N83-28059

SOC-SE-03-01

Volume IV

Mission
Implementation
Concepts

April 1983

SPACE STATION NEEDS, ATTRIBUTES, AND ARCHITECTURAL OPTIONS STUDY—FINAL REPORT

Approved by:

Sherman Schrock
Program Manager

MARTIN MARIETTA AEROSPACE
DENVER AEROSPACE
P.O. Box 179
Denver, Colorado 80201

Page intentionally left blank

Page intentionally left blank

This final report, submitted to National Aeronautics and Space Administration (NASA) Headquarters, Washington, DC 20546, presents the results of the Space Station Needs, Attributes and Architectural Options Study performed by the Space and Electronics Systems Division of the Martin Marietta Corporation under NASA Contract NASW-3686.

THIS PAGE INTENTIONALLY LEFT BLANK

TABLE OF CONTENTS

CONTENTS

CONTENTS

FIGURES

Page

VOLUME IV MISSION IMPLEMENTATION CONCEPTS (cont)

Page

VOLUME IV MISSION IMPLEMENTATION CONCEPTS (cont)

Page

VOLUME IV MISSION IMPLEMENTATION CONCEPTS (cont)

THIS PAGE INTENTIONALLY LEFT BLANK

1.0 INTRODUCTION

1.1 PURPOSE

The purpose of the Mission Implementation Concepts task is to define space station capabilities, characteristics, and configurations required to support potential user payload needs. In addition, this task will evaluate and define a cost effective schedule for implementing and evolving space station capabilities in response to time - phased user needs.

1.2 SCOPE

The primary purpose of this study was to identify, collect, and analyze the science, applications, commercial, U.S. national security and space operations missions that would require or be materially benefited by the availability of a permanent manned space station in low earth orbit and to identify and characterize the space station attributes and capabilities which will be necessary to satisfy these mission requirements. Emphasis is placed on the identification and validation of potential users, their requirements, and the benefits accruing to them from the existence of a space station, and the programmatic and cost implications of a space station program. Less emphasis has been placed on detailed design beyond that necessary for the identification of system attributes, characteristies, implementation approaches and architecture options, and ROM costs.

The study results are presented in six volumes as follows:

Volume I presents an executive summary highlighting the specific results obtained during each phase of the study as described in Volumes II through VI (classified information excepted).

Volume II presents the results of our mission definition activities including the identification, modeling and validation or potential user missions, their requirements and the benefits that could accrue to the users from the existence of a space station.

Volume III presents the space station user requirements, their integration and time phaseing, and the derivation of system and user accommodation requirements. The derivations of user requirements and space station accommodations encompassed a traceability analysis, parametric studies, and an analysis of economic, performance, and social benefits afforded by th existence of a space station.

Volume IV presents the results of our study efforts describing our analyses and defining our recommended space station implementation approaches, architecture options, and evolutionary growth.

Volume V presents the affordability analysis conducted to determine the affordable mission model, quantification of economic benefits, estimate of the ROM costs for each of the architectural options and their associated program and element schedules.

Volume VI presents the results (classified) of our analysis for the DOD National Security mission. This volume was published under a separate cover and is available through the DOD Task Manager at Space Division (SDXR), Los Angeles, California.

The scope of the Mission Implementation Concepts task is clearly illustrated in Figure 1.2-1, the Space Station Study Flow. Activities related to this task are shown cross-hatched. These activities include the definition of top level space station program options compatible with previously generated integrated user mission requirements and mission analyses. Optional architectural configurations and approaches are then derived which lead to the definition of space station characteristics, initial capabilities, capability growth increments, and mature station capabilities in the year 2000.

1.3 GROUND RULES AND ASSUMPTIONS

The following ground rules and assumptions have been used as a basis for the activities performed under this task.

a. Space station IOC is scheduled for early 1991.

b. Space station IOC represents the initiation of manned activities onboard the station.

c. Early elements of the space station may be implemented on-orbit as early as mid-1990, but are not intended to support space station operations before IOC.

d. The maximum number of annual STS flights will not exceed 40, and not all of these will support space station operations.

e. Anticipated early year crew size, overall activities profile, and capability constraints limit OTV launches to two per month.

1.4 CONSTRAINTS

Any constraints related to activities performed under this task have been included as part of the ground rules and assumptions presented in Section 1.3.

1.5 DEFINITIONS

The following definitions are particularly germane to this task.

a. Program Option - A top level, time-phased plan for implementing and evolving space station capabilities based on preselected criteria intended to differentiate between various options.

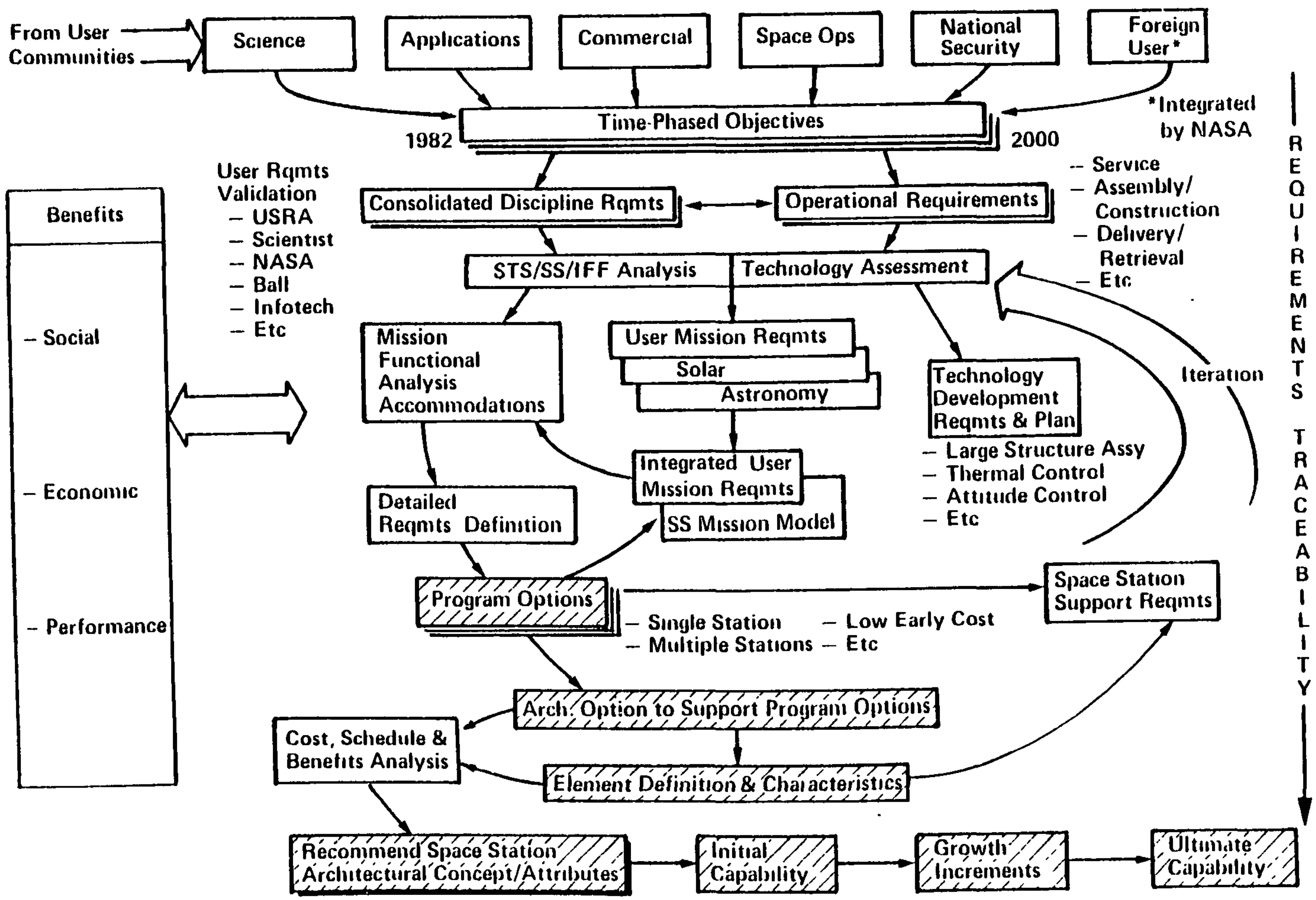


Figure 1.2-1 Space Station Study Flow

b. Viable Program Option - A program option which, based on cost and user support analyses, is considered to be one of the better program options deserving of continuing in depth evaluation.

c. Manned Space Station - Those on-orbit elements of the overall space station capabilities which house and support the space station crew. The manned space station may also include elements and user payloads not directly related to crew support.

d. Platform - An unmanned collection of elements and supporting subsystems intended to support multiple payload operations.

   A platform may operate continuously in the vicinity of the space station, in which case a monitoring and control data interface will be maintained via RF link with the space station.

   Platforms may also operate remotely, i.e. out of RF line-of-sight, from the space station, but still be close enough for servicing operations supported by the space station.

e. Shuttle Derived Vehicle (SDV) - A large cylindrical vehicle, unmanned at launch, which is launched attached to the external tank (ET) in the same manner as the Orbiter. The SDV contains approximately 30000-40000 ft^3 of pressurized volume outfitted with floors and compartments for subsequent manned, orbital operations, 20000-30000 ft^3 of volume for hangar operations, and provisions for the docking of additional modules on-orbit.

f. Architectural Option - A design approach which identifies major space station elements and capabilities, defines functional roles of the elements, evaluates physical configuration to assure orbital stability and an efficient evolution process, and supports the sizing of significant space station capabilities and characteristics.

1.6 APPROACH METHODOLOGY

The approach used to accomplish the Implementation Concepts task is presented as a flow diagram in Figure 1.6-1.

Based on user requirements contained in the Composite Mission Model described in Section 3.2 of Vol. III, early estimation of performance and economic benefits, and additional systems analyses, a series of eight program options was defined which are described in Section 3.0. Each program option was based on pre-selected criteria such as the need for one or two manned space stations and the orbital inclincation at which they would operate. Supporting each program option was a plan for the evolution of space station capabilities in response to time-phased user needs derived from the composite mission model. These options were then subjected to cost, schedule, and user support analyses to identify those which exhibited the best combination of low overall cost and high level of user support.

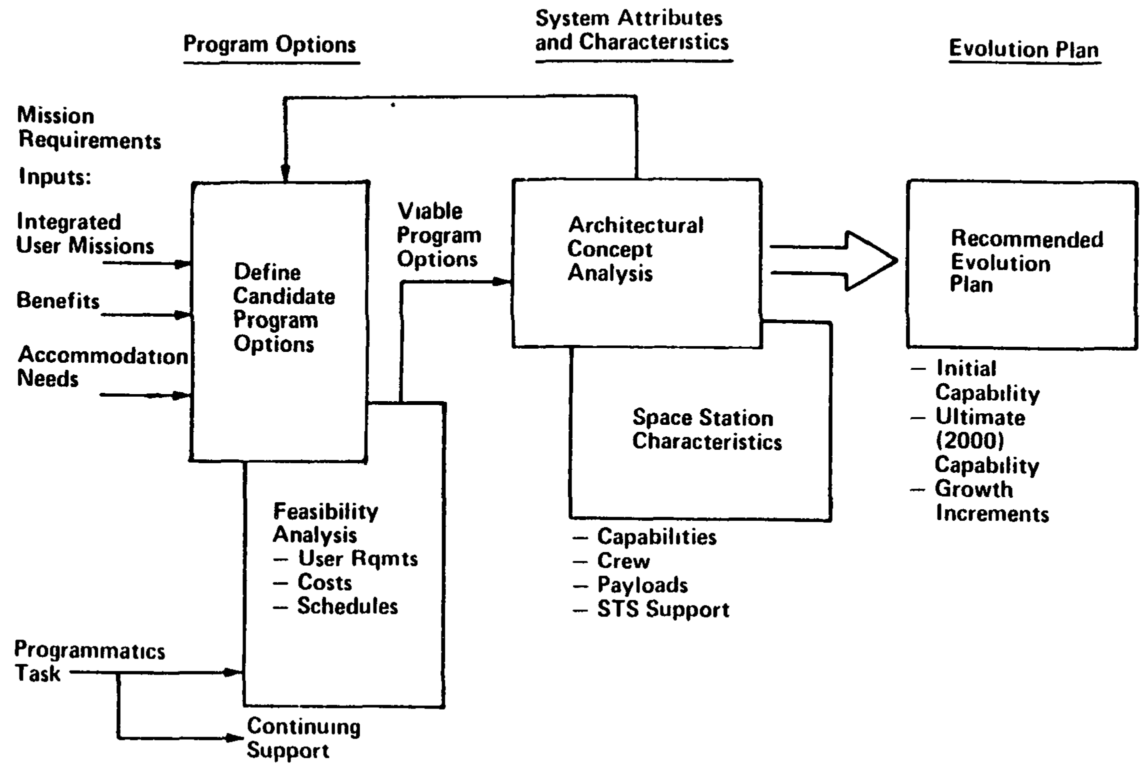


Figure 1.6-1 Implementation Concepts Flow Diagram

Two of the program options considered to be the most viable were then analyzed in greater depth. For each of these viable options, an evolution plan was developed in greater detail.

Additional details on user needs were obtained by descussions with each of the payload discipline specialists and decisions were made regarding the optimum accommodation for each pyalod, and the specific support to be provided by the space station.

With the information developed to date, we then defined and evaluated various architectural options or configurations for the space station. Three different architectural approaches were analyzed as described in Section 4.0. This activity was supported by subsystem requirements analyses and subsystem conceptual design, with emphasis on those subsystem areas which had the greatest station configuration impact. In addition, significant attention was given to crew considerations such as habitability, activities, safety, sizing, and resupply needs.

Following this, the evolution plan was finalized as summarized in Section 7.0. This effort included a recommendation on the required initial space station capability, incremental growth steps, and the ultimate station capability in the year 2000.

2.0 EXECUTIVE SUMMARY

2.1 PROGRAM OPTIONS

Eight top level program options for implementing and evolving space station capabilities were defined and subjected to analyses in the areas of user support, evolution, life cycle cost, and schedule compatibility. The eight varied in terms of the number of manned space stations, either one or two, operating in conjunction with appropriate payload platforms, and the inclination angle at which the stations operated.

The initial program options were:

a. Option A-1, single station at 28.5° with early OTV capability

b. Option A-2, single station at 28.5° with delayed OTV capability.

c. Option A-3, single station at 57° with early OTV.

d. Option A-4, single station at polar orbit with early OTV.

e. Option B-1, early station at 28.5° followed by station at polar orbit in mid 1990's.

f. Option B-2, early polar orbit station followed by 28.5° station in mid 1990's.

g. Option B-3, an early shuttle derived vehicle station at 28.5° followed by polar orbit station.

h. Option C-1, low front end cost approach.

Cost analyses showed that the four lowest cost options were A-1, A-2, A-3 and C-1. By comparison, the four options providing the highest level of user support were options A-1, A-3, B-1, and B-3. This naturally focused attention on options A-1 and A-3 as those considered to be the most viable approaches. In depth user support and evolution analyses were continued with these two options.

The user support analyses indicated that option A-1 will support 79% of the non DOD user missions while option A-3 will support 64% of these missions. Another important factor considered was the fact that option A-1 provided more cost effective support to the largest user class, commercial communication satellites operating at GEO orbit. The results of those analyses and trade studies led to our recommendation that program option A-1 was the optimum space station approach. An important factor in that recommendation is the early availability of the proposed retrievable and space maintainable OTV.

2.2 ARCHITECTURAL OPTIONS

The architecture option studies (section 4.0) resulted in a series of major architectural decisions, the presentation of three space station configurations, and a cursory overview of the space station platform concept.

Working from the top level "given" requirements and the space station mission model results, key trade studies issues, architecture related, were identified. Recommended approaches were selected based on both subsystem analysis (section 6.0) and rationale derived herein. These decisions were then utilized as a common basis for the configuration development.

Two modular space station configurations were developed, one based on STS cargo bay delivery, and the other making use of the cargo bay plus the additional volume afforded by the external tank/aft cargo carrier. A third configuration is based on the shuttle derived vehicle concept.

Our cargo bay (14' diameter) modular design is based on the premise of maximizing commonality between elements and the logic of phased growth. Figure 2.2-1 illustrates the modular design at a mature development stage (approximately 1995). Highlights of the approach include; STS compatibility, commonality, a phased growth approach, and having allowances for unplanned future growth. The major disadvantages associated with this design are; the number of STS flights required to reach a mature configuration, and the complexity involved with the build up and assembly.

An aft cargo carrier concept (ACC) was developed after it became apparent that the STS transportation costs involved with building the station were appreciable, and that many of the STS payloads are volume limited. The ACC approach provides additional volume (12,000 ft^3) which not only permits the transportation of extra elements on a single STS flight, it also allows for elements up to 25 feet in diameter. Figure 2.2-2 presents this configuration. With this approach at least two STS flights involved with building the station can be saved. Other advantages include the use of larger diameter building blocks and retaining the phased growth approach. This configuration also is capable of future growth. ACC disadvantages include the build up complexity previously mentioned, and the cost of developing a new module size.

A space station configuration based on the shuttle derived vehicle payload carrier is illustrated in Figure 2.2-3. This unique approach permits a savings of 3-5 STS flights (build up phase), and achieves a large pressurized volume (mature station requirement) in a single launch. Advantages associated with the SDV station are; reduced

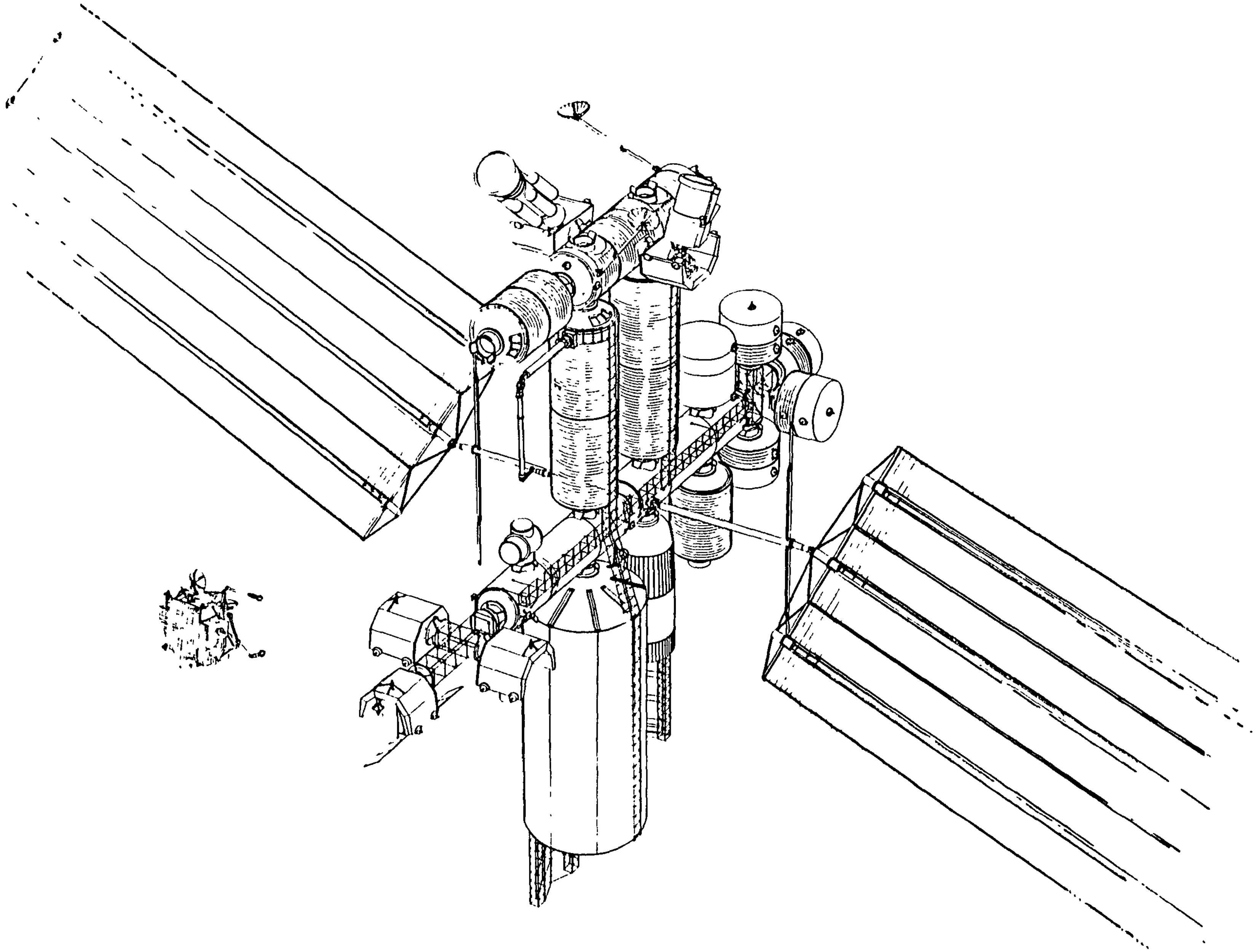


Figure 2.2-1 14' Diameter Modular Space Station

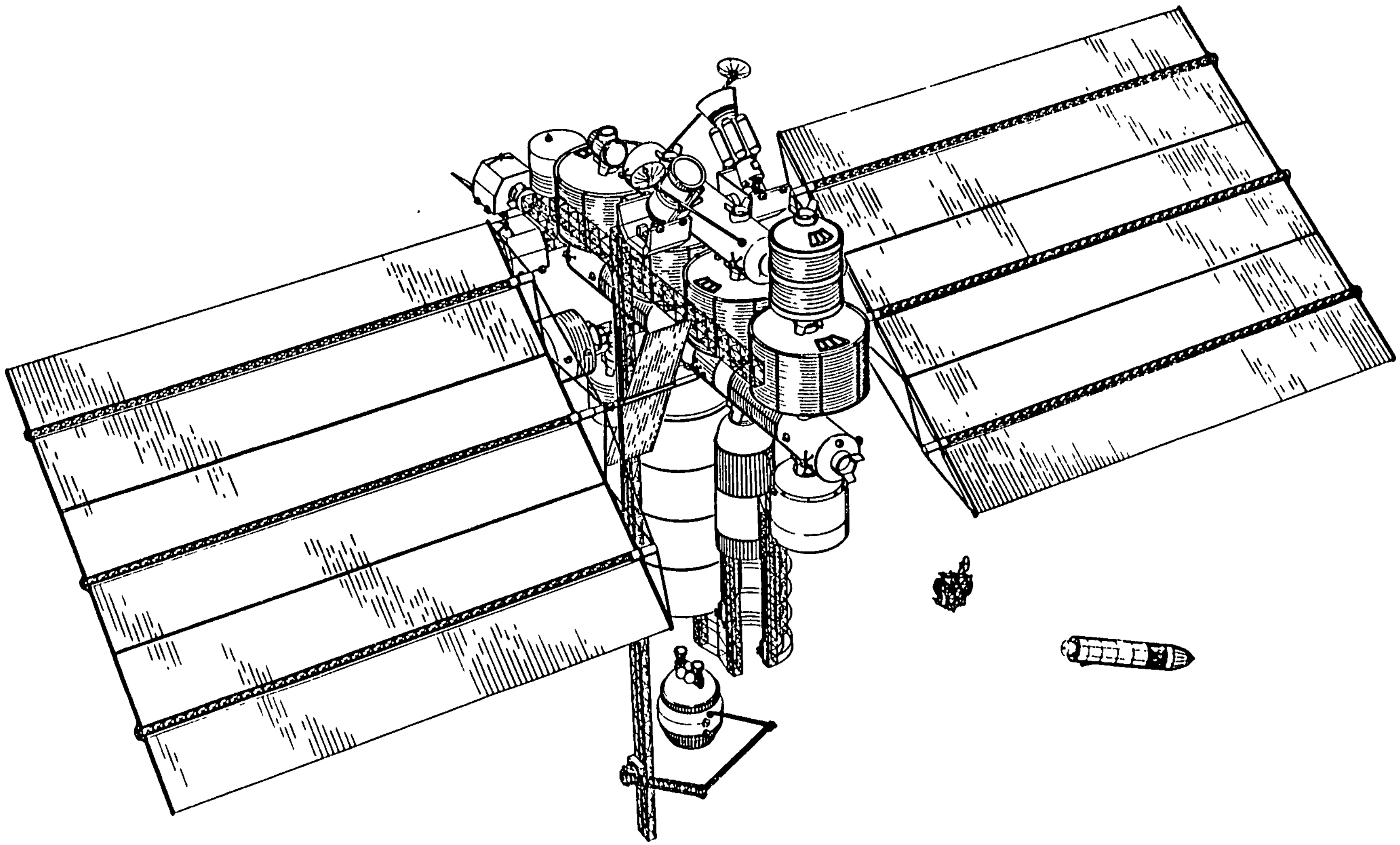


Figure 2.2-2 Modular Aft Cargo Carrier Space Station

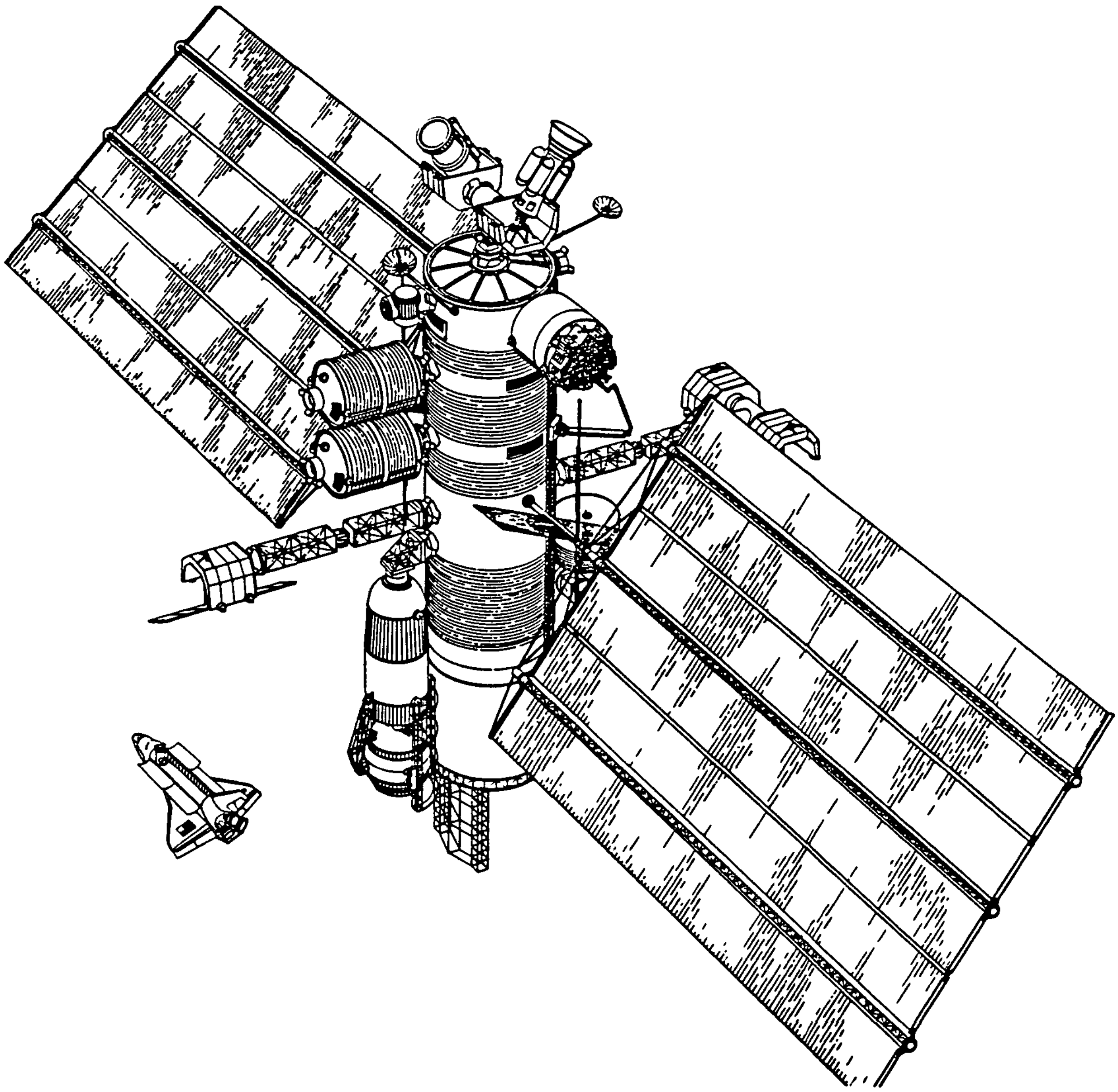


Figure 2.2-3 Shuttle Derived Vehicle Space Station

transportation costs, significant early capability, and crew safety at the initial phase. Reduced growth capability and a commitment to the launch era technology are potential disadvantages.

A limited amount of time was spent on platform designs, but the major conclusion is the selection of five to six platforms including two (astronomy and materials processing) that are colocated with the space station. A preliminary design approach would be to use the MSFC space platform design concept, since compatibility exists between the space station platform requirements and the space platform capabilities.

2.3 MAN/SYSTEMS INTEGRATION

This area of the implementation concepts task addressed such crew related areas as: (1) environmental control-life support (ECLS), (2) EVA operations, (3) social-psychological factors, (4) medical needs, (5) SS pressure, and (6) crew resupply quantities.

The early space station will include an ECLS based on current orbiter technology, and with a limited regenerative capability. This approach in the early years increases reliability, and reduces crew time required for maintenance of the ECLS. Early regenerative capability will be limited to a CO_2 removal system and a condensate water clean-up system to provide hygiene water. Space station evolution and associated crew growth will drive an increasing regenerative capability to avoid sizable ECLS resupply launch costs. The next major evolutionary step will be to incorporate a waste water processing system to provide additional hygiene water for shower and clothes washing. Oxygen and water loop closure will be accomplished in a third major step by addition of a CO_2 reduction system, an O_2 generation system, and additional waste water processing equipment to include water recovery from urine.

The anticipated level of crew EVA activities required for space station integration and maintenance, and for payload large structures assembly dictate that improvements in the current Orbiter extravehicular mobility unit (EMU) be implemented. The primary need is for a higher pressure suit which can avoid totally the need for prebreathing before an EVA, which would require 3.5 hours with the EMU. Based on a Space Station operating pressure of 12-14.7 PSIA, a 6 to 8 PSIA suit would eliminate the pre-breathing requirements. The current EMU operates at 4.3 PSIA. Another improvement would be the elimination of water discharge from the Portable Life Support System (PLSS). In the vicinity of the space station, this discharge, currently at the rate of 1.7 lbs/hours, would present a serious contamination problem for a number of scientific payloads. Finally, the EVA suit component and operational lifetime must be extended. The current EMU is refurbished after 5 EVAs and has a useable life of 30 EVA's. A more appropriate capability would be an operational life of 6000 EVA hours and provision for on-orbit refurbishment.

Considering the social-psychological factors, crew problems may arise for the following reasons: 1) duration of orbital stay, 2) crew inter-relationships, 3) heterogeneous nature of individual crew member backgrounds and assignments, and 4) constraining physical environment of the space station. These factors can result in adverse crew stress reactions leading eventually to decreased performance of assigned tasks. We have proposed a social-psychological design approach which recommends consideration of SS volume requirements, group organization, flexible activity scheduling, cross-training in assignments, and stress management techniques.

A specific space station operating pressure level is not recommended at this point. Section 5.5 discusses the numerous diverse factors that influence selection of an operating pressure. Based on evaluation of these factors a compromise range of pressure levels is suggested, and a tighter control of the pressure level is recommended.

Data were summarized on currently accepted estimates of crew and ECLS resupply needs, EVA suit rotation, and EVA resupply requirements.

2.4 SUBSYSTEM CONCEPTS

Emphasis was placed during the performance of our subsystem analyses on identifying and sizing subsystems which had a direct influence on space station evolution, configuration, or stability. In addition projected technology state-of-the-art required to satisfy a 1991 IOC date as well as future technology development was given serious consideration in all subsystem areas. The following data summarize significant trade study results for the various subsystems.

a. Electrical

- o Reqts. range from 33.5 Kw (IOC) to 78 Kw (1995-2000) at bus
- o Solar array power required at BOL is 75 Kw (IOC) and 187 Kw (1995) with associated size of 6400 ft^2 increasing to 17000 ft^2.
- o Silicon cells selected over GaAs for IOC
- o Modular design includes NiH_2 batteries and 120-160 VDC bus

b. Propulsion

- o Hydrazine used for SS orbit maintenance and attitude control
  - Uses 8 boom-mounted 30 LBM thrusters
- o Hydrazine storage (15000 lbs) in logistics module used to resupply TMS
- o Cryogen storage of 70000 lbs provided to resupply OTV

c. Thermal Control

- o Conventional redundant, pumped heat transport loop (orbiter technology) with body-mounted heat pipe radiators
- o Augment with deployed heat pipe radiator panels if required
- o Consider subsequent upgrading to two phase heat transport loop

d. Attitude Control

- o Gravity gradient attitude control of pitch and roll axes
  - Provides coarse stabilization
  - Fine pointing provided by payloads
- o Early configuration may augment RCS with CMGs
- o Orbital rate (pitch axis) provides gyroscopic stabilization in yaw and roll axes

e. RF Communications

- o RF links possible at UHF, L, S, & Ku bands; at 40-60 GHZ; and at laser wavelength
- o Numerous interfaces with EVA, Orbiter, TMS, OTV, TDRSS, platforms, STDN, DOD
- o Maximum antenna diameter is less than 15 ft

f. Data Processing

- o End-to-end system interfaces SS data bus with ground processor(s) data bus
- o Distributed architecture
- o Adapt commercial, ground systems/concepts to SS use
- o Estimate 50 Mbps data bus and 10^6 FLOPS for some processors
- o Dedicated signal processors and fiber optics interfaces for high rates (in excess of 50 Mbps)

2.5 EVOLUTION APPROACH

An evolution approach was developed for each of the eight candidate program options to provide a basis for subsequent user support, cost and schedule analyses. Following selection of the 28.5° space station option, a more detailed evolution plan was defined, and is presented graphically in Figure 2.5-1.

Significant characteristics of the evolution plan include:

a. Initial launch of unmanned elements about mid-1990.

b. Space station manned IOC occurs early in 1991.

c. As many as ten user payloads in five user categories will be located and operated from the station in its first year of operations.

d. Crew size will grow from four people in 1991 to 10 or 11 people by 2000.

e. OTV operations will be implemented as early as possible, presently scheduled for 1992, to capture sizable benefits gained from the delivery of commercial satellite to GEO.

f. Implementation of a materials processing platform (1993) and combined astronomy/solar physics platform (1994) in the vicinity of the station, and the ISTO/ASO platform (1993) at 57 degrees.

g. Special user support in the form of a materials processing laboratory and limited production facility, and a life sciences research laboratory supporting plant and animal research.

h. Large structure assembly at LEO and transfer by OTV to GEO occurring in the late 1990's.

i. Subsystem growth in terms of capability and technology to support space station growth and increasing user support.

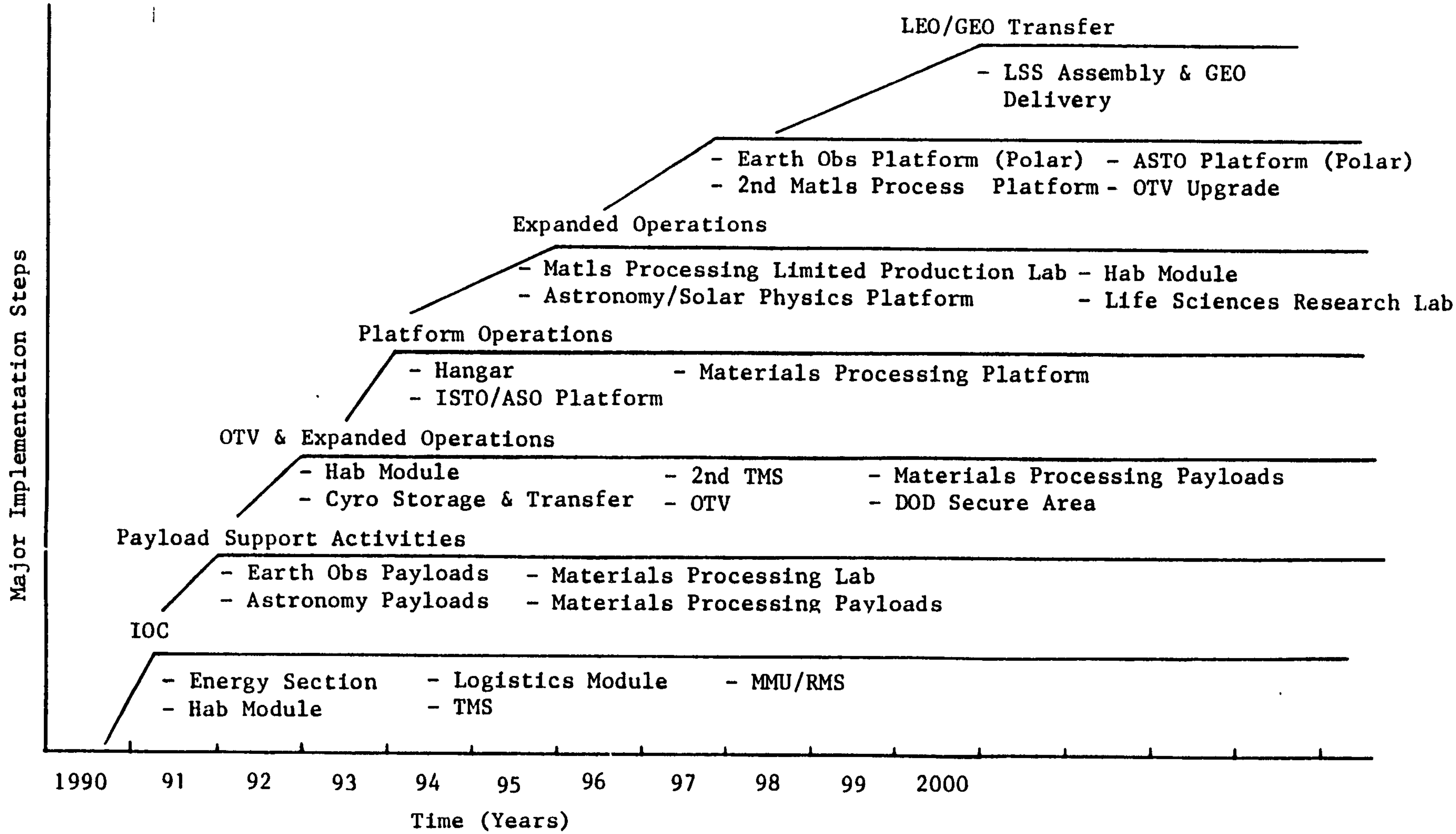


Figure 2.5-1 Recommended Evolution Plan

THIS PAGE INTENTIONALLY LEFT BLANK

3.0 PROGRAM OPTIONS

3.1 PROGRAM OPTION DEFINITION

As indicated in section 1.5, we have defined a program option as a top level plan for implementation and evolution of space station capabilities. Each program option is constrained or driven by certain assumptions which differentiate it from other options. The two primary differentiation factors we have used are the number of manned space stations, either one or two, and their operating orbital inclination angle. Each defined program option was then analyzed in light of the composite mission model.

Time phased, integrated user missions derived from these analyses resulted in a time-phased plan for implementing space station capabilities required to optimize support to user missions.

Eight candidate program options were identified, as listed in Figure 3.1-1. The category A options assume that only a single manned space station will be implemented in conjunction with a number of unmanned payload platforms. Category B options assume that two manned space stations will be implemented at different orbital inclinations, also operating in conjunction with unmanned platforms. Category C contains a single option which emphasizes a low front end cost approach.

In order to derive an evolution plan for each of these eight options, a summary user requirements data base was extracted from the composite mission model; and this summary is presented graphically in Figure 3.1-2. This graph plots the cumulative number of user missions in four disciplines: commercial communication satellites, science, DOD, and applications or materials processing. Each of these disciplines is also broken down by LEO inclination or GEO operations.

Section 3.3 will update this analysis in greater detail for the prime program options, and will take into consideration the user mission affordability criteria applied to the composite mission model data.

3.1.1 Program Option Evolution

This section will describe the evolution plan proposed for each of the eight program options to support the user needs summarized in Figure 3.1-2. Section 3.2 will then present results of the user capture analyses performed, which indicates the percentage of user missions which can be supported by each program option.

It should be noted that this data is presented to more completely describe our overall study approach. Having been completed early in the study, the user requirements and related implementation results are preliminary. More current and pertinent program option and evolution data are presented in sections 3.3 and 3.4.

Category A - Single Manned Space Station Plus Unmanned Platforms

A-1	-	28° Station, Early OTV
A-2	-	28° Station, LEO Support
A-3	-	50° - 57° Station
A-4	-	90° Station

Category B - Two Manned Space Stations Plus Unmanned Platforms

B-1	-	Initial Station at 28°
B-2	-	Initial Station at 90°
B-3	-	Shuttle Derived Vehicle

Category C - Special Emphasis

C-1	-	Low Front End Cost

Figure 3.1-1 Candidate Program Options

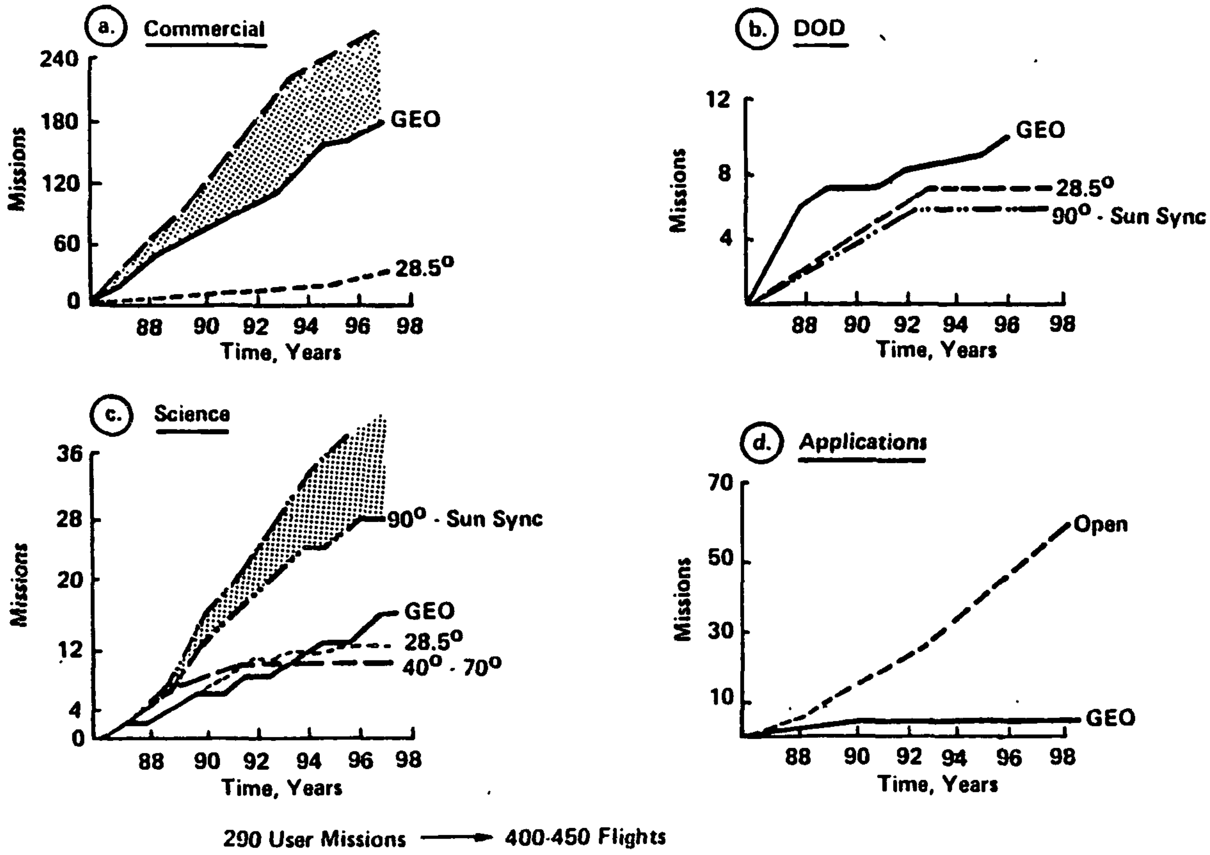


Figure 3.1-2 Requirements Data Base

Figure 3.1.1-1 illustrates a recommended evolution approach for option A-1, which consists of a single manned station at 28.5 degrees. As indicated, IOC capabilities include a crew of four, a TMS along with propellant storage, limited servicing capability, and early payloads. This option recognizes the high volume of commercial communications satellite activity indicated in Figure 3.1-2, and recommends the earliest possible implementation of an OTV capable of LEO plane transfer as well as LEO-to-GEO transfer. To be cost effective, this OTV would be space refuelable and maintainable; and would include some degree of aero-assist capability. A proposed OTV to satisfy this need is described in Vol. III, section 6.0. OTV operations will require cryogen propellant storage and resupply capability, a hangar for storage as well as OTV checkout and payload integration, and a likely increase in crew size. As suggested in figure 3.1.1-1, this early station capability will significantly reduce project STS support to payloads operating at or near 28.5 degrees and for the communications satellites to be delivered to GEO. The next steps in the evolution plan are the implementation of platforms operating in the vicinity of the space station, which support materials processing and science payloads. Following implementation of these platforms with multiple payloads onboard, it then is essential to expand servicing capabilities at the station to properly support these users.

Option A-2 is very similar to option A-1, except that availability of the required OTV is delayed until the late 1990's. Figure 3.1.1-2 portrays this revised evolution approach and shifts early emphasis of the station operations to LEO payloads.

Option A-3, as shown in Figure 3.1.1-3, suggests a single station operating in the range of 50-57 degrees, and using the OTV to support payloads operating between 28 and about 80 degrees.

The final single station option, A-4, as indicated in Figure 3.1.1-4, concentrates operations near polar orbit and is not capable of providing support to lower inclination payloads.

Option B-1, shown in Figure 3.1.1-5, presents the first dual station approach, implementing the first manned station at 28.5 degrees and the second station at polar orbit in the mid 1990's. It was assumed that funding of two stations would preclude early development of the retrievable OTV, and delay its implementation until the late 1990's. Option B-2 is described in Figure 3.1.1-6 and considers implementation of the polar station ahead of the low inclination station.

A somewhat more unique option is suggested by option B-3 and illustrated in Figure 3.1.1-7. This option proposes that the first station implemented be based on the SDV approach. In a single, early STS launch the SDV capability delivered to orbit includes 30000 - 40000 ft^3 of pressurized compartment volume, and 20000 to 30000 of potential hangar volume. The SDV architecture is described in detail in section 4.2, and its launch configuration and possible internal configuration are illustrated in Figure 3.1.1-8.

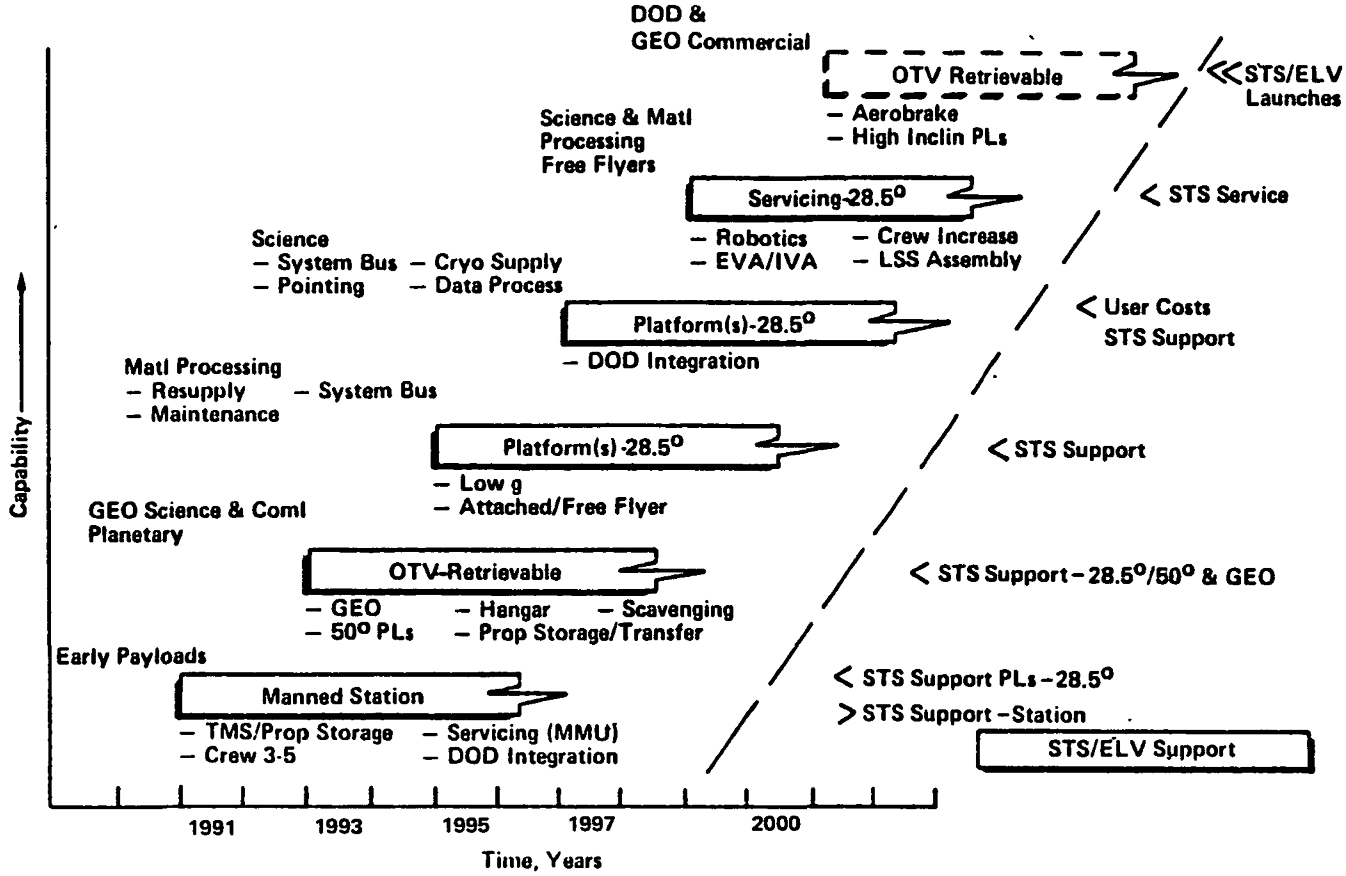


Figure 3.1.1-1 Option A-1: 28.5°-Early OTV

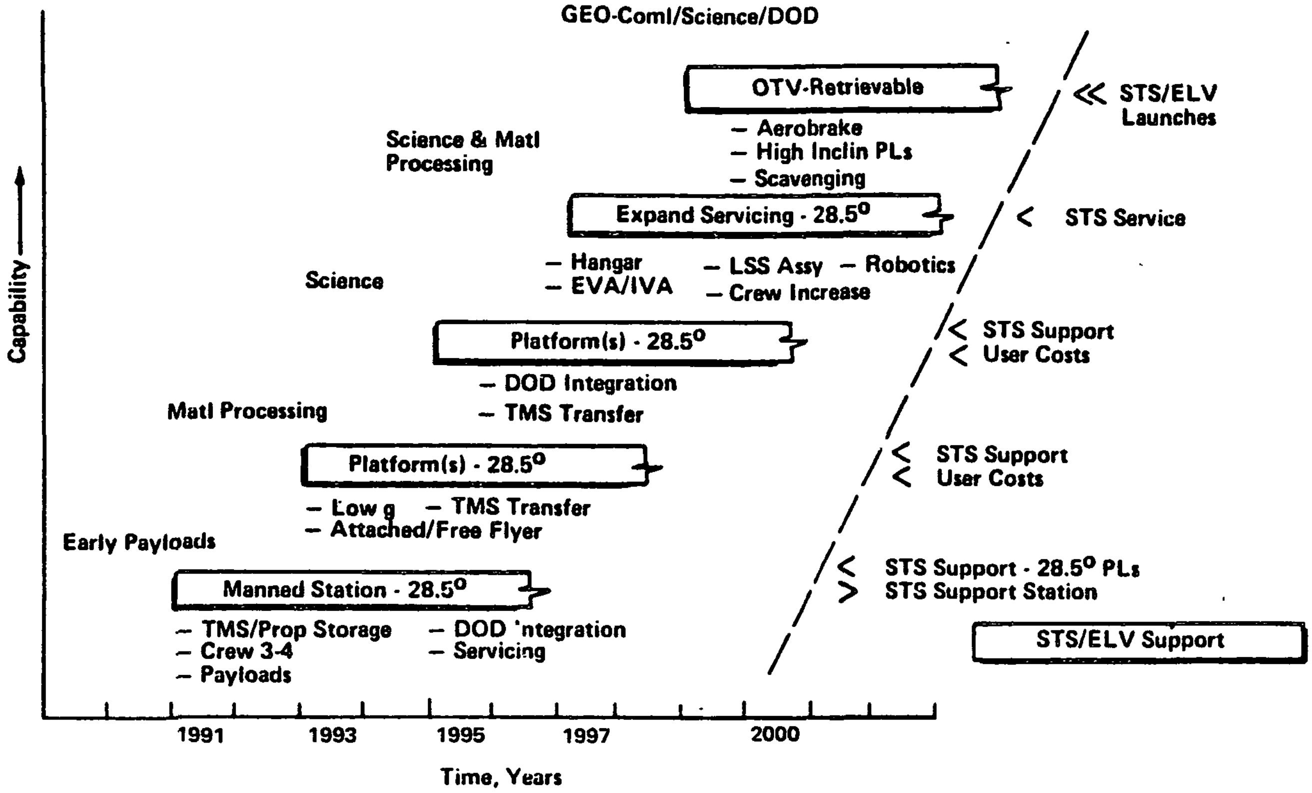


Figure 3.1.1-2 Option A-2: 28.5° LEO Support

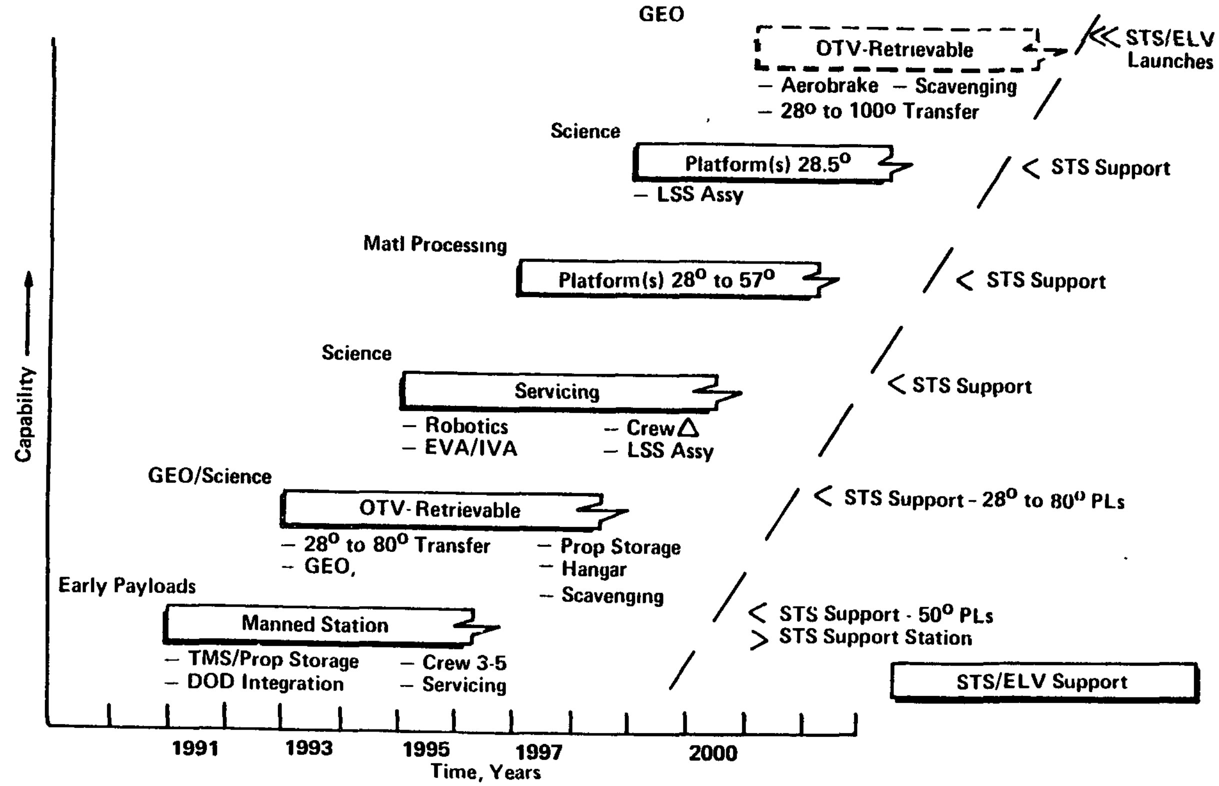


Figure 3.1.1-3 Option A-3: 50° to 57° Station

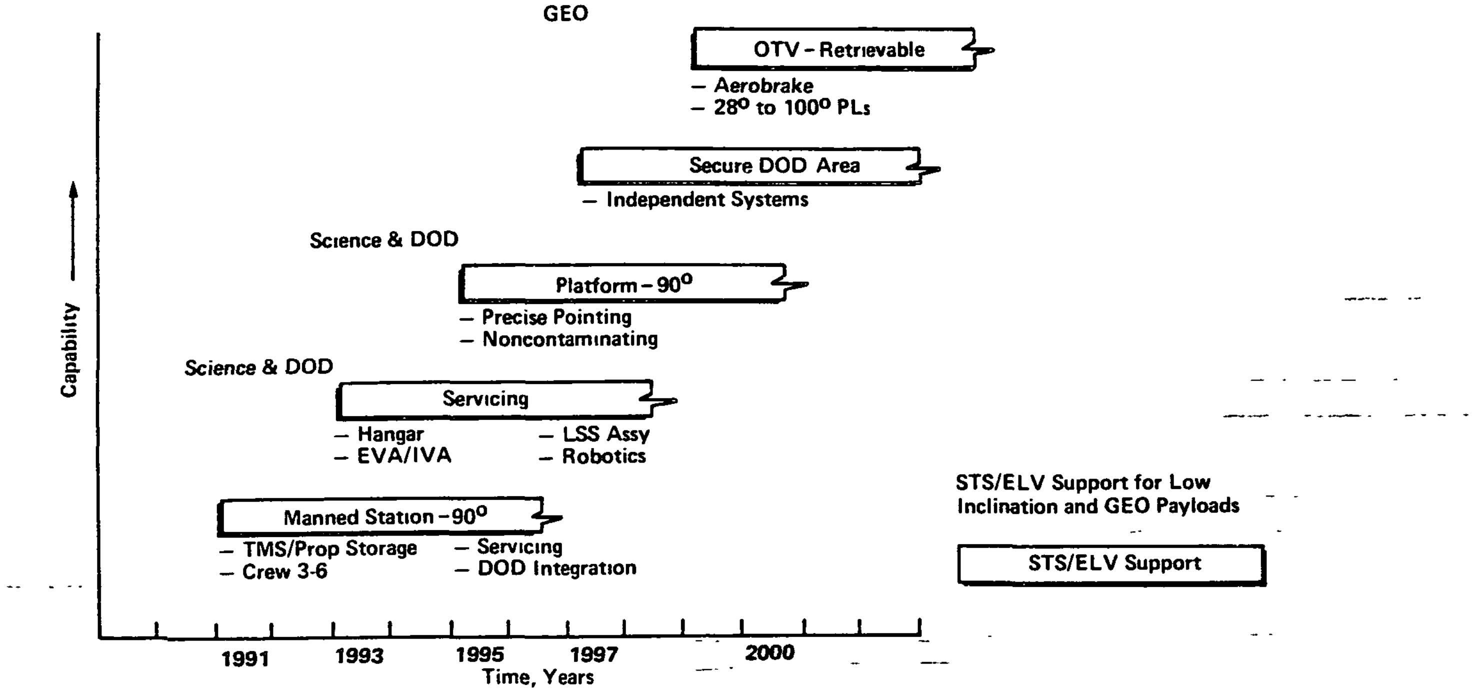


Figure 3.1.1-4 Option A-4: 90° Station

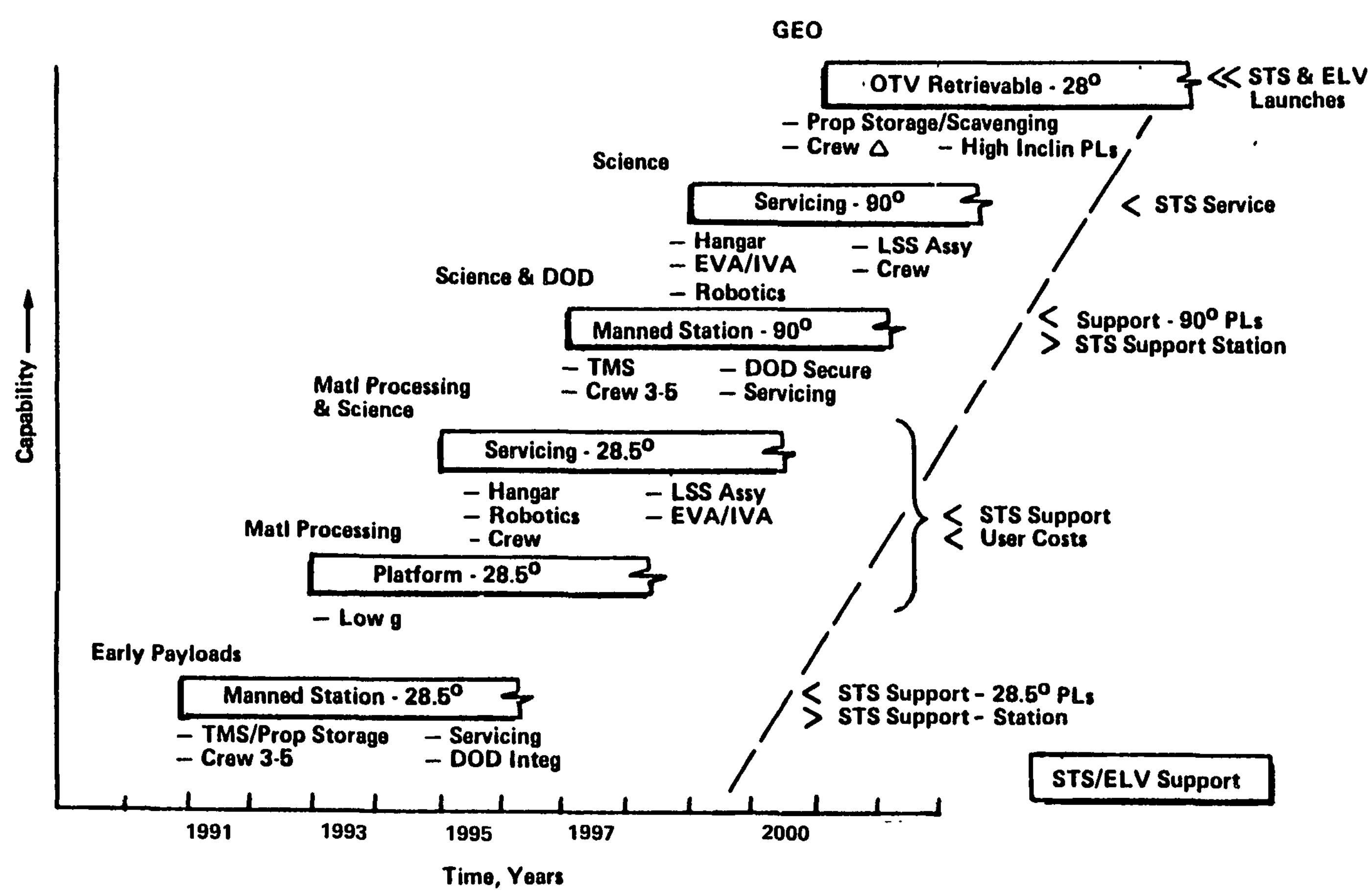


Figure 3.1.1-5 Option B-1: 28.5° & 90° Stations

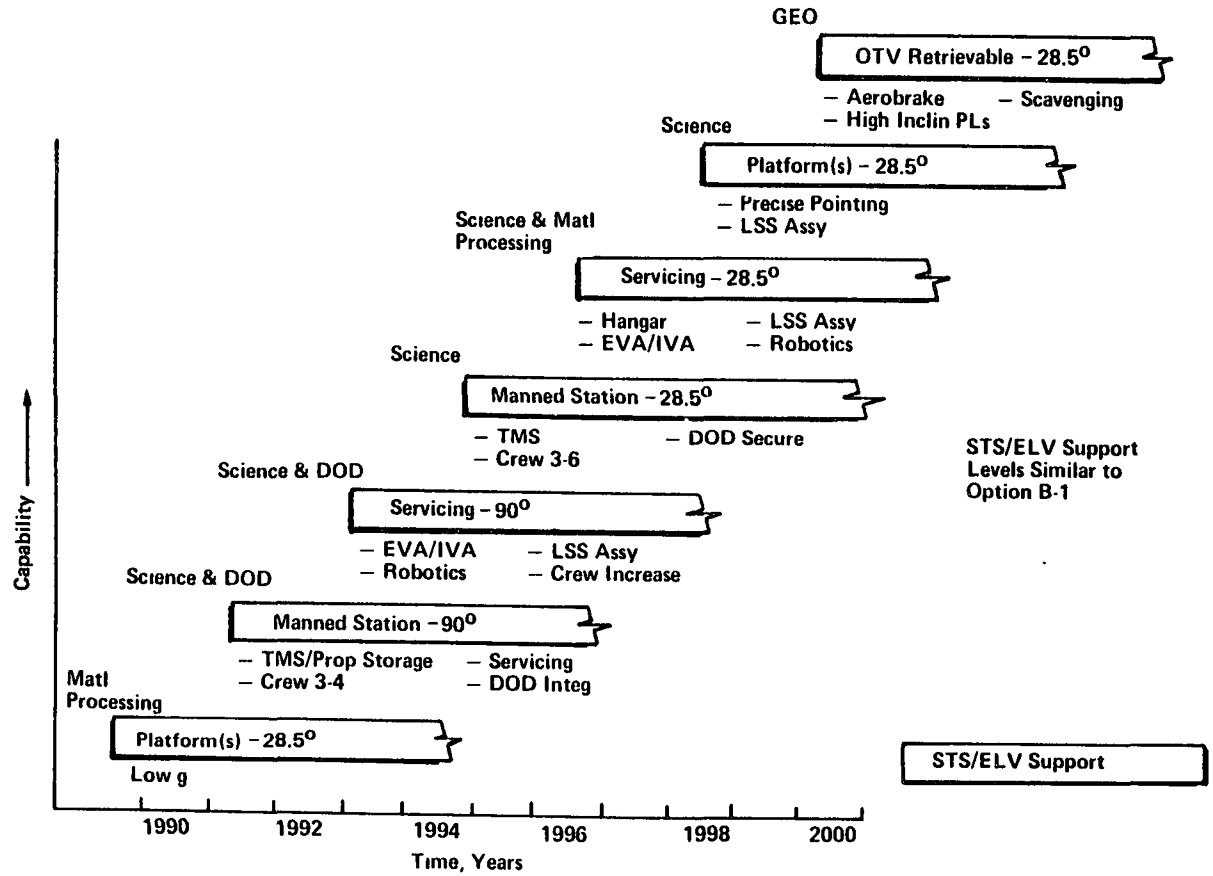


Figure 3.1.1-6 Option B-2: 90° & 28.5° Stations

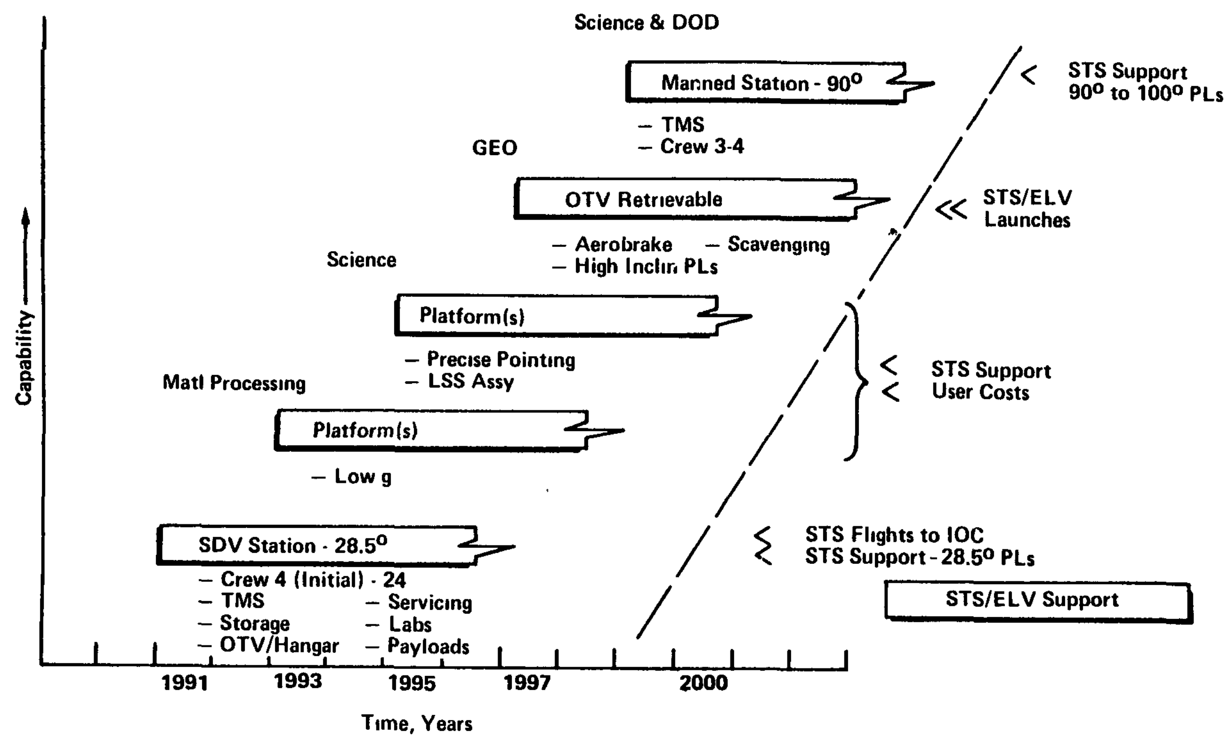


Figure 3.1.1-7 Option B-3: Shuttle-Derived Vehicle Station

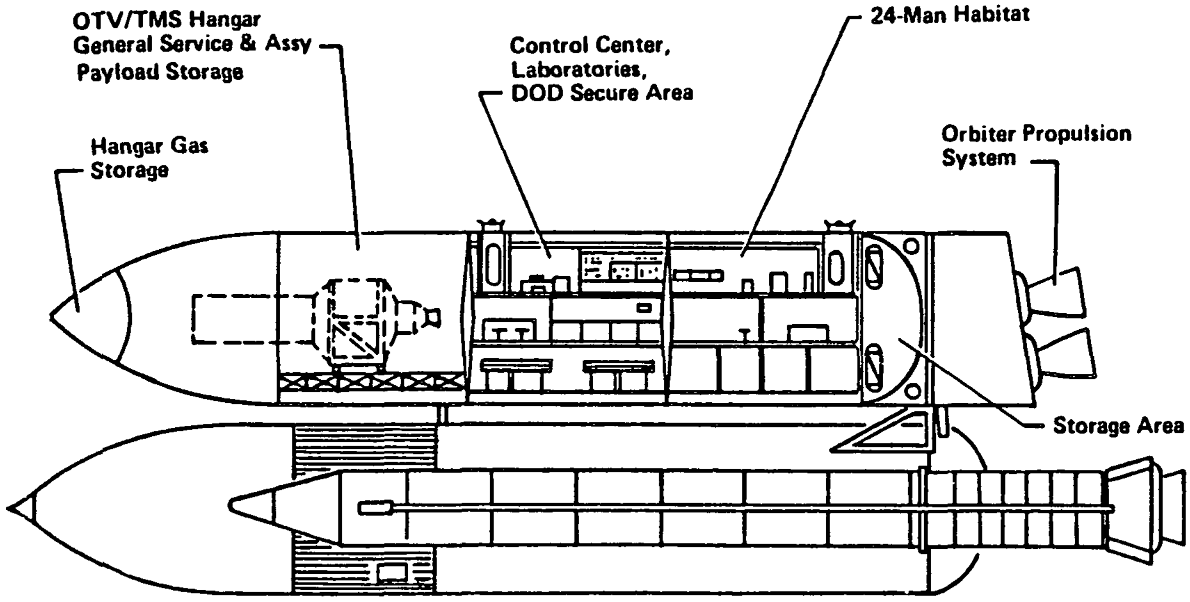


Figure 3.1.1-8 Shuttle-Derived Space Station

Option C-1, presented in Figure 3.1.1-9, suggests front end costs can be reduced most effectively if the manned space station is delayed until the 1994-1995 timeframe. The early emphasis would be placed on implementing payload platforms at 28.5 degrees. Further front end cost reductions can be achieved by delaying the OTV implementation until the late 1990's.

3.2 FEASIBILITY ANALYSES

Analyses were performed on each of the eight program options to estimate their life cycle costs and to determine what percentage of user missions each option could support. The results of these analyses are summarized graphically in Figure 3.2-1. For each of the eight program options, two vertical bars are shown representing total estimated cost of the option (shaded), and capture ranking or the percentage of user missions supported.

Regarding relative costs, options A-1, A-2, A-3, and C-1 fall within the range of 9 to 11 billion dollars; while option A-4 and the three dual station options fall in the range of 13 to 18 billion dollars in terms of 1984 dollars.

With respect to ability to support the integrated user mission requirements, options A-1, A-3, B-1, and B-3 exhibit the highest levels of support, and vary slightly between the four options. The lowest level of user support occurs for option A-4, the single polar station approach. Option A-2 is well below either A-1 or A-3 primarily because of the six year delay in implementing the OTV in option A-2.

3.3 VIABLE PROGRAM OPTIONS

Referring again to Figure 3.2-1, the most viable or effective program options were selected on the basis of lowest option cost and highest level of user support. It becomes rather obvious from Figure 3.2-1 which options are the most cost and performance effective. The lowest cost options, are A-1, A-2, A-3, and C-1; but both A-2 and C-1 have user support levels significantly less than A-1 and A-3. Looking at user support levels, options A-1, A-3, B-1, and B-3 have essentially equivalent levels of support, but options B-1 and B-3 are 3 to 6 billion dollars higher in cost.

We have therefore concluded that program options A-1 and A-3, single manned stations at either 28.5 or 57 degrees, are the most viable options. These two options were then subjected to further analyses to define a more detailed evolution approach and take a much closer look at user mission support. This was an iterative process in which a program option evolution approach would be defined and then discussed with the user requirement specialists on project to determine compatibility with the payload needs. Specific discussions were made for each user mission as to whether it was best accommodated at the space station, on a platform, or as a free flying payload. Ability of each of the two viable options to satisfy specific user needs such as delivery, crew support, instrument upgrade, cryogen resupply, data processing support, subsystem support, and other servicing was evaluated. These analyses further considered the affordability criteria applied to the space station mission model.

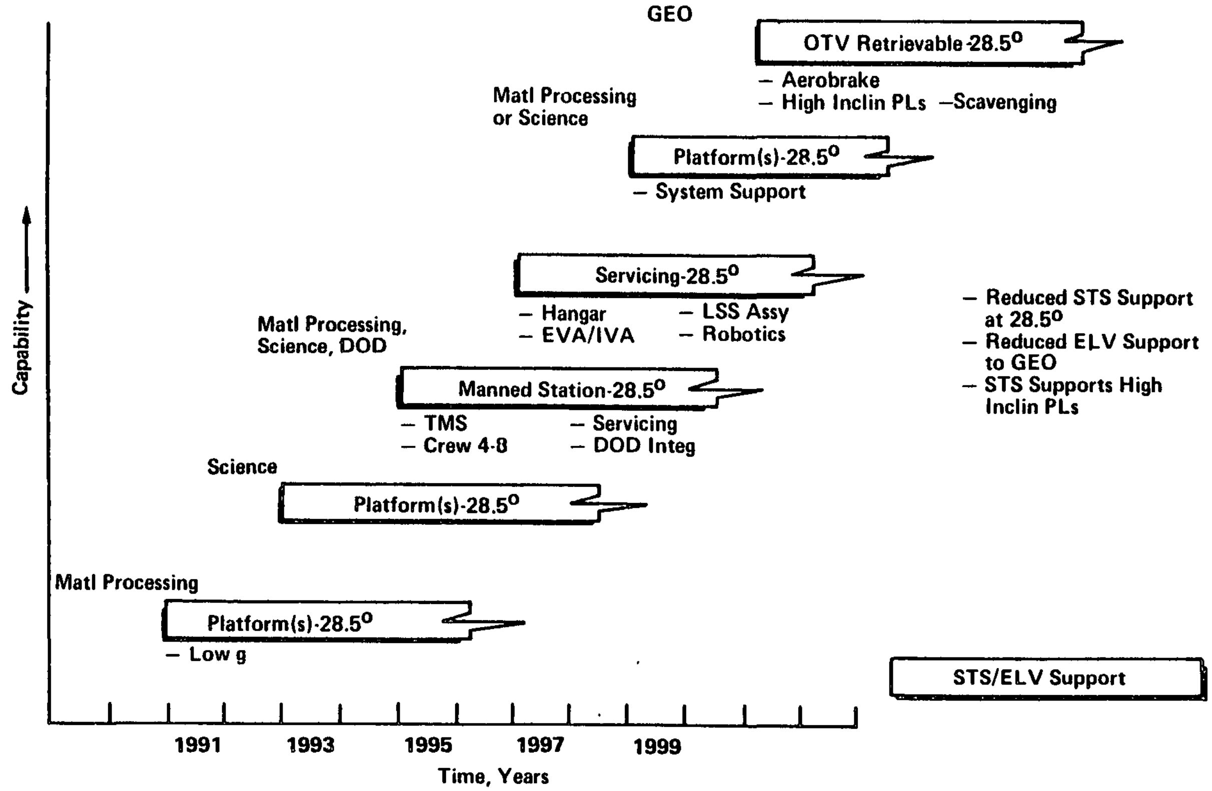


Figure 3.1.1-9 Option C-1: Low Front-End Cost

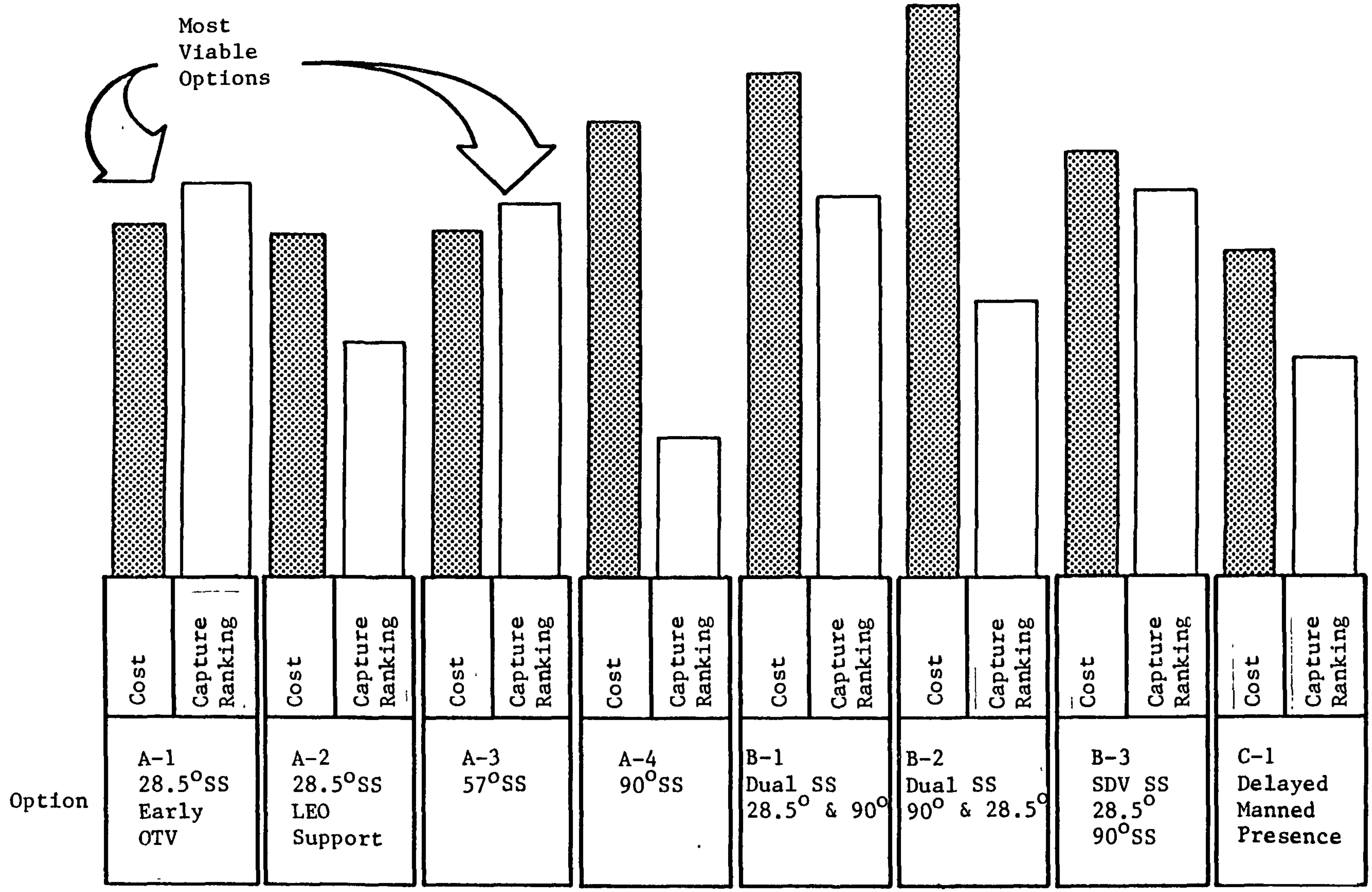


Figure 3.2-1 Program Option Selection

On a mission-by-mission, and year-by-year basis, specific levels of TMS OTV, and STS support were determined for each of the two options. These data also supported the cost analysis of space station user benefits described in section 6.0 of Vol. V.

