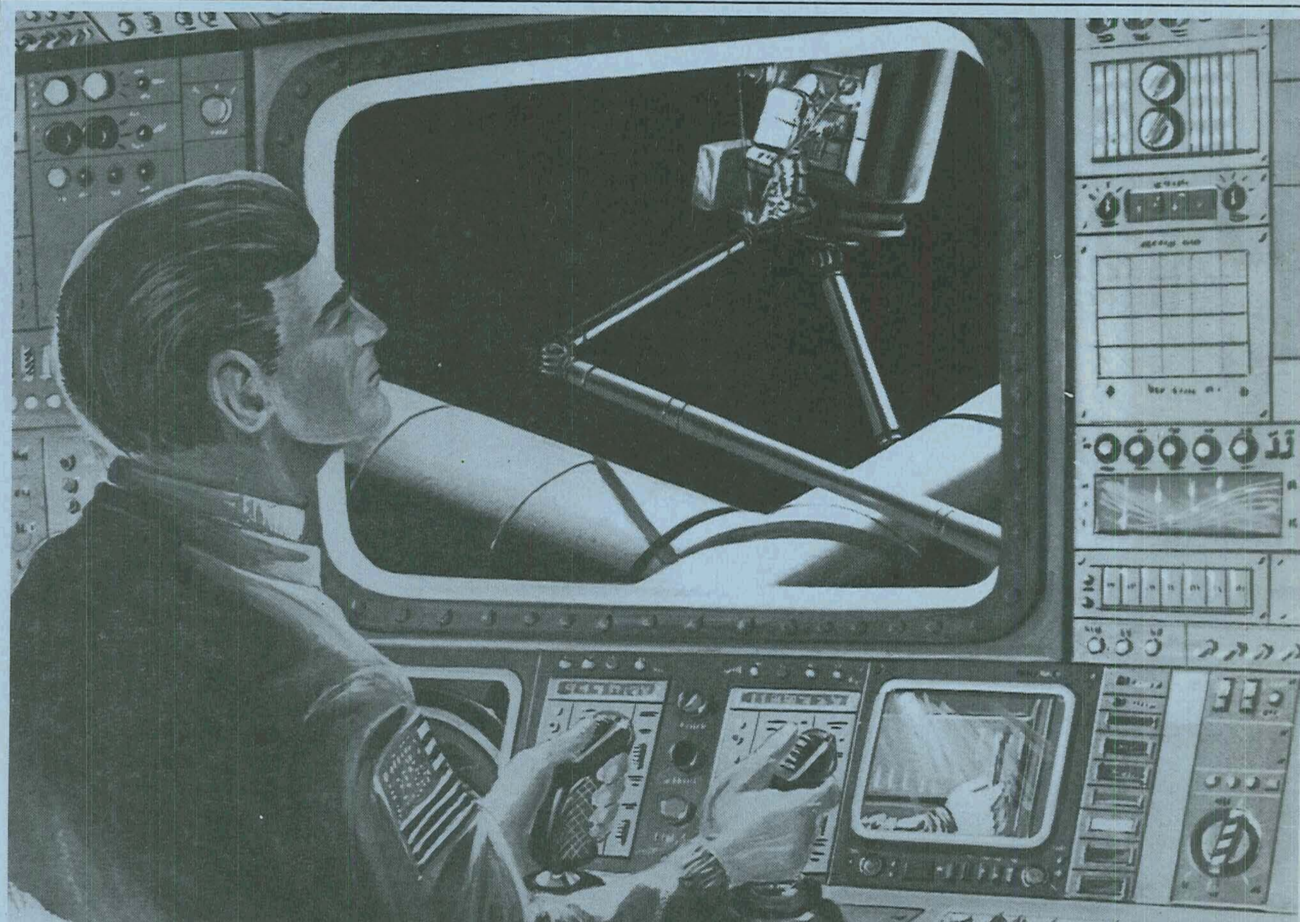
N84-18300

final technical report

SPACE STATION NEEDS, ATTRIBUTES, AND ARCHITECTURAL OPTIONS

volume II - book 2

part I — mission implementation concepts



GRUMMAN

COMSAT GENERAL

GENERAL ELECTRIC

final technical report

SPACE STATION NEEDS, ATTRIBUTES, AND ARCHITECTURAL OPTIONS

volume II - book 2

part I — mission implementation concepts

prepared for

National Aeronautics and Space Administration
Headquarters
Washington, D.C. 20546

under contract NASW-3685
Space Station Task Force
Contracting Study Project Manager — E. Brian Pritchard

by

Grumman Aerospace Corporation
Bethpage, New York 11714

report no. SA-SSP-RP008

20 April 1983

PREFACE

This book consists of four parts and contains the results of studies to develop realistic system architectural concepts and implementation plans for a manned Space Station and its elements. The studies were based on and interated with the mission requirements presented in Book 1 and the programmatics covered in Volume II, Book 3.

Part I of this book describes the approach used to develop the architecture, and presents the results. In addition, the studies performed for the major subsystems and the program evolution are persented.

The results of a special emphasis study of the Space Station Information Management System, by the Space Station Division of General Electric, under subcontract to Grumman are presented in Part II of this volume.

Part III contains the results of a special emphasis study on the Communication Subsystem by COMSAT General, under subcontract to Grumman.

Part IV contains supportive architectural information supplied by MBB/ERNO, Dornier and British Aerospace.

FOREWORD

The study of Space Station Needs, Attributes, and Architectural Options was an eight month effort, focusing on manned space activities during the 1990s that would either require or materially benefit from a Manned Space Station. This study was performed by Grumman Aerospace Corporation, with General Electric and COMSAT General as teammates, under contract NASW-3685 for the National Aeronautics and Space Administration Headquarters' Space Station Task Force.

The NASA Contracting Officer's Representative and Project Study Manager was Mr. E. Brian Pritchard. Technical monitor for the DoD Space Station Working Group was Capt. James Schiermeyer AFSD/XR, who was assisted by Dr. John Baker of the Aerospace Corporation.

This contract was performed within Grumman's Space Station Programs organization directed by Dick Kline. The Grumman Project Study Manager was Ron McCaffrey, who in turn was assisted by Deputy Project Manager Joe Goodwin and Assistant Project Managers Al Alvarado of General Electric and Phil Caughran of COMSAT General. Grumman Task Leaders are:

- o Marty Finkelman - Mission Requirements
- o Don Stein - Concept Development
- o Jim Wilder - Cost and Programmatics
- o Ron Boyland - DoD Assignment

The results of the overall study are described by the following final report documentation.

- o Volume I, Executive Summary, Report No. SA-SSP-RP007, 20 April 1983
- o Volume II, Technical Report No. SA-SSP-RP008, 20 April 1983
  - Book 1 - Mission Requirements
  - Book 2 - Mission Implementation Concepts
  - Book 3 - Cost and Programmatics
  - Book 4 - Military Mission Assessment (Classified)

- o Final Briefing, Report No. SA-SSP-RP009, 5-9 April 1983
  - Part 1 - Summary
  - Part 2 - Mission Requirements
  - Part 3 - Commercialization
  - Part 4 - Technology Development
  - Part 5 - Systems
  - Part 6 - Costing
  - Part 7 - DoD Summary (Classified)
  - Part 8 - National Security (Classified).

Significant contributions were made to the Grumman study effort by their two teammates:

- o COMSAT General defined space station requirements and benefits for commercial communication satellites and the onboard RF communication subsystem. This work was performed within COMSAT Generals' Engineering and Systems Integration organization directed by Mel Savage
- o General Electric, in turn, defined space station requirements and benefits for selected areas of science and applications, commercial processing and remote sensing, and national security missions. In addition, they defined architectural concepts for the data management subsystem. This work was performed within General Electrics' Advanced Earth Observation Programs organization managed by Lew Beers.

Technical progress was reviewed during the study by a seven member Constituency Development Council (CDC). This group met on an "ad hoc" basis to provide senior management/evaluation perspective before each presentation. Members of the CDC include:

- o Dan Huebner, Chairman, Grumman Sr. VP Marketing and Advanced Technology
- o Fred Haise, Grumman VP Space Programs
- o Al Rosenberg, General Electric VP & General Mgr Space System Division
- o Bill Houser, COMSAT General VP System Technology Services
- o Grant Hedrick, Grumman Presidential Asst. for Corporate Technology
- o B/Gen. Dick Rumney (Retired)
- o V/Adm. Forrest Petersen (Retired)

The CDC also provided guidance to parallel corporate funded activities to develop space station advocates and constituents within non-aligned commercial companies. Lou Hemmerdinger, Manager for Space Station Utilization, led Grumman's efforts in this area. Jack Dickinson led similar activities at General Electric. Grumman and General Electric focused on individualized meetings with prospective clients in the pharmaceuticals, metals and semiconductor industries; Clarance Catoe of COMSAT, surveyed the telecommunications industry.

We wish to acknowledge contributions from British Aerospace, MBB/ERNO and Dornier Systems for information on European mission requirements and hardware definitions, for which each company is particularly competent.

Questions regarding this study should be directed to:

Mr. E. Brian Pritchard MFA-13
Contracting Study Project Manager
NASA Headquarters
Space Station Task Force
Washington, D.C. 20546

or

Mr. Ronald W. McCaffrey
Space Station Project Manager
Grumman Aerospace Corporation
Bethpage, N.Y. 11714

KEY SPACE STATION PERSONNEL

	GRUMMAN	GENERAL ELECTRIC	COMSAT GENERAL
Council Review	D. Huebner, Sr. VP F. Haise, VP I.G. Hedrick, Pres. Asst B/Gen. R. Rumney (Ret) V/ADM F. Petersen (Ret)	A. Rosenberg, VP	W. Houser, VP
Program Review	R. Kline	L. Beers	M. Savage
Project Mgr	R. McCaffrey		
Deputy & Asst Mgr	J. Goodwin	A. Alvarado	P. Caughran
Mission Req Leader (Task 1)	M. Finkelman		
Mission Integration	W. Schoen	C. Braunwarth	
Materials Processing	D. Larsen	R. Frost	
Communications	W. Schoen		C. Pentlicki* G. Welti* D. Spinivas*
Science & Applications	R. Pratt T. Hilgeman W. Egan	R. White A. Park M. Bortner	
Military	R. Boyland V. Priebe J. Schultz D. Sullivan J. Corbett	J. McGlinchey	
System Reqmts	R. Pratt		
Proximity Operations	R. Adornato		

*Comsat Labs

	GRUMMAN	GENERAL ELECTRIC	COMSAT GENERAL
Concept Devel Leader (Task 2)	D. Stein		
Concept Design	S. Sherman		
Crew System Design	G. Harms		
Communication	F. Savarese	L. Jones	C. DeViex H. Song
Data Management	E. Ramirez	H. Kraiman R. Barnes E. Heater R. O'Connell	
ECLS	F. Abeles		
GN&C	R. Olsen		
Electrical Power	H. Siegel		
Weights	J. Sims		
Cost Programmatics Leader (Task 3)	J. Wilder		
Cost Analysis	A. Kallas K. Hale R. Black	A. Cohen	
Test	G. Meyers		
DoD Assignment Leader (Task 4)	R. Boyland		
Mission Analysis	J. Schultz D. Sullivan T. Hilgeman	F. Vicente	
Models	V. Priebe W. Brunnig		
Radiation	S. Miluschewa M. Stauber		
Utilization Mgr	L. Hemmerdinger	J. Dickinson	
Marketing	S. Abraham		
Research	D. Larsen		
Prospectus/Industry Survey			C. Catoe

11/2/1

CONTENTS

CONTENTS (contd)

CONTENTS (contd)

Section | Page

ILLUSTRATIONS

ILLUSTRATIONS (contd)

ILLUSTRATIONS (contd)

ILLUSTRATIONS (contd)

ILLUSTRATIONS (contd)

ILLUSTRATIONS (contd)

GLOSSARY

AC	Alternating Current
ACS	Attitude Control Subsystem
ADA	High Level Language
ADF	Automatic Direction Finder
AgZn	Silver Zinc
A/L	Airlock
Amp-Hr	Ampere Hour
Ant	Antenna
AXAF	Advanced X-ray Astrophysics Facility
BER	Bit Error Rate
BOL	Beginning of Life
BPD	Bits Per Day
BPS	Bits Per Second
BW	Bandwidth
C&D	Controls & Displays
C&T	Communications & Tracking
C&W	Caution & Warning
CCTV	Closed Circuit Television
CER	Cost Estimate Relationship
C.G.	Center of Gravity
CH4	Methane
CMD	Command
CMG	Control Moment Gyro
CMOS	Complimentary Metal Oxide Semiconductor
C/O	Check Out
CO_2	Carbon Dioxide
COMM	Communication
COMSEC	Communications Security

CPU	Computer Processor Unit
CRC	Cycle Redundant Code
CRT	Cathode Ray Tube
CY	Calendar Year
DB	Decibel
DBMS	Data Base Management System
D/BS	Docking/Berthing System
DC	Direct Current
DDT&E	Design Development Test & Evaluation
Deg	Degree(s)
DF	Direction Finding
Dia	Diameter
DMS	Data Management Subsystem
DoD	Department of Defense
DOD	Depth of Discharge
DOF	Degrees of Freedom
DOMSAT	Domestic Satellite
DRIRU	Dry Rotor Inertial Reference Unit
EC/LSS	Environmental Control/Life Support Subsystem
EMU	Extra Mobility Unit
EPS	Electrical Power Subsystem
ERP	Effective Radiator Power
ESS	Evolved Space Station
EVA	Extra Vehicular Activity
FEC	Forward Error Correction
FF	Free Flyer
FOV	Field of Vision
FSS	Flight Support Station
FY	Fiscal Year
GaAs	Gallium Arsenide
GAC	Grumman Aerospace Corporation

GEO	Geosynchroneous Earth Orbit
GFE	Government Furnished Equipment
GHz	Gigahertz
GN&C	Guidance, Navigation, & Control
GPS	Global Positioning Satellite
GS	Ground Station
GSE	Ground Support Equipment
GSTDN	Ground Satellite Tracking & Data Network
H_2	Hydrogen
H_2O	Water
HM No. 2	Habitat Module Number 2
HOL	High Order Language
HPA	Handling & Positioning Aid
HSM No. 1	Habitat/Subsystem Module Number 1
HZ	Hertz (Cycles Per Second)
I/F	Interface
IACO	Integration, Assembly & Checkout
IFF	Identification Friend or Foe
IOC	Initial Operational Capability
IPS	Instrument Pointing System
Isp	Specific Impulse, Sec.
ISS	Initial Space Station
IVA	Intra Vehicular Activity
KBS	Kilobytes Per Second
kg	Kilogram
km	Kilometer
KOPS	Kilo Operations Per Second
KSA	Ku-Band Special Acess
KW, Kw	Kilowatt
KWe	Kilowatts Electrical
KW-Hr	Kilowatt-Hour

L-Band	L-Band Frequency
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LiOH	Lithium Hydroxide
L/M	Logistics Module
L/M/P	Logistics Module Pallet
LMSC	Lockheed Missile & Space Company
LO_2	Liquid Oxygen
LOS	Line of Sight
L/V	Local Vertical
M	Meter
M^2	Square Meters
M^3	Cubic Meters
MB	Megabytes
MBA	Multiple Beam Array
MHz	Mega Hertz (Million Cycles)
MISC	Miscellaneous
MMI	Man/Machine Interface
MMU	Maned Maneuvening Unit
MODEM	Modulator/Demodulator
MOTV	Manned Orbital Transfer Vehicle
MRWS	Manned Remote Work Station
MSFC	Marshall Spacecraft Flight Center
NASA	National Aeronautics & Space Administration
N_2	Nitrogen
N/A	Not Applicable
NiCd	Nickel Cadmium
NiH_2	Nickel Hydrogen
N MI	Nautical Mile
N-M	Newton-Meter
N-M-S	Newton-Meter-Sec

O_2	Oxygen
OaS	Operations and Support (Cost)
OCP	Open Cherry Picker
OPS	Operations
ORU	Orbital Replaceable Unit
OTV	Orbital Transfer Vehicle
PAM-D	Payload Assist Module-D
P^3	Programmable Power Processor
PCC&D	Power Control, Conversion, & Distribution
P/L	Payload
POP	Perpendicular to Orbit Plane
PPU	Power Processing Unit
POV	Proximity Operations Vehicle
PROP	Propulsion
Pwr	Power
R&D	Research and Development
RCS	Reaction Control Subsystem
RDT&E	Research Development Test & Engineering
Rec	Recreation
RF	Radio Frequency
RFC	Regenerative Fuel Cell
RMS	Remote Manipulator System
S/A	Solar Array
S-Band	S-Band Frequency
SAWD	Solid Amine Water Desorbed
SCS	Stability and Control Subsystem
SEPS	Solar Electric Propulsion System
Si	Silicone
SIU	Signal Interface Unit
SIMOP	Simulataneous Operation
SIRTF	Shuttle Infrared Telescope Facility

SOA	State of the Art
SRM	Solid Rocket Motor
SS	Space Station
SSA	S-Band Single Access
STS	Space Transport System
TBD	To Be Determined
TCC	Transformer Coupled Converter
TDAS	Tracking and Data Acquisition Satellite
TDRSS	Tracking & Data Relay Satellite Sytems
TFU	Theoretical First Unit (Cost)
TH	Transportation Harbor
TIP	Tended Industrial Park
TIMES	Thermo-Electric Integrated Membrane Evaporation System
TMS	Teleoperator Maneuvering System
TPP	Tended Polar Platform
TV	Television
UHF	Ultra-High Frequency
VCD	Vapor Compression Distillaton
VHSIC	Very High Speed Integration Circuits
Vdc	Volts-Direct Current
VV	Velocity Vector
W/M	Waste Management
WSGT	White Sands Ground Terminal
WTR	Western Test Range

1 - INTRODUCTION

The primary objective of the study of mission implementation concepts is to provide a Space Station architectural concept that is responsive to identified beneficial missions. The concept should be capable of accommodating unforeseen uses and budget variations in an orderly and efficient manner. The approach, guidelines and requirements used to develop an architectural concept are presented in this section. The resulting architecture and its rationale are described in Section 2. Section 3 contains the studies performed for the major subsystems, and Section 4 contains a description of an evolution plan for the architectural concept.

1.1 APPROACH

The approach used for defining the Space Station architecture and its evolution is illustrated in Fig. 1-1 through 1-3. Prior and present mission requirements studies (Fig. 1-1) have resulted in the identification of system architecture and associated subsystem requirements and options as a function of time. Although shown as a straight through task, the development of requirements and architectural concepts is an iterative process considering the missions, cost and programmatic implications.

The next step, shown in Fig. 1-2, is the evaluation and analysis of these requirements. In this process, integrated requirements are subjected to a commonality filter to assure the inclusion of cost effective elements that are responsive to mission needs and do not overly complicate either the specialized mission equipment or the Space Station elements in support of the mission. Commonality and simplicity are major technology and cost drivers, especially for configurations and subsystems. This step is also an iteration process between design commonality, technology status and mission requirements. It results in the identification of top level commonality, a definition of the initial and evolved configurations and enabling technology requirements.

The last step in our approach, shown in Fig. 1-3, derives the evolution plan. It is here that the critical integration of the various items occurs to achieve a cost effective scheduling sequence, with achievable, realizable objectives and budget allocations. The criteria applied for developing the evolutionary strategy included costs, growth capability, complexity and development risk.

The guidelines and requirements used in this approach are presented in the following subsection.

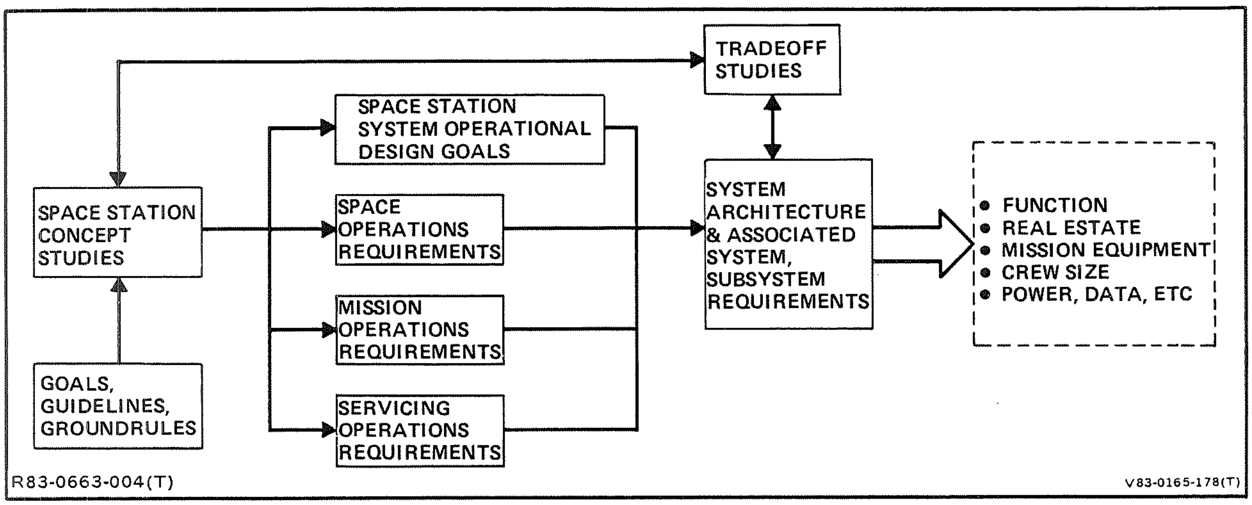


Fig. 1-1 Approach Step 1: Requirements

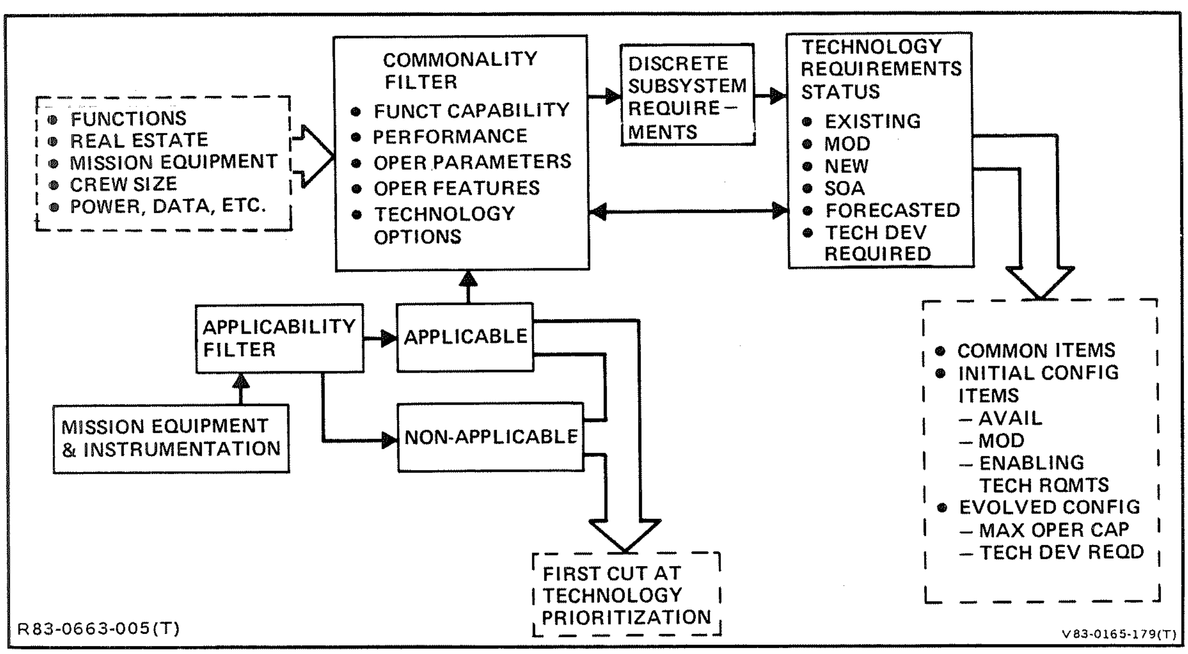


Fig. 1-2 Approach Step 2: Abilities

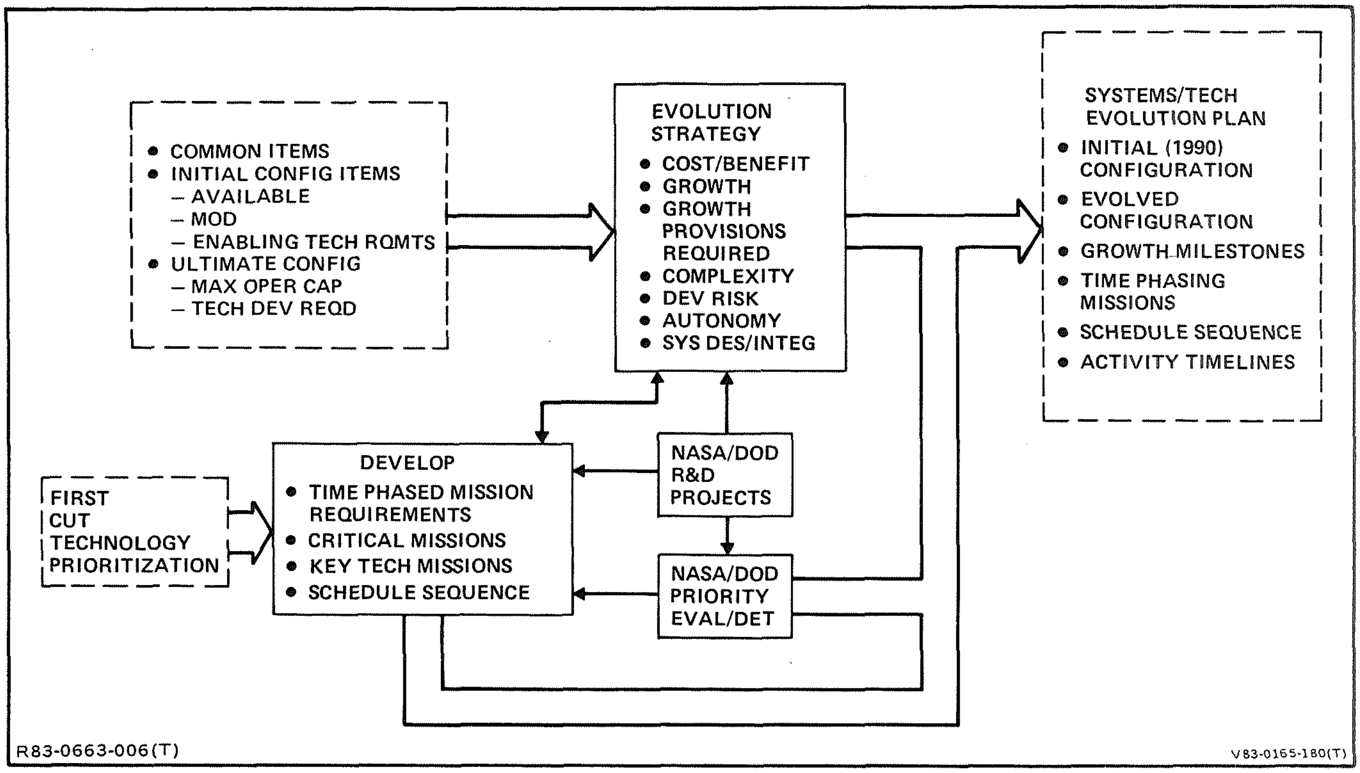


Fig. 1-3 Approach Step 3: Going For It

1.2 GUIDELINES/REQUIREMENTS

The guidelines used for the study of the architectural concepts include:

- Initial capability to use 1986 state-of-art technology
- Initial operational capability by 1990
- Efficient and flexible evolution through 2000, with indefinite use by on-orbit reservice, repair or replacement
- Delivery and support equipment to be shuttle compatible
- Shuttle logistics resupply (men and material) at 90-day intervals
- Safe haven for 14-day crew survival
- Fail operational/fail safe for critical functions and fail safe for non-critical functions.

Specific requirements, such as crew size, power demand, functions to be performed, equipment to be supplied, and other Space Station support elements have been defined in Book 1. These requirements result in a desired set of functional capabilities both for an initial and evolved Space Station. The functional capabilities are:

- Space Test - space equivalent of wind tunnel facilities for research and development and proof-of-concept testing in the space environment
- Transportation Harbor - on-orbit turnaround of propulsion systems for local and remote travel
- Satellite Servicing/Assembly - refuel, repair, checkout and assembly of spacecraft
- Observations - celestial, solar and terrestrial
- Industrial Park - research, development and production of materials in the micro-g space environment.

Satisfying these functional capabilities imposes physical requirements on the architecture.

For the space facility function, desirable architectural features include a structure for assembling and mounting equipments to be tested in the space environment, manipulators to aid in the assembly and test, free flyers to provide remote sensing and internal areas for laboratory and control functions. Illustrated in Fig. 1-4, is the test configuration of a large antenna that represents the most critical requirement identified.

As a transportation harbor, the architectural features include a berthing facility for the shuttle (both for logistics resupply and for emergencies), basing and resupply of upper stages (POV,TMS and reusable OTVs), and stage/payload assembly. Figure 1-5 shows the assembly of a reusable OTV out of the transportation harbor, surrogate bay structure.

A satellite servicing and assembly capability in a Space Station is primarily an external function. Retrieval of satellites for servicing and their subsequent deployment utilizes POVs or TMSs, which can be stored in the area designated as a transport harbor. Similarly, refuel capability can be performed in that area. Handling aids, such as the RMS, HPA, OCP and teleoperators are required to capture, to hold and to maneuver the payloads. These support a servicing function. Shown in Fig. 1-6, is the servicing of an AXAF class satellite.

The functional capabilities for celestial, solar and terrestrial observations require unobstructed fields-of-view, particular levels of pointing accuracy and stability, and an uncontaminated environment. Orbit selection, discussed in Book

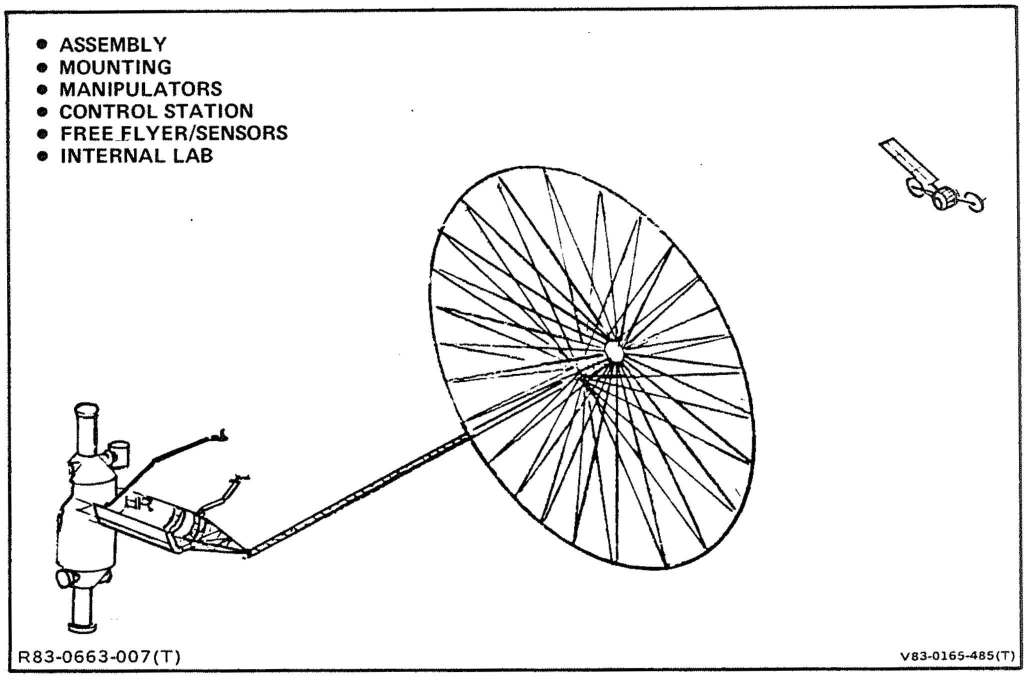


Fig. 1-4 Space Test Facility Requirements

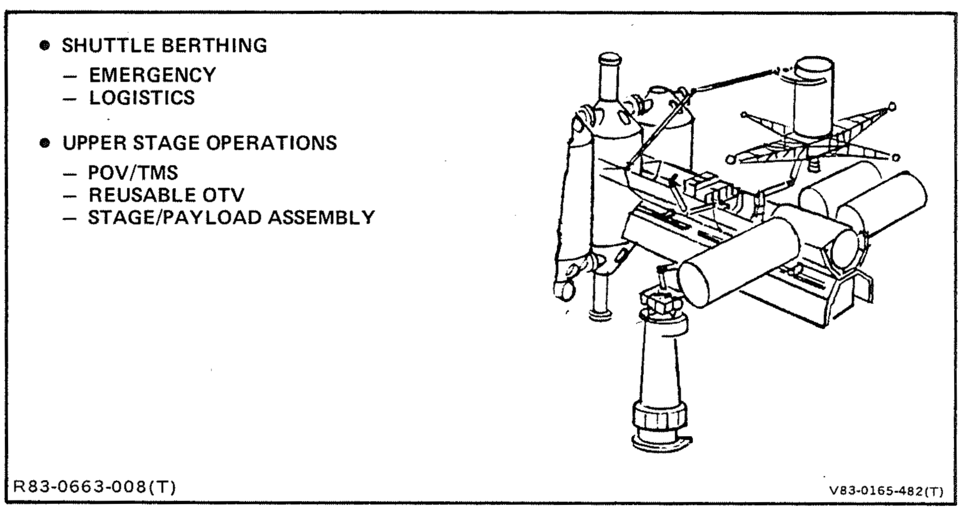


Fig. 1-5 Transportation Harbor Requirements

1, and high data rates, represent additional key drivers for the observation function. Observation missions can utilize free flyers or be incorporated into the Space Station complex (Fig. 1-7), if that orbit inclination is suitable.

The space test facility laboratories include an R&D operation to support an industrial park functional capability. In addition, production facilities for processing materials are required. Such facilities desire micro-g accelerations and generally consume large amounts of power. The micro-g acceleration requirement suggests that the processes be located close to the center of gravity to minimize the effect of rotational accelerations and suspension systems to isolate high frequency vibration. Alternately, an industrial park could be accommodated in a free flying platform, tended by the Space Station, for maintenance and for removal and replacement of processed material. Figure 1-8 shows a tended industrial platform.

In addition, there are architectural requirements which apply to more than one capability. These include the desire to monitor external activity from within a pressurized module (thus the requirement for viewing ports), the availability of ample power, data processing, attitude control and cooling (standardized interfaces) and crew safety considerations (e.g., two ingress/egress paths).

The architecture for implementing these requirements is represented in Section 2.

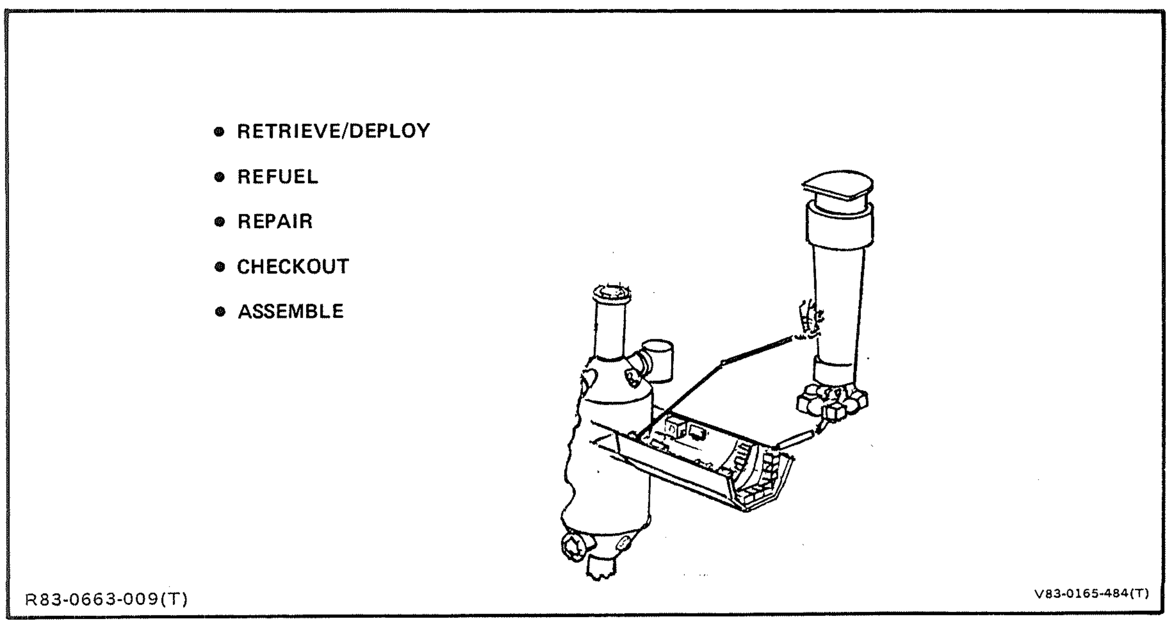
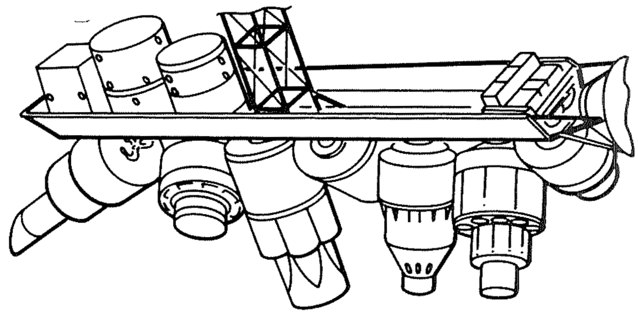


Fig. 1-6 Satellite Servicing & Assembly Requirements

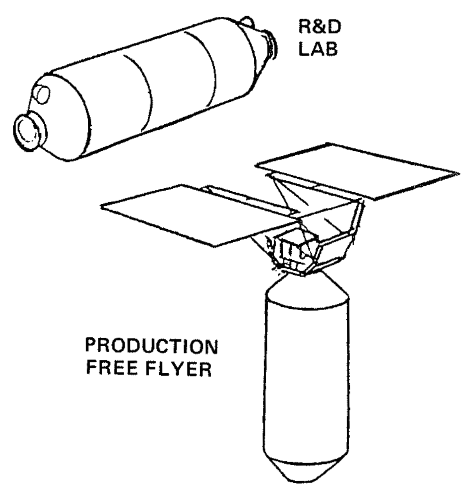
- CELESTIAL
- SOLAR
- TERRESTRIAL
  - – FIELD OF VIEW
  - – POINTING STABILITY
  - – ORBIT
  - – CONTAMINATION
  - – DATA RATES



R83-0663-010(T) V83-0165-483(T)

Fig. 1-7 Observatory Requirements

- MATERIALS R&D
  - – MICRO-G ACCELERATION
- MATERIALS PROCESSING (PRODUCTION)
  - – MICRO-G ACCELERATION
  - – HIGH POWER



R83-0663-011(T) V83-0165-114(T)

Fig. 1-8 Industrial Park Requirements

2 - ARCHITECTURE

A compilation of requirements and trade studies has produced an architecture definition. The resultant architecture consists of a modular manned Space Station with capabilities for Space Test, Transport Harbor, Satellite Services, Celestial/Solar/Terrestrial Observations and an Industrial Park. Solar arrays provide power. The initial Space Station has a 22 kW continuous capability (sufficient to support a three-man crew and the initial operation to support these functions). It can be operational early.

The initial Space Station can evolve with the addition of three-man habitats, additional surrogate structures and laboratories for research and development (R&D), sciences, and industrial processing. By adding solar panels, the power output can be increased to 66 kW continuous. Further evolution beyond this stage can be accomplished.

Included in this configuration is a free flying tended industrial platform for material processing production and a tended (or periodically manned) platform at polar inclination for observations and for satellite service.

This section describes the major elements of the architecture.

2.1 INITIAL SPACE STATION

Figure 2-1 shows the concept for the Initial Space Station (ISS) which fulfills the near-term requirements. It has a pressurized core module to provide habitation, subsystem stowage and an area allocated to laboratory functions. Electrical power is provided by solar arrays and there are external areas for work performed EVA and for scientific observations.

Some of the missions to be flown on the 28.5 deg inclination Space Station have pointing or attitude requirements that influence the selection of the overall station attitude. Figure 2-2 illustrates the collective pointing requirements of those missions and shows a table which considers the impact of these requirements on the

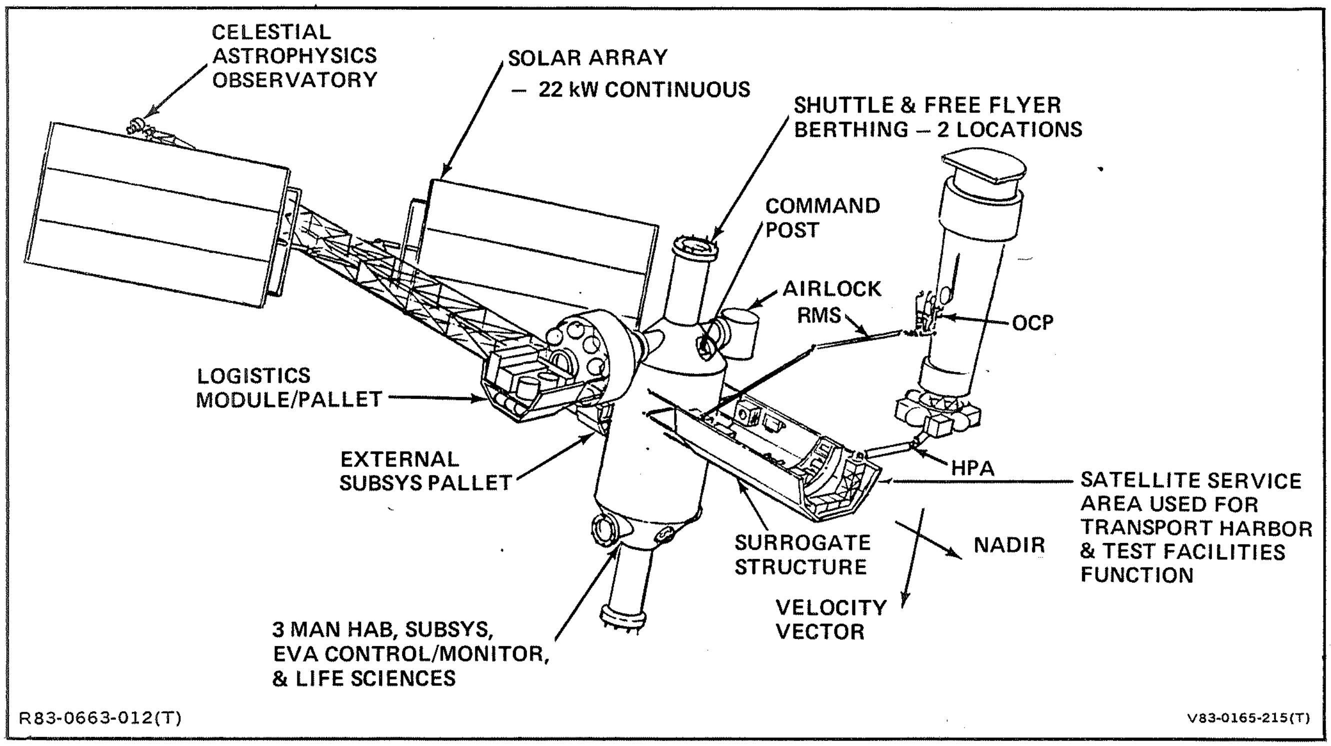
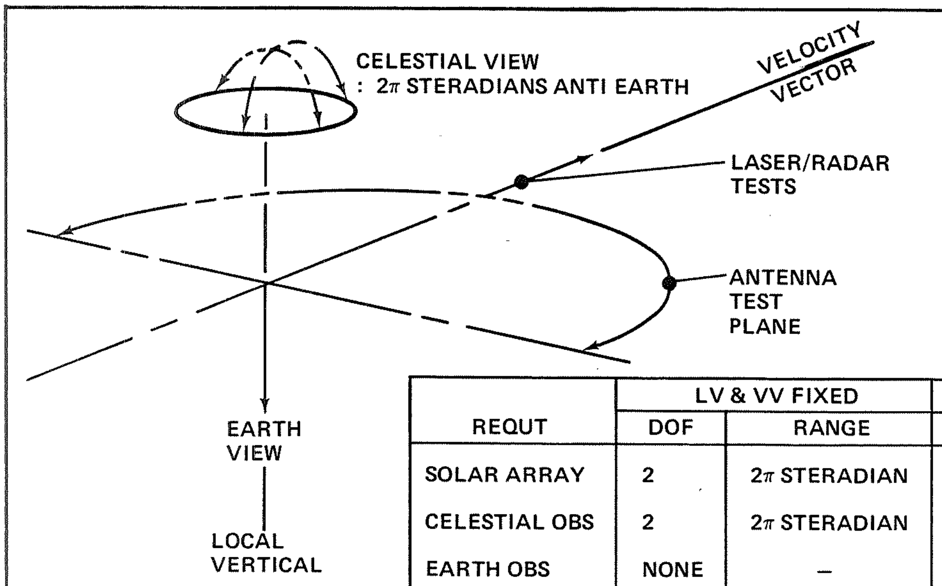


Fig. 2-1 Initial Space Station Configuration



REQUT	LV & VV FIXED		INERTIAL SOLAR FIXED	
	DOF	RANGE	DOF	RANGE
SOLAR ARRAY	2	2π STERADIAN	NONE	–
CELESTIAL OBS	2	2π STERADIAN	2	4π STERADIAN
EARTH OBS	NONE	–	1	180° SWEEP
TEST (VV LOS)	NONE	–	1	360° SWEEP
ANTENNA TEST	1	180° SWEEP	2	180° & 360° SWEEPS
	✓			

VIABLE ORIENTATION OPTIONS

R83-0663-013(T) V83-0165-481(T)

Fig. 2-2 28.5° Space Station – Mission Requirements Impacting Orientation

station in terms of degrees of freedom required and their ranges. Two viable orientations are "Local Vertical & Velocity Vector" fixed and "Inertially Fixed, Sun Pointing." In all but the solar array pointing requirement, the local vertical/velocity vector fixed orientation has the least demanding impact. With solar array pointing, one of the degrees of freedom can be indexed motion. This leaves just one gimballed motion. Furthermore solar array pointing accuracy is not very stringent since cosine losses for, say, 5° misalignment is only 0.5%.

2.1.1 Overall Configuration

Initially, the station has one pressurized core module which houses three men, necessary subsystems, a life sciences laboratory area, and two Extra Vehicular Activity (EVA) command post control/monitor areas. Tunnel extensions provide berthing points for a visiting orbiter. This module is shown and described in Subsection 2.1.2. As will be discussed later, there are two EVA activity areas in the evolved station concept, each of which is controlled and monitored from this module. However with the initial station, only one EVA area and, therefore, one command post is used.

The EVA activity area on this initial station has a 9.15-m long structure, whose cross-section is a trough which simulates the orbiter's cargo bay. This surrogate, with its equipment, caters to satellite servicing; it also performs assembly for space testing and is used as the initial transport harbor.

An external subsystems pallet mounts batteries for dark side power, conversion equipment and control moment gyros (CMG) for attitude control. From this pallet, a mast extends outboard to mount an astrophysics viewing instrument at its tip. The mast length is dictated by the size of the evolved solar array and its movements necessary to track the sun. The tip instrument requires an unocculted view for 2π steradions, anti-earth. The solar array is located so that it does not interfere with EVA activities area, orbiter docking or the unloading of payloads.

The solar array provides 22 kW of continuous power. It is oversized to compensate for loss of effective area due to occultation from the astrophysics instruments and support mast. It has one indexed motion that requires incremental movements to total ±104 deg during one year and a gimballed motion that moves at orbit rate through the sunlit period of the orbit, then backtracks through the dark period.

A logistics module (L/M), described later, is berthed to the pressurized module. An EVA airlock, assumed to be the orbiter developed airlock (A/L), is also berthed to the pressure module, as shown.

Berthing the orbiter to the station uses the orbiter remote manipulator system (RMS) to capture the station and subsequently to berth the two vehicles. The tunnels extending from the pressure module provide two optional berthing ports. The preferred attitude of the orbiter to the station is with its cargo bay parallel to the surrogate. This allows for easy transfer of materials. During the transfer, EVA activity in the surrogate may be restricted and its handling and positioning aid (HPA)-mounted payloads moved out of the way. Alternately, the orbiter can be berthed at 45 deg to this position, which leaves only a small overlap of cargo bay and surrogate but is not as convenient for the transfer of material.

Mission activities to be carried out at this initial station are as follows:

- **Servicing of such satellites as AXAF** - Station involvement starts with launch of a teleoperating maneuvering system (TMS) to retrieve the satellite and rendezvous with the station. The RMS then captures the vehicle and berths it to the HPA arm, as shown in Fig. 2-3. Servicing is performed by an EVA crewman mounted on an open cherry picker (OCP) which is, in turn, mounted to the tip of the RMS. He can control his own movements and/or be controlled by a crewman at the control station in the pressure module. To a great extent, this sharing and monitoring of EVA control function will be influenced by whether or not the EVA man has an EVA buddy or not. A Proximity Operations Vehicle (POV) is available to be flown locally to assist by observing activities with its TV camera.
- **Servicing of such co-orbiting fee flyers as Leasecraft or Eureca** - These free flyers are candidates to mount small space manufacturing experiments, etc. They provide their own propulsion and will rendezvous with the station for servicing of their subsystems and/or their payloads. This is performed in the satellite servicing area and in a similar manner. If their self contained propulsion unit becomes inoperative, the POV or TMS can be sent to retrieve the free flyer.
- **Servicing of POV and TMS propulsion stages** - The HPA mounts the stage while the RMS, with an EVA/OCP crewman, performs the servicing. Fuel resupply is by tank exchange.

- **Assembly of structures or other orbiter payloads for test** - Assembly operations are performed in the satellite servicing area. There is an HPA available to provide a dexterous mount and the RMS can perform assembly tasks either with appropriate end effectors or with an EVA/OCP man at its tip.
- **Earth viewing** - Early on in the program, there is an earth viewing mission required. This instrument is mounted on a pallet at the outboard tip of the surrogate structure to track terrestrial targets.
- **Astrophysics** - IPS mounting for an astrophysics mission is provided at the tip of the solar array mast extension. A 2π steradian, anti-earth coverage is provided.
- **Life sciences experiments** - An area in the pressurized module has been set aside for these experiments.

2.1.2 Three-Man Core Module

The core module, referred to as Habitat/Subsystem Module No. 1 (HSM No. 1), is the primary living and internal work area for a three-man crew. It contains the required subsystems and crew provisions for the Initial Space Station (ISS).

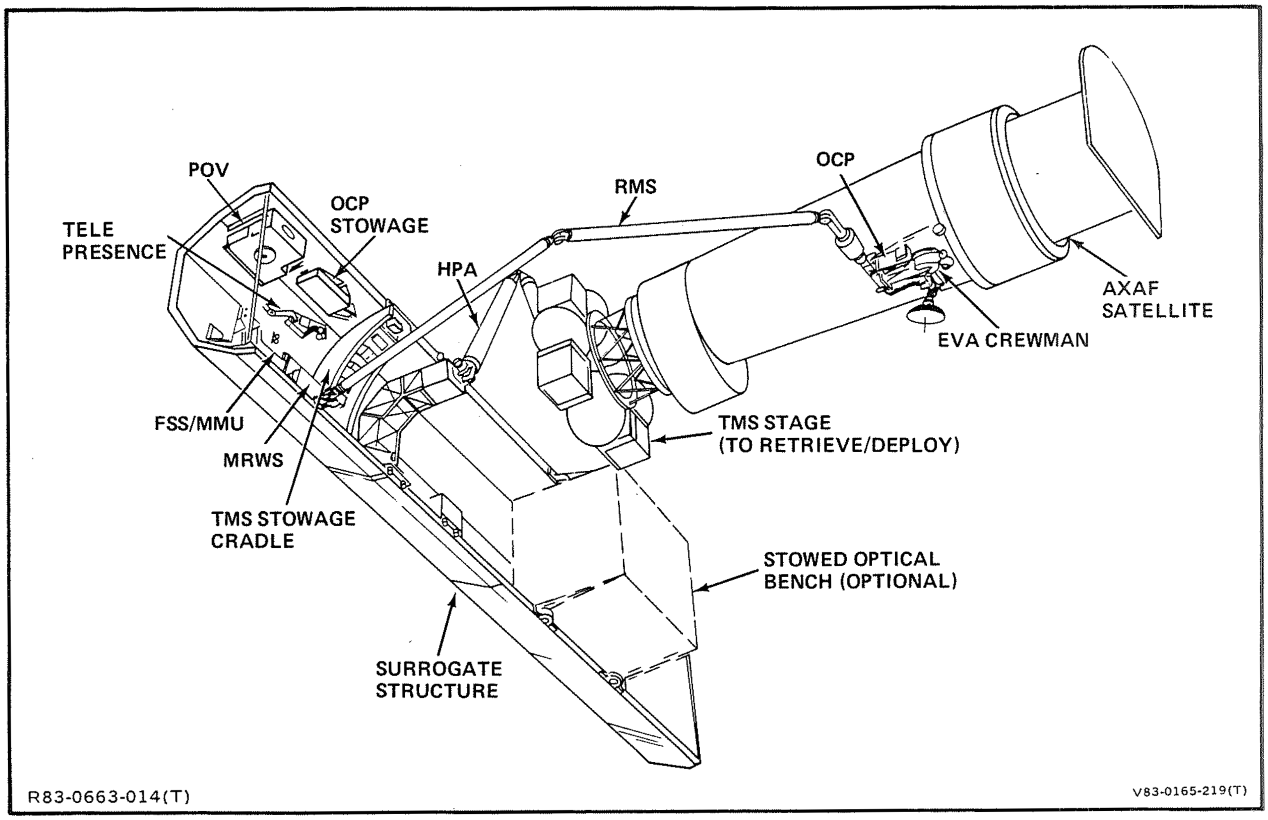


Fig. 2-3 Satellite Service Facility

HSM No. 1 is a 3.45-m dia x 8.7-m long pressurized vessel with two extendable tunnels, each mounting an axial berthing ring to berth the Shuttle during Space Station assembly and during subsequent visits. Each end dome of the pressure vessel contains a work station and two bubble canopy windows to monitor activities in the surrogate structure. Two 1-m diameter tunnels in each end dome, when interconnected with adjacent modules, provide access between the habitats, labs and airlock.

Internally, it is configured to provide crew accommodations, with required subsystems, for a three-man crew. The subsystems hardware is installed in modular racks in the ceiling over the full length of the central aisle and in dispersed areas below the floor boards.

2.1.2.1 **Structures** - Figure 2-4 depicts the structural elements in HSM NO. 1. The module, with stowed radiator panels, has overall dimensions of 3.66-m x 9.15-m long. These dimensions make it possible to fold a surrogate structure around the module when loaded into the payload bay for the first launch (see section A-A in Fig. 2-8).

