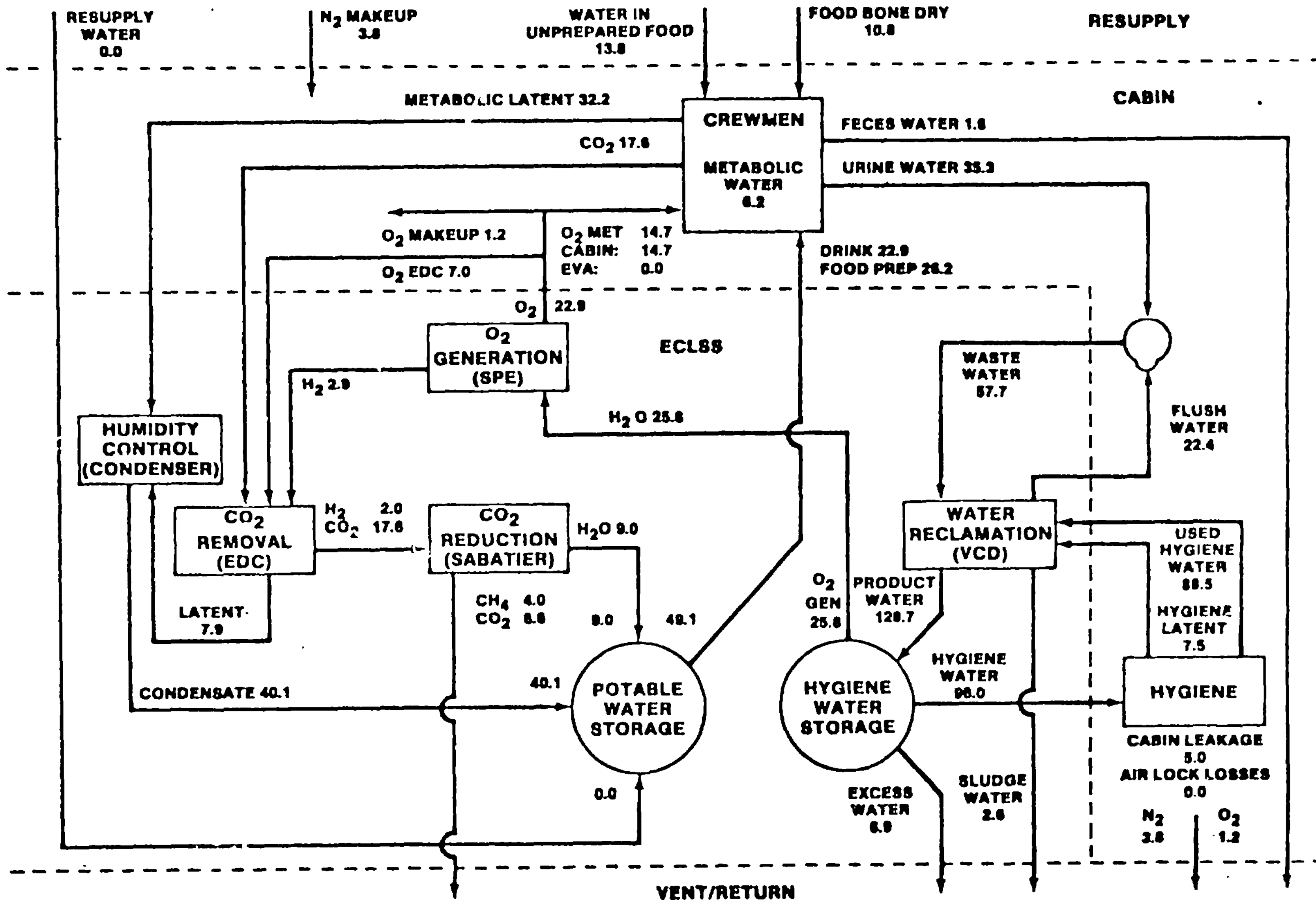

FIGURE 4.1.2 ECLS SUBSYSTEM MASS BALANCE

open-cycle and closed-cycle options. These components include air pressure control panels, condensing heat exchangers, trade contaminant control subsystems, water separators, fans, potable, hygiene and wash water storage tanks, waste water holding tanks, air and water quality monitors, hand wash. water heaters and chillers, slingers (commodes) urin collection units, fire detection and suppression, personnel escape, and EVA support subsystems. Although the integrated ECLS is comprised of the subsystems mentioned above, it is not necessary for every habitable module of the station to have all the subsystems. Depending on the criticality of the functions performed by the subsystems, the subsystems can be economically allocated to provide adequate fail/operational redundancy for the Space Station after IOC. For this purpose, the ECLS subsystems are assembled into the following functional groups:

1. Fire detection and suppression

2. CO2 removal

3. Cabin thermal control - including temperature and humidity controls and ventilation

4. CO2 reduction

5. O2 supply - oxygen generation

6. N2 supply

7. Atmosphere control - including N2/O2 panel, trace contaminant control, atmosphere monitoring and microbiological control

8. Potable water reclamation - including water quality monitoring, pre- and post-treatments

9. Potable water storage

10. Waste management - commodes and urinals

11. Shower

12. Handwash

13. Laundry - clothes washer and dryer

14. Dishwasher

15. Wash water reclamation and storage

16. Emergency escape - POS and Balls (PRS)

17. Emergency supply - 02, N2 and water

18. EVA support - including two suits and backpacks

The recommended functional allocation of the ECLS which provides reasonable capabilities and redundancy to satisfy the increasing requirements and to accommodate the growing configuration of the station, is given in tables 4.1-1 and 4.1-2 for the initial and growth Space Station, respectively. The weight, and volume requirements of the functional groups are given in table 4.1-3. All the subsystems are sized for a four-man capacity. Since some subsystems, such as CO2 removal, 02 generation, and water reclamation have more than two units in the station, the operating policy will produce unnecessary capacity at the expense of extraneous power consumption. To optimize the power consumption, duty cycles have to be set for all subsystem units. The daily average power requirements of the subsystems, based on an evenly distributed metabolic load in the station modules, are given in table 4.1-4 and 4.1-5 for the initial and growth versions of ECLS, respectively.

TABLE 4.1.1 IOC SPACE STATION ECLS FUNCTIONAL ALLOCATION

FUNCTION	C/C1	HM1	LAB1	LAB2	LM
Fire. Det. & Supp.	X	X	X	X	X
CO_2 Removal	X	X	X	X	X
Cabin Thermal Control	X	X	X	X	X
CO_2 Reduction	X	X	X		
O_2 Supply	X	X	X		
N_2 Supply					X
Atmospheric Control	X	X	X	X	X
Potable Water Reclamation	X		X		
Potable Water Storage					X
Waste Management	X	X			
Shower		X			
Handwash	X	X	X	X	X
Laundry		X			
Dishwasher		X			
Wash Water Reclamation & Storage		X			
EVA Support	X				

TABLE 4.1.2 GROWTH (14-MAN)
SPACE STATION ECLS FUNCTIONAL ALLOCATION

FUNCTION	C/C1	HM1	LAB1	LAB2	LM1	C/C2	HM2	LAB3	LAB4	LAB5	LAB6	LM2
Fire Det. & Supp.	X	X	X	X	X	X	X	X	X	X	X	X
CO_2 Removal	X	X	X	X	X	X	X	X	X	X	X	X
Cabin Thermal Control	X	X	X	X	X	X	X	X	X	X	X	X
CO_2 Reduction	X	X	X			X	X	X				
O_2 Supply	X	X	X			X	X	X				
N_2 Supply					X							X
Atmospheric Control	X	X	X	X	X	X	X	X	X	X	X	X
Potable Water Reclamation	X		X			X		X				
Potable Water Storage					X							X
Waste Management	X	X				X						
Shower		X					X					
Handwash	X	X	X	X	X	X	X	X	X	X	X	X
Laundry		X					X					
Dishwasher		X					X					
Wash Water Reclamation & Storage		X					X					
EVA Support	X					X						

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TABLE 4.1.3 ECLS FUNCTIONAL GROUPS WEIGHT AND VOLUME REQUIREMENTS

(NO DUTY CYCLE)

FUNCTION	WEIGHT (LB)	VOLUME (FT^3)
Fire Detec. & Supp.	46	3
CO_2 Removal	190	12
Cabin Thermal Control	390	17
CO_2 Reduction	90	6
O_2 Supply	180	20
N_2 Supply	415	34
Atmospheric Control	360	15
Potable Water Reclamation	600	35
Potable Water Storage	240	16
Waste Management	560	22
Shower	120	44
Handwash	75	6.0
Laundry	90	6
Dishwasher	30	8
Wash Water Reclamation & Storage	400	30
Emergency Escape	240	9
Emergency Supply	TBD	TBD
EVA Support	501	56

TABLE 4.1.4 IOC SPACE STATION ECLS DAILY AVERAGE POWER REQUIREMENTS (W)

FUNCTION	C/C1	HM1	LAB1	LAB2	LM
Fire Det. & Supp.	50	50	50	50	50
CO_2 Removal	220	220	220	220	220
Cabin Thermal Control	320	320	320	320	320
CO_2 Reduction	30	40	40	-	-
O_2 Supply	820	1020	1020	-	-
N_2 Supply	-	-	-	-	-
Atmospheric Control	160	160	160	160	160
Potable Water Reclamation	280	-	280	-	-
Potable Water Storage	-	-	-	-	-
Waste Management	15	15	-	-	-
Shower	-	26	-	-	-
Handwash	30	30	30	30	30
Laundry	-	40	-	-	-
Dishwasher	-	30	-	-	-
Wash Water Reclamation & Storage	-	120	-	-	-
EVA Support	125	-	-	-	-
Subtotals	2050	2061	2110	940	940

Total = 7.8 Kw

* The metabolic load is evenly distributed in 5 modules

TABLE 4.1.5 GROWTH (14-MAN)
SPACE STATION ECLS DAILY AVERAGE
POWER REQUIREMENTS (W)

FUNCTION	C/C1	HM1	LAB1	LAB2	LM1	C/C2	HM2	LAB3	LAB4	LAB5	LAB6	LM2
Fire Det. & Supp.	50	50	50	50	50	50	50	50	50	50	50	50
CO_2 Removal	160	160	160	160	160	160	160	160	160	160	160	160
Cabin Thermal Control	200	200	200	200	200	200	200	200	200	200	200	200
CO_2 Reduction	20	40	40	-	-	30	40	40	-	-	-	-
O_2 Supply	820	870	870	-	-	820	870	870	-	-	-	-
N_2 Supply	-	-	-	-	-	-	-	-	-	-	-	-
Atmospheric Control	150	150	150	150	150	150	150	150	150	150	150	150
Potable Water Reclamation	280	-	280	-	-	280	-	280	-	-	-	-
Potable Water Storage	-	-	-	-	-	-	-	-	-	-	-	-
Waste Management	15	15	-	-	-	15	-	15	-	-	-	-
Shower	-	26	-	-	-	-	26	-	-	-	-	-
Handwash	30	30	30	30	30	30	30	30	30	30	30	30
Laundry	-	40	-	-	-	-	40	-	-	-	-	-
Dishwasher	-	30	-	-	-	-	30	-	-	-	-	-
Wash Water Reclamation & Storage	-	120	-	-	-	-	120	-	-	-	-	-
EVA Support	125	-	-	-	-	125	-	-	-	-	-	-
Subtotal	1860	1721	1720	590	590	1860	1716	1735	590	590	590	590

Total = 14,162. W

* The metabolic load is evenly distributed in 12 modules

4.2 Thermal Control Subsystem Description

4.2.1 Introduction

For this evaluation, unless otherwise noted, the following guidelines, requirements, and assumptions were used:

1. Orbit thermal environment

   a. Inclination: 28½°

   b. Altitude: 270 n. miles

   c. Solar beta angle: 52°

2. Separate cooling systems for fuel cells/electrolysis (160°F average) and crew comfort, station housekeeping, and users (70°F average). As station requirements become better defined, a third temperature level (say 40°F) may be desirable to better accommodate crew metabolic and low temperature experiment requirements.

3. Vehicle waste heat characteristics: Tables 4.2-1 and 4.2-2.

4. Radiator surface coating properties (end of life).

   a. Infrared emissivity (ϵ): 0.78

   b. Solar absorptivity (α): 0.20

5. Radiator fin efficiency (η): 0.85

6. Radiator unit weight: 1.0 lbs/ft^2.

7. Station multilayer insulation characteristics

   a. Number of layers: 20

   b. Unit weight: 0.25 lbs/ft^2.

8. Direct thermal radiation blockage (view factor) effects considered but multiple reflection (gray body factors) between station elements not evaluated.

USER	C/C_1	HAB_1	LAB_1	LAB_2	LOG_1	SOLAR ARRAY BOOM
ECLSS	2.0	4.5	0.5	0.5	0.1	---
THERMAL CONTROL	0.7	0.4	1.0	1.0	0.1	1.5
CREW PROVISIONS	0.2	0.8	0.2	0.2	0.2	---
COMM./TRACKING	0.3	0.2	0.2	0.2	0.1	4.5
DATA MANAGEMENT	0.2	---	0.1	0.1	---	---
DISPLAYS/CONTROLS	1.6	---	0.8	0.8	---	---
GNC	3.6	---	---	---	---	---
POWER SYSTEM	0.4	0.3	2.1	1.1	---	46.0[a]
PROPULSION	0.2	---	---	---	---	0.2
PAYLOADS/EXPERIMENTS	---	---	40.0	20.0	---	---
SUBTOTALS	9.2	6.2	44.9	23.9	0.5	52.2
METABOLIC	0.6	2.4	0.6	0.6	---	---
TOTALS	9.8	8.6	45.5	24.5	0.5	52.5

(a) 41.5 KW ASSUMED AT 160°F AND 4.5 KW AT 70°F. TOTAL HEAT REJECTION LOAD = 141 KW

TABLE 4.2.1 IOC VEHICLE WASTE HEAT CHARACTERISTICS (KW)

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USER	C/C 1,2	HAB 1	LAB_1	LAB_2	LOG 1,2	HAB_2	LAB 3-6	TRUSS
ECLSS	2.0	4.5	0.5	0.5	0.1	4.0	0.5	---
THERMAL CONTROL	0.7	0.4	1.0	1.0	0.1	0.4	0.3	3.0
CREW PROVISIONS	0.2	0.8	0.2	0.2	0.2	0.4	0.2	---
COMM./TRACKING	0.6	1.6	0.6	0.6	0.1	1.6	0.6	5.3
DATA MANAGEMENT	0.2	---	0.1	0.1	---	---	0.1	---
DISPLAYS/CONTROLS	1.6	---	0.8	0.8	---	---	---	---
GNC	3.6/1.7	---	---	---	---	---	---	---
POWER SYSTEM	0.4	0.4	2.2	1.2	---	0.3	0.8	92.0[a]
PROPULSION	0.2	---	---	---	---	---	---	0.2
PAYLOADS/EXPERIMENTS	---	---	40.0	20.0	---	---	15.0	---
SUBTOTALS	9.5/7.6	7.7	45.4	24.4	0.5	6.7	17.5	100.5
METABOLIC	0.6	2.4	0.6	0.6	---	1.8	0.6	---
TOTALS	10.1/8.2	10.1	46.0	25.0	0.5	8.5	18.1	100.5

[a] 83 KW ASSUMED AT 160°F AND 9 KW AT 70°F.

TABLE 4.2.2 GROWTH VEHICLE WASTE HEAT CHARACTERISTICS (KW)

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9. No 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).

4.2.2 Common Design Features

The following discussions provide more detail on some of the thermal control considerations presented in section 3.2.3.5.

4.2.2.1 Radiators

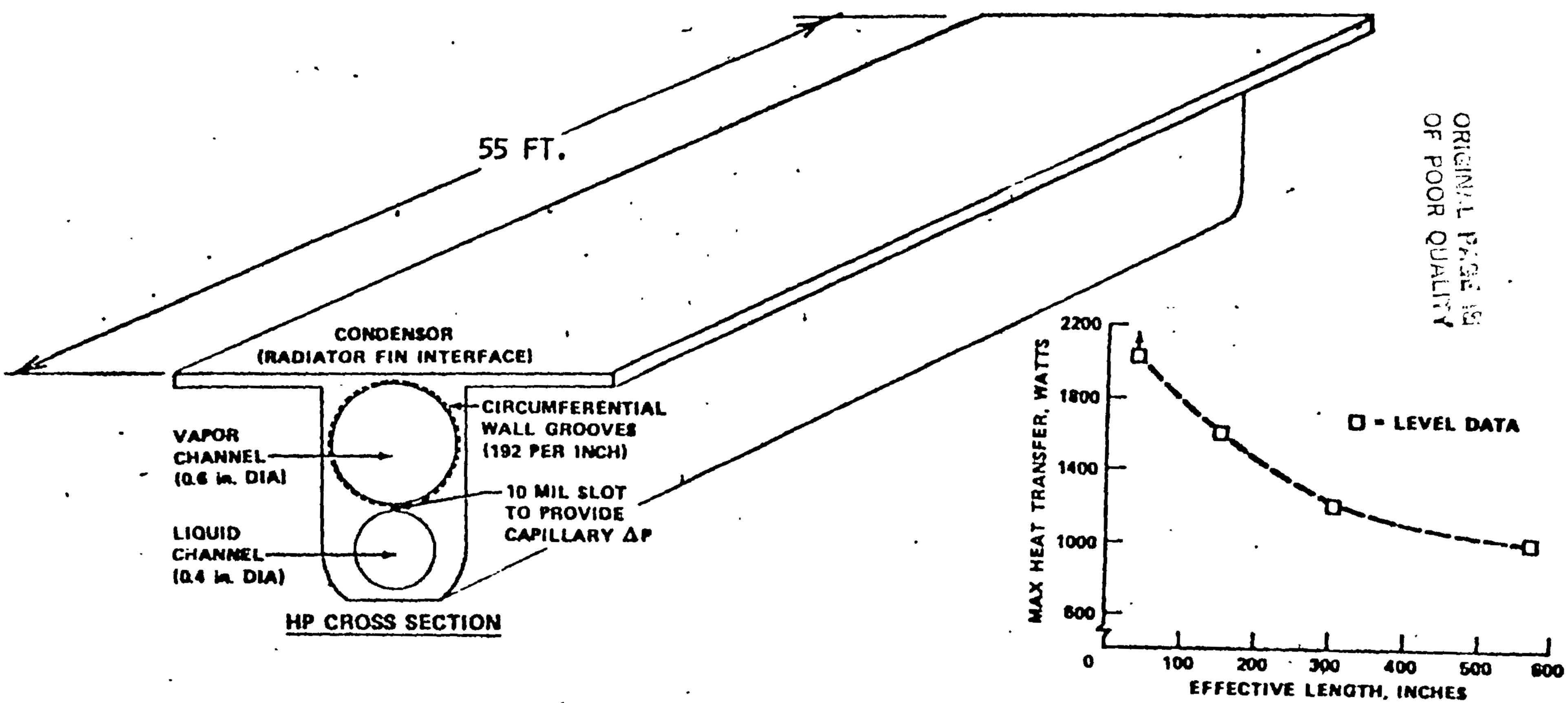
Candidate radiators (body-mounted and deployed/truss-mounted) for the Space Station will use heat pipes as discussed in section 4.2.1.8. The heat pipes will use the high capacity monogroove configuration shown in figure 4.2-1. In this approach, the deployed or truss-mounted radiators will be constructed in space with a RMS as illustrated by figure 4.2-2. Each individual radiator element (about 1' wide by 50' long) can be removed and replaced if damaged. The heat pipe radiator elements are "plugged in" to contact heat exchangers as shown by figure 4.2-3. These heat exchangers provide a loose fit for the heat pipes when they are intiially 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.

4.2.2.2 Heat Acquisition and Transport

As mentioned in section 3.2.3.5.2, the candidate ATCS design uses a "thermal bus" concept. The operation of the "thermal bus" can best be described by referring to figure 4.2-4. First, fluid leaves the condenser as subcooled liquid and then enters a central mechanical pump which raises the liquid line pressure above that of the vapor line. Flow modulation valves control the flow of liquid to each module interface heat exchanger, sensing temperature to vary the valve setting. At each heat exchanger, liquid is vaporized and flows

Figure 4.2-1

HIGH CAPACITY, EXTENDED LENGTH HEAT PIPE



- 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

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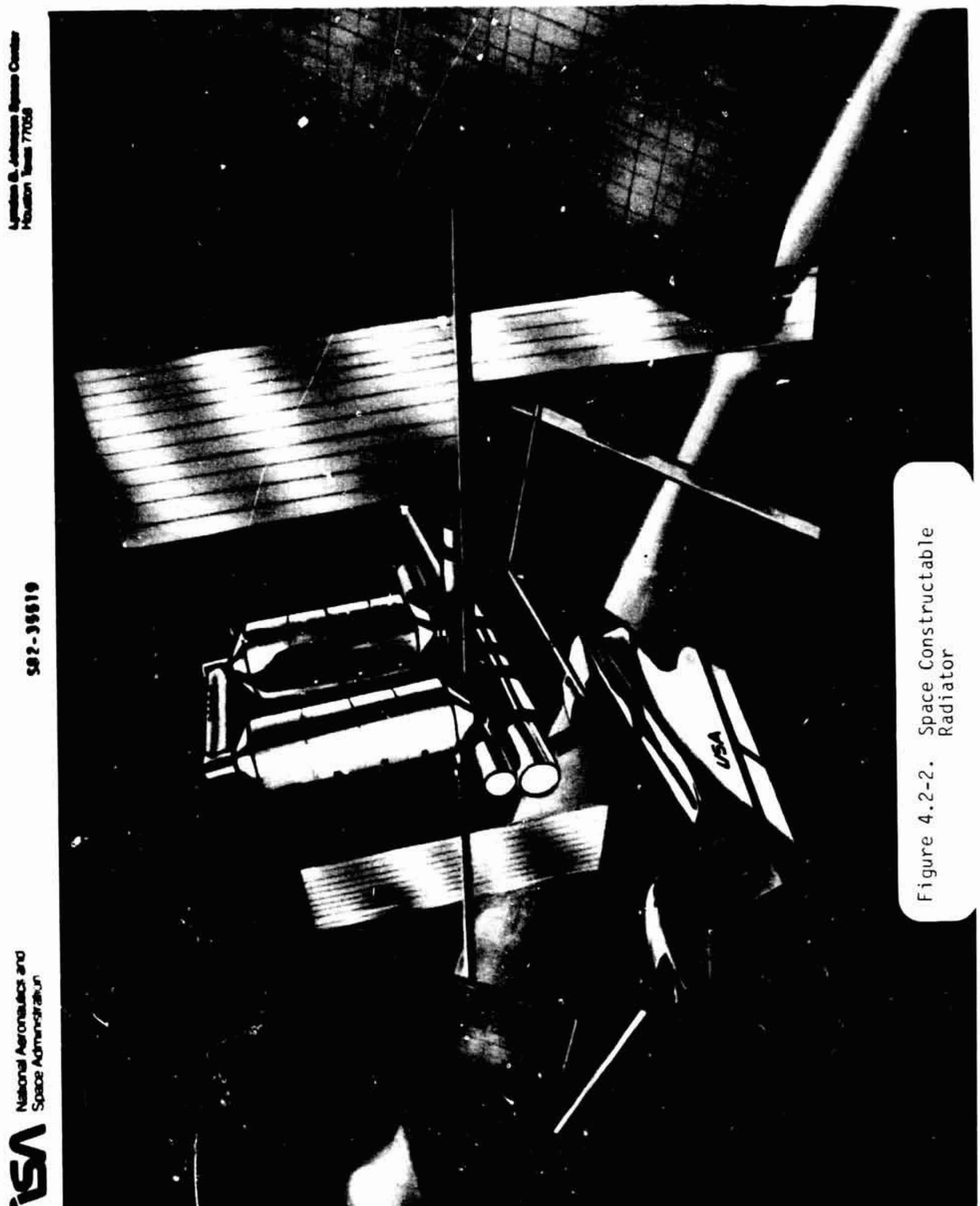

Figure 4.2-2. Space Constructable Radiator

MECHANICAL

CONTACT
HEAT EXCHANGER

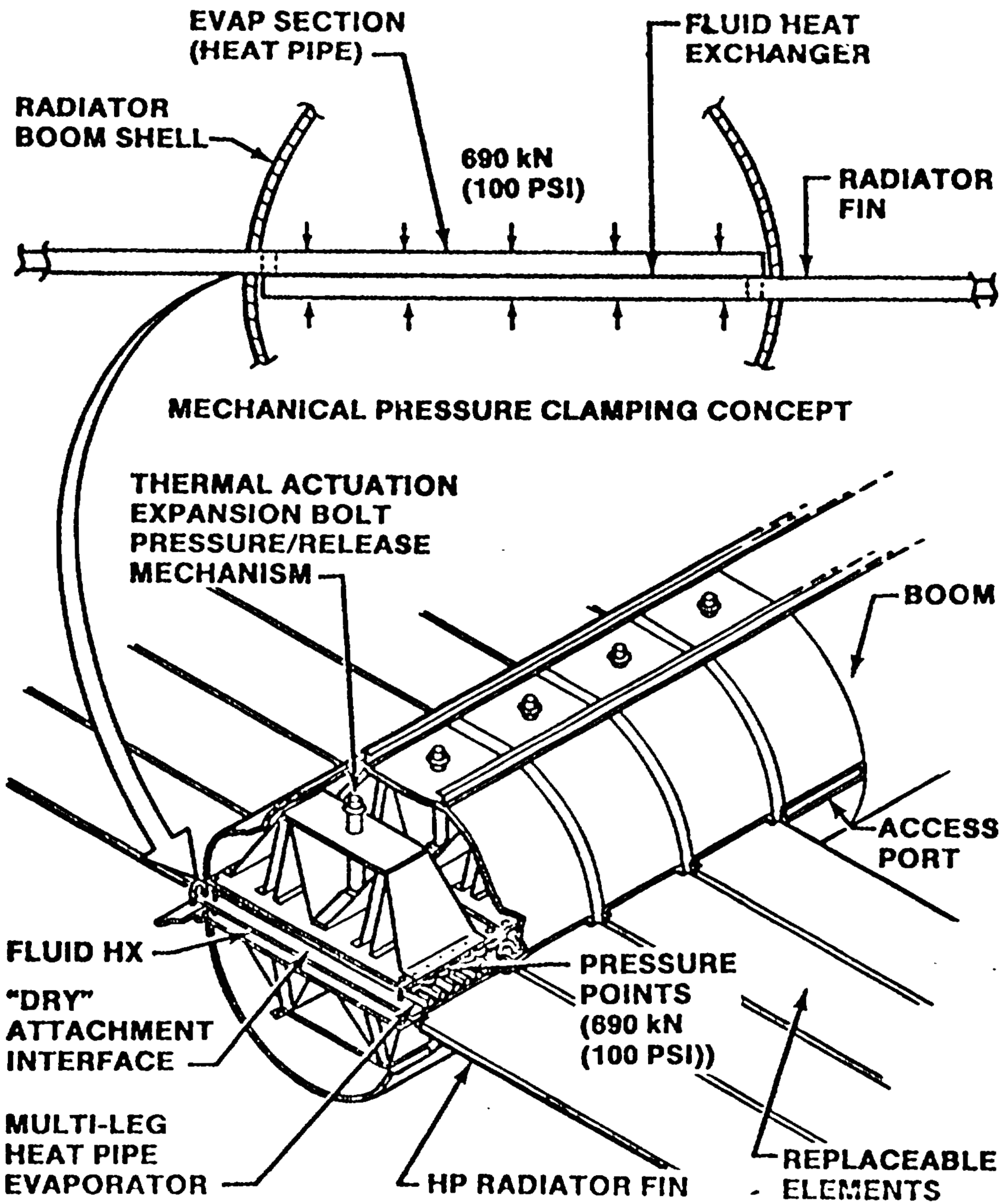

Figure 4.2-3

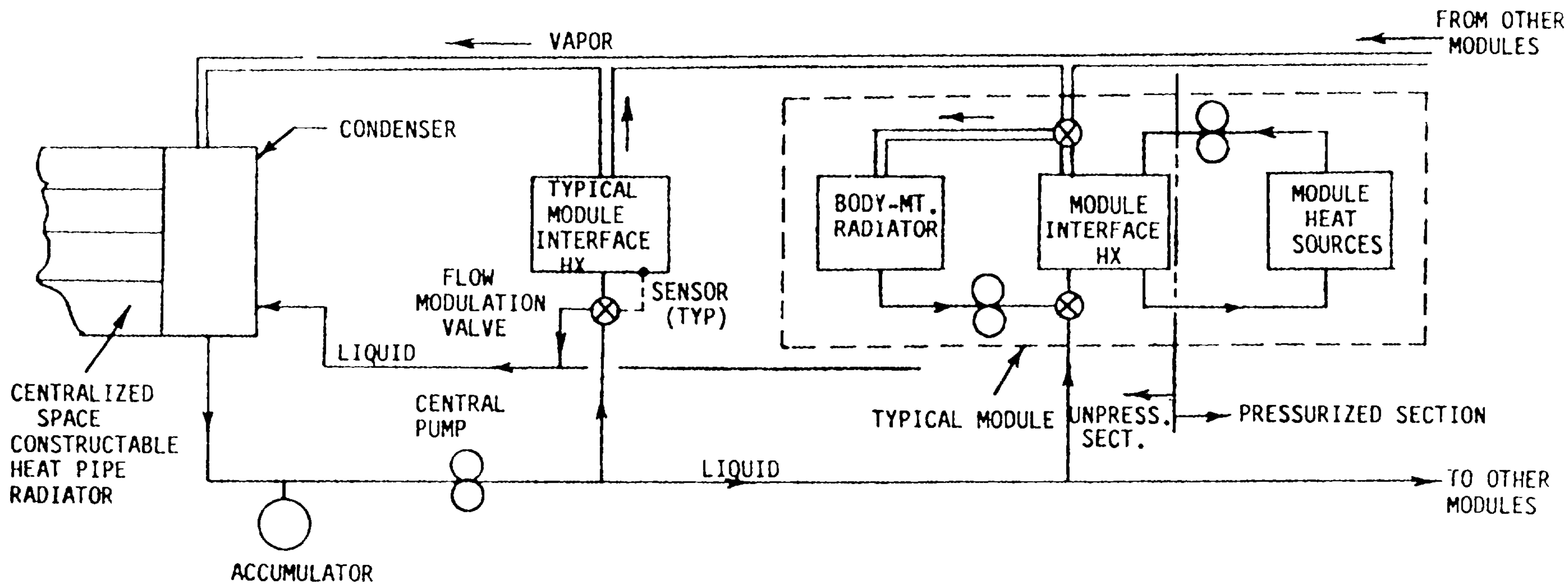

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Figure 4.2-4. Active Thermal Control Subsystem Concept

to the vapor line and back to the condenser. Liquid leaving the condenser will be subcocled prior to entering the pump. The pump will further increase subcooling by raising the liquid pressure. At the heat exchanger entrance, subcooling will be essentially lost as the liquid pressure drops to near saturation across the valve. The heat exchangers in this case need only a modest capillary rise capabilit, similar to that of conventional grooved heat pipes since it is needed only to position the liquid within the exchanger.

In order to control the valve setting and thus the flow of liquid each heat exchanger, sensors must be used to open and close the valves. T could be temperature sensors placed either on the payload itself or on the heat exchanger in a location specifically designed to dry out preferentially. This could be accomplished by the use of a wider groove with a small electrical power input (few watts) to provide a positive signal under no load conditions. An alternative to temperature sensors would be liquid level sensors which sense the presence of liquid in a groove.

In order for these heat exchangers to transfer heat from the thermal bus to the payloads (act as a condenser), the flow modulation valve could be a three-way valve coupled to a small diameter liquid suction line provided to return the condensate to the pump inlet. In reality, the flow modulation valve is a three position/two-way valve since only two ports are open at any one time. As a heat exchanger temperature drops, vapor will flow into the heat exchangers from the vapor line, condense, and flow back to the pump inlet. The same temperature sensor could be used to control the valve setting for evaporation or condensation. Where it is known that only certain payloads will required heat, these heat exchangers would be coupled to the liquid suction line via the three-way valve.

4.2.2.3 Coatings/Insulation

The thermal surface treatment of Space Station elements is an integral part of the overall active and passive thermal design and must be selected to obtain and acceptable heat balance that will not result in local condensation on internal surfaces of pressurized volumes, excessive heat leaks, or violate structural and subsystems temperature limits.

The selection of particular surface properties is highly dependent on the passive thermal design approach; that is, designing the elements as thermos bottles or designing for maximum heat loss to the external environment. The first approach tends to maximize requirements for heat rejection whereas the second can reduce the size and weight of active heat rejection systems. For pressurized elements, passive heat losses in the range of three to five BTU/HR-ft^2 for external surface area are considered achievable.

A wide range of solar absorptivities and emissivities are available in paints and films but these pose some degradation and maintenance problems. Treated metal (anodized and alodined) surfaces on the otherhand require minimal, if any, refurbishment but do not offer the range of surface properties available in paints and films. Specifically, low solar absorptivity to emissivity ratios in the 0.10 to 0.20 range for application to radiators and modules cannot be achieved since the absorptivity tends to increase with emissivity. However, these treatments can be applied where direct solar incidence does not exist or where high solar absorptivity to emissivity ratios are acceptable. Space Station requirements encompass a life time of 30 years in low earth orbit. Degradation of thermal control paints begin to be significant in two to five years due to ultraviolet radiation, contamination sources, and atomic oxygen effects. Films have been found to be stable for five years or more. The problem of refurbishment is the prime driver in applying treated metal

surfaces where practical, or in desensitizing the design to degradation. Since this will only be achievable on a limited basis, systems must be devised for periodic renewal of surface properties.

All pressurized elements and power generation modules will require insulation to control the heat gains and losses. The insulation sizing along with the thermal control coating will be selected to minimize the size and complexity of the active heat acquisition and rejection system.

In the case of the satellite service structure, OMV and OTV hangars, insulation will be applied to modulate the large swing in orbital environments that would be experienced if unprotected and thereby minimize station power requirements and payload thermal systems operations.

CMG's, propulsion modules, propellant tanks, and distribution systems require a thermos bottle design to minimize makeup heat loss requirements.

All of these applications can be accommodated with lightweight, high performance, multilayer insulation systems which are virtually maintenance free. Where micrometeoroid shields exist, such as around modules, the insulation will be located beneath the shield on the module structure with an organically coated aluminized film cover. For exterior surfaces, a glass cloth aluminized film laminate to reduce glare will be applied.

4.2.2.4 Propulsion System

Refer to section 3.2.3.5.2.

4.2.2.5 Antenna System

Refer to section 3.2.3.5.2.

4.2.2.6 Solar Array

Passive thermal control of the solar array can be obtained to a small degree by optimizing the optical properties of the surfaces. Given a typical solar

cell with effective absorptance of 0.68 and emittance of 0.81, a range of solar panel maximum and minimum temperatures at a solar Beta angle of 0° in a solar inertial attitude were computed for various backside emittances and absorptances. The results shown in figure 4.2-5 indicate that the temperature does not vary much for the common range (cross-hatched area) of properties. For a kapton substrate with α and ϵ of 0.44 and 0.8, respectively, the maximum and minimum temperatures are 156°F and -139°F. (Typical of PEP solar array, reference PEP Solar Array Definition Study Final Technical Report, TRW Defense and Space Systems Group, TRW 35515-6001-RU-00, December 1979, page 98.) With worse α and ϵ of 0.5 and 0.7 the maximum and minimum temperatures are 166°F and -135°F, and for α and ϵ of 0.2 and 0.9 they become 138°F and -142°F.

4.2.2.7 Verification

Integrated thermal control system verification experience is based on past programs which considered two types of vehicles (one time use and reuseable). The verification approaches were based primarily on risk assessment and funding available. For example, Apollo required ground thermal vacuum tests and analyses supported by minimal flight tests to provide confidence in the thermal design prior to flight. On the other hand, the Orbiter allowed for major in-flight testing as a result of its systems redundancy, ability to manage the thermal environment and quick return to earth capability which all tended to minimize risk. Ground tests were limited to radiator/heat transport systems, insulations, and minor heater systems and components tests. Verification of the integrated thermal control system was basically by analysis prior to first flight.

The Space Station presents a third type of vehicle which falls somewhat between the first two types. That is, the station is a continuous use vehicle with limited return to earth capability. Table 4.2-3 presents considerations

which must be addressed in determining the overall approach to verification of the integrated thermal control system. These include environment, vehicle operational capability, design sensitivity and margins, degree of systems isolation, design commonality and risk. Table 4.2-4 provides a comparison of ground and flight (Orbiter and station buildup) testing.

Thermal ground test objectives could be extended in scope to include acceptance/verification testing of modules/subsystems, payload interface development/verification, software checkout/verification, habitability and crew interface assessment, procedures checkout/verification and fault isolation and maintainability assessment.

4.2.2.8 Technology Assessment

There are a number of ATCS technology limitations and challenges for Space Station as summarized by Table 4.2-5.

Under OAST sponsorship and as approved by the Space Station Technology Steering Committee, a NASA-wide thermal technology program is underway to satisfy these technology limitations. The objectives of this program as it relates to a Space Station ATCS are summarized by tables 4.2-7, 4.2-8, and 4.2-9. The heat rejection or high capacity heat pipe radiator technology development started in 1979 is at the highest level of technical maturity. Full scale prototype units (1' by 50') will be tested under thermal vacuum conditions in JSC's Chamber B during January 1984 tests. This testing will be complemented by extensive design support testing already successfully accomplished under sea-level laboratory conditions. In addition, tests of 0-G priming characteristics have been successfully demonstrated in KC-135 flights. Most recently, a small-scale heat pipe radiator experiment was flown on STS-8 where thermal performance was successfully demonstrated under actual orbital conditions.