These more detailed user support analyses and updated evolution plans resulted in a 28.5 degree space station option which supported 78% of the non-DOD user missions; and a 57 degree station option which supported 68% of the non-DOD user missions.

3.4 PROGRAM OPTION RECOMMENDATION

The decision on a recommended space station program option came down to selection between the two viable options discussed in section 3.3. The summary results of our analyses indicated that option A-1, a single manned station operating at 28.5^{o} in conjunction with unmanned platforms, provided the highest level of user support and high levels of performance and economic benefits in a cost effective manner. The following rationale supports this decision.

a- A user support level of 78% for the 28.5^{o} option versus 68% for the 57^{o} option. User support across the various user categories is presented graphically in Figure 3.4-1, for the 28.5^{o} option. The user support matrix illustrates year-by-year support for all non-DOD user categories. A fully shaded block indicates a high level of support, a half shaded block indicates support to about 50% of the missions, and lesser shading levels indicates low support levels. The matrix shows that high support levels are achieved for a majority of the disciplines in the second year of space station operations (1992), and moderate to high levels of support are achieved by 1993 in all the user disciplines shown.

b- Greater support to the largest user class, GEO communication satellites, because of higher payload delivery from a lower inclination angle. With an assumed limitation of 1 OTV flight per month for this payload category, option A-1 captures 79% of the post 1991 missions versus 63.5% from 57^{o}, option A-3.

c- Basically the same level of support to the science payloads for both options because of considerable flexibility for many payloads on operating at 28.5^{o} or 57^{o}. The earth observation payloads could accept operation at either a 28.5^{o} or 57^{o} station prior to implementation of a dedicated earth observation platform operating in a polar orbit in 1996. Another notable exception occurred in the Space Physics discipline with the ISTO platform which has a firm requirement to operate at about 57^{o}. Because of ISTO's need for high duty cycle man-in-the-loop support and relatively high data rates, this platform could be better supported from a 57^{o} station with a line-of-sight RF link than from a 28.5^{o} station.

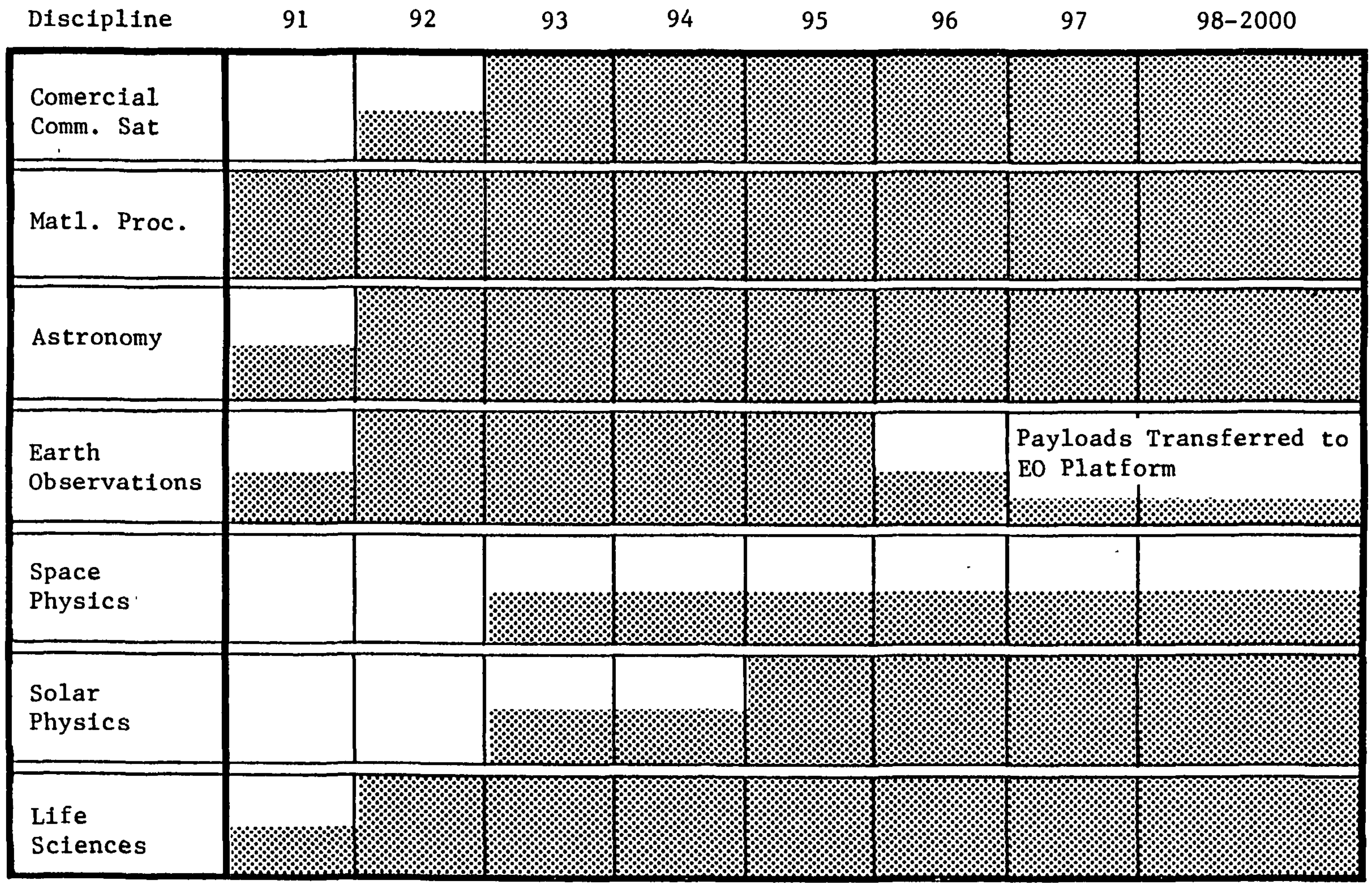


Figure 3.4-1 User Support Matrix

d- Similar levels of support to the Life Sciences requirements and the Materials Processing payloads.

e- A higher level of STS support required for the 57° option because of reduced launch capability.

4.0 ARCHITECTURAL OPTION

4.1 INTRODUCTION

This section describes the architectural option study which includes requirements derivation, a review of the trade studies involved in the configuration selection, and a description of the configurations developed. As used within this study, an architectural option is defined as specific space station configurations that relate to an individual program option step or phased growth increment.

The methodology used in the definition of the space station architecture has been to derive top level requirements from both the selected program option and the mission model, perform trade studies on major architecture issues, and then develop configurations to meet the combination of functional and physical requirements.

Using this approach, three space station configurations have been developed during the course of this study. A modular concept space station based on the STS cargo bay limitations, another modular concept founded on the external tank/aft cargo carrier (ET/ACC), and a station based on the shuttle derived vehicle are presented in the following sections. Platform architecture studies are treated separately in Section 4.5. This study has not reached the detail of architectural options tied to internal module layouts or equipment arrangement.

4.2 ARCHITECTURE ANALYSIS

4.2.1 Requirements

The top level requirements that follow have been assembled from those presented in the statement of work and from customer direction at the various meetings. The requirements are:

1) STS compatibility
2) Early 1990's IOC
3) Permanent manned presence
4) Reasonable budget constraints
5) Compatible with technology growth
6) Using the selected program option
7) Resupply STS flights at approximately 90 day intervals.

The next tier of requirements were derived from the recommended program options as presented in section 3.0. Table 4.2.1-1 identifies the major program option increments and presents both the first level derived requirements and key trade study issues resulting from these requirements.

Table 4.2.1-1 Derived Requirements and Trade Studies

Program Option Step	Derived Requirements/Functions	Key Architecture Trade Study Issues
IOC	- Habitat areas	- Arrangement and safety aspects
	- Station attitude control	- Inertial versus gravity gradient orientation
	- Power system	- Solar array versus nuclear power
	- Station structural buildup	- Preferred attachment scheme
Payload Activation	- Payload accommodations	- Payload compatibility with space station
		- Payload structural interface with station
	- Servicing access	- EVA, TMS, or space crane
OTV & Expanded Operations	- OTV Accommodations	- Storage & service considerations
		- Propellant resupply options
	- Expanded power requirements	- Power system growth options
Expanded Operations/ Growth	- Added elements & payload	- Growth accommodation options

The derived requirements are self-explantory, but some discussion is in order to explain the selection process. First, an attempt was made to limit the listing to only those issues that resulted in significant architecture decisions. Next, similar items, at a following program option step, were only included when they forced a unique decision or trade study. Also, the issues were selected to be basically configuration independent. Finally, the derived requirements assume, as a basis, the previously identified groundrules and requirements.

4.2.2 Trade Studies

The following trade study discussion is limited due to the duplication of many issues in the subsystem sections (section 6.0).

4.2.2.1 IOC Trade Studies - One of the first trade issues to be addressed deals with space station attitude control. The options available are inertial or gravity gradient orientations. The gravity gradient attitude was selected because of the following rationale:

1) A gravity gradient attitude "fails" in an operational mode
2) Gravity gradient forces are substantial at low earth orbits
3) An inertial system would require large momentum storage devices or excessive RCS propellant consumptions
4) The drag make up system (required) can be used to supplement the gravity gradient torques and handle limit cycle problems

An analysis of previously developed concepts illustrates the angular momentum buildup problem associated with the inertial orientation. Even though an inertial system will balance out the gravity gradient torques in one orbit, the half orbit buildup exceeds the storage capacity of the existing control moment gyros (CMG's). Additionally CMG's have, to date, proven to have short life times.

A gravity gradient attitude requires a specific ratio between the station body axes to remain in a fixed attitude and to take maximum advantage of the stabilizing nature that results from the precession effect. Figure 4.2.2.1-1 shows the station coordinate system and desired inertia relationships, and includes a typical shape needed to achieve these relationships. Note that the precession effect (due to orbital rate) provides stability for external yaw and roll influences and that the roll and pitch offsets are acted on by the gravity gradient forces.

Referring to Table 4.2.1-1 once again, another major trade involves the selection of the space station power source. Section 6.1 fully documents the selection of a solar array system primarily on the unavailability of a nuclear source during the station buildup era.

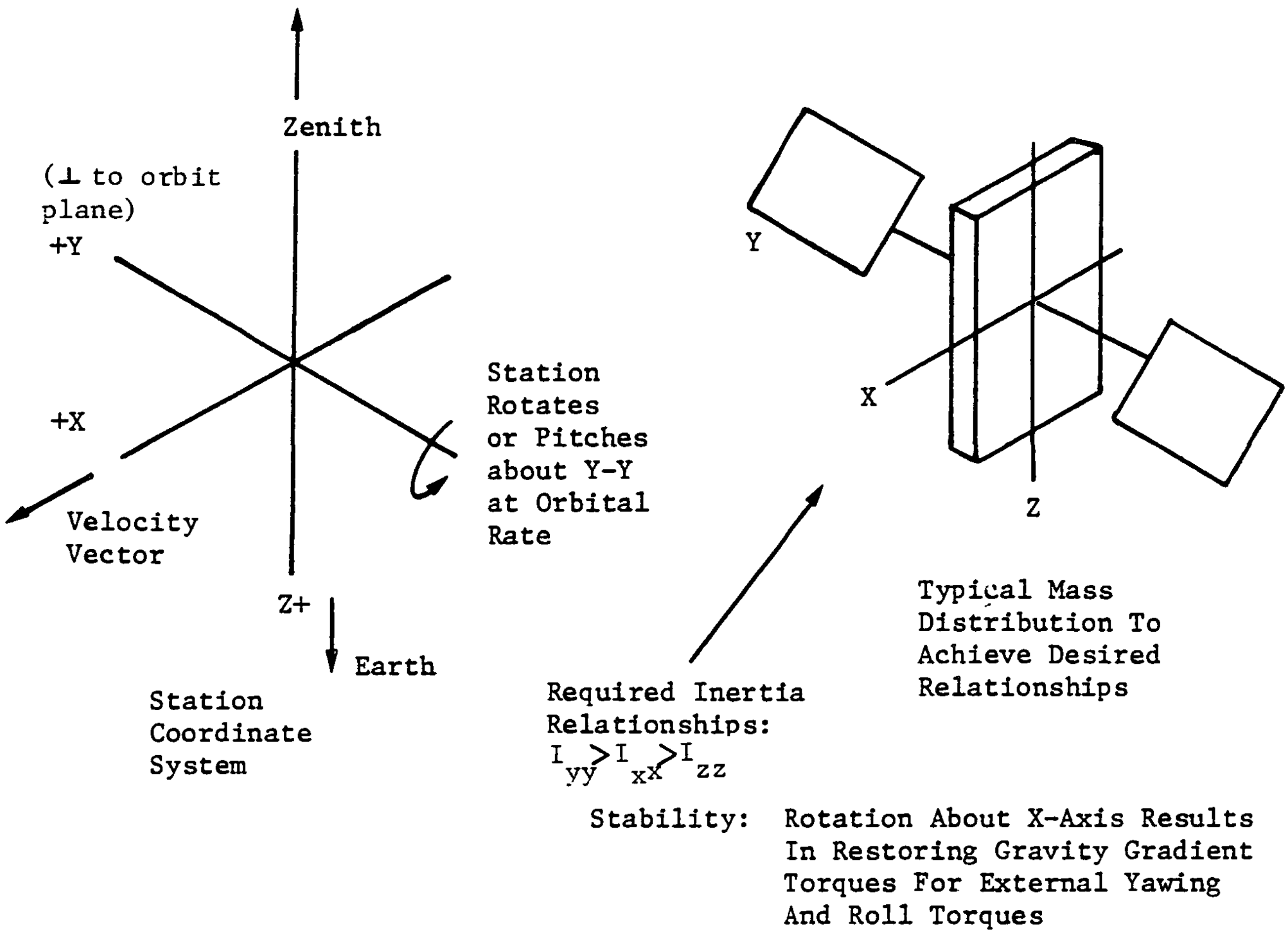


Figure 4.2.2.1-1 Gravity Gradient Body Relationships

The solar array decision must be analyzed relative to the attitude selection since the magnitude of the restoring gravity gradient forces are small, and the angular momentum build up from solar array drives and drag torques can be substantial. The goal is to select a solar array system that minimizes momentum build up. This can be best accomplished by providing continuous motion drives, providing symmetrical array elements, and by minimizing the offset between the space station center of mass and the station center of aerodynamic pressure.

The selected array pointing system uses a two axis drive (see section 6.1 for selection) where a continuous drive is utilized to track the sun's insolation during orbital motion and a slow moving incremental drive is used to track the sun Beta angle motion relative to the orbital plane.

The next two issues evaluated deal with the habitat areas and the physical growth interface scheme. Here the primary issues are safety (crew egress paths) and how the space station interconnection takes place.

A dual egress path was selected for the station as opposed to serial build up concepts due to crew safety considerations. Since the IOC phase will be linited to a minimum number of modules, the initial manned arrangement may not accommodate this feature. During this operational phase, an airlock can be provided for emergency egress via EVA.

The basic structural interface of the space station elements requires some investigation, since the size of the station will eventually lead to flexible body dynamics considerations. Normal design practices are to achieve as great a separation as possible between the body structure natural frequencies and the lowest flexible appendage frequencies in order to prevent coupling. In a normal spacecraft design this is usually easily achieved. The unique space station problem occurs due to the size, mass, number of elements involved, and the restrictions on the interface structures. For example, a 14 foot diameter module with a 14 foot diameter interface structure would achieve great structural rigidity (high natural frequencies). The attachment device, however, must also function as a docking mechanism, commodity/utility interface, and in some cases, a manned passageway. To incorporate these features into a very large diameter interface ring would result in higher costs, difficult circumferential sealing problems, and stowage envelope problems for the opened hatch. These considerations result in the selection of a docking system with an inside diameter closer to the minimum passageway of 40 inches. Now the combined stiffness of the system is reduced and every tolerance or deviation becomes magnified due to the number of mechanisms involved and the moment arms between elements. The conclusion here is that the selected architecture must include structural rigidity considerations as part of the decision process.

4.2.2.2 Payload Activation Trade Studies - Payload activation is the next program option to be analyzed. Here the areas of interest are; what payloads are compatible with the station, how should the payload structural interfaces be accomplished, and how should access & service be accommodated.

Of primary importance is the compatibility assessment of what payloads are best suited for incorporation on board the station. Using the mission model, the candidates are as follows: solar physics, astronomy, earth observation, space physics, and materials processing.

Having selected an earth reference pointing attitude, a natural space station payload candidate becomes the earth observation grouping. this payload class can be characterized in a simplified manner as follows: Nadir or velocity vector pointing, generating high data rates, desiring high inclination orbits, having medium pointing accuracy requirements, and being sensitive to effuents. Although the space station inclination is lower than optimum and there is a potential contamination problem (yet to be quantified), there is still adequate need and benefits (equatorial coverage, availability of early long term viewing, etc-see volume II) to include a portion of these payloads during the early space station flights.

The astronomy and solar physics missions can be addressed together since they share similar requirements. Low inclinations are acceptable to astronomy experiments while the solar payloads prefer a slightly higher inclination range. Both the solar physics and astronomy payloads share a dislike for contaminants and are sensitive to disturbances to the pointing systems. Attachment of these missions to the space station would in most cases require 360° rotation per orbit for target tracking. Since the station does provide good sky coverage over time, and there is a potential for good fields of view (with proper location & configuring of the station), these payload classes should also be included in the preliminary candidate listing.

Materials processing missions have as their unique requirement the need for very low gravity levels. A preliminary calculation of g-level variation with offset from the center of mass for a representative space station geometry and altitude shows that for every foot of offset there is an increase in 1.25×10^{-6}g's. This indicates that an experiment with a threshold of 10^{-5}g's could only be approximately 10 feet from the center of mass. A 10^{-3} limit would allow an offset of almost 1000 feet. Manned disturbances could also have an affect on the materials processing experiments. Here the g levels could be in the neighborhood of 10^{-3}g's, but the impulse (pushoff) would be nearly cancelled by the stopping impulse. Hence over a short time interval, a low net disturbance would occur. These disturbances would be random and of short durations, and may be compatible with the duty cycle of the processing experiments. Until specific analysis proves this compatibility/uncompatibility, materials processing payloads and experiments should be considered for station incorporation.

Space physics incorporates many requirements and features of the other disciplines; earth, sun, and magnetic field lines are the pointing requirements, mid-inclinations are preferred, fine pointing is required on some payloads, and some have contamination sensitivity. Based on the previously discussions, space physics payloads should also be reviewed on an individual basis as possible station payloads.

With a series of candidate payloads in hand, the selection of physical interfaces can be addressed. Direct attachment to the station, tethered approaches, and free flying (but station associated) are potential schemes to be evaluated.

Our selection process eliminated tehers at this time due to; technology issues, compatibility with utility transfer (distances), and docking (STS) effects. There is however a technology development mission identified and the approach would be to operate this TDM on an experimental basis to determine the real applicability of this type of payload operation. At this point in the study both direct station attachment and free flyer operations are selected.

How to physically accomplish the direct attachment scheme is the next topic of discussion. Two options were identified and traded as part of this decision process. Use of the same type of docking adapter as previously identified during this discussion is an obvious choice because of the compatibility with the many docking ports around the station. The other potential approach was to use a STS cargo bay replica (replicated in terms of structural interfaces) for mounting of the various science payloads. The advantages to the docking adapter approach are as follows:

- o Compatible with station docking interfaces
- o Provides standard utility interfaces for power, data, cooling, etc
- o Is adaptable to various pointing orientations
- o Has small weight and volume requirements

Disadvantages are:

- o Require a different interface between the STS and payload for transportation to the station (ie the standard structural attachment fittings or cradle).

- o Limit on number of elements that can added at a particular docking port (assumes a multiple adapter port can handle 4-5 payloads)

The STS type structure has these advantages:

- o Uses same interface as transportation interface
- o Provides high strength/stability
- o Allows many payloads per port

On the con side, the following apply:

- Requires a new umbilical interface with the structure (assumes the interface between the payload and structure should be automatic and common and the the STS wiring and interface system are not adequate)

- Limits viewing orientation to a selected hemisphere (assumes "U" Shaped structure)

From the above data, the docking adapter approach was selected on the basis of the station compatibility and adaptability.

Servicing and access also appears on the "payload activation phase" list of requirements. Once experiments or payloads are emplaced on the station, their maintenance, resupply, and changeout operations must also be satisfied by station capabilities. At the station architecture level, the issue to be decided is one of physical access. This issue also applies to the station elements buildup.

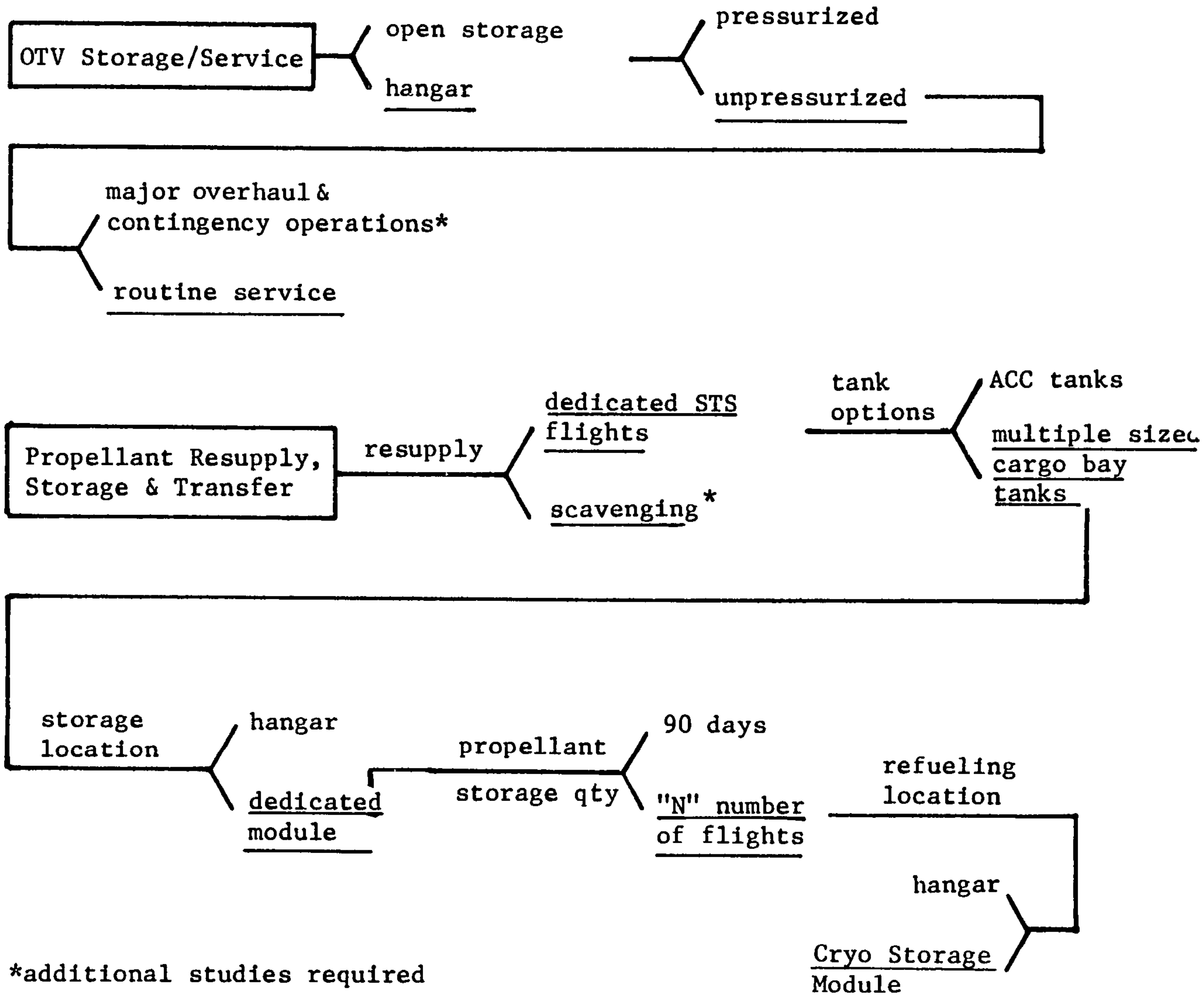
By examining the various stages of buildup and the mature station configurations depicted to date, it is readily apparent that some form of total station access and equipment handling scheme is required. This includes moving modules into position and docking them with the station. The TMS, MMU, EVA, and a traveling crane were all evaluated against these requirements. The MMU and EVA approaches are best suited for access and observation maneuvers, since they have a physical size limit.

Use of the TMS appears to be limited to the transfer mode only. The fine control required for docking can only be achieved by a thruster system that provides both translation and rotational degrees of freedom. The current TMS design studies evaluated outrigger type RCS thruster extension devices, but rejected these due to the need to adapt to multiple center of mass and inertia configurations, and potential stiffness/control problems.

This leaves only the traveling crane or station unique RMS as having the capability of performing the handling activities. In this capacity, the crane must have full access to the station payload/elements, and the units to be handled must be designed with compatible interfaces.

4.2.2.3 <u>OTV Operations Trade Studies</u> - Referring to Table 4.2.1-1, OTV operations, (at the station) is the next incremental program option step selected for driving out configuration decisions. Since the OTV missions and associated benefits have been major drivers in the overall program option selection, the configuration option decisions are equally significant. Table 4.2.2.3-1 depicts in summary fashion the configuration decision logic associated with the two sub-categories of OTV operations.

Table 4.2.2.3-1 OTV Operations Configuration Decisions



*additional studies required

The first branch of the diagram deals with the space station interfaces associated with storage and servicing functions. Choice of the hangar is due to the relatively long periods of time the OTV would be exposed to the micrometeoroid debris environment. An unpressurized hangar was selected based on hatch sealing problems, gas loss/storage requirements, pumping power requirements (assumes saving atmosphere by evaluating the hangar to a storage reservoir), and pressurant gas resupply requirements. Additionally, since the hangar diameter appears to want to exceed the STS cargo bay diameter capability, the problems associated with building up a pressurized hangar on-orbit must also be considered.

If the level of servicing and scope of hands on labor intensifies, then the dexterity advantages of a pressurized hangar needs to be re-evaluated. The next decision to limit OTV servicing to routine/minor operations is also partially tied to the unpressurized hangar decision.

The propellant operations logic flow (Table 4.2.2.3-1) involves both STS operations decisions as well as space station architectural decisions. How the station resupply is handled is more fully discussed in section 6.3. The choice of dedicated STS flights (defined as having cargo bay propellant tanks ground loaded) is predicated primarily on the possibility of not having enough scavenging flights to handle the monthly OTV propellant requirements. Scavenged propellant is assumed limited to the 10,000 lbs that can be effectively transferred from the ET after MECO. A typical month might require 5-7 STS flights on a scavenging basis. Flights are based an average of almost 2 OTV flights/month ,with a 24,000 to 35,000 lb per flight propellant requirement. Additional study is needed in this area, but, since dedicated propellant flights will be required in some instances, a dedicated approach is selected with scavenging also used for supplement on those flights that are not fully loaded.

Tank options investigated included; dedicated cargo bay tanks (in varying size ranges) and external tank/aft cargo carrier tankage. The multiple dedicated tanks permits maximizing the STS cargo load factor on space station bound flights. The dedicated tank approach was selected since there is approximately a 12,000 lb weight penalty associated with achieving orbit and circularizing the external tank/aft cargo carrier (independent of tankage weight).

A cursory analysis was performed on potential tank sizes with a result that two sizes were selected, 25,000 and 50,000 lb propellant capacity. A 10,000 lb tank set is also still under consideration. The 25,000 and 50,000 lb tank sets occupy approximately 25 feet and 36 feet of cargo bay space respectively.

For an on board storage location, a facility integrated with a hangar and a separate module were traded. The separate module approach won out on the basis of integration complexity associated with the hangar/storage facility.

The determination of on board storage requirements was based on potential resupply cycles and also by traffic model considerations. Section 6.3 of this volume presents the analysis behind the selection of a 70,000 lb propellant storage system.

The last OTV operations facility decision involved the selection of the refueling location at the station. Refueling at the cryo storage module was chosen because of the ability to centralize tank loading and OTV loading systems at one location and the advantages of freeing the hangar for other operations during refueling.

Associated with OTV operations, the overall increased activities and expanded operations leads to a need for a crew size growth. This in turn, requires additional power on board the station. Section 6.1 describes the solar array/EPDS system growth steps and approaches to accomplish this. The significant architecture result is that it is feasible to add on the array during growth periods as opposed to having to deploy the entire system at IOC.

4.2.2.4 Expanded Operations Trade Studies - The final program growth step to be evaluated is another expanded operations period. This era involves some large space structures assembly activities, more crew involvement, and payload operations whose definition and scope is not finalized. The main architectural decision here is to determine how best to accommodate future and unforseen growth (the station must provide for some level of unplanned growth or its overall utility will be reduced). The following discussion addresses where the additions or buildups should occur to maintain the proper attitude.

Analysis of previously depicted configurations shows that the addition of a single, laboratory sized, module can significantly effect the station mass properties. On an individual axis, the system inertias could increase by 1/8 to 1/3 of the total inertia depending on weight and location of the added element. This large variation could result in a redistribution of the principal axes and a different flight attitude. Since a gravity gradient orientation desires an inetia axes distribution with $I_{yy} > I_{xx} > I_{zz}$, localized build up along the station x-axis is preferred with a z-axis build up of second choice. This assumes a Y-axis solar array distribution. The size or weight limits for additions will be dependent on the specific architectural configuration selected.

Table 4.2.2.4-1 is a compilation of the selected major decisions for each program step.

Table 4.2.2.4-1 Major Architecture Decisions

Program Option Step	Decisions/Conclusions
IOC	o Gravity gradient attitude o Solar array power system - symmetrical - minimum offset from center of mass - 2-axis drive o Dual egress paths o Docking ports for element attachment, structural rigidity requirements
Payload Activation	o Payload candidates: astronomy, solar physics, space physics, earth observations, and materials processing o Payload integration via direct station attachment or free flying platforms o Use docking port for payload attachment o Payload and station element access and handling requires a traveling space crane
OTV Operations	o OTV Storage via hangar o Hangar is unpressurized o Use dedicated STS cargo bay tanks for propellant resupply (supplement with scavenging) o Provide multiple tank sets to maximize STS load factor o Use a dedicated storage module for propellent o Perform refueling at the storage module
Expanded Operations/Growth	o Station build up along X or Z axes

4.3 MODULAR DESIGNS

4.3.1 Introduction

Previous studies have focused on modular space station arrangements due to the natural volume constraints of the STS delivery vehicle and the logic of phase growth. The basic premise of this approach is to maximize commonality between elements and to build a space station in incremental phases while maintaining integrity at the individual element emplacements.

Our studies resulted in the selection of two modular concepts for presentation, a STS cargo bay modular version and another concept based on the use of the external tank/aft cargo carrier. The following two sections will present these results. All concepts were based on the requirements and major trade study decisions as presented in sections 4.2.1 and 4.2.2.

Since the modular concept is based on a build up of numerous elements, a preliminary analysis of potential construction approaches was undertaken. Using the requirements and architectural decisions from the previous section, triangle and rectangle based concepts were compared.

The triangular approach was selected primarily for the ability to add on elements and retain a dual egress path. The proposed advantage of structural stability only exists if the element attachments are truly pin ended at the joint intersections. Since this is not feasible with the type of module and space construction involved, the structural stability is dependent on the docking port stiffness. As previously discussed, the stiffness achievable is directly related to the diameter of the docking system and the diameter must be limited for reasons previously discussed. Therefore there is no significant structural advantage to the triangular build up approach.

Figure 4.3.1-1 depicts the evolution the triangle concept went through to achieve a workable system. Concept 1 proved unfeasible due to the lack of pressurized volume for crew and equipment. The geometry constraints which limit the module size are due to both STS bay constraints as well as the need to incorporate radial ports for auxiliary module and payload attachments. The result is that it takes 4 1/4 modules of the triangular concept to equal the volume of 3 rectangular modules. Concept 2 presents the next attempt where an additional 3 modules (minimum possible to maintain a closed system) were added to form a pyramid arrangement. This resulting configuration was abandoned for the following reasons:

1) requires an additional STS flight to achieve an FOC configuration comparable to the rectangular concept;

2) a new, ball like, end fitting is required along with a lateral docking capability;

3) the basic shape results in undesired inertia arrangements for gravity gradient orientation (Ixx & Izz $>$ Iyy);

THIS PAGE INTENTIONALLY LEFT BLANK

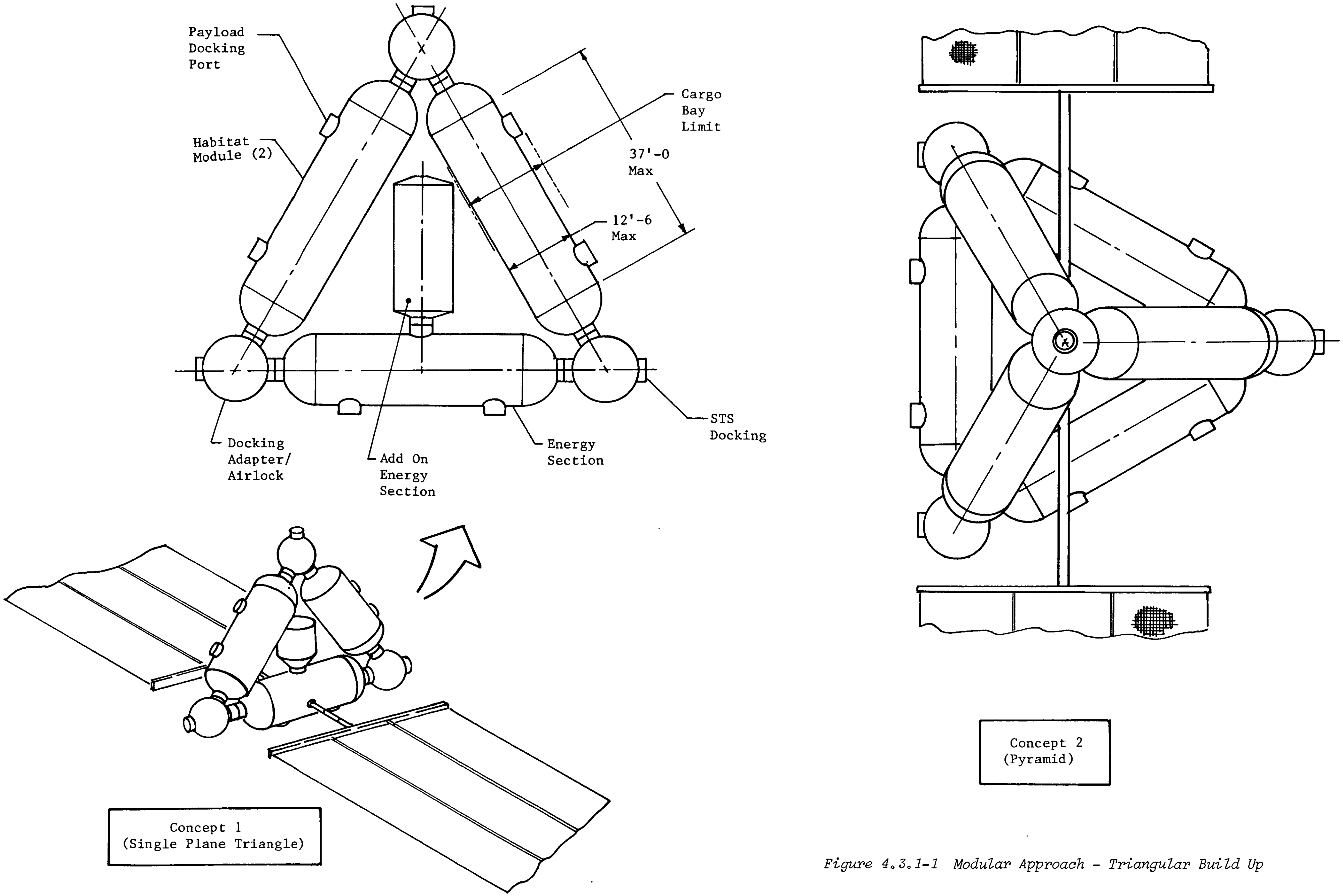


Figure 4.3.1-1 Modular Approach - Triangular Build Up

Page intentionally left blank

Page intentionally left blank

4) the concept has a problem in achieving a balanced solar array location (i.e., extended boom and transition between initial STS flight and FOC configuration);

5) the build up sequence involves the lateral positioning of modules in the proximity of the solar arrays.

4.3.2 Cargo Bay Modular Concept

This approach utilizes the STS cargo bay for the delivery of all major structural elements of the space station.

During this study period two configurations were developed and analyzed. Figure 4.3.2-1 illustrates the "A" concept modular space station. As detailed evaluation of this configuration progressed, a number of configuration related faults became apparent.

As subsystem details became available, it was determined that a single energy section could not accommodate the required volume of equipment. Next, the build up sequence was examined on a STS flight basis. This revealed that at the addition of the tunnel (above the two habitat modules), there was no location at which to dock the orbiter. This would then require "hand off" assembly between a formation flying orbiter and the space station. Additionally, the arrangement is limited in the amount of payloads it can handle, and has little provision for unplanned growth.

Figure 4.3.2-2 depicts the recommended cargo bay modular concept.

On orbit, the station would maintain an earth-reference attitude with the energy sections parallel to the velocity vector. The symmetrical solar arrays are perpendicular to the orbit plane and utilize a two axis drive for sun tracking (continuous rotation about the Y-axis and $\pm$ 52° about the base of the array area).

Approximately 17,000 ft^2 of active solar array area is provided by the 12 panels which make up the array. Each panel is approximately 11.5 feet wide by 123 feet long. The array booms have been minimized in length to provide maximum stiffness, while taking into account the effect of shadowing losses on the array. Shadowing becomes a factor at Beta angles exceeding 30°.

This configuration has dual energy sections which provide adequate volume for the required subsystem equipment (EPDS, TCS, communications, propulsion, DMS, and attitude control), and allow room for an energy section safe haven option. The second energy section permits the phased growth of the power system equipment plus opens up additional docking ports for growth flexibility. The body section is 115 inches in diameter (pressure vessel) and is 588 inches between docking port interfaces. A (deployed on launch) meteroid shield/radiator panel encircles the module at a standoff distance of approximately 12 inches. At launch the first energy section includes the initial solar array compliment, the reaction control system booms, and the initial communication antenna system. These are all deployable at activation. Along with the RCS booms and thruster packages, the interior contains

THIS PAGE INTENTIONALLY LEFT BLANK

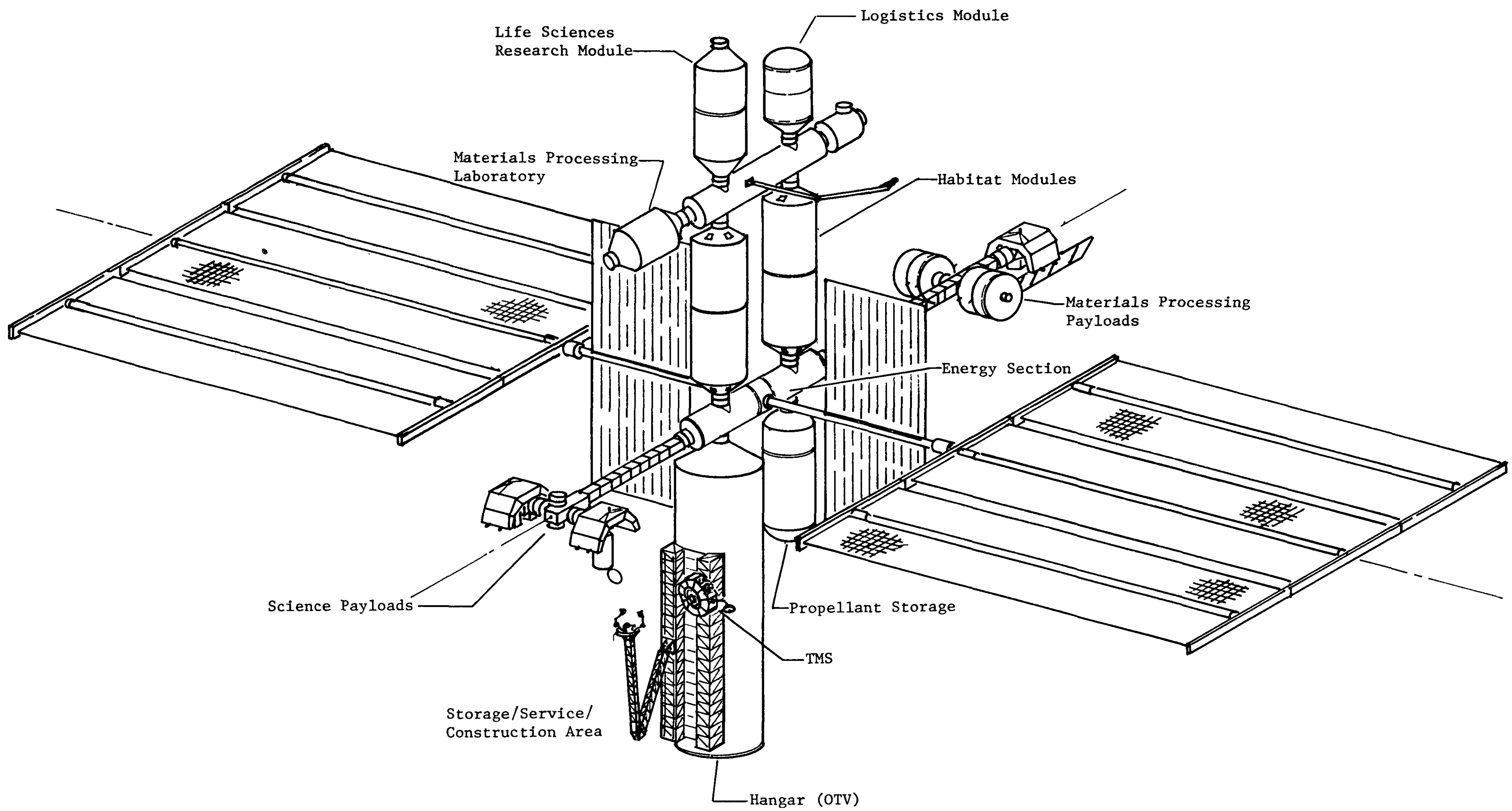


Figure 4.3.2-1 Concept A - Modular Cargo Bay Concept

Page intentionally left blank

Page intentionally left blank

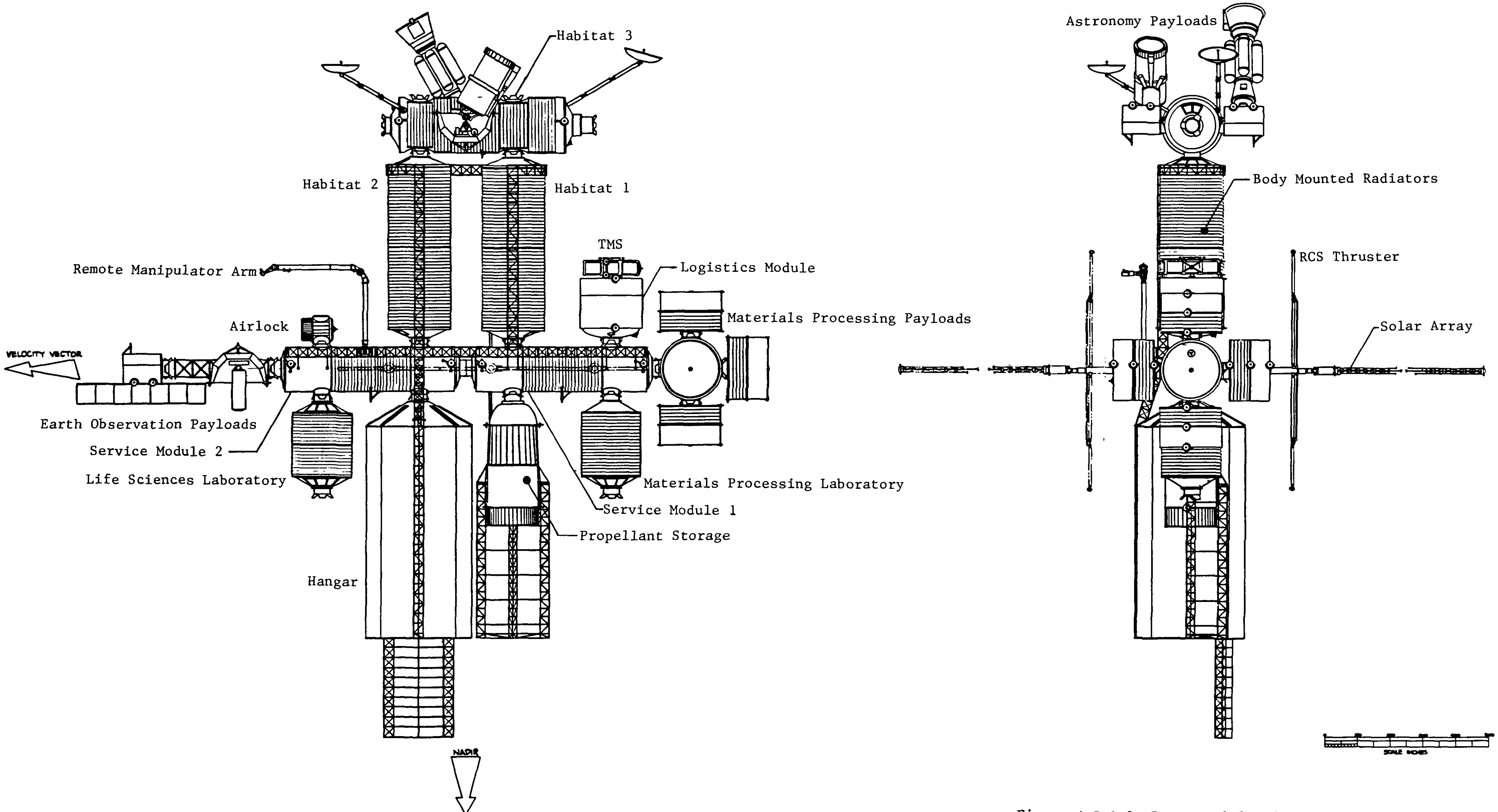


Figure 4.3.2-2 Recommended Modular Cargo Bay Concept

Page intentionally left blank

Page intentionally left blank

approximately 5000 pounds of hydrazine propellant for drag make up purposes.

In this mature station configuration, two habitat modules extend vertically from the energy sections. Each module is approximately 14 feet in diameter (pressure vessel) and again is 588 inches in length between docking port interfaces. Body mounted radiator/shields are also used on these modules, but here the panels must be deployed after removal from the orbiter bay.

Crew quarters, health maintenance facilities, command and control stations, hygiene facilities, food storage and preparation areas, and recreation areas are provided in these modules.

During the activation phase, EVA suit storage and an external airlock are required to permit both normal and emergency EVA's. A safe haven area is also required during the initialization era. In this modular approach, we have selected a division of the first habitat module as the safe haven concept. Primary considerations in this decision were:

1) require redundant ECLS systems and they can be easier split between halves of the same module than in separate modules

2) the food, medical, crew areas, etc can be better integrated into two separate areas within the habitat module as opposed to being split between the hab module and energy section (due to the larger available diameter and better space useage).

The safe haven decision is preliminary, and as previously suggested does not obviate the potential of using the energy section as the safe haven. Internal equipment layouts and arrangements (not completed during this study) would be required to make this final determination.

In our configurations, the logistics module performs the functions of station resupply, waste return, and also serves as the propellant loading location for the TMS. Figure 4.3.2-3 illustrates a logistics module based on these requirements. A pressurized area contains the station resupply items including food (frozen and shelf stable), ECLS resupply and spares, medical spares, water, clothing, and EVA supplies. Lockers and freezers are utilized for launch restraint and storage. Propellant resupply (hydrazine) is required for drag makeup and TMS operations as well as ECLS N_2 generation. For safety, all the N_2H_4 is stored external to the pressure shell in a series of tanks. Multiple tanks permit flexibility in propellant manifesting (dependent on station time varying requirements). Additionally, these tanks may be changed out for water tanks during the initial period of station activation when a closed loop ECLS does not fully exist. Approximately 15,000 lbs of hydrazine can be accommodated in the depicted tanks. This loading fulfills the worst case scenario for 90 days make up, TMS usage, N_2 decomposition, and still has a 90 day reserve for contingency drag make up. Since the energy section had a initial fuel loading of 5000 lbs and is capable of resupply from the logistics module, a degree of conservation exists in these values. Preliminary weight analysis indicates a total (fully loaded) weight of the logistics module of approximately 27,000 pounds.

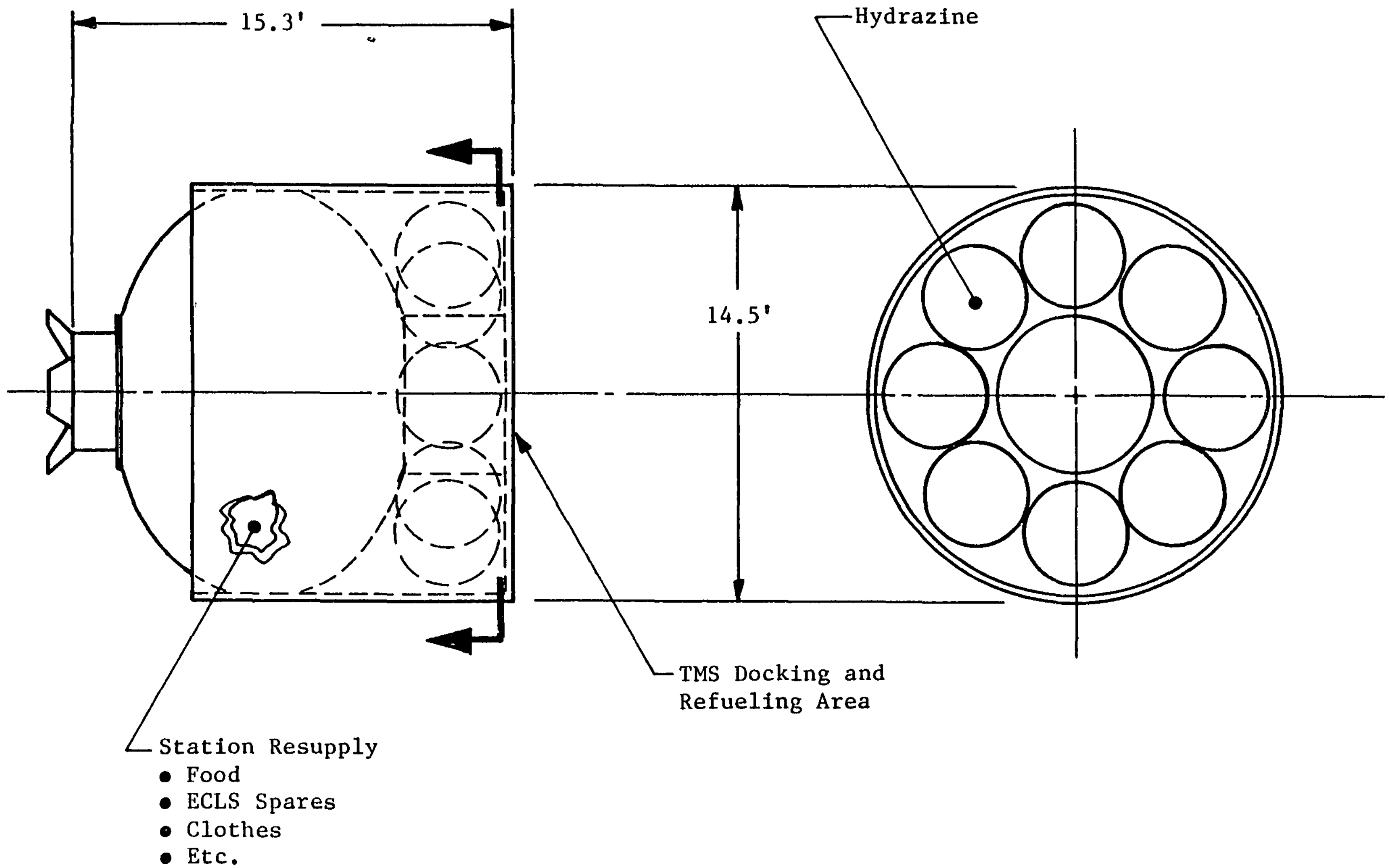


Figure 4.3.2-3 Logistics Module

The TMS refueling operations take place at the rear of the logistics module where there are docking provisions and the necessary propellant transfer equipment.

Our sizing analysis was based on a crew of 8 men for 90 days. The preliminary crew sizing analysis indicates a crew of up to 12 may be required in the latter 90's. There are many alternate approaches for handling an increased crew size with this logistics module. One approach would be to fly an additional logistics module to the station at times of peak need (dock two at the station). Another option would be to increase the frequency of flights. This may be appropriate since the crew change out may be staggered to allow some overlap. In any event, the total number of logistics modules will probably increase to three during the mid to late 90's. The initial compliment of two, assumes a 90 day cycle is adequate for refurbishment and changeout.

As just noted, our selected program option describes a capability build up that exceeds the capacity of the two habitat modules. Since a tunnel segment is required to connect the two hab modules (dual egress path requirement), it was decided to investigate the feasibility of using this tunnel area as an expanded crew quarters.

The tunnel length is approximately the same as the standard habitat modules, but the diameter is limited to around 12 feet due to the addition of radial docking ports. This permits a internal volume of approximately 4800 cubic feet as compared to 6500 cubic feet for the habitability module. Assuming that all the equipment (medical facilities, galley, ECLS, etc) need not be duplicated in the tunnel/module, it seems logical to accept this 26% volume reduction as compatible with the needs of the third habitat module. As the space station has unused docking ports, additional crew areas could be added as required to supplement the tunnel/module area.

The selected hangar concept is of a larger diameter than can be contained within the orbiter cargo bay. Since the space based OTV concepts have not been fully defined, the actual configuration of the hangar also remains open. But, the premise of the OTV having a diameter over 10 feet is likely, since the earth to station transport costs are still length related. To gain adequate access space for servicing tasks and allow for internal support structure, a hangar diameter of approximately 25 feet diameter was selected.

Figures 4.3.2-4 and 4.3.2-5 show a space constructed hangar concept in the partially deployed and deployed configurations respectively. Since this is an unpressurized hangar, an airlock is provided to permit transfer from the station interior via the energy section. An integral part of this concept is the expandable truss, service/construction facility. This system provides the capability to transfer payloads and stages into and out of the hangar. It performs mating/demating operations. Areas internal to the hangar can be utilized for storage of payloads/OTV's. Use of the manipulator permits capture and deployment activities to be conducted from this area.

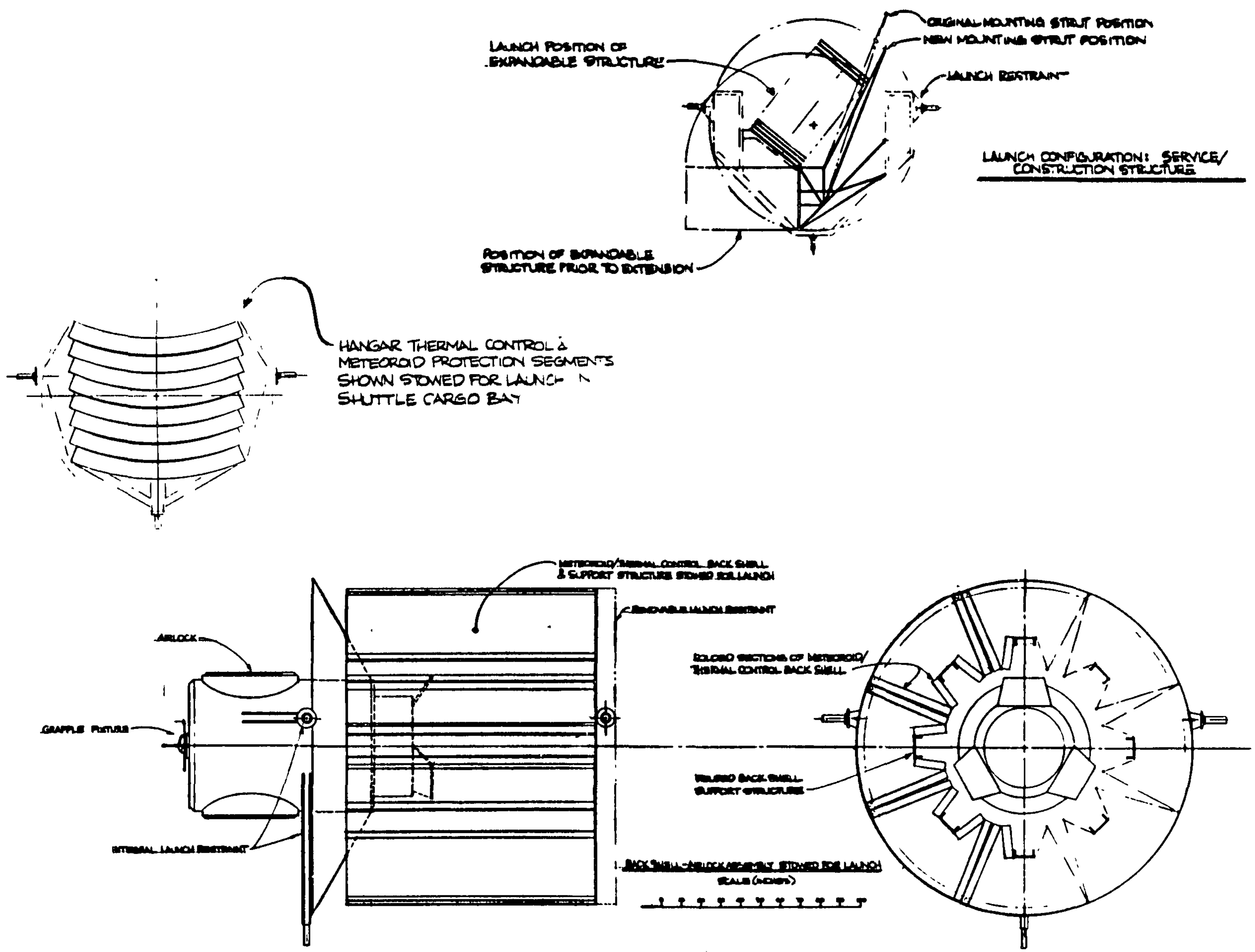


Figure 4.3.2-4 Space Constructed Hangar Concept (Stowed)

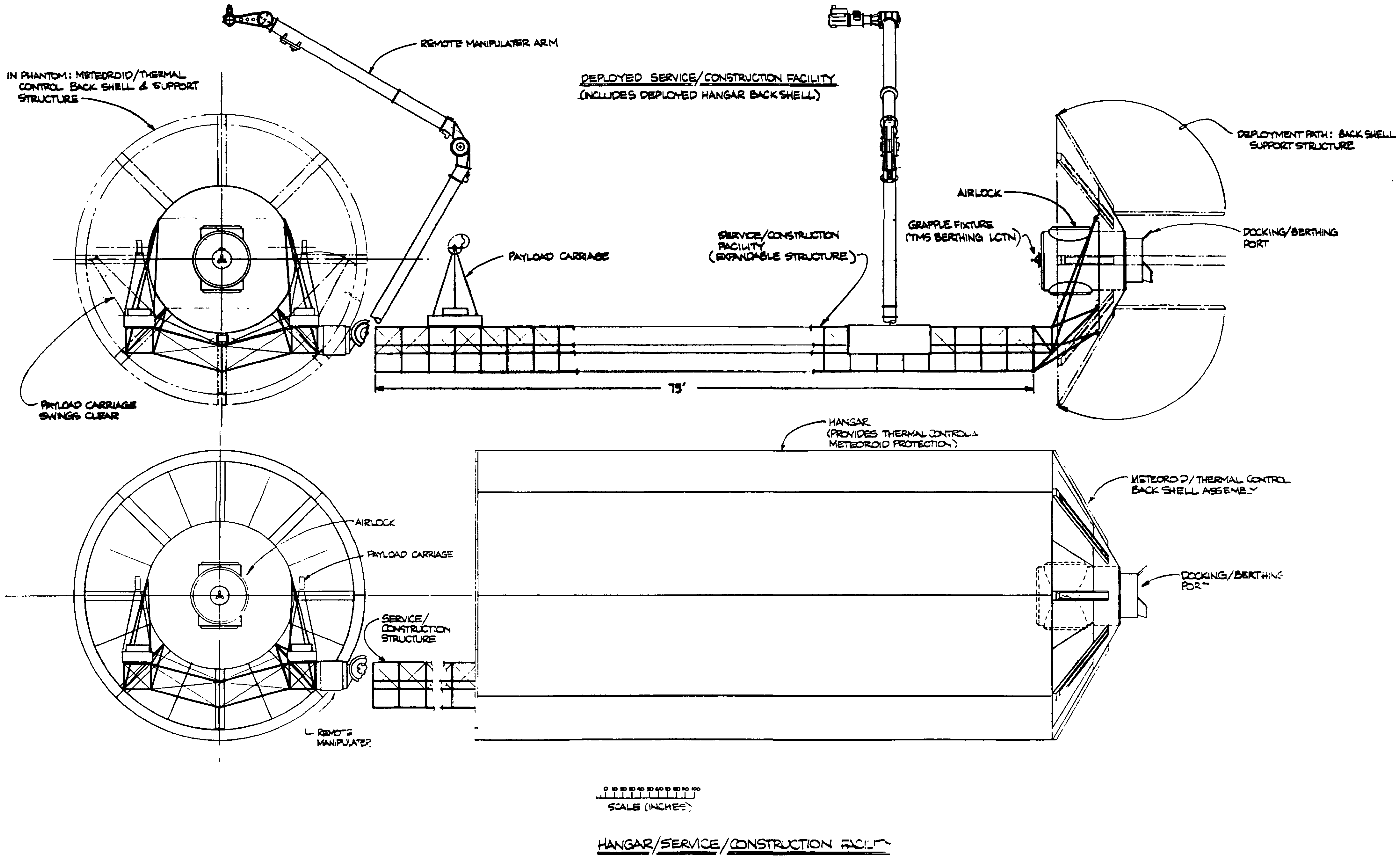


Figure 4.3.2-5 Space Constructed Hangar Concept (Deployed)

Page intentionally left blank

Page intentionally left blank

A unique hangar approach is shown in Figure 4.3.2-6. This concept takes advantage of the existing station structure to limit the amount of large structures build up on orbit. The offset between the solar array center of area and the center of mass of the station (drag torques) resulted in this concept being dropped.

The remaining major built-in space station element is the cryogenic storage module. Section 6.3 describes the selection of the propellant storage system and identifies the rationale for the current total storage requirement of 70,000 lbs of hydrogen and oxygen cryogens. The location on the station is to achieve proximity to the hangar and to the station center of mass. As described in section 6.3, refueling equipment is also incorporated into this tank design. Storage life, tank design details, and tank refueling scenarios are also described in section 6.0.

The selection of the payloads shown on the depicted configuration was based on the results of an affordability analysis (Volume V) of the preliminary mission model. This resulted in a time phased mission model which identified by discipline and specific payload the date at which a payload could be available for integration into the space station system. Additionally, this analysis identified the data when platforms could be affordable. Using these results and the specific payload requirements, it became apparent that many of the payloads could be best suited and would be time matched with the platform approach. Therefore, the payloads selected for direct attachment to the station were determined on the basis of compatibility, availability, and potential station related servicing advantages. Earth observations and electrophoresis payloads predominate since the station environment (orientation/viewing and microgravity) is compatible and there are servicing needs (volume I has payload requirement data). The payloads shown on the figure are:

Earth Observations:
- o Imaging spectrometer (pallet payload)
- o Synthetic aperture radar and microwave radiometer (pallet payload)
- o Geosyn. satellite sensor intercalibration (pallet payload)

Materials Processing:
- o Electrophoresis factory/supply modules - 4 (direct mounted)

Astronomy:
- o SIRTF (pallet payload)
- o Starlab (pallet payload)

Space Physics:
- o Space plasma effects upon large spacecraft (distributed sensors)

The selected option (section 3) identified when the various platforms (earth observation, astronomy, materials processing, etc) fit into the space station system evolution. A number or all of these payloads would be expected to transition from the station to the more advantageous, dedicated platform environment. Although no specific replacement payloads are identified, the expectation is that there will be a steady stream of payloads coming to and going from the station.

THIS PAGE INTENTIONALLY LEFT BLANK

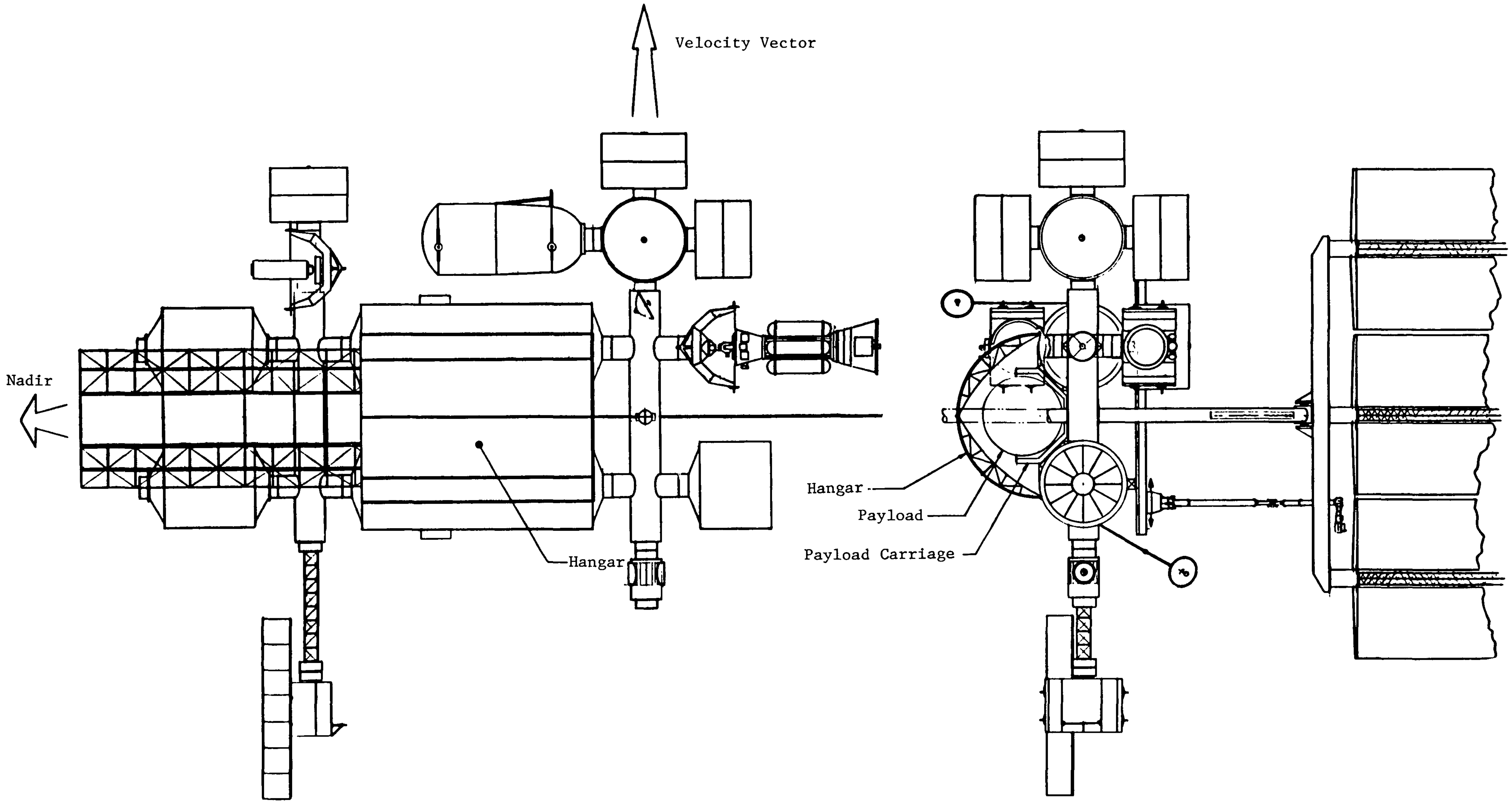


Figure 4.3.2-6 Integrated Hangar Concept

Page intentionally left blank

Page intentionally left blank

Technology development missions, and platform bound science payloads (undergoing a final test and checkout on board the station) are likely candidates for these replacements.

As identified in 4.2.2, the access and servicing of payloads and station build up requires the capability to transfer large elements around the station. A proposed approach is shown on the drawing in terms of a traveling space crane or manipulator arm. By attaching a guide/track system to the station as it is being constructed, the crane has the ability to transfer items from the orbiter bay (hand off via STS/RMS), remove and replace payloads, and dock major station elements. A final design approach and scheme to implement the traveling crane, as the station is building, will require more depth of study.

STS docking access to the station is provided from the end of the tunnel/ module above the logistics module. At this location, the STS is well clear of the solar array envelope and has minimal potential for interference with other station elements. Rendezvous takes place along the velocity vector and final docking would be of the soft variety (i.e., assisted by the station or STS/RMS). A disadvantage to this location is the perturbation to the space station mass properties. The addition of a STS to the station at an offset (STS CG) of 110 feet to the center of mass causes a reorientation of the preferred gravity gradient attitude that would require the expenditure of RCS fuel to counteract.

A final configuration related item to be discussed is the thermal control system radiator approach. This modular build up concept provides adequate heat rejection surface area by using only body mounted radiators. The more efficient, but potentially technology limited, deployable, anti-sun oriented radiator is not used. Section 6.2 presents the analysis and approach that lead to this conclusion.

Figures 4.3.2-7, 8 and 9 show the major steps involved in the modular build up process. It is apparent at the initial deployment step, that the preferred gravity gradient orientation is not identical to the mature station orientation. At this point in the study a decision has not been made on a preferred solution. Either CMG's or use of the onboard RCS could maintain a more desired attitude. CMG's would be a short term solution, but are not needed in the mature era. A thruster approach would involve high propellant usage (limit cycling) but the system is required for the mature system. Since the initial configuration has a drag area of under 1/3 the full capability station, the total fuel consumption may not exceed the mature phase drag make up. A really balanced configuration is not approached until the hangar is added at about STS flight 10 (early 1993).

As depicted in the figure, the build up proceeds quite smoothly using initially the STS RMS for build up, and then relying on station manipulator assistance for the larger reaches. At flight number 6, the first instance of moving an element prior to the addition of a new element occurs. This temporary relocation of a payload grouping would cause a downtime for these experiments while the second energy section is being installed. The solar array addition, also at this time, is seen as needing a combination of EVA and manipulator assembly skills.

THIS PAGE INTENTIONALLY LEFT BLANK

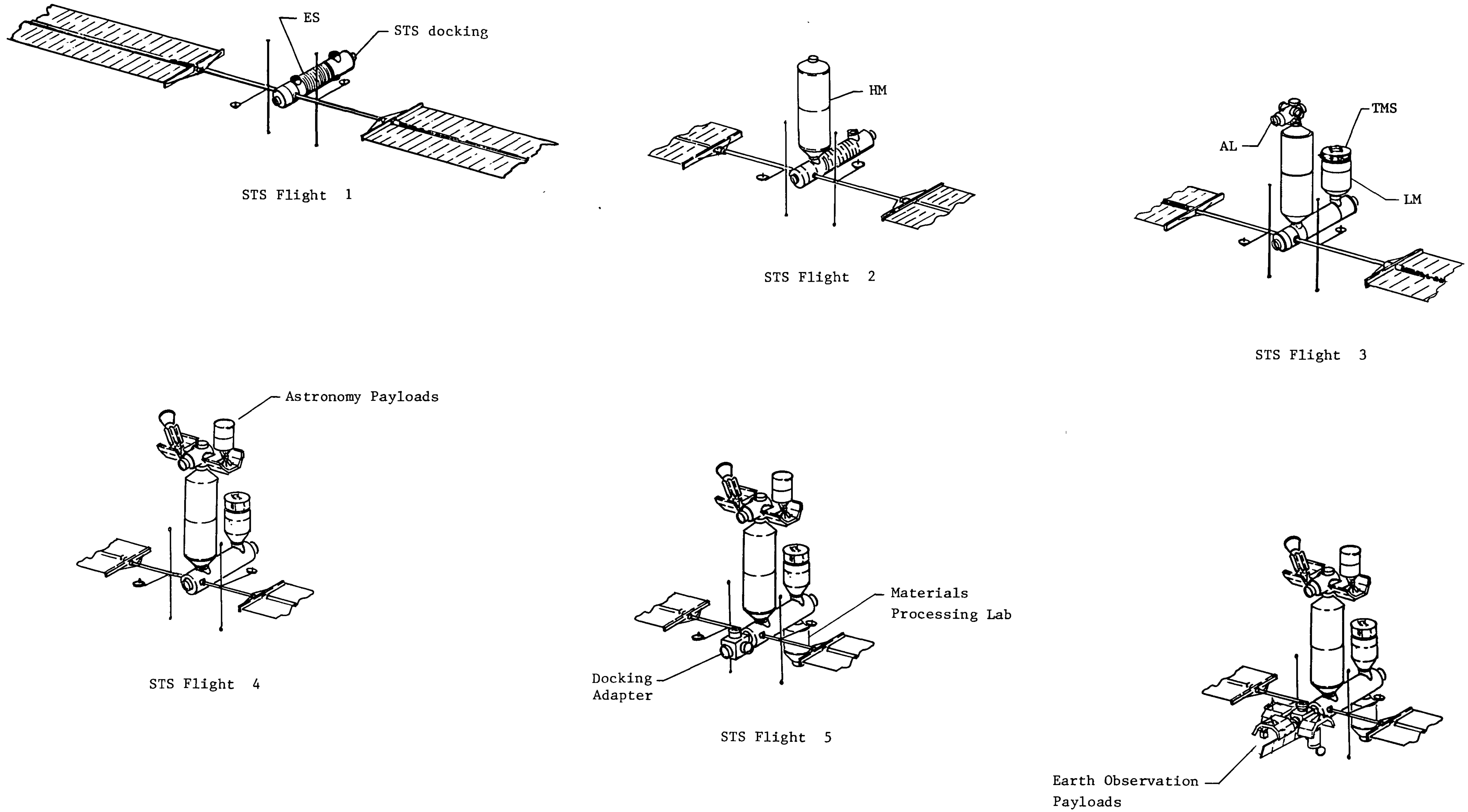
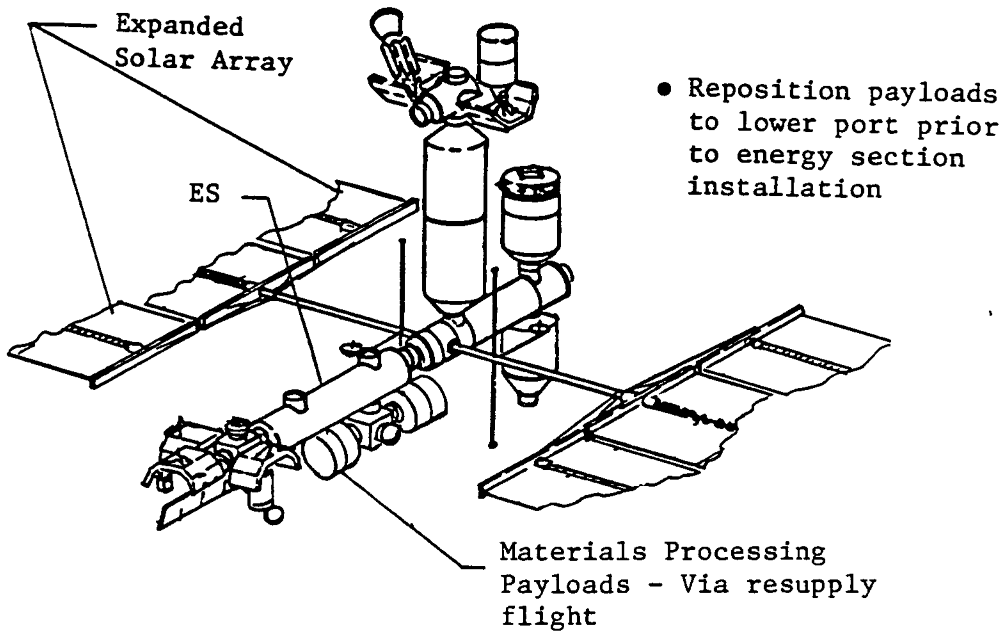


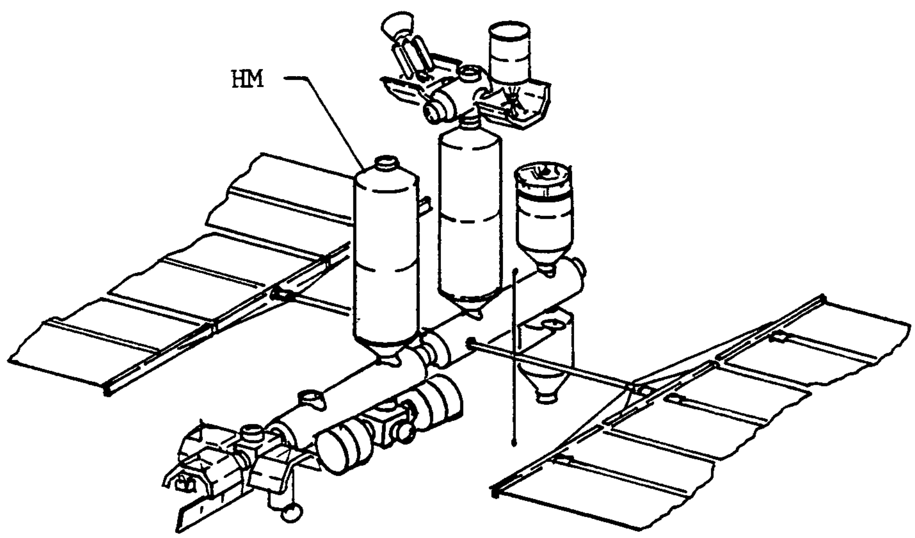
Figure 4.3.2 - 7 Cargo Bay Concept -Build Up Sequence

Page intentionally left blank

Page intentionally left blank

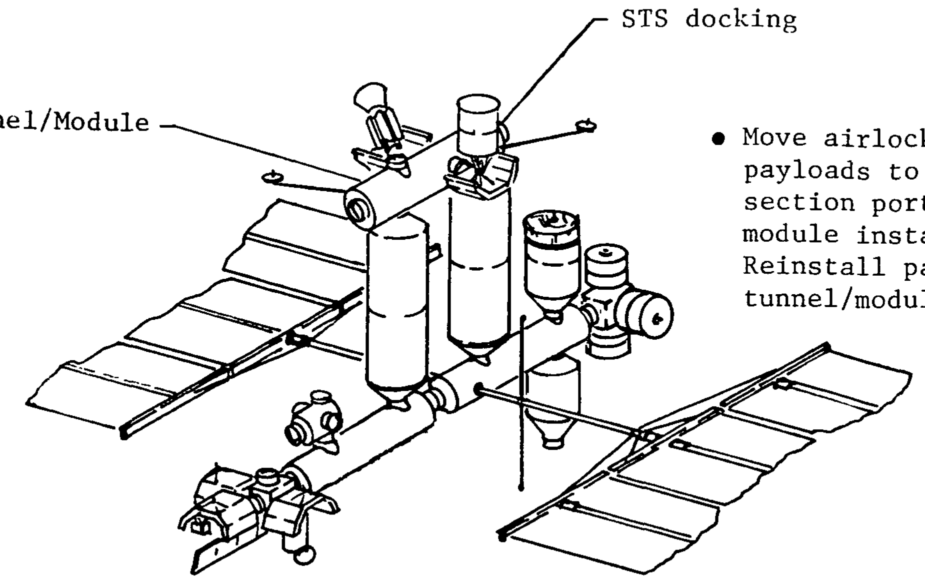


STS Flight 7

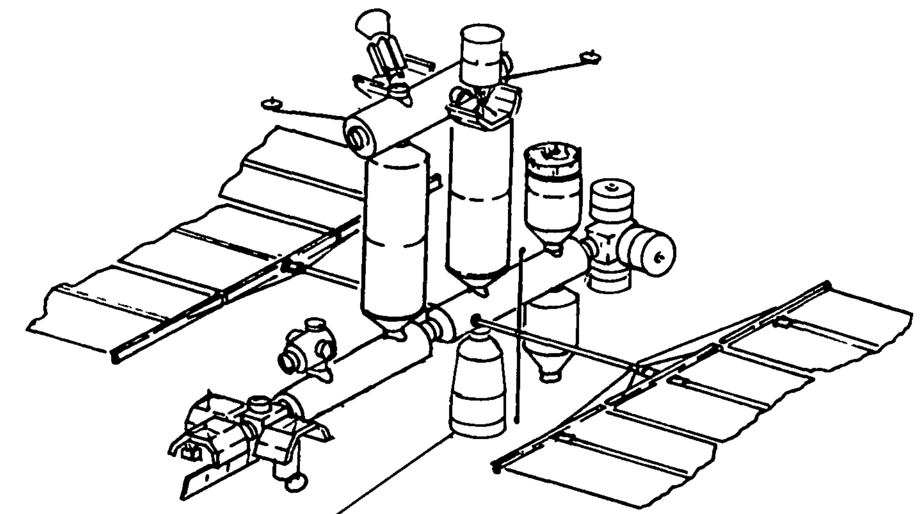


STS Flight 8

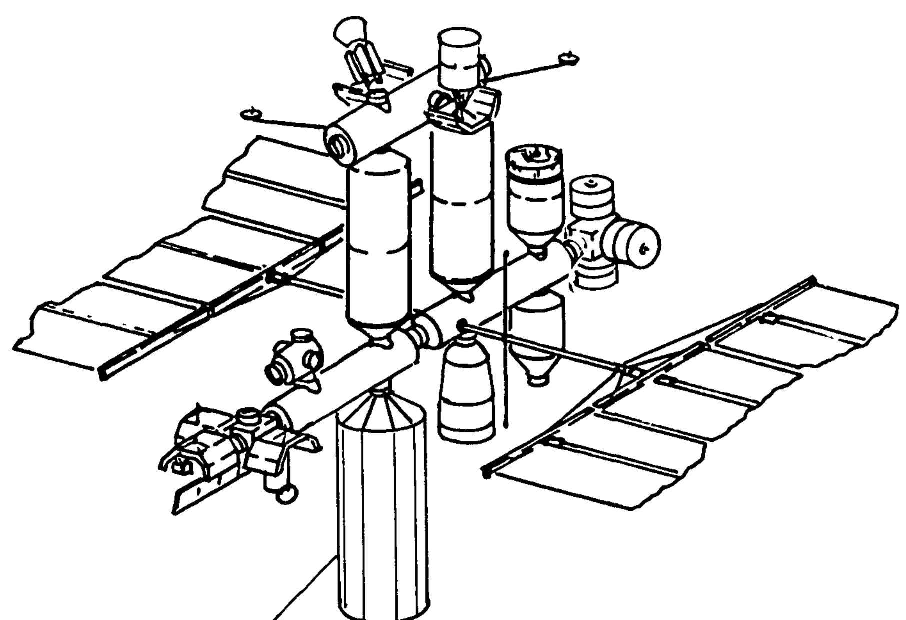
Figure 4.3.2-8 Cargo Bay Concept - Build Up Sequence



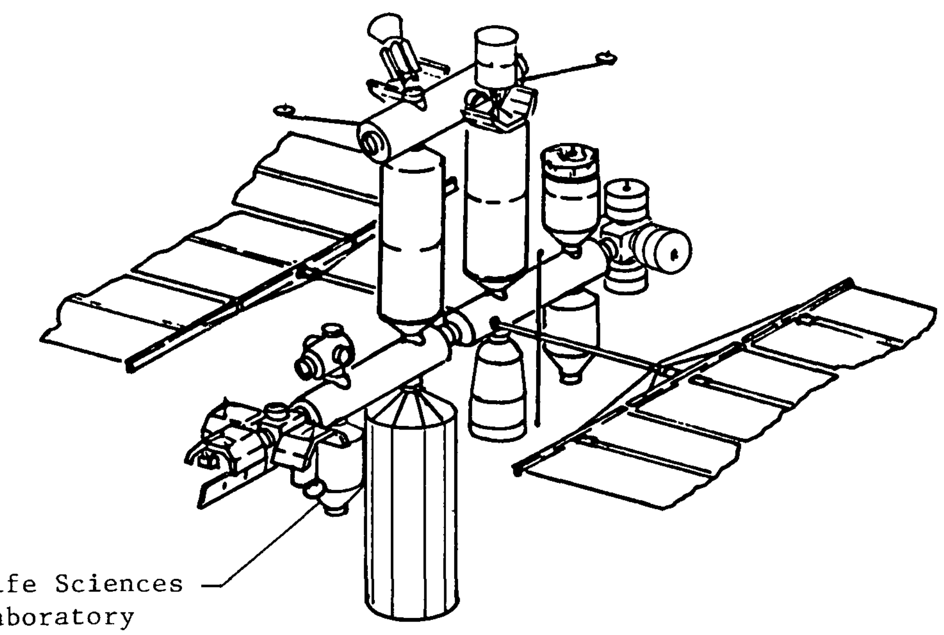
STS Flight 9



STS Flight 10



STS Flight 11



STS Flight 12

Figure 4.3.2-9 Cargo Bay Concept - Build Up Sequence

Page intentionally left blank

Page intentionally left blank

Note that the OTV and propellant storage module are activated prior to the hangar. It is felt that the first OTV flights will be of the test flight category, and as such will be returned to earth for service and checkout. When the hangar becomes fully assembled and operational, OTV servicing at the station will be started.