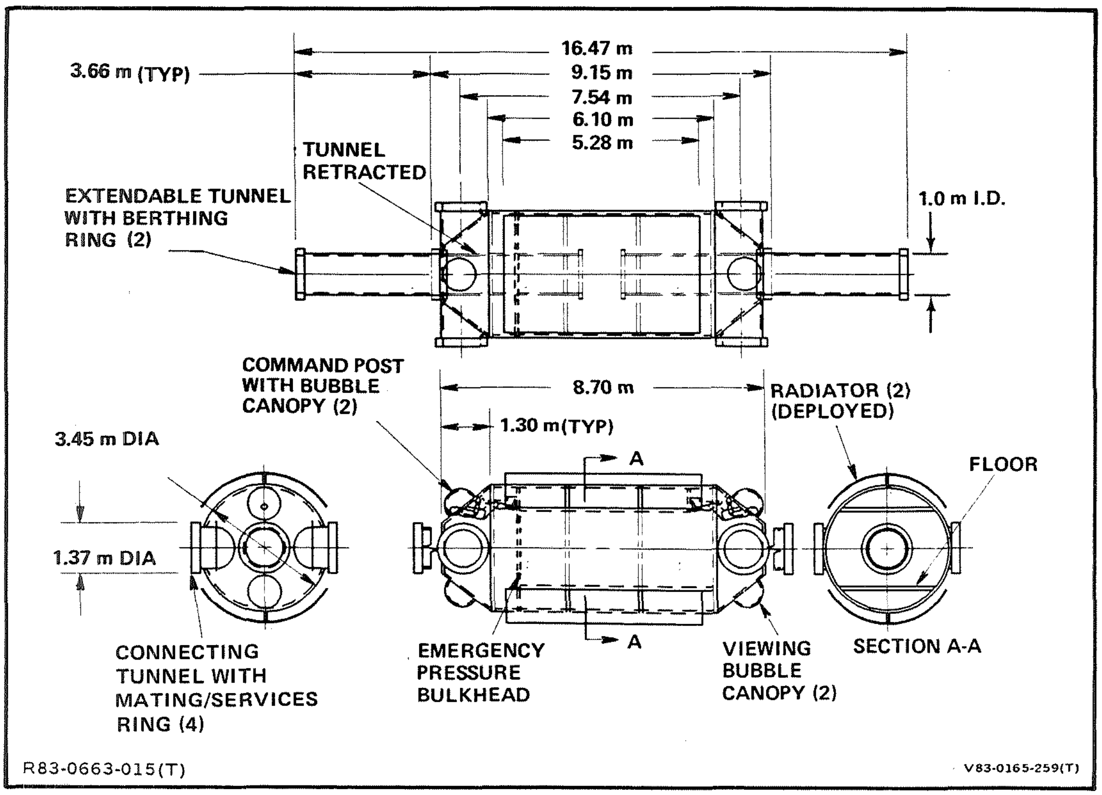


Fig. 2-4 Three-man Core Module (HSM 1) Structural Configuration

The module consists of three elements. The first element is a central cylinder 3.45 m in diameter x 6.10-m long and two end cones each 1.30-m long. The central cylinder mounts two 120 deg x 5.28-m long radiators folded against its outer shell. On orbit these radiators are deployed outward about 0.10-m. Thus, they act as a micrometeoroid bumper shield. The central cylinder contains structure forming a floor and ceiling area. The central aisle floor contains removable floorboards for gaining access to equipment installed below. The ceiling area over the central aisle contains modular racks to which the various subsystem hardware elements are mounted. These units are readily removable for repair, service or replacement.

End cones, 1.30-m long, are attached to either end of the cylinder. Each contains two bubble canopies and two connecting tunnels located 90 deg from the canopies. One bubble canopy, serves a control station used as a command post, while the other canopy is used for general external viewing. Each connecting tunnel mounts a mating/service ring at its end. These are used for interconnecting adjacent modules. When the modules are connected, occupants of the Space Station can move between modules in an IVA condition through 1.0 m inside diameter tunnels.

Tunnels 3.66-m long, with berthing rings, are installed at each end of the domes. At launch these tunnels are retracted into the cylindrical portion of the module. In this configuration the module total length is 9.15 m. When the module has been deployed from the payload bay, the tunnels are manually deployed by EVA astronauts, and permanently attached in place, as shown in Fig. 2-4. The total length of the module is now 16.47 m; the orbiter can berth to the rings on either extended tunnel.

A pressure bulkhead is located inside the cylindrical portion of the module. In the event of pressurization loss in the module, the three astronauts can seal the bulkhead and live in the unaffected volume until rescued.

2.1.2.2 **Mechanisms/Berthing** - The habitat/subsystem module contains several major mechanical devices needed to make this a viable configuration; they are as follows:

- **Extendable Tunnel** - 3.66-m long tunnels are located at each end of the module. These units are retracted into the module at earth launch. The berthing rings at the end of the tunnel are butted up against the end dome while the rest of the tunnel is located inside the module. EVA astronauts deploy these tunnels manually to its fully extended position. The astronauts then permanently attach the end flange of the tunnel to the inner surface of the module domes. A pressure seal is achieved when the end flange is in place. The inside diameter of the tunnel is 1.0 m.
- **Berthing Ring** - The berthing ring is configured to mate two elements to each other (i.e., module-to-module, orbiter-to-module, etc). This ring also contains umbilical interfaces for subsystem support (water, air, electrical and gases)
- **Radiator Panels** - At launch, the two radiator panels are stowed close to the outside diameter of the module in order to fit within the folded surrogate structure in the orbiter P/L bay. After removal from the payload bay, the panels are deployed from the module using remotely controlled actuators. Radiator deployment will place the units about 0.10 m away from the outside shell of the module, thus providing an effective meteorite protection shield.
- **Pressure Bulkhead** - This bulkhead is installed across the full diameter of the module. Its central portion contains a 1.42-m dia opening for passage of occupants within the module. In the event of damage to the module pressure shell, a hatch can be closed over the bulkhead opening. Three astronauts can survive in the undamaged volume of the module until rescue is effected.

2.1.2.3 **Crew Accommodations** - These include the crew quarters, dining area, food stowage/preparation, waste management installation, equipment installation, crew support equipment, control station and crew provisions. Environmental control/life support system (EC/LSS), controls and displays (C&D) and lighting are other interior equipments with which the crew interfaces.

Figure 2-5 depicts a three-man core module.

2.1.2.3.1 Compartment Partitions - The three-man core module (HSM No. 1) requires compartment partitions that will be erected by the flight crew in orbit. During deployment, the extendable tunnels, described in Subsection 2.1.2.1 require a clear path opening 1.42 m in diameter down the central part of the cylinder. When these tunnels have been extended, the central aisle can be reduced to 0.9 m to provide reasonable volume for the wardrooms, galley and waste management areas. The partitions are folded at earth launch, out of the way of the tunnel extension volume, and are then hinged into position on orbit to form the compartments as shown in Fig. 2-5.

2.1.2.3.2 Area Layout - HSM No. 1 is configured with its floor and ceiling running almost the entire length of the cylinder. Elements installed in this area are in the conventional architectural manner. The elements in the end domes are installed perpendicular to those in the cylindrical cross section.

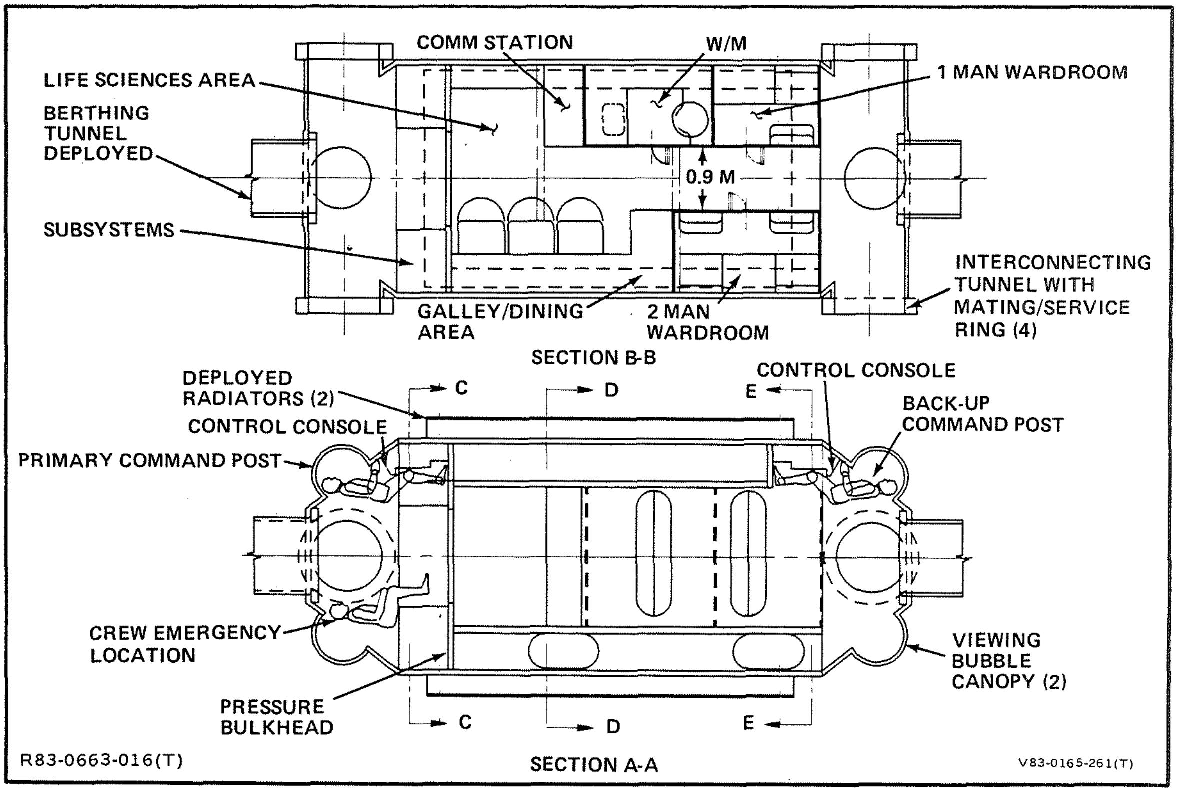


Fig. 2-5 Three-man Core Module (HSM 1) — Interior Config — (Sheet 1 of 2)

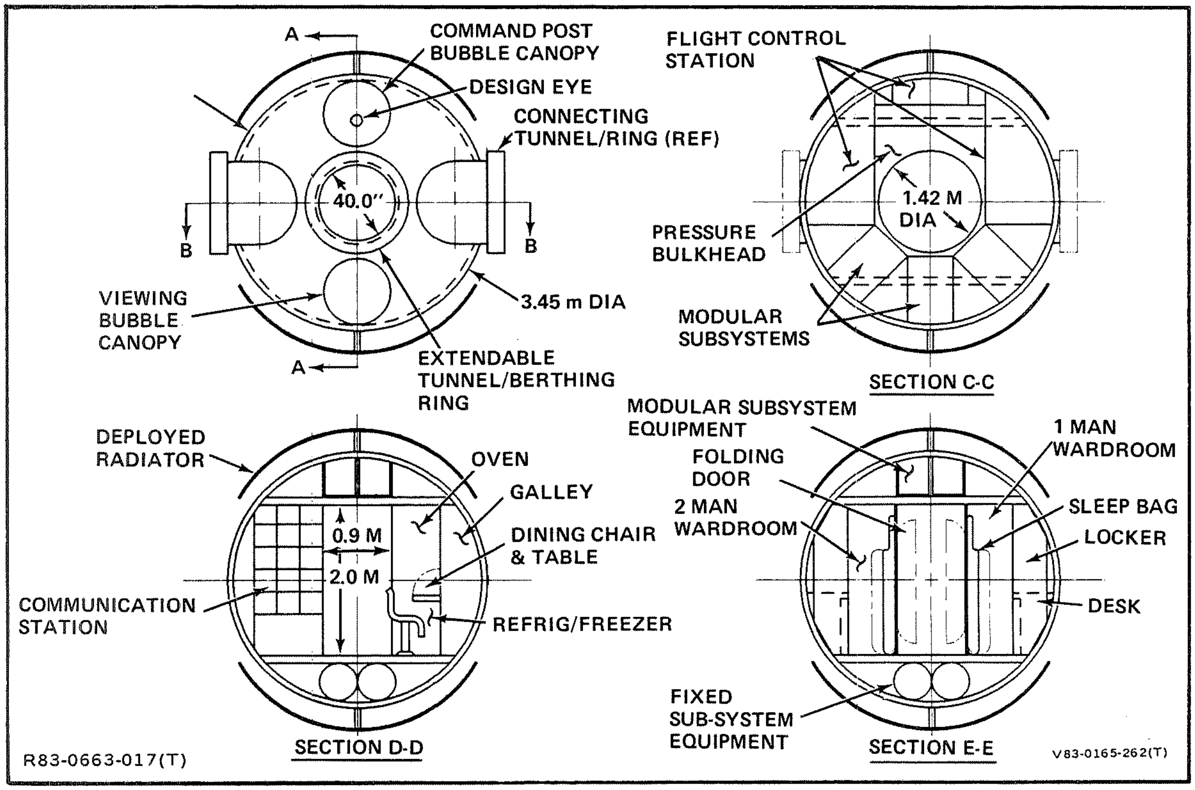


Fig. 2-5 Three-man Core Module (HSM 1) — Interior Config — (Sheet 2 of 2)

2.1.2.3.3 **Crew Wardrooms** - HSM No. 1 contains a one-man and a two-man wardroom. The one-man wardroom has a floor area of approximately 1.28 m^2 and the two-man wardroom about 1.75 m^2. The one-man wardroom contains a Skylab-type sleep bag, writing desk, chair, floor to ceiling locker and a small entertainment center. The two-man wardroom is similarly equipped but with two sleep bags. Entrance to the wardroom is gained through folding doors, similar to those used in Skylab.

2.1.2.3.4 **Food Preparation & Dining** - A dedicated area has been established for food stowage, preparation and dining. Because of stowage volume limitations, a seven-day supply of food is stored in the habitat. A refrigerator and freezer are provided for limited stowage of shelf stable food. The bulk of the 90-day food supply is located in the logistics module. An oven is provided for preparation of food. The dining area can seat three crewmen at one time for each meal. Waste is packaged and stored in the logistics module.

2.1.2.3.5 **Waste Management (W/M)** - A separate compartment is provided for hygiene and waste management functions. This room contains a fecal/urinal collector, body cleansing system and vomitus collector.

2.1.2.3.6 **Life Science Area** - An area has been set aside for crew life science functions. The floor area available (including the central aisle) is about 2.14 m^2. This area is to be used for crew conditioning on the 90-day mission cycles. Since this area is limited in size, the devices used for this function will have to be movable and stowed when not in use.

2.1.2.3.7 **Subsystem Equipment** - The subsystem elements are installed in modular racks located the full length of the ceiling above the central aisle. Controls for these subsystems (EC/LSS, etc) are all located on the units in the ceiling. Wiring, ductings, etc are routed above the ceiling outboard of the subsystem modular elements. The area below the floor contains all fixed hardware items, tubing runs, wiring and ducts. Removable floorboards in the central aisle provides access to the centrally located items.

2.1.2.3.8 **Command Post** - A command post is located at either end of the habitat. A bubble-type dome window is located in the conical sections of the module. An astronaut can stand at his control station with his head inside the bubble and obtain a downward view into the surrogate structure. From this station he can monitor and control all EVA functions in the surrogate structure. The command post will extend down partially into the cylindrical section of the module. The control station at the top of the module will be used as the prime command post and the one at the bottom will be used for backup.

2.1.2.3.9 **Communication Control Station** - A small control station is located in this module. All communications and data management functions are monitored here.

2.1.2.3.10 **Lighting** - Fixtures are located in discrete areas to provide the required internal lighting. Light fixtures are placed on the exterior surfaces to light all external work areas on the Space Station.

2.1.3 External Subsystems Pallet

Some subsystems and their elements can be simplified if they can be serviced by man. This manned servicing is performed most conveniently in shirtsleeves and, therefore, equipment which benefit most from this servicing should be located in the pressurized module. However, this demands premium volume which leads to a larger module or to more of them, and so to higher cost. It follows, therefore, that where practical, subsystem elements should be located outside the pressure module, particularly if they are serviced by being replaced or are hazardous.

Batteries, some power conversion equipment and control moment gyros are suitable candidates for being mounted externally. These items are ground-mounted on a pallet that is compatible with the orbiter cargo bay. The Spacelab pallet is a prime candidate for this job. Location of the external subsystems pallet is shown in Fig. 2-2. It is mounted off the pressure module and it supports the solar array mast.

2.1.4 Initial Surrogate Bay

The EVA activity area on this initial station requires a structure on which to mount EVA equipment. The surrogate bay has a 9.15-m long structure, whose cross section is a trough which simulates the orbiter cargo bay and can provide the same payload mounting points. Thus, orbiter mission equipment developed for mounting in its cargo bay can be used on the Space Station and mounted directly in this surrogate payload bay structure. Similarly, payload transported to the SS orbit in the STS Orbiter can be transferred directly from the Orbiter to this surrogate structure. A further benefit is that, if orbiter equipments are used for station EVA activities and their locations are duplicated, astronaut training is less costly. In the initial station, the surrogate caters to satellite servicing and to assembly for space testing.

The outboard surface of this structure can mount radiator panels to dissipate heat generated by the equipments mounted in the bay.

2.1.5 Initial Observatory

As a discipline, celestial observation requires 2π steradians, anti-earth viewing. Consequently, the instruments must be mounted so that their viewing is not occulted by the solar array and its movements. The array itself has viewing requirements that can only be met by mounting the array on the outboard, anti-earth side of the Space Station. The result of these requirements is that the observatory must be the most anti-earth outboard facility on the station. In the initial station concept, the observatory comprises one instrument. This is all that can be launched as part of the operational assembly resulting from two orbiter launches. Another orbiter launch can, of course, bring up more celestial instruments with their necessary pointing systems. They could be added to the observatory and handled as described in Subsection 2.2.1 for the evolved Space Station observatory.

2.1.6 Airlock

The airlock is that utilized by the orbiter. It is sized to hold two crew members with full extra mobility unit (EMU) equipment. All controls for operation of an airlock are located inside this compartment. The airlock has two ingress/ egress hatches.

On the initial Space Station, it is connected to HSM No. 1, as shown on Fig. 2-1. On the evolved Space Station the airlock is relocated to the mating ring on one of the laboratories, this is described in Subsection 2.2.

2.1.7 Logistic Module

Figure 2-6 illustrates a concept for a logistics module (L/M). For the evolved space station (ESS) three modules will be required. One is attached to the Space Station, one is kept on earth being prepared for flight, and one is a spare. These extra modules provide for rapid turnaround for the 90-day missions.

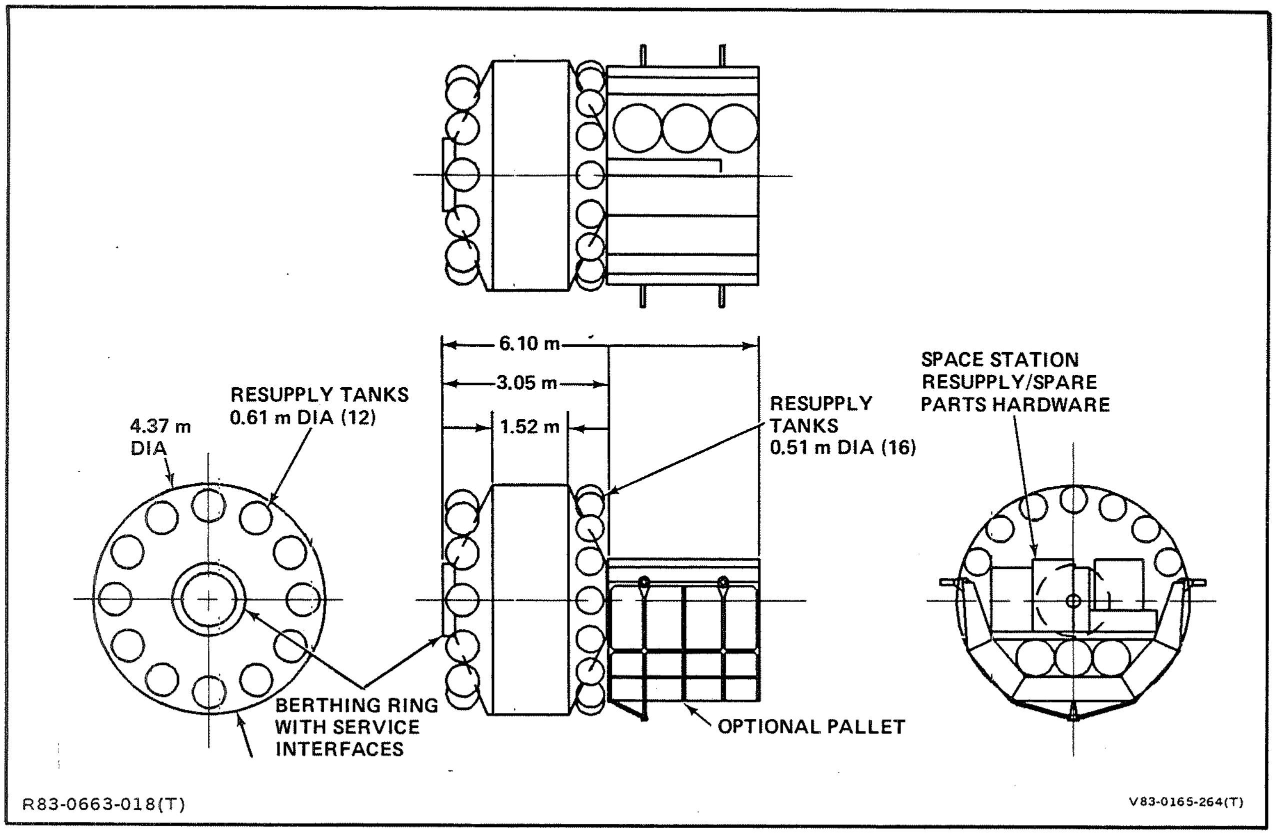


Fig. 2-6 Logistics Module

The L/M is configured as a pressure module 4.37 m in diameter x 3.05-m long. Internally, it contains resupply items for direct access by shirtsleeved crewmen. Externally, the module mounts resupplies which are stored under high pressure or have other safety issues. An STS pallet can be attached to the end of the module on those flights carrying logistics for EVA activities. The total length of the L/M with pallet is 6.10 m.

The pressurized module of the L/M has an approximate volume of 29.0 m^3 with about 20.0 m^3 available for storage. The optional pallet can accommodate about 25.0 m^3 of hardware items.

The end of the pressurized compartment will contain a berthing ring with mechanical, electrical, fluid and gas line interfaces located around its periphery. Pressurized gases and toxic fluids will be plumbed outside the pressurized interface between mating modules. The L/M is connected to the habitat module via the berthing ring interface.

The L/M will be berthed to Habitat/Subsystem Module No. 1 in the ISS configuration and HM No. 2 in the ESS configuration. The habitats are configured to accept the L/M on either end of the module. Once it is berthed, the hatch is opened so that the crew can move freely between mated modules in an IVA condition. The habitat is configured to stow a limited supply of food, clothes, etc. For this reason the Space Station should be off-loaded with equivalent items that are brought on board from the L/M. All wastes are stored in the L/M for return to earth.

Figure 2-7 lists elements to be delivered via the L/M to the Space Station.

2.1.8 Launch & Assembly

The initial operational Space Station can be transported to orbit in two Shuttle launches. The arrangement of equipments in the orbiter cargo bay for each of the two launches is shown in Fig. 2-8.

The top sketch shows the payload bay configuration of STS No. 1. The elements installed in the payload bay include the following:

- Habitat/Subsystem Module No. 1 (3.45-m dia x 9.15-m long)

ITEM	VOLUME, m^3
• FOOD (INCLUDING LOCKERS)	
– SHELF STABLE	4.6
– FROZEN	2.1
– REFRIGERATED	1.4
• PERSONAL HYGIENE	2.3
• HOUSEKEEPING FACILITIES	1.0
• CREW GARMENTS	1.1
• WATER	4.3
• EVA SUPPLIES	TBD
• EC/LSS FILTERS, CHEMICALS, SPARES, ETC	1.1
• GASES (HYDRAZINE, NITROGEN, OXYGEN, ETC)	TBD
• SPACE STATION SPARES	TBD
• OTV SPARES	TBD
R83-0663-019(T)	V83-0165-160(T)

Fig. 2-7 Ninety-Day Resupply Requirements — 9-Man Crew

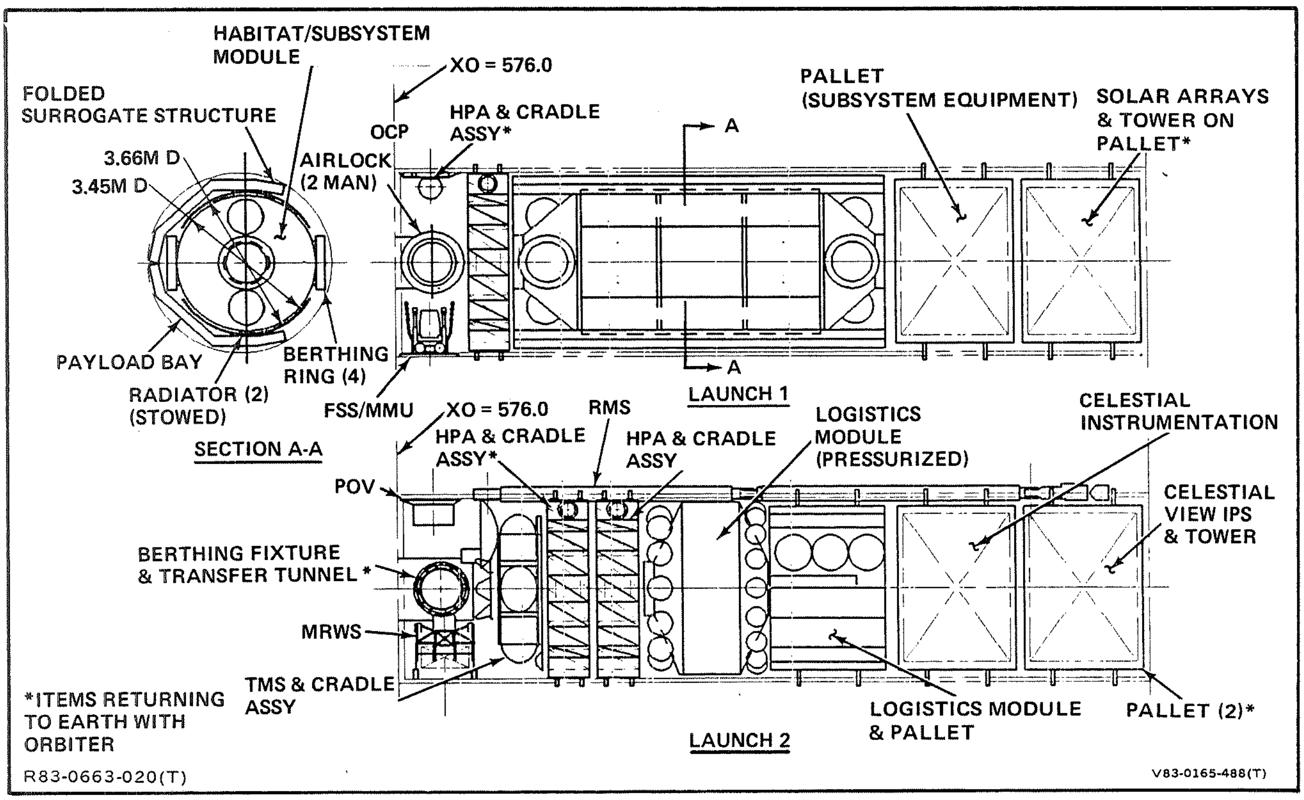


Fig. 2-8 Orbiter Payload Bay Launch Configurations

- Surrogate Structure, 9.15-m long (folded around HSM No. 1) - this scheme saves launch costs for a separately carried surrogate but at the expense of pressure vessel diameter
- Airlock, sized to accommodate two men with EMUs (carried in the STS external airlock location)
- Flight Support Station/Manned Maneuvering Unit (FSS/MMU), (STS development hardware)
- Open Cherry Picker (OCP), (STS Development hardware)
- Pallet with subsystem equipment
- Pallet with solar array and mast (pallet remains with STS)
- HPA and Cradle Assembly (NASA funded development by Grumman - remains with STS).

Assembled components carried on this first launch are shown in Fig. 2-9. Main equipments used in assembly are the orbiter RMS, the assembly HPA, the OCP mounted to the RMS end effector and carrying an EVA crewman, then finally an EVA man with his MMU.

A simplified scenario for assembling the ISS with the first launch equipment is as follows:

2.1.8.1 STS LAUNCH No. 1

- Using the STS RMS, the 9.15-m Habitat/Subsystem Module No. 1 and the wrapped around surrogate structure are deployed clear of the orbiter
- An EVA crewman enters the habitat module and pushes the 3.66-m tunnels outward until seated on the internal surface of the module end domes. The astronaut fastens the tunnels in place. The module is now configured with a tunnel extending 3.66 m out from each end.
- The RMS mounts the module to the HPA for completion of the buildup
- The STS RMS removes the 3.15-m long surrogate structure from around the habitat. An EVA astronaut operates the mechanism which opens the surrogate structure to its full width and attach necessary cross bracing
- The STS RMS and an EVA astronaut move the surrogate structure to its position against the side of the habitat module and attach it in place
- The STS RMS removes the external subsystem pallet, with its contents, from the payload bay and locates it against the opposite side of the habitat module. The EVA astronaut secures it in place

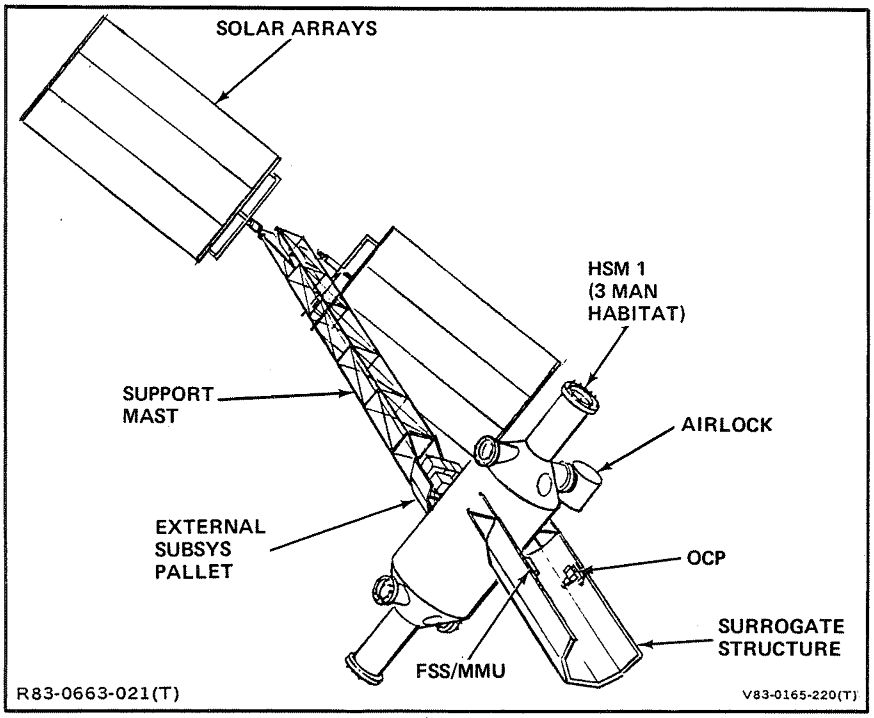


Fig. 2-9 Initial Space Station — Assembly of Launch 1 Components

- The STS RMS removes the airlock from the payload bay and places it in position against one of the short tunnels at the top part of the habitat module. The EVA astronaut will secure the airlock to this tunnel
- The STS RMS removes the contents from the pallet remaining in the payload bay. EVA astronauts will assemble these units into a solar array with supporting mast. This unit is attached to the far end of the external subsystems pallet. The mast is presently conceived as being assembled from about five compact folded segments, each of which is carried by a tethered EVA/MMU crewman, attached, then unfolded. Solar array panels are "SEPS" extensible type and will be transferred and installed by the EVA/MMU crewman, then deployed
- The STS RMS removes the FSS/MMU and OCP from the front of the payload bay and re-installs it in the surrogate structure.

The orbiter then returns to earth leaving the station quiescent and unmanned.

In point of fact, the only element missing from this first launched assembly, which would enable it to be left manned, is necessary logistics such as food, etc. Normally these are supplied by a routine logistics visit with a module but, for this first launch, these supplies could be located somewhere in the pressure module, enough for two men for the period until the second launch.

2.1.8.2 STS Launch No. 2

The second launch orbiter cargo bay arrangement is also shown in Fig. 2-8. The standard docking module shown in the STS payload accommodations document is designed for docking loads and provides an airlock. It is suggested that a berthing fixture, such as that shown in Fig. 2-10, would occupy less cargo bay length and be less of a payload weight penalty. Equipments carried on this launch include:

- Logistics Module with pallet and equipment attached (90-day supply for three men
- Pallet with celestial instrumentation (pallet remains with STS)
- Pallet with celestial viewing IPS and mast (pallet remains with STS)
- HPA and cradle assembly
- HPA and cradle assembly (remains with STS)

- TMS and cradle assembly
- POV
- Manned Restraint Work System (MRWS)
- RMS
- Berthing ring/transfer tunnel (remains with STS)
- RMS (remains with STS).
- The STS RMS removes the Space Station HPA and cradle assembly from the payload bay and installs it in the surrogate structure

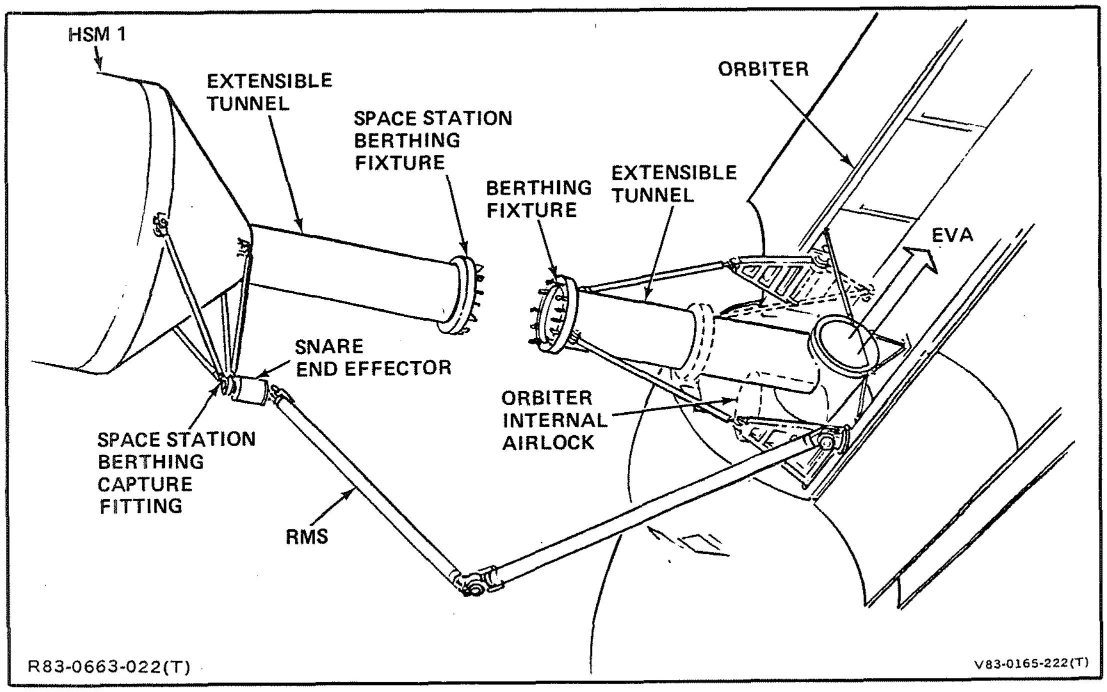


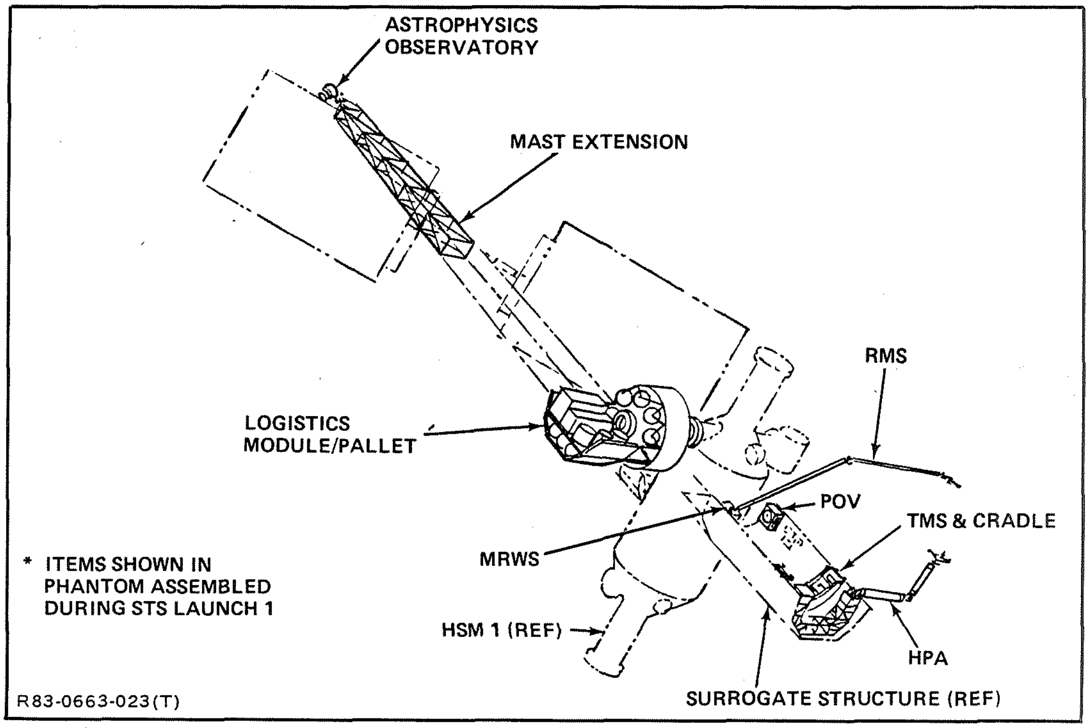
Fig. 2-10 Proposed Berthing System for Orbiter to Space Station Mating

Figure 2-11 shows where these second launched elements are located (the first launched items are shown in phantom). The orbiter berths to the tunnel extension of the pressure module and, using its RMS and EVA man with his aids, the items are assembled to the first launch assembly.

The simplified scenario for assembly is as follows:

- The STS RMS acquires the space station and berths to the extended tunnel on habitat/subsystem module No. 1
- The STS HPA removes the logistics module from the payload bay and berths it to the habitat.
- The RMS removes the mast extension segments from the pallet in the aft part of the payload bay. EVA astronauts will assemble these segments into a celestial viewing tower and platform.

- The STS RMS removes the TMS and cradle assembly from the payload bay and installs it in the surrogate structure
- The STS RMS removes the MRWS from the payload bay and installs it in the surrogate structure
- The STS RMS removes the POV from the payload bay and installs it in the surrogate structure
- The STS RMS removes the second RMS from the payload bay starboard stowage and installs it on the surrogate structure shoulder
- The orbiter berths to the observatory tower, via its HPA. The RMS then transfers and installs the astrophysics instrument, with its IPS, to the tower tip.
- All items comprising the ISS have now been assembled.



V83-0165-221(T)

Fig. 2-11 Initial Space Station — Assembly of Launch 2 Components

2.2 EVOLVED SPACE STATION

The Space Station can grow to the configuration shown in Fig. 2-12. Basically, its physical evolution will be accomplished by the addition of pressurized modules, the addition of surrogate structures to increase EVA activity area, more solar array area to meet increased electrical power demands and increased observation capability.

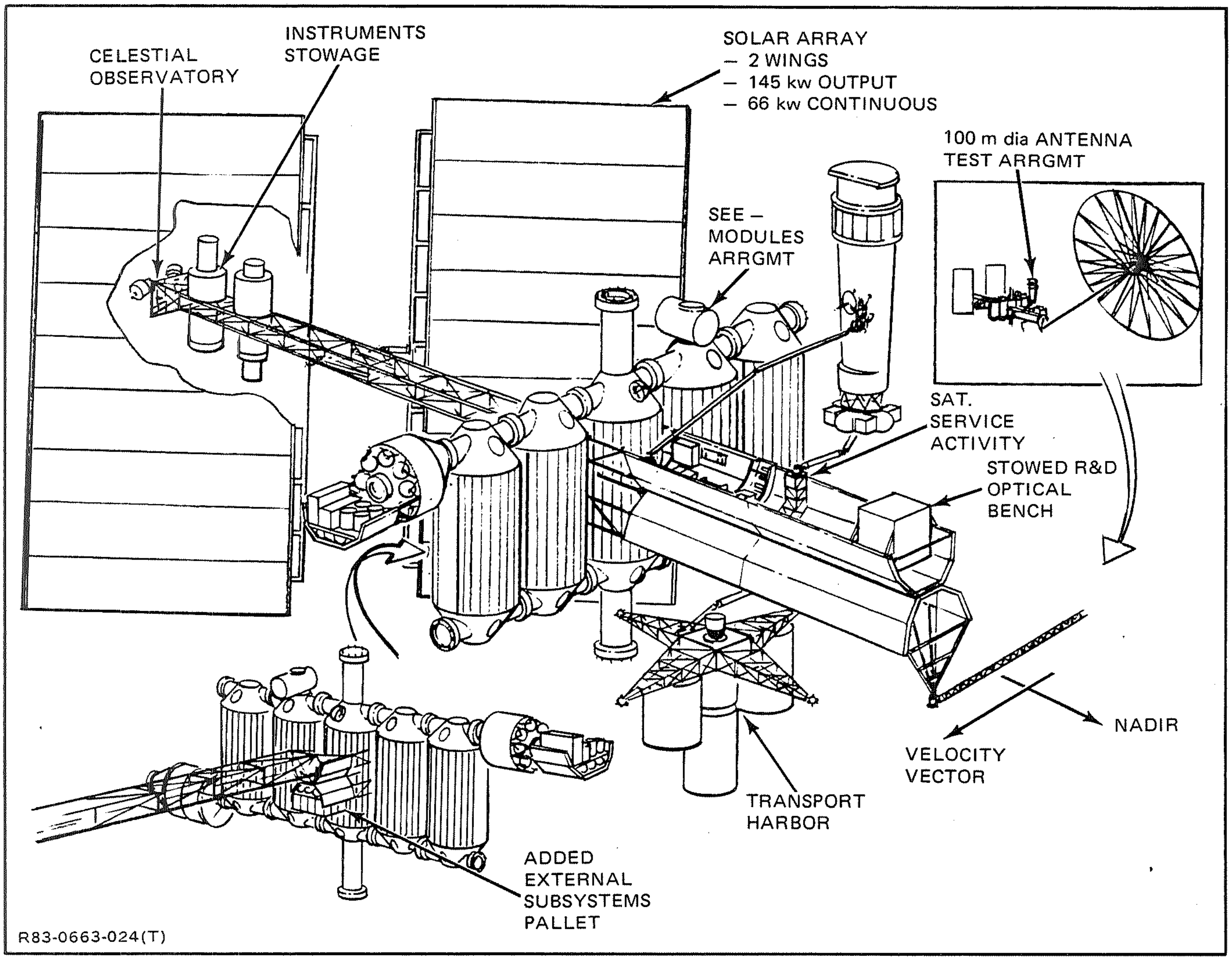


Fig. 2-12 Evolved Space Station Configuration

2.2.1 Overall Configuration

In the early years, a space manufacturing laboratory and a science/R&D laboratory are added to the "initial" station complex. These laboratories use the same shell as the "initial" station habitation and their interior layouts are discussed in Subsection 2.2.4. Manning of the station increases incrementally to a total of nine permanent residents. Therefore, by 1993, two 3-man habitation modules to house six more men are added to the complex. These two modules are replications of the initial habitation module and appear to be more cost effective than a new 6-man module. The costing section (Volume II - Book 3) reflects this conclusion. Thus, the "evolved" station has five identically sized pressurized modules. Their arrangement is shown in some detail in Fig. 2-13. The modules are attached to each other by the tunnels that extend from each cone end. There is now a redundant escape path from each module, and inter-module traffic flow is clear of the main activity area in each module. The outboard tunnels on each of the two end modules can mount logistics modules, air-lock, added labs, etc, as shown in the figure.

When three of the modules are launched to orbit in the Shuttle, a 9.15-m length of surrogate structure is launched with each in the wrapped around launch configuration described in Subsection 2.1.8. Now, with the initially launched surrogate structure, there are four lengths which, when paired, provide the lengthened satellite service/space test EVA area structure and the transport harbor for upper stage service and turnaround.

These incremental additions of pressure modules and of surrogate structures are handled by the orbiter RMS. When the orbiter is berthed to either of the two berthing tunnels extending from the initial habitat module, its RMS unloads the particular equipment and places it in its desired location. The station RMS with an EVA/OCP man at its tip will assist in these operations.

Activities in the satellite service/space test facility continue to be as described for the initial station in Subsection 2.1.1. The facility is now larger to permit storage in the lengthened surrogate and the addition of station equipments which will be identified as later studies evaluate operations in more detail.

One of the space tests identified for the space test facility is the deployment, checkout and calibration of a 100-m dia phased array antenna. This antenna was assumed to be the wire wheel, self-deploying type presently under development by

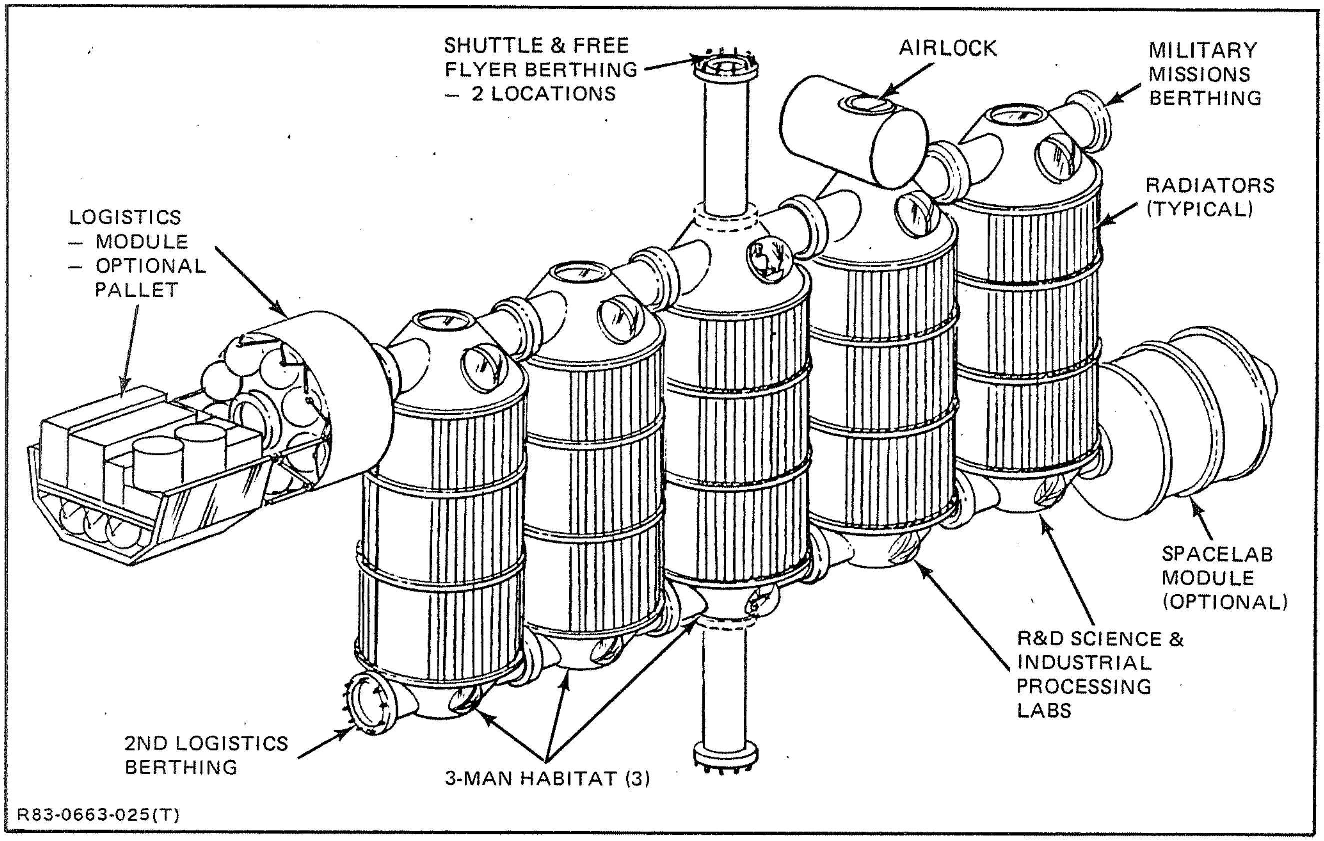


Fig. 2-13 Evolved Space Station — Arrangement of Pressurized Modules

Grumman. As indicated in Fig. 2-12, it will be mounted by its feed end on a pallet that is mounted in either of the two surrogates, whichever is convenient at the time. A following or leading co-orbiting RF source free flyer (using either a POV or TMS) transmits to the antenna while the antenna mast is swung through a 180 deg arc about the local vertical. The antenna is now rotated, say, 30 deg about the mast axis and is then swung back through the 180 deg arc. This is repeated until the antenna surface has been covered and calibrated.

Another space test is the checkout of an optical bench. This bench structure and appendages are contained in a 4-m cube envelope for transport to orbit in the shuttle, but for operation the envelope is extended to be 40-m long x 4-2 m. As shown in Fig. 2-12, the cube envelope is mounted in a surrogate structure. The extension to 40 m will be outboard along the local vertical, then, at the end of the test, the facility will be retracted back to its 4-m cube and returned to earth or, if necessary or convenient, it may remain stored on the surrogate.

OTV turnaround is shown in Fig. 2-14. The OTV is assumed to use storable propellants contained in a single tank assembly. Refueling is by exchanging the empty tank assembly for a ground-filled one. The propulsion stage is serviced at the station. A proposed sequence is shown in the figure. The returned OTV is captured by the RMS and berthed to an HPA arm. The empty propellant tank is removed and stowed by the RMS. Now the propulsion stage subsystems are serviced. A new propellant tank assembly is installed and connected; then, the OTV payloads are installed, as shown. Check out, separation and launch follow.

Power requirements are increased incrementally by adding, an area equivalent to the initial station array to provide a total of 44 kW continuous. Later on, a further addition of the same increment brings the total power output capability to 66 kW continuous, sufficient until at least the year 2000. These increases are achieved by a tethered EVA man on his MMU/WRU transferring each folded solar array panel to its mount on the cross arm, then actuating its SEPS-type deployment.

Extra batteries are required to provide the increased power demand on the dark side of the orbit. These are mounted on the added subsystems pallet, located next to the existing external subsystems pallet (see Subsection 2.2.2).

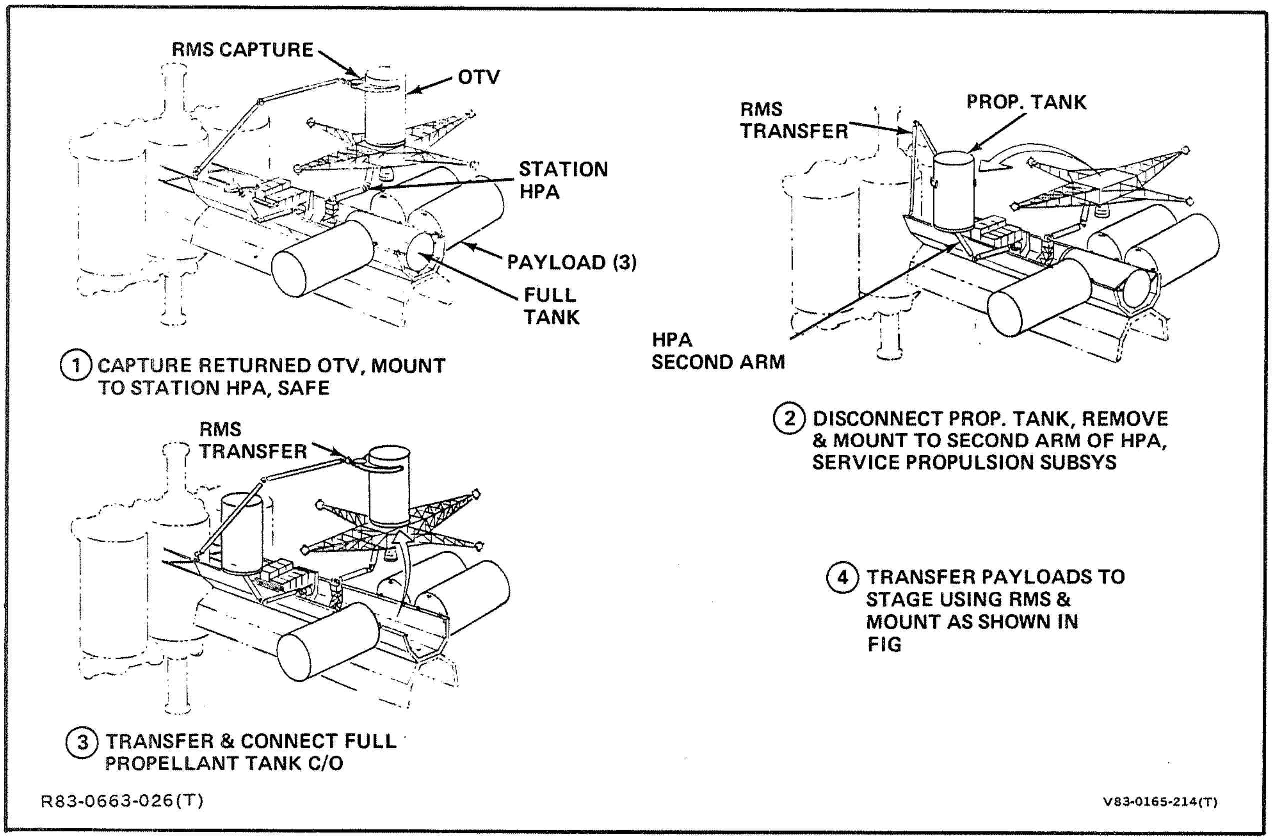


Fig. 2-14 OTV Turnaround

An earth viewing Terrestrial Meterological Support observation system is to operate for a period subsequent to the ISS. It will be accommodated at the outboard end of the surrogate structures where it will mount on an IPS for fine pointing.

It may be necessary to inhibit its viewing for periods while such space tests as antenna deployment calibration or optics test on the extended test bench is going on.

The number of astrophysics instruments increases to six during the evolved period. One problem with many instruments which need physical steering is that their conflicting pointing requirements would lead to either sequential viewing restrictions or to a vast platform that ensures no conflict. For this Space Station, we have selected sequential viewing. As the station evolves, more instruments are brought to this observatory, but only two instruments can operate at one time. Each is mounted to an IPS and operates for a predetermined period. The instruments not in use are stowed. Being a manned station, the instruments can be changed out at the end of the period or if a special event viewing is required.

The stowed instruments are mounted as shown to the outboard section of the mast. Here, they do not inhibit solar array movements. An EVA-operated RMS is mounted in this region to handle the transfer of instruments from IPS mount to stowage and the reverse. Delivery of these added astrophysics instruments to the observatory is by the shuttle berthing with its HPA end effector to a berthing point on the observatory structure. The RMS transfers the instruments from cargo bay to observatory mount. This delivery operation can only take place at a time of the year when the indexed position of the solar array and its associated rotation leaves a suitable corridor.

2.2.2 Additional External Subsystem Modules

Additional power requirements are satisfied by adding solar arrays to the source, and more batteries for dark side power. These batteries, with their associated power conversion equipment, etc, are mounted on a subsystems pallet on the ground and brought up in the Shuttle. When berthed to the station, the orbiter RMS transfers this pallet to be back-to-back with the existing external subsystems pallet. This second pallet has an identical primary structure as the first pallet, with perhaps modified secondary structure to mount more batteries.

2.2.3 Additional Surrogates

Subsection 2.1.4 described the basic surrogate bay structure. It comes in 9.15-m lengths to be compatible with the pressure module around which it is wrapped for launch. In the evolved station configuration, there are two EVA areas, each of which requires an 18.3-m long surrogate structure. Thus, four 9.15-m lengths are required and these, conveniently, can be launched with four pressurized modules. The initial station provided a 9.15-m length and the equipments for satellite servicing. This is now extended to be a full length structure with room for mounting space test assemblies and stowage.

The second EVA activity area is for OTV turnaround and uses a full length surrogate structure that is mounted back-to-back with the other surrogate. Equipments for this activity are discussed in Subsection 2.2.1.

2.2.4 Laboratories

The laboratory modules are sized the same as the habitat modules. They are connected to the habitats by means of the right angle tunnel connectors. Hatches are located at the ends of each tunnel so that the laboratory can be isolated from the rest of the habitable complex. The interior layout of each laboratory will differ from that of the habitats.

Research areas under consideration include, but are not limited to, these discussed in the following subsections.

2.2.4.1 Biological/Pharmaceuticals - This lab, using the electrophoresis and tissue culturing processes, investigates the areas of urokinase, growth hormones, vaccines, interferon, insulin, HLA, Antigens and Erythropoientin.

2.2.4.2 Inorganic Systems - This lab, using the directional solidification and crystal growth processes, investigates the areas of core magnets, super conductors, waveguides, turbine blades, silicon ribbons, crystal detectors, semiconductors, laser glasses, specialty windows, specialty crystals and catalysts.