MAXIMUM AND MINIMUM SOLAR ARRAY TEMPERATURES
VS
BACKSIDE EMITTANCE AND ABSORPTANCE (α_B)

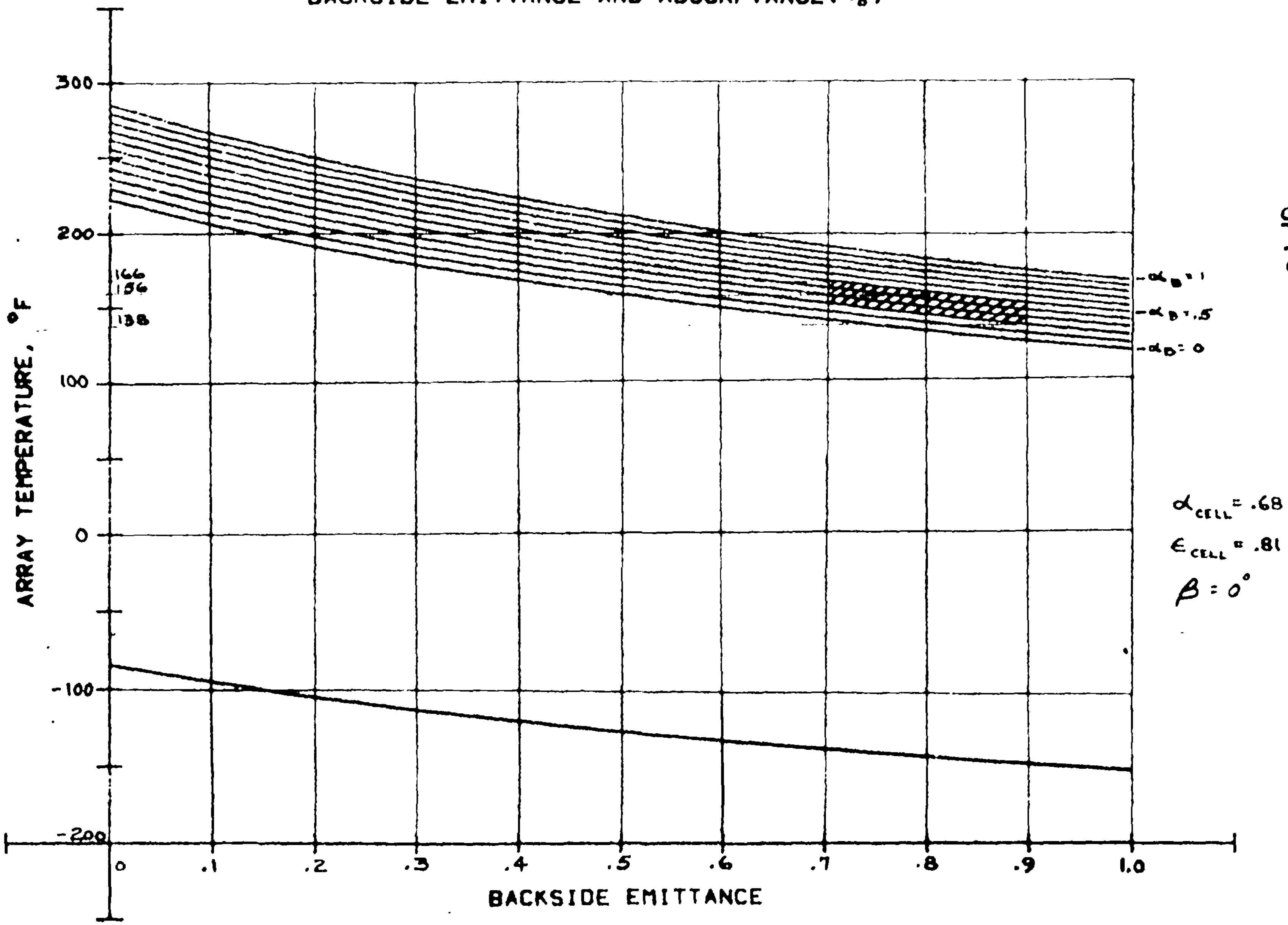


Figure 4.2-5

TABLE 4.2-3

- O SPACE STATION VERIFICATION CONSIDERATIONS
  - o SPACE STATION THERMAL ENVIRONMENT IS A FIXED CYCLE
    - — INITIAL FLIGHT ENVIRONMENT DEPENDENT ON LAUNCH DATE AND TIME WITH ESSENTIALLY NO ON-ORBIT EXTERNAL ENVIRONMENT MANAGEMENT
    - — PAST VEHICLE ENVIRONMENTS COULD BE MANAGED ALLOWING SOME VERSATILITY AND CONSERVATISM IN INITIAL FLIGHT TESTING
  - o SPACE STATION IS A CONTINUOUS USE VEHICLE WITH LIMITED CAPABILITY TO RETURN MODULES TO EARTH FOR MODIFICATION
    - — REUSEABLE QUICK RETURN VEHICLES FIT VERY WELL WITH in-FLIGHT TESTING SINCE ENVIRONMENT CAN BE MANAGED AND MODS (INSTRUMENTATION, COMPONENTS, ETC.) MADE, DURING NON-FLIGHT PERIODS ON THE GROUND, BEFORE SUBJECTING VEHICLE TO MORE SEVERE ENVIRONMENTS
    - — INFLIGHT THERMAL TESTING COULD BE ACCOMPLISHED USING TEST ARTICLES IN ORBITER PLB OR DEPLOYED
      - — DEPLOYED TESTING INFERS TEST ARTICLE ATTITUDE CONTROL CAPABILITY
      - — IN BAY TESTS LIMITS ENVIRONMENT AND REQUIRES ANALYSES AND EXTRAPOLATION TO REAL ENVIRONMENT
    - — FLIGHT THERMAL TESTING OF ACTUAL STATION DURING BUILD UP STATES REQUIRES GROUND CONTROL OF POWER AND OTHER SUBSYSTEM FUNCTIONS AND HAS POTENTIAL OF IMPACTING BUILD UP AND OPERATIONAL SHCEDULES AS WELL AS INFERRING CREW RISKS
    - — GROUND TESTS PROVIDE GREATER FLEXIBILITY IN DETERMINING AND CONTROLLING TEST CONDITIONS, RE-DIRECTION OF TESTING, & TESTING OF REQUIRED MODIFICATIONS WITH LESS PROGRAM SCHEDULE IMPACT
    - — GROUND THERMAL TEST ARTICLES & TEST PROGRAM CAN BE SCOPED TO SUPPORT OTHER SYSTEMS/SUBSYSTEMS ACCEPTANCE/VERIFICATION PROGRAMS
    - — SIZE MAY LIMIT ABILITY TO PERFORM CERTAIN LARGE SCALE GROUND TESTS
  - o DESIGN SENSITIVITY, DEGREE OF ISOLATION, MARGINS
    - — DICTATES NO./TYPES OF MODULES OR ELEMENTS TO BE TESTED (USE SIMILARITY)
    - — TEST ARTICLES SHOULD BE OF FLIGHT QUALITY WITH MINIMUM MODIFICATION AND REFURBISHMENT REQUIRED FOR FLIGHT USE

TABLE 4.2-3 (CONTINUED)

- RISK

  — GROUND TESTING PROVIDES FOR HIGHEST LEVEL OF CONFIDENCE PRIOR TO FLIGHT & EARLIEST OPERATIONAL STATUS

  — FLIGHT TESTS HAVE GREATEST POTENTIAL FOR PROGRAM SCHEDULE IMPACT & SUGGESTS GREATER CREW SAFETY IMPLICATIONS

TABLE 4.2-4
VERIFICATION APPROACH COMPARISON

ITEM	GROUND	FLIGHT
FUNDING	HIGHER INITIAL COST	HIGHER TOTAL COST
CREW RISK	MINIMUM	BUILDUP STAGE TEST - HIGH ORBITER TEST - MINIMUM
SCHEDULE RISK	MINIMUM	BUILDUP STAGE TEST - HIGH ORBITER TEST - MODERATE
DESIGN CONFIDENCE	MAXIMUM	MINIMUM
DESIGN CONSERVATISM REQUIRED	LEAST	MOST
TEST FACILITIES	CHAMBER A/B	BUILDUP STAGES TEST-NONE ORBITER TEST-DEDICATED FLIGHT TIME
TEST ARTICLES	DEDICATED OR REFURB. TO FLIGHT CONFIG. (IOC OR FUTURE GROWTH)	BUILDUP STAGE TEST-FLIGHT ORBITER TEST - DEDICATED OR REFURB. TO FLT. CONFI.
MODIFICATION/RE-TEST	MINIMUM IMPACT	REQUIRES RETURN OF MODULES(S) & MOD FOR RE-TEST (FLT/PROGRAM SCHEDULE IMPACTS)
FLIGHT INSTRUMENTATION	MINIMUM	MAXIMUM
TEST FLEXIBILITY & CONTROL (ENV., FAILURE INVESTIG. (RETEST)	EXCELLENT (SIMULATED ENVIRONMENT,	FAIR (REAL ENVIRONMENT)
SUBSYSTEMS/COMPONENT TESTING PRIOR TO THERMAL TESTS	MODERATE (COULD REPLACE SOME SYSTEMS LEVEL TESTS)	EXTENSIVE
SUPPORT TO OTHER ACCEPTANCE/ VERIFICATION PROGRAMS	HIGH POTENTIAL	LOW POTENTIAL

TABLE 4.2-5
ACTIVE THERMAL CONTROL TECHNOLOGY ISSUES

TECHNOLOGY LIMITATIONS	TECHNOLOGY REQUIREMENT
o SYSTEM INTEGRATION	
CURRENT THERMAL SUBSYSTEMS MAKE LARGE USE OF ELECTRICAL HEATERS AND REQUIRE SIGNIFICANT CREW INVOLVEMENT TO CHANGE SYSTEM OPERATING CONFIGURATIONS AS POWER PROFILES AND HEAT LOADS CHANGE AND SYSTEMS FAILURES OCCUR.	MAKE JUDICIOUS USE OF WASTE HEAT BY MAKING IT READILY AVAILABLE TO SUB-SYSTEMS AND MINIMIZE CREW INVOLVEMENT BY PROVIDING AN INTEGRATED HIGHLY RELIABLE THERMAL UTILITY SYSTEM
o THERMAL ACQUISITION AND TRANSPORT	
CURRENT THERMAL SUBSYSTEMS REQUIRE PRECISE ORDERING OF EQUIPMENT WITHIN THERMAL TRANSPORT CIRCUIT TO MAINTAIN TEMPERATURE CONTROL	MODULARITY/GROWTH CONCEPT OF THE SPACE STATION REQUIRES THAT IT ACCEPT MULTIPLE HEAT LOADS OF VARYING MAGNI-TUDES, HEAT FLUX DENSITY AND LOCA-TIONS WITHOUT CAUSING ADVERSE HEAT SOURCE INTERACTIONS (I.E., BE INSEN-SITIVE TO MULTI-DISCIPLINARY USER LOADS)
o HEAT REJECTION	
CURRENT RADIATOR SYSTEMS REQUIRE LARGE NUMBER INTERCONNECTED FLUID PASSAGES, VALVES, ETC.	LARGE, DEVELOPED RADIATORS ARE RE-QUIRED FOR ANY SPACE STATION CONCEPT. THUS, HEAT REJECTION SIZE AND EFFI-CIENCY IMPROVEMENTS ARE REQUIRED TO MINIMIZE THERMAL SUBSYSTEM COST, WEIGHT AND COMPLEXITY SINCE RADIATOR IS LARGEST AND MOST EXPENSIVE PORTION OF THERMAL SYSTEM.

TABLE 4.2-6
ACTIVE THERMAL CONTROL TECHNOLOGY
LIMITATIONS WITH SHUTTLE-TYPE THERMAL SYSTEMS

- o LOW LONG TERM RELIABILITY
  - o MICROMETEROID/SPACE DEBRIS SENSITIVITY
  - o COMPLEXITY - LARGE NUMBER OF VALVES, CONTROLS, FLUID LINES, REDUNDANT COMPONENTS
  - o DEPENDENCY ON ROTATING EQUIPMENT - HIGH CAPACITY PUMPS

- o COMPLEX/INFLEXIBLE USER INTERFACES
  - o MECHANICAL INTERFACE REQUIRES VERY CONSTRAINED USER PRESSURE DROPS, LEAKAGE SPECS, CLEANLINESS SPEC., ETC.
  - o EACH SUBSYSTEM/PAYLOAD MUST BE PRECISELY LOCATED IN CIRCUIT FOR PROPER CONTROL TEMPERATURE
  - o CONTROL TEMPERATURE LEVEL VARIES DEPENDING ON WHAT UPSTREAM EQUIPMENT IS OPERATING

- o DIFFICULT TO MAINTAIN
  - o MUST BREAK/MAKE FLUID CONNECTIONS
  - o ENTIRE SYSTEM MUST BE SHUTDOWN TO SERVICE (REQUIRES AT LEAST ONE REDUNDANT SYSTEM)

- o HIGH POWER CONSUMPTION
  - o PUMPING POWER APPROXIMATELY 5% OF GENERATED POWER
  - o HIGH CAPACITY PUMPS ARE EXPENSIVE

TABLE 4.2-7
HEAT REJECTION TECHNOLOGY OBJECTIVES

- OBJECTIVE 1 - HIGH CAPACITY HEAT PIPE RADIATOR
  - EXTENDED LENGTH MICROMETEOROID INSENSITIVE HEAT PIPE
  - EFFICIENT, COMPACT EVAPORATOR
  - HIGH CONDUCTIVITY RADIATOR FIN

- OBJECTIVE 2 - DEPLOYABLE/CONSTRUCTABLE RADIATOR SYSTEM
  - ON-ORBIT RADIATOR CONSTRUCTION/SUPPORT
  - RADIATOR-TO-TRANSPORT CIRCUIT INTERFACE DEVICE
  - DEPLOYMENT MECHANISM

- OBJECTIVE 3 - BODY-MOUNTED RADIATOR
  - AUTOMATIC CONTROLLED HEAT PIPES
  - INTERFACE DEVICES
  - DEGRADATION INSENSITIVE COATING

- OBJECTIVE 4 - ENVIRONMENT SENSING RADIATOR SYSTEM
  - MINIMUM ENVIRONMENT (INFRARED AND SOLAR) SENSOR DEVICE
  - RADIATOR SYSTEM DRIVE

- OBJECTIVE 5 - THERMAL COATING MAINTENANCE/REFURBISHMENT
  - CONTAMINATION/DEGRADATION EVALUATION
  - COATING CLEANING
  - COATING REPLACEMENT (IN-SITU APPLICATION)

- OBJECTIVE 6 - ADVANCED RADIATOR CONCEPTS
  - LIQUID DROPLET RADIATOR
  - MOVING BELT RADIATOR

TABLE 4.2-8
HEAT ACQUISITION AND TRANSPORT TECHNOLOGY OBJECTIVES

- OBJECTIVE 1 - CENTRALIZED THERMAL BUS TRANSPORT
  - TWO-PHASE PUMPED HEAT SINK
  - TWO-PHASE CAPILLARY HEAT SINK
  - TWO-PHASE CONTROL SYSTEM
  - TWO-PHASE FLUID DYNAMICS

- OBJECTIVE 2 - HIGH DENSITY HEAT ACQUISITION
  - CAPILLARY PUMPED COLD PLATE
  - EVAPORATOR/HEAT EXCHANGER/COLD PLATES
  - TWO-PHASE HEAT EXCHANGER
  - TWO-PHASE ACCUMULATOR/GAS TRAP
  - CONDENSOR HEAT EXCHANGER/COLD PLATES

- OBJECTIVE 3 - HEAT TRANSFER ACROSS STRUCTURAL BOUNDARIES
  - CONTACT HEAT EXCHANGERS
  - HEAT PIPE THERMAL SLIP RINGS
  - FLUID SWIVEL JOINTS
  - FLEXIBLE HEAT PIPES

- OBJECTIVE 4 - LONG LIFE FLUID SYSTEMS
  - LONG LIFE MAINTAINABLE PUMPS
  - RELIABLE QUICK DISCONNECTS
  - LOW COST, MAINTAINABLE COLD PLATES
  - NON-TOXIC/NON-CORROSIVE SELF-SEALING FLUIDS

TABLE 4.2-9
INTEGRATED SYSTEM TECHNOLOGY OBJECTIVES

- OBJECTIVE 1 - THERMAL STORAGE/LOAD LEVELING/REFRIGERATION
  - TWO-PHASE ENERGY STORAGE
  - LIFE SUPPORT REFRIGERATION
  - HEAT PUMP AUGMENTATION
  - USER SELECTABLE THERMAL INTERFACE
- OBJECTIVE 2 - UTILITY SYSTEM INTEGRATION TEST BED
  - WASTE HEAT UTILIZATION
  - MODULARITY/GROWTH
  - REDUNDANCY/MAINTAINABILITY
  - SYSTEMS FUNCTIONAL VALIDATION
- OBJECTIVE 3 - AUTOMATIC SYSTEM CONTROL/MONITORING/FAULT ISOLATION
  - PASSIVE CONTROL TECHNIQUES
  - AUTOMATIC TEMPERATURE CONTROL
  - WIDE HEAT LOAD RANGE CONTROL
  - MIRCROPROCESSOR APPLICATIONS
- OBJECTIVE 4 - THERMAL COMPUTER MODEL IMPROVEMENT
  - PUMPED TWO-PHASE FLOW ROUTINE
  - TWO-PHASE HX ROUTINE
  - CAPILLARY PUMP ROUTINE
  - HEAT PIPE ROUTINE
- OBJECTIVE 5 - INST. MODULE TEST BED
  - CAPILLARY PUMP
  - CAPILLARY FLAT PLATE
  - PUMPED TWO-PHASE COLD PLATE
- OBJECTIVE 6 - GROUND TEST CAPABILITY
  - TWO-PHASE TESTING TECHNIQUES IN 1-G
  - SCALE MODELING APPLICATIONS
- OBJECTIVE 7 - INFLIGHT HANDLING AND MAINTENANCE
  - ON-ORBIT FLUID CHARGING
  - ON-ORBIT CONSTRUCTION DEVICE
  - ON-ORBIT MAINTAINABILITY EQUIPMENT/REPAIR KITS

4.2.3 General Conclusions

The following conclusions are relatively independent of the vehicle configurations studied:

1. There is not sufficient surface area on the station modules (i.e., body-mounted radiators) to reject the total vehicle waste heat load. Deployed or truss-mounted radiators are required to reject the balance of heat.

2. The use of radiator articulation (i.e. gimballing) and a thermal storage approach will significantly reduce radiator size.

The two-phase thermal bus technology development currently is in transition from breadboard to prototype stage. JSC and GSFC are actively pursuing thermal bus technology through several funded contractor efforts and in-house test bed activities. JSC has a prototype development underway with Grumman for the design, fabrication, and test of a 25 KW capacity system. This prototype hardware should be available for integrated JSC testing at the beginning of FY 1986. GSFC is developing thermal bus technology to accommodate unique experiments or payload thermal control requirements.

4.3 Propulsion System Definition

For the three Space Station configurations specified above, Onboard Propulsion System concepts have been developed to the extent necessary to determine system weights, volume, and configuration differences for comparison purposes. The systems have been sized primarily upon the requirement to provide orbit maintenance at 270 NM. Systems have been sized to provide this function for the worst case (long term) atmospheric density anticipated for the early 1990's. A resupply interval of three months has been assumed and a three months contingency has been provided in the event that one resupply flight has to be postponed. The total impulse to satisfy this function is shown in 4.3-1 along with the respective propellant quantities required for both IOC and growth versions of the three configurations.

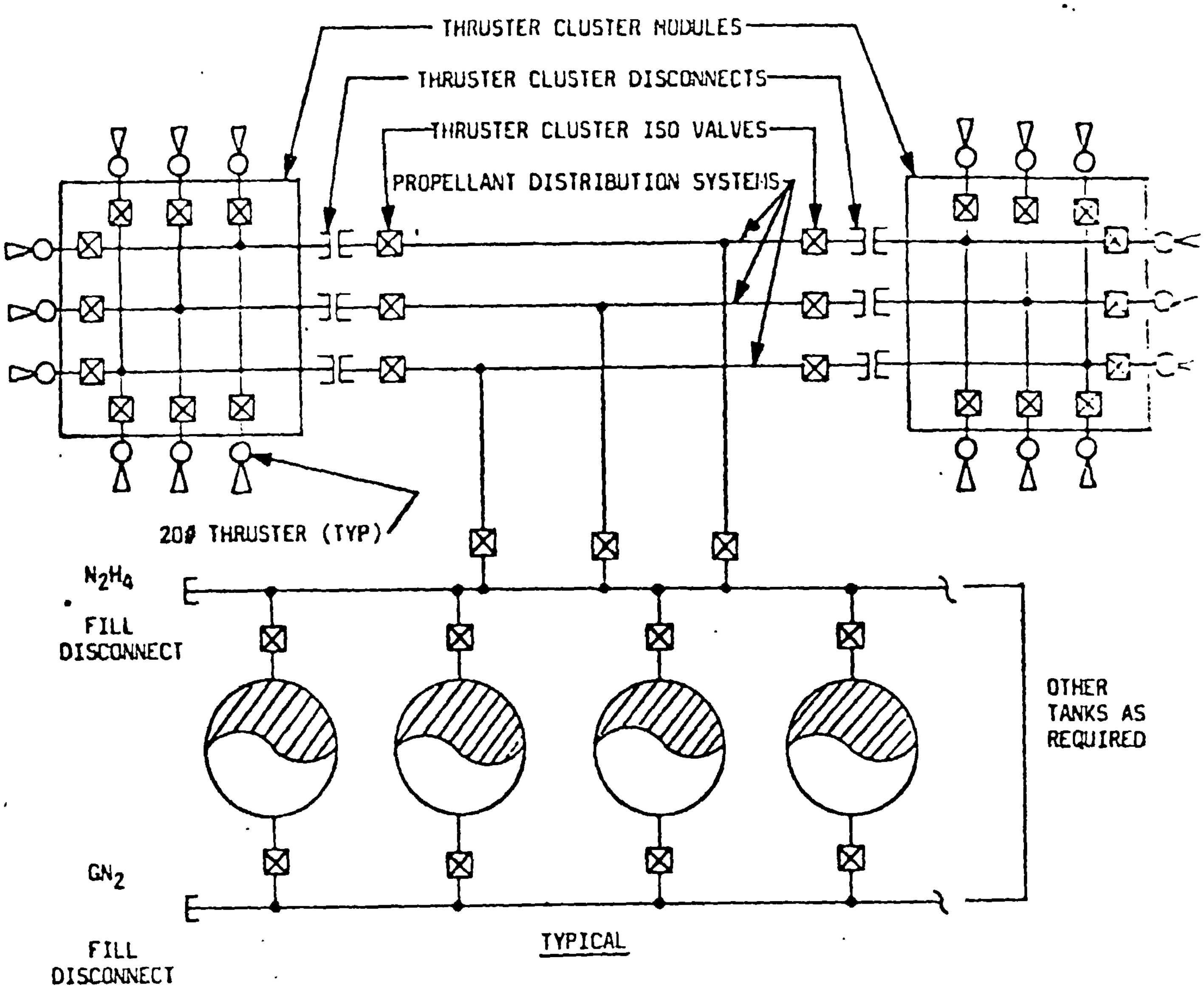
Along within the capability to provide orbit maintenance the systems can provide significant quantities of CMG angular momentum dump capabilities by providing a +X translation with uncoupled thrusters which are also supplying torques in pitch and/or yaw axes. If the CMG momentum dump requirements are small enough and the orbit maintenance burns can be sequenced to coincide with the CMG dump requirements, no significant quantity of CMG dump propellant will be required in addition to the orbit maintenance propellant which is currently provided.

The system concept which has been chosen for this study is monopropellant hydrazine (N_2H_4). This is the most state-of-the-art concept available with a technology basis which spans the previous 20 years. There are dozens of operational systems in earth orbit at the present time on communication satellites. Five years with zero maintenance is a typical design life time. Failure of these systems is usually attributed to depletion of propellant supply since there is currently no capability to resupply the propellant in

these systems.

All proposed systems incorporate a blowdown pressurization system. No active pressurization system components have been included. The propellant tanks have been sized with the volume to provide a blowdown ratio of 2 to 1. The pressurant gas (gaseous nitrogen) would be contained within the propellant tank and would be separated from the liquid hydrazine by AFE-332 elastometric positive expulsion diaphragm. The use of a single propellant with a blowdown pressurization system reduces the complexity of the propulsion system by reducing the number of components to less than one-half of the components in an actively pressurized bi-propellant system. This is expected to significantly enhance the overall reliability of the system and reduce the maintenance required over the life of the station. A typical propulsion system schematic of the type system described above in shown in figure 4.3-1. The blowdown ratio of 2 to 1 will cause the propulsion system thrust level to vary by approximately the same ratio. If a minimum thrust level of 15 lbs. is required then an engine with a thrust range of 15-30 lbs. will be used. The thrusters would provide 30 lbs. of thrust when the propellant tanks are full and 15 lbs. when the propellant tanks are nearly empty.

Use of blowdown pressurization with a monopropellant has another advantage for the Space Station Propulsion System which will require frequent and numerous refuelling cycles. With the use of an elastomeric expulsion diaphragm concept, which has previously demonstrated several hundred cycles of expulsion capability, the pressurizing gas need not be lost at the end of the expulsion cycle or during the refill cycle. This means that the refuelling cycle for a blowdown monopropellant system requires that only a single fluid be resupplied as compared to three fluids for an actively pressurized bi-propellant system which must vent the pressurizing gas during the resupply operation.



ONBOARD PROPULSION SYSTEM SCHEMATIC

MONO-PROPELLANT HYDRAZINE

Two identical systems as shown above rquired with thruster clusters at each corner of solar array as shown in Figure A.

- Triple redundant thrusters
- Triple redundant propellant distribution and isolations
- No oropellant storage redundancy
- Two propellant and pressurant resupply points
- 2:1 bloudown pressurization system

Figure 4.3-1

Hydrazine monopropellant is a midrange performance propellant with a specific impulse of about 220 lbs. sec/lb. -⁴ a liquid density which is approximately that of water.

During this study, locations were selected for the various configurations and the number of thrusters and tanks required for each configuration were determined. This information in addition to the amount of propellant required is shown in table 4.3-1.

From the information shown in table 4.3-1, detailed propulsion system weight summaries were prepared for both the IOC and the growth versions of the three station concepts. This information is shown in table 4.3-2.

4.3.2 Building Block Configuration

4.3.1.1 Propulsion Modules Locations

The most significant difference between the "Building Block" and the "Delta" is due to the stabilizing effect of gravity gradient on the station attitude and the solar array gimbals which allow the solar array to maintain a solar pointing attitude independent of the station attitude. The safety considerations of state attitude control are not as stringent as for the Delta since the array can be pointed in the general direction of the sun regardless of the attitude of the rest of the station.

As a result of placement of other items on the aft end of this configuration and the method of placing solar arrays, the potential locations for propulsion modules are extremely limited. The location which has been selected is on the aft end of the upper command module. Thruster firing directions have been limited to the aft and the up and down directions. Side firing thrusters are not possible from this location because solar arrays are in close proximity. Also, in order to fire in the up direction, it will be necessary to place the

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Table 4.3-1

ON-BOARD PROPULSION

SYSTEM FEATURES	"△"		BUILDING BLOCK		"T"	
	IOC	GRTH	IOC	GRTH	IOC	GRTH
Total Impulse (#-Sec) Required/6-Month	691,000	1,110,000	691,000	1,382,000	325,000	274,000
N_2H_4 Req'd/6-Mo. (lbs) Isp 220 #Sec/#	3,140	5,050	3,140	6,280	1,480	1,250
N_2H_4 Vol Req'd $(FT)^3$	51	82	51	102	24	20
Total Tank Volume Req'd. $(FT)^3$ Pressurization Included	102	163	102	203	48	40
Number Tanks Req'd 36" DIA $16(FT)^3$ EA.	8	12	8	15	4	3
Number of Un-Mod S.S. - APU Tanks Req'd. 29" Dia $7(FT)^3$ EA.	15	24	15	29	7	6
Number of Thruster Clusters Req'd.	2	4	2	2	2	2
Number of Thrusters Req'd. Per Cluster	12	12	12	12	12	12
Total Number Thrusters Req'd.	24	48	24	24	24	24
Number of Tankage Locations Req'd.	2	4	2	2	1	1

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Table 4.3-2

ON-BOARD PROPULSION SYSTEM WEIGHT SUMMARY	"Δ"		BUILDING BLOCK		"T"	
HYDRAZINE MONO-PROPELLANT	IOC	GRTH	IOC	GRTH	IOC	GRTH
Propellant (N_2H_4) Weight	3140	5045	3140	6280	1480	1245
Propellant Tank Weight	480	720	480	900	240	180
Pressurization Weight	82	132	82	165	39	33
Distribution Syst. Weight	100	200	100	150	75	75
Thruster Cluster Weight	120	240	120	120	120	120
Instrumentation Weight	50	100	50	75	50	75
Structural/Mounting Weight	75	126	75	122	49	45
Boom Weight	-	-	200	200	-	-
Therm. Contl. & Meteriod Protection	75	150	75	150	75	75
Dry System Weight	900	1539	1100	1717	609	570
Propellant + Pressurant	3222	5177	3222	6445	1519	1278
Total System Weight	4122	6716	4322	8162	2128	1848

thrusters on short booms and cant the engines slightly outward in order to minimize impingement on other modules which are directly above and below the propulsion module. Tanks will be required on each end of the command module in order to provide enough propellant capacity for the IOC version. Additional tankage may be required in another location to provide enough capacity for the growth version of this configuration.

4.3.1.2 Thruster Redundancy

As a result of the relatively gravity gradient stable station attitude and the independently gimballed solar array the criticallity of maintaining absolute control over the station attitude at all time is not mandatory as for the Delta configuration. That is, it might be feasible to shut down a propulsion system for a short period for repairs, or a lower level of redundancy in thrusters or other components might be acceptable and could result in a substantially reduced development program and/or component costs. This may be an attractive option to be explored in future studies; however, for this study it is assumed that this configuration will require a fail operational/fail safe redundancy. This will require that three thrusters be located in each of four firing directions for each of the two thruster clusters for a vehicle total of 24 thrusters. At this time, it appears that the same thruster cluster configuration is required for the growth version.

4.3.2 Delta Configuration

4.3.2.1 Propulsion Module Location

Propulsion system module locations have been selected on the command modules located at the forward and aft ends of the Delta Space Station. One thruster cluster with 12 thrusters each will be located at each end of one command module with tankage to hold enough propellant for the IOC version of the

Delta. For the growth version of the Delta, the system will be duplicated on the command module on the opposite end of the station. This will increase the tankage volume to that which is required by the growth version of the Delta.

4.3.2.2 Thruster Redundancy

The redundancy considerations of the solar inertial orientation of the Delta and the need to continuously orient the array toward the sun with no gimbal capability, would imply that absolute control of the station would be maintained at all times to keep the station within its design thermal envelope and provide at least some minimum power output level. Should the station lose attitude control and slip into some stable, unfavorable, gravity gradient controlled attitude with the arrays not receiving adequate sunlight, some power would be lost. For this reason, the solar inertial design should have enough redundancy in the CMG and attitude control propulsion system such that momentum control and momentum dump functions not cease during the life of the station. As a minimum it appears that fail operational/fail safe redundancy in the propellant supply and thruster system is required. This necessitates three thrusters for each function identified. On the Delta IOC there would be three thrusters at each of four locations on either end of the command module for a total of 24 thrusters. The growth version would provide identical propulsion systems on each command module.

4.3.3 Big "T" Configuration

4.3.3.1 Propulsion Module Location

The propulsion module location for this configuration station has been selected for the command module on the aft end of the vehicle. As a result of extremely low propellant requirements for this configuration station no

problems are anticipated for propellant storage for either the IOC or growth version. Also, as a result of a more favorable trim attitude for the growth version less propellant may be required for growth than the IOC version. This is a significant departure from the trend of the other two configurations.

4.3.3.2 Thruster Redundancy

The significantly reduced frontal area and the gravity gradient dominated attitude coupled with an increased solar array size and an array orientation which does not require precise solar pointing has produced a unique set of minimum propulsion requirements not attainable with any of the other configurations. First, because of low frontal area very little orbit maintenance is required, and second because of a high degree of inherent attitude stability, propulsion system propellant storage volumes and resupply quantities are significantly smaller than for the other configurations. It also appears that the propulsion system may not be required to operate continuously and may actually remain inactive for long period because of the high degree of attitude stability and the ability of the solar array to provide the design power output in the torque equilibrium attitude without any active attitude control.

The implications of these reduced demands upon the propulsion system design and the ability to tolerate a lower level of system reliability, could have a significantly favorable impact upon propulsion system development cost, hardware cost, redundancy management (software cost) and resupply costs.

If it is assumed that a fail operational, fail safe capability is required for the thruster system. Two thruster clusters, one on either end of the aft command module, each with three thrusters in four firing directions will be required with a total of 12 thrusters per cluster.

4.4 Communications and Tracking Definition

The Communications and Tracking System will be designed to provide communications and tracking services between the Space Station and various space vehicles interoperating with the Space Station, as well as internal communications services. Interoperating vehicles will include the Space Shuttle Orbiter, EVA crew members, Orbiter Transfer Vehicles (OTV's), co-orbiting free flyers, Orbital Maneuvering Vehicles (OMV's), relay satellites, and space platforms.

Each link will provide a variety of services and will entail unique operational requirements. Multiple simultaneous links will also be required. Figure 4.4-1 shows the overall Space Station Communications and Tracking System for the growth phase.

The normal Space Station/ground uplink and downlink channels will operate through a relay satellite at S-band, Ku-band, millimeter wave (mm-wave), or optical frequencies. 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 S-band, Ku-band, mm-wave or optical frequencies.

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 any Communications Security (COMSEC) requirements), and will operate in a Radio Frequency Interference (RFI) environment at all times under normal and emergency operating conditions. Relay of interoperating vehicle data to/from the ground will be provided between the Space Station via a synchronous satellite.

SPACE STATION C&T SYSTEM

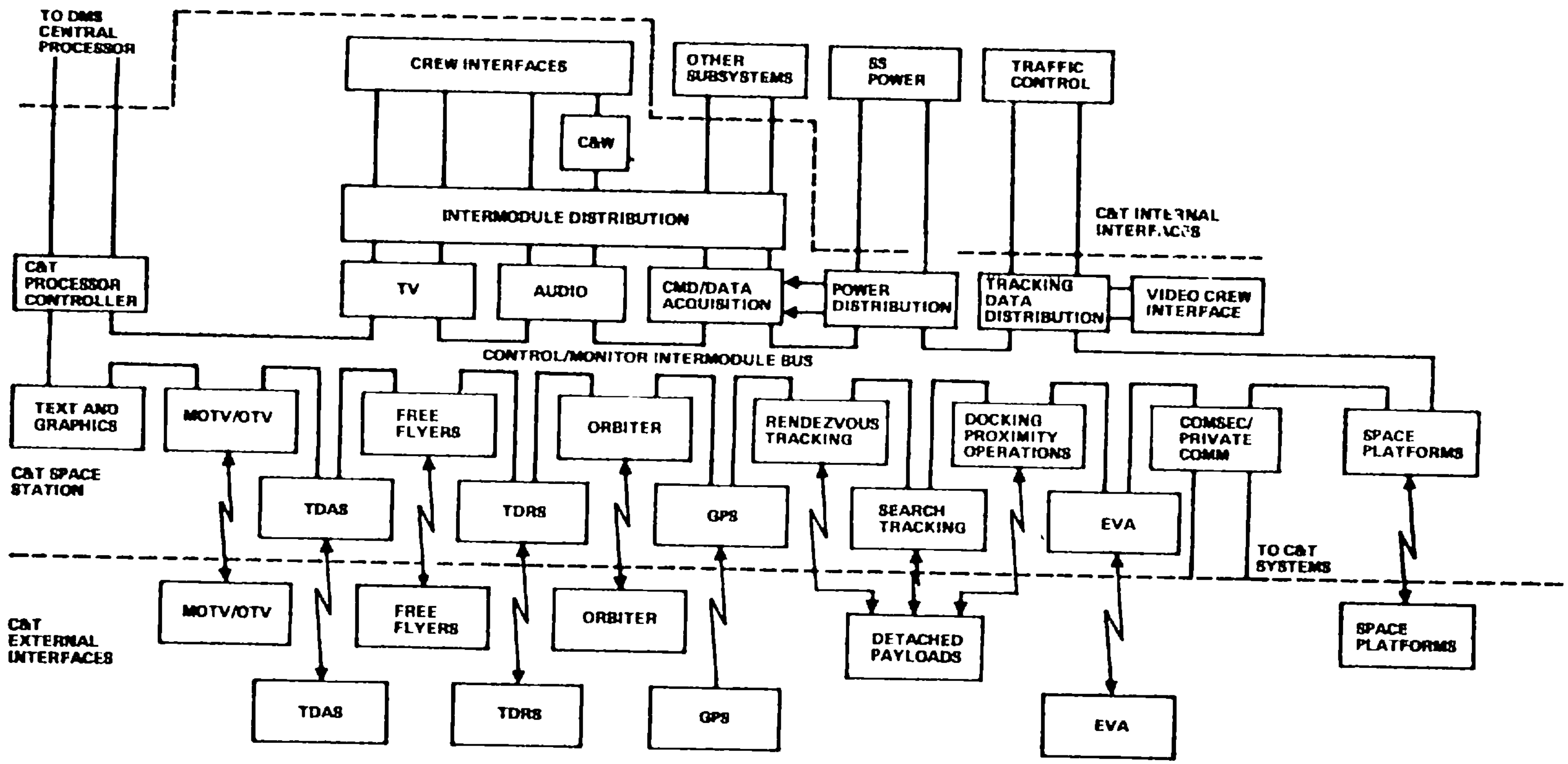

Figure 4.4-1

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The internal C&T system will provide the Space Station intermodule and intramodule C&T. Services provided by the internal C&T system include video, audio, commands, telemtry, data and C&T management/control/distribution. Each replaceable unit will have embedded processors that are software controlled. The C&T system shall be developed and implemented incrementally to support IOC and growth phases. The phases shall incorporate a logical progression in C&T system capability matching the growth in Space Station C&T requirements as the station capabilities are increased and as technology becomes available.

