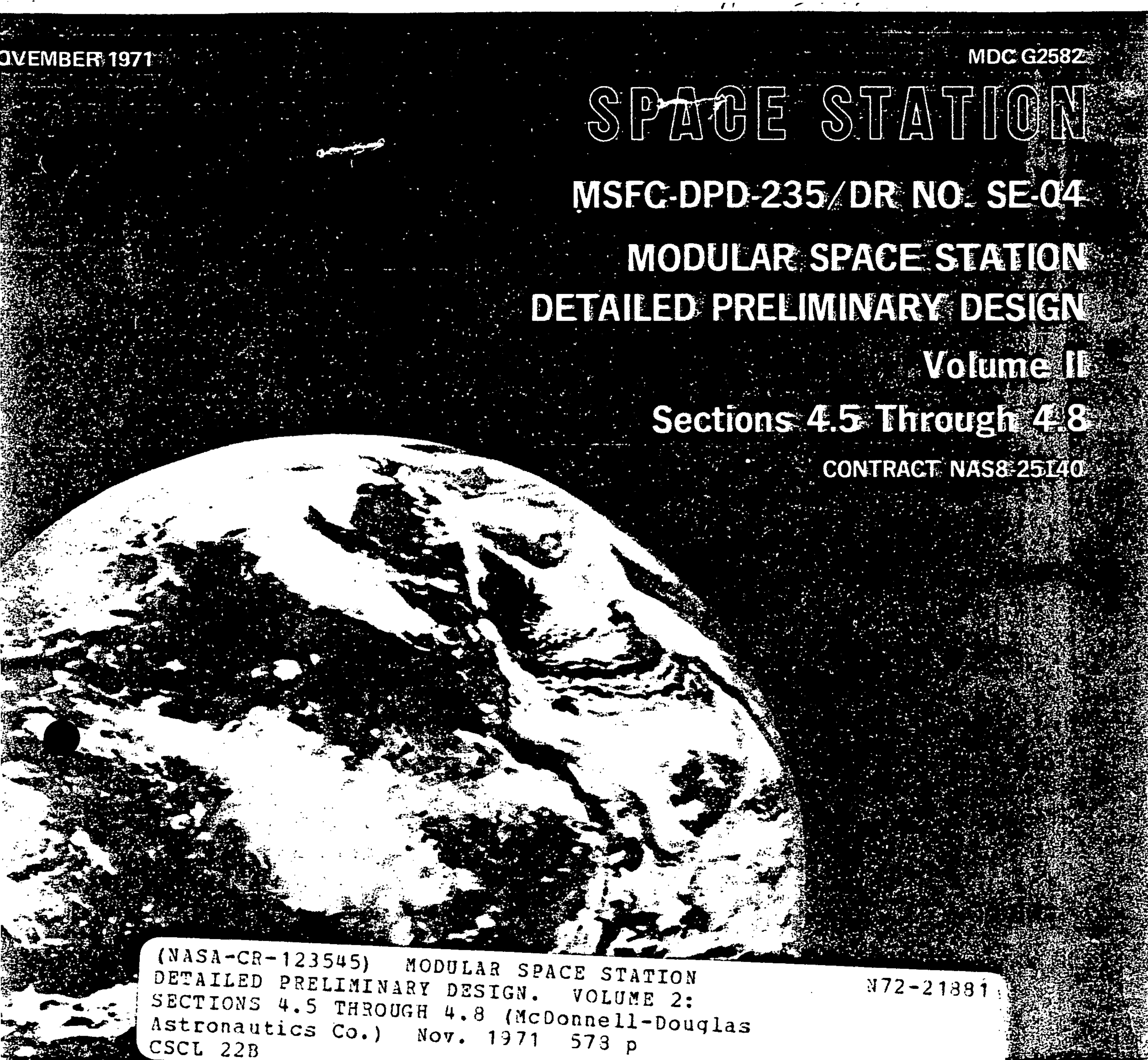


NOVEMBER 1971

MDC G2582

SPACE STATION

MSFC-DPD-235/DR NO. SE-04

MODULAR SPACE STATION DETAILED PRELIMINARY DESIGN

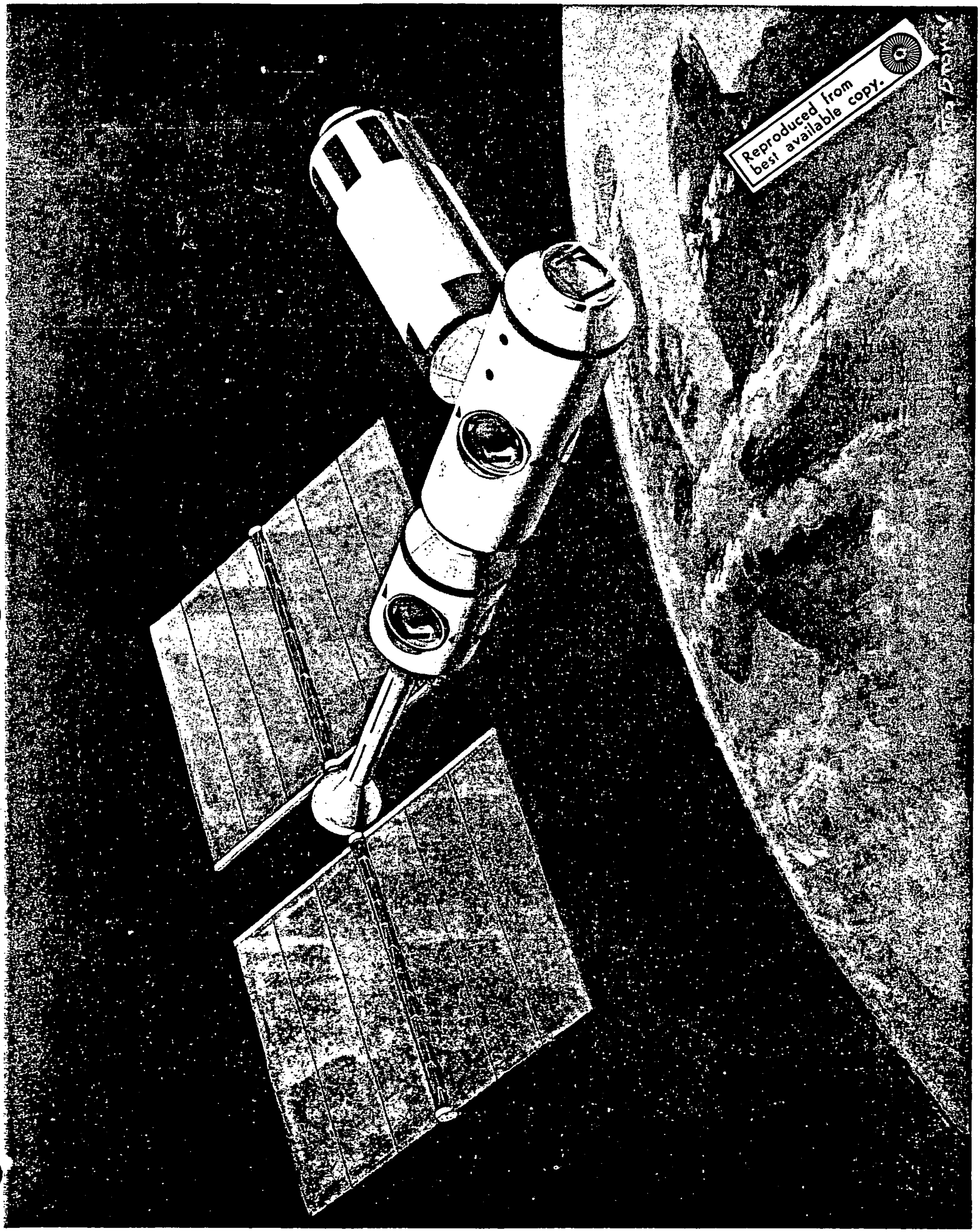
Volume II

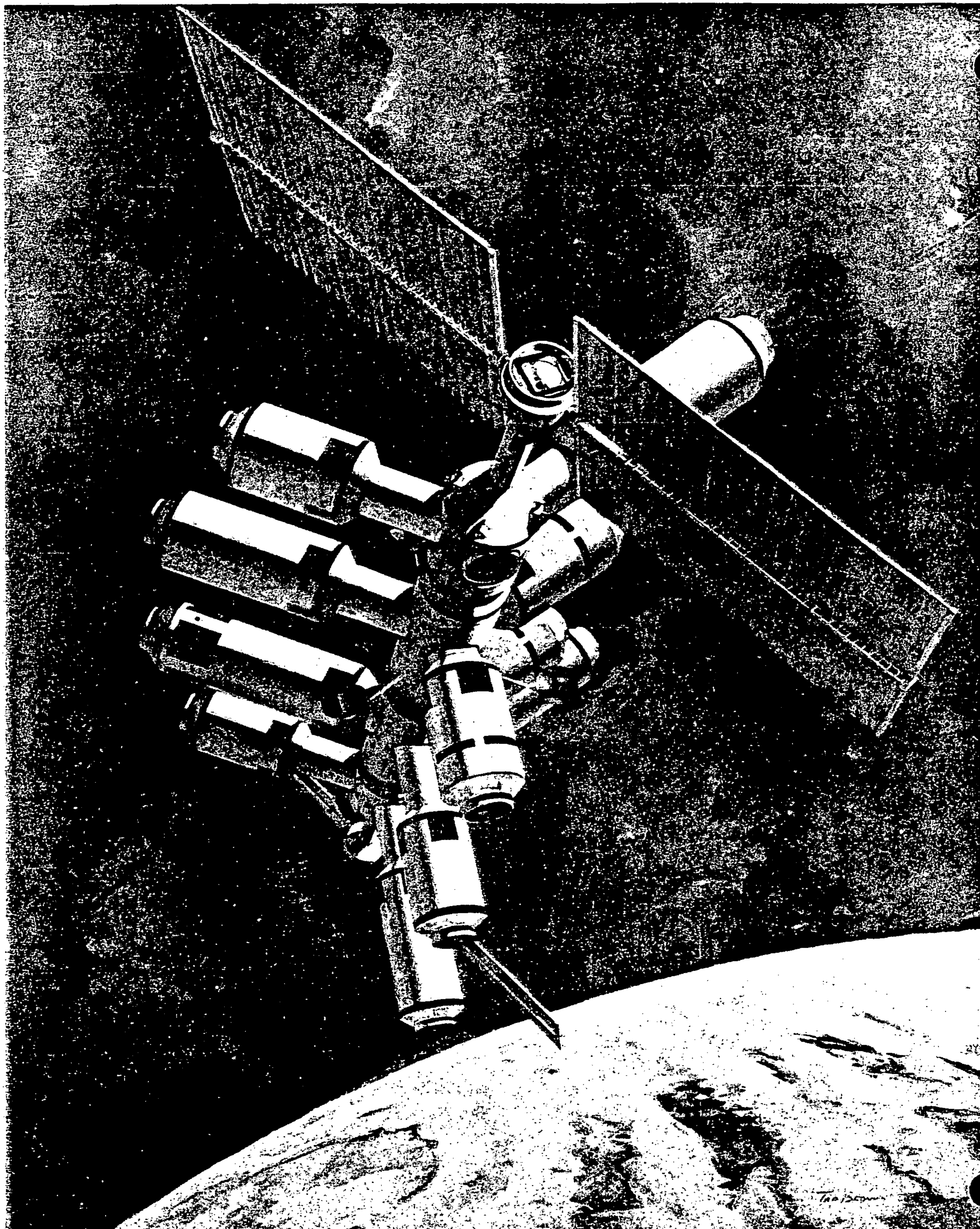
Sections 4.5 Through 4.8

CONTRACT NAS8-25140

MCDONNELL DOUGLAS ASTRONAUTICS COMPANY

MARTIN MARIETTA CORPORATION — IBM International Business Machines Corporation — MCDONNELL DOUGLAS CORPORATION





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PREFACE

The work described in this document was performed under the Space Station Phase B Extension Period Study (Contract NAS8-25140). The purpose of the extension period has been to develop the Phase B definition of the Modular Space Station. The modular approach selected during the option period (characterized by low initial cost and incremental manning) was evaluated, requirements were defined, and program definition and design were accomplished to the depth necessary for departure from Phase B.

The initial 2-1/2 month effort of the extension period was used for analyses of the requirements associated with Modular Space Station Program options. During this time, a baseline, incrementally manned program and attendant experiment program options were derived. In addition, the features of the program that significantly affect initial development and early operating costs were identified, and their impacts on the program were assessed. This assessment, together with a recommended program, was submitted for NASA review and approval on 15 April 1971.

The second phase of the study (15 April to 3 December 1971) consists of the program definition and preliminary design of the approved Modular Space Station configuration.

A subject reference matrix is included on page v to indicate the relationship of the study tasks to the documentation.

This report is submitted as Data Requirement SE-04, Volume II; Volume I contains Sections 1 through 4.4; Volume III contains Sections 4.8 through 6.

DATA REQUIREMENTS (DR's)
MSFC-DPD-235/DR NOs.
(contract NAS8-25140)

Category	Designation	DR Number	Title
Configuration Management	CM	CM-01	Space Station Program (Modular) Specification
		CM-02	Space Station Project (Modular) Specification
		CM-03	Modular Space Station Project Part 1 CEI Specification
		CM-04	Interface and Support Requirements Document
Program Management	MA	MA-01	Space Stations Phase B Extension Study Plan
		MA-02	Performance Review Documentation
		MA-03	Letter Progress and Status Report
		MA-04	Executive Summary Report
		MA-05	Phase C/D Program Development Plan
		MA-06	Program Option Summary Report
Manning and Financial	MF	MF-01	Space Station Program (Modular) Cost Estimates Document
		MF-02	Financial Management Report
Mission Operations	MP	MP-01	Space Station Program (Modular) Mission Analysis Document
		MP-02	Space Station Program (Modular) Crew Operations Document
		MP-03	Integrated Mission Management Operations Document
System Engineering and Technical Description	SE	SE-01	Modular Space Station Concept
		SE-02	Information Management System Study Results Documentation
		SE-03	Technical Summary
		SE-04	Modular Space Station Detailed Preliminary Design
		SE-06	Crew/Cargo Module Definition Document
		SE-07	Modular Space Station Mass Properties Document
		SE-08	User's Handbook
		SE-10	Supporting Research and Technology Document
		SE-11	Alternate Bay Sizes

FOLDOUT FRAME 1

FOLDOUT FRAME 2

SUBJECT REFERENCE MATRIX

LEGEND:
- CM Configuration Management
- MA Program Management
- MF Manning and Financial
- MP Mission Operations
- SE System Engineering and Technical Description

	CM				MA		MF	MP			SE								
	CM-01 Space Station Program (Modular) Specification	CM-02 Space Station Project (Modular) Specification	CM-03 Modular Space Station Project Part I CEI Spec	CM-04 Interface and Support Requirement Document	MA-05 Phase C/D Program Development Plan	MA-06 Program Option Summary Report	MF-01 Space Station Program (Modular) Cost Estimates Document	MP-01 Space Station Program (Modular) Mission Analysis Document	MP-02 Space Station Program (Modular) Crew Operations Document	MP-03 Integrated Mission Management Operations Document	SE-01 Modular Space Station Concept	SE-02 Information Management System Study Results	SE-03 Technical Summary	SE-04 Modular SS Detailed Preliminary Design	SE-06 Crew/Cargo Module Definition Document	SE-07 Modular Space Station Mass Properties Document	SE-08 User's Handbook	SE-10 Supporting Research and Technology	SE-11 Alternate Bay Sizes
2.0 Contractor Tasks																			
2.1 Develop Study Plan and Review Past Effort (MA-01)																			
2.2 Space Station Program (Modular) Mission Analysis								●											
2.3 Modular Space Station Configuration and Subsystems Definition														●					
2.4 Technical and Cost Tradeoff Studies							●												
2.4.4 Modular Space Station Option Summary											●								
2.5 Modular Space Station Detailed Preliminary Design														●					
Mass Properties																●			
2.6 Crew Operational Analysis									●										
2.7 Crew Cargo Module															●				
Mass Properties																●			
2.8 Integrated Mission Management Operations										●									
2.9 Hardware Commonality Assessment													●						
2.10 Program Support					●														
2.11 Requirements Definition																			
Space Station Program (Modular)	●																		
Space Station Project (Modular)		●																	
Modular Space Station Project—Part I CEI			●																
Interface and Support Requirements				●															
2.12 Plans					●														
2.13 Costs and Schedules							●												
2.14 Special Emphasis Task Information Management (IMS)												●							
Modular Space Station Mass Properties																●			
User's Handbook																	●		
Supporting Research and Technology																		●	
Technical Summary													●						
MOD 29																			●
MOD 40						●													

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Volume II

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Volume II

FIGURES

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Volume II

TABLES

4.5 ELECTRICAL POWER SUBSYSTEM

4.5.1 Summary

The Electrical Power Subsystem (EPS) designs developed for the Modular Space Station during this study period were derived in part from earlier studies reported in Contract Document SE-01.

The Modular Space Station baseline EPS is composed of (1) a solar array power source, (2) deployment and orientation mechanisms, (3) a primary switching assembly, (4) energy storage assemblies, (5) power control and regulation assemblies, (6) power transmission assemblies, (7) power conditioning assemblies, (8) power distribution assemblies, including circuit protection and switching elements, and (9) power management assemblies. An EPS functional diagram, shown in Figure 4.5-1, shows the alternative solar panel options considered initially, the functional components of each assembly, and the primary interfaces of the EPS. The locations of the nine major assemblies which constitute the EPS elements are shown in symbolic form by Figure 4.5-2. Further definition of power equipment location is given in the in-board profiles contained in Section 3. A simplified schematic single-line diagram is shown in Figure 4.5-3. The EPS design provides for the predicted power growth profile shown in Figure 4.5-4 from ISS to GSS by replication of the initial solar array after five years of operation with a nominal ISS power capability of 16.7 kw, thus providing growth to 31.1 kw during the period from five to ten years. The GSS requirement at ten years is 30.8 kw.

The solar array is sized to generate 52.1 kw of power initially during the sunlight (56 minutes of a 92-minute orbit is assumed). Of this, 16.7 kw average is used to power loads, 16.8 kw average is used to recharge batteries during sunlight, and 3 kw are losses. The remainder, 15.6 kw, which is allocated to compensate for degradation with time, is not needed initially and could be used for added loads by increasing the conditioning and distribution capacities. The transmission system could transmit the excess power without change within its present 100-percent (dual) redundancy. Degradation of the array with time is expected because of proton and electron damage,

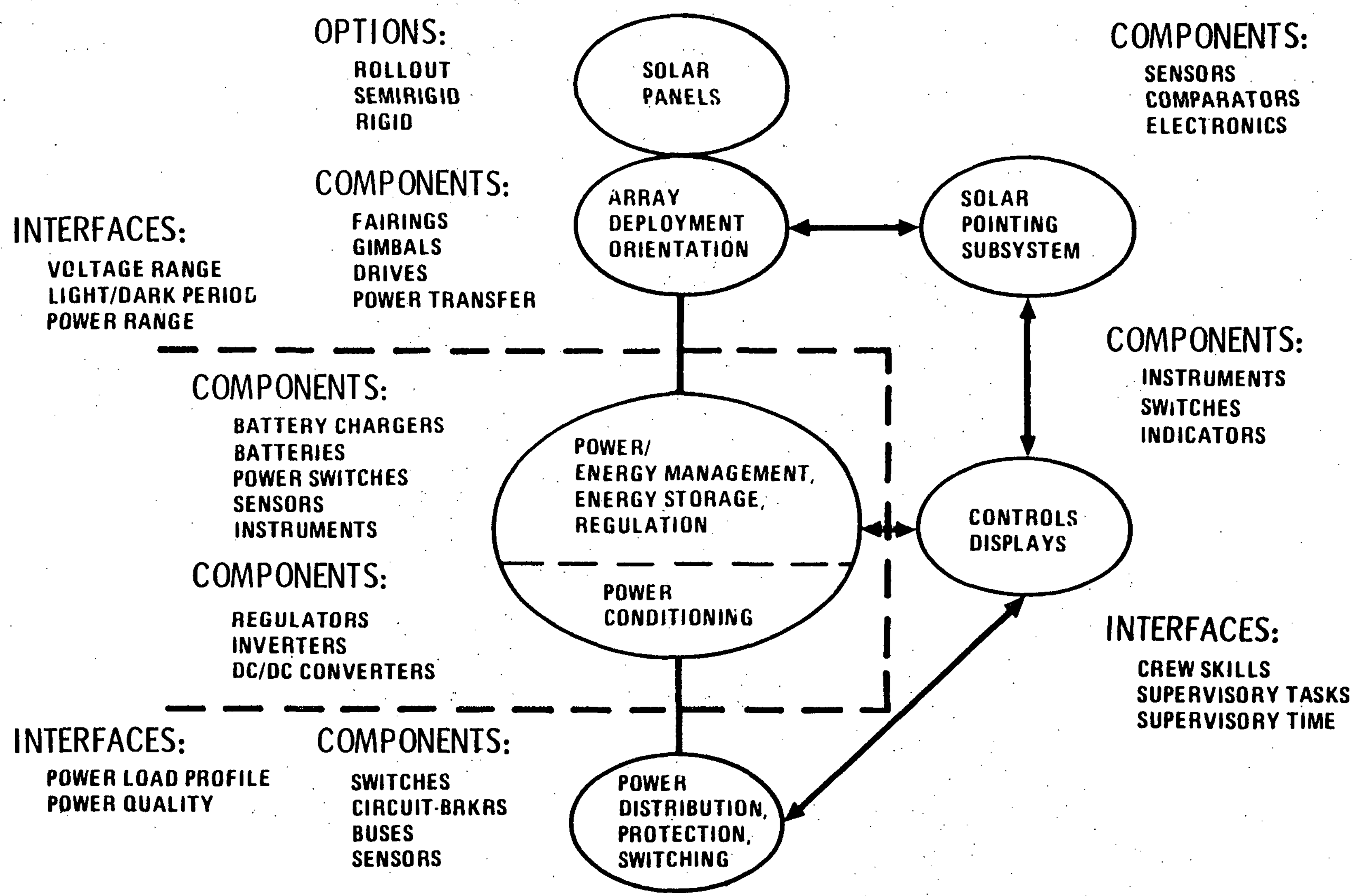

Figure 4.5-1 Electrical Power Subsystem Functional Diagram

CODE	ASSEMBLY	LOCATION PM	CM	GPL
A	SOLAR ARRAY	X		
B	ORIENTATION	X		
C	PRIMARY SWITCH	X		
D	REG & CONTROL	X	✓	
E	TRANSMISSION	X	X	X
F	ENERGY STORAGE	X	X	X
G	CONDITIONING	X	X	X
H	DISTRIBUTION	X	X	X
I	POWER MANAGEMENT	✓	X	✓

X PRIMARY LOCATION
✓ SECONDARY LOCATION

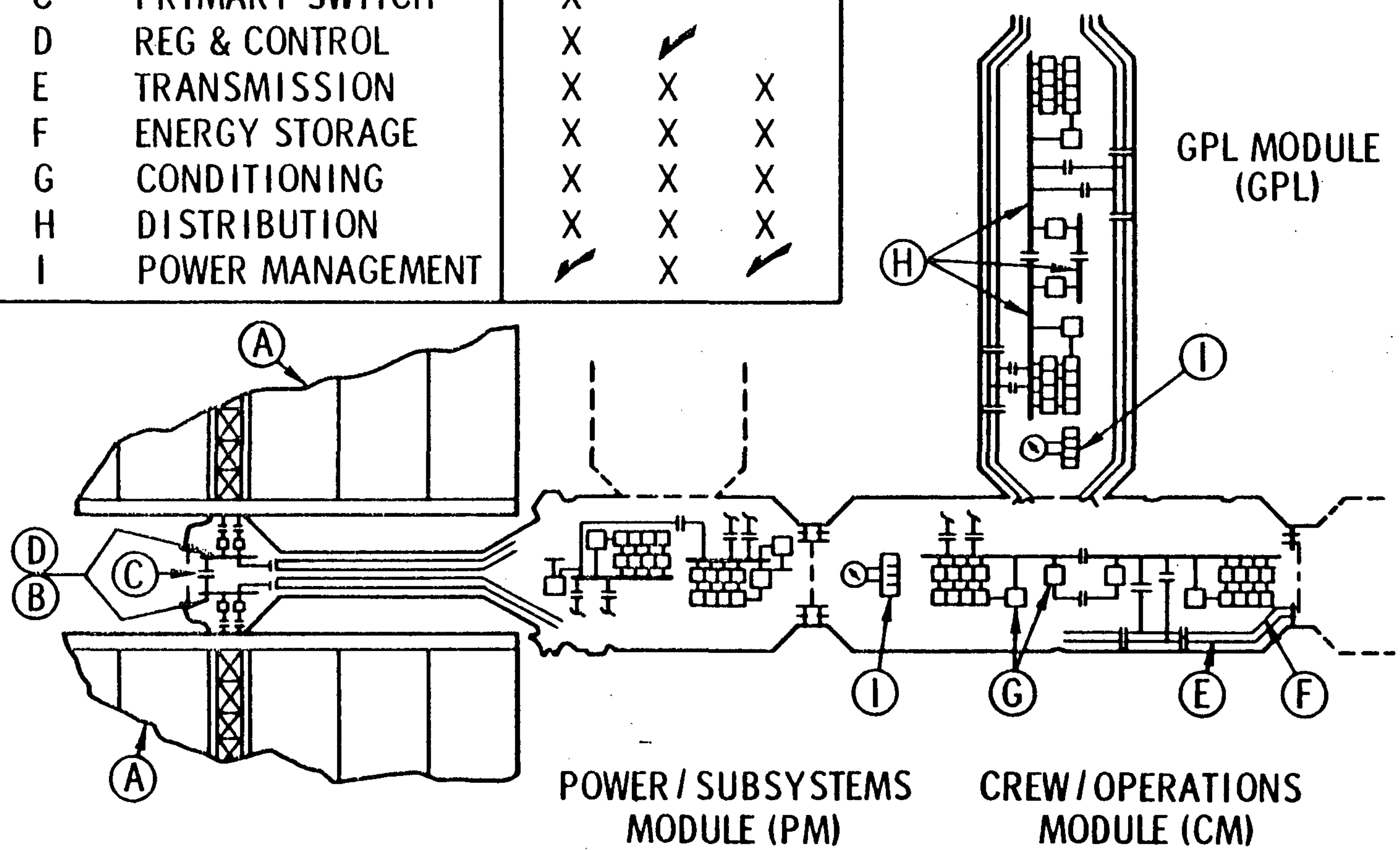

Figure 4.5-2 Electrical Power Subsystem Elements

ultraviolet radiation, and thermal and random failures. These factors, with an allowance of 1-percent per year for micrometeorite damage and contamination result in total degradations of 30 percent in five years and 40 percent in ten years.

At full charge, the Battery System is capable of supporting normal station loads (16.7 kw) for 4.2 hours and emergency loads (3 kw) for 24 hours. If a power source interruption occurs at a the lowest state of charge (65 percent), these values are reduced to 2.7 hours and 2 kw.

The arrays, transmission lines, and conditioning equipment are arranged in two independent, parallel circuits (two cables per circuit) which are normally bused together to meet total power demands. Each two-cable circuit can accommodate full system power. If a major fault occurs in one circuit, that circuit can be isolated. Thus, the system is effectively 100-percent (dual) redundant at maximum power output. Because of this redundancy, plus the backup provided by the Battery System used normally for storage and peak loads, no additional backup is necessary.

The ISS power flow diagram for the selected EPS voltage (115 vdc) and solar array source voltage regulation method (sequential partial shunt regulator, SPSR) is shown in Figure 4.5-5. Solar source power to the main distributor buses is supplied for 56 minutes of each 92-minute orbit; the battery providing power for the 36 minutes of eclipse.

The key issues addressed by the EPS study are listed in Table 4.5-1. Major study efforts were concentrated in the areas of (1) defining the EPS design requirements; (2) conducting the power load analysis using a computer program (P1268) for data storage and analysis; (3) analyzing solar array design options to determine the area required, the shadowing effects, the optimum modularity to obtain power growth at minimum cost, and to determine the orientation requirements, gimballing ranges, and gimbal rates; (4) investigating panel deployment concepts and selecting the panel design options; (5) analyzing energy storage system alternatives and optimizing the selected energy storage system to obtain minimum launch and program

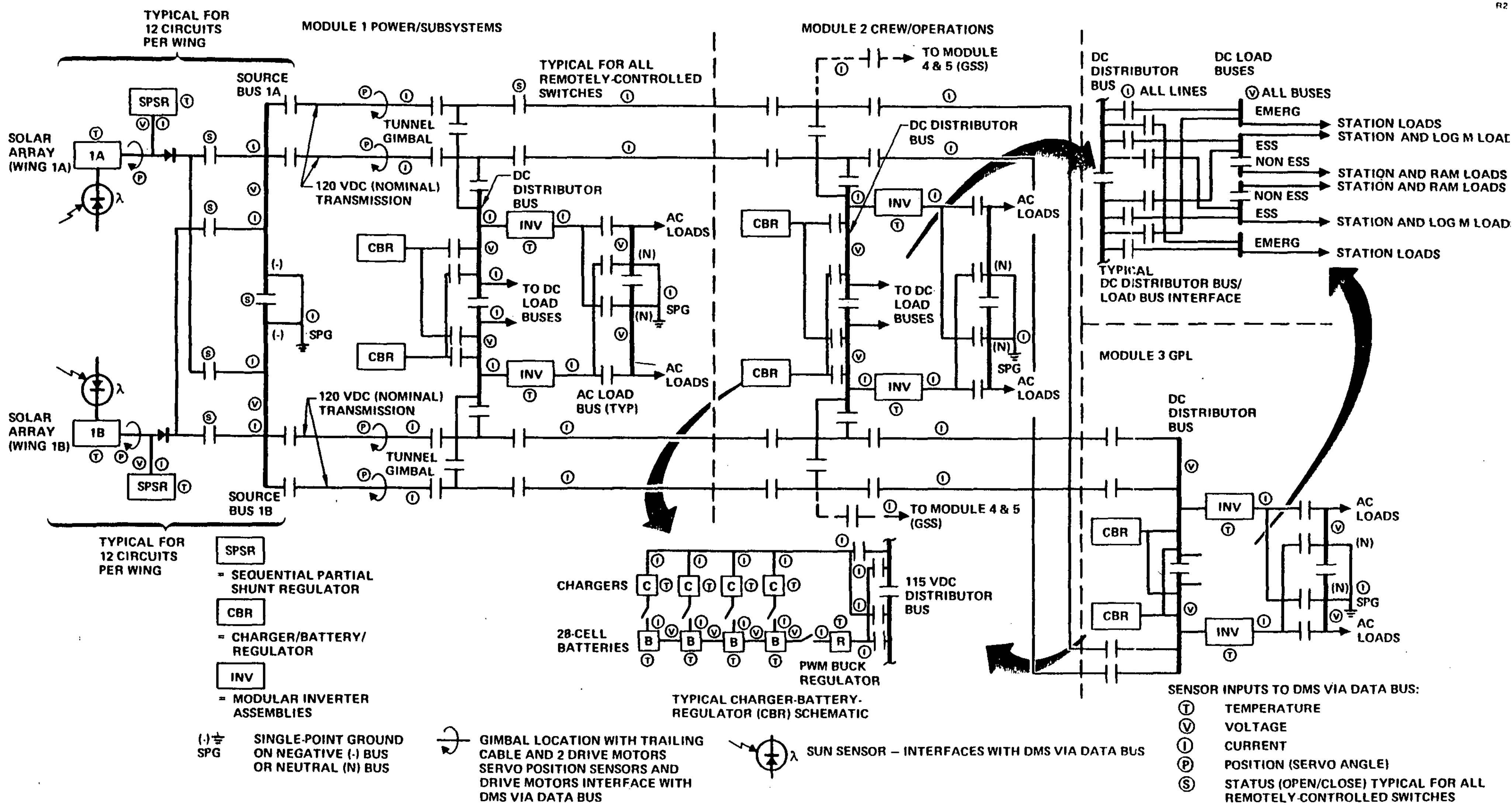

Figure 4.5-3 ISS Baseline EPS Electrical Schematic

R260

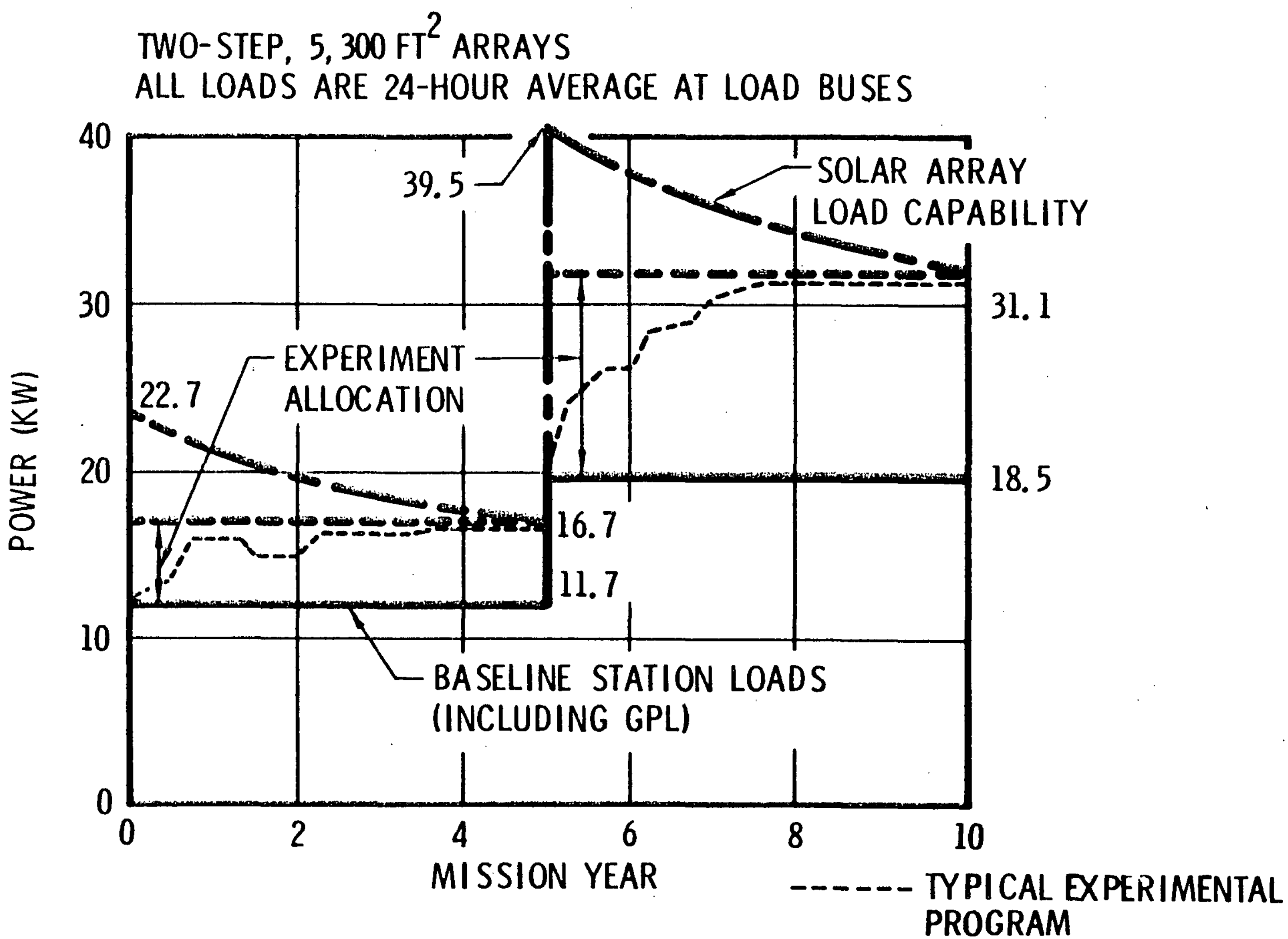

Figure 4.5-4 Power Buildup

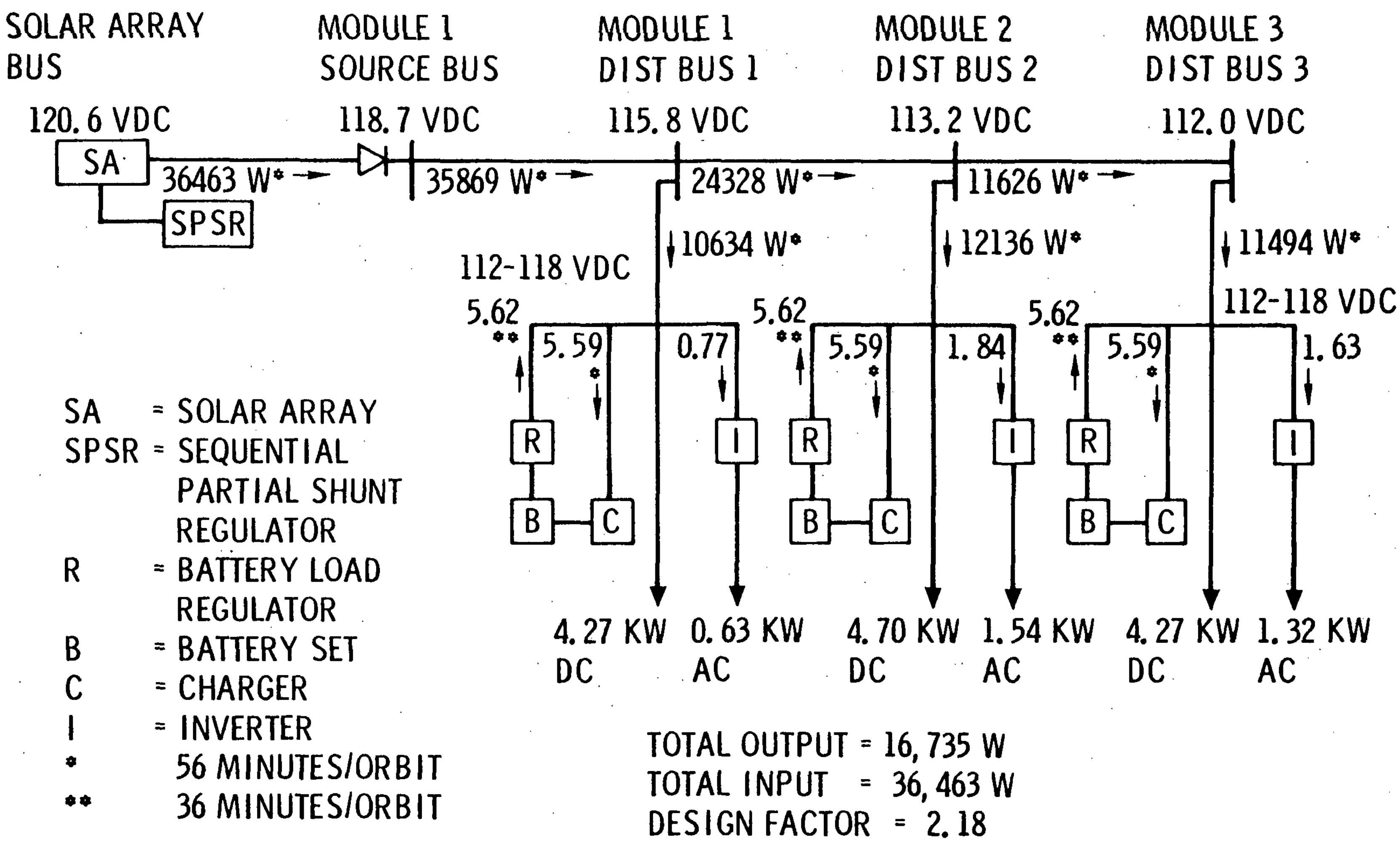

Figure 4.5-5 ISS Power Flow Diagram of 115 vdc System

Table 4.5-1
KEY ISSUES

EPS Design Requirements

Electrical Power Load Analysis

EPS Cost Effectiveness

- Program cost ($M)
- Resource effectiveness (kw-yr or man-yr)

Solar Array Design Analyses

- Area requirements
- Shadowing
- Modularity and power growth accommodation
- Orientation and gimballing
- Panel deployment
- Rigid versus flexible arrays
- Solar array degradation

Energy Storage Design Analysis

- Battery requirements
- Battery system description
- Battery performance (depth of discharge, efficiency, temperature, load profile effects)
- NiCd battery versus fuel cell/electrolysis systems

Transmission, Conditioning, and Distribution (T/C/D) Design Analyses

- T/C/D requirements
- T/C/D system description
- T/C/D performance (efficiency, flexibility)
- Voltage level selection
- Voltage regulation technique

weights and minimum program costs; (6) analyzing solar array source voltage regulation methods; and (7) analyzing the source, transmission, and distribution voltages to permit selection of a preferred voltage level.

The major trade studies, selections, and key selection factors are shown in Table 4.5-2. Complete listing of initial trades and selections are provided in Table 4.5-3. The principal study conclusions are listed in Table 4.5-4, including the conclusions derived from earlier studies which are applicable also to the current baseline Modular Space Station EPS.

Table 4. 5-2

ELECTRICAL POWER TRADE MATRIX

Trade	Options	Characteristics	SELECTION Key Selection Factors
Array type	Rigid	Astromast—foldout	Weight ISS Δ = -1, 536 lb
	$\boxed{\text{Flexible}}$	LMSC development	GSS Δ = -3, 070 lb
		5, 300 ft^2	Development status-cost
Array buildup	$\boxed{\text{Two step}}$	Identical wings	Configuration/operational compatibility
	Three step	2, 650 ft^2 each	
Energy storage	$\boxed{\text{NiCd batteries}}$	100 Amp-hr battery	Cost Δ = -\$91. 6M
	Regenerative fuel cell and electrolysis	Grumman development	Lifetime Electrolysis power reduction = 6 to 7 percent
Voltage selection	$\boxed{\text{115 vdc}}$	Commonality-shuttle	Power Δ = -3, 960 watts
	28 vdc	Shuttle payloads	Weight Δ = -2, 830 lb
T/C/D Regulation	$\boxed{\text{Sequential partial shunt reg}}$	Highest efficiency Inherent redundancy	Power Δ = 1, 132 watts
	Series with limiter hybrid		
$\boxed{}$	Selected Option		

Table 4.5-3 (page 1 of 2)

SA/B EPS INITIAL TRADES AND SELECTIONS

Trades	Alternatives	Selections
Solar array type	Flexible or rigid	Flexible
Gimballing/orientation	Fixed, 1-axis, 2-axis (±90 deg, ±180 deg, 360 deg, other)	2-axis α = ±180 deg β = ±235 deg
Array orientation drive	Synchronous, stepped	α = Synchronous* β = Synchronous*
Array stowage	Inside power tunnel or around power tunnel	Around power tunnel
Power transfer method	Slip-rings, trailing cable, spiral coil, or power clutch	α = Spiral coil β = Trailing cable in tunnel length
Battery type	NiCd, AgCd, AgZn, fuel cell	NiCd
Battery capacity	100 amp-hr, 50 amp-hr, 33 amp-hr, 20 amp-hr	100 amp-hr
Battery depth of discharge and nominal life	30 percent/1 yr, 15 percent/2.5 yr	15 percent/2.5 yr at ISS
Battery charge control	Temperature, pressure, voltage, stabistor, coulometer, amp hr meter, or auxiliary electrodes	Cell voltage limit cutoff Third electrode alternate
Transmission voltage	28, 56, 115, 260 vdc, or 115/200 vac	115 vdc
Distribution voltage	28, 56, 115, 260 vdc, or 115/200 vac, 400 Hz or 60 Hz	115 vdc, 115/200 vac 400 Hz, and 115 vac 60 Hz (GPL only)
Transmission/distribution circuit configuration	Radial, ring, dual; nonredundant or redundant	Dual, redundant

*Arrays are recycled during eclipse to unwind the cables.

Table 4.5-3 (page 2 of 2)
SA/B EPS INITIAL TRADES AND SELECTIONS

Trades	Alternatives	Selections
Voltage regulation	Series, shunt, unregulated	Shunt (array) series (battery)
Solar array voltage control	Switched arrays, switched panels, full shunt, partial shunt, sequential partial shunt, series	Sequential partial shunt
Battery voltage control	Charge regulator, discharge regulator	Charge and discharge regulators
Battery switching	Low voltage, high voltage	Low-voltage charge, high-voltage discharge
Power conditioning		
DC power	Bulk conditioning versus individual conditioning	Bulk regulation for 115 vdc, individual for other requirements
AC power	Parallel versus isolated inverters	Small isolatable units with parallel capability
Power switching	Electromechanical, solid state	Solid state for low power and EM for high power
Power control	Automatic versus manual	Automatic supervisory control with manual backup.
	Remote versus local	Remote control for initial power module and solar deployment. Local control for growth and for manned operation

Table 4.5-4 (page 1 of 2)

PRINCIPAL EPS STUDY CONCLUSIONS

A. Increased emphasis on minimum cost resulted in the following conclusions:

   1. Use smallest array unit size compatible with mission power profile

   2. Buildup by using identical unit arrays.

   3. Use augmentation in preference to replacement of arrays.

   4. Use partial deployment (if feasible) to delay degradation of area not yet required by load.

   5. Use panel or partial array replacement by EVA rather than replacement of a complete array unit, if possible.

B. Shuttle launch weight limitation (20,000 lb) is a primary design driver leading to modular EPS design.

C. The 14-foot diameter payload envelope and 20,000-lb weight are primary design drivers leading to a flexible array preference.

D. Shadowing may reduce array effectiveness intermittently by 1 to 30 percent, depending principally on (a) station/array orientation, (b) station/array configuration, or (c) Experiment/subsystem operation scheduling.

E. The initial solar array may be immobilized in the α-axis gimbal, with either a fixed β-axis before manning and until load growth requires array orientation after initial manning. Approximately 63.6 percent of full power is available in this mode (neglecting shadowing) and a minimum drag profile is achieved to minimize orbital decay.

F. NiCd batteries are preferable to fuel cell/electrolysis units for lower program cost, lower solar array power demand, improved lifetime, and less sensitivity to environment.

G. 115 vdc is preferable to 28 vdc for reduced program cost, lower weight, and lower power losses to be provided by solar array.

H. To minimize initial launch weight, the Crew/Operations Module can be launched with 50 percent of the ultimate battery weight; the Crew/Operations Module and the GPL can be launched without batteries and can be supported by the initial Power/Subsystems Module batteries until manning starts.

I. Battery temperature control (to 0 to 20° C) with a design point of 13° C will substantially prolong battery life and reduce the battery module logistical burden.

Table 4.5-4 (page 2 of 2)

PRINCIPAL EPS STUDY CONCLUSIONS

J. A sequential partial shunt regulation system and 115 vdc nominal system voltage each reduce the source power requirements and the weight.

K. Solar array degradation is predicted to be 30 percent in five years, and 40 percent in ten years.

The EPS design is summarized in Table 4.5-5 in terms of its performance, capacity, modularity, and weight.

Credit is due the following companies, all of whom supported the Phase B study without funding in the areas indicated:

Company	Support Areas	Source of Information
Lockheed Missile & Space Company	Flexible solar array	Derived from NASA-MSC Contract NAS9-11039
TRW	(1) Flexible solar array	In-house
	(2) Rigid solar array	Derived from MDAC Skylab Program Subcontract
Fairchild Industries	(1) Flexible solar array	Derived from NASA-JPL Contract 951969
	(2) Rigid solar array	In-house
TRW	Sequential partial shunt regulator technology	Derived from USAF-APL Contract F33615-70-C-1627, and earlier NASA Contracts for unmanned spacecraft e.g. Voyager
Westinghouse	(1) Conditioning equipment parameters	In-house
	(2) Automated power management technology	In-house
Grumman Aerospace	Batteries	Derived from NASA-MSC Contract NAS9-11074

Table 4.5-5 (page 1 of 2)

ELECTRICAL POWER SUBSYSTEM DESIGN SUMMARY

EPS Type	Flexible foldout solar arrays and NiCd batteries.
Performance	Initial average load of 16.7 kw for 6 men (ISS) for five years. Buildup to 12-man (GSS) load of 30.8 kw after five years. Maintain 12-man load of 30.8 kw to 10-year mission (minimum duration). Dynamic design loads are as follows: • F_N = 0.15 Hz (fundamental mode resonance frequency) • G-tolerance = 0.2-g perpendicular to plane of array and 0.5-g maximum vector sum in three axes. NOTE: This is preliminary pending trades of resultant array penalty versus shock absorber design penalties. Dual full-capacity 115-vdc transmission to all Station modules. Dual independent sources available to all power conditioning assemblies. General distribution at 115 vdc and 115/200 vac, 3-phase, 400 Hz within each Station module and to RAM's. 115 vac, 1-phase, 60 Hz distribution to GPL.
Capacity	
Fully deployed arrays	22.7 kw average at load bus initially from one 5,300 ft^2 array; degrading to 16.7 kw at 5 years. 39.5 kw average at load bus after 5 years from the initial array and a second 5,300 ft^2 array combined, degrading to 31.1 kw at 10 years.
Modularity	
Solar arrays	Two 2,650 ft^2 array wings on one Power/Subsystems Module for ISS. Four 2,650 ft^2 array wings total, with a second Power/Subsystems Module added for GSS. Twelve panels in each 5,300 ft^2 array.

Table 4.5-5 (page 2 of 2)

ELECTRICAL POWER SUBSYSTEM DESIGN SUMMARY

Batteries	Twenty-four SPSR source voltage regulation circuits. Four 100 amp-hr batteries initially in Power/ Subsystems Module. Add twenty 100 amp-hr batteries at time of manning for ISS, as follows: 4 in Power/ Subsystems Module, 8 in Crew/Operations Module, and 8 in GPL. Maintain battery capacity by replacement at 2.5-year intervals. Seven 4-cell, replaceable modules per battery. Add sixteen 100 amp-hr batteries for buildup to GSS, as follows: 9 in second Crew/Operations Module and 8 in second Power/Subsystems Module.
Weight	
Solar arrays	3,056 lb initially for two wings; add two wings weighing 3,056 lb at 5 years. (Orientation system is 1,304 lb additional and the Power Module structural weight is not included.)
Batteries	1,520 lb initially; add 7,600 lb for buildup to ISS manning level; add 6,080 lb at 5 years for GSS.

4.5.2 Requirements

The requirements for EPS design are provided in contract specification documents document as follows: CM-01, Spec PS02925, Para 3.7.1.4.11, -.12; CM-02, Spec RS02927, Para 3.7.1.3.2, and in the PRD in Contract End Item (CEI) Specification, CP-02929 and in Document CM-03, Section 3.2.1.4.

4.5.2.1 Power Load Analysis

The requirements which have the greatest impact upon the EPS design are contained in the power load analysis. The power requirements as a function of mission life are shown by the power buildup curves in Figure 4.5-4. The ISS load bus average power level is shown as 16.7 kw. This represents a modest change from 17.3 kw established on July 15, 1971, as the ISS reference "design-to" value. The initial solar array power is shown as 22.7 kw, allowing 30 percent for degradation in the first five years. A contingency factor of approximately 10 percent is also included in the Station power allocation. Growth to the GSS level of 30.8 kw is provided by the addition of a replicated array of 5,300 ft^2 nominal area. This provides 39.5 kw of initial total load bus power from the two arrays, degrading in the last five years (GSS) to 31.1 kw. This power profile provides an envelope which is sufficient to support a typical experiment program such as 534G.

A detailed assembly-level study was made of the geometry associated with the solar array cell assembly, the array power-collection, wiring-harness losses (1.67 percent), and the energy storage power transmission, conditioning, and distribution efficiencies. The study produced a design factor of 2.18 (solar array delivered power divided by load bus power). Array areas are: (1) 4,820 ft^2 of actual (net) silicon cell area; (2) 5,300 ft^2 of gross active panel area; and (3) 5,496 ft^2 of gross panel (strip) area, including panel hinges, electrical harness, and panel mountings in this area, but not including the 4-in. void spaces between panels, astromast deployment boom space (3.5 ft), inboard support assembly (28 in. by 23.7 ft), or outboard support assembly (28 in by 23. 7 ft).

The modular load analysis is shown in Table 4.5-6 for both ISS and GSS configurations. The double power notation (vis. 5. 2/8. 5) indicates that during ISS operation the GPL requires 5.2 kw for the ISS level of experimental activity and for a 6-man crew. During GSS operation, the GPL requires 8.5 kw for the higher level of experimental activity and for a 12-man crew. A 9. 4-percent contingency factor is allowed as a margin to provide for power growth in the Station electrical loads, for efficiency reductions during hardware design, etc. A 3-percent average allowance for distribution loss is also included for distribution from the load buses to all loads (including experiments). All losses within the EPS from solar arrays to load buses are accounted for in the design factor of 2.18, as defined previously.

The data summarized in Table 4. 5-6 were derived from the Program P1268 data bank, which provides the type of information listed in Table 4.5-7. The data summary is derived from data accumulated from the following levels;

Table 4. 5-6

MODULAR LOAD ANALYSIS—WATTS

Load	Power/ Subsystems	Crew/ Operations	GPL	Crew/ Operations	Power/ Subsystems	RAMS Distributed
Station Subsystems	2. 5	4. 0/4. 1	3. 0/3. 3	2. 7	1. 6	
Contingency	0. 2	0. 4/0. 4	0. 3/0. 3	0. 3	0. 2	
Distribution Loss	0. 1	0. 2/0. 2	0. 1/0. 1	0. 1	0. 1	
Station Operation	2. 8	4. 6/4. 7	3. 4/3. 7	3. 1	1. 9	
Logistics and Crew/ Operations Modules		0. 5		0. 5		
GPL		0. 5/1. 0				
Integral Experiments			1. 3/3. 8			
RAMS Experiments						3. 7/8. 8
Bus Totals	2. 8	5. 1/5. 2	5. 2/8. 5	3. 6	1. 9	3. 7/8. 8
Station Totals	16. 8 - ISS	30. 8 - GSS				

Double notation (e.g.5. 2/8. 5) indicates 5. 2 kw at ISS and 8. 5 kw at GSS.

Table 4.5-7

ELECTRICAL LOAD ANALYSIS

Program P1268–

Provides data bank for electrical load data, parts lists, units, and weight data for mass properties

Prints specific and summation data for each station module at (1) total, (2) subsystem, (3) assembly, (4) subassembly, and (5) functional element levels, as follows:

- Electrical power requirement
  - Average; maximum; standby
- Electrical power form preference
  - ac; dc
- Voltage preference
  - 115 vdc; 28 vdc
- Waveform preference
  - 400 Hz sine-wave; 400 Hz square-wave; 60 Hz
- Load criticality
  - Emergency; essential; nonessential; redundant
- Duty cycle (percent)

---

the lowest level in which the power requirements can be identified provides the basic information (Code CB-2-1, transponder modem, for example). The P1268 subassembly level accumulates these data on a functional basis (Code CB-2-0, S-Band RF Assembly Group, for example). The P1268 assembly level provides the values of power needed to operate the major functional assemblies of the Space Station subsystems (CB-0-0, RF communications, for example). These data are then assembled into the subsystem power requirements shown in Table 4.5-8 (CA-0-0, communications, for example). These data are finally assembled into the power summary shown in Table 4.5-9, which provided data for the modular load analysis shown in Table 4.5-6. All these data are 24-hour average load bus values

Table 4.5-8

ISS MAJOR ASSEMBLIES WEIGHT AND POWER (Sheet 1)

WEIGHT AND AVG PWR REPORT

CONFIGURATION ISS INITIAL SPACE STATION

ITEM CODE	ITEM NAME	MODULE 1 PWR/SUB SY SER NO 1 WT	A PWR	MODULE 2 CREW/OPER SER NO 2 WT	A PWR	MODULE 3 GEN PUR LB SER NO 3 WT	A PWR	MODULE 4 SER NO WT	A PWR	MODULE 5 SER NO WT	A PWR	RAMS (DIST) WT	A PWR	WT	A PWR	TOTAL WT	TOTAL A PWR
AA 0 0	GUID=NAV=CONT SUBSYS	2263	468	20	40	0	0	0	0	0	0	0	0	0	0	2283	508
B 0 0	ATTITUDE CONTROL	147	196	20	40	0	0	0	0	0	0	0	0	0	0	167	236
C 0 0	PRECISION REFERENCE	96	12	0	0	0	0	0	0	0	0	0	0	0	0	96	12
D 0 0	CONTROL MOMENT GYROS	2020	260	0	0	0	0	0	0	0	0	0	0	0	0	2020	260
BA 0 0	EVIRON CONT LIFE SUP	3517	344	2824	1254	1274	744	0	0	0	0	0	0	0	0	7615	2342
B 0 0	ATMOS SUPPLY=CONT	2498	0	99	10	62	10	0	0	0	0	0	0	0	0	2659	20
C 0 0	ATMOSPHERE RECOND	65	140	276	748	264	598	0	0	0	0	0	0	0	0	625	1494
D 0 0	WATER MANAGEMENT	0	0	0	202	0	0	0	0	0	0	0	0	0	0	0	202
E 0 0	WASTE MANAGEMENT	0	0	132	158	0	0	0	0	0	0	0	0	0	0	132	158
F 0 0	IVA EVA	660	0	338	0	714	0	0	0	0	0	0	0	0	0	1712	0
G 0 0	THERMAL CONTROL	274	196	234	136	234	136	0	0	0	0	0	0	0	0	742	468
X 0 0	EXPENDABLES	0	0	1745	0	0	0	0	0	0	0	0	0	0	0	1745	0
CA 0 0	COMMUNICATIONS	261	49	673	260	24	12	0	0	0	0	0	0	0	0	958	321
B 0 0	RF COMMUNICATIONS	249	43	627	222	0	0	0	0	0	0	0	0	0	0	876	265
C 0 0	INTERNAL COMMUNICATE	12	6	46	38	24	12	0	0	0	0	0	0	0	0	82	56
DA 0 0	HI=THRUST PROPULSION	1966	125	230	40	0	0	0	0	0	0	0	0	0	0	2196	165
B 0 0	PRESSURIZATION SUBS	212	16	27	0	0	0	0	0	0	0	0	0	0	0	239	16
C 0 0	PROPELLANT SUBS	413	60	27	0	0	0	0	0	0	0	0	0	0	0	440	60
D 0 0	THRUST SUBS	196	48	176	40	0	0	0	0	0	0	0	0	0	0	372	88
E 0 0	PURGE SUBS	51	1	0	0	0	0	0	0	0	0	0	0	0	0	81	1
X 0 0	EXPENDABLES	1064	0	0	0	0	0	0	0	0	0	0	0	0	0	1064	0
EA 0 0	LO-THRUST PROPULSION	263	143	146	95	0	0	0	0	0	0	0	0	0	0	409	238
B 0 0	PROPELLANT SUBS	99	41	0	0	0	0	0	0	0	0	0	0	0	0	99	41
C 0 0	CONTROL SUBS	[illegible]	96	22	95	0	0	0	0	0	0	0	0	0	0	78	191
D 0 0	THRUST SUBSYSTEM	108	6	124	0	0	0	0	0	0	0	0	0	0	0	232	6
FA 0 0	DATA MANAGEMENT SYS	763	976	1519	1444	2692	1711	0	0	0	0	0	0	0	0	4974	4131
B 0 0	COMPUTATION	230	444	40	20	250	415	0	0	0	0	0	0	0	0	520	879
C 0 0	DATA ACQ AND XFER	192	401	161	264	135	206	0	0	0	0	0	0	0	0	488	871
D 0 0	IMAGE PROCESSING	0	0	0	0	598	87	0	0	0	0	0	0	0	0	598	87
E 0 0	DATA STORAGE	45	63	120	83	310	284	0	0	0	0	0	0	0	0	475	430
F 0 0	DISPLAY AND CONTROL	200	40	1126	1011	1351	675	0	0	0	0	0	0	0	0	2677	1726
J 0 0	UNIQUE CHECKOUT EQPT	96	28	72	66	48	44	0	0	0	0	0	0	0	0	216	138
GA 0 0	ELECTRICAL POWER	9206	176	3732	23	3700	18	0	0	0	0	0	0	0	0	16638	217
B 0 0	SOLAR ARRAY	4030	0	0	0	0	0	0	0	0	0	0	0	0	0	4030	0
C 0 0	ORIENTATION	924	153	0	0	0	0	0	0	0	0	0	0	0	0	924	153
D 0 0	ENERGY STORAGE	3040	0	3040	0	3040	0	0	0	0	0	0	0	0	0	9120	0
E 0 0	PRIMARY SWITCHING	328	7	153	7	110	7	0	0	0	0	0	0	0	0	591	21
F 0 0	REG AND CONTROL	323	5	15	5	0	0	0	0	0	0	0	0	0	0	338	10
G 0 0	TRANSMISSION	231	0	211	0	158	0	0	0	0	0	0	0	0	0	600	0

Table 4.5-8

ISS MAJOR ASSEMBLIES WEIGHT AND POWER (Sheet 2)

WEIGHT AND AVG PWR REPORT
CONFIGURATION ISS INITIAL SPACE STATION

ITEM CODE	ITEM NAME	MODULE 1 PWR/SUB SY SER NO 1 WT	A PWR	MODULE 2 CREW/OPER SER NO 2 WT	A PWR	MODULE 3 GEN PUR LR SER NO 3 WT	A PWR	MODULE 4 SER NO WT	A PWR	MODULE 5 SER NO WT	A PWR	RAMS (DIST) WT	A PWR	WT	A PWR	TOTAL WT	TOTAL A PWR
H 0 0	CONDITIONING	176	0	166	0	202	0	0	0	0	0	0	0	0	0	544	0
I 0 0	DISTRIBUTION	154	11	147	11	190	11	0	0	0	0	0	0	0	0	491	33
HA 0 0	CREW SYSTEMS	267	10	3796	308	1245	0	0	0	0	0	0	0	0	0	5308	318
B 0 0	FOOD MANAGEMENT	0	0	837	238	0	0	0	0	0	0	0	0	0	0	837	238
C 0 0	HYGIENE	100	10	179	10	0	0	0	0	0	0	0	0	0	0	279	20
D 0 0	CREW ACCOMODATIONS	91	0	787	0	309	0	0	0	0	0	0	0	0	0	1187	0
E 0 0	HSEKEEPG/TRASH HNDLG	0	0	299	12	60	0	0	0	0	0	0	0	0	0	359	12
F 0 0	IVA/EVA SUPPORT	10	0	-1	-1	176	0	0	0	0	0	0	0	0	0	186	0
G 0 0	CREW COMPTMTS WK-STA	66	0	1694	48	700	0	0	0	0	0	0	0	0	0	2460	48
H 0 0	FIRE PROTECTION	-1	-1	-1	-1	-1	-1	0	0	0	0	0	0	0	0	0	0
LA 0 0	LIGHTING	137	222	245	532	215	502	0	0	0	0	0	0	0	0	647	1256
B 0 0	LIGHTING	137	222	245	532	215	502	0	0	0	0	0	0	0	0	647	1256
SA 0 0	STRUCTURE	4170	0	5818	0	7324	0	0	0	0	0	0	0	0	0	17312	0
B 0 0	STRUCTURE	2929	0	3754	0	5188	0	0	0	0	0	0	0	0	0	11871	0
C 0 0	ELECTRICAL WIRING	906	0	1338	0	1410	0	0	0	0	0	0	0	0	0	3654	0
D 0 0	RESIDUALS	335	0	726	0	726	0	0	0	0	0	0	0	0	0	1787	0
TA 0 0	METEOROID+THERMAL	1243	0	1530	0	1624	0	0	0	0	0	0	0	0	0	4397	0
B 0 0	RADIATOR	1243	0	1530	0	1624	0	0	0	0	0	0	0	0	0	4397	0
UA 0 0	DOCKING PROVISIONS	1700	0	1651	0	427	0	0	0	0	0	0	0	0	0	3778	0
B 0 0	DOCKING PROVISIONS	1700	0	1651	0	427	0	0	0	0	0	0	0	0	0	3778	0
VA 0 0	LOGISTICS MODULE 1	0	0	0	279	0	0	0	0	0	0	0	0	0	0	0	279
B 0 0	ACTIVE DOCKED PWR	0	0	-1	279	0	0	0	0	0	0	0	0	0	0	0	279
WA 0 0	LOGISTICS MODULE 2	0	0	0	199	0	0	0	0	0	0	0	0	0	0	0	199
B 0 0	DORMANT DOCKED PWR	0	0	-1	199	0	0	0	0	0	0	0	0	0	0	0	199
	BASIC SUBSYSTEM TOT	25806	2513	22184	4474	18525	2987	0	0	0	0	0	0	0	0	66515	9974
1A 0 0	SUBSYSTEM FACTORS	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
1B 0 0	DISTRIBUTION LOSS 3	0	75	0	134	0	89	0	0	0	0	0	0	0	0	0	298
1C 0 0	CONTINGENCY. 10 PCT.	0	251	0	447	0	298	0	0	0	0	0	0	0	0	0	996
	SUBSYSTEM TOTAL	25806	2839	22184	5055	18525	3374	0	0	0	0	0	0.	0	0	66515	11268
2A 0 0	EXPERIMENT PROGRAM	0	0	0	0	5361	1784	0	0	0	0	0	3584	0	0	5361	5368
B 0 0	EXPERIMENTS	0	0	0	0	0	1254	0	0	0	0	0	3584	0	0	0	4838
C 0 0	GENERAL PURPOSE LAB	0	0	0	0	5361	530	0	0	0	0	0	0	0	0	5361	530
	EXP. SUBTOTAL	0	0	0	0	5361	1784	0	0	0	0	0	3584	0	0	5361	5368

Table 4.5-9

ISS SUBSYSTEM WEIGHT AND POWER

WEIGHT AND AVG PWR REPORT

CONFIGURATION ISS — INITIAL SPACE STATION

ITEM CODE	ITEM NAME	MODULE 1 PWR/SUB SY SER NO 1 WT	MODULE 1 A PWR	MODULE 2 CREW/OPER SER NO 2 WT	MODULE 2 A PWR	MODULE 3 GEN PUR LB SER NO 3 WT	MODULE 3 A PWR	MODULE 4 SER NO WT	MODULE 4 A PWR	MODULE 5 SER NO WT	MODULE 5 A PWR	RAMS (DIST) WT	RAMS A PWR	WT	A PWR	TOTAL WT	TOTAL A PWR
AA 0 0	GUID-NAV-CONT SUBSYS	2263	468	20	40	0	0	0	0	0	0	0	0	0	0	2283	508
BA 0 0	EVIRON CONT LIFE SUP	3517	344	2824	1254	1274	744	0	0	0	0	0	0	0	0	7615	2342
CA 0 0	COMMUNICATIONS	261	49	673	260	24	12	0	0	0	0	0	0	0	0	958	321
DA 0 0	HI-THRUST PROPULSION	1966	125	230	40	0	0	0	0	0	0	0	0	0	0	2196	165
EA 0 0	LO-THRUST PROPULSION	263	143	146	95	0	0	0	0	0	0	0	0	0	0	409	238
FA 0 0	DATA MANAGEMENT SYS	763	976	1519	1444	2692	1711	0	0	0	0	0	0	0	0	4974	4131
GA 0 0	ELECTRICAL POWER	9206	176	3732	23	3700	18	0	0	0	0	0	0	0	0	16638	217
HA 0 0	CREW SYSTEMS	267	10	3796	308	1245	0	0	0	0	0	0	0	0	0	5308	318
LA 0 0	LIGHTING	187	222	245	532	215	502	0	0	0	0	0	0	0	0	647	1256
SA 0 0	STRUCTURE	4170	0	5818	0	7324	0	0	0	0	0	0	0	0	0	17312	0
TA 0 0	METEOROID+THERMAL	1243	0	1530	0	1624	0	0	0	0	0	0	0	0	0	4397	0
UA 0 0	DOCKING PROVISIONS	1700	0	1651	0	427	0	0	0	0	0	0	0	0	0	3778	0
VA 0 0	LOGISTICS MODULE 1	0	0	0	279	0	0	0	0	0	0	0	0	0	0	0	279
WA 0 0	LOGISTICS MODULE 2	0	0	0	199	0	0	0	0	0	0	0	0	0	0	0	199
	BASIC SUBSYSTEM TOT	25306	2513	22184	4474	18525	2987	0	0	0	0	0	0	0	0	66515	9974
1A 0 0	SUBSYSTEM FACTORS	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
1B 0 0	DISTRIBUTION LOSS 3	0	75	0	134	0	89	0	0	0	0	0	0	0	0	0	298
1C 0 0	CONTINGENCY 10 PCT	0	251	0	447	0	298	0	0	0	0	0	0	0	0	0	996
	SUBSYSTEM TOTAL	25906	2839	22184	5055	18525	3374	0	0	0	0	0	0	0	0	66515	11268
2A 0 0	EXPERIMENT PROGRAM	0	0	0	0	5361	1784	0	0	0	0	0	3584	0	0	5361	5368
	EXP. SUBTOTAL	0	0	0	0	5361	1784	0	0	0	0	0	3584	0	0	5361	5368
3A 0 0	EXPERIMENT FACTORS	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
3B 0 0	DISTRIBUTION LOSS 3	0	0	0	0	0	53	0	0	0	0	0	107	0	0	0	160
	TOTAL EXP.	0	0	0	0	5361	1837	0	0	0	0	0	3691	0	0	5361	5528
	TOTALS	25306	2839	22184	5055	23886	5211	0	0	0	0	0	3691	0	0	71876	16796

and are identified separately for each Space Station Module. Table 4.9-10 is the GSS Power Summary. The computer printout shows the latest GSS study data (July 22, 1971). Subsequent preliminary design load analyses were limited to ISS only. However, the data changes for ISS which would modify GSS data has been reviewed and the new values are indicated in parentheses on Table 4.5-10.

As Table 4.5-7 indicates, Program P1268 also provides the weight data, the number of units required, the parts lists, peak power, and duty cycle at all levels of detail; in addition to the average power values shown in Table 4.5-6 and 4.5-8 to 4.5-10. It also provides power allocations for each load according to either ac or dc power form, 115 vdc or 28 vdc voltage preference, and according to load criticality (emergency, essential, nonessential, or redundant).

4.5.2.2 Premanning Power Requirements

The period from launch to manning includes (1) the period from launch until the solar array is deployed and initially supplies power to the loads and batteries and (2) the initial unmanned phase or "orbital storage" period when the activation crew is absent and the Station Modules are in a dormant but controlled operational state. The electrical power requirements during the launch to orbit period are time-lined in Figure 4.5-6 and summarized in Table 4.5-11 according to average and maximum power, time duration of the loads, energy requirements, conditioning and distribution losses, ampere-hour demands, and battery depths of discharge for support by the battery complements which can be provided if the Station Module launch weight limitations permit. For the baseline operations profile, the Shuttle provides the power for the Crew/Operations and GPL Modules.

During the "orbital storage" period, subsequent to Station solar array activation (T_{SA}) and before ISS manning commences, the subsystem and module power demands (24-hour average values) are as shown in Table 4.5-12. These load demands for all modules are met by the solar array and four batteries launched with the Power/Subsystem Module.

Table 4.5-10

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SPACE STATION CONFIGURATION

WEIGHT AND AVG PWR REPORT

CONFIGURATION GSS APRIL 17TH BASELINE

ITEM CODE	ITEM NAME	MODULE 1 PWR/SUB SY SER NO 1 WT	MODULE 1 A PWR	MODULE 2 CREW/OPER SER NO 2 WT	MODULE 2 A PWR	MODULE 3 GEN PUR LR SER NO 3 WT	MODULE 3 A PWR	MODULE 4 CREW/OPER SER NO 4 WT	MODULE 4 A PWR	MODULE 5 PWR/SUB SY SER NO 5 WT	MODULE 5 A PWR	MODULE 6 SER NO WT	MODULE 6 A PWR	RAMS WT	RAMS A PWR	TOTAL WT	TOTAL A PWR	
AA 0 0	GUID-NAV-CONT SUBSYS	2191	486	0	8	1	0	4	0	1667	272	0	0	0	0	3871	766	
BA 0 0	EVIRON CONT LIFE SUP	1890	377	3968	1087	1153	899	3983	1087	1890	377	0	0	0	0	12884	3827	
CA 0 0	COMMUNICATIONS	217	61	819	360	24	12	36	18	12	6	0	0	0	0	1108	457	
DA 0 0	HI-THRUST PROPULSION	1309	113	200	32	0	0	54	0	1830	113	0	0	0	0	3293	258	
EA 0 0	LO-THRUST PROPULSION	300	238	124	0	0	0	0	0	300	238	0	0	0	0	724	476	
FA 0 0	DATA MANAGEMENT SYS	739	1178	1573	1687	2588	2764	504	661	436	505	0	0	0	0	5839	6795	
GA 0 0	ELECTRICAL POWER	7226	183	3170	17	3191	22	3191	21	7211	183	0	0	0	0	23989	426	
HA 0 0	CREW SYSTEMS	0	0	638	311	0	0	638	311	0	0	0	0	0	0	1276	622	
LA 0 0	LIGHTING	187	222	245	532	215	502	245	532	179	222	0	0	0	0	1071	2010	
SA 0 0	STRUCTURE	3790	0	4589	0	5147	0	4589	0	3790	0	0	0	0	0	21905	0	
TA 0 0	METEOROID+THERMAL	1030	0	2050	0	1840	0	2059	0	1030	0	0	0	0	0	8009	0	
UA 0 0	DOCKING PROVISIONS	1915	0	2025	0	810	0	2025	0	1915	0	0	0	0	0	8690	0	
VA 0 0	CREW CARGO MODULE	0	0	0	250 (0.5)	0	0	0	250 (0.5)	0	0	0	0	0	0	0	500	(1.0)
	BASIC SUBSYSTEM TOT	21393	(2.5) 2858	19409	(4.6) 4284	14969	(3.3) 4199	17319	(3.2) 2880	20260	1916	0	0	0	0	93350	16137	(15.2)
1A 0 0	SUBSYSTEM FACTORS	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	(0.6)
1B 0 0	DISTRIBUTION LOSS 4	0	114 (0.1)	0	171 (0.2)	0	167 (0.1)	0	115 (0.1)	0	76 (0.1)	0	0	0	0	0	643	(1.4)
1C 0 0	CONTINGENCY 10 PCT	0	285 (0.2)	0	428 (0.4)	0	419 (0.3)	0	288 (0.3)	0	191 (0.2)	0	0	0	0	0	1611	
	SUBSYSTEM TOTAL	21393	(2.8) 3257	19409	(5.2) 4883	14969	(3.7) 4785	17319	(3.6) 3283	20260	(1.9) 2183	0	0	0	0	93350	18391	(17.2)
2A 0 0	EXPERIMENT PROGRAM	0	0	0	0	3502	4580 (4.6)	0	0	0	0	0	0	0	8520 (8.5)	3502	13100	(13.1)
	EXP. SUBTOTAL	0	0	0	0	3502	4580 (4.6)	0	0	0	0	0	0	0	8520 (8.5)	3502	13100	(13.1)
3A 0 0	EXPERIMENT FACTORS	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	
3B 0 0	DISTRIBUTION LOSS 4	0	0	0	0	0	183 (0.2)	0	0	0	0	0	0	0	340 (0.3)	0	523	(0.5)
3C 0 0		0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	
	TOTAL EXP.	0	0	0	0	3502	4763 (4.8)	0	0	0	0	0	0	0	8860 (8.8)	3502	13623	(13.6)
	TOTALS	21393	3257	19409	4883	18471	9548	17319	3283	20260	2183	0	0	0	8860	96852	32014	(30.8)
			(2.8)		(5.2)		(8.5)		(3.6)		(1.9)				(8.8)			

( ) Value in parentheses are in kW. These are derived from the October 25, 1971, updated ISS load analysis.

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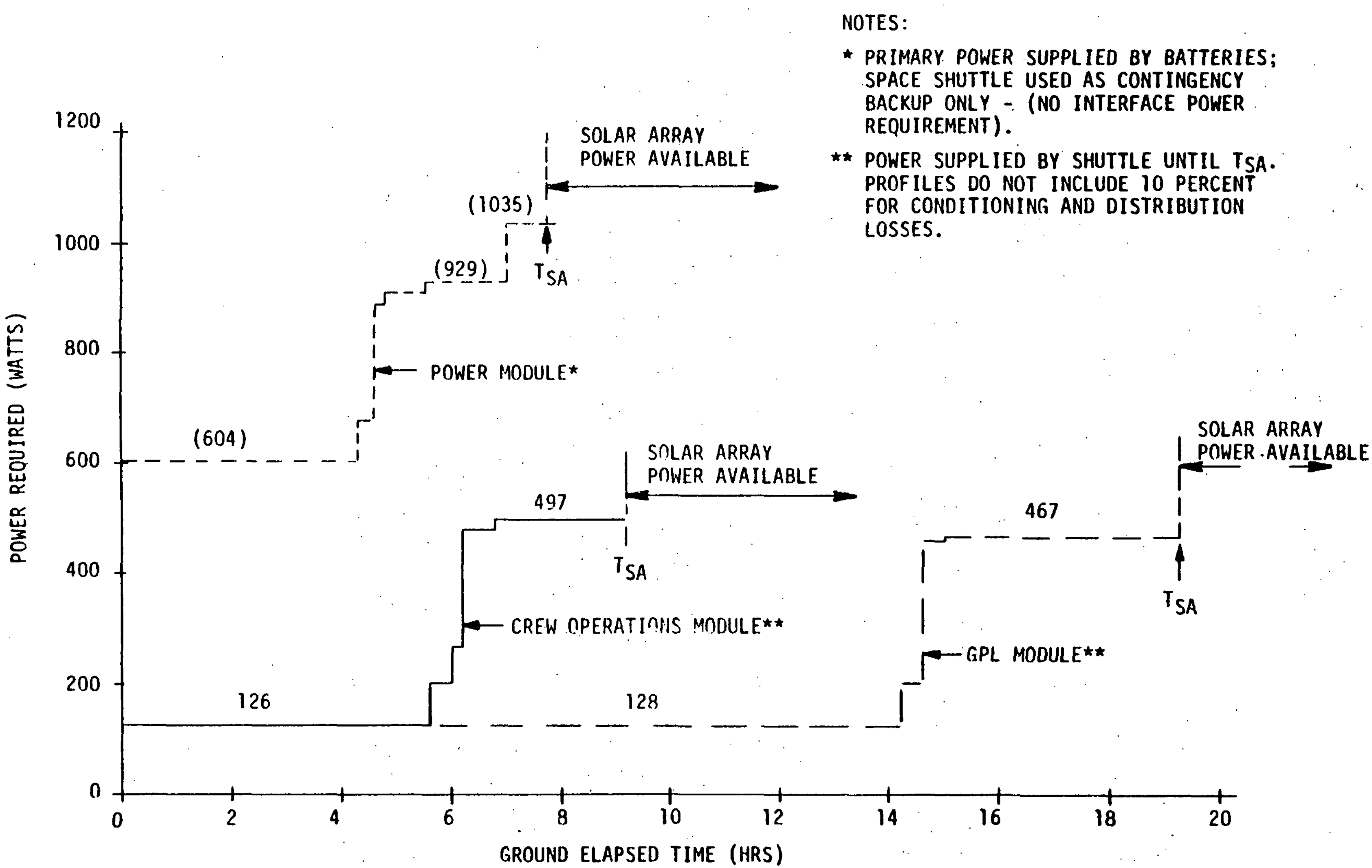

Figure 4.5-6 Space Station Buildup Operations

Table 4.5-11

POWER AND ENERGY REQUIREMENTS SUMMARY

LIFT-OFF THROUGH AVAILABILITY OF SOLAR ARRAY POWER
(T_O Through T_{SA})

Module	Maximum Load Demand (w)	Average Load Demand (w)	Elapsed Time From L-O (hr)	Load Energy Demand (w-hr)	Conditioning and Distribution Losses (w-hr)	Total Energy Demand (w-hr)	Amp-hr Demand 1.2 Volts/Cell Module Battery Set (amp-hr)	Number of Installed Battery Sets	Depth of Discharge Per Battery Set (percent)
1* (Power)	1,035	840	7.8	6,548	655	7,203	53.4	1	53.4
2** (Crew)	497	252	9.2	2,317	232	2,549	19.0	0	–
								1	19.0
								2	9.5
3** (GPL)	467	212	19.2	4,044	404	4,448	33.2	0	–
								1	33.2

*This load demand supplied by on-board (Power/Subsystem Module) batteries.
**These load demands supplied by Space Shuttle power system.

Table 4.5-12

INITIAL UNMANNED MODE ORBITAL STORAGE POWER REQUIREMENTS IN WATTS

Subsystem	Module 1	Module 2	Module 3
GNC	120	---	---
EC/LS	225	264	336
COMM	18	(S-Band Module 1)	(S-Band Module 1)
Prop-HI	90	---	---
Prop-LO	Negligible	Negligible	Negligible
DMS	500	300	400
Lighting	Negligible	Negligible	Negligible
EPS	138	23	23
Totals	1,091	587	759

Note: Power requirements are from solar array deployment until initial manning. Power for all three modules is supplied by the solar array and four batteries launched with the Power/Subsystems Module.

4.5.2.3 Emergency Power Load Analysis

Assessments of one-hour and extended (over 96 hours) emergency power requirements for ISS are shown in Table 4.5-13. The one hour and extended-minimum capability power levels are not sufficient to sustain an experimental program, and some degree of damage to experiments would likely be sustained (e.g., freezing, thermal stress distortion, loss of biocultures and specimens, loss of chemical solutions, etc.).

Sufficient power is maintained, however, to provide for crew safety and for a capability to repair faulty systems needed to restore the Station to normal operations. To preserve the experiments in a condition for restoration to their full experimental status, an additional allowance of 350 watts is provided.

Table 4.5-13

MODULAR SPACE STATION ISS EMERGENCY POWER REQUIREMENTS IN WATTS

		Extended Periods	
	1-Hour	Minimum Capability	Checkout Capability
GNC	---	---	---
ECLS	498	1,217	1,217
COMM	22	44	44
PROP	---	60	60
DMS	126	186	1,739
EPS	63	75	75
Crew	---	341	341
Lighting	625*	1,250*	1,250*
Logistics or Crew/ Operation Modules	279	478	478
Experiments	---	---	---
Totals**			
Unrestricted	1,613	3,651	5,204
Restricted	902	2,925	3,990

*Area and handrail lighting only—does not include portable battery powered emergency lights.

**Unrestricted allows crew to occupy/maintain all three station modules. Restricted allows crew to occupy/maintain only one module.

The emergency/contingency power requirements and durations are shown in Figure 4.5-7. A one-hour period is principally for damage assessment and minimal repairs or switching operations to restore power, thus avoiding need to use the 96-hour emergency pallets onboard for survival/rescue periods. A 24-hour period is indicated also for more extensive

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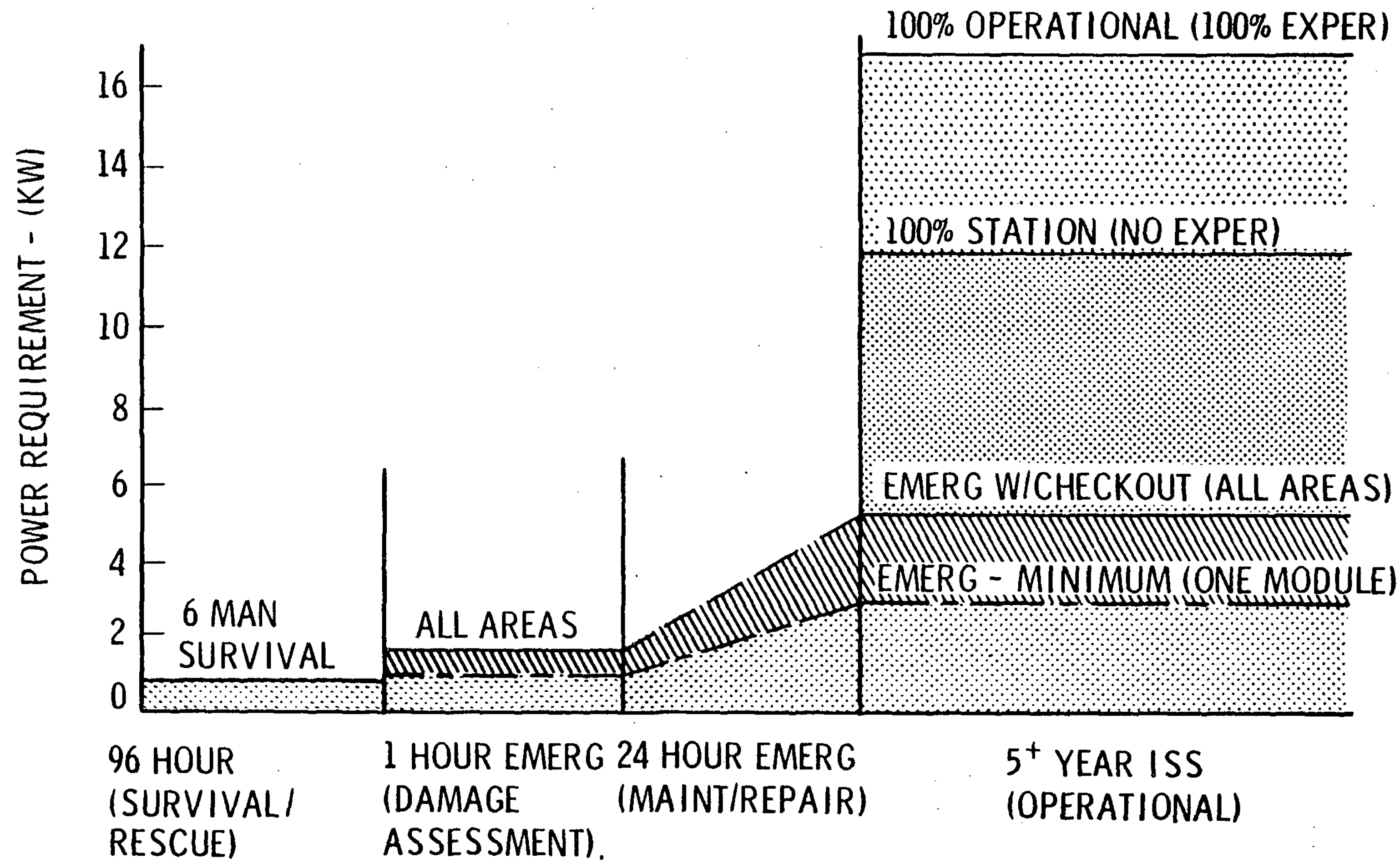


Figure 4.5-7 Emergency/Contingency Power Requirement

maintenance/repair activities. The extended periods allow for resupply of major parts and/or visitation by a repair team.

4.5.2.4 Principal Design Requirements

The PRD specifications previously reference in Subsystem 4.5.2 represent the principal program, project, and system-level design drivers. These are listed in Table 4.5-14 for reference. Specific design impacts are listed in Table 4.5-15, in terms of the design features and characteristics which were developed to accommodate these specifications.

Table 4.5-14

ELECTRICAL POWER SUBSYSTEM REQUIREMENTS

Specification	Paragraph	Requirements
PS 02925	3.7.1.4.11	ISS electrical power will be provided by solar arrays. Minimum average load requirement is 15 kw at the load bus, averaged over a 24-hour period.
PS 02925	3.7.1.4.12	As a goal, no orientation restrictions will be imposed by subsystems, i.e., electrical power, thermal control, communications.
PS 02926	3.6.1.1.1.1.1.7	The Orbiter project shall provide electrical power to the cargo bay payload interface during all mission phases from cargo insertion to completion of the mission as identified by the Space Station projects.
		Space Station projects shall identify electrical power required in the Orbiter cargo bay for all mission phases and shall provide an electrical interface compatible with the Orbiter.
RS 02927	3.7.1.3.2.1	ISS electrical power will be provided by solar arrays. Minimum average load electrical power requirement is 15 kw at the load bus, averaged over a 24-hour period.
RS 02927	3.7.1.3.2.2	As a goal, solar cell arrays shall have a clear unobstructed view of the sun to preclude partial shadowing of their surfaces. If shadowing of the arrays does occur, the arrays shall be designed to provide adequate power during shadowed periods and to preclude shadow-induced damage.

Table 4.5-14

ELECTRICAL POWER SUBSYSTEM REQUIREMENTS (Continued)

Specification	Paragraph	Requirements
RS 02927	3.7.1.3.2.3	The electrical system shall provide circuit protection devices for all station distribution wiring where necessary.
RS 02927	3.7.1.3.2.4	Standard electrical interfaces shall be provided for power transfer between modules and other attachable elements requiring a power transfer interface with the Space Station Module.
RS 02927	3.7.1.3.2.5	Independent power sources shall be provided for each of the independent pressurizable volumes.
RS 02927	3.7.1.3.2.6	Crew supervisory and maintenance and replacement times and skill requirements for the electrical power system (EPS) shall be minimized by the use of automated or semiautomated monitoring and control techniques.
RS 02927	3.7.1.3.2.7	The EPS selected for the Initial Space Station (ISS) shall accommodate the capability for growth to the Growth Space Station (GSS) and shall be electrically and physically compatible with the GSS.
RS 02927	3.7.1.3.2.8	The Modular Space Station EPS shall, as a whole, have a maintained lifetime of not less than 10 years; however, elements of the EPS may be replaced in total or in modular form for maintenance or for growth. As a design goal, during this maintenance or uprating period, the required electrical power levels will be sustained without interruption.
RS 02927	3.7.1.3.2.9	The Modular Space Station EPS shall consist of not less than two independent sources, each of which will be capable of supplying backup power for an extended period (5 years) assuming no second failure mode, and full sustaining power for a duration of 5 years to preserve experiments, instruments, fluid systems, and the like which are required for return to full station operational capability.

Table 4.5-15

PROGRAM REQUIREMENTS DOCUMENT DESIGN DRIVERS

Specification	Paragraph	Impact
PS 02925	3.7.1.4.11	Solar arrays are used for both ISS and GSS. The minimum 24-hour average power available at the load buses during ISS is 16.7 kw and during GSS is 31.1 kw.
RS 02925	3.7.1.4.12	No Station orientation restrictions are imposed by the EPS during normal Station operations. Abnormal conditions such as multiple failures on the turret drive would require (1) POP Station orientation and a slow (6.3 deg/day) Station β-axis roll of ± 78.5 degrees or (2) a reduction of Station power to compensate for panel misorientation, or (3) solar inertial orientation with fixed α and β axis gimbals.
PS 02926	3.6.1.1.1.1.1.7	The Power/Subsystems Module requires no Space Shuttle standby power; however, a circuit connection to the Shuttle Power System is required as backup to the batteries. The Crew/Operations Module requires Space Shuttle support power of 547 watts maximum and energy of 2,549 w-hr (no batteries). The GPL requires Space Shuttle support power of 514 watts maximum and energy of 4,448 w-hr (no batteries).
RS 02927	3.7.1.3.2.1	Solar array power source is required for ISS. Economical growth to GSS by solar array replication is most cost-effective.
RS 02927	3.7.1.3.2.2	(1) Horizontal flight attitude does not meet this goal, and operational techniques to retain power are required during a few quarters of the mission. (2) Solar cell strings are arranged with connections at 104-cell series intervals to parallel eight strings (sub-modules); and each solar cell is in a group of seven paralleled cells. These techniques are more conservative than Skylab design, for which shadow-induced hot-spot phenomena are considered adequately controlled, according to extensive testing and analysis.

Table 4.5-15

PROGRAM REQUIREMENTS DOCUMENT DESIGN DRIVERS (Continued)

Specification	Paragraph	Impact
RS 02927	3.7.1.3.2.3	Individual load circuit breakers are provided with trip ratings to protect the load circuit wiring. Load control is not specifically provided (except by manual operation) and is presumed to be provided at the load.
RS 02927	3.7.1.3.2.4	A 115 vdc power cable connection is provided in each quadrant; each cable is adequate for 50 percent of the maximum solar array power demand (hence 100 percent power redundancy in four cables). A control/instrumentation cable is provided on each quadrant also. Two 115/200 vac 3-Phase 400 Hz power cable connections are provided for RAM's, which can also be utilized by other Station modules if further analysis shows this to be desirable. Present design analysis shows that ac power interchange is not necessary or desirable.
RS 02927	3.7.1.3.2.5	The solar array wings are electrically and physically independent, as are the dual redundant transmission assemblies. The single common element is the power tunnel/turret gimbal (α-axis). Failure to drive this gimbal would reduce power capability. However, this reduction is prevented by changing the station orientation to the principal alternate mode, POP with a very slow β-axis roll (±78.5 degrees at 6.5 deg/day, or 50 days per cycle. Battery strings all are independent, with two strings per station module, and full-capacity sharing (1) within each station module through bus tie circuit-breakers and (2) between station modules through the dual redundant transmission lines.
RS 02927	3.7.1.3.2.6	Preprocessors (RDAU's) will be used to minimize the data rate burden on the central processing computer facility. Out-of-tolerance data will be presented to the central processor for display, analysis, and control reaction.

Table 4.5-15

PROGRAM REQUIREMENTS DOCUMENT DESIGN DRIVERS (Continued)

Specification	Paragraph	Impact
RS 02927	3.7.1.3.2.7	The GSS capability is provided for simple replication of solar arrays on a similar Power/Subsystems Module. The uniform 115 vdc transmission interface is used to transfer and share power, with no synchronization or complex operations.
RS 02927	3.7.1.3.2.8	The solar array may be used for 5 years with 30-percent degradation to the design power value, thereafter degrading an additional 10 percent (40 percent total) after 10 years. After the second (GSS) array is installed to augment the initial array for 3.1 kw at 10 years, the initial array may be replaced if necessary without interruption to ISS power levels. The older array may also be replaced at 5-year mission intervals commencing at 10 years, to continue operation at the GSS power level. These augmentation operations reduce power to the ISS level during replacement periods only. Although EVA is discouraged, it is possible to individually retract, replace, and redeploy the solar panels. Each panel package is 88.6-in. long by 24-in. wide by 3.3-in. deep before deployment. Batteries are replaceable in 4-cell modular units of 40 lb each (dimensions are approximately 8 in. by 7.5 in. by 6 in.). A shunting technique will allow module replacement while the remaining battery is active, if desired. The preferred mode is to replace modules while the battery is out of service, however. Conditioning modules are of modular design and can be deenergized and replaced without interrupting other electrical services. In each case, the remaining power capability is sufficient to supply all essential loads.

Table 4.5-15

PROGRAM REQUIREMENTS DOCUMENT DESIGN DRIVERS (Continued)

Specification	Paragraph	Impact
RS 02927	3.7.1.3.2.8 (Cont)	Power transmission cables are 100-percent redundant, normally operate at 50-percent load, and are not therefore expected to require replacement during the 10-year mission. However, design for replacement is feasible with some weight penalty for ducts, conduits, or accessible raceways.
RS 02927	3.7.1.3.2.9	Each half-panel is electrically separate from the other half-panel due to regulation by separate SPSR circuits (subassemblies) to the source buses. Each solar panel, of 12 panels total, is electrically independent of the other 11 panels. Each solar array wing is electrically and physically independent of the other wing, with separate -axis gimbal drives. The -axis tunnel gimbal drive is common to the two wings; however, see the impact statement for RS 02927. Para. 3.7.1.3.2.5, for relief of this common coupling mode. Less than two panels of twelve panels will be sufficient to provide the emergency power level or the sustaining power level indefinitely, after which normal station operation may be restored. The six battery strings in the ISS (10 strings in GSS) are electrically and physically independent, and can share power indefinitely through any of the four transmission lines. These can be maintained or replaced without reduction of station power by allowing minor usage of the reserve capacity of other batteries below a 15-percent nominal average depth of discharge. The four batteries of each string and the seven 4-cell modules of each battery are physically but not electrically independent.

4.5.3 Selected Subsystem Design

4.5.3.1 Description

The baseline EPS as stated in Subsection 4.5.1 is composed of nine major assembly groups. These are shown in Figure 4.5-8 as an assembly tree. The functional relationships between these assembly groups were shown previously in Figure 4.5-1 with the major assembly options and interfaces. The assemblies are defined in this subsection for the selected baseline EPS. Trade analyses, selection rationale, and related discussions are provided in Subsection 4.2.4, "Design Analyses and Trade Studies." The selected design is summarized in Tables 4.5-16 and 4.5-17.

Solar Array Assembly Design

The selected power source for the ISS is a solar cell array, as required by the Contract Guidelines and Requirements (Reference CM-01 Specification PS02925, Paragraph 3.7.1.4.11). This source is also selected for the GSS in the form of an identical, replicated solar cell array assembly. These solar cell arrays are composed of flexible, foldout solar panels. The array is composed of two wings, each having two quadrants, each quadrant composed of three solar panels; therefore 12 solar panels make up one array. The solar array is shown in Figure 4.5-9, which also indicates the stages of deployment (stages A and B) in phantom. This basic design was provided by LMSC and adapted to the MDAC Modular Space Station requirements. Each panel of the array is 88-in. wide by 62.5-feet long when deployed. When packaged for launch, the panels are in metal cases with embossed Kapton protective pads between panel folds. This design is under development by LMSC under Contract NAS9-11039 with NASA/MSC. The packaged panel is shown in Figure 4.5-10 to be 88.6-in. long by 24-in. wide by 3.3-in. deep, and are fully protected from the launch environment. During recovery, the panels are reassembled in the cases, but are no longer protected by the inter-layer protective pads. These pads are held away from the deployed solar panels by the pad retaining springs shown in Figure 4.5-10 at the inboard support assembly.

The solar array area is dictated primarily by the data from the power load analysis, when increased by the design factor of 2.18 to account for all the

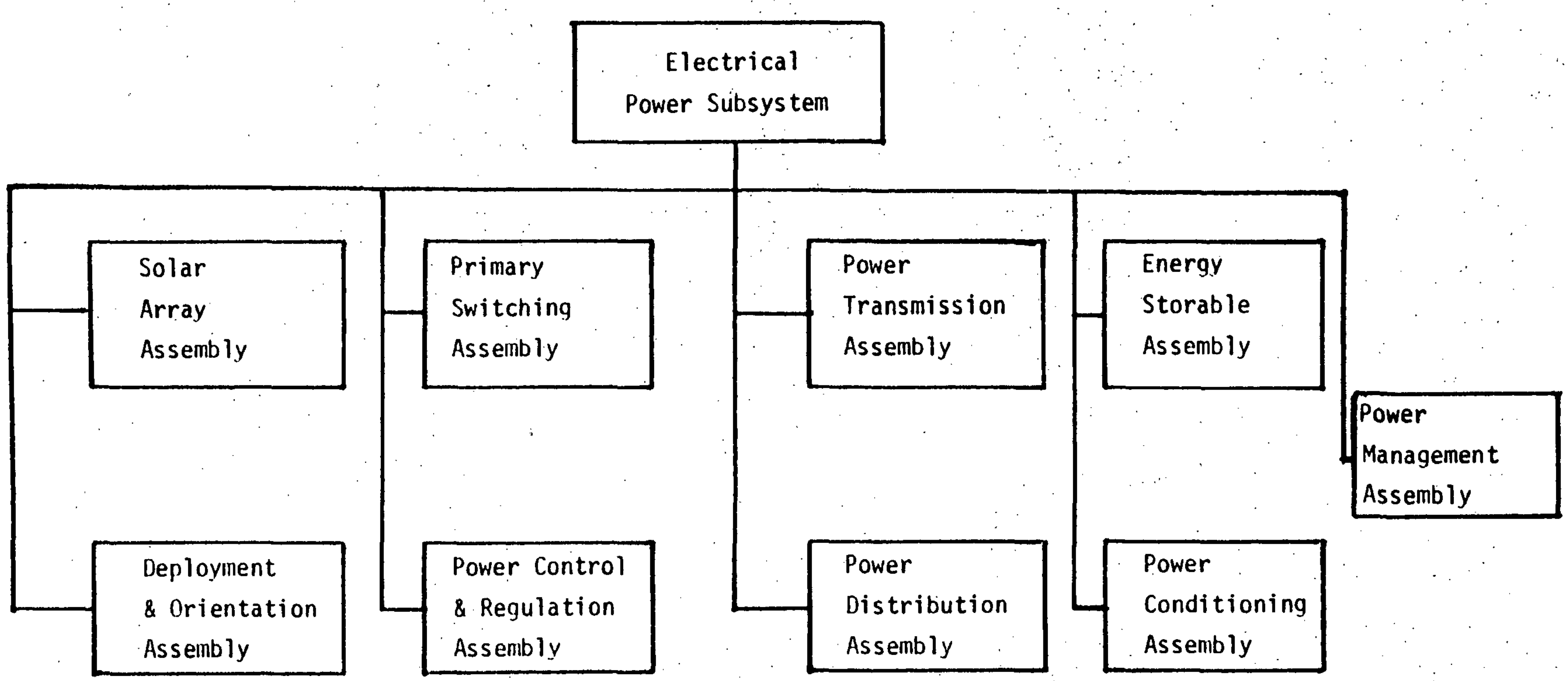


Figure 4.5-8 Electrical Power Subsystem Assembly Tree

Table 4.5-16
EPS DESIGN SUMMARY-1

Electrical load	
"Design to" value - July 15, 1971	ISS - 17.3 kw; GSS - 32.1 kw
Present value - October 27, 1971	ISS - 16.7 kw; GSS - 30.8 kw
Solar array	
Area - ft^2	1 x 5300 (add 1 x 5300 for GSS)
Type	Flexible
Form	Fold-out
Voltage	120 ±1 percent vdc
Energy storage	NiCd - 100 amp-hr (ISS - 24; GSS - 40) "Parallel" charge; "series" discharge 15 percent depth; 2.5-year life Electronic switching Cell voltage cutoff; 3rd Electrode backup
Solar orientation	2-Axis gimbals (±180, ±235); sync drive Trailing cables and spiral coil MDAC turret and drive
Primary switching	Remote control Local manual isolation Electromagnetic for high power Solid-state for low power

EPS losses from the deployment mast to the load buses, and for the storage of energy in the battery which is necessary to provide power during Earth eclipse. The area is further dependent upon the cell arrangement and interconnections, and upon the power collection harness power losses from the solar modules to the root of the wing deployment mast. The area required is shown on Figure 4.5-9 to be 5,300 ft^2, composed of twelve panels (strips) of 441.7 ft^2 each. Each panel is composed of 32 folds, each 23.4 in. by 88 in. in size, or 28.6 ft^2. The panel nomenclature is shown in Figure 4.5-28 of Subsection 4.5.4.1. A typical solar panel and solar cell interconnection plan

Table 4.5-17

EPS DESIGN SUMMARY-II

Regulation and control	Local differential protection Sequential partial shunt regulation (array) Local closed-loop regulation (supervised) Bus: remote control; load: manual control Battery: series load regulation
Transmission	115 vdc Nominal; dual - redundant
Conditioning	Modular Parallel, load-sharing inverters Self-regulated Current-limit protection
Distribution	80-percent dc, 20 percent ac
Load bus voltages	115 ±3 vdc 115/200 ±2-1/2-percent vac; 400 ±1-percent Hz, sine wave and quasi-square wave 115/200 ±5 percent vac; 60 ±1-percent Hz (GPL), sine wave
Power management	Data Management Subsystem (DMS) Central processor support Primary control center in Crew/ Operations Module Secondary control center in GPL

is shown in Figure 4.5-11. The electrical circuitry separates the panel into two lengthwise half-panel strips, each composed of 32 half-folds. Starting at the inboard support assembly nearest to the turret, the 104 cell-group strings in each of four half-folds are connected in series to provide a 121 v (nominal) module. Each cell-group is composed of seven 2 cm by 4 cm solar cells connected in parallel. There are 16 modules and two SPSR regulated circuits per panel; each circuit for the eight paralleled modules of one half-panel. These eight modules are connected in parallel at each fold hinge-line at 30.25 v intervals, as shown in Figure 4.5-11, to reduce sensitivity to shadowed cells and to eliminate "hot-spot" failures.

1 ARRAY	5, 300 FT^2 ACTIVE
2 WINGS	3, 050 FT^2 GROSS
4 QUADRANTS	1, 415 FT^2 GROSS
12 STRIPS	458 FT^2 GROSS
	441.7 FT^2 ACTIVE

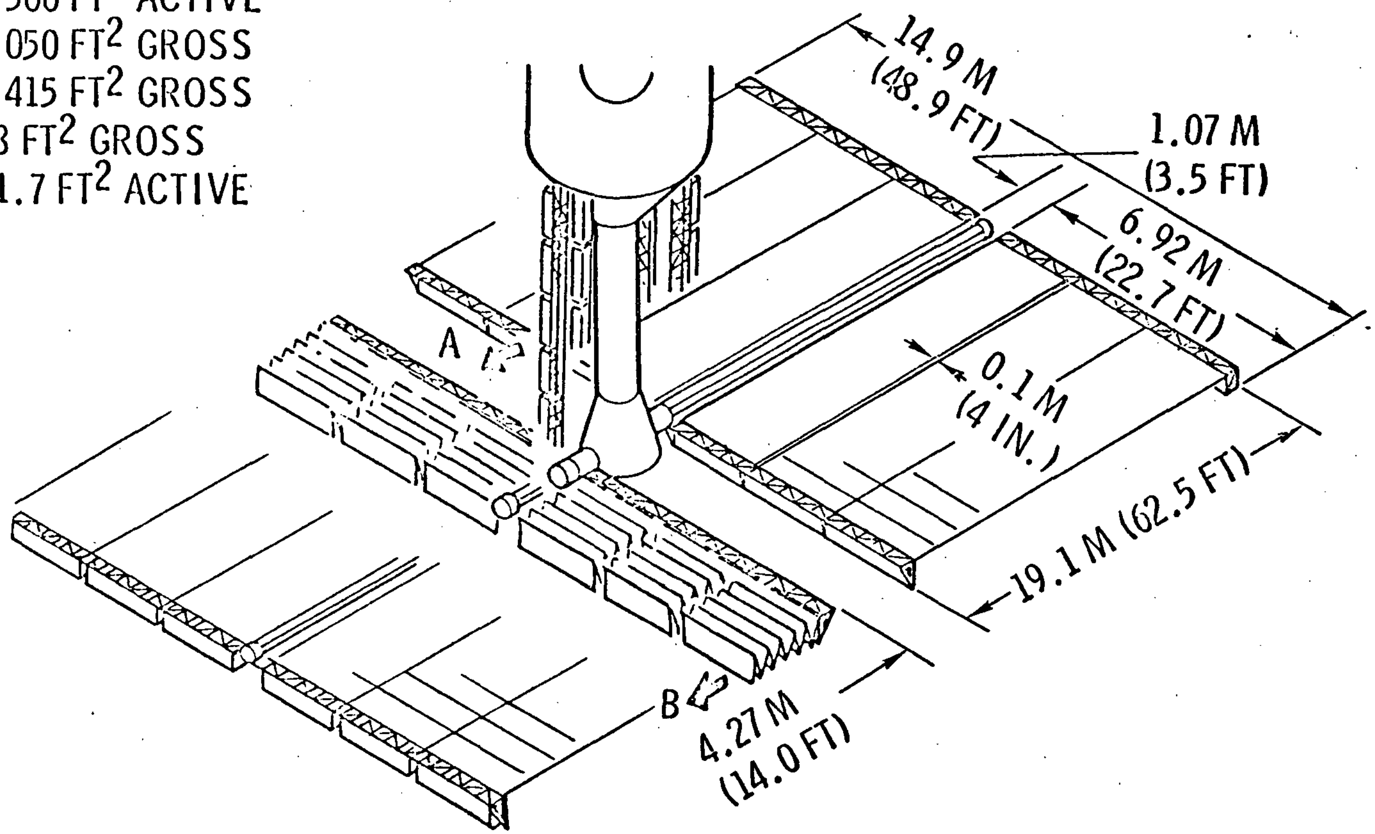


Figure 4.5-9 Solar Array Deployment Sequence

R260

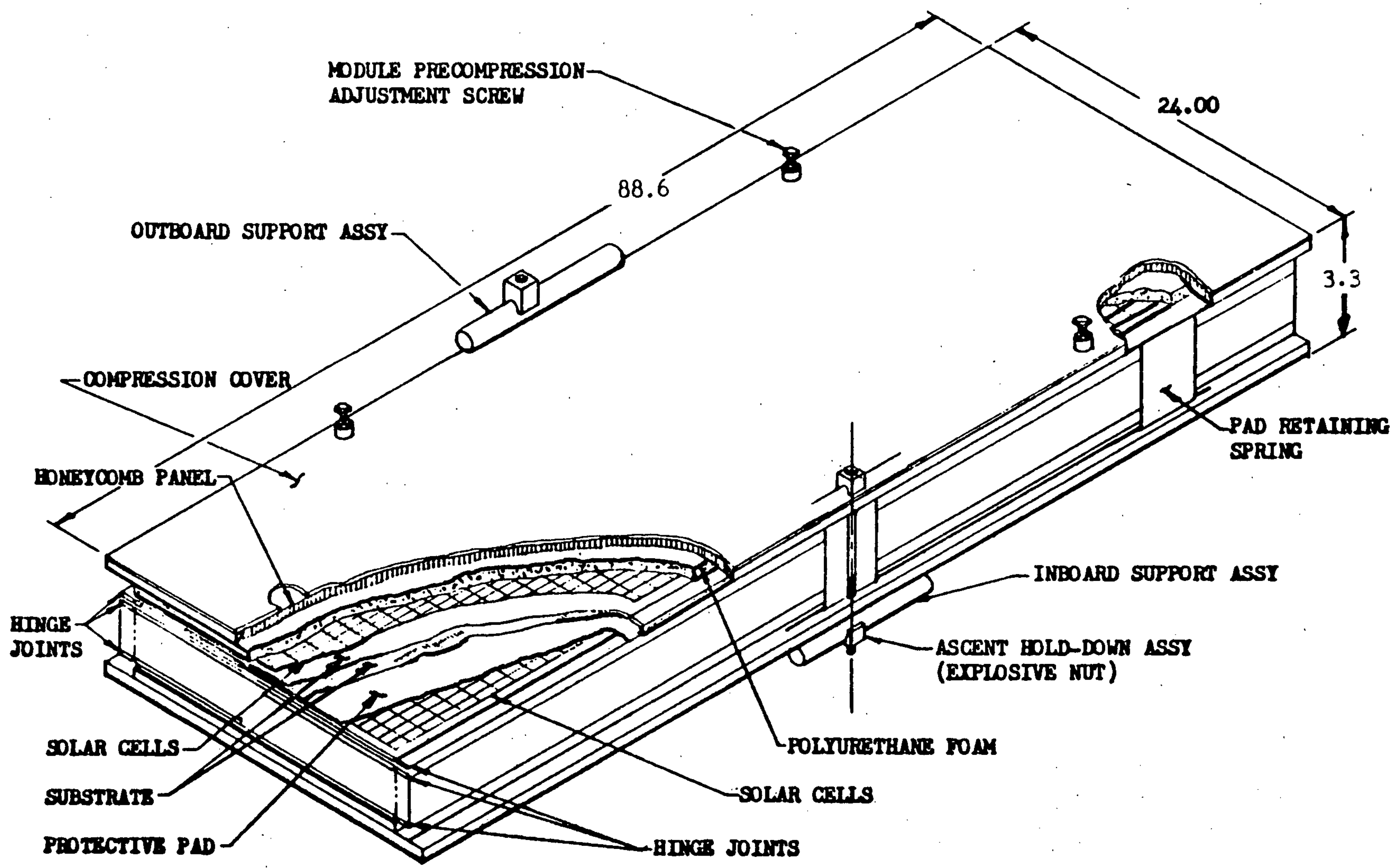


Figure 4.5-10 Array Packaging Assembly

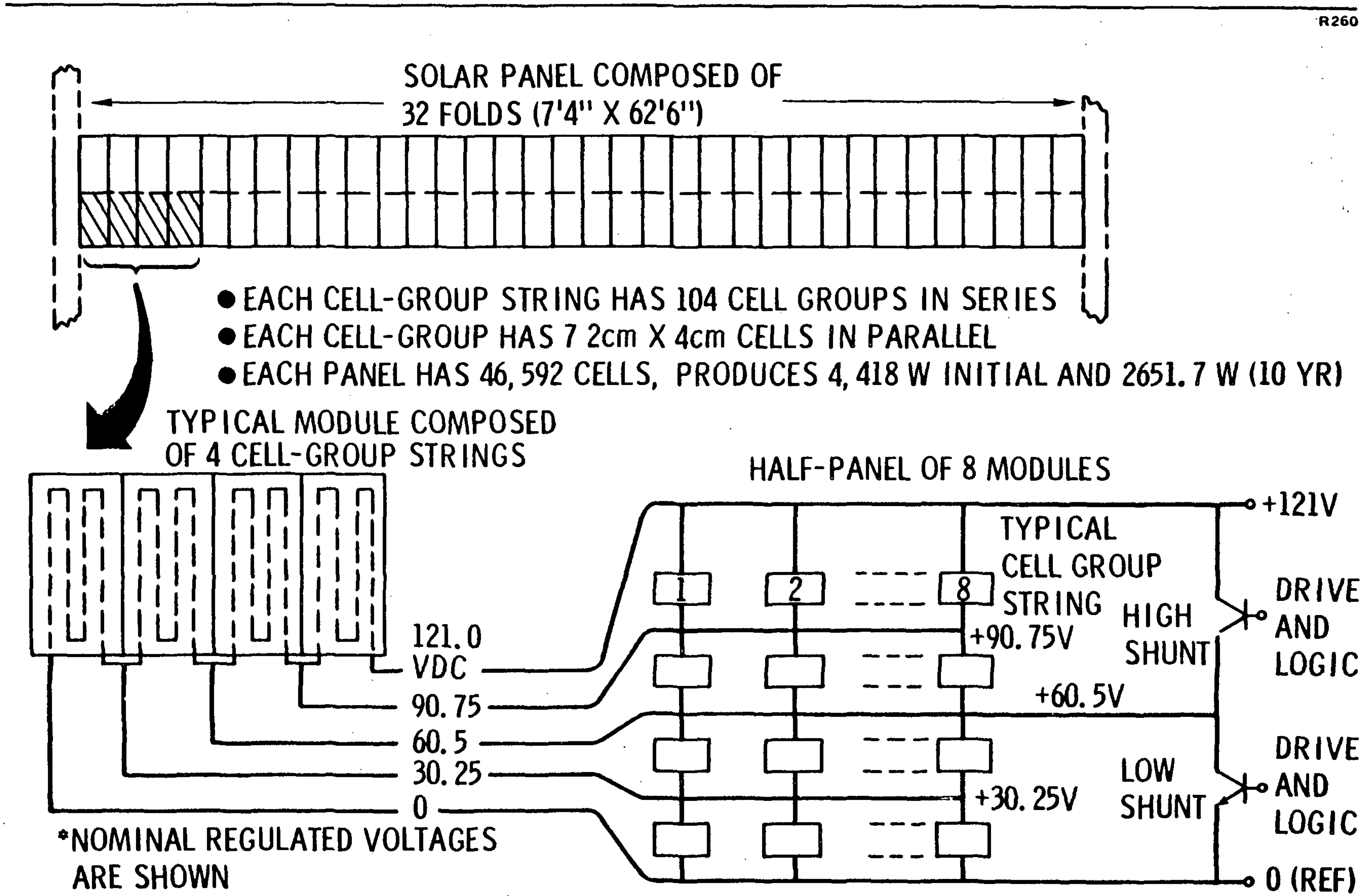


Figure 4.5-11 Solar Panel Interconnection

The high voltage (121 v) end of the eight-module half-panel is connected to the high-shunt terminal of the associated SPSR circuit, the middle voltage (60.5 v) point is connected to the common terminal (between high and low shunts) of the associated SPSR circuit, and the low voltage point (0 v reference) is connected to the low shunt terminal. The SPSR subassemblies are located in the turret, where heat sinks will dispose of the relatively low power dissipation required to regulate voltage by the SPSR technique. This regulation system is described under the topic "Power Control and Regulation Assembly Design." The nominal values of solar array voltage, current, and power (initial, at 5 years and at 10 years), with the dimensions, the areas, the number of strings, and the number of solar cells (total, parallel, and series) are shown in Table 4.5-18.

Details of the LMSC solar array design are shown in Figure 4.5-12, and are incorporated in the MDAC baseline solar array design. The flexible electrical harness, hinge joint, substrate assembly, the packaging assembly, the Astromast deployment system, and the general design features for structural stability and tensioning devices are adopted in the 5,300 ft^2 MDAC design. The module width (88 in.) and the number of cells per string are unique to the MDAC design to meet the voltage requirements. However, the LMSC fold dimension of 23.4 in. is retained. The deployed length of Astromast (62 ft - 6 in.) is less than the LMSC design (84 in.). The turret and drive system described in Subsection 4.2.3 is a MDAC design for pressurized access and shirt-sleeve maintenance while the array is operational.

Deployment and Orientation Assembly Description

The deployment and orientation assembly is composed of structural and mechanical elements which are described in Subsection 4.2.3. Therefore, only the electrical and operational aspects will be described below. The major elements are shown in Figures 4.5-13, -14, and -15.

The panel support assemblies (inner and outer beams, shown in Figure 4.5-13A) reacting through the adapters shown in Figures 4.5-13B and 4.5-13C are first rotated apart from their stowed position as shown by Figure 4.5-14A until they are in the position shown at the top of Figure 4.5-14A. An initial beam extension assembly, shown in Figure 4.5-13C

Table 4.5-18

SOLAR ARRAY

2 cm by 4 cm 75° C AMO 11-Percent Cells

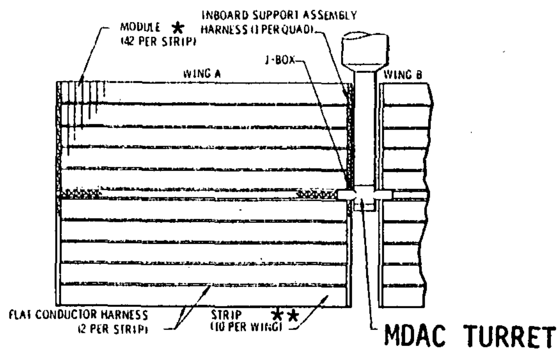
	Section	Module	Panel	Quadrant	Wing	Array
Parallel cells (2 x 4)	7	7	112	336	672	1,344
Series cells (2 x 4)	104*	416	416	416	416	416
Total cells	728	2,912	46,592	139,776	279,552	559,104
Total strings	1/4	1	16	48	96	192
Dimensions – In.	44 in. by 23.4 in.	44 in. by 93.6 in.	88 in. by 749 in.			
Area – m^2 (ft^2)	0.84 (7.15)	2.65 (28.6)	42.6 (458)	127.6 (1,374)	255.2 (2,748)	510.4 (5,496)
Voltage – initial – v	36.9	147.6	147.6	147.6	147.6	147.6
Current – initial – amp	1.9	1.9	29.6	88.8	177.6	355.2
Power – initial – w	69.0	276.1	4,417.9	13,253.8	26,507.5	53,015.0
Voltage – 5 year – v	32.97	131.9	131.9	131.9	131.9	131.9
Current – 5 year – amp	1.5	1.5	23.5	70.6	141.1	282.2
Power – 5 year – w	48.3	193.2	3,091.8	9,275	18,551.0	37,102.1
Voltage – 10 year – v	31.31	125.2	125.2	125.2	125.2	125.2
Current – 10 year – amp	1.3	1.3	21.1	63.4	126.7	253.4
Power – 10 year – w	41.42	165.7	2,651.0	7,953.1	15,906.2	31,812.5

NOTES:

*Strings are connected in parallel at 104-cell intervals to reduce shadow failure hazard.

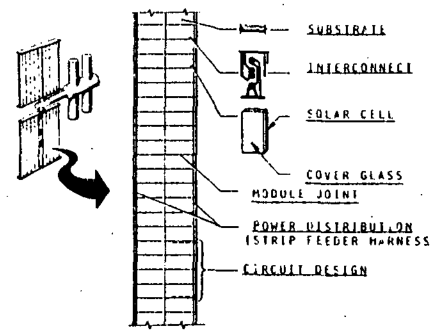
Approximately 1.67 percent less power is available at the mast/turret interface due to voltage drop in the electrical conductors on the panels and the inboard support assembly.

POWER SYSTEM NOMENCLATURE

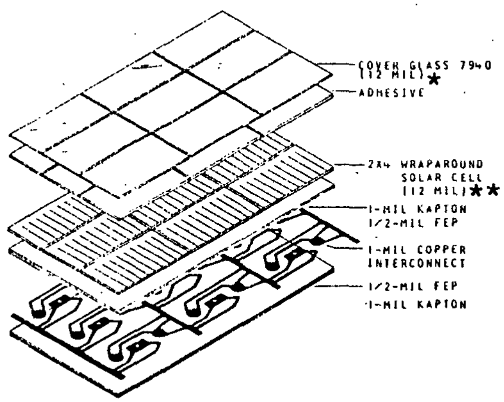


* 32 MODULES PER STRIP IN MDAC DESIGN.
** 6 STRIPS PER WING IN MDAC DESIGN.

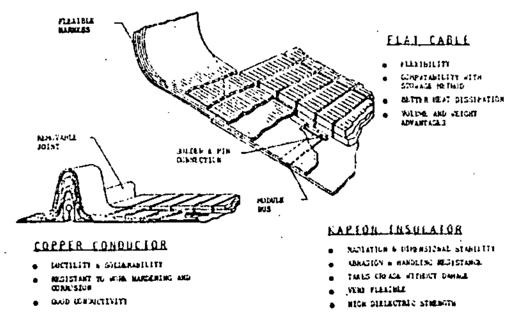
SPACE STATION SOLAR ARRAY BASELINE ELECTRICAL ELEMENTS



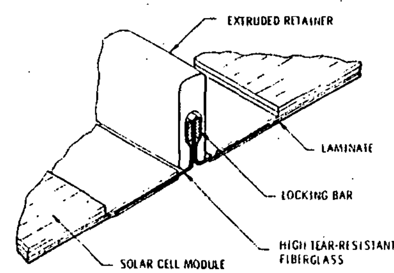
SUBSTRATE ASSEMBLY EXPLODED VIEW



FLEXIBLE FEEDER HARNESS AND CONNECTION

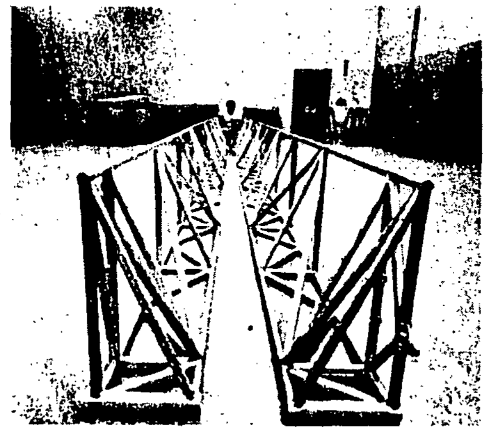


HINGE JOINT



* 6 MIL COVER GLASS IN MDAC DESIGN
** 8 MIL CELLS IN MDAC DESIGN

Figure 4.5-12 Baseline Design - Substrate



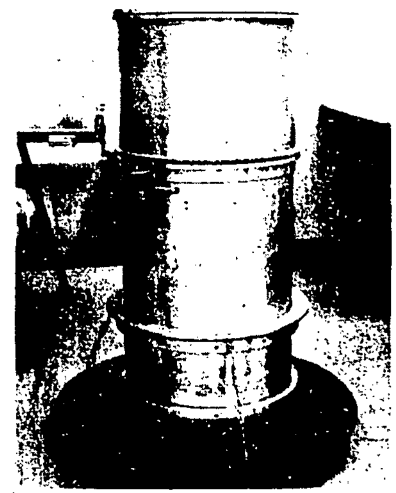


Figure 4.5-13 Major Hardware Fabrication Status

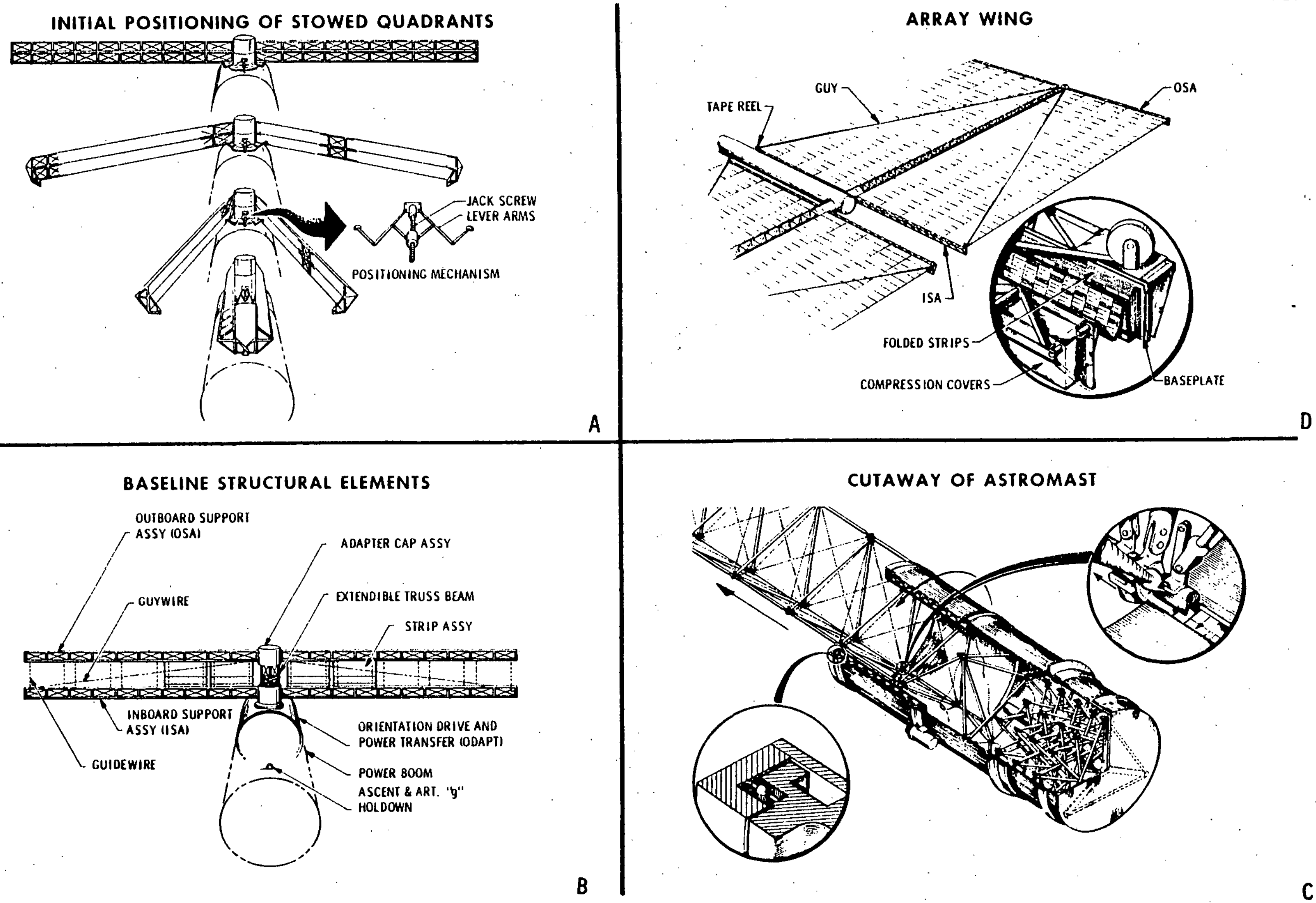


Figure 4.5-14 Baseline Design - Structure

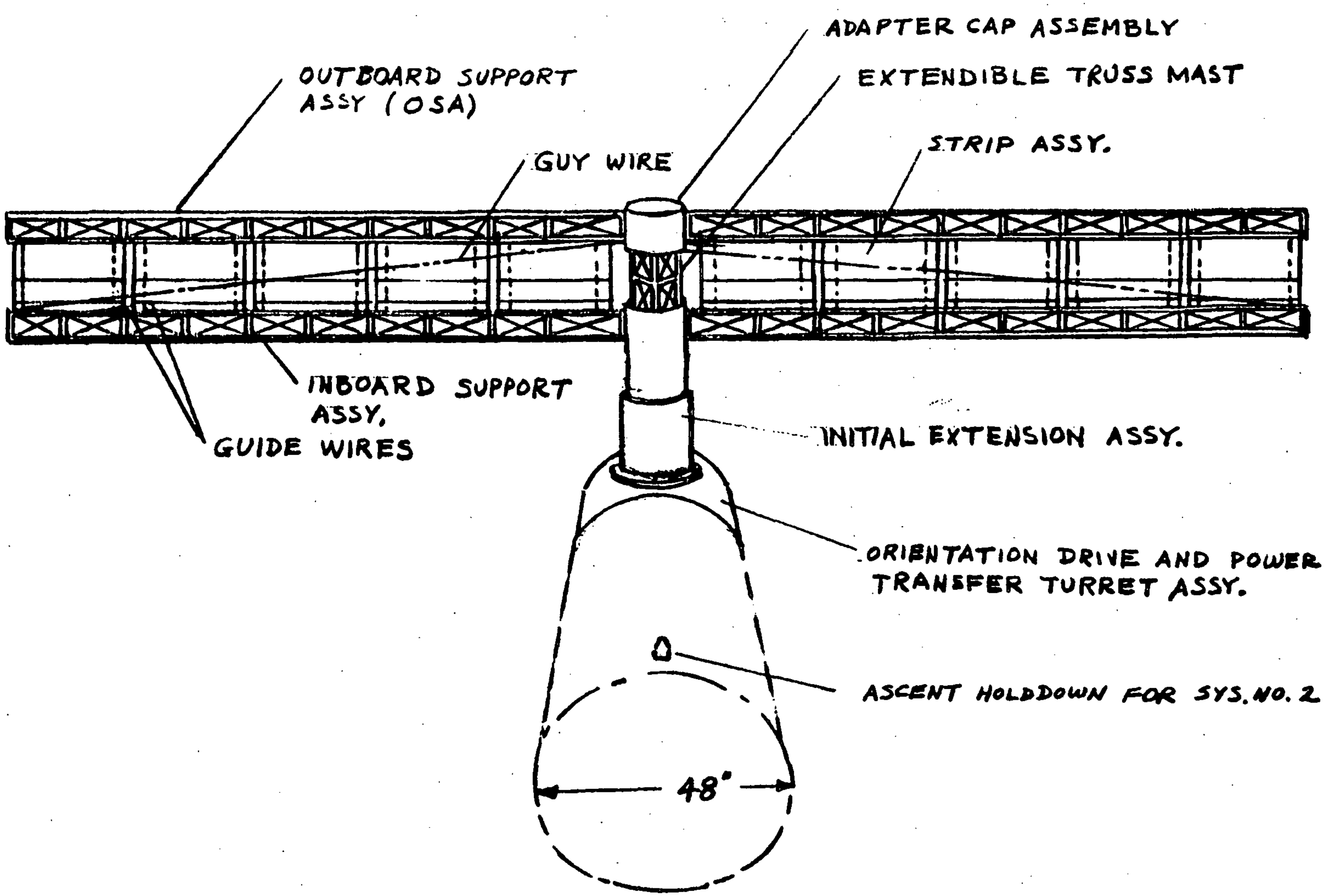


Figure 4.5-15 Array Structural Elements

and in Figure 4.5-15, is then deployed to increase the array standoff from the stowed position. The panels are next deployed simultaneously from their launch cases, as shown in Figure 4.5-14B, extending the Astromast articulated triangular extensible truss beam (shown in Figure 4.5-14C). One fully deployed wing is shown in Figure 4.5-14D. The guy wires and tapes were initially included in the LMSC development model to allow for an artificial-g experiment with a partially-deployed array. However, they are retained for added stability and to improve the dynamic characteristics of the array; although no artificial-g conditions are guidelined for this study.

A capability for replacement of individual solar panels by EVA has been provided; EVA replacement is an alternative replacement method, not the baseline method. The baseline replacement method is to retain the ISS array for 10 years and to replicate it at the opposite end of the Modular Space Station with a second Power/Subsystems Module to (1) increase power for growth to GSS, (2) extend the ISS period to 10 years or longer, or (3) replace a severely damaged or inoperative array or Power/Subsystems Module, if required. The provisions for EVA panel replacement are shown in Figure 4.5-16. Tension is maintained between the pull-out reel and the strip tension spring assembly as the take-up wheel draws the panel back into its launch case on the inboard support assembly. The new panel replaceable unit is mounted in place after removal of the old panel unit. The pull-out cable is then attached, the case explosive bolts are energized, and the new panel is deployed by energizing the motor-driven pull-out reel.

Solar Array Orientation Assembly Description

The orientation system has the capability for any Station orientation. Obviously, the least difficult of these for the solar array would be the solar inertial orientation, with the Station supplying the array orientation. The general inertial case is similar, with the two-axis gimbal capability for initial solar acquisition.

The orientation described as perpendicular-to-orbit plane with roll at orbital rate (POP/OR) requires 2-axis gimballing. A rate of 4 deg/min in the α-axis (orbital rate axis) is required. A gimbal range of ±235 degrees allows the use of either a trailing cable (selected) or a spiral coil for power transfer

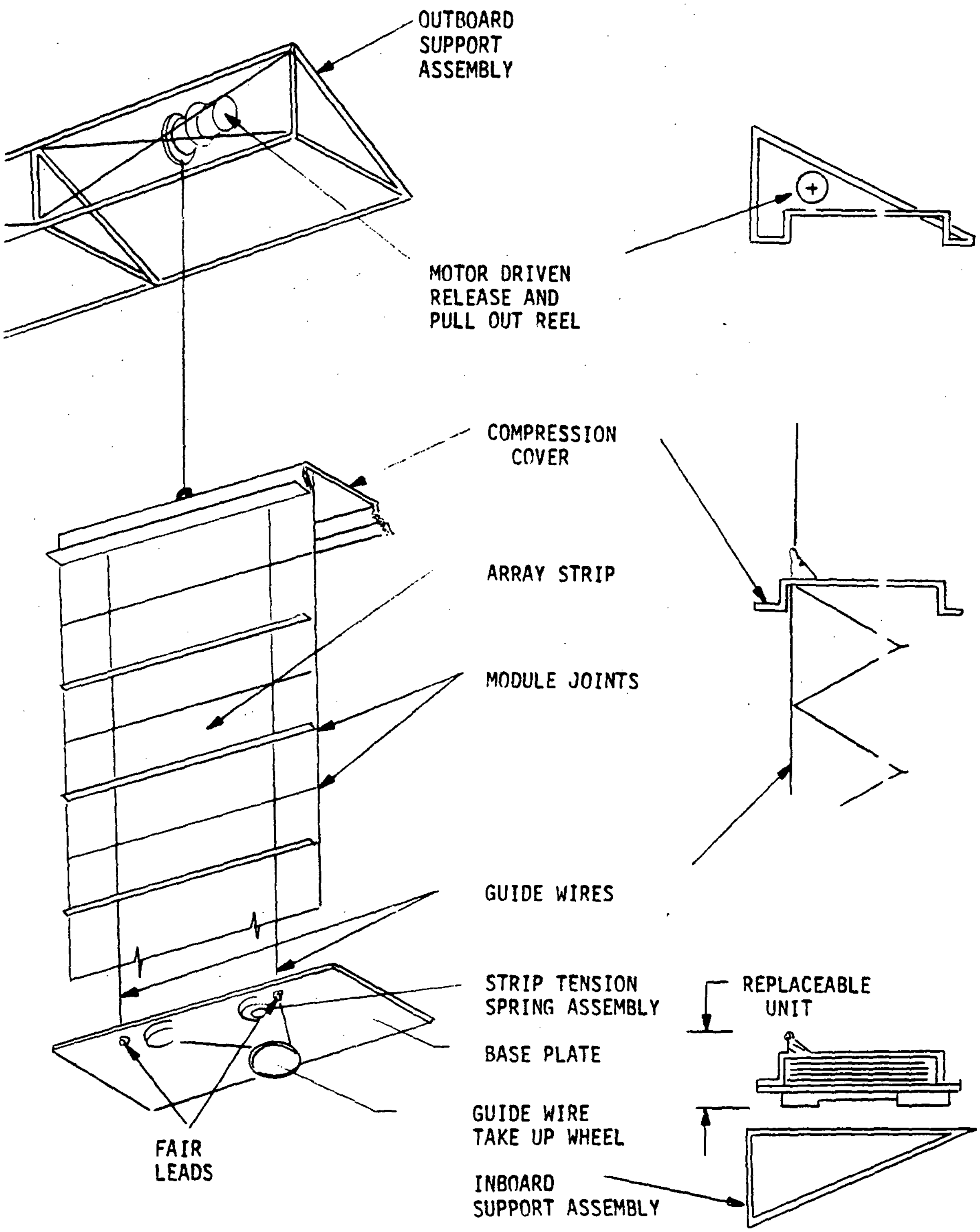


Figure 4.5-16 Strip Replacement

across the tunnel (α) axis. The panels are rotated to minimum drag position (feathered or trailing mode), and are rotated to unwind the power cables (recycled) while in eclipse, at a gimbal drive rate of 22 deg/min maximum. The β-axis (orbital declination axis) will operate at 6.3 deg/day over a gimbal range of ±78.5 degrees. A spiral coil (selected) with a trailing cable provides power transfer across the mast (β) axis. Cable recycling is not required in the β-axis for a POP/OR station orientation.

The baseline station orientation (horizontal or "velocity-vector") is the most demanding upon the solar array orientation system. The α and β axes are not readily identifiable as in the POP/OR case, because they are required to provide combined or composite rotational modes. The mast gimbal range will be from zero while in the high β, continuous sunlight orbit, to the full orbit value of ±235 degrees while in the low-β maximum eclipse orbit. Recycling is provided while in eclipse and a spiral coil is used for power transfer across the mast axis. At both of these limiting orbits, the tunnel gimbal ranges are zero. Between these limiting orbits, the mast and tunnel gimbals share the orientation duty in a composite rotational mode. The range of the tunnel gimbal axis, however, is ±235 degrees, and for the mast gimbal axis it is ±180 degrees. Recycling is provided while in Earth eclipse for trailing cable power transfer across the tunnel gimbals. The gimbal rates are approximately 0 deg/min to 22 deg/min in both axes (with 4 deg/min average). The higher rate (22 deg/min) is used for unwinding the power transfer cables (recycling) and for "feathering" to the low drag position while in Earth eclipse. Provisions for horizontal Station orientation thus provide an operational flexibility which is sufficient to eliminate constraints upon the station orientation.

The DMS computer capability is used to calculate solar orientation angles and to provide commands to the drive servo assembly to assure solar acquisition upon leaving Earth eclipse. Sun sensors on the array and servo control logic in the turret are provided to track the sun during the illuminated periods.

Primary Switching Assembly Description

The primary switching assembly is provided in the turret area for (1) control of power flow from the 24 control and regulation (SPSR) subassemblies to the source buses, (2) control of source bus connections for either parallel or isolated operation, and (3) control of power flow from the source buses to the transmission lines for parallel operation or to deenergize any of the four cables. Primary switching is also provided in each Station Module to (1) sectionalize the transmission lines, (2) control power flow to the main distribution centers from the selected transmission cables, and (3) sectionalize the main distributor buses. Differential protection loops are provided to deenergize faulted transmission lines, source buses, and main distributor buses. The electrical schematic in Figure 4.5-3 shows the primary switching assembly circuit breakers.

Energy Storage Assembly Description

The energy storage assembly in each major Space Station Module (excluding RAM's, Logistics Modules, and Crew/Operations Modules) is made up of eight hermetically-sealed, cold-plated, 100 ampere-hour, nickel-cadmium space batteries; eight battery-chargers; and two pulse-width-modulated (PWM) series buck battery line regulators. These assemblies are each co-located with, and designed to operate in conjunction with, the main distribution center for the Station Module.

Each battery consists of twenty-eight 100 ampere-hour cells made up into seven economically replaceable modules of four cells each, weighing about 40 lb per module. The selected battery depth of discharge is 15-percent average and 35-percent maximum, for a nominal lifetime of 2.5 years. The provisions for a greater depth of discharge during buildup, emergency, or contingency operation (e.g., array shadowing) will not exceed 30-percent average or 70-percent maximum to assure a minimum lifetime of one year.

The batteries are charged concurrently by individual battery chargers at approximately 42 volts, using solar array energy in excess of the Station and experiment power demands. The system capacity is designed to support the 24-hour average power demands and to return the batteries to full charge daily. The batteries are discharged with four batteries in series to the associated Main Distributor Bus at 115 ±3 vdc through the PWM series buck-battery load regulators. These batteries provide all of the electrical power to the Power Conditioning System and the loads during the dark (eclipsed) portion of each orbit. In addition, they supply (1) power for extraordinary peaks during partial reductions of normal solar power, (2) supplemental power during short periods of partial solar array shadowing, (3) emergency Station power in the event of loss of normal solar array power, (4) primary launch and ascent power for the Power/Subsystems Module (supplemented by Space Shuttle power if required), and (5) end-of-mission power when solar arrays are retracted for recovery.

The charging and discharging control functions are automatic within the normal operating range at the most economical rate compatible with the charging source, the electrical loads, and the power available. The individually monitored battery cells provide the voltage limit signals near the end of charge which result in the preferred mode of charge termination. An alternate mode of trickle charging is also available. Cell over-temperature signals will also terminate charging. The third electrode signals provide an approximation of the state of charge, depending upon the previous history of the cells, and are used for a backup mode of charge termination.

The battery modules are temperature-controlled (cold plated) by means of an active coolant loop with a design point of 13°C and a range of 10° to 20°C. The batteries are accessible for periodic inspection, maintenance, and/or replacement. The hermetically-sealed cells are contained in hermetically-sealed 4-cell modules to assure containment of toxic and corrosive KOH in the event of cell rupture. Loss of cell electrolyte automatically gives a failure mode indication that terminates cell charge or discharge. Each cell contains a fourth (recombination) electrode to prevent gas buildup during normal charging conditions.

Instrumentation is provided for normal operation, energy storage status evaluation, remote control data requirements, and telemetry data for ground monitoring.

The emergency power capability of the ISS from a full state-of-charge is about 72 kw-hr or about 46.5 kw-hr from the 35-percent nominal maximum depth of discharge.

The battery switching subassembly (Figure 4.5-17) provides additional cells to replace shorted or open-circuited cells; it also provides for remotely-controlled operation of the batteries, as required during the unmanned periods.

The initial battery complement launched with the Power/Subsystems Module will be four batteries plus spare modules connected in a series string. Five battery chargers (four active and one redundant) and two line regulators (one active and one redundant) will be provided for reliability. They are connected as shown in Figure 4.5-18. The Crew/Operations Module and the GPL are launched initially without batteries but with provisions for future additions of the batteries, and battery chargers. Prior to or concurrent with initial manning for ISS, the Logistics Module missions will supply the remaining 20 batteries (4 for the Power/Subsystem Module, 8 for the Crew/Operations Module and 8 for GPL). These will be installed and placed into operation by the Station activation crew.

Power Control and Regulation Assembly Description

The power control and regulation assembly provides solar array source voltage regulation to the source buses and the transmission assembly. The regulation system is based upon a sequential partial shunt regulation (SPSR) technique. The input power to this regulation method requires the solar panels to be sectionalized into a high-voltage section and a low-voltage section, as shown previously in Figure 4.5-11. The drive and logic subassembly provides for sensing, mode selection, and sequencing of the drive signals for the 12 low-shunts per wing and the 12 high-shunts per wing. Normal regulation is performed by the low-shunt array segments, controlling 50 percent of total array power. When the solar array is very cold after eclipse and/or is new (undegraded), the power capability and voltage are both higher than the design point for a 10-year old array. When the array is cold, the high shunts may be required to operate also. However, the open circuit voltage of a new array can be controlled by the low shunts alone. The basic shunt control modes are Full Off, Linear (proportional) or Full On (saturated). At each control condition, only one shunt per wing will be in the linear, proportional control condition. Of the remainder, enough shunts are in the Full On condition to suppress potential overvoltage, and enough are in the Full Off condition to provide adequate power to the Space Station at the nominal array output voltage of 121 vdc. The power dissipation of a saturated (Full On) shunt is very low, equivalent to approximately 3 volts at I_{SC} (short-circuit current) and about 50-watts for the shunted half-panel.

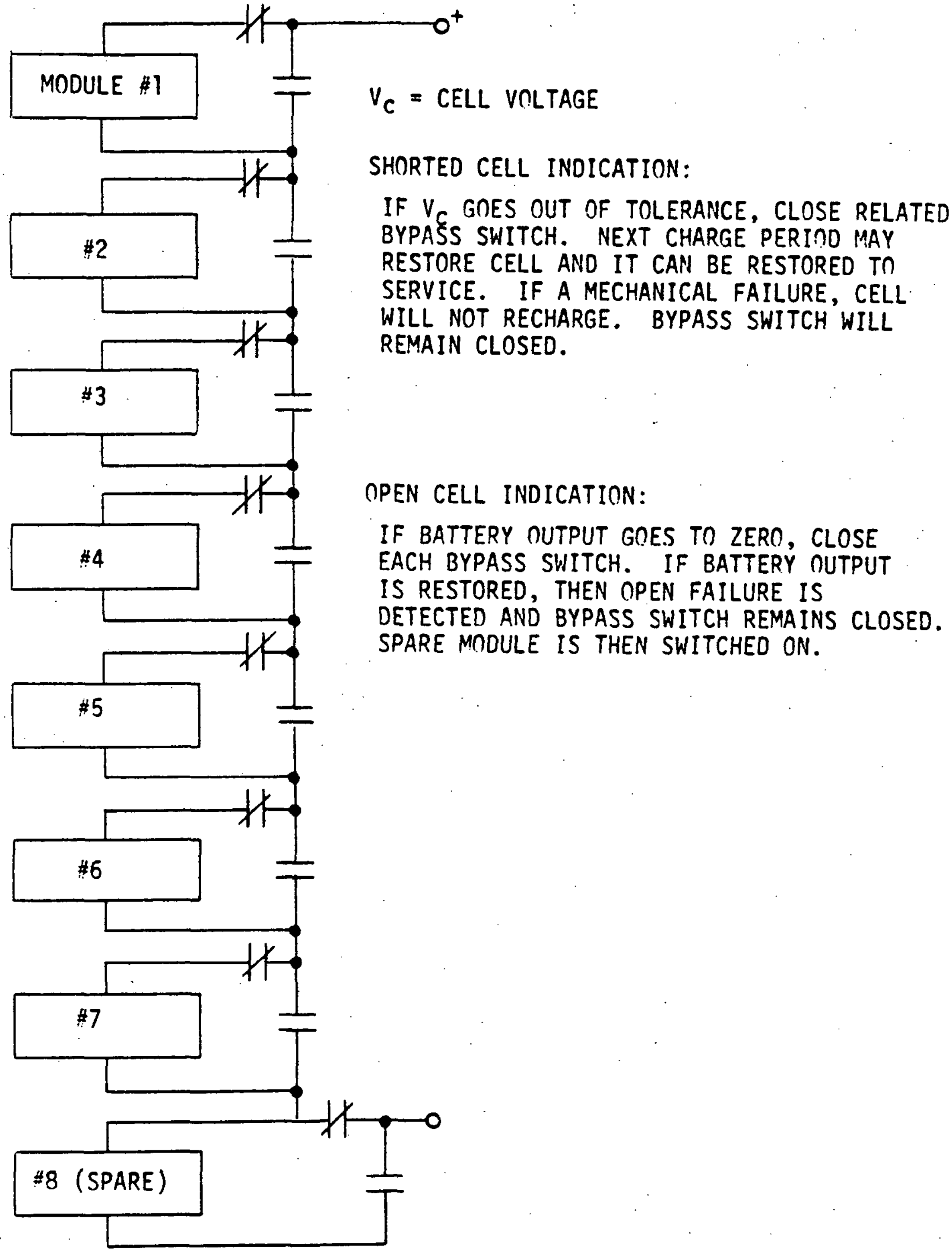


Figure 4.5-17 Battery Cell Failure Accommodation

R260

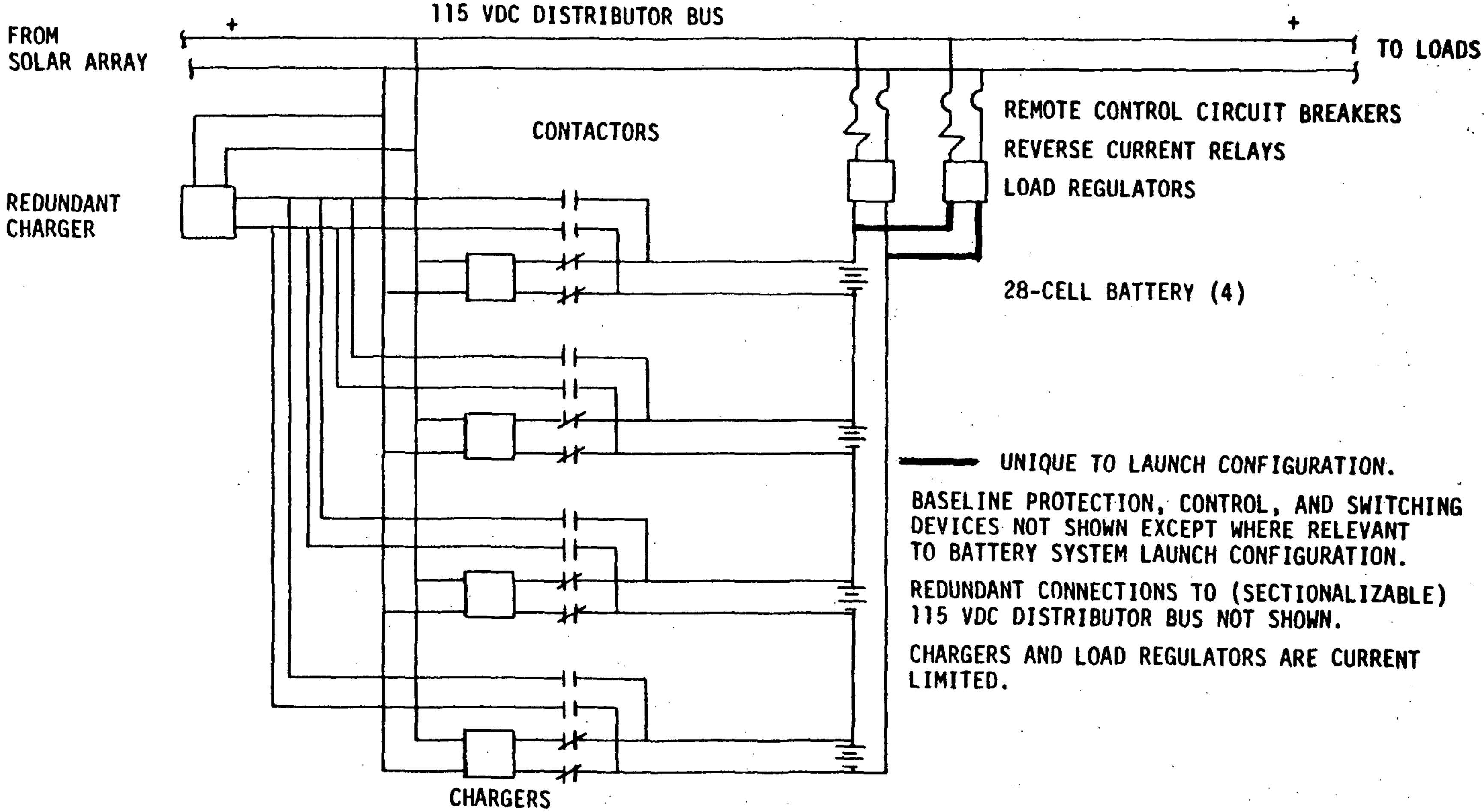


Figure 4.5-18 Module 1 Battery System - Launch Configuration

Power Transmission Assembly Description

The power transmission assembly includes the cabling primary buses, junction units, and primary protection. These are closely related to the primary switching functions described in the primary switching assembly. The single-line schematic diagram was shown in Section 4.5.1, Figure 4.5-3.

The power transmission assembly transmits regulated bulk power from the source buses located in the power turret to the main distribution centers located in each Station Module. The transmission assembly includes four power-transmission cables, any two of which are capable of transmitting the total ISS power requirement within the design voltage band (112 to 118 vdc). Therefore, the transmission assembly is described as dual redundant. The cables are extendible between Station Modules through the four common connector interfaces at each docking port. One manually-secured cable connector is provided in each docking port interface quadrant. The efficiency, reflecting the transmission line power loss, is optimized within reasonable weight and mechanical handling/installation properties by sizing the cable conductors to maintain the required voltage "spread" between Station Modules at maximum load. This is shown in the power flow diagram in subsection 4.5.1, Figure 4.5-5. The selected cable size is AL-1.

