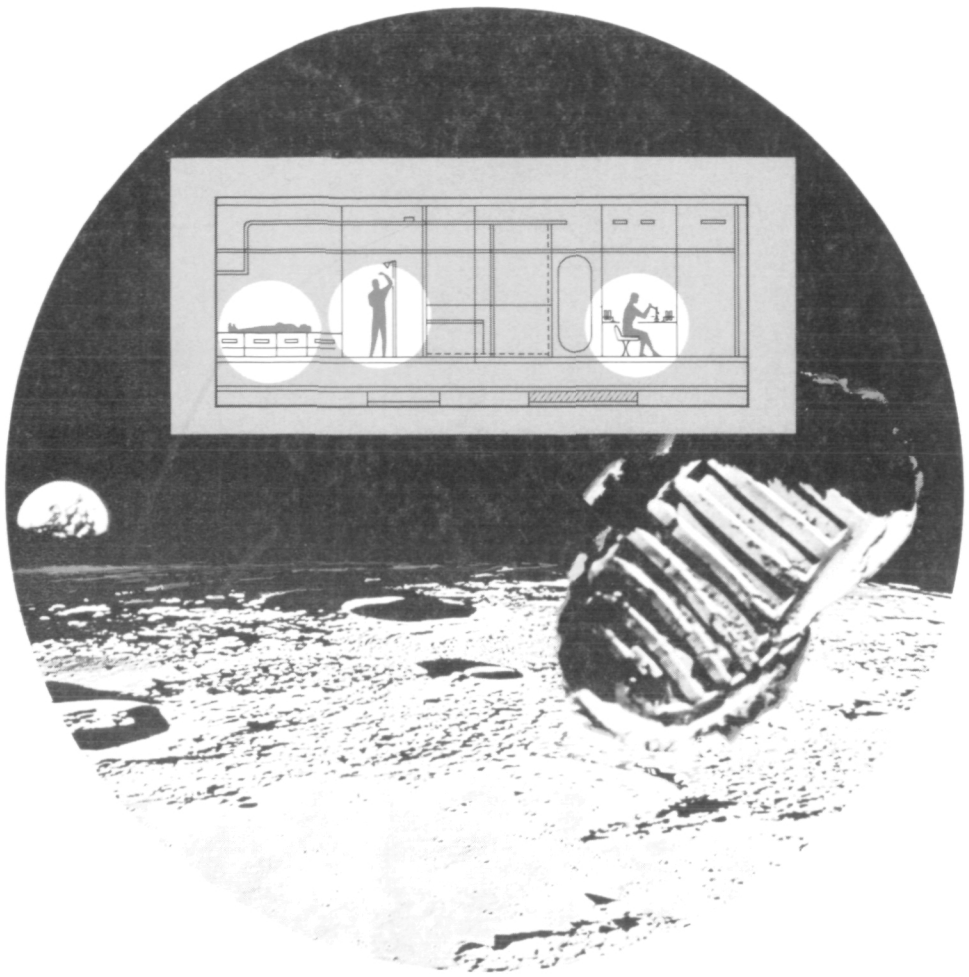
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VOL IV



ORBITING LUNAR STATION (OLS)

PHASE A FEASIBILITY AND DEFINITION STUDY

VOLUME IV

CONFIGURATION AND SYSTEMS ANALYSIS

FINAL REPORT

CASE FILE COPY

Space Division
North American Rockwell

APRIL 1971
Prepared by Advanced Program Engineering

12214 Lakewood Boulevard, Downey, California 90241

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VOLUME IV

ORBITING LUNAR STATION PHASE A FEASIBILITY AND DEFINITION STUDY

VOLUME IV

CONFIGURATION AND SYSTEMS ANALYSIS

(FINAL REPORT)

APRIL 1971

Approved by

L. R. Hagan
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Space Division
North American Rockwell

FOREWORD

This report contains the results of North American Rockwell's analyses conducted under the Orbiting Lunar Station Feasibility and Definition Study (Phase A), Contract NAS9-10924, in accordance with line item 5 of the Data Requirements List (DRL5).

This report is compiled in six volumes for ease of presentation, handling, and readability of the data in the report. In general, each volume is a compilation of the data generated in a specific phase of the study.

This is Volume IV of the report and contains Orbiting Lunar Station (OLS) configurational and subsystems analysis data. The data is presented in parametric form for crew sizes of 4, 8, and 12 men and vehicle diameters of 12, 15, 22 and 33 feet.

The documents comprising the study report are:

Volume I	OLS Objectives
Volume II	Mission Operations and Payloads Analysis
Volume III	OLS Performance Requirements
Volume IV	OLS Configuration and Systems Analysis
Volume V	OLS Configuration Definition
Volume VI	Comparison of OLS Configuration

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1.0 INTRODUCTION AND SUMMARY

This section presents a description and summary of the contents of Volume IV of the OLS final report. In this volume are presented the results of parametric OLS configuration design studies and the characteristics and selection trade study data for various OLS subsystem options. No configuration design or subsystem concept selections are made as this report is limited, by design, to comparisons of concepts and design approaches, not selection of a specific design or subsystem concept approach. The data presented herein was utilized, however, to select design configurations and subsystems concepts for the representative OLS (20 to 33-foot diameter vehicle) and the derivative OLS (based on the earth orbiting Modular Space Station) which are described in Volume V.

1.1 OLS PARAMETRIC CONFIGURATION STUDIES

Section 2.0 of this volume presents the results of configurational design studies. These studies investigated OLS concepts for crew sizes of 4, 8, and 12 men which would satisfy the program objectives, operational criteria, and structural performance requirements documented in Volumes I, II, and III, respectively. Structural parameters which were varied during these analyses were:

Basic vehicle diameter - 12, 15, 22, and 33 feet
Floor orientation - transverse and longitudinal

In these studies, conical, flat, and toroidal pressure bulkhead shapes were considered. Floor arrangement drawings were prepared for each of the above parametric combinations of diameter, floor orientation, and crew size. A weights analysis was performed for each of these OLS concepts.

Based on these analyses it was concluded that for an 8-man OLS required crew size:

1. For modules 12 or 15 feet in diameter limited in length to 30 and 60 feet, respectively, so as to be compatible with the cargo bay of an Earth Orbit Shuttle (EOS) vehicle:

   a. Transverse floor arrangements are more structurally efficient, i.e. yield more usable floor space per pound of module weight, than longitudinal floor arrangements.

   b. The stations composed of 15-foot diameter modules are more structurally efficient than the stations composed of 12-foot diameter modules.

2. For single module concepts with diameters ranging from 20 to 33 feet with no limitations as to length, the Representative OLS groundrule size:

   a. Transverse floor arrangements are more structurally efficient than longitudinal floor arrangements.

   b. The effect of diameter on structural efficiency is small, larger diameters being only slightly preferred.

It should be noted that the conclusion documented above regarding 12 and 15-foot diameter modular concepts did not consider the under floor tunnel arrangement employed in the preferred baseline concept of the earth orbiting Modular Space Station (MSS) design, upon which the Derivative OLS, described in Volume V, is based. This tunnel minimizes aisleway area and hence increases the amount of useful floor area in a longitudinal floor concept, allowing its structural efficiency to approach that of a transverse floor concept. Other advantages of longitudinal floors which led to their selection in the baseline preferred MSS concept are:

1. Longitudinal floor arrangements are less sensitive to module length.

2. Longitudinal floor arrangements provide more single deck area which is preferred for some functions, laboratories in particular.

Artificial gravity criteria to be applied to OLS design configurations is also presented in Section 2.0. These criteria were adopted for OLS use from criteria developed for the Earth Orbiting Space Station (EOSS) and Space Base application. These criteria are based on knowledge of physiological reactions and limitations of crew members in an artificial gravity environment and an evaluation of artificial gravity and coriolis force components resulting from typical crew activities in a space station environment.

1.2 OLS SUBSYSTEM CONCEPTS

Trade data, concept trade trees, and trade study results for OLS subsystem concepts are presented in Section 3.0 for the following subsystems.

1. Environmental Control and Life Support (ECLSS)

2. Electrical Power (EPS)

3. Information (ISS)

4. Guidance and Control (G&C)

5. Reaction Control (RCS)

6. Environmental Protection (ENPS)

Subsystem synthesis parameters considered in the analyses were:

1. Initial, development and resupply relative cost data
2. Initial and resupply weights
3. Safety and Reliability
4. Maintenance and repair considerations
5. Power requirements
6. Heat rejection requirements
7. Volumetric requirements

Wherever applicable, data for these parameters was compiled for various functional concepts capable of satisfying the subsystem functional and performance requirements documented in Volume III. Where appropriate, the data are presented parametrically as a function of crew size (4 to 12 men) and vehicle diameter (12 to 33 feet). In many cases, trade data generated in the EOSS study were re-employed and reprinted herein.

Following each subsystem analysis a section discussing the impact of artificial gravity simulation is included. Consideration of the changes required in each subsystem area if the OLS were required to operate in an artificial gravity, rather than a zero gravity environment, are presented.

2.0 CONFIGURATION DESIGN STUDIES

This section documents the trade studies and analyses performed to provide quantitative data that delineates the interrelationship between several OLS configurational parameters of major interest. These analyses were performed to develop parametric data that would provide rationale for selecting a particular OLS configurational arrangement for a specific crew size.

In these studies the effect on structural weight and availability of floor space as a function of module configuration were key areas of investigation. To provide a basis for conducting the habitability and weights analyses, it was considered essential to prepare a series of gross general arrangement drawings of the OLS that illustrated the effect of each parameter variation. The configurational trade studies were primarily concerned with the investigations of the effect on OLS vehicle weight resulting from variations in number of crew, body diameter, end bulkhead configuration, and internal floor orientation. Subsequently, weight estimates for all of the configurations were generated. In addition, the surface areas, pressurized volumes, gross floor areas, and usable floor areas were calculated. These data were then used for plotting trend curves of the key factors previously mentioned. Detailed discussion of the design assumptions, parametric configurations, final results of the design trade studies weights analysis, and consideration of OLS adaptability to artificial-g operation are provided in the following sections.

2.1 CONFIGURATIONAL GROUNDRULES

An initial step in the design trade studies was to select values and specific ranges of variation for the major parameters previously identified. The following list summarizes the range of variation in these particular parameters selected with customer concurrence:

Number of crew - 4, 8, and 12
Basic vehicle diameter - 12, 15, 22, and 33 feet
Floor orientation - transverse and longitudinal
Bulkhead shape - conical, flat, and toroidal

In addition to these constraints, it was considered necessary to select other second-order limits and to proceed with specific groupings in some cases. In the case of body diameters, the 22- and 33-foot-diameter vehicles were not limited in length, but the 12- and 15-foot-diameter modules were limited to 30 and 60 feet to be compatible with current estimates of the payload compartments of the Earth Orbit Mini Shuttle and Shuttle vehicles, respectively. Although bulkhead configurations were identified as a variable, in order to reduce the number of configurations to a manageable size, specific bulkhead

configurations were selected for the various floor orientations and body diameters. These selections were based upon data and trade studies from current studies, such as Space Tug, Earth Orbit Space Station, and Shuttle-Launched Space Station. The basis for the specific selection included minimum weight and compatibility with the structural and internal arrangement of the module. These selections are shown in the following tabulation:

Body Diameter (ft)	Floor Orientation	Selected Bulkhead Shape
33	Transverse	Toroidal
33	Longitudinal	Flat
22	Transverse	Toroidal
22	Longitudinal	Flat
15	Transverse	Conical
15	Longitudinal	Conical
12	Transverse	Conical
12	Longitudinal	Conical

The primary configurational driver from spatial considerations is the floor area requirement of the basic vehicle. It was necessary, therefore, to compile a list of the major elements that require floor space and to establish values for each. Table 2.1-1 lists the principal elements and the variation in the specific requirements as a function of the number of crew. These values were based upon the requirements established in Volume III of this report, but were suitably modified in some cases where necessary. The OLS safety and crew habitability criteria established in Volume III were observed in the interior design and floor layout of each OLS candidate configuration. In addition to the requirements for spatial allocations shown in Table 2.1-1, it was considered necessary to establish limits on the diameter of the interdeck hatches and width of the aisles to ensure configurational consistency in all of the layouts. In the case of the 22- and 33-foot-diameter vehicles, all interdeck hatches were made 60 inches to permit movement of large cargo packages of this size throughout the vehicle. This is consistent with current EOSS studies and is considered desirable because the 22- and 33-foot-diameter vehicles are single module, and cargo transported to the station must be brought into the vehicle. In the case of the 12- and 15-foot-diameter vehicles, however, the interdeck hatches were reduced to 36 inches to conserve floor space. The stations configured from 12- and 15-foot-diameter modules will always be clustered arrangements with the modules probably docked to a central core module or multiple docking adapter; cargo packages can be taken into the vehicle and broken down into smaller sizes that are compatible with the 36-inch-diameter interdeck hatches of the attached station modules.

Table 2.1-1. OLS Internal Spatial Allocation (ft^2)

Area	Crew Size 4-Man Total	8-Man Total	12-Man Total	Remarks
Stateroom	200	400	600	
Galley (primary)	36	56	66	
Galley (backup)	8	12	18	
Dining	44	44		8-man crews dine in shifts
Recreation	60	80		
Exercise	32	60	60	
Medical	144	144	144	For 15-, 22-, and 33-ft diameter
Medical	90	90	90	For 12-ft diameter only; see Note 3
Personal hygiene (primary)	48	96	96	For 15-, 22-, and 33-ft diameter
(secondary)	41	96	96	
Personal hygiene (primary)	48	90	90	For 12-ft diameter only; see Note 3
(secondary)	41	90	90	
Primary control	80	140	140	For 15-, 22-, and 33-ft diameter
Secondary control	70	120	120	
Primary control	80	90	90	For 12-ft diameter only; see Note 3
Secondary control	70	90	90	
Experiments	300	550	550	
Subsystem installation	220	324	360	
Storage requirements	119	143	166	For transverse floors, see Notes 1 and 4

Additional assumptions:

1. Internal passageways — 60-in. diameter hatches in 22- and 33-ft diameter modules; 36-in. diameter hatches in 12- and 15-ft diameter modules
2. Aisles — 30-in. width in 15-, 22-, and 33-ft diameter modules; 24-in. width (floor) 30-in. width (shoulder) in 12-ft dia. modules

Notes:

1. Storage requirements are defined in Section 9.0, Volume II.
2. The above table lists requirements not the allocated areas of any particular configuration.
3. The useful floor area on one transverse dock of a 12-ft dia. module totals 90 square feet.
4. For longitudinal floor arrangements, the subsystems are located below the floors or above the false ceilings and do not use floor space

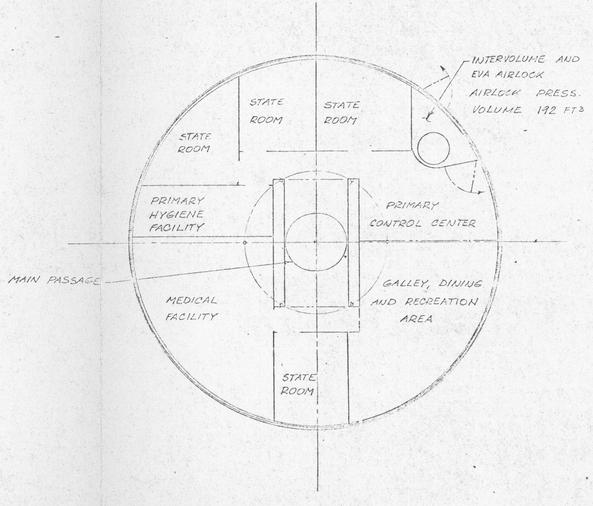
Aisle space within the modules also were selectively established in a similar manner. Thirty-inch-wide aisles were considered satisfactory in the 15-, 22-, and 33-foot-diameter modules, but for the 12-foot-diameter modules, the aisles were reconfigured to provide 24 inches to the floor and approximately 30 inches at shoulder height.

A spatial allocation has been provided for onboard subsystems and is shown as the final entry in Table 2.1-1. The subsystem trades and selection operations were conducted concurrently with the configurational trade studies; therefore, it was necessary to make assumptions for subsystem floor space requirements. EOSS data were used as a basis for the preliminary values shown but were modified as appropriate for the OLS. It should be noted that the allocation shown is for the 12- and 15-foot-diameter modules featuring a transverse floor arrangement. The longitudinal floor arrangements result in considerable volume above the ceiling and below the floor for subsystem accommodations; therefore, no specific spatial allocation is necessary in establishing the module size. In the case of the transverse floor orientation, there is no free volume above or below the floors, and it is necessary to take this into consideration and provide spatial allocation. In the case of the 22- and 33-foot-diameter modules featuring transverse floors, the subsystems are accommodated in the space available between the floor and ceilings and the toroidal pressure bulkheads, thus, no special floor allocation is necessary for these concepts.

2.2 CONCEPT DEFINITION

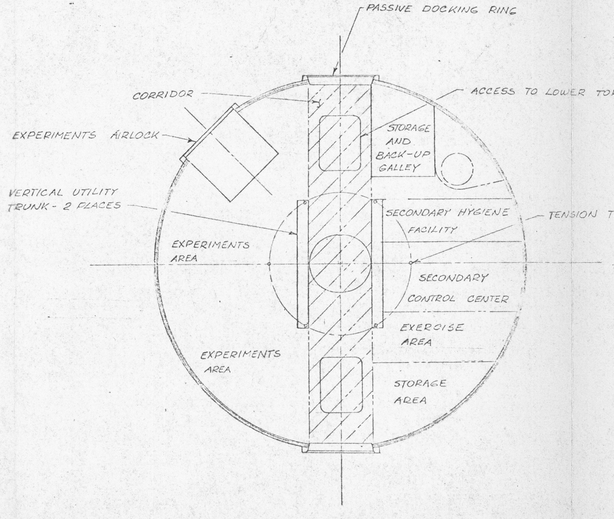
As previously identified in subsection 2.1, three basic configurational parameters were used to establish the spectrum of general arrangement configurations to be generated in this task. A total of 24 individual general arrangement drawings were prepared to cover the three crew sizes, four diameters, and two floor orientations and were individually based upon the floor spatial requirements in Table 2.1-1. The concept drawings (2282-1 through 2282-24) provide views and sections of each station concept and illustrate the gross physical characteristics and spatial arrangements and allocations for each deck. Pressure compartmentation concepts are shown for the 22- and 33-foot-diameter modules. In the case of the 12- and 15-foot-diameter modules, compartmentation is assumed to exist as a result of the modules being clustered about a central core module or multiple docking adapter. A detailed discussion of each drawing is not provided in this section because the spatial requirements upon which the configurations are based is discussed in subsection 2.1. The configurations are straightforward and self-explanatory.

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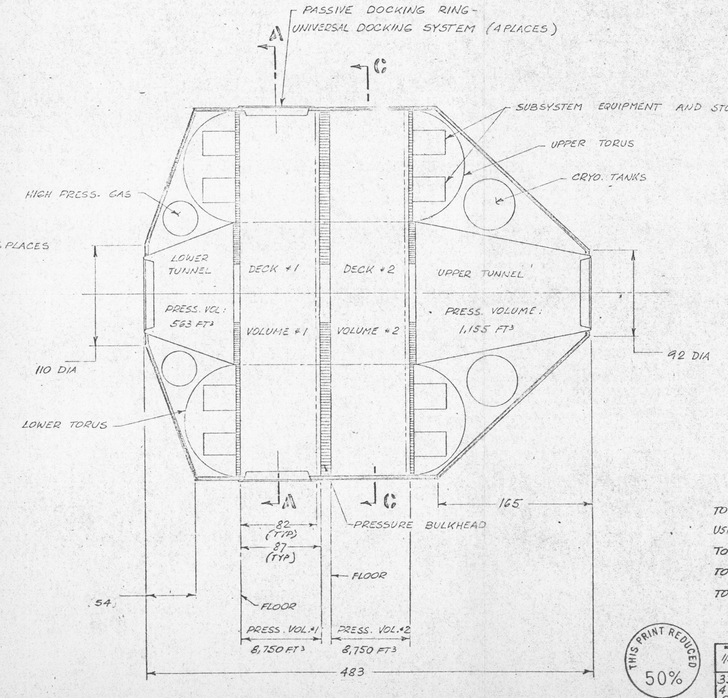
SECTION C-C

DECK #2
VOLUME #2



SECTION A-A

DECK #1
VOLUME #1



TOTAL STORAGE VOLUME: 940 FT³
USEFUL FLOOR AREA: 1,510 FT²
TOTAL FLOOR AREA: 1,640 FT²
TOTAL OUTER AREA: 4,480 FT²
TOTAL PRESS. VOLUME: 19,218 FT³

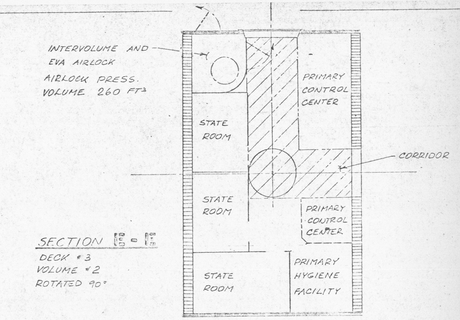
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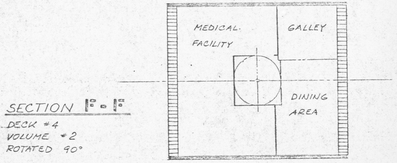
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1/40	DATE 7/27/71	NORTH AMERICAN ROCKWELL CORPORATION
	MODEL	12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA

33 FT DIA, TRANSVERSE FLOOR - 4 MAN OLS MODULE, OLS STUDY

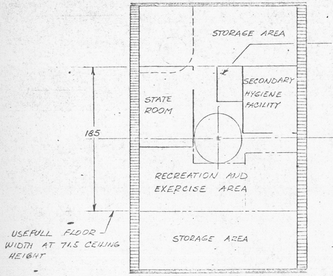
2282-1



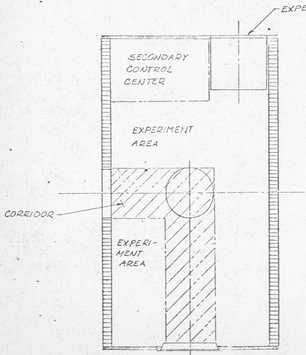
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DECK # 3
VOLUME # 2
ROTATED 90°



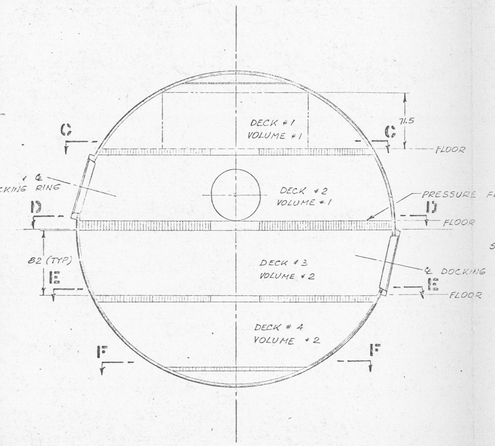
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DECK # 4
VOLUME # 2
ROTATED 90°



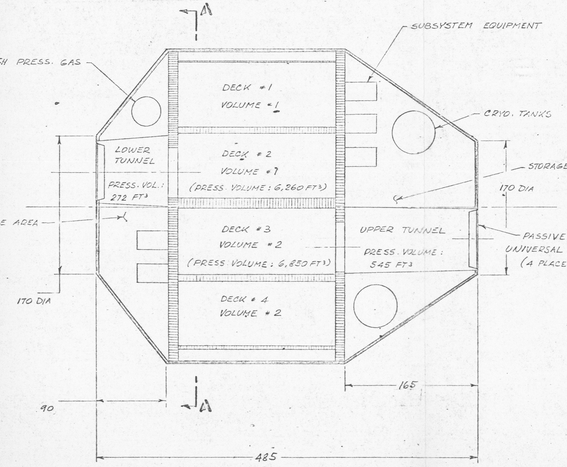
SECTION G - G
DECK # 1
VOLUME # 1
ROTATED 90°



SECTION D - D
DECK # 2
VOLUME # 1
ROTATED 90°
NOTE: EXPERIMENTAL EQUIPMENT TO BE MOVED FROM CORRIDOR AREA DURING CARGO TRANSFER



SECTION A - A



TOTAL STORAGE VOLUME: 800 FT³
(400 FT³ PRESS. VOL., 400 FT³ NON PRESS. VOL.)
USEFUL FLOOR AREA: 1,413 FT²
TOTAL FLOOR AREA: 1,553 FT²
TOTAL OUTER AREA: 4,420 FT²
TOTAL PRESS VOLUME: 13,927 FT³

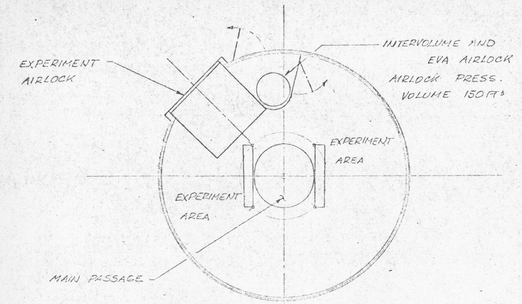
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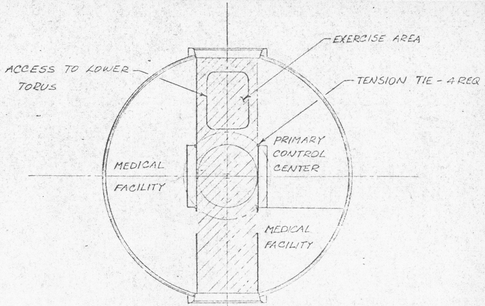
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	DATE	NORTH AMERICAN ROCKWELL CORPORATION
	MODEL	12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA

33 FT DIA, LONGITUDINAL FLOOR-4 MAN, OLS MODULE, OLS STUDY

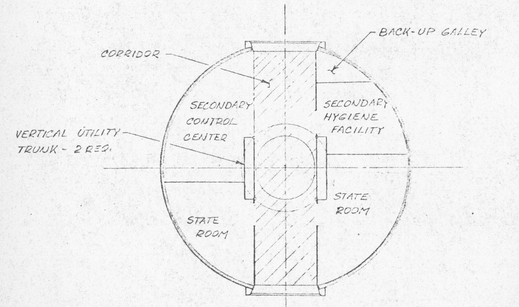
2282-2



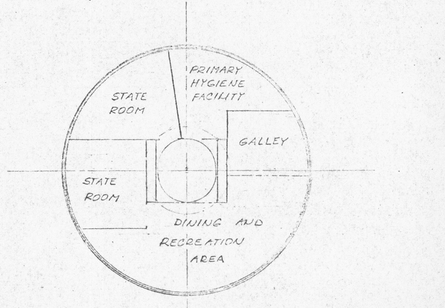
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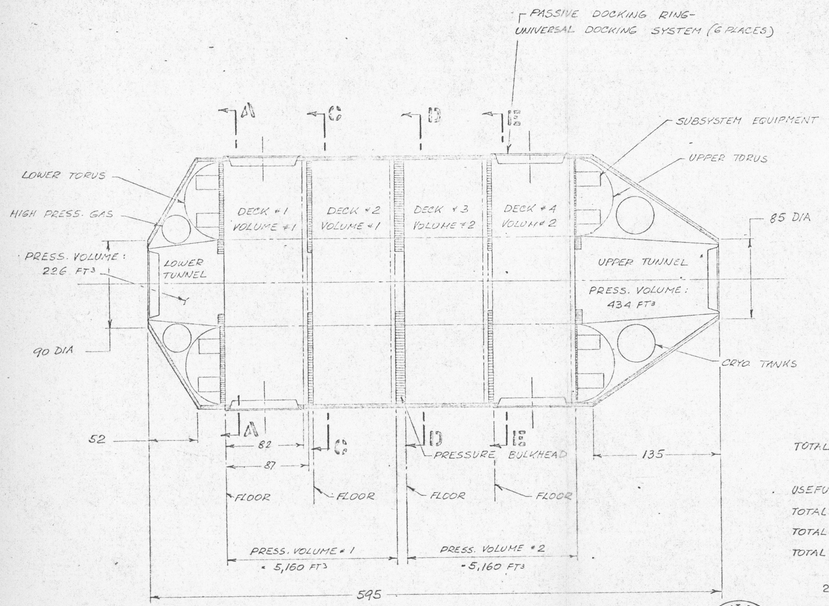
SECTION A-A
DECK # 1



SECTION E-E
DECK # 4



SECTION C-C
DECK # 2



TOTAL STORAGE VOLUME: 750 FT³
(NON-PRESS. VOLUME)
USEFUL FLOOR AREA: 1,378 FT²
TOTAL FLOOR AREA: 1,424 FT²
TOTAL OUTER AREA: 3,420 FT²
TOTAL PRESS. VOLUME: 10,980 FT³

2-9, 2-10 SD 71-207

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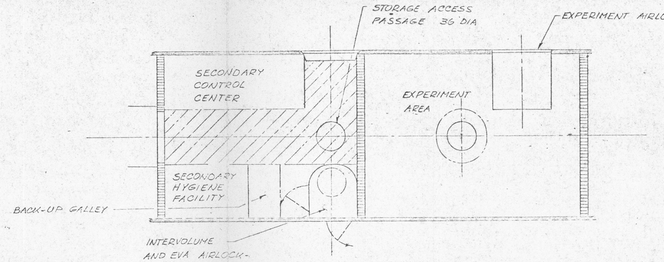
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North American Rockwell

SCALE	DR.	SPACE DIVISION
1/40	DATE	NORTH AMERICAN ROCKWELL CORPORATION
	MODEL	12214 LAKEWOOD BOULEVARD DOWNEY, CALIFORNIA

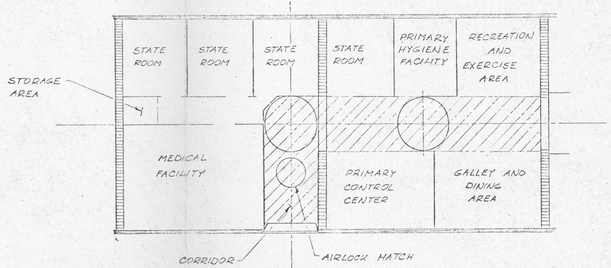
22 FT DIA. TRANSVERSE FLOOR - 4 MAN OLS MODULE, OLS STUDY

2282-3

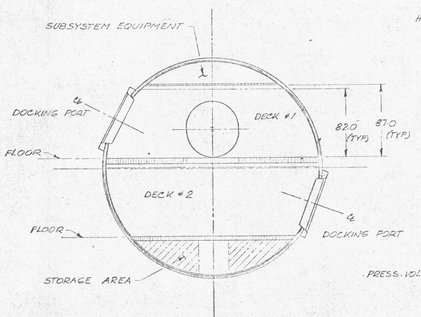
Space Division
North American Rockwell



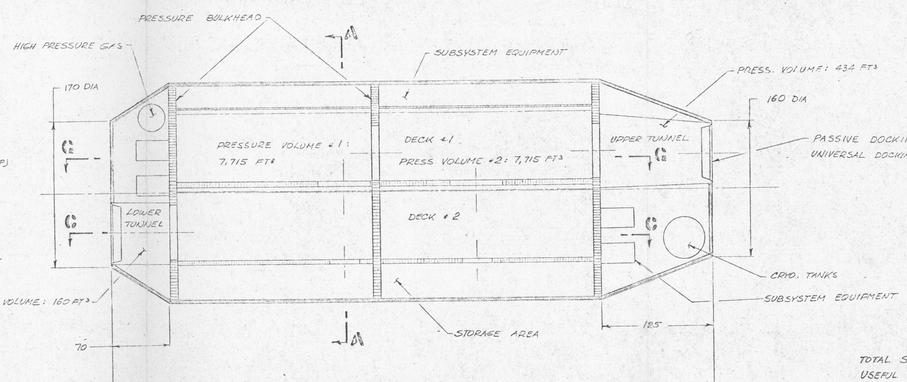
SECTION C - C
DECK #2



SECTION G - G
DECK #1



SECTION A - A



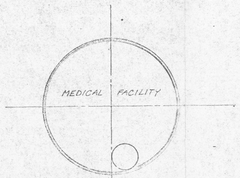
TOTAL STORAGE VOLUME: 800 FT³
USERUL FLOOR AREA: 1,645 FT²
TOTAL FLOOR AREA: 1,503 FT²
TOTAL OUTER AREA: 4,336 FT²
TOTAL PRESS. VOLUME: 16,020 FT³

2-11, 2-12 SD 71-207

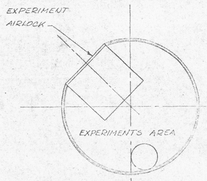
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SCALE: 1/10	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
22 FT DIA. LONGITUDINAL FLOOR-4 MAN OLS MODULE, OLS STUDY		2282-4A

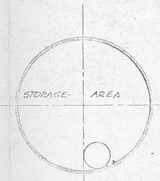
Space Division
North American Rockwell



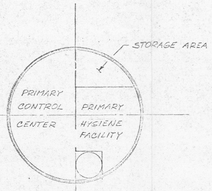
SECTION C-C
DECK # 6
MODULE # 2



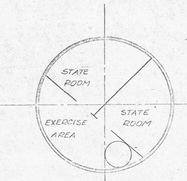
SECTION B-B
DECK # 4
MODULE # 2



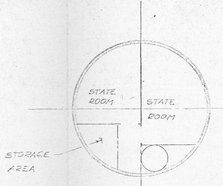
SECTION C-C
DECK # 2
MODULE # 2



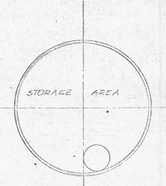
SECTION F-F
DECK # 5
MODULE # 1



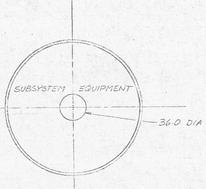
SECTION E-E
DECK # 4
MODULE # 1



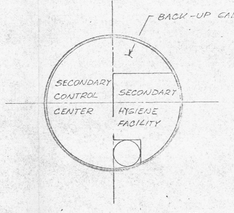
SECTION D-D
DECK # 3
MODULE # 1



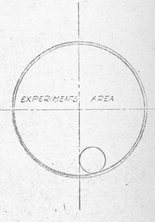
SECTION C-C
DECK # 2
MODULE # 1



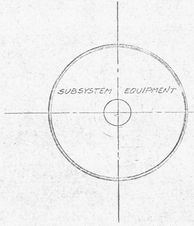
SECTION A-A
DECK # 1
MODULE # 1



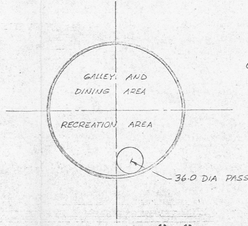
SECTION F-F
DECK # 5
MODULE # 2



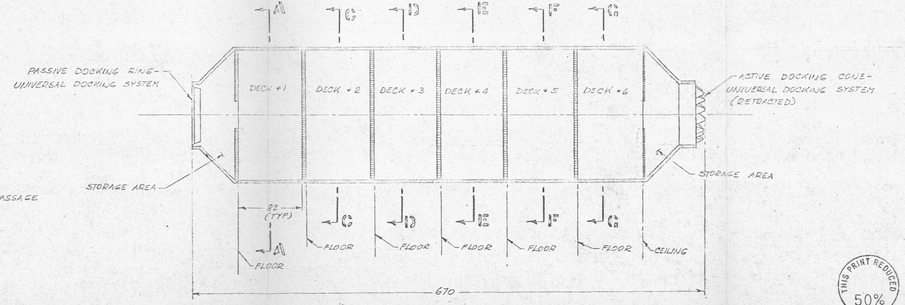
SECTION D-D
DECK # 3
MODULE # 2



SECTION A-A
DECK # 1
MODULE # 2



SECTION G-G
DECK # 6
MODULE # 1



2-13, 2-14 SD 71-207

TOTAL STORAGE VOLUME: 1,200 FT³
USEFUL FLOOR AREA: 1,785 FT²
TOTAL FLOOR AREA: 1,965 FT²
TOTAL OUTER AREA: 4932 FT²
TOTAL PRESS. VOLUME: 14,620 FT³ (7,310 FT³ PER MODULE)
NUMBER OF MODULES 2

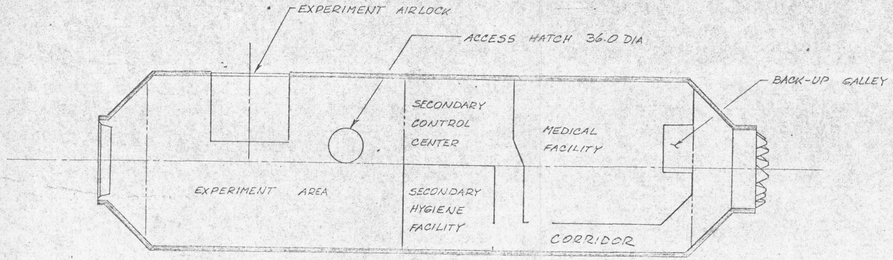
SCALE	BY	SPACE DIVISION
1/40	DATE	NORTH AMERICAN ROCKWELL CORPORATION
MODEL		SEAL BEACH, DOWNEY, CALIFORNIA

15 FT DIA. TRANSVERSE FLOOR-4 MAN OLS MODULE, OLS STUDY

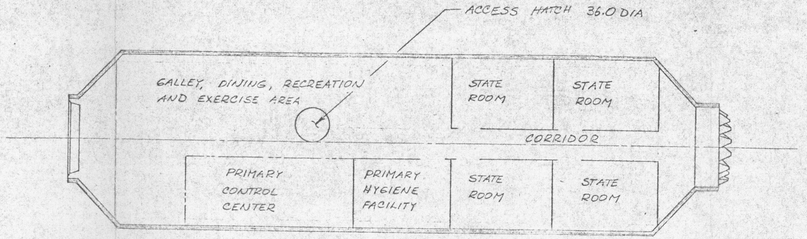
2282-5A

THIS PRINT REDUCED 50%

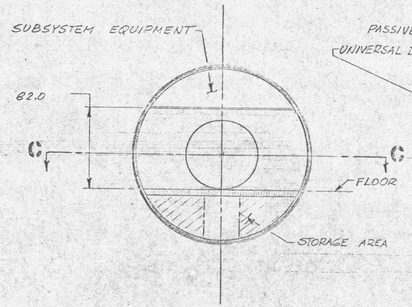
Space Division
North American Rockwell



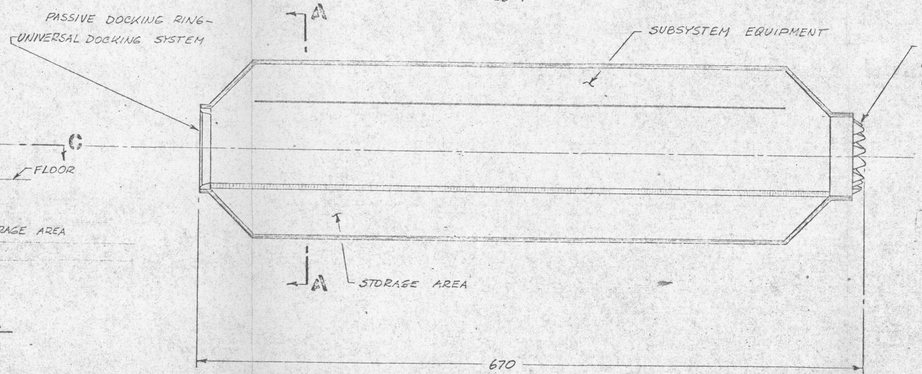
SECTION C - C
MODULE # 2
ROTATED 90°



SECTION C - C
MODULE # 1
ROTATED 90°



SECTION A - A



2-15, 2-16 SD 71-207

TOTAL STORAGE VOLUME: 1,600 FT³
USEFUL FLOOR AREA: 1,130 FT²
TOTAL FLOOR AREA: 1,170 FT²
TOTAL OUTER AREA: 4932 FT²
TOTAL PRESS VOLUME: 14,620 FT³
(7,310 FT³ PER MODULE)

NUMBER OF MODULES: 2

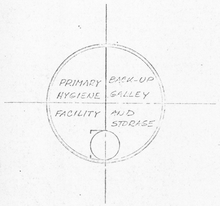
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SCALE 1/40	DR.	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA
	DATE 7/24/71	
	MODEL	

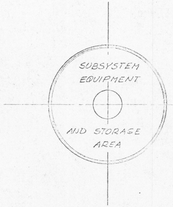
15 FT DIA. LONGITUDINAL FLOOR - 4 MAN OLS MODULE - OLS STUDY

2282-6

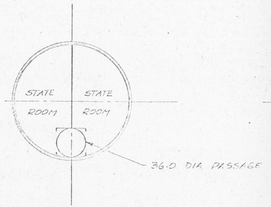
Space Division
North American Rockwell



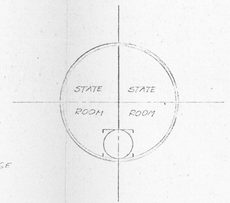
SECTION B - B
DECK = 3
MODULE = 2



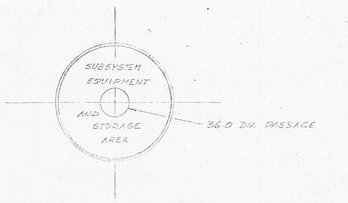
SECTION A - A
DECK = 1
MODULE = 2



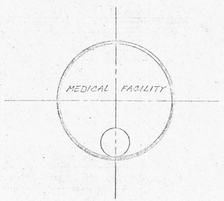
SECTION B - B
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MODULE = 1



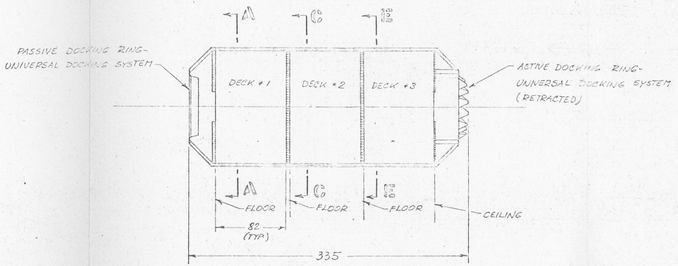
SECTION C - C
DECK = 2
MODULE = 1



SECTION A - A
DECK = 1
MODULE = 1



SECTION C - C
DECK = 2
MODULE = 2



2-17, 2-18 SD 71-207

TOTAL STORAGE VOLUME: 1,200 FT³
USEFUL FLOOR AREA: 1,655 FT²
TOTAL FLOOR AREA: 1,895 FT²
TOTAL OUTER AREA: 6,249 FT²
TOTAL PRESS. VOLUME: 14,154 FT³
(2,359 FT³ PER MODULE)

NUMBER OF MODULES: 6

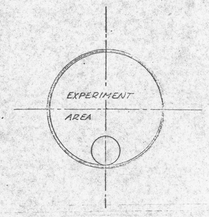
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SCALE	SHT 1	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION SEAL BEACH, CALIFORNIA	
1/40	SHT 2		
	MODEL		

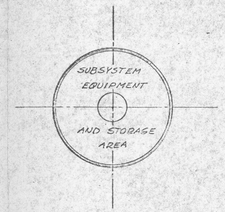
12 FT DIA, TRANSVERSE FLOOR - 4 MAN OLS MODULE, OLS STUDY

2282-7A
SHT 1 OF 2

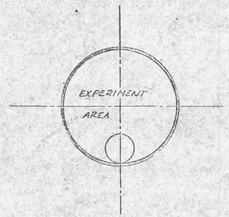
Space Division
North American Rockwell



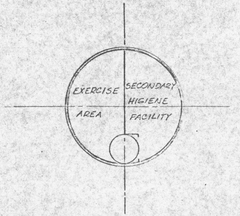
SECTION E - E
DECK = 3
MODULE = 6



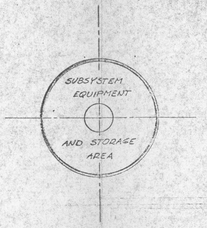
SECTION A - A
DECK = 1
MODULE = 6



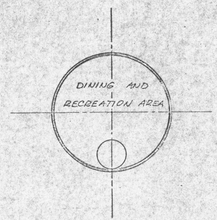
SECTION C - C
DECK = 2
MODULE = 5



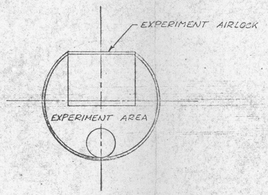
SECTION E - E
DECK = 3
MODULE = 4



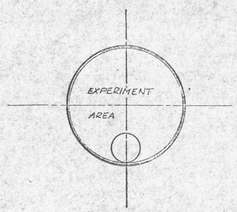
SECTION A - A
DECK = 1
MODULE = 4



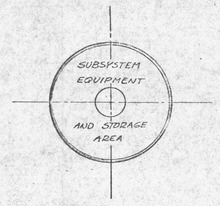
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DECK = 2
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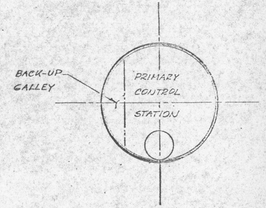
SECTION C - C
DECK = 2
MODULE = 6



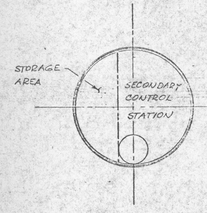
SECTION E - E
DECK = 3
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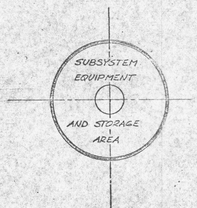
SECTION A - A
DECK = 1
MODULE = 5



SECTION C - C
DECK = 2
MODULE = 4



SECTION E - E
DECK = 3
MODULE = 3



SECTION A - A
DECK = 1
MODULE = 3

2-19, 2-20 SD 71-207

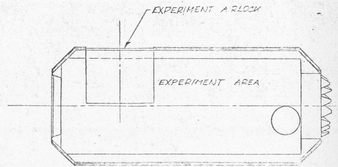
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SCALE 1/10	DR. J.C. 4/10	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA
	DATE 12/6/71	
	MODEL	

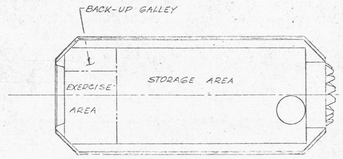
12 FT DIA TRANSVERSE FLOOR - 4 MAN OLS MODULE
OLS STUDY

2282-7A
SHT 2 OF 2

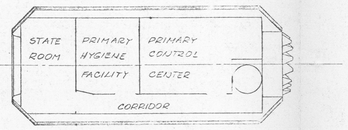
Space Division
North American Rockwell



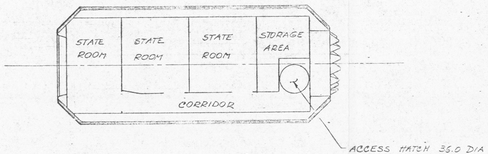
SECTION C - C
MODULE #5



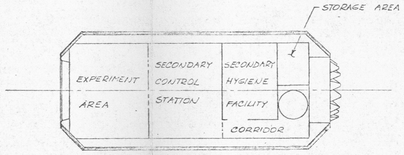
SECTION C - C
MODULE #4



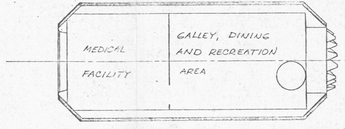
SECTION C - C
MODULE #2



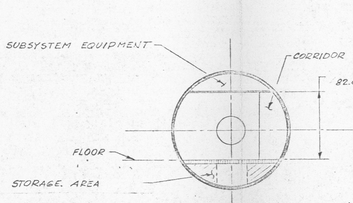
SECTION C - C
MODULE #1



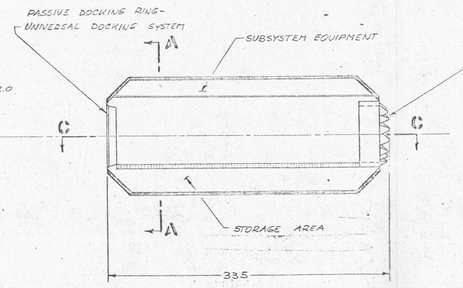
SECTION C - C
MODULE #5



SECTION C - C
MODULE #3



SECTION A - A



2-21, 2-22 SD 71-207

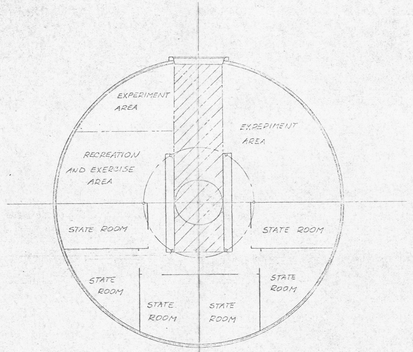
TOTAL STORAGE VOLUME: 2,000 FT³
USEFUL FLOOR AREA: 1,263 FT²
TOTAL FLOOR AREA: 1,323 FT²
TOTAL OUTER AREA. 6,264 FT²
TOTAL PRESS. VOLUME: 14,514 FT³
(2,419 FT³ PER MODULE)
NUMBER OF MODULES: 6

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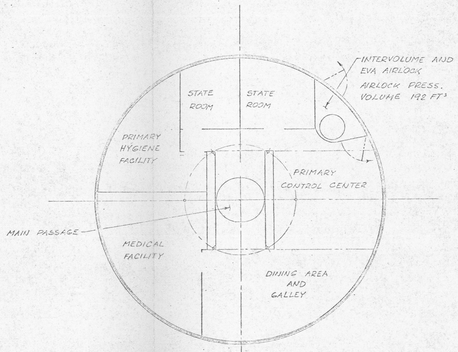
SCALE	DR.	SPACE DIVISION
1/2	DATE	NORTH AMERICAN ROCKWELL CORPORATION
	MODEL	12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA

12 FT DIA. LONGITUDINAL FLOOR 4 MAN OLS MODULE, OLS STUDY

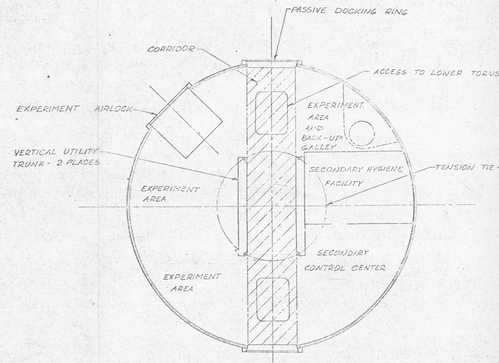
2282-8



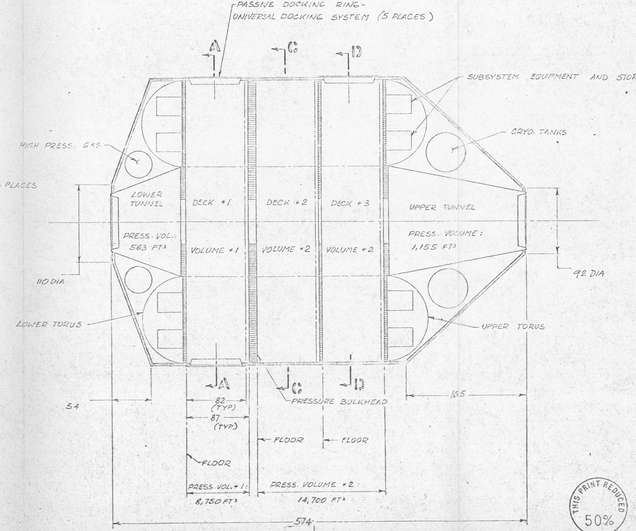
SECTION B-B
DECK #3



SECTION C-C
DECK #2



SECTION A-A
DECK #1



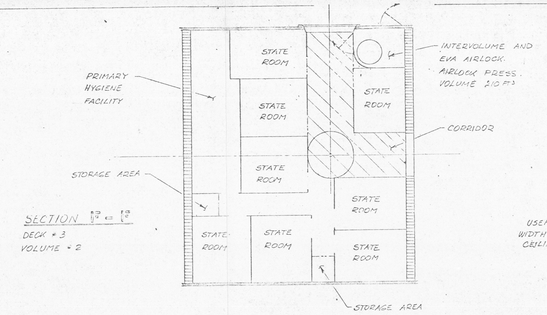
TOTAL STORAGE VOLUME: 940 FT³
USEFUL FLOOR AREA: 2,265 FT²
TOTAL FLOOR AREA: 2,460 FT²
TOTAL OUTER AREA: 5,280 FT²
TOTAL PRESS. VOLUME: 85,168 FT³

2-23, 2-24

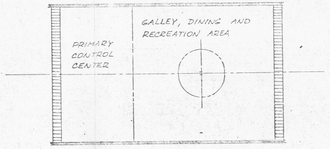
SD 71-207

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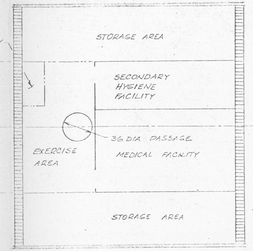
SCALE 1/40	DR. [illegible] DATE 11/22/71	NAR DIVISION: NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
	CHK.		
33 FT DIA, TRANSVERSE FLOOR - 6 MAN OLS MODULE, OLS STUDY			2282-9



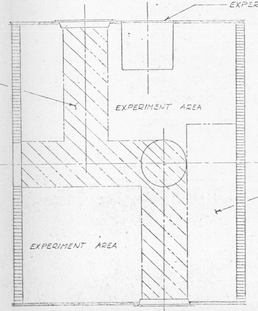
SECTION B-B
DECK = 3
VOLUME = 2



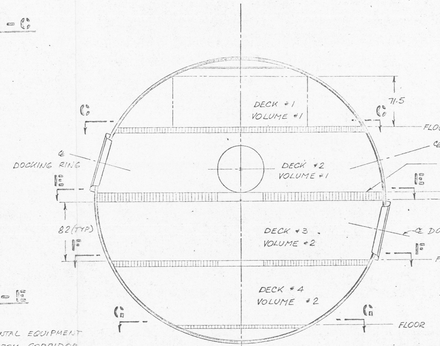
SECTION C-C
DECK = 4
VOLUME = 2



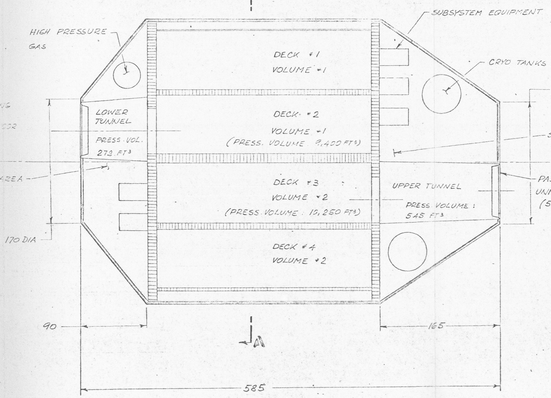
SECTION G-G
DECK = 1
VOLUME = 1



SECTION E-E
DECK = 2
VOLUME = 1
NOTE: EXPERIMENTAL EQUIPMENT TO BE MOVED FROM CORRIDOR AREA DURING CARGO TRANSFER



SECTION A-A



TOTAL STORAGE VOLUME: 870 FT³
(550 FT³ PRESS. VOL, 320 FT³ NON-PRESS. VOLUME)
USEFUL FLOOR AREA: 2,214 FT²
TOTAL FLOOR AREA: 2,331 FT²
TOTAL OUTER AREA: 5,180 FT²
TOTAL PRESS. VOLUME: 20,467 FT³

2-25, 2-26

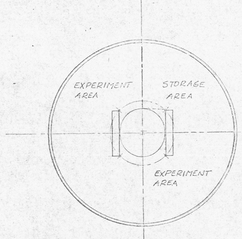
SD 71-207

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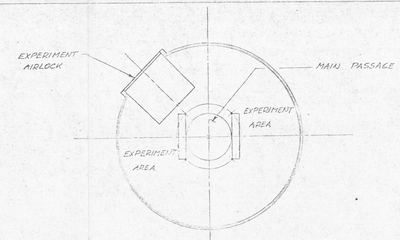
SCALE 1/40	BY	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION SEAL BEACH/NEW ABILITIES, DOWNEY, CALIFORNIA
	DATE	
	MODEL	

33 FT. DIA. LONGITUDINAL FLOOR 9 MAN OLS MODULE, OLS STUDY

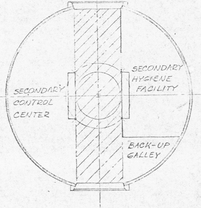
2282-10



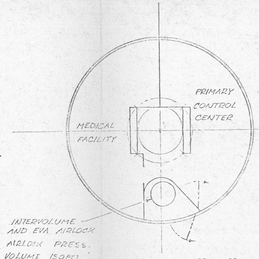
SECTION M-M
DECK = 7
VOLUME = 2



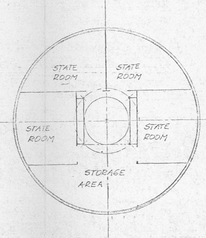
SECTION F-F
DECK = 5
VOLUME = 2



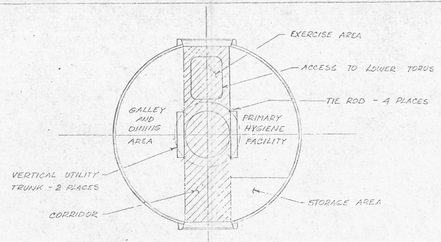
SECTION G-G
DECK = 6
VOLUME = 2



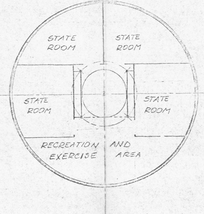
SECTION D-D
DECK = 3
VOLUME = 1



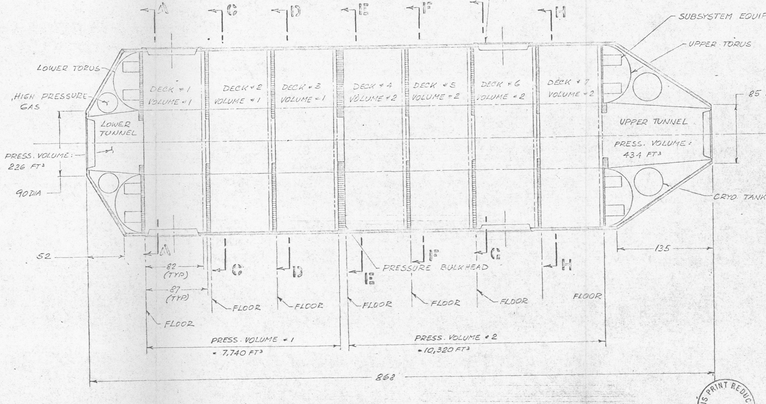
SECTION E-E
DECK = 4
VOLUME = 2



SECTION A-A
DECK = 1
VOLUME = 1



SECTION C-C
DECK = 2
VOLUME = 1



TOTAL STORAGE VOLUME: 270 FT³
(150 FT³ PRESS VOL., 120 NON-PRESS VOL.)
USEFUL FLOOR AREA: 2,220 FT²
TOTAL FLOOR AREA: 2,490 FT²
TOTAL OUTER AREA: 5,020 FT²
TOTAL PRESS. VOLUME: 18,720 FT³

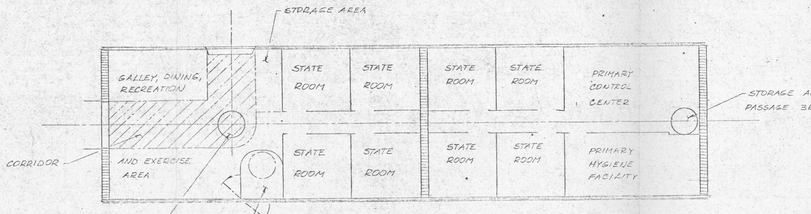
2-27, 2-28 SD 71-207

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Date 1/66	Dr.	MISC. DIVISION	
	Ck'd	NORTH AMERICAN ROCKWELL CORPORATION	
	Appd.	SEAL BEACH, CALIFORNIA	

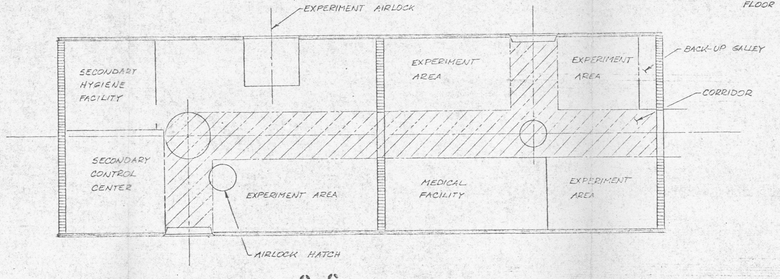
22 FT DIA. TRANSVERSE FLOOR - 9 MAN OLS MODULE. OLS STUDY.

2282-11



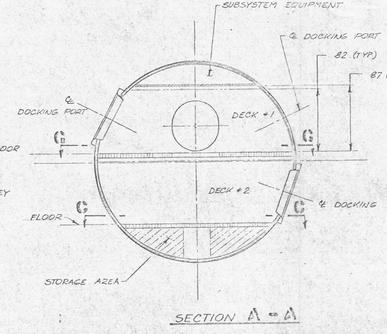
SECTION C - C
DECK #2
ROTATED 90°

NOTE: DINING AND RECREATION EQUIPMENT TO BE MOVED FROM CORRIDOR DURING CARGO TRANSFER

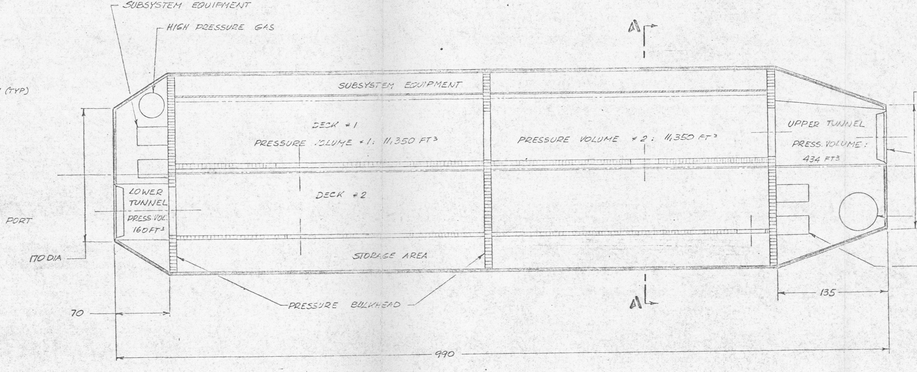


SECTION G - G
DECK #1
ROTATED 90°

NOTE: EXPERIMENTAL EQUIPMENT TO BE MOVED FROM CORRIDOR AREA DURING CARGO TRANSFER



SECTION A - A



TOTAL CARGO VOLUME: 1,200 FT³
USEFUL FLOOR AREA: 2,282 FT²
TOTAL FLOOR AREA: 2,340 FT²
TOTAL OUTER AREA: 5,790 FT²
TOTAL PRESS VOLUME: 23,690 FT³

2-29, 2-30 SD 71-207

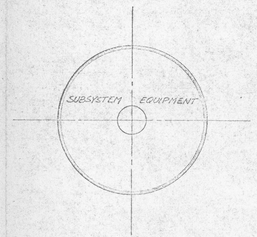
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SIZE	DR.	SPACE DIVISION
40	DATE	NORTH AMERICAN ROCKWELL CORPORATION
	MODEL	12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA

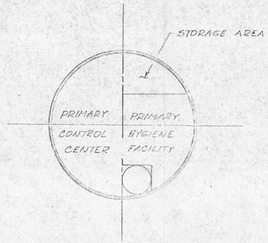
22 FT DIA. LONGITUDINAL FLOOR - 8 MAN. OLS MODULE, OLS STUDY

2282-12A

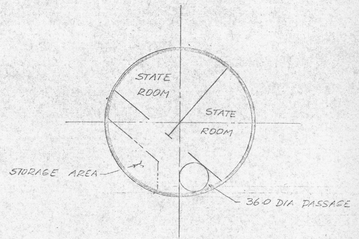
Space Division
North American Rockwell



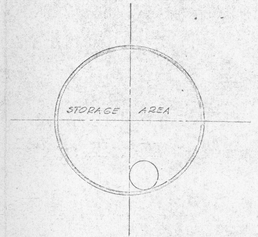
SECTION A-A
DECK #1
MODULE #2



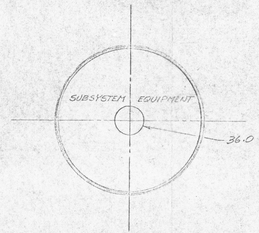
SECTION C-C
DECK #5
MODULE #1



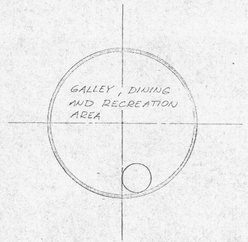
SECTION E-E
DECK #3
MODULE #1



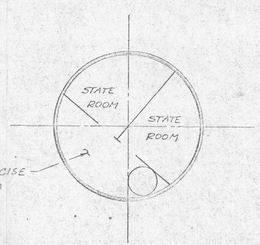
SECTION G-G
DECK #2
MODULE #1



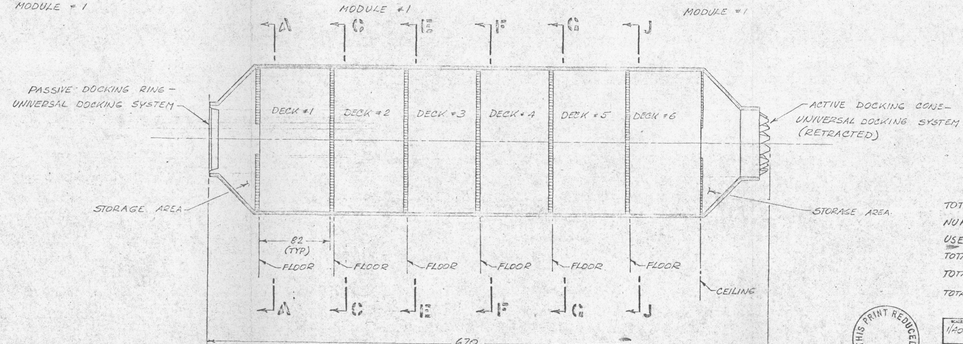
SECTION A-A
DECK #1
MODULE #1



SECTION J-J
DECK #6
MODULE #1



SECTION F-F
DECK #4
MODULE #1



2-31, 2-32 SD 71-207

TOTAL STORAGE VOLUME: 1,800 FT³
NUMBER OF MODULES: 3
USEFUL FLOOR AREA: 2,700 FT²
TOTAL FLOOR AREA: 3,882 FT²
TOTAL OUTER AREA: 7,398 FT²
TOTAL PRESS VOLUME: 21,930 FT³
(7,310 FT³ PER MODULE)

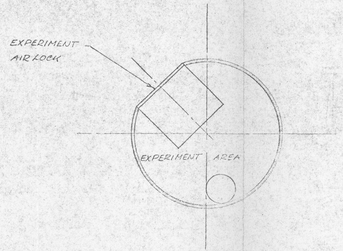
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SCALE 1/20	BY	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION SEAL BEACH AND DOWNEY, CALIFORNIA	
	DATE		
	APPD		

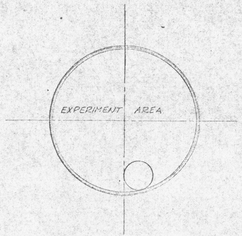
15 FT DIA, TRANSVERSE FLOOR - 3 MAN OLS MODULE. OLS STUDY.

2282-13A
SHT 1 OF 2

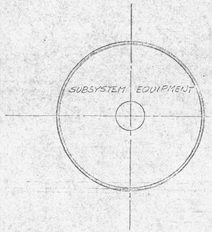
Space Division
North American Rockwell



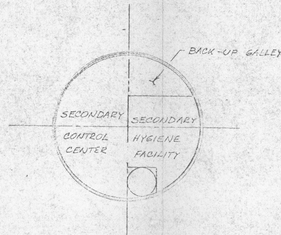
SECTION C-C
DECK = 5
MODULE = 3



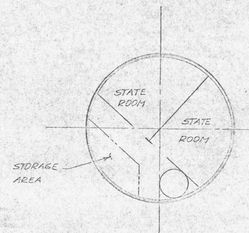
SECTION E-E
DECK = 3
MODULE = 3



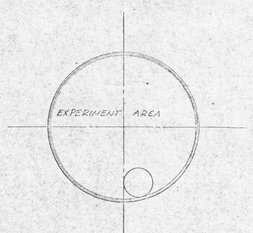
SECTION A-A
DECK = 1
MODULE = 3



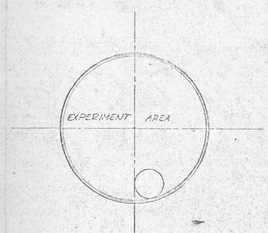
SECTION G-G
DECK = 5
MODULE = 2



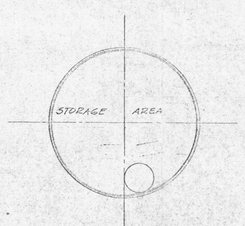
SECTION E-E
DECK = 3
MODULE = 2



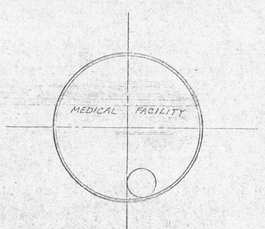
SECTION J-J
DECK = 6
MODULE = 3



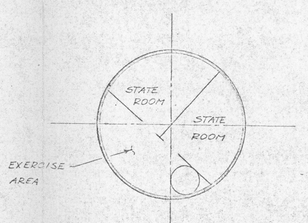
SECTION F-F
DECK = 4
MODULE = 3



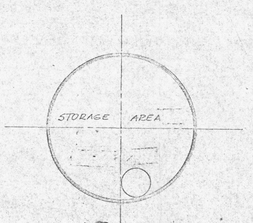
SECTION G-G
DECK = 2
MODULE = 3



SECTION J-J
DECK = 6
MODULE = 2



SECTION F-F
DECK = 4
MODULE = 2



SECTION G-G
DECK = 2
MODULE = 2

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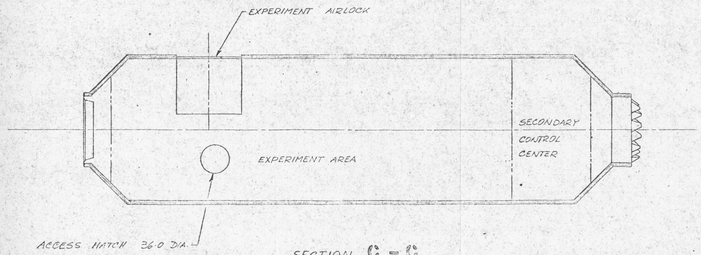
2-33, 2-34 SD 71-207

SCALE	BY	APPROVED	BASIC DIVISION NORTH AMERICAN ROCKWELL CORPORATION
1/40			
MODEL			

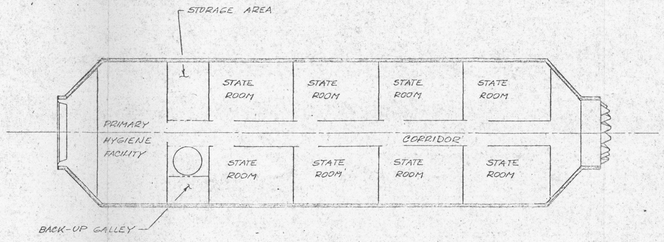
15 FT DIA. TRANSVERSE FLOOR - 8 MAN OLS MODULE
OLS STUDY

2282-13A
SHT 2 OF 2

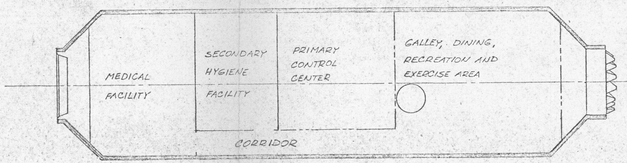
Space Division
North American Rockwell



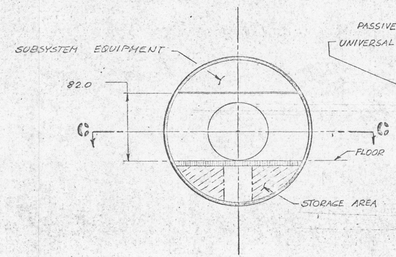
SECTION C - C
MODULE # 2
ROTATED 90°



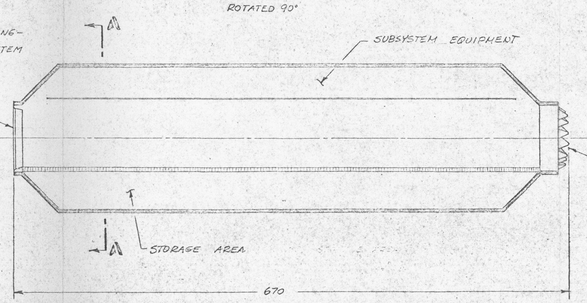
SECTION C - C
MODULE # 1
ROTATED 90°



SECTION C - C
MODULE # 3
ROTATED 90°



SECTION A - A



ACTIVE DOCKING CONE - UNIVERSAL DOCKING SYSTEM (RETRACTED)

2-35, 2-36

SD 71-207

TOTAL STORAGE VOLUME: 2,400 FT³
USEFUL FLOOR AREA: 1,725 FT²
TOTAL FLOOR AREA: 1,755 FT²
TOTAL OUTER AREA: 7,398 FT²
TOTAL PRESS VOLUME: 21,930 FT³
(7,310 FT³ PER MODULE)

NUMBER OF MODULES: 3

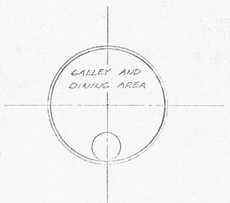
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SCALE	BY	SPACE DIVISION
1/40	DATE	NORTH AMERICAN ROCKWELL CORPORATION
	MODEL	12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA

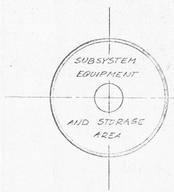
15 FT DIA LONGITUDINAL FLOOR - 3 MAN OLS MODULE, OLS STUDY

2282-14

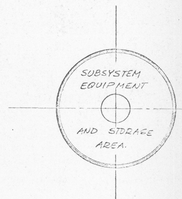
Space Division
North American Rockwell



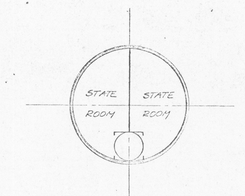
SECTION C-C
DECK # 2
MODULE # 4



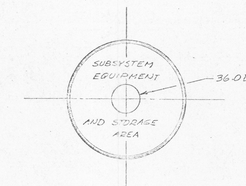
SECTION A-A
DECK # 1
MODULE # 4



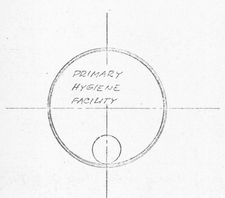
SECTION A-A
DECK # 1
MODULE # 3



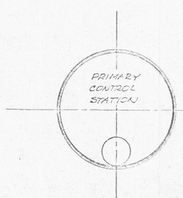
SECTION C-C
DECK # 2
MODULE # 1 AND # 2



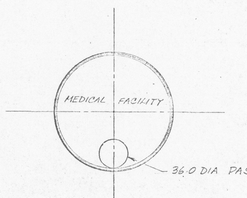
SECTION A-A
DECK # 1
MODULE # 1 AND # 2



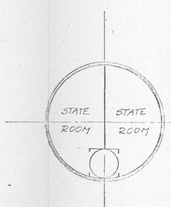
SECTION E-E
DECK # 3
MODULE # 4



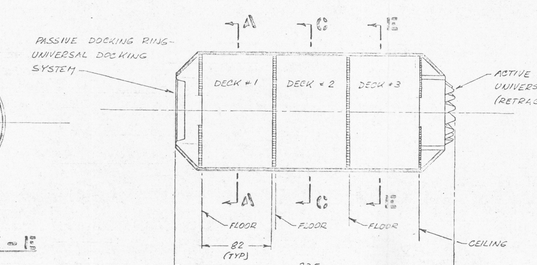
SECTION E-E
DECK # 3
MODULE # 3



SECTION C-C
DECK # 2
MODULE # 3



SECTION E-E
DECK # 3
MODULE # 1 AND # 2



2-37, 2-38 SD 71-207

TOTAL STORAGE VOLUME: 2,300 FT³
USEFUL FLOOR AREA: 2,496 FT²
TOTAL FLOOR AREA: 2,766 FT²
TOTAL OUTER AREA: 9,396 FT²
TOTAL PRESS. VOLUME: 21231 FT³
(2359 FT³ PER MODULE)
NUMBER OF MODULES: 9

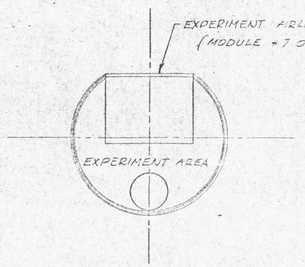
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SCALE	DR.	SPACE DIVISION
1/40	DATE 11-1-71	NORTH AMERICAN ROCKWELL CORPORATION
	MODEL	DOWNEY, CALIFORNIA

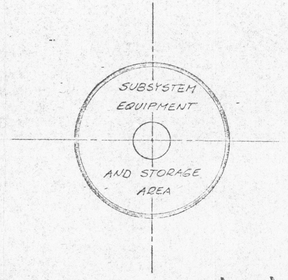
12 FT DIA, TRANSVERSE FLOOR-
6 MAN CLS MODULE,
CLS STUDY

2282-15A
SHT 1 OF 2

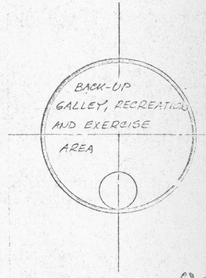
Space Division
North American Rockwell



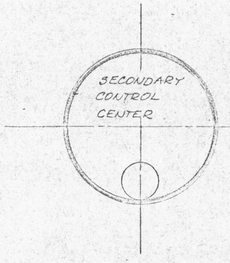
SECTION E-E
DECK #3
MODULE #7, #8 AND #9



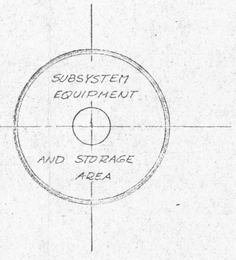
SECTION A-A
DECK #1
MODULE #7, #8 AND #9



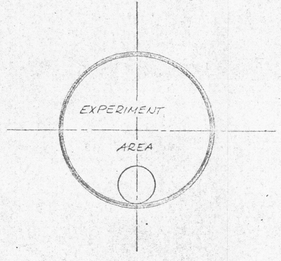
SECTION C-C
DECK #2
MODULE #6



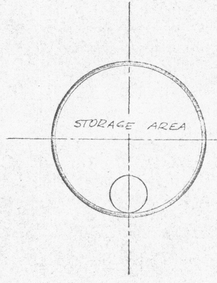
SECTION E-E
DECK #3
MODULE #5



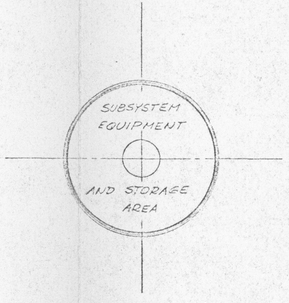
SECTION A-A
DECK #1
MODULE #5



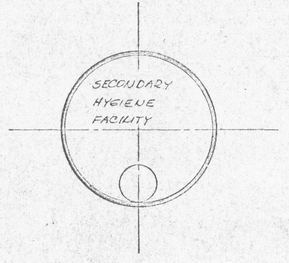
SECTION C-C
DECK #2
MODULE #7, #8 AND #9



SECTION E-E
DECK #3
MODULE #6



SECTION A-A
DECK #1
MODULE #6



SECTION C-C
DECK #2
MODULE #5

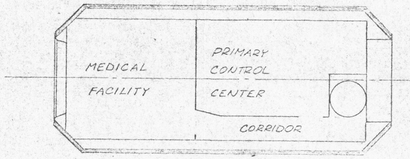
2-39, 2-40

SD 71-207

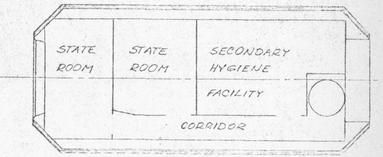
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SCALE 1/40	DR. [illegible]	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
	DATE 12-21-71		
	MODEL		
12FT DIA. TRANSVERSE FLOOR-8 MAN OLS MODULE, OLS STUDY			2282-15A SHT 2 OF 2