Figure 4.3.2-10 is a compilation of the mature station mass properties. This configuration exhibited the most difficulties in trying to establish a balanced gravity gradient attitude. Relocation of modules caused large variations in products of inertia, which resulted in large principal axes offset. This is caused, to a great extent, by the spread out nature of the design and the corresponding large separation distances between elements. The final design will require much more sensitivity analysis to determine an approach that limits these undesireable effects.

A summary of the modular-STS cargo bay concept advantages/disadvantages follows.

Advantages

- o STS compatible
- o stresses commonality/modularity
- o permits phased build up (quarters, power system, capabilities)
- o can take advantage of later technology advances
- o has good growth capability
- o permits return of auxiliary elements or payloads for upgrade
- o satisfies trade studies decisions from table 4.2.2.4-1

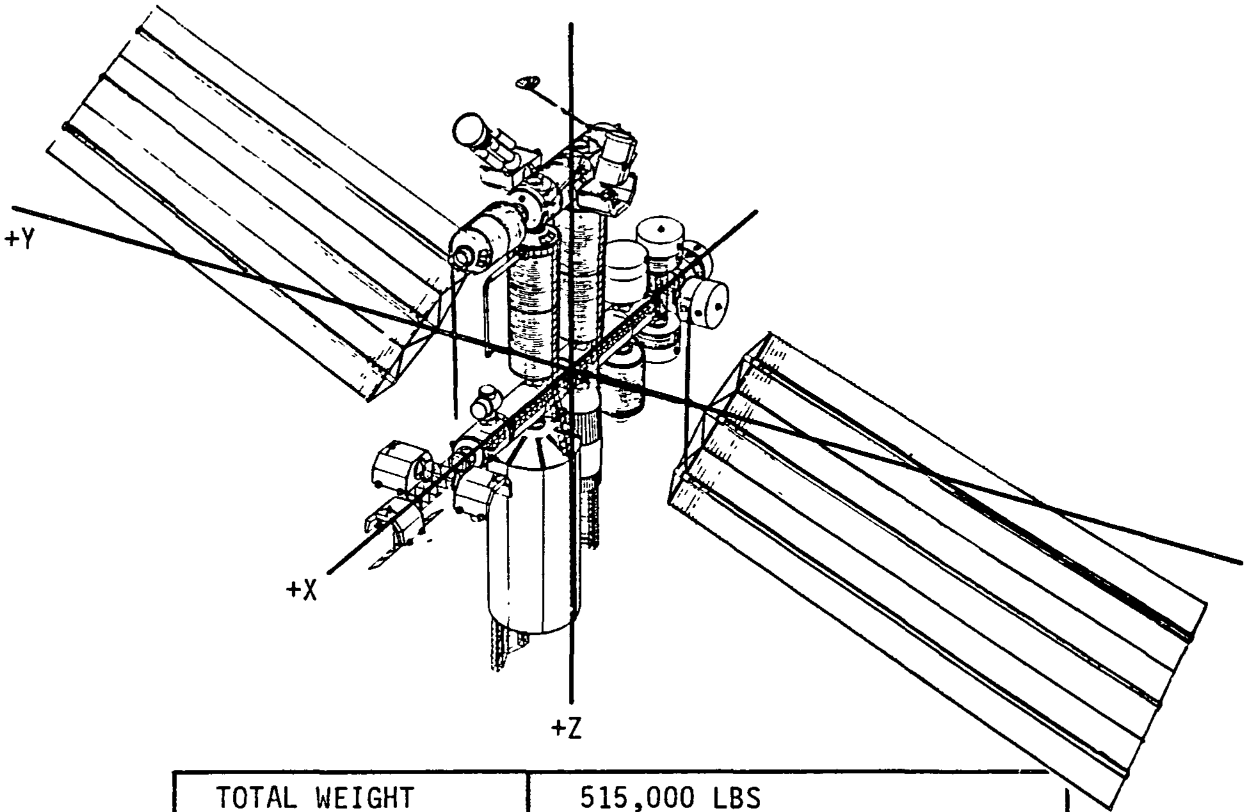
Disadvantages

- o requires multiple STS flights to reach a mature configuration
- o delayed capability implementation
- o initial configuration not gravity gradient stable
- o requires 8 STS build up flights before dual egress established
- o potentially low structural stability (multiple-connected elements)
- o traveling crane buildup lags need

4.3.3 External Tank/Aft Cargo Carrier Modular Concept

Early in this study, it became abvious that the STS flight and operational costs were a large contributor to the overall program costs. One aspect of this architectural option study was to investigate approachs for reducing the number of STS flights required to reach a mature, operating space station configuration. Since a large number of STS payloads tend to be volume limited, a concept that would permit an increase in cargo volume was considered, and on this basis, a series of studies involving the external tank/aft cargo carrier was initiated. The primary focus of these studies was on the aft cargo carrier, since it provided an additional module size capability, and could lead to potentially new configurations.

As a result of previous STS performance studies (liquid boost module, etc), it was determined that not only could a propulsion system be attached to the rear end of the external tank, but that the potential



TOTAL WEIGHT	515,000 LBS
CENTER OF MASS	$\overline{X}$ = -1.0 FT $\overline{Y}$ = -0.2 $\overline{Z}$ = -6.9
MOMENTS OF INERTIA	$I_{XX} = 2.0 \times 10^7$ SLUG FT^2 $I_{YY} = 3.0 \times 10^7$ $I_{ZZ} = 1.8 \times 10^7$
PRODUCTS OF INERTIA	$P_{XY} = 2.1 \times 10^4$ SLUG FT^2 $P_{XZ} = 6.1 \times 10^5$ $P_{YZ} = 2.4 \times 10^5$
PRINCIPAL AXIS OFFSET ANGLES	$\theta_X = 1^o$ $\theta_Y = 15^o$

Figure 4.3.2-10 Modular Cargo Bay Concept Mass Properties

existed to supplement the STS cargo capacity by adding a aft cargo compartment. The Michoud Division of Martin Marietta has been involved in aft cargo carrier (ACC) preliminary design studies for years, and has reached a high level of maturity in the design concept. Figure 4.3.3-1 presents a simplified overview of the major elements involved in this concept, and depicts the available payload envelope. In operation, the flight profile would approximate the following:

1) STS launched with combination of cargo bay and ACC payloads,

2) Separate ACC shroud after SRB cut off,

3) STS flight trajectory would result in a circular LEO for the orbiter/tank combination,

4) Space Station TMS would rendezvous with the STS and mate with ACC payload,

5) Cargo is released from the ACC interface,

6) The external tank is outfitted to perform a controlled deorbit burn,

7) The orbiter and TMS with payload continue to the space station.

The addition of the ACC shroud and external tank modifications contribute to a STS payload weight penalty of around 9400 pounds. The circulization requirement also results in another payload penalty of up to 2300 pounds depending on the STS ascent trajectory used.

In our studies, we evaluated the merits of two approaches based on the availability of the ACC concept. Concept one uses the ACC to transport a new, larger diameter space station module, and continues to use the STS cargo bay for 14 foot modular elements. Concept two takes advantage of the injection of the external tank and uses it for a basis of the space station. This configuration also uses ACC derived elements as well as STS modular elements to continue building the station.

Since the external tank station ends up looking very similar to the Shuttle Derived Vehicle version (section 4.4), no configuration sketch is presented. A comparison of the concepts indicated a preference for the first approach. A summary of the reasoning follows. Although the external tank presents a tremendous amount of volume and materials on orbit, there are still concerns on how best to activate and use this resource. Current thoughts are to use the tank as only a "strongback" or to initially use the tank as the backbone (auxiliary modules would provide habitat and service areas) and outfit the interior as crew quarters, hangar, etc during the station's operational era. In either version, the tank must be compatible with the station needs, requirements and environments. The tank insulation, for example, must be tested and investigated in terms of space station contamination. It is premature to assume that this or a new material could not be made compatible with the launch and space environment requirements. The requirements are, however, so demanding that a

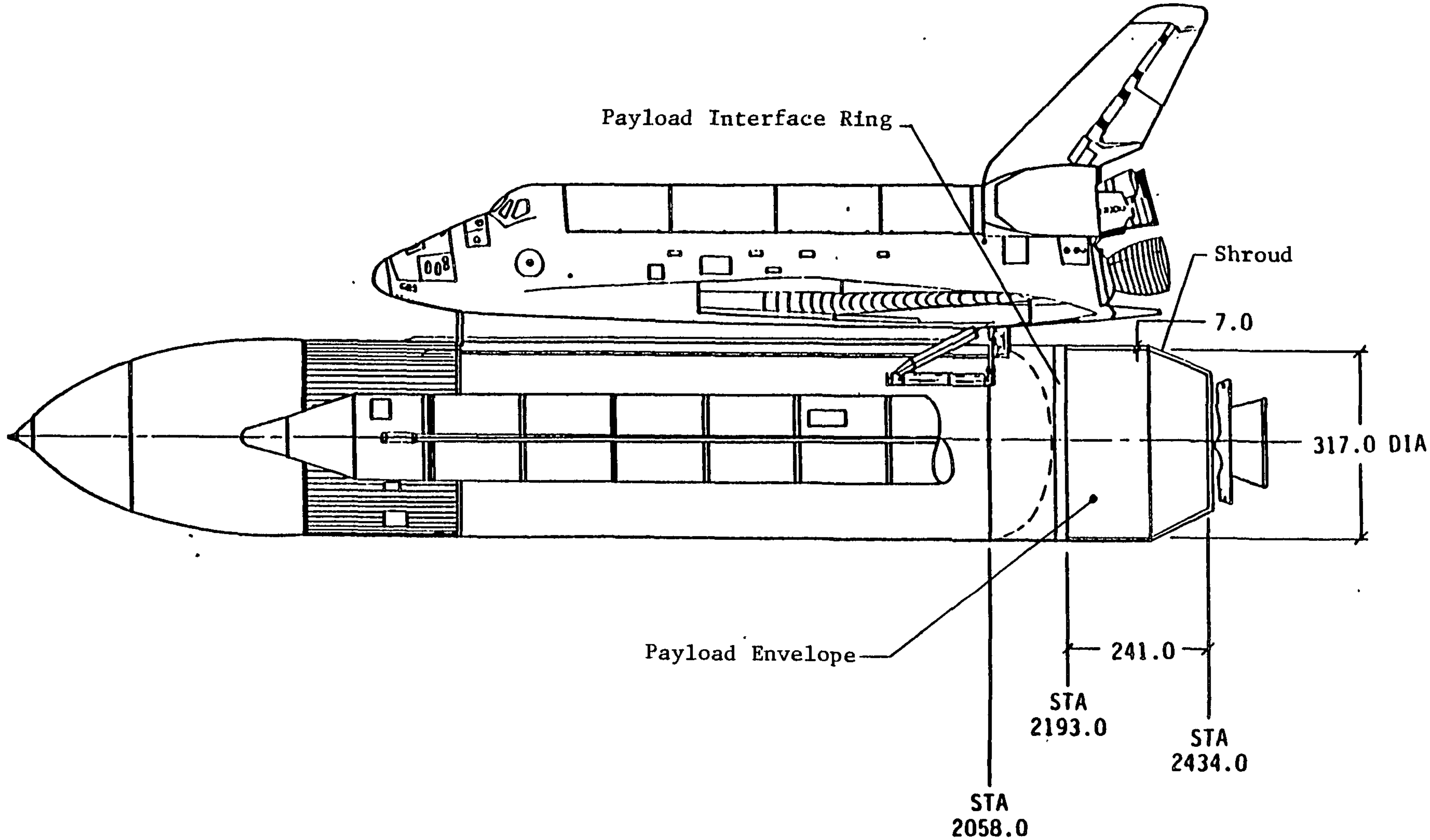


Figure 4.3.3-1 External Tank/Aft Cargo Carrier Overview

better understanding of the current material behavior must be presented before a presumed solution can be accepted. Current IRAD efforts at Michoud should present continuing data towards this objective. This one factor combined with the logistics and manpower involved in on orbit activation (second usage), and the realization that the external tank stiffness contribution as a strongback greatly exceeds any known requirements, caused us to drop this approach. Because the external tank could be an addition to the station on almost any STS flight, there must be a continuing effort directed towards the eventual usage of the tank.

The recommended ACC approach is illustrated in Figure 4.3.3-2. Because this is a modular approach and there are many similarities to the cargo bay modular scheme, only the differences are highlighted in the following configuration description. Referring to the figure, the flight orientation is identical to the modular version, and the solar array size, location and controls are also the same. Because of the reduced height above the energy section, solar array shadowing would be less than on the previous concept for the same solar array boom length.

Dual energy sections of a similar configuration are used, with a length variation occurring due to specific geometry considerations.

The aft cargo carrier concept takes on a new look at the habitability module level. Pursuing the size and volume capabilities of a new module based on transportation with the carrier, lead to the habitat module shown in Figure 4.3.3-3. As indicated on the figure, the volume and floor area compatibility with the cargo bay module indicate that this is a viable useage. An alternate floor layout (baloney slice) was investigated, but only a single floor could be achieved because of ECLS equipment (above ceiling and below floor) and minimum ceiling height requirements. Actual equipment arrangements and layouts were not completed during this study. A launch deployed meteroid shield/radiator is depicted because the compatible volumes and areas could be achieved without undertaking the technical challenges of a deployable on-orbit arrangement. The on-orbit addition of the aft docking port is due to the loss of volume ($1300 ft^3$) associated with a launch deployed port.

The growth or mature configuration incorporates three hab modules compared with two modules and a tunnel/module on the 14 foot concept. A tunnel/module was not incorporated here, partially, to show the alternate approach as discussed in section 4.3.2. The airlock is located on top the third module to provide the EVA egress path.

The aft cargo carrier concept would use the same construction and operational approach as the other modular configuration for the logistics module, tunnel, cryogen storage module, and the aspects of payload integration. One difference is the use of an ACC derived element for the materials processing laboratory.

Another application where the larger diameter elements were deemed useful was the hangar. With the capacity to launch an integral hangar section 25 feet in diameter and over 13 feet in length, it meant that the only space construction involved would be the stacking and

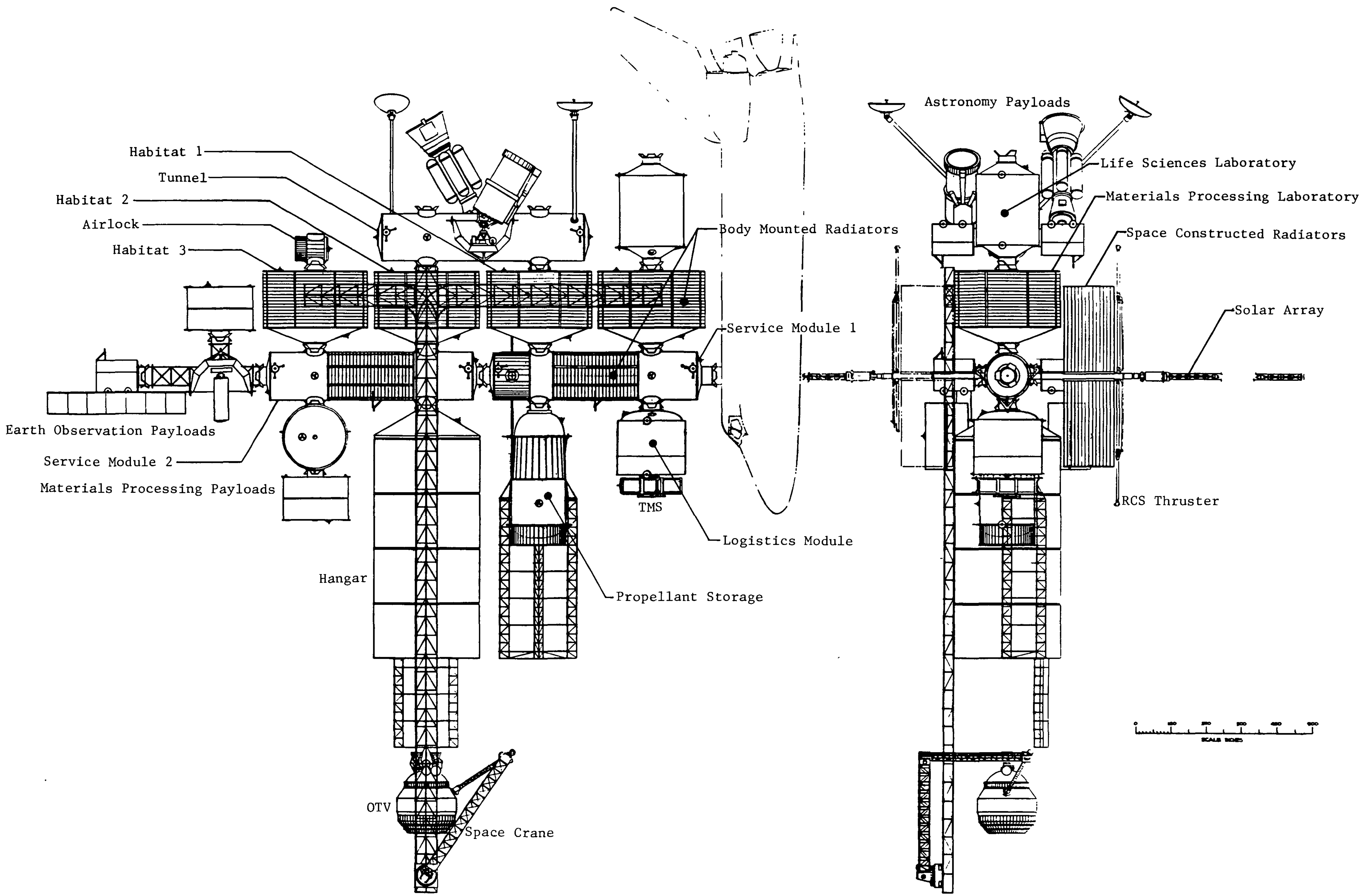


Figure 4.3.3-2 Modular Aft Cargo Carrier Configuration

Page intentionally left blank

Page intentionally left blank

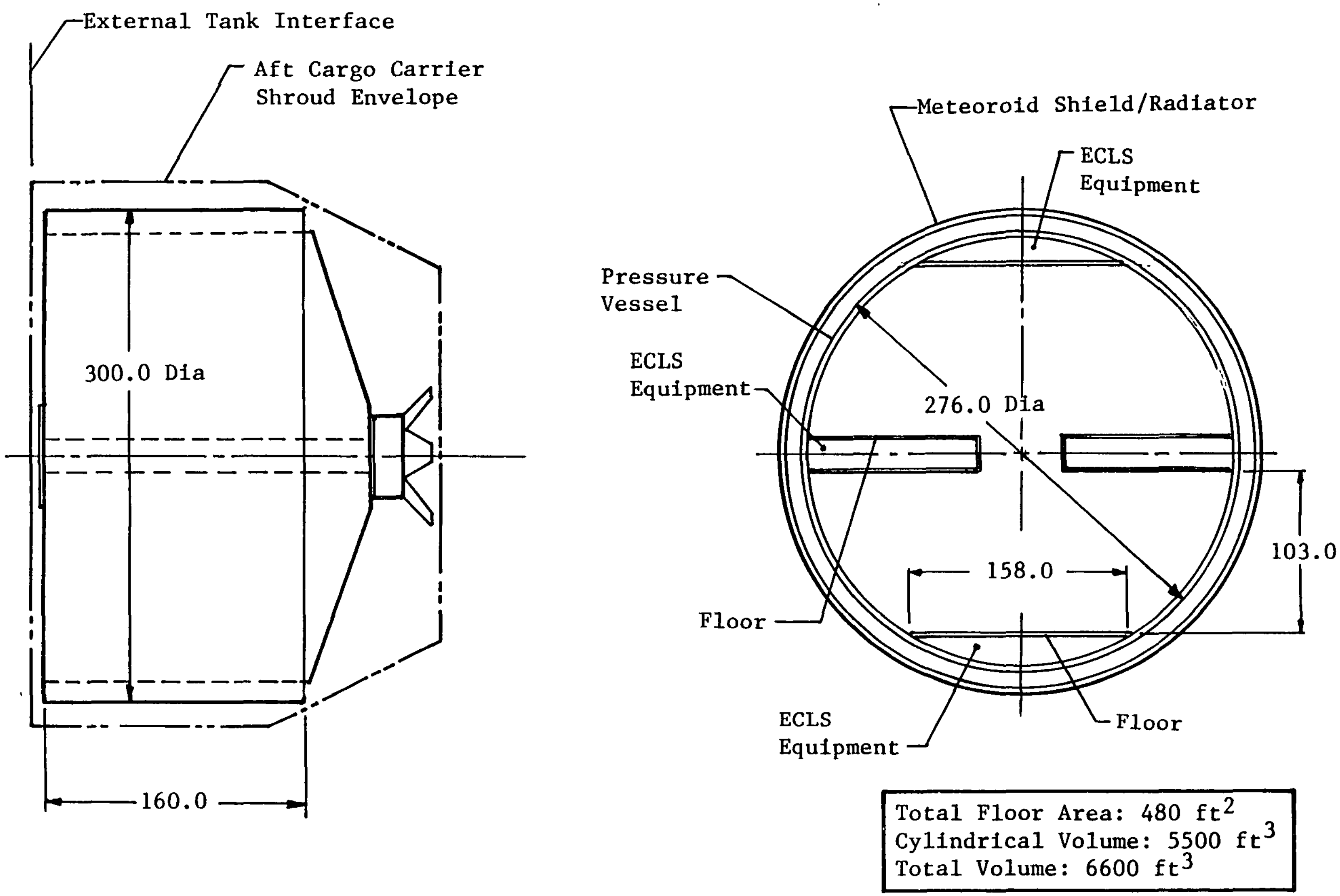


Figure 4.3.3-3 Aft Cargo Carrier Habitat Module

structural attachment of units. The disadvantages of a slow build up did not outweigh the advantages of the reduction in space assembly required by other approaches identified. Since the OTV operations are also phased, a case can be made for a stepwise growth in the hangar capacity. With four ACC flights (shared with normal delivery or station resupply flights), a hangar exceeding 50 feet in length could be achieved.

Thermal control analysis indicated that the larger diameter elements did not provide the same heat rejection surface capability as the standard modular configuration (surface area reduced by approximately 50%). This resulted in the inclusion of the rotatable, deployed heat pipe radiators shown on the figure. Design studies did not result in a solution that would allow these radiators to be packaged with and deployed with the other energy section appendages. At this time, the heat pipe radiator panels must be assembled one by one into the integral, boom mounted heat exchangers. Concern over the required rotary fluid joint lead to an alternate approach where the panel would provide tracking over an angle of $\pm$ 90°. By limiting the swept angle, flex lines could possible be used in place of the rotary joint. For a Beta angle of 0°, this approach would result in the radiator plane being parallel to the sun line except for those periods of the orbit between sunrise, or sunset and local horizontal (approximately 20° each).

The advantages of the ACC configuration in terms of reduced STS flights can be seen in the build up sequence (Figures 4.3.3-4 thru 6). A minimum of two STS flights (station build up flights only) can be saved between the first launch and the year 1995.

Figure 4.3.3-7 presents the results of the mass properties analysis performed on this configuration. The somewhat more compact arrangement lead to a more easily balanced configuration.

Table 4.3.3-1 lists the current assessment of the advantages and disadvantages associated with this concept. The one item that bears discussion is the disadvantage issue about orientation control. Initially the space station configuration (through STS flight 7), does not achieve a balanced gravity gradient orientation. During this period external control (RCS or CMG's) is required to maintain the desired flight attitude.

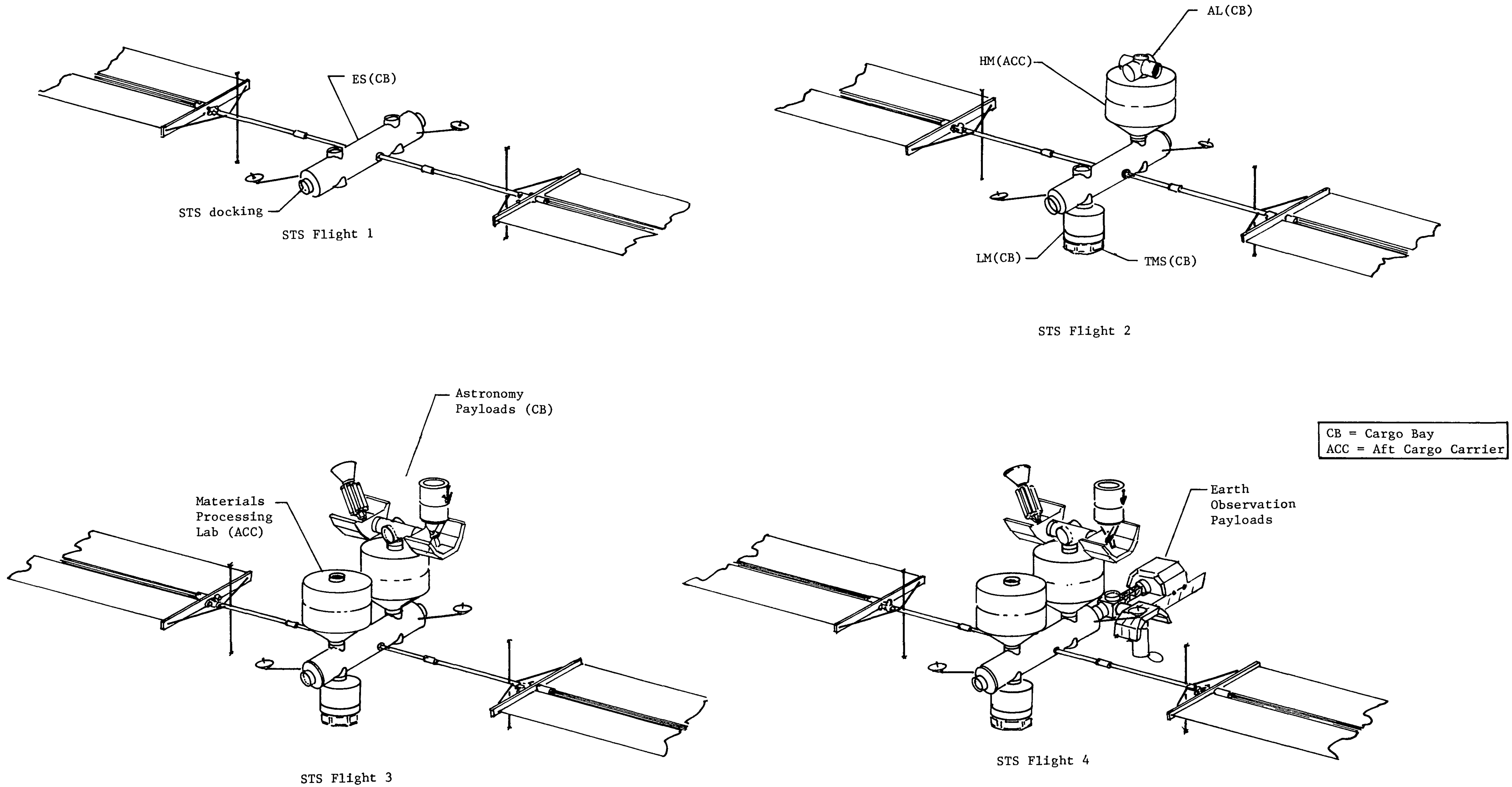


Figure 4.3.3-4 Aft Cargo Carrier Concept - Build Up Sequence

Page intentionally left blank

Page intentionally left blank

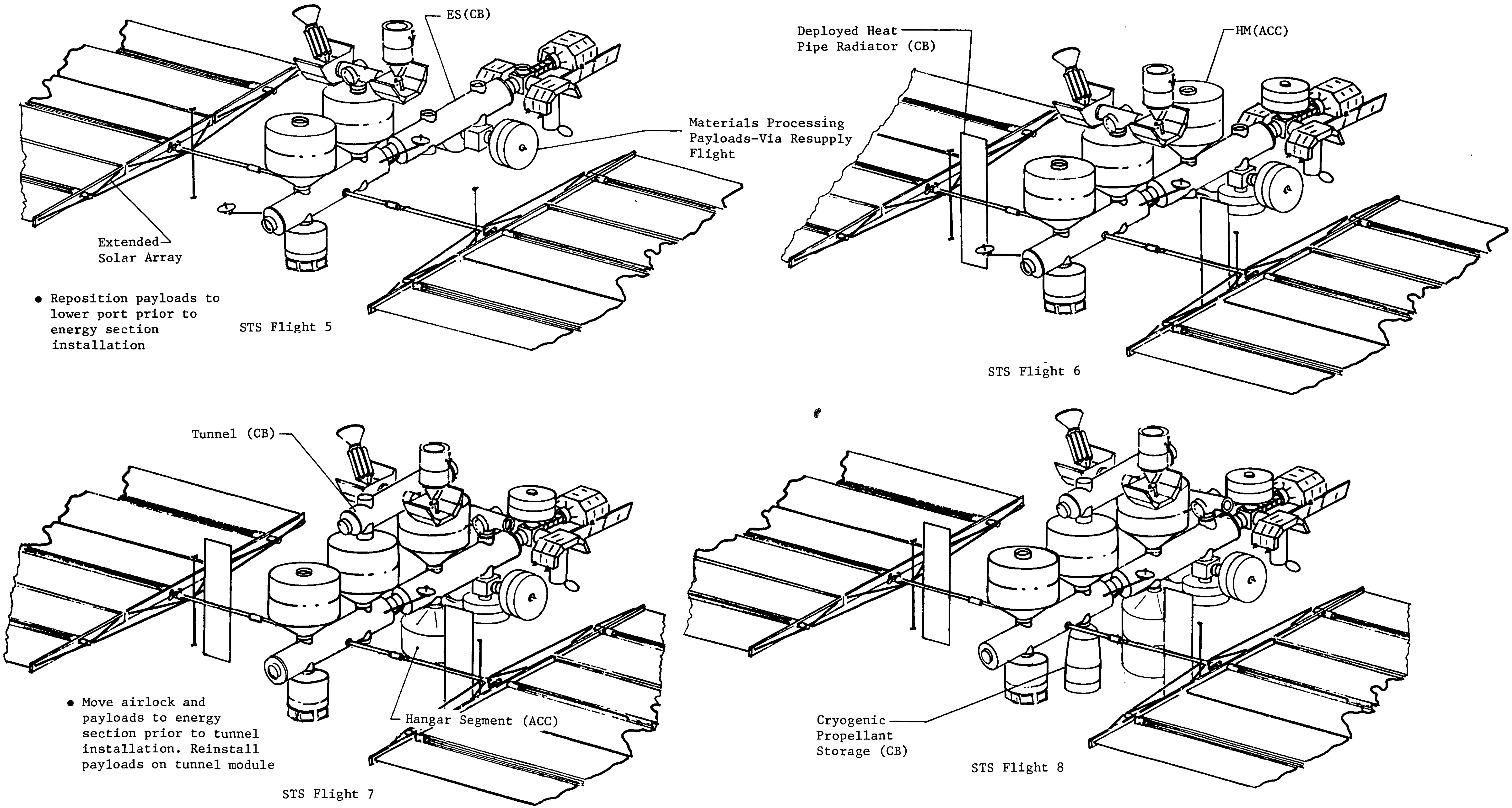
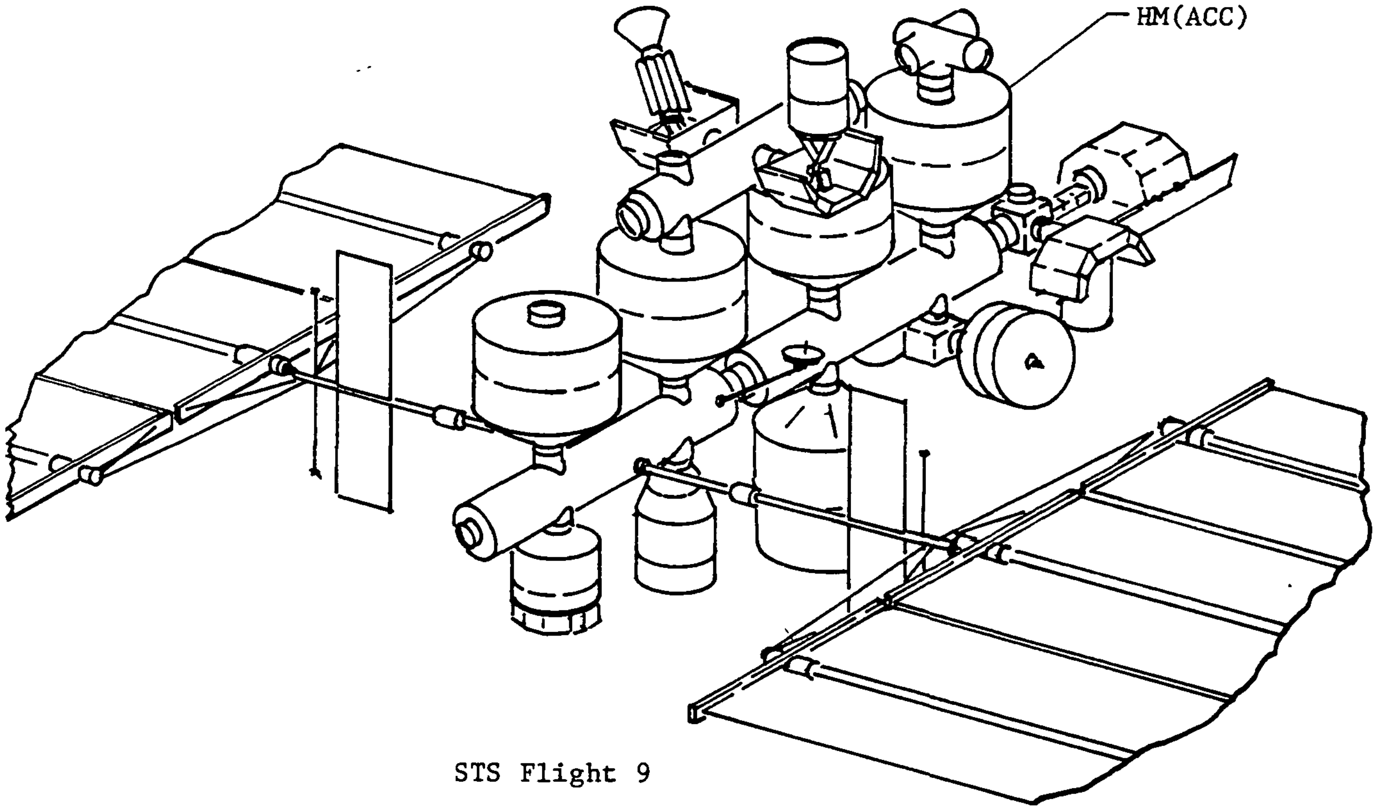


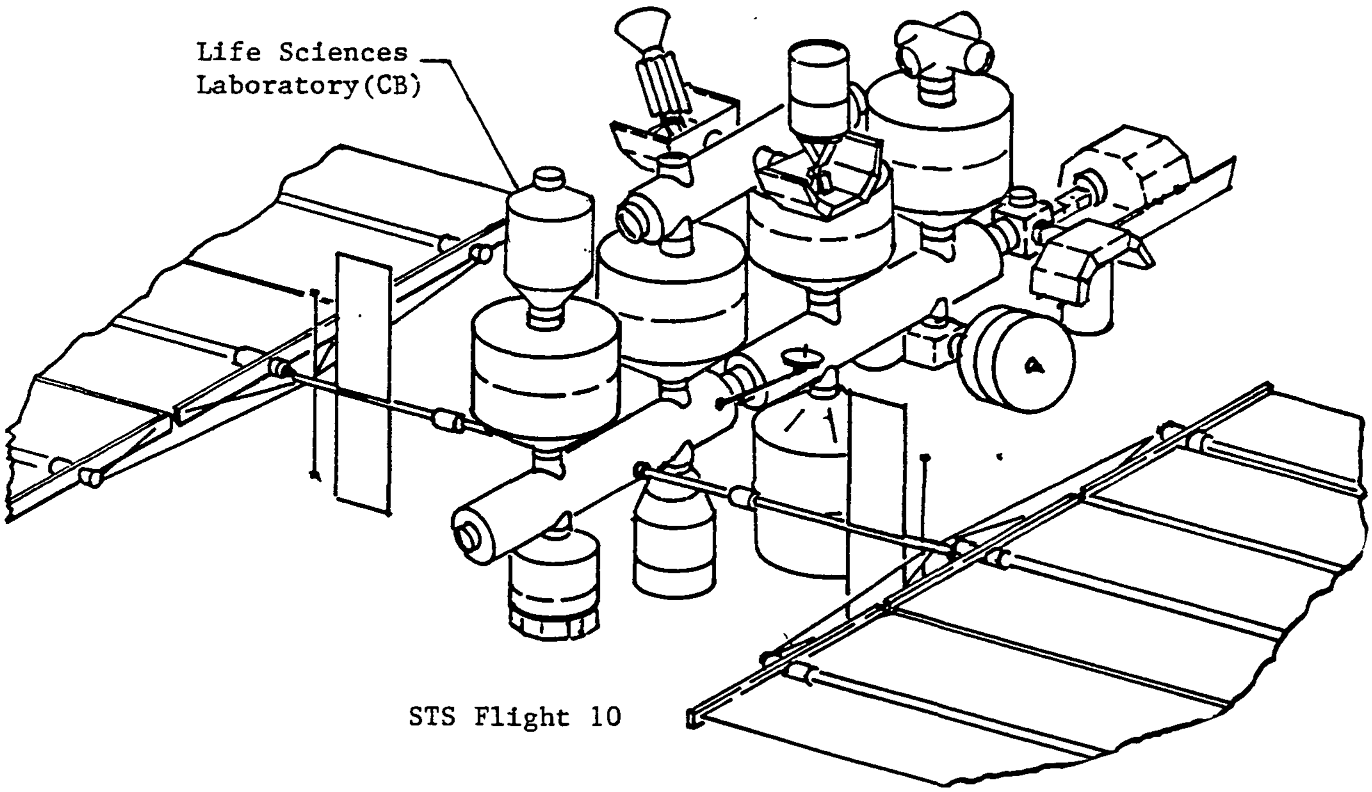
Figure 4.3.3-5 Aft Cargo Carrier Concept - Build Up Sequence

Page intentionally left blank

Page intentionally left blank

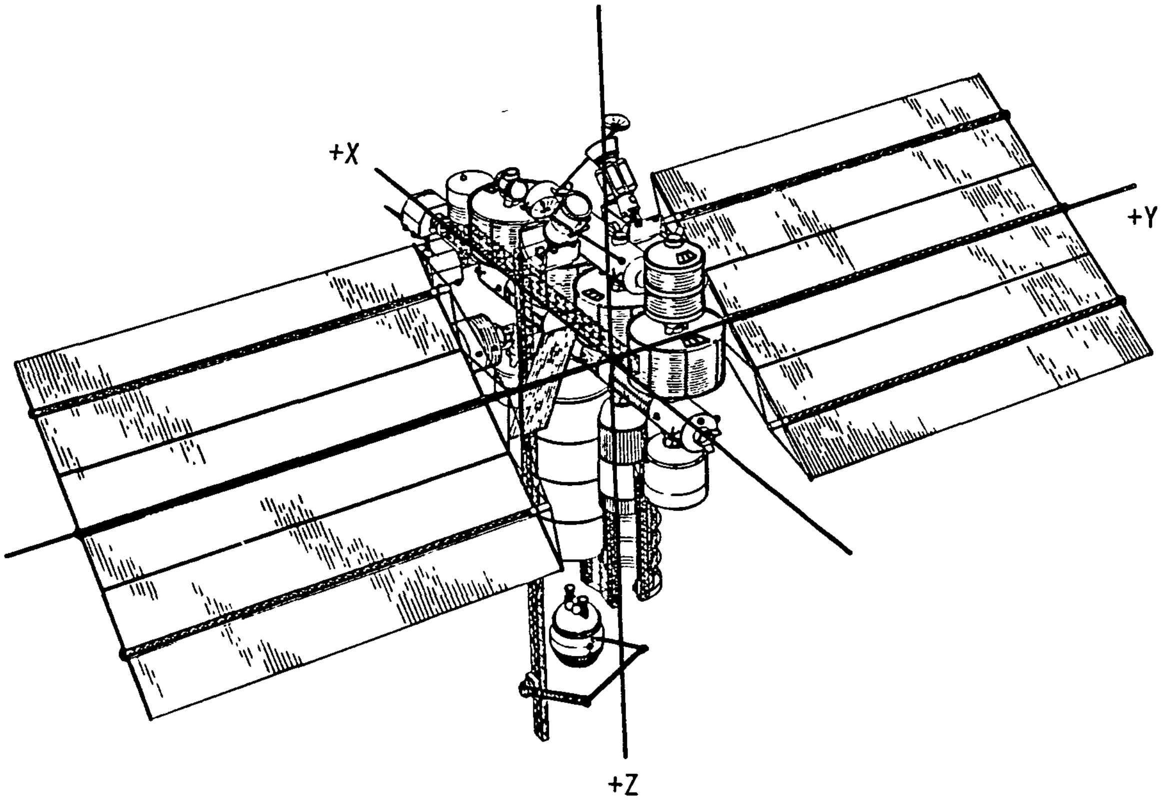


STS Flight 9



STS Flight 10

Figure 4.3.3-6 Aft Cargo Carrier Concept - Build Up Sequence



TOTAL WEIGHT	502,000 LBS
CENTER OF MASS	$\overline{X}$ = +6.8 FT $\overline{Y}$ = -0.2 $\overline{Z}$ = -1.8
MOMENTS OF INERTIA	$I_{XX} = 1.3 \times 10^7$ SLUG FT^2 $I_{YY} = 2.5 \times 10^7$ $I_{ZZ} = 1.9 \times 10^7$
PRODUCTS OF INERTIA	$P_{XY} = 7.5 \times 10^3$ SLUG FT^2 $P_{XZ} = 1.2 \times 10^6$ $P_{YZ} = 1.4 \times 10^5$
PRINCIPAL AXIS OFFSET ANGLES	$\theta_X = 1^\circ$ $\theta_Y = 11^\circ$

Figure 4.3.3-7 Aft Cargo Carrier Concept Mass Properties

Table 4.3.3-1 ACC Advantages and Disadvantages

Advantages	Disadvantages
o Reduced STS transportation costs	o Safe haven concern prior to second hab module and tunnel
o Permits larger diameter elements without assembly	o Potentially low stiffness
o Works volume limited problem	o Initially requires active orientation control
o Permits phased growth	o Requires an additional sized module
o Includes additional growth capability	o Reduces STS cargo bay payload weight capability
	o An abort causes loss of ACC cargo

4.4 SHUTTLE DERIVED VEHICLE

Since one of the charters of this study was to "do some innovative thinking", a unique one-step space station concept was developed around the concept of the Shuttle Derived Vehicle (SDV). Ideally, this approach does not meet the requirements of the statement of work where the following is identified.

- o The permanent facilities defined during this study will be Shuttle launched and Shuttle tended, as required. The Space Shuttle User's Handbook shall be used to provide the associated guidelines.

Although the user's handbook does not contain any reference to the shuttle derived vehicle, the configuration and trade study analyses performed to date show potentially significant advantages by using this approach.

A brief review of the shuttle derived vehicle is presented here for those reviewers who have a limited background in this subject. Two concepts have been recently proposed as methods of increasing the delivery weights and volumes of low earth orbit payloads. These approaches share a common STS major element heritage: the external tank, space shuttle main engines, solid rocket boosters, and orbiter avionics. The first configuration known as the shuttle derived cargo vehicle (SDCV) retains the standard external tank and solids. In place of the orbiter, a cargo carrying element is used (Figure 4.4-1). An expendable payload module and a recoverable propulsion/avionics module are combined to create this element. In this configuration payloads are carried internal to the payload module and are deployed after separating the two halves of the structural assembly along the longitudinal centerline. A lifting body configuration allows for the return of the propulsion module. Dimensions and data on the figure indicate the potential payload volume and weight characteristics.

A derived boost vehicle (DBV) configuration has also been proposed. Here, an external tank mounted payload shroud is combined with the solids and two propulsion/avionics modules. Using only two main engines and a side mount, the propulsion/avionics module configuration achieves a better lift/drag ratio which permits increased use of orbiter developed reuseable surface insulation. Figure 4.4-2 indicates the configuration and characteristics.

Our space station studies are based on the SDCV option since more detailed design and configuration data was available. Also, the added performance characteristics of the DBV could not be justified by the existing space station requirements.

The objectives behind the SDV configuration studies were twofold: reduce STS launch costs and provide other configuration related benefits (i.e., solve other configuration disadvantages). Using the SDCV as presented, provided a major advantage, in that, 44,000 ft^3 of volume could be orbited in one launch. This can within a can approach, however, could obviously be improved by outfitting the complete payload module as the station and incorporating the external envelope as the station exterior. This was the approach selected and depicted in Figure 4.4-3.

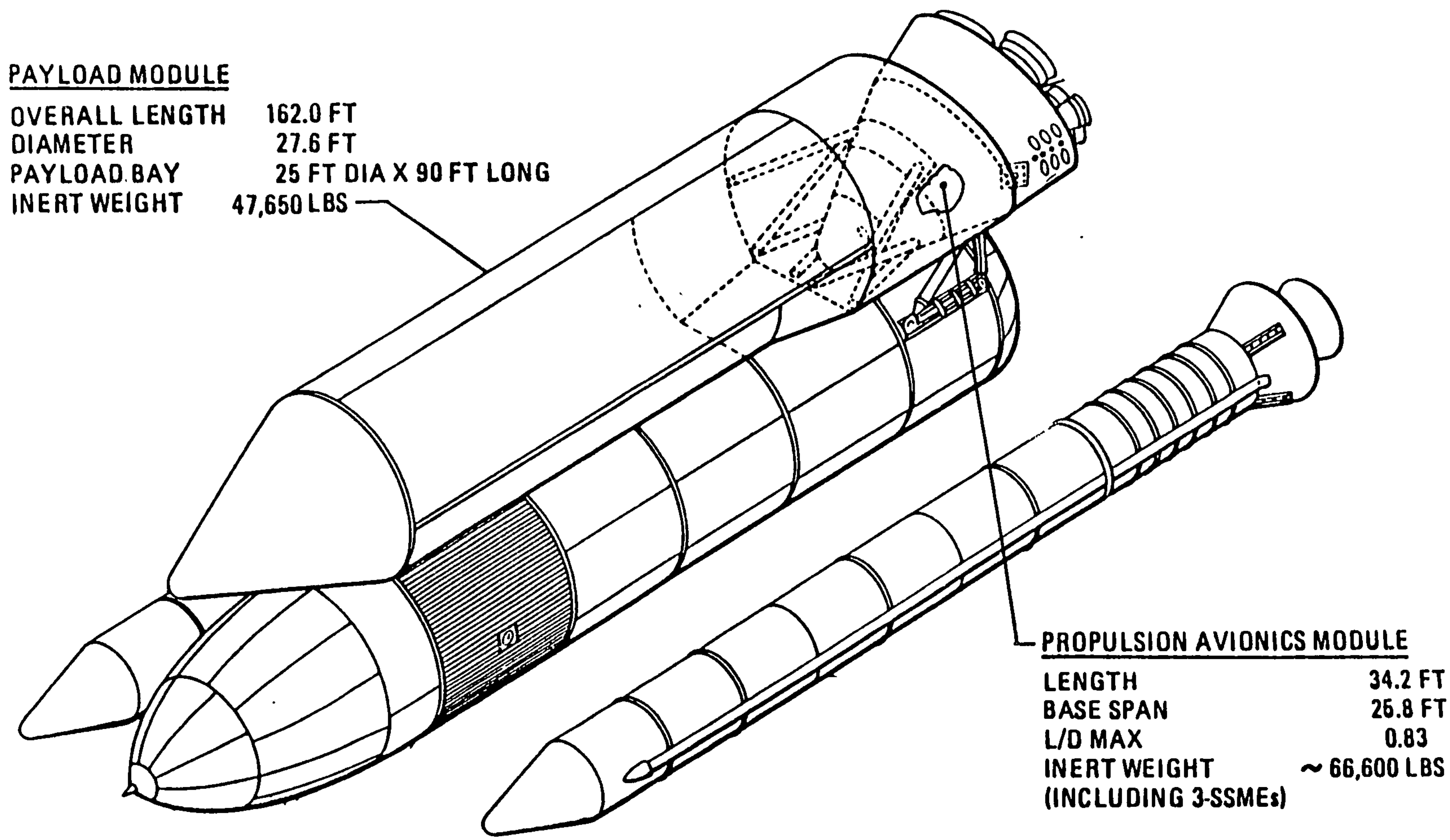
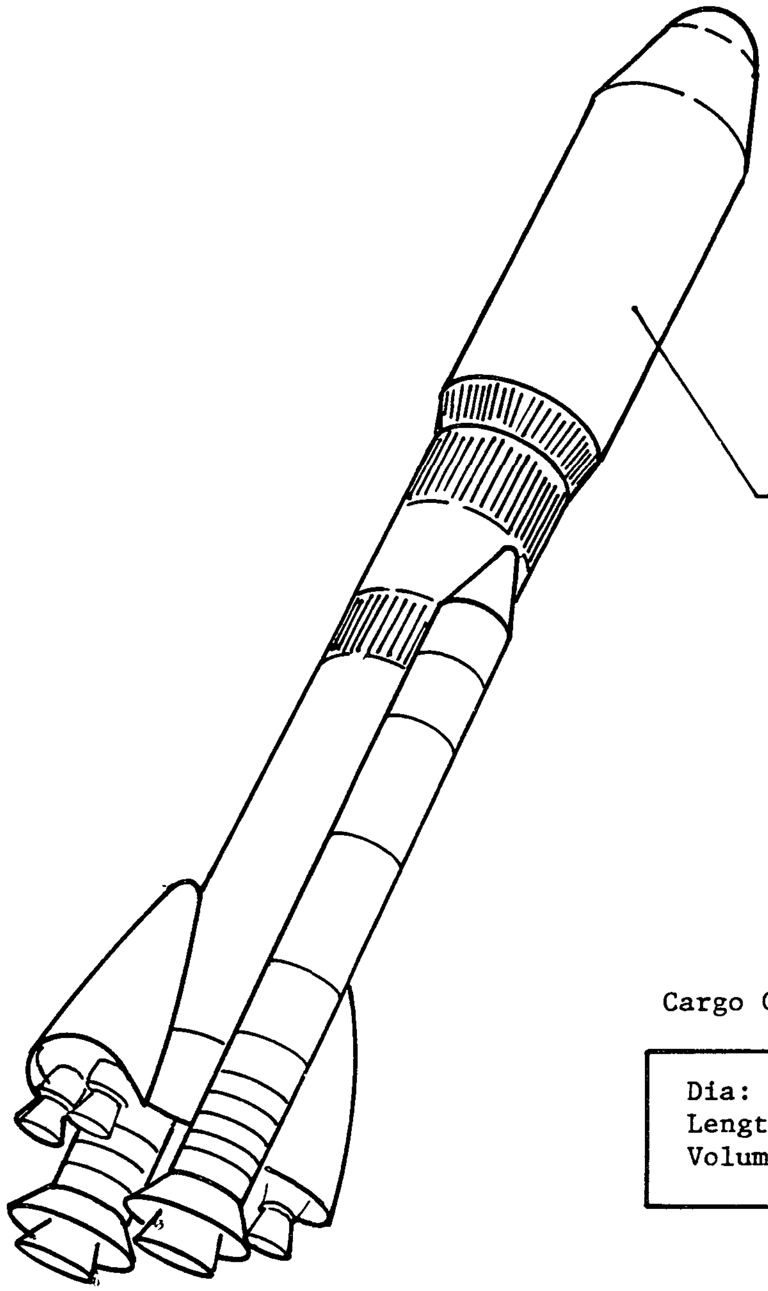


Figure 4.4-1 Shuttle Derived Cargo Vehicle



Cargo Carrier Data

Dia:	33 ft
Length:	100 ft
Volume:	85,500 ft

Figure 4.4-2 *Derived Boost Vehicle*

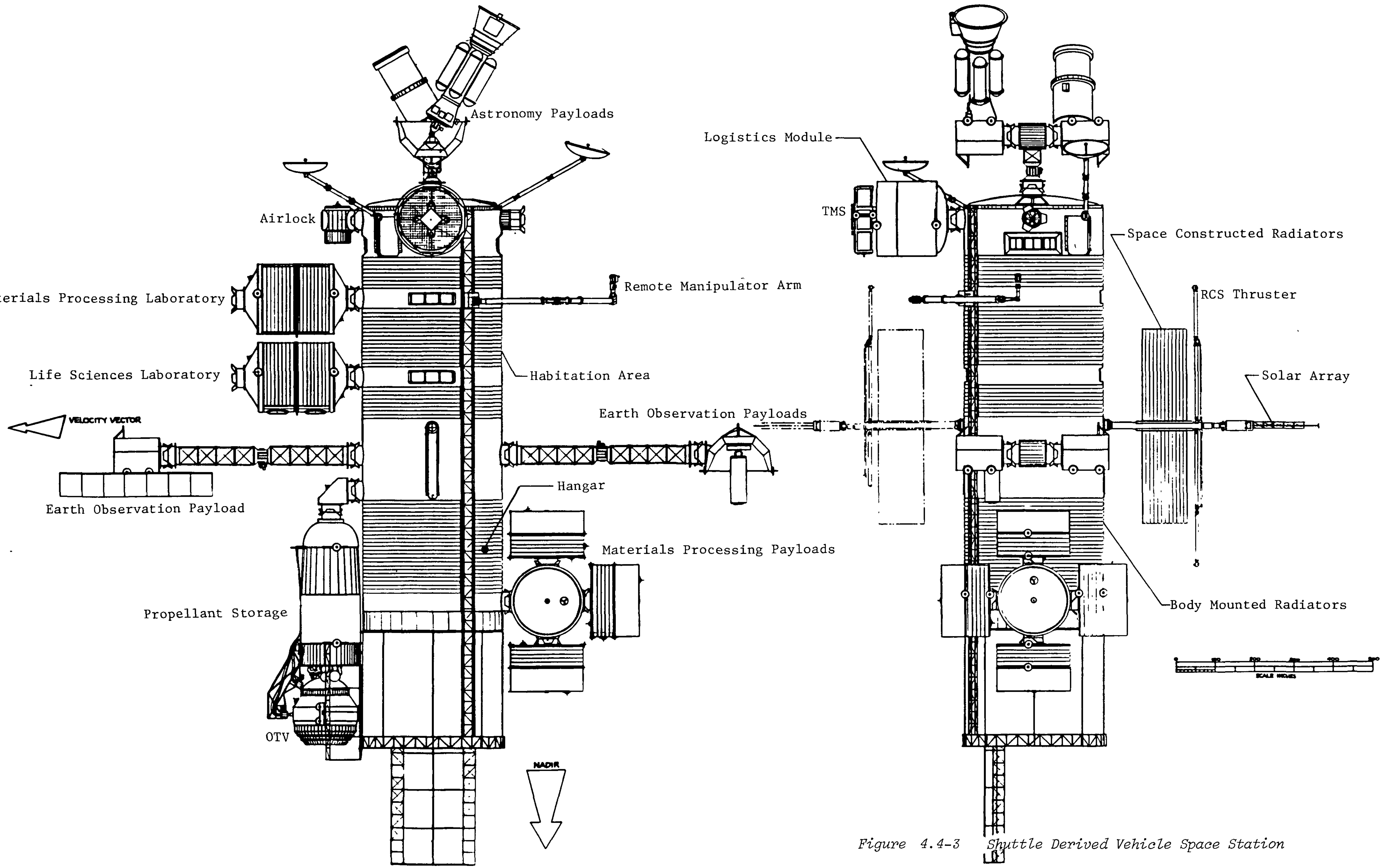


Figure 4.4-3 Shuttle Derived Vehicle Space Station

Page intentionally left blank

Page intentionally left blank

A gravity gradient attitude is also used with this configuration, and in operation, the main body longitudinal axis would be earth pointing. The flight direction indicated on the figure results in the boom mounted payload elements being located forward and aft along the velocity track. The solar array arrangement is similar to those previously depicted.

The SDV configuration is comprised of the main body element and a series of plug in auxillary modules and payloads. Included within the main body pressure vessel are the crew areas, hygiene facilities, health maintenance area, galley, recreation areas, command and control centers, subsystem equipment, and all other internal operating equipment. The pressurized volume is 27.5 feet in diameter with a launch deployed radiator/shield that brings the outer diameter to 29.5 feet. At launch the cylindrical body, which includes 35 feet unpressurized hanger, is approximately 93 feet long. Additional hanger volume has been achieved by add on sections in this mature configuration to bring the cylindrical length to 117 feet.

The selection of what station elements to locate on the interior of the SDV pressure vessel was based on the program option buildup requirements and to a lesser extent the available volume. Initial layouts indicated that a crew of 24 could be housed within the station with volume remaining for equipment and approximately the same hanger area. Since the station requirements only estimate a crew size of 12 maximum, a reallocation of space resulted in the interior space distribution as shown in Figure 4.4-4. volume allocations were based on previous study results and cursory equipment volume studies since no formalized internal layout studies were conducted. Crew quarters amount to approximately the same volume as achieved by three hab modules (or 2 modules plus tunnel/module) in the other configurations. Notice that this design obviates the safe haven concern expressed with the other approaches, since dual egress paths (internal) can be achieved at launch. Subsystem equipment located within the modular energy sections can be easily accommodated in the depicted allocation.

Both the 14 foot and ACC modular concepts used a phased growth approach for the power system. With the SDV approach and the ability to achieve more capability early, an alternate approach would be to launch the entire EPDS equipment compliment on the first flight. We have retained the step growth option on this configuration.

A hanger area is included in the main body because of available volume and the capability it provides to restrain and launch external outfitting equipment such as solar arrays, RCS booms, and communication antenna. Further studies are required to define the limits of launch mounted external equipment since aerodynamic pressures, heating rates, and fairing designs have not been addressed. Expansion of the hanger length can be accomplished by either space assembly or ACC hanger elements.

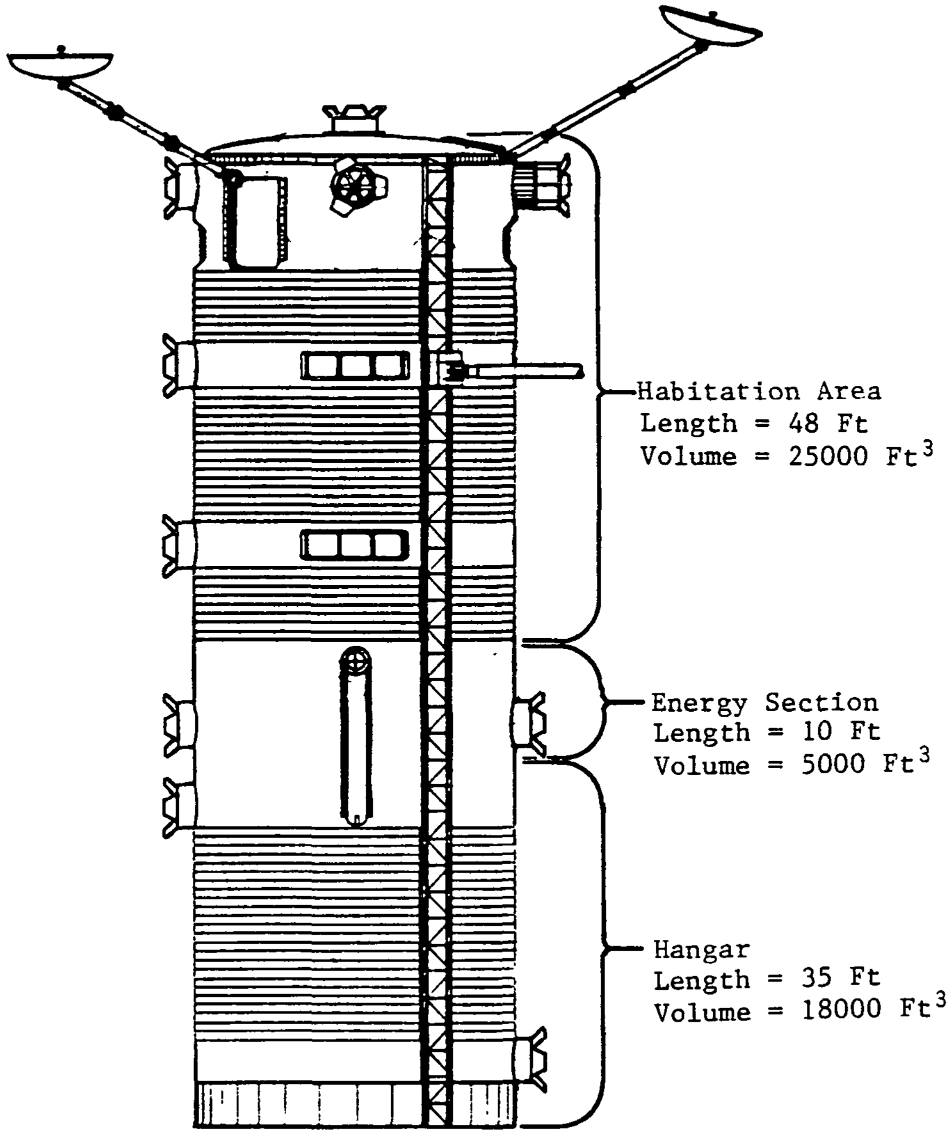


Figure 4.4-4 *Shuttle Derived Vehicle Space Allocation*

Although there is remaining space within the SDV body, it was decided not to include specific areas for the payload laboratories (life sciences and materials processing) since their activation date occurs later in the program evolution. A separate lab facility module also has advantages in terms of isolation and potential for ground refurbishment and upgrades.

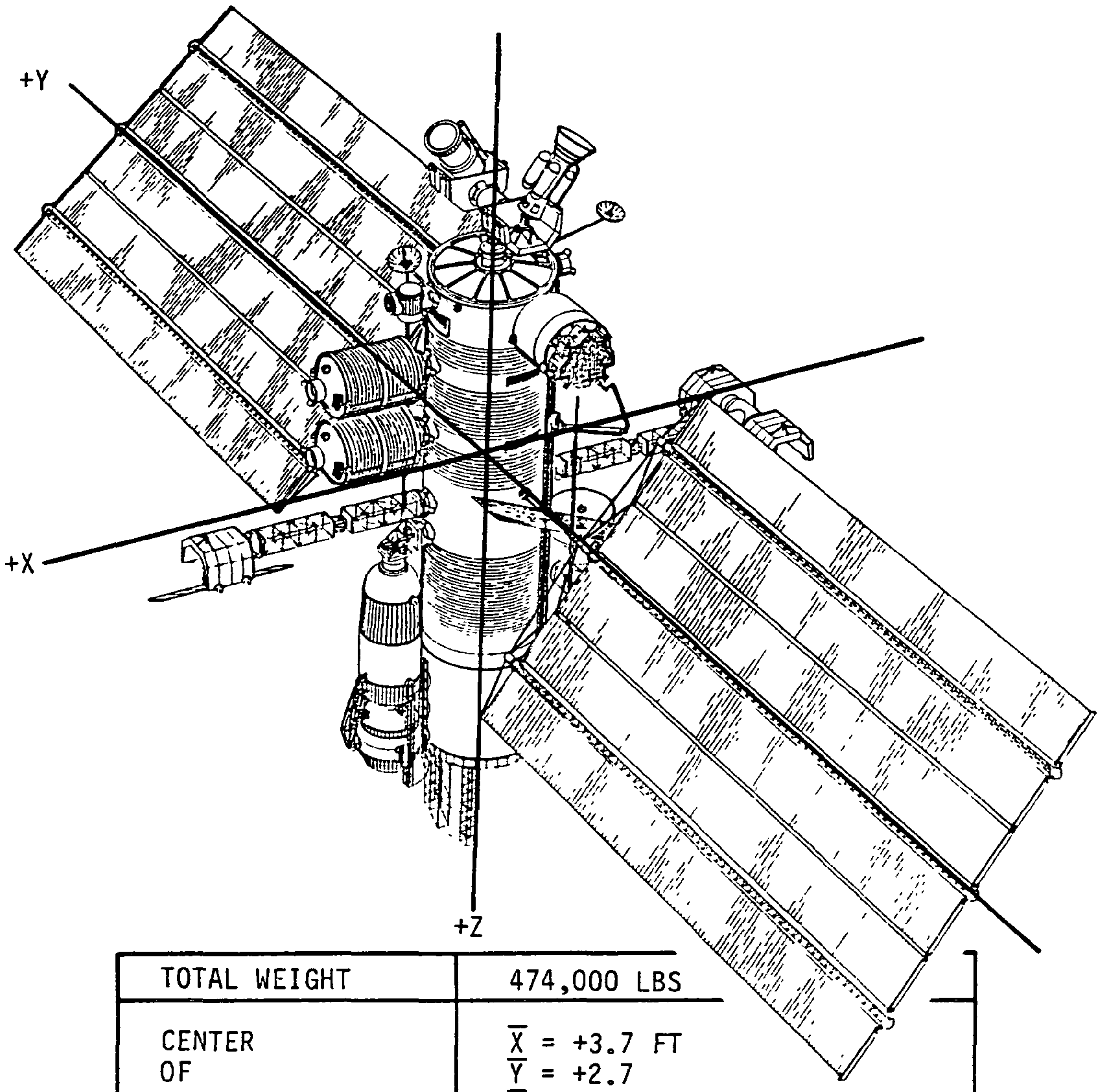
The logistics module. propellant storage, and payloads are similar to the other two options although the mounting orientation has been varied to meet the mass properties and configuration peculiarities. For example, by orienting the cryogenic propellant module perpendicular to the body axis, both payload fields of view and the inertia distribution would be adversely affected.

Figure 4.4-5 presents the mass properties summary for the SDV configuration. The inertia distribution has the proper relationship (i.e., $I_{yy} > I_{xx} > I_{zz}$), but the principal axes offsets are a bit higher than optimum. Since the sensitivity to specific element and payload distribution has been demonstrated, a minor adjustment of locations could be used to achieve a more balanced mature configuration. This configuration has the advantage that the main body presents a major mass and inertia contribution that is only mildly effected by the addition of payloads and auxiliary modules.

The buildup sequence is described on Figures 4.4-6 and 7. A significant feature of this approach is the ability to achieve an operating capability after one SDV and one STS launch, and to reach a mature capability (era 1995) after eight launches. This saves two and four launches when compared to the ACC and 14 foot modular approaches respectively.

A summary of the SDV configuration advantages and disadvantages can be found on Table 4.4-1 . The stiffness advantage may be a very significant space station consideration, but with dynamics and control analysis yet to be initiated, this advantage can only be surmized in qualitative terms. A stable attitude at IOC means that the desired gravity gradient orientation is achieved without requiring supplemental CMG's or RCS control. An examination of the previous space crane/manipulator concepts indicates the need for the crane to install/dock a space station element, which in turn mounts the continuing trackage needed for further build up and assembly. The ability to complete the trackage in one step (SDV concept) has an advantage since a more continuous, less restrictive buildup with greater station access would be possible.

A reduction in the phased growth capability (table 4.4-1) only means that some growth steps require a build up on an equipment piece part basis as opposed to a plug in module approach. The EPDS phased build up is an example of this concern. Batteries, power distribution equipment, and control avionics would require a modular approach capable of transportation, transfer and installation on a piece part basis.



TOTAL WEIGHT	474,000 LBS
CENTER OF MASS	$\overline{X}$ = +3.7 FT $\overline{Y}$ = +2.7 $\overline{Z}$ = -3.6
MOMENTS OF INERTIA	$I_{XX} = 2.2 \times 10^7$ SLUG FT^2 $I_{YY} = 2.3 \times 10^7$ $I_{ZZ} = 1.2 \times 10^7$
PRODUCTS OF INERTIA	$P_{XY} = 2.0 \times 10^5$ SLUG FT^2 $P_{XZ} = 1.4 \times 10^6$ $P_{YZ} = 1.7 \times 10^6$
PRINCIPAL AXIS OFFSET ANGLES	$\theta_X = 9^o$ $\theta_Y = 8^o$

Figure 4.4-5 Shuttle Derived Vehicle Concept Mass Properties

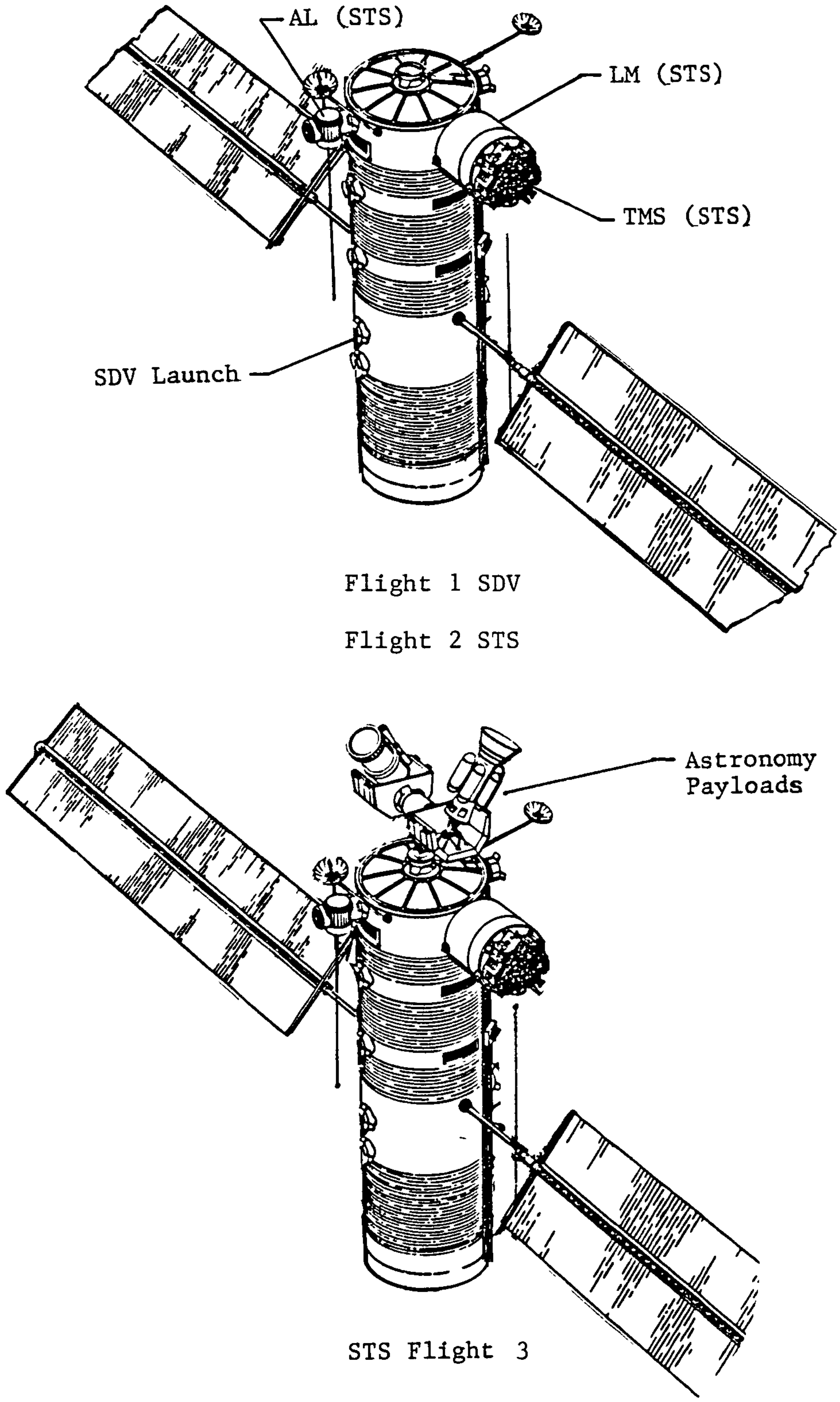


Figure 4.4-6 *Shuttle Derived Vehicle Concept - Build Up Sequence*

THIS PAGE INTENTIONALLY LEFT BLANK

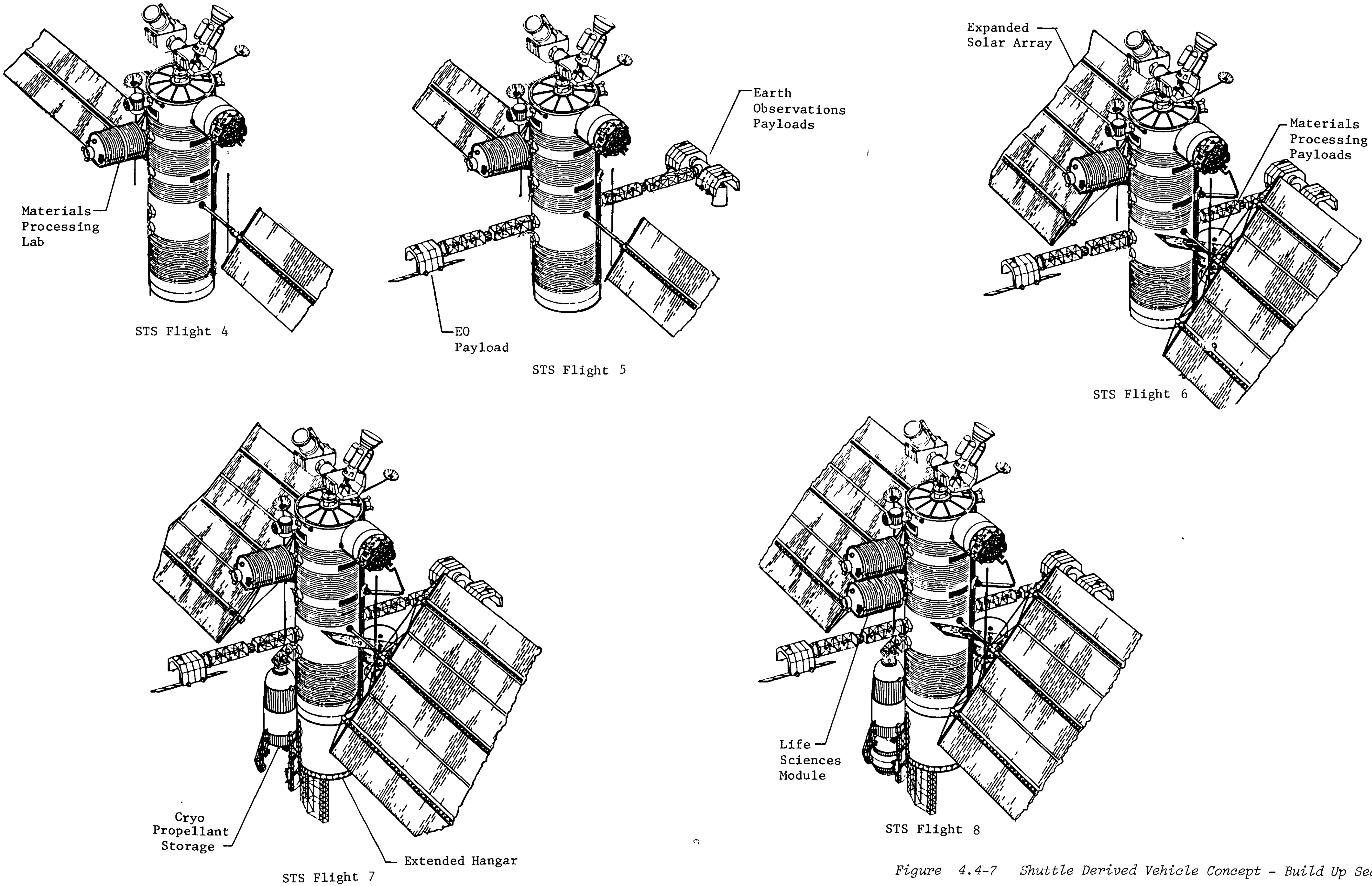


Figure 4.4-7 Shuttle Derived Vehicle Concept - Build Up Sequence

Page intentionally left blank

Page intentionally left blank

The growth capability limits are due to the configuration approach where limited Z axis growth can occur. This can be overcome, to a certain extent, by the use of additional extended, multiple docking ports similar to the payload mounting concept.

Table 4.4-1 SDV Advantages/Disadvantages

Advantages	Disadvantages
o Reduced STS transportation costs	o Committed to launch era technology
o Significant early capability	o Reduced phased growth capability
o Crew safety at initial launch	o Limited growth capability without payload/element docking adapter additions
o Larger available pressurized volume	
o Good main body stiffness	
o Stable attitude at IOC	
o A complete coverage space crane can be achieved earlier	

4.5 PLATFORMS

In section 4.2 there was a discussion on the general requirements of the various science and commercial payloads as regards station mounting compatibility. When the individual payload requirements are examined in detail, the specific wants and needs are so diverse as to require multiple space stations at various inclinations and different pointing attitudes. Some payloads require transfer/delivery, others need a contamination free environment, some require extensive resupply and service. All of these needs obviously could not be satisfied on a single space station. The benefits analysis did show, however, that cost reductions could be achieved by combining payloads and using a single facility or "bus" approach. Based on these results, individual payload requirements were reviewed, and those with common traits and the ability to function together were identified. These became candidate platform payloads.

Our platform architectural studies were limited to a top level requirements derivation, a discipline by discipline compatibility and payload selection analysis, and the definition of a preliminary platform configuration. This discussion will concentrate on the first and last issues since the mission model presents the payload basing mode options (i.e., station, platform, free flyer, etc.)

Table 4.5-1 lists the platform requirements as defined by our preliminary studies. Requirements needing further explanation are: the number of payloads, proximity operations, flight attitude, and use of station components in the platform design.

Both the mission model and cost analysis results were used to select the quantity of payloads for each platform. The mission model identified the potential candidates and availability (year) of payloads by discipline. A cost benefits study showed savings when three or more payloads were combined on a platform. The upper limit of 8 payloads was partially driven by the maximum number of payloads available, and partially by the ability to physically attach and retain access for servicing.