Input power to the transmission assembly is 35.87 kw at 118.7 vdc nominal ±5 percent. The voltage is regulated to maintain a minimum voltage of 112 vdc at the most remote (GPL) main distribution center.

The nominal output power transmitted by the transmission assembly during solar illumination is 10.63 kw to the Power/Subsystems Module, 12.14 kw to the Crew/Operations Module, and 11.49 kw to the GPL.

The transmission circuits are protected by coordinated individual electrical differential protection zones, enclosing source buses and main distribution center buses within the zones of protection.

The transmission cables are afforded mechanical protection against abrasion or other damage, and are arranged for on-orbit removal and replacement, if necessary. However, with only two cables of four required for full power, replacement is not expected to be necessary.

Grounding

A single-point ground is established at the source buses for the negative (low-voltage) terminals of the regulator outputs. The negative source buses are each connected to this ground to allow equivalent grounds during either parallel or isolated operation. This concept is indicated schematically on Figure 4.5-3 in subsection 4.5.1. The purpose of this ground is solely to provide a path for fault current, thus assuring positive tripping of circuit breakers supplying a faulty conductor. Negative conductors are provided in the transmission lines with full load current capacity, and structure is therefore not used for power transmission.

Power Conditioning Assembly Description

The power conditioning assembly includes the quasi-square wave and sine-wave inverters, the battery chargers, and the battery discharge PWM series buck load regulators. The battery chargers and load regulator functions were described under the topic "Energy Storage Assembly Description."

The power conditioning equipment characteristics, quantities, and locations are shown in Table 4.5-19. All subassemblies are capable of replacement in orbit and spare subassemblies are provided to meet the reliability requirements, in addition to the installed redundancy indicated. Selected components can also be replaced within the subassemblies when identified by onboard checkout techniques (e.g., inverter clocks, regulator voltage reference circuits, bias power supplies). The functional subassemblies are designed for self protection by a capability to limit and withstand short-circuit currents for a protective device coordination period, followed by self-tripping before internal damage occurs. Circuit-breakers are provided on the power input side. Contactors are used to switch the output power. Reverse-current relays are used in the battery discharge (buck) regulator outputs to prevent solar arrays or other batteries from discharging into an internal fault.

Table 4.5-19

POWER CONDITIONING EQUIPMENT

Subassembly	Rating	Efficiency Average Load			Spec Wt (lb/kw)	Quantity		
		Power/ Subsystems Module	Crew/ Operations Module	GPL		Power/ Subsystems Module	Crew/ Operations Module	GPL
Inverters:								
400 Hz Sine	750 w/ unit	0.81,	0.75,	0.76	15.6	3(1)*	3(1)*	3(1)*
400 Hz Square	750 w/ unit	0.83,	0.88,	0.85	10.0	3(1)*	2(1)	2(1)
60 Hz Sine	500 w/ unit			0.75	36.0	-	-	1(1)
Battery Charger	1,000 w	0.91	0.91	0.91	5.0	8**	8†	8†
Battery Load Regulator	6,000 w	0.94	0.94	0.94	2.2	2	2	2

*3(1) indicates 3 required plus 1 redundant
**4(1) initial launch, add 3 prior to ISS
†0 initial launch, add 8 prior to ISS

Power conditioning equipment is arranged for parallel operation within each Station Module, but is isolated from the similar buses in other Station Modules (except for RAM's, Logistic Modules, and Crew/Operations Modules. Power transfer between major Station Modules use the transmission assembly only.

The solid-state, power-conditioning subassemblies are cold-plated for temperature control. The temperature of the fluid coolant interfaces with EC/LS Subsystem are 15.6°C (60°F) minimum inlet temperature and 43.3°C (110°F) maximum outlet temperature. Design for heat removal from the cold-plate mounted subassemblies is based on 90-percent conducted and 10-percent radiated heat transfer.

The conditioning equipment is capable of continuous operation in the normal pressurized Station environment or in a partial or full vacuum, and in a zero-g, partial-g, or one-g environment.

Power Distribution Assembly Description

The power distribution assembly includes cabling, load buses, load bus switching, load bus protection, distribution circuit protection, battery switching, and the distribution panels for final distribution to the loads.

The power distribution assembly (1) distributes bulk power from the main distributor bus, where it interfaces with the power transmission assembly, the energy storage assembly, and the conditioning inverters and regulators; (2) distributes both regulated dc power and regulated ac power to the load distribution buses, and (3) provides the load bus facilities to supply load conductors.

Power input to the power distribution assembly is provided in the range of 112 to 118 vdc for the dc load bus and the conditioning assembly; whether it is derived from the batteries through battery load regulators or from the solar array through the SPSR regulation circuits.

Power output from the power distribution assembly is provided of the following types and quantities.

A. 115 ± 3.0 vdc
B. 400 Hz ± 1 percent, 3-phase, 115/200 vac ± 2-1/2 percent, sine wave
C. 400 Hz ± 1 percent, 3-phase, 115/200 vac ± 2-1/2 percent, quasi-square wave
D. 60 Hz ± 1 percent, 1-phase, 115 vac ± 5 percent, sine wave (for GPL instrument power)

Essential load buses and nonessential load buses are provided for each type of power in each Station Module as required by the electrical power load analysis.

The power distribution assembly is controlled automatically through operation of the power management assembly and through internal closed-loop functions. Manual backup and override capabilities are provided for all operational circuits.

Electrical protection is provided for all distribution circuits, with the primary function of protecting the circuit wiring rather than the loads—these have internal controls and any supplementary subcircuit protectors required. Protective devices are coordinated with the power transmission assembly and/or power conditioning assembly protective devices for sequential tripping and fault isolation.

A single-point ground is provided for each electrically independent (isolated) system, such as inverter output circuits, as shown on Figure 4.5-3 in subsection 4.5.1. Neutral, common, or return conductors are provided for all circuits. Structure is not used as a conducting medium, except for return of ground fault currents for the purpose of tripping the circuit-breaker which supplies a faulty circuit.

The distribution assembly components are located in the main distribution centers in each station module; except for cabling and possible special distribution subpanels if required to be located at the load for convenience or safety. The mechanical system provides maximum protection against personnel contact with energized parts and mechanical protection against contact- or damage-induced faults.

The distribution assembly is capable of operation in the normal pressurized atmosphere or in partial or full vacuum, and in a zero-g, partial-g, or one-g environment.

Power Management Assembly

The electrical power management function is provided by integrated subassemblies located in the EPS and the DMS. It includes control of (1) the sequential partial-shunt regulators for voltage control of power from the solar arrays, (2) solar array orientation drive control as required by signals from the sun-acquisition and solar-tracking sensors, and (3) battery charging and discharging electronics. It also provides instrumentation, sensors, and electronics for preprocessing of the operating data from the EPS, to be used for the integrated displays and controls, onboard checkout, and data management functions in the DMS which are related to the EPS. The power management assembly also controls the load buses and the EPS configuration, according to established priorities. These functions are performed automatically, with manual backup or over-ride capability for all essential management functions. The sensors and instrumentation required for monitoring all critical operational parameters are included—among these are voltages at all buses and battery cells, currents in all discrete system cables (such as SPSR circuits, transmission lines, conditioning subassemblies, batteries, and bus tie cables), temperatures of batteries, solar array panels, and coolants, and pressures in battery modules.

The power management assembly provides preprocessing functions with local dedicated computers identified as remote data acquisition units (RDAU's) which monitor and screen sensor data for out-of-tolerance indications and/or rates of change which exceed acceptable reference values. These are transferred to the central processing computer for display, analysis, and/or corrective actions.

Power management of the EPS is possible from the Crew/Operations Module (primary), the GPL (secondary) or the Power/Subsystems Module (initial and emergency). Remote monitoring and control command inputs are provided from the ground via the Data Management and Communications Subsystems.

Equipment and Weight Summary

The major EPS equipments are listed in Table 4.5-20. The quantity of components on the initial launch is reduced as indicated to save weight. These components are added later by Logistics Module launches. The weight summary is shown in Table 4.5-21.

4.5.3.2 Interfaces

The Electrical Power Subsystem has major interfaces with: (a) the Space Station subsystems, (b) the attached RAMS and Logistics Modules, and (c) the Space Station Orbiter (during station buildup). The EPS interfaces with other Modular Space Station subsystems are as follows:

A. EC/LS subsystem for temperature control of electronics and batteries.
B. DMS for power management, onboard checkout, integrated displays, and EPS control.
C. All subsystems for electrical power, as defined by the electrical power load analysis, with the power types listed in subsection 4.5.3.1, "Power Distribution Assembly Description."

Station Module Interfaces

The EPS interfaces between Space Station modules are defined by Table 4.5-22. The interface connections are defined by the common docking interface shown in Figure 4.5-19, which is uniform between all docking ports. The active circuits are connected to the power transmission assembly or the power load buses. The Power/Subsystems and Crew/Operations and Crew/Operations and GPL module interfaces at 115 vac are interconnections for transmission assembly extension. All other interface connections are supplied from cables connected to appropriate load buses in the primary station modules.

Table 4.5-20

ISS/GSS BASELINE EPS EQUIPMENT BUILDUP

	Module 1 Power	Module 2 Crew	Module 3 GPL	Module 4 Crew	Module 5 Power
Solar Array	1	--	--	--	1
Seq Part. Shunt Regulator	2	--	--	--	2
Source Bus	1	--	--	--	1
DC Distributor Bus	1	1	1	1	1
Battery (2.5-yr Life)	8^{Δ}	8*	8*	8*	8*
Battery Charger	8**	8*	8*	8*	8*
Battery Load Regulator	2	2*	2*	2*	2*
3 Phase, 400 Hz Sine Wave Inverter Modules	4	4	4	4	4
3 Phase, 400 Hz Square Wave Inverter Modules	$4^{\Delta\Delta}$	3	3	3	2
1 Phase, 60 Hz Sine Wave Inverter (GPL)	--	--	2	2	--
DC Distribution Panels	1	1	1	1	1
AC Distribution Panels	1	1	1	1	1
Power Control Unit	1	1	--	--	--

*Initial complement is zero (baseline) because module is weight limited.

**Initial complement is 5 because only 4 batteries are installed (one charger per battery plus one switchable redundant unit).

$^{\Delta}$Initial complement is 4 batteries.

$^{\Delta\Delta}$Two inverters for pumpdown motors.

Table 4.5-21

ELECTRICAL POWER SUBSYSTEM WEIGHT SUMMARY

(Pounds per Module)

	ISS and GSS			GSS Only	
	Power/Subsystem*	Crew/Operation	GPL	Crew/Operations	Power/Subsystem*
Solar Array					
Panels and Support	2,377	---	---	---	2,377
Masts	679	---	---	---	679
Orientation					
Turret and Orientation	1,544	---	---	---	1,544
Tunnel and Hatch	354	---	---	---	354
Batteries					
Battery Modules	2,800	2,800	2,800	2,800	2,800
Connecting Hardware	240	240	240	240	240
Primary Switching					
Source Bus/Contactors	164	---	---	---	164
Distribution Bus/Contactors	164	153	110	153	160
Regulation and Control					
Seq Part. Shunt Regulator	308	---	---	---	308
Power Control Unit	15	15	---	---	15
Transmission					
Cables	231	211	158	211	231
Conditioning					
Battery Charger	40	40	40	40	40
Battery Load Regulator	26	26	26	26	26
Square Wave Inverter	32	24	24	24	32
Sine Wave Inverter	48	48	48	48	48
GPL Inverter	---	---	36	---	---
Support Structure	30	28	28	28	30
Distribution					
Wiring to Distribution Panels	35	26	28	28	33
Distribution Panels and Breakers	119	121	162	172	130
Total	9,206	3,732	3,700	3,770	9,211

NOTE: GSS weights are extrapolated from ISS weights on the basis of similarity.
*5,300 ft^2 solar array.

Table 4.5-22

EPS INTERMODULAR INTERFACES

Interface	115 VDC 2 Wire	115/200 vac 3Ø, 4-Wire 400 Hz Sine Wave	Control
Power/Subsystem and Crew/Operation Module	4 active (transmission assembly)	2 inactive	2
Crew/Operation Module and GPL	4 active (transmission assembly)	2 inactive	2
Power/Subsystem Module and RAM's	2 active 2 inactive	2 active	2
Power/Subsystem Module and Logistics Module	2 active 2 inactive	2 active	2
Crew/Operations Module and RAM's	2 active 2 inactive	2 active	2
Crew/Operations Module and Logistics Module	2 active 2 inactive	2 active	2
GPL/RAM's	2 active 2 inactive	2 active	2

Note: Inactive interfaces are connectors only without wiring to the power sources. These "dummy" connections are necessary to assure the identical docking interface for all docking ports shown in Figure 4.5-19.

RAM Interface

The EPS interfaces with the RAM's are described by Table 4.5-23. The same power capability is provided to each RAM docking interface within the full total allocation shown in Table 4.5-23. It is noted that these values are resource allocations. With respect to the 115 vdc interface, a much higher value is technically possible from the docking interface, if the RAM wiring is designed to accommodate it. However, the use of this resource at higher rates than shown in Table 4.5-23 would (1) deprive other subsystems of their allocated share of power or (2) require an increase of the EPS power capacity. The resource allocations provided in Table 4.5-23 were derived from extensive studies of the experiment packages necessary to provide for the experiment blue book. Therefore, the individual RAM allocations should be regarded as advisory or preferred values, while the total allocations for RAM's and integral experiments are limitations on the total experimental program.

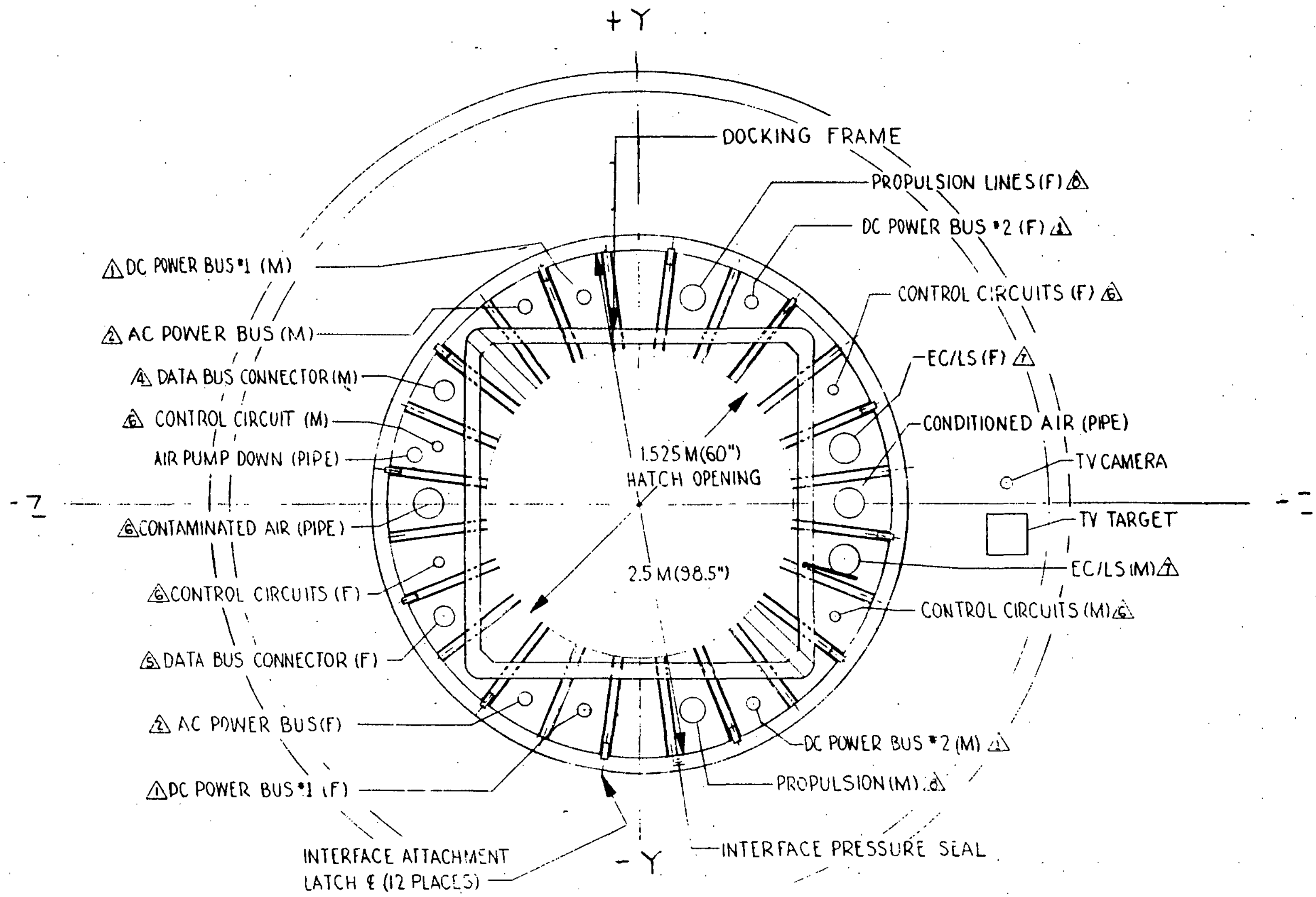


Figure 4.5-19 Docking Interface

Table 4.5-23

ISS/RAM/INTEGRAL EXPERIMENT INTERFACE POWER ALLOCATIONS—KW

	ISS		
	Maximum 24-Hr Average	1-Hr Average	5 Minute Peak
Each RAM:			
115 vdc	2.4**	3.6	--
115/200 vac, 400 Hz	0.5	0.75	--
Total RAM and Integral Experiments:			
115/200 vac 400 Hz	1.0	2.4	2.8
Total ac plus dc power	4.8	7.2	8.4
	GSS		
Each RAM:			
115 vdc	4.0	6.0	--
115/200 vac, 400 Hz	0.5	0.75	--
Total RAM and Integral Experiments:			
115/200 vac, 400 Hz*	2.5	6.0	7.1
Total ac plus dc power	12.1	18.1	21.2

*Includes sine wave and quasi-square wave 400 Hz and sine wave 60 Hz.

**Can be increased to the maximum allocated for experiments (4.8 kW).

Note: These allocations include power for (1) experiments, (2) experiment subsystems, and (3) experiment support and integration (displays/controls), but do not include experiment-related power for GPL.

Logistics Module Interfaces

The EPS interfaces with the docked Logistics Modules are supplied through the common docking-port interfaces. The 24-hour average power allocation for two Logistics Modules is 662 watts of load at 115 vdc (+3 vdc) plus 19 watts (3 percent) for distribution losses from the load bus to the load (641 watts, total). Approximately 33 watts average are required by the subsystems equipments to accommodate docked Free-Flyer Experiment Modules on the GSS.

Shuttle Interface

The Power/Subsystems Module is launched with four batteries installed and the solar array in a stowed position. The module is rotated from the Shuttle bay and the solar arrays are deployed. Checkout of the Power/Subsystems Module is performed while the module remains attached to the Shuttle.

During this initial period, from launch to solar array deployment, the Power/Subsystems Module supplies its own power from on-board storage batteries. Shuttle power will not be required unless an abnormal condition occurs requiring extended operation without deployed solar arrays. To provide for this contingency, a power transfer connection for 115 vdc is provided at the Shuttle/Module interface. Shuttle power available for this contingency mode of operation is 500 watt average, 800 watt peak, and 20 kw-hr. total energy (see References 4.5-1 and 4.5-2).

The Crew/Operations and GPL Modules are launched without batteries and Shuttle power is required until the modules are connected to the solar array/battery power system in the Power/Subsystems Module. Power requirements are as specified in Table 4.5-11.

Power transfer is accomplished by connection to the 115 vdc transmission assembly in each Station Module. The heat dissipation of the electrical loads will be collected and transferred by a coolant loop interface to the EC/LS thermal control assembly.

References for Space Shuttle/Modular Space Station power interface specifications are as follows:

Reference 4.5-1: Payload Design Requirements for Shuttle/Payload Interface, August 9, 1971.

4.2.2.1.1 Orbiter Power

The Orbiter currently allocates limited electrical power to the payload. This power is utilized primarily for Station monitoring of payload critical functions, however, excess power is available to the payload for payload operations. A minimum of 20 kw hours cumulative power is allocated at the rate of 500 watts average and 800 watts peak values by the Orbiter to the payload. Additional electrical power capability shall be payload supplied.

Reference 4.5-2: Attachment to NASA/HQ. Letter, September 7, 1971, "Shuttle/Payload Interface Document—Level I."

Section II. Power—A nominal 20 kw-hr of electrical energy shall be provided by the Orbiter vehicle to the payload via standard connectors. Power supplied will not normally be less than 500 watts average and 800 watts peak.

4.5.3.3 Operations

The operational periods of the Modular Space Station which affect the EPS design and functioning are (1) prelaunch ground operations, (2) launch/ascent operations, (3) orbital storage, unmanned, and (4) on-orbit manned operations.

Prelaunch Ground Operations

The ground support equipment (GSE) provides 115 vdc ground power to the EPS transmission lines through the common docking interface to each Station Module. It is also capable of providing the same power types, voltages, wave-forms, qualities, and quantities as are provided by the EPS in orbit at corresponding EPS test points. The GSE thus accurately simulates the solar array source power profiles and conditioning assembly output power profiles. The GSE also has a capability to provide simulated loads having the same steady-state and simulated transient, overload, and/or faulted characteristics as the actual electrical loads when it is impractical to use actual module loads for testing purposes (e.g. for overloading and fault tests, or for calibration of circuit protective devices).

Launch/Ascent Operations

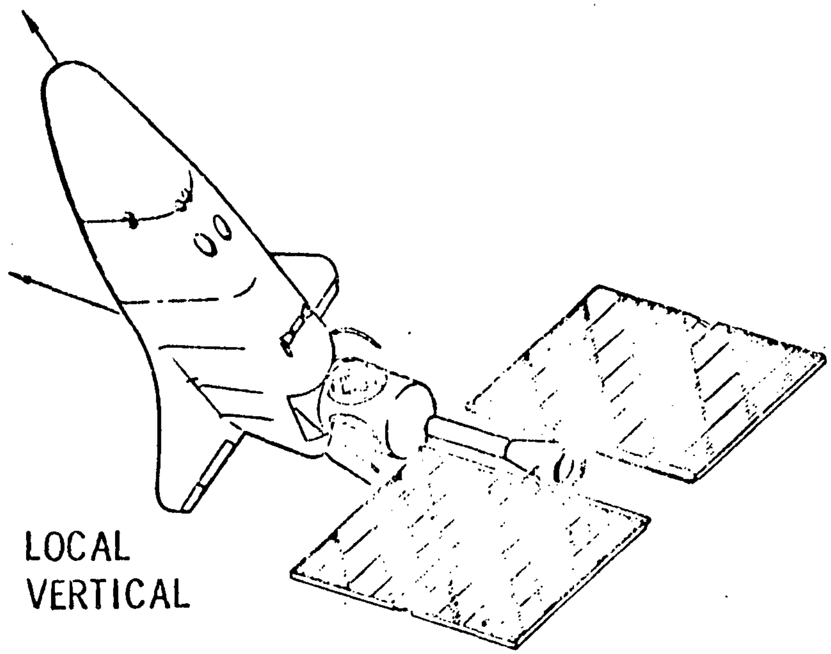
To minimize the initial launch weight, the Power/Subsystems Module is launched with only one set of four batteries. These four batteries provide sufficient power for the Power/Subsystems Module and initial activation period for the limited time prior to solar array deployment. These batteries, with solar array support, are sufficient also to support all three ISS Modules during the orbital storage after solar array power is available and prior to manning. The power requirements are discussed in Subsection 4.5.2 and are shown in Table 4.5-11 and Table 4.5-12.

The Crew/Operations Module and the GPL are launched without batteries, which are later supplied by Logistics Modules prior to and/or concurrent with the ISS manning launches. Therefore, the Space Shuttle provides the Crew/Operations and GPL Modules energy and power levels indicated in Table 4.5-12.

The average and maximum load demands shown in Table 4.5-11 must be increased by 10 percent to account for the conditioning and distribution losses. The total energy demand including 10 percent for conditioning and distribution is also tabulated in Table 4.5-11. The resulting power and energy demands of 547 watts peak and 2,549 watt-hrs for the Crew/Operations Module and 514 watts peak and 4,448 watt-hrs for the GPL represent interface requirements for the Space Shuttle. These requirements are within the Space Shuttle capabilities.

Solar Array Deployment Operations

The deployment sequence for the solar array with Space Shuttle Orbiter manipulation to verify solar orientation is shown in Figure 4.5-20. The sequence of deployment is initiated by verification of internal Power/Subsystems Module battery power. The mechanical operations are described in detail in Subsection 4.2.3. The solar array deployment sequence was also described in Subsection 4.5.3.1. When fully deployed, the Space Shuttle



- VERIFY MODULE ON BATTERY POWER
- TIE-DOWN RELEASE
- EXTEND TRUSS FRAMES
- EXTEND ASTROMAST AND PANELS
- ORIENT MODULE X-AXIS TO SUN
- VERIFY ARRAY VOLTAGE OUTPUT
- COMMAND-ARRAY POWER SWITCHES TO MODULE POWER BUSES
- COMMAND SWITCH BATTERIES TO RECHARGE MODE
- ENABLE SUN SENSOR SYSTEM
- VERIFY ARRAY GIMBAL ACTION WITH ORBITER PITCH AND YAW MANEUVERS
- SHUTTLE MANEUVERS MODULE X-AXIS ALONG ORBIT VECTOR

Figure 4.5-20 Solar Array Activation Sequence

Orbiter X-axis is oriented toward the sun, thus placing the panel face toward the sun. After verifying solar array voltage at the source buses, the source buses and the main distributor buses are switched to the transmission lines and the battery chargers are commanded to the charging mode. The sun sensors on the array are next activated. The Space Shuttle Orbiter is then commanded to perform pitch and yaw maneuvers. The sun sensors are verified to provide the appropriate corrective drive commands by motion of the array to maintain sun tracking. The Power/Subsystems Module is then oriented along the velocity vector for horizontal flight attitude. The arrays are commanded to a trailing, minimum-drag position.

Short-Term Orbital Storage Operations

After the solar array is deployed, but before ISS manning, the orbit is allowed to slowly decay to conserve propellants. To minimize drag during this lower power period, the solar array is fully deployed, but its plane is coincident with the Power/Subsystems Module longitudinal axis and the horizontal velocity vector (in a "trailing" mode), thereby minimizing the solar array drag. The power capability in this position is shown in Figure 4. 5-21. The demand on the illuminated solar array is only 8. 6 kw maximum for 38. 3 minutes to recharge the batteries and to support a 3 kw electrical load. Table 4. 5-12 shows a total requirement of 2, 437 watts for all these modules. Up to 22 kw maximum are available, sufficient to support over 6 kw of constant load with 4 batteries; therefore the "trailing" mode is entirely practical for support of the unmanned power requirements. It is also sufficient to supply more than the emergency power supply requirements during manned operations, in the event of damage or failure modes which may immobilize the array orientation assembly.

Manned Orbital Operations

During manned on-orbit operations, the EPS is principally in an automatic operational mode. The source buses and transmission lines are in parallel; the main distributor buses in each main distributor center are in parallel, and the inverter buses in each main distributor center are in parallel. This mode improves the voltage regulation and minimizes losses. During normal operations, solar array orientation is under control of the sun sensors

HORIZONTAL ORIENTATION - TURRET AXIS ONLY

SOLAR ARRAY POWER - 42.4 MIN
BATTERY POWER - 49.6 MIN

DARK SIDE - BATTERY DISCHARGE - 2.5 KWH
3.0 KW FOR 49.6 MIN

SUN SIDE - BATTERY RECHARGE - 3.6 KWH
5.6 KW FOR 38.3 MIN (AVG)

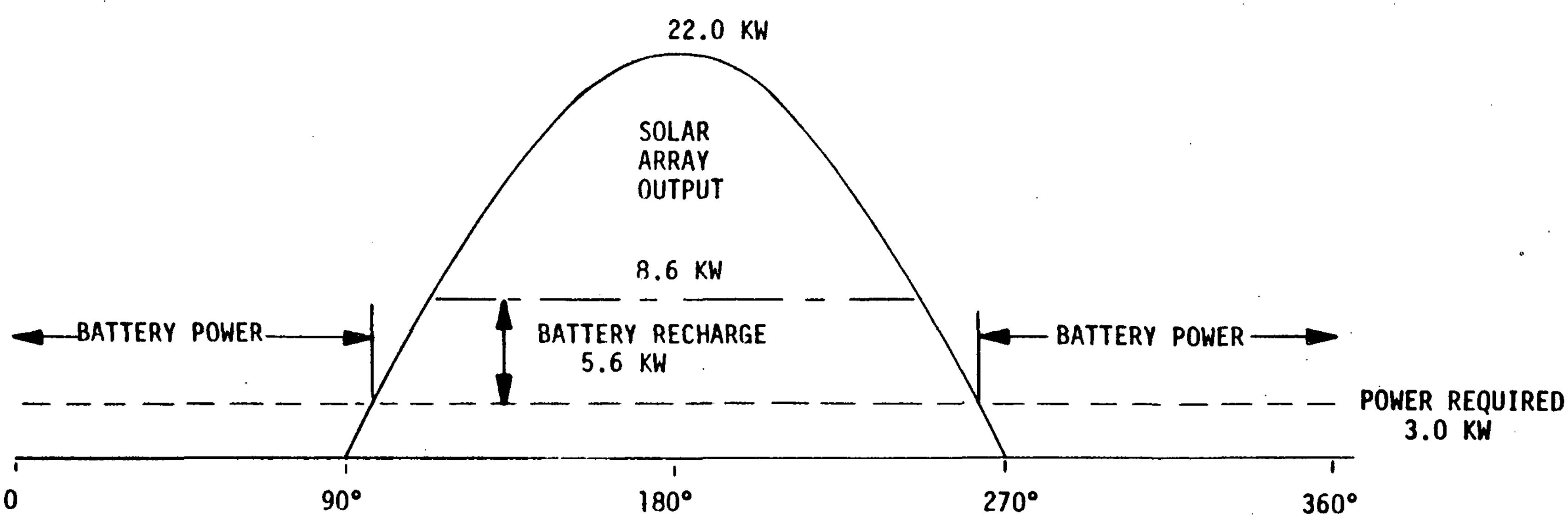


Figure 4.5-21 Premanning Power - Horizontal Orientation - Turret Axis Only

mounted on the arrays. These provide drive signals to the gimbal servo motors within a pointing deadband of ±8 degrees. The DMS receives and stores the gimbal-angle data in the computer memory, and can compare it with the sun-axis directional data in the computer memory to be used in the event that an unexpected solar array power reduction occurs during the normal illumination period. These data also are used in the DMS computations which are necessary to orient the array at the start of each eclipse to the minimum-drag (trailing or feathered) position for minimum propellant consumption, and to reorient the array just prior to the end of each eclipse period in preparation for solar acquisition. The DMS computer provides the required new gimbal angles to the orientation drive electronics and verifies solar acquisition by interrogating the array-mounted sun sensors. These operations are performed in an automatic mode through the data bus and the orientation RDAU. If one line or bus becomes short-circuited, the zonal differential protection system will trip circuit-breakers to remove it from service, and the cable redundancy will continue service. Critical loads will receive power from alternate routes, if necessary. Essential buses will switch over to alternate supplies, if necessary.

Indicators of the problem and the action taken by the central computer will be displayed for the crew. Batteries operating in parallel to the main distributor buses will share-load automatically. As the solar array voltage rises in transition from Earth eclipse to solar illumination, the SPSR automatically controls the voltage to the design range and takes over the power supply function from the batteries, switching the half-panel sections into or out of service, or into a linear control mode as necessary.

If the operator desires to remove a transmission line from service with a minimum system perturbation, he will take the following actions: Assume that the system is in an isolated operational mode (transmission lines or main distributor buses are isolated). The operator will close switches to the desired transmission line or bus, momentarily paralleling the two, then he will switch off the circuit-breaker from the undesired source. When all loads are on the desired line or bus, he can de-energize the undesired source bus , transmission line, main distributor bus, or conditioning subassembly.

It is possible to instruct the computer by inserting the proper commands, then to command "Execute" to accomplish the desired operation.

Emergency, critical, or essential load power functions are normally automatic, with manual command and/or override capability. After a fault indication on a given EPS element, only manual operation is permitted to restore service, thereby increasing personnel safety and minimizing electrical or mechanical shock to the EPS.

Emergency/Contingency Conditions

The requirements for emergency/contingency power are shown in Figure 4.5-6, Section 4.5.2. The capability to supply these loads is shown in Figure 4.5-22. The crew survival/rescue power is available from any two of the four emergency pallets (AgZn batteries) or from the 24 batteries, if fully charged. If any one of twelve panels and four of 24 batteries are available, the 96-hour emergency power level can be supported indefinitely.

The 1-hour emergency period for damage assessment and power restoration can be accommodated by the energy from any one of the 24 Station batteries, even at 65 percent minimum state of charge. However, a DC/DC converter would be required to obtain 115 VDC from one 28-VDC battery. Therefore, the EPS design requires any four of 24 batteries, and this damage assessment period is extended accordingly to four hours at 2-3 kw.

The 24-hour emergency period for repair/maintenance can be accommodated by (a) all 24 batteries if at 100 percent state of charge, (b) any one of 12 solar panels with all 24 Station batteries, or (c) any two of 12 solar panels with any four of 24 Station batteries.

The power requirements for a continuous contingency period requiring different levels of activity (activity either in one module or in all modules) and checkout capability (from minimum crew support to full checkout) are indicated also on Figure 4.5-6 of Section 4.5.2. Comparison with Figure 4.5-22 shows that (a) any two of 12 panels with any four of 24 station batteries will support the crew in an emergency in one module (minimum activity) without

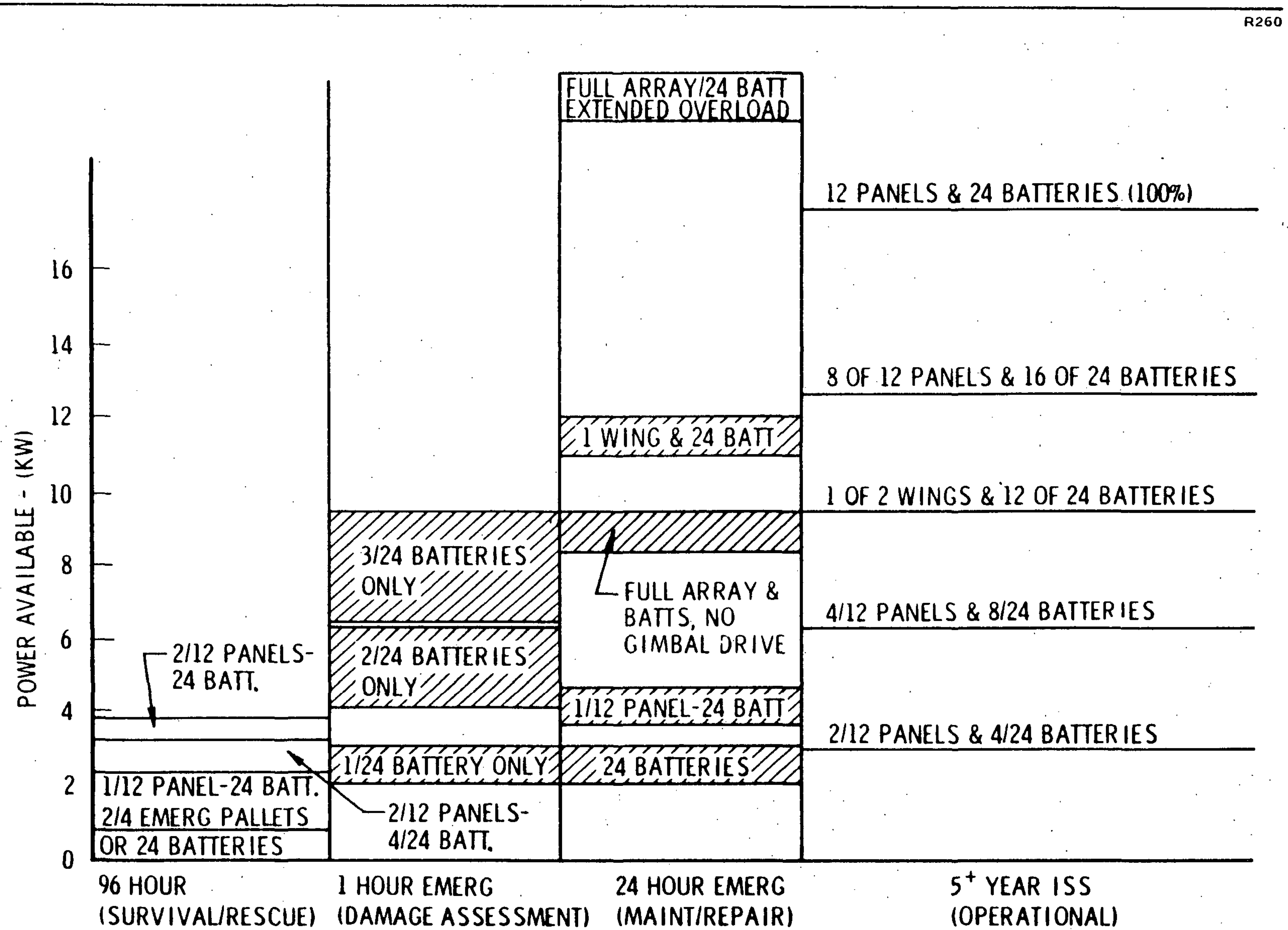


Figure 4.5-22 Emergency/Contingency Power Capability

checkout capability; (b) any four of 12 panels with any eight of 24 batteries (1/3 of the EPS) will extend crew activities to all three modules and will also provide full checkout capability; (c) any six of 12 solar panels with any sixteen of 24 Station batteries (2/3 of the EPS) will support all Station loads without any experiments.

4.5.3.4 Growth Space Station (GSS) Considerations including Nuclear Power

The study of alternatives for growth to GSS considered three principal alternatives: (1) augmentation by adding solar arrays to the ISS solar arrays, (2) replacement of initial ISS solar arrays, and (3) augmentation of the initial solar arrays by a modular nuclear reactor and power boom technique. The cost analyses reported in Section 4.5.4.7 indicates that replacement of a solar array at 5 years life discards 70 percent of the original capability which also represents 70 percent of the recurring cost. As the solar array is the primary cost element, an augmentation technique, in which additional area is added to supplement the degraded array, is preferable. Further study showed that the load requirement for GSS at 10 years is within the total power from one initial solar array, degraded by 40 percent in 10 years, plus a replica of the original array, degraded by 30 percent in 5 years. This replication technique can be repeated indefinitely to sustain the required GSS power level, if desired.

It is not credible that a nuclear reactor can be added to the program more cost-effectively than can solar array replication. However, if a major power growth should occur for GSS, a larger solar array can be added to augment the original array. Because the augmentation power module does not fly independently, it does not require the other autonomous subsystems, and consequently launch of a larger array area is possible. The largest presently in development is 10,000 ft^2. The use of such an array to augment the 5,300 ft^2 ISS array would result in 45.8-kW average at the load buses at the end of the tenth year of the mission—which is well in excess of the presently expected value of 30.8 kW.

The relatively high basic cost of nuclear reactor system development, as determined in the earlier 33-ft Space Station Study, in addition to the ISS solar array system development, must be added to the cost of a new integration program to fit this substantially different source into the EPS transmission, control, and power management complex of the solar array/battery EPS. The conclusion is apparent that a nuclear reactor EPS is not a cost-effective approach for moderate power growth to GSS based on the Guidelines and Constraints used for the Modular Space Station Study. Some different basis for a nuclear reactor EPS would be required, such as cost sharing with another program requiring up to 100 kW, in addition to a major power growth requirement for the Modular Space Station.

The technical feasibility of modularizing a nuclear reactor power source for Space Shuttle launch was examined in preliminary fashion and is reported in the following paragraphs.

Nuclear Power Options

The Modular Space Station Study Guidelines and Constraints directs the use of solar arrays to provide electrical power during the ISS program phase. The option to select an alternate power system for the Growth Space Station (GSS) was maintained, but within the primary NASA objective of a low-cost 10-year program, introduction of an alternate power system at program mid-point is not cost effective. Consequently, the baseline system design for the 10-year Space Station Program has defined a solar array-battery electrical power subsystem which is fully responsive to programmatic constraints and requirements. The applicability of alternate power systems is recognized and their viability will be enhanced for a number of potential programmatic developments, as follows:

A. Extension of Space Station operations beyond 10 years.
B. A major increase in power load requirements (e. g. addition of high power experiment payloads)
C. Shared development (cost) with other programs
D. Changes in Space Station mission and logistics baselines.

Extension of station operations beyond 10 years offers an increased amortization period for write off of a new EPS development cost. Such a mission extension in conjunction with increased power requirements could lead to re-evaluation of the level I study guidelines which established the baseline power system. Increased power requirements could occur as experiment activities evolve into operational applications of greater complexity. Alternate mission altitudes (decreases), increased resupply costs, identification of operational maneuvering constraints, and experiment viewing interferences can all lead to consideration of alternate EPS concepts. Future missions (planetary, lunar, etc.) which select nuclear power systems and which would benefit from early power system demonstration also can justify power system change at the start of GSS.

Since the programmatic variations discussed above represent realistic possibilities, the applicability of reactor and isotope power systems was reviewed. MDAC Phase B design definitions (References 4.5-3 and 4.5-4) for nuclear power systems were used to establish preliminary systems design and to assess Modular Space Station impact. In summary, the previous Space Station—EPS designs can be scaled for integration with the Modular Space Station and no insurmountable problems have been identified for any of the nuclear power system designs. Additional analyses will be required to define specific design and operating points which reflect power system optimization.

The Modular Space Station constraints and requirements that have significant effects on the power system are as follows:

A. Design-to-Shuttle payload capability of 14 ft diameter by 58 ft in length and 20,000 lb.
B. Orbiter's allowable CG travel for launch and recovery.
C. Configuration geometry—crew residence pattern.
D. Unique requirements/operating mode for orbiter recovery of used reactor power modules.

Preliminary reactor EPS designs and integration impacts for the Modular Station are covered below.

EPS Design and Design Modifications

Summary Electrical Power System (EPS) weights taken from Reference 4.5-4, for a 29 kWe power requirement are as follows.

	Reactor T/E	Reactor Brayton
Power System Less Shield (lb)	11,460	13,836
Shield (lb)	13,060	8,930
Disposal System (lb)	3,250	3,250
Total	27,770	26,016

As designed, both replaceable power modules are overweight and require major weight savings to permit launch within the 20,000-lb payload constraint. Elimination, or separate launch, of the reactor disposal system saves 3,250 lb. and elimination is consistent with the results of the Reactor Preliminary Safety Analysis Report (Reference 4.5-5). The most obvious weight savings is to re-evaluate the station configuration and reactor shield requirements. Increasing the reactor-thermoelectric's separations distance to 250 ft and decreasing the dose plane diameter from 130 ft to 110 ft will permit shield weight savings of about 4,560 lb. Comparable changes in dose plane diameter for the Brayton power module shielding will save approximately 1,000 lb and this plus off-loading of one of the three Brayton engines and controls will effect the necessary weight savings of 7,800 and 6,200 lb. Figure 4.5-23 illustrates the Station/Reactor configuration and identifies the dose plane diameter and separations distance relationships. Figures 4.5-24 and 4.5-25 present isometric views of the power modules and indicate the discrete changes made to attain the new module weights listed in Table 4.5-24.

The change in dose plane diameter is responsive to the change in basic Station diameter from 33 ft to 14 ft. Shielding conservatism remains, since the Modular Space Station inherently has an increased crew-reactor separations distance relative to the more compact 33-ft configuration. The shield weight reductions for the thermoelectric system are large and it may be necessary to provide incremental shielding launched separately. It should be noted that the separation boom is launched separately and in advance of the reactor launch. The boom diameters are reduced relative to the design shown in Reference 4.5-4 and this allows further system weight savings. A weight margin exists then for launch within the boom module of either the third Brayton engine or the shield augmentation for the thermoelectric system.

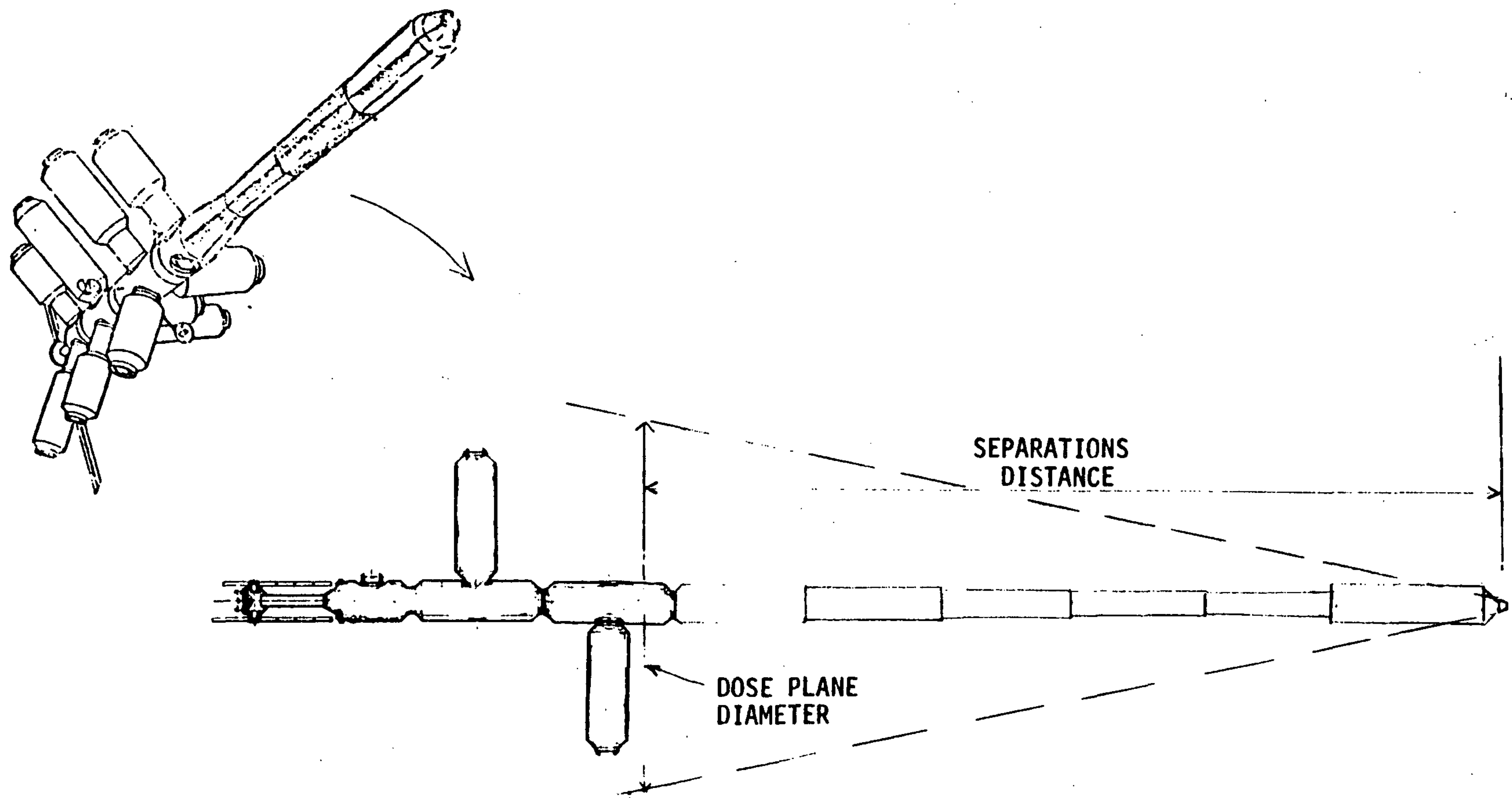
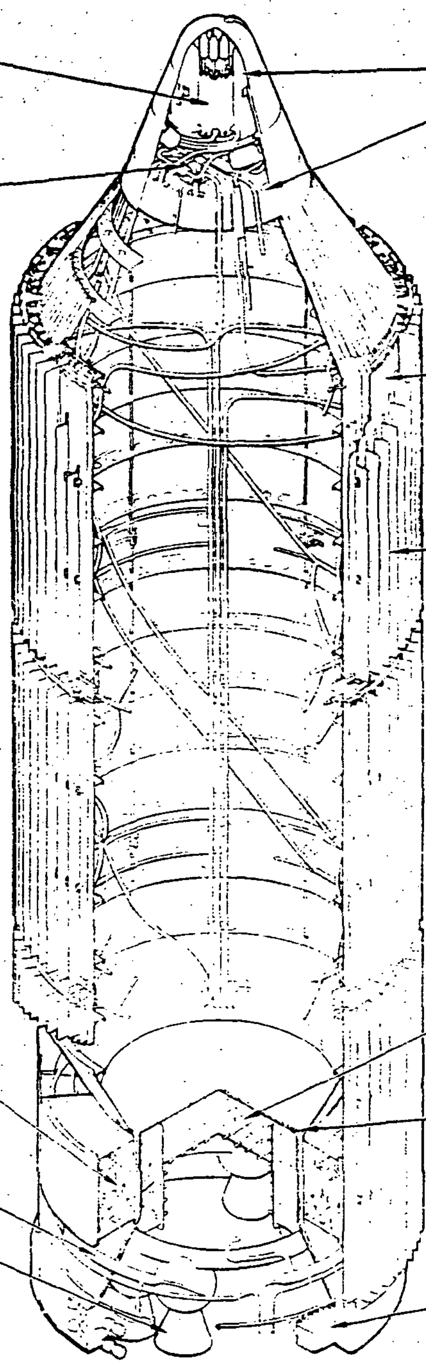


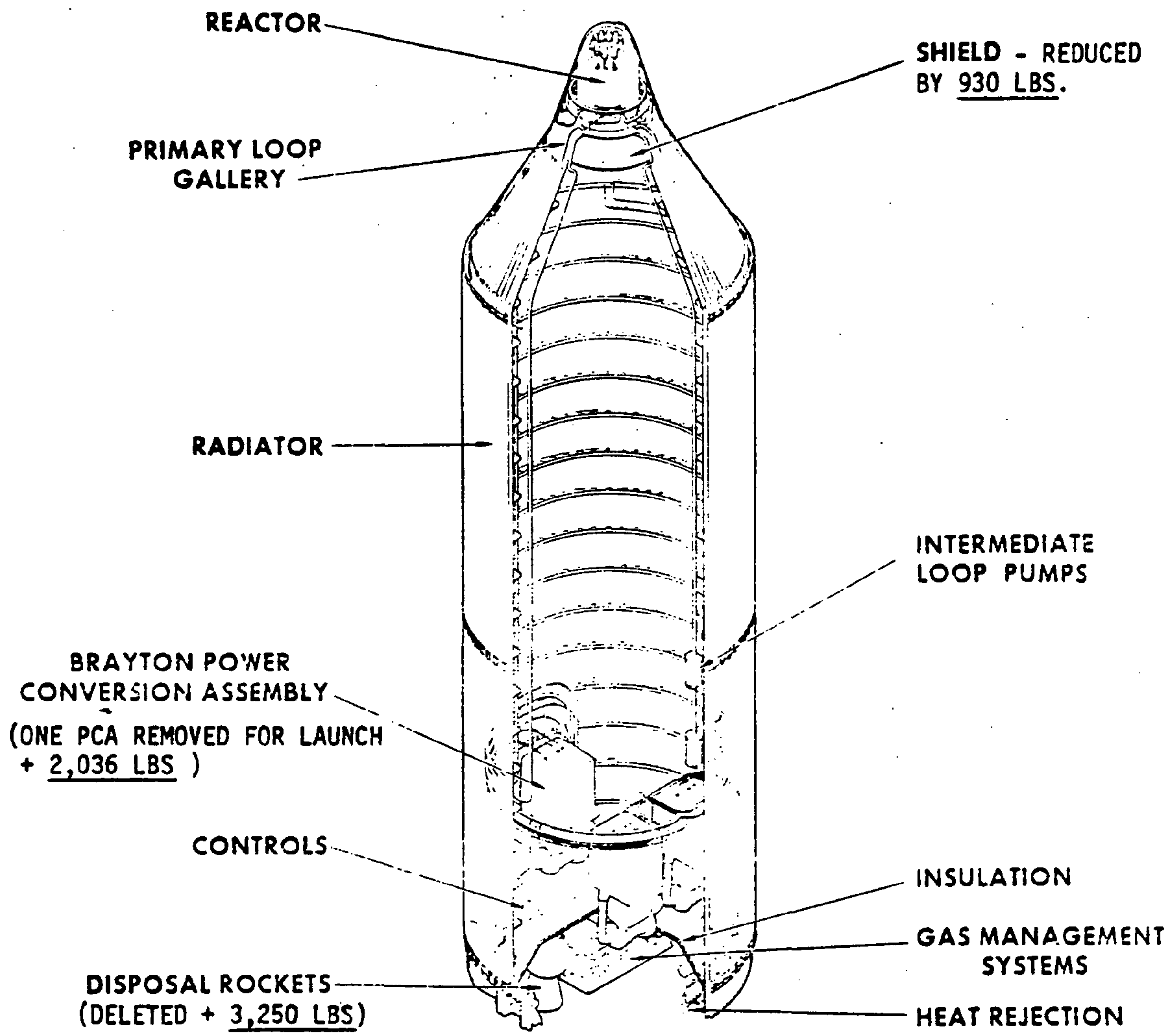
Figure 4.5-23 Radiation Shielding



TOTAL WEIGHT REDUCTION

° SHIELD	4,563 LB.
° DISPOSAL SYSTEM	3,250 LB.
	7,813 LB.

Figure 4.5-24 Reactor Thermoelectric System



TOTAL WEIGHT REDUCTION

° SHIELD	930 LB
° POWER CONV. ASSEMBLY AND CONTROLS	2,036 LB
° DISPOSAL SYSTEM	3,250 LB
	6,216 LB

Figure 4.5-25 Reactor Brayton System

Table 4.5-24

MODULE CHARACTERISTICS

	Reactor T/E		Reactor Brayton	
	Boom Module	Power Module	Boom Module	Power Module
Power (kwe)	--	31.8	--	31.8
Power System less Shield (lb)	--	11,460	--	11,800*
Shield (lb)		8,500**		8,000
Total (lb)		19,960		19,800
Separations distance (ft)		250		200
Dose plane diameter (ft)		110		110
Boom Structure (lb)	6,090		4,420	
Transmission/Conditioning/ Distribution (lb)	5,359		6,281	
Auxiliary Radiator (lb)	2,900		--	
Batteries (lb)	3,000		3,000	
Docking/Lights/Restraints (lb)	400		400	
Total (lb)	17,749		14,101	

*One Brayton PCS and Controls off loaded for launch.
**Shield augmentation may be required.

The reduced capability orbiter also changes the physical size of the power modules from 15 ft to 14 ft in diameter and from 60 ft to 58 ft in length. Additionally, the Modular Space Station (GSS) power requirements increase from 29 to 31.8 kWe. These system changes will have impact on reactor system design and operation, but the effects are secondary relative to the required weight reductions described above. The effects of power increase, radiator area reductions, and revised transmission/conditioning/distribution (T/C/D) designs will perturb system design point optimization; however, the baseline systems (Reference 4.5-4) were designed to accommodate program change of this type. System reoptimization should be done at a future date, but the power growth capability of the reactor systems should be retained. Preliminary consideration was given to the T/C/D impact of integrating the Brayton EPS (400 Hz, AC) into the solar array-DC power system design, since this appeared as a potentially significant impact. However, since the baseline Brayton system was designed to accommodate an 85 percent DC load, the impact will be nominal and further will be limited by the more efficient 115-VDC distribution system used in the Modular Space Station.

The effect of off-loading shielding, disposal propulsion, and Brayton engine to meet the 20,000 lb Shuttle capability results in new CG locations for each power module. The combined effect for the reactor thermoelectric system will be to move the center of gravity closer to the module's mid-point, enabling an increase in module length to about 58 ft. The net effect of reactor Brayton weight changes will shift the CG forward of center, but on-orbit installation of the third engine will limit the shift and an acceptable CG location is expected. Changes in orbiter design continue and currently it is not possible to evaluate Power/Subsystems Module launch and recovery feasibility. The expectation of acceptable Power/Subsystems Module CG locations is based on previous orbiter design constraints.

The effect of reactor Power/Subsystems Module recovery on the Shuttle has not been addressed and future study will be required to define the recovery concept and the recovery-unique requirements. Key questions to be examined include: 1) required reactor shutdown time before Shuttle rendezvous and return, 2) the optimum storage mode for the shutdown power module (either docked or free-flying), 3) orbiter impacts and risks due to radiation and NaK leak/contamination potential, and 4) the operational procedures and equipments necessary for recovery and disposition.

Space Station Impact

Reactor power system integration with the Space Station has been shown (Reference 4.5-4) to have limited impact on Station and subsystem designs. Increased attitude control requirements will exist for extended inertial orientations although this requirement does not exist for baseline Station orientations. Either increased CMG capability and/or high-thrust RCS capability would permit extended inertial orientations. RCS modules could be added at the separations boom—reactor power module interface to minimize propellant requirements. The remaining subsystem impacts are minimal and retraction of the solar arrays will enhance experimental viewing and operational freedom.

Preliminary system analysis indicates that reactor replacement could introduce operating constraints due to radiation if the shutdown reactor remains stowed on the Station during a cool down period for about 100—200 days. Radiation from fission product decay could complicate experiment and EVA operations. For this reason, there is incentive to select reactor Power/Subsystems Module separation from the Station for this decay period. Inherent in this option is a stabilization requirement for the reactor power module to enable subsequent orbiter docking. Power/Subsystems Module free-flight during the cool-down also imposes requirements for navigation/monitoring equipments. A disposal kit could be installed on the Power/Subsystems Module in advance of separation to avoid initial launch weight penalties. The optimum operational mode and recovery-unique requirements should be selected in future studies.

Minor constraints also are identified for the normal experiment operations with a reactor power system. Radiation levels during operation will dictate an incremental weight penalty of 2,000-3,000 lb for film-vault shielding. The sensors for experiment module A5B potentially are sensitive to the reactor radiation levels and study will be required to evaluate means of minimizing radiation interference.

4.5.4 Design Analyses and Trade Studies

4.5.4.1 Solar Array Design Analysis and Trade Studies

A solar array/battery electrical power subsystem has been designed to satisfy the Modular Space Station requirements. A summary of the Solar Array Subsystem requirements and characteristics are shown in Table 4.5-25, which includes the orbital characteristics assumed for the solar array design.

The solar arrays supply all of the electrical power to satisfy the electrical design load requirements discussed in Section 4.5.2 and to recharge the batteries during the illuminated portion of the orbit. The electrical loads are supplied by nickel-cadmium batteries during the eclipsed portion of each orbit. The solar arrays are made up of two wings, one on each side of the Power/Subsystem Module tunnel. Each wing is made up of multiple flexible foldout solar panels.

Modularity and Power Growth Accommodations

The array is sized by the six-man ISS power requirement of 16.7 kw average at the load buses at the end of 5 years. This results in an array area of 5,300 sq ft and an initial power capability of 22.7 kw. After 5 years of operation, growth to the 12-man GSS takes place. A second array identical to the initial array is added to increase the initial capability to 39.5 kw, which degrades to 31.1 kw at the end of 10 years (30.8 kW are required by the GSS). These arrays will satisfy the electrical load and battery charging requirements and will be described in this section. Unless otherwise indicated, power values stated will be average 24-hour total power at the load buses. The solar array power delivered will be greater by a Design Factor of 2.18.

A number of approaches were investigated to provide array power to satisfy the above requirements—these included the use of one, two, and three sets of arrays, and augmentation or replacement methods of growth.

The use of a single array set at the beginning of the mission to satisfy the total 10-year mission requirements would result in a large initial launch

Table 4.5-25

SOLAR ARRAY CHARACTERISTICS

Load Requirements	
ISS	16.7 kw
GSS	30.8 kw
Solar Array	
Area/array	5,300 ft^2
Power available	
Initial	22.7 kw
After 5 yr	16.7 kw
With 2nd array	39.5 kw
After 10 yr	31.1 kw
Orientation	2 axis
Orientation accuracy	±8 degrees
Solar Panels	
Type	Flexible foldout
Deployment	Astromast
Number/array	12
Voltage	121 vdc at deployment mast root
Solar Cells	
Type	Silicon N on P
Size	2 x 4 cm
Thickness	0.008 in.
Efficiency	11 percent bare at amo and 28°C
Cover Glass	0.006 in., 0211 microsheet
Base Resistivity	2 ohm-cm (1-3 ohm-cm range)

weight penalty, a large area to allow for degradation of array that is not required, and a wide variation in the transmission and conditioning equipment requirements. The use of three arrays would result in special design accommodations that penalize the system design. These include interference and shadowing. The use of multiple arrays of several different sizes is not attractive because of the increased development costs as compared to developing one array size only.

The most desirable approach is the use of two identical solar arrays with the first array launched on the first power module and the second array provided on a second Power/Subsystem Module during growth to GSS. Two arrays of 5,300 square feet satisfy the power requirements as shown in Figure 4.5-4 of Section 4.5.1.

Since the initial array (at the start of ISS and the start of GSS) is oversized to compensate for anticipated degradation, the array is capable of producing more power initially than is required. This excess capacity decreases with time as degradation occurs. However, the Baseline transmission, conditioning, and distribution equipment are designed to meet only the required power level and not to accommodate this excess capacity. The excess array capacity represents growth capability that can be utilized by increasing TCD capacity correspondingly (with attendant increases in cost and weight).

Comparison of Flexible and Rigid Array Concepts

A comparison was made of weights for rigid and flexible solar arrays from data supplied by several different array manufacturers, including data supplied by Fairchild Industries, which was extrapolated to 4,500 sq ft for both rigid and rollout flexible designs, and by TRW for rigid and flexible arrays. The TRW rigid arrays were made up from Skylab wing sections and the flexible arrays were foldout arrays with bistem deployment. Data was supplied by LMSC for a flexible foldout array with Astromast deployment. The comparison, shown on Figure 4.5-26, indicates a significant difference in weights between rigid and flexible arrays. Because each array was sized to a slightly different area, the lb/ft^2 and watts/lb factors were calculated for more direct

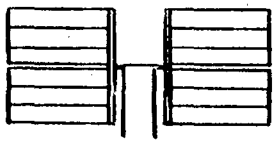
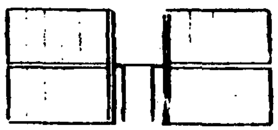
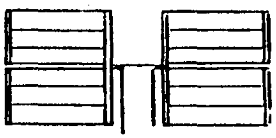
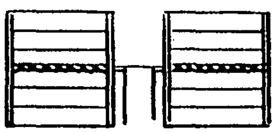
	AREA (FT^2)	RIGID WEIGHT (LB)	RIGID LB/FT^2	RIGID W/LB	FLEXIBLE WEIGHT (LB)	FLEXIBLE LB/FT^2	FLEXIBLE W/LB
 FI (ARTICULATED BEAM)	4500	3,740	0.83	12.0	2,510	0.56	18.0
 TRW (SKYLAB WING SECTIONS)	4850	7,500	1.55	6.5			
 TRW (BISTEM/FOLDOUT)	4700				2200	0.47	21.4
 LMSC (ASTROMAST/FOLDOUT)	4930				2,680	0.54	18.4

Figure 4.5-26 Rigid Versus Flexible Arrays

comparison. As shown, rigid arrays require 0.83 to 1.55 lb/ft^2 compared to 0.47 to 0.56 lb/ft^2 for flexible arrays. In addition, rigid arrays give only 6.5-12.0 W/lb compared to 18.0-21.4 W/lb for flexible arrays.

The array stowage volume, although not presently a critical item for the Modular Space Station design using flexible arrays, could become critical if rigid arrays were selected. Previous analyses have shown that rigid arrays require at least three times the stowage volumes of flexible arrays.

Flexible array concepts are generally characterized by a minimum of structure used only to support other structures, or of hinges, latches, and locking mechanisms to provide rigidity to the deployed panel. In place of many discrete series deployment actions required for rigid arrays, having a reliability loss associated with each step, the flexible array concepts use an infinite number of incremental deployment motions in a single continuous process. The result is a single action, and higher inherent reliability. Elimination of supporting panel volume and weight and the associated structures and mechanisms results naturally in a reduction of stored volume and weight and an increase in the power-to-weight ratio. The flexible array stowed package can be more densely packed to provide tolerance to the launch environment with a minimum of extra support and protection.

The flexible array systems are retractable if desired to permit added reliability for docking, orbital maneuvers, or severe environmental conditions such as solar flares or meteorite activity. Retraction is also potentially available as a means of power control, if desirable, thus reducing or eliminating degradation of cells in space while they are not required. "Rotation" (sequential assignment) of the panels for equal service would be desirable, however, in order to verify their readiness for use. Sequential deployment and retraction/redeployment as operational modes will require materials research to assure tolerance after several years in space. Preferential orientation is the preferred mode for docking.

Experience quoted by manufacturers of both types of panels (rigid and flexible) indicates that as the panel area increases, the cost and development time of flexible panel increases less than does a rigid multi-fold panel. This design is considered to be well beyond the cross-over point. Some of the more important reasons why time and cost favor a flexible approach to a large array, especially where weight and volume are critical, are:

A. The array is more readily protected from dynamic loading due to the launch environment. Thus, the entire design is simplified.

B. Low weight can be achieved without the use of beryllium or other materials which require extensive development.

C. The concept features basic simplicity in the deployment system and eliminates complicated mechanisms, pyrotechnics, or sequencing provisions. Mechanical simplicity reduces the design, manufacturing, test, and qualification effort.

D. Testing is easier. The package size is smaller for launch mode tests. Deployment in a 1-g field can be readily demonstrated with a simple table/roller or vertical counterweight arrangement, thus avoiding the complicated and expensive air-bearing trolley test fixtures.

E. The ground-support equipment is also simpler for the reasons listed in "D", above.

F. No requirement exists for unusual facilities to develop and produce a large rollout panel.

Growth potential for flexible arrays is accomplished by (1) adding similar panels, (2) increasing panel width, (3) increasing panel length, or (4) any combination of these. A given flexible assembly can generally be increased in area by increasing either length or width with a weight increase of less than one to one.

The flexible rollout solar array is selected over the rigid array because of higher inherent reliability, lower weight, stowage volume, retraction capability, growth potential, and flexibility.

Solar Array Design

The selected solar array design is based upon a LMSC design study for NASA-MSC and designed to MDAC requirements (Reference 4.5-6). Data provided by LMSC included orientation drive and power transfer which will not be used in MDAC Modular Space Station design. Other changes from the LMSC design are listed below. Some of the changes are a result of providing requirements early in the study.

A. 5,300 ft^2 array area rather than 4,500 ft^2,
B. No internal stowage in turret tunnel.
C. Reduced number of panels.
D. 8 mil cells rather than 12 mil.
E. 6 mil coverglasses rather than 12 mil.
F. Zero-g operation rather than artificial-g capability.

The change in item No. 1 is a result of increased power requirements and No. 2 to the complexity and undesirable feature of stowage in the turret tunnel. Because of a higher voltage design and more efficient T/C/D concept, the number of panels was reduced to a total of 12 with 6 on each wing. No advantage is seen for thicker cells and coverglasses and the resultant weight increase. While 12 mil cells are optimum today, the thinner cells are expected to be optimum for the Modular Space Station time period. The design for artificial-g results in conservatism for the LMSC design on the MDAC station. Although it requires increased weight, the capability will be retained to maintain solar array mission flexibility. The nomenclature used for the panel is shown in Figure 4.5-27.

Solar Cell Size Selection

Candidate solar cell sizes for selection include 2 by 2, 2 by 4, and 2 by 6 cm cells. Cost consideration for procurement and fabrication on the panels favor the use of larger size cells while breakage and increased loss per single breakage favor a small size. However, the layout and buildup of cells in series and parallel to form modules and panels proved to be the determining factor in cell size selection.

The gross array area required differed significantly when using different cell sizes. Based upon a panel design similar to the LMSC design resulted in gross array area requirements of 5,520 ft^2 and 6,120 ft^2 for 2 by 4 and 2 by 6 cm cells, respectively. The major reason for the difference is the large number of cells in series to provide adequate voltage after 10 years,

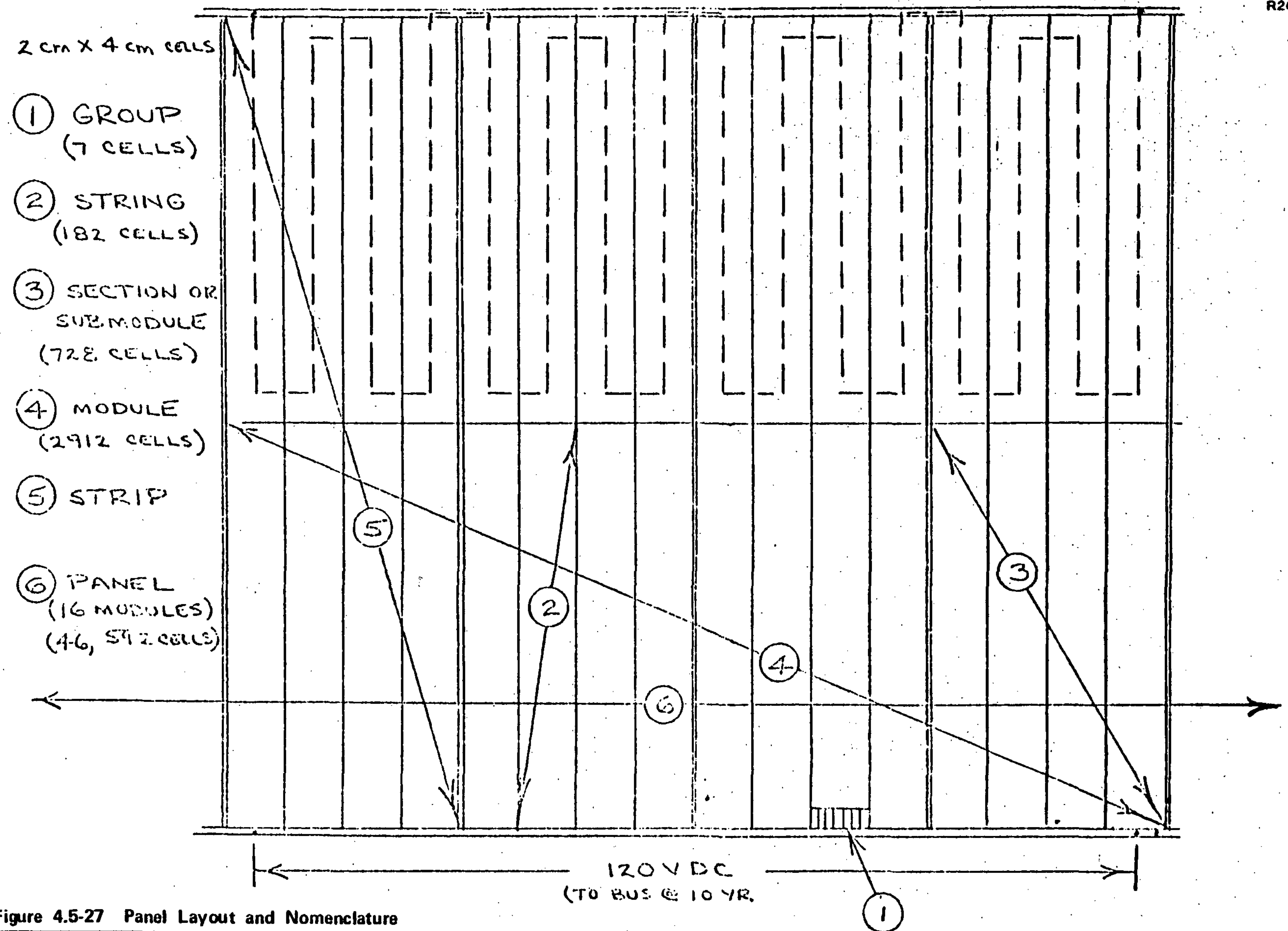


Figure 4.5-27 Panel Layout and Nomenclature

the resultant module sizes and geometry, and the need to provide an even number of modules to satisfy the electrical requirements. Therefore, 2 by 4 cm solar cells are selected for the Modular Space Station arrays.

Solar Array Degradation

The output of the solar arrays at the end of life are used to determine the array sizing at the beginning of life. Many factors must be included in the solar array design and for solar array sizing, and include both time-dependent and time-independent factors. The time independent factors are those for design selection, fabrication and assembly, and mission parameters that cause cyclic output or variations that are recoverable for which the worst case must be considered in the design. The time-dependent factors are those that degrade the solar array with time and are generally not recoverable.

A survey was made of the degradation factors and the results are summarized in Table 4.5-26. They include the Skylab degradation factors for a 300-day mission, the predicted degradation factors for a 10-year mission from LMSC and TRW, and the selected degradation factors for the 10-year Modular Space Station. The selected factors were determined by MDAC analysis and from the inputs from LMSC and TRW. Factors were not supplied from LMSC and TRW for all of the degradations listed. The factors that are shown are in good agreement with each other. The major difference occurs for radiation damage (electrons and protons) between LMSC and MDAC. The difference occurs because of thickness of coverglass used. LMSC uses a 12 mil cover compared to 6 mil covers for TRW and MDAC—12 mil covers result in a decrease in degradation.

The selected array sizing factors are plotted as a function of time in Figure 4.5-28. The factors are grouped slightly different from Table 4.5-26 to show the factors that result in a 10 watts/ft^2 design value and those factors that must be applied to it. The latter factors are the mission-related factors which include time-dependent factors, misorientation which is randomly variable and solar variation, which is cyclic. Each of the output and sizing factors will be discussed briefly.

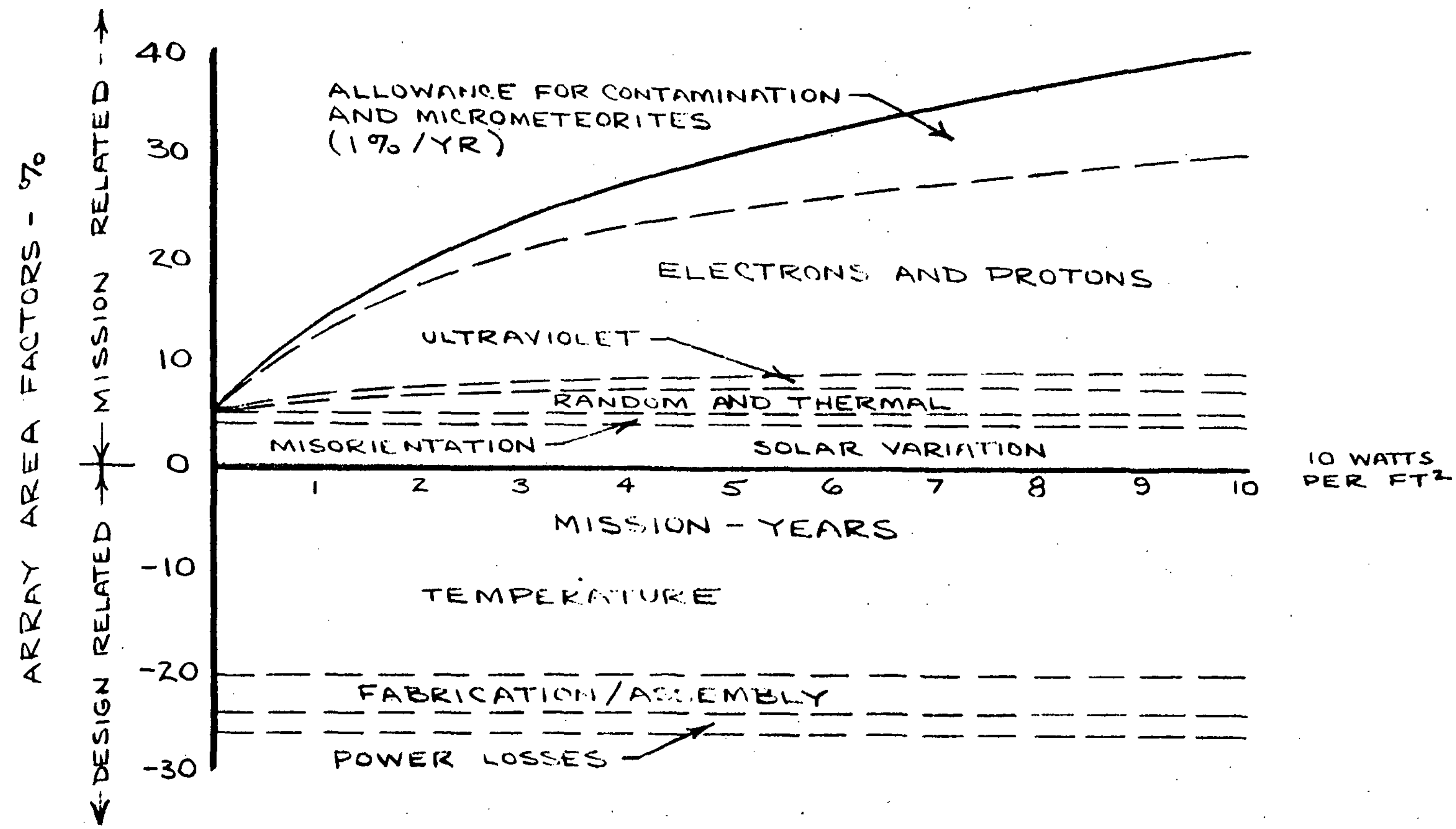


Figure 4.5-28 Solar Array Sizing Factors

Table 4.5-26

SOLAR ARRAY OUTPUT FACTORS

	300 Day	10 Years		
	Skylab	LMSC	TRW	MDAC
Time dependent factors				
Protons (trapped and solar)	4.25%	12.6%	1×10^{14} e/cm^2	14.7% (2.3×10^{14} e/cm^2)
Electrons			(17.0%)	1.2% (1.2×10^{13} e/cm^2)
Ultraviolet	0.9%	2.0% I_{SC} (max)	1.8% est	2.0% (LMSC/TRW)
Micrometeorites	0	5.0% I_{SC} (max)	0	1.0%/yr
Contamination	0	TBD	TBD	
Thermal cycling	1.0%	TBD	2.0% est	2.0% (TRW)
Random mechanical failure	0	TBD		
Time independent factors				
Solar variations	-	-	4%	4%
Cell efficiency	+	+	+	+
Shadowing	-	-	-	-
Misorientation	1% (±8°)	-	1% (±8%)	1% (±8°)
Temperature	-	-	-	20.1%
Power losses	-	-	-	2.5%
Fabrication/assembly	4%	3% I_{SC}	-	4%
Thicknesses (mils)				
Solar cells	14	12	8	8
Cover glasses	6	12	6	6

Trapped Radiation Environment (electrons and protons)—The degradation due to trapped electrons and protons were determined from the radiation *environments in the trapped radiation belts based upon data from* James I. Vette (Reference 4.5-7). The data used was for a 300 nmi orbit which adds a modest safety factor to the analysis. The various integration maps, AP6, AP1, and AP3 gave the proton fluxes as omnidirectional protons per square centimeter per day, and are designed for optimum accuracy at energy ranges of 4-30 MeV, 30-50 MeV, and 50-300 MeV, respectively. These maps consider only protons with energies greater than 4 MeV. However, there are protons with energies less than 4 MeV in the radiation belt, but it is difficult to design instrumentation to monitor such particles. For design purpose these protons need not be considered if a cover glass of 6 mils or greater is used. If no cover glass is used this large population of low energy protons (<4 MeV) will degrade the solar cells very rapidly. The electron fluxes used are an orbital integration of the projected electron environment, for a 1968 environment (maximum year).

It is convenient to refer to proton and electron damage by converting to an equivalent 1 MeV electron flux and then proceed to use the available data for solar cell damage as most damage data is based upon equivalent 1 MeV electrons. Coverglasses are used as shields and reduce the input energy spectrum by a fixed amount—equal to the energy that the selected shield just stops. Therefore, all electron fluxes below this value do not contribute to cell damage. The shield thickness and shield material selected are used to determine a shield thickness factor (gm/cm^2), which in turn is used to determine conversion factors for each energy of proton and electron flux to convert to equivalent 1 MeV electrons. The shield thickness factor for 6 mil microsheet (silica corning 0211) or 6 mil fused silica (silica corning 7940) is 0.035 gm/cm^2.

From the use of the conversion factors for shield thickness for electrons and protons the equivalent 1 MeV electron fluxes are calculated and shown in Table 4.5-27.

Table 4.5-27

EQUIVALENT 1 MEV ELECTRONS

10^{13} ELECTRONS/CM^2/YEAR (300 NMI)

	0°*	30°	60°	90°
Trapped Protons	0.005	1.562	2.296	1.693
Trapped Electrons	0.000	0.049	0.122	0.114
Total	0.005	1.611	2.418	1.807

*Orbit Inclination

The degradation of the solar cells can then be determined from Figure 4.5-29 which shows the maximum power degradation of 1 ohm-cm and 10 ohm-cm N on silicon solar cells as a function of 1 MeV electron/cm^2 flux. From the above fluxes, it is seen that a maximum power degradation of the solar cells will be about 3 percent for the first year and about 16 percent for 10 years due to the trapped radiation environment at a 60 degree orbit inclination.

The proton fluxes are not expected to change significantly from year to year, but the electron fluxes vary significantly over the 11-year solar cycle. The year selected (1968) for the electron environment was a maximum year for electron fluxes. If they are assumed constant at the maximum level it does not affect the yearly degradation as they do not contribute significantly to total cell degradation.

The degradation in the output power of solar cells is not linear even with a constant flux. The greatest degradation occurs early in the mission and becomes less each year. As indicated, a degradation of about 3 percent occurs in the first year. A degradation of about 11 percent occurs in the first 5 years. In the second 5 years, there is only about 5 percent of additional degradation.

Ultraviolet Radiation—The effects of ultraviolet radiation (UV) are not degradation of the solar cells, but a darkening of the adhesive used to attach the cover glass to the solar cell. The darkening results in a decrease in solar transmission of the cell and a decrease in solar cell output. A search

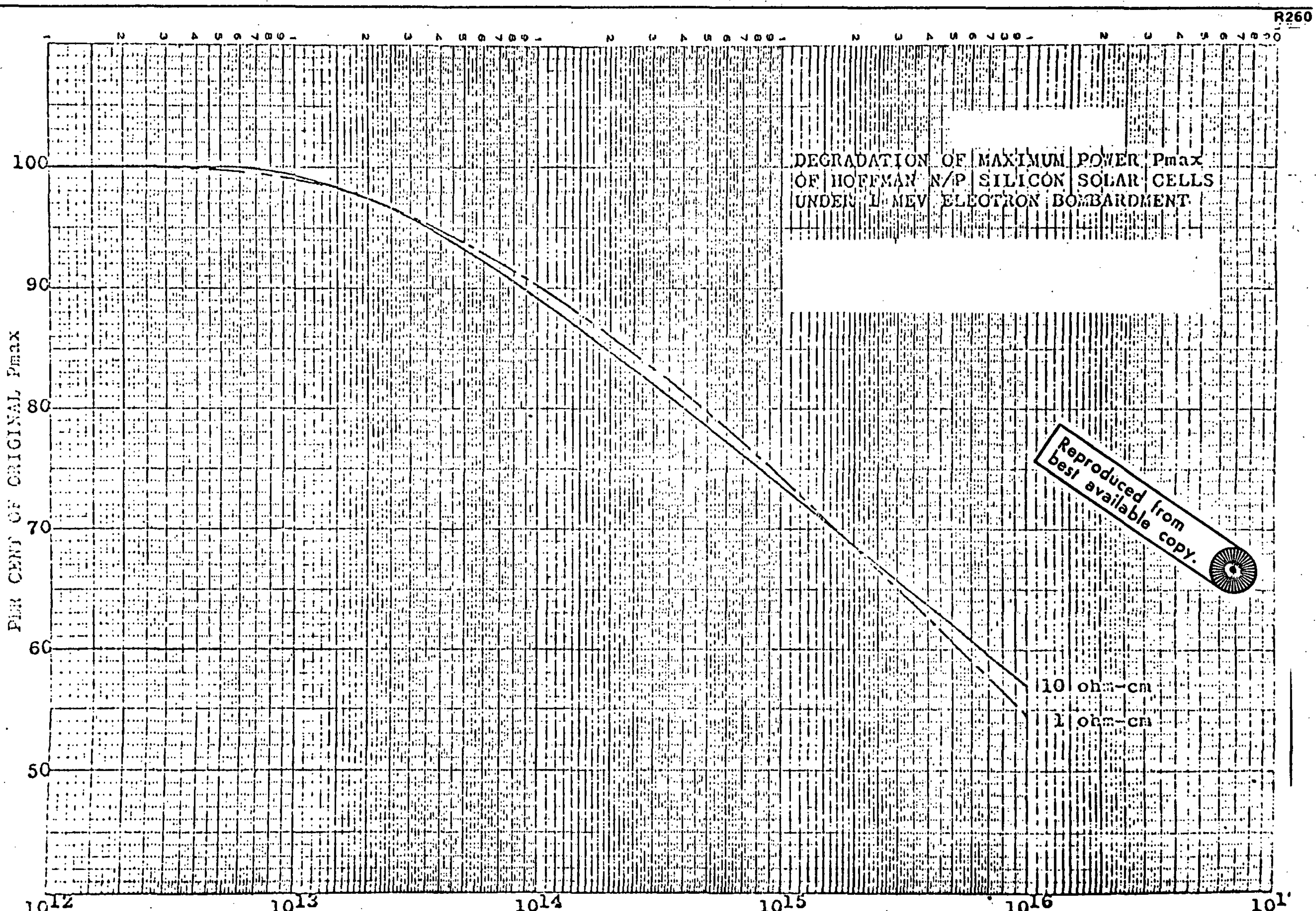


Figure 4.5-29 Electron Flux (1 Mev Electrons/cm^2)

for materials that would be less susceptible to UV resulted in the development of transparent adhesives that darkened only slightly over long periods of UV exposure and were unharmed by penetrating radiation. The selected adhesive, Dow-corning XR-6-3489, has been shown to degrade less than 1 percent under a 1-year equivalent UV exposure, and much less each additional year. Therefore, a degradation of 2 percent was selected for the 10-year mission.

Micrometeoroids—The micrometeoroid population has a mass range extending from cosmic dust particles of the order of 10^{-15} gm or less to asteroids in excess of 10^{10} kg with average velocities of about 30 km/sec. The particle flux near the Earth is much greater than the flux in interplanetary space since particles are trapped by the Earth's gravity field. The effect of micrometeoroid bombardment on solar cells is surface erosion of the cover slide in the form of small craters and occasional punctures by larger particles (see Figure 4.5-30).

Micrometeoroid flux data vary according to the model selected and the models differ widely, indicating the uncertainty in flux. A large uncertainty also exists on the effects of this environment.

Contamination—A potential and undefined source of degradation is the contamination of the environment around the spacecraft and the arrays and the deposits collecting on the solar cells. This contamination may act to decrease the solar transmission to the solar cells and decrease the output (see Figure 4.5-30). The contamination could be caused by P/RCS exhaust, EC/LS effluents, or other sources. The extent of this contamination and its effects on solar array output have not been defined. However, because of the location and size of the arrays the effects should not be uniform and, therefore, an analysis of the effects would be complex.

Because the effects of both micrometeoroids and contamination are not defined, it was necessary to assume a degradation to account for the potential decreases in solar array output. An allowance was assumed of 1 percent per year to account for these effects.

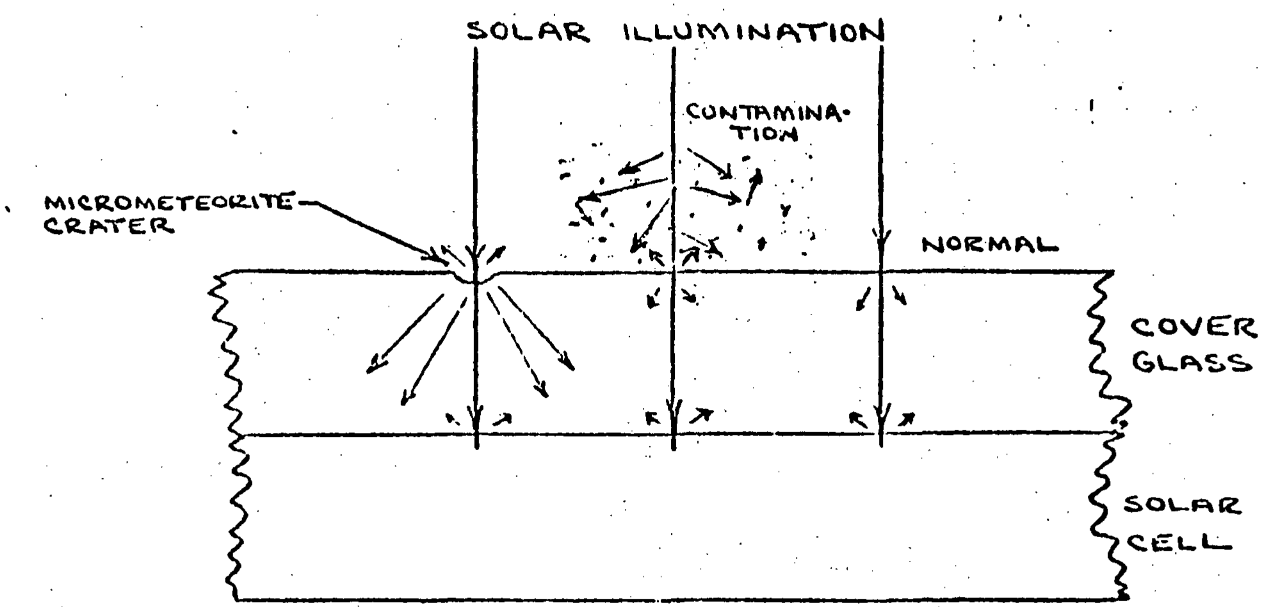


Figure 4.5-30 Effects of Micrometeorites and Contamination on Solar Cell Output

Thermal Cycling and Random Mechanical Failure—Thermal cycling of the array as the array enters and leaves the eclipse portion of each orbit and random mechanical failures are usually associated with open circuits in leads and contacts rather than within the cells. This type of failure is expected to occur early in the mission due to weak contacts and interconnects and much less later in the mission. Therefore, a degradation of about 1 percent was used for the first year with a 10-year total of 2 percent.

Solar Variations—The allowance for solar variation is cyclic and is due to variations in the Astronomical Unit (AU) distance because of the eccentricity of the Earth's orbit. Therefore, for certain periods of the year up to 4 percent excess capability is available from the array. It is not feasible, at this time, to program the loads to eliminate the degradation because of the gradual change and long time at each variation.

Cell Efficiency—The selection of solar cell efficiency has a large impact on the output of the solar array. Solar cells are usually specified at an average efficiency with limits in either direction. The average efficiency is selected by the number of cells required and the cost. For the Modular Space Station, 11-percent-efficient cells are selected but the advantages of higher efficiency cells, if available, is obvious. Therefore, a plus sign is indicated for cell efficiency improvement.

Shadowing—The shadowing of cells can result in decreases in output that are more than proportional to the amount of shadowing. Other degradation effects have also been identified as a result of shadowing. A minus sign is indicated but no values are shown. Further discussion of shadowing effects is given in a subsequent section.

Misorientation—It is not practical and would be difficult to orient the arrays within very tight tolerances of a few degrees or less. An allowance for misorientation of ±8 degrees is an accepted and reasonable orientation tolerance. As the solar array output will vary at approximately the cosine of the angle of misorientation, this misorientation allowance results in a factor of only 1 percent.

Temperature—Increased temperatures will decrease the output of the solar arrays. The effects of temperature on the current and voltage characteristics of a single 2 by 2 cm solar cell is shown in Figure 4.5-31 for 28°C, at which the cells are calibrated, and for the predicted ranges of array temperature expected in low Earth orbit. As shown, the current and voltage do not vary the same nor do they vary linearly. The variation in maximum power due to temperature is shown in Figure 4.5-32 and is nearly linear, but is made up of voltage and current variations which are not linear (see Figures 4.5-33 and 4.5-34). Of particular concern is the large variation in voltage due to temperature, as the system must be designed to handle this variation. Using the data from Figures 4.5-31 through 4.5-34, a maximum power degradation of 20.1 percent occurs as the temperature is increased from 28°C to 77°C. A preliminary thermal analysis for the solar array was prepared by the Lockheed Missile and Space Company (LMSC) on NASA-MSC Contract NAS9-11039. This analysis was reported in Section 4.1.3.7 of the First Topical Report "Evaluation of Space Station Solar Array Technology and Recommended Advanced Development Program," Document LMSC-A981486, December 1970. Thermal cycling test experience was reported in Section 4.1.3.8 of the same document covering a variety of test temperature regimes from -300°F to +285°F, which are much greater than those anticipated (-106.6°F to +103°F). The LMSC laboratory testing with a liquid nitrogen chamber and heaters indicated a lower high temperature at equilibrium (52°C for flexible panels vs 63°C for rigid panels), lower low temperature at equilibrium (-112°C for flexible panels vs -96°C for rigid panels), and higher rates of temperature change (26°F or 14.4°C per minute for flexible vs 20°F or 11.1°C per minute for rigid arrays). LMSC also noted that the upper temperature for rigid arrays (114.5°F or 63°C) matches within 2°F their actual flight experience, and the upper test limits are therefore considered to be valid. The lower temperatures in the laboratory tests are considered to be unrealistically low, because Earth albedo was not simulated; -140°F in the laboratory test being much lower than the minimum actual value of -100°F (-73.3°C) recorded in dozens of LMSC spacecraft flights. The range assumed for the MDAC study is +75°C (+103°F) and -77°C (-106.6°F). Comparing the LMSC lab test (-96°C) to the actual (-73.3°C) for rigid panels, the lower LMSC lab test value

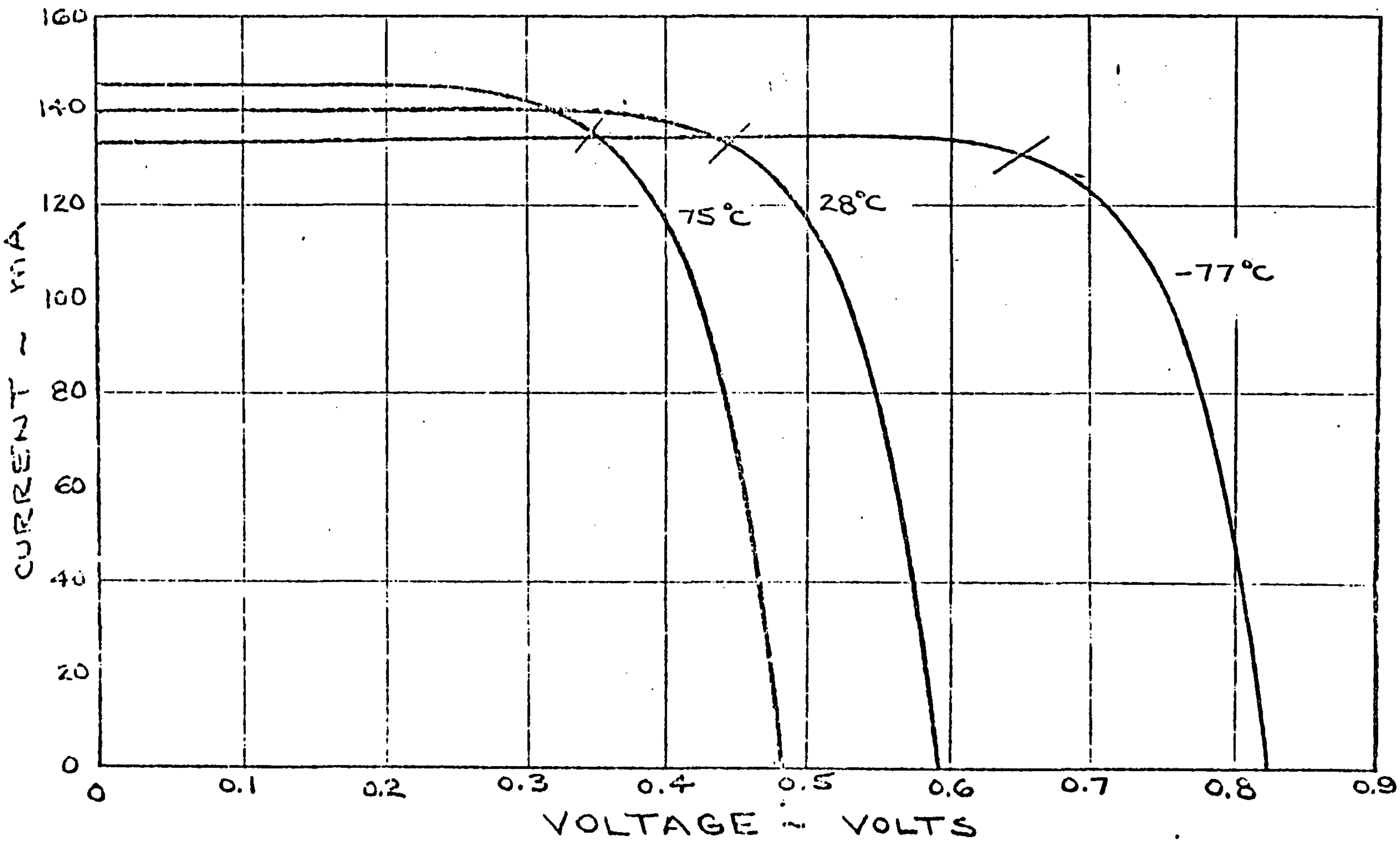


Figure 4.5-31 Solar Cell Thermal Characteristics - 2 x 2 cm Solar Cell

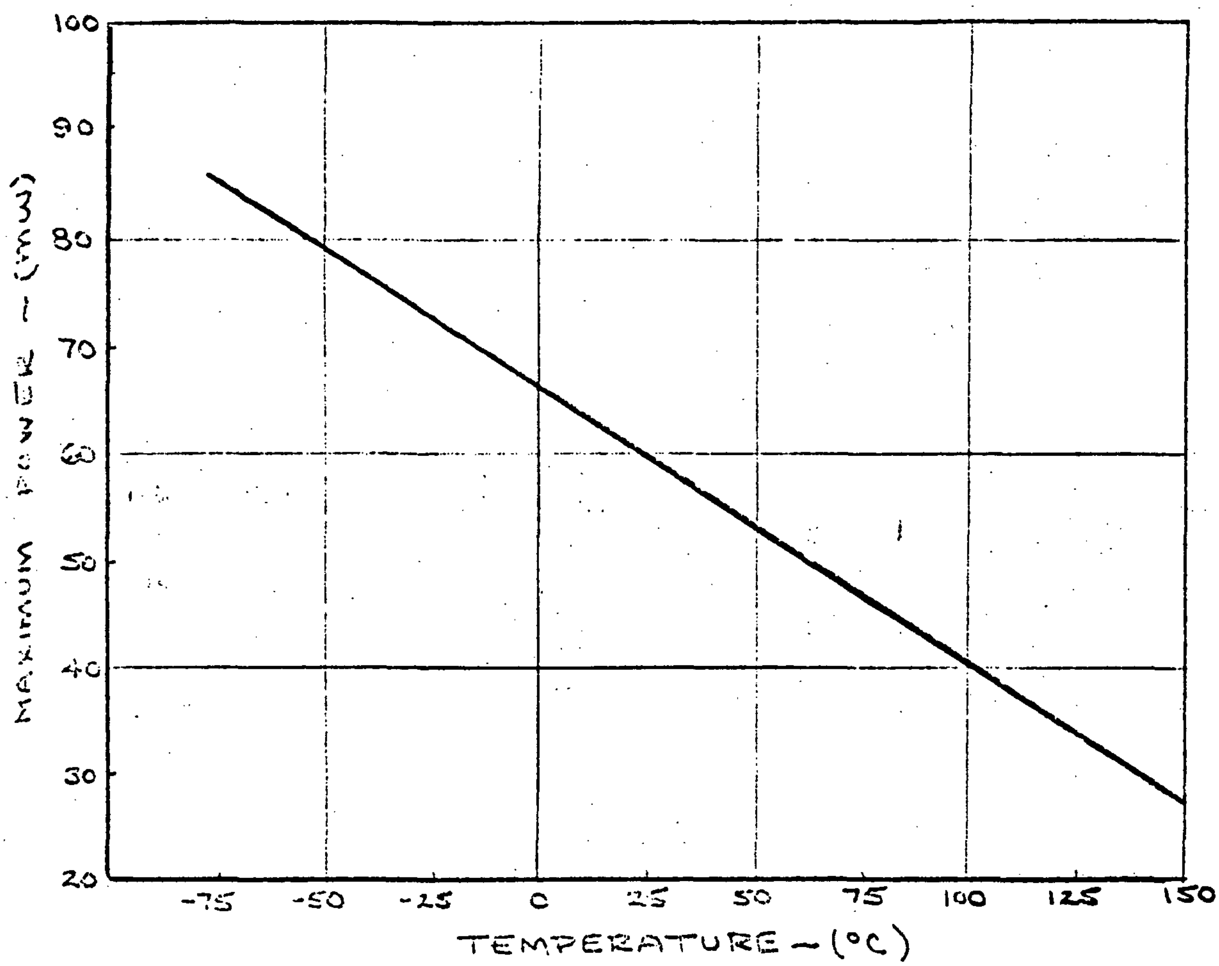


Figure 4.5-32 Solar Cell Thermal Characteristics - 1-3 ohm/cm Base Resistivity

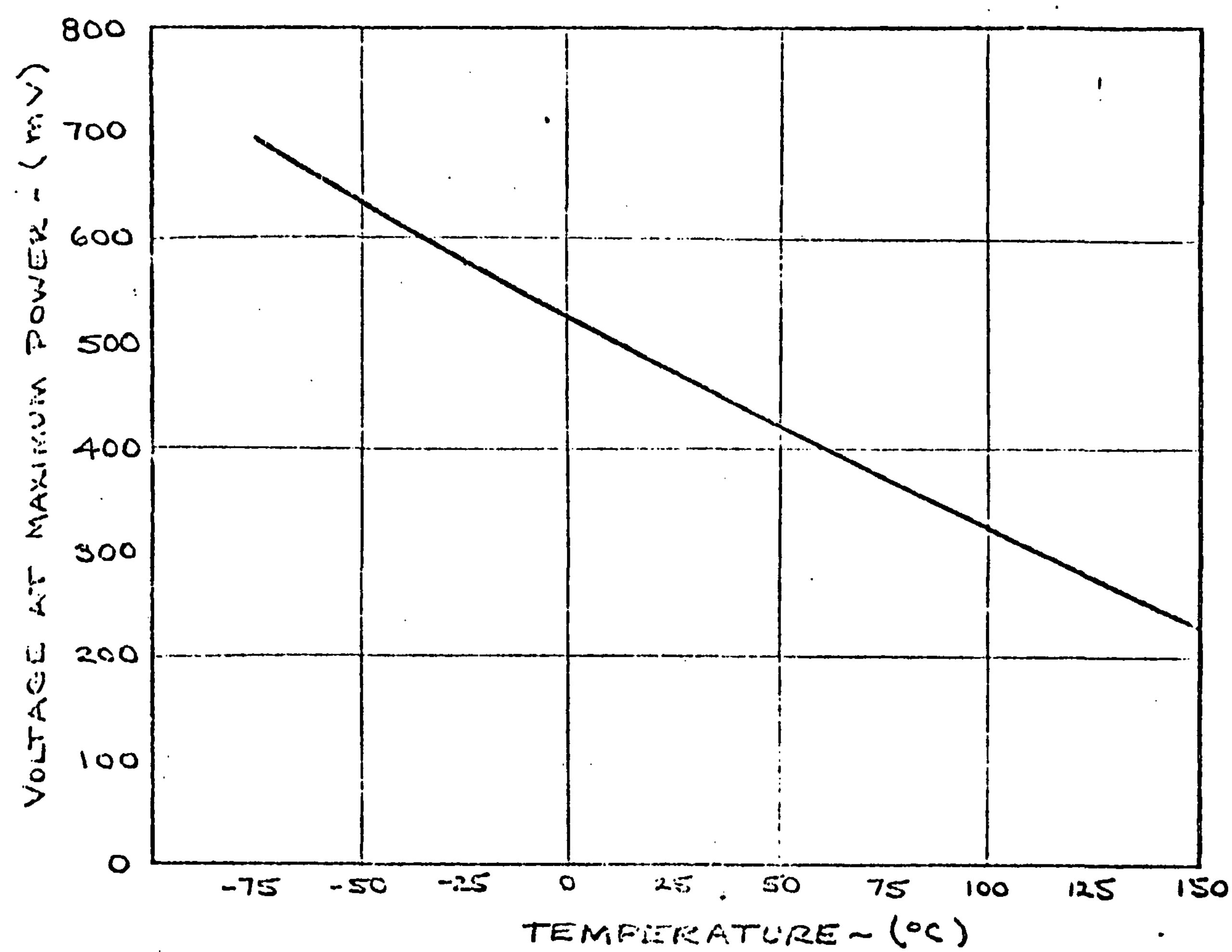


Figure 4.5-33 Solar Cell Thermal Characteristics - 1-3 ohm/cm Base Resistivity

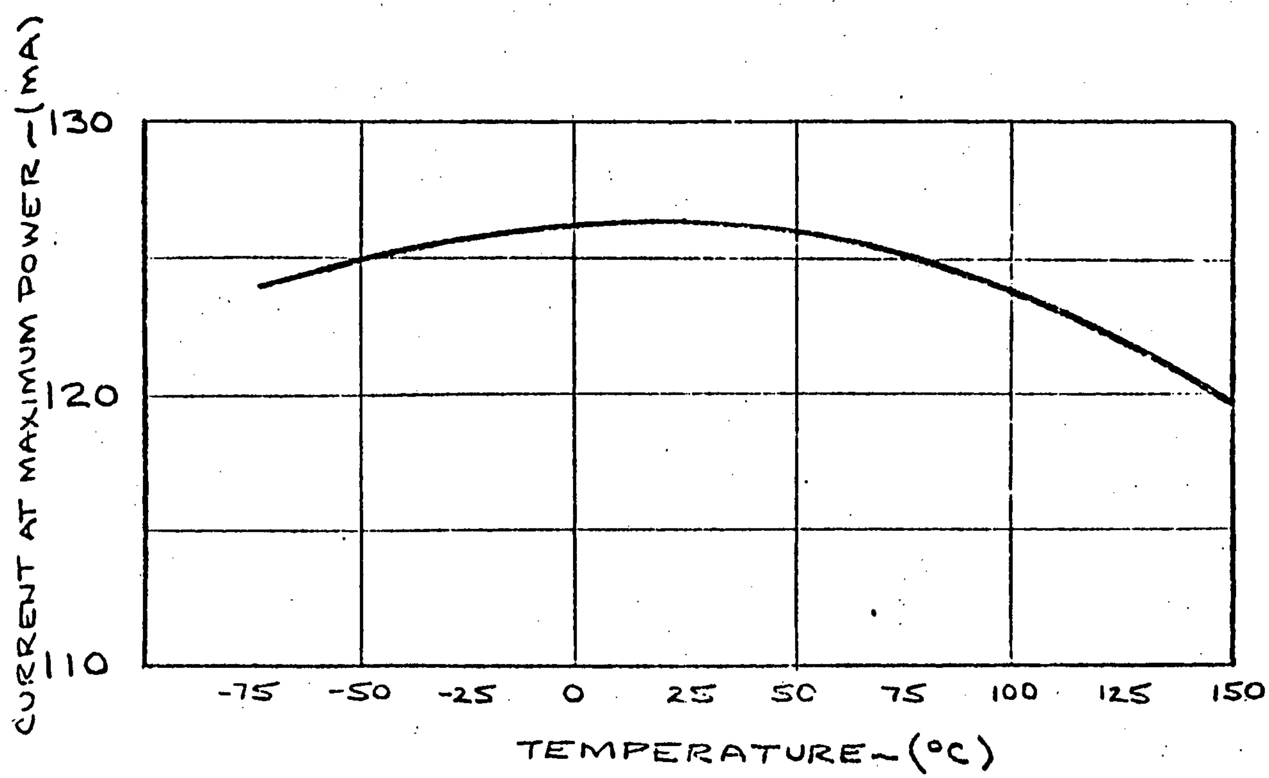


Figure 4.5-34 Solar Cell Thermal Characteristics - 1-3 ohm/cm Base Resistivity

(-112°C) should provide a corresponding expected value of -85.6°C (-124.9°F). The flexible panel is more sensitive to Earth albedo, however, and LMSC has projected -73.3°C (-100°F) as a more optimistic lower value. These values are therefore comparable to the MDAC assumption of -77°C (106.6°F). Furthermore, the low temperature value is not related to performance, and is only used to define a desirable range for thermal cycle life testing. The upper temperature, which affects performance directly, is assumed by MDAC to be higher (+75°C) than LMSC test values, and thus is more conservative than the LMSC expected value of +52°C. These studies are still incomplete for the specific solar panel substrate materials, and such further work is essential during the Phase C/D development program.

Power Losses and Fabrication/Assembly—The power losses are a design factor due to line voltage drops in the transmission of power from the array to the vehicle and includes isolation diode voltage drops. These are accounted for separately. A loss of 1.67 percent provides for 1.5 volts drop in panel edge harness and 0.5 volt on the inboard support assembly. Cable power loss to the source buses and isolation diode loss of 0.8 volt are included in the Design Factor of 2.18.

The fabrication/assembly losses are due to cell mismatch when the cells are connected in series and parallel due to differences in voltage and current between cells and the attachment of coverglasses used to protect against radiation and micrometeorites. The coverglass and adhesive will result in a small decrease in transmission of solar energy to the cell.

Solar Array Shadowing

A preliminary analysis of solar array shadowing was conducted to determine the extent of shadowing that occurs for a typical Modular Space Station configuration in a horizontal and X-POP/OR vehicle orientation, and for several representative orbits with a launch inclination of 55 degrees. The preliminary analysis included 33 photographs of the Modular Space Station with a scaled solar array, positioned on the MDAC Orbital Simulator to show the shadowing of the arrays in representative orbits, at various seasons of the year, for several β angles, and for both horizontal and X-POP/OR orientations with the camera or an observer located at the sun. This analysis has been updated because of changes in vehicle configuration and in array area and shape. However, the changes that were made allowed the use of the previous photographs in updating the analysis.

The results of the updated shadowing analysis are shown in Figures 4.5-35 and 4.5-36 for horizontal and X-POP/OR orientation, respectively. Each figure shows the percent of array shadowed at several orbit positions, with the points connected by straight lines. The solid lines represent a worst case with three modules docked to the power module. If modules are not docked to the power module, the shadowing is reduced to the values shown by the dashed lines.

Shadowing for horizontal orientation is less in magnitude than for X-POP/OR orientation. However, the X-POP/OR shadowing can be reduced to zero by periodically rotating the Space Station by 180 degrees about the Z axis so that the arrays are always between the station and the sun.

For some orbits, excess power is available from the solar arrays, and therefore some shadowing can be allowed without compromising the power requirements. As the β angle increases, the station will see increasing illuminated time and decreasing dark time until at a β angle of 69 degrees or greater the station will be in a totally illuminated orbit. As the dark time decreases, the battery requirements decrease and less energy is required from the arrays during the illuminated periods to recharge the batteries, thus resulting in excess solar array capability. At β angles of 69 degrees and higher, the battery charging requirements are reduced to zero. The excess array capability will allow shadowing of solar array area in excess of 50 percent at the higher β angles.

The maximum allowable shadow is shown by the solid line in Figure 4.5-37 as a function of β angle. Also shown are the maximum shadowing averaged over the total sunlight period, as a function of β angles where higher shadowing is allowed. The maximum average shadow for the X-POP/OR orientation is always below the allowable shadow curve. This is not true for the

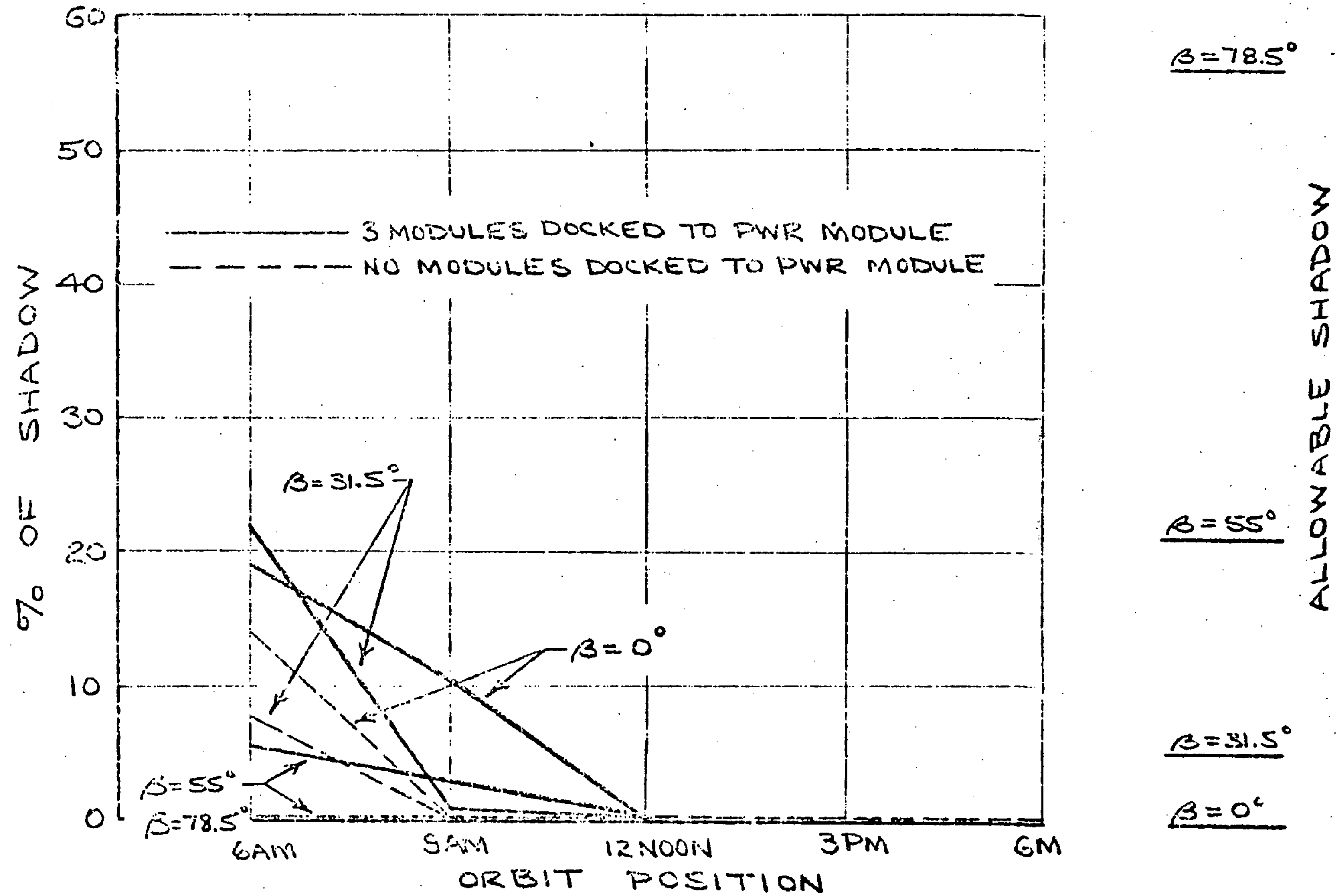


Figure 4.5-35 Solar Array Shadowing - Horizontal Orientation

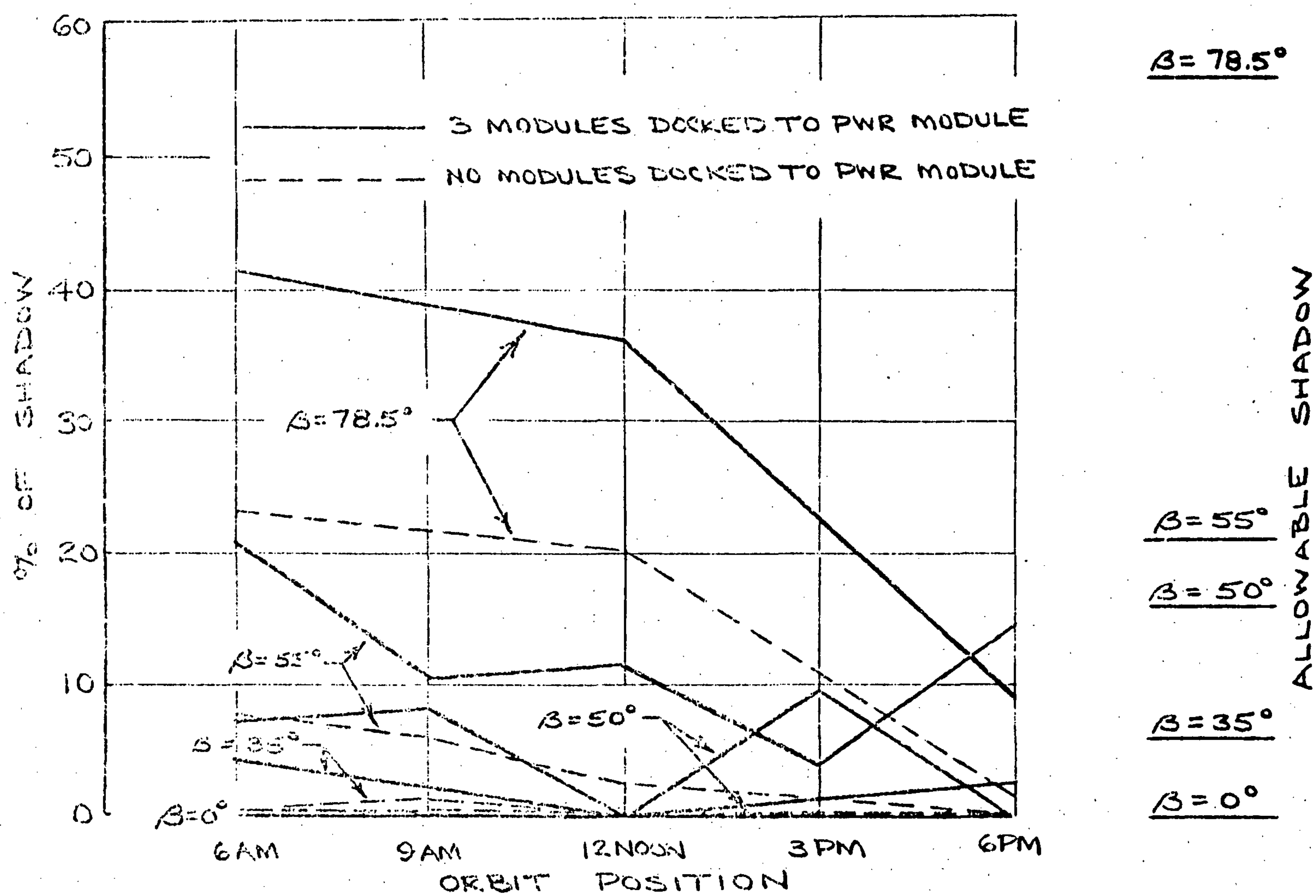


Figure 4.5-36 Solar Array Shadowing - X-Pop/or Orientation

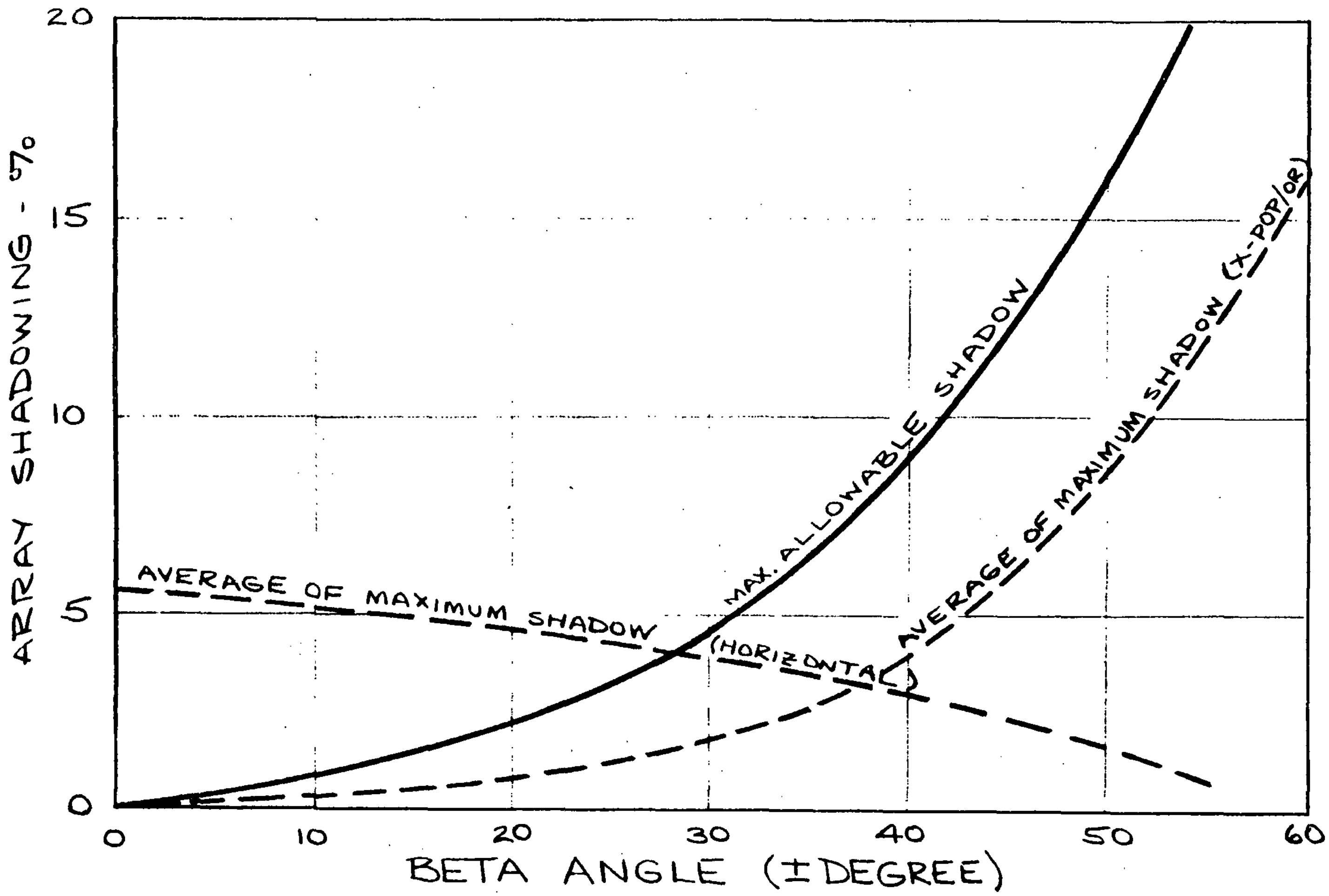


Figure 4.5-37 Array Shadowing Allowance

horizontal orientation, which has been selected as the baseline orientation. At low angles where no shadow is allowable, average shadows of 5.66 percent occur. The Space Station arrays, loads, or operations must be designed to account for this loss.

There are several approaches available to minimize or eliminate the shadowing that occurs at the low β angles in the horizontal orbit. The methods that show merit are shown in Table 4.5-28. The problem is not as bad as it was first thought to be. Because of the excess array capability required to account for array degradation, excess power is available except during the latter part of the ISS mission; that is, a deficiency can occur during quarters 16 through 20 only. During some of these five quarters, the Earth orbit eccentricity recovers up to 4 percent of the deficiency. During the remaining quarters, many of the solutions indicated would be acceptable. Further analysis is required to make the best selection for each quarter from the many approaches that are available to eliminate the shadowing problem. Options 2 and 5 are the simplest. Option 3 would be relatively simple as the skew angle is small, but might interfere with some precisely oriented experiment. Option 6 is practical unless a large number of docked modules are needed. Option 7 is the least desirable for the short period of limitation as it is the most expensive choice.

Table 4.5-28

SHADOWING IMPACT REDUCTION
(Limiting Orbits Occur During Quarters 16 Through 20 Only)

1. Use excess array area allowed for degradation (>5.66 percent except during quarters 16 through 20).
2. Reduce discretionary load during critical periods.
3. Skew station orientation during limiting orbits (low β angles, 0 to ± 28 degrees).
4. Change to X-POP/OR orientation.
5. Schedule experiments requiring high power to periods of high β angles (63 percent of time).
6. Relocate docked modules to other than Power Subsystem Module during low β (0 to $\pm 28°$) angles.
7. Increase solar array area by 5.66 percent.

Solar Array Orientation

A tradeoff analysis was conducted to determine the optimum gimbal axis system for solar array orientation. Each gimbal mechanical design is identical, so no additional development is associated with the second degree of freedom; however, the orientation servo electronics require angular constraints to avoid excessive cable wind-up when used with spiral coils and trailing cables.

Figure 4.5-38 shows the effects of the selected number of gimbal axes on the solar array area—increasing cost, weight, and stowage volume also. All curves are based on the Space Station maintaining a constant horizontal flight attitude for the benefit of the Earth-oriented experiments. Because the system is strongly affected by inclination to the ecliptic plane, the solar array area determined for 0, 1, or 2-axis gimballing is shown as a function of inclination to the ecliptic plane. As the inclination to the ecliptic plane increases, the eclipse period decreases, resulting in a higher average power capability. Because of orbital precession, however, the average power capability varies at each inclination to the ecliptic plane as shown in the cross-hatched areas.

The dashed line shows the reduction of required solar array power which can be realized if orbital precession is prevented. A considerable propellant expenditure would be necessary, however, to maintain an orbit plane that would maximize the sunlight time at high inclination angles.

The design curve (solid line) shows the worst-case array area needed for each kilowatt of average unregulated load power. The bar at the bottom indicates the range of sun angle—this is the sum of the orbit inclination angle and the annual ecliptic plane inclination angle, and varies between ±78.5 degrees.

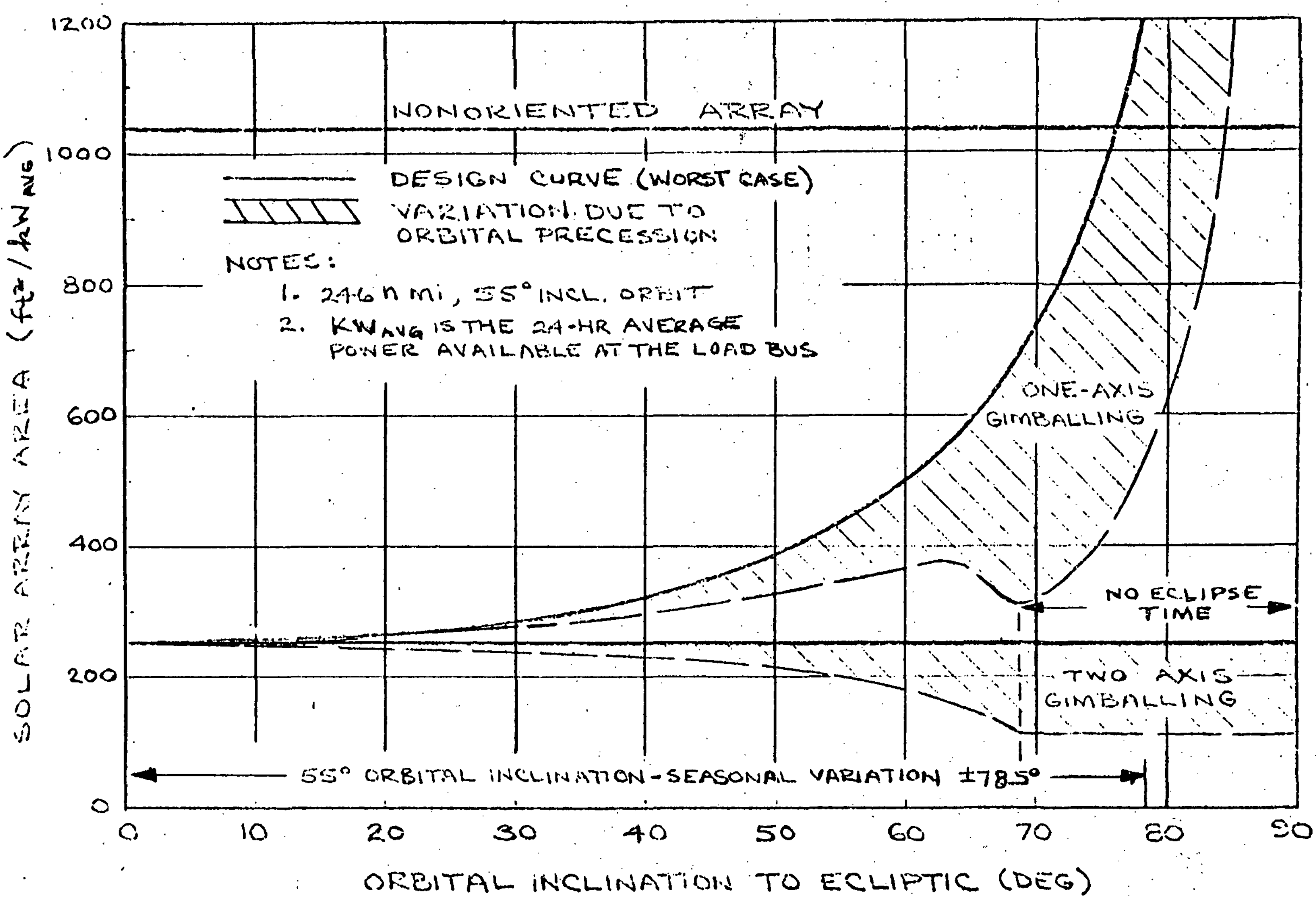


Figure 4.5-38 Solar Array Gimballing

The design curve for the unoriented (zero-axis) array is a constant solar array area of over 1,000 ft^2/kw average. The solar cells could be placed on both sides of the panels so panel area would be one-half of the value shown. The panels would be omnidirectional, set at optimum angles to give a nearly constant effective area for all orientations.

The design curve for one-axis gimballing (semi-oriented) is a simple cosine curve which reflects the tilt of the panel from the sun line, although orbital tracking is performed with one axis. The one-axis gimballing can result in a higher solar cell array area requirement than the non-oriented array if designed for the worst case, which is 78.5° orbital inclination to the ecliptic plane.

The design curve for a fully oriented (two-axis) array is a constant solar array area of about 250 ft^2/kw average due to the ability to fully track the sun, regardless of launch inclination. It therefore offers maximum performance and flexibility. The added gimbal cost is quite low compared with the savings in panel cost, weight, and stored volume. The full two-axis orientation is therefore selected. As shown, some orbits at a 55-degree launch inclination will be total sunlight orbits at high β angles above 69 degrees.

An assessment of gimbal axis ranges and range rates was made, with a brief consideration of the merits and demerits of the several station flight attitudes or orientations. The results are shown in Table 4.3-29 as an impact chart. Although the horizontal orientation is least favorable for solar arrays, it was selected for benefit to experiments and other factors not associated with the EPS. Accommodation of horizontal orientation equips the solar array gimbals to accommodate any other station orientation. Figures 4.5-39 and 4.5-40 show the relatively complex gimbal operations necessary to accommodate horizontal flight and to accomplish feathering and recycling or unwinding of trailing cables during Earth eclipse. Combined gimbal action is clearly indicated.

Table 4.5-29

SPACE STATION ORIENTATION IMPACTS—ELECTRICAL POWER SUBSYSTEM

Impacts	Candidate Orientations: Horizontal (Longitudinal Axis Along Velocity Vector, One Side Toward Earth)	POP/OR Longitudinal Axis ⊥ to Orbit Plane, Roll at Orbit Rate)	Others (i.e., Inertial, Solar as Appropriate for Subsystem Impacts)
Shadow			
Array/array shadowing for GSS	Occurs at low β (worst condition)	Occurs at high β (best condition)	Solar or other inertial—no shadow
Station and RAM/array shadowing	Occurs at low β (worst condition)	Occurs at high β (best condition)	(Except X-inertial away from sun)
Contamination Potential			
Drag makeup thrusting	Lowest (ISS) Highest (GSS)	Low for both ISS and GSS	Variable—suggest interrupted thrusting
Gimbal Requirements			
α Range (outer or tunnel gimbal)	±180 degrees, average 4 deg/min**	±235 degrees* at orbit rate (4 deg/min)	Fixed (0 to 180 degrees)
β Range (inner or mast gimbal)	±235 degrees at 4 deg/min*	±78.5 degrees at orbit declination rate (6.3 deg/day)	Fixed (0 to 180 degrees)
Rotational speed	Highest (22 deg/min) and Lowest (0 deg/min)	Moderate, approximate constant (4 deg/min)	Zero
Motion complexity	Highest (combined action)	Low (single action)	Lowest (no action)

*22 deg/min rate for eclipse period unwinding of cable and feathering.

**Average rate for orbits is shown; range is 0 deg/min to 22 deg/min.

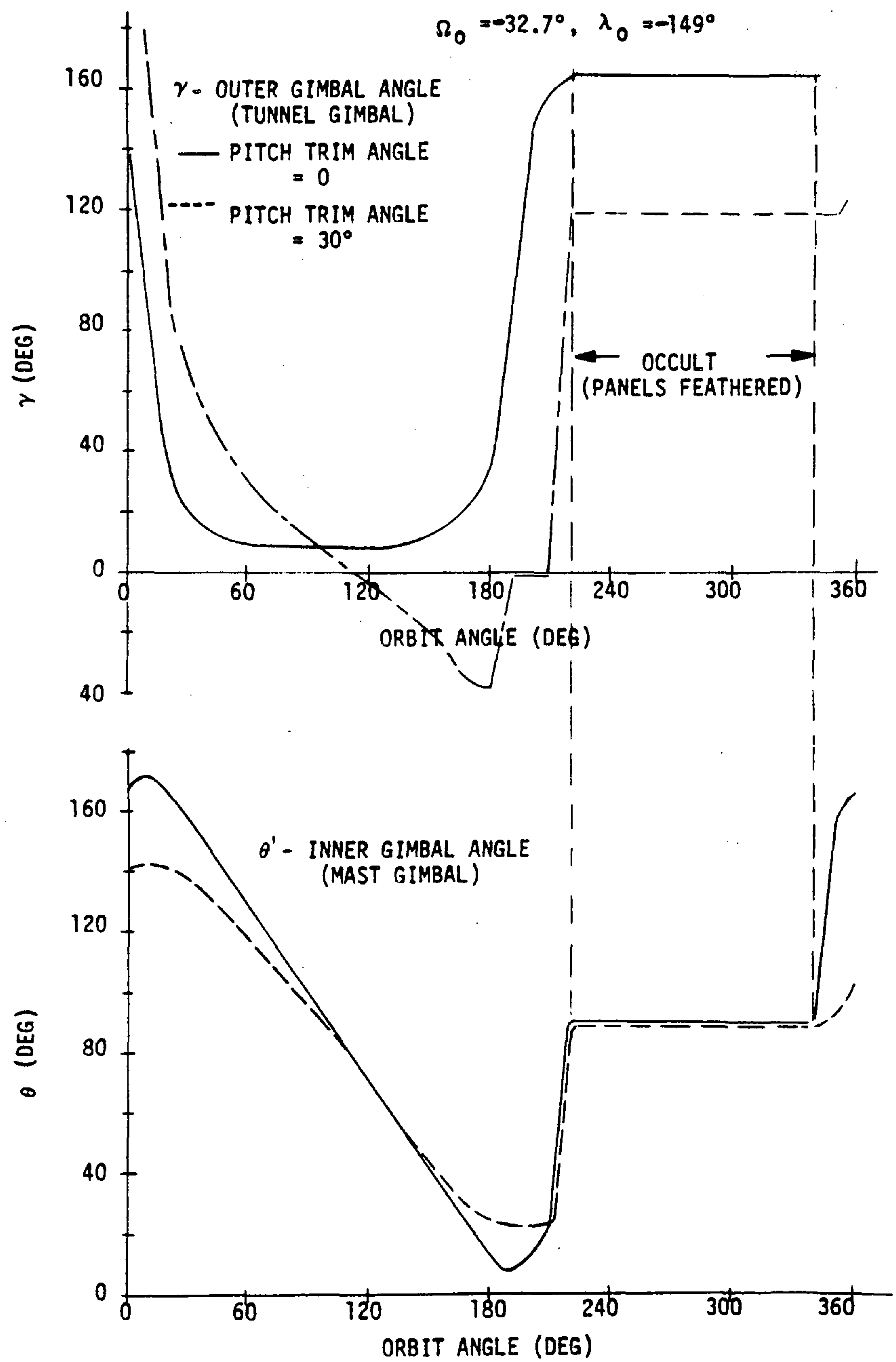


Figure 4.5-39 Solar Panel Gimbal Angles

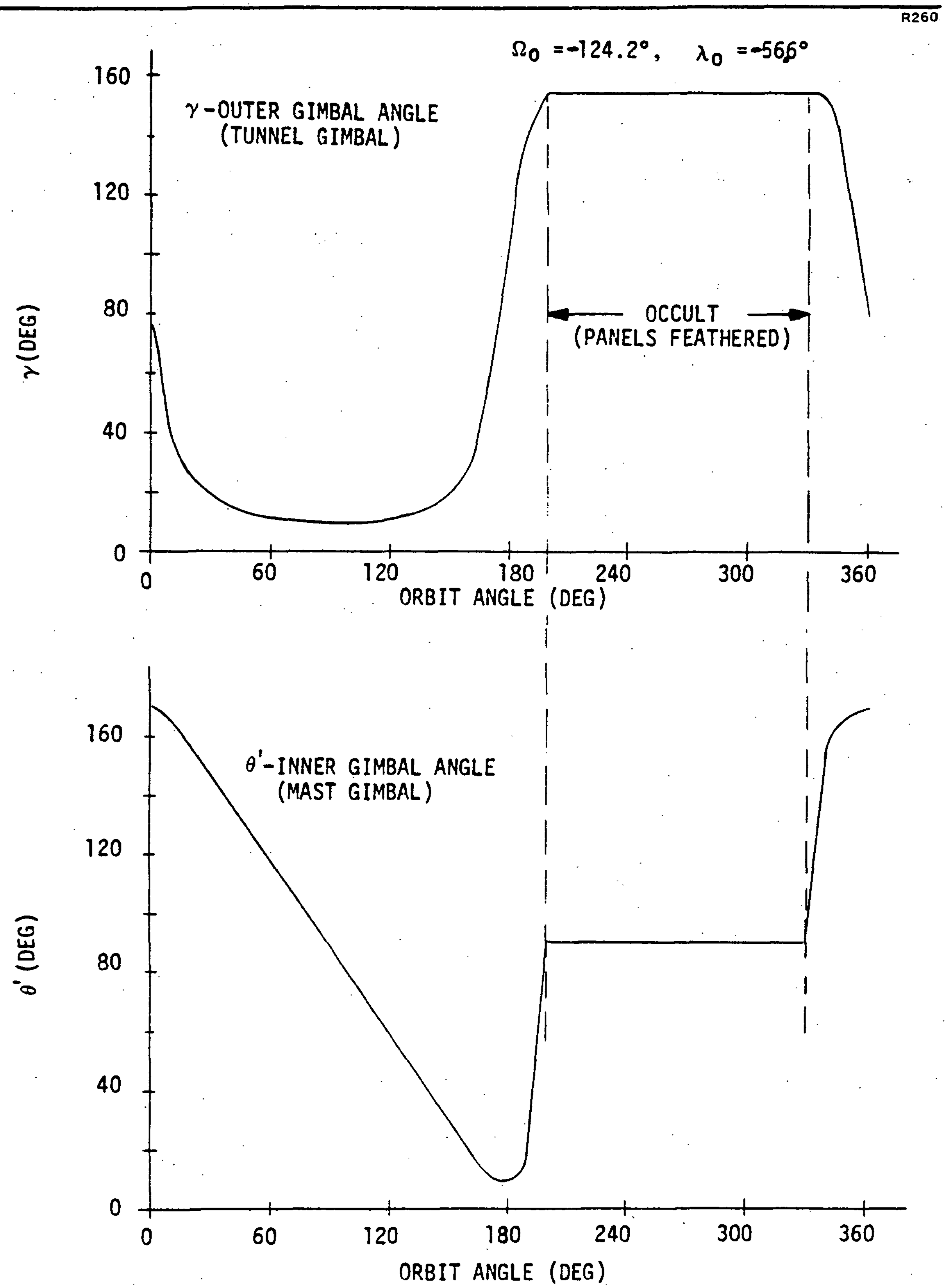


Figure 4.5-40 Solar Panel Gimbal Angles

4.5.4.2 Regulation and Control Assembly Design Analysis and Trade Studies

4.5.4.2.1 Regulation and Control Requirements

System voltage is regulated to provide power quality equal to or better than that specified in MIL-STD-704 for Category B equipment. Control functions are accomplished automatically with provisions for manual override.

Selected functions such as programmable bus switching are controlled through the DMS. Others depend on EPS internal closed loop control, with status outputs to the DMS. Still others, such as array voltage regulation, utilize both internal closed loop and DMS control.

In general, internal closed-loop control is used where fast response and essentially continuous monitoring is required. Redundant sensors and redundant hardwire sensing circuits (internal closed-loop control) or redundant data buses (DMS control) are employed to provide the required reliability.

Comparison of Regulation and Control Alternatives

The principal trades in the area of regulation and control lie in selection of the method for system voltage control. Either series or shunt regulation or a combination of these methods can be used. Considerable attention is given to the relative flexibility afforded by each method. This includes assessments of the effects of errors in predicting design point conditions such as array operating temperature and end-of-life current and voltage capabilities. The ability to utilize excess power available from the array in earlier years of station operation is also examined. These points are covered specifically in the Section titled "Sensitivity Analysis."

The selected method utilizes sequential partial shunt regulators to control solar array voltage, and series buck regulators to control battery load voltage (reference Figure 4.5-3 of Section 4.5.1). Sequential partial shunt regulators are chosen for array voltage control because they require negligible power at maximum power load and end of life and yet dissipate less power than either non-sequential partial shunts or series regulators (with voltage limiters) at minimum load. In addition to providing a high degree of operational flexibility, the sequential partial shunt regulator (SPSR) maximizes EPS overall efficiency, thereby minimizing the required solar array area. Pulse-width-modulated (PWM) buck type regulators are selected for battery voltage control because of their high and relatively flat efficiency characteristic over the anticipated operating load range.

Figure 4.5-41 shows the basic regulating systems which were analyzed in detail. The analysis reflects a nominal system voltage of 115 vdc. Shunt limiters are incorporated in the series regulator schemes to protect the regulators against overvoltage as the array emerges from an eclipse and array temperature is low. A limit of 150 vdc was assumed for this study.

Weight and power penalties shown for the series schemes are referenced to the sequential partial shunt regulator (SPSR) scheme as a baseline. All penalties are based on ISS load conditions. The 1,200 watt power penalty reflects an assumed series regulator efficiency of 97 percent. Weight penalties represent the difference between the weight of the sequential partial shunt regulators and the sum of series regulator plus shunt limiter circuitry (weight for the additional solar array area required to make up for the 1,200 watt loss is not included in the ΔW values shown). No penalties are included for the use of a peak power sensor in the series schemes. This is dealt with in detail in the analysis of regulation sensitivity to array operating temperature.

An alternate scheme which locates the series regulators at the main distributor bus in each module was also considered. In this scheme, battery power is fed through the same regulator that is used to condition solar array power—this is a feature of both the ATM and OWS/AM power systems. This concept is slightly less efficient and heavier than the series schemes shown in Figure 4.5-41.

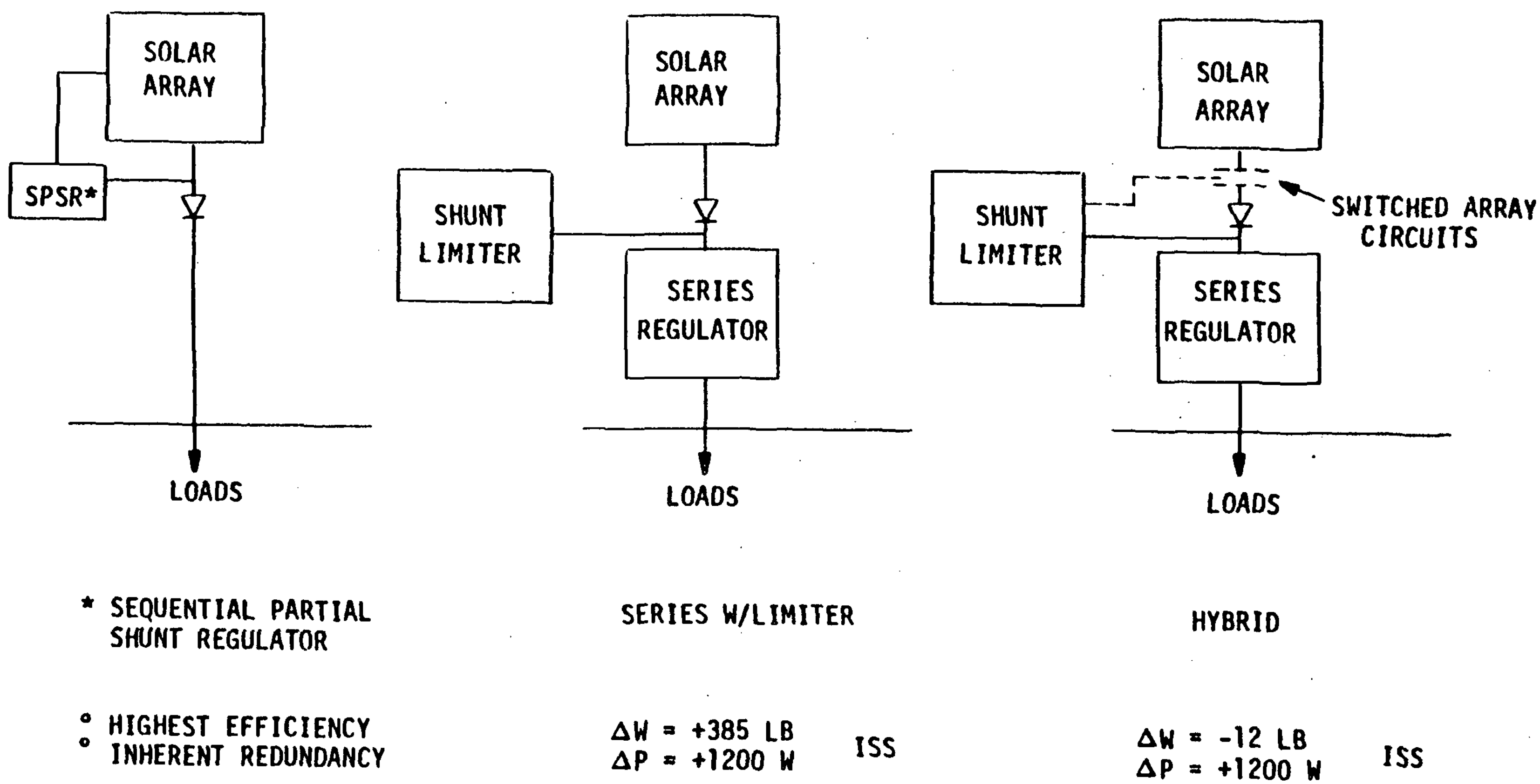


Figure 4.5-41 Regulation Trades - 115 vdc Distribution

Sensitivity Analysis

The selection of the SPSR over the series regulator reflects comparison of the operational flexibility and sensitivity to off-design conditions as assessed in the following paragraphs.

Sensitivity of Shunt and Series Regulation Methods to Solar Array Operating Temperature

A comparative analysis was made of the sensitivities to solar array operating temperatures of the sequential partial shunt and series regulation methods. The study covered a range of ±20°C around the nominal design temperature of 75°C. The characteristics for each regulator type at 75°C are shown on Figures 4.5-42 and 4.5-43.

Figure 4.5-42 shows solar array current-voltage (I-V) characteristics at the nominal stabilized array operating temperature (75°C) after 10 years of operation. Maximum power available from the array is 40 percent less than for the same array when new. The maximum power point P_{max} is the design point for the SPSR; array capability just matches load requirements. The required voltage is available with no dissipation in the regulator shunt elements.

If the same array is controlled by the series regulating method, the design point (array operating point at end of life) is displaced from P_{max} by the peak power sensor bias of 3.5 percent. This is shown in Figure 4.5-43. The array voltage at this point is fixed by the regulator full load drop. Assuming a regulator drop of 3 VDC, the array must operate at 123 VDC to provide 120 VDC at the regulator output. The regulator operating point on the 120-VDC constant voltage line is then established by the regulator efficiency, assumed to be 97 percent. The resulting regulator output is 93.5 percent P_{max}. Since available regulated power with the SPSR is P_{max}, the series regulator provides only 93.5 percent as much power for the loads as the SPSR at array end of life (10 years) and an array operating temperature of 75°C. The results of this study are summarized in Table 4.5-30 to indicate the operating characteristics after 10 years.