In translating the general Communications and Tracking System requirements into a Space Station System Concept, it has been necessary to formulate a number of basic groundrules and assumptions. These are summarized below:

a. Free-flyers (co-orbiting spacecraft) can rendezvous and dock with Space Station. At IOC, it is assumed that the Space Station will be capable of simultaneously supporting one free-flyer out to 8 km, and three free-flyers out to 2,000 km. For the growth phase, four free-flyers will operate out to 8 km, and six free-flyers out to 2,000 km.

b. Space Station/OTV line-of-sight communications are required out to 100 nm. Space Station will control and monitor OTV out to 100 nm; OTV will revert to ground control beyond 100 nm.

c. Communications between Space Station and ground facilities will be via a relay satellite. Relay satellite links will be via TDRSS initially with transition to Tracking and Data Acquisition System (TDAS) in the mid 1990's. Thus, relay satellite links will be TDRSS compatible.

d. Communications with the ground via relay satellite will be continuous within the coverage limitations of TDRSS.

e. Space Station/OMV communications are required out to 2,000 km.

f. Multiple duplex voice channels are required between Space Station and ground facilities.

g. Command uplink required (minimum duplication of Space Station panel funtions on ground).

h. Secure/private communications is required. All links will be bulk encrypted for security; all command channels will employ command authentication.

i. NTSC standard color television required between ground and Space Station (two-way), and one-way TV to Space Station from free-flyers, OMV's, OTV's, EVA's, and SSO.

j. Capability for duplex slow scan TV required between the Space Station and EVA's.

k. Four-man EVA communications system required.

l. Wideband data transmission from free-flyers greater than 20 Mbps (if required) will be direct to ground via a relay satellite.

m. Capability must be provided in the growth phase for simultaneous communication between Space Station and ten free-flyers and four EVA's.

n. Command transmission required on forward link to free-flyers.

o. Telemetry transmission required from free-flyers to Space Station.

p. TV will be digital and will utilize video compression techniques to achieve 22.0 Mbps rate and 400 kbps rate slow scanning.

q. Voice will be digital and will utilize advanced coding techniques to achieve a rate of 16 kbps.

r. Hardware will be modularized with separate operating modules for baseband, Intermediate Frequency (IF), and RF. The C&T hardware shall be distributed to within the various Space Station modules as needed to provide required C&T services and to optimize performance and maintainability.

s. The Space Station will communicate with Space Shuttle Orbiter (SSO) through the Orbiter network communications system via an RF link. Signal and RF designs will be compatible with the existing SSO S-band system where practical.

t. The Space Station shall be capable of reception and processing of GPS navigational signals.

u. The space platform will be treated as a free-flyer for communications to/from the Space Station.

A summary of the Communications and Tracking external RF link characteristics is shown in Table 4.4-1 as a function of the Space Station buildup. Table 4.4-2 is a similar summary for the intra-Space Station C&T characteristics. The performance requirements are given in Tables 4.4-3 and 4.4-4.

TABLE 4.4-1

COMMUNICATIONS AND TRACKING LINK CHARACTERISTICS

FUNCTION	PHASE	COMM. AND TRACKING LINK CHARACTERISTICS	CAPABILITIES	RATIONALE
SS RELAY SATELLITE	IOC	KU-BAND SINGLE ACCESS	25 MBPS FORWARD 300 MBPS SYMBOL RATE RETURN	BASIC CAPABILITY
	IOC	S-BAND SINGLE	96 KBS RETURN	BASIC CAPABILITY
	GROWTH	ADD W-BAND SINGLE ACCESS UPGRADE RETURN LINK	50 KBPS RETURN LINK 600 MBPS SYMBOL RATE	TDAS TRANSITION
SS-SSO	IOC	S-BAND SINGLE ACCESS	(1 SIM VEHICLE) VOICE, CMD, 128 KBS DATA RANGING	BASIC CAPABILITY
	GROWTH	ADD TV** RETURN LINK	2 SIM VEHICLES	PHASED CAPABILITY
SS-OTV	GROWTH	GROUND TT&E COMPATIBLE	(2 SIM VEHICLES) CMD, 128 KBS DATA, RANGING TV RETURN**	PHASED CAPABILITY

TABLE 4.4-1

COMMUNICATIONS AND TRACKING LINK CHARACTERISTICS
(CONTINUED)

FUNCTION	PHASE	COMM. AND TRACKING LINK CHARACTERISTICS	CAPABILITIES	RATIONALE
SS-MOTV	GROWTH	GROUND TT&E COMPATIBLE	(2 SIM VEHICLES) VOICE, CMD, 128 KBPS DATA, RANGING TV RETURN**	PHASED CAPABILITY
SS-FREE FLYER	IOC	DEDICATED SYSTEM	(4 FREE FLYERS) CMD, DATA, RANGING, TV RETURN ONE VEHICLE	BASIC CAPABILITY
	GROWTH	MULTI-ACCESS* SYSTEM	(10 FREE FLYERS MULTIPLE NETS VOICE, CMD MBR DATA AND RANGING TV RETURN** 4 SIM VEHICLES	PHASED CAPABILITY
SS-OMV	IOC	DEDICATED SYSTEM	(1 VEHICLE) CMD, DATA, RANGING TV RETURN**	BASIC CAPABILITY
	GROWTH	MULTI-ACCESS* SYSTEM	(2 VEHICLES) 16 STATIONS MULTIPLE NETS VOICE, CMD MBR DATA AND RANGING TV RETURN**	PHASED CAPABILITY

TABLE 4.4-1

COMMUNICATIONS AND TRACKING LINK CHARACTERISTICS
(CONTINUED)

FUNCTION	PHASE	COMM. AND TRACKING LINK CHARACTERISTICS	CAPABILITIES	RATIONALE
SS-EVA	IOC	SHUTTLE EVA SYSTEM	(2 EVA'S) 3-WAY FULL DUPLEX +2 LBR DATA CH.	BASIC CAPABILITY
		S-BAND FM	EMU TV	
	GROWTH	MULTI-ACCESS* SYSTEM	(4 EVA'S) 16 STATIONS MULTIPLE NETS VOICE, CMD, FWD SLOW SCAN TV, MBR DATA AND RANGING TV RETURN**	PHASED CAPABILITY
SPACE STATION POSITION	IOC	GPS OMNI-DIRECTIONAL (L-BAND)	POSITION, VELOCITY, TIME	SPACE STATION NAVIGATION
INTERIOR COMMUNI-CATIONS	IOC	ELECTRO-MAGNETIC	CREW-TO-CREW COMMUNICATIONS AND CREW INTER-FACE WITH EX-TERNAL LINKS	ACTIVITY COORDINATION AND CREW MOBILITY, SECURITY
STRUCTURE	IOC	NARROW BEAM, INFRARED, ONE-WAY	PRECISE DE-TECTION OF RANGE AND ANGLE CHANGES	STRUCTURAL INTE-GRITY AND INSTRU-MENT POINTING ACCURACIES
DOCKING	IOC	CONTINUOUS WAVE, TONE MODULATED, INFRARED	AUTOMATIC DOCKING; DATA FOR MANUAL DOCKING	DOCKING WITHOUT VEHICLE DAMAGE

* FREE FLYERS, OMV, EVA, ALL SERVED BY THIS MULTI-ACCESS COMM SUBSYSTEM
** FREE FLYERS, OMV, SSO, OTV, EVA ALL SERVED BY A DEDICATED FOUR CHANNEL TV SYSTEM

TABLE 4.4-1

COMMUNICATIONS AND TRACKING LINK CHARACTERISTICS
(CONTINUED)

FUNCTION	PHASE	COMM. AND TRACKING LINK CHARACTERISTICS	CAPABILITIES	RATIONALE
PROXIMITY OPERATIONS	IOC	NARROW BEAM KU-BAND OR INFRARED	ACCURATE RANGING OVER SHORT PATHS WITH DIRECTIONAL SENSITIVITY	PRECISE MANEUVERABILITY
SHUTTLE RENDEZVOUS	IOC	KU-BAND, PULSED, IDENTIFY-CODED, OMNI-DIRECTIONAL	RELIABLE EFFICIENT REN-DEZVOUS OF SHUTTLE WITH SS	SHUTTLE DELIVERIES DURING CONSTRUCTION PHASE
RENDEZVOUS	IOC	KU-BAND, PULSED, IDENTIFY-CODED, OMNI-DIRECTIONAL	RELIABLE, EFFICIENT REN-DEZVOUS OF SS WITH OMV AND OTHERS	SPACE STATION SERVICE AND LOGISTIC MISSIONS
MULTIPLE TRACKING	GROWTH	SKIN TRACK RADAR WITH PHASED-ARRAY ANTENNA SYSTEM	SEARCH AND TRACK MULTIPLE TARGETS TO OBTAIN POSI_ TION AND VELOCITY DATA	NUMBER OF OBJECTS TO BE TRACKED SIMU-TANEOUSLY OVER A LARGE VOLUME

TABLE 4.4-2

INTRA SPACE STATION
COMMUNICATIONS AND TRACKING CHARACTERISTICS

FUNCTION	PHASE	CHARACTERISTICS	CAPABILITIES	RATIONALE
SUBSYSTEM CONTROL/MONITOR	IOC	COMM AND TRACKING SYSTEM PROCESSOR/CONTROLLER CONNECTED TO ALL SUB-SYSTEM LRU'S VIA A FIBER OPTIC LOCAL AREA NET-WORK	WILL PROVIDE INTERFACE WITH SPACE STATION DISTRIBUTED DATA MANAGEMENT SYSTEM. WILL CONFIGURE, CON-TROL AND MONITOR ALL C&T SUBSYSTEM LRU'S	NETWORK TOPOLOGIES, ARCHITECTURE, PROTO-COL, INTERFACE COM-PLEXITY, FLEXIBILITY, RELIABILITY, NOISE IMMUNITY
TEXT AND GRAPHICS	IOC	SOLID STATE SCANNER AND HARDCOPIER ON SPACE STATION AND GROUND EXTENSIVE USE OF LARGE SCALE INTEGRATION	PROVIDES FOR TRANS-MISSION OF HARDCOPY MATERIAL BETWEEN GROUND AND SPACE STATION	ONLY VIABLE OPTION FOR PROVIDING REQUIRED CAPABILITIES
COMMAND/DATA ACQUISITION	IOC	ALL DIGITAL SYSTEM EXTENSIVELY UTILIZING LARGE SCALE INTE-GRATION, PACKETIZED DATA FOR SCIENCE/COM-MERCIAL USERS	PROVIDES COMMAND AND TELEMETRY SIGNAL PROCESSING ON SPACE STATION FOR DATA RECEIVED FROM GROUND, ON-BOARD, AND ALL INTEROPERATING VEHICLES. COMSEC AND COMMAND AUTHEN-TICATION INCLUDED	BASIC CAPABILITY
CLOSED-CIRCUIT	IOC	NTSC STANDARD COLOR TV	COMPLETE CLOSED CIRCUIT TV SYSTEM FOR SPACE STATION INCLUDING SENSORS, VIEWERS, DISTRIBU-TION, CONTROL	HOLOGRAPHIC SENSORS, 3D VIEWING STATIONS COMMAND/CONTROL OF ALL SENSORS FROM EACH VIEWING STATION SYNCHRON-OUS SYSTEM, DUAL REDUNDANCY FIBER OPTIC DISTRIBUTION TO BE PHASED

TABLE 4.4-2

INTRA SPACE STATION

COMMUNICATIONS AND TRACKING CHARACTERISTICS
(CONTINUED)

FUNCTION	PHASE	CHARACTERISTICS	CAPABILITIES	RATIONALE
AUDIO SUBSYSTEM	IOC	ALL DIGITAL AUDIO SYSTEM WITH SYNTHESIZED SPEECH, AUTOMATIC SPEECH RECOGNITION	INTERNAL VOICE COMM. IN ALL HABITABLE MODULES, AUDIO DISTRIBUTION BETWEEN ALL USERS LOCATIONS, SYNTHESIZED SPEECH, AUTOMATIC SPEECH RECOGNITION, AND PRIVATE VOICE LINE CAPABILITY AT ALL COMM. STATIONS.	NO SYSTEM LEVEL OPTIONS IDENTIFIED
INTERIOR COMMUNICATIONS	IOC	DIFFUSED INFRARED OR RADIO FREQUENCY	CREW-TO-CREW COMMUNICATIONS AND CREW INTERFACE WITH EXTERNAL COMMUNICATION LINKS	ACTIVITY COORDINATION AND CREW MOBILITY; SECURITY

TABLE 4.4-3

EXTERNAL SPACE STATION

COMMUNICATIONS AND TRACKING PERFORMANCE REQUIREMENTS

FUNCTION	PERFORMANCE REQUIREMENT	REMARKS
TELEMETRY DATA	BER $\leq 10^{-5}$	BER MEASURED AT EARLIEST BIT SYNCHRONIZER OUTPUT
COMMAND DATA	BER $\leq 10^{-5}$ COMMAND AUTHENTICATION REQUIRED	BER MEASURED AT EARLIEST BIT SYNCHRONIZER OUTPUT
WIDEBAND DATA	BER $\leq 10^{-5}$	BER MEASURED AT EARLIEST BIT SYNCHRONIZER OUTPUT
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
VOICE ANALOG	SNR $\leq$ 16 Db VOICE RECOGNITION REQUIRED VOICE INTELLIGIBILITY 90 PERCENT	SNR MEASURED AT INPUT TO THE VOICE TERMINAL EQUIPMENT. 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	
TELEVISION ANALOG	Spp/Nrms $\geq$ 35 Db NTSC STANDARD RESOLUTION = 250 LINES	MEASURED AT THE TV TERMINAL

TABLE 4.4-3

EXTERNAL SPACE STATION

COMMUNICATIONS AND TRACKING PERFORMANCE REQUIREMENTS
(CONTINUED)

FUNCTION	PERFORMANCE REQUIREMENT	REMARKS
TELEPRINTER	BER $\leq 10^{-5}$	BER MEASURED AT EARLIEST BIT SYNCHRONIZER OUTPUT
COMPUTER DATA	BER $\leq 10^{-9}$	BER MEASURED AT INPUT TO COMPUTER TERMINAL
DOPPLER TRACKING	TBD	
PSEUDO-NOISE RANGING	TBD	
TRAFFIC CONTROL SHORT RANGE	MAX RANGE - 100 Km COVERAGE - TBD ACCURACIES: ANGLE $\leq$ 10 Mrad (3σ) RANGE $\leq$ 20 M (3σ) VELOCITY $\leq$ 0.1 m/s (3σ)	
TRAFFIC CONTROL LONG RANGE	MAX RANGE - 2,000 km COVERAGE - TBD ACCURACIES: ANGLE $\leq$ 25 mrad (3σ) RANGE $\leq$ 1 Km (3σ) VELOCITY $\leq$ 1 m/s (3σ)	SPECIFIED ACCURACIES ARE FOR AUGMENTED TARGET VEHICLES
PROXIMITY OPERATIONS	MAX RANGE - 0 TO 8 Km COVERAGE - 4 pi STERADIANS ACCURACIES: ANGLE - 10 mrad (3σ) RANGE - 0.01 XRANGE (3σ) VELOCITY - 0.01 m/s (3σ) ACCELERATION - 0.01 M/S (3σ)	

TABLE 4.4-3

EXTERNAL SPACE STATION

COMMUNICATIONS AND TRACKING PERFORMANCE REQUIREMENTS
(CONTINUED)

FUNCTION	PERFORMANCE REQUIREMENT	REMARKS
DOCKING SENSORS	COVERAGE	
	MAXIMUM RANGE - 300 METERS	
	CONE ANGLE RADIUS - 20 DEG.	
	ACCURACY	
	ANGLE - 0.1 DEGREES	
	RANGE - 0.5 CENTIMETER	
	VELOCITY - 1.0 cm/sec	
	ATTITUDE - 0.5 DEGREE	
	ANGLE OUTPUT DATA	
	MAXIMUM - 20.0 DEGREES	
	RESOLUTION - 0.01 DEGREE	
	WORD SIZE - 12 BITS	
	RATE, MAX. - 5.0 DEGREES SEC.	
	RATE, MIN. - 0.05 DEGREES/SEC.	
	RATE, WORD SIZE - 8 BITS	
	RANGE OUTPUT DATA	
	MAXIMUM - 300 METERS	
	MINIMUM - 0.002 METERS	
	RATE, MAXIMUM 10 METERS/SEC.	
	RATE, MINIMUM 0.001 METER/SEC.	
	RATE, WORD SIZE - 15 BITS	
	SCAN	
	HORIZONTAL - 500 ELEMENTS	
	VERTICAL - 500 LINES	
	RECEIVER	
	LENS DIAMETER - 0.07 METER	
	MINIMUM SIGNAL - 5.0 NANOWATTS	
RENDEZVOUS SENSORS	MAX RANGE - 600 Km	
	COVERAGE - TBD	
	ACCURACIES:	
	ANGLE - 10 mrad (3σ)	
	RANGE - 10 m (3σ)	
	VELOCITY - 0.1 M/s (3σ)	

TABLE 4.4-4

TABLE 4.4-4
INTRA-SPACE STATION
COMMUNICATIONS AND TRACKING PERFORMANCE REQUIREMENTS

FUNCTION	PERFORMANCE REQUIREMENT	REMARKS
TELEMETRY DATA	BIT ERROR RATE (BER) $\leq 10^{-9}$	BER MEASURED END-TO-END OVER THE INTENDED COMMUN-ICATION PATH
COMMAND DATA	BER $\leq 10^{-9}$	BER MEASURED END-TO-END OVER THE INTENDED COMMUN-ICATIONS PATH
WIDEBAND DATA	BER $\leq 10^{-9}$	BER MEASURED END-TO-END OVER THE INTENDED COMMUN-ICATIONS 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 $\leq 10^{-7}$ AT OUTPUT OF TV DIGITAL TO ANALOG PROCESSOR Spp/Nrms $\geq$ 40 dB RESOLUTION = 350 LINES	
HIGH RESOLUTION ANALOG	Spp/Nrms $\geq$ 48 dB RESOLUTION = 700 LINES	
NTSC ANALOG	Spp/Nrms $\geq$ 40 dB RESOLUTION = 350 LINES	

TABLE 4.4-4

INTRA-SPACE STATION
COMMUNICATIONS AND TRACKING PERFORMANCE REQUIREMENTS
(CONTINUED)

FUNCTION	PERFORMANCE REQUIREMENT	REMARKS
VOICE DIGITAL	BER $\leq 10^{-5}$ VOICE INTELLIGIBILITY $\geq$ 90 PERCENT. VOICE RECOGNITION REQUIRED	WORD INTELLIGIBILITY MEASURED USING HARVARD PHONETICALLY BALANCED WORD LISTS
VOICE ANALOG	SIGNAL-TO-NOISE RATIO (SNR) $>$ 16 dB. VOICE INTELLIGIBILITY $\geq$ 90 PERCENT. VOICE RECOGNITION REQUIRED	
CONTROL/MONITORING/ STATUS DIGITAL DATA	BER $\leq 10^{-9}$	BER MEASURED END-TO-END OVER THE INTENDED COMMUNICATIONS PATH

4.5 Avionics Subsystems Definition

The Space Station Avionics has been partitioned into nine independent subsystems. These subsystems are:

- Navigation
- Guidance and Control
- Integrated Displays and Controls
- Data Management
- Power Distribution and Control
- Facilities Management
- Operations Planning and Scheduling
- Payload Operations
- Traffic Control

The rationale for defining the boundries and scope of these subsystems is discussed in more detail in JSC Internal Note #18740. Essentially the goal was to achieve a distributed data management approach where each subsystem or cluster is as autonomous and independent as possible. Interactions and dependencies between clusters or subsystems has been reduced to a minimum. Another goal of the partitioning was to identify manageable subsystems with clear boundries and interfaces that could be developed independently and that could grow and change without affecting other subsystems.

Brief descriptions of these subsystems are provided in the following sections. In addition, more comprehensive descriptions of these subsystems can be found in JSC Internal Note, #19099.

The Power Distribution and Control Subsystem will be discussed in another section of this report.

4.5.1 Avionics/Configuration Dependency

It should be noted that most of the Space Station Avionics will be located within the Command/Control Module. The various Space Station configurations do not affect and are not affected by the Avionics subsystems. Essentially, the subsystems are transparent to the various configurations.

However, there is one notable exception: the Guidance and Control Subsystem. The configuration sensitive aspects of this subsystem will be addressed in great detail in another section of this report - 3.2.3.2, "Structural Dynamics and Control."

4.5.2 Avionics Subsystems Guideline/Assumptions

In order to lay out these subsystems at this early (conceptual) stage of the Space Station Program, a number of assumptions had to be made. It should be remembered that this is a first cut or rough estimate of the Avionics subsystems characteristics. The weight, power and dimensions are rough estimates based on experience and expectation of future capabilities. These estimates will certainly change, but are provided so that rough estimates of the the overall Space Station power and weight can be made.

Certain assumptions had to be made relative to redundancy and architecture. Basically, a distributed approach was assumed with each subsystem being as independent as possible and having its own hardware. It is recognized that eventually the purity of a true distributed approach will have to be compromised and hardware will have to be shared between subsystems. In addition, fault tolerant processors will reduce hardware proliferation and still allow achievement of the necessary redundancy and reliability. In summary, the following subsystem descriptions include our best estimates at this time.

4.5.3 Navigation

4.5.3.1 Description

The navigation subsystem maintains the current position and velocity state vector and inertial attitude for the entire Space Station. The sensors to be utilized include the Global Positioning Satellite receivers (located in the tracking subsystem), and a number of star trackers located on the various modules. Additional short-term attitude data may be obtained from rate gyro packages, and/or strapdown laser gyro reference systems on some of the principal modules. Some linear accelerometer sensors may also be required, depending upon the outcome of future studies.

The conventional navigation task consists of routine propagation of precision measured state vectors from the GPS sensors, forward/backward extrapolation of state vectors as required by operational planning (i.e., rendezvous, payload launch, etc.), and providing sunset/sunrise times for solar array management. Use of star occultation data from the star trackers will be available for infrequent periods of fully autonomous operation, in which less precise state vectors may be acceptable.

Attitude determination will be maintained almost continuously by fixed-mount, non-articulated star trackers. Dual-redundant tracker assemblies consisting of three small-field, high sensitivity trackers will provide 3-axis inertial attitude at a sample rate of at least 10 Hz. Each module equipped with such an assembly will provide its unique attitude to the central attitude processor for use in relative-module control. During brief intervals between tracker sightings, as when a particular tracker assembly is blocked by the earth or sun, or by a nearby vehicle, the inertial attitude will be derived from a gyro package. Initialization of the gyros by the tracker data is expected to

yield a continuous and very precise knowledge of inertial attitude. The large star catalog required will dictate the use of a separate, large capacity memory device. Either magnetic bubble technology, or the new optical disk devices may be required. A dedicated processor will be necessary for the navigation task and another for the attitude determination system. As much as possible of the attitude software will be resident in the sensors themselves. See figure 4.5-1 for a functional flow of the navigation subsystem.

4.5.3.2 Technical Characteristics

1. State vector accuracy:

   ± 100 feet

   ± .1 ft/sec

2. Attidue sensing accuracy:

   ± 0.01 degree

   ± 0.01 deg/sec

3. Attitude output bandwidth = 10 Hz

4. Mass Properties: (per unit)

Item	Qty	Total Volume (ft^3)	Total Weight (lbs)	Total Power (Watts)
Tracker Assy	12	1	30	30
Nav. Processors	3	1	20	50
BIU's	3	1	20	50

4.5.3.3 Design Basis

Selection of GPS as a navigation source was favored because of the high precision offered. Various other navigation schemes can provide the autonomous navigation data desired, but with consider ly less accuracy. Deriving orbital elements from : occultation data as a backup system does provide a viable autonomous mode without much additional onboard equipment.

Attitude determination to an accuracy less than 0.1 degree requires star trackers. Other sensors considered included horizon sensors, sun sensors, GPS (relative position mode). The degree to which the star trackers are used to provide attitude information directly, rather than to aline the gyros, is a subject of current study. Some limited number of gyro-type sensors will likely be required to cover occasional intervals of "blinded" trackers. The quality and accuracy of rate gyro data is comparable to the derived rate data from the trackers.

Linear accelerometers, if needed, will provide feedback to the attitude control system to assist in damping vibrations of the module clusters. If the attitude rate data is sufficient, these devices will not be needed. These accelerometers may also be used during orbit re-boost maneuvers.

4.5.4 Guidance and Control (G&C)

4.5.4.1 Description

Provides appropriate sensor information; accepts information and commands from ground, crew, and other subsystems; processes appropriate software logic and issues torque or delta V commands to momentum exchange devices or reaction control propulsion devices such that Space Station attitude and orbital altitude can be controlled with appropriate response speed and accuracy. A top level list of the required inputs, processing functions, and outputs to other subsystems is shown in figure 4.5-2. A more comprehensive listing of functional and performance requirements and subsystem design trade considerations can be found in Space Station Yellow Books 3 and 5.

The basic principle of operation is that of a multi-input/multi-output controller, where desired states are compared with actual (measured and/or estimated) states, and the errors are processed in accordance with a goal of

LEVEL I - INPUT/OUTPUT FUNCTINAL FLOW

INPUTS REQUIRED FROM OTHER SUBSYSTEMS	SUBSYSTEM: NAV	OUTPUTS REQUIRED BY OTHER SUBSYSTEMS	
FACILITY MANAGEMENT - VEHICLE	MAJOR SUBSYSTEM FUNCTIONS:	SS NAV STATE VECTOR	G&C TRAFFIC CONTROL TRACK
CONFIGURATION	o MAINTAIN ATTITUDE REFERENCE		
TRACKING - GPS DATA AND TIME	o MAINTAIN NAV POSITION AND VELOCITY	RELATIVE MODULE ATTITUDE	G&C
SIGNALS			PROPULSION
GROUND SV UPDATE	o MAINTAIN RELATIVE ATTITUDES	SS NAV ACCELERATIONS DURING BURNS (X,Y,Z)	
DISPLAYS AND CONTROLS - SENSORS	o MAINTAIN DRAG AND GRAVITY MODELS		
MODING	o MAINTAIN MASTER TIMING UNIT	SENSOR AND SUBSYSTEM STATUS	FACILITY MGMT
	o MEASURE ACCELERATIONS	Cmd. Data & Acquisition	
DMS		D&C	
G&C		DMS	
		GENERAL PURPOSE POSITION, ATTITUDE & POINTING DATA TO VARIOUS USERS - FLUID SYSTEMS - COMM - POWER GENERATION - EPDC - ACTIVE THERMAL CONTROL - PROPULSION STAGE MGT. - PAYLOAD OPS	

FIGURE 4.5-1

LEVEL 1 - INPUT/OUTPUT FUNCTIONAL FLOW

INPUTS REQUIRED FROM OTHER SUBSYSTEMS	SUBSYSTEM: GUIDANCE & CONTROL	OUTPUTS REQUIRED BY OTHER SUBSYSTEMS
DISPLAYS & CONTROLS	MAJOR SUBSYSTEM FUNCTIONS:	RCS TO"QUE/FORCE CMDS
	DELTA V CMDS	o MAGNITUDE
	o GENERATE TARGETS	o ON-TIME
	o ESTIMATE ACCELERATION	o POLARITY
MODE CCNTROL	o COMPUTE FUEL OPTIMAL THRUSTING ATTITUDE	FACILITIES MANAGEMENT
	o COMPUTE THRUSTER on-OFF COMMANDS	
V (ORBIT CHNG)/INHIBIT ATTITUDE CMDS	RIGID BODY CONTROL	o HARDWARE STATUS
	o ATTITUDE HOLD	o IMPENDING MANEUVERS
	o GENERATE TORQUE CMDS	
NAVIGATION	o COMPUTE AND COMPENSATF FOR SYSTEM BIAS	DISPLAYS AND CONTROLS
STATE VECTORS (1)	o ATTITUDE MANEUVER	o SUBSYSTEM STATUS
	o GENERATE ACCELERATION PROFILE	
EPHEMERIS DATA	o LIMIT AND SHAPE ACCELERATION (AVOID EXCITING BENDING MODES)	o ALARMS
		o IMPENDING MANEUVERS
	o TRACK MANEUVER PROGRESS	
FACILITIES MGMT	o SHAPE ACCELERATION CMD (AS ABOVE)	
SS MASS PROPERTIES	o INITIATE DECELERATION	
- STRUCTURES DYNAMIC DATA	o BLEND RCS/MED ACCELERATION/DECELERATION TORQUES	
G&C HDW CONFIG CMDS		CMD DATA ACQ. & PROCESSING
	MOMENTUM MANAGEMENT	
POWER CONSTRAINTS	o STATUS MED'S	o SUBSYSTEM STATUS
POWER GENERATION	EQUALIZE STORED MOMENTUM BETWEEN MED'S	o HDW STATUS
	o ISSUE DESATURACTION CMD'S	
DMS	o DISTRIBUTE CONTROL TOQUE CMD TO AVOID INDIVIDUAL WHEEL SATURATION	POWER GENERATION
FLUIDS MGT		DMS
	FLEX BODY CONTROL	NAVIGATION
	o VIBRATION SENSING	
RCS	o ACTIVE DAMPING OF BENDING MODES	
STRUCTURES		
TRAFFIC CONTROL		

FIGURE 4.5-2

optimizing a performance criteria, to form effector commands.

4.5.4.2 Design Basis

Design requirements flow from mission requirements to basic functional requirements and finally to specific derived performance requirements. The mission requirement drivers are:

- Orbital altitude
- Low g environment
- Pointing (direction, accuracy and stability)
- Disturbance inputs (requirement to minimize influence)
  - Payload assembly
  - OTV servicing
  - Cargo transfer
  - Crew motion
  - Gravity gradient and aerodynamics
  - Docking/berthing momentum exchange
  - Flexible body dynamics
- Orbital altitude maintenance and momentum dump propulsive fluid resupply (requirement to minimize).

The tradeoff between all propulsive attitude control and use of momentum exchange devices overwhelmingly dictate the latter. Sub-trades within this group appear to favor large double-gimballed CMG's. State of the art technology will probably dictate initial use of steel rotor-ball bearing

technology (to delay development costs), phasing into composite fiber rotors with magnetic bearing suspension to improve reliability and reduce ownership costs.

Rate gyro assemblies are baselined in the equipment list because they may be required as an aid in flex body dynamics control. In the final design, they may be dispensable due to availability of rate information from other sources (star trackers, IRU's, panel articulatron and CMG gimbal drive tachometers, i.e.).

4.5.4.3 Technical Characteristics

1. Nominal flight modes:
   - o Principal axes offset from LVLH adaptively for zero momentum accumulation - under CMG control
   - o Inertial hold with one principal axis POP under CMG control
   - o Attitude maneuvering for short time period in any orientation under RCS/CMG control
2. Pointing accuracy*: $0 < 0.5 \leq 1.0$ deg.
3. Pointing stability*: $0 \leq 0.05 \leq 0.1$ deg/sec
4. Linear acceleration*: $10^{-5} \leq 10^{-4} \leq 10^{-3}$ g
5. Momentum storage**: 4,000 - 35,000 ft-lb-sec (probable bounds on requirements)
6. Orbit makeup ΔV: TBD - wide variation due to uncertain (W/C_D) A and atmospheric characteristics

* at experiment mount base

** full up station, CMG requirements are TBD at this time and are very sensitive to the configuration and the orientation mode chosen. The total full up station momentum requirements will probably fall in a range between 4,000 and 35,000 ft-lb-sec.

4.5.4.4 Mass Properties and Power

ITEM/UNITS	DIMENSIONS (INS)	PER UNIT WEIGHT (LBS)	AVE POWER (WATTS)
CMG'S/5	40 x 36 DIA CYL	650	350
PROCESSORS/4	7.5 x 10.5 x 15	20	50
BIU'S/4	7.5 x 10.5 x 15	20	50
MAGNETOMER/2	3 x 3 x 5	TBD	TBD
TORQUE BARS	TBD	TBD	TBD
GYRO ASSEMBLIES/5	1 CU FT	50	150
ACCELERATION (5) ASSEMBLIES	1 CU FT	20	80

4.5.5 Integrated Displays and Controls

4.5.5.1 Description

Provides crew visibility into all subsystems (data management, propulsion, life support, etc.) simultaneously and provides crew capability for subsystem startup, initialization, operational moding, manual proportional inputs, safing, and shutdown. The displays and controls subsystems provide the following functions: visual alphanumerics, visual graphics, discrete and analog manual controls, voice control* and annunciation*, aural tones* and klaxons*, and all lighting functions. A top level list of the required inputs, major subsystem functions, and outputs to other subsystems is shown in figure 4.5-3.

LEVEL I - INPUT/OUTPUT FUNCTIONAL FLOW

INPUTS REQUIRED FROM OTHER SUBSYSTEMS	SUBSYSTEM: INTEGRATION DISPLAYS AND CONTROLS	OUTPUTS REQUIRED BY OTHER SUBSYSTEMS
CREW - MANUAL INPUTS	MAJOR SUBSYSTEM FUNCTIONS: 1) GENERAL LIGHTING 2) INTEGRATED DISPLAY PRESENTATION 3) CREW COMMAND RECEPTION 4) VISUAL ALPHANUMERICS* 5) VISUAL GRAPHICS* 6) AURAL TONES AND KLAXONS* 7) AURAL SYNTHESIZED VOICE* 8) DISCRETE MANUAL CONTROLS 9) ANALOG MANUAL CONTROLS 10) VOICE CONTROLS 11) PRINTER/PLOTTER* 12) CREW VISIBILITY INTO ALL SUBSYSTEMS UPON COMMAND 13) SUPPORT TO SUBSYSTEM LOCAL D&C (PORTABLE TERMINAL)	VISUAL OUTPUTS TO CREW AUDIO - VOICE SYNTHESIS CMDS. AUDIO - C&W WARNING TONE CMDS.
AUDIO - PROCESSED VOICE CMDS. TV TEXT & GRAPHICS - TV FOR NUMERICS AND GRAPHICS		FACILITIES MANAGEMENT - CREW INITIATED COMMANDS TO ALL SUB-SYSTEMS THROUGH FACILITIES MGMT
FACILITIES MANAGEMENT & ALL SUBSYSTEMS - SUBSYSTEM STATUS, MEASUREMENT DATA, C&W OUT-OF-LIMIT DATA, DISPLAY FORMAT DATA	* INDICATES FUNCTIONS WHICH ARE ALSO IN OTHER SUBSYSTEMS.	FACILITIES MANAGEMENT & ALL SUBSYSTEMS - CREW REQUEST FOR SPECIFIC DATA
		USING SUBSYSTEM AS REQUIRED - PROPORTIONAL AND DISCRETE CMDS THROUGH MANUAL D&C INPUT DEVICES TEST AND GRAPHICS - DATA FOR HARDCOPY OR FOR TRANSMISSION TO GROUND

FIGURE 4.5-3

4.5.5.2 Design Basis

The current Orbiter cockpit contains many toggle switches, thumbwheels, rotary switches, and circuit breakers. There are many sets of mechanical displays and instruments, electromechanical talkbacks and annunciators for depicting subsystems operation/status, vehicle attitude, and navigation aids. Many of these devices are required only during specific phases of the mission.

Despite extensive tradeoffs to provide the best physical layout of this large number of display and control (D&C) devices, the STS cockpit design has tended to be unwieldy, inflexible, and inefficient. Utilization of this D&C configuration is laborious and time-consuming for the crew and requires extensive and expensive crew training to master. Also, the reliability of these devices, particularly the electromechanical talkbacks, has presented problems. Many paper data books are stowed onboard for reference as needed, which greatly adds to the clutter in the cockpit area and results in inefficient flight crew operations.

The integrated D&C subsystem for the Space Station should employ color CRT's and/or flat panel devices for multi-purpose displays, and devices such as liquid crystals for dedicated displays and replacement of electromechanical talkbacks. Multi-functional control devices such as programmable legend pushbutton switch keyboards, touchpanel CRT overlays, etc., should be employed to reduce the large quantity of switching devices such as is used on the Orbiter. The above devices should be designed for high reliability, lower power and weight, and less volume and rea. Flight data files could be digitized and stored for subsequent electronic display.

4.5.5.3 Technical Characteristics

1. Display size and type (i.e., Raster/Stroke) - TBD
2. Display luminance - TBD
3. Display resolution - TBD
4. Display color capability - TBD
5. Display graphics capabilities (i.e., 3-D plot) - TBD
6. Display refresh rate - TBD

*In conjunction with the Audio Subsystem

4.5.5.4 Mass Properties and Power

ITEMS/UNITS	PER UNIT DIMENSIONS (INS)	WEIGHT (LBS)	AVE POWER (WATTS)
CRT DISPLAY UNIT/25 12 C/C, 9 HAB, 2 EACH LAB	10 x 13 x 15	19	50
DISPLAY ELECTRONICS UNIT/25 12 C/C, 9 HAB, 2 EACH LAB	11 x 12 x 8	21	100
MULTI-FUNCTION KEYBOARD/4 2 C/C, 1 EACH LAB	10 x 12 x 4	10	40
FLAT PANEL DISPLAY/29 12 C/C, 9 HAB, 4 EACH LAB	8 x 5 x 1	1	20
PROCESSORS/7 3 CC, 2 HAB, 1 EACH LAB	7.5 x 10.5 x 15	20	50
BIU'S 4	7.5 x 10.5 x 15	20	50
MANUAL PROPORTIONAL DEVICE/3	3 x 8 x 16	8	5
PORTABLE TERMINAL 13 2 C/C, 9 HAB, 1 EACH LAB	18 x 12 x 2	9	20
DEDICATED CIRCUIT AND SWITCHES	———	60	0
LIGHTING HARDWARE	———		
INTERIOR	———	200	1000
EXTERIOR	———	140	2500
D&C-BRIDGE STATION ASSEMBLY/PANELS		200	
D&C EXPERIMENT STATION ASSEMBLY/PANELS		100	
D&C LOOK-IN STATION ASSEMBLY/PANELS		30	

4.5.6 Flight Data Management System (FDMS)

4.5.6.1 Description

The FDMS is the integrating system for the Space Station. It provides the mechanism for all inter-Space Station data and information exchange among subsystems, payloads, D&C, other vehicles and ground. Using a distributed avionics concept, each subsystem will be charged with responsibility for providing its own internal data processing, including its own applications software design, development, and verification. For purposes of total system integration, verification, and hardware/software commonality, the FDMS will specify a common bus interface unit (BIU) and high order language (HOL) for subsystem applications software.