2.2.4.3 Industrial Processes - This lab investigates the processes for producing refined metals of high quality. Figure 2-15 depicts a typical layout of an industrial process laboratory with 12 ovens. The raw stock is loaded on one end of the oven and the processed metal is captured on the other end as it comes out of the oven. Filled collection containers are transported to the logistics module for return to

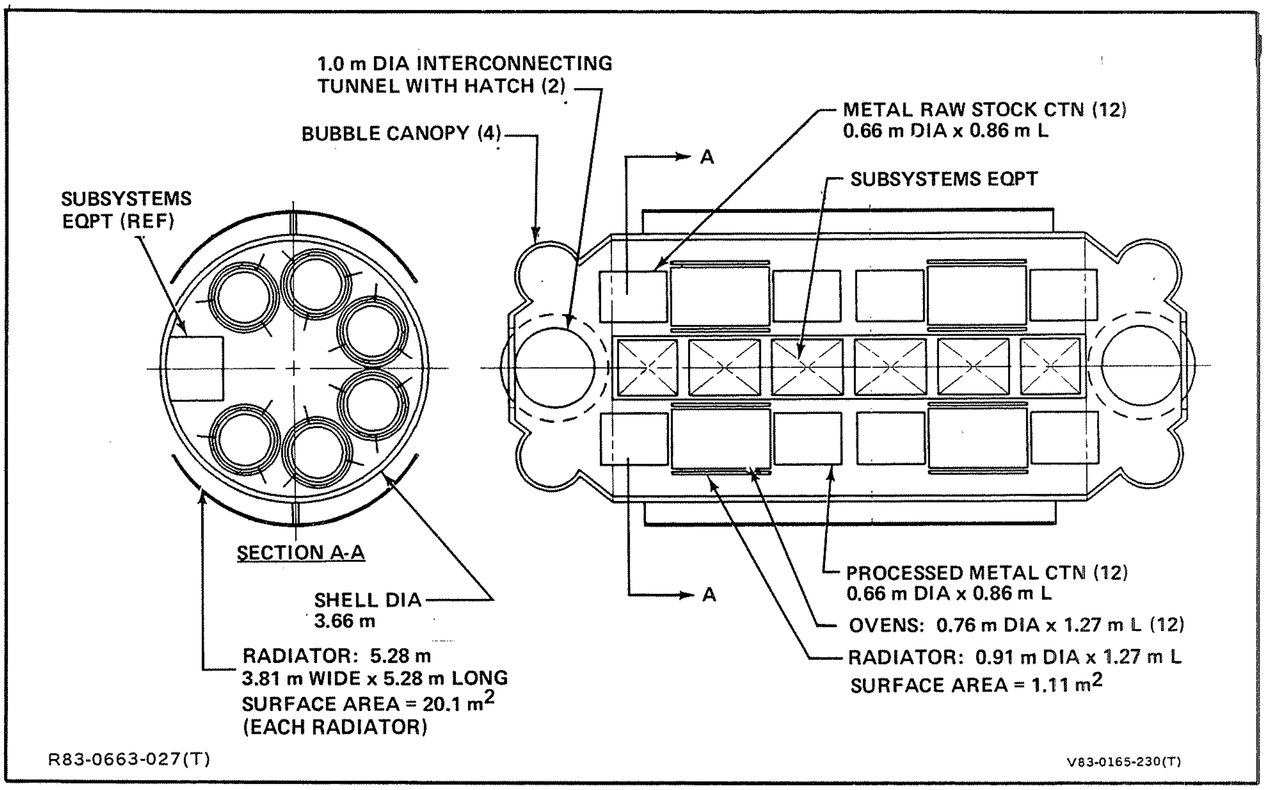


Fig. 2-15 Industrial Process Laboratory

earth. Each collection container is sized to pass through the 1.0-m dia tunnels that interconnect the modules.

The ovens shown are sized at 0.76-m dia x 1.27-m long. A radiator envelopes each oven, producing 1.11 m^2 of cooling surface area. Each oven is supported at the top and bottom by six shock mounts to minimize induced loads into the oven.

The metal processing is completely automatic, requiring no work on the astronaut's part. The only task he has is to load the raw stock bins and remove the processed metal.

An indicator panel is located in the module to monitor the performance of each oven. Any anomaly in oven operation will be displayed at the habitat control panel and the oven will shut down automatically.

2.3 TENDED INDUSTRIAL PLATFORM

Commercial materials processing facility can be incorporated in the 28.5 deg inclination Space Station complex or it can be a free flyer. The micro g requirements and very high power demands for materials production are difficult to satisfy as part of the main Space Station. Therefore, the materials processing facility is a school of free flyers that co-orbit with the main station.

Four free flyers are required for the program, and Fig. 2-16 shows a typical one. The standard pressurized module used for habitations and laboratories on the Space Station provides the pressurized shell and the appropriate subsystems. There may be additional subsystems elements and these will be installed in the module. The end cones of the pressure shell have such standard features as tunnels and viewing bubbles blanked off. As with the main station, a pallet mounts external subsystems like batteries, power processing and CMGs. The power source solar array has no gimballing requirements since the satellite will be flown inertially fixed relative to the sun, thus simplifying the array and minimizing undesirable accelerations.

Potentially, there are 40 units called for in the materials processing requirements section to be aboard in later years. Their allocation to the four free flyers will be on the basis of duty cycle. Therefore, the power requirement will vary from free flyer to free flyer and the solar array size will vary accordingly. With a total power requirement of around 110 kW continuous, the average is 28 kW per free

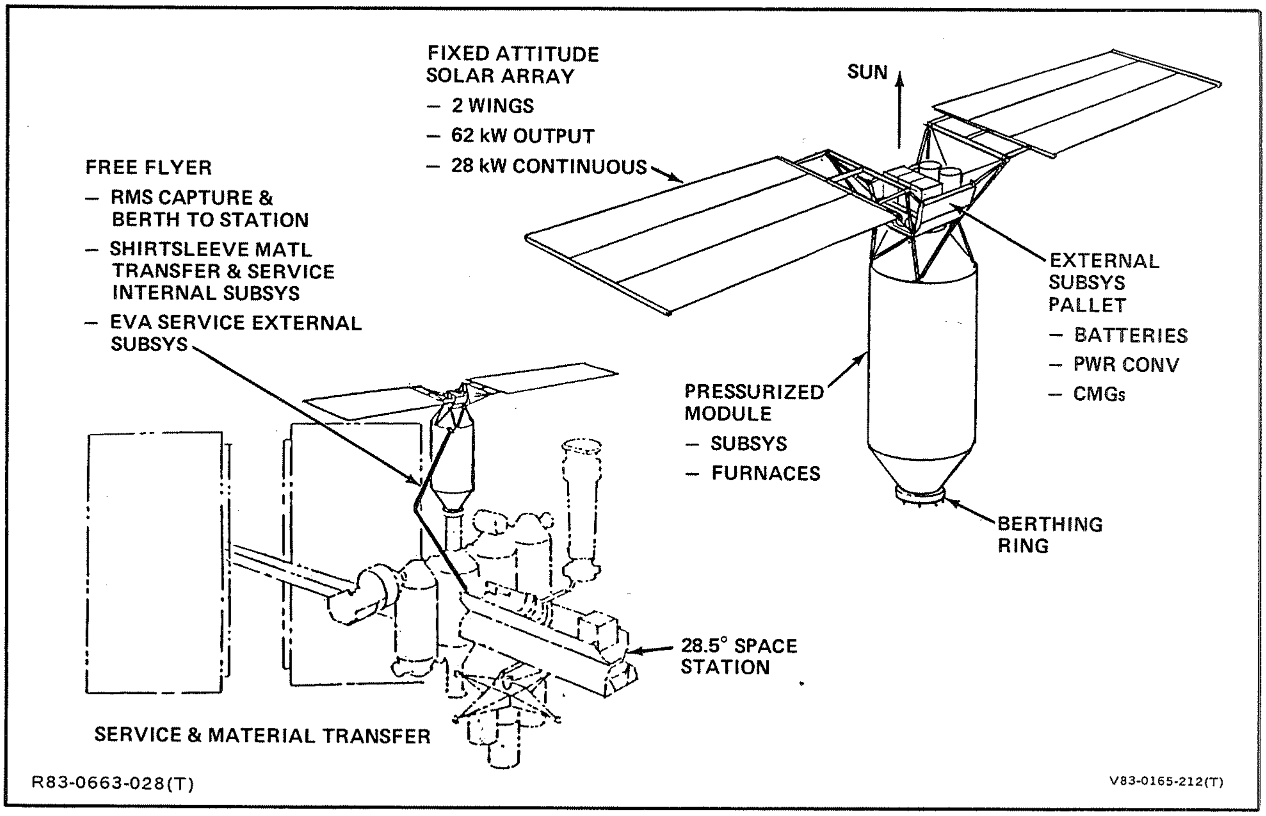


Fig. 2-16 Tended Industrial Platform

flyer and that is the size of the array shown. It uses the same SEPS-type modular panels as the main station and can vary in 7 kW increments.

The operational sequence of a free flyer is that after installation of materials to be processed, it boosts itself to a higher orbit using onboard propulsion. The new orbit is dictated by the duty cycle time of the furnaces, because the flyer is allowed to orbit decay in that time period so that at the end of the duty cycle, its location is suitable for rendezvous, capture and berthing to the Space Station berthing port. This is shown in the figure where berthing to the habitation module allows shirtsleeve exchange of materials and servicing of internal subsystems. External subsystems are serviced EVA.

2.4 TENDED POLAR PLATFORM

Requirements call for a total of three astrophysics missions, three solar observation missions and 12 terrestrial observation missions to be aboard a LEO facility in high inclination orbit by the year 2000. Some of the earth observation missions dictate a noon sun synchronous orbit, to provide light/dark contrasts.

System analysis, reported on earlier, shows that a premanently manned facility is unnecessary for this observatory. It is an unmanned platform, visited by the Shuttle at approximately six-month intervals to service the platform, change out observation instruments and to retrieve, service, then deploy two satellites.

Initially, the platform caters to earth viewing and is configured as shown in Fig. 2-17. A standard three-man habitation module, replicated from the 28.5 deg inclination Space Station, houses subsystems and can provide extended living volume for a visiting orbiter crew. When visiting, the orbiter berths to the module to enable shirt sleeve servicing of the subsystems.

Two standard surrogate bay structures, replicated from the 28.5 deg station, mount IPSs. These, in turn, mount the packages of earth observation instruments. Outboard of the surrogate structures, solar array panels are mounted on structures designed to support solar observation instruments at a later date. The array is sized to give 14.5 kW of continuous power and is comprised of panels replicated from the 28.5 deg station array. Each of the two wings requires a single axis orbit rate gimbal to track the sun. The gimbal mechanism utilizes the 28.5 deg station system.

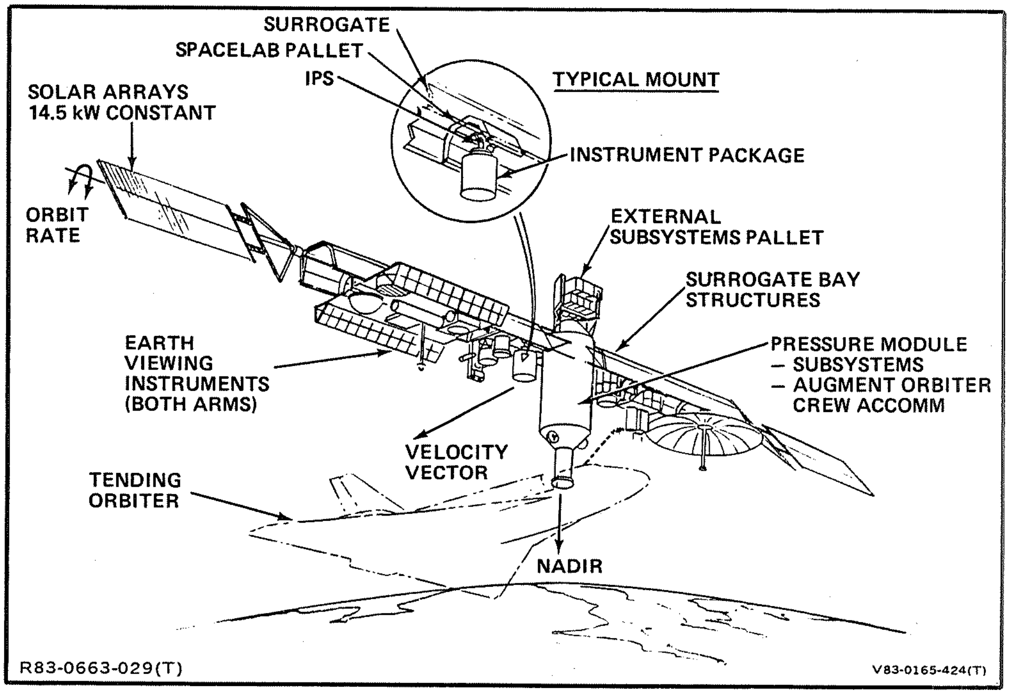


Fig. 2-17 Tended Polar Platform – 'Initial' Configuration

The evolved platform is shown in Fig. 2-18. To accommodate the full complement of earth observation missions, a surrogate bay structure is added orthogonally to the two existing structures. Another surrogate is added to mount satellite service equipment that includes a TMS to retrieve the satellite, an HPA on which to mount it, an MMU for EVA crewman maneuvering and an OCP to accommodate an EVA worker at the tip of the orbiter RMS. On the 28.5 deg Space Station, an RMS was part of the station standard satellite service equipment. Since this platform only performs the manned servicing function when tended by an orbiter, then the orbiter RMS can be used.

For a service mission, the orbiter berths to the platform, checks out the TMS, then sends it to retrieve the satellite. On its return, it is captured by the orbiter RMS, which berths the TMS/Satellite combination to the HPA arm end effector. The orbiter RMS then picks up the OCP at its tip, an EVA crewman boards the OCP, then the satellite is serviced by the EVA crewman from spares brought up in the orbit cargo bay. If necessary, the TMS can be refueled at this time by exchanging full tanks for empty. The MMU is also available for a second EVA crewman to assist in the operations.

After service and check out, the TMS returns the satellite to its orbit. It then separates and returns itself to the platform for safing, service and storage.

The IPS-mounted, solar viewing mission equipments are located on the solar array wings support structures. Their gross pointing is provided by the solar array gimbal.

A mast is added along the local vertical to mount a celestial observation instrument which requires a viewing field of 2π steradians, anti-earth. The mast extends from the external subsystem pallet.

Two other celestial observation packages are mounted, as shown, to the back, anti-earth face of the surrogate structures. Their viewing requirement is local zenith and, therefore, they are located outboard of the volume swept out by the movements of the mast-mounted celestial instrument. These two packages are each mounted on an IPS to provide fine pointing.

The solar array wings are extended by adding panels to provide a total continuous power of 29 kW.

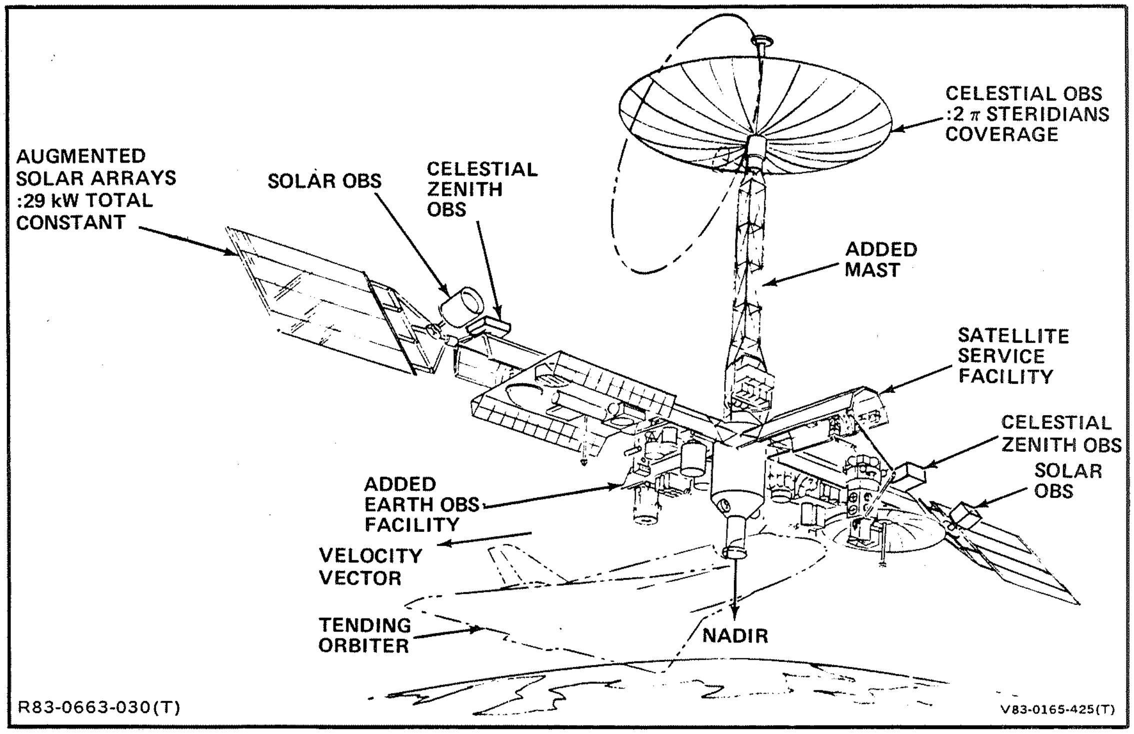


Fig. 2-18 Tended Polar Platform — 'Evolved' Configuration

Further growth of EVA facilities can be achieved by extending the two surrogate structures lying along the velocity vector.

Berthing points for the orbiter will be provided on the mast and surrogate structures. The orbiter can berth its HPA end effector to a suitably located point to enable its RMS to reach an instrument to change it out.

2.5 MASS SUMMARY

This subsection summarizes the masses that are the basis for costing in this study. It also provides the mass properties used in the attitude control investigation of the various configurations.

The massess are presented in station building block form (Fig. 2-19 through 2-24) and are summarized in Fig. 2-25 to show the total station dry masses. Expendables, given in Fig. 2-26, have been added, together with representative mission payloads, to provide total station mass characteristics (Fig. 2-27 through 2-31).

Masses have been obtained from as many verifiable sources as possible. Subsystem mass substantiations may be found in the appropriate section of this report. Structure masses are based on unit area masses and, where possible, have been verified by comparison to masses in previous studies.

The mass properties summary of the Initial Station (Fig. 2-27) gives a total mass of 47,000 kg. It includes an AXAF in operating position with a TMS attached, the SIRTF celestial instrument as a representative mass and three-man 90-day logistic supplies. The data are given for the station with and without the shuttle orbiter attached. This is also the case for the Evolved Station where a tended industrial platform is berthed and an OTV is in the surrogate for a total mass of 139,000 kg. Representative furnaces have been added to both laboratories and the SIRTF and the Starlab celestial instruments are operating in position. Similarly, the tended Platforms are shown with appropriate payloads given in Mission Requirements Volume II, Book I, Part I.

2.6 USER-FRIENDLY ATTRIBUTES

The attributes that make a system "user-friendly" are easy interfaces, simplicity, timely availability and cost effectivity. These features, above and beyond those of necessity, whet the appetite of a user and encourage a friendly atmos-

	3-MAN CORE MODULE kg	LABORATORY kg	TENDED MODULE kg
STRUCTURE	3650	3650	3650
TUNNELS	500	—	250
BERTHING	400	270	135
EPS (DISTRIB)	100	120	30
ECLS	1314	50	50
THERMAL	670	1500	1500
CONT & DISPLAYS	140	40	40
DATA MGMT	510	50	250
COMM	550	—	150
GN&C	120	—	120
CREW ACCOM	1050	120	—
TOTALS	9004	5800	6175

R83-0663-032(T) V83-0165-187(T)

Fig. 2-19 Pressure Vessels Subsystem Masses

	INITIAL STATION kg	EVOLVED STATION kg	INITIAL TENDED POLAR PLATFORM kg	EVOLVED TENDED POLAR PLATFORM kg
STRUCTURE	964	3856	3856	5784
EPS	70	200	200	300
THERMAL CONT	100	400	—	—
DATA MGMT	25	100	150	150
POV	156	156	—	—
RMS	393	786	—	—
HPA	798	1596	—	798
OCP	227	227	—	—
TELE END EFF	160	160	—	—
MMU	145	290	—	145
TOTALS	3038	7771	4206	7177

R83-0663-033(T) V83-0165-188(T)

Fig. 2-20 Surrogate Masses

	INITIAL STATION Kg	EVOLVED STATION Kg	INITIAL TENDED POLAR PLATFORM Kg	EVOLVED TENDED POLAR PLATFORM Kg	TENDED INDUST PLATFORM Kg
PALLET(S)	741	1482	741	741	741
TOWER(S)	750	750	500	1250	–
SOLAR ARRAYS	660	1980	445	910	660
S.A. GIMBALS/BOOMS	1000	1000	1000	1000	350
BATTERIES	660	1970	436	872	660
PWR CONDITIONING	220	650	145	290	220
PWR DISTRIB	500	500	145	145	100
CMG'S	636	636	636	636	636
RCS	340	340	340	340	340
TOTALS	5507	9052	4398	6184	3707

R83-0663-034(T) V83-0165-274(T)

Fig. 2-21 External Subsystems Masses

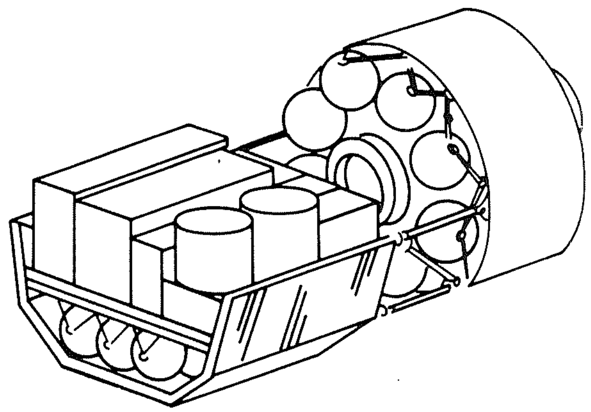
	INITIAL STATION Kg	EVOLVED STATION Kg	INITIAL TENDED POLAR PLATFORM Kg	EVOLVED TENDED POLAR PLATFORM Kg
IPS	1160	2320	4640	13920
SUPPORT STRUCT	200	200	1500	3100
TOWER	750	750	–	–
RMS	–	393	–	–
TOTALS	2110	3663	6140	17020

R83-0663-035(T) V83-0165-189(T)

Fig. 2-22 Observatory Masses

ITEM	WEIGHT, kg
PRESSURE VESSEL	820
PALLET	741
EQUIPMENT SUPTS	210
BERTHING MECH	68
EPS	50
ECLS	75
THERMAL	50
COMMUNICATIONS	15
DRY WEIGHT	2029

R83-0663-036(T)



V83-0165-190(T)

Fig. 2-23 Logistics Module Mass

	PROP UNIT Kg	TANK Kg
MAIN PROP TANKS	–	202
BODY SHELL	–	216
THRUST STRUCT/EQUIP SUPT	82	–
PAYLOAD INTERFACE	91	41
TANK INSULATION/PURGE	–	36
MAIN ENGINE/GIMBALS	122	–
FILL FEED VENT DRAIN PRESSURIZATION	91	45
ACS	86	–
EPS	204	–
AVIONICS	181	–
SUB TOTAL	857	540
CONTINGENCY	127	82
DRY MASS	984	622
RESIDUAL FLUIDS/GASES MPS		189
EPS, RCS	14	
END OF MISSION	998	811

R83-0663-037(T)

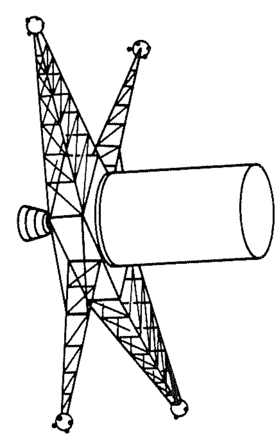


Fig. 2-24 Storable OTV Masses

	INITIAL STATION Kg	EVOLVED STATION Kg	INITIAL TENDED POLAR PLATFORM Kg	EVOLVED TENDED POLAR PLATFORM Kg	TENDED INDUST PLATFORM Kg
PRESSURE MODULES –					
• 3-MAN CORE	9004	27012	9004	9004	–
• LABORATORY (2)	–	11600	–	–	–
• TENDED	–	–	–	–	6175
AIRLOCK	900	900	–	–	–
SURROGATES	3038	7771	4206	7177	–
EXT SUBSYSTEMS	5507	9052	4398	6184	3917
OBSERVATORIES	2110	3663	6140	17020	–
LOGISTICS MODULE	2029	2029	–	–	–
TOTAL MASS	22,588	62,027	23,748	39,385	10,092

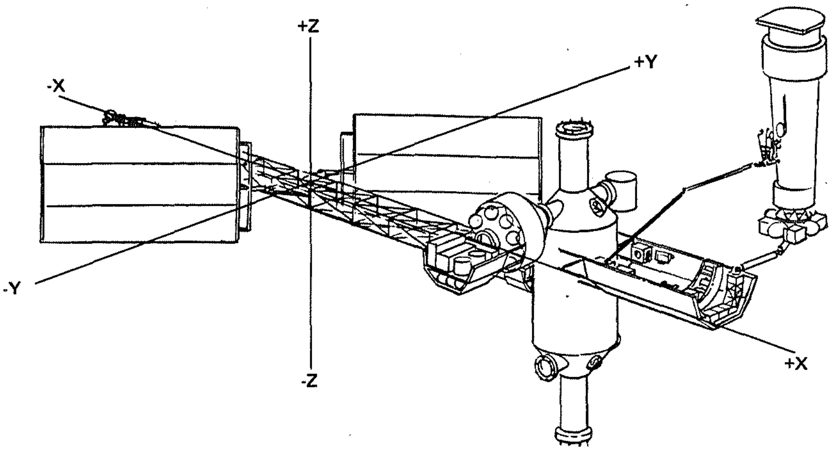
R83-0663-031(T) V83-0165-325(T)

Fig. 2-25 Summary Configuration Dry Masses

ITEM	3-MAN STATION kg	9-MAN STATION kg
FOOD	227	681
WATER	30	90
N_2	90	270
PERSONAL HYGIENE	113	340
CLOTHING	84	251
HOUSEKEEPING	27	81
SPARES	125	250
CONTROL PROPELLANT (COLD GAS)	280	710
EVA PROPELLANT	370	1100
EVA RESUPPLY	500	1500
TANKS & PACKAGING	184	527
TOTAL MASS	2030	5800

R83-0663-038(T)

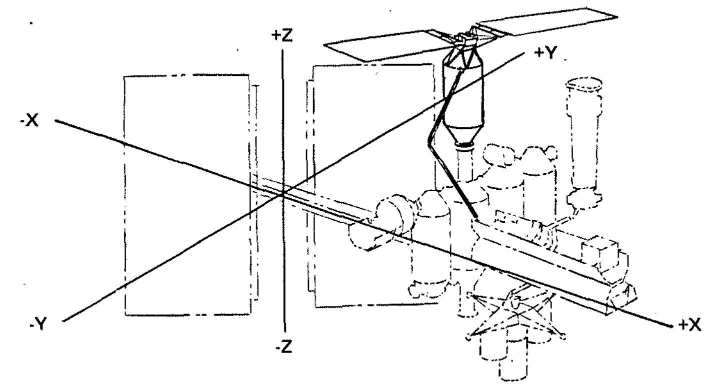
Fig. 2-26 Initial Space Station Logistics Masses for 90 Day Period (Exluding Mission Items)



ITEM	MASS Kg	X-ARM M	Y-ARM M	Z-ARM M	I(XX) KM²	I(YY) KgM²	I(ZZ) KgM²	I(XY) KgM²	I(XZ) KgM²	I(YZ) KgM²
DRY WT	22588.0	17.120	-0.444	-0.295	270704.	4415958.	4506227.	-58365.	-62436.	12902.
AXAF	11000.0	32.000	12.000	8.000	181461.	181461.	18824.	0.	0.	0.
TMS	3550.0	32.000	12.000	3.000	2367.	2367.	2367.	0.	0.	0.
LOG P/L	2030.0	23.000	-6.000	1.300	8120.	4060.	8120.	0.	0.	0.
CEL INST	8100.0	-23.000	0.0	0.0	5400.	5400.	5400.	0.	0.	0.
TOTAL	47268.0	15.068	3.242	2.013	2696953.	21291424.	22350688.	2781056.	1615351.	876758.
SHUTTLE	89108.0	32.000	0.0	-10.000	1170000.	9100000.	9500000.	0.	400000.	0.
TOTAL	136376.0	26.150	1.120	-5.850	8633396.	43660240.	41000336.	1090970.	-4246622.	2075857.

R83-0663-039(T)

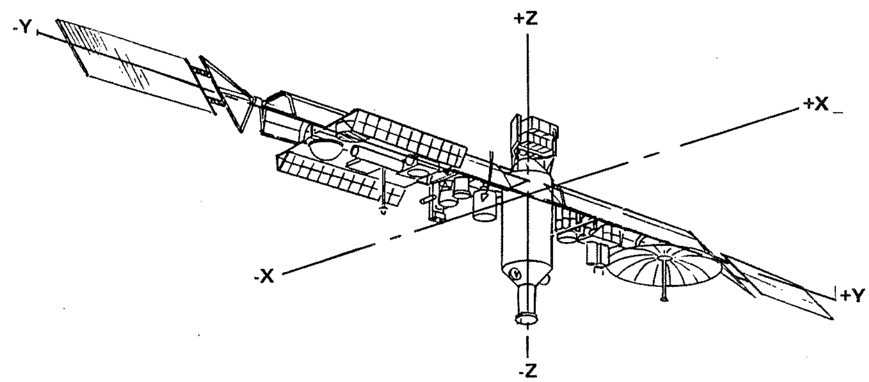
Fig. 2-27 Mass Properties Summary Initial Station at 28.5°



ITEM	MASS Kg	X-ARM M	Y-ARM M	Z-ARM M	I(XX) KgM²	I(YY) KgM²	I(ZZ) KgM²	I(XY) KgM²	I(XZ) KgM²	I(YZ) KgM²
DRY MASS	62027.0	20.170	-0.940	0.030	2002782.	10413305.	11012385.	-165100.	-106400.	130000.
TMS	3550.0	29.000	0.0	1.300	2367.	2367.	2367.	0.	0.	0.
LOG P/L	5800.0	23.000	-6.000	1.300	23200.	11600.	23200.	0.	0.	0.
CEL INST	11380.0	-23.000	0.0	0.0	7587.	7587.	7587.	0.	0.	0.
OTV	29000.0	34.000	0.0	1.300	66168.	169022.	169022.	0.	0.	0.
FURNACES	5000.0	23.000	7.000	0.0	38750.	38750.	10000.	0.	0.	0.
FURNACES	5000.0	23.000	7.000	0.0	38750.	38750.	10000.	0.	0.	0.
IND FF	16627.0	23.000	0.0	10.000	265161.	241904.	318684.	0.	0.	0.
TOTAL	138384.0	20.408	-0.167	1.575	4578704.	39591264.	39586768.	-59989.	895783.	119407.
SHUTTLE	89108.0	32.000	0.0	-10.000	1170000.	9100000.	9500000.	0.	400000.	0.
TOTAL	227492.0	24.948	-0.102	-2.959	13012868.	63237856.	56372240.	44924.	-5977509.	14648.

R83-0663-040(T)

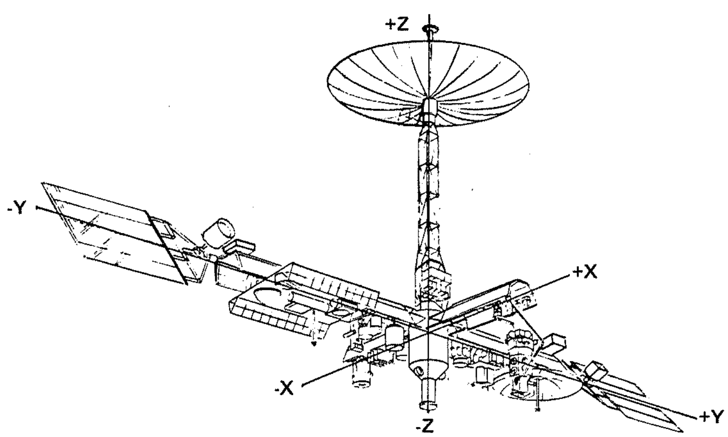
Fig. 2-28 Mass Properties Summary Evolved Station at 28.5°



ITEM	MASS Kg	X-ARM M	Y-ARM M	Z-ARM M	I(XX) KgM^2	I(YY) KgM^2	I(ZZ) KgM^2	I(XY) KgM^2	I(XZ) KgM^2	I(YZ) KgM^2
CORE MOD	9004.0	0.0	0.0	-1.800	69781.	69781.	18008.	0.	0.	0.
SURROGAT	4206.0	0.0	0.0	0.0	653683.	11216.	653683.	0.	0.	0.
SA LH	227.0	0.0	41.500	0.0	6148.	946.	7056.	0.	0.	0.
SA RH	227.0	0.0	-41.500	0.0	6148.	946.	7056.	0.	0.	0.
GIM/SUPT	500.0	0.0	24.400	0.0	333.	333.	333.	0.	0.	0.
GIM/SUPT	500.0	0.0	-24.400	0.0	333.	333.	333.	0.	0.	0.
EXT SUBS	2944.0	0.0	0.0	4.000	10158.	7996.	5888.	0.	0.	0.
OBSERV	6140.0	0.0	0.0	0.0	954258.	16373.	954258.	0.	0.	0.
INSTS	14000.0	0.0	0.0	-3.000	2175833.	37333.	2175833.	0.	0.	0.
TOTAL	37748.0	0.0	0.0	-1.230	5399102.	290423.	5199709.	0.	0.	0.

R83-0663-041(T)

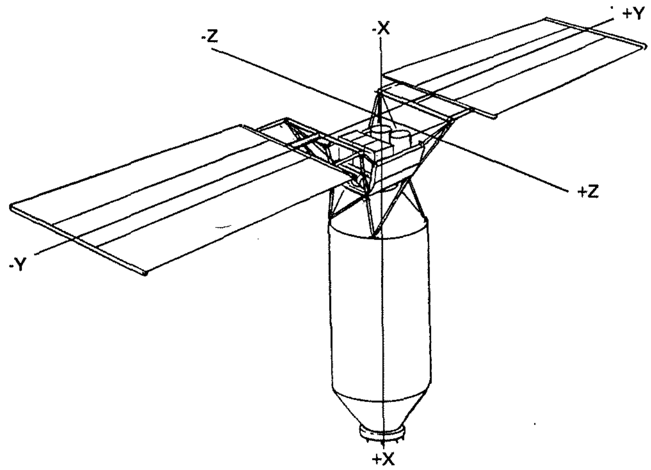
Fig. 2-29 Mass Properties Summary Initial Tended Polar Platform



ITEM	MASS Kg	X-ARM M	Y-ARM M	Z-ARM M	I(XX) KgM^2	I(YY) KgM^2	I(ZZ) KgM^2	I(XY) KgM^2	I(XZ) KgM^2	I(YZ) KgM^2
CORE MOD	9004.0	0.0	0.0	-1.800	69781.	69781.	18008.	0.	0.	0.
SURROGAT	1014.0	-7.600	0.0	0.0	2704.	8197.	8197.	0.	0.	0.
SURROGAT	1014.0	7.600	0.0	0.0	2704.	8197.	8197.	0.	0.	0.
SURROGAT	4206.0	0.0	0.0	0.0	653683.	11216.	653683.	0.	0.	0.
SA LH	455.0	0.0	41.500	0.0	12323.	7470.	19717.	0.	0.	0.
SA RH	455.0	0.0	-41.500	0.0	12323.	7470.	19717.	0.	0.	0.
GIM/SUPT	500.0	0.0	24.400	0.0	333.	333.	333.	0.	0.	0.
GIM/SUPT	500.0	0.0	-24.400	0.0	333.	333.	333.	0.	0.	0.
TOWER	750.0	0.0	0.0	16.000	21250.	21250.	2000.	0.	0.	0.
HPA/MMU	943.0	7.000	0.0	0.0	2515.	7623.	7623.	0.	0.	0.
EXT SUBS	3827.0	0.0	0.0	4.000	13205.	10395.	7654.	0.	0.	0.
OBSERV	17020.0	-1.370	1.600	2.700	4870000.	982536.	4220000.	36846.	63391.	-73025.
INSTS	64712.0	0.0	-0.850	2.000	21400000.	1230000.	20340000.	0.	0.	-600000.
TOTAL	104400.0	-0.160	-0.266	1.786	29638928.	2889652.	27743616.	-4909.	30293.	-659901.

R83-0663-042(T)

Fig. 2-30 Mass Properties Summary Evolved Tended Polar Platform



ITEM	MASS Kg	X-ARM M	Y-ARM M	Z-ARM M	I(XX) KgM^2	I(YY) KgM^2	I(ZZ) KgM^2	I(XY) KgM^2	I(XZ) KgM^2	I(YZ) KgM^2
SA LH	330.0	0.0	14.000	0.0	9237.	4463.	13645.	0.	0.	0.
SA RH	330.0	0.0	-14.000	0.0	9237.	4463.	13645.	0.	0.	0.
SUPPTS	350.0	0.0	0.0	0.0	5046.	2479.	7292.	0.	0.	0.
SUBSYST	2907.0	2.000	0.0	0.0	12113.	8237.	8237.	0.	0.	0.
HABITAT	6175.0	8.000	0.0	0.0	47856.	47856.	12350.	0.	0.	0.
FURNACES	7000.0	8.000	0.0	0.0	54250.	54250.	14000.	0.	0.	0.
TOTAL	17092.0	6.507	0.0	0.0	267099.	252930.	329710.	0.	0.	0.

R83-0663-043(T)

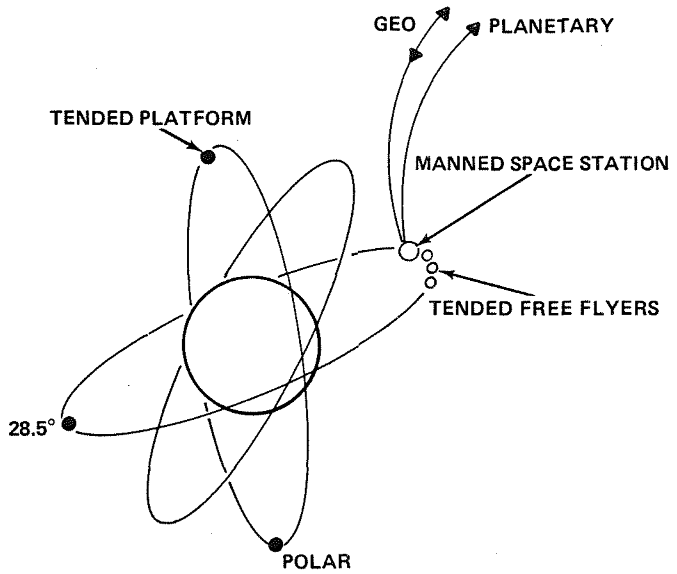
Fig. 2-31 Mass Properties Summary Tended Industrial Platform

phere. In terms of a potential Space Station user, these attributes are reflected in standardized interfaces with many options that are simple and have broad limitations. Ample physical and utility resources preclude concerns of availability. Sharing with other users the resources required to sustain a manned Space Station provides cost effectivity. Design efforts, subsequent to this study, will be required to detail user-friendly interfaces.

Illustrated in Fig. 2-32 are those user-friendly attributes of a general nature. This figure and the subsequent figures identify the attributes in terms of interfaces, resources and facilities for various missions. In addition to standardized utilities, a general feature of the architecture, described in the preceding subsections, is that of a common mounting and utilities interface between the orbiter cargo bay and the Space Station surrogate. General resources available in the Space Station include extensive equipments. Facilities to support the functional capabilities round out those user-friendly attributes that are of a general nature.

For commercial missions (see Fig. 2-33), the additional attributes of the Space Station, in terms of interfaces and resources, are power and thermal which are primarily for materials processing. Facilities supporting the commercial missions,

- INTERFACES
  - COMMON MOUNTINGS & UTILITIES
  - ORBITER CARGO BAY & SPACE STATION SURROGATE
  - STANDARDIZED POWER (28 VDC), THERMAL (COLD PLATE), DATA (DATA BUS) & MAN/MACHINE INTERFACES
- RESOURCES
  - AMPLE VOLUME, POWER, THERMAL, DATA, CHECK-OUT, FUEL, TOOLS, PERSONNEL
  - EXTENSIVE SERVICES/ASS'Y-RMS, HPA, WRU, OCP, POV, TMS, TELEOPERATORS
- FACILITIES
  - TWO 18.3 M SURROGATE BAYS
    - ○ TRANSPORT HARBOR, TEST RANGE & SAT SERVICES (28.5°)
  - FOUR 18.3 M SURROGATE BAYS
    - ○ TERRESTRIAL OBSERV (POLAR)
  - TWO (+) 5.5 x 13.6 METER LABS
  - FOUR (+) FREE FLYERS
  - OBSERVATORIES - 28.5° & POLAR



0663-044(T)

Fig. 2-32 User Friendly Attributes — General

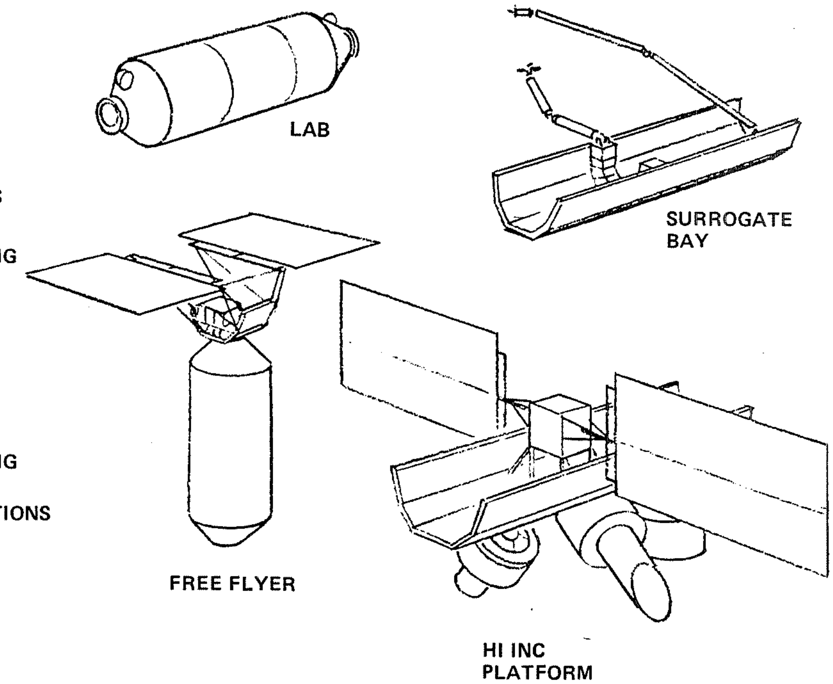
such as laboratories for R&D, surrogates for transportation and large structure developments, free flyers/platforms for material processing and observatories, are also shown in that figure.

The most significant additional interface and resource required to make the Space Station user-friendly for the scientific and application missions is the capability of handling and compacting data for transmission to the users on the ground. Also shown in Fig. 2-34 are the multiple facilities available to support the diverse requirements.

For technology development missions, attributes of a Space Station (shown in Fig. 2-35) include the availability of all on-board systems for technology enhancement, the availability of surrogates for EVA capability development, laboratories for experimentation and free flyers for the space testing of such devices as low-level thrusters and tether dynamics.

Of special note in Fig. 2-36 are the facilities that can be provided by the international community, either on a leased or barter basis or as adjuncts to a Space Station program for international missions.

- INTERFACES & RESOURCES
  - – POWER, THERMAL
  - – PROPRIETARY DATA
- FACILITIES
  - – R&D LABS (28.5°)
    - o COMPONENTS/SYSTEMS
    - o BIOLOGICAL LAB
    - o MATERIALS PROCESSING
    - o OBSERVATORIES
  - – SURROGATE BAY (28.5°)
    - o TRANSPORT
    - o LARGE ANTENNA
    - o MATERIALS R&D
  - – FREE FLYERS (28.5°)
    - o MATERIALS PROCESSING PRODUCTION
  - – TERRESTRIAL OBSERVATIONS
    - o HIGH INCLINATION PLATFORM.

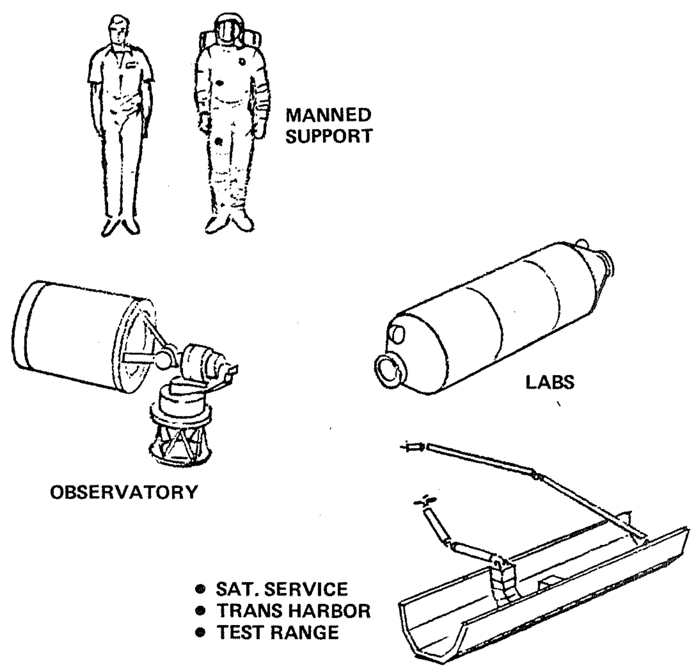


R83-0663-045(T)

V83-0165-113(T)

Fig. 2-33 User Friendly Attributes — Commercial

- INTERFACES
  - – POWER, THERMAL, HIGH DATA RATES, ATTITUDE CONTROL, EPHEMERIS
- RESOURCES
  - – SERVICES/ASS'Y
  - – DATA COMPACTION/ TRANSMISSION
  - – R&D PERSONNEL SUPPORT
- FACILITIES
  - – CELESTIAL OBSERV
    - o TWO IPS AT 28.5°
    - o THREE IPS AT POLAR
  - – TERRESTRIAL OBSERV
    - o SEVEN IPS AT POLAR
  - – SOLAR OBSERV
    - o TWO IPS AT POLAR
  - – INTERNAL LABS
    - o BIOLOGICAL/BIOMEDICAL
    - o BASIC SCIENCE
    - o MATERIALS SCIENCE
  - – SAT SERVICE, TRANSPORT HARBOR & TEST RANGE.

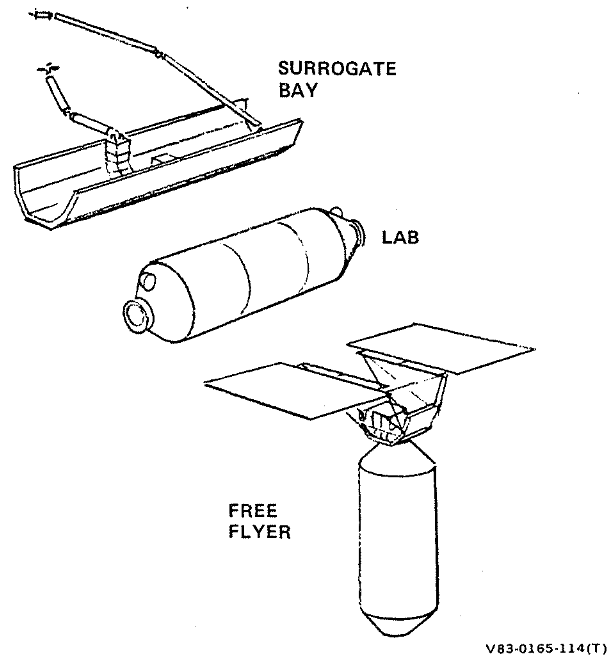


R83-0663-046(T)

V83-0165-112(T)

Fig. 2-34 User Friendly Attributes Science & Applications

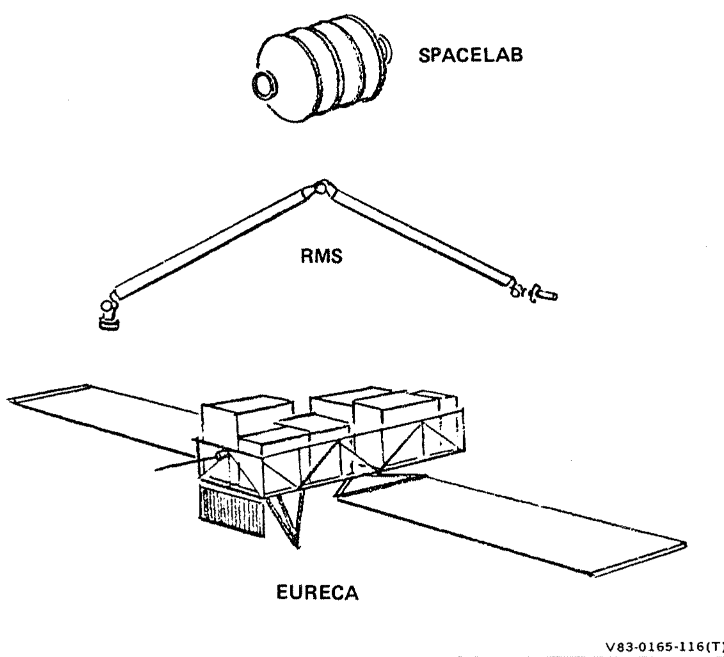
- MATERIALS R&D
  - MICRO-G ACCELERATION
- MATERIALS PROCESSING (PRODUCTION)
  - MICRO-G ACCELERATION
  - HIGH POWER



0663-047(T)

Fig. 2-35 Industrial Park Requirements

- INTERFACES
  - POWER, THERMAL, DATA/COMM, EC/LSS
- RESOURCES
  - BERTHING & UTILITIES FOR LABS
  - SERVICES FOR FREE FLYERS
  - TRANSPORT HARBOR
- CANDIDATE FACILITIES FOR INTERNATIONAL PARTICIPATION
  - LABS
  - FREE FLYERS
  - SPECIALIZED MISSION EQUIPMENT.



0663-048(T)

Fig. 2-36 User Friendly Attributes International

The user-friendly attributes associated with national security missions are shown in Fig. 2-37. Provisions for secure and encrypted/decrypted data and for isolating/securing a command post and work area have been provided. Within the core module of the Space Station, two command posts (one on each end) are available. Either can be secured with hatch closures. Laboratories can be similarly isolated. A shield can be erected to prevent IVA personnel from seeing restricted external operations.

As the design effort progresses, these attributes should be added to and defined in more detail.

2.7 ALTERNATE CONCEPTS

A variety of configuration alternates were studied to arrive at a baseline. Evaluation criteria included costs, commonality of elements and mission performance capability.

2.7.1 Configurations

The alternate concepts investigated have concentrated on the 28.5 deg inclination Space Station; addressing such issues as whether or not facilities like the transport harbor and/or industrial park and/or observatories should be incorporated on the Space Station or co-orbit as free flyers. At a lower level, we looked at the options for arranging the elements of the Space Station.

The incorporation of all required facilities on the one station complex has an advantage in shared subsystems and crew skills. However, there are basic requirements for some facilities (such as micro-g for space manufacturing and high power levels) that are particularly restrictive on the configuration of such a station. The configuration shown in Fig. 2-38 is a typical concept for a versatile multimission station. It utilizes an STS external tank as a structural spine and puts the longitudinal axis along the local vertical, thus using gravity gradient effect.

A disadvantage of this configuration is the high drag, resulting from the side-on tank body and from the large solar arrays necessary for a continuous power output of approximately 170 kW. Furthermore, this concept carries high cost for modifying the tank to add external rails for mounting facilities and to provide an aft cargo compartment. The high cost to modify the external tank and extensive subsystem requirements did not justify the use of this configuration.

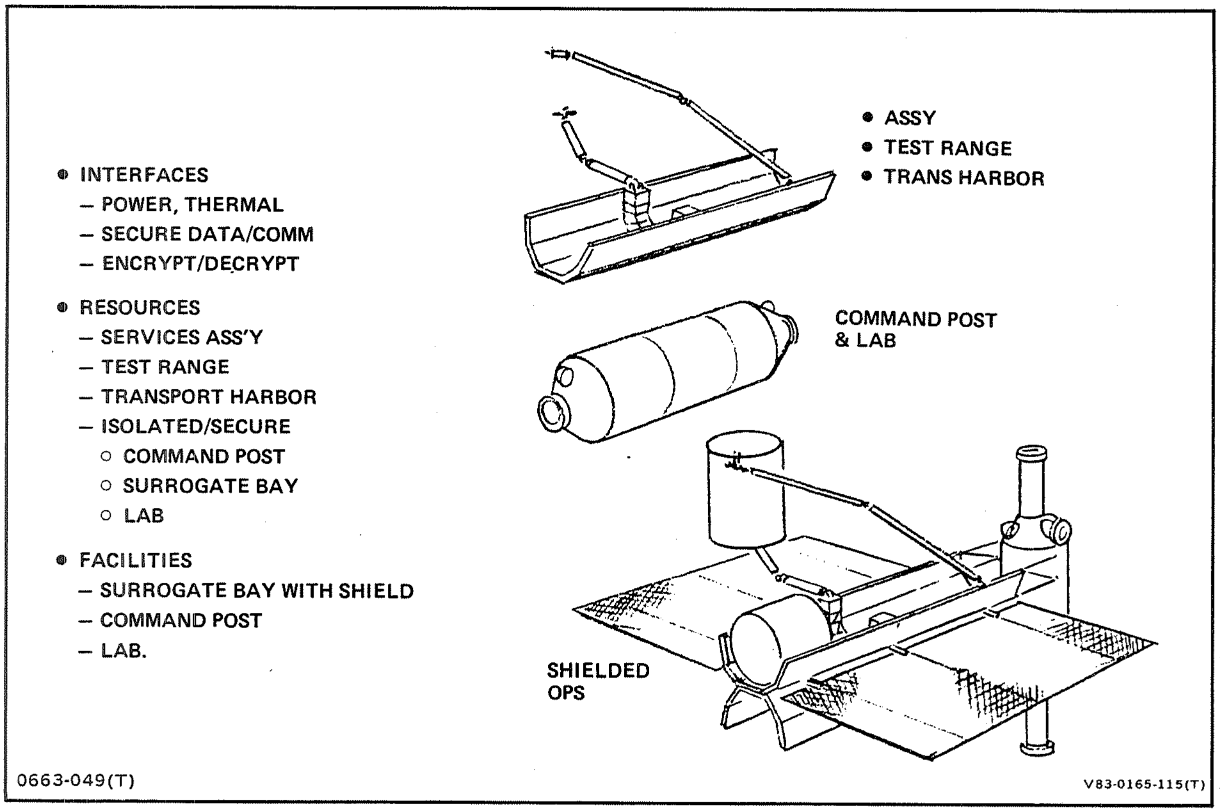


Fig. 2-37 User Friendly Attributes National Security

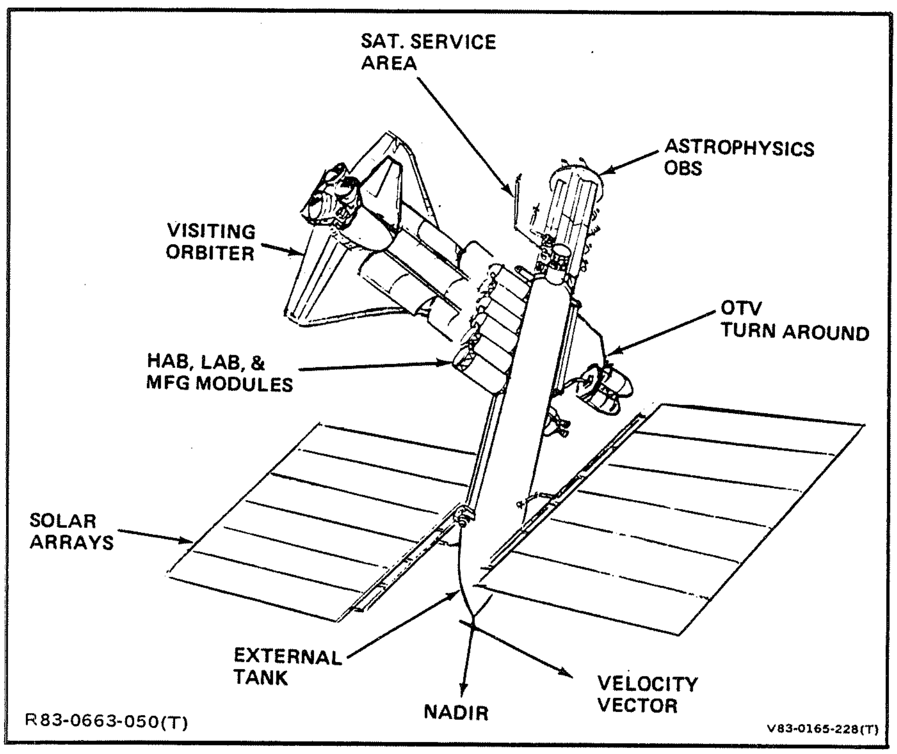


Fig. 2-38 STS External Tank Option

As an alternative, requirements can be met by an assembly of pressure modules to provide internal working volume, structures for EVA external work areas, observatories and a power source.