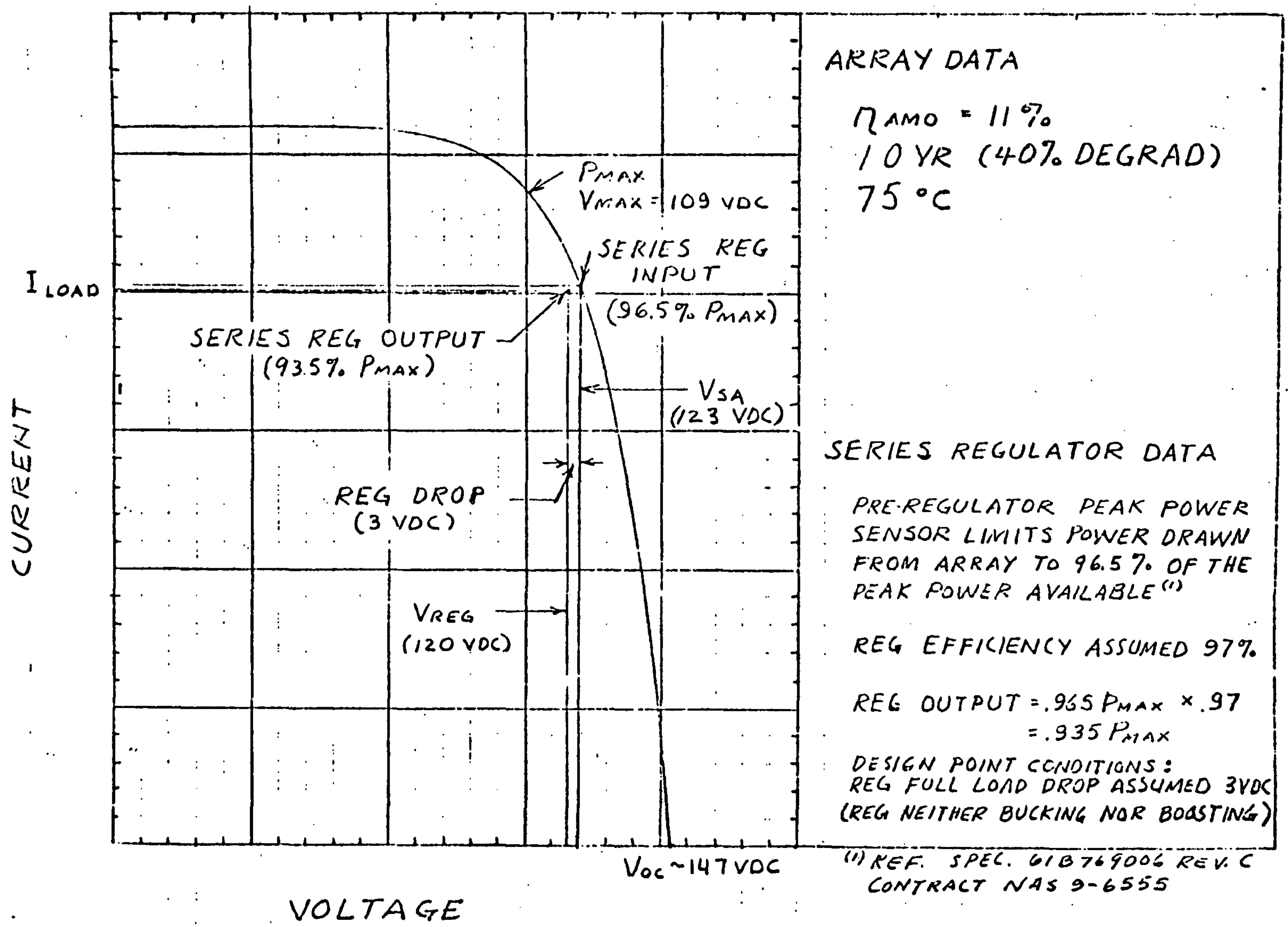


Figure 4.5-42 Solar Array/Voltage Regulator I-V Characteristics

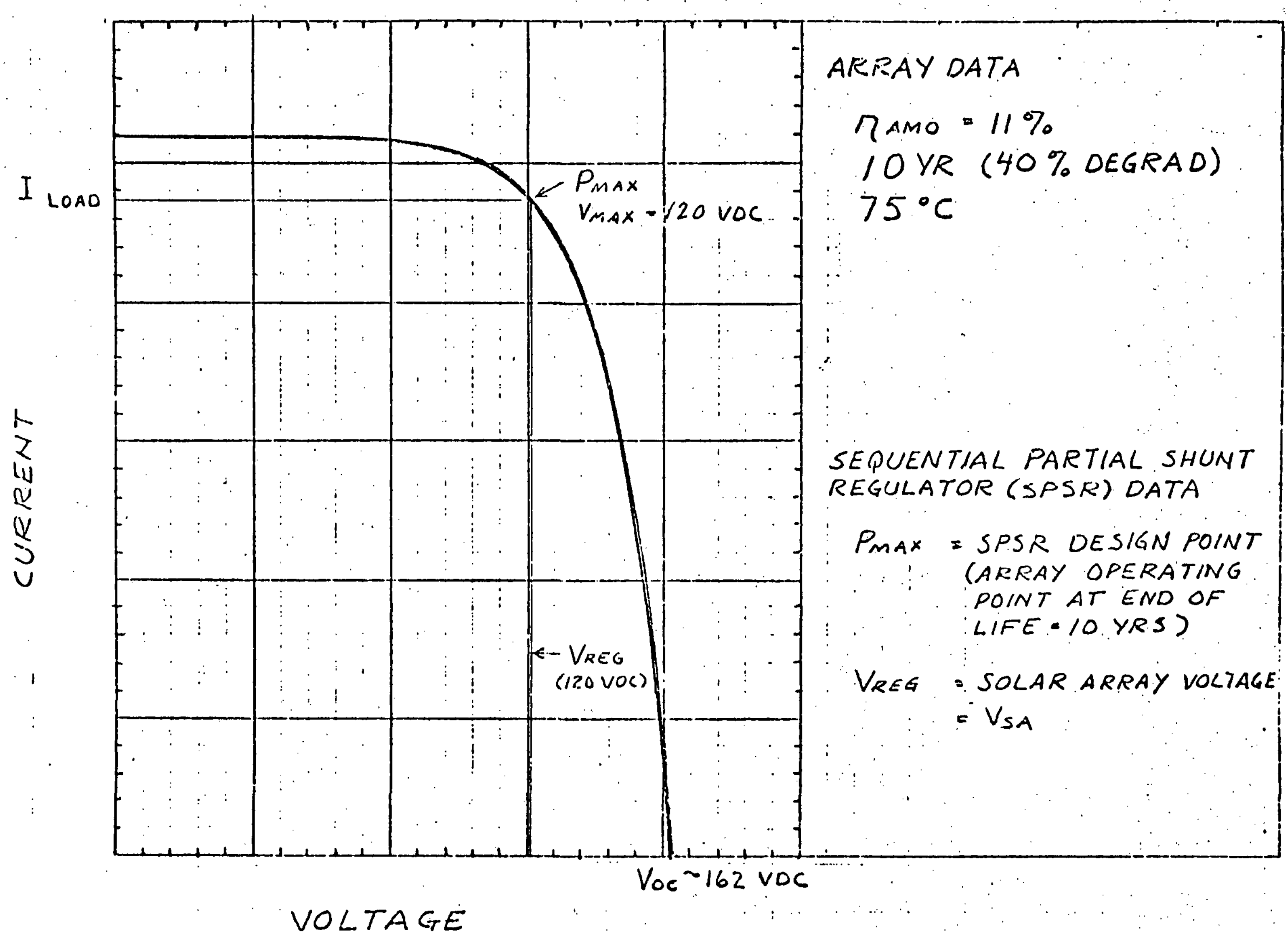


Figure 4.5-43 Solar Array/Voltage Regulator I-V Characteristics

Table 4. 5-30

COMPARISON OF SHUNT AND SERIES REGULATION METHODS FOR A FIXED SOLAR ARRAY AREA

Operating Condition	Sequential Partial Shunt Regulator (SPSR)		Series Regulator* (Buck-Boost)	
75° C - 10 yr (Design Point)				
Available regulated power**	100. 0	(100)	93. 5	(93. 5)
Array voltage - vdc	120. 0		123. 0	
95° C - 10 Yr				
Available regulated power	98. 5	(90. 0)	93. 5	(85. 3)
Array voltage - vdc	120. 0		110. 7	(Boost)
55° C - 10 Yr				
Available regulated power	96. 3	(104. 5)	93. 5	(101. 6)
Array voltage - vdc	120. 0		133. 0	(Buck)

*Series regulator output voltage is 120 ± 1 percent vdc, same as SPSR. Series regulator efficiency is assumed to be 97 percent. Peak power sensor limits power drawn from array to within 3. 5 percent of the peak power available.

**Percent of array maximum power (P_{max}) at the specified operating temperature. Enclosed numbers are percent of P_{max} at the design point temperature (75° C).

Similar analysis of solar array/voltage regulator characteristics at the 5-year point shows the SPSR system continues to yield higher power output at 75°C and 95°C (where power availability is most critical and 104.5 percent of design is still available), but yields lower power output at 55°C (where power availability is least critical). The higher power capabilities resulting from decreased degradation (30 percent degradation at 5 years versus 40 percent at 10 years) requires oversizing the series regulator for any temperature in the 55°C to 95°C range if loads in excess of the end-of-life design point capability are to be accommodated. The SPSR on the other hand is less sensitive to temperature and degradation, requiring oversizing only if array temperature is high and the array is relatively new.

End-of-Life Voltage Mismatch

From Figure 4.5-42, the SPSR end-of-life design point for voltage is 120 VDC. The operating point for this voltage corresponds to P_{max} on the I-V characteristic. Assume that the actual array voltage at end-of-life is 125 VDC and that P_{max} occurs here instead of at 120 VDC as planned. The constant voltage line V_{REG} will then be shifted slightly to the left of its position in Figure 4.5-42. The intersection of the new 120 VDC V_{REG} line with the I-V curve establishes the new end-of-life operating point for the SPSR. The resulting power capability is 98.8 percent of that at 125 VDC. This compares with a capability of 93.5 percent for the series regulator (series regulator capability is 93.5 percent of P_{max} for all operating conditions provided the input voltage limiter does not force the regulator to operate in the 3.5 percent peak power sensor buffer zone).

Excess Array Power (Long-Term Operation)

The ability to provide additional regulated power when excess power is available from the array can be an important consideration in evaluating regulation methods. Excess array power is greatest when the array is new. The availability of excess power decreases as array output degrades with time on orbit. Excess array power is also available when array operating temperature is low, as when emerging from an eclipse. This is a transient condition that lasts only until the array warms up to its stabilized operating temperature.

This section addresses the relative capabilities of the SPSR and series regulator to provide regulated power with respect to long-term array degradation. A comparison of regulator performance for the transient case is given in the next section.

Table 4.5-31 summarizes operating characteristics of the two systems as a function of years on orbit for a stabilized array temperature of 75°C. The significance of the asterisked footnote is that it points up an apparent conflict between requirements for power tracking and voltage limiting. If the series regulator input voltage is limited to 150 vdc as previously assumed, the array will be forced to operate within approximately 0.5 percent of P_{max}, well into the 3.5 percent buffer zone specified for tracker operation.

Power savings benefits with the SPSR are clearly in evidence at the 10-year design point and 5-year operating point shown in Table 4.5-31. When the array is new, the advantage lies with the series regulator. The break-even point where the two systems provide the same amount of regulated power is approximately 1 year. The relatively short-term advantage of the series

Table 4.5-31

	SPSR			Series Regulator		
	0 Yr	5 Yr	10 Yr	0 Yr	5 Yr	10 Yr
Available Regulated Power—Percent of P_{max} at 75°C	89.1	98.0	100.0	93.5	93.5	93.5
Array Voltage—VDC	120.0	120.0	120.0	153.5*	130.0	123.0

*Series regulator input voltage is required to limit power drawn from the array to 96.5 percent of P_{max}.

system is attributable to the highly exponential nature of array power degradation. Over one-third of the first 5 years degradation occurs in the first year; over one-fourth of the total 10-year degradation occurs in the first year.

Excess Array Power (Post-Eclipse Operation)

An analysis was made to determine the relative capabilities of the SPSR series regulator to make excess array power available to the loads during the warm-up period when the array emerges from an eclipse. Estimates of mean array temperature for nominal 2-minute increments during warm-up were made using cooling and heating rates for flexible solar panels from Lockheed Topical Report LMSC-A981486, dated December 1970, from NASA MSC, Contract NAS9-11039.

The I-V curves for these temperatures were constructed for a new array. For both regulating methods, the volt-ampere capacity available to the loads was determined at each mean operating temperature from analysis of the required operating points. The capacities were then integrated over the total warm-up period (11 minutes) and added to the capacity available during (a) the remainder of the illuminated period, and (b) the remainder of the orbital period, to establish figures of merit expressing the relative power capabilities of the two methods. The results show the SPSR is capable of supplying slightly more excess array power to the loads (equivalent to approximately 2 percent in terms of average power) than the series regulator.

Several simplifying assumptions were made to facilitate the analysis. These included disregarding power penalties resulting from buffer zone requirements for tracker operation. A more rigorous analysis taking into account specific requirements for tracker operation and potentially conflicting voltage limiting requirements would further increase the indicated advantage of the SPSR.

4.5.4.2.2 Voltage and Waveform Selection

The preceding comparisons of regulation schemes assumed a nominal 120-volt solar array. Higher voltages are favored from the standpoint of improved transmission efficiency, but are not as attractive for direct utilization by the loads. Conversely, lower voltages result in reduced transmission efficiency and increased weight, but offer the advantage of using well-established designs and practices developed for modern 28-volt systems. The use of a higher voltage for transmission and a lower voltage for distribution and utilization is an attempt to combine the best features of both systems. Each of these approaches is represented on Figure 4.5-44, which shows block diagrams of typical dc configurations examined in the system voltage and configuration study.

An alternate approach to the dc configurations given in Figure 4.5-44 is to invert the output of the solar arrays for transmission and distribution to the loads as 3-phase ac power. This scheme is characterized by a low source-to-load efficiency at the standard utilization voltage of 115/200 vac, 3-phase, 400 hertz. It is most compatible with systems having high ac/dc load ratios. The ac/dc ratio for the Modular Space Station is less than 0.25. Analysis shows that even with significantly higher ratios, 400 hertz ac transmission and distribution systems must be rejected on the basis of excessive dissipation and solar array area penalties.

Table 4.5-32 presents a qualitative comparison of 28 vdc and 115 vdc system voltages. A quantitative comparison is provided by the power flow diagrams on Figures 4.5-45 and 4.5-46 for trade configurations A and D, respectively, on Figure 4.5-44. As indicated by the required power output at the solar arrays, a power penalty of 42,114—38,610 = 3,504 watts is incurred by the 28-volt system in supplying ISS loads. A weight penalty of 3,076 pounds for transmission, conditioning and distribution is also incurred by the 28-volt system.

Detailed power flow analyses were conducted for the 115-volt configuration and the 28-volt configuration, considering both average power for loss analysis and peak power for sizing EPS hardware.

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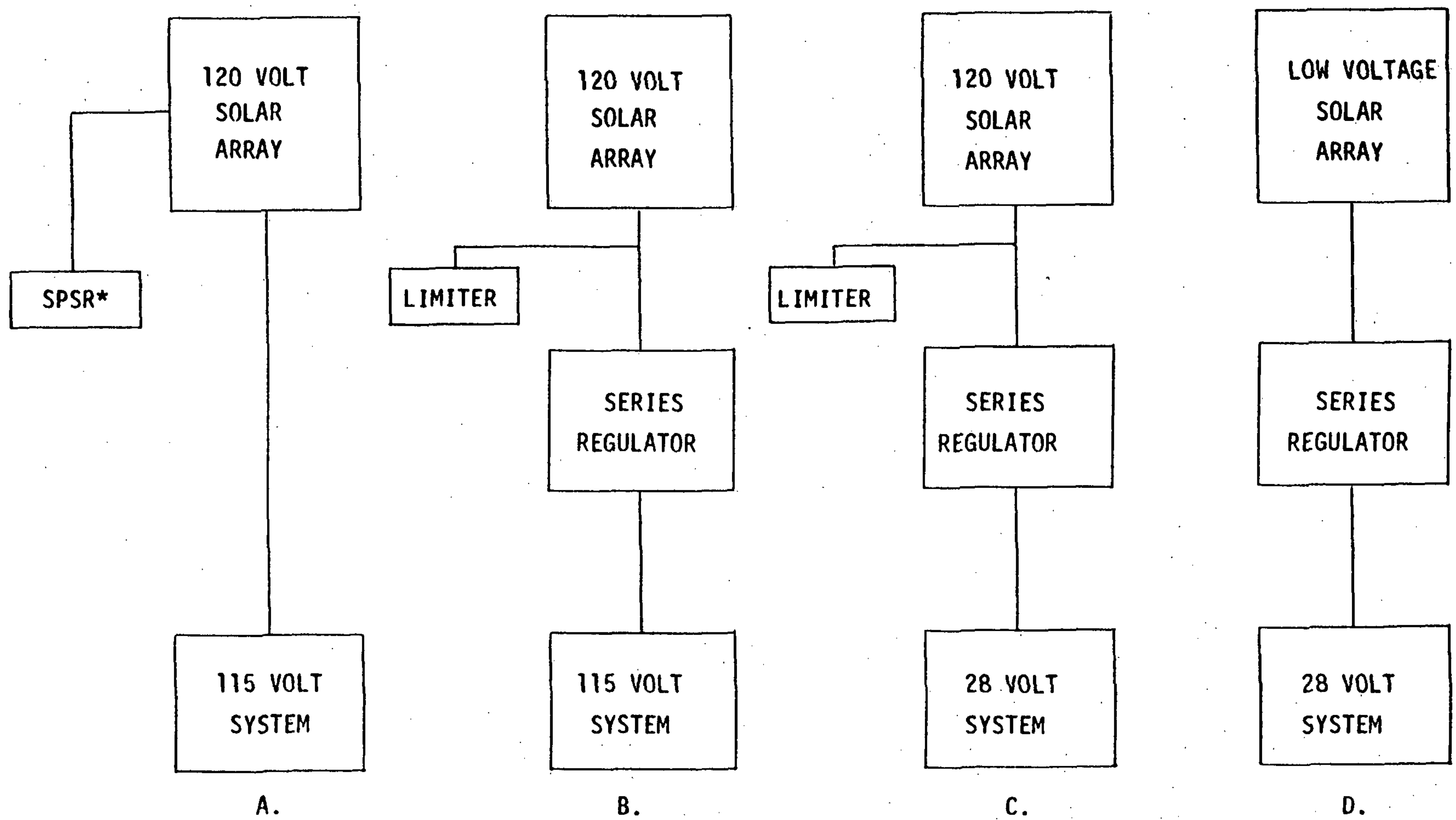


* SPSR-SEQUENTIAL PARTIAL SHUNT REGULATOR

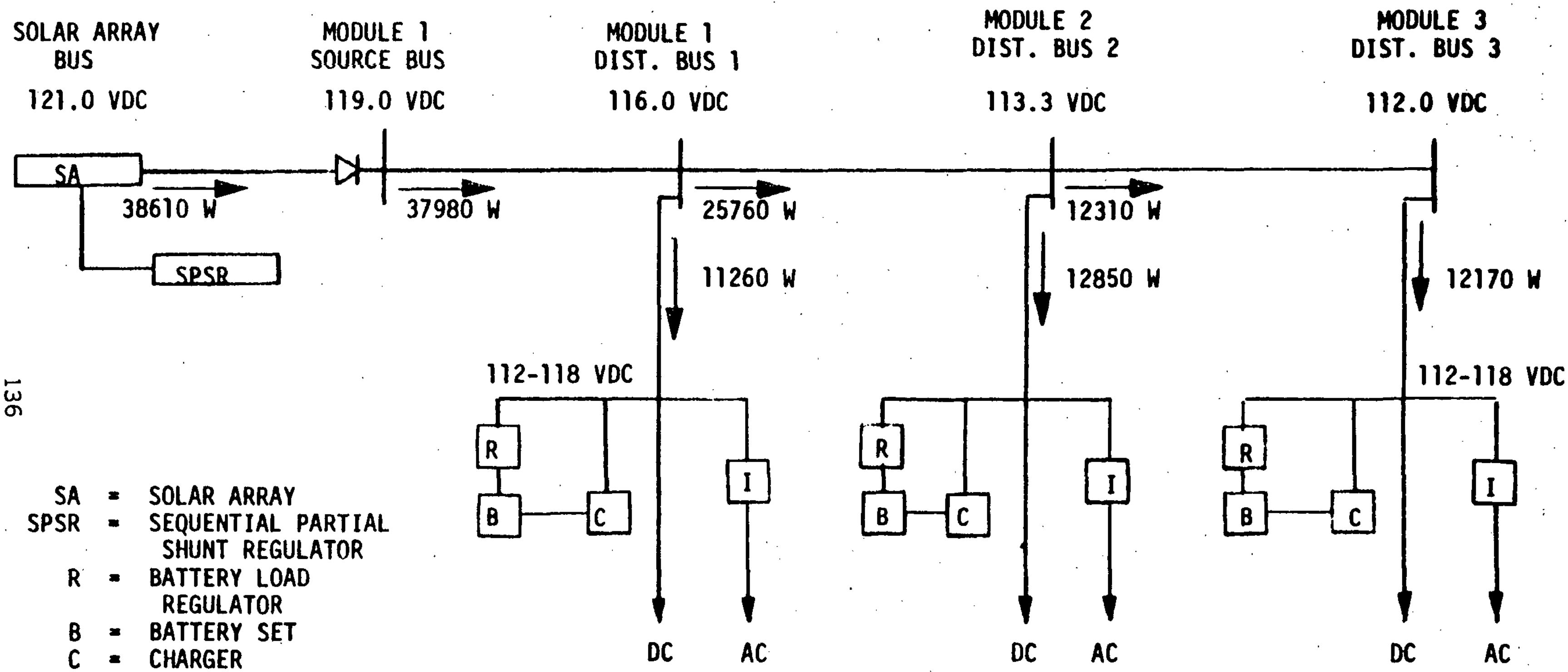
Figure 4.5-44 System Voltage and Configuration Trade Study

Table 4.5-32
SYSTEM VOLTAGE COMPARISON

	28 VDC	115 VDC
Cost	-	+
Commonality	+	+ (Shuttle)
Program Flexibility	-	+
Electrical Efficiency	-	+
Voltage Regulation	-	+
Weight	-	+
Volume	-	+ (Ratings)
Growth Potential	-	+
Development Risk	+	-
Operational Simplicity	+ (No SPSR)	-
Component Availability (1979)	+	±

An alternate 28-volt system using the Orbital Workshop (OWS) charger-battery-regulator configuration was also analyzed. This scheme eliminates the separate regulator for supplying array power to the loads. However, the overall conditioning losses increase somewhat due to the flow of load power through the charger and load regulator in series. The weight of the conditioning equipment also increases slightly, but this is discounted as not being significant within the accuracy of the analysis.

Power and weight penalties for the alternate 115-volt and 28-volt systems shown on Figure 4.5-44 (configurations B and C, respectively) are less than those given above for configuration D of Figure 4.5-44, when compared with configuration A. Minimum penalties are incurred by the 115-volt systems.



SA = SOLAR ARRAY
SPSR = SEQUENTIAL PARTIAL SHUNT REGULATOR
R = BATTERY LOAD REGULATOR
B = BATTERY SET
C = CHARGER
I = INVERTER

Figure 4.5-45 ISS Power Flow Diagram - 115 vdc System

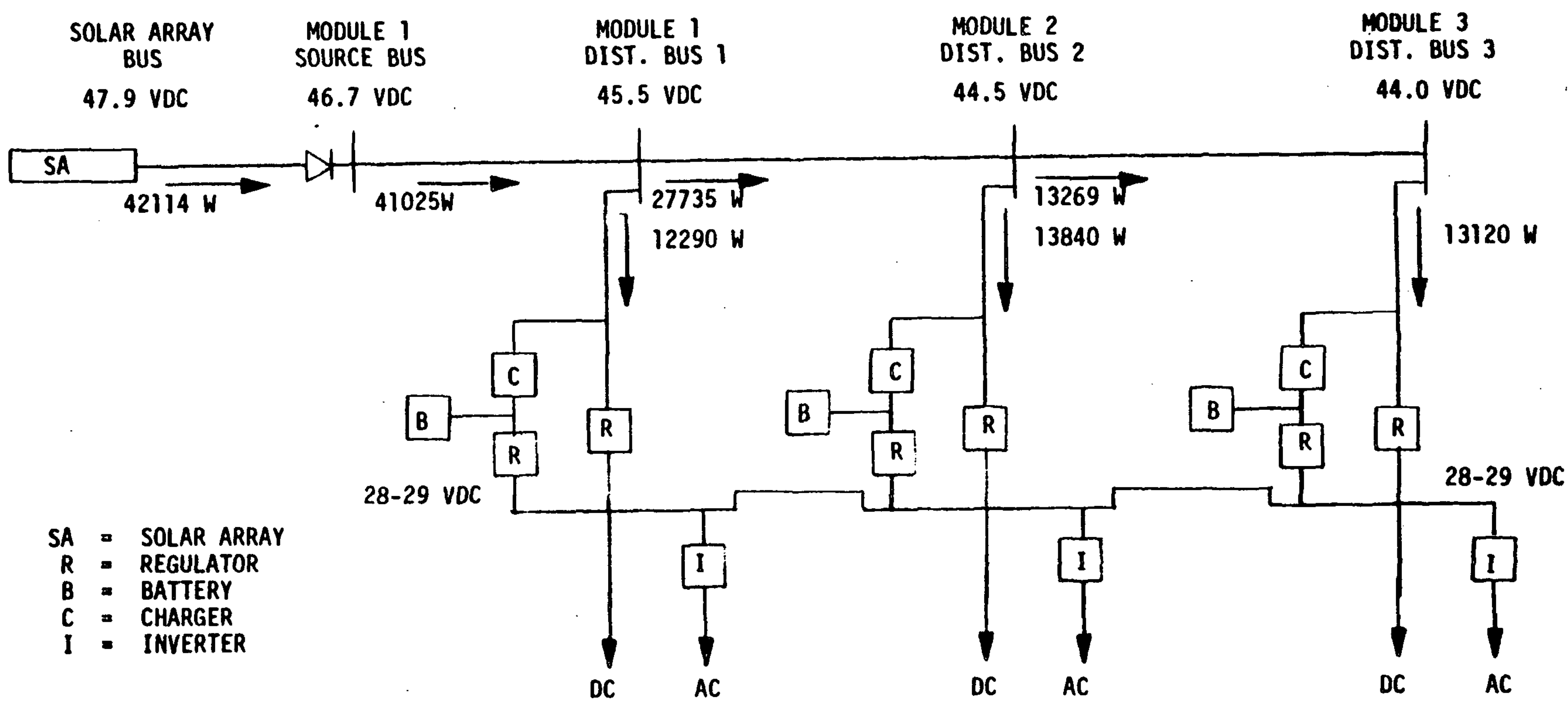


Figure 4.5-46 ISS Power Flow Diagram - 28 vdc System

Considering the qualitative comparison on Table 4.5-32, it is seen that 115 vdc is the preferred choice in those categories which can most readily be assessed on a quantitative basis. The areas which tend to favor 28 vdc are more difficult to evaluate. Development risk for example may be evaluated much differently by different evaluators. Considerable experience exists with component and equipment design for 28 vdc; however, much of this experience and 115 V commercial experience can be applied directly to 115 VDC applications.

The main impact of using 115 vdc for direct utilization is on avionics equipment where a change in dc/dc input converters is required. In the electromechanical area, solenoids (valves) must be modified to use 115-vdc coil motors. These changes, as well as requirements for higher voltage ratings in the area of power control, can be accomplished within the present state-of-the-art.

Component availability may be limited in some cases due to a limited demand for high-voltage applications in the past. There is a growing tendency, however, to turn to higher voltage dc for proposed high power systems. It is not unreasonable to assume, therefore, that component availability for 115-vdc applications will be improved within the time frame of Modular Space Station development. This in turn will contribute to minimizing development risk.

In view of the decided advantages of reduced weight and losses offered by higher voltage dc, and a generally favorable qualitative comparison of moderately high voltage dc relative to 28 vdc, a value of 115 vdc is selected as the nominal system voltage for the Modular Space Station. Twenty-eight vdc power will be provided only on a restrictive and local basis where impact analysis clearly precludes use of 115 vdc.

4.5.4.2.3 Control Logic and Control Modes

Solar Array Regulation/Control

The solar array consists of two wings, each controlled by its own sequential partial shunt regulator. The array wings normally operate in parallel but may be operated isolated if required. The control scheme for regulating solar array voltage is indicated in block diagram form on Figure 4.5-47 for the ISS configuration. The system is basically capable of regulating solar array voltage to within 1 percent of the design point voltage.

In the normal or parallel mode, Wing 1 voltage equals Wing 2 voltage and each load bus receives power from both wings. Control switching (indicated by the dashed lines on Figure 4.5-47) is provided to permit use of common references and other low-level components when operating in this mode. The control switching scheme also provides the capability for completely independent control of each array wing if the system is split into two parts. It further offers a means of implementing certain kinds of redundancy; however, redundancy requirements and selection of redundancy techniques are logically tasks for a more detailed design/development effort.

The solar array wing/load bus discriminator matrix shown on the right of Figure 4.5-47 identifies the source for each load bus, either Wing 1 or Wing 2 or both. Information in the matrix is maintained current by monitoring the position of selected bus and feeder switches. Identification of the power source for each load bus is readily determined by the open/close status of switches in the matrix. The discriminator routes the sampled load bus voltages to the "voltage select" and "average voltage" processor units for Wing 1 or 2, or both, as required.

As shown on Figure 4.5-47, solar array wing voltages are also routed to "voltage select." This unit selects either the wing voltage or load bus voltage for further processing depending on whether the array is operating at or near its stabilized temperature or is undergoing a thermal transient after emerging from an eclipse.

Under stabilized temperature conditions the unit passes only load bus voltages (V_{LB}). Each discrete load bus voltage is compared with a voltage reference to determine if it is in or out of predetermined limits. If the voltage value is within limits, the processor turns to the "average voltage" function. This function averages the load bus voltages for comparison with a reference equal to the desired nominal load bus voltage (115.0 vdc). The

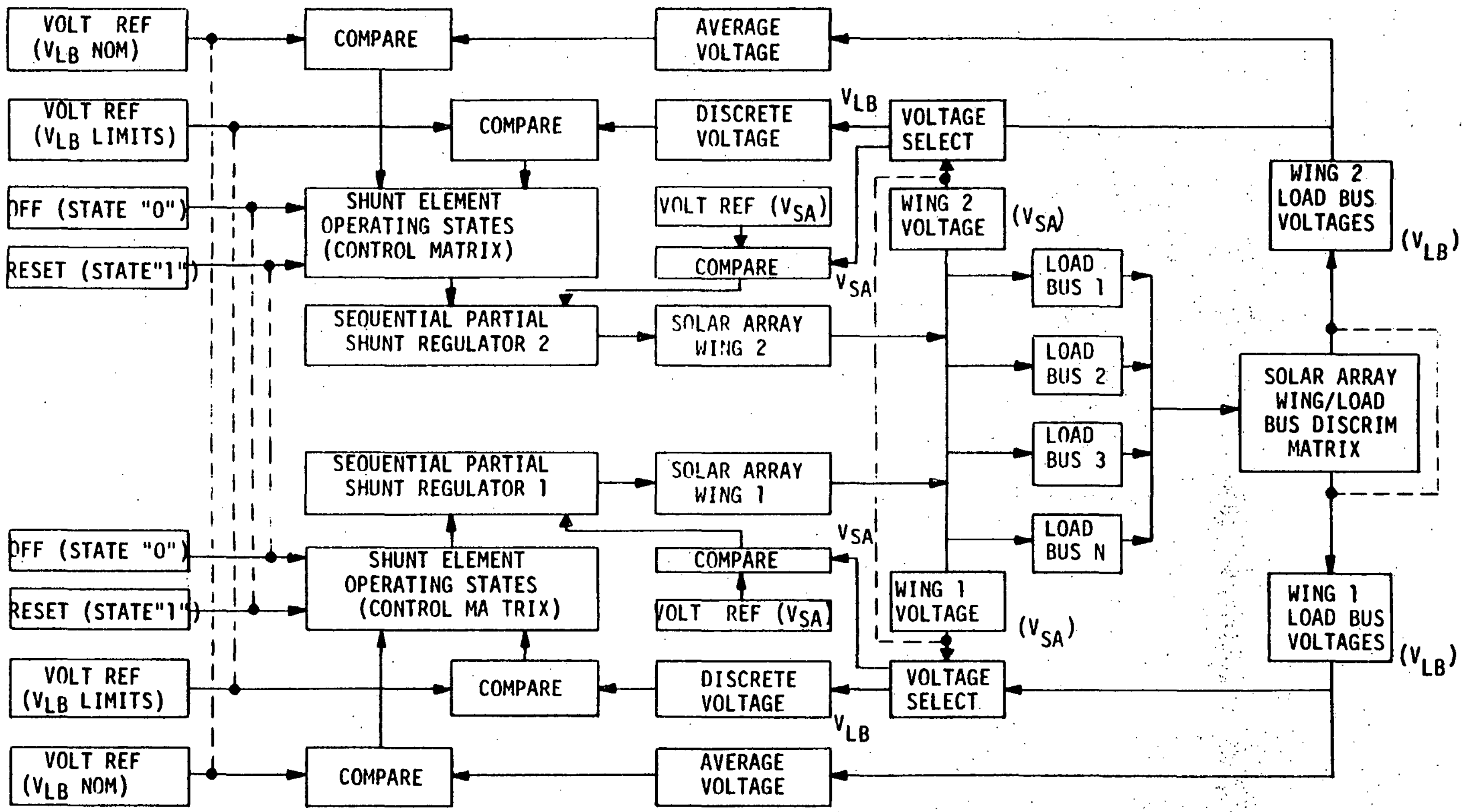


Figure 4.5-47 Solar Array Regulation/Control - ISS Configuration

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Table 4.5-33

SPSPR CONTROL MATRIX

			HIGH SHUNT ELEMENTS												LOW SHUNT ELEMENTS											
			ACTIVE	SWITCHED																						
REGULATOR STATE	OPERATING CONDITION	SHUNT STATES	1AH	1BH	2AH	2BH	3AH	3BH	4AH	4BH	5AH	5BH	6AH	6BH	1AL	1BL	2AL	2BL	3AL	3BL	4AL	4BL	5AL	5BL	6AL	6BL
"0"	ECLIPSE	ALL SHUNTS OFF	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O
"1"	SENSE SUN	LOW SHUNTS SATURATED	O	O	O	O	O	O	O	O	O	O	O	O	S	S	S	S	S	S	S	S	S	S	S	S
	$V_{SA} > 120$ VDC	INITIATE HIGH SHUNT OPERATIONS																								
2	ACTIVE SHUNT ON	SWITCHED SHUNTS SATURATED	L	S	S	S	S	S	S	S	S	S	S	S	S	S	S	S	S	S	S	S	S	S	S	S
3	RECYCLE NO. 1	1BH OFF		O																						
4	2	1AH OFF			O																					
5	3	2BH OFF				O																				
6	4	3AH OFF					O																			
7	5	3BH OFF						O																		
8	6	4AH OFF							O																	
9	7	4BH OFF								O																
10	8	5AH OFF									O															
11	9	5BH OFF										O														
12	10	6AH OFF											O													
13	11	6BH OFF												O												
	$V_{SA} < 120$ VDC	INITIATE LOW SHUNT OPERATIONS																								
14	117.5 – 118.0	1AL LINEAR	O	O	O	O	O	O	O	O	O	O	O	O	L	S	S	S	S	S	S	S	S	S	S	S
15	117.0 – 117.5	1BL LINEAR													O	L										
16	1165 – 117.0	2AL LINEAR														O	L									
17	116.0 – 116.5	2BL LINEAR															O	L								
18	115.5 – 116.0	3AL LINEAR																O	L							
19	115.0 – 115.5 V_{LB} (AVG)	3BL LINEAR																	O	L						
20	114.5 — 115.0	4AL LINEAR																		O	L					
21	114.0 – 114.5	4BL LINEAR																			O	L				
22	113.6 – 114.0	5AL LINEAR																				O	L			
23	113.0 – 113.5	5BL LINEAR																					O	L		
24	112.5 – 113.0	6AL LINEAR																						O	L	
25	112.0 – 112.5	6BL LINEAR																							O	L
26	$V_{LB} < V_{LIM}$	"RAISE" BIAS	SHIFTS REGULATOR STATES 14-24 BY PLUS ONE																							
27	$V_{LB} > V_{LIM}$	"LOWER" BIAS	SHIFTS REGULATOR STATES 15-25 BY MINUS ONE																							

O – SHUNT ELEMENT OFF (NON CONDUCTING)

L – SHUNT ELEMENT IN LINEAR MODE (SHUNT ELEMENT IS REGULATING)

S – SHUNT ELEMENT SATURATED (OPERATING AT SATURATION VOLTAGE) error signal voltage generated by this comparison serves as an input to the unit identified as "shunt element operating states." This unit contains a control matrix which commands the specific regulator states shown on Table 4.5-33, depending on input conditions.

A detailed operational analysis was made of the sequential partial shunt voltage regulation system to verify adequate regulation during all operating conditions. Figure 4.5-48 shows a simplified schematic of the sequential partial shunt regulator. The complete assembly (excluding controls) consists of 12 regulating circuits for each wing, one for each of the half-panels, as shown previously in Figure 4.5-11. Each regulating circuit contains a "low shunt" element and a "high shunt" element; each of which shunts half of each half-panel in a fully saturated ON state, a linear (proportional) state, or an OFF state. Thus, the term "partial shunt" is used. Since only one of the 12 circuits is in a linear state at one time and this linear control acts in sequence with the fully-saturated modes, the term "sequential partial shunt regulator" is derived. The "high shunt" elements are used only during the relatively short time the array requires to reach stable operating temperature following each eclipse. The "low shunt" elements are used for continual regulation. It is noted that the circuit breakers on each regulated circuit (half panel) also can be used to transfer the panel output to either source bus or to disconnect the circuit entirely.

4.5.4.2.4 Regulation/Control Installations and Thermal Control

The sequential partial shunt regulators are installed in the turret section of the Power/Subsystems Module. The shunt elements are conductively cooled by the station fluid cooling system. The temperature of the fluid coolant will be between 65°F (minimum inlet temperature) and 80°F (maximum outlet temperature). Design for heat removal from the shunt elements will be based on 90 percent minimum conducted and 10 percent maximum radiated heat transfer, with emphasis on increasing the conducted capability.

Maximum heat rejection from each low-shunt system is approximately 910 watts. This is for a new array wing operating essentially at no load. The regulating element dissipates 360 watts. Each of the remaining low shunts dissipates approximately 50 watts (saturation loss). For a minimum

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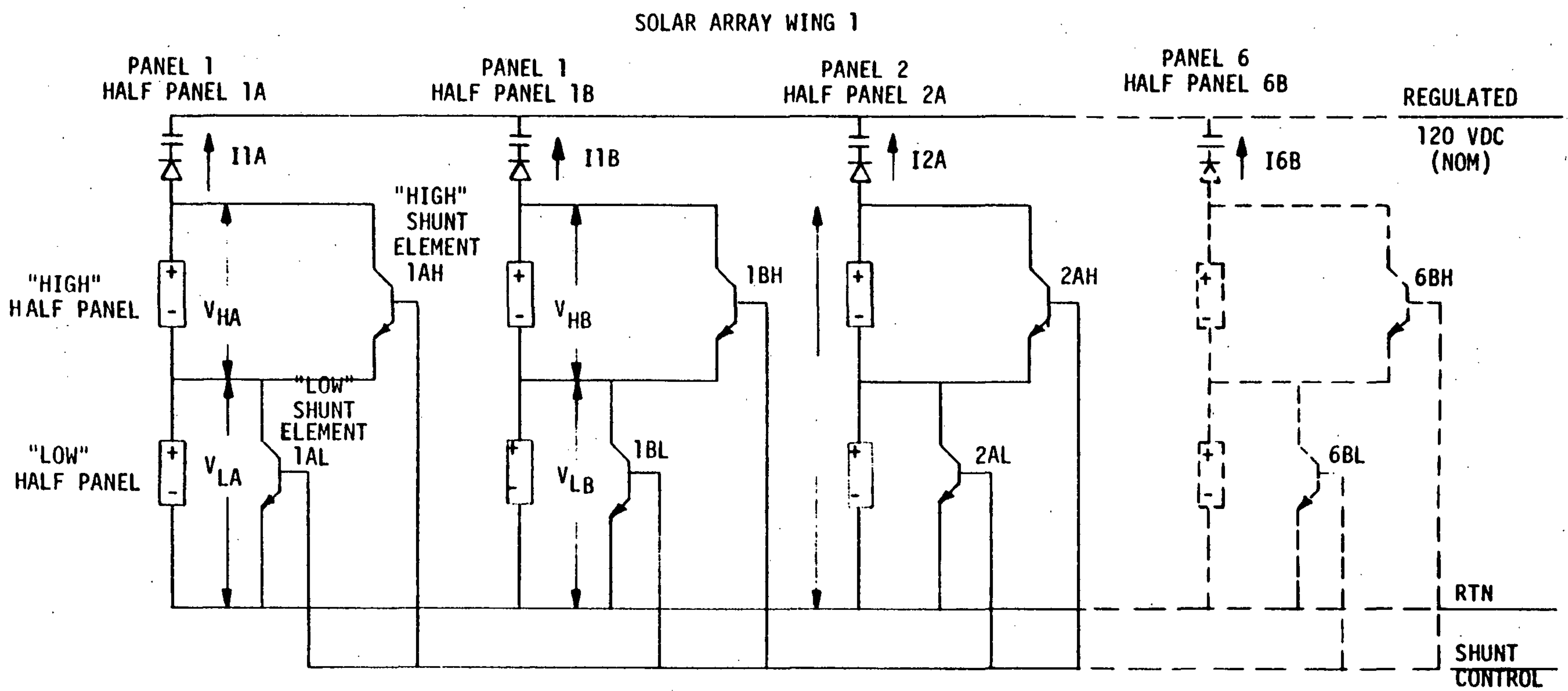

6 PANELS/WING
2 HALF PANELS/PANEL
1 REG CCT/HALF PANEL
2 REG CCTS/PANEL
12 REG CCTS/REG ASSY
1 REG ASSY/WING
2 REG ASSYS/ARRAY

Figure 4.5-48 Sequential Partial Shunt Regulator - Simplified Schematic

loadon orbit equal to 80 percent of the average design load, the total dissipation is approximately 660 watts for each SPSR or 1,320 watts for the two SPSR's required for the ISS array.

Maximum heat rejection from the SPSR occurs during the thermal transient condition generated as the array exits from eclipse. Peak dissipation in each active high-shunt element (one per SPSR) is approximately 1,920 watts. For the minimum load condition given above, total calculated dissipation from the active high-shunt element, switched high-shunt elements, and (saturated) low-shunt elements is 2,800 watts. Peak transient dissipation for the two SPSR's is then 5,600 watts.

The main memory units required by the SPSR control scheme are assigned to the DMS subsystem computer. Control logic and other control processing functions for the SPSR's are installed in the power control unit (PCU). The PCU is located in the main section of the Power/Subsystems Module. This unit is suitable for packaging/cooling in accordance with the standard designs proposed for the replaceable logic modules (see Section 4.12). This is in sharp contrast to the installation of the SPSR shunt elements where the high dissipation environment dictates more direct cooling provisions.

4.5.4.3 Energy Storage Assembly Analysis

Energy Storage Assembly Requirements

The energy storage system is made up of 100-ampere-hour nickel-cadmium batteries, each composed of 28 cells connected in series. Four batteries are connected in a series set to provide a nominal voltage of 115 VDC when discharged in series to the DC load buses and power conditioning subassemblies. The cells are grouped in economical replacement modules of four cells each, and each module is capable of temperature control. The batteries are cold-plated to an active coolant loop for thermal control of the batteries between 0° and 20°C for optimum battery performance. Each battery is about 8 inches high, 7.5 in. wide, and 42 in. long, and weighs 380 lbs with connectors, cases, and racks.

Eight batteries or two battery sets are assigned to each of the three ISS or five GSS modules; however, only one set is required at initial launch with the second set supplied in a subsequent launch, either prior to or concurrent with ISS manning. The batteries are located in accessible areas of the pressurized volumes of each of the Space Station modules and are designed with module containment to isolate potential effluent potassium hydroxide (KOH) from the Space Station atmosphere. Access is required for periodic maintenance and module replacement.

Each battery is recharged from the 115-VDC main distributor bus through its own charger/regulator. A modified constant-potential charger will be used. Individual cell monitoring provides a cell voltage limit signal to terminate battery charging. A backup signal is also received from a third electrode near the end of charge. A recombination (fourth) electrode will prevent premature termination of charge because of residual oxygen pressure within the cell. The nominal battery depth of discharge was selected as 15 percent as the result of the trade studies reported below. The battery characteristics are shown in Table 4.5-34.

Battery Weight and Cost Optimization

The number of batteries required for either a 15 percent (2.5 year life) or a 30 percent (1.0 year life) depth of discharge, based upon current NiCd battery data, can be obtained from Figure 4.5-49 as a function of average power. The batteries nominally provide 28 volts DC and the electrical power system bus voltage was tentatively selected at 115 VDC; therefore, the total number of batteries must be in multiples of four. An increase in the number of batteries results in a smaller depth of discharge, a longer life, and possibly fewer resupplies. A decrease in the number of batteries results in larger depths of discharge, shorter life, and increased resupplies. Using a range of batteries from 8 to 40, in multiples of four, sized for a nominal ISS power of 17.3 kw average, a weight and cost optimization study was conducted to select the number of batteries and depth of discharge for the ISS. The quantity was increased according to Figure 4.5-49 for the GSS also.

Table 4.5-34

BATTERY CHARACTERISTICS

Battery Type	Nickel-Cadmium	
Case	Stainless Steel	
Ampere-Hour capacity	100	
Number of cells/battery	28	
Size of replaceable module	4 cells	
Design life	2.5 year (nominal); 1 year (minimum)	
Operating temperature	0° to 20°C (13°C design point)	
Depth of discharge	15 percent (nominal) 70 percent (maximum)	
Electrodes	Signal Electrode (3rd) 1 required/battery 1/module available Recombination electrode (4th) 1 required/cell	
Charge method	Modified constant-potential cell voltage limit to end charge	
Battery requirements	ISS	GSS
Number of batteries	24	40
Battery weight (lb)	9,120	15,200
Battery volume (ft^3)	36	60

Each quantity of batteries resulted in values for initial ISS battery weight and depth of discharge. The depth of discharge was then converted to battery life. Figure 4.5-50 shows the initial ISS weight for 17.3 kw average, plotted versus battery life. The resupply weight per year was calculated from the lifetime and is also shown in Figure 4.5-50. The resupply weight optimizes at about 2.5 years, which corresponds to 24 batteries and a 15-percent depth of discharge. The optimum resupply weight is associated with an increased initial battery weight as shown, which is necessary to extend the life to 2.5 years.

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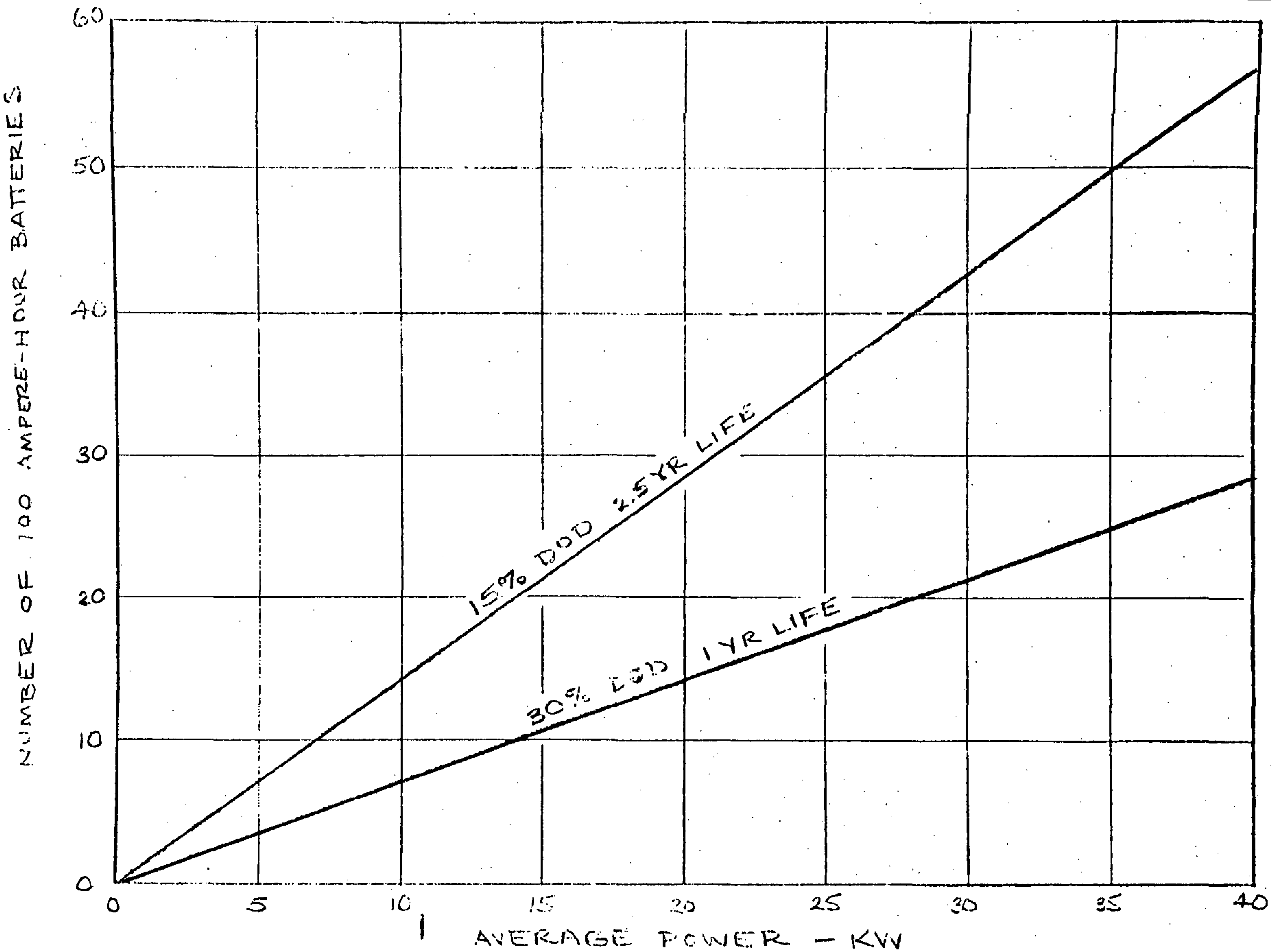


Figure 4.5-49 Battery Requirements

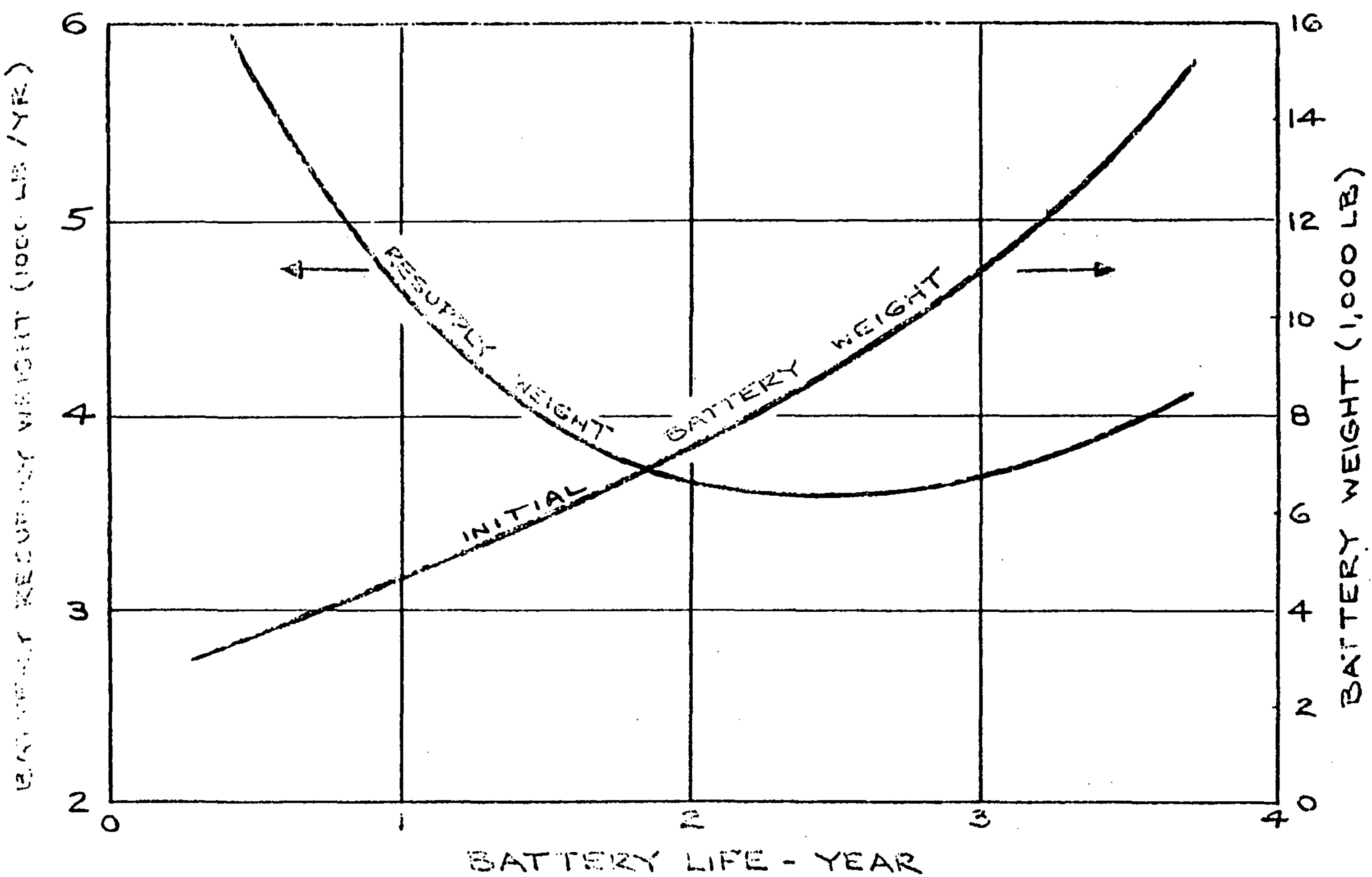


Figure 4.5-50 Battery Resupply Requirements Weight - 17.3 kw_{avg}

A battery cost optimization study was also conducted, and the results are shown in Figure 4.5-51. The costs include battery recurring cost, penalties for weight and volume for launch and resupply, and costs for crewtime required for maintenance and replacement. The cost factors and penalties used are as follows:

Battery recurring cost	\$50,000/battery
Weight (launch and resupply)	\$250/lb
Volume (launch and resupply)	\$1,500/ft^3
Crewtime	\$1,940/man-hour

The most significant cost factor was the weight; therefore, it would be expected that the cost would also optimize at a 2.5-year battery life, and this prediction is found to be correct.

Conclusion

A comparison of the weight and cost optimizations for a 5-year mission is shown in Figure 4.5-52. Both weight and cost optimize at about a 2.5-year battery life which results in the selection of 24 batteries and a 15-percent depth of discharge. A review of the study and conclusions for the 16.7 kw ISS and 30.8 kw GSS load analyses shows the selections to remain valid.

Battery Load Profile Analysis

A solar array/battery electrical power system can normally provide for peak load requirements because the solar arrays are designed to about 218 percent of the average load at the buses to account for battery charging, line losses and system inefficiencies, and because the batteries are designed for a nominal depth of discharge of about 15 percent with a maximum expected depth of discharge of 35 percent. With the development of a new power load profile, it was necessary to determine the effect of this profile on the battery capability.

The specific analysis was made for a 1-year life design, for which depths of discharge are 30 percent average and 70 percent maximum. The results are also applicable to the selected 2-1/2 years life reference design, for which depths of discharge are 15 percent average and 35 percent maximum.

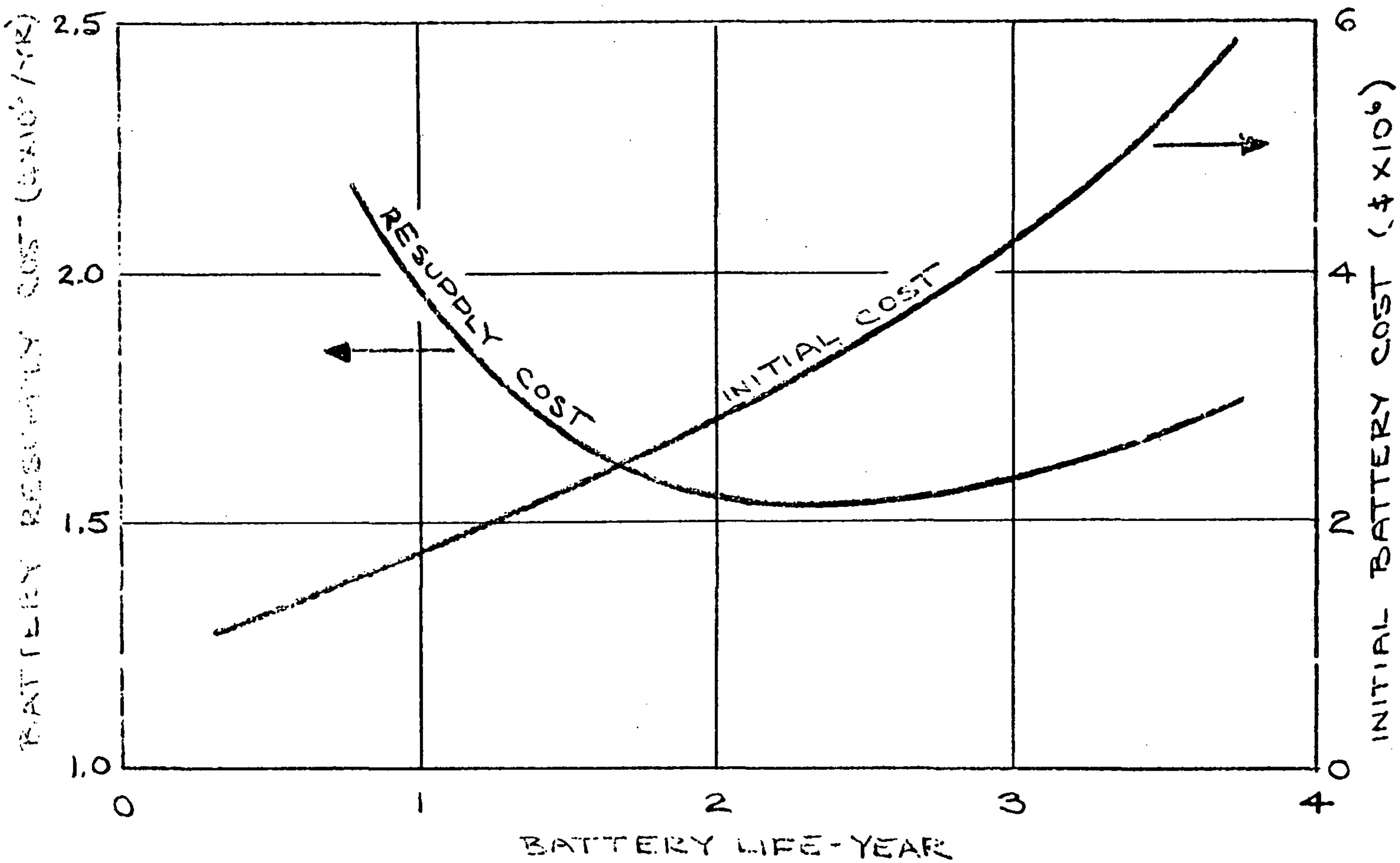


Figure 4.5-51 Battery Resupply Requirements Cost - Recurring Costs Only, 17.3 kw_{avg}

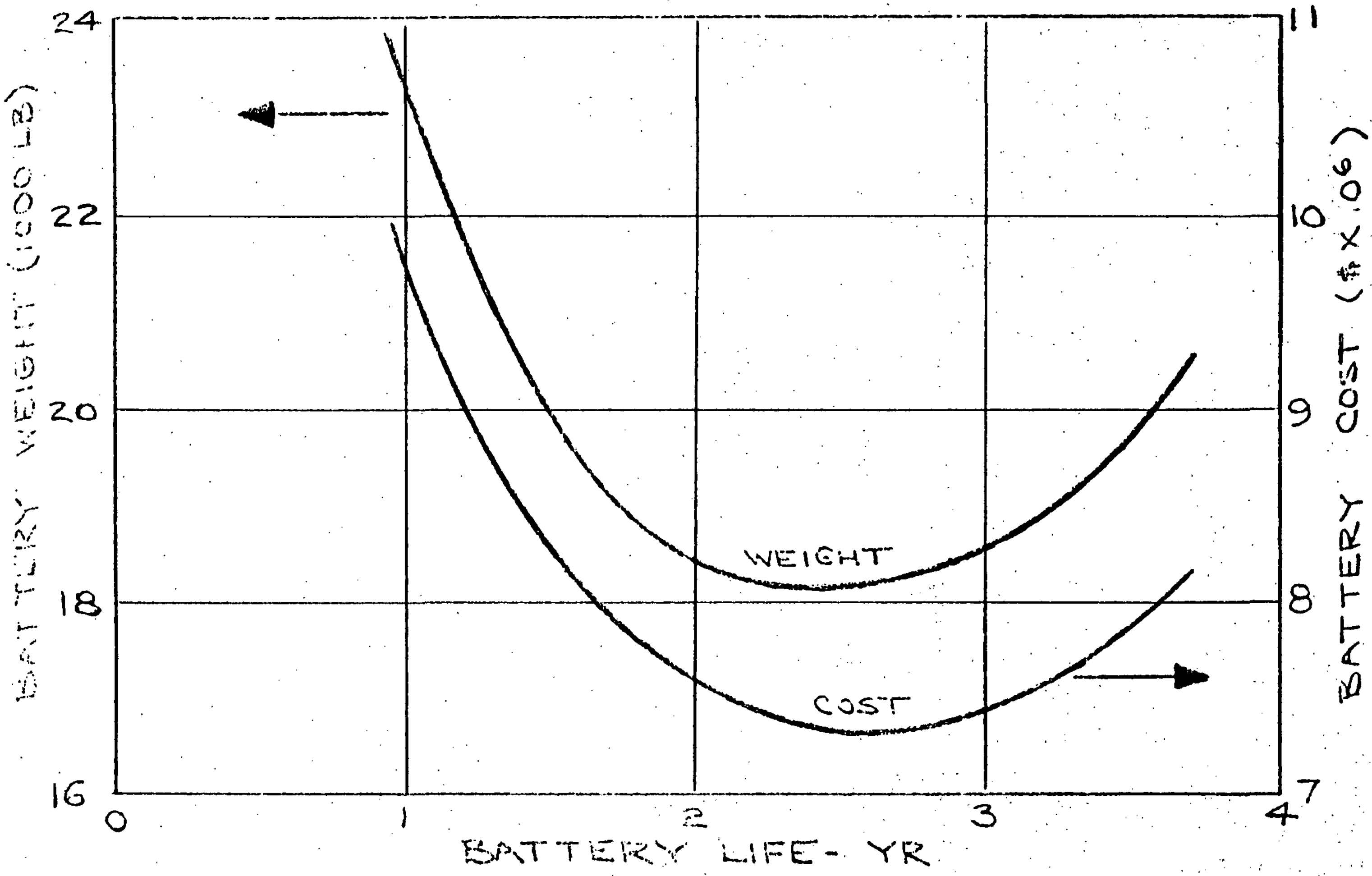


Figure 4.5-52 Battery 5-Year Weight and Cost - Recurring Cost Only, 17.3 kw_{avg}

The study consisted of a worst case analysis to determine effects of the 48-hour power load profile on the battery charge/discharge characteristics as indicated by state-of-charge profiles. The 48-hour load profile is the result of an earlier study and represents the 48-hour period of maximum activity, thus providing the most demanding period for the batteries.

The first case was a standard 92 minute orbit (56 minutes in sunlight, 36 minutes in eclipse) with the peak load during eclipse. Battery state-of-charge ranged from 30 percent minimum to 100 percent once in every 48 hours.

The second case allowed 60 minutes in sunlight and 32 minutes in eclipse (also 92 minutes/orbit) with peak loads in eclipse as before. The result of four minutes more charge and four minutes less discharge are dramatic — battery state-of-charge varied from 60-percent minimum to 100 percent in almost every cycle (92 percent or higher charge was achieved in every cycle).

The third case was derived from the first case (56 min/36 min) by shifting the peak load to the illuminated part of the orbit. This improved the state of charge to 40-percent minimum and 100-percent charge once in each 36 hours.

The fourth case corresponds to peak load scheduling, in that each peak load, the duration of which exceeded the illuminated period, was rescheduled to occur with maximum illuminated time. The result was to improve a little upon the third case, and resulted in a 39-percent minimum state-of-charge with 100-percent charge once in each 35-1/2 hours.

Conclusions

The battery design will satisfy the electrical power load profile requirements in the worst case conditions and can provide excess power during the other operating conditions investigated. A comparison of analyses conducted is shown in Figure 4.5-53 for the worst periods. The figure shows the worst case conditions for (1) 36-minute eclipse with eclipse time peaks, (2) average eclipse periods of 32 minutes, (3) 36 minutes eclipse with illuminated period peaks, and (4) modified profile for illuminated peaks at 36-minute eclipse

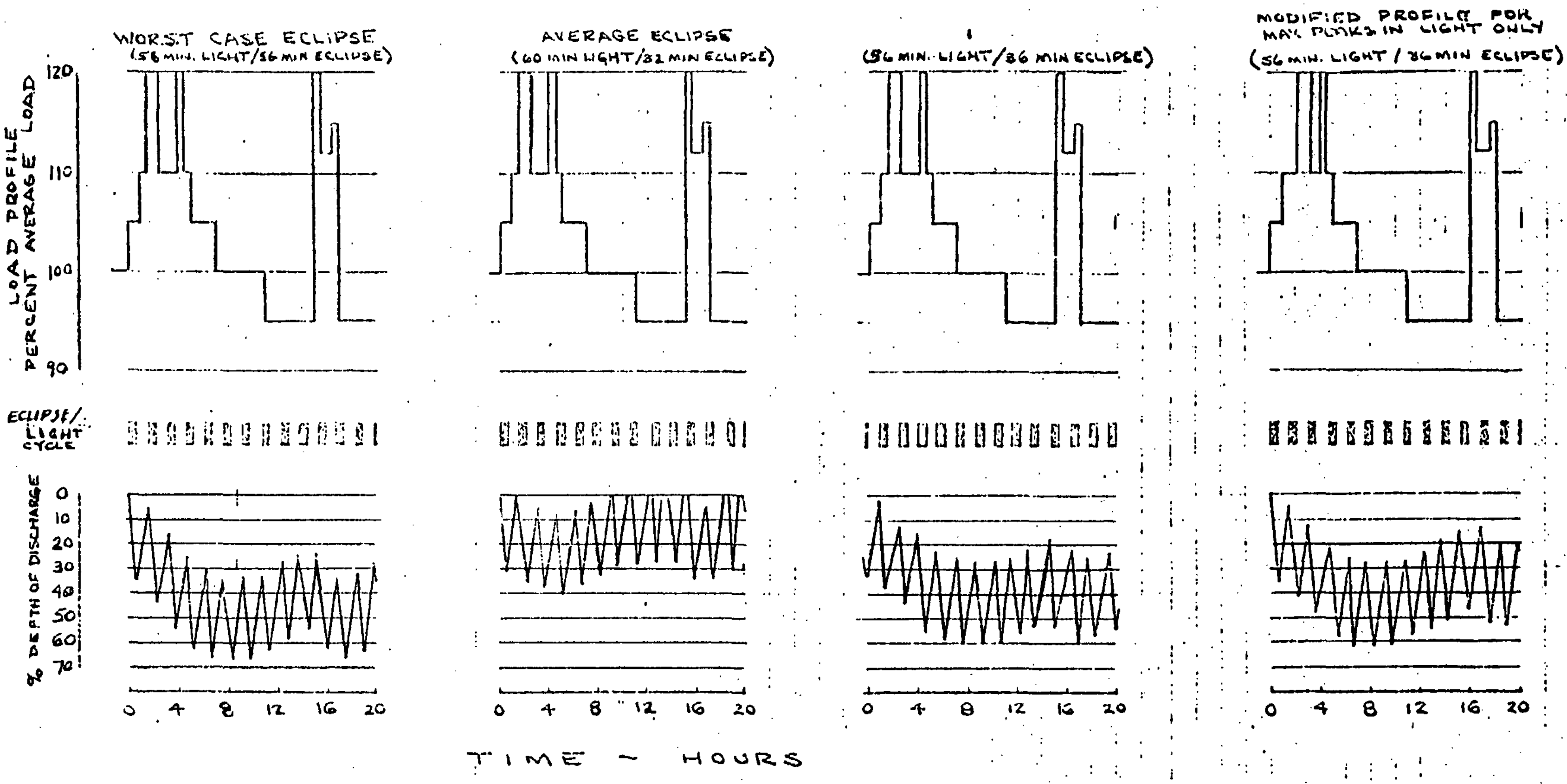


Figure 4.5-53 Battery Charge-Discharge Characteristics

periods. The effects on battery charge and discharge histories are evident except for the latter two cases, which are very similar.

In the worst case conditions, the batteries reach a low depth of discharge and therefore provide minimum emergency capability, reaching 60-70 percent depth of discharge. Selection of a shallow depth of discharge will increase emergency power capability and will increase battery life, but will also increase initial launch weight. A 15-percent nominal depth of discharge and a 35-percent maximum expected depth of discharge, as selected for the Modular Space Station, will approximately reduce each charge and discharge segment to one-half the range shown in Figure 4.5-53.

Battery Ampere-Hour Efficiency

Previous analyses and sizing of solar arrays and battery electrical power systems have used an ampere-hour efficiency for nickel-cadmium batteries of 80 percent and a range of voltage efficiencies based upon the end-of-life, end-of-charge and end-of-discharge voltages, resulting in energy efficiencies in the neighborhood of 50 to 60 percent. The solar array was correspondingly sized to provide about 2.5 times the average power required at the buses, to provide for battery charging, line losses, and system inefficiencies. An increase in battery efficiency, the lowest of all components, could result in a substantial savings of array area. Therefore, a survey and analysis of battery efficiency data was conducted to establish more realistic efficiency values — particularly for the ampere-hour efficiencies.

The nickel-cadmium battery charge efficiency data analyzed (References 4.5-8 through 4.5-11) indicated high ampere-hour efficiencies. Most of the data was for new batteries and for batteries with a depth of discharge of 100 percent. It was reported that the use of lower depths of discharge will sharpen the knee of the ampere-hour efficiency curve and will increase the overall efficiency. It was also reported that higher charge rates in the early stages of charge will also result in higher efficiencies. Typical ampere-hour efficiencies versus states of charge for nickel-cadmium batteries are shown in Figure 4.5-54 as a function of charge rate and for

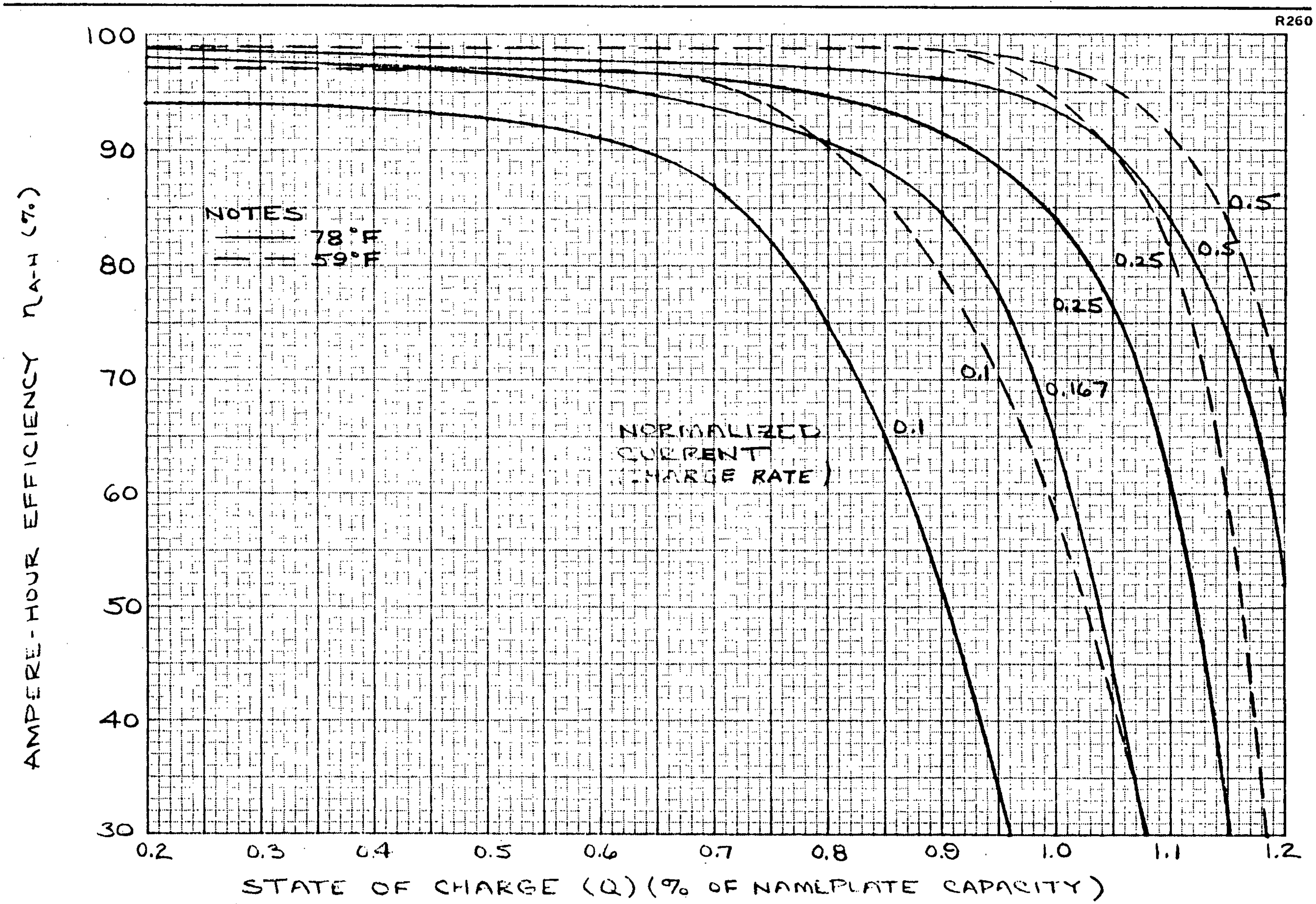
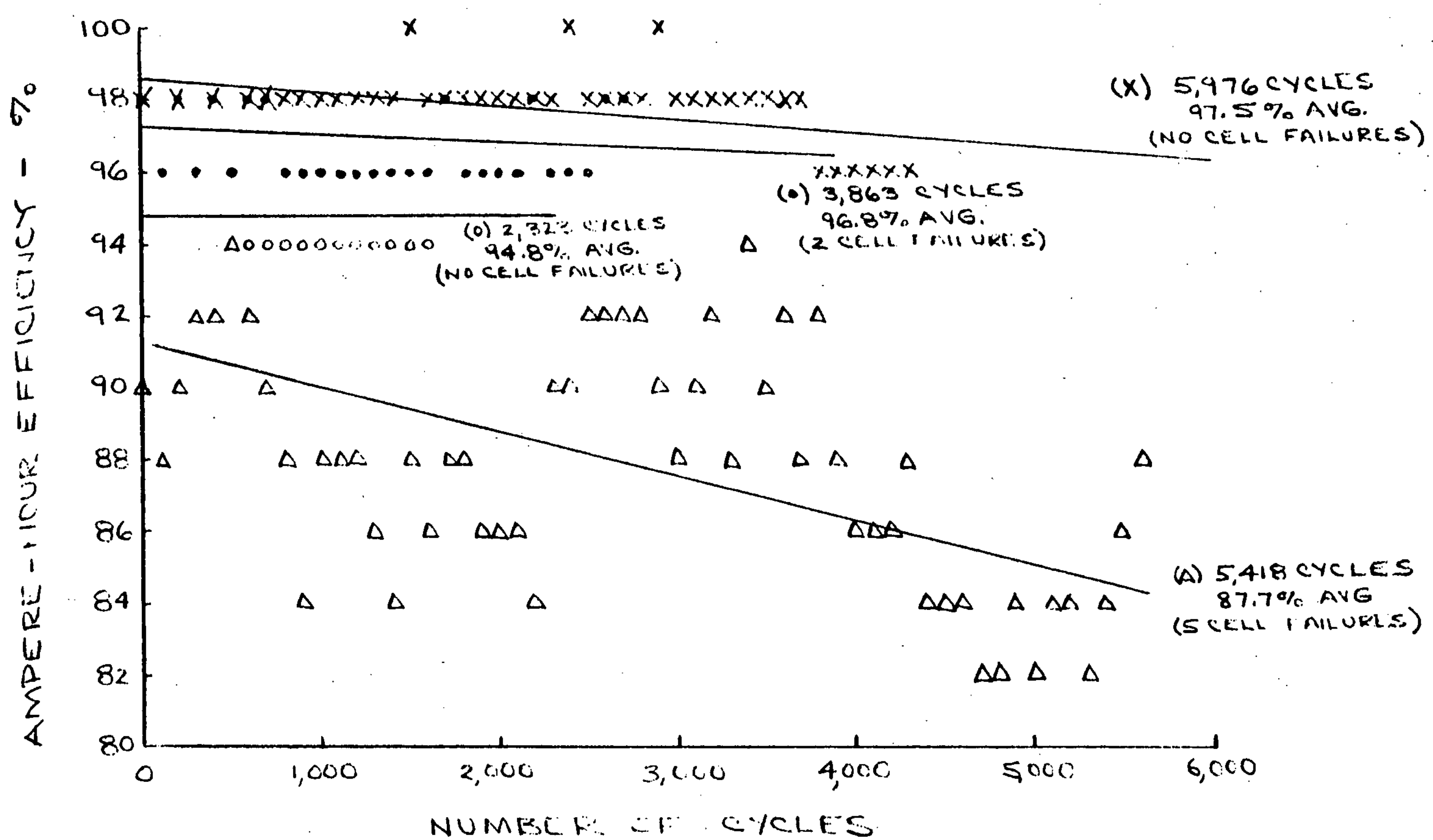


Figure 4.5-54 Ampere-Hour Efficiencies of Nickel-Cadmium Cells

two different temperatures (Reference 4.5-9). As shown, the ampere-hour efficiency is highest at initial charge and increases at both increased charge rates and decreased temperatures. The design charge rate for the Modular Space Station will be between the 0.25 and 0.5 charge rates, and closer to the 0.5 charge rate, with a temperature range of 0 to 20°C (32-68°).

A study by the Martin Marietta Corporation for the NASA Goddard Space Flight Center investigated the characteristics of a nickel-cadmium battery for several depths of discharge, charge rates, and charge methods from beginning-of-life to end-of-life (Reference 4.5-8). End-of-life was assumed when half of the cells had failed, however, some tests were terminated prior to failure. The analysis included the measurements of voltage, current, and efficiencies (ampere-hour and energy) throughout the cycle life of the battery. Typical results of the study for the ampere-hour efficiency as a function of cycle life is shown in Figure 4.5-55. The four representative cases shown were operated in a manner similar to the Space Station battery operating requirements. Two of the batteries (x and ∘) had initial ampere-hour efficiencies of about 98 percent and degraded to about 96 percent at the end-of-life, with averages of 97.5 percent and 96.8 percent. The data points on Figure 4.5-55 are from computer plots which plotted at 2 ampere-hour intervals. The third battery (o) kept a constant ampere-hour efficiency throughout its life of 94.8 percent. The representative lines are shown to indicate a trend in the ampere-hour efficiency, all of which gradually decrease with cycle life. The fourth battery (△) showed a randomly varying ampere-hour efficiency with a trend based upon beginning of life, end of life, and a given average of the ampere-hour efficiencies. Analysis of the data did not reveal the reason for the wide variations. The data also indicated a general increase in energy efficiency with increased cycle life. This is in conflict with other data, however, and therefore the fourth battery is discounted for this analysis. The batteries (o and △) also used a lower charge rate than for the Modular Space Station and would therefore also have a lower ampere-hour efficiency.

Cycle life tests at TRW Systems showed an ampere-hour efficiency of 95-96 percent for a new battery (References 4.5-9 and 4.5-10) when



DATA FROM REFERENCE 4.5-8

Figure 4.5-55 Nickel-Cadmium Battery Ampere-Hour Efficiency operated at the Modular Space Station requirements. Recent battery tests at MDAC-WD have also shown high nickel-cadmium battery ampere-hour efficiencies (see Figure 4.5-56). The MDAC data are also for new batteries and show an ampere-hour efficiency of 96-98 percent at the 0.5 or $^{C}/2$ charge rate.

Conclusions

Nickel-cadmium battery ampere-hour efficiencies are very high initially, and tests have shown a gradual decrease in efficiency with cycle life. The beginning-of-life efficiency is expected to be about 95-98 percent and is assumed to degrade to 94 percent at the end-of-life. A design value of 94 percent has been selected.

The energy efficiency will vary with the method of charging the battery and is expected to be in the neighborhood of 60 percent to 70 percent as compared with the 80 percent ampere-hour efficiencies used previously. These increases are reflected in decreased solar array area requirements.

Battery Temperature Effects

Battery performance is affected by the temperature at which it operates, and this temperature can be controlled. The temperature effect on the Modular Space Station batteries is shown in Figure 4.5-57. The curves are based upon the use of 24 100-ampere-hour nickel-cadmium batteries of 28 cells each at a depth of discharge of 15 percent. An increase of temperature from 20°C to 30°C results in a decrease in cycle life of about 5,000 cycles (almost 1 year in orbit), and an increase in resupply weight of over 100 lb/month. A decrease in temperature from 20°C to 10°C results in an increase in cycle life by almost 5,000 cycles, and a decrease in resupply weight by about 70 lb/month. The effects of discharge are not included but will decrease the discharge efficiency at lower temperatures. The optimum battery temperature is between 10 and 15°C. Because this close tolerance would be difficult to maintain, a range between 10°C and 20°C is selected for best battery performance. A design point temperature of 13°C represents the preferred cell temperature.

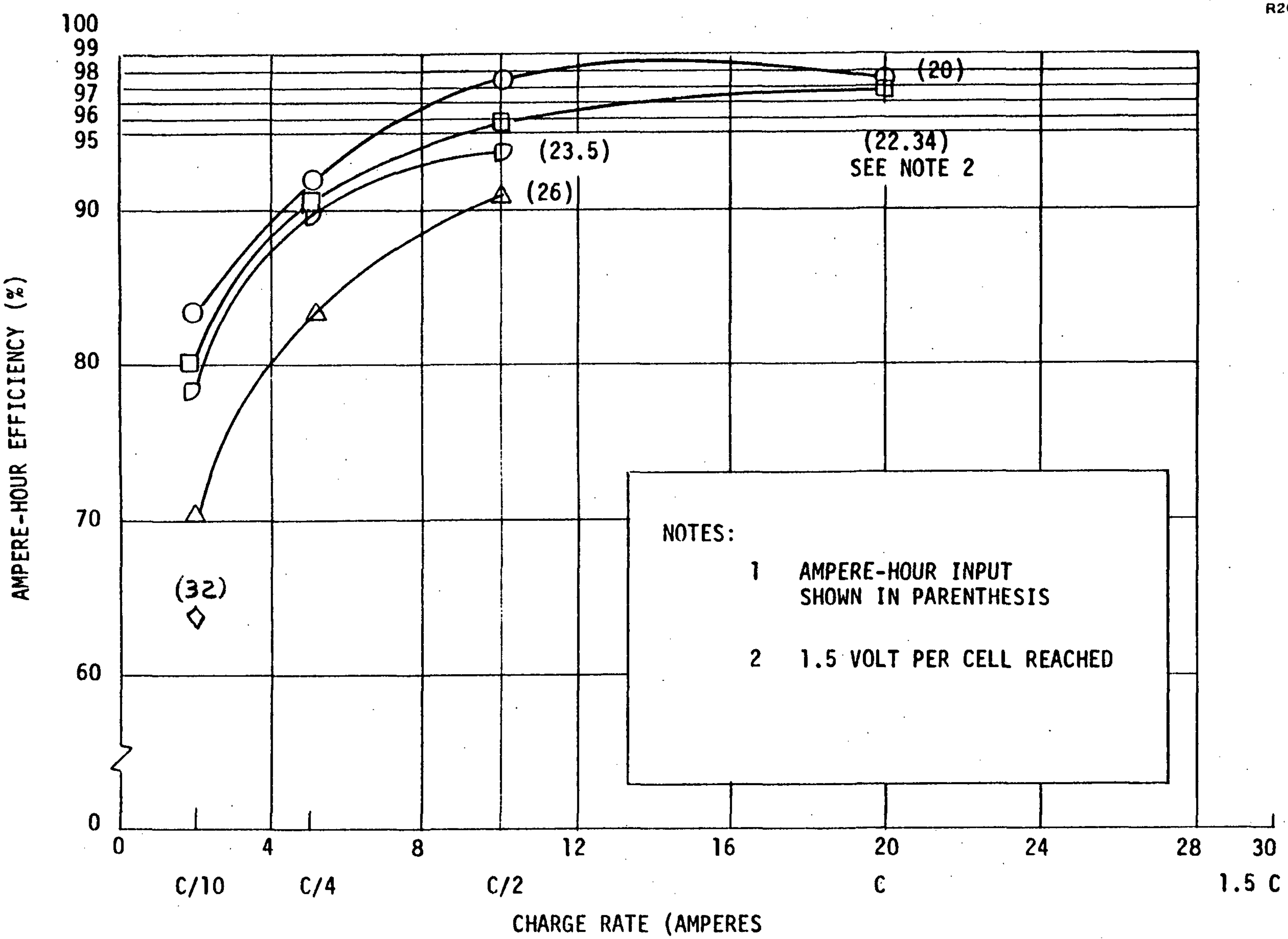


Figure 4.5-56 Charge Rate and Ampere-Hour Efficiency

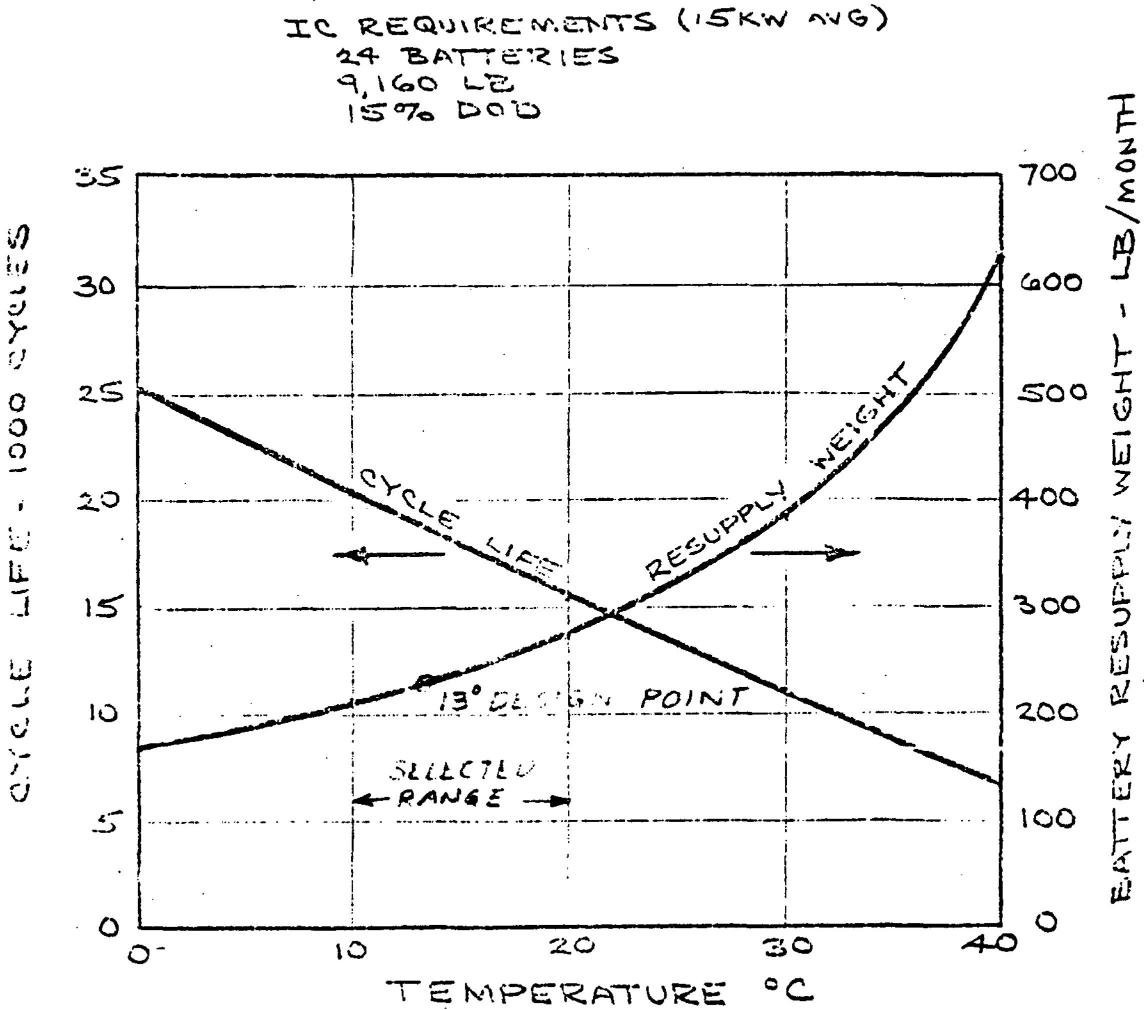


Figure 4.5-57 Battery Temperature Effects

Fuel Cell/Electrolysis vs. Batteries

A study was conducted to evaluate the use of a fuel cell/electrolysis assembly as an alternative to the nickel-cadmium battery assembly. With a solar array/fuel cell electrical power source, the fuel cell provides all of the eclipse time electrical power. The by-product is electrolyzed during the sunlight period of each orbit, utilizing solar array power.

The fuel cell design is based upon the Space Shuttle fuel cell so that development cost will be minimized. However, the details of the design are not well-defined at the present. Pratt and Whitney and General Electric have parallel contracts for the Space Shuttle fuel cell development. The NASA MSC guidelines for fuel cell design are shown in Table 4.5-35. The Pratt and Whitney fuel cell data for the MDAC design are used for this analysis, because the concept is further advanced and because it has a higher power capability.

Electrolysis units are not presently developed and any requirement for their use would incur a large development cost chargeable to the electrical power systems, because the EC/LS subsystem does not presently have a requirement for these units. Development work at Allis Chalmers was recently discontinued. The electrolysis unit selected for this analysis is the General Electric Solid Polymer Electrolyte water electrolysis system. The characteristics of this electrolysis unit are shown in Table 4.5-36.

The important tradeoff parameters between a fuel cell/electrolysis assembly and a battery assembly are shown in Table 4.5-37 for comparison. The GSS model used had six station modules, rather than the current MDAC design for five station modules.

The fuel cell/electrolysis assembly shows a definite advantage over a nickel-cadmium battery assembly in initial launch weight and resupply weight. However, in all other comparison factors the nickel-cadmium battery assembly shows significant advantages. The operating life of the fuel cell/electrolysis assembly is projected to about 1 year of operation taking into account the off-time, while the battery life shown is available with current

Table 4.5-35

NASA/MSC SPACE SHUTTLE FUEL CELL GUIDELINES

Weight, lb/kw	40 to 60
Sustained Power, kw	5 to 7
Voltage Level	28, 56, 112, 120
Voltage reg, 0 to 7 kw	±7.5 percent
SRC*lb/kwh	0.7 to 0.8
Operating life, hr	2,000 to 5,000
Reactant supply pressure, psia	200 or higher
Start-stop cycles	Unlimited
Reactant purity	Propulsion grade
Heat rejection means	Coolant and other
MDAC DESIGN	
Weight, lb	245
Volume, ft^3	4.2
Sustained power, kw	7
Voltage Level	120
SRC, lb/kwh	0.95

*Specific reactant consumption

Table 4.5-36
ELECTROLYSIS UNIT CHARACTERISTICS
GENERAL ELECTRIC DESIGN

Type	Solid polymer electrolyte
Nominal 0_2 rate	2.16 Lb/hr (2.59 lb/hr maximum)
Input voltage	40 to 60 vdc
Nominal input power	5,600 Watts (6,790 w maximum)
Installed weight	236 Lb
Maximum 0_2 production rate	36.3 Lb/day
Heat rejection (maximum)	
Power conditioner	1,830 Btu/hr
Electrolysis module	1,340 Btu/hr
Operating temperature	190°F
Cell current density	200 ASF (amperes/ft^2)

technology. The resupply weights are based upon these operating lives. An improvement in battery life and/or failure to qualify a 1-year fuel cell or electrolysis life will reduce the weight advantage shown for the fuel cell/ electrolysis assembly.

The electrolysis unit requires more solar array power than the batteries. An additional 4.1 kw of solar array power is required for ISS and an additional 8.2 kw is required for GSS. This corresponds to an additional solar array area of about 410 ft^2 for each of the two arrays. The fuel cell/electrolysis assembly has a significantly larger volume requirement for initial launch and resupply, and a much larger radiator area for heat rejection than the battery. The batteries are designed for 2.5 years of operation; replacement and maintenance are achieved through relatively simple mechanical and electrical interfaces. A philosophy for the replacement of fuel cells and electrolysis units has not yet been developed, but is expected to be more

Table 4.5-37

ENERGY STORAGE COMPARISON

	Fuel Cell/Electrolysis	Nickel-Cadmium Batteries
Operating life	2,000 to 5,000 hours (Assume 1 yr – 3,500 hr)	2.5 Yr (15 percent DOD)
Cost, (10 yr)	202.4×10^6	110.8×10^6
Weight, lb	3,500	9,120
Volume, ft^3	128	36
Recharge Power, kw	19.5 (ISS) 39.0 (GSS)	15.4 (ISS) 30.8 (GSS)
Resupply		
Materials	Fuel cells, electrolysis	Battery modules
Weight, lb	2,794 (ISS) 5,588 (GSS)	3,648 (ISS) 6,080 (GSS)
Volume, ft^3	51 (ISS) 102 (GSS)	14 (ISS) 24 (GSS)
Maintenance/replacement	No philosophy developed	Module or battery
Interfaces	Mechanical, Electrical, Liquid, Gas	Mechanical, Electrical
Start/stop capability	Fuel cells – good Electrolysis – good	Excellent

complex because of the liquid and gas interfaces in addition to the electrical and mechanical interfaces. The fuel cell/electrolysis assembly also has a significantly higher cost than the nickel-cadmium battery assembly. The costs shown in Table 4.5-37 are complete program costs for 10 years of operation, taking into account the ISS and GSS power requirements, and include recurring, nonrecurring, and resupply costs. A comparative breakdown of costs for fuel cell/electrolysis and nickel-cadmium batteries is shown in Table 4.5-38 based upon cost trade criteria used during the Modular Space Station study. The crewtime cost was not included but would probably increase the difference because of the more complex fuel cell/electrolysis interfaces and because of the more frequent replacement requirements. Also, the fuel cell/electrolysis assembly does not lend itself to a modularity similar to the batteries within an increase in weight and volume.

Conclusions

The fuel cell/electrolysis system disadvantages cited above result in the selection of a nickel-cadmium battery assembly as the best energy storage assembly for use with the solar arrays on the Modular Space Station. However, fuel cell/electrolysis assemblies should continue to be developed and compared with nickel-cadmium batteries for other missions because of their potential weight savings. Further development of fuel cells and electrolysis units may result in reductions in the disadvantages listed above, and other missions may find this concept to be more advantageous. A significant reduction would occur if electrolysis were a requirement for EC/LS and the development cost would be shared. Additional advantage would occur if (1) H_2 and 0_2 propulsion tanks are available and would be shared, (2) the missions are relatively short to reduce total life, (3) frequent return to Earth would permit replacements and maintenance to be performed on Earth. There merits are found on the Space Shuttle and the SOAR missions and support a fuel-cell selection for those missions.

4.5.4.4 Conditioning Assembly Analysis

Power-Conditioning Requirements

The power-conditioning assembly is required to provide battery charging power from the solar array, regulated DC load power from the batteries,

Table 4.5-38

ENERGY STORAGE MISSION COST COMPARISON
($ x 10^6)

	Fuel Cell/ Electrolysis	Nickel-Cadmium Battery
Initial weight ($250/lb)	0.87	2.28
Initial volume ($1,500/ft^3)	0.19	0.05
Recharge power ($7,650/watt-10 yr)	111.87	88.35
Development	42.00	3.60
Hardware	1.80	1.21
Operations (10 yr)		
Weight ($250/lb)	9.78	9.88
Volume ($1,500/ft^3)	10.73	0.23
Hardware	25.20	5.25
Total	202.44	110.85

Note: Crew time for replacement was not assessed but is expected to be greater with the increased resupply frequency of the fuel cell/electrolysis approach ($1,940/man hour).

and a regulated AC load power from the solar array/battery sources. The power conditioning assembly employs modular design with components placed throughout the Station in main distributor centers.

A battery charger is provided for each battery. The charger provides a modified constant potential charge with charge termination on a signal from the cell producing the limiting voltage of 1.5 vdc, or from a battery third electrode.

Battery load regulators provide regulated voltage at the DC load buses from the high voltage battery sets. In the normal or parallel mode of operation, the regulators are biased to share load equally. This forces the battery sets to operate at equal depths of discharge thereby assuring uniformity of charge on subsequent charging cycles.

Static inverters provide regulated voltage at the AC load buses. Three-phase sine wave inverters supply 400 Hertz power to transient-sensitive and transient-free AC loads. Three-phase quasi-square wave inverters supply 400 Hertz power to transient-producing and transient-tolerant AC loads. Single-phase sine wave inverters supply 60 Hertz power to GPL loads.

All power conditioners (battery chargers, battery load regulators, and inverters) incorporate current limiting features to prevent damage from external overloads.

Conditioning Modularity and Growth Accommodation

Each battery charger supplies one battery in a battery set (four batteries). No provision is made for switching the charger to another battery. Redundancy is achieved at the battery-set level. If a charger fails, the associated battery set is isolated. System operational requirements are met with no degradation in performance, using the remaining battery sets. The failed charger is replaced at the earliest opportunity. The chargers are sized to provide a short-term, high-rate charging capability when the station emerges from an eclipse, and maximum excess power is available from the solar arrays.

A battery load regulator is provided to condition the output of each battery set. No provision is made for switching the regulator to another battery set. Failure of a regulator produces the same result as failure of a charger; system operational requirements continue to be met with no degradation in performance. The battery load regulators are sized to handle, as a minimum, the contribution of a single battery set to the full system peak load with one battery set out of service.

The three-phase inverters are modularized. Identical three-phase inverter modules in each Space Station Module operate either one at a time or in multiple, and either isolated or paralleled, as may be required to maximize operating efficiency. The single-phase inverters operate one at a time. Identical inverters are synchronized to a common clock; however, inverters in one Station Module cannot be paralleled with like inverters in another Station Module.

Switchable redundant inverter modules are provided in the Station Modules to meet reliability requirments. The three-phase modules are sized to optimize efficiency over the expected load range in the individual Station Modules. The number of three-phase modules making up an inverter assembly in a given Station Module is chosen to provide the required capacity for the maximum Station Module load plus the RAM's AC load demand on that Station Module. Single-phase inverters are sized for the allocated 60 Hertz AC load in the GPL.

Growth accommodation for DC power conditioning is provided at the battery set level, i.e., four chargers and one load regulator must be added for each additional battery set. Growth accommodation for AC power conditioning is at the three-phase inverter module level for either sine wave or square wave power, and the single-phase inverter level for 60 Hertz power.

Conditioning Module Installations and Thermal Control

The conditioners are installed in the main distributor center in each Station Module. All conditioners are conductively cooled by the Station fluid cooling system. The temperature of the fluid coolant will be between 60°F (minimum inlet temperature) and 110°F (maximum outlet temperature). Design for heat removal from power conditioning components will be based on 90-percent conducted and 10-percent radiated heat transfer.

Battery chargers and battery load regulators are located adjacent to their respective battery sets. Charger control logic is integral with each charger. Battery load regulator logic is contained in the power conditioning unit (PCU). A functional description of charger and regulator controls is given in the next section.

Inverter controls also are integral with each inverter. Automatic paralleling, load sharing, and protection circuitry are incorporated in each three-phase inverter module. Controls for the single-phase inverters are much simpler; these units are not designed for paralleling.

Conditioning Module Characteristics

Battery Charger

The battery chargers employ pulse-width modulated (PWM) buck-type regulators to supply charging power at a nominal 42 VDC. Each charger weighs 5 lb and occupies 0.1 cu ft. Output rating is 1.0 kw. As previously indicated in Section 4.5.3.1, Figure 4.5-18, the four chargers for each battery set are paralleled at their inputs but isolated at their outputs. Each charger supplies a separate battery in the series - connected battery set. Each charger is biased on and off by outputs from its control logic. An elementary diagram of the charge control scheme is shown in Figure 4.5-58.

Inputs to the control logic are derived from the battery (individual cell limit voltage or third electrode signal), the PCU (sun sensor, load bus volts low, SPSR state "0", SPSR state "1"), and from the control console keyboard (battery recondition charge). Logic states shown on the diagram are for the high rate charger mode. When the cell limit voltage or third electrode signal is received from the battery, the output of the AND gate to the HIGH RATE function will change from a 1 to a 0; a 1 will appear at the input to TERMINATE CHARGE, and the charger will be biased to its off state. The charger is also biased off by an output from the INHIBIT function.

Battery Load Regulators

The battery load regulators also employ the PWM buck-type design. Regulator output voltage is adjustable from 112 vdc to 118 vdc and controllable to within ±1 percent of the set point. Output rating is 6 kw. Each regulator weighs 13 pounds and has a volume of 0.4 cu ft. Reverse current relays are provided in the regulator outputs to prevent parallel sources from feeding into an internal fault.

The battery load regulator for a given battery set is biased off when the battery chargers are on. This is indicated by the logic elements in the battery set regulation/control diagram on Figure 4.5-59. If the chargers are on (not inhibited), a 0 will be present at the output of the CHARGER SET 1 INHIBITED function in the lower right hand corner of the figure. Therefore,

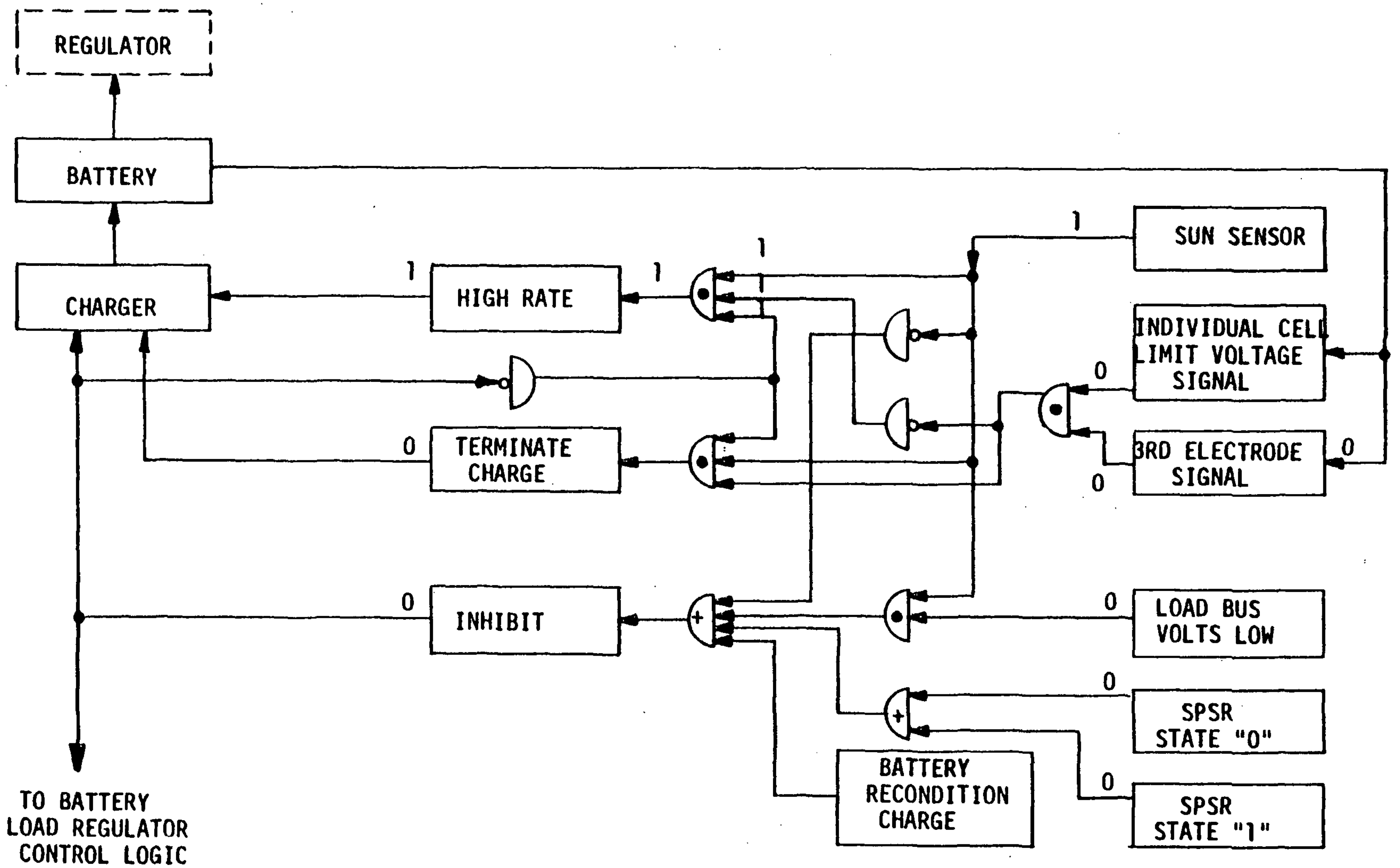


Figure 4.5-58 Battery Charge Control Logic (High Rate Mode Shown)

R260

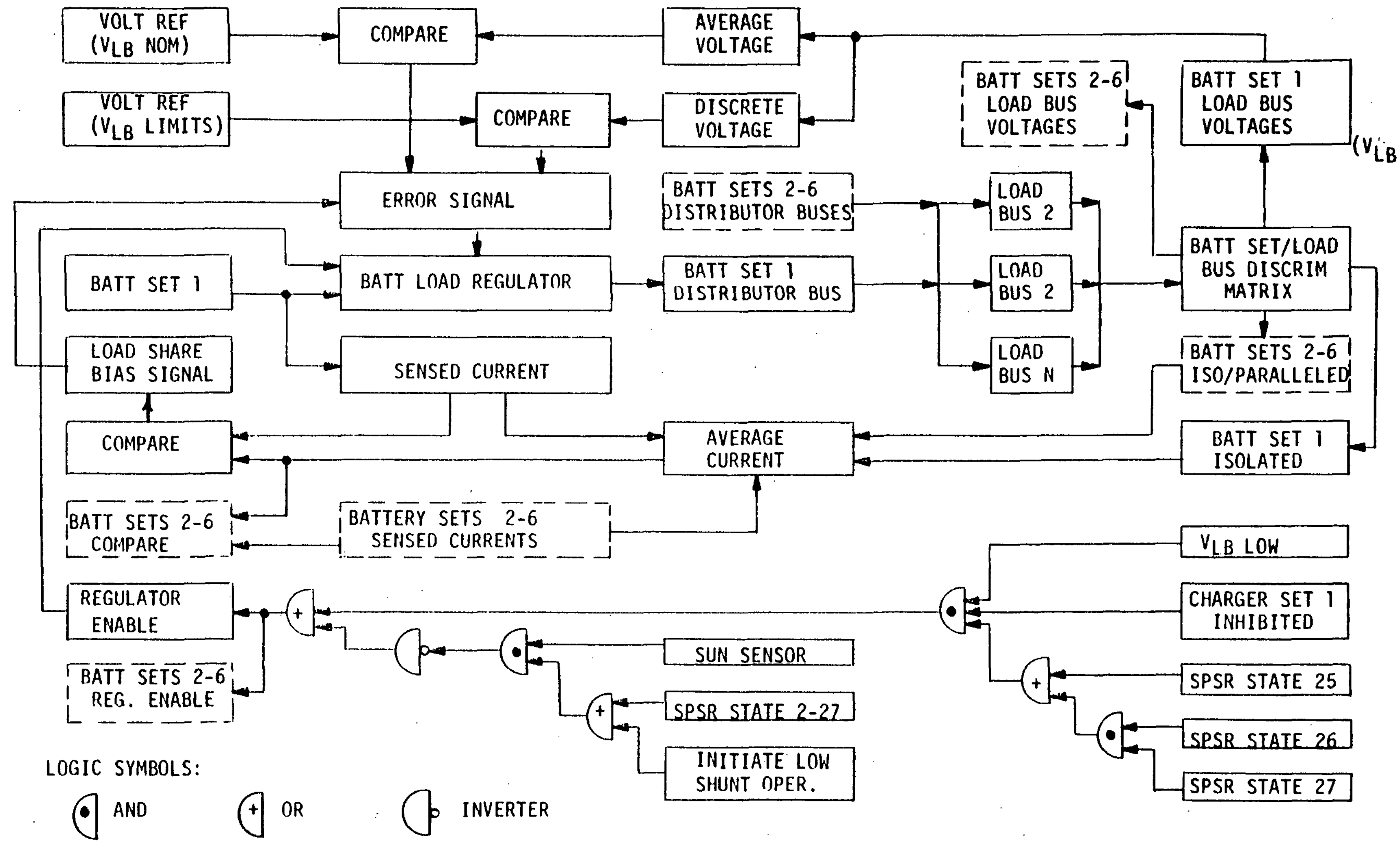


Figure 4.5-59 Battery Set Regulation/Control - ISS Configuration

a 0 will be present at the output of the function AND gate and at the input to the following OR gate. From Figure 4.5-59, if the chargers are not inhibited, the sun sensors must be on. Accordingly a 0 also will be present at the second input to the OR gate on Figure 4.5-59 so that the regulator is not enabled. In short, the regulator is off when the chargers are on (array operation normal), and on when the chargers are off (eclipse and post-eclipse battery operation, or array operation not normal).

Referring to the rest of Figure 4.5-59, it is seen that several battery control functions are identical to those shown on Figure 4.5-47 of Section 4.5.4.2.3 for the solar array. The battery set/load bus discrimator matrix performs the same basic function as the solar array wing/load bus discriminator matrix; namely, identifying the source or sources of power for each load bus. It is in fact part of the same matrix. The bus voltages associated with each source are first limit tested, then averaged as before for comparison with the 115-vdc nominal reference. In addition, the current from each battery set is sensed and fed into a processor which determines the average current for each group of paralleled sets. Identification of paralleled and isolated sets is established by the open/close status of switches in the discriminator matrix.

The average current of the paralleled groups is compared with the current from each set in the group to generate load share bias signals. These signals are used to bias the error signals generated by the voltage comparisons. The biased error signal causes the regulator to raise or lower its voltage as required to reduce the load share bias signal to zero, while maintaining the average load bus voltage as close as possible to the desired nominal value. In this way, all battery sets in the paralleled group will operate at equal depths of discharge. This is the normal mode of operation-- all load regulator/battery set units operating in parallel. A load bus voltage out-of-limits signal overrides the load share bias signal and immediately requires corrective action by the regulator or regulators supplying the affected bus.

Inverters

The three-phase sine wave and quasi-square wave inverter modules are sized to supply 750 VA (peak) at 115/200 VAC ±2-1/2 percent, 400 Hz

±1 percent. Each sine wave module weighs 11.7 lb and occupies 0.3 cu ft. Each square wave module weighs 7.5 lb and occupies 0.2 cu ft. Identical modules can be paralleled in any number to increase power capacity. The basic power switching and protection and control circuits can be separately packaged as interchangeable submodules.

All inverter submodules are driven from a master frequency reference which provides equalization of frequency among the paralleled modules and elimination of beat frequency interference from isolated modules. Load division circuitry limits unbalanced current among paralleled modules to less than 10 percent of module rated current. Protection is provided for over-voltage, under-voltage, abnormal frequency, and over-current (current-limiting). The control circuits provide for automatically switching in or switching out parallel modules as a function of load and efficiency requirements, in addition to providing regulation functions.

The single-phase sine wave inverters are sized to supply 500 VA (peak) at 115 VAC ±5 percent, 60 Hz ±1 percent. A single-phase module weighs 18 lb and occupies 0.4 cu ft. Protection features are identical to those for the three-phase modules. These units are not designed for parallel operation.

Design for Maximum Efficiency

Emphasis is placed on designing power conditioning equipment for maximum efficiency. High efficiency is achieved primarily by component derating. This also improves reliability. These gains result in increased conditioning equipment cost and weight; however, appreciable net program cost savings are realized using the weighted trade factors for launch weight ($250/lb) and power ($7,650/W-10 years). Power conditioning efficiencies used in this report have not been optimized for these trade factors, however.

4.5.4.5 Distribution Assembly Analysis

Distribution Requirements

The basic requirements of the distribution assembly are safety, reliability, and efficiency. Specific requirements include the following:

A. Provide for priority distribution of power based on load criticality assignments.
B. Provide redundant load buses and load circuits for system reliability and operational flexibility.
C. Provide for switching of buses and load circuits to meet system reconfiguration requirements.
D. Provide bus and circuit capacity to meet peak load demands within voltage regulation limits.
E. Provide coordinated protection of load buses and load circuits.

Load on-off switching and control are not requirements on the baseline EPS distribution assembly. These functions are included in the power-up and power-down provisions of the DMS/load systems interfaces as noted under the discussion of interface requirements for the Power Management Assembly, Section 4.5.4.6.

Operation of the distribution system will be simplified to the greatest extent consistent with performance requirements. Buses, feeders, and equipment will be located to facilitate maintenance and will be mechanically protected to minimize involvement of other circuits by faults. Vehicle structure will be used for mechanical protection where possible.

Circuit/Bus/Panel Configurations

The distribution assembly supplies DC and AC loads from separate DC and AC panels configured to reflect the following load criticality categories;

A. Emergency loads – operation is required to ensure crew safety for emergencies up to one hour.

B. Essential loads — operation is required to provide crew habitability, malfunction analysis, and repair capabilities for sustained emergencies.

C. Nonessential loads — not required for crew safety or crew habitability and emergency repair capability. Experiment loads and experiment support loads are included in this category.

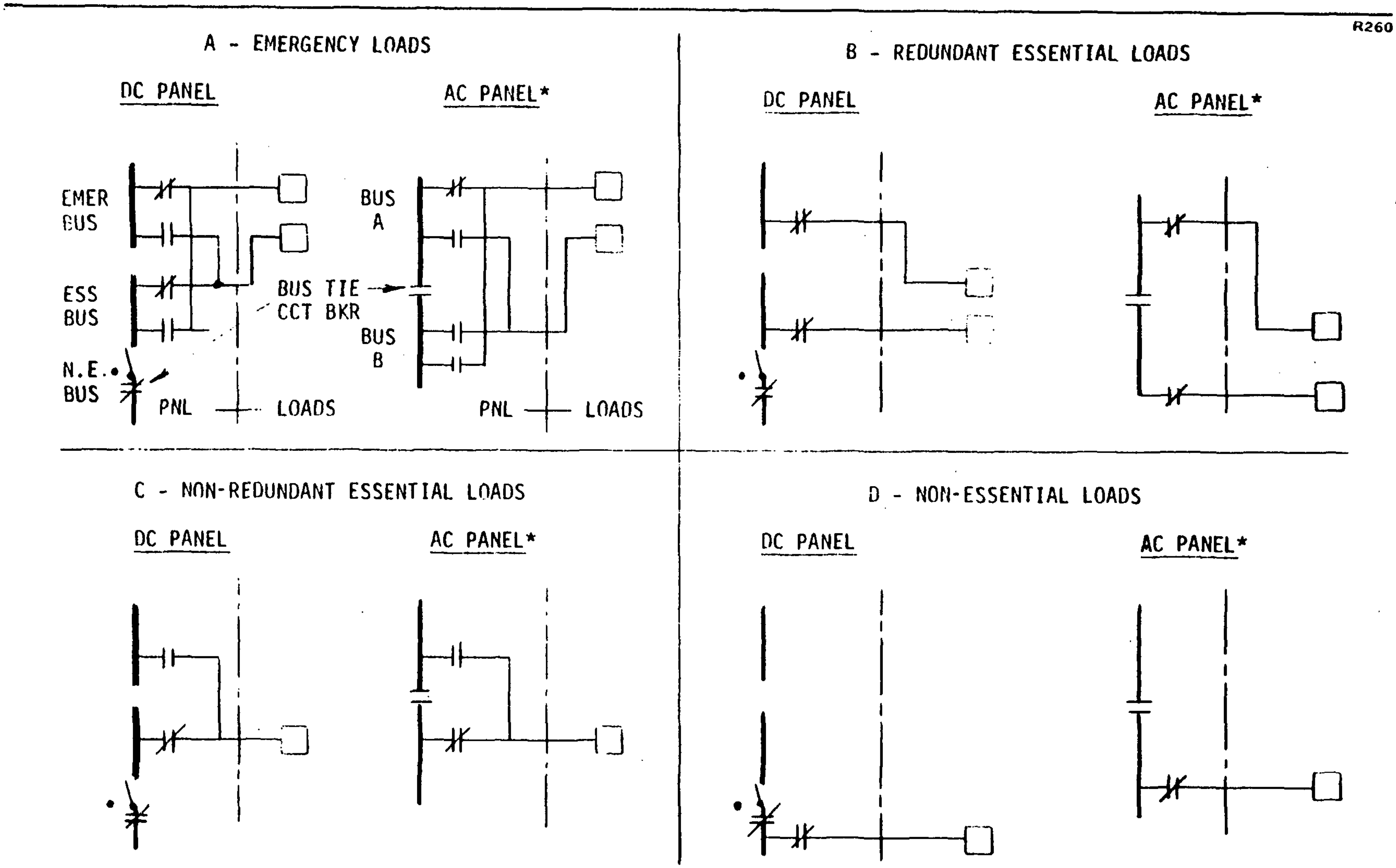
Redundant load buses, load circuit (feeder) terminations, and associated switches are installed in each panel in accordance with the following criteria.

A. Emergency loads, with their circuits and control should be 100 percent redundant, for safety reasons. Redundancy includes supply feeders. Emergency loads will be readily reconnectable to an alternate bus section in preparation for bus maintenance or in the event of a bus fault.

B. Essential loads are not required to be redundant for safety reasons, but may be selectively redundant for other reasons. Nonredundant essential loads will be readily reconnectable to an alternate bus section in preparation for bus maintenance or in the event of a bus fault. Nonredundant essential loads may be automatically deenergized during emergency conditions. No essential load will be deenergized until all nonessential loads have been deenergized.

C. Nonessential loads are discretionary in nature and are not redundant. Nonessential loads, principally consisting of experimental loads, are supplied by single buses in the AC and DC distribution systems. DC nonessential bus loads will be automatically deenergized by the power management system during overload or emergency operating conditions. Return to normal operation will be performed manually, after the fault or other abnormal condition has been identified and cleared.

Circuit and bus configurations for the basic types of loads in the DC and AC distribution panels are shown in Figure 4.5-60. One DC panel and one AC panel are installed in each Station module. Each panel provides the total DC or AC load circuits for the module and its interfacing loads (RAM's and Logistics Module) as required.



* EACH AC PANEL CONTAINS 400 HZ SINE WAVE AND SQUARE WAVE CIRCUITS. THE AC PANEL IN THE GPL ALSO CONTAINS 60 HZ SINE WAVE CIRCUITS. CONFIGURATIONS SHOWN HERE ARE TYPICAL FOR ALL AC POWER TYPES.

Figure 4.5-60 Load Bus and Load Circuit Configurations in Distribution Panels

Regulation and Electrical Characteristics

The 115-vdc distribution system is designed for a maximum drop of 4 vdc from the panel load buses to the loads. The 115/200 vac systems are designed for a maximum drop of 4 vac (rms) from the sine-wave and square-wave load buses to the individual AC loads. Table 4.5-39 summarizes the characteristics of all power types distributed to the loads.

Distribution Control/Protection Provisions

Each DC emergency and essential load bus can be supplied from either one or both sections of its associated distributor bus. The required switching is accomplished by the remote circuit breakers indicated in the distributor bus/load bus interface detail shown on Figure 4.5-61. These breakers are part of the primary switching assembly. They also provide overcurrent protection for the load buses and their supply feeders. In addition, they are under keyboard control and are operated in this manner for routine bus maintenance, for bus reassignments due to special loading conditions, or for system reconfiguration following a fault.

Each DC nonessential load bus is controlled by its bus tie circuit breaker, and where provided, by a bus selector switch. The bus tie breaker can be operated remotely. Under control of the power management assembly, it opens to dump nonessential bus load in the event of a sustained system overload. The breaker opens automatically for a fault in the bus itself.

The AC load buses are controlled by contactors in the supply circuits from the inverters, and by their respective bus tie breakers. Referring again to Figure 4.5-61, the two sections of each 400 Hertz AC load bus can be supplied supplied from either or both inverter sources; or can be isolated from each other and supplied separately if required. The 60 Hertz bus sections can be paralleled, but their inverters cannot.

All load circuits, DC and AC, are protected by magnetic circuit breakers. These breakers are not remotely controlled. No provision is made for remote control since there are no requirements for automatically transferring any load to an alternate bus or for using the circuit breakers for

Table 4.5-39

TYPES OF POWER DISTRIBUTED TO LOANS

Power	Steady-State Voltage Limits at the Loads (1)
115 vdc, 2-wire (2)	108 – 118 vdc (3)
400 Hz ± 1 percent, 3-phase 115/200 vac, 4-wire, quasi-square wave	108 – 118 vac line to neutral
400 Hz ± 1 percent, 3-phase 115/200 vac, 4-wire, sine wave	108 – 118 vac line to neutral
60 Hz ± 1 percent, 1-phase 115 vac, 2-wire, sine wave (4)	105 – 120 vac

(1) Normal operation.

(2) 28 vdc power can be provided for special loads, e.g., integral experiments where impact analysis warrants.

(3) Tentative limits. May be relaxed to 106 – 124 vdc, the same as proposed for Shuttle, if commonality warrants.

(4) 60 Hz power is distributed to GPL loads only.

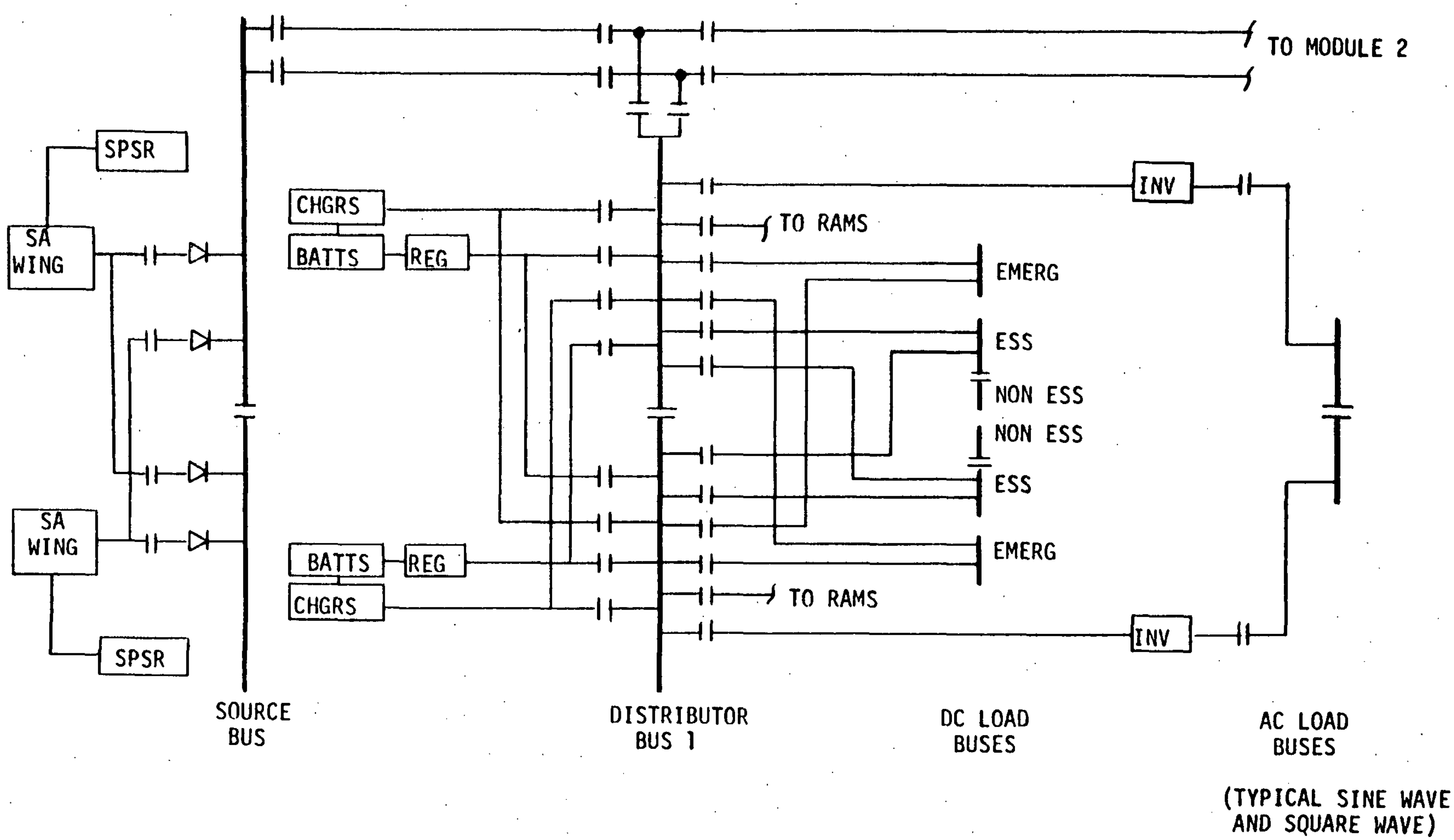


Figure 4.5-61 Module 1 Busing and Switching Scheme

programmed on-off switching of the loads. It is assumed that automatic load control will be provided at the load by control from the DMS. Circuit breaker pairs provided for transferring selected load circuits to an alternate bus are operated with only one breaker normally closed to minimize disturbance on one bus system in the event of a fault on the other.

4.5.4.6 Power Management Assembly Analysis

4.5.4.6.1 Interface Requirements

The power management assembly is required to interface with other EPS assemblies to provide monitor, checkout and/or control functions for contactor and circuit breaker switching, solar array voltage regulation, battery charging and discharging, and system/equipment protection. These functions may be in a central processor or an internal closed loop system. They may be either automatic or manual, or a combination of central/internal and automatic/manual, depending on operational requirements of the interfacing assembly.

The power management assembly interfaces with the DMS (displays and controls, data bus, and multiprocessor) to implement many of these functions. Through the DMS, the assembly also interfaces with all subsystem, RAMs, and Logistics Module loads to control overall Station loads according to established priorities, and to provide system isolation, test, and reconfiguration capabilities.

4.5.4.6.2 Instrumentation/Sensors

The power management assembly consists of interface instrumentation, sensors, displays, transducers, and control devices within the EPS and the data management subsystem and the onboard checkout subsystem which allow monitoring, display, control, and a substantial degree of automated power system supervision. Critical parameter values are read on hardwire or sampled by the data bus. The data are fed into a computer that reviews and compares the data with predetermined operating ranges. If not within the predetermined range, modifications will be made according to preprogrammed conditions. The values from the sensors are also capable of being read by command on the display panel.

Sensed parameters include voltage, current, temperature, frequency, power factor, position, pressure, and mode (state). These are primarily analog measurements. Commands on the other hand are primarily bilevel functions. Table 4.5-40 shows the distribution of EPS commands (stimuli) and measurements (responses) by signal type.

Measurement sensors, transducers, and signal conditioning for the EPS are provided as an integral part of that subsystem. The signal interface between the EPS and DMS is in the form of a DC voltage for each measurement. The voltage levels are in the ranges of 0-40 mv and 0-5 v.

4.5.4.6.3 Dedicated Computer/Preprocessor Functions

The power management assembly utilizes the DMS data bus, subsystem computer, and peripheral equipment for most monitor and essentially all checkout functions. Control functions primarily use hardwire signal transmission circuits and dedicated computer/preprocessors to perform operations such as gating, averaging, comparing and amplifying, especially where very high or near-continuous parameter sampling rates are required.

Table 4.5-40

EPS COMMANDS AND MEASUREMENTS

Module	Total Parameters	Stimuli		Response			Status Monitoring		
		An	Bi	An	Bi	Dig	Caution	Warning	Noncritical
1	1132	112	194	622	196	8	19	0	362
2	609	37	119	319	127	7	19	0	192
3	603	35	111	329	121	9	19	0	192
Total	2344	184	424	1270	444	24	57	0	746

Typical applications include low-level circuitry used in the PCU for controlling the solar array and battery voltage regulation systems, and the differential protection relaying equipment. Both are examples of automatic internal closed loop functions. Other examples include the inverter automatic paralleling and load-sharing control submodules, the current-limiting functions of conditioning equipment, the battery charge/discharge control equipment, and the circuit breaker and reverse-current protective relaying equipment. In addition to their automatic functions, these elements of the power management assembly provide status displays and other operating data to the command and control centers, and to the data management and onboard checkout subsystems via the Station data bus.

The DMS is utilized where status information already available from the data bus is needed in support of a control function, e.g., the source bus/load bus discriminator matrix in the array voltage regulation scheme. The DMS is also used to provide manual backup or override capability.

4.5.4.6.4 Monitor/Checkout/Control Operations

<u>Monitoring</u>

The EPS requires a minimum of crew supervision after initial array deployment. Monitoring of tunnel and mast drive motor positions, battery status displays, and readout of selected bus voltages provides the basic information for evaluating system performance. The ability to call up the status of other parameters such as array feeder currents, inverter output currents, or contactor and remote circuit breaker open/close status as may be deemed necessary for evaluation of a particular operational condition, provides the flexibility required to ensure adequate status assessments at any given time.

Continuous monitoring is required to detect out-of-tolerance conditions for parameters such as bus and battery voltages. Continuous or near continuous monitoring is also required to detect abnormal events. These include relay trips, contactor and circuit breaker trips, and power conditioner overload (current limiting signal).

Checkout

Checkout functions are those necessary to verify operational status, detect and isolate faults, and to verify proper operation following fault correction. Specific requirements include stimulus generation, sensing, signal conditioning, limit-checking, trend analysis, and fault isolation.

Periodic checkouts will be performed at intervals ranging from once per week to once each 6 months depending on equipment or parameters to be checked. Complexity of checkout varies from simple readouts of parameters such as voltage or temperature to injection of test currents into current transducer loop circuits to simulate fault conditions seen by differential protection relays. No major shock-producing tests, such as powerline faults or fault clearing, are planned. Tests for relay, circuit breaker, and contactor operations can generally be accomplished on line during periods of relatively low-scheduled experiment activity; system switching effects will be minimal.

Controls

The EPS is primarily controlled by automatic internal closed-loop functions, with selected operational functions under control of the DMS. Additional controls are provided in the DMS to support checkout functions, and to provide manual backup and override capability. These controls also provide for either manual or programmed reconfiguration of the EPS following automatic fault-clearing operations, as well as for facilitating reconfiguration to match changing loads or other operational conditions. The EPS portion of the primary and experiment/secondary command and control centers represent the EPS output and input terminals for these control modes.

Functions which are controllable from the command and control centers include the following:

A. Solar array deployment
B. Orientation drives
C. Tunnel pressure
D. SPSR shunt element drives
E. SPSR error bias adjustments

F. Battery module bypass
G. Battery charger on-off and mode select
H. Battery load regulator on-off and error bias adjusts
I. Protective relay test currents
J. Contactor and remote circuit breaker operation

4.5.4.7 EPS Cost Analysis Summary

A preliminary cost-effectiveness analysis was conducted during the early studies covering the key issues of solar array modularity and power-growth accommodation. These analyses addressed the closely-related issues of total program cost and the resulting resource effectiveness measured in kw-years available to the Space Station. Both ISS and GSS periods and power levels were included.

Subsequent studies were conducted for two-step and three-step growth options for the Baseline Modular Space Station, using two arrays or three arrays during the 10-year mission, respectively. These studies were based on arrays of both equal size and different sizes. Equal sizes resulted in the lowest costs due to commonality reduction of the non-recurring costs. A larger array followed by a smaller array incurred a larger cost for the initial array than for an equal-size approach, although the second (smaller) array benefited in cost by scaling down the more expensive initial array development cost. Higher program cost will result from a higher total array area during the program; or for equal total array areas, for (1) a large array followed by a small array, (2) unequal array sizes, (3) capability in excess of requirements, (4) an array modularity in excess of two, or (5) replacement rather than augmentation of arrays, because useful area is discarded when replacement occurs.

Concurrently, the power load profile underwent changes in both the time of growth to GSS and in the ISS and GSS power requirements. The two-step and three-step arrays were therefore studied with array growth occurring at different times from 2 to 6 years in the mission. An effort was made to satisfy the dual criteria of (1) meeting the projected power growth profile and (2) minimizing total program cost. A kw-year resource effectiveness was computed for each case to produce a cost-effectiveness evaluation in $ million/kw-year. The costs are total estimated program costs and the kw-year values are the integrated products of end-of-period kw for each growth step and the time durations in years. An alternative mode of augmentation prior to addition of the second Power/Subsystems Module consisted of a dedicated Shuttle launch with a "plug-in" solar array. The dedicated Shuttle flight cost estimate is $5.1M, thus adding $4.6M to the launch cost for the solar array weight on a cooperative launch.

The cost-effective values for the spectrum of cases studied ranged from $0.720M/kw-year to $0.788M/kw-year. This corresponds closely to the cost basis of $0.765M/kw-year used for power on all trade studies. The lowest-cost two-step approach yielded a cost-effectiveness of $0.760M/kw-year for two 5,500 ft^2 arrays, giving 16.1 kw at 5 years and 28.0 kw at 10 years, but having less complexity than the three-step approaches with plug-in arrays.

The solar arrays are the largest cost elements in both non-recurring and recurring costs. Therefore, the cost-effectiveness is directly dependent upon the overall electrical power system efficiency. The data used during these studies were based on a conservative system design factor of 2.5 (solar array delivered power during sunlight divided by load bus average power). The preliminary design phase indicates that the design factor should be reduced by 12.8 percent to 2.18, reflecting reduced power losses for transmission at 115 vdc rather than a lower voltage, for the sequential partial-shunt regulator rather than a series regulator, and for the thermally-controlled battery ampere-hour efficiency of 94 percent rather than 80 percent for room temperature operation.

The selected approach with a 5,300 ft^2 initial array which is replicated at GSS, and a design factor of 2.18, produces a corresponding program cost estimate of $161.5M for the EPS; a resource effectiveness of 239.0 kw-year (16.7 kw for 5 years followed by 31.1 kw for 5 years); and a cost-effectiveness rating of $0.676M/kw-year. This rating is 11 percent better than the value obtained with a 12.8 percent poorer design factor, thus indicating the close correlation predicted between EPS efficiency and power cost-effectiveness.

4.5.5 References

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4.6 ENVIRONMENTAL CONTROL/LIFE SUPPORT SUBSYSTEM

4.6.1 Summary

The Environmental Control/Life Support Subsystem (EC/LS) provides cabin atmosphere control and purification, water and waste management, pressure suit support, and thermal control for the entire Modular Space Station. The preliminary design described in this section has been developed to a depth sufficient to arrive at reasonable estimates of weight, volume, power and performance requirements, and operating characteristics. In addition to these design data, EC/LS interfaces and design impacts on other Space Station elements are identified.

In performing the Phase B Study of the Modular Space Station, data and designs in the areas of CO_2 removal and water recovery were provided by the Hamilton Standard Division of United Aircraft under a subcontract to MDAC. Assistance was also provided by numerous unfunded subcontractors.

The selected EC/LS design provides a habitable atmosphere, nearly the same as that at sea level, with a selectable temperature of 18.3 to 29.4°C (65 to 85°F). Two 6-man EC/LS units located in the Crew/Operations and GPL Modules are provided. Normally, each EC/LS unit processes the air in one separable, habitable compartment; however, in emergencies, either EC/LS unit can serve the entire station.

The EC/LS Subsystem provides full water recovery; that is, urine, wash water, and condensate are processed for reuse. This provides more water than is required for drinking and washing so that excess water is available. An open oxygen system is provided which complies with the guidelines for low initial and total program cost. However, provisions are included to retrofit the orbiting station with closed oxygen at any time it becomes advantageous. Oxygen for metabolic use is resupplied as high-pressure gas; oxygen and nitrogen for atmospheric makeup in case of leakage is stored as a gaseous reserve.

Crew metabolic CO_2 removed from the cabin atmosphere is collected and stored for use as propellant in the low-thrust propulsion subsystem. Expulsion of the CO_2 at high temperature and velocity minimizes potential contamination interference with the experiments.

Interference with the experiments is also minimized by providing a fecal collection and processor design which precludes dumping water vapor overboard. The feces are processed by heat and vacuum methods with an onboard pump; fecal water is stored and returned to Earth during scheduled Logistics Module return flights.

SeparateThermal Control Systems, consisting of a Freon 21 radiator loop and an integral water loop, are provided for each core module to facilitate station buildup and to allow any module to be removed for refurbishment or replacement. A separate interchange loop is included which allows transfer of heat between modules. This loop provides continuous thermal control within a module when its own thermal control loop is inadequate or inoperative.

EC/LS process heat is derived from a solar collector mounted on the solar array support structure. Solar heat is absorbed by the solar collector surface which has a highly absorptive coating for trapping solar energy. Two separate circulating water loops pass through redundant tubes in the solar collector. Each loop transports heat to each of the EC/LS units located in separate compartments to provide heat for CO_2 removal, urine recovery, and water storage.

Crew support provisions are included for suited or unsuited IVA and for suited EVA. Additionally, a 96-hour pallet is provided which contains all crew life-support needs for 4 days of emergency operation. Two pallets are located in the GPL Module and one in each attached module (during initial buildup one is also located in the Power/Subsystems Module). The main purpose of the 96-hour pallet is to support the crew during an abort operation when major Space Station systems have become permanently disabled.

A number of backup modes of operation are provided for the critical EC/LS functions. Both the normal and contingency atmosphere stores are available either through the normal-use manifold or through manually operated valves located in the storage areas. Atmosphere reconditioning units are designed for easy maintenance; redundant units are also provided. The 96-hour pallets supply additional capacity for atmosphere reconditioning.

A 30-day contingency water supply backs up the water recovery unit which is maintainable at the component level. The thermal control system has considerable redundancy since each core module has a separate system, external radiators are redundant and because an interchange loop is incorporated between the basic S/S modules. Additional capability is provided in the 96-hour pallet for potable water and for thermal control.

In order to arrive at a refined preliminary design, a number of trade studies and analyses were performed to decide key issues in the EC/LS design. The major results are highlighted in Table 4.6-1.

The trade studies on the EC/LS Subsystem were performed on both qualitative and quantitative bases. Each concept considered for the design is required to meet absolute criteria of performance and safety and to be capable of being developed for a 1979 launch date. Concepts which meet these absolute criteria have been evaluated from a quantitative cost standpoint which includes costs for launch weight, launch volume, power requirements, crew time for maintenance, resupply costs and hardware costs. Qualitative evaluation criteria considered include flexibility, growth potential, interface sensitivity, development risk and complexity. Table 4.6-1 gives the major considerations which influenced each trade result.

The thermal control design involved a major analytical effort to provide data for design decisions and to determine the performance adequacy of the final design. The selected radiator design meets all performance requirements for all modules in "worst case" conditions of peak internal heat loads, orbits yielding maximum incident heating, maximum values for orbital heating constants, least favorable docked module configuration and vehicle attitude and for a degraded surface coating.

Table 4.6-1

KEY TRADES AND ANALYSES IN EC/LS DESIGN

Item	Disposition
Degree of oxygen-loop closure	Open-loop O_2 selected because of low initial and total program cost
	Oxygen recovery retrofit capability provided
Degree of water-loop closure	Closed-water loop chosen because of low cost
Process heat	Solar collector selected because of low cost and safety
Compartment pumpdown	Pumpdown selected to reduce large air resupply required for nonscavenging approach
	Sufficient onboard accumulator capacity for isolation test facility
Water recovery by CO_2 hydrogeneration	Not selected because large water makeup needed caused by low water-recovery rate
EC/LS modularity	Six-man EC/LS size chosen for lowest weight, volume, and power
EC/LS backup	Dual EC/LS preferred with oxygen and water contingency supply
	Ninety-six-hour pallets provide emergency provisions
Atmosphere storage method	Gaseous storage selected because of low initial cost
Regenerable versus nonregenerable charcoal	Nonregenerable charcoal used because of low cost and simplicity
CO_2 control method	Molecular sieve chosen because of lower cost and advanced development status
Urine recovery method	Air evaporation selected because it is well developed, therefore, can be obtained at low cost
Wash-water recovery method	Reverse osmosis chosen because of low expendable requirements and better flexibility
Air distribution ducting and coolant water line sizes	Optimized for minimum total penalties
Radiator capacity	Analyzed using computer program; adequacy of design confirmed

Although an active radiator concept was selected, less conventional designs were considered including refrigeration cycles, heat pipes and hybrid systems. The active radiator concept was selected based on extensive data from previous studies and in particular the 10-m (33 foot) diameter Space Station. Additionally, the selected concept was proven successful on previous space vehicles (Apollo and Gemini). Alternate systems have the disadvantages of requiring extensive development which would result in higher costs. The detailed design of the thermal control loop involved a number of subtrades to arrive at an efficient design. These involved studies for determining radiator tube geometry and layout on the radiator surfaces, fluid temperature and flow control, and meteoroid protection.

4.6.2 Requirements

4.6.2.1 General Requirements

Table 4.6-2 tabulates program and project requirements which have a significant influence on the definition of the EC/LS Subsystem. These requirements are extractions from the Space Station Project and Program (modular) Specifications No. PS02925 and No. RS02927 of NASA reports MSFC-DPD-235/DR No. CM-01 and CM-02. The originating paragraph in the Specifications follows each requirement in the table.

A number of key requirements exist which particularly impact the EC/LS design. These requirements stress the need for EC/LS overcapacity and for multiple EC/LS units. The general requirement for "two separate pressurized habitable compartments with independent life support capability" requires at least two separate sources for each major life support function. Closely related to this requirement is the need to provide for crew overlap. The two independent sources for each life support function will in most cases also satisfy the crew overlap requirement.

The guideline for "safety as a mandatory consideration" impacts the design by emphasizing safety in EC/LS concept selection and in the design for redundancy and backup provisions.

Table 4.6-2

GENERAL EC/LS REQUIREMENTS (Page 1 of 3)

1. The Initial Space Station will be sized to accommodate at least six crewmen. Provisions for double occupancy will be provided in case they are required during relief crew overlap periods. (3.7.1.2.1)

2. A minimum of two separate pressurized habitable compartments with independent life support capability and provisions and other essential services will be provided at each manned stage of cluster buildup and operation. (3.2.1.2.2)

3. The Space Station will be capable of accommodating a mixed male-female crew. (3.1.3.1.5)

4. The Space Station shall be divided into at least two pressurized habitable volumes so that any damaged module can be isolated as required. Accessible modules will be equipped and provisioned so that the crew can safely continue a degraded mission and take corrective action to either repair or replace the damaged module. (3.2.6.2.1)

5. At least 30 days consumables, including subsystems and experiments, will be available beyond the scheduled resupply mission. (3.4.2)

6. The Initial Space Station shall have the capacity for independent operation with the full crew for a period of 120 days. This capacity can be included in a cargo module. (3.4.1)

7. Redundant equipment and electrical or fluid paths shall be physically separated, where possible, to minimize the probability of damage to one when the other is damaged. (3.2.6.2.4)

8. Two or more suited crewmen will participate in any pressure-suit activity and rescue provisions will be provided. (3.2.6.4.4)

9. All materials selected for use in habitability areas will be nontoxic, nonflammable, and nonexplosive to the maximum extent practical. (3.3.2.2.7)

10. Safety is a mandatory consideration through the total program. As a goal, no single malfunction or credible combination of malfunctions and/or accidents shall result in serious injury to personnel or to crew abandonment of the Space Station. (3.2.6.1.1)

11. Provisions and habitable facilities shall be adequate to sustain the entire crew for a minimum of 96 hours during an emergency situation requiring Shuttle rescue. (3.2.6.6.2)

12. Access to an EVA and IVA airlock suit station(s) shall be provided for all credible emergency conditions. Airlock chamber(s) shall be provided to permit crew access for EVA/IVA operations. (3.2.6.5.4)

Table 4.6-2

GENERAL EC/LS REQUIREMENTS (Page 2 of 3)

13.	Critical onboard subsystems will be designed to minimize risk of loss of modules, injury to the crew or damage to the Shuttle and other interfacing vehicles. (3.2.6.2.7)
14.	The EC/LS system will provide a shirtsleeve environment within habitable areas for crew activities during the buildup, activation periods, and module replacement period. (3.7.1.3.1.1)
15.	The Space Station structure and subsystems will be designed for an oxygen/nitrogen mixture at a normal operating pressure of 101 kn/m^2 (14.7 psia). (3.2.1.1.16)
16.	Carbon dioxide partial pressures will be maintained below 0.4 kn/m^2 (3.0 mm Hg) in all habitable areas. (3.7.1.3.1.2)
17.	The environmental control and life support subsystem shall be designed with a closed wash water loop. Closure of other functional loops will be based on the appropriate trade data. (3.7.1.3.1.3)
18.	An active temperature control system shall be provided with external fluid radiators. The cabin temperature will be selectively maintained between 18.4°C and 24.0°C (65 and 85°F). The mean radiant wall temperature, referenced to the crew, will be maintained between 15.5°C and 26.6°C (60 and 80°F). The maximum surface temperature of surfaces that may be contacted by the crew shall not exceed 40°C (105°F). The atmosphere velocity will be maintained, in habitable regions, between 0.1 and 0.25 m/sec (20 and 50 ft/min). The partial pressure of the cabin atmosphere water vapor will nominally be maintained about 1.1 kn/m^2 (8 mm Hg) and shall not exceed 1.7 kn/m^2 (13 mm Hg). Short transients to 0.8 kn/m^2 (6 mm Hg) will be allowed. No condensation shall form on internal surfaces. (3.7.1.3.1.4)
19.	The potability of resupply water must be verified prior to its use on the Space Station. The potability of water used by crewmen will be monitored and controlled. (3.7.1.3.1.5)
20.	Atmosphere stores and subsystem capacity sufficient for one repressurization of the largest pressurized habitable volume shall be maintained on the Space Station during manned operations and be available to independently supply any pressurized habitable volume. (3.7.1.3.1.6)
21.	The atmosphere constituents, including harmful airborne trace contaminants and odors, will be monitored and controlled in each pressurized habitable volume. (3.7.1.3.1.7)

Table 4.6-2

GENERAL EC/LS REQUIREMENTS (Page 3 of 3)

22. Space Station modules shall be launched at orbital operating pressure of standard atmosphere of 101 kn/m^2 (14.7 psia) with no programmed venting. However, overpressure control and emergency venting shall be provided. Systems shall be designed to operate at nominal atmospheric conditions as well as survive pressures of 0.00138 kn/m^2 (0.01 mm Hg). Systems are not required to operate at 0.00138 kn/m^2 (0.01 mm Hg). (3.1.3.2.5)

23. Heat transport fluids located within pressurized crew compartments shall be nontoxic and nonflammable at ambient atmosphere pressure and composition. (3.3.2.2.6)

The requirement for a low 0.4 kn/m^2 (3.0 mm Hg) CO_2 partial pressure impacts the design in the areas of concept selection for CO_2 removal and air distribution design. Large ventilation rates are needed to maintain low CO_2 level throughout the Space Station. Additionally a 0.4 kn/m^2 (3.0 mm Hg) PCO_2 favors removal concepts which control to low CO_2 levels, i.e., H_2 depolizer and carbonation cell.

Low cost is emphasized as a program requirement to minimize initial cost and total program cost. This requirement favors low cost subsystems and suggests that trades be performed from a cost standpoint rather than the classical minimum weight standpoint.

4.6.2.2 Functional Requirements

Table 4.6-3 tabulates key functional requirements for the EC/LS Subsystem by assembly group. A more detailed presentation of the functional requirements is given in the CEI Specification, DR Number CM-03.

4.6.2.3 Design Requirements

Design level requirements are listed in Table 4.6-4 at the functional assembly group level. These requirements were derived from the Phase B Space Station Study and from the Space Station Prototype Program.

Table 4.6-3

EC/LS FUNCTIONAL REQUIREMENTS

Assembly Group	Requirement
Atmosphere Supply and Control	Provide makeup O_2 and N_2
	Provide contingency O_2 and N_2 supply
	Maintain atmosphere pressure and composition control
	Provide for compartment pressurization/ depressurization
Atmosphere Reconditioning	Provide atmosphere temperature control
	Provide atmosphere humidity control
	Provide atmosphere CO_2 control
	Provide atmospheric trace contaminant control
	Provide oxygen recovery from CO_2 (if applicable)
Water Management	Process and render potable water from urine, condensate, and wash water
	Store and deliver potable water
Waste Management	Collect and transfer urine
	Collect, process, and store fecal waste
EVA/IVA	Provide support for EVA and IVA pressure suit activity
	Provide portable crew O_2 supply
Thermal Control	Collect waste heat from all liquid cooled equipment.
	Provide process heat for EC/LS equipment
	Transfer waste heat to the radiator for rejection to space.

Table 4.6-4

EC/LS DESIGN REQUIREMENTS (Page 1 of 3)

Atmosphere Supply and Control	
Depressurizable compartment volume	23.2 m^3 - 440 m^3/30 days (820 - 15,600 ft^3/30 days)
Depressurizable compartment time, minimum	24 hr for 115 m^3 (4078 ft^3)
Repressurization/contingency supply	Onboard storage for one repressurization of largest compartment
Repressurization time	6-hr maximum
Leakage	Negligible
Atmosphere relief	Relieves cabin pressure at 105.5 ± 1.4 kn/m^2 (15 ± 0.2 psia). Dump largest compartment to 6.89 kn/m^2 (1 psia) or less in 3 minutes.
Atmosphere	
Oxygen partial pressure	21.4 kn/m^2 (3.1 psia)
Total pressure	101 kn/m^2 (14.7 psia)
Atmosphere Reconditioning	
CO_2 partial pressure	Normal - 0.4 kn/m^2 (3 mm Hg) or less Emergency - 1.0 kn/m^2 (7.6 mm Hg) maximum for 7 days.
CO_2 generation rate, peak/average	0.354/0.260 kg/hr (0.78/0.575 lb/hr) (6 men)
O_2 use rate, average	0.218 kg/hr (0.48 lb/hr) (6 men)
Trace contaminants	Same as Phase B SS Study (see Ref 1)
Free moisture in atmosphere	None allowed
Particulate filtration level	Class 100,000 clean room
Atmosphere heat load	Crew metabolic +20% of net electrical power output
Metabolic levels	Normal - 136 watts (465 Btu/hr) for 24 hr. Design - 2 men at 235 watts (800 Btu/hr) 4 men at 161 watts (550 Btu/hr)
Atmosphere temperature	18.4 to 23.9°C (65 to 85°F) selectable
Dewpoint temperature	7.2 to 14.5°C (45 to 58°F) with transients to 4.5°C (40°F) allowable

Table 4.6-4

EC/LS DESIGN REQUIREMENTS (Page 2 of 3)

Atmosphere Reconditioning	
Mean radiant wall temperature	18.4 to 23.9°C (65 to 85°F)
Velocity in occupied regions	0.1 to 0.25 m/sec (20 to 50 ft/min)
Design latent load	
Crew	640 watts (2180 Btu/hr)
Crew equipment	385 watts (1313 Btu/hr)
Experiments	306 watts (1042 Btu/hr)
Water Management	
Urine rate, average	1.64 kg/man-day (3.61 lb/man-day)
Urine solids rate, average	0.0728 kg/man-day (0.16 lb/man-day)
Urine flush water, average	0.163 kg/man-day (0.36 lb/man-day)
Wash water rate, average	22.2 kg/man-day (48.9 lb/man-day)
Wash water rate, peak	109 kg/hr (240 lb/hr) for 10 min
Wash water output temperature	40.5°C (105°F)
Potable water delivery temperature, hot	71°C (105°F)
Potable water delivery rate, hot	54.5 kg/min (120 lb/min) for 3 sec
Potable water delivery temperature, cold	7.2°C (45°F)
Potable water delivery rate, cold	54.5 kg/min (120 lb/min) for 3 sec
Sterilization requirements	Capability to sterilize required
Potable requirements	Determined by Space Science Board, National Academy of Sciences
Waste Management	
Frequency of defecation	1/man-day
Duration of defecation	5 minutes/event
Fecal solids	0.036 kg/man-day (0.08 lb/man-day)
Fecal water	0.112 kg/man-day (0.25 lb/man-day)
Frequency of micturations	6/man-day
Duration of micturation	1 minute/event
Urine solids	0.0715 kg/man-day (0.16 lb/man-day)
Urine water	1.54 kg/man-day (3.45 lb/man-day)

Table 4.6-4
EC/LS DESIGN REQUIREMENTS (Page 3 of 3)

EVA/IVA	
EVA metabolic rate, peak	586 watts (2,000 Btu/hr) 2 hr
EVA metabolic rate, average	352 watts (1,200 Btu/hr) 4 hr
IVA metabolic rate, peak	470 watts (1,600 Btu/hr) 2 hr
IVA metabolic rate, average	234 watts (800 Btu/hr) 4 hr
Preconditioning time per suited event	6 to 8 hr
Suit pressure	25.5 kn/m^2 (3.7 psia)
Average number of EVA activities	1.5 events/month
Number of crewmen	2 crewmen/event
Thermal Control	
Orbit inclination	55 degrees
Orbit altitude	444 to 500 km (240 to 270 nmi)
Orientation	No restrictions allowed
Heat leaks	Minimize
Structural interface	Integrated meteoroid shield/radiator
Radiator reliability	0.99 for each module for 10 years

4.6.3 Selected Subsystem Design

The selected design described below is based on the requirements noted in subsection 4.6.2 and design analyses and trade studies reported in subsection 4.6.4 of this report. The description includes estimated weight, volume, power requirements, interface characteristics, and operational considerations of the EC/LS Subsystem to a depth consistent with or exceeding the requirements of the Phase B study and emphasizes definition of those interfaces which have a significant effect on other subsystems of the Modular Space Station. The description applies to the ISS level of buildup; subsection 4.6.3.4 describes the growth to the GSS.

4.6.3.1 Subsystem Description

The EC/LS Subsystem provides cabin atmosphere control and purification, water and waste management, pressure suit support, and thermal control for the entire Space Station. Figure 4.6-1 shows a subsystem-to-assembly breakdown for the baseline design. Primary characteristics of the EC/LS Subsystem include the following:

A. A 101 kn/m^2 (14.7 psia) sea level atmosphere is provided with a partial pressure of oxygen constant at 21.4 kn/m^2 (3.1 psi).