The standardized BIU will interface the user (subsystem) with the network. The network will be reconfigurable and adaptable to support Space Station buildup, operational growth, and contingency control. The BIU shall conform to the International Standards Organization Open Systems Interconnection (ISO/OSI) reference model. Functionally, any device with a BIU shall be able to communicate over the data network with any other device containing a BIU. Furthermore, the BIU shall provide services such as error detection and automatic retransmission of packets, redundancy tasks, initialization, etc. It shall also perform self-tests and maintain a status of its health.

For purpose of this documentation, the subsystem processors and their BIU's have been listed with their respective subsystems. This is the result of a "bottom up" approach to definition of processing requirements and the fact that the FDMS architecture and the FDMS entity is still TBD.

In addition to the function of an intra-Space Station data and information system, the FDMS will provide a common timing reference and other TBD data

services, such as archival data storage and temporary dynamic bulk data storage and retrieval.

4.5.6.2 Design Basis

No specific design "straw man" is postulated herein; however, due to anticipation of extremely high data rate requirements, fiber optics data busses will probably be used. Also, it is anticipated that the Space Station will benefit from current DOD development of standardized airborne microprocessor computers using very large scale integration (VLSI) technology. See figure 4.5-4 for a functional flow diagram of the data management subsystem.

4.5.6.3 Technical Characteristics - TBD

4.5.6.4 Mass Properties and Power - TBD

ITEM	PER UNIT DIMENSIONS	WEIGHT (LBS)	POWER (WATTS)
FREE PROCESSING STATION (3)			
PROCESSORS (3)	7.5 x 10.5 x 15	20	50
BIU'S (3)	7.5 x 10.5 x 15	20	50
ARCHIVAL STORAGE (3)	1 CU FT	20	50
TIME REFERENCE (3)	1 CU FT	20	50

LEVEL I - INPUT/OUTPUT FUNCTIONAL FLOW

INPUTS REQUIRED FROM OTHER SUBSYSTEMS	SUBSYSTEM: DATA MANAGEMENT	OUTPUTS REQUIRED BY OTHER SUBSYSTEMS
	MAJOR SUBSYSTEM FUNCTIONS:	
FORMATTED DATA AND STRUCTURED	o INTER-SUBSYSTEM DATA ROUTING AND TRANSFER	FORMATTED DATA AND STRUCTURED
MESSAGES - INTERFACE CONTROL	o RECONCILE TRANSMISSION ERRORS	MESSAGES INTERFACE CONTROL SIGNAL
	o SUPPLY SUBSYSTEM POWER CONTROLLER ENERGIZING SIGNAL	DISCRETE CONTROL SIGNAL(S)
SUBSYSTEM DATA FROM ALL SUB-SYSTEMS	o DATA ACQUISITION, RETENTION & RETRIEVAL	SUBSYSTEM DATA
BIU INPUTS FROM ALL SUBSYSTEMS		BIU OUTPUTS TO ALL SUBSYSTEMS (WHERE APPLICABLE)
GMT TIME	o MAINTAIN AND DISTRIBUTE COMMON TIME REFERENCE SIGNAL	TIMING SIGNAL AND/OR CODE
RE[illegible]ENCE CODE		
	o ARCHIVAL DATA STORAGE	
CODED MESSAGE REQUESTS	o REPORT FUNCTIONAL STATUS	CODED MESSAGE RESPONSE
DMS		
SUBSYSTEM POWER, AND COOLING		

FIGURE 4.5-4

4.5.7 Facilities Management (FM)

4.5.7.1 Description

The FM subsystem has three important major aspects. First, it performs as a dynamic data base of the complete Space Station configuration, both physical and operational, including attached payloads, OTV/TMS/MMU, consumable inventory subsystem status, etc. The second major aspect is that of an operational software program that permits the crew and/or ground to off-load their activity into automated operations. The third major aspect is that of performing specific assigned duties, such as computing mass properties, providing integrated display formats, performing subsystem test and checkout at a level higher than self-contained subsystem specific built-in-test capability.

The basic functions of FM, as defined to date, along with its associated inputs and outputs, are shown in figure 4.5-5.

4.5.7.2 Design Basis

The subsystem, along with four other operations oriented (or management) type subsystems (Operations Planning and Scheduling, Payload Operations, Propulsion State Management, and Traffic Control) evolved from a project that attempted to identify all Space Station functions involving data processing. These functions were then processed through a set of screening criteria to logically group them into subsystems (reference JSC 18740). The manner in which these management subsystems relate to each other, and to the other subsystems, is discussed in more detail in JSC Internal Note #19099.

LEVEL I - INPUT/OUTPUT FUNCTIONAL FLOW

SUBSYSTEM: FACILITIES MANAGEMENT

INPUTS REQUIRED FROM SUBSYSTEM OTHER SUBSYSTEMS	MAJOR SUBSYSTEM FUNCTIONS:	OUTPUTS REQUIRED BY OTHER SUBSYSTEMS
CREW COMMANDS VIA D&C PLANS, INSTRUCTIONS, ETC. VIA OPERATIONS & PLANNING SUB-SYSTEMS (WITH CREW APPROVAL) GROUND INPUTS VIA UPLINK SUBSYSTEM STATUS FOR EACH SUB-	o SUBSYSTEM MGT - INITIALIZATION - STATUS/HEALTH MONITORING - SUBSYSTEM RECONFIGURATION - COMMANDS IN UNMANNED OPERATIONS - DEACTIVATION o FACILITY STATUS - VEHICLE CONFIGURATION MONITORING - MAINTAIN VEHICLE MASS PROPERTIES - EXECUTE CREW COMMANDS RECEIVED VIA D&C - EXECUTE PREPLANNED OPERATIONS, EMERGENCY PROCEDURES, ETC. RECEIVED FROM CREW VIA D&C AND THE OPERATIONS & PLANNING SUB-SYSTEMS o LOGISTIC SYSTEM o MAINTENANCE o SAFETY o RESOURCE ALLOCATION	VEHICLE, SUBSYSTEM STATUS TO CREW VIA D&C RECONFIGURATION COMMAND TO SUBSYSTEMS OPERATING COMMAND TO SUB-SYSTEMS DOWNLINK DATA & STATUS SPARES, STORES, INVENTORY STATUS TO GROUND DOWLINK MASS PROPERTIES DATA TO G&C SUBSYSTEM

Figure 4.5-5

4.5.7.3 Technical Characteristics - TBD

4.5.7.4 Mass Properties and Power

The assumption at this time is that the facilities management subsystem will reside in redundant regional processors.

ITEM/UNITS	DIMENSIONS (INS)	PER UNIT WEIGHT (LBS	AVER. POWER (WATTS)
PROCESSORS (4)	7.5 x 10.5 x 15	20	50
BIU'S (4)	7.5 x 10.5 x 15	20	50

4.5.8 Operations Planning and Scheduling (OPS PLAN)

4.5.8.1 Description

The process of integrating information from ground planners, principal investigators, data bases, and Space Station status to provide the top level authoritative direction and plans for both short term and long term Space Station activities. In other words, OPS PLAN as a subsystem entity is principally a dynamic data base that serves a purpose similar to that of the Mission Operation Plans and Flight Procedures Manuals for Shuttle but also provides unique display format and user interactive computer programs that permit ready access to, and manipulation of the data base. Some of the top level functions and interfaces of OPS PLAN are shown in figure 4.5-6.

4.5.8.2 Design Basis

No detailed design or trade-off option selections are available at this time. The evolution of P/L OPS as a subsystem concept is the same as that described under FM.

4.5.8.3 Technical Characteristics: TBD

4.5.8.4 Mass Properties and Power

The assumption at this time is that the Operations Planning and Scheduling subsystem will reside in one or more regional processors.

ITEMS/UNITS	DIMENSIONS (INS)	PER UNIT WEIGHT (LBS)	AVER. POWER (WATTS)
PROCESSORS (3)	7.5 x 10.5 x 15	20	50
BIU'S (3)	7.5 x 10.5 x 15	20	50

LEVEL I - INPUT/OUT FUNCTIONAL FLOW

INPUTS REQUIRED FROM OTHER SUBSYSTEMS	SUBSYSTEM: OPERATIONS PLANNING & SCHEDULING	OUTPUTS REQUIRED BY OTHER SUBSYSTEMS
	MAJOR SUBSYSTEM FUNCTIONS:	
CMD DATA ACQUISITION - UPLINK COMMANDS	o PERSONNEL SCHEDULING	
DATA MGT - CREW INPUTS - DATA RETRIEVAL	o INTEGRATED CREW SCHEDULING o OPERATIONS SCHEDULING - NEAR TERM (DAILY) - LONG TERM (WEEKLY, MONTHLY)	DATA MGT - DATA STORAGE - DATA FOR ROUTING & TRANSFER
DISPLAYS AND CONTROLS - CREW INPUTS	o CUSTOMER SERVICING	
MEDICAL - SICK CALL	- MANUFACTURING - PAYLOAD MONITORING - ETC.	DISPLAYS & CONTROLS - DATA FOR CREW DISPLAY
FACILITIES MGT - VEHICLE CONFIGURATION - SUBSYSTEM STATUS	o ONBOARD TRAINING	PROPULSION STAGE MANAGEMENT - PLANNED ACTIVITY & SCHEDULES
	- PROCEDURES REVIEW - SIMULATIONS	MEDICAL - CREW AVAILABILITY
PAYLOAD OPERATIONS - PAYLOAD STATUS	o PLANNING - MISSION OPERATIONS PLANS - CONTINGENCY PLANS	PAYLOAD OPERATIONS - PLANNED ACTIVITY & SCHEDULES
PROP STAGE MANAGEMENT - PROP STAGE STATUS	- EMERGENCY PLANS - SUBSYSTEM RECONFIGURATION PLANS - ETC.	FACILITIES MANAGEMENT - PLANNED ACTIVITY & SCHEDULES

FIGURE 4.5-6

4.5.9 Payload Operations (P/L OPS)

4.5.9.1 Description

Payload operations, as a subsystem entity, will consist of enabling hardware and software that provides the interface between payloads and the core facility, crew and payloads, and to some degree between payloads and the FDMS. To date, only data flow requirements have been addressed, and these are shown in figure 4.5-7. P/L OPS should have responsibility for secondary structure required for payload mounting, facility attachments for power, cooling and data, berthing facilities for maintenance, repair and storage, checkout equipment, manipulator systems, and software that would enable the ground or crew to automate routine repetitive operations. This software would also maintain status of all attached and detached payloads and provide control in a manner analogous to the way FM interfaces with the core subsystems and be the repository for data on payloads, similar to that currently being documented in the Cargo Systems Manuals for the Shuttle Orbiter.

4.5.9.2 Design Basis

No detailed design or trade-off option selections are available at this time. The evolution of P/L OPS as a subsystem concept is the same as that described under FM.

4.5.9.3. Technical Characteristics: TBD

4.5.9.4 Mass Properties and Power: TBD

The assumption at this time is that the P/L OPS subsystem will reside in one or more regional processors.

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LEVEL I - INPUT/OUTPUT FUNCTIONAL FLOW

SUBSYSTEM: PAYLOAD OPERATIONS

INPUTS REQUIRED FROM OTHER SUBSYSTEMS

NAVIGATION
- SS NAV DATA TO USERS

TRAFFIC CONTROL
- EPHEMERIDES OF EXT SAT
- SAT. STATE VECTORS
- COMMAND ACKNOWLEDGEMENT
- TRAFFIC CONTROL SENSING
- EXT. SAT NAV DATA

COMMAND DATA ACQUISITION
- UPLINK

DISPLAYS & CONTROLS
- CREW INPUTS

DATA MGT SYSTEM
- TIME REF SIGNAL
- DATA ROUTING & TRANSFER
- ARCHIVAL DATA STORAGE

PROPULSION STAGE MANAGEMENT
- MANAGEMENT
- LAUNCH PROCESSING
- CHECKOUT

FACILITIES MANAGEMENT
- FACILITY STATUS (CONFIG)
- UPLINK COMMANDS
- SUBSYSTEM STATUS
- RESOURCE ALLOCATION

OPS PLANNING & SCHEDULING
- OPS UPDATE
- EMERGENCY PLANS
- CONTINGENCY PLANS

MEDICAL

COMMAND DATA ACQUISITION
- EXTERNAL PAYLOAD DATA

MAJOR SUBSYSTEMS FUNCTIONS:

- o PAYLOAD CHECKOUT
- o PAYLOAD MAINTENANCE & REPAIR
- o PAYLOAD MANAGMENT
- o PAYLOAD DEPLOYMENT & RETRIEVAL
- o SPACE STRUCTURE CONSTRUCTION SUPPORT
- o USER DATA DISTRIBUTION MANAGEMENT

NOTE: FOR THE PURPOSES OF THIS STUDY, A PAYLOAD IS DEFINED AS A GENERIC TERM FOR CARGO BROUGHT TO THE SPACE STATION THAT WILL FULFILL A SCIENTIFIC, COMMERCIAL OR USER NEED. PAYLOADS CAN BE LOCATED INTERIOR OR EXTERIOR TO THE SPACE STATION. EXAMPLES OF PAYLOADS INCLUDE FREE FLYERS, SATELLITES, ONBOARD EXPERIMENTS, SPACE MANUFACTURING, ETC.

OUTPUTS REQUIRED BY OTHER SUBSYSTEMS

DATA MANAGEMENT
- ARCHIVAL DATA
- TIME REFERENCE

PROPULSION STAGE MANAGEMENT
- LAUNCH PROCESSING
- MANAGEMENT

FACILITIES MANAGEMENT
- SUBSYSTEM
- SAFETY

OPS PLANNING & SCHEDULING
- CUSTOMER SERVICING

CMD DATA ACQUISITION
- DOWNLINK
- SCIENTIFIC/EXP. DATA

DISPLAYS AND CONTROLS
- CREW DISPLAYS

TRACKING
- SATELLITE DATA (RANGE, RANGE RATE, AZIMUTH, EL.)

TRAFFIC CONTROL
- CMD ACKNOWLEDGEMENT

FIGURE 4.5-7

ITEMS/UNITS	DIMENSIONS (INS)	PER UNIT WEIGHT (LBS)	AVER. POWER (WATTS)
PROCESSORS (3)	7.5 x 10.5 x 15	20	50
BIU'S (3)	7.5 x 10.5 x 15	20	50

4.5.10 Traffic Control Subsystem (TCS)

4.5.10.1 Description

Accepts sensor information from Tracking Subsystem relative to vehicles being tracked and transmitted state vector information from satellites with GPS. This information is used by the Traffic Control Subsystem to produce relative state vectors of all vehicles being tracked in addition to ephemerides of those vehicles. This information is then utilized to determine desired positions meeting payload requirements and insuring collision avoidance. Delta V commands are generated and transmitted via the Communication Subsystems. The Traffic Control Subsystem has control responsibility for satellites, incoming rendezvous vehicles and outgoing OTV and TMS units within a defined control boundary tentatively defined as a 8 Km sphere. Docking vehicles will be precisely controlled with both delta V and attitude commands. Vehicles to be launched from the Space Station will be provided with a desired trajectory and ignition time by Traffic Control. Traffic Control will provide display information for monitoring purposes. Operations will be automatic but Traffic Control will accept manual commands.

The basic principle of operation is that of a multi-input/multi-output controller where desired states are compared with measured states and the errors processed to optimize a performance criteria in forming effector commands.

4.5.10.2 Design Basis

Design requirements flow from Space Station mission and satellite mission requirements to basic functional requirements, and finally to specific derived performance requirements.

Specific requirement drivers are:

1. Satellite mission requirements
   - Pointing considerations and accuracies
   - Viewing considerations and accuracies
   - Proximity considerations
2. Launch vehicle targeting requirements
3. Docking mechanism interface limitations
   - Position accuracy requirements
   - Orientation accuracy requirements
   - Velocity constraints
4. Safety considerations
   - Minimum separation distance between vehicles
   - Maximum allowable closure rates
   - Failure protection requirements and conditions
5. Tracking subsystem limitations
   - Angle sensing accuracies
   - Range measurement accuracies
6. Transmitted satellite GPS update rate limitations

This subsystem as currently conceived does not have a great deal of hardware:

Redundant processors and bus interface units. Most of the design trade-offs considered thus far have been concerned with task partitioning between the various subsystems and the data which must flow between them. Later design trades will involve the processor selection (speed, memory, size, and instruction set) and the software implementation of the control algorithms, in addition to the redundancy management schemes for the multiple processors.

4.5.10.3 Technical Characteristics

1. Operating modes (on individual vehicle/satellite basis)

   o State vector monitoring only

   o State vector monitoring and position control

   o State vector monitoring with both position and attitude control (for special vehicles only)

   o Rendezvous vehicle state vector monitoring and prediction

   o Docking vehicle monitoring and control (position and attitude)

   o Docking mechanism control to effect final latch up

2. Satellite position control accuracy: + 200 feet with satellite GPS: + TBD without satellite GPS.

3. Satellite velocity control accuracy:

   + TBD ft/sec with satellite GPS

   + TBD without satellite GPS

		PER UNIT	
ITEMS/UNITS	DIMENSIONS (INS)	WEIGHT (LBS)	AVE. POWER (WATTS
PROCESSORS/3	7.5 x 10.5 x 15	20	50
BIU'S/3	7.5 x 10.5 x 15	20	50

4.6 Structures Definition

4.6.1 Truss

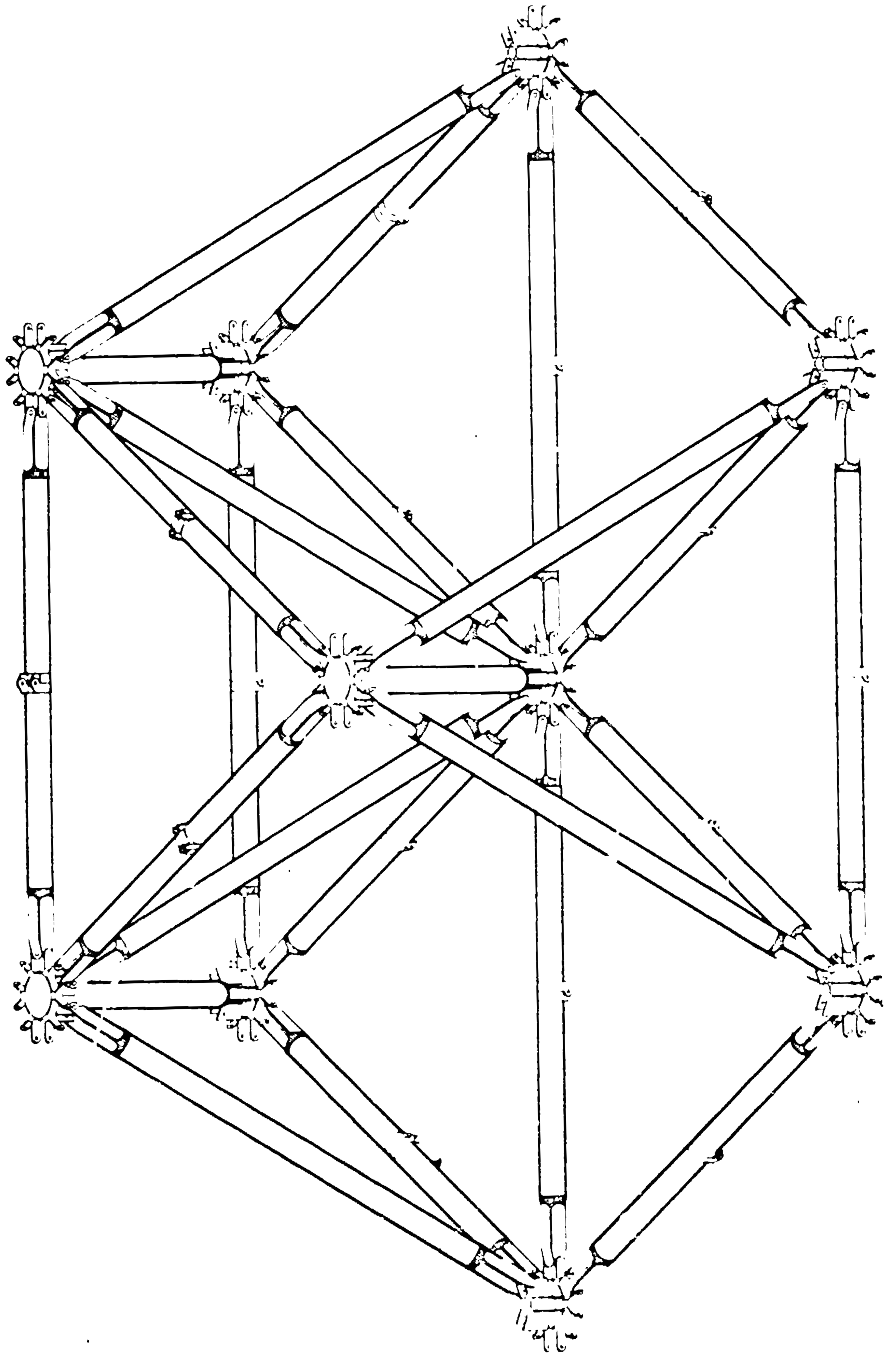
The purpose of this section is to familiarize the reader with large planar trusses as related to Space Station construction. Large planar trusses have been proposed to provide a solid foundation for construction of the Space Station. In this study, large planar trusses for the backbone of two (big "T" and Delta) of the three configurations considered. Not only will they serve to secure the various modules but they will also serve to provide a solid support structure for mounting radiators, solar arrays, plumbing and electrical runs, storage areas for payloads, OTV's and tank farms, and large work areas for the construction of large antennas and the servicing of large boosters.

Transportation of the large trusses to orbit requires that they be packaged in a collapsed state to fit within the confines of the Orbiter payload bay. This requirement can be met by either packaging each individual truss member and associated hardware, then physically constructing the truss from these components while in orbit, or the truss can be designed to be folded to fit the payload bay, then automatically deploys to its extended configuration when in orbit. For this study, an automatically deployable structure was selected to facilitate system definition. The earth assembled and packaged structure concept also has the desirable feature of element quality assurance/rejection prior to launch. In terms of constructing a large area of structure in the shortest amount of time, the deployable concept is unmatched by even the most optimistic estimates of on-orbit fabrication/assembly schemes.

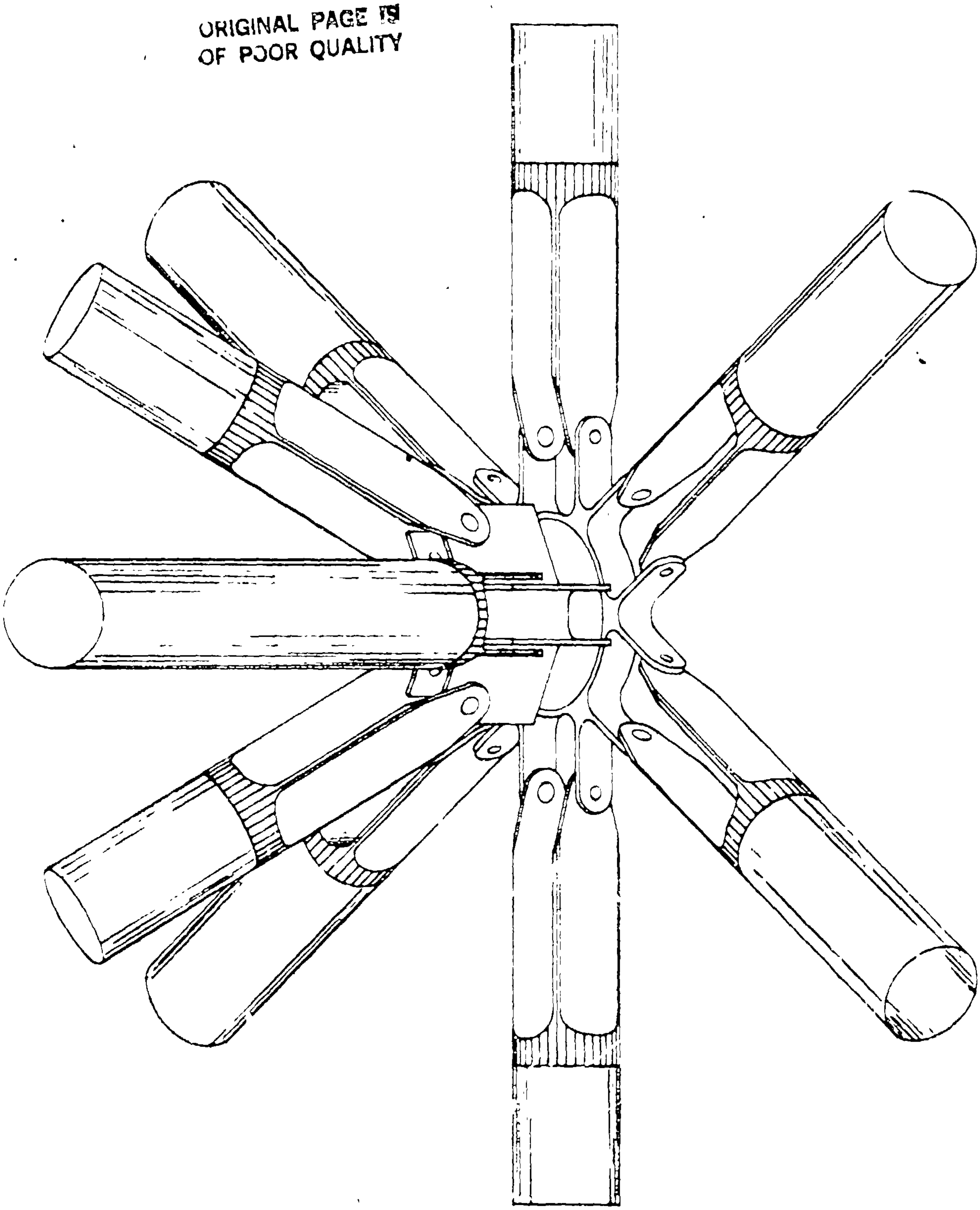
Review of various future space projects including space solar power satellites, communication antenna, and other work platform facilities indicate that the modular planar truss structure provides the necessary building block for strength and stiffness. Evolving from these studies, the tetrahedral planar truss has been shown to provide the lightest weight structure and the highest first mode natural frequency when compared to other generic trusses of equal depth and planform area. In addition, the structural arrangement of the members within the tetrahedral truss module causes this structure to be highly redundant which allows alternate load paths should any of its members become damaged.

Figure 4.6-1 shows a typical tetrahedral truss cell composed of energy contained foldable members that allow the cell to be collapsed to a compact package and automatically deployed. The foldable members are hinged at their midpoint and fold towards the center of the planar truss. That is, the foldable members on the top fold down and those on the bottom fold up. The non-foldable members simple hinge at the nodes. The final package forms a small volume unit and is ideally suited for stowage in the Orbiter payload bay. Many of these typical cells linked together form the ultra-large planar area required for the Space Station truss frame. It should be noted that the packaging density of the cell is a function of the diameter of the truss member node fittings shown in figure 4.6-2. When the truss is collapsed, each of the node fittings in each surface will lie adjacent to each other. For a typical 2.0 inch diameter truss member, the node fitting diameter will be 8.5". Using a member length of 10 feet, the packaged volume of the truss will be about .6% of the fully expanded truss volume.

Success of packaging and automatic deployment of this concept is dependent on the foldable member joint design. A typical foldable joint design is shown in



Typical Tetratruss Cell
Figure 4.6.1

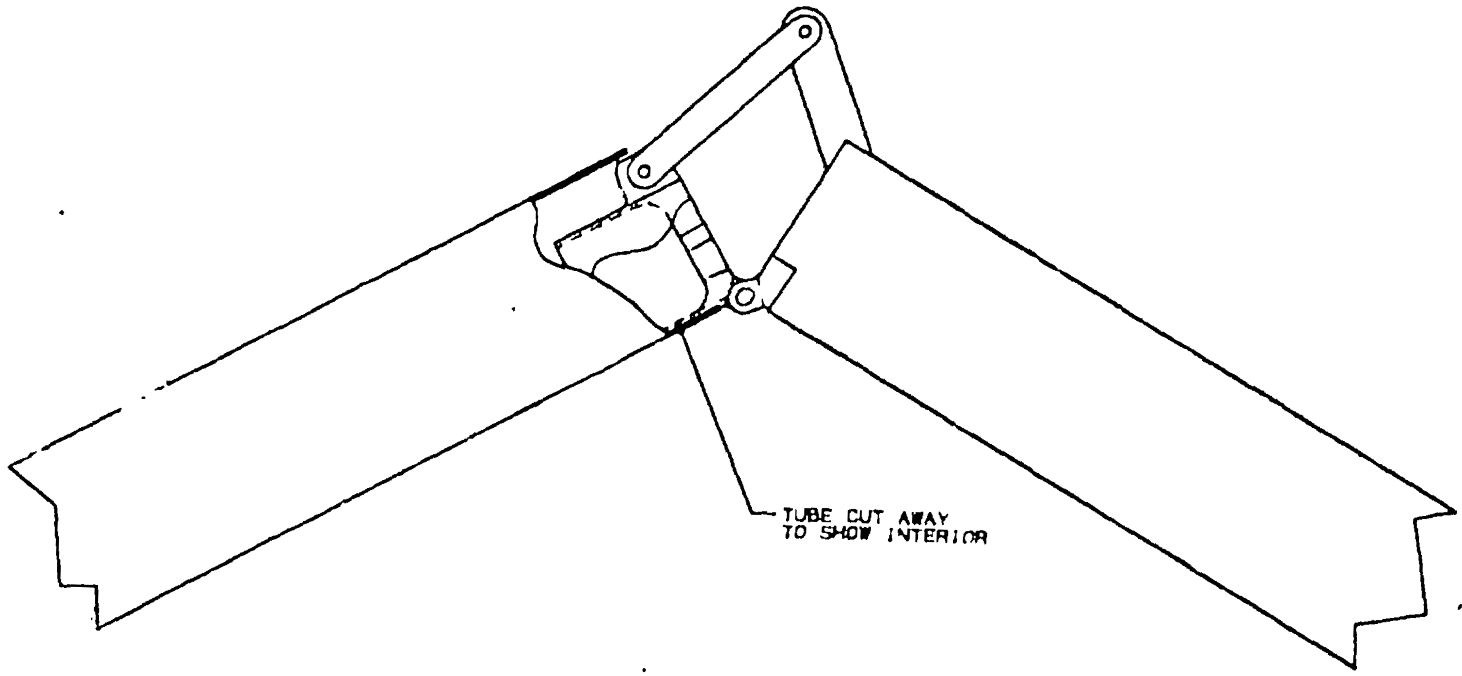


Node Fitting
Figure 4.6.2

figure 4.6-3 which contains an off-center hinge that will allow the member to fold in half for compact stowage. Also designed into this joint, is a spring energy and locking mechanism that will provide the force necessary to rotate the member from its stowed position to its fully extended position and lock it in place. The lock and hinge also provide load paths and bearing surfaces to react member tension and compression loads. Additional precompression of the joint is achieved by a cable system running through the extended truss to apply additional compression to the hinged joint thus causing any joint clearances to be taken up in one direction. This eliminates a zero stiffness "dead band" in the joint and thus eliminates a degraded overall natural frequency.

Due to the large surface area exposed to the thermal environment, it is highly desirable to manufacture the truss components from materials that have a very low, if not zero, coefficient of thermal expansion. Prior studies of the tetrahedral truss for Space Station applications considered manufacturing the tubular truss members from a balanced ply lay-up of high modulus graphite/epoxy composite. Using this material yielded a coefficient of thermal expansion of 0.5×10^{-6} inches/inch/°F. The balanced ply lay-up of the composite to produce this low coefficient of thermal expansion dictated that the composite must be composed of seven plies of graphite/epoxy which resulted in a tubular wall thickness of 0.035 inches.

Additional stress analyses performed on this tubing showed that the structural member was not critical for the imposed ultimate design loading and that the large planar truss made from this tubing had a high first mode natural frequency of 5.44 H_z for the Delta configuration. It has also been proposed that due to the large number of member cluster and end fittings required to construct the ULTRA-large tetrahedral truss, these fittings be made of molded



OVER-CENTER HINGE JOINT

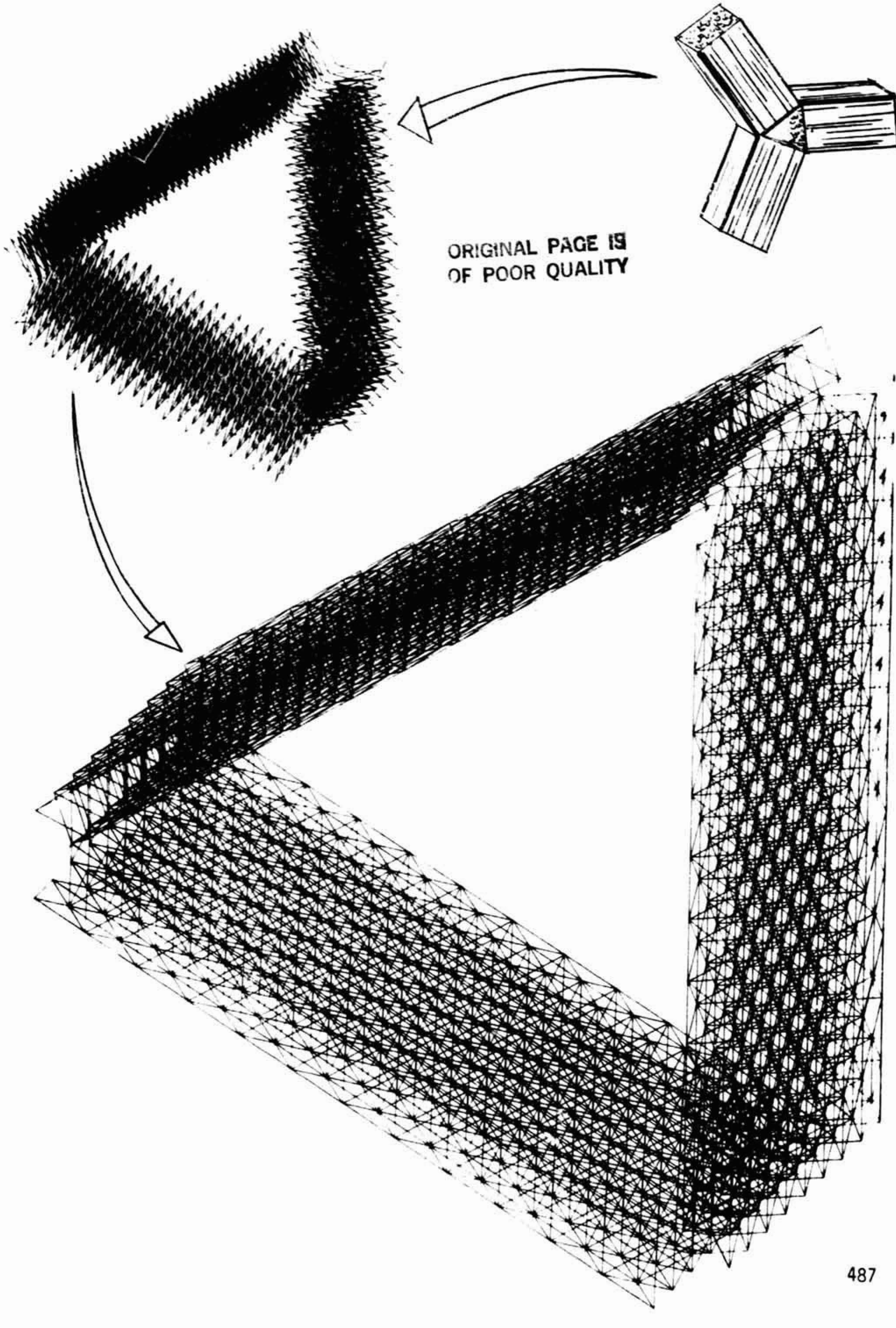
FIGURE 4.6.3

graphite/epoxy composites. Strength analyses and contacts with well known manufactures of this type of product indicate that the proposal is feasible and would be a cost reduction in the manufacture of the truss hardware. Figure 4.6-4 shows the sequential deployment of the planar trusses in the delta configuration. One outstanding feature of this deployable truss system is that it is stable during deployment. Actually, the system can be mathematically described as a single degree of freedom during deployment. Therefore, the entire station trusses can be deployed simultaneously limited only by the Orbiter payload bay space. The sides of the delta IOC can all be stowed within the Shuttle payload bay with room for other equipment. The effect of this concept is to provide the basic foundation of the station in the first launch.

4.6.2 Modules

Previous Space Station studies have shown that the structural sizing of a Shuttle launched Space Station module is generally governed by the loads experienced for the Shuttle flight environments rather that the activity that the module is to perform while attached to the station. It would appear that since the structural design environment is known, it would be economically feasible to design one module and let this design be used for any function that might be required such as habitat, laboratory, etc. Only one set of tooling would be required for manufacturing and strength and life certification would certainly be minimized.