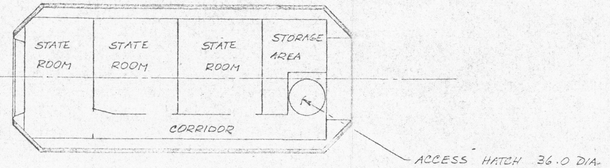
Space Division
North American Rockwell



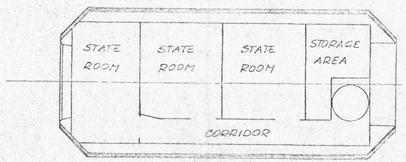
SECTION G - G
MODULE #4



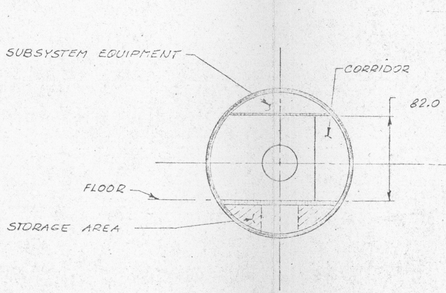
SECTION G - G
MODULE #2



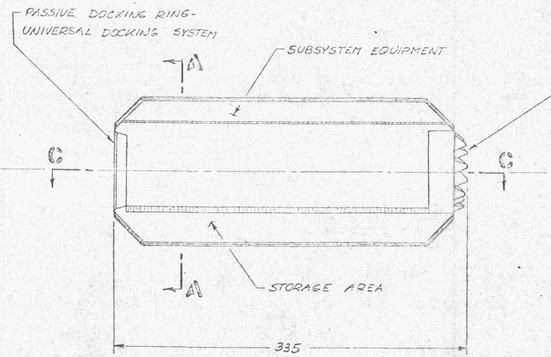
SECTION G - G
MODULE #1



SECTION G - G
MODULE #3



SECTION A - A



2-41, 2-42 SD 71-207

TOTAL STORAGE AREA:	2,300 FT³
USEFUL FLOOR AREA:	1,899 FT²
TOTAL FLOOR AREA:	1,989 FT²
TOTAL OUTER AREA:	9,396 FT²
TOTAL PRESS. VOLUME:	23,040 FT³
	(2,560 FT³ PER MODULE)

NUMBER OF MODULES: 9

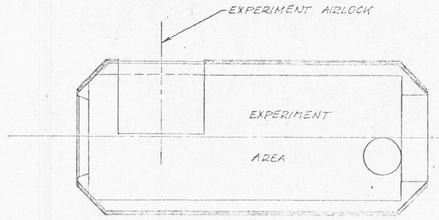
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SCALE 1/40	DR	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
	CH		
	MODEL		

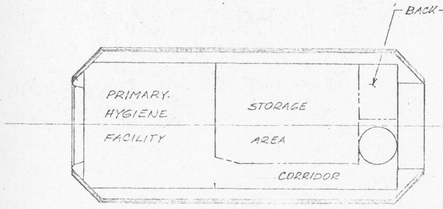
12 FT DIA, LONGITUDINAL FLOOR-
8 MAN OLS MODULE,
OLS STUDY

2282-16
SHT 1 OF 2

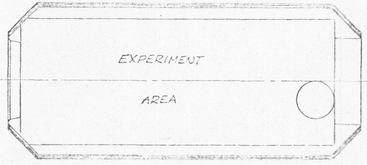
Space Division
North American Rockwell



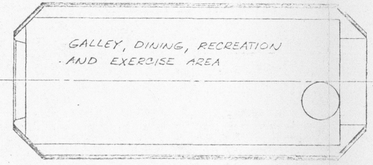
SECTION ⓒ - ⓒ
MODULE #8



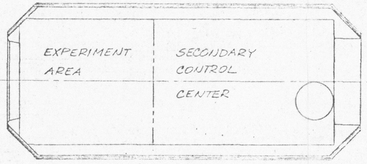
SECTION ⓒ - ⓒ
MODULE #6



SECTION ⓒ - ⓒ
MODULE #9



SECTION ⓒ - ⓒ
MODULE #7



SECTION ⓒ - ⓒ
MODULE #5

2-43, 2-44 SD 71-207

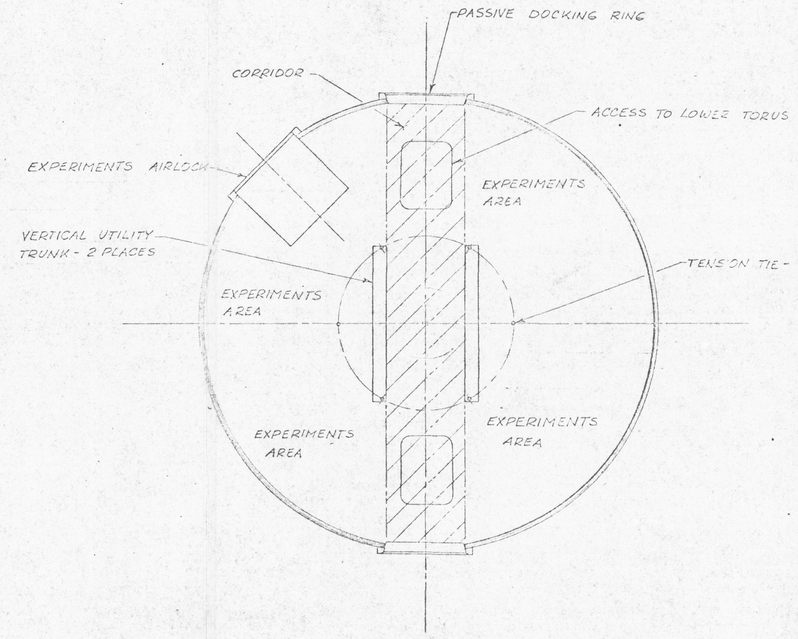
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SCALE 1/40	DR.	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA
	DATE	
	MODEL	

12 FT DIA. LONGITUDINAL FLOOR-8 MAN OLS MODULE, OLS STUDY

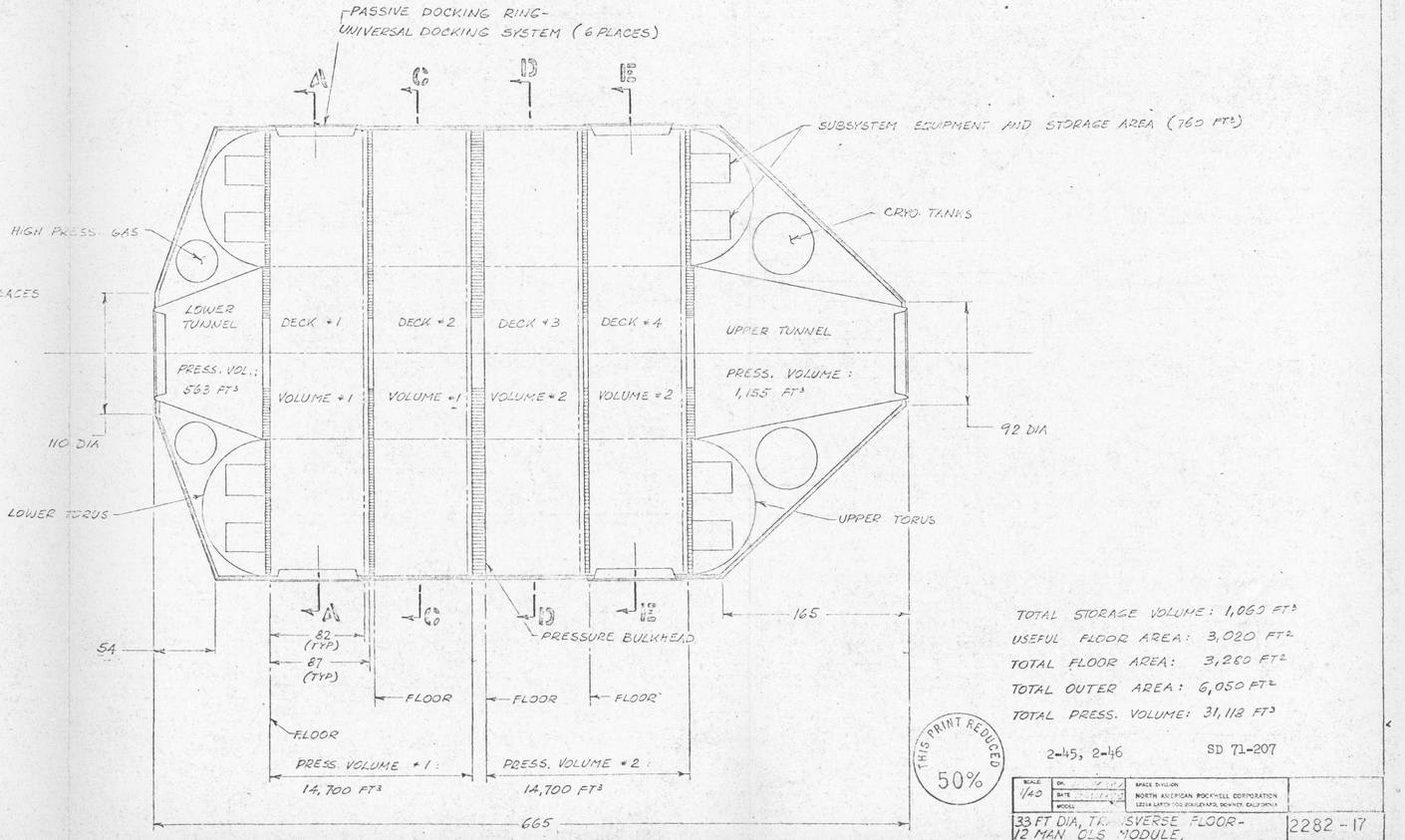
2282-16
SHT 2 OF 2

Space Division
North American Rockwell



SECTION A - A

DECK #1
VOLUME #1



TOTAL STORAGE VOLUME: 1,060 FT³
USEFUL FLOOR AREA: 3,020 FT²
TOTAL FLOOR AREA: 3,280 FT²
TOTAL OUTER AREA: 6,050 FT²
TOTAL PRESS. VOLUME: 31,118 FT³

2-45, 2-46

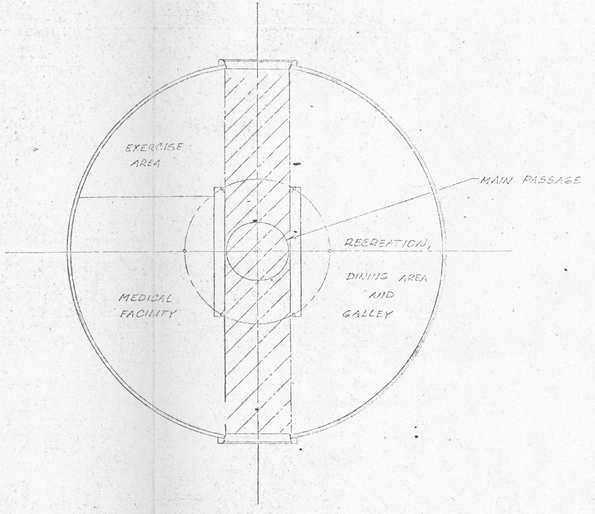
SD 71-207

THIS PRINT REDUCED 50%

SCALE 1/40	DR.	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA
	DATE	
	MODEL	

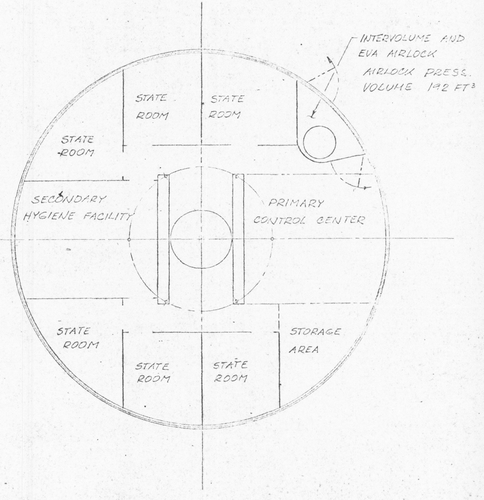
33 FT DIA, TRANSVERSE FLOOR - 2 MAN OLS MODULE, OLS STUDY

2282 - 17
SHT 1 OF 2



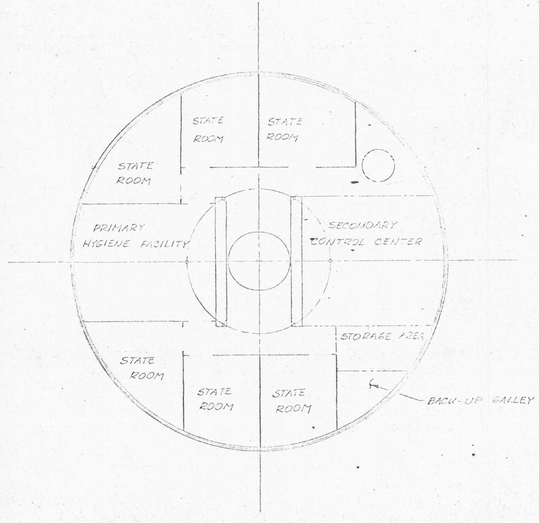
SECTION E - E

DECK # 4
VOLUME # 2



SECTION D - D

DECK # 3
VOLUME # 2



SECTION C - C

DECK # 2
VOLUME # 1

2-47, 2-48 SD 71-207

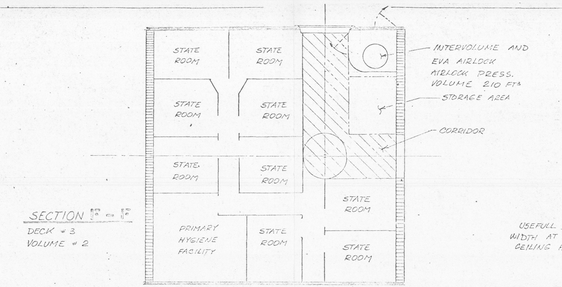
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SCALE 1/10	DR.	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA
	DATE 11/12/71	
	MODEL	

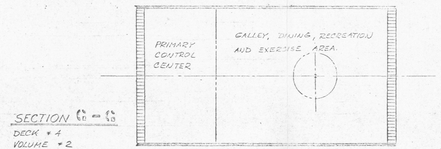
33 FT DIA. TRANSVERSE FLOOR-12 MAN OLS MODULE, OLS STUDY

2282-17
SHT 2 OF 2

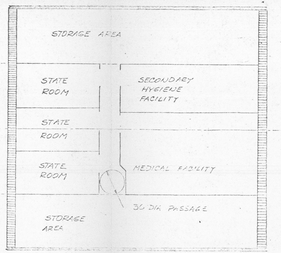
Space Division
North American Rockwell



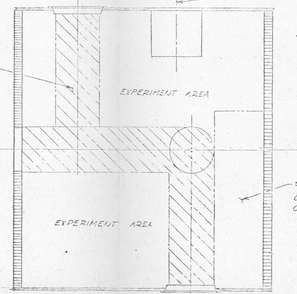
SECTION E - E
DECK = 3
VOLUME = 2



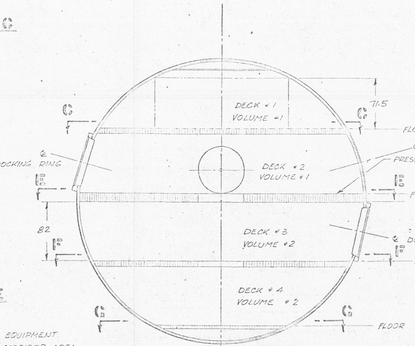
SECTION G - G
DECK = 4
VOLUME = 2



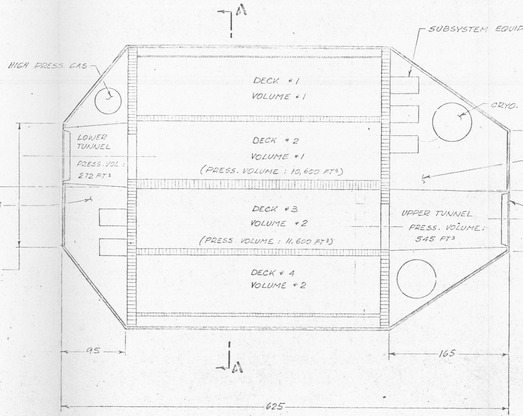
SECTION C - C
DECK = 1
VOLUME = 1



SECTION D - D
DECK = 2
VOLUME = 1
NOTE: EXPERIMENTAL EQUIPMENT TO BE MOVED FROM CORRIDOR AREA DURING CARGO TRANSFER



SECTION A - A



TOTAL STORAGE VOLUME: 1,000 FT³
(650 FT³ PRESS. VOL., 350 FT³ NON-PRESS. VOL.)
USEFUL FLOOR AREA: 2,488 FT²
TOTAL FLOOR AREA: 2,805 FT²
TOTAL OUTER AREA: 5,830 FT²
TOTAL PRESS. VOLUME: 23,017 FT³

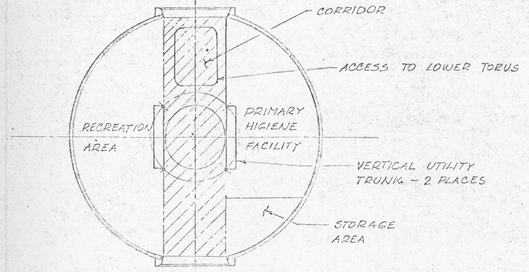
2-49, 2-50 SD 71-207

THIS PRINT REQUIRED 50%

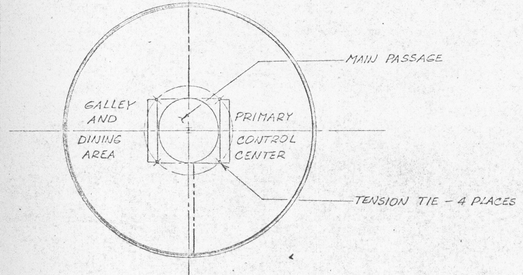
ISSUE	BY	DATE	SPACE DIVISION
1/10			NORTH AMERICAN ROCKWELL CORPORATION
			DOWNEY, CALIFORNIA

33 FT DIA LONGITUDINAL FLOOR-12 MAN OLS MODULE. OLS STUDY

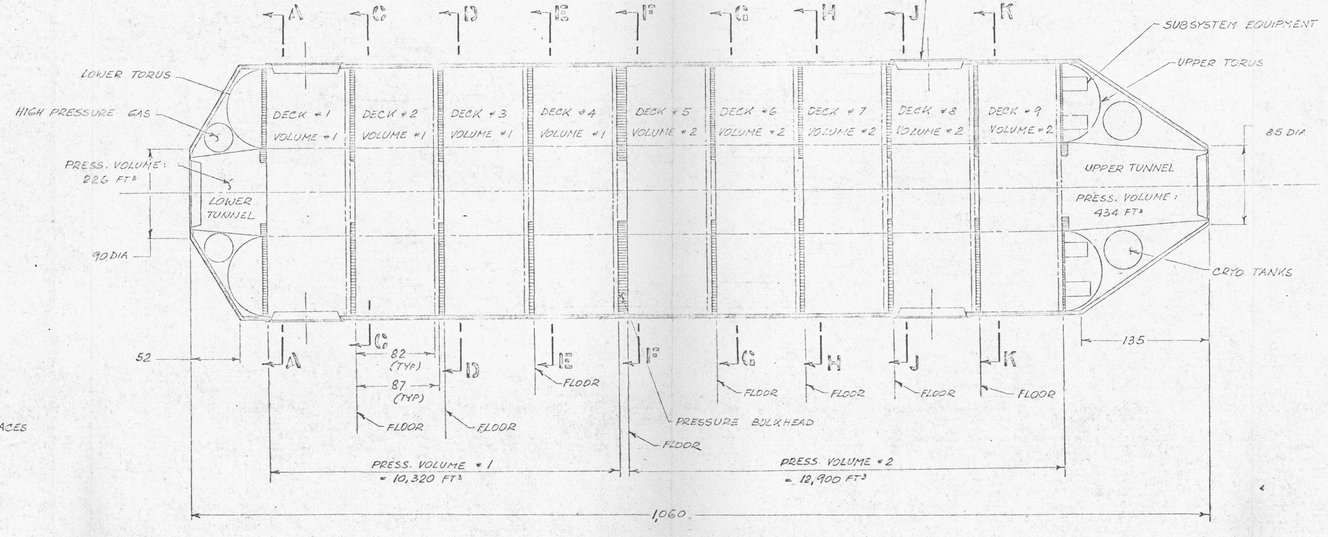
2282-18



SECTION A-A
DECK #1
VOLUME #1



SECTION C-C
DECK #2
VOLUME #1



TOTAL STORAGE VOLUME: 1,050 FT³
USEFUL FLOOR AREA: 2,862 FT²
TOTAL FLOOR AREA: 3,204 FT²
TOTAL OUTER AREA: 6,060 FT²
TOTAL PRESS. VOLUME: 23,880 FT³

2-51, 2-52 SD 71-207

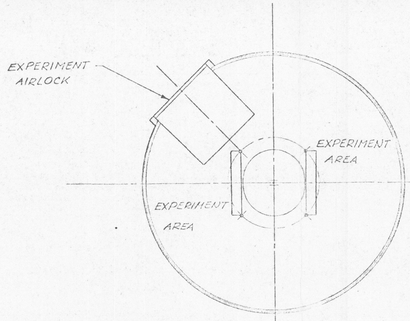
THIS PRINT REDUCED 50%

SCALE: 1/40	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA
DATE:	
MODEL:	

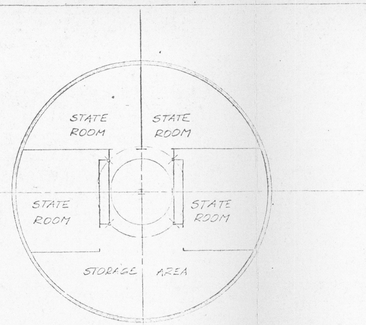
22 FT DIA, TRANSVERSE FLOOR - 12 MAN OLS MODULE, OLS STUDY

2282-19
SHT 1 OF 2

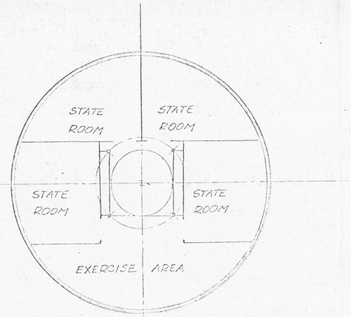
Space Division
North American Rockwell



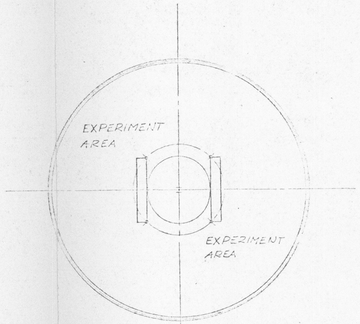
SECTION H-H
DECK # 7
VOLUME # 2



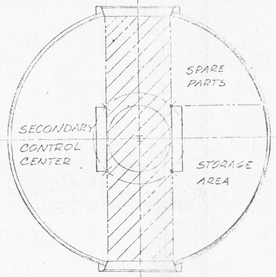
SECTION I-I
DECK # 5
VOLUME # 2



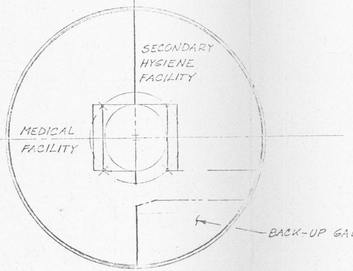
SECTION D-D
DECK # 3
VOLUME # 1



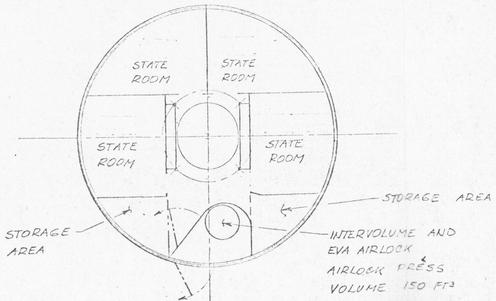
SECTION K-K
DECK # 9
VOLUME # 2



SECTION J-J
DECK # 8
VOLUME # 2



SECTION G-G
DECK # 6
VOLUME # 2



SECTION E-E
DECK # 4
VOLUME # 1

2-53, 2-54

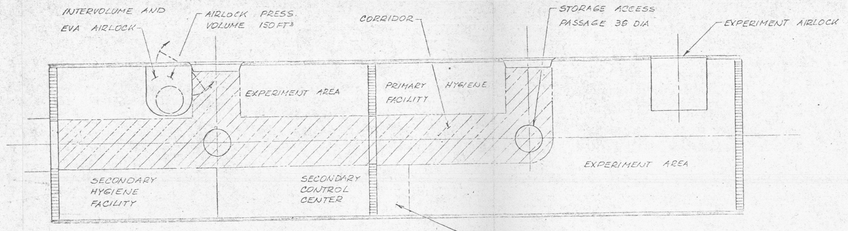
SD 71-207

THIS PRINT REDUCED 50%

SCALE 1/40	DR.	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA
	DATE	
	MODEL	

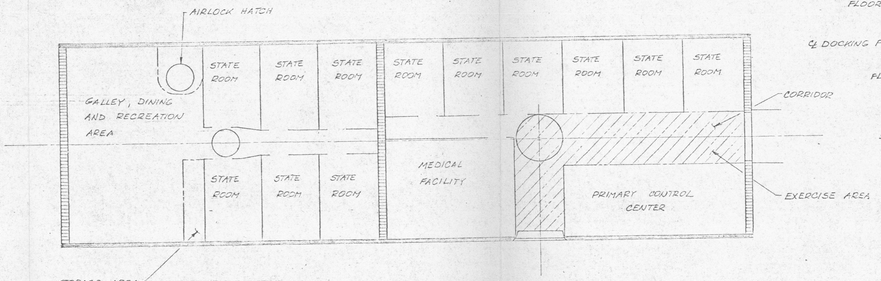
22 FT DIA. TRANSVERSE FLOOR-12 MAN OLS MODULE OLS STUDY

2282-19
SHT 2 OF 2



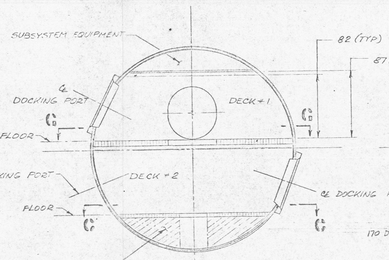
SECTION G-G

DECK = 2
ROTATED 90°
NOTE: EXPERIMENTAL EQUIPMENT TO BE MOVED FROM CORRIDOR AREA DURING CARGO TRANSFER

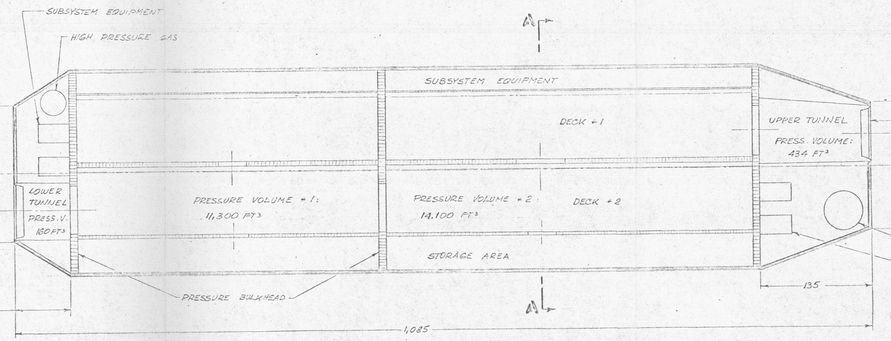


SECTION G-G

DECK = 1
ROTATED 90°



SECTION A-A



USEFUL FLOOR AREA: 2,577 FT²
TOTAL FLOOR AREA: 2,635 FT²
TOTAL OUTER AREA: 6,340 FT²
TOTAL PRESS. VOLUME: 25,490 FT³
TOTAL CARGO VOLUME: 1,350 FT³

2-55, 2-56 SD 71-207

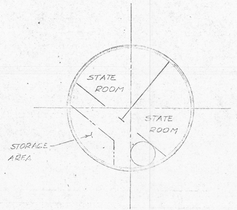
THIS PRINT REQUIRED 50%

BUILD	DR.	SPACE DIVISION
1/66	DATE	NORTH AMERICAN ROCKWELL CORPORATION
	MODEL	DOWNEY, CALIFORNIA

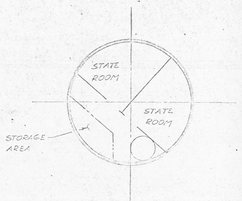
22 FT DIA. LONGITUDINAL FLOOR - 12 MAN OES MODULE, OES STUDY

2282-20A

Space Division
North American Rockwell



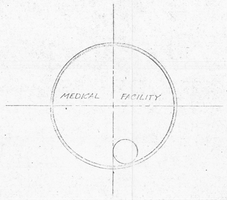
SECTION G - G
DECK = 5
MODULE = 2



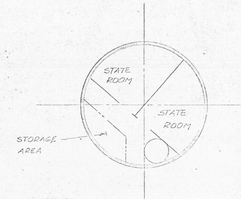
SECTION H - H
DECK = 3
MODULE = 2



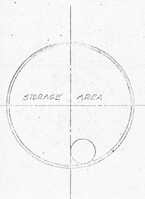
SECTION A - A
DECK = 1
MODULE = 2



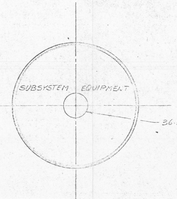
SECTION C - C
DECK = 5
MODULE = 1



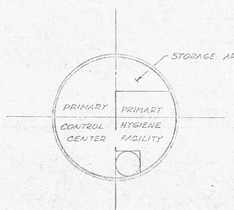
SECTION E - E
DECK = 3
MODULE = 1



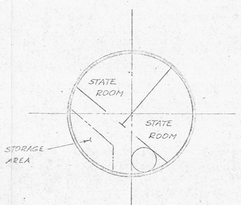
SECTION G - G
DECK = 2
MODULE = 1



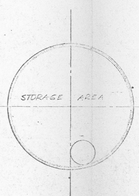
SECTION A - A
DECK = 1
MODULE = 1



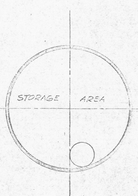
SECTION J - J
DECK = 6
MODULE = 2



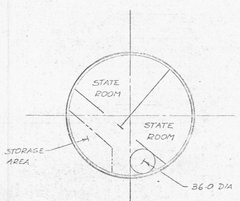
SECTION F - F
DECK = 4
MODULE = 2



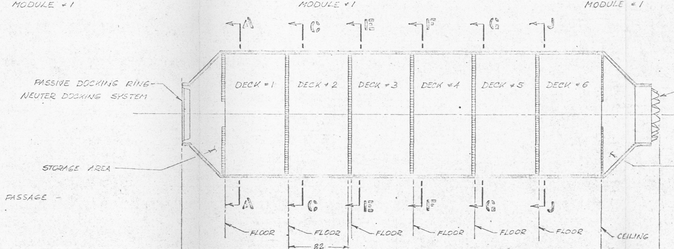
SECTION G - G
DECK = 2
MODULE = 2



SECTION J - J
DECK = 6
MODULE = 1



SECTION F - F
DECK = 4
MODULE = 1



2-57, 2-58 SD 71-207

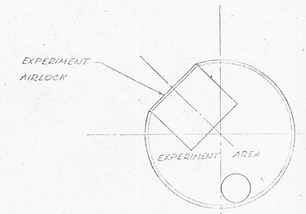
TOTAL STORAGE VOLUME: 2,000 FT³
NUMBER OF MODULES: 4
USEFUL FLOOR AREA: 3,920 FT²
TOTAL FLOOR AREA: 3840 FT²
TOTAL OUTER AREA: 9764 FT²
TOTAL PRESS. VOLUME: 29,240 FT³ (7,310 FT³ PER MODULE)

THIS PRINT REVIEWED 50%

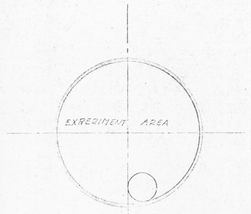
SCALE	BY	SPACE DIVISION
NO	DATE	NORTH AMERICAN ROCKWELL CORPORATION
	APPD	DOWNEY LABORATORY RESEARCH, DOWNEY, CALIFORNIA

15 FT DIA. TRANSVERSE FLOOR-12 MAN OLS MODULE, OLS STUDY

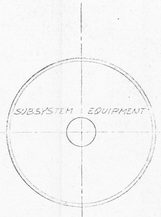
22882-21A
SHT 1 OF 2



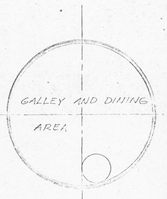
SECTION G - G
DECK = 5
MODULE = 4



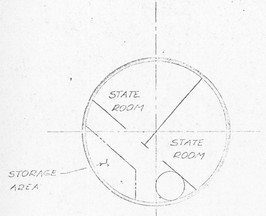
SECTION E - E
DECK = 3
MODULE = 4



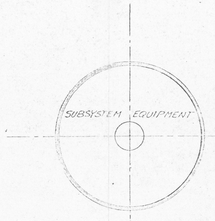
SECTION A - A
DECK = 1
MODULE = 4



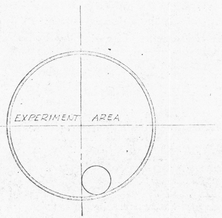
SECTION G - G
DECK = 5
MODULE = 3



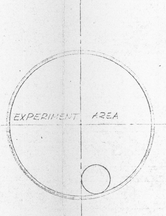
SECTION E - E
DECK = 3
MODULE = 3



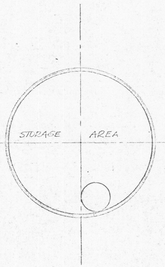
SECTION A - A
DECK = 1
MODULE = 3



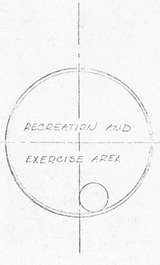
SECTION J - J
DECK = 6
MODULE = 4



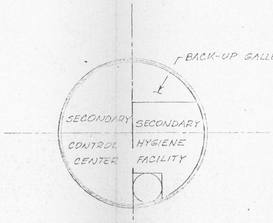
SECTION F - F
DECK = 4
MODULE = 4



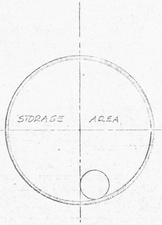
SECTION C - C
DECK = 2
MODULE = 4



SECTION J - J
DECK = 6
MODULE = 3



SECTION F - F
DECK = 4
MODULE = 3



SECTION C - C
DECK = 2
MODULE = 3

THIS PRINT REDUCED 50%

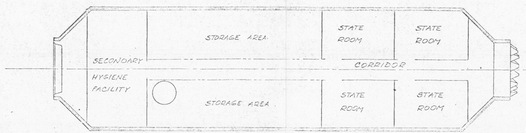
2-59, 2-60 SD 71-207

SCALE 1/40	DR	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION SEAL BEACH, CALIFORNIA, DOWNEY, CALIFORNIA	
	DATE		
	MODEL		

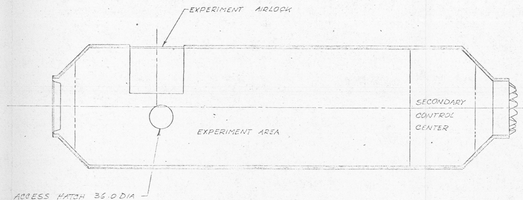
15 FT DIA. TRANSVERSE FLOOR - 12 MAN OLS MODULE. OLS STUDY

2282-21A
SHT 2 OF 2

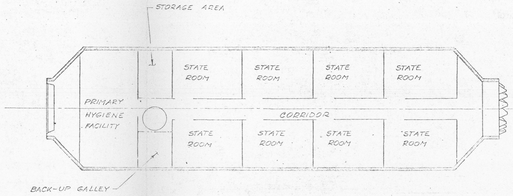
Space Division
North American Rockwell



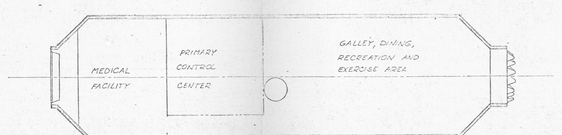
SECTION C ⇔ C
MODULE × 1
ROTATED 90°



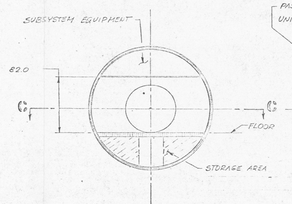
SECTION C ⇔ C
MODULE × 3
ROTATED 90°



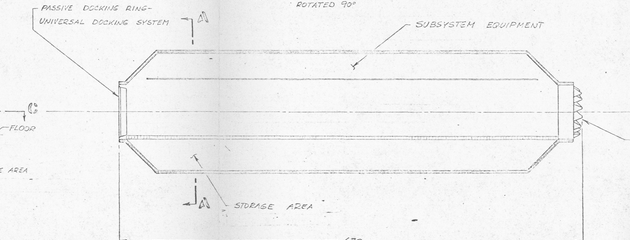
SECTION C ⇔ C
MODULE × 1
ROTATED 90°



SECTION C ⇔ C
MODULE × 2
ROTATED 90°



SECTION A ⇔ A



2-61, 2-62 ED 71-207

TOTAL STORAGE VOLUME: 3,200 FT³
USEFUL FLOOR AREA: 2,220 FT²
TOTAL FLOOR AREA: 2,320 FT²
TOTAL OUTER AREA: 9,864 FT²
TOTAL PRESS. VOLUME: 29,240 FT³
(7,310 FT³ PER MODULE)

NUMBER OF MODULES: 4

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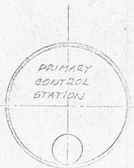
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	CH.		
	APPD.		

15 FT DIA. LONGITUDINAL FLOOR
12 MAN. DOS MODULE.
DOS STUDY

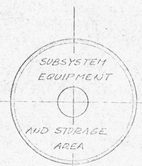
2282-22



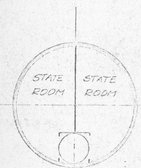
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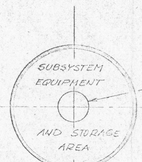
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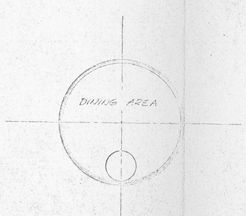
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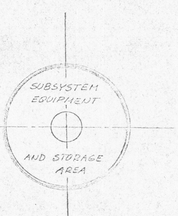
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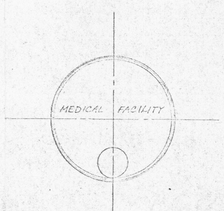
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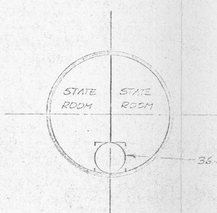
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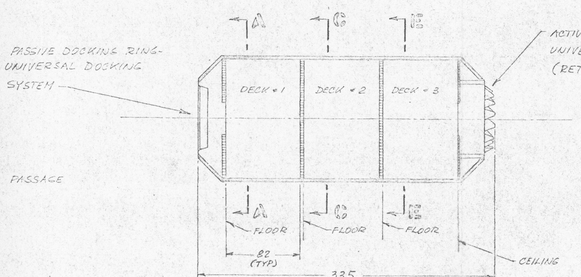
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MODULE = 5



SECTION
DECK = 2
MODULE = 4



SECTION
DECK = 3
MODULE = 1, = 2 AND = 3



2-63, 2-64 SD 71-207

TOTAL STORAGE VOLUME:	2,000 FT³
USEFUL FLOOR AREA:	2,775 FT²
TOTAL FLOOR AREA:	3,075 FT²
TOTAL OUTER AREA:	14,960 FT²
TOTAL PRESS VOLUME:	23,590 FT³
	(2,359 FT³ PER MODULE)

NUMBER OF MODULES: 10

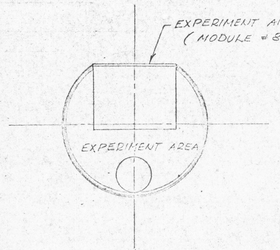
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	DATE	
	APPD.	

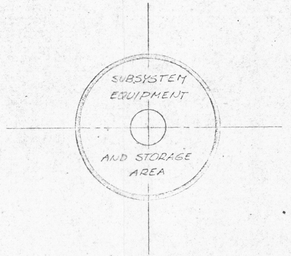
12 FT DIA TRANSVERSE FLOOR-
12 MAN OLS MODULE
OLS STUDY

2282-23A
SHT 1 OF 2

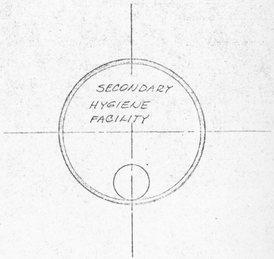
Space Division
North American Rockwell



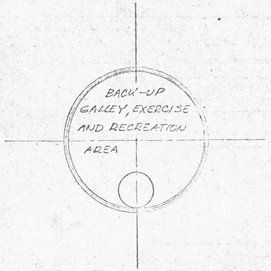
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DECK #3
MODULE #8, #9 AND #10



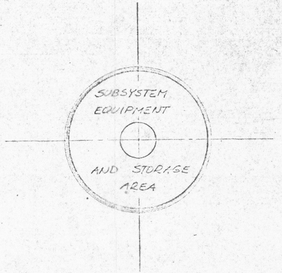
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DECK #1
MODULE #8, #9 AND #10



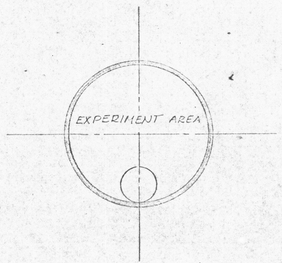
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DECK #2
MODULE #7



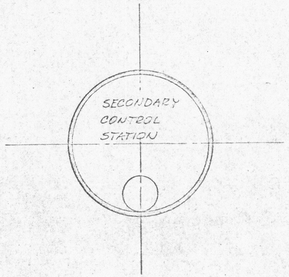
SECTION B - B
DECK #3
MODULE #6



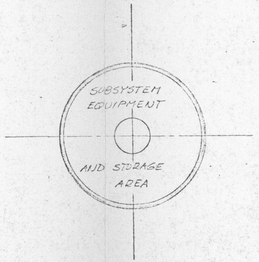
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DECK #1
MODULE #6



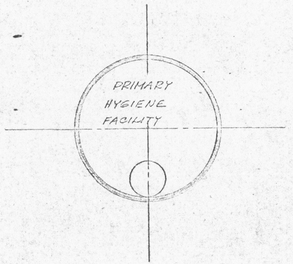
SECTION C - C
DECK #2
MODULE #8, #9 AND #10



SECTION B - B
DECK #3
MODULE #7



SECTION A - A
DECK #1
MODULE #7



SECTION C - C
DECK #2
MODULE #6

2-65, 2-66

SD. 71-207

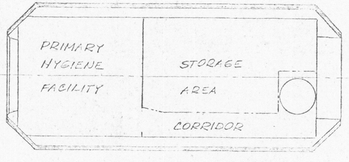
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	DATE 8/31/70	
	MODEL	

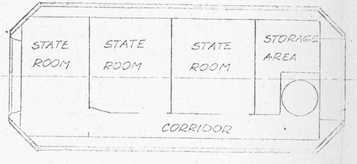
12 FT DIA. TRANSVERSE FLOOR-
12 MAN OLS MODULE,
OLS STUDY

2282-23A
SHT 2 OF 2

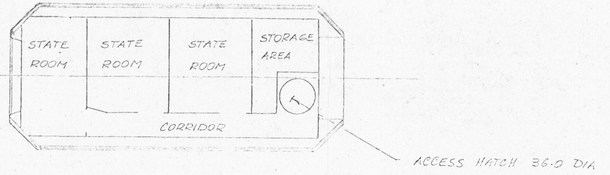
Space Division
North American Rockwell



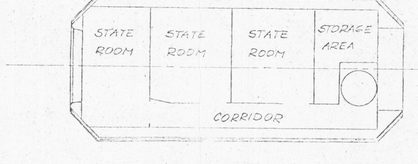
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MODULE #4



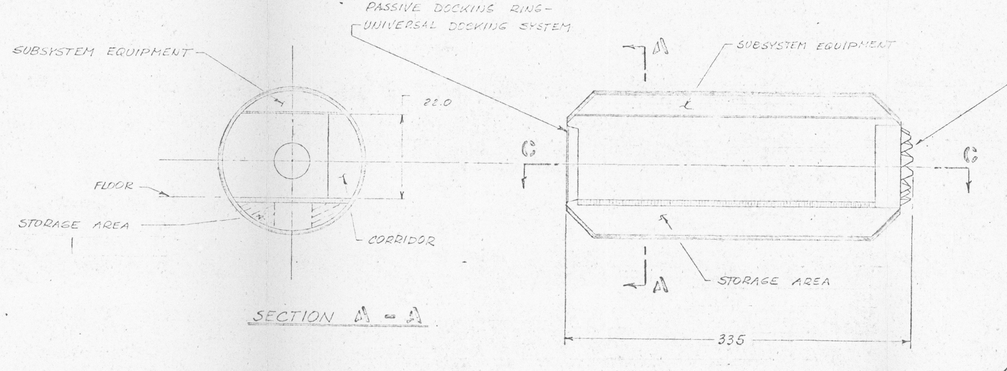
SECTION C6 - C6
MODULE #2



SECTION C6 - C6
MODULE #1



SECTION C6 - C6
MODULE #3



SECTION A - A

2-67, 2-68 SD 71-207

TOTAL STORAGE VOLUME: 3,350 FT³
USEFUL FLOOR AREA: 2,321 FT²
TOTAL FLOOR AREA: 2,431 FT²
TOTAL OUTER AREA: 14,550 FT²
TOTAL PRESS. VOLUME: 28,160 FT³
(2,560 FT³ PER MODULE)

NUMBER OF MODULES: 11

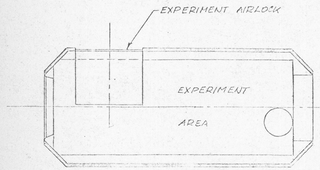
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	DATE	
	APPVD.	

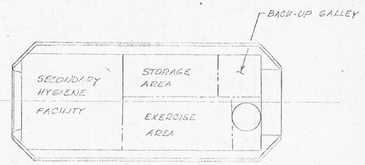
12 FT DIA, LONGITUDINAL FLOOR-
12 MAN OLS MODULE,
OLS STUDY

2282-24
SHT 1 OF 2

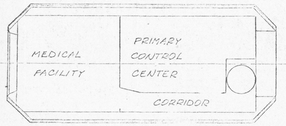
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SECTION C - C
MODULE - 10



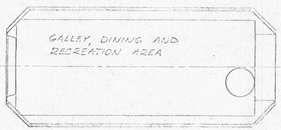
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MODULE - 8



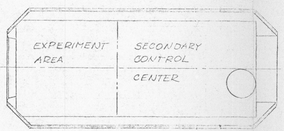
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MODULE - 6



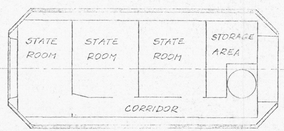
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MODULE - 11



SECTION C - C
MODULE - 9



SECTION C - C
MODULE - 7



SECTION C - C
MODULE - 5

2-69, 2-70 SD 71-207

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SCALE 1/20	NO.	DATE	BY
DATE 6-4-72			
APPD.			

SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION
12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA

10 FT DIA. LONGITUDINAL FLOOR-
12 MAN OLS MODULE,
OLS STUDY

2282-24
SHT 2 OF 2

2.3 CONFIGURATION ANALYSIS

Through use of the 24 general arrangement concept drawings, it was possible to conduct structural weights analyses and calculate the other vehicle module characteristics of interest. Table 2.3-1 summarizes the major physical characteristics of the 24 parametric concepts with the calculated volumes, surface areas, floor areas, available cargo volumes, and structural weights. Two different vehicle weights are tabulated. The column entitled "total core module weight" indicates the complete structural weight of each module, which includes pressure shell, unpressurized structure, docking ports, meteoroid protection, and thermal protection. In the case of the 12- and 15-foot diameter concepts, in which more than one module is clustered to complete the total OLS, the individual module weights are shown in addition to the total assembled weight, which includes the weight of the multiple docking adapter (MDA) or adapters required to join the individual modules, plus the weight of supporting structure required during the delivery from earth orbit to lunar orbit via cislunar shuttles. Shown in Figure 2.3-1 are sketches of OLS concepts made up of 12-foot diameter, 30-foot long modules assembled in a CLS delivery configuration. The 4-man configuration requires six 12-foot diameter modules and one MDA in both the transverse and longitudinal floor versions. The 8-man configuration requires nine 12-foot diameter modules and two MDA's in both the transverse and longitudinal floor versions. The 12-man configuration made up of 12-foot diameter modules with transverse floors requires 10 modules and two MDA's while the longitudinal floor version requires 11 modules and two MDA's. Shown in Figure 2.3-2 are sketches of OLS concepts made up of 15-foot diameter, 60-foot long modules assembled in a CLS delivery configuration. The 4-man configuration requires two 15-foot diameter modules and one MDA in both the transverse and longitudinal floor versions. The 8- and 12-man configurations made up of 15-foot diameter modules require three and four modules respectively regardless of floor version. Both the 8- and 12-man 15-foot diameter modular concepts require one MDA and both require side support to reduce the moment at the junction between the side docked modules and the MDA. The 12-man configuration requires four side support frames while the 8-man configuration requires two. All of the modular configurations depicted in Figures 2.3-1 and 2.3-2 require an interstage support between the CLS and the first MDA in the OLS cluster. A listing of the pressure shell weights also is included for each concept because it will provide useful comparison data related to this item separated from other structural items. As previously stated, the basic objective of the parametric conceptual trades was to obtain data that would delineate the interrelationship between crew size, module size, and floor orientation and could potentially be used to permit selection of optimum design points.

Three characteristics were considered to be key factors that would possibly illustrate significant trends and were calculated and shown in Table 2.3-1. Because all of the vehicles were initially configured to satisfy specific floor area requirements, it seemed reasonable to assume that a measure of the configurational efficiency could be obtained by examining the

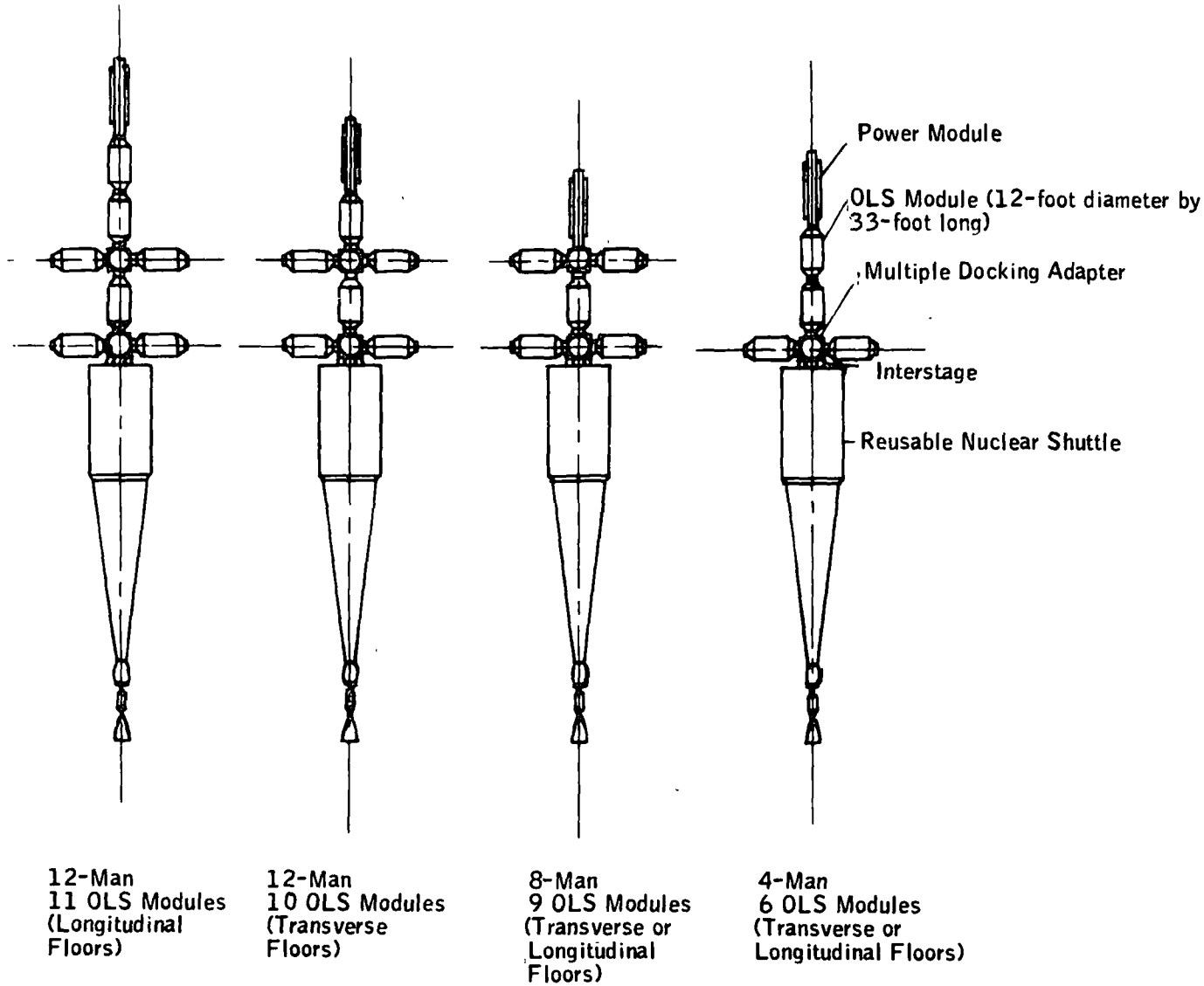


Figure 2.3-1. OLS Modular Assemblages - 12-Foot-Diameter Modules

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North American Rockwell

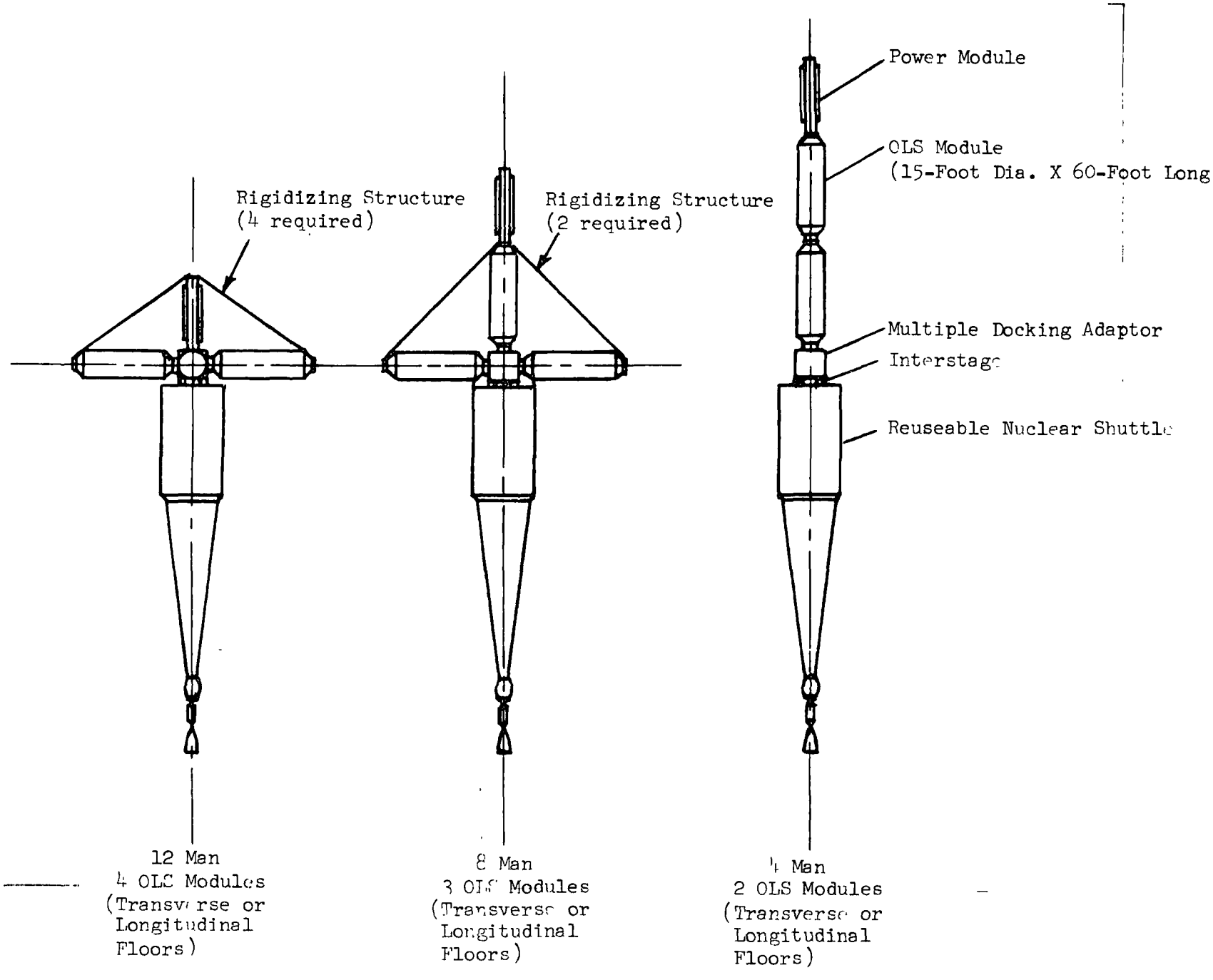


Figure 2.3-2 OLS Modular Assemblages - 15-Foot Diameter Modules

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Table 2.3-1. OLS Module Characteristics Summary

No. of Crew	Diameter (ft)	Floor Orientation	End Bulkhead Configuration	Pressure Volume Total (ft^3)	Surface Area Total (ft^2)	Floor Area Total (ft^2)	Floor Area Useful (ft^2)	Cargo Volume (ft^3)	Total Docking Ports	Number of Modules	Number of MDA	Module Length (ft)	Total OLS Structural Weight (lb)	Pressure Shell Structural Weight (lb)	Surface Area - Useful Floor Area	Pressure Shell Weight Useful Floor Area (lb/ft^2)	Total OLS Structure Weight Useful Floor Area (lb/ft^2)	Reference Drawing No.
4	33	Transverse	Toroidal	19,218	4,480	1640	1510	940	4	1	0	40.2	31,609	7,752	2.97	5.13	20.9	2282-1
		Longitudinal	Flat	13,927	4,420	1553	1413	800	4	1	0	40.5	40,491	18,470	3.13	13.1	28.6	2282-2
	22	Transverse	Toroidal	10,980	3,480	1424	1272	750	6	1	0	49.6	23,445	7,449	2.73	5.85	18.4	2282-3
		Longitudinal	Flat	16,020	4,336	1503	1445	800	4	1	0	69.5	29,863	13,085	3.0	9.05	20.6	2282-4
	15	Transverse	Conical	14,620	4,932	1905	1785	1200	5 (2)	2	1	55.7	10,152 (3) 25,604 (4)(5)	3,010(3) 6,020 (5)	2.76	3.37	14.35 (5)	2282-5
		Longitudinal	Conical	14,620	4,932	1170	1130	1600	5 (2)	2	1	55.7	9,618 (3) 24,536 (4) (5)	3,010(3) 6,020 (5)	4.35	5.32	20.5 (5)	2282-6
	12	Transverse	Conical	14,154	6,264	1845	1665	1200	5 (2)	6	1	27.9	4,973 (3) 35,138 (4) (5)	1,452(3) 8,712 (5)	3.76	5.23	21.1 (5)	2282-7
		Longitudinal	Conical	14,514	6,264	1323	1263	2000	5 (2)	6	1	27.9	4,821 (3) 34,226 (4)(5)	1,452(3) 8,712 (5)	4.95	6.9	27.1 (5)	2282-8
8	33	Transverse	Toroidal	25,168	5,280	2460	2265	940	6	1	0	47.7	38,551	9,855	2.33	4.35	17.0	2282-9
		Longitudinal	Flat	20,467	5,180	2331	2214	870	5	1	0	48.7	49,309	20,488	2.34	9.26	22.3	2282-10
	22	Transverse	Toroidal	18,720	5,020	2492	2220	870	6	1	0	72.3	34,707	12,460	2.76	5.62	15.65	2282-11
		Longitudinal	Flat	23,690	5,790	2340	2282	1200	5	1	0	82.5	40,378	18,123	2.53	7.95	17.65	2282-12
	15	Transverse	Conical	21,930	7,398	2880	2700	1800	5 (2)	3	1	55.7	10,302 (3) 36,206 (4)(5)	3,010(3) 9,030 (5)	2.73	3.34	13.4 (5)	2282-13
		Longitudinal	Conical	21,930	7,398	1755	1725	2400	5 (2)	3	1	55.7	9,768 (3) 32,754 (4)(5)	3,010(3) 9,030 (5)	4.28	5.25	19.0 (5)	2282-14
	12	Transverse	Conical	21,231	9,396	2766	2496	2300	9 (2)	9	2	27.9	4,957 (3) 55,063 (4)(5)	1,452(3) 13,068 (5)	3.77	5.25	22.1 (5)	2282-15
		Longitudinal	Conical	23,040	9,396	1989	1899	2300	9 (2)	9	2	27.9	4,805 (3) 53,545 (4)(5)	1,452(3) 13,068 (5)	4.95	6.88	28.3 (5)	2282-16
12	33	Transverse	Toroidal	31,118	6,050	3280	3020	1060	6	1	0	55.3	45,390	12,146	2.0	4.02	15.0	2282-17
		Longitudinal	Flat	23,017	5,630	2605	2488	1000	5	1	0	52.0	53,795	21,301	2.26	8.57	21.6	2282-18
	22	Transverse	Toroidal	23,880	6,060	3204	2862	1050	6	1	0	88.3	42,492	16,255	2.12	5.68	14.8	2282-19
		Longitudinal	Flat	25,990	6,340	2635	2577	1350	5	1	0	90.5	45,757	20,981	2.46	8.15	17.8	2282-20
	15	Transverse	Conical	29,240	9,864	3840	3600	3000	5 (2)	4	1	55.7	10,256 (3) 46,724 (4)(5)	3,010(3) 12,040(5)	2.74	3.35	13.0 (5)	2282-21
		Longitudinal	Conical	29,240	9,864	2320	2280	3200	5 (2)	4	1	55.7	9,722 (3) 44,588 (4) (5)	3,010(3) 12,040 (5)	4.32	5.30	19.6 (5)	2282-22
	12	Transverse	Conical	23,590	10,500	3075	2775	2000	9 (2)	10	2	27.9	4,981 (3) 60,110 (4)(5)	1,452(3) 14,520 (5)	3.87	5.25	21.7 (5)	2282-23
		Longitudinal	Conical	28,160	11,550	2431	2321	3350	9 (2)	11	2	27.9	4,808 (3) 63,188 (4)(5)	1,452(3) 15,972 (5)	4.98	6.38	27.2 (5)	2282-24

(1) Total floor area less inter floor hatch area
(2) Number of docking ports available after modular assembly of OLS
(3) Weight per module
(4) Includes multiple docking adapters (5000 lb ea.) used for modular buildup, and support and adapter structural weight for transporting to lunar orbit by cislunar shuttles

(5) The 12- and 15-foot-diameter multimodule concept structural weight estimates do not include an allowance for cryogenic storage module requirements. These data are therefore not directly comparable with 22- and 33-foot diameter single module concepts which can store cryogenics in the unpressurized torus volumes at each end of the module

surface area, pressure shell, and total module weight as a function of usable floor area for each configuration. The computed ratios of these factors are shown in the table. These data were then plotted against module diameters for each of the three different crew sizes and for both the transverse and longitudinal floor orientations. Figure 2.3-3 presents plots for the three parameters previously identified for the 4-man vehicles. It should be noted that the data points on the graphs are not connected between the 15- and 22-foot-diameters because an obvious discontinuity exists in this region since diameter elements are single modules. Within each of these two distinct areas, the data points are relevant and are connected. In the absence of a third midpoint to establish the shape of the curve, a straight line was used to connect the two end data points. Another functional difference between the multimodule 12- and 15-foot-diameter concepts which should be kept in mind while reviewing the following weights estimates is that they are for structural weight only and do not include subsystems weight. Furthermore, in the case of cryogenic storage, both structural weight and RCS subsystem tank weight will be less for the single module concepts. This is due to two factors, both of which became evident during the analysis of the representative and derivative OLS configurations in Volume V:

1. The amount of cryogenic storage required for the single module concepts is less than for multimodular concepts because atmospheric leakage makeup and orbital maintenance requirements are lower for single module concepts.

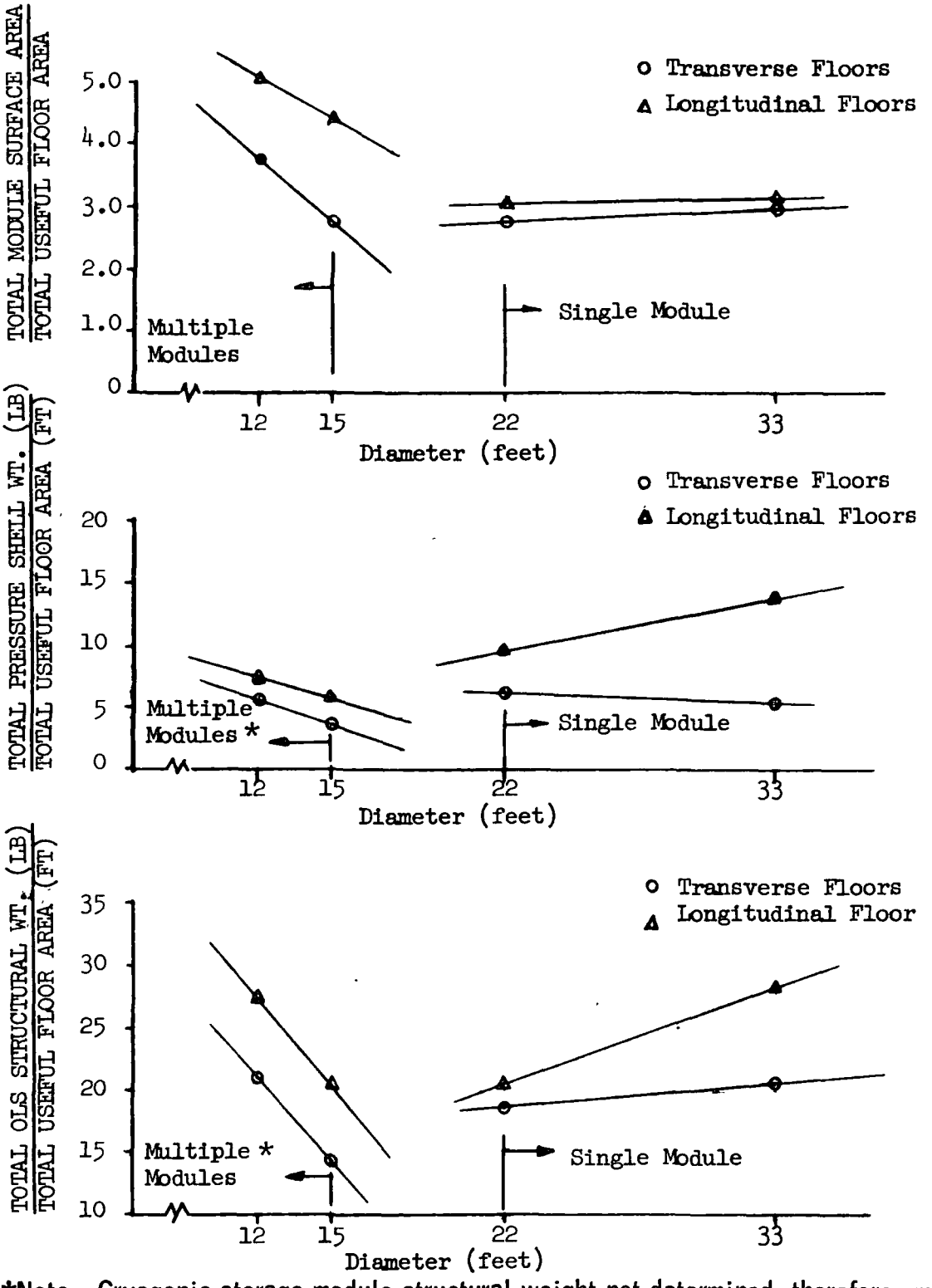
2. In the single module concepts, space for cryogenic tanks is available in the unpressurized volumes located at each end of the modules between the toroidal pressure bulkheads and the conical end close-outs. In the multimodular concepts, however, an additional unpressurized cryogenic storage module or modules would be required to safely house the necessary cryogenic storage tanks. Thus, the multimodular concepts structural weight should be increased by an unknown amount to account for the structural weight involved in these added cryogenic storage modules.

These effects may more than offset the apparent structural weight advantage of the multimodular concepts over the single module concepts. The following conclusions are reached through examination of Figure 2.3-3, which presents data for the 4-man-sized station concepts.

1. Transverse floors result in a lower surface area to floor area ratio for all module sizes.

2. The surface area to floor area ratio is essentially a minimum and is the same for a station composed of 15-foot-diameter modules as it is for a single 22-foot-diameter station.

3. The slope of the surface area to floor area ratio is relatively shallow between the 22- and 33-foot-diameter points and would indicate a low level of sensitivity. The minimum value, however, is at the 22-foot diameter.



*Note: Cryogenic storage module structural weight not determined; therefore, multiple module concepts are not directly comparable with single module concepts.

Figure 2.3-3. Module Surface Area, Pressure Shell Weight, and Total OLS Structural Weight Versus Useful Floor Area - 4-Man Capacity

4. The pressure shell weight to floor area ratio is lower for the transverse floors than for longitudinal floors.

5. The pressure shell weight to floor area ratio is lower for 12- and 15-foot diameter modules than for single 22- and 33-foot-diameter modules.

6. The lowest pressure shell weight to floor area ratio for the 15-foot-diameter module is 30 percent less than for a single 22-foot diameter module.

7. The total core module weight to floor area ratios are less for all diameters featuring transverse floors.

8. The lowest total core module weight to floor area ratio is for the 15-foot-diameter modules. The value is 22 percent lower than the value for the 22-foot-diameter module. From purely structural weight considerations, this can be considered to be the most efficient concept for a 4-man station. As noted, however, the structural weight required for a cryogenic storage module by the modular concepts was not determined.

Figure 2.3-4 presents the parametric data for the 8-man-sized station concepts. The following conclusions are reached through examination of the figure.

1. Transverse floor arrangements are more structurally efficient than longitudinal floor arrangements.

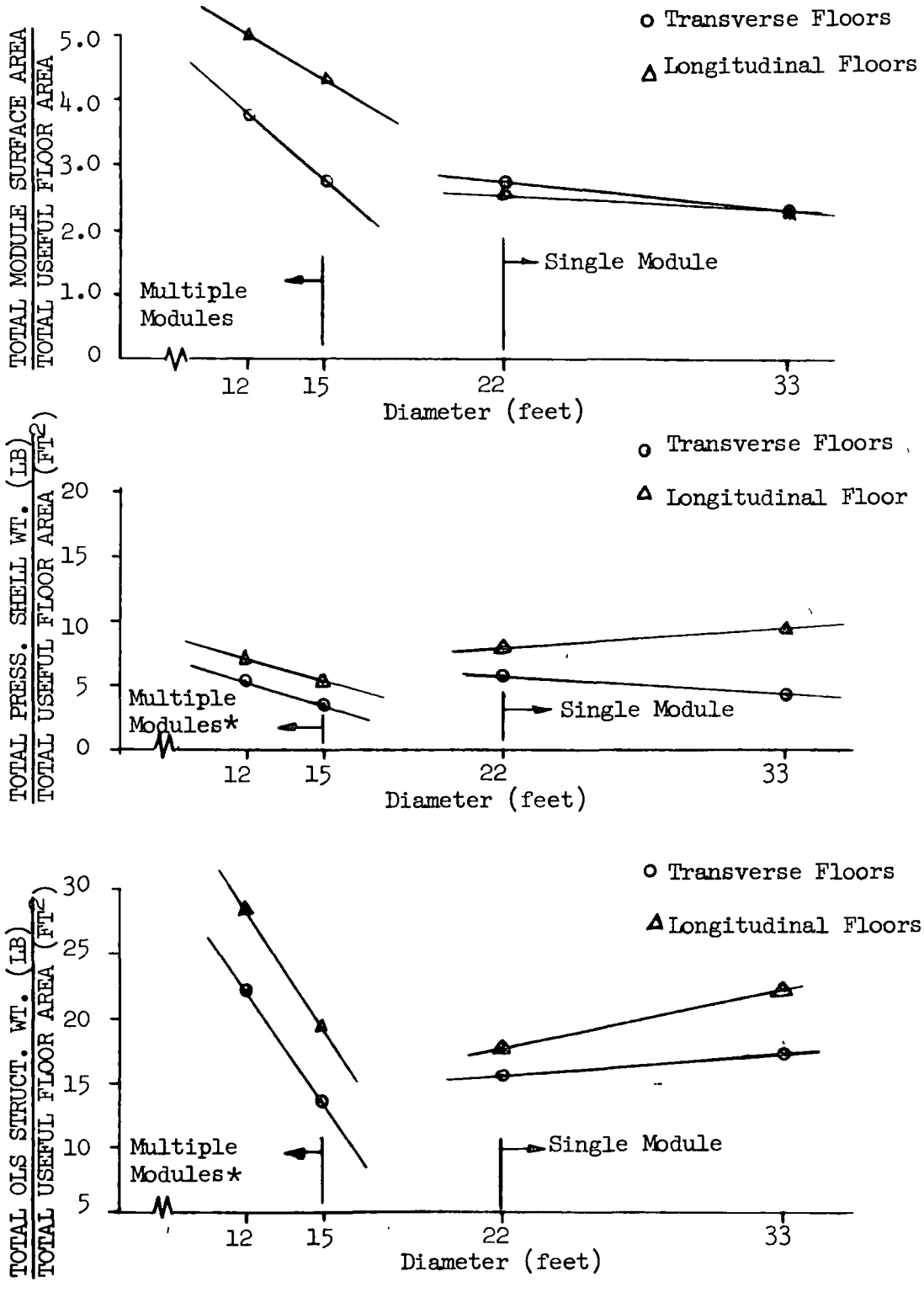
2. The surface area to floor area ratio is lower for single modules in the 22- to 33-foot-diameter modules.

3. The pressure shell weight to floor area ratio is lower for the 12- and 15-foot diameter modules. The 15-foot-diameter module has a 42 percent lower ratio than the 22-foot-diameter module.

4. The total core module weight to floor area ratio is lowest for the 15-foot-diameter clustered modules. This would indicate that from structural weight considerations, the 15-foot-diameter modules are more efficient. As noted previously, the structural weight required for cryogenic storage modules by the multimodule concept was not determined.

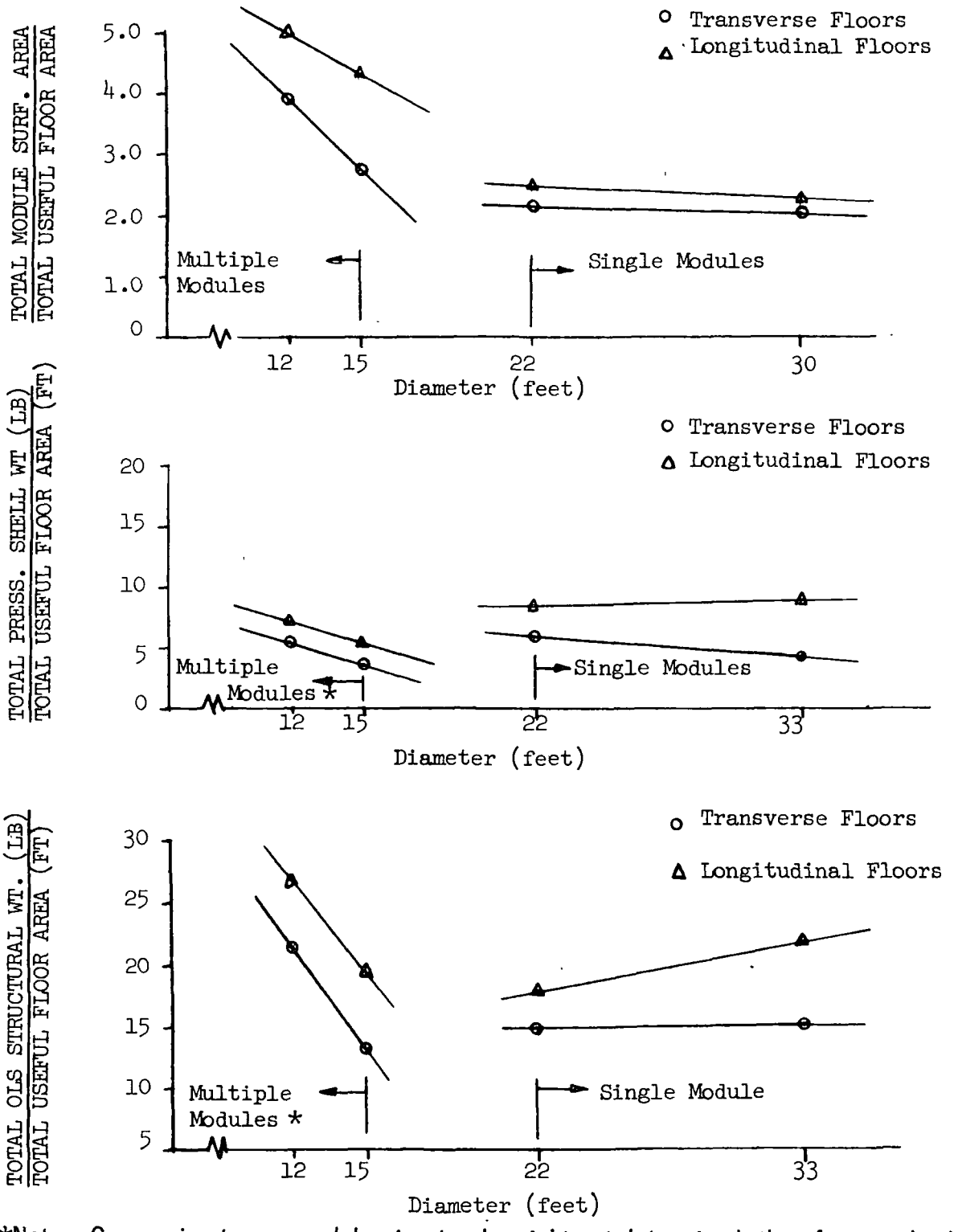
Figure 2.3-5 summarizes the data for the 12-man capacity stations. The following conclusions can be stated.

1. All transverse floor arrangements for a particular module diameter are more efficient structurally than are the longitudinal floor arrangements.



*Note: Cryogenic storage module structural weight not determined; therefore, multiple module concepts are not directly comparable with single module concepts.

Figure 2.3-4. Module Surface Area, Pressure Shell Weight, and Total OLS Structural Weight Versus Useful Floor Area - 8-Man Capacity



*Note: Cryogenic storage module structural weight not determined; therefore, multiple module concepts are not directly comparable with single module concepts.

Figure 2.3-5. Module Surface Area, Pressure Shell Weight, Total OLS Structural Weight Versus Useful Floor Area - 12-Man Crew Capacity

2. The pressure shell weight to floor area ratio is lower for the clustered 12- and 15-foot-diameter modular approach than for the single 22- and 33-foot-diameter modules.