B. Two 6-man atmosphere reconditioning subsystems are provided, one in the Crew/Operations Module and one in the GPL. Each unit is capable of processing the entire Station atmosphere including attached modules. Normally the unit in the Crew/Operations Module will process all atmosphere except the GPL which will provide its own processing. In a contingency, either unit will process the total atmosphere. Each module contains separate atmosphere cooling provisions.

C. The ISS employs an open oxygen loop but provisions are included to permit adding oxygen recovery at any time desired. CO_2 removed from the atmosphere is used in the resistojet low-thrust propulsion system.

D. The subsystem provided has full H_2O recovery; that is more water is recovered in the Space Station than is required for drinking and washing. A water management system is located in the Crew/Operation Module and a 30-day contingency water supply is located in the GPL.

E. The total heat generated in the Space Station is rejected to space through segmented radiators integrated with the micrometeoroid shield. Each core module contains independent thermal control loops. A separate water loop between core compartments provides a sharing of cooling capacity. A solar heat collector is mounted on the solar array structure to provide EC/LS process heat.

All normal operations of the EC/LS are automatic; the crew will only occasionally review status and initiate seldom-used operations such as pumpdown. Primary control is by simple controls located on the EC/LS assemblies; the data management system performs checkout diagnosis and monitoring operations. This arrangement is chosen to make the EC/LS nearly independent of other subsystems, thereby enhancing crew safety.

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ENVIRONMENTAL CONTROL/LIFE SUPPORT

1000 ATMOSPHERE SUPPLY AND CONTROL

- 1100-1 O_2 GAS STORAGE
- 1100-2 N_2 GAS STORAGE
- 1200 PRESSURE REDUCTION
- 1300 PRESSURE CONTROL
- 1400 COMPARTMENT REPRESSURIZATION
- 1500 DUMP AND RELIEF
- 1600 PUMPDOWN ACCUMULATOR
- 1700-1 DOCKING PORT PUMPDOWN PRESSURE CONTROL
- 1700-2 AIRLOCK PUMPDOWN PRESSURE CONTROL
- 1800 AIRLOCK PUMP
- 1900 PLSS RECHARGE

2000 ATMOSPHERE RECONDITIONING

- 2100 HUMIDITY CONTROL
- 2200 TRACE CONTAMINANT CONTROL
- 2300 CO_2 REMOVAL
- 2600 VENTILATION
- 2700 TRACE CONTAMINANT MONITORING
- 2800 AIR TEMPERATURE CONTROL
- 2900 ATMOSPHERE DISTRIBUTION

3000 WATER MANAGEMENT

- 3100 URINE WATER RECOVERY
- 3200 WASH WATER AND CONDENSATE RECOVERY
- 3300 RECOVERED WATER STORAGE
- 3400 CONTINGENCY WATER STORAGE

4000 WASTE MANAGEMENT

- 4100 WASTE COLLECTION STORAGE
- 4200 FECAL WATER STORAGE

5000 EVA/EVA

- 5100 PORTABLE LIFE SUPPORT SYSTEM
- 5200 UMBILICAL LIFE SUPPORT
- 5300 IVA SUPPORT
- 5400 PORTABLE O_2 BOTTLES
- 5500 EMERGENCY EC/LS PALLET

6000 THERMAL CONTROL (1)

- 6100 COLLANT WATER RECIRCULATION
- 6200 RADIATON CONTROL
- 6300 RADIATOR RECIRCULATION
- 6400 INTERCHANGE LOOP RECIRCULATION
- 6500 WASTE HEAT EXCHANGER
- 6600 HEATING WATER RECIRCULATION
- 6800 COLDPLATE

[1] RADIATOR AND INSULATION ARE PART OF STRUCTURE

Figure 4.6-1 EC/LS Subsystem Assembly Breakdown

A total mass balance for the EC/LS Subsystem is shown in Figure 4. 6-2. Inputs are food, water contained in the food, and gaseous oxygen makeup. Outputs are fecal water, miscellaneous solids associated with the metabolic process, nonrecoverable water from urine purification, carbon dioxide utilized by the Propulsion Subsystem, and a water surplus, part of which is used for EVA cooling. With the exception of the CO_2 used as propellant and the leakage gases, no products from the EC/LS are dumped or vented overboard.

The wash water and condensate recovery assembly purifies 80 percent of the condensate and wash water; the 20-percent residue is cycled to the urine water recovery assembly. There the residue, the urine, and the urine flush water are purified at a 99-percent efficiency; the only water lost is that contained in the replaceable wicks. The purified water from the water recovery units provides wash water, water for EVA cooling, and the water consumed by the crew in excess of that provided in the food. Oxygen required for crew metabolic usage is resupplied in the form of gas. The carbon dioxide is transferred to the Propulsion System where it is used as resistojet propellant for orbit-keeping and control moment gryo (CMG) desaturation.

The excess water not used for cooling during extravehicular activity (EVA) events provides a contingency that can be used for experiments and an allowance for uneaten food or water lost in trash disposal.

Figures 4. 6-3, 4. 6-4, and 4. 6-5 are assembly-level schematics of the EC/LS Subsystem which show how the assemblies are interrelated and how the subsystem is integrated within the 3-core modules. Figure 4. 6-6 illustrates the type of oxygen recovery unit which could be added during the mission if constraints on logistics resupply or other factors indicated the need. As previously noted, the assemblies provided in the Crew/Operation Module and the GPL Module each have the capability to support six men and either system can provide reconditioned atmosphere to any core module. If the EC/LS is inoperative in either compartment, the remaining unit can accommodate the entire ISS through the interconnecting ducting. However, the two habitable volumes are processed separately although some intermixing will occur.

With this concept, a major emergency such as a fire, decompression, or massive contamination will affect only the atmosphere in half of the Space Station. The crew will be able to isolate the other compartment and continue

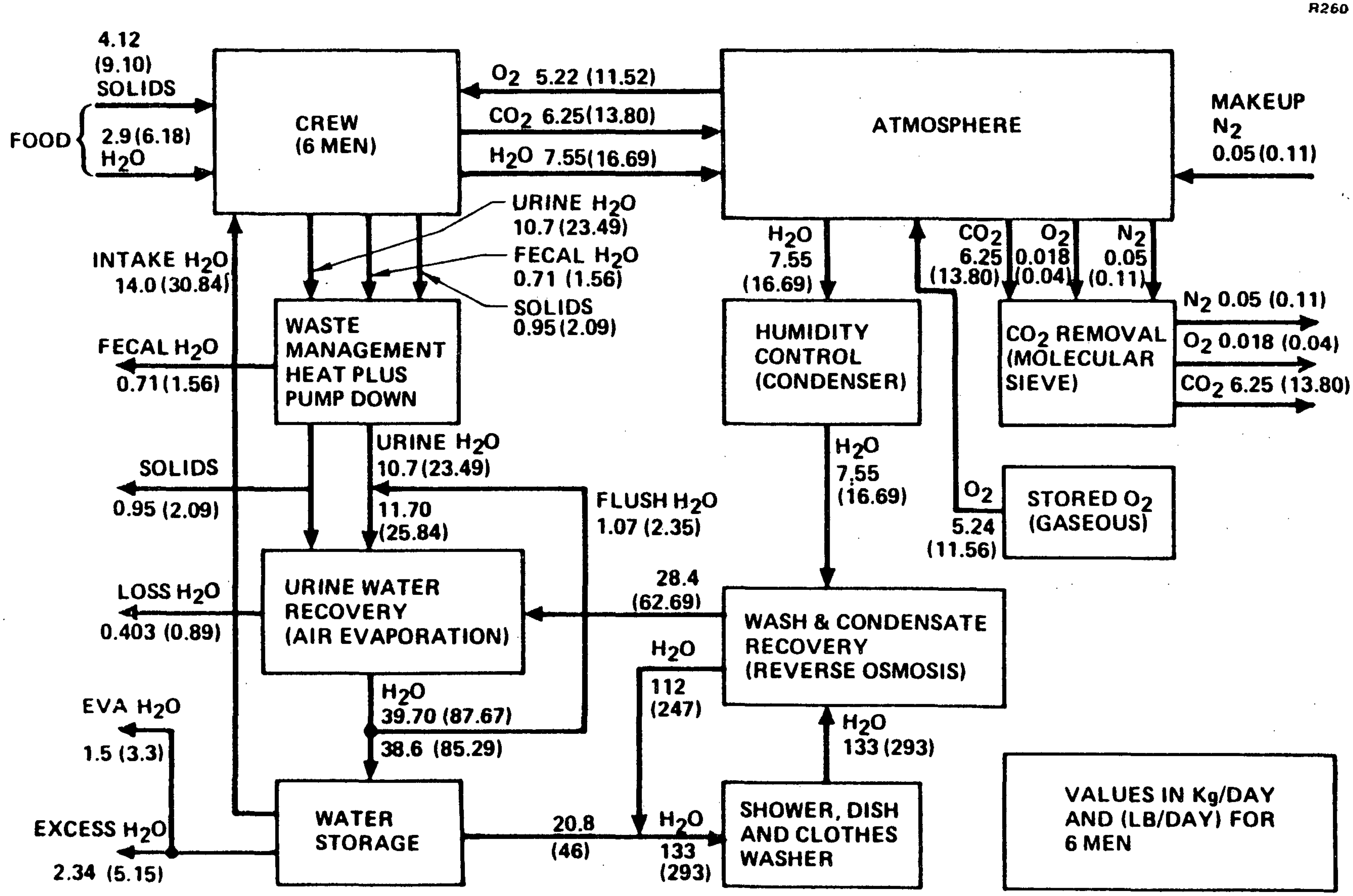

Figure 4.6-2 EC/LC Mass Balance for Modular Space Station

operation limited only by the amount of consumables onboard at the time of the emergency. With both units operating, the system easily accommodates a 12-man crew during the overlap period.

Separate thermal control loops are necessary so that any core module can be returned to Earth for refurbishment. If a single loop were provided for all core modules, removal or incorporation of a module could result in loss of thermal control for the entire ISS. However, to make full use of the multiple loop concept, an interchange loop is provided between core modules. This interchange water loop is normally inactive and is activated only when there is insufficient cooling in one of the individual modules. Heat is removed from the deficient module and transported to the remaining modules where it is rejected to space. The interchange loop also passes through critical electronics so that loss of the entire thermal control system in a module will not result in complete loss of the module.

Nearly all of the surface of the core modules is needed for heat rejection under design environmental conditions. For this reason, and because radiator failures may be difficult to repair, full redundancy is provided in the radiator circuitry. Segmentation and circuit isolation further protect against major thermal control system loss.

Table 4.6-5 tabulates the weight, volume, power, assembly dimensions, spares, and expendables for the EC/LS Subsystem.

A 96-hour pallet is provided which contains all essential EC/LS services, food, or emergency power for the crew. This assembly does not rely on any onboard system for support and is self contained for ease of location and movement throughout the Space Station. The pallet contains the following provisions: (1) oxygen, (2) water for crew intake and cooling, (3) food, (4) LiOH for CO_2 control, (5) a water boiler, and (6) miscellaneous medical and personal hygiene provisions. Two of these 3-man pallets are normally located in the GPL and one pallet is located in each attached module. During buildup, a pallet is also located in the power module before the docking of a Logistics Module.

Subsections 4.6.3.1 through 4.6.3.6 of this report present a more detailed description of the EC/LS Subsystem by assembly group.

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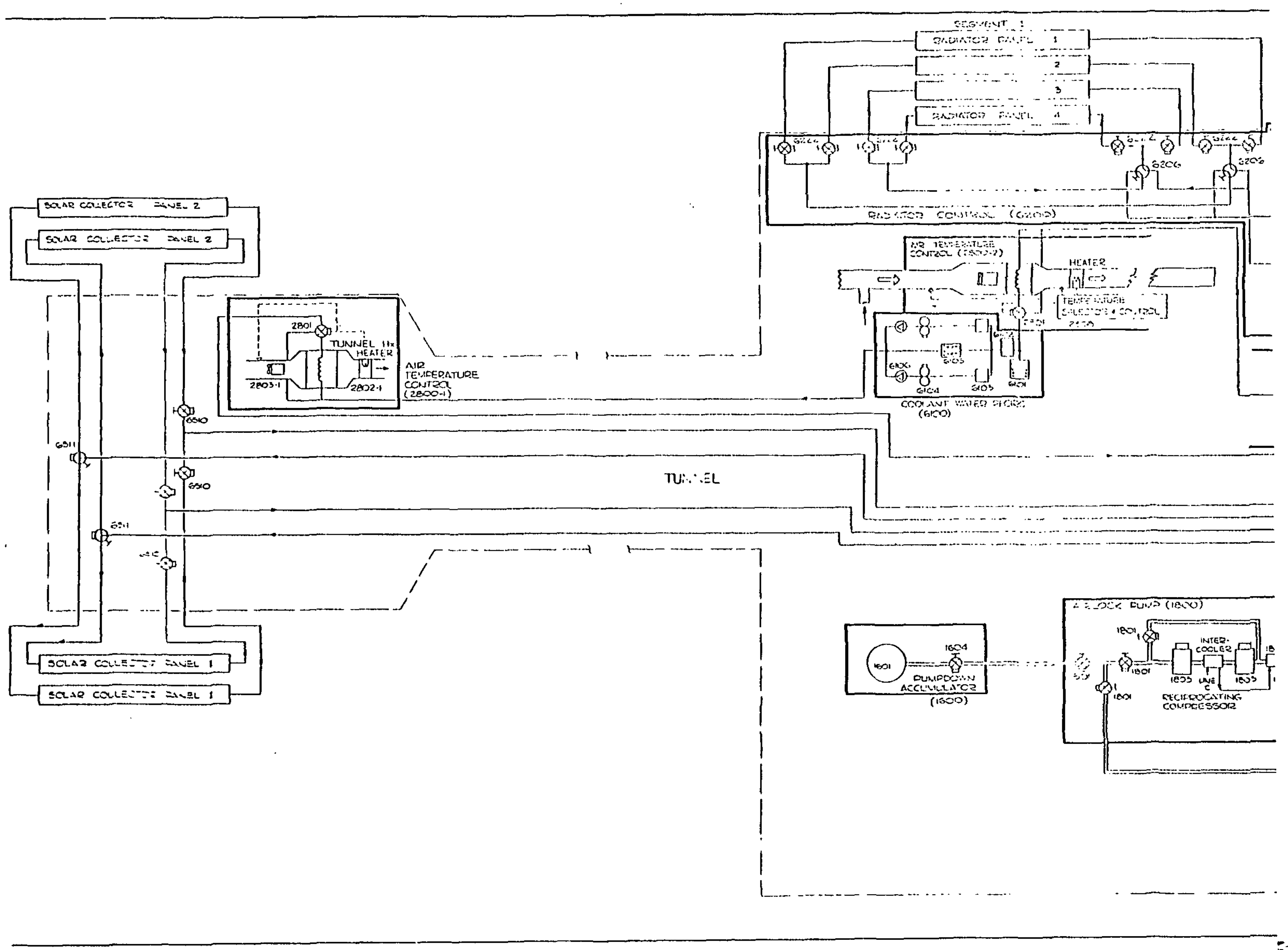

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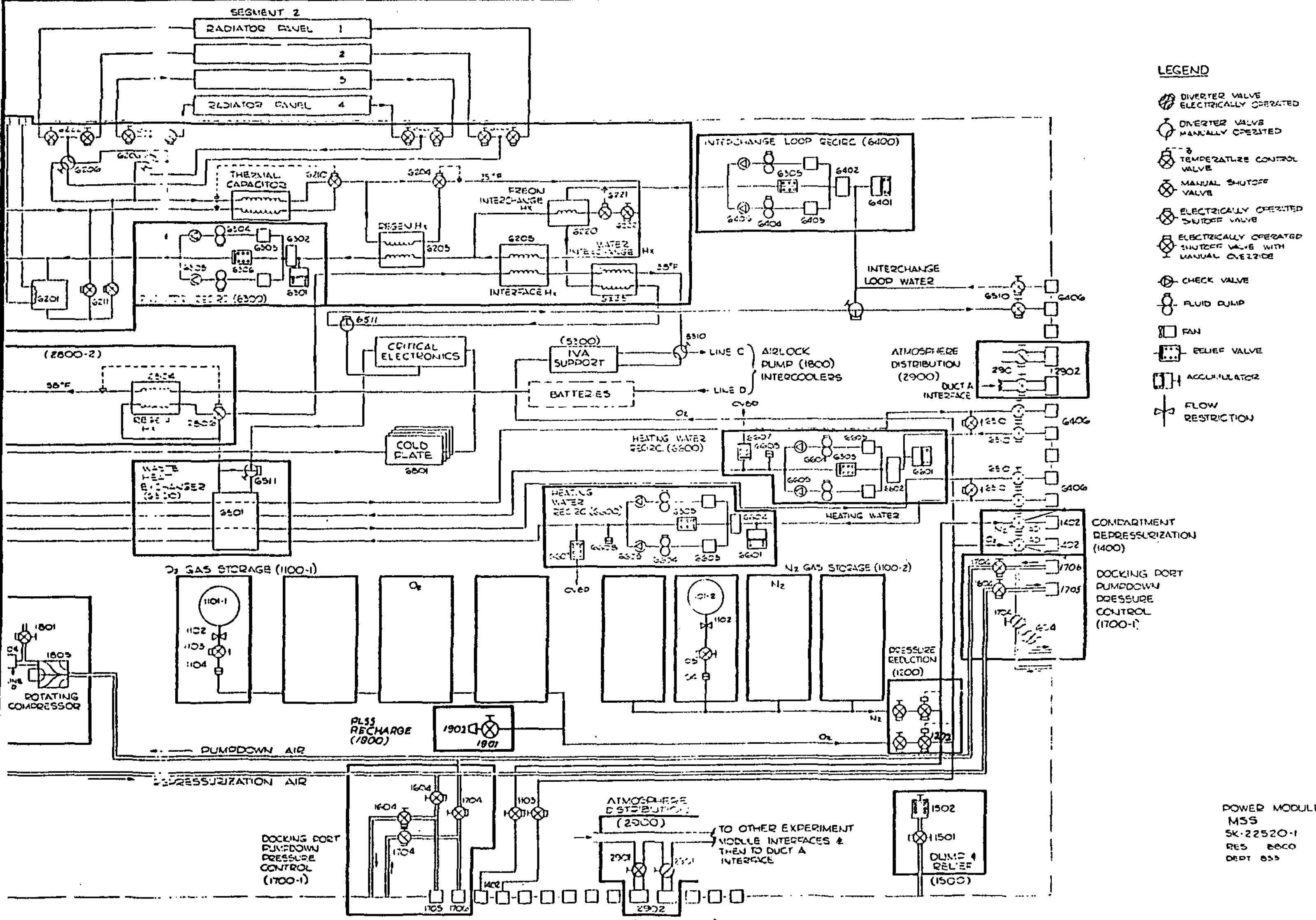

Figure 4.8-3 EC/LS Schematic for Power/Subsystems Module

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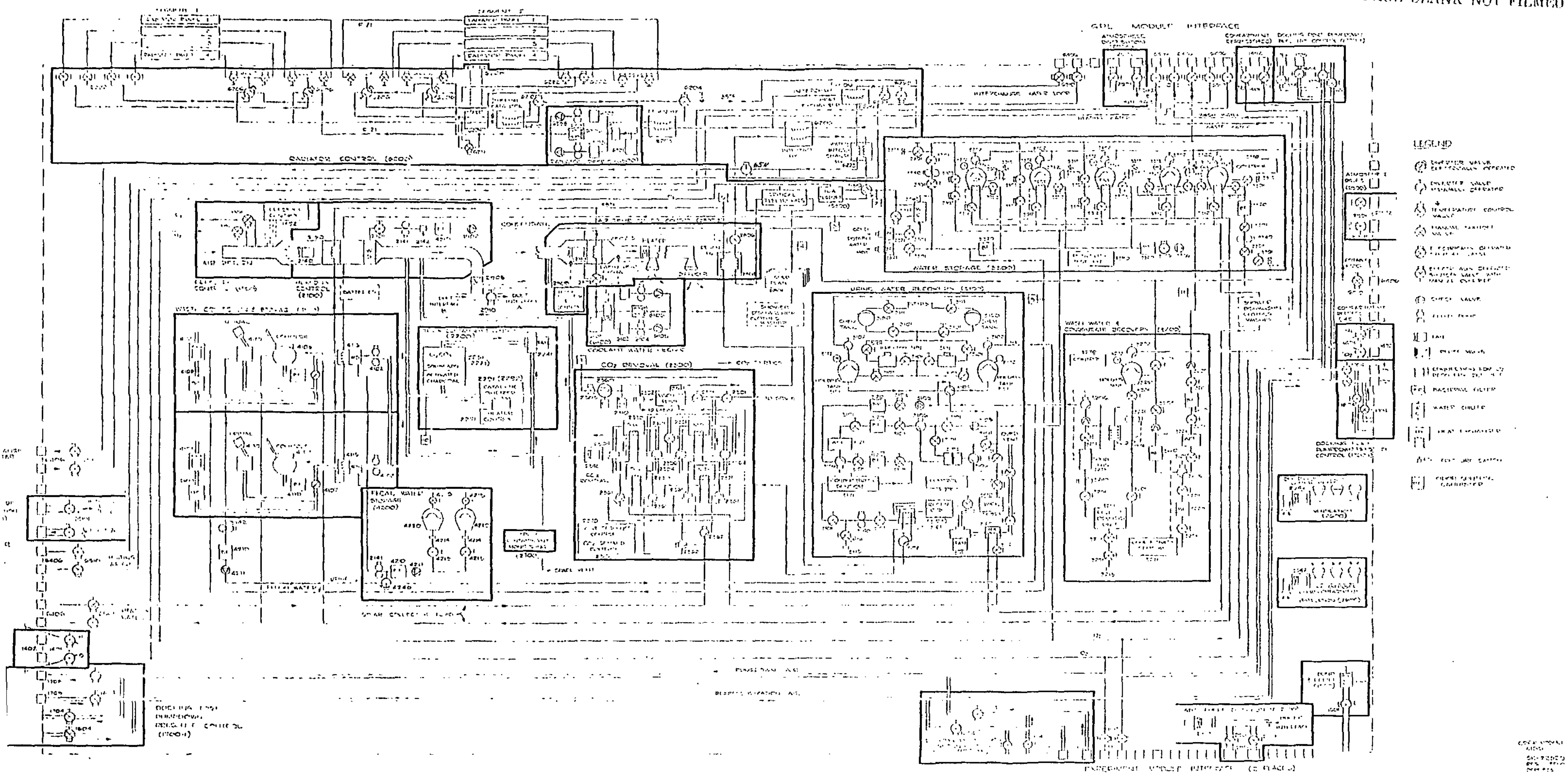

Figure 4.6-4 EC/LS Schematic for Crew/Operations Module

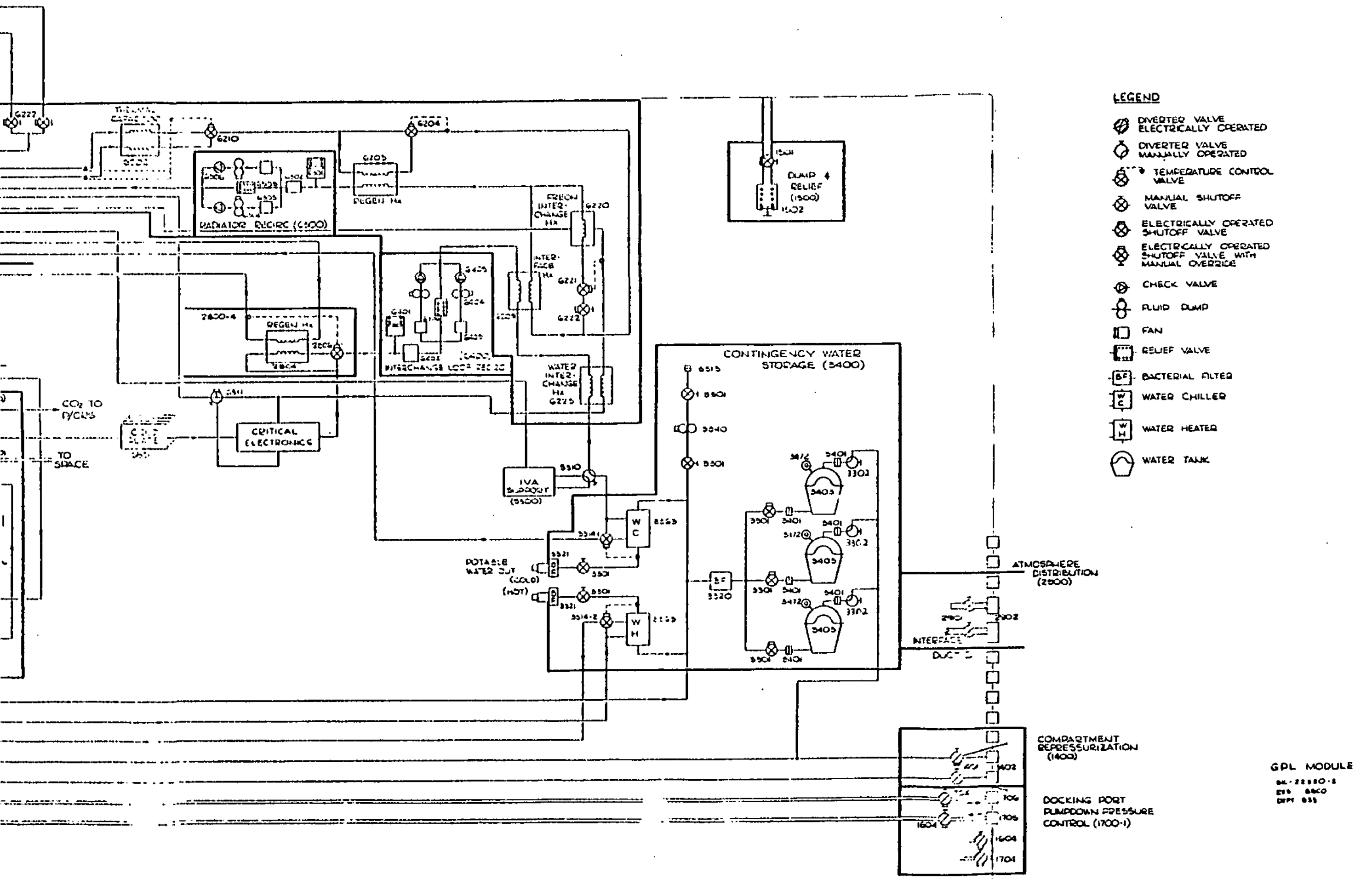

Figure 4.6-5 EC/LS Schematic for GPL Module

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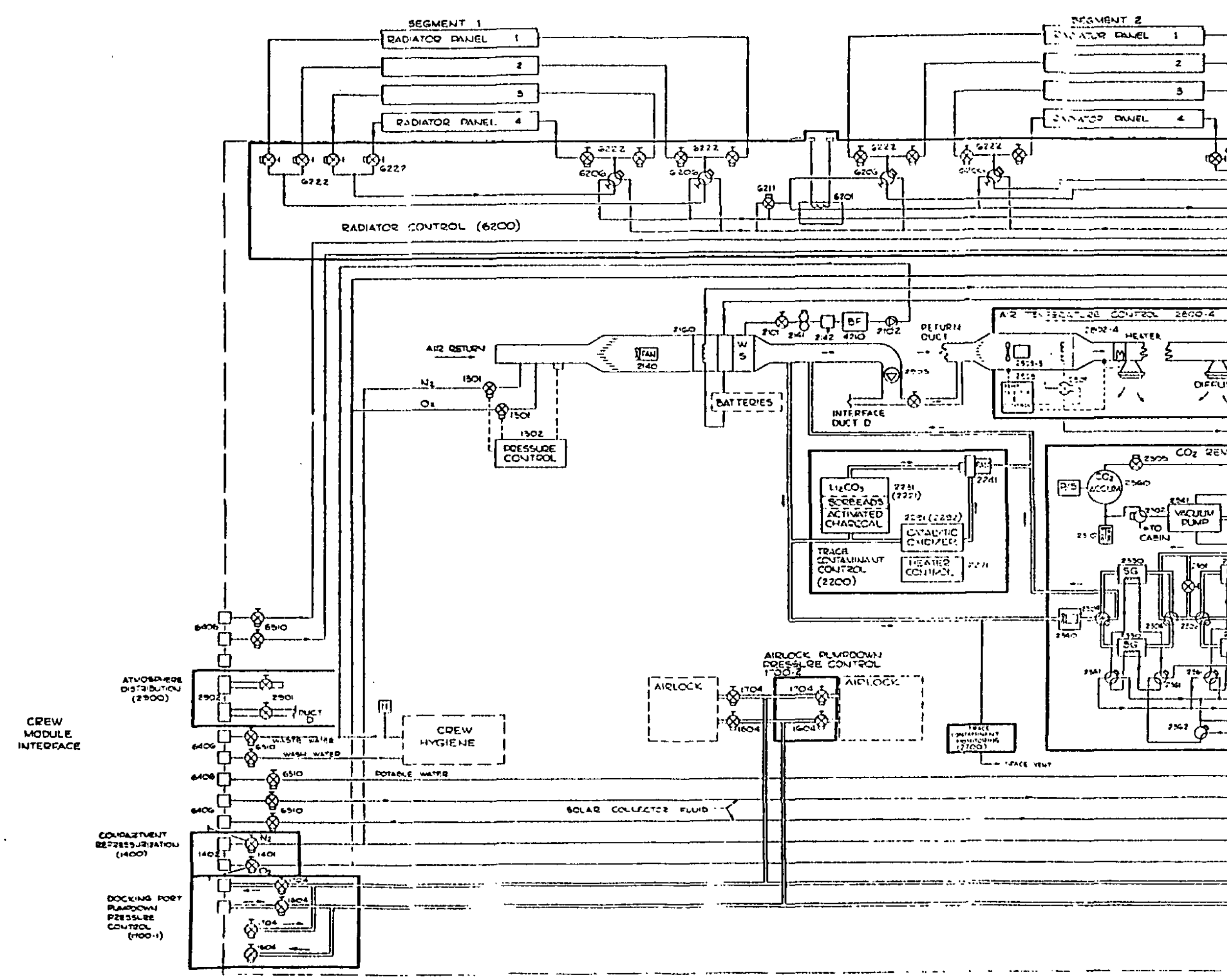

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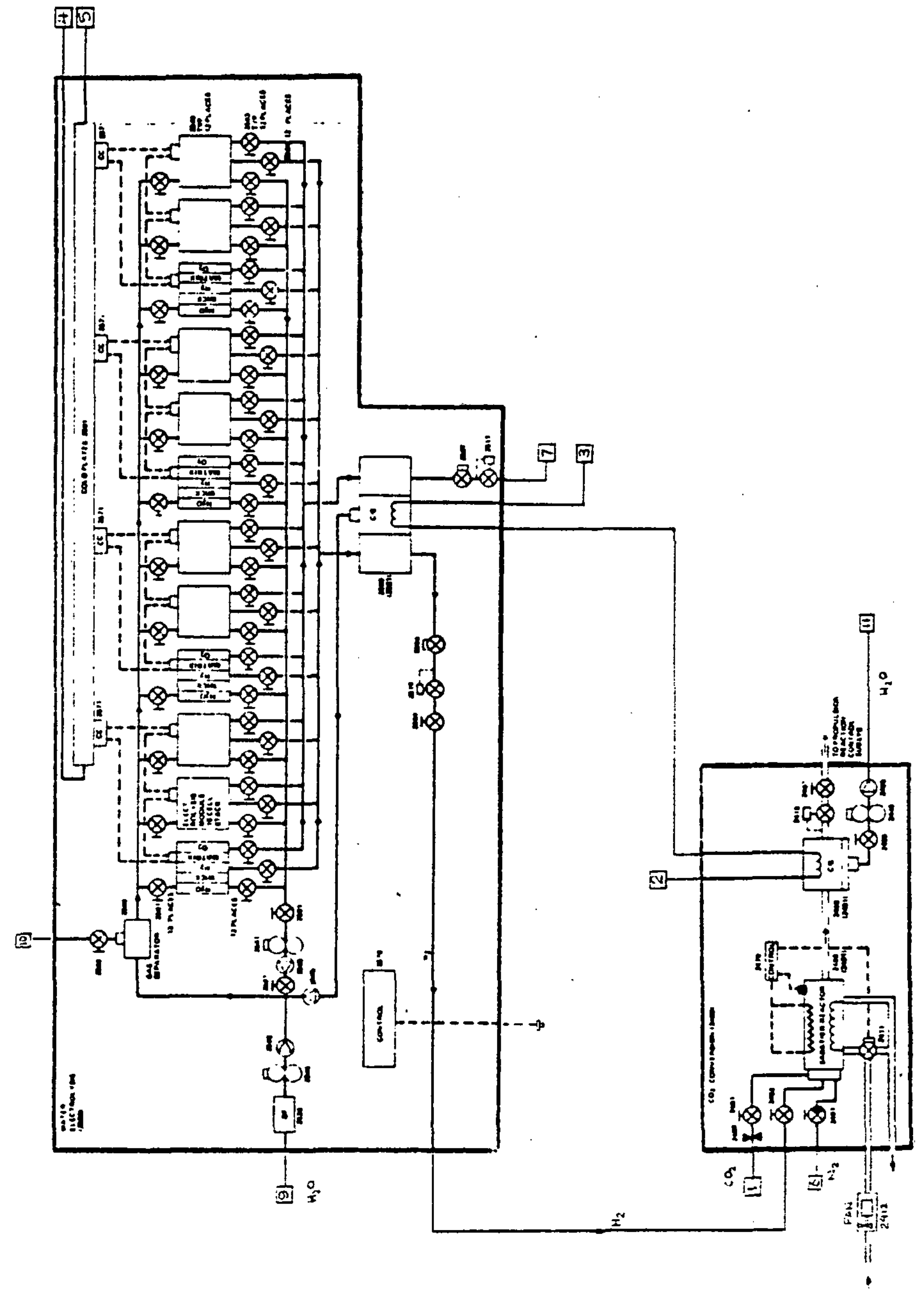

Figure 4.6-6 Recovery O_2 Retrofit Package

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4.6.3.1.1 Atmosphere Supply and Control

The major functions of the Atmosphere Supply and Control Group are as follows:

A. Provide for makeup O_2 and N_2, as required.

B. Provide a contingency supply of O_2 and N_2 for emergencies.

C. Maintain atmosphere pressure and composition control.

D. Provide for compartment pressurization and depressurization.

Figure 4.6-1 shows a breakdown of the assemblies which were designed to meet these functional requirements.

Oxygen and nitrogen storage requirements are shown in Table 4.6-6. Makeup gas consists of oxygen and nitrogen for pumpdown makeup and oxygen for crew metabolic makeup. This makeup gas is resupplied as high pressure gas at 2.06×10^4 kn/m^2 (3,000 psia) in tanks located in the Logistics Module. The gas is withdrawn from the Logistics Module at a reduced pressure of 410 kn/m^2 (60 psia) as required by the Space Station and distributed to the core modules through a common manifold running to all modules.

The contingency gas supply consists of O_2 and N_2 for compartment repressurization and a 30-day supply of crew metabolic O_2. This supply is stored as 2.06×10^4 kn/m^2 (3,000 psia) gas in tanks located in the power module (Figure 4.6-3). The oxygen requirement of 240 kg (530 lb) is contained in four O_2 gas storage assemblies (1100-1) consisting of tank and supporting connectors and valves. The nitrogen requirement of 231 kg (510 lb) is contained in four N_2 gas storage assemblies (1100-2) which are identical in design to the O_2 gas storage assemblies (1100-1). Pressure transducers located on each tank provide signals for quantity measurements and orifices are located in tank outlet lines to limit flow output.

The contingency gas supply is not normally used except during emergencies. If the supply is used, tank replacement is provided for with the disconnect 1104.

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Table 4.6-5

MODULAR SPACE STATION EC/LS ASSEMBLY CHARACTERISTICS (Page 1 of 4)

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Assembly (No. and Title)		No. Reqd	Location	Fixed Equipment Wt, kg (lb)	Fixed Equipment Vol, m^3 (ft^3)	Power (Average/Peak) (W) dc	Power (Average/Peak) (W) ac	Dimensions, cm (inches) $L_1 \times L_2 \times L_3$	Access Reqmts	Spare Wt, kg (lb) Onboard	Spare Wt, kg (lb) 30-Day Resupply	30-Day Expendables Wt, kg (lb)	30-Day Expendables Vol, m^3 (ft^3)	Expendable Description
1000	Atmosphere Supply and Control			1,204.9 (2,659.5)	3.90 (137.8)	20/NA	0/NA			43.1 (94.5)	1.1 (2.4)	370.0 (815.6)	0.73 (25.5)	
1100-1	O_2 Gas Storage	4	Power/Subsystems Module	522 (1,155)	1.02 (36)	0/10	0/0	85 cm (31 in.) sphere	A	5.5 (12)	0.1 (0.2)	352 (776)	0.69 (24.2)	162Kg (356 lb) O_2 gas at 20.6x1 KN/m^2 (3,000 psia)
1100-2	N_2 Gas Storage	4	Power/Subsystems Module	504 (1110)	1.06 (37.6)	0/10	0/0	85 cm (31 in.) sphere	A	5.5 (12)	0.1 (0.2)	18 (39.6)	0.04 (1.3)	8.22 (18.1 lb) N_2 gas at 20.6 x 10^3K N/m^2 (3,000 psia)
1200	Pressure Reduction	2	Power/Subsystems Module near gas storage	15.4 (34)	0.06 (2)	0/0	0/0	32 x 22 x 45 (12x8x16.5)	F, B, D	1.8 (4)	0	0	0	
300	Pressure Control	2	GPL and Crew/ Operations Module	16.3 (36)	0.06 (2)	20/30	0/0	65 x 16 x 32 (24 x 6 x 12)	D, F, B, A	8.2 (18)	0.6 (1.3)	0	0	
400	Compartment Repressurization	5	Docking ports between Core Modules	4.5 (10)	0.03 (1.0)	0/0	0/0	8 x 8 x 11 (3 x 3 x 4)	B, D, F	1.8 (4)	0	0	0	
500	Dump and Relief	3	Each Module near overboard duct	20.4 (45)	0.13 (4.5)	0/0	0/0	41 x 41 x 31 (15x15x11.5)	D, F, B, A	5.5 (12)	0.1 (0.3)	0	0	
1600	Pumpdown Accumulator	2	Power/Subsystems Module	38.1 (84)	1.24 (44)	0/0	0/0	112 cm (41 in.) sphere	A	1.4 (3)	0.1 (0.2)	0	0	
1700-1	Docking Port Pumpdown Pressure Control	10	Docking ports normally used, 4 in Power/Subsystems Module, 5 in Crew/Operations Module, and 1 in GPL Module	54.5 (120)	0.17 (6)	0/20	0/0	27 x 19 x 19 (10 x 7 x 7)	D, A, B, F	2.7 (6)	0	0	0	
1700-2	Airlock Pumpdown Pressure Control	3	Near isolation test facility in GPL	6.8 (15)	0.03 (1.2)	0/0	0/0	22 x 19 x 19 (8 x 7 x 7)	D, A, B, F	2.3 (5)	0	0	0	
1800	Airlock Pump	1	Power/Subsystems Module	22.7 (50)	0.10 (3.5)	0/0	170/2060 (1)	65 x 49 x 38 (24 x 18 x 44)	D, A, B, F	8.2 (18)	0.1 (0.2)	0	0	
1900	PLSS Recharge	1	Power/Subsystems Module	0.2 (0.5)	0	0/0	0/0	5.5 x 5.5 x 11 (2 x 2 x 4)	D, A, B	0.2 (0.5)	0	0	0	

Table 4.6-5

MODULAR SPACE STATION EC/LS ASSEMBLY CHARACTERISTICS (Page 2 of 4)

Assembly (No. and Title)		No. Reqd	Location	Fixed Equipment Wt, kg (lb)	Fixed Equipment Vol, m^3 (ft^3)	Power (Average/Peak) (W) dc	Power (Average/Peak) (W) ac	Dimensions, cm (inches) $L_1 \times L_2 \times L_3$	Access Reqmts	Spare Wt, kg (lb) Onboard	Spare Wt, kg (lb) 30-Day Resupply	30-Day Expendables Wt, kg (lb)	30-Day Expendables Vol, m^3 (ft^3)	Expendable Description
2000	Atmosphere Reconditioning			767.6 (1,692)	3.89 (137.6)	233/NA	1126/NA			224.2 (495)	2.9 (6.6)	19.2 (42.4)	0.05 (1.9)	
2100	Humidity Control	2	Crew/Operations Module and GPL	31.8 (70)	0.17 (6.0)	0/0	107/107	44 x 44 x 55 (16 x 16 x 20)	A, B, D	18 (40)	0.4 (0.8)	0.9 (2.0)	0 (0.1)	Wicks, water transfer discs
2200	Trace Contaminant Control	2	Crew/Operations Module and GPL	85.2 (188)	0.47 (16.6)	133/204	103/158	65 x 55 x 82 (24 x 20 x 30)	A, B, D	20.4 (45)	0.1 (0.3)	16.8 (37)	0.05 (1.7)	Charcoal, abso ent beds, filter
2300	CO_2 Removal	2	Crew/Operations Module and GPL	500 (1,102)	2.50 (88.5)	10/20	456/456	175 x 49 x 147 (64 x 18 x 54)	A, B, D	139 (307)	2.0 (4.5)	0	0	
2600	Ventilation	2	Crew/Operations Module, Crew Quarters area	5.5 (12)	0.14 (5.0)	0/0	75/150	82 x 32 x 32 (30 x 12 x 12)	A, B, D, E	2.3 (5)	0 (0.1)	0	0	
2700	Trace Contaminant Monitoring	2	Crew/Operations Module and GPL	30.0 (66)	0.06 (2.0)	90/90	0/0	49 x 22 x 32 (18 x 8 x 12)	A, B, D, F	10.0 (22)	0.1 (0.2)	1.5 (3.4)	0 (0.1)	Absorption columns
2800-1 2800-2 2800-3 2800-4	Temperature Control	3	Each Core Module	56.6 (125)	0.37 (13.0)	0/0	355/355	60 x 49 x 49 (22 x 18 x 18)	A, B, D	23.6 (52)	0.2 (0.5)	0	0	
								49 x 38 x 38 (18 x 14 x 14)	A, B, D					
								65 x 52 x 52 (24 x 19 x 19)	A, B, D					
								16 x 16 x 8 (6 x 6 x 3)	A, B, D					
2900-1 2900-2 2900-3	Atmosphere Distribution	3	In ducting between modules	58.5 (129)	0.18 (6.5)	0/0	30/30	Part of ducting	A, B, D, E	10.9 (24)	0.1 (0.2)	0	0	
3000	Water Management			702.5 (1,548)	3.36 (119)	64/NA	138/NA			125.7 (275)	0.9 (1.7)	30.3 (66.7)	0.09 (3.3)	
3100	Urine Water Recovery	1	Crew/Operations Module	134 (295)	0.99 (35)	18/24	96/128	164x65x115 (60x24x42)	A, B, D	73.5 (160)	0.5 (0.9)	11.5 (25.4)	0.06 (2.1)	Wicks, Charcoal, filters, pretreatment chemicals
3200	Wash Water and Condensate Recovery	1	Crew/Operations Module	38.5 (85)	0.40 (14)	24/32	27/36	131x63x120 (48 x 23 x 44)	A, B, F	23.6 (52)	0.1 (0.2)	14.4 (31.7)	0.02 (0.7)	Separator men branes cartrid filters and charcoal

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Table 4.6-5

MODULAR SPACE STATION EC/LS ASSEMBLY CHARACTERISTICS (Page 3 of 4)

	Assembly (No. and Title)	No. Reqd	Location	Fixed Equipment Wt, kg (lb)	Fixed Equipment Vol, m^3 (ft^3)	Power (Average/Peak) (W) dc	Power (Average/Peak) (W) ac	Dimensions, cm (inches) $L_1 \times L_2 \times L_3$	Access Reqmts	Spare Wt, kg (lb) Onboard	Spare Wt, kg (lb) 30-Day Resupply	30-Day Expendables Wt, kg (lb)	30-Day Expendables Vol, m^3 (ft^3)	Expendable Description
3300	Recovered Water Storage	1	Crew/Operations Module	68.0 (150)	1.07 (38)	22/22	15/20	164 x 65 x 125 (60 x 24 x 46)	A, B, D	23.2 (51)	0.2 (0.4)	3.4 (7.4)	0.01 (0.4)	Bacteria filters
3400	Contingency Water Storage	1	GPL Module	462.0 (1,018)	0.90 (32)	0/20	0/20	82 x 82 x 164 (30 x 30 x 60)	A, B, E	5.4 (12)	0.1 (0.2)	1.0 (2.2)	0 (0.1)	Bacteria filters
4000	Waste Management			60.0 (132)	0.27 (9.4)	0/NA	78.6/NA			4.0 (8.7)	0.3 (0.6)	8.2 (18)	0.03 (1.2)	
4100	Waste Collector and Storage	2	Crew/Operations Module Waste Management	52.7 (116)	0.23 (8.0)	0/0	78.6/200	60 x 49 x 49 (22 x 18 x 18)	A, B, E, F	3.0 (6.5)	0.3 (0.5)	5.5 (12)	0.01 (0.4)	Wipes, liner filters, wate transfer disk
4200	Fecal Water Storage	1	Crew/Operations Module Waste Management Area	7.3 (16)	0.04 (1.4)	0/0	0/0	27 x 44 x 36 (10 x 16 x 13)	A, B, F	1.0 (2.2)	0 (0.1)	2.7 (6)	0.02 (0.8)	Fecal water, storage tank
5000	IVA/EVA			777.9 (1,715.6)	1.97 (70.0)	0/NA	0/NA			5.4 (12.0)	0	10.7 (23.6)	0.09 (3.2)	
5100	Portable Life Support System	4	2 in GPL and 2 in Power/Subsystems Module	187.0 (412)	0.80 (28.4)	0/0	0/0	112 x 87 x 49 (14 x 32 x 18)	None	0	0	10.7 (23.6)	0.09 (3.2)	LiOH, Batteries
5200	Umbilical Life Support	4	2 in GPL and 2 in Power/Subsystems Module	83.4 (184)	0.34 (12)	0/124	0/0	49 x 65 x 27 (18 x 24 x 10)	B, C, F	0	0	0	0	
5300	IVA Support	3	1 in each core module	13.6 (30)	0.03 (1.2)	0/0	0/0	19 x 27 x 27 (7 x 10 x 10)	A, B, C, D, F	4.5 (10.0)	0	0	0	
5400	Portable O_2 Bottles	6	Crew/Operations Module	13.9 (30.6)	0.04 (1.4)	0/0	0/0	19 x 19 x 22 (7 x 7 x 8)	None	0.9 (2.0)	0	0	0	O_2 at 20.6 N (300 psi) for event
5500	Emergency EC/LS Pallet	3	2 in GPL, 1 in Power/Subsystems Module	480.0 (1,059)	0.76 (27)	0/0	0/0	100 x 65 x 49 (36 x 24 x 18)	B	0	0	0	0	
6000	Thermal Control			336.9 (742)	0.58 (20.2)	0/NA	468/NA			119.4 (263)	0.5 (1.1)	0	0	
6100	Coolant Water Recirculation	3	Each core module	13.6 (30)	0.03 (0.9)	0/0	180/180	22 x 22 x 22 (8 x 8 x 8)	A, B, F, D	4.5 (10)	0.1 (0.2)	0	0	
6200	Radiator Control	3	Each core module	210.0 (462)	0.32 (11.2)	0/10	0/0	68 x 41 x 33 (25 x 15 x 12)	A, B, F, D	93.0 (205)	0.2 (0.5)	0	0	

Table 4.6-5

MODULAR SPACE STATION EC/LS ASSEMBLY CHARACTERISTICS (Page 4 of 4)

Assembly (No. and Title)		No. Reqd	Location	Fixed Equipment Wt, kg (lb)	Fixed Equipment Vol, m^3 (ft^3)	Power (Average/Peak) (W) dc	Power (Average/Peak) (W) ac	Dimensions, cm (inches) $L_1 \times L_2 \times L_3$	Access Reqmts	Spare Wt, kg (lb) Onboard	Spare Wt, kg (lb) 30-Day Resupply	30-Day Expendables Wt, kg (lb)	30-Day Expendables Vol, m^3 (ft^3)	Expendable Description
6300	Radiator Circulation	3	Each core module	13.6 (30)	0.06 (2.1)	0/0	228/228	82 x 55 x 33 (30 x 20 x 12)	A, B, F, D	4.5 (10)	0.1 (0.2)	0	0	
								87 x 60 x 36 (32 x 22 x 13)						
6400	Interchange Loop Recirculation	1	Power/Subsystems Module	5.5 (12)	0.02 (0.8)	0/0	0/90	38 x 33 x 22 (14 x 12 x 8)	A, B, F, D	2.7 (6.0)	0	0	0	
6500	Waste Heat Exchanger	1	Power/Subsystems Module	2.7 (6.0)	0.02 (0.6)	0/0	0/0	33 x 22 x 30 (12 x 8 x 11)	A, B, D	2.7 (6.0)	0	0	0	
6600	Heating Water Recirculation	2	Power/Subsystems Module	5.5 (12.0)	0.02 (0.8)	0/0	60/60	38 x 33 x 22 (14 x 12 x 8)	A, B, F, D	2.7 (6.0)	0.1 (0.2)			
6800	Cold Plates	19	Equipment bays in core modules	86.0 (190)	0.11 (3.8)	0/0	0/0	(82 x 55 x 1.4) 30 x 20 x 1/2	F	9.3 (20)	0	0	0	
										521.8	5.7	438.4	0.99	
				3,849.8	139.7	317/NA	1,810.6/NA			(1,148.2)	(12.4)	(966.3)	(35.1)	
				(8,489.1)	(494.0)									

(1) The air lock pump is periodically used and is scheduled for use during low power demand periods. Power for this assembly is therefore not included in the totals.

Table 4.6-6
GASEOUS STORAGE REQUIREMENT

Item	On Board Storage (lb)	Resupply (lb/30 days)
Compartment repressurization O_2	184	
Compartment repressurization N_2	510	
Crew contingency O_2	346	
Crew metabolic O_2		346.0
Pumpdown makeup O_2		6.5
Pumpdown makeup N_2		18.1
EVA O_2		3.6
Totals	1,040	374.2

Normally the tank shutoff valves are closed so the contingency gas supply is not used. If gas is required from the contingency supply, the normally-closed tank shutoff valves (1103) are opened. The high-pressure gas from the tanks is reduced to a working pressure of 410 kn/m^2 (60 psia) in the pressure reduction assembly (1200). The assembly also contains shutoff valves to facilitate maintenance on the pressure regulator (1202). Lines and downstream components are protected from overpressure due to a failed pressure regulator by a pressure relief provision in the regulator. Pressure transducers are located in the supply manifold to measure supply gas pressure. Temperature sensors are also located upstream of the pressure reduction assembly to warn against too cold a gas supply which can occur during rapid tank depletion. Downstream regulators and equipment can be damaged and condensation or frosting can occur on the supply lines at very low temperatures.

Normal cabin makeup O_2 and N_2 is supplied to the cabin by the pressure control assembly; one is located in each compartment. An electronic control (1302) provides the control logic for gas addition based on atmosphere composition sensor signals. The control actuates solenoid valves (1301) which admit controlled amounts of makeup gas. The amount of N_2 added to the cabin is determined by the pressure control to provide a backup to tankage quantity sensing and to give early warning of excessive cabin leak rates. This is accomplished by adding makeup gas with timed impulses of the solenoid valves. Because upstream pressure is regulated, a set amount of gas is added through a flow restrictor in the solenoid valve each time it is actuated. The number of impulses along with the valve flow characteristics establish gas use rates. These data are accessed from the onboard computer and presented in the multipurpose display on the primary control console upon demand.

Diagnostic instrumentation is provided in the pressure control assembly consisting of (1) current sensors measuring controller output and (2) flow sensors on the O_2 and N_2 lines.

Compartment repressurization valves (1400) are located at module interfaces and provide for (1) core module repressurization from either side of the core module hatch and (2) O_2 and N_2 line shutoff capability when no module is attached or to isolate a module due to downstream equipment malfunction.

Excess compartment pressure is prevented by the dump and relief assembly (1500); one of these assemblies is located in each compartment. In addition, this assembly allows cabin atmosphere dump for fire suppression or to purge a contaminated atmosphere. The dump feature of this assembly is activated by manual operation of a switch on the control panel. A shutoff valve (1501) is located in the overboard duct to prevent loss of atmosphere in the event of a failed valve and to allow replacement of the valves. Both the shutoff valve and the dump and relief valve are electrically actuated from another compartment and have manual override features. A flow sensor located in the overboard duct senses excessive leakage.

An airlock pump assembly (1800) is provided to reclaim atmosphere from airlocks, adapter sections, experiment modules, and the isolation test facility which are periodically pressurized and depressurized. The air is pumped into the accumulator (1600) if the pumpdown volume is 23.2 m^3 (820 ft^3) or less, which is the volume of the isolation test facility.

Attached modules which have larger pumpdown volumes will have self-contained accumulators but will use the onboard pumpdown facilities. The onboard pumpdown accumulator assembly (1600) is located in the Power/Subsystems Module and contains a shutoff valve (1604) to isolate the accumulator in the event of equipment failure or long-term gas storage. Accumulator pressure and temperature are sensed as required for pumpdown operation control.

The airlock pump assembly (1800) consists of three compression stages separated by intercoolers (1804). The firststage pump (1805) is a heli-rotor design and the remaining stages are reciprocating compressors (1803). During initial phases of operation, the reciprocating compressors are bypassed and pumping is performed by the heli-rotor unit. At higher compression ratios, all three units are used.

The assembly contains shutoff valves (1801) to prevent backflow through the compressors when inoperative. Shutoff valves are also provided so that air can be pumped to either the onboard accumulator (1600) or the accumulators in the attached modules via the pressurization line.

A flow sensor in the discharge line monitors performance of the pumps. Docking adaptors, airlocks, and experiment compartment pumpdown and repressurization are controlled by the pumpdown pressure controls (1700-1 and 1700-2). Two versions of the pumpdown pressure control are required; one for airlocks and experiment compartments and one for docking ports. They both serve the same function; however, the equipments required differ somewhat. In the case of airlocks, two item 1704 valves are provided, one for pumpdown and one for repressurization. The assembly for docking ports provides four valves; one pair (1604 and 1704) to service the volume between the Space Station and the experiment module and another pair to service the modules themselves. In addition, the docking port pumpdown pressure control contains the disconnects (1705 and 1706) for connection to the docked vehicle. Orifices are provided for both assembly types to limit the rate of repressurization.

Instrumentation is provided in each depressurizable volume for pressure and temperature to aid in the pumpdown operation.

Recharging of the Portable Life Support System (PLSS) oxygen tank is provided by PLSS recharge assembly (1900). It contains a manually-operated shutoff valve (1901) and disconnect (1902) which mates with PLSS O_2 tank to pressurize it to 5,850 kn/m^2 (850 psia).

4.6.3.1.2 Atmosphere Reconditioning

The functions of the Atmosphere Reconditioning Assembly Group are as follows:

A. Control atmosphere humidity and temperature.
B. Remove and collect CO_2 from the atmosphere.
C. Provide for ventilation.
D. Control and monitor atmospheric trace contaminants.

The equipment necessary to accomplish these functions have been grouped into assemblies as shown in Figure 4.6-1. Component level schematics for the 3-core modules are shown in Figure 4.6-3, 4.6-4, and 4.6-5.

Atmosphere humidity is controlled by two assemblies located in the Crew/ Module and the GPL Module. The unit located in the Crew/Operations Module dehumidifies air in the Crew/Operations Module, Power/Subsystems Module and any attached modules. The GPL unit normally dehumidifies air only in the GPL Module.

Humidity control is effected by drawing process gas into the condensor (2160) where the process air temperature is cooled to about 10°C (50°F) by the cold circulating water on the liquid side of the condensor. This temperature is below the cabin dew point. Excess moisture is condensed from the process air, separated from the air stream by a face type wick separator and pumped to the water management assembly group for processing.

Each humidity control assembly has sufficient capacity to remove 1,330 watts (4,535 Btu/hr) which is comparable to the design point for the Space Station Prototype Program. Allowances are included for the crew at high metabolic loads in a warm 24°C (75°F) cabin, crew equipment such as showers and washers at 386 watts (1,313 Btu/hr) and 306 watts (1,042 Btu/hr) from experiments.

Separate air temperature control assemblies (2800) are provided in each module which remove the air heat load to maintain the selectable temperature at 18.4 to 29.4°C (65 to 85°F). Cabin air is passed through a liquid-to-air heat exchanger where the air is cooled by the circulating cooling water. The cabin temperature is controlled by the temperature control valve (2801) which bypasses cooling water based on a signal from the temperature selector and control (2808). Input to the controller originates from the temperature sensors located in the process gas duct.

A constant circulating water temperature of 14.4°C (58°F) is provided at the cabin heat exchanger inlet to avoid inadvertent moisture condensation in the process gas. The temperature control valve (2806) bypasses sufficient warm circulating water through the regenerative heat exchanger (2804) to maintain the control water temperature into the temperature control assembly.

The air temperature control assemblies have sufficient cooling capacity to remove crew thermal loads, EC/LS calculated sensible loads, and equipment sensible loads.

Where at all practical, electrical loads will be liquid cooled since liquid cooling represents a smaller penalty than air cooling. It is estimated that 80 percent of all electrical heat loads other than EC/LS loads can be removed by liquid cooling.

Trace contaminants are controlled by the trace contaminants and odor control assembly (2200); one of which is located in the GPL and another in the crew module. Process gas from the condenser outlet is passed through the non-regenerable charcoal cannister (2231) where heavy molecular weight contaminants, such as hydrocarbons, are removed. The fan (2241) provides the high flow 2.54 m^3/min (90 cfm) for the charcoal canister and a lower flow 0.156 m^3/min (5.5 cfm), for the catalytic oxidizer (2291). Carbon dioxide, hydrogen, and methane are oxidized in the catalytic oxidizer to produce water and other products which can be removed by other EC/LS equipment.

In addition to regenerable charcoal, the canister (2231) contains copper sulfate beads to remove ammonia and lithium carbonate sorbent to remove acid gasses. Removal of these products prevents poisoning of the catalyst bed. An on-off heater control (2271) maintains catalytic bed temperature at the design temperature of 371°C (700°F).

Molecular sieves are used to remove and control the level of CO_2 in the atmosphere. One assembly is located in the Crew/Operations Module and one assembly is located in the GPL Module. Each unit is capable of removing 0.354 kg/hr (0.78 lb/hr) of CO_2 which corresponds to 2 men at 235 watts (800 Btu/hr) and 4 men at 161 watts (550 Btu/hr). The units are completely independent from one another; each unit is serviced by different process heat and coolant water loops. This arrangement reduces the likelihood of losing both CO_2 removal assemblies at one time.

An isometric view of the CO_2 removal assembly is shown in Figure 4.6-7. Air drawn from downstream of the humidity control assembly by the process flow fan (2340) passes through a liquid-cooled, adsorbing desiccant bed (2330) where the stream is dried to a dewpoint of approximately -65°C (-85°F). The air continues through a liquid-cooled, absorbing molecular sieve canister (2331), where CO_2 is removed by adsorption on zeolite. Effluent air returns to the cabin through the desorbing desiccant canister where desorption of the contained water rehumidifies the air and regenerates the desiccant bed.

The silica gel beds operate on a 34-minute adsorption, 34-minute desorption time period. For approximately 28 minutes each orbit, the silica gel beds are inoperative.

The components remaining are engaged simultaneously in recovering carbon dioxide adsorbed previously. The canister that had been adsorbing carbon dioxide from cabin air is isolated from the other canisters by valves (2306) and is ready for desorption, which is a sequenced operation. In the first phase, atmospheric gas filling the void volume in the isolated desorbing zeolite canister is first returned to the cabin by the vacuum pump (2341) through valve (2302). The accompanying reduction in canister pressure causes partial desorption of air and CO_2. This sequence is necessary to assure delivery of high-purity CO_2 to the accumulator (2360) during the next phase. The

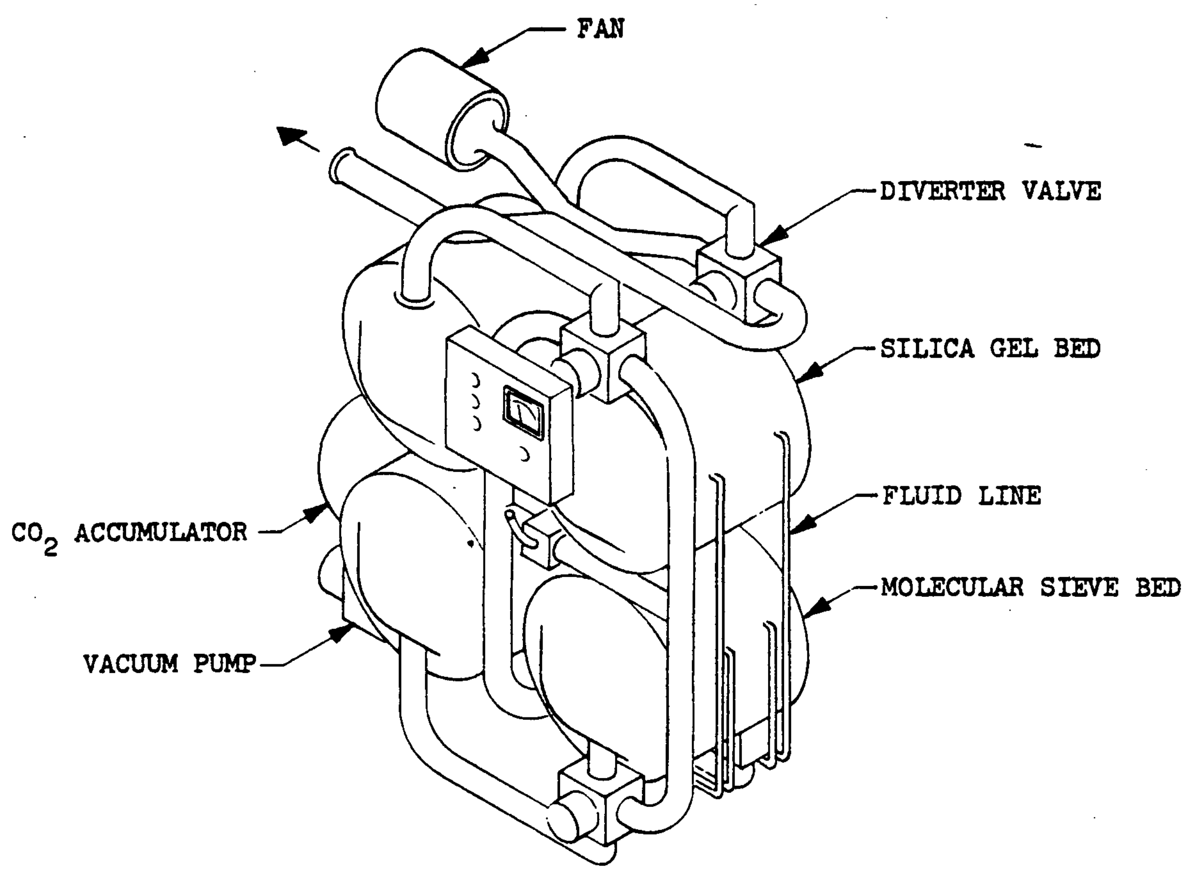

Figure 4.6-7 Molecular Sieve CO_2 Removal System

molecular sieve beds operate on a 68-minute adsorption, 68-minute desorption cycle. For approximately 28 minutes each orbit, the beds are inoperative.

In the second phase of the operation, the vacuum pump discharge is diverted into the accumulator by a solenoid operated valve (2302). The compressor maintains reduced pressure in the desorbing zeolite canister and transfers the carbon dioxide to the accumulator as it is desorbed. The heat required for the desorption process is supplied by the 132°C (270°F) water (process heat). Near the end of the desorption cycle, the molecular sieve bed is pre-cooled before adsorption. Assembly operation is controlled by a timer (2370) that cycles fluid valves (2361 and 2362) and gas valves (2306 and 2304) in an automatic, predetermined cycle.

The demand control (2371) senses the partial pressure of CO_2 in each module through sensors located in assembly 2100 and operates the assembly that is located in the module with the highest CO_2 partial pressure. After the timed cycle is completed, the process is repeated. An overboard dump line is provided through valve 2301 to desorb the molecular sieve canister to space if desired. Electrical heaters are provided in both the silica gel and molecular sieve beds because periodic bakeout may be required.

A dew point sensor provides information as to the amount of water vapor in the gas stream during startup of CO_2 concentrator after long-term shut down. When the gas stream is adequately dry, the bypass valve (2301) is closed and the flow is directed through a molecular sieve bed for CO_2 removal.

Atmosphere ventilation is provided to ensure adequate velocity throughout the Space Station and to prevent buildup of gas products or temperature in any region. A velocity of 0.1 to 0.25 m/sec (20 to 50 ft/min) is provided by the main air distribution system in each module which distributes cooled air from the air temperature control assembly (2800) and discharges it through diffusers. The cooled air flow along with the diffusion-induced flow is sufficient to maintain the required velocity range throughout the Space Station. This distribution system also provides sufficient air mixing to avoid any significant buildup of CO_2, humidity, temperature, or trace contaminants within each module. Electrical heater units are provided in the main air distribution duct just downstream of the air temperature control assemblies to provide atmosphere heating during low air-cooling conditions.

Some special ventilation is needed in the crew quarters because no atmosphere mixing can occur with the doors closed. The crew quarters located in the crew module require high ventilation flow but low conditioning flow because the heat load is small. Normally, the preferred crew quarters temperature will be no higher than the average deck temperature since the crew will be at a low metabolic rate in that area. Because of the long periods of time at low heat load, the crew quarters temperature will approach the supply air temperature which can be as low as 15.5°C (60°F); therefore it cannot be added directly to the crew quarters. To maintain the proper crew quarter's temperature, air is drawn from the main deck area with a fan and distributed to the crew quarters by ducting and diffusers. Each unit supplies 14.1 m^3/min (500 cfm) to three crew quarters, which is sufficient flow to provide normal crew cooling with a 1.8°C (2°F) temperature rise in the gas stream. This flow rate is also sufficiently large to prevent appreciable buildup of crew metabolic products in the crew quarters. The reconditioned air from the CO_2 removal assembly (2300), the trace contaminant control assembly (2200) and the humidity control assembly (2100) is expelled into the main distribution system which distributes the air throughout the Space Station. The air returns are located in the hygiene and galley areas so that the air movement towards those areas will prevent odors from escaping to the remainder of the station.

The atmosphere distribution assembly (2900) distributes air to the various modules serviced by the atmosphere purification equipment in each EC/LS. Normally the Crew/Operations Module atmosphere purification assemblies service the Power/Subsystems Module, the Crew/Operations Module and any attached modules (excluding the GPL). The GPL assemblies normally service only the GPL but can also service the entire ISS during contingency operation.

The flow arrangement for purified air from the Crew/Operations Module is of particular significance and this is shown in Figure 4.6-8. Values given in the figure for dew point temperature and CO_2 partial pressure represent design

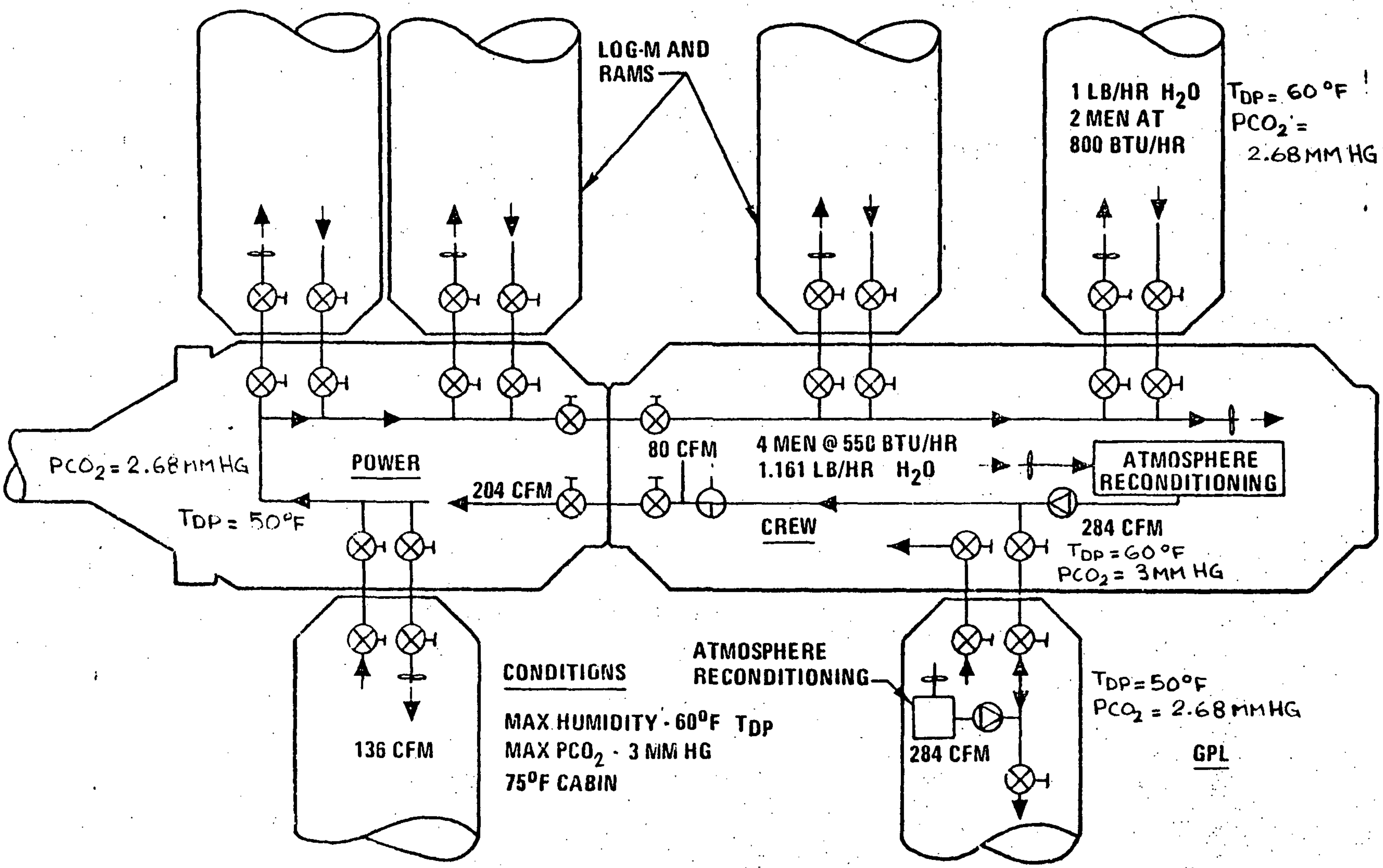

Figure 4.6-8 Mode Ventilation–Design Point

point conditions for humidity control and CO_2 removal. The design point condition for atmosphere distribution is based on 2 crewmen located in an attached module working at 234 watts (800 Btu/hr). Equipment humidity loads assumed in the attached module are 360 watts (1, 042 Btu/hr) and in the Crew/Operations Module are 386 watts (1, 313 Btu/hr). Purified air is delivered to the modules in a series arrangement. This approach minimizes the total air which must be processed and delivered. It also minimizes the required capacity of atmosphere purification equipment because the series arrangement results in a higher concentration of H_2O vapor and CO_2 entering the atmosphere purification equipment.

Although the design point for atmosphere distribution assumes two men at high metabolic load in an attached module, a greater number of crewmen can be accommodated in one attached module if they are working at lower levels. Additionally, there is sufficient flow to the Power/Subsystems Module and the attached modules so that the entire crew can be in those areas working at their average metabolic rates without exceeding design atmosphere conditions. The values given assume worst case steady-state conditions; it would take time for CO_2 and water vapor to buildup to the maximum allowable level under higher crew and equipment loads.

Figure 4. 6-9 shows contingency operation when a reconditioning unit becomes inoperative in one of the core modules. If the Crew/Operations Module unit is inoperative the atmosphere distribution duct valves are opened between compartments and the shutoff valve is closed in the GPL distribution duct which is normally used. Backflow of air through the Crew/Operations Module reconditioning unit is prevented by a check valve. The atmosphere leaving the Power/Subsystems Module passes through the Crew/Operations Module and then to the GPL carrying the generated water vapor and CO_2 back to the GPL reconditioning unit.

If the GPL reconditioning unit fails, the atmosphere ducting valves are opened between the GPL and Crew/Operations Modules and a portion of the reconditioned air from the Crew/Operations Module is directed to the GPL. Backflow is prevented through the GPL reconditioning unit by a check valve. The flow

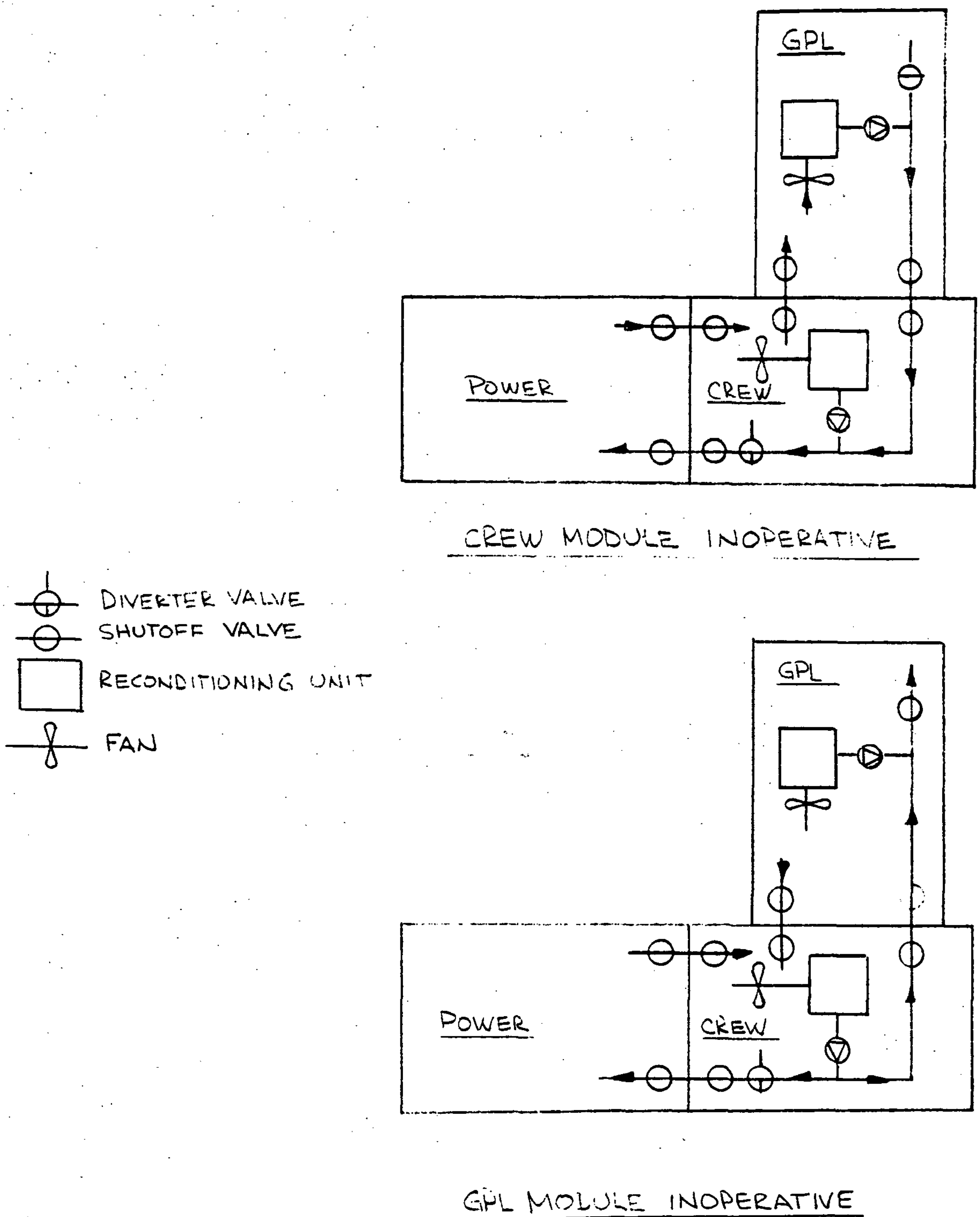

Figure 4.6-9 Atmosphere Interchange Design During Inoperative Atmosphere Reconditioning Unit

split between the GPL and Crew/Operations Modules is adjusted with the shutoff valves located in the distribution ducts to correspond to the water vapor and CO_2 loads in the modules.

A diverter valve is located in the Crew/Operations Module which allows the Power/Subsystems Module to be isolated if unoccupied. Distribution duct shutoff valves are closed between the Crew/Operations Module and Power/ Subsystems Module and the diverter valve is positioned so that the entire flow from the reconditioning unit is distributed within the Crew/Operations Module.

4.6.3.1.3 Water Management

Functions of Water Management Assembly Group are as follows:

A. Process urine, wash water, and condensate to obtain potable water.
B. Store and deliver potable water and wash water.

Equipment necessary to accomplish these functions have been grouped into four assemblies: urine water recovery (3100); wash water and condensate recovery (3200); recovered water storage (3300); and contingency water storage (3400). The location of these assemblies within the core modules can be seen in the EC/LS Subsystem schematics; Figures 4.6-4 and 4.6-5. The contingency water supply (3400) is located in the GPL; all other water management assemblies are located in the Crew/Operations Module.

A water balance shows that there is sufficient water recovered onboard the Space Station to meet crew needs. This is possible because water is gained due to the water contained in food 2.8 kg/day (6.18 lb/day) and due to metabolic water generated by the crew 2.13 kg/day (4.70 lb/day). This can be seen in EC/LS mass balance, Figure 4.6-2, which gives values for a 6-man crew. Water is lost due to water recovery inefficiency and the unrecovered fecal water. These losses are small compared to the excesses onboard and as a result an excess of 3.83 kg/day (8.45 lb/day) exists. It is estimated that 1.5 kg/day (3.3 lb/day) of this will be used for EVA cooling, which leaves 2.34 kg/day (5.15 lb/day) of excess water available. This amount can be considered as an available contingency to account for the inability to estimate water needs exactly and for other onboard needs such as experiments.

All normal onboard water needs are provided by recovered water from urine, urine flush, wash water and condensate. The contingency water is standby only and is only used during emergencies. Most of the waste water sources originate in the Crew/Operations Module. Small amounts of condensate and wash water originate in the GPL and are transferred to the Crew/Operations Module for processing. Both hot and cold potable water are available to the crew in the GPL and Crew/Operations Modules.

The urine water recovery assembly (3100) recovers potable water from urine, urine flush water, and reverse osmosis residuum by means of a closed-cycle air evaporation process. An isometric view of this assembly is shown in Figure 4.6-10. The assembly is designed to process the daily rate in 18 hours so that the unit has the capacity to catch up and to process overloads during crew overlap in the event a recycle is necessary. However, it is normally operated 24 hours per day at a reduced flow rate to avoid unnecessary shutdowns and startups.

Air evaporation is an ambient pressure distillation process in which a carrier gas is used to evaporate water from wicks saturated with waste water. In the closed concept urine and waste water from the wash water and condensate recovery assembly (3200) are collected in a pretreatment tank (3151) where a pretreatment chemical is added automatically to fix the free ammonia and kill the bacteria chemically. The tanks (3150) holding the pretreatment chemical are installed redundantly because the pretreatment chemical is replaced by removing the empty tank and replacing it with one that is full. Two holding tanks (3151) are used: one receives urine while the other discharges pretreated urine to the wick evaporator (3190) through the feed cylinder (3194).

The carrier gas evaporates water from the wicks and leaves the evaporator nearly saturated and at a reduced temperature. From there, the humidified air goes through the condensing heat exchanger (3162) where the vapor is condensed and separated from the gas. Finally, the carrier gas is heated by hot water from the solar collector in the heater (3160) and circulated back to the evaporator. The condensate is removed continuously from the gas stream and pumped through a series of charcoal and bacteria filters to the water

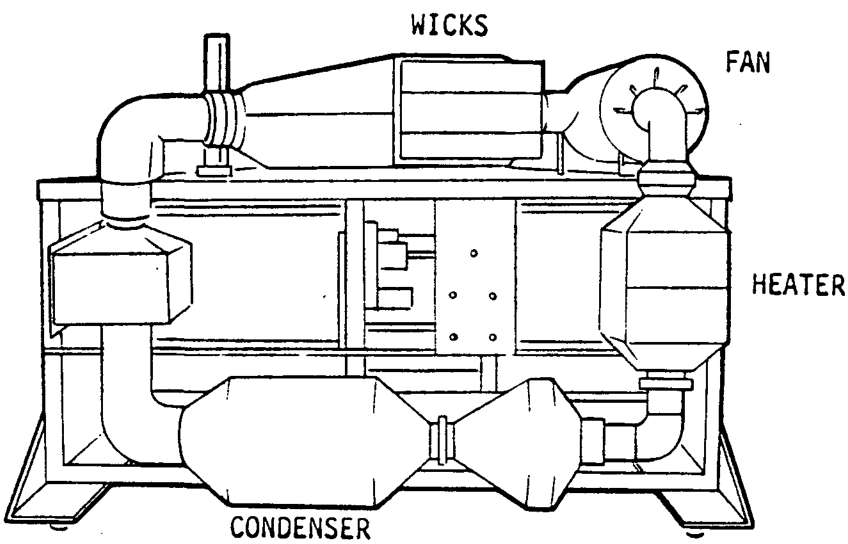

Figure 4.6-10 Isometric View of Urine Water Recovery Assembly

storage assembly (3300). The regenerative exchanger (3161) reduces the amount of process heat added by the heater (3160).

The wick has a predetermined, known life. Therefore, it must be replaced periodically. The evaporator is isolated before wick replacement with manual shutoff valves (3107); the wick evaporator is designed so that no volatile contaminants escape to the cabin atmosphere. After removal from the assembly, the wicks are placed in a bag containing a disinfectant which prevents bacteria growth in the wicks prior to Earth return. The wicks can be replaced in one to two hours; therefore, there is no need to operate the other assembly during wick replacement. The present cycle requires wick replacement every 20 days. Further, development of this technology could provide larger wicks which would increase the period between wick replacements. In detailed design, it would be desirable to optimize wick size based upon replacement time, feed distribution considerations, and pressure drop.

The product water is pumped (3140) through a conductivity sensor (3171), bacteria filter (3120), activated charcoal filter (3121), a second bacteria filter (3120), and the feed cylinder (3194) before it goes to the water storage assembly. The control (3170) and the feed cylinder are designed so that the urine fed to the wick exactly replaces the volume of water pumped out of the assembly automatically; thus, wick drying or flooding is eliminated. The control also senses the quantity of urine in the holding tank through sensor (3172) and adjusts the position of control valve (3112) on the heater. When the quantity of urine in the tank is low, the amount of heat added to the heater is reduced. This slows the process rate to match requirements based upon tank quantity; in addition, it operates automatically at the lowest evaporator temperature possible, thus reducing the amount of volatiles carried over to the product water. The conductivity sensor activates valve 3106 automatically and transfers bad water back to the holding tank for recycling.

The quantity sensors on the urine holding tank also position the 4-way valve (3104) so that when one is filled it becomes the supply tank and the other becomes the collection tank. After the 4-way valve is positioned, the injector (3191) automatically transfers a complete charge of pretreatment chemical into the empty tank. If there is a large imbalance in the collection rate and the processing rate, the control shuts the assembly down and the crew is alerted by the onboard checkout subsystem.

In the event that the air evaporation loop becomes contaminated with bacteria, the unit can be operated in a sterilization mode. Valve (3112) is set manually to heat the air loop to 121°C (250 F). The loop is recirculated continuously with the wick removed and the urine feed shut off until the contamination is eliminated. The water side of the air evaporation loop is sterilized by connecting a flex hose (not shown) manually from the 3313 disconnect downstream of the 3340 pump in the water storage assembly to the corresponding disconnect on the discharge side of the 3140 pump in this assembly, and recirculating until the contamination is eliminated. The urine side is decontaminated chemically if necessary.

Wash water and condensate are purified by means of the reverse osmosis process. Normally, the assembly is operated 24 hours per day but it is designed to process the daily batch in 18 hours so that the unit can catch up in the event a recycle is necessary or during heavy use periods such as crew overlap. Reverse osmosis (RO) is a process that reclaims water by using pressure of 690 to 1, 380 kn/m^2 (100 to 200 psi) to force waste water through a semipermeable membrane. The semi-permeable membrane prevents the passage of solids and other contaminants.

Waste water consisting of wash water and condensate is received and stored in holding tank 3253 in the system shown in Figure 4. 6-4. The waste water is pressurized and discharged by a pressurization pump (3241) into the reverse osmosis circulation loop and recirculated by a low-pressure/high-flow pump (3240). In the reverse osmosis unit, the high pressure forces the water through the semi permeable membrane, leaving the solids in the circulation loop. The reclaimed, processed water is removed continuously and pumped through a series of charcoal (3221) and bacteria (3220) filters. The concentrated waste liquid that builds up in the circulation loop is routed to the urine water recovery assembly (3100) upon a signal from the solids sensors.

Membrane pressure differential is measured across the membrane with the pressure transducer. If a low differential is sensed, indicating a membrane failure, the feed pump to the reverse osmosis unit is turned off and corrective action is taken.

The reverse osmosis unit has been designed to operate on a batch-processing basis. A solids sensor in the recirculation loop determines when the batch is complete and positions the solenoid valve (3206) so that the residuum can be transferred to the urine water recovery assembly (3100) for final processing. Water recovered from the RO unit is passed through a conductivity sensor (3271) which can position a solenoid valve (3206) to divert the flow from the wash water storage tank to the holding tank if water quality is not satisfactory. A quantity sensor (3272) is provided in the holding tank so that the total water supply in the wash water loop can be determined.

This assembly also can be sterilized in a manner similar to that provided for the urine water recovery assembly. Disconnects are provided just downstream of the clean water side of the reverse osmosis unit and in the wash water pump. The dirty side of the reverse osmosis unit is heated by bypassing one of the wash water user equipments (not shown).

Recovered potable and wash water is stored in the recovered water storage assembly (3300) at a pasteurization temperature of 71. 1°C (160°F) to prevent bacteria growth. Three storage tanks (3352 - a fill tank, test tank, and use tank) have been provided in the potable water portion of the assembly. These tanks have 24-hour use water capacities and are cycled alternately through the fill, test, and use functions. An 18-hour microbiological test is conducted to assure that the water is not contaminated. The quantity sensors (3372) indicate when cycling should occur. Each potable water tank has a capacity of 42. 6 kg (94 lb) of water. Valve (3312) controls each tank at the pasteurization temperature of 71. 1°C (160°F) using the heating water high temperature water. Solenoid valves (3303) control the flow of water into and out of the tanks. Pump (3340) recirculates heated water continually through the use tank and the supply line and supplies the pressure required at potable water outlets. A potability test kit (3390) is provided to check the water for potability. Checks include bacteria culture, turbidity, conductivity, and taste.

The wash water holding tanks (3363) are operated in a similar manner except that a test tank is not required. Each tank has a capacity of 73 kg (161 lb) of water. The wash water is delivered at tank temperature through bacteria filter 3320. A second (3340) pump recirculates heated wash water continuously through the use tank and supplies the pressure required by the user equipment.

Hot water is delivered directly at tank temperature through a bacteria filter (3321). Cold water is supplied by cooling the hot water in an accumulator 3363 and feeding through another 3321 bacteria filter. Because the cold water used for drinking provides a possible location for bacteria growth, the coolant to the heat exchanger is turned off periodically and the cold portion of the loop is flushed with 71.1°C (160°F) water by a line (not shown) from the cold potable water outlet to the tank through disconnect 3313.

The system is designed so that it can be sterilized in the event that it becomes contaminated with bacteria. Valve 3312 can be positioned manually to control sterilization temperature at 121°C (250°F) rather than the normal 71.1°C (160°F) pasteurization temperature. Water at this temperature is recirculated until sterile conditions are achieved. This mode is also used to sterilize the water loop in the urine water recovery assembly and the wash water and condensate recovery assembly.

A contingency water storage assembly (3400) is provided in the GPL which stores a 30-day supply of crew intake water. This stored water is available for use in the event of a nonrepairable failure in the urine water recovery assembly or due to a loss of the entire Crew/Operations Module. The water is stored in 3 tanks; each holding 150 kg (308 lb) of water. This water is not maintained at pasteurization temperatures because the tanks contain only ground launched water which contains neglible substrates for supporting bacteria growth. Normally the tanks are isolated from the recovered water loop by normally-closed shutoff valves (3301) and a bacteria filter (3320). The tanks need not be normally pressurized when not in use; the 3-way valves (3302) isolate the 410 kn/m^2 (60 psia) N_2 source. When the contingency supply is needed, the tanks are pressurized by positioning the 3-way valve (3302) and opening the shutoff valves (3301). Tanks are recharged by manual replacement; water transfer lines are not justified because little use of the contingency supply is anticipated.

The contingency water storage assembly contains hot and cold water sources for use by the crew in the GPL. This function is provided by accumulator (3363), shutoff valves (3301), temperature control valve (3314), and bacteria filters (3321) which are provided by the same design as located in the Crew/ Operations Module.

4.6.3.1.4 Waste Management

Functions of the Waste Management Assembly Group are as follows:

A. Collect and transfer urine.

B. Collect, process, and store fecal waste.

These functions are accomplished by the waste collection and storage assembly (4100) and the fecal water storage assembly (4200). Two collector assemblies and one fecal water storage assembly are located in the Crew/ Operation Module as shown in Figure 4.6-4.

The waste collection and storage assembly consists of a seat container/ processor, removable liner, motor driven deflector, vacuum pump, air blower, bacteria and odor filters, condenser/water separator, urinal controls and gauges as shown in Figure 4.6-11.

To operate the system, the control handle is set to "operate." This puts the selector valve (4107) temporarily in an intermediate position which allows cabin air to back flow through the blower (4101) and brings the collector/ processor (4103) up to the cabin pressure. The selector valve then moves

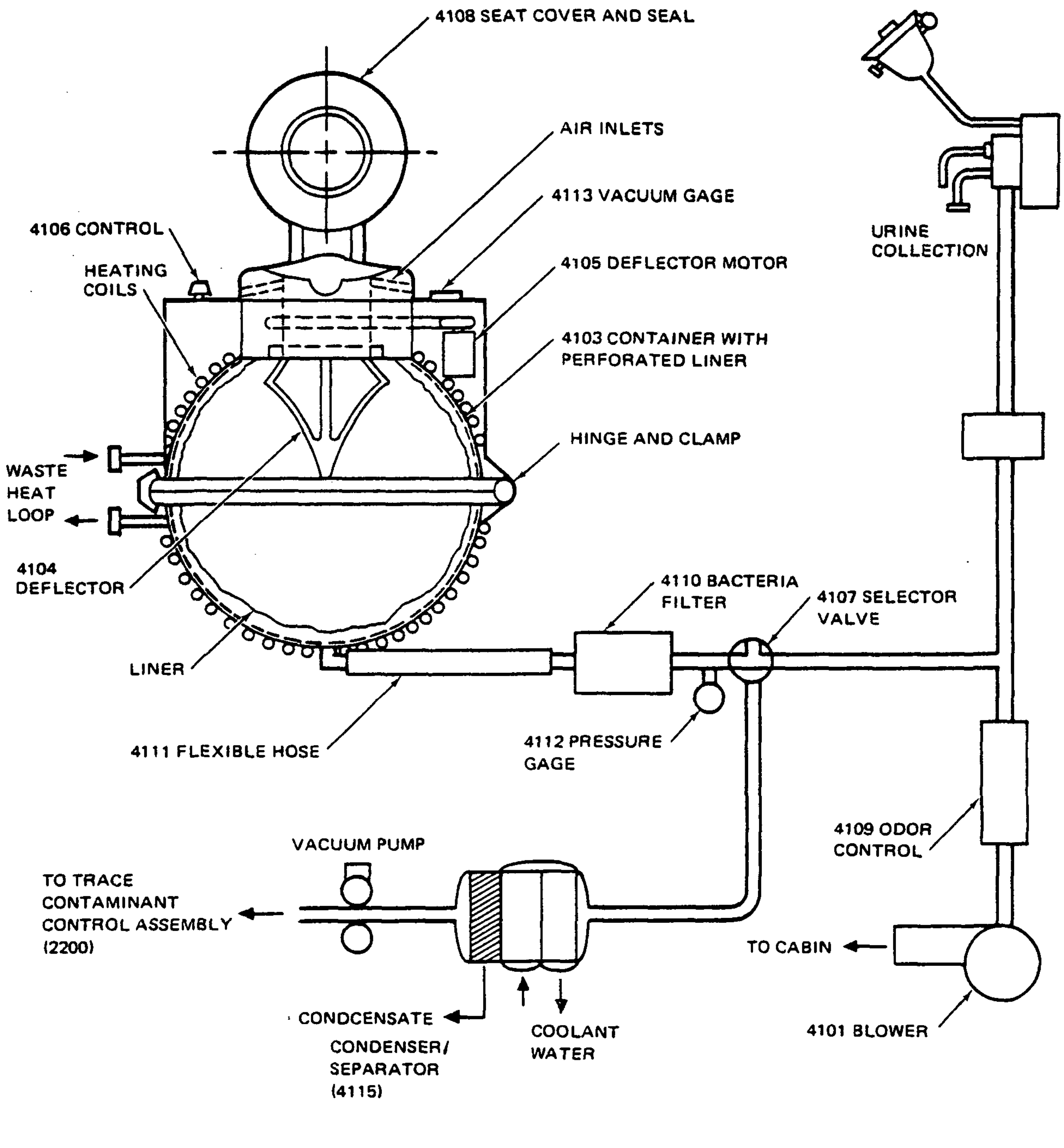

Figure 4.6-11. Waste Collection and Storage Assembly (4100)

to full open and the blower is started automatically. The seat (4108) is then opened manually and this action permits cabin air to flow through the system. The deflector motor (4105) is also activated at the same time. Air flowing through holes in the seat draws the feces down into the deflector which dices and deflects the feces onto the linered inner wall of the collector/processor. The deflector is contoured in such a manner that the diced fecus is distributed evenly over the inner surface of the collector/processor which provides optimum drying conditions as the collector/processor is heated externally by the liquid waste heat loop. Conventional wipes are used and are drawn into the chamber by airflow. The airflow volume and velocity provided by the blower 0.282 m^3/min at 0.372 m/sec (10 cfm at 1.22 ft/sec) also assures against the escape of any odors through a bacteria filter (4110) and an odor control filter (4109) before it is returned to the cabin.

After the wipes are drawn into the collector/processor, the seat is closed manually. The control handle is then set to the "process" position which shuts off the blower and activates the vacuum pump (4102). This action draws the cabin air trapped in the collector/processor through the bacteria filter, pumps it through the odor control filter, and returns it to the cabin. As the air is removed from the collector, the contents are heated by the heating water from the solar collector, thereby increasing the water vapor pressure in the collector. The condenser located on the collector outlet duct cools the exiting gases to about 7.22°C (45°F) with the circulating coolant water. Due to the water vapor differential between the condenser and collector, water vapor flows from the collector and is condensed. The vacuum pump (4102) continually operates to remove the noncondensables from the collector.

The collector/processor liner bag is removed and replaced easily because the lower half of the collector/processor is hinged to swing down and expose the outer surface of the liner. The liner will require replacement every 63 man days in normal use.

After being condensed and separated from the gas stream, the fecal water is pumped through a bacteria filter (4210) to the storage tanks (4220). The tanks are equipped with quick disconnects for ease of tank removal for Earth return and installation of empty tanks.

The urine collector (4130), shown in Figure 4. 6-12, is part of the waste collection and storage assembly (4100) and can be operated separately or in conjunction with fecal collection. The assembly for male crew members are described here. A second design is also provided for female crew members. These are launched as required. The urine collector is comprised of a covered receptacle, a liquid/air separation unit, an air separation membrane, and a manual control valve. The unit uses potable water for flush from the water storage assembly (3300) and the waste collector blower (4104) for air entrainment flow.

The receptacle cover is opened and the control handle is turned clockwise which opens the urine "in" valve, cocks a spring and starts the blower. The blower draws air through the receptacle into the liquid/air separation unit, out through the liquid gas separation membrane, through filters, and into the cabin. After use, the receptacle is closed and the "flush" button is depressed. The control handle is then turned counter-clockwise. This closes the urine "in" line and opens the urine-to-recovery-system line, stops the blower, and releases the urine transfer spring. The spring forces the urine into the urine recovery system tank and readies the system for reuse. The unit is sized to handle a volume of 1, 000 cc per usage.

The water balance shown in Figure 4. 6-2 indicates a normal water surplus of 2. 34 kg/day (5. 15 lb/day). Some of this surplus may be used in experiments, but at certain times, it may be necessary to store and return the excess water to Earth. This water will be stored in the fecal water storage assembly and returned to Earth by manual replacement of the tanks. When the quantity sensor in the urine holding tank indicates a full position, the diverter valve (4240) will be positioned automatically to direct urine to the fecal water storage tanks.

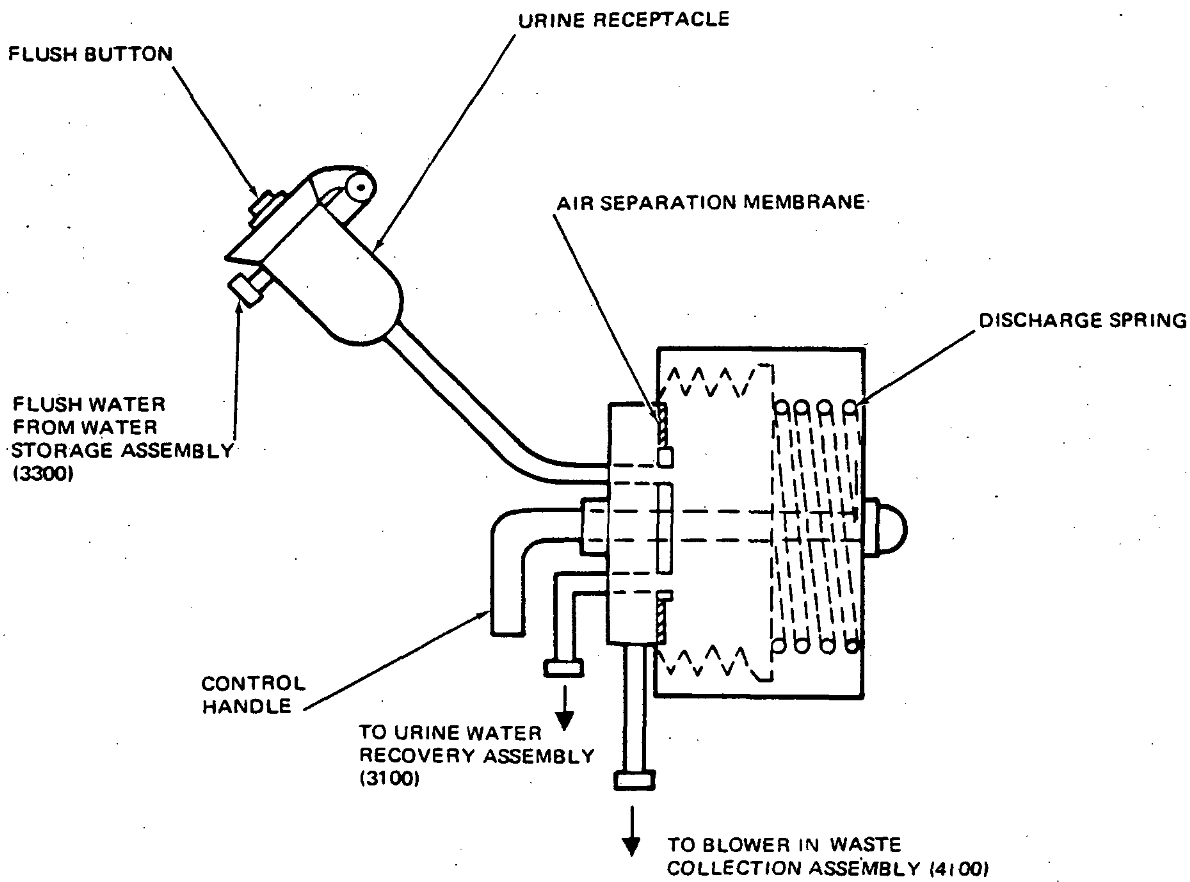

Figure 4.6-12. Urine Collector (4130) for Male Crewmembers

4.6.3.1.5 IVA/EVA and Emergency Life Support

Functions of the IVA/EVA assembly group are to provide support to suited or unsuited crewmen during intravehicular (IVA), to suited crewmen during extravehicular (EVA) activities and to provide emergency life support for the crew during emergencies.

Suited EVA support is provided by Portable Life Support Systems (PLSS) (5100) identical to the back packs developed for the Apollo Program. Two of these assemblies are located in the GPL and two are located in the Power/Subsystem Module. Their use frequency is estimated at 1-1/2 times per month with an event duration of 4 hours per man. Two crewmen will normally be required for each EVA mission.

The Apollo PLSS currently developed is a self-contained backpack that provides life support, voice communication, and telemetry for astronauts performing up to four hours extravehicular tasks. Its cooling capacity is 1,420 watts (4,816 Btu/hr) and can support metabolic loads ranging from 117 watts (400 Btu/hr) to short-term peaks of 586 watts (2,000 Btu/hr). An oxygen purge system (OPS) is mounted on top of the PLSS as an emergency backup unit. In the event of a PLSS malfunction, the OPS provides oxygen (for a minimum of 1/2 hour) for respiration, forced convective cooling of the astronaut, and visor defogging.

With the PLSS, an oxygen ventilation circuit provides oxygen flow through the suit and cools, dehumidifies, and removes CO_2 and trace contaminants from the recirculated oxygen flow. A feedwater loop provides H_2O to a sublimator which rejects metabolic and waste heat to space. The liquid transport loop recirculates cooled water through a liquid-cooled garment worn under the pressure suit for sensible metabolic heat removal.

During IVA the crewmen are supported by the umbilical life support assembly (5200) which provides open-loop regulated oxygen ventilation and pressure control for a suited crewman and closed-loop cooling water that interfaces with the pressure suit thermal loop to provide for the removal of the sensible heat load. Oxygen and coolant for the umbilical life support assembly pass through an umbilical which attaches to the IVA support assembly (5300).

This assembly can be used either in a pressurized compartment or in a depressurized environment. An alternate mode of operation is to provide metabolic oxygen via a face mask for an unsuited crewman. Four umbilical life support assemblies are provided; two are stored in the Power/Operations Module and two in the GPL.

The IVA support assembly (5300) provides suit cooling water, high-pressure oxygen, and communications cables to the umbilical connections. An IVA support assembly is provided in each core module so located, so any point can be reached withing the Space Station.

Three sets of connectors are provided per assembly; however, the number of crewmen that can be supported is limited by their metabolic activity levels and by the coolant-water flow rate delivered to that assembly. Nominally, this coolant-water flow rate is 188 kg/hr (413 lb/hr) per assembly, which is adequate to support two crewmen at nearly their peak metabolic rate or three crewmen at slightly more than half the peak metabolic rate.

A diverter valve permits isolation of the assembly when not in use to minimize water coolant pump power consumption. A relief valve is installed in the design to permit continuous flow independently of the number of pressure suits on line.

Six portable O_2 bottles (5400) are located in the Crew/Operations Module for emergency crew breathing by face mask. Each bottle supplies 1/4 hour of O_2 for a crewman breathing at a high rate.

The emergency EC/LS pallet (5500) supplies 96 hours of EC/LS provisions for a three-man crew during emergencies requiring a Shuttle rescue. Two of these pallets are located in the GPL for six-man capability and one pallet is located in each attached module. During buildup stages one pallet is also located in the Power/Subsystems Module to provide a second EC/LS source before a Logistics Module is attached.

Figure 4.6-13 gives a summary of provisions included in the pallet. The unit is completely self-sufficient and provides for all essential EC/LS and

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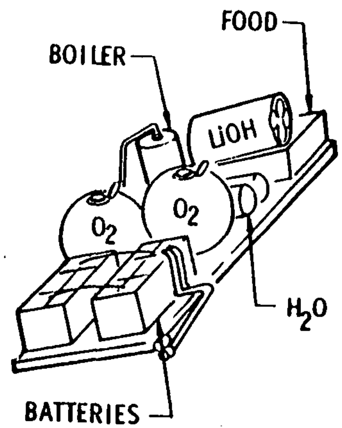

3-MAN PALLET

PROVISION	REQUIREMENT	WEIGHT Kg (LB)
OXYGEN	METABOLIC O_2	23.6 (52)
WATER	CREW INTAKE + COOLING	64.4 (142)
FOOD	2700 K CAL DIET	13.6 (30)
LiOH	MAINTAIN .102KN/M^2(7.6 mmHg) PCO_2	19.0 (42)
BATTERIES	960 WATT-HR	22.6 (50)
WATER BOILER	740 WATT (1600 BTU/HR) COOLING	3.2 (7)
MISC. SUPPLIES		2.3 (5)
PALLET/PACKAGING		11.3 (25)
	TOTAL	160.0 (353)

Figure 4.6-13 96-Hour Emergency Provisions

crew systems functions. The basic Space Station EC/LS normally has two separately located sources available for each major EC/LS function. Therefore, only emergency situations of a most severe nature will result in use of the pallets.

4.6.3.1.6 Thermal Control

The function of the Thermal Control Assembly Group is to collect, transport, distribute and reject Space Station heat so that crew and equipment are maintained within required temperature limitations. This is accomplished with active thermal control loops consisting of internally circulating water loops and external freon 21 radiator loops. Heat is transferred from the air and various types of equipment to the circulating water loops, it is then transferred through interface heat exchangers to the freon loops and rejected to space from the radiators. The radiator design provided is capable of rejecting Space Station heat loads for high Beta angle (78.5°), least favorable orientation, worst docked module configuration and 3σ hot case environment.

A separate water and freon loop is provided in each core module. This approach is favored over single loops for all three core modules so that loss of one loop or module will not cause an abort situation. Additionally, separate loops in each compartment allows the removal and Earth return of a module for refurbishment without loss of thermal control in the remaining modules.

An interchange circulating water loop is also provided so that cooling loads can be shared between modules. This provision enables a module to continue operating even though its primary thermal control function is inoperative. Additionally, the interchange loop can transfer peak thermal loads from one module with insufficient capacity to other modules with excess capacity.

Process heat is required on the Space Station by a number of EC/LS assemblies, namely, CO_2 removal, urine water recovery, recovered water storage, and waste collection and storage. This heat is supplied by a solar collector mounted on the solar array structure and delivered to the core modules by two heating water loops. Separate loops are provided for the EC/LS in the Crew Operations Module and in the GPL Module.

A more detailed description of the Thermal Control Assembly Group follows below.

Coolant Water Loops

The coolant water loops differ slightly for each of the three core modules because the cooling requirements differ for each. Details of the loops can be seen in Figures 4.6-3, 4.6-4, and 4.6-5. Heat balances showing major thermal loads are shown in Figures 4.6-14, 4.6-15, and 4.6-16.

All three coolant water loops contain thermal loads from batteries, air temperature controls, electrical equipments and (when in use) IVA support. In addition to these cooling loads, the Crew/Operations Module cooling includes humidity control, urine water recovery, CO_2 removal, potable water cooling and crew equipment such as shower, dish washer and clothes washer. The GPL has additional cooling loads for humidity control, CO_2 removal, and potable water cooling.

The arrangement of the water loops places the cold-temperature requirements at the low-temperature point by order of temperature; namely (1) IVA, (2) potable water cooling, (3) battery, and (4) condenser. These components all require a temperature below 7.22 °C (45 °F). The air temperature control heat exchanger requires an inlet temperature of around 14.5 °C (58 °F). A temperature lower than this can result in unwanted condensation on the heat exchanger surfaces. Although the humidity for worst case, continuous operation could reach 15.5 °C (60 °F), condensation should not occur in the air temperature control heat exchanger because of the temperature drop across the heat exchanger plates.

A minimum inlet temperature of 14.5 °C (58 °F) is ensured to the air temperature heat exchanger by heating the inlet coolant water with a regenerative heat exchanger (2804). The heat required is obtained from the coolant water leaving the cold plates. Temperature control valve (2806) controls hot-side water flow in the regenerative heat exchanger to obtain the 14.5 °C (58 °F) temperature for cold-side coolant flow.

Total flow in each loop is set by the allowable cold-plate outlet temperature of 43.3 °C (110 °F). A high temperature is desired for cold-plate outlet temperature to maximize radiator temperature and performance. The coldest temperature in the water loop is set at 3.3 °C (38 °F) which is sufficiently

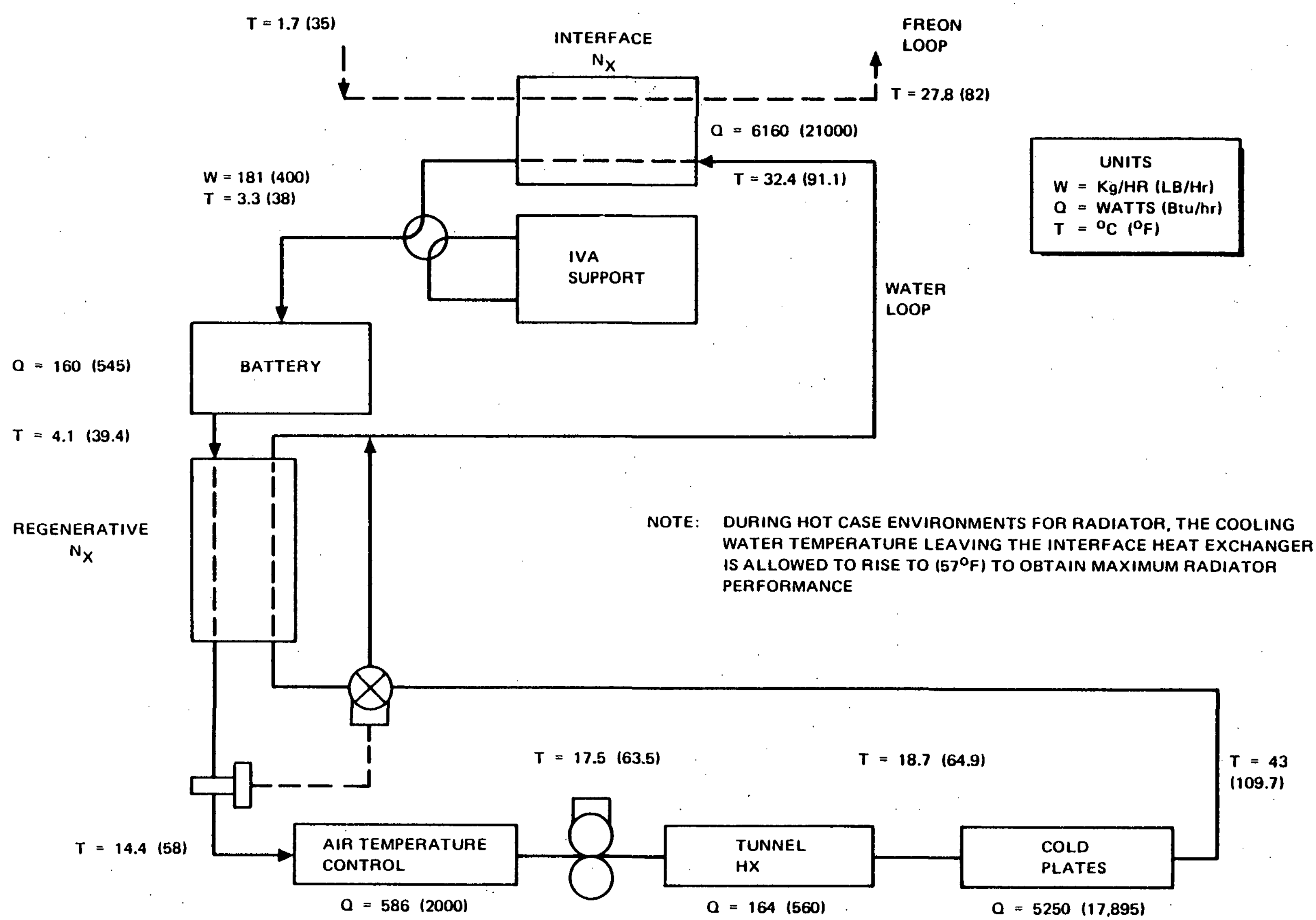

Figure 4.6-14. Power Module Heat Balance (Max Heat Load–65°F Cabin)

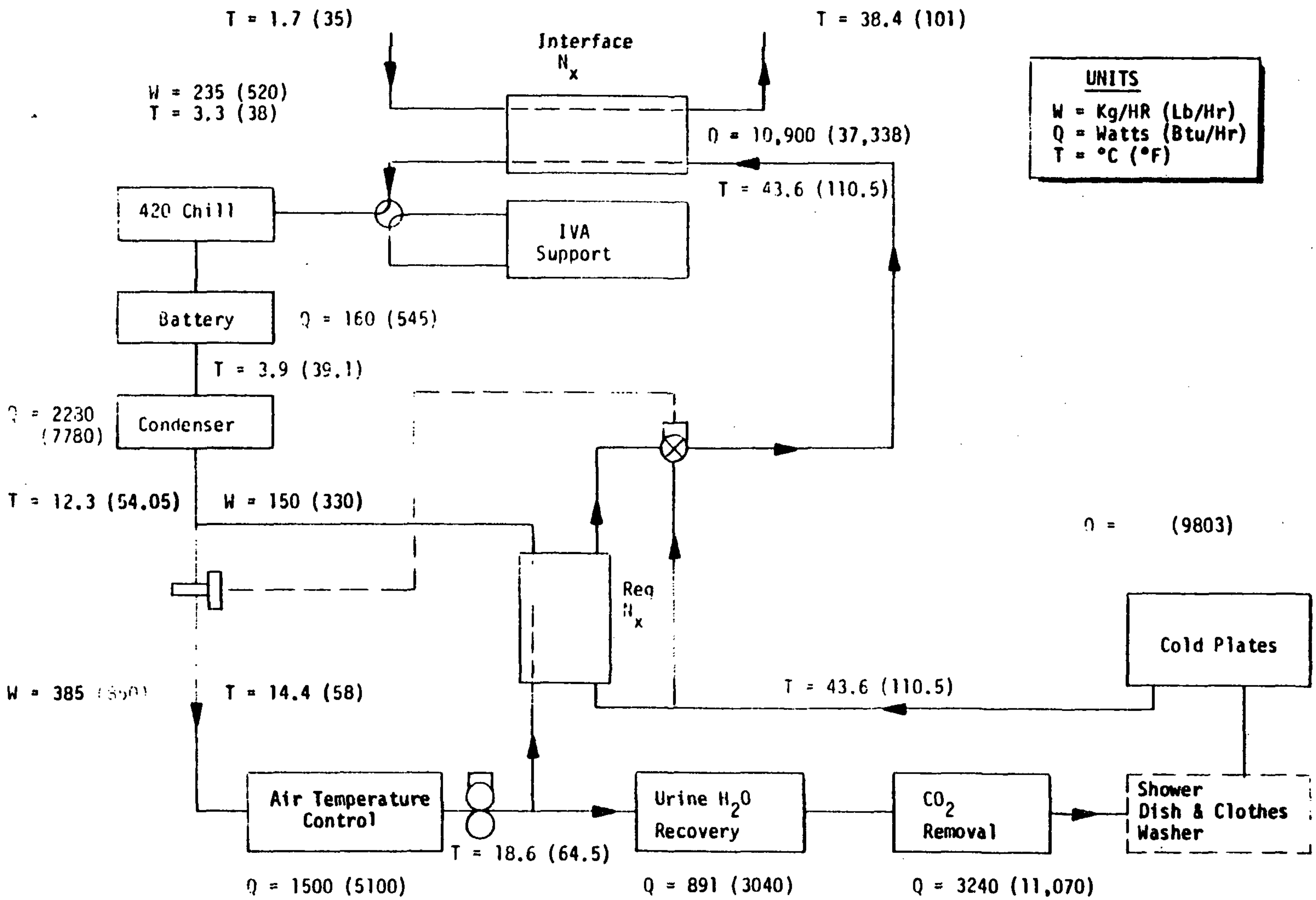

Figure 4.6-15 Crew/Operations Module Heat Balance (Maximum Heat Load - 65°F Cabin)

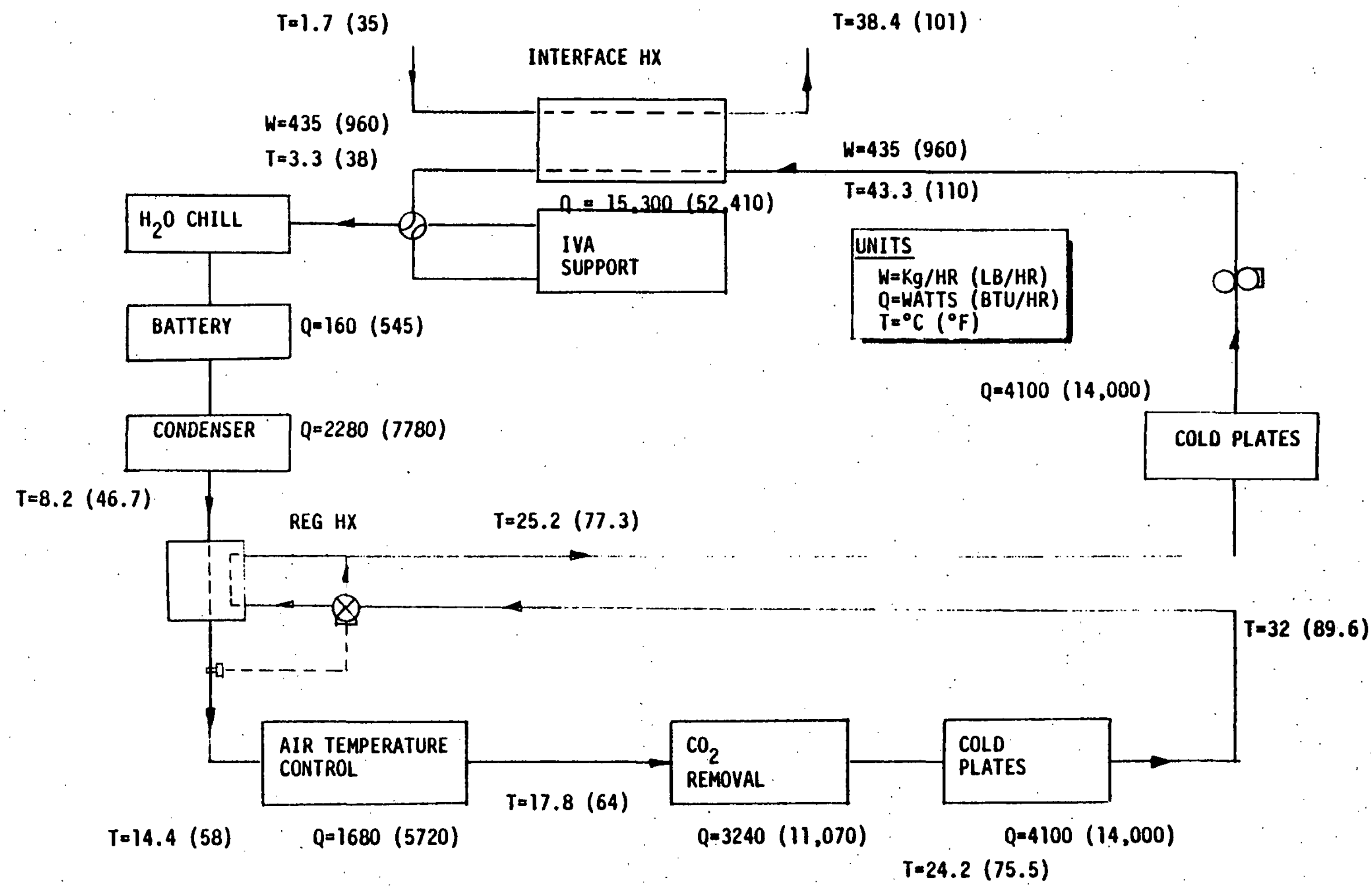

Figure 4.6-16 GPL Module Heat Balance (Maximum Heat Load – 65°F Cabin)

above 0°C (32°F) to avoid water-loop freezing. The flow set by the heat rejection requirements and thus temperature limits provides adequate flow rates for all Space Station assemblies except for the air temperature control assembly in the Crew/Operation Module. Sufficient cooling capacity is available but a higher flow than 235 kg/hr (520 lb/hr) is required. The higher flow requirement 385 kg/hr (850 lb/hr) is obtained with a bypass loop around the air temperature control assembly, see Figures 4.6-4 and 4.6-15. The coolant water recirculation assembly (6100) is located at the air temperature control assembly outlet so that flow in the primary water loop and the bypass loop can be provided by one assembly for simplicity.

Due primarily to the humidity control assembly requirements for 7.22°C (45°F) coolant water, the coolant-water temperature in the Crew/Operations and GPL Modules must not appreciably exceed 3.3°C (38°F) out of the interface heat exchanger. This is not true for the Power Subsystems Module where the temperature can be allowed to rise to 14.5°C (58°F). This may be necessary and desirable during worst-case radiator environments to obtain sufficient heat rejection. Battery life is enhanced by lower temperatures, under 10°C (50°F), however, the increase in battery resupply weight for short-term temperature excursions to 14.5°C (58°F) are minimal.

A second option for maintaining Power Subsystems Module temperature below 10°C (50°F) is to use the interchange loop. This loop is normally not operated, but is automatically activated if the coolant water temperature in any module exceeds 5.55°C (42°F) at the interface heat exchanger outlet. The interchange loop interfaces with the freon and water loops through heat exchangers (6220 and 6223) located in each core module. Heat is either picked up or rejected by the interfacing water loop in each module depending on the interchange loop temperature entering the core modules. If the temperature is over 5.55°C (42°F), the shutoff valve (6222) opens to admit a portion of the cold freon to the freon interchange heat exchanger (6220) which cools down the interchange loop water. The amount of freon admitted to the freon heat exchanger corresponds to the normal excess radiator capacity of that module. The interchange loop then passes through the coolant water loop but little heat exchange occurs.

If the interchange loop temperature entering a module is less than 5.55 °C (42 °F), shutoff valve (6222) remains closed and no cooling occurs. However, when the interchange loop flow through the water interchange heat exchanger, the water loop is cooled. In this way, the interchange loop transfers heat loads between modules.

A second function of the interchange loop is to backup the primary water cooling loops. Each basic station module contains critical equipment which are essential to continuing station operation. The interchange loop provides this backup so that in the event of a failed primary cooling-water loop, the critical electrical equipment continues to operate normally.

Both the primary and the interchange cooling loops pass through separate passages in the critical-equipment cold plates. Due to the large amount of built-in redundancy, and the inherent reliability of cooling-loop equipment, an inoperative primary cooling loop is highly unlikely. However if the flow sensor in the primary loop indicates a no-flow condition, the data management system switches diverter valve (6511). The interchange-loop pumps are activated and the crew is warned of the failure. The interchange loop continues to operate until the primary loop is repaired and placed back into operation. This backup cooling is provided at all stages of buildup. A bypass in the interchange loop allows the interchange loop to operate when the Power/Subsystem Module is the only module on station.

The bulk of the critical cooling loads are in the Power/Subsystems Module. These include the gyroscope assemblies, communication equipment, the power source bus and power regulation equipment. Other critical cooling loads exist in the Crew/Operations Module and GPL Module. These include data management, batteries, battery chargers, power distribution busses, and inverters. Much equipment is duplicated and does not require backup cooling. These include EC/LS equipment, computers, and consoles.

Heating-Water Loops

Process heat is provided for EC/LS equipment by a solar collector which is distributed to the equipment by 2 heating water loops. The two water loops are completely independent; one loop serves the EC/LS in the Crew/Operations Module and the other loop serves the EC/LS in the GPL Module.

The solar collector consists of two flat panel surfaces located on the solar array framework. The panels are of a similar design to the radiators with redundant sets of fluid tubes attached to the panel. The panel faces consist of sand blasted aluminum which produces a solar absorbtivity/thermal emissivity ratio (α/ϵ) of 2.35. This coating should be very stable and relatively insensitive to effluent impingement and degradation due to solar radiation. A surface coating with an α/ϵ of 2.35 gives good performance in that the percentage of solar radiation which is transferred to the hot fluid loops is large for the surface area. An area of 15.8 m^2 (170 ft^2) produces on the average 3,460 watts (11,800 Btu/hr) at 132°C (270°F) which is adequate for the Modular Space Station EC/LS. Although on the average 3,460 watts of heating is provided, a peak of about 7,000 watts is obtained during the sunlight portion of the orbit.

Each solar collector has two sets of parallel tubes; each heating-water loop uses one set of tubes. Valves (6510 and 6511) can be used to circulate each heating-water loop through either or both solar collector panels. Additionally, these valves isolate the solar panel tubes in the event of a pressure decay indicating a panel leak. A large leak may result in loss of water from the heating-water loop. After the damaged area has been isolated, the water loop is recharged from the onboard water supply.

During Space Station buildup, the solar collector heat is not always needed. During this time the heat can conveniently be transferred to the radiator loops by the heat exchanger (6500). This prevents possible freezing of the radiator loops and also allows operation of the solar collector at moderate temperatures.

If a no-flow condition occurs in the solar collector, a maximum temperature of 221°C (430°F) can occur which corresponds to a pressure of 2370 kn/m^2 (344 psia). The heating water loops are designed for this pressure so that pressure relief is not necessary. Relief valves (6607) are added however in the event that the solar collector coating degrades to a higher value of α_s.

Valves (6510) are provided to (1) provide a flow path for the heating water loop before the Crew/Operations and GPL Modules are added.

Radiator Loops

Each module is provided with a separate freon radiator loop which picks up heat from the coolant-water loops through the interface heat exchanger (6205) and transfers the heat to the radiator for heat rejection. The radiator loop design is similar for each module; Figures 4.6-3, 4.6-4 and 4.6-5. The freon temperature into the interface heat exchanger is controlled to 1.67° C (35° F) by a regenerative heat exchanger (6203) and control valve (6204). Sufficient warm radiator inlet flow is diverted through the regenerative heat exchanger to maintain a 1.67° C (35° F) fluid. This temperature is selected to avoid possible water coolant freezing due to the freon temperature being too cold.

A thermal capacitor (6202) is provided in the radiator loop to even out temperature extremes in the freon fluid at the radiator outlet. These temperature extremes occur due to changes in radiator performance as the environment changes around the orbit. Radiator outlet changes also differ greatly between radiator segments; segments facing the sun and Earth are warm while segments facing open space are cold. Flow direction and split through the segments are controlled to obtain maximum radiator performance. This is done by flow control valve (6210) which controls flow split and flow reversal valves (6206) which alter the flow direction to obtain optimum radiator performance.

Each radiator panel is equipped with shutoff valves which isolate the panels when not in use or when damage has occurred. Panels are automatically closed when a rapid pressure decay occurs in a radiator panel. In the event freon fluid is lost in the freon loop within the module, recharging capability is provided. A gas separator (6302) in the radiator recirculation assembly (6300) removes entrained air which can occur due to leakage or component repair.

The radiator recirculation assembly contains redundant pumps (6304) and filters (6303). In the event the operating pump fails, the standby pump is automatically actuated and the crew alerted to the failure through the Data Management Subsystem. Check valves (6305) prevent back flow through the inoperative pump. A relief valve (6306) is provided to prevent damage to the pumps and other equipment due to a radiator loop obstruction.

A ground heat exchanger and water boiler (6201) is provided to cool the radiator fluid during ground operation before and during launch. Cold fluid from the ground support unit is supplied via quick disconnects (6208). The freon loop is recirculated through valves (6211) during this period to avoid losing cooling capacity from the radiator. During launch, water is drawn from the potable water supply and evaporated in the unit to provide cooling.

Radiator Design

Each core module has a separate external radiator which rejects the heat from the circulating freon 21 radiator fluid. A serpentine tube layout with a combination of series and parallel passes was found necessary for all modules. Assuming a minimum allowable tube size of 0.19 inches, turbulent flow with $R_e > 10,000$ cannot be achieved with parallel tubes only. In general, it is desirable to maintain a Reynolds number above 10,000 to maximize heat rejection capacity. Figure 4.6-17 shows a schematic for the serpentine tube layout for a typical module. For clarity, only one tube is shown. Depending on the module, the one tube represents the tube layout for 3 to 9 parallel tubes.

The cylindrical surfaces of the GPL, Crew/Operations and Power/Subsystems Modules are divided into four separate circuits. Each circuit encompasses 180 degrees of vehicle circumference and half the vehicle length in the axial direction. The vehicle was divided into 180-degree segments to permit higher heat rejection capacity. While additional segmentation will further increase radiator capacity, the improved performance cannot be justified because of the resulting increase in system complexity and corresponding

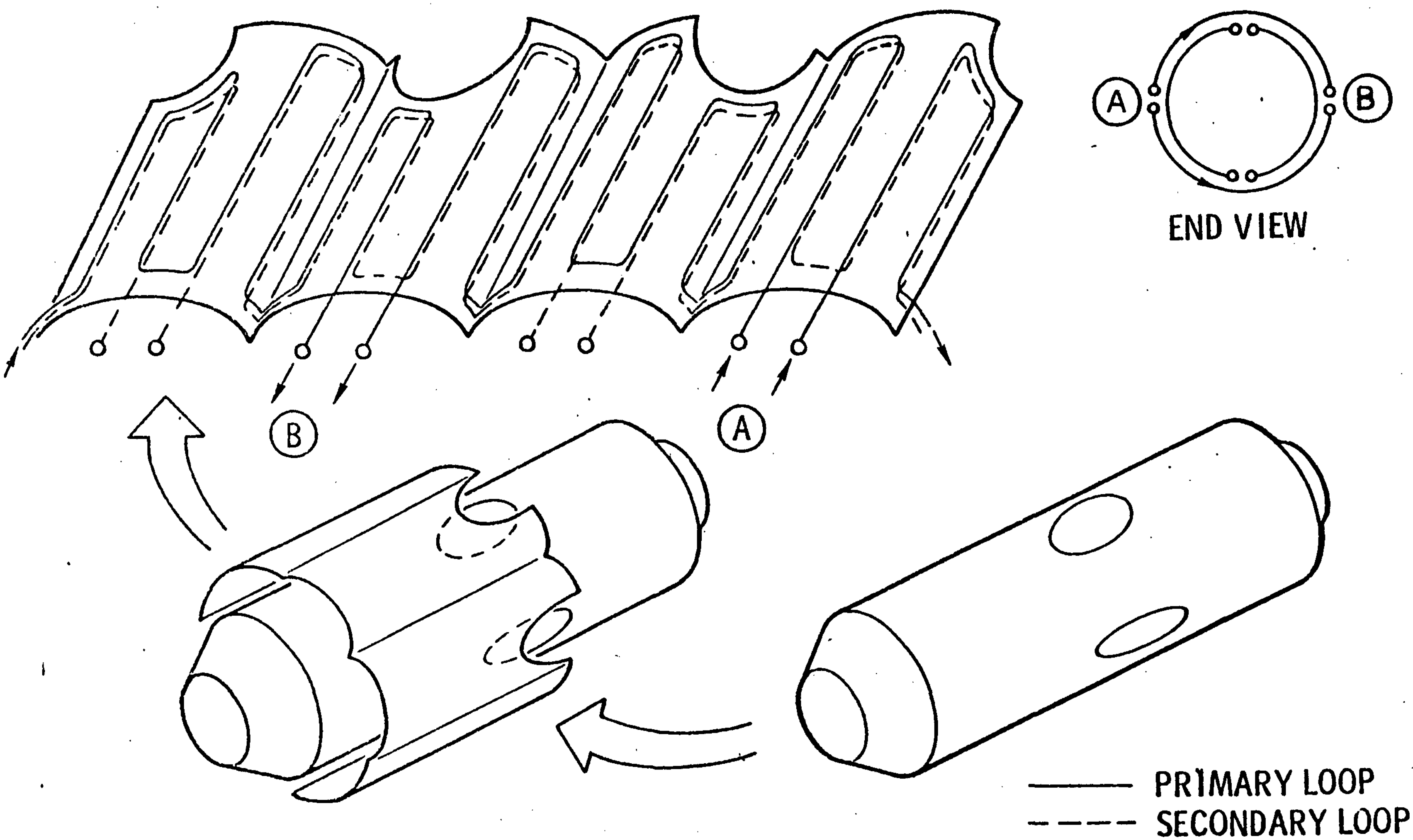

Figure 4.6-17 Radiator Panel Tube Layout

decrease in reliability. The vehicle is divided into separate circuits in the axial direction in order to simplify tube layout in the vicinity of the docking ports. The area on the forward conical section of the Power/Subsystems Modules is covered with radiator surface.

Figure 4.6-18 shows the method of attaching tubes to the radiator fin. The tube supports are not required to withstand high aerodynamic compression loads since the vehicle will be enclosed within the Shuttle Orbitor during launch and reentry. This allows a design for the method of attaching the tubes which is much improved from that previously selected for the 10m (33-ft) diameter station. In particular, the current design is lighter and yields a higher overall radiator effectiveness.

Flow reversal valves (6206) are provided to reverse the direction of flow in a particular radiator segment. Thus, for those modules where the radiator panels facing the sun switch within the orbit, valves are provided which would automatically switch flow direction when the condition occurs. Such a condition will occur for Earth-oriented Station for $\beta = 0$ certain modules.

The flow reversal capability also will be useful for the core modules to compensate for changes in sun-vehicle orientation due to precessing of the vehicle orbit.

A flow control valve (6210) has been incorporated to adjust the proportion of radiator fluid to the two 180-degree radiator segments. Here, more efficient use is made of available radiator area by increasing flow to the panel with the coldest sink temperature while correspondingly decreasing the flow to the panel with the hottest sink temperature.

Secondary radiator circuitry is provided to increase reliability of the thermal control system and to enable selection of more favorably oriented radiator panels. The inlet manifolds for the secondary circuitry are staggered by 90-degree increments around the vehicle circumference. This will allow selection of the circuits which have the better orientation with respect to the sun for a particular orbit. Thus, the primary and secondary circuitry will not be used in the conventional sense. That is, the secondary circuitry will

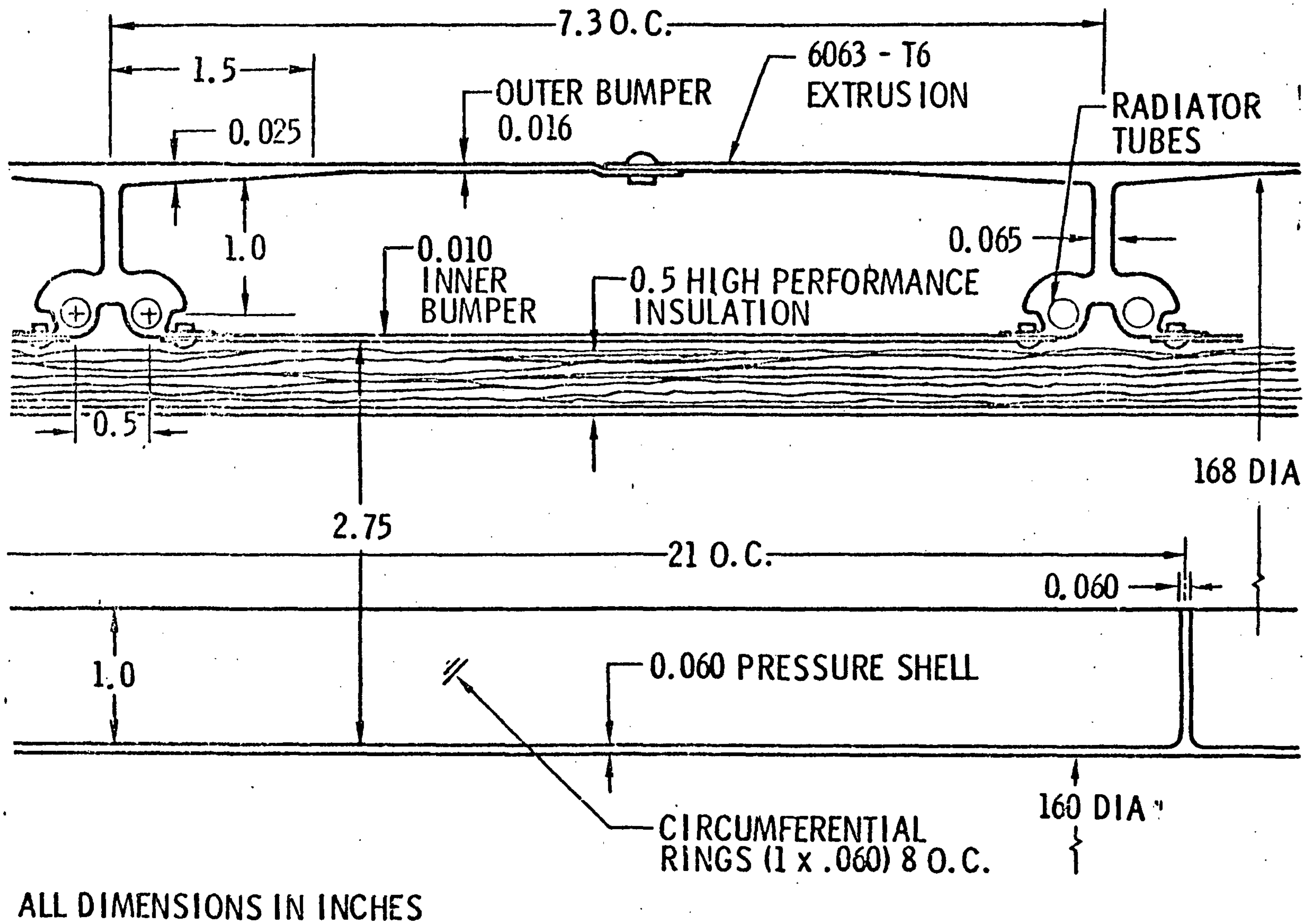

Figure 4.6-18 Fin and Tube Extrusion

not be used strictly only upon failure of the primary circuit, but will be used at any time the position of its inlet manifold offers a better orientation with respect to the sun. The concept, however, does not offer complete redundancy. Should the primary circuit fail, the secondary circuit would not have the same orientation with respect to the sun. Assuming the primary circuits had the better orientation, some degraded radiator performance might occur during periods when the radiation environment is the most severe. By using flow modulation the decrease in radiator performance could be minimized to an acceptable value.

The radiator is coated with paint with a relatively low solar absorptance and high thermal emittance. Using S-13-G, ZhO in methyl silicone with potassium silicate treatment, an initial solar absorptance, αs, of 0.18 can be achieved. Maximum degradation of exposure to 1,580 hours of solar radiation during space flight was found to be about 65 percent which corresponds to $\alpha s = 0.3$. For design purposes, an initial value of 0.2 with a maximum degradation to 0.4 after 10 years has been assumed. The thermal emittance is assumed to remain constant at 0.9.

Detailed performance analyses have been performed and are reported on in subsection 4.6.4.5. The results show that sufficient radiator performance is available for worst case conditions of high β angle (78.5 deg), least favorable manifold location, degraded radiator coating ($\alpha/\epsilon = 0.4$), worst docked module configuration and 3σ hot-case environment.

4.6.3.2 Interfaces

Table 4.6-7 describes the interfaces between the Space Station core module and attached RAMS and Logistics-Modules, between core modules and the Shuttle, and between the core modules. Only ISS modules are included except a description is included for the interface required in the ISS Crew/Operations Module which mates with GSS buildup modules. Additional interfaces exist with other Space Station subsystems such as electrical power and data management system and these are covered in the descriptions of those subsystems.

All interfacing lines contain shutoff valves on both sides of the interface. Additionally, these lines will be capped during launch and when not in use to prevent contamination and as a backup to the shutoff valves.

Space Station and Logistics Module

Atmospheric O_2 and N_2 lines are provided between these modules so that the normal use supply can be transfered from the Logistics Module. Additionally, pump down and repressurization lines allow use of the Space Station pump down system for operation of the Logistics Module air lock. Logistics Module atmosphere is purified by the EC/LS on board the Space Station and lines are provided for this service. The interfaces discussed above exist at all docking ports so that there are no restrictions on docked location of the Logistics Module.

Space Station and Shuttle

During prelaunch, the Space Station thermal control system is "cold soaked" and it acts as a cooling medium during launch. The wax material in the radiation control assembly (6200) provides much of the required cooling during launch. Ground connections to provide cold circulating fluid are necessary between the Space Station module and ground equipment via the Shuttle to attain the "cold soak" condition.

During some periods of the buildup phase, the atmosphere reconditioning equipment is not functioning on the Space Station. This function is then provided by the Shuttle EC/LS. Purified and temperature controlled air is withdrawn from the Shuttle EC/LS and directed to the Space Station through flexible "drag in" ducts.

Space Station and RAMS

Pumpdown and repressurization interfaces are provided so that Space Station pumpdown equipment can be used for the depressurization and repressurization of RAM experiment chambers. Purified air is distributed to each RAM interface to provide atmosphere reconditioning.

Space Station Core Modules

Oxygen and nitrogen gas are provided at each core module interface so that

Table 4.6-7

EC/LS INTERFACE DESCRIPTION

Interfacing Elements	Description (No.)	Type	Purpose
Space Station and Log Module	O_2 supply (1)	O_2 gas at 410 kn/m^2 (60 psia)	Normal O_2 supply
	N_2 supply (1)	N_2 gas at 410 kn/m^2 (60 psia)	Normal N_2 supply
	Pumpdown (1)	Air at 0 to 1 atmosphere	Pumpdown of Log Module airlock
	Repressurization (1)	Air at 0 to 2,050 kn/m^2 (300 psia)	Repressurization of Log Module airlock
	Atmosphere purification (2)	Air at 1 atmosphere	Provide purified air to Log Module
Space Station and Shuttle	Ground cooling connections (2)	Coolant at -6.7°C (20°F)	Provide ground cooling to Space Station
	Atmosphere purification	Air at 1 atmosphere	Provide atmosphere conditioning during buildup
Space Station and RAMS	Pumpdown (1)	Air at 0 to 1 atmosphere	Pumpdown of experiment chambers
	Repressurization (1)	Air at 0 to 2,050 kn/m^2 (300 psia)	Repressurization of experiment chambers
	Atmosphere purification (2)	Air at 1 atmosphere	Provide purified air to RAM
Power/Operations and Crew/Subsystem Modules	O_2 supply (1)	O_2 gas at 410 kn/m^2 (60 psia)	Supply contingency O_2 from power to Crew/Operations Module(s)
	N_2 supply (1)	N_2 gas at 410 kn/m^2 (60 psia)	Supply contingency N_2 from power to Crew/Operations Module(s)
	Coolant water (2)	Water at 5.6 - 38.4°C (42-101°F)	Transfer cooling loads between modules
	Heating water (4)	Water at 60-132°C (140-270°F)	Transfer heat from solar collector to Crew/Operations Module(s)
	Pumpdown (1)	Air at 0 to 1 atmosphere	Transfer pumpdown air to power module
	Repressurization (1)	Air at 0 to 2,050 kn/m^2 (300 psia)	Repressurization of other Space Station modules
	Atmosphere purification (2)	Air at 1 atmosphere	Provide power module atmosphere purification from EC/LS in Crew/Operations Module(s)
Crew/Operations and GPL Modules	O_2 supply (1)	O_2 gas at 410 kn/m^2 (60 psia)	Supply contingency O_2 from power to Crew/Operations Module(s)
	N_2 supply (1)	N_2 gas at 410 kn/m^2 (60 psia)	Supply contingency N_2 from power to Crew/Operations Module(s)
	Coolant water (2)	Water at 5.6-38.4°C (42-101°F)	Transfer cooling loads between modules
	Heating water (2)	Water at 60-132°C (140-270°F)	Transfer heat from solar collector to GPL module
	Potable water (1)	Water at 410 kn/m^2 (60 psia)	Supply potable water from crew module to GPL Supply contingency water from GPL to crew module
	Purified wash water (1)	Water at 410 kn/m^2 (60 psia)	Transfer wash water from Crew/Operations Module(s) to Hygiene facility in GPL
	Unpurified wash water (1)	Water at 410 kn/m^2 (60 psia)	Transfer used wash water from hygiene facility in GPL to crew module EC/LS
	Pumpdown (1)	Air at 0 to 1 atmosphere	Transfer pumpdown air from GPL to Power/Subsystem Module
	Repressurization (1)	Air at 0 to 2,050 kn/m^2 (300 psia)	Repressurization of other Space Station modules
	Atmosphere purification (2)	Air at 1 atmosphere	Emergency air purification between Crew/Operations and GPL Modules
Crew/Operations (ISS) and Crew/Operations (GSS) Modules	O_2 supply (1)	O gas at 410 kn/m^2 (60 psia)	Supply contingency and Logistics Module O_2 supply between modules
	N_2 supply (1)	N gas at 410 kn/m^2 (60 psia)	Supply contingency and Logistics Module N_2 supply between modules
	Potable H_2O (1)	Water at 410 kn/m^2 (60 psia)	Transfers water between Crew/Operations Module(s) to meet crew demands
	Pumpdown (1)	Air at 0 to 1 atmosphere	Provides pumpdown of GSS Modules from ISS Power/Subsystems Module
	Repressurization (1)	Air at 0 to 2,050 kn/m^2 (300 psia)	Provides repressurization of GSS Modules from accumulator in ISS Power/Subsystems Module
	Atmosphere purification (2)	Air at 1 atmosphere	Emergency air purification between ISS modules and GSS modules

the atmosphere can be replenished in any module using the normal supply from a Logistics Module or the contingency supply in the Power/Subsystems Module.

A coolant-water interface exists between all core modules so that heat can be transferred between the modules with the interchange loop of the thermal control system. This enables supplementary cooling of a module with an inoperative of deficient radiation loop.

The solar collector, which is located on the Power/Subsystems Module, provides heat to the Crew/Operations and the GPL Modules for EC/LS process heat. Two heating-water loops transport this heat and this function requires heating-water interfaces between the core modules.

Pumpdown and repressurization lines interface between all core modules so that the pumpdown system in the Power/Subsystems Module can service the airlocks in the GPL and the docking ports on the Power/Subsystem and Crew/Operations Modules.

The Space Station design is designed so that either of the two atmosphere reconditioning units can process air for the entire Space Station. This requires purified air ducts for supply and return between all core modules.

Transfer of potable, used wash water, and purified wash water are required because the water recovery equipment is located in the Crew/Operations Module but a portion of the water is used in the GPL. Additionally the contingency water supply is located in the GPL and may be required for use in the Crew/Operations Module. Therefore interfaces exist between the GPL and Crew/Operations Modules for potable water, purified wash water and unpurified wash water.

The interface between Crew/Operation Modules for the GSS, configuration are identical as for the Logistics Module and in addition includes a potable water line. This is necessary to maintain a water balance between the two water recovery units; located in each Crew/Operations Module. Cross transfer of potable water is necessary if potable water use rate or waste water generation differs greatly between modules.

Line sizes for the various types of interfaces are given below.

Description	Diameter cm	(inches)
O_2 Supply	1.27	(1/2)
N_2 Supply	1.27	(1/2)
Pumpdown	7.61	(3)
Repressurization	5.06	(2)
Atmosphere Purification	20.3	(8)
Water Lines	1.9	(3/4)

4.6.3.3 EC/LS Operation

EC/LS operation differs with each mission phase, based on the EC/LS functions required. For instance, during unmanned operations, atmosphere purification and water processing is not required. These functions are not activated until just prior to manned operations. This includes humidity control, CO_2 removal, trace contaminant control, and urine and wash water recovery. Other EC/LS functions, such as fluid circulation through the solar collector, must be activated as soon as the solar arrays are activated. In the following paragraphs the required EC/LS functions and startup sequence are described for the various mission operation modes.

Operation of the key EC/LS assemblies is controlled by individual controllers on each assembly. Therefore, key functions will not rely on the Central Data Management System. These key functions include pressure control, humidity control, CO_2 removal, air temperature control, water management, and thermal control. A habitable environment will be maintained within the Space Station with these functions. The control needs of these EC/LS functions are simple in nature and can be performed without complex computer logic. They consist of simple timers, switches, and sensors. Other functions are essentially manual in nature and are primarily crew-controlled and operated. These include airlock pumpdown, waste management and IVA/EVA.

Major usage is made of the Data Management System (DMS) for alerting the crew to malfunctions and potentially hazardous conditions in the EC/LS. Additionally the DMS is used for performance trend analysis, inventories of expendables, mass balances and equipment checkout and diagnosis.

Prelaunch

During prelaunch the EC/LS Subsystem is checked out to ensure proper operation of all equipment. Performance checking will be performed where this is compatible with the desired launch condition; (e.g., contaminated water will not be added to check out the water recovery units). Wash water storage tanks and urine storage tanks will be partly filled with pure water to ensure ullage as required for normal system operation. The water storage tanks will also be filled but need not be heated to pasteurization temperature because no substrate will be contained in the water to support bacteria growth.

Both silica gel beds and molecular sieve beds will be fully regenerated and ducts capped off so that, when normal operation begins, a backout cycle will not be required.

During ground checkout, dry and purified air is supplied to the Space Station. This will enable the checkout crew to work in the module without active EC/LS operation. Additionally, this will reduce the possibility of moisture condensation followed by material degradation during unmanned operation.

Just prior to launch, the module EC/LS equipment and atmosphere are precooled to provide for a heat sink during launch. The thermal capacitor which operates by material phase change provides much of the cooling capacity during launch.

Launch and Deployment

During launch the Thermal Control Systems are operated with the freon loop in the bypass mode of operation. Due to precooling, the Thermal Control System can dissipate the heat load until orbit when cooling is provided by the ground heat exchanger/water boiler. The radiators are activated when the module is erected from the Shuttle bay and provides cooling from that point on.

During launch and deployment the bulk of the EC/LS assemblies are inactive; active assemblies are required for air temperature control and for coolant water and radiator loop recirculation.

Activation for Initial Unmanned Operation

Initial unmanned operation requires activation of a number of EC/LS functions in addition to those used during launch and deployment. They are as follows:

A. O_2 supply
B. N_2 supply
C. Pressure control
D. Solar collector activation

The Power/Subsystems Module is launched without the contingency O_2 and N_2 supply. These supplies must be transferred from the logistics module prior to power module activation for unmanned operation.

Heating water recirculation is initiated just prior to solar array deployment to avoid excessive solar collector temperatures. The heat from the solar collector is transferred to the radiator loops within the three modules via the waste heat exchanger and the interchange loop. This also provides heat to the radiators which prevents possible Freon 21 freezing due to the low heat dissipation required during unmanned operation. After docking of the Crew/Operations Module, the heating water will bring the recovered water storage assembly up to pasteurization temperatures.

Activation for Manned Operations

Additional equipment besides that described above for unmanned operation must be activated for manned operations. The Thermal Control System and Pressure Control System are first activated followed generally by the activation of functions in the order as required by the crew. Humidity control is initiated because humidity buildup is rapid. Next the CO_2 removal and trace contaminant control assemblies are activated. These assemblies are activated by positioning switches; no special startup sequences are required. After docking of the Crew/Operations Module to the Power Subsystem Module, the atmosphere distribution fan is activated.

Urine water recovery is activated prior to use of the waste management equipments. One of the two urine storage tanks initially contains pure water and this supply must be processed through the urine recovery assembly so that, prior to urine availability for processing, potable water for crew consumption is made up. This same procedure is required for the wash water recovery assembly. Since no wash water is available for initial operation, pure water is provided in the wash water storage tank to provide wash water system initial charge. The remaining EC/LS assemblies are activated as required by the crew. These include airlock pumpdown, waste management, and ventilation assemblies.

Standby Unmanned Operation

In the event of a major failure in the Space Station modules or the logistics elements, sustained unmanned orbital operation in a standby mode may be required for a period up to one year. This contingency mode may also involve removal of one or more modules from the Space Station cluster and return of these modules to the ground for refurbishment.

If the cluster remains intact, on-orbit, or if any module other than the Power/Subsystems Module is removed, the Station will remain in a stable horizontal orientation and all functions other than those required for life support and experiment support will remain active. In the EC/LS Subsystem, primarily the thermal control loops will remain active while atmosphere conditioning, water recovery, and other processes will be shut down.