In the attempt to arrive at a common structural entity that can be used for all proposed module configurations, it has been proposed that a common cylindrical section of 88" be selected. By adding or subtracting the cylindrical segments, the several modules proposed in this study can be constructed. The habitat module will contain five of these segments; the



Typical Deployment Sequence For DELTA Configuration

Figure 4.6.4

logistics module will contain two of these segments, etc. The segments can be designed for the most critical loads environment and mass produced; however, this implies that for some uses, the common cylindrical segment will be over designed, or heavyweight.

The following list specifies general requirements that have been identified for the Space Station module.

1. The module should be made of materials that will provide a service lift of 10 years or greater without refurbishment.

2. Module gross weight and overall dimensions to utilize maximum Shuttle capabilities.

3. Provide strength integrity to sustain a manned shirtsleeve environment of 14.7 psia.

4. Provide adequate internal attachment structure for module function configuration.

5. Provide meteoroid protection for 10 year life or better.

6. Provide docking or berthing capability to other modules and the Space Shuttle.

7. Provide dual ingress and egress capability.

8. Structural ultimate factors of safety

   — 2.0 for pressure loading

   — 1.4 for mechanical and thermal loading

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, are acceptable leakage rates. It is expected that it would be difficult to maintain an acceptable leakage

rate for a module made of conventional riveted skin-stringer construction for the long 10 year 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 aluminum plate.

Shuttle cargo bay dimensions dictate that the module payload should be a long cylindrical package to utilize the maximum cargo space available. Therefore, it follows that the basic shape of the module should be a cylindrical pressure vessel terminated on both ends with domes. The configuration already established for the cylindrical portion of the module is an all welded integrally machined skin-stringer shell. The configuration for the end domes should be selected to preclude high discontinuity stresses at the dome/cylinder interface. From an extensive study of dome end closures in reference 4.6-1, a recommended shell geometry to be used for the module is the cassinan dome. This particular dome is for a general class of shells of which the hemispherical and ellipsoidal domes are special cases. It also has impressive properties, from a structures point of view, of having negligible discontinuity stresses at the dome/cylinder interface and no compressive membrane stresses for internal pressure loading. It is also attractive because the fatigue life at the dome/cylinder interface will be minimized for internal pressure and thermal cycling loads. It is anticipated that due to its large diameter, the domes would be manufactured from gores and welded together. At this point, it would appear feasible to weld in external stiffeners along the gore interfaces to provide attachment for the required meteroid protection shield.

Structural sizing of the module should consider the largest payload to be carried by the Orbiter. The largest module proposed by this study is the habitat module which has an overall length of 528" and is comprised of five

cylindrical elements and two end domes. The cylindrical length of the module is 400" which is only 43" shorter than the detailed structural design study of the Shuttle launched station module of reference 4.6-2. The module in this reference was sized for Orbiter launch and landing loads and also contained meteoroid protection for a 10 year life. Since the two module configurations are essentially identical, the design data that was generated will be applicable to the present study. The ultimate design loads for structural arrangement of the cylindrical portion of the module were:

$$V = 128{,}026 \text{ LBS (ULT)}$$

$$V = 22.801 \times 10 \text{ IN-LBS. (ULT)}$$

$$V = 8.842 \times 10 \text{ IN-LBS. (ULT)}$$

Meteoroid protection considered that there would be a 90% probability of not having a meteoroid penetration of the module for 10 years. This criteria dictated the meteoroid bumper concept which consists of a thin aluminum shield separated from the module wall that will allow the meteoroid debris to be scattered over a larger area. Results of this criteria showed that the aluminum bumper thickness should be .040" thick and 4.0" away from the module skin. The module wall must be .060" thick to resist further penetration. It should be noted that the meteoroid criteria dictates the thickness of the module skin rather than the module internal loads requirements.

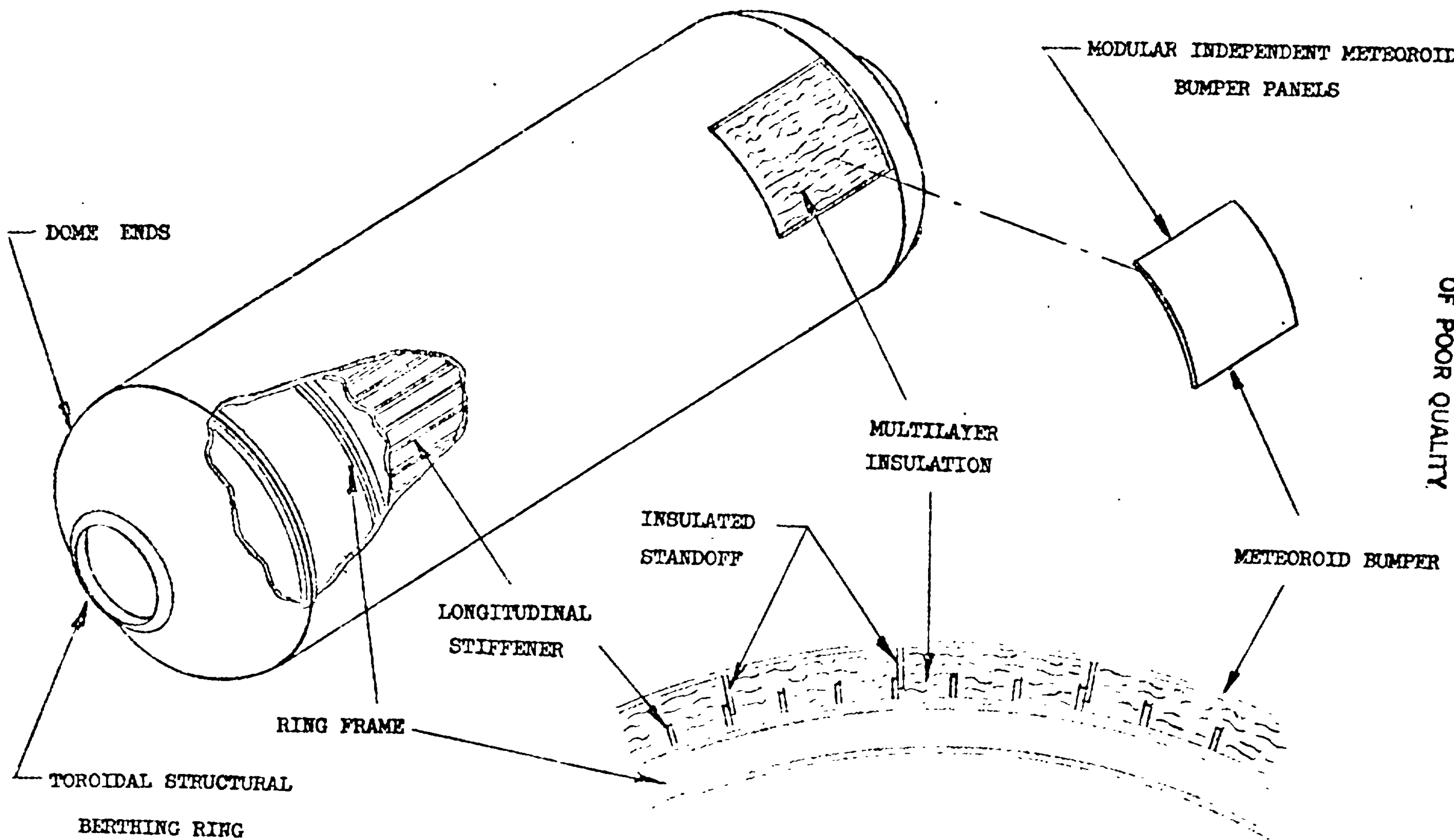
The module primary load carrying structure must be as light as possible to allow for growth in equipment weight required for any module function. This requires a high strength, low density material that can be easily and economically formed into the structural shapes needed to carry the design loads. To have a module that will serve any given function efficiently, the primary load carrying structure should be clear of the interior module space, giving abundant work and storage volume. 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 spaced around the outer circumference to resist body bending and axial loads from the Shuttle flight environment.

Placement of the stringers on the outer surface of the cylinder skin will not only produce a clean internal volume, but will also produce a more efficient structure to preclude general instability due to compressive loads. Ring frames 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.

Detailed design and analyses of the cylindrical portion of the module performed in reference 4.6-2 for the ultimate design loads given previously, resulted in the ring and external stiffener structure shown in figure 4.6-5. Total weight of the stiffened cylindrical shell for the proposed habitat module is 3828 pounds and results in 766 pounds per common 88" cylindrical segment.

An extensive weight study was made in reference 4.6-1 for various end closures used with a 14 foot cylindrical pressure vessel. The results of this study are shown in table 4.6-1 and shows that the cassini and elliptical shaped domes will be the lightest weight design primarily due to the discontinuity stress condition at the dome/cylinder interface. Also included in this analysis was the capability to support the module by the docking or berthing structure at the APEX of the dome. It is not desirable to transmit bending loads through the dome shell due to local shell instability problems; if a configuration, such as the BBC, transmits bending loads through end caps, then struts must be added to transfer these loads directly to the stiff cylindrical structure. If the modules are attached to a truss, as is the case with the



STRUCTURAL ARRANGEMENT
FIGURE 4.6.5

ENC CLOSURE WEIGHT COMPARISON FOR A 14-FOOT DIAMETER
CYLINDRICAL TANK UNDER 29.4 PSI PRESSURE
TABLE 4.6-1

GEOMETRIC SHAPE	COMPONENTS							
	DOCKING CYLINDER	TORUS	COLLAR	DOOR	CORE OR BEAMS, RINGS, ETC.	SKIN	WELD AND WELD LANDS	TOTAL
CASSINIAN DOME	128	60	34	183	160	156	37	758
ELLIPTIC DOME	138	60	34	183	160	156	37	758
CONIC DOME	128	60	34	183	215	230	37	887
SPHERICAL DOME	128	60	34	183	160	202	37	804
FLAT SANDWICH	141	---	60	183	540	192	50	1166
FLAT BEAN/SKIN/ STRINGER	188	60	34	183	545	103	47	1160

delta and the "T," then the major load path is through the truss rather than through the end caps. Therefore, the modules can be much lighter on the "T" and delta configurations than on the BBC.

4.7 Power

4.7.1 Introduction

The power system for the Space Station consists of three elements: Energy Conversion Subsystem (ECS), Energy Storage Subsystem (ESS), and Power Management and Distribution Subsystem (PMAD). Figure 4.7.1-1 illustrates the relationship between these elements and their functions.

Within each of these elements, there are several technologies available that may be utilized. For instance, in the ECS, the photovoltaic technologies are planar silicon solar arrays and gallium-arsenide (GaAs) concentrator solar arrays. Technologies in the other subsystems, ESS and PMAD, are listed in Table 4.7.1-1, along with those of the ECS.

For this study effort, one option in each subsystem was chosen for a baseline design. The other options will be the subjects of future studies. For the ECS, the planar silicon photovoltaic array utilizing the large area (5.9 x 5.9 cm) cells was selected. The alkaline Regenerative Fuel Cell (RFC) was chosen for the ESS and the high voltage, high-frequency AC system was selected for PMAD. These selections may not be optimum; however, they provide workable choices that allow for design insights.

4.7.2 Energy Conversion Subsystem

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. These cells were developed as a part of the PEP program and are currently planned for use on the SAFE experiment and the Air Force MILSTAR program. These cells were developed to reduce the total number of cells required by increasing the area of each cell from the standard 2 cm x 4 cm (8 cm) to 5.9 cm x 5.9 cm (34.81 cm^2. This reduces the total number of cells by a factor of four, thereby

ELECTRICAL POWER SYSTEM ELEMENTS

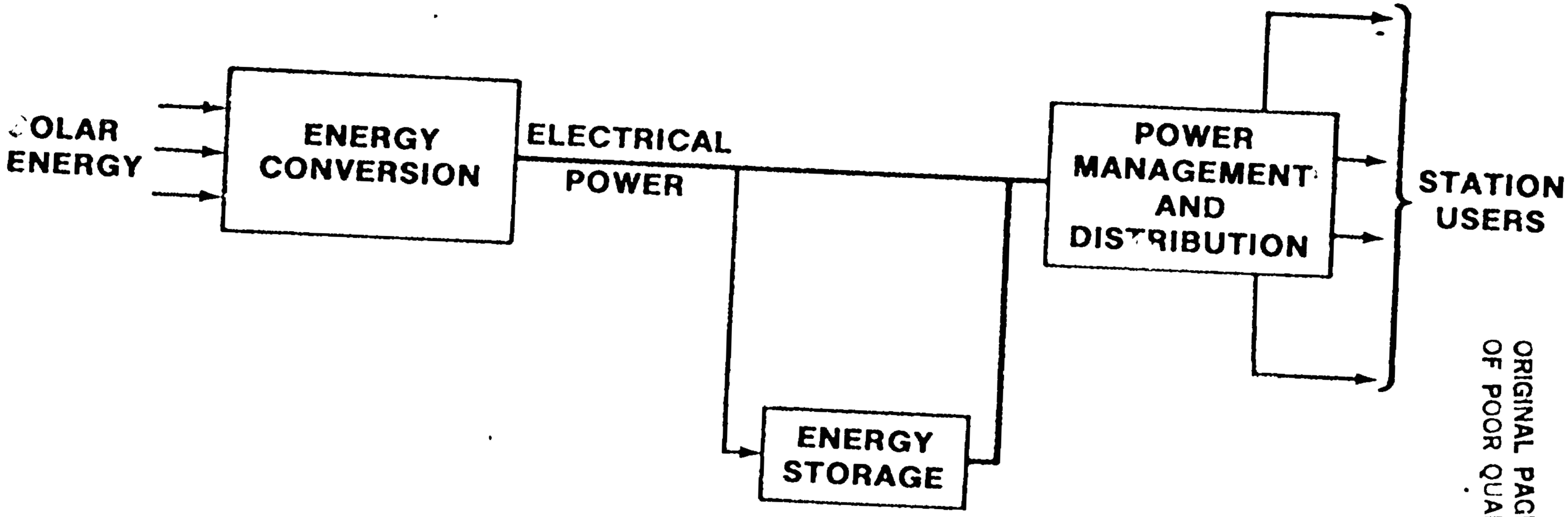


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FIGURE 4.7.1.1

ELECTRICAL POWER SYSTEM OPTIONS

ENERGY CONVERSION	ENERGY STORAGE	POWER MANAGEMENT AND DISTRIBUTION
● **PHOTOVOLTAIC**	REGENERATIVE FUEL CELLS	
• SILICON PLANAR	Ni - H_2 BATTERIES	HIGH-VOLTAGE, HIGH-FREQUENCY AC
• Ga - As CONCENTRATOR	Ni - Cd BATTERIES	
● **DYNAMIC**	FLYWHEELS	HIGH-VOLTAGE DC
• BRAYTON	PHASE-CHANGE MATERIALS	
• RANKINE		
• STIRLING		

TABLE 4.7.1.1

reducing the manufacturing and handling steps required in array fabrication. During the development program for these cells, an average efficiency of 12.9% was achieved during a production run of 2,300 cells.

The cells will be attached to a flexible kapton (or similar material) substrate instead of the more conventional rigid aluminum honeycomb. The cells will probably be welded to the attached circuitry in order to provide for the long duration cycle life that will be experience in LEO. Since this blanket will have little or no structural stiffness, a means of support will be provided. During this study, a unit weight for the solar array blanket of 0.3 lb/ft^2 was used. For the Delta and the "T" configurations, this weight was used directly and the structure was accounted for separately; however, for the building block configuration, a unit weight of 0.4 lb/ft^2 was used to include for the support structure. These values were derived on the current weight at the SAFE experiment.

A design life goal of 10 years was chosen for the solar array. This goal dictates two significant considerations:

1. Since the station will last longer than the array life goal, provisions must be made to change out solar array blankets.

2. Rroughly a 10% degradation in cell performance will occur over the 10 year life of the array.

A typical solar array wing is shown in figure 3.2.3.6.2-2. This unit would allow both deployment and retraction so that it can be considered an Orbit Replaceable Unit (ORU). The mast would be required on the building block configuration but not on the delta or "T." The storage container and tension devices will be required for all configurations. In general, the box dimensions can be designed for any desired panel size.

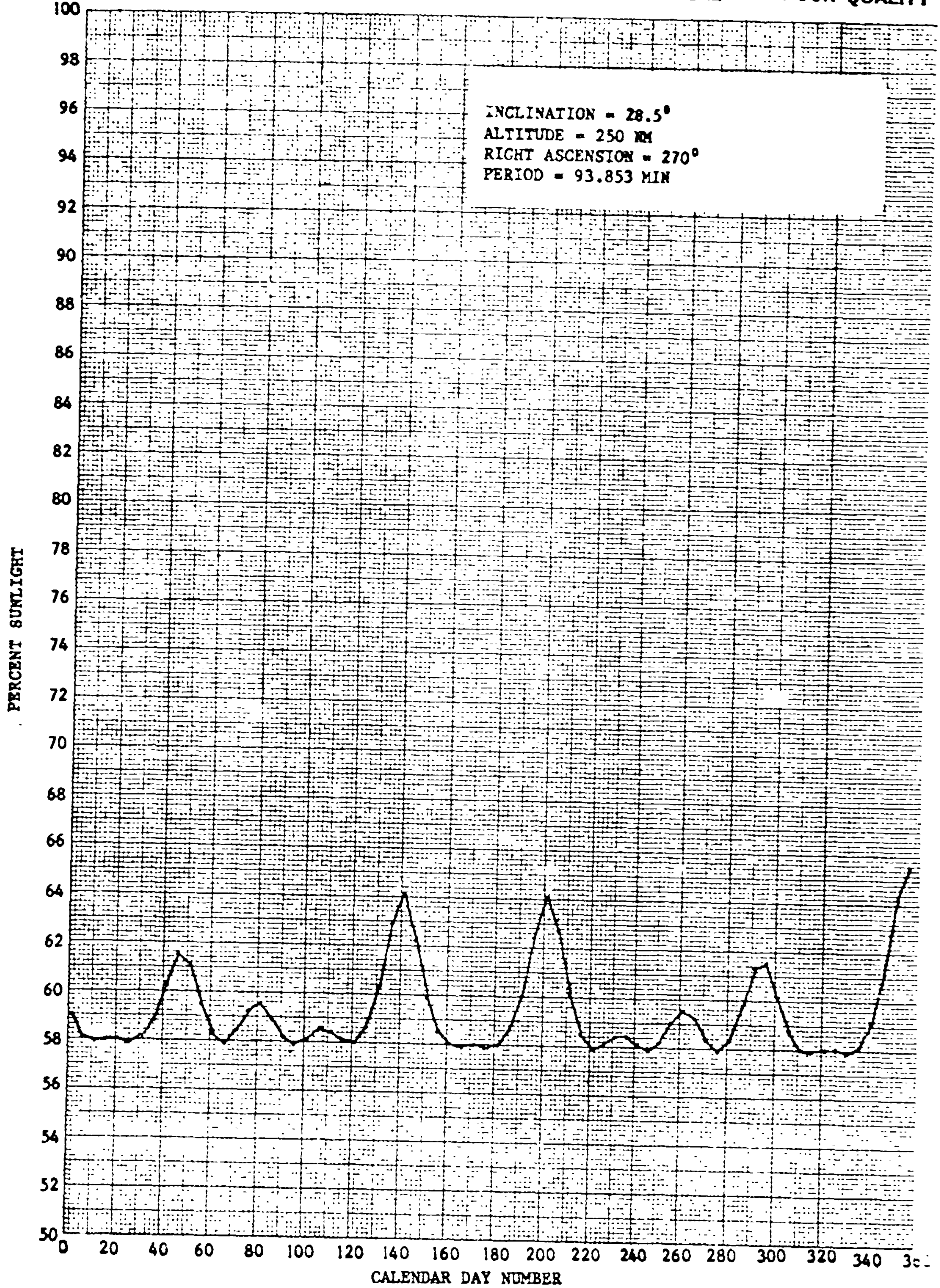
The required solar array size is a function of the power level array orientation, the storage system efficiency, storage system efficiency, distribution system efficiency, expected degradation, and the orbital parameters. It should be noted that the output of the array will change during the year due to changes in the amount of sunlight available. A typical case of this is shown in figure 4.7.2-1. For this study, this variation was not considered; however, in the future this excess capability will certainly be utilized.

The array area for each configuration is shown below:

CONFIGURATION	IOC	GROWTH
BUILDING BLOCK	16,396 FT	33,792 FT
DELTA	18,229 FT	36,458 FT
"T"	30,000 FT	60,000 FT

These array sizes are approximate but they do include allowances for system efficiencies, orientation, and end-of-life degradation.

FIGURE 4.7.2.1 PERCENT SUNLIGHT VERSUS TIME



4.7.3 Energy Storage Subsystem

The ESS will consist of a RFC with its required supporting hardware. The major items required for a RCS system are as follows:

1. Fuel Cell (F/C) - For this study, an alkaline fuel cell power plant was chosen, based on the Shuttle design. A unit that would have an output of 2.5-30 KW was selected and it would weigh about 391 lbs. and occupy about 8.2 ft^3. The fuel cell will provide the station with electrical power during the occulted part of each orbit. The fuel cell will consume hydrogen and oxygen and will produce electricity and water.

2. Electrolysis Unit (ECU) - The electrolysis unit was also assumed to be of alkaline type and is sized to be compatible with the fuel cell. The weight and volume are 400 lbs. and 12.25 ft^3 respectively. During operation, the electrolysis unit takes the water produced by the fuel cell and electrolizes it into gaseous H_2 and O_2 which are then stored as ambient temperature gases until required by the fuel cell. This process is repeated each orbit as electrical energy becomes available from the solar array.

3. Tankage - A minimum of three tanks are required, i.e. one hydrogen tank, one oxygen tank and one water tank. These tanks will have to be thermally controlled such that the water doesn't freeze and the water vapor in the H_2 and O_2 tanks doesn't condense. This probably means that the tanks and fluids will be maintained at a relatively constant 160°F during all operations.

4. Heat Exchanger - A heat exchanger will be required to interface between the F/C and ECU and the radiator or heat rejection system.

5. Power Management and Distribution (PMAD) - There is a requirement for voltage and power control in the energy storage module. A weight of 210 lbs.

was allocated for this activity and the output of the PMAD will be conditioned power ready for transmission to the station.

6. Module Structure - The above components will be contained in a single module. Many variations to the mounting arrangement are possible to meet the packaging requirements; however, the final package will have to satisfy several conditions as follows:

a. Structural support - The structure must provide for supporting the components, mounting in the Orbiter and on the station.

b. Thermal Control - An environment must be maintained to satisfy the operational requirements of the RFC and PMAD components.

c. Micrometeoroid - The enclosure must provide the desired protection.

d. Mounting of External Radiators - Provisions must be made to mount the radiators and contact heat exchanger.

A summary of the weights for a 25 KW energy storage module is shown in table 4.7.3-1. This module may well form the basis for an orbital replacement unit since it will be mounted externally, i.e., in a non-pressurized location. It should be noted that the power and energy storage quantities and sizes may be changed to meet the needs of a particular configuration.

Finally, it should be noted that a reserve requirement of two hours was provided for in the ESS design. This would provide the loss of one complete charge cycle, i.e, no output from the arrays for one light period. The energy storage resulting from this requirement is 50 KW-HR, and the reactant tanks and water tank are the only items affected.

A drawing of this concept is shown in figure 4.7.3-1.

4.7.4 Power Management and Distribution Subsystem

The PMAD subsystem transfers electrical power from the source to the user

Table 4.7.3-1. 25 KW Module Weight

ITEM		WEIGHT
FUEL CELL		391.0
ELECTROLYSIS UNIT		400.0
REACTANTS		47.2
H2 TANK		196.0
02 TANK		80.0
PLUMBING		200.0
PMAD		210.0
BOX STRUCTURE		400.0
	TOTAL	1,932.2
ARRAY		709.0
MAST		183.0
BOX & ECT.		165.0
	TOTAL	3,000.0
RADIATOR (100 FT)		100.0
DEPLOYMENT MECH		25.0
	TOTAL	125.0
	TOTAL	3,114.2
	MIS	385.8
	GRAND TOTAL	3,500#

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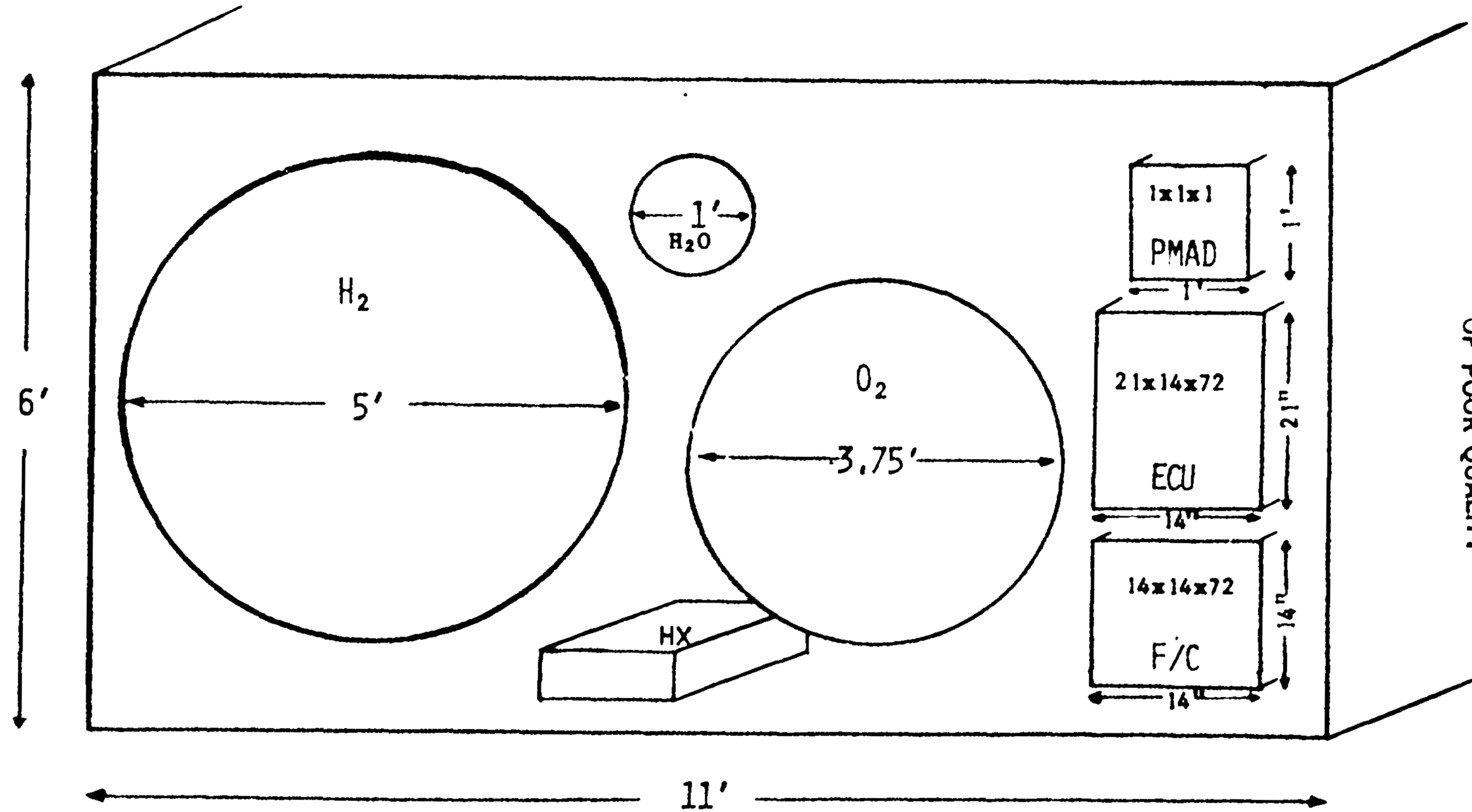

FIGURE 4.7.3.1
ESS CONCEPT

interface. It conditions and controls the power and provides protection for the power/user interface, power sources, and PMAD hardware and wiring. The subsystem manipulates sources, loads, and buses as required for most effective utilization and provides the mechanization for recharging the energy storage devices.

The accompanying diagram (Figure 4.7.4-1) depicts a generic configuration proposed for the Space Station. Although it is not intended to reflect an actual design, it does define the principles upon which the flight hardware will be designed. Modification to the layout pictured may easily be made to accommodate the final cluster configuration, whatever from it may take.

AC or DC power from the generation and storage devices is converted to 20 kHz, three-phase power for transmission via four redundant 25 kva primary loops throughout the orbiting assembly. Power generation is confined to one or two locations, depending on the final cluster configuration. Energy storage shown distributed throughout the cluster in this diagram may also be centralized in fewer locations than shown, but never less than two to insure crew safety.

Each module of the cluster has up to four similar distribution strings, depending on module criticality. Each string consists of a substation with manual overrides for selected switches, an Automated Power Management System (APMS) function interface, primary AC bus, secondary AC and DC buses, and user utility outlets. Disconnects and crossties controlled by the module 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 PMAD-dedicated data bus. It provides protection, current limiting, and

power generated by solar cells. For those conditions, the choice between AC or DC for primary power transmission could go either way as far as weight, volume, and cost are concerned. Nevertheless, future requirements for an upgraded Space Station or an advanced program will surely force the subsystem to AC. In addition, rotating machinery is a very real contender for energy storage and power generation which will cause AC to be a practical necessity. Because of this, AC was selected both for its known advantages and its vast potential.

Design of the circuitry was chosen to most advantageously utilize the desirable characteristics of AC for the enhancement of the Space Station requirements noted above. A primary goal of the PMAD design has been the development of a utility-type subsystem as nearly independent as possible from outside support from the crew or another subsystem. Such a design promotes reliability, enhances independent subsystem development, and increases autonomy.

Figure 4.7.4-2 illustrates redundancy management and bus configuration. Each module contains a total of four distribution busses. For redundancy management, the supply busses will be switchable between two distribution busses.

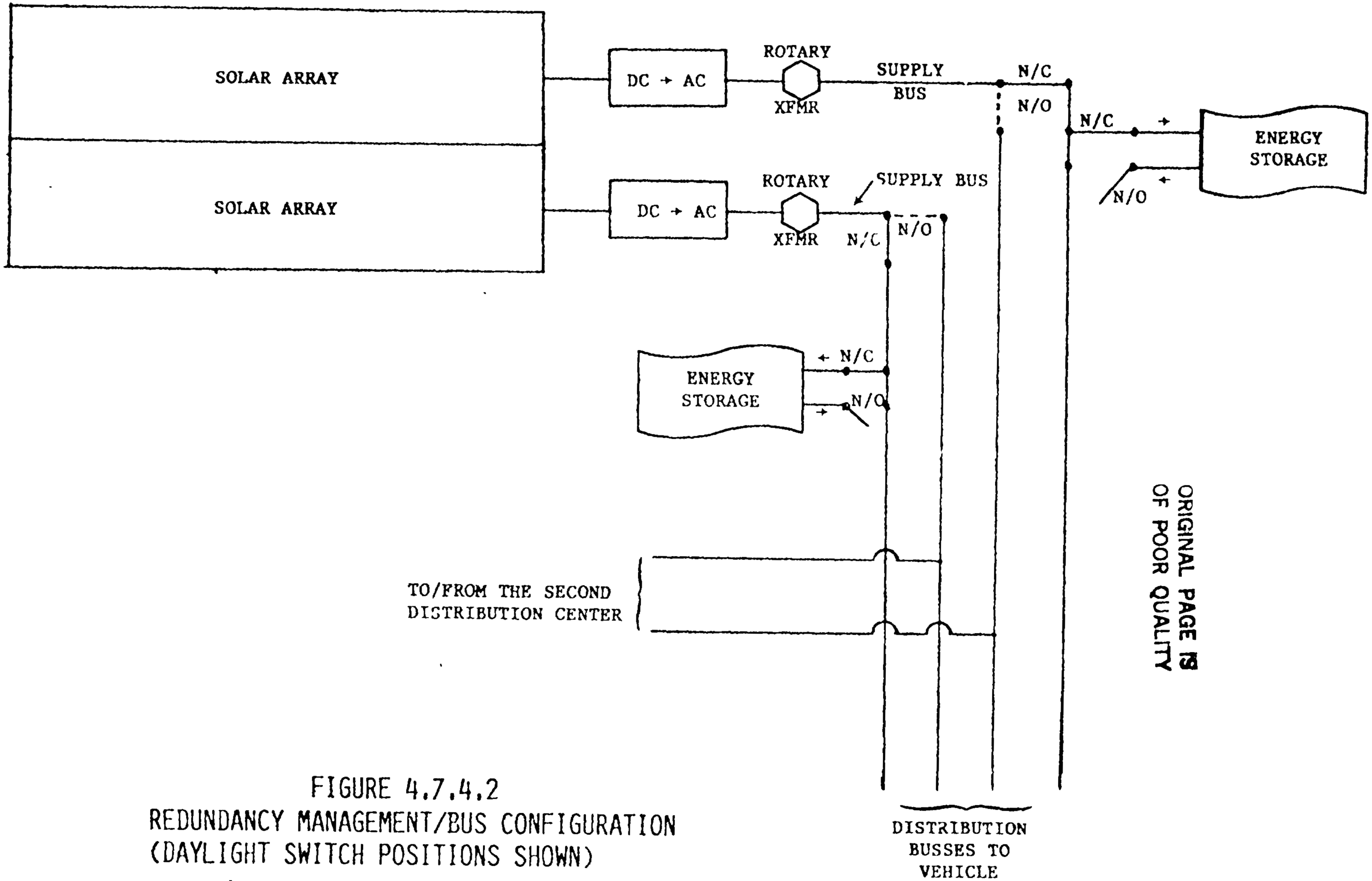

FIGURE 4.7.4.2
REDUNDANCY MANAGEMENT/BUS CONFIGURATION
(DAYLIGHT SWITCH POSITIONS SHOWN)

The technical characteristics of the PMAD subsystem are tabulated below:

Characteristics	Min	Nom	Max	Unit
Frequency (primary)	19	20	30	kHz
Voltage (primary)	350	400	450	Volts
Current (total primary)	124	144	192	Amps
Power (total)	75	100	150	kw
Efficiency (electrical)	TBD	90	TBD	Percent

NOTE: The UPC provides power to the user in whatever voltage and frequency it desires; i.e., 28V DC, 400 Hz 115V AC, etc.

4.8 EVA Support Requirements

4.8.1 Overall EVA Requirements

EVA will be a significant function aboard Space Station during all phases of the program. These guidelines were used to produce these early requirements. For the IOC phase, one two-man EVA/day will be performed as often as six days/week. For the growth phase, two shifts of two-man EVA's will be used. Each EVA will have a duration of eight hours from the beginning of airlock depress to the completion of repress. There will not be a requirement for a prebreathe period. Each crewman shall have a maneuvering unit available for use of distant excursions.

4.8.2 Airlock

The Space Station shall provide at least two airlocks for EVA support. One is the main airlock which has a two crewman capability and the other is a service airlock available for the efficient transfer of equipment.

There are two approaches to providing a main airlock. The first is to provide an airlock which serves as the stowage and servicing area for the EMU. The second is to have as small as possible an airlock which serves as a transfer

lock only. The EMU would be stowed and serviced inside the station. The first approach was used for the latter penalty estimates. The inclusion of a third airlock which acts as a backup to the main airlock may be required to eliminate manlock failure modes. The station shall provide the following services for EMU operations in the airlock:

o As much of the airlock gas as practical shall be pumped to the cabin during depress. Lost airlock gas shall be made up by the station ARS.

o The maximum time from the beginning of depress to the opening of the outer hatch shall be five minutes.

o Power shall be provided at 28V D.C. at 1500 watts maximu

o Oxygen shall be provided at no less than 1,000 psia at a rt of 12 lbs/hr for the IOC phase and at 24 lbs/hr for the growth phase.

o 5,000 BTU/HR cooling with a cooling return temperature not higher than 45°F shall be provided.

o All depress and repress functions shall be controllable from inside the station, inside the airlock or outside the airlock.

The dimensions of the main airlock must be larger than the Shuttle airlock to accommodate the larger EMU and added airlock system. For preliminary study purposes, the dimension shall be 70" diameter x 90" length.

4.8.3 EVA Hardware Configuration

4.8.3.1 EMU Configuration

The Space Station EMU will use a modular configuration . The spacesuit assembly and the basic module combined, form the EMU. These two modules contain the long life items such as batteries, regulators, prime movers, hard suit items, etc. Attached to the basic EMU are the easily removable functional modules. The modular approach facilitates easy

regeneration, repair, or replacement. There may be more than one technical approach for each module depending on the evolution of technology, EVA requirements, and limitations on expendables. For this study, it is assumed that the H_2O removal module and the CO_2 removal module will be regenerated in the station near the ARS. The power module and heat rejection module will be resupplied while attached to the basic EMU. The LCLV will require washing at least once per three EVA's.

4.8.3.2 MMU Configuration

The MMU design for Space Station will include the capability for near in stationkeeping, short distance maneuvering and long distance traversing. The MMU will be modular with the modules to be removable by an EVA crewman and transferred to the station through the service airlock. The MMU will use rate gyros, liquid propellants, and cold gas propellants for maneuvering.

4.8.4 EVA Hardware Spares and Maintenance

4.8.4.1 EMU Spares and Maintenance

The EMU will be fully maintained in the Space Station. There will be one spare, each of the H_2O removal, CO_2 removal, power, heat rejection, comm and computer controller modules. There will be two spares of all crewman specific hardware such as the arms, LTA, gloves, and the LCVG. Each module will be repairable at the component level. Repairs will be simple in nature and require minimal tools (i.e., replacement of a computer card). All repairs will be performed inside the Space Station.