When configuring a Space Station which includes several pressurized modules, there are two basic ways of arranging the modules. One way is to cluster them; the other is to put them in line. Figure 2-39 shows an inline arrangement. The external EVA areas radiate from a control module and the added growth labs, habitations, etc are strung out in line. This arrangement makes use of the gravity gradient effect if the modules are strung out along the local vertical, but it provides no redundant escape paths from module to module, unless paralleling tunnels are provided. This concept requires a relatively long journey to go from end to end, and there is the inconvenience of inter-module traffic passing through the main activity area of each module: Furthermore, growth is restrictive with the inline concept if a power source is located at one end of the string, as shown in Fig. 2-39. Now there is only one place to add an inline module. Unless the mission growth requirements and the desired growth pattern coincide, it is unlikely that "like function" modules will be together, a desirable feature. If the power source were located elsewhere, a module could be added to each end, but this is still a restrictive pattern. Stiffness of the assembly is another issue, particularly if the "inline" string gets too long. This is a design restriction which can only be satisfied by additional structure and its resultant mass penalty. For all these reasons, the "inline" arrangement concepts were not pursued.

Station Configurations that include the materials processing production facility end up with a total demand of about 170 kW for the evolved station. If large solar arrays, such as those shown in Fig. 2-39 are to be avoided, alternate power sources must be developed. A nuclear power source is shown in Fig. 2-40 as an extension of the concept shown in Fig. 2-39. Here the solar array, used until the nuclear source is introduced, becomes the emergency power source and an extension is added to the existing tower to mount the nuclear power source.

With the elimination of in line concepts, cluster concepts were investigated. To ensure provision of the micro-g needs of space manufacturing, that facility is provided by a series of free flyers. Thus, the need for a single large power source, such as nuclear, disappears and can be met by individual solar arrays. The baseline station configurations evolved to be a cluster of modules that provide

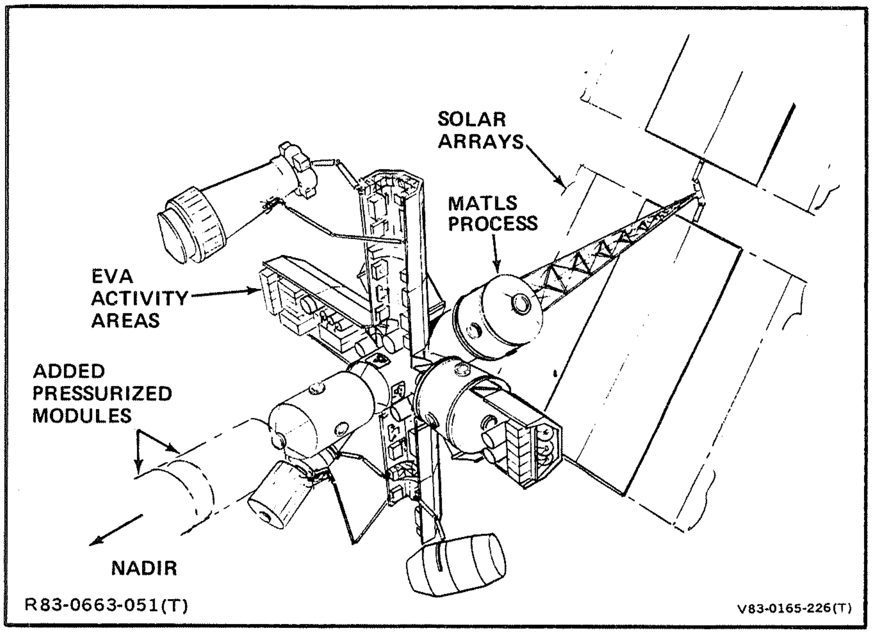


Fig. 2-39 In-Line Module Growth Option

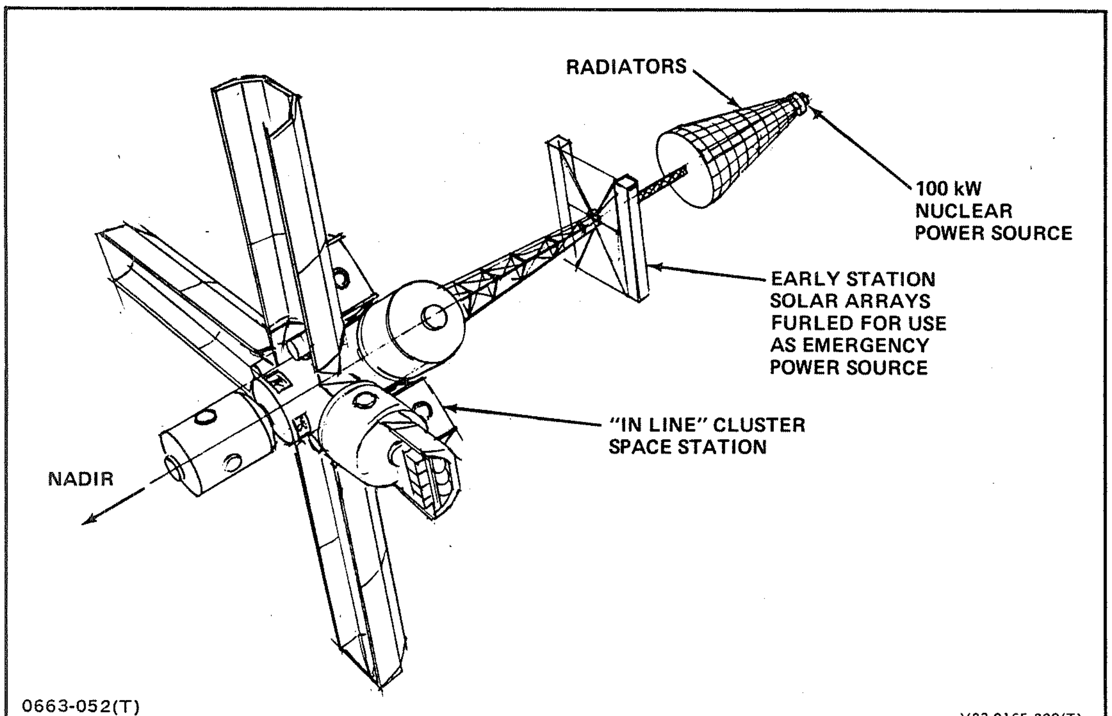


Fig. 2-40 'In Line' Growth Concept, Nuclear Power Source

habitation, internal work areas and laboratories to which are attached EVA work areas. Power generation is provided by solar arrays.

A clustered module Space Station is typified by the arrangement shown in Fig. 2-41. This is a three-man Space Station with a structure for EVA activities supporting the power source. Growth, as shown in Fig. 2-42, is provided by adding a "handed' version of the original three-man assembly to make a nine-man station. However, this arrangement does not entirely satisfy celestial viewing requirements. In addition, the placing of EVA external structures at each end of the pressure module complex restricts the adding of modules in functionally similar groupings.

The baseline configuration for a 28.5 deg inclination Space Station evolved to be the clustered concept which was described in the preceding subsections.

2.7.2 Commonality

The issue of commonality plays an important role in the selection of a preferred Space Station system from a series of alternates. In particular the impact on costs for using common elements as replicated items is a high level discriminator. Commonality is an issue that affects the entire Space Station system. In our baseline, the system comprises a manned station plus materials processing free flyers in 28.5 deg inclination and an observation platform in polar orbit. Our design philosophy is to optimize the basic 28.5 deg station design and to use its components as building blocks for the free flyers and high inclination platform. Additionally, we look for commonality in the components that make up the basic station. Figure 2-43 shows the four basic building blocks and their replication in the Space Station system facilities.

A pressurized core module provides the pressure shell used on the manned station for habitation and laboratory modules. Subsystems within those modules have common features, particularly in the life support area, and are replicated to a great extent.

The tended industrial platform free flyers use the same pressure shell and many of the same subsystem elements including control, communications, data handling and life support. The module is, therefore, a version of the basic station core module. To house furnaces, it is stripped internally of everything except necessary subsystems elements, the majority of which are replications of the baseline.

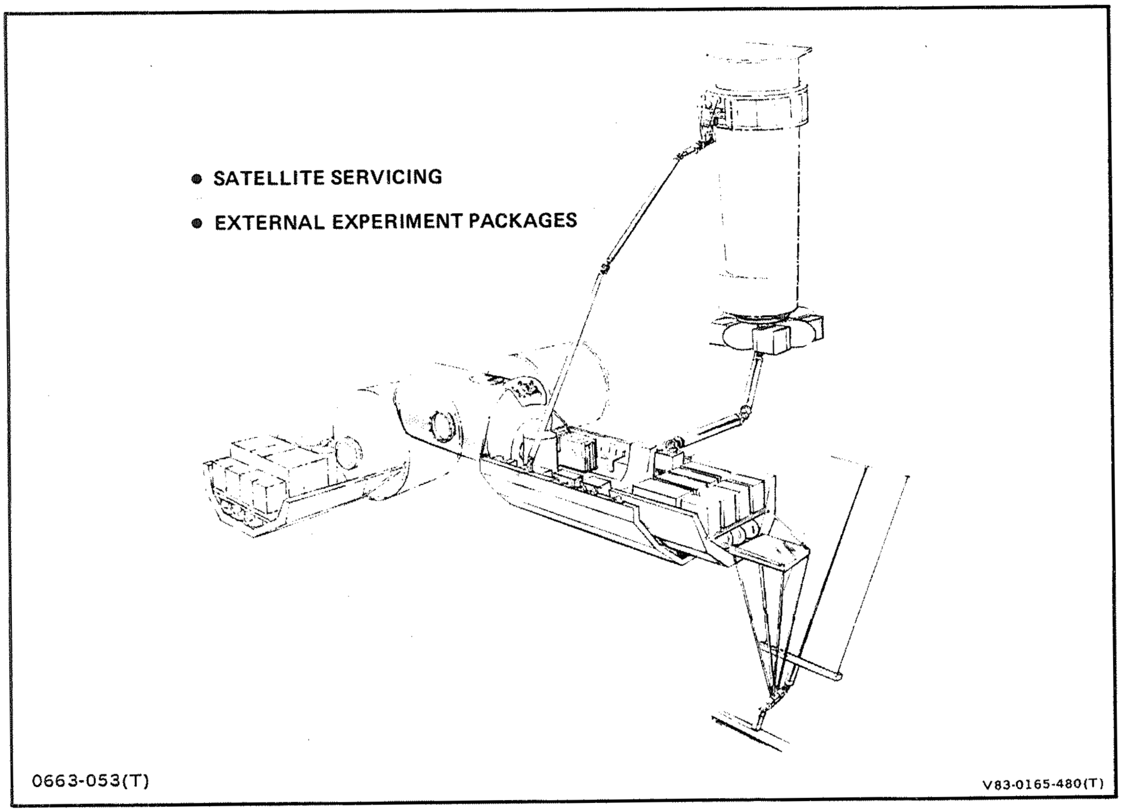


Fig. 2-41 Clustered Three-Man Space Station Option

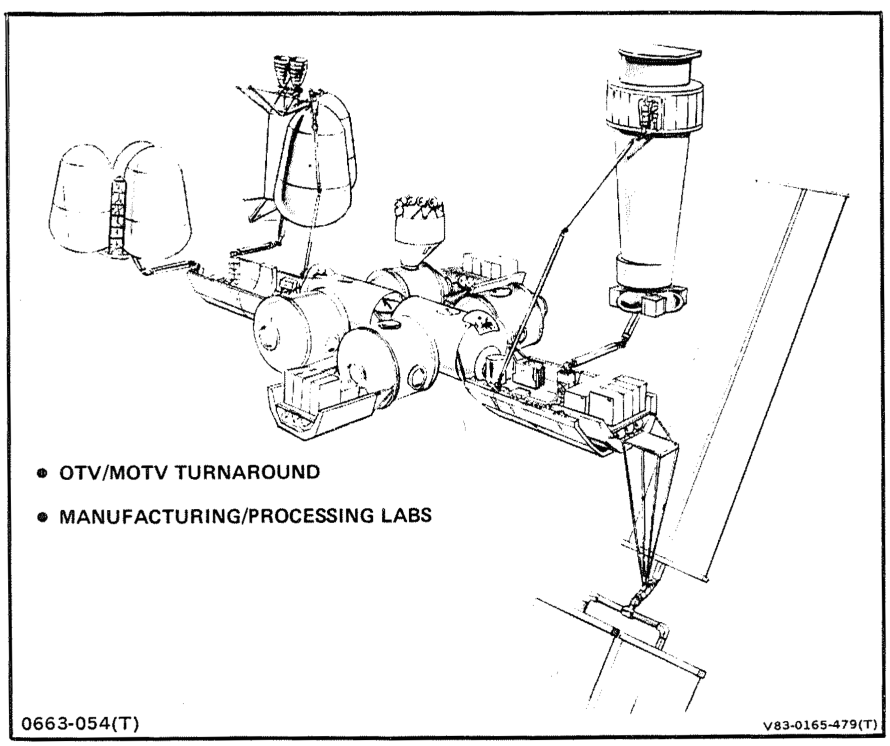
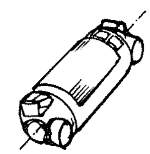
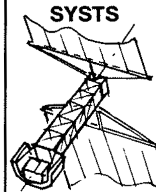
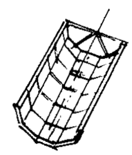
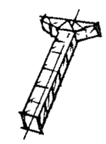


Fig. 2-42 Clustered Nine-Man Space Station Option

INCLIN		IOC	PRESS. VESSEL CORE 	EXT SUB SYSTS 	SURROGATE BAY 	OBSERV TOWER 
28.5°	INITIAL STATION	'90	3 CREW HAB	22 kW		
	GROWTH	'91 '93	3 CREW HAB LAB	PLUS 44 kW		
28.5°	TENDED INDUST PLATFORM	'92 ON	M.P.S. X4	28 kW X4		
97°	TENDED POLAR PLATFORM	'94	TENDED HAB	14.5 kW		
	GROWTH	'96		PLUS 14.5 kW		
			10	6	10	2

REPLICATION TO HOLD DOWN COSTS

0663-055(T)

V83-0165-227(T)

Fig. 2-43 Space Station System — Basic Building Blocks

Similarly, the unmanned high inclination platform requires a module to house subsystems. It is a replication of the core module that is used for 3-man habitation on the 28.5° station.

The power source, support mast and external subsystems pallet combination are used on all the system facilities. The size of the array differs as indicated in the figure, but each array is assembled from identical panels. The mast length differs with each facility, but each is a multiple of standard sections. The subsystems on their common pallet mount are multiples of the same battery, the same cmg, etc.

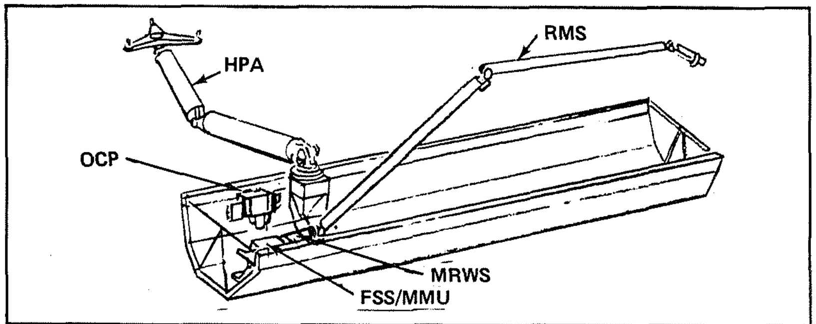
A surrogate bay is the structure which mounts EVA equipments. In itself, it is a replicated structure for each application. It is used on the manned station and on the high inclination platform. The equipments mounted in the structure are common to a great extent. Figure 2-44 shows these common equipments.

A tower to support observation instruments is used on the manned station and the solar platform. For most of its length, it uses the same standard sections as the solar array support mast.

The functions performed in the EVA activities surrogate structure are R&D, satellite service and assembly, and OTV turnaround on the 28.5 deg station and observations on the Tended Polar Platform. Typical subsystems and equipments which have common usage for these functions are described in Fig. 2-44.

2.7.3 Tethers

Figure 2-45 lists the candidate functions for the use of a tether on a Space Station, identifies its potential applications and comments on the pros and cons of using a tether. Although there has been sufficient studies to date to indicate the feasibility of their use, more development is necessary to establish their role before baselining them for operations.



SUBSYS & EQPT / OPERATIONS FUNCTIONS	28.5° SPACE STATION			TENDED POLAR PLATFORM
	R&D	SERVICE & ASSY	OTV SERVICE	
LIGHT, POWER, COOLING, DATA, VIDEO	√	√	√	√
HANDLING EQPT: RMS, HPA, MMU, WRU, OCP	√	√	√	RMS
TOOLS & PARTS STOWAGE	√	√	√	√
PROP. HANDLING & UPPER STAGE SERVICE		TMS & POV	√	

0663-056(T) V83-0165-223(T)

Fig. 2-44 Surrogate Bay Subsystem Equipment – Common Use for Operational Functions

FUNCTION	POTENTIAL APPLICATION	COMMENTS
TRANSFER MOMENTUM	• ORBITER/EXTERNAL TANK/PAYLOADS/ SPACE STATION-ALTITUDE CHANGE	• RDT&E REQUIRED FOR TETHER & RELEASE TETHER DYNAMICS • LIMITED APPLICABILITY TO INITIAL SPACE STATION
STABILIZE ATTITUDE	• CONTROL DRAG • ENHANCED GRAVITY GRADIENT STABILIZATION • VARIABLE "G" BETWEEN TETHERED VEHICLES	• RISK & COMPLEXITY REQUIRE MORE ANALYSES • PREMATURE FOR BASE LINING
RETRIEVE/BERTH	NON-COOPERATIVE SATELLITES/DEBRIS	• POV/TMS REQUIRED TO DOCK TETHER TO NON-COOPERATIVE ELEMENTS • SAVES RETURN FUEL • RISK, COMPLEXITY & DEV COST OUTWEIGH SAVINGS FOR LOW VOLUME LOCAL TRAFFIC
DEPLOY/RETRIEVE/BERTH	COOPERATIVE SATELLITES/PLATFORMS/ ORBITER	• NON-CONTAMINATING COMPETES WITH COLD GAS SYSTEM • SAVES FUEL & SEPARATE COMM/ POWER • RISK, COMPLEXITY & DEV COST OUTWEIGH SAVINGS FOR LOW VOLUME LOCAL TRAFFIC
ELECTRODYNAMICS MORE RDT&E REQUIRED BEFORE BASELINING TETHER OPERATIONS	• ELECTRICAL POWER (GENERATOR) • PROPULSION (MOTOR)	• REQUIRES FUEL MAKEUP FOR ORBIT KEEPING IF POWER GENERATOR • REQUIRES ELECTRICAL POWER IF PROPULSION • RDT&E REQUIRED BEFORE BASE-LINING

0663-057(T) V83-0165-141(T)

Fig. 2-45 Tether Considerations

3 - SUBSYSTEMS

The following five subsystems studied were selected because they can have significant impact on the architecture and program costs. They are:

- Electrical Power
- Communications and Tracking
- Data Management
- Environmental Control/Life Support
- Guidance, Navigation and Control.

Trade studies were performed within the functional areas of these subsystems to define the optimum subsystem design characteristics (based on cost, availability and performance criteria). These trade study results were subsequently used in generating the recommended Space Station element architecture.

The trades were performed in accordance with the guidelines defined in Subsection 1.2. The following subsections describe the major trades performed, recommended follow-on studies, and baseline subsystem design characteristics for the ISS, ESS, TIP and TPP elements. In addition, the subsystem's evolution and corresponding technology development is also presented.

3.1 ELECTRICAL POWER

The Electrical Power Subsystem (EPS) functional areas and their respective candidate components are summarized in Fig. 3-1. Technical and programmatic data describing these components were reviewed, and projected component design parameters were prepared. This component review led to the identification of five candidate subsystem concepts for the initial and evolved elements of the Space Station. These five subsystem concepts were compared on a power figure (watts per kg) and relative cost basis.

Four candidate subsystems use solar arrays to provide the power for loads and recharging the energy storage equipment during the sunlight portion of the orbit. Nickel hydrogen batteries or regenerative fuel cell/electrolysis cell energy storage devices represent the projected 1986 state-of-the-art (SOTA) and provide the required electrical power during the shadow portion of the orbit. Similarly, silicon and galluim arsenide solar cells represent the projected SOTA that would be utilized during the operational life of the Space Station program. Two candidate concepts use modified Solar Electric Propulsion (SEP) silicon arrays plus either a battery or a regenerative fuel cell (RFC) energy storage section. Advanced gallium arsenide solar arrays plus a battery or RFC energy storage section complete the solar array candidate concepts. The fifth candidate concept would be a man-rated modified SP-100 nuclear space power system.

3.1.1 EPS Trade Studies

3.1.1.1 **Power Generation** - The three generic power generation candidates represent the classical competing technologies that have been considered for space applications over the past 25 years.

In low radiation environments (i.e., 28.5 deg, 370 km), the weight and efficiency of the solar arrays are the driving factors in array weight. Thin silicon cells result in the lightest weight blankets. Gallium arsenide (GaAs) must be produced 2 mils thick and 18% efficient to compete on a weight basis with silicon cells in these orbits. A number of silicon and GaAS solar cells were evaluated for near-term (1986 technology available) and far-term (early 1990s technology available) applications. Representative solar cell and array parameters are illustrated in Fig. 3-2 with a projected 30% improvement in specific weight for the early 90s (far-term) solar cell. A modified SEP array design has been used in developing the near-term parameters.

With the high power (22 to 172 kW) and high voltage (140 to 270 vdc) requirements being considered, the method used to transfer the power across a rotary joint becomes an important factor. In addition, the ability to control such a system with an orientation drive becomes more challenging.

Concentrator solar array configurations require solar pointing accuracies on the order of ± 1 deg or less, compared to accuracies on the order of ± 5 deg for planar arrays. Concentrator arrays are not considered a viable concept for the ISS application which requires 1986 SOTA. Concentrator arrays require development of

	POWER GENERATION	ENERGY STORAGE	POWER CONDITIONING & CONVERSION	POWER DISTRIBUTION
CANDIDATE COMPONENTS	SOLAR ARRAY – SILICON – GaAs – PLANAR – CONCENTRATED PRIM. FUEL CELLS NUCLEAR REACTOR ~ SP-100	N_iCd BATTERY N_iH_2 BATTERY REG. FUEL CELLS FLYWHEELS	HI VOLT REG LO VOLT REG BATTERY CHARGE CONT PEAK POWER TRACKER POWER TRANSFER – SLIP – ROLL – TWIST	HARNESS – CU VS AL – ROUND VS FLAT SWITCH GEAR – MAG. LATCH RELAY – SOL. STATE
VIABLE SYSTEM CONCEPTS	• MOD SEPS ARRAY + N_iH_2 BATTERY + NEAR TERM PC, C&D • MOD SEPS ARRAY + REG. FUEL CELL + NEAR TERM PC, C&D • ADV. ARRAY + N_iH_2 BATTERY + ADV. PC, C&D • ADV ARRAY + REG. FUEL CELL + ADV PC, C&D • NUCLEAR REACTOR			
ANCILLARY TRADES	– EMERGENCY POWER SOURCES			

R83-0663-058(T) V83-0165-129(T)

IRAD

Fig. 3-1 Electrical Power Subsystem Trade Areas

	TECHNOLOGY READY 1986	TECHNOLOGY READY EARLY 90's
CELL		
TYPE	SILICON	GaAs
JUNCTION TYPE	SHALLOW	SHALLOW
SIZE CM x CM	2 x 2	2 x 2
THICKNESS ~ MILS	2	2
EFFICIENCY ~ PERCENT	13.5	18
MAXIMUM POWER mW/CM^2	18.27	24.35
RADIATION DEGRADATION @ 1 x 10^{15} 1 Me V e/CM^2 ~ PERCENT	22	20
WEIGHT g/CM^2	0.016	0.0279
ARRAY (BLANKET & STRUCTURE)		
WATTS/M^2 B.O.L.	117	201
WATTS/M^2 WITH RADIATION DEGRADATION	91	161
WATTS/KG B.O.L.	231	295
WATTS/KG WITH RADIATION DEGRADATION	180	235
ARRAY & AUXILIARIES – WATTS/KG	72.6	110
PERFORMANCE FIGURE – KG/KW	13.8	9.1

R83-0663-059(T) V83-0165-130(T)

IRAD

Fig. 3-2 Electrical Power Solar Cell & Array Parameters small area GaAs cells operating in a high temperature environment, stable long-lived optics and accurate pointing capability. This technology could be used on the ESS.

Primary fuel cells would require approximately 18,000 kg of reactants over 90 days to support the initial Space Station energy requirements. External tank scavenging may provide 4500 to 7000 kg of reactants per Space Shuttle flight, but the reactant requirements for the initial and evolved stations are beyond the practical application range of the fuel cell.

The SP-100 nuclear space power system is nominally a 100 kw power source for unmanned spacecraft applications that require high specific power and long life. The current SP-100 conceptual design, under evaluation by Jet Propulsion Laboratory and Los Alamos National Laboratories, is being studied by the Grumman Research Department for application to the Space Station. The radiation shield characteristics proposed for the unmanned spacecraft mission have been revised and result in a significant increase in shield mass for the ISS. A mass summary is shown in Fig. 3-3 and includes 15,000 kg for a 2π shield design having a layer of high atomic number material (tungsten or tungsten/lithium hydride) between the inner and outer layers of lithium hydride. This shield design reduces the reactor dose to approximately 5 REM per quarter year at 60 m from the reactor source.

ITEM	MASS ~ KG
• NUCLEAR SUBSYSTEM	
– REACTOR, HEAT PIPES, REACTOR CONTROL	910
• RADIATION SHIELD	15,000
• CONVERSION/RADIATOR SUBSYSTEM	640
• POWER CONTROL SUBSYSTEM	130
• USER INTERFACE EQUIPMENT	70
SUBTOTAL	14,750
• ADDITIONAL INTEGRATION STRUCTURE	1,475
TOTAL SYSTEM MASS	18,225

0663-060(T) V83-0165-137(T)

IRAD

Fig. 3-3 Nuclear Reactor Mass Summary

3.1.1.2 **Energy Storage** - Nickel-cadmium (Ni-Cd) batteries have been principally used to provide energy storage over the past 20 years. Due to their inherent cycle life limitation and temperature sensitivity, these batteries are derated for the specific application by use at a limited "depth of discharge" (DOD) or fraction of rated capacity that has been typically 15 to 25% for low earth orbit. These figures have been derived from cycle life test programs and flight experience.

Over the past 12 years, development of a cell technology to replace Ni-Cd has proceeded. The nickel-hydrogen (N_iH_2) cell is derived from the Ni-Cd cell design by substituting a hydrogen (catalytic) electrode for the cadmium electrode. This cell exhibits an improved rate capability, and the effect of higher temperatures on life should be reduced due to the improved stability of the negative electrode and separator material. Life should therefore be improved, especially at the higher DODs and higher temperatures where Ni-Cd cells were more life limited. Although the rapid evolution of cell designs has limited the development of long-term life data, high stress tests of recent cell designs suggests that in low earth orbit applications N_iH_2 cells may have twice the cycle life of NiCd cells. At present, N_iH_2 cells developed for low earth orbit applications are comparable in weight to NiCd cells but enable approximately a 50% weight reduction via improved DOD.

For the Initial Space Station requiring near-term engineering development, nickel-hydrogen appears to be the most attractive energy storage technology based on its high specific energy, specific power and efficiency. The N_iH_2 design parameters used in developing trade and architectural data are shown in Fig. 3-4.

The regenerative fuel cell-electrolysis cell energy storage system represents a viable concept for near-term applications. Although it has been considered as a closed-loop dedicated storage concept in this study, the potential advantages of in-

100% DEPTH OF DISCHARGE (DOD)

- SPECIFIC ENERGY DENSITY 40.0W-HR/kg
- VOLUMETRIC ENERGY DENSITY .... 0.025W-HR/CM^3

~ 5720 CYCLES PER YEAR IN 28.5° LEO

~ 60% DOD FOR 5 YEAR LIFE (~ 28600 CYCLES)

PROJECTED LEO PERFORMANCE (WITH 10% CELL REDUNDANCY)

- 22W-HR/kg
- 0.015W-HR/CM^3

0663-061(T) V83-0165-131(T)

IRAD

Fig. 3-4 N_iH_2 Battery Energy Storage Design Parameters

11/2/1

tegrating this concept with life support, attitude control and other propulsion system requirements need to be evaluated. Reviews were held with the alkaline (United Technologies, Inc.) and solid polymer electrolyte (General Electric) cell developers to obtain realistic performance and design characteristics. Figure 3-5 presents the design parameters used and are based on projected near-term solid polymer electrolyte technology for the electrolysis cell and alkaline technology for the fuel cell.

- STORAGE SECTION ELECT TO ELECT EFFICIENCY – 55%
- FUEL CELL REACTANT CONSUMPTION: 2575W-HR/kg (0.8545Lb/KW-HR)
- ELECTROLYSIS (SOLID POLYMER) ENERGY CONSUMPTION
  - 5.15 kW-HR/.kg OF O_2 (2.34 W-HR/lb OF O_2)
- TOTAL FUEL CELL & ELECTROLYSIS CELL
  - SPEC WT 10 kg/ kW (22lb/kW)
  - SPEC VOL 0.018 m^3/kW (0.6ft^3/ kW)
- 200 psi GASEOUS STORAGE (INCONEL) TANKS
  - O_2 TANK ~ 2.5 kg/kg OF O_2
  - H_2 TANK ~ 40 kg/kg OF H_2
- HEAT REJECTION – 4 kg PER kW OUTPUT FROM FUEL CELL

0663-062(T) V83-0165-134(T)

IRAD

Fig. 3-5 Regenerative Fuel Cell Energy Storage Design Parameters

3.1.1.3 Power Conversion & Control - Several high voltage high power converters have been under development to satisfy the requirements of NASA large power system programs. The MSFC Programmable Power Processor (P^3) regulator, developed for the 25-kW Power System produces 2.5 kW at 32 V and over 10 kW at 140 V. The LMSC Transformer Coupled Converter (TCC) regulator delivers 6 kW at 32 V for the Space Shuttle PEP program. The TRW series resonant Power Processing Unit (PPU) regulator provides 2.5 kW for the SEP ion engine power at 1100 V and other voltages. Power semiconductor device development and availability have supported these new power converter designs in a timely fashion.

The power controller functions required include:

- Solar array voltage limiting/regulation
- Solar array peak power tracking
- Battery charge control (current/voltage regulation)

- Bus voltage regulation
- Load voltage transformation and regulation.

These functions are distributed to the appropriate regulators, converters, and the PC&C area is projected to require 10 kg/kW in the initial Space Station application.

3.1.1.4 Power Distribution - Lightweight power conductors are needed for large space power systems and ongoing R&D offers the potential of lightweight high current density conductors. Aluminum conductors used for the large power cables would result in a considerable weight savings compared to copper conductors. Aluminum is being considered for solar array harnesses. However, the disadvantages of aluminum (i.e., low tensile strength, poor flexibility and poor crimp terminability) must be evaluated against the potential weight savings. The use of high voltage itself will significantly reduce harness weight and the potential for aluminum to creep, causing looseness in the connector, eventual arcing and, finally, an open circuit leads to the selection of copper wire for the required harnesses.

The power system switchgear is relevent to both the power conversion function, since it uses switches and controls, and to the harnessing function in that the switches are distributed throughout the power distribution lines. The solid-state switchgear technology now ready for application appears to be adequate for high voltage, high power systems. The power distribution function is projected to require 8 kg/kW in the initial station application.

3.1.1.5 Candidate Concepts Evaluation & Baseline Selection - Five candidate system concepts were synthesized and compared on a weight-trade basis. They are identified in Fig. 3-1. Figures 3-6 through 3-8 present block diagram descriptions for the solar array/battery, regenerative fuel cell and nuclear power system, respectively.

The weight performance factors described in the previous subsections were used in developing the overall power figure for each concept. Overall energy storage section efficiency differences result in requiring 2.2 and 2.7 array output watt for each load watt provided by the N_iH_2 and regenerative fuel cell section, respectively.

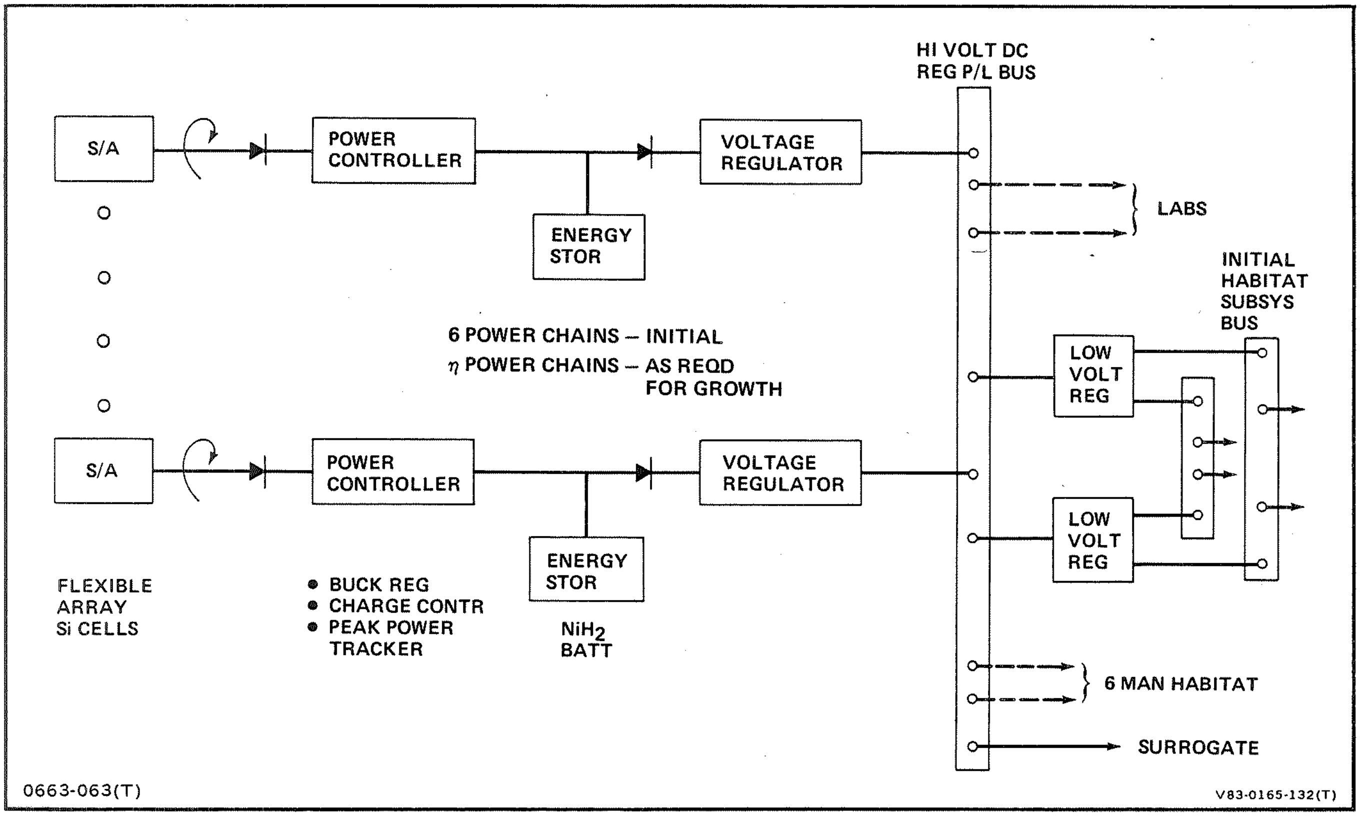


Fig. 3-6 Solar Array/Energy Storage Block Diagram

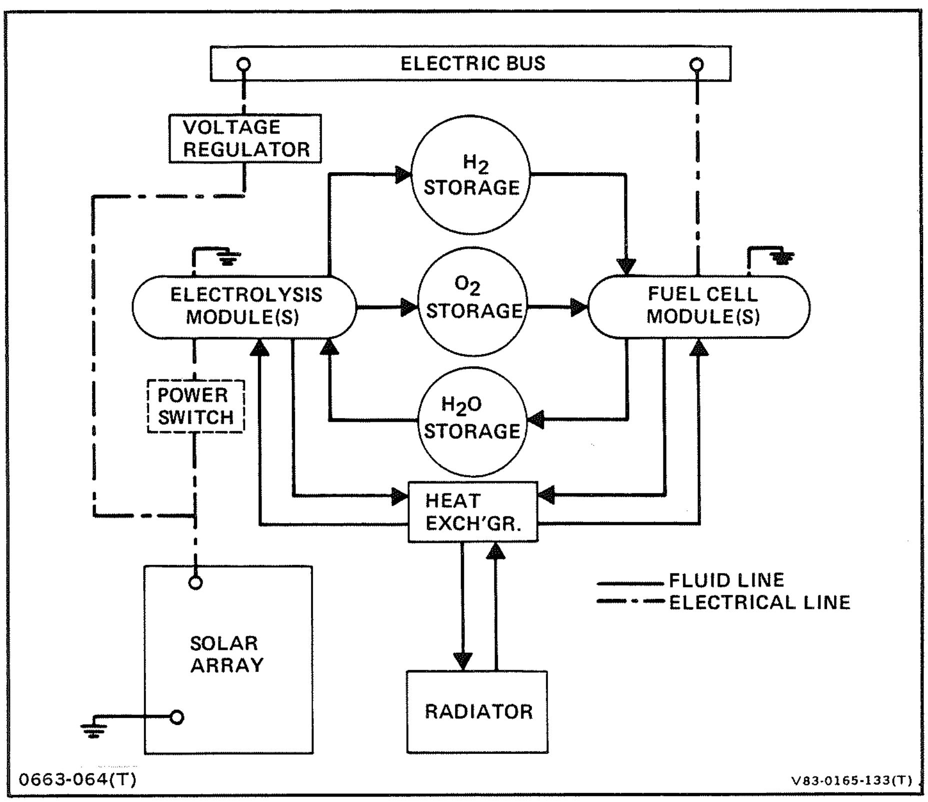


Fig. 3-7 Regenerative Fuel Cell Orbital Energy Storage Block Diagram

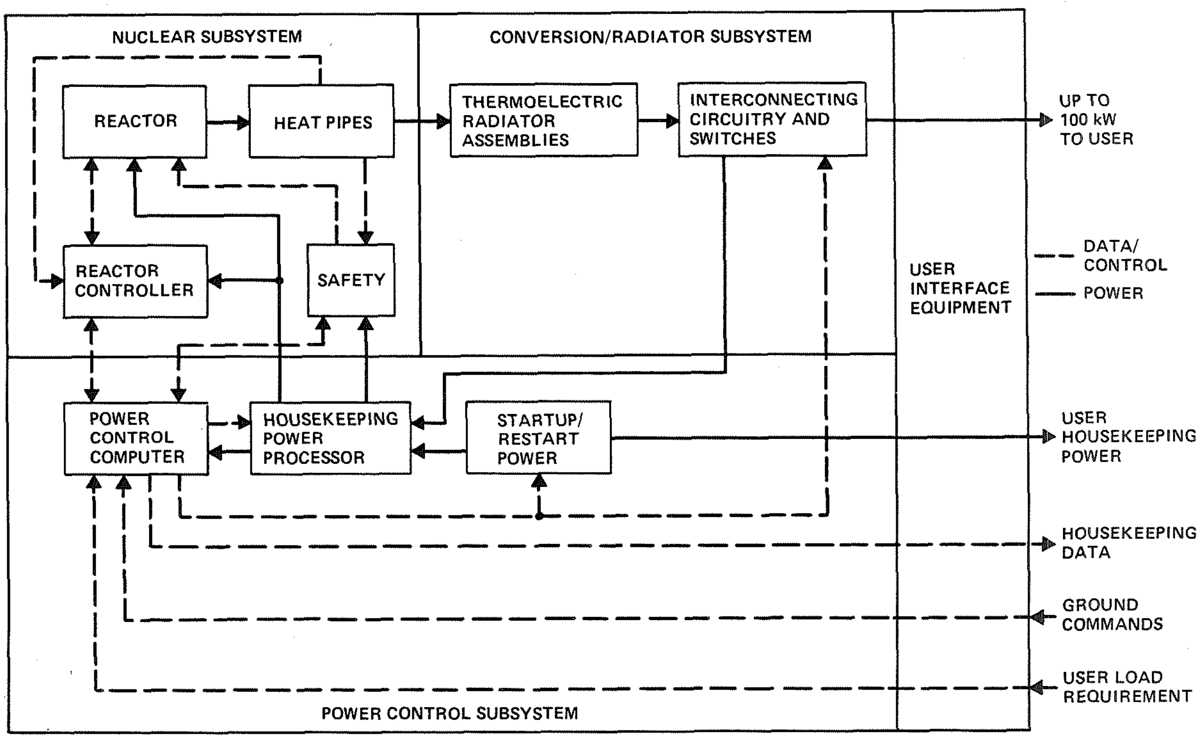


Fig. 3-8 Nuclear Power System Block Diagram

The overall power figures are presented in Fig. 3-9 and indicate that the advanced array/RFC concept would provide the lightest weight system; whereas, the modified SP-100 nuclear system would be the heaviest system. The advanced array concepts would be available in the early 1990s and represent viable alternatives for the evolved configurations. The modified SEPS/N_iH_2 concept would weigh approximately 300 kg more for the ISS than the modified SEPS/RFC approach, which translates into approximately $1 million in launch costs. However, RFC development costs are estimated to be significantly more than N_iH_2 development costs and negate the RFC weight advantage. The apparent complexity of the RFC approach is another factor in favoring the nickel-hydrogen approach for the ISS. The modified SEPS array/N_iH_2 energy storage concept represents the most cost effective approach when combining shuttle launch, development and production costs.

The modified SEPS/nickel-hydrogen energy storage concept is selected as the baseline considering cost, availability and reliability/complexity. This concept is also used (in modular power chains) in developing the EPS for the other elements of the program.

CONCEPT DESCRIPTION	POWER FIGURE (WATTS/KG)	IOC	RELATIVE DEV. & PROD. COST
MOD/SEPS ARRAY + N_iH_2 BATT + PCC&D	13.3	1990	LO
MOD SEPS ARRAY + REG FUEL CELL + PCC&D	15.0	1990	MED
ADV ARRAY + N_iH_2 BATT + PCC&D	16.8	EARLY 90's	MED
ADV ARRAY + REG FUEL CELL + PCC&D	20.4	EARLY 90's	MED-HIGH
NUCLEAR REACTOR (MOD SP-100 TYPE)	5.5	MID-LATE 90's	VERY HIGH

PCC&D = POWER CONV, COND & DISTRIB
0663-065(T) V83-0165-138(T)

IRAD

Fig. 3-9 EPS Candidate Concepts Evaluation Criteria

3.1.1.6 Emergency Power Considerations

The complete failure of the baseline primary EPS is a remote possibility but power/energy for such an emergency condition is required to satisfy program requirements. The emergency power section needs to be stored for long durations; have immediate operational capability; be maintainable and reliable.

Two approaches were evaluated:

- Primary high-capacity silver zinc (Ag-Zn) batteries
- Shuttle fuel cells plus gaseous (400 psi) reactant storage.

Energy requirements could vary from 600 kW-h to 2000 kW-h with Ag-Zn batteries weighing from 2500 to 8200 kg to satisfy these requirements. Fuel cell section weights are estimated to be from 1100 to 3600 kg for this application.

For a representative 14-day (3.5 kW average load) requirement (i.e. approximately 1200 kW-h) the fuel cell approach would weigh 2730 kg less and save more than $6 million in launch costs. This cost saving would increase significantly with emergency energy growth requirements. Therefore, the fuel cell is selected as the baseline emergency power source based on projected overall cost savings and flexibility for future growth.

3.1.1.7 Follow-On Trade Studies - A benchmark energy management study is recommended to assess the viability of integrating the EPS, EC/LSS, Thermal Control and GN&C requirements and resulting configurations. Two areas (common LH_2-LO_2 reactant storage and electrical energy storage/attitude control via energy wheels) may offer significant logistic supply advantages when compared to non-integrated approaches.

A study to determine the high voltage dc system level is also recommended. The study should include cost trades of available components vs optimized components requiring development; as well as the primary trade of voltage levels vs power distribution cable, harness and bus weights.

A study to compare the cost of solar array assemblies of various designs is recommended. Near-term advances in solar cell and cover performance, lightweight structural design and automated array fabrication capability require close monitoring. As a result of these possible major improvements, solar array life cycle costs (including Shuttle weight and volume considerations) for all elements of the Space Station program should be synthesized and evaluated. In addition, on-orbit assembly of add-on modules for increased capacity, servicing and repair of existing arrays should be included in this total array cost evaluation.

An emergency power supply study is recommended which would identify equipment required for crew safety and define achievable rescue time. Candidate power supplies would be evaluated to meet these requirements.

3.1.2 Subsystem Description

3.1.2.1 Requirements - Stationkeeping electrical power requirements were developed for three phases of the LEO Stations evolution. Electrical power requirements were developed for initial, mid and evolved configurations tied to three-, six- and nine-man support capability. The stationkeeping requirements range from 11 kW for the initial configuration up to ~ 26 kW for the evolved nine-man station. Figure 3-10 lists the functional subsystem requirements versus the crew size/operational timeframe.

FUNCTION	POWER REQUIREMENTS, kW		
	INITIAL CONFIG (1990 IOC) 3 MEN	MID (1993) 6 MEN	EVOLVED (1996) 9 MEN
• EC/LS, FOOD, HYGIENE	2.8	6.0	9.0
• THERMAL CONTROL	0.5	1.8	2.5
• COMM, INSTRUMENTATION, CONT. & DISPLAYS	2.0	2.5	3.0
• DATA MANAGEMENT	2.0	3.0	4.0
• G&C/PROP	0.4	0.6	0.8
• ILLUMINATION & EPS	3.0	4.5	5.7
• MISC (REC, ON-BOARD C/O)	0.3	0.6	1.0
TOTAL LOAD	11	19	26

0663-066(T) V83-0165-144(T)

IRAD

Fig. 3-10 Station Keeping Electrical Power Requirements — 28.5 LEO

Figure 3-11 lists the stationkeeping, civil mission payloads and Tended Industrial Platform (TIP) power requirements. These requirements are then used to define the baseline LEO implementation options (Fig. 3-12) which range from 22 to 66 kW with TIP vehicles obtaining power via their own individual power supplies. In addition, the power levels for a station with an integral industrial park are also shown in Fig. 3-12.

The solar array is sized by the load power required, array performance characteristics, sun/shadow orbit parameters and the overall power system inefficiencies, including environmental degradation. Figure 3-13 illustrates the percent shadow time for the two orbits under study. The energy storage capacity design condition is the same for the 28.5 deg orbit and the sun-synchronous "noon" orbit.

	POWER REQUIREMENTS, kW										
BASIC FUNCTIONS	'90	'91	'92	'93	'94	'95	'96	'97	'98	'99	2000
• STATION KEEPING (CREW SIZE)	11 (3 MEN) →	11	11	19 (6) →	19	26 (9) →	26	26	26	26	26 →
• CIVIL MISSION PAYLOADS (SCI & APPLIC; TECH DEV)	11	22	22	22	22	40	40	40	40	40	40
• TENDED INDUSTRIAL PLATFORM (OR INDUSTRIAL PARK)	1	12	17	25	40	45	57	75	86	97	106

0663-067(T) V83-0165-145(T)

IRAD

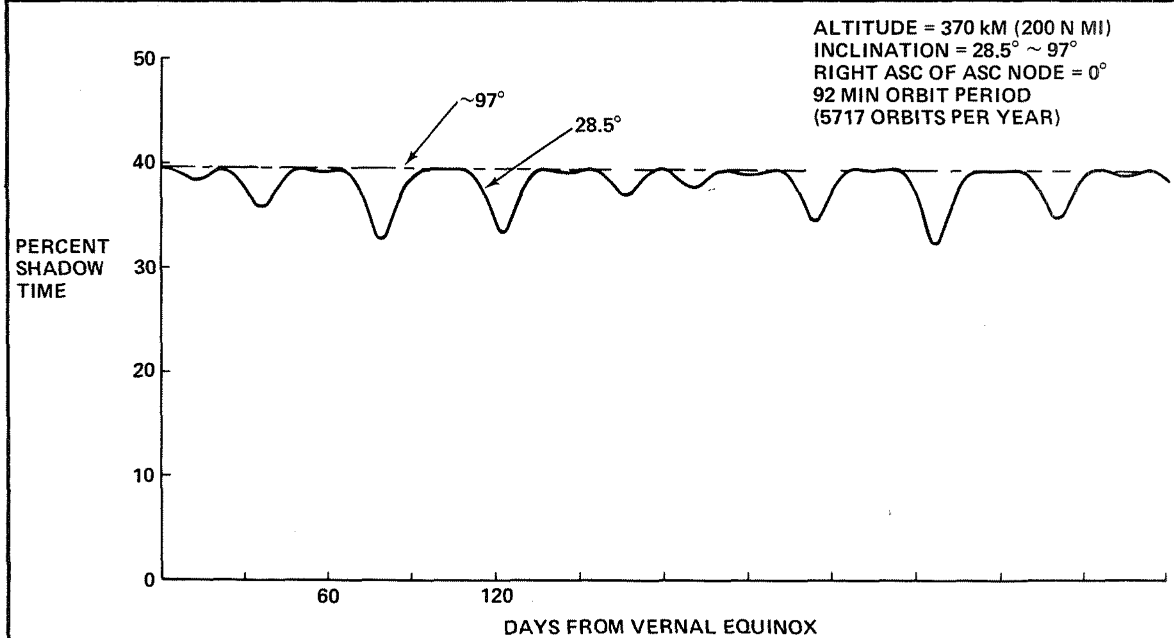
Fig. 3-11 EPS Power Requirements Basic Functions 1990 – 2000

	POWER REQUIREMENTS, kW										
IMPLEMENTATION OPTIONS	'90	'91	'92	'93	'94	'95	'96	'97	'98	'99	2000
• STATION KEEPING CIVIL MISSION P/L SPACE STATION	22	33	33	41	41	66 →					→
• TENDED INDUSTRIAL PLATFORM	1	12	17	25	40	45	57	75	86	97	106
• STATION KEEPING + CIVIL MISSION P/L + TENDED INDUSTRIAL PLATFORM	23	45	50	66	81	111	123	141	152	163	172

0663-068(T) V83-0165-146(T)

IRAD

Fig. 3-12 EPS Power Requirements Implementation Options 1990 – 2000



- ENERGY STORAGE CAPACITY DESIGN CONDITION SAME FOR ALL INCLINATIONS
  – 39% SHADOW (36 MIN)
  – 61% SUN (56 MIN)

0663-069(T) V83-0165-147(T)

Fig. 3-13 Solar Array Energy Storage Orbit Considerations

The energy storage section is sized to provide 13.2 kW-hr for the initial 28.5 deg station and 36 min of the required average power for all other stations and platforms.

The emergency power requirement (3.5 kW average) has been assumed to be needed over 14 days before rescue can be effected.

3.1.2.2 **Baseline Architecture** - The initial 28.5 deg Space Station power and energy requirements are satisfied by use of modified SEP arrays, nickel hydrogen batteries and appropriate power control, conversion and distribution components. The power and energy requirements for the evolved 28.5 deg Space Station is also supplied by additional power system modules incorporating the same basic technology used for the initial station. These two power systems are briefly described in Fig. 3-14 and 3-15 for the initial and evolved configurations.

To provide redundancy, each wing is mounted to its own gimbal system which points the array to within 5° of the sun line. In its flight attitude, the ISS has the longitudinal axis along the local vertical. The solar array is outboard with reference to the earth. During the sunlit period, the whole array rotates at orbital rate about an axis normal to the orbit plane. This rotation is from 90° before, to 90° past the sun/earth line. For the remaining sunlit part of the orbit, cosine losses are experienced by the system and result in approximately 0.5% overall power loss. To obviate the need for slip rings, the orbital rotation is not continuous. After 180° rotation during sunlight, rotation reverses during the dark period. The second degree of freedom provided by the gimbal system is necessary to track the north/south movement of the sun caused by orbital precession. This movement is less than 1°/day and is provided for by indexing at intervals.

The emergency primary fuel cell/gaseous reactant storage section is sized to provide 1200 kW-hr and the section components are described in Fig. 3-16.

The Tended Industrial Platform and Tended Polar Platform use the power system components developed for the initial 28.5 deg Space Station. The solar array, energy storage, control and conversion description can be found in Fig. 3-14. The power distribution weights would be reduced to 180 kg for the TIP configuration.

The Tended Polar Platform would use a scaled-up version of the ISS/EPS. Eight modular power chains would be required, compared to six power chains for

• LOAD POWER REQUIRED	22 kW
• SHADOW TIME ENERGY REQUIRED	13.2 kW-HR
• SOLAR ARRAY & AUXILIARIES	660 kg
– 2 MOD SEPS WINGS – 13.5% EFF. Si CELL	425 m^2
– (EACH WING 18.5 m x 11.5 m)	
• ENERGY STORAGE	
– 6 Ni H_2 BATTERIES	660 kg
– EACH BATTERY: 22.5 AMP-HR, 180V, 4 kW-HR CAPACITY 110 kg, 0.17 m^3	1.02 m^3
• POWER CONTROL & CONVERSION	220 kg
– BATTERY CHARGE CONTROL	0.24 m^3
– PEAK POWER TRACKER; REGULATORS	
• POWER DISTRIBUTION	670 kg
– MAST HARNESS	
– HABITATION MODULE	
– SURROGATE STRUCTURE	
0663-070(T)	V83-0165-148(T)

IRAD

Fig. 3-14 EPS Description Initial Space Station

• LOAD POWER REQUIRED	66 kW
• SHADOW TIME ENERGY REQUIRED	39.6 kW-HR
• SOLAR ARRAY & AUXILIARIES	1980 kg
– 2 EVOLVED WINGS	1240 m^2
– (EACH WING ~ 18.5 m x 33.5 m)	
• ENERGY STORAGE	1970 kg
– 18 N_i H_2 BATTERIES	3.06 m^3
• POWER CONTROL & CONVERSION	650 kg
	0.72 m^3
• POWER DISTRIBUTION MAST, HAB MODULES, LABS & TRANSPORT HARBOR/SURROGATE STRUCTURE	1240 kg
0663-071(T)	V83-0165-149(T)

IRAD

Fig. 3-15 EPS Description Evolved Space Station

• ORBITER FUEL CELLS (2)	186 kg
• RADIATOR & POWER CONDITIONING	90 kg
• O_2 & H_2 REACTANTS	466 kg
• O_2 TANK (INCONEL @ 400 PSI)	518 kg 2.9 m dia
• H_2 TANK (INCONEL @ 400 PSI)	1040 kg 3.6 m dia
TOTAL WEIGHT	2300 kg
0663-072(T)	V83-0165-237(T)

IRAD

Fig. 3-16 Emergency Power Section Components the ISS. The section/component weight, area and volume requirements for the Tended Polar Platform are listed below:

• Solar Array and Auxiliaries	910 kg
2 Wings-each 18.5 m x 15.2 m	562 m^2
• Energy Storage	872 kg
8 N_iH_2 Batteries	1.4 m^3
• Power Control and Conversion	290 kg
	0.32 m^3
• Power Distribution	545 kg.

The array output capability of 64 kW would provide an average load capability of 29 kW.

3.1.3 Subsystem Evolution

The EPS selected to satisfy the initial Space Station requirements is based on a varied technology base. The solar array would be developed by modifying the array technology being pursued under the MSFC Solar Electric Propulsion (SEP) Program. This technology development program has been under way since 1975, and provides a credible base upon which to designed and manufacture the station array. The other major EPS sections (i.e., energy storage) require more advances in the present SOTA to bring the selected components up to the 1986 technology ready level. One area needing further analysis and development with a significant impact on the subsystem design is the degree of autonomy (operation with no involvement of man) planned for the Space Station and subsystems.