If the Power/Subsystems Module is removed, power, communications, attitude control and other functions will be terminated. Insufficient power is available in the Crew/Operations and GPL Modules to maintain operation of the active thermal control loops and operation in a passive thermal control mode will become necessary. Since the Station heat balance is highly dependent on orientation, shadowing and solar absorptivity of the surface, the equilibrium temperature will depend on the state of these parameters during the period of unmanned, "standby," operation. Therefore, the design of the EC/LS subsystem must accommodate the resulting temperature extremes without incurring damage that cannot be readily corrected during the manned reactivation operations. If the timing and conditions associated with such an event permit, subsequent reactivation phase. These procedures, which remain to be developed in detail, may include such things as: removing or relocating highly temperature-sensitive components, removing waste materials and other sources and nutrients for bacterial growth, sterilization of lines and tanks, depressurizing lines, closing valves and hatches, etc.

If periodic servicing of the Station occurs (e.g., if the Shuttle docks to initiate an orbit altitude change), assessment of on-board status and checkout of key components will be performed to facilitate planning for subsequent reactivation.

Contingency Operations

A number of alternate and degraded modes of operation are available for providing each essential EC/LS function. In most cases at least one alternate mode of operation is provided wherein the normal vehicle operation is not interupted in the event of a failure. This provision results in no crew reaction being required until the next normal maintenance period. If more than one failure occurs in the equipment providing an essential function, prompt maintenance and repair is recommended. In the event an essential function cannot be restored, derated operation or use of the onboard contingency supplies may become necessary. In the following paragraphs, the alternate modes of operation available for each essential service will be discussed.

A number of alternate means of pressure control are available in the event the primary equipment fails. Two pressure control assemblies are provided, one in the Crew/Operations Module and one in the GPL Module. Either device can pressurize the entire Station through open hatches. Also, two sources of O_2 and N_2 gas are available; the normal supply in the Log Module and the contingency supply in the Power/Subsystems Module. A third alternate method of pressure control is afforded by the repressurization valve at each docking port which can be actuated from either side of the docking port.

If all normal means of pressure control fail, the oxygen partial pressure would degrade at a rate of 0.92 kn/m^2 (0.35 psi) per day. Therefore, several days would pass before a dangerous situation would exist. The nitrogen level would degrade at a slower rate because of the low leakage rate in the Space Station.

A small supply of O_2, sufficient for three crewmen's metabolic needs, are available on each of the 96-hour pallets as a final pressurization backup.

Completely redundant atmosphere purification units are located in the Crew/ Operations Module and the GPL Module. These provide for CO_2 removal, humidity control, and odor and trace contaminant control. If one unit fails, valves are positioned in the atmosphere purification ducts so that the entire Space Station is serviced by the operative atmosphere purification unit.

The three major functions provided by the atmosphere purification units are essentially independent. That is, if the CO_2 removal system is inoperative, the unit still provides for humidity, odor, and trace contaminant control. An exception is the CO_2 control unit which relies on the humidity control condenser for low dew-point temperature process gas. If the condenser becomes inoperative, the CO_2 removal unit would degrade in performance and an occasional silica gel bakeout cycle would have to be performed. If both CO_2 removal units become inoperative, the CO_2 level would rise at a rate of 0.034 kn/m^2 (0.25 mm H_g) per hour and the contingency level of 1.01 kn/m^2 (7.6 mm H_g) would be reached in 18 hours. After this time the emergency pallets would be used; each pallet affords 6 man days of CO_2 and humidity control.

The humidity level in the Space Station is not as critical as CO_2 level. If no humidity control was present in the Space Station, the level would rise to around (60°F) dew point temperature, after which condensation would occur in the air temperature control heat exchanger or on the walls. This condition is undesirable because the presence of free water could cause some impairment in equipment operation. Serious damage, however, is not likely to occur and this condition could probably be tolerated in a short-term emergency.

Odor and trace contaminant control is not a short-term essential function and loss of this function can be tolerated in emergencies. A good amount of odor and trace contaminant control is afforded by the molecular sieve beds and the condensing heat exchanger. If these units are in operation some control will be available in the Space Station. The major disadvantage to operation without the odor and trace contaminant system is that no atmosphere cleanup capability is present in the event of a fire or spillage of a contaminant in the cabin.

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Air temperature control is a major function which is not backed up in each module. The reason is that reaction time is long in the event of a failure, and the Air Temperature Control Assembly is adaptable to a quick change approach to maintenance. Additionally, a fail unit does not result in a serious threat to continued full operation of the station. Some crew discomfort may result and crew activity may have to be reduced in the effected locations. Some continuing air cooling will be afforded by the humidity control condensors and the free air movement through open hatches of adjacent modules which are operating normally. Following multiple failures of all other means of temperature control, the 96-hour pallets will be used.

Loss of operation of the atmosphere distribution fan in the Crew/Operations Module can be substituted by closing the hatch between the Crew/Operations Module and the Power/Subsystems Module. In this manner, purified atmosphere is forced through the atmosphere distribution ducting by the atmosphere purification fan. The fan in the atmosphere purification unit located in the GPL Module is a backup to the force. Due to the ventilation afforded by the air temperature control unit, good distribution is obtained in all modules. This, along with the process of natural diffusion, should result in a viable atmosphere throughout the Space Station even if the entire atmosphere distribution system should fail.

Short-term failure of the urine water recovery unit can be tolerated because ample storage facilities are provided for urine and potable water. The unit is designed with excess capacity so that the unit can catch up after the failed condition is rectified. If the failure cannot be repaired on orbit, and resupplied parts are required, the 30-day contingency supply would be used. Additional water stores are available on the 96-hour pallet.

Loss of a single thermal control loop in a module will not incapacitate the Space Station because sufficient contingency cooling will be available normally from other modules via the interchange loop. Loss of radiators or overload conditions can be tolerated from a short-term standpoint by use of the ground heat exchanger/water boiler. Minimal heat rejection capacity is also available on the 96-hour pallets. This is sufficient to provide crew cooling and pallet equipment cooling.

Redundant tube sets and redundant heating water loops are provided on the solar collectors to protect against major failures. Additional backup is provided in the CO_2 removal system by electrical heaters which can be activated if the heating-water loops become inactive.

4.6.3.4 Growth Space Station (GSS) Considerations

Minimum EC/LS modifications are required in the designs of the second Crew/Operation and Power/Subsystems Modules added to obtain the GSS configuration. The EC/LS design for the add-on modules can be identical to their counterparts in the ISS, however, some provisions can be conveniently deleted. These include the airlock pumpdown, the secondary heating water loop, and contingency gas storage. The airlock pumpdown facility located in the ISS Power/Subsystem Module is sized for pumpdown requirements for both the ISS and GSS and an additional pump is not required. They may be added however if it is desired to reduce the pumpdown times. Addition of pumpdown accumulators at the GSS level would enable the RAMS accumulators to be eliminated or reduced in size.

Two separate heating water loops are provided at the ISS stage to service the two EC/LS Subsystems located in the Crew/Operations and GPL Modules. At the GSS, only one additional EC/LS Subsystem is added, only one heating-water loop is required. Redundant loops are not required because there is ample EC/LS backup capacity in the first Crew/Operations Module and GPL. However, redundant circuits in the solar collector should be retained because of the possible occurrence of unrepairable failures there.

A second contingency O_2 and N_2 storage facility may not be required at the GSS level because the requirement for capacity to pressurize the largest habitable volume is identical for ISS and GSS. The largest habitable volume at the ISS include the Crew/Operations and Power/Subsystem Modules. The second habitable volume is the GPL Module. At the GSS, the largest habitable volume remains as the Crew/Operations and Power/Subsystems Modules. Prior to the buildup to GSS, there should be sufficient ISS data on repressurization needs to influence the decision on inclusion of additional contingency O_2 and N_2 at the GSS level.

A number of provisions are included in the ISS design to facilitate Station growth to the GSS. It was previously mentioned that airlock pumpdown and N_2 and O_2 storage were sized to accommodate both ISS and GSS. In addition, the docking port interface between the first Crew/Operations Module (ISS) and the second Crew/Operations Module (GSS) contains interface connections as required for growth. Atmosphere purification ducts are included to allow interchange of purified air between Crew/Operations Modules during emergencies. Therefore, if either EC/LS is inoperative, adequate atmosphere conditioning is obtained from the operative EC/LS in the adjacent module.

N_2 and O_2 connections are provided at the interface so that the atmosphere contingency stores in the ISS Power/Subsystems Module are available for GSS module use. Additionally, these connections allow Logistics Module gaseous stores to be used in any core module in the station independent of Logistics Module docked location.

Pumpdown and repressurization line connections are provided at the ISS/GSS interface so that the pump facility in the ISS can be used to pumpdown modules attached to second Crew/Operations or Power/Subsystems Module.

A potable water line is provided at the buildup interface so that a water balance can be maintained between water management equipment located in the Crew/Operations Modules. This is necessary since the crew may be consuming potable water and producing waste water in different proportions in each Crew/Operations Module.

A number of EC/LS improvements must be considered for the GSS level to (1) improve performance and (2) to provide a testbed for advanced systems. Addition of oxygen recovery is a major candidate for addition at GSS. The ISS level Crew/Operations Module would be retrofitted with the O_2 recovery equipment and the same design would also be added to the add-on Crew/Operations Module for GSS. Addition of O_2 recovery would greatly reduce the resupply requirement at the expense of more costly systems and higher power usages.

Due to continuing development of monitoring equipment during the ISS stage, more advanced and better performing monitoring equipment should be available for GSS. Two prime candidates for upgrading are the trace contaminant and water potability monitors.

Radiator coating degradation data will be derived due to the ISS mission phase. This data along with continuing ground testing and development will result in radiator coatings which are less susceptible to radiation and effluent contamination. These should be included on module add-on's for the GSS.

Much crewtime is consumed during the ISS for O_2 prebreathing before EVA. This loss of valuable crewtime can be eliminated or greatly reduced by the availability of a higher pressure suit than the current 25.4 kn/m^2 (3.7 psia) design. Development of a Portable Life Support System (PLSS) must accompany such a suit design and is a prime candidate for the GSS mission. Additional improvements may be available in the PLSS in the areas of reducing expendables by using regenerable materials and by making the PLSS maintainable.

From time to time, expendables must be replaced in EC/LS equipment. Expendables requiring replenishment include filters, absorption beds, wick packages and membrane stacks. The design allows replenishment of these items with a minimum of interruption to normal operation. Replenishment can normally be scheduled by the onboard Data Management System based on time duration or performance degradation. The EC/LS is designed for replenishment of the expendables with minimum time requiremenets and to minimize the risk of EC/LS fluids escaping into the cabin.

Much of the EC/LS equipment is operated continuously, regardless of the processing load. This includes humidity control, air temperature control, and the thermal central system. Operation of these assemblies at reduced rates is not practical. However, this is not the case for CO_2 removal and the water recovery assemblies which represent large power penalties.

The CO_2 removal unit operates on a demand basis but the control is designed so that only one of the two units provided operate at a time. Each is sized for a 6-man crew working at a high CO_2 production rate so that an over-capacity of about 200 percent is provided. The control activates the CO_2 removal unit in the module where the CO_2 level is the highest. For instance, if the CO_2 level is highest in the GPL Module, the unit located there will be activated. When the CO_2 level is reduced to a level below that in the Crew/Operations Module, the CO_2 removal unit will be shut down at the end of its normal 80 minute cycle. The unit will then be activated in the GPL Module until the CO_2 level again builds up in the Crew/Operations Module.

The urine water recovery unit is operated at reduced rates when the demand is low by reducing the amount of heat added to the air stream. In the event maintenance operations have caused the unit to fall behind schedule, sufficient capacity is provided for a "catch up" mode of operation.

The reverse osmosis unit operates in a batch mode and is shut down periodically during low wash-water use times. Reactivation requires only the activation of a switch to bring the unit back on line.

The EC/LS is designed for maintainability down to the component level. Maximum use is made of quick fasteners to minimize crew time. All maintenance can be performed without EVA. Equipment containing the radiator loop freon equipment is located in a separate air-tight compartment. This prevents freon 21 from escaping into the module atmosphere due to a failure. In the event maintenance is required, a hatch provides access to faulty equipment. If a leak should occur within the compartment, the freon is purge overboard by vent valves which allow a small cabin air flow through the compartment. This purge gas flow can be maintained during the maintenance activity. The use of the air tight compartment for freon equipment prevents freon from escaping into the atmosphere and does not require EVA for EC/LS maintenance.

4.6.4 Design Analyses and Trade Studies

Trade studies have been performed at three distinct levels in accordance with the study plan for the Modular Space Station preliminary design. These levels, along with the major trades performed at each level, are shown in Table 4.6-8.

Table 4.6-8

EC/LS TRADES AT EACH PROGRAM LEVEL

Subsystem Level

- Degree of oxygen loop closure.
- Degree of water loop closure.
- Process heat trade.
- Compartment pumpdown trades.
- Water recovery by CO_2 hydrogenation.
- EC/LS modularity.
- EC/LS backup options.

Assembly Level

- Atmosphere storage method.
- Regenerable versus nonregenerable charcoal.
- CO_2 control method.
- Urine water recovery method.
- Wash water recovery.

Assembly Integration Level

- Air distribution ducting size.
- Coolant water line size.

Not shown in Table 4.6-8 is the thermal control trades and analyses which was a continuing effort at all program levels. This effort addressed the following areas of analysis:

- Thermal coatings.
- Deployable or fixed radiator design.
- Radiator tube meteoroid protection.
- Radiator fluid selection.
- Radiator fluid temperature control.
- Radiator fluid flow regime (laminar, turbulent).
- Radiator segmentation.
- Radiator tube layout and spacing.
- Independent or integrated radiators for the modules.

These areas of effort will be presented in detail in subsection 4.6.4.5.

A summary of key trades and analyses are presented in Table 4.6-1. Also given is the disposition for each activity.

Essentially, the subsystem level trades establish which processes are to be used for providing the major EC/LS functions. The trades are performed only in the case where more than one competitive method is available. Results of the 10.06 m (33-ft) diameter Space Station Study were used to establish candidates. Quantitative data for the subsystem level trades were based on what appeared to be the most competitive method. For instance, the urine water recovery trade assumed air evaporation for the recovery process since it was selected in the 10 m (33-ft) diameter Space Station study. Where necessary, trades considered more than one concept to ensure valid results. As an example, is the O_2 recovery trade which considered both molecular sieve and hydrogen depolarization for CO_2 control methods.

The oxygen and water trades at the subsystem level are dependent on the penalty used for process heat which differs for solar collection, isotope sources and solar cell electrical power. The closure trades were performed for electrical heat as well as for solar collection. The trades did not include options using isotope sources because the penalty is nearly identical to that for solar collection.

After the subsystem level trades determined the EC/LS processes to be used, the assembly level trades were performed to select the methods to perform each process or function. The results of the 33-ft diameter Phase B Space Station study were examined and the number of candidate methods reduced to a minimum. An industry survey was performed to obtain the latest available data on assembly level equipments. This data was useful in the selection of candidate methods.

The assembly integration level trades were performed to establish line sizes between assemblies and to determine the optimum arrangement of assemblies.

The thermal control analyses consisted of two distinct efforts: (1) a digital computer analysis to determine the heat rejection radiator performance, and (2) detailed design analyses and trades to optimize design characteristics for the radiator system elements. The computer analysis was a transient lumped parameter analysis which used the MDAC PO-333 computer program to obtain orbital heating rates and the MDAC JA-15 computer program to perform the transient thermal analysis around the orbit. Additional analyses were performed to establish the required radiator tube arrangement, radiator fluid flow, direction and flow splits between panels, inlet manifold locations and optimum tube geometry. Heat transfer and meteroid protection considerations were involved in selecting the optimum tube geometry.

4.6.4.1 Selection Criteria/Approach

Criteria for selection of preferred concepts were divided into three categories: absolute, quantitative, and qualitative.

<u>Absolute Criteria</u>

Absolute criteria are those that are mandatory; i.e., if the concept does not meet minimum requirements then it is considered unacceptable. The following criteria are considered absolute:

A. Performance
B. Safety
C. Development Status

Normally, any concept which could not be designed and qualified in time to meet the scheduled launch date would be considered unacceptable under the development status constraint. However, recognizing that breakthroughs could occur, that advanced development is possible, or that the program schedule could be delayed; the development status constraint was not held rigid. However, the risk that would be incurred with a particular concept was considered under qualitative criteria.

Quantitative Criteria

The criteria which are assessed quantitatively are as follows:

A. Launch weight.
B. Launch volume.
C. Power requirements.
D. Crew time for maintenance.
E. Expendable requirements.
F. Spares requirements.
G. Hardware cost.

Cost effectiveness was used to correlate or weigh the relative merits of each of these criteria. Figure 4.6-19 shows how this method was implemented. It should be noted that total relative cost factor is not an actual cost, but should be considered strictly as a number by which to quantitatively compare the relative merits of different concepts.

Figure 4.6-19 also shows the values for the dollar normalization factors which were used in the trades. The rationale for the development of these values is contained in Data Requirements Document MP-01.

Analysis of power requirements merited further consideration because the EC/LS subsystem is the largest single user of electrical power on the Space Station outside of the experiment program. With a solar cell/battery type of power system, there are different penalties associated with the different types of electrical power. Table 4.6-9 shows the values which were used in the tradeoffs.

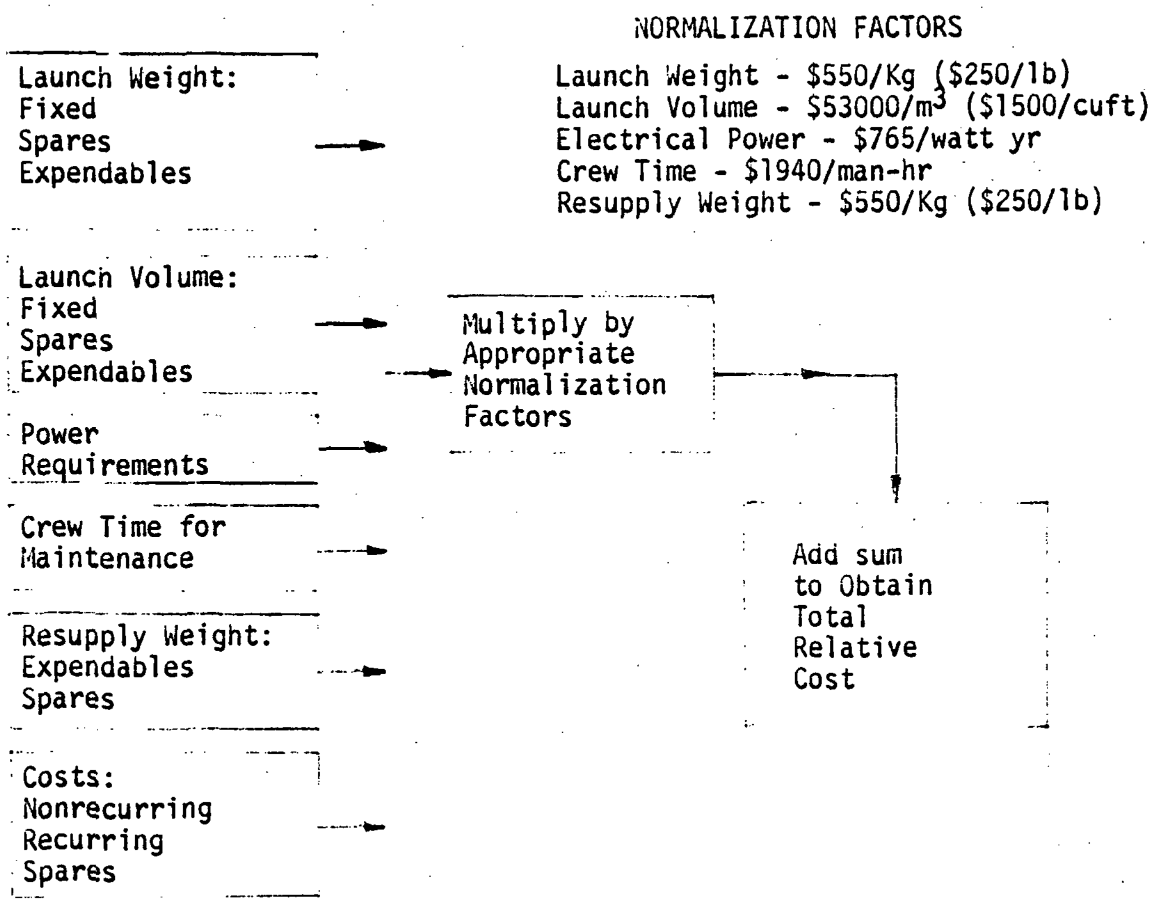

Figure 4.6–19. Cost Effectiveness Implementation

Table 4.6-9

SOLAR CELL/BATTERY POWER PENALTIES

Type of Power Used	Penalty-kg/kwe	(lb/kwe)
Continuous (92 min/orbit)	342	(752)
DC (load bus)	342	(752)
AC	354	(780)
Illuminated Only		
DC (source bus)	98	(216)
DC (load bus)	134	(294)
AC	147	(323)
Eclipse Only		
DC (load bus)	256	(563)
AC	268	(590)

These factors were used in the individual tradeoffs to help determine the optimum design for a particular concept. For example, CO_2 removal concepts were considered which operated on unregulated dc, sunlight side only, because this results in a lighter combined EC/LS-EPS System by virtue of the factors above.

Costs were initially compared without discounting, and the effects of discounting on the results were estimated. Where it appeared to affect the comparison a revised analysis was prepared using cost discounting.

<u>Qualitative Criteria</u>

The criteria which are assessed qualitatively are as follows:

A. Flexibility—to accommodate alternate mission/requirements.
B. Growth Potential.
C. Interface Sensitivity.
D. Development Risk.
E. Complexity.

While these factors are more difficult to quantify than those given in the previous list, they are extremely important in comparing alternatives and must be given equal consideration in the trade studies. In performing tradeoffs and making concept selection, it is also important to consider interactions between the subsystems—especially in those areas where a synergistic advantage may result. This has been done in many cases. Significant advantages result from satisfying requirements on a system level rather than treating each concept independently. An example is the resistojet trade reported in subsection 4.8.4 which considers the full impacts between the Propulsion Subsystem and the EC/LS Subsystem.

4.6.4.2 Subsystem Level Trades

4.6.4.2.1 Degree of Oxygen Loop Closure

Oxygen makeup for crew metabolic needs can be either resupplied in the form of gas or it can be recovered from the crew produced CO_2. Oxygen recovery concepts are more complex and costly initially than O_2 resupply, however, the cost to resupply O_2 becomes large for long-mission durations. Only partially closed oxygen was considered since, with a partially wet food diet, partially closed O_2 produces all the water required for O_2 production.

One of the two oxygen recovery concepts considered consists of molecular sieve CO_2 concentration, Sabatier reactor for CO_2 hydrogenation, and a wick-feed water-electrolysis unit. These were the selected concepts for the 10 m (33-ft) diameter Space Station and can be validly assumed here because of the similar trade factors used for the two Space Station studies.

The other concept is similar except a hydrogen depolarized concentrator replaces the molecular sieve. This offers a weight reduction and produces some electrical power.

Data for open oxygen loop assumes gaseous O_2 resupply and vacuum dump molecular sieve for CO_2 control. Gaseous storage was used rather than cryogenic storage because of the low initial cost guideline. An assembly-level trade was performed which resulted in selection of gaseous storage;

this trade is reported in subsection 4.6.4.3.1. Additional penalties occur if CO_2 must be saved for use in a resistojet low-thrust propulsion system and these are included in a separate resistojet trade study reported in subsection 4.8.4.

Oxygen Recovery Concept—Molecular Sieve CO_2 Removal

Carbon dioxide is removed from the cabin atmosphere by a regenerative molecular sieve unit. Relatively dry process air is drawn from the humidity control assembly and passed through silica gel beds which further dry the air prior to passage through the molecular sieve beds. Both the water vapor and CO_2 are removed by absorption processes. CO_2 is removed from the beds by heat addition and pumpdown. The silica gel beds are regenerated by recycling hot dry process air through them.

Since CO_2 is pumped from the molecular sieve beds in a cyclic fashion, an accumulator is provided to damper out the CO_2 pressure variations. This results in a relatively constant CO_2 flow to the Sabatier reactor.

The Sabatier reactor reacts H_2 and CO_2 to produce water and methane. The water is collected in a condenser and pumped to the electrolysis cell which produces gaseous O_2 and H_2. The H_2 is used in the Sabatier reaction and the O_2 is supplied to the cabin for metabolic and leakage makeup. Water electrolysis produces sufficient H_2 to react about two-thirds of the available CO_2. The remaining CO_2 and methane gas are available for use in a resistojet low-thrust propulsion system.

The amount of water produced by the Sabatier is about one-half the total required by the electrolysis unit. The additional water required is assumed to be made up with excess water from the Space Station water management system. Sufficient excess water is available in the baseline design due to the selected crew diet which is 40-percent water.

Oxygen Recovery Concept—Hydrogen Depolarizer (HDC) CO_2 Removal

The HDC receives cabin air from the humidity control system and removes CO_2 by an electrochemical process that transports CO_2 from a low partial pressure in the stream through an electrolyte to a high partial pressure at the concentrated CO_2 outlet. The reaction is driven by an electrical potential produced by a fuel cell reaction of O_2 in the process stream and H_2 on the concentrated CO_2 side. Water vapor produced in the fuel cell reaction is removed in the concentrated CO_2 side of the concentrator cells. Excess H_2 is fed into the CO_2 cavity and is delivered with the concentrated CO_2 to the Sabatier reactor.

An electrolysis unit provides hydrogen for reaction in the HDC and the Sabatier reactor and oxygen for reaction in the HDC and for metabolic makeup. The amount of CO_2 recovered and water makeup required is the same as oxygen recovery concept using molecular sieves.

Open Oxygen Concept

The open oxygen concept consists of a vacuum desorbed molecular sieve unit for CO_2 control with crew metabolic O_2 makeup being resupplied as gaseous oxygen. Because there is no need to save the CO_2 for O_2 recovery, it is rejected directly overboard. This reduces vehicle penalties associated with heating and mechanical pumpdown required for CO_2 save.

Sufficient gaseous O_2 is stored onboard the Space Station for crew metabolic needs for 30 days. The normal 90-days resupply is provided onboard the Logistic Module and is withdrawn as needed by an attached line to the Space Station. Redundant O_2 tankage is not considered here because of the following reasons: (1) high-pressure gas storage is inherently reliable; (2) tankage can be separated into two or more banks to minimize potential losses; and (3) the onboard contingency supply is available for emergencies.

Trade Results

Figure 4.6-20 shows the results of the cost tradeoff for two methods of providing process heat, i.e., electrical and solar heat. The results show that oxygen recovery is not cost effective for the planned 10-year program duration for either form of process heat. Based on solar heat, open oxygen costs 13 million dollars less initially and 1.5 million dollars less for the 10-year mission. Comparing costs for the electrical heat designs, open

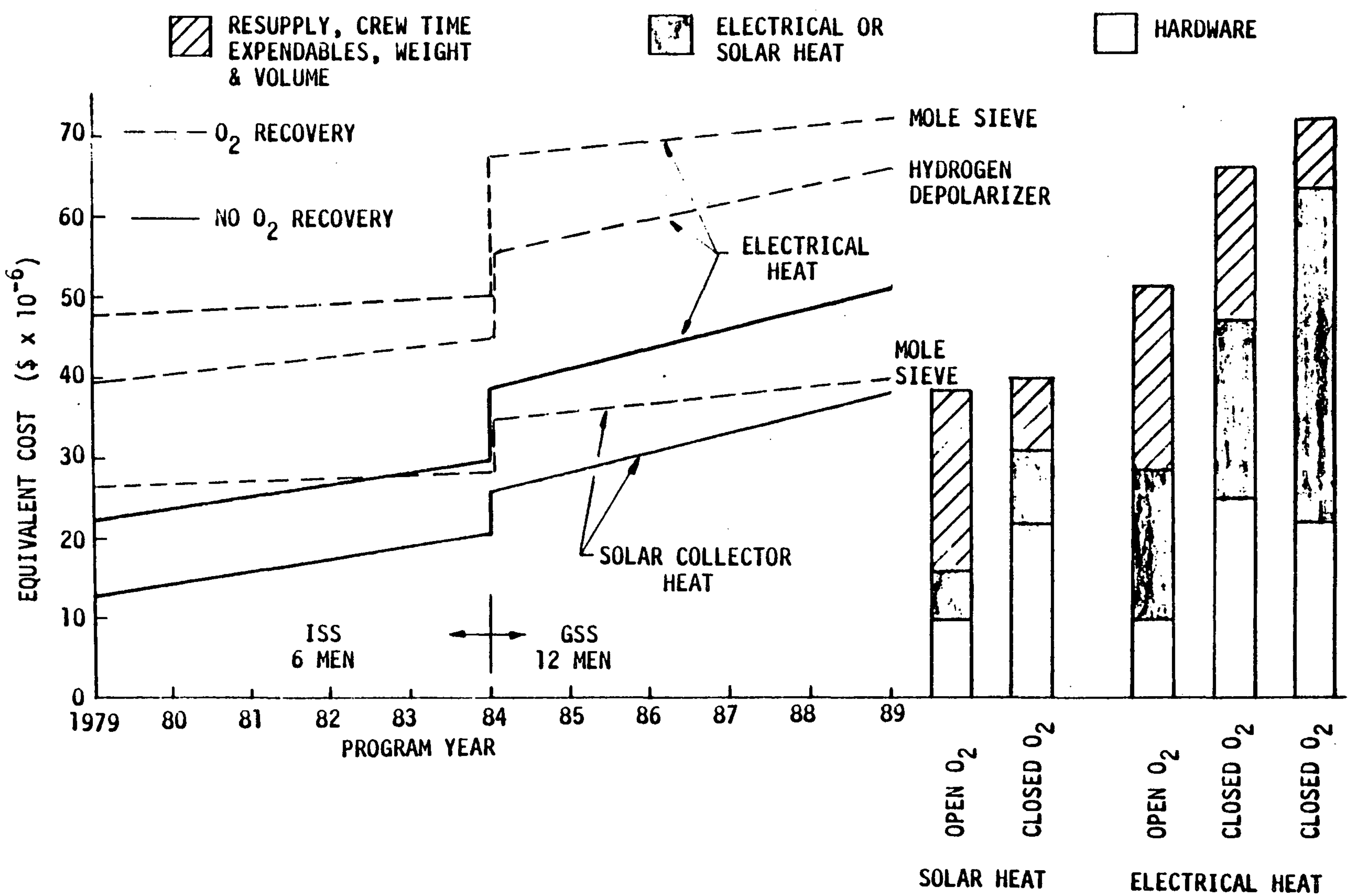

Figure 4.6–20. Oxygen Recovery Trade

oxygen costs 26 million dollars and 17 million dollars less initially and 21 million dollars and 15 million dollars less for the 10-year mission for the molecular sieve and hydrogen depolarization concepts, respectively. Considering initial cost and total program cost, an open oxygen loop is favored.

The qualitative consideration which is expected to have the greatest impact involves the limitation of one Shuttle launch per month. The closed oxygen design reduces resupply by about 700 lb per month during GSS. Resupply requirements will be increasingly important as the station builds up and logistics requirements grow. Therefore, oxygen recovery may be desirable for growth versions of the Space Station.

Based on cost and qualitative considerations presented above, open oxygen is selected for the initial Space Station launch. It is recommended that provisions for oxygen recovery retrofit be incorporated in the design so it can be added at a later date if desired. This will enable large reductions in logistics resupply during buildup phases of the mission.

The trade data for oxygen recovery assumes the availability of water makeup for electrolysis. This is a valid assumption with a wet food diet containing at least 40-percent water. If dry food were resupplied and reconstituted from the onboard potable water supply, water for electrolysis would have to be resupplied and an additional cost would be incurred. This cost would amount to $8 million for the 10-year period. On the other hand, the open oxygen concept results in excess onboard water with a wet food diet. This excess water can be used for experiment water makeup or a dry food diet can be adopted for a net savings of about $8 million in resupply costs.

Both the open and partially closed oxygen systems result in CO_2 and methane gases which must be removed from the Space Station in some manner. Dumping them directly overboard will interfere with some experiments and this is undesirable. Ejecting the gases overboard through a resistojet propulsion system results in less experiment interference but the effects are not totally known. If ejection overboard of the gases is found to be unacceptable, the gases must be stored and returned to Earth which results in sizable penalties for compression and tankage. A better solution appears to be to close the O_2 loop completely and react the CO_2 to carbon which can be easily returned to Earth. For this reason retrofit to full O_2 recovery is a possible design option to be considered if experiment interference by CO_2 and CH_4 appears likely from further testing and analysis.

4.6.4.2.2 Degree of Water Loop Closure

Water is required by the crew for intake, body washing, clothes washing, and food preparation. Part or all of this water may be supplied or it may be recovered from waste water sources, i.e., urine, condensate, and wash water. Water recovery from feces is not considered in the trade because of the high penalty for recovery and the small amount of water involved. Water recovery from wash water and condensate was guidelined and trades performed early in the Phase B extension period confirmed this decision. Generally speaking, water can be recovered from wash water and condensate at a low penalty and it is convenient to do so. This is because several acceptable concepts are available which do not require complex phase change processes.

Urine water recovery is more costly because candidate concepts require a phase change. The trade performed herein is based on the air evaporation technique which was selected in the 33-ft-diameter Phase B study. Due to the similar trade criteria, the results of that study are applicable to the Modular Space Station.

Urine Water Recovery Concept

Air evaporation is an ambient pressure distillation process in which a carrier gas is used to evaporate water from wicks saturated with urine which has been chemically pretreated to fix ammonia and kill bacteria. The air evaporation unit employs a closed loop consisting of a heater, wick package, a regenerative heat exchanger, a condenser/separator, and a fan. Air is heated by the heater and then passed over the evaporator wicks where water evaporates into the air process stream. The moist air then passes through heat exchangers where relatively pure water is condensed from the gas stream. The thermal energy required by the distillation process can be provided either by electrical heat or by solar collector heat.

Further treatment of the condensate is necessary by passage through charcoal beds and filters before storage. Other peripheral equipment for the air evaporation unit consists of pretreatment chemical tanks, ejectors, urine storage tanks, feed equipment, valves, and controls. Three small water storage tanks are required to ensure the recovered water meets potability standards. This requires a fill tank, a use tank, and a standby tank on test status.

Two complete urine water recovery units and the associated tanks, pumps, and controls are provided at initial launch. Resupply is required for filters, air evaporation wicks, charcoal beds, spares, and pretreatment chemicals. An additional urine water recovery unit is launched in year 1984 followed by the crew buildup from 6 men to 12 men. Resupply requirements also increase due to larger GSS Space Station crew size.

Stored Water Concept

The alternate to recovering urine water for reuse is to resupply potable water for crew intake. In the latter case, urine is pretreated to prevent bacteria growth and then stored for return to Earth. Earth-return is preferable to overboard dumping for contamination reasons.

The stored water concept is highly reliable and represents less initial cost than recovery because of hardware simplicity. However, large quantities of water and tanks must be resupplied.

Before use, each water tank must be checked for water purity. It is assumed that water storage at pasteurization temperatures will not be necessary because resupply water is low in nutrients. Therefore, electrical energy is small, and required only for valves, control and pumps.

The initial launch provides potable water, urine pretreatment chemical and tankage for the 6-man crew for 30 days. Water is supplied with the

Logistics Module for an additional 90 days. Redundant water supplies are not provided because of its inherent reliability and ability to store the supply in two or more separate locations. Additional tankage is launched in the year 1984 to accommodate the crew buildup. Resupply rate also increases due to the crew size increase.

Trade Results

Figure 4.6-21 gives the accumulated cost for urine water recovery and the stored-water concept. Urine water recovery using solar heat trades favorably after four years of operations. Cost savings are much larger after the crew buildup to larger crews due to the increased resupply costs for the larger crew. For the ten-year mission, urine water recovery costs about 23 million dollars less than water resupply.

The dominant costs are: hardware and power costs for the recovery concept and resupply costs for stored-water concept. Urine water recovery using solar heat costs about six million dollars less than the same concept using electrical heat due to the lower cost of solar heat energy.

Initial cost of the stored water concept is about nine million dollars less than the recovery concept, however, the large resupply requirement for stored water may be excessive for the logistics system during early portions of the mission. The stored-water concept has a major impact on the logistics system design since about 6,200 lb of expendables are required every 90 days for 12 men.

Reliability, safety, and complexity considerations all favor the stored-water concept because it is very simple. However, it offers little mission flexibility because of the large logistics requirements which must be launched on a scheduled basis. Although the stored-water concept has a major impact on the logistics system, the interfaces with the Space Station is minimal.

Urine water recovery requires large amounts of process heat and some electrical power and hence requires the greatest amount of integration with other systems. Growth potential favors water recovery because the concept

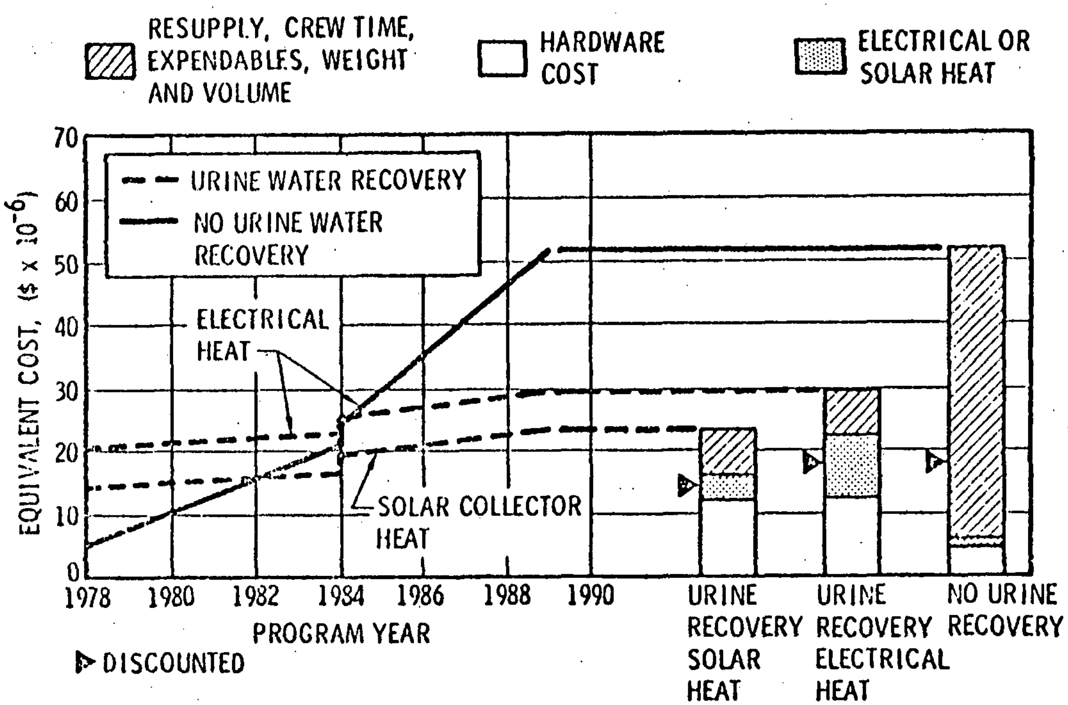

Figure 4.6–21. Urine Water Recovery Trade–Option B

has much greater capacity than is normally used. Because the recovery units are sized to allow for down time for maintenance, the concept has capacity for up to 16 men at the ISS and 24 men at the GSS.

From a qualitative standpoint, the advantages due to simplicity for the stored-water concept are offset by the advantage of flexibility for the urine recovery. This selection could be reversed if even more emphasis is placed on low initial cost. However, an increase in logistics costs would further substantiate the selection for urine water recovery.

4.6.4.2.3 Process Heat Trade

Process heat is required in the Modular Space Station EC/LS Subsystem for CO_2 control, urine water recovery, and stored-water pasteurization. This heat can be provided by electrical resistance heaters, solar collection or radio-isotope sources. Electrical power and solar collection are sunside only operations and so continuous EC/LS Subsystem operation requires some form of energy storage such as batteries or heat sinks. The continuous versus sunside only operation trade is an assembly level trade and will be discussed in subsection 4.6.4.3 of this report. Sunside operation is assumed here because it traded favorably for the 33-ft-diameter Space Station. Continuous operation is considered for the isotope heat source concept.

Solar Cell Electrical Heat

Solar cell electrical concept is a simple, efficient source for EC/LS process heat. Electrical resistance heaters are provided in the equipment requiring heat and they are actuated by electrical controls. Efficient use is obtained because electrical power is used only as required. No energy loss occurs due to line and heat-exchanger losses. It may be necessary, however, to schedule large power demands carefully to avoid exceeding power system capability.

Isotope Heat Source

This concept consists of a centrally located isotope heat source with a circulating fluid loop to transport heat from the source to the EC/LS equipment. This approach was selected over individual isotope sources in each

EC/LS assembly because the central isotope source can more easily be designed for secondary emergency heat dissipation. Additionally, a fluid loop provides a convenient means of modulating heat addition to assemblies requiring nonconstant heat addition. An example is molecular sieve beds in the CO_2 control system which must be alternately cooled and heated.

The isotope heat source is constructed of 50-watt size plutonium sources. This size was chosen because of inherent safety for small sizes and low cost for fabrication and qualification. The central isotope heat source is designed to provide a highly reliable, secondary-heat-rejection path for safety.

Use of isotopes for EC/LS process heat is a simple and convenient approach. The major disadvantage involves safety and handling problems associated with plutonium isotopes. Although high safety can be demonstrated for the isotope packaging design, a certain amount of hazard remains due to the extreme toxicity of the isotope material.

Solar Collection

The solar collection concept consists of a surface located outside the vehicle which absorbs sun energy in the form of heat. Fluid is circulated through tubes attached to the solar collector surface to pick up the heat energy and transport it to the EC/LS Subsystem. The solar collector surface is coated for a high absorptivity (α) for solar energy and a low emissivity (ϵ) for infrared radiation. This produces a surface with a high equilibrium temperature. Large quantities of heat can be withdrawn at temperatures required by the EC/LS Subsystem.

High α/ϵ ratios are desirable because they reduce solar collector sizes, however, high equilibrium temperatures can result if fluid flow in the panel is terminated inadvertently. Excessive equilibrium temperatures can cause structural damage due to high temperature effects and high pressures from boiling fluids. An α/ϵ value of 2.35 is assumed here for an oriented solar collector resulting in an equilibrium temperature of 221°C (430°F), a reasonable design value.

An important consideration in solar collector selection is selecting a fluid which is compatible with the temperature extremes encountered between sun side and shade side. Ideally, the same fluid should be used in the solar collector and the Space Station interior. Water is ideal for use in the cabin due to its nontoxicity and superior heat-transfer characteristics. Unfortunately, water is not ideal for solar collector fluid because of its relatively high freezing point and low boiling point. The fluid can potentially freeze during shade side of the orbit and high pressures can build up if fluid flow is terminated on sun side.

Both oriented and nonoriented solar collectors were considered. The oriented collector yields more heat for a given size because the surface can be oriented perpendicular to the sun rays, thereby intercepting the maximum amount of solar energy. The major disadvantages of the oriented solar collector are due to the need for an orientation mechanism and articulating joint. These disadvantages are reduced somewhat on the design considered here by locating the collector on the sun oriented solar array structure. The solar array is restricted to ±235-degree movement in two planes which enables flexible lines to be used for articulating joints.

Water is tentatively selected as a heat transport fluid. The freezing problem is avoided by providing an interchange heat exchanger which transfers waste heat from the thermal control loop to the collector loop during shade side operation. As an additional precaution a tube design is considered which can tolerate expanding fluid due to water freezing. Redundant sets of tubes are provided on the solar collector panel. Each tube set has a separate pump so that the probability of flow interruption in the panel is highly remote. Other alternative solutions to the freezing problem are to use a low temperature fluid or heat pipes. An alternative solution to the boiling problem is to allow the water to vent overboard or into a reservoir.

An alternate to the oriented solar collector is the fixed solar collector. A number of separate collectors are located around the periphery of the vehicle and fluid flow is directed to the collector facing the sun. The fixed solar collector eliminates the orientation mechanism and articulating joints.

However, larger areas are required because the collector is generally not oriented directly facing the sun and performance is degraded.

Two types of fixed solar collectors were considered: (1) flat plate, and (2) solar concentrator. The flat plate locates a number of flat plates around the vehicle. The basic design of each plate is similar to the oriented solar collector. The solar concentrator design locates reflectors around the module in a manner similar to the flat-plate design. Each concentrator is trough shaped, running lengthwise along the Space Station. The cross-sectional shape is circular and concentrates solar energy on a tube bundle located in the center of the reflector.

The fixed solar collectors require substantial areas to provide sufficient heat for the EC/LS Subsystem. Available radiator area is thereby reduced below the required amounts for conventional heat-rejection techniques. Additionally, the solar concentrator has a high equilibrium temperature, 310 to 515°C (590-960°F) for α/ϵ range of 2.35 to 7.7. This high equilibrium temperature represents a difficult design problem. For these reasons, the fixed solar collector is not considered competitive for use in the Modular Space Station.

<u>Results</u>

Figure 4.6-22 shows the cost trade results for the three candidate heat sources. Results show that for ISS solar cell electrical costs about five million dollars more initially and is never cost effective. The 12-man Growth Space Station shows that solar cell electrical costs about 10 million dollars more than isotope sources or solar collection. Cost differential is small between isotope sources and solar collection. Only direct costs are shown, however, and additional costs would be incurred for isotope safety and handling provisions.

Although cost savings for isotope or solar heat collection are moderate, consideration of program flexibility and growth potential provides additional advantages. Using solar electrical heat EC/LS process heat requirements would increase the area of each solar array by about 139 m^2 (1,500 ft^2). Addition of this amount of area would complicate solar array design which is

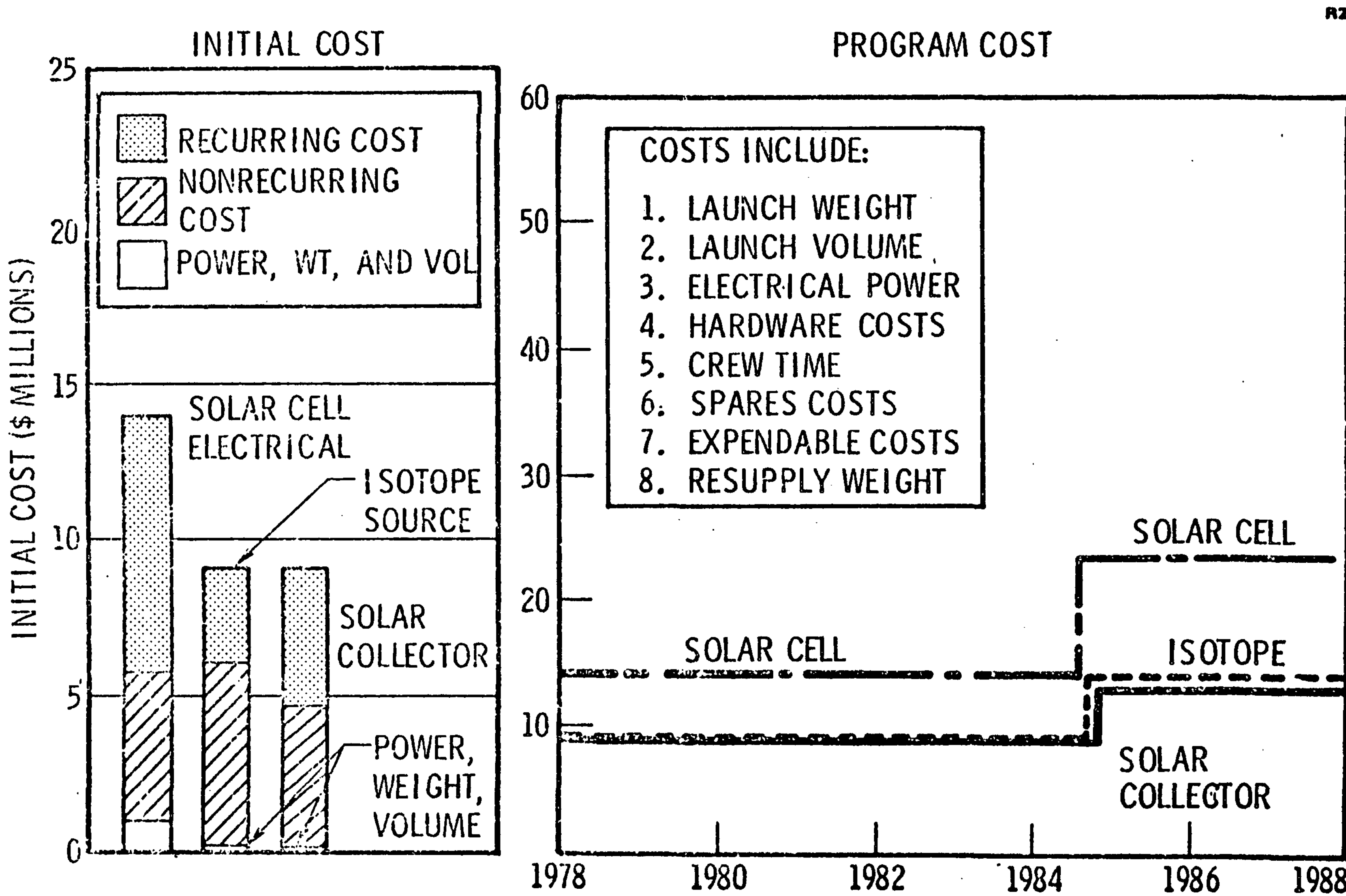

Figure 4.6–22. EC/LS Process Heat Trade

already near the upper limit for manageable area. Growth requiring large solar cell power would not be easily accommodated because of the already large array sizes. Growth in EC/LS process heat requirements can be more easily accommodated by the isotope or solar collector designs. Based on the above considerations, solar cell electrical is not preferred from cost or qualitative consideration.

Since cost differential is small between isotope and solar collection designs, selection is based largely on qualitative considerations. Isotopes represent less mechanical complexity, however, they are subject to safety criticism. Although high safety can be demonstrated with isotope sources, they still require unique design and operational considerations due to the extreme hazardous nature of plutonium material. On the other hand, the mechanical complexity of solar collectors can be largely eliminated in design. Major collector problems involve the articulating joint and freezing/boiling considerations. Use of flexible lines and ±235-degree movement eliminates the articulating joint deficiency with little impact to the Space Station design. Freezing/boiling problems can be eliminated or reduced by using heat pipes or by selecting a more compatible heat transport fluid.

The mechanical complexities of solar collection appear not to be basic to the concept and appear to have design solutions. On the other hand, the hazardousness of acceptable isotope sources is inherent and will remain a subject of concern. Therefore, solar collection is selected to provide EC/LS process heat for the Modular Space Station.

4.6.4.2.4 Compartment Pumpdown Trade

The Modular Space Station contains airlocks, experiment modules, and equipment compartments which must be depressurized and later repressurized as a part of normal operating procedure. Atmosphere contained in these volumes can be vented directly overboard and then repressurized from onboard stores or the atmosphere can be reclaimed by pumping to an accumulator. Pumping to the cabin is not considered for large airlocks because the resultant pressure rise is intolerable to the experiment program. However, the two small scientific airlocks, about 1 m^3 (35 ft^3) each, could be pumped to the cabin without exceeding the cabin pressure control range. Table 4.6-10 lists detailed requirements and constraints for airlock pressurization. Two alternates are provided for EVA: (1) use of the isolation test facility, and (2) use of the Logistics Module airlock. Experiment modules attached to the Space Station contain compartments requiring pump-down and this requirement is considered as a parameter in the tradeoff.

Pumpdown time is based on the largest identified volume to be pumped and on the peak power which can be tolerated by the Electrical Power Subsystem. Based on these considerations, a pumpdown time of 24 hr is selected, which results in a peak pump power of 2,060 watts. A weight tradeoff of pump power and gas loss weight for varying final pressures shows that a low pressure is optimum. A final pressure of 3.44 kn/m^2 (0.5 psia) is chosen since this level is obtainable with a reasonable number of pump stages and also recovers about 97 percent of the atmosphere. The accumulator volume of 1.19 cu m (42 ft^3) represents an airlock to accumulator ratio of approximately 20. Ratios greater than this result in excessive pumping power.

Table 4.6-10

REQUIREMENTS AND CONSTRAINTS FOR AIRLOCK PRESSURIZATION

Airlock Volumes	
Isolation Test Facility, m^3 (ft^3)	23.2 (820)
Integral experiment scientific airlocks, m^3 (ft^3)	(35)
Attached experiment modules, m^3 (ft^3)	33 to 115 (1163 to 4078)
Pumpdown time (largest volume)	24 hours
Final airlock pressure	3.44 (kn/m^2) (0.5 psia)
Accumulator volume	
Space Station m^3 (ft^3)	1.19 (42)
Experiment Module, m^3 (ft^3)	Up to 4.52 (160)

Expendable Gas Concept

The airlock is depressurized by venting the gas to space. Repressurization is from onboard high-pressure gas stores which are periodically replenished by the logistics craft. Theoretically, any atmosphere pressure could be considered from 24 kn/m^2 (3.5 psia) O_2 to 101 kn/m^2 (14.7 psia) air. However, airlock ingress by crewmen would require an intermediate airlock and O_2 prebreathing for lower-pressure levels. Because of these requirements for additional airlocks and crew operational constraints, only 101 kn/m^2 (14.7 psia) air is considered for airlock pressurization.

With the expendable gas concept, each airlock or module volume contains an on/off valve which vents the gas to space. An orifice is provided to control maximum vent rate and a shutoff valve in the overboard line is used to isolate the upstream line for component replacement or emergency shutoff.

Repressurization is accomplished through separate O_2 and N_2 pressurization lines which function in a similar manner. Gas from multiple storage tanks passes through tank isolation valves and a heater to warm up the cool expanding gas. This prevents damage to downstream components due to very low temperatures and reduces condensation or frosting on supply lines in occupied cabin areas. The supply line to the pressurization control shutoff valves is maintained at a pressure of 686 ± 6.9 kn/m^2 (100 ± 10 psia) so the pressurization can be controlled at a constant rate with a decreasing storage tank pressure. Orifices are provided to limit maximum gas inflow rate. Instrumentation is installed at strategic locations to monitor status and operation of critical components.

Airlock Pumpdown Concepts

Airlock atmosphere is reclaimed by the pumpdown concept by pumping the gas to a separate accumulator. Repressurization is accomplished by venting the accumulator back to the airlock.

Two accumulators are used to limit the size for handling and ease of packaging. Three stages of compression are used with a heli-rotor positive displacement first stage followed by two reciprocating second stages. An intercooler is used to reduce pump power. A final pumpdown pressure of 3.44 kn/m^2 (0.5 psia) is reasonable with this arrangement. A solenoid valve just downstream of the compressors prevents gas backblow when the compressors are inoperative.

The two accumulators will accommodate pumpdown of the isolation test facility, however, they will only partially suffice for pumpdown of candidate attached RAMS. The RAMS must be equipped with accumulators to handle the remaining pumpdown volume requirements.

Repressurization is by a separate manifold with pressure regulation for uniform pressurization rate control. Orifices in the pressurization lines limit the maximum pressurization rate.

Evaluation of Candidates

Due to the variability of the frequency of integral airlock and experiment module depressurizations, the quantitative criteria were analyzed parametrically with volume per 90 days as the variable. The study indicated that the volumetric rates would vary from 69.5 to 2720 m^3 (2,460 to 96,400 ft^3) per 90 days.

Selection of Preferred Approach

The selection criteria were applied to the candidate techniques as described in subsection 4.6.4.1. Both concepts satisfy the absolute criteria of performance, safety, and development status. The expendable approach has performance advantage because with this concept the airlock can be cycled more rapidly.

Cost comparisons of the candidates are shown in Figure 4.6-23. Venting costs about $6 million less initially than the pumpdown concept. However, total mission costs for the venting are $36 million more expensive than the pumpdown and the crossover occurs in the second year. Dominant cost is the atmosphere resupply which is $48 million and peaks at approximately 2270 kg/month (5,000 lb/month) for Shuttle resupply. This large resupply requirement represents a major portion of the Logistics Module capacity.

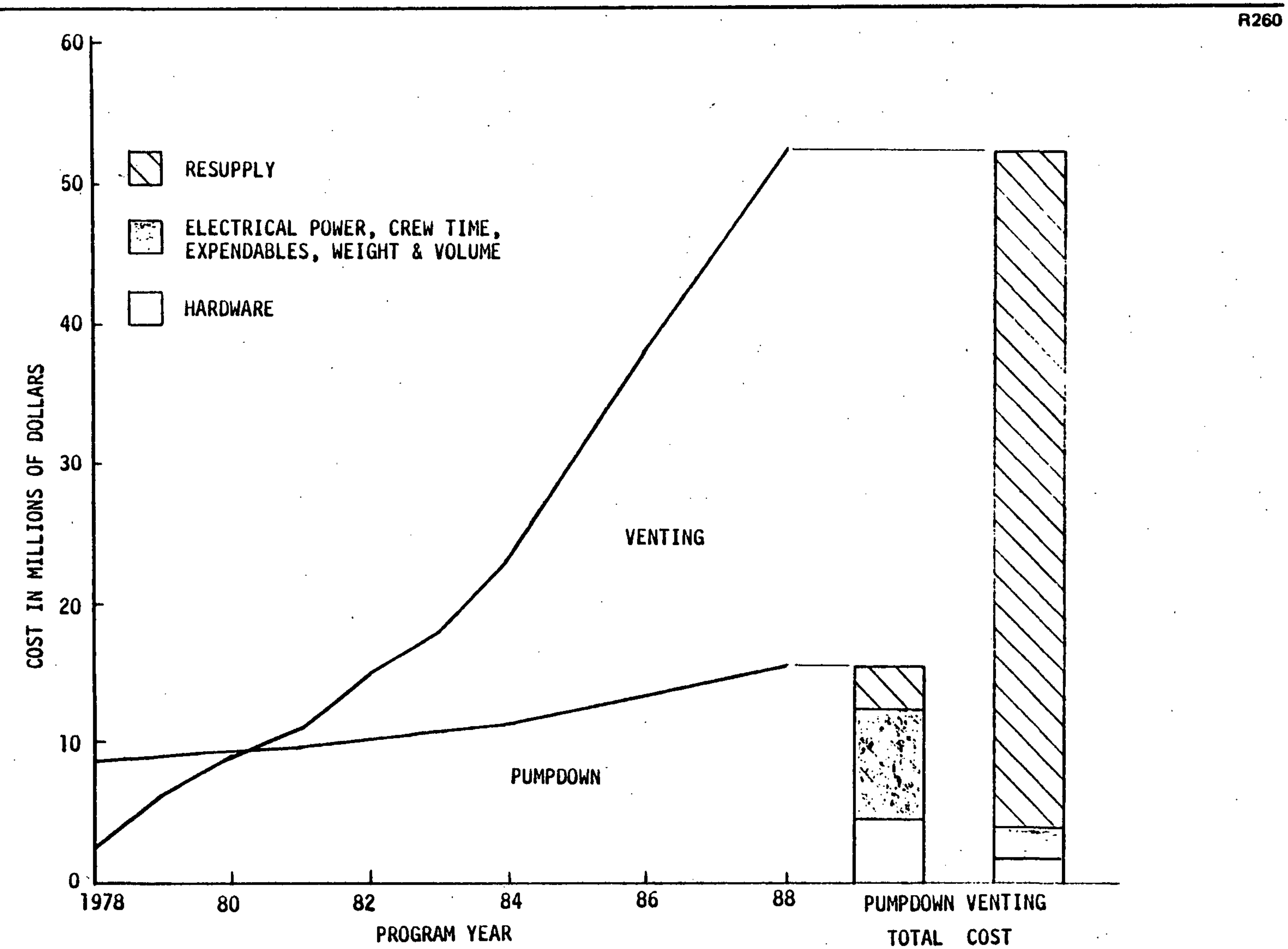

Figure 4.6–23. Cost Comparison of Airlock Pumpdown Versus Venting

The pumpdown concept is more flexible from a qualitative standpoint because increases in logistics costs and airlock volume has little effect on subsystem design. Interface requirements with the logistics mission element are minimal with pumpdown because resupply requirements are small. On the other hand, pumping interacts in a major way with the Electrical Power System due to the relatively large peak power of about 2,000 watts.

Performance margin can more easily be designed into the pumpdown system because more airlock operations can be had with only a small increase in expendables. The opposite is true with the expandable gas concept where total system penalty is nearly proportional to number of airlock operations. Therefore, an increase in airlock use would increase expendables 30 times more for the expendable concept than for the pumpdown concept.

Based on the lower relative cost, and largely on the advantage in flexibility, the pumpdown concept is selected for Space Station airlock pressurization.

A further consideration might be to launch with the venting concept and retrofit the pumpdown concept at a later date. For example, retrofitting in 1981 would allow for the low-cost initial savings of $6 million but would drive the total cost to $23 million for the combination system. This is $10 million more than the total pumpdown concept, and is therefore not recommended.

4.6.4.2.5 Water Recovery by CO_2 Hydrogenation

This subsection presents a trade study which compares two ways of recovering water to offset a portion of the water resupply requirements; (1) urine water recovery by air evaporation, and (2) water recovery from CO_2 hydrogenation by Sabatier reaction. The trade study was performed for the Modular Space Station consisting of a 6-man crew for the first five years, which then grows to 12 men for the remainder of the mission.

CO_2 Hydrogenation

CO_2 hydrogenation is a relatively simple process used in a partial O_2 recovery system which uses a sabatier reactor to produce water from CO_2. In the oxygen recovery system the water is electrolyzed to produce O_2 for crew men and H_2 for use in the Sabatier reactor. This electrolysis reaction is complex and consumes high power. Use of the relatively simple Sabatier reactor to produce water without the electrolysis appeared to be an attractive alternate to the urine water recovery for generating water for crew use. In this way the rather complex urine water recovery system could be eliminated. The concept uses the CO_2 which is driven from the molecular sieve beds of the CO_2 removal assembly by pumpdown. It is then fed with stored H_2 into the CO_2 conversion (Sabatier) assembly where the constituents are reacted, producing H_2O and CH_4. The CH_4 is rejected overboard or collected as biowaste for use in the propulsion subsystem. The thermal energy required for the beds can be provided either by electrical heat or by solar collector heat.

The H_2 required in the Sabatier reaction may be resupplied in the form of super-critical hydrogen by sets of tanks ferried in the Logistics Modules. After docking, the hydrogen tanks will remain in the Logistics Module and H_2 gas transferred by attachable lines. Tanks will be provided for each Logistics Module and will be reusable for the duration of the mission. This method will result in minimum cost for the hydrogen assembly.

The Sabatier reactor uses a catalytic bed operating at approximately 315°C, (600°F). CO_2 and H_2 are proportionately mixed and fed to the catalytic bed where the CO_2 is hydrogenated to form water vapor and methane (CH_4). The constituents are then passed through a condensor-separator where the H_2O is removed and pumped into storage. The CH_4 gas stream is then directed either to the propulsion system for resistojet use or dumped overboard.

For each of the assemblies discussed two 6-man units must be launched at the mission start and an additional unit launched later to accommodate crew buildup to 12 men. Resupply also increases as the crew size increases.

Since the Sabatier reaction output water is insufficient for total crew needs, makeup water must be resupplied.

The supplementary storage of potable water is highly reliable due to hardware simplicity. However, large quantities of water and tanks must be resupplied. Prior to use, each water tank must be checked for water purity. It is assumed that water storage at pasteurization temperatures will not be necessary if the resupply water is low in nutrients. Therefore, electrical energy is small and required only for valves, controls and pumps.

The initial launch provides 30 days onboard and 90 days in the Logistics Module of potable water and tankage for the 6-man crew. Redundant water supplies are not provided because of its inherent reliability and ability to store the supply in two or more separate locations. Additional tankage is launched when the station builds to the GSS level to accommodate the crew buildup. The resupply rate also increases proportionately.

Urine H_2O Recovery—Air Evaporation (Closed)

The urine water recovery assembly recovers potable water from urine, urine flush water, and reverse osmosis residuum by means of a closed-cycle, air-evaporation process.

The assembly is designed to process the daily rate in 18 hours so that the unit has the capacity to catch up in the event a recycle or maintenance is necessary. However, it is normally operated 24 hours per day at a reduced flow rate to avoid unnecessary shutdowns and startups. A description of the air evaporation process is presented in subsection 4.6.4.2.2 and will not be repeated here.

It is assumed for trade purposes that 2 to 6-man units are launched initially along with two water storage assemblies. Additional 6-man units are added with the modules for growth to a 12-man crew.

Solar collector heat is assumed to provide the process heat for the urine water recovery and for pasteurization of the stored water.

Trade Results

Both concepts satisfy the absolute criteria of performance, safety and development status. The urine water recovery system is considered safer because CO_2 hydrogenation involves large stored amounts of H_2.

Figure 4.6-24 depicts the quantitative results which show about the same initial cost for both concepts but a lower total program cost by 21 million dollars for the urine water recovery concept.

A dominant cost for the CO_2 hydrogenation concept is for water resupply to make up for the lower H_2O production rate of the concept. A cost for solar energy is charged to the H_2 hydrogenation concept in an amount which is required to store the CO_2 for use rather than dumping it overboard.

If a hydrogen storage system is included for the propulsion subsystem the reduction in the nonrecurring costs of hydrogen tankage development for Sabatier is 300 thousand dollars. This cost saving is not sufficient to change the trade results appreciably.

The CO_2 hydrogenation concept is less complex than the urine water recovery but lacks flexibility and growth potential due to its high resupply demands. Longer resupply periods or increased launch costs both favor urine recovery because very little resupply is needed. Increased crew sizes can be more conveniently handled by the urine water recovery concept because it has a much higher capacity than is normally used.

If both units are operated at the ISS level, a 16-man capacity is available and at the GSS a 24-man capacity is potentially available.

Because of the total program cost savings and the slight qualitative advantages, the urine water recovery concept is selected.

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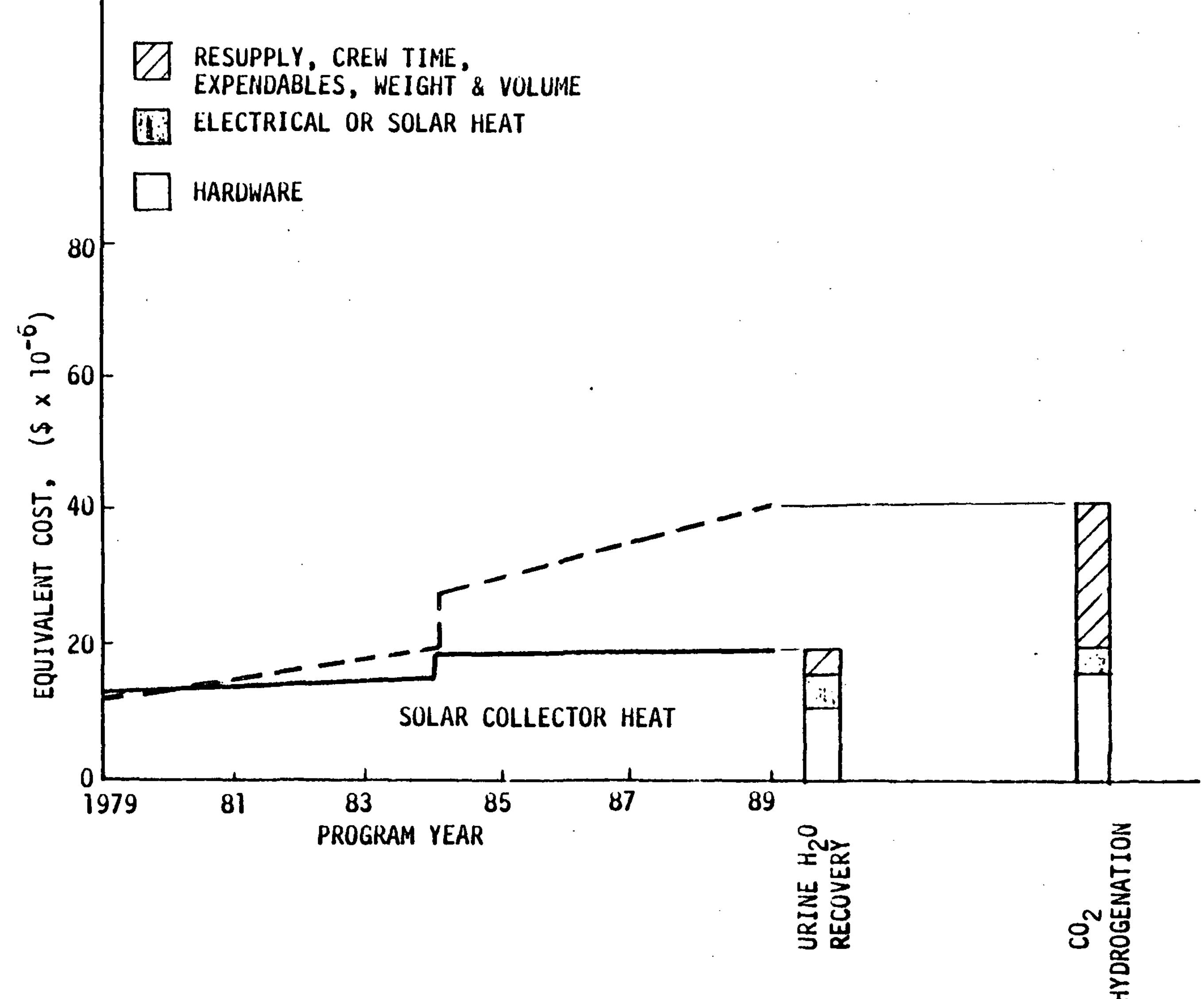

Figure 4.6–24. Cost Comparison of Urine Water Recovery and Water Recovery by CO_2 Hydrogenation

4.6.4.2.6 EC/LS Subsystem Modularity Trade

A tradeoff was performed on the early Modular Space Station baseline to determine the optimum size of EC/LS Subsystem. Not all EC/LS assemblies are involved in the trade; only the assemblies are included which interface directly with the crew and are amenable to modularity based on crew size. These include the following:

- Humidity control.
- CO_2 removal.
- Water recovery.
- Waste management.

Trade results are shown in Table 4.6-11 which gives the weight, power, expendables, and relative recurring cost for each modularity option. Non-recurring cost is expected to differ little between options.

Each modularity option complies with the following ground rules:

A. Sufficient capacity is provided to handle the normal crew plus a 6-man overlap crew.

B. Each option has at least one backup provided.

Based on these ground rules, candidate options are shown for the 6-man ISS level and the 12-man GSS level. The results show it is less costly for both crew levels to provide the larger EC/LS Subsystem size, i.e., 6-man or 12-man. The reason for this result is that weight increases little with EC/LS Subsystem size and larger size EC/LS Subsystems are more efficient from a power standpoint and an expendable usage standpoint. Additionally, the larger sized EC/LS Subsystems involve less recurring costs.

On the basis of these data, the 6-man EC/LS Subsystem is selected for ISS. Since ISS development costs preceed GSS, it is not cost effective to change to a 12-man configuration for GSS; rather, additional 6-man units should be used.

4.6.4.2.7 EC/LS Subsystem Backup Options

The modularity trade study reported in subsection 4.6.4.2.7 considered only one type of backup, provision of one completely redundant EC/LS

Table 4.6-11

EC/LS MODULARITY COMPARISON

Option	Modularity	Weight, kg (lb)	Average Power (watts)	Expendables, kg (lb/30 days)	Relative Recurring Cost
ISS					
6-Man Crew	2 to 6's	2,840 (6,250)	4,965	468 (1,031)	1.0
6-Man Overlap					
1 EC/LS Backup	4 to 3's	3,100 (6,822)	5,303	470 (1,038)	2.0
GSS					
12-Man Crew	2 to 12's	4,610 (10,177)	9,149	900 (1,988)	1.0
6-Man Overlap	3 to 6's	4,730 (10,413)	9,328	913 (2,012)	1.5
1 EC/LS Backup	6 to 3's	5,200 (11,496)	10,100	930 (2,054)	3.0

Subsystem module. In this subsection, an additional method of backup is considered, i.e., open-loop backup.

Figure 4.6-25 shows the result of this trade which gives weight and power for various crew sizes. Data are presented for a single EC/LS Subsystem, single EC/LS Subsystem with 30-day open loop backup, and for a dual EC/LS Subsystem. Theoretically, the single EC/LS Subsystem option is not acceptable from a safety standpoint but is included in the figure for reference only.

Results of the trade show that the dual-loop EC/LS Subsystem is preferred from a weight standpoint but requires 70 to 80 watts more power. The higher power results from the fixed power required for controls and similar equipment for each EC/LS Subsystem. This type of power is twice as large for the dual EC/LS Subsystem as compared to the single EC/LS Subsystem. Because of the large weight advantage for providing dual EC/LS, this approach is favored from a quantitative standpoint. The major disadvantage to the dual EC/LS Subsystem is due to the similarity of equipment. It is advantageous to have a backup system which is of a different design than the primary system. This is because if a design induced failure occurs in one EC/LS Subsystem, the probability is high that it will also occur in the backup system. Also with a dual EC/LS Subsystem they both interface with the same equipment so that a failure at one interface is more likely to affect both units.

Although the results of this study indicate the desirability of dual EC/LS Subsystem, each function was considered separately during the detailed definition of the EC/LS Subsystem and the decision on backup was based on the unique considerations of each function. As a result of this trade, dual EC/LS Subsystems were selected except for water recovery. Only one urine and one wash water recovery unit is selected, located in the Crew/Operations Module, because only one personal hygience facility is provided. Little would be gained by providing water recovery units in a module that did not contain sources and supply points for water. Backup is provided, however, by the 30-day contingency water recovery unit located in the GPL.

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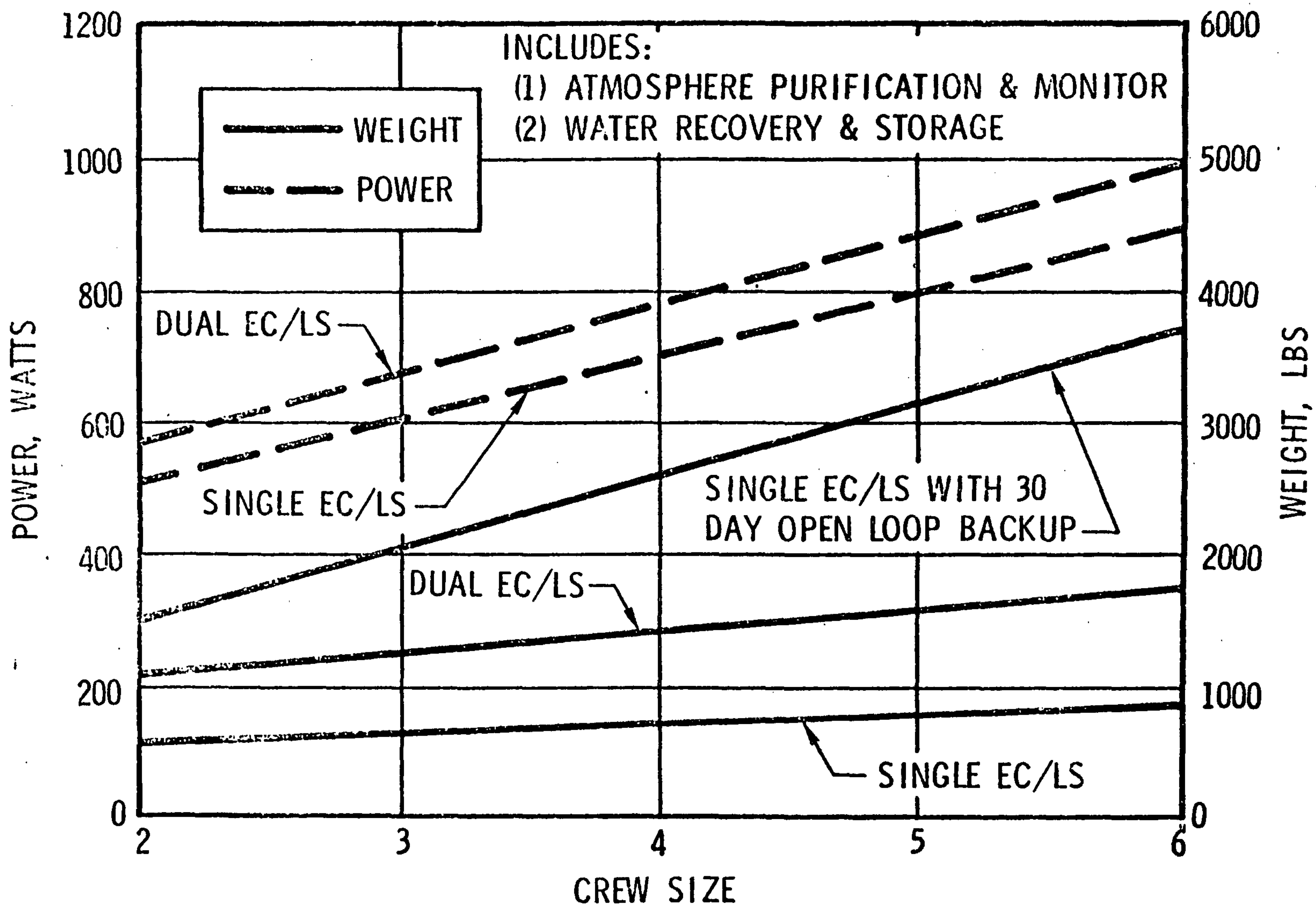

Figure 4.6–25. EC/LS Options

4.6.4.3 Assembly Level Trades

4.6.4.3.1 Atmosphere Storage Method

Oxygen is required for crew metabolic makeup, module repressurization, and makeup of overboard losses. Nitrogen gas is required for module repressurization and for makeup of overboard losses. Crew metabolic 0_2 makeup is needed continuously while all other gas needs are periodic.

Crew metabolic requirements for 0_2 are 0.87kg/man-day (1.92 lb/man-day). Overboard losses are not predictable but are expected to be small and no separate stores are assigned for this purpose. The module repressurization gas requirement is the largest pressurized compartment which is taken as the sum of the Power/Subsystem and the Crew/Operations Modules. This represents the largest habitable volume in the Space Station. Some makeup air will also be needed for airlock makeup but this amount is small for a pumpdown system which scavenges the bulk of the air before evacuation, therefore, the effects of airlock makeup are small and can be neglected for the purposes of this trade.

Storage concepts evaluated include the following:

- Cryogenic
- High-pressure gaseous
- Chemical

<u>Cryogenic Atmosphere Storage</u>

Cryogenic storage concept is acceptable only for makeup of losses which occur continuously. This is necessary to prevent boil-off of cryogenic fluid during periods where fluid is not withdrawn. Boil-off is to be avoided to minimize the detrimental effects of effluents on experiments.

Metabolic 0_2 supply is the only makeup requirement which has a relatively continuous withdrawal rate and it is therefore a candidate for cryogenic storage. Based on the guidelines and constraints, a 30-day supply must be provided beyond the normal resupply period. This supply is not normally used and is best located on the Space Station to minimize Logistics Module load. Since it is a standby supply, this 30-day standby supply cannot be stored cryogenically.

The normal metabolic resupply makeup can be conveniently located on the Logistics Module in cryogenic form and supplied to the Space Station by an attached line. This approach is favored over manual transfer of tanks or cryogenic fluid transfer to tanks located on the Space Station. The cryogenic tank insulation is based on the longest resupply period of 90 days. The same tank design is used for resupply periods of shorter duration but the electrical heater power is slightly higher to obtain the required withdrawal rates of 0_2.

Supercritical cryogenic tankage is assumed because it is well developed and within existing state-of-the-art. The slightly better performance obtained with subcritical storage is negated by the added cost and development risk.

Three tank sets are provided at the ISS level; one for each Logistics Module. An additional three tank sets are provided at the GSS level for the three additional Logistics Modules.

High-Pressure Gaseous Storage

Gaseous O_2 and N_2 stored at 20.6 x 10^3 kn/m^2 (3, 000 psia) is acceptable for all gas makeup requirements because this concept can be used for standby and continuous usage. Shelf life is nearly unlimited and therefore high pressure gas storage is particularly adaptable for contingency supplies.

High-pressure gas is potentially hazardous due to the high pressure but acceptable levels of safety and reliability can be demonstrated. The concept is the best developed of all storage concepts.

A major advantage of the gaseous concept is that it is an acceptable concept for all atmosphere storage needs. As a result, nonrecurring cost is held at a minimum. This low initial cost is offset somewhat by the relatively high penalty for tank weight.

The 90-day metabolic resupply requirement is satisfied by locating tanks on the Logistics Module. The tanks remain on the Logistics Module and the 0_2 is supplied to the Space Station as required by an attached line. This approach precludes the requirement to transfer tanks manually or transfer gas to tanks located on the Space Station. Six tank sets are provided for the three logistics modules at the ISS level. This tankage requirement is doubled at the GSS level.

Chemical Atmosphere Storage

Chemical atmosphere storage is not as well developed as the gaseous and cryogenic concepts and as such represents a higher development risk. Additionally, chemical concepts represent safety hazards due to toxic nature of materials involved.

Chlorate candles are the best developed concept for chemical 0_2 supply. This concept has been used in submarines, and aircraft for emergency oxygen and as normal 0_2 supply in an advanced Space Portable Life Support System. High safety is obtainable but the 0_2 generation rate cannot be closely controlled. Therefore, the approach is unacceptable for normal metabolic 0_2 supply. Since the storage penalty for chlorate candles is comparable to high-pressure gaseous storage, no advantage is seen for their use on Space Station.

Other chemical sources of oxygen such as peroxides, superoxides, and hydrogen peroxide have control problems or involve hazardous materials. Therefore, they do not now appear applicable to Space Station.

A number of chemical forms of nitrogen have been proposed which store N_2 in the form of CH_4 or N_2O_4/N_2H_4. Some development has occurred with CH_4 disassociation but there has been little effort on N_2O_4/N_2H_4 systems. These candidate chemical forms are hazardous to the crew and as such are not acceptable for use in Space Station. This decision is compatible with the Space Station guideline which states, "All materials selected for use in habitability areas will be nontoxic, nonflammable, and nonexplosive to the maximum extent practical."

Based on the reasons stated above, chemical storage concepts are rejected for Space Station use.

Cryogenic Versus Gaseous Storage

Gaseous O_2 and N_2 supply are the only concepts which are acceptable for supply needs which meet standby-type requirements. This includes gas for repressurization, makeup of overboard losses and 30-day contingency metabolic makeup.

A cost trade was performed for metabolic O_2 makeup and includes gaseous and cryogenic storage concepts. Figure 4.6-26 shows the results of this trade which includes only the normal makeup supply.

The trade was performed initially without cost discounting and no clear cut advantage was obvious with either concept. The trade was then redone with cost discounting and the results are shown in Figure 4.6-26. Gaseous storage is not assessed with development costs since tank design for the contingency supply can be used. The results show that gaseous storage costs about two million dollars less initially but about three million dollars more for the total 10-year program. The cross-over point where cryogenic storage is the least costly occurs after about four program years.

Cryogenic storage is more complex than gaseous; however, its lower weight allows an increase in Logistics Module payload. The smaller weight penalty for cryogenics increases the logistics module capacity by about 1370kg (3,000 pounds) per year for a 6-man crew. Program flexibility slightly favors gaseous because of standby capability. This would be important during times of reduced crew numbers or shutdown modes.

Based on a lower initial cost and simplicity, the gaseous O_2 concept is selected. However, if in the final design phase the Logistics Module capacity is found to be inadequate, a conversion to cryogenic storage is recommended. If this occurs after initial use of gaseous storage, the Logistics Modules can be conveniently retrofitted for cryogenic storage. The cost of this approach is shown in Figure 4.6-26 which shows the conversion in 1984. This

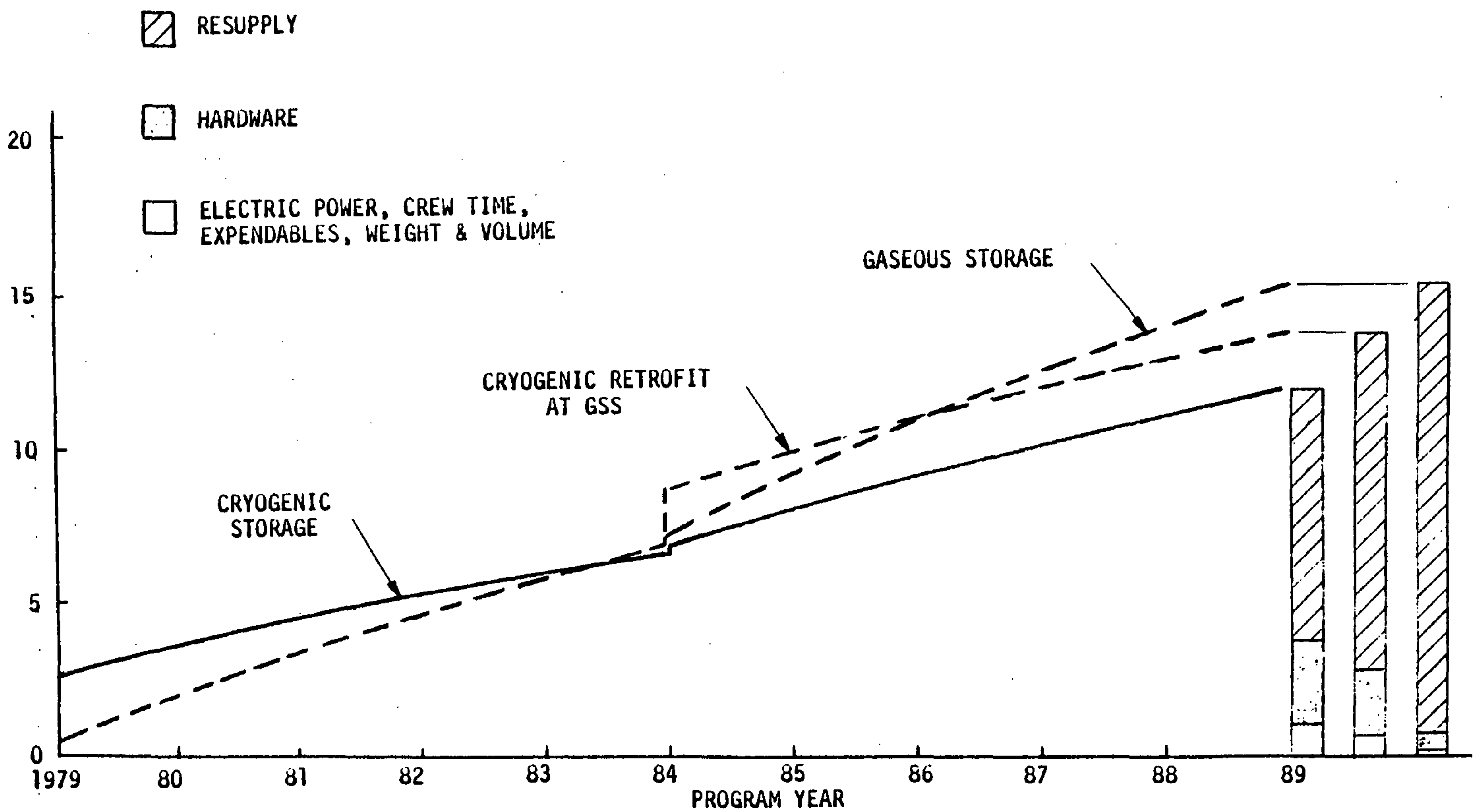

Figure 4.6–26. Cyrogenic Versus Gaseous O_2 Trade

retrofit would result in a discounted cost which is still about 1-1/2 million dollars under that for gaseous storage for the entire 10-year program.

4.6.4.3.2 Regenerable versus Nonregenerable Charcoal

Two concepts for trace contaminant and odor control were evaluated; catalytic oxidation with nonregenerable charcoal and catalytic oxidation with regenerable charcoal. Most low molecular weight contaminants can be catalytically oxidized to produce water vapor and gases which can be conveniently removed by other EC/LS process; CO_2 can be removed by the CO_2 removal unit. Catalytic oxidation is the only strong candidate for this function. However, many other heavier molecular weight contaminants must be removed by absorption on activated charcoal. The trade reported here compares the use of expendable charcoal, which is the selected concept, versus a concept which regenerates the charcoal for reuse. This regeneration process involves heating the charcoal bed and exposing it to space vacuum.

Since continuous contaminant removal is desirable, the regenerable charcoal method uses two beds. While one bed is online removing contaminants, the second bed is being regenerated.

Regenerable charcoal concept is more complex because it requires air diverter valves, switches, timers, and heaters. On the other hand, the expendable charcoal concept requires more resupply to replace the expended charcoal.

Concept Evaluation

Both trace gas control candidates are believed capable of performing the control function adequately. Neither concept presents a significant safety hazard. Stored charcoal might provide fuel for a fire or a place for bacteria growth, but with proper precautions neither concept presents a safety hazard.

Although the expendable charcoal concept is better developed, both concepts could be ready for the Modular Space Station Program.

Figure 4.6-27 shows the relative cost of the two candidates. Included are costs for launch weight, launch volume, power consumption, crew time, hardware costs, and resupply costs; initial costs are $400K less for the nonregenerable concept. For the total mission the regenerable concept saves $700K. The crossover point occurs at the initiation of GSS level.

From a qualitative standpoint, the nonregenerable charcoal concept is less sensitive to interfaces because it has fewer electrical and vacuum connections. Regenerable charcoal has more growth potential and flexibility because it does not depend on a regular logistics resupply.

The expendable charcoal concept is selected because it is better developed and has a lower initial cost. These factors more than offset the advantages of regenerable charcoal which are: lower total program cost and less reliance on the logistics system. This selection could be reversed if resupply costs increased significantly.

4.6.4.3.3 CO_2 Removal Method

The following tradeoff study compares two concepts, molecular sieve and carbonation cell, for the CO_2 removal and concentration function on the Modular Space Station. The purpose of the concentrator is to remove carbon dioxide, controlling its level in the cabin air, and deliver it to the resistojet subsystem for use in vehicle attitude control.

Two important techniques, the hydrogen depolarized cell and steam desorbed amine concepts, were considered but not evaluated in detail. The hydrogen depolarized concept was not evaluated since the oxygen recovery was not selected at the subsystem-level trade studies. Hydrogen depolarized requires hydrogen for operation which results in hydrogen carryover with the concentrated CO_2, and is therefore best used in a closed O_2 cycle life support system using a hydrogenation process such as the Sabatier or Bosch. The steam desorbed amine concept delivers a high latent load to the cabin air. Since this concept did not trade favorably for the 33-foot-diameter-station, the concept was not considered a candidate for this study.

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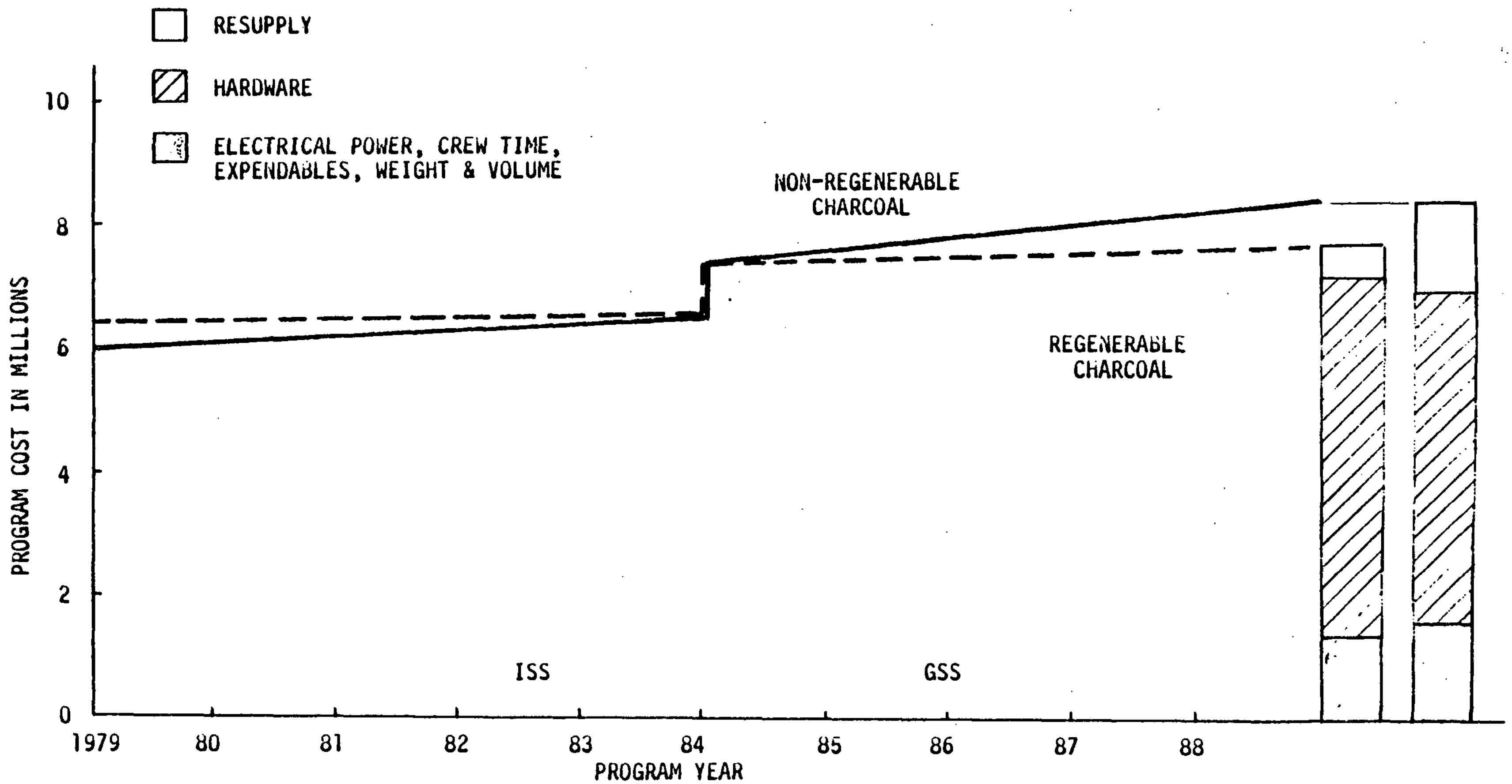

Figure 4.6–27. Cost Comparison of Regenerable Versus Nonregenerable Charcoal

Requirements

Specifically, the concentrator must maintain a maximum CO_2 partial pressure of three millimeters mercury in the cabin by removing an average of 1.04kg (2.3 pounds) of CO_2 per man-day in a 6- or 12-man vehicle. In addition, it must deliver concentrated CO_2 at (or above) the cabin total pressure of 101 kn/m^2 (14.7 psia). Although no purity requirement has been established, a penalty is applied for loss of oxygen or nitrogen with the CO_2. An emergency CO_2 partial pressure limit of 2 kn/m^2 (15 mm Hg) applies to a period of two hours, corresponding to a period of peak metabolic activity. The concentrator is required to operate with heat transfer fluids (if needed) at temperatures of 132° C (270° F) and 19.5° C (67° F).

A number of general requirements are also important. During the first half of the 10-year mission, the 6-man crew may be present at either of two isolatable locations within the vehicle, or may be divided between them. One 6-man CO_2 concentrator is used at each location. Operation alternates from one unit to the other, so that the one in the location with the higher CO_2 concentration is always running. This approach provides its own redundancy, because either unit can support the entire crew if the other unit fails. During the second half of the mission, the crew is expanded to 12 men, and a third 6-man concentrator is added to provide the required capacity and redundancy. Other requirements include sufficient spare equipment to meet reliability goals for a period of 120 days throughout the mission.

Molecular Sieve Concentrator

A molecular sieve CO_2 concentrator may be designed to operate in one of a number of modes. A preliminary study resulted in the choice of a thermal swing unit operated continuously around the orbit. To achieve this type of operation (with electrical power supplied by solar cells and thermal power supplied by hot fluid), both a solar collector and a thermal storage unit are needed. These units are not considered part of the concentrator, although an appropriate penalty is assessed. Pressurized liquid water appears to be the most suitable thermal storage medium.

The molecular sieve concentrator is illustrated schematically in Figure 4.6-28. It is basically a four-bed cyclic adsorption system consisting

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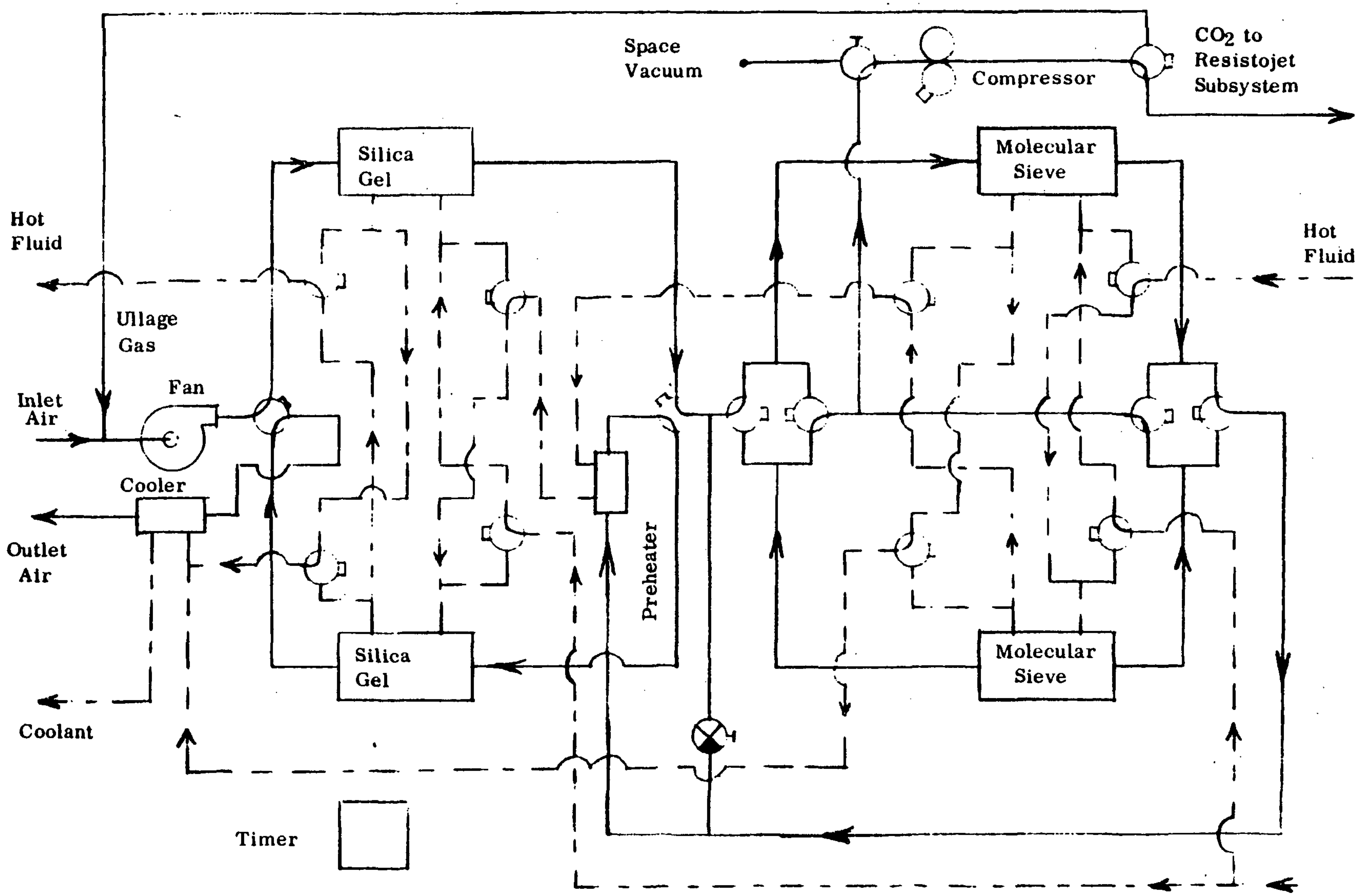

Figure 4.6–28. Molecular Sieve

of two silica gel desiccant beds and two molecular sieve CO_2 removal beds. The desiccant prevents moisture poisoning of the molecular sieve material. Each desiccant and molecular sieve bed is fitted with an integral heat exchanger for cooling the beds during the adsorption phase of the cycle and for heating the beds during the desorption phase. Coolant is provided at 19.5°C (67°F), and hot fluid is provided by the solar collector-heat storage subsystem at 132°C (270°F). A vacuum pump-compressor removes adsorbed CO_2 from the desorbing molecular sieve bed and delivers it to the resistojet subsystem. In addition, the concentrator includes a process flow fan; valves to direct the flow of gas, coolant, and heating fluid to the appropriate beds; and a timer to actuate all valves in the proper sequence.

In normal operation, the process flow fan first directs the air (from the Humidity Control Subsystem) through the adsorbing silica gel bed to remove nearly all water vapor from the air. The dry air then continues through the adsorbing molecular-sieve bed, which removes much of the CO_2. Next, the air stream is heated before exiting through the desorbing silica-gel bed, where it picks up the water removed from the air during the previous half-cycle. Finally, the air is cooled before returning to the cabin atmosphere. Meanwhile, the vacuum pump-compressor first discharges ullage air from the desorbing molecular sieve bed to the fan inlet, and it then delivers pure CO_2 to the resistojet subsystem.

An early version of the type of molecular sieve concentrator described here was successfully run as part of the NASA Langley Integrated Life Support System, with a desorption temperature of 191°C (375°F). Preliminary design of a 6-man molecular sieve concentrator similar to the one described here was completed in 1971 as a part of the NASA/MSC Space Station Prototype program. Thus, molecular sieve technology is at a stage where detailed design of a final flight prototype can be started immediately. The pacing item is the vacuum pump-compressor. Performance of a prototype unit was entirely satisfactory, but sealing and bearings problems require further work.

Carbonation Cell Concentrator

The carbonation cell concept is a two-stage electrochemical process. The first stage removes CO_2 and oxygen from the cabin air, while the second stage removes oxygen from the CO_2-oxygen mixture, yielding pure CO_2. Accessory equipment includes a process flow fan, a humidifier-dehumidifier, blowers for air cooling the electrochemical cell stacks, valves, and controls. The carbonation cell concentrator is illustrated schematically in Figure 4. 6-29.

In normal operation, the process flow fan first directs cabin air through the humidifier-dehumidifier. Here, water vapor is transferred through a membrane from the effluent air to the incoming air. This prevents dehydration of the electrochemical cells. The humidified air then passes through the first-stage cell stack before returning through the humidifier-dehumidifier to the cabin atmosphere. In the first-stage cells, CO_2 and oxygen are absorbed from the air and transferred across an alkaline matrix to the anode. At the anode, the CO_2 and oxygen evolve as a gaseous mixture, which is directed to the second stage. As the gas mixture passes through the electrochemical cells of the second stage, nearly all of the oxygen is transferred across an acidic matrix, leaving nearly pure CO_2. The CO_2 is delivered to the resistojet subsystem, and the oxygen is returned to the cabin atmosphere. Each of the electrochemical stages, which operate at 60°C (140°F), is cooled by cabin air blown over the unit. Available information does not indicate the need for a makeup water supply. However, water vapor losses are likely (despite the humidifier-dehumidifier), and addition of a makeup water supply would complicate the system significantly.

The carbonation cell system is in the laboratory prototype phase. Single-cell testing includes parametric testing of both stages. In addition, endurance testing amounts to 2, 400 hours on the first stage and 6, 700 hours on the second stage, including 1, 300 hours of coupled operation. A four-cell first-stage module was fabricated and tested, and a 15-cell first-stage module was fabricated but apparently tested only in the hydrogen depolarized mode. Information on specific development problems is not available.

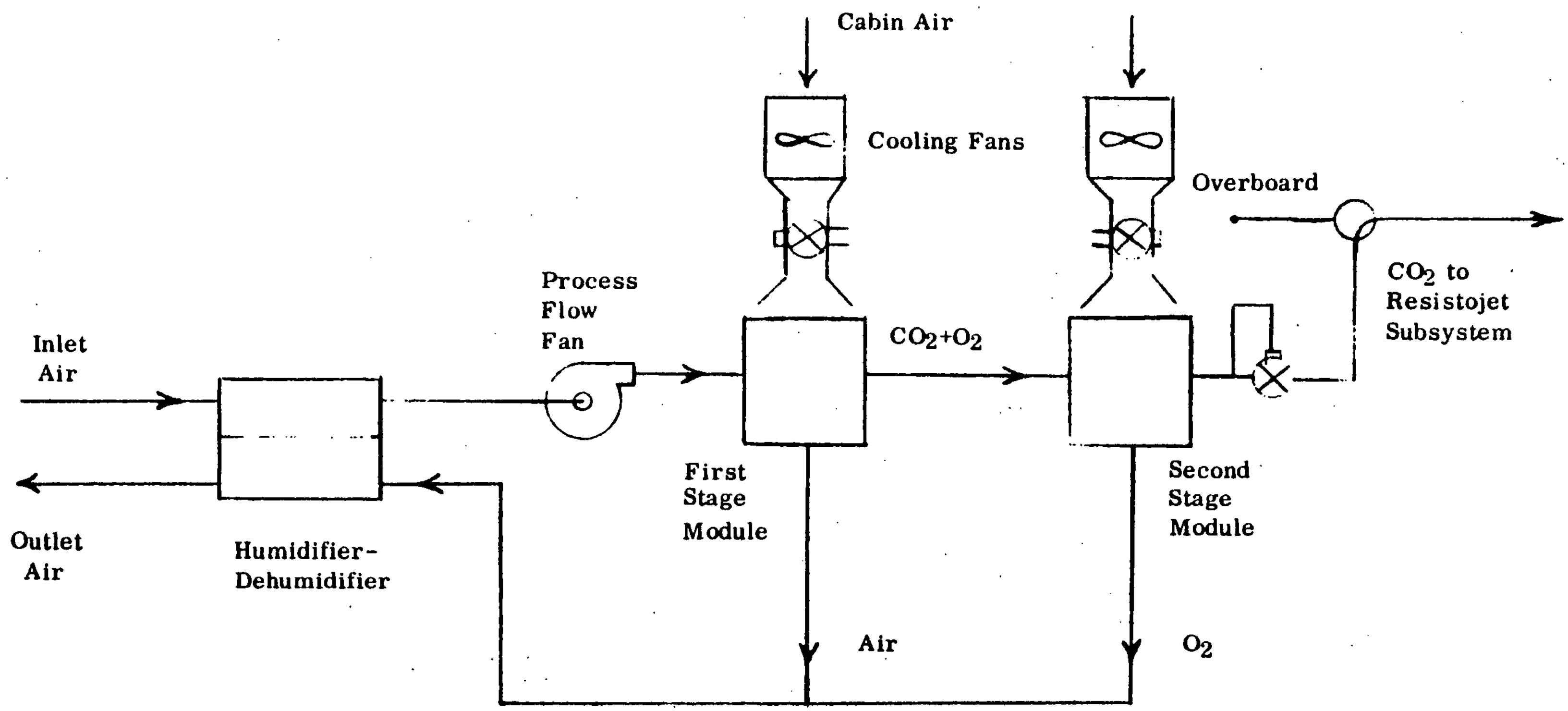

Figure 4.6–29. Carbonation Cell

Concept Evaluation

Both concepts are capable of meeting performance, however, the carbonation cell has more potential for controlling to low CO_2 levels. Due to the presence of high corrosive electrolyte in the carbonation cell, the molecular-sieve concept is considered safer. Additionally, molecular sieve is far better developed than the carbonation cell.

Figure 4.6-30 depicts the quantitative results of the trade. The molecular sieve has a slight advantage of about 4 million dollars initially and about 1-1/2 million dollars for the total program. Based on these results, neither concept has a clear-cut advantage from a cost standpoint.

The major determining factor in the trade is development risk which clearly favors the molecular-sieve concept and for this reason it is selected. The major factor which could reverse this decision at a later date would be highly encouraging performance from carbonation cell development testing or if the allowable CO_2 level was lowered much below the current $0.4 kn/m^2$ (3 mm Hg).

4.6.4.3.4 Urine Water Recovery Method

The purpose of this trade is to evaluate two water reclamation concepts for use on the Space Station to recover available water from urine, urine flush, and reverse osmosis residuum diverted from the wash water recovery assembly. The concepts considered are vapor compression and air evaporation.

Urine, which represents an important portion of the input to the reclamation system, has two characteristics that have special impact on its reclamation treatment. Although it is essentially sterile when voided by a healthy person, it is an excellent growth medium for bacteria, and rapid bacterial contamination of the storage facilities is to be expected. Also, it contains a significant quantity of relatively volatile nitrogen compounds that break down under bacterial action and a temperature of approximately 120°F, releasing gaseous ammonia. Because both the bacteria and ammonia pose a contamination hazard, it has been found advisable to pretreat the urine to

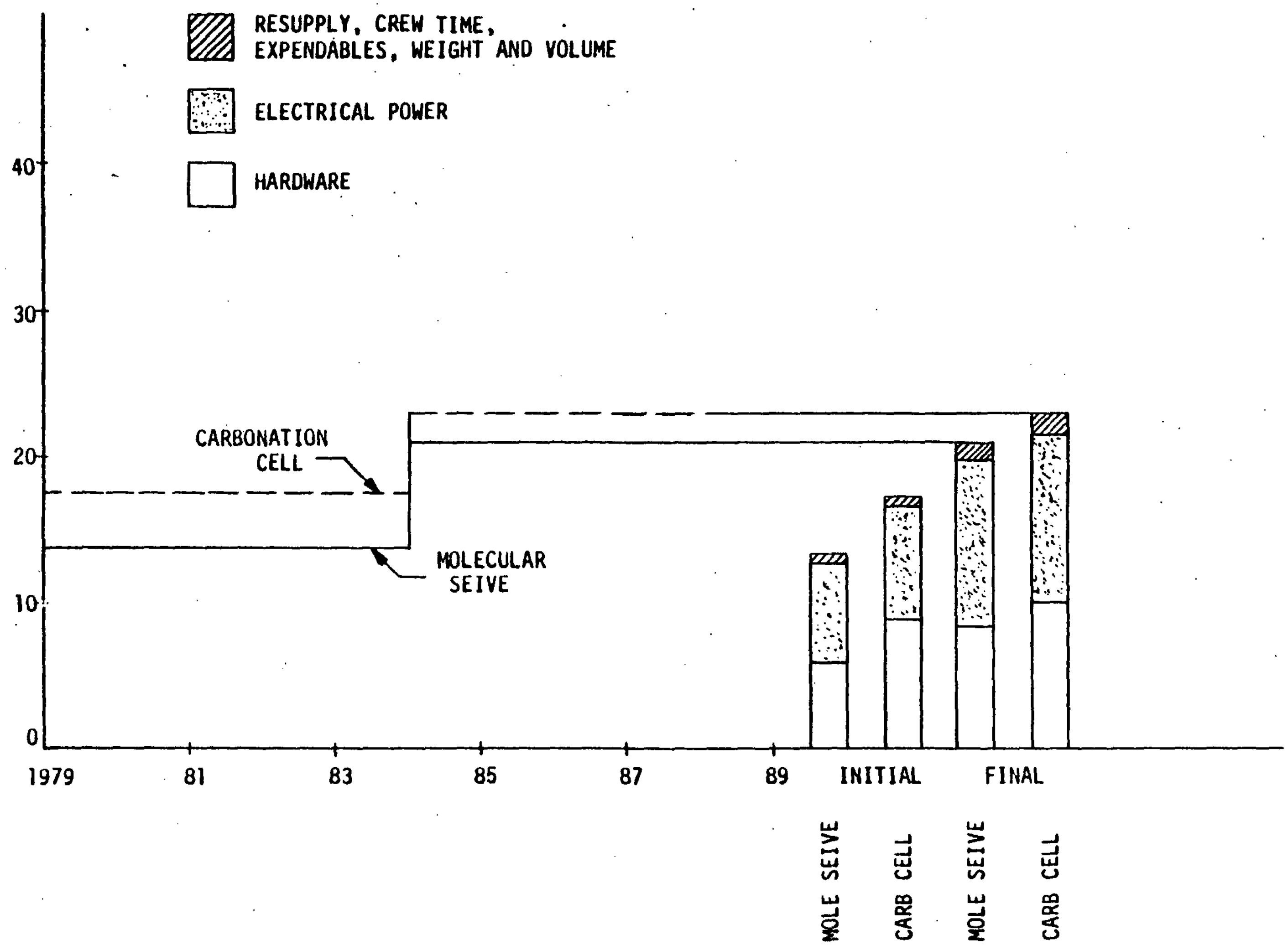

Figure 4.6–30. Carbon Dioxide Control

inhibit bacterial growth and to chemically fix the volatile ammonia. The vapor compression and other evaporation concepts considered in this study for urine processing use a mixture of chromium trioxide and sulfuric acid in the urine flush water as pretreatment for the urine before processing. Flash evaporation is not considered because it is basically similar to two other more advanced concepts, vacuum distillation, and vapor compression. Thermoelectric distillation is not considered further because of its aborted development, and its basic similarity to vapor compression. Because of the high penalty for thermal energy used in this study, vapor diffusion was not evaluated in detail. It does offer significant advantages in positive bacteria control and permits waste handling in liquid form. Because of these advantages, it warrants periodic reevaluation for inclusion in the system.

Following is a list of requirements for the urine-water reclamation assembly.

Urine-water rate	1.78 kg (3.915 lb)/man-day
Flush-water rate	0.18 kg (0.392 lb)/man-day
R.O. residuum	4.75 kg (10.45 lb)/man-day
Recovery efficiency	99 percent
Condenser coolant temperature	15°C (59°F)
Design process rate	8.95 kg (19.7 lb)/man-day
Number of men	
0 to 5 years	6
5 to 10 years	12
Number of urine-reclamation assemblies	
0 to 5 years	2
5 to 10 years	3
Initial launch spares	120 days
Initial launch expendables	120 days
Resupply period	30 days
Mission length	10 years

Air Evaporation

Air evaporation is a distillation process in which recirculating gas is used to evaporate water from wicks saturated with waste water. The evaporated water is transported by the recirculating gas to a condenser-separator for recovery. This concept can be applied for processing of all types of waste water and is defined schematically in Figure 4. 6-31.

In normal operation, pretreated urine and urine flush water is metered to one of two installed wick modules. The rate of urine to the wick module is proportioned, to the rate of water processed by the system, by a metering device. This eliminates the possibility of flooding or drying-out the wicks.

Gas is circulated by a fan, in a closed loop, through a heat exchanger where it is heated. This gas is then directed through a wick module where water is evaporated from the urine. The moisture-laden air flow then passes through a regenerative heat exchanger where it is precooled prior to entering a condenser/separator. Here water is condensed from the gas stream and separated.

Condensed water from the regenerative heat exchanger and the condenser/separator is pumped through a conductivity sensor which controls a three-way valve. This valve normally directs acceptable water flow through charcoal and bacteria filters to the potable water storage tank. If the water is not acceptable, the conductivity sensor diverts the valve to recycle the water back into the waste water storage system for further treatment. Two wick modules are included to allow replacement of an expended wick module without interrupting the process flow. The wick modules are sized for a 30-day replacement period. When maintenance of the wicks is required, a three-position valve is actuated to bring another wick module on line and to isolate the expended module from the main gas flow. Approximately 10 percent of the recirculated flow is directed to the expended wick package for complete drying before replacement so that bacteria growth is prevented during storage.

Air evaporation was first developed more than 10 years ago. Laboratory prototype units have been delivered to two NASA Centers for testing.

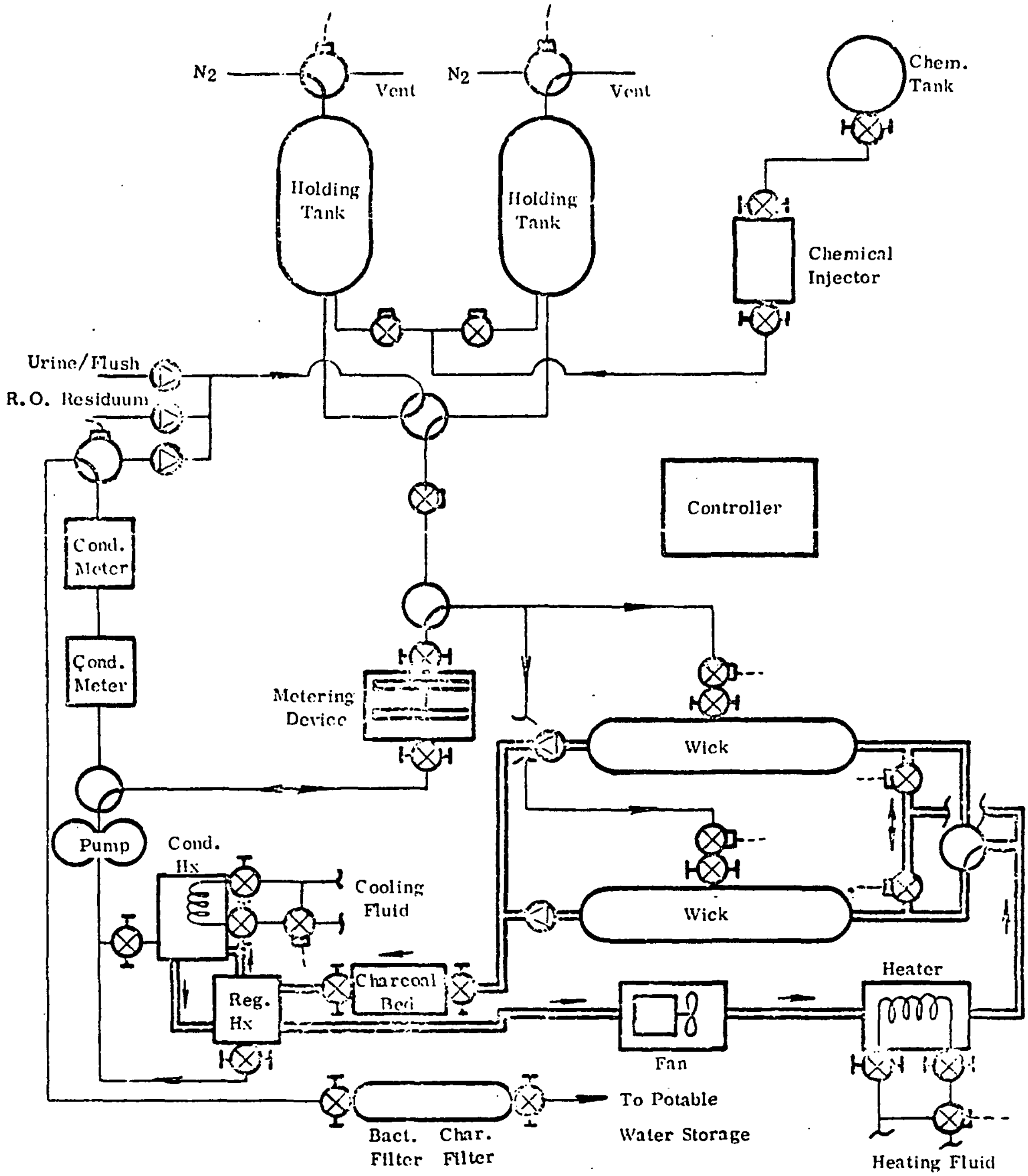

Figure 4.6–31. Air Evaporation

Considerable operating experience exists with this concept. A flight system is considered feasible for Space Station with normal development effort. Areas requiring further development are wick replacement and storage concepts and waste water feed control.

Vapor Compression

Vapor compression is a vacuum distillation process carried out in a rotating still. Its major features are low power and heat rejection obtained by recovering the heat of condensation for use in vaporizing the waste water. This concept, which can process any type of waste water, is defined in Figure 4.6-32.

In normal operation, a pump circulates waste water through a cycle tank, the vapor compression stills, a solutes sensor, and back to the pump. As water distills out of the circulating mixture, the mixture volume decreases. When the cycle tank diaphragm reaches a low limit position, the cycle tank is automatically refilled from the waste water holding tank. Operation continues in this manner until the concentration of solutes (which are retained in the mixture) reaches 50 percent. This is detected by the solutes sensor. The highly concentrated waste water is then ejected from the cycle tank for further processing by the wet waste management subsystem.

As the waste water passes through the rotating vapor compression still, a portion of it evaporates at a pressure of 5.1 kn/m^2 (0.74 psia). The vapor passes through a demister screen to eliminate entrained water droplets, and is then compressed to a pressure of 7.35 kn/m^2 (1.14 psia). This compressed vapor condenses as it passes over the evaporator-condenser heat transfer plate. Because of the higher condensing pressure, the condensing temperature is also higher. Heat, therefore, flows from the condensing surface to the evaporation surface. Thus, the use of the compressor permits complete recovery and reuse of the latent heat. Reduced pressure in the stills is maintained by a vacuum pump. The vented gas passes through a urine separator, and water is returned to the waste water holding tank. Gases pass through a bacteria filter and a sorbent bed to the cabin. The filter prevents bacteria carry-over into the cabin atmosphere, and the sorbent removes ammonia and volatile organics.

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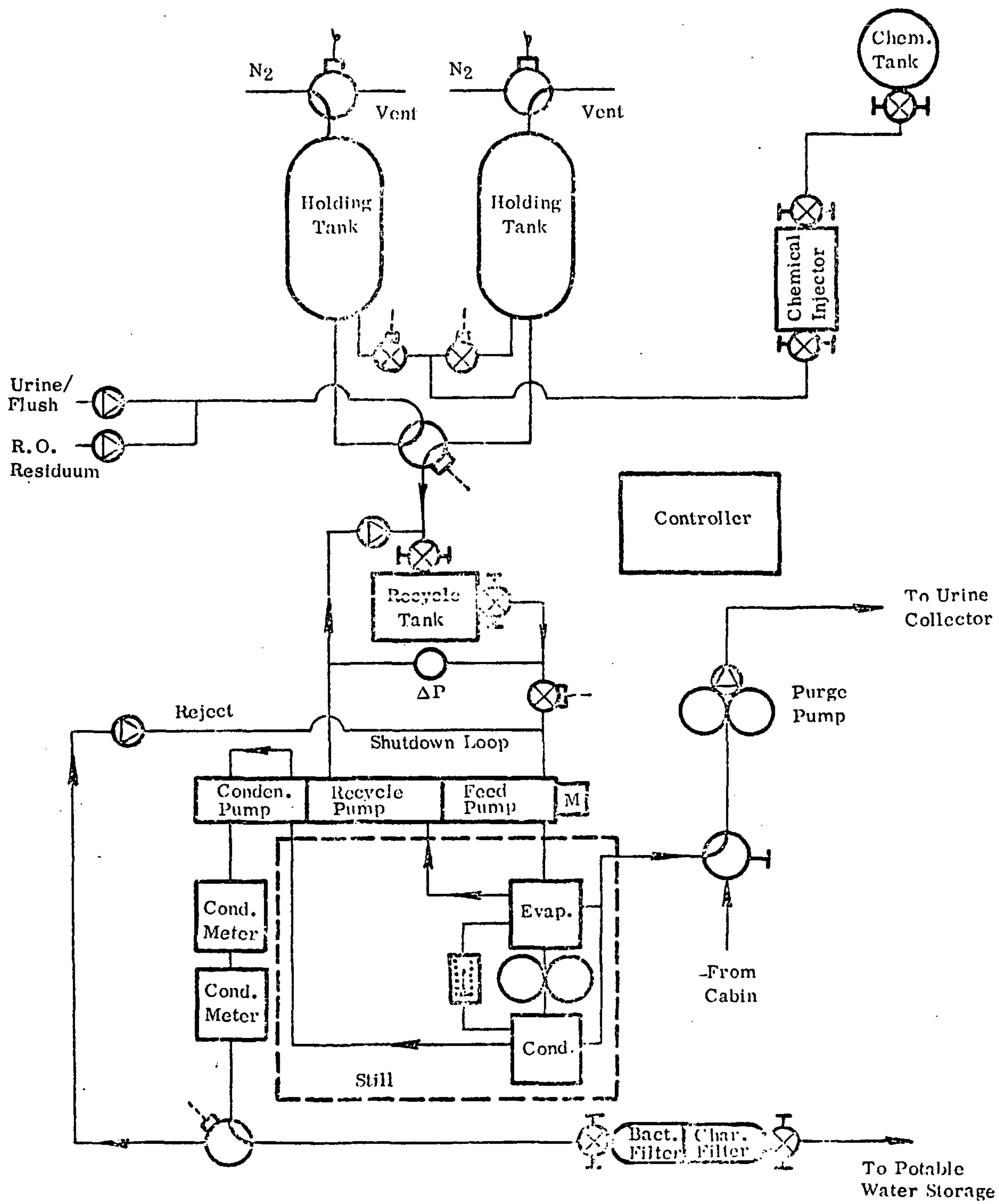

Figure 4.6–32. Vapor Compression

Product water from the rotating stills is pumped through a conductivity sensor, a bacteria filter, and a charcoal filter to provide water storage. The sensor monitors the product water and automatically recycles it to the waste water holding tank if specific conductivity is excessive. The first filter removes bacteria from the product water. The charcoal removes dissolved ammonia and trace organics that are distilled with the product water.

A development prototype of the vapor compression system is presently being tested. This testing will provide data for NASA's Space Station Prototype (SSP) Program. Certain development problems do exist, however, namely a reliable sensor to monitor solids concentration within the process loop and continued effort to control bacteriological contamination in the product water. Total system development is well advanced though, and there is high confidence that a flight unit can be fabricated for Space Station.

Concept Evaluation

Both concepts have shown the ability to product acceptable quality water. One of the main advantages of the air evaporation technique is that it is capable of recovering 99 to 100 percent of the waste water processed. Extensive testing has confirmed that water of excellent potability is produced once the system has been pasteurized.

The vapor compression system has successfully demonstrated the capability to produce water that meets water quality standards. A water recovery efficiency of 95 percent is considered practical. The stills must be vented to vacuum during normal operation so a vacuum pump which runs continuously is required. The added complication of a bacteria filter and sorbent bed to prevent cabin contamination from the vented still is a drawback of this process.

Figure 4.6-33 presents the cost comparison for the two candidates. This date shows that air evaporation is favored from an initial cost standpoint and from a total cost standpoint by from 2.5 to 3.5 million dollars.

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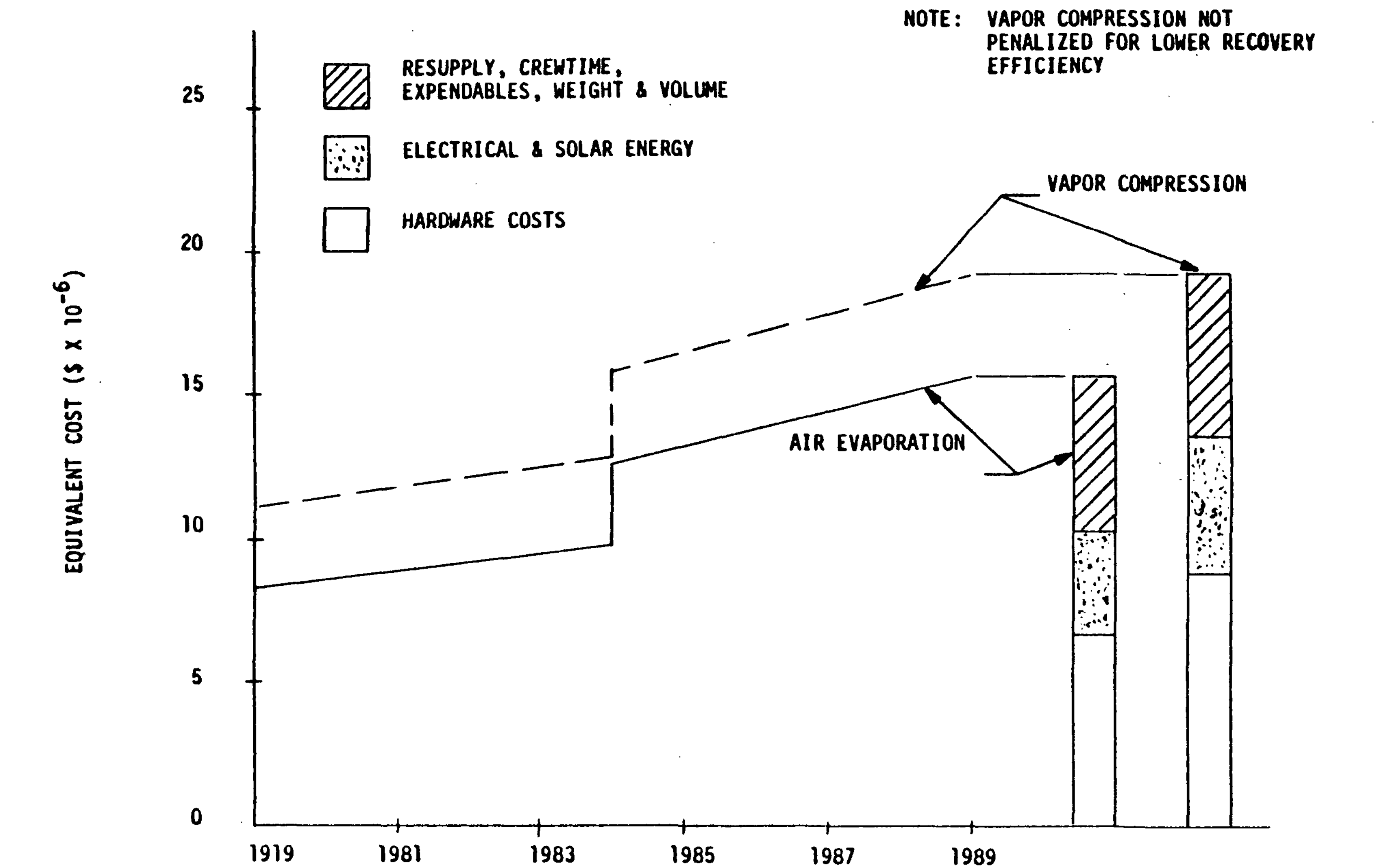

Figure 4.6–33. Cost Comparison for Urine Water Recovery – Millions

No process heat is required for the vapor compression technique which gives it an advantage considering interface sensitivity. However, the air evaporation technique appears to be better developed and somewhat simpler.

Based on the cost advantage, and because it represents less development risk, the air evaporation technique is selected.

Since there is currently a slight excess of water available, the vapor compression concept was not penalized for its lower efficiency. If the excess water was found to be useful for experiments or some other use, the higher recovery efficiency for air evaporation would become important.

4.6.4.3.5 Wash Water Recovery Method

This tradeoff study evaluates two concepts for the reclamation of wash water and humidity condensate; reverse osmosis and multifiltration. Reverse osmosis is shown to be preferred. Phase change processes such as air evaporation, vapor compression, and vapor diffusion were not considered because of the high penalties associated with phase change processes and the relatively low level of contamination of the wash and condensate water does not warrant such treatment. Both of the selected concepts have demonstrated adequate performance and their development status is such that a high level of confidence exists as to their availability for application on Space Station.

Following is a list of the requirements for the wash water and condensate reclamation assembly.

- Wash water rate — 22 kg (48.60 lb)/man-day
- Humidity condensate — 1.26 kg (2.78 lb)/man-day
- Potable water make-up — 3.48 kg (7.67 lb)/man-day
- Maximum use rate — 109 kg (240.00 lb)/hr. for 10 min.
- Design process rate — 31.2 kg (68.82 lb)/man-day
- Reverse osmosis efficiency — 80 percent
- Number of men
    - 0 to 5 years — 6
    - 5 to 10 years — 12

- Number of wash water assemblies
  - 0 to 5 years 2
  - 5 to 10 years 3
- Initial launch spares 120 days
- Initial launch expendables 120 days
- Resupply period 30 days

Reverse Osmosis

The reverse osmosis concept is based on the fact that when a solution of impurities in water is pressurized above its osmotic pressure, water from the solution will permeate through a suitable membrane as pure water. Thus, pure water can be collected with no phase change.

In normal operation, waste water is pumped to a pressure of 2.82×10^3 kn/m^2 (410 psia). At this pressure, the waste water enters one end of the reverse osmosis module. As it flows through the module, most of the water permeates the hollow fibers, and the resulting purified water flows out through the fibers. The waste water which does not permeate the fibers becomes a concentrated brine solution when it reaches the other end of the module.

Figure 4.6-34 schematically depicts the reverse osmosis concept. The holding tanks for waste water are each sized to accommodate one day's wash water and condensate accumulation. The bacteria and charcoal filters are combined as a single unit and sized on the basis of grow-through life for the bacterial filter (10 days). A final bacteria filter is added as a precaution tion against back growth of bacteria.

The heart of the reverse osmosis concept is the membrane. In the past, available membranes produced water of questionable quality and the membrane life was limited. However, recent membrane development programs have greatly improved the performance of the reverse osmosis concept. Additional programs now underway are expected to result in acceptable membranes which could be flight qualified in a reverse osmosis unit for the Space Station launch data.

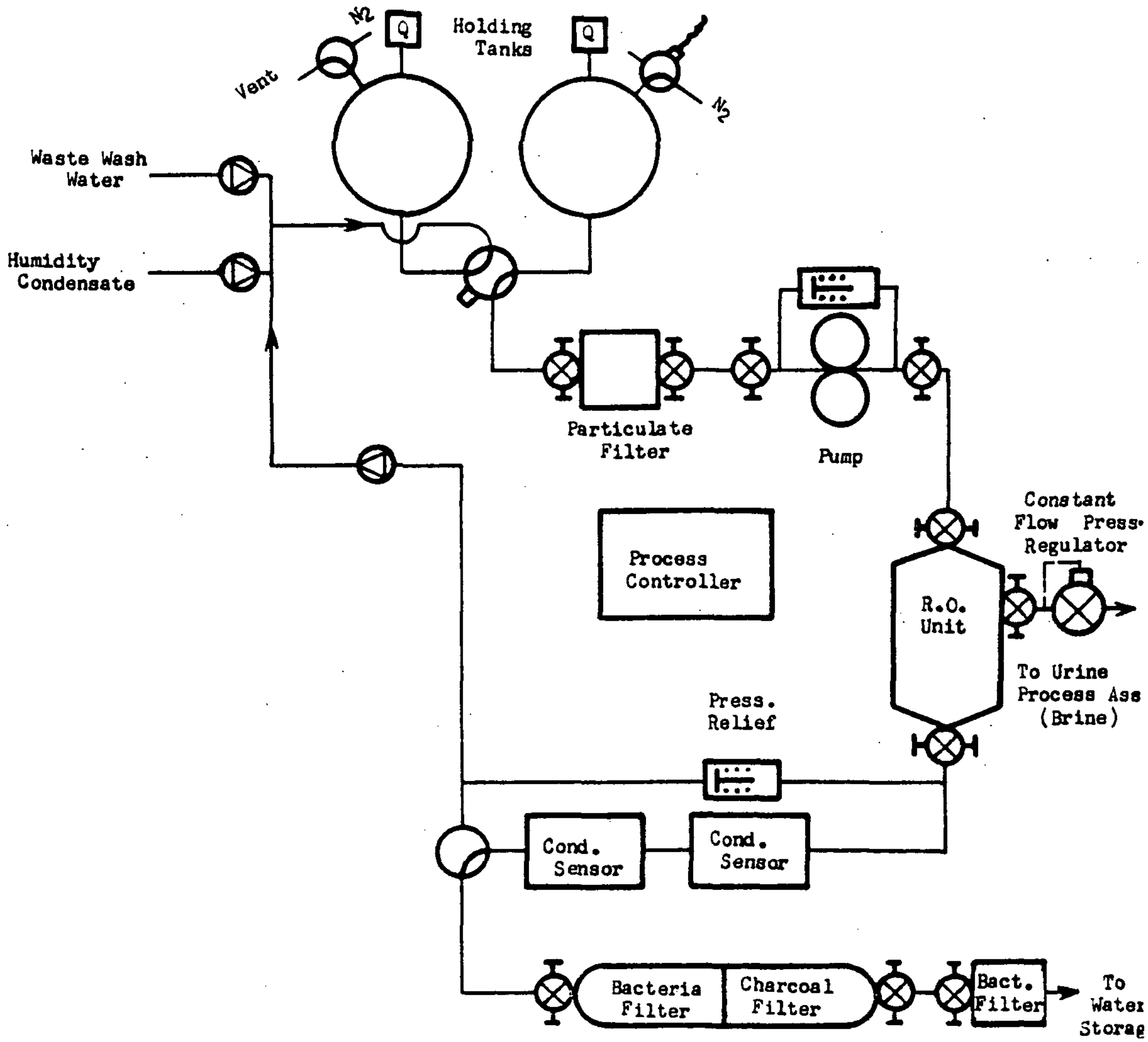

Figure 4.6–34. Reverse Osmosis

The development of this membrane is an item considered pacing technology and thus requires considerable future effort.

Multifiltration

In the multifiltration process, Figure 4.6-35, waste water passes through a series of static components, each of which removes a specific type of contaminant. The components contain expendable materials or replaceable items. The most common processes are mechanical filtration, adsorption, and deionization. This concept is theoretically suited for processing all waste water, but previous studies have shown that including urine in the waste water increases the expendables weight to a noncompetitive value.

In normal operation, waste water is pumped in series through a particulate filter, bacteria filter, charcoal bed, ion exchange bed, another charcoal bed, and out through a final bacteria filter. The particulate filter consists of two or three mechanical filters to remove particulates from the waste water.

The first filter keeps particles, larger than one micron, from clogging the bacteria filter (about 0.3 micron) which follows it. The charcoal bed removes cleaning agents and other organic contaminants, thereby eliminating odors. The deionization bed is in two parts. The upstream section is a mixed bed ion exchange resin, which removes both positive and negative ions to a very low concentration. The downstream section is an anion exchange resin, needed to raise the pH of the product water to a satisfactory value between 6.0 and 8.0. The second charcoal filter adsorbs odors that may be released by the resin during initial operation. A bacteria filter prevents backgrowth of bacteria into the charcoal bed or the ion exchange resins, where they would multiply rapidly.

The holding tanks for waste water are each sized to accommodate one day's wash and condensate accumulation. The bacteria and charcoal filter are combined as a single unit and sizing based on growth-through life for the bacteria filter. For the primary bacteria/charcoal filter, the grow-through life is extended to 30 days by incorporating a small amount of silver chloride

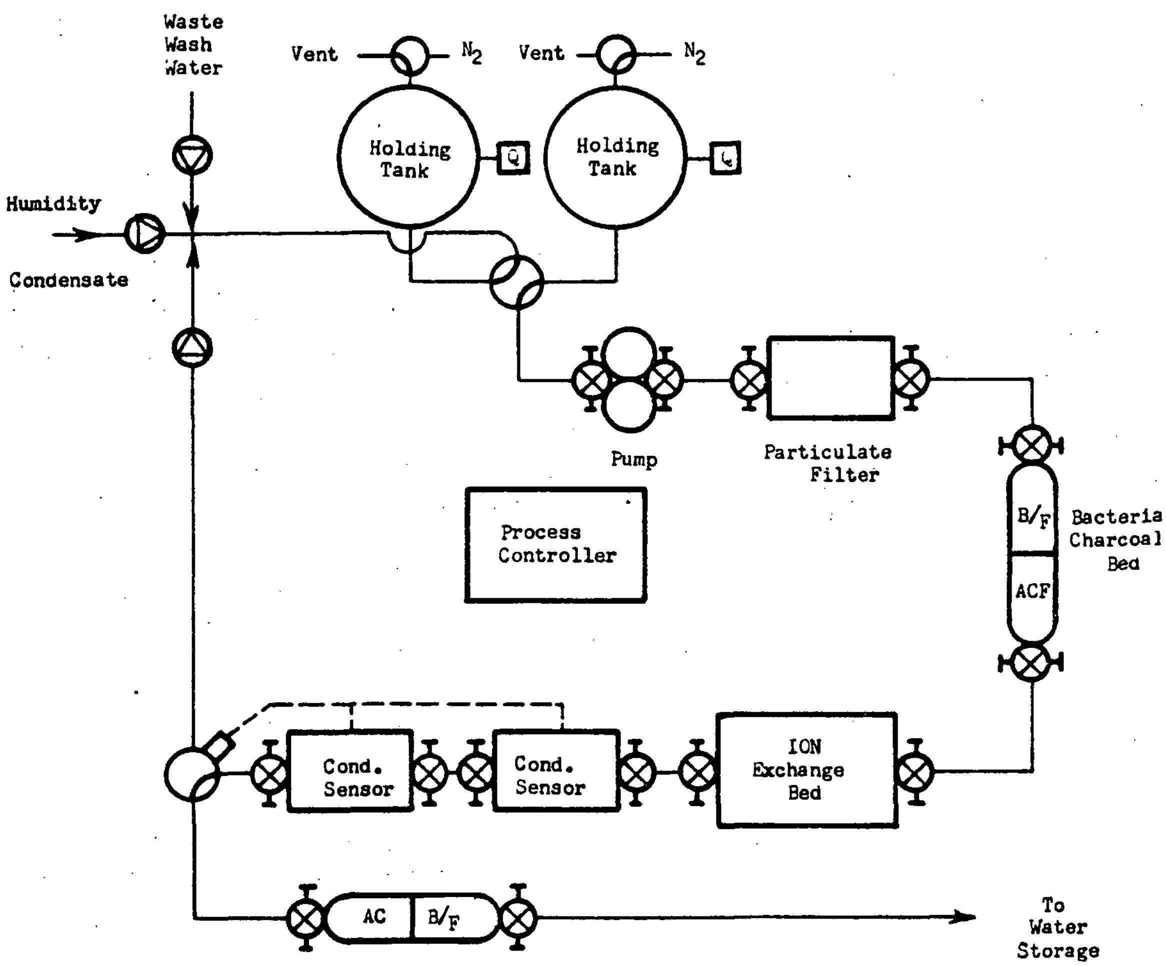

Figure 4.6–35. Multifiltration

beads at the inlet. This technique is applicable only to this filter since ion exchange resins are required downstream to remove any silver ions from the effluent water.

Charcoal bed sizing is based on interpolated data from previous testing. The quantity of charcoal for these tests ranged from 0. 15 kg (0. 33 lb)/day (MDAC 90-day test) to 8. 15 kg (18 lb)/day (N. A. Steel et al) based on the specified processing rate. The quantity used for this application is an optimistic value based on recent Chemtric Corporation data of 1. 36 kg/day (3 lb/day).

Development of multifiltration is well advanced and resultant water quality is simply dependent on quantity of expendables. Components and materials required for this system are available. Additional development work on expendable rates for the particular contamination levels anticipated is required.

Concept Evaluation

Both concepts have shown the ability to produce acceptable quality water. The required preprocessing of 20 percent of the waste-water flow in the urine reclamation assembly to increase recovery efficiency is a penalty for reverse osmosis that reduces its attractiveness. It is anticipated, however, that the preprocessing rate may be significantly reduced with the development of the new membrane materials. This in turn will reduce the overall penalty of the reverse osmosis assembly.

For multifiltration, the use of sufficient expendables and/or beds make the product water as pure as desired. Water lost when changing beds is minimal. Thus, multifiltration recovers essentially all of the water processed.

Reverse osmosis is a high-pressure process, and leakage of waste water could therefore be a potential problem. Operating conditions are conducive to growth of microorganisms, and the membrane is not an absolute barrier to microorganisms. In the multifiltration process, the deionized bed may become contaminated, and bacteria will multiply rapidly within the system.

However, with the system protected by bacteria filters, this does not normally occur.

Figure 4.6-36 shows the quantitative results for the two concepts. It can be seen that there is little cost differential between concepts. It was mentioned above that a large portion of the reverse osmosis cost is for processing the residuum in the urine water recovery system. This amounts to a cost of about 7.2 million dollars which might be greatly reduced for later versions of reverse osmosis units. The assumed recovery efficiency of 80 percent is rather pessimistic and recovery efficiencies of up to 95 percent are predicted with new membrane materials.

Reverse osmosis has the advantage regarding flexibility because it does not depend as heavily on resupply. However, it is a more complex concept and is not as well developed as multifiltration.

Reverse osmosis is selected for reclamation of wash water and condensate. The primary reason for its selection over multifiltration is on the basis of the qualitative comparison of the two concepts. Multifiltration is a good choice except for its high expendables requirement, which reduces its appeal in the areas of maintainability potential, weight, volume, and resupply penalty. Reverse osmosis has an advantage in the important areas of flexibility and growth potential.

4.6.4.4 Assembly Integration Level Trades

4.6.4.4.1 Atmosphere Distribution Ducting Size

A trade was performed to determine the optimum ducting size for the atmosphere distribution system. As ducting size increases, the volume and weight penalty increases but fan power penalty decreases due to a lower pressure drop. A classical tradeoff curve results and this is shown in Figure 4.6-37. It can be seen that minimum cost penalty occurs between 21.8 cm (8-in.) and 27.3 cm (10-in.) diameter. A 21.8 cm (8-in.) duct size is selected for the Modular Space Station.

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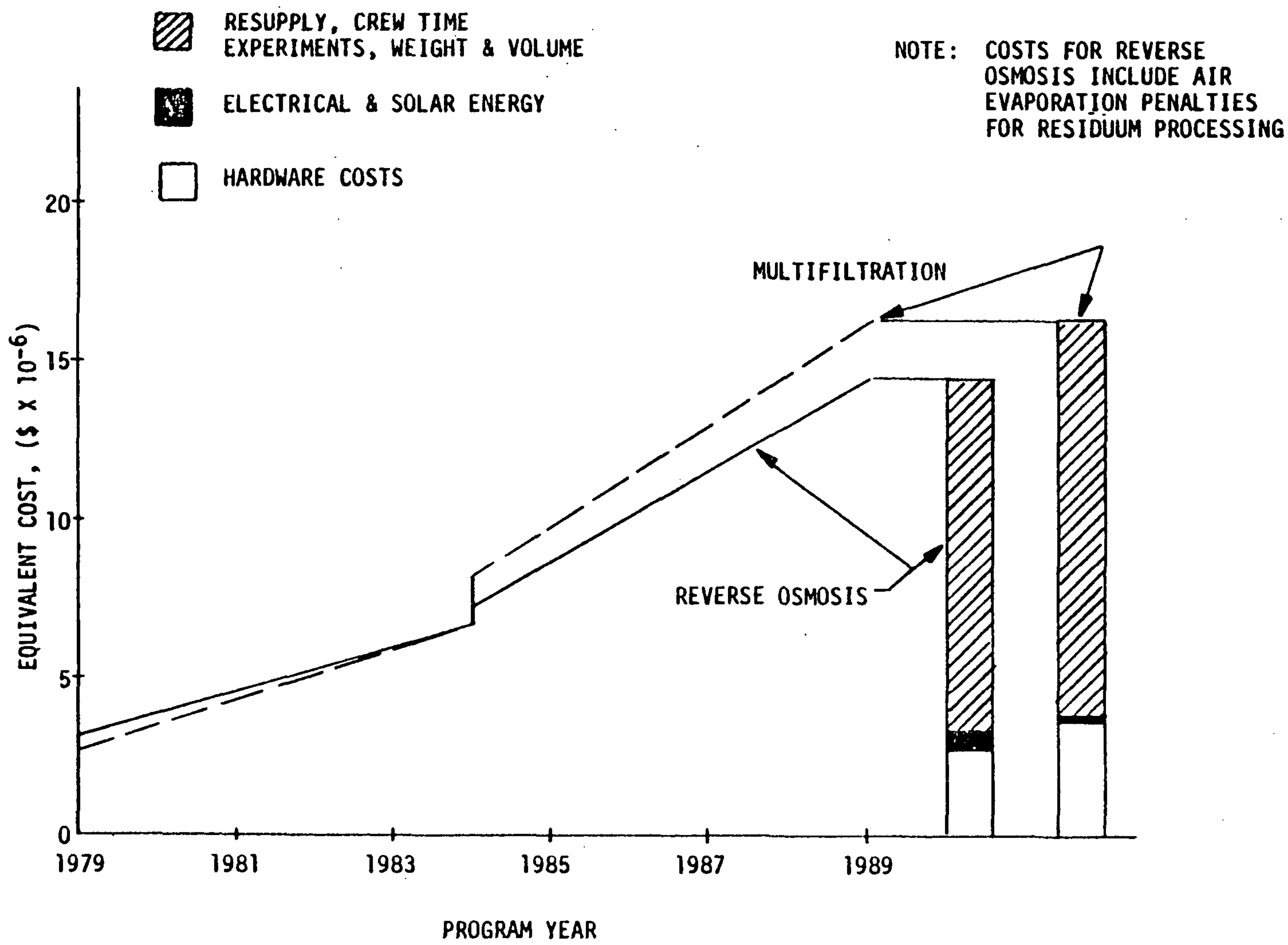

Figure 4.6–36. Cost Comparison for Wash and Condensate Water Recovery Methods

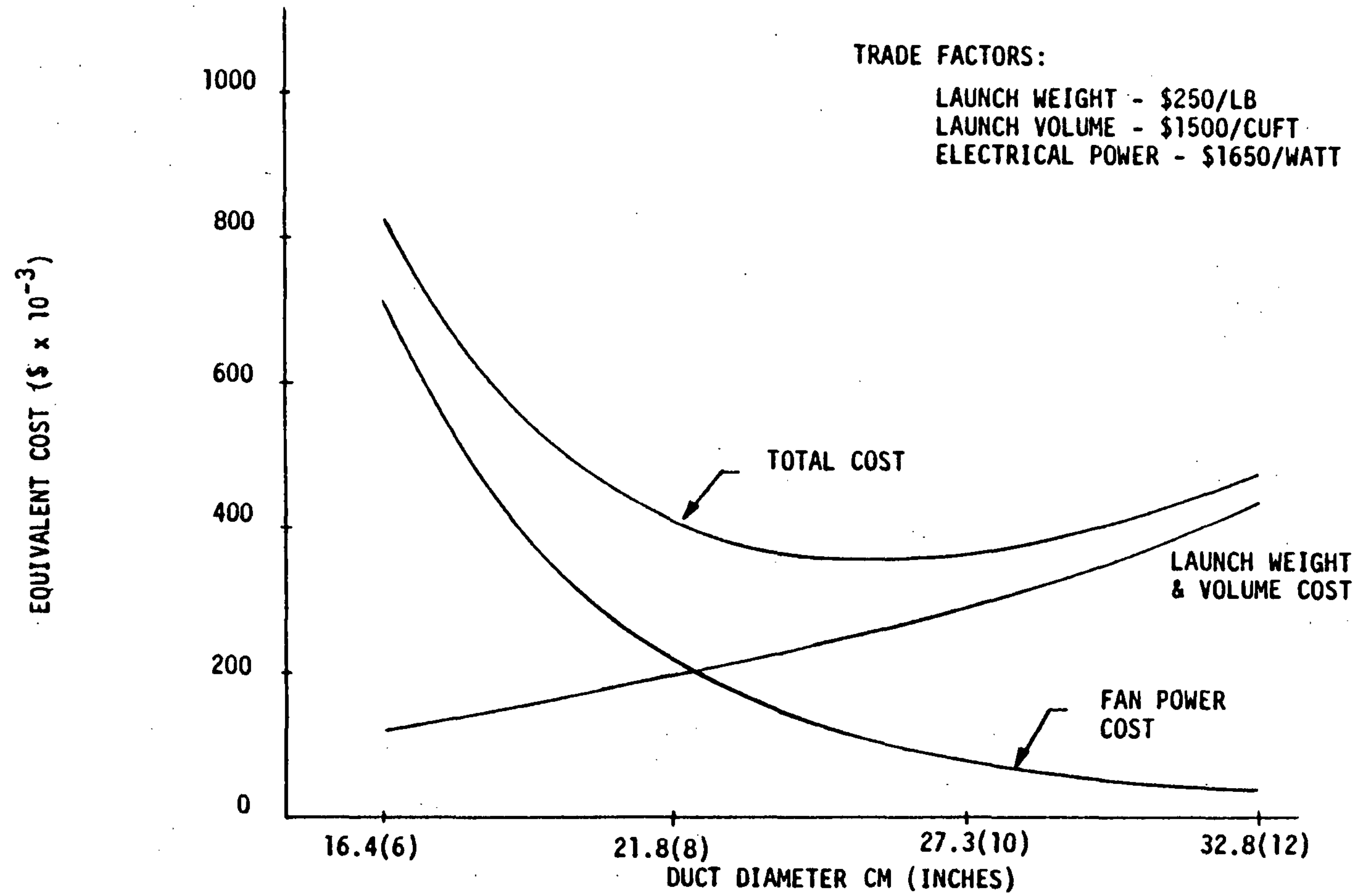

Figure 4.6–37. Ventilation Duct Size Optimization

4.6.4.4.2 Coolant Water Line Size

A coolant water line size optimization was performed in a manner similar to the atmosphere ducting optimization presented above.

The result is shown in Figure 4.6-38 which shows minimum cost occurs between 2.05 (3/4-in.) and 2.73 cm (1-in.) diameter. Based on these results a line size of 2.05 cm (3/4-in.) diameter is selected for the Modular Space Station.