Table 4.5-1 Platform Requirements

- o STS cargo bay compatibility
- o Provide onboard drag makeup system
- o Compatible with TMS servicing/access
- o Design to accommodate between 3-8 payloads
- o Provide STS docking capability for manned access
- o Proximity operations to space station desireable
- o Inertial attitude preferred
- o Design life exceeding 5 years
- o Maximum utilization of space station subsystem equipment

By co-orbiting platforms with the space station, service, resupply and changeout operations can be more easily accomplished since less propellant consuming transfer operations are required (TMS or OTV), and line of sight communications can be maintained.

An inertial attitude has been selected for most science platforms due to their preference for celestial or solar pointing. Some solar terrestrial and earth observation payloads do however, require earth nadir and limb pointing.

Preliminary subsystems analysis on the platforms has been restricted to an identification of power requirements and to a lesser extent some physical accommodations evaluations. In summary, a number of the space station subsystems can and should be considered for use on the platforms. The propulsion/RCS equipment would be similar in design (less drag make up required and less tankage) and installation. Commonality would occur to some extent on the communication systems as well.

The data management system requires further evaluation before an assessment on commonality can be made. Some compatible thermal control system equipment can be expected, but detailed analysis of heat rejection requirements, platform orientation, and individual equipment requirements must be completed before a definitive position can be established. Section 6.1 presents a platform by platform preliminary power summary which indicates a range between 7 and 30 Kw. At these power levels, much of the modular EPDS equipment could be utilized on the platforms. As an example, the basic solar array panel design could be modified by reducing the length of deployed blankets. A new attitude/orientation approach would involve a different equipment list for the guidance and control subsystem.

One of the results from the mission requirements studies was to identify the type of platforms required. This was accomplished by first determining the payload accommodation potential of the preliminary space station configurations. Then the most space station compatible payloads were selected. Finally the free flyer candidates were identified, which left a series of platform payloads (with corresponding platform orbital locations and inclinations). The preliminary platform selection includes:

1) Earth observations
2) Astronomy
3) Materials processing
4) Initial solar-terrestrial observatory (ISTO)
5) Advanced solar terrestrial observatory (ASTO)

Both the astronomy and materials processing platforms can be co-located with the 28.5^{o} inclination space station. The astronomy payloads are compatible with 28.5^{o} and the materials processing payloads are not sensitive to inclination. The ISTO platform desires an inclination of 57^{o} and the ASTO and earth observation platforms require polar orbits.

Selection of a platform configuration involved a review of previous platform concepts and a cursory platform definition study. The MSFC space platform is a natural candidate for the space station platforms, because of the compatible power levels, pointing capabilities, and ability to mount the numbers of payloads under consideration.

Our platform studies concentrated on a materials processing platform where the requirements are power, heat rejection, servicing, and microgravity driven. A platform concept where the space station energy section is used for the spacecraft bus was also investigated. Figures 4.5-1 and 2 show these configurations which were prepared prior to final definition of the platform payloads.

The material processing platform attempted to accommodate a maximum number of payloads, and was developed around the premise of frequent TMS servicing intervals. Because the materials processing class of payloads have requirements that are unique when compared to the science payloads, a new platform design approach may be beneficial. Further studies are recommended to better define the requirements and develop alternate design approaches.

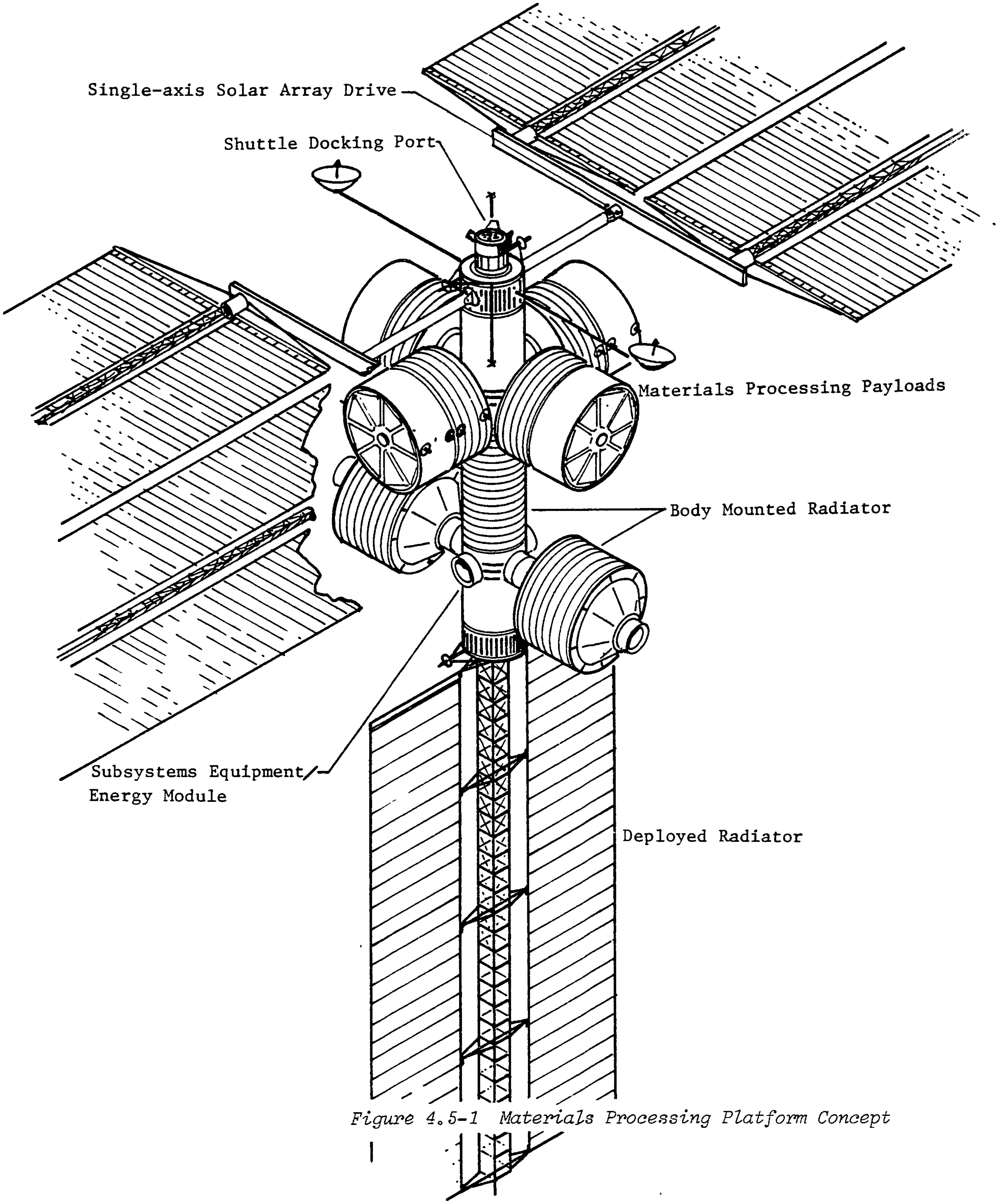


Figure 4.5-1 Materials Processing Platform Concept

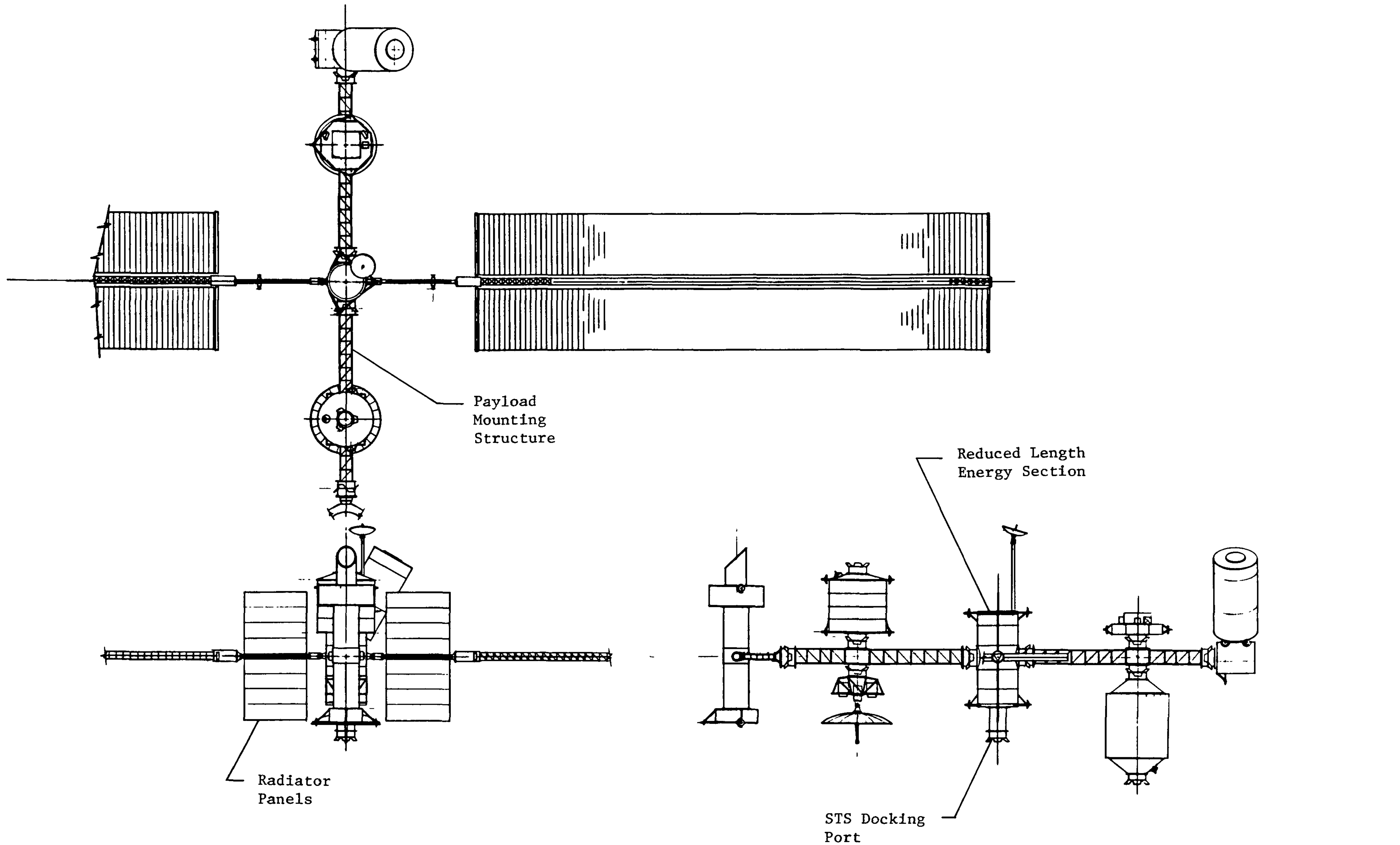


Figure 4.5-2 Energy Section Based Platform Concept

Page intentionally left blank

Page intentionally left blank

The energy section approach looked at the accommodation of diverse payloads only because the selection activity for payloads was not yet complete. A review of the subsystem equipment requirements showed that the concept involving the space station energy section far exceeded the volumetric requirements (twice the required volume). This arrangement also attempted to maintain a balanced solar array configuration which resulted in a less then desireable payload arrangement.

By comparison the space platform satisfied all the specific requirements identified at the beginning of this section, and has had extensive feasibility and subsystem performance analysis completed. This then became the logical platform choice for use in this study.

An adaptation of the space platform concept to a dedicated astronomy payload set is shown in Figure 4.5-3.

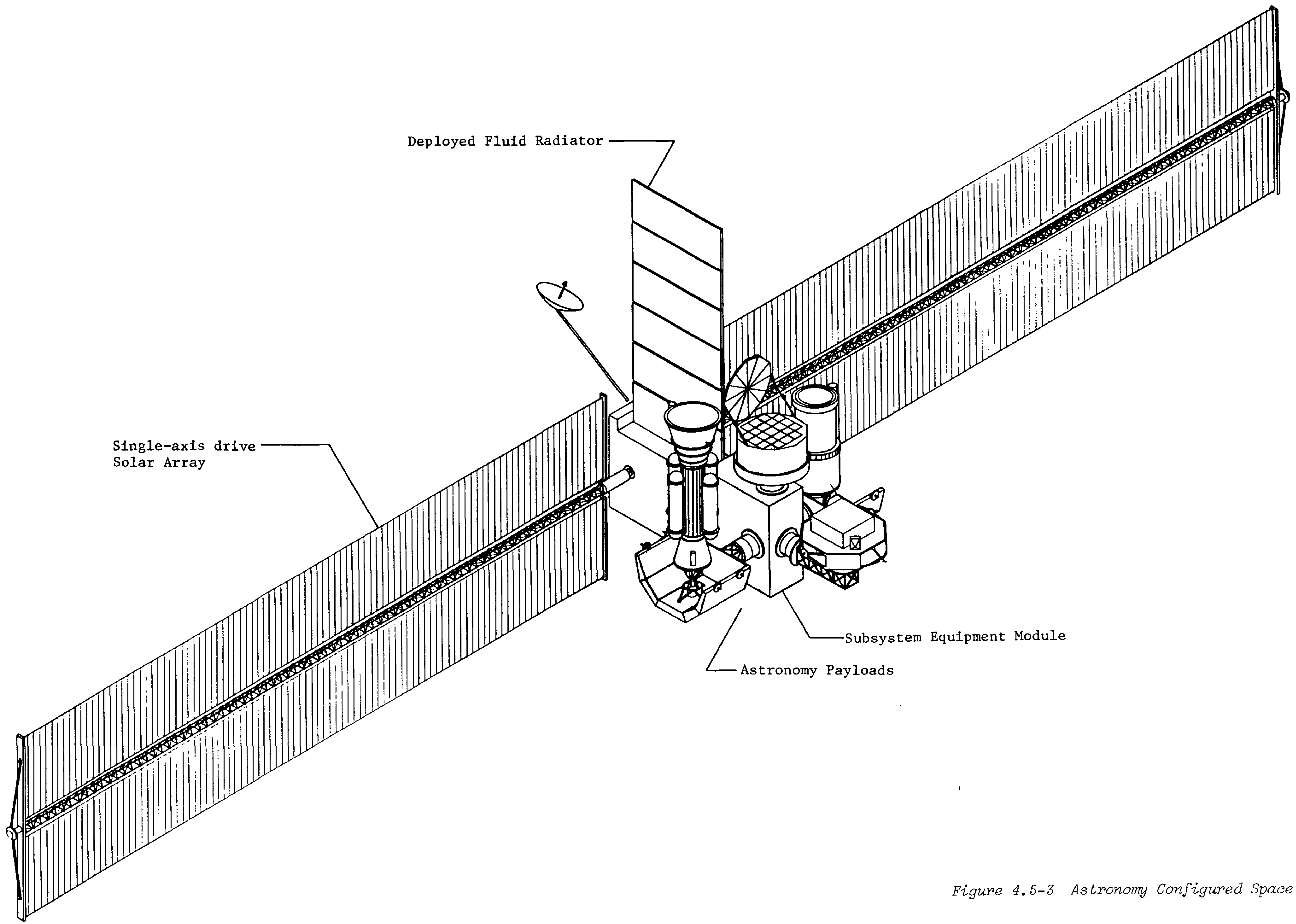


Figure 4.5-3 Astronomy Configured Space Platform

5.1 SPACE STATION ENVIRONMENTAL CONTROL-LIFE SUPPORT (ECLS)

The Space Station will be capable of being continuously manned for the 10 to 20 years of its expected useful life. One of the most significant aspects of habitability module design is the Environmental Control - Life Support (ECLS) design philosophy and systems approach.

Unlike the Shuttle orbiter, which obtains its orbital power from fuel cells, the full operational Space Station will obtain its power from a solar cell array. This change greatly affects the selection of ECLS equipment needed for the Space Station crew, since the flow of fresh water resulting from the fuel cell reaction and a large supply of cryogenic oxygen (that may be drawn upon for crew metabolic use) may not be available. Resupply of crew metabolic oxygen is a considerably less penalty than resupplying the crew's water needs, but it is still significant. When the cost of resupplying water and oxygen is considered, it becomes obvious that regenerative life support equipment will ultimately provide a significant payback for the Space Station.

At this point in the Space Station development, an evolutionary growth approach appears reasonable. The present ECLS system technology development plan will support a variety of requirements, keeping options open until a specific Space Station approach is defined by NASA.

This analysis presents a "strawman" ECLS system defining a probable initial implementation and the major steps necessary to close the oxygen and water loops. Many Space Station architectural factors affect the timing of these steps. These factors and their influence on the ECLS system design are as follows.

5.1.1 Reserved

5.1.2 Possible Space Station ECLS Evolution

The basic functions provided by the ECLS equipment to support man in a spacecraft are independent of the spacecraft architecture or mission. However, proper selection of equipment required to provide the ECLS functions is very much dependent on spacecraft architecture and mission requirements. Table 5.1.2-1 shows the subsystem breakdown of Environmental & Thermal Control and Life Support equipment employed on a space station.

ECLS equipment for manned spacecraft flown to date can generally be classified as "open loop". The relatively short mission duration and the use of a fuel cell power supply for past and current spacecraft justified the use of expendables for atmosphere revitalization and replenishment, the one-time use of water, and the use of water vaporization for cooling. Early operational and mission scenarios may permit an early station to use mostly current technology ECLS with minimal penalty, but for the eventual full capability station it will be desirable to use "closed loop" ECLS in order to avoid high operations costs of large resupply penalties. The weight advantage of utilizing regenerative CO_2 removal, water processing, and oxygen

Table 5.1.2-1 Space Station ECLS

Cabin Ventilation & Thermal Control	Atmosphere Revitalization	Heat Transport & Rejection	Atmosphere Supply
△ - Ventilation, Filtration △ - Air Cooling Heat Exchangers △ - Cold Plate (Equipment Cooling)	* - Humidity Control Heat Exchangers * - CO_2 Removal • - CO_2 Reduction - Odor Control * - Contaminant Control * - Atmospheric Monitoring	△ - Freon Coolant Pump Package △ - Water Coolant Pump Package △ - Freon to Water Heat Exchanger with Thermal Control Valve	* - O_2 Storage & Supply System * - N_2 Storage & Supply System - Hydrazine Decomposition * - Emergency O_2 Storage & Supply System * - Emergency N_2 Storage & Supply System • - Electrolysis * - O_2/N_2 Control Valves * - Cabin Dump & Relief Valves
Water Management	**Health & Hygiene**	**EVA/IVA Support**	**System Control**
• - Wastewater Storage/Pretreat • - Evaporation/Purification Units • - Water Quality Monitoring • - Potable Water Storage/Post-Treat * - Emergency Water Storage • - Condensate Water Processing	* - Waste Collection & Storage * - Hot Water Supply - Cold Water Supply * - Whole Body Shower * - Hand Wash - Clothes Washer/Drier - Trash Compactor * - Food Refrigeration * - Food Freezer * - Oven - Dishwasher * - Emergency Waste Collection	- Suits - PLSS's - Umbilical Life - POS - Recharge Station - Emergency Escape System - Airlock Pump	- ECLS Central Control & Display - Portable Maintenance Control/Display * - Local Microprocessor Control

* = Basic ECLS Hardware
• = Closed Loop Hardware
△ = Thermal Control and Heat Removal Hardware

recovery is shown in Figures 5.1.2-1, 5.1.2-2, and 5.1.2-3, respectively. Significant volume advantages are also realized by utilizing regenerative technology. It is important that the Space Station design concept consider the eventual closed loop ECLS with an objective to avoid discarding early ECLS equipment which can be used as building blocks or as backup components.

The probable first ECLS system step will be to use Shuttle and Spacelab technology with the possible incorporation of a regenerative CO_2 removal system, a condensate water clean-up system to provide hygiene water, and radiators to provide total heat rejection. The next major evolutionary step would be to incorporate a waste water processing system to provide additional hygiene water for shower and clothes washing. Oxygen and water loop closure would be accomplished in a third major step by addition of a CO_2 reduction system, an O_2 generation system, and additional waste water processing equipment to include water recovery from urine. These ECLS growth options are shown in Figure 5.1.2-4 in more detail. These options are shown relative to two power system scenarios. The impact of the power system concept selection on the ECLS system is discussed further in the next section.

The programmatic value of the evolutionary growth approach for Space Station ECLS can be evaluated with the assistance of Figure 5.1.2-5. Assuming a Space Station program start of about 1986, in-orbit operations starting about 1991 and a 10-year growth period, there are two initial funding paths which can be taken for the ECLS system. Early program lower costs can be achieved be designing "open loop" ECLS equipment retaining during the design, the option to evolve the ECLS system into "closed loop" capability. Retaining the evolution option is not a significant program cost. The higher early program cost approach would be to implement a "closed loop" design at program start. As can be seen from Figure 5.1.2-5, this latter option will incur the lowest total program cost. The cost analysis assumes that Space Station mission requirements will have a need for the crew sizes versus time indicated on the figure. The analysis further assumes that the ECLS equipment basic module size is for a crew of 4. The cost of sizing larger equipment is insignificant and a reasonable maximum crew size for the largest single volume launched by Shuttle is about 4.

After initiating in-orbit Space Station operations, assuming the early program lower cost path was followed, a decision can be made to evolve the ECLS system as discussed earlier or to stay with relatively "open loop" operation. However, when mission activity requires larger crews, the figure shows that resupplying an "open loop" ECLS system will be a significant cost penalty. Throughout the time period shown, a resupply cost of $1,000 per pound was used. This launch cost may be high early if Shuttle payloads are not full, further supporting use of an "open loop" ECLS system. However, when Space Station operations are mature even small volumes in the payload will probably be used for mission equipment thereby reaching the Shuttle's full payload weight limit.

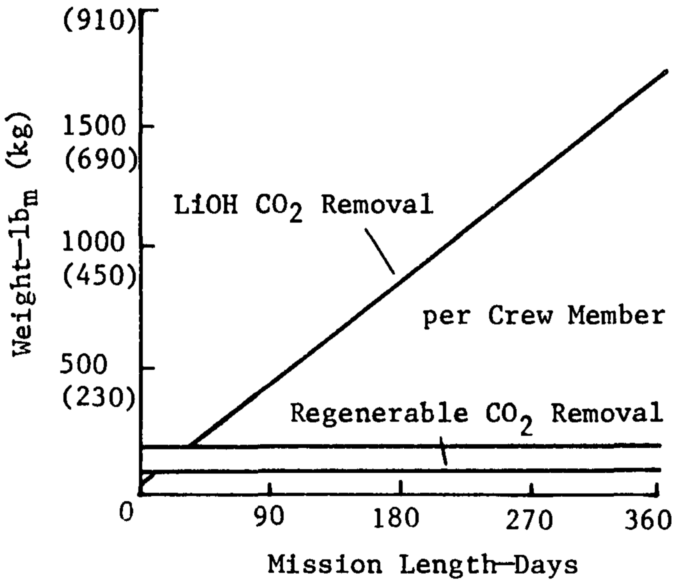


Figure 5.1.2-1 Regenerable CO_2 Removal Benefit

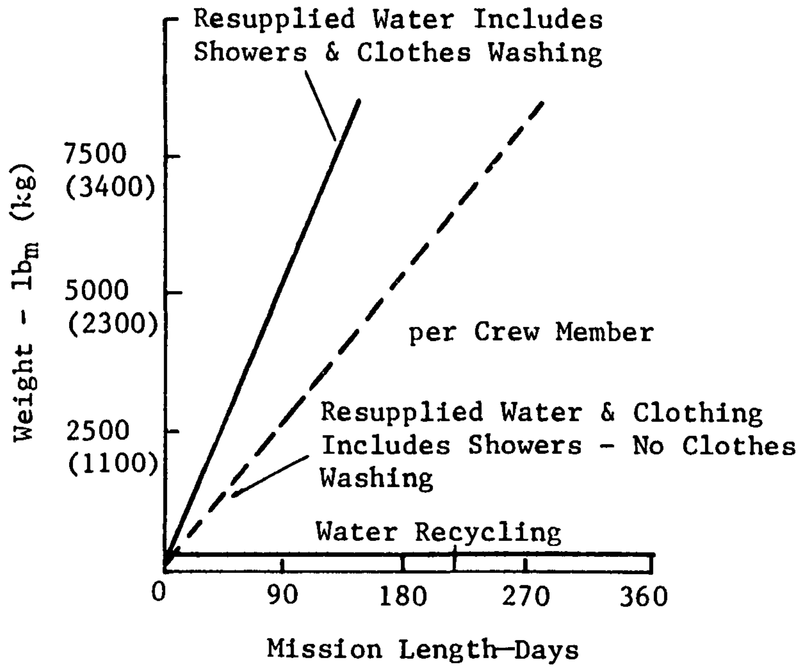


Figure 5.1.2-2 Water Recycle Benefit

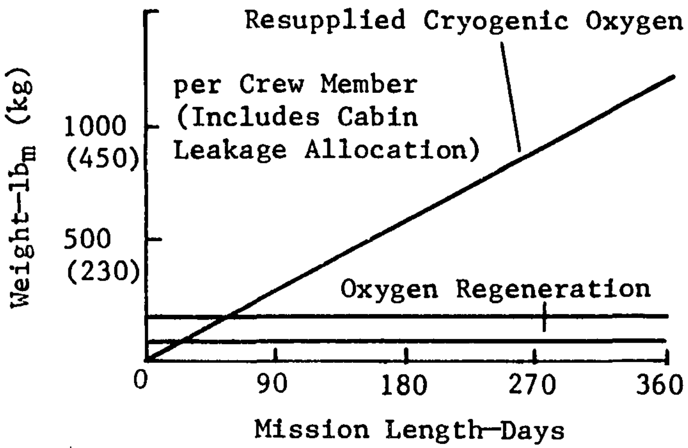


Figure 5.1.2-3 Oxygen Regeneration Benefit

ECLS Functions	Early Station Crew of 2 or 3		Increased Capability Station Crew of 4 to 6		Mature Full Capability Station Crew of 8 to 12	
	Fuel Cells or Fuel Cells + Solar Array	Batteries + Solar Array	Fuel Cells + Solar Array	Batteries + Solar Array	Fuel Cells + Electrolysis Cells + Solar Array	Batteries + Solar Array
Cabin Ventilation & Thermal Control	Fans, Heat Exchangers Cold Plates	Same	More of Same if Station Modules Added	Same	More of Same for Additional Modules	Same
Heat Transport & Rejection	Pumped Water Loops in Pressurized Volume, Pumped Freon Loops External & Radiators	Same	More of Same if Station Modules Added	Same	More of Same for Additional Modules	Same
Atmospheric Supply	Resupplied Crye O_2 (Same Source as Fuel Cells), Resupplied Crye N_2	Resupplied Crye O_2 & N_2	Same as Early Station Fuel Cell System	Same as Early Station Battery System	O_2 From Water Electrolysis, Resupplied Crye N_2 (Electrolysis May be in Power Sys)	O_2 From Water Electrolysis, Resupplied Crye N_2 (Electrolysis must be an ETCLS Subsys)
Air Reventilation	Humidity Control by Condensing H/X, Regenerative CO_2 Removal (Overboard Dump CO_2), Trace Gas Catalytic, Oxidizers, Odor Control Canisters with LiOH Capability for Emergency CO_2 Backup	Same	More of Same if Modules Added	Same	More of Same for Additional Modules without CO_2 Dump and CO_2 Reduction Added, LiOH Net Required for Emergency	Same
Water Processing &	Potable & Hygiene Water from Fuel Cells	Resupplied Potable Water, Hygiene Water from Humidity Condensate (Quantity Sufficient for Hand Wash & Sponge Both no Showers)	Same as Early Station Fuel Cell System	Resupply Potable Water, Add Wash Water Processing (Capable of Processing Urine for Mature Station) For Additional Hygiene Water, Add Rudimentary Water Quality Monitoring	Potable and Hygiene Water from Wastewater Processing Eqpt, Add Potable Water Quality Monitoring, Emergency Potable Water Tank Storage	Same
Health & Hygiene	Vacuum Dried Fecal Waste Collectors, Hot/Cold Water Supply, Hand Wash, Even, Resupplied Clothing (Water Available for Showers if Desired)	Same (Except Water not Available for Showers)	Same as Early Station Plus Showers, Clothes Washer/Drier, Trash Compactors, Refrigerator/Freezer	Same	More of Same as Increased Capability Station for Additional Modules Plus Possibly Dish Washer	Same

Figure 5.1.2-4 Space Station ECLS Growth Options

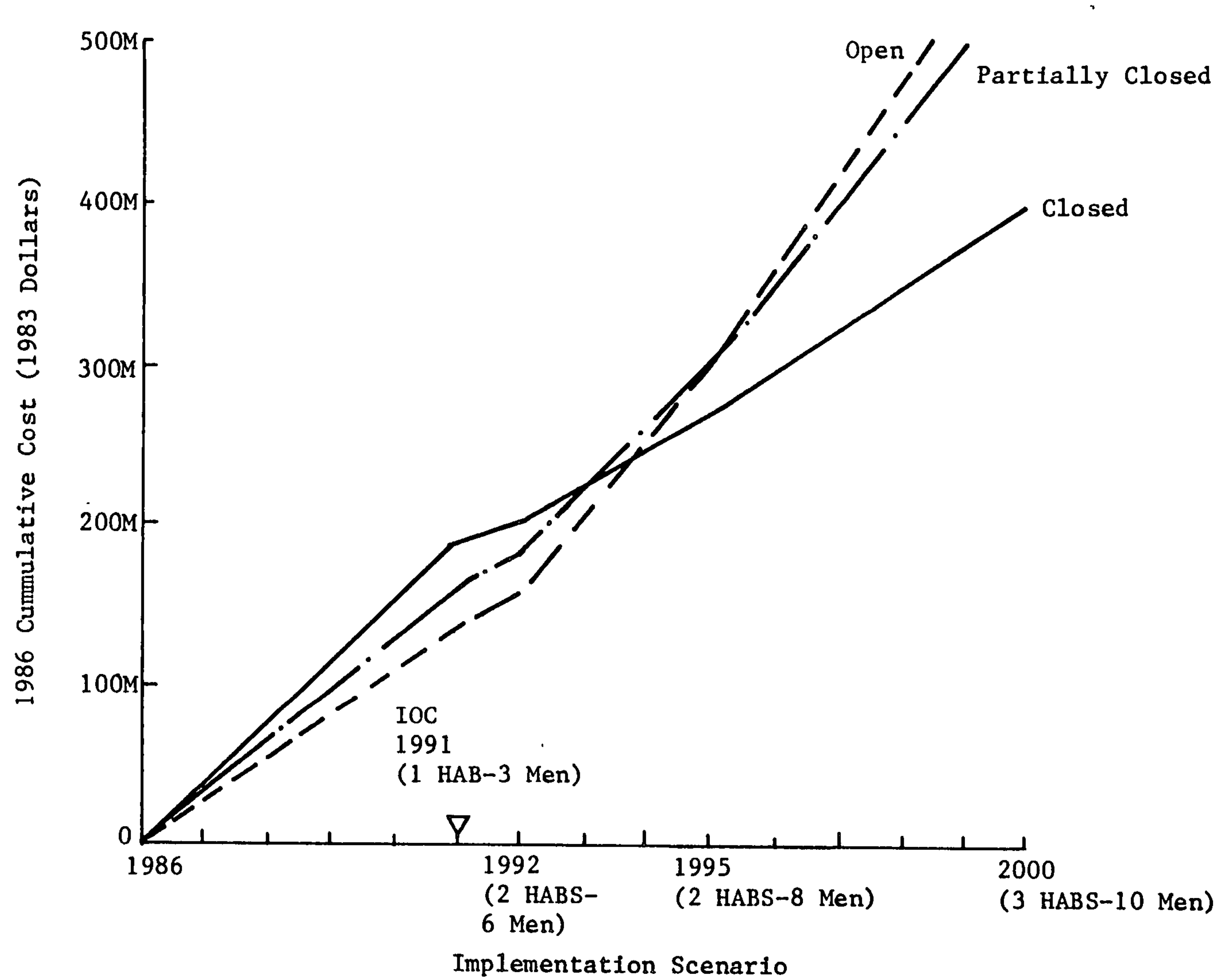


Figure 5.1.2-5 Space Station ECLS Program Options Costs

5.1.3 Options For ETCLS System Loop Closure Equipment

Table 5.1.3-1 contains a list of concept options for those major ETCLS functions involved in closing the water and oxygen loops or for which venting to space is a consideration. The concepts ultimately selected to provide the listed ETCSL functions are dependent upon and can influence other Space Station systems selected. Table 5.1.3-1 attempts to provide trade-off information so that interactions between systems can be considered and optimum Space Station ETCLS, power, Cryogen Storage, ACS and Logistics systems can be selected. Each ETCLS functional area shown in the table is discussed briefly below:

5.1.3.1 Water Supply - The NASA is developing equipment which will produce potable quality water from wastewater (including urine). However, medical and safety approval to drink reprocessed water will require extensive equipment certification probably including on-orbit long term test certification. As a minimum, drinking water may have to be resupplied to the station during the early years of Space Station operation in order to allow adequate on-orbit certification time. If OTV operations, ET scavenging or other operations scenarios result in large quantities of cryogens to be stored in orbit, drinking water (and probably hygiene water) can be provided from some "dark side" fuel cell operation at little or no penalty. Even if advanced insulations and cryo sub-cooling techniques are used, it would seem that eventually a steady state boil-off condition would exist. This boil-off could be used to provide "dark side" fuel cell power, reduced solar array area and potable quality water for the crew.

5.1.3.1 CO_2 Removal - The concept selected for CO_2 removal from the crew compartments is dependent upon restrictions on venting to space and upon the need for oxygen recovery from the CO_2. If oxygen is available from cryo storage boil-off for crew metabolic and vehicle leakage needs, CO_2 reduction may not be required.

Since water is the intermediate step in CO_2 reduction, electrolysis of the water is required to recover the oxygen for crew metabolic reuse. The electrolysis system can be in a regenerative fuel cell system or a separate ETCLS unit.

It should be kept in mind when reviewing Table 5.1.3-1 that CO_2 or CH_4 could be used to provide some thrust by passing it through resistor jets or could be entrained in the ACS plumes to accelerate the potential contaminants away from the Space Station, possibly reducing the venting concern.

5.1.3.3 Oxygen Supply - Electrolysis of water is the largest single power consuming process in a closed loop ETCLS system. As stated previously, if cryogenic O_2 is available with little penalty, a trade-off may show no advantage for the additional complexity of water electrolysis for O_2 generation.

Table 5.1.3-1a Options for ECLS System Loop Closure Equipment

ETCLS Function	Potential Space Station Concepts	Advantages	Disadvantages
Water Supply	- Resupply All Water (17.6 lbm/ Man-Day H_2O	- Low Equipment Cost, Low Risk, Potability of Water not a Concern - No Processing Power Required - Low Crew Time, Low Ground Op's	- Resupply: 25.1 lb/Man-Day (With Water Tankage) 2.5 lb/ Man-Day Disposable Clothing - Return: All of Above or De-orbit by Concept "C" Under Waste Storage Disposal
	- Resupply All Water by O_2/H_2 & Process through Fuel Cells (17.6 lbm/ Man-Day O_2/H_2)	- Same Advantage for Concept A - 1.95 kW/Man Dark Side Pwr Provided by Fuel Cells - 1.40 kW/Man Less Solar Array Pwr Required with Regenerable Fuel Cell Energy Storage Concept - Can Utilize Cryo Boil-Off (Less Cryo Storage Technology Development Required) - Is Compatible With Clothes Washing which will Increase Above Nos. by 250% - Is Also Compatible with Using Humidity Condensate for Hygiene use which will Reduce Above Nos. by 23%	- Resupply: 22.9 lb/Man-Day Fuel Cell Cryogens with Tankage (May not be Disadvantage if Cryo Boil-Off Available) - 2.5 lb/Man/Day Disposal Clothing - Return: 25.1 lb/Man-Day Wastewater - 2.5 lbm/Man-Day Clothing - Or Deorbit by Concept "C" Under Waste Storage/ Disposal

Table 5.1.3-1b (cont)

ETCLS Function	Potential Space Station Concepts	Advantages	Disadvantages
	- Process All Water to Potable Quality (1.95 lbm/ Man-Day Chemicals)	- No Water Resupply Required	- Resupply: 2.8 lb/Man-Day Chemicals with Tankage Spares - Return: 3.2 lb/Man-Day Chemicals with Tankage & Waste Solids - 134 Watts/Man Required to Process Wastewater - High Equipment Cost & Higher Risk - $17M Approximate DDT&E Cost Delta to Concepts A&B. $2.5M Recurring Cost to Provide Processing Capability
CO_2 Removal	- Chemical (LiOH Absorption	- Simple & Low Risk - No Venting to Space	- Resupply: 5.8 lb/Man-Day Including Canister & Rack Wt - Return: 6.6 lb/Man-Day Including Canister, Rack & CO_2 Weight - High Cost of Canisters. Approximately $1000 per Man-Day Based on Shuttle Size Canister (+ Cost to Resupply 5.8 lb/Man-Day)
	- Regenerable with CO_2	- Low Risk, Low Cost - No Resupply if O_2 Available from Cryogen Boil-Off	- 2.2 lb CO_2 per Man-Day Vented to Space - 200 Watts Light Side Power per Man
	- Regenerable with CO_2 Liquid Storage	- Low Risk, Relatively Low Cost - Low Vol Method of Returning to Space	- Resupply: 0.7 lb/Man-Day Tankage Weight - Return: 2.9 lb/Man-Day Tankage & CO_2 Weight - 40 Watts/Man Delta Pwr to Concept B for Compressor to Store CO_2 as Liquid
	- Regenerable with CO_2 Reduction	- No O_2 Resupply	- 0.9 lb/Man-Day CH_4 Vented to Space - $5.5M DDT&E Cost Delta to Concept B. $1.0M Recurring Cost Delta to Concept B - Need Electrolysis of Produced H_2O to Eliminate O_2 Resupply - 0.17 lb/Man-Day H_2 Needed from Cryogen Boil-Off to Recover all the O_2

Table 5.1.3-1c (concl)

ETCLS Function	Potential Space Station Concepts	Advantages	Disadvantages
Oxygen Supply	- Resupply O_2	- Low Eqmt Cost, Low Risk - No Processing Power - Can Use Cryogen Boil-Off	- None Assuming Cryogen Boil-Off Avail. 1.84 lb/Man-Day O_2 Rqd Plus Vehicle Leakage
	- Electrolize H_2O	- Closed Loop (with CO_2 Reduction - Water Balances Usually Show Sufficient Water for Vehicle Leakage O_2 Makeup	- 340 Watts/Man Rqd + Pwr for Leakage O_2 Makeup - \$9.0M Approx DDT&E Cost for Electrolysis if Separate from Regenerable Fuel Cell System. \$1.5M Recurring Cost for Each Electrolyzer Unit
Waste Storage/ Disposal	- Vacuum Freeze Dry Shuttle Type Concept	- Developed Eqmt	- Resupply: 0.43 lb/Man-Day Container Replacement (210 Man-Days/Container) - Return: 0.83 lb/Man-Day Canister & Contents - 0.53 lb/Man-Day Gas Vented to Space
	- Frozen Storage	- No Venting to Space	- \$5M Approx DDT&E Cost for Developing New System - 20 Watts per Man Light Side Pwr to Freeze & Maintain Waste in Frozen State
	- Deorbit Daily Waste (Use only Materials Which are Changed to Gases on Reentry)	- No Venting to Orbit Path of Space Station - No On-Orbit Storage Vol for Wastes Rqd - Biocides not Needed - Compatible w/ Culinary, Fecal, used Clothing & Liquid Wastes - No Return Penalty on Logistics Module	- Resupply Bags to Contain Waste Until Reentry (Approx 0.5 psia Bag Pressure During Deorbit) - Requires an Undesirable Space Vacuum Interface as well as a Vacuum Pump & Contaminant Filter

5.1.3.4 Water Storage/Disposal - The present Shuttle toilet is vented to space vacuum in order to freeze dry the waste material and make it bio-stable. Venting can be avoided by integrating a refrigeration system (probably a thermoelectric concept) with the toilet bowl in order to keep the waste frozen and bio-stable. Another concept which would significantly reduce the station trash management problem without venting to the orbit path would be to seal all wastes in bags and deorbit the bags periodically. If the waste material is deorbited on a daily basis no biocides would be required. Before deorbiting, the trash container would be evacuated by pumping the air through a filter to the cabin to avoid gas loss to the orbit path. An internal bag pressure of about 0.5 psia would keep any waste liquid disposed of with the trash from boiling. Only trash materials which would completely gasify on reentry would be disposed of in this manner. Cullinary, toilet waste, disposable clothing, paper and plastic waste, and waste liquids including urine, chemicals and hygiene water could all be disposed of by this technique and thereby avoiding a logistics module return penalty (and a resupply penalty for waste storage containers).

5.1.4 Space Station Architecture Influence on ECLS System

In defining an ECLS system three significant issues arise. First, an ECLS system growth scenario should be defined which has an acceptable fiscal year cost versus total program cost. Second, the ECLS system design should not dictate the Space Station architecture or have a negative impact on mission operations. Third, the approaches selected for other (non-ECLS) systems can have a major influence on the ECLS subsystems technology selections and system growth steps as described below.

5.1.4.1 Power System - Space fuel cells, fuel cells/solar cells, fuel cells/electrolysis cells/solar cells, or batteries/solar cells are all possible power system concepts for the initial phase of a growth Space Station. Assuming the availability of cryogens (O_2 and H_2), fuel cells supporting an early station would provide ample potable quality water for crew consumption and hygiene needs. Later, solar cells and loop closure with electrolysis cells would greatly change this scenario, providing significant payoff for water recycling. The use of batteries and solar cells would make water recycling more desirable for an early station. As a result, the power system concept selection and growth scenario has a significant impact on the degree and timing of optimum ECLS system water loop closure. An open water loop could require that all waste water (hygiene and urine) be returned to earth to avoid contaminating the Space Station environment.

5.1.4.2 Reaction Control System - The use of either hydrogen/oxygen or hydrazine for space station orientation and orbit keeping has an influence on the valid options available for ECLS system design. If hydrogen/oxygen is used, the ECLS system electrolysis subsystem (or the power system electrolysis subsystem) could provide hydrogen/oxygen from resupplied water. Due to the fuel rich ratio used by the reaction control system (about 6 lbs O_2 to 1 lb H_2), sufficient O_2 may be available from water electrolysis (8 lbs O_2 to 1 lb H_2) to satisfy crew metabolic and cabin leakage makeup needs. If hydrazine propulsion is used, a decomposition subsystem can be employed to obtain N_2 for cabin leakage makeup needs thereby taking advantage of common resupply between the reaction control and ECLS systems. Furthermore, the H_2 from the decomposition process can be used in the ECLS CO_2 reduction subsystem to supplement electrolysis H_2 in order to react more CO_2, thereby minimizing O_2 loss.

5.1.4.3 Cryogenics Sources - Cryogenics stores in-orbit will eventually be required when orbital transfer vehicles (OTVs) become a part of Space Station operations. These cryogens may either be transported to orbit as a Shuttle payload or by scavenging the Shuttle external tanks (ET) to bring cryogens not consumed during the launch into orbit. This technique, after appropriate modifications are made to an orbiter, can provide significant quantities of cryogens in-orbit at almost no penalty to the payload capacity. Early employment of ET scavenging, before cryogens are needed for OTVs, could make cryogens available for an early station's power system if fuel cells are utilized. The cryogenic O_2 can also be used for crew metabolic and cabin leakage makeup needs. Even when OTV operations become routine, the boil-off from large cryo storage tanks would probably be sufficient for ECLS needs. Thus, cryogenics availability has a significant impact on the ECLS system design and growth scenario.

5.1.4.4 Resupply/Shuttle Visit Rate - The frequency of Shuttle visits will have a pronounced effect on the steps taken to close and the degree of closure of the ECLS oxygen and water loops. Payloads for mission operations may require Shuttle visits to the Space Station at a much more frequent rate than to 30 to 90 day resupply rates. Recent studies have shown that the Shuttle payload is generally volume limited rather than weight limited. If this is the case, high density resupplies such as potable water or cryogenics might be included in void areas of the payload bay. The use of these volumes, which do not occupy payload volume, may provide essentially "free" expendables resupply, therefore, affecting loop closure trade studies of the ECLS system.

5.1.5 Station Growth Steps

During the Station's buildup, the pressurized volume(s) and crew size will determine the ECLS subsystem's module size. An early Station may be Shuttle tended and/or have a crew size small enough that an open loop ECLS is adequate. Major objectives of the Space Station ECLS system design are to provide a no-throw-away growth approach and to select optimum subsystem concepts and module sizes. With proper planning it should be possible to grow from an open loop to a full capability closed loop ECLS system without discarding of equipment as the system grows. It is important that the system design allows interfaces and packaging volumes for on-orbit growth of the ECLS system.

Proper subsystem module size selection can save significant program cost. For example if mission analysis shows that only a single vehicle module and a two-man crew is required early but the station will rapidly grow to two modules and an eight-man crew, it would be appropriate to size ECLS equipment for a four-man crew in each module. The equipment would be oversized for the early station but the cost of redesign and certification of the ECLS hardware would be avoided.

5.1.6 Space Station ECLS Requirements for 8 Crewmen (4 per module)

The research and development of regenerative life support equipment has progressed to the point where concepts have been demonstrated and the Space Station program can plan to use this equipment.

Space Station mission success and the health and safety of the Space Station crew are dependent on the continuing reliable performance of the Space Station ECLS equipment. A fail operational/fail safe philosophy has been established as the minimum acceptable design criteria for the ECLS equipment. Table 5.1.6-1 summarizes the ECLS system Life Support performance requirements proposed for a 8-man, 90-day mission Space Station. The "operational" column represents normal operation without failures for the crew population distribution of four in two habitability modules. The "90-Day Degraded" column is acceptable in complying with the "fail operational" performance criteria resulting from a single worst failure of non-maintainable equipment. The "14-21 Day Emergency" column is acceptable in complying with a second consecutive worst failure of non-maintainable equipment.

The full significance of these performance requirements is manifested in their influence on reliability and redundancy considerations, which in turn dictate the number of primary systems that Space Station must carry. Non-maintainable equipment such as main distribution tubing, major wiring distribution bundles and equipment support structure can be assumed to have a reliability of nearly "one". All equipment with a reliability less than "one' which can be practically maintained in space will be designed for replacement and an appropriate complement of spares will be provided.

In order to minimize the spares complement on-board the Space Station, commonality should be a design requirement. With proper systems engineering common valves, fans, pumps, instruments, controllers etc can be designed so that one spare can be used in many different places. Average ECLS design loads are shown in Table 5.1.6-2.

5.1.7 Basic ECLS & Closed Loop Hardware

For purposes of this study, the equipment has been grouped into three categories: (1) Basic Environmental Control and Life Support (ECLS) Hardware: (2) Closed Loop Hardware, i.e., hardware required to employ regenerable water usage and oxygen and hydrogen production; and (3) Thermal Control and Heat Removal Hardware.

Table 5.1.6-1 ECLS Performance Requirements

Parameter	Units	Oper-ational	90 Day *Degraded	14-21 Day Emergency
CO_2 Partial Pressure	fmmHg	3.8 Max	7.6 Max	12 Max
Temperature	^oF	65-75	60-85	60-90
**Dew Point Temperature	^oF	40-60	35-70	30-75
Ventilation	ft/min	15-40	10-100	5-200
Wash Water	lb/Man Day	40 min	20 min	0
***O_2 Partial Pressure	PSIA	2.6 or 3.1	2.4-3.8	2.3-3.9
Total Pressure	PSIA	10.0 or 14.7	10.0-14.7	10.0-14.7
Trace Contaminants	-	****24 hr Ind Std	****8 hr Ind Std	****8 hr Ind Std
Maximum Crew Number	per SOC	8	6	12
Maximum Crew Number	per HAB/MOD	4	8	8

*Degraded Level is Acceptable to Meet a "Fail Operational" Reliability Criteria

**In No Case Shall Relative Humidities Exceed the Range of 25-75%

***In No Case Shall the O_2 Partial Pressure Exceed 26.9% or be Below 2.3 PSIA

****hr Ind Std = Hour Industrial Standard

Table 5.1.6-2 Space Station ECLS Design Average Loads

1 - Metabolic O_2	1.84 lb/Man Day
2 - Leakage Air	5.00 lb/Day Total SOC
3 - EVA O_2	1.22 lb/8 hr EVA
4 - EVA CO_2	1.46 lb/8 hr EVA
5 - Metabolic CO_2	2.20 lb/Man Day
6 - Drink H_2O	4.09 lb/Man Day
7 - Food Preparation H_2O	1.58 lb/Man Day
8 - Metabolic H_2O Production	0.76 lb/Man Day
9 - Cloths Wash H_2O	27.50 lb/Man Day
10 - Hand Wash H_2O	4.00 lb/Man Day
11 - Shower H_2O	6.00 lb/Man Day
12 - EVA H_2O	9.68 lb/8 hr EVA
13 - Perspiration & Respiration H_2O	4.02 lb/Man Day
14 - Urinal Flush H_2O	1.09 lb/Man Day
15 - Urine H_2O	3.31 lb/Man Day
16 - Food Solids	1.60 lb/Man Day
17 - Food H_2O	1.00 lb/Man Day
18 - Food Packaging	1.00 lb/Man Day
19 - Urine Solids	0.13 lb/Man Day
20 - Fecal Solids	0.07 lb/Man Day
21 - Sweat Solids	0.04 lb/Man Day
22 - EVA Wastewater	2.00 lb/8 hr EVA
23 - Charcoal Required	0.13 lb/Man Day
24 - Metabolic Sensible Heat	7000 Btu/Man Day
25 - Hygiene Latent H_2O	0.96 lb/Man Day
26 - Food Preparation Latent H_2O	0.06 lb/Man Day
27 - Experiments Latent H_2O	1.00 lb/Day
28 - Laundry Latent H_2O	0.13 lb/Man Day
29 - Wash H_2O Solids	0.44%
30 - Shower/Hand Wash H_2O Solids	0.12%
31 - Vehicle Heat Leak & Non-ECLS Thermal Loads	TBD
32 - Air Lock Gas Loss	1.33 lbs/Use
33 - Trash	1.80 lb/Man Day
34 - Trash Volume	0.10 ft^3/Man Day

Figures 5.1.7-1, 5.1.7-2, and 5.1.7-3 shows the weight, volume, power and cost penalty associated respectively with these groupings. Groups (1) and (2) are shown parametrically versus number of crew members, while group (3) is shown parametrically versus the amount of heat removal.

It should be noted that the reason a maximum of crew of six members and heat rejection of 20 Kw is that these values represent about the maximum that can be accommodated by a single module that could be fitted into the Shuttle cargo bay. For crews in excess of six and internal heat loads in excess of 20 Kw, an additional module would be required.

Figure 5.1.7-1 shows the weight, volume and power penalties for the basic (open loop) ECLS and closed loop ECLS systems. The closed loop system employs all the equipment used in the open loop, except for cryogenic oxygen storage, and adds the equipment necessary for loop closure.

The closed loop equipment considered for the study includes:

a) Evaporative water processing (washwater or washwater and urine reclamation)
b) Water filtration for post-treatment
c) Water electrolysis for oxygen generation
d) CO_2 reduction.

This study does not include the weight, volume, power and cost penalties for the additional sets of ECLS equipment necessary for operational redundancy to meet failure mode requirements. For the present, a factor of 1.5 can be used for weight, volume and production costs to meet redundancy requirements, while power requirements will remain essentially unchanged.

A regenerable CO_2 removal system has been included in the basic ECLS hardware since previous trade studies have shown that the payback for using a regenerable system rather than an expendable system (e.g., LiOH) is almost immediate.

Figure 5.1.7-2 shows the Initial Operational Capability (IOC) costs for the open loop and closed loop systems. The three elements of the IOC cost are:

a) DDT&E for all equipment
b) the production cost for a shipset of equipment
c) the launch cost of putting the mass of equipment into orbit ($1200/$lb_m$).

The DDT&E costs are reflected in the intercept (0 crewmembers) while the production and launch costs are reflected in the slope of the curve.

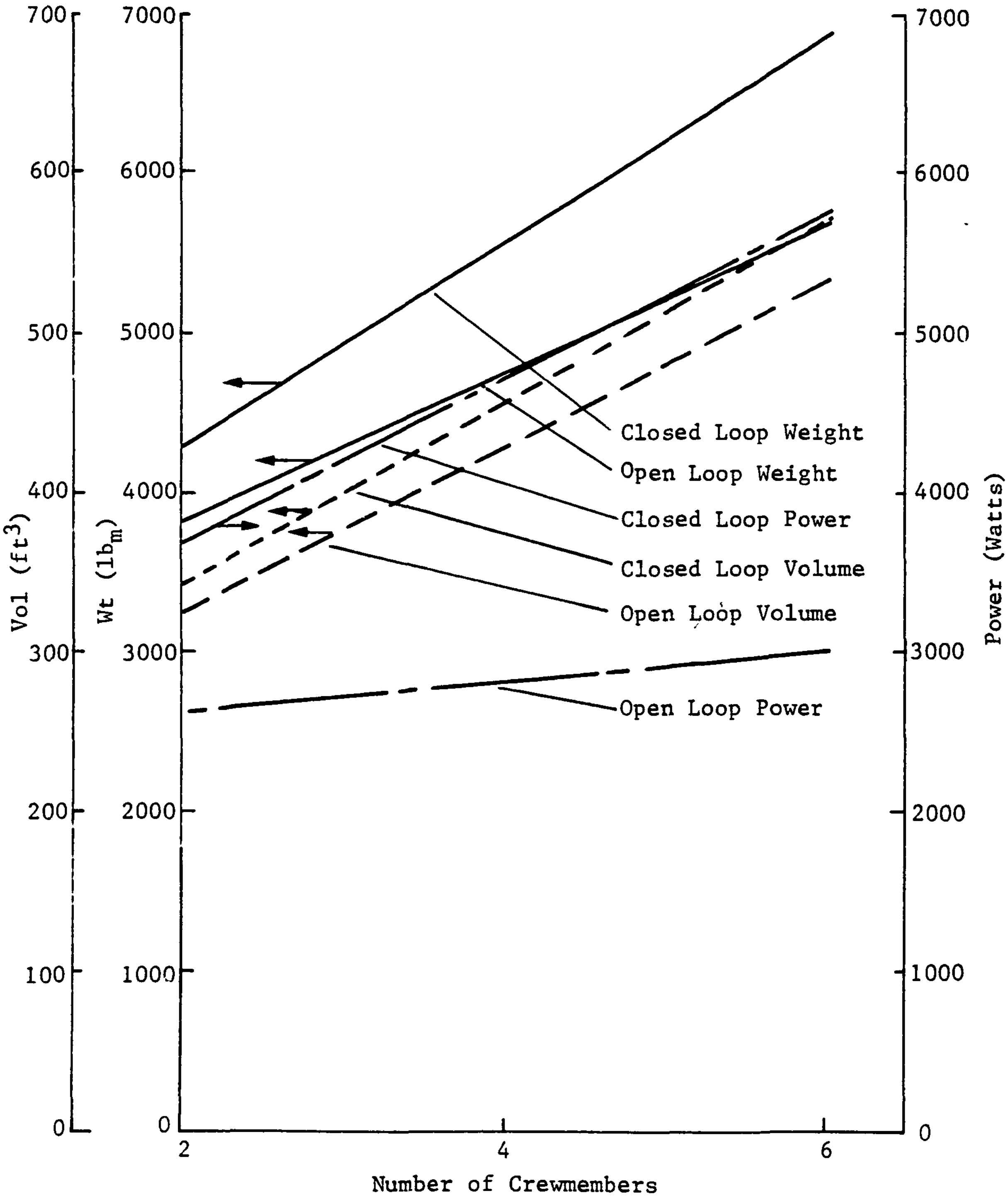


Figure 5.1.7-1
Basic ECLS Hardware Per Habitat Module W/O Thermal Control and Heat Rejection Equipment

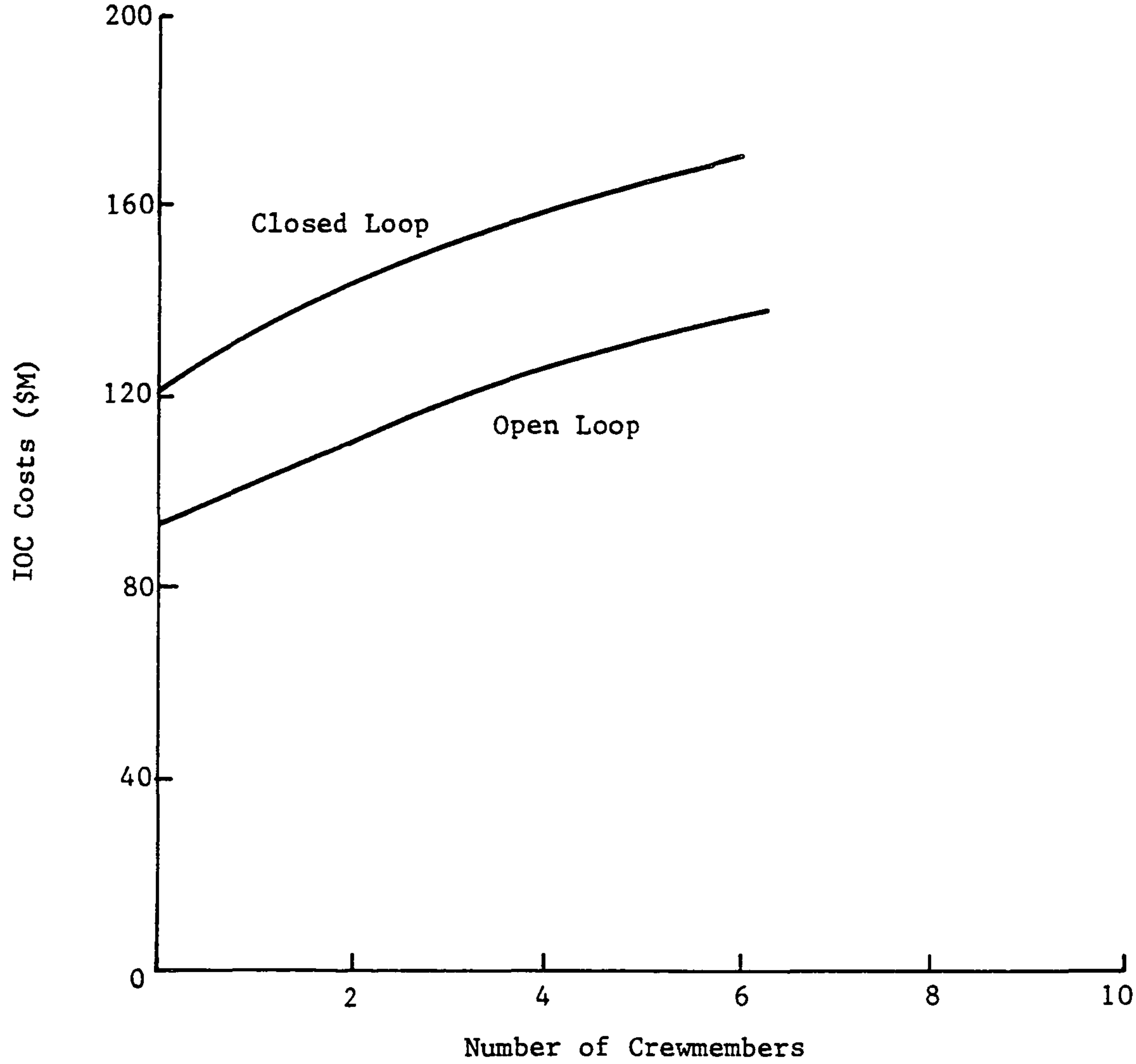


Figure 5.1.7-2
IOC Costs (DDT&E, Shipsets and Launch Costs) for ECLS Hardware without Thermal Control and Heat Rejection Equipment

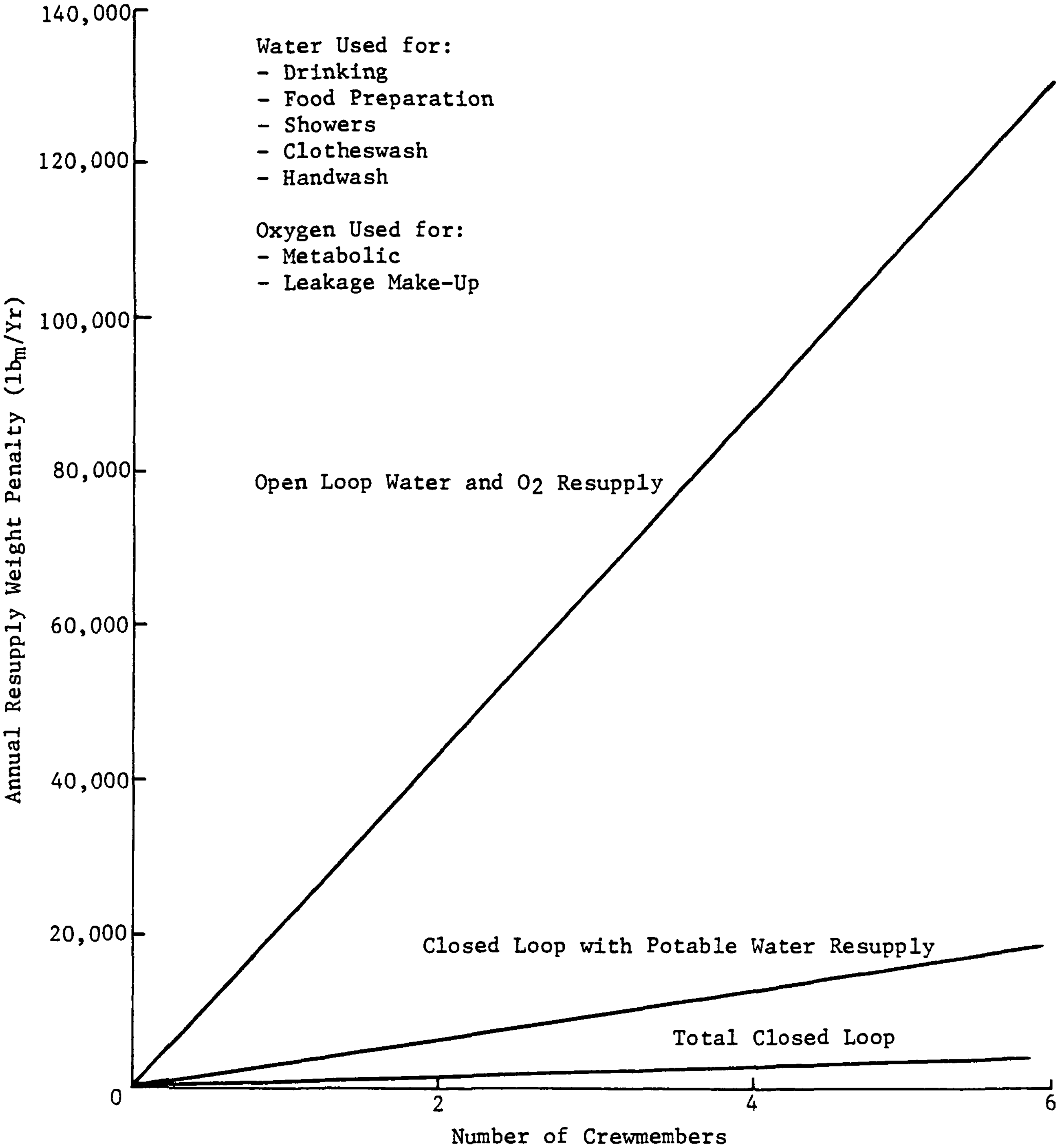


Figure 5.1.7-3
Open Loop vs Closed Loop Operation Annual Water and Oxygen Resupply Penalties

5.1.7.1 Open Loop Versus Closed Loop Resupply - Figure 5.1.7-3 shows the annual resupply saving possible by incorporating the loop closure equipment referred to in the previous text. The curve labeled "Closed Loop With Potable Water Resupply" assumes that the closed loop equipment is being used but that potable water is being resupplied from the ground. The "Total Closed Loop" curve assumes that the water processing system can be used to process urine and, therefore, produce potable water.

One can observe that, at a resupply penalty of $1200/lb$_m$, the annual savings in resupply costs in going to closed loop equipment quickly offsets the IOC costs of that equipment.

5.1.8 Cabin Ventilation-Thermal Control and Heat Transport and Rejection Subsystem

Figure 5.1.8-1 shows the weight, volume, power and IOC costs of the Cabin Ventilation-Thermal Control and Heat Transport and Rejection Subsystem equipment as a function of heat removed from the interior of a pressurized module. For all levels of heat rejection, the split between airborne to direct liquid loop heat rejection is assumed to be approximately 60/40.

The zero intercept for the IOC costs represents DDT&E costs for equipment. The slope of the line is determined by production costs of a shipset of hardware and launch weight penalty costs.

As noted previously, the maximum value used in the parametric study was 20 Kw because values larger than this would normally be associated with crews of greater than six. Therefore, for a larger station, with additional modules, the weight, volume and power parametric values are multiplied by the number of modules to calculate these parameters for the total station. The cost penalties would be much lower for additional modules because the DDT&E cost would already have been paid.

Using the information in Figures 5.1.7-1 and 5.1.8-1, one can size the vehicle heat rejection necessary to accommodate additional heat loads such as experiments. This is best demonstrated by an example:

> Suppose, after performing mission modeling and requirements studies it is determined that each habitat module should be designed for a maximum crew of four during normal operation. Suppose also that it is decided to size the heat rejection system to remove 18 Kw of internally generated heat.
>
> From Figure 5.1.7-1, assuming four crewmen, the open loop and closed loop ECLS heat rejection requirements are 2825 watts and 4725 watts, respectively. (The closed loop value is actually lower than this because a portion of the electrolysis power goes into splitting the water molecules, hence does not go into sensible heat. However, for this example, the full electrolysis power can be assumed).

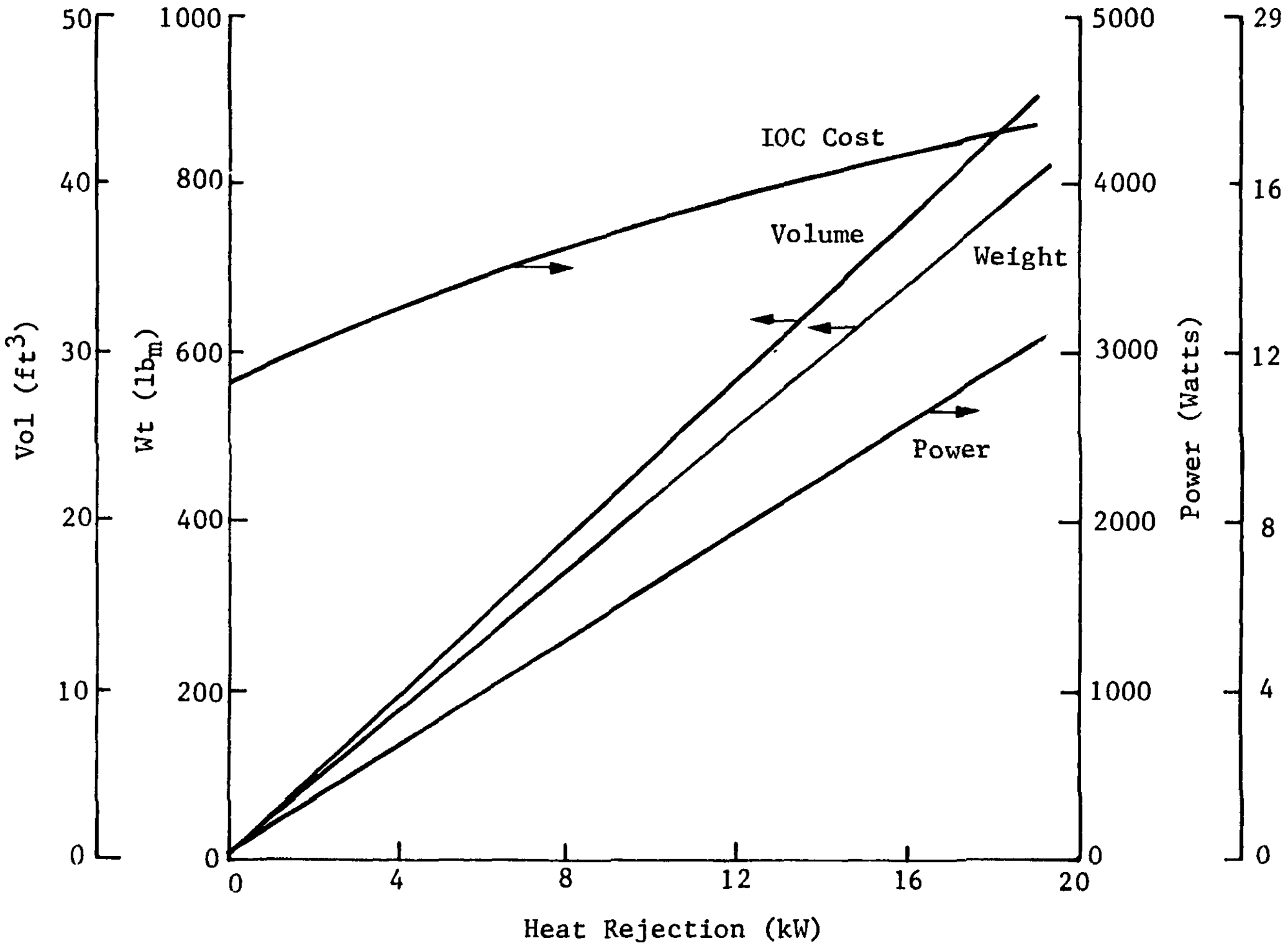


Figure 5.1.8-1
Cabin Ventilation - Thermal Control Subsystem Plus Heat Transportation and Rejection Subsystem (Excluding Vehicle External Heat Loads)

Using Figure 5.1.7-1, at 18 Kw heat rejection, the power required to run the heat rejection-removal equipment is 2920 watts.

Therefore, the amount of heat rejection capability remaining for other vehicle requirements such as lighting, communications, etc., plus experiments is:

18,000 - (2825 + 2920) = 12,255 watts for the open loop system

and

18,000 - (4725 + 2920) = 10,355 watts for the closed loop system.

5.1.9 ECLS Equipment Weights

Curves defining ECLS equipment weights, and weights of expendables and spares for various stages of ECTLS closure are shown in Figures 5.1.9-1, 5.1.9.2 and 5.1.9-3. These curves define ECTLS equipment weights and annual resupply weights for open loop, closed loop with potable H_2O resupply, and completely closed loop ETCLS equipment as a function of crew size.

Figure 5.1.9-1 displays weight of ETCLS equipment for various stages of loop closure. The closed loop system employs all the equipment used in the open loop, except for cryogenic oxygen storage, and adds the equipment necessary for loop closure.

The partial closed loop system employs all the equipment required for closed loop operation; however, drinking water is resupplied. All 3 system weights reflect the necessary weight for cabin ventilation-thermal control and heat transport and rejection subsystems in addition to the basic ECLS hardware. The system weights have also been upgraded to reflect additional sets of ECLS equipment necessary for operational redundancy to meet a failure mode requirement of fail operational/fail safe.

Figures 5.1.9-2 and 5.1.9-3 display annual spares and expendables weights for various stages of ECLS closure. The weight values on these curves must be added together to determine total resupply requirements.

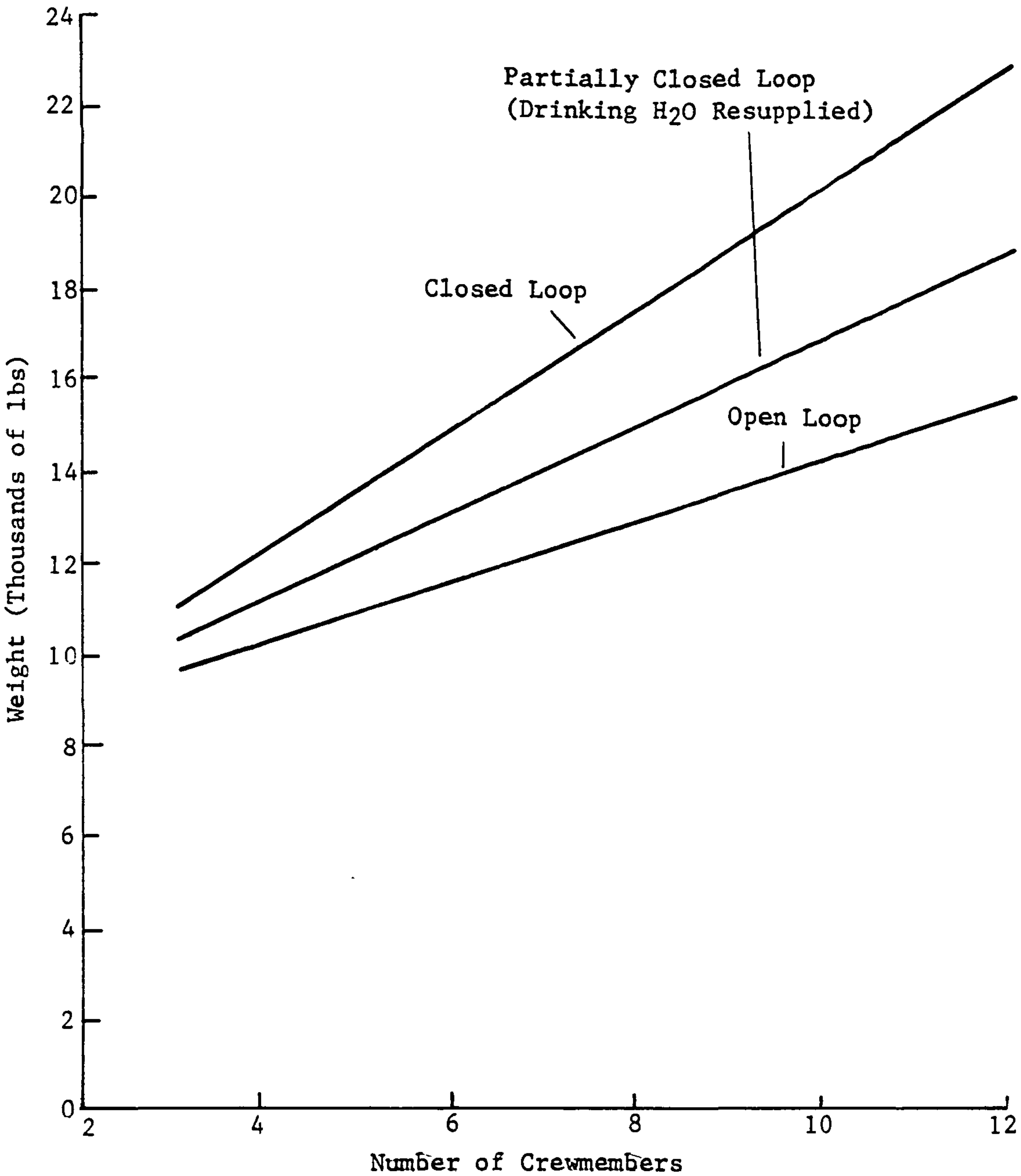


Figure 5.1.9-1 Open Loop vs Closed Loop ECLS Equipment Weight

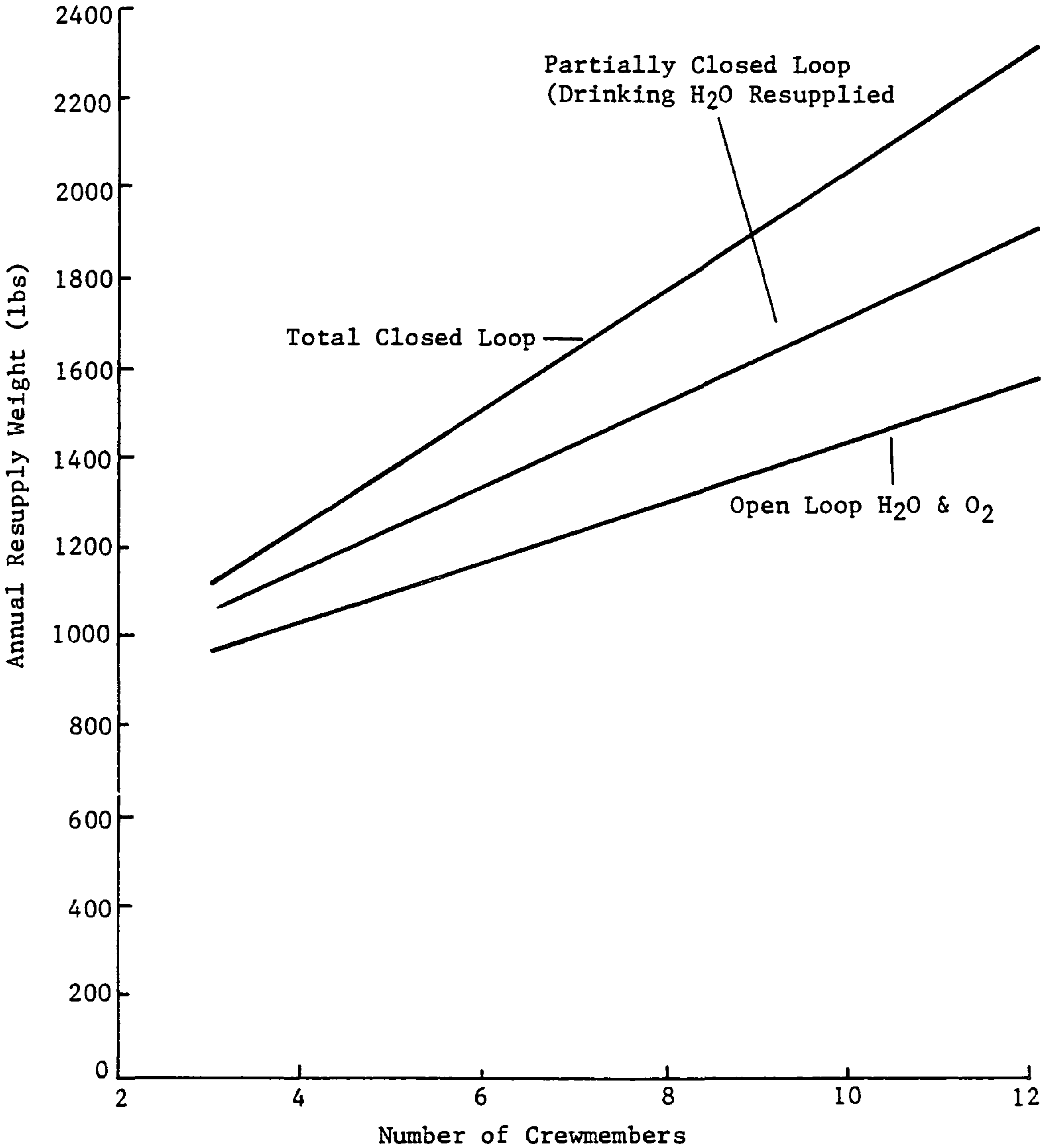


Figure 5.1.9-2 Open Loop vs Closed Loop Annual Resupply Weight of Spares

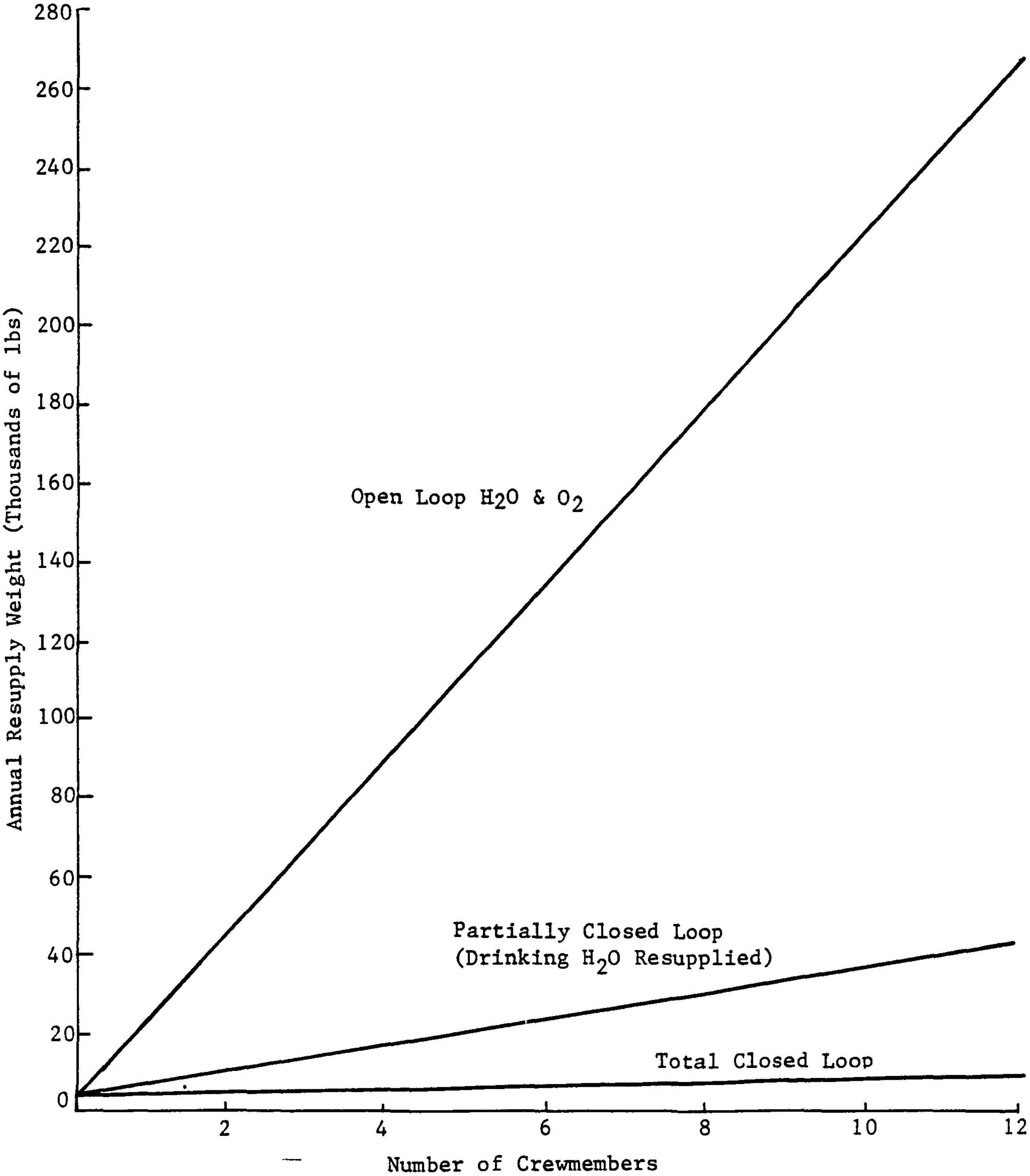


Figure 5.1.9-3
Open Loop vs Closed Loop Annual Resupply Weight of Expendables

5.2 EVA OPERATIONS

5.2.1 Advanced EVA Pressure Suit

The current Shuttle Extravehicular Mobility Unit, (EMU) is a significant advancement over the apollo/Skylab A7LB suit. However, the Shuttle suit still has deficiencies when continuous on-orbit EVA servicing is considered. The components of the present EMU are shown in figure 5.2.1-1. The areas of EMU improvement needed are:

5.2.2 Higher Pressure Suit

The current EMU operates at 4.3 psi a crewman must prebreath pure O_2 for 3.5 hours prior to the pressure reduction from the 14.7 psi Space Station pressure. This pre-breath purges the N_2 from the crewmans system & prevents him from getting the binds during this large pressure drop. The rule of thumb, while breathing air, is that a rapid pressure reduction greater than 50% of the operating pressure can induce the bends.