3.1.3.1 Required Technology Development - In low radiation environments (i.e. 28.5 deg orbit), the weight and efficiency of the solar cells are the driving factors in array weight. Additional development is required in thin (2-mil) solar cells and low weight cell covers to define the cost effective assembly. The higher radiation orbits (i.e., polar) could use the 28.5 deg array design with the array capability reduced in proportion to the expected radiation degradation.

Power transfer assemblies previously designed for satellite application are susceptible to the corona discharge and electrical breakdown problem of the internal gas at critical pressure. Prototypes of competing technology (i.e., sealed slip ring, twist flex) should be designed and evaluated. Similarly, orientation drive prototypes should be designed and evaluated to meet the technology ready time-frame.

Nickel-hydrogen appears to be the most practical energy storage technology. Further analytical and test evaluation as proposed in the NASA Fuel Cell Program Plan is warranted to determine whether the projected system weight and performance make it a viable competitor to the nickel-hydrogen battery.

Power control and conversion development will be impacted by the degree of autonomy selected for the space station and subsystems.

The use of high voltage distribution will significantly reduce power distribution weight as a fraction of the total power subsystem. However, although copper wire appears adequate and reliable from initial applications, development of reliable aluminum terminals would provide significant conductor weight reduction.

The 1986 technology base used in providing the initial 28.5 deg Space Station EPS would be adequate in meeting the requirements for the evolved Space Station and all other elements of the program.

The modular design of the EPS would be effected by providing a number of parallel power chains as illustrated in Fig. 3-6.

The capability required for the initial station and the development effort needed to bring the current technology up to that new level is summarized in Fig. 3-17. The more significant advancements will be required in designing and providing the solar auxiliaries and automated power management components. With the increasing complexity of spacecraft systems operation, the need to minimize the crew's involvement in housekeeping activities and reduce the spacecraft life cycle costs, an autonomous power management capability becomes a real requirement. The advent of advanced microprocessor and computer technology make it feasible to consider this capability for the Space Station program. The NASA OAST Spacecraft Automation Technology task will trade the degree of autonomy vs the attendant complexity and cost. This task should provide the technology to support a cost-effective subsystem design. The solar array technology advancement will focus primarily on providing the mechanical and electrical design-to-manufacturing array capabilities, rather than on component research and development. The ongoing DoD- and NASA-sponsored programs on nickel-hydrogen and regenerative fuel cell energy storage sections, respectively, should provide the required technology. A nominal development effort will be required in providing the power distribution components.

AREA	CAPABILITY REQUIRED	RELATIVE EFFORT
SOLAR ARRAY – BLANKET – STRUCTURE	HI VOLT OPERATION, MINIMUM LONG TERM RADIATION DEGRADATION	CREDIBLE '83 SOTA WITH NOMINAL ADVANCEMENT
SOLAR AUXILIARIES – POWER TRANSFER – DRIVE	HI VOLT OPERATION WITHOUT CORONA DISCHARGE AND ELECTRICAL BREAKDOWN	SIGNIFICANT DEVELOPMENT
ENERGY STORAGE	RELIABLE LONG TERM OPERATION	CONTINUE ON-GOING DEVELOPMENTS
POWER CONV. & CONTROL (AUTOMATED POWER MGMT)	SUBSYSTEM CONVERSION, REGULATION AND CONTROLS PLUS AUTOMATED ELECTRONIC FUNCTIONS, I.E. COMMAND, TELEMETRY, FAULT ISOLATION	SIGNIFICANT DEVELOPMENT
POWER DISTRIBUTION – HARNESS – SWITCH GEAR	RELIABLE TERMINALS, HEAT SINKING HARNESS TECHNIQUES MINIMUM WEIGHT & POWER LOSS, FAULT ISOLATION SPEED	NOMINAL DEVELOPMENT

0663-073(T)

IRAD

Fig. 3-17 Technology Development Summary Initial 28.5° Space Station

3.1.3.2 Modular Evolution - The increased EPS capability required to satisfy the station's evolutionary growth will be provided by adding modular power chains similar in size, function and performance capability developed for the initial configuration. The evolved configuration will be used in sizing the array's structural support mast and distribution harness. The initial baseline configuration includes these provisions.

The solar array/nickel-hydrogen modular power chain approach appears to offer a practical and cost-effective design to meet the growth envisioned at this time. The use of other more weight-optimum concepts must trade the savings in launch costs against the anticipated development and production costs. As an example, an advanced solar array using 18% efficient GaAS cells may provide a 60% array area and 40% array weight reductions. In addition, there would be a reduction in attitude control requirements. The array weight reduction would translate into approximately $700,000 savings in launch costs (assuming approximately $1200/lb to 28.5 deg orbit). However, development (unless shared with the Department of Defense) and production costs would probably negate the launch cost savings for this element of the program. Launch costs to sun-synchronous orbits are approximately three-times more expensive than to 28.5 deg orbits, so the technology developments in solar cells would be monitored closely vs the anticipated mix of station element requirements. The higher radiation environments encountered in polar orbits is another incentive to closely monitor GaAS technology for possible use in high inclination applications.

The projected weight savings for a regenerative fuel cell section would approach approximately 30% of the nickel-hydrogen concept. The launch cost savings (in providing the 13.2 kW-hr delta required for the energy storage capacity in the six-man station) would be approximately $500,000. However, significant development costs and section complexity appear to override the estimated launch cost savings.

In summary, the baselined solar array/nickel-hydrogen modular power chain concept offers a cost-effective and flexible approach to satisfy the evolving station's increasing requirements. However, if the power loads grow to 200 to 300 kW (possibly due to attached commercial production facilities), the use of SP-100 nuclear space power generation technology would be considered. Array area requirements would then approach 4000 m^2 - 7000 m^2 for these load levels and could produce significant GN&C requirements.

3.2 COMMUNICATIONS & TRACKING

The communications & tracking subsystem study results, attributes, characteristics, key issues and baseline architecture are provided in the paragraphs below under the following three major subject headings: Analysis & Tradeoffs; ISS/ESS Subsystem Configurations; Subsystem Evolution. The information and data herein is complementary to that available in Volume II, Book 2, Part III, prepared by COMSAT General.

3.2.1 Analyses & Tradeoffs

A study roadmap defining the interconnection and sequence of major study areas is presented in Fig. 3-18. The areas identified for study are those considered most significant and influential to systems/subsystems design features. Items designated as key issues, tradeoffs and future follow-on studies are derived for each area and are identified and discussed in Subsections 3.2.1.7 and 3.2.1.8.

3.2.1.1 Project Guidelines - Project guidelines and development influences affected by these guidelines are identified in Fig. 3-19. These guidelines have been derived from analyses herein and NASA documentation. Autonomy and the inherent automation requirement strongly narrows down the system/subsystem architecture options. The technology differentials between the initial and evolved configurations bear heavily on an evolution growth plan and intricately relate to technology/state-of-the-art status. Technology transparency, 1986 technology and fail operational - fail safe criteria influence redundancy and commonality as design guidelines.

3.2.1.2 System Operational Guidelines - A listing of system operational guidelines is provided in Fig. 3-20. These guidelines are derived from system analyses herein and NASA documentation. In association with RF LINK simultaneous operation (SIMOP) requirements, an RF LINK topology overview is presented in Fig. 3-21, which clearly indicates the Space Station as a RFI "SINK" (similar to Heat Sink) relative to the following interactions: Space Station-terrestrial; Space Station-RF terminals; Space Station onto itself. RFI is a key issue item since it can inhibit Space Station RF LINK operations.

The RF LINK topology illustration suggests the use of the Space Station (28.5 deg inclination) as a "Communications Node" for those prior designed satellites not serviceable by the Space Station. This particular operation would involve Space Station reception of data from the satellite which in turn receives commands from the Space Station. This is an advantage for those satellites not possessing a TDRS

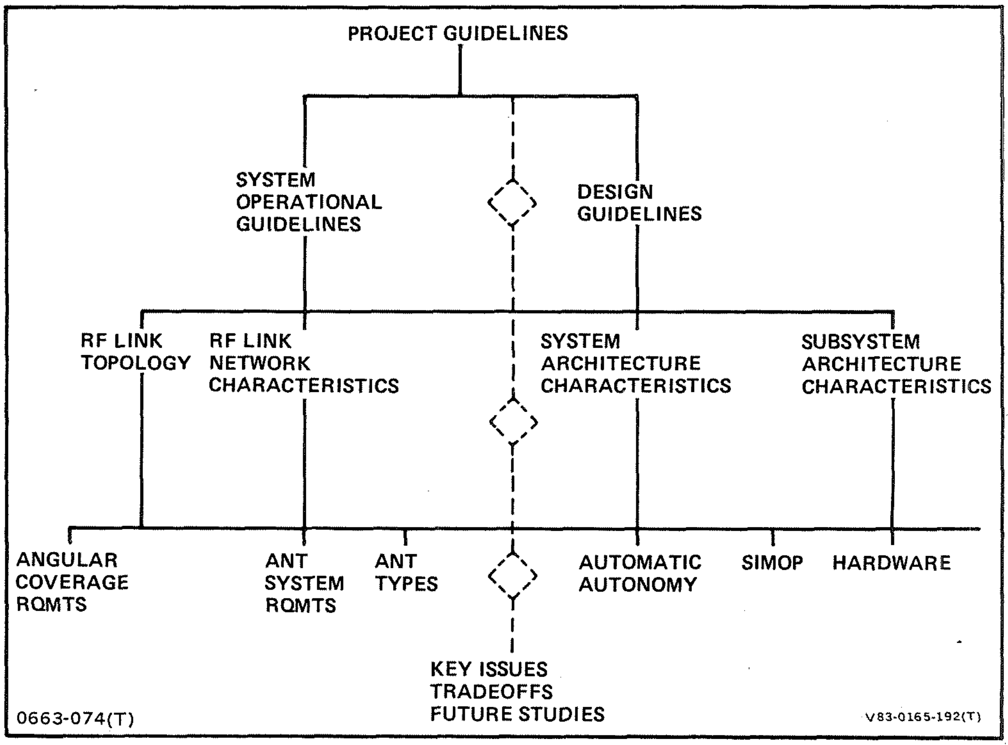


Fig. 3-18 Communications/Tracking Analyses — Tradeoffs Study Roadmap

IRAD

GUIDELINE

- LAUNCH INITIAL SPACE STATION CONFIGURATION 1990
- LAUNCH EVOLVED SPACE STATION CONFIGURATION 2000
- LOW COST – LOW RISK
- INITIAL CONFIG – 1986 TECHNOLOGY
- TECHNOLOGY TRANSPARENCY
  – EVOLUTION GROWTH W/O MAJOR REDESIGN FOR ULTIMATE CONFIGURATION

AREAS INFLUENCED

- TECHNOLOGY/HARDWARE
- DESIGN GUIDELINES
- TECHNOLOGY/STATE-OF-THE-ART
  – PRESENT
  – FORECASTED
  – DEV REQD
- EVOLUTION PLAN

GUIDELINE

- ALTITUDE NOMINAL 370 km/28.5°
  – ORBITAL PERIOD ≈ 90 MIN
- FAIL OPERATIONAL – FAIL SAFE
- AUTONOMY – SYSTEM/SUBSYSTEM INDEPENDENCE
  – CREW
  – GSTDN
  – AUTOMATION

AREAS INFLUENCED

- SYSTEM OPERATIONAL GUIDELINES
- DESIGN GUIDELINES
- SYSTEM/SUBSYSTEM ARCHITECTURE

0663-075(T)

V83-0165-193(T)

Fig. 3-19 Project Guidelines

IRAD

- RF LINK TERMINALS
  - EVA/FREE FLYER/ORBITER/GPS/GSTDN/TDRS (1990) – TDAS (2000)
- SIMULTANEOUS OPERATIONAL RF LINK RQMTS
- INDEPENDENT/AUTOMATED SUBSYSTEM OPERATIONS
  - MANUAL/GSTDN OVERRIDE CONTROL
- RF LINK INTERRUPTIBILITY
  - EVA CONTINUOUS/NO INTERRUPTIONS
  - OTHERS WITHIN ORBITAL GEOMETRY BLOCKAGE CONSTRAINTS
  - NO INTERRUPTIONS FOR DOCKING/CLOSE PROXIMITY MANEUVERING
- NAVIGATION GPS UTILIZATION
- SPACE STATION QUIESCENT STATE
  - GSTDN MONITORING/COMMAND CONTROL/ACTIVATION – DEACTIVATION
- EXCLUSIVE USE AS REQUIRED OF ONE TDRS (KSA) BEAM
- EARTH LINK
  - PRIMARY MODE TDRS RELAY
  - BACK UP MODE DIRECT (S-BAND)
  - INITIAL (1990) OPER PHASE DIRECT (S-BAND)

R83-0663-076(T) V83-0165-194(T)

IRAD

Fig. 3-20 System Operational Guidelines

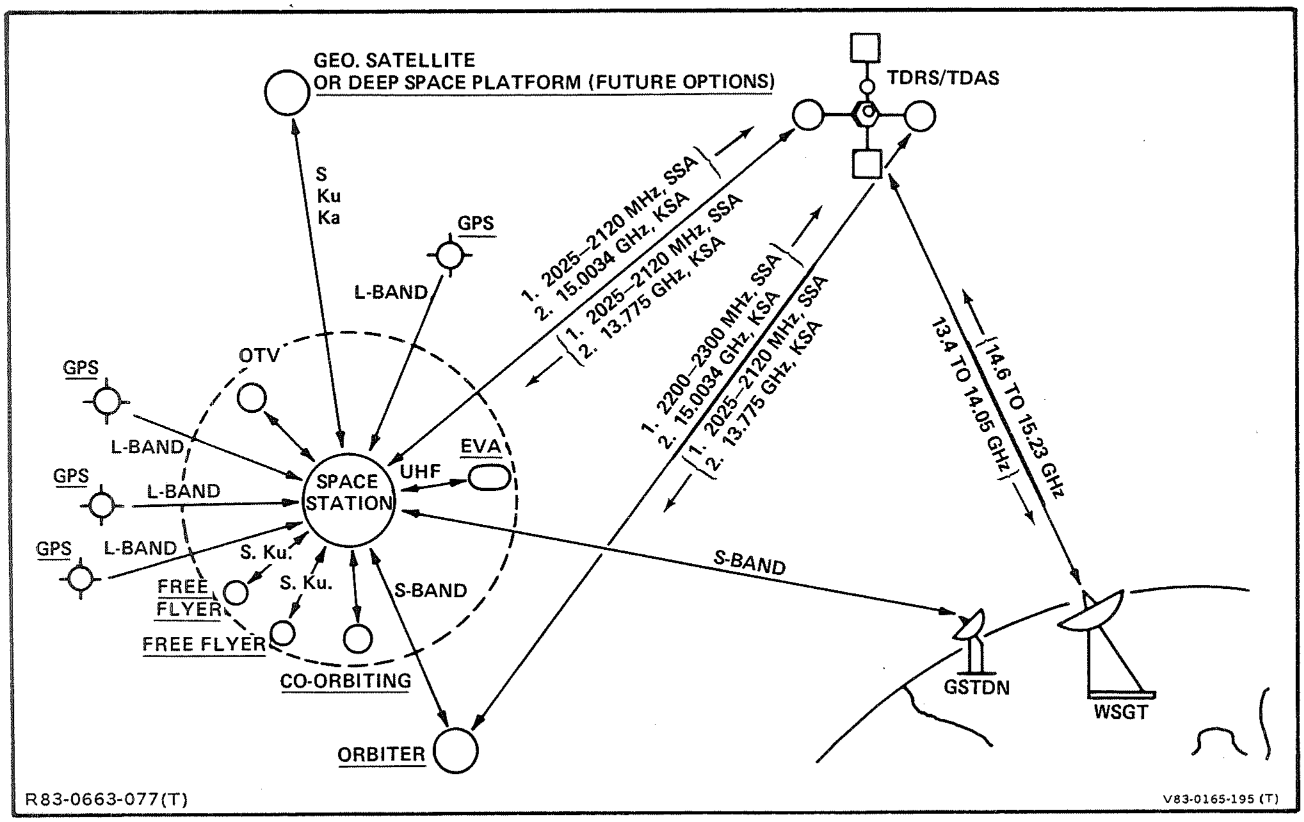


Fig. 3-21 RF Link Topology Overview

link, and for those that do this Space Station operation would ease the TDRS S-band loading. This possible Space Station operation is identified as a follow-on study effort.

The RF LINK network characteristics associated with the topology are identified in Fig. 3-22. RF SIMOP requirements must be practical, realizable and not represent a potentially dangerous situation to the Space Station crew such as might occur if two docking maneuvers were attempted simultaneously. This suggests that initial SIMOP include only one docking maneuver as a safety measure. Those terminals involving relatively large ranges and docking (Orbiter, Tended Industrial Platform, OTV,) inherently require more consideration of the following performance/design parameters: antenna gain; transmitted power; pointing; ranging/tracking; and dedicated vs shared antennas. In general, **a priori** information/data is available for each of these terminals:

- Orbiter - ground launch trajectory and flight path to Space Station
- Tended Industrial Platform - set in proximity formation flying position by Space Station, orbital decay variation known
- OTV - launched by Space Station to known higher altitude position, flight path pre-planned/programmed.

Therefore, antenna acquisition requirements are expected to be moderate. Auto-tracking mode will be used after acquisition for all links. In particular, stationkeeping of the platforms is most effectively accomplished by each having GPS receivers whereby the position data for each can be transmitted to the Space Station, which can then process this data in conjunction with its own GPS data. Since data from four GPS satellites are required for position/velocity determination without clock synchronization, effectiveness is questionable for GPS use by the OTV. This would involve integration of such items as: position at time of launch; flight path (geometric) and total time; time/interval and crossover point during which GPS above OTV and time interval GPS below OTV; and OVT acceleration and velocity.

The maximum simultaneous operation (SIMOP) capacity depicted in Fig. 3-22 suggests a clear distinction be made for ranging/tracking, traffic control, and surveillance radar. For initial Space Station purposes, traffic control is represented by the DMS integrated data processing output display/data for each SIMOP terminal having a ranging/tracking link. Additional possibilities and surveillance radar are discussed as a follow-on study effort.

S/S (TO-FROM)	MAX NO.	FREQ BANDS	MAX RANGE km	ANGULAR SECTOR COVERAGE	RF CHANNEL SERVICE REQUIREMENTS
EVA	4	UHF	10	• 4π STERADIANS ANYWHERE WITHIN SPHERE • 2π COVERAGE CLOSE PROXIMITY • DISCRETE ANGULAR SECTOR – LONG RANGE	• TO S/S: VOICE, DATA, TV • FROM S/S: VOICE, DATA
ORBITER	2	S, K_u, mm	2000	2π STERADIANS PREDOMINANTLY LOWER HEMISPHERE (NADIR)	• TO S/S: VOICE, DATA, RENDEZVOUS/TRACKING • FROM S/S: VOICE, DATA, TRACKING, RANGING • DOCKING
TDRS (1990) TDAS (2000)	1	S, K_u, mm, LASER	38000	2π UPPER HEMISPHERE PREDOMINANTLY. SOME SLIGHTLY LOWER REQD DEPENDENT ON ORBITAL INCLIN	TO S/S: DATA, COMMANDS, TV FROM S/S: DATA, TV
GPS	4	L-BAND	18500	2π UPPER HEMISPHERE PREDOMINANTLY SOME SLIGHTLY LOWER REQD	24 GPS SAT – 3 ORBITS 8/ORBIT. REQUIRE 4 SAT DATA W/O CLOCK SYNC S/S RECVS ONLY POSITION/VEL DETERMINATION

0663-078(T) V83-0165-196(T)

Fig. 3-22 RF Link Network Characteristics (Sheet 1 of 2)

S/S (TO-FROM)	MAX NO.	FREQ BANDS	MAX RANGE km	ANGULAR SECTOR COVERAGE	RF CHANNEL SERVICE REQUIREMENTS
FREE FLYER	5	S, K_u mm, LASER	2000	360° AZIMUTH 2 MILES ABOVE/BELOW SPACE STATION	• TO S/S: DATA, TV, NAV & RENDEZVOUS • FROM S/S: COMMANDS, TRACKING/RANGING • DOCKING
GDSTD	1	S	2000	2π LOWER HEMISPHERE	TO S/S: VOICE, COMMMAND FROM S/S: VOICE, TELEMETRY, DATA, TV
OTV	2	S, K_u mm, LASER	38000	2π PREDOMINANTLY UPPER HEMISPHERE SLIGHTLY LOWER MAYBE REQ'D	• TO S/S: DATA, TELEMETRY, TV, RENDEZVOUS • FROM S/S: COMMAND, DATA, TRACKING • DOCKING

0663-079(T) V83-0165-197(T)

Fig. 3-22 RF Link Network Characteristics (Sheet 2 of 2)

3.2.1.3 Design Guidelines - The major design guidelines are presented in Fig. 3-23. These guidelines are derived from a combination of present space operations, system design techniques, and space station requirements. The system operational requirements for RF LINK SIMOP, and independent subsystem operations in conjunction with autonomy inevitably lead to an all automated system/subsystem architecture with predominantly DMS interfaces. Therefore, each functional subsystem multiplicity/commonality requires consideration relative to SIMOP and providing insensitivity to mission models. The basic functional system architecture necessary is illustrated in Fig. 3-24. Both the DMS and Operations Control Center provide for two levels of operational mode selection as follows:

- Systems level
  - Selection of particular subsystem(s), antennas for angular sector
  - Selection of associated program(s) for link operations
  - Initial acquisition beam pointing position
- Subsystem level
  - Selection/activation of subsystem equipment/component elements
  - oper mode; channel(s), bandwidth(s), pwr transmission (low, med, high)
  - Selection/standby of backup subsystem equipments/components

In addition, the DMS provides for the overall processing (digitizing, multiplexing, formatting, switching, intended routing) of raw baseband data distributed to the Signal Interface Unit (SIU). The SIU could also contain elements pertaining to automatic tracking. The quantity of subsystems shown is relative to the particular link terminal. Antenna switching matrices are required for flexibility of SIMOP links and assigned angular sector coverage for each.

A subsystem functional architecture representation appears in Fig. 3-25. Redundancy is implemented by duplication of subsystem elements (100%); i.e., effectiveness of transfer switching common elements is an issue requiring further followon studies. A listing of some subsystem elements appears also in the figure. It is evident that redundancy by 100% duplication of equipments/components provides maximum power/mass impacts whereas common element transfer switching tends to lower these effects. Commonality is a much more extensive issue relative to standardization of functional subsystems and associated frequency allocations. This is also identified for further follow-on study effects.

3.2.1.4 Frequency Allocation & Management - Space Station SIMOP requirements, in conjunction with present and future frequency allocations, presented in Fig. 3-26,

- ALL LINKS DIGITAL MAX EXTENT
  - HIGH OPERATIONAL FLEXIBILITY
  - OPTIMUM TDRS CAPABILITY UTILIZATION
- SOLID STATE AMPLIFIERS MAX EXTENT
- 5-10 YEAR LIFE
- SPACE STATION ATTITUDE STABILITY ±2°
- ALL TIMING DERIVED FROM SPACE STATION CLOCK
- 100% REDUNDANCY CRITICAL ELEMENTS
  - MASS/VOLUME CONSTRAINTS
  - COMMON ELEMENT TRANSFER SWITCHING
- AUTOMATIC TRACKING TECHNIQUES (NARROW BEAM)
- ARCHITECTURE INSENSITIVE TO MISSION MODELS
- TV: EVA (MONITORING & RETURN), DOCKING
- FREQUENCY ALLOCATIONS: EXISTING (S, Ku, L, UHF)/FUTURE (mm, LASER)
- DMS
  - HIGH VOLUME DATA STORAGE: 1990/MISSION 10^{13} B/DAY-OPER 10^{10} B/DAY
2000/MISSION 2 x 10^{13} B/DAY-OPER 10^{1} B/DAY
  - AUTOMATED SUBSYSTEM OPERATION/MONITORING/CONTROL
  - COMPUTER CONTROL (A PRIORI PROGRAMMING) ACQUISITION, POINTING.

0663-080(T) V83-0165-198(T)

Fig. 3-23 Design Guidelines

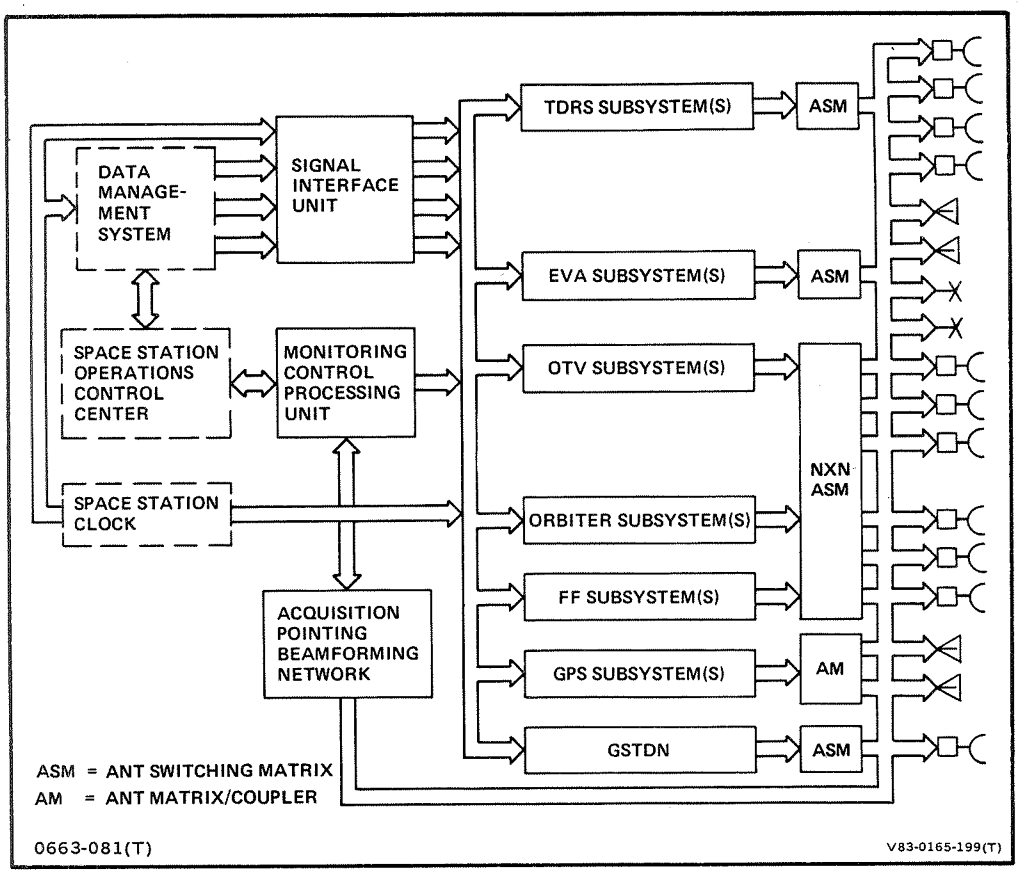


Fig. 3-24 System Architecture

IRAD

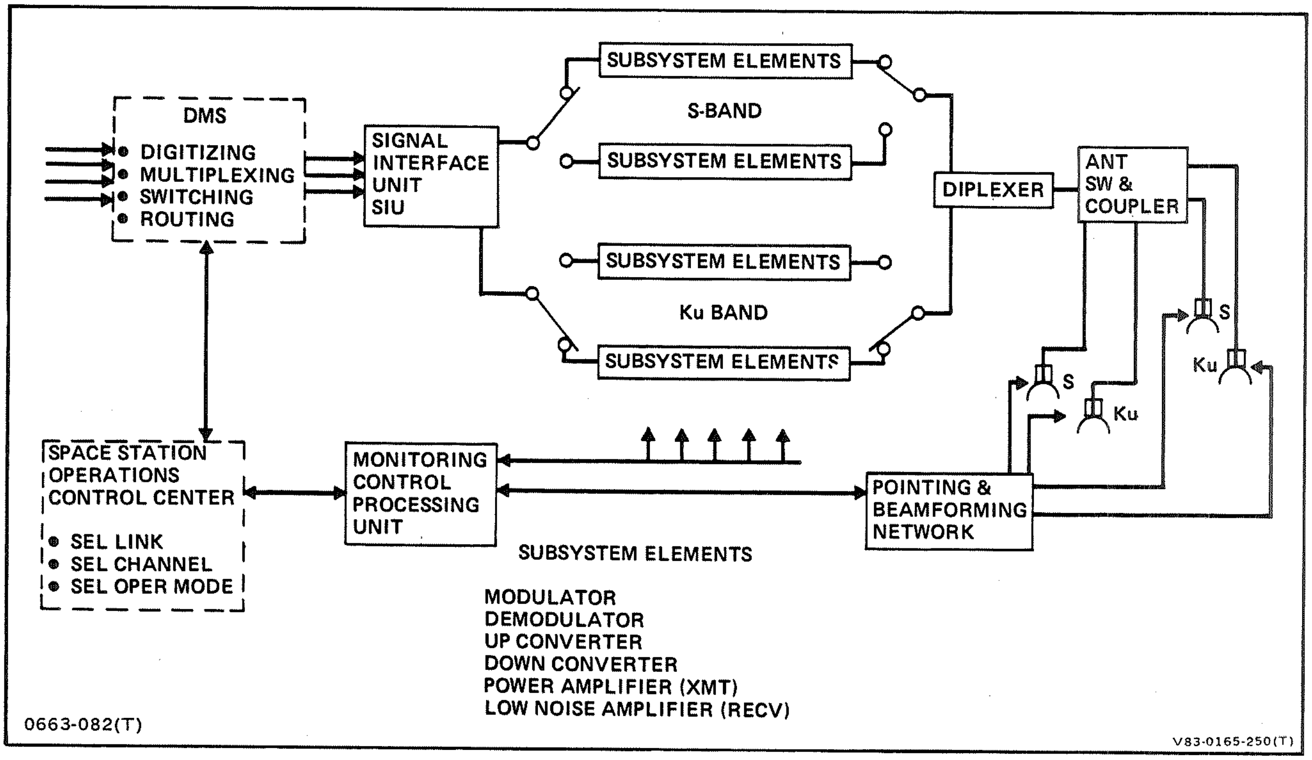


Fig. 3-25 Subsystem Functional Architecture

IRAD

GENERAL

GD-SAT
UPLINK 2025-2120 MHz
DOWNLINK 2200-2300 MHz
DEEP SPACE 2290-2300 MHz

TDRS
Ku BAND 13.4-14.2 GHz
14.5-15.35 GHz

S-BAND 2025-2120 MHz
2200-2300 MHz

1979 WARC INTERSATELLITE SERVICE ALLOCATIONS

FREQUENCY		BANDWIDTH (GHz)
BAND	ALLOCATION (GHz)	
K (18-26.5)	22.55 – 23.55	1
Ka (26.5-40)	32 – 33	1
U (40-60)	54.25 – 58.2	3.95
V (50-75)	59 – 64	5
F (90-140)	116 – 134	18
G (170-220)	170 – 182	12
G (170-220)	185 – 190	5

ORBITER

FUNCTION/SYSTEM	ORBITER TRANSMIT	ORBITER RECEIVE
STDN PM-1	2287.5 MHz	2106.406 MHz
STDN PM-2	2217.5 MHz	2041.947 MHz
AFSCF PM-1	2287.5 MHz	1831.787 MHz
AFSCF PM-2	2217.5 MHz	1775.733 MHz
STDN/AFSCF FM	2250.0 MHz	NONE
DFI FM	2205.0 MHz	NONE
NASA PAYLOADS (APPROXIMATELY 800 CHANNELS)	2025.833 TO 2118.722 MHz	2200.0 TO 2300.875 MHz
AF PAYLOADS (20 CHANNELS)	1763.721 TO 1839.795 MHz	2202.5 TO 2297.5 MHz
EVA COMMUNICATIONS	296.8 MHz OR 259.7 MHz	259.7 MHz OR 296.8 MHz AND 279.0 MHz
Ku-BAND COMMUNICATIONS	15.0034 GHz	13.7756 GHz
Ku-BAND RADAR ACTIVE MODE PASSIVE MODE	13.883 GHz CENTER - 13.883, 5 STEP FREQUENCY DIVERSITY OF 52 MHz PER STEP	13.883 GHz SAME AS TRANSMIT

0663-083(T)
V83-0165-251(T)

Fig. 3-26 Frequency Allocations/Uses

necessitate closer detailed scrutiny of significant frequency management limitations, deficiencies and pitfalls for the Space Station configuration beyond the initial (1990). It appears that, based on present identification of ISS SIMOP link requirements, frequency congestion for the initial (1990) configuration may not be severe. Since allocation is presently confined to narrow portions of S-band and Ku-band, and both NASA and DoD share the same frequency assignments, RFI can become especially critical.

The 1979 WARC additional satellite frequency allocations are viewed as alleviating the space operating frequency congestion by providing availability of the millimeter spectrum, especially for those Space Station links that will be new (e.g., OTV, GEO, TDAS). This use will depend on the cost-effectiveness of the systems to which the millimeter frequencies will be applied. In particular, the TDAS (future replacement for TDRS) will likely employ these millimeter frequencies and/or laser communications. Since this is presently in the planning stages, it is difficult to assess the implementation impact to the Space Station configuration beyond 1990 (i.e., initial Space Station-TDRS link in S- and Ku-bands).

The Space Station era, in conjunction with present technology transitions (e.g., transition of point-to-point microwave links to fiber optics), is a prime time for re-assessing frequency allocations and uses. In particular, the Space Station may be a first step to similar frequency-subsystem standardization as previously performed for terrestrial-based systems (e.g., TACAN, IFF, Loran, radio, ADF, etc). This, in conjunction with the inevitably of additional frequency band utilization, has led to a partial tradeoff study (viability and technical feasibility) and suggested follow-on studies, for frequency band sharing and system parameters involved (see Paragraph 3.2.1.9).

The new millimeter wave allocations will require large to full DDT&E costs and time for system functional applications that are to be designated for the Space Station. The inherent large bandwidths and higher data rates may not be actually needed for the OTV, Free Flyer, future GEO, satellites. RF link cost effectiveness will depend on the operational range distance required, antenna size (gain), receiver sensitivity, transmitted power, noise background, pointing accuracy/stability, data rate requirements. Millimeter wave equipments/components also have inherently higher manufacturing and material costs due to very tight dimensional tolerances and surface uniformity associated with the smaller wavelengths. The specific use of the 59 to 64 and/or 116 to 134 GHz frequency bands due to the high atmospheric attenuation, which thereby minimizes severe RFI with terrestrial terminals, may be limited to cost-effective application for earth-satellite links (i.e., self defeating). Space diversity using two or more earth-based terminals to offset inclement weather at site(s) requires new site construction. Frequency use for satellite-to-satellite links is offset by those RF link parameters previously cited. Space Station operational use of millimeter waves requires further evaluation and analysis as to cost effective application. An overview of frequency band advantages/disadvantages appears in Fig. 3-27.

3.2.1.5 Hardware - An assessment of present space qualified hardware status is provided in Fig. 3-28 for the higher frequency bands. It is evident that up to Ka-band equipments potentially fall into the following categories:

- Existing as is
- Existing to be modified (function, performance).

Therefore, equipment (design, development, and/or modification) is not viewed as a key issue or show-stopper for the initial configuration.

3.2.1.6 Antenna System - Antenna system considerations are summarized in Fig. 3-29. The Space Station configuration's inherent geometric complexity and dimensions prohibit a single antenna from providing 4π steradian coverage. In addition, multiple RF SIMOP links with varying degrees of structural blockage necessitates an antenna system selection criticality of antenna type, locations and quantity, in concert with overall Space Station operational modes. An antenna system based on angular sector coverage would be the most effective and appropriate. Omni-directional to low-gain antenna types are not a problem due to their variety, relatively small sizes and ability to provide discrete angular sector coverage.

High gain antennas conventionally are gimbal dish types which have significant limitations relative to the Space Station requirements. Although the actual limitations will be ultimately dependent on the final actual Space Station configuration, a first order evaluation of antenna quantity can be made relative to functional/ performance capabilities required. The approximation for maximum dish antenna quantity is provided by first applying one antenna for each RF link. This quantity is then increased by a factor determined from dividing each link's spherical solid-angle coverage requirement by the discrete solid angle coverage limitations (blockage, etc) of its antenna. For each antenna's designated angular sector there will be times when SIMOP occurs for which the dish antenna cannot accommodate

FREQUENCY	ADVANTAGES	DISADVANTAGES
UHF/VHF	• TECHNOLOGY WELL ESTABLISHED • LOW RISK, LOW COST	• FREQ CONGESTION • POTENTIAL RFI • LIMIT TO LOW RATE USERS
S-BAND	• TECHNOLOGY WELL ESTABLISHED • LOW RISK, LOW COST	• FREQ CONGESTION • POTENTIAL SEVERE RFI • LOW TO MODERATE RATES
Ku BAND	• TECHNOLOGY PROVED • RELATIVELY LOW COST, LOW RISK • HIGH RATE	• ADDITIONAL DEV COST (e.g. PWR AMPLIFIER) • POTENTIAL FREQ CONGESTION (LATE 1990) • ACQUISITION PROBLEM (NARROW BEAM)
Ka BAND	• TECHNOLOGY AVAILABLE • FREQ/BW HIGHLY AVAILABLE • HIGH TRANSMISSION RATE • RFI IMMUNITY (10-20TH HARMONIC)	• TECHNOLOGY ENABLING REQD COST & SOME RISK • INCREASED POINTING PROBLEM
W-BAND (60 GHz)	• HIGH TRANSMISSION RATE • FREQ/BW HIGHLY AVAILABLE • INCREASED RFI IMMUNITY (30TH HARMONIC)	• TECHNOLOGY R&D REQD • HIGHER COST & RISK • SEVERE POINTING PROBLEM
OPTICAL (LASER)	• VERY HIGH TRANSMISSION RATE • FREQ/BW HIGHLY AVAILABLE • NO RFI PROBLEM	• TECHNOLOGY R&D REQD • HIGH COST & RISK • SEVERE POINTING PROBLEM

0663-084(T) V83-0165-252(T)

Fig. 3-27 Frequency Bands Overview

HARDWARE	AVAILABILITY (VENDOR)	EXISTING OPERATIONAL SYSTEMS	COMMENT
GPS L-BAND RCVR & PROCESSER ASS'Y	OFF THE SHELF (MAGNAVOX)	LANDSAT-4	NO RISK, LOW COST
S-BAND ANTENNA & CHANNEL EQPT	OFF THE SHELF (COMMUNICATIONS INDUSTRIES IN GENERAL)	NASA SYSTEMS (ORBITER, LANSAT-4 TDRS, ETC)	NO RISK, LOW COST
Ku BAND ANTENNA & CHANNEL EQPT	COMMERCIALLY AVAILABLE OFF THE SHELF (COMMUNICATIONS INDUSTRIES IN GENERAL)	LANDSAT-4 ORBITER TDRS, ETC	LOW RISK, LOW TO MEDIUM COST, NEED WIDEBAND IMPROVEMENT
Ka BAND ANTENNA & CHANNEL EQPT	TECHNOLOGY AVAILABLE TECHNOLOGY ENABLING DEV REQD	• JAPANESE DOM-SAT • PLANNED SYSTEMS – ACTS – ITAL SAT	MEDIUM TO HIGH RISK & COST FOR WB APP-LICATIONS
W BAND ANTENNA & CHANNEL EQPT	TECHNOLOGY IN GENERAL AVAILABLE, R&D REQD FOR ENABLING	NO OPERATIONAL SATELLITE SYSTEM AT PRESENT	MEDIUM TO RELATIVELY HIGH RISK & COST
LASER XMTR RCVR ACQUISITION TRACKING	TECHNOLOGY IN GENERAL AVAIL-ABLE, SUBSTANTIAL R&D NEEDED FOR ENABLING TECHNOLOGIES	NO OPER SYSTEM EXPMNTL DEMO COMM SYSTEMS EXIST	MEDIUM TO HIGH RISK & COSTS, SYSTEM NEAR FUTURE REALIZATION NOT LIKELY FOR WB APPLICATION

0663-085(T) V83-0165-253(T)

Fig. 3-28 Hardware Status

- SPACE STATION CHARACTERISTICS
  - COMPLEX GEOMETRIC CONFIGURATION
  - VARYING DEGREES OF SHADOWING
  - RF LINK SIMOP
  - MULTIPLE FREQUENCIES
  - VARIABLE RANGES
- DISH ANTENNAS
  - ONE DISH PER LINK
  - EACH DISH 2π COVERAGE → TWO DISHES PER LINK
  - DISH BEAM POINTING LIMITED, ONE DIRECTION ONLY, MECHANICAL
  - SINGULAR BEAM TRACKING
  - REDUNDANCY
- SOLUTIONS
  - MULTI BEAM ARRAY ANTENNAS
  - ANTENNA SHARING/REDUCTION OF SIMOP
  - DUAL FREQ FEED FOR EACH DISH ANTENNA
  - PLANAR/CONFORMAL ARRAYS
  - ELECTRONIC BEAM POINTING
  - MULTI BEAM TRACKING

0663-086(T) V83-0165-253(T)

IRAD

Fig. 3-29 Antenna System Considerations

(i.e., more than one terminal point will exist in the sector). This situation is further compounded in view of the SIMOP links likely involving different frequencies. In addition, some RF link terminals inherently have variable operating ranges for which variable antenna gain needs are encountered. The aspect of redundancy must also be accounted for.

Solutions to offsetting dish antenna operational limitations and associated quantities can be provided by the following:

- o Multibeam array antennas - planar and/or conformal
- o A dual frequency feed for each dish antenna with a beam-forming network for each frequency at a common focal point
- o Antenna sharing with an associated reduction in SIMOP requirements.

This latter approach is least desirable. A dual frequency feed, particularly for S- and Ku-bands, appears especially appropriate for RF terminals involving both frequencies; e.g., TDRS, Orbiter with S-band communications and Ku-band radar, OTV and Free Flyers similar to Orbiter. The multibeam antenna array is especially appropriate for designated angular sectors and SIMOP in the sector. The possibility also exists of gimbal-mounting the array antenna for offsetting the limitations in off-axis beam-pointing characteristics. The conformal array is most applicable where it could be designed as a 360 deg cylindrical antenna providing complete azimuth coverage and limited elevation coverage. This approach appears most effective for a surveillance radar system comprised of integrating various sensors/detectors; multiple RF frequencies, IR, TV (optics). It would be unlikely to expect the evolved Space Station configuration to afford a space for such an array antenna; however, cost effectiveness for surveillance radar may lead to such space provisions (see Paragraph 3.2.2.2). Further information, alternative considerations and tradeoffs are provided as follows:

- o Fig. 3-30 Antenna Design Tradeoffs
- o Fig. 3-31 Antenna System Architecture Alternatives
- o Fig. 3-32 Alternative Architecture Assessment.

At present Multible Beam Array (MBA) technology lies predominantly in the military domain and, therefore, the associated MBA data in Fig. 3-30 is conservative. Radar systems incorporating MBA in conjunction with Direction Finding (DF) are operational, and more advanced systems are in the demonstration phase. These systems applications are mostly for ECM/ECCM on mobile and airborne vehicles. It is highly probable that Space Station application of such a system would

ASPECT	GIMBALLED PARABOLIC DISH	WITH GIMBALLED SCANNABLE SUB REFLECTOR	PHASED ARRAY ELEMENT ANTENNA (MBA)	PARABOLIC DISH WITH PHASED ARRAY FEED
GAIN	HIGH	MEDIUM TO HIGH	LOW TO MEDIUM	LOW TO MEDIUM
POINTING ACCURACY	HIGH	MEDIUM	LOW	LOW
DATA RATE	HIGH	MEDIUM	LOW	LOW
ACQ/TRK PWR RQT	HIGH	MEDIUM	LOW	LOW
ANT SLEWING TORQUE NOISE	HIGH	MEDIUM	NONE (HIGH IF GIMBALLED)	NONE (HIGH IF GIMBALLED)
COMPLEXITY	LOW	HIGH	MEDIUM TO HIGH	MEDIUM
RELIABILITY	HIGH	LOW	LOW	MEDIUM
COST	LOW	MEDIUM TO HIGH	HIGH	MEDIUM
SPACE STATION PREFERRED CHOICE	HIGH	LOW	LOW TO MEDIUM	HIGH V83-0165-255(T)

Fig. 3-30 Antenna Design Tradeoffs

OPTION I	SINGLE BEAM ANTENNA SYSTEM (WITH GIMBAL DRIVE & RF SWITCH) • 0.5m DISHES FOR BOTH S & Ku BANDS • OMNI, HEMI OR HORN ANTENNAS FOR S, Ku, L, UHF (CLOSE PROXIMITY).
OPTION II	PHASED ARRAY FEED MBA SYSTEMS (SECTOR COVERAGE) PLUS OMNI AND/OR HORN ANTENNAS, RF SWITCHES • PHASED ARRAY FEED MBA SYSTEMS FOR S-BAND • PHASED ARRAY FEED MBA SYSTEMS FOR Ku-BAND • OMNI, HEMI OR HORN ANTENNAS FOR S, Ku, L, UHF (CLOSE PROXIMITY).
OPTION III	MULTIPLE ACCESS (MA) ARRAY ANTENNA SYSTEM PLUS PARABOLIC DISHES & OMNI OR HORN ANTENNAS, RF SWITCHES • MA ARRAY ANTENNA SYSTEM FOR S-BAND • 0.5m DISHES FOR Ku BAND • OMNI, HEMI OR HORN ANTENNAS FOR S, Ku, L, UHF (CLOSE PROXIMITY).
OPTION IV	PHASED ARRAY FEED MBA SYSTEM (WITH GIMBAL DRIVE) PLUS PARABOLIC DISHES & OMNI OR HORN ANTENNAS, RF SWITCHES • PHASED ARRAY FEED MBA FOR S-BAND • 0.5m DISHES FOR BOTH S & Ku • OMNI, HEMI OR HORN ANTENNAS FOR S, Ku, L, UHF (CLOSE PROXIMITY).
NOTE: ALL OPTIONS INCLUDE 4m AND 0.5m DISHES FOR TDRS LINK *PARTICULAR EMPHASIS FOR INITIAL SPACE STATION V83-0165-245(T)	

Fig. 3-31 Antenna System Architecture Alternatives*

	NUMBER OF ANTENNAS REQD BY COVERAGE REQUIREMENT	COMMENTS
OPTION I, DISH WITH GIMBAL DRIVE	• 8 UNITS OF 0.5 m DISH (S & Ku) • 8–12 OMNI OR HORN ANTENNAS – CLOSE PROXIMITY (S, Ku, L, UHF).	• LOW COST • LIGHT WT • GOOD COVERAGE • TECHNOLOGY PROVED.
OPTION II, SECTOR COVERAGE	• 6 UNITS OF PHASED ARRAY FEED MBA SYSTEM FOR S-BAND • 2 UNITS OF PHASED ARRAY FEED MBA SYSTEM FOR Ku BAND • 8–12 OMNI OR HORN ANTENNAS FOR S, Ku, L, UHF.	• HIGH COST • HEAVY WT • OPTIMAL COVERAGE • TECHNOLOGY AVAILABLE • DEV REQD FOR SPACE STATION.
OPTION III, MA ANTENNA SYSTEM	• 6 UNITS OF MA ARRAY ANTENNAS FOR S-BAND • 4 UNITS OF 0.5 m DISHES FOR Ku BAND • 8–12 OMNI OR HORN ANTENNAS FOR S, Ku, L, UHF.	• HIGH COST • HEAVY WEIGHT • OPTIMAL COVERAGE • TECHNOLOGY AVAILABLE • SPACE OPER SYSTEM IN TEST (TDRS).
OPTION IV	• 2 UNITS OF PHASED ARRAY FEED MBA SYSTEM FOR S-BAND • 6 UNITS OF 0.5 m DISHES FOR S & Ku BANDS • 8–12 OMNI OR HORN ANTENNAS FOR S, Ku, L, UHF.	• MEDIUM COST • MEDIUM WEIGHT • VERY GOOD COVERAGE • TECHNOLOGY AVAILABLE • DEV REQD FOR SPACE OPERATIONS.
*PARTICULAR EMPHASIS FOR INITIAL SPACE STATION		V83-0165-246(T)

Fig. 3-32 Alternate Architecture Assessment*

involve moderate costs in relation to functional modifications, not development. The MBA antenna itself is a passive item and can be modified for a different operational frequency by frequency scaling. Active components based on frequency would require replacement. The signal processing technique could be retained, modified or replaced. Further investigation and study in this area has been recommended; see item (f) in Subsection 3.2.1.8.

3.2.1.7 Key Issues - The previous discussions have provided a basis for the following derived key issues:

- 4π coverage
- Frequency allocation and management
- RFI
- Commonality - system and subsystem level
- Millimeter/laser link assignments
- Future TDAS configuration
- Subsystem mass/power restrictions
- Redundancy vs common element transfer switching
- Gimbal dish antenna vs multibeam array
- Substantial use of TDRS KSA beam for extended time intervals; only two beams available
- Technology/state-of-the-art advances forthcoming vs Space Station initiation/acceleration of advances
- Hardware modifications relative to commonality doctrine to be established.

These key issues by their nature will not be resolved conclusively in a short period of time. This is particularly so due to their relationship/dependency on future events and/or interrelations with each other.

3.2.1.8 Recommended Additional and/or Follow-on Studies/Investigations - The following additional and/or follow-on studies/investigations are recommended on the basis of their impact to key issues and their derivation from various portions of this study, especially the above communications/tracking tradeoffs and analyses.

a) 28.5 deg Space Station as a communications node for other satellites
b) Space Station subsystem functional frequency standardization and related impacts to existing technology/equipments
c) Millimeter/laser link assignments in conjunction with TDAS configuration and related impacts to Space Station subsystems/elements

d) System/subsystem levels of commonality in conjunction with items b) and c) above and relative impacts to redundancy vs common element transfer switching

e) Review, evaluation and analyses of existing space and terrestrial (earth-based and airborne) systems for cost effectiveness applicability to Space Station operations. This will include potential frequency sharing and inter-system operational characteristics between the Space Station and terrestrial system (refer to Subsection 3.2.1.9)

f) Review, evaluation and analyses of existing array antenna systems and performance characteristics, relative to Space Station applicability and associated modifications/costs

g) Surveillance radar system operational objectives, criteria, type sensors/detectors and required system architecture. This involves an iteration process with item (f) above

h) Frequency allocation and associated levels of RFI susceptibility (in band/ out-of-band). This will include the Space Station as an RFI SINK and present RFI vulnerability of existing operational spacecraft.

3.2.1.9 **Frequency Sharing** - With proper engineering there is significant potential applicability of existing terrestrial (ground-based and airborne) systems to Space Station operations. In addition existing terrestrial system technology transitions, such as fixed point microwave links to fiber optics, make particular frequencies more readily available for sharing. The basic Space Station application categories would be:

- Same system frequency and function
  - Modified hardware for space qualification
  - Modified hardware for function and/or performance
- Same frequency
  - New function and new hardware.

Frequency sharing does **not** inhibit new technology development since new technology will be used for a new system application of the frequency (e.g., array antenna replacing gimbaled dish antenna).

Frequency Sharing Viability - Frequency sharing is recognized now as non-avoidable and a "must" internationally. Viability considerations are summarized in Fig. 3-33. There are existing procedures participated in by the OTP, ITU, FCC, NTIA for reviewing, evaluating and deciding on proposed system frequency sharing. These are summarized in Fig. 3-34. Such proposals have been previously invoked and are presently in process (e.g., DSCS and JTIDS). The time interval for this review process is three to six years and therefore a recommended additional followon study must be continued now.

There is presently a technology transition from earth-based point-to-point microwave links to fiber optic cables. The fiber optic signal (light) may be directly modulated (e.g., digital encoding and pulses) or may have the microwave signal modulated onto the light signal. This latter approach inherently immunizes the microwave signal from RFI (electrical interference), thereby increasing the viability and technical feasibility of Space Station frequency sharing for these particular microwave frequencies.

Technical Feasibility - Frequency sharing feasibility issues are summarized in Fig. 3-35. The primary concern for frequency sharing is RFI between the Space Station and terrestrial systems. There are a number of points favoring frequency sharing for the Space Station. These points will vary depending on the particular terrestrial frequency band and present system application.

General system operational characteristics supporting frequency sharing are: range and associated space loss; potential for avoiding (predesign planning) of antenna main-beam-to-antenna-main-beam line of sight (LOS); varying finite field-of-view (FOV) time; and intended (by design) Space Station link operational angular sector zone. In addition, further support is provided by new technology such as: signal modulation and coding; EMC techniques; antenna pattern nulling; and antenna beam shaping.

The FOV between the Space Station and earth/airborne based systems will be intermittent and time-limited. For fixed earth-based systems, the FOV periodicity will be dependent on the Space Station altitude and inclination. For mobile/airborne systems, the FOV is of a probabilistic nature and the intermittent time intervals will be of a random characteristic.

Preliminary design criteria and approach for Space Station utilization of the shared frequency would be the following:

- Space Station RF links to lateral terminals or higher altitude terminals. Space Station has high gain antenna, terminal has low gain antenna

VIABILITY

- UNAVOIDABLE & A "MUST" INTERNATIONALLY
- EXISTING PROCEDURES FOR NEW PROPOSED SYSTEMS REVIEW/EVALUATION
  - JTIDS
  - DSCS; AN/MSC-60, MSC-59
- TECHNOLOGY TRANSITION
  - FIBRE OPTICS REPLACING TERRESTRIAL POINT-TO-POINT MICROWAVE LINKS
- VARIETY OF TERRESTRIAL FREQUENCIES-SYSTEMS WITH SPACE STATION APPLICABILITY CRITERIA.

0663-087(T)

V83-0165-247(T)

IRAD

Fig. 3-33 Frequency Sharing Viability

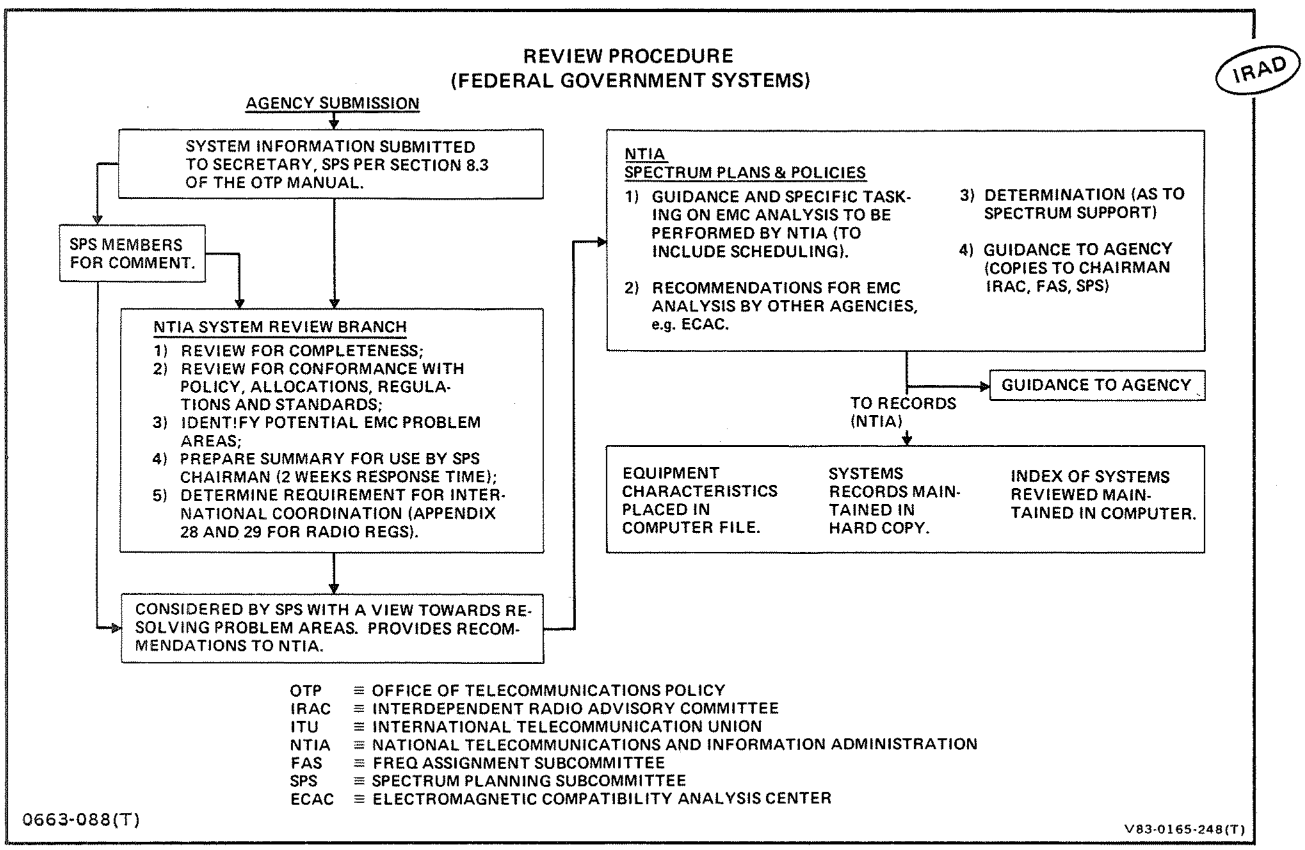


Fig. 3-34 Frequency Sharing Review Procedure

- PRIMARY CONCERN – RFI
- POINTS FAVORING FREQUENCY SHARING
  - RANGE & ASSOCIATED SPACE LOSS
  - PREDESIGN TO MINIMIZE ANT.-TO-ANT. LOS GAIN
  - INTERMITTENT FIELD OF VIEW (FOV)
  - PREDESIGN SPACE STATION LINK OPERATIONAL ZONE
  - TECHNOLOGY PROGRESS: SIGNAL MODULATION &/OR CODING, EMC TECHNIQUES, ANTENNA PATTERN NULLING, ANTENNA BEAM SHAPING
- DESIGN CRITERIA/APPROACH
  - SPACE STATION HIGH GAIN ANTENNA, TERMINAL LOW GAIN ANTENNA FOR LATERAL POSITION/HIGHER ALTITUDE TERMINALS (OTV, MOTV, FF, GEO SAT)
  - SPACE STATION LOW GAIN, TERMINAL HIGH GAIN FOR LOWER ALTITUDE TERMINALS
  - PREDICTABLE FOV TIME LINE INTERMITTENCY/PERIODICITY FOR FIXED EARTH BASE SYSTEMS
  - PROBABILISTIC FOV TIME INTERVALS FOR MOBILE/AIRBORNE SYSTEMS.