4.6.4.5 Thermal Control System Radiator Design and Analysis

One of the major objectives of the Modular Space Station Phase B Study has been the development of a heat rejection system concept which will meet the stringent requirements of high heat-rejection capacity over a 10-year life without compromising the design of the Station configuration. This objective has been accomplished by selecting an active space radiator approach using Freon 21 in the external loops and water as the heat transfer medium in the internal cabin loops. In the selected design, the external radiators are fixed to the outer structure and integral with the meteoroid shield. Sufficient radiator surface is available on the cylindrical structure to obviate the need for deployable segments. Less conventional alternate designs were considered including refrigeration cycles, heat pipes, and hybrid systems. While the alternatives offer potential advantages, the space radiator concept has been studied extensively in previous studies and in particular, the 10 m (33-ft) Space Station, proved successful on previous space vehicles (Apollo, Gemini, etc.). Alternate systems would require extensive system analysis and testing to achieve workable designs and the high development costs involved would override any potential advantages. Concentration on the conventional active radiator approach has permitted in-depth design analysis to verify performance and accurately establish tube sizing, spacing, and other construction and design details.

A brief description of the concepts selected for the radiator design is given in subsection 4.6.3.1. A detailed thermal analysis of the design indicates that there is adequate heat rejection capacity for all modules for a combination of "worst case" conditions. These include peak station internal heat loads, orbits yielding maximum incident heating, maximum values for

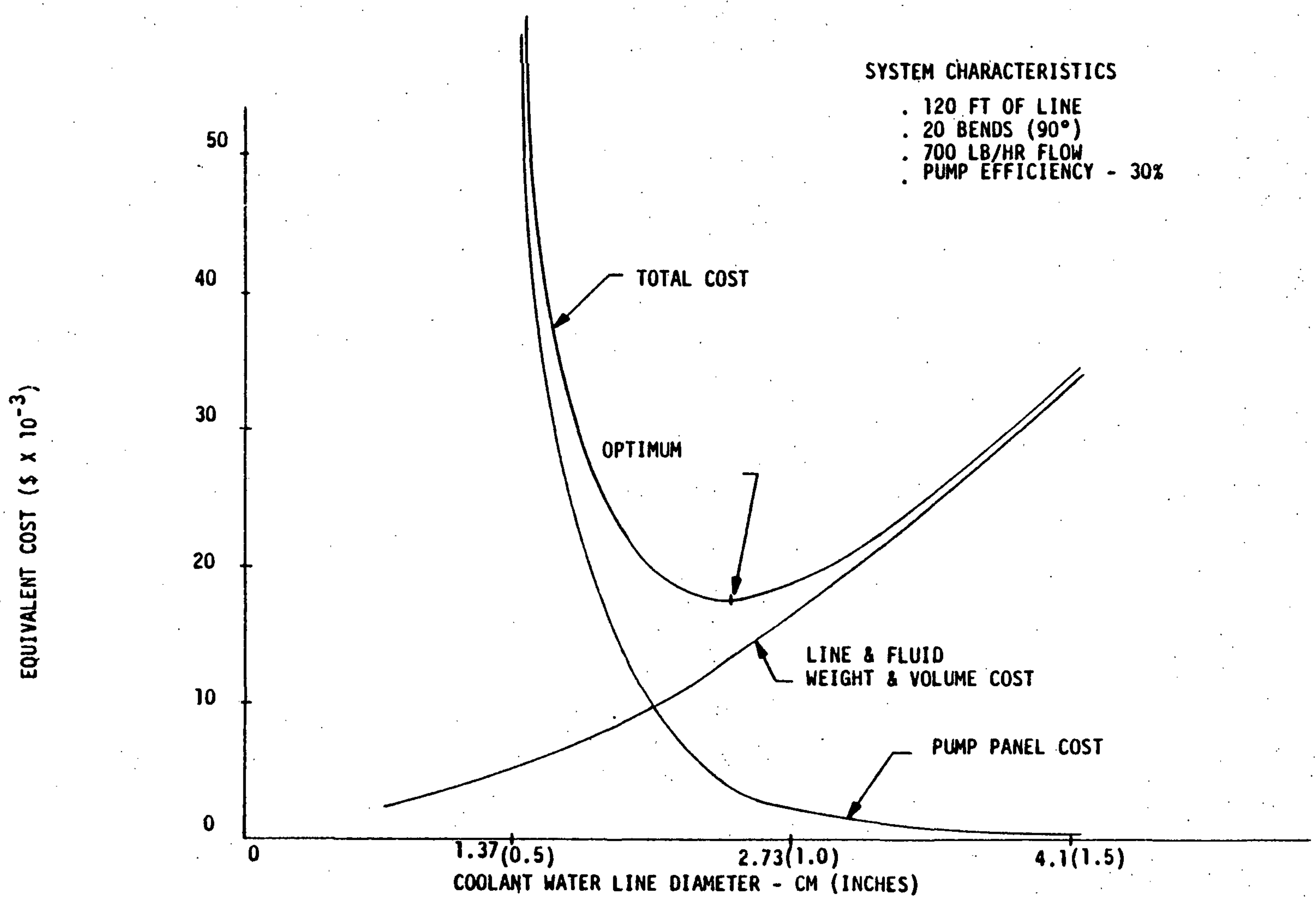

Figure 4.6–38. Coolant Water Line Optimization

orbital heating constants, most degraded surface optical properties (α_s = 0.4), and least desirable vehicle attitude. The resulting performance is satisfactory with or without docked modules which can cause shadowing of incident heating and blockage of radiation view factor to space.

The principal tradeoffs requiring consideration in radiator design are summarized below:

- Thermal coatings
- Deployable or fixed design
- Meteoroid protection
- Radiator fluid
- Radiator fluid temperature control
- Radiator fluid flow regime (laminar, turbulent)
- Radiator segmentation
- Tube layout and spacing
- Independent or integrated radiators for the modules

Several of these tradeoffs were investigated in depth in prior studies and are applicable to the Modular Space Station. In particular, those analyses performed for the 10.06 m (33-ft) diameter Space Station (Reference 4.6-2) provided valuable data to support the current design effort.

The thermal control coating selected, designated S-13-G, consists of a pigment of ZnO in methyl silicone with potassium silicate treatment (Reference 4.6-3). This coating has an initial solar absorptance α_s, of 0.18. Maximum degradation for exposure to 1,580 hours of solar radiation during spaceflight was found to be about 65 percent (Reference 4.6-3). This corresponds to α_s = 0.3. There is insufficient test data for extrapolation to a ten-year mission. For design purposes, a conservative approach has been taken by assuming an initial α_s = 0.2 with a maximum degradation to 0.4. It is assumed that should this value be exceeded, the module coating will be refurbished or replaced either in space or on return to Earth. The coating thermal emittance is assumed to remain constant at 0.9.

Fixed radiators using the cylindrical surfaces of each module have been selected rather than a deployable system. The fixed radiator will provide the minimum weight system with the least complexity. Deployable radiators would warrant consideration only if there were not sufficient area available on the cylindrical surfaces for meeting load requirements.

A design using external Freon 21 loops interfacing with an internal water-cooling loop has been selected, based on the results of previous studies. Freon 21 has the advantages of a low freezing point 138°K (-211°F) and flat viscosity-temperature curve. The low freeze point allows consideration of rather simple methods of radiator fluid temperature control such as by-pass or regenerative heat exchanger control. The flat viscosity curve will assure that the radiator flow resistance will remain essentially constant over a wide range of varying sink temperatures.

A previous study evaluated the relative merits of several methods for radiator outlet temperature control (Reference 4. 6-2). Bypass, regenerative and stagnation methods were evaluated. In by-pass control, a valve modulates flow to the radiator and a line by-passing the radiator so that the mixed temperature of these flows satisfies the control requirement. With this method of control, Freon 21 will freeze in the radiator for heat loads less than 0. 7 of the design value. In the stagnation method, outlet temperature is controlled by affecting a change in effective radiator area by selectively allowing the fluid in certain radiator tubes to freeze when heat loads are low and to melt when heat loads increase. This method offers appreciable performance advantages, but was not selected because the associated design complexity results in high costs. In the regenerative method, heat rather than fluid flow is allowed to bypass the radiator by means of a regenerative heat exchanger. As heat load decreases, more cold radiator fluid is directed to the heat exchanger to lower radiator inlet temperature which in turn lowers the radiating temperatures, thus reducing heat rejection. This method was selected because of its simplicity and wide range of operating capability. For heat loads greater than 0. 2 of design value, the system will operate without the Freon 21 freezing.

The methods discussed above for temperature control are devised to compensate for fluctuation in Station internal heat loads. A thermal capacitance device has been incorporated to damp out fluctuations of temperature due to variations in sink temperature around an orbit. A phase changing material consisting of a mixture of tetradecane and hexadecane provides the thermal storage function of the capacitance device. The capacitor also provides thermal mixing of the two fluids from opposite segments of the radiator. The amount of material required for the capacitors was sized for the orbit presenting the greatest fluctuation in sink temperature. Estimated total weights for the capacitors for the power, crew, and GPL Module are 15.5 kg (34 lb), 27.8 kg (61 lb), and 38.6 kg (85 lb), respectively. The thermal capacitor, which is part of the radiator control assembly (6200), also provides a heat sink during launch; the wax is frozen just prior to launch. The thermal capacitor provides cooling until the launch vehicle is beyond the atmosphere after which cooling is provided by the ground heat exchanger/ water boiler (6201). This unit becomes inactive, except for emergency operation, after radiator activation.

The radiator design has been influenced considerably by the need to maintain turbulent flow with $Re \geq 10,000$ in the tubes. The advantages of maintaining a high Reynold's number have been given previously (Reference 1). In particular, the heat transfer coefficient for Freon 21 at $Re = 10,000$ is approximately 10 times that at $Re = 1,100$.

The selected design of the space station modules provides separate thermal control systems for each module. This offers considerable advantage during buildup of the station. Systems or individual modules can be checked out and operated on-orbit without having to rely on the docking of the other modules to provide heat rejection capacity. In addition, a reliability and safety advantage is provided since a module can still reject heat even if the thermal control systems fail in the other modules. Additional flexibility is provided should removal and refurbishment of one of the modules be required.

Completely independent systems have the disadvantage that a radiator failure in a particular module would remove the heat rejection capacity from that module. A transfer loop has been added which will allow heat load transfer between modules when necessary. Several methods were evaluated for accomplishing the load transfer. The method selected uses a water-transfer loop and no Freon 21 interfaces which might present a safety hazard to the crew are required between modules. Figure 4.6-39 illustrates the operation of the system. The logic for operation of the system is explained in detail in subsection 4.6.3.1.6 and will not be repeated here.

Using this basic concept for the radiator design, detailed analyses were performed to optimize the performance of the selected thermal control system. These analyses resulted in:

- Determination of tube size, spacing, and layout.
- Location of inlet/outlet manifolds.
- Determination of dimension of a fin and tube extrusion with sufficient protection from meteoroid damage.

Details concerning these items are described in the following subsections.

4.6.4.5.1 Tube Size, Spacing, and Layout

A serpentine tube layout is required for all modules. Assuming a minimum allowable tube size of 0.483 cm (0.19 in.), turbulent flow with $R_e > 10,000$ cannot be achieved with parallel tubes only. Thus, a combination of series and parallel tube passes is required to achieve the desired flow regime in the individual tubes. Total radiator flow for each module was based on maximum heat rejection capacity for that module and the maximum allowable temperature change for the fluid in the radiator. For the Crew/Operations and GPL Modules this maximum change is based on specifications of 311.4°K (101°F) for the radiator inlet and 274.7°K (35°F) for radiator fluid outlet temperature. The maximum inlet temperature is set by the maximum allowable temperature for the electronics cooled by the cold plates. The minimum is required for adequate humidity control.

Because of the high heat-load requirements and smaller radiator area available in the Power/Subsystems Module, use of higher inlet/outlet temperatures than on other modules is necessary to meet maximum heat-rejection requirements under worst-case heating conditions. The required

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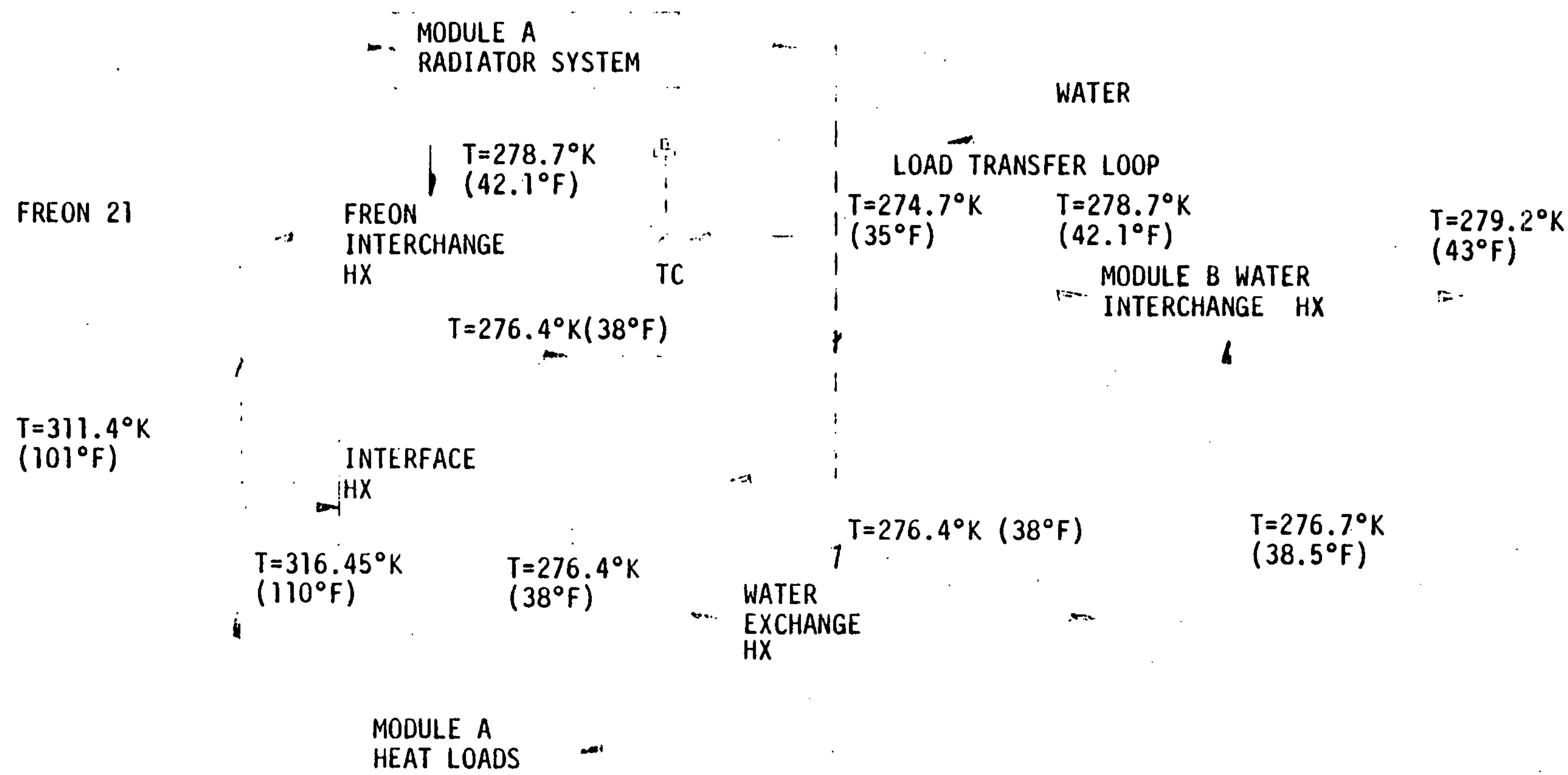

Figure 4.6–39. Thermal Control System Using Water Transfer Loop

inlet/outlet temperatures are 315.3/285.3°K (108/54°F), respectively. This will provide adequate control of all heat loads in the power module with the exception of latent and sensible heat loads in the module's cabin atmosphere. These loads result from losses to the air from equipment and plumbing and from crewmen when the module is occupied. These loads must be handled by the Crew/Operations Module air conditioning system.

To maximize heat-rejection capacity, it is necessary to use the area located between the docking ports on the Power/Subsystems and Crew/Operations Modules. Approximately 14.2 m^2 (153 ft^2) can be gained for each of these modules by using the area. For this reason, the serpentine tube passes are arranged in the axial direction to simplify layout of tubes in the vicinity of the docking ports. The overall length for a series pass is not significantly different from passes made in a circumferential direction, therefore no pressure-drop penalty results. Total cylindrical area including the area near docking ports is 56.5 m^2 (543 ft^2), 122 m^2 (1,314 ft^2), and 153 m^2 (1,648 ft^2), respectively for the Power/Subsystems, Crew/Operations, and GPL Modules.

The forward conical section of the Power/Subsystems Module is also used to increase the heat-rejection capacity of this module. An additional 24.5 m^2 (264 ft^2) is made available by using this surface.

Figure 4.6-40 through 4.6-43 show the selected tube layout for the three modules. A summary of important radiator design parameters is given in Table 4.6-12. Heat rejection requirements are summarized in Table 4.6-13.

4.6.4.5.2 Location of Inlet Manifold

The location of the inlet manifold for the primary and secondary circuits has an important effect on vehicle heat-rejection capacity. Previous results have indicated that for high Beta angles, and neglecting the effect of the Earth, maximum heat rejection capacity will result when there is an angle of zero between the vehicle-sun vector and the plane formed by the inlet manifold of the active circuit. Thus, flow in both radiator panels is in a direction away from the sun. This condition is illustrated by Figure 4.6-44.

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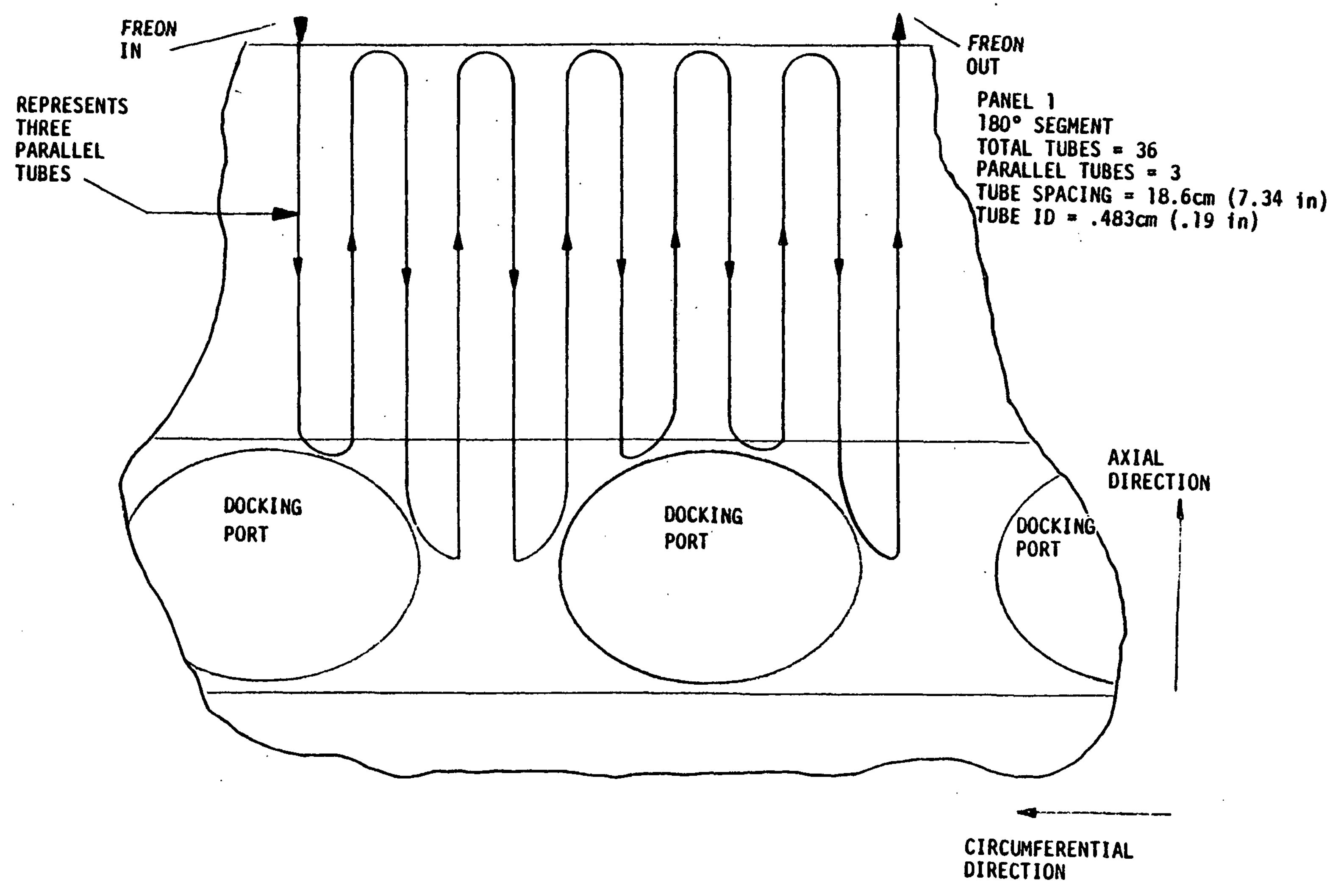
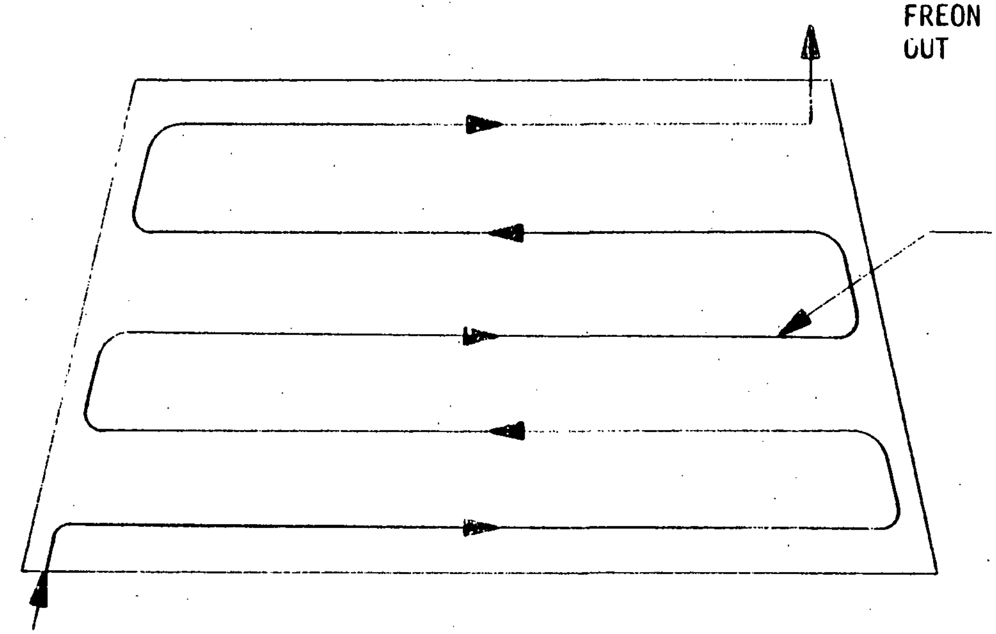


Figure 4.6–40. Power/Subsystems Module Radiator (Cylinder Part)



PANEL 5
TOTAL TUBES = 15
PARALLEL TUBES = 3
TUBE SPACING = 17.8 cm (7.0 in)
TUBE ID = .483 cm (.19 in)

NOTE: CONICAL RADIATOR HAS TWO PRIMARY AND TWO SECONDARY CIRCUITS

Figure 4.6–41. Power/Subsystems Module Forward Cone Radiator

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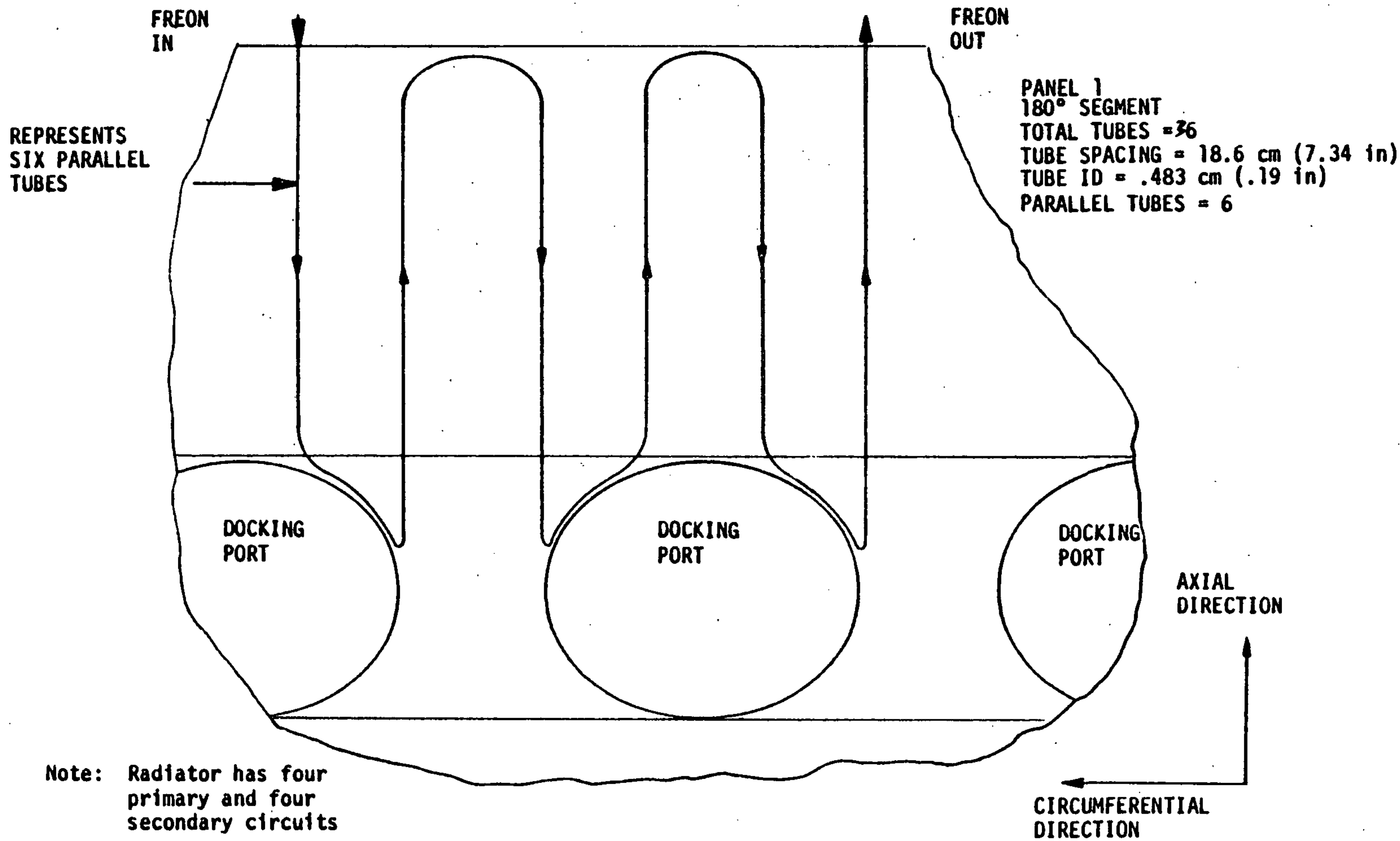


Figure 4.6–42. Crew/Operations Module Radiator

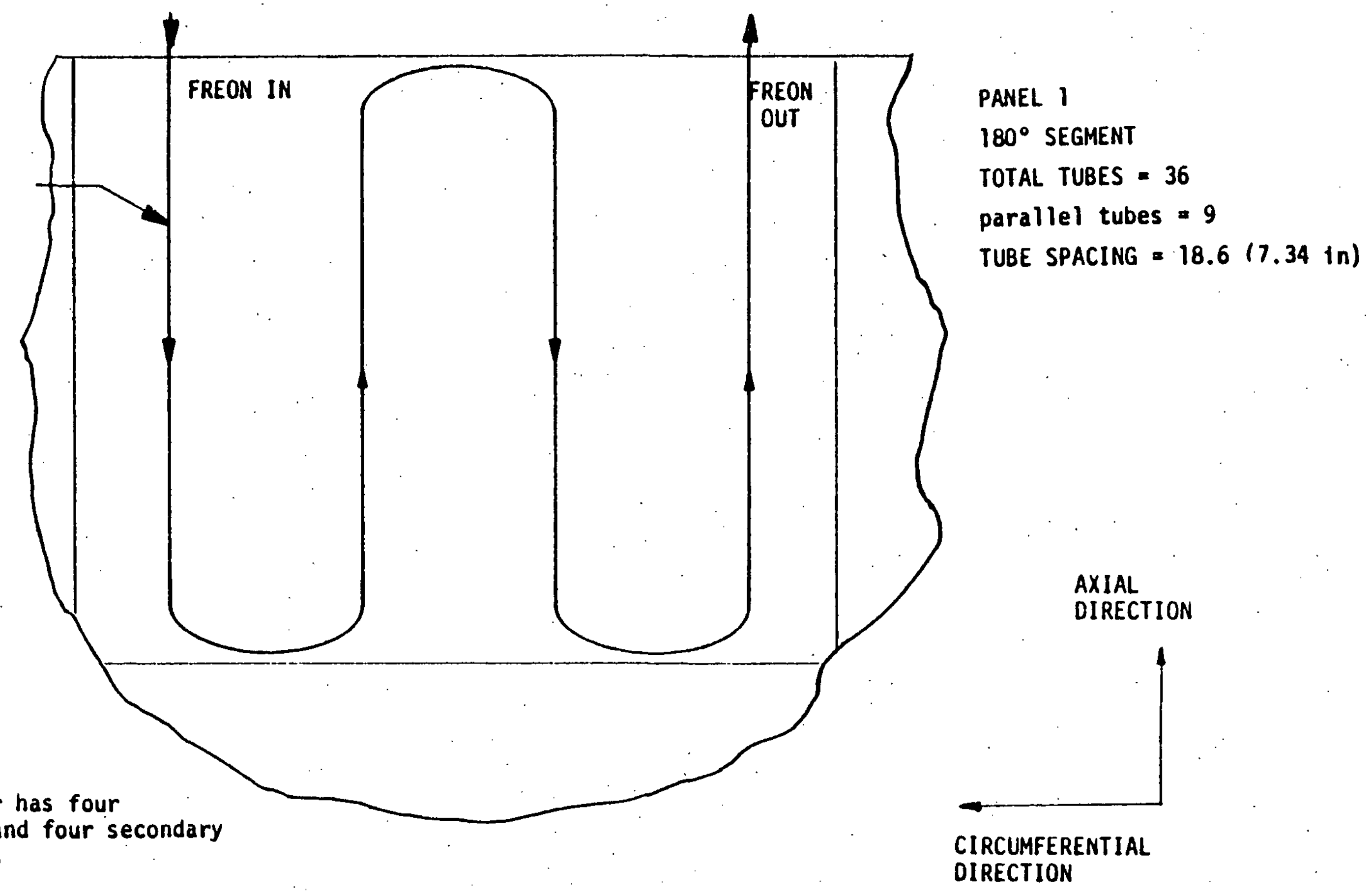


Figure 4.6–43. GPL Radiator

Table 4.6-12

RADIATOR DESIGN PARAMETERS

Module	No. of Circuits	Area m^2 (ft^2)	D_{ID}, cm (in.)	L_f, cm (in.)	W_T, kg/sec (lb/hr)	$\dot{W}_{tube}$, kg/sec (lb/hr)	R_e	N_T	N_P	N_S
GPL	4	153 (1,648)	0.483 (0.19)	18.6 (7.34)	0.526 (4,180)	1.46×10^{-2} (116)	11,060	36	9	4
Crew/ Operations	4	122 (1,314)	0.483 (0.19)	18.6 (7.34)	0.350 (2,780)	1.46×10^{-2} (116)	11,000	36	6	6
Power/ Subsystems										
Cylinder	4	50.5 (543)	0.483 (0.19)	18.6 (7.38)	0.1714 (1,360)	1.42×10^{-2} (113)	10,800	36	3	12
Cone	2	24.5 (264)	0.483 (0.19)	17.8 (7.0)	0.083 (660)	1.39×10^{-2} (110)	10,500	15	3	5
Total		75 (807)								

Where:

D_{ID} = tube diameter

L_f = spacing between

W_T = total flow rate in radiator

$\dot{W}$ = flow rate per tube

R_e = Reynold's number

N_T = total number of tubes in 180 deg segment

N_P = number of parallel tubes in 180 deg segment

N_S = number of series passes for a tube in 180 deg segment

Table 4.6-13

BASELINE HEAT REJECTION REQUIREMENTS

Configuration		Type	Average Electrical, Watts (Btu/hr)	Average Metabolic, Watts (Btu/hr)	Average Solar Collector, Watts (Btu/hr)	Average Total, Watts (Btu/hr)	Peak Loads (Btu/hr)
ISS	GSS	Power/ Subsystems	6,954 (23,713)	0	---	5,930 (20,213)*	6,150 (21,000)*
ISS	GSS	Crew/ Operations	6,149 (20,968)	545 (1,860)	1,700 (5,800)	9,420 (32,120)*	10,920 (37,338)
ISS	GSS	GPL	10,770 (36,726)	273 (930)	1,700 (5,800)	12,750 (43,456)	15,380 (52,410)
	GSS	Crew/ Operations	4,199 (14,318)	410 (1,400)	2,270 (7,730)	6,890 (23,448)	---
	GSS	Power/ Subsystems	5,587 (19,000)	410 (1,400)	---	5,980 (20,400)	---

*Approximately 1.03 kw (3,500 Btu/hr) of sensible heat dissipated in Power/Subsystems Module is removed by air conditioning supplied by Crew/Operations Module EC/LS Subsystem.

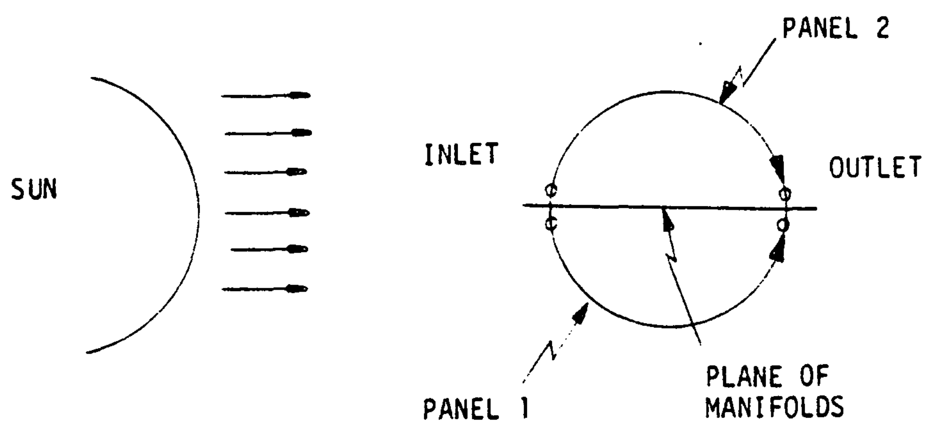


Figure 4.6–44. Optimum Orientation for Maximum at β = 78.5 Deg

Figure 4.6-45 shows the locations selected for the inlet manifolds for the core modules. These locations should nearly maximize heat-rejection capacity for the worst-case heating conditions which occur at a Beta angle (β) of 78.5 deg. The baseline attitude with the ES-1G and CN-1B RAMS pointed towards the Earth was assumed in selecting the inlet position. Inlet/outlet manifolds are positioned every 90 degrees measured from the centerline of the RAM docking ports.

Figure 4.6-46 shows the angle between the manifold plane and sun vector for β = 78.5 degrees. At the top and bottom of the orbit, the angle is 11.5 degrees. At the sides of the orbit, the angle will be zero.

The position of the manifolds with respect to the sun for β = 0 is shown in Figure 4.6-47.

For a sun-oriented vehicle, the secondary circuit has an orientation which maximizes heat rejection. For an earth-oriented vehicle (the preferred orientation), the radiator flow reversal valves are used to keep the flow in the secondary circuit directed away from the sun throughout the orbit.

The location for the inlet manifolds for the GPL also has been selected so that the flow in each panel is in a direction away from the sun. The plane of the manifold is perpendicular to the lateral axis of the vehicle core modules. (See Figure 4.6-48.)

The manifold positions selected will minimize the angle between manifold plane and the sun-vehicle vector for the baseline vehicle attitude. Should the vehicle be rolled for operational or experimental purposes, this angle will increase, thus decreasing radiator performance. Since the system has been designed with primary/secondary inlet/outlet manifolds every 90 degrees, the maximum angle of the manifold plane from the sun will be 45 degrees. The worst-case position of the manifolds for the 10.06 m (33-ft) diameter Station was assumed to be more severe. In this study a worst-case angle of 90 degrees had to be assumed since inlet/outlet manifolds were located every 180 degrees. The 90-degree positioning for the Modular Space Station represents a design improvement.

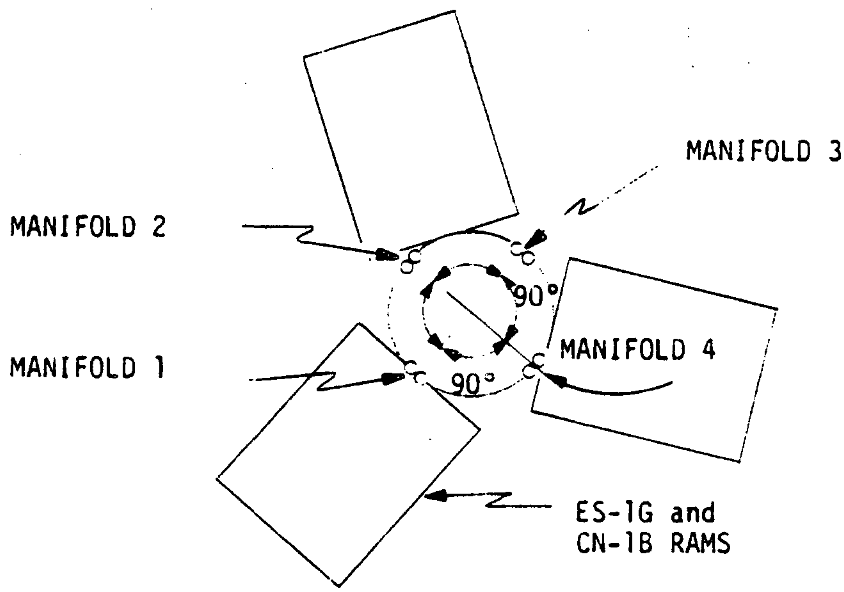


Figure 4.6–45. Selected Manifold Locations

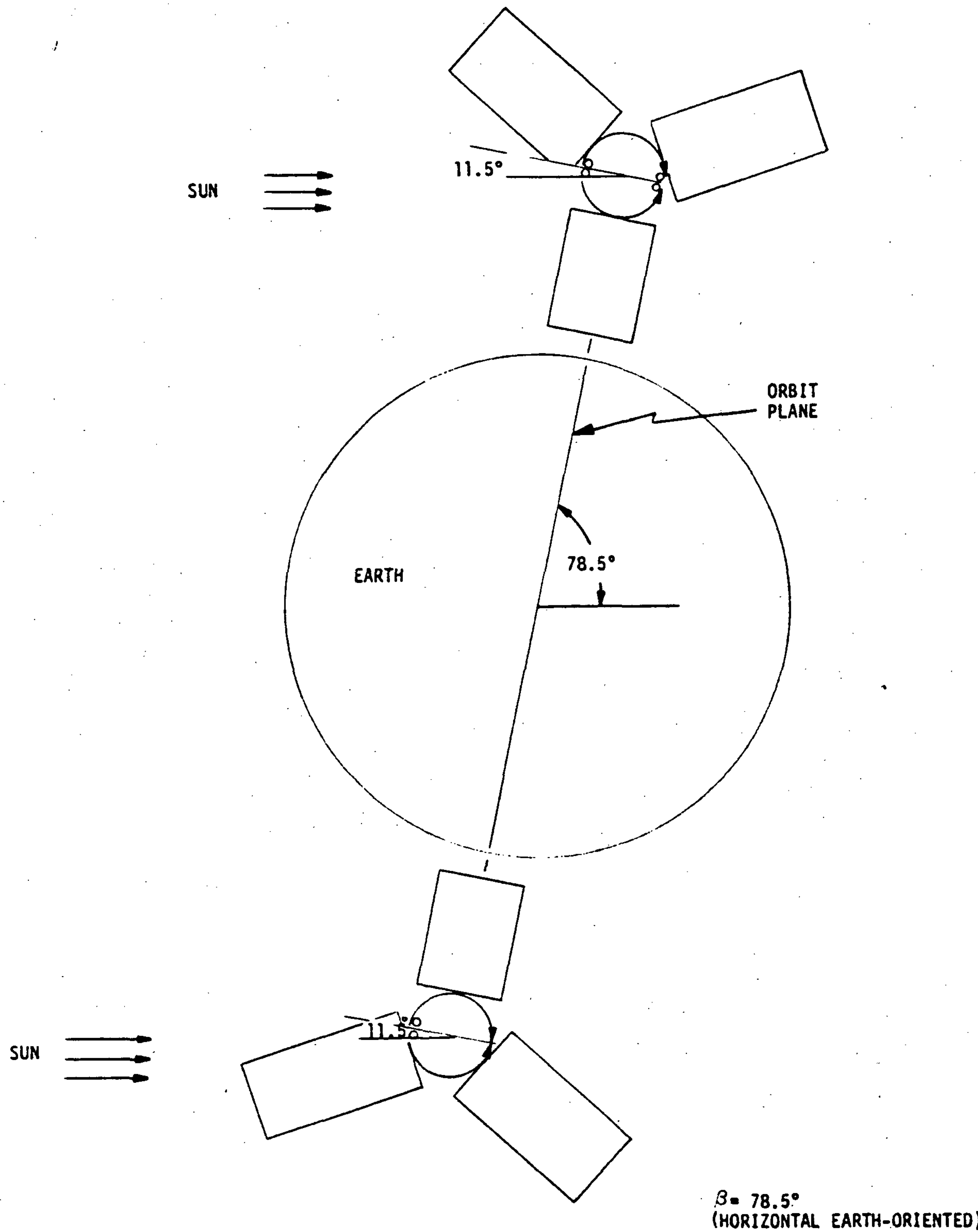


Figure 4.6–46. Vehicle–Sun Orientation β = 78.5 Deg (Horizontal, Earth Oriented)

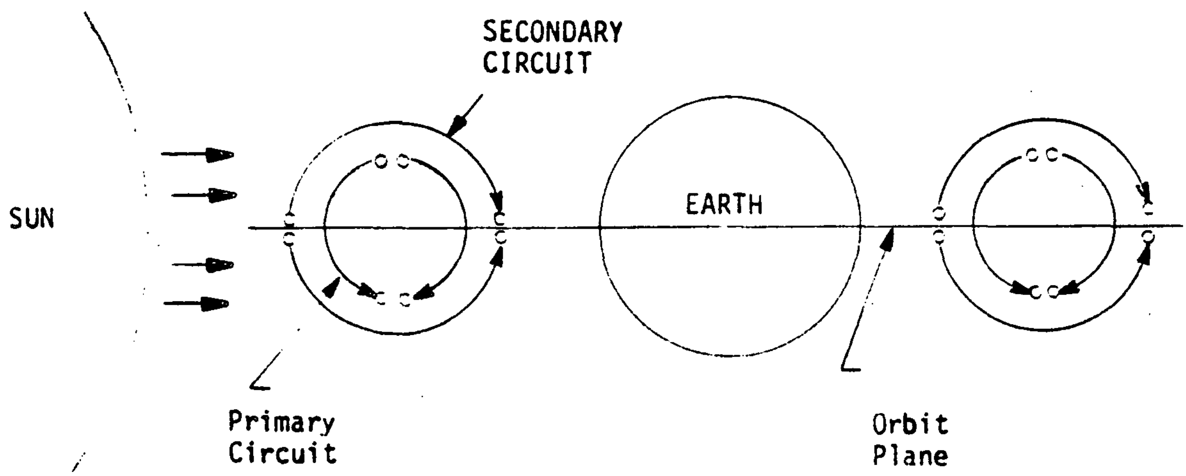


Figure 4.6–47. Orientation of Manifolds for β = 0 (Sun–Oriented Vehicle)

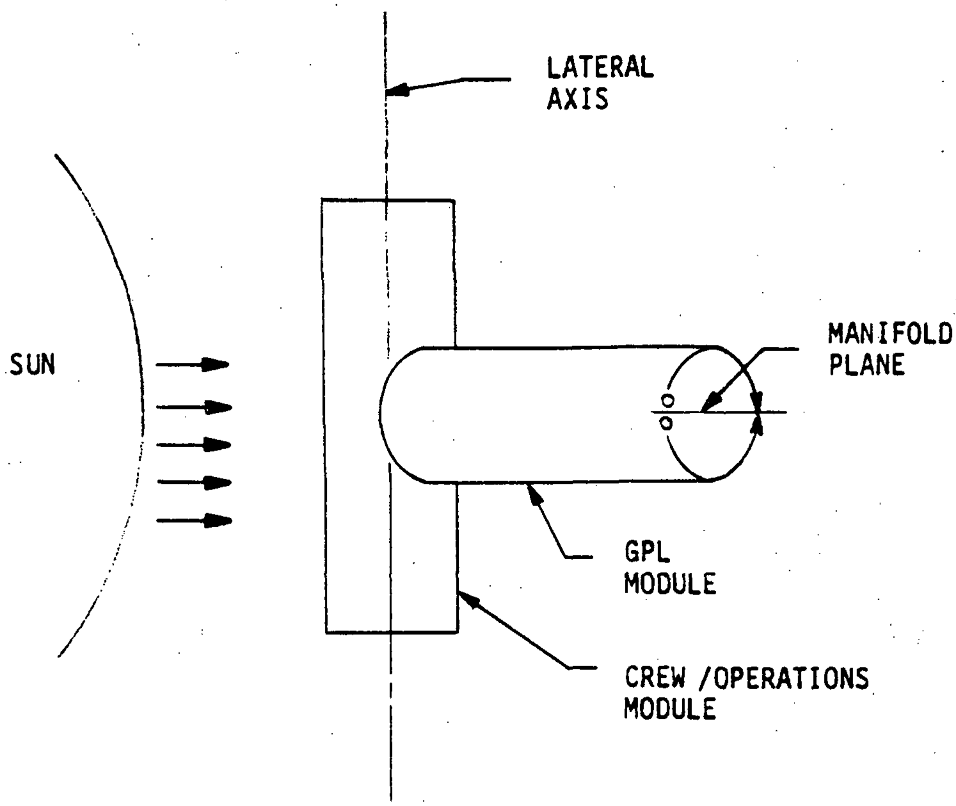


Figure 4.6–48. GPL Manifold Location

4.6.4.5.3 Design of Meteoroid Protection for Radiator

The need to protect the radiator from damage due to meteoroid impact influences the radiator design. A double bumper providing a system reliability of 0.99 has been incorporated into the selected design. Tubes are separated from the radiator fin surface by a standoff of suitable dimensions. The radiator fin surface acts as a primary bumper while the tube wall acts as the secondary bumper. Fin and tube wall thickness, and standoff thickness and length, were sized to meet requirements for meteoroid protection and heat rejection. An integral extrusion has been designed which includes the tube, standoff, and fin surface. The assumptions and calculation procedures used for determining the design are outlined in subsection 4.2.3.3.

4.6.4.6 Computerized Parametric Thermal Analysis of Radiator System

A detailed computer simulation of the Space Station was prepared to evaluate performance under a set of worst-case heating conditions. Two computer programs were used to accomplish the analysis. The MDAC PO-333 Thermal Radiation Heating Program was used to determine absorbed heating rates for the vehicle and the graybody shape factors from surfaces to deep space. The total heating includes direct and reflected solar, albedo, and earth infrared. The MDAC JA-15 (CINDA) Thermal Analyzer Program was used to determine heat-rejection capacity by means of a lumped parameter simulation of the radiator. Effects were included for variation in sink temperature due to orbit position, shadowing due to the presence of radial modules, and blockage of radiation view factor to space also due to the presence of the radial modules.

A set of parameters was defined which brackets the conditions for which the radiator must perform adequately. These include vehicle configuration values for heating constants, range of solar absorptivity, and vehicle attitude and orbit. The set of computer cases which were run using these parameters is shown in Table 4.6-14.

A brief description of the rationale for the selection of parameters is given in the following subsections.

Table 4.6-14
PARAMETRIC THERMAL ANALYSIS

Set No.	Orbital Heating Constants	Surface Coating Properties		Stage of Buildup			Orientation				Remarks
		α_S	ϵ_T	Configuration	Station Modules	RAMS and LM	Type	Roll Angle (deg) Ψ	Sun Angle (deg) β	Manifold Plane Angle (deg) ϕ	
I	Maximum (per References 4.6-5 and 4.6-6)	0.2	0.9	ISS	1, 2, 3	None	Horizontal	33.5	78.5	45 to 67	β = Most severe
		0.3	0.9	ISS	1, 2, 3	None	Horizontal	33.5	78.5	45 to 67	Ψ = Severe. Maximum heat const. No shadowing or blockage.
		0.4	0.9	ISS	1, 2, 3	None	Horizontal	33.5	78.5	45 to 67	
II	Maximum (per References 4.6-5 and 4.6-6)	0.4	0.9	ISS	1, 2, 3	None	Horizontal	0	78.5	11.5	β = Most severe. Ψ = Nominal Value. Maximum heat const. No shadowing or blockage.
III	Maximum (per References 4.6-5 and 4.6-6)	0.4	0.9	ISS	1, 2, 3	None	Horizontal	33.5	78.5	45 to 67	Same as Set I. Except two-thirds of total flow modulated to cold panel.
IV	Maximum (per References 4.6-5 and 4.6-6)	0.4	0.9	ISS	1, 2, 3	None	Solar Inertial	0	0	0	β = Severe. Ψ = Most severe. Maximum heat const. No shadowing or blockage.
V	Maximum (per References 4.6-5 and 4.6-6)	0.4	0.9	ISS	1, 2, 3	RAMS 1, 2, 3, 4 LMI	Horizontal	33.5	78.5	45 to 67	β = Most severe. Ψ = Severe. Maximum heat const. Shadowing and blockage.
VI	Maximum (per References 4.6-5 and 4.6-6)	0.4	0.9	ISS	1, 2, 3	RAMS 1, 2, 3, 4 LMI	Horizontal	0	78.5	11.5	β = Most severe. Ψ = Nominal value Maximum heat const. Shadowing and blockage
VII	Nominal (per References 4.6-5 and 4.6-6)	0.2	0.9	ISS	1, 2, 3	RAMS 1, 2, 3, 4 LMI	Horizontal	0	78.5	11.5	β = Most severe. Ψ = Nominal value Nominal heat const. Shadowing and blockage.
		0.3	0.9								
		0.4	0.9								

4.6.4.6.1 Configuration

Effort was concentrated on a detailed analysis of the ISS configuration. Results for the GSS stage of buildup can be ascertained by extrapolating data from the ISS stage. The ISS was analyzed with and without docked RAM's or Logistics Modules to obtain the worst-case heating environment. Figure 4.6-49 shows the ISS stage of buildup.

4.6.4.6.2 Heating Constants

Incident heating rates were determined using heating constants which will yield the maximum heating corresponding to a 3σ deviation from nominal values. The data used in determining these constants was taken from References 4.6-5 and 4.6-6. Table 4.6-15 summarizes nominal values, 2σ and 3σ minimum values, and 2σ and 3σ maximum values for incident Earth infrared and solar heating, and Earth albedo. The values presented for the 2σ and 3σ variations for solar heating include the effects of seasonal variation due to the eccentricity of the Earth's orbit.

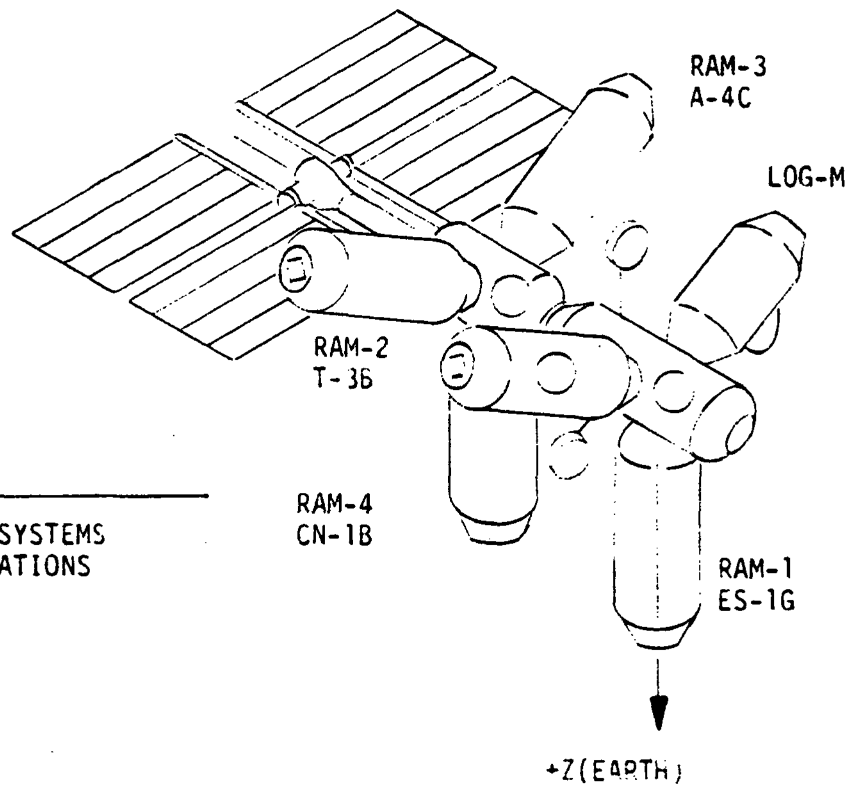
4.6.4.6.3 Surface Absorptivity

A radiator coating has been selected with an initial solar absorptance, α_s, of 0.18 and thermal emittance of 0.9. The thermal emittance is assumed to remain constant over the 10-year life of the spacecraft. The degree of α_s degradation is a complex function of exposure time to solar radiation and meteoroid impact. Because of many uncertainties involved in predicting values for α_s, for design purposes α_s is assumed to lie within the range of 0.2 to 0.4. Parametric runs were performed for surface absorptivities of 0.2, 0.3, and 0.4 to bracket radiator performance.

4.6.4.6.4 Vehicle Attitude and Orbit

The contract guidelines state that the radiator must perform adequately for any possible orientation. For analysis purposes, two orbit conditions were considered:

1. β = 78.5 degrees with vehicle in a horizontal-Earth orientation
2. β = 0 degrees with vehicle sun oriented (inertially oriented)



MODULE SEQUENCE	PRIMARY FUNCTION
1	POWER/SUBSYSTEMS
2	CREW/OPERATIONS
3	GPL

Figure 4.6–49. ISS Configuration

Table 4.6-15

SUMMARY OF CURRENT ORBITAL HEATING CONSTANTS

(References 4.6-5 and 4.6-6)

Average Values	Tolerances	Seasonal Variation
$\overline{E} = 0.238 \frac{kw}{m^2}$ (75.3 Btu/hr-ft^2)	$\sigma_E = 0.01$ (6.65)	+3.43 percent -3.26 percent
$\overline{A} = 0.3$	$\sigma_A = 0.06$	
$\overline{S} = 1.358 \frac{kw}{m^2}$ (429 Btu/hr-ft^2)	$\sigma_S = 0.0067$ (2.12)	

	Incident Heating, $\frac{kw}{m^2}$ (Btu/hr-ft^2)				Background Temperatures ^oK (^oR)				
	Minimum		Maximum		Minimum		Nominal	Maximum	
Parameter	3σ	2σ	2σ	3σ	3σ	2σ		2σ	3σ
Earth IR	0.174 (55)	0.196 (62)	0.28 (88.6)	0.3 (95.2)	233 (420)	241 (433)	252 (454)	263 (474)	268 (482)
Albedo*	0.12	0.18	0.42	0.48	---	---	---	---	---
Solar	1.29 (409)	1.3 (411)	1.41 (448)	1.42 (450)	5,640 (10,192)	5,680 (10,250)	5,750 (10,365)	5,820 (10,480)	5,850 (10,530)

*Refers to fraction of incident solar heat reflected by Earth

Previous analysis has indicated that for a horizontal, earth orientation, the worst-case heating will occur when the angle Beta (β), between the vehicle-sun vector and orbit plane is 78.5 degrees. For this angle, the vehicle is in sunlight for the complete orbit. The location of the inlet manifolds was selected in order to minimize the effect of solar heating. For β = 78.5 degrees, the angle between the plane formed by the inlet and outlet manifolds and the vehicle-sun vector will be 11.5 degrees. Thus, flow in both radiator panels will be in a direction away from the sun. Figure 4.6-50 illustrates the relationship between vehicle, sun, and Earth for β = 78.5 degrees. Cases were run for this condition with and without docked radial modules on the side facing the sun in order to assess the effects of module shadowing of radiant heat and blockage of view factor to space.

Because of the possibility of the vehicle being rolled for operational purposes, analysis was performed also for a rolled condition which presented the worst case angle between the manifold plane and vehicle-sun vector. Figure 4.6-51 shows the vehicle attitude for the assumed worst-case roll angle of Ψ = 33.5 degrees. The angle ϕ between the manifold and sun vector varies between 45 and 67 degrees at the top and bottom, respectively of the orbit. The angle at the sides of the orbit is 56.5 degrees. This condition is analyzed with and without docked radial modules.

Analysis was also performed for conditions with β = 0 degrees and the vehicle in a solar inertial orientation. This condition causes the most severe environment for direct solar, reflected solar (albedo), and Earth infrared on the sunlighted portion of the orbit. For this condition the vehicle will be in sunlight for approximately 62 percent of the orbit. For the remainder of the orbit, the vehicle will be in the Earth's shadow. The radiator outlet temperature has the most extreme fluctuations for β = 0 degrees. Thus, the results of this analysis were used to size module thermal capacitance devices for damping out fluctuations due to variations in effective sink temperature around the orbit. Figure 4.6-52 shows the vehicle-sun relationship for β = 0 degrees. The orientation shown in Figure 4.6-52 presents the maximum heating condition for the Power Subsystem and Crew Operations Modules. A slightly different condition with

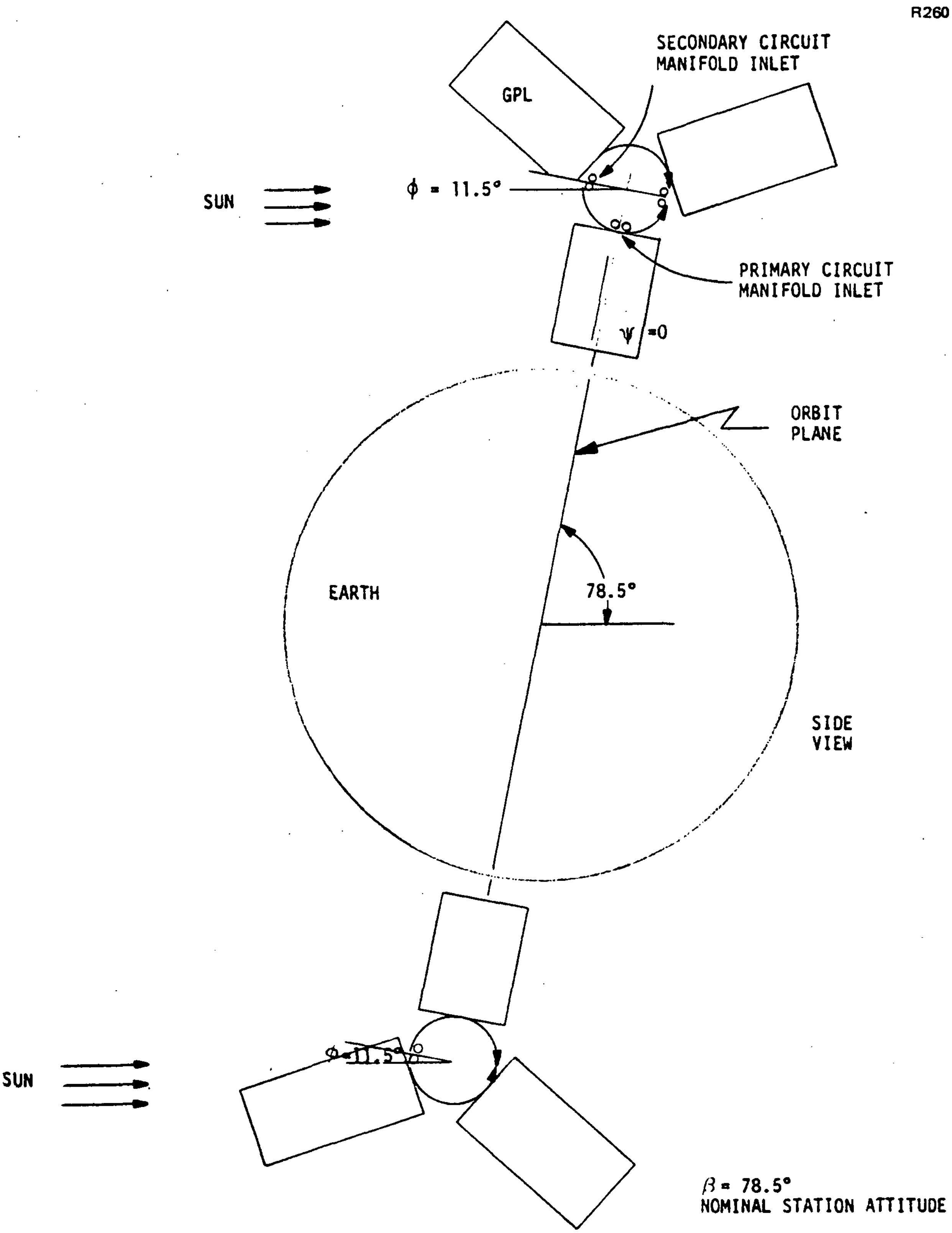


Figure 4.6–50. Vehicle–Sun Orientation β = 78.5 Deg (Nominal Station Attitude)

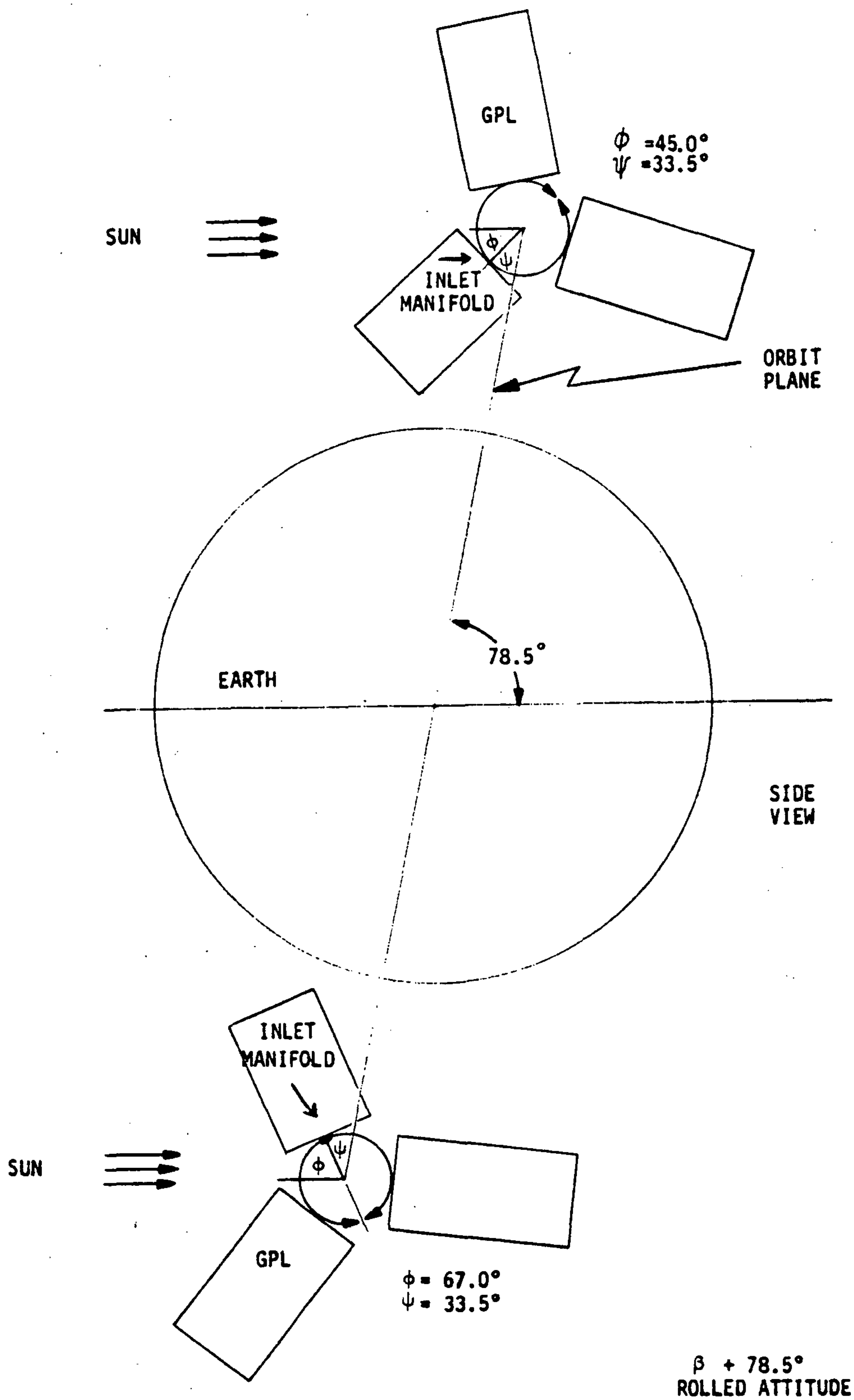


Figure 4.6–51. Vehicle–Sun Orientation β = 78.5 Deg (Rolled Attitude)

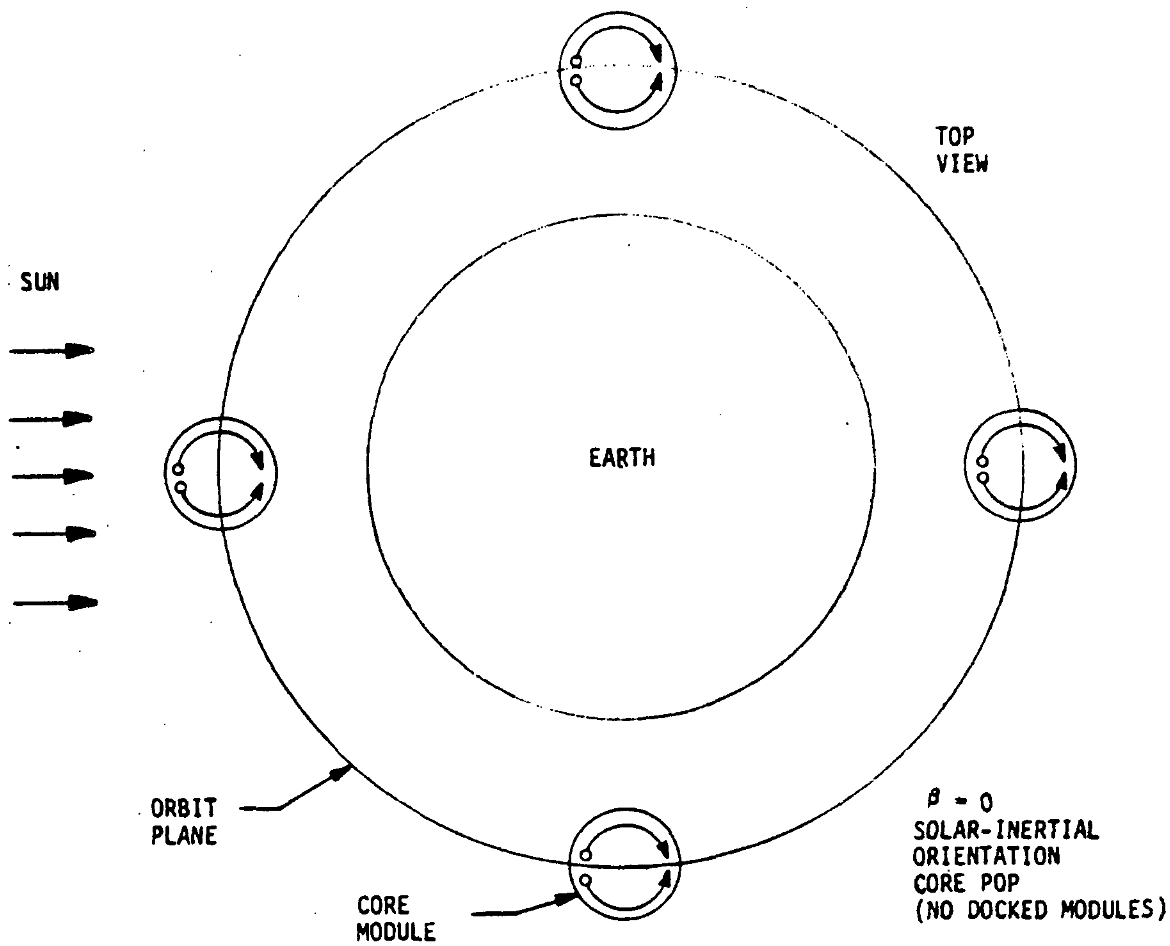


Figure 4.6–52. Vehicle–Sun Orientation $\beta = \phi$

the Station rolled so that the GPL Modules are oriented perpendicular to the sun-earth line was used for analysis of the GPL Module.

4.6.4.6.5 Determination of Incident Heating and Heat-Rejection Capacity
Orbital heating rates for the Station were determined using the MDAC PO-333 Thermal Radiator Heating Program. For analysis purposes, the Station is modeled as a number of surfaces suitably connected to simulate the vehicle configuration. Surfaces are described as portions of rectangles, disks, trapezoids, cylinders, cones, spheres, or paraboloids. The program will accommodate up to 140 surfaces and will calculate shadowing effects and diffuse reflections between these surfaces.

Using cathode ray tube (CRT) equipment, orthographic projections of the analytical model may be requested along with a three-dimensional view to verify model input data. Since the calculation of the black-body view factor, the gray-body interchange factors, and the absorbed heat fluxes by each of the modes can consume large amounts of computer time, it is advantageous to use plotter subroutines to verify correct model surface, orbit, and orientation data. Plots were obtained for all vehicle configurations and orbits which were considered. Figure 4.6-53 is a sample of a three-dimensional view of the Station for the orientation considered in Set VI. A plot of the Station attitude for the orbit of Set VI is shown in Figure 4.6-54.

After the geometric input data for the Space Station surfaces and appropriate orbit configuration were verified, gray-body radiation interchange factors and total absorbed heat fluxes (including reradiation and reflected energy) which depend on the solar absorptivity for each surface mode were determined. These results are output on punched cards and magnetic tape in a format compatible with input data for the CINDA Thermal Analyzer Program which is used for a detailed model analysis of the temperatures of the radiator fluid and structure.

The thermal analogue used to simulate the performance of the Crew Operations, Power Subsystem, and GPL Modules consisted of separate nodes for fluid, structure, and fin surfaces. Nodes covered approximately

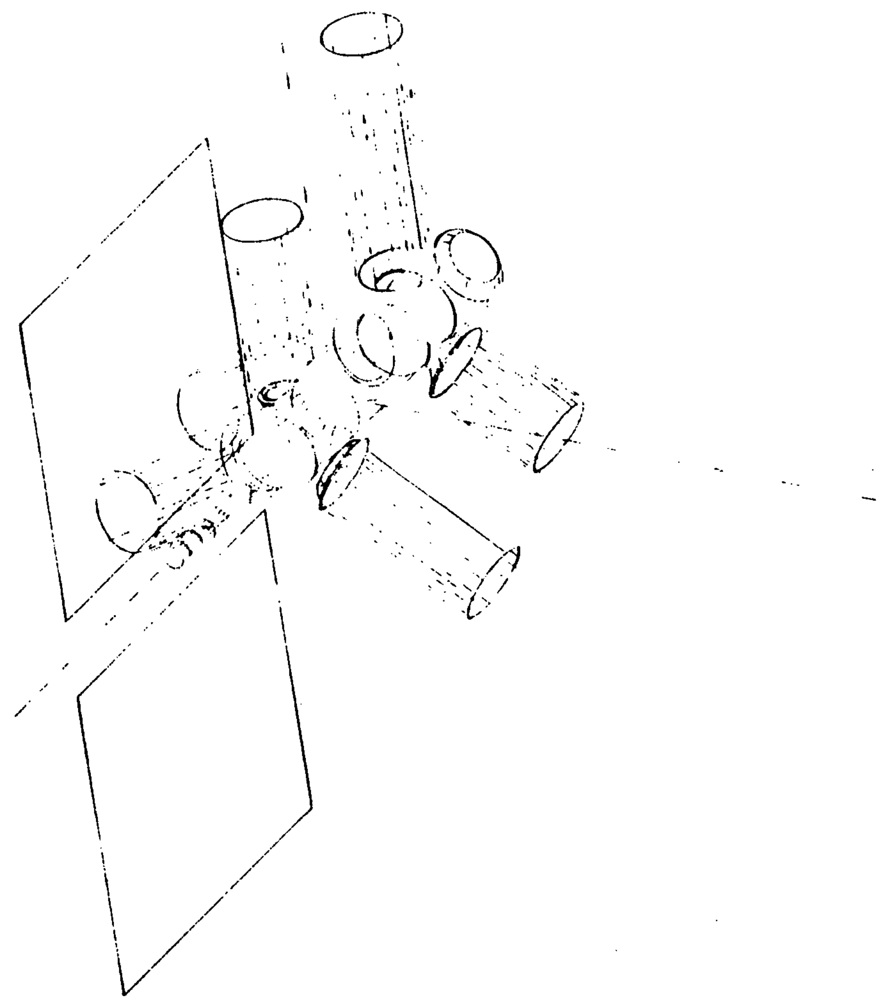


Figure 4.6–53. Graphic Plot–Modular Space Station

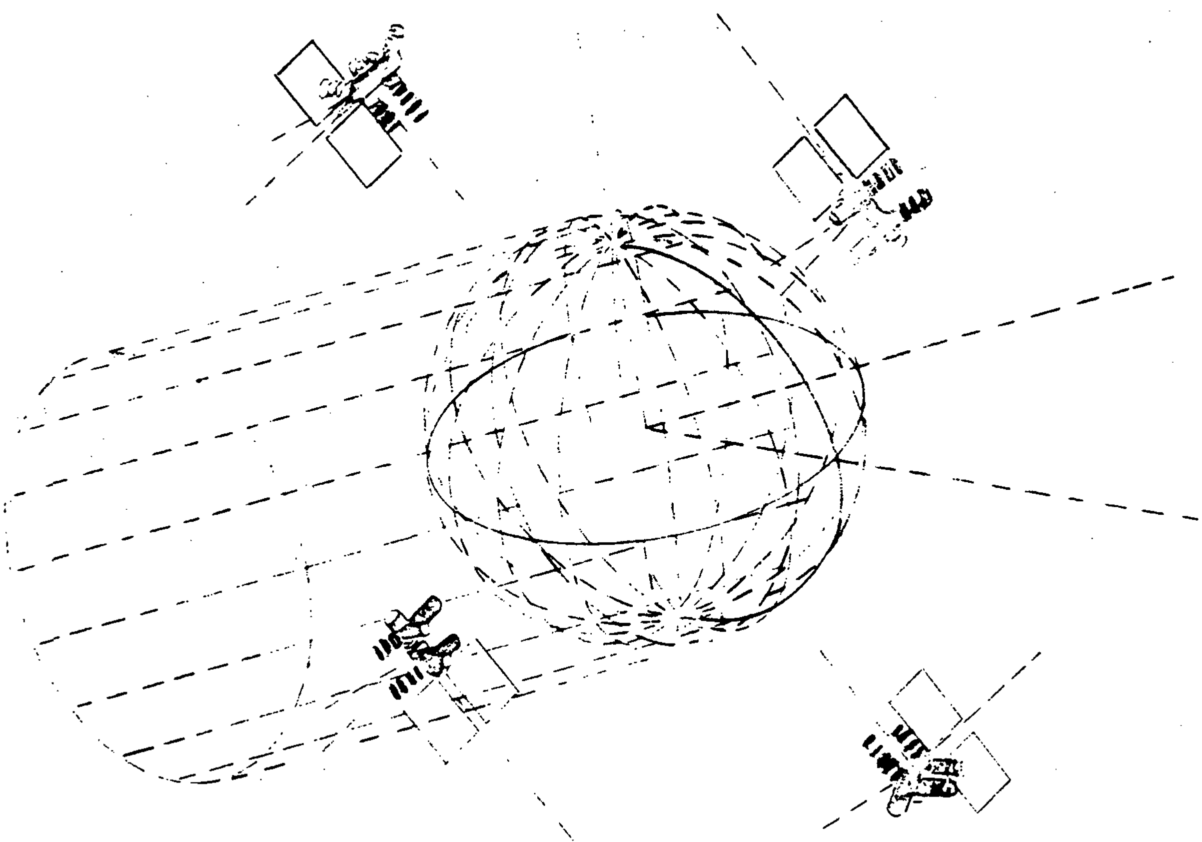


Figure 4.6–54. 3–D View Orbit Set VI, β = 78.5 Deg, Ψ = 0

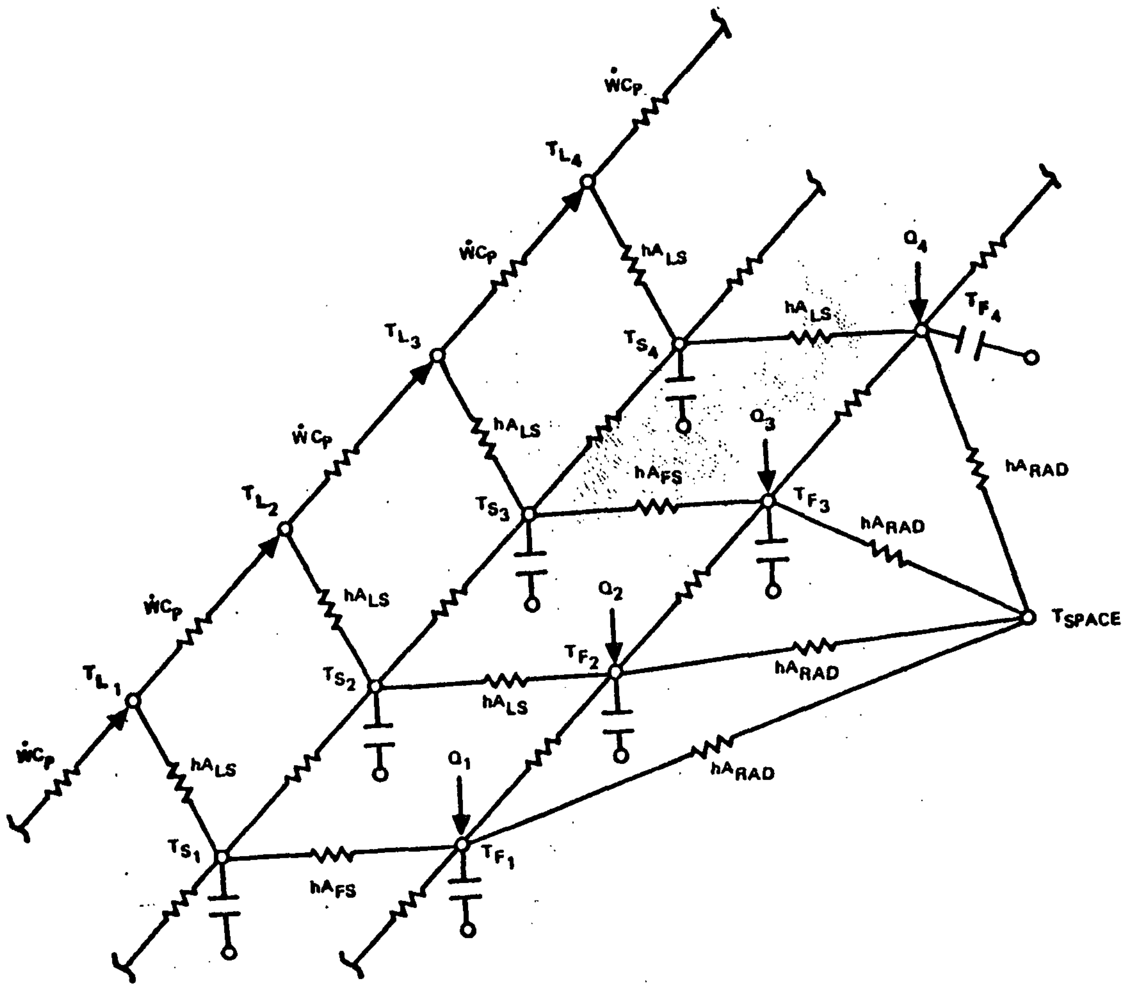
every 0.61 m (2 ft) of tube and fin length in the direction of flow. Parallel tubes were lumped together in the analysis. Part of the thermal network used to simulate a portion of a typical radiator is shown in Figure 4.6-55.

The lumping of capacitances and determination of equivalent conductances for the network was performed according to the techniques and analytical methods discussed for the 10.03 m (33-ft) diameter Station (Reference 4.6-3). In particular the equivalent conductance from Station to fin was determined from the analytical solution of the differential equation for heat transfer on the fin surface. This technique is described in detail in Appendix B of Reference 4.6-3.

4.6.4.6.6. Discussion of Results

Transient temperature calculations were performed for all of the sets of conditions defined in Table 4.6-15. Plots were generated of radiator outlet temperature versus time in orbit. The results were integrated to determine an average outlet temperature for the orbit. Results of the calculations are tabulated in Table 4.6-17. Plots of average radiator outlet temperature for maximum and nominal conditions are given in Figures 4.6-56 through 4.6-58. Some of the important conclusions are listed below:

- Sufficient heat-rejection capacity is available to meet requirements during conditions of peak internal heat loads, orbits with maximum heating, least desirable attitude, and a degraded surface absorptivity, a_s, equal to 0.4. Requirements can be satisfied with or without the presence of docked radial modules.
- The heat-rejection values presented in Table 4.6-16, -17, and -18 can be obtained (or exceeded) while satisfying the minimum and maximum temperature requirements by suitable modulation of the total radiator flow. However, for the selected flow rates, heat rejection capacity is limited to maximum values of 7,980, 13,440, and 20,200 watts (27,200, 45,800, and 69,000 Btu/hr), respectively, for the Power/Subsystems, Crew/Operations, and GPL Modules.
- For the Power/Subsystems and Crew/Operations Modules there is slightly less heat-rejection capacity with docked radial modules. The effect of blockage of view factor to space is more important



Q_1	Heat Absorbed on Surface of Fin. Watts/°C (Btu/Hr°F)
T_{F_1}	Temperature of Fin Node, °C (°F)
T_{S_1}	Temperature of Structural Node, °C (°F).
T_{L_1}	Temperature of Fluid Node, °C (°F).
WC_p	Flow conductance, Watts/°C (Btu/Hr°F)
hA_{LS}	Equivalent conductance from liquid to structure, Watts/°C (Btu/Hr°F)
hA_{FS}	Equivalent Conductance from structure to fin, Watts/°C (Btu/Hr°F)
hA_{RAD}	Equivalent Radiation Conductance from fin to space Watts/°C (Btu/Hr°F)

Figure 4.6–55. Thermal Network Radiator Simulation

Table 4.6-16

POWER/SUBSYSTEM MODULE RADIATOR PERFORMANCE

Set No.	Absorptivity α_s	Orbit Sun (deg) β	Manifold (deg) ϕ	Active Radiator Circuit	Flow Modulated	Docked Modules	Inlet Temperature °C	(°F)	Outlet Temperature °C	(°F)	Heat Rejection Watts	(Btu/hr)	Heat Rejection Requirement Watts	(Btu/hr)
I	0.4	78.5	45	Primary	No	None	42.44	(108)	10.42	(50.7)	8,504	(29,000)	6,158	(21,000)
	0.4	78.5	45	Primary	No	None	32.8	(91)	6.89	(44.4)	6,921	(23,600)	6,158	(21,000)
	0.3	78.5	45	Primary	No	None	42.44	(108)	5.56	(42.0)	9,794	(33,400)	6,158	(21,000)
	0.2	78.5	45	Primary	No	None	42.44	(108)	0.67	(33.2)	11,080	(37,800)	6,158	(21,000)
II	0.4	78.5	11.5	Secondary	No	None	40.8	105.5	12.2	54.0	7,630	(26,000)		
III	0.4	78.5	45	Primary	75 percent Cold Panel	None	42.44	(108)	5.27	(41.5)	9,840	(33,600)	6,158	(21,000)
IV	0.4	0	0	Secondary	No	None	42.44	(108)	11.2	(52.2)	8,260	(28,200)	6,158	(21,000)
V	0.4	78.5	45	Primary	No	Yes	42.44	(108)	12.65	(54.8)	7,888	(26,900)	6,158	(21,000)
		78.5	45	Primary	75 percent Cold Panel	Yes	42.44	(108)	5.0	(41)	9,960	(33,800)	6,158	(21,000)
VI	0.4	78.5	11.5	Secondary	No	Yes	42.44	(108)	12.0	(53.7)	8,035	(27,400)	6,158	(21,000)
VIII	0.4	78.5	11.5	Secondary	No	Yes	42.44	(108)	8.70	(47.7)	8,944	(30,500)	6,158	(21,000)
	0.3	78.5	11.5	Secondary	No	Yes	42.44	(108)	4.64	(40.33)	10,029	(34,200)	6,158	(21,000)
	0.2	78.5	11.5	Secondary	No	Yes	42.44	(108)	0.222	(32.4)	11,202	(38,200)	6,158	(21,000)

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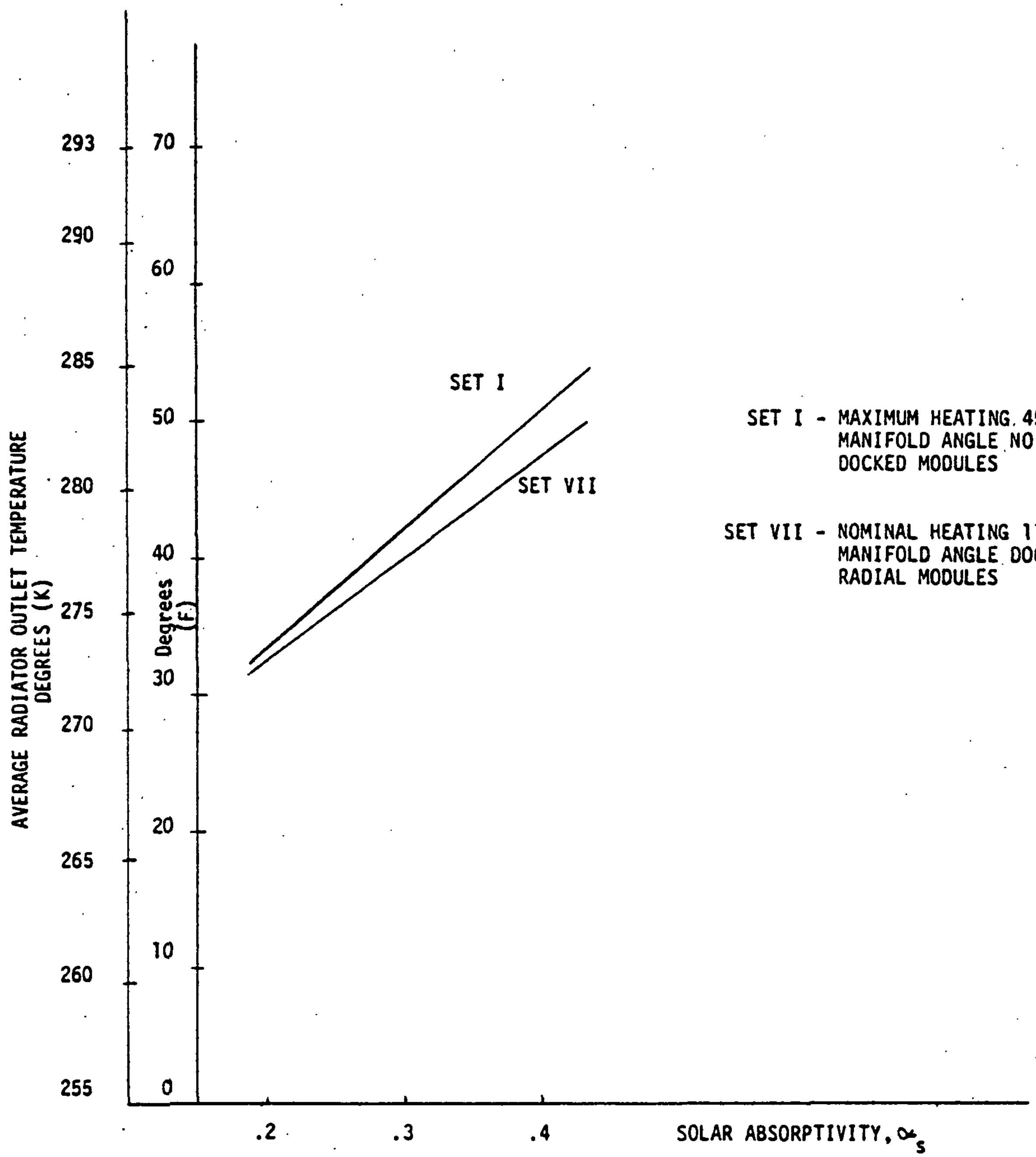

Figure 4.6–56. Power/Subsystem Module Radiator Outlet Temperature

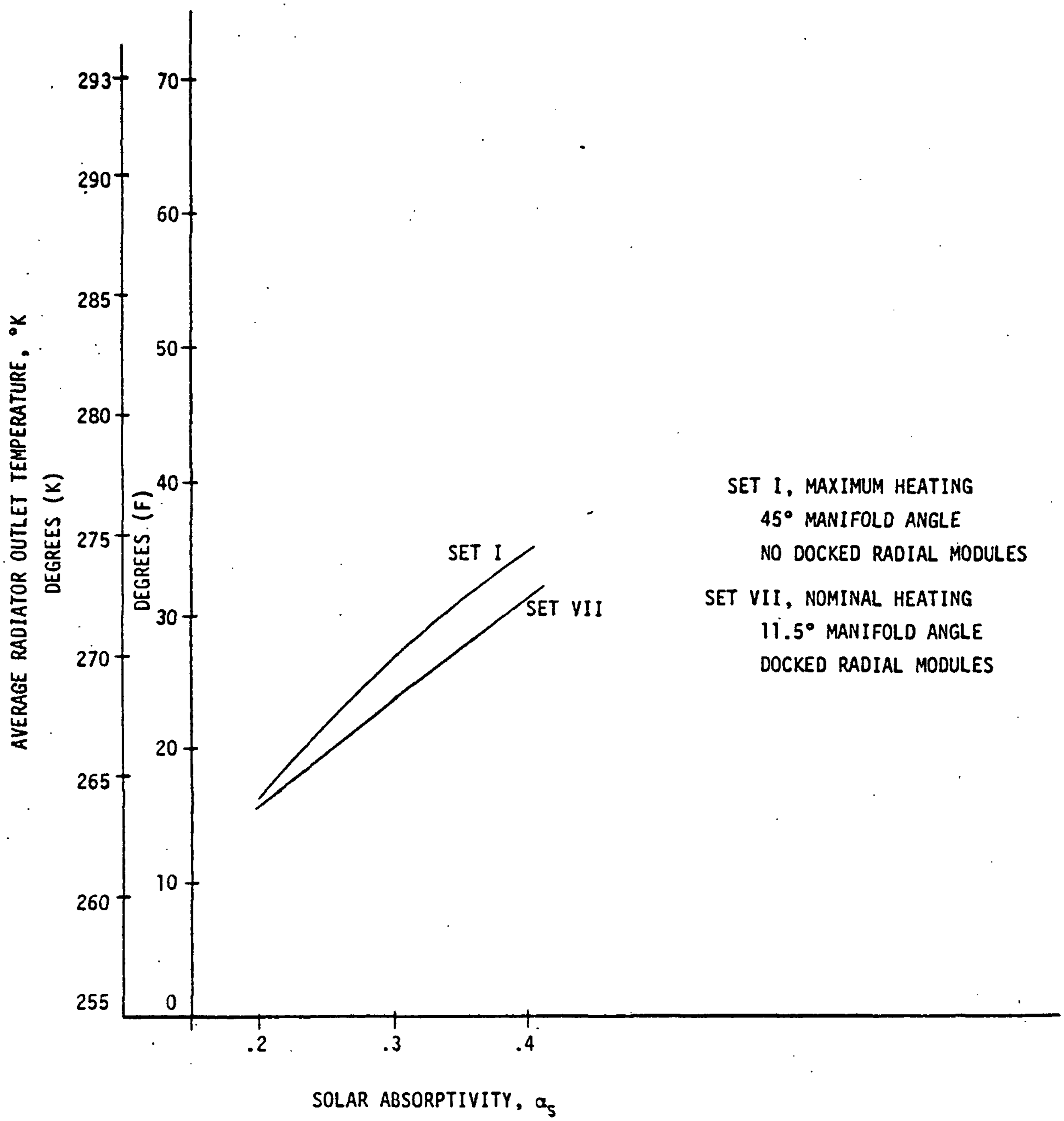

Figure 4.6–57. Crew/Operations Module Radiator Outlet Temperature

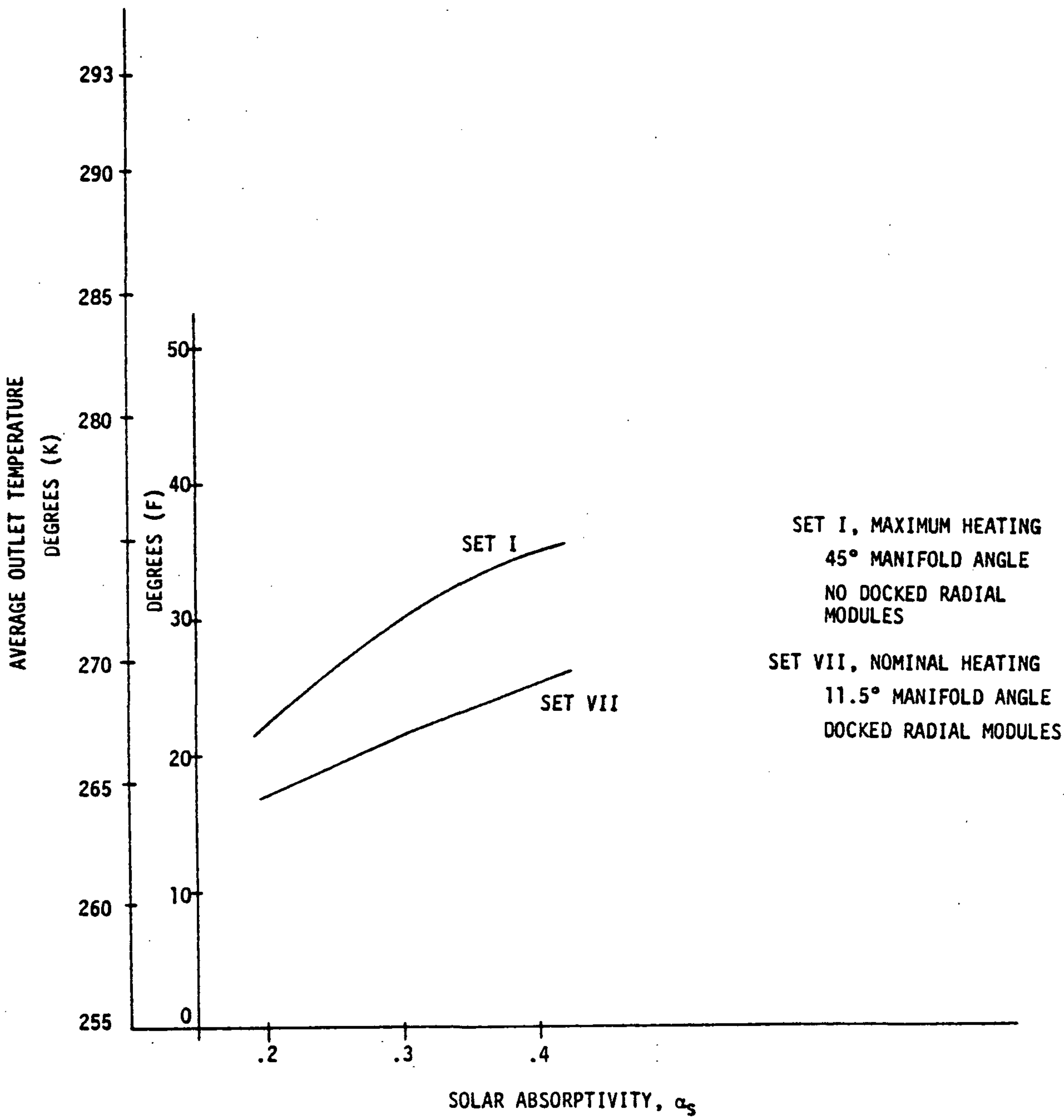

Figure 4.6–58. GPL Module Radiator Outlet Temperature

than shadowing. For the GPL there is slightly greater capacity with radial modules. Here the shadowing of incident albedo and earth IR by docked radial modules has an important influence on the GPL's rejection capacity.

- Some improvement in performance is obtained by modulating radiator flow so that 75 percent of the available flow is directed to the segment with the coldest sink temperature. At β = 78.5 degrees for the Power/Subsystems Module, the improvement amounts to 15.8 to 25.6 percent without and with docked radial modules respectively. For Set V for the assumed inlet temperature, modulation is necessary to maintain the desired outlet temperature of 285°K (54°F). For the Crew/Operations Module, an improvement of approximately 5 percent is realized without and with docked modules at β = 78.5 degrees.
- For β = 78.5 degrees (Set I), satisfactory heat-rejection performance for the Power/Subsystems Module can be obtained with inlet temperatures of either 315°K (108°F) or 308°K (91°F). The outlet temperature using a 305.8°K (91°F) inlet temperature is 280°K (45°F). This outlet temperature should result in slightly longer lifetime for batteries cooled by the Power/Subsystems Module cooling loops.
- Three modes of operation were evaluated at β = 0 degrees for the Crew/Operations Module. Figure 4.6-59 shows the direction of flow in the radiators for conditions when the secondary and primary circuits are active. With the primary circuit active, the direction of flow was reversed 180 degrees at the subsolar point and switched back to its original direction on the opposite side of the orbit on the dark side of the Earth. This mode was investigated for conditions of equal flow in the segments and modulated flow with 75 percent of total flow directed to segment facing away from the sun. The results indicate that there is slightly better performance with the secondary circuit active. Flow reversal and flow modulation thus cannot be used advantageously for the orbit.

FOLDOUT FRAME

Table 4.6-17

CREW/OPERATIONS MODULE RADIATOR PERFORMANCE

Set No.	Absorptivity α_s	Orbit Sun (deg) β	Manifold (deg) ϕ	Active Radiator Circuit	Flow Modulated	Docked Modules	Inlet Temperature °C	Inlet Temperature (°F)	Outlet Temperature °C	Outlet Temperature (°F)	Heat Rejection Watts	Heat Rejection (Btu/hr)	Heat Rejection Requirement Watts	Heat Rejection Requirement (Btu/hr)
I	0.4	78.5	45	Primary	No	None	38.3	(101)	2.3	(36.1)	13,200	(45,000)	10,960	(37,338)
	0.3	78.5	45	Primary	No	None	38.3	(101)	-2.9	(26.7)	15,100	(51,500)	10,960	(37,338)
	0.2	78.5	45	Primary	No	None	38.3	(101)	-8.66	(16.4)	17,250	(58,800)	10,960	(37,338)
II	0.4	78.5	11.5	Secondary	No	None	38.3	(101)	1.4	(34.5)	13,500	(46,200)	10,960	(37,338)
III	0.4	78.5	45	Primary	75 percent Cold Panel	None	38.3	(101)	0.3	(32.6)	13,900	(47,500)	10,960	(37,338)
IV	0.4	0	0	Secondary	No	None	35.2	(95.4)	1.66	(35.0)	12,300	(42,000)	10,960	(37,338)
	0.4	0	90	Primary	No, Flow Rev	None	38.3	(101)	4.4	(40.0)	12,480	(42,500)	10,960	(37,338)
	0.4	0	90	Primary	No, Flow Rev +75 percent Cold Panel	None	38.3	(101)	4.0	(39.2)	12,580	(42,800)	10,960	(37,338)
V	0.4	78.5	45	Primary	No	Yes	38.3	(101)	7.5	(45.5)	11,319	(38,600)	10,960	(37,338)
		78.5	45	Primary	75 percent Cold Panel	Yes	34.3	(93.8)	1.66	(35.0)	12,000	(40,800)	10,960	(37,338)
VI	0.4	78.5	11.5	Secondary	No	Yes	34.6	(94.4)	1.66	(35.0)	12,100	(41,200)	10,960	(37,338)
VII	0.4	78.5	11.5	Secondary	No	Yes	38.3	(101)	-0.4	(31.3)	14,193	(48,400)	10,960	(37,338)
	0.3	78.5	11.5	Secondary	No	Yes	38.3	(101)	-4.56	(23.8)	15,777	(53,800)	10,960	(37,338)
	0.2	78.5	11.5	Secondary	No	Yes	38.3	(101)	-9.0	(15.8)	17,360	(59,200)	10,960	(37,338)

Table 4.6-18

GPS MODULE RADIATOR PERFORMANCE

Set No.	Absorptivity α_s	Orbit Sun (deg) β	Manifold and Roll (deg) ϕ/Ψ	Active Radiator Circuit	Flow Modulated	Docked Modules	Inlet Temperature °C	(°F)	Outlet Temperature °C	(°F)	Heat Rejection Watts	(Btu/hr)	Heat Rejection Requirement Watts	(Btu/hr)
I	0.4	78.5	0/33.5	Primary	No	None	35	(95)	1.66	35.0	18,400	(62,700)	15,350	(52,410)
	0.3	78.5	0/33.5	Primary	No	None	38.3	(101)	-0.9	30.3	21.700	(73,800)	15.350	(52.410)
	0.2	78.5	0/33.5	Primary	No	None	38.3	(101)	-5.38	22.3	24.100	(82.400)	15.350	(52.410)
II														
III														
IV	0.4	0	0/30*	Primary	No	None	29.4	(85)	1.66	35.0	15,400	(52,600)	15,350	(52.410)
	0.4													
V	0.4	78.5	0/33.5	Primary	No	Yes	35.2	(95.4)	1.66	35.0	18,520	(63,200)	15.350	(52.410)
VI	0.4	78.5	0/0	Primary	No	Yes	38.3	(101)	-0.8	30.5	21,600	(73.700)	15.350	(52.410)
VII	0.4	78.5	0/0	Primary	No	Yes	38.3	(101)	-3.6	25.5	23,200	(79,000)	15,350	(52.410)
	0.3	78.5	0/0	Primary	No	Yes	38.3	(101)	-5.9	21.4	24,400	(83,200)	15,350	(52,410)
	0.2	78.5	0/0	Primary	No	Yes	38.3	(101)	-8.3	17.0	25,800	(87,800)	15,350	(52,410)

*Roll angle, Ψ = 30 degrees, defined at subsolar point for solar inertial orientation

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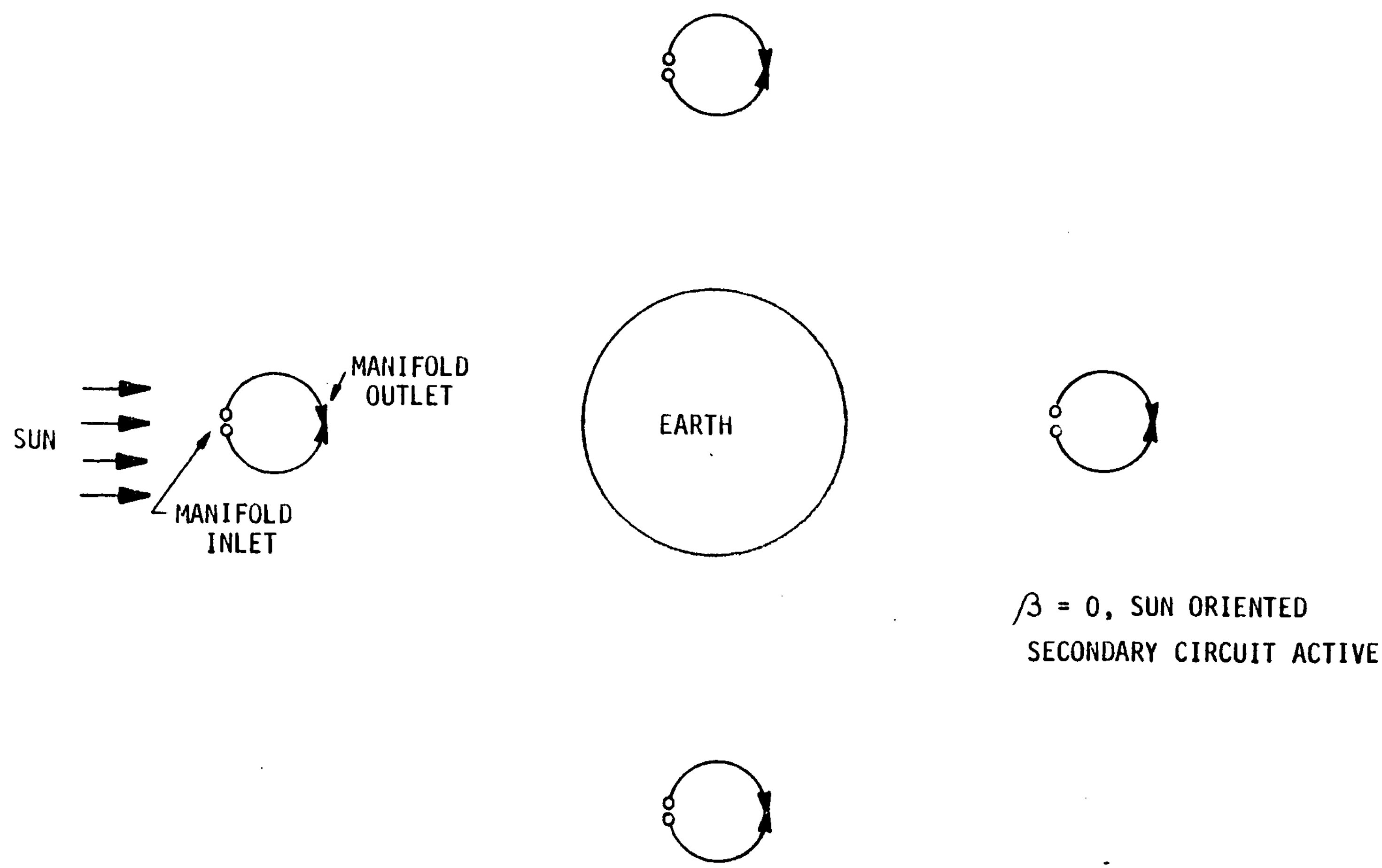

Figure 4.6–59a. Direction of Radiator Flow When Primary and Secondary Circuits are Active

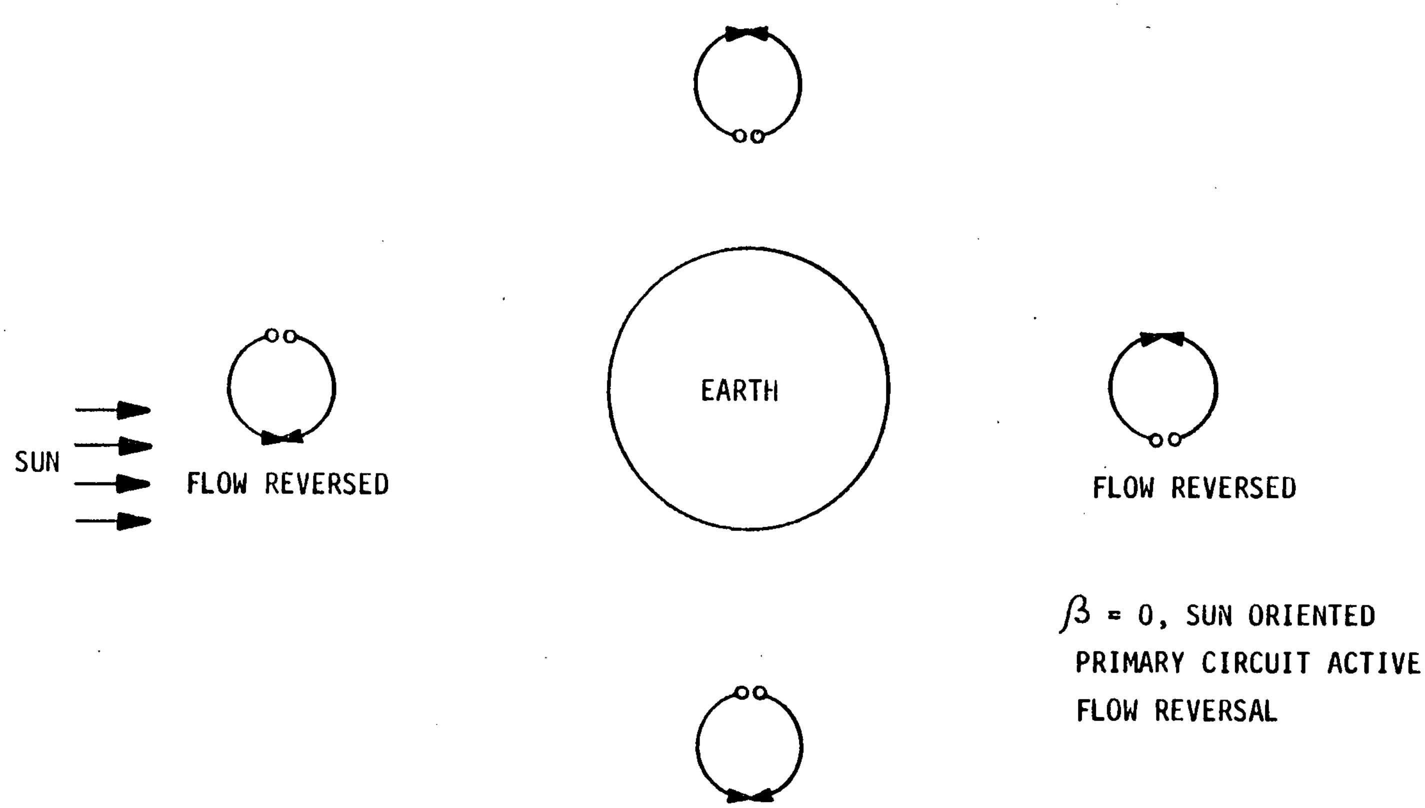

Figure 4.6–59b. Direction of Radiator Flow When Primary and Secondary Circuits are Active

- Use of the MDAC Thermal Analysis Computer Programs resulted in a precise prediction of the station's heat rejection capacities. For vehicles with geometries involving shadowing or blockages of view factors to space, such an analysis would be very difficult without a detailed computer program and simpler analyses could not be relied upon. This fact is illustrated by certain anomalies in expected results. For example, heat rejection capacity for the power module was slightly greater with the vehicle in the more severe roll attitude (see Sets I and II). This anomaly is attributed to reflections of albedo and IR off the docked modules near the active circuit outlet manifold. The relative importance of shadowing versus blockage of view factor to space is also difficult to predict without a detailed analysis. As discussed previously, shadowing is more important than blockage on the GPL while the reverse is true for the Crew/Operations and Power/Subsystems Modules.
- Despite the effects of shadowing and blockage, and differences in relative attitude with respect to the sun and Earth, heat-rejection capacity per unit area for the worst-cases analyzed was found to be equal to approximately 97.8 watts/m^2 (31.0 Btu/hr-ft^2) for all three modules.
- Heat rejection requirements in Tables 4.6-17, -18, -19 correspond to peak internal loads. There is considerable margin between average requirements and available capacity for the Crew/Operations and GPL Modules. Thus, a margin for growth in heating requirements is provided.

4.6.5 References

4.6-1. "MSFC-DRL-160, Line Item 8, Space Station Definition, Volume V, Subsystems, Book 2, Crew Systems," Contract NAS8-25140.

4.6-2. "MSFC-DRL-160, Line Item 13, Space Station Definition, Volume V, Subsystems, Book 2, Crew Systems," Contract NAS-25140.

4.6-3. "MSFC-DRL-231, Line Item 8, Space Station Report on Selected Update Tasks for Baseline Space Station, Volume II, Thermal Control," Contract NAS8-25140.

4.6-4. "ECLS Radiation Configuration Study," MOL DD Form 1423 Line Item 69, DAC 57137, 15 August 1966, page 5.

4.6-5. NASA TM X-53865, "Natural Environment Criteria for the NASA Space Station Program (Second Edition), August 20, 1970.

4.6-6. NASA TM X-53957, "Space Environment Criteria Guidelines for Use in Space Vehicle Development (1969 Revision), October 17, 1969.

4.7 GUIDANCE, NAVIGATION AND CONTROL SUBSYSTEM

4.7.1 Summary

The Guidance, Navigation and, Control (GNC) subsystem provides the following functions:

A. Stabilization and attitude control
B. Attitude and rate data for experiment support operations
C. Navigation
D. Orbit altitude control

The GNC subsystem senses, computes, and receives the commands and data for these functions and the Propulsion Subsystem and a part of the GNC subsystem (the control moment gyros) generates the actuation forces and torques for executing these functions. The sensing and computation of the Space Station attitude and angular rates are provided through the control functions while the navigation data is provided by the ground tracking network. The GNC subsystem must provide these four functions previously identified for a period of 10 years throughout the ISS buildup phase which exhibits a large variation in the physical characteristics of the Modular Space Station.

The three major trade studies performed for the GNC subsystems are: (1) primary orientation, (2) control actuation selection and sizing, and (3) pointing and experiment accommodation capability determination. The first two trade studies are presented in subsection 4.7.4, Design Analysis and Trade Studies, while the third one is presented in subsection 4.7.2, Requirements.

The selected design for the GNC subsystem is shown in Figure 4.7-1. The particular location of the GNC equipments in the various modules is indicated by the dashed lines. The GNC subsystem has four sensors all located in the Power/Subsystems Module which provide an all-attitude capability. Two gyro triads provide the primary attitude sensors and are used in all orientations of the Space Station to provide continuous attitude and rate data for attitude control and to support. Optical sensors are required to realign the attitude reference periodically to compensate for gyro drift. In Earth-centered

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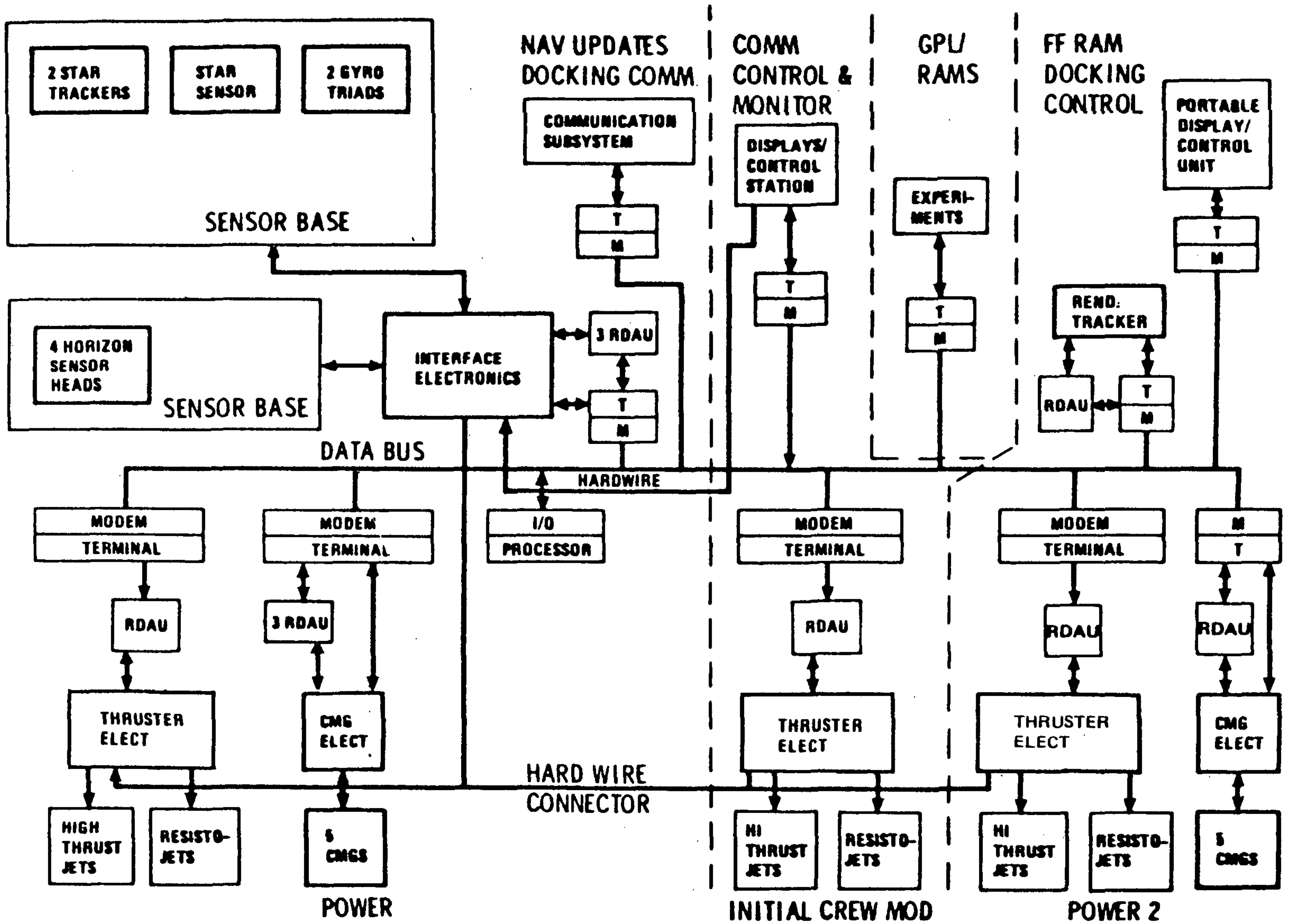

Figure 4.7-1. GNC Subsystem

orientations the star sensor, which is located on the side of the power mode facing away from the earth, is used to update the attitude information in trimmed orientations.

The horizon sensor which is located on the Earth facing side of the Power/ Subsystems Module provides a backup capability for the Earth-centered orientation but is limited to smaller trim angles than the Star sensor. The horizon sensors primary function is to provide wide angle capability which is used for initial acquisition of the Earth-centered orientation.

In all inertial and pseudo inertial orientations the gimbaled star tracker provides the drift-free inertial measurements to update the Space Station attitude reference system. Because ground tracking is used for navigation data, both inertial and Earth-centered attitude reference information is available in all orientations. The interface electronics performs the required conversions to standardize the sensor outputs for the data processing. The attitude reference trade studies for the sensor selection were performed by Honeywell Corporation.

The primary control actuation is provided by four control moment gyros (CMG's) which are installed in the Power/Subsystems Module after the first manning. A fifth CMG is in a standby mode. The low thrust resistojets are used for the orbit keeping impulse requirements and normal CMG desaturation. The high thrust jets are used to combat the docking disturbances and to provide a backup capability to the resistojets. During the unmanned phase, the high thrust jets provide the control torques for attitude control and stabilization of the Modular Space Station.

The Crew/Operations Module contains a second set of high- and low-thrust jets. The interface of the thrustor electronics is made through the data bus terminals of the Data Management Subsystem (DMS). The Crew/ Operations Module also contains the display/control station which has a hard-wire connection from the manual controller to the high thrust jets through the interface electronics.

The GNC Subsystem uses the DMS computer for its computational requirements. Station attitude and rate reference data is supplied to the dedicated experiment computer in the GPL for user support.

The GNC subsystem which includes sensors, interface electronics, reaction control electronics, CMG control electronics and CMG's has a weight of 1037 Kg (2, 283 lb), a volume of 4. 13 m^3 (141. 5 cu ft) and an operating power of 360 W.

The second Power/Subsystems Module shown in Figure 4. 7-1 is used for the Growth Space Station (GSS). This module contains the additional momentum storage capacity required, which is identical to that of the first Power/ Subsystems Module. The thrustors used on the second module are also identical to those on the first module.

The primary orientation of the Space Station is trimmed horizontal, which is illustrated in Figure 4. 7-2. To implement the trimmed horizontal orientation, the Station is initially aligned so that the roll or X-axis is aligned to the orbital velocity vector, the yaw or Z-axis towards the Earth aligned to the vertical and the pitch or Y-axis is aligned with the normal to the orbit plane. This orientation is the horizontal orientation. The trimmed horizontal orientation is obtained by an angular deflection about the pitch axis so that the bias gravity-gradient and aerodynamic torques about the pitch axis are zero. This orientation minimizes the attitude control propellant and is a near-minimum orbital drag configuration. The horizontal orientation can also be trimmed with an angular deflection about the yaw or Z-axis. This case of trimming produces a near-maximum drag configuration and can be implemented subject to experiment program requirements. This orientation will be referred to as Z-axis, trimmed horizontal. The GNC subsystem has an all-attitude capability and can accommodate any inertial orientation subject to experiment program requirements for an indefinite time period. The restriction to maintaining indefinite operation in the alternate attitudes is the required propellant expenditures and potential contamination associated with the use of the high thrust jets. The attitude control propellant and equivalent impulse penalities associated with several Space Station orientations are given by Table 4. 7-1.

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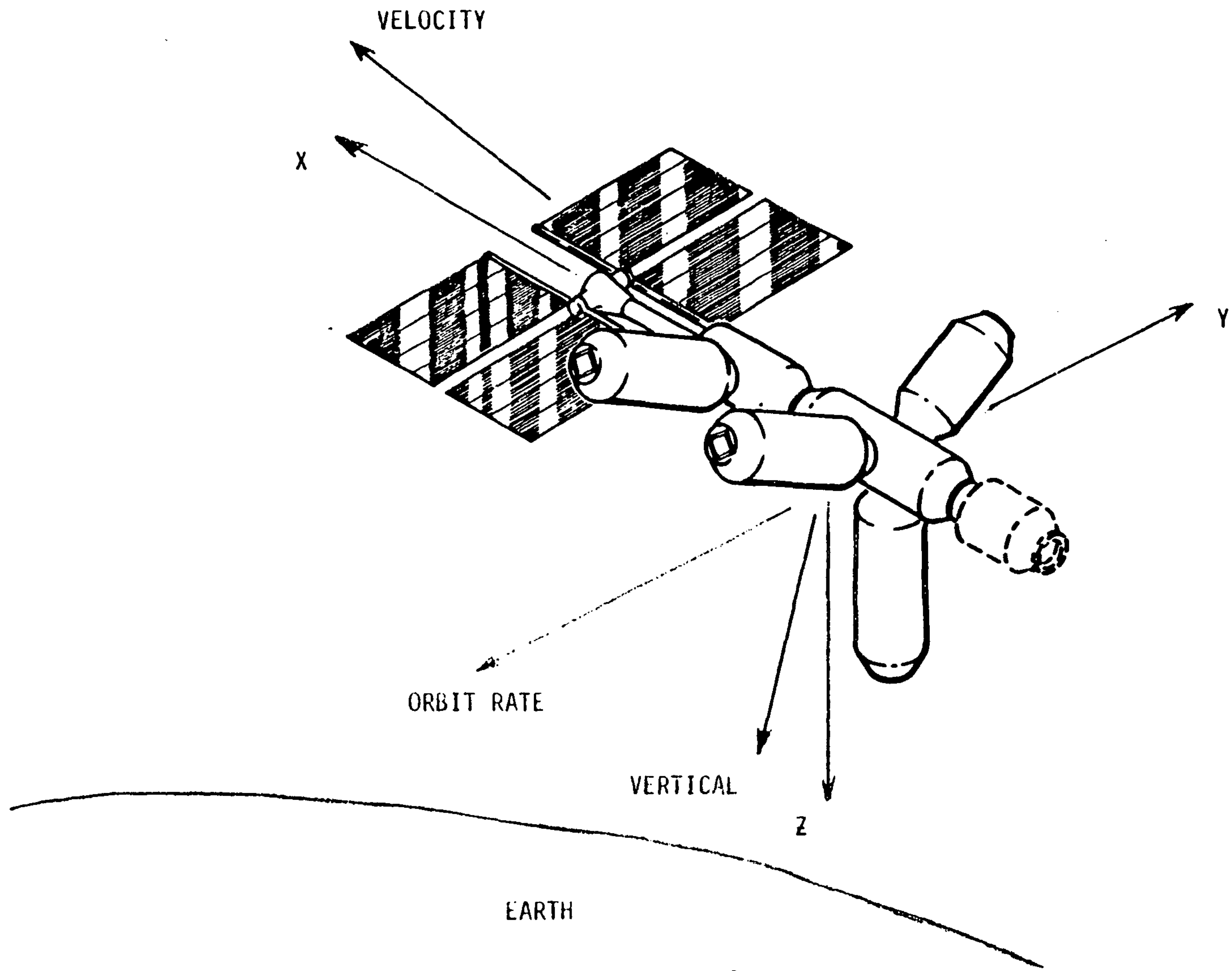

Figure 4.7-2. Trimmed Horizontal Orientation

Table 4.7-1

ISS ATTITUDE CONTROL PROPELLANT PENALTIES

Orientation	Attitude Control Requirements	
	Impulse (n-m-sec/orbit)	Propellant* (kg/day)
Trimmed Horizontal (Long term)	315	0.23
Untrimmed Horizontal	24,100	17.2
Worst Case Earth-Centered	50,000	204.0
Inertial		
Average	19,500	18.6
Worst case	28,700	118.0

*Propellant I_{sp} = 180

In the long-term orientation, the orbit keeping and bias attitude control requirements are provided by the resistojet system. The bias attitude control, or CMG desaturation, is performed concurrently with the orbit-keeping function. The biowaste resistojet capability for the six-man crew is 6.35 kg/day which is sufficient for the long-term orientation.

The control moment gyro (CMG's) are sized for the largest cyclic impulse of the inertial orientation. This orientation imposes the largest momentum storage requirements and therefore the CMG's will have sufficient capacity for all other orientations. The selected configuration of four CMG's provides up to 16,280 n-m-sec (12,000 lb-ft-sec) momentum storage in the pitch and yaw axes and 8,140 n-m-sec (6,000 lb-ft-sec) in the roll axis. The CMG's are capable of maneuvering the Station at a minimum of 0.05 deg/sec in pitch. The CMG selection and the CMG configuration trade study were performed by the Bendix Corporation.

The functional requirements tree for the GNC subsystem is shown in Figure 4.7-3. The five basic GNC functions are shown as attitude determination, navigation, attitude control, experiment support, and checkout and calibration. In the attitude determination function, four sensor data processing modes are identified along with the acquisition logic. Under the attitude control function, both CMG and high thrust control functions are specified. Inertial and Earth-centered orientations as well as rate stabilization and maneuver modes of operation are identified for high-thrust and CMG control.

The attitude and rate performance requirements for the GNC subsystem are given as the following:

Attitude control (all-attitude)	±0.25 deg
Rate control (stability)	±0.005 deg/sec

This performance is adequate for all experiments and is readily achieved with developed hardware.

The trimmed horizontal orientation provides good Earth-viewing and celestial-viewing capabilities for the attached experiments. Examination of the Earth-centered FPE's (earth surveys, communication/navigation, and physics) as presently accommodated indicate they have no preference for trimmed or not-trimmed horizontal orientation leaving the selection to Station optimization. Celestial experiments which are performed on the Station are subject to common viewing limits in any Earth-centered orientations. Only the inertial orientation provides unconstrained view to a particular celestial target. Most station-based experiments require short viewing times and are readily accommodated in the Earth-centered orientations.

A matrix of the key trade studies for the GNC subsystem is given in Table 4.7-2. The factors influencing the selection for the three major trades is also given. The attitude reference trade study was performed in the 33-ft Space Station study and were documented in Reference 4.7-1. The selected assembly levels are indicated in the table.

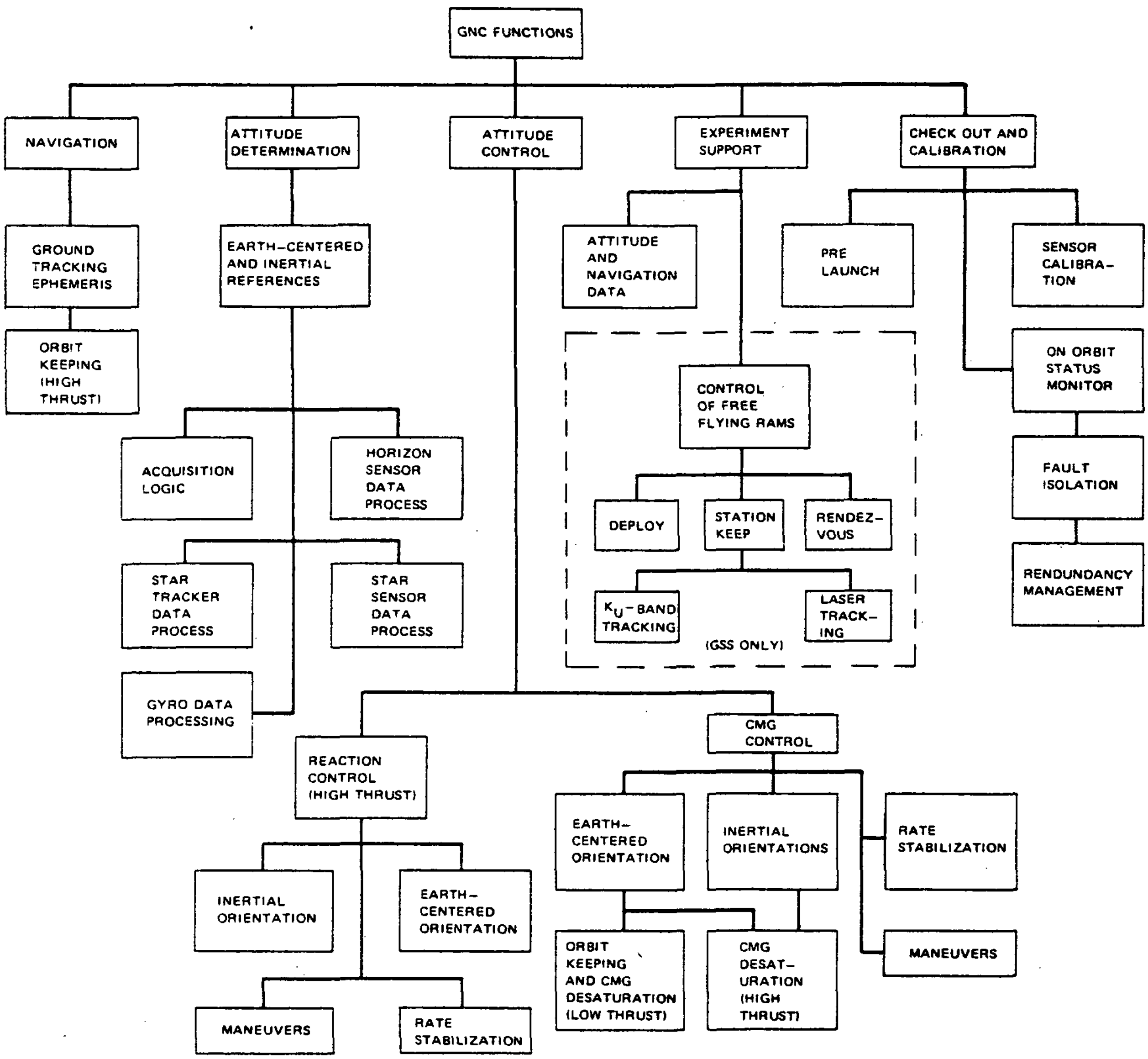

Figure 4.7-3. Functional Requirements Tree

Table 4.7-2

GNC KEY TRADE STUDIES

	1	2	3	4	5	Rationale
Orientation	$\boxed{\text{Trimmed horizontal}}$	Horizontal	Inertial	Others		Minimizes propellant requirements and provides good accommodation for Earth-centered experiments
Attitude reference	Star trackers Horizon sensor gyros	Electronic star tracker Horizon sensor gyros	ESG's Star sensor Horizon sensor gyros	$\boxed{\text{Star sensor Star trackers Horizon sensor gyros}}$	Star sensor Horizon sensor gyros	Most responsive to orientation requirements at medium cost and development risk.
Control actuation	High thrust	ATM CMG's	$\boxed{\text{Improved ATM CMG's}}$	Advanced CMG's		Minimizes actuation costs.

$\boxed{}$ Modular Selection

4.7.2 Requirements

The function of the GNC subsystem is to control the orientation of the Space Station during all phases of orbital operation. This capability starts after separation of the Power/Subsystems Module from the Shuttle-Orbiter and continues throughout the ISS buildup phase. During the operational phase of the ISS, the GNC Subsystem provides control capability for the Space Station cluster which may include as many as five modules (RAM's and Logistics Modules) docked to the ISS. To support the experiment program, the GNC Subsystem must provide Earth-centered, inertial and solar orientation capability as well as vehicle attitude, rate, and navigation information.

The GNC requirements are derived from three basic sources:

1. PRD including program guidelines and constraints
2. Analysis of experiment requirements
3. Analysis of vehicle design features and operations

Table 4.7-3 lists several of the key GNC requirements and the Program requirements having a significant effect on the design selection. The requirement that total program cost be a primary consideration (subsection 3.1.1.2) was the driving factor that led to the utilization of CMG's as the primary actuator. Considerable savings to the total program cost is realized by this selection as discussed in subsection 4.7.4.3, Control Actuation Trade Study. This same requirement (subsection 3.1.1.2) also specifies that primary emphasis is on minimum cost to the IOC which led to the selection of "off-the-shelf" hardware to minimize development cost.

Another key factor in the selection of a GNC design approach is the requirement to use automated systems to the greatest extent possible to free the crew from routine operation (subsection 3.2.1.1.4). The selected GNC Subsystem satisfies this requirement in that all functions are automated and require a minimum of crew time for mode selection and replacement or repair of failed equipment.

The functions of the GNC subsystem are attitude reference determination, attitude control, position determination, and orbit keeping. The performance

Table 4.7-3

KEY GNC AND PROGRAM REQUIREMENTS

Subsection No.	Item
3.1.1.2	Total cost of the program is a primary consideration. Primary emphasis is on minimum cost to the IOC.
3.2.1.1.5	The Space Station shall be designed to have a minimum operational life of 10 years with resupply. The Space Station design shall provide for damage control and repair.
3.2.1.1.4	The crew shall be freed of routine operations to the greatest practical extent by use of automated systems.
3.2.1.1.8	All subsystems for new program elements shall contain operational instrumentation consistent with system operation and fault isolation and repair to the lowest replaceable unit.
3.7.1.4.12	As a goal, no orientation restrictions will be imposed by subsystems, i.e., electrical power, thermal control, communications.
3.1.2.2	The ISS shall be capable of use in an orbit of 55-deg inclination at an attitude of between 445 km to 500 km (240 to 270 nmi).

requirements for these functions are derived primarily from the evaluation of the experiment pointing, stabilization, and navigation requirements.

The attitude control and attitude determination performance requirements are derived from the pointing and stabilization requirements of the attached and integral experiments shown in Table 4.7-4. Pointing requirements derived from Reference 4.7-2 generally specify pointing requirements for the experiment sensors themselves although some experiment groups do not specify Space Station requirements as shown in the table. Reference 4.7-2 which was used for the Modular Space Station study does not specify navigation requirements. It is assumed that the ±1.0 nmi requirements specified in Reference 4.7-3 is still valid.

Table 4.7-4
EXPERIMENT REQUIREMENTS

Mode	Sub Group	Title	Orientation Viewing Req'mts	Pointing: Experiment	Pointing: Station	Stability: Experiment	Stability: Station	Hold Time	G Level	Comments
A	A4-A, B	Large UV telescopes	Inertial	1 $\overset{\frown}{sec}$	0.5 deg	0.5 $\overset{\frown}{sec}$	--	240 min	10^{-3}	Gimbals provided
A	A4-C	Small UV telescopes	Inertial	0.5 deg	--	0.6 $\overset{\frown}{min}$	0.3 deg	15 min	10^{-3}	Gimbals provided
A	A5-B	High energy experiments	Inertial	1 $\overset{\frown}{sec}$	--	1 $\overset{\frown}{sec}$	--	--	10^{-3}	
A	A-6	IR astronomy	Inertial	1 $\overset{\frown}{sec}$	--	1 $\overset{\frown}{sec}$	--	4 Hrs	--	
I	P1-A	Space physics	Velocity vector	2 $\overset{\frown}{min}$	--	$0.1 \frac{deg}{sec}$	--	--	--	
I	P1-BC	Cometary and meteoroid SCI	Away from Earth	2 deg	--	$0.05 \frac{deg}{sec}$	--	--	--	
I	P1-E	Small astronomy	Inertial	0.5 deg	--	$0.01 \frac{deg}{sec}$	--	--	--	Gimbals provided
A	P2-A	Wake measurements	View ±90 deg of velocity	0.5 deg	0.5 deg (1)	0.5 $\overset{\frown}{sec}$	--	--	--	(1) Flush mounted Do not gimbal
A	P2-C, D, E	Plasma and particle physics	Geomagnetic	1.0 deg	(2)	1.0 deg	--	--	--	(2) Maintain within ±45 deg gimbals
A	P-3	Cosmic ray physics	Zenith hemisphere	--	(3)	--	--	--	--	(3) Knowledge of attitude to ±1.0 deg
I	P-4	Physics and chemistry	Velocity vector and inertial	1.0 deg	--	--	--	--	10^{-4}	Boom experiments
A	ES-1	Earth surveys	±45 Deg from nadir	0.5 deg	--	$0.01 \frac{deg}{sec}$	--	--	--	Gimbals provided
A	C/N-1	Com/Nav experiments	Earth and synchronous satellites	0.01 deg	--	0.01 deg	--	10 min	--	Gimbals provided
I	T1-A	Contamination experiments	Solar	0.5 deg	--	$0.05 \frac{deg}{sec}$	--	--	--	
I	T-4C	Adv spacecraft system test	Solar and anti-solar	0.1 deg	--	0.005 $\overset{\frown}{sec}$	--	--	--	
	MS-1	Materials Science	--	--	--	--	--	--	10^{-3}	

The key experiment requirements from which the Space Station pointing and stabilization are derived are indicated by the numbers enclosed in the bold squares. The other experiments are either so stringent that it is not feasible to accommodate these with the Space Station or are relaxed enough so that they lie within those identified. These two experiments (A4-A and P2-A), specifically identify a Space Station pointing requirement of ±0.5 degree. With these experiments being in attached modules, a large part of the allowable pointing error must be allocated for alignment between the GNC reference axes and the experiment reference axes.

The Space Station pointing requirement defined during the option period (Reference 4.7-4) was ±0.25 degree which was based on similar experiment requirements. With these requirements the identified experiments would have to be aligned to within ±0.434 degree of the GNC reference in order to meet the total pointing requirements. Reduction of the GNC pointing error below ±0.25 degree does not significantly change the alignment requirements when these errors are statistically summed. For example, if the GNC pointing error were to be reduced by an order of magnitude (±0.025 degree), the increase in allowable misalignment would be ±0.05 degree. This small relaxation in alignment requirements has an insignificant impact on the alignment technique utilized. However, the reduction in GNC pointing errors could significantly affect the design approach selected, whereas the ±0.25 degree requirement can be achieved with relative ease.

The ±0.25 degree requirement was also used in the experiment accommodation study to define the mode of accommodation (integral, attached, free-flying) and to define gimbal requirements for the experiments.

For these reasons, the previously defined pointing requirement of ±0.25 degree has been maintained for the Modular Space Station. The stability requirement of ±0.005 degree-per-second was also derived during the option period and used in the experiment accommodation study. Since this performance is adequate for all experiments and is readily achieved with developed hardware, it is also adopted as a requirement for the Modular Space Station.

Only one experiment (P-3) identifies an attitude reference requirement. However, reference 4.7-2 does mention use of the Space Station attitude reference to initialize a number of the experiment reference systems although performance requirements are not specified. The ±1.0 degree required for P-3 will be adequate for experiment reference initialization due to the large acquisition capability of most optical sensors. At the same time, the 1.0 degree error is small enough so that the probability of acquiring the wrong star is negligible. In this instance, as in the pointing performance, the major error source in the knowledge of the experiment attitude is the relative misalignment between the experiment and GNC reference axes. Since inertial reference systems having accuracies of less than 0.1 degree are readily available, nearly all of the ±1.0 degree allowable error can be allocated for alignment.

The remaining functional and performance requirements are derived from an assessment of operational considerations. To support the variety of operations required of the Space Station Program, the GNC Subsystem must provide attitude control for several orientations of the Space Station. These orientations are defined below in terms of the Space Station body axes: (Figure 4.7-4)

A. Horizontal—The Z-axis is aligned with the vertical, the X-axis is aligned with the orbital velocity vector, and the negative Y-axis is parallel to the orbital rate vector.

B. X-POP—Perpendicular to Orbit Plane - The X-axis is parallel to the orbital rate vector and angular rate about X-axis is maintained at nominally zero degree/sec.

C. X-POP/OR—Perpendicular to Orbit Plane/Orbit Rate - The X-axis is parallel to the orbit rate vector and the Z-axis is aligned with the vertical. The vehicle rotates about the X-axis at orbit rate.

D. Inertial— Fixed vehicle attitude with respect to the celestial sphere.

In addition to these orientations, the GNC Subsystem shall provide a rate stabilization capability in which the body rates are controlled and the attitude is allowed to drift and seek a minimum torque orientation.

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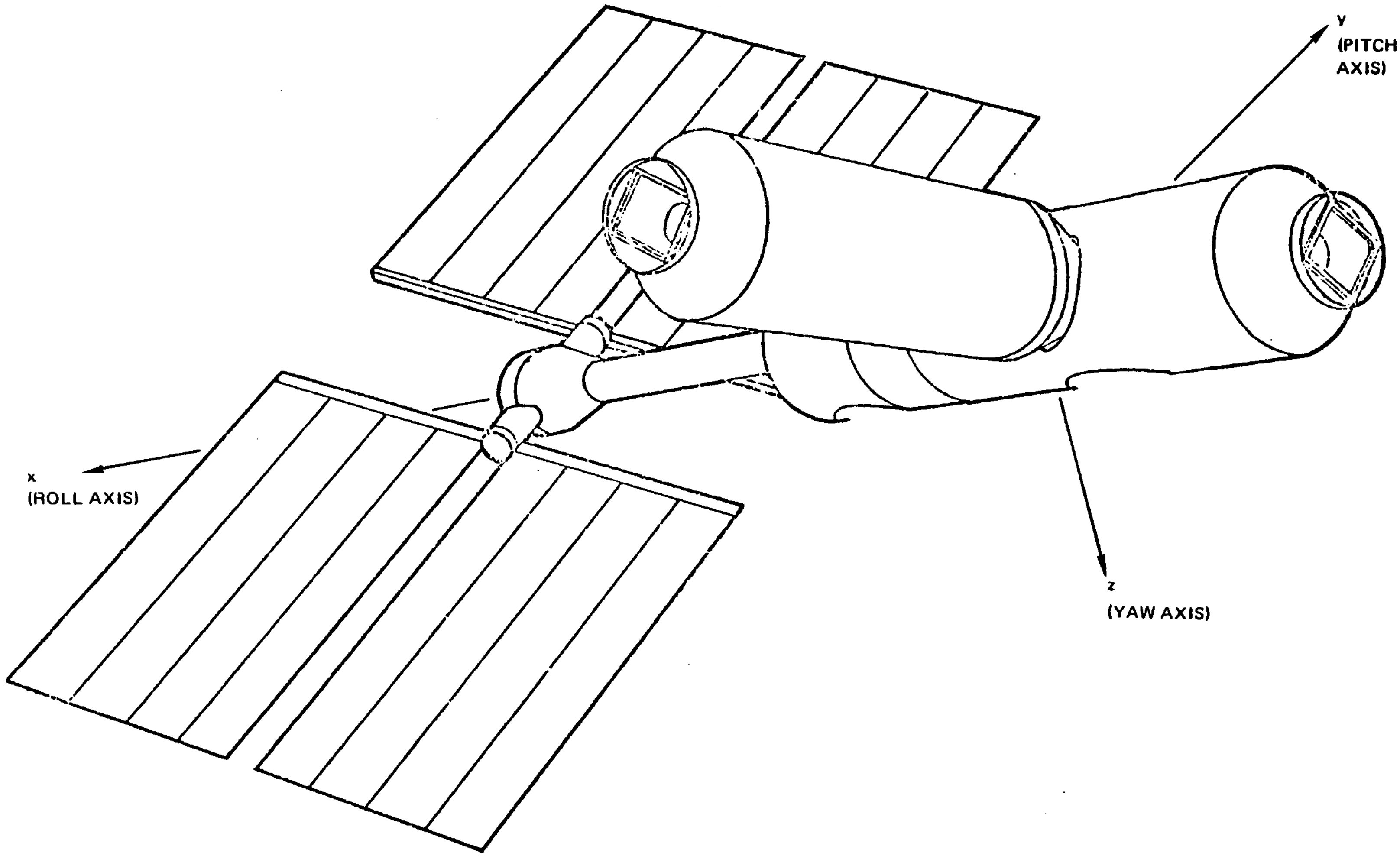


Figure 4.7-4. Space Station Body Axis Definition

On a noninterference basis with the experiment, the GNC Subsystem shall provide an attitude trim capability for the Earth-centered orientations. While in the horizontal orientation, the GNC Subsystem shall maintain the pitch axis in a trimmed attitude which minimizes control propellant consumption and in the POP/OR orientation the roll axis shall be maintained in a trimmed attitude which minimizes the attitude control propellant consumption. consumption.

The functional and performance requirements are summarized in Table 4.7-5.

Table 4.7-5
FUNCTIONAL AND PERFORMANCE REQUIREMENTS

Function	Performance
Orientation	
Primary	Trimmed horizontal
Others	All attitude
Attitude control (All attitude)	±0.25 deg
Rate control (Stability)	±0.005 deg/sec
Attitude reference data	±0.1 deg
Rate reference data	Compatible with rate control
Navigation	±0.1 nmi

The trimmed horizontal orientation identified in Table 4. 7-5 aligns the geometric X and Z-axes within the orbit plane such that the bias torque about the Y-axis is zero and the X-axis is near the orbital velocity vector.

The propellant requirements and the momentum storage requirements for the the ISS configuration are given by Table 4. 7-6 with the supporting analysis given in Section 4. 7. 4.

The primary Propulsion System, which is the Biowaste System, has the capability to provide 570 kg/90 days. This will satisfy all the orientations but those inertial orientations with large bias impulse requirements.

The four operational CMGs have a maximum capability of 16, 280 n-m-sec (12, 000 lb-ft-sec) at 12, 000 rpm and a nominal capacity of 10, 840 n-m-sec (8, 000 lb-ft-sec) at 8, 000 rpm.

Table 4. 7-6

ISS PROPELLANT AND MOMENTUM STORAGE REQUIREMENTS

	Orientation			
Function	Trimmed Horizontal*	X-POP/OR	X-POP	Inertial
Propellant** (kg/90 days) (I_{sp} = 180 sec, 1981 atmosphere)	230	260	290	1, 850
CMG capacity	9, 150	11, 400	8, 550	9, 700

*Selected long-term orientation

**Orbit Keeping plus attitude control propellant

4.7.3 Selected Subsystem Design

The GNC subsystem provides the Modular Space Station with the capability to maneuver and hold any orientation to support the orbital and experiment operations in the presence of the orbital disturbance environment.

The GNC subsystem is designed to maximize the operational effectiveness of the Modular Space Station throughout the buildup phase with varying Space Station physical characteristics while constraining the required propellant and electrical power resources to a reasonable level.

The primary orientation of the Modular Space Station is trimmed horizontal, which is an Earth-centered orientation. This orientation aligns the geometrical X and Z axes in the orbit plane so that the bias torque on the Y-axis is zero with the X-axis near the velocity vector. Other orientations, such as inertial, may be imposed by the experiment operations. The Station can accommodate any inertial orientation for an indefinite time period subject to propellant expenditure and potential contamination associated with use of the high-thrust system. Normal attitude control is performed by control moment gyros (CMG's) which provide sufficient capacity for the worst-case orientation.

Attitude and rate information for attitude control and experiment support is determined during both Earth-centered and inertial orientation.

A description of the GNC subsystem and its major assemblies is given in Subsection 4.7.3.1 and its subparagraphs. Definition of the attitude reference system and its software requirements was provided by Honeywell Incorporated, the subcontractor for the attitude reference assembly group. Definition of the CMG system and its software requirements was provided by the Bendix Corporation Navigation and Control Division, the subcontractor for the CMG assembly group. Additional assistance was provided by numerous unfunded subcontractors.

4.7.3.1 Subsystem Description

A schematic diagram of the GNC subsystem is shown in Figure 4.7-5. The attitude gyros, star trackers, star sensors, horizon sensors, and CMG's

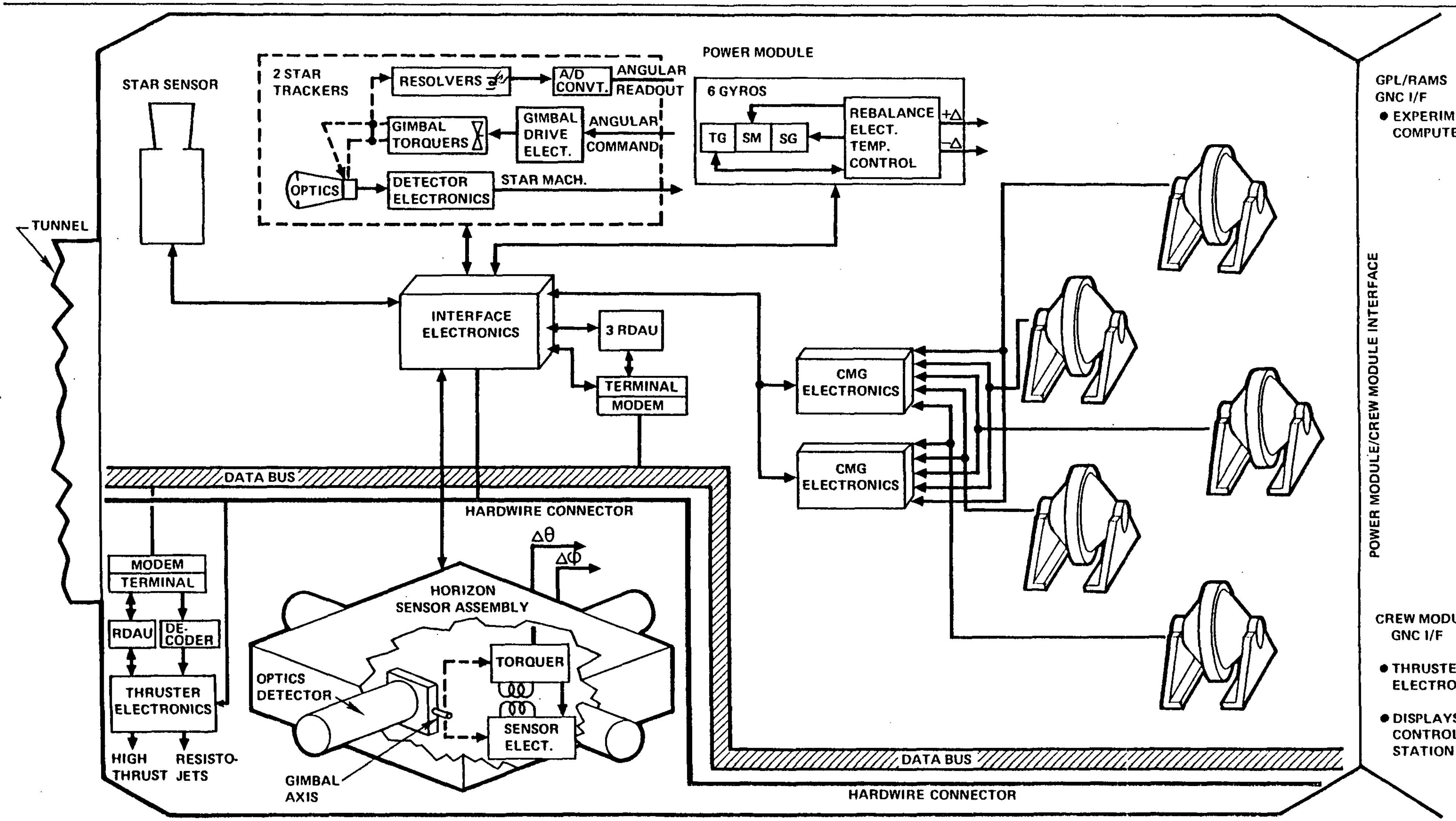


Figure 4.7-5. GNC Schema

are all located in the Power/Subsystems Module. High- and low-thrust engines (Propulsion Subsystem) are located on the Power/Subsystem Module and the Crew/Operations Module. During the normal manned operation of the ISS, the primary control actuation is provided by the CMGs, and the CMG desaturation and orbit-keeping functions are provided by the Propulsion Subsystem's resistojets. The high-thrust jets provide a backup capability for the resistojets and CMGs and provide the primary control actuation during docking operations. Also, during the unmanned phase of the ISS when the CMGs are not operational, the Propulsion Subsystems high-thrust jets provide the control actuation function.