3. The total core module weight to floor area ratio is lower for the clustered 15-foot-diameter module concept. As noted previously, the structural weight required for the cryogenic storage modules by the multimodule concept was not determined.

To summarize overall conclusions for the 4-, 8-, and 12-man crew sizes, the following statements can be made.

1. Transverse floor arrangements have lower structural weight and surface area to floor area ratios than do the longitudinal floor arrangements.

2. Regardless of crew size, in the range of 4 to 12 men, the 15-foot-diameter module cluster stations with transverse floors provide the lowest total OLS structural weight to floor area ratio. The next lowest total structural weight to floor area ratio for all crew sizes is provided by the 22-foot-diameter station, followed closely by the 33-foot-diameter station, both with transverse floors. As noted previously, however, the structural weight of the 12- and 15-foot-diameter modular concepts shown in Figures 2.3-3, 2.3-4, and 2.3-5 should be increased by an undetermined amount to account for the structural weight of cryogenic storage module(s).

3. Based on the data shown in Figures 2.3-3, 2.3-4, and 2.3-5, if a modular concept, capable of delivery to earth orbit in the earth orbit shuttle (EOS) cargo bay is either desired or required, a 15-foot-diameter module is clearly preferred over a 12-foot or smaller diameter module, regardless of crew size.

4. For single module stations in the size range of 22 to 33 feet in diameter for modules with transverse floors, the smaller diameter is always preferable regardless of crew size. However, the increase in the ratio of total structural weight to useful floor area with station diameter is shallow, especially for the 12-man crew size. Concept selection would probably be based upon other program issues.

5. For the required OLS 8-man crew size, the OLS made up of 15-foot-diameter modules with transverse floors has a total structural weight to useful floor area ratio of 13.4, the lowest value of all possibilities studied. This value would

increase, however, if the undetermined structural weight of the cryogenic storage module(s) required were accounted for. The single module versions with transverse floors have ratios of total structural weight to useful floor area of 15.65 and 17.0 for modular diameters of 22 and 33 feet, respectively. The difference in these values is not large and concept solution would not be governed by this factor alone.

2.4 OLS STRUCTURE SUBSYSTEM WEIGHT

The OLS structure subsystem parametric weight analysis took into consideration 24 configurations in order to encompass the prime variables of crew size (4 through 12), core module diameter (12 to 33 feet), and use of transverse (circular) floors versus longitudinal floors. The weight and other specific data of the structure subsystem constituents used in the analysis are shown in the parametric weight matrix of Tables 2.4-1 and 2.4-2 for the circular bulkheads and longitudinal bulkheads respectively.

The weight analysis drew heavily upon the EOSS as a source of basic data. The 12- and 15-foot-diameter OLS configurations are considered as shuttle-launched vehicles while the 22- and 33-foot-diameter configurations are considered as earth-launched vehicles. A 5-g boost loading criteria was assumed for the earth-launched configurations (same as that of the EOSS). However, the stiffness criteria imposed by the natural frequency requirement of OLS configurations consisting of 22- to 33-foot-diameter modules with side-docked tugs and cargo modules were the controlling weight factor. Structural loadings for the 22- and 33-foot diameters with side-docked tugs and cargo modules were considered similar to those on the EOSS.

For the shuttle-launched configurations, a maximum loading of 3 g was assumed (exclusive of crash loadings). Also, it was assumed that loading rather than stiffness criteria was the controlling weight factor because of no side docking ports. Certain portions of the structure (noted in the discussion) have considered these factors in the establishment of their weights, which are reflected in structural unit weights in pounds per square foot.

2.4.1 WEIGHT ESTIMATES

Lower Skirt. The lower skirt exists only on the 22- and 33-foot-diameter configurations. A unit weight of 2.02 lb/ft^2 was used on both the 8- and 12-man configurations because the design was governed by the stiffness criteria of assembled side-docked modules. This value was lowered to 1.28 lb/ft^2 on the 4-man configuration because this much shorter module was assumed to be more critical to the 5-g boost load criteria than to stiffness criteria. In all cases, skin-stringer design was assumed. No skirt exists on the 12- and 15-foot-diameter modules because these are assumed to be transported to earth orbit within the cargo bay of the Earth Orbit Shuttle (EOS) vehicle.

Cislunar Shuttle Adapter. (Required for 12- and 15-foot-diameter stations only) For 12- and 15-foot-diameter modules, an interstage structure between the multiple docking adapter (MDA), used to link station modules together, and the cislunar shuttle (CLS). This interstage structure is estimated to weigh 300 pounds. For 15-foot-diameter 60-foot long modules, frame supports are required to reduce the moments of the MDA docking ports; see Figure 2.3-2. These supports are estimated to weigh 400 pounds for the four frames required

Table 2.4-1. OLS Structure Parametric Weight Matrix - Transverse Floors (Circular)

Crew Size	4				8				12			
Core Module (dia. - ft/requirement)	12/(6)	15/(2)	22	33	12/(9)	15/(3)	22	33	12/(10)	15/(4)	22	33
Drawing Number 2282-	-7	-5	-3	-1	-15	-13	-11	-9	-23	-21	-19	-17
Total core module structure weight, lb (Including docking ports, environment protection, and radiators)	(4973)	(10,152)	(23,445)	(31,609)	(4957)	(10,302)	(34,707)	(38,551)	(4981)	(10,256)	(42,492)	(45,390)
Body structure	--	--	(17,077)	(24,964)	--	--	(26,374)	(30,495	--	--	(32,805)	(35,933)
Lower nonpressurized compartment, wt, lb	--	--	(455)	(1046)	--	--	(557)	(1328)	--	--	(557)	(1328)
Lower skirt												
Area, ft^2	--	--	138	380	--	--	138	380	--	--	138	380
Specific weight, lb/ft^2	--	--	1.28	1.28	--	--	2.02	2.02	--	--	2.02	2.02
Weight, lb	--	--	177	486	--	--	279	768	--	--	279	768
Lower end cap (closeout)												
Area, ft^2	--	--	400	877	--	--	400	877	--	--	400	877
Specific weight, lb/ft^2	--	--	0.57	0.57	--	--	0.57	0.57	--	--	0.57	0.57
Weight, lb	--	--	228	500	--	--	228	500	--	--	228	500
Attachments	--	--	50	60	--	--	50	60	--	--	20	20
Pressurized compartment weight, lb	(2664)	(6188)	(15,606)	(21,511)	(2664)	(6188)	(24,741)	(26,761	(2664)	(6188)	(31,172)	(32,199)
End pressure bulkheads (2)												
Area, ft^2 (total)	404	504	943	2230.5	404	504	943	2230.5	404	504	943	2230.5
Constructure	Conic	Conic	Torus	Torus	Conic	Conic	Torus	Torus	Conic	Conic	Torus	Torus
Specific weight, lb/ft^2	1.0	1.0	0.83	0.83	1.0	1.0	0.83	0.83	1.0	1.0	0.83	0.83
Weight, lb	404	504	783	1851	404	504	783	1851	404	504	783	1851
Attachments	40	40	--	--	40	40	--	--	40	40	--	--
Internal pressure bulkhead (honeycomb)												
Area, ft^2 (total)	300	960	1441	1672	300	960	2528	2508	300	960	3249	3344
Number required	3	6	4	2	3	6	7	3	3	6	9	4
Specific weight, lb/ft^2	1.0	1.2	1.64	2.18	1.0	1.2	1.64	2.18	1.0	1.2	1.64	2.18
Weight, lb	300	1150	2363	3645	300	1150	4146	5468	300	1150	5328	7290
Ceiling weight, lb	50	80	--	--	50	80	--	--	50	80	--	--
Outer cylindrical shell (aluminum skin-structure)												
Length, ft	22.5	44.7	31.1	16.0	22.5	44.7	54	23.3	22.5	44.7	69.17	31.1
Area, ft^2	848	2106	2099	1597	848	2106	3683	2325	848	2106	4716	3100
Specific weight, lb/ft^2	1.0	1.0	2.2	2.27	1.0	1.0	2.5	2.27	1.0	1.0	2.72	2.27
Weight, lb	848	2106	4618	3625	848	2106	9208	5278	848	2106	12,817	7389
Tunnel - upper												
Area, ft^2	--	--	265	482.7	--	--	265	482.7	--	--	265	482.7
Specific weight, lb/ft^2	--	--	1.85	1.85	--	--	1.85	1.85	--	--	1.85	1.85
Weight, lb	--	--	490	893	--	--	490	893	--	--	490	893
Tunnel - lower												
Area, ft^2	--	--	136	257.6	--	--	136	257.6	--	--	136	257.6
Specific weight, lb/ft^2	--	--	1.85	1.85	--	--	2.9	2.9	--	--	2.9	2.9
Weight, lb	--	--	252	477	--	--	394	747	--	--	394	747
Tunnel - upper RCS	--	--	25	38	--	--	25	38	--	--	25	38
Tunnel - lower RCS	--	--	37	55	--	--	37	55	--	--	37	55
Tunnel attachment	--	--	91	91	--	--	91	91	--	--	91	91
Tension tie (titanium rod)												
No. req.	--	--	6	6	--	--	6	6	--	--	6	6
Length, ft (per rod)	--	--	31.1	15.3	--	--	54	22.33	--	--	69	31
Weight, lb	--	--	165	336	--	--	581	490	--	--	742	666
Docking hatch - number and weight, lb	(2) 160	(2) 360	(6) 1080	(4) 720	(2) 160	(2) 360	(6) 1080	(5) 900	(2) 160	(6) 360	(6) 1080	(6) 1080
Ground access hatch provision, number and weight	--	--	(2)186	(2)186	--	--	(5)465	(2)186	--	--	(7) 651	(2)186
Pressure bulkhead hatch - number and weight, lb	--	--	(1)160	(1)160	--	--	(1)160	(1)160	--	--	(1) 160	(1)160
Inertia bulkhead hatch - number and weight, lb	(4) 80	(7) 140	(9)260	(5)144	(4) 80	(7)140	(15)440	(7)203	(4) 80	(7) 140	(19)560	(9)260
Torus access hatch - number and weight, lb	--	--	(4)140	(4)140	--	--	(4)140	(4)140	--	--	(4) 140	(4)140

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Table 2.4-1. OLS Structure Parametric Weight Matrix - Transverse Floors (Circular) (Cont'd.)

Crew Size	4				8				12			
Core Module (Diameter - ft/requirement)	12/(6)	15/(2)	22	33	12/(9)	15/(3)	22	33	12/(10)	15/(4)	22	33
Internal airlock	--	--	(1) 350	(1) 350	--	--	(1) 350	(1) 350	--	--	(1) 350	(1) 350
Windows and ports - number and weight, lb	(2) 36	(5) 90	(3) 84	(2) 56	(2) 36	(5) 90	(6) 168	(3) 84	(2) 36	(5) 90	(8) 224	(3) 84
Partitions - vertical (foam FG honeycomb)												
Area, ft^2/level	(3) 77/L	(6) 58/L	(4) 422/L	(2) 630/L	(3) 77/L	(6) 58/L	(7) 422/L	(3) 630/L	(3) 77/L	(6) 58/L	(9) 422/L	(4) 630/L
Specific weight, lb/ft^2	0.36	0.36	0.36	0.36	0.36	0.36	0.36	0.36	0.36	0.36	0.36	0.36
Weight, lb	83	126	636	454	83	126	1063	680	83	126	1367	907
False ceilings (foam FG honeycomb)												
Area, ft^2/level	(3) 29/L	(6) 45/L	(4) 98/L	(2) 220/L	(3) 29/L	(6) 45/L	(7) 98/L	(3) 220/L	(3) 29/L	(6) 45/L	(9) 98/L	(4) 220/L
Specific weight, lb/ft^2	0.5	0.5	0.6	0.6	0.5	0.5	0.6	0.6	0.5	0.5	0.6	0.6
Weight, lb	44	135	235	264	44	135	412	396	44	135	529	528
Storage provisions												
Area, ft^2	100	180	892	1048	100	180	1565	1528	100	180	2007	2038
Specific weight, lb/ft^2	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.60
Weight, lb	60	110	535	629	60	110	939	917	60	110	1204	1223
Vertical utility trunks	307	815	390	433	307	815	685	632	307	815	876	843
Horizontal utility trunks	74	230	242	180	74	230	424	270	74	230	544	360
RCS access dome outer covers - number and weight, lb	--	--	(4) 200	(4) 200	--	--	(4) 200	(4) 200	--	--	(4) 200	(4) 200
Window covers	(2) 12	(5) 30	(3) 18	(2) 12	(2) 12	(5) 30	(6) 36	(3) 18	(2) 12	(5) 30	(8) 48	(3) 18
Brackets	27	84	89	102	27	84	157	153	27	84	201	204
Contingency	39	88	155	200	39	88	245	291	39	88	309	366
Upper nonpressurized compartment weight, lb	--	--	(1016)	(2406)	--	--	(1076)	(2406)	--	--	(1076)	(2406)
Upper skirt	--	--	--	--	--	--	--	--	--	--	--	--
Area, ft^2	--	--	115	239	--	--	115	239	--	--	115	239
Specific weight, lb/ft^2	--	--	1.03	1.62	--	--	1.03	1.62	--	--	1.03	1.62
Weight, lb	--	--	118	387	--	--	118	387	--	--	118	387
Upper end cap (closeout) - weight	--	--	658	1572	--	--	658	1572	--	--	658	1572
Area, ft^2	--	--	598	1182	--	--	598	1182	--	--	598	1182
Specific weight, lb/ft^2	--	--	1.1	1.33			1.10	1.33			1.10	1.33
Fairing adapter weight	--	--	218	325	--	--	218	325	--	--	218	325
Attachments - weight	--	--	82	122	--	--	82	122	20		82	122
Induced environment protection	(1281)	(2672)	(3486)	(4424)	(1281)	(2672)	(5109)	(5162)	(1281)	(2672)	(6177)	(5947)
Thermal protection weight	(154)	(322)	(471)	(576)	154	322	(669)	(659)	154	322	(807)	(747)
Coatings	6	12	24	24	6	12	24	24	6	12	24	24
Insulation (40-layer Mylar; 1-inch thick)												
Main cylindrical shell	86	213	238	181	86	213	417	264	86	213	535	352
Lower skirt	--	--	15	43	--	--	15	43	--	--	15	43
Lower end cap	11	23	46	100	11	23	46	100	11	23	46	100
Upper skirt	--	--	13	27	--	--	13	27	--	--	13	27
Upper end cap	11	23	68	134	11	23	68	134	11	23	68	134
Miscellaneous	10	10	26	26	10	10	26	26	10	10	26	26
Vent fitting	30	41	41	41	30	41	60	41	30	41	80	41
Meteoroid protection weight at 0.9 #/ft^2	(1127)	(2350)	(3015)	(3848)	(1127)	(2350)	(4440)	(4503)	(1127)	(2350)	(5370)	(5200)
Docking structure weight	(800)	(800)	(2020)	(1360)	(800)	(800)	(2020)	(1690)	(800)	(800)	(2020)	(2020)
Lower end port - weight	520	520	420	420	520	520	420	420	520	520	420	420
Upper end port - weight	280	280	280	280	280	280	280	280	280	280	280	280
Side ports - number and weight	--	--	(4) 1320	(2) 660	--	--	(4) 1320	(3) 990	--	--	(4) 1320	(4) 1320
Radiator weight at 0.368 lb/ft^2	(230)	(492)	(862)	(862)	(214)	(642)	(1204)	(1204)	(238)	(596)	(1490)	(1490)
[illegible]s/module ft^2	937	1338	2343	2343	654	1745	3273	3273	540	1619	4048	4048

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Table 2.4-2. OLS Structure Parametric Weight Matrix - Longitudinal Floors

Crew Size	4				8				12			
Core Module Dia. - Ft/Req.	12/(6)	15/(2)	22	33	12/(9)	15/(3)	22	33	12/(11)	15/(4)	22	33
Drawing No. 2282-	-8	-6	-4	-2	-16	-14	-12	-10	-24	-22	-20	-18
Total core module structure weight, lb (Including docking ports, environment protection, and radiators)	(4821)	(9618)	(29,863)	(40,491)	(4805)	(9768)	(40,378)	(49,309)	(4808)	(9722)	(45,575)	(53,795)
Body structure	--	--	(23,335)	(23,080)	--	--	(31,871)	(40,404)	--	--	(35,965)	(44,242)
Lower nonpressurized compartment weight, lb	--	--	(535)	(1136)	--	--	(637)	(1418)	--	--	(637)	(1418)
Lower skirt, wt-lb	--	--	177	486	--	--	279	768	--	--	279	768
Area, ft^2	--	--	138	380	--	--	138	380	--	--	138	380
Specific weight, lb/ft^2	--	--	1.28	1.28	--	--	2.02	2.02	--	--	2.02	2.02
Lower end cap (closeout), wt-lb	--	--	308	590	--	--	308	590	--	--	308	590
Area - ft^2	--	--	541	1036	--	--	541	1036	--	--	541	1036
Specific weight, lb/ft^2	--	--	0.57	0.57	--	--	0.57	0.57	--	--	0.57	0.57
Attachments	--	--	50	60	--	--	50	60	--	--	50	60
Pressurized compartment weight, lb	(2512)	(5654)	(21,647)	(28,655)	(2512)	(5654)	(29,781)	(35,697)	(2512)	(5654)	(34,175)	(39,535)
End pressure bulkhead												
Area, ft^2 (total)	404	504	722	1672	404	504	722	1672	404	504	722	1672
Design	Conic	Conic	Flat	Flat	Conic	Conic	Flat	Flat	Conic	Conic	Flat	Flat
Specific weight, lb/ft^2	1.0	1.0	5.6	7.5	1.0	1.0	5.6	7.5	1.0	1.0	5.6	7.5
Weight, lb	404	504	4044	12,540	404	504	4044	12,540	404	504	4044	12,540
Attachments	40	40	--	--	40	40	--	--	--	40	--	--
Internal pressure bulkhead												
Area, ft^2	--	--	361	550	--	--	361	817	--	--	361	935
Specific weight, lb/ft^2	--	--	5.6	6.0	--	--	5.6	7.0	--	--	5.6	7.5
Weight, lb	--	--	2022	3300	--	--	2022	5710	--	--	2022	7012
Inertial bulkheads (honeycomb)												
Area, ft^2 (total)	242	745	1139	1200	242	745	1800	1782	242	745	2016	2040
Number required	1	1	2	3	2	1	2	3	1	1	2	3
Specific weight, lb/ft^2	.8	1.0	2.5	2.5	.8	1.0	2.7	2.7	.8	1.0	3.0	3.0
Weight, lb	194	745	2848	3000	194	745	4860	4811	194	745	6048	6120
Ceiling and floor supports, lb	70	200	300	--	70	200	400	--	70	200	500	--
Outer cylindrical shell (aluminum skin - structure)												
Length, ft	22.5	45	42	18.8	22.5	45	65.5	26.9	22.5	45	73	30.4
Area, ft^2	848	2106	2837	1884	848	2106	4426	2694	848	2106	4936	3052
Specific weight, lb/ft^2	1.0	1.0	2.5	2.27	1.0	1.0	2.7	2.27	1.0	1.0	3.0	2.27
Weight, lb	848	2106	7092	4277	848	2106	11,950	6115	848	2106	14,808	6928
Tunnel - upper												
Area, ft^2	--	--	257	293	--	--	257	293	--	--	257	293
Specific weight, lb/ft^2	--	--	1.85	1.85	--	--	1.85	1.85	--	--	1.85	1.85
Weight, lb	--	--	475	542	--	--	475	542	--	--	475	542
Tunnel - lower												
Area, ft^2	--	--	112	161	--	--	112	161	--	--	112	161
Specific weight, lb/ft^2	--	--	1.85	1.85	--	--	1.85	1.85	--	--	1.85	1.85
Weight, lb	--	--	207	298	--	--	207	298	--	--	207	298
Tunnel - upper RCS	--	--	25	38	--	--	25	38	--	--	25	38
Tunnel - lower RCS	--	--	37	55	--	--	37	55	--	--	37	55
Tunnel attachments	--	--	91	91	--	--	91	91	--	--	91	91
Docking hatch - number and weight, lb	(2) 160	(2) 360	(4) 720	(4) 720	(2) 160	(2) 360	(5) 900	(5) 900	(2) 160	(2) 360	(5) 900	(5) 900
Ground access hatch provisions - number and weight, lb	--	--	(2) 186	(1) 93	--	--	(2) 186	(1) 93	--	--	(2) 186	(1) 93
Pressure bulkhead hatch - number and weight, lb	--	--	(3) 480	(3) 480	--	--	(3) 480	(3) 480	--	--	(3) 480	(3) 480
Inertial bulkhead hatch - number and weight, lb	(1) 20	(1) 20	(2) 57	(5) 143	(1) 20	(1) 20	(2) 57	(5) 143	(1) 20	(1) 20	(5) 57	(5) 143
Internal airlock	--	--	350	350	--	--	350	350	--	--	350	350
Windows and ports - number and weight	(2) 36	(5) 90	(3) 84	(3) 84	(2) 36	(5) 90	(3) 84	(3) 84	(2) 36	(5) 90	(3) 84	(3) 84

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Table 2.4-2. OLS Structure Parametric Weight Matrix - Longitudinal Floors (Cont'd)

Crew Size	4				8				12			
Core Module Dia. - Ft/Req.	12/(6)	15/(2)	22	33	12/(9)	15/(3)	22	33	12/(11)	15/(4)	22	33
* Partitions - vertical												
Area, ft^2/level	(2) 117/L	(2) 290/L	(2) 403/L	(4) 265/L	(2) 117/L	(2) 290/L	(2) 637/L	(2) 379/L	(2) 117/L	(2) 290/L	(2) 713/L	(4) 450
Specific weight, lb/ft^2	0.36	0.36	0.36	0.36	0.36	0.36	0.36	0.36	0.36	0.36	0.36	0.36
Weight, lb	85	210	290	382	85	210	459	546	85	210	513	648
* False ceilings												
Area, ft^2/level	(1) 215/L	(1) 530/L	(2) 192/L	(4) 92/L	(1) 215/L	(1) 530/L	(2) 290/L	(4) 132/L	(1) 215/L	(1) 530/L	(2) 248/L	(4) 156/L
Specific weight, lb/ft^2	0.5	0.5	0.6	0.6	0.6	0.5	0.6	0.6	0.5	0.5	0.6	0.6
Weight, lb	107	265	230	221	107	265	348	317	107	265	298	374
* Storage provisions	160	450	967	856	160	250	1528	1271	160	250	1712	1455
Specific weight, lb/ft^2	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6	0.6
Weight, lb	100	150	580	514	100	150	917	763	100	150	1027	873
* Vertical utility trunks	120	300	649	649	120	300	649	649	120	300	649	649
* Horizontal utility trunks	100	300	302	270	100	300	477	400	100	300	534	459
* RCS access dome outer cover, number and weight, lb	100	100	(4) 200	(4) 200	100	100	(4) 200	(4) 200	100	100	(4) 200	(4) 200
* Window covers	(2) 12	(5) 30	(3) 18	(3) 18	(2) 12	(5) 30	(3) 18	(3) 18	(2) 12	(5) 30	(3) 18	(3) 18
* Brackets	80	150	150	200	80	150	237	297	80	150	265	340
Contingency	36	84	210	200	36	84	308	257	36	84	360	300
Upper nonpressurized compartment, weight, lb	--	--	(1153)	(3289)	--	--	(1153)	(3289)	--	--	(1153)	(3298)
Upper skirt												
Area, ft^2	--	--	--	--	--	--	--	--	--	--	--	--
Specific weight, lb/ft^2	--	--	--	--	--	--	--	--	--	--	--	--
Weight, lb	--	--	--	--	--	--	--	--	--	--	--	--
Upper end cap (closeout) weight	--	--	853	2842	--	--	853	2842	--	--	853	2842
Area, ft^2	--	--	812	2138	--	--	812	2138	--	--	812	2138
Specific weight, lb/ft^2	--	--	1.10	1.33	--	--	1.10	1.33	--	--	1.10	1.33
Fairing adapter weight	--	--	218	325	--	--	218	325	--	--	218	325
Attachments	--	--	82	122	--	--	82	122	--	--	82	122
Induced environment protection	(1281)	(2672)	(4306)	(5189)	(1281)	(2672)	(5913)	(6011)	(1284)	(2672)	(6430)	(6373)
Thermal protection weight	(154)	(322)	(536)	(639)	(154)	(322)	(713)	(730)	(154)	(322)	(770)	(770)
Coatings	6	12	24	24	6	12	24	24	6	12	24	24
Insulation (40-layer Mylar, 1-in. thick												
Main cylindrical shell	86	234	316	212	86	234	493	303	86	234	550	343
Lower skirt	--	--	--	--	--	--	--	--	--	--	--	--
Lower end cap	11	23	61	118	11	23	61	118	11	23	61	118
Upper skirt	--	--	--	--	--	--	--	--	--	--	--	--
Upper end cap	11	23	92	242	11	23	92	242	11	23	92	242
Miscellaneous	10	10	26	26	10	10	26	26	10	10	26	26
Vent fittings	30	41	41	41	30	41	41	41	30	41	41	41
Meteoroid protection weight at 0.9 lb/ft^2	(1127)	(2349)	(3770)	(4550)	(1127)	(2349)	(5200)	(5281)	(1127)	(2349)	(5660)	(5603)
Docking structure weight	(800)	(800)	(1360)	(1360)	(800)	(800)	(1690)	(1690)	(800)	(800)	(1690)	(1690)
Lower end port - weight, lb	520	520	420	420	520	520	420	420	520	520	420	420
Upper end port - weight, lb	280	280	280	280	280	280	280	280	280	280	280	280
Side ports - weight, lb	--	--	(2) 660	(2) 660	--	--	990	(3) 990	--	--	(3) 990	(3) 990
Radiator weight, lb at 0.368 lb/ft^2	(230)	(492)	(862)	(862)	(214)	(642)	(1204)	(1204)	(217)	(596)	(1490)	(1490)
Area, ft/module	625	1338	2343	2343	582	1745	3273	3273	589	1619	4048	4048

*Items considered as secondary structure

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for the 12-man configuration, and 150 pounds for the two frames required by the 8-man configuration. No frames are required for the 4-man configuration.

End Pressure Bulkhead. The end pressure bulkhead is toroidal on all of the 22- and 33-foot-diameter configurations with transverse floors and is assumed to be of aluminum monocoque construction. The unit weight was taken as 0.83 pounds per square foot.

The unit weight per square foot was based on an assumed aluminum honeycomb sandwich construction considered for the EOSS internal pressure bulkhead. As noted earlier, the end pressure bulkhead is flat on the 22- and 33-foot-diameter configuration with longitudinal floors,and conical on all 12- and 15-foot-diameter configurations. The following unit weights were used:

Configuration	Circular Floor	Longitudinal Floor
12-foot diameter	1.0 lb/ft^2	1.0 lb/ft^2
15-foot diameter	1.0 lb/ft^2	1.0 lb/ft^2
22-foot diameter	0.83 lb/ft^2	5.6 lb/ft^2
33-foot diameter	0.83 lb/ft^2	7.5 lb/ft^2

Internal Pressure Bulkhead. Based on an aluminum honeycomb design, the following unit weights were used for the OLS. In the case of circular floors, the internal pressure bulkhead is also a floor. In the case of longitudinal floors in 22-foot-diameter configurations, the circular internal pressure bulkhead dividing the module into two separate pressure volume is still required in addition to the longitudinal floor.

Configuration	Circular Floor	Longitudinal Floor
12-foot diameter	DNA	-
15-foot diameter	DNA	-
22-foot diameter	5.6 lb/ft^2	-
33-foot diameter	7.5 lb/ft^2	6.0 lb/ft^2 4-man 7.0 lb/ft^2 8-man 7.5 lb/ft^2 12-man

The difference in unit weight is attributed chiefly to floor support span of diameter and tension tie cantilever span. It should be noted that the analyses on this bulkhead as well as for the internal circular floors assume the use of tension ties. A constant loading per square foot of floor area on all configurations with common loading criteria is assumed as noted previously.

Inertia Bulkheads. The internal bulkheads (floors) are effected by cantilever span and tension tie spacing. The following unit weights were assumed:

Configuration	Circular Floor	Longitudinal Floor
12-foot diameter	1.0 lb/ft^2	0.8 lb/ft^2
15-foot diameter	1.2 lb/ft^2	1.0 lb/ft^2
22-foot diameter	1.64 lb/ft^2	2.7 lb/ft^2 8-man 2.5 lb/ft^2 4-man 3.0 lb/ft^2 12-man
33-foot diameter	2.18 lb/ft^2	2.7 lb/ft^2 8-man 2.5 lb/ft^2 4-man 3.0 lb/ft^2 12-man

Cylindrical Shell. In predicting the cylindrical shell weight, it was necessary to consider: (1) the boost loads on the 22- and 33-foot-diameter configuration, and (2) stiffness criteria on all diameters. There was some compensation of body slenderness versus lower module weight assumed for a given crew size. Taking these factors into account, the following unit weights were assumed:

Configuration	Circular Floor	Longitudinal Floor
4-Man		
12-foot diameter	1.0 lb/ft^2	1.0 lb/ft^2
15-foot diameter	1.0 lb/ft^2	1.0 lb/ft^2
22-foot diameter	2.2 lb/ft^2	2.5 lb/ft^2
33-foot diameter	2.27 lb/ft^2	2.27 lb/ft^2
8-Man		
12-foot diameter	1.0 lb/ft^2	1.0 lb/ft^2
15-foot diameter	1.0 lb/ft^2	1.0 lb/ft^2
22-foot diameter	2.5 lb/ft^2	2.7 lb/ft^2
33-foot diameter	2.27 lb/ft^2	2.27 lb/ft^2
12-Man		
12-foot diameter	1.0 lb/ft^2	1.0 lb/ft^2
15-foot diameter	1.0 lb/ft^2	1.0 lb/ft^2
22-foot diameter	2.72 lb/ft^2	3.0 lb/ft^2
33-foot diameter	2.27 lb/ft^2	2.27 lb/ft^2

Tunnels. All tunnel weights regardless of geometric variation were based on a unit weight of 1.85 lb/ft^2.

Tension Ties. Six radial titanium rod tension ties were assumed for the 22- and 33-foot-diameter configurations with circular floors. As mentioned previously, the unit loading on the floors per square foot was assumed to be constant for all configurations with common loading criteria. Thus, tension-tie weight was determined by module length and number of floors. No tension ties were assumed for the longitudinal floor configuration.

Hatches. Hatch weights were based on EOSS experience and on the number required for the various configurations shown in Tables 2.4-1 and 2.4-2.

Airlock. Airlock weight was based on EOSS data.

Partitions. Partition weight was predicted as a function of module diameter and length and was based on a determined value of area per level (floor). The unit weight per square foot of partition was taken as 0.36 lb/ft^2.

False Ceilings. Ceiling weight was a function of floor (or ceiling) area and was assumed as 0.6 lb/ft^2 on the 22- and 33-foot-diameter modules and 0.5 lb/ft^2 on the 12- and 15-foot-diameter modules.

Storage Provisions. Weight for storage areas was based on volume ratios (i.e., diameter and length). Unit weight was taken as 0.6 lb/ft^2.

Utility Trunks. Utility trunk weights were predicted as a function of module length (vertical trunks) and magnitude of systems involved. These weights were scaled from EOSS data.

Windows, Ports and Brackets. Unit weights for windows, ports, and brackets were based on EOSS data.

Contingency. Contingency weight for the pressurized compartment was assumed to be approximately 1 percent of total compartment weight.

Upper Skirt. The upper skirt is required only on the 22- and 33-foot-diameter transverse floor configurations. The following unit weights (based on the same considerations that applied to the lower skirt) were used:

22-foot diameter	1.03 lb/ft^2
33-foot diameter	1.62 lb/ft^2

Lower End Cap. The lower end cap is essentially a nonload-carrying structural item on the 22- and 33-foot diameter and assumes a unit weight of 0.57 lb/ft^2 on all configurations. On the 12- and 15-foot diameter it is a pressure wall with a unit weight of 1.0 lb/ft^2.

Upper End Cap. End cap weight was predicted as a function of loading and geometry. Unit weights were assigned as follows:

Configuration	Circular Floor	Longitudinal Floor
12-foot diameter	1.0 lb/ft^2	1.0 lb/ft^2
15-foot diameter	1.0 lb/ft^2	1.0 lb/ft^2
22-foot diameter	1.1 lb/ft^2	1.1 lb/ft^2
33-foot diameter	1.33 lb/ft^2	1.33 lb/ft^2

Fairing Adapter. A fairing adapter was assumed to be required on the boosted 22- and 33-foot-diameter configurations. These weights were predicted as a function of loading and scaled from EOSS predictions.

Insulation (Induced Environment Protection). All vehicles were assumed to have a 1-inch-thick, 40-layer Mylar insulation blanketed over the entire exposed surface between the meteoroid shield and structure. Unit weight of this type insulation is 0.1136 lb/ft^2.

Meteoroid Protection. Meteoroid protection weight was based on a 10-year life and a probability of no failure of 0.9. A meteoroid shield of approximately 0.2 lb/ft^2 is constant up to a value of 1 x 10^{11} for the product of area (M^2) times time (seconds). Checks on configuration geometry indicated that this value of unit weight could be used on all configurations. To this unit weight, however, weight had to be added for support closeouts and hardware. This brought the total installed weight to 0.9 lb/ft^2. This weight was applied to the complete vehicle exposed area (including radiator area). This was done because the radiator weight includes only the tubular fluid passages and the tube meteoroid protection boss, while in actual fact the radiator also requires fins between cooling tubes and mounting hardware.

Docking Structure Weight. Docking port weights are based on EOSS data and the specific OLS configurations requirements. The 12- and 15-foot-diameter modules, however, have a heavier weight docking port on one end of the module to provide for the assumed use of a multiple docking adapter that is used as a central docking fixture to which the individual OLS modules are end-docked.

Radiator. The previously noted radiator weight (0.368 lb/ft^2) consists of only the fluid passages and meteoroid protection for those passages. In predicting radiator weight, it was necessary to consider several factors, such as required area, available area of the configurations, the efficiency (effective area/available area), area required per module on cylindrical portion and upper conic and arc angle. All of these data are tabulated in Table 2.4-3 which includes the arc angle required for the radiator and the predicted weight.

Table 2.4-3. Estimated OLS Radiator Requirements for Weight Estimate

	Number of Modules n	1 Required Effective Area (ft^2)	2 Effective Area / Available Area	③ Area Required /Mod (①/②n)	4 (Circular Floors) Total Area (ft^2) Available per Module Cyl. + V. Conic	Arc Angle (Degrees) 360 (③/④)	Radiator Weight /Mod (lb) at 0.368 #/ft^2	Remarks
4-man		1874						
12-ft diameter	6		0.50	625	848	266	230	Full length cylinder
15-ft diameter	2		0.70	1338	2106	229	492	Full length cylinder
22-ft diameter	1		0.80	2343	2355 + 598 = 2953	285	862	Full length cylinder + conic
33-ft diameter	1		0.80	2343	2216 + 1182 = 3398	248	862	Full length cylinder + conic
8-man		2618						
12-ft diameter	9		0.50	582	848	247	214	Full length cylinder
15-ft diameter	3		0.50	1745	2106	298	642	Full length cylinder
22-ft diameter	1		0.80	3273	3936 + 598 = 4534	260	1204	Full length cylinder + conic
33-ft diameter	1		0.80	3273	2944 + 1182 = 4126	284	1204	Full length cylinder + conic
12-man		3238						
12-ft diameter	10		0.50	648	848	274	238	Full length cylinder
15-ft diameter	4		0.50	1619	2106	277	596	Full length cylinder
22-ft diameter	1		0.80	4048	4969 + 598 = 5567	262	1490	Full length cylinder + conic
33-ft diameter	1		0.80	4048	3719 + 1182 = 4901	297	1490	Full length cylinder + conic
12-man (Longitudinal Floor)		3238						
12-ft diameter	11		0.50	589	848	250	217	Full length cylinder

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Docking Adapters. Multiple docking adapters (MDA's) were used to connect the individual modules into an assemblage. Table 2.4-4 shows estimated weight of these adapters for the various 12- and 15-foot-diameter modules. These weights are based on a 6-port MDA at 5000 pounds apiece. The use of core modules instead of MDA's was considered but rejected as being heavier than MDA's. The modular arrangements are shown in Figures 2.3-1 and 2.3-2.

Table 2.4-4. Module Adapter Estimated Weights

	Transverse Floors	Longitudinal Floors
	MDA Weight (pounds)	MDA Weight (pounds)
4-Man		
12 ft diameter	5,000	10,000
15 ft diameter	5,000	5,000
8-Man		
12 ft diameter	10,000	10,000
15 ft diameter	5,000	5,000
12-Man		
12 ft diameter	10,000	10,000
15 ft diameter	5,000	5,000

2.4.2 PARAMETRIC WEIGHT CURVES

Floor and Bulkhead Weights. Figure 2.4-1 shows a plot of floor and bulkhead weights versus module diameter. These weights include both the pressure bulkhead and inertia bulkheads. To be noted is the minimal weight at 12- and 15-foot diameters for the longitudinal bulkheads.

Structural Cylindrical Shell. Figure 2.4-2 shows the weight of the main cylindrical shell for the three crew sizes versus module diameters. In this case, minimum weight occurs at four-man 33-foot diameter transverse floor and maximum at 22-foot diameter for the 12-man longitudinal floor configuration.

Environmental Protection Weight. Figure 2.4-3 shows the thermal plus meteoroid protection weight versus module diameter for the three crew sizes. The weight is minimal at the 22- and 33-foot diameter at the larger crew sizes but tends to shift to a 22-foot diameter as minimum at the 4-man crew case.

Radiator Weight. Figure 2.4-4 shows the radiator weight versus module diameter for the three crew sizes. Minimal weight is at the larger diameter.

Total Structure Subsystem Weight. Figures 2.4-5 and 2.4-6 present the total structure subsystem weight including docking, environment protection, and radiator versus crew size for the 12-, 15-, 22-, and 33-foot-diameter modules, respectively. For the smaller 12- and 15-foot diameter (shuttle-launched), the longitudinal floor configurations appear to be minimum weight except for the 12-man 12-foot-diameter modules. For the 22- and 33-foot

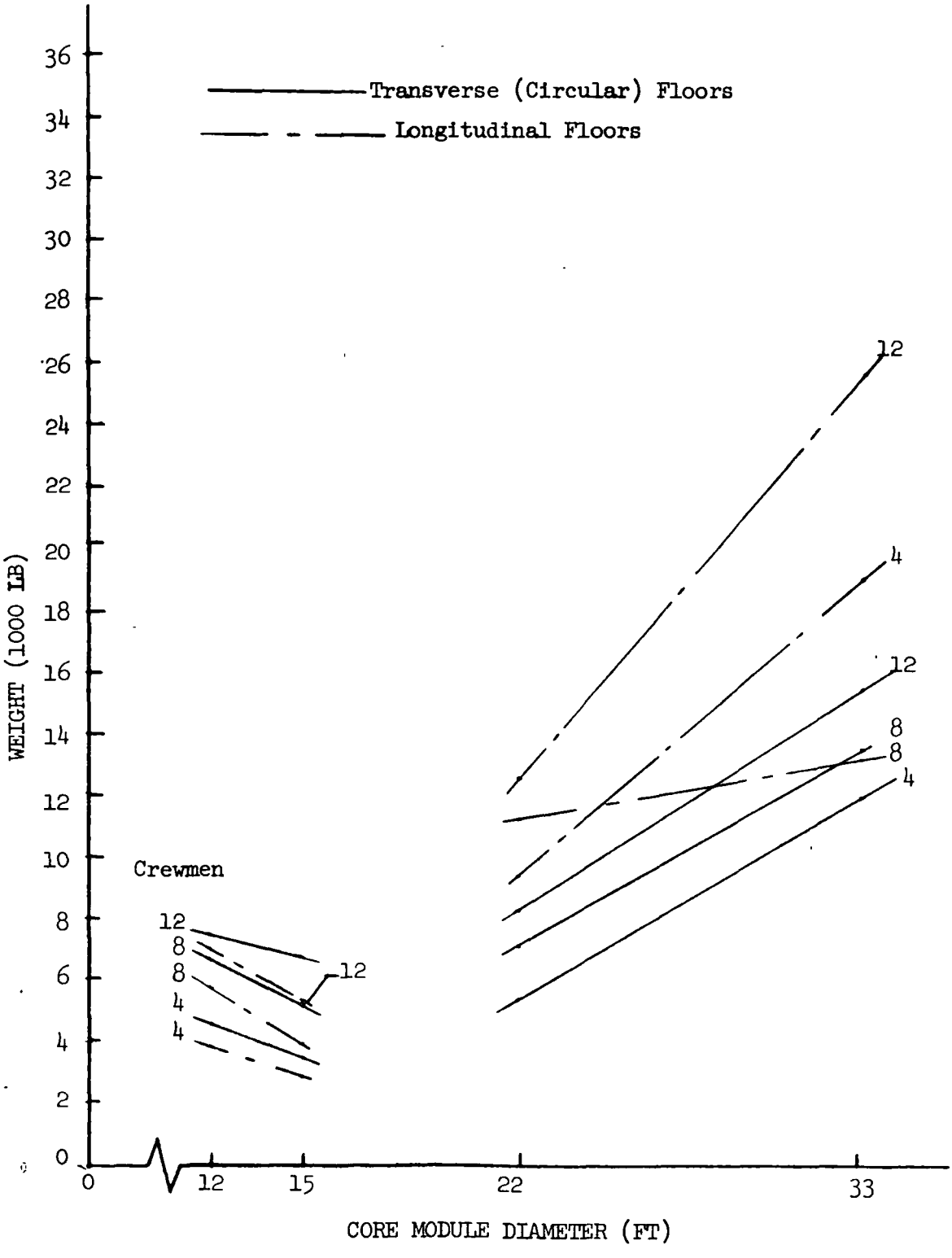


Figure 2.4-1. OLS Floor and Bulkhead Weights

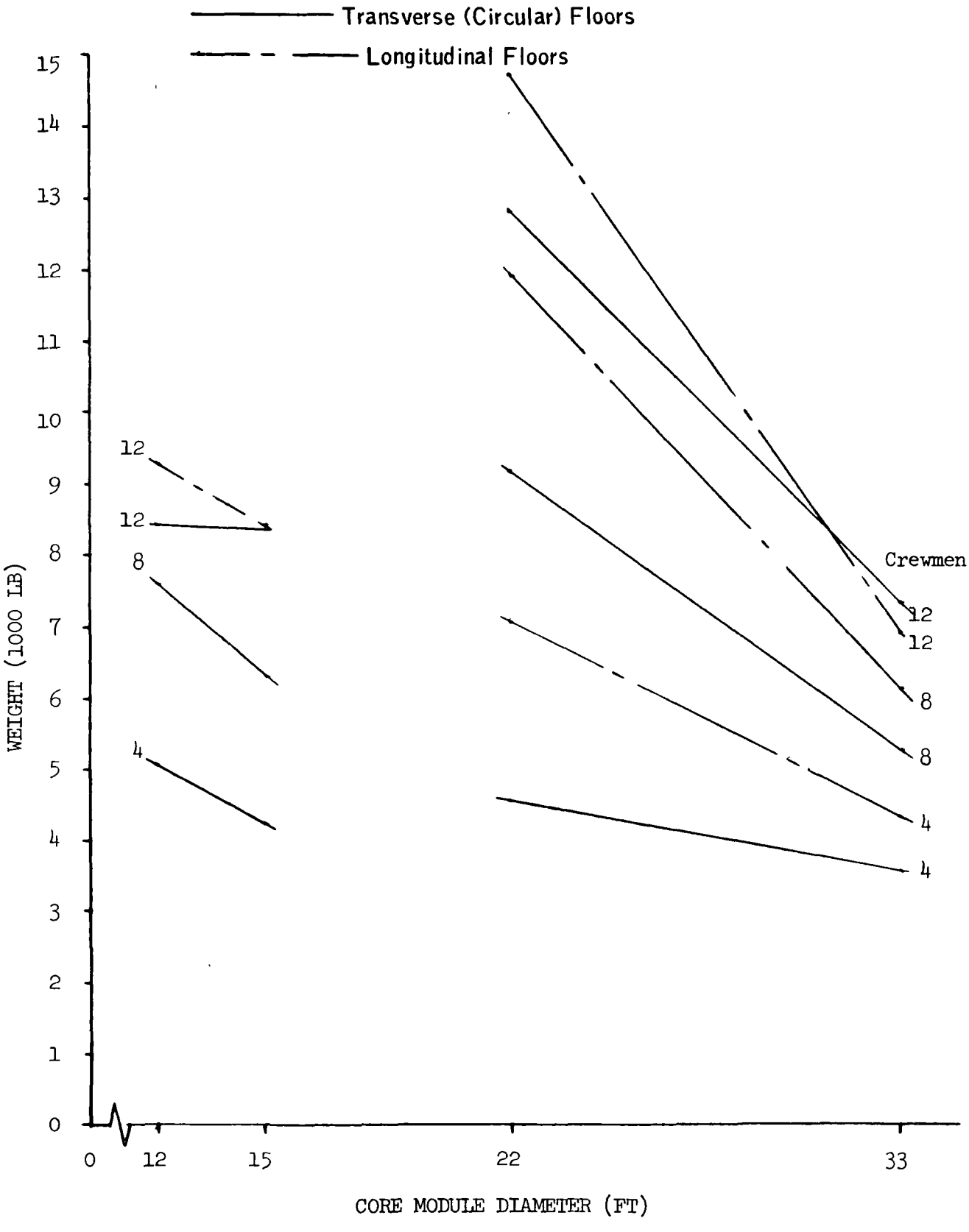


Figure 2.4-2. OLS Structural Cylindrical Shell Weight

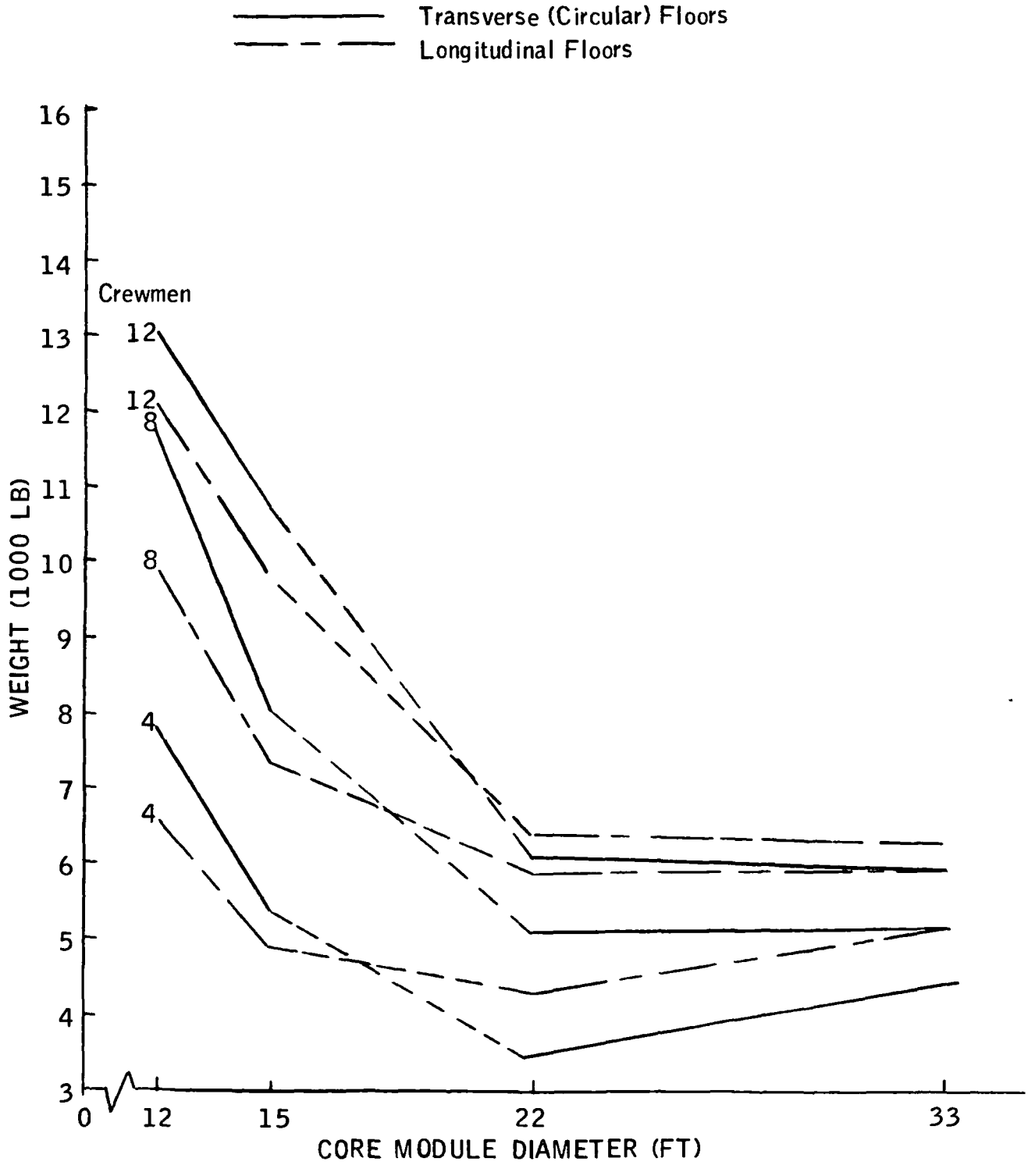


Figure 2.4-3. OLS Environmental Protection Weight

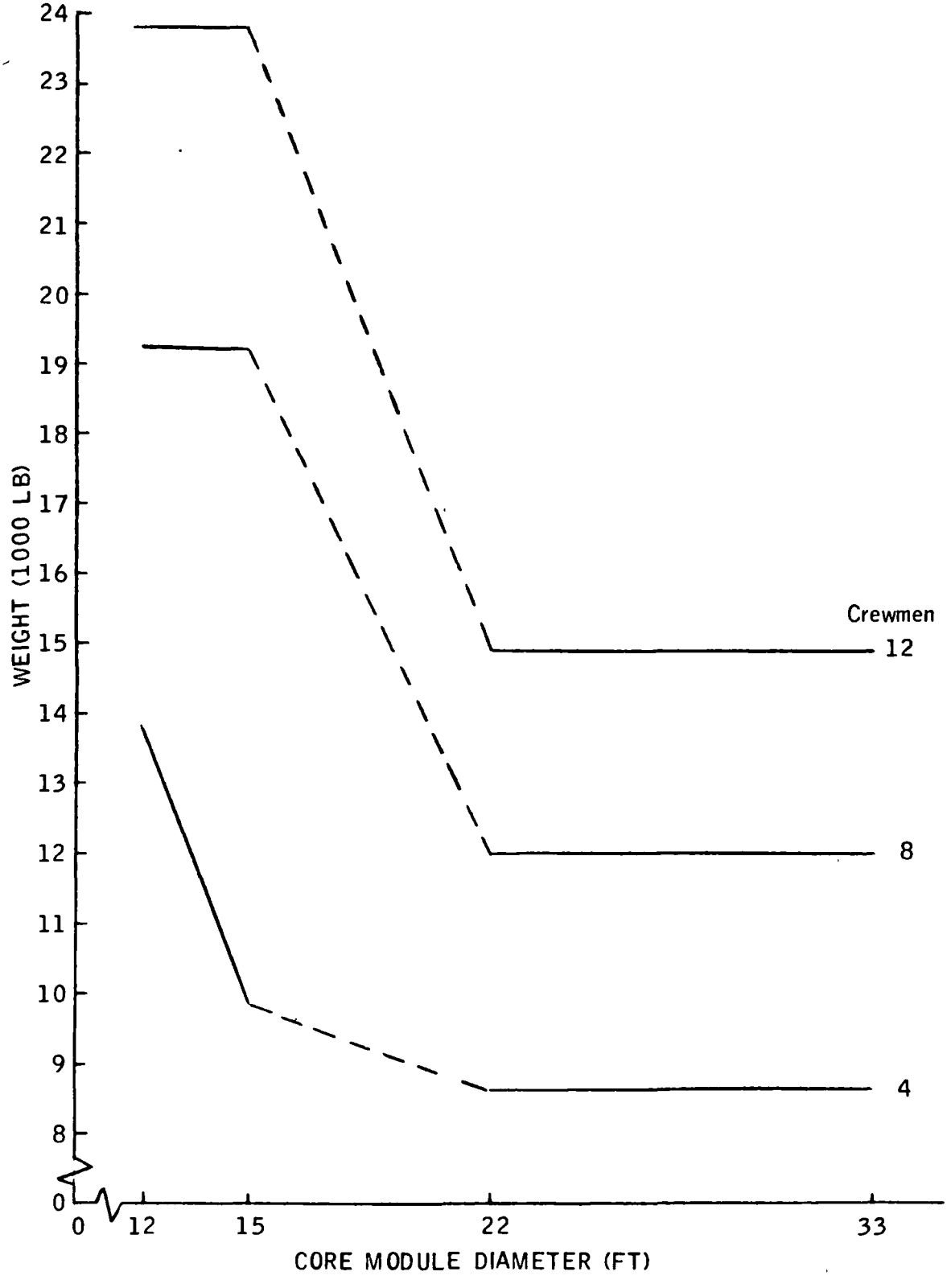


Figure 2.4-4. OLS Radiator Weight
(Tubes and Tube Meteoroid Protection)

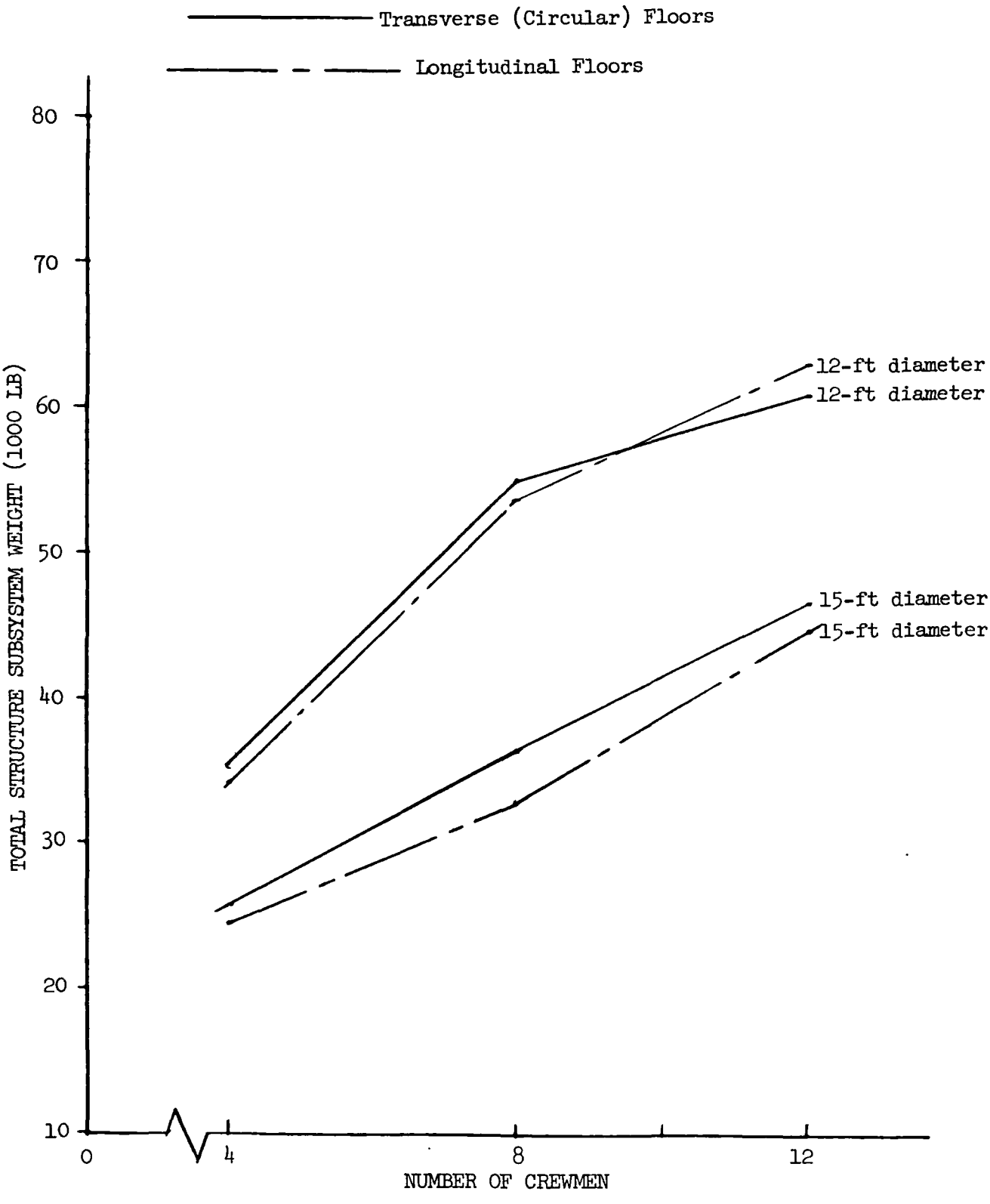


Figure 2.4-5. OLS Total Structure Subsystem Weight, 12- and 15-ft Diameter (Including Docking, Environment Protection, Radiators, Multiple Docking Adapters, and Cislunar Shuttle Adapters)

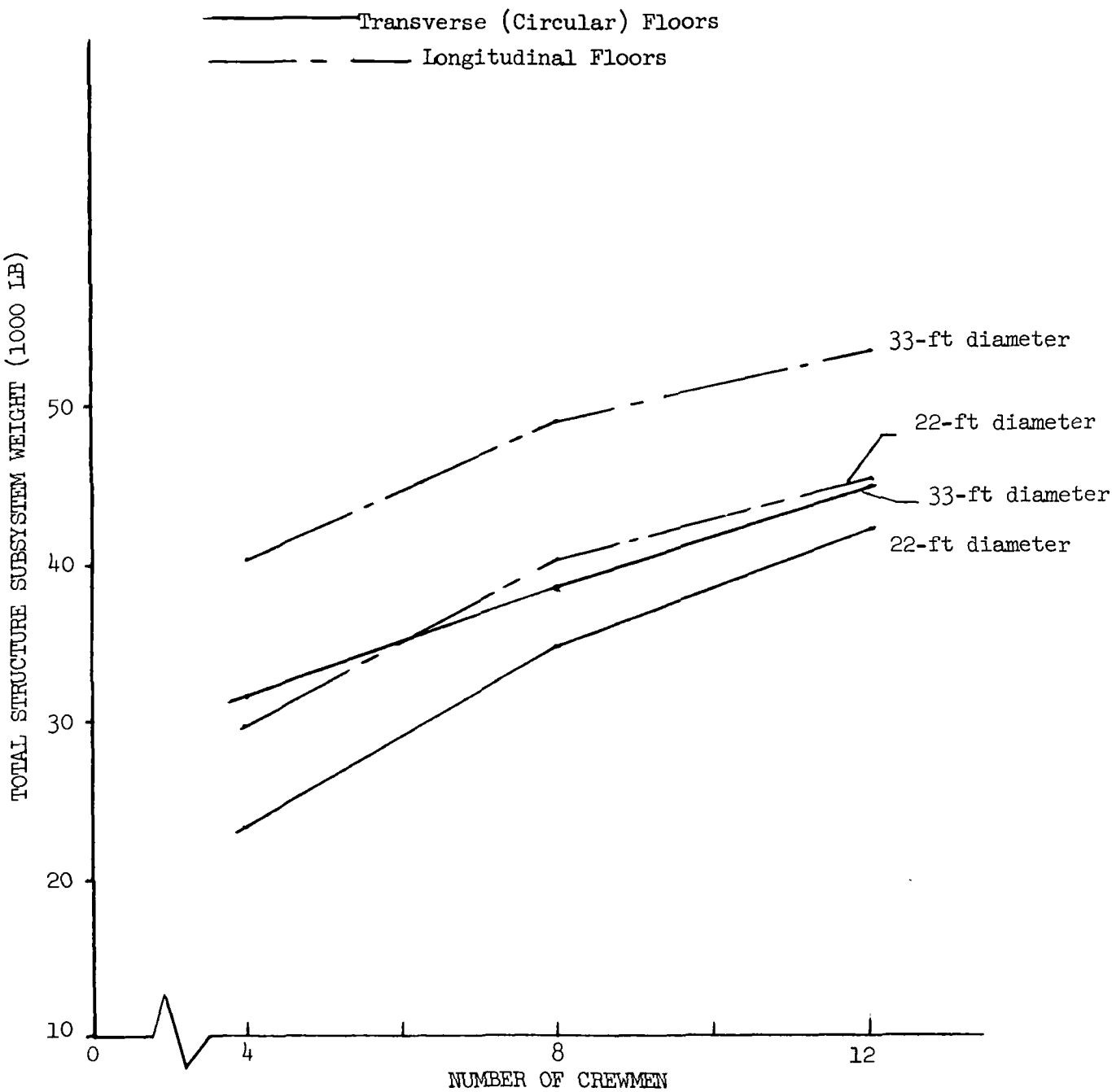


Figure 2.4-6. OLS Total Structure Subsystem Weight, 22- and 33-ft Diameter (Including Docking, Environment Protection, and Radiators)

diameter (earth launched), the 22-foot-diameter circular floor configuration is the minimum weight case. For the 12-man case, however, the 22-foot-diameter longitudinal floor and 33-foot-diameter transverse floor configurations are essentially equal in weight.

It should be noted that the weights include the multiple docking adapters required for assembly of the modular OLS and the cislunar shuttle adapter weights.

Moment of Inertia

In support of guidance and control studies, a very rough moment-of-inertia estimate was made of the arrangements shown in Figure 2.4-7. The inertias assumed 15,000 pounds for each cargo module (C), 84,615 pounds for each tug module (T), 5400 pounds for the power boom module (PM) (less solar panels), and 5000 pounds for the multiple docking adapter (MDA) in the cruciform configuration using four 15-foot-diameter modules. These inertias are summarized in Table 2.4-5.

Inertia of the solar panels were carried out separately since they must be considered in various orientations, and is summarized in Figure 2.4-8.

2.4.3 SUMMARY AND CONCLUSION

In summary, the weight data of the OLS total structure subsystem weight has been cross-plotted in Figures 2.4-9, 2.4-10, and 2.4-11 for the 4-, 8-, and 12-man crew size, respectively. These curves give the weight trend for a given crew size.

For a 4-man crew (Figure 2.4-9), minimum weight is the 22-foot diameter configuration with transverse floors. The weight trends (in descending order) are shown in the following listing.

Maximum weight	33-foot diameter longitudinal floor
	12-foot diameter transverse floor
	12-foot diameter longitudinal floor
	33-foot diameter transverse floor
	22-foot diameter longitudinal floor
	15-foot diameter transverse floor
	15-foot diameter longitudinal floor
Minimum weight	22-foot diameter transverse floor

For the 8-man crew (Figure 2.4-10), the minimum weight concept is the 15-foot diameter longitudinal floor arrangement. The following list shows the weights trends in descending order.

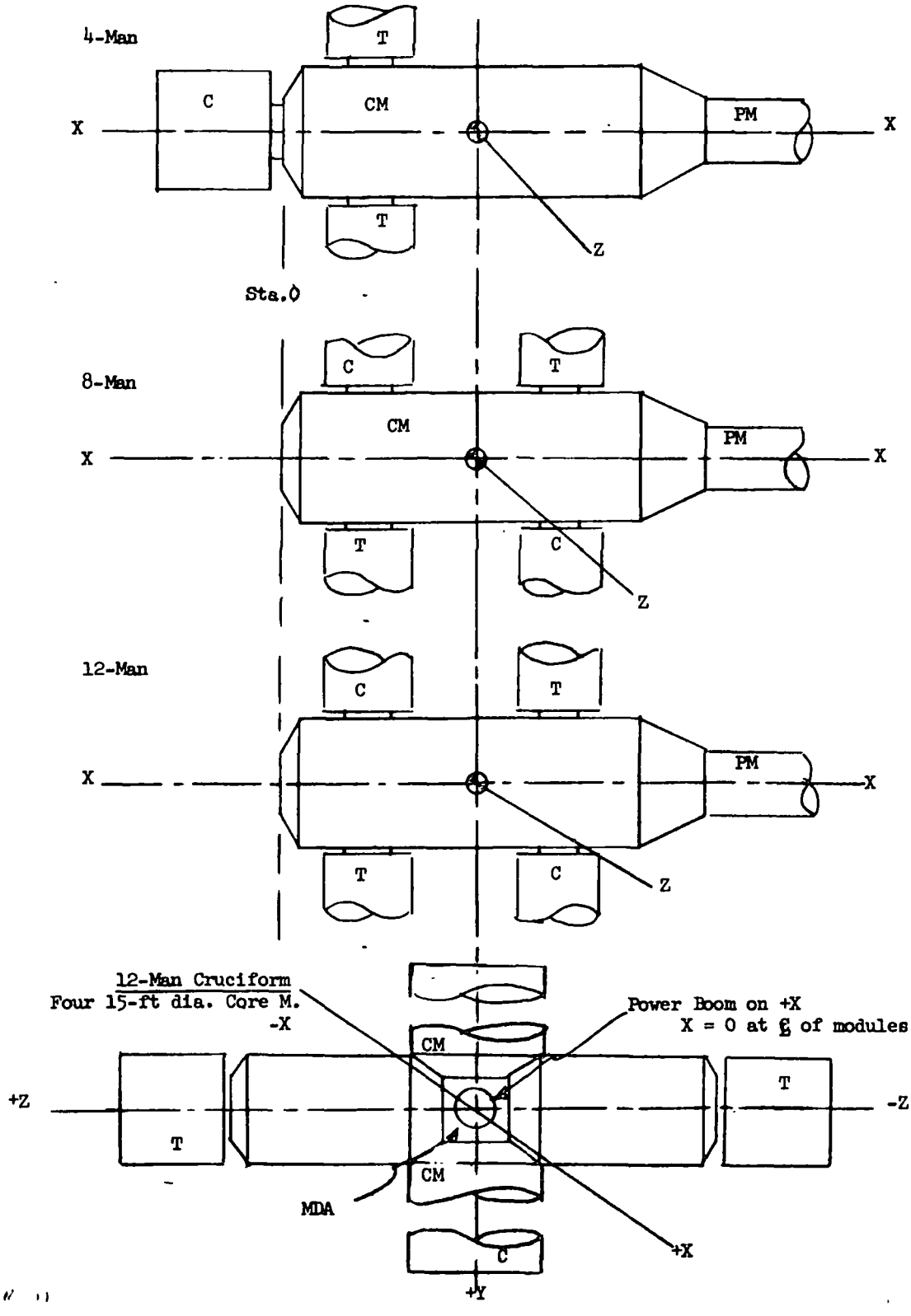


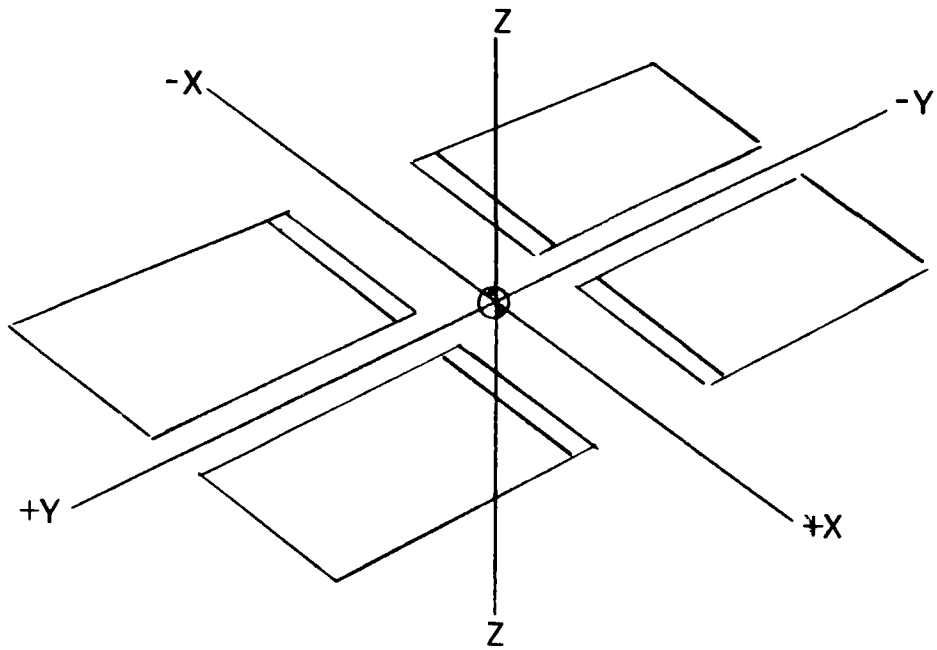
Figure 2.4-7. Vehicle Axis and Arrangement for Moment of Inertia

Table 2.4-5. OLS Estimated Mass Properties with Cargo and Tugs

	Weight* (lb)	CG Sta 0 = Aft Port			Moment of Inertia Slug ft^2 x 10^{-6}			Product of Inertia Slug ft^2 x 10^{-6}		
		x	y	z	I_x	I_y	I_z	P_{xy}	P_{xz}	Pyz
4- man										
22-ft diameter (1 cargo + 2 tugs)	237,022	151.2	0.0	0.0	9.4	1.42	10.59	0.01	0.001	0.002
33-ft diameter (1 cargo + 2 tugs)	249,630	152.2	0.0	0.0	11.96	1.1	12.71	0.01	0.002	0.002
8-man (estimated)										
22-ft diameter (1 cargo + 2 tug)	270,000	374.0	0.0	0.0	9.7	4.06	13.6	--	--	--
33-ft diameter (1 cargo + 2 tug)	279,000	240.0	0.0	0.0	12.5	1.5	13.6	--	--	--
12-man										
22-ft diameter (2 cargo + 2 tug)	300,030	479.5	0.0	0.0	10.12	6.71	16.60	4.87	0.004	0.004
33-ft diameter (2 cargo + 2 tug)	308,784	298.0	0.0	0.0	13.06	1.96	14.53	2.38	0.004	0
Cruciform (four, 15-ft diameter) (2 cargo + 2 tug)	351,150	4.9	0.0	0.0	15.76	11.09	5.43	0	0	0

*These total weights were estimated for inertia purposes only and should not be considered for any other purpose.

Space Division
North American Rockwell



Module	wt-lb	cg			Moment of Inertia slug-ft^2 x 10^6			Product of Inertia slug-ft^2		
		x	y	z	I_x	I_y	I_z	I_{xy}	I_{xz}	I_{yz}
4-Man	3600	0	0	0	0.108	0.044	0.15	-.006	0.0	0.0
8-Man	4720	0	0	0	0.36	0.058	0.42	-.005	0.0	0.0
12-Man	5572	0	0	0	0.71	0.068	0.78	-.005	0.0	0.0

	Total Area - ft^2
4-Man	7800 ft^2
8-Man	12000 ft^2
12-Man	15180 ft^2

Area, ft^2 (total fuel 20% package factor)
(Panel 0.267 lb + ft^2 + 380 lb-housing for each of four panels)

Figure 2.4-8. OLS Solar Panel Weight and Moment of Inertia

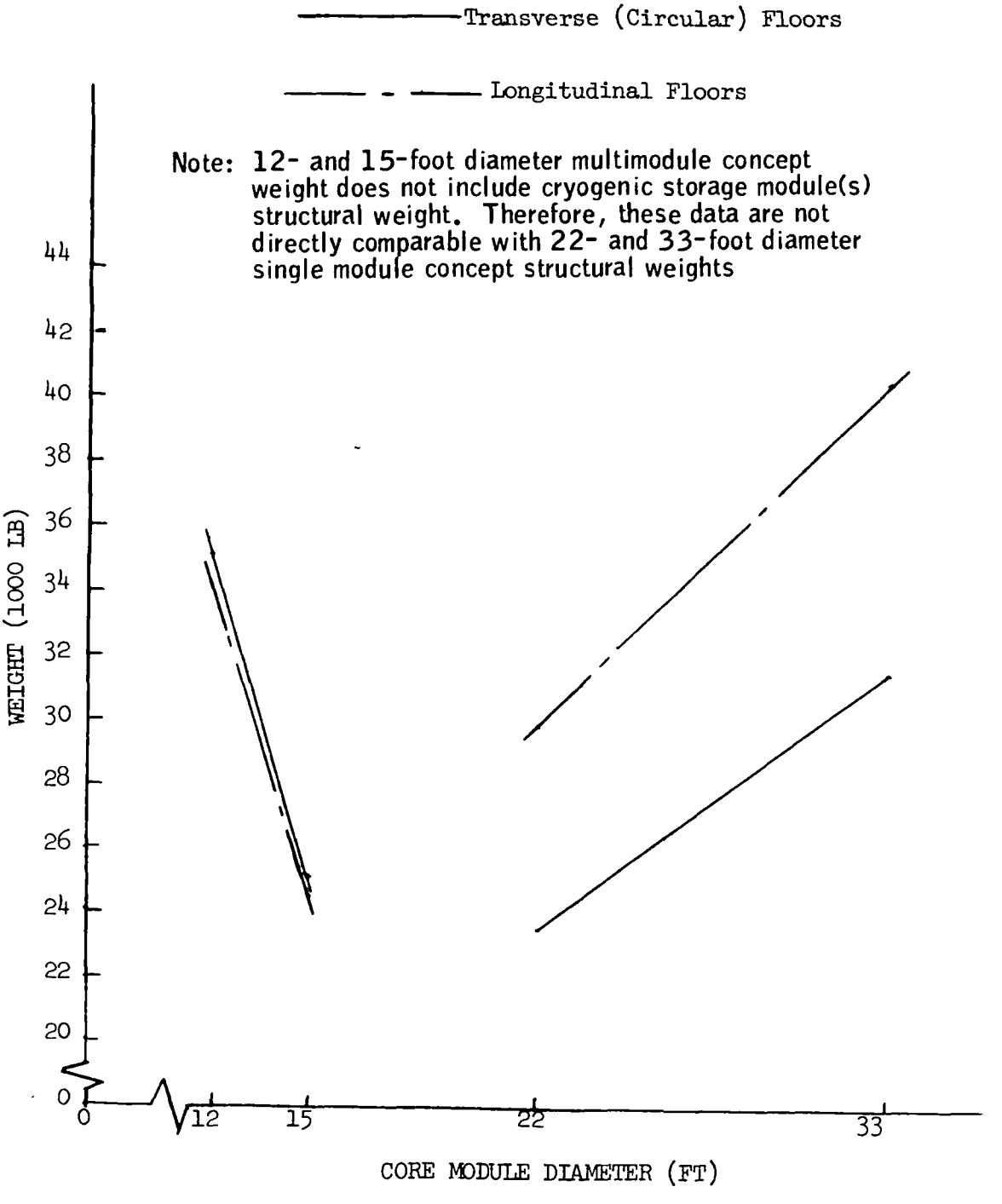


Figure 2.4-9. OLS Total Structure Subsystem Weight (4-Man Crew)

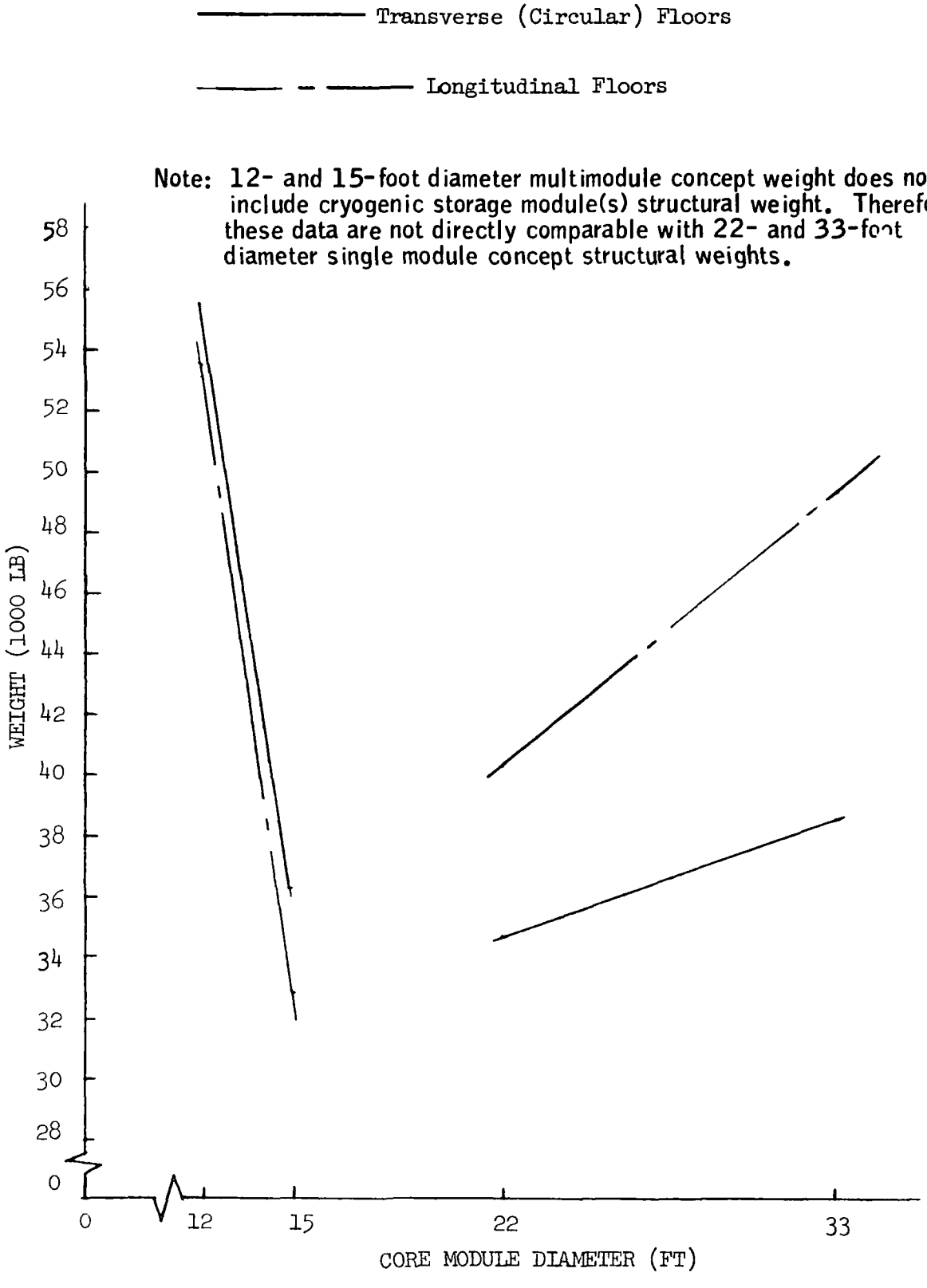


Figure 2.4-10. OLS Total Structure Subsystem Weight (8-Man Crew)

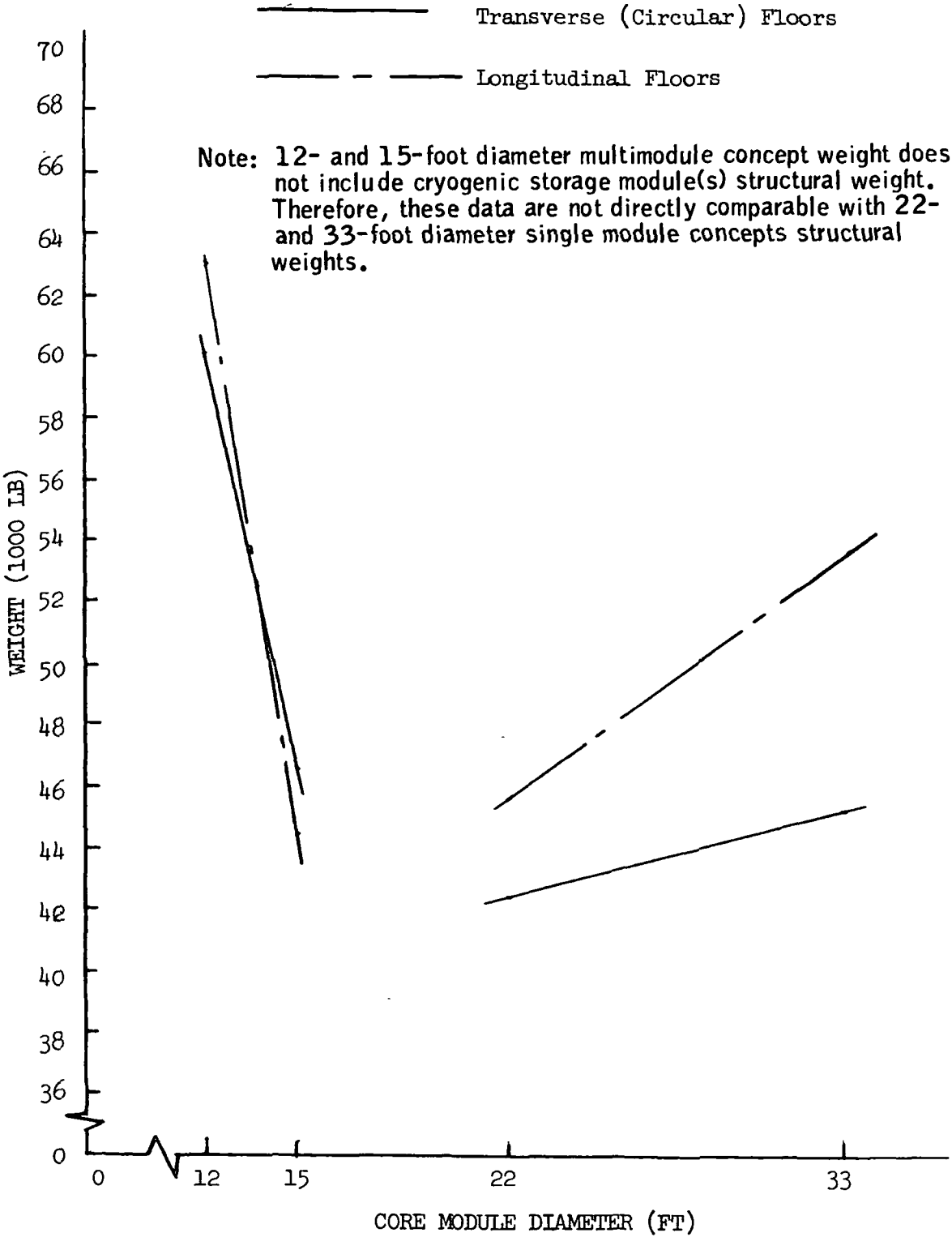


Figure 2.4-11. OLS Total Structure Subsystem Weight (12-Man Crew)

Maximum Weight:

Single Module Concepts	Multimodule Concepts
33-foot diameter longitudinal floor	12-foot diameter transverse floor
22-foot diameter longitudinal floor	12-foot diameter longitudinal floor
33-foot diameter transverse floor	15-foot diameter transverse floor
22-foot diameter transverse floor	

Minimum Weight: 15-foot diameter longitudinal floor

Figure 2.4-11 shows the total structural weight summary for the 12-man crew. A minimum weight is shown for the 22-foot diameter transverse floor concept. The following lists the weight trends in descending order of weight.