0663-089(T) V83-0165-249(T)

IRAD

Fig. 3-35 Frequency Sharing Technical Feasibility

- Space Station RF links towards earth (lower altitude terminals) tentatively would place high gain antenna on terminal and low gain antenna on Space Station
- Space Station orbital periodicity (for present study nominally 90 minutes) and associated fixed earth-based stations in LOS
- Determination of susceptibility level (RFI)
- Performance parameters and system operational characteristics of terrestrial systems.

There are a number of earth-based/airborne systems that may be applicable to Space Station frequency sharing and possibly direct/modified systems utilization for Space Station operations. Only one example is provided in the following discussion.

Weather Radar Application - Airborne weather radar operates at a frequency of 9.345 GHz. This system may be applicable for Space Station surveillance of debris and would appear to be directly usable for debris density (i.e., airborne system operation relative to moisture/precipitation density). Potential Space Station use, therefore, may be for one of the following:

- Frequency only - new system and function for Space Station
- Frequency and airborne system directly modified for space operation
- Frequency and modified airborne system (functionally) plus modification for space operation.

Characteristics of the airborne system are as shown in Fig. 3-36. First level identification of system parameters are illustrated in Fig. 3-37. A top level susceptibility determination can be made from the data presented. The RF level to the aircraft (excluding Space Station parameters) is approximately 50 dB below the aircraft receiver sensitivity level; i.e., space loss, aircraft antenna gain - receiver sensitivity. Since the Space Station utilization may be for space debris (forward and AFT Looking antennas) surveillance, only a sidelobe would be viewing the aircraft. It is not likely the sidelobe ERP would equal 50 dB. Conversely, if the Space Station receiver sensitivity were -100 dBm, the aircraft signal would be approximately 11 dB below this sensitivity. It is evident this susceptibility analysis requires further detailed refinement; however, this would also include on the Space Station side those techniques (EMC, antenna pattern nulling, antenna beam shaping, etc) that will be used to minimize RFI susceptibility, and the sidelobe ERP level viewing the aircraft.

3.2.2 ISS/ESS Subsystem Configurations

A summary of the communications/tracking subsystem architectures for the initial (ISS) and evolved (ESS) Space Station configurations is provided in the following discussions. Basically the subsystem architecture is that previously discussed for Fig. 3-24 and 3-25.

3.2.2.1 Initial Space Station Configuration - The communications/tracking subsystem approach for the initial Space Station configuration appears as follows:

- Fig. 3-38 RF Link Operational Requirements
- Fig. 3-39 RF Link Performance Requirements Summary
- Fig. 3-40 Subsystem Component Type Listing
- Fig. 3-41 Orbiter/Free Flyer Antenna System
- Fig. 3-42 TDRS Link Antenna System
- Fig. 3-43 EVA/GPS/GSTDN Antennas.

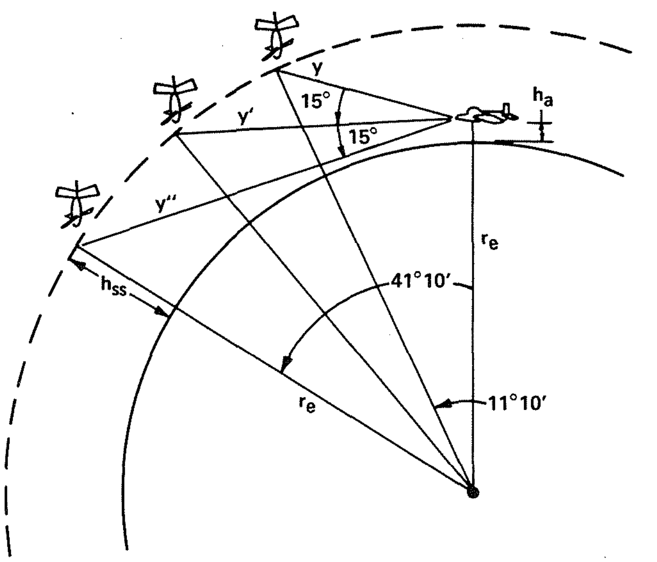
A redundancy of 100% is provided for critical subsystem elements. Frequency bands are S, Ku and UHF in accordance with present space operations. The orbiter/free flyer antenna system is compatible with their similarity of RF link performance requirements. The dish antennas are arranged to favor the lower hemisphere, where operations will predominate. The antennas located on the surrogate will provide elevation coverage and are in the same plane as the solar panel to off-

- AIRBORNE WEATHER RADAR
  - FREQ = 9.345 GHz, PULSED
  - PEAK POWER 10 kW, AVG POWER 4-5W
  - ANT. GAIN 31.2 dB (18" DIA SLOT ARRAY)
  - RANGE 250 MILES
  - AUTOMATIC ANTENNA SCAN IN AZIMUTH
    - WIDEBAND 120° (±60°)
    - NARROWBAND 60° (±30°)
  - ELEVATION AUTOMATICALLY MAINTAINED TO A/C HEADING MANUALLY TILTABLE ±15°
  - RECEIVER SENSITIVITY −98 dBm
- SPACE STATION APPLICATION
  - FREQ ONLY, NEW SYSTEM
  - FREQ & SYSTEM FOR SPACE DEBRIS SURVEILLANCE
  - FREQ & MODIFIED SYSTEM FUNCTION
  - MOD FOR SPACE OPER REQD.

0663-090(T) V83-0165-313(T)

IRAD

Fig. 3-36 Airborne Weather Radar Application – Frequency Sharing



ORBITAL PERIOD ≈ 90 MIN
CIRCULAR ORBIT - 200 MILES
r_e = 3953 MILES
h_a = 6 MILES

y = SHORTEST DISTANCE ≈ 1375 Km
SPACE LOSS ≈ 180 dB

y'' = MAX DISTANCE ≈ 4674 Km
SPACE LOSS ≈ 188 dB

FOV TIME = 7.5 MIN

NOTES:
1. AIRCRAFT IN ORBITAL PLANE; 2-DIMENSIONAL REPRESENTING MINIMUM DISTANCES
2. SPACE STATION SPACE DEBRIS SURVEILLANCE RADAR (FWD & AFT) ANTENNAS HAVE SIDELOBE FOV TO AIRCRAFT

0663-091(T) V83-0165-265(T)

IRAD

Fig. 3-37 Airborne Weather Radar Top Level Inter-System Parameters

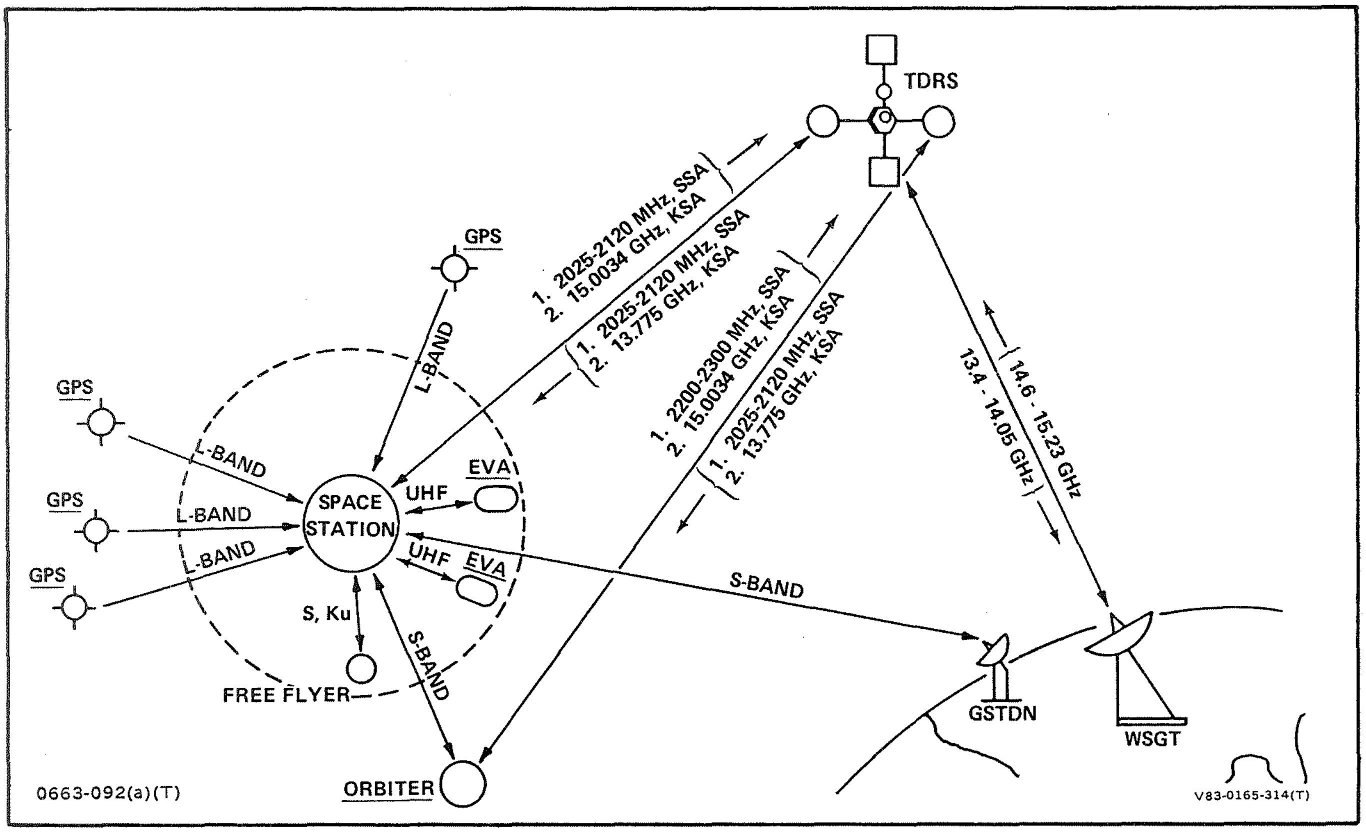


Fig. 3-38 RF Link Operational Requirements

LINK T F S/S		SERVICES	FREQ BAND	NO. SIMOP	MAX RANGE (km)	ANGULAR COVERAGE	CHARACTERISTICS
TDRS (RELAY)	TO	COMMAND DATA	S	1	38,000	2π UPPER	• S/S & TDRS MODERATE ACQUISITION • NAV EPHEMERIS DATA • LARGE DATA VOLUME
	FROM	DATA/TV	Ku				
GPS	TO ONLY	NAVIGATION DATA POSITION DETERMINATION & VELOCITY	L	1-4	18,500	2π UPPER	24 GPS (3 ORBITS, 8/ORBIT) REQUIRE 4 GPS DATA WITHOUT CLOCK SYNC
FREE FLYER	TO	TELEMETRY, MISSION DATA, TV, NAV DATA	S, Ku	1	2000	AZIMUTH SMALL UPPER & LOWER	• DOCKABLE TO S/S • RANGE VARIABLE
	FROM	COMMAND, TRACKING RENDEZVOUS	S, Ku				

0663-092(b)(T) V83-0165-315(1/2)(T)

Fig. 3-39 RF Link Performance Requirements Summary — (Sheet 1 of 2)

LINK T→/←F S/S		SERVICES	FREQ BAND	NO. SIMOP	MAX RANGE (km)	ANGULAR COVERAGE	CHARACTERISTICS
ORBITER	TO	VOICE, DATA, RENDEZ-VOUS, TRACKING	S, Ku	1	2000	2π LOWER	• PRE DETERMINED TRAJECTORY • MODERATE ACQUI-SITION • DOCKABLE TO S/S
	FROM	VOICE, DATA, TRACKING	S, Ku				
EVA	TO	VOICE, DATA, TV (NB, SLOW SCAN ANALOG)	UHF	2	8-10	4π	• CONTINUOUS COVER-AGE REQD • 2π COVERAGE CLOSE PROXIMITY • DISCRETE ANGULAR SECTOR LONG RANGE
	FROM	VOICE, DATA	UHF				
GSTDN (BACKUP)	TO	VOICE, COMMAND	S	1	2000	2π LOWER	• PREPROGRAMMED LOS & TIMING INTERVALS ON S/S • MAJOR LINK PRIOR TO S/S OPER VERIFICATION
	FROM	VOICE, TELEMETRY, DATA, TV	S				

0663-093(T) V83-0165-315(2/2)(T)

Fig. 3-39 RF Link Performance Requirements Summary (Sheet 2 of 2)

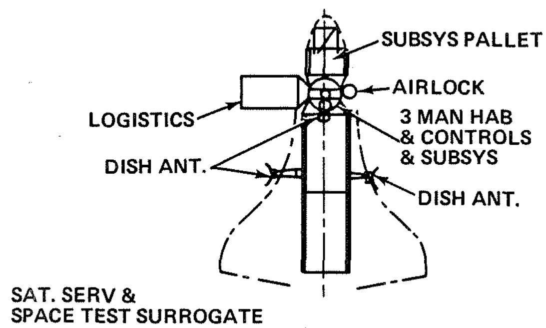
- ANTENNAS
- FEED HORNS & BEAM FORMING NETWORKS
- RF SWITCHES
- LOW NOISE RECEIVER AMPLIFIER
- TRANSMIT POWER AMPLIFIER
- UP/DOWN CONVERTERS
- POINTING & CONTROL MECHANISM (GIMBALS)
- PROCESSOR FOR POINTING & CONTROL
- MODULATOR/DEMODULATOR (MODEM)
- SIGNAL INTERFACE UNIT
- FORWARD ERROR CORRECTION (FEC)
- MONITOR & CONTROL
- INTERFERENCE CONTROL
- DOPPLER PROCESSING.

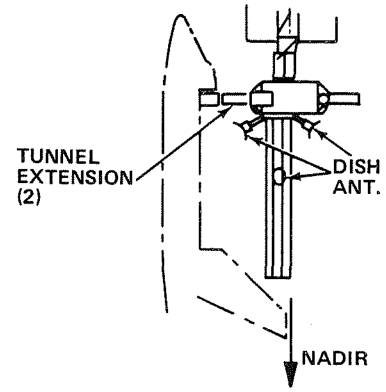
0663-094(T) V83-0165-316(T)

Fig. 3-40 Subsystem Component-Type Listing

- ORBITER/FREE FLYER SIMILAR CHARACTERISTICS
  - VARIABLE RANGE 0-2000 km
  - RENDEZVOUS, DOCKING
  - FREE FLYER 360° AZIMUTH, ±2 MILES VERTICAL DISTANCE
  - ORBITER LOWER HEMISPHERE PREDOMINANTLY
- FOUR DISHES
  - EACH DISH Ku & S-BAND
  - 0.5m DIAMETER
  - S-BAND 20° BW
  - Ku BAND 3° BW
- FOUR OMNI ANTENNAS (NOT SHOWN)
  - SHORT RANGE OPERATION
  - 2 S-BAND, 2-Ku BAND

IRAD





NOTES: *REFER TO PART III FOR FURTHER DETAILS.
NOT DRAWN TO SCALE.

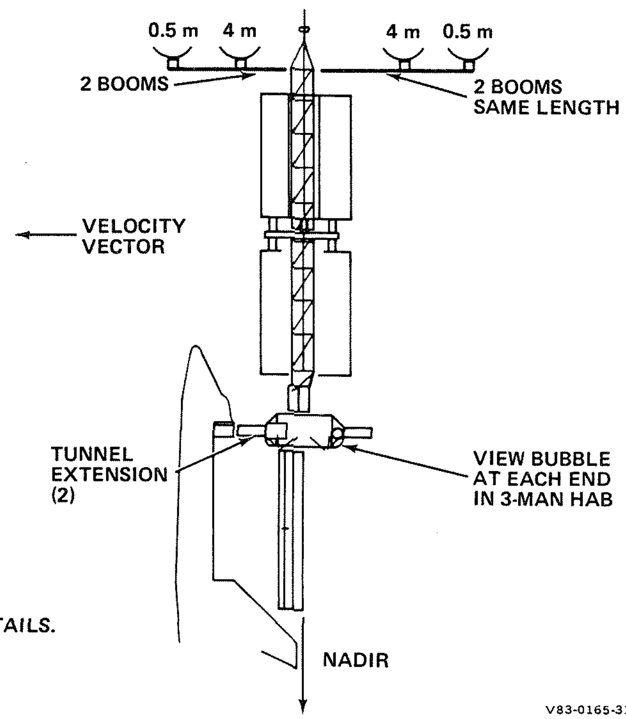
0663-095(T)

V83-0165-317(T)

Fig. 3-41 Orbiter/Free Flyer Antenna System*

- EACH DISH Ku & S-BAND
  - SEPARATE BEAM FORMING NETWORK
- 0.5 METER DIAMETER DISH–ACQUISITION
  - S-BAND 20° BW
  - Ku 3° BW
- 4 METER DIAMETER–COMMUNICATIONS
  - S-BAND 2.5° BW
  - Ku 0.4° BW
- TWO SETS OF ANTENNAS
  - BACKUP
  - MAINTAIN LINK CONTINUITY WHEN SWITCHING ONE TDRS TO OTHER TDRS
- AUTOMATIC TRACKING AFTER ACQUISITION.

IRAD



NOTES:
* REFER TO PART III FOR FURTHER DETAILS.
NOT DRAWN TO SCALE.

0663-096(T)

V83-0165-318(T)

Fig. 3-42 Initial TDRS Link Antenna System* set (minimize) blockage from docked vehicles (orbiter or free flyer). Docking operations and maneuvering will occur for only one vehicle at a time. The extent of blockage for the four dish antennas will be dependent on the boom length for each.

Requirements for closed circuit TV (CCTV) EVA monitoring is minimal due to the viewing port windows on the manned core module. EVA UHF communications antennas will be omni types (dipoles, crossed dipoles, etc) for close proximity operations within the surrogate area, or above and about the subsystem pallet. For ranges of 2- to 10-km EVA operations will be confined to a particular angular sector which may be accommodated by horn antenna(s) mounted on gimbals. The horn antenna pattern angular coverage is well controlled by design (geometry, dimensions). In addition, the use of ridged horn techniques provides for a 35 to 50% reduction in size relative to conventional horn design.

The initial Space Station configuration represents a minimum RF link topology. Consideration was given to extending the support structure above the subsystem pallet to provide a 360 deg azimuth cylindrical region. Since this would be more effective for the ESS configuration (i.e., increased RF link topology, surveillance radar), the additional weight and costs did not appear justified for the initial Space Station configuration. This approach is discussed in Subsection 3.2.2.2 below.

- EVA
  - MINIMUM TWO EVA SIMULTANEOUSLY
  - 0-10 km RANGE, UHF BAND
  - EVA ACTIVITIES PREDOMINANTLY LOWER (SURROGATE) AREA
  - MINIMUM FOUR OMNI ANTENNAS
  - LOW RF POWER
- GPS
  - RECEIVE FUNCTION ONLY
  - L-BAND
  - MINIMUM TWO OMNI ANTENNAS
  - MINIMIZE STRUCTURAL BLOCKAGE LOCATIONS
  - LOCATED UPPER PORTION OF SPACE STATION
- GSTDN
  - USE HORN ANTENNA OR ORBITER/FF DISH ANTENNA (SHARED OPERATION)
  - S-BAND.

*REFER TO PART III FOR FURTHER DETAILS.

0663-097(T) V83-0165-319(T)

IRAD

Fig. 3-43 EVA/GPS/GSTDN Antennas*

3.2.2.2 **Increased Structure - Complete Azimuth Coverage** - The support structure above the subsystem pallet affords an excellent opportunity to provide 360 deg azimuth coverage with limited elevation dependent on the extent to which the structure length is increased (refer to Fig. 3-44 and 3-45). This angular coverage is especially appropriate for the ESS configuration where maximum RF link topology will be encountered in addition to a sophisticated surveillance radar system. An issue arises as to whether the cost/weight/effort is greater for early implementation (ISS configuration) or later implementation (ESS configuration). Docked vehicle blockage would be limited for short time intervals for moderate increased structure lengths. Higher length increases could provide minimum to no significant blockage. Figure 3-44 and 3-45 illustrate the elevation angles associated for a 30-ft increase in support structure length, for two orthogonal views of the Space Station configuration. Elevation angles are shown for a position located at the center of the structure, and at the end of a 20-ft boom from the structure center. Although these data are for only two possible conditions, it is evident that a combination of increased height and boom could reduce antenna operational blockage to almost zero. Another consideration is the antenna beam axis vertical range as a function of elevation angle and horizontal distance from the Space Station. These data are provided in Fig. 3-46.

The array to be placed around the increased support structure could have various shapes, and either be adjacent to the structure or offset by additional structure. These possibilities, in addition to a comparison of dish vs multibeam array, are illustrated in Fig. 3-47.

3.2.2.3 **Tended Polar Platform** - The TPP communications link will be predominantly via TDRS as a relay due to its inherent line-of-sight (LOS) limitations to both ground stations and the Space Station. The subsystem capability requirements of the platform should be available as duplicators of the applicable main Space Station subsystems. If a direct RF link between the platform and Space Station is required, it could readily be accommodated for. The platform LOS time intervals and associated periodicity to the Space Station are determinable for pre-programming. In addition, navigation data are available for updating this programming.

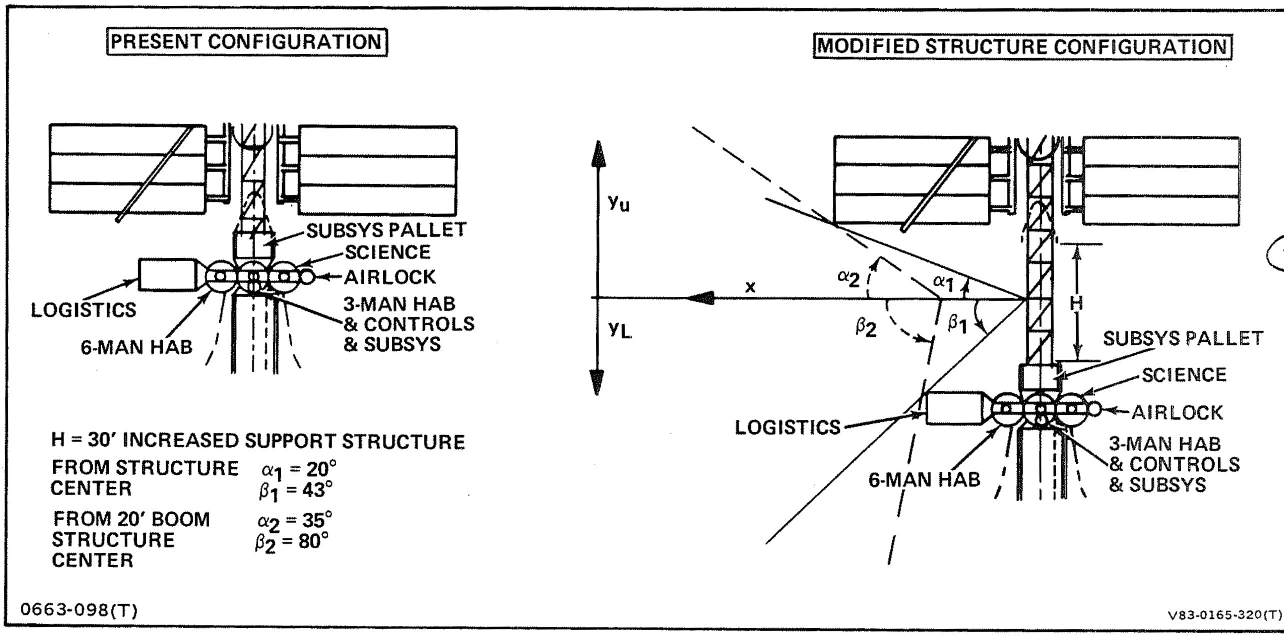


Fig. 3-44 Communications/Tracking Increased Structure, Complete Azimuth Coverage – View I

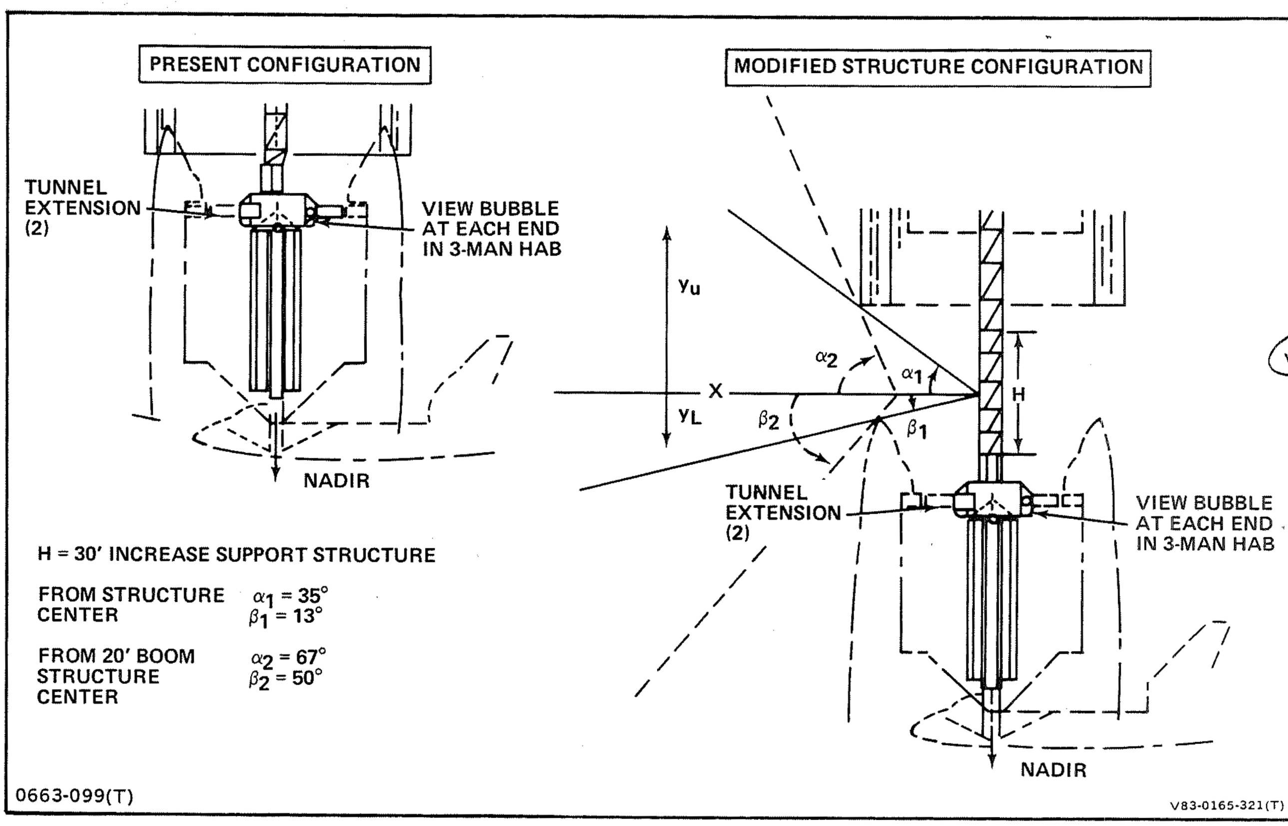


Fig. 3-45 Communications/Tracking Increased Structure, Complete Azimuth Coverage – View II

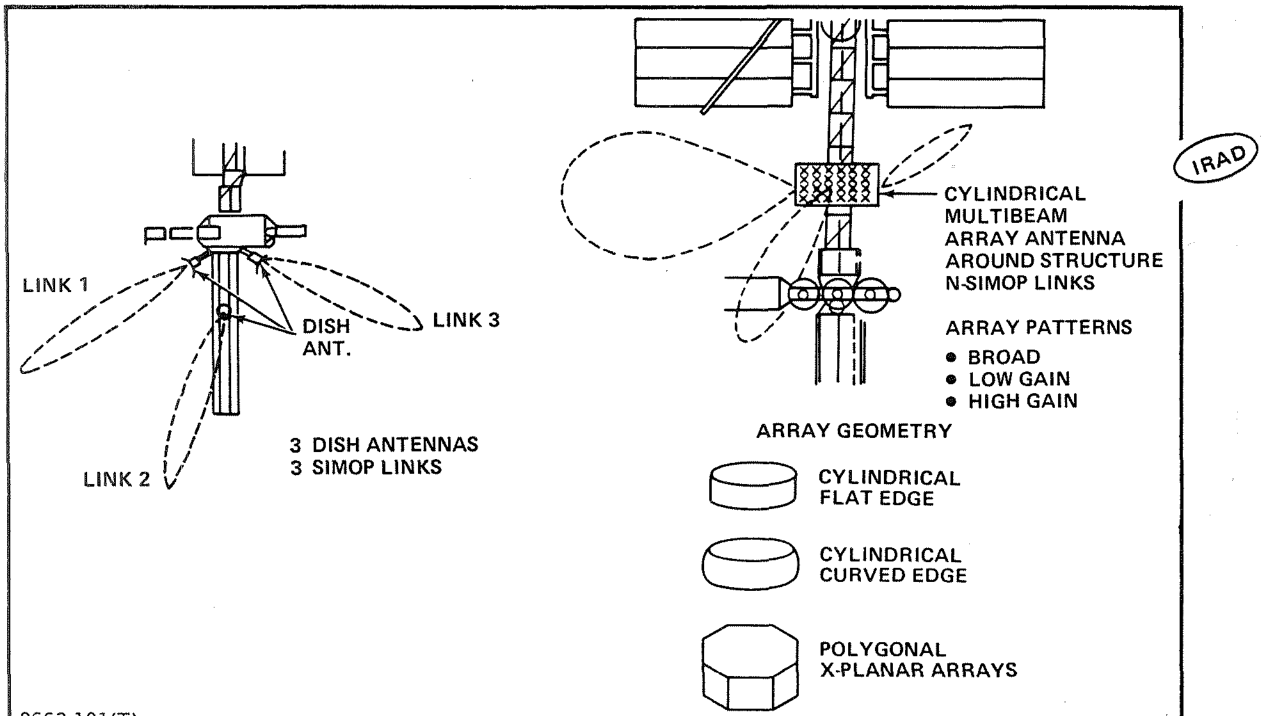
	X = 10 km				X = 200 km				X = 3000 km			
	α_1		α_2		α_1		α_2		α_1		α_2	
y_u, km	20°	35°	35°	67°	20°	35°	35°	67°	20°	35°	35°	67°
	3.64	7	7	23.6	72.8	140	140	472	10.92 (10^2)	21 (10^2)	21 (10^2)	70.8 (10^2)

IRAD

	X = 10 km				X = 200 km				X = 3000 km			
	β_1		β_2		β_1		β_2		β_1		β_2	
y_L, km	13°	43°	50°	80°	13°	43°	50°	80°	13°	43°	50°	80°
	2.31	9.33	11.9	56.7	46.2	186.6	238	1134	6.93 (10^2)	27.99 (10^2)	35.7 (10^2)	170.1 (10^2)

0663-100(T) V83-0165-322(T)

Fig. 3-46 Communications/Tracking Increased Structure y_u, y_L = f (α, β, χ)



0663-101(T) V83-0165-396(T)

Fig. 3-47 Communications/Tracking Increased Structure Complete Azimuth Array Configuration

3.2.2.4 ESS Configuration - The ESS communications and tracking architecture configuration will essentially be an expansion of the ISS architecture to include additional subsystems (i.e., increased RF links and associated SIMOP requirements). It is expected that multibeam arrays will be employed in place of earlier dish and omni antennas. In particular, the complete azimuth coverage approach (Subsection 3.2.2.2) will likely be implemented.

3.2.3 Subsystem Evolution

Subsystem evolution aspects require further distinction of technology transparency to two areas as follows:

- Functional capability and capacity
- Technology design base and performance parameters.

It is evident that the ISS Communications and Tracking Subsystem architecture exhibits complete transparency for the former area (i.e., requirement for additional SIMOP RF links accommodated by additional independent subsystems). The latter transparency area can be misleading and may result in significant future space station implementation impacts unless proper engineering precautions are taken early. This arises essentially relative to the need for compatibility in system design/integration commonality features, technology advances, and future NASA programs.

The addition of independent subsystems basically is providing increased capability and performance enhancement with supplementary equipments and/or replacements. Therefore, system integration commonality features and doctrine should be developed and established prior to the ISS so that growth provisions do not inherently involve severe time/cost implementation impacts (i.e., avoid unique interface tailoring). This is particularly true for the communications and tracking subsystem wherein technology advances will impact the equipment designs (i.e., operational frequency, power requirements, signal modulation/coding techniques, etc). Technology advance forecasts are:

- Antennas: Multibeam (MBA)
  - Phased array
  - Lenses

- Feed horns and beam forming networks:
  - Variable power divider
  - Variable phase control
  - Monolithic microwave integrated circuits (MMIC)
- Switches
  - For combining antenna systems
  - For beam switching (tracking purposes)
- Low-noise amplifiers: GaAs FET
- Power amplifiers: TWTAs, GaAs FETs
- Modulation techniques: QPSK, OQPSK, BPSK, MSK, M-ARY PSK, Coded Phase Modulation
- Interference control techniques
- Doppler control technique
- Processors: LSI, VLSI, on-board data processor, etc.
- Development of higher frequency bands (Ka-band or W-band).
- Development of optical communications links
- Development of flexible modulation techniques characterized by:
  - bandwidth and power efficiency
  - robustness
  - insensitivity to enviornmental interference and doppler shifts
- Space station application of multiple access techniques.

Establishing subsystem functional frequency standardization early (prior to 1990) will help reduce long-range improvement uncertainty. In particular, this must include future projected TDRS capabilities.

3.2.3.1 Advanced TDRS Capabilities - Compatibility with Evolution to the Year 2000* - The Tracking and Data Acquisitions System (TDAS), which is the planned advanced TDRSS, is an outgrowth of TDRSS having greater data rate capacity (gigabits/second), increased capability and direct downlink from a TDAS relay satellite to user ground terminal. The planned TDAS satellite will possess new or enhanced capabilities such as on-board data processing, on-board intelligent switches, sophisticated antennas, reliable high power amplifier at S-band, Ku-band, Ka-band, W-band, and laser systems, etc.

---

*The TDRSS evolution is based on the study results of Tracking and Data Acquisition System, by STI, Inc. and GSFC/NASA.

The Ka-band, W-band and/or laser are candidate frequencies for cross-link between two TDAS spacecraft. S-, Ku-, W-band and/or laser are considered for the inter-satellite link between TDAS and user (low orbiting) spacecraft. Ku-band and Ka-band are the most likely choices for the TDAS space-ground link. The TDAS spacecraft architecture, constellation, user RF interface and operational interface are to be defined with respect to frequency used, end-to-end communications connectivity, and user service and performance requirements.

This evolution plan is based on the projected needs of the user community coupled with projected advances. Some planned improvements and enhancements of TDRSS for incorporation into TDAS in 1990 are highlighted below:

- Improve the TDRSS coverage capability for low orbiting users to 100% communication coverage
- Increase TDRS return data rate capacity up to 1 GBPS (at least 500 Mbps as required by space station requirements)
- Consider laser or W-band for TDRS - Space Station high rate communications
- Add more single access channels to each TDRS so as to adequately support medium and high rate users real-time communications
- Improve TDRS/user satellite acquisition and auto-tracking capacity
- Provide high data rate direct downlink (Ka-band) to a ground terminal dedicated to the space station
- Introduce new technologies including: bandwidth efficient moduiation techniques; data compression, advanced navigation and tracking; intr-satellite link; high power TWTAs (100-200 w); highly efficient IMPATT or FET amplifiers; multiple beam antennas system (reflector, microwave devices, phased array).

It is expected that there will be constraints associated with the planned TDRSS network operations. These constraints come from three sources:

- Those defined in the TDRSS system requirements or specifications (see TDRSS Users Guide)
- Those inherent to the NASA network operations, such as sun interference, user mutual interference and GRFI (Ground Radar Radio Frequency Interference)

- Those coming from the TDRSS system design and implementation such as TDRSS ground station message buffer size and response time limitations, service real-time reconfiguration, KSA auto-track system detection and false lock, etc.

These TDRSS system constraints imposed on users such as the space station may become an operational problem if they yield any intolerable condition to the user communications operations. In order to mitigate such constraints, some desirable TDRSS improvement may be essential to the TDAS end users.

Typical examples of desirable TDRSS improvements as derived from the projected space station requirements in order to cope with such constraints are:

- Develop a complete frequency plan to resolve any possible frequency congestion problem among TDRSS/TDAS, space station and other users
- Improve the TDRSS ground network (e.g., WSGT, NASCOM, NCC and POCC data and real-time message routing capability and capacity to improve the simultaneous operational efficiency and data throughput of the user end-to-end operations)
- Increase the WSNGT/NASCOM capability to enable concurrent operations of high data rate/analog/TV channels, which may very likely occur during simultaneous operations of space station, orbiter, LANDSAT-4 and other users
- Augment TDRSS channel reliability (no channel loss during real-time communications or reconfigurations)
- Enhance the TDRSS capability in detecting channel loss and in recovering channels
- Allocate a dedicated beam exclusively tc the space station - TDRS/TDAS communications
  - White Sands Ground Terminal (WSGT)
  - NASA Communications (Network) (NASCOM)
  - Network Control Center (NCC)
  - Payload Operational Control Center (POCC)
  - White Sands NASA Ground Terminal (WSNGT).

3.3 DATA MANAGEMENT

Within the last decade there have been many dramatic advances in computer technology. To apply the many potential benefits of this rapid expanding field to the Space Station, a Data Management Subsystem (DMS) was defined which assumes and exercises control of all activities performed on the Space Station. Requirements for DMS were developed and used to establish the level of processing performance that must be met. We have defined three major areas in identifying the DMS: trade issues; subsystem description; and evolving technologies.

3.3.1 Data Management Subsystem/System Trades

During the course of the study, many alternatives to key Data Management issues were investigated. The main areas include Architecture, Operational Autonomy, High Order Language, Network Topology and Transmission Media technology. Details of the Data Management Trades appear in Volume 2 - Book 2 - Part II. The results of the trade study are discussed in the following subsections.

3.3.1.1 Architecture - A critical factor in minimizing the Space Station system life cycle cost is subsystem integration and overall space avionics system architecture. The system architecture addresses the processing distribution of those Information Management System functions that will be performed on-board the space station. The allocation issues between on-board and ground will be presented in the next subsection. Figure 3-48 traces the flow for identification of on-board architecture and software requirements.

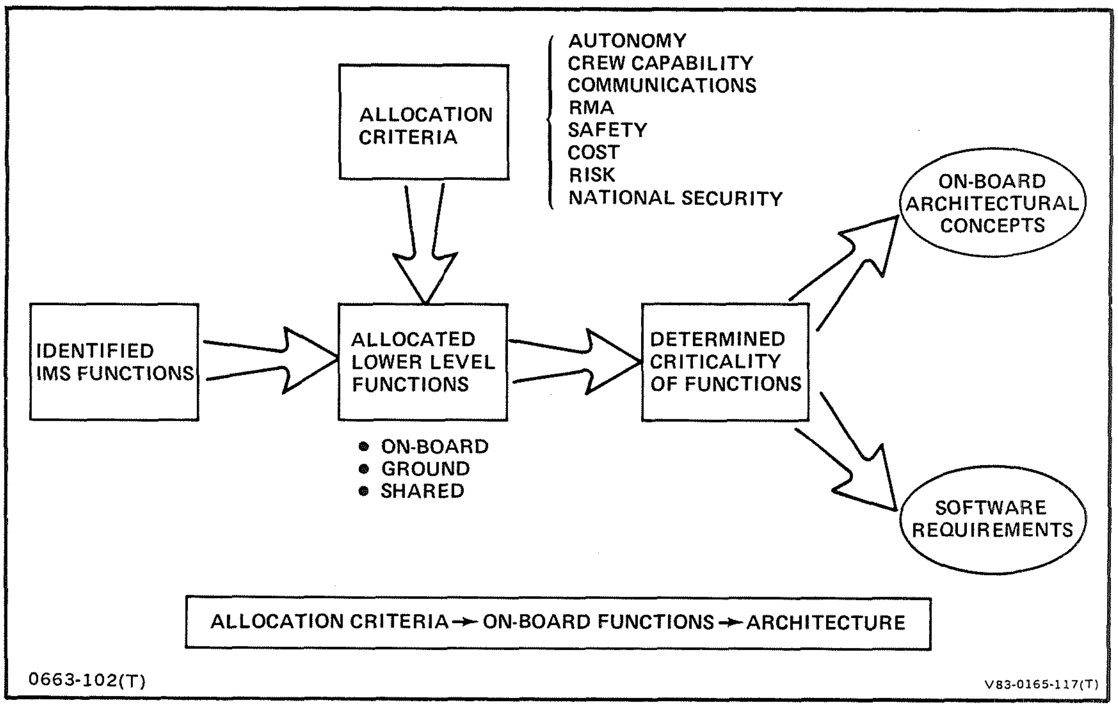


Fig. 3-48 Functional Allocation

The objectives of the DMS architecture are shown in Fig. 3-49. To meet these objectives several candidate configurations were considered. Each alternative architecture must satisfy the software requirements estimates for the Space Station DMS shown in Fig. 3-50. Readily identifiable approaches that can fulfill the requirements are the Centralized and the Distributed Processing architectures. Centralized is defined as an architecture where subsystems do not have any processing capability, thus overloading all system requirements into one computer. A single processor with the capacity to time-share all of the diverse programs would have to be extremely large and complex. Programs that would be used would be difficult to manage. The distributed processing architecture does not have these shortcomings.

- PERFORMANCE
  - IMPLEMENTS THE 68 ON-BOARD (& SHARED) FUNCTIONS
  - SUPPORTS PEAK & AGGREGATE THROUGHPUT REQUIREMENTS
  - SUPPORTS PEAK & AGGREGATE COMMUNICATION REQUIREMENTS
- SAFETY/RELIABILITY
  - FAULT DETECTION, ISOLATION, RECOVERY
  - FAIL OPERATIONAL/FAIL SAFE PERFORMANCE FOR CRITICAL FUNCTIONS
  - AUTONOMOUS OPERATION OF CREW SAFETY FUNCTIONS
- EXPANSION POTENTIAL
  - ABILITY TO ADD & DELETE MISSIONS
  - ABILITY TO ADAPT TO STATION RECONFIGURATIONS
- TECHNOLOGY TRANSPARENCY
- COST EFFECTIVE.

0663-103(T) V83-0165-118(T)

Fig. 3-49 DMS Architecture — Objectives

FUNCTION	LINES OF CODE	MAIN MEMORY	MASS MEMORY
STATION OPERATIONS	25k	1 MB	150 MB
MISSION OPERATIONS	25k	1 MB	150 MB
COMMUNICATION MANAGEMENT	15k	0.5 MB	100 MB
PERSONNEL SUPPORT	15k	0.5 MB	150 MB
ASTRONAUT PERSONAL DATA	15k	0.5 MB	150 MB
STATION SUBSYSTEMS	15k	1 MB	–
SUPPORT SOFTWARE	75k	(INCLUDED IN ABOVE ESTIMATES)	
	185k	4.5 MB	700 MB

*BASED ON BOTTOMS-UP ESTIMATES USING SIMILAR FUNCTIONS PERFORMED BY PAST SPACE PROGRAMS

0663-104(T) V83-0165-119(T)

Fig. 3-50 Software Estimates* for Space Station DMS

In a distributed architecture, the processing resources are incremental to the nodes minimizing impact of other nodes in both hardware and software. A trade was conducted to determine the degree of distribution (number of nodes) that should be considered as a result of supporting the total DMS functions. The methodology is identified in Fig. 3-51. The requirements of each function were first identified; then the functions were grouped into alternate functions so as to establish a degree of distribution between a fully centralized data processing system and a fully distributed system. Evaluation criteria that include hardware and software costs and expansion potential were used, in determining the optimum number of processings nodes that should be selected. The preferred configuration has four processor nodes in a distributed configuration. This is shown in Fig. 3-52. The evaluation work sheet for the selected configuration is shown in Fig. 3-53; the system architecture of the DMS is illustrated in Fig. 3-54. Note that each of the station operation processors is triply redundant, conforming to the fail-op, fail-safe requirements of the system.

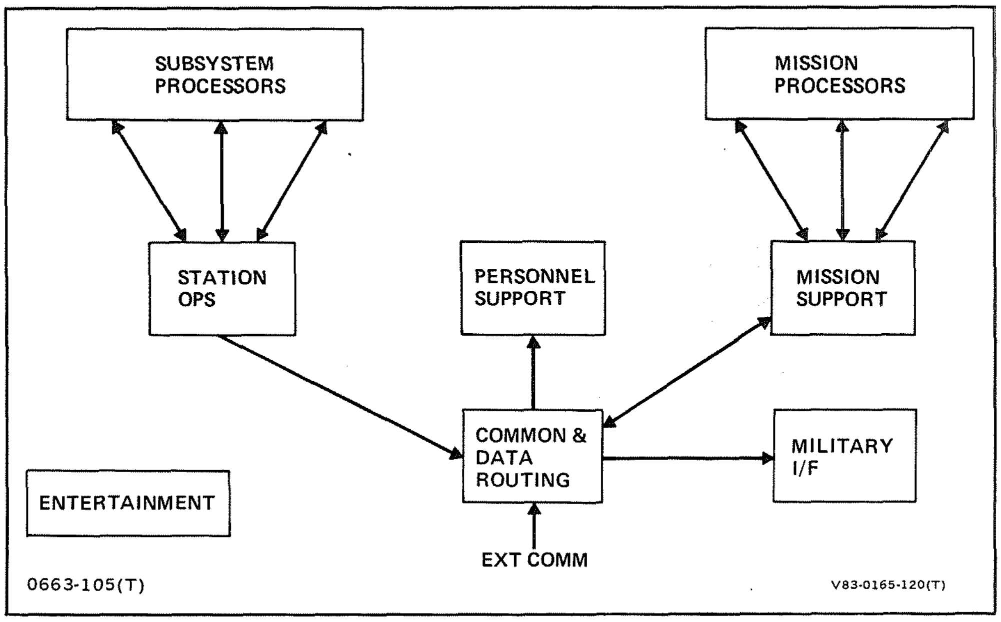


Fig. 3-51 DMS Processor Distribution

3.3.1.2 Autonomy - Autonomy can be further partitioned into machine autonomy and operational autonomy. Machine autonomy minimizes the involvement of the flight crew or ground personnel by having functions that are traditionally performed by man relegated to a machine. The DMS incorporates these philosophies in its conceptual design. This study, however, concentrated on the Operational Autonomy question: "Where is it most cost effective to perform a function, on the ground or aboard the Space Station?"

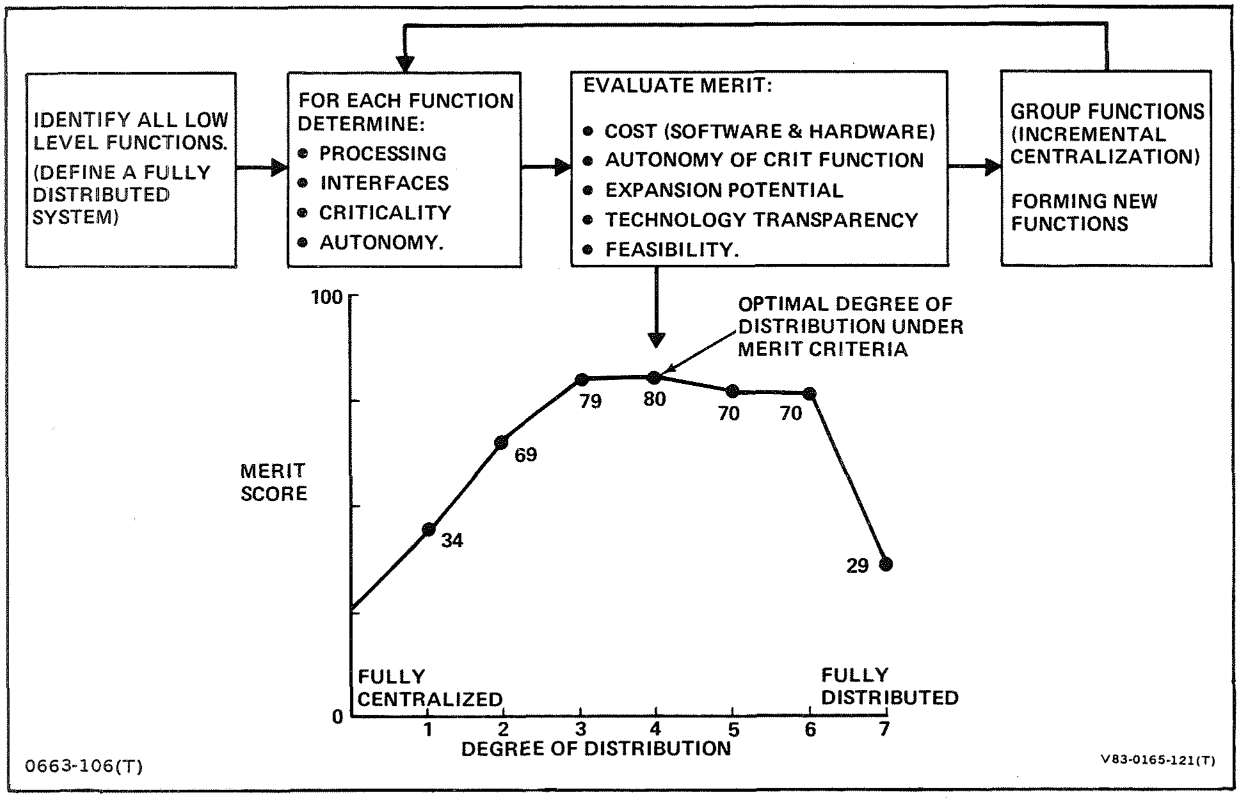


Fig. 3-52 Distribution Methodology

PROCESSORS IN THE STATION OPERATIONS SUBSYSTEM WERE COMBINED BASED UPON THROUGHPUT & INTERFACES. TO ALLOW FOR EXPANSION POTENTIAL, THE REQUIRED THROUGHPUT OF ANY PROCESSOR DOES NOT EXCEED 55% OF THE 700 KOP MAXIMUM.

CRITERIA	RATIONALE	SCORE (1-10)	WEIGHTED SCORE
COST	HARDWARE SOFTWARE INTEGRATION	7 7 7	7 3.5 3.5
AUTONOMY	CRITICAL FUNCTIONS ARE RELATIVELY INDEPENDENT BUT NOT AS GOOD AS ALTERNATIVE 3	8	16
EXPANSION	NO PROCESSOR IS MORE THAN 55% LOADED	8	16
TECH TRANS	CHANGES TO SUBSYSTEMS AFFECT TWO OR THREE FUNCTIONS	8	16
FEASIBILITY	FEWER INTERFACES THAN ALTERNATIVES 3	9	18
		TOTAL	80

0663-107(T) V83-0165-122(T)

Fig. 3-53 Evaluation Work Sheet

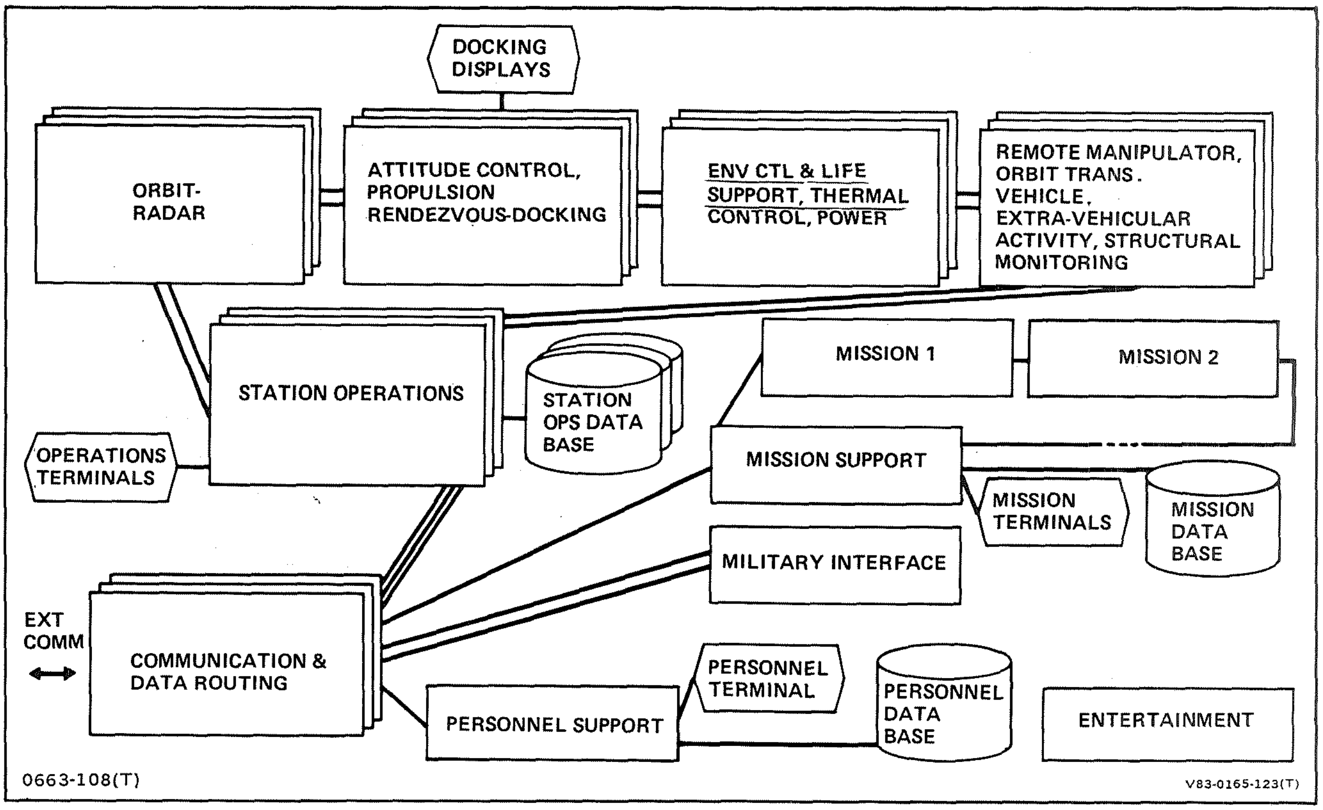


Fig. 3-54 DMS Architecture

To accomplish this, 18 major functions were identified and then decomposed to 84 lower level functions. The functions were allocated to the three areas where processing would be performed; on-board or flight operation, ground or shared-in both places (see Fig. 3-55). Tradeable functions were then re-evaluated based on their characteristics. The functions are listed in Fig. 3-56. A cost function algorithm, shown in Fig. 3-57, was then used in the evaluation, with the mission function and weighted values represented in Fig. 3-58. Results of the trade are summarized in Fig. 3-59. The trade concluded that 5 of the 21 functions re-evaluated resulted in a reverse allocation.