The space station operating pressure will be between 12-14.7 psi. The current 4.3 psi suit would again require the 3.5 hr. pre-breath. This requirement makes a significant impact on crew timelining and prevents a quick reaction EVA for servicing and/or emergency operations.

A 6 to 8 psi suit will allow the pre-breath requirement to be deleated. The crewman can go EVA directly from a 14.7 PSI cabin pressure.

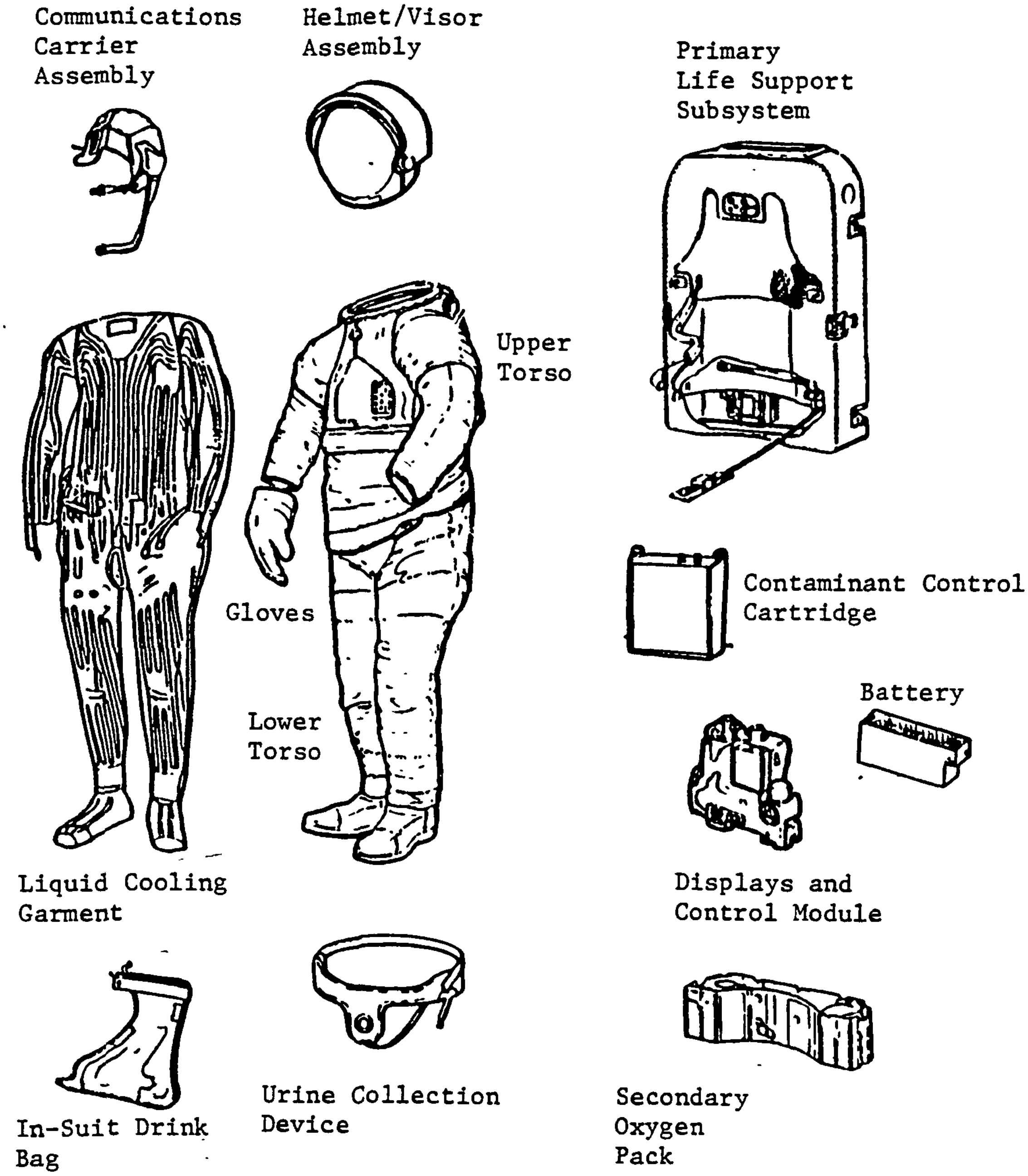
Hamilton Standard is presently producing an 8 psi "feasability" suit to be tested in the JSC-WIF during the second quarter of 1983. In a the past, the problem with higher pressure fabric suits is that limb rigidity increases with pressure and mobility is lost. However, new joint designs are expected to equal or better the current suit joint mobility.

5.2.3 Non-Contaminating PLSS

The current Portable Life Support System (PLSS) thermal control unit, (water sublimator) discharges water at a rate of 1.72 lbs/hr. (10.3 lbs/6 hr. EVA). This water can condense on surfaces and is a source of worksite contamination. Hamilton Standard is currently building a prototype, non-venting heat sink under a NASA-JSC contract.

5.2.4 Extended Life EMU Components

The use of Extravehicular activity will extend throughout the 10-20 yr. operational lifetime of the space station. The present EMU has a total life of 30 EVA's and must be refurbished, on the ground, after 5 EVA's. This relatively short EMU life is adequate for Shuttle use and possibly during the early phases of the Space Station. However, longer life components must be incorporated into the EMU. The fabric arm and leg joints are the EMU components that deteriorate quickly and will be upgraded first.



5.2.1-1 Shuttle EMU Components

The following discussion defines baseline requirements related to EVA and the EMU, followed by a description of two types of EVA suits. The baseline Space Station EMU can be basically a Shuttle EMU which is upgraded for extended life (EL EMU). This extended life EMU consists of two-major subsystems, the Space Suit Assembly (SSA) and the Advanced Primary Life Support System (PLSS). Each will be addressed separately in later sections.

5.2.5 Space Station EMU Requirements

The Space Station EMU, be it a current model EMU or an upgraded no-vent EMU or the advanced EL EMU, will be used according to the following set of requirements:

1. Each EMU is used for a maximum of one EVA per day.
2. Each EVA-dedicated crewmember is provided with his own EMU.
3. Two men per EVA as a minimum.
4. Minimum of 12 hours for EMU recharge.
5. Each EMU is replaced on orbit every 90 days.
6. On-orbit checkout of EMU is accomplished via the Caution and Warning System.
7. Recharge of EMU accomplished through the services and cooling umbilical connections. (Battery, Oxygen, Thermal Control).
8. All EVA-related work equipment (i.e., MMU, tools) will be stowed on the external shell of Space Station.
9 Planned EVA sorties for up to eight hours (max).
10. The EMU shall be capable of passing through internal hatches of 40-inch diameter in both a manned and unmanned mode.
11. The Liquid Cooling and Ventilation Garment (LCVG) will be replaced every 90 days. The LCVG may be washed in the space station washing machine and laundered every six to ten EVA's. A chiffon body stocking will be worn under the LCVG to pick up the majority of waste products (water, hair, skin, etc.) an will be laundered after each use (each body stocking can support up to ten EVA's, weight 5 oz. each, and has a volume of 10-20 in^3 each).

Initial Space Station operations, which may not warrant a heavy EVA schedule an therefore could use the existing 4.3 psi EMU. However, as EVA frequency increases, the launch weight penalty for expendables becomes prohibitive, and the advanced regenerative extended life EMU becomes attractive.

5.2.6 Extended Life EMU

The Extended Life EMU will eliminate the requirement for prebreathing prior to EVA (i.e., the EMU pressure in conjunction with Space Station cabin pressure) and in doing so will allow immediate egress from the airlock. This condition will also increase crew EVA preparation efficiency. The major differential between the current 4.3 psi EMU and the EL EMU lies within the construction of the Space Suit Assembly.

5.2.7 Extended Life Space Suit Assembly (EL-SSA)

The current EMU SSA provides approximately nude range mobility and supports a suit operating pressure of 4.3 psi. The use of an EL SSA will not decrease EVA crewmember mobility. New joint technology will replace current EMU tucked fabric joints within rolling colvolute joints, toroidal convolute joints, and four-bearing joints. These new joints will allow current EMU mobility capability for a range of suit operating pressures reaching 8 psi. The EL SSA could be available by 1986. The EL Space Suit Assembly will also incorporate features facilitating on-orbit EMU maintenance.

The current EMU SSA is checked out extensively prior to each Shuttle flight. Ground testing hardware consists of:

- Liquid Cooling and Ventilation Garment test rig and Space Suit Assembly leakage test rig

- Space Suit Assembly cleaning station.

To have this same capability on orbit, plus the capability to repair any malfunctions identified during tests would require a large volume. On-orbit checkout using current design philosophy would be limited to the following;

- Leakage check of the Space Suit Assembly.

- Cleaning of soft goods

  - o Liquid Cooling and Ventilation Garment - use on-board washing machine, clean after every 6-10 EVA's.

  - o Urine Collection Device - throw out after use.

  - o Fecal - diaper, throw out after use.

  - o Cleaning, drying Protective Garment Assembly - odor, bacteria control, use a stericide wipe.

The use of the EMU SSA on-oribt and related maintenance is directly related to SSA functional life. The life of the current EMU soft goods is 6 years. EVA operational life is 180 hours when the current EMU is pressurized at 4.3 ± 0.1 psig. All repair of the current EMU occurs on ground. On-orbit spares consist primarily of gloves.

The Space Station EL SSA should have an operational life of 6,000 EVA Hours and accommodate an on-orbit maintenance philosophy (replacement of joints) which requires only a minimum of spares, this being due to the commonality of parts and the use of sizing rings to quickly adjust arm and leg lengths.

The proposed Space Station EL SSA will be constructed of a single wall laminate bladder to facilitate easy cleaning using a microbial wipe. This will allow on-orbit cleaning of the SSA without requiring elaborate cleaning and drying stations.

The proposed, on-orbit maintainable Space Station EL SSA should have an inventory of the following spares:

	Weight Each (lb.)	Volume Each (in3)
Gloves	2.70	3601
LCVG	6.50	1445
UCD	0.56	100 - 120
Body Stocking	0.45	10 - 20
Arms	8.51	1656
Lower Torso Assembly	33.80	5508

The Space Station EL SSA would be capable of supporting an EVA schedule approaching 65 EVA's per 90-day resupply period.

5.2.8 Space Station EMU PLSS

When EVA frequency becomes high, the launch weight penalty is driven up, as shown in Figure 5.2.8-1. This will drive the requirement for an on-orbit, regenerative, maintainable EMU system.

5.2.9 Space Station EMU Recommendations

At this point in the Space Station design development, it is recommended that the EMU evolve in three phases. Specifically when these EMU configurations are implemented into the Space Station evolution are dependant on EVA frequency needs, PLSS venting limitations and resupply weights; the recommended EMU phasing is as follows:

- o Configuration 1 - Uses the existing Shuttle EMU if the projected EVA's are less that 80 per year and venting is allowed.
- o Configuration 2 - Uses the Shuttle EMU with a non-venting PLSS if the projected EVA's are less than 80 per year and venting is not allowed due to contamination requirements.
- o Configuration 3 - When EVA frequency becomes greater than 80 per year, an extended life SSA and non-venting, regenerable PLSS is recommended. If this EMU configuration is used at 8 psi, it will have a mixed gas atmosphere, probably 3 psi O_2 and 5 psi N_2.

These three configurations are shown pictorially in Figure 5.2.9-1 with data for each shown in Figure 5.2.9-2. These charts show that the EMU weight goes up significantly (242-435 lbs) and PLSS volume also increases. Program cost trade-off dta, for these three configurations, is shown in Figure 5.2.9-3.

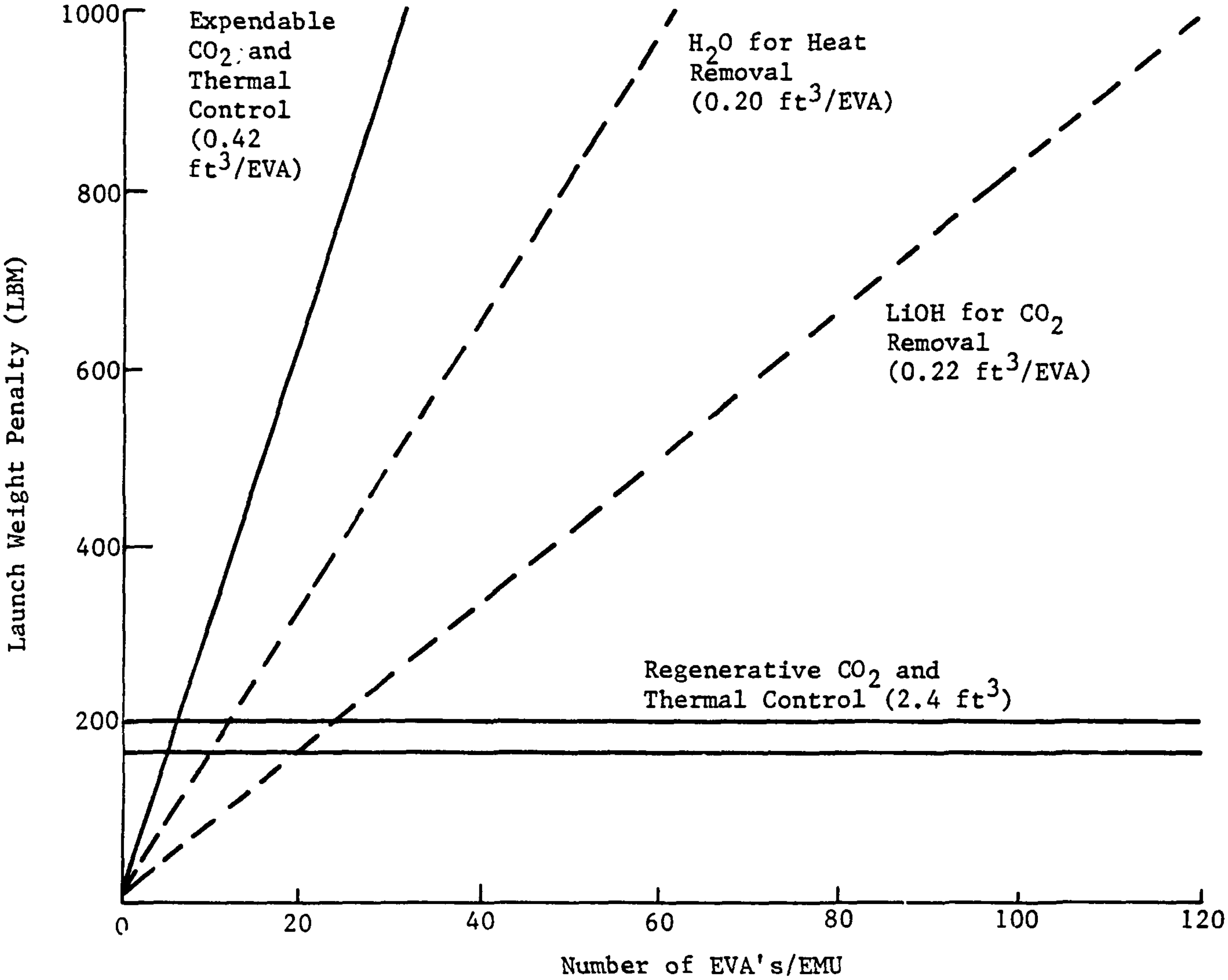


Figure 5.2.8-1
Launch Weight Comparison Expendable vs Regenerative PLSS Subsystems

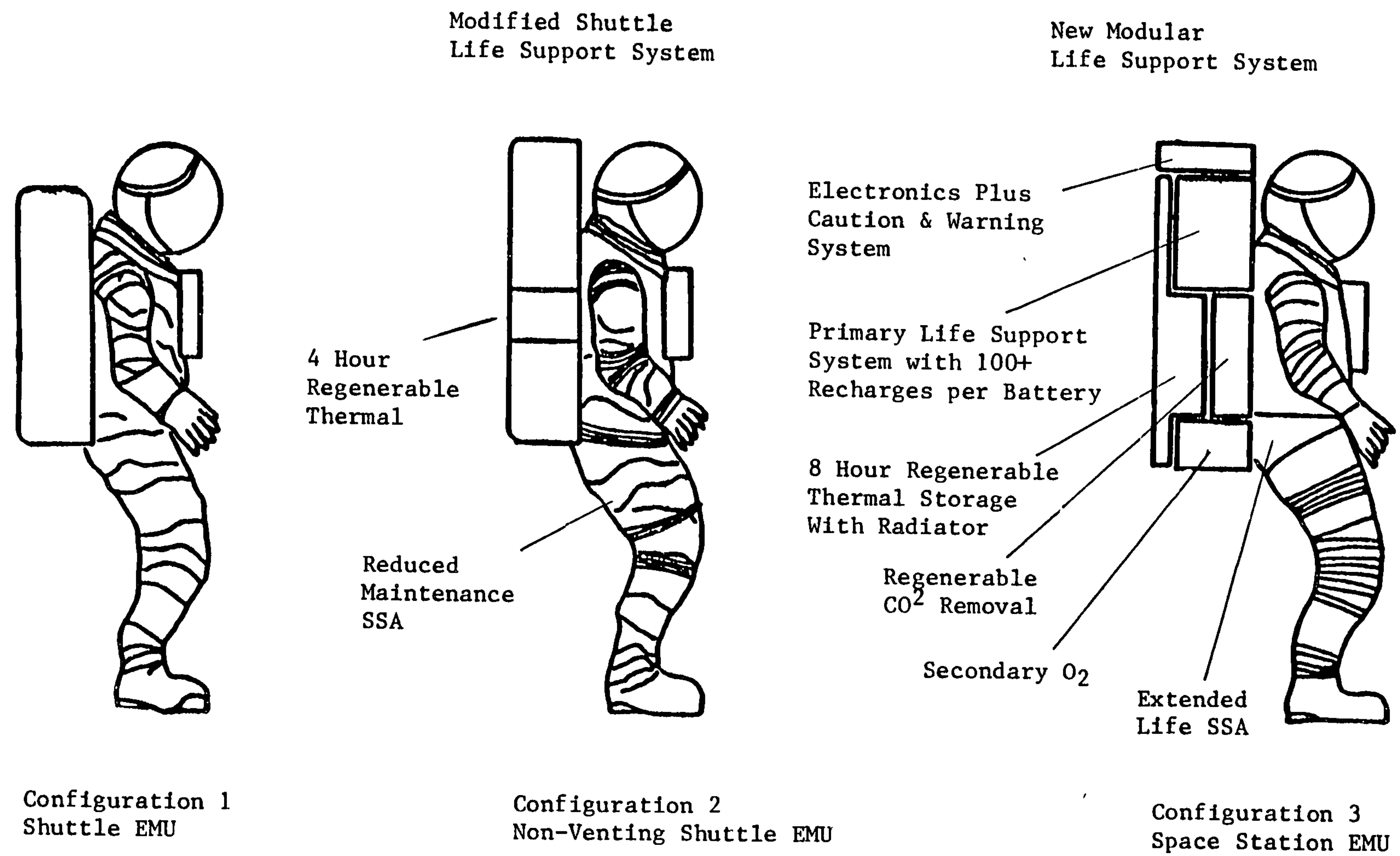


Figure 5.2.9-1 EMU Evolution

Configuration	1	2	3
Technology Availability	Present	1985-1990	1990-1995
Weight (lb) (Charged)	242	310	435
Expendable Penalty per EVA (lb/EVA)	25.1	15.4	1.0
Volume (Stowed) ft^3	16.2	17.1	22
Regeneration Power (Watts)	50	190	440
EVA's Between Refurbishment	5	15	65
Total Life (EVA's)	30	30	750

Figure 5.2.9-2 EMU Configuration Comparisons

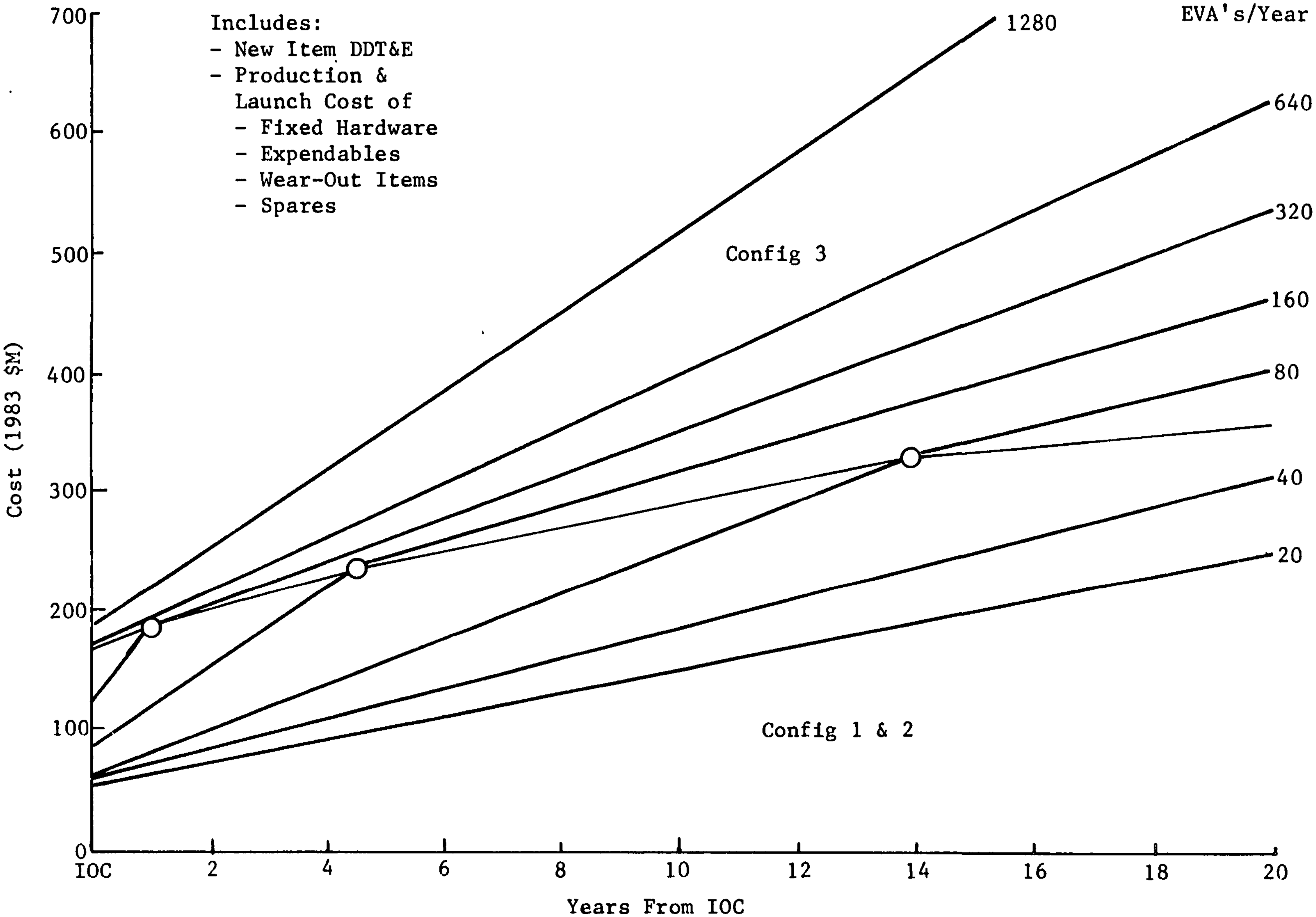


Figure 5.2.9-3 EVA Cost Trade-Off

5.3 SOCIAL-PSYCHOLOGICAL CONSIDERATIONS

5.3.1 Introduction

The space station design must be increasingly attentive to social-psychological variables if the space missions of the future are to be fully accomplished. The increased saliency of space related social-psychological issues stems from the following requirements inherent in the design and missions of the proposed space station:

- o The crew assignments on the space station will be of relatively long duration--three months and perhaps longer under certain circumstances.

- o The space station, once it is fully developed, will require from eight to twelve members. The accomplishment of station's objective will require the cooperation and interdependancy between these crew members.

- o The composition of the crew will be hetrogenious--ranging from scientists, to maintenance personnel, to career astronauts who will assume the responsibility and control over all station systems.

- o The environment within the Space Station will be constraining, thereby limiting the number and kinds of sensory experience to which crew members can be exposed.

All of these requirements are assumed to interweave into a causative network that can lead to the development space related stress reactions among crew members. If such stress is allowed to accumulate the mission objectives and indeed the safety of crew members and the station can become jeoparized.

5.3.2 Background

During the period of the 60's and 70's considerable research was conducted on the behavioral and psychological effects of long term confinement and isolation of personnel from their normal environment. This research was based on interviews, observations, and before/after psychological tests of participants. The environments ranged from the Artic, submarienes, simulated isolation chambers, to space craft (e.g., Gemini, Vostok and Skylab).

The evidence, though far from consistent, points to the following generalizations:

- Stress reactions were frequently observed and reported as manifested by mood changes, irritability, depression, insomnia, anxiety, interpersonal hostility, and sleep disorders.

- Performance decrements of some degree (e.g., reaction time, judgement, memory, and learning ability) were reported, primarily within the setting of isolation chambers.

- Based on a compilation of a large number of studies (Roth, 1968) there is a small but consistent trend pointing toward the negative effects of confinement and isolation becoming more pronounced over time.

5.3.3 Stress Reaction

Given the probable occurrence of stress reactions developing after some period of environment confinement, a central question arises: what features in a confined situation (as in a space station) contribute to stress? This question can best be answered by treating environmental confinement as a mediating variable that is associated with other conditions which directly or indirectly cause stress reactions. These conditions include: (1) a drastic change in the kind of social-physical environment which an individual has became accustomed to over the course of normal living patterns; (2) living in a constrained environment which on the one hand imposes on crew members the requirement to have frequent and intensive contacts with others while concurrently reducing opportunities for privacy, (3) the imposition of a demanding high work load for each crew member, and (4) and the requirement to work and interact with individuals of varying backgrounds and interests. Each of these conditions requires explanation.

5.3.3.1 New Social/Physical Environment - The space station is intended to provide each crew member with a safe and within limits confortable habitat. Nevertheless, because the Station imposes clear physical boundaries, the opportunities for varied and novel environmental experiences are limited. There is a large body of literature which indicates that individuals have a strong need for environmental stimulation and, perhaps more importantly, opportunities for change and variety in their experiences. Based on observations of individuals in isolated enmonments, the inability to satisfy this need can lead to psychological reactions ranging from marked performance decrements, perceptual distortion to mood changes. The design of the Space Station thus represents a challenge from the standpoint of ensuring adequate environmental stimulation and experiential variety.

5.3.3.2 Reduction in Personal Privacy - Within the limited space available in the module, a relatively large number of crew members will be living and working together. It follows there will be frequent an close interaction between crew members. Given the limited volume within the Station, each crew member will experience a reduction in his/her psychological space that defines a personal boundary between each crew member and others. It is assumed that individuals need privacy in much the same way that individuals need environmental stimulation. The inability to satisfy this need will also contribute to space related stress reactions.

5.3.3.3 A Demanding High Work Load - The accomplishment of many of the scheduled Space Station missions will impose on crew member moderate to high motor and intelectual demands. The likehood of such workloads when coupled with physiological disturbances such as motion sickness and autonomic nervous system adjustments represents another causative condition for stress. Further, research in group dynamics has shown that individuals within a group sitting exhibit higher production and increased morale when given opportunities to plan and implement organizational goals (space missions). The impositions of a rigid nonparticipatory work schedule during space missions may therefore represent a further source of frustration and stress to crew members.

5.3.3.4 Heterogenious Space Crew - Since the space station will be composed of crew members of varying backgrounds, interests, and skills, barriers to interpersonal communication can exists. For example, the technical goals and language of one crew members might not be clearly understood by others even though all crew members share a common cultural language. The possibility therefore exists that the accomplishment of a work related goals may be disrupted by the inability of one member to clearly communicate technical requirements to others. This means that the work objectives of a given crew member may be thwarted with the result that interpersonal hostility will emerge as an additional manifestation of space related stress.

5.3.4 Proposed Social Psychological Design of the Space Station

Given the above analysis (section 5.3.3), it is proposed that an effort be directed to plan and implement a social/psychological habitat within the space station analogous to the work of various kinds of engineers who will plan, design, and implement the space structure consistent with identified functional requirements. The following social/psychological interventions are recommended strategies for reducing predicted space related stress.

5.3.4.1 Volume Requirements - Given the total structural limitation of any space station, the available volume per crew member must be limited. Nevertheless, it is suggested that a minimum space requirement is needed for each crew member. The recommended volume suggested by Roth (1968) is approximately 600-700 cubic feet per man. This figure represents a total volume allocation per man which in turn must subdivided into the work area, personal crew space, a public section (where all crew members may gather--for example in a dinning area), and a service area. Of the total recommended 600-700 cubic feet per individual, Roth further suggests that approximately 20% of that total volume be allocated to a personal rest area--i.e., approximately 120 cubic feet for each crew member.

This space allocation for personal use is viewed as critical since it will provide each crew member with a minimum volume required to satisfy privacy needs including a personal "space" enabling a crew member to maintain some degree of separation and autonomy from others.

Related to the proposed space or volume requirements, there are other social/psychological design considerations. As was pointed out, the stringent confines of a space station means that the environment within the structure will lack the variety of experiences and stiumli to which individuals are normally exposed. The station habitat needs to incorporate features and capabilities to enhance environmental variety including the following design candidates:

- Flexible panels to allow each crew member to alter the configuration of his/her rest area.
- A varied food menu.
- Capability to privately communicate with friends and relatives.
- Games which can be used individually or with others.
- Movies which can be played on a video playback device.
- Availability of books and magazines.
- Generous use of colors (both bright and pastel colors) throughout the space station.
- Generous use of window (within design and safety limitations) to ensure crew members have frequent and varied views of space.

5.3.4.2 Group Organization - Research on organizational development has concluded that the type organizational structure (i.e., whether it operates in bureaucratic, "top to down direction" or in a democratic participatory framework) should be compatible with the kinds of organizational objectives and goals a group has evolved. Groups with well defined goals point toward an organizational structure which is centralized in terms of a top to down flow of control and direction. Groups with less structured goals point toward an organizational framework which accommodates and encourages member participation in the planning and implementation of group goals. This generalization applies to the space station, where it may be assume that many of its missions are not comprehensively defined--i.e., in terms of the definition of all of the procedural steps to be followed in implementing a mission. The absence in some instances of well defined missions strongly suggest a group organization within the space station which encourages a high degree of crew participation in decision making and a less centralize pattern of leadership in implementing a mission.

5.3.4.2 Activity/Work Scheduling - Consistent with the recommended group structure within the space station, it is also suggested that crew members have autonomy, within the limits of safety and the accomplishment of space mission objectives, to organize their daily work schedules. Thus, Bluth (1980) reports that the crews from Salyut 6 and the Skylab expressed the desire to exercise control over their daily work schedules. This recommendation to allow crew members some degree of autonomy in scheduling is also consistent with the strategy for facilitating variability in crew experiences as a countermeasure against boredom and monotony.

5.3.4.4 Cross Training of Assignments - Within the limits of system safety and the accomplishment of mission objectives, it is recommended that crew members have the opportunity to interchange selected tasks normally performed by other crew members. This cross training could be accomplished as part of the pre-flight training and orientation program.

The rationale for cross training also follows from the concept of environmental variety, in this case by providing crew members with an opportunity to change their daily work routines.

5.3.4.5 Work Socialization - Since a variety of space objectives and missions will be performed during the course of the life cycle of the Space Station, the accomplishment of these objectives will require a crew with diverse professional backgrounds skills and interests. As was pointed out earlier, crew heterogenity may contribute to communication barriers and interpersonnel hostility. It is thus suggested that as part of the preflight training program, the assigned crew receive an intensive orientation about the professional roles of other crew member. Part of this role socialization/training would be devoted to acquainting each crew personnel with the unique professional vocabulary and concepts of other crew members as well as the kinds of missions to be performed by others. This procedure would aim to reduce or eliminate communication misunderstandings as a source of interpersonal conflict and personal stress.

5.3.4.6 Stress Management - A variety of techniques have evolved over past few years whose objectives is control the stress responses. These techniques include Bio-feedback, deep breathing techniques, and self hypotheses. Research with stress management methods has been promising as evidenced by studies showing that hypertension can be reduced and control through the application of deep breathing techniques in the absence of any hypertensive drug intervention. These results are sufficiently encouraging to suggest that crew members be trained to use stress management methods as part of the pre-mission training program. Once acquired, a stress management technique can then be used by a crew members as a response against experienced stress. The design recommendation for a minimum personal living quarters on the Space Station represents ideal setting for applying stress management methods.

5.3.5 A Summary of Ideas

The central assumption of section 5.3 is that a stress reaction represents a probable response to confinement in a space environment. Once stress interfers with the accomplishment of mission objectives, efforts must be directed to identifying strategies for minimizing the stress reaction and thereby enhancing the likelihood that space objectives can be accomplished. The emphasis place in this section is that the Space Station can be engineered from a Social Psychological perspective to provide a set of environmental conditions which mitigate against the stress reaction.

The recommended social/psychological engineering principles involves structuring the Station's environment in ways which are assumed to enhance opportunities for crew members to control their work activities as manifested by increase opportunities to formulate work schedules and to more fully participate in the planning and implementation of space missions. Further recommended steps have been directed toward designing an Space Station environment which will provide increased opportunities for varied sensory experiences.

5.3.6 Research Requirements

The era of the Space Station is not an unrealistic expectation especially in light of the recent success of the Space Shuttle. Already a large program of medical and physical research has been proposed for the Space Station. Regrettably little or no social/psychological research has been planned even though man in the system loop represents a critical element in the success or failure of space missions. Accordingly, what is needed is research on optimizing the social/psychological conditions of a space crews of from 8-12 individuals. The fundamental objective of this research is not the desire to make people in space happy and congenial but rather to reduce projected stress which if accumulated can jeopardize the safety of presonnel, the space station, and the completion of mission objectives.

The research needs are clear—namely, to: (1) identify and define required research objectives, (2) develop the appropriate methodology to the observe study variables within the setting of the Space Station; and (3) implement and evaluate study findings. As suggested in earlier paragraphs of Section 5.3, there is a clear need to conduct research focusing on the following categories of social/psychological engineering:

- o Defintion of an optimal group structure in a space station.
- o Designing the Space Station interior to maximize the opportunities for environmental stimulation and variety.
- o Selecting and evaluating stress management techniques applicable to space situations.

5.4 MEDICAL AND PHYSIOLOGICAL CONSIDERATIONS

The proposed Space Station involves complex space construction, extended periods of space confinement, and frequent EVA missions. Both the construction and operation of the Space Station requires personnel of varied backgrounds including scientists, maintenance staff, construction workers as well as the required astronaut crew. This heterogenious crew will perform varied activities both within an external to the station. These activities clearly imply the potential for personal accidents, infections, and adverse physiological changes associated with microgravity. Clearly, then, the conceptualization and detailed design of the proposed space station must incorporate the requirement for medical treatment and health maintenance including consideration of needed facility space, equipment and the required crew members to support the facility.

An overall objective of the space station's medical function is to ensure that the work efficiency of the crew is maintained, thereby ensuring that mission objectives of the Space Station can be accomplished in a timely and safe fashion. Considering the range of activities in operating the Space Station, the following categories of medical conditions are projected:

I Common Medical Problem of Crew Members
- A - Non work related problems such as infections, allergies;
- B - Work related problems such as bruses, cuts, fractures;

II Space Related Medical Problems
- A - Effects of Microgravity such as space sickness
- B - Radiation injuries

III Health Maintenance/Preventive Medicine
- A - Exercise Activities, diet planning, etc
- B - Mental Health and Psychological Maintenance

The above outline highlights the scope of the Medical activities. Note that these activities include Mental Health support, anticipating the potential for stress reactions resulting from relatively long term confinement within the environment of the space module. Section 5.3 (Volume IV) presents a detailed description of the possible causes of space related stress as well as a discussion of counter measures which could be incorporated into the planning of the Space Station. Additionally, the medical-health concept also includes the provision for exercise activities to ensure positive body tone in a zero gravity environment.

Medical planning for the Space Station has been conceptualized as progressing through four stages of development. Briefly, medical activities are initially performed on the Space Shuttle (Category 1). As work on the Station progresses, the Space Shuttle will no longer have the capabilities to support medical requirement. The medical activities and facility will be shifted to the Station. (A description of the four categories of medical support is included in section 8.4.3, Vol II).

Once the Station becomes operational in the late 80's and early 1990's it is assumed that the Category II medical facility will support the Health needs of the Space crew during this time frame. Briefly, Category II includes the following components:

a. First Aid Station - Located in the Space Station, this facility will treat injured crew members and contain essential equipment and medication including introvenous fluids, oxygen, defibulator, etc.

b. Space Station Medical Kit - This kit represents an integral part of the First Aid Station, and will contain additional drug supplies and surgical equipment.

c. Hyperboric Chamber - This facility will be designed to withstand a minimum of three atmospheres and will be used for treating decompression sickness.

d. Exercise Area - This facility will be incorporated as part of the recreation area. It will contain a treadmill, exercise bicycle, etc.

For the category II facility, one of the basic crew members will be trained as an emergency medical technician. All the remaining crew members will be trained in a variety of first aid skills which could be applied on an emergency basis.

Over the course of the expected life cycle of the station, the medical function will be expanded into category three and ultimately category four capability. With this added capability, increased attention should be directed toward preventative medicine including mental health. The preventative activity would be supported by health specialists who would function in dual role of researchers in the Life, Medical, and Behavioral Sciences as well as health practitioners. Thus over successive missions the space station crew would include Physicians as well as behavioral specialists (e.g., a Psychologist or Psychiatrist).

5.5 SPACE STATION PRESSURE & EVA PRESSURE SUIT CONSIDERATIONS

The factors that must be considered in the selection of cabin pressure are fire saftey, metabolic oxygen requirements, oxygen toxicity, oxygen pre-breathe requirements for EVA, EVA suit design, personnel and equipment air flow (fan sizing) requirements and orbiter compatibility. The following discussion is partially an except from previous Space Station Studies. Most parameters have not changed and the recommended Space Station cabin pressure cannot be made at this point in the system evolution. Some studies recommend a pressure in the 11 to 12 psia range, with an associated EVA suit pressure of 5 to 6 psia. Other studies recommend a full atmosphere, 14.7 psia and an 8 psia EVA suit.

The following discussion focuses on the factors which must be considered during the cabin pressure selection process.

5.5.1 Commonality With Shuttle Cabin Pressure

A common pressure in both the Space Station and Shuttle is highly desirable and possibly a requirement during the docked phase. A differential pressure that requires an air lock between the two vehicles would be highly undesirable. Should other factors drive the Space Station to a lower pressure than the Shuttle's one atmosphere, (i.e. 11 to 12 psi) the Shuttle could operate at a reduced pressure for the docked equipment/crew transfer time period.

5.5.2 Eliminating Pre-Breathe

The present 4.3 psi suit, when used in the Shuttle 14.7 psi atmosphere, requires the crewman to breathe 100% O_2 for 3.5 hrs prior to the actual EVA. This is required to purge the nitrogen from his system and prevent the "bends" during the 14.7 to 4.3 psi pressure reduction. Suit preparation and donning takes 1.5 hrs. at the end of the 3.5 hrs period. During the 2.0 hr. initial pre-breathe time the crewman can perform other tasks while using a portable oxygen mask. However, the fact remains, the crewmen cannot go EVA without the 3.5 hr. preparation time. When considering the Space Station scenarios which utilizing many EVA's per week and possible quick reaction time EVA's this pre-breathe time significantly impacts timelines. The elimination of the initial 2.0 hr. pre-breathe and possibly a more efficient suit preparation sequence could reduce the pre-EVA timeline from the present 3.5 hrs to between 1.0 and 1.5 hrs. The relationship between suit pressure to avoid pre-breathe and cabin pressure is shown on figure 5.5.2-1.

5.5.3 Oxygen Toxicity

The partial pressure of oxygen in a breathable atmosphere must be limited to avoid toxic effects. The upper limit of oxygen partial pressure selected for Apollo and Skylab cabins was 5 psia, and in the case of Skylab this was for continuous use. There were some medical evidence of undersirable oxygen toxicity in these programs, as reported in the literature ("Extravehicular crewman Work System Study Program", Final Report, Vol. II, Construction, July 1980, Contract NAS 9-15290 R. C. Wilde, Hamilton Standard). There has also been evidence of toxicity revealed in tests run since then, but there does not seem to be a real consensus of the degree of seriousness ofthese observed effects. An oxygen concentration as high as 4 psi O_2 partial pressure could probably be tolerated continuously in the Space Station cabin, but this is a moot point because the Space Station will utilize a two gas atmosphere making this high a PPO_2 unnecessary, as shown on the left-hand vertical scale of Figure 5.5.2-1.

Oxygen toxicity during EVA is a different matter. First, EVA will occur for an individual crew member for a maximum of about 25 percent of his total in orbit time, and second, the atmosphere in the suit is now pure oxygen. There is evidence that 8 psi pure oxygen pressure in the suit will result in unacceptable toxicity effects, as described in the literature (NADC-74241-40, "Physiological Responses to Intermittent Oxygen and Exercise Exposures", E. Hendler,NADC, Warminster, PA, 1974). For eight hours a day, a 4 psia level is generally accepted. The maximum allowable suit level of pure oxygen for EVA probably lies between 4 and 8 psia. But this not a black or white matter, and considerable differences in tolerance between individuals undoubtedly occurs. A limit of 6 psi (100% O_2) is logical since 4 psia is acceptable and 8 psia is not, but this a tentative limit, not clearly defined. The presently proposed 8 psi suit (configuration 3 in Figure 5.2.9-1 will be mixed gas with an approximately ratio of 3.0 psi O_2 and 5.0 psi N_2.

5.5.4 Weight of Stored Cabin Pressurization Gas

The leakage flow through any hole or leak in the vehicle pressure wall is directly proportional to cabin pressure. Space Station cabin leakage is expected to be about 5 pounds of air a day. Another 5.3 pounds of air per day is expected to be lost in use of airlocks on a EVA day assuming pump down to 2 psia for 14.7 psia cabin. This total air loss can be made up from oxygen produced from wastewater, by electrolysis, and by nitrogen obtained from the decomposition of 9.3 lb/EVA day of hydrazine. Capability for one complete repressurization utilizing stored high pressure gas weighs approximately 750 lb, plus tankage. The weight of the above varies as follows with cabin pressure:

Design Cabin Pressure	Resupply Hydrazine Required For Nitrogen Makeup Per 90 Days	Stored Repressurization Gas, Including Tankage	Resupply Water Including Tankage Required For Oxygen Makeup Per 90 Days
14.7 psia	775	1321	270
11 psia	580	989	202
9 psia	474	809	166

5.5.5 Vehicle Mechanical Strength

Thickness of the Space Station vehicle skin is dictated by the need for protection from meterorites and space debris. Reducing the vehicle cabin pressure would therefore not reduce skin weight.

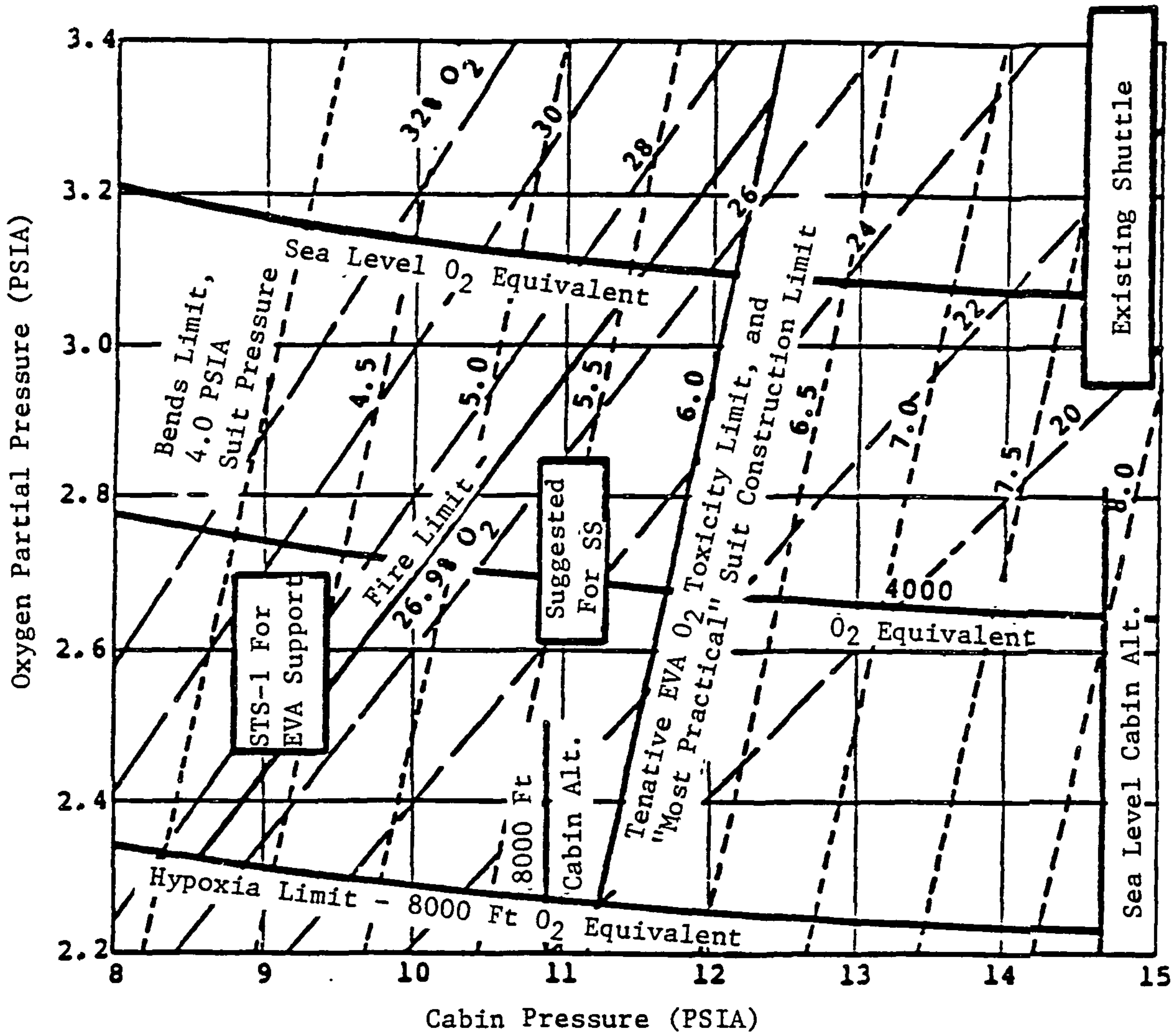


Figure 5.5.2-1 Space Station Cabin and Space Suit Pressure Considerations

5.5.6 Hypoxi

Is a physiological response (generally reduced brain function) to the relative lack of oxygen in the body, caused by a low partial of oxygen. A partial pressure of oxygen in the Space Station corresponding to 4,000 feet or less is considered a requirement by NASA ("Medical Science Position on Space Cabin and Site Atmospheres" Position paper by NASA JSC/SD, May 1980). An altitude as high as 8,000 feet equivalent oxygen level is considered to be an acceptable level for commercial aircraft pressurization.

5.5.7 Flammability

The famability of materials, used inside the Space Station, is a function of the partial pressure of O_2 within the total cabin atmosphere. The sea level oxygen concentration of 21 percent in the Space Station cabin would be desirable from a flammability standpoint. Only one major material used in Shuttle, a silicon fiberglass line insulation, has failed to meet fLammability tests at 35 percent O_2, and this material will be replaced in later Shuttle vehicles. The cabin pressure control tolerances for the current Shuttle result in a maximum normal oxygen concentration of 23.8 percent O_2. A caution and warning light is set of Shuttle to trip at the 25.9 percent level with a 26.9 percent O_2 absolute maximum level. These same levels are probable going to be inherited by Space Station as the flammability requirement. The relation between flammability and cabin pressure is shown on Figure 5.5.2-1.

5.5.8 The Effects of Selected Cabin Pressure on ECLS System Components

The specific cabin pressure level selected for design has many ramifications. One of these is the fact that a lower cabin pressure makes rejection of heat from the cabin air to the radiator coolant fluid more costly in terms of system size, complexity, and power consumption. This is because cabin air is the first stage coolant for rejection most of the heat load generated in the cabin. This heat transfer is a function of air mass flow, not CFM, and therefore reduced air density increases the power needed to circulate the airflow required for heat transfer. This study considers a sea level cabin pressure as baseline. If this hardware were built and developed, and the cabin pressure were then reduced, the baseline heat rejection cabability of the baseline ECLS would degrade as shown on Figure 5.5.8-1. This higher temperature may be undesirable so changes to the system may have to be made to accommodate lower cabin pressures. These changes need by made only in the components involved in the cabin air temperature control and ventilation functions, since the other ECLS systems components are unaffected.

The simplest change which can be made to the ECLS system to compensate for reduced cabin pressure would be to increase fan air handling capacity in order to maintain the design value of mass airflow, and accept the power and noise suppression penality which would occur as a result. Figure 5.5.82 shows how fan power would increase to hold airflows, and therefore heat transfer constraint. Unfortunately, this solution of increasing fan size to accommodate a lower cabin design

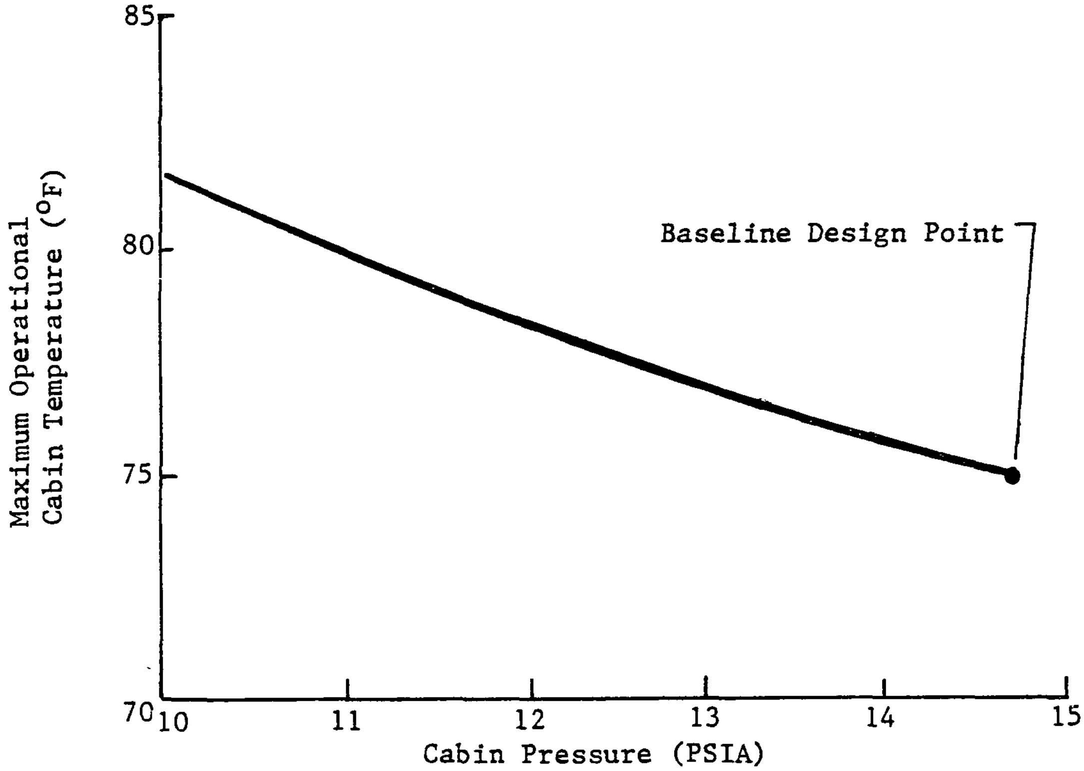


Figure 5.5.8-1 Cabin Pressure Effect on Cabin Temperature

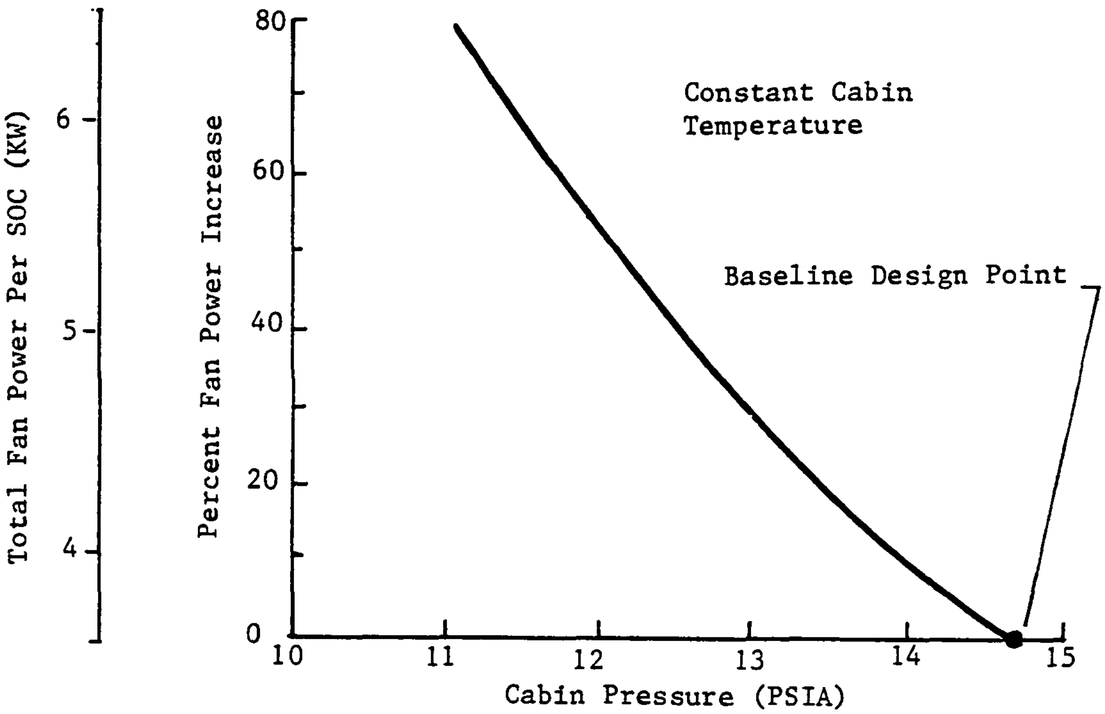


Figure 5.5.8-2 Cabin Pressure Effect on Fan Power

pressure would add significantly to the electric load demand of the ECLS. Figure 5.5.8-2 shows for example that an increase in power is required per full Space Station from 3.6 kw to 6.5, or a delta increase of 2.9 kw, in dropping cabin pressure from sea level to 11 psia. Battery weight needed to provide this 2.9 kw of power on the dark side would weigh 910 pounds.

This weight does not include the weight of hydrazine which must be resupplied to keep an additional 2.9 kw of solar array in orbit.

An alternate approach would be to redesign all air handling components of the ECLS to maintain required airflow while holding the fan power increase to a minimum. This solution requires larger heat exchangers, filters and distribution air ducting, as well as larger fans. The result of a family of such system designs is shown on Figure 5.5.8-3. Note on this Figure that the fan power delta is now only 1.0 kw in going from a sea level to 11 psia cabin. This is preferable to the 2.9 kw delta which results from changing only the fans, as shown on Figure 5.5.8-2. This lower power eenalty is obtained by increasing the size of other air handling components in the system by 28 lb. and 1.7 ft^3.

5.5.9 Conclusions Regarding Selection of Cabin Pressure

It is beyond the scope of this study to recommend the design value of cabin pressure which should ultimately be selected for Space Station. As the preceding sections have pointed out. There are many diverse factors to consider that the final selection is a compromise. The following is a set of individual conclusions which may be reached concerning these factors:

1. Figure 5.5.2-1 shows that the logical cabin pressure for Space Station lies within the boundaries of a traingle formed by the 26.9 percent oxygen Fire Limit on the left, the tentative O_2 Toxicity Limit and "Practical" Suit Limit on the right, and the 8,000 foot equivalent oxygen level at the bottom of the triangle. Existing Shuttle pressures are shown on this Figure for reference.

2. There is a preponderence of medical/health logic to favor selecting the Space Station pressure cabin toward the upper right portion of the traingular boundaries, mainly because man obviously works best near his ancestral sea level environment. The power, weight, and volume penalities of operating toward the upper right portion of the triangle, as opposed to operating toward the lower left portion, are not great. These penalities are about one percent of the total resources of Space Station.

3. A normal cabin pressure error band of $\pm$ 0.2 psi is recommended for Space Station based on this value being used on the current Shuttle.

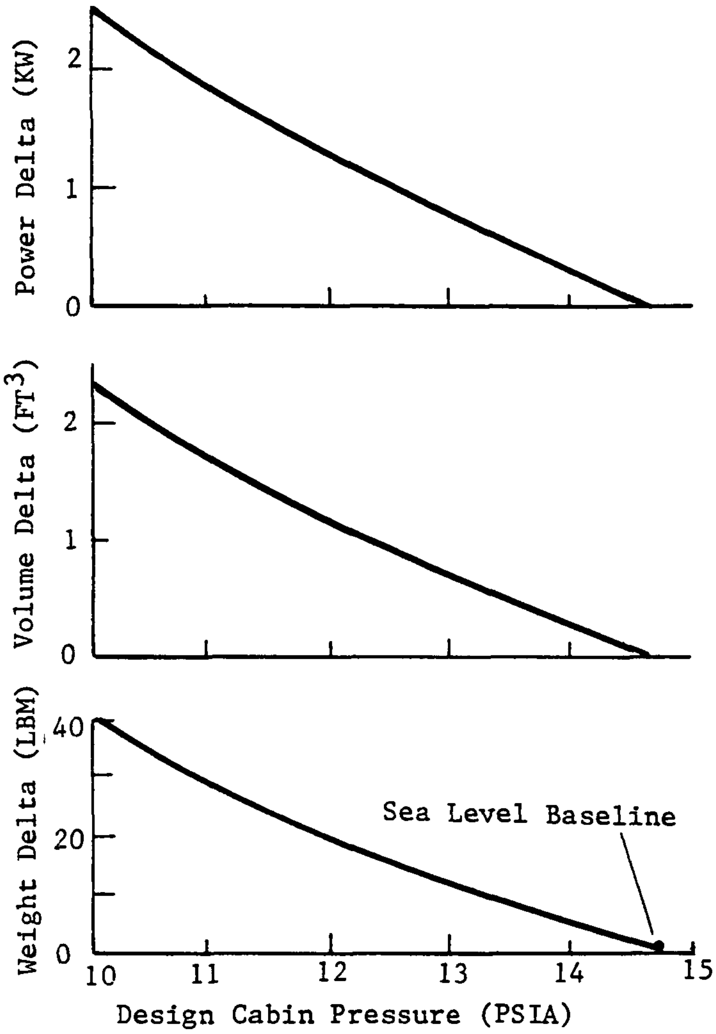


Figure 5.5.8-3
Penalty On Reduced Cabin Pressure (Family of Optimized Systems)

4. The boundaries of the traingle call for tighter control on normal oxygen partial pressure level than is exercised on the existing Shuttle. A control of about $\pm$ 0.1 psia oxygen partial pressure is recommended for Space Station. Automatic control on Space Station, should be at least this accurate.

5. The box labeled " suggested for Space Station" on Figure 5.5.2-1 is just that, a suggested compromise between the many diverse factors involved. Based on information available during this study, it is a logical, but not firm, selection.

5.6 FOOD AND WATER

In order to establish a Space Station baseline for food and water requirements, the Shuttle crews diet and water consumption rates were used for reference. The only exception is that the Space Station crews diet should be supplemented with frozen food. A reference mission crew size of 4 men for 90 days was (360 man-days) was selected. (360 man-days = 1080 meals). The following can be used for volumetric and weight planning:

Consumables	Weight (lb./man/day)	Weight (360 man days)	Volume (ft^3)
oShelf Stable Food			
-Food	1.6		
-Individual Packaging	1.0	936 lbs	1080 meals packaged at 18 meals/ft^3
	2.6 lbs		= 60 ft^3 meals/ft^3
-Packaging Factor (racks,containers,etc.) =1.6			60 ft^3x = 96 ft^3
oFrozen Food			
-Food	1.0 lb.	360 lbs.	15 ft^3
-Freezer		610 lbs.	20 ft^3
oWater			
-Drinking	1.5	540	
-Rehydration	2.7	972	
	4.2 lbs	1512 lbs	

Hardware*	Weight (LBS)	Volume (ft^3)
.Refrigerator	20	5
.Oven	75	5
.Dishwasher	105	8
.Accessory Storage	20	5
.Warning Table	40	2
.Trahs Compactor	60	2
.Water Tankage (empty)	370	24

*All wgts & volumes are estimates

Food Storage Distribution Guidelines

1. Frozen and shelf stable food must be redistributed after the arrival of a logistics Module to ensure that there will always be sufficient contingency food available if one storage location is evacuated and resupply is not available for 90 days.

2. Contingency rations should be the same as used for the normal diet, including the frozen food. Dedicated "c-ration" type food should not be used.

3. Contingency rations should be at least 1.4 lbs/man-day. This provides 55% of the standard 90 day food supply at any time and therefore, in a contingency situation the crew will eat only 55% as much as they eat normally.

4. If a food storage area must be evacuated, assume that it's food stores are not available for retrieval. A severe packaging penalty would be required for vacuum exposure if this food was retrievable.

5. Stored food distribution:

Logistics Module	-	2/3 frozen, 1/3 shelf stable
Habitability Module(s)	-	1/3 frozen, 2/3 shelf stable

6. Plan on consuming all of the frozen food prior to a resupply.

7. The following requires technology development:

   a. Freezers - 10^{o}F and refrigerators 40^{o}F.

      Consideration should be given to recent breakthroughs in frozen food technology which permit storage of many frozen foods at 20^{o}F rather than 0^{o}F. Although all frozen foods cannot be stored at 20^{o}F, a considerable savings in energy can be realized by storing some at 20^{o}F.

   b. Oven

   c. Dishwasher

   d. Food preparation and serving equipment, i.e., trays, tables, chairs, dispensers for beverages, etc.

   e. Capability for growing fresh salad greens

   f. Trash Management

   g. Food Development - Baking in space, milk based beverages, *freezing ice cream*.

5.7 CREW SYSTEMS WEIGHTS AND VOLUMES

The following data is provided for sizing a logistics module. The items included are related only to crew systems support. The weights and volumes are those for an almost totally closed ECLS system and include expendables and spares. The system also includes cloth washing.

CREW SUPPORT

	Resupply Weight (LBS)	Resupply Volume (FT^3)
o Cabin ventilation and thermal control	12	1.75
o Air revitalization	90	4.5
o Atmospheric supply	540	23.8
o Heat transportation and rejection	6	0.5
o Water processing and management	135	7.1
o Health and hygiene	280	55
o System control and display	2.25	1.5
o Food - Dried	936	50
- Frozen	360	15
- Resupply freezer		40
o Clothing	40	3.0
o Personal items	40	4.0

EVA PRESSURE SUIT

Item	Weight (LBS)	Volume (FT^3)
o Shuttle EMU	277.6	16.2
- Water used per 8 hr. EVA	12.9	0.2
- LiOH Canister per EVA	6.5	0.22
o Closed loop 8 PSI Suit	435	22.0

THIS PAGE INTENTIONALLY LEFT BLANK

6.1 ELECTRICAL POWER

6.1.1 Requirement

The conceptual design of the electrical power subsystem (EPS) was based on the set of general requirements shown in Table 6.1.1-1 as well as the power requirements defined for the space station subsystem and users. For sizing purposes, the power generated must satisfy the average needs of both areas as well as the peak power levels anticipated. The basic design life established by the statement of work as 10 years. However, continued use of the space station should be planned for after this time period. The power capability will be less due to system degradation as time progresses past the ten year mark. Operation of the station could continue at a reduced power level or an incremental growth step taken when required to raise the power to a specified level.

The anticipated launch in the early 1990's was used to provide a basis for evaluating the power technology items being developed and define their availability for the initial station configuration or possible use as a growth item. The EPS design must address the possible incorporation of advancing technologies at initial launch as well as during the life of the station. Significant pre-planning must be performed to assess what is and is not possible to incorporate later. For example, if the initial launch employs batteries as the energy storage device and a further advanced energy storage device (e.g., bipolar NiH2 or fuel cell) becomes available, its use may require that some provision be made at initial station launch. Also, advanced solar cells, or a nuclear power source capability would require some pre-thought to enable their method of incorporation during the life of the station.

It is not desired at this time to define in detail an EPS design. The necessary level of design is to the point at which the EPS effects the architecture of the station. For example, the choice must be made regarding the type of power source to be used. A solar array effects the architecture and dynamics in one way while the selection of a nuclear source would present a different station architecture. However, it is not necessary in this study to select power semiconductors, relays, or the specific power conditioning technique if station architecture is not effected.

Another requirement seen in Table 6.1.1-1 addresses maintainability. It is necessary in the concept selected to be able to perform maintenance on the items selected since the station life is so long. Component failure or degradation may require replacement by either extra vehicular activity (EVA) or intra vehicular activity (IVA). In the case of a nuclear power source, very few items related to the source are maintainable.

Table 6.1.1-1 Electrical Power Subsystem Requirements

—Provide Power to Users as well as Subsystems.
—10 Year Basic Design Life, Low Earth Orbit
—Launch in 90's
—Define Growth/Evolutionary Concepts
—Limit Design to the Level that Architecture is Effected.
—Consider Incorporation of Future Technology Developments During Life.
—Identify Technical Issues/Concerns.
—Compatible with STS Launch Capabilities.
—Subsystem Maintenance to be Considered.

In addition to the general items of Table 6.1.1-1, the power levels seen in Table 6.1.1-2 were provided. This reflects how each subsystem power requirement is expected to change over the first six years of the station. These were based on the user needs addressed in section 4.0 of this report. Here it is expected that the main station will be at full capability in 1995. The user levels seen reflect the build up of the station. It is interesting to note that a significant portion of the power required is because of the presence of man. By observing the Life Support entry for the years 1991, 1993, and 1995 then comparing these values with the power totals, it is seen that they comprise approximately 30% of the required power level. It can be seen that a 20% contingency has been placed on the provided requirements. It is felt that at this early stage, the power requirements must be considered as "soft" (i.e., highly uncertain). Therefore, a significant, though not unusual, contingency factor has been applied.

These values have been graphically displayed in Figure 6.1.1-1. The solid stairsteps reflect the totals of Table 6.1.1-2. The two levels seen reflect the daytime (upper) and eclipse period (lower) power requirements. The difference is caused by operating some environmental control equipment during the sunlit portion of the orbit but not during eclipses. This reduces the requirement on the energy storage device and permits a smaller solar array size. Figure 6.1.1-2 illustrates how the selected program option incorporates the use of unmanned platforms and their required power level. The platforms are different from the manned station in that they are expected to be unmanned and have full power capability at launch. The EPS conceptual design for these are based on that of the main station to maximize design and hardware selection transfer and minimizing cost. Since the manned station EPS design will be modular, it is recommended that the main station power modular hardware be implemented on the platforms to the maximum extent possible. Several of the power levels seen in the Figure 6.1.1-2 are comparable to the initial station second year power value. This may permit direct transfer of design and hardware selection. However, those platforms designated for geostationary location would require further design analysis. The time phasing of the platforms appear to be easily accommodated. The platform launch years do not coincide with the main station incremented power requirement increases. This would permit an efficient EPS hardware design and production schedules.

6.1.2 Conceptual Design

Before the conceptual design could be formulated to meet the identified requirements, several trades were performed. Table 6.1.2-1 tabulates the most significant trades. The trades that had the most impact on space station architecture are those dealing with the power source, and the energy storage device selection. The remaining trade items were important in considerations of evolution, modularity, subsystem efficiency, maintenance, weight and volume.

Table 6.1.1-2 Defined Power Requirements - Manned Station

	Year					
	90	91	92	93	94	95
Comm	241	241	256	496	596	856
Life Support (Day/Night)	9.2 Kw / 5.1 Kw	9.2 Kw / 5.1 Kw	18.2 Kw / 10.1 Kw	18.2 Kw / 10.1 Kw	18.2 Kw / 10.1 Kw	27.3 Kw / 15.1 Kw
CMD/Data	205	205	490	490	490	880
Prop	200	200	1000	1000	1000	1000
ACS	450	450	450	450	450	450
TCS	750	1010	1750	1750	1750	2260
EPS	297	297	674	674	674	890
Subsystems Total	11.3 Kw / 7.2 Kw	11.6 Kw / 7.5 Kw	22.8 Kw / 14.7 Kw	23.0 Kw / 14.9 Kw	23.1 Kw / 15.0 Kw	33.6 Kw / 21.4 Kw
Users	1 Kw	16.3 Kw	25.7 Kw	26.2 Kw	26.2 Kw	30.7 Kw
Total	12.3 Kw / 8.2 Kw	27.9 Kw / 23.8 Kw	48.5 Kw / 40.4 Kw	49.2 Kw / 41.1 Kw	49.3 Kw / 41.2 Kw	64.3 Kw / 52.1 Kw
20% Contingency	2.46 Kw / 1.64 Kw	5.58 Kw / 4.76 Kw	9.7 Kw / 8.1 Kw	9.84 Kw / 8.2 Kw	9.86 Kw / 8.24 Kw	12.86 Kw / 10.42 Kw
Total	14.8 Kw / 9.7 Kw	33.5 Kw / 28.6 Kw	58.2 Kw / 48.5 Kw	59.0 Kw / 49.3 Kw	59.2 Kw / 49.4 Kw	77.2 Kw / 62.5 Kw

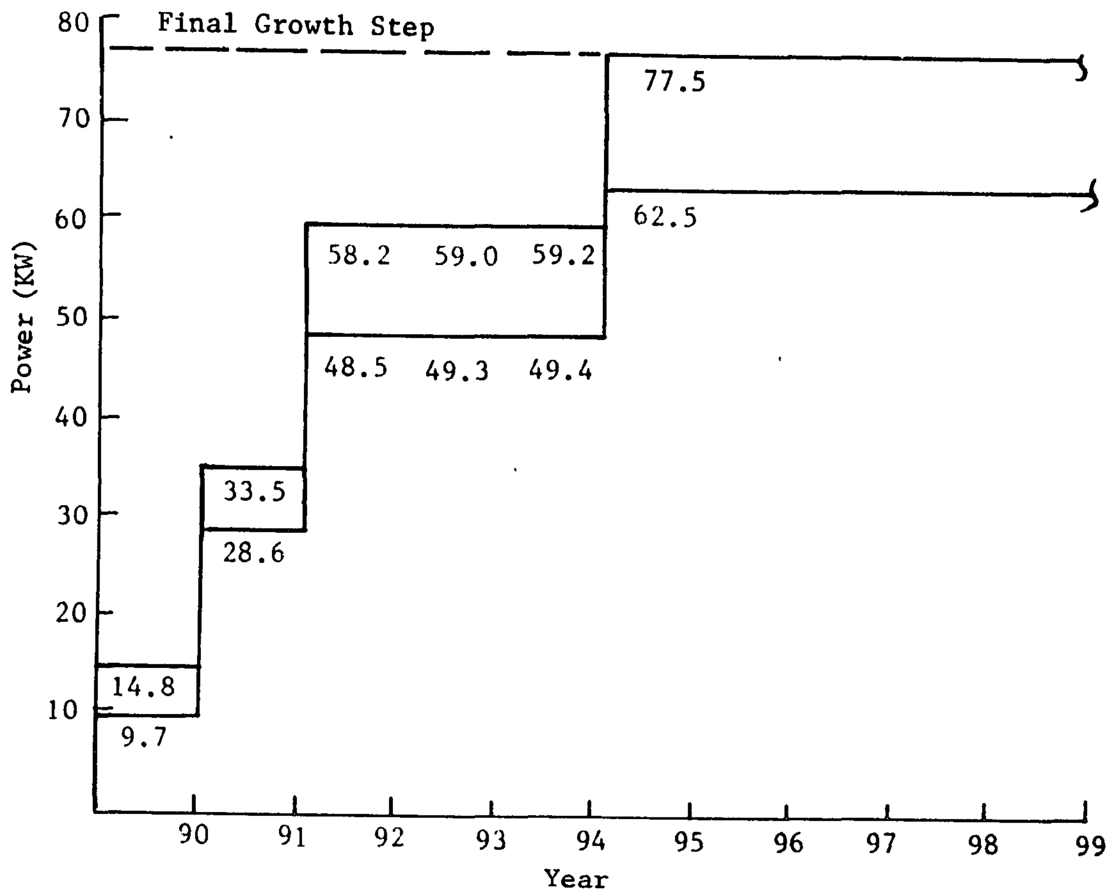
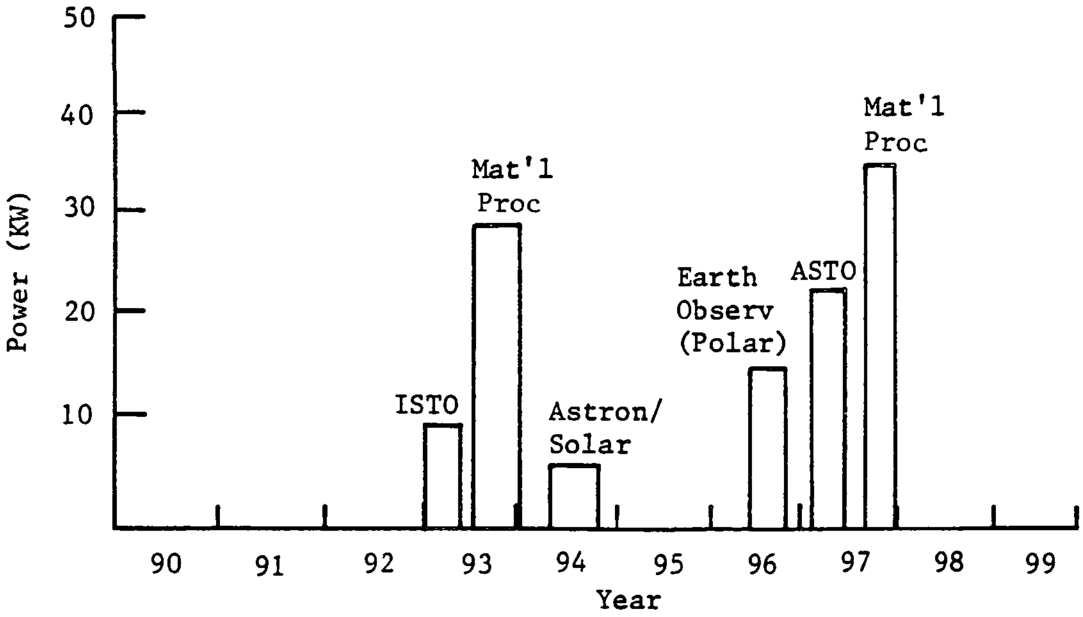


Figure 6.1.1-1 Time Phased Power Requirements—Manned Station



- Full Power Capability at Platform Launch
- Capable of Maintenance
- Independent from Main Station
- EPS Design Based On Manned Station

Figure 6.1.1-2 Power Requirements - Platforms

Table 6.1.2-1 EPS Trades

- Power Source
- Energy Storage
- Direct Energy Transfer Versus Peak Power Tracking
- Transmission Voltage Level
- Preliminary Equipment Selection
- Growth Increments
- Automation Level
- Power Transfer Scheme
- Power Management Approach
- Solar Array Degrees of Freedom
- Modularity Level

A summary of the power source trade is seen in Table 6.1.2-2. The items considered included a nuclear power source, non-regeneration fuel cells, and solar arrays of various cell types. A nuclear source offers a low drag profile and high power levels with one launch. The approach also results in a relatively simple EPS design, low drag, and low component quantity. The difficulties with the nuclear approach are challenging. The reactor development schedule for a unit compatible with a manned space station will likely result in a qualified reactor availability in the mid 1990s. The reactor being pursued by Los Alamos has the characteristic of leaking fision gases which must be captured and contained if used in a manned program. The present design uncertainties include the reactor type to be used, the power conversion device (e.g., thermoelectric, rankine cycle, Brayton cycle, Sterling cycle) and the heat rejection technique. Further, the development schedules in these support areas is unclear.

The world safety concern may require that the nuclear device be at safe orbit at the beginning of the mission to preclude accidental de-orbiting of nuclear materials. This may require a long tether if the space station is to be at 250 nmi and the nuclear source at 300 to 350 nmi. This altitude for the nuclear device would be required to assure a 300 to 400 year deorbit period. If other means for reactor disposal are to be employed this would require redundant design to assure operational success.

The use of fuel cells as the prime source of power requires large amounts of fuel based on the approximation of .98 lbs of fuel per kWH. For a 30 day period at initial station capability of 30 kW over 20,000 lbs of fuel would be required. For a full capability station the number increases proportionately. Even with cryogenic scavenging of the external tank, the space station would become dependent on cryogenic delivery schedules.

The solar array option has several possible solar cells and configurations feasible for power generation. Silicon, silicon concentrators, GaAs, GaAs concentrators being the most viable for an early 1990's launch date. GaAs cells have the advantage of higher efficiency over silicon, more power at high operating temperatures and seem to be more radiation resistant. Due to its operating characteristics a GaAs solar array is approximately 35% smaller than the comparable silicon array. However, there is no GaAs cell production capability at this time. The MANTECH program being performed by Applied Solar Energy Corp (ASEC) will provide a 1000/week capability. This will therefore improve the cell availability issue. However, it is doubted that the other technical issues surrounding GaAs could be solved by the first launch. Potential problems with GaAs is the lack of flight data, cell laydown development, and lower reverse voltage characteristics. San Marcos and LIPS programs used some GaAs cells but there were small quantities and small array areas. Great care can be taken in laying down cells for a small array, however this would not be the case for the space station array.

Table 6.1.2-2 Power Sources Trade Summary

Item	Pro	Con
Nuclear Power Source	- Low Drag - Low Weight & Volume for Power Level - Excess Power Available - Simplifies Power System (i.e., Provides Eclipse Power) - Full Capability in One STS Launch	- Development Status/Cost - Development/Qualification Schedules - Design Uncertainties i.e., Reactor Type, Heat Rejection Method Selection, Power Conversion device Selection - Safe Altitude Distance From Space Station - Safety - Maintainability - Single Point Failure
Fuel Cells (Non-Regenerative)	- Provides Eclipse Power - No Drag on Station	- Dependent on STS for Resupply - High Fuel Consumption - Creates Excessive Water
Solar Array - Silicon (Planar) - GaAs (Planar)	- Known Technology - Much Flight History - Low Launch Volume - Reduced Area - MANTECH Program in Progress - Higher Efficiency than Si - Better Temperature Characteristics - Potential for Annealing - Appear More Radiation Resistent than Si	- Largest Drag Configuration - Little Flight History - Little Cell Laydown Data - Cells More Brittle Than Si - Reverse Voltage Characteristic Lower than Si - Cost Prediction Low Fidelity - Ga Material Availability & Recovery Uncertain
- Concentrators	- Improved Efficiency at High Concentration Ratios - Potential for Reduced Array Size	- Still in Development - Required Launch Volume Order of Magnitude Larger than Si or GaAs - Weight Greater - 2^{o} Pointing (100(Sun Concentration) - Reduced Packing Factor
Thin Cells, Large Cells, Multi-Band Gap Cells		- Still in Early Development Stages

Baseline Selection: Silicon Planar Array

Silicon has the advantage of much flight history, proven cell laydown technology, lower cost per cell (at this time), and an abundance of available material. The concentrator technology is being developed with concentration ratios of 6 to 100 suns. The advantage of these is that less solar cell material is required with some improvement in efficiency demonstrated at the higher concentration ratios. However, the pointing requirements for a concentrator is more stringent (0^{o} to 2^{o} required at 100:1 concentration), more difficult array assembly and alignment is required, each cell requires assembly with its concentrator, and presently the effect of partial to full shadows on a cell is presently unknown.

Other potential cells include thin cells, large area cells, multiband gap cells, and the plasmon cell. These are all in various stages of research and development and therefore can not be depended upon for successful development by the first launch of the space station. It may be possible that some of these could be fully developed in time for addition to the station as an evolutionary step but this is presently unknown. GaAs has the best potential for being part of a station growth step and possibly its initial launch.

Based on all of the above, the benefits offered by the Silicon array i.e., flight history, known technology, and compatible availability schedules, Silicon selected as the preliminary space station baseline. The development of the concentrator technology, GaAs cell development improvement, and the possible nuclear development are retained as options at this time. This baseline will result in a solar array size of 17024 ft^2 in the final space station configuration. This would result in each array being approximately 100 ft long by 85 ft wide. The planar array is selected over the concentrator primarily for the greatly reduced launch volume.