4.8.4.2 MMU Spares and Maintenance

The MMU will be fully maintainable. The modules will be repaired inside the Space Station. Repairs to the main MMU structure must be made by an EVA crewman.

4.8.5 EVA Hardware Reservicing

4.8.5.1 EMU

The EMU will be partially reserviced in the airlock and partially reserviced in the Space Station. The airlock will provide power, potable water, waste water collection, O_2 and cooling to the EMU. EMU modules which can be regenerated using these utilities will be resupplied in the airlock.

Modules requiring more complex interfaces will be regenerated in the station taking full advantage of the station ARS. Chemical modules such as the CO_2 and H_2O removal modules shall be regenerated in the station.

To maximize EVA crewman productivity, resupply shall be as automatic as practical. The lowest level of crewman interface desired for routine reservicing is removal and replacement of modules. The LCVG shall be designed to minimize spacesuit cleaning. The LCVG will require routine washing.

To maximize Space Station crewman productivity, the EMU performance shall be automatically verifiable.

Water and nutrition shall be provided for an EVA crewman while EVA. There shall be food and liquid container cleaning and refilling equipment in the station.

4.8.6 Power Tools

EVA crewman productivity shall be enhanced through the use of powered tools. These tools will be recharged inside or outside the airlock. Tools shall be repaired in the station at the component level.

4.8.7 Communication, Data Transmission, and TV

One communication net will be required for every two EVA crewmen. At least one common data link will be provided for the EMU to station transmission. Full page data transmission from the station to each EMU will also be provided.

4.8.8 Translation Devices, Restraints, and Lights

The placement of translation aids, stationkeeping restraints, and flood lights shall depend on the station design and the EVA activity required. The penalties for these were left TBD.

4.8.9 Weight, Volume/Power Penalties

Figure 4.8-3 presents the best estimates of penalties for Space Station technology EVA hardware. Table 4.8-1 and Table 4.8-2 present a power usage breakdown and a total power usage profile for Space Station.

TABLE 4.8-1
EVA WEIGHTS/VOLUME/POWER PENALTIES (IOC PHASE) TWO CREWMEN

SYSTEM	VOLUME (FT)	WEIGHT (LBS)	POWER (WATT)	TOTAL ENERGY (WATT HRS/DAY)
AIRLOCK				
- MAIN	200	940	1,500	1,300
- SERVICE	8	TBD	25	10
- BACKUP	90	400	25	0
EMU (2)*	34	960	340	510
EMU RESERVICE STATIONS				
- IN AIRLOCK	53	465	520	6,280
- IN STATION	10	216	317	4,280
EMU SPARES	12	300	1,500**	12,370**
MANEUVERING UNIT (2)	62	800	----	----
MANEUVERING UNIT SERVICE STATION (2)	260	450	210	3,360
MANEUVERING UNIT REPAIR BAG (IN STATION)	10	200	150	NOT REGULAR
MANEUVERING UNIT SPARES	6	100	----	----
EVA EQUIPMENT				
- STORAGE	25	200	200	1,600
- EQUIPMENT	---	400	----	----
- SPACE STATION SPECIFIC EQUIP. STORAGE	TBD	TBD	TBD	TBD
- SPACE STATION SPECIFIC EQUIP.	TBD	TBD	TBD	TBD
- REPAIR IN STATION	3	70	----	----
- TRANS AIDS/ RESTRAINTS/ LIGHTS	TBD	TBD	TBD	TBD

TABLE 4.8-2
TYPICAL EVA DAY

TIME PERIOD	OPERATIONAL HARDWARE (WATTS)	POWER (WATTS)	TOTAL POWER
0-1 EVA PRES	AIRLOCK LIGHTS MMU RECHARGE EMU POWER MOD	100 210 100	40[illegible]
1.00-1.75 IVA OPS	AIRLOCK LIGHTS EMU IV OPS	100 340	440
1.75-2.00 DECOMPRESSION	AIRLOCK LIGHTS AIRLOCK COMPRESSOR	100 1,400	1,500
2.00-9.00 EVA	AIRLOCK LIGHTS	100	100
9.00-10.00 EVA	AIRLOCK LIGHTS MMU RESERVICE MMU CHECKOUT	100 210 100	410
10.00-10.75 POST EVA OPS	AIRLOCK LIGHTS MMU RESERVICE EMU IV OPS EVA TOOLS & EQUIP.	100 210 340 200	850
10.75-11.00	AIRLOCK LIGHTS EMU RESERVICE EVA EQUIP. & TOOLS MMU RESERVICE	100 520 120 210	950
11.00-24.00	EMU RESERVICE (ALK) EVA EQUIP. & TOOLS MMU RESERVICE EMU RESERVICE (STATION) SUIT DRYING	520 100 210 320 150	1,300*

* THIS HAS CHANGED DUE TO SUIT DRYING NEEDS.

POWER dt = 20 KW HRS

NOTE: DOES NOT INCLUDE PENALTIES FOR ARS GENERATION OF WASTE WATER AND CO or 0 GENERATION.

4.9 Crew Accommodations

4.9.1 Habitability Module Architectural Layout

The Habitability Module (HM) for IOC is a single cylinder 440 inches long plus end cones and adapters. It has a single heads-up orientation throughout the entire module. Half of the module contains the private compartments of the eight individual crewmembers. A pictorial representation of this module is presented in figure 4.9-1.

Beginning at the end of the HM adjacent to the command and control module, one enters the galley/wardroom. One the left is the galley which includes the following items:

- Microwave oven
- Convection/conduction oven
- Refrigerator
- Freezer
- Dishwasher
- Trash disposal/compactor
- Hot and cold water dispensers
- Bulk drink dispenser
- Hand washer
- Retractable working shelf
- Storage lockers

On the right are two foldable tables. Each if 48" long and 24" wide. The ends are 22" apart and the long sides of the table are parallel to the length of the HM. This provides a simultaneous eating facility for eight crewmembers. In the HM wall behind the tables are two windows mounted to provide a broad view of the earth. Using a pull down screen mounted in the end cone and an overhead projector, the crew may enjoy TV viewing together.

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TOP VIEW HABMOD #1

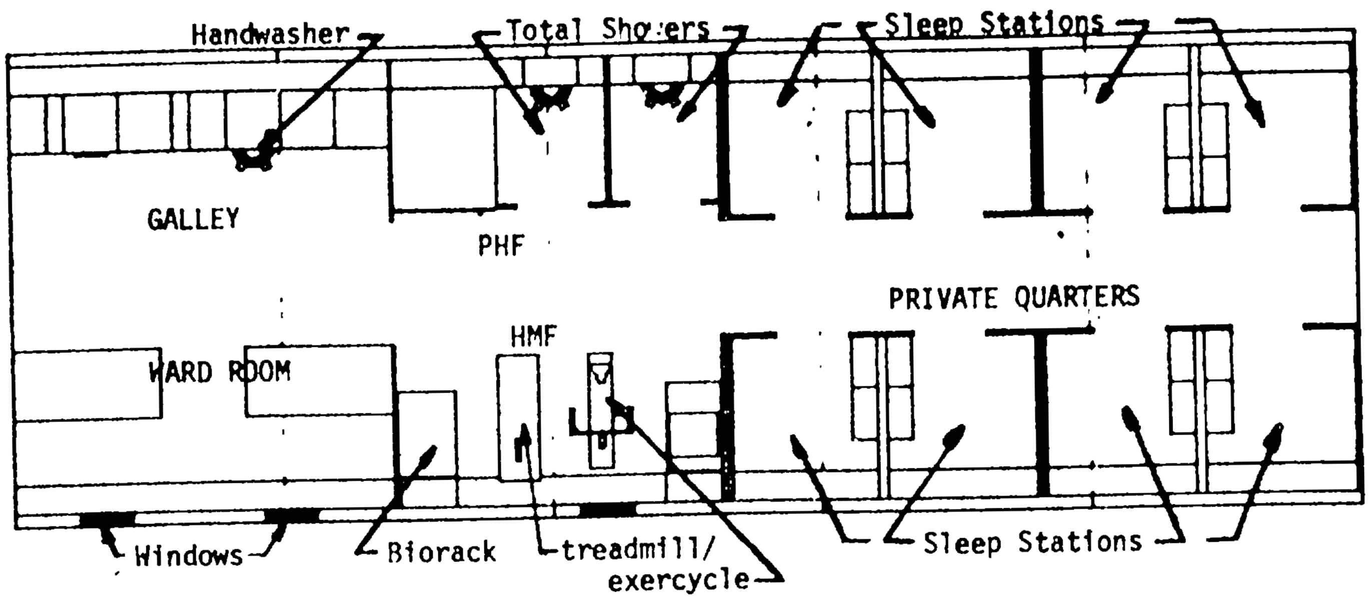

Figure 4.9-1

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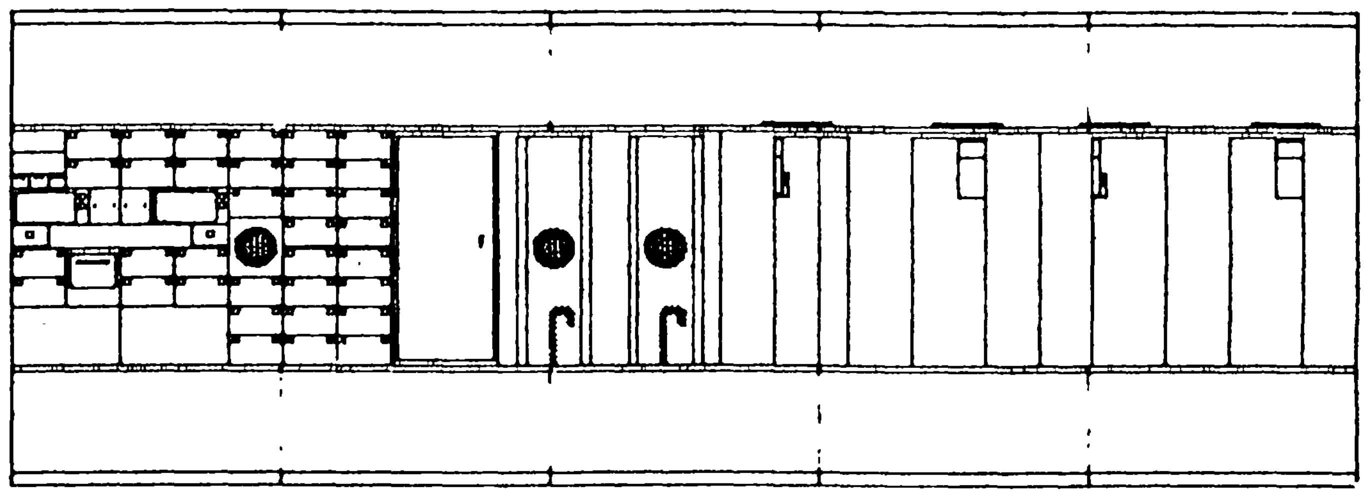
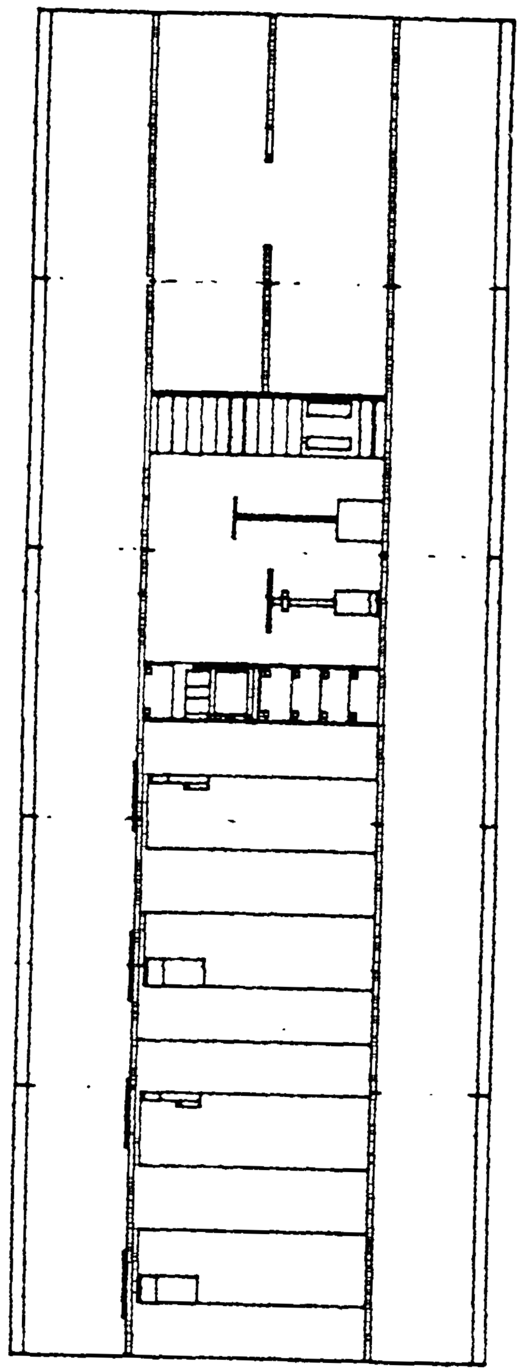
LEFT SIDE VIEW HABMOD #1



MOD012

Figure 4.9-1a

RIGHT SIDE VIEW HABMOD #1



MOD012

Figure 4.9-1b

On the left side of the next section is the Personal Hygiene Facility (PHF). This consists of three cubicles with latching doors for privacy and each is large enough for convenient doffing and donning of clothing as is necessary. The first cubicle contains the commode. The next two cubicles are identical, each containing a combined unisex urinal, a hand washer, and a full body shower.

On the opposite side of this compartment are the Health Maintenance Facilities (HMF) consisting of the biomedical instrumentation rack, the treadmill, the exercycle, storage cabinets, and a look-in station. The latter is required for monitoring the station status, activity scheduling, etc. On the outer wall is a window and a TV screen for viewing during exercise periods.

The rest of the HM contains eight private crew quarters, four per side, separated by a hallway 42" wide. This allows standard size equipment through the HM. The floor-to-ceiling height is 84" throughout the HM and other modules. The volumes below the floors and above the ceilings will be used for fans, pumps, motors, electronics and power modules, storage tanks, cable runs, duct runs, and miscellaneous stowage. These compartments are readily accessible through removeable panels and lids in the floors and ceilings of all modules in the Space Station.

Each private compartment contains a sleep restraint, a video/computer terminal for work and entertainment, an audio system with controls, a bulletin board and desk combination, a ceiling light for room illumination, an adjustable reading light, and storage lockers. There is approximately 100 ft^3 of free space in each private compartment. The doors, walls, floors, and ceiling are acoustically insulated. Double walls are located between compartments and between the compartments and the health HMF/PHF section of the There is a

"breakaway" panel between each compartment allowing emergency egress from one compartment to another.

4.9.2 Specifications for the Habitability Module Number 1

4.9.2.1 General Guidelines

- o Private quarters for eight crewmembers
- o Floor to ceiling height must accommodate a 95 percentile male and all equipment should be convenient for use by persons in the five percentile female to the 95 percentile male ranges.
- o All facilities and equipment should be designed for ease of cleaning and maintenance.
- o A single heads-up orientation for each module.
- o Emergency supplies and equipment for eight crewmembers.
- o Access to all equipment and spaces behind panels, lockers, and equipment.
- o Closeout panels and moldings in appropriate places to prevent small items from lodging behind equipment.
- o Logistics module will be used as a store room.
- o Resupply from logistics module every seven days for food.
- o Resupply from logistics module every 30 days for clothing, etc.
- o Storage volumes based on current Shuttle usage and Skylab experiences.
- o All equipment must be removable and/or relocatable for configuration changes and repairs.

4.9.2.2 Private Quarters

- o The total free volume of each compartment should be at least 100 ft^3. This minimum allows for emergency egress from the sleep restraint, clothing changes, and personal entertainment.

o Designed for single occupancy and containing a video/computer terminal, audio system and controls, bulletin board, a desk area, and temperature/ventilation controls. The desk and video/computer terminal should be conveniently accessible when the crewmember is in the sleep restraint.

o Eight to 10 ft^3 of storage using standard lockers as are used throughout the Space Station.

o Acoustic insulation of doors, floors, ceiling, and walls.

o Double walled panels for acoustic and vibration insulation between compartments and between the compartments and HMF/PHF section of the HM.

o Access to a common hallway which is no less than 42" wide and 80" high. Alternate access should be available through breakway panels between compartments.

o The "Gatling Gun" or cylindrical arrangement of the crew compartments was considered, but it not recommended for the following reasons:

1. It deviates from the standard design that is used throughout the rest of the Space Station.

2. A single section of the HM 7 ft. long does not provide for enough free volumes for the eight individual crewmembers in their respective compartments.

3. It does not allow enough volume for duct work and cabling.

4. Not easily reconfigured.

o A handwasher and unisex urinal for each private compartment were considered, but are not recommended for the following reasons:

1. Potential odor problems with the urinal.

2. Potential water hazards.

3. Elaborate plumbing.

4. The close proximity of .aa PHF in the proposed HM layout.

5. Uneconomical use of compartment space.

4.9.2.3 Wardroom Area

o Table facilities for group dining by eight crewmembers.

o Window(s) for earth viewing by the crew.

o Group TV viewing capability.

4.9.2.4 Personal Hygiene Facility

o Unisex urinal

o Handwashing facilities

o Privacy facilities for full body showering with free volumes for doffing and donning clothing.

o Commode

o A commode in the logistics module was considered but is not recommended because of:

1. Anticipation that the present Orbiter commode will evolve into a system compatible with long-term use.

2. Unnecessary cargo to be flown in each logistics module.

3. A second commode is conveniently available in the Command and Control Module.

4.9.2.5 Health Maintenance Facility

o Should be placed to cause the least disturbance to the sleep area.

o The equipment should include an exercycle, a treadmill, a biomedical rack for recording and observation, and storage for support equipment.

o An observation window and/or TV monitor for entertainment and diversion during exercise periods.

o All HMF equipment and associated stowage should be designed for easy removal to the Life Sciences module during the buildup phase.

o The volume vacated by the movement of the HMF will become an expanded recreational area or used for other functions as needed.

4.9.2.6 Look-in Station

o Equipment required: portable terminal, video monitor, display electronics unit, flat panel display, processor, and bus interference unit.

4.9.2.7 Galley

o Equipment required: refrigerator, freezer, microwave oven, convection/conduction oven, dishwasher, trash disposal/compactor, hot and cold water dispenser, bulk drink dispenser, retractable work surface, hand washer, and storage lockers for food, food preparation, housekeeping, loose equipment, and entertainment and emergency supplies and equipment.

o Total storage volumes 70 ft^3.

o Housekeeping equipment volume 8 ft^3.

o Conveniently cleaned.

4.9.3. Second Habitability Module Layout

The second habitability module, HM2, will be needed for the Growth Configuration. It is laid out in the same general plan of the first HM. Like HM1, HM2 is a cylinder 440" long plus end cones and adapters.

Entering HM2 from the galley/wardroom end, one sees the galley on the left wall. A pictorial representation of this module is presented in figure 4.9-2. It is approximately half the size of the galley in HM1 and includes the following items:

- Microwave oven
- Refrigerator

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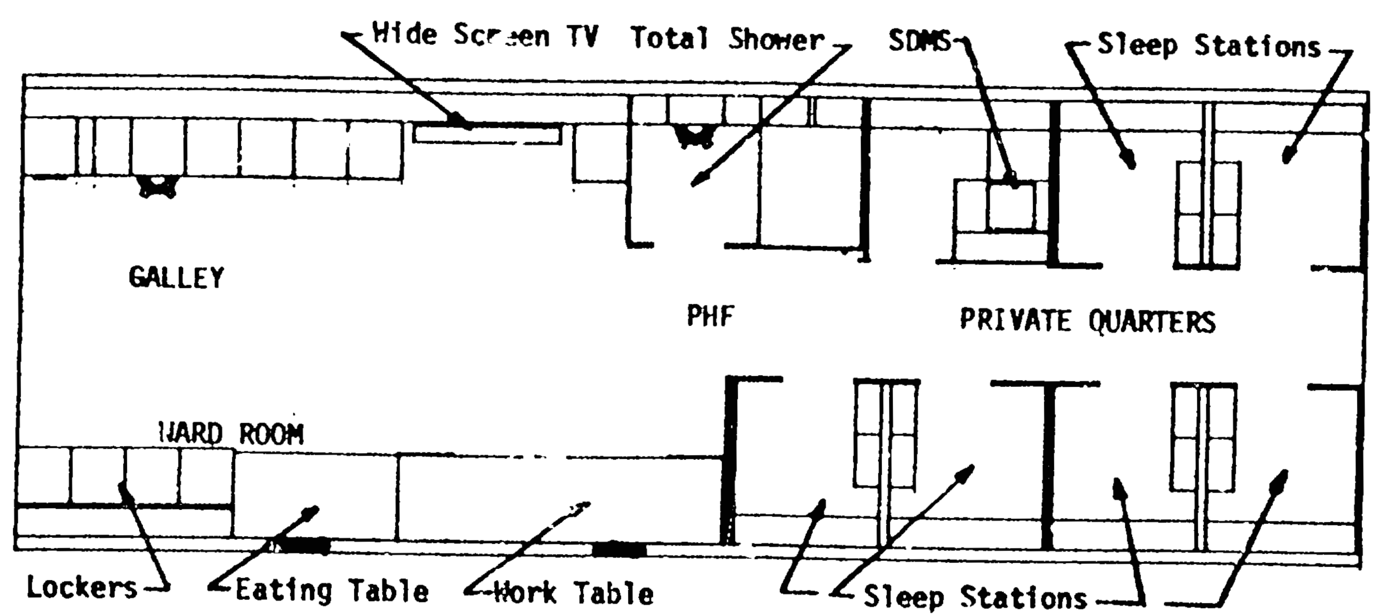

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Figure 4.9-2

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LEFT SIDE VIEW HARMOD #2

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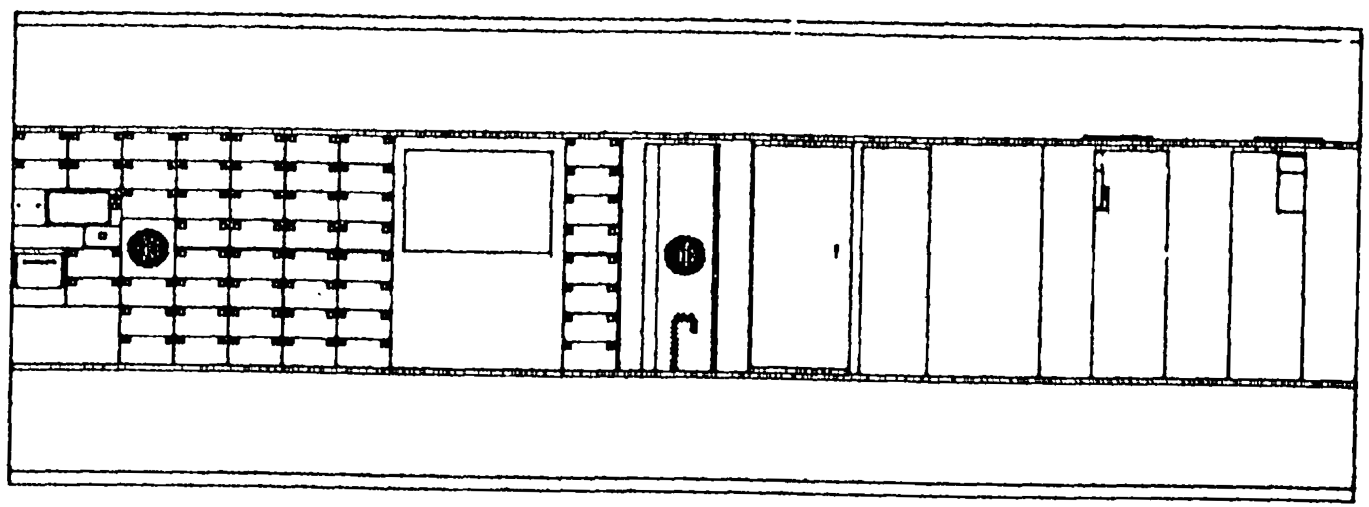

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Figure 4.9-2a

RIGHT SIDE VIEW HABMOD #2

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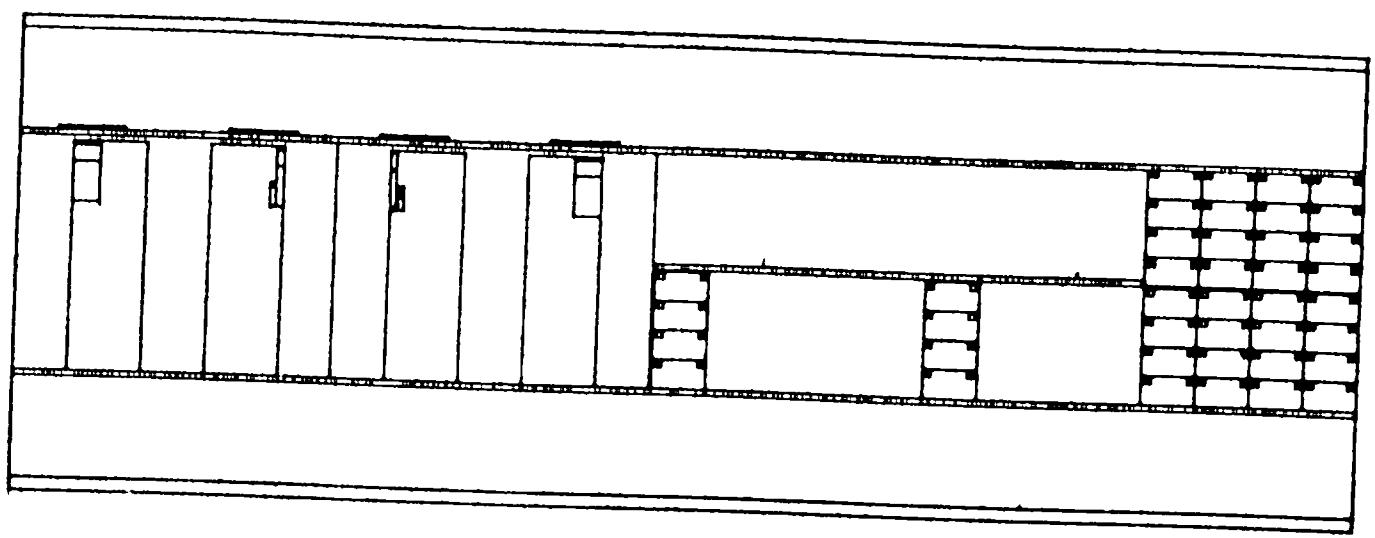

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Figure 4.9-2b

- Freezer
- Trash disposal/compactor
- Hot and cold water dispensers
- Bulk drink dispenser
- Hand washer
- Storage lockers

Further on the left is a large screen for projection TV and then a column of lockers against a bulkhead.

On the right side of the module, in the wardroom area, are four columns of lockers and then a long table. This provides an eating surface for six crewmembers plus a work bench for equipment maintenance and repair. Above the table are two viewing windows.

Exiting the galley/wardroom, on the left, is the Personal Hygiene Facility consisting of the shower/hand washer/urinal in one encloser and the commode in the next one. Next on the left is an enclosure for the Space Station Commander's Office. It contains a standard "look in" station, an audio control (including ground communications capability), a writing surface, and storage lockers.

The rest of HM2 contains the six private crew quarters, two on the left side and four on the right. They are configured like the private quarters in HM1. As in HM1, acoustic and vibration insulation is vital for the quiet, undisturbed rest of the crew members when they are in their private compartments.

4.10 Estimated Power Profiles

The initial step in generating a Space Staton power profile was the development of a conceptual station timeline for 48 hours of representative

activities. Each crew person's activity for a 24 hour period was divided into three major areas:

- o 9 hours - Space Station science and payload operations
- o 7 hours - Personal time: meals, exercise, hygiene, scheduling, recreation, training, housekeeping, maintenance
- o 8 hours - Sleep

The following major events were scheduled:

FIGURE	CREW	ACTIVITY
4.10-1	Red	Rendezvous, science
4.10-2	Blue	RMS/OMV servicing, materials processing
4.10-3	Red	EVA, science
4.10-4	Blue	Science, materials processing

Assumptions and scheduling techniques used:

1. Several activities were scheduled for four crew persons simultaneously for simplicity in scheduling and analysis as well as being close to they way things may be done:

   a. Schedule - ½ hour for review of personal schedule for the next day.

   b. Med Test - ½ hour to record medical data on each person.

   c. Exercise - ½ hour assumed, two simultaneous exercises would go on. Stagger four over schedule and med test time.

   d. Pre and post sleep - ½ hour each, personal hygiene.

   e. Handover - ½ hour all eight crew persons together every 12 hours to review status of work and do scheduling/planning.

   f. Meal - one hour per person, three times per day (except on EVA day)

2. Other activities defined for Space Station:

a. Training - one hour, 1-2 crewmen per day when possible; crew reviews procedures or tutorial lessons.

b. Data management - 25-30 minutes per 12 hours to do onboard computer maintenance and keep tapes.

c. Recreate - one hour per crewman per 24 hours when time was available.

d. Med Tech - ½ hour per 12 hours to record, maintain, service, initiate ongoing medical experiments.

e. Nav - 5 minutes per 24 hours to maintain navigation platforms.

f. Personal Time - 8-9 hours per person every 7 days (only shown once in this example.)

g. Housekeep/maintenance - ½ hour per 12 hours to do regular station cleaning and maintenance.

3. Science, rendezvous, EVA and materials processing were scheduled to represent possible activity mixes the Space Station working group has baselined.

A preliminary Electrical Power System (EPS) analysis has been performed for the IOC configuration of Space Station based on this conceptual 48-hour timeline of representative activites. The electrical equipment list of approximately 900 line replaceable units (LRU) and preliminary utilization information was provided by the subsystem experts from the Engineering Directorate in the form of activity blocks. Payload data was extracted from reference 4.10-1 and was scheduled based on data provided by the ED and MOD. The results of the preliminary EPS analysis of the IOC configuration of Space Station with payloads is presented in figure 4.10-5. Peak loads approach 111 KW, but the average power requirements over the 48-hour period is 98 KW.

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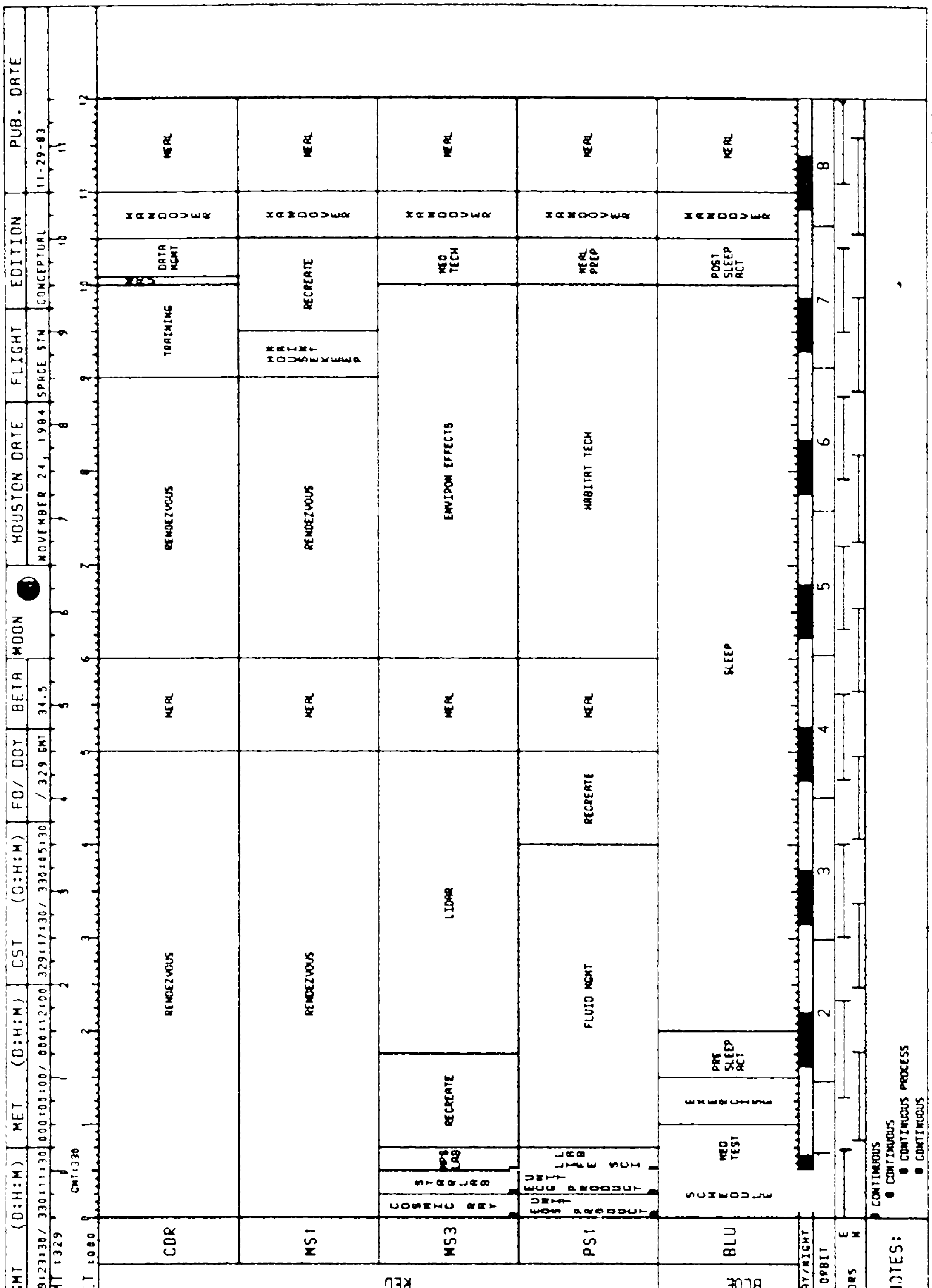

Figure 4.10-1. Conceptual 43-hour Space Station Timeline

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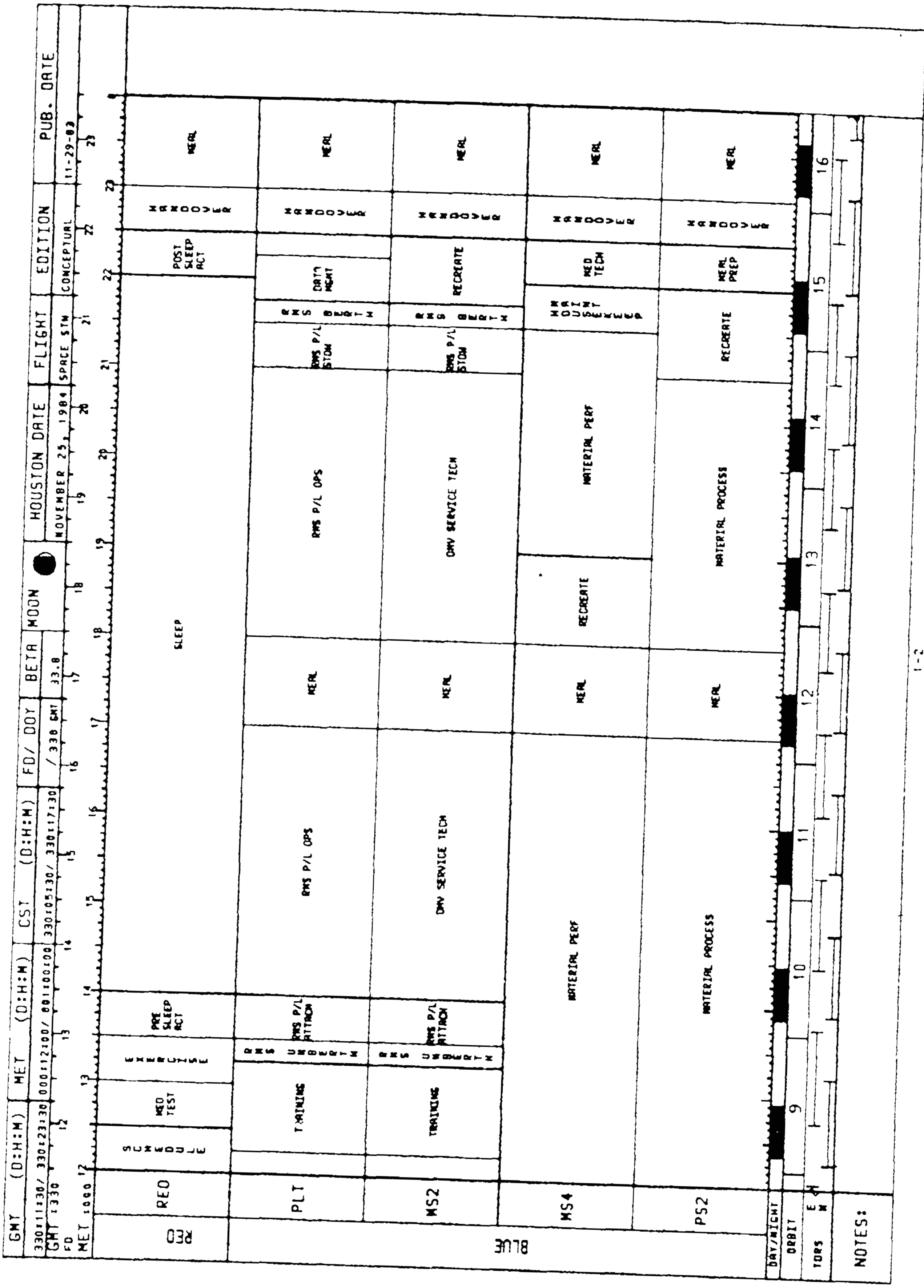