3.3.1.3 **High Order Language** - Software cost have dominated the DMS costs for the last decade. To minimize the escalation and hopefully drive the life cycle costs of software down, a trade study was performed between several candidate languages that can be utilized in the Space Station DMS. These included Ada, Fortran and Jovial. Fortran and Jovial are established high order languages, but they lack many requesites for reliable, maintainable, long life cycle systems such as the Space Station. Ada, on the other hand, was designed specifically for this type of application and includes features such as exception handling and real time control.

	ON-BOARD	GROUND	SHARED
USER/PI INTERFACE	0	3	2
SYSTEM COMMAND & CONTROL	4	2	0
MISSION SUPPORT	2	1	0
S/S HARDWARE MAINTENANCE	3	0	3
S/S SOFTWARE MAINTENANCE	1	0	4
CREW HEALTH MONITORING/MAINTENANCE	2	0	4
SPACEBORNE EXPERIMENTATION	2	1	3
S/S ONBOARD SUPPORT	13	0	1
S/S SUPPORT SUBSYSTEM C&C	2	0	0
S/S MISSION SUBSYSTEM C&C	2	1	0
S/S SUPPORT SUBSYSTEM MONITORING	4	1	0
S/S MISSION SUBSYSTEM MONITORING	4	1	0
MISSION DATA DISTRIBUTION	1	3	1
ENTERTAINMENT	3	0	1
DATA STORAGE	1	1	1
PERFORMANCE EVALUATION	2	2	0
MILITARY SUPPORT	1	0	0
TRAINING & SIMULATION	1	0	0
TOTALS	48	16	20

V83-0165-124(T)

Fig. 3-55 Allocation Summary

THE FOLLOWING FUNCTIONS HAVE BEEN SELECTED FOR TRADE-OFF RE-EVALUATION BASED ON THEIR CHARACTERISTICS. NEXT TO EACH FUNCTION IS FOUND A PRE-ALLOCATION CODE AND A SECOND CODE (H, M, OR L), WHICH REFERS TO ITS CRITICALITY.

0663-109 V83-0165-125(1/3)(T)

Fig. 3-56 Tradable Functions (Sheet 1 of 3)

MISSION ORIENTED FUNCTIONS

• MISSION OPERATIONS SCHEDULING	OB, L
• MISSION SUBSYSTEM COMMANDING	OB, M
• MISSION OPERATION	OB, L
• SHORT-TERM MISSION PERFORMANCE EVALUATION	OB, M
• LONG-TERM TREND ANALYSIS	G, L
• MISSION DATA COLLECTION	OB, L
• MISSION DATA PRE-PROCESSING	OB, L
• MISSION DATA PROCESSING	G, L
• DATA RECORDING SPACE BORNE EXPERIMENTS	OB, L
• DATA ANALYSIS SPACE BORNE EXPERIMENTS	G, L

0663-110 V83-0165-125(2/3)(T)

Fig. 3-56 Tradable Functions (Sheet 2 of 3)

SUPPORT OPERATIONS FUNCTIONS

• HARDWARE FAULT DETECTION	OB, H
• HARDWARE CORRECTIVE ACTION	OB, H
• SOFTWARE FAULT DETECTION	OB, H
• SUBSYSTEM SUPPORT LOGISTICS	OB, L
• SUPPORT SUBSYSTEM C&C S/S COMMANDING	OB, H
• SUPPORT SUBSYSTEM C&C PROCEDURE DISPLAY/PROCESSING	OB, H
• MISSION SUBSYSTEM C&C PROCEDURE DISPLAY/PROCESSING	OB, M
• SUPPORT SUBSYSTEM TREND ANALYSIS	G, L
• LONG-TERM SYS PERFORMANCE EVALUATION	G, M
• SHORT-TERM SYS PERFORMANCE EVALUATION	OB, H
• LONG-TERM MISSION PERFORMANCE EVALUATION	G, L

0663-111(T) V83-0165-125(3/3)(T)

Fig. 3-56 Tradable Functions (Sheet 3 of 3)

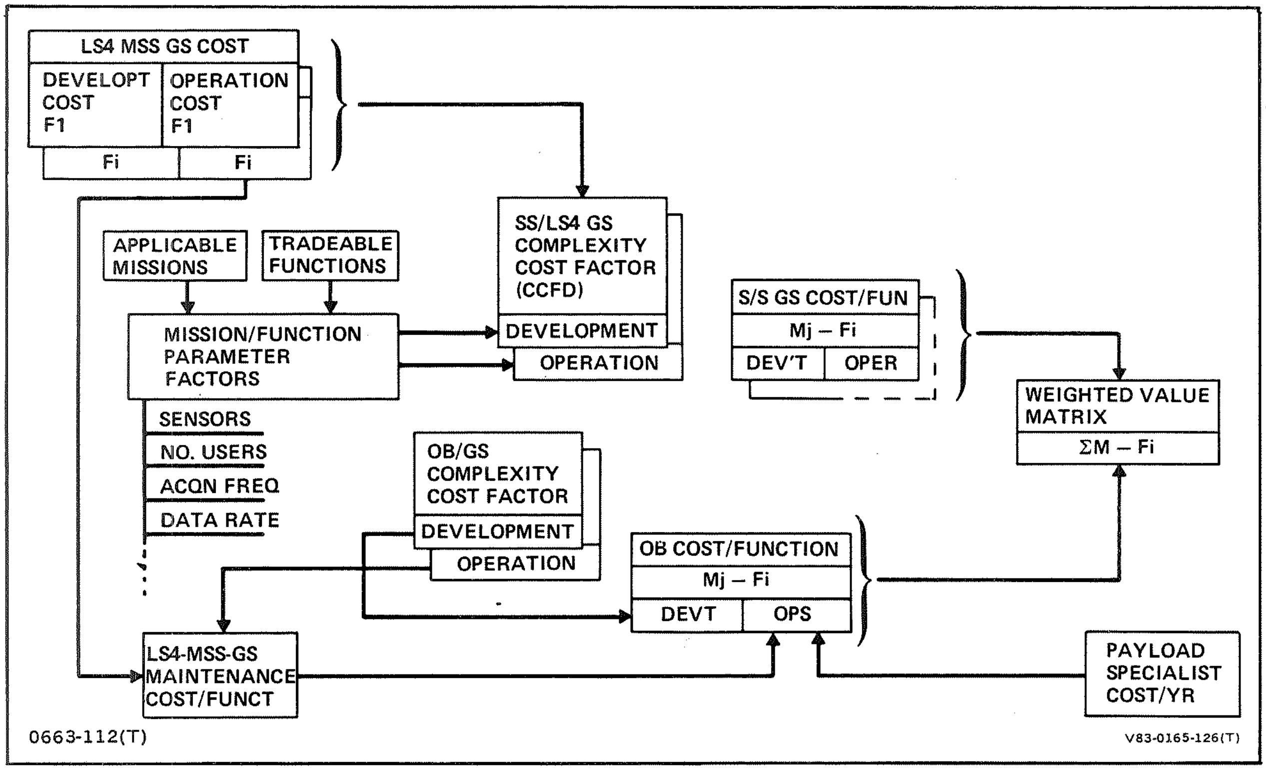


Fig. 3-57 Space Station Cost/Function Algorithm

MISSION FUNCTION	ASSIGNED VALUES			WEIGHTED VALUES	
	ASSIGNED WEIGHT	ON GND VALUE	ON BRD VALUE	ON GND VALUE	ON BRD VALUE
MISSION OPS SCHEDULING					
1.0 AUTONOMY	20	0.00	1.00	0.00	20.00
2.0 HEALTH & SAFETY	–	–	–	0.00	0.00
3.0 CREW CAPABILITY	–	–	–	0.00	0.00
4.0 CREW LOAD	–	–	–	0.00	0.00
5.0 COST	60	0.57	0.43	34.20	25.80
6.0 REL/AVAILABILITY	10	0.70	0.30	7.00	3.00
7.0 MAINTAINABILITY	10	0.70	0.30	7.00	3.00
8.0 COMM LOAD	–	–	–	0.00	0.00
9.0 TECHNICAL RISK	–	–	–	0.00	0.00
10.0 PROCESSING LOAD	–	–	–	0.00	0.00
11.0 USER ACCESS	–	–	–	0.00	0.00
12.0 CO-LOCATION	–	–	–	0.00	0.00
13.0 BACK-UPS	–	–	–	0.00	0.00
MISSION OPS SCHEDULING	–	–	–	48.20	51.80
0663-113(T)					V83-0165-127(T)

Fig. 3-58 Allocation Weighted Value Matrix

MISSION FUNCTIONS	PRE-ALLOCATION (CRITICALITY)	OB/GS WEIGHTED VALUE RATIO	REMARKS
MISSION OPS SCHEDULING	OB(L)	52/48	MAINTAIN PRE-ALLOCATION
MISSION SUBSYS COMMANDING	OB(M)	50/50	MAINTAIN PRE-ALLOCATION
MISSION OPERATIONS	OB(L)	51/49	MAINTAIN PRE-ALLOCATION
SHORT-TERM MISSION PE	OB(M)	69/31	CONFIRM PRE-ALLOCATION
LONG-TERM MISSION TREND ANALYSIS	G(L)	69/31	REVERSE PRE-ALLOCATION
MISSION DATA COLLECTION	OB(L)	32/68	REVERSE PRE-ALLOCATION
MISSION DATA PREPROCESSING	OB(L)	34/66	REVERSE PRE-ALLOCATION
MISSION DATA PROCESSING	G(L)	37/63	CONFIRM PRE-ALLOCATION
DATA RECORDING SS EXPERIMENTS	OB(L)	27/73	REVERSE PRE-ALLOCATION
DATA NALYSIS – SS EXPERIMENTS	G(L)	22/79	CONFIRM PRE-ALLOCATION

0663-114(T) V83-0165-128(1/2)(T)

Fig. 3-59 Space Station Function Summary Value Allocation (Sheet 1 of 2)

SUPPORT OPERATIONS	PRE-ALLOCATION (CRITICALITY)	OB/GS WEIGHTED VALUE RATIO	REMARKS
H/W FAULT DETECTION	OB(H)	63/37	CONFIRM PRE-ALLOCATION
H/W CORRECTIVE ACTION	OB(H)	63/37	CONFIRM PRE-ALLOCATION
S/W FAULT DETECTION	OB(H)	63/37	CONFIRM PRE-ALLOCATION
S/S SUPPORT LOGISTICS	OB(L)	58/42	CONFIRM PRE-ALLOCATION
SUPPORT S/S C&C COMMANDING	OB(H)	48/52	MAINTAIN PRE-ALLOCATION
SUPPORT S/S C&C PROCEDURE DISPLAY/ PROCESSING	OB(H)	48/52	MAINTAIN PRE-ALLOCATION
MISSION C&C PROCEDURE DISPLAY/PROCESSING	OB(M)	50/50	MAINTAIN PRE-ALLOCATION
S/S TREND ANALYSIS	G(L)	53/47	MAINTAIN PRE-ALLOCATION
L/T SYS PERFORMANCE EVALUATION	G(M)	52/48	MAINTAIN PRE-ALLOCATION
S/T SYS PERFORMANCE EVALUATION	OB(H)	50/50	CONFIRM PRE-ALLOCATION
L/T MISSION PERFORMANCE EVALUATION	G(L)	40/60	REVERSE PRE-ALLOCATION

0663-115(T) V83-0165-128(2/2)(T)

Fig. 3-59 Space Station Function Summary Value Allocation (Sheet 2 of 2)

In Fig. 3-60 criteria pertinent to the Space Station were used to tradeoff the three high level languages. If we assume that Ada will meet its design goals and that the compiler and the support environment are available on schedule, then this language will provide the best solution for minimizing the software costs of the DMS.

3.3.1.4 **Transmission Media Technology** - The communication of data between computers and terminals has traditionally been accomplished via hardwire. Although fiber optic technology has been available since the early 1960s, fiber optics communication was not viewed seriously until the 1970s, when 20-dB/km attenuation was achieved. Today, reliable and economical fibers, connectors, emitters and detectors are readily available. Fiber optics provides virtually unlimited bandwidth, insensitivity to electromagnetic interference and lightweight. When fiber optics is compared to hardwire for the criteria shown in Fig. 3-61, the results show that this technology is the best approach in all these categories. Fiber optics only suffers when considering interconnect complexities. But greater application of the technology is reducing the complexity. The selection of fiber optics as the transmitting medium technology does not preclude that all issues associated with fiber technology are eliminated. There are still design issues (shown in Fig. 3-62) that must be addressed when the Space Station design becomes more mature.

3.3.1.5 **Network Topology** - There are many diverse network systems used in data communication. The space station architecture uses distributed processing with five network nodes. Local area network, as this technology is called, allows for the distribution of data at rates specified by the performance requirements. A network topology trade was conducted with several candidate configurations. These were investigated, not only for linking the DMS network nodes, but for internal application to the mission support and station operation areas. The ring (unidirectional and bidirectional), the multidrop or global bus, the star and the fully connected topologies were all considered. In addition to the criteria listed in Fig. 3-63, attention was also given to the need for a centralized/decentralized controller and to the complexity of the protocols that each topology requires. The results of the topology trade was the selection of the bidirectional ring to the station operation and mission support areas and the star network selected for the DMS network.

CRITERIA	WEIGHT (Wi)	Ada*		FORTRAN		JOVIAL	
		SCORE	Wi	SCORE	Wi S	SCORE	Wi S
LANGUAGE FEATURES	3	3	9	1	3	2	6
MATURITY OF LANGUAGE	4	1	4	3	12	2	8
MAINTAINABILITY	4	3	12	2	8	2	8
RELIABILITY	5	3	15	1	5	2	10
FAULT TOLERANCE	5	3	15	1	5	2	10
SUPPORT TOOLS	3	3	9	3	9	2	6
EFFICIENCY OF CODE GENERATION	5	3	15	2	10	2	10
TRAINING	1	1	1	3	3	1	1
TOTAL (Σ Wi S)		80		55		59	

*PROJECTED – ASSUMES Ada MEETS DESIGN GOALS AND HAS BEEN IN USE

SCORE: 1 – FAIR
2 – GOOD
3 – EXCELLENT

V83-0165-161(T)

Fig. 3-60 High Order Language Trade Study

- FIBER TYPE (GRADED OR STEP)
- WAVE LENGTH OF OPERATION
- FIBER SIZE
- JACKET MATERIAL
- CONNECTING VERSUS SPLICING
- INPUT/OUTPUT SIGNAL AND IMPEDANCE INTERFACE
- FIBER NA
- FIBER LOSS CHARACTERISTICS
- SYSTEM ARCHITECTURE
- TYPE OF LIGHT SOURCE
- TYPE OF RECEIVER
- OPTICAL LOSS BUDGET
- OPTICAL TIME DELAY
- SYSTEM FREQUENCY RESPONSE
- BEND RADIUS CONSIDERATIONS
- SIZE AND WEIGHT OF R/Ts
- ENVIRONMENTAL EFFORTS

0663-116(T) V83-0165-162(T)

Fig. 3-61 Transmission Media Technology Trade

OPTIONS	CRITERIA	COMMENTS
FIBER OPTICS	• BANDWIDTH/DATA RATE	• FIBER OPTICS IS BEST APPROACH IN ALL CATEGORIES
	• EXPANDABILITY	
	• TECHNOLOGY TRANSPARENCY	
HARDWIRE	• COMPLEXITY	
	• RELIABILITY	
	• LIFE CYCLE COST	
	• SIZE, WEIGHT, POWER	

0663-117(T) V83-0165-163(T)

Fig. 3-62 Fiber Optic Design Issues (Typical)

	REDUNDANT LINKS REQUIRED	VULNERABILITY TO "CHATTY" NODES	COMPLEXITY OF ROUTING	FIBER OPTIC IMPLICATIONS	OPERATION IN PHYSICALLY DISTRIBUTED SYS	PERFORMANCE WITH MANY NODES	EXPANSION (ADD/ DELETE NODES)	UTILIZATION
RING (UNI-DIRECTIONAL)	YES	MODERATE	LOW	ALL-POINT TO POINT SIMPLEX	GOOD-FEW LINKS	GOOD	FAIR	
RING (BIDI-RECTIONAL	NO	LOW	MODERATE	ALL-POINT TO POINT FULL DUPLEX	GOOD-FEW LINKS	GOOD	FAIR	STATION OPERATIONS & MISSION SUPPORT
MULTIDROP	YES	HIGH	MODERATE	NO. OF DROPS LIMITED BY COUPLING LOSSES, FULL/ HALF DUPLEX	GOOD-FEW LINKS	GOOD (WIRE LINKS)	FAIR	
STAR	YES	LOW	–	ALL-POINT TO POINT. SIMPLEX/HALF DUPLEX	POOR-FOR LARGE NO. OF NODES	POOR	GOOD	DATA MANAGEMENT SYSTEM NODES
FULLY CONNECTED	NO	LOW	HIGH	ALL-POINT TO POINT. FULL DUPLEX	POOR-FOR LARGE NO. OF NODES	FAIR	POOR	V83-0165-164(T)

Fig. 3-63 Network Topology Trade

3.3.1.6 Fault Tolerance Trades - The reliability requirements of the Space Station specify that all critical functions be fail-operational/fail-safe, as a minimum. There are alternative approaches of applying this requirement that were studied. These fault-tolerant trades were performed for two categories; the orbital replaceable unit and the communication and the network area of the Data Management Subsystem. The criteria used were fault detection and fault isolation, automatic recovery, cost and complexity. Candidate approaches are listed in Fig. 3-64. Fault-tolerant implementations for the processor and networks are shown in Fig. 3-65. The results of the fault tolerant trades are documented in Fig. 3-66. Conclusions reached are that a three-processor with majority voting is the preferred choice for critical functions. Reconfigurable spares exist for all processing functions. It is envisioned that the evolving Space Station will automatically reconfigure the processing elements. Automatic self-test will be performed in the background while the application programs are being executed (foreground processing). Ada is a highly structured language and has been selected as the high order language. Its attributes will enhance reliable software.

Networks will have redundant links and will use cycle redundant codes (CRC). The trade study shows that automatic reconfiguration around failed nodes by use of the redundant links is the preferred approach.

CATEGORY	METHOD	FAULT DETECTION	FAULT ISOLATION	AUTOMATIC RECOVERY	COST	COMPLEXITY	RECOMMEND FOR
ORBITAL REPLACE-ABLE UNIT	5 PROCESSOR MA-JORITY VOTING	EXCELLENT	GOOD	EXCELLENT	VERY HIGH	HIGH	
	4 PROCESSOR CROSS CHECK	EXCELLENT	GOOD	EXCELLENT	VERY HIGH	HIGH	
	3 PROCESSOR MAJORITY VOTING	GOOD	GOOD	GOOD	HIGH	MODERATE	CRITICAL
	2 PROCESSOR, HOT STANDBY	FAIR	FAIR	FAIR	MODERATE	LOW	
	2 PROCESSOR, COLD STANDBY	FAIR	FAIR	FAIR	MODERATE	LOW	
	RECONFIGUR-ABLE SPARES	POOR	POOR	NONE	LOW	LOW	ALL
	REDUNDANT INFORMATION	FAIR	FAIR	FAIR	HIGH	HIGH	V83-0165-165(T)

Fig. 3-64 Fault Tolerance (Sheet 1 of 2)

CATEGORY	METHOD	FAULT DETECTION	FAULT ISOLATION	AUTOMATIC RECOVERY	COST	COMPLEXITY	RECOMMEND FOR
COMMUNI-CATION & NETWORK AREA	REDUNDANT IN-FORMATION (CRC)	GOOD	GOOD	FAIR	LOW	MODERATE	ALL
	REDUNDANT LINKS	GOOD	GOOD	GOOD	LOW	LOW	CRITICAL
	FUNCTIONAL REASSIGN-MENT BETWEEN PROCESSORS	GOOD	GOOD	FAIR	HIGH	HIGH	
	FAULT TOLER-ANCE AT ORBITAL REPLACEABLE UNIT LEVEL	GOOD	GOOD	GOOD	MODERATE	LOW	ALL
	FAULT TOLER-ANCE AT SYSTEM LEVEL	FAIR	POOR	FAIR	MODERATE	HIGH	V83-0165-166(T)

Fig 3-64 Fault Tolerance (Sheet 2 of 2)

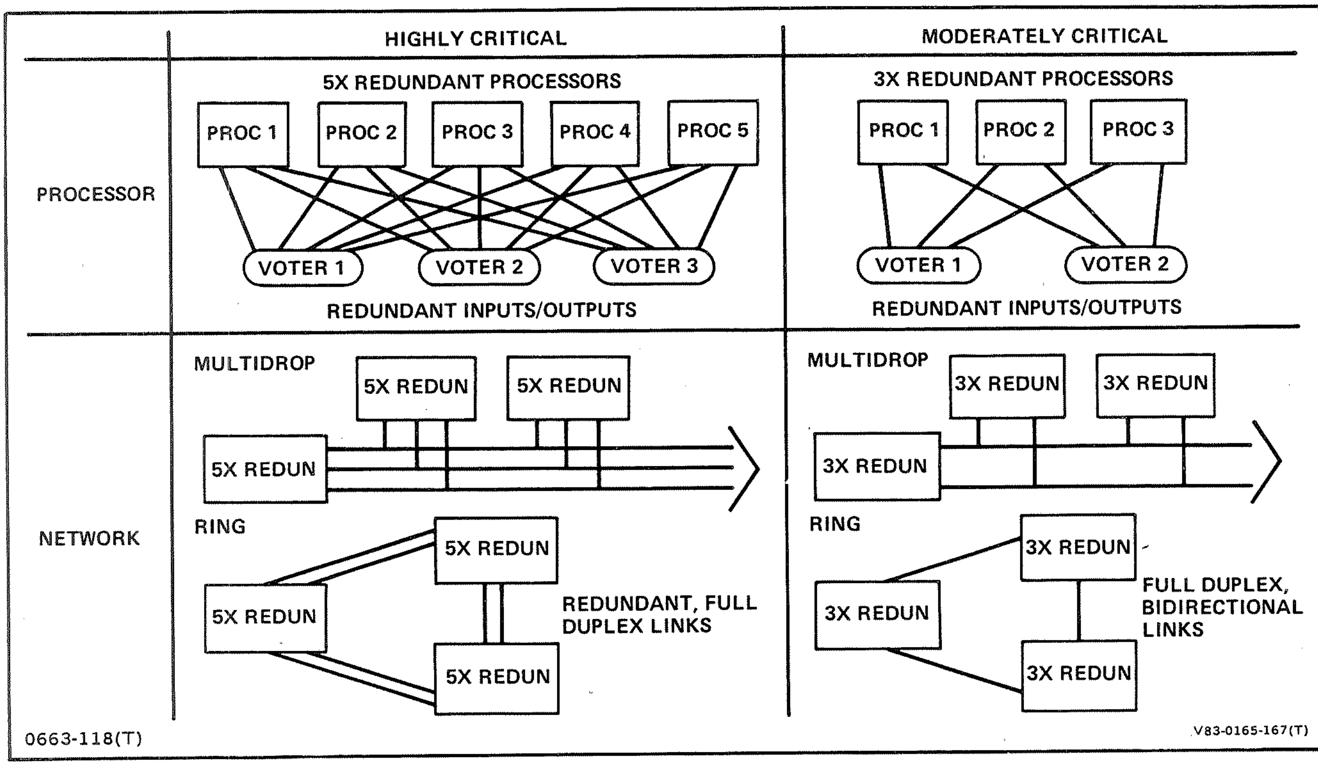


Fig. 3-65 Fault Tolerant Implementations

- HARDWARE
  - 3 PROCESSOR MAJORITY VOTING FOR CRITICAL FUNCTIONS
  - RECONFIGURABLE SPARES FOR ALL FUNCTIONS
  - AUTOMATIC SELF TESTING IN BACKGROUND
- SOFTWARE
  - HIGH LEVEL STRUCTURED LANGUAGE
  - FAULT TOLERANT SOFTWARE METHODS REQUIRE FURTHER ANALYSIS
- NETWORKS
  - CRC CODES
  - POLLING
  - AUTOMATIC RECONFIGURATION AROUND FAILED NODES
  - INSURE 2 COMMUNICATION LINKS TO ALL CRITICAL NODES

0663-119(T) V83-0165-168(T)

Fig. 3-66 Fault Tolerance DMS Implementation

3.3.1.7 Summary Trade Studies - Many trades were conducted during the study so as to develop the most cost effective DMS for the Space Station time period. Each of the elements of the DMS was traded off using the most important drivers of the technology. A summary of the technologies and the selected options are found in Fig. 3-67. Of the trades conducted, two technologies are herewith highlighted. In the main memory area, which requires non-volatility, CMOS was selected over magnetic core based on the higher density and lower power characteristics. Another requirement, that of not using electro-mechanical components, precluded the use of optical discs in the auxiliary memory area. This technology promises high density and low cost per megabyte.

TECHNOLOGY	OPTIONS	CRITERIA	COMMENTS
PROCESSOR-CPU	$\boxed{\text{• MICRO-COMPUTER}}$ • MINI-COMPUTER • LARGE SCALE	• FAULT TOLERANCE • COST • COMPLEXITY • ARCHITECTURE	• 700-1000 KOPS BY 1995 • RAD HARD/SPACE QUALIFICATION WILL IMPACT SELECTION
MAIN MEMORY	$\boxed{\text{• CMOS}}$ • MAGNETIC CORE • PLATED WIRE	• WT/SIZE/POWER • COST • ACCESS TIME	• RAD HARDENING IS A MAJOR FACTOR • NON-VOLATILITY REQUIRED – SOLID STATE CMOS WITH BATTERY BACKUP MEETS THIS REQUIREMENT
AUX MEMORY	$\boxed{\text{• BUBBLE MEMORY}}$ • CCD MEMORY	• WT/SIZE/POWER • CAPACITY	• AVOIDS USE OF ELECTRO-MECHANICAL COMPONENTS • EXPECTED TO ACHIEVE 10^9 BITS/CHIP BY 1990 • 1.5×10^6 BITS/SEC – TRANSFER RATE

V83-0165-185(1/2)(T)

Fig. 3-67 DMS Technology Trade Summary (Sheet 1 of 2)

TECHNOLOGY	OPTIONS	CRITERIA	COMMENTS
DISPLAYS	$\boxed{\text{• CRT}}$ • FLAT CRT • FLAT PANELS	• WT/SIZE/POWER	
SOFTWARE	$\boxed{\text{• Ada}}$ • FORTRAN • JOVIAL	• STANDARDIZATION • STRUCTURED LANGUAGE • EFFICIENCY	• Ada IS DOD'S STANDARD LANGUAGE COMMONALITY IS SOUGHT WITH NASA/WORLD WIDE
NETWORK	$\boxed{\text{• RING}}$ • MULTIDROP $\boxed{\text{• STAR}}$ • FULLY CONNECTED	• REDUNDANCY • EXPANDABILITY	• RING SELECTED FOR MISSION SUPPORT & STATION OPERATIONS INTERNAL COMMUNICATION • STAR SELECTED FOR DATA MANAGEMENT NODES
TRANSMISSION MEDIA	$\boxed{\text{• FIBER OPTICS}}$ • WIRE	• BANDWIDTH • WT/SIZE/POWER	V83-0165-185(2/2)(T)

Fig. 3-67 DMS Technology Trade Summary (Sheet 2 of 2)

3.3.2 Data Management Subsystem Description

Having as a foundation the DMS performance requirements shown in Fig. 3-68 and completed technology trades leading to the selection of preferred data process- ing elements, a preferred DMS that efficiently melds and preserves the individual technology gains was identified. Figure 3-69 illustrates the preferred architecture and its elements. A detail description of the DMS is found in Part II.

KEY PARAMETERS		HABITAT	MILITARY FACILITY	SPACE TEST FACILITY	SATELLITE SERVICE FACILITY	INDUS-TRIAL PARK	OBSER-VATORY	TRANSPORT HARBOR
*DATA RATES (KPS)	MISSION	11.6	704	1875	0.5	1.9	3300	–
	OPERATIONS	45	23	6	6	6	15	15
*STORAGE (x 10^9 CAPACITY BPD)	MISSION	1.0	61	86	0.04	0.17	280	–
	OPERATIONS	0.6	0.6	0.2	0.2	0.2	0.2	0.2
*PROCESSING SPEED (x 10^6 OPS)	MISSION	0.005	1.4	0.5	0.0003	0.0004	2.1	–
	OPERATIONS	1.3	1.3	0.1	0.1	0.1	1	0.2
*COMMUNICATION RATES (x 10^6 BPS)		0.008	4.2	7.5	0.001	0.001	13	0.1

*AVERAGED OVER A 24 HOUR PERIOD

0663-120(T) V83-0165-173(T)

Fig. 3-68 DMS Performance Requirements

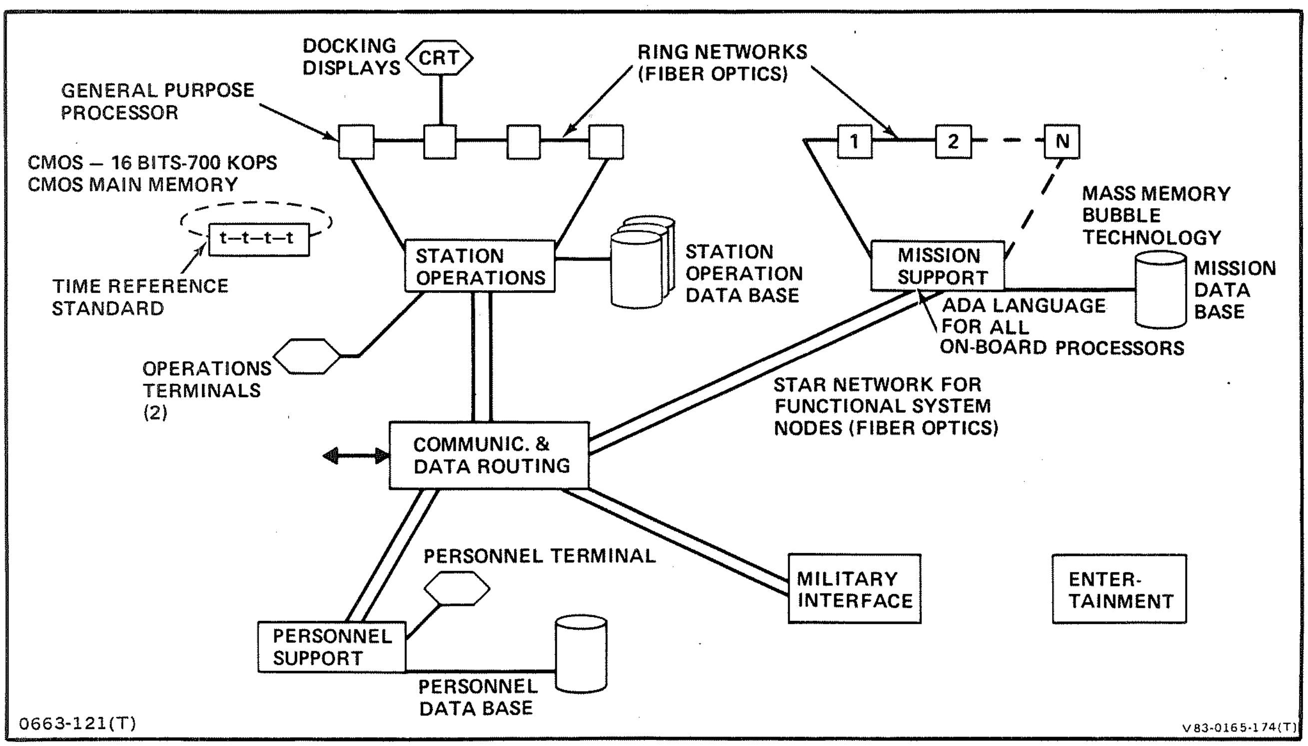


Fig. 3-69 Data Management Subsystem

3.3.2.1 **Hardware Description** - The system consists of a distributed processing system containing five non-homogeneous nodes where multiple computer resources are available. Each processing element is backed up by two other redundant elements resulting in satisfying fail-operational/fail-safe requirements. The major functions timeshare processing elements with less critical functions, thereby minimizing the total number of processing elements. This architecture supports the physical buildup of the Space Station and the incremental software development for the subsystems. A time reference standard for subsystem use will be an integral part of the DMS. The distributed processor resources at each node will be controlled centrally.

3.3.2.2 **Software Description** - Ada is projected as the implementation language for the Space Station software. The software for the DMS can be partitioned into the Application Software and the System (Executive) Software utilized to manage and support the application software (see Fig. 3-70). Included in the system software are modules for data base management and support software tools. The operating system module will include the Ada run time system, permitting resource management and redundancy management.

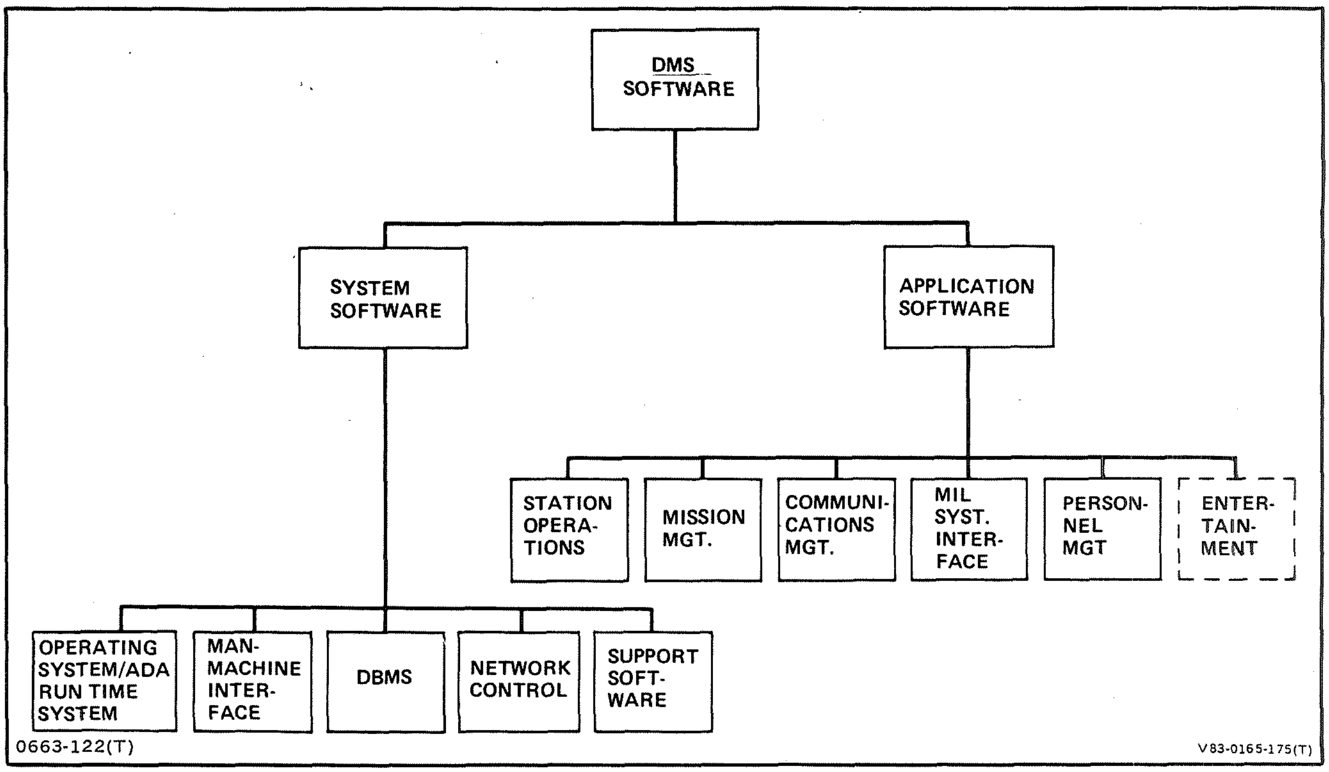


Fig. 3-70 DMS Software Overview

The application software has been partitioned into the five functional system nodes, with a sixth node as the entertainment center. A diagram showing the distribution of the software functions is shown in Fig. 3-71.

3.3.2.3 **Physical Characteristics** - Physically, the DMS consists of processors with main memory, interface units, auxiliary memories, CRTs and transmission networks between various nodes. In Fig. 3-72 the total number of these components are identified and their physical characteristics, weight, power and volume described.

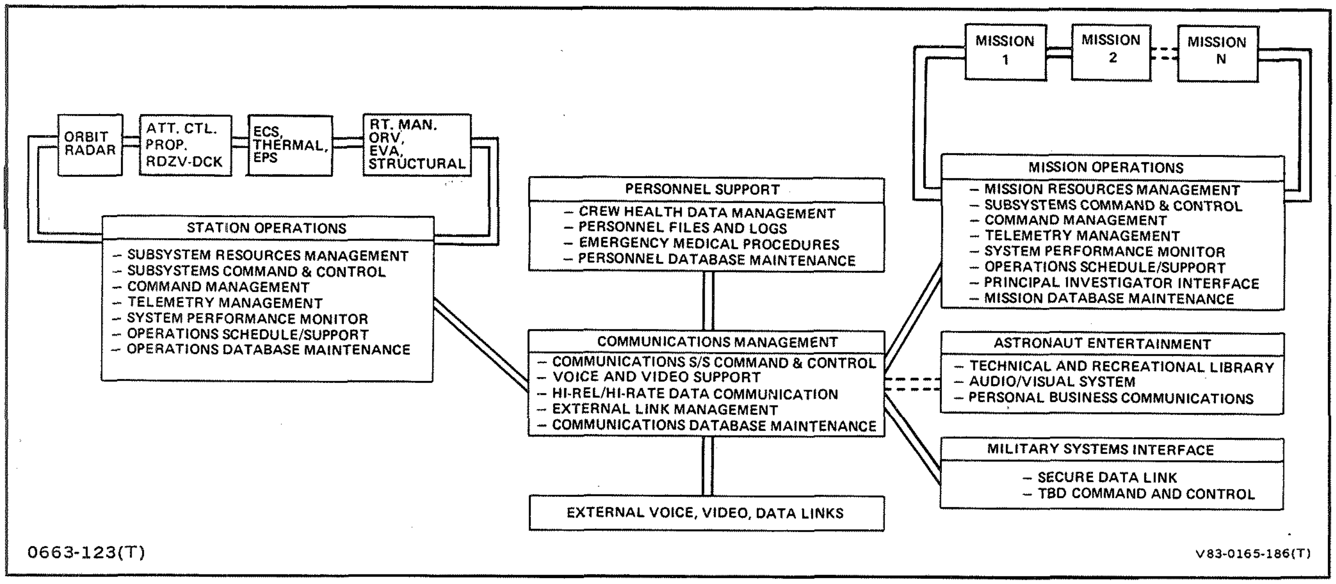


Fig. 3-71 Applications Software Overview

COMPONENT	QUANTITY	UNIT WEIGHT (kg)	TOTAL WEIGHT (kg)	UNIT POWER (WATTS)	TOTAL POWER (WATTS)	UNIT VOLUME (CUBIC FT)	TOTAL VOLUME (CUBIC FT)
MASS MEMORY	5	*	108	*	1200	*	4.75
CRT	9	15.9	143	51	459	1.7	15.3
PROCESSORS & INTERFACE UNITS	22	**	39	10	200	**	3.5
TOTAL			290		1919		23.55

* NOT APPLICABLE – PACKAGED IN 3 PHYSICAL AREAS
** NOT APPLICABLE – PROCESSORS PARTITIONED TO 5 NODES

0663-124(T) V83-0165-268(T)

Fig. 3-72 Data Management System

3.3.3 DMS Evolution

The DMS of the year 2000 Space Station, will provide additional capabilities and increased reliability over the Initial Space Station of 1990. Performance and reliability is not gained, however, without a structured plan that permits for technology insertion by use of technology transparency. In the DMS, new technology or improvements will take place as a result of using standard interfaces, thus providing the desired technology transparency. In the following subsections, we will discuss the DMS of the year 2000, identifying the difference in technology and benefits derived. We will identify development areas and indicate the necessary steps required to bring the technology to fruition.

3.3.3.1 DMS Technologies - The DMS for the year 2000 will contain the baseline architecture development for the 1990 Space Station (see Fig. 3-73). Within the Station Operations Node, additional processing elements are envisioned which will be implemented to function as standby redundant/automatic reconfigurable modules. The benefit derived from using this concept is increased resource sharing, thereby providing greater reliability. This processing node has, as a baseline, a cen-

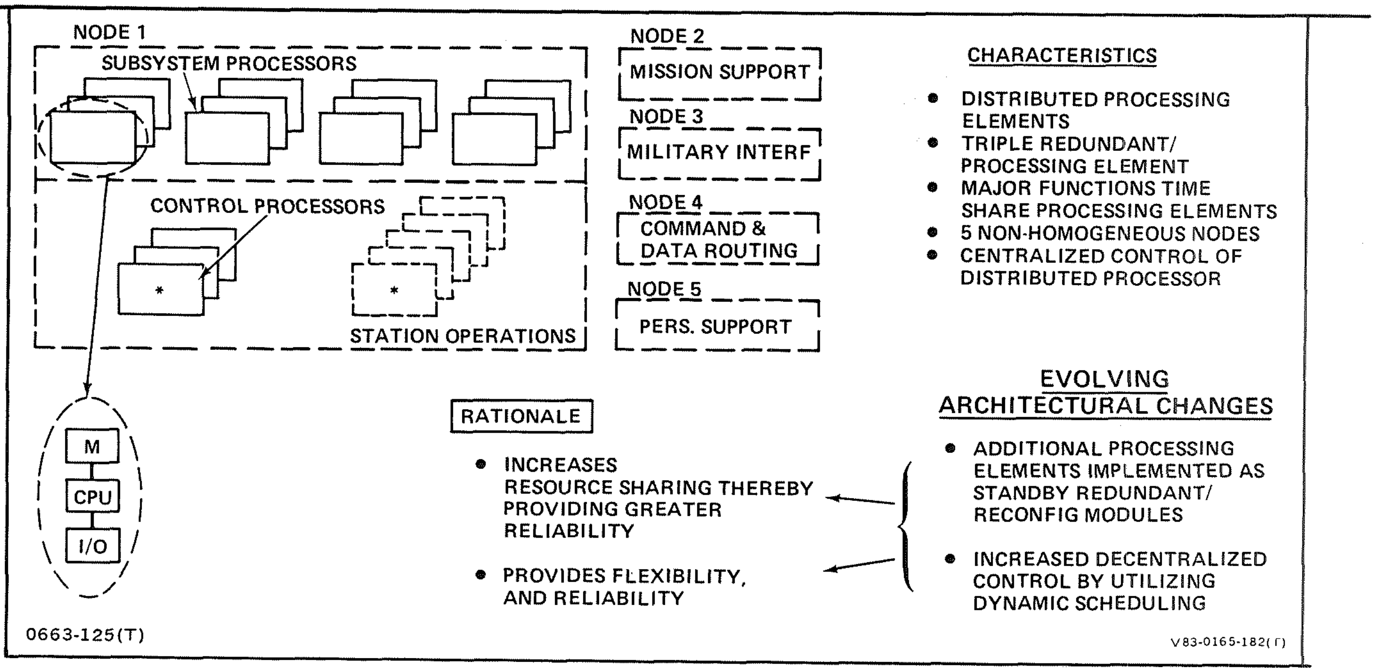


Fig. 3-73 DMS Architecture 1990 — 2000

tralized control. It is preferred, in an evolving manner, to increase the decentralization of the control function by using dynamic scheduling. Flexibility and increased reliability are the attributes of this approach.

The characteristics of the DMS elements are listed in Fig. 3-74. It is anticipated that hardware performance improvements will be made based on more maturing technology. This transition will result in increasing the capability of the DMS to meet the anticipated growth requirements of an advanced Space Station.

Software will permit greater sophistication in applications in the year 2000 Space Shuttle. As the mission experiments mature, the use of high order languages will evolve to permit Artificial Intelligence applications. Robotics is a definable output that can increase the machine autonomy functions. In Fig. 3-75, the evolved features of software are presented. The improvements that will be made in software will be as a result of using more mature software and the increasing trends of standardizing the software in a manner similar to the hardware. An increased use of Very High Speed Integrated Circuits (VHSIC) for Space Station applications will increase the use of Ada.

3.3.3.2 **Technology Development Plan** - The technologies listed in Fig. 3-76 have been identified as those requiring to be developed so as to provide the increased capability Space Station in the year 2000. Auxiliary memory and display technology will provide the greatest rewards since both of these technologies in their present form have a large penalty in terms of physical characteristics. High density, high speed, non-volatile random access main memory in a low power technology can also enhance the capability of the DMS. With maturing software and accomplishing the objectives of the technologies listed, the year 2000 Space Station will meet challenging requirements of the future.

3.4 ENVIRONMENTAL CONTROL/LIFE SUPPORT SUBSYSTEM (EC/LSS)

3.4.1 Trade Studies

A major trade study has been conducted to aid in the selection of the Space Station EC/LSS. The study was to choose between: an upgraded open loop EC/LSS

	1990	2000	RATIONALE
GENERAL PURPOSE PROCESSOR			
THRUPUT	700 KOPS	1 MOPS	HIGHER PERFORMANCE
WORD LENGTH	16 BITS	32 BITS	SAME
TECHNOLOGY	CMOS/SOS	CMOS/BULK & GALLIUM ARSENIDE	HIGHER DENSITY, LOWER POWER
RADIATION TOLERANT	LOW	HIGHER	IMPROVED RAD. TOLERANCE LEVELS
OTHER CHARACTERISTICS	–	VIRTUAL MEMORY	PROVIDES MEMORY PROTECT
TESTABILITY	OFF CHIP	ON CHIP	MORE RAPID FAULT DETECTION
MAIN MEMORY			
ACCESS TIME	350 NANO SEC	100 NANO SEC	BETTER PERFORMANCE
TECHNOLOGY	CMOS	CMOS TYPE	LOWER POWER
OTHER CHARACTERISTICS	–	BUILT IN ERROR DETECT & CORRECT	INCREASED RELIABILITY
MODULARITY/CHIP	64 KBITS	128 KBITS	HIGHER DENSITY

0663-126(1/2)(T) V83-0165-183(T)

Fig. 3-74 Processing Element Characteristics (Sheet 1 of 2)

	1990	2000	RATIONALE
AUX MEMORY			
TECHNOLOGY	BUBBLE MEMORY	BUBBLE MEMORY TYPE	
DENSITY (BITS/CHIP)	1.6×10^7	10^9	REDUCED PERIODICITY & FEATURE SIZE
ACCESS TIME MILLISECONDS	40	20	
TRANSFER RATE/SEC	70 KILOBITS	1.5 MEGABITS	

0663-126(2/2)(T) V83-0165-184(T)

Fig. 3-74 Processing Element Characteristics (Sheet 2 of 2)

ITEM	1990	COMMENTS	ITEM 2000	COMMENTS
• LANGUAGE	ADA 82	• 1982 VERSION – STANDARD LANGUAGE • AMERICAN NATIONAL STAND INST • EUROPE INTERN STAND ORGANIZ	ADA? (LATER VERSION) & LANGUAGES TO PERMIT ARTIF INTELLIG APPLICATIONS	• ADA IS EXPECTED TO BE UPDATED • ROBOTICS • KNOWLEDGE BASE SYSTEMS
FEATURES	– STRUCTURE DESIGN – TOP DOWN METHODOLOGY – STRONG DATA TYPE	• ATTRIBUTES FOR SOFTWARE RELIABILITY	• INCREASE EASE OF MAINTENANCE BASED ON GREATER ACCEPTANCE & UTILIZATION	• INCREASED SOFTWARE RELIABILITY BASED ON MATURITY V83-0165-156(T)

Fig. 3-75 Software (Sheet 1 of 3)

ITEM	1990	COMMENTS	2000	COMMENTS
• SUPPORT ENV.	Ada PROGRAM SUPPORT ENVIRONMENT	TOOLS FOR SUPPORTING Ada	• MORE TOOLS	
COMPILER	– AVAILABILITY	AVAILABILITY IS INCREASING (VAX 11-780, IBM ETC)		
	– EFFICIENCY	GOAL – HIGH		
LIBRARY	– STRONG PROGRAM LIBRARY	• REPOSITION OF SOFTWARE COMPONENTS		V83-0165-157(T)

Fig. 3-75 Software (Sheet 2 of 3)

ITEM	1990	COMMENTS	ITEM 2000	COMMENTS
• OPERATING SYSTEM	– VIRTUAL MEMORY – RESOURCE MANAGEMENT – REDUNDANCY MANAGEMENT		• STANDARD OPERATING SYSTEMS IMPLEMENTED IN SILICON	• STANDARD APPLICATION/OPERATING SYSTEM INTERFACES • STANDARD OPER SYSTEM BASED ON ADA TASKING & RUN TIME SYSTEM
• COMPATIBILITY WITH HARDWARE	HIGH WITH INSTRUCTION SET ARCHITECTURE MICROPROCESSORS	• FEATURES AS – SINGLE INSTRUCTION PROCEDURES/CALL – CONTEXT SWITCHING	• HIGH DENSITY CHIPS SUCH AS VHSIC WILL BE SUPPORTED BY ADA	• VHSIC CHIPS WILL BE PROGRAMMED IN HOL (ADA) V83-0165-158(T)

Fig. 3-75 Software (Sheet 3 of 3)

(as presented in Fig. 3-77) based upon Shuttle and Spacelab Technology incorporating a regenerative CO_2 removal system, a condensate water clean-up system to provide hygienic water and radiators for heat rejection; or a closed loop EC/LSS (as presented in Fig. 3-78), which takes the upgraded open loop and incorporates into it an oxygen regeneration system and a water reclamation system, both of which are currently being developed by NASA.

The cost of the closed loop has been estimated to be approximately $7.5 million more than the upgraded open loop. Five million dollars is required for completing the development of the Sabitier Reactor and the Solid Polymer H_2O Electrolysis system, both of which are required for O_2 regeneration. The remaining $2.5 M is required for completing the development of the water reclamation system [either the Vapor Compression Distillation (VCD) or the Thermoelectric Integrated Membrane Evaporation System (TIMES)]. A closed loop sized for three men weighs 206 kg more than the upgraded open loop; however, it requires 13.7 kg less per day in resupply (H_2O, O_2 and tankage). A trade-off of the two systems (illustrated in Fig. 3-79) shows that after approximately 15 days (45 mandays), the initial difference in launch weight is made up and, if the cost of resupply is estimated at $1200 per lb, the additional system cost is made up in another 208 days (624 man-days). The closed loop system becomes more cost effective than the upgraded open loop system in less than nine months of operation and therefore has been selected for the Space Station EC/LSS.

3.4.2 EC/LSS Areas for Further Study

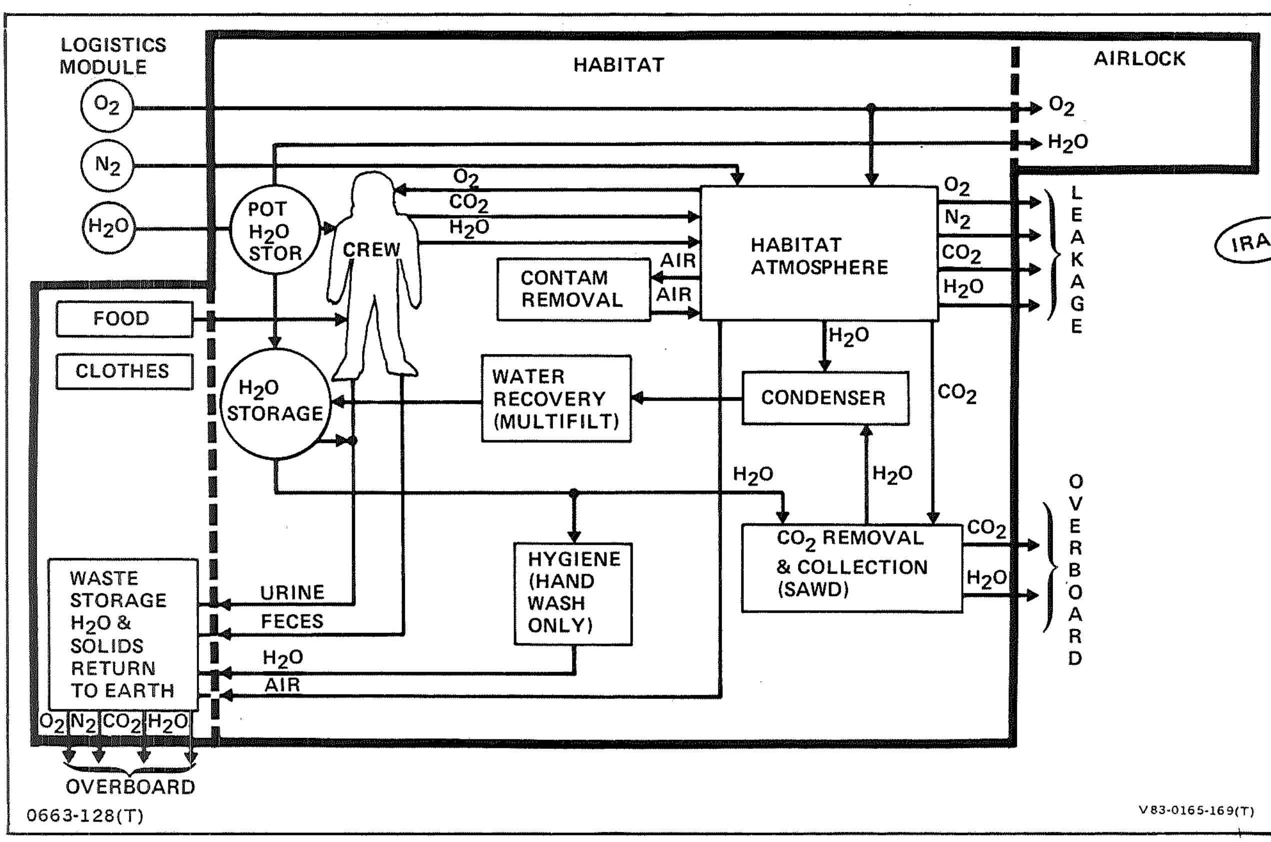
Areas recommended for further study related to the development of the Space Station EC/LSS are outlined in Fig. 3-80, and are discussed in more detail in the following paragraphs.

3.4.2.1 Total Cabin Pressure - The selection of cabin pressure level is related to Space Station missions. A 14.7 psia earthlike pressure is preferred by scientists for most on-board experiments. It also favorably effects EC/LSS design since lower ventilation rates are required for cooling than when operating at lower cabin pressures. Conversely, because current technology levels limit space suits to operate between 5 and 6 psia, it is necessary for the astronaut to "decompress" prior to EVAs from a 14.7 psia 2-gas atmosphere. Cabin pressures of between 8 and 12 psia are recommended for eliminating the need to decompress. Lower cabin pressure will also reduce cabin leakage rates. An 8-psia cabin pressure has been assumed in Grumman's baseline design.