Attitude control and navigation computations are performed in the Data Management Subsystem's computer located in the Power/Subsystems Module. Normal control signals are routed from the computer to remote engines via the DMS data bus.

The GNC interface from the Power/Subsystems Module to the Crew/Operations Module requires an electrical connection via the data bus to connect the thrustor drivers in the Crew/Operations Module to the thrustor control logic in the Power/Subsystems Module. A DMS data bus terminal supplying attitude and rate data from the computer is provided at each docking port for use by attached RAMs.

From Figure 4.7-5, the star trackers, star sensors, and gyros are located on the minus Z-axis side of the Power/Subsystems Module, while the horizon sensor is on the opposite side (+Z axis) facing the Earth in the horizontal orientation. Both the star trackers and horizon sensors are externally mounted to provide the large unobstructed area required for scanning and tracking. The star sensor, however, is stationary with respect to the vehicle and relies on the vehicle's motion to scan the celestial sphere. Because the star sensor is fixed with respect to the vehicle it can be internally mounted requiring only a small (approximately six inches in diameter) optically-flat window to provide the required field of view. Since this is the primary optical sensor used in the GNC subsystem, internal mounting is desirable for easy access for repair and maintenance.

Both sensor mounting bases are constructed so that the bases can be rotated within the Power/Subsystems Module for externally mounted sensor replacement in a shirtsleeve environment. The operation for this sensor replacement requires a dome cover, placed and bolted over the sensor base. The atmosphere with the dome cover is depressurized to the space environment and the sensor base is rotated up from the seal. A hatch cover is then rotated down against the seal which closes the volume within the dome from the space environment. The volume within the dome is then pressurized and the dome cover can then be removed to inspect and/or replace the sensors. After replacement the dome cover is installed and the same procedure is used to rotate the sensor base back into place. The same dome cover is used for both sensor bases.

In all operating modes and orientations, the gyros provide the high-frequency rate and attitude information necessary to supplement the data from the stellar sensors and the horizon sensors. The horizon sensors are used for initial acquisition of the Earth-referenced coordinates and provide a backup for the Earth-centered orientations.

The horizon sensor assembly contains four sensor heads that are aligned with the geometric axes of the Space Station in a particular arrangement to scan the horizon. Present horizon sensor designs have a linear range up to 10 deg. It is estimated that this linear range can be extended to 20 deg with only minor modification. This increased linear range of the horizon sensor is not sufficient to accommodate all the trimmed horizontal attitudes of the ISS phase which are greater than 20 deg nearly 40 percent of the time. Therefore in a backup mode some propellant penalty may result.

The primary method of providing the trimmed horizontal reference is to use a strapdown star sensor in the Earth-centered reference mode. Using the star sensor and the gyro assembly, the Earth-centered references (horizontal) and X-POP/OR) can be provided for any conditions of single- or three-axis trimming. The use of the star sensor for the large trim angles rather than a major modification of the horizon sensor is estimated to be more cost effective since the star sensor is an existing design and is also less costly on a per unit basis. It is also desirable to utilize the star sensor as the primary optical sensor as the primary optical sensor because it is internally mounted to facilitate repair and maintenance.

In all Earth-centered orientations the constant rotation rate required of the vehicle to maintain this orientation provides the scanning motion for the star sensor which is completely passive and provides no tracking capability. The star sensor provides the long-term, drift-free inertial reference data while the gyros provide the short-term, high-frequency attitude and rate information. The star sensor also provides inertial reference data to support attached RAMs requiring inertial reference data during the long-term, Earth-centered orientation.

The gimballed star trackers are primarily used for inertial orientations of the Modular Space Station. Because of the lack of angular rotation of the Space Station in this orientation, the sensors must provide their own tracking and scanning capability to acquire and track the desired reference stars. The inertial orientations are used primarily in support of the experiments.

The star trackers can also provide reference information in an Earth-centered orientation, but because of their lower reliability it is desirable to restrict their use to a backup mode to the star sensors. In this way the star sensors and star trackers complement each other. The star trackers with their large angular capability provide the operational flexibility required, while the star sensors provide the increased reliability necessary to achieve the long life requirements of the Space Station Program.

The sensor interface electronics controls the flow of information from the sensors to the data bus and converts all sensor inputs to a standardized format.

The primary attitude control actuation is provided by four double gimbal CMGs each capable of producing between 2, 700 n-m-sec (2, 000 lb-ft-sec) and 4, 070 n-m-sec (3, 000 lb-ft-sec) of angular momentum. A variable speed (8, 000 - 12, 000 RPM) improved ATM CMG provides this angular momentum range. The arrangement of the CMGs, shown in Subsection 4. 7. 4, was selected on the basis of the momentum envelope requirements of the modular space station and the ease of operational redundancy. The five CMGs (four operational and one standby) are mounted in the Power/ Subsystems Module so that the CMG outer gimbals are aligned with the body yaw or Z-axis. In the initial position, the CMG inner gimbal axes are aligned with the body pitch or Y-axis and the four CMG spin vectors are self-cancelling. By providing the capability of spinning the rotors in either direction, the fifth CMG provides for the loss or maintenance downtime of any one of the other four CMGs.

The lower level trades for the GNC assemblies such as star trackers, gyros, CMGs, etc., were performed in the 33-foot Space Station study and were documented in Reference 4.7-1. These lower level trades are the same for the Modular Space Station with the exception that some of the GNC sensors are not required.

The weight, volume, and power requirements of the GNC subsystems are given in Table 4.7-7.

Table 4.7-7

GNC PHYSICAL CHARACTERISTICS

Function	Weight (kg)	Volume (m^3)	Operating Power (w)
Attitude reference (gyros, horizon sensor, star sensor and star tracker)	74	0.133	100
Interface electronics	27.2	0.0028	30
Reaction control electronics	18.2	0.0228	40
CMG control electronics	9.1	0.0114	30
CMGs (5)	908	3.83	100
Totals	1,036.5	4.0	360
	(2,283 lb)	(141 ft^3)	

4.7.3.1.1 Attitude Gyros

Description

The attitude gyro assembly consists of six inertial-grade high-performance, single-degree-of-freedom, strapdown gyros which provide the inertial reference. The gyros are of the spin motor type, using a gas bearing to support the rotor. Gas pressure for suspending the rotor is generated by rotor motion, causing the bearing elements to be separated by a thin film of gas. Since there is no physical contact of moving parts, no wear occurs during operation. However, since bearing contact occurs during starting and stopping, the bearing life is limited by the number of on-off cycles. It is therefore desirable to keep the gyros running and avoid recycling as much as possible.

For a gyro failure, the effected gyro package, which contains three gyros, is replaced. In this manner, single gyro alignment and calibration procedures are eliminated. A shirtsleeve environment is provided for the maintenance and replacement tasks of the gyros.

The strapdown gyro package provides both rate and attitude information. The inertial attitude data derived from the gyros are used for short periods between star tracker updates.

Schematic

The schematic of the gyro and the rebalance electronics is shown in Figure 4.7-6. The gyro uses a moving coil permanent magnet torque generator with high torquing capability and a signal generator for operation at very high excitation frequencies. The rebalance electronics provides the pulse rebalance loops for the gyros. The loops operate on a pulse-on-de demand principle and provide precisely scaled rebalance torquing pulses to the gyro torquing coil as a function of output axis angular displacement. The scaling or pulse weight must be equal to an integral submultiple of two to allow direct interfacing with the computer. When the vehicle rotation angle exceeds the pulse weight, the signal generator secondary signal will be sufficient to trigger the level detector. A strobe pulse will then be synchronous with either a positive or negative level detector output to form a

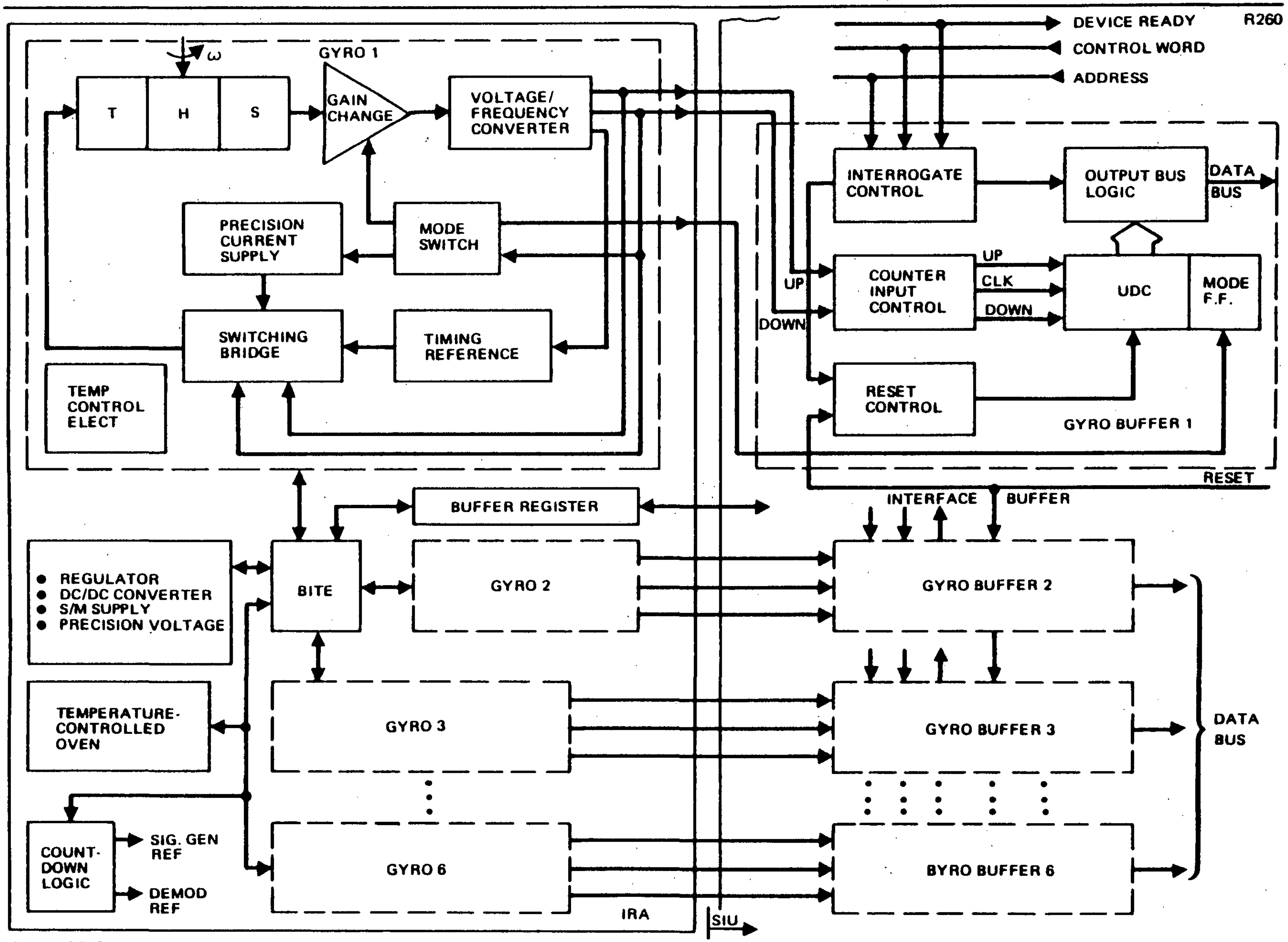


Figure 4.7-6. Gyro/Interface Buffer

gating pulse to drive the gating circuitry. These gate drive pulses are the increment attitude angle pulses. The gating circuit will furnish a precise pulse weight of current to the zero torque generator, which torques the gyro about its output axis toward its null position. These pulses are also sent to the computer for guidance and navigation computations. A second torquer winding of much smaller size than the primary is provided for compensation of the gyro.

Physical Characteristics

The two gyro assemblies consists of six gyros, none of which have co-linear input axes. This design approach used two orthogonal trials with the input axes of each gyro aligned along a common cone with half angle of 54.75 degrees.

The physical characteristics of the gyro package are as follows:

Assembly weight: 11.3 kg (25 lb)
Assembly volume: 2 x 0.0082 m^3 (7-1/2 by 7-1/2 by 9 in.)
Assembly power: 45 w normal, 130 w peak
Number of assemblies operating: 2
Number of assemblies: 2

The physical characteristics of the gas bearing gyro are as follows:

Angular momentum: 1.5×10^5 $gm\text{-}cm^2/sec$
Start power: 8 w
Run power: 3 w
Life: 45,000 hr
MTBF: 50,000 hr
Start/stop cycles: 10,000
Size: Diameter: 0.051 m (2.0 in.)
Length: 0.089 m (3.5 in.)
Weight: 0.454 kg (1 lb)
Random drift: 0.005 deg/hr
RMS noise: 0.0001 deg/sec
Bandwidth: 6 Hz

4.7.3.1.2 Horizon Sensors

Description

The basic principle of horizon sensor operation is the detection of the IR emission given off by the Earth's disk. From the observation of this emission, the vehicle attitude relative to the local vertical about two axes is defined by four optical heads whose four scan planes are situated 45 degrees from the $\pm X$ and $\pm Y$ axes of the vehicle. The horizon sensor is used to provide a coarse Earth-centered reference and to provide a method of initially acquiring the Earth-centered reference.

In the event of a horizon sensor head failure, each of the four heads can be replaced in a shirtsleeve environment as identified previously. With one of the four horizon sensor heads out, pitch and roll attitude data are available at nearly the same accuracy level as that for a full complement of sensor heads.

Figure 4.7-7 shows the schematic of the horizon sensor which is an edge-tracking system. The tracking signal is obtained by comparing the infrared radiance seen in the two fields of view which are separated by a fixed angle in the elevation plane of each tracking head. The two fields of view are servo-positioned within the scan plane so as to measure the declination angle to the horizon relative to the vehicle. Horizon edge tracking is accomplished by applying a fixed downward drive to the servoed assembly to direct the lower field of view into the horizon by an amount sufficient to generate a signal equivalent to this fixed downward drive. During track, the lower field of view rides on the horizon edge. This is functionally accomplished by amplifying the dc signals of two opposite thermopiles which are situated in the focal plane of the tracking telescope. This technique eliminates the requirement for the high-speed optical chopping or scanning normally used in edge-tracking horizon sensors. Vehicle tilt relative to the local vertical is obtained by measuring the difference between the declination angles from each optical head.

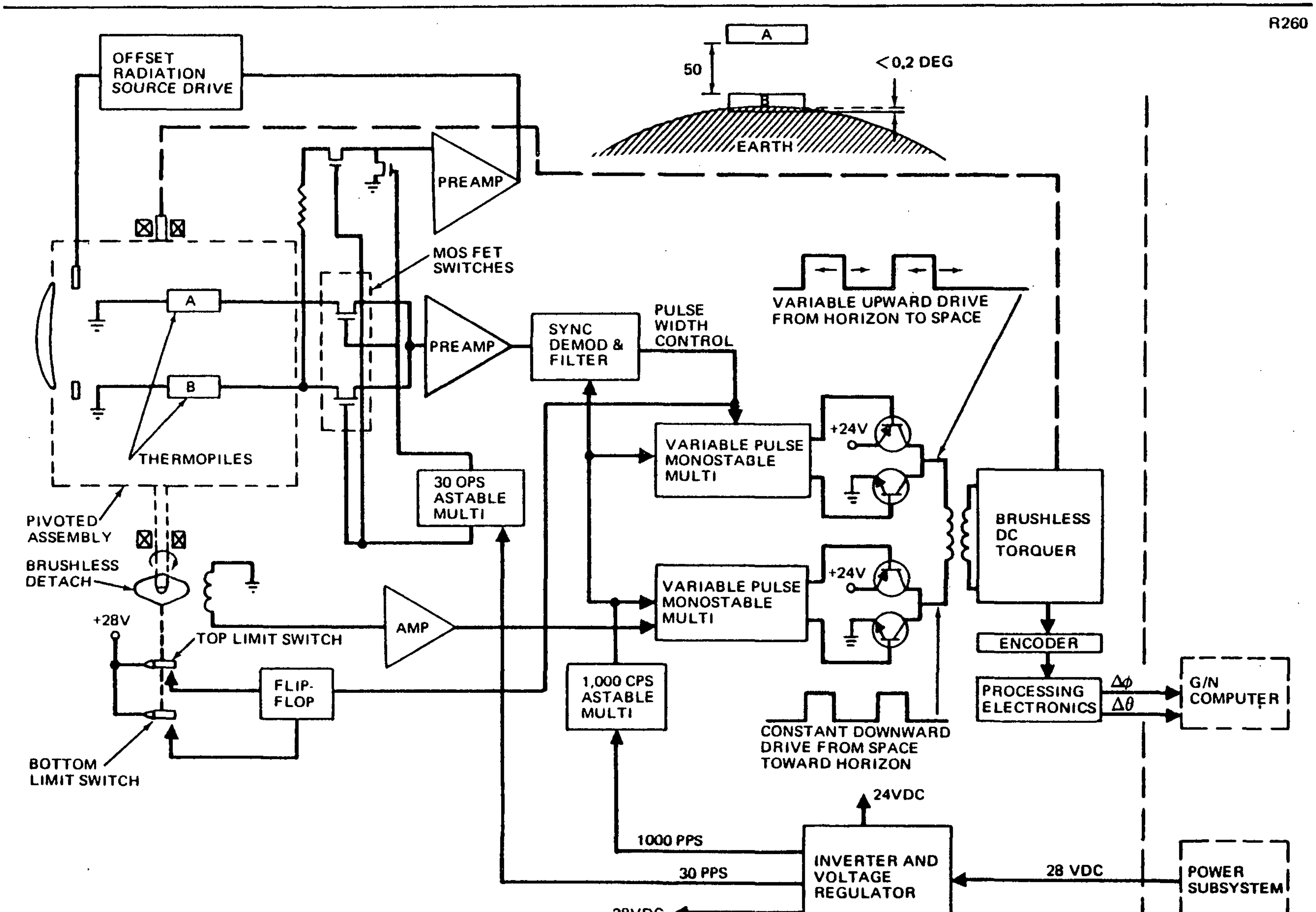


Figure 4.7-7. Horizon Sensor Block Diagram

Physical Characteristics

Figure 4.7-8 is a cross section of an horizon edge tracker designed by Barnes Engineering Co. The characteristics of the assembly are as follows:

Operating range: 70 to 100,000 mi

Accuracy: 0.1 deg (instrument only)

Linear range: ±10 deg (±20 deg with minor modification)

Field of view: 1 x 8 deg

Optical bandwidth: 14 to 16μ

Operating temperature range: -40 deg to 160 deg F

Assembly power: 7 w

Assembly weight: 5.9 kg (13 lb)

Assembly volume: 0.00492 m^3 (300 $in.^3$)

MTBF: 150,000 hr

Number of assemblies operating: 1

Number of assemblies: 1

4.7.3.1.3 Interface Electronics

Description

The interface electronics performs the analog-to-digital, pulse-to-digital, and digital-to-digital conversions which are required to standardize the sensor outputs for the data processing. Functionally, this assembly accepts the outputs of all the guidance, navigation, and control sensors and CMG's in a variety of forms and converts them to standard format digital words to be accessed by the computer.

Figure 4.7-9 presents a schematic of the interface electronics assembly. This assembly will contain the buffer registers where they are necessary to isolate the precounters from the input/output operations. Also, since the binary format resulting from the counter operation may not be consistent with the format used by the computer, the interface electronics will provide the necessary conversion logic.

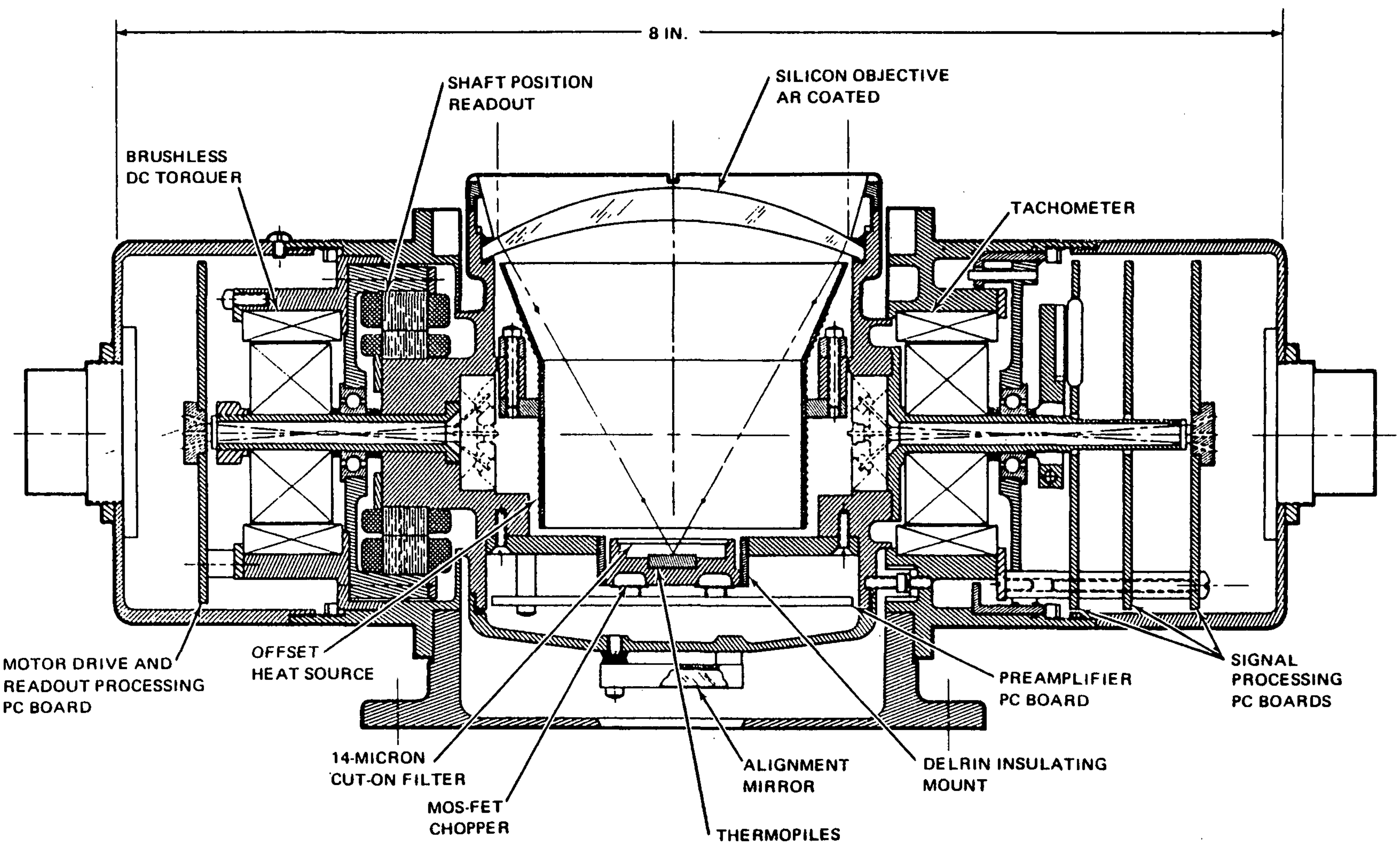


Figure 4.7-8. Cross Section of Thermopile Horizon Edge Tracker Head

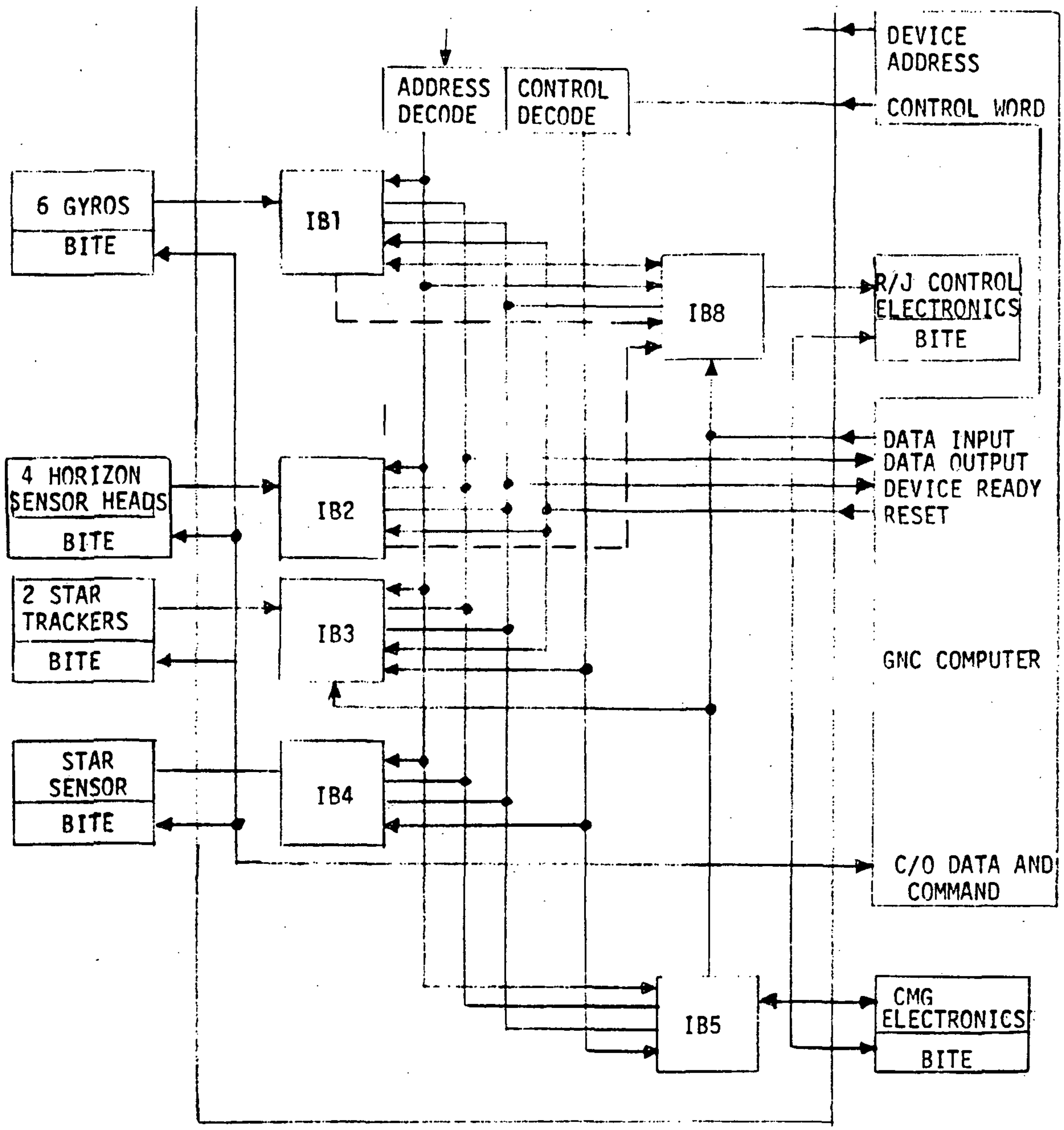


Figure 4.7-9. Interface Electronics

Physical Characteristics

The physical characteristics of the sensor interface electronics assembly are as follows:

Assembly weight: 13.6 kg (30 lb)

Assembly volume: 0.0131 m^3 (800 $in.^3$)

Assembly power: 30 w

Number of assemblies operating: 1

Number of assemblies: 2

4.7.3.1.4 Star Sensors

Description

Star sensors provide a highly accurate, drift-free, inertial reference for the Space Station in the Earth-centered orientation. In this orientation, the vehicle rotation is utilized to provide the scanning motion for the strapdown star sensors. Because of the intermittent nature of the stellar reference information, this type of sensor is used in conjunction with a gyro package. The gyro data provide an essentially continuous attitude reference, and the star sensor data are used to bound the long-term gyro drift.

The star sensor is a passive device because it does not operate in a closed tracking loop but depends on the vehicle to provide the scanning motion necessary for attitude determination. The star sensor detector consists of a number of photosensitive slits arranged in a spokelike array to take advantage of the relative motion of the star field caused by the motion of the vehicle in the Earth-centered orientation. The basic star measurement is the transit time when the image of a star crosses one of the photosensitive slits. Correlation of the knowledge of star celestial coordinates with the transit time provides partial information on vehicle inertial attitude. A schematic of the star sensors provided by Honeywell Incorporated is given in Figure 4.7-10. Multiple transits on different stars and on slits of varying angles with respect to vehicle motion provide the necessary data for complete attitude update.

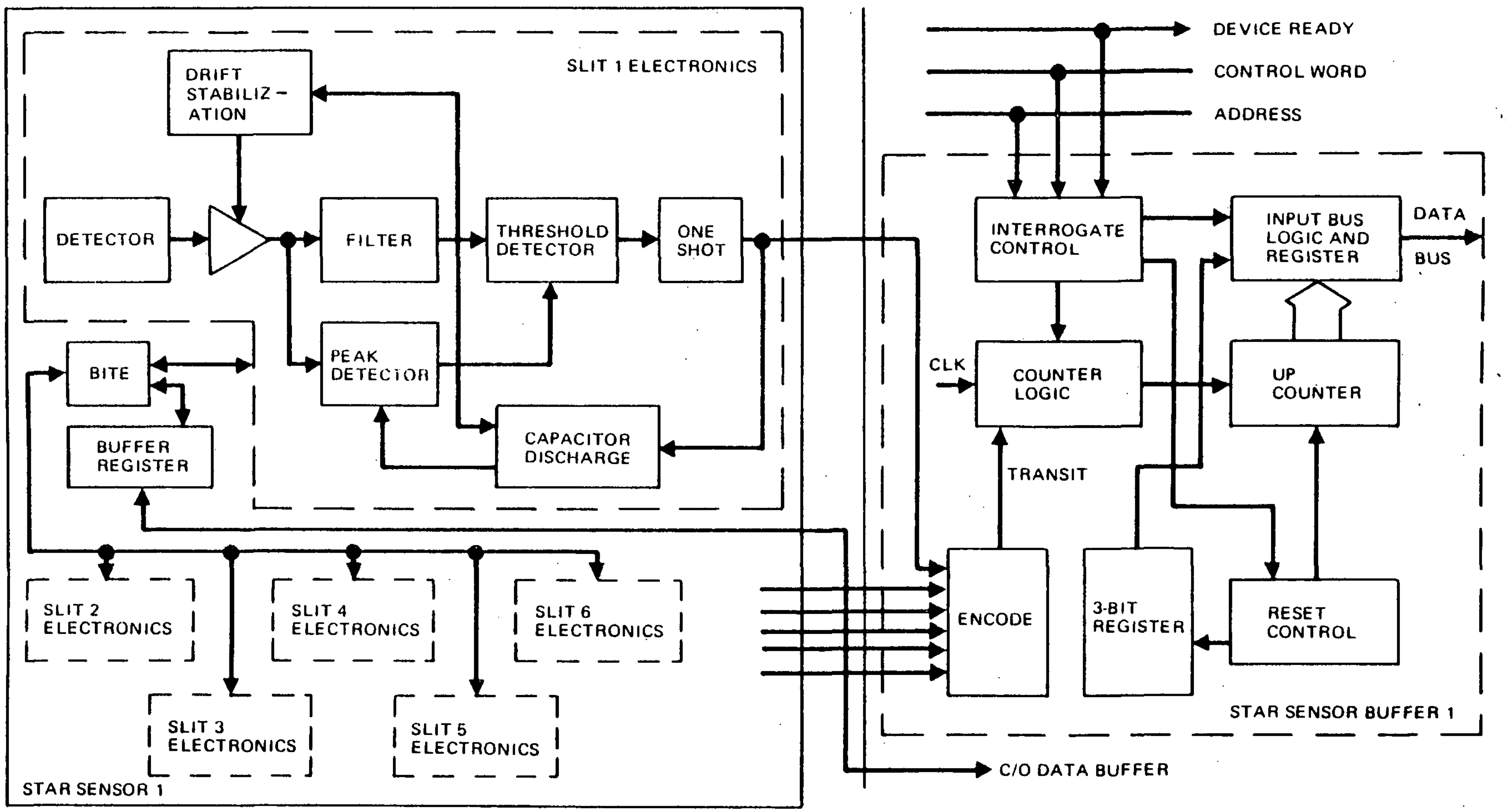


Figure 4.7-10. Star Sensor/Interface Buffer

Physical Characteristics

The physical characteristics for the star sensors are as follows:

Accuracy: 10 sec

Field of view: 10 deg

Assembly weight: 7.25 kg (16 lb)

Assembly power: 40 w

Assembly volume: 0.0082 m^3 (500 in.^{3})

Detector: Cadmium sulfide

MTBF: 35,000 hr

Development status: Presently undergoing environmental testing

Number of assemblies operating: 1

Number of assemblies: 1

4.7.3.1.5 Star Trackers

Description

The star trackers provide a highly-accurate, drift-free inertial reference for the Space Station in an inertial orientation. In the inertial orientation, the star trackers provide their own pointing capability by mechanical gimbals. The star trackers can be used in conjunction with a gyro package, with the star trackers bounding the long-term gyro drift.

In the event of a tracker failure, a single star tracker can be used to provide the inertial reference update.

Schematic

A schematic of the star tracker assemblies and their interface buffer is shown in Figure 4.7-11. The star trackers are two-axis, mechanically gimbaled trackers with digital encoders on each gimbal to indicate angular position. The encoder provides a string of pulses representing increments of attitude to a counter.

Included in the interface electronics are the circuits necessary for commanding specific gimbal angles. Since the command angle and the actual angle are stored in the computer, the command error signal (difference between the actual and commanded angles) is transmitted to the interface

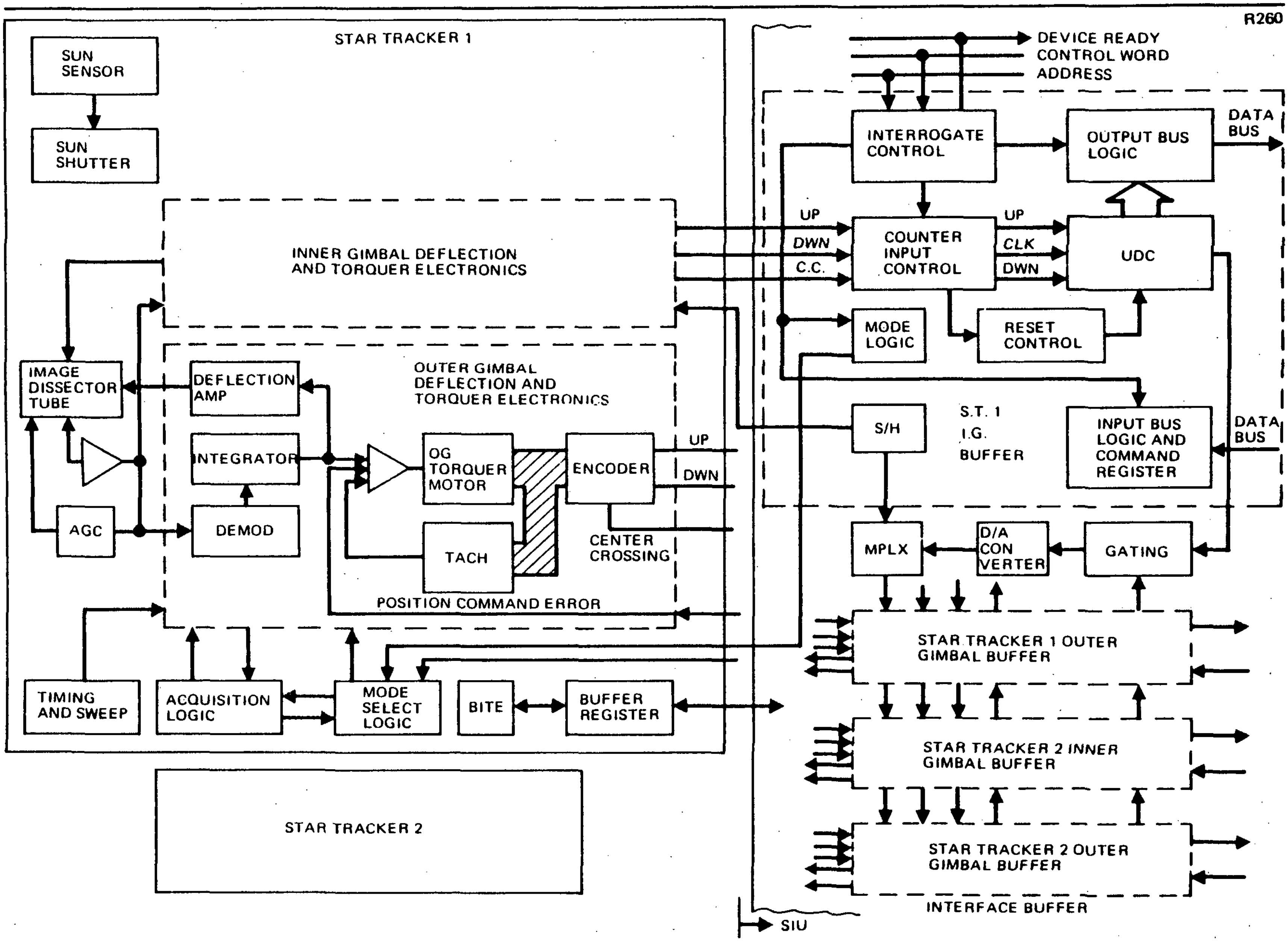


Figure 4.7-11. Star Tracker/Interface Buffer

buffer and stored in a command register. The error signal is conditioned and transmitter to the torque motor drive amplifier. The mode control logic for the various tracker modes is controlled by mode logic under command of control word.

Physical Characteristics

The size of the tracker is indicated in Figure 4.7-12. The physical characteristics are as follows:

Field of view: 1.0 deg

Gimbal freedom: ±60

Star magnitude sensitivity: 2.0

Accuracy: 10 arc-sec

Assembly power (including electronics): 39 w

Assembly weight (including electronics): 18.1 kg (40 lb)

Assembly volume: 0.0492 m^3 (3,000 in.^{3})

MTBF: 21,000 hr

4.7.3.1.6 Control Moment Gyro Electronics

Description

The control moment gyro electronics assembly provides the command signals to the gimbal torquers of the CMG's and it provides the analog-to-digital conversion for the interface assembly. The CMG electronics assembly also includes all the CMG gimbal electronics, mode control switching, and spin motor control electronics. This electronic assembly also contains the interface for monitoring CMG parameters such as wheel speed and bearing temperature.

The CMG electronics assembly is made up primarily of integrated electronics circuits and is located with the CMG's. Two of these assemblies are provided to give the required redundancy. Maintenance for this assembly is by modular replacement.

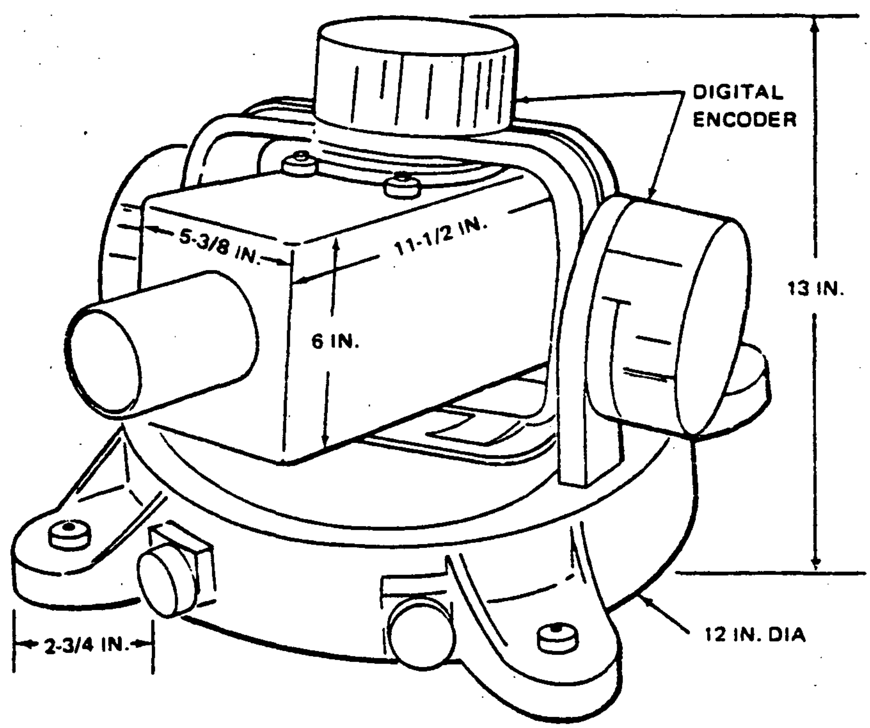


Figure 4.7-12. Star Tracker

Physical Characteristics

The physical characteristics of the CMG electronics assembly are as follows:

Weight: 4.54 kg (10 lb)

Volume: 0.0041 m^3 (250 in.^{3})

Power: 30 w

Number of assemblies operating: 1

Number of assemblies: 2

4.7.3.1.7 Control Moment Gyros

Description

The primary control actuation is provided by four double-gimbal CMG's. The double-gimbal CMG is a high-speed running wheel enclosed by two gimbals with the capability of approximating a spherical volume with the angular momentum vector. The CMG's provide torques for attitude control and rate stabilization and limited attitude maneuvers.

The double-gimbal CMG consists of the following assemblies: (1) the inner gimbal, an evacuated housing which contains the rotor, supporting bearings, spin motor, speed pickoff, bearing monitoring devices, evacuation valve, and lubricant reservoir, (2) an actuator assembly and sensor assembly (angle transducer and tachometer), mounted on the outer gimbal ring to provide control of the inner gimbal; and (3) an identical set of actuator and sensor assemblies, mounted on the frame to provide control of the outer gimbal ring.

The specific CMG selected is the improved ATM CMG designed by the Bendix Corporation, which has a selectable speed in the range of 8,000 to 12,000 rpm and an unlimited gimbal freedom. The angular momentum capacity is between 2,700 and 4,070 n-m-sec (2,000 to 3,000 lb-ft-sec).

The improved ATM CMG's have replaceable assemblies, such as spin bearings, gimbal torquers, and spin motors, so that the entire CMG unit does not have to be replaced. This reduces the logistics weight requirements for the CMG's by several times.

Schematic

The control system of the double-gimbal CMG is a high-gain velocity servo which will provide linear gimbal rate control over a specified range. The block diagram shown in Figure 4.7-13 illustrates the functional modules of the gimbal control system. Each gimbal servo loop consists of a DC torquer and drive train, a DC tachometer, several stages of operation amplifiers to provide voltage gain the forward loop, and an operational amplifier which is used to amplify the gimbal rate reference signal. The resolver and potentiometer outputs shown provide functions of gimbal position angles for implementing the CMG in the attitude control system. The gimbal limit switches are adjustable to any gimbal angle.

The selected mounting arrangement of the CMG's is given in Subsection 4.7.4, Design Analyses and Trade Studies.

The four double-gimbal CMG's will respond to gimbal rate commands to the inner gimbal and outer gimbal torques. The magnitude of each gimbal rate command as a function of vehicle rate and position error signal is determined by the CMG control law.

Physical Characteristics

The physical characteristics are as follows:

Angular nomentum	Max 4,070 n-m-sec (3,000 lb-ft-sec)
Wheel speed	Max 12,000 rpm
Wheel weight	65.7 kg (145 lb)
Wheel diameter	0.559 m (22 in.)
Total CMG weight	181.5 kg (400 lb)
CMG envelope	1.07 m x 1.09 m x 1.02 m (42 by 43 by 40 in.)
2 Brushless DC Spin Motors	
Steady-state power	30 w
Run-Up power (peak)	414 w
Run-Up energy	588 w-hr
Run-Up time	2 hr

R260

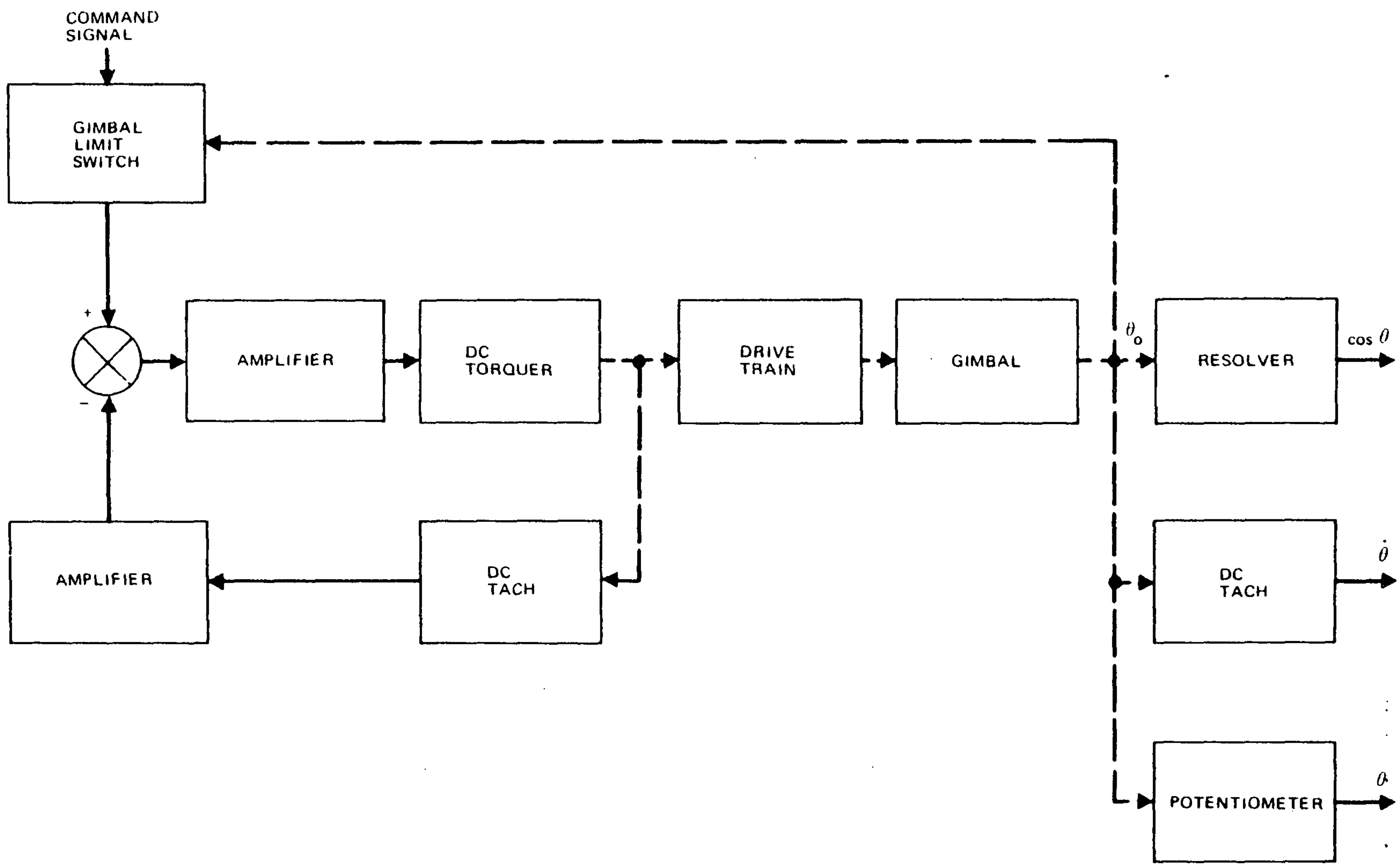


Figure 4.7-13. Gimbal Rate Control Functional Block Diagram

Gimbal Torquers	
Brushless DC torquers	9.5 n-m (7 ft-lb)
Gear ratio	28.6:1
Brushless DC tach	
Unlimited gimbal freedom	
Slip rings	
Performance data	
Max torque	271 n-m (200 ft-lb)
Max gimbal rate (To be a function of gimbal angle)	
Frequency responses	10 Hz
Monitoring	
Wheel speed	
Bearing temperature	
Bearing vibration	
Vacuum	

Number of assemblies operating: 4

Number of assemblies: 5

4.7.3.1.8 Reaction-Jet Control Electronics Assembly

Description

This assembly consists of logic, holding circuits, and power switching elements to activate both resistojets and the high-thrust bipropellant jets. Instrumentation for thrustor operation fault detection will be provided, which includes an excessive thrustor activity monitor.

The reaction-jet control electronics assembly and interface buffer are detailed in Figure 4.7-14. The primary operating mode uses commands from the DMS computer through the interface buffer. The commands are in the form of on/off commands to the jet driver. In the backup mode, the assembly generates an attitude reference (based on gyro and/or horizon sensor inputs) and a control law to provide jet on/off commands. Mode logic, attitude commands, and display signals communicate directly with

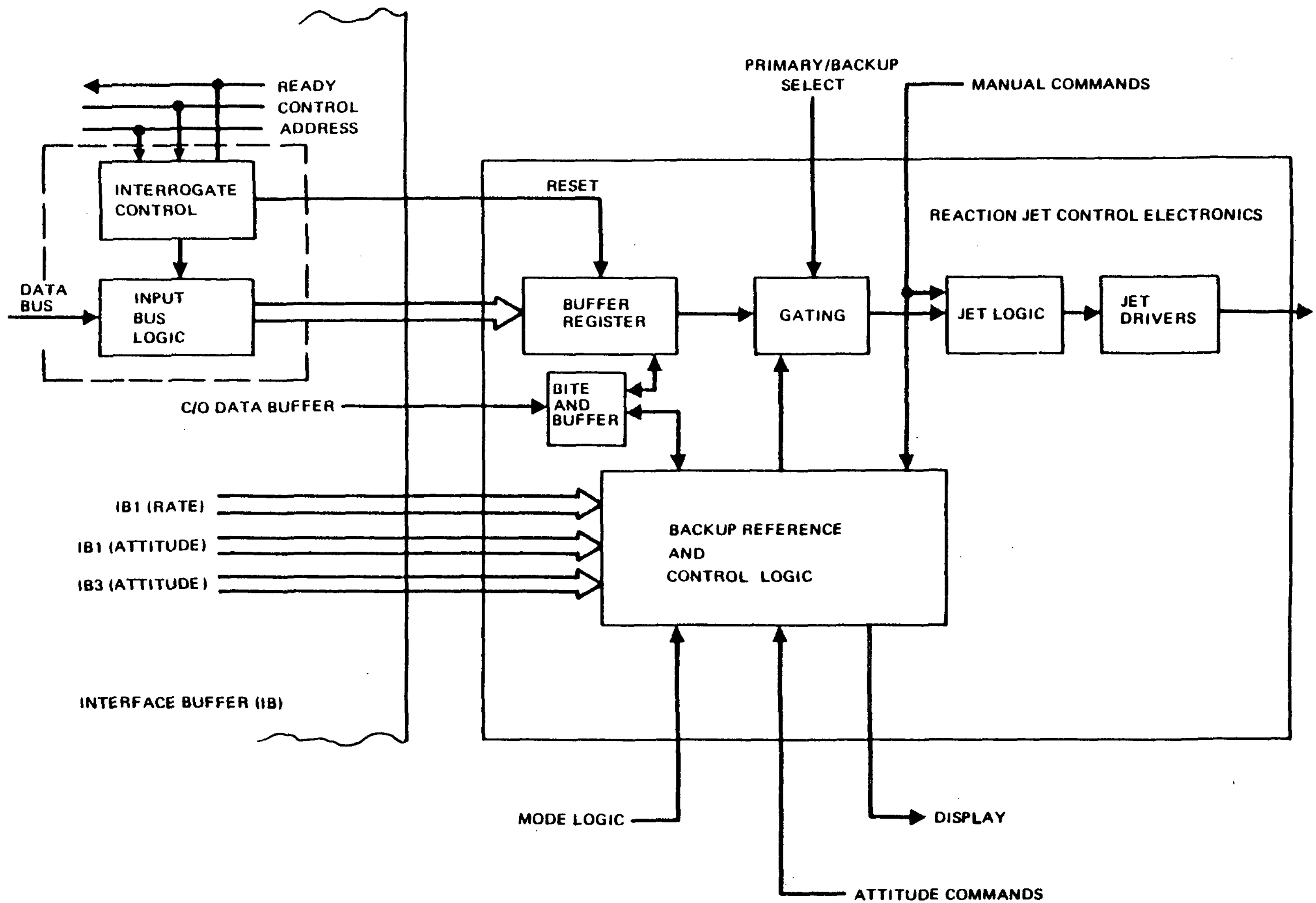


Figure 4.7-14. Reaction Jet Control Electronics/Interface Buffer

the display console. A signal to inhibit the reaction control system (RCS) operation is provided so that the disturbances or contamination during critical experiment operation can be minimized.

Schematic

Figure 4.7-15 is a block diagram of the resistojet control. The accelerometer provides the information for the orbit-keeping function. The logic selects the proper jets (depending on the Space Station orientation) and determines the level of I_{sp} for the throttle command. The gimbal angles of the CMG's are continuously monitored to determine the desaturation requirement. The desaturation function also provides inputs to the jet selection and throttle command logic assemblies.

Physical Characteristics

The physical characteristics of the reaction-jet control electronics assembly are as follows:

Assembly weight: 9.0 kg (20 lb)
Assembly volume: 0.0098 m^3 (600 $in.^3$, 5 by 6 by 20 in.)
Assembly power: 40 w and 8 w per jet driver
Number of assemblies operating: 1
Number of assemblies: 2

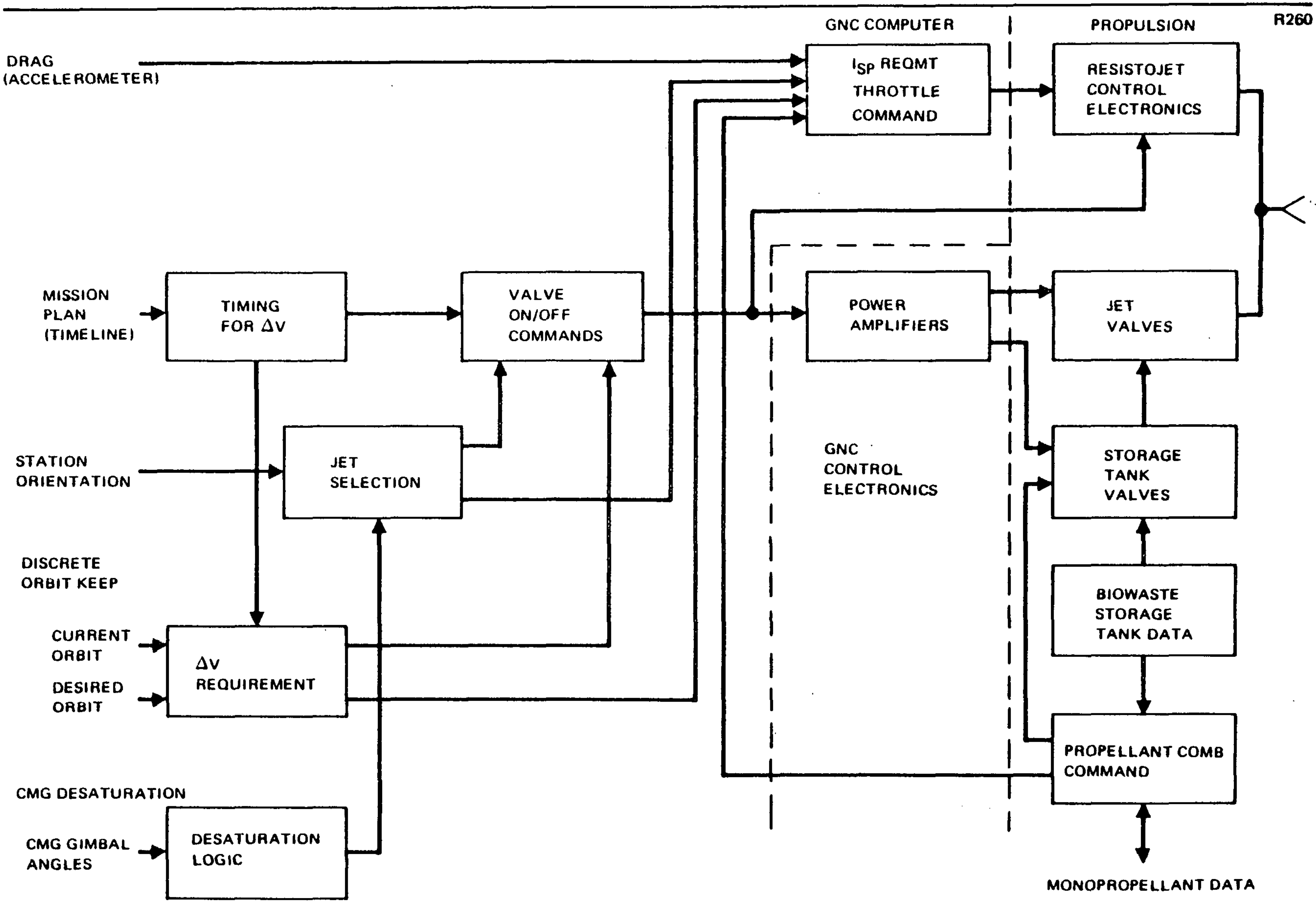


Figure 4.7-15. Resistojet Control

4.7.3.2 Interfaces

In the ISS configuration, the GNC equipment (sensors, electronics, and CMGs are located in the Power/Subsystems Module. The attitude control and navigation computations are performed in the Data Management Subsystem's (DMS) computer located in the Power/Subsystems Module. The interfaces between the sensors and control actuation (CMGs and high and low thrustors) are provided via data bus terminals. In using the DMS computer instead of a GNC dedicated computer, the GNC interface between the modules is provided through the data bus terminals of the DMS. Certain backup modes of operation requiring hardwire connections around the computer are required.

The GNC Subsystem provides the electrical signals for firing the high and low thrustors of which both types are located on the Power/Subsystems Module and the Crew/Operations Module. The GNC subsystem also provides attitude and rate reference data and the monitor and alarm functions to the primary and secondary display/control console in the Crew/Operations Module and the GPL, respectively. The attitude and rate reference data are also supplied to the dedicated experiment computer in the GPL for those attached RAMs requiring such. These interfaces from the Power/Subsystems Module to the Crew/Operations Module and GPL are provided via the data bus terminals of the DMS. Any additional (presently not identified) GNC interface requirements to other modules can be easily provided via the data bus with negligible impact to the GNC subsystem.

The GNC subsystem also has a hardwire connection from the sensor interface assembly to the high-thrust jets on the Power/Subsystems Module and the Crew/Operations Module. This hardwire connection provides backup attitude control using the high thrust jets. A second level of backup control using the high thrust jets is provided through a hardwire connection from the hand controller in the Crew/Operations Module.

Figure 4.7-16 shows a schematic of the GNC subsystem. The heavy blocks indicate GNC equipment while the light blocks show the DMS interface equipment. The dashed lines show which equipments are contained in the

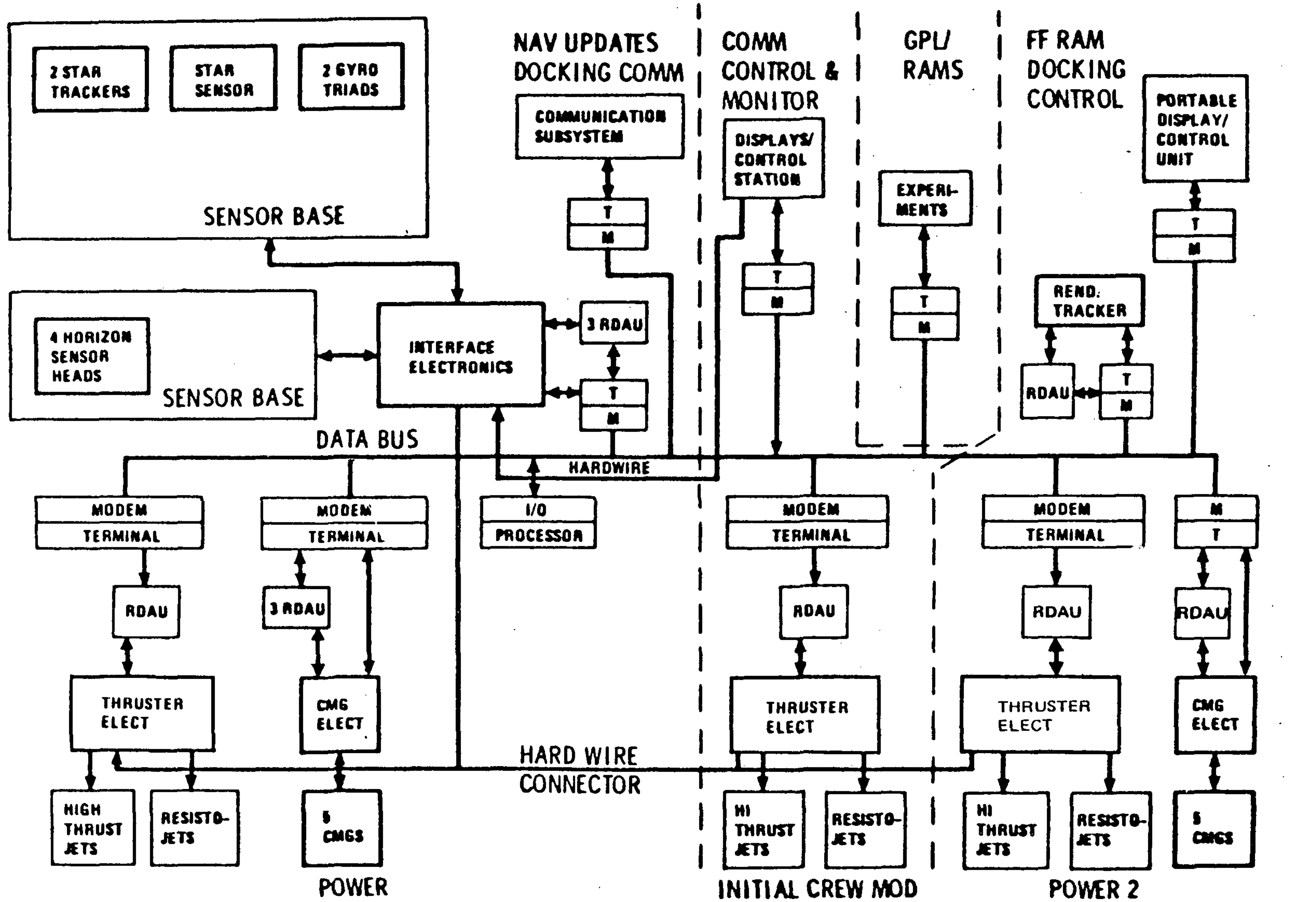


Figure 4.7-16. GNC Subsystem

Power/Subsystems Module, Crew/Operations Module, and GPL. A second Power/Subsystems Module is also shown which is used for the GSS.

This illustrates the negligible impact to the GNC interface requirements in using the data bus terminals of the DMS.

Table 4.7-8 shows the interfaces between the Shuttle and modules for the GNC subsystem along with the functions or operations and the type of interface. Only the Crew/Operations Module and the GPL have interface requirements to the GNC subsystem located in the Power/Subsystems Module. The GNC interface to the GPL is through a data bus terminal while the Crew/Operations Module has a hardwire GNC interface, as well as the data bus terminal for the GNC attitude reference data. There are no physical interface requirements of the GNC subsystem to the Shuttle or attached RAMs. Any GNC functions supplied the attached RAMs, such as attitude reference data, are provided from the experiment computer in the GPL by a data bus terminal. The Logistics Module and GNC have no interface requirements.

The module interfaces of the GNC subsystem are electrical, but GNC assemblies have other interfaces besides electrical. The attitude gyro package interfaces with the Power Subsystem, the EC/LS for environmental control, the structure for alignment, and the sensor interface electronics to receive the output of the gyros.

The horizon sensors interface with the Power Subsystem, the structure for alignment, and the sensor interface assembly, which receive the output of the horizon sensor electronics.

The interfaces of the star sensor and star tracker are with the Power Subsystem, the structure for alignment, and the interface electronics.

The interface electronics assembly interfaces with the Power Subsystem, the EC/LS for environment control, the complement of sensors (gryos, star sensors, star tracker and horizon sensor), and the DMS computer by the data bus terminal.

Table 4.7-8

SHUTTLE AND MODULE INTERFACES OF GNC

(ISS Configuration)

Interface	Function or Operation	Type of Interface
Power/Subsystems Module to Crew/ Operations Module	Electrical signals to high thrustor and low thrustor valves.	(1) Data bus terminal (2) Hardwire connection to high thrustors
	Attitude reference data and monitor and alarm functions for display/control console	Data bus terminal
	Manual Controller.	Hardwire
Power/Subsystems Module to GPL	Attitude reference data and monitor and alarm functions for display/control console.	Data bus terminal
	Attitude and rate reference data for Experiment support.	Data bus terminal
RAM's	Attitude reference and navigation for Experiment support.	Data bus terminal
Shuttle	Attitude and navigation data for initialization.	Data bus terminal

The CMG electronics assembly interfaces with the Power Subsystem, the EC/LS for environment control, the CMGs and the data bus terminal.

The CMGs interface with the power subsystem, the structure for alignment and torque transmittal, and the CMG electronics assembly.

The reaction-jet control electronics assembly interfaces with the power subsystem, the EC/LS for environment control, the data distribution assembly, the high-thrust jets and the resistojets.

4.7.3.3 GNC Operations

The GNC operations are primarily automatic with selection of the various operating modes and orientations being controlled by the crew via the display and control panel or via uplink commands from the ground control center. The GNC Subsystem also relies on the crew for repair and replacement of failed equipment. Although the system is primarily automatic, manual attitude control and maneuvering capability is provided at either display and control console through a hand controller which operates the high-thrust propulsion.

4.7.3.3.1 Initial Operation

All GNC equipments are located in the Power/Subsystems Module with the exception of the control electronics used to operate the propulsion thrustors in the Crew/Operations Module. The control moment gyros are not launched with the Power/Subsystems Module but are brought up in the first Logistics Module.

During the prelaunch phase, the GNC Subsystem will be checked out on the launch pad using the onboard checkout subsystem software and auxiliary ground checkout equipment. After checkout, the equipments are turned off with the exception of the gyro package which will remain on and electronically caged during launch. When operating, the gyros are less susceptible to the launch environment. This also eliminates the delay that would be necessary during on orbit checkout if the gyros were activated from a cold start condition.

After orbit insertion, the GNC Subsystem is checked out while the Power/Subsystems Module is attached to the Orbiter. Operation of all sensors, electronics and the DMS interface are verified before the Power/Subsystems Module separates from the Orbiter. After separation, commands from the Shuttle are issued to the GNC Subsystem to place the Power/Subsystems Module in the horizontal orientation. Normally the module will have been released in an orientation which will allow the GNC horizon sensors to view the Earth. If for some reason such as high tip-off rates, the Power/Subsystems Module has rotated far enough so that the horizon sensors do not view the Earth, the GNC Subsystem will automatically operate in the rate-stabilization mode until an acquisition command is received from the Shuttle or ground control at which time the GNC Subsystem will automatically command a sequence of maneuvers to allow the horizon sensors to acquire the Earth. After horizon sensor acquisition, the vehicle will be maintained in the local horizontal orientation. Any subsequent loss of the horizon by two or more horizon sensor heads will return the system to a rate stabilization mode. The use of an acquisition capability which requires ground command rather than an automatic initiation precludes the possibility of depleting the propellant supply due to a failure which causes repeated acquisition sequences.

The rate-stabilization mode also provides a backup to the primary horizontal mode which minimizes vehicle rates during docking. The backup rate-stabilization capability uses a minimum of hardware to provide a reliable alternate to the primary mode. This capability can also be exercised without DMS computational support.

The Modular Space Station will operate throughout the buildup sequence in either the horizontal or rate-stabilization mode using the high-thrust system. Both of these modes are compatible with the Shuttle docking operations. During the time when the Shuttle is attached to the Space Station, the Shuttle will control the attitude of the combined vehicle. Thrustor firing of both vehicles will be inhibited automatically when the docking mechanisms latch to prevent unnecessary stressing of the docking mechanism. The thrustors will be inhibited until the mechanism is fully retracted and latched, at which time the Shuttle will assume the attitude control function. Manual override of the inhibit signal is provided.

The on-orbit operations begin after the first logistics flight. The first logistics module will contain the control moment gyros which will be installed in the Power/Subsystems Module by the crew. After the CMGs are spun up and checked out, the GNC Subsystem will be fully operational.

4.7.3.3.2 On-Orbit Operation

The primary operation of the GNC Subsystem is in the trimmed horizontal orientation using CMG's as the primary control actuators. To minimize propellant consumption, this mode of operation will be used whenever it does not interfere with the experiment operations. Simultaneous orbit keeping and CMG desaturation is automatically provided using resistojets. In this mode, the primary sensors are the gyros which provide continuous attitude information for the control function. The gyro assembly contains six gyros which allow two failures to occur without disrupting the attitude information. The failures are automatically detected and the GNC software is modified to process only the information from the working sensors. At the same time, a signal is sent to the crew to inform them of the failure. The attitude computation is periodically corrected using data from the star sensor to remove the error accumulated as a result of the inherent drift in the gyros. Navigation information is used in this orientation to convert the inertial reference obtained by the sensors into an Earth-centered reference required for attitude control.

Other orientations are selected by the crew through the control and display console as required to support a particular experiment schedule. These orientations are described in subsection 4.7.2, Requirements. Operation in each of the orientations is described in the following paragraphs.

- Horizontal Orientation - Operation in this mode is the same as the trimmed horizontal with the exception that the pitch axis is not trimmed and therefore the longitudinal axis of the station is aligned with the orbital velocity vector. This mode is used to support Earth-pointing experiments such as Earth resources.
- Z-Axis, trimmed horizontal - Operation in this mode is the same as the horizontal orientation with the exception that the yaw axis is rotated to an angle that minimizes the bias torques on the pitch axis. This mode is used to support Earth-pointing experiments.
- Perpendicular to Orbit Plane/with Orbit Rate (POP/OR) Orientation (Trimmed and Untrimmed) - These orientations use the same equipment as the horizontal orientations with the pitch and roll body axes interchanged (see subsection 4.7.2).

In all the earth-centered orientations (horizontal and POP/OR), the star sensors provide the drift-free attitude information to compensate for the gyro drifts. This sensor and electronics are designed with builtin redundancy which allows up to two failures in the sensor itself without interruption in the normal attitude control function. A backup mode using horizon sensor information to update the gyro-derived attitude reference is also available. This backup capability does limit the trim angle capability to ±20 degrees, however, no navigation data are required. This capability can also be used as a backup if navigation information is not available due to some failure in the onboard or ground equipment.

- Inertial Orientation - In this orientation, the gimbaled star trackers are used to update the basic gyro reference. Since there is no motion of the vehicle with respect to the celestial sphere, the star sensors which have no tracking capability of their own are not usable and the star positions must be obtained by acquiring and tracking stars using the gimbaled sensors. The inertial orientation is used to support inertially oriented experiments such as the stellar astronomy experiments. This orientation is the most expensive in terms of propellant usage. Also, because of the larger disturbing torques, the high-thrust capability of the Propulsion Subsystem must be used for the CMG desaturation and no orbit keeping is performed while the vehicle is operating in this orientation. Although two star trackers are normally used for the orientation, the GNC does provide the capability to operate with one tracker failed.
- Perpendicular to Orbit Plane (POP) Orientation - The POP orientation is a pseudo inertial orientation which results in considerably less propellant consumption than the inertial orientation. This orientation will be used with attached pointing experiments which have sufficient gimbal freedom, field of view, or are of a short enough duration that the orbit regression rate does not interfere with the experiment. It is this small rate (4 degrees/day) which is the basic difference between this orientation and the pure inertial orientation. The same sensors are used in both the inertial and POP orientations. The POP orientation also has the advantage that the low thrust system can and will be used for orbit keeping and desaturation.

Primary actuation for all on-orbit operations is provided by four, double-gimbaled control moment gyros (CMGs). The GNC Subsystem also provides the capability to operate with three CMG's, in the event of a failure, until the standby unit can be spun up. In addition to the primary actuation, the GNC may also operate using the high thrustors as the actuation device as it does during the unmanned buildup phase. This option is selectable by the crew and may be used for maneuver, recovery from docking and separation transients, or in the event of a failure which renders all CMG's inoperative such as a DMS failure. Because the high thrustors are hardwired to the GNC electronics, attitude control can be provided independent of the CMGs or DMS. Although the high-thrust engines will not normally be fired unless commanded by the crew, an attitude error exceeding 2 degrees or an attitude rate exceeding 0.1 degree-per-second will cause activation of the high thrustors to maintain the vehicle within these boundaries. Since these boundary conditions are only exceeded in the event of a system failure, the larger control torques provided by the thrustors result in a fast recovery to prevent high-rate buildup and tumbling which may result from a thrustor which has failed "on" or some other failure resulting in the expulsion of material causing large disturbing torques. The occurrence of the anomaly also results in a caution signal to the crew.

The low-thrust system is used for CMG desaturation and orbit keeping in all orientations except the inertial orientation. The orbit-keeping function may also be performed using the high thrustors after long periods of inertial orientation or due to errors accumulated with low-thrust operation. The high-thrust system will also be used for orbit changes which are required to avoid collision with another object since a fast response may be necessary. Because the Space Station relies on the ground for navigation data and provides no capability for measuring velocity changes, all orbit maneuvers using the high thrustors are performed by timed firings of the high thrustors. The firing time is determined by ground control and the initiation may be performed through the control panel or via uplink command from ground control.

A one-year storage requirement for the three primary modules—Power/Subsystems, Crew/Operations, and GPL would require that the three-module configuration be boosted to an orbital altitude of 500 km (270 nmi) and placed in a gravity-gradient stabilized attitude. With the solar panels feathered, the orbital altitude of the three-module configuration will remain above 455 km (246 nmi) for a period longer than one year. A more detailed description of the various operating modes and features of the GNC subsystem may be found in subsection 4.7.4.5, Software Definition Task.

4.7.3.3.3 On-Orbit Checkout and Maintenance

The GNC checkout software provides for fault detection, fault isolation, calibration, and redundancy management. The most common method of fault detection is performed within the Remote Data Acquisition Units (RDAU's) which continuously check test points and interrupt the DMS processor if an out-of-limit signal is received. Limits for each test parameter are stored in the memories of the RDAU's. Software to control the RDAU operation is part of the DMS executive.

Detection of an out-of-limit condition results in an immediate notification of the crew. Initiation of further action is a crew option except where an out-of-tolerance condition is considered critical. Twenty critical functions have been identified for the GNC subsystem; they are the bearing temperatures and vibration levels of the CMG's. If any of these parameters are exceeded, the corresponding CMG is automatically despun and the attitude control software is modified to allow operation with three CMG's running.

These parameters are also examples of fault isolation by limit checking. Where an out-of-tolerance parameter isolates as well as detects the fault, no further diagnostics are required. In addition to the monitoring performed by the RDAU's, the GNC provides continuous fault detection of the inertial unit, horizon sensor assembly, and star sensor by using comparative techniques on the outputs of the actively redundant sensors as described previously. Detection of a fault by this method also results in immediate crew notification. Further action is a crew option since the GNC subsystem will continue to operate with any single failure in these sensors. The GNC software also provides a fault indication whenever the attitude error exceeds 2 degrees or the attitude rate exceeds 0.1 degree-per-second.

In addition to the continuous orbital monitoring performed by the RDAU and within the GNC subsystem as described above, periodic checks will also be performed. These periodic checks will use the same software employed for fault isolation. This software must be capable of fault isolation to the line replaceable unit (LRU) level. The LRU's for the GNS subsystem are presented in Table 4.7-9. The rationale for the selection of these LRU levels

Table 4.7-9

LINE REPLACEABLE UNITS

Line Replaceable Unit	Number Active	Standby
Horizon Sensor Head Assembly	4	0
Inertial Reference Assembly (3 gyros per assembly)	2	0
Star Sensor	1	0
Star Tracker	2	0
Interface Electronics	1	1
Control Logic Electronics	2	2
CMG Electronics	1	1
CMGs (less spinmotor, torquers and bearings)	4	1
CMG Bearings	-	-
CMG Torquer	-	-
CMG Spin Motor	-	-

is based primarily on the use of currently designed components used on existing programs or in the process of being developed for existing programs. This approach takes maximum advantage of existing design and development to minimize design risk and cost. Lower level LRU's are not recommended in the GNC subsystem because of the development cost of special packaging requirements which also generally increase the weight of the device. Also most of the GNC sensors require tight alignment and calibration which is more easily performed in a bench-maintenance environment. Refurbishment of the LRU's will be performed on the ground where the special test equipment is available.

An exception to this general rationale is the replacement of the CMG bearings, torquers, and spin motors. This advanced ATM design which allows on-orbit replacement of components is under development by Bendix Corporation and is expected to be in operation for Space Station use. The large logistics weight savings which results by this maintenance technique and the expected availability favor the on-orbit maintenance approach selected for the CMG's.

Periodic checks are performed to ascertain whether certain parameter degradations which are not obviously detectable have occurred and to detect failure in inactive or standby equipment. Calibration is a subtask of the periodic checkout task and will be conducted during the periodic event. Checkout intervals are nominally once a month based on predicted performances of the components. The star trackers for inertial orientation are used infrequently and will be tested prior to entering the inertial mode of operation.

The fault isolation is accomplished through the technique of introducing calibrated stimuli at the first practical point in the forward path of the GNC loop and monitoring subsequent downstream points for fault isolation. Many of the downstream monitoring points are DMS-computed data such as attitude and attitude errors. In this case, the test sequence begins with verification through self-diagnositc routines of the GNC software and DMS/GNC interfaces.

4.7.3.3.4 Redundancy Management

Several levels of redundancy management are used in the GNC Subsystem. Failures in the inertial reference unit are detected and the software is reconfigured to ignore the failed channel and continue to operate by processing the remaining five gyros. The process is also used for the second failure since fault isolation is possible with any two failures. With the resupply interval for the Space Station, no provisions for operation beyond two failures have been made. Replacement with new or refurbished hardware should have been accomplished before the third failure. Although the GNC Subsystem provides automatic fail operational capability in this area, the failed gyro is not turned off unless some critical parameter which may destroy the instrument and propagate further failure is found to be out of tolerance. For the gyro, these include the gyro temperature and spin speed monitors. Out-of-tolerance detected in this parameter will cause automatic shutdown of the gyros.

The horizon sensors redundancy is controlled in the same way as the inertial reference unit except that no automatic shutdown is required unless an overtemperature condition occurs.

The star sensor contains some builtin redundancy and will continue to function whenever two out of three slits on each side of the sensor are operating. Detection of these and other faults result in crew notification. Further action is a crew option. This is also true of the Star Tracker although calibration and checkout procedures are initiated prior to any use of the star trackers. If one tracker is found to be faulty, the crewman will flag the GNC software so that operation with a single tracker can be accomplished until the faulty sensor can be replaced.

The only other critical parameters are the 16 CMG parameters, bearing temperature, and vibration. Out-of-tolerance on any of these parameters will automatically initiate shutdown of the faulty CMG and flag the control software to modify the CMG control laws for three gyro operations. Although a fifty gyro is available on standby, it is only actuated through crew initiation. The interface electronics and control electronics likewise employ standby redundancy which are only activated by crew initiation.

In general, the redundancy management function is primarily handled by the crew through activation of standby hardware or replacement of failred LRU's. Exceptions to this are the critical gyro and CMG parameters which cause automatic shutdown of faulty components for safety reasons. In no instance does the GNC activate any redundant hardware. The rationale for this philosophy is that most GNC failures are not hazardous or critical from a standpoint of crew safety or mission success and that a costly and complex automatic redundancy management technique cannot be justified.

4.7.3.4 Growth Space Station (GSS) Considerations

4.7.3.4.1 GSS Requirements and Capabilities

The GSS has only one additional requirement of the GNC Subsystem over those of the ISS configuration. Since the ISS phase does not have the free flying RAM's as the GSS does, the GSS must provide the control for rendezvous and docking of the free flyers. A laser tracker is used to provide the rendezvous function for the free flyers and manual control is used for the docking operation.

The GSS has the same all-orientation capability as the ISS and also the same primary orientation of trimmed horizontal. Because of the mass properties, the GSS requires much higher rate of attitude control propellant consumption in the worst-case inertial orientation than that of the ISS.

The attitude and rate performance capabilities of the GSS are the same as those for the ISS. These GSS capabilities are listed as the following:

Attitude pointing error	+0.25 deg
Stability - rate control (rigid body rate)	+0.005 deg/sec
Attitude reference data	+0.02 deg
Rate reference data	+0.001 deg/sec

4.7.3.4.2 GSS Equipment Location and Additional Requirements

The second Power/Subsystems Module, which is for the GSS, has the same thrustor and CMG complement as the first Power/Subsystems Module. The basic difference in the Power/Subsystem Modules is the second Power/ Subsystems Module does not contain the attitude reference assemblies or the interface electronics assemblies of the GNC subsystem.

The second Power/Subsystems Module requires a data bus terminal for an electrical connection to the actuator control electronics in the first Power/Subsystems Modules. Data bus terminals for this connection will go through both Crew/Operations Modules. This data bus terminal also provides the monitor and control functions for the CMG's to the display/control console in the Crew/Operations Modules.

The high-thrust and low-thrust jets located on the second Power/Subsystems Module require a data bus terminal for connecting to the GNC subsystem in the first Power/Subsystems Module. The GSS uses the thrustors located on both Power/Subsystems Modules for the attitude control functions. The high thrustors in the second Power/Subsystems Module also have the hardwire interface to the GNC Subsystem for the secondary control mode.

The GNC Subsystem provides a rendezvous tracker for the free-flying RAM's during the GSS phase. The rendezvous tracker is located in the second Power/Subsystems Module along with a portable display/control unit for the rendezvous operation.

The laser rendezvous tracker is an optical tracking device which transmits a coherent, parallel light beam in a direction determined by electronic deflectors. This device detects the reflected light beam from the object being tracked when the beam is pointed toward the object. Directional and range information to the object is determined by the scanning laser tracker and the rendezvous commands are generated. The rendezvous sensor is gimbal mounted to increase the field of view potential.

The scanning laser radar within the gimbal mount consists of two modules; the sensor and the supporting electronics. The sensor contains the laser generators, beam steerer, optical amplifier, receiver optics, and scanning optical reflector. The laser transmitter is a diffraction-limited, gallium arsenide semiconductor laser. The beam steering is accomplished by a piezoelectric beam deflector.

The physical characteristics of the rendezvous tracker are given by the following:

Weight - 17.7 kg (39 lb)
Volume - 0.0427 m^3 (1.5 ft^3)
Power - 35 w

Additional software requirements over the ISS are imposed by the GSS. There are twice the number of CMG's as well as the additional free-flyer RAM control equipment. These additional software requirements will be primarily for checkout and the rendezvous operation. The same CMG control laws, as identified for the ISS, are used with the eight CMG configuration.

4.7.3.4.3 Control Actuation Sizing

The major impact of the GSS configuration to the GNC Subsystem is the effect of the attitude control actuation sizing. The mass properties of the largest GSS configuration give inertia properties three times that of the largest ISS configuration. These increased inertia characteristics increase the control actuation requirements. (The mass and inertia characteristics for the ISS and GSS buildup sequences are given in Tables 4.7-11 and 4.7-12 Subsection 4.7.4, Design Analysis and Trade Studies.)

A summary of the control actuation sizing for the GSS is given in Table 4.7-10.

A more detailed breakdown of the control actuation sizing requirements are given in Table 4.7-18 of Section 4.7.4. Of the four orientations considered, the primary orientation, trimmed horizontal has the minimum propellant requirements with a near minimum momentum storage requirement. By using the same number and size of CMG's (improved ATM-4070 N-M-Sec) in the second Power/Subsystem module as in the first, sufficient momentum capacity will be provided for all but the inertial orientation.

These improved ATM CMG's are the selected baseline. This assumes that two of the 10 CMG's are in the standby and the operational CMG's are operating at maximum speed. The inertial orientation momentum storage requirements can be satisfied by operating all 10 CMGs at a maximum speed of 12,000 rpm. For the advanced 6,000 H CMG-type, only five CMGs (four

Table 4.7-10

GSS CONTROL ACTUATOR SIZING

Actuation	Trimmed Horizontal	X-POP/OR	X-POP	Inertial
CMGS n-m-sec (ft-bl-sec)	28,500 (21,000)	34,000 (25,100)	26,300 (19,400)	42,000 (31,000)
Required Propellant* kg/90 days (lbs/90 days)	1,010 (2230)	1,180 (2600)	1,380 (3050)	2,900 (6400)
Biowaste output	1,140 (2520)	1,140 (2520)		
Supplement to biowaste system	Not required	36 (80)	230 (510)	1,750 (3860)

*I_{SP} = 180 sec, max. solar activity, 455 km orbit

operational and one standby) would be required for the GSS momentum requirements. These 5 larger CMGs would provide a weight savings over the 10 smaller baseline CMG's of 170 kg (375 lb). The momentum capacity of the advanced CMG at the maximum speed of 12,000 rpm is 12,200 n-m-sec (9,000 lb-ft-sec).

For the long-term orientation (trimmed, horizontal) the propellant requirement for the orbit-keeping and attitude control function, other than docking operations, can be provided entirely by the biowaste system. Note: These propellant requirements are based on the worst-case atmosphere. This assumes a biowaste output from the nominal 12-man crew of the GSS. Supplement propellant is needed for other orientations as indicated by the table.

4.7.4 Design Analyses and Trade Studies

4.7.4.1 Momentum Storage and Propellant Requirements

The momentum storage and propellant requirements for the attitude control and orbit-keeping functions of the Modular Space Space Station are derived from the physical characteristics of the Station, the disturbance environment, and possible operational constraints imposed on the GNC Subsystem. These requirements are used to define the propellant tankage capacity and the number and size of the CMG's of the GNC Subsystem for the Modular Space Station. The selected CMG's are the improved ATM type, with a maximum angular momentum capacity of 4070 n-m-sec (3000 lb-ft-sec) per CMG.

4.7.4.1.1 Modular Space Station Buildup

The Modular Space Station has highly variable mass and inertia characteristics throughout the mission. A buildup sequence for the Modular Space Station is given in Figures 4.7-17 through 4.7-20. This buildup sequence was derived with certain constraints of module arrangement identified by the experiment timeline for the attached RAM's. These constraints are identified with the buildup sequence as follows.

From the third step of the buildup shown in Figure 4.7-17 the GPL is radially docked to the Crew/Operations Module, which is a permanent position. The remaining two radial docking ports and the end docking port on the crew module are used primarily for logistics docking. The docking ports on the Crew/Operations Module are used in lieu of those on the Power/Subsystems Module to ease the task of unloading the Logistic Modules. For the docking of the Logistic Modules, it is assumed that the incoming Logistic Module does not dock at the same docking port as that used by the outgoing logistic module. Therefore, with two docked Logistic Modules, a third open docking port must be available. This mode of operation reduces the docking operations of the Shuttle.

At step 4 in Figure 4.7-17, there is only one of the three docking ports on the Crew/Operations Module for the Logistic Module placement that is

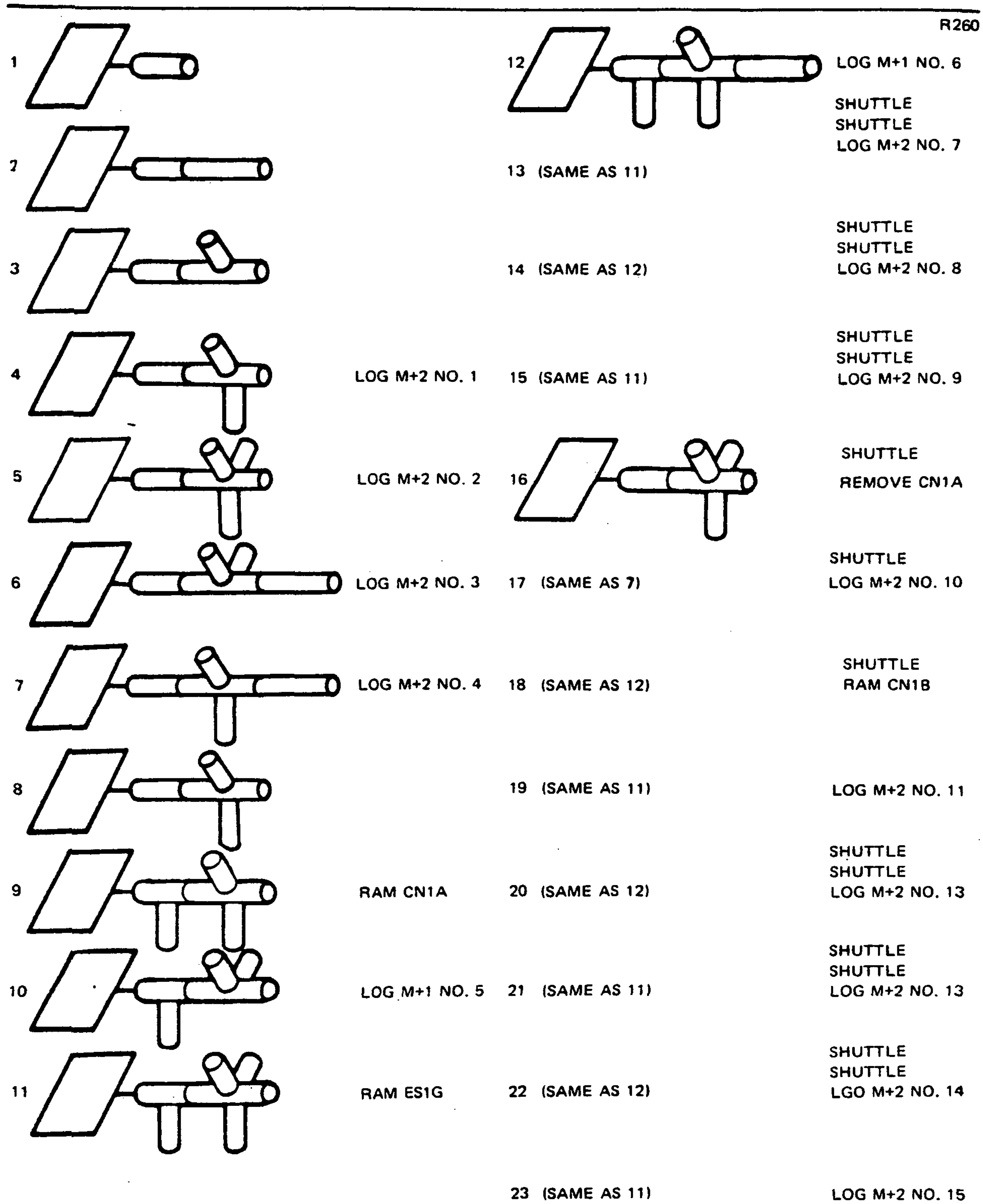


Figure 4.7-17. ISS Phase of Space Station Program

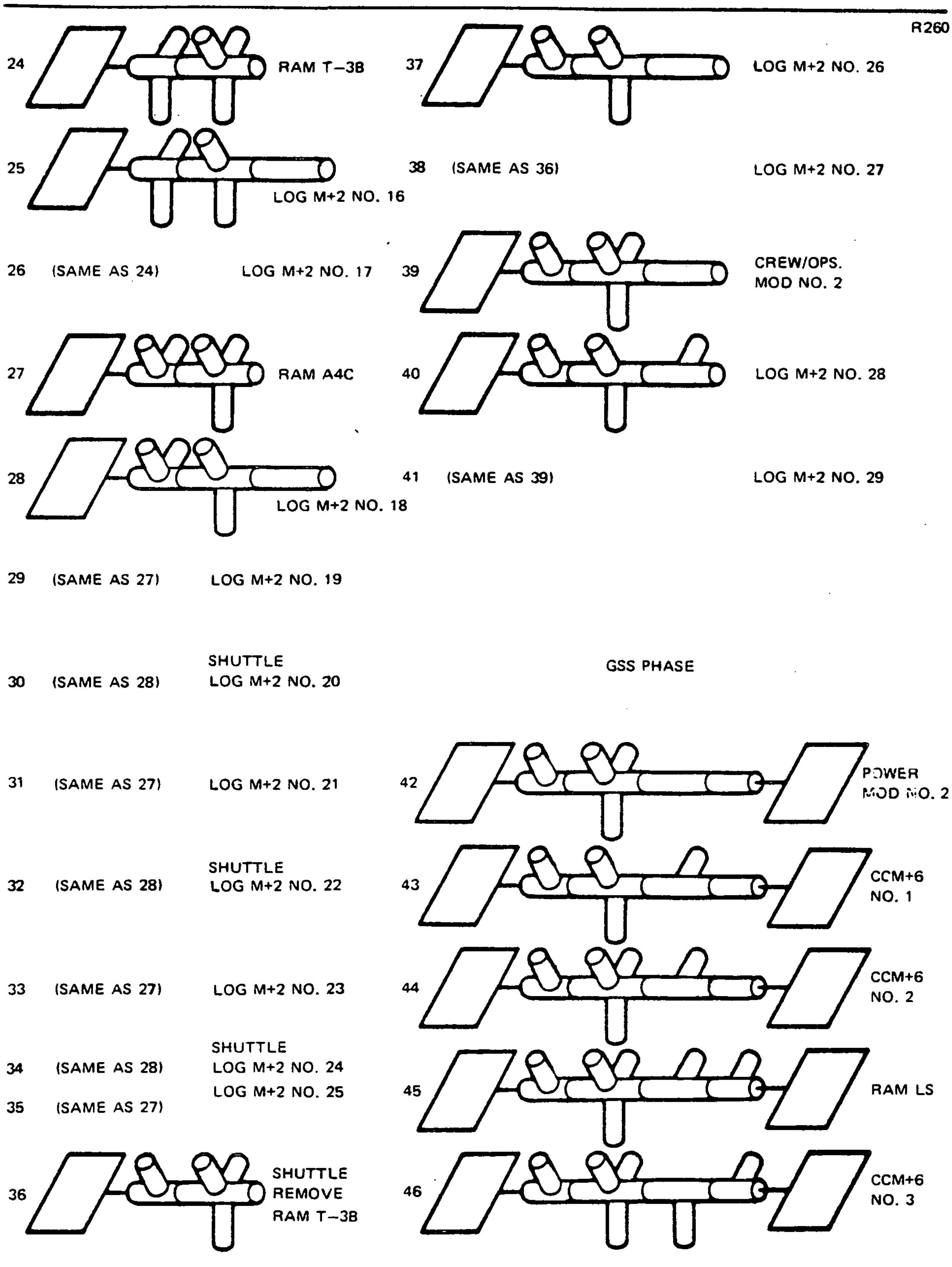


Figure 4.7-18. ISS and GSS Phase of Space Station Program

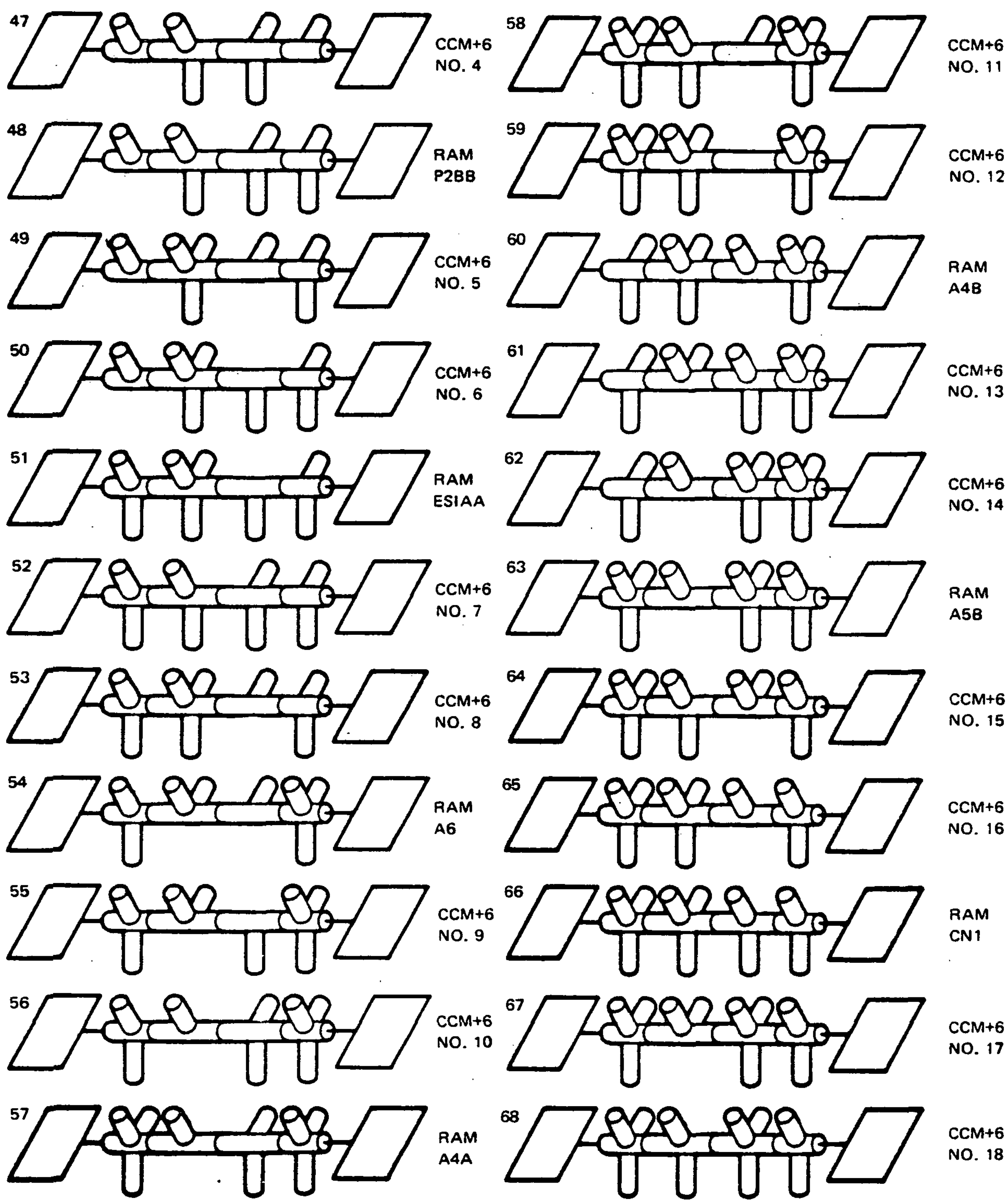


Figure 4.7-19. GSS Phase of Space Station Program

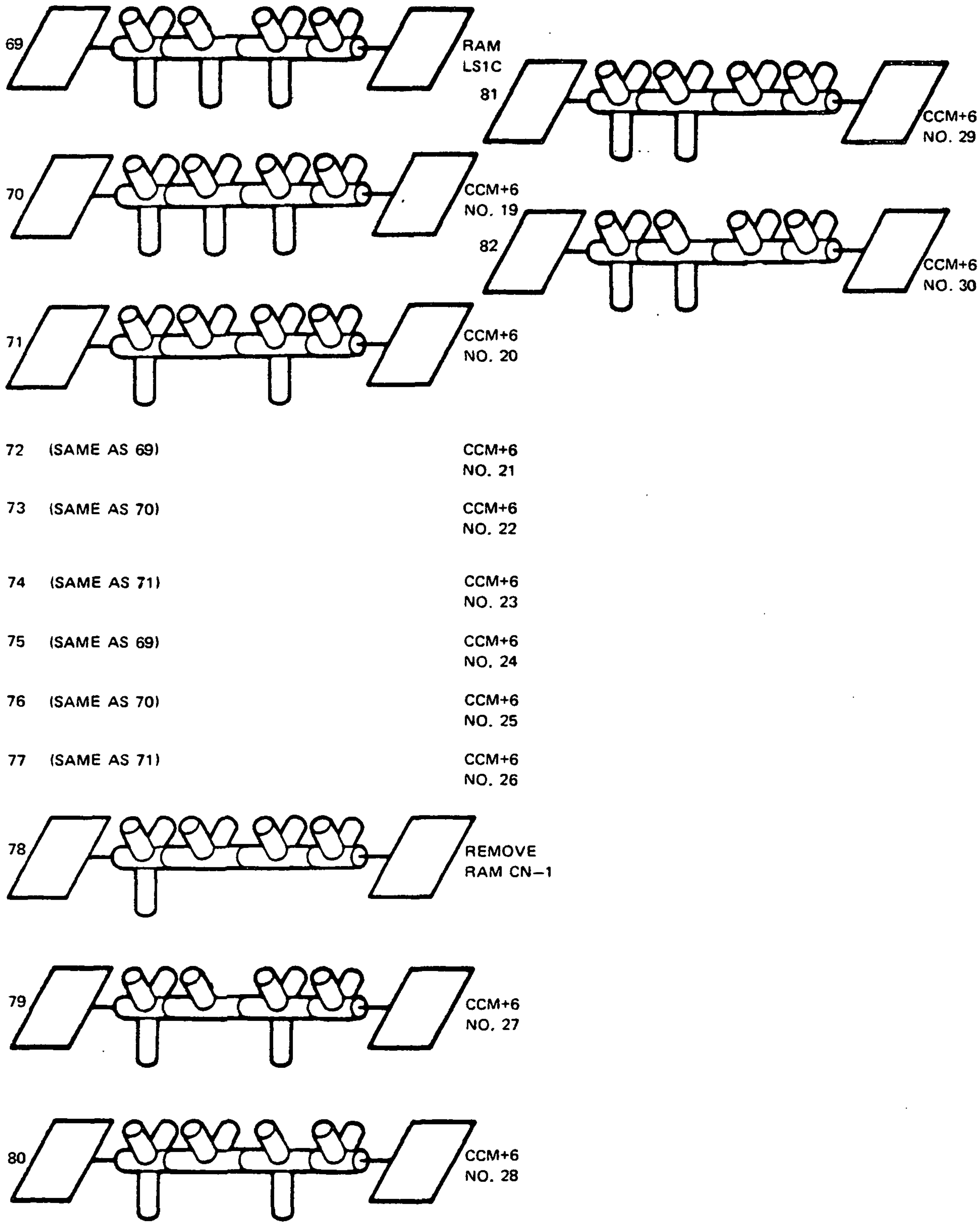


Figure 4.7-20. GSS Phase of Space Station Program

compatible with the experiment module placement in steps 11 and 38. The Earth-Science experiment module at step 11 as well as the Communication, Navigation experiment module desires the Earth-facing position (+Z axis alignment). At step 10, the +Z axis port on the Crew/Operations Module must be vacant for the Earth-Science experiment module. The rotation of the Logistic Modules from steps 11 through 38 produce a vacant end-docking port on the Crew/Operations Module so the second Crew/Operations Module can be end docked at step 39.

On reaching the GSS phase which is after the second Power/Subsystems Module is docked (step 42), the Crew Cargo Modules are docked to the ports that produce the least off-axis CCM shifts and the least principal axis rotation about the pitch or Y-axis. The reason for the constraints of the Y-axis will be identified later. Noting Figures 4.7-19 and 4.7-20, the maximum number of docking ports used during the GSS phase is 10 out of the 12 docking ports.

Figures 4.7-21 shows the axis definition of the Modular Space Station. The Station numbers are in meters and referenced to the docking-port interfaces.

The physical properties of several key configurations during the buildup phase are given in Table 4.7-11 for the ISS and Table 4.7-12 for the GSS. The parameters given are mass, cg Station numbers, moments of inertia, products of inertia, and principal moments of inertia. The moments of inertia and products of inertia are referenced to the axes shown in Figure 4.7-21.

Several of the flight numbers shown in Table 4.7-11 exhibit a large cg offset from the X-axis. In some cases, the cg is not contained within the Station. To combat this large cg shift, thrusters are provided in two locations which provide the necessary negative-and positive-direction torques. These thrustor locations are on the end of the forward Power/Subsystem Module

Table 4.7-11

MASS AND INERTIA CHARACTERISTICS
(ISS BUILDUP)

Flight No.	Mass (kgx10-3)	cg Station (m)			Inertia ($kg\text{-}m^2 x 10^{-6}$)								
		X	Y	Z	I_x	I_y	I_z	I_{xy}	I_{xz}	I_{yz}	*I_{xp}	*I_{yp}	*I_{zp}
1	9.1	108.2	0	0	0.038	0.398	0.48	0	0	0	0.038	0.398	0.48
2	18.2	100.3	0	0	0.077	1.86	1.94	0	0	0	0.077	1.86	1.94
3	27.3	97.8	-3.18	-1.68	0.77	2.02	2.39	0.438	0.228	0.217	0.597	2.1	2.5
4	36.4	97	-2.54	2.19	3.03	4.45	2.76	0.59	-0.28	0.38	2.97	4.75	2.52
5	45.4	96.2	-0.92	-0.39	1.88	2.89	3.26	0.202	0.071	0.148	1.85	2.83	3.35
6	54.5	93.3	-0.92	-1.6	1.69	5.63	6.66	0.18	0.041	0.45	1.7	5.58	6.69
10	59	97.8	-0.93	0.82	4.32	6.03	4.72	0.34	1.38	0.304	3.04	6.1	5.89
11	65.7	97.4	-0.79	3.1	6.36	8.2	4.73	0.255	0.713	0.462	4.4	8.26	6.62
12	65.7	94.8	-1.75	3.66	5.49	11.22	7.58	0.23	1.15	0.94	4.92	11.4	7.93
25	75	96.1	0.1	2.28	8.43	13.0	10.2	-0.484	0.315	1.35	8.06	13.45	10.1
27	75	98.1	0.01	-0.58	7.73	7.65	7.37	0.658	-0.914	-0.418	6.22	8.57	7.95
28	75	95.8	-0.71	-0.01	7.26	11.26	10.66	0.847	-1.105	-0.23	6.75	11.4	11.0

* Principal moments of inertia

R260

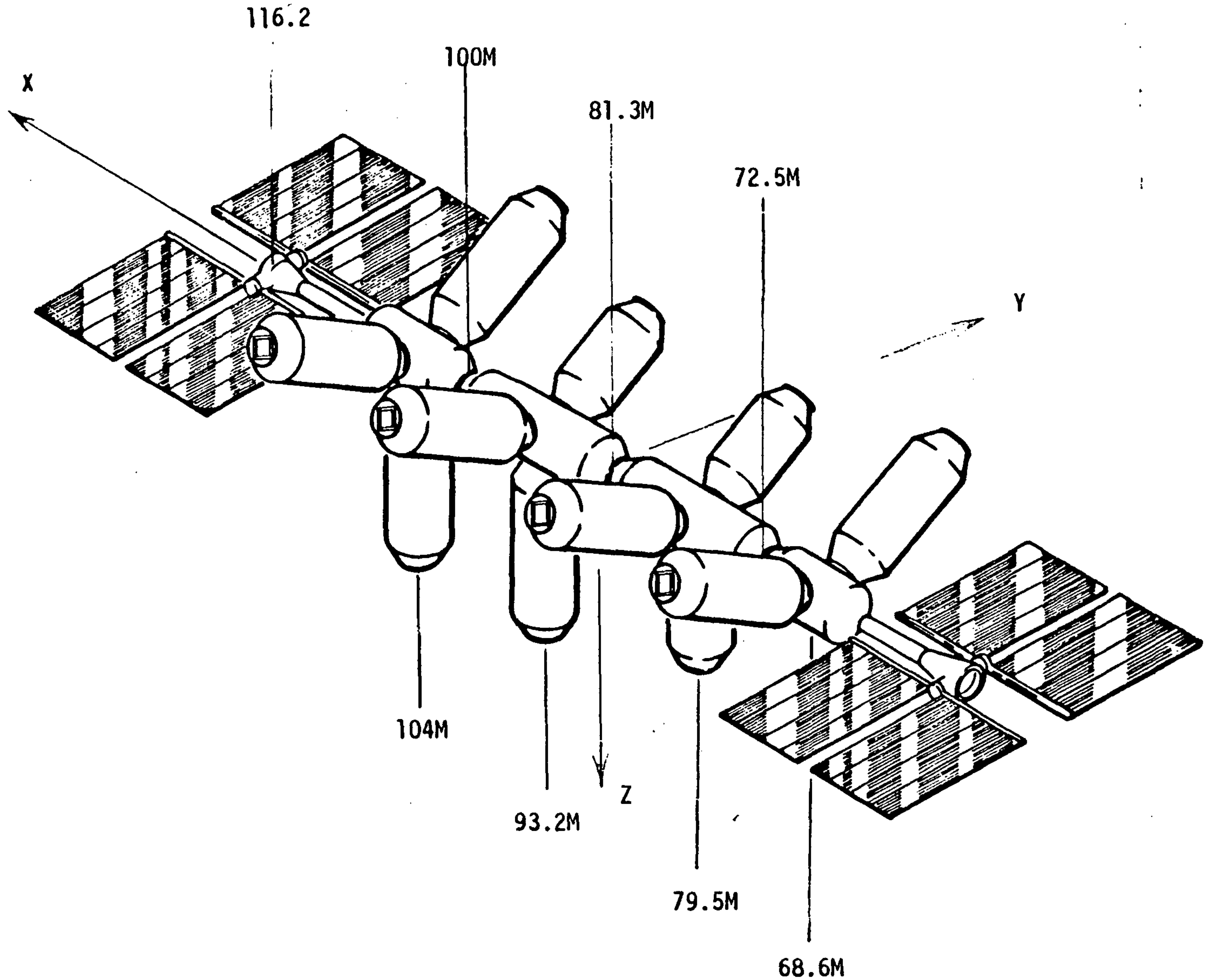


Figure 4.7-21. Coordinate Axes and Station Numbers

Table 4.7-12

MASS AND INERTIA CHARACTERISTICS (GSS BUILDUP)

Flight No.	Mass ($kg \times 10^{-3}$)	cg Station (m)			Inertia ($kg\text{-}m^2 \times 10^{-6}$)								
		X	Y	Z	I_x	I_y	I_z	I_{xy}	I_{xz}	I_{yz}	*I_{xp}	*I_{yp}	*I_{zp}
37	65.7	94.8	-3.34	0.97	4.52	9.93	7.81	0.896	-0.467	-0.33	4.41	10.27	7.58
39	77	94.6	0.895	-0.1	5.38	10.6	9.85	-0.65	-0.73	1.26	5.05	11	9.7
40	77	92.5	-1.48	0.36	5.37	12.4	10.8	0.572	0.147	-1.54	5.04	12.9	10.6
42	88.5	90.6	0.79	-0.076	5.71	20.1	19.3	-0.657	-0.76	1.53	5.6	20.4	19.2
44	100	89.3	-0.5	-0.076	6.06	21.1	19.8	0.33	0.102	-1.67	5.88	21.4	19.7
45	109	87.6	0.63	-0.71	8.35	25.3	25.2	-0.57	1.44	-4.1	7.28	26.3	25.2
52	127	86.1	0.05	2.06	11.1	34.4	30.8	-0.44	1.95	-4.42	10.1	35.2	31
54	127	85.7	0	-0.33	9.6	34.3	34.1	-0.33	1.79	-2.8	9.15	34.6	34.2
61	136	85.2	0.41	-0.076	11.95	35.8	35.9	-1.02	0.72	1.4	11.9	37	34.8
66	145	88	-0.89	1.27	12.8	38	34.8	0.335	-0.38	2.65	12.5	38.4	34.7
69	145	86.9	-0.025	0.13	14.5	38.2	36.9	-0.78	3.38	-0.58	14	38.6	37

*Principal moments of inertia

and the aft end of the Crew/Operations Module which is end-docked to the Power/Subsystems Module. The thrustor locations and individual thrustor functions are defined in subsection 4. 8, Propulsion Subsystem.

Table 4. 7-13 presents the angles of rotation between the body axis and its respective principal axis. The ISS configurations (Nos. 3 to 37) exhibit larger rotation angles of the principal axes than the GSS configuration (Nos. 40 to 66). The effect of this principal axis rotation will be identified in a following section.

A configuration timeline of a few of the most predominate configurations during the ISS phase are shown in Figure 4. 7-22. The nominal time period for the unmanned phase is 90 days with a maximum period of 120 days. During the unmanned phase, the attitude control is provided by the high-thrust reaction jets. The orientation of the first two modules (Power/ Subsystems and Crew/Operations Modules) aligns the roll or X-axis to the minus orbital velocity vector, therefore solar panels trailing, and the Z-axis towards the Earth along the vertical. Since the electrical power requirements are small during the unmanned phase, the solar panels are set to a minimum drag position, but positioned about the X-axis to obtain the necessary power sustaining requirements.

Table 4. 7-14 presents some of the ISS and GSS sizing parameters. The C_DA of the configuration without the solar panels is given for several flight numbers in the X-horizontal and X-POP/OR orientations. The effect drag of the solar panels is also shown and is based on a solar panel area of 510 m^2 (5, 500 ft^2).

At the time the GPL or third module is docked to the configuration, the principal axes and not the geometric axes are aligned to the same Earth-centered reference. The solar panels are still in the trailing position for this condition with the option to feather the panels if desired.

Table 4.7-13

PRINCIPAL AXES ROTATION

Flight No.	X_p-Axis Rotation From X-Axis (degrees)	Y_p-Axis Rotation From Y-Axis (degrees)	Z_p-Axis Rotation From X-Axis (degrees)
3	20.4	30.7	23.9
4	25.2	23.4	23.9
10	41.9	18.5	46.1
11	21.4	8.2	22.2
12	25.7	13.4	28.8
25	15.6	18.3	10.8
27	36.8	35.9	49.3
28	19.9	11.3	16.2
37	10.3	19.9	22
40	11.8	19.6	15.7
43	7.2	17.5	15.9
45	13.8	17.5	13.9
52	11.8	10.4	5.7
66	6	8.4	5.9

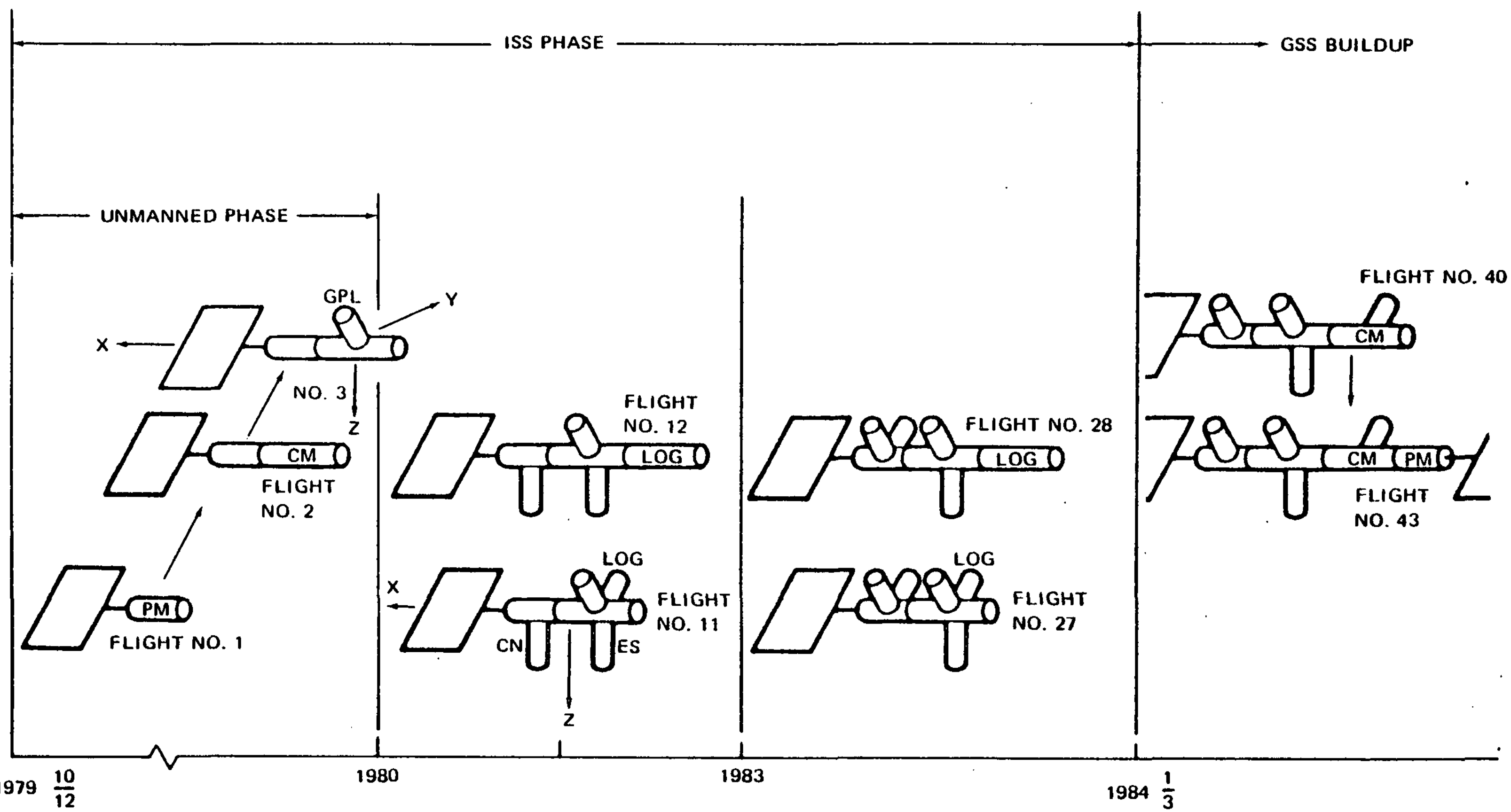


Figure 4.7-22. Configuration Timeline

Table 4.7-14

ISS AND GSS SIZING PARAMETERS

Flight	Body Drag C_DA (m^2) X-Horizontal	Body Drag C_DA (m^2) X-POP/OR	Solar Panel Effective Drag C_DA (m^2)	Solar Panel Lever Arm (m)
6	452	-	270	21
7	350	-	270	23.8
11	552	-	270	19.8
12	434	544	270	21.3
27	643	670	270	19.1
28	566	827	270	21.1
40	566	827	270	24.6
GSS	1,410	2,740	540	1.2

This alignment of the principal axes rather than the geometric axes reduces the attitude-control propellant consumption.

During the unmanned phase, the orbit-keeping function is eliminated. Initially the Power/Subsystems Module is placed in a 500 km (270 nmi) orbit and the Station is allowed to decay in altitude during the entire unmanned phase. If the solar panels are fully aligned to the sun for the unmanned phase, the Modular Space Station at the end of 120 days would decay from 500 km (270 nmi) to approximately 467 km (252 nmi).

The three basic modules—Power/Subsystems, Crew/Operations, and GPL—can be placed in a 500 km (270 nmi) orbit in a gravity-gradient stabilizing mode for over a period of one year without orbit keeping and not decay below 467 km (246 nmi). This particular gravity-gradient position is for the case of aligning the longitudinal axis of the GPL in the orbit plane in a trailing position and the solar panels feathered. Initially at the 500 km (270 nmi) orbit, the orbit decay for this configuration is 0.045 km (0.024 nmi/day).

After the Station is manned, the CMG's are installed in the Power/ Subsystems Module and they are used for the primary attitude-control actuation. The normal- or long-term orientation of the Station is changed to the trimmed horizontal which aligns the X and Z axes in the orbit plane so that the bias torque about the Y-axis is zero and the X-axis is near the velocity vector. The solar panels are in front with full solar pointing operation.

As indicated by Figure 4.7-22, Flights 11 and 12 are the predominate configurations between 1980 and 1983. The flights alter between these two configurations until Flight 24. For the next year or so, the configurations alter between those shown by Flights 27 and 28.

Flight number 39 is the beginning of the buildup for the GSS configuration. At Flight 43, the configuration has both Power/Subsystem Modules, two Crew/Operations Modules, and two GPL Modules. Considering the GSS phase, the GNC control-actuation sizing must be capable of providing sufficient capacity for Flight 40. It is at Flight 43 that the additional GNC control actuation is installed in the second Power/Subsystems Module.

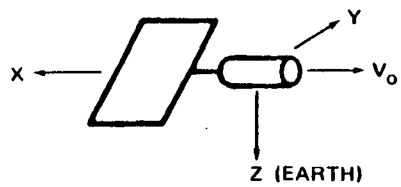
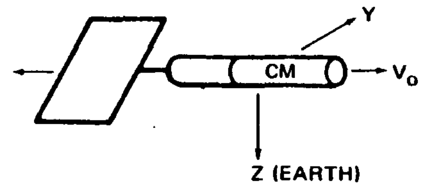
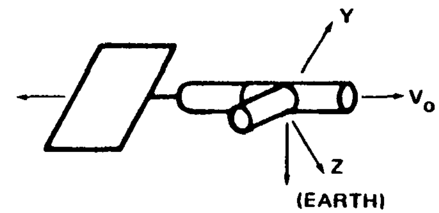
4.7.4.1.2 Unmanned Configuration

The attitude-control propellant requirements for the three unmanned configurations are presented in Table 4.7-15. The specific Earth-centered orientation is the same as previously described and is illustrated in the table. The attitude control propellant and the minimum control impulse is given for each axis. It is noted that the roll requirement is several times that of the pitch and yaw axes. This is due to the relatively small moment of inertia for the roll axis. The 120-day propellant requirement is based on that for each of the three configurations plus the worst-case, 30-day period. The attitude control propellant for the maximum 120-day period is given as 290 kg (640 lb).

The high-thrust propellant tankage has sufficient capacity for this requirement. An optional orientation is the gravity-gradient orientation. This orientation can be implemented to reduce the attitude control propellant requirements if desired.

Table 4.7-15

UNMANNED PHASE MODULAR SPACE STATION

CONFIGURATION / FUNCTION	 30 DAYS	 30 DAYS	 30 DAYS
ATTITUDE CONTROL (EARTH–CENTERED, TRIMMED)	Kg/30 DAYS		
PITCH AXIS (Kg/30 DAYS) (5^o LIMIT CYCLE) MINIMUM CONTROL IMPULSE	6 (13.2) 38 N–M–SEC (28 LB–FT–SEC)	6.5 (14.4) 152 N–M–SEC (112 LB–FT–SEC)	7.4* (16.3) 190 N–M–SEC (140 LB–FT–SEC)
YAW AXIS (Kg/30 DAYS) (SAME AS PITCH)	6 (13.2)	6 (13.2)	6.3* (14)
ROLL AXIS (Kg/30 DAYS) (10^o LIMIT CYCLE) MINIMUM CONTROL IMPULSE	41 (91) 28.5 N–M–SEC (21 LB–FT–SEC)	25.5 (57) 28.5 N–M–SEC (21 LB–FT–SEC)	80* (176) 70 N–M–SEC (52 LB–FT–SEC)
*ASSUMES ALL AXES TRIMMED			
TOTAL PROPELLANT Kg/30 DAYS (LBS/30 DAYS)	53 (117)	38 (184)	99.7 (220)

120 DAY PROPELLANT REQUIREMENT = 290 Kg

4.7.4.1.3 ISS CMG and Propellant Requirements

The CMG and propellant impulse requirements are given for the ISS in several Space Station orientations. These orientations are: X-horizontal which aligns the X-axis to the orbital velocity vector and the Z-axis along the vertical towards the Earth; X-POP/OR which aligns the X-axis with the orbit rate vector and the Z-axis along the vertical towards the Earth; X-POP which is the same as X-POP/OR without the orbit rate vector; and inertial.

These impulse requirements are derived from the solar panel disturbances and the gravity-gradient torques. The atmosphere density model used for the aerodynamic disturbances is generated by the MDAC MZ10 Computer Program. This atmosphere model is based on the $+2\sigma$ 10.7 cm solar flux level and the $+2\sigma$ geomagnetic index. A comparison of the MDAC density model with the aerodynamic density data provided in a NASA, Huntsville Memorandum (S&E-AERO-MM-40-71 dated June 18, 1971) was made and noted that good agreement between the two sources does exist. The average annual density profile throughout the 11-year solar cycle is shown in Figure 4.7-23. This was obtained with the aid of the MZ-10 computed program.

This program is also used to generate an average orbital density profile. This density profile is then curve-fitted and used in several other computer programs to generate the aerodynamic torques and resulting impulse produced by the double-gimballed solar panels. The different programs are for the various Station orientations considered.

Another source of disturbance in the Earth-centered orientations are the product-of-inertia parameters. For aligning geometric axes to the Earth-centered reference, these product of inertia terms produce cyclic and bias gravity-gradient torques. The cyclic components are negated by the CMG's and the bias components produce attitude-control propellant requirements. Unless the Space Station attitude is restricted by the attached RAM's, the Station's attitude will be trimmed to eliminate the bias gravity-gradient torque. The trimming operation for the X-horizontal orientation places the

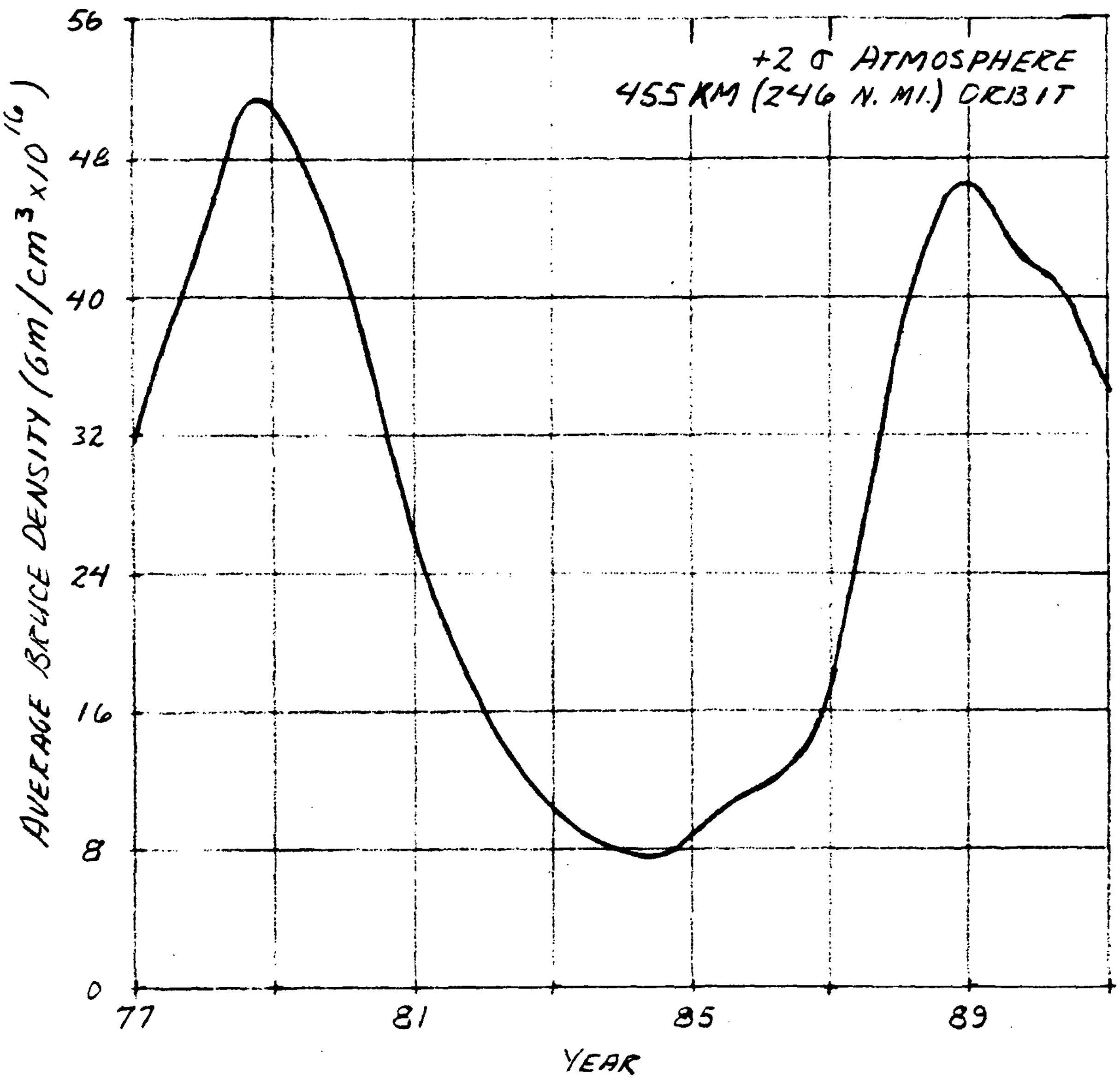


Figure 4.7-23. Annual Average Density

X-principal axis in the horizontal plane and the Z-axis in the orbit plane. This pitch trim angle will vary between 18 to 38 degrees throughout the ISS phase. Using the trim option for trimming about the z-axis, the yaw trim angle will vary between 50 to 78 degrees.

The CMG's are sized to provide sufficient capacity for all orientations of the Space Station and all module placements defined for the ISS buildup. The CMG sizing is based on the cyclic disturbance impulse for each body axis and a controllability factor derived from the minimum threshold of attitude and rate error signals. The controllability factor is based on limiting the CMG gimbal response from the attitude and rate threshold signals. The signal thresholds for the attitude reference system of the Space Station are estimated to be 0.02 degree of attitude and 0.1×10^{-3} deg/sec of rate. With these parameters and assuming the gimbal rates of the CMG's will be proportional to the rate and attitude error signals, the controllability factor is given in Figure 4.7-24. The controllability factor is plotted as a function of the system natural frequency and was derived to limit the CMG gimbal response to an amplitude of 6 degrees. The CMG capacity for the controllability factor times the moment-of-inertia of the particular axis. The selected design point is based on a system natural frequency of 0.17 rad/sec. The system natural frequencies are expected to be less than the 0.17 rad/sec.

The CMG size for each axis of the Space Station is obtained by adding the disturbance impulse of that axis to the respective controllability factor.

Table 4.7-16 shows a breakdown of the momentum storage requirements for each axis of Flight 12 of the ISS in the four different orientations. The controllability factor is that defined in Figure 4.7-24 for the design point. The solar panel disturbance is based on a 1981 solar activity level. In the X-horizontal orientation, the specific solar panel gimbal arrangement produces gimbal reaction torques as well as the panel aerodynamic torques.

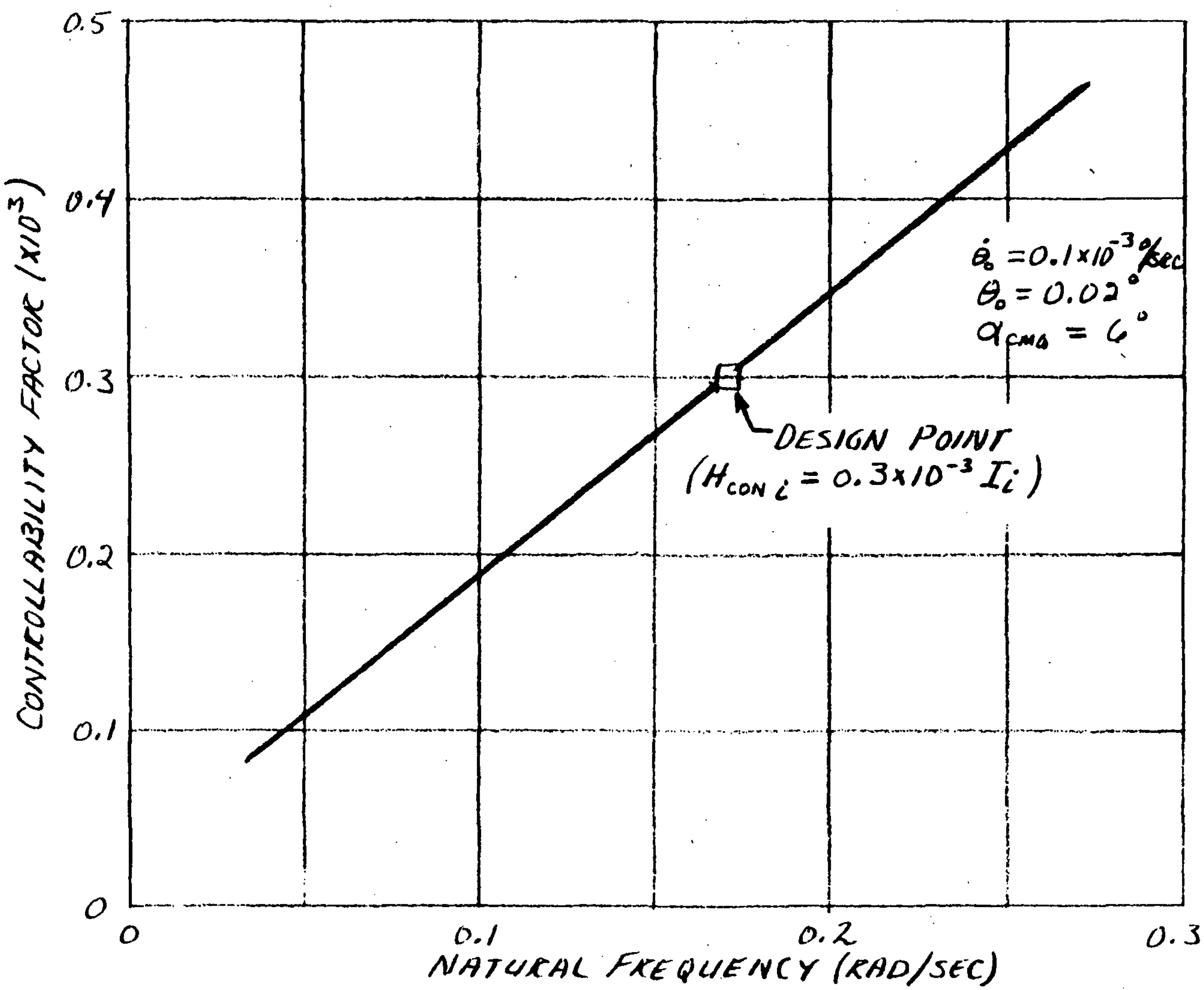


Figure 4.7-24. CMG Controllability Factor

Table 4.7-16

FLIGHT NO. 12 CMG REQUIREMENTS (1981)

Orientation	Angular Momentum Requirements (n-m-sec) Controllability Factor	Solar Panel Disturbance	Gravity Gradient	Total Capacity
X-Horizontal				
H_x (Roll)	1,650	690	4,070	6,410
H_y (Pitch)	3,360	875	0 (32,800)*	4,240
H_z (Yaw)	2,280	740	4,070	7,090
Inertial				
H_x	1,650	0	2,920	4,570
H_y	3,360	1,640	2,520	7,520
H_z	2,280	1,640	5,450	9,370
X-POP/OR				
H_x (Roll)	1,650	0	0 (26,900)*	1,650
H_y (Pitch)	3,360	1,770	5,150	10,280
H_z (Yaw)	2,280	2,270	5,150	9,700
X-POP				
H_x	1,650	0	2,590	4,240
H_y	3,360	2,350	960	6,670
H_z	2,280	0	960	3,240

*Bias component removed by attitude trim or thrustors each orbit period.

The resulting impulse requirements for these two torques with this particular configuration are about the same. For the two Earth-centered orientation, X-horizontal, and X-POP/OR, it is assumed that the resulting gravity-gradient bias impulse is trimmed. From Table 4.7-13, it is noted that the geometric X-axis would have to be pitched down about 25 degrees to eliminate the bias gravity-gradient torques are obtained from the inertial orientation and the inertial orientation maximizes the CMG's.

The CMG requirements for Flight 28 are presented in Table 4.7-17. This particular configuration has larger inertia characteristics than Flight 12 with less solar panel disturbance. The reason for the latter is the reduced solar activity level from 1981 to 1983. It is noted that Flight 12 imposes larger CMG requirements than Flight 28. In the ISS buildup sequence, Flight 12 imposes the largest CMG requirement up through Flight 40. Therefore, Flight 12 will be used to size the CMG's for the ISS configuration.

During the manned portion of the ISS buildup, the maximum propellant requirements occur at Flight 11 and are about 20-percent higher than that of Flight 12.

Table 4.7-18 presents the CMG and propellant requirements for the ISS configuration in four orientations. The trimmed horizontal orientation has been selected as the baseline. This orientation gives a maximum CMG requirement for the roll (X) axis and next to minimum total angular momentum capacity, H_t. The roll and yaw (Z) CMG disturbance components are nearly 180 degrees out of phase while the peak pitch (Y) and yaw components occur at nearly the same time. The total CMG requirement is given as the square root of the sum of the peak pitch component (H_t) squared; the peak yaw component (H_z) squared, and the roll controllability component squared.

Table 4.7-17

FLIGHT NO. 28 CMG REQUIREMENTS (1983)

Orientation	Angular Momentum Requirements (n-m-sec) Controllability Factor	Solar Panel Disturbance	Gravity Gradient	Total Capacity
X-Horizontal				
H_x	2, 180	550	1, 910	4, 640
H_y	3, 380	530	0 (31, 800)*	3, 910
H_z	3, 190	580	2, 060	5, 830
Inertial				
H_x	2, 180	0	350	2, 530
H_y	3, 380	630	3, 520	7, 530
H_z	3, 190	630	3, 920	7, 740
X-POP/OR				
H_x	2, 180	0	0 (6, 580)*	2, 180
H_y	3, 380	680	4, 970	9, 030
H_z	3, 190	880	4, 970	9, 040
X-POP				
H_x	2, 180	0	510	2, 690
H_y	3, 380	900	930	5, 210
H_z	3, 190	0	930	4, 120

*Bias component removed by attitude trim or thrustors each orbit period.

Table 4.7-18

FLIGHT NO. 12 CONTROL ACTUATION SIZING

Function	Orientation			
	Trimmed Horizontal	X-POP/OR	X-POP	Inertial
CMG Capacity N-M-Sec				
H_x	6,410	1,650	4,240	4,570
H_y	4,240	10,280	6,670	7,520
H_z	7,090	9,700	3,240	9,370
H_t in n-m-sec (lb-ft-sec)	9,150 (6750)	11,400 (8400)	8,550 (6,300)	9,700 (7,150)
Propellant: kg/90 days (I_{sp} = 180, 1981 Atmosphere 456 km)				
Orbit Keeping	230	260	290(4)	250
Attitude Control	120(1)	70(2)	33(3)	1,600(5)
Total Propellant-kg/90 days (lbs/90 days)	230 (510)	330 (730)	290 (640)	1,850 (4,080)
Supplement Propellant with Biowaste System	-340 (excess) (750)	-240 (excess) (530)	-280 (excess) (-620)	1,280 (2,820)

NOTES: (1) Assumes a 2.1 m (7 ft) lever arm and pitch attitude trim
(2) Assumes a 13.7 m (45 ft) lever arm and roll attitude trim
(3) Assumes a 13.7 m (45 ft) lever arm and trim and pitch and yaw axes
(4) Assumes a 90-percent efficiency factor
(5) Assumes a 13.7 m (45 ft) lever arm

It is noted that the trimmed horizontal orientation has the minimum attitude control and orbit keeping propellant requirements of the four orientations. Both the maximum and 11-year average propellant requirements are given. The attitude-control propellant requirements for the trimmed horizontal orientation assume the pitch attitude is trimmed to an offset angle (25 degrees) to eliminate the bias pitch attitude control propellant requirements. Without this trim capability, the total attitude control propellant is 1250 kg/90 days. By combining the orbit-keeping and CMG desaturation functions, the total propellant requirement is 230 kg/90 days with the attitude trim function.

For a six-man crew onboard the ISS, the biowaste thrustor system (resistojets) has a propellant capability of 570 kg every 90 days at an I_{sp} of 180 seconds. Therefore, the total orbit-keeping/attitude-control propellant requirements for the ISS can be provided by the biowaste thrustor system.

From Table 4.7-18, the X-POP/OR orientation imposes the largest pitch and yaw CMG requirements of all the orientations. The total CMG momentum capacity (H_t) is the largest of the four orientations. In this orientation, the pitch and yaw angular momentum components are 180 degrees out of phase. The total CMG capacity (H_t) is given as the square root of the sum of the peak pitch (H_y) component squared, and the roll controllability component squared.

The orbit-keeping impulse for the X-POP/OR orientation is 13 percent more than that of the trimmed horizontal orientation. Since the total attitude control propellant is for the pitch axis, the orbit-keeping and attitude-control functions can be combined for this X-POP/OR orientation. The total propellant requirement is then the larger of the two functions, or the orbit-keeping propellant in this case. The biowaste thruster system has sufficient capacity to meet the orbit-keeping/attitude-control propellant requirements for the maximum solar activity period.

The propellant and CMG requirements for the option of trimming the horizontal orientation with a yaw axis rotation will be about the same as those of the X-POP/OR orientation. This is due to the large yaw angular rotation (approaches 80 degrees) requirement.

The X-POP orientation imposes the smallest total angular momentum (H_t) requirement of the four orientations. The components shown in Table I for the X-POP orientation occur at nearly the same time and therefore the total (H_t) is obtained as the square root of the sum of the squares of the three components. The X-POP orientation has the largest orbit-keeping requirement of the four orientations. This requirement is based on the assumption that the orbit keeping is done in a continuous manner using four different thruster locations spaced 90 degrees around the X-axis of the station. Using one thrust location for a 90-degree orbit-angle firing, yields a 90-percent efficiency factor of aligning the thrust vector to the orbital velocity vector. This is compatible with the resistojets' thrust levels. Since the attitude control impulse for the X-POP orientation is divided nearly equal between the pitch and yaw axes, the orbit-keeping and attitude-control functions can be combined as done for the Earth-centered orientations. The total propellant requirement for this case is the attitude-control requirement. Again, the biowaste thruster system has sufficient capacity to meet these propellant requirements. The attitude-control propellants are based on trimming the pitch and the yaw attitudes.

The CMG requirements for the inertial orientation are the second to the largest. The attitude-control propellant for this orientation is the largest of the four orientations. It assumes that each axis is placed in the worst-case bias impulse position for 15 days. The total propellant requirement for the inertial orientation is given as the sum of the orbit-keeping and the attitude-control propellants.

To have the capability to operate in any of the four orientations shown in Table 4.7-18, the CMG system for the ISS configuration must be capable of providing 6,410 n-m-sec in the Z-axis with a total momentum capacity of 11,400 n-m-sec.

The selected CMG's are the improved ATM CMG's that have a maximum angular momentum capacity of 4,070 n-m-sec at 12,000 rpm. A minimum of four operational improved ATM CMG's would be required for the ISS configuration. A fifth CMG in a standby mode would be added so that periodic maintenance and spin bearing replacement could be done with the full CMG configuration in operation.

The baseline arrangement of these four operational CMG's aligns the outer gimbals of the CMG's to the yaw axes of the Station and the inner gimbal of the CMG's to the Station pitch axis. The initial or zero position of the CMG gimbals align the momentum vectors of two CMG's with the +X axis and the other two CMG's with the -X axis. This initial starting position may have a small inner-gimbal angle for each CMG so that the total angular momentum vector has zero components at the initial position. Operating the improved ATM CMG's running at 9,600 rpm would provide a capacity of 6,500 n-m-sec in the roll or X-axis, and 13,000 n-m-sec in the pitch (Y) and yaw (Z) axes. This will provide sufficient momentum storage capacity for the ISS in any orientation.

4.7.4.1.4 GSS, CMG, and Propellant Requirements

For the GSS configurations, the inertial orientation produces the largest control-actuation requirements. Hence, the GSS configuration chosen for the sizing study is one made up of the most modules. This configuration is the same as Flights 66, 67, and 68, shown in Figure 4.7-19 with the exception that one more module is docked to the Power/Subsystems Module. This leaves one docking port open for the exchange of modules.

Table 4.7-19 presents the CMG and propellant requirements for the GSS configuration in the four orientations. The CMG requirements are derived from the same three sources, namely the controllability factor, solar panel, and the product of inertia disturbances. The trimmed horizontal orientation has the second to the smallest total CMG requirements and the smallest propellant requirements. The attitude-control propellant assumes the pitch attitude is trimmed to eliminate the pitch axis propellant requirement. Without the trim, the pitch attitude-control propellant requirement would be 2,040 kg/90 days with a thruster-lever arm of 21.4 m. For the GSS

Table 4.7-19

GSS CONFIGURATION CONTROL ACTUATION SIZING

Function	Orientation			
	Trimmed Horizontal	X-POP/OR	X-POP	Inertial (5)
CMG Capacity n-m-sec				
H_x	12, 100	6, 200	8, 000	7, 100
H_y	17, 200	29, 300	18, 700	35, 200
H_Z	22, 000	28, 800	19, 100	35, 800
H_T: n-m-sec (lb-ft-sec)	28, 500 (21, 000)	34, 000 (25, 000)	26, 300 (19, 400)	42, 000 (31, 000)
Propellant (kg/90 days) 180 I_{sp} and 456 km)				
Orbit Keeping	1, 010	1, 180	1, 320 (4)	1, 090
Attitude Control	195 (1)	14 (2)	1, 380 (3)	1, 800 (3)
Total Propellant: kg/90 days (lb/90 days)	1, 010 (2220)	1, 180 (2600)	1, 380 (3040)	2, 900 (6400)
Supplement Propellant with Biowaste System: kg/90 days for Max.; (lb/90 days)	-130 (excess) (287)	36 (73)	230 (467)	1, 750 (3870)

NOTES: (1) Assumes a 6.5 m (21.4 ft) lever arm with pitch attitude trim of 6.5 degrees.
(2) Assumes a 6.5 m (21.4 ft) lever arm with roll attitude trim of 33 degrees.
(3) Assumes a 6.5 m (21.4 ft) lever arm.
(4) Assumes a 90-percent efficiency factor for orbit-keeping.
(5) Assumes gravity-gradient torques only.

configuration with a 12-man crew, the biowaste propellant output is 1,140 kg/ 90 days at an I_{sp} of 180 seconds which can meet the propellant requirements of the trimmed horizontal orientation during the peak solar activity years.

The CMG and propelland requirements for the X-POP/OR orientation are within 20 percent of those for the trimmed horizontal. Since the GSS configuration has solar panels at both ends, the resulting aerodynamic impulse requirement from these panels is small compared with that for the IC/ISS configuration. The major portion of the CMG requirement is from the controllability factor and the product of inertia disturbances. The attitude-control propellant for the X-POP/OR orientation assumes a roll trim angle of 33 degrees to eliminate the product of inertia contribution. Without this trim angle, the attitude-control propellant would be 850 kg/90 days at a 26.4 m thruster arm. The biowaste thruster system can nearly provide the propellant requirements of the X-POP/OR orientation during the peak solar activity.

The X-POP orientation yields the minimum CMG requirements but only by a few percent. The attitude-control propellant for the X-POP orientation can be eliminated with a pitch attitude trim of 6.7 degrees and a yaw attitude trim of 2 degrees. This eliminates the bias torques produced by the product of inertia terms. The biowaste thruster system must be supplemented with 230 kg of propellant during the peak solar years.

The CMG and attitude control propellant requirements for the inertial orientation are based on gravity-gradient torques only. These impulse requirements for the GSS in the inertial orientation are obtained in the same manner as those for the ISS configuration.

From Table 4.7-19, the maximum roll capacity (12,100 n-m-sec) is about one-third of the maximum pitch or yaw capacity with a maximum vector sum of 42,000 n-m-sec. Using the baseline ISS capacity of five CMG's in each Power/Subsystems Module, the CMG's would have to be operated at their maximum speed of 12,000 rpm to satisfy the GSS requirements. The CMG's have a speed selection between 8,000 and 12,000 rpm. These GSS requirements can also be met with four of the advanced CMG's that have three times the capacity as the improved ATM CMG's (baseline selection) at the same speed. Five of the advanced CMG's would offer a weight savings of 170 kg (375 lb) over 10 of the improved ATM CMG's.

4.7.4.2 Orientation Trade Study

The trimmed horizontal orientation is selected as the long-term orientation. The trimmed horizontal orientation is an Earth-centered orientation and it is defined as that which eliminates the bias gravity-gradient torque about the pitch axis. For the trimmed horizontal orientation, the X and Z axes are in the orbit at a position such that the bias torques about Y-axis are zero with the X-axis near the velocity vector. Generally, the trimmed horizontal orientation has the minimum orbit-keeping requirements and a near minimum momentum storage requirement of all the orientations considered.

The trimmed horizontal orientation provides a good accommodation for the Earth-centered experiments and can adequately accommodate the presently-defined celestial survey experiments. The design approaches of the orientation sensitive subsystems—Electrical Power, EC/LS, GNC, Propulsion, and Communications—while optimized for the trimmed horizontal orientation, do not preclude operation in other orientations. The all-orientation capability of the Space Station is limited only by the onboard-propellant quantity.

The orientation trade study evaluated the experiment and the subsystems impacts for the candidate orientations of the Space Station. The subsystems considered are Electrical Power, EC/LS, GNC, Propulsion, and Communications. Orientations required to accommodate either the Space Station experiment program or certain operational events were identified and their impacts on the station subsystem design approaches were assessed.

The designs for the Electrical Power and the EC/LS Subsystems for the trimmed horizontal orientation also provides for an all-orientation capability. The horizontal orientation imposes the more complex solar gimbal capability of the Electrical Power Subsystem along with the highest thermal capacity for the EC/LS subsystem.

The baseline CMG configuration for the ISS configuration uses four improved ATM CMG's which provide sufficient capacity for all orientations of the

Space Station. This redundant CMG capacity and the supplement high thrust propellant provides the Space Station with an all-orientation capability. Only the worst-case inertial orientation would be time-limited due to the onboard propellant.

The GNC sensor complement, horizon sensors, rate integrating gyros, star sensor, and star trackers provide an all-orientation capability for the Space Station. The star sensor and the gyros provide the Earth-centered reference. The horizon sensors provide the acquisition capability for the Earth-centered reference and it can also provide the Earth-centered reference along with the gyros for limited cases of attitude trim. The star trackers provide the update capability to the gyros for inertial orientations.

The Communications Subsystems has several antenna locations and the antenna switching is designed so that it is compatible with all orientations.

The experiments impose conflicting orientation requirements on the Space Station. The horizontal orientation provides the best accommodation for both Earth- and celestial-pointing experiments since the celestial survey modules contain a gimbal system for the experiment sensor. This gimbal system provides the required isolation and pointing requirements of the celestial experiment in an Earth-centered Station orientation.

4.7.4.3 Control Actuation Trade Study

The trade study for the control actuation selection is based on the ISS configuration in the 3-axis, trimmed horizontal orientation. The trade study evaluates a pure reaction-jet Control System (RJCS) versus a CMG plus RJCS system along with several types of CMGs. The trade study will consider the weight, volume, and power penalties along with the desirable features of one system with respect to the other. The 3-axis trimmed horizon orientation is selected since it yields the minimum disturbance profile which will favor the RJCS.

The disturbance impulse of the Space Station in the 3-axis trimmed horizontal orientation is produced by the aerodynamics and gimbal reaction torques of the solar panels. A computer program simulates the gimbal angle history for the solar pointing and computes the disturbance torques and their resulting impulse. The results of this program have been used in sizing the CMG requirements presented in Subsection 4.7.4.1, Momemtum Storage and Propellant Requirements. The CMG capacity includes the disturbance impulse for the solar panels plus the controllability factor. For this latter term, the CMG capacity is directly proportional to the moment of inertia of the Space Station axes. Five CMGs, four operational and one standby, are selected for the ISS configuration to provide the momentum storage requirements. Each CMG has a maximum momentum storage capacity of 4070 n-m-sec (3,000 lb-ft-sec) and weighs 183 kg (400 lb).

For the case of CMG actuation with the Earth-sentered orientation, the orbit-keeping and CMG-desaturation functions can be combined which reduces the total propellant requirements. These thrustors are aligned such that the thrust is along the velocity vector. The firing on one thrustor will produce a torque (at a lever arm equal to the radius of the vehicle) and a translational velocity. A pure translational velocity can be obtained by firing the thrustors in pairs. For the ISS configuration in the 3 axis trimmed X-horizontal orientation, the total CMG desaturation is provided by the orbit-keeping impulse. This same operation can also be accomplished in the POP/OR orientation with different thrustor arrangements.