Maximum Weight:

Single Module Concepts	Multimodule Concepts
33-foot diameter longitudinal floor	12-foot diameter longitudinal floor
22-foot diameter longitudinal floor	12-foot diameter transverse floor
33-foot diameter transverse floor	15-foot diameter transverse floor
	15-foot diameter longitudinal floor

Minimum Weight:

Single Module Concepts

22-foot diameter transverse floor

Referring to Figures 2.4-9, 2.4-10, and 2.4-11, it can be seen that for all crew sizes, the 15-foot diameter modular concepts with longitudinal floors have lower total structure weight than those with transverse floors. As the ratio of structure weight to useful floor area was shown earlier to be lower for transverse floors, the above conclusion seems incongruous. However, the longitudinal floor concept does not require as much floor area or utility tank structure because the utility runs and subsystem equipment units can be located below the floors and above the false ceilings. In the larger diameter single module stations, this effect is offset by the added weight of the flat end pressure bulkheads required by longitudinal floor concepts which weigh much more than the toroidal end bulkheads employed on the transverse floor concepts.

2.5 OLS ADAPTABILITY TO ARTIFICIAL-G OPERATION

One task of the next phase of the OLS study will be to evaluate the adaptability of a representative OLS configuration to artificial-gravity operations. A set of criteria to be applied to design configurations and crew provisions has been derived and primarily based on present knowledge of physiological reactions and limitations of crew members in an artificial-gravity environment and an evaluation of artificial gravity and coriolis force components resulting from typical crew activities in the Earth Orbiting Space Station and Space Base (References 1, 2, and 3). Those criteria, which are equally applicable to the OLS, are listed in Table 2.5-1.

Figure 2.5-1 presents a sketch of a typical artificial-gravity module, defining the coordinate system and nomenclature related to direction of motion. The fundamental equations describing the acceleration components encountered in an artificial-gravity environment are given in Figures 2.5-2 and 2.5-3, along with sketches to help achieve a physical understanding of the terms. The basic artificial-gravity acceleration level and the relative magnitude of coriolis accelerations are plotted versus distance from the spin axis for two spin rates (3 rpm and 4 rpm) in Figure 2.5-4 and Figure 2.5-5. These plots show that the relative magnitudes of coriolis effects increase drastically as the spin axis is approached. However, for the range of radii expected for normal artificial-gravity living and working quarters (75 feet to approximately 140 feet), the coriolis effects are relatively small, varying from about 10 to 40 percent of the basic artificial-gravity level.

It is of interest to note that a motion in the axial direction produces no coriolis accelerations. A motion in the tangential direction results in coriolis acceleration in the radial direction. If the direction of motion is in the direction of spin, the two acceleration components add and the apparent weight is increased. If the tangential motion is counter to the spin direction, the opposite effect is true. This could result in a crewman "walking off the floor" if the coriolis acceleration ($2\omega R\,\dot{\theta}$) approaches the basic artificial-gravity level ($R\omega^2$).

Radial motions, such as a man ascending or descending a ladder or stairway, will produce coriolis accelerations in the tangential direction. If the direction of motion is toward the spin axis, the tangential coriolis forces is in the direction of rotation, and the force direction is reversed for motion in the opposite direction. This relationship leads to a basic design guideline: radial transport systems such as ladders or stairs should be oriented in the X-Z plane, and the direction of motion should be controlled so that coriolis effects tend to push a man against the device rather than away from it.

An interesting relationship between artificial gravity and coriolis forces is depicted in Figure 2.5-6. For the case of an OLS module with flat floors, the centrifugal force vector will be inclined to the floor at

Table 2.5-1. Artificial-Gravity Criteria

Criteria	Rationale
Under artificial gravity, normally occupied compartments shall have acceleration levels of not less than 0.2 g nor more than 1.0 g. Design spin radius and rotational rates should correspond to the latest physiological data on vestibular effects.*	Undesirable physical and physiological effects of artificial-g operations should be minimized to obtain maximum habitability of the station. The given criteria include the most significant aspects of these considerations (this is one of the guidelines listed in the OLS statement of work.
Flat floors will be employed in the artificial-gravity modules, which are normal to the radius of rotation at the centerline of the module.	Floors curved so as to be normal to the radius of rotation at all points would have radii varying from approximately 70 to 140 feet. Tooling and fabrication costs for this type of structure would be very high. Space station modules would also lose capability for universal application (artificial gravity or zero gravity location). The curved floors and the equipment mounted normal to the floors could contribute to confusing visual stimuli for the crewmen. It is believed this would be more distressing than the inclination of the centrifugal force vector associated with flat floors.
Space station design will minimize undesirable artificial-gravity and coriolis effects by orienting compartments so that the predominant direction of traffic is parrallel to the axis of rotation and by minimizing requirements for crew occupancy or activity in regions where maximum inclination (with respect to floors) of the centrifugal force occurs. However, special compartmentation to control direction of crew movement in the artificial-gravity environment is not mandatory.	Undesirable physical and physiological effects of artificial-gravity operations should be minimized to obtain maximum habitability of the station. The given criteria include the most significant aspects of these considerations. However, the coriolis forces resulting from normal crew motions are relatively small and should not produce dangerous or overly tiring situations. Therefore, special compartmentation to rigorously control the direction of crew motions is not considered mandatory.
Detailed design considerations: 1. Crew duty station positions shall be oriented so that during normal activity, the lateral axis through the crewman's ears is parallel to the axis of rotation (i.e., crewman facing in or counter to the direction of rotation) and his long-body axis (when standing or sitting) is perpendicular to the axis of rotation. 2. Work console instruments, displays, controls, etc., shall be designed so that left-right head rotations and up-down arm motions are minimized. Side-to-side head movements shall not exceed 45 degrees out of the plane of rotation (90 degrees total). Frequently viewed or critical displays shall be located within a 30-degree lateral field of view (15 degrees out of the plane of rotation) when a crewman's head is in the neutral position (in the plane of rotation). 3. Equipment and associated operational tasks shall be designed to minimize rapid, large-scale, or precise movements.	Undesirable physical and physiological effects of artificial-gravity operations should be minimized to obtain maximum habitability of the space station where artificial gravity is expected to be the normal living environment.
*In no case should the rotational rate be greater than 4 rpm	

Table 2.5-1. Artificial-Gravity Criteria (Continued)

Criteria	Rationale
4. The presence of potentially confusing visual stimuli shall be minimized. For example, observation ports or windows should be covered when not in use. 5. Space station radial traffic (toward or away from the axis of rotation) shall be kept to a minimum. For crew movement between levels, the long axes of rungs or steps shall be parallel to the axis of rotation. Crew movement toward or away from this spin axis shall be made on the side of the ladder (or passageway) that opposes the coriolis inertial reaction. Transport across the spin axis and crew activity at the spin axis shall be held to a minimum. 6. Provide crew locomotion aids or traverse mechanisms (handholds, rails, or other mobility aids, etc.) as well as placarding in aisles or passageways and other areas where adverse coriolis effects may be experienced, especially at and approaching the spin axis. 7. All critical switches and controls shall be guarded to prevent inadvertent actuation. Horizontal handrests below each row of such controls are desirable. 8. The centrifugal force vector shall not be inclined at an angle greater than 20 degrees with the normal to floors in any inhabited region of the space station artificial-gravity assembly.	Avoid excessive physical strain on crewmen to maintain balance. Obtain maximum habitability of the space station by minimizing undesirable artificial gravity and coriolis effects, including vestibular stimulation, occulogyral and oculogravic illusions, visual stimulation, and coriolis force components.
Basic requirements: 1. Mass balancing of the artificial-gravity assembly and structural coupling to the zero-gravity assembly shall be accomplished so that the maximum acceleration on experiment modules in the zero-gravity assembly does not exceed 1×10^{-3}g because of artificial-gravity operation. 2. System capability for spinup or despin at any time during the operational life of the space station shall be maintained.	Provide artificial gravity as a continuing environment in crew living and recreation quarters and in some work stations to achieve the following major benefits: 1. Reduce psychological difficulties of crew adaptation from earth to space environment. 2. Prevent physiological deterioration of crewmen. 3. Provide assistance in performance of some service operations and experiment activities (e.g., pouring fluids, weight determination of labaratory samples by mass balance, etc.) Maintain an environment acceptable to the majority of planned zero-gravity experimental operations (additional isolation or decoupling may be necessary in some cases) within the practical limits of mass balancing capability. Provide for temporary termination of artificial-gravity operations to allow for space station maintenance, modification, and buildup or because of emergency conditions.

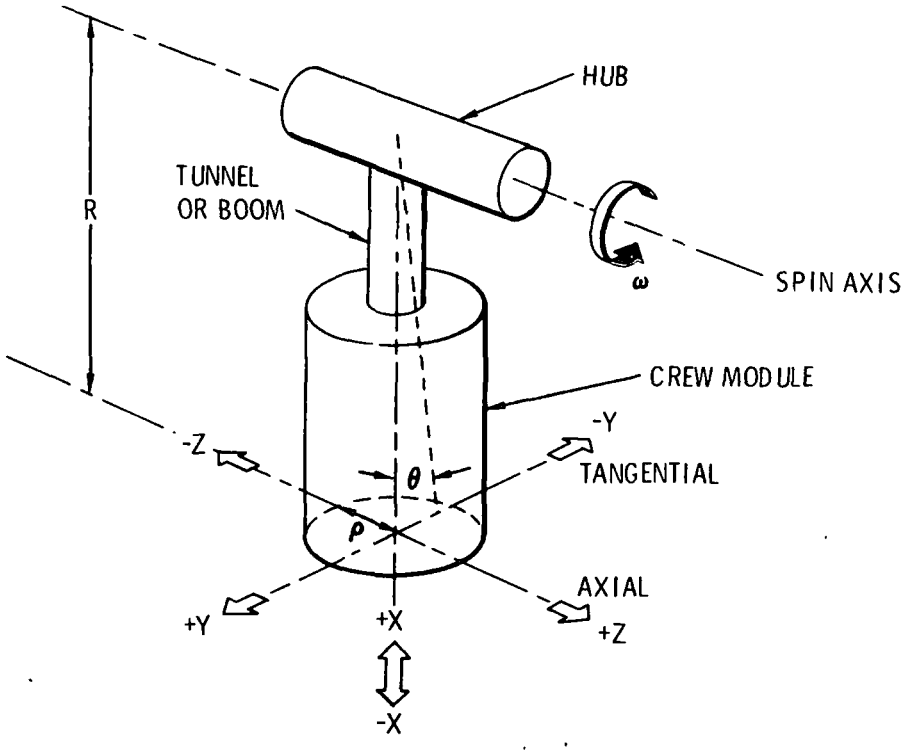


Figure 2.5-1. Coordinate System Orientation

RADIAL ACCELERATION	=	CENTRIPETAL ACCELERATION DUE TO ROTATION OF THE SYSTEM (ARTIFICAL G)	+	RELATIVE ACCELERATION DUE TO TANGENTIAL MOTION OF M	+	CORIOLIS ACCELERATION DUE TO TANGENTIAL MOTION OF M	+	RADIAL ACCELERATION OF M (INITIATION OF MOTION)
a_R	=	$R\omega^2$	+	$R\dot{\theta}^2$	+	$2\omega R\dot{\theta}$	+	$\ddot{R}$

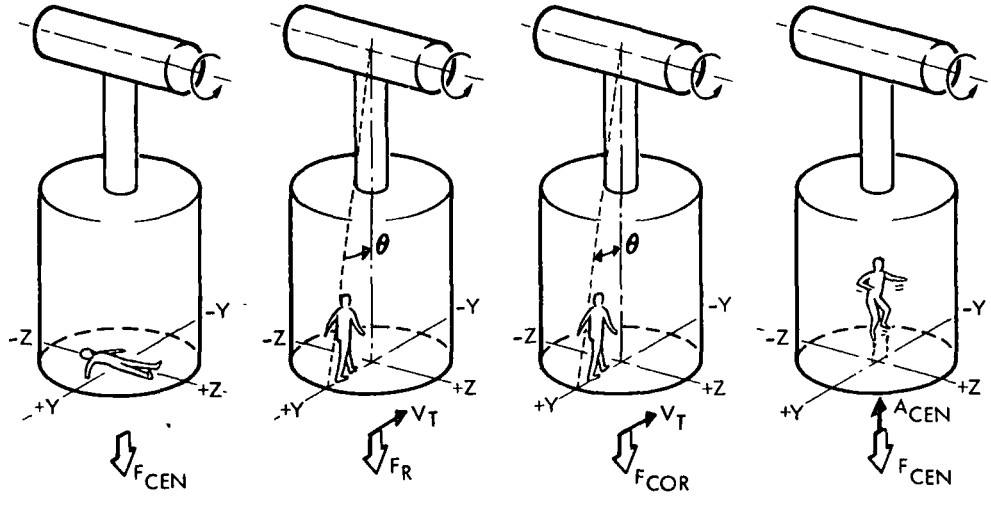


Figure 2.5-2. Radial Acceleration

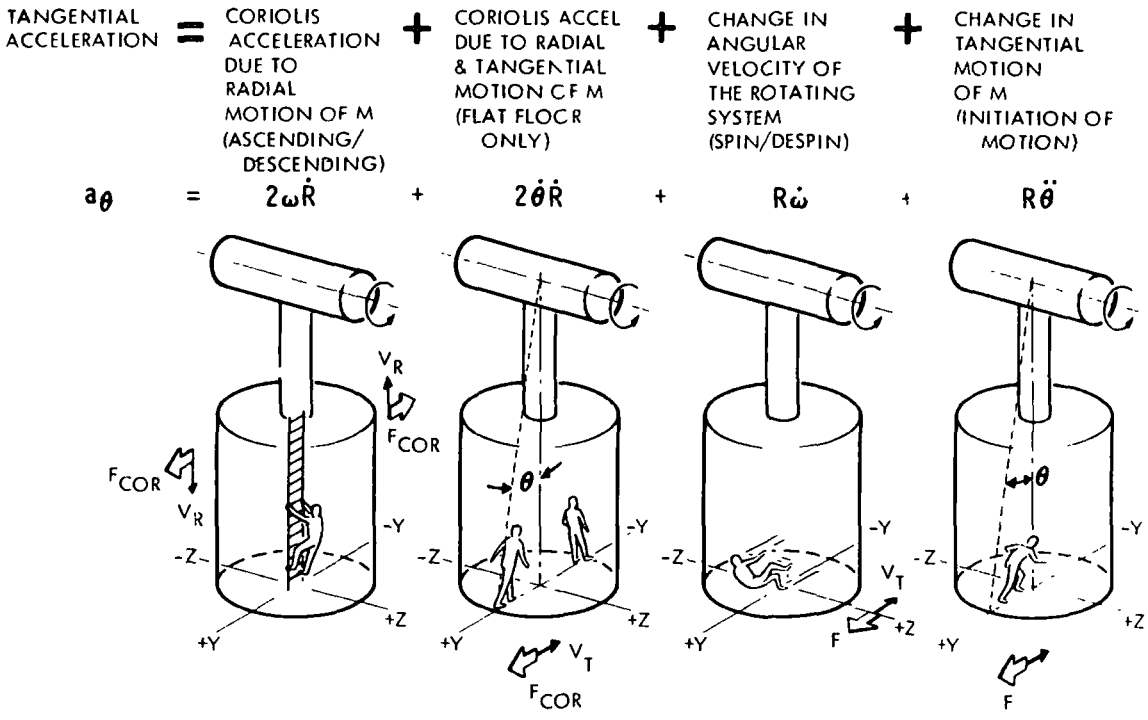


Figure 2.5-3. Tangential Acceleration

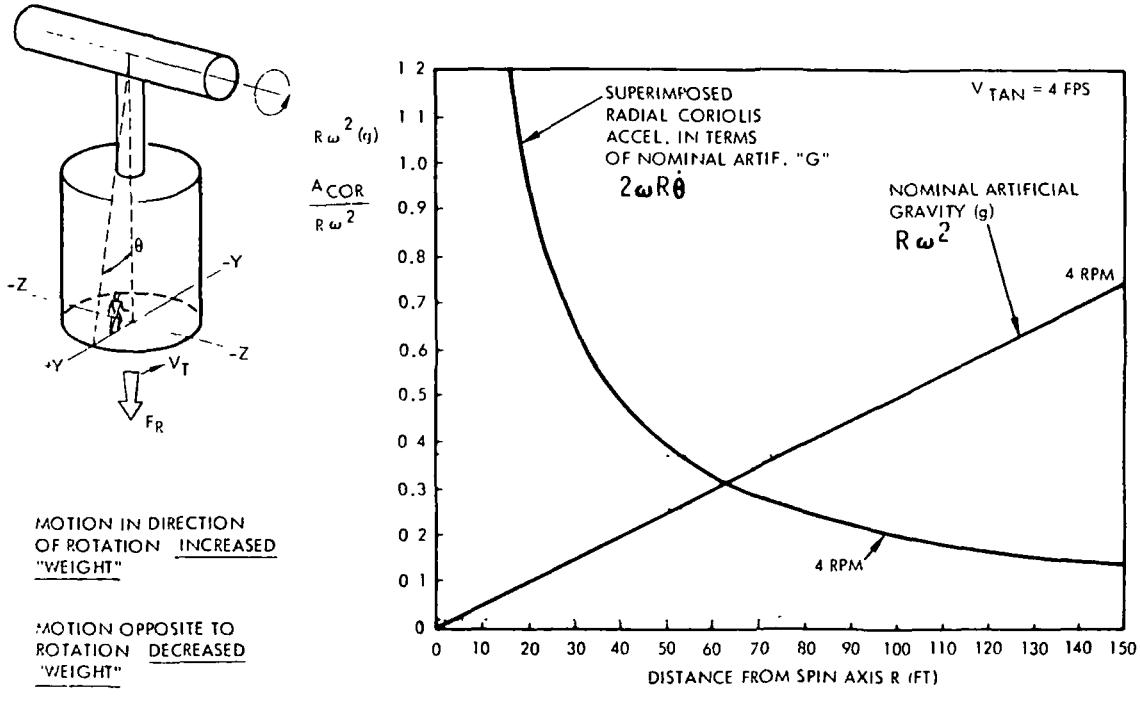


Figure 2.5-4. Radial Coriolis From Tangential Motion

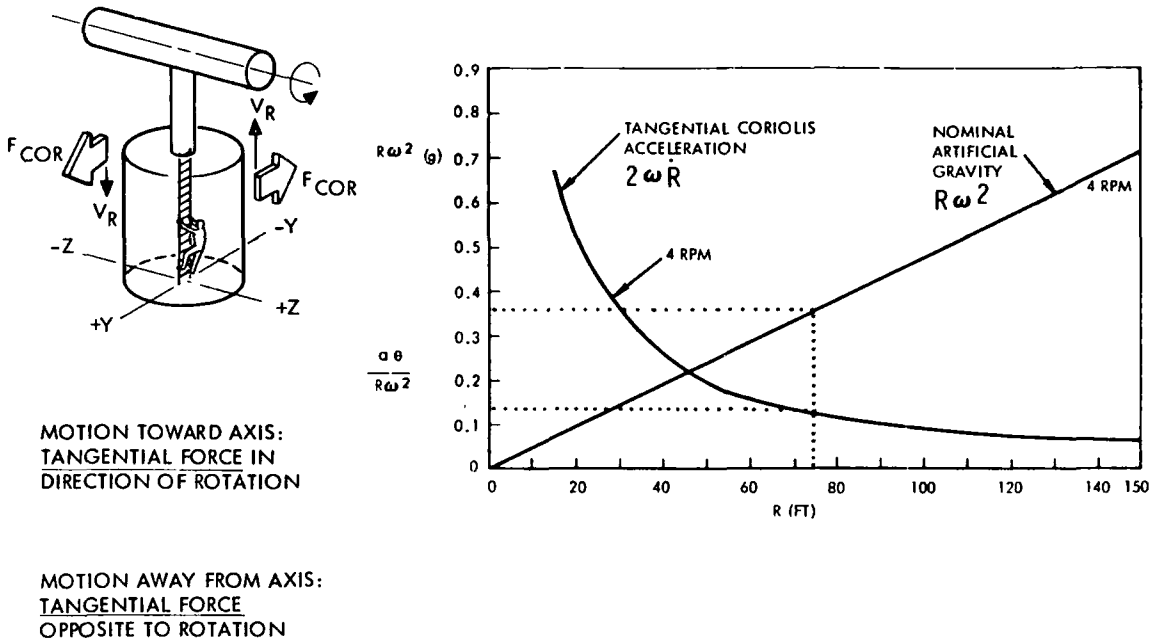


Figure 2.5-5. Tangential Coriolis From Radial Motion

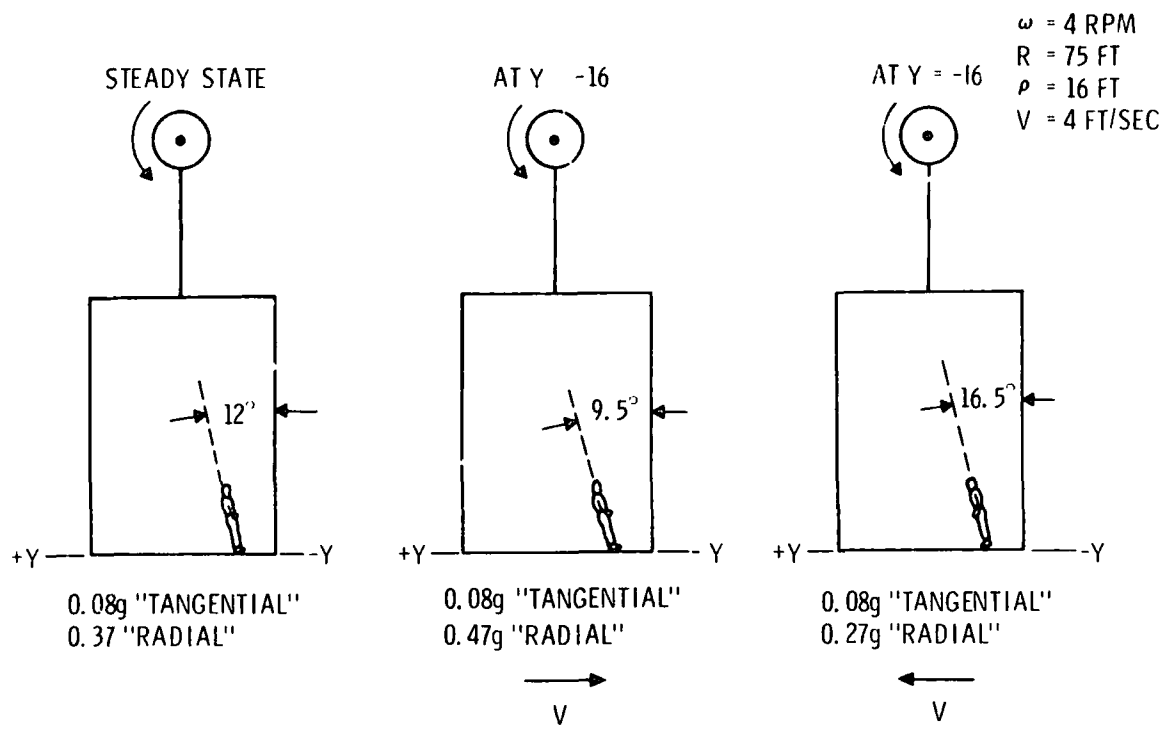


Figure 2.5-6. Flat Floor - Angular Relationship

locations in the Y-Y direction away from the centerline of the module. For the case of a crewman at rest at the maximum possible distance from the centerline of the module (approximately 16 feet), the inclination of the centrifugal force vector is 12 degrees (assuming a distance to the spin axis of 75 feet). If the crewman is moving in a tangential direction at a rate of 4 feet per second, radial coriolis forces will combine with artificial-gravity centrifugal force, and the resulting inclination of the resultant force vector would change to 9.5 or 16.5 degrees, depending upon the direction of motion. However, the forces and "lean-angles" necessary to maintain balance are relatively small, and a man would easily and quickly adapt to them, probably with no conscious effort.

A summary of artificial-gravity and coriolis force components for typical conditions and various crewman motions is given in Table 2.5-2.

A synthesized block diagram of the human "guidance" system is given in Figure 2.5-7, and a simplified description of human sensors is shown in Figure 2.5-8. Some of the illusory effects resulting from the stimulus of an artificial-gravity environment on these sensors are summarized in Figures 2.5-9 and 2.5-10. These physiological reactions provide the basis for design guidelines to eliminate or minimize undesirable effects of an artificial-gravity environment. Two examples of these guidelines, which are followed in the detailed definition of OLS configurations, are given in Figures 2.5-11 and 2.5-12.

Table 2.5-2. Coriolis Effects for Typical Crew Activities in an Artificial-Gravity Environment

Artificial Gravity Parameters
Rotation rate = 3.5 rpm
Distance to spin axis = 75 feet
Nominal artificial g acceleration = 0.31 g
Flat floors in artificial g module

Activity	Coriolis Effect		
	Acceleration (g)	Force[1] (lb)	Force Direction
Walking in direction parallel to spin axis	0	0	--
Walking in spin direction near X-X[2] axis (4 ft/sec)	-0.09	-15	Radial - away from spin axis
Walking counter to spin direction near X-X axis (4 ft/sec)	0.09	15	Radial - toward spin axis
Walking in tangential direction remote from X-X axis (Y = 16 ft, V_{tan} = 4 ft/sec)	±0.09 ±0.022(max)	±15 ±3.6	Radial Tangential[3]
Ascending ladder or stairs (toward spin axis) at 2 ft/sec	0.045	7.5	Tangential - in spin direction
Descending ladder or stairs (away from spin axis) at 2 ft/sec	-0.045	-7.5	Tangential - opposite to spin direction
Rapid hand motions (20 ft/sec) Axial direction	0	0	--
Tangential direction	±0.45	--	Radial
Radial direction	±0.45	--	Tangential

[1] Based on 165-lb crewman

[2] Refer to Figure 2.5-1 for coordinate system

[3] Due to radial component of velocity associated with flat floors

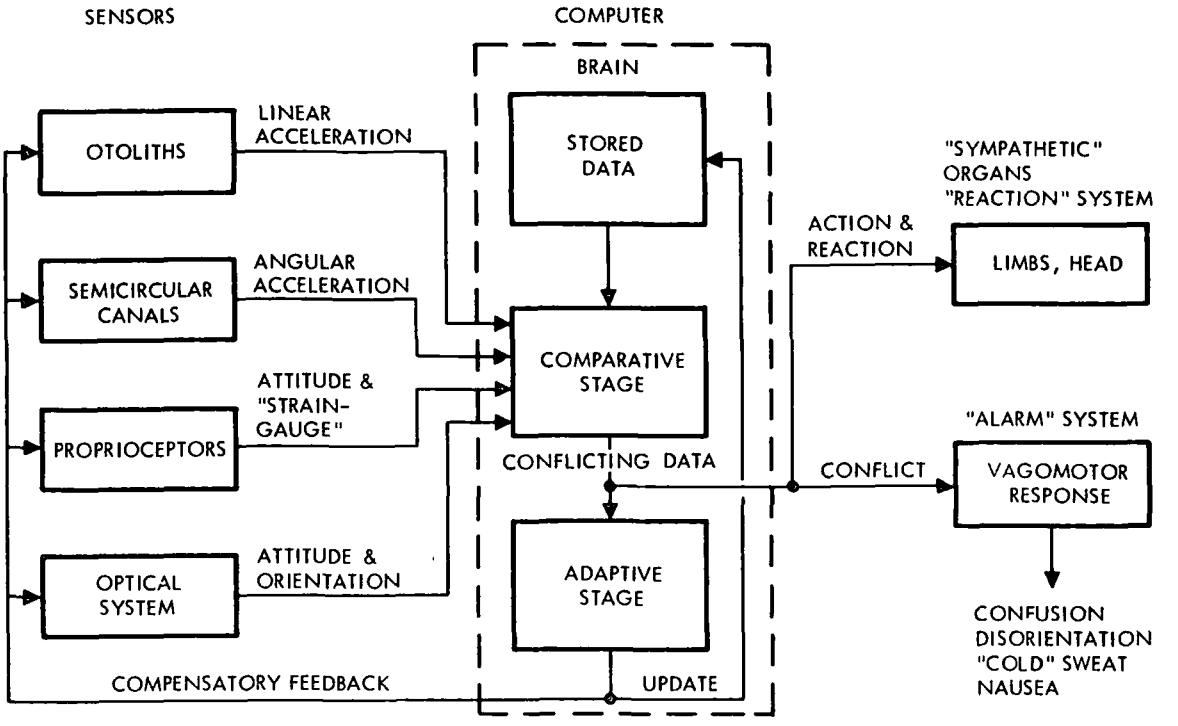


Figure 2.5-7. Functional Diagram of Human "Guidance" System

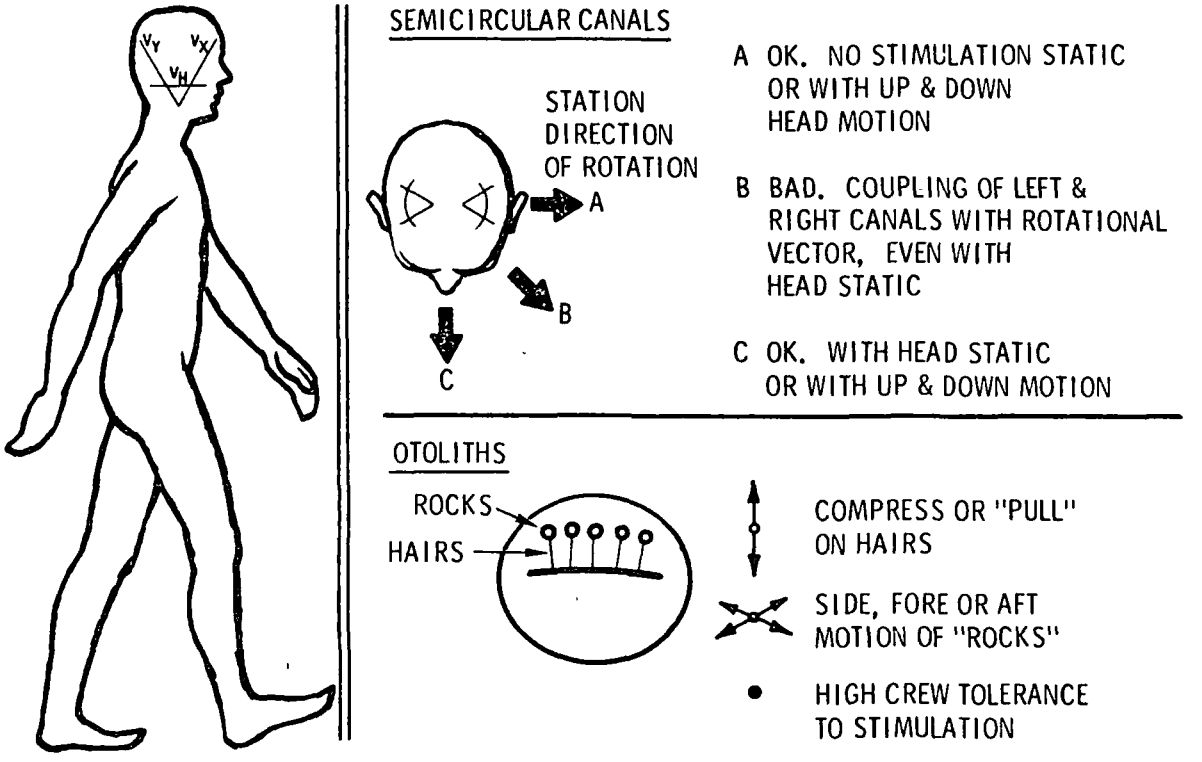


Figure 2.5-8. Vestibular Sensors - Typical Stimuli

- ANGULAR ACCELERATIONS SENSED BY SEMICIRCULAR CANALS
- APPARENT MOTION OF STATIONARY OBJECTS DUE TO RELATIVELY SMALL MOVEMENTS OF CREWMAN'S HEAD (IN DIRECTION OF SYSTEM ROTATION)
- EXCESSIVE RATES OF ANGULAR ROTATION OF HEAD OR BODY MAY RESULT IN TRANSIENT NYSTAGMUS ("HUNTING" OF EYE-BALLS)

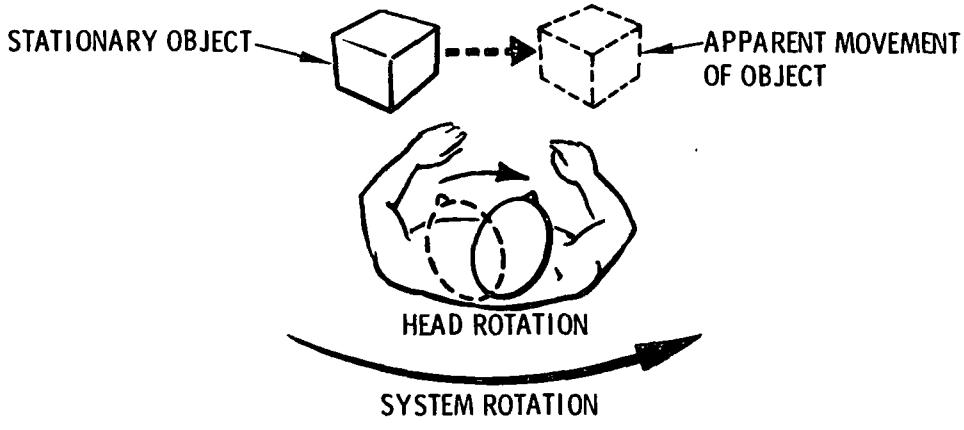


Figure 2.5-9. Oculo-Gyral Illusion

- INDUCED BY LINEAR ACCELERATIONS - SENSED BY OTOLITHS
- WELL TOLERATED
- ADAPTATION TO FLAT FLOOR OCCURS IN A FEW HOURS TO DAYS (VARIABLE WITH INDIVIDUAL)
- NEVER ADAPT TO CURVED FLOOR (UNDESIRABLE VISUAL STIMULI)

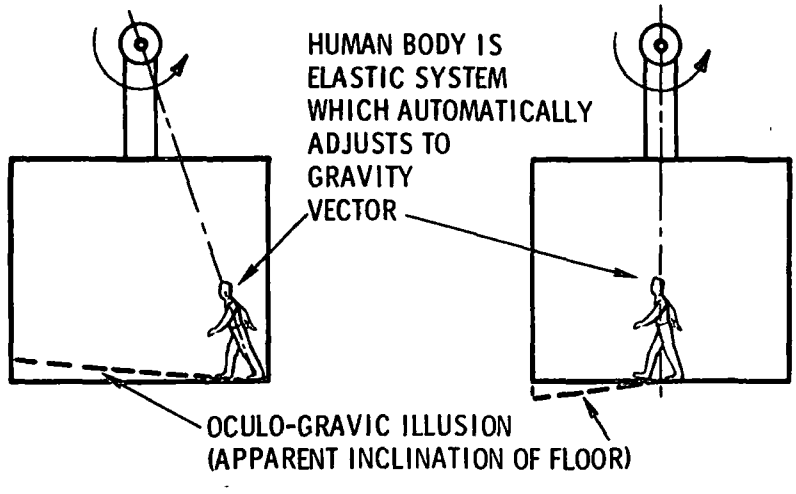


Figure 2.5-10. Oculo-Gravic Illusion

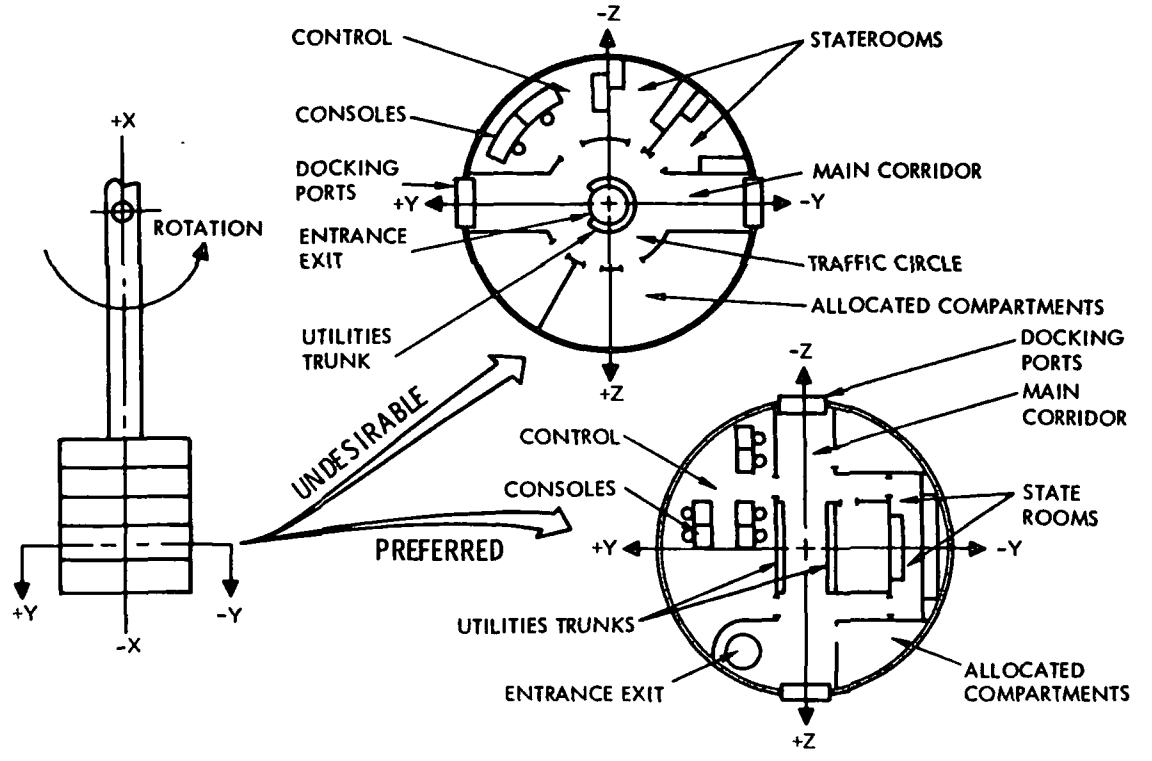


Figure 2.5-11. Compartment-Equipment-Console-Corridor Orientation for Artificial Gravity Space Station

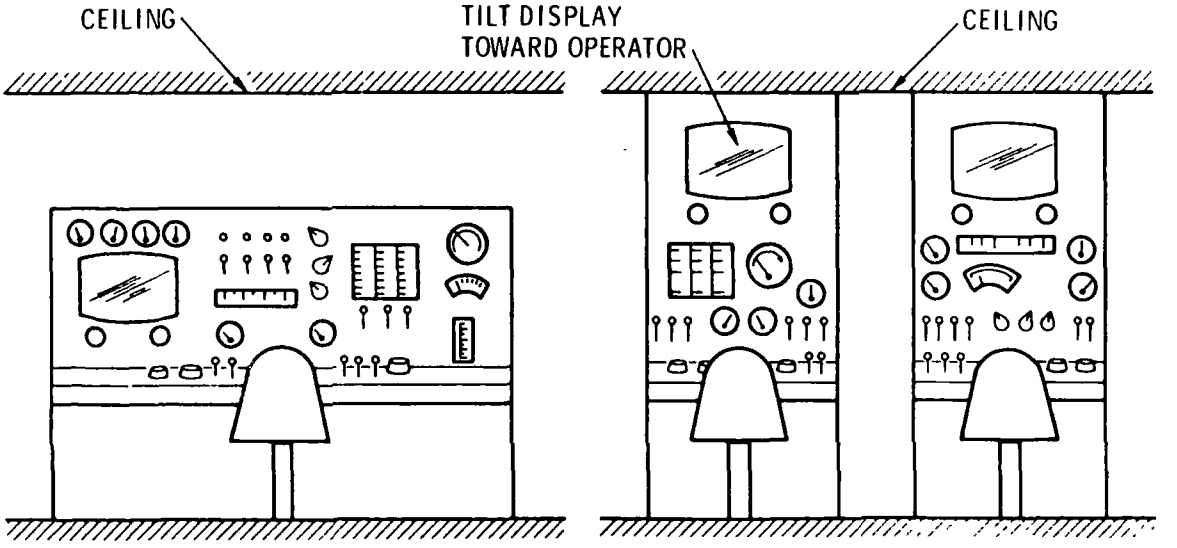


Figure 2.5-12. Console Configuration for Artificial Gravity Space Station

REFERENCES

1. Space Station Program - Phase B Definition, Special Emphasis Tasks Summary Report, SD 69-611 (12 December 1969).

2. Space Station Program - Phase B Definition, Special Emphasis Tasks Summary Report, SD 70-145 (13 March 1970).

3. Space Station Program - Phase B Definition, Special Emphasis Tasks Summary Report, SD 70-531 (July, 1970).

3.0 SUBSYSTEM ANALYSIS

This section presents the trades, parametric data, and comparison matrices developed in the analysis of candidate subsystem concepts for the OLS. The prime prerequisite for a concept to become a candidate was that the concept satisfy the functional and performance requirements defined in Volume III, OLS Performance Requirements.

Where appropriate, data are presented parametrically as a function of crew size (4 to 12 men) and OLS module diameter (12 to 33 feet). This section is divided into six subsections corresponding to the following subsystem:

3.1 Environmental control/life support (ECLSS)
3.2 Electrical power (EPS)
3.3 Information management (ISS)
3.4 Guidance and control (G&C)
3.5 Reaction control (RCS)
3.6 Environmental protection (ENPS)

The analysis approach and data format for subsystem trades and subtrades vary as a function of the sensitivity to crew size and OLS configuration and the applicability of EOSS data.

The ECLSS is directly dependent upon consumables, and, therefore, almost all data are presented parametrically as a function of crew size.

Although total power requirements are a function of crew size, the primary trades of the EPS are relatively independent of crew size over the range of interest for the OLS. Therefore, EPS data consist of both parametric and comparison matrices of basic power system concepts that are essentially independent of the OLS configuration.

OLS and EOSS ISS functional and performance requirements are quite similar. Therefore, EOSS data were used extensively in the ISS subsection. The primary analysis was the evaluation of OLS external communication links.

Vehicle configuration is a significant parameter in the tradeoffs of the torquer of the G&C. Tabular comparisons of various configurations and flight modes are presented. The remaining elements of the G&C are basically common with those used for an EOSS. Trade data developed in the EOSS study for these elements therefore were presented for completeness.

RCS concept selection is based upon the OLS configuration, flight mode and the RCS interfaces with EPS, ECLSS, and G&C. Thus, RCS data are presented parametrically as a function of impulse requirements and qualitatively as a function of potential interfaces. In addition, resupply concepts for the viable propellant concepts are discussed.

Performance requirements for ENPS are derived and presented parametrically. Included are descriptions of the OLS thermal, meteoroid, and nuclear radiation environments together with concepts for protection of the OLS from these environments.

When applicable subsystem concept trade trees are presented. Also presented are comparison data (weight, power, volume, technical development risk, and cost) for the options defined. In some cases EOSS-generated trade data were reprinted (for completeness) in this document as OLS trade data. All data are presented objectively. Neither a preferred nor a selected concept is identified.

3.1 ENVIRONMENTAL CONTROL/LIFE SUPPORT SUBSYSTEM

3.1.1 INTRODUCTION AND REQUIREMENTS

The environmental control life support system (ECLSS) provides the functions listed in Table 3.1-1. These functions also are ECLSS hardware assemblies. This report contains parametric data on each assembly so that a typical OLS ECLSS can be described and defined in terms of weight, power, volume, development time, and complexity. The parametric curves give representative data based upon nominal operating conditions. To fix crew size, the metabolic rate was held constant and 24-hours-per day operation was assumed for the process hardware. The curves can be ratioed by hours for different operational time periods. The discussions presented below for each parametric curve note the limitations of the curve in question.

The general requirements utilized for this study are listed in Table 3.1-2. Deviations from these requirements and assumptions that may occur because of the parametric nature of this study are given in the text or as notes to the specific curves.

This report is broken into several sections; each ECLSS function is discussed in a separate section and other separate sections are included that discuss the total system.

3.1.2 ATMOSPHERIC STORAGE

This function involves repressurization of the vehicle, leakage makeup, and pumpdown systems for airlocks. CO_2 management impacts this function because metabolic oxygen must be stored if it is not recovered. Also, if a CO_2 management water electrolysis unit is used, leakage oxygen may be made up by electrolysis of excess water from H_2O management and is not stored If a Sabatier unit is chosen for CO_2 reduction in CO_2 management, then hydrogen may be stored by the atmospheric storage assembly. RCS and EPS also impact this function because they may use or store hydrogen and oxygen.

Repressurization

Gases for repressurization are normally not utilized if repressurization of the vehicle is not scheduled. OLS requirements do not specify any scheduled vehicle repressizations (excluding airlocks which are discussed under systems later in this section). Therefore, the gas would probably be stored in 3000-psia tanks. If cryogenic tanks are used, gas will steadily be lost through boil-off. Only if repressurization is considered as a potential secondary performance parameter by utilizing existing gas stores (as oxygen for RCS, EPS, and ECLSS, and cryogenic leakage makeup N_2) is cryogenic storage of repressurization gas considered feasible. However, with this concept as the gases are used for their primary purpose, the amount of gas remaining for

Table 3.1-1. ECLSS Functions and Description

Function/Subfunction	Description of Function
Atmospheric storage Repressurization Gas makeup Pumpdown	Provides for storage of atmospheric gases as well as for pumpdown and repressurization of vehicle or airlock
CO_2 Management CO2 removal CO2 reduction O_2 generation	Removes CO_2, generates oxygen
Atmospheric control Circulation Temperature/humidity Trace contaminate control	Provides for circulation of purified and tempered air; regulates vehicle total and partial oxygen pressures
Active thermal control Cabin loop Heat rejection loop Waste heat loop	Provides for equipment and air (via atmospheric control) cooling; rejects heat to space and transports waste heat (if available) to appropriate equipment
Water management Storage Recovery Water purity	Maintains storage facilities for potable and waste waters; processes waste waters for reclamation, and monitors and maintains potable water purity
Waste management Urinals Toilets Other trash	Provides receivers and storage facilities for biological, non-biological, wet and dry wastes; also processes several types of waste
Hygiene Showers Sinks Clothes	Provides hardware for whole body and partial body cleaning; crew clothing (or laundry facility), personal effects, and toilet articles also included
Food management Storage Preparation Serving Cleanup Monitoring	Weight of food, type, and hardware associated for storage, preparation, etc., included
Special life support Emergency supplies IVA/EVA Fire detection and control	Provides special hardware associated with listed subfunctions as well as any other miscellaneous services that may exist

Table 3.1-2. ECLSS Requirements

OLS MISSION IMPOSED DESIGN REQUIREMENTS	
1. Crew size	Nominal: 8 men; 16-day overlap up to 18 men Range: 4-12 men; 16-day overlap up to 14 - 24 men
2. Mission duration	Nominal: 10 years
3. Resupply period	Nominal: 109 days (Sect. 7.0, Vol.II) Design: 180 days
4. Gravity	Nominal: 0 g 0 to 1 g (if artificially induced gravity required)
5. Vehicle environment	Space vacuum to 14.7 psia
6. Maintenance	Inflight repair and/or replacement, and maintenance
7. Power supply to ECLSS	
8. Emergency reserve on expendables	30 days of normal OLS crew consumables (Section 7.0, Vol. II)
	graph 5.1.1)
9. Launch expendables/spares	109-day supply
10. Vehicle configuration	Single or multiple modules depending upon crew size and module size (see Section 2.0)
11. Vehicle total volume	20,000 ft^3 nominal (see Section 2.0)
12. Vehicle leakage rate	13-17 lb/day O_2/N_2 nominal case
13. Heat rejection	Space radiators (passive or active)

CREW REQUIREMENTS (NOMINAL)	
1. Metabolic heat (total)	11,200 Btu/man-day
2. Latent load	4,110 Btu/man-day
3. Sensible load	7,090 Btu/man-day
4. Oxygen consumption	1.84 lb/man-day
5. CO_2 production	2.25 lb/man-day
6. Food consumption (dry)	Supply: 1.68 lb/man-day Intake: 1.50 lb/man-day
a. Dried foods	1.04 lb/man-day
b. Wet foods (dry part)	0.64 lb/man-day
c. Water in wet foods	0.96 lb/man-day
d. Food packaging	1.18 lb/man-day
7. Latent water loss	3.09 lb/man-day

Table 3.1-2. ECLSS Requirements (continued)

8. Urine production	
a. Water	3.45 lb/man-day
b. Solids	0.13 lb/man-day
9. Feces production	
a. Water	0.25 lb/man-day
b. Solids	0.13 lb/man-day
SUBSYSTEM DESIGN REQUIREMENTS	
1. Total pressure	Nominal: 14.7 psia Design minimum: 10.0 psia
2. O_2 partial pressure	Nominal operation: 3.1 psia Maximum: 3.5 psia
3. Diluent	Nitrogen
4. CO_2 partial pressure	Nominal: 5.0 mm Hg Maximum: 7.6 mm Hg Emergency maximum: 15.0 mm Hg for 2 hr
5. Atmosphere temperature	Selective by room between 65 F and 75 F. Control tolerance ±3 F of selected temperature
6. Cabin ventilation rate	Nominal: 40 ft/min Minimum: 15 ft/min Maximum: 100 ft/min
7. Cabin humidity	Absolute humidity range: 8 mm Hg to 12 mm Hg pp H_2O Relative humidity range: 30 to 50% approximate Maximum dewpoint: 57 F
8. Water requirements model	
a. Crew water consumption	Nominal: 6.32 lb/man-day
b. Hand and face wash	4.0 lb/man-day
c. Shower	17.2 lb/man-shower
d. Dishwashing & housekeeping	3.0 lb/man-day
e. Experiments	Design: 35 lb/day of potable water
f. Urine flush	3.5 lb/man-day
g. Laundry	4.0 lb/man-day
9. Special LSS	16 man-hr
a. IVA	16 man-hr
b. EVA	4 times/month with 2 men
c. Emergency OLS requirement	(See item 8 of Table 4.1-1)
d. Emergency LEP requirement	

repressurization steadily declines. Also, as soon as gas is used for repressurization, a new resupply of the gases (for leakage, RCS, or EPS) may be required ahead of schedule depending upon the amount of gas remaining in the tanks.

Figure 3.1-1 presents the weight of oxygen and nitrogen required for OLS repressurization. Density factors in Figure 3.1-1 permit conversion of the weight of gas to the volume occupied at cryogenic temperatures or at 3000 psia and 70 F. The repressurization requirements are a function of vehicle gas volume and pressure and not crew size. However, a relative crew size factor has been assigned so that the data can be read as a function of crew size. The reference data points shown in Figure 3.1-1 are for the 12-man, 30,000-ft^3 EOSS. Figure 3.1-1 does not include tank weight penalties. High-pressure and cryogenic tank penalties are given in Table 3.1-3. The tank penalty varies depending upon the space environment and gas usage rates. Also, high-pressure storage requires no heater penalties, while cryogenic storage will require a heater penalty, which will be on the order of 140 to 150 Btu per pound of oxygen or nitrogen required. It is assumed that the tanks are within the environmental shield of the OLS in unpressurized areas, so the penalties do not include the weight of fairings for mounting of the tanks outside the mold line.

Table 3.1-3. Tank Penalties

Gas	Storage State	Tank Penalty (lb tank/lb gas)
O_2	High pressure (3000 psia)	1.20
	Subcritical cryogenic	0.08 to 0.12
N_2	High pressure (3000 psia)	1.20
	Subcritical cryogenic	0.12 to 0.20

Gas Makeup

It is assumed that metabolic and leakage oxygen will be made up solely by the oxygen generation unit of the CO_2 management assembly and/or the RCS. Therefore, only parametric data are presented for nitrogen makeup. Figure 3.1-2 presents seal leakage (oxygen and nitrogen) as a function of seal length. Also, leakage caused by six or eight docking ports is indicated on the seal length axis. The docking port leakage was estimated for the 1985 time frame. This rate requires technological advancements in seals.

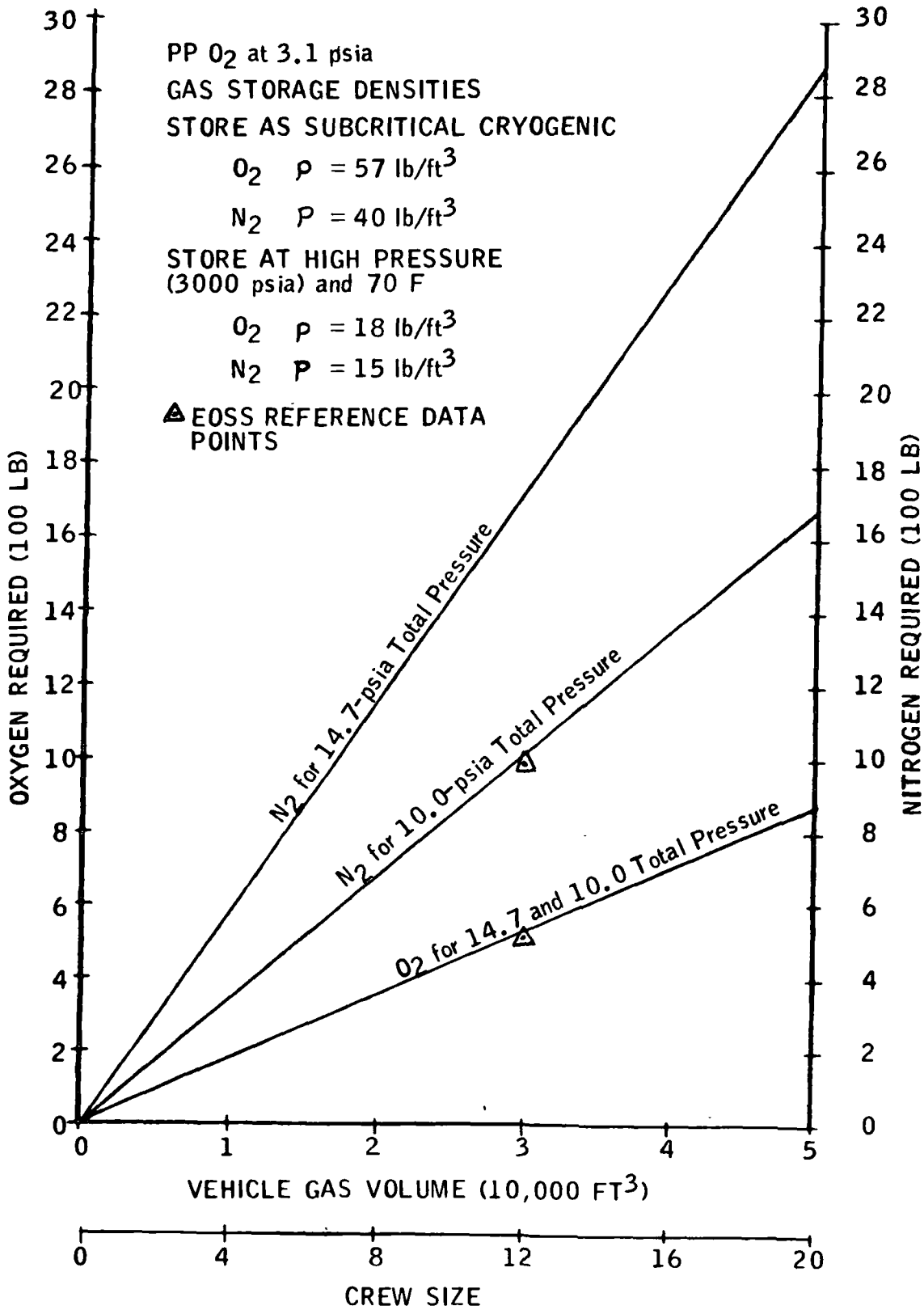


Figure 3.1-1. Repressurization Gas

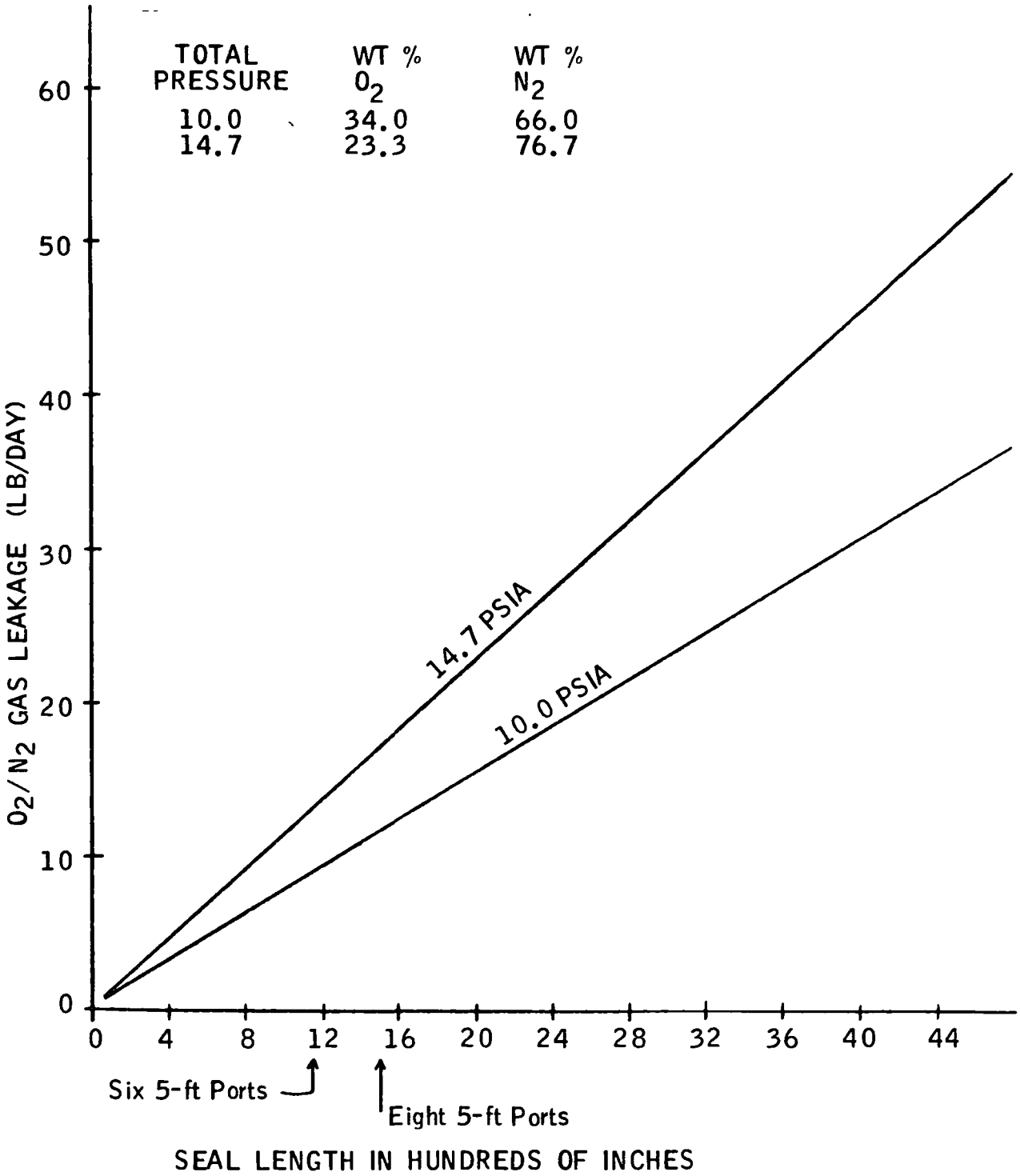


Figure 3.1-2. O_2/N_2 Gas Seal Leakage

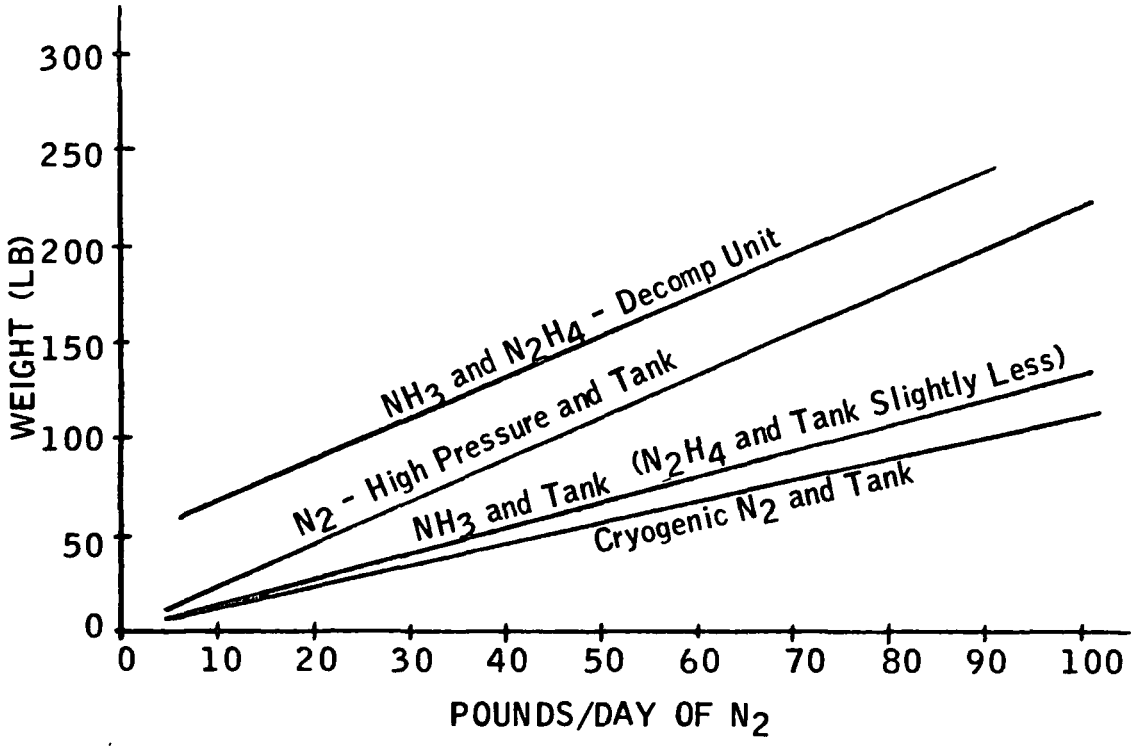
To use Figure 3.1-2, the seal lengths for all ports, airlock, etc., are determined and totaled; then the O_2/N_2 leakage is read off the appropriate curve as a function of seal length and internal pressure.

The leakage indicated in Figure 3.1-2 consists only of seal leakage (vehicle leakage). The leakage does not include oxygen and nitrogen losses caused by venting to vacuum as in the toilets and gas condensers, nor does it include pumpdown gas losses. These venting losses are strongly affected by configuration and cycling of equipment. These other gas losses are an order of magnitude less than the seal leakage losses as indicated in Table 3.1-4. Thus, for Phase A these venting losses can be neglected or approximated based on the data of Table 3.1-4.

Table 3.1-4. Gas Losses for 30,000-ft^3 Station

Gas Loss Type	Amount of Loss (O_2/N_2) (pounds/day)
Seal leakage	20
Waste management	1.2
Pumpdown	1.6

With metabolic oxygen and oxygen leakage being madeup by the oxygen generation unit of the CO_2 management assembly, only storage of makeup nitrogen need be considered. Figures 3.1-3 and 3.1-4 compare high pressure, cryogenic, and chemical decomposition nitrogen storage concepts. Nitrogen makeup requirements are based upon leakage only. Figure 3.1-3 is read for a decomposition unit by using the daily rate of required N_2 to size the decomposition unit then using the NH_3 and tank weight line to obtain the daily storage penalty which, when multiplied by the storage period (days) will give the storage weight. In Figure 3.1-3, chemical NH_3 and N_2H_4 have 10-percent tank penalties, while the cryogenic N_2 tank has a 20-percent penalty. No credit for hydrogen production is shown in Figures 3.1-3 and 3.1-4. Ammonia dissociation produces 0.214 pound of H_2 per pound of N_2 and hydrazine produces 0.145 pound of H_2 per pound of N_2. Storage of hydrogen cryogenically (subcritical) requires 1.43 pound of tank weight per pound of hydrogen. Subcritical hydrogen has a density of 3.2 lb/ft^3. Thus, less total storage volume is required if hydrogen is stored chemically as NH_3 or N_2H_4 rather than cryogenically as subcritical H_2. The presence of hydrogen cannot be considered a credit for ammonia or hydrazine decomposition systems, unless the hydrogen can be used by RCS in resistojets or by ECLSS for the CO_2 reducing Sabatier reactor concept. Figure 3.1-4 is a continuation of Figure 3.1-3 and is utilized in the same manner as Figure 3.1-3.



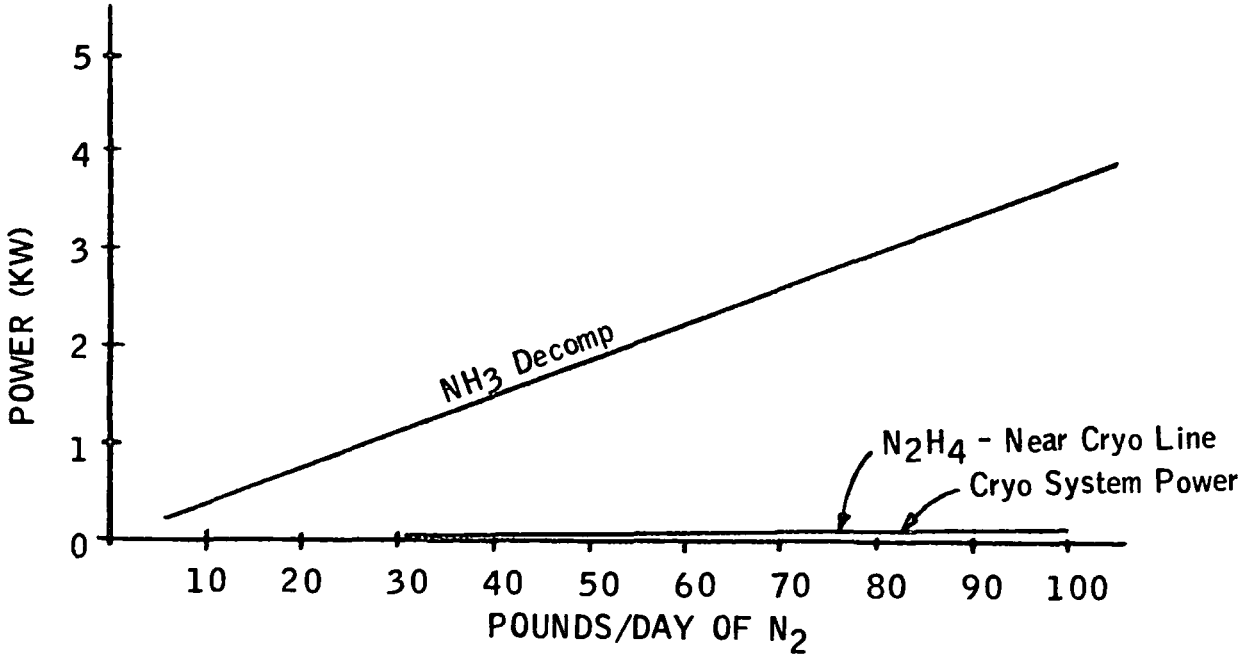


Figure 3.1-3. N_2 Storage - Weight and Power

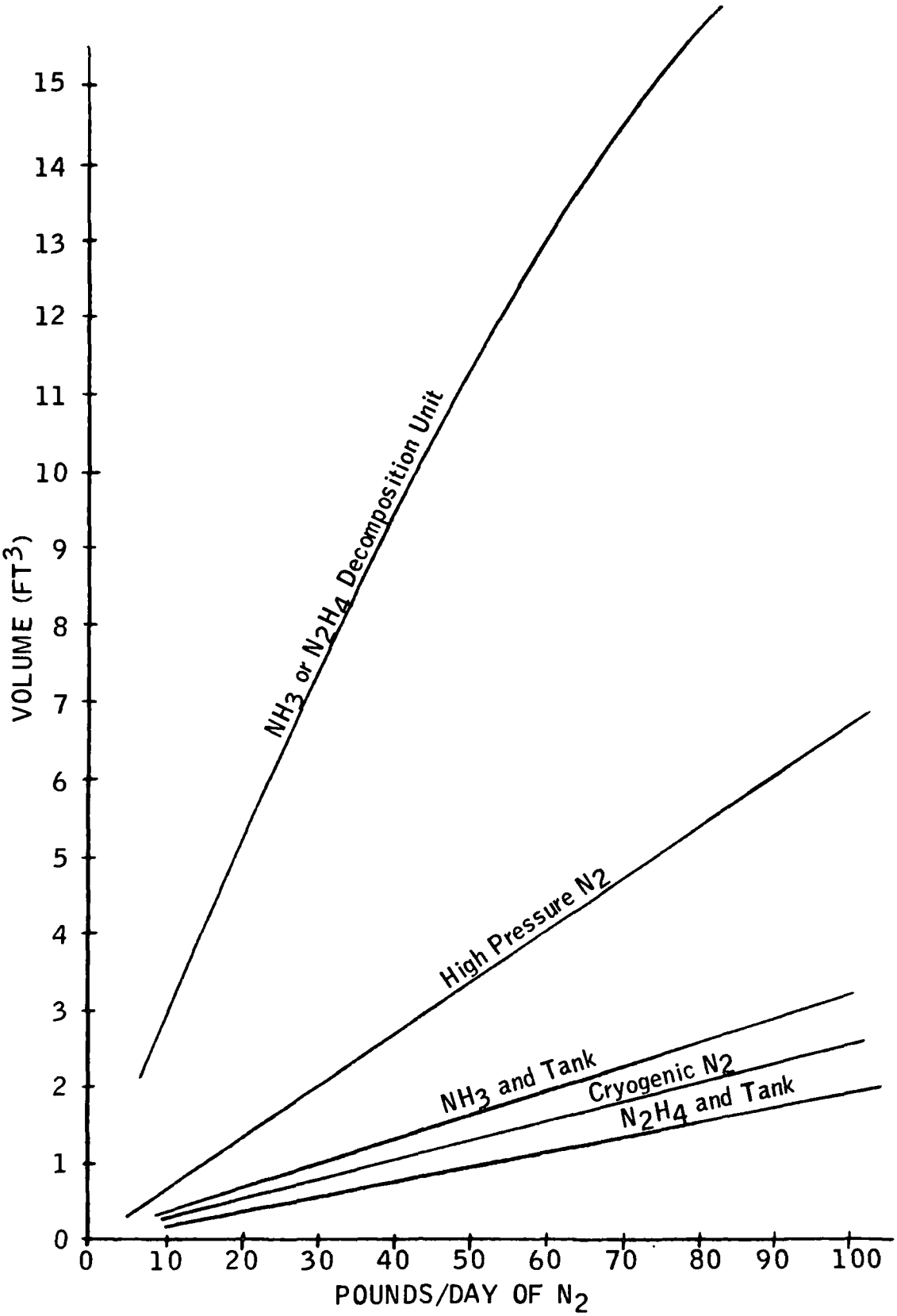


Figure 3.1-4. N_2 Storage - Volume

Pumpdown Systems

Pumpdown systems may be used to salvage atmospheric gases trapped in airlocks or experiment volumes, which are periodically exposed to vacuum. The number of airlock repressurizations and dumps to space determine the amount of gas lost if a pumpdown system is not employed. The amount of gas salvaged through use of a pumpdown system must be compared to the weight and power penalties of the system.

The pumpdown system trade is configuration sensitive because the frequency of pumpdown and airlock or module sizes being pumped need to be known. As with the EOSS, several pumpdown systems may be used instead of one central pumpdown system. The EOSS as defined by NR in SD 70-510-1, dated July 1970, has a large four-stage pumpdown system with a reservoir in one pressure volume for pumping down a volume of 2400 cubic feet or less. The other EOSS pressure volume uses small one stage compressors pumping to the cabin atmosphere because in that volume only the docking ports require pumpdown. Figure 3.1-5 presents the weight characteristics of the composite EOSS pumpdown system as a function of pumpdown frequency. This figure includes the combined frequencies of the various sized volumes included in the EOSS pumpdown system. The power penalty shown in one curve is based upon 0.5 lb/watt and the 700-watt 24-hour average power required during an actual pumpdown of the large four-stage compressor. For the EOSS, a large attached module, with frequent pumpdown requirements contributed heavily to the amount of gas loss. Thus, Figure 3.1-6 was generated with the large attached module influence removed. Without the attached module, the large four-stage compressor and reservoir was not required, and this unit was replaced by multiple one stage compressors. Thus, the weight and power penalties of Figure 3.1-6 are less than those of Figure 3.1-5.

An approximate method of relating crew size to pumpdown frequency is to ratio frequency to crew size by using the EOSS reference of 12 men and 13.12 cycles per month. The variables that size the pumpdown system in addition to the frequency of pumpdown are final pressure after pumpdown and time for pumpdown. The final pressure determines the amount of energy required for pumpdown. This final pressure refers both to final reservoir pressure and final pumped volume pressure. (The EOSS has a final reservoir pressure of 1000 psia and a final pumped volume pressure of 1.0 psia.) Pumpdown time influences the size of the compressor and the power required.

3.1.3 CO_2 MANAGEMENT

This function deals with the removal of carbon dioxide and its reduction to oxygen. All of the CO_2 reduction concepts considered, except for the solid electrolyte concept, produce water instead of oxygen and, therefore, require water electrolysis. For those concepts that require water electrolysis, metabolic and leakage oxygen can be made up by integration with water management. Thus, this function impacts atmospheric storage and water management. Also, if the Sabatier reactor is chosen, it produces methane (CH_4) that may impact the RCS depending upon whether the RCS can use CH_4 in resistojets.

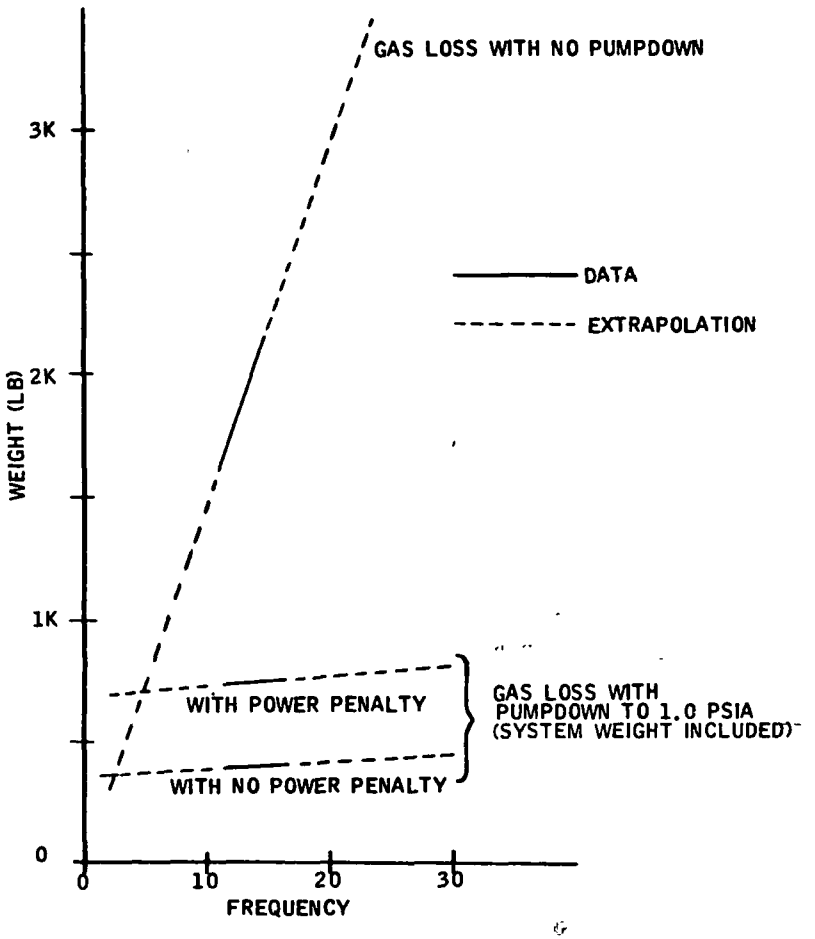


Figure 3.1-5. Earth Space Station Pumpdown Trade (Includes Attached Module Factor)

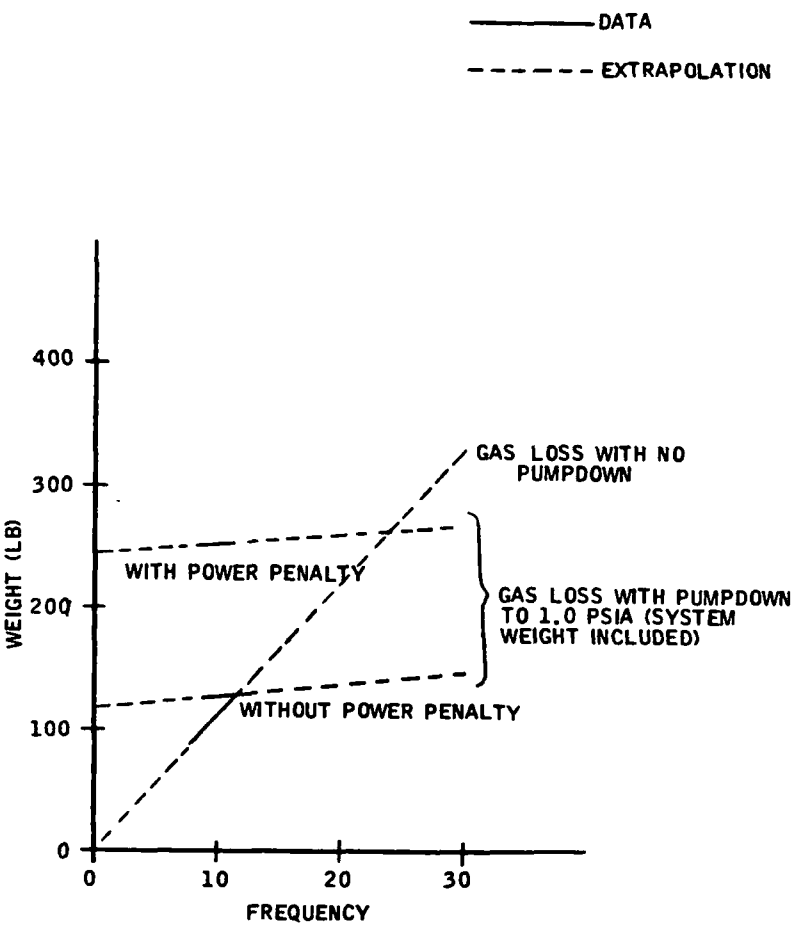


Figure 3.1-6. Modified Earth Space Station Pumpdown Trade (Attached Module Factor Removed)

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CO_2 Removal

The CO_2 removal concepts being considered for OLS use are the molecular sieve (Figure 3.1-7), steam desorbed resin (Figure 3.1-8), solid amine (Figure 3.1-9), and hydrogen depolarized cell (Figure 3.1-10). Figure 3.1-11 presents single unit system weights and spare weights for these systems. The weights shown are only valid for comparison purposes because the molecular sieve may not operate 24 hours a day. The curves were plotted based upon 2.25 lb/man-day of CO_2 removal for a 24-hour operating period.