TECHNOLOGY/OBJECTIVE/APPROACH	RISK	AVAILABLE DATE
• BUBBLE MEMORY – DEVELOP LIGHT WEIGHT HIGH DENSITY, LOW POWER BUBBLE MEMORY DEVICES COMPATIBLE WITH SPACE STATION ENVIRONMENT – DEVELOP MULTICHIP PACKAGES, FABRICATE AND TEST MEMORY DEVICES	LOW	1987
• FLAT PANEL DISPLAYS	LOW	1987
• NON-VOLATILE RANDOM ACCESS MEMORY (HIGH DENSITY-HIGH SPEED)	HIGH	1993
• SINGLE FIBER-SINGLE MODE FIBER OPTICS	LOW	1987
• HELMET MOUNTED DISPLAYS – WIDE FIELD-OF-VIEW VISORS	LOW	1987

0663-127(T) V83-0165-159(T)

Fig. 3-76 Technology Development Plan



IRAD

Fig. 3-77 Upgraded "Open" Loop EC/LSS

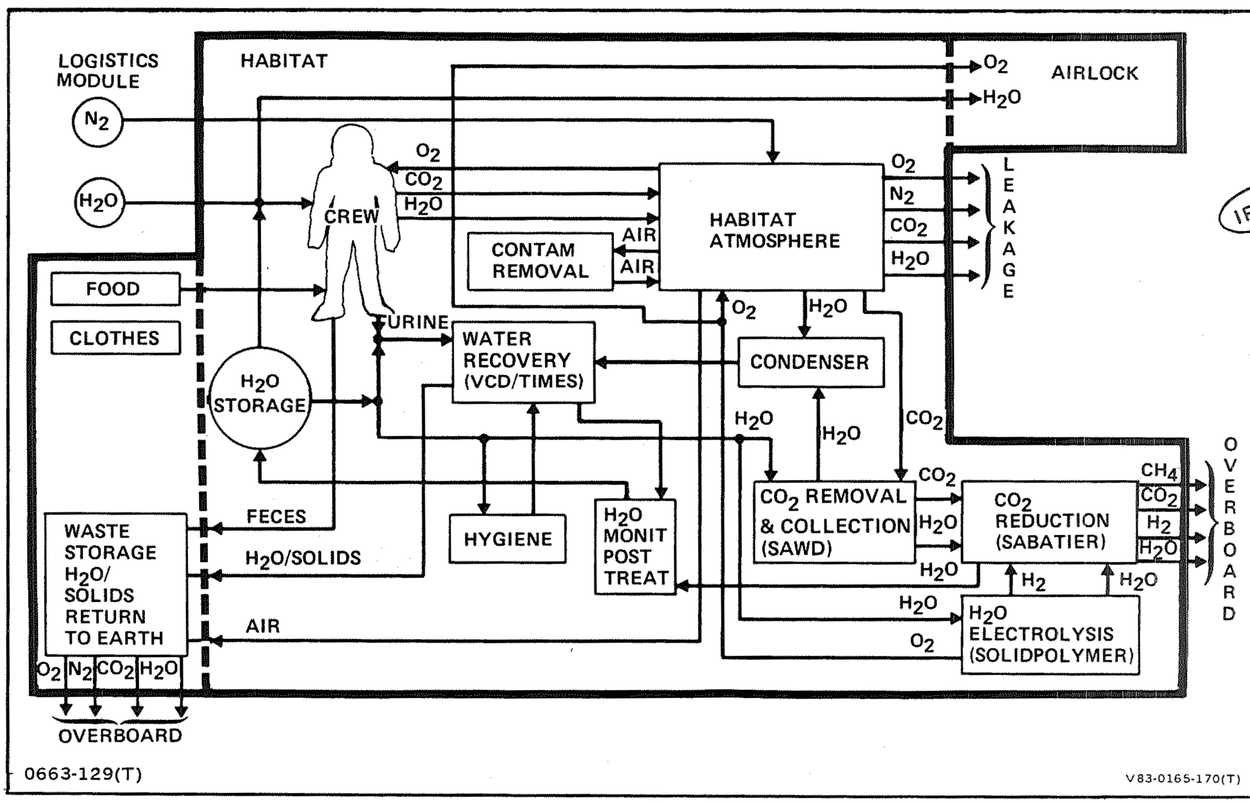


Fig. 3-78 Closed Loop EC/LSS

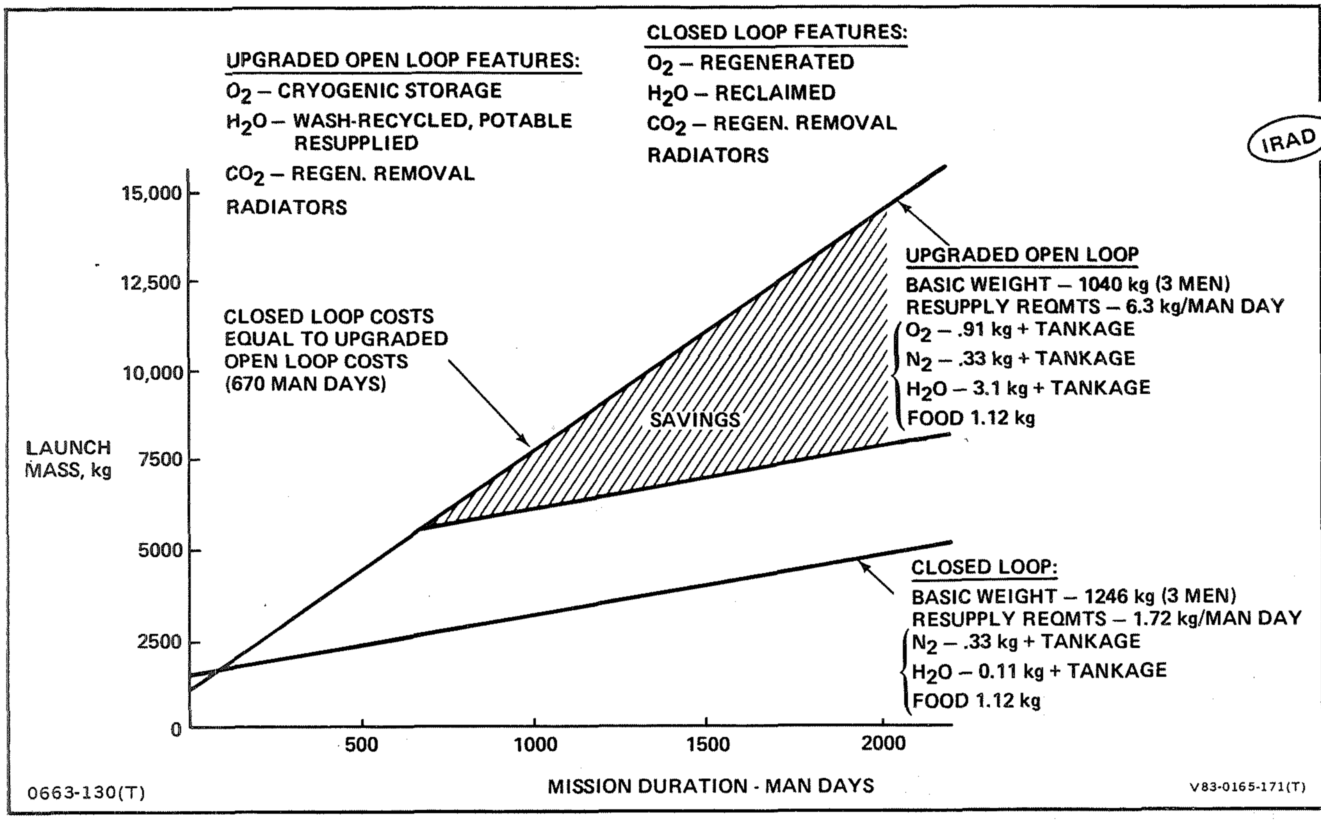


Fig. 3-79 EC/LSS Trade Study Closed Loop vs Upgraded Open Loop

AREA FOR FURTHER STUDY	PROBLEM
• TOTAL CABIN PRESSURE	• CABIN PRESSURE OF 14.7 PSIA EASES OPERATION & MAINTENANCE OF EQUIPMENT & PERFORMANCE OF EXPERIMENTS. CABIN PRESSURES BETWEEN 8 & 12 PSIA ELIMINATES NEED TO DECOMPRESS FOR EVAS AND REDUCES CABIN LEAKAGE (BASELINE ASSUMES AN 8 PSIA SYSTEM)
• SYSTEM REFINEMENT TO LIMIT OVERBOARD VENTING TO SPACE	• OVERBOARD DUMPING OF GASES FROM VARIOUS PROCESSES MAY BECOME A PROBLEM
• TRACE CONTAMINANT IDENTIFICATION & REMOVAL	• REDUCED CABIN LEAKAGE RATE & PROCESS GASES MAY CAUSE BUILDUP OF CONTAMINANTS NOT REMOVED BY CURRENT SYSTEMS.
• RECYCLING OF SOLID WASTES	• SOLID WASTE STORAGE WILL REQUIRE MORE VOLUME AS CREW SIZE & MISSION DURATIONS INCREASE & EXPERIMENTS GET MORE COMPLEX
• TWO PHASE THERMAL CONTROL	• LIQUID LOOP RADIATORS WILL INCREASE IN SIZE & WEIGHT AS CREW SIZE & THE NUMBER OF EXPERIMENTS INCREASE. A TWO PHASE THERMAL CONTROL SYSTEM OFFERS HIGHER HEAT FLUX CAPABILITY, THEREFORE REDUCED SIZE & WEIGHT.

0663-131(T) V83-0165-172(T)

IRAD

Fig. 3-80 EC/LSS Areas for Further Study

3.4.2.2 System Refinement to Eliminate Overboard Venting - The baseline design includes process equipment that requires overboard dumping of waste gases from the Sabatier reactor and from the waste storage system. This could become a problem area because of the potential leakage hazard involved and methods for eliminating or limiting overboard venting should be investigated.

3.4.2.3 Trace Contaminant Identification and Control - Long-term operation of a Space Station and reduced leakage rates will most likely introduce into the habitat trace contaminants that are generated from process equipment, experiments, furnishings and the crew, which heretofore have not been identifiable nor considered a problem on short-duration missions. Since these contaminants could become hazardous, methods for their identification and removal should be studied.

3.4.2.4 Recycling of Solid Wastes - Solid wastes management on the Space Station currently calls for the wastes to be compacted and stored in the logistic module until returned to earth during resupply. Increase in crew size, mission duration and such activities as biological experiments and medical treatment will tend to increase the quantity of solid wastes. Since these waste products will most likely occupy more volume than the original source, methods for the recycling of wastes should be investigated.

3.4.2.5 Two Phase Thermal Control - The thermal control section currently planned for the Space Station is based on Shuttle technology i.e., pumped liquid (Freon-21) loop with integral liquid loop radiator. However, advanced two-phase systems offering potentially significant advantages are being developed by NASA. One concept for an Instrument Management Subsystem will provide coolant to a group of scientific instruments by capillary pumping, thus eliminating the need for a mechanical pump. Another much larger two-phase system (Thermal Bus), is comprised of evaporating and condensing cold plates and heat exchangers that provide temperature control for all Space Station subsystems. Ultimately the station waste heat load is rejected to a Space Constructable Radiator comprised of individual heat pipe radiator elements.

This new technology offers many advantages:

- The heat pipe radiator can be constructed, maintained and expanded in orbit and provides much improved reliability against micrometeroid and debris damage

- The two-phase heat transport loop provides much higher heat flux capability, low equipment-to-coolant temperature difference, much lower coolant flow rate and a constant loop temperature not dependent on equipment placement or heat load.

Current development contracts are for ground prototype tests. If favorable, flight experiments will be required to demonstrate operation in zero-g, to demonstrate suitable assembly hardware and procedures to join the thermal systems of individual modules in orbit. With continued emphasis by NASA, a Thermal Control Section, with considerably higher performance than the Shuttle system will be space-qualified and available for application to Space Station.

3.4.3 EC/LSS Description

The EC/LSS baselined for the Space Station is a closed loop regenerable system (Fig. 3-78) designed to require a minimum resupply of consumables. A list of the subsystem's functional requirements is presented in Fig. 3-81. A brief description of each major section that comprises the EC/LSS is provided in the following paragraphs.

3.4.3.1 **Atmosphere Supply & Revitalization** - This section supplies the O_2 and N_2 that make up the cabin's 8-psia, 37.5% O_2/62.5% N_2 atmosphere. It also removes CO_2, removes trace contaminants, controls cabin pressure and composition and controls cabin humidity.

- N_2 Supply - Nitrogen is stored in the non-pressurized area of the logistics module in cryogenic form. The quantity required is a function of cabin leakage
- O_2 Supply - Oxygen is supplied to the cabin atmosphere from a Solid Polymer Water Electrolysis oxygen generator which electrolyzes water into O_2 and H_2. The H_2 is utilized in the CO_2 removal system
- CO_2 Removal - CO_2 is removed from the cabin air and concentrated by the Solid Amine Water Desorption process (SAWD) and then delivered to the Sabatier Reactor for reduction
- CO_2 Reduction - The Sabatier Reactor combines the CO_2 from the SAWD with H_2 from the electrolysis unit to form water and methane (CH_4). The methane is vented overboard and the water delivered to the water reclamation system

- **Trace Contaminant Control** - Cabin air is stripped of trace contaminants by passing it through a contaminant removal system that contains a combination of filters, sorbants and a catalytic oxidizer
- **Atmosphere Composition & Control** - Cabin pressure and composition is maintained by a series of valves, O_2 and N_2 sensors and a pressure regulator
- **Humidity Control** - Cabin air is drawn into a humidity control heat exchanger where excess moisture is condensed out and removed by a water separator. The condensate is delivered to the water reclamation system.

3.4.3.2 **Water Management** - This section stores and delivers water to be used for drinking, food preparation and hygiene. It also reclaims water from waste wash water, urine and condensate water. Virtually all of the water needed is reclaimed; however, a small quantity of make-up water may be required and a water tank is carried in the logistic module.

- **Water Recovery** - Two systems under consideration for the recovery of water are the Vapor Compression Distillation (VCD) and the Thermo Electric Integrated Membrane Evaporation System (TIMES).

3.4.3.3 **Waste Management** - The waste management section collects and disposes feces, urine and refuse. Feces are collected from the commode, then vacuum dried and stored until returned to earth. Urine is collected from the commode and urinals, then transferred to the water reclamation system. Refuse is compacted and stored until returned to earth.

3.4.3.4 **Thermal Control** - The function of the thermal control section is to remove and reject waste heat from the cabin and electronics/experiments. The system consists of a water loop that removes heat from the cabin air by an air-to-water heat exchanger. The electronics/experiments are cooled either by cold plates or by air-to-water heat exchangers. The water loop interfaces with the radiator Freon loop.

3.4.3.5 **Emergency Stores** - A 14-day supply of emergency stores has been provided. These stores are comprised of food, H_2O, gaseous O_2 and LiOH for CO_2 removal.

3.4.3.6 **Summary** - Figure 3-82 shows a weight, volume and power breakdown for each of the EC/LSS sections sized for a three-man crew. The data presented

SECTION	FUNCTIONS
• ATMOSPHERE SUPPLY & REVITALIZATION	• N_2 SUPPLY • O_2 SUPPLY & REGENERATION • CO_2 REMOVAL • TRACE CONTAMINANT REMOVAL • CONTROL ATMOS PRESS. & COMP • DEHUMIDIFICATION
• WATER MANAGEMENT	• WATER SUPPLY • WATER RECLAMATION
• WASTE MANAGEMENT	• SOLID WASTE COLLECTION
• THERMAL CONTROL 0663-132(T)	• CABIN ATMOSPHERE • ELECTRONICS/EXPERIMENTS • HEAT REJECTION V83-0165-176(T)

IRAD

Fig. 3-81 EC/LSS Functions

SECTION	WT, kg ①	VOL, M^3	PWR, kW
• ATMOSPHERE SUPPLY & REVITALIZATION	522	1.15	1.235
• WATER MANAGEMENT	337	0.86	0.175
• WASTE MANAGEMENT	80	0.43	
• THERMAL CONTROL	171 ②	0.30	1.400
• EMERGENCY STORES	333	0.71	–
• MISCELLANEOUS (PLUMBING, ETC)	136	0.28	–
TOTAL	1579	3.33	2.810

NOTES:

① WEIGHTS INCLUDE 50% REDUNDANCY FACTOR.

② THERMAL CONTROL DOES NOT INCLUDE WT ALLOWANCE FOR ELECTRONICS/EXPERIMENTS COLD PLATES OR RADIATORS.

0663-133(T) V83-0165-177(T)

IRAD

Fig. 3-82 EC/LSS System Weight, Volume, & Power Rqmts (Based on 3 Man Crew)

reflect actual component manufacturer's data (reference Hamilton Standard Parametric Data for Space Station) and, where data were not available, estimates were based on experience and judgment. Weights include recommended spares and expendables; however, an additional 50% redundancy factor has been included. A more detailed reliability analysis should be done as the program develops to arrive at a more complete and possibly lighter system.

3.4.4 EC/LSS Evolution

The basic three-man EC/LSS provided for the Initial Space Station and described in Subsection 3.4.3 serves as the building block for the nine-man EC/LSS on the Evolved Space Station (see Fig. 3-83). The basic subsystem is used for both the ISS and ESS because it incorporates technology that is expected to be available prior to 1986 that will "close" the loop (regenerate oxygen and reclaim water) and no other major advances in EC/LS technology are expected to occur prior to the time the design of the ESS is frozen. It is anticipated that use of the basic EC/LSS as a building block for both the ISS and the ESS represents the most practical and cost effective approach.

The ISS will carry a basic three-man EC/LSS that includes the spare parts and redundant systems needed to assure Space Station reliability requirements. Emergency stores capable of sustaining the three-man crew for 14 days will also be provided. Except for expendables and the water storage tank which are stored in the logistics module, all of the EC/LSS is located in the habit.

The addition of two three-man habitats to create the ESS requires increasing the capacity of the ISS's EC/LSS from three to nine men. This is accomplished by providing basic three-man EC/LSS modules in each of the additional three-man habitats. The modules are similar to the one located in the Initial Habitat except that no provision is made for spares or redundancy. Additional emergency stores for only three men for 14 days are provided. The reduced need for spares and redundancy is possible since the EC/LSS on the ISS already makes this provision. The reduced need for emergency stores is possible because operation of two of the three basic EC/LSS modules will support nine men in a degraded mode.

The EC/LSS to be provided on the Science Laboratories will be comprised of air circulation fans and air ducts which distribute air from the habitats through the laboratories.

Weights, volumes and power requirements for the EC/LSS on both the ISS and the ESS are presented in Fig. 3-84. The EC/LSS for the ESS weighs slightly more than twice the EC/LSS for the ISS, though it provides three-times the capacity because of the reduced need for spares, redundant systems and emergency stores.

3.5 GUIDANCE, NAVIGATION & CONTROL (GN&C)

3.5.1 GN&C Trade Issues

GN&C for the Space Station primarily encompasses three areas: attitude control, attitude determination and orbit control. Figure 3-85 summarizes the general requirements and parameters for the basic station configurations. A nominal altitude of 370 km was chosen for design tradeoffs. A local vertical (station longitudinal axis) perpendicular to the orbit plane (solar array gimbal axis) orientation was specified by the initial configuration designs, primarily to support payloads and solar array requirements. Control accuracy of 5 deg was chosen to simplify control; with precision pointing and control provided by the individual payloads as required.

The following guidelines are used in the subsequent system trades:

- Minimize the need for expendables
- Emphasize proven techniques
- Emphasize existing components
- Minimize Contamination Effects
- Emphasize design analysis without the Orbiter (assume Orbiter augments control when attached).

3.5.1.1 Attitude & Velocity Control Issues - The major GN&C design issues for the Space Station are summarized in Fig. 3-86. The first three issues are closely related. Selection of attitude orientation is initially made based on mission requirements (e.g., payload pointing) and required support of other subsystems such as solar arrays. Superimposed on these requirements is the desire to minimize environmental disturbance torques to reduce actuator demands and propellant requirements.

The major disturbance sources at the selected altitude are gravity gradient and aerodynamic torques. In addition, there are internal disturbances caused by mass motion of fluids and equipment. The latter effect can be minimized to some extent by self-compensation and careful placement of equipment. Gravity gradient torques

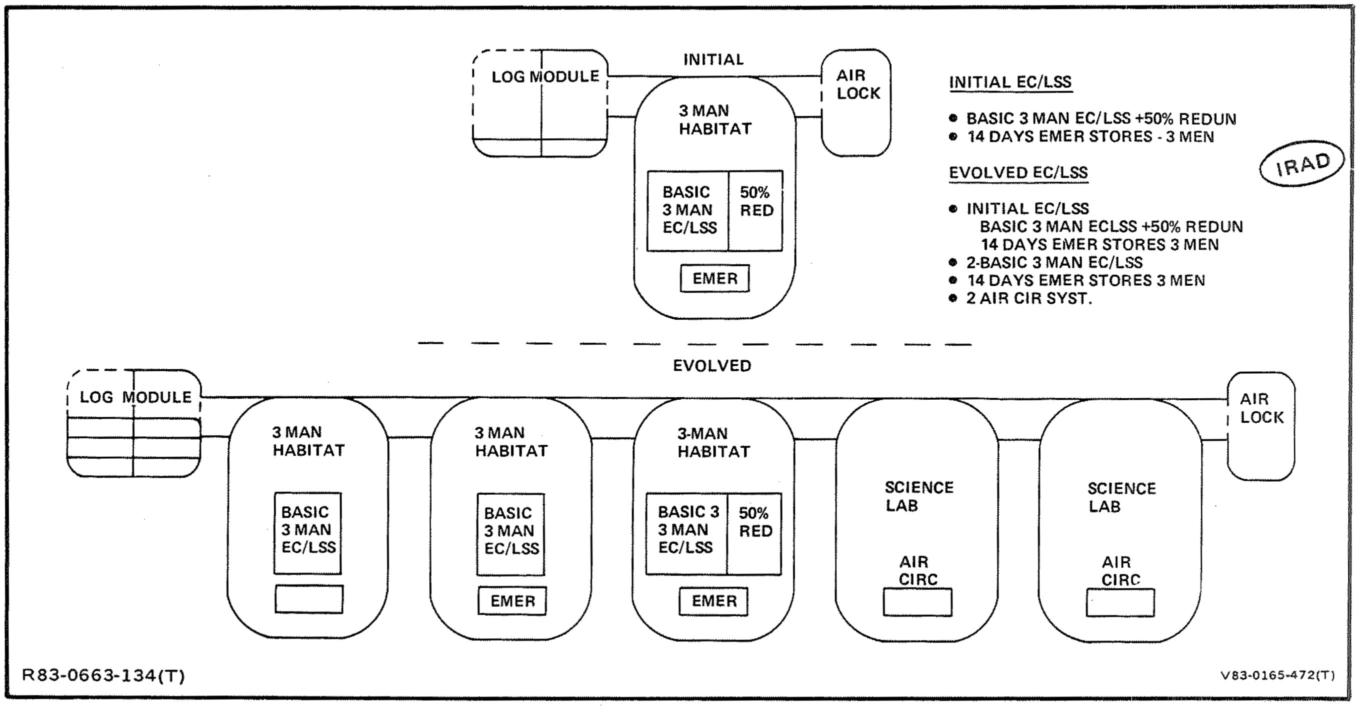


Fig. 3-83 EC/LSS Evolution

INITIAL	WT kg	VOL M³	PWR-KW
• BASIC 3 MAN EC/LSS +50% REDUNDANCY	1246	2.62	
• EMERGENCY SUPPLIES (3 MEN 14 DAYS)	333	.71	
TOTAL	1579	3.33	2.8
EVOLVED			
• INITIAL EC/LSS			
– BASIC 3 MAN EC/LSS +40% REDUNDANCY	1296	2.62	
– EMERGENCY SUPPLIES (3 MEN 14 DAYS)	333	.71	
• 2 BASIC 3 MAN EC/LSS	1480	3.65	
• EMERGENCY SUPPLIES (3 MEN 14 DAYS)	333	.71	
• 2 LAB ATMOS CIRC SYS	95	.57	
TOTAL	3937	8.26	9.0

IRAD

R83-0663-135(T) V83-0165-151(T)

Fig. 3-84 EC/LSS Evolution Weight, Volume, & Power Requirements are a function of cross products of inertia and attitude relative to local vertical. Aerodynamic torques are a function of the drag cross section area, the altitude and the cg location.

Our initial approach is to select the nominal local vertical (L/V) perpendicular to orbit plane (POP) orientation to support the many earth-oriented payloads and to estimate disturbance torques and momentum buildup. The capabilities of existing actuators and unloading techniques can then be evaluated, leading to trades of attitude orientation vs system weight.

CONFIGURATION	ALTITUDE (n mi)	INCLIN (deg)	ORIENTATION	FEATURES
1. MANNED SPACE STATION	370 ± 10	28.5	L/V – POP ± 5°	• MUST INCLUDE PROVISIONS FOR CELESTIAL OBSERVATIONS • FREQUENT SHUTTLE VISITS • CONTROL WITH TIP ATTACHED.
2. TENDED INDUSTRIAL PLATFORM	370 ± 10	28.5	L/V – POP ± TBD	• LOW-g PAYLOADS • PERIODIC MANEUVER CAPABILITY TO & FROM 1.
3. TENDED POLAR PLATFORM	370 ± 10	97	L/V – POP ± 5°	• SUN-SYNCHRONOUS ORBIT • EARTH-ORIENTED IPS (6) • INFREQUENT SHUTTLE VISITS.

0663-136(T) V83-0165-231(T)

Fig. 3-85 GN&C Requirements

ISSUE	RESOLUTION	STUDY APPROACH
ATTITUDE ORIENTATION	ESTABLISHED BY MIN AERO, MIN GRAVITY GRADIENT, & EARTH-ORIENTED PAYLOADS	SELECTED NOMINAL L/V-POP ATTITUDE, INITIALLY
ACS ACTUATORS	USE AVAILABLE CMGs; NUMBER TBD	ESTIMATE PEAK DISTURBANCE & MOMENTUM BUILDUP
UNLOADING TECHNIQUE	USE ATTITUDE STEERING WITH SOME JET UNLOADING	EVALUATE COUNTERACTING DISTURBANCES; USE ORBIT ADJUST MANEUVERS
ORBIT KEEPING TECHNIQUE	SCHEDULED DRAG MAKEUP THRUSTING	COMPUTE PROPELLANT FOR WORST-CASE DRAG, CONTINUOUS THRUSTING
PROPELLANT	NOT RESOLVED, EMPHASIZE LOW CONTAMINATION	EVALUATE ALTERNATE PROPELLANTS

0663-137(T) V83-0165-232(T)

Fig. 3-86 Attitude & Velocity Control Issues

3.5.1.2 Disturbance Torque Summary - The magnitudes of the dominant environmental disturbances are presented in Fig. 3-87 for the basic configurations at the nominal 370 km altitude. The X-axis is earth-oriented with the Y-axis perpendicular to the orbit plane. The values are computed for one particular position of the gimbaled payloads and the solar arrays. The total torque shown is the sum of the aero and gravity gradient torques. The evolved Space Station and Tended Polar Platform configurations have some very sizable disturbances which are discussed in the next subsection.

3.5.1.3 Configurations Comparison - Figure 3-88 compares the Space Station Configurations qualitatively in terms of aero drag profile and the distribution of the major elements (e.g., solar arrays, surrogate payload bay, habitability modules, etc). The approximate cg location is indicated which permits one to grossly assess the dominant sources of aero and gravity gradient disturbances and their variation due to their changing positions.

In all cases except the Tended Industrial Platform configuration, there are off-axis masses that seriously effect the inertial symmetry, causing significant gravity gradient disturbance torques.

3.5.1.4 Bias Momentum Per Orbit - The total angular momentum accumulated in an orbit as a result of the aerodynamic and gravity gradient disturbance torques are presented in Fig. 3-89. Gravity gradient torque is the dominant effect in two configurations because of the non-optimum inertia characteristics.

A Skylab cmg has a maximum torque capability of 165 newton-meter, more than an order of magnitude higher than the maximum required torque. However, the momentum storage capacity of 3118 N-m-sec is only suitable for the requirement of the Tended Industrial Platform configuration. Frequent unloading is called for at this momentum buildup level. The torques required are too large for a conventional magnetic unloading system. Initial calculations of the propellant quantity required for monopropellant and bipropellant systems appear prohibitive, suggesting configuration and/or flight control modifications. It is urged that attitude steering similar to Skylab and juidcious operation of gimbaled elements, such as the Instrument Pointing Systems, can be used to significantly reduce the momentum storage requirements to an acceptable level. Attitude steering involves programming the

CONFIG	DRAG FORCE (NEWTON)	AERO DIST TORQUE (N-M)			GRAV GRAD DIST TORQUE (N-M)			TOTAL BIAS TORQUES (N-M)		
		T_X	T_Y	T_Z	T_X	T_Y	T_Z	T_X	T_Y	T_Z
TENDED POLAR PLATFORM	0.156	0.20	0.19	0	0	3.12	1.60	0.20	3.31	1.60
MANNED SPACE STA (EVOLVED)	0.600	0.03	8.81	0	0	1.19	0.49	0.03	10.01	0.49
TENDED INDUSTRIAL PLATFORM (AT 200 N MI)	0.013 - 0.249	0	0.008	0	0.20	0.10	0.10	0.20	0.11	0.10

0663-138(T) V83-0165-233(T)

Fig. 3-87 Disturbance Torque Summary

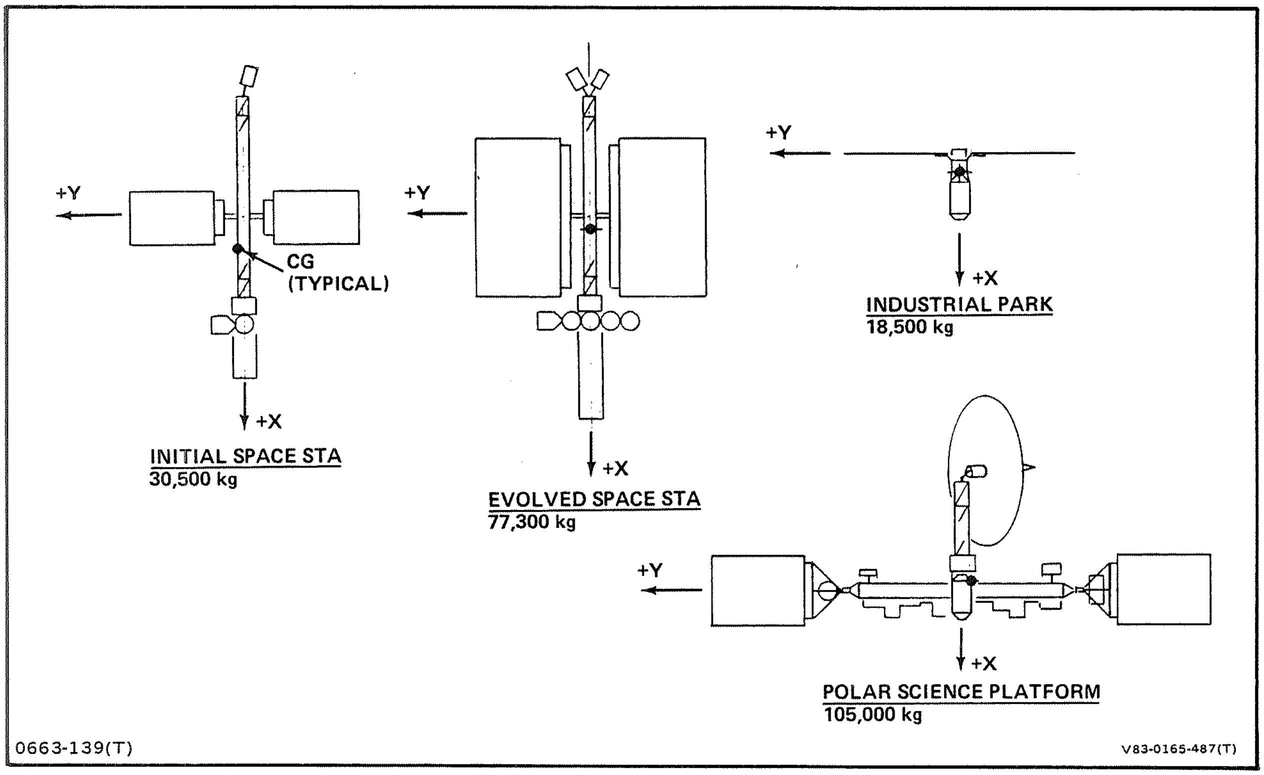


Fig. 3-88 Configuration Comparisons Drag Areas & Mass

CONFIG	ANGULAR MOMENTUM (NEWTON-METER-SEC)	PRIMARY CAUSE
TENDED POLAR PLATFORM	18,240	GRAV GRAD TORQUE ABOUT Y AXIS
MANNED SPACE STATION	55,160	AERO DRAG TORQUE ABOUT Y AXIS
TENDED INDUSTRIAL PLATFORM	1120	GRAV GRAD TORQUE ABOUT X AXIS
0663-140(T)		V83-0165-234(T)

Fig. 3-89 Bias Momentum Per Orbit

attitude of the Space Station as a function of time to minimize momentum accumulation using a math model of the disturbance environment and vehicle characteristics.

It is realistically assumed that this technique can reduce the momentum buildup per orbit to 10% of these values. These reduced values have been used in the subsequent propellant trade.

3.5.1.5 Propellant Consumption Estimates - Propellant estimates were generated for each configuration using three candidate propellants: cold gas (I_{sp} = 60 sec), monopropellant (I_{sp} = 150 sec) and bipropellant (I_{sp} = 280 sec) and are summarized in Fig. 3-90. The momentum per orbit used for these calculations is 10% of the total bias momentum, as discussed previously. The momentum arms used assume their capability on all axes which may require the addition of a "thruster arm" to the configurations.

The Tended Polar Platform is assumed to be resupplied at two-year intervals, whereas the manned Space Station and Tended Industrial Platform can be resupplied at 90-day intervals. The latter two require 711 kg and 32 kg respectively, of cold gas which are acceptable amounts. A two-year requirement of 430 kg of biopropellant fuel for the Tended Polar Platform is marginally acceptable. However, the contaminating properties of biopropellant are considered undesirable for the optical payload.

The results of this tradeoff suggest that the Tended Polar Platform be moved to a higher orbital altitude at the beginning of each two-year cycle. The objective is to achieve conditions which will permit the use of a cold gas system. The use of cold gas has already been shown to be feasible for the manned Space Station and Tended Industrial Platform.

CONFIG	NET** BIAS MOMENTUM PER ORBIT (N-M-S)	AVAIL MOMENT ARM (METER)	GAS CONSUMPTION PER ORBIT/PER DAY (kg)		
			I_{sp} = 60 sec COLD GAS	I_{sp} = 150 MONOPROP	I_{sp} = 280 BIPROP
TENDED POLAR PLATFORM	1824	18.3*	0.18 / 2.86	0.07 / 1.1	0.036 / 0.59
MANNED SPACE STA	5516	18.3*	0.5 / 7.9	.23 / 3.6	0.11 / 1.82
TENDED INDUSTRIAL PLATFORM	112	9.15	0.023 / 0.36	0.009 / 0.14	0.005 / 0.09

*MAY REQUIRE "THRUSTER ARM"
**ASSUMES ONLY 10% OF TOTAL BIAS MOMENTUM MUST BE UNLOADED BY JETS
0663-141(T) V83-0165-235(T)

Fig. 3-90 Propellant Consumption Estimates

3.5.1.6 **Conclusions** - The basic design approach we have selected features on-board autonomous sensors and actuators for attitude control. Precision attitude determination is provided when required by payload sensors. Orbit control is based on navigation via the Global Positioning System with periodic orbit adjust maneuvers scheduled on a non-interference basis with mission operations.

Future trade studies should address the following:

- Preferred attitude orientations vs configuration design
- Orbit altitude vs system weight and complexity
- Integrated magnetic torquers vs reaction control unloading
- LO_2-LH_2 propellants integrated with EC/LSS and EPS requirements.

3.5.2 Guidance, Navigation & Control Subsystem Description

Figure 3-91 shows the major GN&C System components for all Space Station configurations. The primary attitude control sensors are three-axis rate integrating gyros with attitude updates provided by earth, sun and star sensors. The magnetometers are used by the magnetic torquing system. A group of three 2-degree-of-freedom control moment gyros provide redundant control torques with cold gas and magnetic unloading. Payload-mounted or internal fine error sensors provide precision control inputs as required. The GN&C electronics provides all computations required for autonomous attitude and velocity control with minimum ground station involvement.

The subsystem components required for attitude and velocity control are specified in Fig. 3-92 for the baseline concept. Typical unit weights and power are given and a preliminary recommendation is made for the quantity required which will

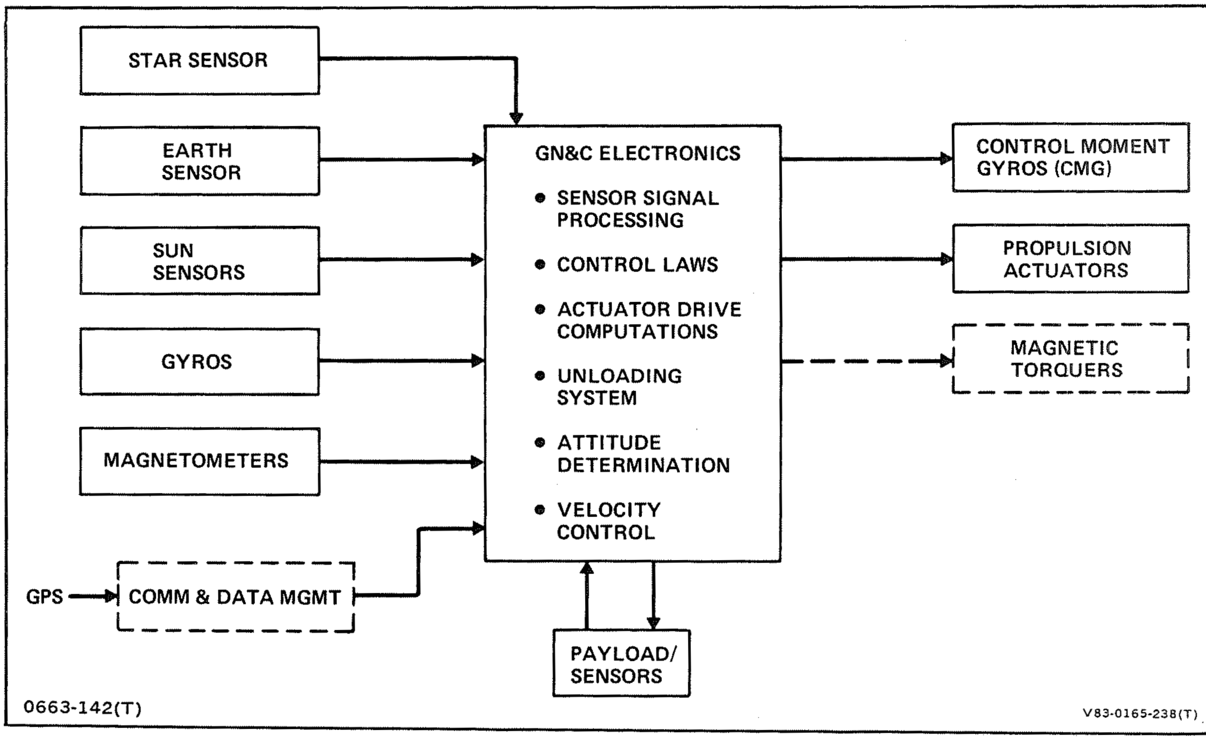


Fig. 3-91 GN&C Block Diagram

COMPONENT	UNIT WT, kg	UNIT POWER, W	NO. REQD
STAR SENSOR (TRACKER)	7.7	18	3
SUN SENSOR (ANGLE)	0.9	0.7	3
EARTH SENSOR (CONICAL SCAN)	1.8	6	2
GYROS(DRIRU)	13.2	17	2
CMG (DOUBLE GIMBAL)	212	80 (170 PEAK)	3
MAGNETOMETER (3-AXIS)	0.45	–	2
MAGNETIC TORQUERS	TBD	TBD	–

0663-143(T) V83-0165-236(T)

Fig. 3-92 GN&C Subsystem Components Characteristics provide redundancy and maintainability while operational. The specification of specific thrusters and propellant tanks require further configuration development before a selection can be made.

3.5.3 GN&C Subsystem Evolution

The specific equipment (e.g., the sensors and actuators) required for the initial and subsequent Space Station is essentially available presently. The one possible exception to this is the hardware elements to provide a low-g platform for the materials processing missions. These include dynamic isolation systems to overcome the forces resulting from control actuators which requires development.

The key enabling technology for GN&C concerns the analytical and experimental validation of the subsystem design for a modular, incremental buildup of a Space Station. In addition to the development of the analytical tools to be used for configuration design, integrated modular assemblies of GN&C subsystem equipment must be developed which permit the safe, reliable buildup and growth of the Space Station. Such a "construction control package" concept is required using the available hardware elements already identified. Smooth transitions of control authority must be insured as a configuration evolves, and also during normal maintenance operations.

Although the basic technology required for the Space Station program is available, there are GN&C technology improvements and refinements that can be developed on the Space Station. The availability of a manned laboratory in orbit offers the possibility of developing improvements in GN&C equipment. For example, long-life rotating components applying the principles of tribology can be demonstrated in the zero-g environment under close observation leading to improved reactor wheels, gyros and other components. These improvements may be incorporated in Space Station updates leading to reduced maintenance and improved performance. The general development of more readily maintainable hardware is also suggested by this on-orbit test and evaluation capability.

The Space Station affords the opportunity to develop more accurate models for thruster plume dynamics and related design parameters which may form a basis for more efficient thruster designs. The feasibility of higher I_{sp} thrusters to prolong station life as well as reducing detrimental effects on Space Station payloads and equipment may be demonstrated.

One further area of improvement is for the development of clean, high I_{sp} thrusters of relatively low thrust 0 to 25 lbf which can be readily integrated into evolving Space Station configurations. Precise control torques and high system reliability are the goals of such development.

A number of state-of-the-art improvements are desirable that could greatly enhance the efficiency and performance of the GN&C subsystem. An integrated energy storage/attitude control device could significantly reduce the combined weight of the GN&C and electrical power subsystems. Combined electrical energy and angular momentum management using rotating mass components is a possible approach that should be developed further.

Another area of improvement concerns the development of high performance magnetic torquers for momentum unloading. Both discrete components and integrated large cross sectional area concepts should be evaluated. Possible interactions with other equipment (e.g., electromagnetic interference) should be investigated.

4 - EVOLUTION

The space station system/subsystem conceptual architecture designs and evolution discussed in Sections 2 and 3 provide a perspective amongst objectives, technology, risk, and costs. This was accomplished by placing particular emphasis on the following areas:

- Initial-evolved system operational requirements and associated differentials
- Present-future technology base
- Architecture based on modular design, system/subsystem levels
- Combination of near and long-term planning.

The ISS core architecture is predominantly near-term technology (1983-1986) which minimizes technological risk, provides for early availability by utilizing well known and established design techniques, costs less, and exhibits higher reliability. This avoids the usual difficulties encountered with use of new sophisticated technology such as:

- Lengthy development, testing and high costs
- Cost overruns
- Delayed schedules.

The parametric tradeoffs and analyses provide optimization towards the ISS early missions. However, sensitivity to future requirements have been recognized and identified herein (Subsection 4.2.1). In addition, the independent modular design ensures capacity and ease of growth, ability to benefit from technological advances, and with tight interface controls will minimize future implementation impacts. Future growth implementation impacts are further reduced by consideration of the ESS requirements in the ISS architecture design. In order to minimize the impact of uncertainties, iterative mission/technology assessments and forecasts must be performed throughout the early phases of the space station program.

4.1 MODULAR COMMONALITY BASE

The space station system/subsystem conceptual architecture designs provide a base of common modules which permit adequate flexibility in developing a variety of configurations. This modular commonality base is summarized in Fig. 4-1.

4.1.1 Space Station

The incremental growth stages and associated configurations are described and discussed in Section 2. The evolved configuration is not restricted to parallel additions but can be assembled as shown in Fig. 4-2. The arrangement also indicates possible growth beyond the evolved configuration.

4.1.2 Subsystems

The evolved space station complex will consist of the main space station assembly (i.e., attached modules) and associated cluster (Free Flyers, co-orbiting, etc). A multiplicity of common subsystem modules and elements will be distributed within this complex. Unique adapters/interface elements for any of the subsystem applications (e.g., EPS for Tended Industrial Platform) will be kept to a minimum since early planning should consider all applications of each subsystem module. As a result each subsystem will require an evaluation and summary of all applications directed towards determining design and performance requirements to be incorporated in the pre-ISS Enabling Technology phase.

4.2 ENABLING TECHNOLOGY REQUIREMENTS

The total enabling technology requirements for the ISS must include the following:

- Subsystem performance, design features
- Awareness of subsystem sensitivities to advanced technology and growth
- System design/integration commonality needs.

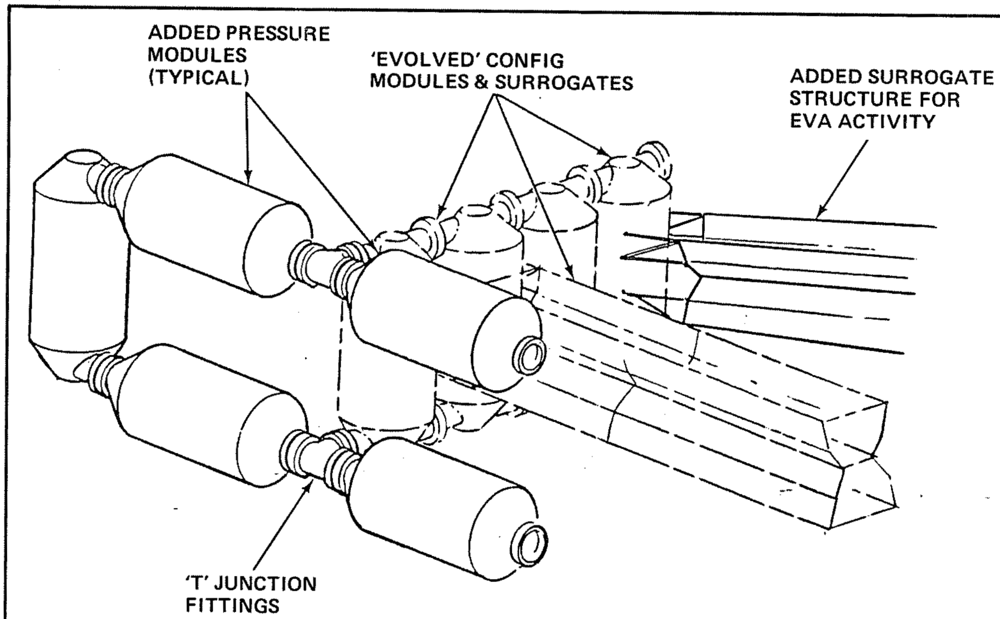
Enabling technology encompasses the complete spectrum from available off-the-shelf hardware to research. Figure 4-3 summarizes subsystem ISS enabling technology requirements based on their architecture concept presented in Section 3. Basically the 1986 technology base and associated design techniques will be adequate for the ISS. Automation development is required for all subsystems (refer to Subsection 4.2.2).

- SPACE STATION
  - PRESSURIZED CORE MODULE
  - SURROGATE BAY
  - HANDLING EQUIPMENT (RMS, HPA, MMU, WRU, OCP)
  - SUPPORT MAST STRUCTURE
  - SUBSYSTEM PALLET
- EPS
  - SOLAR ARRAY PANELS
  - GIMBALLING
  - POWER CONDITIONING & CONVERSION COMPONENTS
  - POWER DISTRIBUTION COMPONENTS
  - ENERGY STORAGE
- DMS
  - PROCESSORS
  - MAIN MEMORY MODULES
  - DISPLAYS
  - NETWORK COMPONENTS INCLUDING FIBRE OPTICS
  - MASS MEMORY DATA BASE
- COMMUNICATIONS & TRACKING
  - ANTENNAS, SWITCHES
  - RECEIVER/TRANSMITTER ELEMENTS
  - SIGNAL PROCESSOR ELEMENTS
- EC/LSS
  - COMPONENTS/ELEMENTS (ATMOS SUPPLY, WATER/WASTE MGT, ETC)
- GN&C
  - COMPONENTS/ELEMENTS (SENSORS, GYROS, ACTUATORS, ETC)

0663-144(T) V83-0165-477(T)

IRAD

Fig. 4-1 Evolution Modular Commonality Base



0663-145(T) V83-0165-478(T)

IRAD

Fig. 4-2 Space Station Growth Beyond Evolved Configuration

	BASELINE	ENABLING TECHNOLOGY
EPS	• SOLAR ARRAY • NiH_2 BATTERIES • 180V DIST	• THIN CELL & HIGHER EFFICIENCY 2 • CELL MFG PROCESSES 2 • BATTERY DEVELOPMENT 2 • HI VOLT COMPONENT DEVELOPMENT 1
DMS	• Ada • FIBRE OPTICS • CMOS MAIN MEMORY WITH B/U BATTERY • BUBBLE AUX MEMORY	• MEETING EXISTING Ada SCHEDULE 1 • LOW LOSS COUPLERS 1 • DEV HIGHER DENSITIES 2 • SPACE QUALIFICATION & HIGHER DENSITIES 2
COMM & TRKNG	• S, K_u BAND SUBSYSTEMS • DISH, OMNI ANTENNAS • TDRS • SIMOP	• MODULATION/CODING/BANDWIDTH 1 • DES/DEV FOR APPLICATION 1 • ACQUISITION/TRACKING/DATA RATE 1 • RFI PROTECTION 1
EC/LSS	CLOSED LOOP	EXISTING HARDWARE WITH MODIFICATIONS 1
GN&C	ATTITUDE CONTROL VELOCITY CONTROL STABILIZATION SENSORS	EXISTING HARDWARE WITH MODIFICATIONS 1

IRAD

1 (1983-1986) TECHNOLOGY BASE & DESIGN TECHNIQUES ADEQUATE
2 TECHNOLOGY ADVANCE REQUIRED

0663-146(T) V83-0165-475(T)

Fig. 4-3 Subsystem Enabling Technology Requirements

4.2.1 Sensitivities

Subsystem sensitivities to growth and advance technology are summarized in Fig. 4-4. The EC/LSS is not identified because it benefits from maturing technology and its implementation is finite and piecemeal during the space station evolutionary stages. Sensitivities are discrete for individual subsystems and also involved in intersubsystem actions (i.e., effects on one subsystem impacting other subsystems).

4.2.2 System Design/Integration

The two major drivers for system design/integration features are commonality and autonomy automation. These two areas strongly influence growth implementation and uncertainty impacts for the space station. These impacts will be minimized by providing unique growth provisions in the ISS and/or possible earlier technology development for installation into the ISS. The major time-period for this effort is prior to ISS (i.e., must be part of ISS enabling technology).

Some of the commonality items required to be implemented by all subsystems prior to ISS are:

11/2/1

- Lowest modular element and maintenance category
  - Organizational level, replacement
  - Depot (lab, earth), repair
- Packaging
  - Standard modular design
- Installation/removal techniques
  - Tools, fasteners, mounting
- Connectors
- Wiring/bundling/routing.

- EPS
  - INCREASED POWER, LARGER ARRAYS INTERFERENCES WITH ASTROPHYSICS VIEWING, EVA & COMMUNICATIONS/TRACKING
  - GAs, NEW MODULES/ELEMENTS – SPACE STATION MODIFICATION, DOWNTIME/REDUCED OPERATION
  - REGEN FUEL CELLS, SAME IMPACTS AS WITH GAs
- COMMUNICATIONS/TRACKING
  - DIFFERENT OPER FREQUENCIES, MODIFICATIONS/NEW RCVRS, XMTRS, ANTENNAS
  - TDAS COMPATIBILITY
  - LASERS, NEW ADDITIONAL EQP & TIGHTER TRACKING
  - MULTIPLE ACCESS/CODING, MODULATION, SIGNAL PROCESSING
- DMS
  - ADDED OPERATING SYSTEM RUN TIME REQMNTS
  - INCREASED SUBSYSTEMS, ADDITIONAL PROCESSORS, SPACE/VOLUME CONSTRAINTS
  - ADDED THROUGHPUT AND BANDWIDTH COMPATIBILITY
- GN&C
  - INCREASED SPACE STATION STRUCTURE, MASS – STABILITY, DOCKING MANEUVERS, THRUSTER LOCATIONS

0663-147(T) V83-0165-474(T)

IRAD

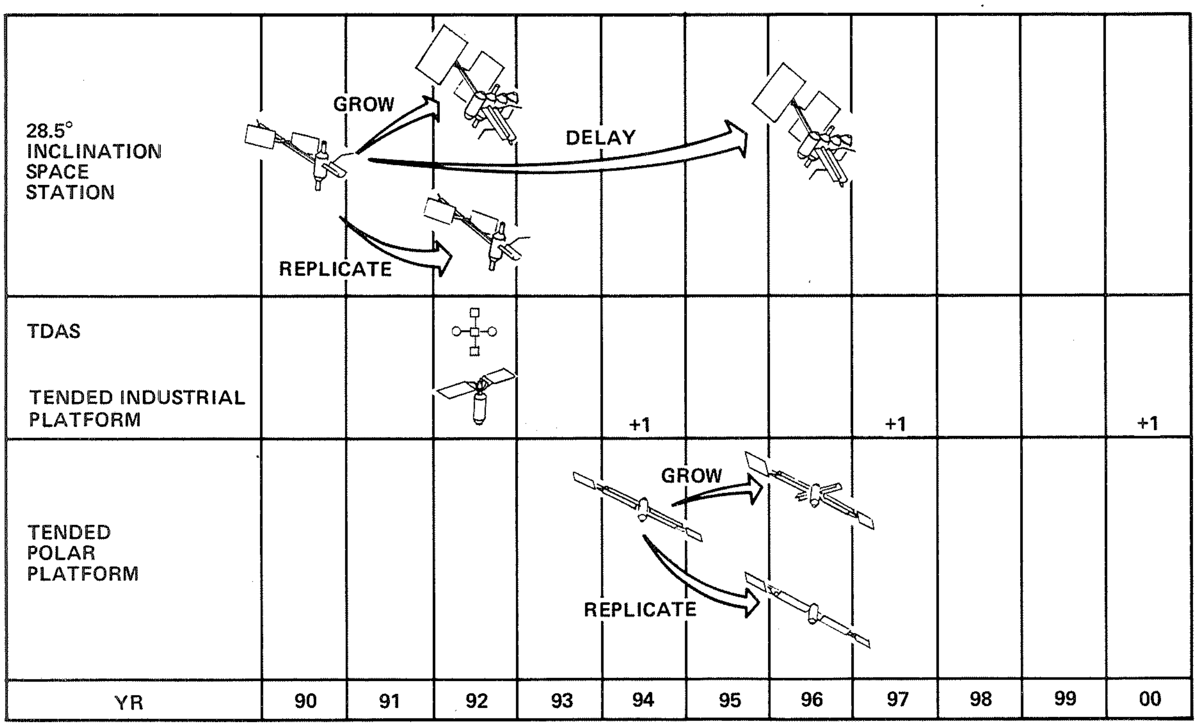
Fig. 4-4 Subsystem Growth/Advanced Technology Sensitivities

The items listed above are considered minimum and most appropriate for effective incremental growth steps by replacement with new and/or modified elements. The continuing rapid advances in electrical/electronic technology are for the most part characterized by smaller volume, less weight, and lower power. In addition, implementation and uncertainty impacts would be further minimized since doctrine, procedures and specifications would be established well in advance.

The automation aspect of Autonomy is a more complex task. The initial DMS will be designed to accommodate the automatic control, monitoring, diagnostic operations for the subsystems. The basic requirement here is that each subsystem element possess the necessary internal failure/malfunction detection, and standardization of signals to/from the DMS. In addition overall space station autonomy is viewed as also evolving in time. In particular, the extent of the ISS equipment internal automated design features will probably be moderate and evolve to maximum for the ESS.

4.3 EVOLUTION PLAN

Various growth options for the evolved configuration appear in Fig. 4-5. A more detailed evolution plan is presented in Volume II - Book 4. Many sequence and time interval combinations are possible. However, in general, all will basically consist of the following three phases.



V83-0165-494(T)

Fig. 4-5 Growth Options

4.3.1 Phase I 1990-1992

- First STS Launch
  - Three Men-Core Module
  - External Subsystem Pallet
  - Surrogate
- Second STS Launch
  - HPA, RMS
  - Logistics/Personnel
  - Celestial and Terrestial Observatory
- Basic Space Station Module Assembly Activities Performed

- System/subsystem operational verification conducted
- Tended Industrial Platform into orbit and Space Station interoperation verified
- Initial mission operations performed
  - Test Facility
  - SAT Services
  - Observatory
  - Initial Harbor.

4.3.2 Phase II 1992-1996

- Confidence achieved by operational experience in previous stage
- TDAS launched and system operational tests performed
- Space station - TDAS interoperation verified
- Additional pressurized modules and surrogates launched
  - Operational checkout of growth modules
- Crew size increase
- Additional Tended Industrial Platforms
  - Operational checkout
  - Operation
- Mode I of Tended Polar Platform launched
  - Assembly and checkout
  - Operation
- First reusable OTV available
  - Assembly and checkout
  - Operation
- Increased space station operational experience
  - Test facility
  - Transport harbor
  - Satellite services
  - Observatories
  - Industrial park.

4.3.3 Phase III 1996-2000

- Mission support at maximum level
- Mode II of Tended Polar Platform launched and put into operation
- Additional Tended Industrial Platforms
- Crew size 9-12
- Space station operational capacity at maximum
- Maximum autonomy achieved
- Space station ready for additional growth
- Prepared for space station cluster about earth (e.g., additional space stations).

GRUMMAN

0663-83
0188-83