Another major trade was to select the energy storage device once the decision was made to employ a solar array. Critical to this trade is the selection of the transmission voltage. The preliminary selection is the 120 Vdc + 165 Vdc bus over the higher 270 $\pm$ 15% level. This selection was based on keeping the array voltage below 400 Vdc to minimize possible interaction with the plasma in space, and to permit maximum utilization of development work previously accomplished in this voltage range in the programmable power processor and space platform efforts.

At this voltage, regenerative fuel cells become a more viable option. This is due to the fact that the cell size presently used on the shuttle is as small as can be used without detrimentally affecting current density and fuel cell efficiency. With the cell size a 135 Vdc fuel cell in basically a 30 kW unit and at 270 Vdc this becomes a 60 kW unit. The 60 kW is too large an increment to employ on a practical basis since power system modularity would be limited to one or two power modules. This becomes difficult to use in a growth step and becomes an inefficient application. The 30 kW unit is a more appropriate step even with operation at a 50% level operating point. The use of four such units was traded against the NiH2 and NiCd battery as the energy storage device.

The development status of the fuel cell and the NiH2 battery technologies are roughly equivalent with the NiH2 rate of development somewhat greater than the fuel cell. The reason for this may be that it is easier for many contractors to experiment with an NiH2 cell due to its size and availability rather than with the larger and more complex fuel cell. The NiCd battery has much flight history and has known depth-of-discharge (DOD) characteristics, abuse limitation; and failure modes. The NiH2 battery already has demonstrated superior DOD, abuse and failure characteristics over the NiCd battery and appears extremely promising as a near and far term energy storage device. The regenerative fuel cell system is more complicated and does not have as much data compiled against its operating and failure modes especially in space cycle operating modes. In a comparison between the NiH2 and the fuel cell, given that four 30 kW units would be used for a completed space station the fuel cell required much less volume, slightly more weight, approximately 15% more solar array due to its lower efficiency, and a greater cost. It offers the benefits of generating water, oxygen and hydrogen if required. The NiH2 battery is simpler, more efficient, is progressing at a faster rate, and requires less solar array and cost.

Based on the above, the NiH2 battery has been selected as the preliminary energy storage device in the MMC baseline. The regenerative fuel cell is retained as an option to be evaluated again in the future. See Table 6.1.2-3.

The other listed trades were evaluated to a lesser level of detail since their effect on the station architecture was either minimal or non existent. However, it was felt they should be addressed since several are technology issues. To summarize the results,

a. A peak power tracking approach has been selected to take advantage of all the available power on the array. A direct-energy-transfer scheme is not able to take advantage of the full beginning-of-life power from the array. This could amount to between 10% and 20% more power in the early array lifetime and in each phase of evolutionary growth.

b. The transmission voltage preliminary selection is the 120 Vdc $\pm$ 165 Vdc for the reasons stated earlier.

c. The preliminary equipment selection is covered later on in this section.

d. The growth increments selected are seen in the evolution section, 6.1.3. The three steps are compatible with expected solar array technology and add-on capability.

e. The automation level has been defined in a preliminary fashion. Basically the EPS monitors itself, takes emergency corrective action to a point then informs and/or requests the main station computer to take more action if deemed necessary.

Table 6.1.2-3 Energy Storage Trade Summary

Item	Pro	Con
NiCd Batteries	- Much Flight History - Known Technology	- Low DOD for Long Life - Weight
NiH_2 Batteries	- Greater DOD Possible over NiCd - Much Development Activity - Absorbs Abuse Well	- Greater Volume
Regenerative Fuel Cell	- Reduced Equipment Volume - Can Generate Water or Oxygen - Some Development Activity	- Lower Efficiency Over 8 Batteries - More Solar Array Required - More Complex than Batteries - Higher Cost - Greater Heat Rejection Requirement
Momentum Wheels		- Needs Development
New Battery		- Needs Development

Baseline Selection: NiH_2 Batteries

f. The power transfer scheme employs slip rings. This prevents rewinding the large solar array each orbit and permits a known one-direction continuous torque on the station rather than cycling positive and negative torques.

g. The tentative power management scheme is to employ the ground computer. Doing this task can be added to the EPS computer at a later date if it is deemed cost effective and efficient to do on-board.

h. Two degrees of freedom have been selected for the array to successfully point at the sun from the gravity gradient flying station.

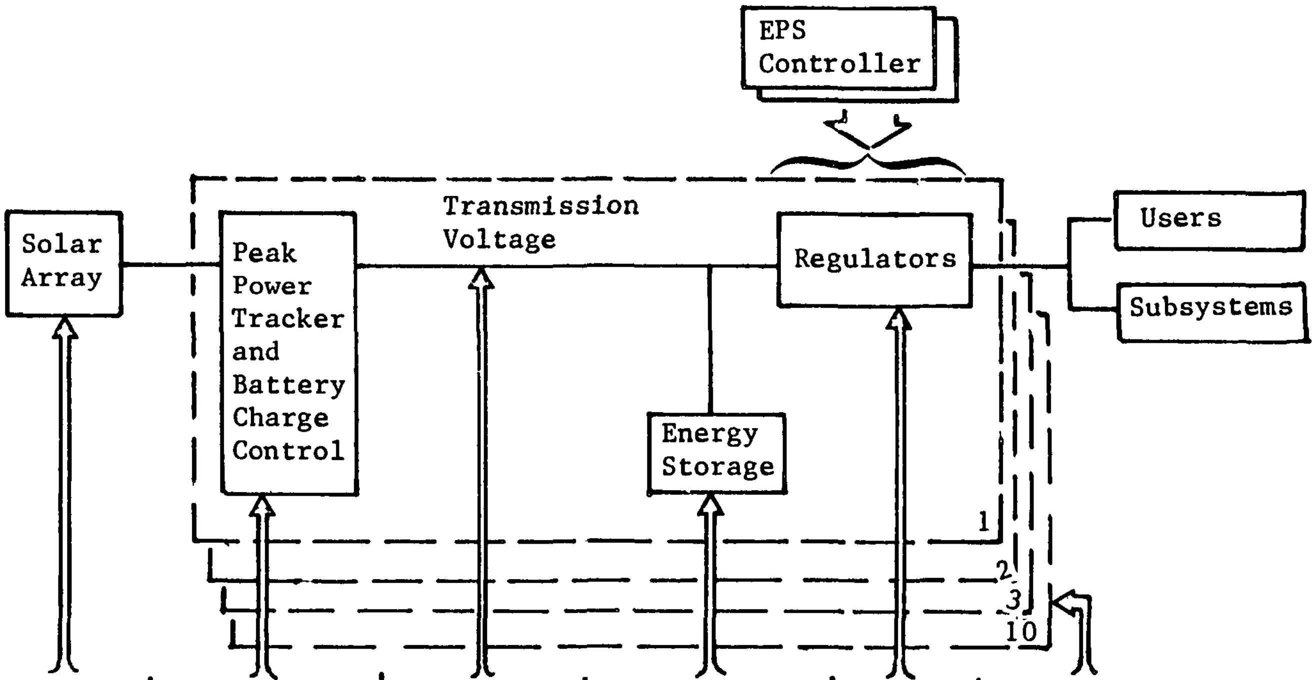
i. The completed station modularity level is 10 EPS modular units. This is based on the desired battery depth of discharge for long life and the redundancy requirements from the initial to the completed station.

The EPS conceptual approach can be seen in Figure 6.1.2-1. This reflects the baseline selection with the viable options also listed. This diagram also shows the redundant EPS controllers that monitor and exercise limited control over the subsystem. The peak power trackers and regulators are of the P^3 type. It is likely that some modifications to this unit, which is in a high state of development, is likely should higher array voltages become feasible and higher power components become developed. The modularity level of the EPS comprises a peak power tracker/battery charger, the battery and the output regulator(s).

Table 6.1.2-4 shows a preliminary equipment list, quantities, weights, and volumes. This provides an approximate measure of what would be required for the EPS equipment. This is provided to enable the appropriate sizing of the power section of the station. While the actual equipment used will be unknown for some time, the equipment listed in representative of what is reasonable to expect.

6.1.3 Evolution

The graphical presentation of the time phased power requirements of Figure 6.1.1-1 are repeated in Figure 6.1.3-1 with the recommended growth steps. For the station to incrementally grow in capability the recommended steps are seen as dashed lines in this figure. The first step would be sufficient for two years, the second for three years, and the third for the remaining five years. The steady power requirement level after 1995 implies that further space station growth would not occur at the manned station but by the use of platforms or possibly another manned station.



Baseline	Silicon Planar	P^3 Type	120-165 VDC	NiH2 Batteries	P^3 Type	10 Power Modules
Options Still Being Considered	- GaAs - Concentrators - Nuclear	- P3 - Solar Array Switching	- 270 + 15% VDC	- Regenerative Fuel Cells	- P3	

Figure 6.1.2-1 Power System Approach

Table 6.1.2-4 Preliminary Equipment List

Item	Qty	Total Weight (lbs)	Total Volume (ft^3)	Remarks
Initial Station				
- Solar Array Wings	2	3012	N/A	Each Wing-3216 ft^2 Deployed -37.9 kW BOL Each
- Batteries	5 (110 Cells ea)	3819	80	NiH2 Baselined
- Pwr Conditioning	5	310	10.4	P^3 Derivative
- Regulators	5	310	10.4	P^3 Derivative
- Distributors	1	50	2.78	Assume $\frac{20Lx20Wx12h}{(12)^3} = 2.78\ ft^3$
- Cables	1 set	∿550	N/A	
Growth Step Configuration				
- Solar Array Wings	2	5624	N/A	Each Wing-5964 ft^2 Deployed -68.4 kW BOL Each
- Batteries	8 (110 Cells ea)	6111	128	NiH2 Baselined
- Pwr Conditioning	8	496	16.6	P^3 Derivative
- Regulators	10	620	20.8	P^3 Derivative
- Distributors	3	150	8.34	
- Cables	1 set	∿1150	N/A	
Final Station Equipment				
- Solar Array Wings	2	8024	N/A	Each Wing-8512 ft^2 Deployed -93.4 kW BOL Each
- Batteries	10 (110 Cells ea)	7639	160	NiH2 Baselined
- Pwr Conditioning	10	620	20.8	P^3 Derivative
- Regulators	14	868	29.1	P^3 Derivative
- Distributors	3	150	8.34	
- Cables	1 set	∿1540	N/A	

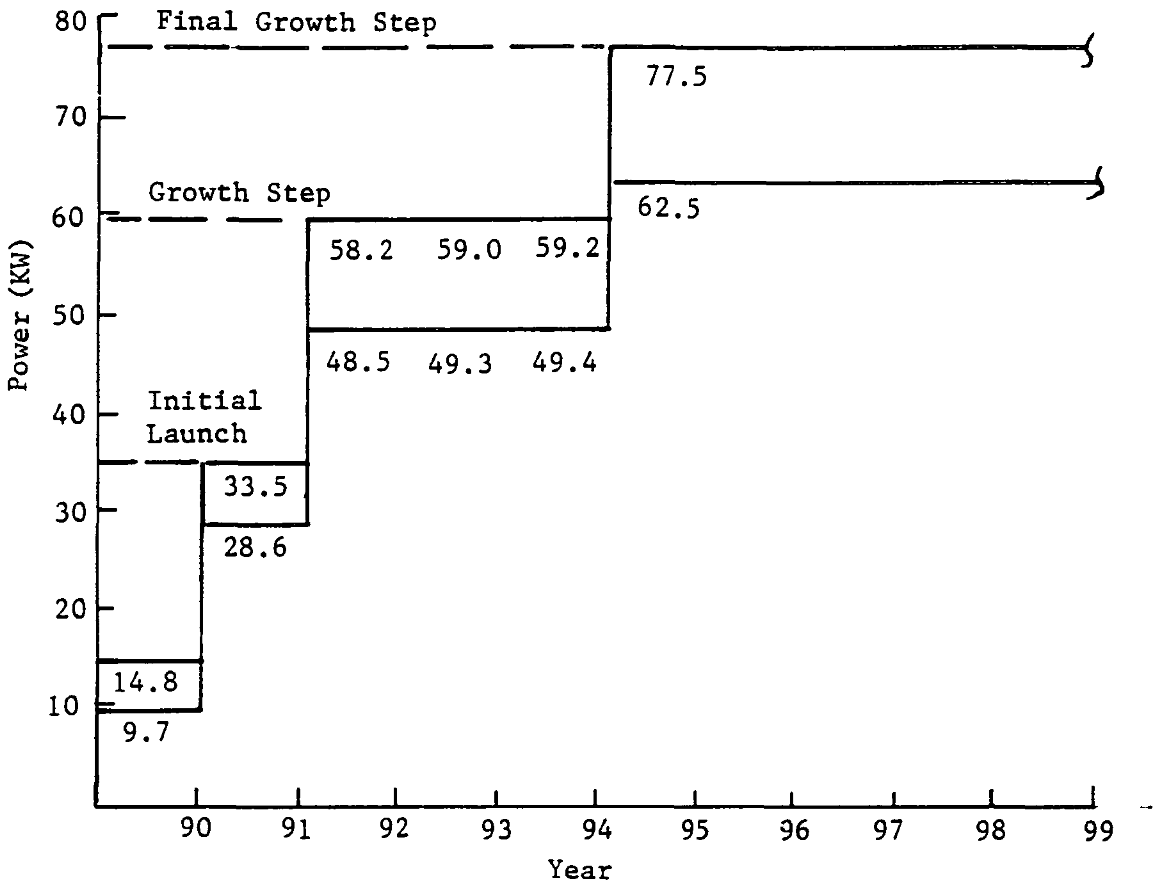


Figure 6.1.3-1 EPS Growth Increments - Manned Station

The performance of the EPS as it grows can be seen in a preliminary fashion in Figure 6.1.3-2, -3, and -4. These are the initial space station, the 1st growth step, and the final growth step (i.e., the completed station) respectively. This level of detail was performed to arrive at the probable array size required. Each of these diagrams show the split between user loads, and subsystem loads, the modularity of the EPS, the anticipated depth-of-discharge on the batteries and the solar array beginning-of-file (BOL) and end-of-file (EOL) values. The manner in which the array can be increased in power is shown in Figure 6.1.3-5. This would also permit the incorporation of advanced cell technology as well. The array blankets added in growth steps could contain the advanced cells and fewer panels or some panels with no cells if the initial length is desired; or simple tensioning devices could be used with no extra panels to reduce the overall drag.

The first array launched would require an area of 6431 ft^2. This entails two wings, each wing being approximately 100 ft long by 32 ft wide. The second growth step would place additional blankets on the arrays in an evenly distributed manner to prevent the generation of uneven drag torques on the station. To allow the array mast to be evenly loaded, tensioning devices instead of blankets can be placed on the other side of the mast. The second increment would add 5497 ft^2 to the array. The third step adds a final increment of 5096 ft^2 to the array. The power conditioning equipment associated with the EPS growth can be increased in three steps to follow array growth. There are other methods of evolving uniformly. This method appeared the most feasible while retaining a symmetrical growth pattern.

The modular conditioning and energy storage section growth increments can be installed as a unit minimizing crew black box installation and connector hook ups.

6.1.4 Technical Issues and Concerns

The technical issues using the baseline approach are summarized in Table 6.1.4-1. The interaction between the solar array voltage level and space plasma must be understood to assure that the selected array design does not degrade or become damaged due to this phenomena. Solar Array Flight Experiment (SAFE) II, if approved will provide data on this subject. The degree of automation that should be and/or can be implemented in the EPS should be assessed further. Many technology advancements in power are in a state of development these must be reviewed periodically to be aware of which can be successfully developed and qualified at a reasonable cost for the space station. Also, those than can be implemented as growth steps must be assessed for the best method of incorporation into a station that is in orbit.

A phenomena not totally understood but that can radically effect the present approach to lightweight solar arrays is the oxygenation (i.e., atomic oxygen interaction) of materials in space, specifically its effect on kapton. The phenomena entails a significant weight loss of material and a possible change in the material characteristics. This could effect absorbtivity, emissivity, reflectance, brittleness, etc. All of these could detrimentally effect a large lightweight array employing kapton substrates. This phenomena must be understood as it applies to solar array related materials.

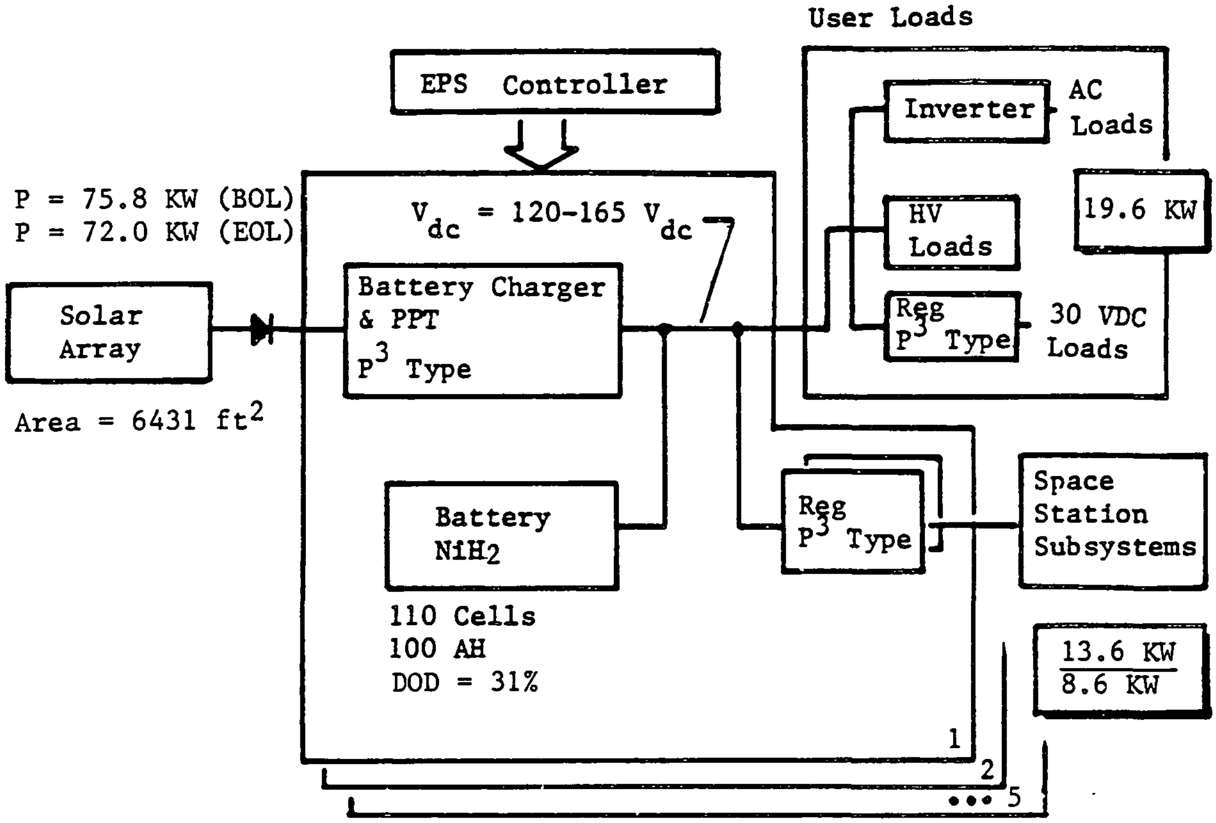


Figure 6.1.3-2 Initial EPS Performance

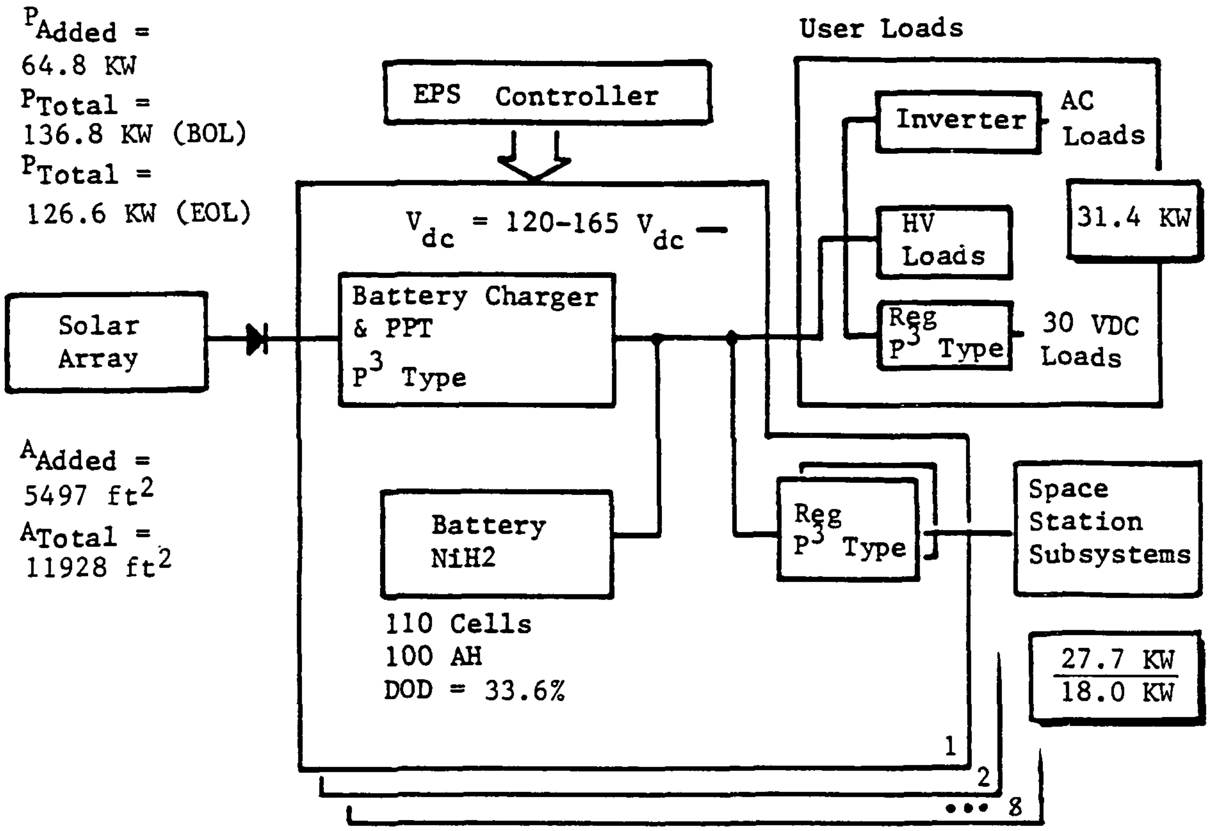


Figure 6.1.3-3 First EPS Growth Step

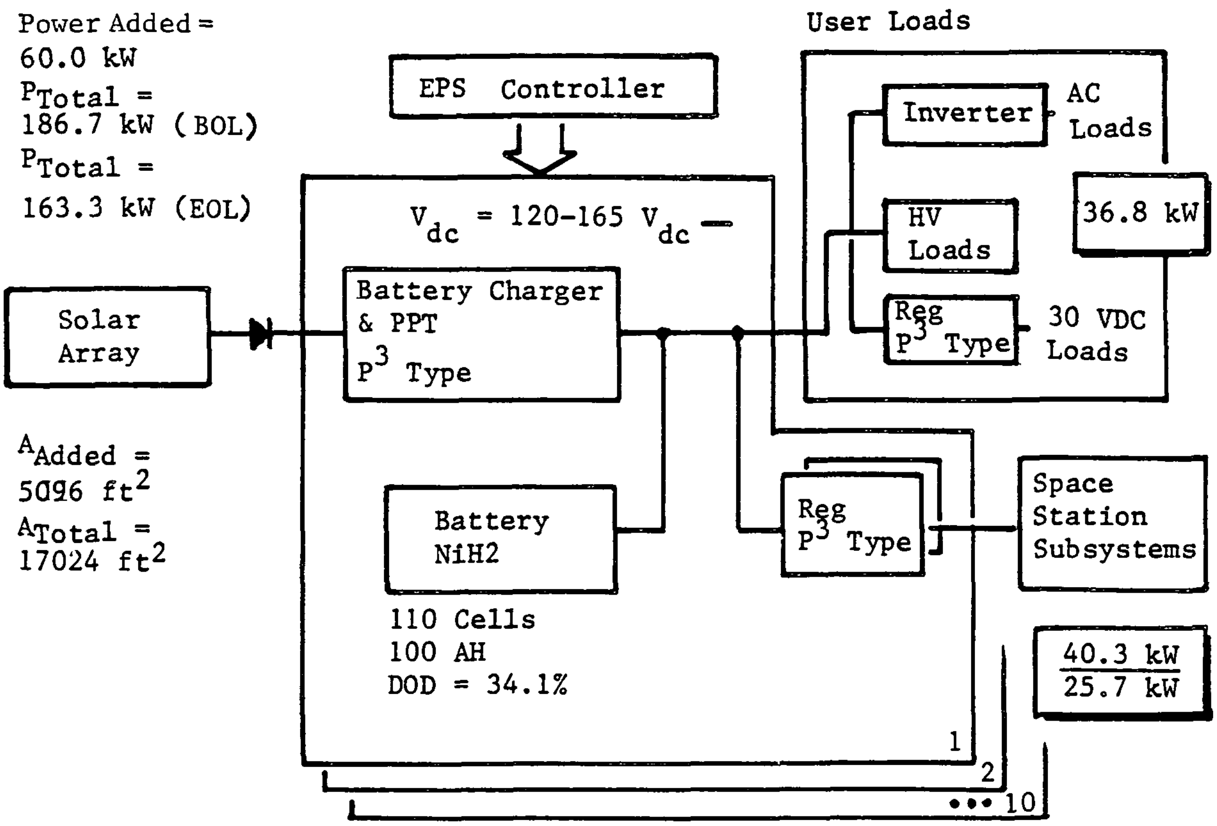


Figure 6.1.3-4 Final Growth Step Performance

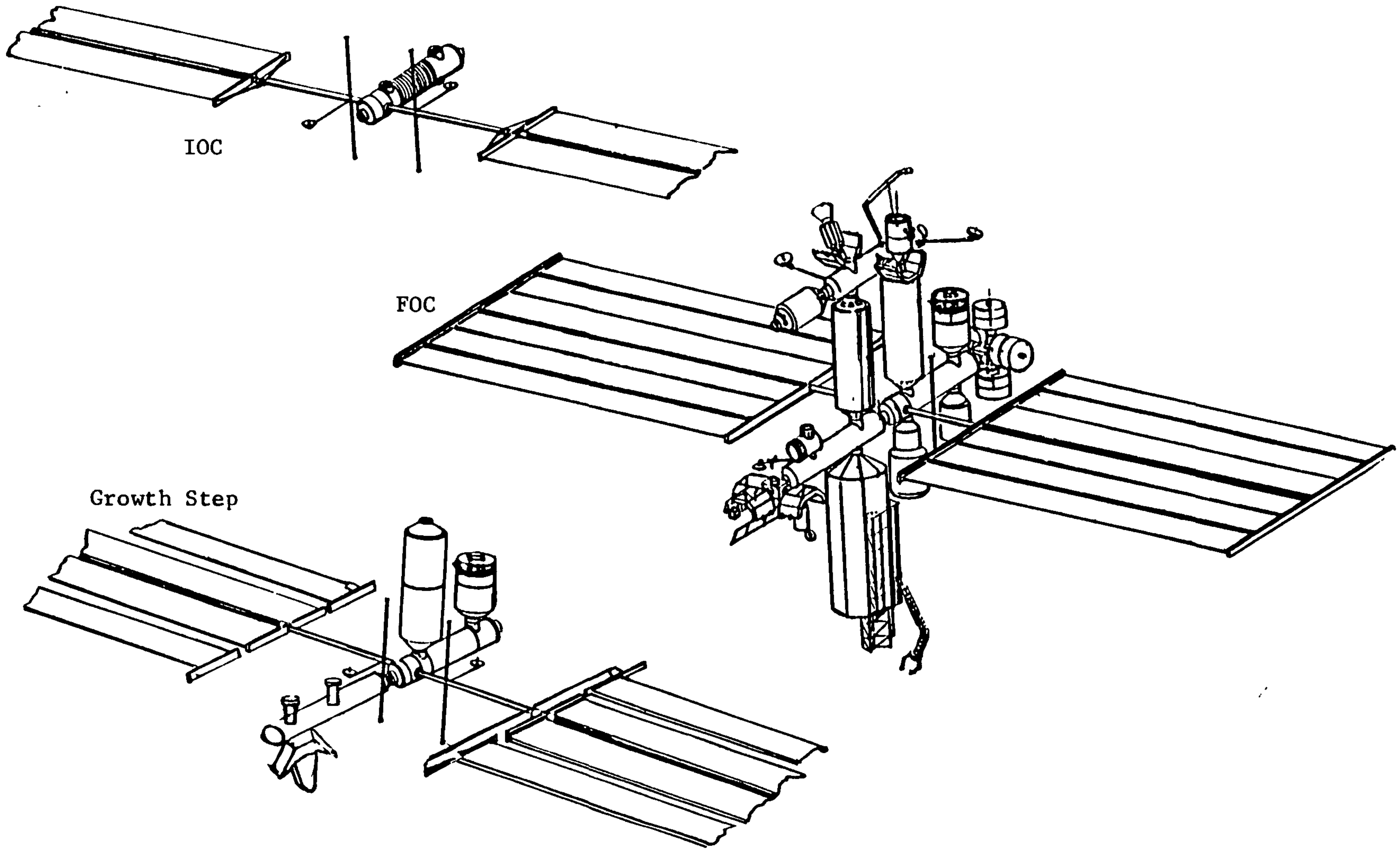


Figure 6.1.3-5 Solar Array Growth

Table 6.1.4-1 Technical Issues and Concerns

- Plasma Effects
- Degree of Autonomy/Automation
- Technology Advancements Forthcoming
  - GaAs
  - Concentrators
  - Thin Cells
  - Multiband Gap Cells
  - NiH_2 Batteries
  - Regenerative Fuel Cells
  - Nuclear Power
  - Light Weight Solar Array Concepts
  - Power Components
- Incorporation of Advancements Throughout Station Life
- Possible Oxygenation of Critical Materials
- High Voltage/Power Component Availability
- Shadowing Considerations with Gravity Gradient Mode
- Solar Array Dynamics
- Transmission Voltage Level

Should a nuclear electrical power source be selected, its appearance in a system would likely take the form of that shown in Figure 6.1.4-1. It would not be the only power source on board since it would be a single failure point. Therefore, some sort of long term power source must be provided for emergency modes that would permit operating at a minimum station power level for as long as 90 days. A fuel cell could be employed especially if there is an availability of oxygen and hydrogen fuels. A solar array/battery combination could also perform in this manner. But the employing of a nuclear device entails technical issues and concerns separate from those listed in the baseline system. These are shown in Table 6.1.4-2. Aside from development schedules for not only the reactor type selected and the supporting elements (i.e., power conversion device, heat rejection method, shield, and control system) issues include assuring world and crew safety, deciding and developing the method of a reliable disposal means. Also, mean time between failures (MTBF) must be assessed and how and/or what can be maintained on a nuclear device should a failure occur. All of these items must be successfully addressed before a nuclear device can become a viable power source candidate.

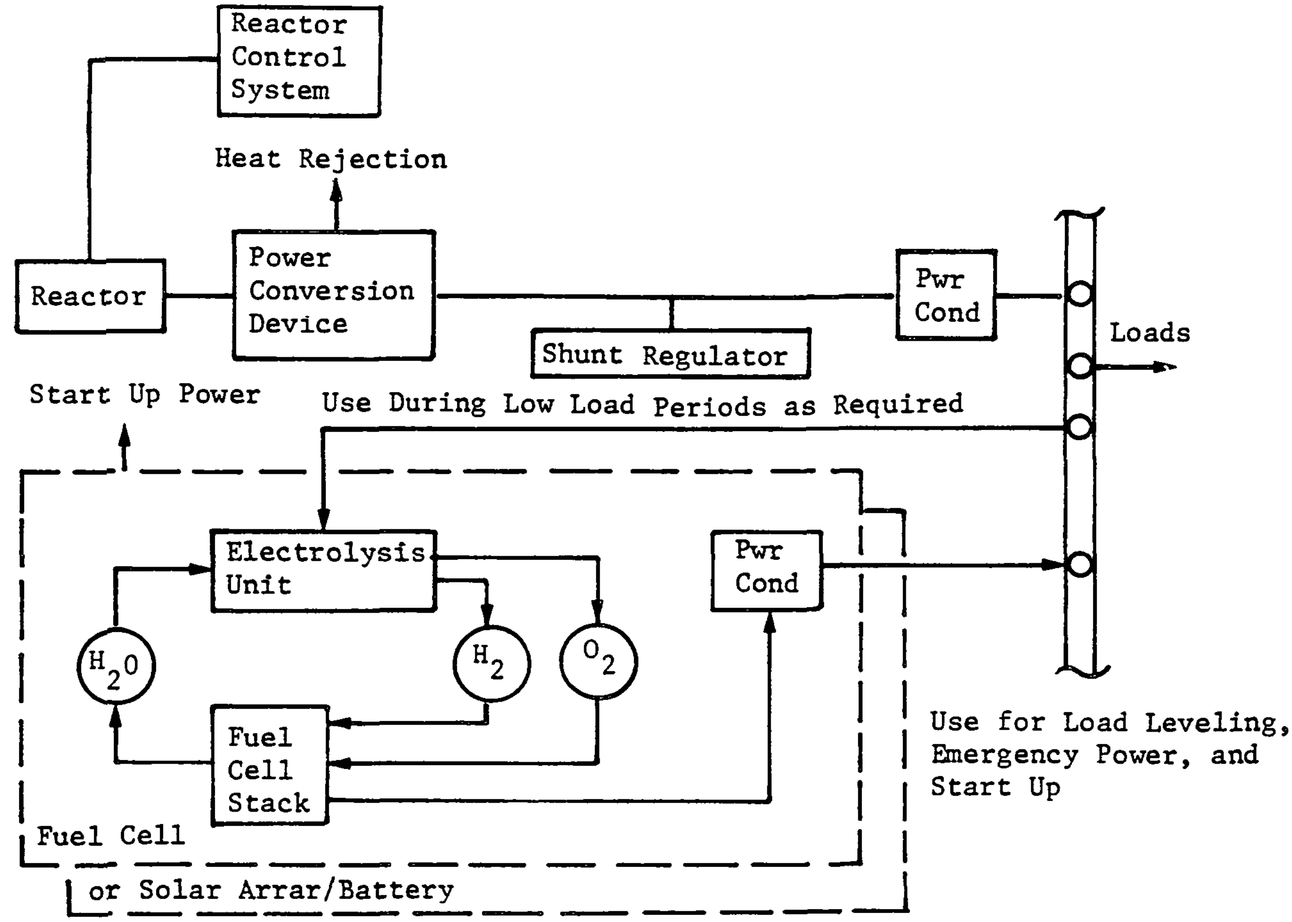


Figure 6.1.4-1 Potential Nuclear Electrical Power System

Table 6.1.4-2 Issues and Concerns—Nuclear Power Source

- Reactor Development Schedule
- Safety
- Disposal
- Supporting Technology Development Schedules
  - Power Conversion Device
  - Heat Rejection Method
  - Shield
  - Reactor Control/Start Up/Shutdown
- Mean Time Between Failures
- Maintainability
- Shadow or 4 π Shielding
- Implementation Scenarios Using STS
- Flight Unit Delivery Schedules
- Cost

6.2 THERMAL CONTROL

This section involves the evaluation of options for the various functions of heat-transport and heat rejection to arrive at a baseline concept. A preliminary determination of initial, interim and ultimate capability is measured against time-phased user requirements. The following presents the requirements, options and baseline selection, evolutionary capability and some recommendations for future technological advancements.

6.2.1 Requirements

The function of the thermal control system is to collect, transport, distribute and reject Space Station heat such that crew and equipment are maintained within required temperature limitations. A 10 year system life is the design goal with indefinite life achieved through orbital replacement and maintenance. Systems level requirements are given in Table 6.2.1-1 and derived requirements (heat loads, temperature limits) are addressed in the subsequent section 6.2.4.

6.2.2 Trade Studies

Thermal Control Coating - Foremost among the long-life thermal design problems associated with the thermal radiation mechanism for energy rejection is the deterioration of thermal control coating performance resulting from contamination, UV degradation, etc. This degradation specifically results in the increasing of solar absorptance (α_s) values. Allowances for degradation of optical properties of the radiator surface frequently result in radiators that are too large. This problem can be overcome in one of two ways: (1) design a thermal control system that is relatively insensitive to the coating degradation, and (2) use a thermal control coating that will withstand long-term exposure in space without significant degradation of performance, or, that can be easily replaced.

A surface coating with a low α/ε ratio that would not significantly degrade on exposure to UV and various contaminants would significantly simplify the thermal control system by:

a) the use of less radiator area
b) reducing the frequency of refurbishment/replacement of radiator panels
c) eliminating the need for solar avoidance locations for radiator panels

Unfortunately no single coating currently exists with all the desirable features of low α_s / ε, optically stable, low susceptibility to contamination, low outgassing and electrically conducting. The optical solar reflector (OSR) has a higher probability of meeting these requirements than any other coating however, it is not attractive for large surface areas because of the major drawbacks of cost, difficulty of application, and weight.

Table 6.2.1-1 TCS System Level Requirements

TIMING & GROWTH

- Baseline concepts for 1986 technology readiness for early 1990 missions.
- Accommodate stepwise growth in heat rejection load.

LIFE, MAINTENANCE, RELIABILITY

- 10 Year life design goal.
- Indefinite life with orbital replacement and maintenance.
- Redundancy and micrometeoroid (debris) protection to achieve survivability.
- Replaceability of major subsystem elements (e.g. pumps).
- Fault detection and isolation.

ENVIRONMENTS

•Orbit Altitude:	250 n.m.
•Orbit Inclination:	28.5°
•Orbit β Angle:	0° to 52°
•Thermal:	Solar, Earth, Albedo, Module/External Elements Interaction

SAFETY

- Non-toxic and non-flammable coolants in pressurized areas.
- No contact temperatures above 105°F.

INTERFACES

- Minimum obstruction to Scientific Viewing Payloads.
- Minimum aerodynamic drag.
- Avoid unwanted moments due to unfavorably placed deployed masses.
- Avoid physical interference with gimballed solar arrays or payloads.
- Minimize payload contamination threat due to fluid leakage.

The most promising currently available coatings are Silverized Teflon and Zinc Orthotitanate (ZOT). After 5 years in low Earth orbit both degrade to an α_s/ε of approximately 0.23. On the basis that the Silverized Teflon coating can be more easily replaced this is the preferred coating selection pending development of a new coating.

Radiator Location - In general the most attractive location for the radiator from the purely thermal standpoint would be where the radiator panel is oriented edge to the sun and such that both sides of the panel are active. Such a radiator might typically be represented in total by a planar panel 80 ft x 40 ft for a 100 KW space station. Unfortunately even when subdivided a radiator of this size, mounted probably on a boom, may not be physically compatible with many S.S. operations such as docking, remote maneuvering of arms etc. Additional unwanted moments and aerodynamic drag are incurred as well as possible interference with payload viewing and installation. This type of radiator also requires the development of a 4-pass rotary fluid joint, or long-life flex-hoses, depending on the approach to maintaining an edge-to-sun orientation. Thus body mounted radiators, integral with the micro-meteoroid bumper, should be incorporated to reduce the size of the deployed radiator area. An important factor involved in this recommendation is that since in delivering the space station elements the Shuttle is volume limited (Ref. Section 4.0), the weight disadvantages of the body mounted radiator is not a consideration. The principal design problem incurred by body mounted radiators is that they are subject to and sensitive to solar degradation of the thermal control coating. Thus an optically stable thermal control coating development would more significantly benefit the radiator sizing and maintenance frequency problems of body mounted radiators than deployed radiators. Table 6.2.2-1 summarizes the considerations addressed in selecting the radiator location.

Radiator Panel - Essentially two candidate radiator panel approaches are available for consideration, (1) the conventional pumped liquid loop radiator, and (2) the heat pipe radiator. A third type, the liquid droplet radiator offers the promise of low weight and is impervious to the micro-meteoroid environment. It is, however, only in the conceptual design stage and requires significantly more development before it can become a meaningful candidate.

To achieve a maximum probability of success in a micro-meteoroid environment with the conventional radiator, redundant liquid loops are integral with each panel and separately manifolded. Each liquid loop is capable of rejecting the full heat load and thus the redundant loop is a standby or back up loop. The flow tubes are shielded from micro-meteoroid penetration. A significant disadvantage is that a puncture or leak in a flow tube results in the loss of the complete loop. Numerous sub-loops as shown in Fig. 6.2.2-1 can increase the reliability but this increases the complexity.

A new and advanced heat pipe radiator is currently under prototype development by NASA that is less susceptible to the micro-meteoroid environment. Each heat pipe is independent of the others so that a puncture of a heat pipe results only in that fractional loss of performance associated with the damaged heat pipe. A feature of this approach is that a damaged panel (one for each heat pipe) can be

Table 6.2.2-1 Thermal Radiator Location Considerations

Consideration	Deployed Radiator	Body Mounted Radiator
•Heat Rejection Surface Area Limits	No Specific Bounds	Bounded by Module Surface Area
•Stepwise Cluster Heat Load Growth	Expandable Radiator	Grows with Cluster Additions
•Emergency Thermal Management-Safe Refuge	Central System	Autonomous Modules
•Unwanted Moments	Impacts Control System	No Control System Impact
•Aerodynamic Drag	Impacts Control System	No Control System Impact
•Instrument and Payload View Blockage	Increased Blockage	Minimizes Blockage
•Heat Transport Across Moving Joint	Fluid Swivel Joint Development	No Moving Joints
•Orientation Sensitivity for Heat Rejection	Optimum, Edge on to Sun	Solar Side Less Effective
•Thermal Control Coating Solar Degradation	Not Exposed to Direct Radiation	Liable to and Sensitive to Degradation

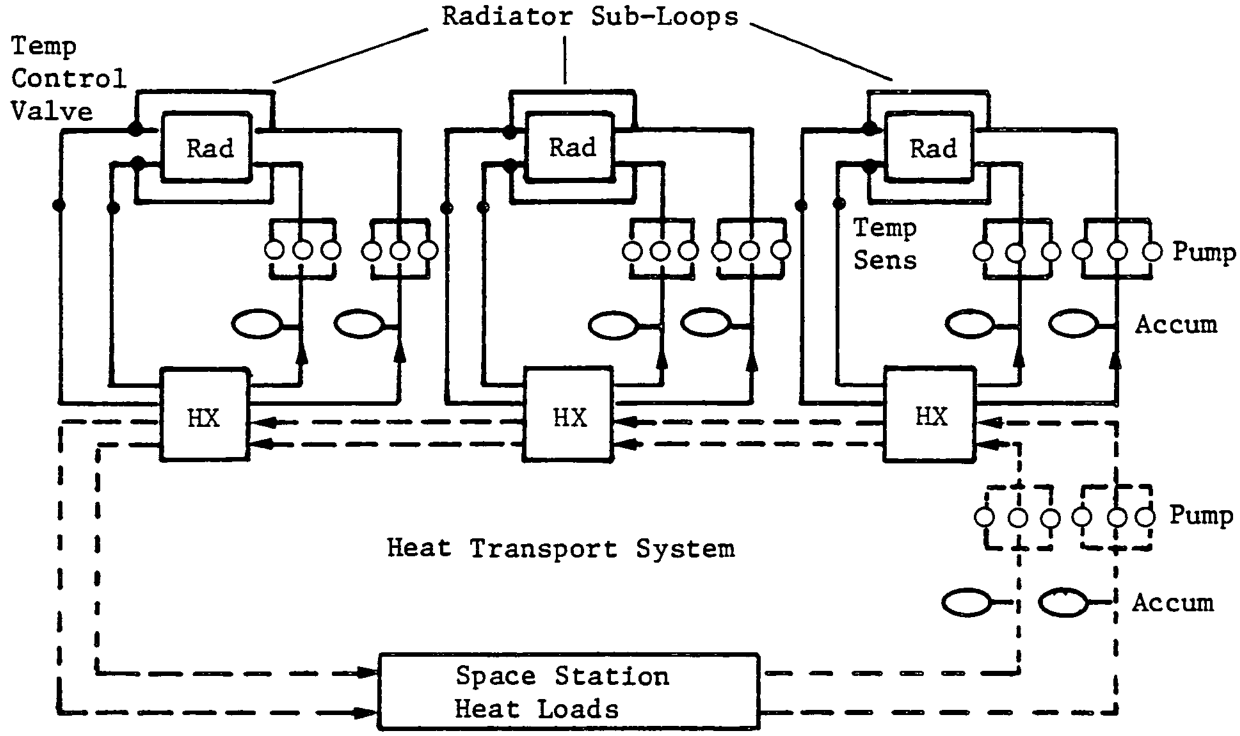


Figure 6.2.2-1 Multiple Radiator Sub-Loops

removed and replaced without breaking fluid connectors in the interfacing equipment heat transport loop. This thermal interface comprises a contact heat exchanger that the heat pipe is "plugged into". A loose fit for the heat pipe is provided initially. A clamping action is then activated to give the contact force for good heat transfer. The panel removal feature not only permits damaged heat pipes to be replaced but also degraded thermal control coatings can be replaced in a suitable work environment.

In view of the superior survivability characteristics, and panel removal developments in progress, heat pipe radiators are the recommended approach for the S.S.

Heat Transport Loop - For manned spacecraft the selection of a heat acquisition and transport approach can be made principally from pumped single phase and two phase fluid systems. Single phase systems have performed with excellent reliability in manned spacecraft for the past 20 years, such as in Gemini, Apollo, Skylab and Shuttle. Two phase systems are under study for the S.S. by NASA (Ref. contract NAS9-16781) because of the long physical distances involved in transporting large quantities of heat and the diverse interface requirements. Some of the systems such as the osmotic heat pipe and capillary heat pipe offer the potential of long life and require no power because of their passive nature. While possessing many attractive features these two passive concepts result in a larger lead time and development risk than the other candidates. It is felt that laboratory study should continue to establish their characteristics and limitations but that their lack of technological maturity eliminates them from further consideration for initial space station application.

A concept that offers the most promise for near term two-phase system development is the mechanical pump augmented heat pipe. It has good performance characteristics, requires the least development, and requires little power. Coolant circulation is by a pump located in the liquid portion of the loop. As with all two-phase approaches it operates at constant temperature over the entire length of the loop. This is because heat is transferred by evaporation and condensation rather than by the sensible heat changes of a single phase conventional coolant loop. However, because all heat is rejected at the minimum system temperature the radiator area becomes very much larger than that for a conventional coolant loop. Examples of low operating temperature ($\sim$40°F) equipment are electrophoresis experiments, humidity control heat exchangers, and Ni - H_2 batteries (desirable). On the other hand fuel cells and furnace processes operate near 200°F or higher. In supporting such multi-disciplinary requirements, considerable reduction in radiator area is achieved by splitting the system into two or more discrete temperature level loops. Fig. 6.2.2-2 shows a schematic of a dual temperature radiator for this approach. A pre-charged accumulator (pressurized with gaseous nitrogen) located just upstream of each pump sets the saturation pressure of each loop and therefore the desired operating temperature level. The radiators condense and slightly sub cool the coolant from the vapor/liquid state it attains after collecting the waste heat loads.

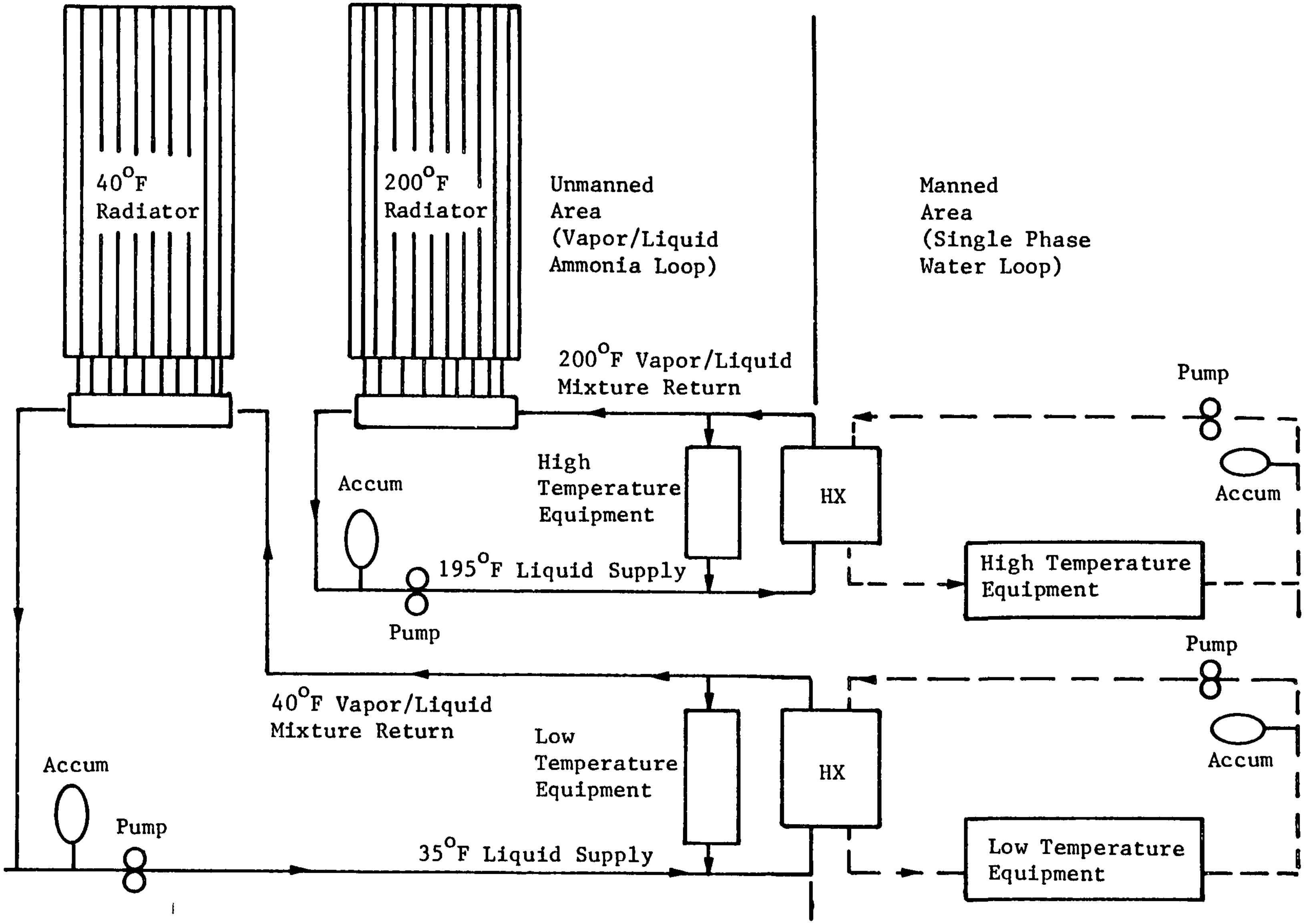


Figure 6.2.2-2 Pump Augmented Heat Pipe—Split Radiator Concept

Two phase flow systems present many more complex problems to the designer than are involved in liquid loop design. Some of the factors that contribute to the difficulties in analyzing two phase flow are:

1) Several types of flow patterns (regions) can be formed, depending on the vapor-liquid ratio (heat load), making pressure drop and heat transfer rate predictions complex.

2) Maldistributions of liquid and vapor in heat exchangers and cold plates is difficult to overcome and results in thermal performance degradation. Typically the 2-phase mixture is separated and then recombined at entry to the heat exchanger in order to control the mixing and distribution.

3) The presence of non-condensable gases in the vapors can appreciably decrease the condensing film heat transfer coefficients.

4) The mechanisms of evaporating and condensing heat transfer are not well understood for zero-g and thus will probably require considerable experimentation in space.

5) System pressure drops cause the liquid to boil or flash, thereby generating additional vapor, increasing the mixture velocity and reducing temperature.

6) The vapor-liquid ratios, and consequently pressure drop, will vary principally because of heat load fluctuations. Thus the pump will be working against changing system resistances and therefore the design limits of the pump must accommodate this required flexibility.

Ammonia has been shown to be the optimum two-phase fluid. However it is not acceptable inside the manned areas. Therefore this concept will require a separate water loop in the manned areas, as is needed with conventional pumped freon loops.

Because many more problems are inherent in the design of two phase systems, that do not occur in single phase systems, and from the standpoint of technology readiness, it is recommended that the conventional pumped (single phase) liquid system be selected for the initial S.S. modules. It is likely that at least the subsystems in the early S.S modules would not benefit substantially from the equipment interface versatility, constant temperature, etc features offered by a two phase system. Furthermore, it appears that the only available selection in the near future for manned areas will be the conventional pumped water loop.

6.2.3 Baseline Conceptual Design

From the previous trade studies a thermal control system has been defined. Radiator sizing consistent with the phased EPS requirements of section 6.1 is addressed in section 6.2.4 on Evolution. Heat pipe space radiators, integral with the S.S. micro-meteoroid bumper, are employed where possible and any additional requirement for radiator area is accommodated using a deployed radiator, i.e., mounting "plug-in" heat pipe panels on the solar array booms, see Fig. 6.2.3-1.

For the initial studies Silverized Teflon is selected as the thermal control coating and a degradation in excess of 100% to an α_s/ε of 0.21/.76 is assumed. This corresponds to a time span in the region of 5 years before such an α_s/ε value is reached. After this point the options are to replace the coating or expand the radiator area to compensate for the decrease in radiator performance. This latter approach is configuration dependent and is addressed in section 6.2.3-2 for each S.S option. For the purposes of heat pipe and coating replacement, the body mounted radiator panels are removable in segments. It is envisioned that an adaptation of the heat-pipe and contact heat exchaners under development for the deployed radiator can be utilized (NASA Contracts: NAS9-15965 and NAS9-16582).

Coolant loops flowing in pressurized areas use water as the transport fluid because of toxicity considerations. Heat rejection loops, interfacing with the heat pipe radiator panels, use Freon 21 as the transport fluid to avoid freeze-up problems. The water and freon loops interface via a heat exchanger similar to the Orbiter payload heat exchanger. Fig. 6.2.3-2 shows the schematics of the reference concepts for the three S.S. Configuration Options derived in Section 4.0. For the Energy Modules, utilized in both the Modular and Aft Cargo Carrier options, the coolant loop is Freon 21 because internal volume limitations result in the associated cold-plated equipment being located external to the pressure shell. TCS component sizing will be highly dependent on the S.S. configuration final selection and the heat load in each module. An examination of the coolant flowrates and heat transfer rates consistent with the three schematics in Figure 6.2.3-2, indicates that extensive use can be made of coolant loop components and technology developed for the Shuttle. Redundant flow loops are used; one active and one standby (or backup) to provide the needed reliability. Only one of the two redundant loops is shown in Fig. 6.2.3-2 for clarity. Each redundant system is capable of transporting the full heat load. Also, each of the dual loops has a redundant pump so that the possibility of neither of the pumps in the standby loop not starting if the primary loop fails is small. Thus, two pump failures are allowed before the standby loop is activated. Full capacity is still provided after three pump failures. The pumps are similar in type to the Sunstrand pumps used in the Orbiter and Spacelab which are virtually a wearout free design. The demonstrated life on test rigs is 3.2 years for the Orbiter freon pump and 4.2 years for the Orbiter water pump without failure. Figure 6.2.3-3 shows the performance of both Orbiter pumps.

Subsystem cold plates are similar to those utilized on Spacelab for which performance data is shown in Fig. 6.2.3-4. The number of cold plates required in each subsystem was ascertained to be determined by the equipment footprint area rather than by heat transfer considerations. A requirement for air cooling of equipment and experiments has not yet been established but this could be provided in a similar manner to the Spacelab air cooling provisions. Depressurizing of the area in which the equipment is housed will of course result in loss of cooling. Therefore the critical nature of the equipment must be evaluated before this option is incorporated.

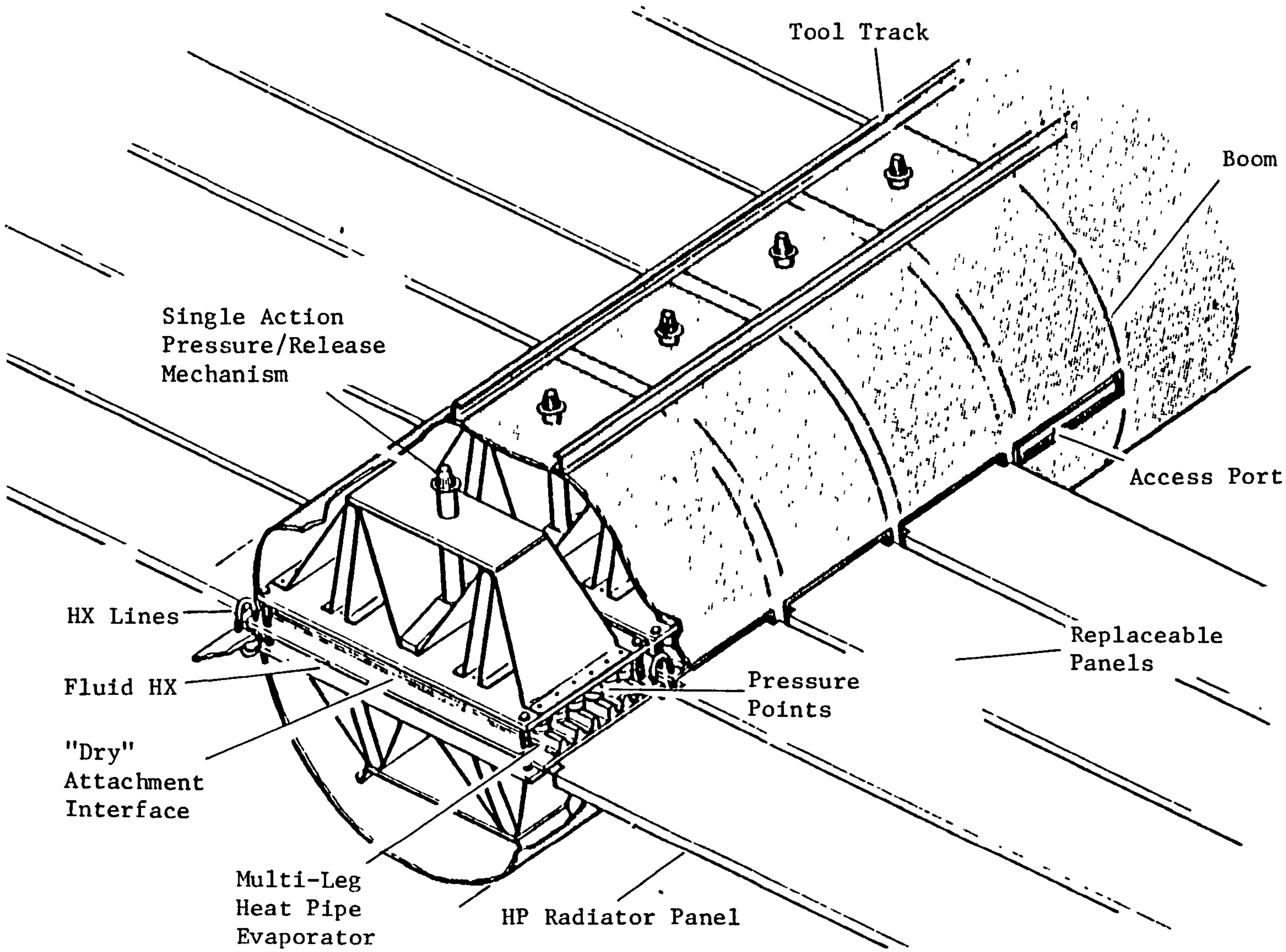
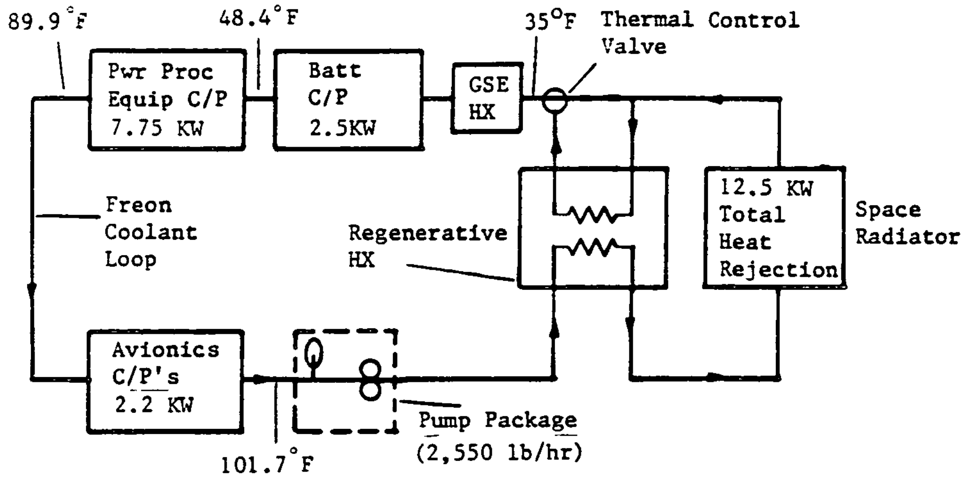
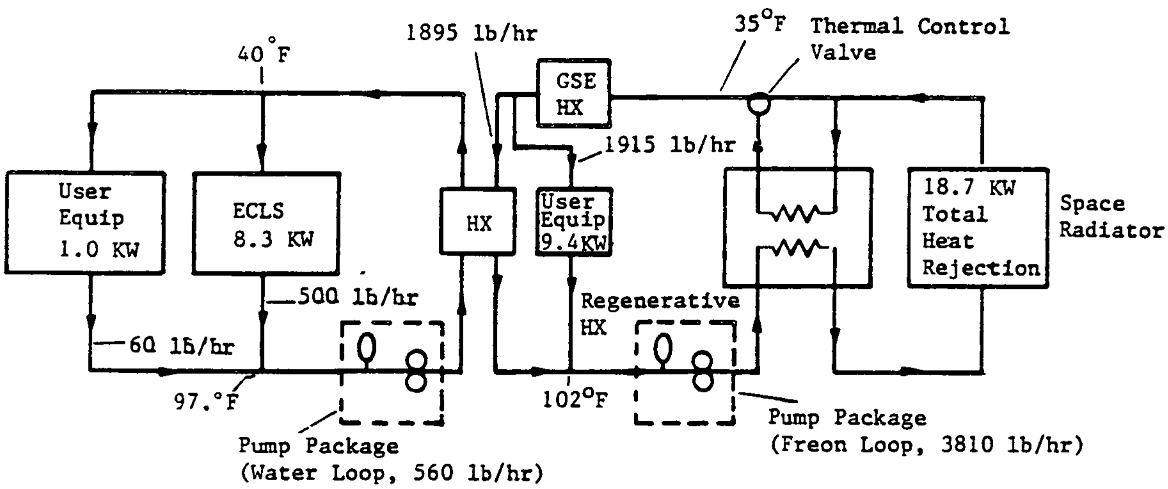


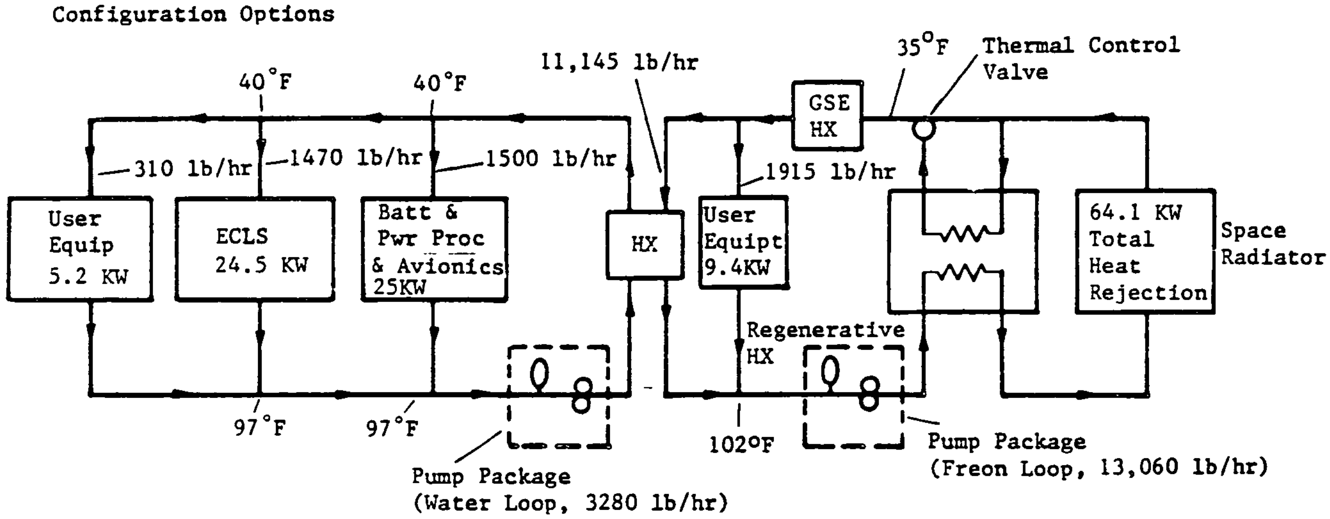
Figure 6.2.3-1 "Plug-In" Heat Pipe Radiator Panels



(a) Energy Module Thermal Control Reference Concept - Modular and ACC Configuration Options



(b) Habitat Module Thermal Control Reference Concept - Modular and ACC Configuration Options



(c) Shuttle Derived Vehicle Thermal Control Reference Concept

Figure 6.2.3-2 Thermal Control Reference Concept Schematics

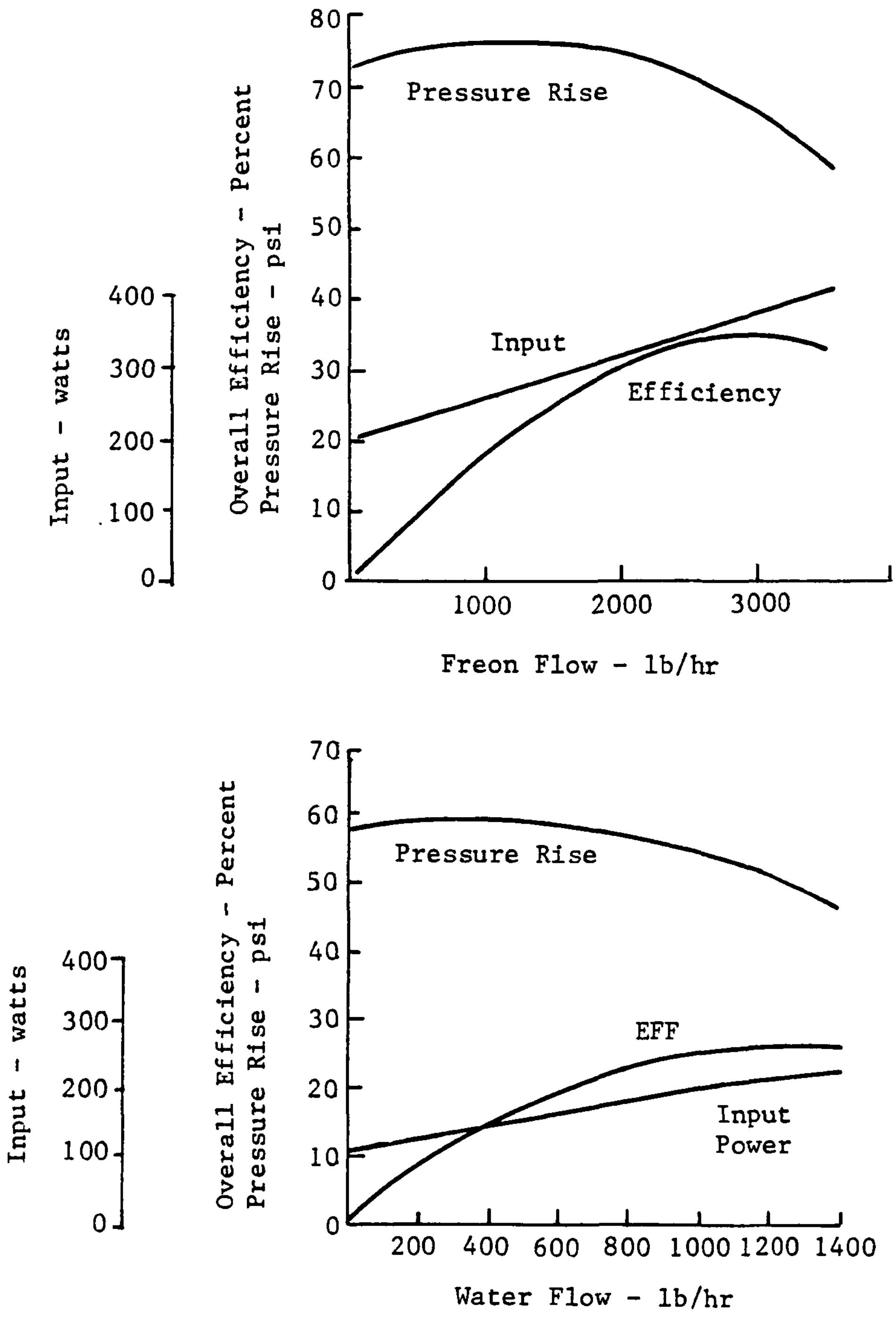


Figure 6.2.3-3 Orbiter Freon and Water Pump Performance

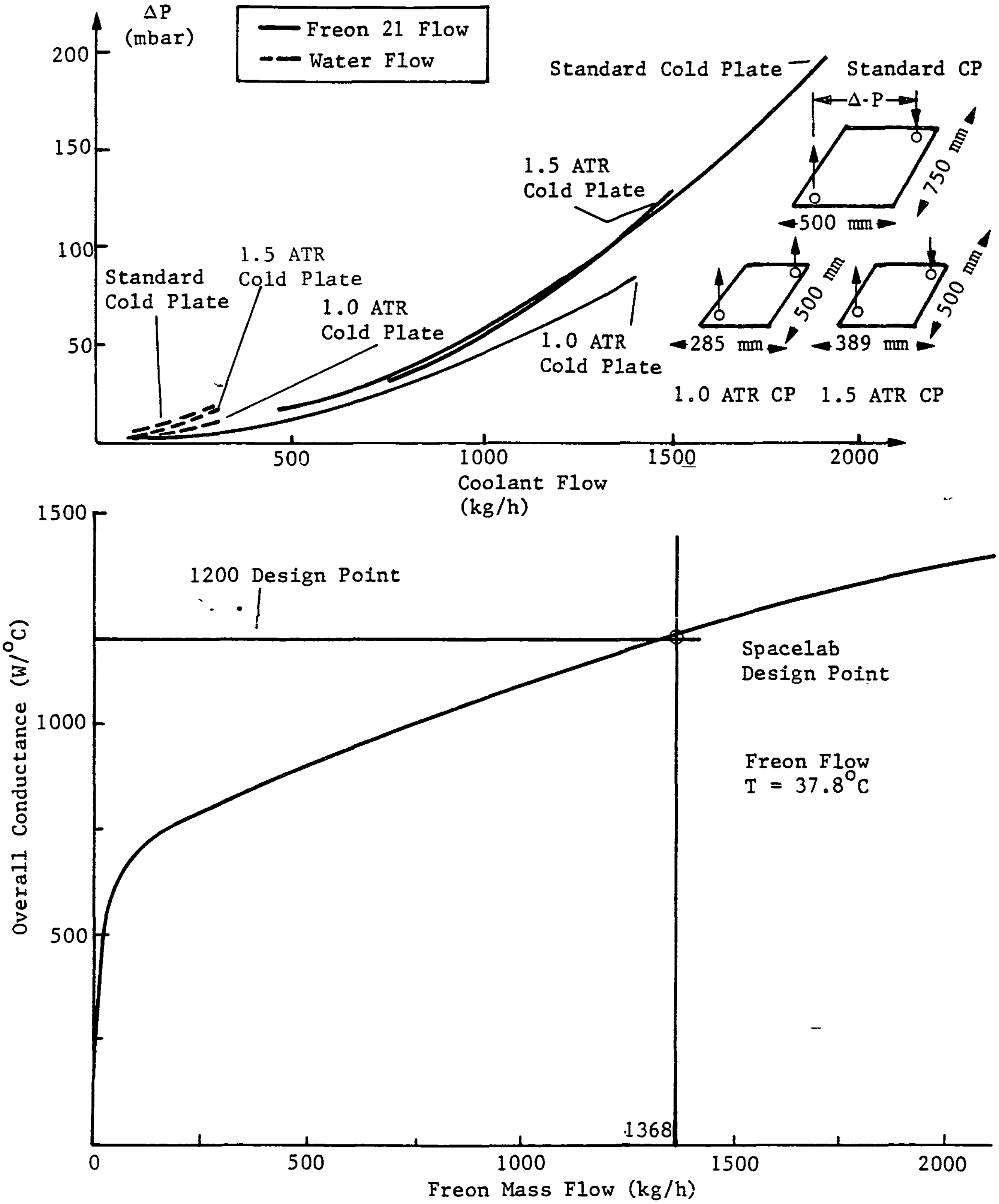


Figure 6.2.3-4 Spacelab Coldplate Pressure Drop and Thermal Performance

The interface approach for modules that require coupling to the heat transport loop after docking is shown in Fig. 6.2.3-5. By using a heat exchanger at each docking interface, and with the interface disconnects in the docked module, a failure in a disconnect would not jeopardize the integrity of the S.S. thermal control system. With this approach, the docked module has its own independent heat transport loop (pumps, accumulator, valves, etc). All docked module payload heat exchangers are in parallel. This permits the same temperature coolant to be available at each docking interface.

Thermal Capability Growth for S.S. Configuration Options - Of the three options derived in Section 4.0, the Modular and Aft Cargo Carrier (ACC) options both expand in a modular fashion to ultimately be capable of accommodating 12 crew in 1995. This permits the TCS capability to grow as the S.S expands when the individual Energy and Habitat modules contain independent pumped loops each with an associated set of body mounted radiators. With these two S.S. options it is feasible to incorporate technology developments such as two phase coolant loops in the build of the later modules. In the next section on Evolution 6.2.4, the extent to which the TCS stepwise growth phases in with the heat load buildup is addressed.

The third S.S option, the Shuttle Derived Vehicle (SDV), represents an "all-up" approach. The TCS must be sized to reject the entire ultimate heat load from the S.S. Careful consideration must therefore be paid to operation at the early low heat-loads to avoid freeze-up. This problem could be overcome by the selection of the temperature control system for the radiator outlet and/or electrical heaters in the coolant circuit, since in 1990 considerable excess power is available from the solar array (See EPS Growth Requirements - Manned, Fig. 6.1.3-1). It is likely that a regenerative heat exchanger control system will be capable of providing the necessary turn down ratio (maximum heat rejection/minimum heat rejection).

A large surface area is available on the shell of this S.S. option for the mounting of radiators. The extent to which this corresponds to the area required for rejection of the ultimate heat load is addressed in the next section dealing with Evolution, 6.2.4.

6.2.4 Evolution

TCS evolution is impacted by the time-phased buildup of the S.S. heat loads over the S.S. life and the physical expansion of the S.S. In this section the heat rejection requirements are compared with the heat rejection capability of the body mounted radiator areas available on each of the three S.S. configuration options. This leads to a determination of the size of the deployed radiator required for each option. The deployed radiator size is of concern because it is potentially one of the principal drivers in the overall configuration.

Phased Buildup of Heat Loads - Heat rejection requirements are summarized in Table 6.2.4-1 for the 1990 to 2000 time period. These include the waste heat dissipated in the EPS, metabolic and humidity control heat loads in the ECLS, and all bus power heat loads. It is recognized that the battery heat loads and the charger heat loads do

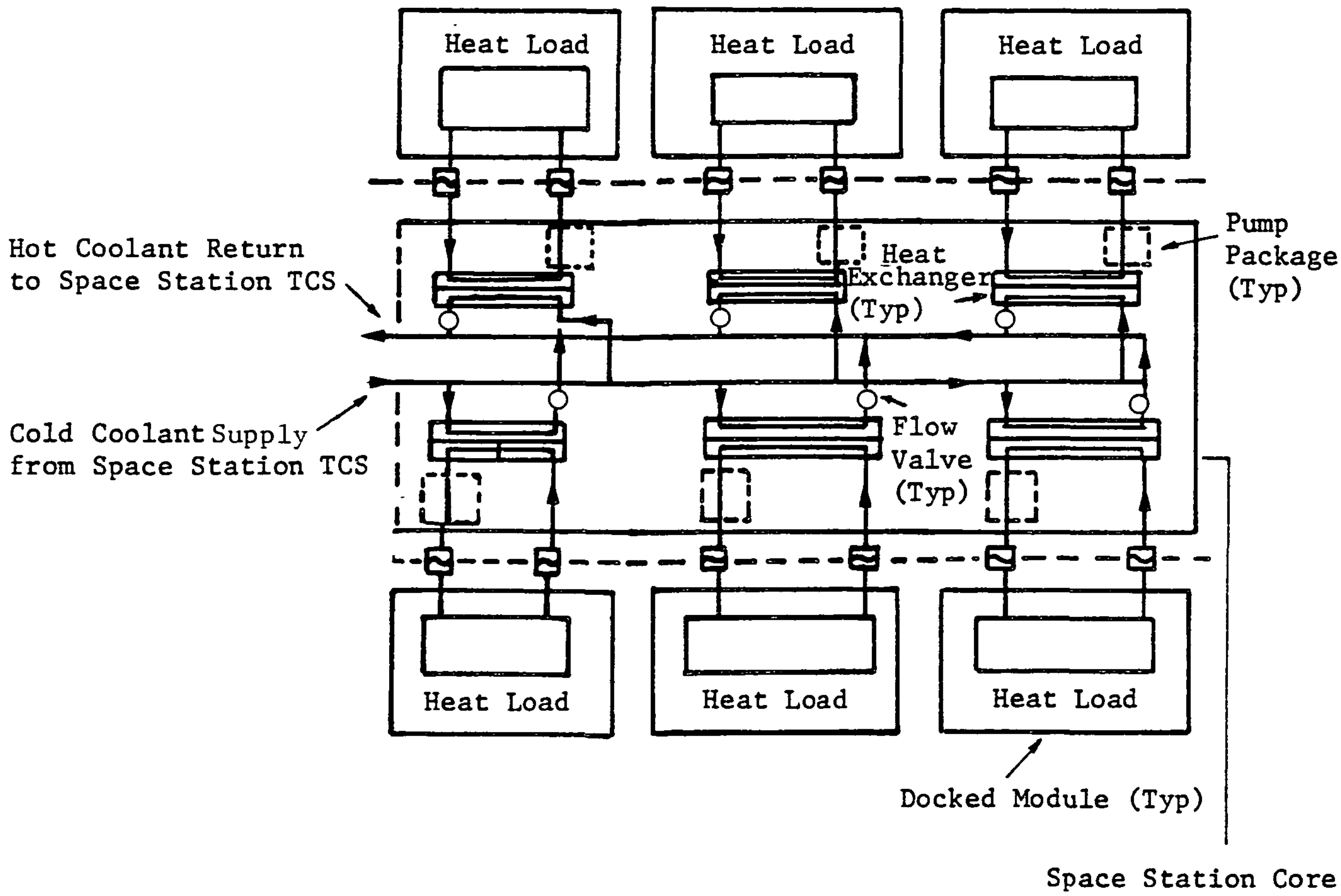


Figure 6.2.3-5
Payload Heat Exchangers at Docking Ports - Six HX in Parallel Flow Arrangement

Table 6.2.4-1 Time Phased Heat Rejection Loads

	1990	1991	1992-94	1995-2000
•Power System Equipment				
NiH_2 Batteries	961	2,295	3,985	5,050
Battery Charger	621	1,482	2,583	3,333
Regulator	7,970	4,703	8,292	10,717
Switching	100	239	478	693
Electronic Controls	248	297	562	742
	3,900w	9,016w	15,900w	20,535w
•Environmental Control System and Life Support				
Cabin Air Loop				
- Fans and Blowers				
- Metabolic				
- Humidity				
- Electrical				
- Wall Heat Leak				
Air and Water Management				
- Air Revitalization				
- O_2 Generation				
- H_2O Processing				
- Hygiene/Health				
	8,248w	8,248w	16,333w	24,480w
•Subsystems				
Communications	241	241	596	856
Data Management	205	205	490	880
Propulsion	200*	200*	1,000*	1,000*
Thermal Control System	1,010	1,010	1,750	2,260
Attitude Control System	450	450	450	450
	2,106w	2,106w	4,286w	5,446w
*Not Cold Plated				
•User Requirements	1,000w	16,350w	24,550w	29,050w
•TOTAL COOLANT LOOP LOAD	15,054w	35,520w	60,069w	78,511w

not occur simultaneously. However, due to the preliminary design phase of the S.S., their orbital average heat loads at ß angle 0° were selected as representative for the purpose of TCS sizing. The ECLS system heat load determination requires detailed consideration of crew size and activity, and space station size. Accordingly the values indicated in Table 6.2.4-1 are very preliminary. The heat load levels assume a four crew level for 1990, 8 crew for 1992 and 12 crew for 1995.

Heat Rejection System Evolution - Radiator sizes consistent with the previously derived phased EPS requirements are determined in Table 6.2.4-2 for the Modular S.S. option, in Table 6.2.4-3 for the ACC S.S. option and in Table 6.2.4-4 for the SDV S.S. option.

Modular S.S. Option - For this option it is demonstrated in Table 6.2.4-2 that the extensive use of body mounted radiators obviates incorporating a deployed radiator. The radiators are integral with the micro-meteoroid bumpers of the Energy-Modules and Habitats. A dedicated high temperature radiator (260°F inlet, 50°F outlet) is integral with the shell of the MPL for thermal control of the furnaces. Table 6.2.4-2 shows that the first Energy-Module has the capability to reject all the EPS and Avionics heat loads in the period 1990 to 1991. The delivery of the second Energy-Module in 1992 provides the added capability to manage all EPS and Avionics heat loads onward from 1992. The first Habitat has the capability to reject the ECLS and user heat loads generated in the period 1990 to 1991. From 1992 to 1994 the incremetal increase in these heat loads is managed by the second Habitat and onward from 1995 by the third Habitat. Thus for this configuration option the TCS modular growth with S.S. expansion is in phase with the heat load buildup. A reasonable margin is shown to be available for growth in heat loads and/or for compensating for degradation of the Silverized Teflon beyond the α_s degraded design value of 0.21. For any necessary further growth the use of the large surface area provided by the Hangar would be explored.