Figure 3.1-11 presents approximate initial spare requirements as a percentage of the system weight. The spares requirement was calculated based upon surveys of past histories of similar components making up the various ECLSS units. Component life and random failure data were utilized to arrive at the number of spares required for those various units. In cases where data were not specifically available (i.e., new components), a percentage factor -- based upon data already collected for other component types and adjusted for the amount of similarity in operation and design -- were utilized to obtain a spares weight.

For a volume comparison, an average fixed system and spares density of 29 pounds per cubic foot can be used for all concepts but the H_2 depolarized cell. For this concept, the density is 61 pounds per cubic foot for system weight, and 24 pounds per cubic foot for the spares. Figure 3.1-12 presents the power requirements and corresponding heat rejection penalties (which are continuous requirements for 24-hour operation). The power requirements shown can only be used for comparison because the more detailed EOSS molecular sieve studies resulted in the molecular sieve power requirement increasing from 1.2 kilowatts to 2.3 kilowatts for 12 men. This updated power for the molecular sieve is not shown in Figure 3.1-12 because detailed studies for the other concepts were not performed and, therefore, their potential power increases are unknown. In reality, the H_2 depolarized cell concept only requires 60 watts power for 12 men. However, some electrolysis of water to produce oxygen and hydrogen to operate the cell is required, and this causes the large power requirement shown in Figure 3.1-12 as well as a weight delta, which is included in the weight curve.

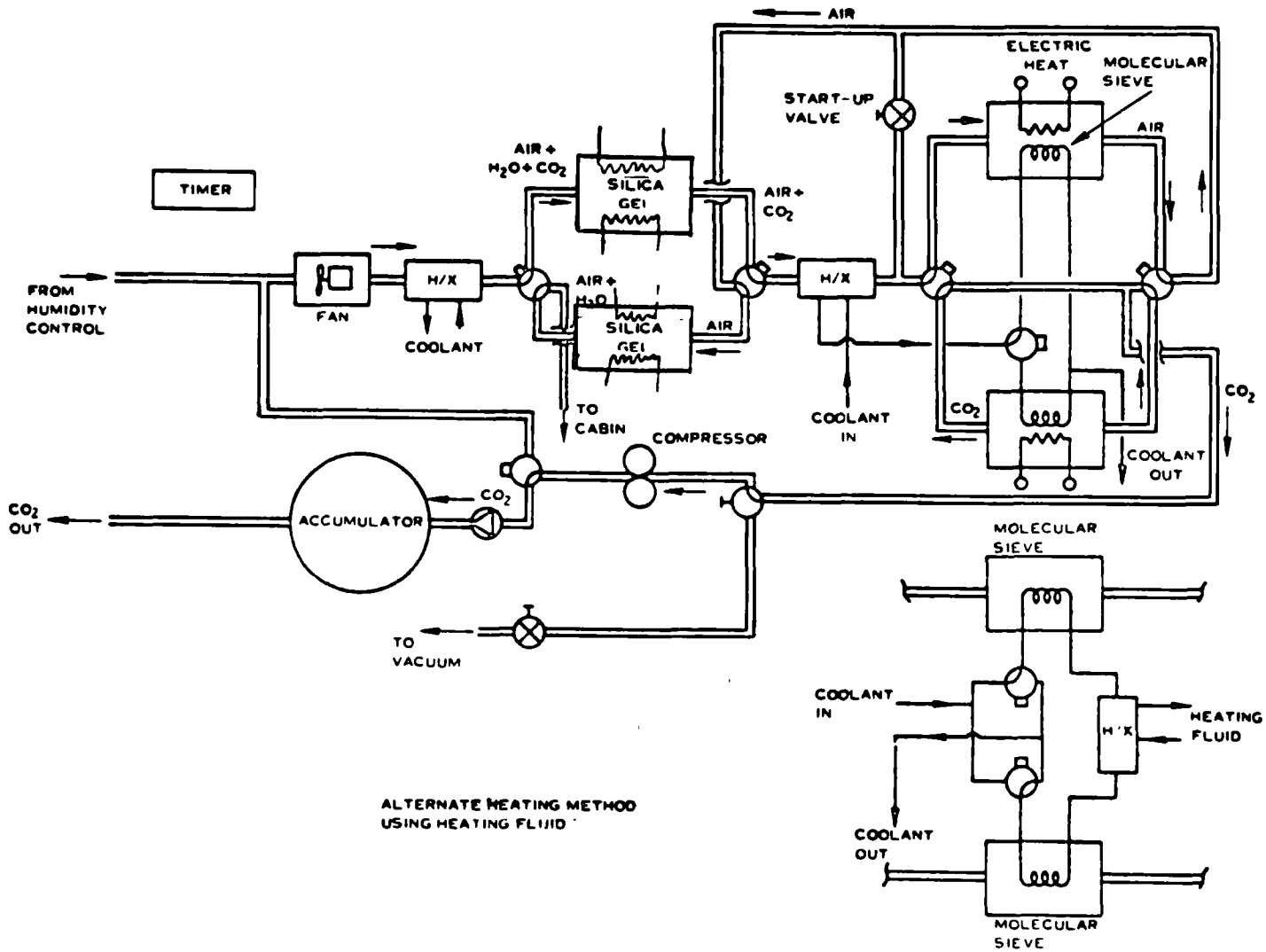


Figure 3.1-7. Molecular Sieve Concept

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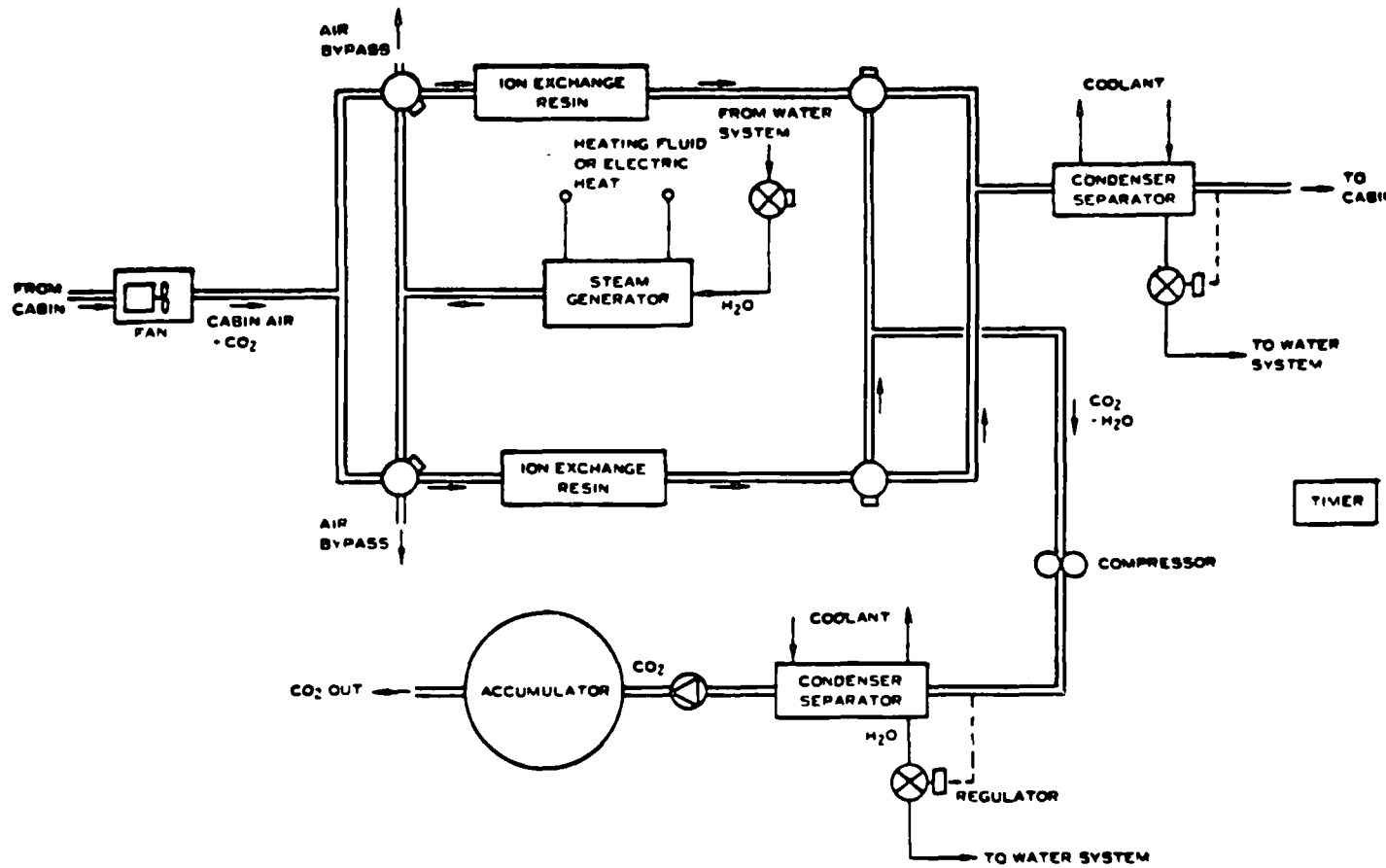


Figure 3.1-8. Steam Desorbed Resin Concept

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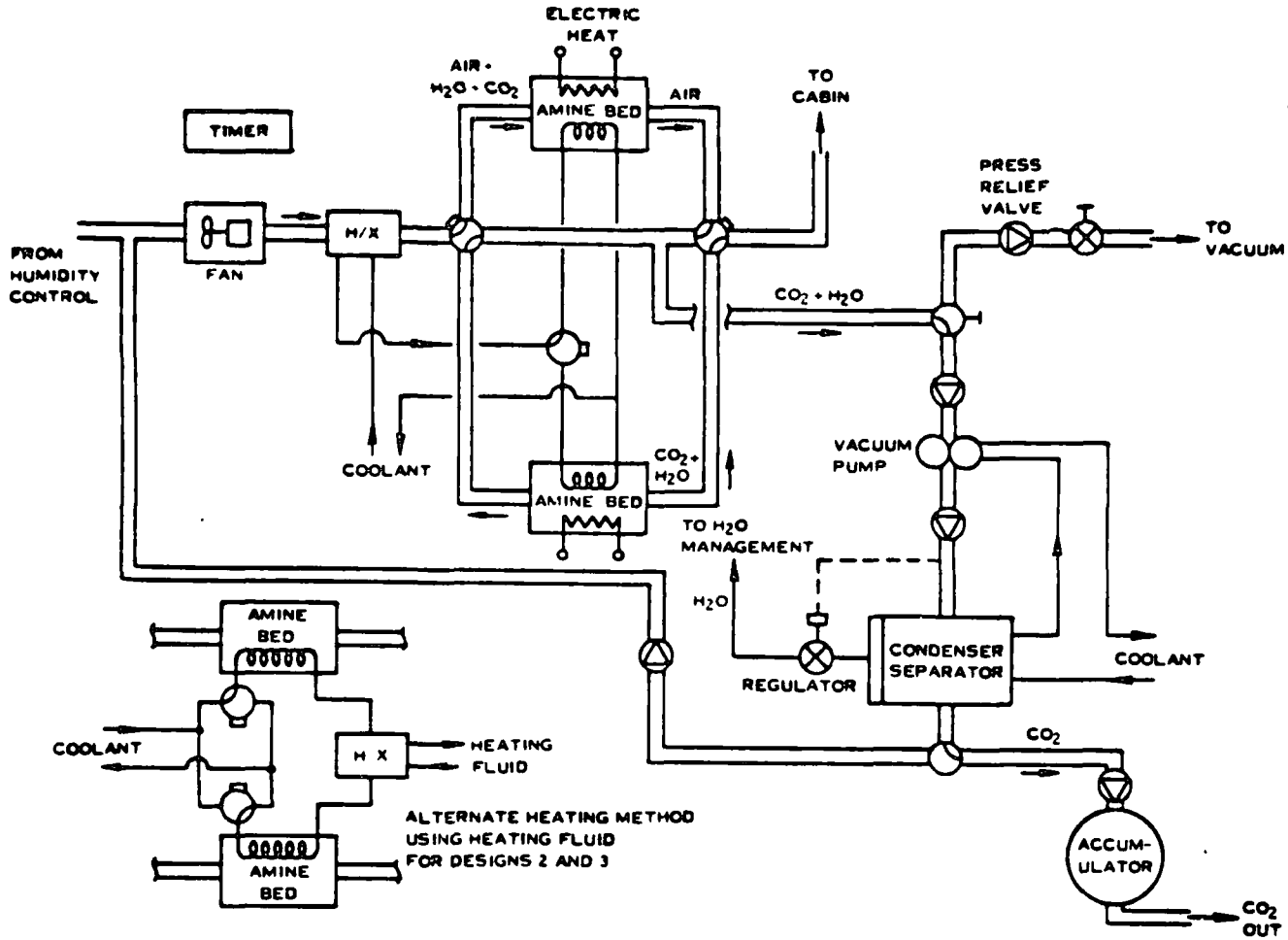


Figure 3.1-9. Solid Amine Concept

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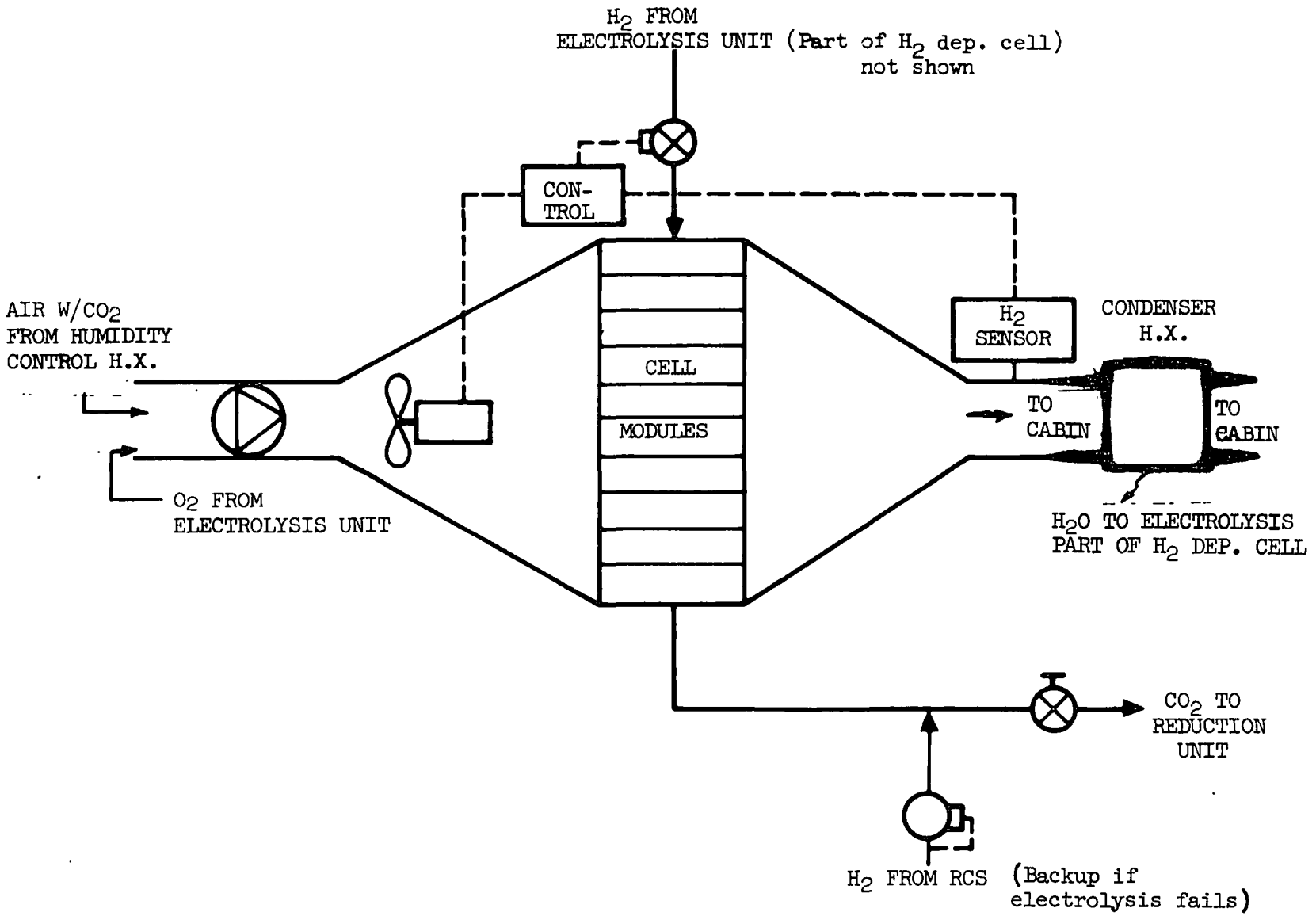
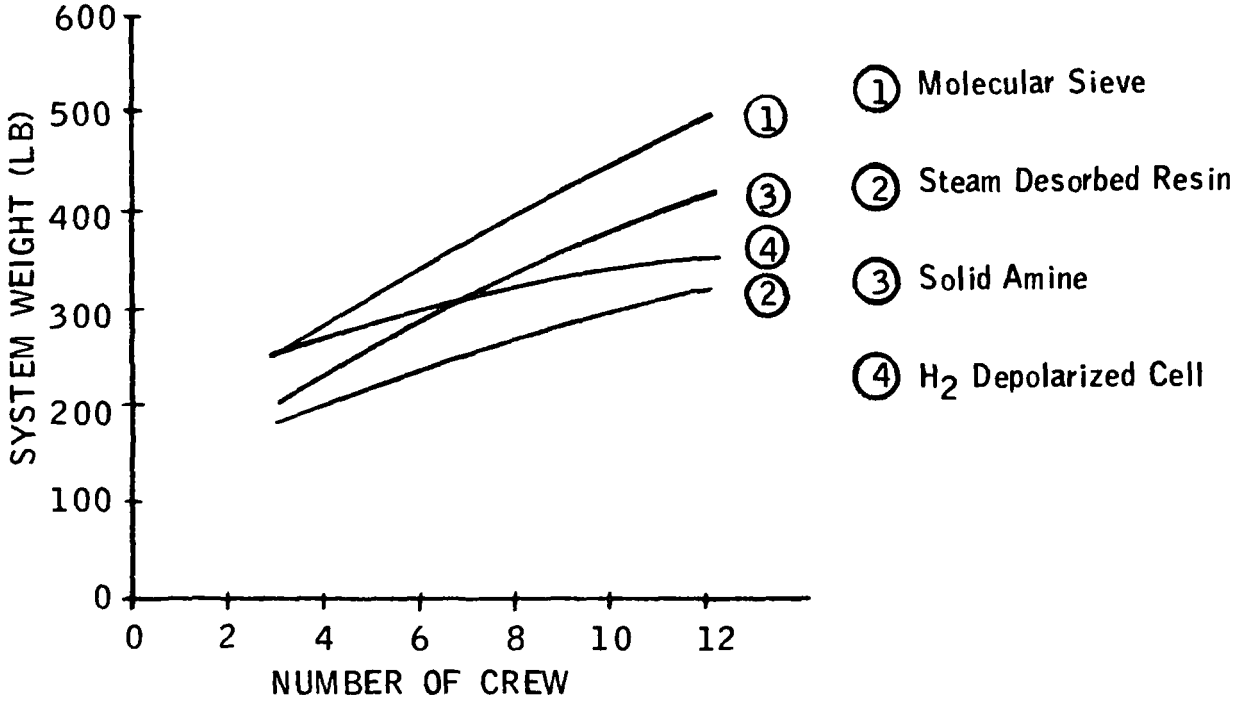


Figure 3.1-10. H_2 Depolarized Cell

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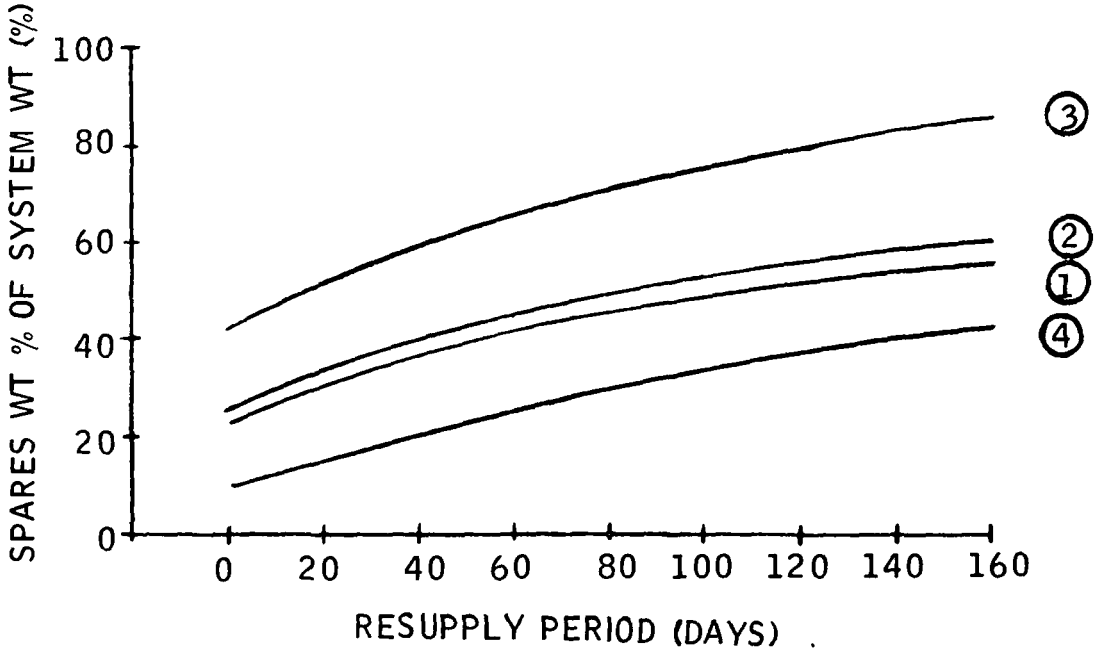


Figure 3.1-11. CO_2 Removal System Weight Trade Study

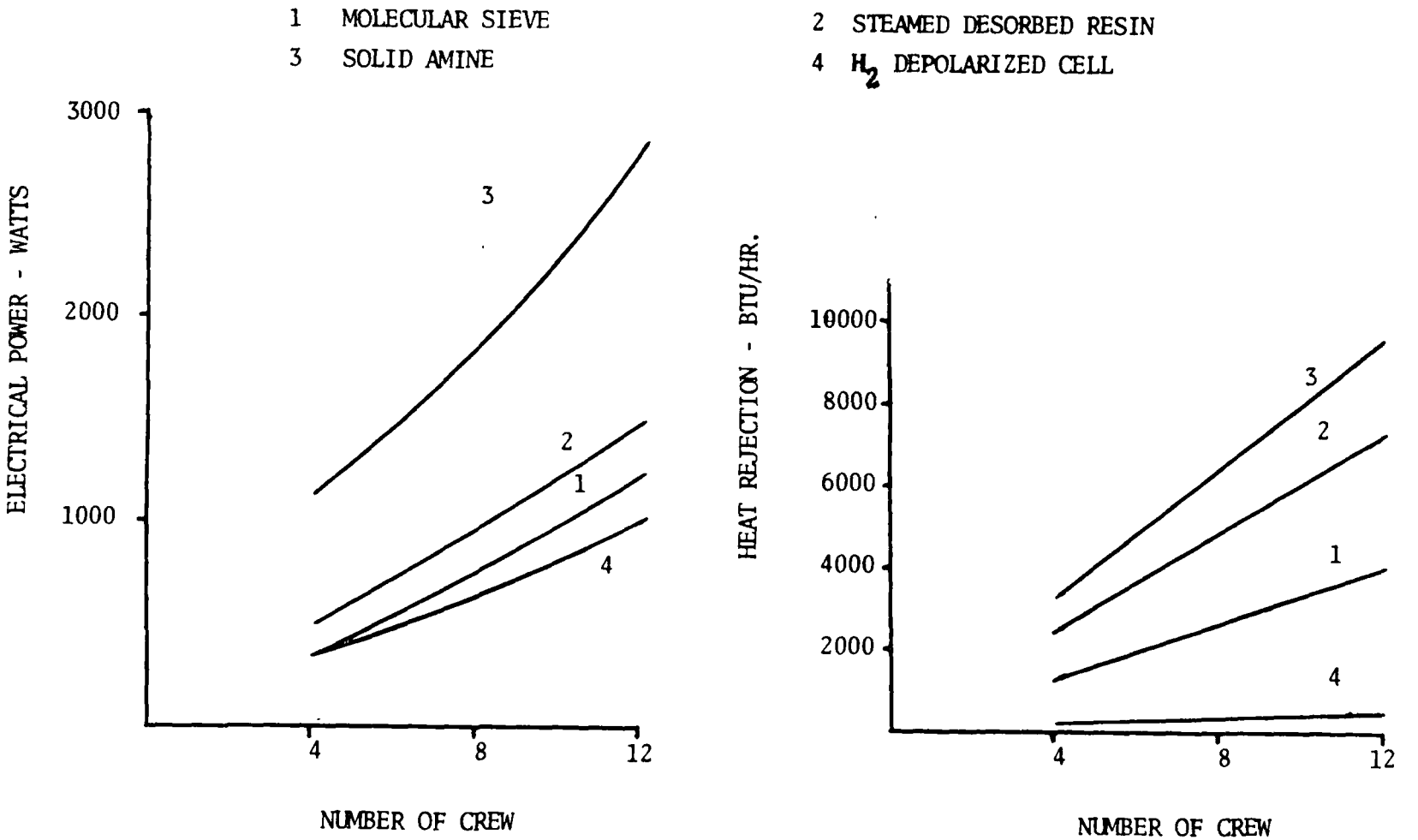


Figure 3.1-12. CO_2 Removal - Electrical and Heat Rejection Penalties

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CO_2 Reduction

Four concepts were studied for CO_2 reduction. They are Sabatier with methane dump (Figure 3.1-13), Bosch (Figure 3.1-14), solid electrolyte (Figure 3.1-15), and Sabatier with methane cracking (Figure 3.1-16). Figure 3.1-17 presents the single unit system and spare weights for the four reduction units. For Figure 3.1-17, the average system density factor is 11.3 pounds per cubic foot. The spares density is 22.5 pounds per cubic foot. Figure 3.1-18 compares the expendables required for these CO_2 reduction units. The expendables consist of hardware expendables such as catalysts, filters, etc. Only the Sabatier/methane dump concept requires a hydrogen gas expendable. The tank penalty included in the figure is 38 percent (0.38 lb tank/lb gas), which is low in terms of cryogenic storage or high-pressure storage according to newer data.

The power curves and heat rejection penalties for the four concepts are given in Figure 3.1-19. All the concepts shown, except the Sabatier/methane dump concept require power to maintain the required chemical reactions. The Sabatier/methane dump concept only requires 75 watts startup (for 12 men) and then 14 watts continuous for instrumentation because it does not attempt to end up with carbon as a product.

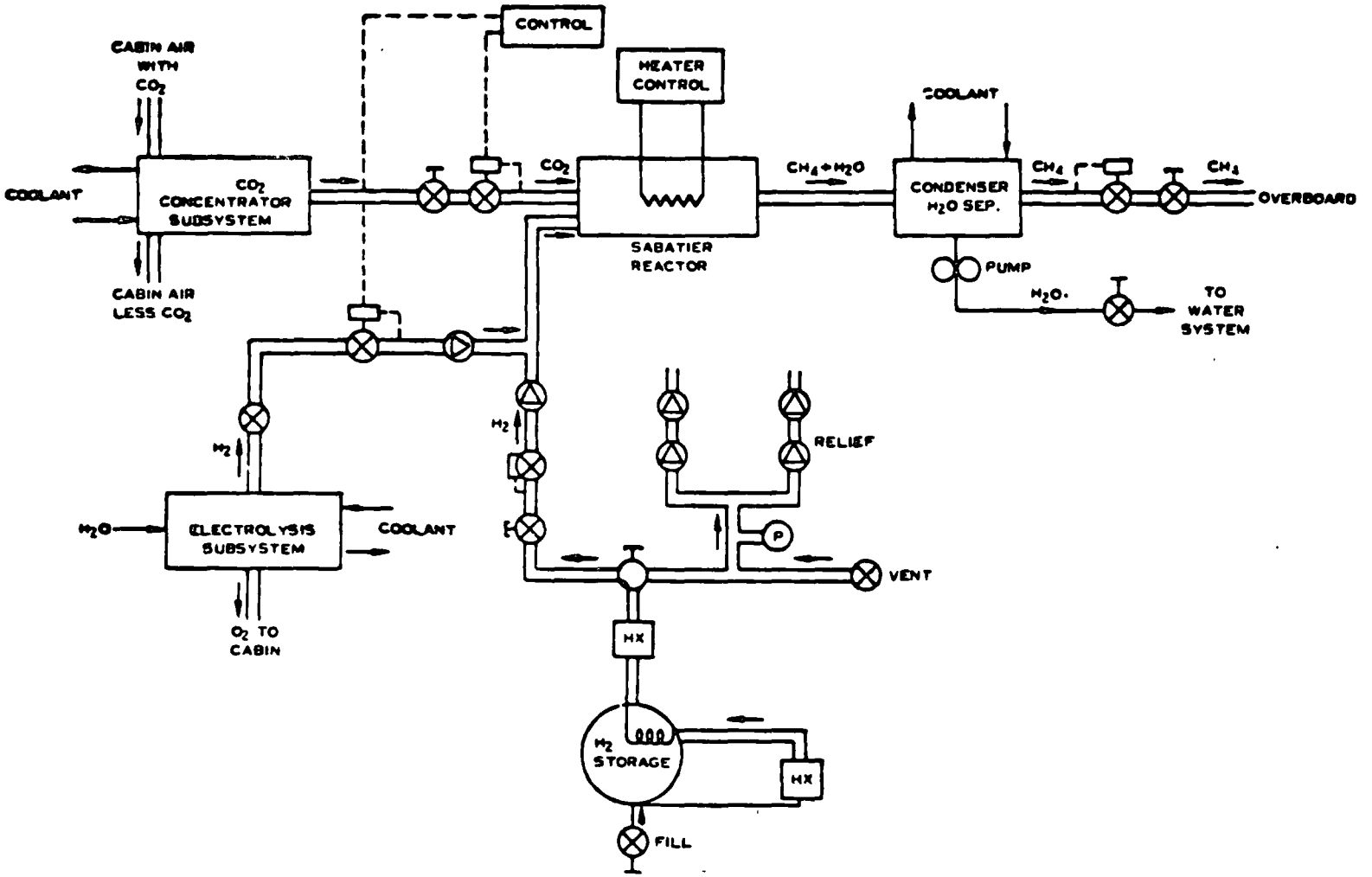


Figure 3.1-13. Sabatier CO_2 Reduction with Methane Dump Concept

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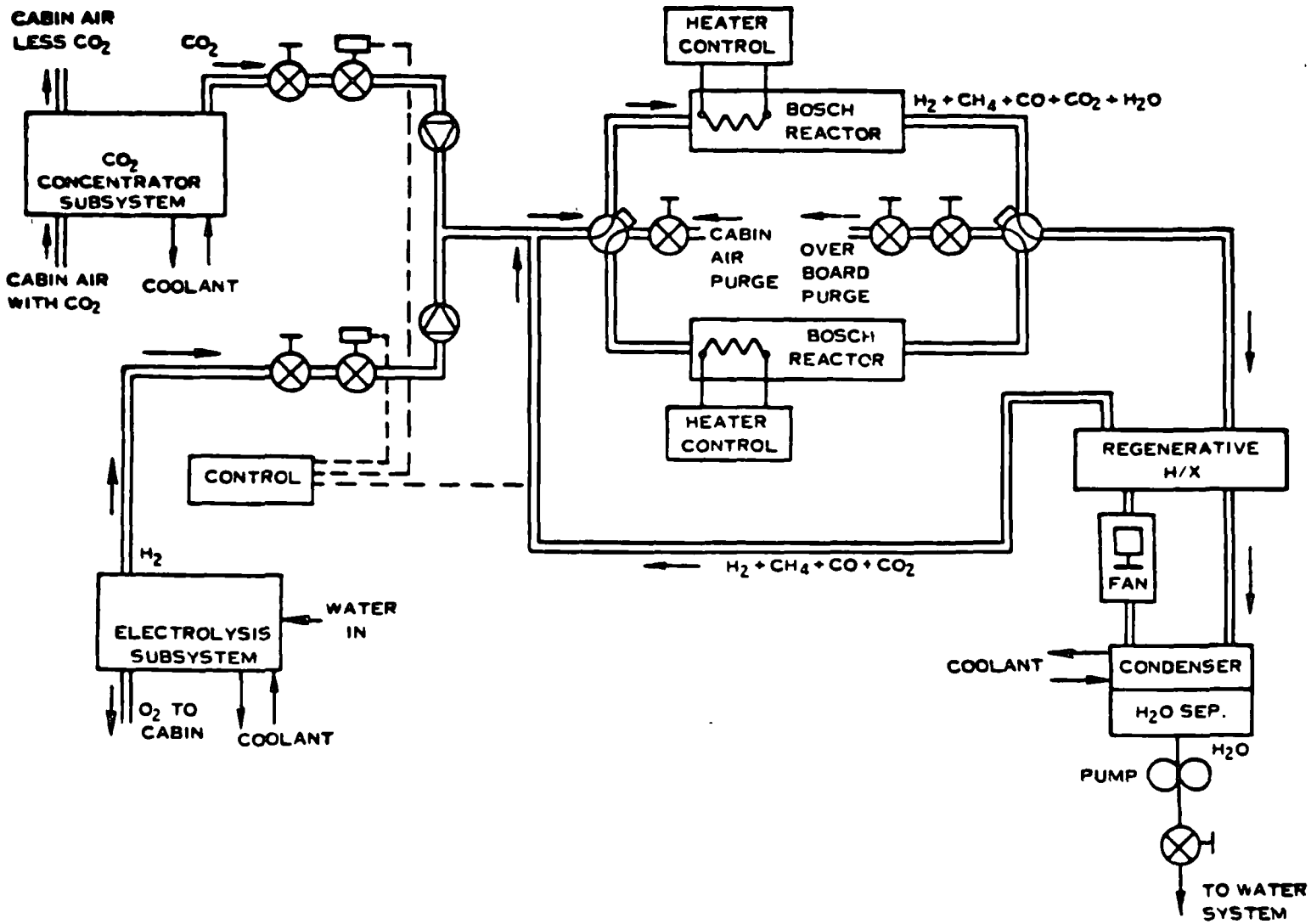


Figure 3.1-14. Bosch CO_2 Reduction Concept

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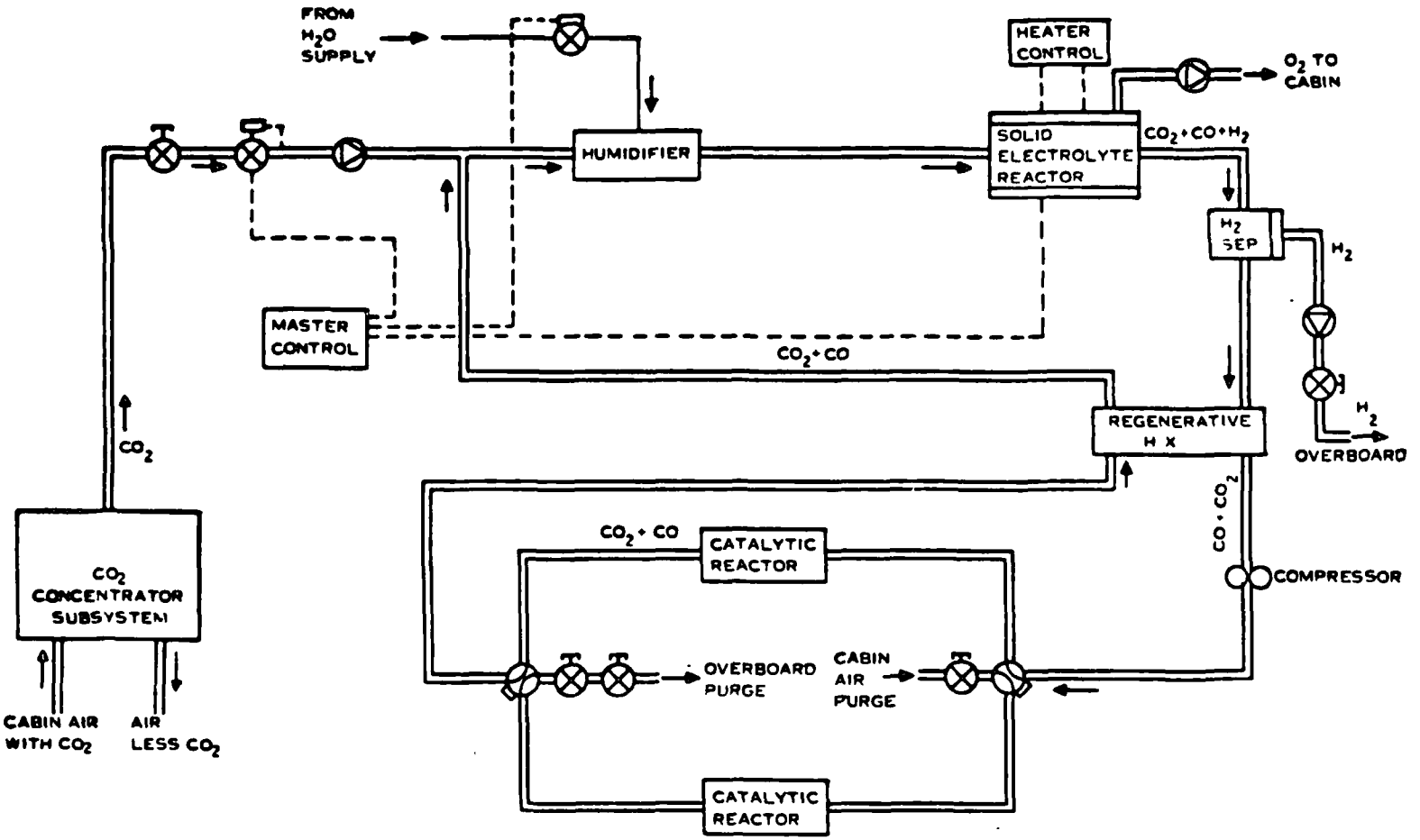


Figure 3.1-15. Solid Electrolyte Concept

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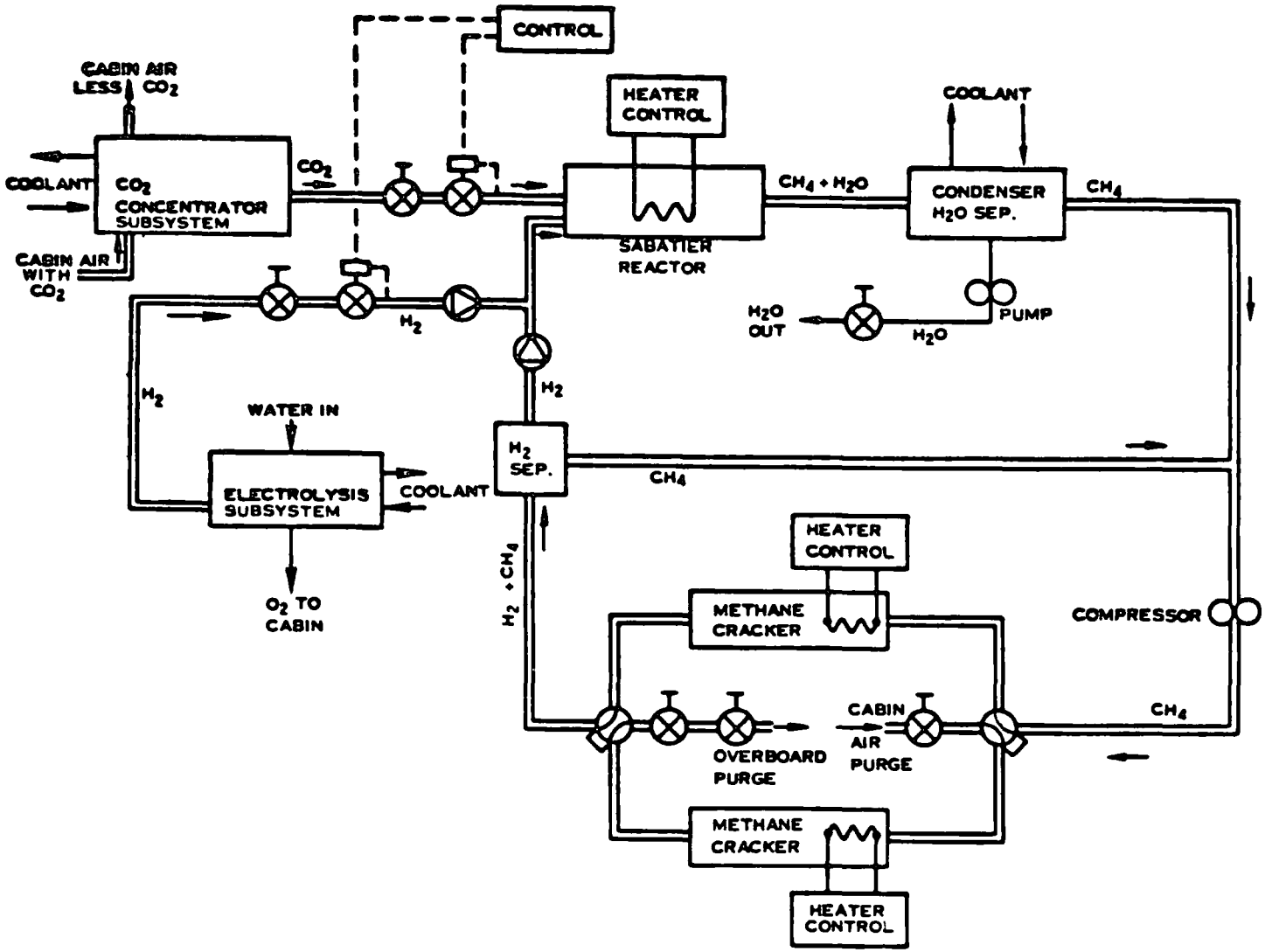
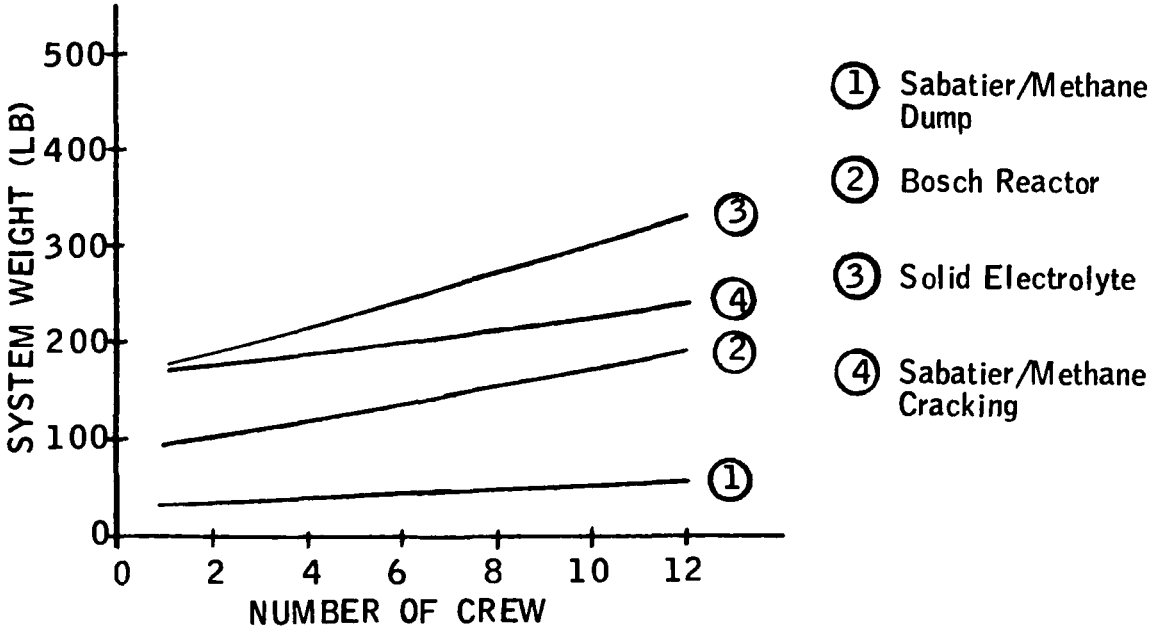


Figure 3.1-16. Sabatier CO_2 Reduction with Methance Cracking Concept

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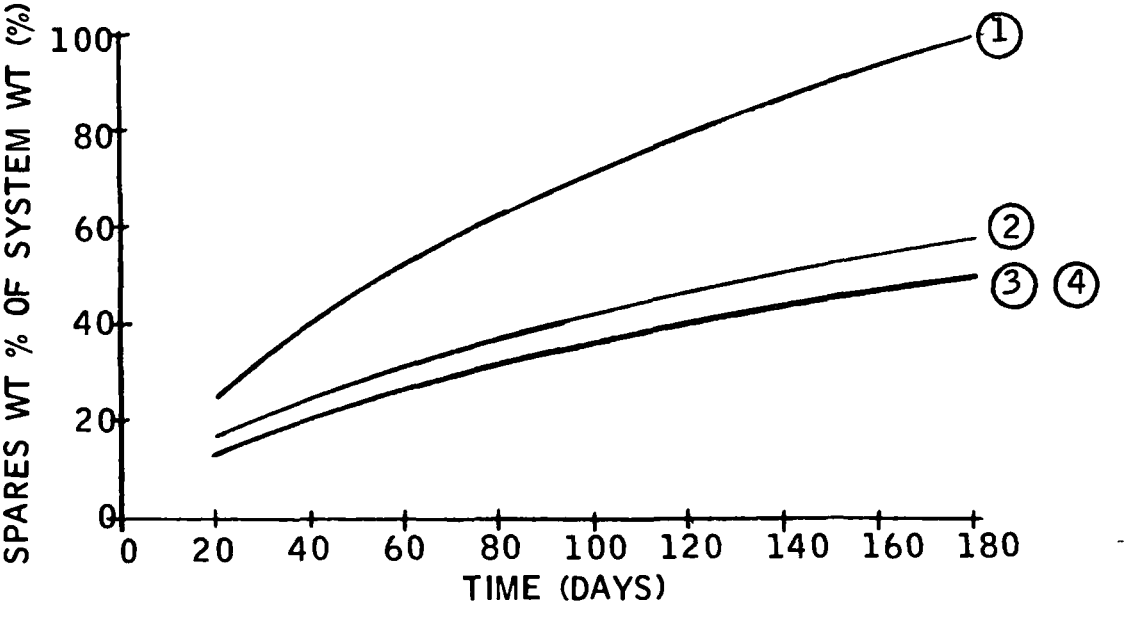
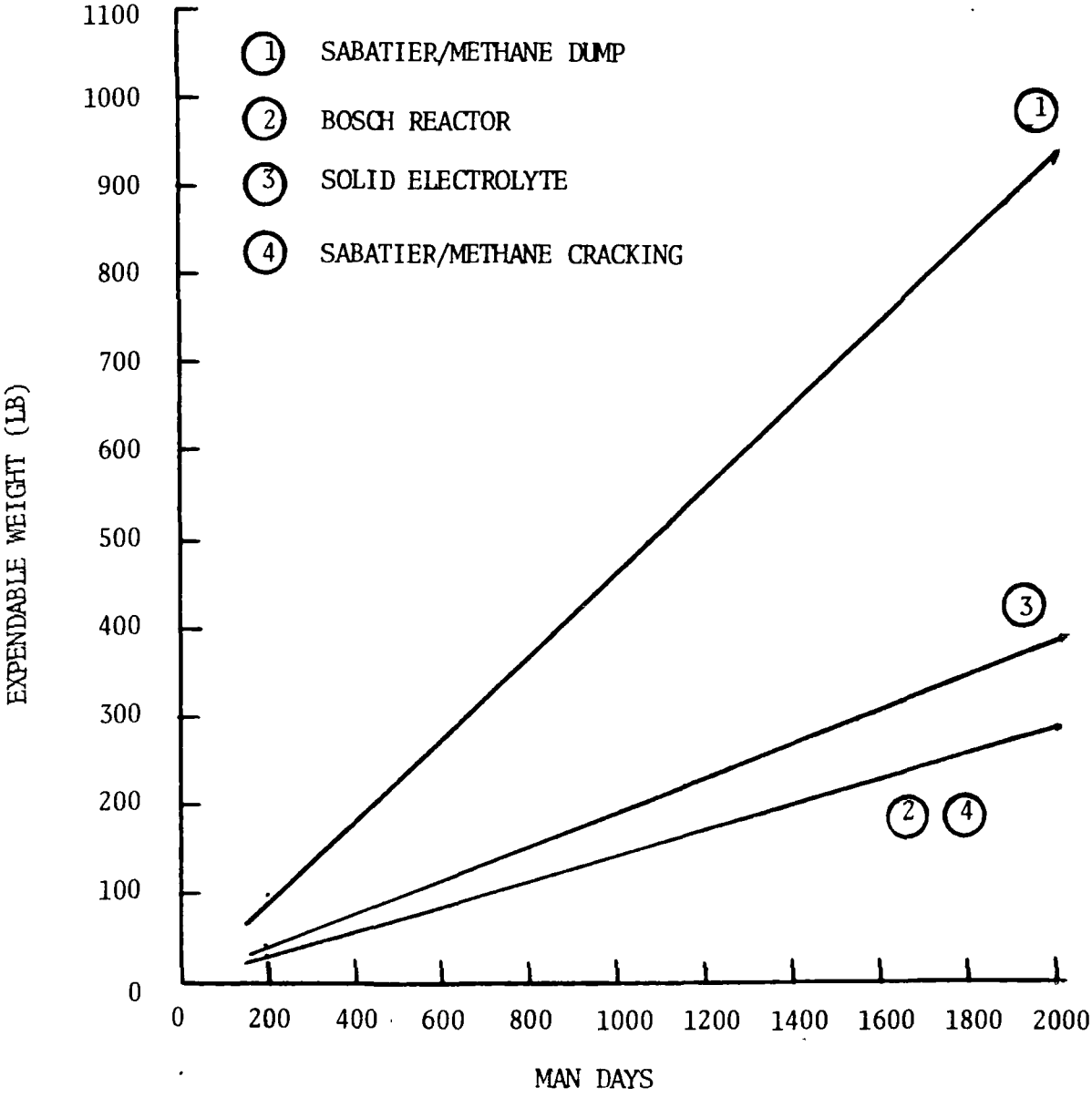
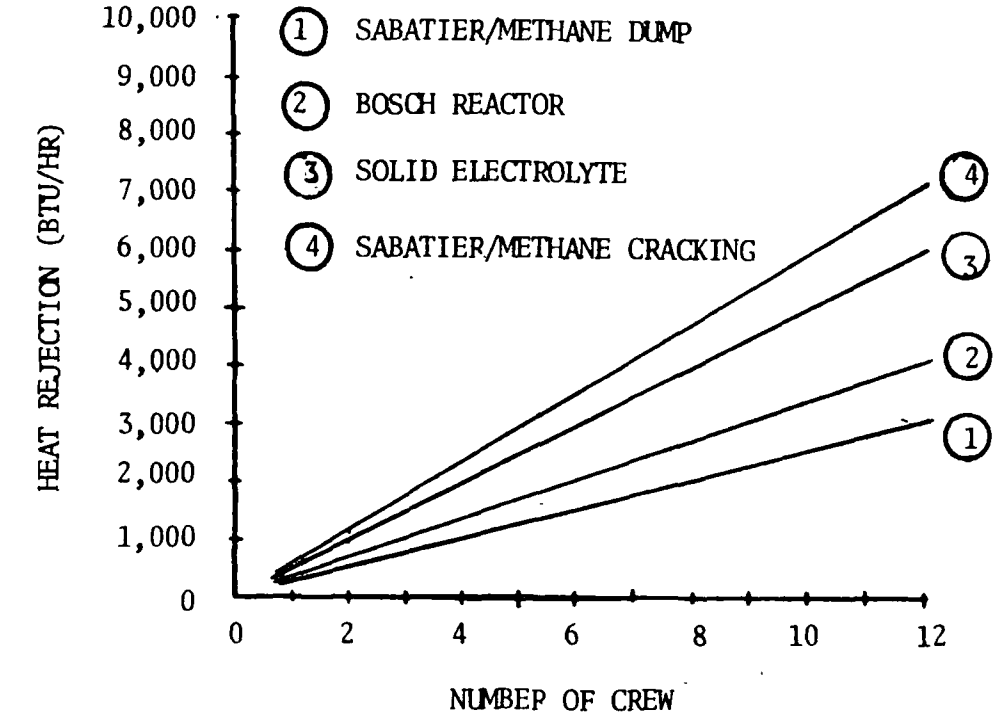


Figure 3.1-17. CO_2 Reduction - System Weight Trade Study



NOTE: EXPENDABLES INCLUDE SUCH ITEMS AS FILTERS, CATALYSTS OR CHEMICALS NORMALLY PROGRAMMED FOR REPLACEMENT.
(H_2 WITH 138-PERCENT TANK PENALTY INCLUDED IN CONCEPT)

Figure 3.1-18. CO_2 Reduction - Expendable Weight Trade Study



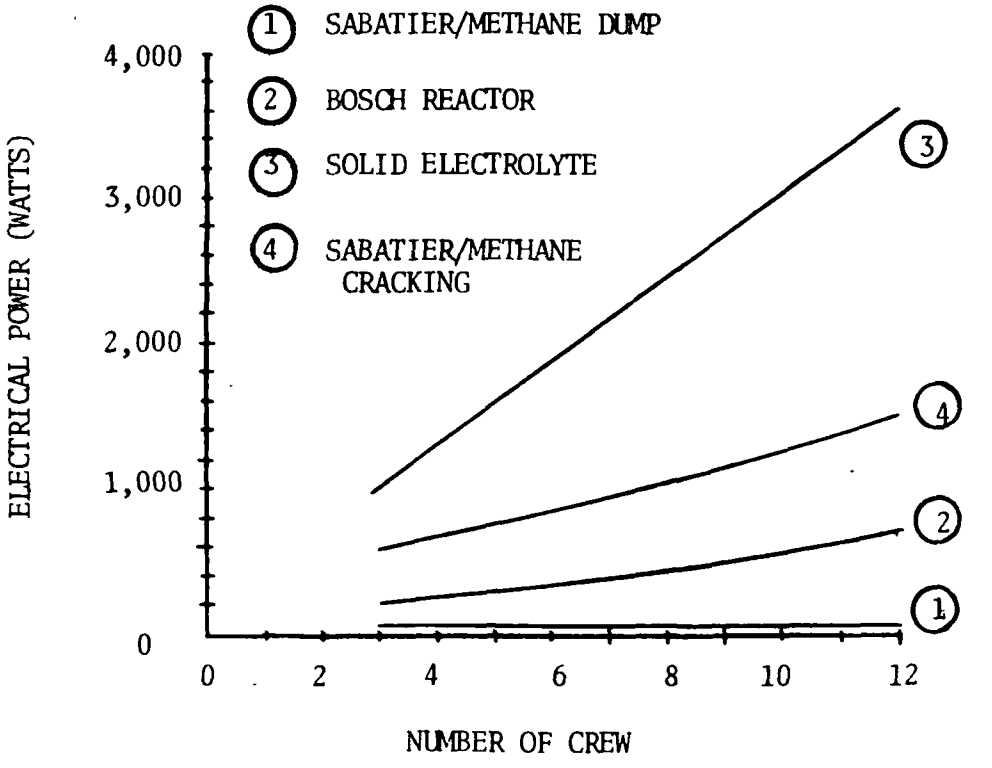


Figure 3.1-19. CO_2 Reduction - Electrical Power and Heat Rejection Penalty

Oxygen Generation

All of the CO_2 reduction concepts, except for solid electrolyte, involve chemical reactions that produce water instead of oxygen. Thus, a water electrolysis unit is required to convert the water into hydrogen (for the CO_2 reduction units) and oxygen. The solid electrolyte unit directly converts CO_2 to oxygen.

Those units that require water electrolysis can utilize makeup water to produce metabolic and leakage oxygen. However, the solid electrolyte unit cannot produce leakage oxygen makeup nor the total metabolic oxygen requirement. (All the CO_2 reduction units cannot produce the total metabolic oxygen from only CO_2 because the exhaled CO_2 does not contain all the inhaled oxygen. Some oxygen is expelled from the human body as metabolic water.)

Figures 3.1-20 through 3.1-24 are schematics of the various electrolysis concepts. Previous EOSS studies indicate that the concepts to consider for OLS use are wick feed, gas circulation, and solid polymer. The system weights for these concepts are shown in Figure 3.1-25. The densities for the electrolysis concepts are 26.9 lb/ft^3 for the system weights and 26.5 lb/ft^3 for the spares. These electrolysis weights and volumes are relative and not absolute. Redundancy, maintenance, leakage, and operating requirements drastically effect weight and power. The curves show relative weight differences for a nominal 13 to 14 hour/day operational cycle, nonredundant electrolysis unit. Figure 3.1-26 shows relative power and heat rejection levels that can vary over a wide range of values, depending upon the hourly flowrate (which is a function of the operating time and the daily average flowrate). For redundancy these unit weights must be added; therefore, redundancy must be considered in making concept selection.

3.1.4 ATMOSPHERIC CONTROL

The atmospheric control system is directly related to the interior vehicle configuration. Thus, only an approximation for the various OLS subassemblies could be made in terms of weight, power, and volume. The only viable options in this assembly involved the trace gas control portion of the trace contaminant control assembly.

Circulation, Temperature-Humidity, and Pressure Control

These subassembly units can be localized, centralized, or partly centralized and partly localized. Because of the possible configuration variations, the concept utilized in the EOSS was chosen as a basis for approximating the OLS weight, power, and heat rejection, and volume requirements shown in Figures 3.1-27, 3.1-28, and 3.1-29. The NR EOSS concept consisted of local circulation fans for ventilation and centralized ducting for temperature and humidity control. Pressure control also was centralized. The weights and volumes in Figures 3.1-27 and 3.1-28 are for a redundant system. This means that the centralized functions are each sized for 12 men and are fully redundant. One set is located in each of the two EOSS pressure volumes.

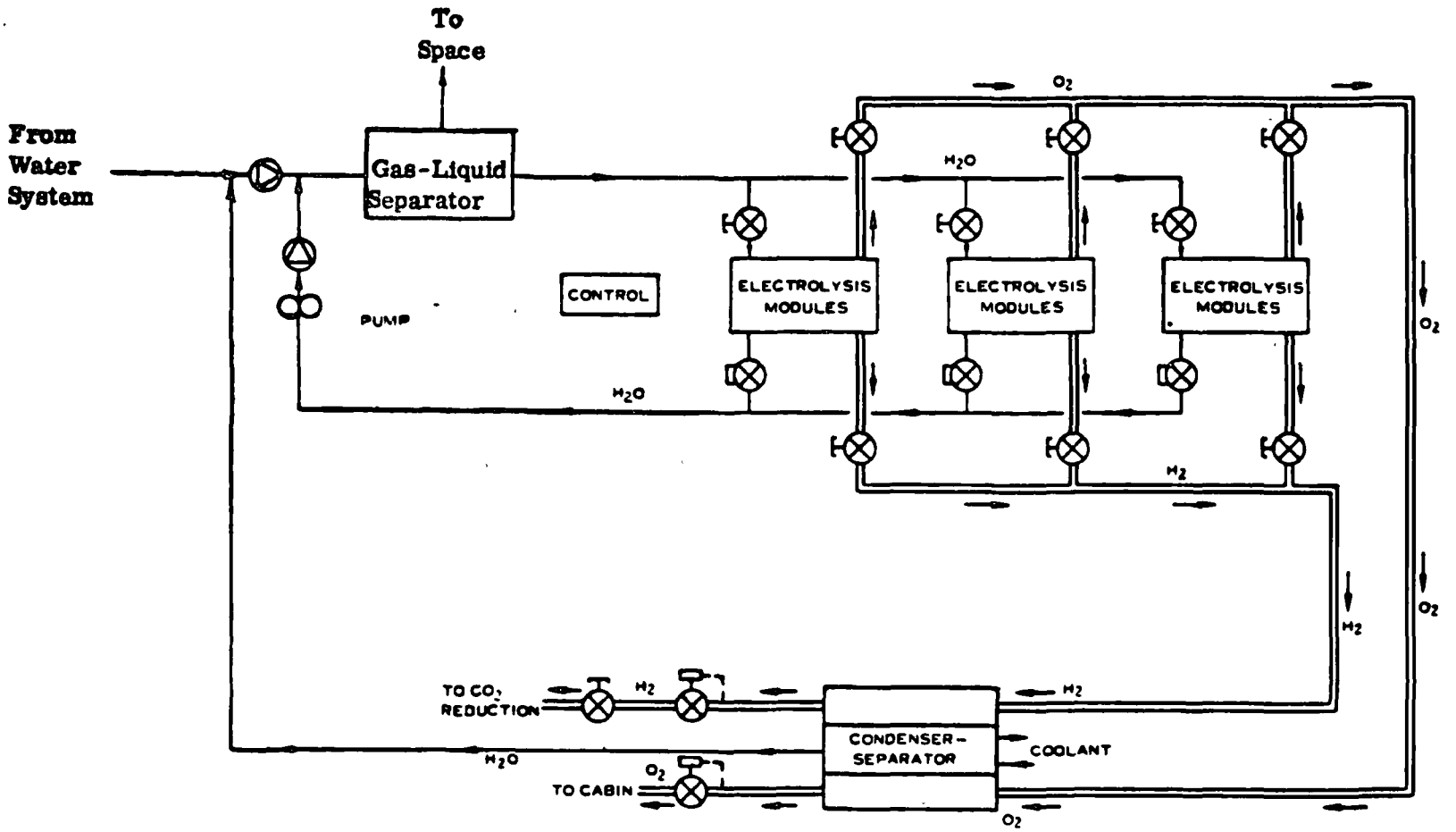


Figure 3.1-20. Wick Feed Concept

Space Division
North American Rockwell

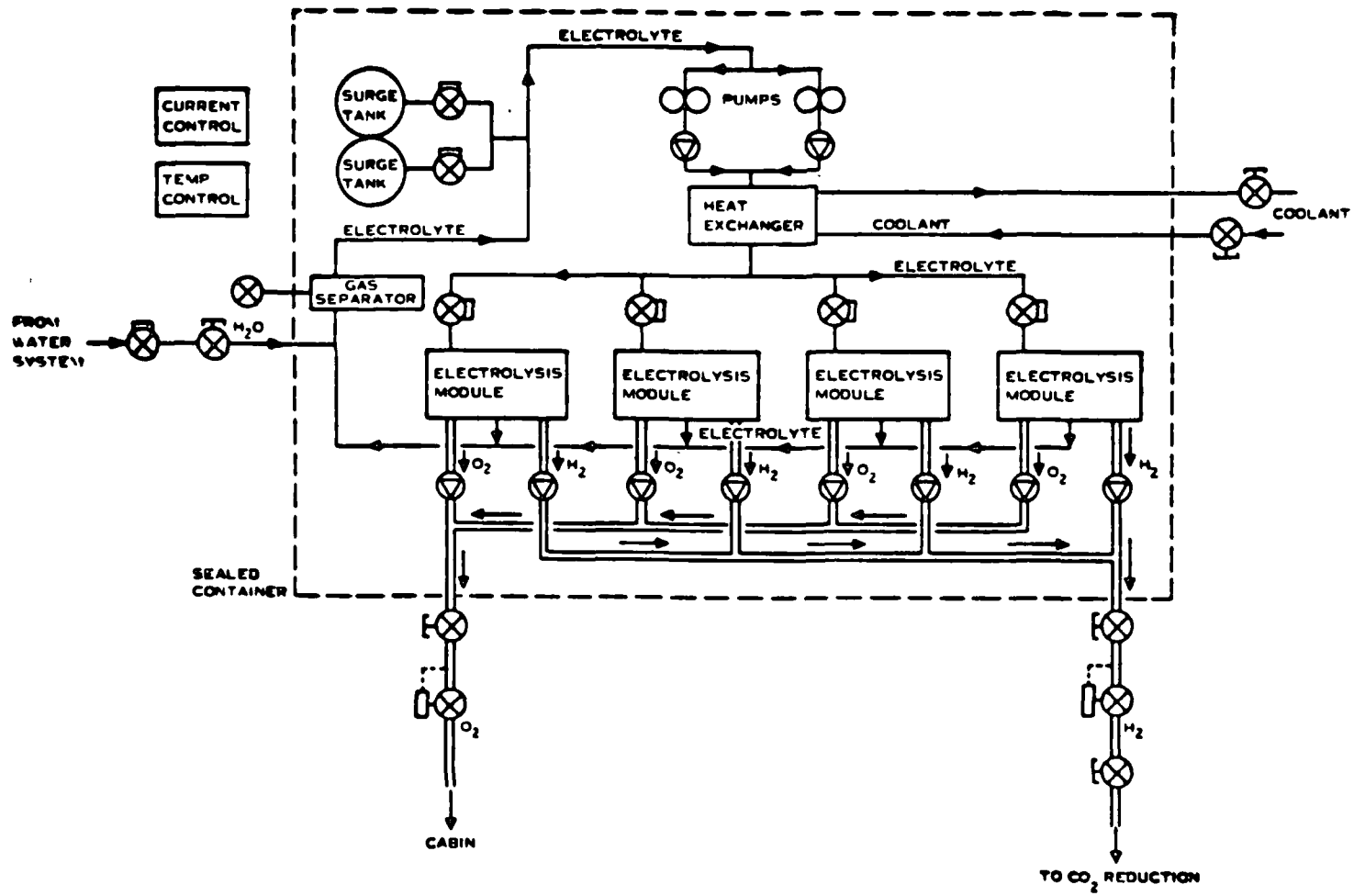


Figure 3.1-21. Circulating Electrolyte Concept

Space Division
North American Rockwell

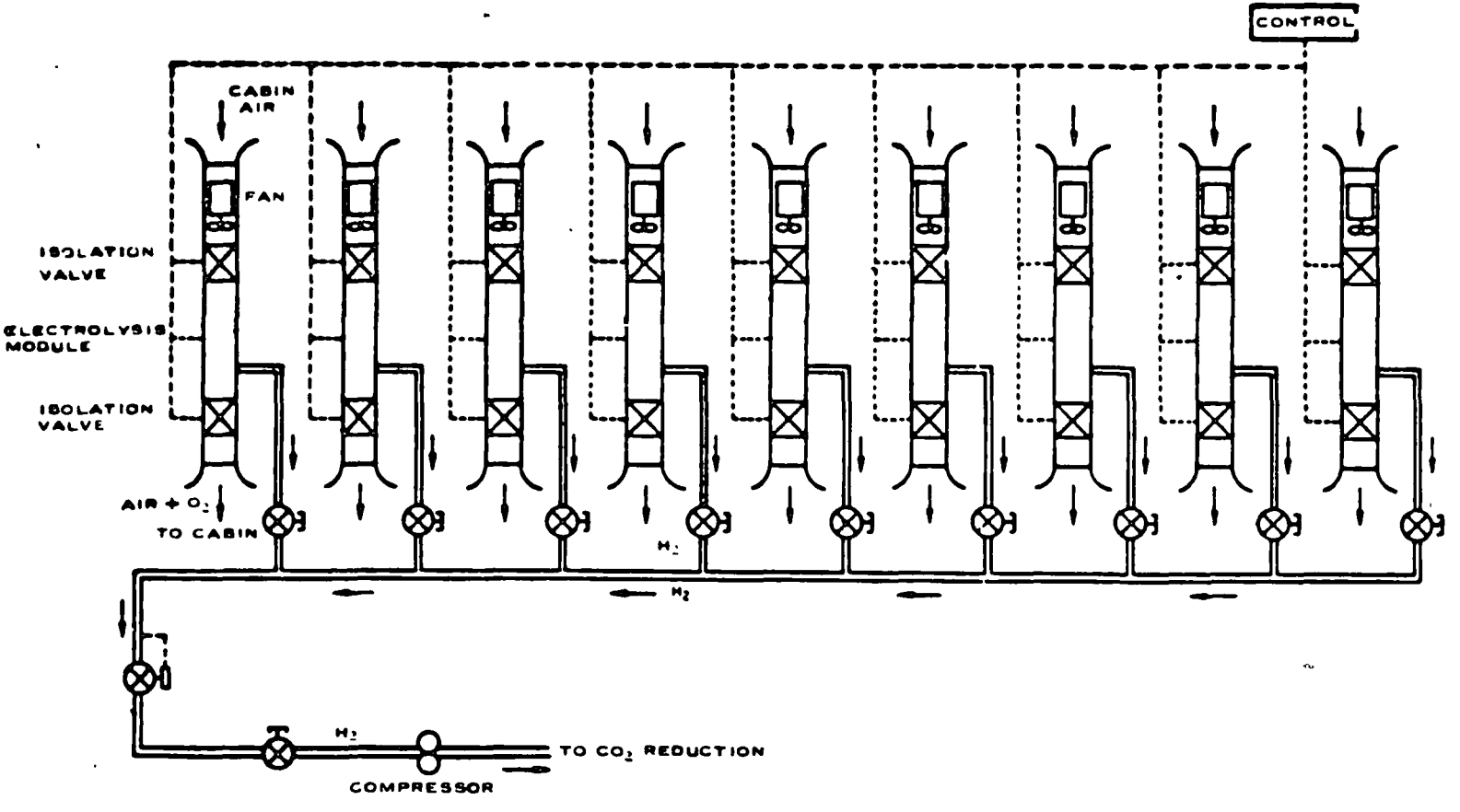


Figure 3.1-22. Cabin Air Concept

Space Division
North American Rockwell

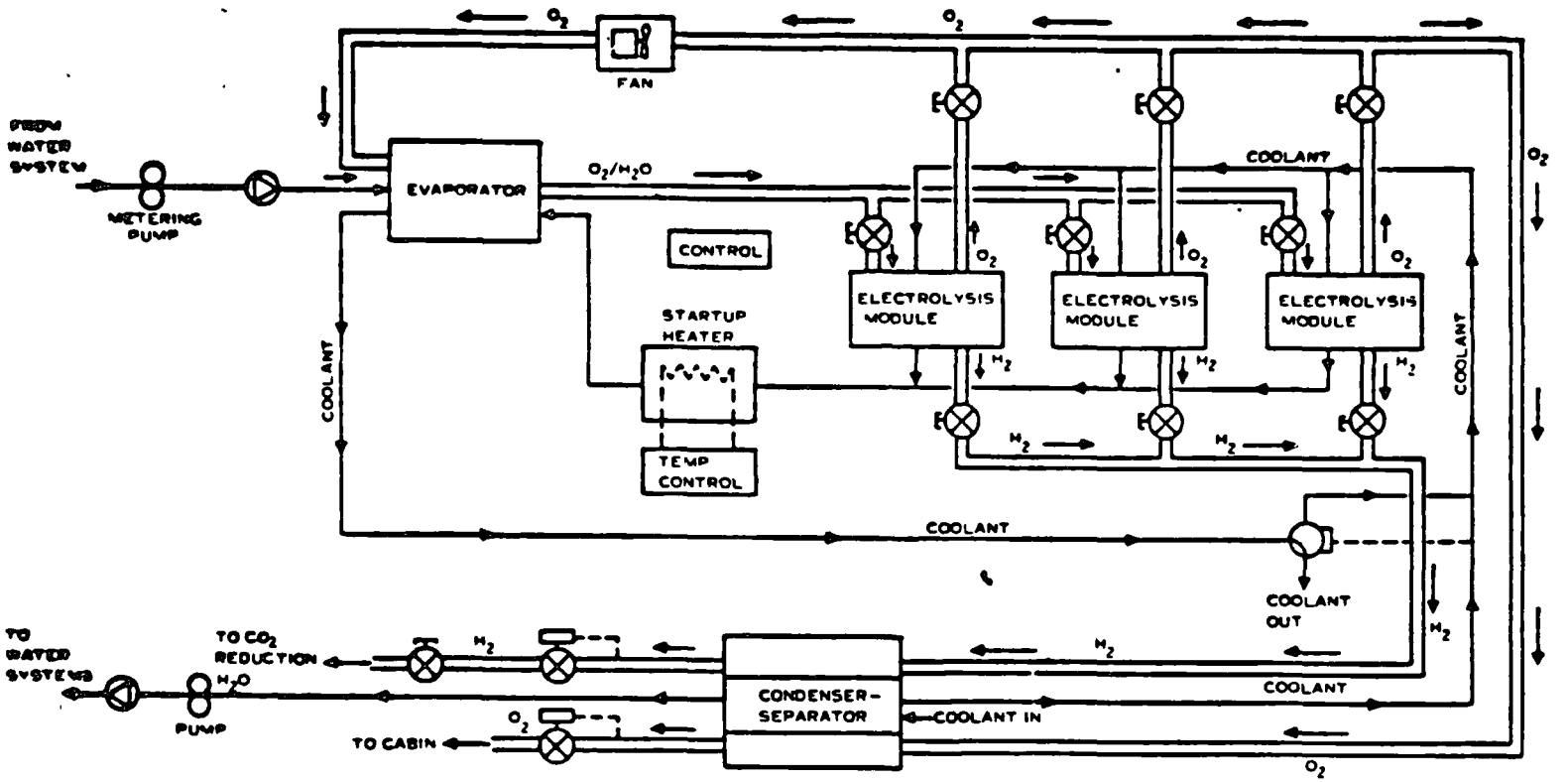


Figure 3.1-23. Gas Circulating Concept

Space Division
North American Rockwell

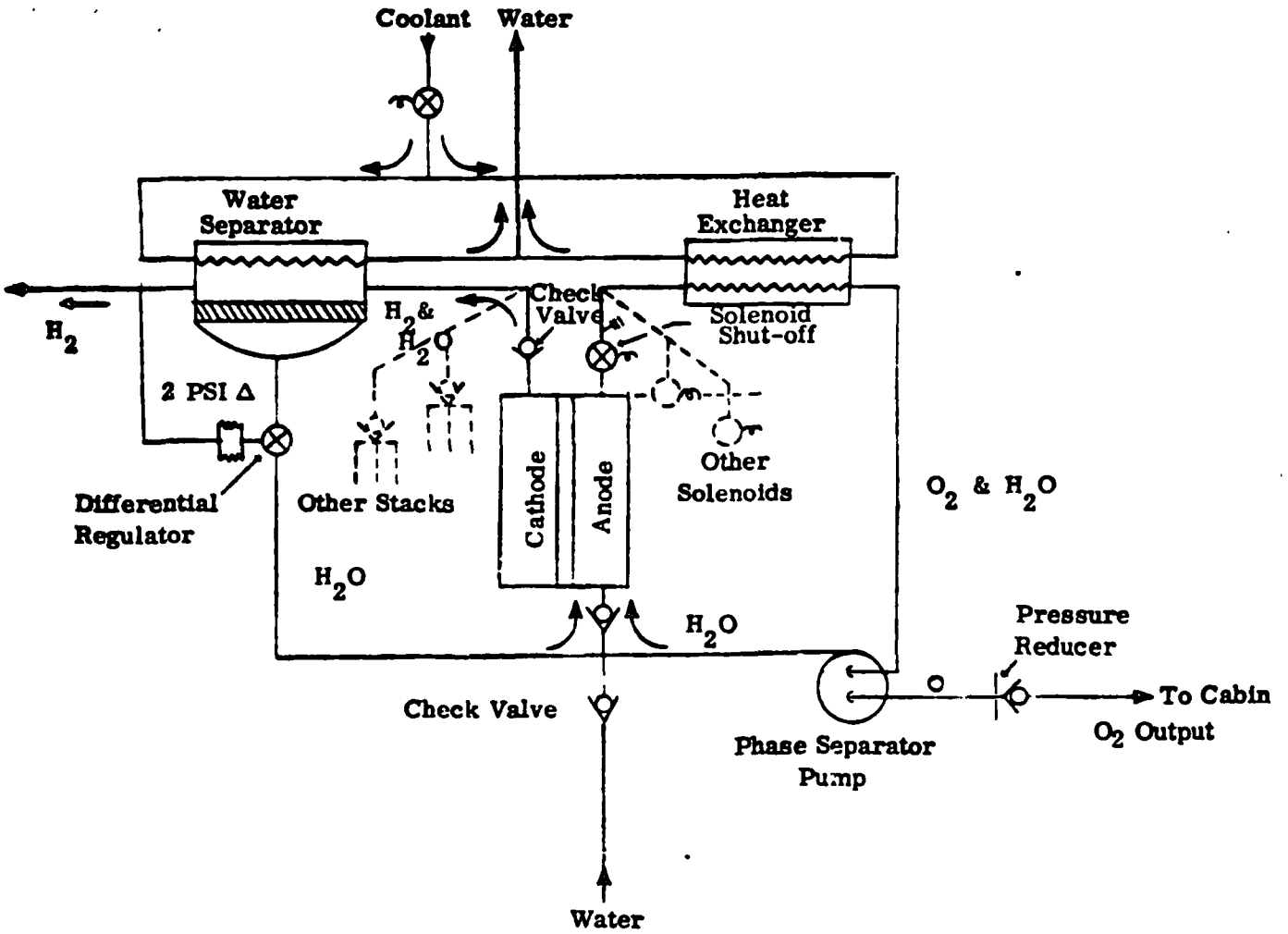
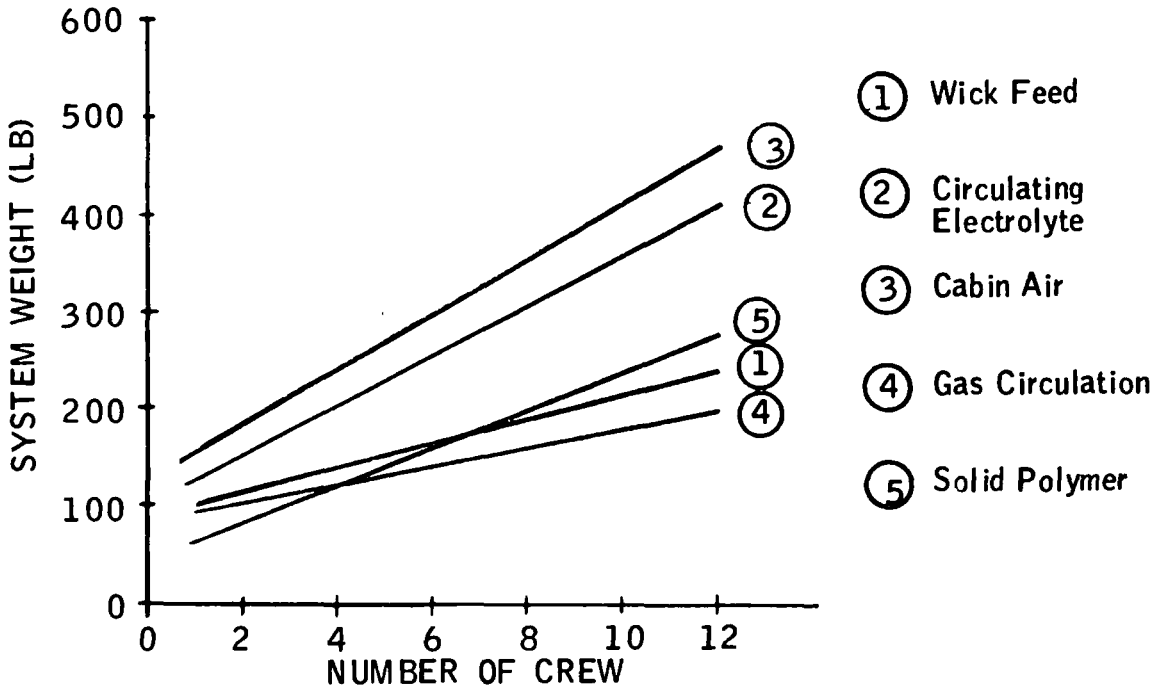