Figure 4.10-2. Conceptual 48-hour Space Station Timeline

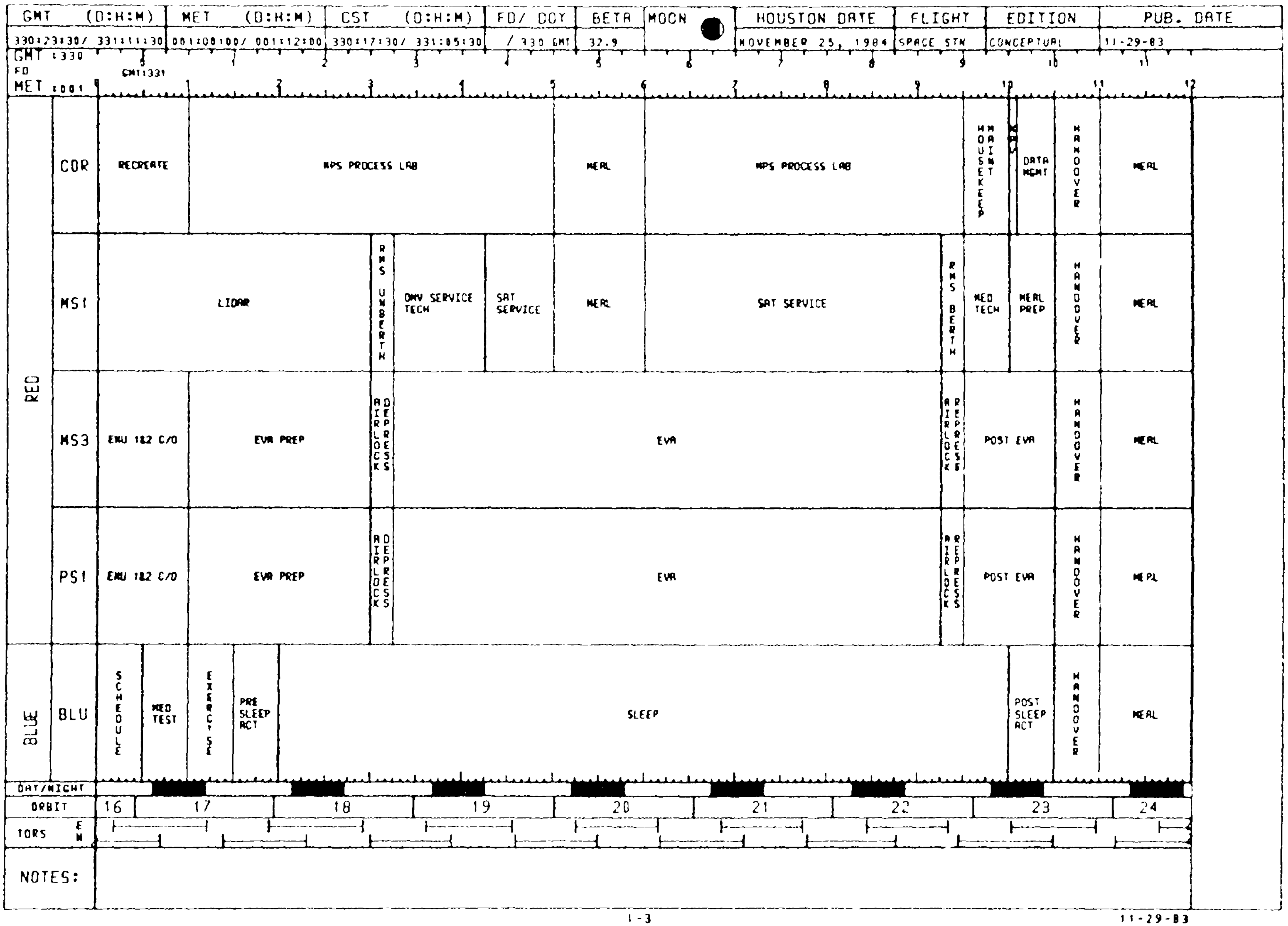

Figure 4.10-3. Conceptual 48-hour Space Station Timeline

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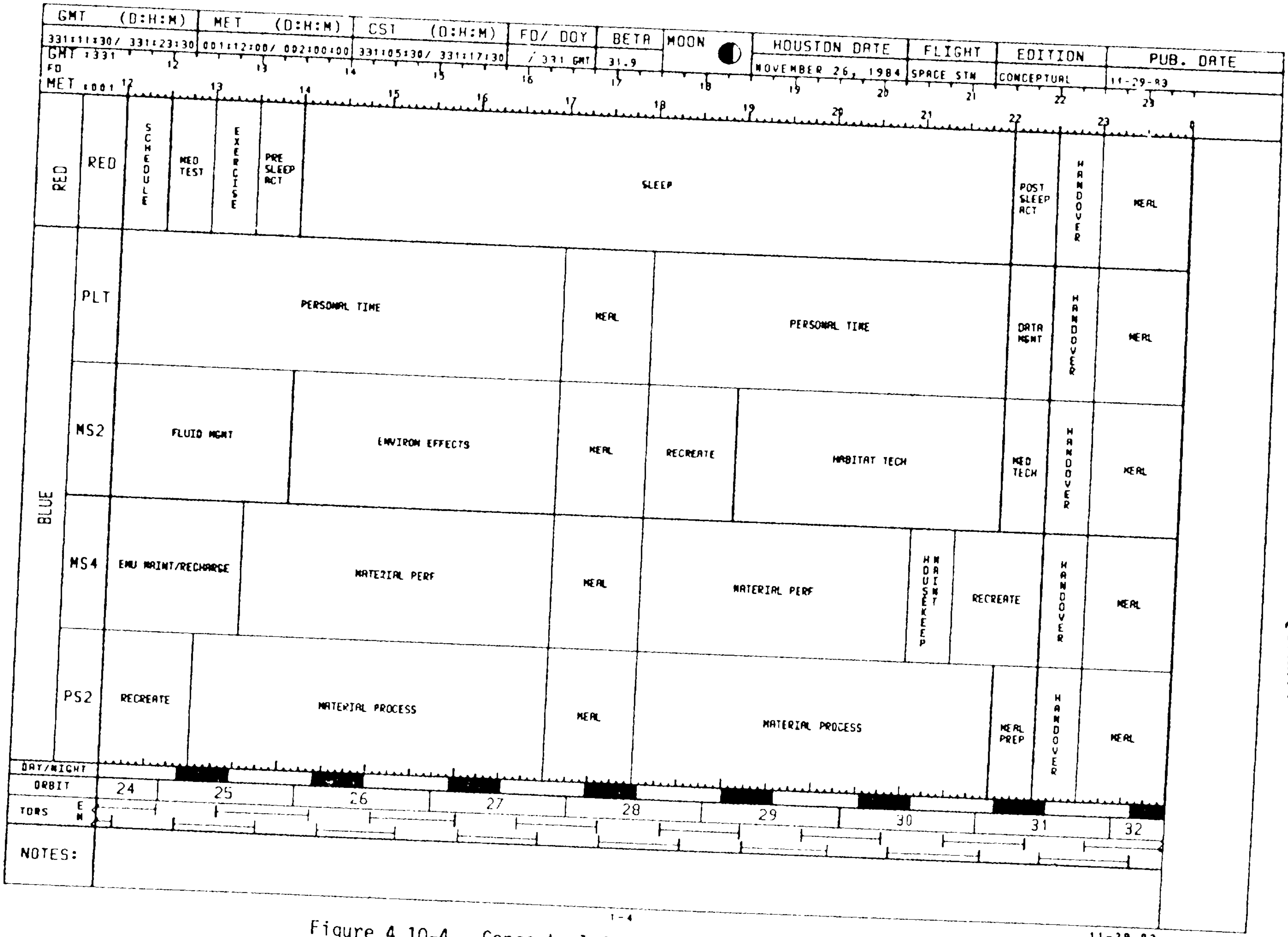

Figure 4.10-4. Conceptual 48-hour Space Station Timeline

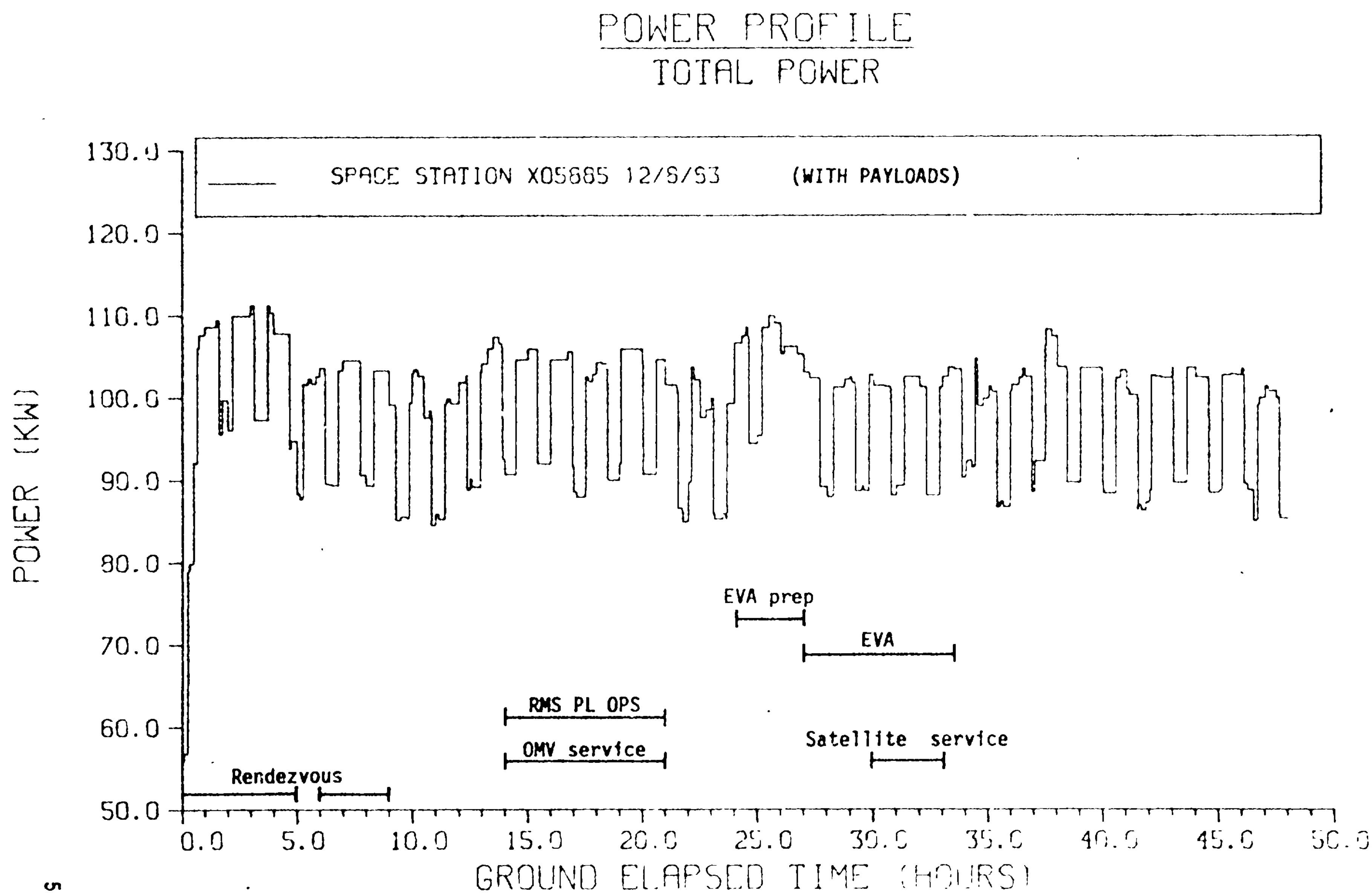

Figure 4.10-5. Space Station (IOC) Total Power with Payloads

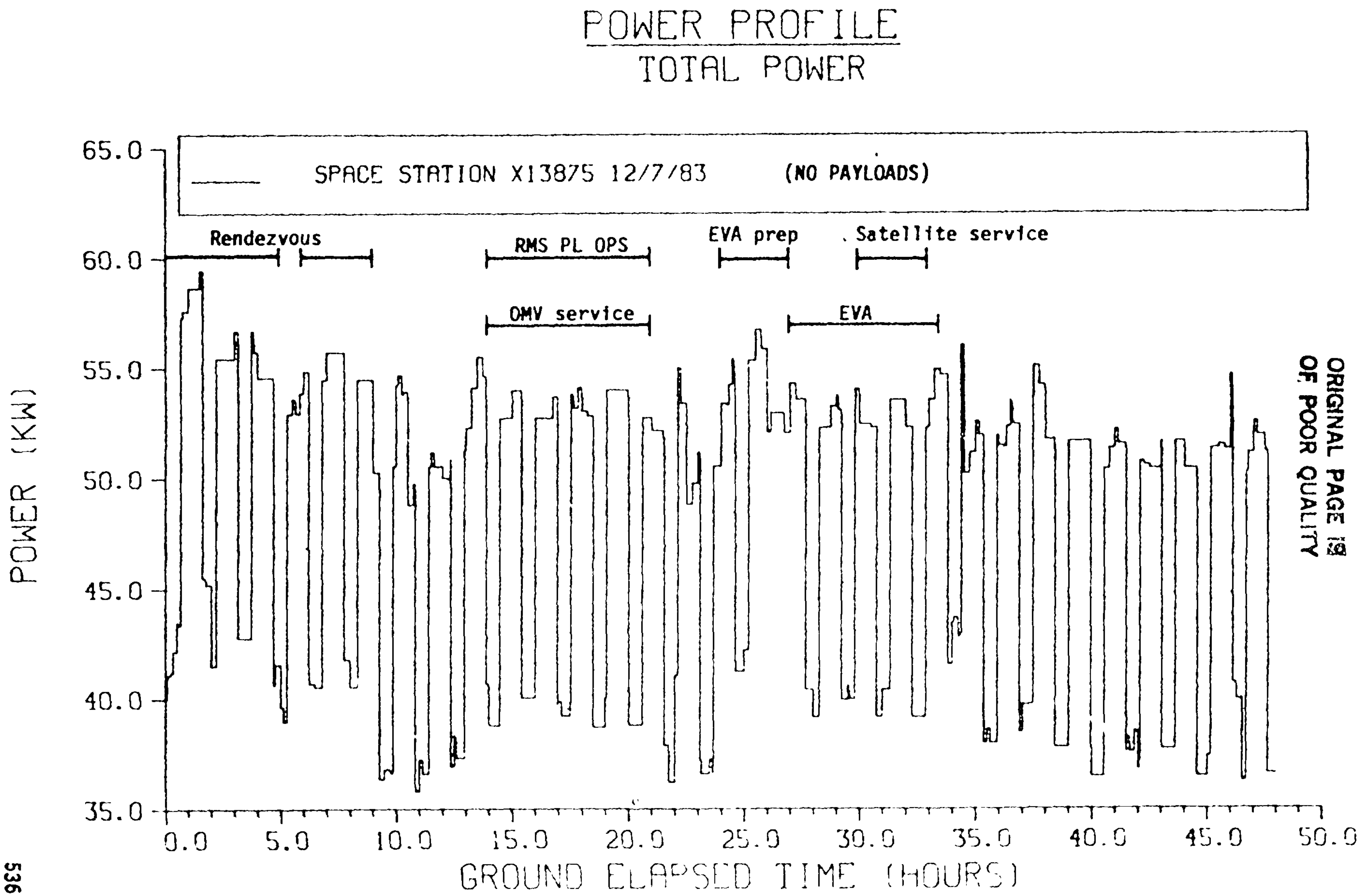

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Figure 4.10-6. Space Station (IOC) Total Power - No Payloads

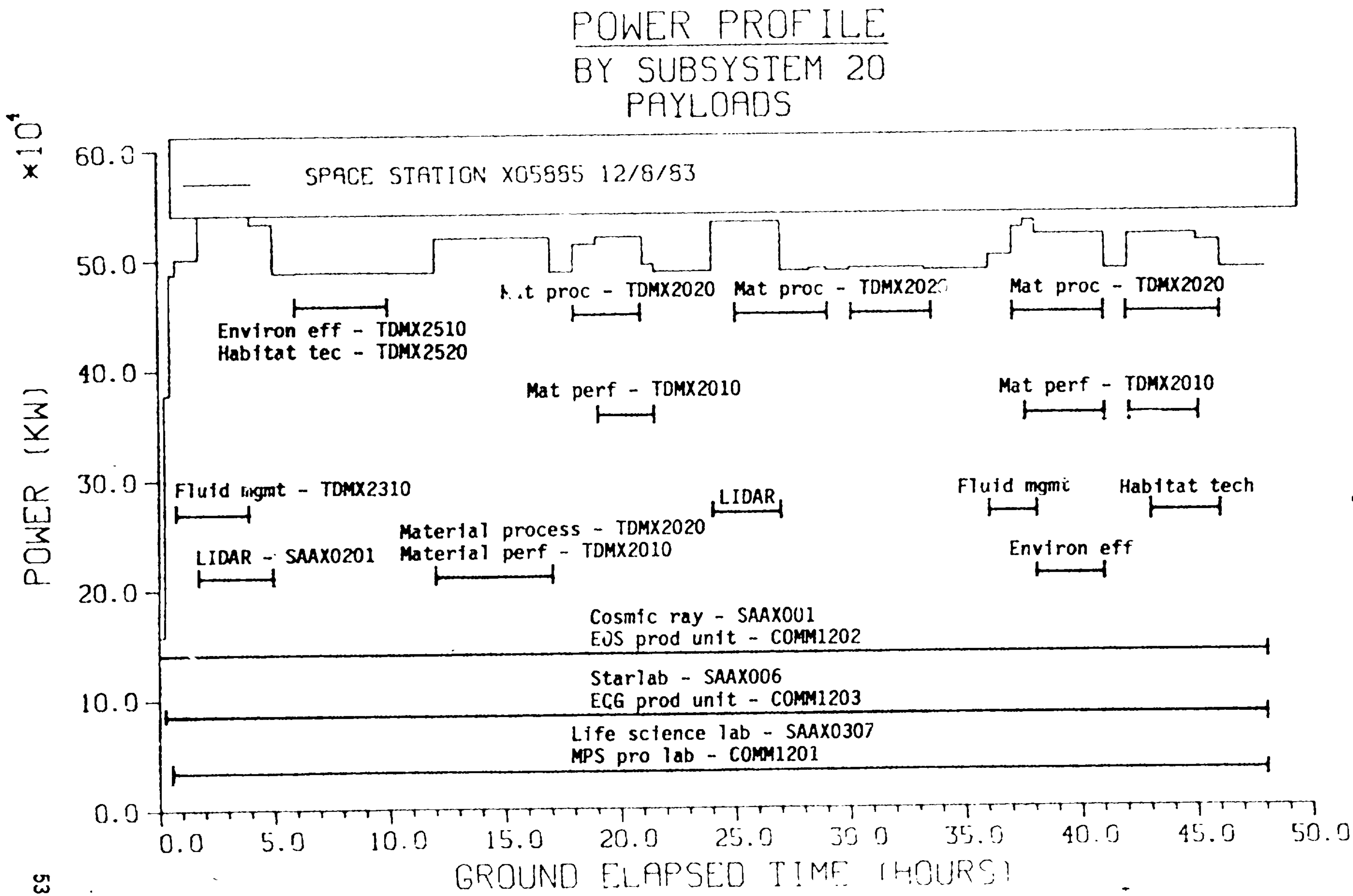

Figure 4.10-7. Space Station Typical Payload Power Profile

Table 4.10-1

SPACE STATION EPS ANALYSIS
SUBSYSTEM SUMMARY

		TOTAL POWER, KW (ALL COMPONENTS ON)	AVERAGE POWER, KW (OVER 48 HR MISSION)
01	AUDIO	0.63	0.63
02	COMMUNICATION	6.27	2.23
03	CREW ACCOMM	8.71	1.38
04	DATA ACQ & PROC	1.71	1.09
05	DOCKING/BERTHING	2.39	0.23
06	DATA MGMT	1.95	1.77
07	ECLSS	23.46	14.00
08	GUID AND CONTROL	3.09	3.08
09	DISPLAYS AND CONTROL	9.58	5.79
11	NAVIGATION	0.38	0.35
12	POWER DIST & CONTROL	7.20	7.20
13	POWER GENERATION	0.40	0.40
14	ONBOARD PROP	1.00	0.45
15	STRUCTURES	0.00	0.00
16	ATCS	5.88	5.68
17	TV	3.25	2.36
18	TRACKING	4.15	0.33
19	FACILITY MGMT	0.80	0.80
20	PAYLOADS	58.15	50.30
21	OPS PLANNING	.06	0.06
	TOTAL	139.06 KW	98.13 KW

Figure 4.10-6 presents the Space Station subsystem requirements alone, without payloads. This figure demonstrates that the Space Station housekeeping power requirements average 48 KW. A more detailed representation of power usage is shown by individual subsystem in table 4.10-1. Table 4.10-1 presents both the average power of the subystem over the 48 hour conceptual timeline and the total power requirement if all components in that subsystem were continuously activated. The power requirement with all Space Station and payload components on is 139 KW.

The payload power profile assumed for this EPS analysis is presented in figure 4.10-7. An average power of 50 KW is dedicated to the paylaods alone.

4.11 Rendezvous and Proximity Operations

4.11.1 Operational Control Zones

The approaching Space Station era requires that station operations will encompass many orbiting elements, each with different operational descriptions. These elements, i.e., OMV, OTV, MMU, free-flyers, etc., will be used in conjunction with the station to define various operational requirements. Some of these requirements are:

- Standardize flight planning
- Standardize crew planning
- Minimize Space Station and orbiting element hardware and software requirements
- Provide collision avoidance
- Define communications and telemetry requirements
- Define command and control requirements

In order to best satisfy these numerous requirements, the concept of Operational Control Zones (OCZ) has been adopted. In essence, this is the definition of specific operational flight regimes, whereby control may be maintained for each discipline's area of operation. At this point, eight zones have been defined to support Space Station traffic control activities. Following are brief descriptions of the eight zones. The reader should refer to Figure 1 in conjunction with the descriptions.

4.11.1.1 ZONE 1

Within this zone, all proximity operations (PROX OPS), including docking and berthing, will take place. In addition, all MMU and EVA activity is restricted to this zone. It consists of the region enclosed by a 3,000 ft. sphere centered on the Space Station.

4.11.1.2 Zone 2

All rendezvous with the station will be targeted to arrive within this zone. As shown in figure 4.11-1, it extends from TBD ft. (approximately one n.mi.) ahead of the Space Station to 100 n. mi. behind it. Its location and size have been designed to be consistent with the current Stable Orbit Rendezvous Technique (SOR). In SOR, the chaser vehicle arrives at an offset point some distance behind the target and performs its closing maneuvers from this point. Upon entering Zone 1, the chaser will move from the rendezvous to the proximity operations phase.

4.11.1.3 Zone 3

All departures from the Space Station will nominally take place within this zone after initial deployment and separation maneuvers are performed in Zone 1. This zone extends from the forward edge of Zone 2 to a point TBD n. mi. (approximately 100 n. mi.) ahead of the Space Station. It has been located

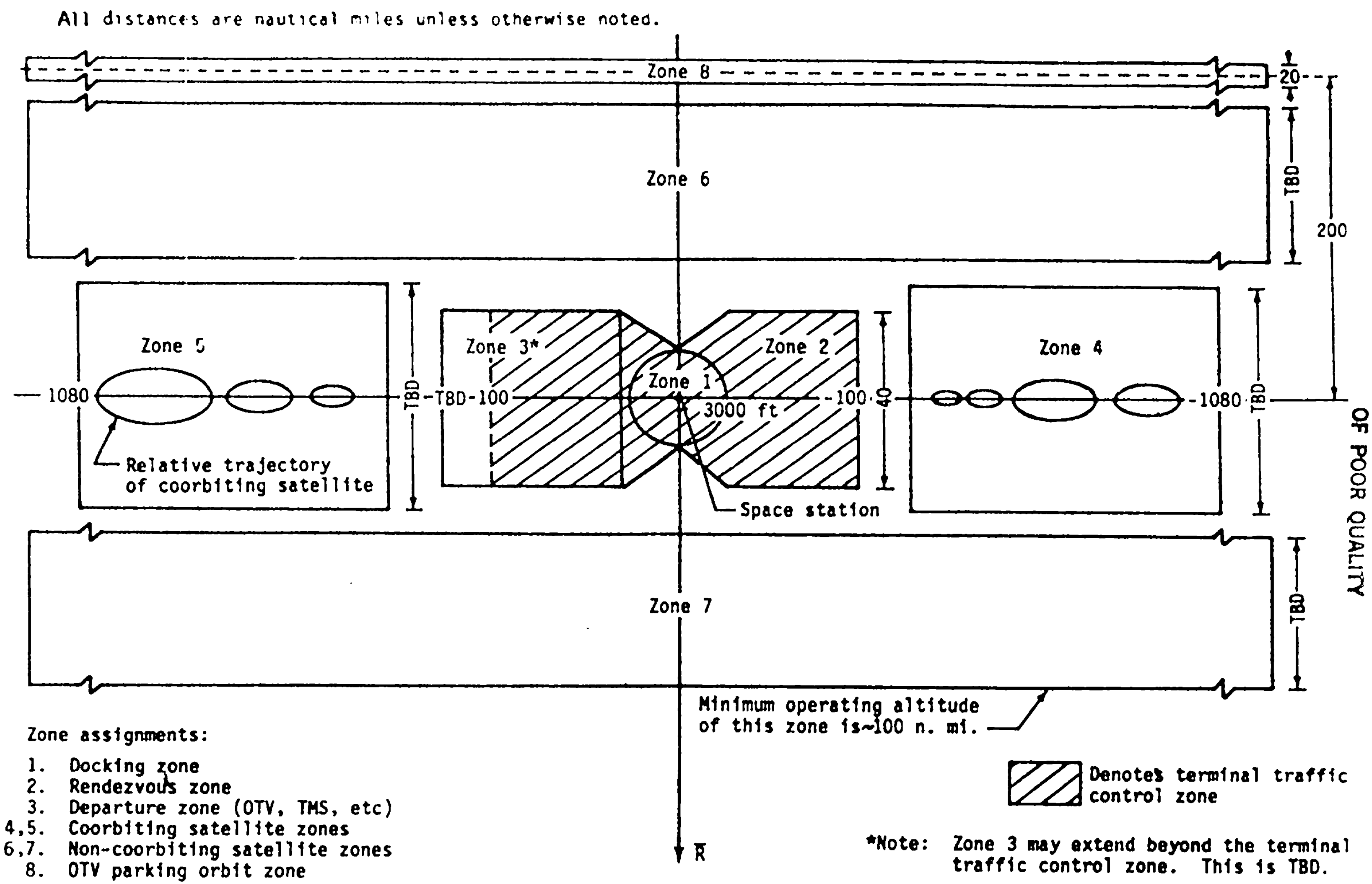

Figure 4.11-1. Operational control zones.

ahead of the Space Station to aid in collision avoidance between departing vehicles and the Space Station.

4.11.1.3.1 Terminal Traffic Control Zone

Within this zone, traffic concentrations become such that continuous communications and control are required by the Space Station for collision avoidance between spacecraft. This zone is approximately cylindrical and is centered on the Space Station. It has a radius of approximately 20 n. mi. and extends 100 n. mi. along the Space Station orbital flight path.

4.11.1.4 Zone 4 & 5

These zones are dedicated to Co-orbiting Satellite (COS) operations. A COS is a type of free-flyer (FF) that possesses some stationkeeping capability and utilizes this capability to remain co-orbital with the Space Station. As seen in figure 4.11-1 the cylindrical COS zones are centered about the Space Station velocity vector. Two zones are allocated for these FF, to allow for greater operational flexibility. Zone 4 begins TBD n. mi. behind the Space Station. It extends to range of -1,080 n. mi (the maximum line-of-sight (LOS) range for communications and tracking). Zone 5 begins TBD n. mi. ahead of Zone 3. It extends to a range of 1,080 n. mi. ahead of the Space Station.

4.11.1.5 Zone 6 & 7

These zones contain the Non Co-orbiting Satellites (NCOS). These are detached payloads that are not under Space Station command and control. These zones are located both above and below the Space Station orbit track. Two zones have been allocated for NCOS operations in order to accommodate user requirements.

4.11.1.6 Zone 8

This zone is the parking orbit utilized by OTV's returning from geosynchronous

TABLE 4.11-1. TIMELINE FOR -VBAR SEPARATION SEQUENCE

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EVENT NO.	TIME (PET) HH:MM:SS.S	SEPARATION EVENT
1	0:00:00.0	0.2 fps -Vbar sep burn mnvr in Inertial Attitude Hold (IAH). The other sep burns will do the same initial burn (opposite the docking port).
2	0:00:00.8	Coast in Local Vertical Local Horizontal Attitude Hold (LVLH).
3	0:10:00.0	1.0 fps +Rbar sep burn mnvr in IAH. All sep mnvrs will do this sep burn for the second burn.
4	0:10:02.6	Auto mnvr to Local Vertical Retrograde sep burn attitude.
5	0:15:31.9	End auto mnvr; continue coast to sep burn in LVLH to achieve a separation distane of 1000 ft or more.
6	0:30:00.0	3 fps Retrograde OMS sep burn in IAH (this will get the vehicle approx. 10 n mi away from the Space Station)
7	0:30:01.7	End sep burn and coast for one revolution in LVLH to the Zone-Of-Departure (ZOD).
8	2:00:00.0	Range = approx. 10 n mi.

orbit. It may also be used by spacecraft returning from lunar or planetary missions. Zone 8 is located +200 n. mi. above the Space Station orbit track.

4.11.1.7 Rendezvous

A simplistic definition of rendezvous may be expressed as the act of bring 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 pre-determined maneuvers designed to ensure that the chaser will rendezvous with the target at a desirable position in the target's orbit, within a prescribed amount of time. The desired target position at rendezvous depends mainly on lighting conditions, while the amount of time can range from hours to days. In general, the planned rendezvous maneuvers will leave the chaser vehicle in a stable position within a specified distance to the target (approximately 1,000 feet). The final approach to the target (inside 1,000 feet) is considered a part of PROX OPS.

The Space Station rendezvous problem varies slightly from that of current Shuttle operations. To illustrate the rendezvous scenario, consider the Space Station in a 270 n. mi circular orbit as the "target," and the OMV in a 470 n. mi. circular orbit as the "chaser." In addition, assume that the two orbits are coplanar.

The initiation of the rendezvous sequence begins with a phase-adjust maneuver (executed by the chaser OMV), denoted as NC1. This maneuver is designed to allow the OMV to "catch up" to the station in terms of the differential phase angle between the two vehicles by changing the orbital period of the chaser. For the analysis referenced here, the phase angle was chosen so as to achieve a rendezvous in approximately 24 hours.

The next step is to place the OMV in an elliptical orbit with perigee being at the same altitude as the station. This is accomplished via a Hohmann transfer designated as NH1 which occurs 12.5 revolutions (revs) following NC1. One-half rev after NH1, a second phasing maneuver, NC2, is executed by the OMV to further diminish the catchup rate between the OMV and the Space Station. This is done at a range of approximately 25 n. mi.

Proper lighting conditions are critical during the final portions of the rendezvous when the maneuvers are to be performed visually by the crew (manned vehicles) or by way of a traffic controller using relayed Closed-Circuit Television (CCTV) data. Furthermore, the chaser and target vehicles must remain illuminated long enough to allow for completion of the rendezvous and PROX OPS phases. In the case of unmanned rendezvous, with maneuvers being computed using on-board software, there is no concern with lighting requirements. However, it is important to mention here that all early maneuvers are designed to provide the correct conditions for the final maneuvers.

The corrective combination maneuver, or NCC, is executed by the OMV, a full rev after NC2. This maneuver is designed to place the OMV approximately 1,000 feet ahead of, or behind the station, depending on the type of return desired, approximately 320° of target travel later. The NCC maneuver is executed approximately eight n. mi. behind the station. An additional responsibility of the corrective combination is to take out any dispersions in the OMV trajectory, i.e., height, phasing, and plane dispersions.

The final maneuver in the Space Station rendezvous sequence is the NSR, or "null slow rates" maneuver. NSR nulls any out-of-plane velocity between the OMV and the Space Station. At this point, the rendezvous phase has been completed and the return profile PROX OPS begins.

As a final note, it is important to realize that the above discussion was presented for an OMV returning to the Space Station from above. The same descriptions would hold for an OMV rendezvous from below. The procedure is the same; only the altitudes and the direction of travel (down vs. up) change.

4.11.1.8 Proximity Operations

PROX OPS is concerned with that portion of a mission when two bodies are within 1,000 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 positive velocity vector). PROX OPS maneuvers may include transitions to various offset points, stationkeeping, approaches, flyarounds, and the separation to a "stand-off" position.

Numerous operational considerations must be addressed regarding the final position achieved by the Orbiter and the PROX OPS techniques used to maneuver the vehicle to this position. A major concern is the effect of the RCS plume on the target (plume impingement). This plume impingement creates problems of contamination and disturbance (over-pressure) experienced by the target. In addition, RCS propellant usage is also critical. Propellant consumption influences the choice of braking maneuvers and stationkeeping techniques. A second concern 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; i.e., Crew Optical Alignment Sight (COAS), Rendezvous Radar (RR) data, etc. In any case, crew visibility and procedural simplicity have considerable impact on the selected maneuvers. Another major consideration for PROX OPS is lighting, especially during the

return of a vehicle to a space station docking port. For the simplicity of this discussion, assume an OMV and the Orbiter as unmanned and manned return vehicles, respectively.

First, consider the approach of a manned vehicle (Orbiter) toward a Space Station docking port on the positive velocity vector. During the close-in approach, crew members will desire a highly visible "target." In order to achieve this end, the approaching Orbiter must arrive in the vicinity of the station sometime during daylight, i.e., after sunrise and before orbital noon. This will place the sun in a position somewhat "over the shoulder" of the Orbiter, highly illuminating the Space Station and its port, providing the crew with optimum visibility. Waiting until after noon to approach along the positive velocity vector could reduce the sun line-of-sight (LOS) angle to unacceptable limits, causing the crew to have to look into the sun as they approach. However, this would be the proper time to approach the station along the negative velocity vector, as the sun would now be in a position to provide the "over-the-shoulder" illumination from the opposite direction. Approaches along the two out-of-plane directions (angular momentum vectors) and from above (along the negative radius vector) are acceptable any time during daylight. In these cases, as long as the sun and the docking port are on the same side of the orbit, the sun will generally be in a position to provide the desired illumination of the station. An approach from below (along the positive radius vector) would be highly dependent upon the relative positions between the sun and the station, i.e., time of day and year. However, lighting is not the only criteria used when evaluating the desireability of an approach direction. Relative motion and orbital mechanics effects are important in the execution and control of PROX OPS maneuvers. In general, approaches occurring from below, exhibit a tendency for the

approaching vehicle (chaser) to move away from the target vehicle due to the effects of orbital mechanics. Likewise, vehicles on the velocity vector exhibit a tendency to remain stationary relative to one another except for differential drag effects. Therefore, in order to begin the desired motion of the chaser toward the target (establish a positive closing rate), motion must be initiated by the chaser. This is not the case for out-of-plane approach scenarios. While the two vehicles are at the same altitude, their orbits are not coplanar. This implies a nodal intersection of the orbits at two points during a given revolution.

Therefore, as the vehicles approach an orbital intersection point, they will already be moving toward each other due to the orbital wedge angle. A positive closing rate is clearly established. Thus, frequent jet firings would be required to inhibit or reduce this rate. These concerns greatly complicate stationkeeping activities and classify out-of-plane approaches as the least desireable from the standpoint of crew considerations.

The above approaches assumed an LVLH stabilized target vector with ports along the velocity, radius, or angular momentum vectors. However, a station stabilized in an inertial attitude presents different aspects on lighting and relative motion. Consider a docking port configuration consisting of a simple cubic structure with six ports, one on each face of the cube. For a given daylight scenario, with the sun in an arbitrary position, the following situation exists. Due to the inertial attitude of the station and its ports, their positions are fixed relative to the sun. In a specific case where the sun angle (b) is 0°, and one of the ports points toward the sun, four of the ports (on the cube) will be at 90° angles with respect to the sun LOS. A fifth will essentially be "on" the LOS at 0°, with the final port being on the LOS at + 180°. In this last situation then, a manned vehicle would have to

look into the sun during the approach no matter when the approach occurs (except darkness). The five remaining ports would have adequate lighting for all daylight situations.

Likewise, for relative motion considerations during the approach, the inertially stabilized station would require a different approach philosophy. The initial approach would be along some pre-determined direction (nominally the positive velocity vector) as with the LVLH station. However, at some range (approximately 200 feet) the chaser would go to an inertial attitude profile to match attitude rates with the target and continue the approach.

Considerations for approaches of unmanned vehicles could contrast with those of the previously discussed manned Orbiter. In both cases, the sun position must be relevant to the visual direction of the person/CCTV in control of the approach. Furthermore, there is a dependence on the type of navigation system that the vehicle employs. Assume the situation of an OMV approaching a station along the negative velocity vector. If a "traffic controller" in the station desires a visual acquisition, it is advantageous to time the approach to occur between sunrise and noon, due to favorable lighting conditions. However, if the OMV is to relay CCTV data back to the station, by which the operator can maneuver the OMV toward the dock, then it becomes desirable to have the OMV "backlit," i.e., the approach should occur after noon, and prior to sunset. This leads to an inevitable situation--one that calls for the operator to view the OMV as it approaches, in addition to monitoring the CCTV data output (via a TV screen). The sun LOS is also important if OMV guidance is dependent on a device sensitive to light magnitudes (stars, planets, etc.). If such is the case, a sun LOS angle of less than 30° may cause saturation of the sensor and the signal may be lost.

These comments provide the background for drawing the following conclusion concerning Space Station designs. By providing the various configurations with as many docking ports as possible (on the major LVLH axes) there may be fewer restrictions on approach timelines. This could be an important factor when considering contingency planning or emergency operations.