The pure RJCS system must provide the disturbance impulse produced by the solar panels. This includes the cyclic and bias disturbance impulse requirements which were computed from the previously mentioned computer program. The computer runs based on a 455 km (246 nmi) orbit altitude at a maximum +2 solar activity. The disturbance impulse variation in terms of orbit altitude and solar activity are obtained by the ratioing of the atmosphere densities.

Figure 4.7-25 is a plot of the accumulated weight for the pure RJCS and the CMG plus RJCS systems from 1979 through 1985. The influence of the solar panels and the solar panel activity is portrayed by the slope of the pure RJCS curve around the year 1982. About this time the solar activity is approaching a minimum which is proportional to the attitude control propellant. The total resupply weight which includes tankage and pressurant is about 1.56 times the pure RJCS curve shown in Figure 4.7-25. Note the Propulsion Subsystem section for a detialed breakdown of these weights.

It is assumed that the ATM GMG's have a lifetime of 1.5 years and that 5 CMG's are required every 1.5 years for the ISS configuration. The weight of these five CMG's is 950 kg (2,100 lb). The improved ATM CMGs have replaceable spin bearings, spin motors, and torquers so that entire unit does not need to be replaced. These improved ATM CMGs yield a weight savings of 2,600 kg (5,700 lb) for the nearly six-year duration.

The CMG's will provide a substantial weight savings over a pure RJCS system in any of the four orientations and Space Station configurations considered. Whether it is cost effective, depends on the weight savings and its cost compared to the development cost of the CMG's. Minor costs of the difference in power, volume, and crew time are also considered in this control actuation trade study. These CMG costs are given in Table 4.7-20. These costs data were provided by the Bendix Corporation.

Table 4.7-20
CMG COSTS

Item	Cost in $ X 10^6	
	ATM	Improved ATM
CMG hardware (5)	1.77	1.92
Launch cost (5 CMG) Power—$7,650/10-w-yr Volume—$1,500/ft^3 Weight—$250/lb	2.17	2.15
Total launch cost	3.94	4.07
Resupply (5 CMGs/ 18 months)	2.0	0.2

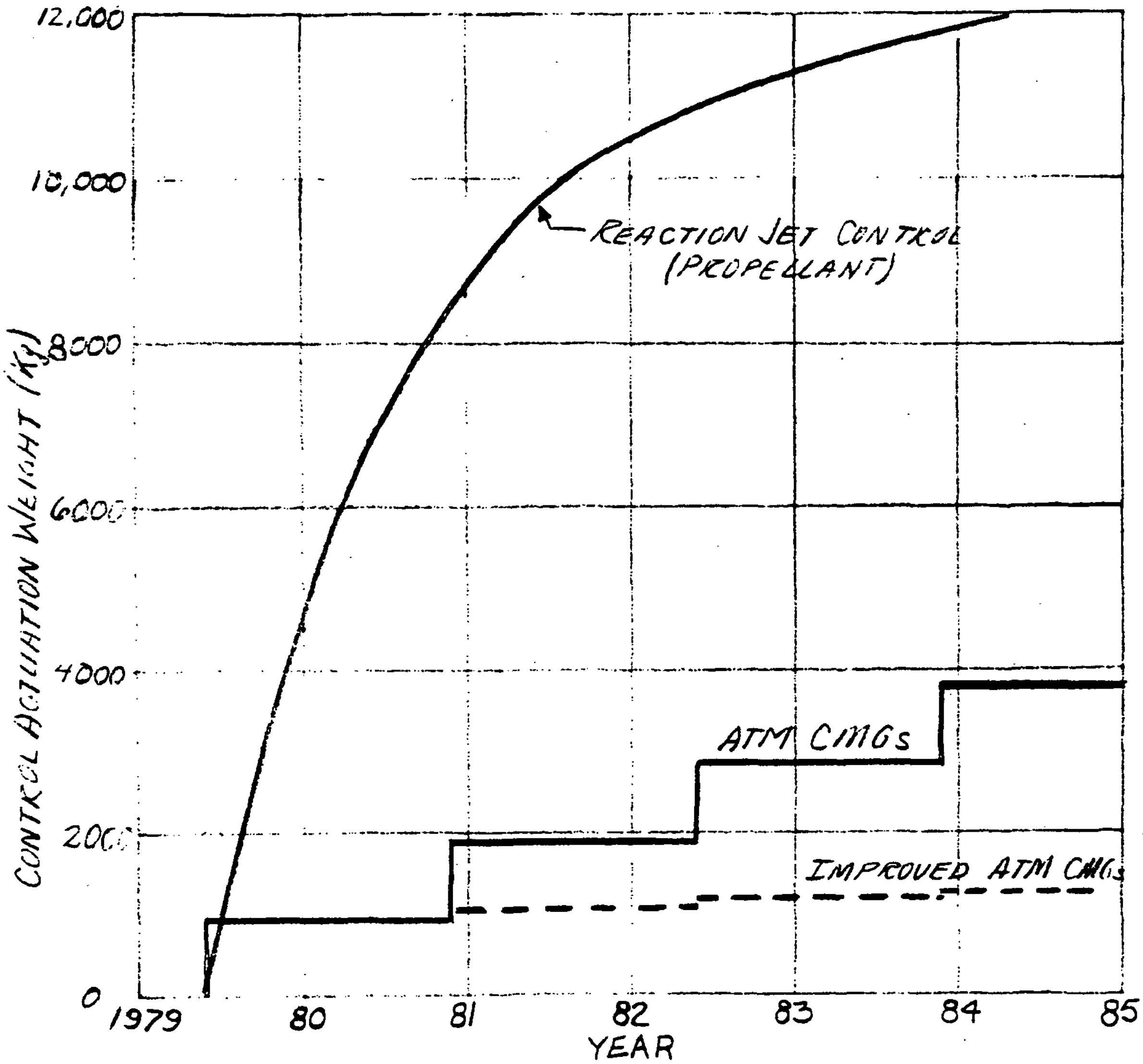


Figure 4.7-25. Control Actuation Trade Study

It is noted that the cost of the Improved ATM CMGs is slightly higher than the cost of the ATM CMG. For a mission duration of 10 years, the Improved ATM CMGs will certainly be more cost effective.

Using the CMG costs given in Table 4.7-20 and the propellant weight given in Figure 4.7-25 multiplied by the 1.56 factor to account for the RJCS resupply weight, a cost comparison of these candidate control actuators is presented in Figure 4.7-26. The results shown in this figure indicate that the improved ATM CMGs is about one-half the cost of the ATM CMGs or the reaction-jet control actuators. Using the Improved version of the CMGs will save over $5 million

Other than the cost somparison, the CMGs offer several advantages in operation. The CMGs provide an increase in attitude and rate stabilization performance and a reduction of the g-level disturbance. The use of CMGs will also decrease the high-thrust contamination. These advantages are very desirable for the onboard optical experiments with sensitive pointing requirements. The use of CMGs will also reduce the maintenance problems associated with the high thrust reaction jets, which is an EVA activity.

4.7.4.4 CMG Control Laws

A description of the operation for the baseline CMG configuration is given by the following.

From the momentum sizing study presented in Subsection 4.7.4.1, Table 4.7-18, it is noted that the roll (x) axis has about two thirds the momentum storage requirements as the pitch (y) or yaw (z) axes. The maximum momentum storage requirements of the roll, pitch, and yaw axes are 6,410, 10,280 and 9,700 n-m-sec, respectively, where the roll axis component is determined from the horizontal orientation and the other two are from the X-POP/OR orientation. The Space Station also has the capability to provide inertial or Earth-centered orientations. In the Earth-centered orientations, a gimbal lock problem of the CMGs will be present unless the CMG's are provided with slip rings eliminating the gimbal lock or the CMGs can be mounted in a particular configuration to eliminate the gimbal lock

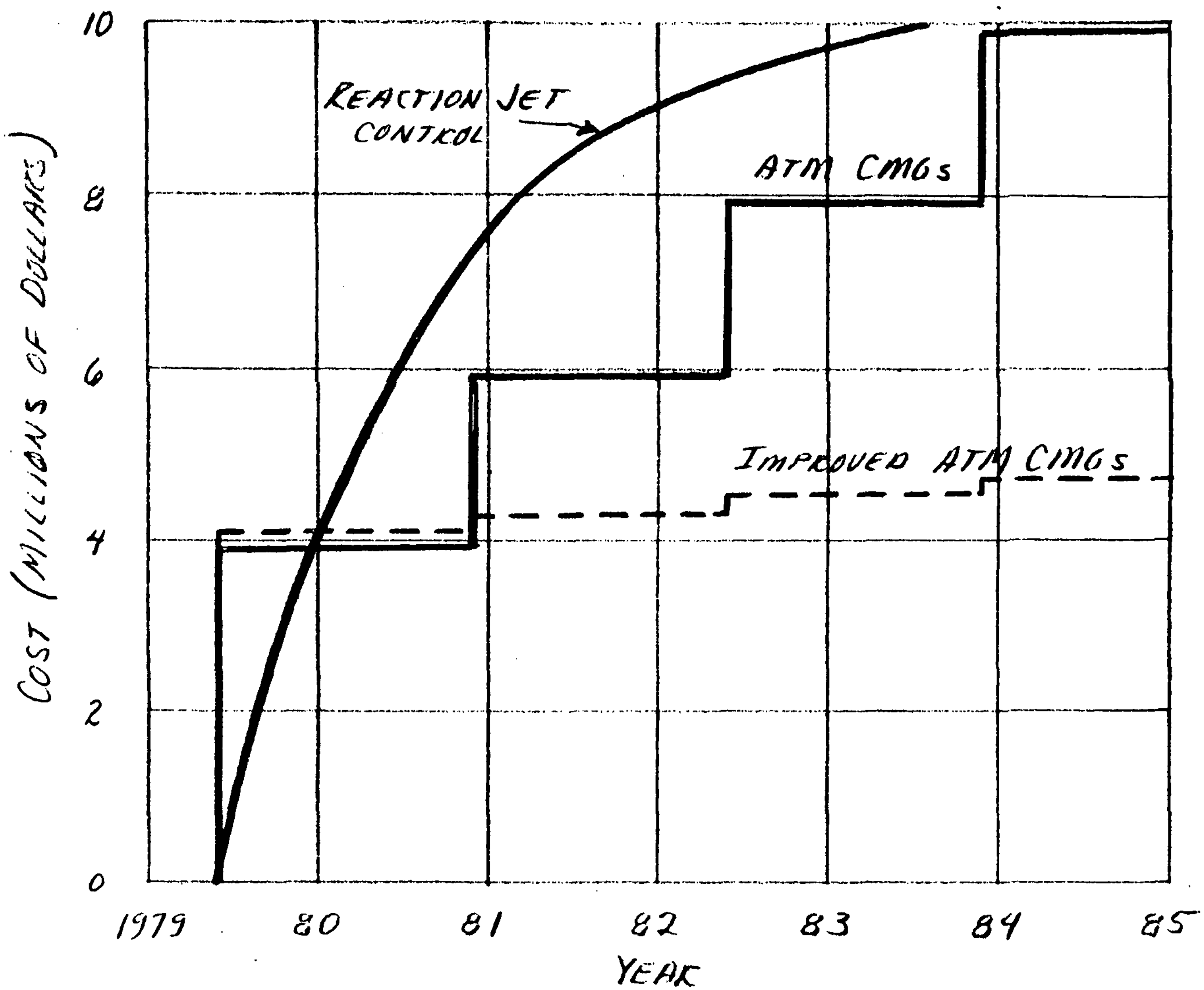


Figure 4.7-26. Control Actuation Cost

problem. Although the selected CMG unit (improved ATM type) has the slip-ring feature for the gimbals, the CMGs are mounted in a configuration which eliminates the need for the slip rings. This configuration also provides some operational advantages which will be identified.

The baseline CMG configuration is shown in Figure 4.7-27. This configuration has four CMGs arranged so that their outer gimbal axes are parallel to the body Z-axis of the Space Station and the inner gimbal axes are parallel to the body gimbal Y-axis. The outer gimbal rates are given by $\dot{\alpha}_o$, $\dot{\alpha}_{o2}$, $\dot{\alpha}_{o3}$ and $\dot{\alpha}_{o4}$ while the inner gimbal rates are given by $\dot{\alpha}_{i1}$, $\dot{\alpha}_{i2}$, $\dot{\alpha}_{i3}$ and $\dot{\alpha}_{i4}$, and the CMG angular momentum is given as H. Two of the CMGs are operating with their momentum vectors in the -X direction. At the initial position of the CMG configuration the inner gimbal of CMG No. 1 is deflected a small angle (17 degrees) in the positive direction and the inner gimbal of CMG No. 2 is deflected 17 degrees in a negative direction. The inner gimbal starting position for CMGs No. 3 and 4 are similar to 1 and 2.

At this position, the net angular momentum vector is zero. Pitch or Y-axis control is obtained gimballing only the outer gimbals of all four CMG's. To obtain a positive Y-axis torque, the outer gimbals of CMG's 1 and 2 are rotated in a positive direction while the outer gimbals of CMG's 3 and 4 are rotated in a negative direction. The total pitch axis momentum storage capacity is equal to the capacity of the four CMG's. The maximum outer gimbal deflections are then ±90 degrees and at this maximum deflection the CMG's require a desaturation torque about the Y-axis.

The inner gimbal torques are used for both the roll (X) and yaw (Z) control. In the long-term horizontal orientation the roll and yaw axes have a momentum exchange due to the gyroscopic effects produced by the orbit rate vector. Therefore, when one axis has a maximum angular momentum requirement the second axis will be near the minimum angular momentum requirement. To obtain a positive Z-axis torque the inner gimbals of CMGs No. 1 and 2 are rotated in a minus direction while the inner gimbals of CMGs No. 3 and 4 are rotated in a positive direction. The total Z-axis capacity is then equal to that of the four CMGs and the maximum inner gimbal deflections are ±90 degrees.

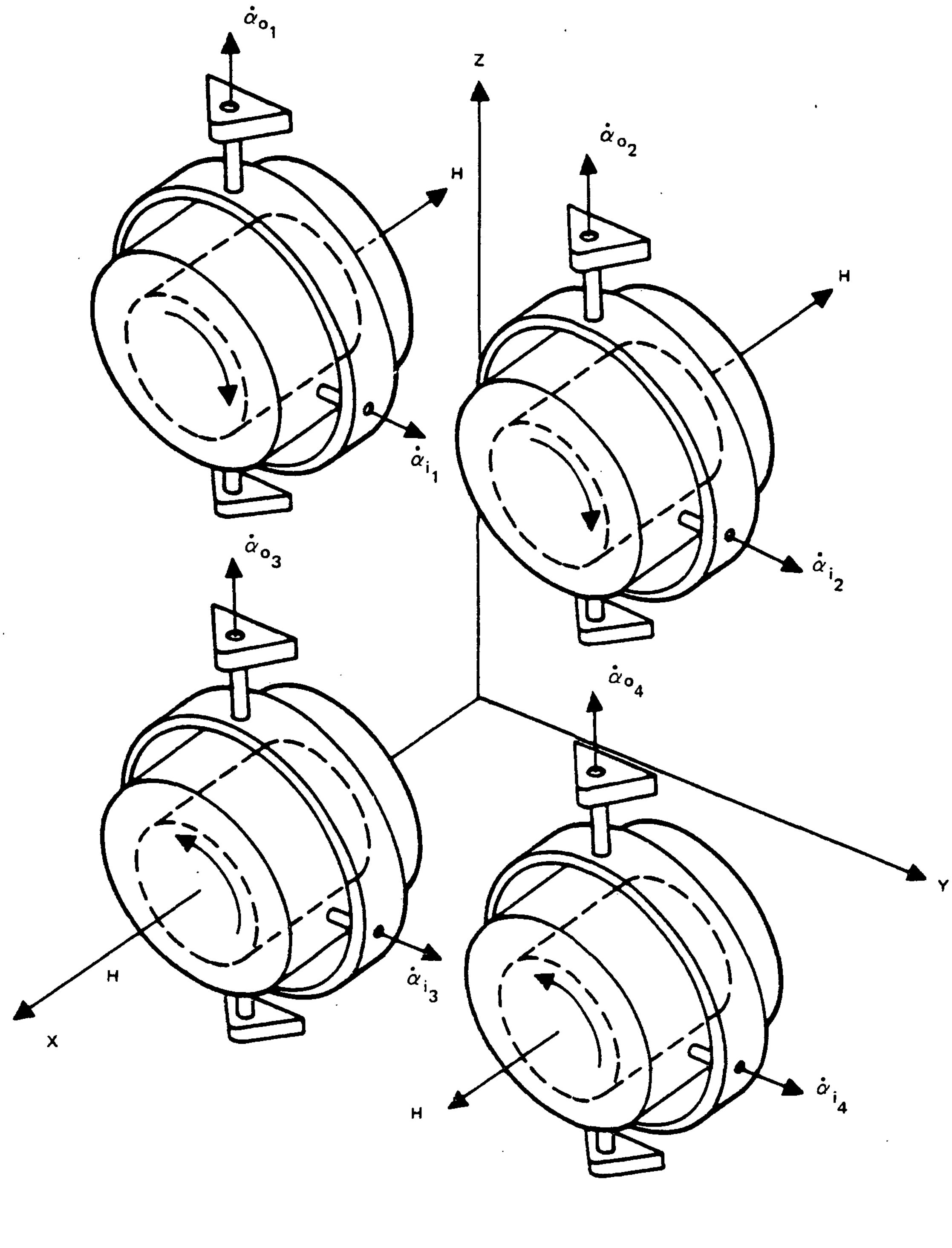


Figure 4.7-27. Selected Mounting Configuration (Antiparallel Configuration)

The X-axis control is obtained by the inner gimbal torques. The relative position of the inner gimbals will determine which CMG inner gimbals are torqued. For a positive X-axis command, if CMGs 3 and 4 (see Figure 4.7-27) have the inner gimbals deflected to give positive and negative Z-axis components, then the inner gimbals of CMGs 3 and 4 are commanded to rotate their momentum vectors away from the X-Y plane. The inner gimbals of CMGs 1 and 2 remain fixed for this X-axis command. Again, for the same positive X-axis command, if all the CMGs have inner gimbal deflections in the same Z-direction, the inner gimbals of CMGs 1 and 2 are commanded to rotate their momentum vectors towards the X-Y plane while the inner gimbals of CMGs 3 and 4 are commanded to rotate their momentum vectors away from the X-Y plane. This same logic is used for a negative X-axis command only the CMG 1 and 2 are used in the case of different Z-axis directions. The total inner gimbal torque commands are the sum of the X-axis command and the Z-axis command.

These CMG gimbal rate commands use an attitude and rate control law and the dynamic cross coupling terms due to body rates and unwanted CMG gimbal rates are removed.

4.7.4.5 Software Definition Task

The purpose of this task is to determine logical breakpoints in the GNC software (excluding the executive functions) that allow identification of several distinct modules of software and to define the functional and performance requirements for each software module with respect to the various mission phases.

The approach used in performing the software definition task is to first identify all the functions to be performed by the GNC software as shown in the functional requirements tree of Figure 4.7-28. These functions are divided into five primary areas: attitude determination, attitude control, navigation, experiment support, and on-orbit checkout and calibration. In the experiment support area, many of the functions are not required until the GSS phase when free-flying experiment modules are used. These functions are identified by the dashed lines in the table. Although it is not

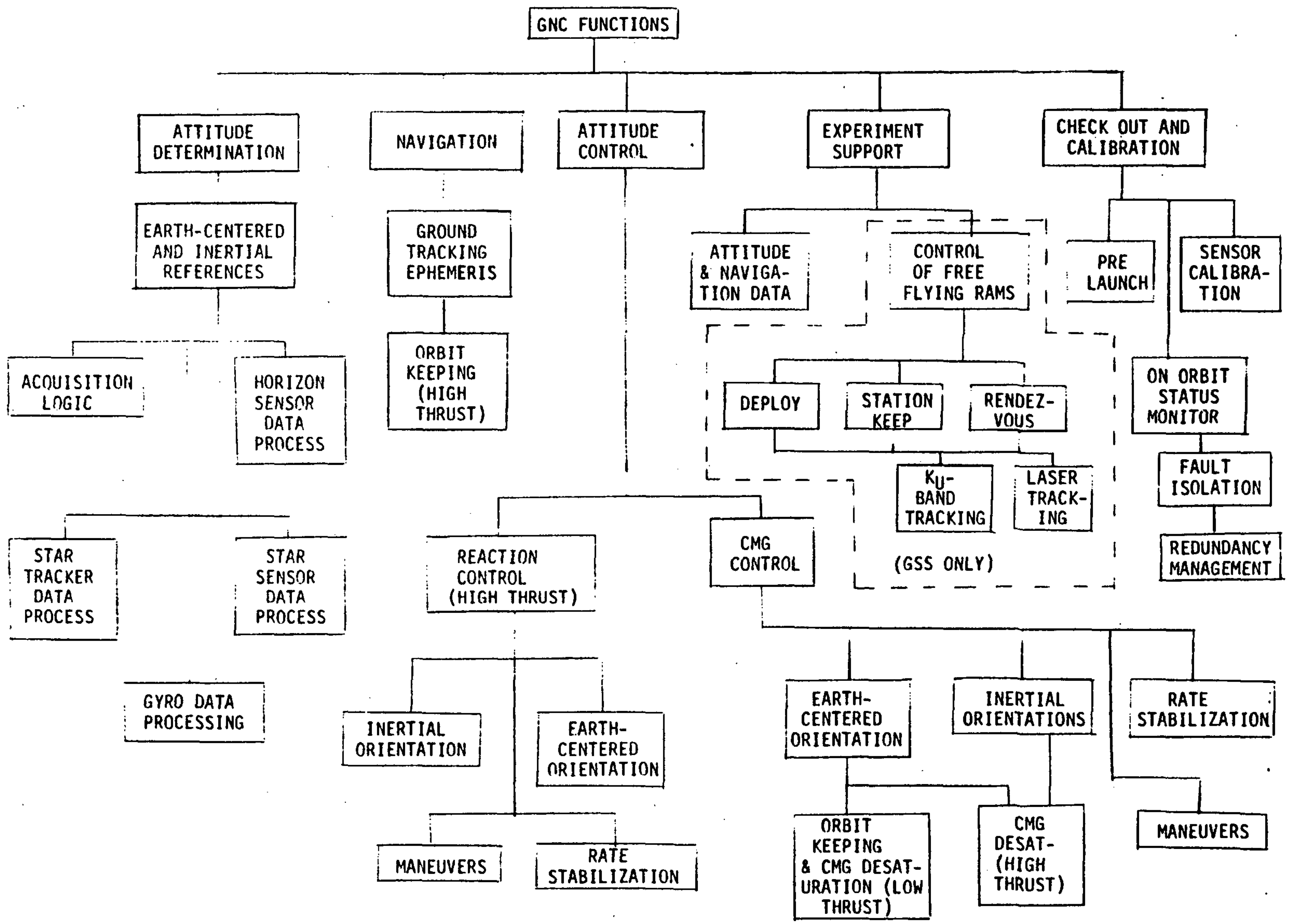


Figure 4.7-28. Functional Requirements Tree

indicated in the table, additional software is also required in the checkout area to handle the added equipment for the GSS phase.

The relationship between the five primary functions with respect to the mission phases is shown in the top-level flow diagram (Figure 4.7-29). The GNC subsystem has few requirements during the prelaunch and boost phase of the mission. During the prelaunch phase, it is desirable to use the onboard checkout software to the greatest extent possible as indicated on the figure. The dashed box indicates the interface with the ground checkout computer which provides the additional software required specifically for ground checkout. This software will be required to provide additional stimuli to those GNC components which rely on the on-orbit environment for checkout stimuli such as the gyros which use the vehicle motion for checkout and the star sensors which rely on actual star transits for checkout. Most of the GNC components such as those which use comparative techniques of several sensors for checkout and fault isolation will require special software for ground checkout.

Similarly, there is no function requirements for the GNC during launch. Prior to launch, the GNC subsystem will be configured for the launch phase as shown in the figure. The function of this software is to assure that the GNC components are in the operating configuration which will best withstand the launch environment. For example, during the launch the gyro spin motors will be running and the gyros will be electronically caged during launch to protect the unit from the launch environment. The major functional requirements for the GNC software of course result from the on-orbit operating requirements.

One of the main purposes of modularizing the GNC software is to permit flexibility to changes which may occur later in the program. To be flexible, the modules must be separable discrete functions which can be identified as an entity in itself. In addition, the modular package should be one which is used repeatedly in several places in the overall software package, for if it is isolated within a particular portion of the software, then changes required of this function are also isolated to a small part of the software package and

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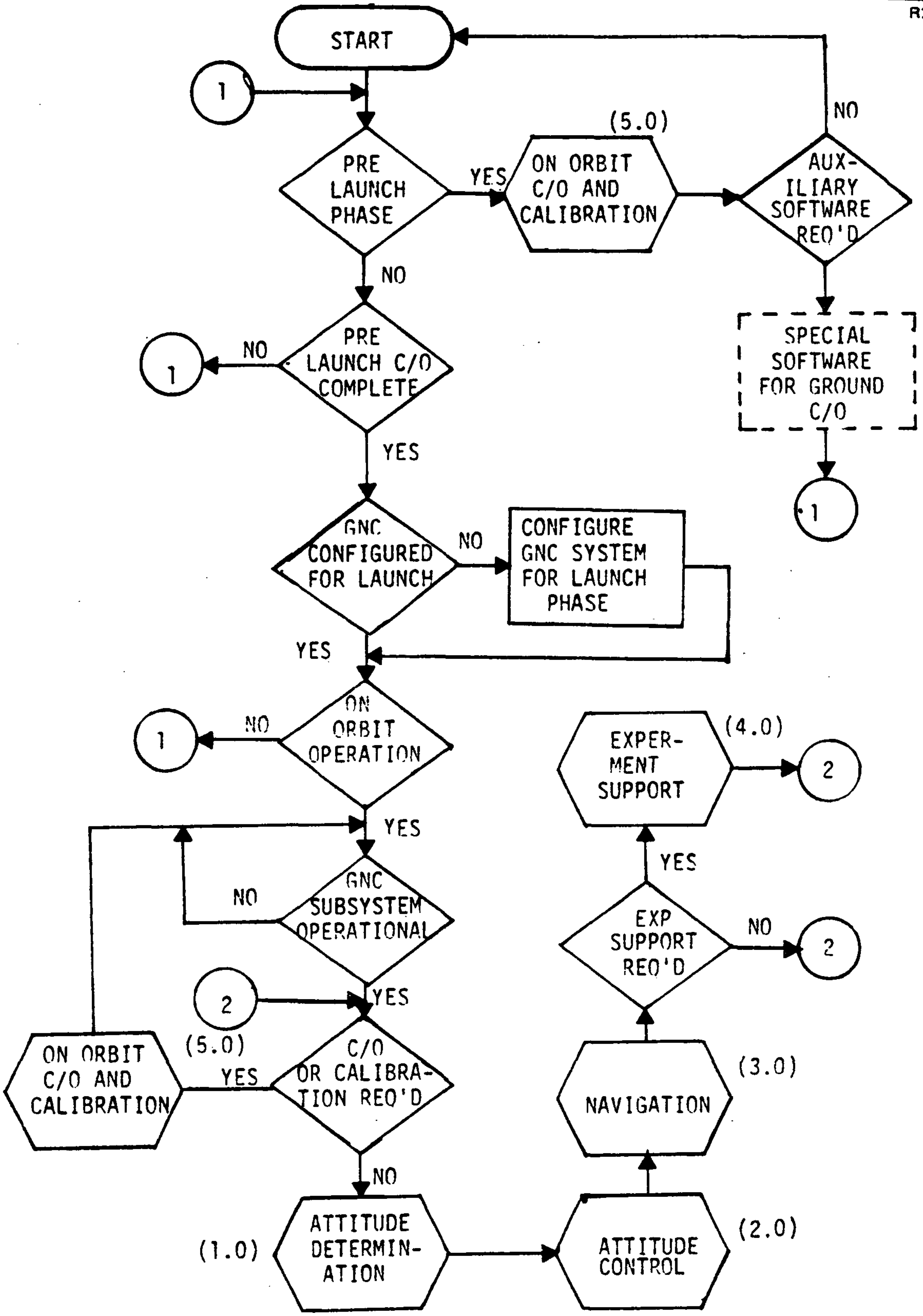

Figure 4.7-29. Top Level Flow

there is no reason to further isolate it through modularization. For this approach to work, the module-to-module interfaces must be maintained so that changes in one module will not affect other modules. Therefore, it is necessary to define the inputs and outputs of each module.

Another factor considered in determining the areas to be sectioned into the various software modules is where changes are likely to occur. In the attitude determination area, for instance, the primary function is the processing of sensor information to determine the vehicle attitude. As a result, some of the software modules are defined according to the particular sensor being processed so that changes in the sensors used could be accommodated by changes to one software module only. Other software changes which can be anticipated in the attitude determination area are improved filtering techniques. In some cases, the filtering will also be associated with processing data from a particular sensor.

Each of the five major functional areas were examined in depth to determine the lower level functional requirements and to determine logical breakpoints in the GNC software. A total of 20 modules have been defined for ISS operation. Four additional modules are required for GSS operation. A breakdown of the number of modules for each major function is summarized in Table 4.7-21. Tables 4.7-22 through -25 summarize the recommended software module breakdown and the functional requirements for each module. The modules are grouped according to the five major areas defined previously. Since the navigation function consists of a single module, it is not included in the table.

4.7.4.6 Attitude Reference Error Analyses

An error analyses of the attitude reference was performed by Honeywell, Inc., for all primary operating modes. The system block diagram of Figure 4.7-30 shows the complement of sensors and electronics necessary for the primary attitude reference modes which include the horizontal reference using the passive star sensors and the inertial and POP reference using the gimbaled star trackers. Performance in the trimmed horizontal, POP/OR and trimmed POP/OR is identical to the horizontal mode. Analysis of the rate measurement error and a solar reference for solar panel acquisition were also performed.

The errors associated with the sensors and navigation data included in the analyses are summarized in Table 4.7-26. The results of a detailed error analysis combining the effects of the individual errors are summarized as follows.

Mode	Performance (1) Attitude (deg)	Rate (deg/sec)
Horizontal (POP/OR)	0.015	0.001
POP	0.011	0.001
Inertial	0.011	0.001
Solar	0.027	0.001

4.7.4.7 Flexible Dynamics Consideration

The control system bandwidth is a parameter which affects the momentum requirements and influences the selection of CMG size for the GNC Subsystem. Although a particular CMG size can provide a range of control bandwidths, it is necessary to determine the region of control system frequencies which bound the Space Station requirements to assure that the selected GNC design and CMG size will meet these requirements.

Table 4.7-21

SOFTWARE MODULE SUMMARY

	Major Functional Area	No. of Software Modules
1.0	Attitude determination	Nine
2.0	Attitude control	Four
3.0	Navigation	One
4.0	Experiment support	One - ISS Four - GSS
5.0	On-orbit checkout	Five - ISS Six - GSS

Table 4.7-22

MODULAR BREAKDOWN OF THE ATTITUDE DETERMINATION FUNCTION

Module		Function
1.1	IRU processing	• Determine body rate and attitude • Provide fault isolation and software modification
1.2	Horizon sensor processing	• Determine local horizontal attitude • Provide fault detection and software modification
1.3	Horizontal acquisition logic	• Acquire local horizontal orientation from any random orientation
1.4	Star tracker processing	• Determine inertial attitude • Control star tracker pointing • Provide for one tracker operation
1.5	Star sensor processing	• Determine inertial attitude
1.6	Attitude filter	• Provide compensation for gyro errors • Attenuate sensor noise
1.7	Attitude transformations	• Transform attitude and rate from horizontal to inertial, POP/OR, POP and vice versa
1.8	Attitude error and rate	• Determine attitude error and rate for control signals
1.9	Subroutines	• Provide general purpose subroutines including: matrix multiply, matrix inversion, square root, trigonometric functions, etc.

Table 4.7-23

MODULAR BREAKDOWN OF THE ATTITUDE CONTROL FUNCTION

Module		Function
2.1	CMG control	• Provide CMG control with and without attitude trim • Provide backup CMG control for three CMG's
2.2	Desaturation (high thrust)	• Provide logic for CMG desaturation using high thrust operation
2.3	Low-thrust control	• Provide simultaneous CMG desaturation and orbit-keeping using low thrust
2.4	High-thrust attitude control	• Provide attitude control using high thrust - Primary during build-up - Backup during ISS and GSS operation

Table 4.7-24

MODULAR BREAKDOWN OF THE EXPERIMENT SUPPORT FUNCTION

Module		Function
4.1	ISS experiment support	• Provide attitude and navigation data to attached and integral experiments
4.2	RAM deployment (GSS)	• Provide initialization of RAM GNC • Acquire RAM with rendezvous sensor • Determine and command ΔV for deployment
4.3	RAM rendezvous (GSS)	• Determine and command ΔV for initial burn • Provide closed loop rendezvous • Control rendezvous tracker
4.4	Station keeping (GSS)	• Determine and command ΔV to station keep up to three RAM's simultaneously

Table 4.7-25

MODULAR BREAKDOWN OF THE ON-ORBIT CHECKOUT AND CALIBRATION FUNCTION

	Module	Function
5.1	IRA fault isolation	• Isolate faults detected by comparing gyro outputs
5.2	Horizon sensor fault isolation and calibration	• Isolate faults detected by comparing horizon sensor outputs • Calibrate horizon sensor using star sensor
5.3	Star sensor fault isolation and calibration	• Isolate star sensor failures • Calibrate using star trackers
5.4	Star tracker fault isolation and calibration	• Isolate star tracker failures • Calibrate using B.I.T.E. and comparing tracker data
5.5	Excessive rate or attitude fault isolation	• End to end check of GNC subsystem except IRA and horizon sensors for fault isolation

Of the factors which influence the choice of control system frequencies, the flexible body dynamics of the Modular Space Station is the most important. Preliminary analyses of the Space Station flexible body dynamics were made to identify a bandwidth region for the body bending frequencies. The McDonnell Douglas Computer Program D402, which is limited to single-axis dynamics, was used to generate the bending data. This bending model simulates the flexible dynamics for the pitch or yaw axes. Torsional modes, simulating the flexible dynamics for the roll axis, were not generated.

The Space Station configuration selected for this analysis is one of the most representative and appears consistently throughout the ISS phase of the Space Station Program (see Configuration 27 in subsection 4.7.4.1). This configuration is shown in Figure 4.7-31. The configuration was modeled as indicated by the EI and distributed-weight diagrams. The side-docked modules are represented by the concentrated weights shown in the figure.

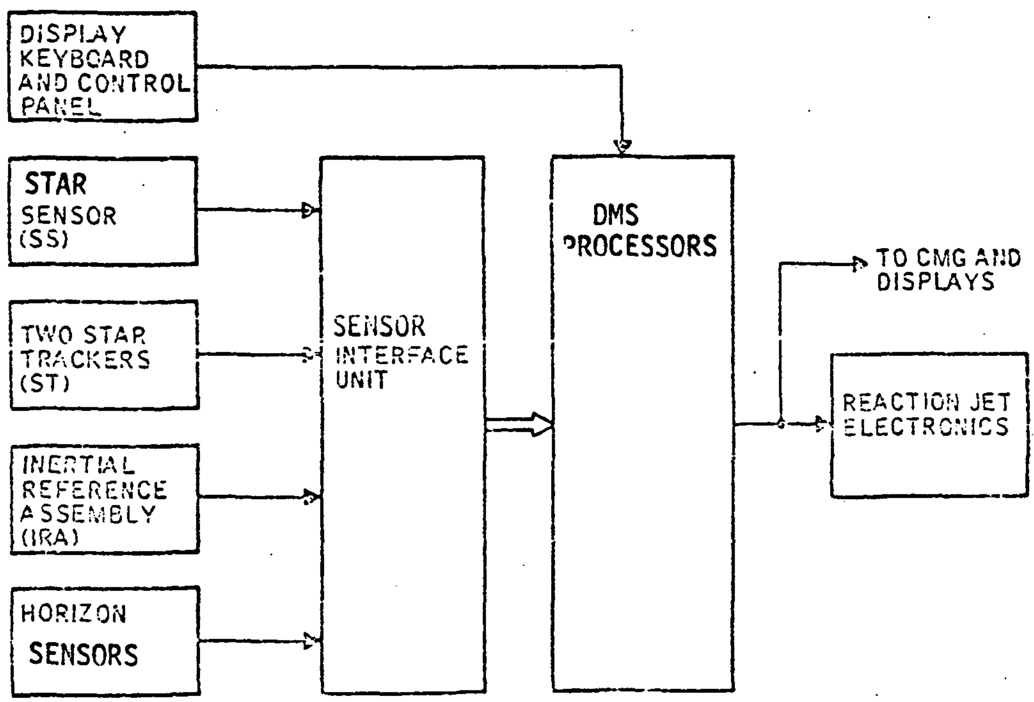

Figure 4.7-30. Attitude Reference Block Diagram

Table 4.7-26

MAGNITUDES OF PRINCIPAL ERRORS

Assembly or Device	Source	Symbol(s)	Magnitude (1-σ for random)	Type of Error	Comment
Inertial Reference Assembly	Constant drift	E_{QR}, E_{PB}, E_{RR}	0.72 deg/hr	Bias	Can be removed by corrections from star trackers or star sensor
	Random noise	E_{QR}, E_{PR}, E_{RR}	0.11 deg/hr	Random	
	Alignment	E_{QR}, E_{QA}, E_{RA}	3 arc sec	Bias	Not significant at low rates
Gimbaled Star Tracker	Optics	E_{STOP}	10 arc sec	Combination	Treated as worst-case bias error in inertial mode analysis
	Gimbal readout	$E_{STG\theta}$	20 arc sec	Random	
	Gimbal readout	$E_{STG\phi}$	20 arc sec	Random	
	Alignment	E_{STA}	8 arc sec	Bias	Magnitude is subject to change; treated as worst-case error in inertial mode analysis
Orbit Ephemeris (Error magnitudes supplied by MDAC except orbit precession which is assumed to be 10 percent of maximum value)	Orbit inclination, i	E_{ORi}	$\frac{0.02}{3}$ deg	Random	Updated at 30-minute interval
	Nodal longitude, Ω	$E_{OR\Omega}$	$\frac{0.02}{3}$ deg	Random	
	Orbit rate W	E_{ORW}	$\frac{5 \times 10^{-6}}{3}$ deg/sec	Random	Orbit angle updated at 30-minute intervals
	Orbit precession Ω	E_{ORP}	$\frac{0.7}{3}$ deg/day	Random	Nodal longitude updated at 30-minute intervals
	Orbit angle α	$E_{OR\alpha}$	$\frac{0.02}{3}$ deg	Random	
Star Vector	Earth velocity	E_{VEL}	20.6 arc sec worst-case 17 arc see in simulation	Bias (if not modeled)	The effect of E_{VEL} can be negated by including in equations

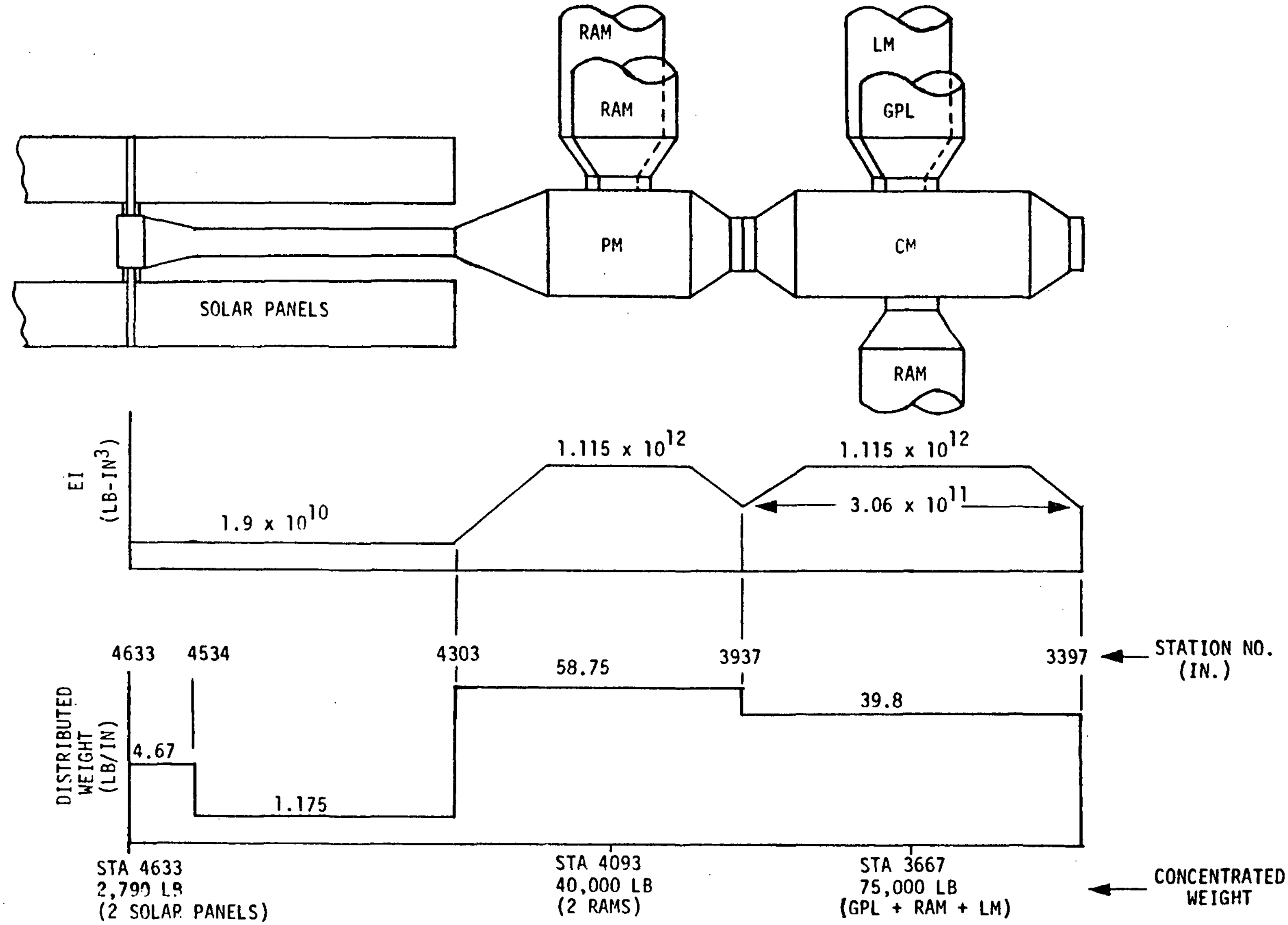

Figure 4.7-31. Dynamics Model (Not to Scale)

Two cases were run for this model. In the first case, the concentrated weights were rigidly attached to the main structure. In the second case, the concentrated weights were attached as spring mass systems with the following frequencies:

Solar panels	0.1 cps
GPL	0.03 cps
All other modules	0.0227 cps

The solar panel frequency was estimated from the data in Reference 4.7-5. The module frequencies are based on a minimum stiffness on the docking port and surrounding structure.

The structural models are used to bound the range of bending frequencies and define a range of control system frequencies. The control system frequency range is selected to minimize the dynamic coupling with the bending frequencies and disturbance frequencies. The highest disturbance frequency that the control system must counteract results from the gravity-gradient torque and is twice orbit rate frequency (0.00036 Hz). The control-system frequency is chosen to be 10 times that of the disturbance frequency. For Space Station application, this factor of 10 provides adequate control response. Separating the control frequencies from the bending frequencies by a factor of ten will provide sufficient decoupling between the frequencies which in turn will yield adequate stability margins. In those cases where the control system frequencies and control signals can be used to obtain the desired stability margins.

Figure 4.7-32 shows the range of the body bending frequencies obtained from the two models. The lowest body bending frequency, 0.052 Hz, results in a minimum-frequency separation between the disturbance frequency, the control frequencies and the bending frequencies. Figure 4.7-32 shows that the pitch and yaw control frequencies can be one-tenth or less than the minimum bending frequency and at the same time the control frequency will be 10 times higher than the disturbance frequency. With the higher expected bending frequencies, the separation of the control frequencies becomes more favorable.

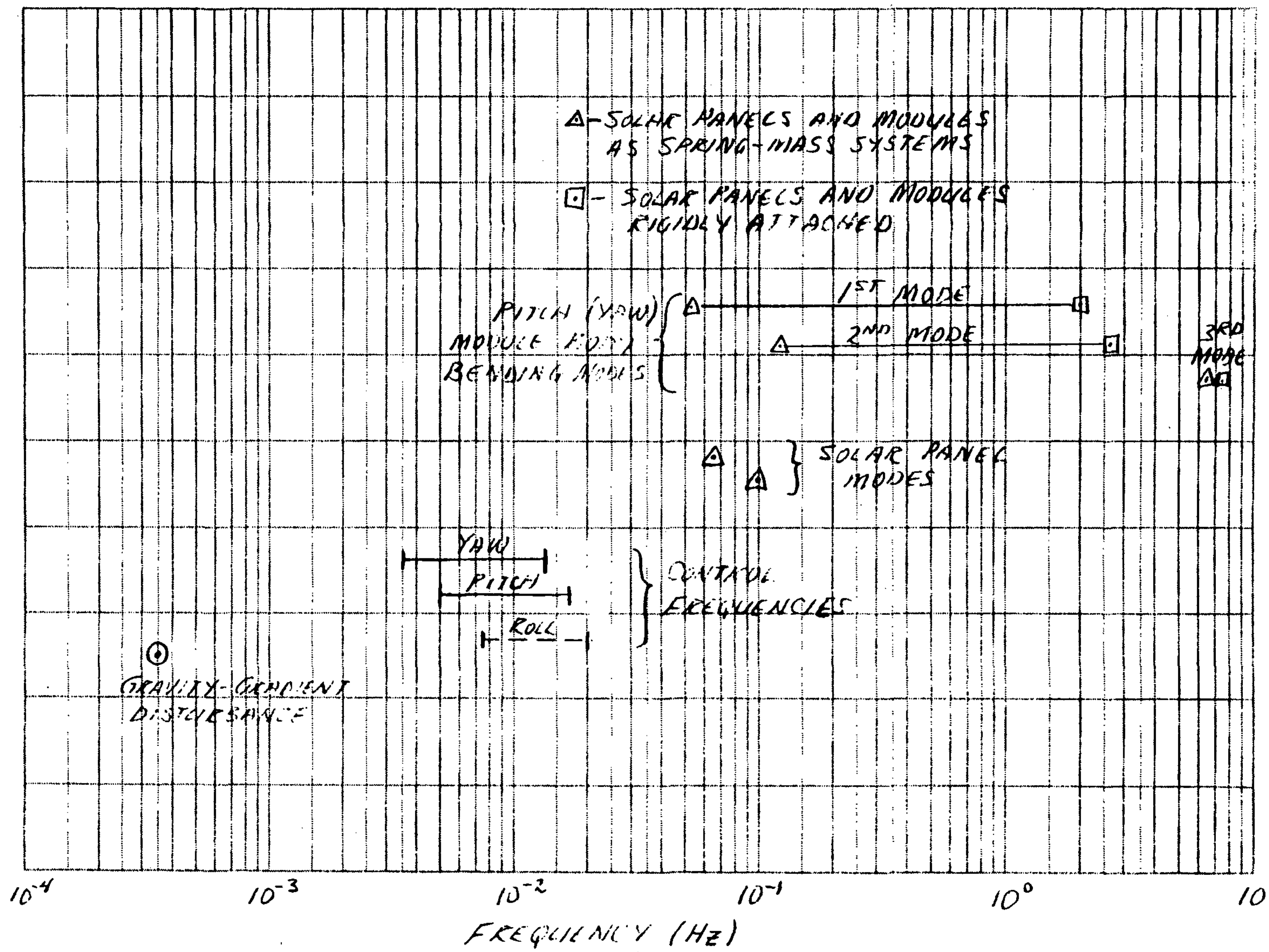

Figure 4.7-32. Bending and Control Frequence Ranges

Although a model for the body torsional mode of the roll axis is not available, it is estimated that these torsional modes will have about the same frequency range as that shown for the pitch or yaw bending modes in Figure 4.7-32. For this case, the roll control frequency range shown in the figure provides adequate stability margin.

The separation between roll, pitch, and yaw control frequencies was selected to reduce the gyroscopic coupling between these control loops. The control-frequency ranges identified in Figure 4.7-32 are compatible with the baseline CMG sizing for the ISS configuration and these control-frequency ranges provide adequate stability margins for the body bending. The control system designed has the capability for large variations in the control-loop frequencies by means of gain changes within the software. This provides the design flexibility to minimize the coupling between the control loops and body bending modes for the varying physical characteristics of the Modular Space Station.

4.7.5 References

4.7-1 "Space Station Definition," MSFC-DRL-160 Line Item 8, Volume V, Book 5, Utility Services, Contract NAS8-25140, McDonnell Douglas Astronautics Company, July 1970

4.7-2 "Experiment Requirements Summary for Modular Space Station and Space Shuttle Orbital Applications and Requirements," Volume II, Martin Marietta, Denver Division, April 1971

4.7-3 "Experiment Requirements Summary for the Space Station Program," Revision II, Martin Marietta, Denver Division, August 1970

4.7-4 "Report on Selected Update Task for Baseline Space Station," MSFC-DRL-231 Line Item 8, Volume I, Selected Update Tasks, Contract NAS8-25140, McDonnell Douglas Astronautics Company, February 1971

4.7-5 "The Study of Dynamic Interactions of Solar Arrays with Space Stations and Development of Array Structural Requirements," Contract NASI-10155, Interim Report, Fairchild Hiller Corp, February 1971

4.8 PROPULSION SUBSYSTEM

4.8.1 Summary

The Modular Space Station Propulsion Subsystem is required to provide for the functions of Attitude control and Shuttle de-docking disturbance control prior to initial manning and CMG installation. After the initial manning and CMG installation, the Propulsion Subsystem is required to provide for:

- Orbit-keeping (orbit altitude maintenance)
- Shuttle de-docking disturbance control
- Maneuvers
- Backup attitude control (CMG-out situation)
- Free-flying RAM propellant resupply (GSS only)

Additional design requirements have been imposed on the system that are of a more qualitative nature, but are necessary to insure a usable and efficient design. The more important of these design requirements are (1) maintainability, (2) redundancy, (3) minimuzed contamination, (4) provision of resupply capability, and (5) use of proven systems, common to other program elements if feasible. Minimized cost, particularly with regard to initial development cost is also a major goal of the design.

To accomplish these functions, a dual-propulsion system was selected. A low-thrust Boiwaste (CO_2) Resistojet Subsystem provides for orbit-keeping and can, if desired, desaturate the CMG's. A high-thrust hydrazine (N_2H_4) monopropellant subsystem provides for the other functions (see Table 4.8-1). These choices were made as a result of trade studies involving high-thrust system propellant selection and the use of only a high-thrust subsystem vs. a combination high-thrust, low-thrust propulsion subsystem.

The high-thrust propellant selection involved a review of the prior Space Station selection, N_2H_4, to determine if the modular concept, in some way, might invalidate the earlier work. Results indicate that N_2H_4 is also preferred for the Modular Station. The prime reasons remain as before; low contamination potential, simplicity, minimum development required, and minimized resupply development.

Table 4.8-1

MODULAR SPACE STATION — P/RCS SUBSYSTEM
DIVISION OF FUNCTIONS

Area	Hi-Thrust (Monopropellant)	Lo-Thrust (Biowaste Resistojet)
Orbit Keeping	Supplementary	Primary
CMG Desaturation	Back-Up	Secondary*
Free Flyer (Docking Disturbances)	Primary	
Shuttle		
Separation Disturbance	Primary	
Docking Disturbance (No Latch)	Primary	

*Primary method is attitude trimming to provide gravity gradient desaturation torques

The high-thrust/low-thrust trade showed that CO_2 resistojets, when combined with the open loop oxygen EC. LS system, minimized total program costs, reduced Space Station effluent, and provided maximum attitude freedom.

The high-thrust subsystem uses refillable, stainless steel, bellows tankage pressurized with regulated GN_2. The propellant and pressurant tanks are located in the unpressurized-but-pressurizable conic section at the forward end of the Power/Subsystems Module. The thrustors are grouped in eight thrustor modules; each module contains five thrustors of 111 N (25 lbf) thrust each. Four modules are located at the forward end of the Power/Subsystems Module and four are located at the aft end of the Crew/Operations Module. This arrangement provides for translation in the +X direction and/or true couples for attitude control in all axes. The overall design contains a high degree of built-in redundancy. In normal day-to-day operations, the system is in a stand-by mode, being used primarily when docking or attitude maneuvers are required.

The low-thrust subsystem is composed of CO_2 compression pumps and storage spheres that feed regulated CO_2 to the biowaste resistojet thrustors. The pumps and tanks are mounted in the same area as the high-thrust tankage. Four 0.111N (0.025 lbf) biowaste thrustors are located in the same eight thrustor modules as the N_2H_4 thrustors noted above. The storage spheres are sized to hold 2 day's output (12.5 Kh/27.6 lbm) of CO_2 from the EC/LS subsystem. As with the hi-thrust system, the design incorporates a high degree of built-in redundancy. The low-thrust subsystem will be used on a nearly continuous basis to provide orbit-keeping impulse and assist, if needed, with Control Moment Gyro (CMG) desaturation.

The location of modules in the ISS phase is illustrated in Figure 4.8-1. The build-up to the GSS Space Station is implemented by the duplication of the propulsion elements in the second Power/Subsystems Module.

4.8.2 Requirements

The basic requirements of the ISS Propulsion System are to maintain orbital altitude and provide attitude control over the operational life of the Space Station. Detailed requirements are presented in Table 4.8-2.

4.8.2.1 Functional Performance Requirements

4.8.2.1.1 Orbit Keeping Requirements

The aerodynamic drag at the selected altitude of 240 to 270 nmi requires that impulse be applied regularly to maintain altitude. The impulse required varies markedly over the 11-year solar activity cycle, as shown in Figure 4.8-2. The figure indicates that there is a relatively small difference in total requirement between the two orientations shown. The small steps in impulse requirement that occur periodically are caused by configuration changes in the Space Station, such as the addition of attached RAM's or basic station modules.

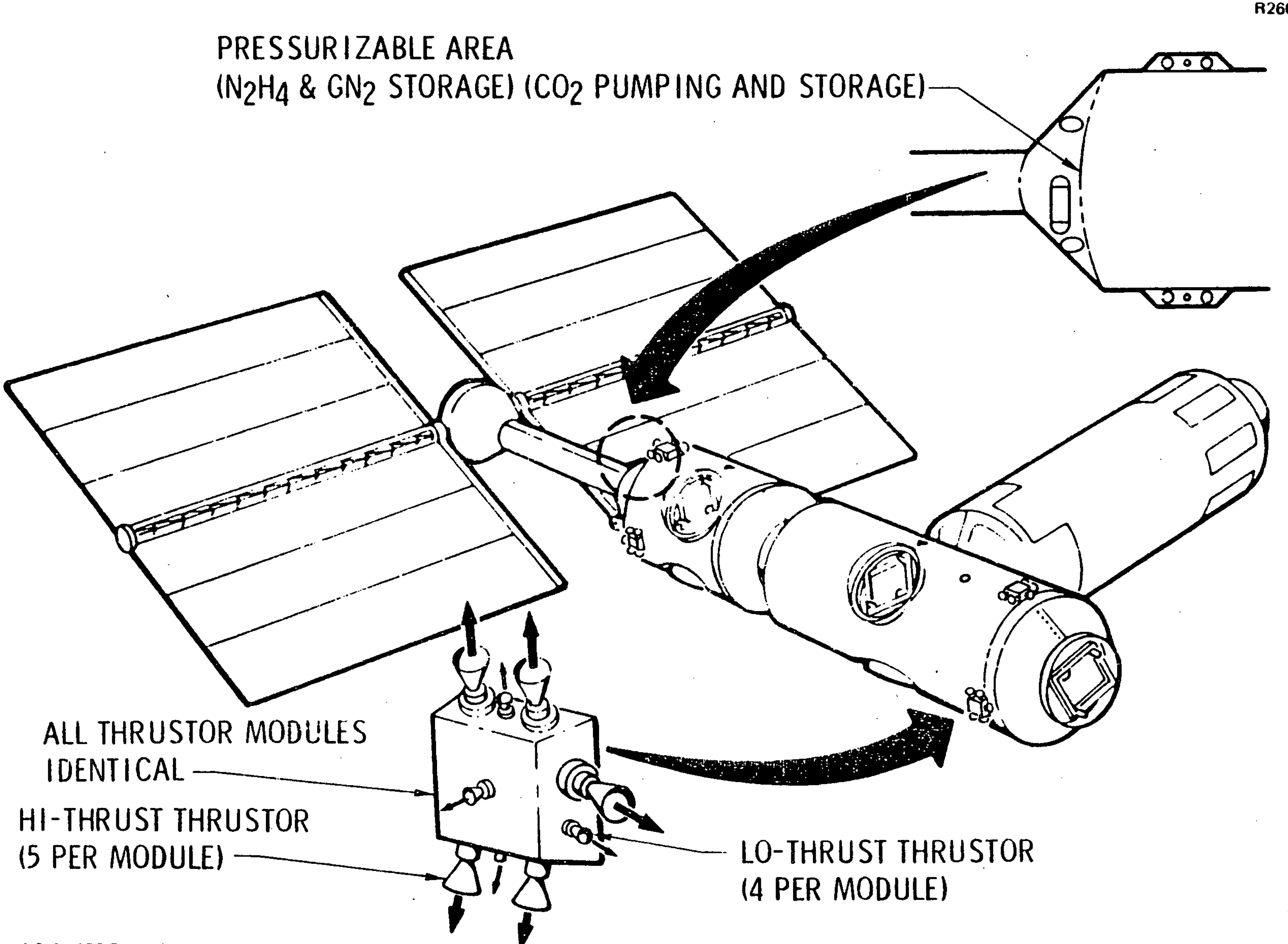

Figure 4.8-1. ISS Propulsion Installation

Table 4.8-2

PROGRAM REQUIREMENTS (PS 02925)

Requirement (No.)	Description
3.1.1.3	Total cost of the program is a primary consideration. Primary emphasis is on minimum cost to the IOC. IMPACT: Selection of N_2H_4 monopropellant to reduce cost to IOC. Selection of biowaste resistojet system to reduce total cost.
3.1.1.6	"Commonality" is a primary consideration throughout the study. As a goal, common module structures, systems, subsystems and assemblies for Space Station modules, crew cargo modules, and Research and Applications Modules should be developed. IMPACT: Selection of N_2H_4 for high-thrust system to be common with free-flying RAM modules.
3.1.3.2	Shuttle launch frequency, to support the Space Station Program, will be no greater than one every 30 days. IMPACT: Affects propellant storage requirements.
3.1.3.7	All modules on orbit shall be capable of being placed into a standby, unmanned mode and be reactivated after a period of up to one year. This capability shall be provided even when any one module has been returned to the ground for major repair. Resupply flights, if required, are permissible. (Moved from 3.1.3.1.3) IMPACT: N_2H_4 propellant selection for storable capability.
3.2.1.1	Provision shall be made for the return and recovery or suitable passivation and disposal of all expended Space Station hardware and waste products. Solid wastes shall not be dumped in space. IMPACT: Biowaste resistojet selection to dispose of CO_2 at high temperature and velocity.

Table 4.8-2

PROGRAM REQUIREMENTS (PS 02925) (Cont)

Requirement (No.)	Description
3.2.3.1	The "design-to" weight of Shuttle-transported modules shall not exceed 20,000 pounds.
3.2.4.1	Maintenance and repair will be accomplished on the ground when cost effective. Module return will be traded against on-orbit repair and maintenance.
	IMPACT: Requires purging provisions for high-thrust system, separable connectors and increased onboard checkout and fault isolation. Location of propulsion system in unpressurized but pressurizable area.
3.2.6.1.2	Safety is a mandatory consideration through the total program. As a goal, no single malfunction or credible combination of malfunctions and/or accidents shall result in serious injury to personnel or to crew abandonment of the Space Station.
	IMPACT: Affects location of propellant tankage (uninhabited area) and routing of propellant lines (external). No hydrazine inside pressurized area. Redundant system.
3.2.6.1.3	All safety hazards shall be identified in order of criticality. Catastrophic or critical hazards shall be eliminated or reduced to controllable or acceptable risk levels.
	IMPACT: Design redundancy - backup systems; equipment location; design margins on tanks and lines.
	<u>PROJECT REQUIREMENTS (PS02927)</u>
3.1.1.3	(See para. 3.1.1.3 Program Requirements - same impact.)
3.1.2.1	The Space Station will be capable of use in an orbit of 55° inclination at an altitude between 240 and 270 nautical miles.
	IMPACT: (See next item - 3.1.3.1.6.)

Table 4.8-2

PROGRAM REQUIREMENTS (PS 02925) (Cont)

Requirement (No.)	Description
3.1.3.1.6	Drag makeup impulse shall be provided. IMPACT: Requires orbit keeping be provided, met with low-thrust propulsion system to minimize resupply requirements and contamination potential.
3.2.1.1.4	No effluents (waste, propulsion, ventings, material outgassing, etc. shall deleteriously affect the Space Station, attached modules, adjacent spacecraft, or any of their experiments or measurements. The effects of effluents on the elements shall be minimized by design, including selection of material, outlet locations, direction of flow, sequencing, filtering, or crew EVA, experiments, optical devices, logistics and ancillary vehicle docking, structures, thermal control, and effective engine performance. IMPACT: Selection of low-thrust propulsion system to minimize resupply requirements and contamination potential; minimize use of high-thrust systems.
3.2.1.1.7	Subsystems shall be designed to incorporate functional product improvements and design growth without change in the interface. IMPACT: Selection of biowaste resistojet low thrust system is compatible with potential upgrading of EC/LS to a partially closed (sabatier) cycle.
3.2.1.8.2	The Space Station shall have means of producing control torques which are capable of meeting the statilization and control requirements for all of orbital operations after separation from the Shuttle Orbiter. IMPACT: See next item - 3.2.1.8.3.

Table 4.8-2

PROGRAM REQUIREMENTS (PS 02925) (Cont)

Requirement (No.)	Description
3.2.1.8.3	The Modular Space Station shall provide the orientation control system for any combination of modules to accommodate integral and attached module experiment accuracy and orbit-keeping requirements. IMPACT: Thrust level selection and thrustor module location for high thrust propulsion system (111 N/thrustor; 25 lbf/thrustor)
3.2.1.8.4	The Space Station must be stabilized for initial manning and buildup. IMPACT: Requires high thrust propulsion system on power module.
3.2.2.2	(See 3.2.3.1 under Program Requirements).
3.2.3.1	Space Station Project reliability shall provide 10 year useful operational life in which mission objectives can be accomplished. Reliability shall be obtained by (1) utilizing component with high reliability; (2) redundancy; and (3) maintenance. IMPACT: Selection of high- and low-thrust propulsion systems provides high degree of backup. Multiple thrustors, propellant tanks, regulators, etc., provide further redundancy. In-flight maintenance is provided.

R260

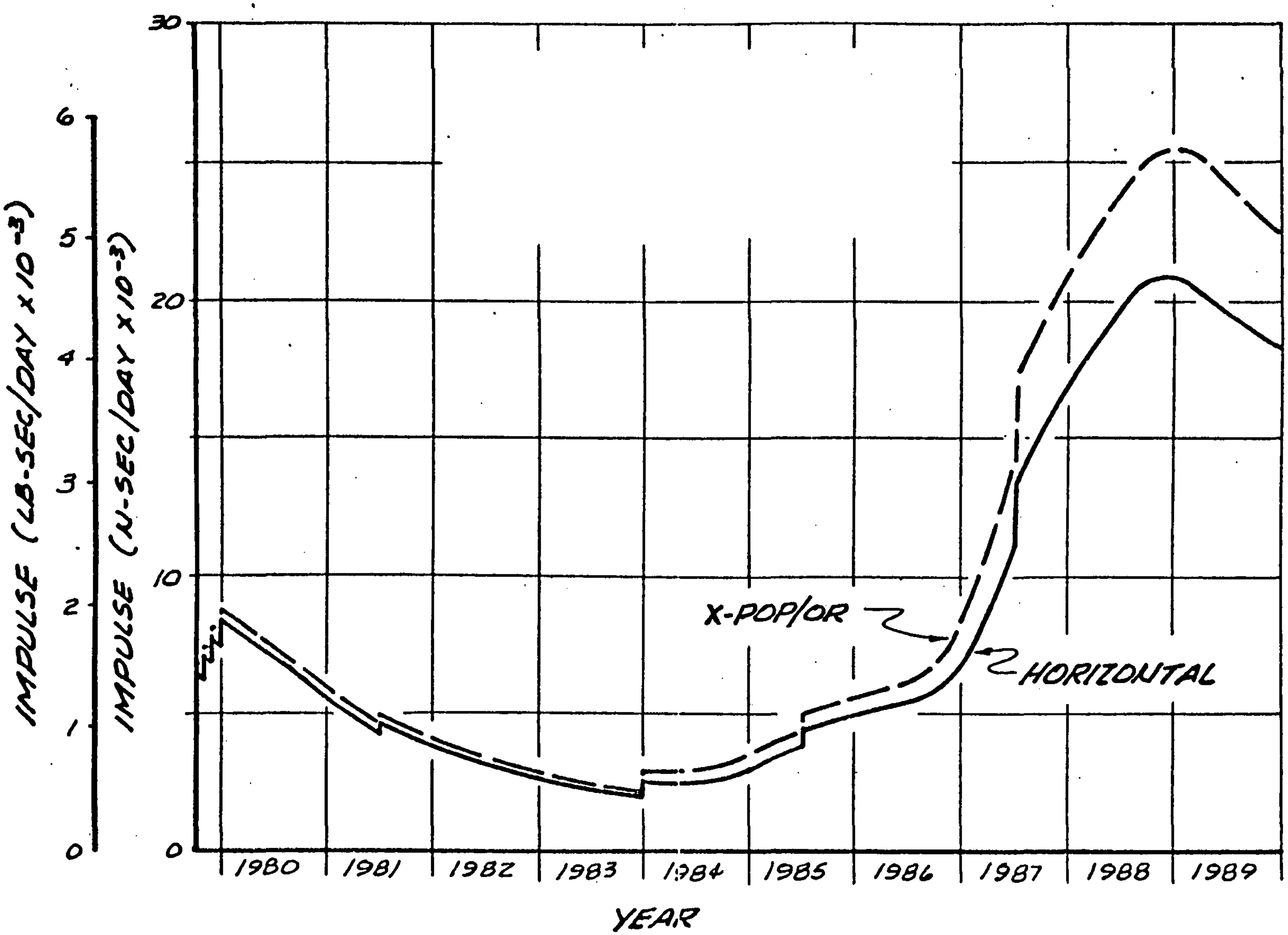

Figure 4.8-2. Orbit Keeping Impulse Requirement (242 N. Mile Circular)

The impulse indicated is equivalent to propellant (N_2H_4) requirements of 34 to 430 Kg/month (74 to 950 lb/month). During the initial 90-day buildup period, the Space Station modules are launched into a higher orbit than required and allowed to decay to the operational (243 nmi) altitude. This technique permits delaying the initial orbit keeping until 120 to 150 days after the initial launch.

4.8.2.1.2 Attitude Control Requirements

The attitude control requirements of the Space Station are divided between the control moment gyros (CMGs) in the Propulsion and the Guidance Navigation and Control (GNC) subsystems. The CMGs are sized to control cyclic torques during the manned operations, control of large disturbances such as docking, attitude maneuvers, and the attitude control prior to manning are performed with the P/RCS. Attitude control of the combined ISS and Orbiter, when docked, will be done with the Orbiter.

During the initial buildup, the attitude control requirements are minimized because the solar panel is fixed in a trailing orientation. Attitude control is needed primarily to minimize aerodynamic drag and to position the Space Station for docking operations. Roll control insures that one axis of the solar panels is normal to the sun.

As stated earlier, the CMG's provide the basic attitude control for day-to-day operations; however, the CMG's must be desaturated periodically to remove the effects of bias torques. The Propulsion Subsystem shall have the capability to desaturate the CMGs. Attitude control impulse requirements are summarized by phase in Table 4.8-3 and by mission year in Table 4.8-4.

4.8.3 Selected Subsystem Design

The subsystem design selected consists of a low-thrust, high-duty cycle Biowaste Resistojet subsystem and a high-thrust, low-duty cycle Monopropellant Hydrazine subsystem.

Table 4.8-3

SPACE STATION IMPULSE REQUIREMENTS SUMMARY*

Mission Phase	Functional Requirement	Total Impulse (n-sec x 10^{-6})	Total Impulse (lb-sec x 10^{-6})	(Remarks)
Initial Unmanned	Attitude Control	0.383	0.0861	0-90 days
Buildup	Maneuvers	0.127	0.0287	90-120 days
(0-90 days)	Shuttle De-dock	0.0505	0.0113	10% contingency
(4th Qtr 1979)		0.5605	0.1261	Total
ISS Phase	Orbit-Keeping	6.86	1.54	78%
(1980-1984)	Attitude Maneuvers Shuttle De-dock Disturbance	1.93	0.433	22%
		8.79	1.973	100%
GSS Phase	Orbit-Keeping	21.4	4.82	75%
(1985-1989)	Attitude Maneuvers Shuttle Docking Free Flying Module	7.05	1.59	25%
		28.45	6.41	100%
		Propellant Weight (Kg)	**Propellant Weight (lbs)**	
	Free Flying Propellant Resupply	4440	9780	

*For baseline "trimmed" horizontal attitude and case 534g Experiment Program

Table 4.8-4

ATTITUDE CONTROL IMPULSE REQUIREMENTS

SI UNITS (n-sec)

Year	Maneuvers (K - N-Sec / Event)	Maneuvers No. Events	Maneuvers Total (K N-Sec)	Shuttle Dock/De-dock (K N-Sec / Event)	Shuttle Dock/De-dock No. Events	Shuttle Dock/De-dock Total (K N-Sec)	F. F. Dock/De-dock (K N-Sec / Event)	F. F. Dock/De-dock No. Events	F. F. Dock/De-dock Total (K N-Sec)
1980	12	24	288	2.4	11	26.4			
1981	12	24	288	2.4	9	21.6			
1982	12	24	288	2.4	9	21.6			
1983	12	24	288	2.4	9	21.6			
1984	12/32	12/12	144/384	2.4/24	5/6	12/144			
1985	32	24	768	24	13	312			
1986	32	24	768	24	12	288			
1987	32	24	768	24	11	264	1.6	10	16.0
1988	32	24	768	24	8	192	1.6	18	28.8
1989	32	24	768	24	6	144	1.6	18	28.8
			5520			1135			73.6
Totals					5730				

ENGLISH UNITS (lb-sec)

Year	Maneuvers (K - N-Sec / Event)	Maneuvers No. Events	Maneuvers Total (K N-Sec)	Shuttle Dock/De-dock (K N-Sec / Event)	Shuttle Dock/De-dock No. Events	Shuttle Dock/De-dock Total (K N-Sec)	F. F. Dock/De-dock (K N-Sec / Event)	F. F. Dock/De-dock No. Events	F. F. Dock/De-dock Total (K N-Sec)
1980	2.7	24	65	0.54	11	5.9			
1981	2.7	24	65	0.54	9	4.9			
1982	2.7	24	65	0.54	9	4.9			
1983	2.7	24	65	0.54	9	4.9			
1984	2.7/7.4	12/12	32/87	0.54/5.4	5/6	2.7/32			
1985	7.4	24	173	5.4	13	71			
1986	7.4	24	173	5.4	12	65			
1987	7.4	24	173	5.4	11	59	0.36	10	3.6
1988	7.4	24	173	5.4	8	43	0.36	18	6.5
1989	7.4	24	173	5.4	6	32	0.36	18	6.5
			1240			255			16.6
Totals					1510				

The Biowaste Resistojet subsystem, referred to as the low-thrust subsystem, provides the orbit keeping impulse by heating waste CO_2 from the EC/LS subsystem. For the baseline altitude and attitude, the CO_2 production from a nominal crew is adequate to overcome orbital drag. If desired, the low-thrust subsystem may also provide CMG desaturation torques in combination with orbit keeping thrust without additional propellant requirements.

The Monopropellant Hydrazine Subsystem, referred to as the High-Thrust Subsystem, provides the torques for control of major attitude disturbances or maneuvers, back-up and/or supplementary orbit keeping, and a propellant supply system for the free flying RAM's (GSS only). As much Propulsion Subsystem equipment as possible is located in the forward conic section of the Power Module. This area may be pressurized or vented as desired, thus allowing for isolation in the event of N_2H_4 leakage or during system repair. Secondly, this location provides additional safety regarding rupture of tanks or components due to the distance from normally inhabited areas. Figure 4.8-1 shows the general arrangement of the system components and thrustor modules. Table 4.8-5 contains a complete listing of components for both the High-Thrust and Low-Thrust Subsystems extracted from MDAC Program P1268. Information on quantity, weight, power, and volume is included.

4.8.3.1 Description

4.8.3.1.1 <u>Low-Thrust Subsystem (Biowaste Resistojet)</u>

The low-thrust Propulsion Subsystem consists of compression pumps, heat exchangers, accumulators, thrustors, and the necessary valves and controls for system operation and checkout. A schematic of the system is shown in Figure 4.8-3.

The biowaste CO_2 is available from the EC/LS subsystem molecular sieves which are located in the Crew and GPL Modules. This CO_2 is carried to the Power Module under a pressure ranging from 275 to 103 N/m^2 (42-15 psi) where it is compressed for storage.

The CO_2 interconnection plumbing is 8 mm internal diameter (3/8-in. outside diameter). Line routing is from the molecular sieve accumulators

Table 4.8-5 (Sheet 1)

OCTOBER 15, 1971 PROGRAM P1268

MODULAR SPACE STATION

ISS CONFIGURATION

ITEM CODE	ITEM NAME	UNIT WT	AVG PWR	UNIT VOL	***** UNIT QUANTITIES **** MOD 1	MOD 2	MOD 3	TOTAL
DA 0 0	HI-THRUST PROPULSION	0	0	0.0	0	0	0	0
B 0 0	PRESSURIZATION SUBS	0	0	0.0	0	0	0	0
1 0	PRESS STG ASSY	0	0	14.0	0	0	0	0
1	STORAGE SPHERE	52	5	0.0	2	0	0	2
2	FILL DISCONNECT	1	0	0.0	1	0	0	1
3	VENT SOL (NC)	1	0	0.0	1	0	0	1
4	BURST DISK ASSY	1	0	0.0	2	0	0	2
5	RELIEF VLV ASSY	2	0	0.0	2	0	0	2
6	TEMP XDUCER	0	0	0.0	2	0	0	2
7	PRESS XDUCER	0	0	0.0	2	0	0	2
8	ISO SOL (LTG)	2	0	0.0	2	0	0	2
9	XFER SOL (LTG)	2	0	0.0	2	0	0	2
10	PLUMBING	15	0	0.0	1	0	0	1
11	BRACKETS	15	0	0.0	1	0	0	1
2 0	PRESS CONTROL ASSY	0	0	1.0	0	0	0	0
1	INLET FILTER	1	0	0.0	2	0	0	2
2	INLET SOL (LTG)	2	0	0.0	2	0	0	2
3	REGULATOR	2	0	0.0	2	0	0	2
4	PRESS XDUCER	0	0	0.0	2	0	0	2
5	OUTLET SOL (LTG)	2	0	0.0	2	0	0	2
6	PRESS SWITCH	1	1	0.0	4	0	0	4
7	ISO VLV (HD)	1	0	0.0	2	0	0	2
8	PLUMBING	4	0	0.0	2	0	0	2
9	BRACKETS	2	0	0.0	2	0	0	2
3 0	PRESS RESUPPLY ASSY	0	0	2.0	0	0	0	0
1	FILL UMBILICAL	1	0	0.0	3	3	0	6
2	FILL FILTER	1	0	0.0	3	3	0	6
3	FILL VLV (LTG)	1	0	0.0	3	3	0	6
4	DISPENSING UMB	1	0	0.0	3	3	0	6
5	DISPENSING VLV(HD)	1	0	0.0	3	3	0	6
6	PLUMBING	3	0	0.0	3	3	0	6
7	BRACKETS	1	0	0.0	3	3	0	6
C 0 0	PROPELLANT SUBS	0	0	0.0	0	0	0	0
1 0	PROPELLANT STG ASSY	0	0	38.0	0	0	0	0
1	ULL INLET SOL(LTG)	2	0	0.0	4	0	0	4
2	PROPELLANT TANK	50	15	0.0	4	0	0	4
3	PRESS XDUCER	0	0	0.0	6	0	0	6
4	TEMP XDUCER	0	0	0.0	4	0	0	4
5	QUANTITY GAGE	2	0	0.0	4	0	0	4

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Table 4.8-5 (Sheet 2)

OCTOBER 15, 1971 PROGRAM P1268 [redacted]

MODULAR SPACE STATION

ISS CONFIGURATION

ITEM CODE	ITEM NAME	UNIT WT	AVG PWR	UNIT VOL	***** UNIT QUANTITIES **** MOD 1	MOD 2	MOD 3	TOTAL
6	RELIEF VALVE	2	0	0.0	4	0	0	4
7	BURST DISK ASSY	1	0	0.0	4	0	0	4
8	ULL VENT SOL (NC)	1	0	0.0	4	0	0	4
9	PROP ISO VLV (HD)	1	0	0.0	4	0	0	4
10	PROP OUT SOL (LTC)	2	0	0.0	4	0	0	4
11	PROP FILL VLV (HD)	1	0	0.0	4	0	0	4
12	TH MOD ISO VLV(HD)	1	0	0.0	4	0	0	4
13	PLUMBING	10	0	0.0	4	0	0	4
14	BRACKETS	15	0	0.0	4	0	0	4
2 0	PROP RESUPPLY ASSY	0	0	2.0	0	0	0	0
1	FILL UMBILICAL	1	0	0.0	3	3	0	6
2	FILL FILTER	1	0	0.0	3	3	0	6
3	FILL VALVE (HD)	1	0	0.0	3	3	0	6
4	DISPENSING (UHR)	1	0	0.0	3	3	0	6
5	DISPENSING VLV(HD)	1	0	0.0	3	3	0	6
6	PLUMBING	3	0	0.0	3	3	0	6
7	BRACKETS	1	0	0.0	3	3	0	6
D 0 0	THRUST SUBS	0	0	0.0	0	0	0	0
1 0	THRUSTOR MODULE	0	0	3.5	0	0	0	0
1	THRUSTOR ASSY	5	2	0.0	20	20	0	40
2	TEMP XDUCERS	0	0	0.0	24	24	0	48
3	PRESS XDUCERS	0	0	0.0	24	24	0	48
4	ISO VLV (HD)	1	0	0.0	4	4	0	8
5	FILTER ASSY	1	0	0.0	4	4	0	8
6	PLUMBING	2	0	0.0	4	4	0	8
7	BRACKETS/STRUCT	15	0	0.0	4	4	0	8
E 0 0	PURGE SUBS	0	0	0.0	0	0	0	0
1 0	PURGE ASSY	0	0	2.0	0	0	0	0
1	PRESS SPHERE	25	0	0.0	2	0	0	2
2	REGULATOR	2	0	0.0	1	0	0	1
3	ISO VLV (HD)	1	0	0.0	2	0	0	2
4	PRESS XDUCER	0	0	0.0	4	0	0	4
5	PLUMBING	5	0	0.0	1	0	0	1
6	BRACKETS/STRUCT	20	0	0.0	1	0	0	1
X 0 0	EXPENDABLES	0	0	0.0	0	0	0	0
1 0	PROPELLANT	0	0	0.0	0	0	0	0
1	N2H4	1000	0	0.0	1	0	0	1
2 0	PRESSURANT	0	0	0.0	0	0	0	0
1	GN2	64	0	0.0	1	0	0	1

Table 4.8-5 (Sheet 3)

OCTOBER 15, 1971 PROGRAM P1268

MODULAR SPACE STATION

ISS CONFIGURATION

ITEM CODE	ITEM NAME	UNIT WT	AVG PWR	UNIT VOL	***** UNIT QUANTITIES **** MOD 1	MOD 2	MOD 3	TOTAL
EA 0 0	LO-THRUST PROPULSION	0	0	0.0	0	0	0	0
B 0 0	PROPELLANT SUBS	0	0	0.0	0	0	0	0
1 0	COLLECT/STG ASSY	0	0	20.0	1	0	0	1
1	PUMP ASSY	8	17	0.0	2	0	0	2
2	SUPPLY VLV	1	0	0.0	2	0	0	2
3	PUMP OUT CK VLV	1	0	0.0	2	0	0	2
4	SPHER ISO VLV(LTG)	2	0	0.0	2	0	0	2
5	TEMP XDUCERS	0	0	0.0	4	0	0	4
6	PRESS XDUCERS	1	0	0.0	4	0	0	4
7	RELIEF VLVS	2	0	0.0	2	0	0	2
8	PROP OUT VLV (LTG)	0	0	0.0	2	0	0	2
9	STORAGE SPHERE	18	3	0.0	2	0	0	2
10	PLUMBING	5	0	0.0	2	0	0	2
11	BRACKETS	10	0	0.0	2	0	0	2
C 0 0	CONTROL SUBS	0	0	0.0	0	0	0	0
1 0	FLOW CONTROL ASSY	0	0	2.0	1	0	0	1
1	INLET FILTER	1	0	0.0	2	0	0	2
2	PRESS XDUCER	1	0	0.0	4	0	0	4
3	TEMP XDUCER	1	0	0.0	2	0	0	2
4	REGULATOR	2	0	0.0	2	0	0	2
5	PROP OUT VLV (LTG)	2	0	0.0	2	0	0	2
6	PLUMBING	3	0	0.0	2	0	0	2
7	BRACKETS	6	0	0.0	2	0	0	2
2 0	POWER DIST/CONTROL	0	0	.6	1	0	0	1
1	INVERTER	6	95	0.0	2	0	0	2
2	CONTROLS	16	0	0.0	2	0	0	2
D 0 0	THRUST SUBSYSTEM	0	0	0.0	0	0	0	0
1 0	THRUSTOR MODULE	0	0	.3	4	4	0	8
1	THRUSTORS ISS5/GS7	2	0	0.0	16	16	0	32
2	TEMP XDUCER	1	0	0.0	16	16	0	32
3	PRESS XDUCER	1	0	0.0	16	16	0	32
4	ISOLATION VLV (HP)	1	0	0.0	4	4	0	8
5	TRANSFORMER (1,3)	1	0	0.0	16	16	0	32
6	PLUMBING	5	0	0.0	4	4	0	8
7	BRACKETS/STRUCT	5	0	0.0	4	4	0	8

FOLDOUT FRAME 1

Table 4.8-5 (Sheet 4)

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OCTOBER 15, 1971

PROGRAM P126B

SPACE STATION CONFIGURATION

FOLDOUT FRAME 2

ISS PARTS LIST

ITEM CODE	ITEM NAME	UNIT WT (LBS)	(1) AVG PWR (WATT)	MAX PWR (WATT)	(2) STDBY PWR (WATT)	UNIT VOL (CU FT)	(3) 115 VDC	(4) 28 VDC	400 SQ	400 SIN	(5) 60 SIN	FUG	ESS	NON ESS	RED	DUTY CYC (PCT)	PWR FORM	DEL
DA 0 0	HI-THRUST PROPULSION	-0	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
~~B 0 0~~	~~PRESSURIZATION SUBS~~	~~-0~~	~~-0~~	~~-0~~	-0	-0,0	0	0	0	0	0	0	0	0	0			
1 0	PRESS STG ASSY	-0	-0	-0	-0	14,0	0	0	0	0	0	0	0	0	0			
1	STORAGE SPHERE	52	5	15	-0	-0,0	1	0	0	0	0	0	0	1	0	33	DC	
2	FILL DISCONNECT	1	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
3	VENT SOL (NC)	1	0	50	-0	-0,0	1	0	0	0	0	0	1	0	0	01	DC	
~~4~~	~~BURST DISK ASSY~~	~~1~~	~~-0~~	~~-0~~	~~-0~~	~~-0,0~~	0	0	0	0	0	0	0	0	0			
5	RELIEF VLV ASSY	2	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
6	TEMP XDUCER	-0	-0	-0	-0	-0,0	1	0	0	0	0	0	1	0	0	ON	DC	
7	~~PRESS XDUCER~~	-0	-0	-0	-0	-0,0	1	0	0	0	0	0	1	0	0	ON	DC	
8	ISO SOL (LTG)	2	0	50	-0	-0,0	1	0	0	0	0	0	1	0	0	01	DC	
9	XFER SOL (LTG)	2	0	50	-0	-0,0	1	0	0	0	0	0	0	1	0	01	DC	
~~10~~	~~PLUMBING~~	~~15~~	~~-0~~	~~-0~~	~~-0~~	~~-0,0~~	0	0	0	0	0	0	0	0	0			
11	BRACKETS	15	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
2 0	~~PRESS CONTROL ASSY~~	~~-0~~	~~-0~~	~~-0~~	-0	-1,0	0	0	0	0	0	0	0	0	0	33	DC	
1	INLET FILTER	1	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
2	INLET SOL (LTG)	2	0	50	-0	-0,0	1	0	0	0	0	0	1	0	0	01	DC	
~~3~~	~~REGULATOR~~	~~2~~	~~-0~~	~~-0~~	~~-0~~	~~-0,0~~	~~0~~	~~0~~	~~0~~	0	0	0	0	0	0			
4	PRESS XDUCER	-0	-0	-0	-0	-0,0	1	0	0	0	0	0	1	0	0	ON	DC	
5	OUTLET SOL (LTG)	2	0	50	-0	-0,0	1	0	0	0	0	0	1	0	0	01	DC	
~~6~~	~~PRESS SWITCH~~	~~1~~	~~1~~	~~1~~	-0	~~-0,0~~	1	0	0	0	0	0	1	0	0	ON	DC	
7	ISO VLV (HD)	1	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
8	PLUMBING	4	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
~~9~~	~~BRACKETS~~	~~2~~	~~-0~~	~~-0~~	~~-0~~	~~-0,0~~	~~0~~	0	0	0	0	0	0	0	0			
3 0	PRESS RESUPPLY ASSY	-0	-0	-0	-0	2,0	0	0	0	0	0	0	0	0	0			
~~1~~	~~FILL UMBILICAL~~	~~1~~	~~-0~~	~~-0~~	-0	-0,0	0	0	0	0	0	0	0	0	0			
2	FILL FILTER	1	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
3	FILL VLV (LTG)	1	0	50	-0	-0,0	1	0	0	0	0	0	0	1	0	01	DC	
~~4~~	~~DISPENSING UMB~~	~~1~~	~~-0~~	~~-0~~	~~-0~~	~~-0,0~~	~~0~~	0	0	0	0	0	0	0	0			
5	DISPENSING VLV(HD)	1	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
6	PLUMBING	3	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
~~7~~	~~BRACKETS~~	1	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
C 0 0	PROPELLANT SUBS	-0	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
1 0	PROPELLANT STG ASSY	-0	-0	-0	-0	38,0	0	0	0	0	0	0	0	0	0			

~~(1) WORST CASE--24 HOUR PERIOD~~
(2) INCLUDED IN AVG PWR
(3) BASELINE FOR DC LOADS
~~(4) RESTRICTED USAGE--SELECTION MUST BE SUPPORTED BY IMPACT ANALYSIS~~
(5) FOR USE IN GPL ONLY

Table 4.8-5 (Sheet 5)

SPACE STATION CONFIGURATION

ISS PARTS LIST

ITEM CODE	ITEM NAME	UNIT WT (LBS)	(1) AVG PWR (WATT)	MAX PWR (WATT)	(2) STDBY PWR (WATT)	UNIT VOL (CU FT)	(3) 115 VDC	(4) 28 VDC	400 SQ	400 SIN	(5) 60 SIN	EMG	ESS	NON ESS	REF	DUTY CYC (PCT)	PWR FORM	DEL
1	ULL INLET SOL(LTG)	2	-0	50	-0	-0.0	1	0	0	0	0	0	1	0	0	01	DC	
2	PROPELLANT TANK	50	15	45	-0	-0.0	1	0	0	0	0	0	1	0	0	33	DC	
3	PRESS XDUCER	-0	-0	-0	-0	-0.0	1	0	0	0	0	0	1	0	0	ON	DC	
4	TEMP XDUCER	-0	-0	-0	-0	-0.0	1	0	0	0	0	0	1	0	0	ON	DC	
5	QUANTITY GAGE	2	-0	-0	-0	-0.0	1	0	0	0	0	0	1	0	0	ON	DC	
6	RELIEF VALVE	2	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
7	BURST DISK ASSY	1	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
8	ULL VENT SOL (NC)	1	0	50	-0	-0.0	1	0	0	0	0	0	1	0	0	01	DC	
9	PROP ISO VLV (HD)	1	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
10	PROP OUT SOL (LTG)	2	0	50	-0	-0.0	1	0	0	0	0	0	1	0	0	01	DC	
11	PROP FILL VLV (HD)	1	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
12	TH MOD ISO VLV(HD)	1	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
13	PLUMBING	10	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
14	BRACKETS	15	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
2 0	PROP RESUPPLY ASSY	-0	0	0	0	2.0	0	0	0	0	0	0	0	0	0			
1	FILL UMBILICAL	1	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
2	FILL FILTER	1	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
3	FILL VALVE (HD)	1	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
4	DISPENSING (UMB)	1	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
5	DISPENSING VLV(HD)	1	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
6	PLUMBING	3	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
7	BRACKETS	1	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
D 0 0	THRUST SUBS	-0	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
1 0	THRUSTOR MODULE	-0	-0	-0	-0	3.5	0	0	0	0	0	0	0	0	0			
1	THRUSTOR ASSY	5	2	6	-0	-0.0	1	0	0	0	0	0	1	0	0	33	DC	
2	TEMP XDUCERS	-0	-0	-0	-0	-0.0	1	0	0	0	0	0	1	0	0			
3	PRESS XDUCERS	-0	-0	-0	-0	-0.0	1	0	0	0	0	0	1	0	0			
4	ISO VLV (HD)	1	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
5	FILTER ASSY	1	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
6	PLUMBING	2	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0	ON	DC	
7	BRACKETS/STRUCT	15	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0	ON	DC	
E 0 0	PURGE SUBS	-0	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
1 0	PURGE ASSY	-0	0	0	0	2.0	0	0	0	0	0	0	0	0	0			
1	PRESS SPHERE	25	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
2	REGULATOR	2	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			

(1) WORST CASE--24 HOUR PERIOD
(2) INCLUDED IN AVG PWR
(3) BASELINE FOR DC LOADS
(4) RESTRICTED USAGE--SELECTION MUST BE SUPPORTED BY IMPACT ANALYSIS
(5) FOR USE IN GPL ONLY

Table 4.8-5 (Sheet 6)

~~OCTOBER 15, 1971~~ PROGRAM P1268

SPACE STATION CONFIGURATION

ISS PARTS LIST

ITEM CODE	ITEM NAME	UNIT WT (LBS)	(1) AVG PWR (WATT)	MAX PWR (WATT)	(2) STDBY PWR (WATT)	UNIT VOL (CU FT)	(3) 115 VDC	(4) 28 VDC	400 SQ	400 SIN	(5) 60 SIN	EMG	ESS	NON ESS	RED	DUTY CYC (PCT)	PWR FORM	DEL
3	ISO VLV (HP)	1	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
4	PRESS XDUCER	-0	-0	-0	-0	-0.0	1	0	0	0	0	0	0	1	0	ON	DC	
~~5~~	~~PLUMBING~~	5	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
6	BRACKETS/STRUCT	20	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
X 0 0	EXPENDABLES	-0	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
1 0	PROPELLANT	-0	0	0	0	-0.0	0	0	0	0	0	0	0	0	0			
~~1~~	~~N2H4~~	1000	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
2 0	PRESSURANT	-0	0	0	0	-0.0	0	0	0	0	0	0	0	0	0			
~~1~~	~~GN2~~	64	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			

(1) WORST CASE--24 HOUR PERIOD
~~(2) INCLUDED IN AVG PWR~~
(3) BASELINE FOR DC LOADS
(4) RESTRICTED USAGE--SELECTION MUST BE SUPPORTED BY IMPACT ANALYSIS
~~(5) FOR USE IN GPL ONLY~~

Table 4.8-5 (Sheet 7)

SPACE STATION CONFIGURATION

ISS PARTS LIST

ITEM CODE	ITEM NAME	UNIT WT (LBS)	(1) AVG PWR (WATT)	MAX PWR (WATT)	(2) STDBY PWR (WATT)	UNIT VOL (CU FT)	(3) 115 VDC	(4) 28 VDC	400 SQ	400 SIN	(5) 60 SIN	EMG	ESS	NON ESS	RED	DUTY CYC (PCT)	PWR FORM	DEL
EA 0 0	LO-THRUST PROPULSION	-0	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
B 0 0	PROPELLANT SUBS	-0	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
1 0	COLLECT/STG ASSY	-0	-0	-0	-0	20,0	0	0	0	0	0	0	0	0	0			
1	PUMP ASSY	8	17	68	-0	-0,0	0	0	1	0	0	0	0	1	0	25	AC	
2	SUPPLY VLV	1	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
3	PUMP OUT CK VLV	1	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
4	SPHER ISO VLV(LTG)	2	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
5	TEMP XDUCERS	-0	-0	-0	-0	-0,0	1	0	0	0	0	0	1	0	0	ON	DC	
6	PRESS XDUCERS	1	-0	-0	-0	-0,0	1	0	0	0	0	0	1	0	0	ON	DC	
7	RELIEF VLVS	2	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
8	PROP OUT VLV (LTG)	-0	0	50	-0	-0,0	1	0	0	0	0	0	1	0	0	01	DC	
9	STORAGE SPHERE	18	3	9	-0	-0,0	1	0	0	0	0	0	0	1	0	33	DC	
10	PLUMBING	5	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
11	BRACKETS	10	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
C 0 0	CONTROL SUBS	-0	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
1 0	FLOW CONTROL ASSY	-0	-0	-0	-0	2,0	0	0	0	0	0	0	0	0	0			
1	INLET FILTER	1	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
2	PRESS XDUCER	1	-0	-0	-0	-0,0	1	0	0	0	0	0	1	0	0	ON	DC	
3	TEMP XDUCER	1	-0	-0	-0	-0,0	1	0	0	0	0	0	1	0	0	ON	DC	
4	REGULATOR	2	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
5	PROP OUT VLV (LTG)	2	0	50	-0	-0,0	1	0	0	0	0	0	1	0	0	01	DC	
6	PLUMBING	3	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
7	BRACKETS	6	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
2 0	POWER DIST/CONTROL	-0	-0	-0	-0	,6	0	0	0	0	0	0	0	0	0			
1	INVERTER	6	95	380	-0	-0,0	1	0	0	0	0	0	0	1	0	ON	DC	
2	CONTROLS	16	0	50	-0	-0,0	0	0	0	0	0	0	0	0	0	01	DC	
D 0 0	THRUST SUBSYSTEM	-0	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
1 0	THRUSTOR MODULE	-0	-0	-0	-0	,3	0	0	0	0	0	0	0	0	0			
1	THRUSTORS ISS5/GS7	2	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0	02	AC	
2	TEMP XDUCER	1	-0	-0	-0	-0,0	1	0	0	0	0	0	1	0	0	ON	DC	
3	PRESS XDUCER	1	-0	-0	-0	-0,0	1	0	0	0	0	0	1	0	0	ON	DC	
4	ISOLATION VLV (HB)	1	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			
5	TRANSFORMER (1,3)	1	-0	-0	-0	-0,0	0	0	0	0	0	0	0	0	0			

(1) WORST CASE--24 HOUR PERIOD
(2) INCLUDED IN AVG PWR
(3) BASELINE FOR DC LOADS
(4) RESTRICTED USAGE--SELECTION MUST BE SUPPORTED BY IMPACT ANALYSIS
(5) FOR USE IN CPL ONLY

FOLDOUT FRAME 1

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FOLDOUT FRAME 2

Table 4.8-5 (Sheet 8)

OCTOBER 15, 1971

PROGRAM P1268

SPACE STATION CONFIGURATION

ISS PARTS LIST

ITEM CODE	ITEM NAME	UNIT WT (LBS)	(1) AVG PWR (WATT)	MAX PWR (WATT)	(2) STDBY PWR (WATT)	UNIT VOL (CU FT)	(3) 115 VDC	(4) 28 VDC	400 SQ	400 SIN	(5) 60 SIN	EMG	ESS	NON ESS	RED	DUTY CYC (PCT)	PWR FORM	DEL
6	PLUMBING	5	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			
7	BRACKETS/STRUCT	5	-0	-0	-0	-0.0	0	0	0	0	0	0	0	0	0			

(1) WORST CASE--24 HOUR PERIOD
(2) INCLUDED IN AVG PWR
(3) BASELINE FOR DC LOADS
(4) RESTRICTED USAGE--SELECTION MUST BE SUPPORTED BY IMPACT ANALYSIS
(5) FOR USE IN GPL ONLY

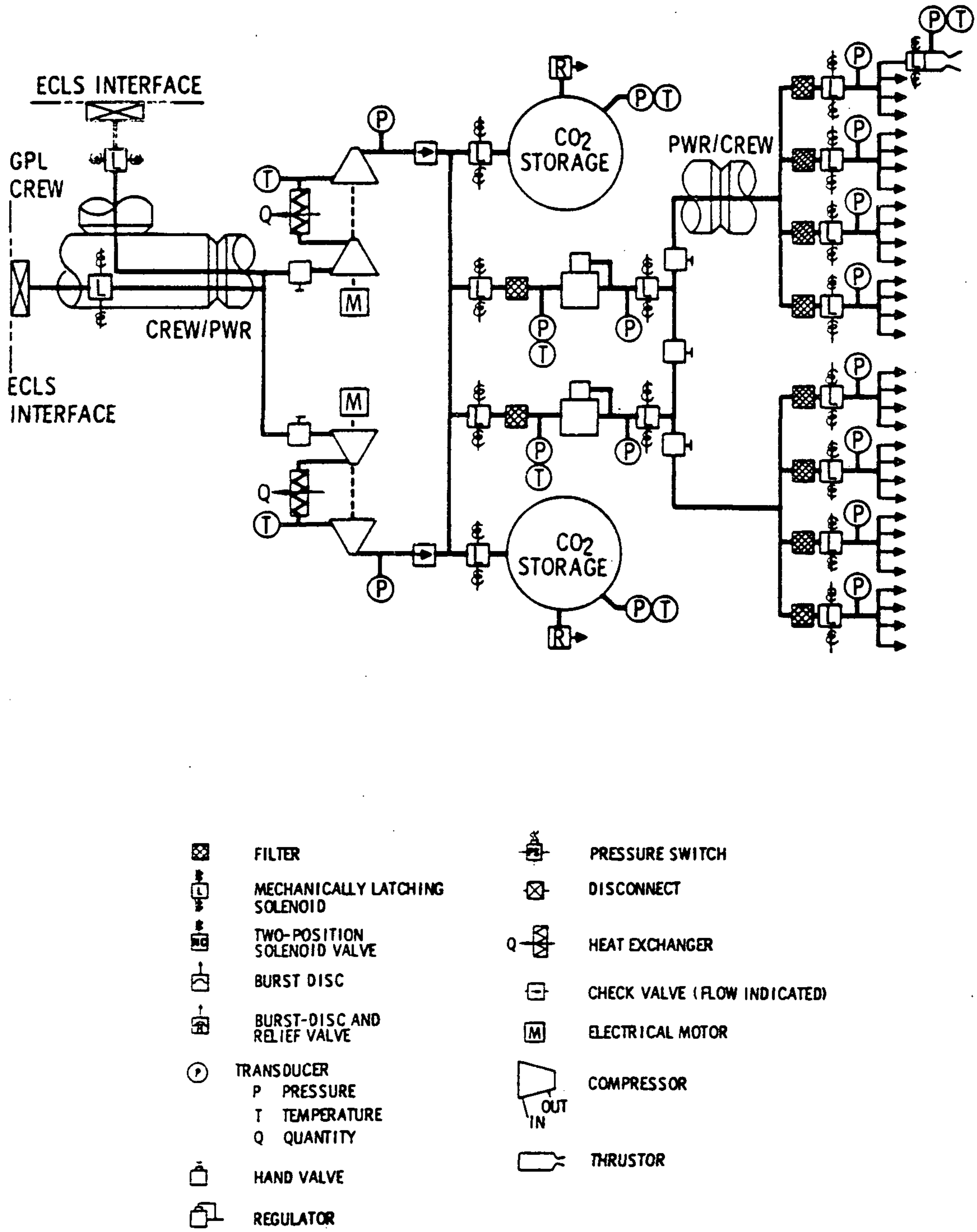


Figure 4.8-3. Low-Thrust Propulsion Subsystem Schematic

through the internal station docking ports, i.e., GPL/Crew and Crew/ Power Module interfaces. Disconnects are provided at the interface although the attachment is essentially permanent. The feed lines are routed separately through the interface to provide redundancy from the EC/LS accumulators to the compressors. There is a crossover line upstream from the compressors to allow either pump to use CO_2 from either or both EC/LS units.

The pumps are two stage compressors with intercooling and each handle 1.5×10^{-6} Kg/s (1.2 lb/hr), which is twice the average CO_2 flow rate. Thus, one pump may fail without degrading system operation, even during periods of crew overlap. The pump outlet pressure may be as high as 2.07×10^{6} Kg/m^2 (300 PSIA) with an inlet pressure of 7×10^{4} Kg/m^2 (10 PSIA). With this pressure rise and 50 percent efficiency, the average power is 40 watts. If a single-stage compresor were used, the power requirement would almost double.

The CO_2 is stored in two spherical accumulators, each with a capacity of 6.28 Kg (13.8 lb) at 2.07×10^{6} Kg/m^2 (300 PSIA). The spheres have a diameter of 0.716 m (28 in.) and will weigh approximately 8.18 Kg (18 lb) each. The accumulators are protected from overpressure in two ways: one, the pump power and accumulator fill valve may be interlocked with the accumulator temperature and pressure, and two, a relief valve is provided on the accumulator. The plumbing from the compressors to the accumulators allows either compressor to pump to either accumulator and also to be on line to the Thrustor Modules.

The CO_2 is regulated to 0.303×10^{6} N/m^2 (45 PSIA) in either of two parallel regulators. The regulators are designed to operate over a flow range of 0-0.520×10^{-3} Kg/S (0-1.14×10^{-3} Lb/Sec) which allows the number of thrustors in use to vary from zero to eight at any time.

The regulated CO_2 is distributed to four Thrustor Modules on the Power Subsystem module and four thrustor modules on the crew module (see Figure 4.8-1). The thrustor modules are protected from solid contamination by a 5-micron absolute filter element located upstream from the module inlet valve. Thus, the filter protects both the valve and thrustor assemblies. A pressure measurement is provided in each module to assist in determination of inlet valve status, filter condition, and, possibly thrustor valve leakage.

The system solenoid valves are all two position mechanically latching, i.e., an electrical signal is required to change position but no current or signal is required to hold in either position. This design has been selected because most valves are required to stay in position, either open or closed, for hours at a time. The instrumentation for the lo-thrust subsystem is as shown below in Table 4.8-6. In addition to this instrumentation, bi-level state indicators for all valves are required.

Table 4.8-6

LOW-THRUST SUBSYSTEM INSTRUMENTATION

Parameter Name	Range	Total Number
CO_2 Pump Inlet Pressure	0 to 50 PSIA	2
CO_2 Pump Interstage Temperature	460 to 800°R	2
CO_2 Pump Outlet Pressure	0 to 400 PSIA	2
CO_2 Storage Temperature	460°to 800°R	2
CO_2 Storage Pressure	0 to 400 PSIA	2
CO_2 Regulator Inlet Temperature	400 to 700°R	2
CO_2 Regulator Inlet Pressure	0 to 400 PSIA	2
CO_2 Regulator Outlet Pressure	0 to 60 PSIA	2
Thrustor Module Inlet Pressure	0 to 60 PSIA	8
Thrustor Chamber Pressure	0 to 60 PSIA	32
Thrustor Chamber Temperature	200 to 3000°R	32

4.8.3.1.2 Hi-Thrust Subsystem

Along with the instrumentation and controls necessary for subsystem operation and checkout the Hi-Thrust subsystem (Monopropellant N_2H_4) consists of GN_2 pressurant storage spheres, metal bellows propellant storage tanks, regulators, plumbing, and thrustors. A schematic of the hi-thrust subsystem is shown in Figure 4.8-4.

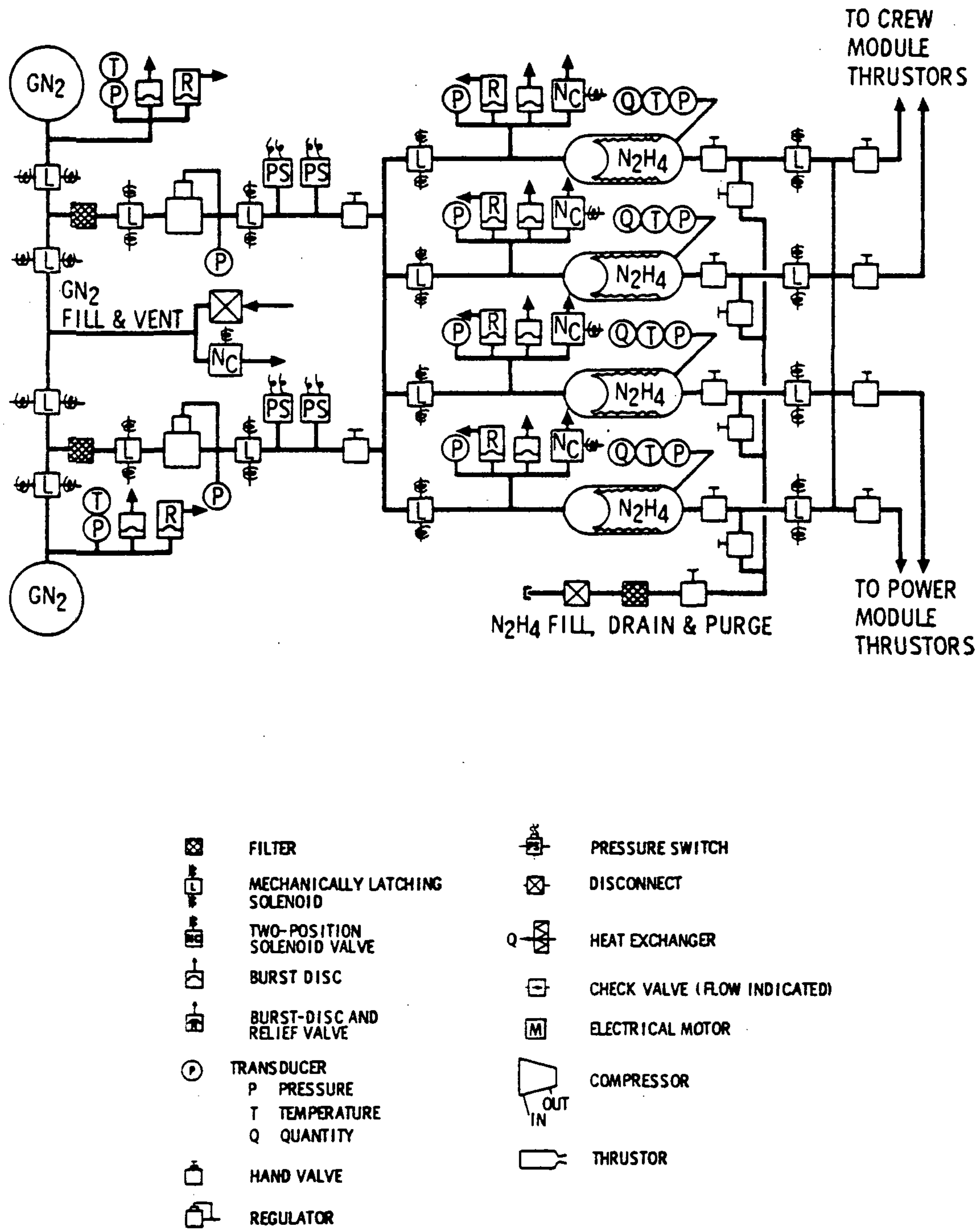


Figure 4.8-4. High-Thrust Propulsion Subsystem Schematic

The hi-thrust subsystem is the primary propulsion subsystem, in that it is capable of performing all the functions necessary for the Space Station. In cases where the function may be performed more efficiently by a lo-thrust biowaste subsystem, the hi-thrust subsystem serves as a back-up. Prior to initial manning, only the hi-thrust subsystem is operational. The hi-thrust subsystem is located in the power module to provide control during the early phases of buildup.

There are four propellant tanks of 114 Kg (250 LBM) capacity each composed of a titanium shell with a stainless steel bellows. The tanks are 46 cm (18 in.) in diameter and 115 cm (45 in.) long including the end domes and weigh 26.4 Kg (58 lbm) each. This sizing gives a total capacity of 445 Kg (1,000 lbm) which is adequate without resupply until initial manning or during any 90-day period thereafter, unless free flying RAM resupply is required. See Section 4.8.3.4 for further discussion of free flying RAM resupply. The tank working pressure is 2.06×10^6 N/m^2 (300 PSIA).

The pressurant is stored in two ambient temperature titanium spheres of 51 cm (20 in.) diameter which will weigh 23.6 Kg (52 lbm) each. The spheres are protected from overpressure by relief valves and burst disks. The relief valves set at 22.1×10^6 N/m^2 (3200 psia) with a burst disk which is set at 23.4×10^6 N/m^2 (3200 psia). This arrangement protects against a small, continuous loss caused by a leaking relief valve and also against a sudden total loss from a ruptured burst disk. Storage sphere pressure and temperature transducers provide an easy method to determine the available GN_2 and serve to initiate a warning of pressurant loss.

The GN_2 is regulated through one of two parallel regulators to a pressure of 2.06×10^6 N/m^2 (300 psia). Overpressure protection is provided by pressure switches which will in the first failure mode shut one regulator off and open the other. If the second regulator is not or line or available, the pressure switch will control pressure in a "bang-bang" mode. In the band-bang mode, the pressure switch shuts off a solenoid valve until the pressure decays to the switch drop-out setting. When the switch drops out, the valve is opened until the switch picks up and closes the valve again.

With the low-propellant consumption rates, this is a very satisfactory second back-up.

The four propellant tanks are plumbed together in parallel so that any one may be in use and the others depressurized. Each tank is protected with a vent valve, relief valve, and burst disk on the ullage side of the bellows. Each tank may also be isolated from the others by use of a latching solenoid valve on the ullage inlet and propellant outlet.

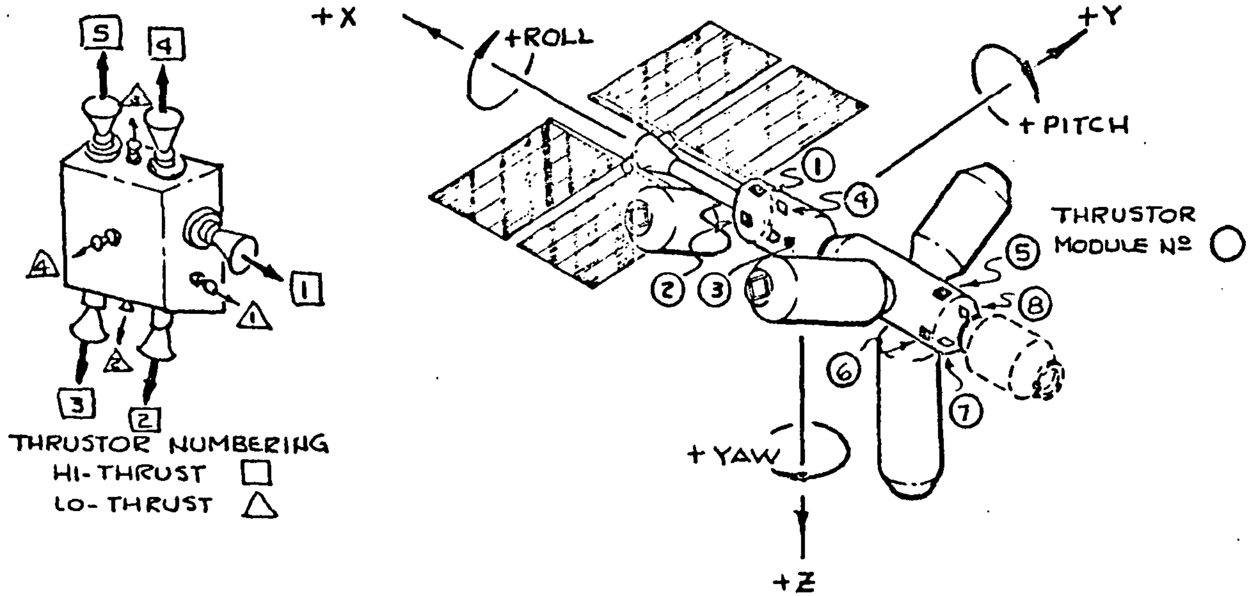
The tanks are refilled by venting the ullage side of the bellows and refilling each tank, one at a time, through the propellant fill hand valve. If necessary, the propellant may be expelled through the same valve and dumped through a catalytic burner to prevent contamination of spacecraft surface with raw N_2H_4.

The propellant lines from the tanks to the thrustor modules are routed outside the pressure shell and under the radiator, meteoroid shield, and insulation. They are thermally bonded to the pressure shell to prevent freezing. This location is inaccessible for maintenance or repair; therefore, a redundant propellant line is installed.

At the docking ports, the lines must be accessible for mating and checkout. Both accessibility and isolation are provided by routing the lines inside the umbilical frame but providing a sleeve that may be installed over the lines and the anulus vented to space vacuum (See Subsection 4.8.3.2.1).

The thrustors are mounted in eight modules of five thrustors each. Each module contains both the 0.111 N (0.25 lbf) and the 111 N (25 lbf) thrustors and all are identical and interchangeable. Figure 4.8-1 shows the mounting arrangement.

Shuttle Cargo Bay clearance prevents mounting a high-thrust thrustor in a radial position as can be done with the low-thrust thrustors. Therefore, to preserve redundancy, the modules must contain two tangential thrustors in each direction. Figure 4.8-5 indicates the use to which each thrustor may be put and indicates the redundancy of the system.



	Module / Thrustor / Function	1-1	1-2	1-3	1-4	1-5	2-1	2-2	2-3	2-4	2-5	3-1	3-2	3-3	3-4	3-5	4-1	4-2	4-3	4-4	4-5
HI-THRUST	X TRANSLATION	-					-					-					-				
HI-THRUST	Y TRANSLATION		-	-	+	+							+	+	-	-					
HI-THRUST	Z TRANSLATION							+	+	-	-							-	-	+	+
HI-THRUST	ROLL		-	-	+	+		-	-	+	+		-	-	+	+		-	-	+	+
HI-THRUST	PITCH	+						-	-	+	+	-						+	+	-	-
HI-THRUST	YAW		-	-	+	+	-						+	+	-	-	+				
LO-THRUST	X TRANSLATION	-					-					-					-				
LO-THRUST	Y TRANSLATION		-	+						+			+	-						-	
LO-THRUST	Z TRANSLATION				+			-	+						-			-	+		
LO-THRUST	ROLL		-	+				-	+				+	-				+	-		
LO-THRUST	PITCH	+			-			-	+			-			+			+	-		
LO-THRUST	YAW		-	+			-			+			+	-			+			-	

	Module / Thrustor / Function	5-1	5-2	5-3	5-4	5-5	6-1	6-2	6-3	6-4	6-5	7-1	7-2	7-3	7-4	7-5	8-1	8-2	8-3	8-4	8-5
HI-THRUST	X TRANSLATION	+					+					+					+				
HI-THRUST	Y TRANSLATION		+	+	-	-							-	-	+	+					
HI-THRUST	Z TRANSLATION							-	-	+	+							+	+	-	-
HI-THRUST	ROLL		+	+	-	-		+	+	-	-		+	+	-	-		+	+	-	-
HI-THRUST	PITCH	-						-	-	+	+	+						+	+	-	-
HI-THRUST	YAW		+	+	-	-	+						+	+	-	-	-				
LO-THRUST	X TRANSLATION	+					+					+					+				
LO-THRUST	Y TRANSLATION		+	-						+			-	+						-	
LO-THRUST	Z TRANSLATION				+			-	+						-			+	-		
LO-THRUST	ROLL		+	-				+	-				+	-				+	-		
LO-THRUST	PITCH	-			+			-	+			+			-			+	-		
LO-THRUST	YAW		+	-			+			-			+	-			-			+	

Figure 4.8-5. Thrustor Functions

The instrumentation for the High-Thrust Subsystem is shown below in Table 4.8-7. In addition to this instrumentation, bi-level state indicators for all valves are required.

Table 4.8-7
HI-THRUST SUBSYSTEM INSTRUMENTATION

Parameter	Range	Total No.
GN_2 Storage Press.	0 to 3500 PSIA	2
GN_2 Storage Temp	450 to 600°R	2
GN_2 Regulator out. Press.	0 to 400 PSIA	2
N_2H_4 Ullage Press.	0 to 400 PSIA	4
N_2H_4 Prop. Press.	0 to 400 PSIA	4
N_2H_4 Prop. Temp	460 to 600°R	4
N_2H_4 Prop. Quantity	0 to 100%	4
Thrustor Mod Prop. Press.	0 to 400 PSIA	16
Thrustor Temp	300 to 3000°R	40

4.8.3.2 Interfaces

The major system interfaces with the other program elements are propellant and pressurant resupply from the Logistics Module and propellant resupply to the Free-Flying RAMs. There is no propulsion command interface with the Shuttle during docking; the interface is confined to the Station GNC subsystem which commands the Hi-Thrust Subsystem either through hardware interconnections or via the DMS. The major internal interfaces are with the EPS, GNC, ECLS, and DMS.

4.8.3.2.1 Logistics Module Interface

The Logistics Module/Propulsion interface provides a means of transferring propellant (N_2H_4) and pressurant (GN_2). The propellant lines and couplings are sized for a flow rate of 0.364 kg/sec (6 GPM) which will yield a transfer time of approximately 10 minutes for 227 Kg (500 LBM) at full flow. The total time for the operation from initiation to securing requires 20 to 30 minutes. For a pressure drop of 173×10^3 N/M^2 (125 psia) the lines and fittings must have a nominal diameter of 2 cm (3/4 in.).

The propellant umbilical connection normally will be purged and vented so that no special care must be exercised during module separation. It will be pressurized and filled with propellant only during the 20 minute loading noted above. As further precaution, the disconnect is shrouded in such a manner that any leakage will be vented the space (See Figure 4.8-6). Thus, the disconnect is maintained in a secured position except during transfer operations and then any leakage would be vented overboard.

The pressurant (GN_2) disconnect does not present a toxicity hazard such as the N_2H_4 does and therefore may be considerably simpler. The disconnect is 0.65 cm (0.25 in.) diameter and is repeatedly used. Both halves are self sealing so that no problem will be encountered during a rapid disconnect. No vacuum annulus is required; however, a double seal is employed to simplify leak detection and improve reliability.

The GN_2 and N_2H_4 disconnects are located at each docking port. No requirement for these connections exists for the CREW/GPL interface and the docking port at the exterior end of the GPL.

4.8.3.2.2 Internal Interfaces

EPS—Propulsion Interface

The Hi-Thrust Propulsion Subsystem requires only 115 volt dc power. Of the average load of approximately 75 watts, most is heater power for thrustor catalyst bed heaters and propellant temperature control. The solenoid valves will be designed for 115 volts dc and are nearly all of the mechanically latching-type that require no signal or command to stay in the position of last command. Thus, the control power requirements are minimal.

The Lo-Thrust Subsystem propulsion power load is approximately 220 watts average. Each of the two compression pumps will consume about 17 watts of square wave ac power if all CO_2 is pumped from 103×10^3 to 2070×10^3 N/M^2 (15 to 300 psia). The remainder of the power is 115 volt dc. The Lo-Thrust Subsystem's power distribution and control assembly contains

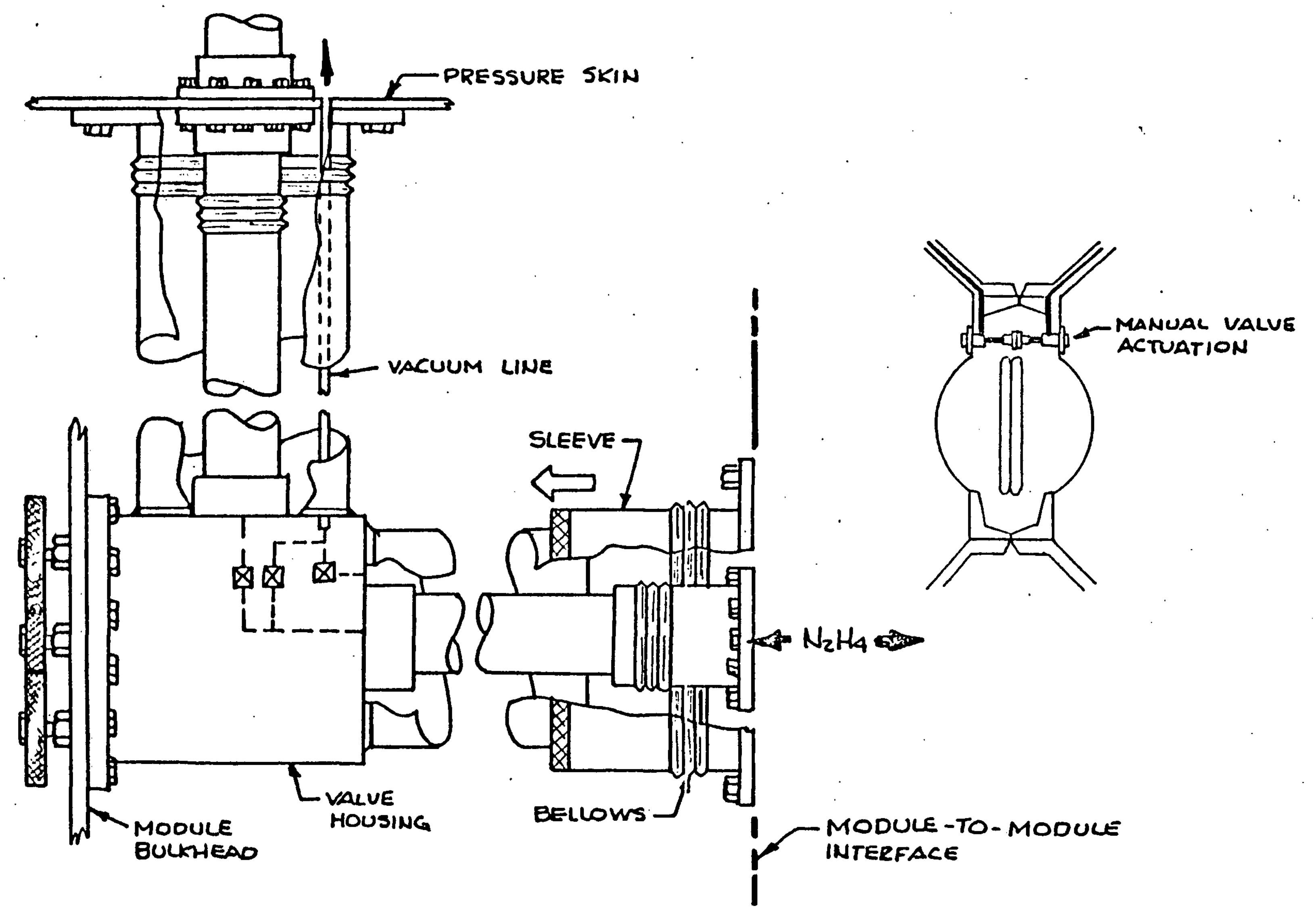


Figure 4.8-6. N_2H_4 Interface Connection

inverters and transformers to condition the 115 dc for the resistojet thrustors. An efficiency factor for the inverter, transformer combination of 75 percent has been included in the loads.

The Lo-Thrust power loads consider that all available CO_2 is pumped to 2.07×10^6 N/M^2 (300 psia) for storage and expelled at 1600°K (2880°R). Under certain conditions it may be possible to save a significant portion of the power by operating the resistojets at a lower temperature.

GNC—Propulsion Interface

The GNC Hi-Thrust Subsystem interface is basically a command link where the GNC Subsystem sends firing commands to the thrustors. The GNC Subsystem determines the torques and thrust vector changes necessary from attitude and navigation data. The thrust requirements are compared with Propulsion Subsystem status data so that the appropriate thrustors may be selected. Normal firing commands are sent over the Data Bus System. Back-up hardwire command links are provided so that control may be maintained even with a loss of the data bus. This hardwire command bypasses much of the GNC electronics and is a "fly-by-wire" technique.

The GNC/Propulsion low-thrust interface is much the same as the high-thrust with the exception that no hardwire back-up is provided and that the decision-making process for thrustor firing must include propellant availability vs. projected needs in the near future as well as current space station impulse requirements.

Thus, if little orbit keeping has been required due to low atmospheric density and the CO_2 storage is filled to capacity, low-temperature (reduced-power) or even non-propulsive thrustor operations may be required.

EC/LS—Propulsion Interface

The EC/LS interfaces only with the Low-Thrust System. CO_2 is available from the EC/LS units in both the Crew and GPL modules. The average CO_2 production rate will be 1.05 kg/man-day (2.3 lbm/man-day). Therefore, the ISS propellant availability is 6.30 Ka/Day (13.8 lbm/Day).

The CO_2 supply is intermittently available from the EC/LS molecular sieves with small tubing of approximately 0.65 cm (0.25 in.) diameter to carry the gas to the low-thrust biowaste compression pumps.

If a low-thrust system failure prevents CO_2 collection storage or use, it will be dumped temporarily by the EC/LS Subsystem. This is, however, extremely unlikely since a large degree of hardware and operational redundancy exists.

DMS—Propulsion Interface

The DMS/Propulsion interface is through the individual RDAUs in the Space Station modules. The transducers provide either a 0 to 5 volt or 0 to 40 millivolt output to the RDAU stations while the bi-level inputs are 0 to 5 volts. Command signals of 5 volts are used to switch the 115 volts required by the control valves.

4.8.3.3 Propulsion Subsystem Operation

4.8.3.3.1 Ground Servicing

The High-Thrust Subsystem is designed to be resupplied with propellant under zero-g conditions and contains positive expulsion metal bellows in the fuel tanks.

Prior to propellant loading, a leak and functional check is required to ensure that the system is ready for use. This check includes a leak test of all connections with GH_e at operating pressure and then a functional check of valves which includes actuation times and seat leakage. After this check is complete, the GN_2 system is filled and secured, a pressure decay leak check is conducted, and the preparations are made for propellant loading.

The metal bellows are an asset in the ground loading because they allow "vacuum loading" to be used. The propellant tanks must be purged with an inert gas (GN_2) and then evacuated with a ground vacuum pump. The required 700 lbm of N_2H_4 is loaded after all the purge gas is eliminated. This provides enough propellant to last until manning has taken place.

The Low-Thrust System is not operational nor does it contain propellants until after manning. Therefore, ground servicing consists only of a check-out and functional certification followed by a securing procedure.

4.8.3.3.2 Flight Operations

The High-Thrust system is operational during the first 90 days (build-up period). Propellant loaded prior to launch is adequate for 120 days to allow a 30-day contingency period. The system is activated for checkout after the Power Subsystems Module is erected from the Shuttle Cargo Bay but prior to release. Once proven functional, the module may be released and the high-thrust system allowed to assume control of the module attitude. The station will be held in a local-horizontal, roll-stabilized attitude with the solar panels feathered and trailing. This attitude reduces propellant consumption while still providing adequate electrical power.

When the second (Crew/Operations) module arrives, the Station can be maneuvered to a more favorable attitude for docking. Once docked, the activation crew is required to connect and leak check the docking port disconnect for the N_2H_4 propellant lines to the crew-module thrustors. This time has been included in the activation time line. The arrival of the third module (GPL) does not require any propulsion related activation. The Low-Thrust system operation commences with the arrival of the permanent station crew members. When the EC/LS subsystem begins CO_2 production, the accumulator pressure is sensed and the CO_2 compressor started, from that point the system functions automatically, following the GNC commands. CO_2 consumption will be governed by availability predictions, i.e., when an excess is predicted some CO_2 will be expelled non-propulsively through opposing jets. If a deficit is predicted, consumption will be in the most efficient manner through the most effective thrustors.

The long term operations after the Space Station is activated include monitoring, fault isolation, maintenance, checkout, and resupply. The monitoring is handled automatically by the On-board Checkout System (OCS) which observes status on a near continuous basis. Items such as unscheduled valve closure or out-of-tolerance regulation would cause a switch to redundant components and notification of the crew. Then this condition is investigated by the crew, either by calling on OCS sub-routines and/or by a physical inspection. Once a failed part is identified, maintenance is accomplished in one of three ways. The propulsion components, other than thrustors and propellant lines, are located in an unpressurized but pressurizable compartment and, therefore, operations could be performed under either IVA or shirtsleeve conditions.

If the failure and repair does not involve a concern over leakage of toxic materials, the parts will be replaced and checked out subsequently in a shirtsleeve environment. If the failure involves concern over contamination of the cabin atmosphere, the repair will take place under IVA conditions with the cabin atmosphere isolated from the propulsion compartment.

Failures involving thrustors must be handled by EVA. The need for this activity has been minimized by having both a High- and a Low-thrust system and and by providing redundant thrustors in all axes. Nominally, a number of thrustors may be allowed to fail before EVA is required and then, all failed thrustors will be replaced in one excursion. The thrustors will be developed specifically to include a plug-in capability. The high-thrust propellant plumbing is routed outside the pressure shell and includes installed redundant lines. Presently, there are no plans to make repairs on the external portions of the propellant lines as they are routed between the meteoroid bumper/insulation blanket and the pressure shell and are not accessible. All exterior connections will be welded.

The maintenance mentioned above is on an "as-required" basis with all components permitted to operate until degraded performance or failure is apparent.

The high-thrust subsystem must be resupplied with N_2H_4 and GN_2 at periodic intervals. This is done by bulk transfer from propellant tanks in the Logistics Module. Sequentially, after the docking port connections (See Subsection 4.8.3.2.1) are mated, the Space Station N_2H_4 tanks are vented, interconnecting lines evaluated, Logistics Module tanks pressurized, and then propellant flow is initiated. Receipt of a 100 percent signal from the quantity gage on the Station tanks causes flow to be diverted to the next empty tank; if none is left, the flow is halted.

A weight, volume, and power allocation has been made for a purge assembly for use during maintenance or resupply operations. This purge assembly consists of GN_2 storage spheres, regulator, valves and assorted controls. Prior to the opening of a line or component exposed to propellant the item would be purged with GN_2 and allowed to vacuum dry. Significant quantities of N_2H_4 would be discharged through opposing thrustors to minimize contamination.

4.8.3.4 Growth Space Station (GSS) Considerations

The growth to GSS is accomplished by replication of ISS elements in the additional modules. The only exception to a strict duplication of the initial modules occurs with the second Crew/Operation Module which does not have thrustors.

The second Crew/Operations Module will have an EC/LS-Propulsion interface for the low-thrust subsystem and GN_2 and N_2H_4 plumbing for the high-thrust subsystem. There is no requirement for thrustors. The second Power Module contains the same elements as the first. This, when connected to the initial system, provides for a total of 910 Kg (2000 lbm) N_2H_4 storage, 38.2 (128 lbm) GN_2 storage, and 25 Kg (55.2 lbm) CO_2 storage.

The thrustor modules on the initial Crew/Operations Module are deactivated because their lever arm is reduced to almost zero. The thrustors on the two Power Modules are used for all functions. The first use of free-flying RAM modules occurs in the GSS phase. This brings with it a requirement to resupply the RAM N_2H_4 which is done somewhat differently than the resupply of the Space Station high-thrust subsystem. The free-flying modules use "blow-down" propellant tanks and loading must take place at a high-pressure (2.41×10^6 N/M^2 vs. 0.344 N/M^2 (350 psia vs. 50 psia)). With a blow-down system, the tanks are loaded against the pressurizing gas which is recompressed to the original operating pressure. The requirements for free-flyer propellant resupply are as follows:

	N_2H_4/Event (Avg)		N_2H_4 (Total)	
Year	Kg	LBM	Kg	LBM
1988	88.3	194	880	1936
1989	99.1	218	1785	3924
1990	99.1	218	1785	3924
TOTAL	---	---	4450	9784

NOTE: 534G experiment schedule.

The use of a blow-down system on the experiments modules eliminates GN_2 transfer but does not actually reduce consumption. The GN_2 will be consumed by the Logistics Module or the Space Station, whichever supplied the propellant. Consumption will total 285 Kg (625 lbm) over the three years noted in the listing above.

4.8.4 Design Analyses and Trade Studies

The prime propulsion trade study was one to determine whether a High-Thrust, Monopropellant Subsystem or a combination High-Thrust, Monopropellant and Low-Thrust, Biowaste Resistojet Subsystem best meets the overall propulsion requirements.

4.8.4.1 High-Thrust/Low-Thrust Trade Study

The basic considerations for this trade study are (1) cost (both initial development and total system cost); (2) contamination of the Space Station and its environment, and (3) operational flexibility and reliability. The high-thrust system is needed in any case to handle attitude control during build-up and docking or other major disturbances during the operational phase. Four systems with different degrees of CO_2 disposal control were selected for comparison.

All four systems require the high-thrust propulsion and one combines the high-thrust with a biowaste (CO_2) resistojet:

A. High-Thrust + CO_2 Resistojet
B. High-Thrust + CO_2 Continuous Vent
C. High-Thrust + CO_2 Store and Dump
D. High-Thrust + CO_2 Store and Return

System costs are subdivided into "building block" elements in a manner which allows the elements to be combined as appropriate for each alternate.

4.8.4.1.1 Alternate Systems (Selected)

Alternate A - High thrust + CO_2 Resistojet

This alternate combines a High-Thrust Subsystem to perform all propulsion functions except orbit keeping which is performed by the low-thrust biowaste resistojet subsystem. This system provides control over the CO_2 disposal by allowing storage of up to two days output if contamination-sensitive experiments are in operation. The system would be operated with the CO_2 exhaust directed away from the Space Station at a high temperature and high velocity, which minimizes the quantity that could collect around the Space Station as a "cloud".

Alternate B - High Thrust + Continuous CO_2 Vent

The continuous CO_2 vent makes this system the most unfavorable with regard to contamination; however, at the same time, the initial cost is lowest. The high-thrust system performs all propulsive functions, is developed for a higher duty cycle and requires additional spare parts over Alternate A. Resupply costs include N_2H_4 for orbit keeping.

Alternate C - High-Thrust + CO_2 Store and Dump

This alternate was included as it provided a portion of the control over CO_2 disposal attained by Alternate A and did not have development costs for a CO_2 Biowaste Resistojet, Low-Thrust Subsystem. As in Alternate A the storage capacity is two days. Power costs are similar to Alternates A and D, because the pumping power overshadows the resistojet thrustor power. N_2H_4 resupply for orbit keeping is required.

Alternate D - High Thrust + CO_2 Store and Return

This combination offers the absolute minimum with regard to the total mass of effluent; however, the N_2H_4 exhaust weight is higher than Alternate A because of the orbit-keeping requirement. The waste CO_2 from the EC/LS system is compressed for storage in bottles brought up in the Logistics Module. The bottle weight penalty is severe (1.3 Kg of bottle for each 1.0 Kg of CO_2 returned). The resupply costs are based only on bringing up empty bottles with no return weight cost.

4.8.4.1.2 Cost Elements

The fixed costs were segregated into development and initial hardware costs for each system plus an additional hardware cost to provide hardware for the GSS buildup. In the case of the High-Thrust subsystem, an additional cost due to higher usage for orbit keeping was included.

The power costs considered that all CO_2 was compressed for storage (except for Alternate B) with an increment for resistojet thrustor power for Alternate A. The costs for each five-year period, ISS and GSS, were spread out as a resupply cost.

The resupply costs included molecular sieve expendables, CO_2 return bottles, N_2H_4 for maneuvers, and N_2H_4 for orbit keeping as well as the power mentioned above. The orbit-keeping propellant requirements were determined from Figure 4.8-7.

4.8.4.1.3 Results

Figure 4.8-8 shows the comparative system costs as a function of time from IOC to program completion; and the division between fixed, power and resupply costs and the total Space Station effluent quantity from each system.

Table 4.8-8 shows the cost elements that were included with each system alternate with back-up information shown in Tables 4.8-9 and 4.8-10. The ideal system would minimize both initial development and resupply costs, be least sensitive to an increase in the resupply cost factor, and minimize the power requirements and total effluent.

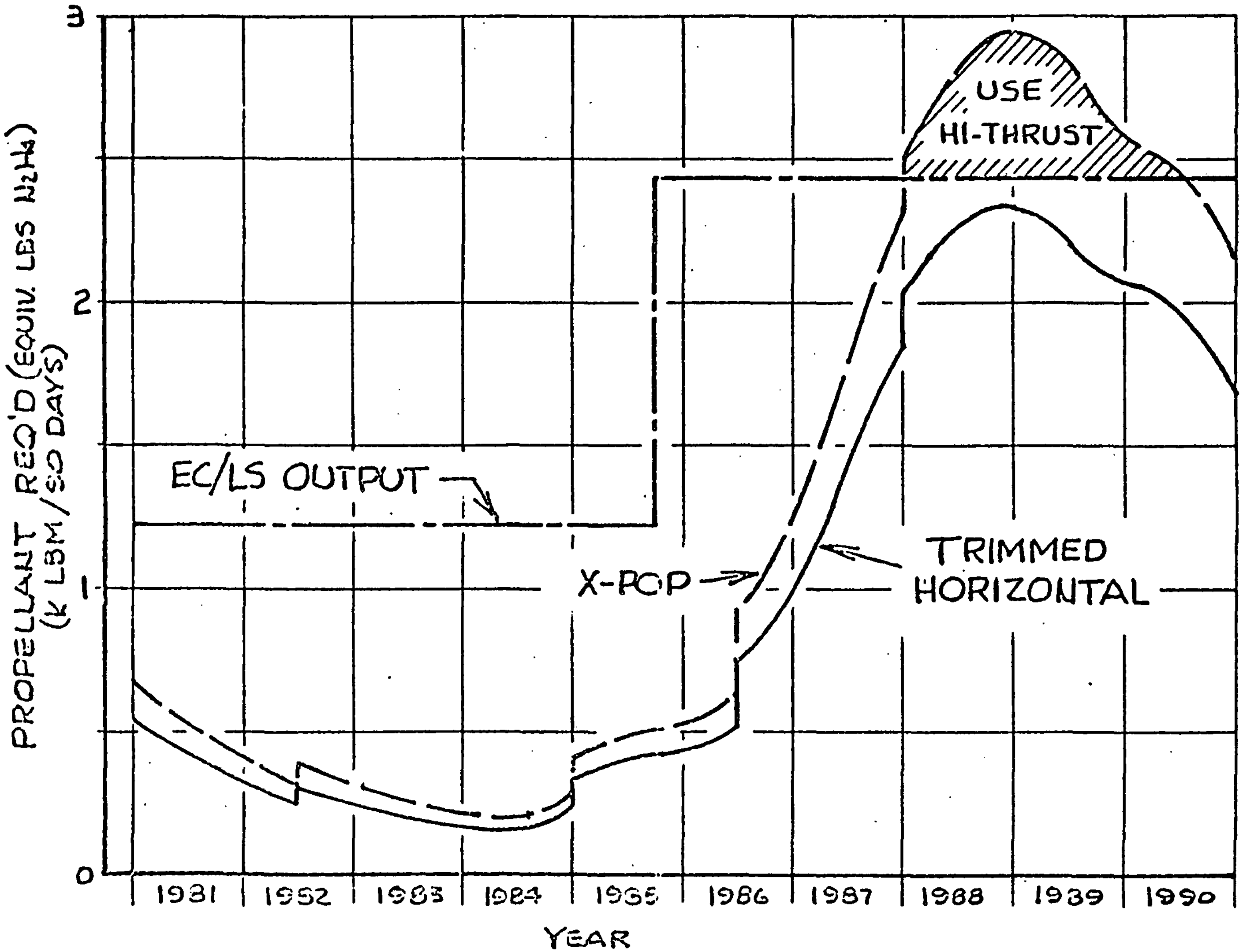


Figure 4.8-7. Orbit Keeping Propellant (242 NMI Circular)

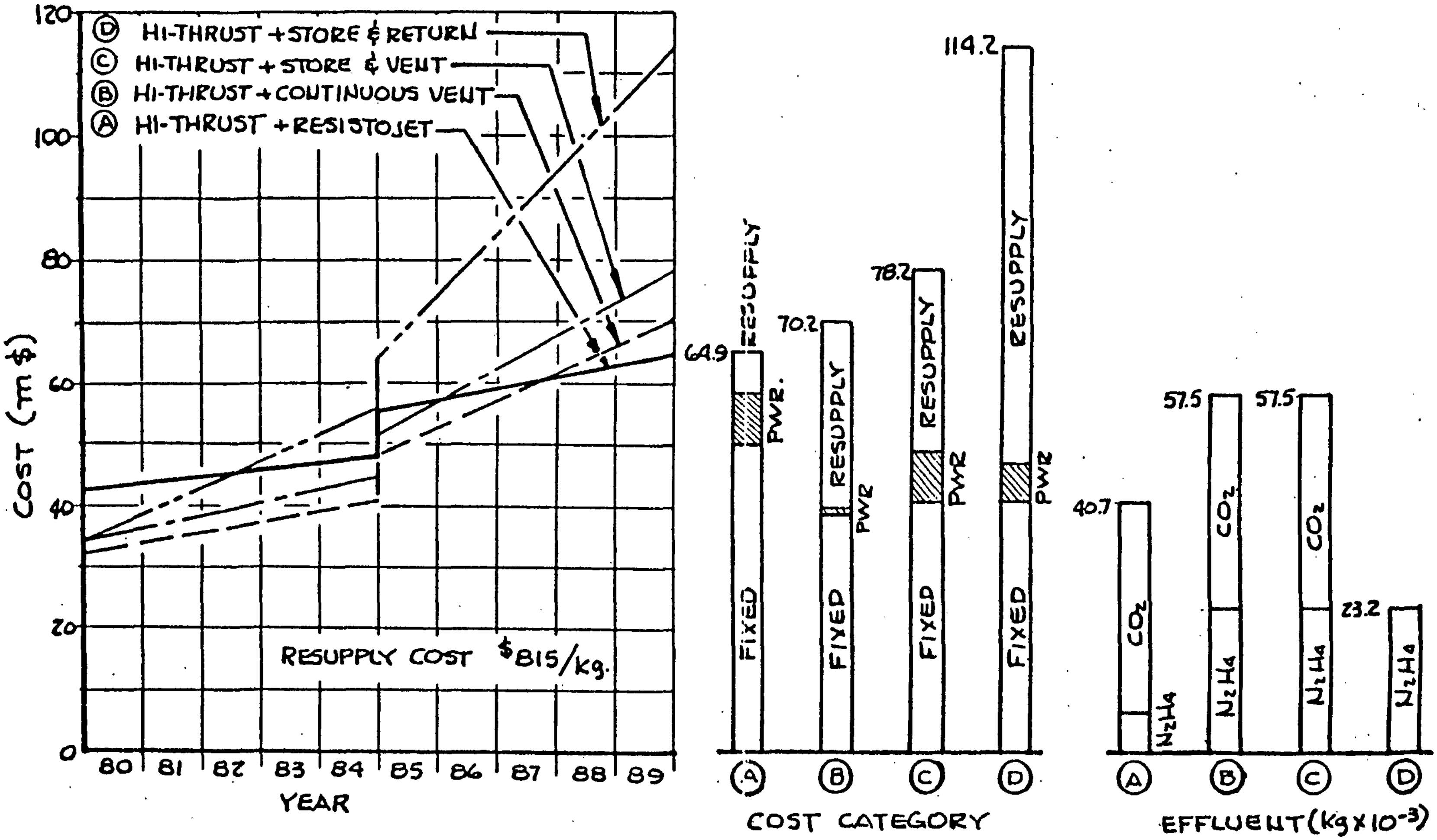


Figure 4.8-8. Cost Comparison – CO_2 Disposal (242 NMI Circular)

Table 4.8-8

SYSTEM COSTS (MILLION $)

Cost Category	System	System Selection (High Thrust Plus Molecular Sieve Plus....) Resistojet	Continuous Vent	Store & Dump	Store & Return
ISS Fixed	MOL Sieve	10.5	10.5	10.5	10.5
	CO_2 Pump	0.0	0.0	0.0	0.0
	High-Thrust	18.4	18.4	18.4	18.4
	Δ High-Thrust	0.0	3.3	3.3	3.3
	Low-Thrust	13.7	0.0	0.0	0.0
	Total	42.6	32.2	33.7	33.7
ISS Resupply	MOL Sieve	0.4	0.4	0.4	0.4
	CO_2 Pump Power	2.0	0.0	2.0	2.0
	CO_2 Return	0.0	0.0	0.0	12.0
	N_2H_4 (Maneuvers)	2.2	2.2	2.2	2.2
	N_2H_4 (Orbit Keep)	0.0	6.2	6.2	6.2
	Resistojet Power	0.7	0.0	0.0	0.0
	Total	5.3	8.8	10.8	22.8
ISS Total		47.9	41.0	44.5	56.5
GSS Fixed	MOL Sieve	3.1	3.1	3.1	3.1
	CO_2 Pump	0.0	0.0	0.5	0.5
	High-Thrust	3.0	3.0	3.0	3.0
	Δ High-Thrust	0.0	0.5	0.5	0.5
	Low-Thrust	1.5	0.0	0.0	0.0
	Total	7.6	6.6	7.1	7.1
GSS Resupply	MOL Sieve	0.8	0.8	0.8	0.8
	CO_2 Pump Power	4.0	0.0	4.0	4.0
	CO_2 Return	0.0	0.0	0.0	24.0
	N_2H_4 (Maneuvers)	3.2	3.2	3.2	3.2
	N_2H_4 (Orbit Keep)	0.0	18.6	18.6	18.6
	Resistojet Power	1.4	0.0	0.0	0.0
	Total	9.4	22.6	26.6	50.6
GSS Total		17.0	29.2	33.7	57.7
Total Fixed		50.2	38.8	40.8	40.8
Total Resupply		14.7	31.4	37.4	73.4
Grand Total		64.9	70.2	78.2	114.2

Table 4.8-9
FIXED COSTS
(Million $)

Area	Year							Totals		
	74	75	76	77	78	79	84	ISS	GSS	Total
Resistojet (CO_2)										
Development	0.50	1.50	2.50	4.00	3:00					
Procurement					0.70	1.38	1.38			
Launch Weight						0.06	0.06			
Launch Volume						0.06	0.06			
								13.70	1.50	15.20
High-Thrust (N_2H_4)										
Development	0.70	1.50	3.00	5.50	3.50					
Procurement					1.20	2.40	2.40			
Launch Weight						0.54	0.54			
Launch Volume						0.06	0.06			
								18.40	3.00	21.40
ΔHigh-Thrust (N_2H_4)*										
Development	0.25	0.50	0.75	1.00	0.50					
Procurement						0.30	0.50			
Launch Weight						----	----			
Launch Volume						----	----			
								3.30	0.50	3.80

*Additional high-thrust costs due to high duty cycle for orbit keeping.

Table 4.8-10

RESUPPLY AND POWER COSTS

Resupply Costs		lbm/mo (Avg)	$/lbm	Kg/Mo (Avg)	$/Kg	Total (m$)
ISS 60 Months	Molecular Seive	18	370	8.2	815	0.40
	N_2H_4 (Maneuvers)	65	578[1]	29.5	1272[1]	2.25
	N_2H_4 (Orbit Keep)	180	578[1]	82	1272[1]	6.24
	CO_2 Return	540[2]	370	246	815	12.00
GSS 60 Months	Molecular Seive	36	370	16.4	815	0.80
	N_2H_4 (Maneuvers)	93	578[1]	45.5	1272[1]	3.23
	N_2H_4 (Orbit Keep)	535	578[1]	243	1272[1]	18.60
	CO_2 Return	1080[2]	370	492	815	24.00

(1) $370 x 1.56 (Factor to include tankage and pressurant)
(2) Wt of bottles carried up to return CO_2

Power Costs		Avg Pwr (W)	$/w. 5 Yr.	Total (m$)
ISS	CO_2 Pumping	525	3825	2.01
	Resistojet	185	3825	0.71
GSS	CO_2 Pumping	1050	3825	4.02
	Resistojet	370	3825	1.43

Alternative B was not expected to be a viable candidate because it is not responsive to the effluent guideline that forbids uncontrolled discharge. It does, however, serve as a baseline minimum cost for initial development.

Alternate C is a minimum-response approach and considered only marginally acceptable regarding contamination. The system does not reduce the total mass of effluent but only controls the time of CO_2 discharge within a two-day limit. Due to the power and resupply costs, it is next to the highest total cost, \$13.3M over the minimum. Thus, this is not a favorable candidate.

Alternate D eliminates the emission of CO_2 but at a severe resupply cost penalty. Even though the initial cost is quite favorable, this is by far the most expensive and also most sensitive to resupply cost changes.

Alternate A, the High-Thrust plus CO_2 Biowaste Resistojet subsystem, meets the overall demands in a well-balanced manner when compared with Alternated C and D, the only viable candidates.

Regarding the minimization of effluents, the total mass is markedly reduced over the store and dump (Alternate C) because the CO_2 is usefully expelled and, therefore, reduces the N_2H_4 resupply and consequent consumption by some 16,800 Kg (37,000 lbm). This same fact also provides a system with the least sensitivity to resupply costs.

The comparison with the "CO_2 Store and Return" (Alternate D) regarding contamination is very nearly even. Even though the total effluent is higher with the dual system, the acount of the N_2H_4 exhaust products is markedly reduced. Thus, because CO_2 is considered less harmful than N_2H_4 and NH_3 from the high-thrust subsystem, the overall effect is quite even.

In other regards, not shown in these figures and tables, the dual system is very favorable. For example, reliability is enhanced in two ways:

1) The high-thrust subsystem usage is reduced.
2) The low-thrust subsystem is completely backed up by the high-thrust subsystem. Also, in the area of operational flexibility, the baseline CMG desaturation requires that the Space Station body axes by trimmed to eliminate bias torques on the CMG's. With a low-thrust system, other attitudes may be flown and the CMG desaturation accomplished while orbit keeping, without additional propellant requirements.

In summary, the High-Thrust plus CO_2 Resistojet subsystem was selected because it offers the lowest total cost, a very favorable contamination improvement, a very low resupply cost, reliability, and flexibility for a moderate increase in initial development costs.

4.8.4.2 Propellant Selection—High-Thrust Subsystem
The propellant selection was a major trace study in the 10-m (33-ft) diam Space Station. That study resulted in the selection of monopropellant hydrazine (N_2H_4). This section constitutes a review of that trade study to determine if the conclusions are valid for the Modular Space Station.

The major criteria used were (1) performance, weight, and volume, (2) design subsystem characteristics, (3) propellant storability, (4) plume contamination potential, and (5) cost. Supporting criteria were questions of reliability, maintainability, commonality, and safety. These criteria are equally valid for the Modular Space Station. The propellants considered in the study were (1) storable monopropellant (N_2H_4) and (3) cryogenic bi-propellant (O_2/H_2).

N_2H_4 was chosen in the 10-m (33-ft) diameter Space Station study chiefly because of the system simplicity offered by a single propellant, good storage characteristics, low contamination potential, and low development cost. The disadvantages of N_2H_4 are of less consequence on the Modular Space Station because of both the smaller system size and total propellant consumption. The Modular Space Station storage is 455 Kg vs 3180 Kg and total consumption is 4000 Kg vs 17,000 Kg for the 10-m (33-ft) diameter Space Station.

Monopropellant (N_2H_4) specific impulse is the lowest (180 sec in a pulsing mode) however, even when compared with O_2/H_2 at a system impulse of 330 sec, the total resupply penalty is only \$2. 3 m. This is more than overcome by lower development costs and commonality with the free-flying RAMs.

The other area that monopropellant was not the most favorable choice is surface contamination from exhaust products. O_2/H_2 is considered less contaminating but only by a small margin. This advantage is further reduced by the low consumption. There is no reason to alter the choice of propellant. The simplicity of the monopropellant system is even more significant because there are a number of inter-module connections that must be made during the build-up phase on the Modular Space Station. These are reduced by a factor of two.