ACC S.S. Option - This option physically expands in a similar manner to the Modular option and utilizes the same design Energy-Module. Thus the EPS and Avionics heat load build-up is managed by the Energy-Module TCS exactly as in the previous configuration option. The overall body mounted radiator capability is lower however because the radiator surface area available on the Habitat-Module designed for this option is approximately half that of the Modular option Habitats, see Table 6.2.4-3. This radiator area shortfall is compensated for by a 144 ft^2 gimballed radiator mounted on the solar array boom. Essentially, this additional capability using a deployed radiator provides for the heat loads incurred by the Electrophoresis Labs in the third quarter of 1992. Dedicated radiators for these labs was considered but the area available in total on the 4 labs was determined to be marginal for the 6 Kw (total) low temperature rejection requirement. In keeping this deployed radiator size to a minimum and also to postpone its need for installation until late 1992, a portion of the MPL surface is employed as an extension of Habitat No. 1 radiator area, see Table 6.2.4-3. The remaining portion of MPL surface provides for a dedicated high temperature radiator for the furnaces. In effect the MPL radiator is a split temperature radiator. With this S.S. option, heat rejection growth can be accommodated by expanding the deployed radiator i.e., by

Table 6.2.4-2 Evolution For Modular Space Station Option

	Coolant Loop		Heat Rejection System				
Year	Source	Load, KW	Radiator Location	Radiator Area Avail. Ft^2	Rej. Capability, KW	Rej. Capability, ÷ Load	Deployed Radiator Area Rqmt Ft^2
1990	EPS & Avionics	5.8	Energy Mod No. 1	1,325	15.9	2.7	0
	ECLS HMF Total	8.3 1.0 9.3	Habitat No. 1	2,115	25.4	2.7	0
1991	EPS & Avionics	10.9	Energy Mod No. 1	1,325	15.9	1.5	0
	ECLS HMF Payloads (6) Total	8.3 1.0 9.4 18.7	Habitat No. 1	2,115	25.4	1.4	0
	Furnaces	6.0	MPL	832	32.0*	5.3	0
1992 to 1994	EPS & Avionics	19.2	Energy Mods No. 1&2	2,650	31.8	1.7	0
	Same as 1991	18.7	Habitat No. 1	2,115	25.4	1.4	0
	ECLS DOD Area Comm & DOD P/L Integr Electrophoresis Total	8.1 1.6 0.6 6.0 16.3	Habitat No. 2	2,115	25.4	1.6	0
	Furnaces	6.0	MPL	832	32.0*	5.3	0
1995 to 2000	EPS & Avionics	25.0	Energy Mods No. 1&2	2,650	31.8	1.3	0
	Same as 1991	18.7	Habitat No. 1	2,115	25.4	1.4	0
	Same as 1992-4	16.3	Habitat No. 2	2,115	25.4	1.6	0
	ECLS HMF Life Science Total	8.1 2.0 2.5 12.6	Habitat No. 3	2,115	25.4	2.0	0
	Furnaces	6.0	MPL	832	32.0*	5.3	0
			Total Deployed Radiator Requirement				0

*High Temperature Radiator ($50^{o}F$ to $260^{o}F$)

Table 6.2.4-3 Evolution for ACC Space Station Option

Year	Coolant Loop		Heat Rejection System				
	Source	Load, KW	Radiator Location	Radiator Area Avail. Ft^2	Rej. Capability, KW	Rej. Capability, ÷ Load	Deployed Radiator Area Rqmt Ft^2
1990	EPS & Avionics	5.8	Energy Mod No. 1	1,325	15.9	2.7	0
	ECLS HMF Total	8.3 1.0 9.3	Habitat No. 1	1,047	12.5	2.7	0
1991	EPS & Avionics	10.9	Energy Mod No. 1	1,325	15.9	1.5	0
	ECLS HMF Payloads (6) Total	8.3 1.0 9.4 18.7	MPL & Habitat No. 1	1,874	22.5	1.2	0
	Furnaces	6.0	MPL	220	8.4*	1.4	0
1992 to 1994	EPS & Avionics	19.2	Energy Mods No. 1&2	2,650	31.8	1.7	0
	Same as 1991	18.7	HabitatNo. 1 & MPL	1,874	22.5	1.2	0
	ECLS DOD Area Comm & DOD P/L Integr Electrophoresis Total	8.1 1.6 0.6 6.0 16.3	Habitat No. 2	1,047	12.5	0.8	130
	Furnaces	6.0	MPL	220	8.4*	1.4	0
1995 to 2000	EPS & Avionics	25.0	Energy Mods No. 1&2	2,650	31.8	1.3	0
	Same as 1991	18.7	Habitat No. 1 & MPL	1,874	22.5	1.2	0
	Same as 1992-4	16.3	Habitat No. 2	1,047	12.5	0.8	130
	ECLS HMF Life Science Total	8.1 2.0 2.5 12.6	Habitat No. 3	1,047	12.5	0.99	14
	Furnaces	6.0	MPL	220	8.4*	1.4	0
			Total Deployed Radiator Requirement				144 Ft^2

*High Temperature Radiator Segment ($50^{o}F$ to $260^{o}F$)

Table 6.2.4-4 Evolution For SDV Space Station Concept

Year	Coolant Loop		Heat Rejection System				
	Source	Load, KW	Radiator Location	Radiator Area Avail. Ft^2	Rej. Capability, KW	Rej. Capability, ÷ Load	Deployed Radiator Area Rqmt Ft^2
1990	EPS, Avionics, ECLS, & HMF	15.1	SDV Space Station	4,282	51.4	3.4	0
1991	EPS, Avionics, ECLS, HMF, & Payloads (6)	29.6	SDV Space Station	4,282	51.4	1.7	0
	Furnaces	6.0	MPL	832	32.0*	5.3	0
1992 to 1994	Same as 1991+(Growth in EPS, Avionics, ECLS), DOD Area, Communication, & DOD P/L Integr.	48.2	SDV Space Station	4,282	51.4	1.1	0
	Furnaces	6.0	MPL	832	32.0*	5.3	0
	Electrophoresis	6.0	EOS Labs (5)	880	7.6*	1.3	0
1995 to 2000	Same as 1994 + (Growth in EPS, Avionics, ECLS, HMF)	64.1	SDV Space Station	4,282	51.4	0.8	500
	Furnaces	6.0	MPL	832	32.0	5.3	0
	Electrophoresis	6.0	EOS Labs (5)	880	7.6**	1.3	0
	Life Science	2.5	Life Science Module	832	10	4.0	0
			Total Deployed Radiator Requirement				500

*High Temperature Radiator (50^{o}F to 260^{o}F)
**Low Temperature Radiator (32^{o}F to 41^{o}F)

the addition of further "plug-in" radiator panels. This assumes that the provision of additional contact-heat exchangers would be anticipated in the design phase. Alternatively, as in the previous S.S. option the use of the large surface area provided by the Hangar would be explored.

SDV S.S. Option - Unlike the two previous options the only body mounted radiator time-phased expansion is by the addition of one MPL, five Electrophoresis Lab Modules and one Life Science Module to the large center core (SDV). These added modules have dedicated heat rejection systems thus lowering the total heat rejection requirement of the SDV radiators to that for the systems/users accommodated in its interior. Table 6.2.4-4 shows that a heat rejection area shortfall exists for the SDV which requires the addition in 1995 of 500ft^2 of gimballed radiator mounted on the solar array boom. As discussed in the previous S.S. option evolution, growth in heat rejection requirements would be accommodated by expanding this deployed radiator.

6.2.5 Technology Advancements

Advancements are needed in a number of TCS technology areas to support the future long life and flexible thermal management system for S.S.

1) Liquid systems development are needed for four pass rotary fluid joints, no leakage quick disconnects and contact heat exchangers.

   The proven life and capacity of the Shuttle-type pumps needs extrapolating. Variable speed pump development is desirable.

   Versatile approaches to "cold-plating" the diverse types of equipment are needed.

   In the area of 2 phase flow, research is needed into the mechanisms of zero-g evaporative and condensing heat transfer. Two phase cold-plate heat exchanger, and radiator, component development is needed.

2) Radiator development needs include an optically stable, contamination resistant 10 year life coating with an EOL $\alpha_s/\varepsilon \leq$ 0.2/0.8. Also on-orbit coating cleaning, refurbishment, and replacement techniques are needed. High heat-transport/extended length heat pipe development is needed for radiators as well as methods, procedures and tools for their orbital assembly.

6.3 PROPULSION

The space station propulsion subsystem has two main tasks, space station Orbit Adjust and Attitude Control (OA/ACS) and propulsion system resupply and servicing are both the space station itself (and its TMSs) and a reuseable OTV service center.

6.3.1 Requirements. Space station propulsion requirements are composed of ground rules of the study as well as user derived requirements. Study groundrules include the 90 day resupply interval, STS compatibility, 10 year life, evolutionary growth, modularity, end of life deorbit, and manned presence. The rest were user derived. These are shown in Table 6.3.1-1 and discussed below.

6.3.1.1 Orbit Maintenance and Attitude Control. Using nominal space station configurations and the worst case atmospheric densities a maximum drag level of 0.15 lb_f was determined. This is for a FOC station in a high solar year. SS dynamic response limits the acceptable acceleration to 0.05 g's. Torque requirements from disturbances are expected to need approximately 200 ft lb_f to counteract. An end of life deorbit requirement was assumed which requires 200 fps of Δ velocity within 10 minutes to assure a compact foot print. Drag and resupply cost considerations were used to arrive at a nominal operating altitude of 250 nm.

Several other requirements could not be quantified at this time but desired trends were identified i.e., minimum contamination, plume impingement and propellant usage were identified as requirements. Low cost, low risk was assumed a requirement in order to keep front end costs low. Life cycle cost studies may relax this in favor of low operating cost at the expense of front end costs. The implications of this are discussed under 6.3.2.3, Alternate Approaches. Likewise reliability/redundancy are assumed to be the maximum obtainable. Fail safe/fail operational will be followed, as is standard practice. STS safety guidelines are to be used. Propulsion systems are to be welded to the fullest extent possible; leak proof mechanical joints used where necessary. Automated system performance monitoring and long life components are to be used to minimize crew involvement in system trouble shooting and maintenance. Front end costs increase due to software development but a more efficient crew utilization results. A potential requirement for space debris collision avoidance has been identified but no quantification has been determined.

6.3.1.2 Propellant Resupply and Servicing.

Space Station and TMS. Resupply and servicing of the space station has been baselined to occur on at least 90 day intervals. The TMS and space station are to use a common propellant to ease cost and complexity of resupply logistics. Resupply is effected via a logistics module supplied by the shuttle. Pressurant resupply is also to be provided by the logistics module. Planned servicing for TMSs and similar propulsion system is to be provided for.

OTV. Examination of the mission model identified support of GEO bound vehicles as a key function. A reusable OTV is therefore deemed vital, and assumed available when needed. Support of the OTV is to include not only resupply of propellant but any necessary replacement/refurbishment planned for the OTV. The break point for cost effective service performed at the space station as opposed to ground has not been determined. It is anticipated that only routine service and some limited contingency servicing will be provided. The cost of providing training and tools for all contingencies is prohibitive and a return to ground would be necessary for any servicing outside that planned for.

Table 6.3.1-1 Propulsion System Requirements

Space Station Orbit Maintenance and Attitude Control

Requirement	Value	Comment
Altitude	250 nm	
Attitude	Gravity Gradient	Minimize Prop. Consumption
Torque	<200 ft-lb_f	
Drag	.001 - .15 lb_f	
Acceleration	< 0.05 g's	
Resupply Internal	90 Days	
Reliability	Fail Safe/Fail Operational	
End of Life Deorbit ΔV	200 fps	
Time	10 min	
Component Life	10 yr	
Contamination	Minimize	
Collision Avoidance, Time	TBD	
ΔV	TBD	

Propellant Resupply and Service

Requirement	Value	Comment
OTV Flights	24/yr max	
Capacity	35 Klbm	
SS Tank Capacity	70 Klbm	
SS Resupply Interval	14 days nominal 90 days max	
Quantity Resupplied	25-50 Klbm	
Bailoff	<1%/30 days	
OTV Maintenance Interval	20 missions	
OTV Storage		

Cryogenic oxygen and hydrogen have been identified as the propellants for the OTV. Mission model analysis indicates a requirement of up to 24 OTV flights per year. A maximum propellant capacity for the OTV of 35,000 lbm at 6:1 MR was determined. A nominal space station cryogen storage capacity of 70,000 lbm has thus been identified. This is for a space station cryogen resupply frequency of two weeks with either 50 K lbm or 25 K lbm of propellant being brought up. (Depending upon immediate need.) Boil off from the space station tank would be limited to less than 1% over a 30 day period.

OTV servicing will be limited to planned operations as in TMS and space station servicing. A 20 mission life between refurbishments has been identified. The OTV would then be brought back down for major overhaul. On orbit refurbishment would be limited to items determined to be cost effective. OTV life limitations and payload manifesting will require two OTVs be based at the space station during overlap periods. A hangar will be required to service and store the OTVs and payloads. Checkout and verification will also be performed in the OTV hangar prior to refueling the OTV.

6.3.2 Trade Studies. Due to the preliminary nature of the space station design, the normal propulsion system trades cannot be very detailed or the results considered as other than preliminary. Three high level trades were performed as well as numerous lesser trades or considerations. These lesser trades will be discussed in relation to their conclusions as they occur in Section 6.3.3. The higher level trades performed were electric vs chemical propulsion, propellant selection, and finally a central vs distributed propulsion subsystem design. These trades are summarized in Table 6.3.2-1 and discussed below.

6.3.2.1 Electric vs Chemical Propulsion. The main incentive for using an electric propulsion system for the space station is the potential to reduce the operating costs of the station. This primarily results from a reduced launch cost for supplying space station propellant due to the substantially higher specific impulse (Isp) possible with electric propulsion. This is offset by a higher program front end cost due to the relative unmaturity of the technology. In addition, the low thrust available from electric propulsion systems will complicate ACS control torques and certain high thrust maneuvers such as controlled deorbit. Chemical assist back up systems will thus still be needed. The high power required is a further penalty for the electric system. It is certain that a future for electric propulsion exists and with the increased emphasis on space use and lower operating costs placing a premium on high Isp systems the space station could play an important role in developing electric propulsion. This could be either as an orbiting laboratory or a space station orbit adjust module. Due mainly to the relative state of maturity, the chemical system was chosen but it is recommended that the trade be revisited to compare chemical and electrical propulsion on a LCC basis.

6.3.2.2 Propellant Selection. The three common propellant types were considered, mono-propellant hydrazine, earth storable bi-propellant and cryogenic bi-propellant. Exotic propellants were considered inappropriate due to lack of development and perhaps incompatibility with a manned space station. Earth storable bi-propellants were eliminated because their performance advantage relative to mono-propellant would not offset the increased complexity and contamination. Increased complexity and high front

Table 6.3.2-1 Propulsion System Trade Studies

Chemical vs Electric Propulsion

Electric		Chemical	
Pro	Con	Pro	Con
High Isp	High Power Consumption	Mature Technology	Low Total Impulse
Low Operating Cost	Heavy	Low Initial Cost	High Contamination
Low Contamination	High Technology Risk	Simple	High Resupply Cost
High Total Impulse	Limited Thrust	Reliable	
	High Initial Cost	Unlimited Thrust Levels	

∴ Choose Chemical; Low Initial Cost, Risk

Propellant Selection

N_2H_4	MMH/NTO	LH_2/LO_2
Low Contamination	High Contamination	Low Contamination
Low Isp	Medium Isp	High Isp
Simple	Moderate Complexity	Most Complex
Low Cost Components	Moderate Cost	High Cost Components
Qualified Components Available	Qualified Components	Few Qualified Components
Simplest Resupply	Moderate Resupply Complexity	Complex Resupply

∴ Choose N_2H_4; Low Cost, Contamination, & Simple Resupply

Distributed vs Central Propulsion Subsystem

Central		Distributed	
Pro	Con	Pro	Con
Low Weight	Configuration Dependent	Flexible	Heavy
Simple Resupply	Limited Growth	Accommodates Growth	More Complex ACS
Low Crew Involvement	Long Line Runs	Simple SS Interface	High Crew Involvement

∴ Choose Central; Low Weight, Crew Involvement end cost also eliminates cryogens in favor of hydrazine. However, if an integrated cryogen system consisting not only of space station propulsion but OTV, EPS fuel cells and ECLS water supply were to prove desirable, then the cost of developing the special low thrust ($\sim$30 lbf) engines and componentry could be justified. Presumably, OTV cryogen boil off would be utilized for all functions other than OTV propellant. This means placing a vapor storage and handling system on board the space station to serve these functions. This would be advantageous if "cheap" cryogen propellants could be supplied to the space station. However, hydrazine was chosen as the baseline propellant, primarily because of its mature development status and favorable contamination and handling qualities.

6.3.2.3 Central vs Distributed Propulsion System. Due to the modular nature and evolutionary growth for the proposed space station concepts, a distributed propulsion system is attractive. For this concept, small, self contained (tanks, heaters, valves, and thrusters) propulsion modules would be placed about the space station where needed. Their only interface would be mechanical for attactment and electrical for data, command and heater power. These modules would be removed for resupply and/or reconfiguration of SS propulsion. Module propellant capacity would be a compromise between the desire to reduce the resupply frequency (large tank) and the desire to ease handling problems (small tank). This system places no constraints on space station configuration or evolution and would allow for space station propulsion system growth in both size and type. It is rather crew intensive in that module replacement and resupply would require crew activity.

On the other hand, a centralized system with one set of tanks feeding remotely placed thrusters through the appropriate plumbing in the conventional manner would be less crew intensive. Due to attitude control considerations, space station evolution is expected to occur in such a manner that the center of mass will remain in roughly the same spot, i.e., growth will be in symmetrical increments about the center of mass to ease updating of control algorithms. This allows for placement of thrusters in a permanent location. Since some propulsion will be needed for the first space station unit placed in orbit, a permanent installation of tanks and valves is implied. Growth would be accomplished through enlarged external tanks (Logistics Module) and higher thrust replacement thrusters (if needed). In light of these considerations the central system will be chosen as baseline because it simplifies SS resupply and reduces crew activity relative to the distributed system.

6.3.3 Conceptual Design. As mentioned in the beginning, the propulsion system has been divided into the space station on board Orbit Adjust and Attitude Control system, and Propellant Resupply and Servicing. The latter includes an OTV, TMS and the space station itself. This section will cover the baseline design of the two systems. It is apparent that there will be some overlap between the systems and the following discussion will illuminate these interrelationships.

6.3.3.1 Baseline - Orbit Adjust and Attitude Control. The previous section on trade studies touched on the reasons for selecting the mono-propellant hydrazine centralized system for the baseline. The centralized system was enabled by attitude control limitations imposed upon space station growth which held the center of mass within a relatively small envelope. Thrusters can then be placed about this envelope and sized to provide the necessary control torques. The energy module will house the propulsion system tanks, valves, lines and thrusters since it will be the first space station element placed in orbit. This enables a fully welded propulsion system with the exception of QDs for resupply and possibly for thruster replacement. Thrusters would be placed on deployable booms with flexible lines to enable boom deployment. With a centralized system, resupply is simplified relative to the distributed system since only the space station tanks need to be resupplied. Integration with TMS is also simplified, as will be discussed in Section 6.3.3.2.

Blowdown tankage for 5000 lbm of hydrazine was selected for permanent installation on the space station. Blowdown pressurization was selected to avoid requiring resupply of high pressure pressurant. Blowdown operation will not adversely affect system performance since tank pressure can be readily resupplied via the logistics module. This will be more fully covered in Section 6.3.3.2. The value of 5000 lbm N_2H_4 for on board storage is the worst case 90 day propellant consumption derived from a .15 lb_f drag level for 90 days. With additional N_2H_4 available from the logistics module this is adequate.

The thrusters are to be deployed on 4 booms arrayed about the space station center of mass with 2 thrusters per boom (see Figure 6.3.3-1). The thrusters are canted 15° into the Z axis from the X axis to provide for roll control. All 8 thrusters will be aft pointing (-X) to minimize propellant consumption. Attitude control torques are obtained by firing only one thruster which results in both a torque for ACS and a Δ V for orbit adjust. Pure Δ V is obtained by firing two thrusters on opposite sides of the center of mass. Thruster size and boom length determination awaits final space station mass properties and ACS analysis results. 10-30 lb_f thrusters on 10-30 ft booms are envisioned. Boom dynamic response will be taylored to provide for some damping of thruster pulses to ease space station dynamics. This too awaits further analysis to fully quantify.

A nominal space station propulsion system schematic is shown in Figure 6.3.3-2 with an exploded view shown in Figure 6.3.3-1. The space station propellant delivery system is shown with resupply lines for both propellant and pressurant with the redundant QDs backed up by latch valves. A single line is used for each boom since each thruster valve is backed up by a latch valve (necessary when thrusters are removed). If the thruster design allows for thrust chamber removal down stream from the thruster valve then the boom latch valves can be eliminated. The system would than be divided into A and B systems with both A and B lines running out each boom. Latch valves are shown on each tank to allow for selective tank filling and draining. As designed, the system shown can be run directly from the logistics module when desired. This will depend to some extent upon the QD joint fidelity, i.e., if a leak check reveals a small leak then the joint will only be "wet" long enough to affect resupply of the space station tanks.

The baseline system is shown for the modular buildup architectural option, however the design is applicable to either of the other two options. The tank size and orientation and such details may change but the basic system elements will remain.

An end-of-life deorbit requirement has been identified. The Δ Velocity requirement of 200 fps in 10 minutes implies a force of approximately 1000-2000 lb_f and total impulse of $.6\text{-}1.2\times10^6$ lb_f-sec. This is beyond the capability of the baseline OA/ACS shown above.

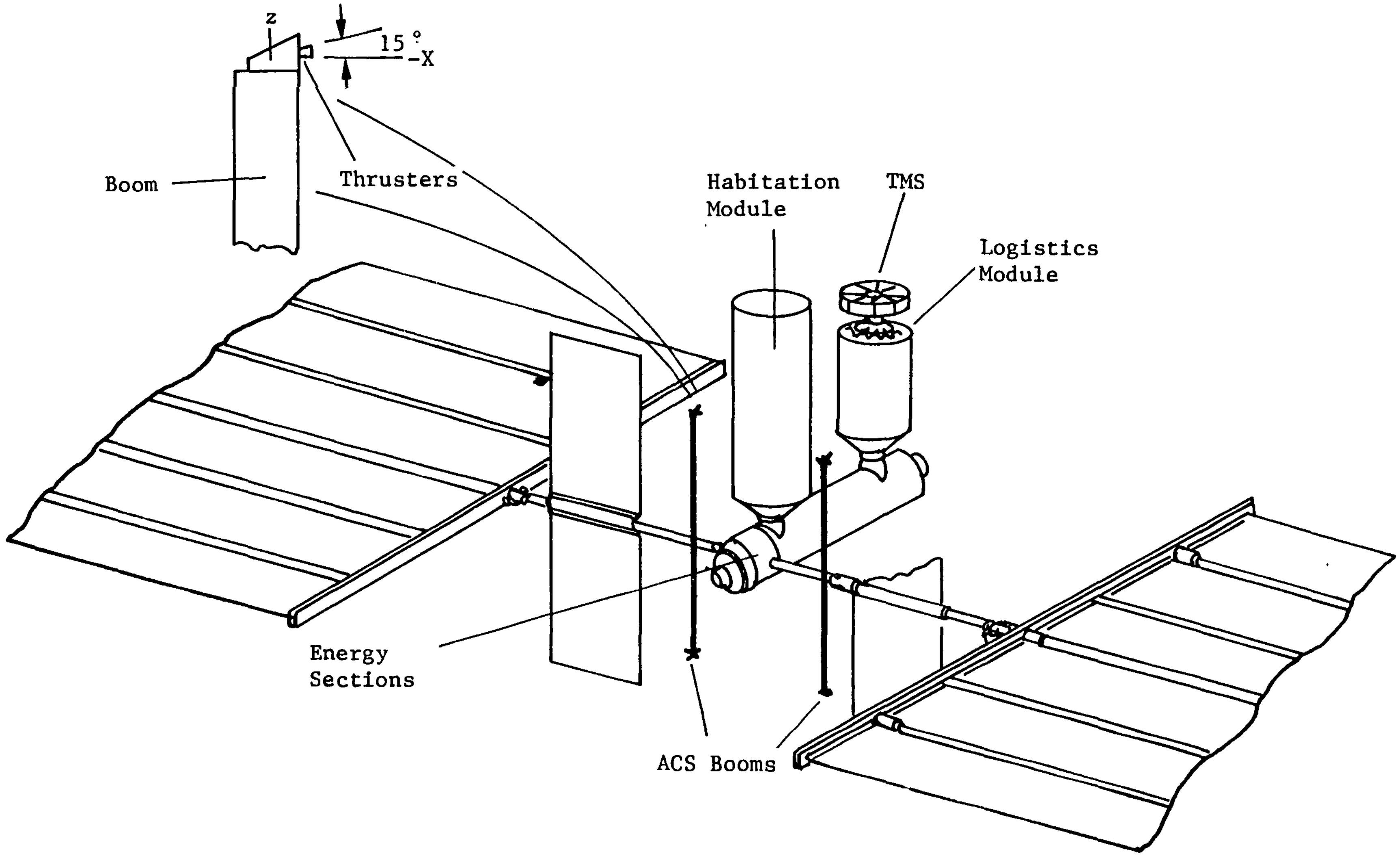


Figure 6.3.3-1 SS-LM-TMS Exploded View

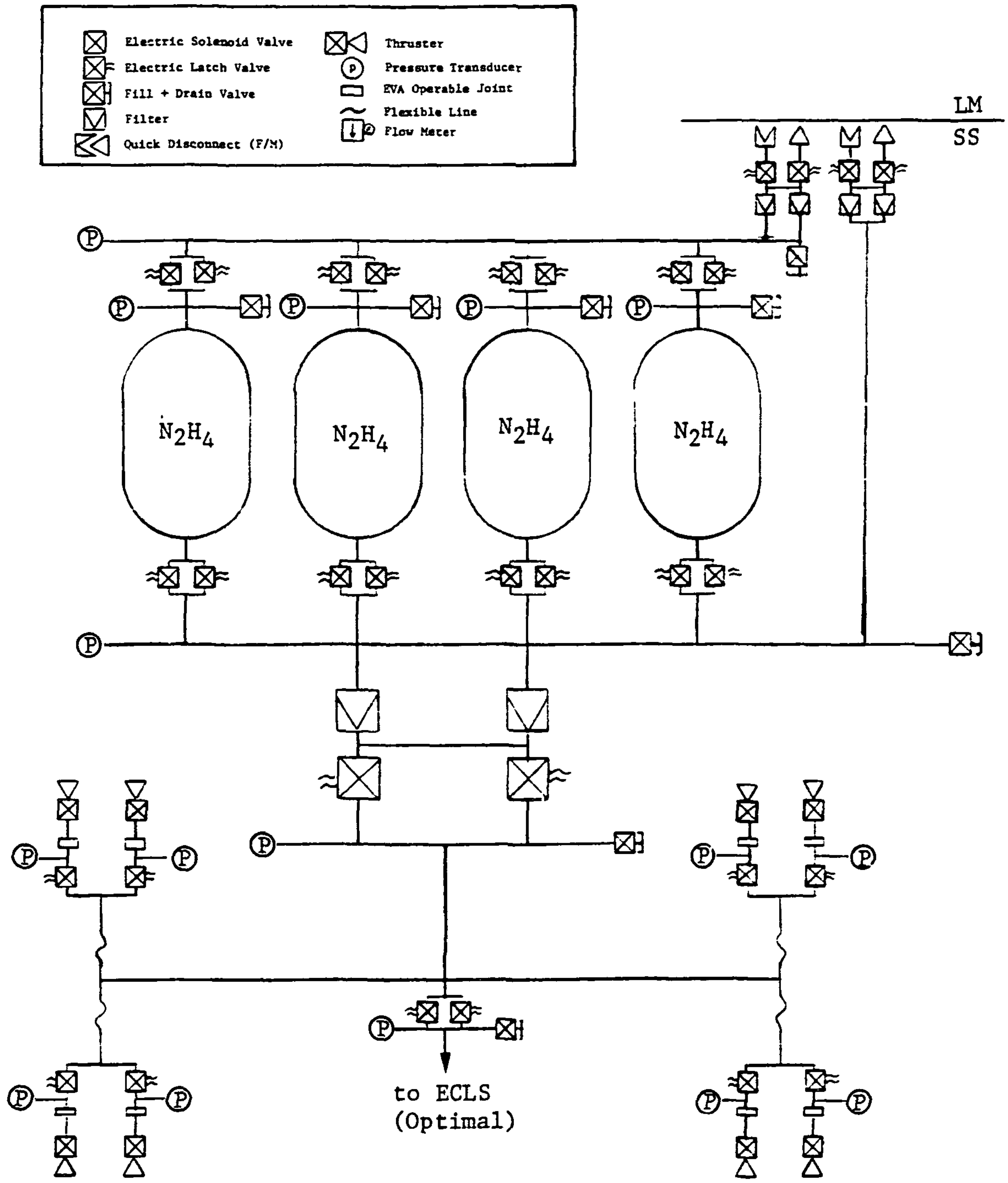


Figure 6.3.3-2 Space Station - Propulsion System Schematic

Therefore a special deorbit propulsion module will be needed with the baseline system supplying the necessary attitude control up to space station breakup. This module would be brought up and attached to a position along the x-axis to thrust through the space station center of mass. Alternately, 4 of the baseline thrusters could be replaced by 250-500 lb_f thrusters and the LM left attached so that the baseline system could be utilized. The deorbit system is left independent of the OA/ACS because it is a planned function and required only for the end-of-life condition. A contingency deorbit capability would drive the OA/ACS design and was not considered necessary.

6.3.3.2 Baseline - Propellant Resupply and Servicing. The propellant resupply and servicing function is further divided into the space station TMS subsystem and the OTV subsystem. The two systems will be functionally similar but the actual hardware will of course be different.

Space Station and TMS Resupply and Servicing. The logistics module will be used to resupply the space station and the TMS with propellant and pressurant. As currently configured, the TMS will be resupplied from the aft end of the LM while the space station will be resupplied through the LM docking interface. Heater power and transfer control signals will be provided by the space station. The TMS will be mechanically attached to the LM on its aft end to allow resupply while a payload is attached to the TMS.

The LM pressurant and propellant schematic is shown in Figure 6.3.3-3. The tanks shown are for illustration purposes only as the number of tanks has not been determined. Provisions for up to 22,500 lbm of N_2H_4 and 96 lbm of helium will be made will the initial LM will carry only 15000 lbm. The plumbing layout was designed to accommodate several modes of operation, primarily resupply of the TMS and space station. However, in emergency, propellant transfer from the TMS to/from space station is possible. For maximum expulsion efficiency, the tanks are designed to be drained sequentially. To avoid blowing gas into the propellant delivery lines a valve and line is provided from the tank outlet into the pressurant side of the next tank to allow blowing the last 2% (nominally) of propellant and pressurant into the downstream tank. This will be enabled by a properly designed in tank propellant management device (PMD).

Also shown is a quad redundant "bang-bang" pressure regulation setup to allow for a wide range of regulated output pressure. This enables the TMS and space station to operate at their respective optimum pressures, as well as allows pressure regulation for transfer between space station and TMS. Shown as part of this pressure regulation system is a manifold of sufficient volume to damp pressure pulses generated by the on-off actuations of the electric valves. No check valves were used but a full scale system analysis may reveal a need. The flow distribution latch valves are considered adequate at the present. Over pressure protection is provided by plumbing the downstream end of the pressure regulation system to the vent mast with latch valves and pressure transducers. This implies a system status subsystem with enough intelligence to detect an over pressure and open the appropriate valves and alert the space station of the problem. Mechanical relief valve/burst discs could be provided as backups. The elctro-mechanical approach was chosen to reduce the ground refurbishment costs of the LM.

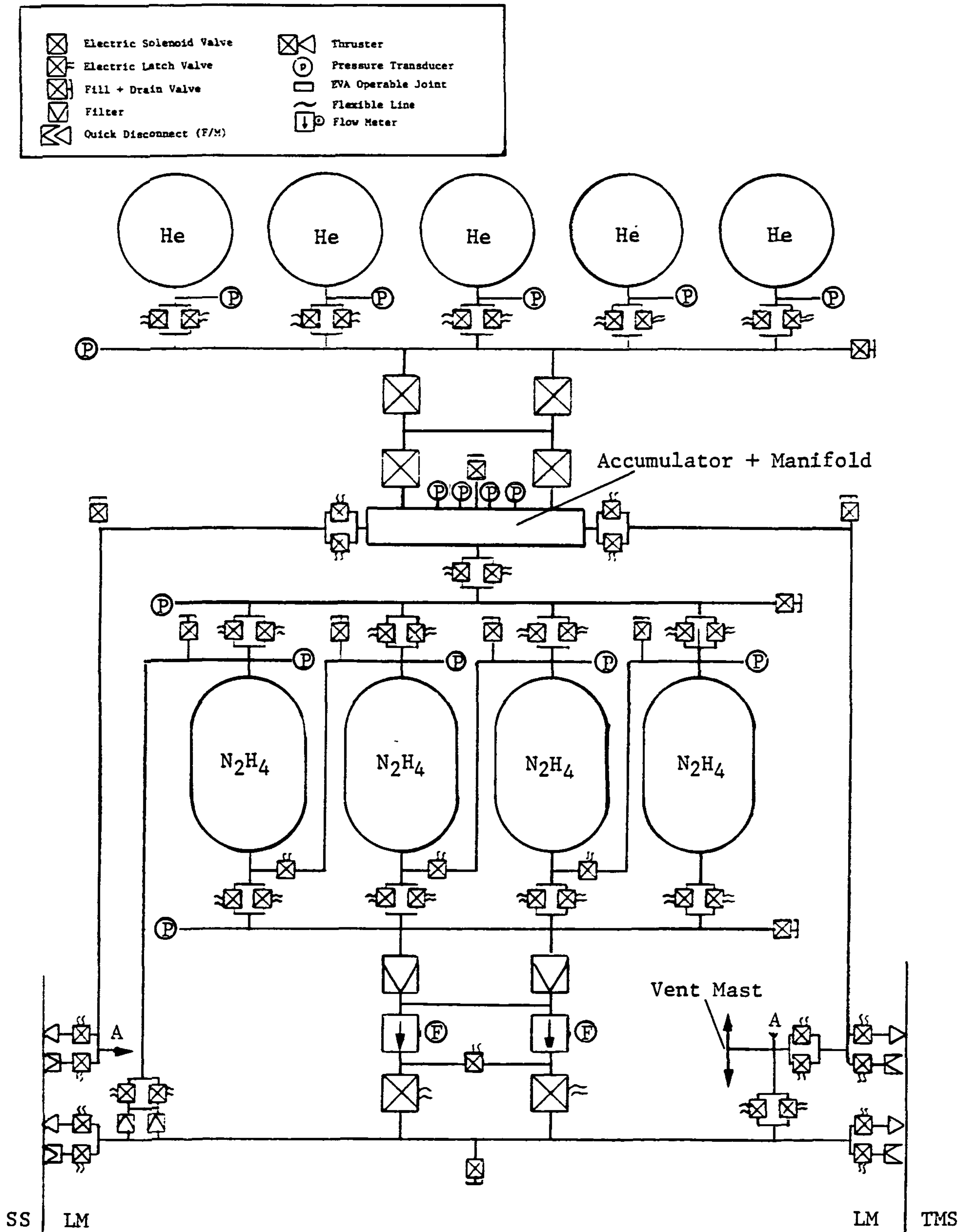


Figure 6.3.3-3 Logistics Module Propellant Supply Schematic

The vent mast is provided to allow evacuating pressurant and propellant lines when necessary as well as evacuating the TMS tanks prior to reloading if the TMS PMDs are suspected of having ingested gas. Down loading propellant/pressurant from the TMS is provided for, however this is not expected to be a routine operation. Back flowing through the propellant tank outlets is not allowed because of the possibility of ingesting gas into the propellant lines and specifically the LM propellant tank PMDs thus defeating their function. Hence the redundant one way flow meters. A latch valve is included in the cross over between the flow meters and control latch valves to control flow in the event one of the latch valves fails closed.

TMS and space station service will be primarily scheduled maintenance; i.e., replacement of LRUs. Implicit to the LRU philosophy is the use of a permanent EVA or IVA operable fluid joint. In the current OA/ACS design these are used only for the thrusters. Possibly failing into this category are critical latch valves and filters. Other than the thrusters, it is anticipated that long life components will prove more cost effective and will therefore be welded into the system, failures being provided for through redundancy. A similar philosophy will be followed for the TMS.

Not shown but also needed will be the propellant transfer control and monitor system. This system will include the ability to contain spills or leaks of a predetermined (small) magnitude and provide a proper system response in the event of such a contingency. Leaks would be detected through either monitoring pressures at various points within the flow system or propellant sensors placed near likely leak locations (QD's) or, more likely, a combination of both. A leak containment system could consist of a shroud surrounding likely leak locations and connected to a suitable disposal unit (vent mast, catalyst bed, etc). Since these systems are in need of further development, no design has been selected but are discussed because they will have an impact on the final design.

OTV Resupply and Servicing. Resupply of the OTV will be handled in a similar manner to the TMS, i.e., the OTV will be docked to the aft end of the cryogen storage module. Unlike the hydrazine LM however the space station cryogen tanks will be a permanent part of the space station, replaced only if a larger tank is needed. The space station tanks would be resupplied through tanks fitted in the orbiter. For flexibility in STS payload manifesting the STS tanks would be either of two sizes, 25 K lbm or 50 K lbm. The STS tanks would mount to the orbiter through an interface mechanically similar to the space station cryogen tanks OTV interface. This is shown in Figure 6.3.3-4. Thus the space stations tanks will be resupplied through the OTV interface and the STS tanks will mount to ASE in the orbiter which will house the abort dump hardware (see Figure 6.3.3-5). The STS tanks will be loaded on the ground or through ET scavenging. To support the projected launch rate of 24 OTVs per year, the STS must bring up one 25 K lbm and one 50 K lbm tank each month, implying that one of the orbiters will have a semi permanent installation of the STS tank ASE. The use of the 25 K lbm and 50 K lbm STS tanks will not only aid STS payload manifesting but that of the OTV as well since not all of the OTV missions will require the full 35,000 lbm propellant capacity.

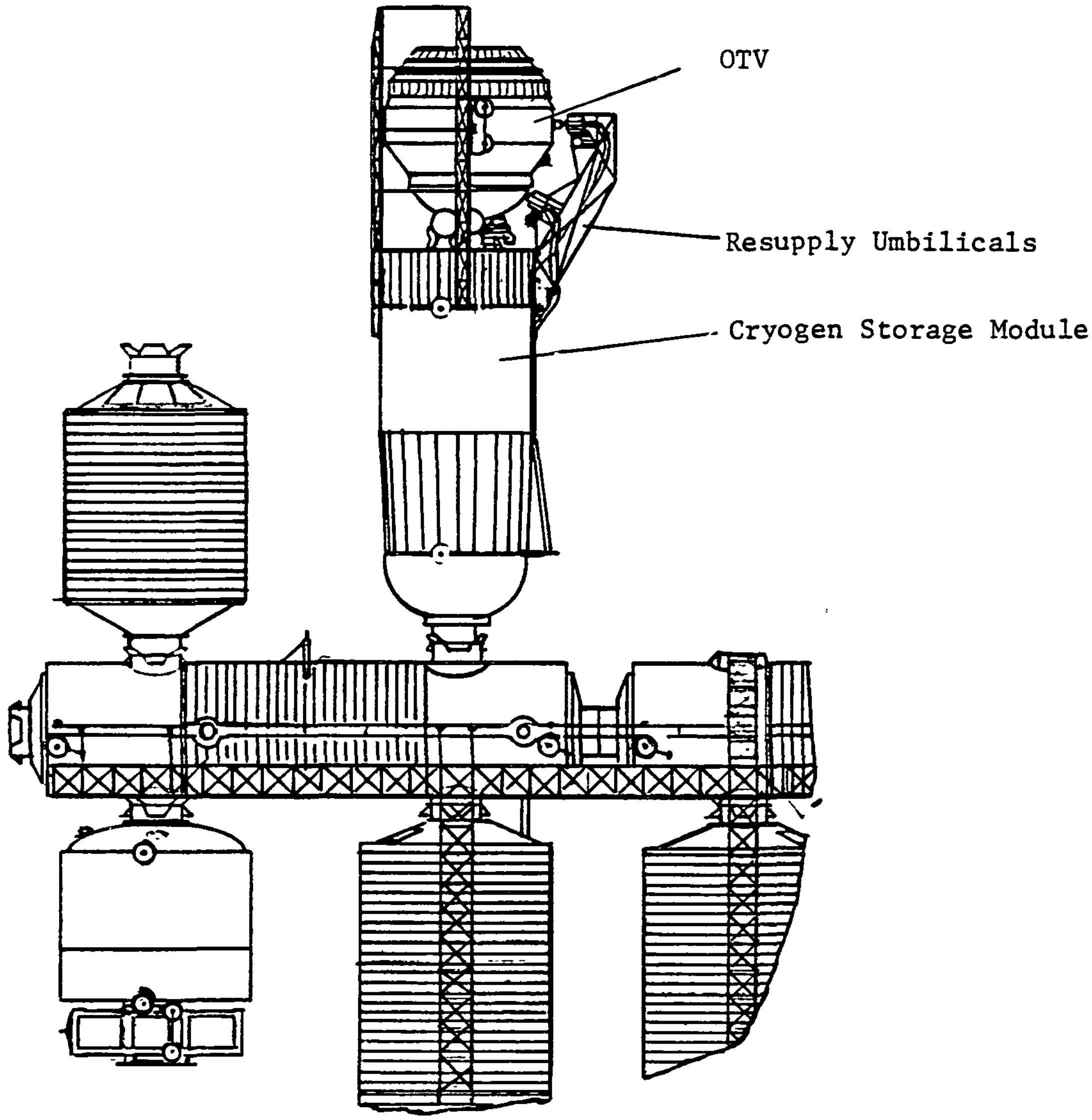


Figure 6.3.3-4 OTV Tankage + Resupply Module

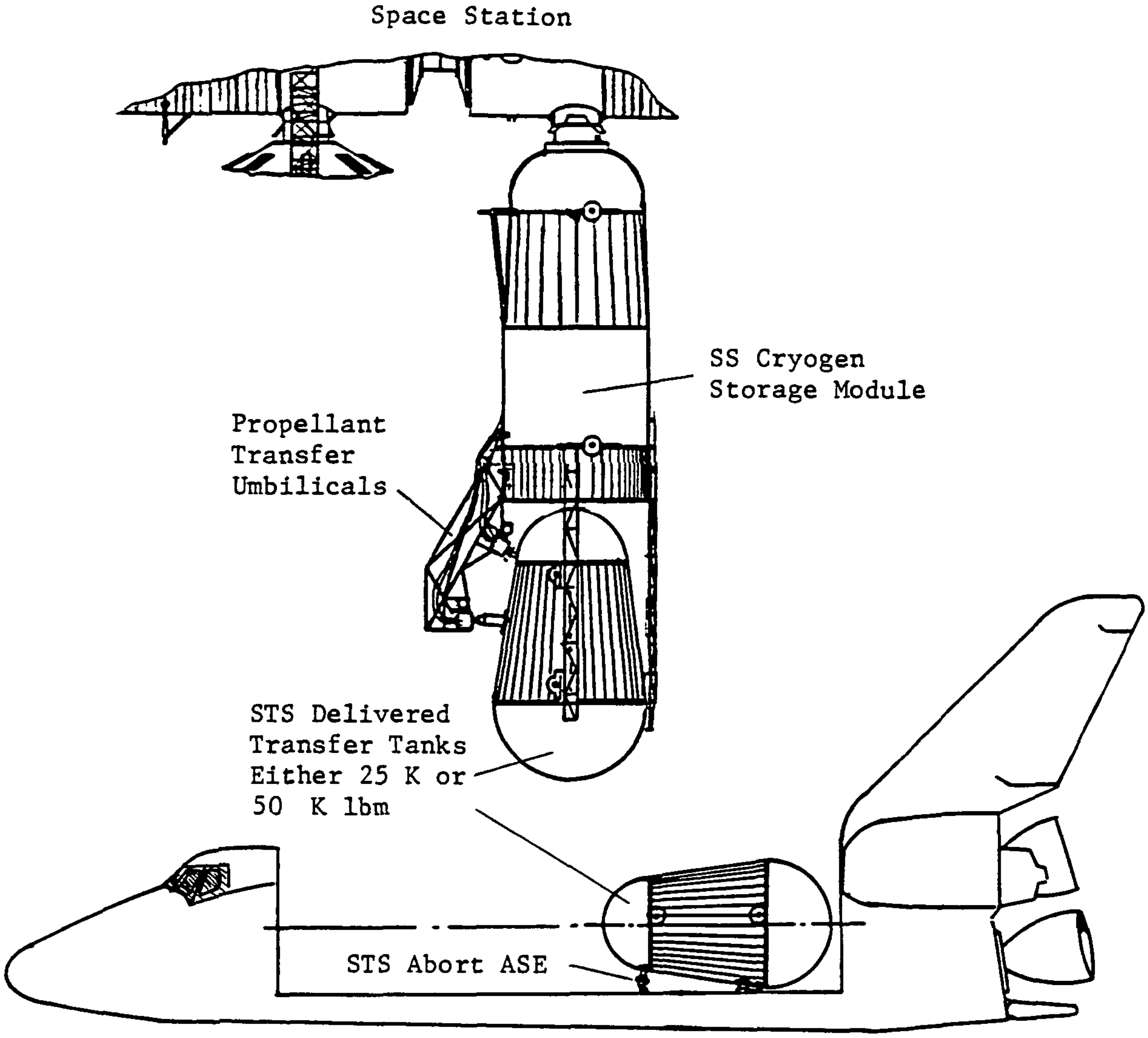


Figure 6.3.3-5 STS Cryogen Tanks for SS Resupply

As currently envisioned, the STS tanks will include a set of helium tanks to effect propellant transfer from the STS tanks into the space station tanks. The space station tanks would use an autogenous system to drive the cryogens into the OTV. This presumes the space station tanks will have a system to capture and store the cryogen vapors for use on board the space station. This does not address resupply of the OTV pressurant, which will either come from the LM, STS cryogen tanks or not be needed. The latter is the desired method where by the OTV would use autogenous pressurization for start up and proper pump design to effect operation once the engines are fired.

As in the TMS, all propellant transfer operations will be remotely performed, monitored, and to a large extent, controlled. It is not considered desirable or even necessary to have EVA astronauts handling hoses, etc. Remote mechanisms can be used to effect all the connections required for the propellant transfer operations. The degree of automation implies a rather involved control system. SS crew will however be provided with a monitoring capability and a limited manual override ability to contain unforseen events. Details of this system have not been fully worked out.

An OTV hangar will be provided to enable suited IVA servicing of the OTV and for OTV and payload storage. Maintenance of the OTV will be limited to LRU replacement and some troubleshooting and repair. OTV hangar design will need to be coordinated with the OTV design so that proper tooling, fixtures, spares etc will be on hand in order to affect efficient use of the space station crew. The OTV in the hangar will need to be empty primarily for crew safety reasons. This implies the loss of a small amount of cryogens to rechill the OTV tanks. Therefore OTV maintenance schedules will need to be coordinated to minimize the cryogen losses. This cryogen loss may not be fully recoverable and will have a direct effect on OTV operating costs.

A cryogen boil off recovery system may well prove worth while in view of the high cost of boosting the cryogens up to the space station and the potential uses they could put to (see Section 6.3.2.2). The system would primarily consist of a set of gaseous storage tanks, pumps and valves to collect the boil off which would be housed in the space station cryogen tank module. The gaseous hydrogen and oxygen would then be pressure regulated and supplied to the space station through the tank module/space station interface. Space station demand would need to match the average supply dictated by the gaseous storage capacity and the cryogen boil off rate. Within bounds this will be possible. More boil off can always be provided and, indeed there will be control of the space station tanks such that the boil off can be controlled to a higher amount when deemed necessary. This will be moderated by the cost of providing the space station with cryogens.

6.3.4 Evolution. In general, the subsystem growth or evolution will in part be dictated by the station growth. The propulsion system itself is not easy to grow since it does not lend itself to modularization in the space station sense. Thruster size and location must be determined in the early design phase and, once placed, cannot be easily moved from location to location without this being provided for. It is also undesirable to route propellant plumbing across space station element interfaces, limiting thruster placement locations. OTV support system growth is more easily accommodated in that they are self contained modules which in theory could be replaced by growth versions as needed. Cost will however limit the number of actual increments to one or two.

6.3.4.1 Space Station Orbit Adjust and Attitude Control System Evolution. With space station growth restricted to maintain the center of mass within a relatively small envelope, the thruster sizes and locations can be chose to accommodate growth without the need to be relocated. Thruster size can be changed with the current design by replacement since thruster refurbishment will require removal of the thrusters. Propellant capacity can likewise be accommodated by increasing the tankage in the logistics module. This however is the limit to space station OA/ACS growth with the baseline system. Change over to different propulsion technology would be difficult due to the degree to which the system is integrated into the space station energy section. Switching to an electric propulsion system would not be possible in the current design. For this reason, the AO/ACS will be designed to accommodate all space station growth configuration within the initial capability. It is not expected that the growth of either capacity or thrust level will be needed, but these remain as options.

6.3.4.2 Propellant Resupply and Servicing Evolution. A need for space station resupply system growth is not anticipated but could be done by making the appropriate design modifications in the LM. The TMS will likely be grown only be replacement of the TMS or rebuilding on the ground. Servicing of other S/C propulsion can be evolved as the S/C are developed with this capability and the training and tooling are supplied to the space station crew.

The OTV support system will need to evolve as the OTV is phased into operation. This is also brought about because of the size of the OTV support units, specifically the cryogen storage tanks and OTV hangar. OTV operation from the space station will be needed early to support GEO payload delivery. Therefore space station use for OTV development is minimal. The OTV is assumed developed, complete with aerobrake capability. The OTV support system evolution is geared toward transferring OTV operations from STS based to space station based.

To support OTV operations, the first element to be brought up will be the space station cryogen tank module (empty, to ease tank design), followed by the OTV itself and the OTV support hangar being the last element. These are shown scheduled in the space station evolution plan. OTV maintenance requirements may force the installation of the hangar to an earlier date than shown if EVA servicing is not adequate for the interim period.

Once these three elements are in place, no real growth is seen except in OTV launch frequency which will require a stepped up maintenance schedule and stream lining of operations. Late in the space station evolution a larger ($\sim$90,000 lbm) OTV may be utilized which will require a larger space station cryogen tank module. Because of uncertainty of requirements in the 2000 year time frame, however, this capability is not shown in the space station growth model.

6.3.5 Technical Issues and Status. Propellant resupply represents the major technical issue for the propulsion subsystem. Of primary concern is the QD design, mass flow measurement, receiver tank conditioning and the propellant transfer procedures. Also of concern is the means of monitoring the performance of the transfer operation. Component design for long life (10 + years) may be a concern, especially for the OA/ACS thrusters. The OTV operations will add to the above with the increased thermal concerns. While the OTV itself represents a wealth of technical issues they are not addressed here except for their impact to the space station OTV support system design.

Efforts are under way currently to address the above noted issues relative to propellant transfer, both for earth storable and cryogenic propellants. Several QD designs will be needed ranging from the QD for LM-TMS which will see a high cycle rate but short connect time through the mechanical joint used for propulsion system LRUs which will have a small number of cycles but a long connect time. Similar designs will be needed for cryogens where material properties and thermal design considerations will alter hardware design. Mass flow in the zero gravity environment presents some technical issues compounded by space reliability/redundancy requirements. This is needed both for space station status monitoring and to assure proper loading of the OTV and TMS. Preferably two independent means of measuring mass transferred will be used on top of the usual loading of contingency margins to assure adequate propellant has been placed on board the spacecraft.

For both the OTV and TMS, the condition of the propellant tanks need to be established before transfer can begin. If the OTV has been idle long, its tanks will need to be cooled down before they will accept the new propellants. The TMS tanks will likely need to be evacuated prior to loading. Both of these operations will need support hardware and procedures to accomplish. These procedures are somewhat driven by the use of screen type PMD's which depend upon a gas free liquid fill in the channels.

On board the space station the crew will need to know how the OA/ACS is performing and the systems status. A series of temperature and pressure transducers will be distributed throughout the system to aid in this status reporting. Normally the data will be monitored within the OA/ACS electronics and be displayed only when asked for or when an anomaly is detected. Thruster on times will be accumulated to aid in predicting the useable life left for the thruster. Similarly pressure drop could be used to signal filter replacement (or purge). While not a technology issue (except for mass measurement) in itself, developing and testing the software and integrating it into the space station computer system will require effort early in the design phase to assure a functional system when the space station is deployed.

6.4 ATTITUDE CONTROL SUBSYSTEM (ACS)

6.4.1 ACS Requirements

Attitude Control System (ACS) requirements for SS do not appear to drive extreme stability or pointing capability for the basic station. Special payload requirements can be accommodated by SS through fine pointing or isolation devices either attached or separated from the manned station. Obviously, the solar arrays, radiation panels and communications antennas can be controlled from the relatively uncontrolled massive station so long as its earth centered attitude in known. In order to conserve energy, particularly expulsive fuels which must be delivered from earth, the orbital influences are utilized for basic stability. Gravity gradient and orbital rate momentum torques could provide a natural earth pointing and hold control for the platform. The attitude control system would then provide second order functions for producing torque commands to initialize attitude, to limit rate and attitude excursions, and to provide thrust commands for orbit maintenance. Accelerations, angular rates, attitudes and positions developed by the system would be used to point solar arrays, cooling panels and antennas, and to provide inputs for payload special functions.

6.4.2 Stabilization Analyses and Trade Studies

Orbital configuration stabilization of space station has been evaluated on a rough order of magnitude basis. To do this a "representative" space station was characterized as described in Table 6.4.2-1. Also characteristic values of orbiting parameters were assigned, as listed in Table 6.4.2-2. First order elements of these two tables were used in the following assessments of forces and torques acting on the space station.

a. Atmospheric Drag

Calculations of atmospheric drag forces on space station were made with the simplified equation:

$$\text{Drag} = \rho v^2 (2-f_n) \text{ Ag where } F_n = 0.8$$

Table 6.4.2-1 Space Station Characteristics

Mass - 2.3×10^5 lbs (7.14×10^3 slugs)

Moment of Inertia

I_{xx} - 2.9×10^7 slug ft^2
I_{yy} - 2.9×10^7 slug ft^2
I_{zz} - 8.3×10^6 slug ft^2

Areas
Solar Arrays - 23,000 ft^2
Radiators - 2500 ft^2
Modules - 2900 ft^2

Offsets

Center of Solar Pressure to Center of Mass-10 ft.
Center of Drag to Center of Mass-10 ft.

Hydrazine Reaction Jet I_{sp} - 228 sec.

Table 6.4.2-2 Orbit Conditions

Altitude (h) - 250 NMi

Inclination (i) - 28.5°

Solar Pressure - 1×10^{-7} lb/ft^2

Atmospheric Density (ρ) - 1×10^{-14} slug/ft^3 - high average

- 5×10^{-16} slug/ft^3 - low average

Orbit rate (ω) - 1.11×10^{-3} rad/sec

Orbit period - 94 minutes

Velocity (v) - 25000 ft/sec

Deorbit ΔV - 140 ft/sec in 20 minutes

The area (A) was determined for an orbit average with solar arrays following the sun at $\beta = 0^o$ and the radiator edge to the sun and coaxial with the solar arrays.

$$1 \times 10^{-14} (25000)^2 (2-0.8) [(23000 + 2500) 0.637 + 2900]$$

Drag = 0.144 lbs_f - high average density

Drag = 0.0072 lbs_f - low average density

Hydrazine fuel required for orbit maintenance would be (lb_m):

	Atmosphere	
	High Average	Low Average
Per Orbit	3.6	0.18
Per Day	55	2.7
Per Year	20,000	1,000

As noted above the calculations were made for $\beta = 0^o$, a highly unlikely long term condition. As β increases, solar array drag decreases as the cosine of the angle. At an orbit plane inclination angle of 28.5^o, long term drag is only reduced about 2% from the $\beta = 0^o$ case.

b. Drag Torque

At 10 feet offset of drag center of pressure to SS mass center, maximum drag torque is 2 ft.-lb. With reaction jets located 20 feet from the center of mass, an average force of 0.1 lb is needed. This is about the same as the force required for velocity maintenance.

c. Drag Acceleration

Acceleration due to drag for high average atmospheric drag and solar arrays at maximum drag attitude would be:

$$a = \frac{0.2}{2.3 \times 10^5} = 1 \times 10^{-6} g \text{ (approx.)}$$

d. Solar Pressure and Resulting Torque

At 1×10^{-7} lb/ft^2 acting on 23,000 ft^2 solar arrays, force is 0.0023 lb and torque applied at 10 feet offset is 0.023 ft-lb. These values are lower than those for atmospheric drag at low average density.

e. Gravity Gradient Torque

A rough cut four body SS was devised to obtain an order of magnitude gravity gradient torque (pitch axis).

Using the dumbell equation:

$$T = \frac{3GM}{R_3} (m\ell^2)(\sin 2\theta_1 - \sin 2\theta_2))$$

$$T = 0.38 \text{ ft-lb/degree}$$

This indicates the torque due to gravity gradient which would offset other station torques assuming small angles.

f. Momentum

A gravity gradient stabilized SS has a pitch rate of one revolution per orbit. This is represented by a momentum whose vector is along the -Y axis. The momentum of our "representative" SS is

$$I\omega = 2.9 \times 10^7 \times \frac{2\pi}{94 \times 60} = 3.2 \times 10^4 \text{ ft-lb sec}$$

This momentum will stabilize the SS about the yaw axis and can help gravity gradient stabilization of the roll axis.

g. Acceleration (gravity gradient)

Each mass particle of the orbiting SS is acted upon by opposing centrifugal and gravitational forces. The net effect is that the center of mass of the SS is subjected to equal and opposite centrifugal and gravitational forces:

$$Mg_o \left(\frac{r_o}{r_1}\right)^2 = M\omega^2 r_1$$

From which is determined as $1.111693401 \times 10^{-3}$ rad/sec.

A part of the SS displaced radially from the center of mass has a net acceleration acting on it:

$$a = \frac{g_o r_o^2}{r_2^2} \div \omega^2 r_2$$

The acceleration is about 1.15×10^{-7} g per foot of radial displacement from the center of mass.

h. Man Push Off Acceleration

If a man were to push off at one end of a rigid SS, the acceleration produced would be the sum of SS linear acceleration and the linear affect of angular acceleration

$$a = \frac{F}{M} + \alpha \frac{\ell}{2}.$$

A 50 lb push off at 100 feet from the center of mass would result in 0.00075g. For a flexible SS, the acceleration would be locally greater, diminishing with distance from the push off point. Since he cannot travel far, the impulse will be nearly cancelled by his stoping impulse. In any event, over a period of random movement, the accelerations should null to zero.

i. Thruster Acceleration

Thruster force will be 10 to 30 lb and the displacement from the SS center of mass will be less than the man push off discussed above. The thrusters will probably be "soft" mounted to main sections of SS. Accelerations caused by thrusters will, therefore, be about an order of magnitude lower than for man push off.

j. Docking (Berthing)

Docking or berthing of another space element to the SS could encompass the range from virtually no impulse through a catastrophic "crash." Mechanisms and maneuvering procedures will have to be developed which preclude damaging either spacecraft. They may also be designed to produce impulses below annoyance levels to personnel and disturbance levels to payloads, antennas and solar arrays.

k. Miscellaneous Internal Disturbance

Effects of SS disturbances or payload moving systems (including drives, pumps, and fans), liquid motion (sloshing and transfer) and venting will all require special attention. Designs must consider the impact on all other SS and payload systems. A basic approach will probably be to design for interfaces well below the man disturbance level.

l. Controlled Deorbit

For emergency deorbit or planned eventual decommissioning of the SS a controlled ΔV will be required. If it assumed that 140 ft/sec must be imparted in 20 minutes time the thrust force is:

$$F = 7.14 \times 10^3 \times \frac{140}{20 \times 60} = 833 \text{ lbs}$$

Fuel required is $\frac{833 \times 20 \times 60}{228}$ = 4400 lbs hydrazine

6.4.3 ACS Conceptual Design

A simplified block diagram for the SS attitude control elements could be as shown in Figure 6.4.3-1. Miscellaneous sensors for specialized payload control and isolating systems, as well as platform bending error detection are not shown. Also, the solar array may require bending mode sensors for dynamic damping control. As shown, three axis rate gyro information is processed to provide rates and attitudes in the attitude control processor. Two axis earth sensors produce earth centered attitude updates. This two angle system (pitch and roll) is supplemented by gyrocompassing for yaw angle. GPS inputs provide accurate earth centered positioning and timing. From this system, sun, stars, planets and the TDRS can be targeted.

The early SS configuration may consist of only a power system. In this case the power system attitude control system would be used as long as appropriate as suggested in Figure 6.4.3-2. Eventually the CMG's will wear out over one to two years. The magnetic torquers would be undersized and the thrusters would not operate about the evolved SS center of mass. This could lead to dismantling and restructuring for the full-up station.

6.4.4 ACS Conclusions

The SS will be in low earth orbit of about 250 NM to provide realistic revisit capability for the Orbiter. At low earth orbit, aerodynamic drag will be a significant and inescapable disturbance which must be overcome. Efforts can be made to minimize drag but it will be ever present, and with large solar arrays tracking the sun it will be appreciable. Calculated for a "representative" SS at 0° beta angle, it could be 0.007 to 0.14 pounds average over an orbit for solar effects between low and high average.

Using known technology, expulsive devices are needed. In the future, ion or other electric propulsion may be developed for an evolved SS; however, in its earlier forms, its drag makeup will be accomplished with 10 to 30 pound force reaction jets.

Given that a propulsion system is required for drag makeup, it can be sized and distributed to provide attitude control simultaneously by firing only +X thrusting jets. The jets are located on the structure or booms at 10' to 30' from the average center of mass which varies with SS buildup. Firing a rear pointing jet, coupled with the center of mass, produces a ΔV and a moment simultaneously.

Torques acting on a "representative" SS have been assessed. They derive from drag and solar pressure offsets from center of mass, gravity gradient, magnetic dipole, mechanism motions, fuel and other liquid slosh, venting, docking and random motions of the crew. Some of

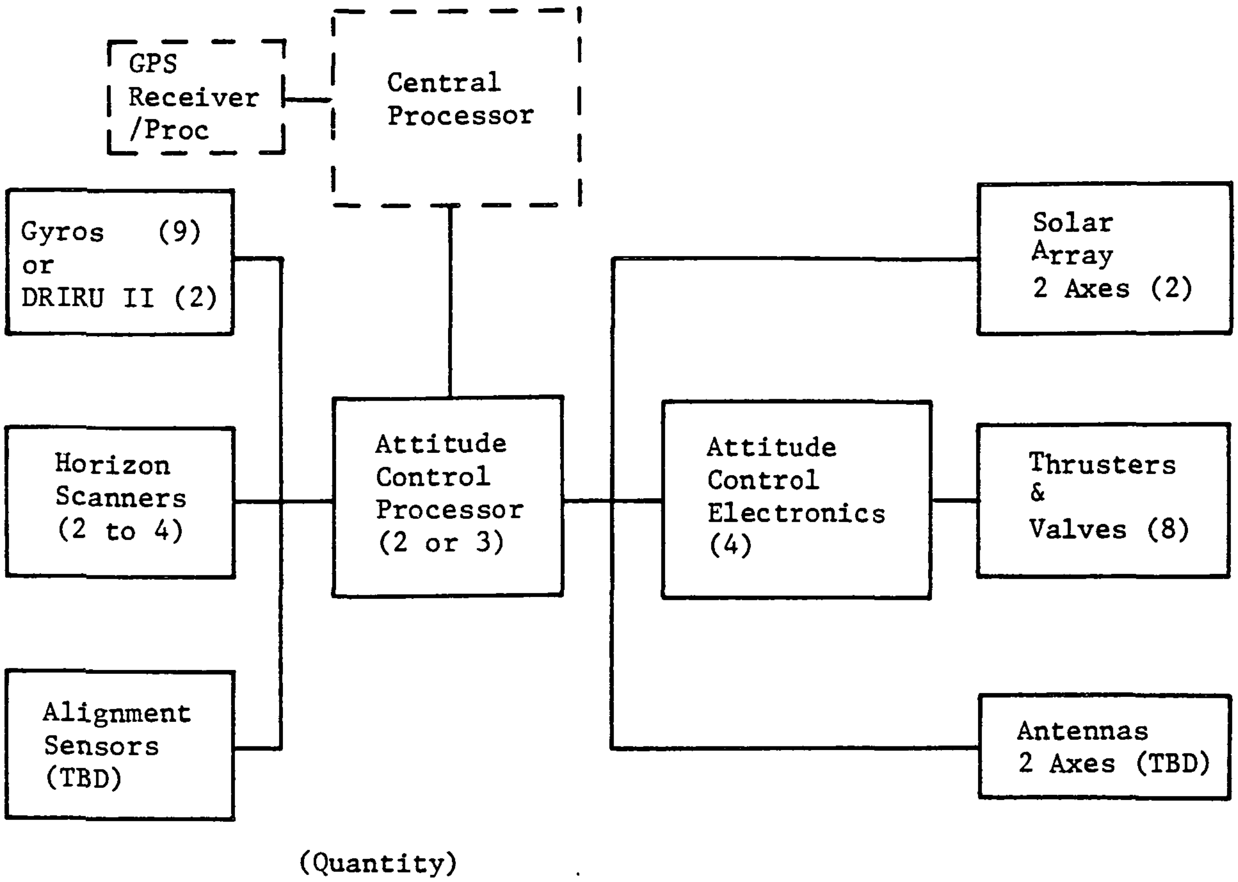


Figure 6.4.3-1 Space Station ACS Concept

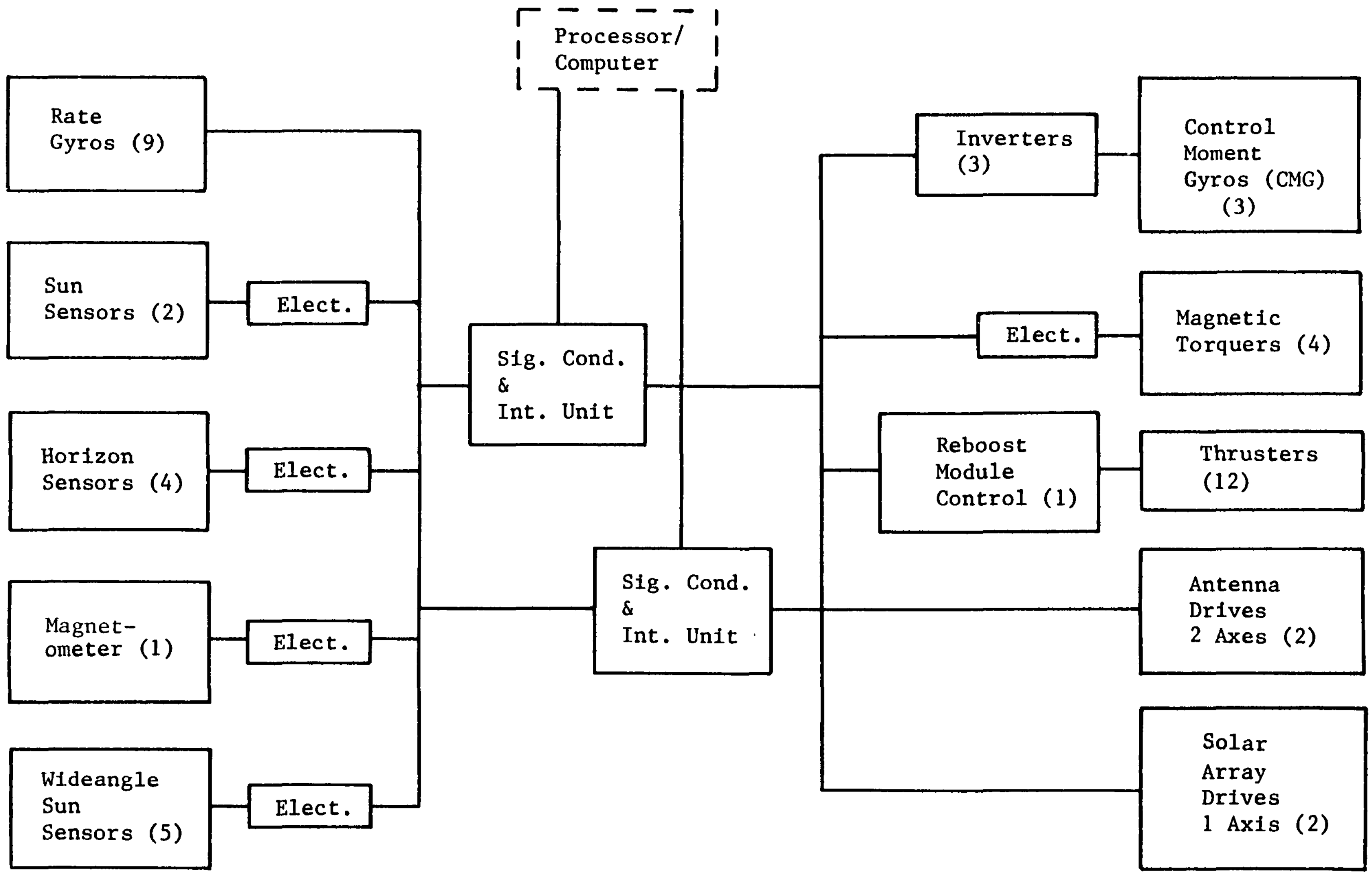


Figure 6.4.3-2 Candidate Power Module ACS

the small cyclic torques could be accommodated by momentum storage devices; however, continuous momentum buildups would have to be countered (desaturated) by large magnetic torquers or the propulsion system.

The most viable approach to attitude control is to configure the SS at each evolutionary step of buildup for minimum moment of inertia about the earth radial and maximum moment of inertia perpendicular to the orbit plane. Gravity gradient then torques the SS to long axis vertical attitude and the one revolution per orbit momentum about the major principal axis maintains yaw angle. Small variations in attitude caused by miscellaneous torques will be tolerated. If a specific disturbance is too great, the reaction jets will be turned on to damp the effect.

Control laws for the reaction jet system will be modified to optimize control of the flexible SS as the mass properties and dynamics vary from changes in expendables, evolutionary buildup and attachment/separation of the Orbiter and other free-flyers.

6.5 RF COMMUNICATIONS

6.5.1 Communication Requirements

The space station must have the capability to establish an RF interface with a variety of orbital spacecraft. A summary of the required interfaces is presented in Table 6.5-1, together with data on the expected or proposed operating frequence range, communication/data signals, data rates, and maximum operating range. The following rationale also supports identification of these requirements.

a. TDRSS - It has been groundruled that the TDRSS will provide the primary communications interface between the space station and ground support facilities.

b. STDN - An interface with the STDN tracking sites provides a communications back-up to the TDRSS. Because of the limited sites available and very brief communication contact periods, this is considered an emergency, rather than nominal, support capability. It is estimated that contact with the STDN will be possible for no more than 10-15% of the time.

c. Orbiter - Communications will occur between the station and orbiter during rendezvous periods prior to the docking of orbiter with the SS.

d. GPS - The SS will monitor navigation transmissions from the Global Positioning System (GPS) satellites in order to maintain very accurate position-in-space knowledge and universal time updates.

e. OTV - Communications with the OTV will be required to support rendezvous and docking with the SS and also to support launch and retrieval of the OTV.

f. EVA - Since extra-vehicular activity (EVA) will be required by the SS crew, an RF link will be required for monitoring of the crewmen and EVA systems.

g. Intra-SS - As the SS evolves into its mature configuration, it will be desirable to provide low power RF links to communicate with remote sections of the station. This interface may include communications with payloads attached to the SS.

h. Platforms - Payload platforms operating in the vicinity of the SS will be capable of maintaining a continuous link between the SS and platform. This link will enable control of payloads and the transfer of payload data to the SS for integration and incorporation of the data into the TDRSS downlink. For the significantly higher payload rates, signal processing will be required at the SS before the data can be transferred via the TDRSS.

Table 6.5-1 Space Station RF Interfaces

Terminal	Frequency Band	Data	Rate - BW	Range
TDRSS	K S	Status Commands Video Voice Science Tracking	50-100 Kbps 1 Kbps 10-50 Mbps 48 Mbps 1-300 Mbps ---	25000 NM
STDN	S	Status Commands Voice	50-100 Kbps 2 Kbps 16-32 Kbps	1750 NM
GPS	L	Nav	1 Mbps	12000 NM
Orbiter	K S	Radar Voice	NA 16 Kbps	15 NM 100 NM
OTV	S	Data	10 Kbps	100 NM
TMS	S	Data Commands Video	2 Kbps 1-2 Kbps 3 Mbps	1500 NM
Intra-SS	TBD	Data Commands	TBD	500 FT
Docking Vehicles	TBD	Radar	TBD	20 NM
EVA	UHF	Data Commands Video	1 Kbps	
Platform	S-K	Data	10 Kbps-300 Mbps 1 Kbps	1000 NM

i. TMS - The TMS will often operate within line-of-sight of the SS, but will, at times, operate over-the-horizon. When possible, the SS will control the TMS; and over-the-horizon control will be transferred to the TMS ground control. Very early in the program, two TMSs will be based at the SS, and can be operating simultaneously.

j. Rendezvous Radar - A variety of vehicles will be docking with the SS on a regular basis, including the Orbiter, TMS, and OTV. This radar may also be required to support initial tracking and retrieval of the OTV.

6.5.2 Conceptual Design Analyses

Link Analyses of several of the interfaces contained in Table 6.5-1 indicate that nominal RF power levels between 10 and 25 watts and omni-directional antennas or small parabolic antennas (less than 3 feet diameter) will satisfy several of the requirements identified. This includes the data/command interfaces required for the Orbiter, TMS, EVA, GPS, STDN, and intra-SS links. Since the primary intent of our subsystem analyses is to drive out SS configuration drivers, attention was then focused on the TDRSS and the platform links to determine what antenna sizes were required.

6.5.2.1 TDRSS Links - Although the downlink (return) data rates shown for TDRSS vary from relatively low rates to as high as 300 Mbps, a nominal rate for this link might be 50 Mbps. This assumes digitally multiplexed science, video, voice, and status data. Assuming a data rate of 50 Mbps, a 20 watt transmitter power level, and quadraphase modulation, the achievable data rate (ADR) on the KSA Link is 33.8 dB plus the radiated power of the SS. Taking into consideration cable losses of 5 dB; link losses of 5 dB (other than the propagation attenuation losses) including pointing error, polarization, and TDRS transponder losses; and a desired margin of +6dB, an antenna gain of +46.2dB is required. This gain level can be achieved at a 15 GHz frequency with a parabolic antenna 6 feet in diameter. Accommodation of the maximum rate of 300 Mbps, would require a parabolic dish size of 15 feet assuming other parameters remain fixed. However, an increase in transmitter power to the 30-40 watt range could reduce antenna size to 10-12 feet.

If it is necessary to uplink wideband data to the SS, this link and its associated antenna may create more of a configuration impact than the downlink. For example, an analysis was performed for uplinking a 25 Mbps signal via TDRSS. Assuming nominal link and cable losses, together with a receiver noise figure of 2.5 dB, results in the need for an antenna diameter of 20 feet to provide the antenna gain indicated. This size antenna could cause problems if mounted on some type of boom, as is likely, to avoid blockage from the SS structure because of the anticipated structural flexibility of a mature SS configuration.

6.5.2.2 Platform Relay Link - Recent space station studies have identified the potential need to transfer wideband data at rates up to 300 Mbps between the SS and a payload operating in the SS vicinity. The payloads involved are either free-flyers or similar payloads mounted on a common, free-flying platform. They generally require an undisturbed environment and have very precise pointing requirements. The long term, and sometimes unpredictable high data rates may place a burden on a TDRS interface. As an option, the wideband data could be relayed to the space station during communications line-of-sight conditions, and be processed before transfer to the ground at a lower rate. Maximum range is estimated to be 1000 NM, and use of the K band is assumed. Assuming a 20 watt transmitter power at the platform, 5dB of cable losses, and 2dB of combined link losses, a SS antenna 3 ft. in diameter would generate a +4dB margin and a bit-error-rate of 10^{-5}. This analysis also assumes the same size parabolic antenna on the platform, although a phased array antenna with the equivalent +40 dB gain may be preferrable and produce less disturbances for the pointing system.

6.5.3 Communication Capability Evolution

The early SS communications capability must support the RF interfaces with the Orbiter, STDN, TMS, GPS, and TDRSS. Since the SS evolution plan integrates high data-rate payloads on the SS within its first year of operation, the full TDRSS capability should be implemented early. Two of the early payloads have unprocessed data requirements exceeding 50 Mbps. The OTV interface will be required about one year after IOC, and it would be cost effective to incorporate this RF system with the initial hardware.

Platform data relay links are not required until 1994, and the current platform payload complement does not indicate the need to transfer data rates in the 100 Mbps range. The interface with the materials processing platform, in fact, is probably 10 Kbps or less.

6.6 SPACE STATION DATA MANAGEMENT AND PROCESSING

The Data Management and Processing Subsystem for Space Station presents some unique requirements for an onboard computer system, while the Space Station's concept of operation and physical features will allow some spaceborne data handling architectural approaches which have been inappropriate or impossible in past programs, none of which have had either the magnitude nor the wide range of processing requirements as Space Station. The function of data management can be broken down into 3 high level activities: the acquisition of data; the processing of data; and the dissemination or distribution of data. A system as large as the Space Station will have several classes of data which must be collected, processed and disseminated, including subsystem status, subsystem and facility control data, sensor status, crew status data, and experiment sensor (raw) data.

The Space Station will be a new generation of space system in the sense that the quantity of data to be managed will be much greater than in any previous system, with physically larger subsystems having proportionately larger monitoring and control requirements and a larger complement of much more sophisticated sensors generating much greater quantities of sensor data. All of the data collected must be processed, with the processing workload being naturally much greater for some types of data than for others such as status and health measurements. The data management and processing system must disseminate the data it has collected and processed, and some of the dissemination will be to ground stations while other data sets will be presented to the onboard crew for evaluation and decision. This last function, an operator-system interface, is a significant extension of previous space programs, and in effect allows us to think of the Space Station as a laboratory and onboard workshop in low earth orbit. A natural point of departure for the onboard data management system, given this analogy, would be the computer system for a ground based laboratory and workshop and thus we come to a high level description of the Space Station's data management and processing system: a multi-user, distributed system, in communication with other remote computers, collecting data from experiments and controlling the sensors. In addition, the conceptual system has real time control functions such as controlling a free-flying satellite or a Teleoperator Maneuvering System (TMS), which are similar to real time industrial process control, also typical of a ground based workshop.

6.6.1 Data System Requirements

The approach taken in the development of data handling and processing concepts for the Space Station System was to consider the end to end system as a whole. Our concept is that the ground segment and spaceborne segment should operate together as a single distributed processing system, with all processors in the system interconnected by communications system which appears logically to the processors as a single unified data bus.

A fundamental assumption made early in this study was that a historical trend in space program data systems would continue. Namely, the gradual transition of data processing functions from ground to the spacecraft. Nevertheless, early in the Space Station's lifetime, ground mission operations will play a proportionally very large role in system operation, with the role decreasing as the flight system builds up incrementally to full capability. Simultaneously, as operating experience allows procedures to mature, the flight crew can assume a greater share of system interface from ground operators. So, we see that the allocation of functions between the ground and flight data systems is one which will change with time.

Figure 6.6.1-1 illustrates a partitioning of high-level Space Station System functions between the ground and spaceborne segments. This figure shows the onboard crew with an important role, interfacing with all elements of the Space Station, with the crew interface being supplemented by the Station Control function. As discussed above, however, the initial partitioning will change with time, although the ground system's total workload will probably stay approximately constant, since the onboard system's share of the data processing responsibility will increase as the Space Station grows incrementally from the first unmanned module through Full Operation Capability (FOC) and beyond.

A series of comprehensive system level trade studies must be initiated early in the Space Station program, with the goal of partitioning functions between the ground and onboard data systems and of scheduling the transition of responsibilities. Functions requiring outside consultation such as long range mission planning would be more appropriate for allocation to the ground segment, while day to day detailed crew planning could easily be accomplished in the onboard segment. Many other functions are appropriately handled by the ground segment while, simultaneously, many are more appropriate to the spaceborne segment. However, in many cases the appropriateness is not readily distinguishable, and must be subjected to further analysis.

Data Systems requirements are heavily influenced by the Station's complement of payloads and sensors at a given point in its lifetime. In particular, the volume of data required to be stored onboard prior to either onboard processing or relay to earth is a direct measure of the onboard mass storage required as well as an indication of the magnitude of the data base management task. Table 6.6.1-1 lists the experiments or sensors to be attached, either on board or off board, to Space Station showing the maximum data rates of the sensors and a total maximum of TBD megabits per second and a total onboard mass data storage requirement of more than 1100 gigabits. Given the amount of data collected of more than 400×10^{12} bits each day, it is clear that the onboard DMPS must be capable of reducing this flood of data to manageable proportions.