4.11.1.9 Space Station Separation Profile: Mission Overview

There are a number of possible separation scenarios which include separations from ports on the radius, velocity, or angular momentum vectors. These are "stardard" choices, but they are dependent on the location of the docking ports which are, in turn, a function of the individual station configurations. In order to illustrate a general scenario, consider an OTV docked to a port on a given Space Station configuration. Initially, a small sep burn will be performed to begin the OTV movement away from the port. The direction of the burn will be directly away from the docking port to maximize an opening rate. Another small, intermediate burn may be required to ensure a favorable geometry between the OTV and station for the large separation burn to follow. After a coast, the OTV will perform the large sep burn to ensure a separation range of approximately 10 n. mi. at OTV orbital transfer burn ignition. It is at this point that the separation sequence has essentially been completed. It should be noted that all maneuvers performed in the vicinity of the Space Station will be designed such that protection of the Space Station and any co-orbiting satellites is considered consistent with safety considerations. These include possibilities of recontact, plume impingement, and the explosion range of the OTV upon ignition of its upper stage. This range from the station will be achieved by the OTV through a combination of coast times and separation delta V's, and will be consistent with the baslined OCZ. In addition, minimizing propellant consumption will be an important objective.

Finally, it is assumed that the separation sequences of the OMV and the Orbiter will be similar to that of the OTV, i.e., differences in the vehicular mass properties will result in very minor changes in the respective relative trajectories and differing amounts of propellant consumed. However, significant differences in these trajectories will appear after the orbital transfer burns due to the different thrust-to-weight ratios of the three vehicles.

The separation procedures for the previously mentioned scenarios, presented here in more detail, reflect the maneuvers of an OTV from the Space Station Building Block configuration which maintains a local-vertical, local-horizontal (LVLH) attitude. Due to various groundrules and assumptions these procedures are assumed to relate equally well to the Big "T" and Delta configurations. The only exception to this assumption are the differences in configuration ballistic numbers and the fact that the Delta configuration, maintaining an inertial attitude, may never rotate such that its "offset" ports are aligned on or near one of the "standard" axes described previously. The differences in ballistic numbers would result in minor changes to the relative trajectories, while the docking port alignment may necessitate the development of a new separation procedure.

Table 4.11-1 lists a detailed timeline for a separation sequence along the negative velocity vector. In this timeline and the accompanying relative motion plots, all burns are referenced to the LVLH axes. There are several noteworthy features contained in this timeline. First of all, it is important to see that this is a standardized timeline, i.e., event number one is a 0.2 fps -Vbar separation burn in Inertial Attitude Hold (IAH), and would be the same for all docking ports. The only exception is that the burn will be in a

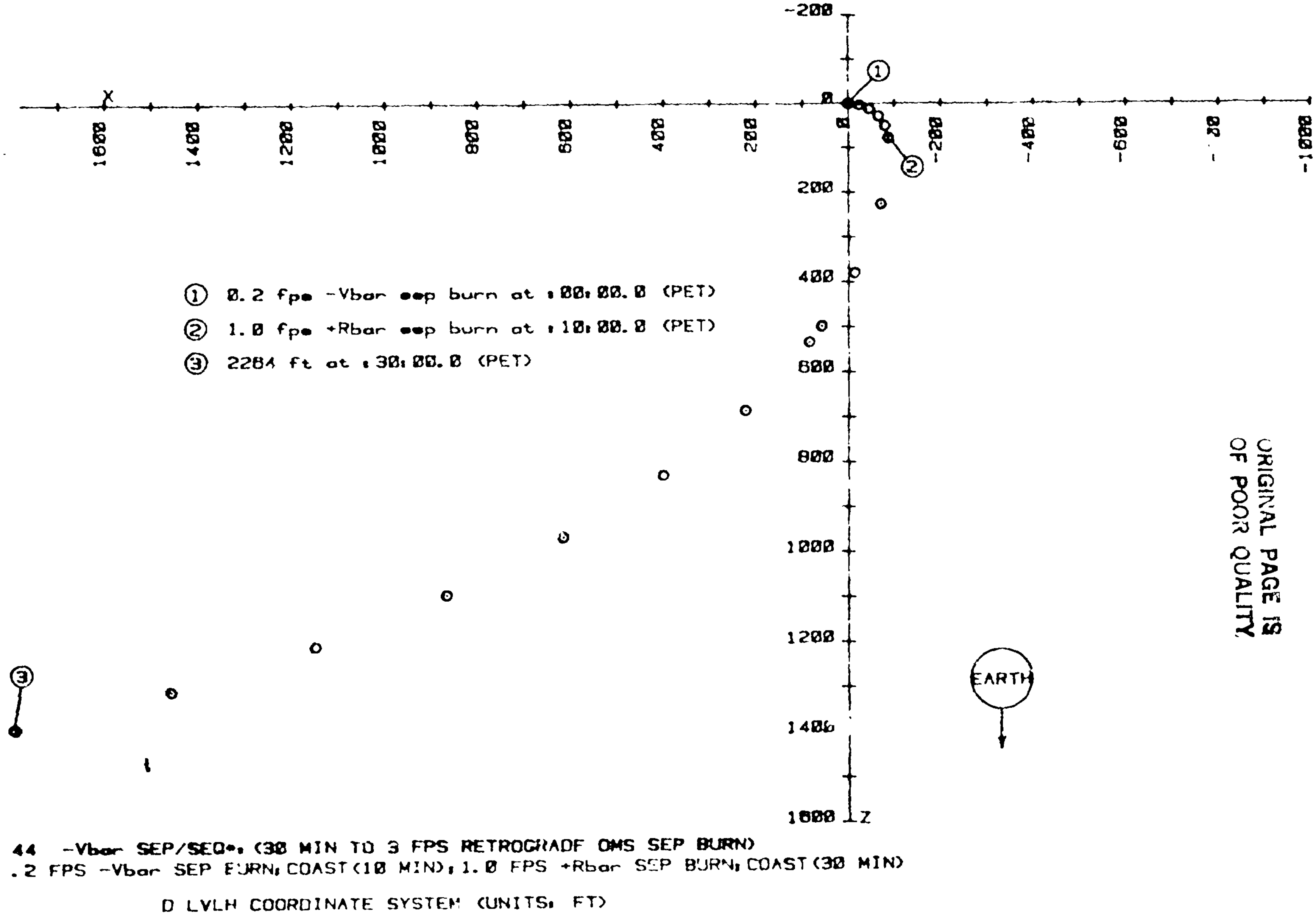
direction away from the port of departure. Event three, a 1.0 fps radial separation burn in IAH is also identical in all sequences. This event, when coupled with events 4-7, will place the OTV approximately [illegible] n. mi. ahead of the station in the departure zone. Note that events 4-6 are designed to place the OTV in an attitude such that the three fps OMS type burn is purely retrograde.

Figure 4.11-2 illustrates the relative motion of the OTV with respect to the Space Station in LVLH coordinates. The plots show the separation from initiation through 30 minutes (near-field), post-OMS burn (intermediate range), and two hours (farfield) respectively. Similar timelines and relative motion plots result for separations on each of the other major LVLH axes. They are not presented here, however, to avoid repetition.

4.11.1.10 Space Station Return Profile: Mission Overview

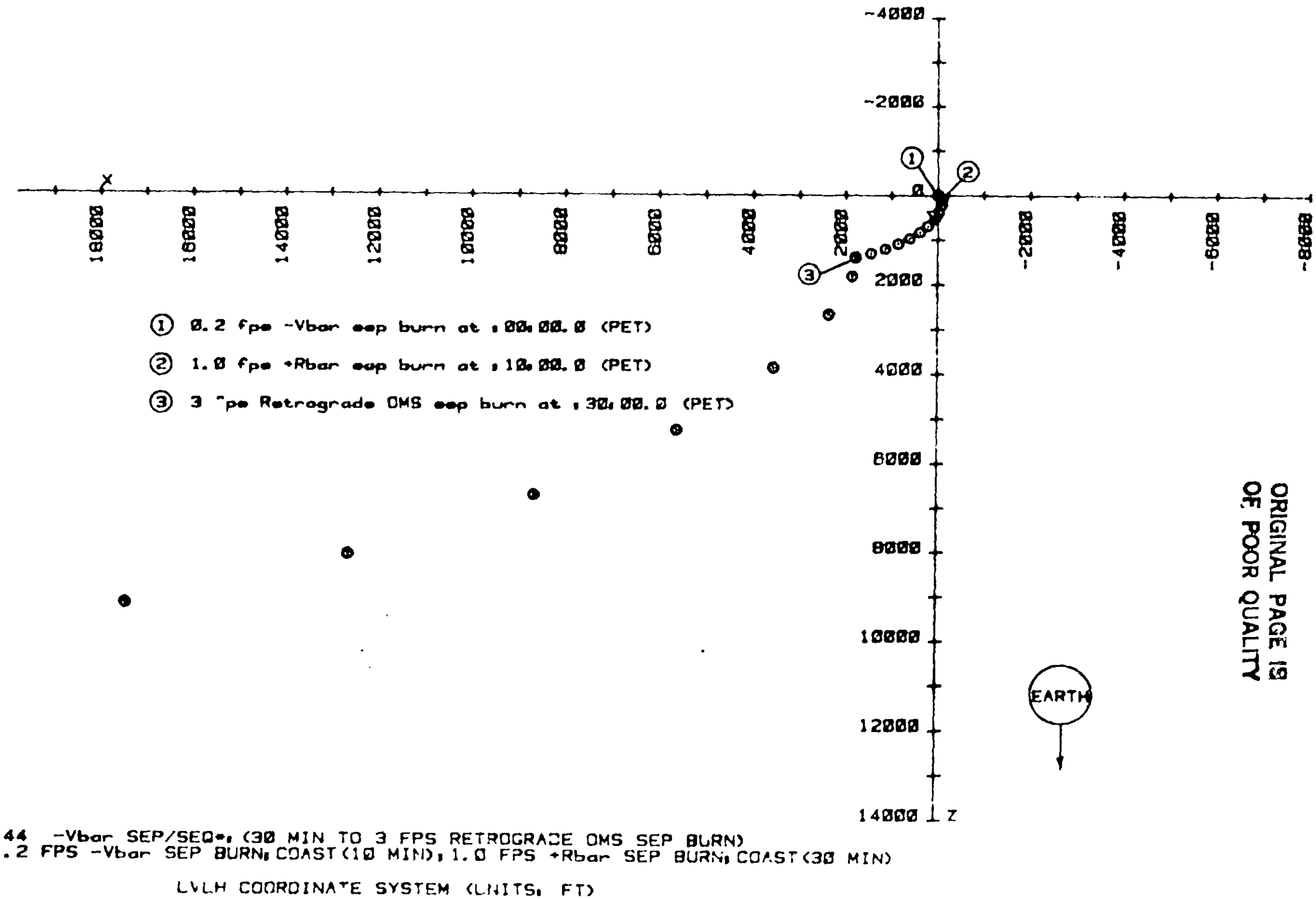
There are also a number of possible return scenarios which include returns to docking ports located on the radius, velocity, and angular momentum vectors. Again, these are "standard" choices, dependent on the docking port locations which are in turn a function of the individual station configurations. An additional choice would be an inertially located port.

For a general mission overview, consider an OMV returning to a docking port on a given Space Station configuration. Since the vehicle's return covers the PROX OPS range of 1,000 feet from the station to the docking port, it is assumed that a successfully executed return rendezvous profile has placed the vehicle approximately eight n. mi. behind the station in the rendezvous zone. An auto-return is then executed to bring the OMV on in to the 1,000 foot offset point, either ahead of or behind the station depending on the type of return. All maneuvers performed from the eight n. mi. offset point on in to dock are assumed to be performed automatically with an available manual

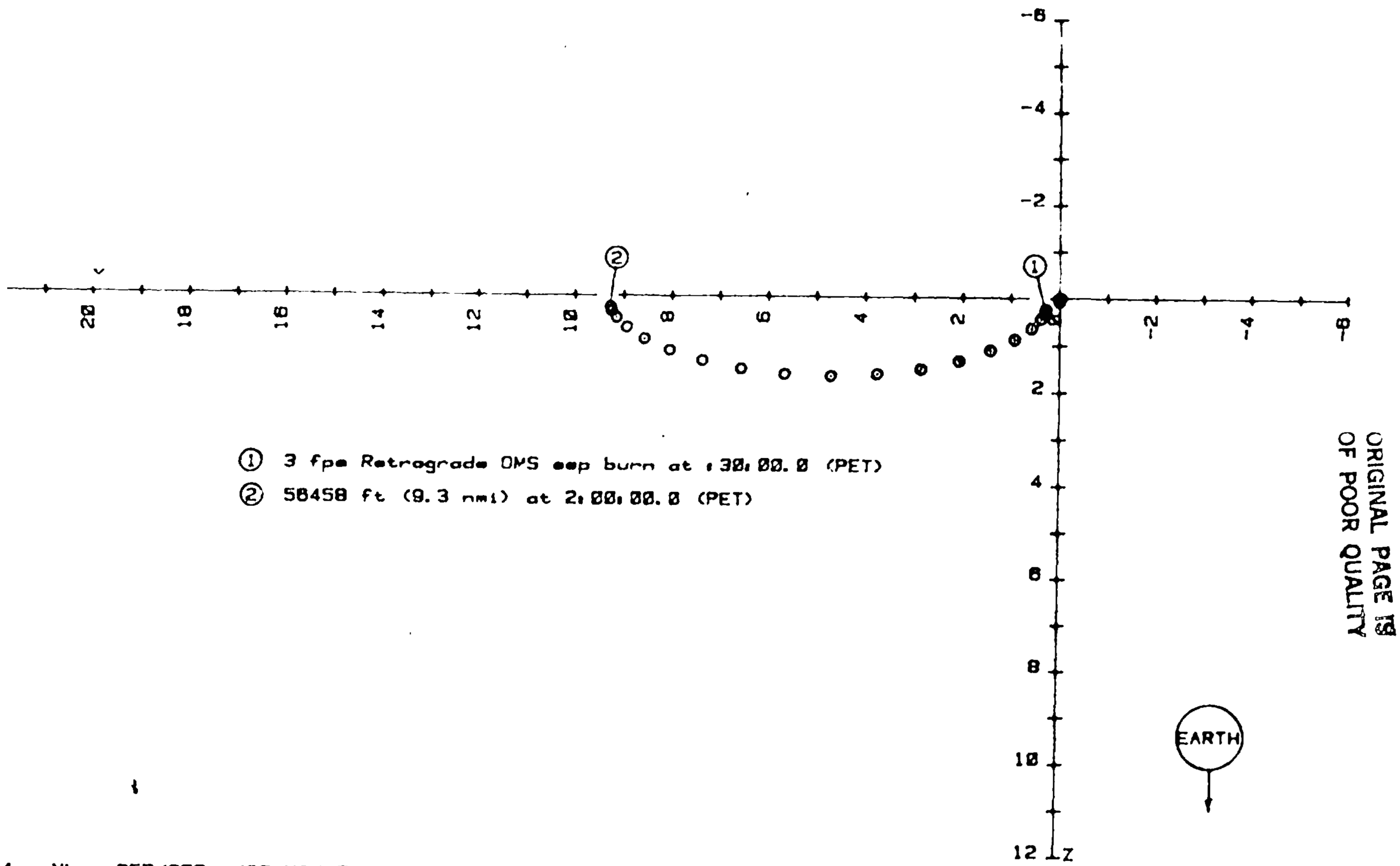


(a) Close-in Motion.

Figure 4.11-2. Motion of the OTV Relative to the SS for Separations from the Negative Velocity Vector Decking Port



(b) Intermediate Range Motion.
Figure 4.11-2. Continued



44 -Vbar SEP/SEQ*: (30 MIN TO 3 FPS RETROGRADE OMS SEP BURN)
.2 FPS -Vbar SEP BURN, COAST (10 MIN), 1.0 FPS +Rbar SEP BURN, COAST (30 MIN)

LVLH COORDINATE SYSTEM (UNITS, NMI)

(c) Long Range Motion.
Figure 4.11-2. Concluded

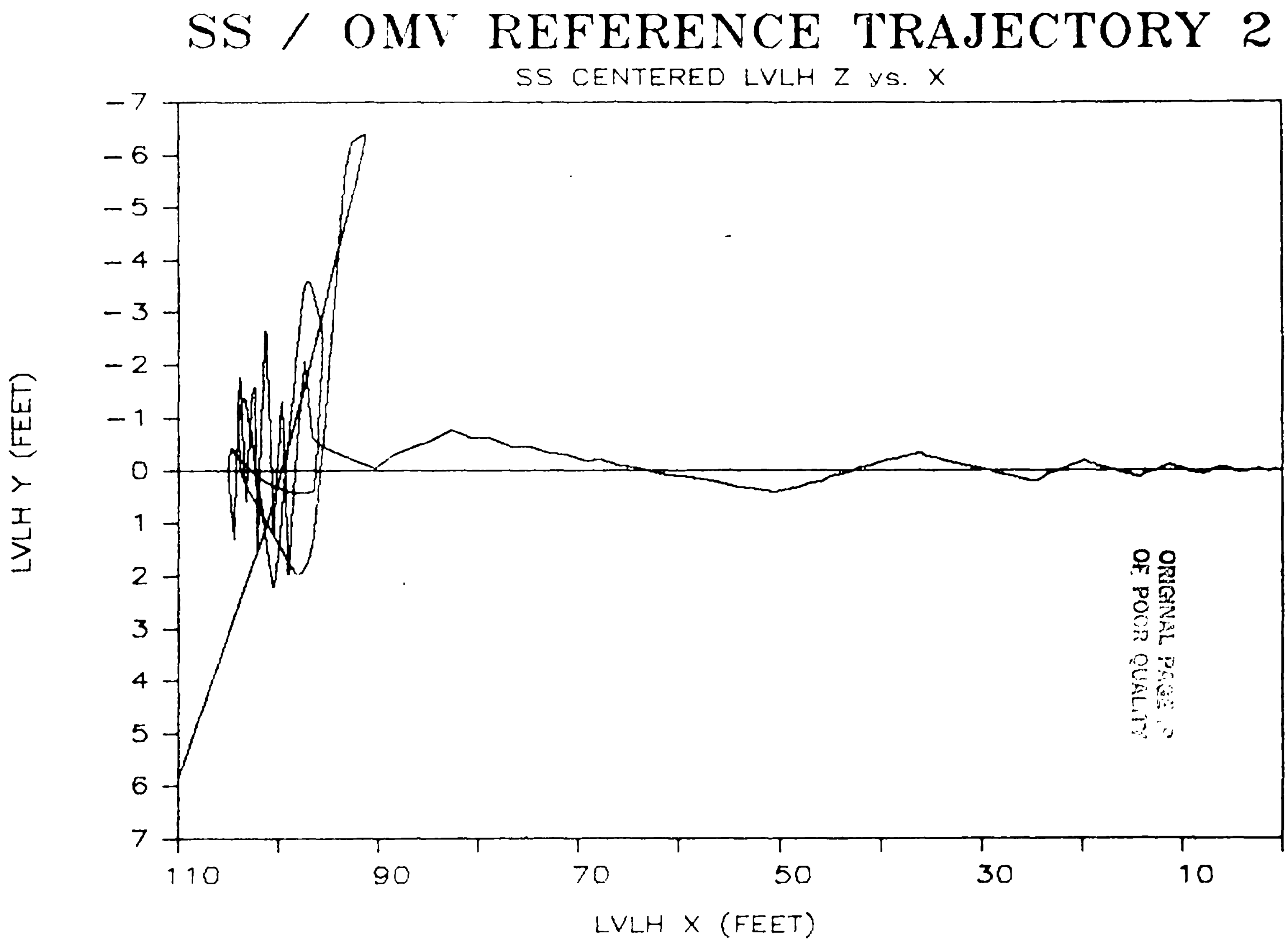
override. Furthermore, assume for this discussion that the port is located on the station's positive Vbar. The return begins with five minutes of stationkeeping at the 1,000 foot offset point, ahead of the Space Station. A controlled set of range rate gates is then flown to a position between 100-200 feet ahead of the station. At this point, stationkeeping may be established to await proper closure conditions or the approach to a dock may be continued. At the docking port, a soft dock occurs, the OMV control system is deactivated and a hard dock is performed.

As with the separation sequences, all maneuvers performed in the vicinity of the Space Station will be designed such that protection of the station and any co-orbiting satellites from plume impingement and recontact is considered, consistent with safety considerations.

A return to a port along the radius or angular momentum vectors would follow closely the approach presented for the velocity vector, with two exceptions. First, after the initial period of stationkeeping at 1,000 feet (behind the station in this case), a targeted 20 minutes transfer to 100 feet behind the station is executed. Then, the following additional statekeeping at 100 feet, a constant range flyaround is performed, aligning the OMV with the port along the radius or out-of-plane vector, as necessary. From this point the profiles are identical except for the port of approach. Returning to an inertially located port also follows the scenario presented above except in this case at 100 feet, the OMV would execute a constant range, Eigenvector flyaround to the port's inertial attitude, prior to beginning its final approach.

Figure 4.11-3 illustrates the relative motion of the OMV with respect to the station for the +Vbar return in LVLH coordinates. The plots illustrate the

maneuvers performed inside eight n. mi, 1,000 feet respectively. Similar motions result for the other return scenarios but plots are not presented here.



(a) Close-in Motion.
Figure 4.11-3. OMV/Space Station Relative Motion for Returns to a +Vbar Port

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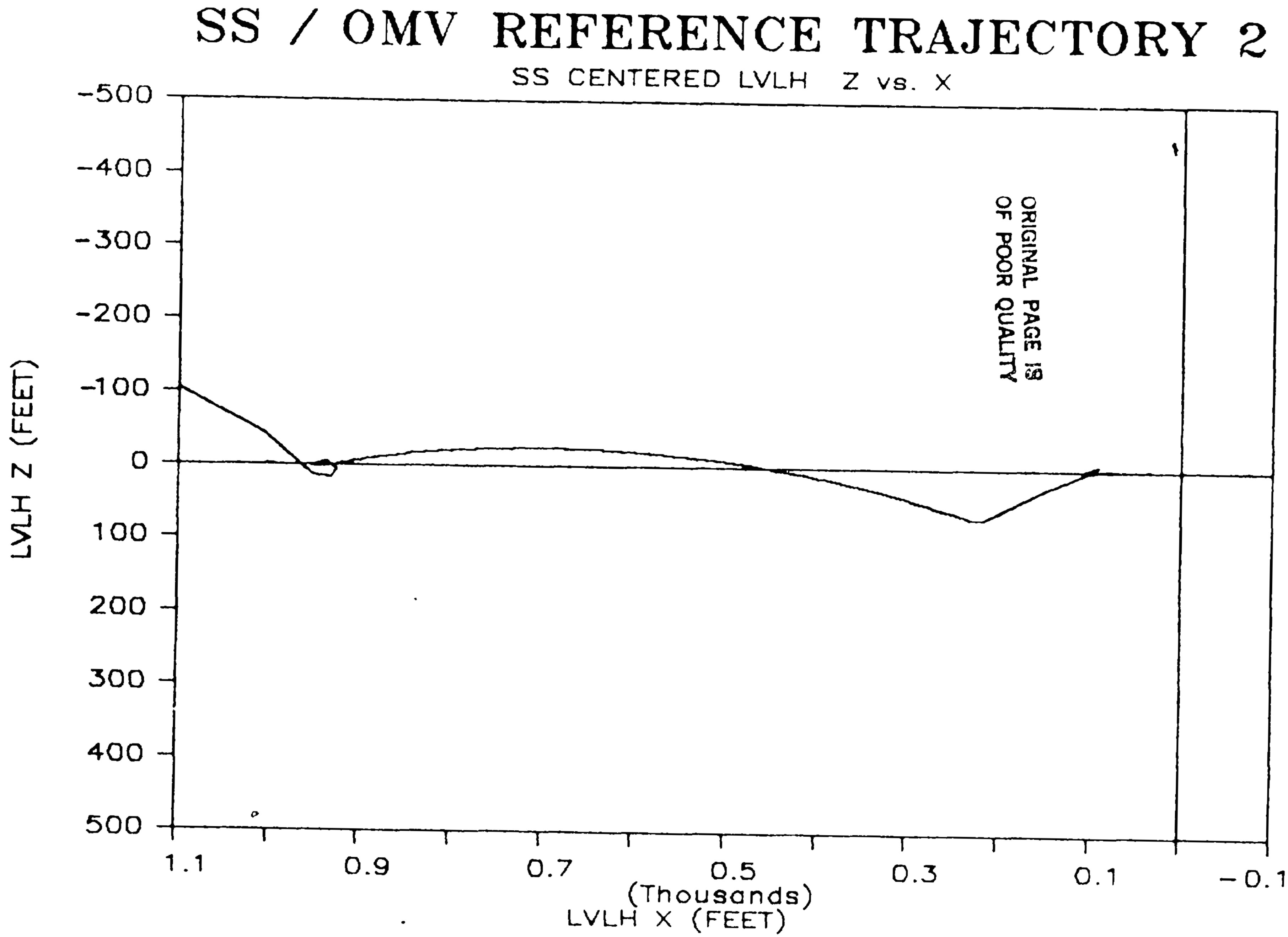
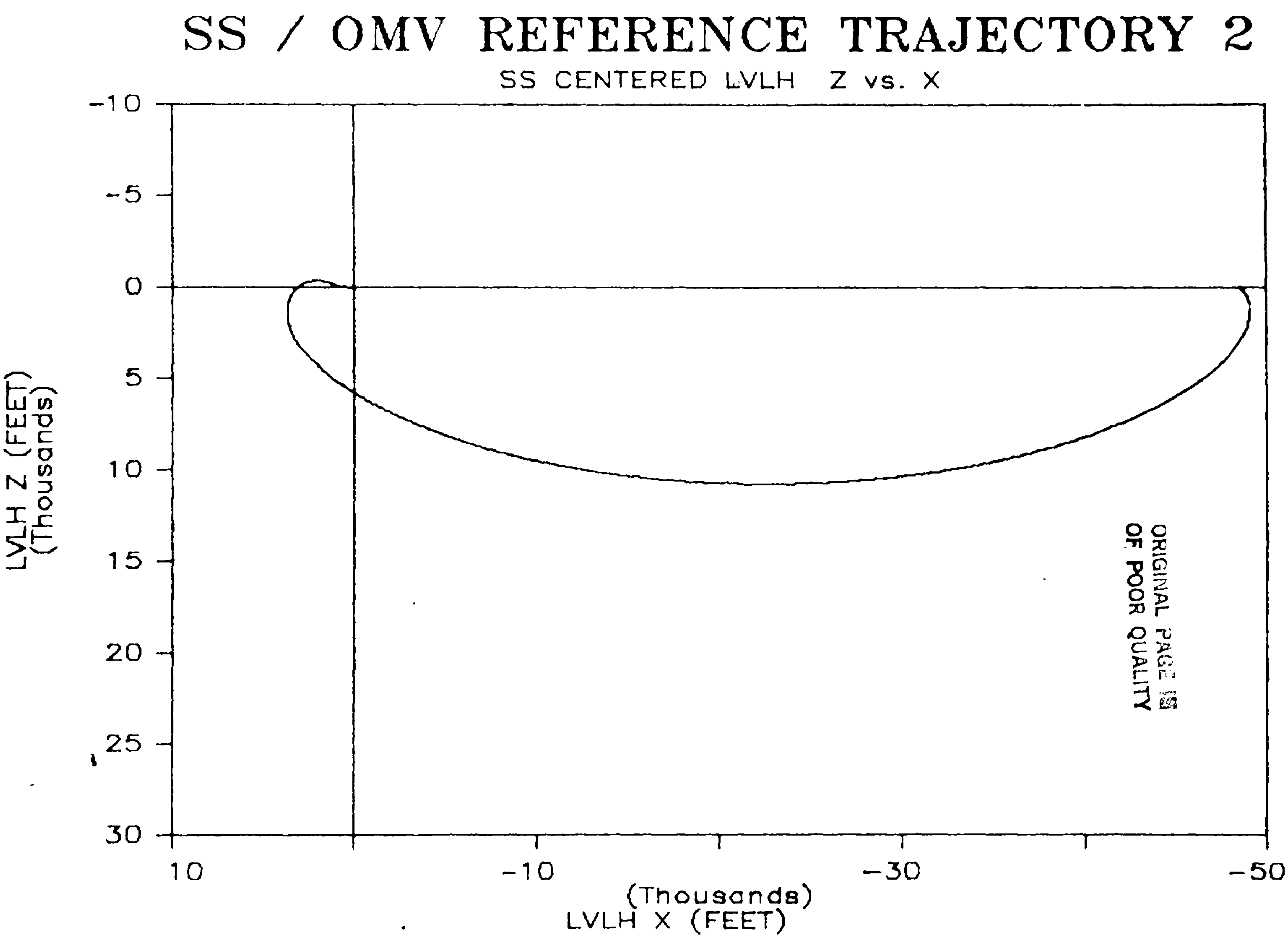


(b) Intermediate Range Motion.
Figure 4.11-3. Continued



(c) Long Range Motion.
Figure 4.11-3. Concluded

5.0 ISSUES

5.1 Introduction

This section provides a summary of the requirements and design issues identified but not resolved during this study. An issue was defined as a consideration where questions existed in one or more of the following areas:

1. Whenever requirements definitions were inadequate.
2. Where options or alternative approaches exist for future study.
3. Where inadequate information exists to permit detailed evaluation.
4. Where development questions such as risk and technology availability exist.

The list of requirements issues applies to all three of the configurations studied. In the list of design issues, the particular configurations (Buidling Block, Delta truss, or big "T") in which the issue was most pronounced is identified.

5.2 Requirements Issues

1. Thermal control for hangars, satellite servicing areas, payloads, and instrument racks.
2. Proximity operations and co-orbiting satellite communications continuous coverage.
3. Station operations power.
4. Payload bay docking module requirements for buildup and operations.
5. Orbiter hard docking/berthing.
6. Module return.
7. Single (44 ft) or double (22-ft) laboratory module
8. Crew activity, equipment, and utility for OMV, OTV, and satellite servicing.

9. Pressurized module viewing.

10. Relationship between platforms and Space Station.

5.3 Design Issues

1. Alternate power source options (solar thermal).

2. Alternate approaches to crew safety (dual egress vs. safe haven).

3. Alternate module arrangements (linear vs. racetrack vs. raft vs. "spoke").

4. RCS location.

5. Fixed vs. oriented boom-mounted radiators - Building Block

6. Deployed vs. erectable structure.

7. Assembly of truss elements - Delta, big "T".

8. Interface definition for other elements with trusses - Delta and big "T".

9. Connecting tunnel interfaces with truss - Delta and big "T".

10. Plume impingement effects.

11. Maintenance of boom rotary joints.

12. Solar array high voltage - plasma interactions.

13. Viewing capability from modules - Delta, big "T".

14. Long distance EVA - Delta, big "T".

15. RMS requirements and implementation.

16. Local shadowing and/or blockage of solar arrays.

17. Use of Orbiter vs. station RMS vs. automatic mechanisms vs. EVA for establishing interfaces during buildup.

18. Sizing of utilities in modules for growth.

19. Number of pressurized ports to be provided on interface modules - Delta and big "T".

5.4 Considerations and Design Issues - Operations Accommodations

In consideration of the analysis presented herein, it is evident that separation and rendezvous/returns are indeed feasible.

Refinements of these techniques and procedures will be evaluated as the Space Station project moves toward an operational era.

The design problem of separation and return lighting constraints was discussed in detail previously. However, a more detailed treatment of lighting considerations will be necessary when particular mission scenarios are evaluated.

A second important issue that will influence the design of these proximity operations procedures is Space Station traffic control. This aspect of station operations leads to additional concerns; the first of these being communication. The communications systems of the various spacecraft, in addition to the station itself, will have range limitations that must be met. Thus, communication constraints will become important design considerations.

The OCZ concept also induces relevant comments that may affect future mission design. For instance, the definite location of a co-orbiting satellite nest, will place restrictions upon the upper stage ignition points of separating vehicles. These points of ignition must be selected such that once the burn is initiated, the resulting transfer trajectory will not carry the vehicle into the co-orbiting satellite zone. Thus, the control zone concept places somewhat of a restriction on the design of the separation procedures proposed for the station.

The sequences presented in this volume are designed such that the separating vehicle ultimately reaches an upper stage ignition point located approximately 10 n. mi. ahead of the station in the designated departure zone. The

restriction of having a single zone of departure located in front of the station makes separations from ports located above and behind the station more complicated than is desireable. In addition, plume impingement is higher than revised procedures would dictate.

Establishing a second departure zone behind the station would exhibit two significant advantages. First of all, a departure from above would become much simpler requiring only an initial separation burn from the port, and a second burn to take it to an upper stage ignition point. A separation from behind would also benefit from a departure zone behind the station. In this case, although a third burn may be required (the second burn, nominally a radial burn, may be be sufficient to put the vehicle at the proper ignition points.), concerns with plume impingement and recontact would be greatly reduced. Separations from ahead of and below the station would be targeted for the zone leading the station as before. However, the burn sequences (including the number of burns required) would parallel those just presented with the same advantages.

The second advantage arising from the implementation of a second departure zone is a reduction in station contamination. In the preceding discussion, the burn used to place the vehicle on a trajectory for its ignition point may be executed by the vehicle in an attitude that allows the burn to be directed away from the station. Therefore, plume impingement on the station is reduced.

The disadvantages arising from the addition of a new zone are threefold. The fact that a rendezvous zone is already established behind the station would warrant additional coordination of separation/return activities in an effort to avoid possible collisions. Also, standardization of separation sequences is hindered. The baselined sequences presented in this volume were

standardized for ports along the three major LVLH axes and in all cases the departing vehicle ended up ahead of the station. The procedures to be employed in the event of addition a second departure zone would be more difficult to standardize because of the possibility of needing additional burns for separations occurring from above or behind. Of course, this conclusion is a function of the number and location of ports on the Space Station configurations.

Finally, additional concern becomes evident with regard to the vehicle orbital transfer trajectories. Placing the vehicles behind the station at upper stage ignition requires that the vehicle travel back toward the station on the transfer trajectory. In this case, the 10 n. mi. separation range may not be sufficient, and a larger OMS-type separation delta V (and/or more coast) would be needed to increase the range. Therefore, final definition of the OCZ limits will be dependent on these trajectory considerations.

The proximity operations that will apply to a nest of co-orbiting satellites is one example of a future issue that is also of importance to Space Station design. For example, consider the procedures that may be required if these satellites are attached to each other via a tether or similar device. It may be necessary to employ entirely different procedures when dealing with co-orbiting satellites.

Another possible study is that of quiescent free-flyer (FF) separations. Here, differential drag effects may be coupled with orbital maneuvers in such a way that a FF will separate from the station at a positive rate, coast out, and eventually begin returning to the station. Problems may arise here in the area of OCZ definitions, i.e., how to control the FF movement so as to keep the vehicles in the "proper" zones.

5.5 Communications and Tracking

A requirement still exists fro obscuration profiles to be performed in order to determine if the location of the antennas selected will provide the coverage needed to meet the link requirements. If blockage does exist, alternate locations or additional antennas may be necessary.

Another issue which still has to be considered, is the location of the antenna and RD equipment and the location of other subsystem hardware such as radiators and solar panels which could offer interference.

A third issue is the firmness configuration orientation and the coverage dictated for operation of the station. Coverage requirement significantly effects antenna system selection and coverage is significantly effected by Space Station orientation.

5.6 User Accommodations

For IOC, is it better to have one 44 foot module or two 22 foot modules? Common equipment such as safety equipment, TV monitors, and life support equipment is needed for two modules. The common equipment could be reduced in one 44 foot module which will be developed by IOC. The reduction of common equipment would reduce weight and volume taken by household equipment and cost. It would allow the users to have more volume for their equipment, thus increasing the flexibility of the user facilities at IOC. This seems feasible, and we would prefer a 44 foot facility while others would prefer a 22 foot. Why not provide a 44 foot test bed to the users at IOC called a multi-discipline manned research laboratory that would stay throughout Space Station life.

6.0 ACKNOWLEDGEMENTS

The Conceptual Space Station Design Study was accomplished by an interdisciplinary team of engineers, designers, and scientists located at the NASA-Johnson Space Center, Houston, Texas. The approach utilized to document the study results was to first develop an outline of the report contents and then assign an individual primary responsibility for preparing a specific section of the report. The assigned individuals were support by a team of contributors representing the disciplines involved in the particular report section. The following lists identify the individuals with primary responsibility for the various report sections and acknowledge the supporting personnel according to their JSC organization.

6.1 Document Section Responsibilities

1.0 Introduction and Background	T. Redding/KA3
2.0 Concept Definition	L. Livingston/ET4
3.0 Concept Function Description and Evaluation	
Configuration Discussion	
User accommodations	C. Robinson/SE
Crew accommodations	C. Dumis/DF4
Engineering	
Assemby and Growth	L. Livingston/ET4
Structural Dynamics and Control	W. Schneider/ES2
Communications	S. Lenett/EE3
Elements/utilities interfaces	J. Hirson/ET5
Thermal	W. Morris/EC2
Power	R. Rice/EP5

	Operation Accommodations	C. Gott/FM
	Configuration Design Considerations	
	RMS Reach Capability	
	Safety Accommodations	T. Edwards/NB
	Cost	R. Whitlock/BN
4.0	S/Systems Definition	
	ECLSS	W. Morris/EC2
	Thermal	W. Morris/EC2
	Propulsion	D. Kendrix/EP4
	Communication and Tracking	S. Lenett/EE3
	Avionics System	T. Barry/EH3
	Structures	W. Schneider/ES2
	Power	R. Rice/EP5
	EVA Support	W. Morris/EC2
	Crew Accommodations	J. Mitchell/EW4
	Power Profile	C. Gott/FA
	Proximity Operations	C. Gott/FA
5.0	Issues	ALL
6.0	Acknowledgements	

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ND/D. Berlad

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