Figure 3.1-24. Water Electrolysis Solid Polymer Electrolyte

Space Division
North American Rockwell



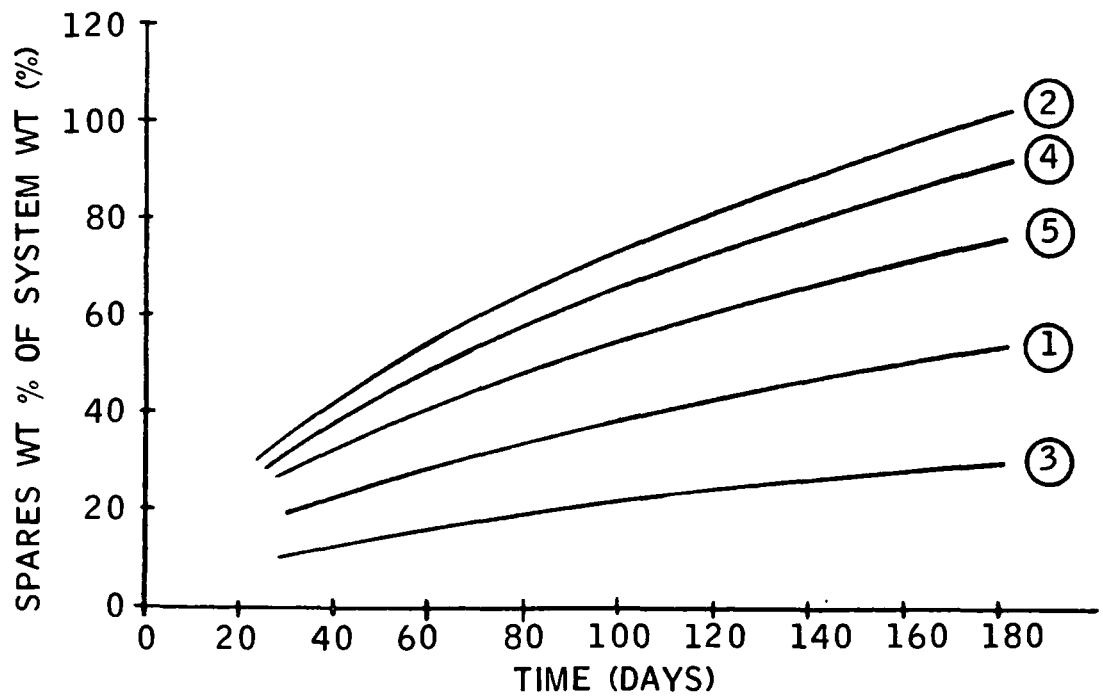
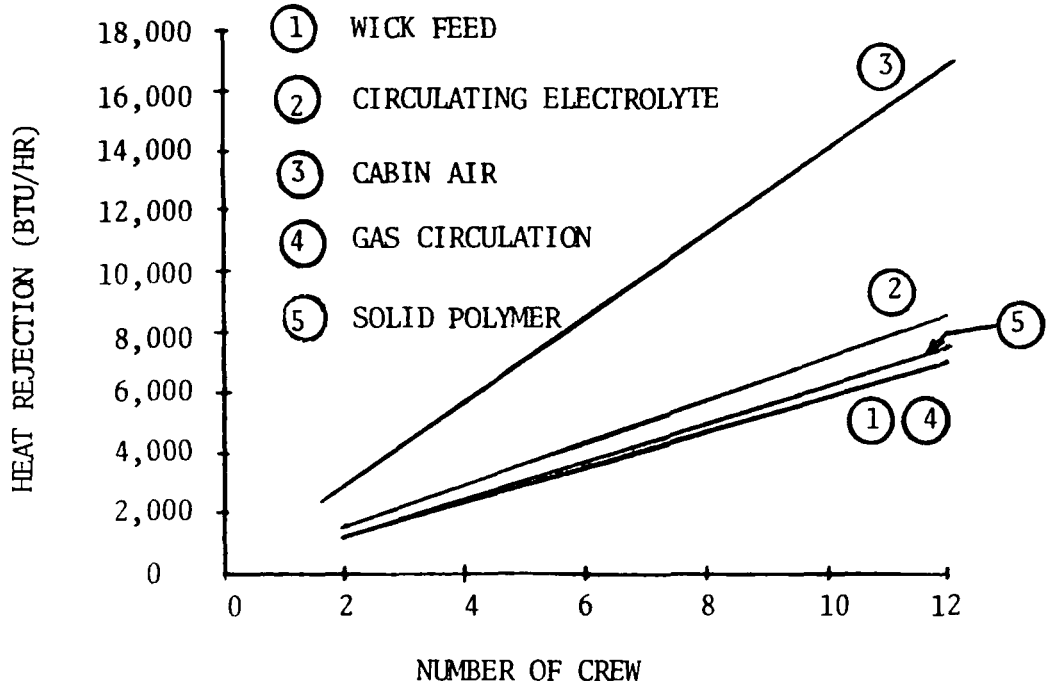


Figure 3.1-25. Water Electrolysis - System Weight Trade Study



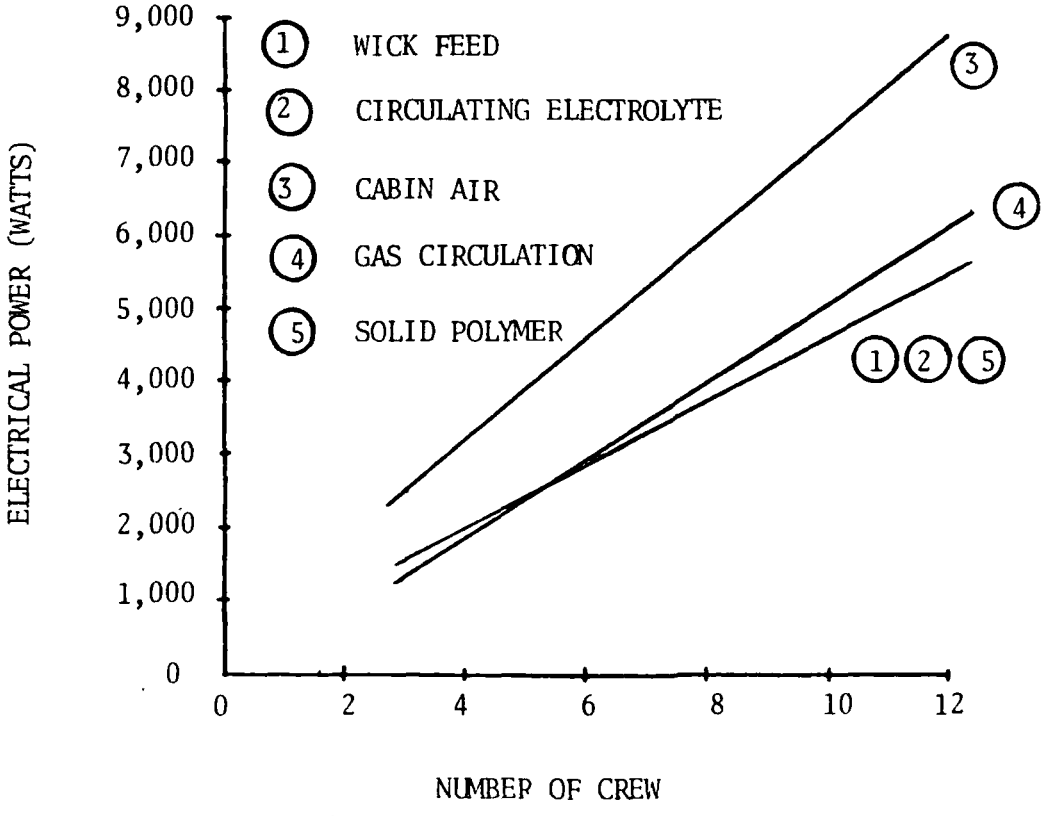


Figure 3.1-26. Water Electrolysis - Electrical Power and Heat Rejection Penalty

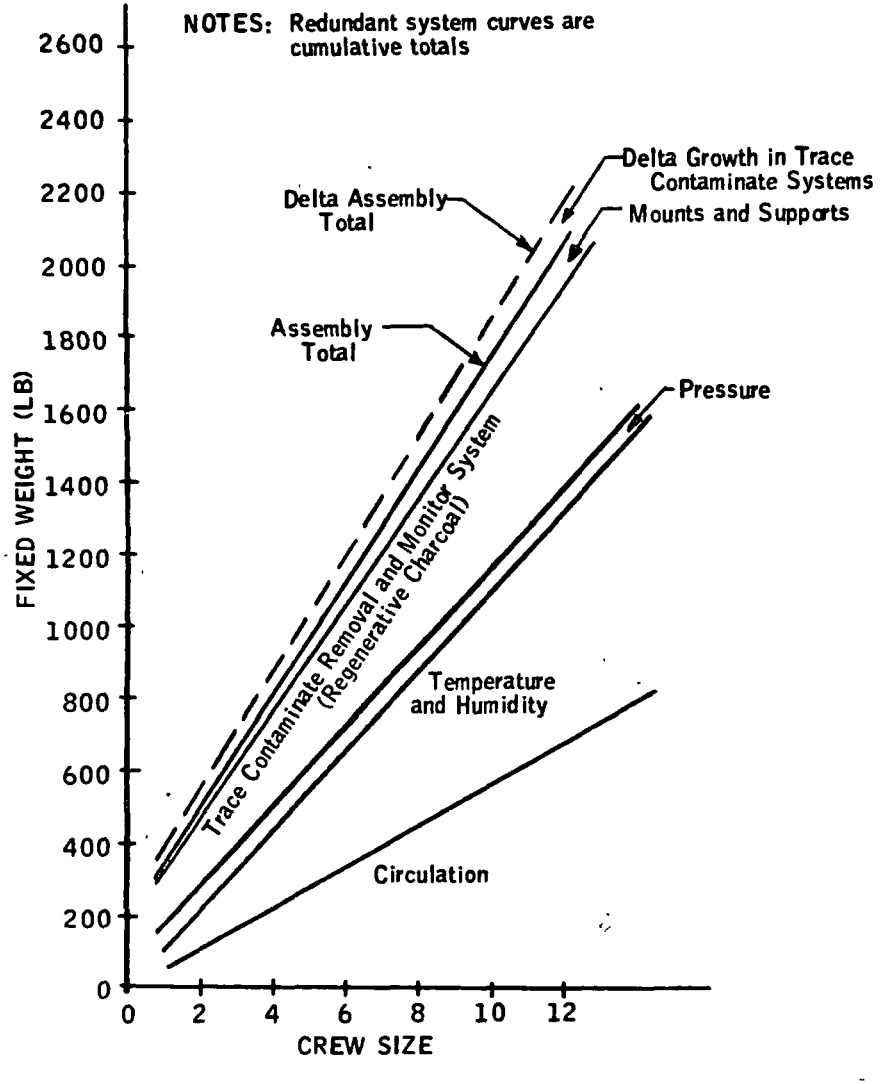


Figure 3.1-27. Atmospheric Control - Weight

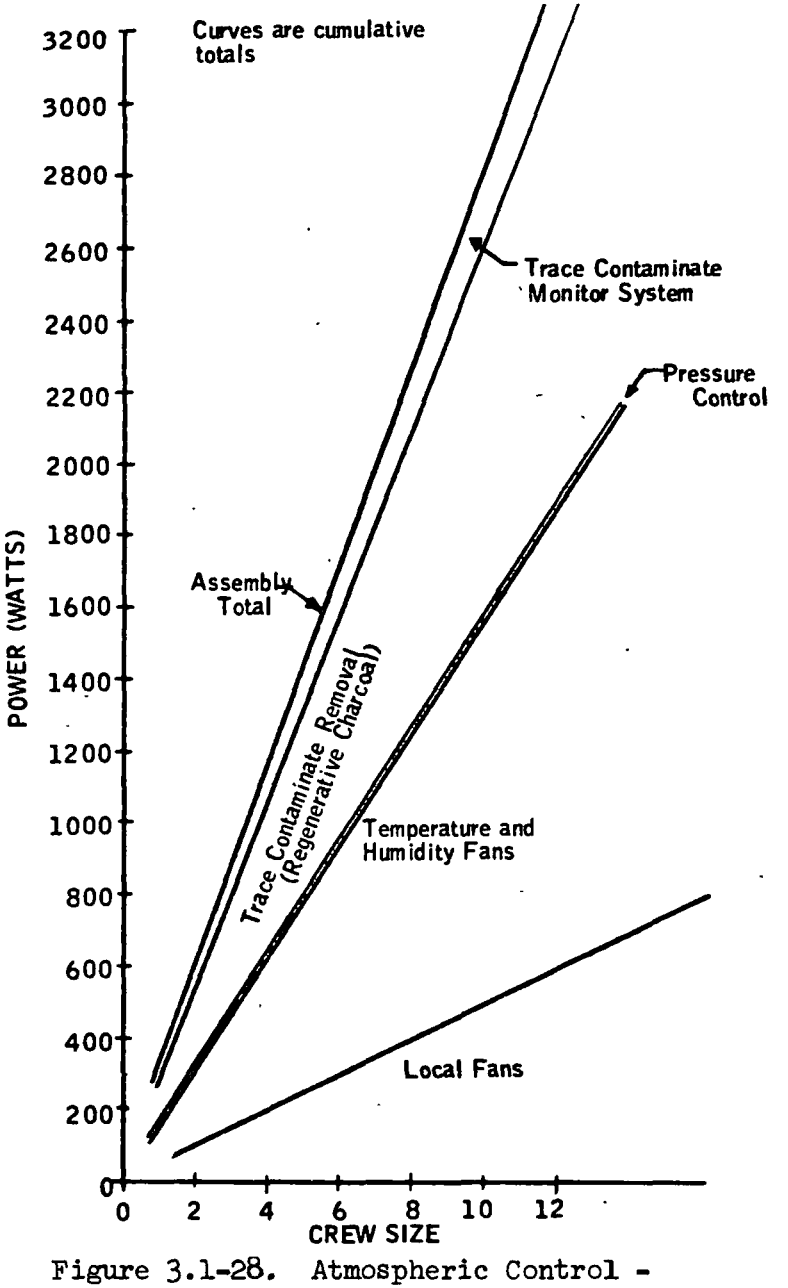


Figure 3.1-28. Atmospheric Control - Power

Space Division
North American Rockwell

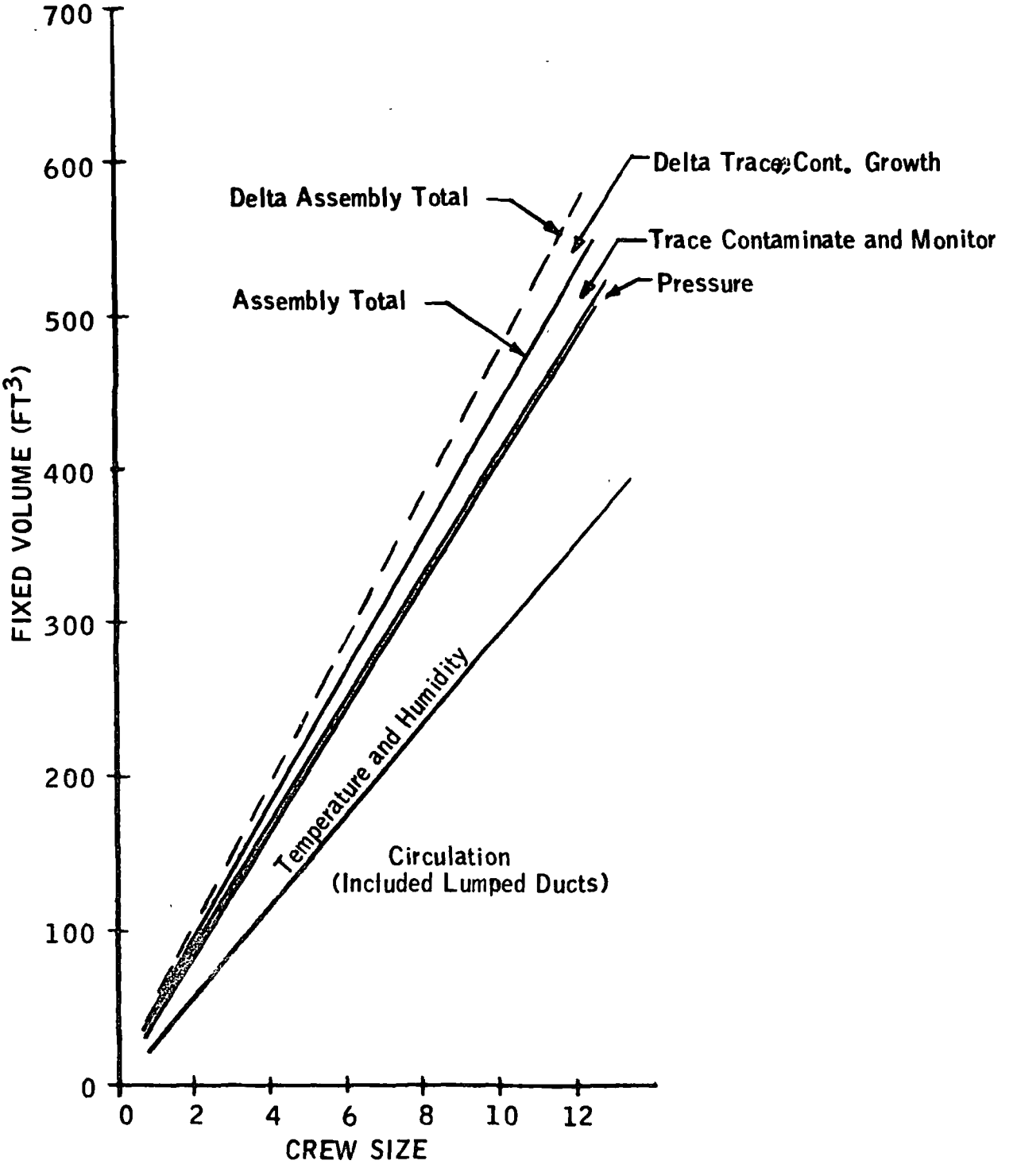


Figure 3.1-29. Atmospheric Control - Volume

The circulation subassembly unit includes local fans and the circulation ducts. The volume listed for circulation includes the lumped volume of the ducts. The temperature-humidity subassembly unit includes the main units plus some localized EOSS personal hygiene and exercise heat exchangers. The pressure control subassembly unit is a valving-regulator assembly that remains relatively constant with respect to crew size.

It should be re-emphasized that the weights, powers, and volumes shown in Figures 3.1-27 through 3.1-29 are only approximate. The vehicle configuration and air heat load profile can greatly change the characteristics of the circulation and temperature-humidity subassembly units.

Trace Contaminate Control

This subassembly unit consists of removal and monitoring of trace gases and bacteria. The monitoring system characteristics are based upon the EOSS, which it is anticipated will use a gas chromatograph and mass spectrometer for trace gas detection and an Agar tape-incubator-detector unit for bacteria detection. Bacteria removal is accomplished by filtration, while trace gases can be removed by several methods.

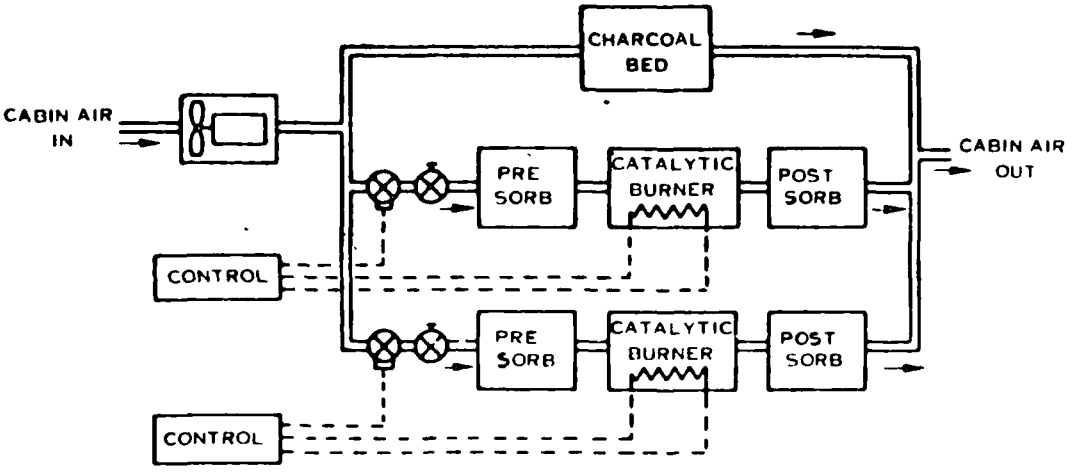
The options are shown in Figures 3.1-30 through 3.1-32. Each figure presents a schematic for two heat sources - radioisotope (or waste heat) and electrical. For the type of OLS missions envisioned (i.e., long duration), the nonregenerable charcoal and catalytic oxidation concept can probably be discarded because of its large resupply requirements. The EOSS selected a regenerable charcoal and catalytic oxidation concept followed closely by the catalytic oxidation-sorbent bed concept. Figures 3.1-27 through 3.1-29 include the EOSS regenerable charcoal concept. Figures 3.1-33 through 3.1-35 present the relative characteristics. It should be noted that a more detailed analysis of the EOSS regenerable charcoal concept caused a weight increase over that shown in Figure 3.1-33. Thus, a delta weight and volume curve is shown in Figures 3.1-27 and 3.1-29 to illustrate the subassembly growth. Detailed analysis of the other concepts was not performed, thus the old regenerable charcoal characteristics were used in Figures 3.1-33 through 3.1-35 for relative comparisons. The weights, etc. in Figures 3.1-33 through 3.1-35 are for single units. The approximate density factor for Figure 3.1-33 is 17.8 lb/ft^3 for the system weight and 33.2 lb/ft^3 for the spares.

For atmospheric storage as a whole using the EOSS data points (12 men), the initial 180-day spares are 16 percent of the fixed system weight. EOSS data indicate expendables and resupply spares would vary approximately as shown in Table 3.1-5.

Table 3.1-5. Six-Month Resupply Spares and Expendables Weight

Men	4	8	12
Resupply spares	37	75	112
Expendables	71	141	212

ELECTRICAL POWER



RADIOISOTOPE HEAT SOURCE

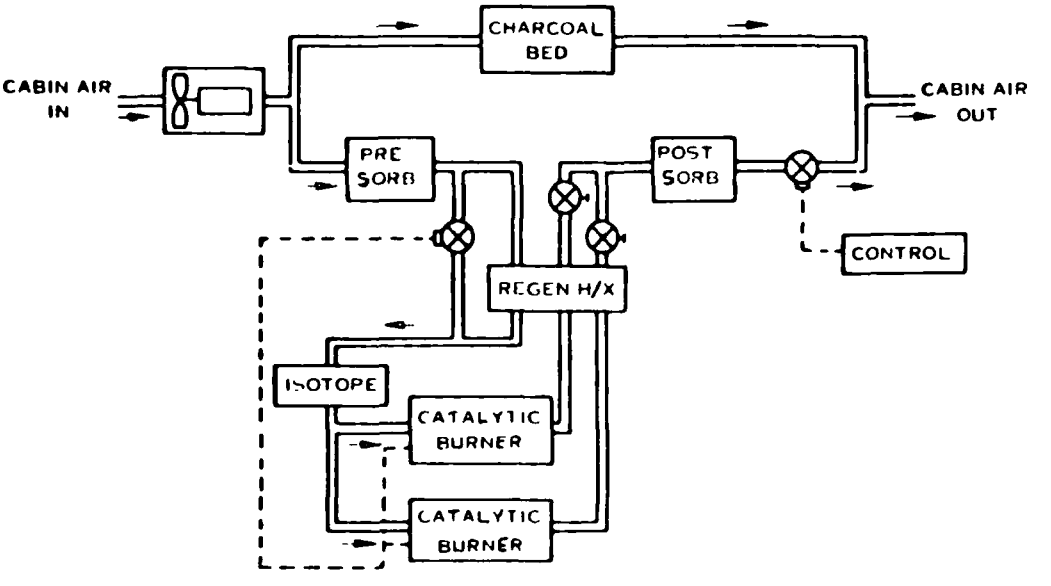


Figure 3.1-30. Nonregenerable Charcoal/Catalytic Oxidation

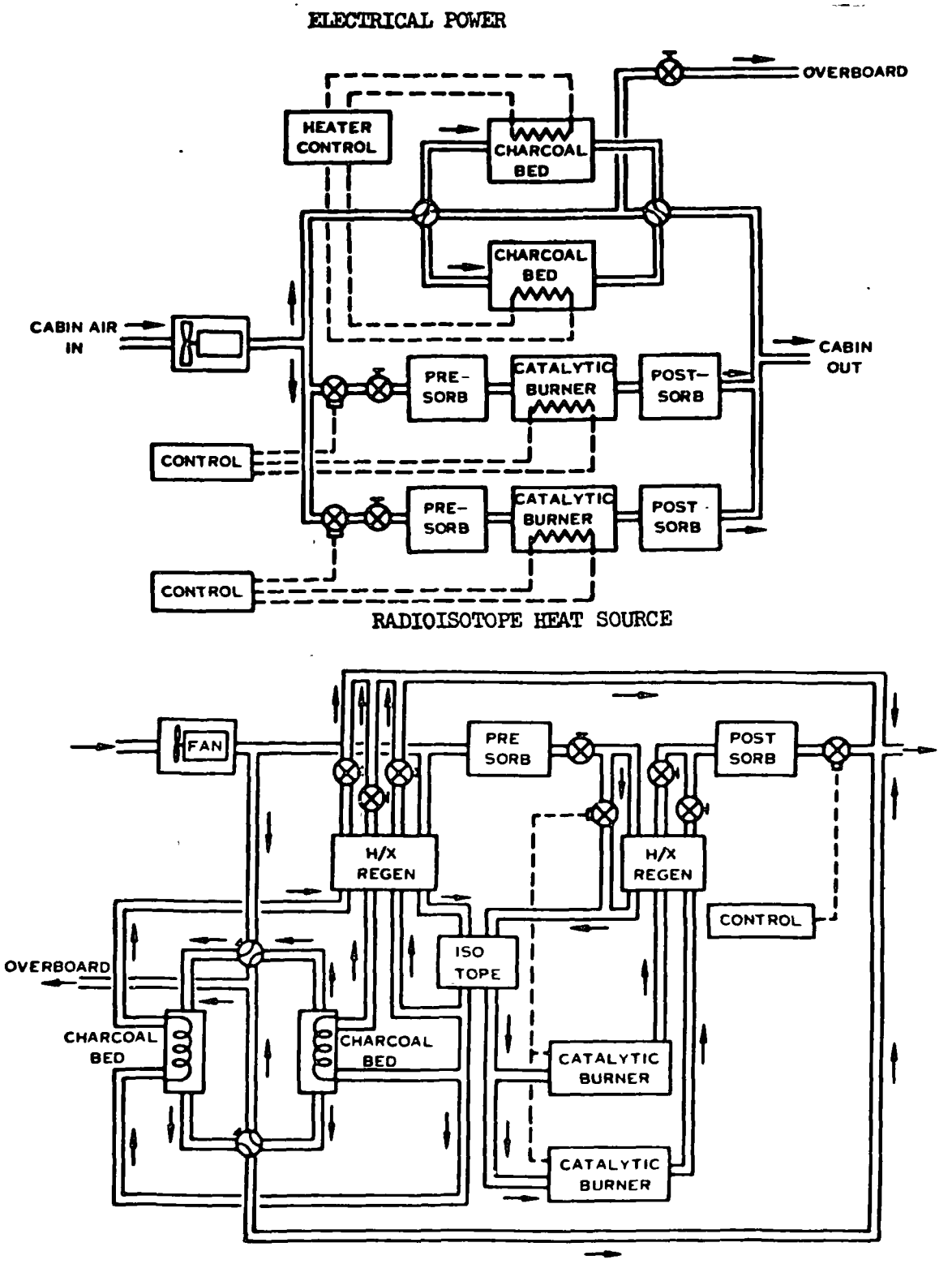
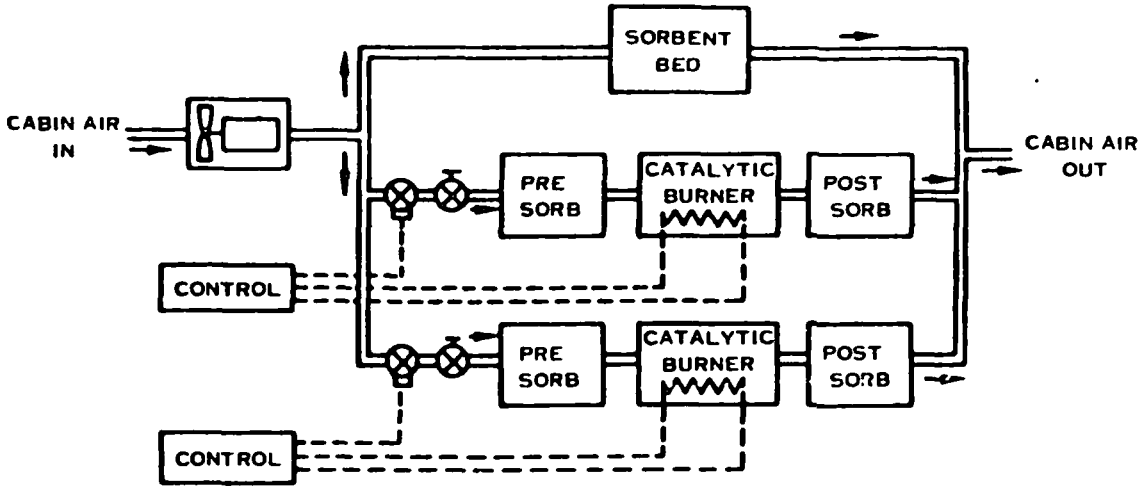


Figure 3.1-31. Regenerable Charcoal/Catalytic Oxidation

ELECTRICAL POWER



RADIOISOTOPE HEAT SOURCE

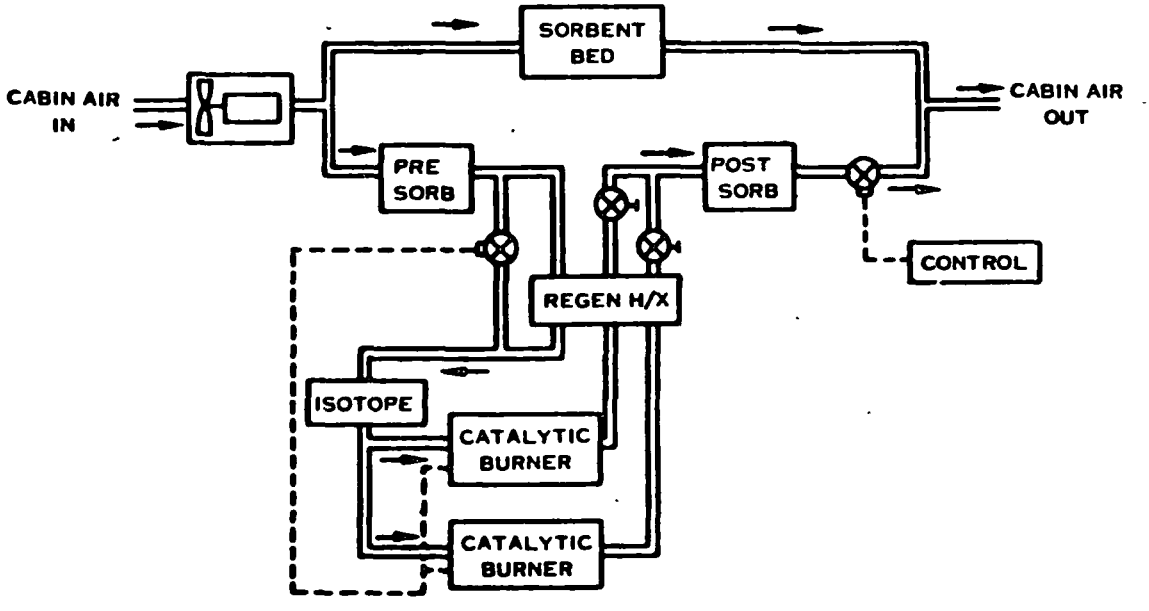


Figure 3.1-32. Catalytic Oxidation/Sorption

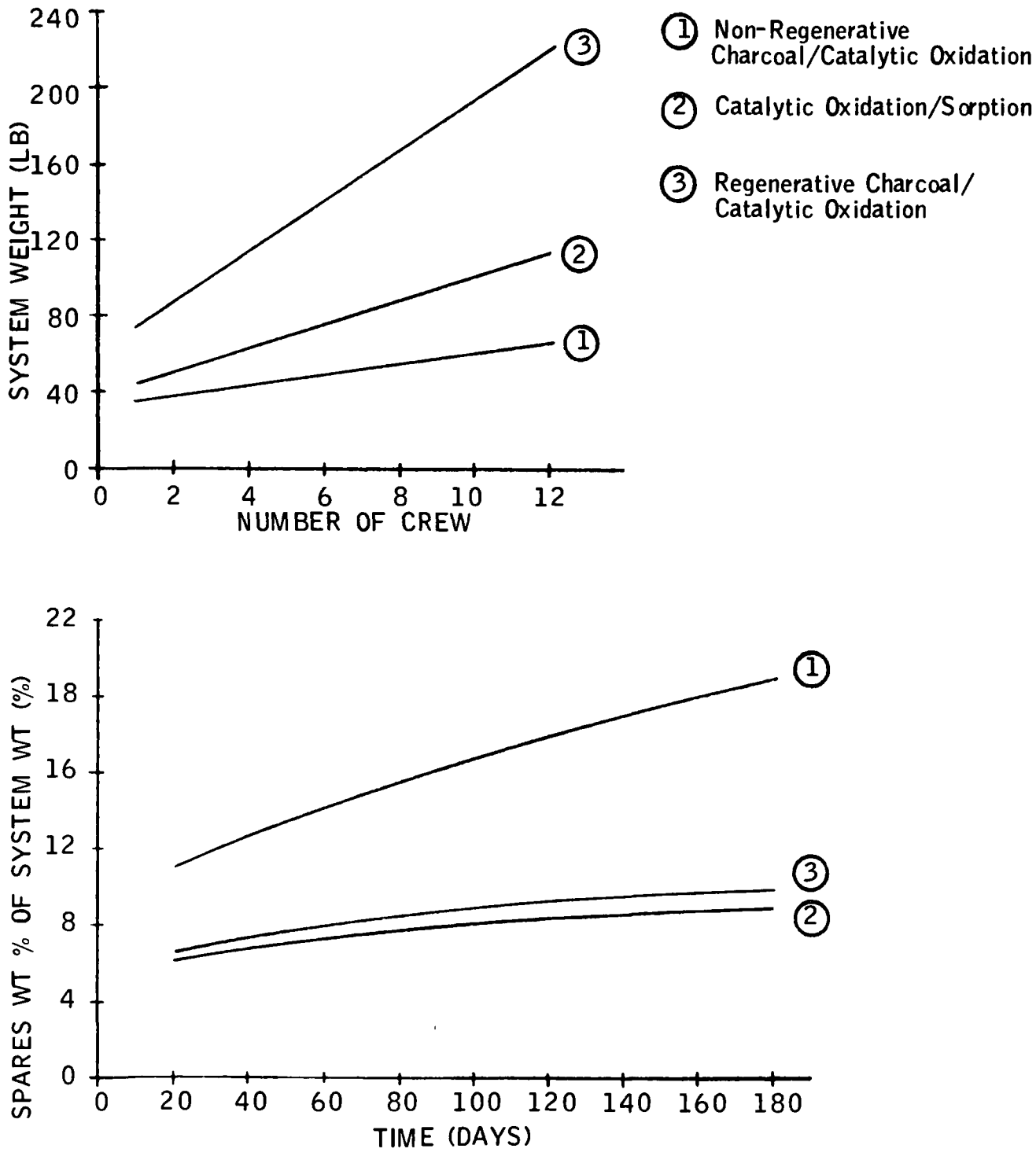
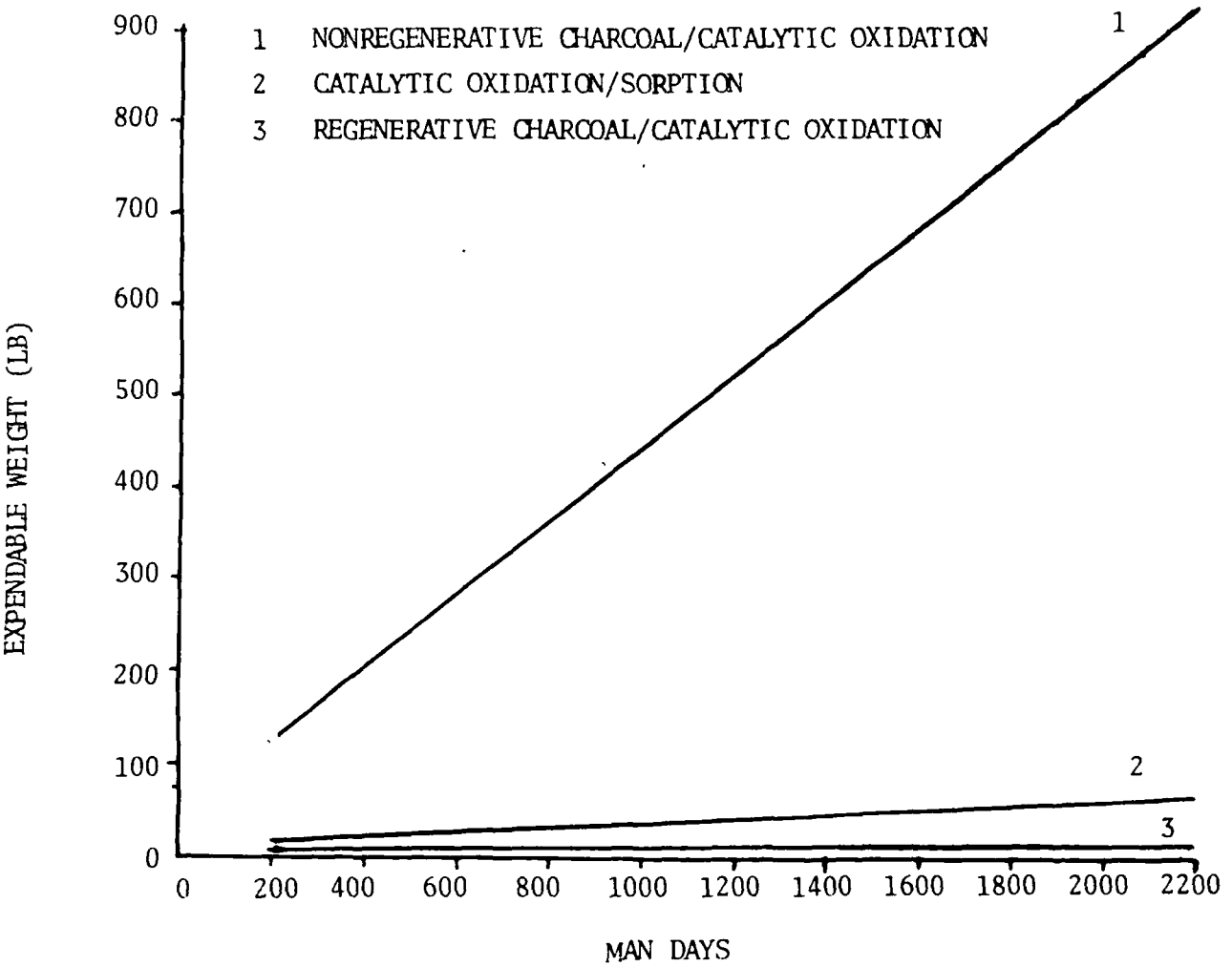
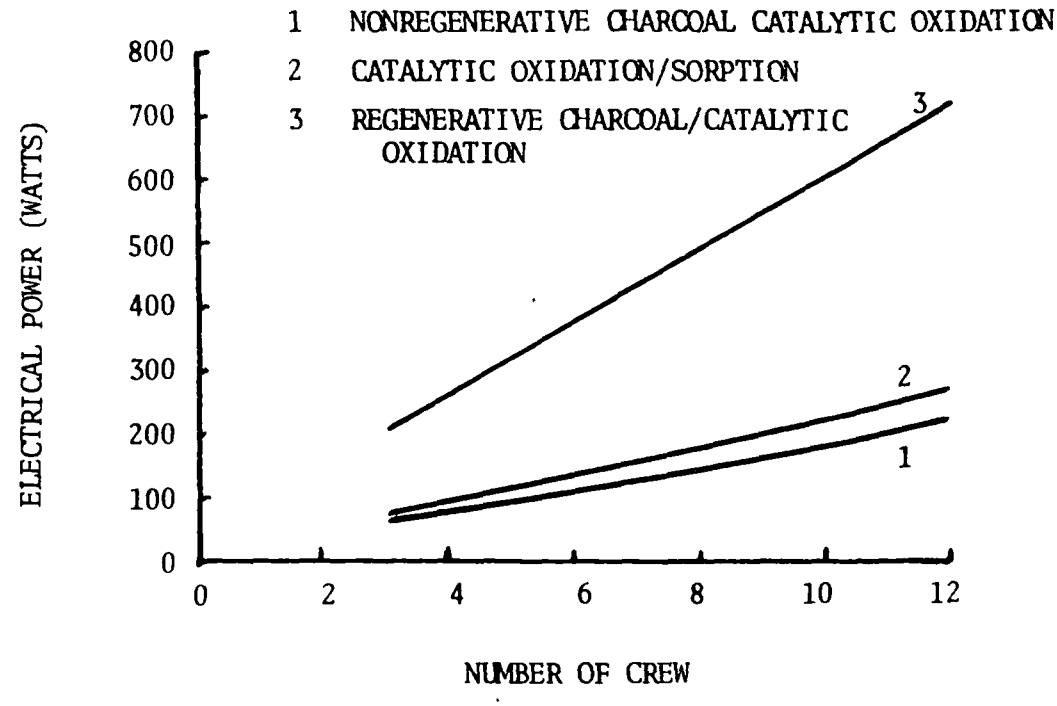


Figure 3.1-33. Contaminant Control - Weight Trade Study



NOTE: EXPENDABLES INCLUDE SUCH ITEMS AS FILTERS, CATALYSTS OR CHEMICALS NORMALLY PROGRAMMED FOR REPLACEMENT.

Figure 3.1-34. Contaminant Control - Expendables Weight Trade Study



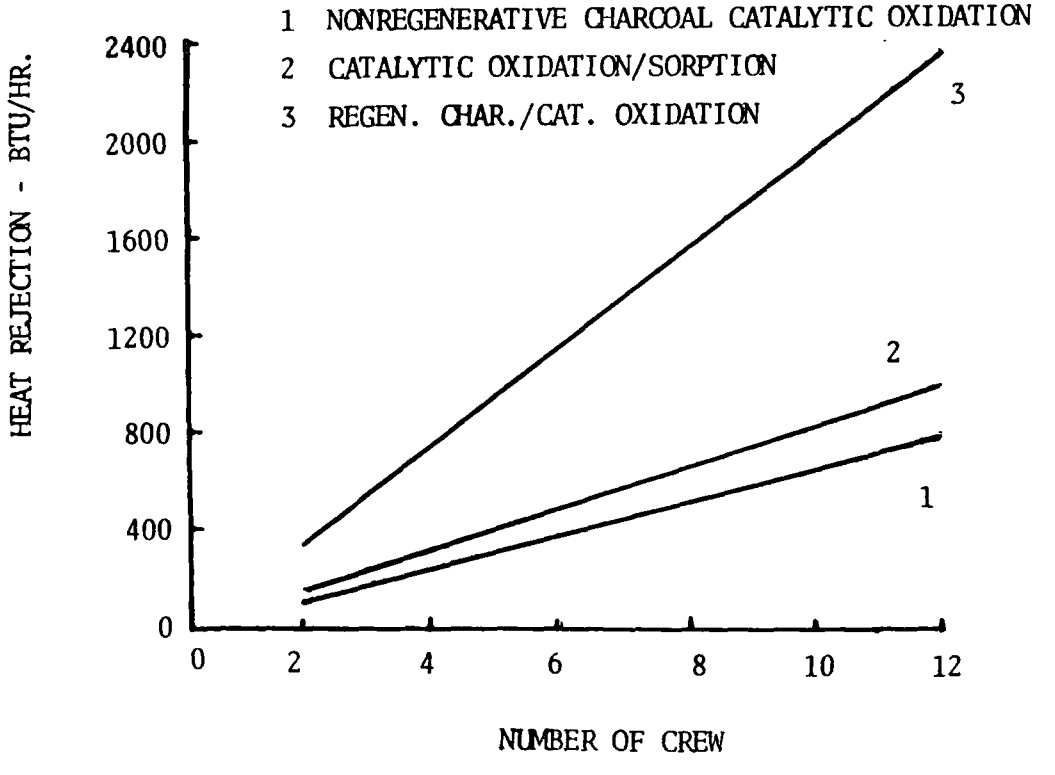


Figure 3.1-35. Contaminant Control - Electrical Power and Heat Rejection Penalties

3.1.5 ACTIVE THERMAL CONTROL

Section 3.6.1 presents parametric data for both passive and active thermal control concepts. Consideration of active thermal control concept characteristics only is given here. In this section, it is envisioned that the OLS active thermal control will consist of an active thermal control concept with at least two fluid loops: a cabin-cooling loop and a heat rejection loop. If waste heat is available, then a third waste heat loop may be added. However, for waste heat to be useable, it must be on the order of 300 F.

Thermal Loops

OLS requirements being similar to EOSS requirements in this area the characteristics of the OLS cabin, heat rejection, and waste heat loops are similar to those employed in the EOSS. The cabin loop is a water loop and includes coldplates, plumbing, heat exchangers, and pumps. The heat rejection loop is a freon loop, which includes the interchange heat exchangers between the water and freon. The heat rejection loop includes pumps and freon reservoirs. Pumps have redundancy in the various loops. The heat rejection loop is 100 percent redundant, while the cabin loop is only partially redundant. The partial redundant loop was designed for the critical components.

The waste heat loop (shown as a delta weight) is redundant in pumps and interchange heat exchangers with the waste heat source, and it is redundant for the components it serves because they are redundant.

Figures 3.1-36 and 3.1-37 present characteristics of the anticipated OLS active thermal loop. All thermal control loops are configuration and power-load sensitive. Vehicle configuration and equipment locations determine pipe runs, pressure drops, and pump power. The radiator is discussed later in this section. It is sufficient to say here that two weight penalties can be assigned to the radiator, one with and one without the inclusion of meteoroid protection.

Radiator

The net amount of heat arising from electrical power inputs, metabolic inputs, and vehicle environmental heat gains (or losses) determines the total heat that must be rejected. Experience from the EOSS and other manned space stations indicate radiators usually cover a large portion of the available surface area so that vehicle heat gain or loss is minimized.

Figure 3.1-38 schematically illustrates the two extreme types of polar orbits with three different vehicle flight modes. The vehicle flight modes, in this preliminary analysis, does not significantly affect the heat rejection. Only the orbit was examined. The terminator orbit is preferred over the subsolar orbit from a heat rejection viewpoint. However, propellant will be required to hold a terminator orbit. If the OLS is held in a fixed orbit relative to the moon, the earth-moon-sun system will cause the OLS to vary from terminator to subsolar orbits in approximately ninety days. For design the worst-case subsolar orbit will be assumed.

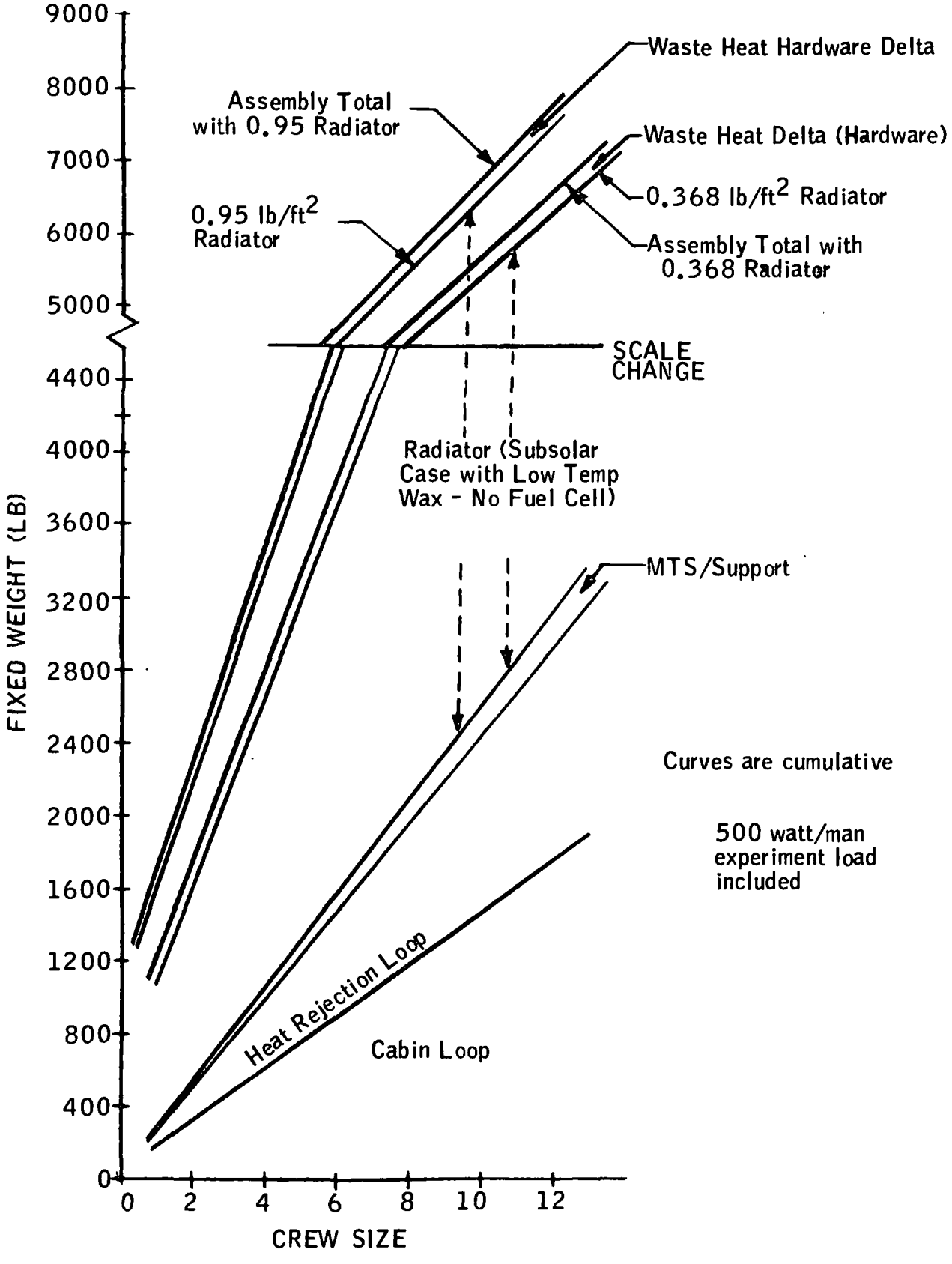


Figure 3.1-36. Active Thermal Control - Fixed Weight

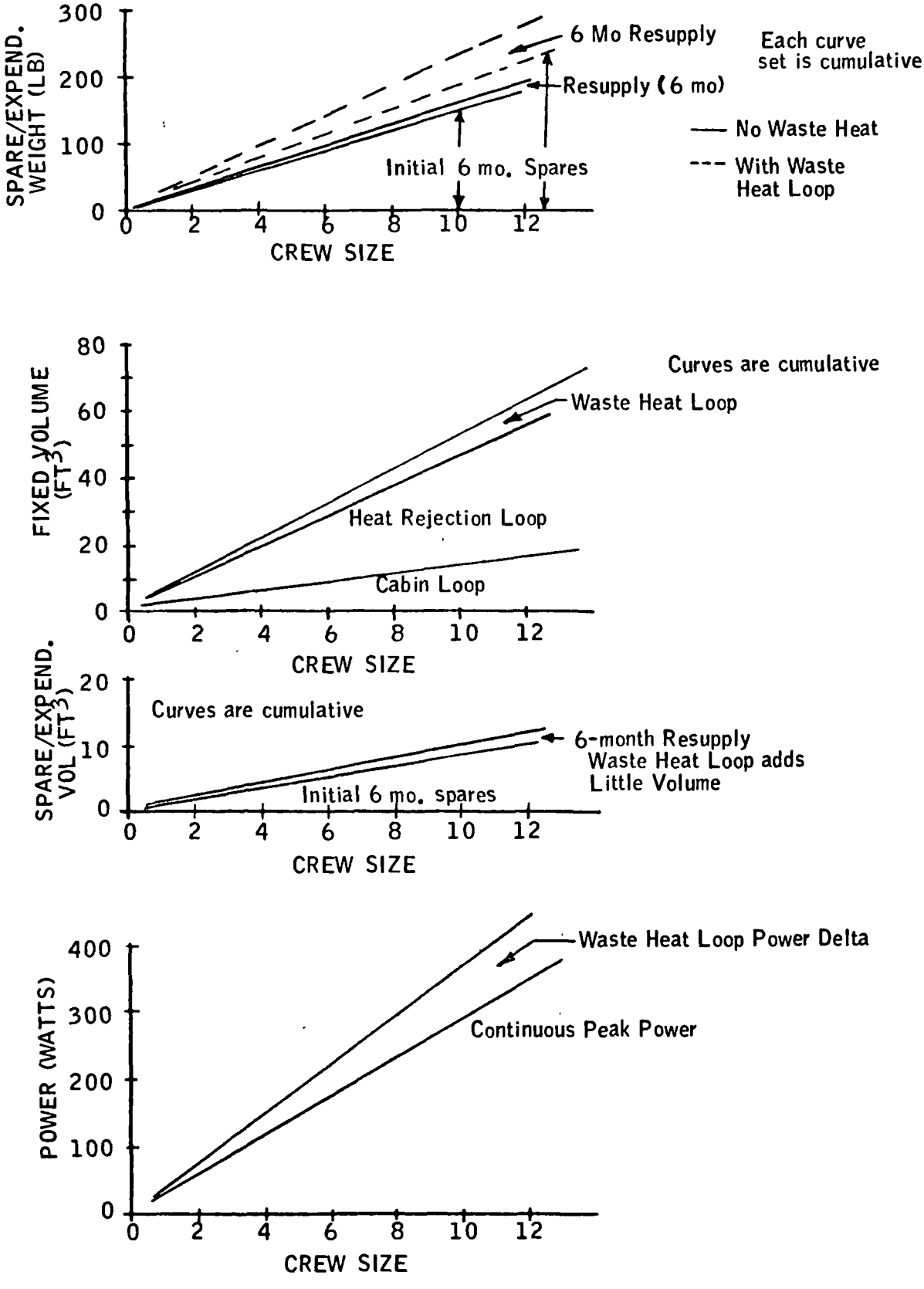
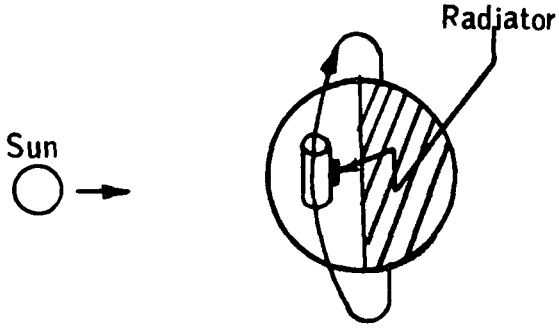


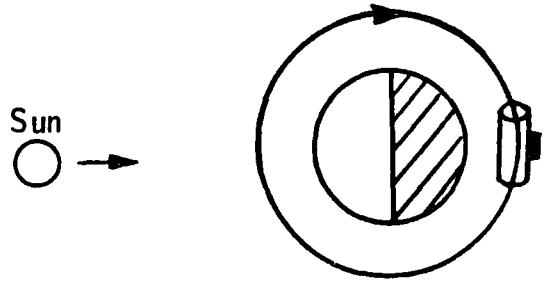
Figure 3.1-37. Active Thermal Control

TERMINATOR CASE

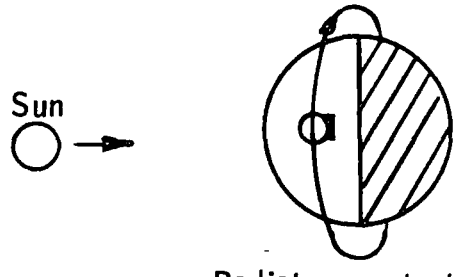
SUBSOLAR CASE



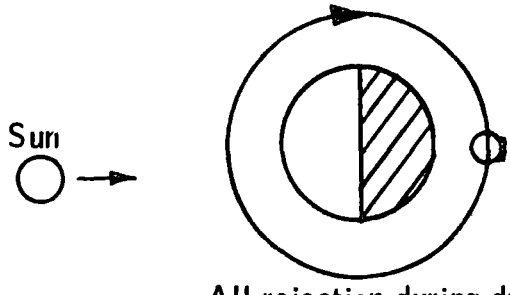
Radiator constantly sees deep space. One-half of cylinder available.



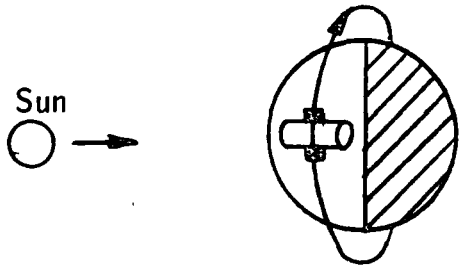
All rejection during dark side of orbit. Need heat sink during light side. Total cylinder available for each 1/2 orbit.



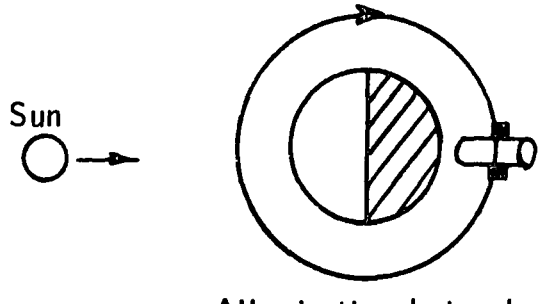
Radiator constantly sees deep space. One-half of cylinder available.



All rejection during dark side of orbit. Need heat sink during light side. Total cylinder available for heat rejection during dark side of orbit



One-half of cylinder available. Half of cylinder sees deep space.



All rejection during dark side of orbit. Need heat sink during light side. Total cylinder available for heat rejection during dark side of orbit.

Figure 3.1-38. Vehicle Orientation for Radiation - Polar Orbit

The analysis used for determining the required radiator area assumed a cylinder that rejected to deep space. No attached modules or other obstructions were considered. Rejection was to deep space because the low temperature (55 F) ECLSS radiators cannot reject to the hot sunlit moon or to the sun.

Six cases were studied for radiator sizing. Three of the cases assumed that all radiation would be low temperature (55 F). However, if fuel cells are used for EPS power, some high temperature (180 F) radiation is required. Thus, the same three cases were repeated with the high-temperature radiator requirement added. Table 3.1-6 summarizes the concepts. It should be noted that the heat sink material only affects system weights and not radiator area. In subsection 3.6 both 55 F and 70 F active as well as 70 F passive radiator systems are considered. Note that 70 F systems require a heat pump (refrigeration loop).

Table 3.1-6. Lunar Orbit Radiator Concepts

Polar Orbit Type	Concept Description
Terminator	Low-temperature radiator constantly views and radiates to deep space. No heat storage required. Propellant required to hold orbit.
Terminator	Low-temperature radiator and high-temperature radiator constantly view and radiate to deep space. No heat storage required. Propellant required to hold orbit.
Subsolar	Low-temperature radiator views and radiates to deep space during dark side of orbit. During sun side of orbit, ice is used as a heat sink.
Subsolar	Low- and high-temperature radiators view and radiate to deep space during the dark side of the orbit. During the sun side of the orbit, ice is used as a heat sink.
Subsolar	Low-temperature radiator views and radiates to deep space during dark side of orbit. During sun side of orbit, tetradecane compound is used as a heat sink.
Subsolar	Low- and high-temperature radiators view and radiate to deep space during dark side of orbit. During sun side of orbit, tetradecane is used for the low-temperature heat sink and transit heat 90 is used for the high-temperature heat sink.

Figure 3.1-39 presents the radiator areas for the concepts given in Table 3.1-6. The radiator areas represent the net area that must see deep space to reject the required heat loads. The addition of the high-temperature radiators to the low-temperature ones increased the net required area because more heat is being rejected. Also, the subsolar cases only reject during the dark period of the orbit, thereby increasing the radiator level to twice that for continuous rejection (assuming equal dark and light periods during the orbit). The radiator can cover any part of the cylinder for the subsolar cases. For the terminator cases, the area must not be on the side that views the sun so that only one-half to two-thirds of the cylinder area is available.

Figures 3.1-40 and 3.1-41 present radiator weights. The weights include the dry radiator weight and, where applicable, the weight of the heat sink material such as ice or equivalent heat of fusion material. No penalties for packaging or plumbing of the heat sink materials were included. Figure 3.1-40 has a radiator penalty of 0.95 lb/ft^2, which includes radiator tubes, braze, connectors, manifolds, paint, meteoroid bumper and fins, stiffeners, tube meteoroid protection, reinforcements, closeouts, and supports. Figure 3.1-41 has a radiator penalty of 0.368 lb/ft^2 and includes all the items for the 0.95 lb/ft^2 radiator, except meteoroid bumper and fins, stiffeners, reinforcements, closeouts, and supports.

An analysis initiated during the EOSS study was adapted to the OLS to observe the effect of a refrigeration loop to raise the 55 F radiator temperature to 140 F. Tables 3.1-7 and 3.1-8 were compiled for comparison purposes. Table 3.1-7 uses a radiator weight penalty of 0.95 lb/ft^2; Table 3.1-8 uses a penalty of 0.368 lb/ft^2. The 140 F radiator can only reject 41 $Btu/hr\text{-}ft^2$ to space when the radiator is oriented towards the sun, compared to the heat rejection to deep space of 200 $Btu/hr\text{-}ft^2$. Thus, a higher temperature refrigeration loop radiator may be desired. Possibly, the refrigeration loop could be used to enable integration of the ECLSS radiator with the fuel cell radiator.

Tables 3.1-7 and 3.1-8 show the effect of the refrigeration loop on radiator area and weight. With a higher heat flux, the radiator area is decreased. However, the added power of the refrigeration system increases the amount of heat rejection required. The radiator weight increases because of the condenser, evaporator, compressor, and plumbing of the refrigeration loop, which is added to the radiator weight. (Ammonia weight is excluded.) The last two columns of the tables present the percentage increases and decreases associated with the conversion of a low temperature 55 F radiator to a high temperature 140 F radiator and refrigeration loop.

3.1.6 WATER MANAGEMENT

The water management subassembly interfaces with waste management, hygiene, food management, and CO_2 management. This subassembly receives various waste waters, processes these waters, and then stores and provides fresh water on demand. A single or multiple concept system can be used for water recovery.

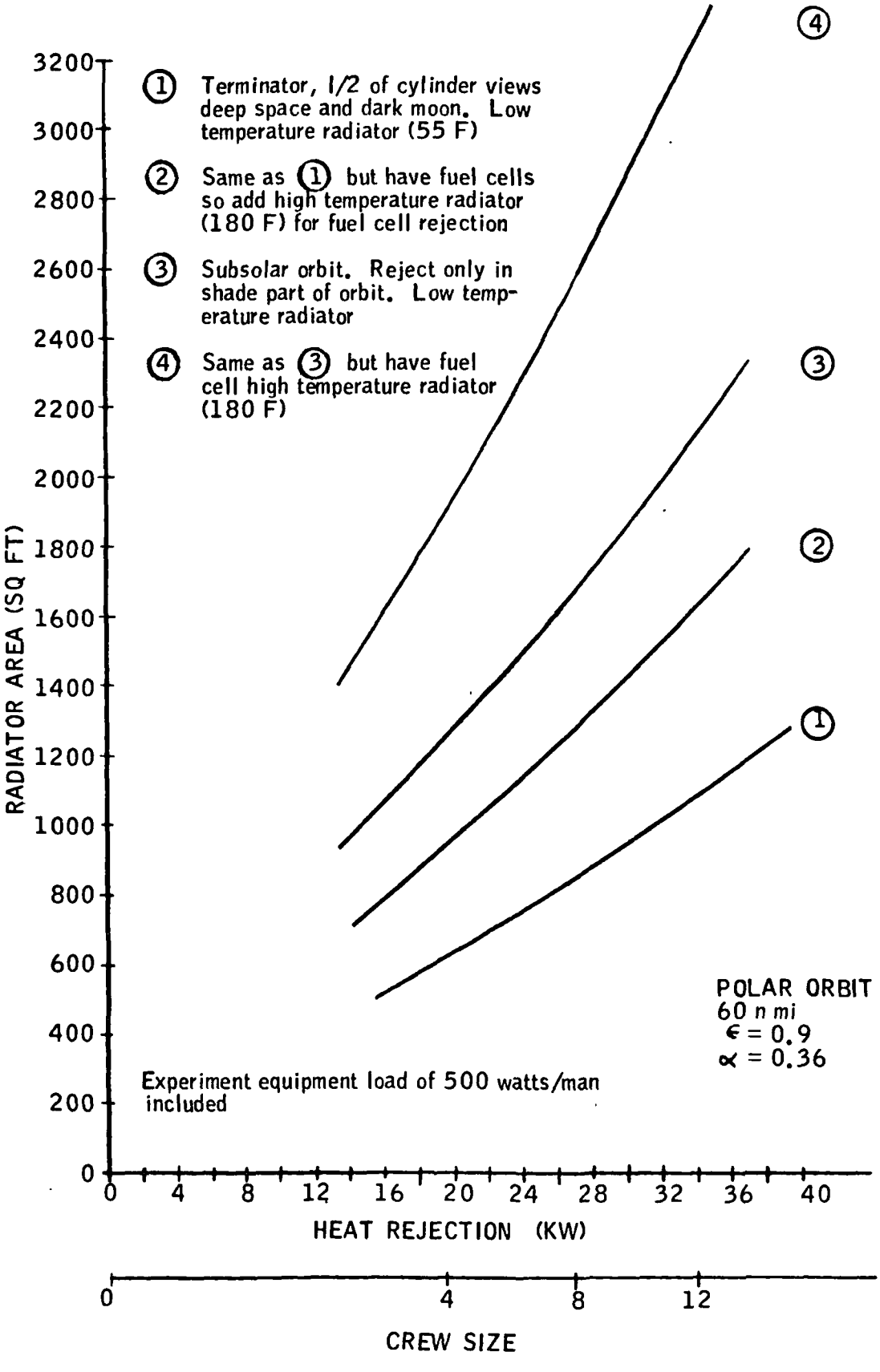


Figure 3.1-39. Radiator Area

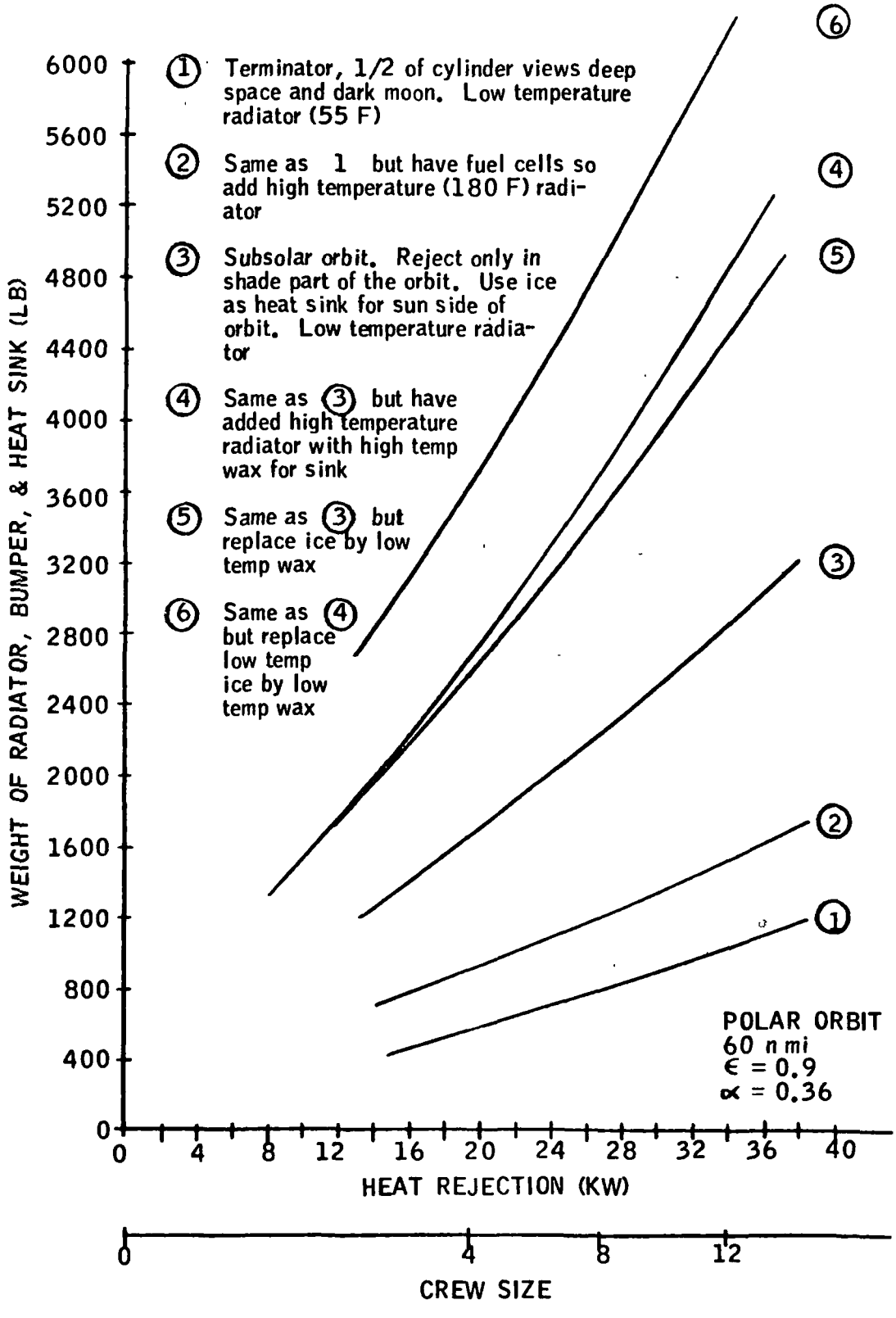


Figure 3.1-40. Radiator Weight (0.95 lb/ft^2)

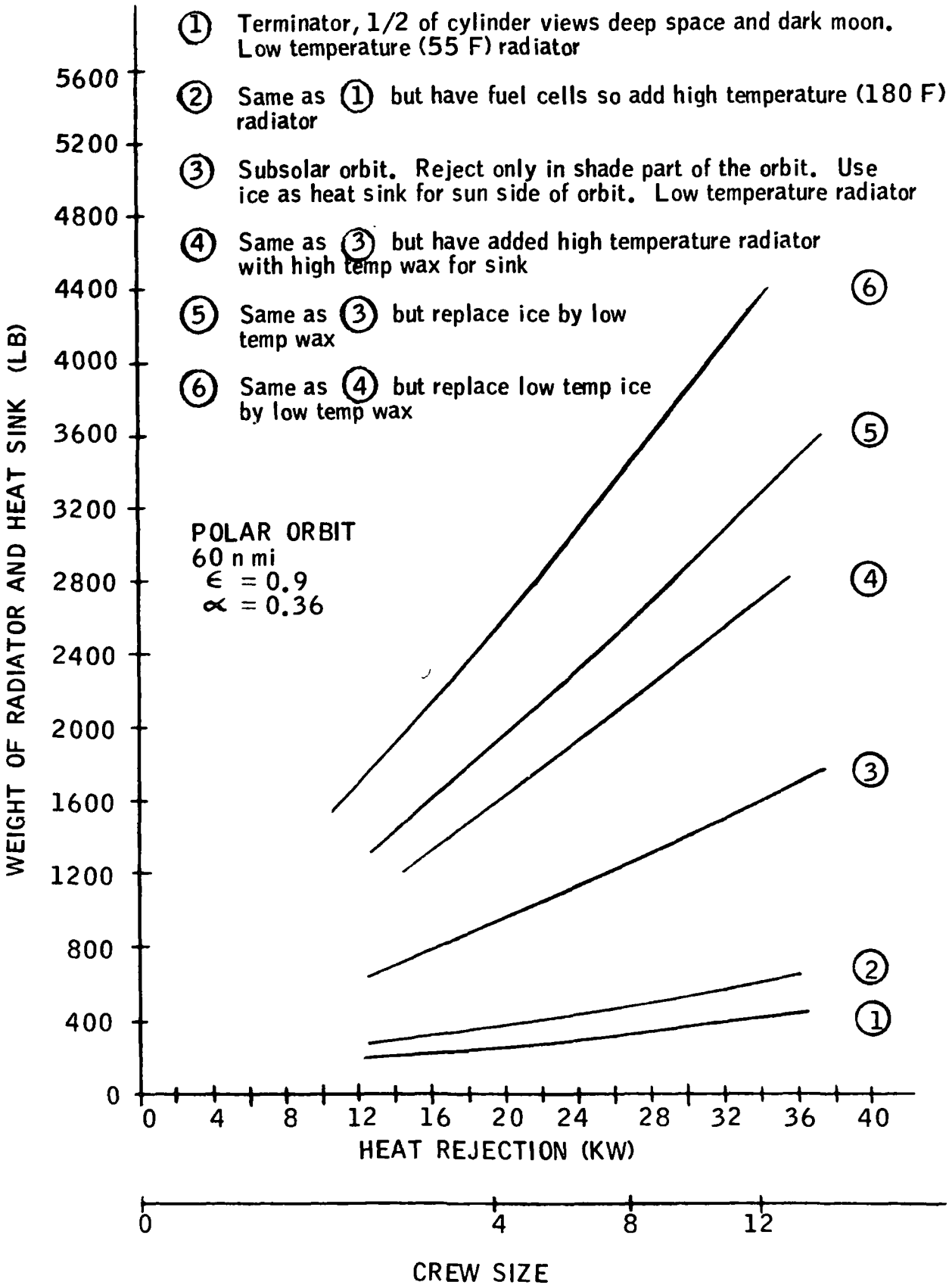


Figure 3.1-41. Radiator Weight (0.368 lb/ft^2)

Table 3.1-7. Low-Temperature Radiator Versus Radiator with Refrigeration Loop
(Radiator Weight = 0.95 lb/ft^2)

No. of Men	Case	55 F Radiator		140 F Radiator and Refrigeration Loop *				
		Radiator Area (ft^2)	Radiator** Weight (lb)	Radiator Area (ft^2)	Radiator** & Refrig. Wt. (lb)	Power Delta (kw)	Area Decrease (%)	Weight Increase (%)
4	Terminator orbit solar array-battery	613	581	470	1146	8	30	197
4	Subsolar orbit solar array-battery, tetradecane wax heat sink	1226	2562	940	3692	16	30	144
8	Terminator orbit solar array-battery	855	811	655	1377	11	31	170
8	Subsolar orbit solar array-battery tetradecane wax heat sink	1710	3574	1310	4707	22	31	132
12	Terminator orbit solar array-battery	1067	1011	786	1557	12	37	154
12	Subsolar orbit solar array-battery tetradecane wax heat sink	2134	4450	1572	5544	24	36	124

* NH_3 refrigeration loop

** Radiator based upon 0.95 lb/ft^2

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Table 3.1-8. Low-Temperature Radiator Versus Radiator with Refrigeration Loop
(Radiator Weight = 0.368 lb/ft^2)

No. of Men	Case	55 F Radiator		140 F Radiator and Refrigeration Loop *				
		Radiator Area (ft^2)	Radiator** Weight (lb)	Radiator Area (ft^2)	Radiator** & Refrig. Wt. (lb)	Power Delta (kw)	Area Decrease (%)	Weight Increase (%)
4	Terminator orbit solar array-battery,	613	226	470	873	8	30	386
4	Subsolar orbit solar array-battery, tetradecane wax heat sink	1226	1852	940	3146	16	30	170
8	Terminator orbit solar array-battery,	855	314	655	996	11	31	317
8	Subsolar orbit solar array-battery, tetradecane wax heat sink	1710	2578	1310	3942	22	31	153
12	Terminator orbit solar array-battery	1067	392	786	1099	12	37	280
12	Subsolar orbit solar array-battery, tetradecane wax heat sink	2134	3214	1572	4629	24	36	144

* NH_3 refrigeration loop
** Radiator based upon 0.368 lb/ft^2

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Water Reclamation

A single concept recovery system utilizes one system concept and one type of water purity. A multiple concept, which was selected for the EOSS, utilizes one system to recover and process potable waters and reverse osmosis to process wash waters. Provided that the major portion of the water processed is wash-type water, the multiple concept increases total system efficiency and reduces overall weight, power, and volume because wash-type water does not have to be processed to the same degree as does potable water. Thus, the system efficiency is increased because the potable water system processes a smaller flow rate and the reverse osmosis unit (by sending its concentrate to the potable system) is effectively 100 percent efficient. A water balance would be required to really determine which method is best for OLS use.

Figures 3.1-42 through 3.1-46 present schematics of the various water recovery systems being considered for OLS use. All but the reverse osmosis concept are able to recover any type of normal waste water. Reverse osmosis cannot recover urine, but it can recover wash-type water. Figure 3.1-47 is a relative comparison of various water recovery concepts for potable water based on a 0.618 lb H_2O/man-hr water flowrate. For comparison, the EOSS uses a vapor compression unit at a flowrate of 0.750 lb H_2O/man-hr and one unit weighs 540 pounds. One hundred pounds of this weight can be attributed to the more detailed analysis performed on the vapor compression unit after it was selected for the EOSS.

The 0.618 lb H_2O/man-hr is a typical potable water load for the OLS in which dishwasher, urine, urinal flush, experiments, and reverse osmosis concentrate waters are processed. The reverse osmosis unit would process shower, handwash, and humidity waters (with possibly the addition of the Sabatier water prior to its electrolysis). The reverse osmosis unit will typically process 1.03 lb H_2O/man-hr. Both the potable and reverse osmosis rates depend upon the final water rates established for the various water using devices.

Figure 3.1-47 does not include radiator penalties for the heat rejection required. All the concepts require radiator penalties, except for the vapor compression concept. Figures 3.1-47 to 3.1-49 present representative weight and power characteristics for the various water reclamation units. The densities that apply to the weights in Figure 3.1-47 are 10 lb/ft^3 for the spares. The characteristics, while based upon a constant water flowrate and the assumption of a multiple concept water system, still present a relative comparison of concepts. The reverse osmosis characteristics required for the multiple concept water system is shown in Figure 3.1-50 (based upon 1.03 lb H_2O/man-hr).