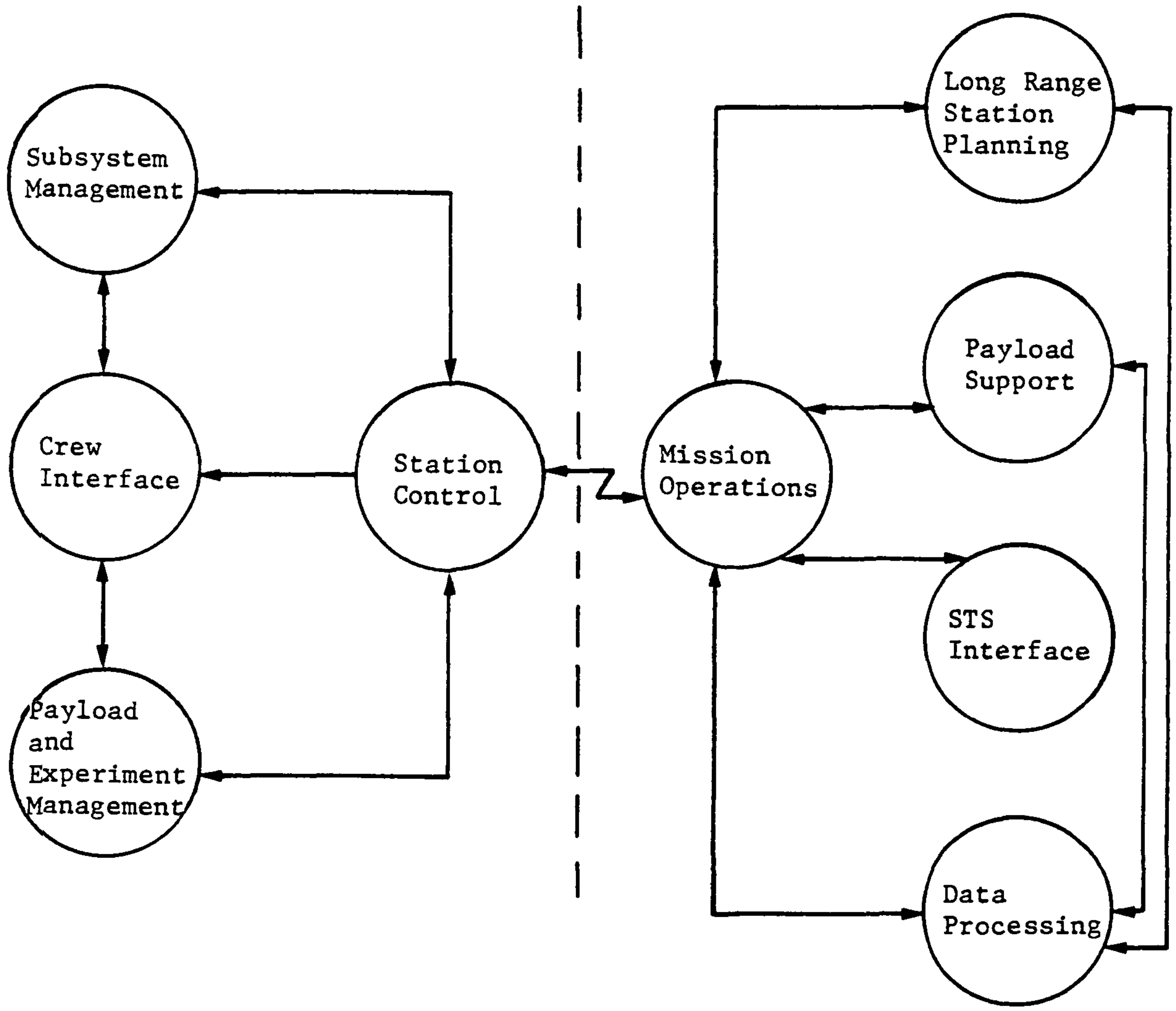


Figure 6.6.1-1 End to End Data System Functions

Table 6.6.1-1. Experiment/Sensor Data

Program	Year	Data Rate Megabits/Sec	Data Volume Gigabits/Day	Data Storage Gigabits
Starlab	1991	8	230	3
SIRTF	1991	2	60	1
LAMAR	1993	10	288	4
FOT	1991	2	60	1
LDR	1996	10	576	10
SOT	1994	1	2	2
Pinhole Camera	1995	1.5	2	2
SAR	1992	100	10^5	10^3
Imaging Spectrometer	1992	300	$3x10^5$	
Passive Microwave	1992	1	10^2	10^2
Satellite Calibrator	1994	1	Negl.	Negl.
Space Plasma Effects	1994	1	Negl.	Negl.
Materials Processing	1992	0.1	Negl.	Negl.
Biological Lab	1991	50	2	40
TOTAL	--	500	$4x10^5$	$1.2x10^3$

6.6.2 Conceptual Design

The design of the Data Management and Processing Subsystem (DMPS) is driven by several important requirements which are peculiar to the Space Station. Primary among these is the requirement that the system be modular and expandable because of the Station's evolutionary development. Additionally, the design must be flexible in order to accommodate a varying complement of experiments and payloads whose characteristics are as yet not well defined.

The overall architecture selected for the Space Station System (onboard plus ground components) DMPS is illustrated in block diagram form in Figure 6.6.2-1. The DMPS is separated into two parts: an onboard segment and a ground segment, each with its own data bus, with TDRSS connecting the two. The key feature of this design is the link adapters whose function is to link the ground and the onboard data busses, making these logically a single bus, giving the astronauts access to any files in the ground system, or vice versa. Mission Operations personnel can access any file or sensor onboard Space Station with the same procedures and commands as the astronauts. This architecture will therefore allow a graceful transfer of procedures between Mission Operations and the Station as operating experience matures and the Station assumes greater autonomy over the system's life time.

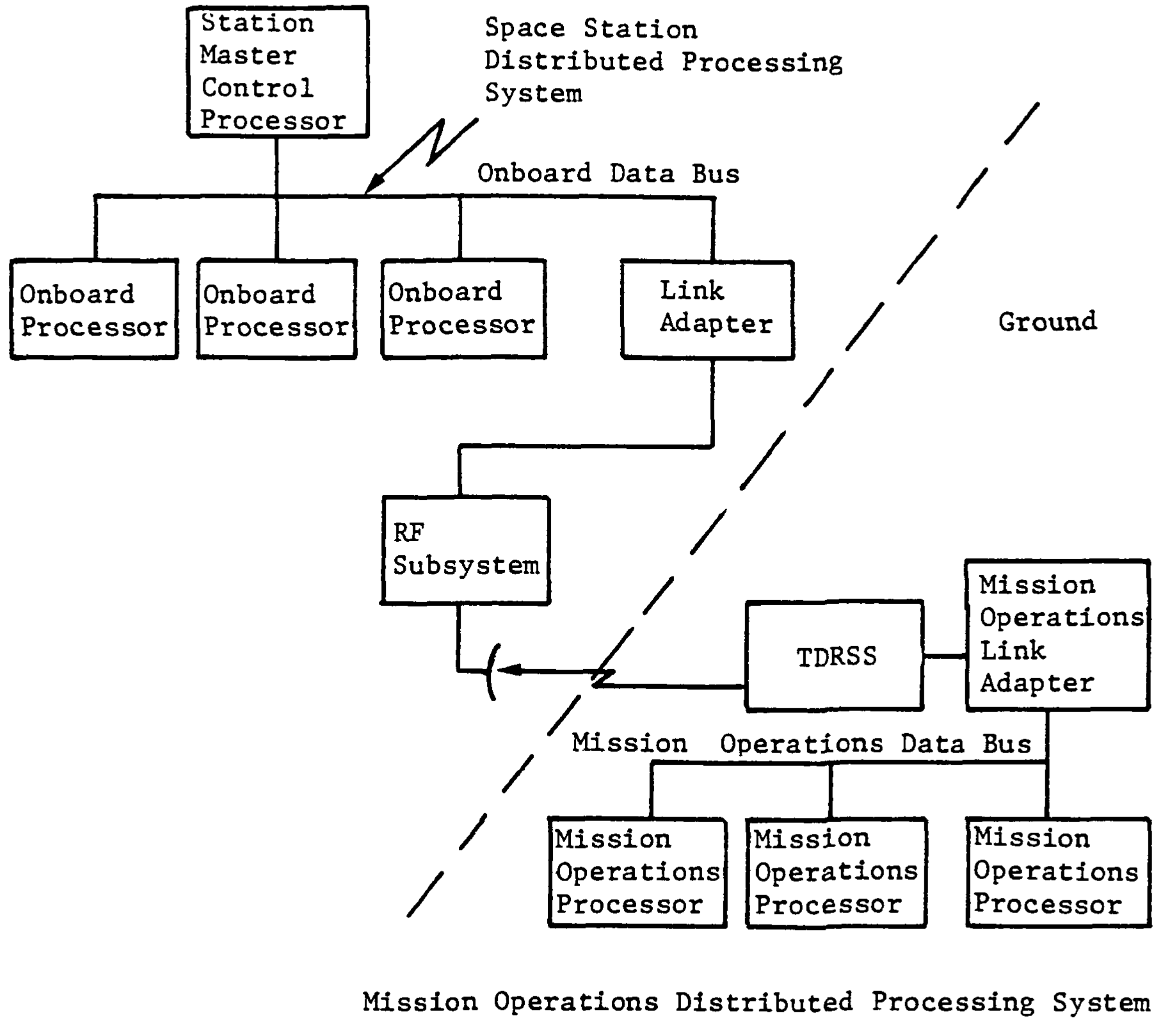


Figure 6.6.2-1 Space Station System Data Management Architecture

6.6.2.1 Architecture Trades - An early decision which must be made in designing a spacecraft data management and processing system is the partitioning of functions between the ground-based supporting processor system and the onboard processor. Factors affecting this decision which at the present time are not well known are the data rate characteristics of the experiments and payloads projected for the Space Station, and the rate of TDRSS bandwidth expansion likely to be imposed during the station's lifetime. Corollary to these tradeoff factors is the cost of operating a ground based data processing system as opposed to maximizing the utilization of onboard processors. The partitioning study of November 1982 by JSC's Data Systems and Analysis Directorate includes some preliminary results, and provides an excellent starting point for a more detailed examination of all of the issues affecting this tradeoff.

The trend in spacecraft data systems has been to shift a larger and larger proportion of the processing and data reduction workload away from large dedicated ground based data processing systems. We have assumed that this trend will continue, and have therefore assigned such functions as subsystem control to an autonomous onboard system, and have assumed that the Space Station, in its fully matured form, will be nearly autonomous and will accomplish a significant proportion of payload data processing in its onboard data processing system. This high level assumption needs to be confirmed before other architectural trade studies and design efforts can be conducted, since the complexity of computations and magnitude of onboard autonomy have a direct impact on the size of the onboard data system. Another trend which has been ongoing for some time, in both spaceborne and ground systems, has been the off-loading of detailed control of sensors and peripherals from the central processor. This is the "smart sensor" concept, and we have assumed that this trend will also continue. The Space Station must therefore have an architecture which is compatible with this concept, and can be easily interfaced to a succession of sensors and payloads and several differing subsystem controllers.

Table 6.6.2.1.-1 summarizes the characteristics of the choices which must next be made in designing the architecture of the Space Station's data processing and handling system. This table shows the elements of a series of tradeoff studies which must be undertaken in order to quantify and put the distributed versus centralized data processing architecture on a firm basis. Some of the elements in the table, however, are not susceptible to any degree of precise quantification and will remain as "qualitative" characteristics favoring one architectural approach over the other. Such a characteristic is Modularity, which is important to the Space Station because of its incremental buildup in function and capability. This characteristic clearly favors a system whose functional partitioning can be approximated by a physical partitioning.

Table 6.6.2.1-1 Data Processing Architecture Factors

	Growth Capability	Modularity	Bus Traffic	Maintainability	Reliability	Adaptability	Automation/ Autonomy	Hardware Cost	Software Cost	Computational Speed
Centralized	Constrained	Poor	Moderately Low	Moderately Complex	Low	More Complex	Difficult	Moderate	Moderate	Limited
Distributed	Easy	Excellent	High	Simpler	High	Simple	Simpler	High	High	No Hard Limit

6.6.2.2 Technical Issues - Tradeoff elements such as those discussed in the previous paragraph can be impacted by carefully structured technology development programs during the early years of system implementation. For example, spacecraft independence from the ground while conducting a variety of experiments implies a need for onboard mass data storage, and some degree of data processing. However, state of the art equipment is presently limited to tape recorders for data storage, and flight computers have limited throughput and a relatively low precision 16-bit word length.

A solution to the problem of onboard mass data storage does not appear to be further development of magnetic tape recorders, but rather in qualifying existing ground-based mass storage devices for the space station mission. The first possibility is magnetic bubble solid state memory devices which have a number of features which make them, conceptually at least, very attractive for space missions. The fact that magnetic bubbles are non-volatile and have no moving parts make them attractive, but unfortunately they consume a great deal of power and are heavy.

Another approach to onboard mass data storage would be the rotating magnetic disk. These devices have seen an explosive growth in use and in technology development for ground based applications, but their utilization in space has been limited by the fact that they are vibration sensitive when operating, and that the head-disk assembly requires an air cushion to maintain the extremely close spacing (less than 50 micron) between the head and the rotating disk. These limitations disappear in the Space Station's habitation modules, however, with their pressurized atmosphere and lack of vibration, and therefore, suitably modified rotating disks, possibly with removable packs, should be adaptable to use in the Space Station. Utilization of rotating disks would also simplify employment of ground-based software operating systems (possibly with suitable modifications), which would have the benefit of greatly reducing software development costs, and would add greatly to the versatility of the onboard computer system.

Table 6.6.2.2-1 summarizes some of the pertinent characteristics of state of the art disk and bubble systems, and, although it is not intended to be an exhaustive comparison, it serves to illustrate some of the points made above, particularly in weight and volume of a comparable storage capacity.

Table 6.6.2.2-1. Comparison of Magnetic Bubble and Disk Mass Storage

	Magnetic Bubbles	Magnetic Disk
Bit Stored	10^7	$3.6x10^9$
Weight (lbs)	26	100
Volume (ft^3)	0.38	2.77
Power Consumption (Watts)	94 Active 15 Stby	150
Access Time (msec)	13	31.3 (Avg)

The second major element of the computer system requiring development is the processor itself. The space station program will, over the course of its lifetime, be required to process massive quantities of data from the many onboard sensors as discussed earlier. A significant proportion of this processing load is certain to be computationally intensive reduction of very wide bandwidth data, such as earth resources imaging sensors, requiring a scientifically oriented computer. There are several potential solutions to this requirement, and studies should be undertaken to determine the most feasible and cost effective approach for the Space Station. A potential solution is to design and develop (or adapt) dedicated signal processors employing the array processing techniques and hardware developed by DOD's VHSIC program. This technology development program should bear fruit in time to be mature for Space Station. A potential solution to Space Station's need for a standard processor is to adapt a commercial super-minicomputer (as typified by Digital Equipment Corporation's VAX family) to the spaceborne environment. Arguments in favor of this approach are similar to those discussed earlier for adapting commercial rotating disk drives: reduced software development costs and utilization of a mature family of very sophisticated software packages and operational procedures. A further advantage of a ground-based computer adaptation is the fact that software and hardware for operating a distributed processor architecture is already in existence, and has been commercially available for several years. This is another area of technology requiring development: definition of an optimum data bus for Space Station. This area should include development of the transmission medium (potentially fibre optics), and the bus adaptors to interface the computers with the bus. In addition, a standard bus protocol must also be defined and standardized in order to insure that the data processing system installed in the initial version of the Space Station can easily be upgraded later in the station's life: the so-called "technology transparency" factor. It is important, also, that the Space Station's data bus protocol be defined and a standard set early in the program, because a variety of systems must interface to the DMPS, each of which will have different development schedules.

6.6.2.3 Recommended Approach - A factor which heavily influences the approach recommended for Space Station's Data Processing and Handling System is the requirement for modularity and incremental buildup in capability. This implies that the space station's data system will be initiated with minimum capability and will grow as modules are added to the space station. The subject of how the data system will evolve during space station buildup is covered in a subsequent section of this report. It is not implied, however, that modularity and incremental build up are the only driving requirements, but rather that they have the greatest impact on the system's overall architecture, and consequently on the recommended approach.

It is felt that electronics for the Space Station can be significantly degraded from conventional space program practice. For example, electronics components for STS are rated to withstand 100 launches, whereas Space Station components will only be subjected to a single launch environment, or possibly as many as 2 or 3 launches in the case of components which failed in flight and were returned to earth for repair. Such equipment could be protected further from its environment, thus opening the possibility of components of near-commercial grade for Space Station's onboard data system.

A logical starting point for the development of the space station processor system is the processor itself, with a primary goal of minimizing life cycle costs for the entire system. Although a number of trade studies whose results could impact the final design are yet to be accomplished, technological trends which are well established permit some broad assumptions upon which a preliminary design can be based. The architecture of our design and its evolution are discussed in some detail in Section 6.6.3, and is based on some principles which are repeated here for emphasis.

First, the baseline computer for Space Station should not be a completely new design optimized for the mission, nor are previous guidelines of long lifetime and extremely rigorous and costly screening of parts and manufacturing quality control required. Rather, the computer's design should be an emulation of a ground computer capable of the Space Station computational workload, with appropriate margin allowances. Alternatively, the upgrade of a commercial (or possible military) computer to the Space Station's environment should be evaluated. Several commercial computers have been ruggedized for the military environment, and this has proven to be cost effective for many military applications. This tradeoff should be accomplished early in the technology development program. As discussed earlier in Section 6.6.2.2, a standard computer in the Space Station based on a commercial system allows use of a commercially based operating system and software packages, including HOL compilers such as FORTRAN and ADA, and a large body of applications packages and operating experience, all of which combine to reduce software development costs.

The second departure from standard spacebased system practice is the use of disk-based operating procedures. This recommendation is really an extension of the previous recommendation, since all medium scale or larger ground-based systems use disk-based operating systems and procedures, and adaptation of these procedures and systems will allow the Space Station to take advantage of the wealth of experience and operational maturity developed over many years in ground systems. Hardware required for the system described above must be of the mass storage, random access type, as typified by the rotating magnetic disk seen as a part of nearly every ground computer system. Rotating magnetic disk technology has been progressing, and continues to progress, at a very rapid rate and should be adaptable to the Space Station environment. An alternative to rotating magnetic disk is the optical disk. This technology is just emerging, and at the present is limited to read only applications, so the technical risk of an optical system is considerably greater than magnetic disks. Optical disks have the considerable advantage, however, of much greater data density per unit of storage medium, thus reducing onboard weight and space requirements.

6.6.3 Evolution

One of the Space Station's key features is the fact that it will build up, or evolve, over a extended period of time. This means that system capability will start from some minimum configuration to a fully expanded system, as operational experience matures and the station's payload of experiments and sensors increases to a maximum.

The Space Station will evolve starting with the first (unmanned) energy module placed in orbit. This first increment will include a minimum capability data processing system, since the first increment requires support primarily in attitude control and status and health monitoring of the included subsystems. Figure 6.6.3-1 is a simplified block diagram of the on board processor system required by the energy section placed in orbit on the first launch. The primary goal in designing this system was to install a computer to fulfill the minimum requirements of attitude and electrical power control and monitoring, and to be fully compatible with the growth versious as the space station incrementally builds up on subsequent launches. During the lifetime of the initial increment, the station would be under full time ground control, thus the onboard processor would require only minimal computational capability. The key feature is its capacity for being integrated into a distributed system during subsequent station upgrades. The figure shows our estimates of the data flow within the system composed of Status and Health (S&H) data, both serial and discrete, Command messages, both serial and discrete, and feedback data such as solar array position in serial digital form. It should be pointed out that this initial system will not require onboard mass data storage, and consequently none has been included.

Figure 6.6.3-2 illustrates the first incremental upgrade of the Space Station's DMPS. This upgrade will occur, according to our evolution schedule, during the second and third STS launches, which will assemble the first habitation module and the TMS/Servicing area to the Station. Note that the Station is still unamanned, so command authority for the system still resides in Mission Operations. This version of the DMPS has a redundant data bus and two (one backup) Station Control Processors, one of which will be located in the Habitation Module, with the second located in the Space Station's Safe Haven, for accessibility in the event of an emergency requiring crew evacuation.

The key feature of this first upgrade can be appreciated by comparing Figures 6.6.3-2 and -1: this is a major upgrade to the primitive DMPS placed in orbit on Launch #1, and we have now implemented a distributed system, although without any operators onboard the Station. This first upgrade includes mass storage, and the system can be exercised and tested from Mission Operations, using the concept of a link adapter to link the ground data bus with the onboard data bus as discussed earlier.

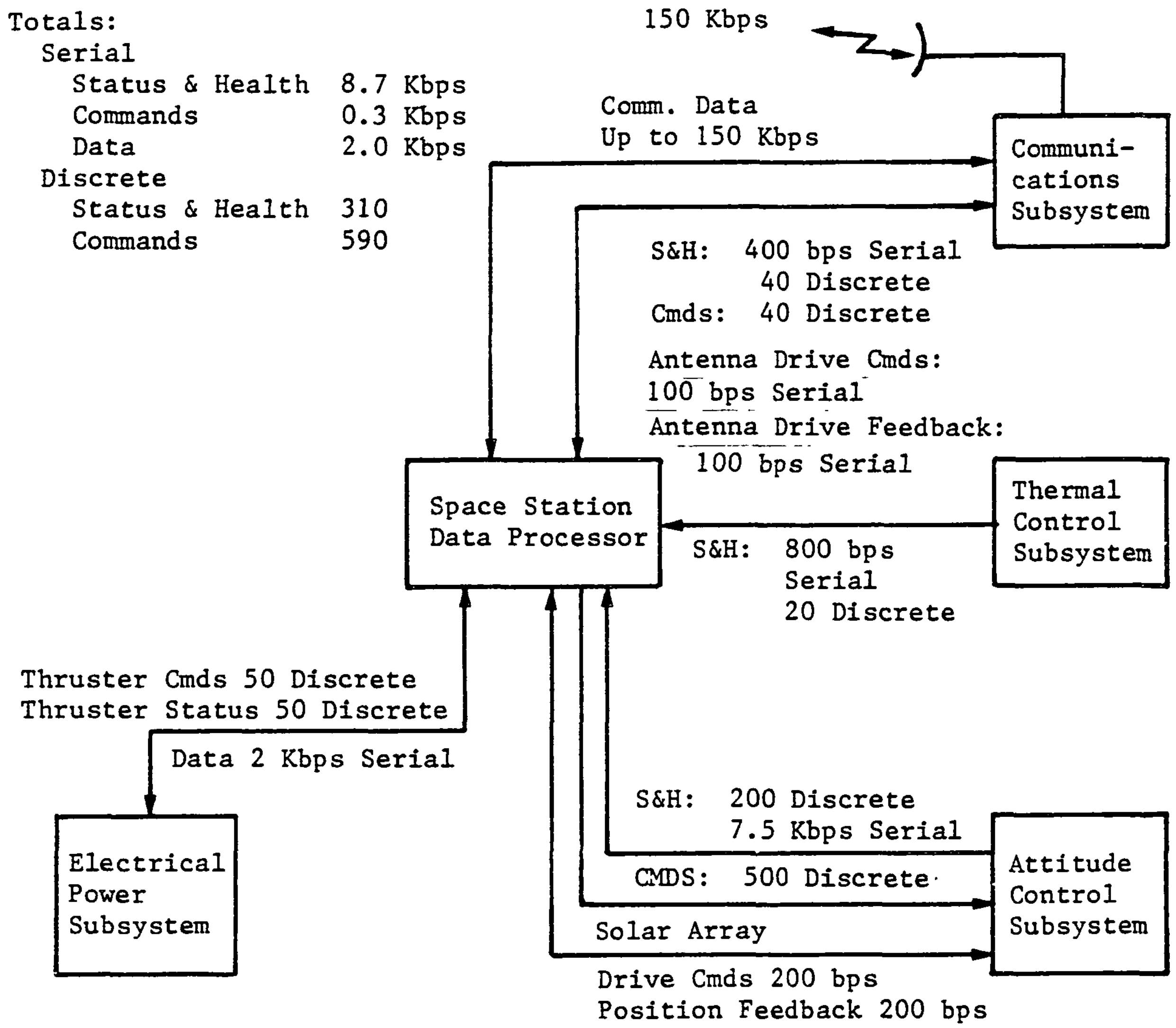


Figure 6.6.3-1 Data Processing System for First (Energy Module) Launch

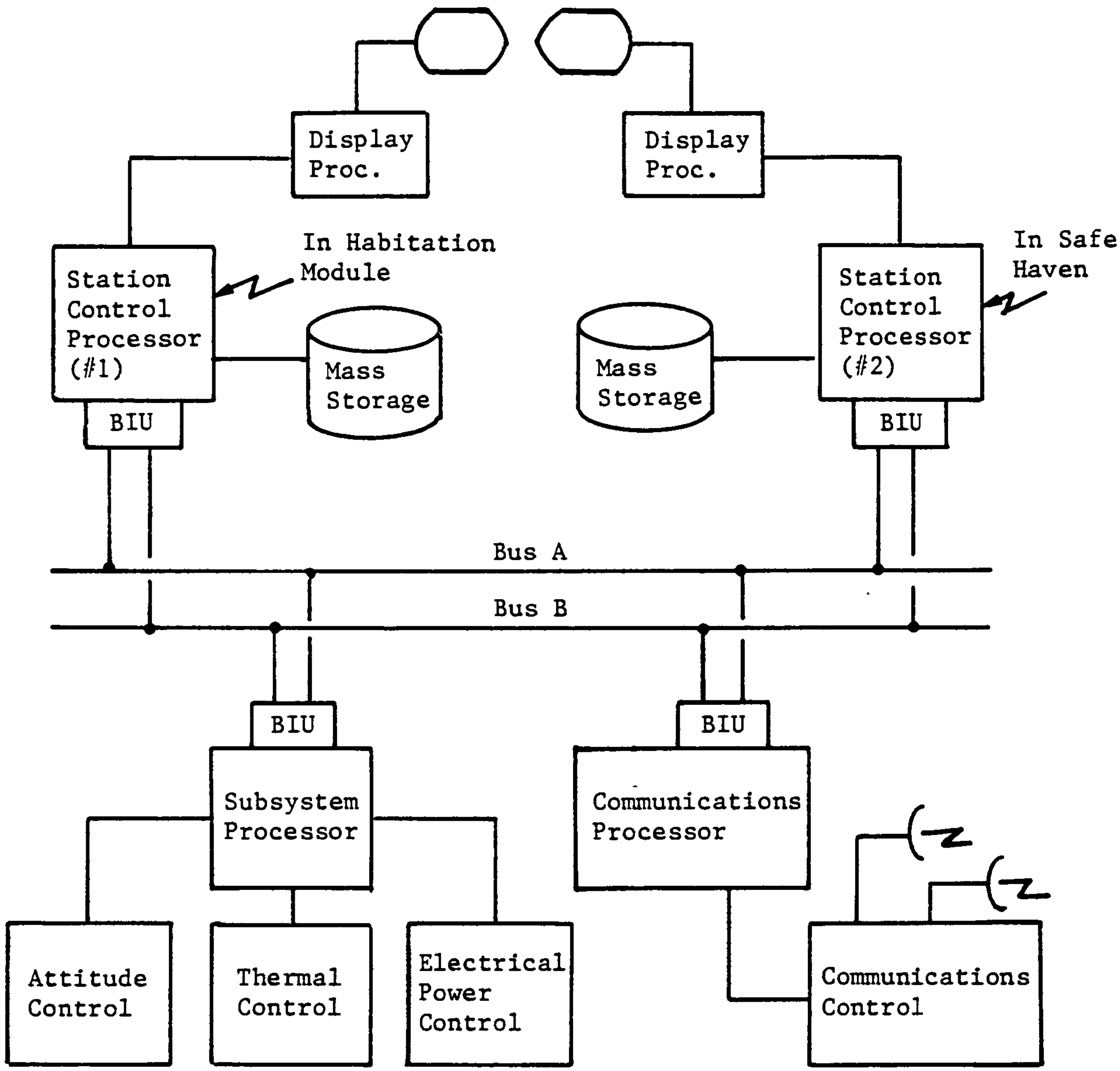


Figure 6.6.3-2 First Incremental Upgrade

The second incremental upgrade of the DMPS is illustrated in Figure 6.6.3-3, showing an additional processor added to the bus. This processor's primary function will be to control the Remote Manipulator System (RMS), with the added workload of test, checkout and control of the TMS. Since this upgrade will also be the Space Station's IOC, the operator-system interface will be required for the first time, and the added processor will give the added processing power required for the RMS and TMS functions.

The next two launches (launches 5 and 6) will add another processor to Space Station's DMPS to control the various sensors, experiments and payloads included on these and subsequent launches. Figure 6.6.3-4 shows another processor added to the bus, with unspecified experiments, one of which would be a Materials Processing Laboratory, implemented on the next STS launch (#7).

The previous discussion illustrates the character of the Space Station's DMPS evolution: a rapid buildup of hardware from the first launch with limited capability to a fully distributed system with substantial performance margins by Station IOC at Launch #4, approximately 1 year later. As operational experience with the system grows and the Station achieves a larger degree of autonomy, with the number of onboard control and data reduction functions increasing, the system performance margins will decrease, until eventually a processor is added to the data bus, with this process being repeated when necessary.

An extermely important function to be added to the Space Station will be the implementation of the Orbital Transfer Vehicle (OTV) by STS Launch 11 in 1992, approximately 2 years into Space Station's lifetime. Test, checkout and control of this vehicle will be a major set of new functions for the DMPS which might be assigned to the RMS/TMS control processor, depending on whether the TMS and OTV are tested, checked out and/or operated simultaneously. If they are, then an additional processor can be added to the bus, dedicated to OTV support.

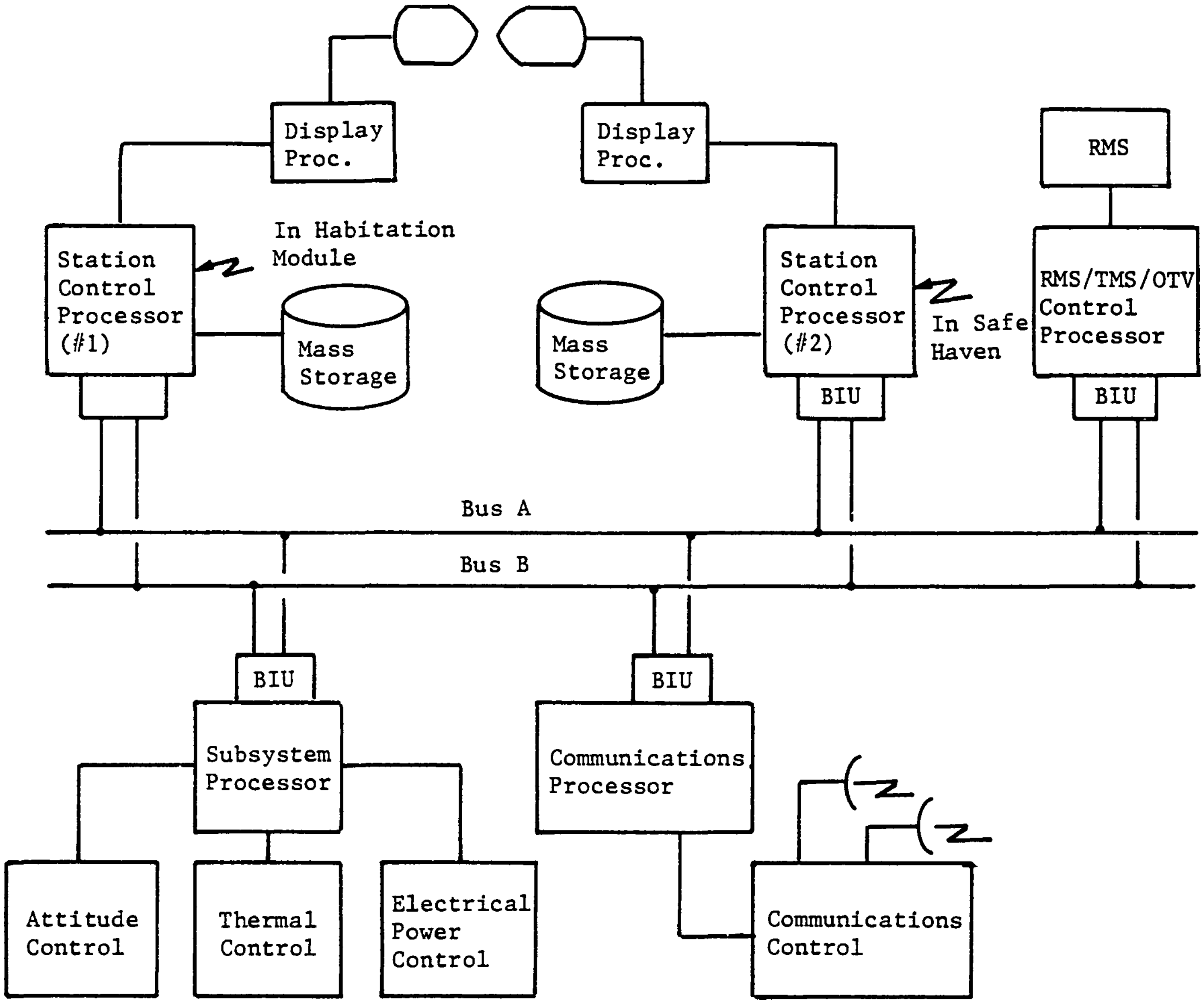


Figure 6.6.3-3 Second Incremental Upgrade

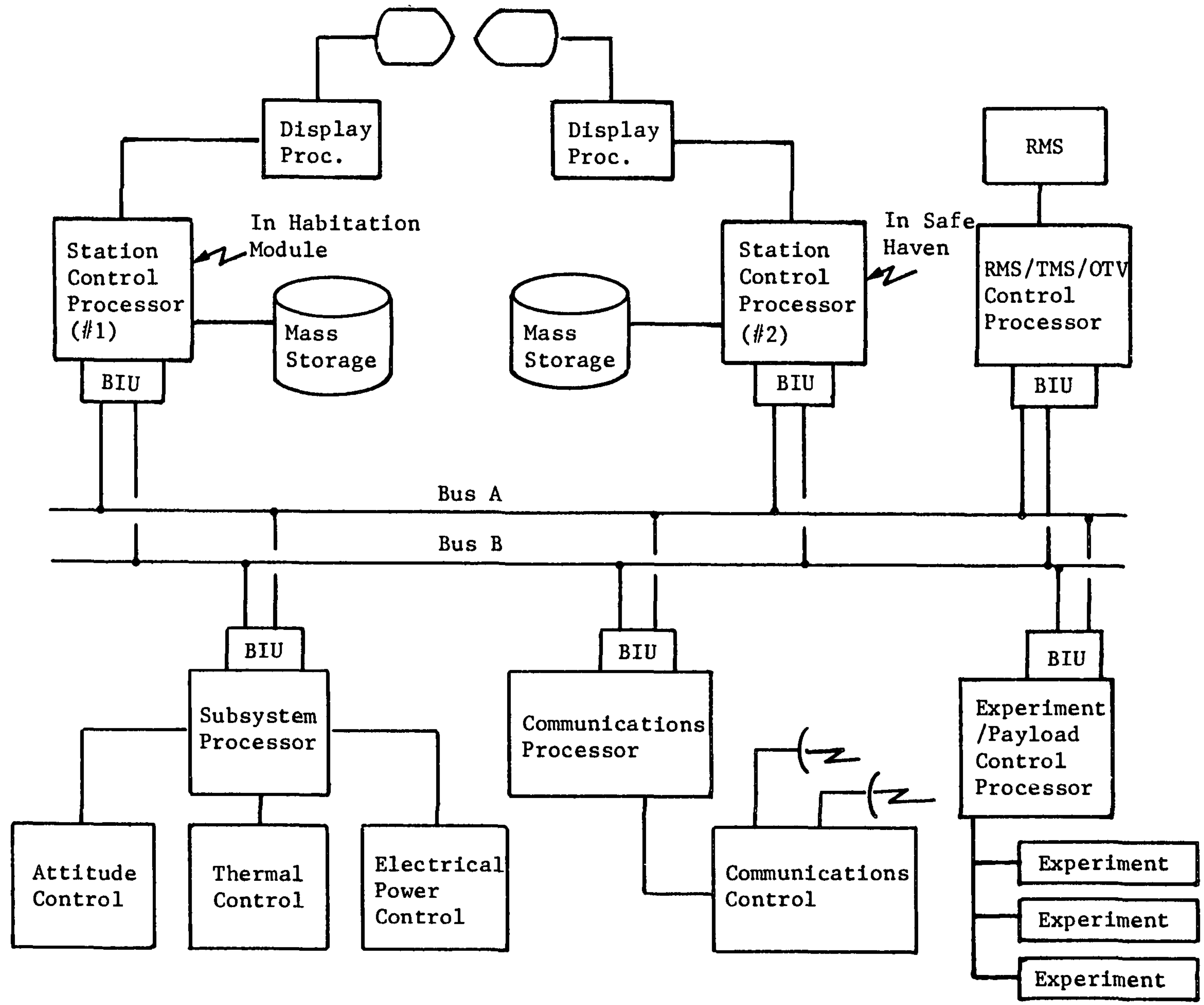


Figure 6.6.3-4 Third Incremental Upgrade

7.0 SPACE STATION EVOLUTION

7.1 EVOLUTION PLAN

A detailed evolution plan has been developed for the recommended space station program option, the manned station operating at 28.5° in conjunction with several unmanned platforms. The proposed evolution plan is presented graphically in Figure 7.1-1. The following commentary will present supporting rational on a year-by-year basis.

a. 1990 Implementation of unmanned station elements is initiated in the second half of 1990 with delivery of the energy section, habitability module including a category II health maintenance facility (HMF), and a TMS. For the SDV architectural option, delivery of these items would be delayed and combined in a single launch with the items implemented in 1991.

b. 1991 Space station IOC will occur early in 1991 with delivery of a logistics module, MMU, servicing robotics, and the initial crew of four people.

Following station checkout and a brief learning period, scientific payloads will be delivered for attachment to and operations from the station. These payloads include:

1. SAR/Passive Microwave (Earth Observ.)
2. Imaging Spectrometer (Earth Observ.)
3. Satellite Calibration (Earth Observ.)
4. Solar Optical Telescope (Solar Physics)
5. Solar Soft X-Ray Telescope Fac. (Solar Physics)
6. Starlab (Astronomy)
7. SIRTF (Astronomy)
8. Space Plasma Effects (Space Physics)
9. EOS (Materials Processing) (2)

Toward the end of 1991, a materials processing (MP) laboratory will be implemented for MP research and development activities. Servicing and resupply of earlier free flying MP payloads operating in a 28.5° orbit will also be initiated using the TMS.

c. 1992 In preparation for the initiation of OTV operations, a cryogen storage tank and a second TMS will be implemented. Because of the crew support for OTV operations, a second habitability module is implemented, followed by the retrievable OTV; and OTV delivery of NASA and DOD payloads to LEO and GEO will begin during the third quarter of the year.

An additional 2 MP payloads will be supported on the station for a total of four.

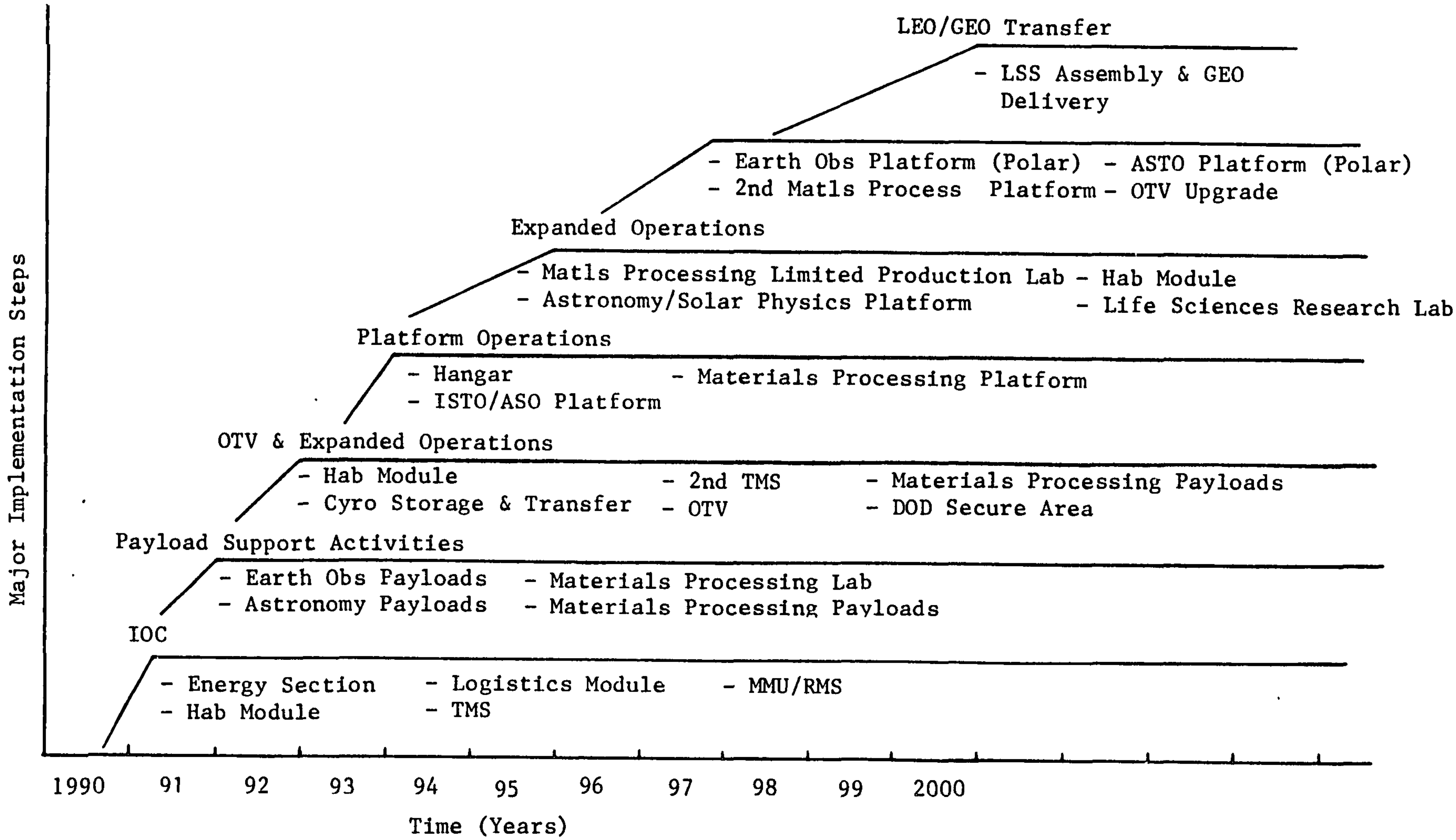


Figure 7.1-1 Recommended Evolution Plan

With availability of the OTV and associated increased DOD operations, it may be necessary to add a secure area or module at this time.

The high level of activity scheduled for this year precludes implementation of a hangar until early 1993.

The OTV activities will continue in subsequent years at a level of 1 or 2 OTV missions monthly.

d. 1993 Early in this year, hangar assembly will begin and continue intermittently through much of the year, interspersed with other activities.

The combined ISTO/ASO platform will be implemented at a 57° orbit, with future servicing support from the 28.5° station via OTV transfer.

A MP platform and MP payloads will be implemented and begin operations in the vicinity of the station, and regular TMS resupply missions will be intiated.

e. 1994 The MP laboratory will be expanded to include a limited production facility which will allow increased production for the more promising processes without full commitment to a complete payload.

An Astronomy/Solar Physics platform will be implemented and operate in the vicinity of the station with continuous communications possible between the two. The platform will support four astronomy and four solar physics payloads between 1994 and 2000.

f. 1995 A life sciences research module will be implemented to conduct plant and animal experiments. A third habitability module will be implemented to accommodate a total crew of 12 people.

g. 1996 A dedicated Earth Observations platform will be implemented in a polar orbit, and will be integrated and supported by the STS since our recommended OTV will not be capable of 28 to 90 orbit plane transfer.

h. 1997 A second MP platform may be required at this time to accommodate commercial payloads whose processes were previously developed in he MP laboratory and limited production facility. This platform will operate in the vicinity of the station and be supported with regular resupply missions using a TMS.

An OTV upgrade may be appropriate at this point to either increase payload delivery capability or to add a thrust control capability which will allow the OTV to carry sizable, but flexible payloads or platforms from LEO to GEO.

The ASTO space physics platform will be implemented in a polar orbit and receive further support from the STS.

i. 1998 The earth observations Passive Microwave payload will require on-orbit assembly support at or near the space station, and will be transported to GEO by the OTV.

At about this point in time, crowding of the available GEO communications satellite orbit may require assembly of a multi-payload platform at the station and subsequent OTV delivery to GEO.

j. 1999-2002 During this period, the GEO-STO space physics platform will require assembly at the station and OTV delivery to GEO.

Similar support will be required by the space physics Very Large Radar.

7.2 CREW SUPPORT AND SIZING

Based on the space station evolution plan described in section 7.1, the following five areas of crew support have been identified:

a. Space station maintenance and evolution and TMS operations.

- Initial checkout and continuing maintenance.
- Crew participation in station evolution, e.g. module additions, hangar assembly.
- Checkout, fueling, and control of all TMS missions.

b. Payload Support

- Scientific payloads attached to the station or on platforms in the station's vicinity.
- Life sciences activities and crew medical/physiological support.
- MP laboratory, attached payloads support, and handling of raw/finished materials.

c. OTV Operations

- OTV refurbishment
- Payload integration
- OTV checkout
- Refueling
- Launch, mission operations, and retrieval

d. DOD operations.

e. Large Structure Assembly

The estimated crew manpower level required to provide the above support is presented graphically in Figure 7.2-1 for the 1991 through 2000 period.

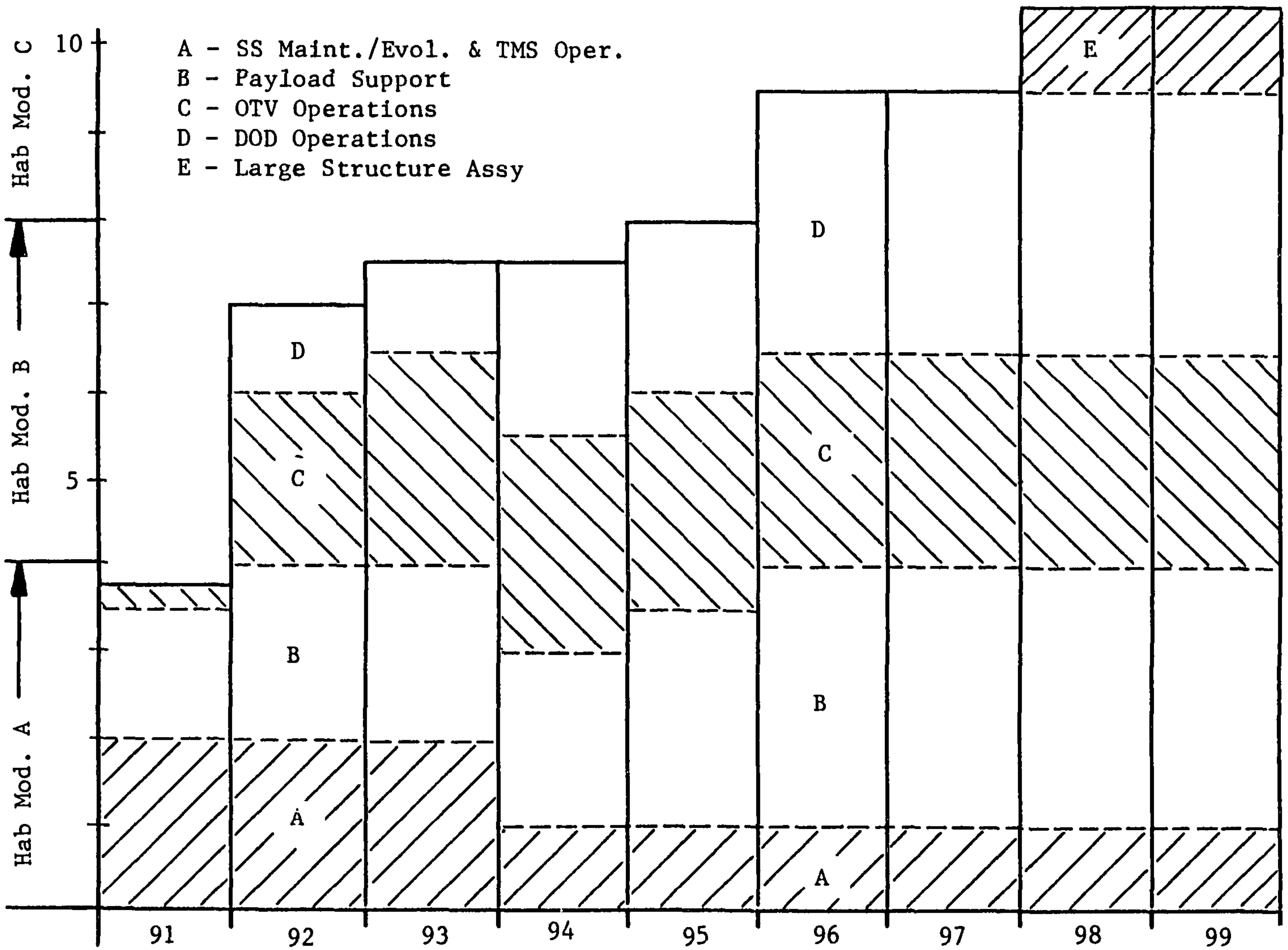


Figure 7.2-1 Crew Activities Profile

7.3 STATION CHARACTERISTICS AND CAPABILITIES

The evolving space station capabilities and significant characteristics are summarized in this section. Overall space station growth is indicated in terms of the physical modules or elements, special station facilities available for user support, and payload platforms supported by the station. Capability growth is also measured in terms of OTV missions and payloads supported, as well as in terms of the crew size available to carry out station and payload activities. Finally, significant subsystem characteristics and capacity is defined throughout the station's evolution.

The space station initial capability and characteristics are presented in Table 7.3-1 for the period from IOC through the end of 1991. Capability growth for the intermediate period from 1992 through 1994 is summarized in Table 7.3-2; and the mature station capabilities for the period between 1995 and 2000 are contained in Table 7.3-3.

Table 7.3-1 Space Station Initial Capability

Capability	Period IOC - 1991	
Modules/elements	Energy Section[1] Logistics Module Habitability Module[1]	Airlock
Special Facilities	TMS, MMU, Robotics Materials Processing Laboratory Health Maintenance Facility (Category II)	
Platforms	-------	
OTV Operations Missions /Yr. Payloads/Yr. Propellant Storage	-------	
Crew Size	4	
Electrical Subsystem		
Bus Load	33.5 Kw (Day)/28.6 Kw (Night)	
S.A. Power	76 Kw BOL/72 Kw EOL (1991)	
S.A. Size (Area)	6430 ft^2	
Power Groups	5 with 500 ampere-hours (batteries)	
User Support	20 Kw	
Thermal Loads		
SS Systems	11.1 Kw	
ECLS	8.3 Kw	
User Payloads	16.4 Kw	

1 - In SDV configuration, a separate module is not req'd.

Table 7.3-2 Space Station Intermediate Capability

Capability	Period: 1992 - 1994
Modules/Elements	Habitability Module #2[1] Tunnel[1] Hangar[1] Cryogen Propellant Storage Energy Section #2[1]
Special Facilities	TSM #2 Materials Processing Limited Production Facility
Platforms	ISTO/ASO Astronomy/Solar Physics Materials Processing
OTV Operations	
Missions/Yr.	16 missions/yr. (ave.)
Payloads/Yr.	24 payloads/yr. (ave.)
Propellant Storage	70,000 lbs
Crew Size	7-8
Electrical Subsystem	
Bus Load	60 Kw (day)/49 Kw (Night)
S.A. Power	137 Kw BOL/127 Kw EOL (1994)
S.A. Size (Area)	11,900 ft^2
Power Groups	8 with 800 amp-hours (batteries)
User Support	31.4 Kw
Thermal Loads	
SS Systems	20.2 Kw
ECLS	16.3 Kw
User Payloads	24.5 Kw

1 - In SDV configuration, a separate module is not req'd.

Table 7.3-3 Space Station Mature Capability

Capability	Period: 1995 - 2000
Modules/Elements	Habitability Module #3[1] Life Sciences Research Laboratory
Special Facilities	Health Maintenance Facility (Category III)
Platforms	Materials Processing Platform #2 Earth Observations Platform (Polar Orbit)
OTV Operations	
Missions/Yr	19 missions/yr. (ave.)
Payloads/Yr.	30 payloads/yr. (ave.)
Propellant Storage	70,000 lbs
Crew Size	8-11
Electrical Subsystem	
Bus Load	77.5 Kw (Day)/62.5 Kw (Night)
S.A. Power	187 Kw BOL/163 Kw EOL (1999)
S.A. Size (Area)	17000 ft^2
Power Groups	10 with 1000 amp-hours (battery)
User Support	36.8 Kw
Thermal Loads	
SS Systems	26 Kw
ECLS	24.5 Kw
User Payloads	29 Kw

1 - In SDV configuration, a separate module is not req'd.

7.4 STS SUPPORT FLIGHTS

The level of support required by the space station over the period from mid-1990 to 1999 has been determined for the evolution plan presented in section 7.1. The year-by-year needs of individual user missions and growth in crew size were also considered in arriving at the STS support level. STS flights were required for the following transportation activities: (1) space station element delivery, (2) payload delivery, (3) materials processing resupply needs, (4) crew rotation, (5) crew and ECLS resupply using the logistics module, (6) OTV cryogen resupply, and (7) OTV and TMS refurbishment.

Maximum support of 7 to 8 flights per year was required for delivery of payloads, followed by a need for 6-7 flights/year for OTV propellant resupply, assuming a limited scavenging capability. These estimates apply to the post-1992 period. Crew and EVA suit rotation and logistics module resupply required an average 1.5 flights per year.

STS overall support ranged from 3 to 11 flights from 1990 through 1992, and averaged between 18 and 20 flights per year thereafter.

APPENDIX A ACRONYMS AND ABREVIATIONS

A	Angstrom
AC&S	Attitude Control and Stabilization
ACC	Aft Cargo Carrier
ACS	Attitude Control Subsystem
ACTS	Advanced Communications Satellite Corporation
AFB	Air Force Base
AHUT	Animal Holder and Unit Tester
AIAA	American Institute of Aeronautics and Astronautics
AIE	Advanced Interplanetary Explorer
AL	Airlock
ALCOA	Aluminum Company of America
AMIMS	Advanced Meteorological Infrared & Microwave Soander
AMPTE	Active Magnetosphere Particle Tracer Experiment
AO	Announcement Opportunity
AP	Action Potential
ARC	Arnold Research Center
ASE	Airborne Support Equipment
ASO	Advanced Solar Observatory
ASTO	Advanced Solar Terrestrial Observatory
ATP	Authority to Proceed
AXAF	Advanced X-Ray Astrophysics Facility
B	Billion
BASD	Ball Aerospace Division
BCK	Blood Collection Kit
BIT	Built-In Test

APPENDIX A ACRONYMS AND ABREVIATIONS

BITE	Built-In-Test-Equipment
BIU	Bus Interface Unit
BOL	Beginning of Life
BTS	Biotelemetry System
BYU	Brigham Young University
C	Core
c	Centigrade
Ca	Calcium
CB	Cargo Bay
C&DH	Command and Data Handling Subsystem
CDP	Coronal Diagnostic Package
CDR	Critical Design Review
CELSS	Controlled Environment Life Support System
CER	Cost Estimating Relationship
CF	Construction Facility
CG	Center of Gravity
CIT	California Institute of Technology
Cl	Chloride
CLIR	Cryogenics Limb Scanning Interferometer & Radiometer
CM	Command Module
CMD	Command
CMG	Control Moment Gryo
CMM	Composite Mission Model
CO_2	Carbon Dioxide
COBE	Cosmic Background Explorer

APPENDIX A ACRONYMS AND ABREVIATIONS

COMPMM	Composite Mission Model
COMSAT	Communications Satellite Corporation
COSMIC	Coherent Optical System Modular Imaging Collector
CR	Comet Rendezvous
CRM	Chemical Release Module
CRMF	Chemical Release Module Facility
CRO	Cosmic Ray Observatory
CRT	Cathode-Ray Tube
CSR	Comet Sample Return
CU	Colorado University
CZCS	Coastal Zone Color Scanner
DBS	Direct Broadcast Satellite
DBV	Derived Boost Vehicle
DDT&E	Design Development, Test and Evaluation
DEMS	Dynamic Environment Monitoring System
DMPS	Data Management and Processing System
DOD	Department of Defense
DRM	Design Reference Mission
DSN	Deep Space Network
DVM	Doctor of Veterinarian Medicine
EAAR	Earth Approaching Asteroid Rendezvous
ECG	Electrocardiograph
ECLS	Environmental Control Pipe Support
ECLSS	Environmental Control/Life Support Systems

APPENDIX A ACRONYMS AND ABREVIATIONS

ECS	Environmental Control System
EEG	Electroencephalogram
e.g.	Example
EKG	Electromyogram
ELS	Eastern Launch Site
EMC	Electromagnetic Compatibility
EMG	Electromyogram
EMI	Electromagnetic Interference
EMU	Extravehicular Mobility Unit
ENG	Electonystagnogram
EOL	End of Life
EOS	Electrophoresis Operations In Space
EOTV	Expendable Orbital Transfer Vehicle
EPS	Electrical Power
EPDS	Electrical Power and Distribution System
ERB	Earth Radiation Budget
ET	External Tank
ETCLS	Environmental and Thermal Control and Life Support
EUVE	Extreme Ultraviolet Explorer
EVA	Extra-Vehicular Activity
Exper	Experimeter
Expmt	Experimeter
fps	Feet per Second
FCC	Federal Communications Commission
FDMA	Frequency-Division Multiple Access

APPENDIX A ACRONYMS AND ABREVIATIONS

FF	Free Flyer
FILE	Feature Identification and Location Experiment
FLOPS	Floating Point Operations Per Second
FOC	Full Operating Capability
FOCC	Flight Operations Control Center
FOT	Faint Object Telescope
FSF	First Static Firing
FUSE	Far Ultraviolet Spectroscopy Explorer
FY	Fiscal Year
g	Gravity
GG	Gravity Gradient
G_Z	Vertical Gravity Acceleration Component
GaAs	Galium Arsemide
GEO	Geosynchronous Earth Orbit
GEOSTO	Geosynchronous Solar Terrestrial Observatory
GFP	Government-Furnished Property
GG	Gravity Gradiometer
GHZ	Gigadertz
GND	Ground
GPS	Global Positioning System
GPWS	General Purpose Work Station
GRIST	Grazing Incidence Solar Telescope
GRO	Gamma Ray Observatory
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center

APPENDIX A ACRONYMS AND ABREVIATIONS

GSS	Ground Support System
GSSI	Geosynchronous Satellite Sensor Intercalibration
GTE	Gamma Ray Timing Explorer
H	Hangar
H_2O	Water
H/W	Hardware
HM	Habitation Module
HMF	Health Maintenance Facility
HNE	Heavy Nuclei Explorer
HOL	Higher Order Language
I&C	Installation and Checkout
I/F	Interface
ID	Identification
INCO	International Nickel Company
INTELSAT	International Telecommunications Satellite Organization
IOC	Initial Operating Capability
IPS	Instrument Pointing System
IR	Infrared
IRAS	Infrared Astronomy Satellite
IRD	Instrument Research Division
IS	Imaging Spectrometer
ISP	Initial Specific Impulse
ISPM	International Solar Polar Mission
ISTO	Initial Solar Terrestrial Observatory
IUE	International Ultra Violet Explorer
IVA	Intravehicular Activity

APPENDIX A ACRONYMS AND ABREVIATIONS

J&J	Johnson and Johnson
JEA	Joint Endeavor Agreement
JHU	John Hopkins University
JPL	Jet Propulsion Laboratory
JSC	Johnson Space Center
K	Potassium
Kbps	Kilobits Per Second
KG, kg	Kilogram
KSC	Kennedy Space Center
KW, kw	Kilowatt
lbm	Pounds
LAMAR	Large Area Modular Array Reflectors
LAMMR	Large Antenna Multifrequency Microwave Radiometer
LaRC	Langley Research Center
LBNP	Lower Body Negative Pressure
LBNPD	Lower Body Negative Pressure Device
LDR	Large Deployable Reflector
LEO	Low Earth Orbit
LeRC	Lewis Research Center
LIDAR	Light Detection and Ranging
LiOH	Lithium Hydroxide
LM	Logistics Module
LMMI	Large Mass Measurement Instrument
LSEPS	Large Spacecraft Effects on Proximate Space

APPENDIX A ACRONYMS AND ABREVIATIONS

LSLE	Life Sciences Laboratory Equipment
LSLF	Life Sciences Laboratory Facility
LSM	Life Support Module
LSRF	Life Sciences Research Facility
LSRM	Life Sciences Research Module
LSS	Life Support Systems
LRU	Line Replaceable Unit
LWA	Long Wavelength Antenna
mV	Millivolt
M	Million
MAM	Main Belt Asteroid Multirendezvous
Mbps	Megabits Per Second
MD	Medical Doctor
MDAC	McDonnell Douglas Astronautics Company
MeV	Million Electron Volts
MGCM	Mars Geochemistry/Climatology Mapper
MIT	Massachusetts Institute of Technology
MMC	Martin Marietta Corporation
MML	Martin Marietta Laboratories
MMS	Multimission Modular Spacecraft
MMU	Manned Maneuvering Unit
MOHM	Megaohms
MOTV	Manned Orbital Transfer Vehicle
MP	Materials Processing
MPN	Mars Probe Network

APPENDIX A ACRONYMS AND ABREVIATIONS

MPS	Materials Processing in Space
MR	Microwave Radiometer
MRICD	Medical Research Institute for Chemical Defense
MRWS	Mobile Remote Work Station
M-SAT	Mobile Satellite
MSFC	Marshall Space Flight Center
MWPC	Multi-Wire Proportional Counter
MWS	Microwave Sounder
N/A	Not Applicable
NAS	National Academy of Sciences
NASA	National Aeronautics and Space Administration
NiH_2	Nichel Hydrogen
NM	Nautical Miles
NMR	Nuclear Magnetic Resonance
NOAA	National Oceanic and Atmospheric Administration
NRL	Naval Research Laboratory
ODSRS	Orbiting Deep Space Relay Station
OIST	Orbiting Infrared Submillimeter Telescope
OMP	Ocean Microwave Package
OMS	Orbital Maneuvering Systems
O_2	Oxygen
O_2/N_2	Oxygen/Nitrogen
OPEN	Origin of Plasma in the Earth Neighborhood
OSA	Optical Society of America
OTV	Orbital Transfer Vehicle
OVLBI	Orbital Very Long Baseline Interferometer

APPENDIX A ACRONYMS AND ABREVIATIONS

P	Phosphorous
PDR	Preliminary Design Review
PET	Position Emission Tomography
PhD	Doctorate of Philosophy
PH	Level of Acidity
PI	Principal Investigator
PIDA	Payload Installation and Deployment Aid
P/L	Payload
PLSS	Portable Life Support Systems/Personal Life Support System
PMD	Propellant Management Device
PMS	Physiological Monitoring System
P/OF	Pinhole/Occulter Facility
PS	Payload Specialist
psi	Pounds per Square Inch
psia	Pounds per Square Inch Absolute
PTE	Plasma Turbulence Explorer
QD	Quick Disconnect
R&D	Research and Development
R&T	Research and Technology
RAHF	Research Animal Holding Facility
RBC	Red Blood Cell
RCA	Radio Corporation of America
RCS	Reaction Control System
REM	Roentgen Equivalent, Mass

APPENDIX A ACRONYMS AND ABREVIATIONS

RF	Radio Frequency
RFP	Request for Proposal
RMS	Remote Manipulator System
ROM	Rough Order of Magnitude
ROSS	Remote Orbital Servicing System
ROTV	Reusable Orbital Transfer Vehicle
SAO	Smithsonian Astronomical Observeratory
SAR	Synthetic Aperture Radar
SARSAT	Search and Rescue Satellite - Aided Tracking
SAT	Satellite
S/C	Spacecraft
SCADM	Solar Cycle and Dynamics Mission
SCDM	Solar Coronal Diagnostic Mission
SCE	Solar Corona Explorer
SDCV	Shuttle Derived Cargo Vehicle
SDV	Shuttle Derived Vehicle
SERV	Servicing
SEXTF	Solar EUV/XUV Telescope Facility
SHEF	Solar High Energy Facility
SIDM	Solar Interior Dynamics Mission
SIDF	Solar Interior Dynamics Facility
SIRTF	Shuttle Infrared Telescope Facility
SIS	Solar Interplanetary Satellite
SL	Spacelab
SLFRF	Solar Low Frequency Radio Facility

APPENDIX A ACRONYMS AND ABREVIATIONS

SMMI	Small Mass Measurement Instrument
SOMS	Shuttle Orbiter Medical Systems
SO/P	Saturn Orbiter/Probe
SOT	Solar Optical Telescope
SP	Scientific Payload
SPELS	Space Plasma Effects on Large Spacecraft
SPIE	Society Photo-Optics Instrument Engineers
SRB	Solid Rocket Booster
SRR	Systems Requirements Review
SS	Space Station
SSCAG	Space System Cost Analysis Group
SSEC	Solar Systems Exploration Committee
SSF	Solar Shuttle Facility
SSL	Space Sciences Laboratory
SSMM	Space Station Mission Model
SSR	Solar Spectrometer/Radiometer
SSRMS	Space Station Remote Manipulator System
SSXTF	Solar Soft X-Ray Telescope Facility
ST	Space Telescope
STDN	Space Tracking and Data Network
STO	Solar Terrestrial Observatory
STS	Space Transportation System
SVI	Stereo Visual Image
TAT	Thinned Aperture Telescope
TBD	To Be Determined
TBR	To Be Required

APPENDIX A ACRONYMS AND ABREVIATIONS

TBS	To Be Supplied
TCS	Thermal Control Subsystem
TDAS	Tracking and Data Acquisition System
TDM	Technology Development Mission
TDMA	Time-Division Multiple Access
TDRS	Tracking and Data Relay Satellite
TDRSS	TDRS System
TEM	Transmission Electron Microscopy
THM	Tethered Magnetometer
TIMI	Thermal Infrared Multispectral Imager
TM	Technical Memorandum
TMS	Teleoperator Maneuvering System
TOPEX	Ocean Topography Experiment
TP	Thermal Panels
TPS	Thermal Protection System
TSS	Time Sharing System
TV	Television
um	Micrometer = micron
usec	Microsecond
uvolt	Microvolt
UARS	Upper Atmosphere Research Satellite
UC	University of California
UCSF	University of California, San Francisco
UHF	Ultra High Frequency

APPENDIX A ACRONYMS AND ABREVIATIONS

Ult	Ultimate
UM	University of Maryland
UM	University of Michigan
UMS	Urine Monitoring System
U.S./USA	United States/United States of America
US	Upper Stage
USRA	University Space Research Association
UT	University of Texas
UV	Ultraviolet
V	Velocity
VAP	Venus Atmospheric Probe
VAFB	Vandenberg Air Force Base
VCU	Virginia Commonwealth University
Vdc	Volts Direct Current
VFR	Vestibular Function Research
VHEO	Very High Earth Orbit
VHSIC	Very High Speed Integrated Circuit
VLR	Very Large Radar
VLST	Very Large Space Telescope
VRF	Vestibular Research Facility
VRM	Venus Radar Mapper
WARC	World Administration Radio Conference
WBS	Work Breakdown Structure
WLS	Western Launch Site
WRU	Work Restraint Unit

APPENDIX A ACRONYMS AND ABREVIATIONS

XGP	Experimental Geostationary Platform
XRO	X-Ray Observatory
XTE	X-Ray Timing Explorer
Zero g	Zero Gravity
β angle	Angle Between Orbit Plane and Solar Vector
α_s	Coating Solar Absorptance
ε	Coating Emmitance
ω	Watts

THIS PAGE INTENTIONALLY LEFT BLANK

APPENDIX B REFERENCES

I. SPACE OPERATIONS CENTER

A. Boeing Aerospace (NAS9-16151)

Monthly Progress Report #1	June 1980
Monthly Progress Report #2	July 1980
" " " #4	Sept 1980
" " " #6	Oct 1980
" " " #7&8	Jan 1981
" " " #9	Feb 1981
" " " #10	Mar 1981
First Quarter Briefing	Sept 1980
Mid Term Review	Dec 1980
Final Briefing	June 1981
Executive Summary	June 1981
Final Report, Vol I - Executive Summary	July 1981
Final Report, Vol III - System Definition	July 1981
Final Report, Vol IV (1 of 2) - System Analysis	July 1981
Final Report, Vol IV (2 of 2) - System Analysis	July 1981
Progress Systems Requirements Document	July 1981
Monthly Progress, Study Extension #1	Oct 1981
Mid Term Review, " " #2	Oct 1981
Final Briefing " "	Jan 1982
Executive Summary " "	Jan 1982
Final Report, Vol IV - Addendum I	Apr 1982

B. Rockwell International (NAS9-16153)

Monthly Progress Report #1	Aug	1980
" " " #2	Sept	1980
" " " #3	Oct	1980
" " " #5	Dec	1980
" " " #6	Jan	1981
" " " #7	Feb	1981
First Quarter Review	Aug	1980
Mid Term Review	Dec	1980
Final Briefing	April	1981
Mid Term Review, Study Extension #3	Oct	1981
Monthly Progress Report #4, Study Extension	Nov	1981
Mid-Term Review, " "	Oct	1982
Final Review, " "	Feb	1981
Executive Summary, " "	Feb	1982
Final Report, Vol I " "	Feb	1982
Final Report, Vol II " "	Feb	1982

C. Martin Marietta Denver Aerospace

Year End IRAD Report	Dec	1982
Denver/Michoud Technical Interchange	Jan	1982

Space Station Needs, Attributes, and Architectural Options Study (NASW-3686)

(Cont)	C.	Monthly Progress Report #1	Oct 1982
		" " " #2	Nov 1982
		" " " #3	Dec 1982
		" " " #4	Jan 1983
		" " " #5	Feb 1983
		Mid Term Review	Nov 1982
	D.	Johnson Space Center	
		Concept Analysis I	Nov 1979
		Concept Analysis II	Nov 1979
		Conference	Nov 1979
		SOC Status Review	Sept 1981
		Requirements Document	Nov 1981
		Program Plan	Nov 1981
		Technology Plan	Oct 1981
		Mission Plan	Apr 1982
		Mission Model Addendum	Nov 1981
		Boeing Program Plan	Nov 1981
		Advanced Mission Studies	Feb 1982
		STS Capabilities for Supporting the Space Station Program	Sep 1982
	E.	Marshall Space Flight Center	
		Use of External Tank for SOC	Dec 1981
		25 KW Power System	Sept 1979

F. International Data

Status of Space Station Study in Canada	Dec 1982
ESA; Space Station: Status of Planning Activities	Dec 1982
German Presentation for Space Station Conference	Dec 1982
Italian Presentation: Potential Application of the Tethered Satellite System Concept to the Space Station	Nov 1982
Japanese Study Status on Space Station	Dec 1982
European Utilization Aspects of a U.S. Manned Space Station	Dec 1982

II. GEOSTATIONARY PLATFORM

A. General Dynamics - Convair (NAS8-33527)

System Concept Definition	July 1981

III. SCIENCE AND APPLICATIONS SPACE PLATFORM

A. McDonnel Douglas (NAS8-33592)

First Interim Briefing	(DR-3)	Aug 1981
Second Interim Briefing		Nov 1981
Final Briefing		Feb 1982

B. NASA

Science and Applications Requirements 30th Space Station, Draft Reports, NASA/GSFC.	Nov 1982
Space Science Platform: Panel Reports from UAH/NASA Workshop. NASA/MSFC.	Dec 1978

IV. TELEOPERATOR MANEUVERING SYSTEM (TMS)

A. Martin Marietta Denver Aerospace (NAS8-34581)

Mark II Concept	Dec 1979
" " Summary	
" " AIAA Study	July 1981
Technical Proposal	Mar 1980
Reference Missions	Oct 1981
Monthly Progress Report #1	Nov 1981
" " " #2	Dec 1981
" " " #3	Jan 1982
" " " #4	Jan 1982
" " " #5	Mar 1982
" " " #6	Mar 1982
" " " #7	May 1982
Requirements Review	Nov 1981
Executive Overview	Dec 1981
Mid Term Review	Feb 1982

(Cont) A.

Executive Summary	Apr 1982
Final Review	May 1982
Final Report	June 1982

B. Vought Corporation

TMS Study	April 1981
Interim Review	Dec 1980

Final Review	July 1981
Executive Summary	July 1981
Technical Summary	July 1981
System Definition Study	
Vought/MMC TMS Comparisons	July 1978
Low Energy Stage Study	Oct 1978
Mission Requirements Systems Definition-Tech Report	May 1982
Mission Requirements System Definition-Final Review	May 1982

C. Marshall Space Flight Center

Modular Propulsion concept	June 1981

V. SHUTTLE DERIVED VEHICLES

A. Martin Marietta Denver Aerospace (NAS8-341B3)

Final Report	July 1981

VI. ORBITAL TRANSFER VEHICLE

A. Grumman Aerospace Corporation

System Analysis Study Extension	Feb 1981

B. Boeing Aerospace (NAS8-33532)

Concept Definition Study	1980

C. General Dynamics - Convair (NASI-33533)

Concept Definition Study	Feb 1981

APPENDIX B REFERENCES (CONT)

VII. SPACE CONSTRUCTION

A. General Dynamics Convair (NAS9-16120)

Space Construction Experimentation
Definition Study — July 1981

B. Rockwell International (NAS9-15718)

Space Construction System Analysis — Mar 1981

Cost and Program — Apr 1980

C. Grumman Aerospace (NAS9-16120)

Orbiter-Based Construction Equipment Study (OBCES) — Apr 1981

VIII. SATELLITE SERVICING

A. Grumman Aerospace (NAS9-16120)

Final Briefing (Part II) — July 1981

Executive Summary — July 1981

Mid Term Briefing — Dec 1981

B. Lockheed Missiles and Space (NAS9-16121)

Final Review — July 1981

Executive Summary — July 1981

Mid Term Review — Dec 1981

C. Martin Marietta Denver Aerospace

Rendezvous and Docking System — June 1981

Integrated Orbital Servicer Vol I — April 1978

" " " Vol II — April 1978

D. Lincom Corporation Automated Rendezvous (NAS9-16130) Oct 1981

IX. DEFINITION OF TECHNOLOGY DEVELOPMENT MISSIONS FOR EARLY SPACE STATION

-Satellite Servicing

A.	Martin Marietta Denver Aerospace	(NAS8-35042)	
	Monthly Progress Report #1		Nov 1982
	" " " #2		Dec 1982
	" " " #3		Jan 1983
	" " " #4		Feb 1983
	Mid Term Review		Oct 1982
	Environmental Analysis		Jan 1983
B.	TRW	(NAS8-35081)	
	Mid Term Review		Oct 1982

-Orbital Transfer Vehicle

A.	General Dynamics Convair	(NAS8-35039)	
	Mid Term Review		Oct 1982
-	Large Space Structures	(NAS8-35043)	
	Mid Term Review		Oct 1982

X. MISSION MODELS

A. NASA

Space Systems Technology Model	May 1980
Space Systems Technology Model	Sept 1981
STS Flight Assignment Baseline	Dec 1980

B. Martin Marietta Denver Aerospace

National Mission Model	Sept 1981
TMS Mission Model	Oct 1981
Space Station Composite Mission Model	Nov 1982

C. Eagle Engineering

National Mission Model	June 1981

D. Battelle Columbus Laboratories

Low Energy Model	Oct 1980

E. National Academy of Sciences

Astronomy and Astrophysics for the 1980's	1982
A Strategy for Space Astronomy and Astrophysics for the 1980's	1979

F. American Institute of Aeronautics and Astronautics

Space Transportation Services Through 2000	Jan 1981

G. Technology for Space Astrophysics: The next 30 years — Oct 1982

XI. ENVIRONMENT

A. AIAA

Bluth, B. J., Social and Psychological Problems of Extended Space — May 1980

B. NASA

Terrestrial Environment (Climatic) Criteria Guidelines for use in Aerospace Vehicle Development Technical Memorandum 78118 — 1977

Space and Planetary Environment Criteria Guidelines for use in Aerospace Vehicle Design Development Technical Memorandum 78119 — 1977

Space Vehicle Design-Criteria (Environment) Model of Earths Atmosphere (90-2500 km) Sp-8021 — Mar 1973

The NASA/MSFC Global Reference Atmospheric Model MOD-3 (With Spherical Harmonic Wind Model) CR-3256 — Mar 1980

Tables of Atmospheric Temperature, Density, and Composition Derived from Satellite and Ground Based Measurements Vol 1, 2 — Jan 1979

Four Dimensional World Wide Atmospheric Models Allied Research Associates, INC — July 1972

Roth, Emaneul Compenduim of Human Responses to the Aerospace Environment, Report CR-1205 — 1968

XII. LIFE SCIENCES

A. NASA

Life Sciences Considerations for Space Station, NASA/HQ&Bionetics, Inc. — Sept 1982

Life Sciences Research and the Science and Applications Space Platform, NASA/ARC. — Jan 1982

Medical Operations and Life Sciences Activities on Space Station NASA/JSC TM 58248. — Oct 1982

D.L. DeVincenez;, J.R. Bagby
Orbiting Quarantine Facility: The Antaeus Report
NAS SP-454 — 1981

Fabricant, J.D., "Life Sciences Experiments for a Space Platform/Station," Society of Automotive Engineers, Inc., (MSFC SAE-TP82834). 1982

T. Bannister, Man Tended - Life Sciences Research Facility: A Conceptual Design and Analysis Study, (MSFC-PD-A5555) Jan 1982

T.W. Halstead, D.R. Beam, P.L. Russell, _Descriptions of Current Research Projects on the NASA Space Biology Program_, NASA/HQ 1983

Johnston, R.S., Dietlein, L.F., Berry, C.A., _Biomedical Results of Apolls_, NASA/JSC 1975

Johnston, R.S. Dietlein, L.F., _Biomedical Results from Skylab_, NASA/JSC 1977

Spacelab 4: Mission Science Requirements Document, NASA/JSC June 1982

Space Motion Sickness Symposium Proceedings, GE; POT-1830G; NASA/JSC Nov 1978

Space Lab 1: Mission Operations Recurements Dcoument, NASA/TSC, LS 2000 8-4 May 1980

Simmonds, R.C., Bournc, G.H., The Use of Non-human Primates in Space, NASA Conference Publication 005 Dec 1974

B. Richard S. Johnston & Associates
Review of Present and Future Trends of Materials Science and Life Sciences Flight Experiments: Part I Life Sciences. Feb 1981

C. Other
CELSS Principle Investigators Workshop Abstracts. Dec 1982

National Research Council Space Science Board, Life Beyond the Earth's Environment: The Biology of Living Organisms in Space (The Bricker Report) 1979

Furukawa, S., Operational Medicine Support in Long-Duration Manned Space Missions - Low Earth Orbit and Beyond, McDonnel Douglas Technical Services Company - KSC, Feb 1982

T.W. Halstead, "The NASA Space Biology Program," The Physiologist, Vol 25, No. 6, Suppl. 1982

XIII. HUMAN FACTORS

A. NASA

Man/System Requirement for Weightless Environments

MSEC-STD-512 Rev A Dec 1976

B. Department of Defense

Human Engineering and Facilities.

MIL-STD-1472 Rev C May 1981

XIV. MISSION ANALYSIS

A. Johnsom Space Center, STS Capabilities for Supporting the Space Station Program, Spetember 1982.

Johnson Space Center, Mission Planning and analysis Division, Maneuver Strategies and Performance Requirements for Sphere of Influence Determination, March 1982.

Johnson Space Center, Joseph P. Loftus, Jr., Advanced Missions Discussions, Martin-Marietta Denver, November 16, 1982.

Johnson Space Center, L. E. Livingston, Co-Orbiting Mechanics, June 22, 1982.

B. Marshall Space Flight Center, Space and Planetary Environment Criteria Guidelines for Use in Space Vehicle Development, 1977 Revision, PP 2-4.

C. Goddard Space Flight Center, A. E. Hedin, Tables of Atmospheric Temperature, Density and Comp]osition Derived from Satellite and Ground Based Measuremetns, Volume 2 of 3 (AP=20), January 1979, P-608.

XIV. MISSION ANALYSIS

Goddard Space Flight Center, A. e. Hedin, Tables of Atmospheric Temperture, Density and Composition Derived from Satellite and Ground Based Measurements, Volume 1 of 3 (AP+4), January 1979, P-108.

D Martin Marietta Corporation, Composite Mission Model (including DOD flts), Space Station Needs, Attributes, and Architectural Optios Study, Initial Release, November 1982.

Martin Marietta Corporation, Tethered Satellite Advanced Mission Development, Project No. D-29D (Proprietary Studies), 1 November 1982.

Martin Marietta Aerospace, Teleoperator Maneuvering System/Mark II Propulsion Module Study, Final Report, June 1982.

Martin Marietta Aerospace, System Technology Analysis of Aero Assisted Orbital Transfer Vehicles: Low Lift/Drage (0-0.75), Folder 1, Technical Proposal, July 1982.

XIV. MISSION ANALYSIS

Martin Marietta Corporation, SDV Technology Requirements Study, Phase 1, Final Report Contract NAS8-24183, MMC-SDV-DR-6-1, July 1981.

D. California Space Institute, Utilization of the External Tanks of the Space Transportation System, Draft Report, October 8, 1982.

XV. TETHERED SATELLITE

A. Martin Marietta Denver Aerospace
Advanced Mission Development (Proprietary Studies) Nov 1982

XVI. COST MODELS

A. Unmanned Spacecraft Cost Model, U. S. Air Force, Space Division, Fifth Edition, June 1981.