Water Storage and Purity

Depending upon the various water cycles and reserves desired, water storage will vary. Generally storage for one to two days worth of waste and fresh water is desired, excluding emergency reserves. (The EOSS has a 1-day capability.) This storage capacity is required to accept water (waste or processed) if a reclamation unit fails. Thus, redundancy of reclamation units

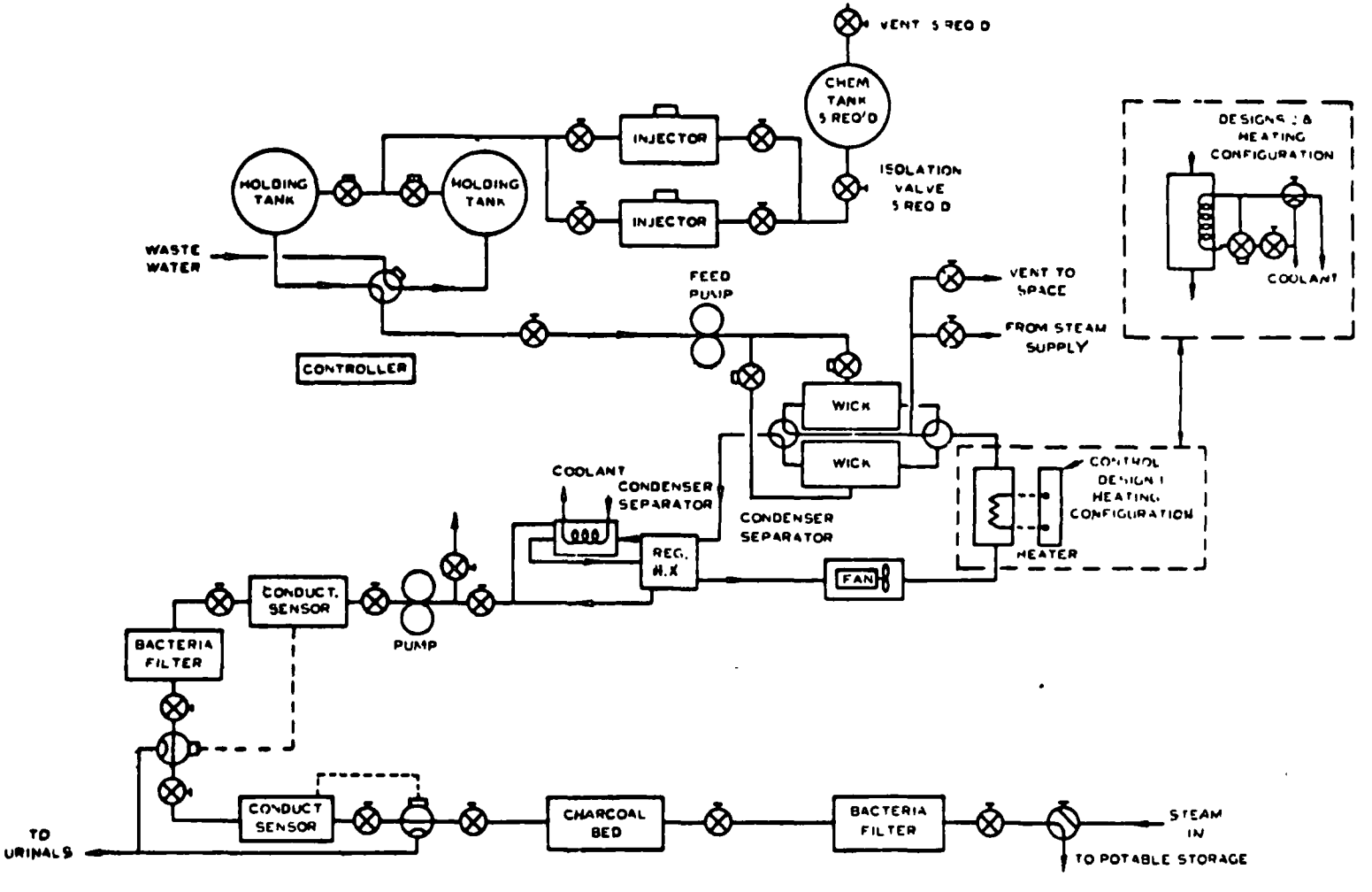


Figure 3.1-42. Closed Cycle Air Evaporating Concept

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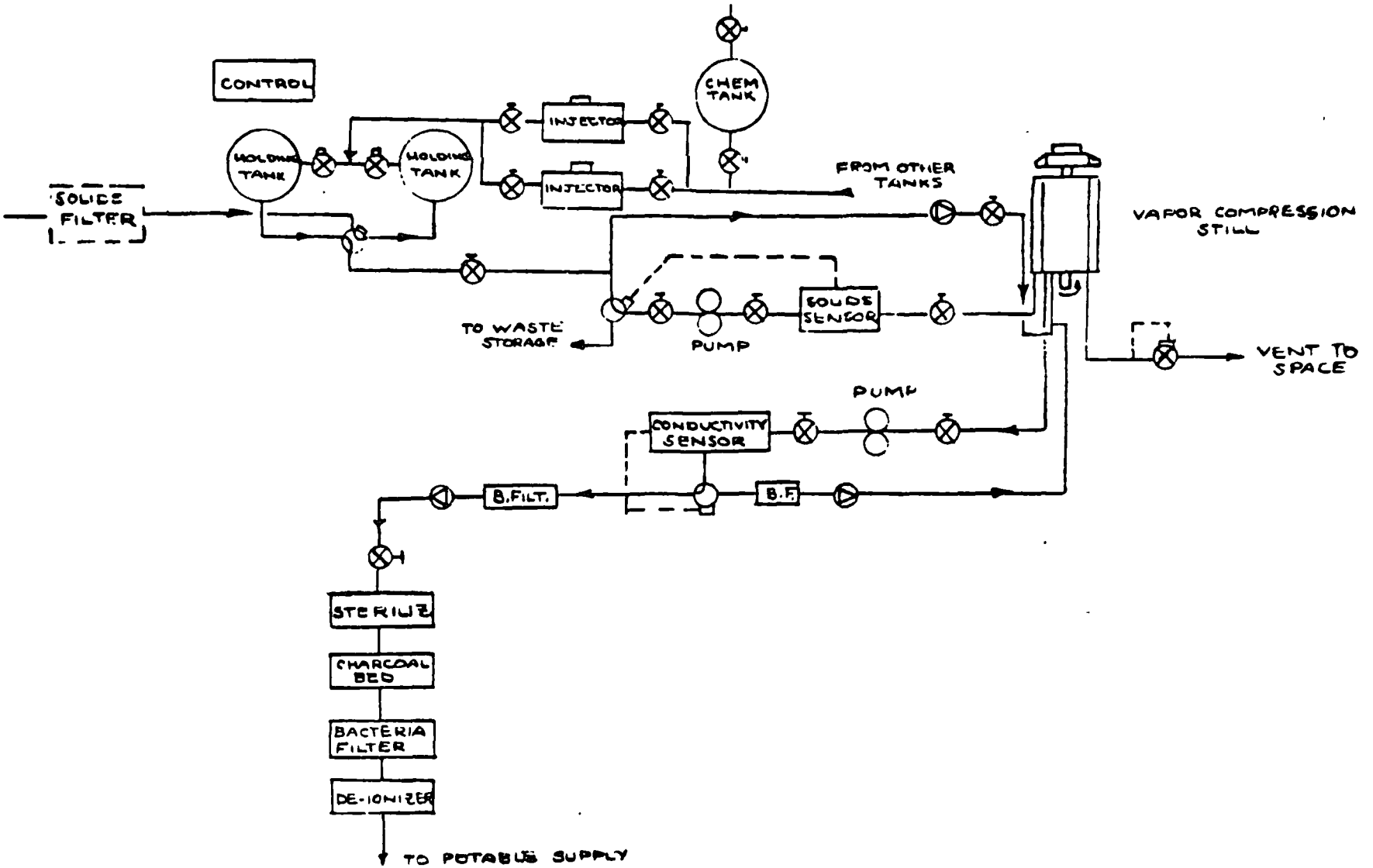


Figure 3.1-43. Vapor Compression

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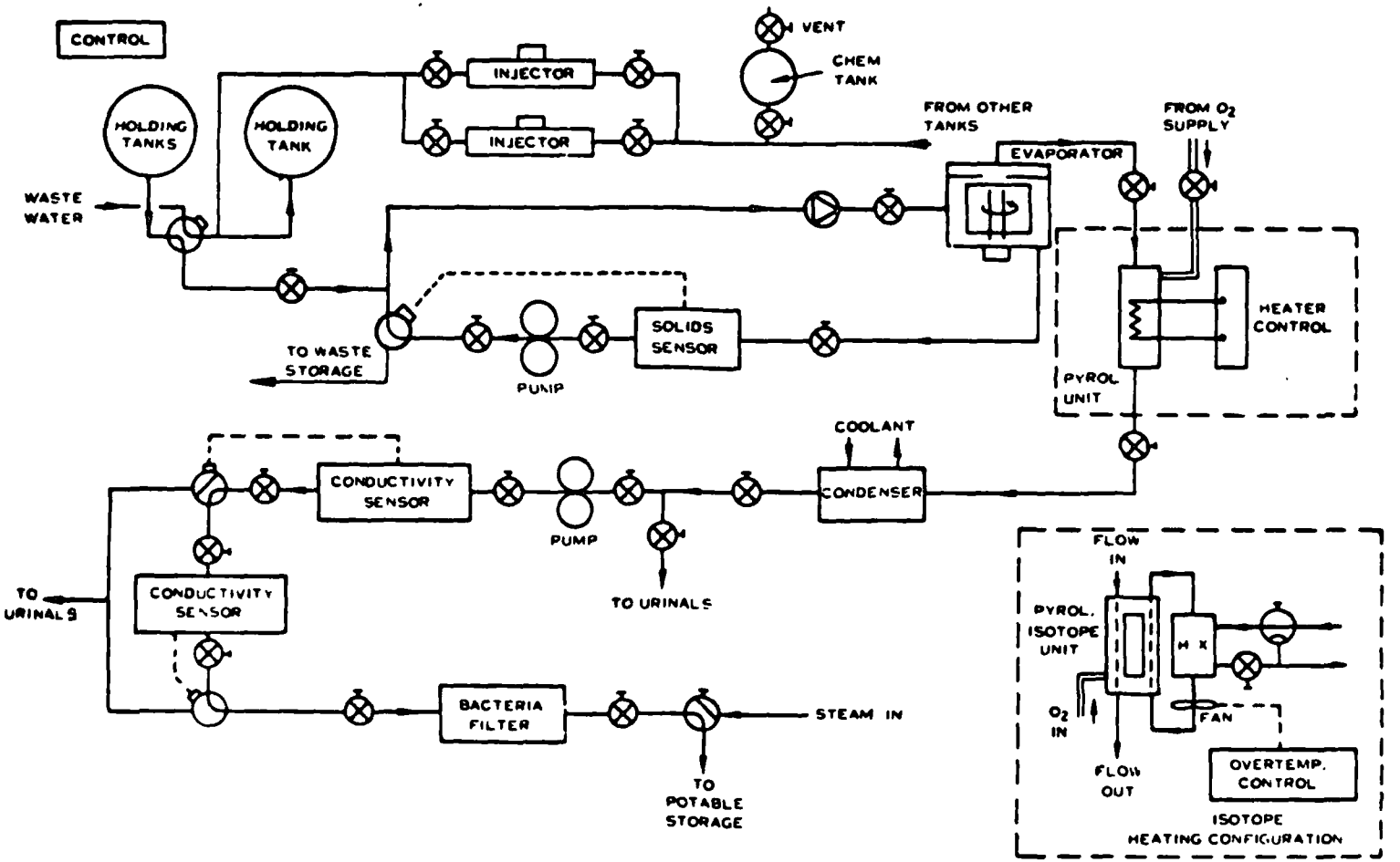


Figure 3.1-44. Vacuum Distillation/Pyrolysis Concept

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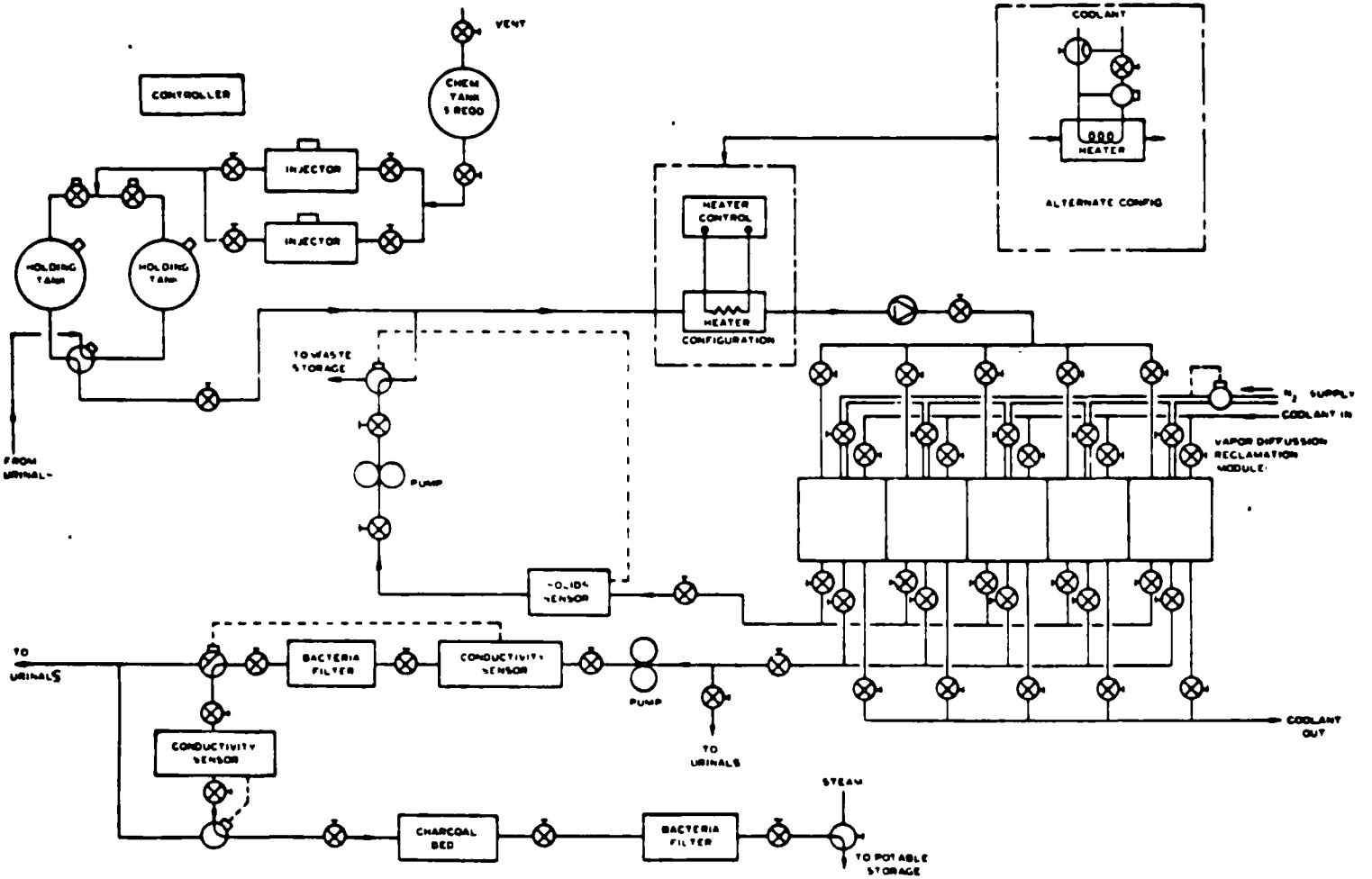


Figure 3.1-45. Vapor Diffusion Concept

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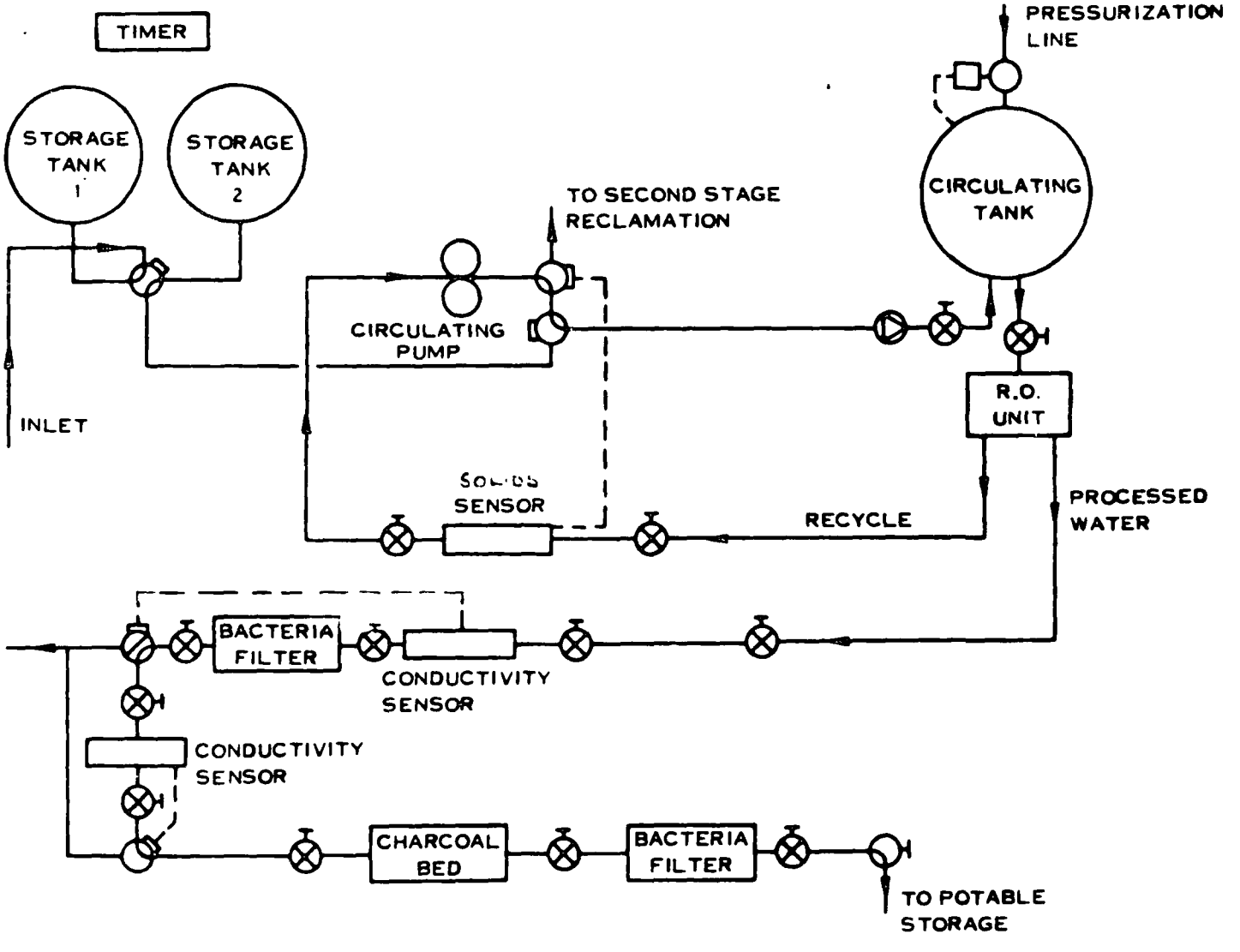


Figure 3.1-46. Water Reclamation - Reverse Osmosis

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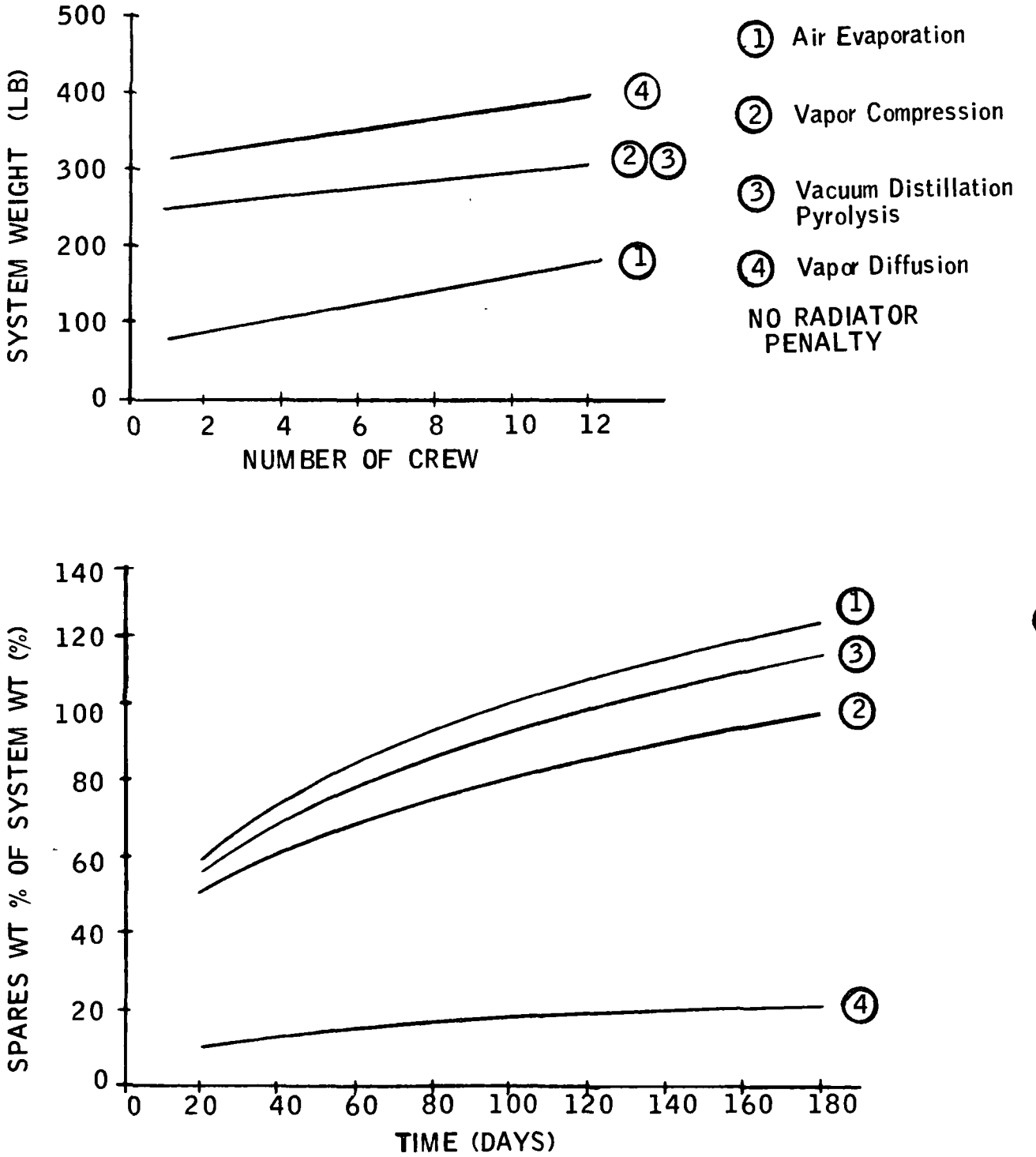
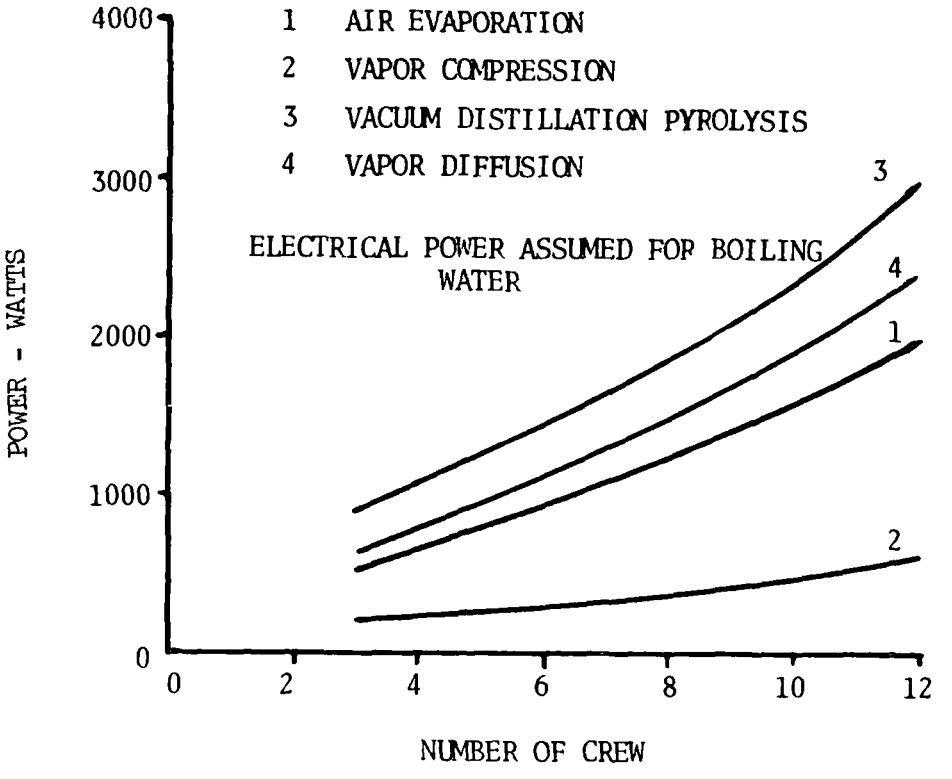


Figure 3.1-47. Water Reclamation - System Weight Trade Study

NOTE: EXPENDABLES INCLUDE SUCH ITEMS AS FILTERS, CATALYSTS OR CHEMICALS NORMALLY PROGRAMMED FOR REPLACEMENT.

Figure 3.1-48. Water Reclamation - Expendables Weight Trade Study

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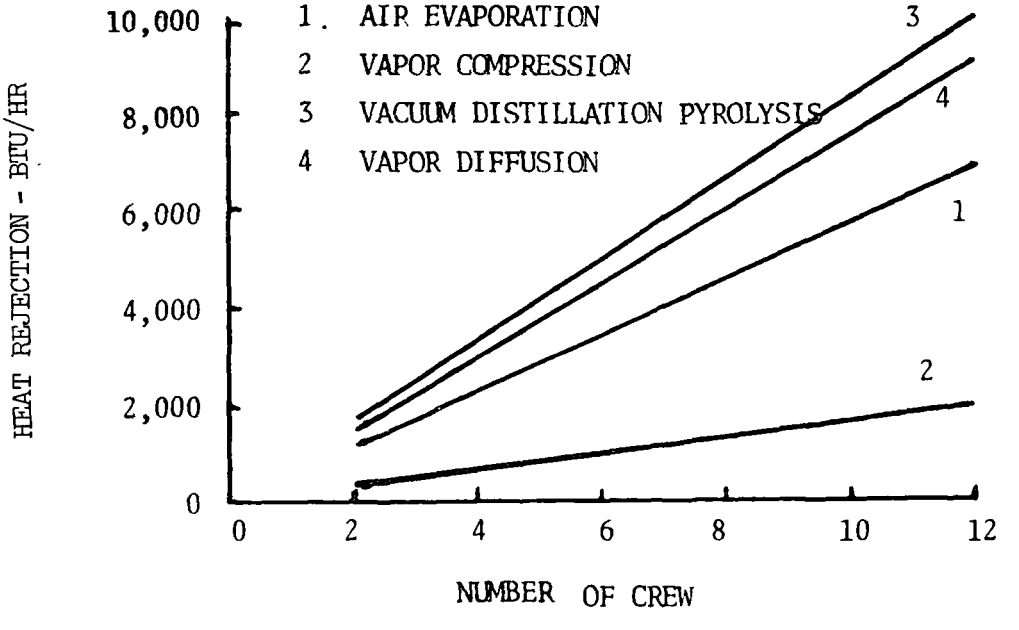


Figure 3.1-49. Water Reclamation - Electrical Power and Heat Rejection Penalties

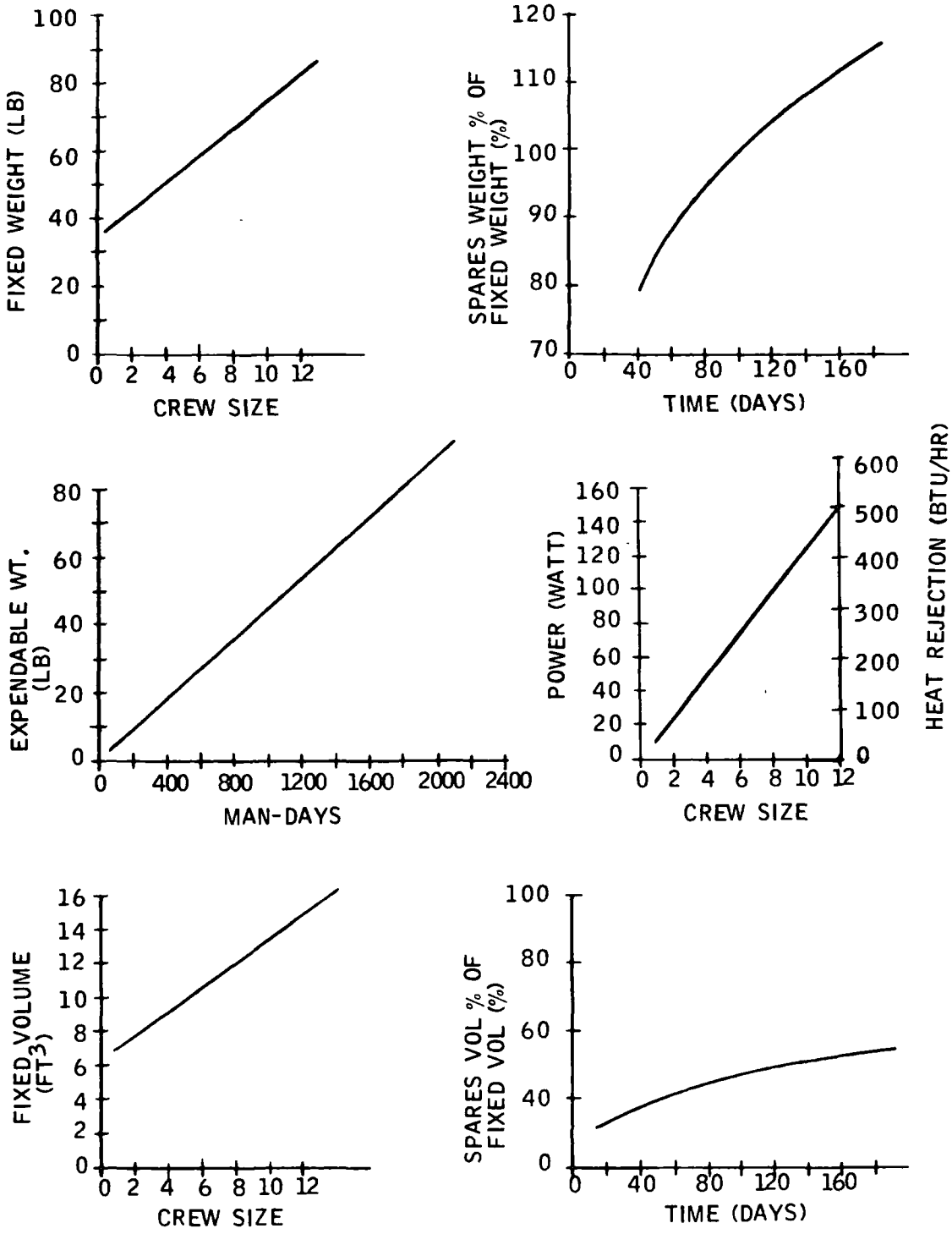


Figure 3.1-50. Water Reclamation - Reverse Osmosis, Parametric Data

affects storage sizing. Water tank penalties are on the order of 10 percent, and the volume of storage required can be approximated by using water's 62.4 lb/ft^3 density. The water reclamation curves include a waste tank for waste water storage so that only the fresh water storage tanks must be considered separately.

Water purity control will probably consist of a concept similar to the one selected for EOSS use. In this process, the potable water is in at least two tanks. One tank has previously monitored water (the use tank), and the other has water under test. Two water samples are **taken** from the tank (automatically). One is metered through a detector to test conductivity. Another sample has nutrient added and then is metered through an incubation chamber before tests for conductivity are made. In this manner, bacteria can be detected through electrical conductivity differences. Both samples have electrolyte added so that conductivity is measurable. Potable water storage tanks will be kept at a temperature of 160 F to prevent bacteria growth once the water is tested and cleared.

Should a tank become contaminated, it can be sterilized by heating it to 250 F for a given time period. Upon leaving the storage tank, the water is cooled to 60 F by a heat exchanger, and then silver ions are added for bacteria control within the plumbing. The silver ions are removed at the various water outlets. Bacteria filters also will be employed at each water outlet.

The water purity control units weight, volume, and power requirements are (respectively): 12 pounds per unit, 2 ft^3, and 150 watts while operating. A purity control and tank heating system is required for each isolated fresh-water tank farm system. However, the number of isolated tank farms required depends upon redundancy and system concept. The EOSS concept has three isolated fresh storage systems: two redundant potable storage and one wash storage tank farms.

3.1.7 WASTE MANAGEMENT

This subassembly consists of the equipment to collect, process, and store various wastes. The items included are urinals, toilets, and a waste processing unit. No options are considered in this area. The same system selected for the EOSS was chosen for the OLS because the rationale used for EOSS concept selection applies equally to the OLS. The parametric data presented here are for the EOSS-type system. Air-water flush standup urinals that can operate in zero gravity are depicted in Figure 3.1-51. Air flow is used during urination to move the urine from the man through the urinal. After urination, the urinal opening is covered, and a water flush cycle cleans out the residual urine.

A dry john/air entrainment toilet is shown in Figure 3.1-52. The toilet unit also includes a sitdown urinal unit (similar to the standup urinal). Feces and toilet paper are collected by air entrainment, shredded, and then exposed to vacuum for drying. The toilet holding tank is replaced when full. The toilet uses air entrainment to transfer the feces and toilet paper in zero gravity.

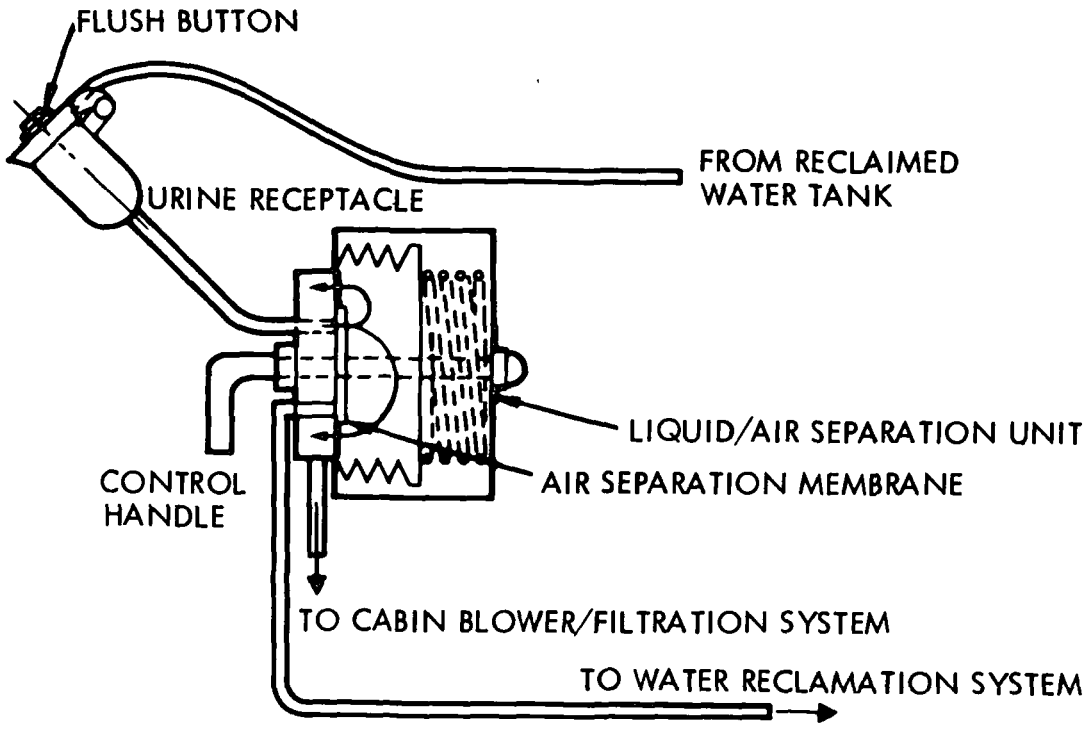


Figure 3.1-51. Urine Collection Subassembly

A waste processing unit is employed for processing nonfecal trash. Food packagings, experiment wastes and like material that could support bacteria growth are dried in a relatively low-temperature (250 F) drying chamber to remove water and other liquids that can be safely vented to space without condensing on the vehicle surface. The dried mass is then compacted with a motor-screw drive unit. After compaction, the deactivated waste is stored until it can be disposed of. Trash that does not support bacteria (scratch pads, magnetic tape, etc.) may be sent directly to the compactor, by-passing the drying chamber.

The waste management concept is chosen to emphasize two distinct goals: (1) minimization of contamination both exterior and interior to the station; and (2) a high degree of user acceptance and minimum handling. From a performance point of view, the venting of gases that might condense on the exterior surfaces (radiator, solar panel, or telescope lenses) is minimized by using low-temperature processing.

Figures 3.1-53 through 3.1-55 present the waste management characteristics. The system was sized so that at least two urinals and toilets exist for the 0- to 4-man crew size. One large or two small waste-processing units can be used. The large size refers to the single unit size used for 4 to 12 men. The number of urinals increase with crew size, four urinals being required for 12 men (two/pressure volume). In the event of a failure

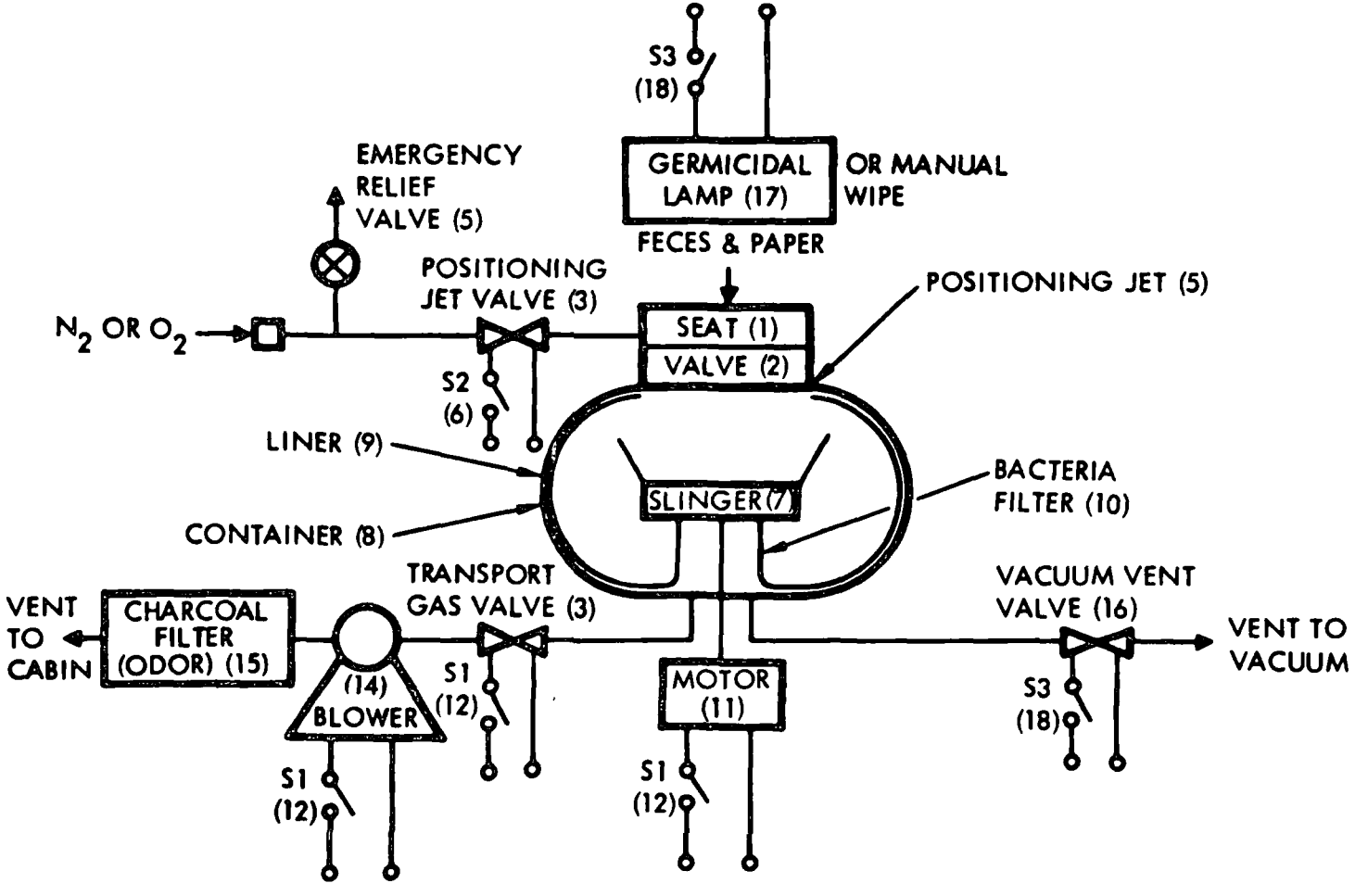


Figure 3.1-52. Fecal Collection Subassembly

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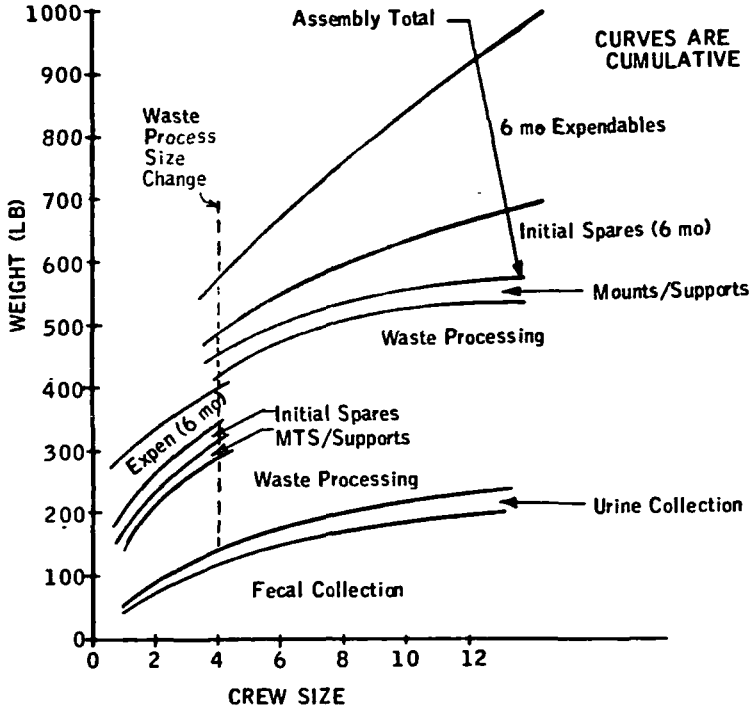


Figure 3.1-53. Waste Management - Weight

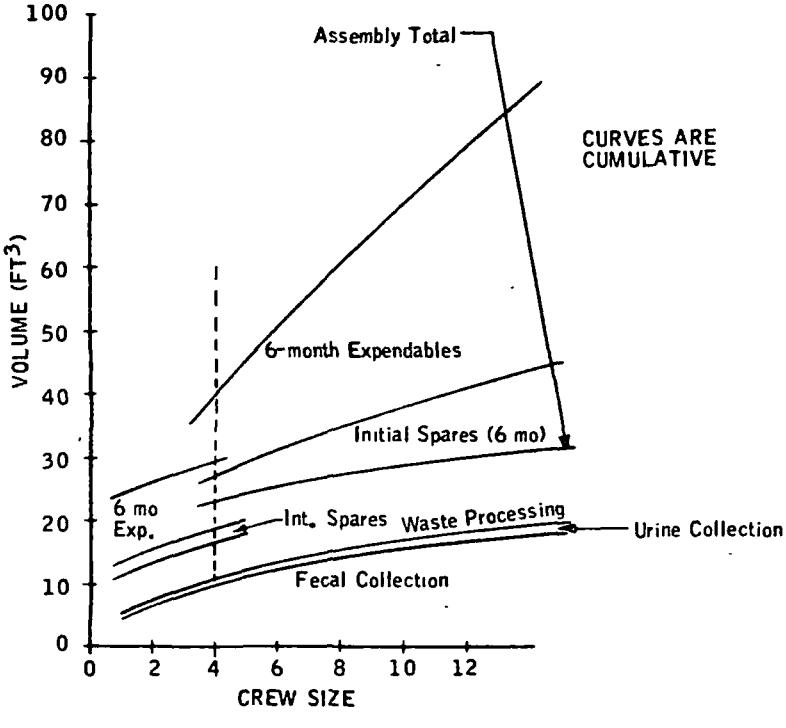
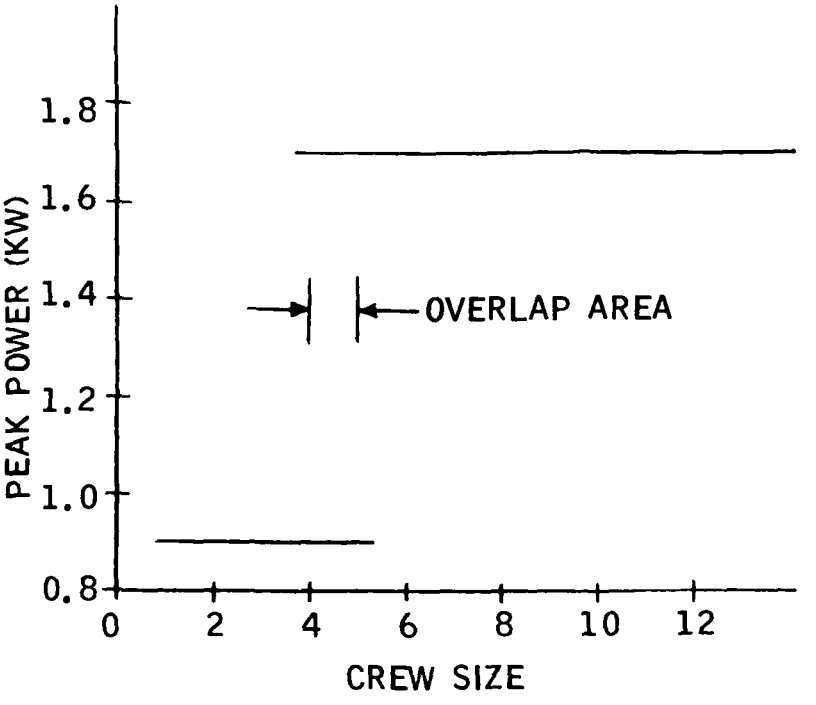


Figure 3.1-54. Waste Management - Volume



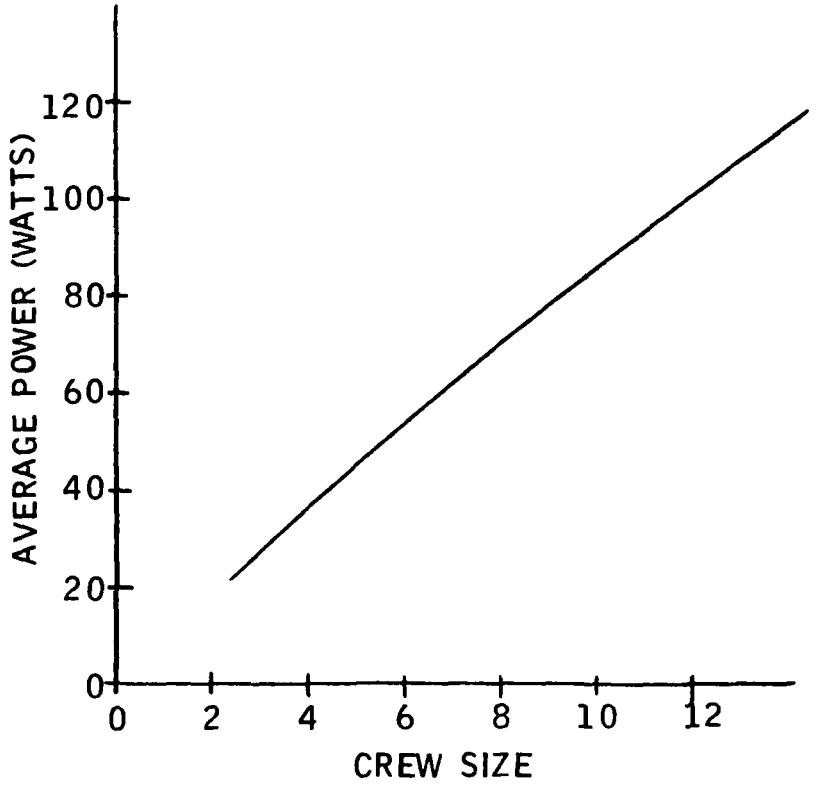


Figure 3.1-55. Waste Management - Power

of one pressure volume, the twelve men would have two urinals in the habitable volume. The number of toilets required increase to four for 12 men also. Two large waste-processing units are required for crews of 4 to 12 men. These curves, again, are approximate because both crew size and vehicle configuration, especially walking distance, determines the number of required urinals and toilets. These curves only present the effect of crew size on the number of toilets and urinals.

The cutoff of four men for waste-processing unit size (or number) change is arbitrary as is the peak power overlap. Specific waste modeling is really required for the waste-processing unit sizing. The peak power depends upon the assumption of the number of units that are operating simultaneously. The lower peak power line shown in Figure 3.1-55 assumes that one urinal, one toilet and the waste-processing unit are operating. The higher peak power line (for crews larger than four) assumes that two urinals, two toilets, and two waste-processing units are operating. Peak power is not a true indication of the power. Average power, which is based on the daily frequency and single incident time duration any particular unit (urinal, toilet, etc.) is operated, is the better indicator.

3.1.8 HYGIENE

The hygiene subassembly includes the housekeeping functions and crew cleaning functions. No options for shower, sink, and vacuum cleaner were considered. The characteristics of equipment selected were for the most part determined in connection with EOSS studies. Laundry equipment data are included as well as data describing crew personal equipment.

Laundry Considerations

Preliminary OLS laundry data, based upon EOSS data, were compiled. In summary, five concepts were studied. Table 3.1-9 lists the five concepts. Utilizing these concepts, a clothing model and laundry hardware model were established. Weight and power curves are presented in Figure 3.1-56.

The fixed weight in Figure 3.1-56 has a fixed washer size but varying laundry loads. The expendable weight includes resupply clothing for resupply or wear out, and detergent. Deltas to the water recovery system for laundry are also included in Figure 3.1-56.

Three conclusions can be drawn from Table 3.1-9. Concept 1 is preferred based upon fixed weight, and Concept 3 is preferred based upon resupply weight. Concepts 1 and 2 both require the same crew time and power. Concept 1 has less total weight (fixed and expendable) than does Concept 2. Thus, the viable options are Concepts 1 and 3. Concept 3, the preferred laundry case, has the problem of requiring matched replacement crews because the clothing is handed down from crew to crew.

Table 3.1-9. Laundry Concepts

Concept	General Type	Description	Crew Time Required (12 men) (hr/wk)
1	No laundry	Maximum clothes wear frequency	1.5
2	No laundry	Nominal clothes wear frequency	1.5
3	Total laundry	Nominal wear frequency - new replacement crew uses clothing already onboard from the previous crew	6.0
4	Total laundry	Nominal wear frequency - new replacement crew brings own clothing; previous crew brings back their clothing; based on 12-man crew replacement every 6 months	6.0
5	Partial laundry	Maximum wear frequency for outer garments - no laundering of outer garments; nominal wear frequency of undergarments, laundering of undergarments	3.5

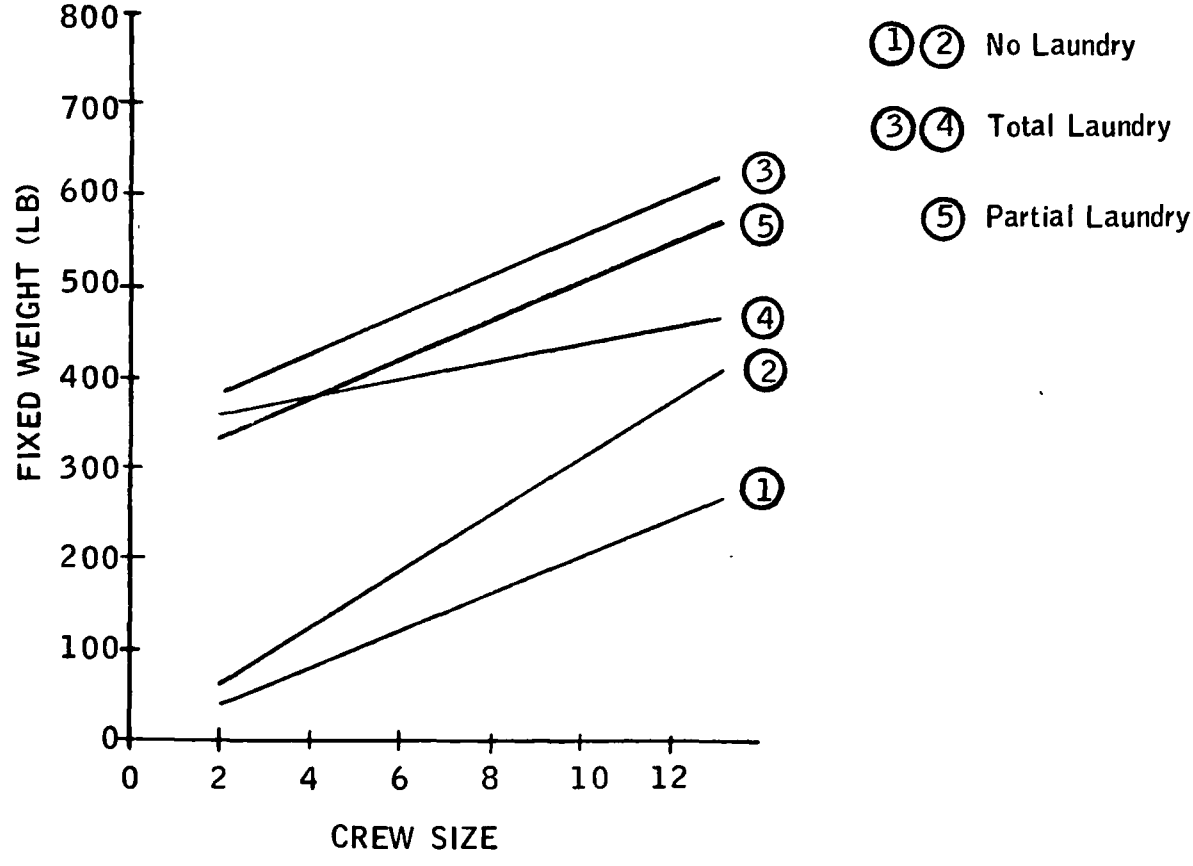
NOTE: Wear frequency refers to the length of time between changes of the clothing. Maximum wear frequency implies clothes are changed less than for the nominal case.

Discussion of Hygiene Parametric Curves

For the purposes of generating parametric hygiene subsystem characteristics the no laundry concept was chosen. If a laundry is required the data from Figure 3.1-56 should be used to modify the curves of Figures 3.1-57 through 3.1-60 as follows:

1. Remove the clothing fixed weight delta of Figure 3.1-57 and add the laundry fixed weight delta for the particular laundry concept chosen from Figure 3.1-56.

2. Remove the clothes and linens expendables weight delta of Figure 3.1-58 and add the laundry expendables weight delta for the particular laundry concept chosen from Figure 3.1-56.



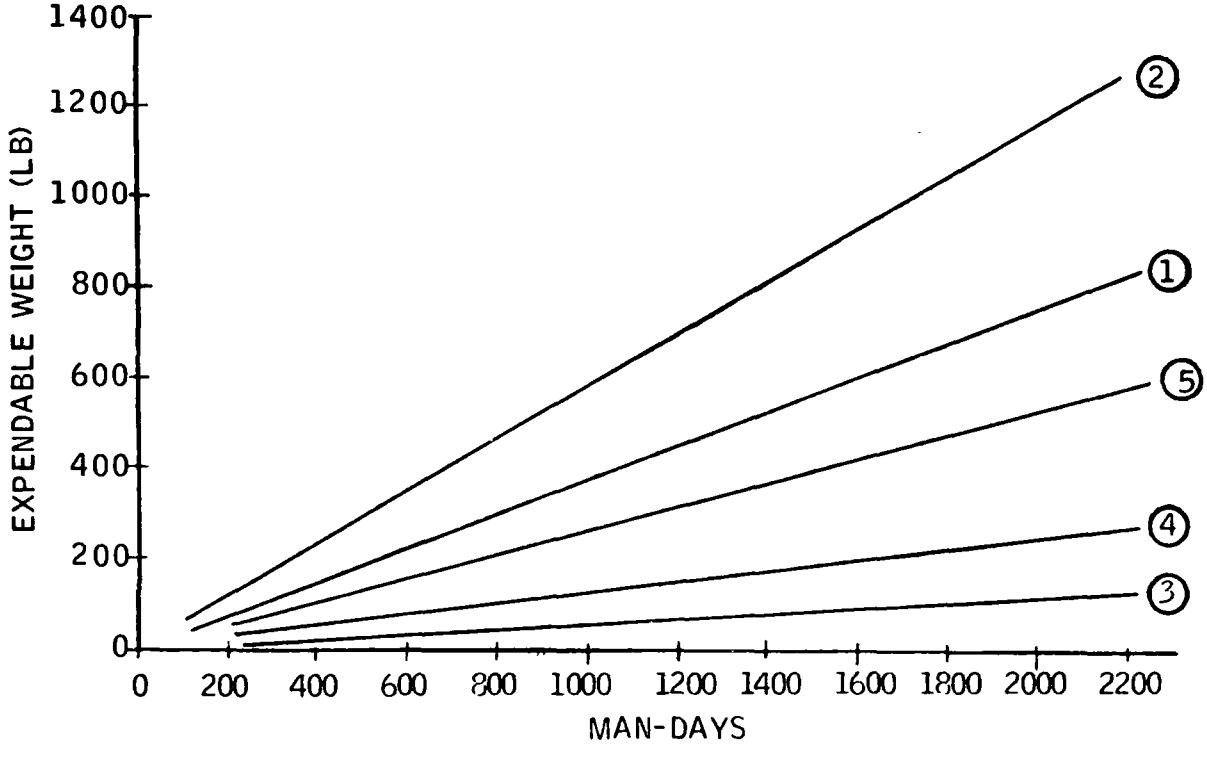


Figure 3.1-56. Laundry Trade Data
(Sheet 1 of 2)

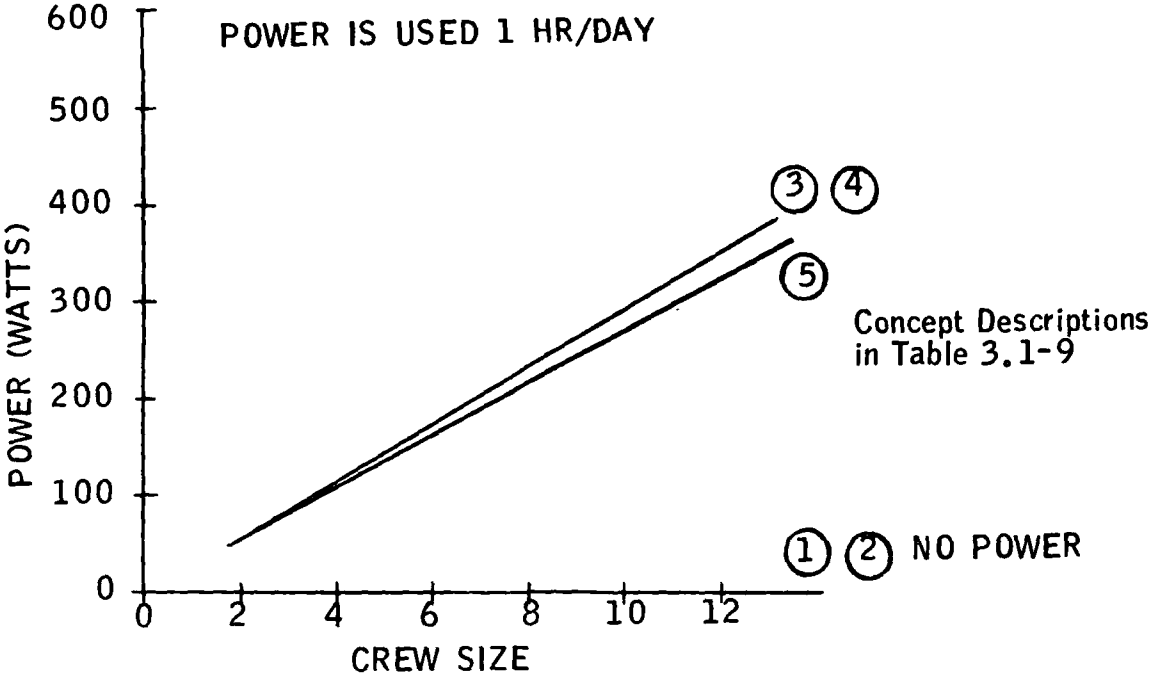


Figure 3.1-56. Laundry Trade Data
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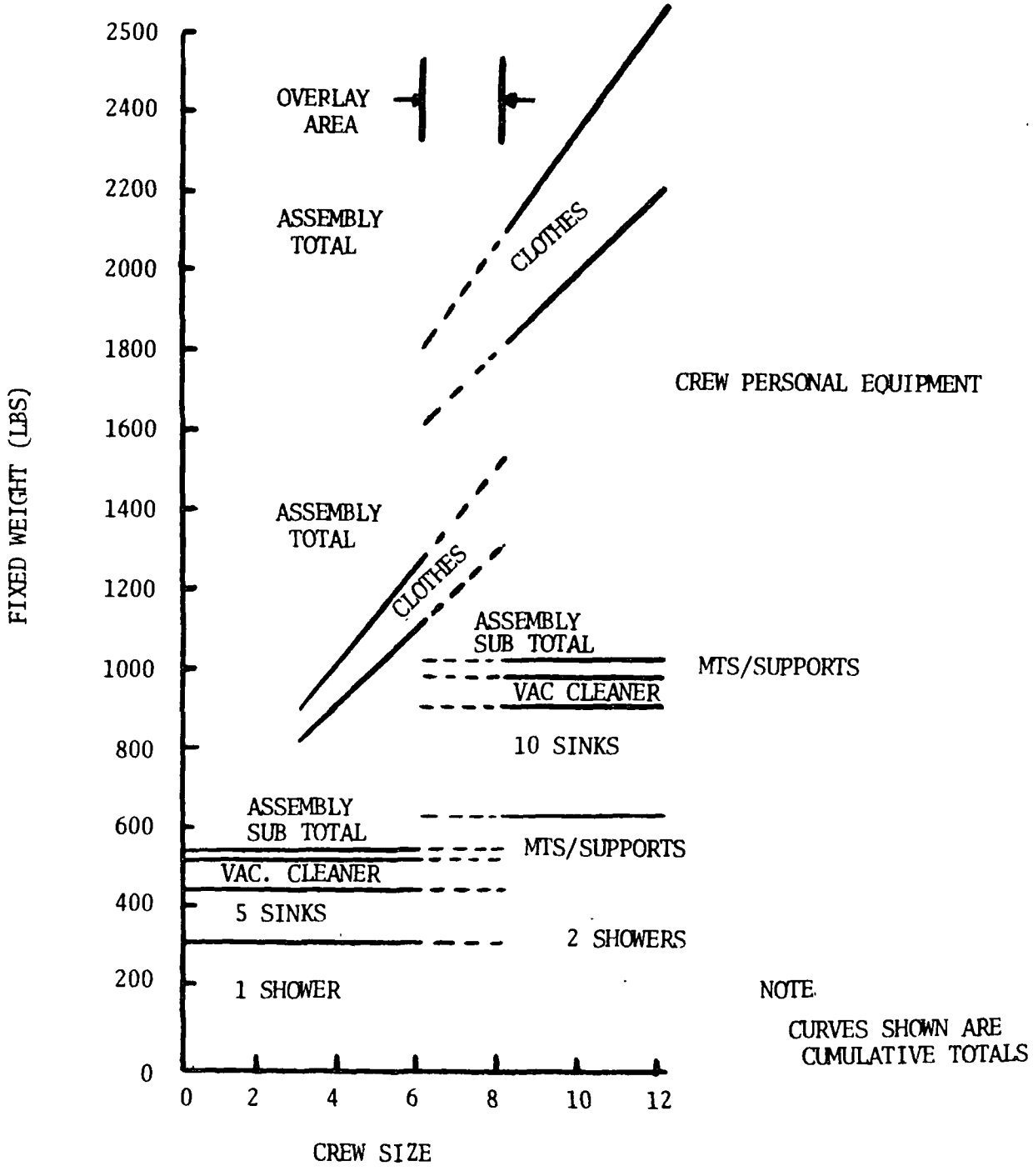


Figure 3.1-57. Hygiene - Fixed Weight, No Laundry

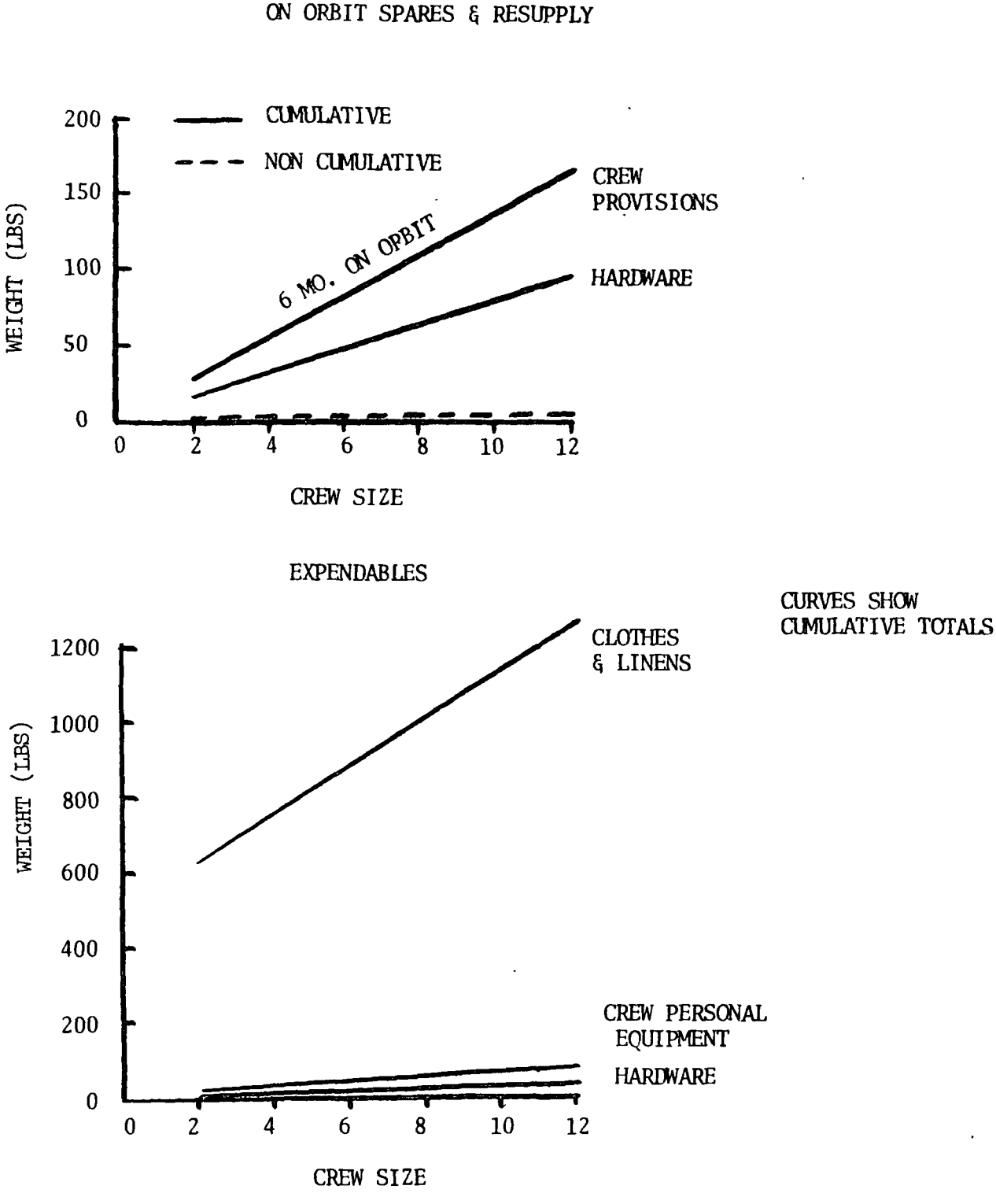
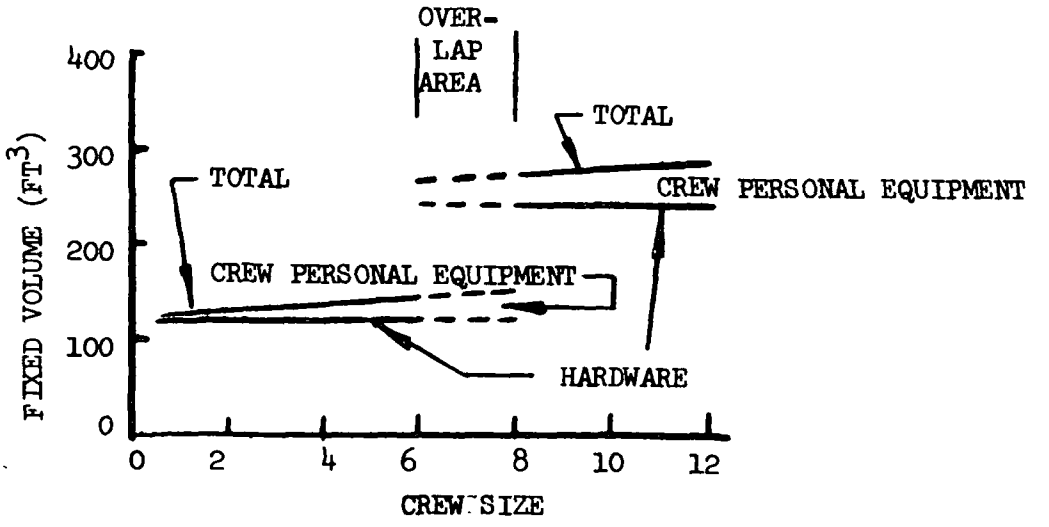
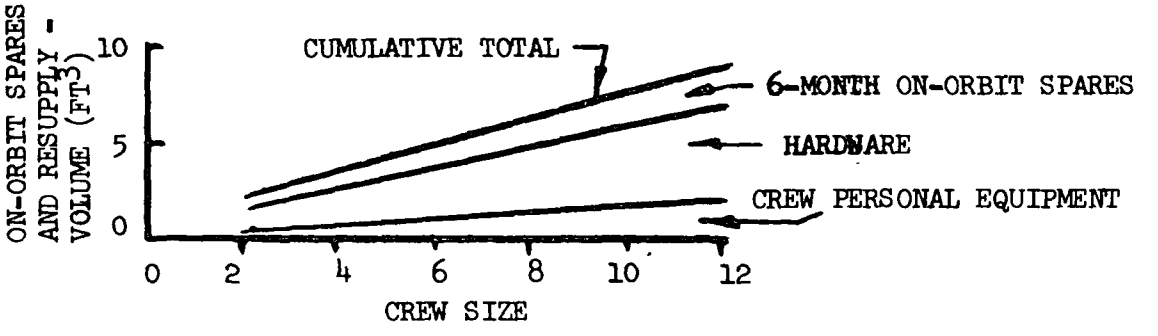


Figure 3.1-58. Hygiene - Weight, No Laundry





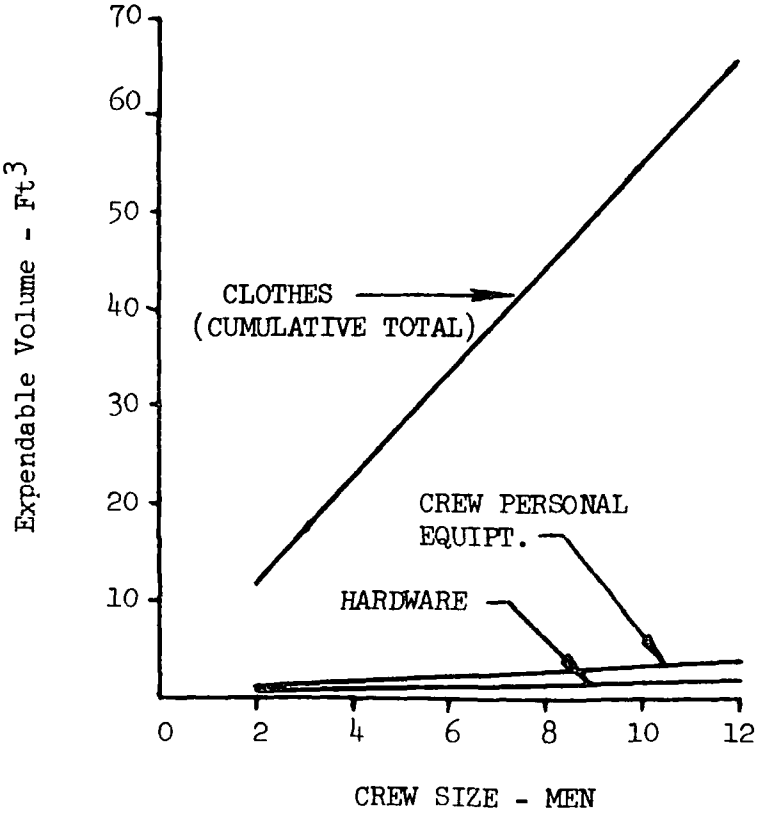


Figure 3.1-59. Hygiene - Volume, No Laundry

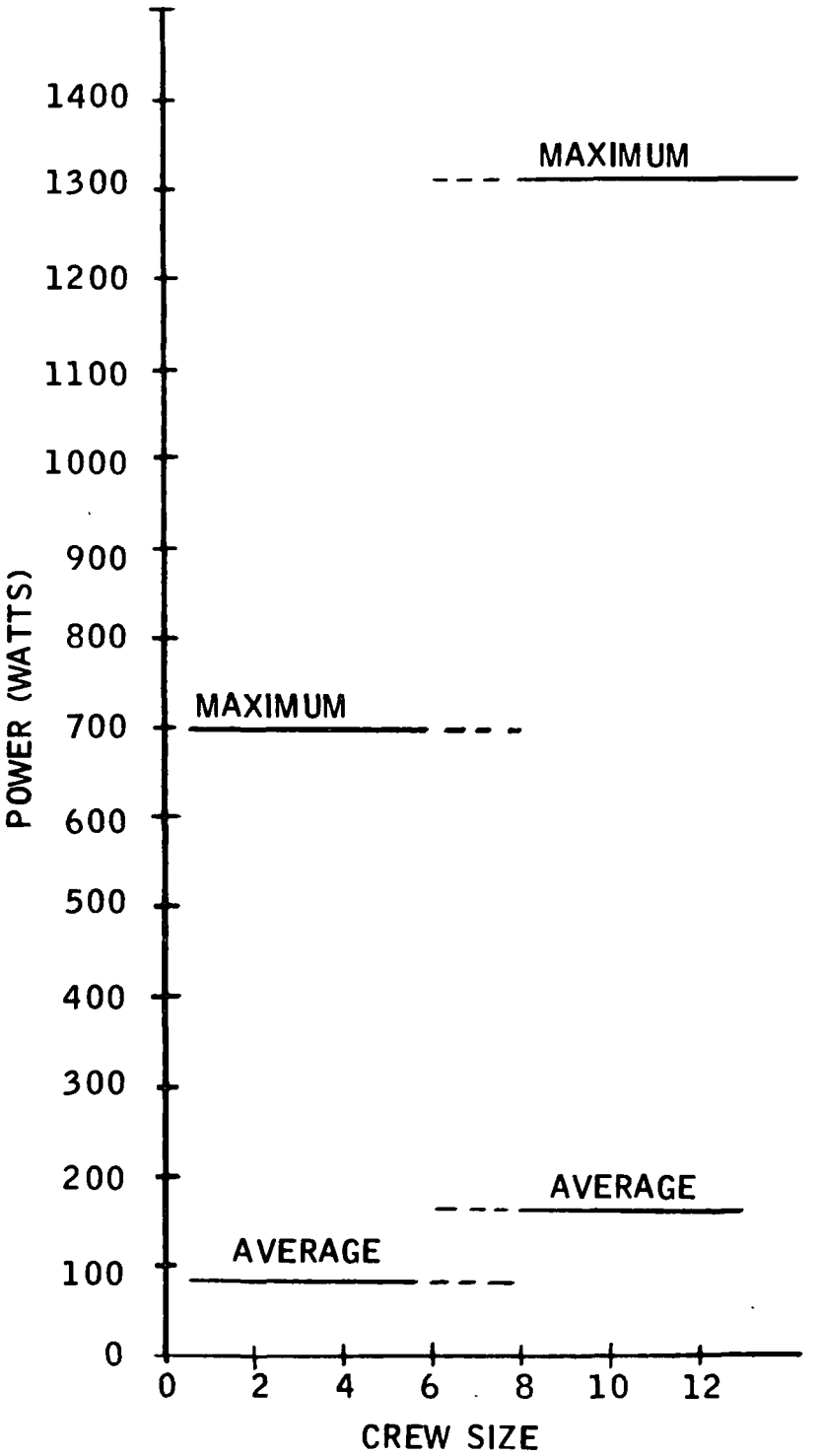


Figure 3.1-60. Hygiene - Power, No Laundry

3. An approximate density value for a laundry is 20 lb/ft^3. To modify the expendables volume curve of Figure 3.1-59, the clothing delta would have to be replaced by the expendable weight delta of Figure 3.1-56 divided by this 20 lb/ft^3.

4. The maximum hygiene power of Figure 3.1-60 would increase by the laundry power of Figure 3.1-56. It increases with crew size. The average laundry power delta would increase by 1/24 of the power requirements from Figure 3.1-56.

Figures 3.1-57 to 3.1-60 present the hygiene subsystem characteristics assuming no laundry. The basic hardware consists of showers, sinks, and vacuum cleaners. It should be noted that as in the waste management section, an arbitrary overlap was assumed at the 6- to 8-man OLS crew range. For six or less men, the number of showers and sinks were halved because the smaller vehicle volume for less men requires less hygiene facilities. The number of vacuum cleaners (three) were held constant. Figure 3.1-61 shows the shower concept. Also shown in Figures 3.1-57 to 3.1-60 are crew personal equipment expendables. The crew personal equipment includes disposable clothing, linens, personal effects, and toilet articles. The comments on power mentioned previously for the waste management power curve also applies to the hygiene power curve. The higher maximum power curve includes simultaneous operation of two showers, three sinks, and one vacuum cleaner. The lower maximum power curve has one shower, 1-1/2 sinks, and one vacuum cleaner on simultaneously.

3.1.9 FOOD MANAGEMENT

The food management subassembly includes all hardware for storing, preparing, and serving food as well as for cleaning up food preparation areas. Shelf, table, or counter penalties are excluded. The parametric curves presented in Figures 3.1-62 and 3.1-63 also include food plus packaging.

Using the EOSS concept as the model, the OLS food management hardware consists of freezer, refrigerator, resistance oven, microwave oven, reconstitution unit, utensils for cooking and eating, and a dishwasher. In Figures 3.1-62 and 3.1-63, the hardware weight includes the main galley hardware in one pressure volume plus a backup reconstitution unit in the other pressure volume for the emergency backup galley.

The OLS food mix conceptually is the same as for the EOSS - thermostabilized, freeze-dried, frozen, and fresh foods. The packaging penalties are included in the weight. The food rate is 1.68 lb/man-day with 1.18 lb/man-day packaging. The EOSS food mix is based on Zero-g shirtsleeve operations for an energy output of 11,900 Btu/man/day, Apollo data, and data from the Compendium of Human Responses to the Aerospace Environment, NASA CR-1205, Volume III, November 1968. In addition to providing a well-balanced diet, the food mix will promote the psychological well-being of the crew by resembling an earth environment food mix as closely as possible.

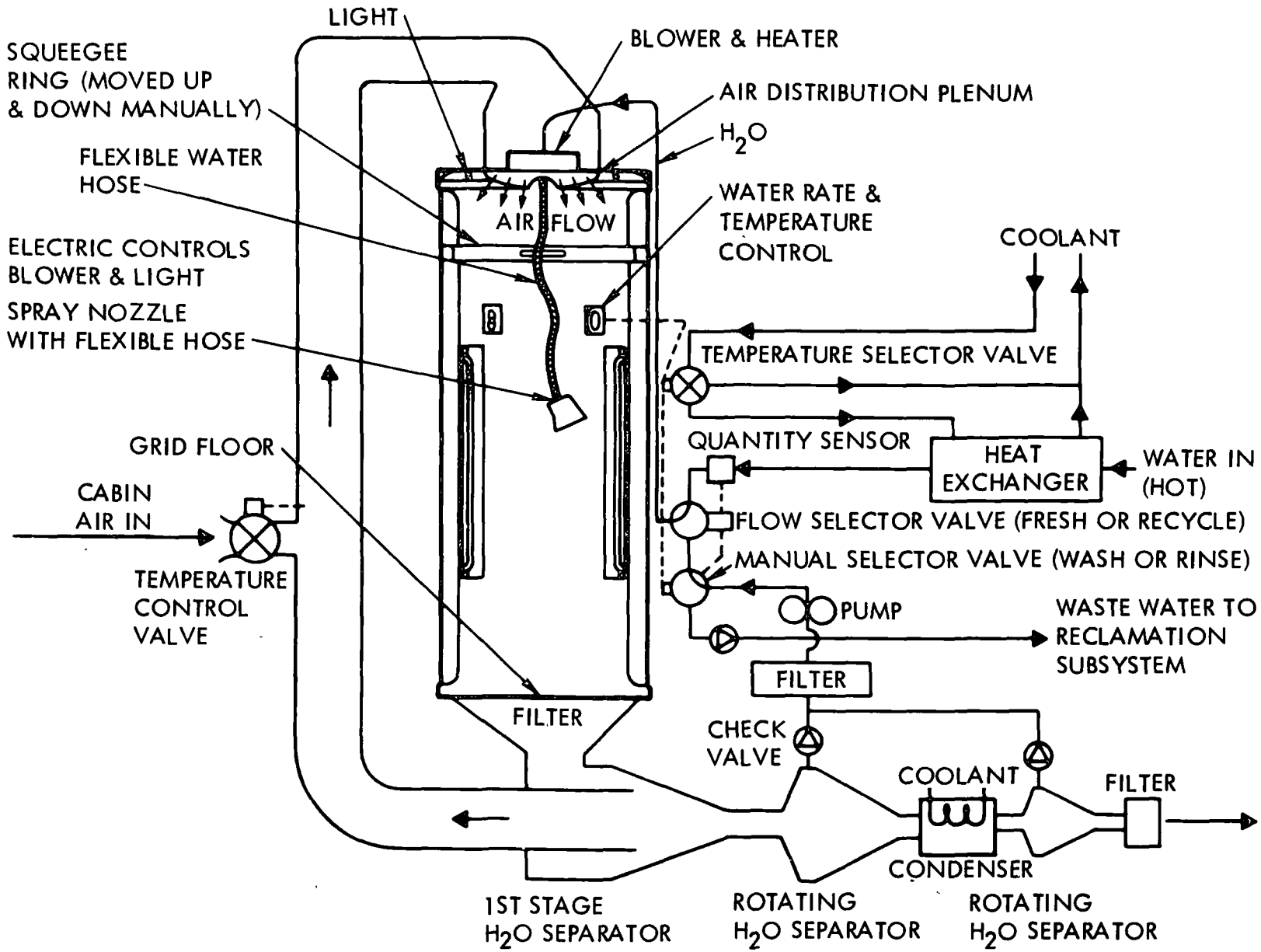


Figure 3.1-61. Zero-Gravity Shower

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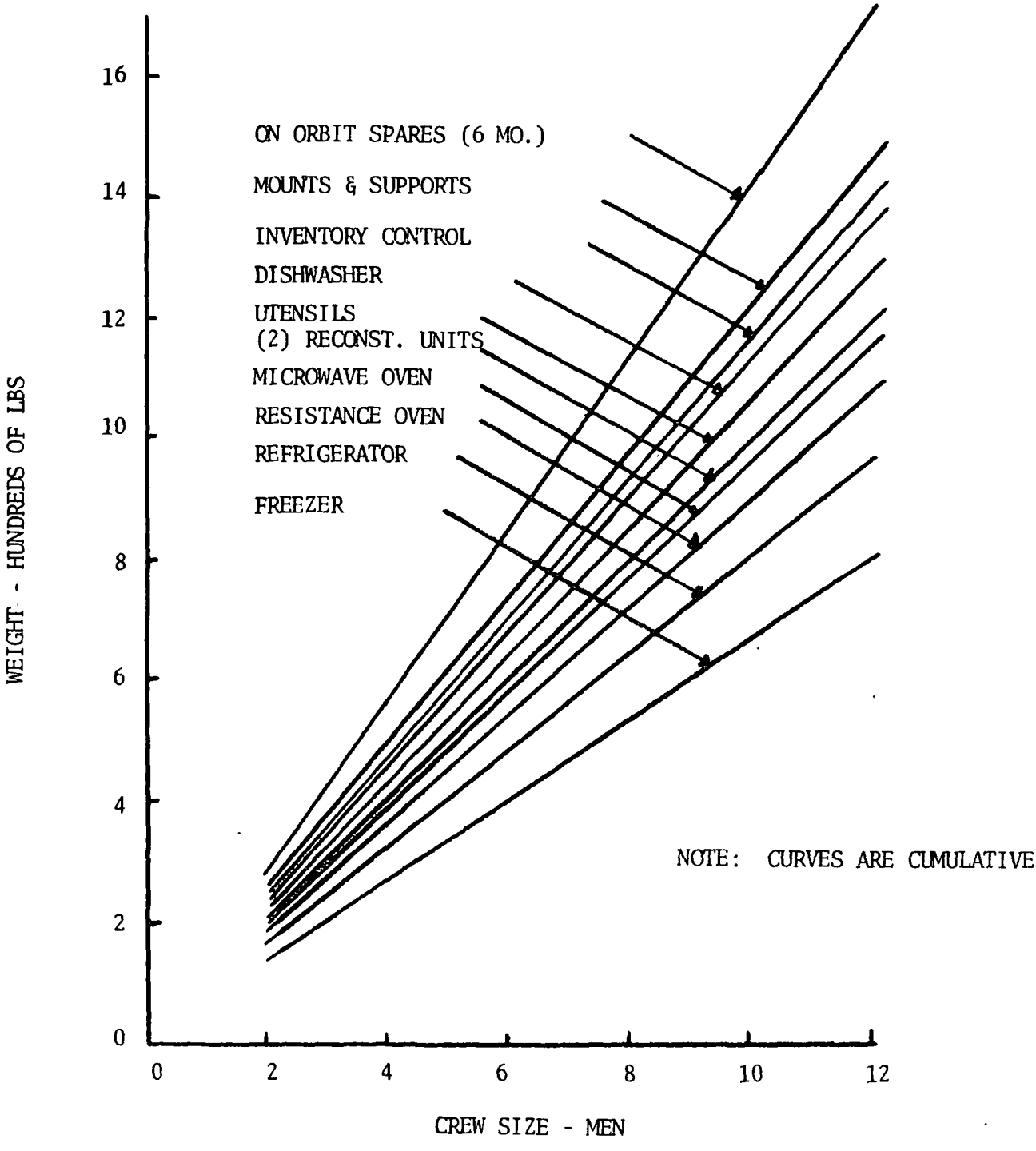


Figure 3.1-62. Food Management - Weight
(Sheet 1 of 2)

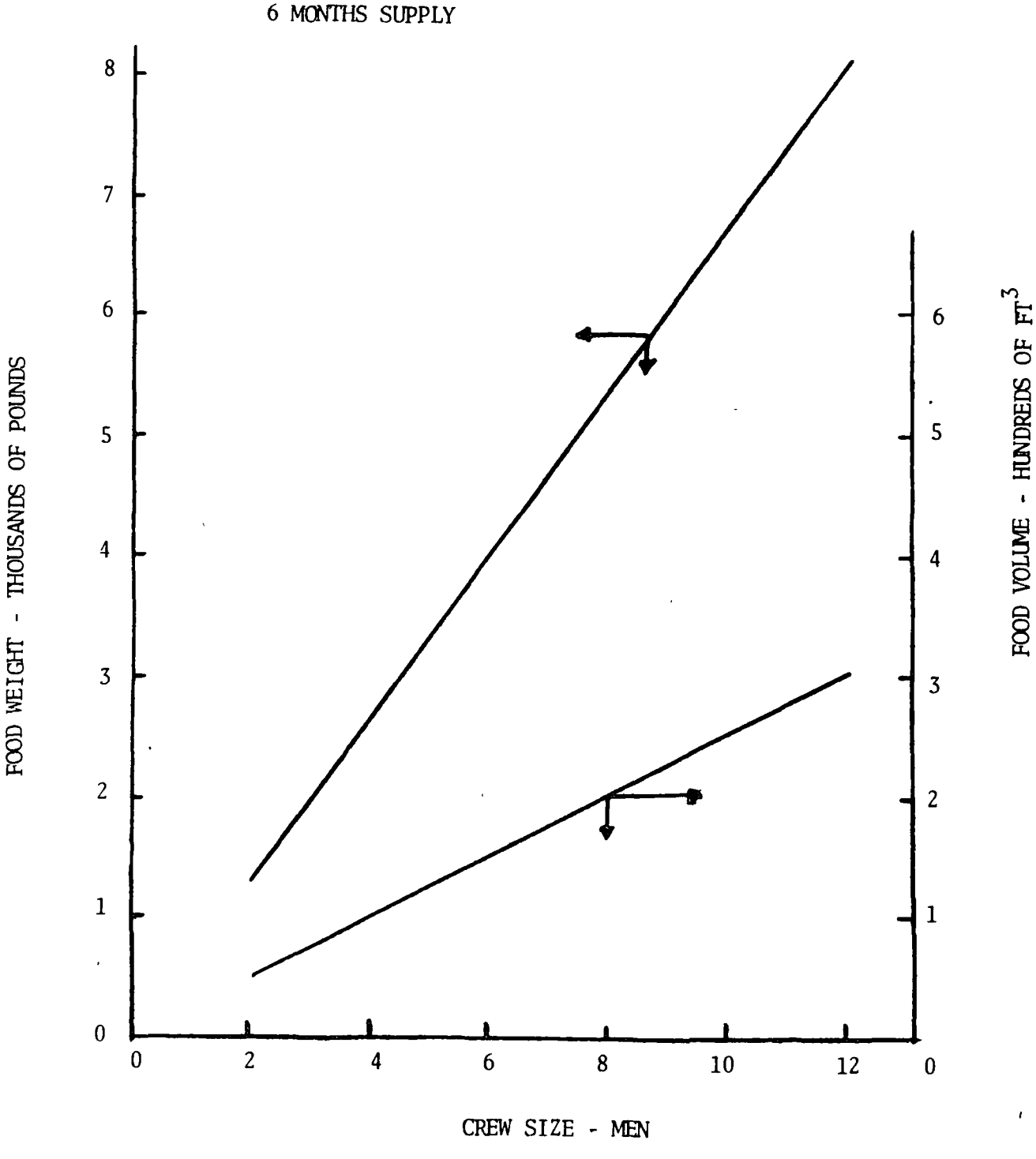


Figure 3.1-62. Food Management - Weight
(Sheet 2 of 2)

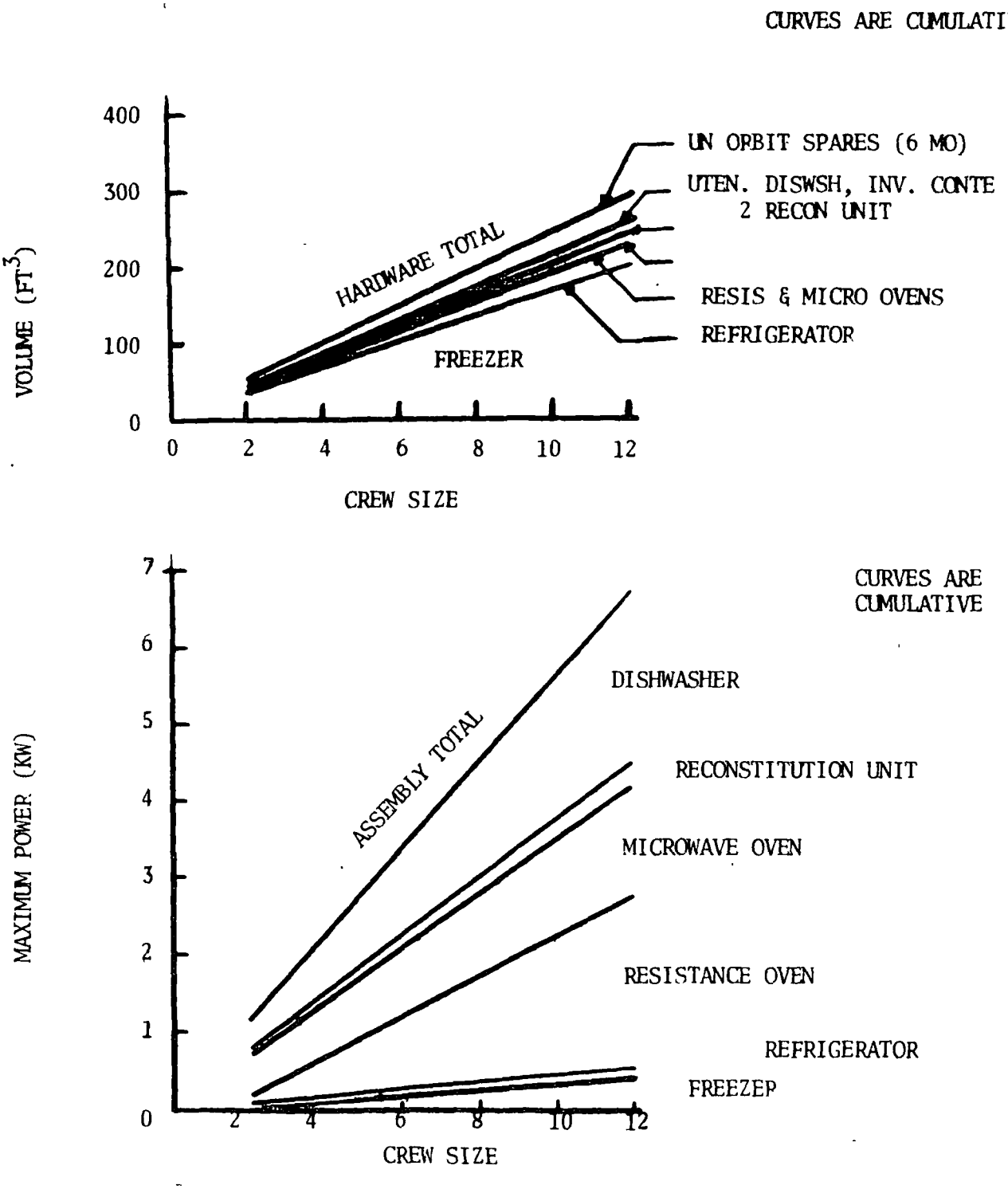


Figure 3.1-63. Food Management - Volume and Power
(Sheet 1 of 2)

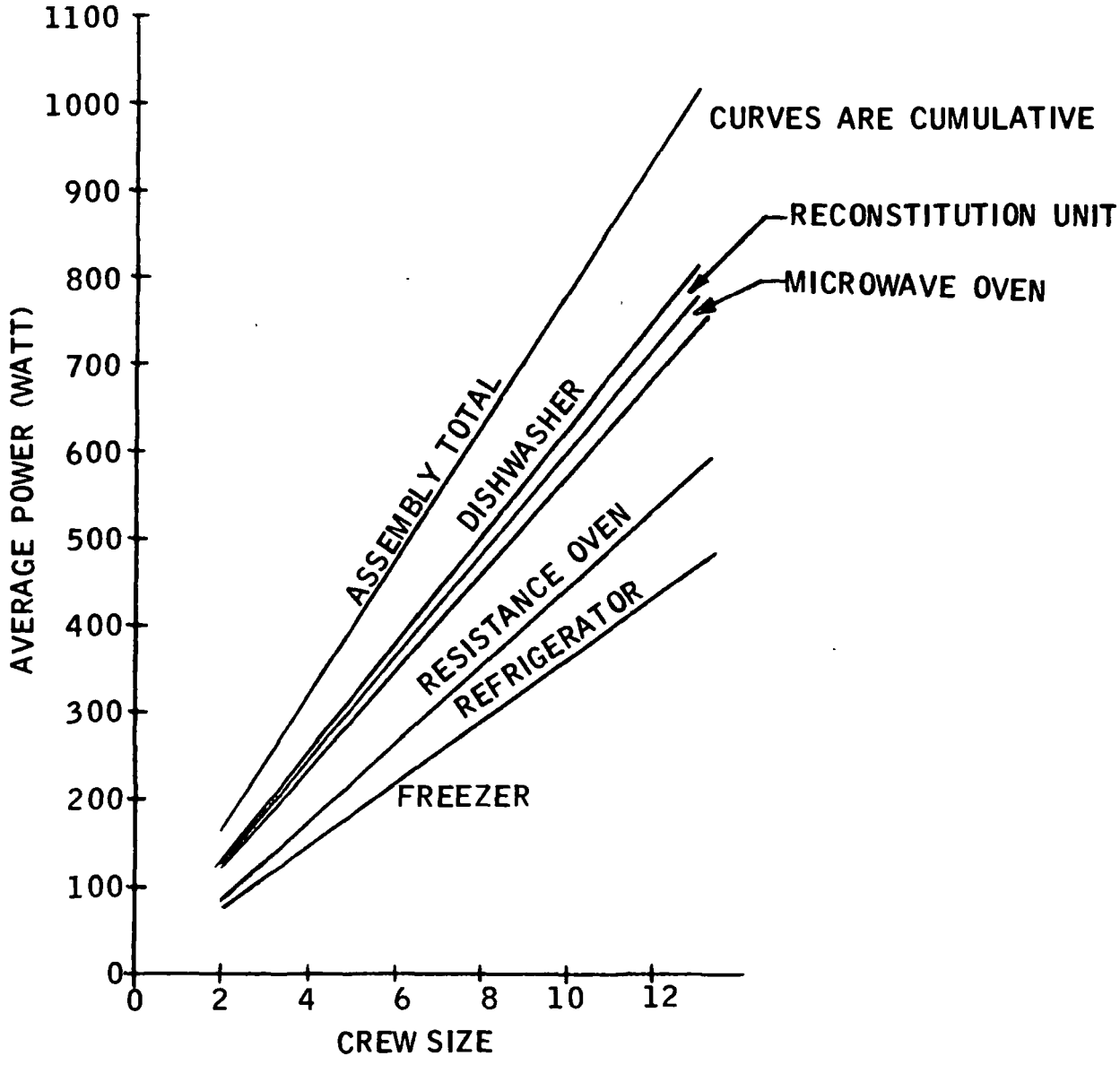


Figure 3.1-63. Food Management.-
Volume and Power
(Sheet 2 of 2)

3.1.10 SPECIAL LIFE SUPPORT

Special life support is a miscellaneous subassembly, which includes emergency oxygen, water, and food storage; IVA and EVA support; and fire control equipment.

Figure 3.1-64 presents parametric data for emergency supplies. The "pounds" curve is based upon 20 lb/day O_2/N_2 leakage plus other air venting, both of which are independent of the number of crewmen. Although the two curves are independent, total consumables for a given number of days can be determined by adding the product of the number of crewmen and the per man consumables to the leakage consumables. Table 3.1-10 lists the rates utilized in generating the curves in Figure 3.1-64. For the oxygen storage, 3000 psia tanks were assumed. The gas tank penalty will vary from 1.0 to 1.2 lb tank/lb gas, depending upon the arrangement and number of tanks.

Intervehicular activity (IVA) and extravehicular activity (EVA) support will vary depending upon vehicle configuration. However, some general data are presented in Table 3.1-11.

The IVA umbilical loop performs no gas processing. High-pressure oxygen is sent through plumbing and umbilicals to the pressure suit and then is vented from the suit. A water-to-water heat exchanger connects the liquid-cooled garment (LCG) through umbilicals to the space station cooling loop. IVA support requirements are oxygen for the flood flow suits, water for cooling the water in the LCG, and instrumentation cable for biomedical data and communications. The umbilical and suit weights are part of crew habitability and not the ECLSS. Also, the umbilical lengths (and therefore the weights) are a function of the vehicle configuration.

EVA support is provided by the Apollo portable life support system (PLSS) and the emergency oxygen purge system (OPS), which is an emergency backup to the PLSS. Table 3.1-11 lists various characteristics for the IVA and EVA systems.

For fire control, two types of fires were considered. For large fires, the affected pressure volume is evacuated and then dumped to space to extinguish the fire. For small fires, a less drastic method of control is provided. Each deck will have two firefighting units. Each unit consists of a CO_2 extinguisher, an oxygen bottle, and a full-face oxygen mask. Twenty-minute oxygen capability is provided. If a fire cannot be controlled in that time by two men, it is classified as a large fire. One firefighting unit weighs 13.6 pounds and occupies 0.7 ft^3.

3.1.11 ECLSS COMPLEXITY AND STATE OF DEVELOPMENT

To present a relative indication of the OLS ECLSS subassembly complexity, Table 3.1-12 was compiled. The actual units that make up a subassembly are broken out in the table, and the number of different major components are tabulated. In terms of maintenance, these major components give an indication of the possible number of inflight replaceable units (IFRU)

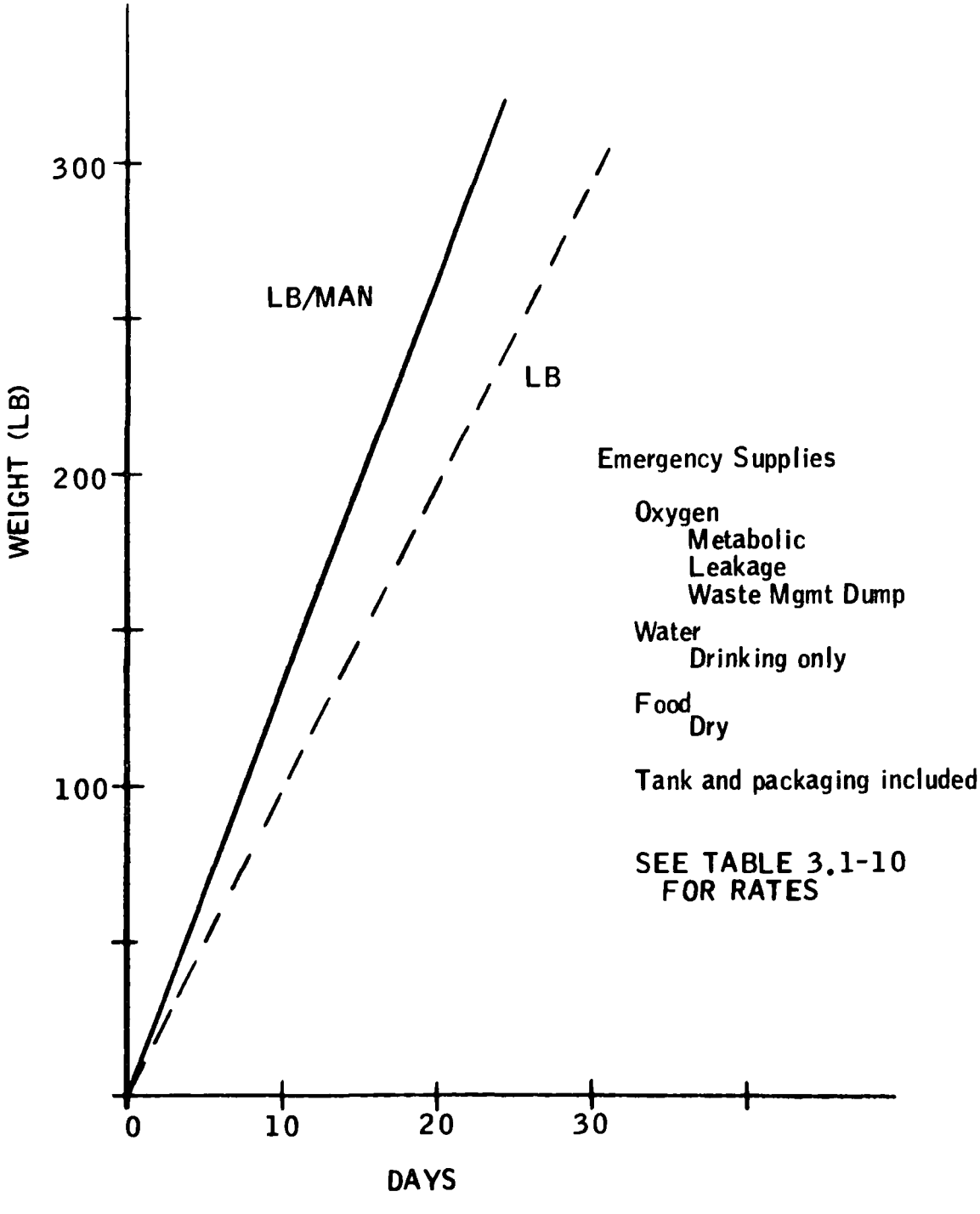


Figure 3.1-64. ECLSS Emergency Consumables

Table 3.1-10. Emergency Consumable Data used in Figure 3.1-64

Item	Total Rates		Detailed Rates		
	lb/man-day	lb/day			
Oxygen	3.68	9.8	Metabolic		
			Gas	1.84	lb/man-day
			Tank	1.84	lb/man-day
			Seal Leakage		
			Gas	4.7	lb/day
			Tank	4.7	lb/day
			Toilet Venting		
			Gas	0.2	lb/day
			Tank	0.2	lb/day
Water	6.95	0.0	Drinking Water (no wet food)		
			Water	6.32	lb/man-day
			Tank	0.63	lb/man-day
Food	2.41	0.0	Dry Food		
			Food	1.68	lb/man-day
			Packaging	0.73	lb/man-day

Table 3.1-11. IVA and EVA Data

Characteristics	Requirement
Physical	
IVA	
Weight	Installed 61.8 lb total (3 two-man connector units) Inital spares 30.0 lb No resupply
Power	
Maximum	500 watt (operating) (4 men attached)
24-hr average	0 watt
Volume	
Installed	1.2 ft^3 (3 two-man connector units)
Initial spares	0.4 ft^3
EVA	
Weight (1 unit)	PLSS: 80 lb with expendables OPS: 40.5 lb with expendables
Power	Self-contained
Volume (1 unit)	PLSS: 3.10 ft^3 OPS: 1.75 ft^3
Performance	
IVA	
Nominal O_2 flow	8 lb/man-hr
Inlet O_2 temperature	40-64 F
LCG pump flow	240 lb/man-hr
Inlet H_2O temperature	43 F
Heat load	2000 Btu/man-hr*
Operation time	4 hr
EVA	
PLSS	
Pounds O_2/charge	1.6
Charge pressure	1000 psia minimum
Temperature	70 to 90 F
Pounds H_2O/charge	11.0
Operation time	4 hr
Pounds LiOH/charge	5
Battery unit weight	5.5 (replace each time or recharge: 240 watt-hr)
OPS	
Pounds O_2/charge	5.7

*Three-hour limit for this maximum load

Table 3.1-12. Major Subassembly Complexity

Assembly, Subassembly Options	Components						Interfaces																
							ECLSS																
	Dynamic			Static			Atmospheric Store	CO_2 Management	Atmospheric Control	Active Thermal	Water Management	Waste Management	Hygiene	Food Management	Special Life Support	EPS	RCS	Structure	Environmental Protection	Crew Habitability
	Valve	Fan	Pump	Heat Exchanger	Chemical Beds and Reactors	Filters and Wicks																	
Atmospheric storage																							
High pressure	2	0	0	0	0	1		X	X						X								
Cryogenic	3	0	0	1	0	1		X	X	X								X	X				
NH_3/N_2H_4 decomp unit	5		1	1	3	1		X	X	X						X	X	X					
CO_2 Management																							
CO_2 Removal(a)																							
Molecular sieve	9	1	1	1	4	0		(b)	X	X						X		X		X			
Steam desorb res	7	1	1	2	2	2		(b)	X	X	X					X		X		X			
Solid amine	7	1	1	2	2	1		(b)	X	X						X		X		X			
H_2 depolarized cell*	√	√	0	0	√	0		(b)	X	X	X					X							
CO_2 Reduction(b)																							
Sabatier	12	0	1	2	1	1	X	(a/c)		X	X					X	X	X					
Bosch	10	1	1	2	2	1	X	(a/c)		X	X					X		X					
Solid electrolyte	9	0	1	2	4	0	X	(a)	X							X	X	X					
Water electrolysis(c)																							
Wick feed	15	0	1	1	3	2		(b)	X	X	X					X		X		X			
Solid polymer	6	0	1	2	3	1		(b)	X	X	X					X							
Gas circulation	10	1	2	2	3	1		(b)	X	X	X					X							
Atmospheric control																							
Temperature/ humidity	√	√	0	√	0	√		X		X				X		X				X	X		
Trace gas control																							
Catalytic oxidation and regeneration charcoal	7	1	0	0	8	0				(X)*						X	X						
Catalytic oxidation and sorbal bed	4	1	0		7	0				(X)*						X							
Monitoring system	√	0	0	0	√	√																	
Active thermal control	√	0	√	√	0	√ (in pumps)	X	X	X		X		X	X	X	X	X	X	X				
Water management																							
Vacuum distillation	26	0	2	1	2	2			X	X		X	X	X		X							
Vapor compression	20	0	2	0	5	4			X			X	X	X		X		X					
Air evaporation	29	1	2	2	3	4			X	X		X	X	X		X		X					
Vapor diff/ compress	55	0	2	0	3	2			X	X		X	X	X		X							
Reverse osmosis	20	0	1	0	1	3			X				X			X							
Waste management	√	√	√	√	√	√			X		X		X			X		X					
Hygiene	√	√	√	√	√	√				X	X	(X)**				X							
Food management	√	√	√	√	√	√			X			X	X	-	X	X							
Special life support	√	0	√	√	√	√	X			X	X			X		X		X					

*Waste heat use possible
**If use disposal clothes

required for maintenance. Reliability relationships also can be inferred from the relative number of static-versus-dynamic components. Dynamic components have a higher probability of failure. Complexity is further indicated by the number of subassembly and assembly interfaces required. The EPS interface refers to the need for power, RCS interfaces refer to gas interfaces such as H_2 or CH_4. Structure interfaces indicate that vacuum areas or sealed areas are required or that the unit needs a vent line to vacuum. An environmental protection interface implies that heat leakage affects the ECLSS unit or subassembly. Crew habitability interfaces indicate that output or inputs of the unit are directly matched to metabolic or other human requirements.

Figure 3.1-65 presents development risk by presenting the year of availability, or years after go-ahead, for the various ECLSS options or subassemblies. Usually, development status is a strong indicator of cost because each stage of development tends to have a fixed cost.

3.1.12 DURABILITY

Concept evaluations for the long duration lunar orbit missions must consider durability and flexibility as more significant criteria than the standard (and more readily quantifiable) weight, power, volume criteria. To aid in the concept evaluations, several significant ECLSS components were rated relative to durability. The durability assessments reflect the complexity of the components as listed in Table 3.1-12.and the ECLSS equipment studies of Reference (AILSS) and space station, Reference (Volume III).

The durability assessments are presented in Table 3.1-13 for the significant ECLSS equipment. A maximum durability rating of 25 was chosen and the various concepts were rated relative to each other. The 25 point total durability rating was determined from the following contributions:

8	Mean Time Before Failure (MTBF)
4	Scheduled Maintenance
12	Unscheduled Maintenance
1	Overall View
25	Total Maximum Rating

The MTBF used to estimate the MTBF rating and the maintenance time estimates, based on 500 hours of equipment operation, are listed on Table 3.1-13. Durability comparisons presented in Table 3.1-13 are valid only between concepts of one function. A further definition of the elements which are a measure of durability is given below:

1. MTBF - Since this is a measure of the probable number of failures that will occur, it gives an indication of how often the operation of the equipment will be interrupted for some remedial action. One-third of the durability measure is assigned to this parameter.

2. Scheduled Maintenance Time - This is a measure of the amount of crew time expected to be expended for replacement of limited-life items and expendables. Since it is more easily predicted than unscheduled maintenance time, only one-sixth of the durability measure is assigned to this parameter.

Figure 3.1-65. Major Subassembly State-of-the-Art

Table 3.1-13. Durability

FUNCTION	CONCEPT	TOTAL SCORE	OVERALL VIEW ±1	MTBF Hours	MTBF 8 Max Rating	SCHED. MAINT. Hours	SCHED. MAINT. 4 Max Rating	UNSCHED. MAINT. Hours	UNSCHED. MAINT. 12 Max Rating
GAS STORAGE	HIGH PRESS.	25	+1	328,000	8	0	4	0.05	12
	CRYOGENIC	20	0	80,600	6	0	4	0.2	10
	N_2H_4	21	-1	112,000	7	0	4	0.1	11
	NH_3	21	-1	112,000	7	0	4	0.1	11
CO_2 REMOVAL	MOLE SIEVE	21	0	8,415	6	0	4	2.9	11
	STEAM DES.	20	0	12,550	8	16	0	2.0	12
	HS-B AMINE	21	0	11,479	7	8	2	2.1	12
	H_2 DEP CELL	24	+1	11,750	7	0	4	2.3	12
CO_2 * REDUCTION	SAB. (CH_4 DUMP)	20	0	4,740	8	26.8	4	11.2	8
	SAB. (CH_4 CRACK)	13	-1	4,310	6	126.8	1	11.7	7
	BOSCH	15	0	4,450	6.5	126.8	1	11.5	7.5
	SOLID ELECT.	14.5	-1	3,480	2	110.8	1.5	6.9	12
H_2O ELECTROLYSIS	WICK FEED	18.5	0	5,700	6.5	10	1	3.1	11
	CIRC. ELEC.	20	0	5,300	6.0	4	3	3.0	11
	CABIN AIR	18	0	2,500	3.5	0	4	3.6	10.5
	GAS CIRC.	18	0	5,700	6.5	8	2	4.2	9.5
	SOLID POLY.	20.5	0	7,100	8.0	8	2	3.3	10.5
CONTAMINANT CONTROL	CHARCOAL	21.5	+1	78,100	7	17.5	1.5	0.3	12
	REGEN CHAR	19	-1	96,500	8	13	2	0.5	10
	SORB. BED	22	-1	78,000	7	0.7	4	0.3	12
URINE RECLAIM	AIR EVAP	21	0	20,000	8	207	1	1.8	12
	VAPOR COMP.	17	0	13,600	5	131	4	3.5	8
	VAC. DISTIL.	17	0	16,000	6	56	3	2.3	11
	VAPOR DIFF.	18	0	14,000	5	38	3	2.6	10

* ALL CONCEPTS INCLUDE WATER ELECTROLYSIS EXCEPT SOLID ELECTROLYTE

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3. Unscheduled Maintenance Time - This is a measure of the amount of crew time expected to be expended for replacement of randomly failed components. This parameter is assigned one-half the durability measure, three times the amount assigned to scheduled maintenance because:

   a. The estimates for unscheduled maintenance time are based solely on "nut turning" time, and do not include the time required to isolate the malfunction or that required to verify that the maintenance has solved the problem.

   b. Random failures occurring may also affect other components or subsystems.

4. Resupply of Spares and Expendables - Whereas the previous two elements were a measure of the "time" penalty, the weight and volume of spares for limited-life items and random failure items plus the weight of expendables are the weight and volume penalties associated with the lack of durability. These quantities were added to the fixed weight and volume curves. They are not considered further in the durability measure.

3.1.13 INTERFACES

The purpose of this section is to discuss in one section all the effects, relations, and interfaces the OLS ECLSS subassemblies have with each other and with other OLS subsystems. Crew overloading is discussed under the crew habitability interface. Also presented is a discussion of the effects of artificial gravity upon the ECLSS.

ECLSS Subsystem-to-Subsystem Interfaces

Atmospheric storage size is determined by the types of gases the ECLSS must store. If oxygen is recovered, it does not have to be stored. Thus, atmospheric storage is related to CO_2 management. Because some CO_2 management concepts (CO_2 reduction units) require hydrogen, the type of concepts selected for CO_2 management influence atmospheric storage. Special life support emergency gas storage also impacts atmospheric storage.

CO_2 management is affected by the atmospheric storage subassembly in terms of oxygen leakage and possibly hydrogen makeup. If an electrolysis concept is chosen, the oxygen leakage can be made up. Also, water management is impacted because additional water would be required for the oxygen leakage makeup. Atmospheric control and active thermal control impact CO_2 management because the vehicle atmospheric conditions (as humidity level) and the type of cooling and heating (as waste heat loop) influence the CO_2 management sizing.

Atmospheric control is basically sized by the breakdown between air cooling and coldplate cooling (active thermal control). The CO_2 management subassembly requires humidified or dry air, depending upon the various concepts selected. This influences humidity control requirements. Food management also influences humidity control because cooking of foods involves release of humidity.

Active thermal control is affected by almost all of the ECLSS subassemblies because most involve equipment that must be cooled (or heated if a waste heat loop is available). The requirements of the other ECLSS subassemblies greatly determine the cabin-loop portion of the active thermal control.

Water management is influenced by those subassemblies that use water and produce waste water: atmospheric control (humidity); waste management (urine); hygiene (sinks, showers); and food management (water for drinking and food preparation). CO_2 management impacts water management because make-up water may be required for oxygen leakage makeup. Also, several water management concepts impact active thermal control.

The ECLSS subassembly interfaces for waste management only involve atmospheric control (air flow through units needs to be purified), water management (urine is recovered), and food management (major trash source). Hygiene interfaces with active thermal control (water cooling), water management, and waste management (disposable clothes).

Food management interfaces with atmospheric control (cooling of galley and humidity removal), water management, waste management (food waste, packaging waste), and special life support (emergency food). The special life support subassembly only interacts with atmospheric storage (emergency and IVA/EVA oxygen), water management (emergency water), and food management (emergency food). Also, the IVA umbilical loop interfaces with active thermal control because a water to water heat exchanger is required.

ECLSS Subsystem-to-System Interfaces

Electrical Power System (EPS)

If EPS selects a system that requires storage of oxygen or hydrogen, then atmospheric storage is impacted. Either ECLSS, EPS, or RCS would be made responsible for gas storage. EPS also impacts active thermal control. If EPS produces usable waste heat, then active thermal control may have a waste heat loop. Also, the EPS power source concept influences radiator design because fuel cells require high-temperature radiators, while solar arrays require no radiators.

Reaction Control System (RCS)

As mentioned for the EPS, RCS impacts atmospheric storage of oxygen and hydrogen if RCS-chosen concepts utilize O_2 or H_2. Also, if a Sabatier reactor is chosen for CO_2 reduction in CO_2 management, then RCS may be able

to use the CH_4 waste product for resistojet use, depending upon the RCS concept selection and thrust requirements.

Structure and Vehicle Configuration

Cryogenic gas storage requires vacuum areas, and hydrazine requires a sealed pressurized area in which to operate. Thus, atmospheric storage impacts structure. Also, the freon loop (active thermal control) requires a sealed area in which to operate because freon leakage into the catalytic oxidizer (trace gas control) will cause toxic gases to be produced. Vehicle modularization shades vehicle areas and impacts available radiator area (active thermal control).

Most of the ECLSS subassemblies contain equipment that needs to vent either waste products or noncondensable gases to space. The need for lines and structure penetrations is an important interface.

Vehicle configuration affects atmospheric storage because seal length of docking ports, airlocks, etc., affect leakage, and the volumes of the ports, etc., affect the pumpdown system (if one is required).

Environmental Protection

Atmospheric storage interfaces with environmental protection if cryogenics make up part of the system. Also, any other ECLSS tanks that are placed in vacuum areas will need appropriate thermal coatings or insulation. Atmospheric control and active thermal control strongly interface with environmental protection. Temperature and humidity requirements determine the wall dew (or condensation) point which must not be exceeded. Vehicle heat gain or loss affects the cooling, heating, and heat rejection requirements.

The meteoroid protection function is required to protect ECLSS radiators and to prevent hole punctures that will cause rapid gas pressurization loss.

Crew Habitability

Crew habitability directly affects most of the ECLSS because ECLSS equipment is sized to remove man's waste products and to provide his metabolic requirements. Also, the waste management, hygiene, and food management subassemblies are designed to give man psychological and functional comfort. Special life support is present mainly for man's protection in the hostile space environment.

Experiments

The ECLSS may provide all of its functions for experiments or only a few functions (supply water and gas). The amount of interfacing depends upon the type of experiments to be performed as well as the servicing requirements. In some cases, the basic vehicle ECLSS can service some experiments

with little impact. However, other experiments could impact the ECLSS enough to make a separate experiment ECLSS worthwhile.

Summary

The parametric data given in this report are of two types: (1) component curves; and (2) total system curves. The ECLSS is generally redundant. Most large space vehicles will have two pressure volumes for safety. Thus an 8-man vehicle generally would have two 8-man ECLSS systems. For the potential OLS crew overloads of 70 days, a triple ECLSS system may be required. The ECLSS for the normal crew load usually would be redundant, only one system per volume. Thus, for a crew size twice the normal size, all ECLSS units would be turned on to full capacity. In this situation, no stand-by redundant units would exist. If a critical unit failed, severe problems could exist. Therefore, the major assembly components will probably need one active and two redundant units. (Another solution is to size single units based upon the crew overload size and make them redundant.) Those total system parametric curves that already include redundancy would require multiplication by 1.5 to obtain double redundancy. The other curves that present characteristics on a unit basis would be tripled instead of doubled. These crew overload rules are general. Specific cases may deviate. The EOSS, for example, has one electrolysis unit with four stacks. Only three stacks are required normally. Thus, this unit is redundant on a component level rather than on the unit level. Such results might be applied to various OLS components. However, the parametric data in this report are at the unit and subassembly levels and not the component levels.

Table 3.1-12 summarizes various ECLSS subsystem to system interfaces. Figure 3.1-66 presents an overall schematic view of possible OLS ECLSS interfaces.

3.1.14 IMPACT OF ARTIFICIAL GRAVITY

Artificial gravity decreases ECLSS complexity. Units designed for zero gravity will work better with gravity. If artificial gravity could be guaranteed all of the time, ECLSS cost and weight would decrease. Zero-gravity condensers, urinals, toilets, sinks, and showers present design problems. Their design is simplified for artificial gravity. Air flow patterns, fluid dispensing, and fire detection are simpler with artificial gravity. Fluid tanks do not require bladders, and fluid lines can be bled of dissolved gases when a gravity gradient exists.

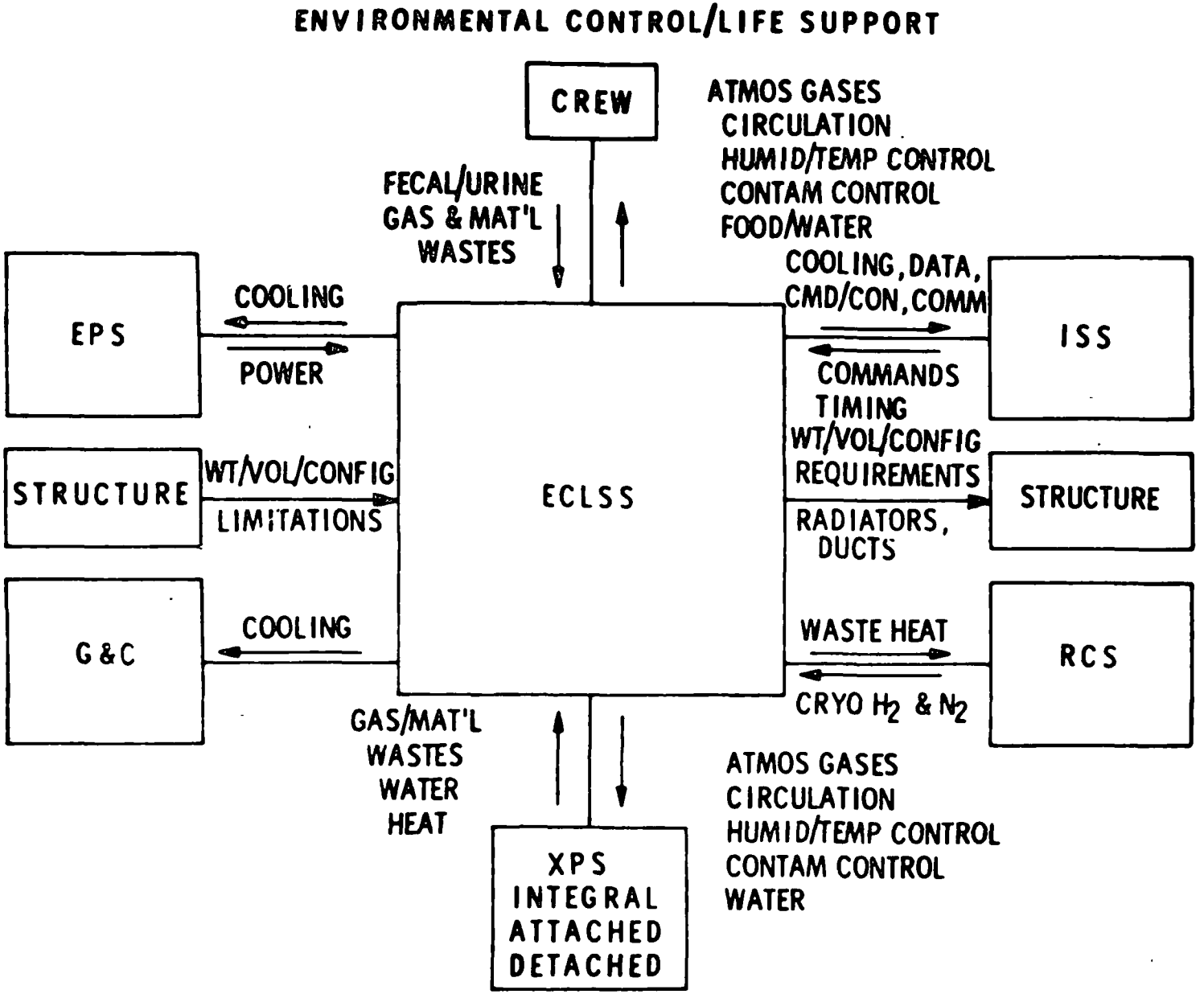


Figure 3.1-66. ECLSS Interface Functional Block Diagram

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3.2 ELECTRICAL POWER SUBSYSTEM (EPS)

The primary function of the EPS is to provide continuous electrical service for the Orbital Lunar Station (OLS). To accomplish this, the EPS shall generate, regulate, condition, store, control, and distribute electrical power required for the life of the OLS (10 years), including power for backup and emergency contingencies. Besides power distribution, the EPS shall provide for the general lighting (interior and exterior) and the sequential mechanization or power switching needs of the OLS. In addition, the EPS shall be capable of transferring power to peripheral equipment, lunar landing vehicles, and docked logistics vehicles (cislunar tugs, etc.). Major EPS functions are summarized below:

Primary power generation
Secondary power generation
Energy storage
Power conditioning
Distribution control and wiring
Lighting

3.2.1 STUDY OBJECTIVES

Study objectives are to present the parametric trade data and rationale necessary for a preferred EPS selection. In selecting the EPS, it is first necessary to establish an initial set of OLS power and performance requirements and a number of candidate concepts and then perform trade analyses leading to the EPS concept finalist. Power conditioning, distribution, and OLS lighting are not driving functions on the electrical power source selection. For this reason trade analyses for these functions will not be performed in this study.

3.2.2 REQUIREMENTS

Mission Requirements

The initial mission requirements and constraints for the OLS are defined in Volume III. Briefly, the EPS must supply continuous conditioned power for all mission phases and must be capable of both manned and unmanned operations. Pertinent EPS requirements follow:

a. Nominal power supplied with one credible failure
b. Critical functions but reduced power with two failures
c. Separate, independent system for emergency
d. Safety features - automatic switching on critical loads, isolation of critical life support
e. Normal maintenance without primary power shutdown
f. Common electrical interface at docking ports
g. Modes of operation

Modes of Operation	Representative OLS		Derivative OLS	
	Period	Power	Period	Power
(1) Unmanned launch and EO premanning	5 days	2.0 kw	5 days 20 min	100 w(C1A) 2 kw(PM+C1A)
(2) Manned EO operations	2 days	13.5 kw	1 day	2.5 kw
(3) Unmanned EO, translunar flight and lunar orbit operations	7 days	5.5 kw	7 days	2.5 kw
(4) LO quiescent storage	1 year	1.9 kw	1 year	1.9 kw
(5) Initial manned LO operations	2 days	15.0 kw	16 days	15 kw
(6) Manned LO operations	continuous	20.0 kw	continuous	20 kw
(7) *Backup (fail degraded)	90 days	13.5 kw	90 days	13.5 kw
(8) *Emergency	30 days	4.0 kw	30 days	4.0 kw
(9) Earth eclipse	3.5 hr	15.0 kw	3.5 hr	15 kw
(10) Solar flare events	72 hr	10.0 kw	72 hr	10 kw

*Based on 8-man crew

EPS Performance Requirements

The EPS for the OLS has the following operational performance requirements:

1. Operate unmanned for launch and both premanned and manned after launch.
2. Operate automatically when manned with crew override ability.
3. Fail operational, fail safe for critical and essential loads; fail degraded for general-purpose (not critical) loads.
4. Provide emergency power upon loss of primary power for 48 hours in earth orbit and 30 days in lunar orbit.
5. Provide EPS hardware to operate from the information subsystem (ISS).
6. Operate for all critical or essential loads with equipment compartment pressurized or depressurized.
7. Design all hardware including its installation so it can be maintained.

Power Profile

The EPS must supply a 24-hour average power level, which is dependent upon mission phase and crew size. Power requirements for 4-, 8-, and 12-man crews are given in Tables 3.2-1 to 3.2-3, respectively. An updated version of the 8-man electrical load analysis appears in Volume IV based on ECLSS subsystem data that was not available at the time Tables 3.2-1 through 3.2-3 were constructed. For the comparative purposes of this section, the data presented here is adequate. A typical representative OLS mission power profile based on an 8-man crew is shown in Figure 3.2-1. A typical lunar orbit load profile (8-man) is illustrated in Figure 3.2-2. The power profile is based on the following assumptions:

1. The maximum use of experiment power will be on a one-shift basis, and some operational loads will be staggered to other shifts (i.e., electrolysis, portion of the lighting, G&N, etc.).

2. The batteries/electrolysis will be fully recharged/completed each orbit.

3. The weighed system and battery charging/electrolysis efficiencies are 80.5 and 58.8 percent, respectively.

4. The orbital characteristics are:

Earth	Lunar
258 nautical-mile circular orbit Inclination angle = 31.6 degrees Orbit time = 1.569 hours (worse case = 38 percent dark)*	60 nautical-mile circular orbit Inclination angle - 90 degrees (polar) Orbit time = 1.987 hours (worse case = 39 percent dark)*

*Does not include eclipses of the sun by the earth.

5. Several times per year, the OLS may experience periods of shadowing because of the eclipse of the sun by the earth (eclipse duration $\leq$ 210 min.). For these periods, solar-powered EPS's will be required to revert to the emergency mode of operations. Nuclear-powered EPS's will not be affected.

Table 3.2-1. OLS Electrical Load Analysis (Four-Man Crew)

Subsystem/Equipment	Sustained*	Average	Backup	Emergency	Premanned
Atmospheric storage	400	232	40	0	0
CO_2 management	3,200	1,870	1,870	0	0
Atmosphere control	1,328	1,250	1,250	753	108
Active thermal control	175	175	175	175	175
Water management	345	290	290	0	0
Waste management	854	54	54	13	0
Hygiene	654	55	55	0	0
Food management	2,348	407	407	0	217
Special life support	500	0	0	0	0
Total ECLSS	9.804	4,333	4,141	941	500
Communication/tracking	2,409	1,810	721	310	310
Data management	1,165	977	501	255	255
Display and control	1,455	730	650	350	0
Total ISS	5,029	3,517	1,872	915	565
CMG's	1,370	470	470	0	0
Navigation	304	139	128	98	34
IMU	198	161	161	198	198
RCS control	0	2	2	2	40
Landmark tractor	80	40	40	40	40
Total G&C	1,952	812	801	298	312
RCS	554	5	5	5	5
Lighting	2,400	1,200	500	500	25
Experiments	6,000	4,000	800	0	0
Total electrical loads (less EPS and distribution losses)		14,867	8,119	2,659	1,407

*The sustained load represents a feasible load that can be applied for up to one hour duration and not more than twice in a 24-hour period.

Table 3.2-2. OLS Electrical Load Analysis (Eight-Man Crew)

Subsystem/Equipment	Sustained*	Average	Backup	Emergency	Premanned
Atmospheric storage	800	467	73	0	0
CO_2 management	6,390	3,740	3,740	0	0
Atmosphere control	2,660	2,510	2,510	1,500	216
Active thermal control	350	350	350	350	350
Water management	690	579	579	0	0
Waste management	1,706	107	107	25	0
Hygiene	1,309	111	111	0	0
Food management	4,695	812	812	0	435
Special life support	500	0	0	0	0
Total ECLSS	19,100	8,676	8,282	1,875	1,001
Communication/tracking	2,409	1,810	721	310	310
Data management	1,165	977	501	255	255
Display and control	1,880	1,155	650	350	0
Total ISS	5,454	3,942	1,872	915	565
CMG's	1,370	470	470	0	0
Navigation	304	139	128	98	34
IMU	198	161	161	198	198
RCS control	0	2	2	2	40
Landmark tractor	80	40	40	40	40
Total G&C	1,952	812	801	298	312
RCS	554	5	5	5	5
Lighting	3,600	2,400	1,000	500	25
Experiments	6,000	4,000	800	0	0
Total electrical loads (less EPS and distribution losses)		19,835	12,760	3,593	1,908

*The sustained load represents a feasible load that can be applied for up to one hour duration and not more than twice in a 24-hour period.

Table 3.2-3. OLS Electrical Load Analysis (12-Man Crew)

Subsystem/Equipment	Sustained*	Average	Backup	Emergency	Premanned
Atmospheric storage	1,200	700	120	0	0
CO_2 management	9,583	5,604	5,604	0	0
Atmosphere control	3,637	3,474	3,474	1,610	216
Active thermal control	350	350	350	350	350
Water management	1,035	867	867	0	0
Waste management	1,706	107	107	25	0
Hygiene	1,309	111	111	0	0
Food management	6,785	950	950	0	435
Special life support	500	0	0	0	0
Total ECLSS	26,105	12,163	11,583	1,985	1,001
Communication/tracking	2,409	1,810	721	310	310
Data management	1,165	977	501	255	255
Display and control	1,880	1,155	650	350	0
Total ISS	5,454	3,942	1,872	915	565
CMG's	1,370	470	470	0	0
Navigation	304	139	128	98	34
IMU	198	161	161	198	198
RCS control	0	2	2	2	40
Landmark tractor	80	40	40	40	40
Total G&C	1,952	812	801	298	312
RCS	554	5	5	5	5
Lighting	4,700	3,600	1,500	500	25
Experiments	6,000	4,000	800	0	0
Total electrical loads (less EPS and distribution losses)		24,522	16,561	3,703	1,908

*The sustained load represents a feasible load that can be applied for up to one hour duration and not more than twice in a 24-hour period.

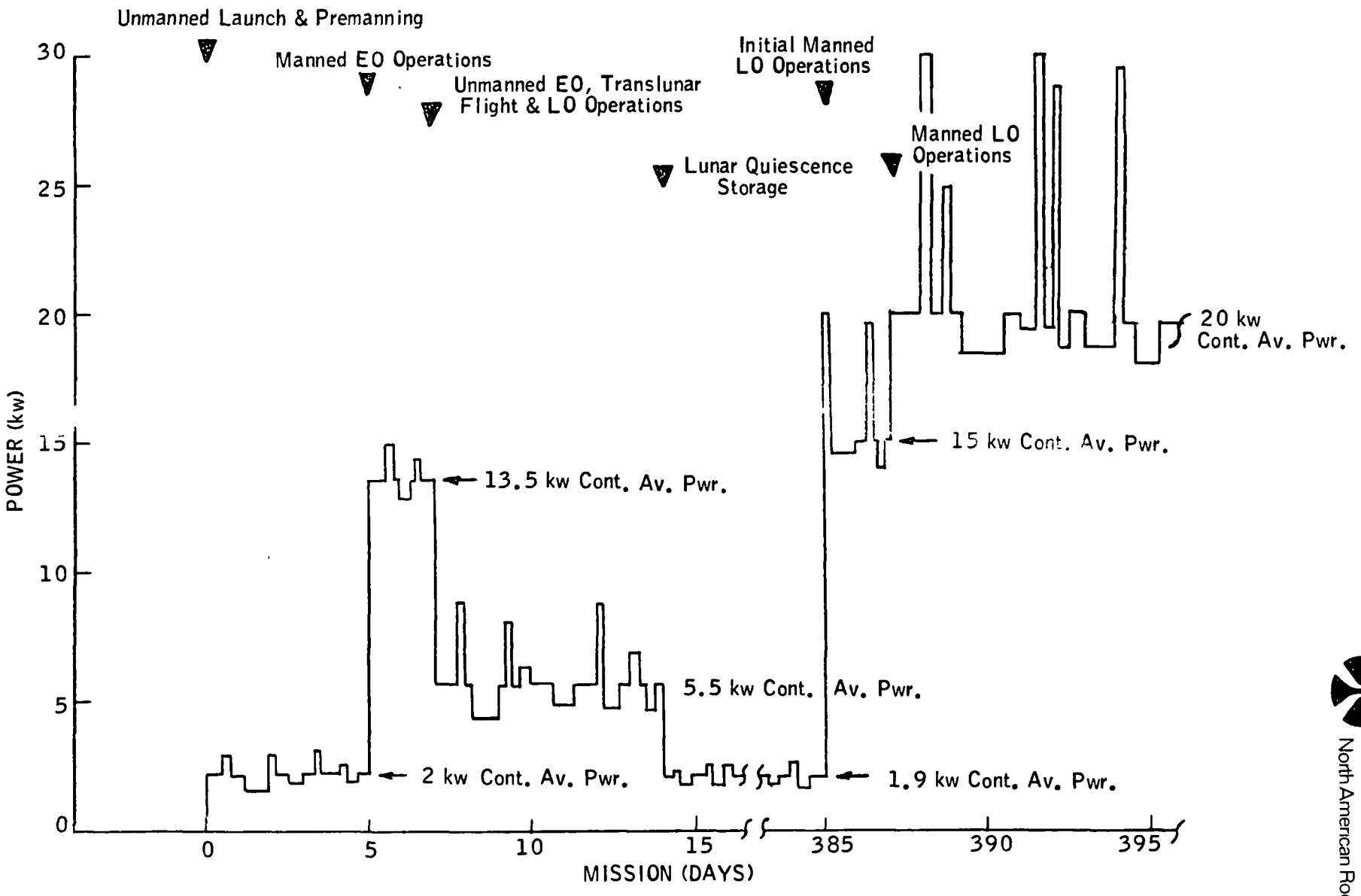


Figure 3.2-1. Typical OLS Mission Power Profile (8-Man)

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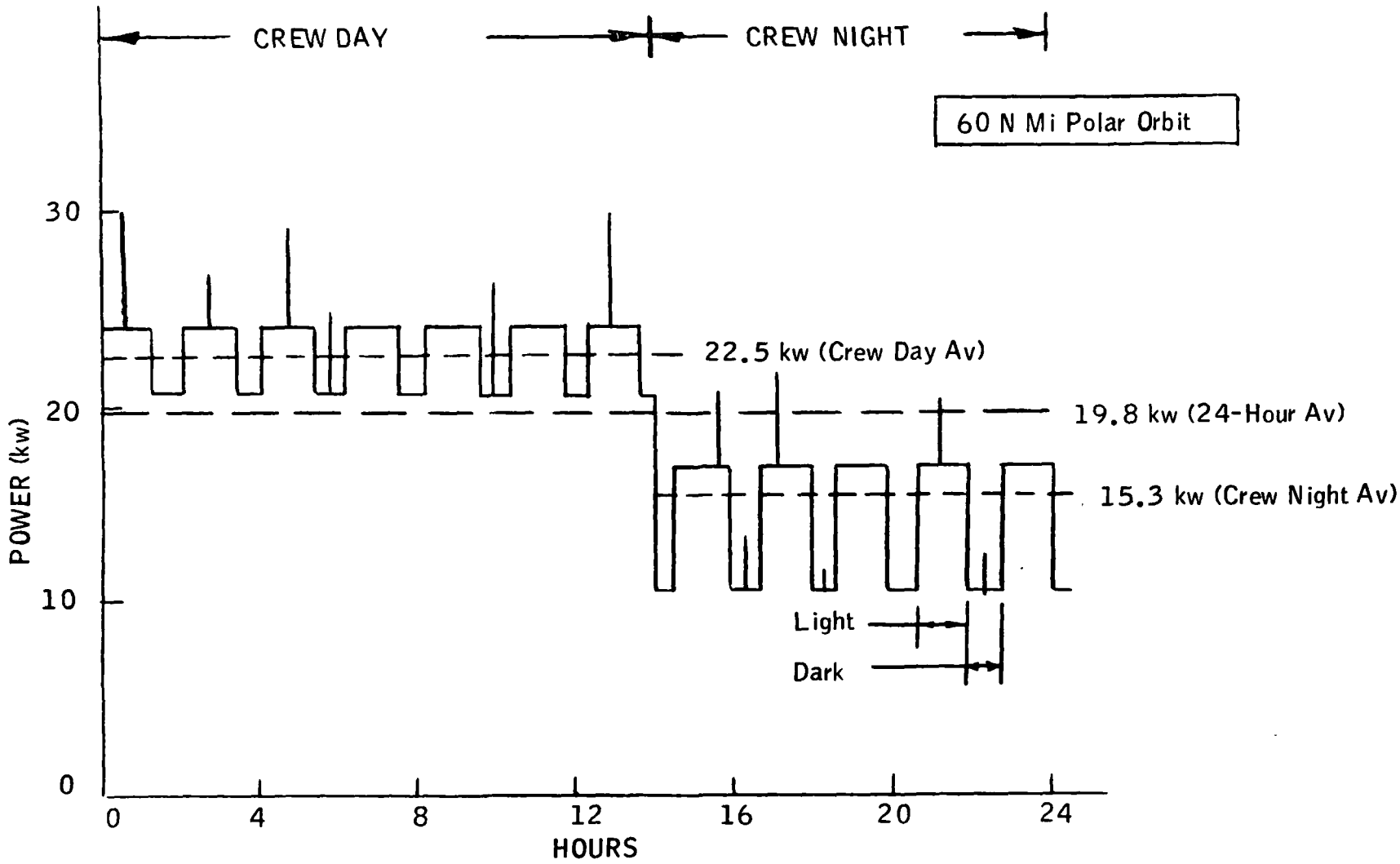


Figure 3.2-2. Typical Orbit Load Profile for OLS Profile (8-Man)

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3.2.3 ENERGY SOURCES

Solar (light), chemical, and nuclear energy sources are considered potential OLS candidates. Because of the unique mission requirements or natural phenomenon, a combination of these energy sources may be required to meet the electrical power requirements.

Requirements for each candidate energy source is dictated by the conversion efficiency and total electrical load demands. Electrical loads are sensitive to both mission phases and the size of the crew.

Solar Energy

Solar energy (sun), available only during the light orbital periods, can be converted to electricity using static (solar cells) or dynamic (collectors) (heat engine) concepts. The dynamic concept was eliminated because of the large size requirements and the stowage, deployment, orientation, and pointing accuracy problems associated withe collectors.

Use of solar energy in conjunction with solar cells (static concept) emerged as a strong energy source candidate because it requires no consumable fuel and because technology for its conversion is well advanced.

Solar energy systems have a number of drawbacks; i.e., impact of cell equilibrium temperature, radiation degradation, power requirement for solar array orientation if employed, requirements for power conditioning and power energy storage equipment, etc. Thus, not only must the solar array panel be sized for end-of-the-mission performance, but also for providing electrical energy needed to recharge the storage devices required during the dark period of operation. The factors impose weight and cost penalties on solar array use. On the other hand, solar array systems have demonstrated high reliability for space applications.

Chemical Energy

The two electrochemical energy sources considered for the OLS are fuel cells and batteries. It is difficult to isolate the chemical energy sources and the associated conversion functions (i.e., batteries). For this reason, the electrochemical candidates will be treated as a direct electrical energy source.

Fuel cells are high thermally efficient devices for the direct conversion of chemical energy into electricity. Two reactants (fuel and an oxidizer) supplied to the fuel cell are consumed in an electrochemical reaction producing electricity, water, and heat. The hydrogen-oxygen fuel cell, because of its high thermal efficiency and water production capability, becomes the most practical means of electrical power generation for manned orbiting applications in cases where the water byproduct is required to fulfill mission requirements. (Water conversion efficiency equals approximately 98 percent; hence, little fuel penalty is associated with fuel cells.) If the fuel weight is chargeable to the EPS (byproduct water considered a waste in this case), then the fuel cells are less attractive as a means of electrical power generation.

The OLS mission constraints (resupply at 109 days and storage for 180 days) and the relatively high power levels required eliminate fuel cells as a primary source candidate on the basis of a high weight resupply penalty. An average OLS load of 20 kw would require approximately 57,000 pounds of reactants (neglecting boil-off losses). Sample calculations follow:

$$\frac{20 \text{ kw} \times 109 \text{ days} \times 24 \text{ hours} \times 0.87 \text{ lb/kwh}}{0.80 \text{ system efficiency}} = 57{,}000 \text{ lb}$$

Tankage weights could increase this value by about 25 percent.

Fuel cells may be used with solar energy systems for supplying power during the dark periods of operation (including eclipses). In this case, a portion of the water byproduct may be used by the ECLSS to meet crew water requirements and the balance dumped overboard. The combination of fuel cells and electrolysis cells (fuel cells operating in a back-to-back regenerative mode) is a strong candidate for supplying power during dark periods in conjunction with solar arrays and for meeting peak power requirements with nuclear sources. The consumable fuel requirements would be minimized to cycle losses only (about 2 percent). Fuel cells can also be used as energizing power sources (i.e, emergency/standby).

Secondary alkaline batteries are potential energy sources for both the storage and generation of electrical power when used with solar array systems and as peak power batteries employed to optimize selection of the fuel cells and nuclear power sources. In addition, batteries can function as an emergency power source. Batteries as a primary energy source candidate has been eliminated by the initial and resupply weight penalties. Sample calculations based on an average 20 kw load and a depth of discharge (DOD) of 100 percent are as follows:

$$\frac{20 \text{ kw} \times 10^3 \times 109 \text{ days} \times 24 \text{ hours}}{0.80 \text{ efficiency} \times 60 \text{ wh/lb}} = 1{,}090{,}000 \text{ lb}$$

The batteries must be replaced each resupply period.

Nuclear Energy

Two nuclear energy sources (reactor and radioisotope) are prime candidates for the primary EPS. Nuclear reactor energy sources were eliminated as sources for secondary and emergency EPS's, because nuclear safety requirements result in extremely heavy energy sources when considered in relationship to the relatively low-level power requirements of the secondary and emergency EPS's. Two nuclear reactor design concepts were considered: a reference zirconium hydride reactor and an advanced high-temperature reactor. The high-temperature fast reactor was rejected, because it is unlikely that it would be ready for application by the 1983 OLS IOC date.

Radioisotope energy source concepts are advanced to the stage that OLS power requirements ($\leq$ 25 kw) can with reasonable confidence be met by 1983. Studies by NR on the EOSS favor Pu 238 as the prime fuel contender. The high cost for Pu 238 ($\geq$ \$540/Watt$_t$) is a paramount parameter in source selection. The OLS would require about 100 kw$_t$ heat sources costing about \$54M based upon two heat sources and a 25 percent power conversion cycle efficiency. At present, Pu 238 is available only in limited amounts but is expected to be available in the required amounts by 1983. Safety aspects (i.e., shielding, controlled reentry and recovery, end-of-life disposal, etc.) present design problems, the solutions of which result in high-weight penalties for the heat source. However, the resupply requirements are low, because the source has a relatively long half life ($>$ 86 years).

3.2.4 ENERGY CONVERSION

The function of energy conversion equipment is to convert the source energy into raw electrical energy. Because this source energy may be heat, light, or electrochemical by nature, conversion equipment will vary as a function of the source energy (i.e., solar photovoltaic arrays for light energy sources, thermodynamic cycles and thermoelectric devices for nuclear energy sources, and batteries and fuel cells for electrochemical energy sources). The power conversion methods (both static and dynamic) are discussed in the following paragraphs.

Solar Photovoltaic (static) Concepts

A number of solar array configurations have been considered in previous NR studies. Several different design approaches are possible for large area solar array systems. Four solar array concepts were investigated and compared to determine performance parameters. These are:

1. General Electric roll-up solar array
2. Hughes flexible roll-up solar array
3. Boeing large area solar array
4. ATM array

General Electric Roll-Up Solar Array

The General Electric concept for the roll-up array was evolved during the feasibility study (JPL Contract 951970). This concept features two-array substrates that are deployed by a single BI-STEM deployable boom. The array structural stiffness is provided by tension in the two substrates. In the deployed state, the array has a single-point attachment at the vehicle interface. A calculated power-to-weight ratio of 32.3 watt/pound for this 250 ft^2 array is made possible by the use of beryllium for the storage drums and the leading edge member.

Hughes Flexible Roll-Up Solar Array

A 1.5-kw roll-up array, which has a power-to-weight ratio of 21 watt/pound, is under development by Hughes (Air Force Contract F33615-68-C-1676). This concept deploys two substrates from a common drum with four BI-STEM rods mounted in two actuators. This contract culminates in a flight demonstration of the array in the third quarter of 1971. In addition to the basic roll-up array, the contract includes the associated two-axis orientation and control system along with the necessary power conditioning, energy storage, and instrumentation.

Boeing Large Area Solar Array

The Boeing large area solar array is a 1250 ft^2 (gross module area) folding-panel array, which has a power-to-weight ratio of 21 watt/pound. This concept utilizes a beryllium framework panel with the solar cells mounted on a fiberglass tape substrate, which is stretched on this frame. The preliminary design of this system was completed in October 1968 under Contract No. 951934.

ATM Array

The ATM array, which has a power-to-weight ratio of 2.76 watts/pound, is composed of four wings with a total module of 1230 ft^2. This array system will be flown on the NASA SKYLAB Program. The array consists of aluminum honeycomb panels that are hinged to each other for deployment by a scissor mechanism.

Comparison

A qualitative comparison of these four large area array concepts was made to determine the configuration, which is best suited for application on lunar orbital missions based on a set of mission related performance areas. A three-point rating system has been used to rank each candidate against each performance criteria. Each of these performance factors is discussed along with the rationale used to rank each array concept. The results of this comparison are shown in Table 3.2-4.

Table 3.2-4 Solar Array Comparison Matrix

Performance Factor	GE Roll-Up	Hughes Roll-up	Boeing Fold-out	ATM Array
Maximum productivity				
(wt/lb)	1	2	2	3
(wt/ft^3)	1	1	3	3
Cost ($/kw)	2	2	3	1
Minimum constraint imposed on operation	1	1	3	3
Minimum degradation and resupply	2	2	2	1
Maximum ability to achieve primary power operation (reliability)	1	2	2	1
Code: 1 = Best; 2 = Good; 3 = Worst				

Maximum Productivity

This generalized performance area has been divided into two specific measures of productivity (i.e., power-to-weight ratio and power-to-stowed-volume ratio). The power-to-weight ratio is a function of the basic array design if it is assumed that the solar cell and cover glass are the same for each approach. The GE roll-up array ranks "best", because the system has the highest power-to-weight ratio (30 watts/pound). Both the Hughes roll-up and the Boeing fold-out arrays have a power-to-weight ratio of approximately 21 watts/pound. The ATM array is the obvious choice for the last place because of the 2.76 watts/pound ratio. The scaling up of the existing concepts to the sizes required for the OLS should not change the relative ranking of these concepts. Also, the structural modifications required to withstand the artificial gravity modes of operation should have similar weight penalties for all approaches if such an operation becomes a requirement.

The stowed volume of the array also is an important measure of the performance of each concept. Both roll-up array approaches have been ranked "best" because of the inherent compactness of these designs (approximately 300 watts/ft^3). By comparison, both fold-out panel approaches have large stowed volume requirements (approximately 60 watts/ft^3).

Cost

A quantitative comparison of array system costs would be difficult to determine; however, the relative ranking of each concept can be estimated based on present development status and inherent hardware costs associated with each design. The ATM array is ranked as the lowest cost because of its advanced state of development and relatively low hardware cost. Both roll-up array approaches are presently in the same state of development (i.e., design qualification of a prototype model). Both of these systems should have approximately the same hardware costs. The Boeing fold-out array must be ranked as the highest program cost because of the lack of present funding and the high hardware costs associated with the formed beryllium framework.

Minimum Constraint Imposed on Operations

Both roll-up array approaches are ranked "best" in this category because of the capability for repeated orbital extension and retraction cycles. In addition, these roll-up arrays can be retracted into a relatively small package for removal and replacement in orbit. The mechanical interface with the station is relatively simple for either roll-up array approach. On the other hand, the folding panel array configurations are not capable of orbital retraction without major redesign of the linkages.

Minimum Degradation and Resupply

All candidates should rank approximately the same in this category, except for a small advantage associated with the ATM array with its honeycomb substrate. This heavy substrate will afford backside shielding against radiation damage as well as increased thermal mass to increase the low temperature during eclipse periods. This reduced operating temperature excursion may reduce thermal cycling damage to the solar cell interconnections during lunar orbit operation.

Maximum Ability to Achieve Primary Power Operation

This performance factor can be related to the inherent reliability of the array system. Because the reliability of the solar cells and associated interconnection should be the same for all candidates, the reliability of the deployment operation is the factor that must be assessed to determine the relative ability to achieve primary power operation. The GE roll-up array was ranked "best", because deployment is accomplished with one deployable boom in one actuator. The Hughes roll-up array requires four deployable booms in two actuators. The Boeing fold-out array is a multiple fold system with flip-out side panels. The ATM array was also ranked "best" because of the flight experience that will be logged prior to the finalized design of the OLS array.

Array Configuration Conclusion

Based on this qualitative comparison, it is concluded that the two roll-up array concepts have many distinct advantages over the fold-out panel approaches. It is more difficult to make a selection between the two

roll-up concepts because of the performance similarities between these approaches. However, the GE single deployable boom configuration has the following additional advantages over the Hughes two-boom concept:

1. It requires no synchronization of boom deployment/retraction.
2. It has a single hard point attachment to the supporting structure in the deployed condition.
3. It is potentially a lighter weight approach because of less structure required to deploy the same solar cell area.

Thermoelectric (Static) Power Conversion

Thermoelectric power systems produce electrical power by the direct conversion of heat (normally from nuclear heat sources) to electricity. The Seebeck (thermoelectric) effect offers advantages of static operation and high reliability potential. The net cycle efficiencies of present-day converters are in the 4- to 6-percent range with efficiencies up to 11 percent predicted for 1980-technology converters. The higher efficiencies are obtained through use of relatively high heat source coolant temperatures and large radiator areas (large temperature differences). Because of the relatively low cycle efficiency potential for such devices when compared with dynamic power conversion cycles such as the Brayton cycle, the thermoelectric converters are not considered as viable power conversion candidates for the OLS. The lower efficiencies and higher heat source temperatures required for increased cycle efficiency result in heavier system weights and/or lower system reliability (lifetimes) based upon previous NR EOSS studies.

Dynamic Power Conversion

The dynamic power conversion techniques used in conjunction with nuclear heat sources that have received most attention are the mercury-Rankine, Brayton, and organic Rankine. The mercury-Rankine is a low efficiency (5 to 8 percent) dynamic system with a resultant poor overall system productivity. Both the Brayton cycle and organic-Rankine cycle offer over twice the efficiency of the mercury-Rankine cycle. Thus, if a dynamic conversion system is to be selected with the obvious disadvantages of dynamic machinery, the selection should represent as high an efficiency as is feasible in order to maximize productivity. Comparisons show that both the Brayton and organic-Rankine cycles offer significant improvements in productivity over a mercury-Rankine. Because both nuclear reactor and radioisotope heat source concepts will accommodate all of the previously mentioned conversion systems, the higher efficiency cycles would be preferred. Thus, the mercury-Rankine cycle has been eliminated from further OLS consideration. The organic-Rankine cycle exhibits relatively high net cycle efficiencies of about 12 to 20 percent while operating at lower temperatures (heat source coolant temperatures of 800 to 900 F). The net effect of these operating conditions is that more energy (longer life) can be obtained from reactors operating at lower temperatures. However, radiator area requirements are increased for lower heat rejection temperatures. The Brayton cycle offers

more flexibility for a fixed radiator area (i.e., the Brayton cycle also has the potential for much higher heat source temperatures). High temperature reactors and radioisotope heat sources can be expected to evolve in the 1980 time period. The availability of higher-temperature heat sources and the greater state of development of the Brayton cycle over the organic-Rankine cycle makes the Brayton cycle conversion system the most favorable of the dynamic systems.

3.2.5 ENERGY STORAGE CANDIDATES

All candidate EPS's require storage of electrical energy to meet OLS electrical peak, eclipse and/or emergency power requirements. The trade tree for the energy storage concepts is shown in Figure 3.2-3 and discussed in the following paragraphs. Two candidates considered were secondary batteries and regenerative fuel cells (separate fuel and electrolysis cells).

Battery Candidate

The requirement of long reliable life for a high number of charge and discharge cycles (25 to 30 thousand) is the primary factor favoring a nickel-cadmium battery. Cell size of 100-ampere-hours is tentatively anticipated although a 300-ampere-hour size more adequately meets the requirement for solar array configurations. Silver-zinc batteries may be considered for peaking power requirements. However, silver-zinc batteries have a short wet stand life and a lower charge/discharge cycle capability at the depth of discharge required for OLS application. Silver-zinc battery capability is expected to be increased to at least 1-year operation with a few hundred cycles charge/discharge capability at 15 percent DOD. Although lighter than nickel-cadmium batteries (from 2 to 5 times lighter), the shorter life of silver-zinc batteries as compared with nickel-cadmium batteries ($\approx$ 2 to 3 years) may impose a greater logistics and maintenance problem. The silver-cadmium batteries are excluded from consideration for large scale application because of excessively high cost. Such batteries are limited to applications where minimum magnetic field interference with experiments is a requirement. Long battery life requirements are best met by operating at low temperatures ($<$ 32 F) and shallow depth of discharge ($\leq$ 15 percent). Thermal control of the battery employs a coldplate with three separate cooling loops with the excess heat rejected to the ECS radiator. Loss of any one cooling loop will in no way degrade battery performance whereas loss of any two loops will result in some performance degradation.

Cells are packaged on a replaceable three-cell module concept to facilitate in-orbit maintenance. Each cell has provisions for bypassing a failed cell, sensing voltage, temperature, and pressure, and a load resistor for reconditioning the cell. The parameters previously mentioned are required for computer monitoring and control of the individual cells. Cells are arranged to minimize the effects of magnetic fields.

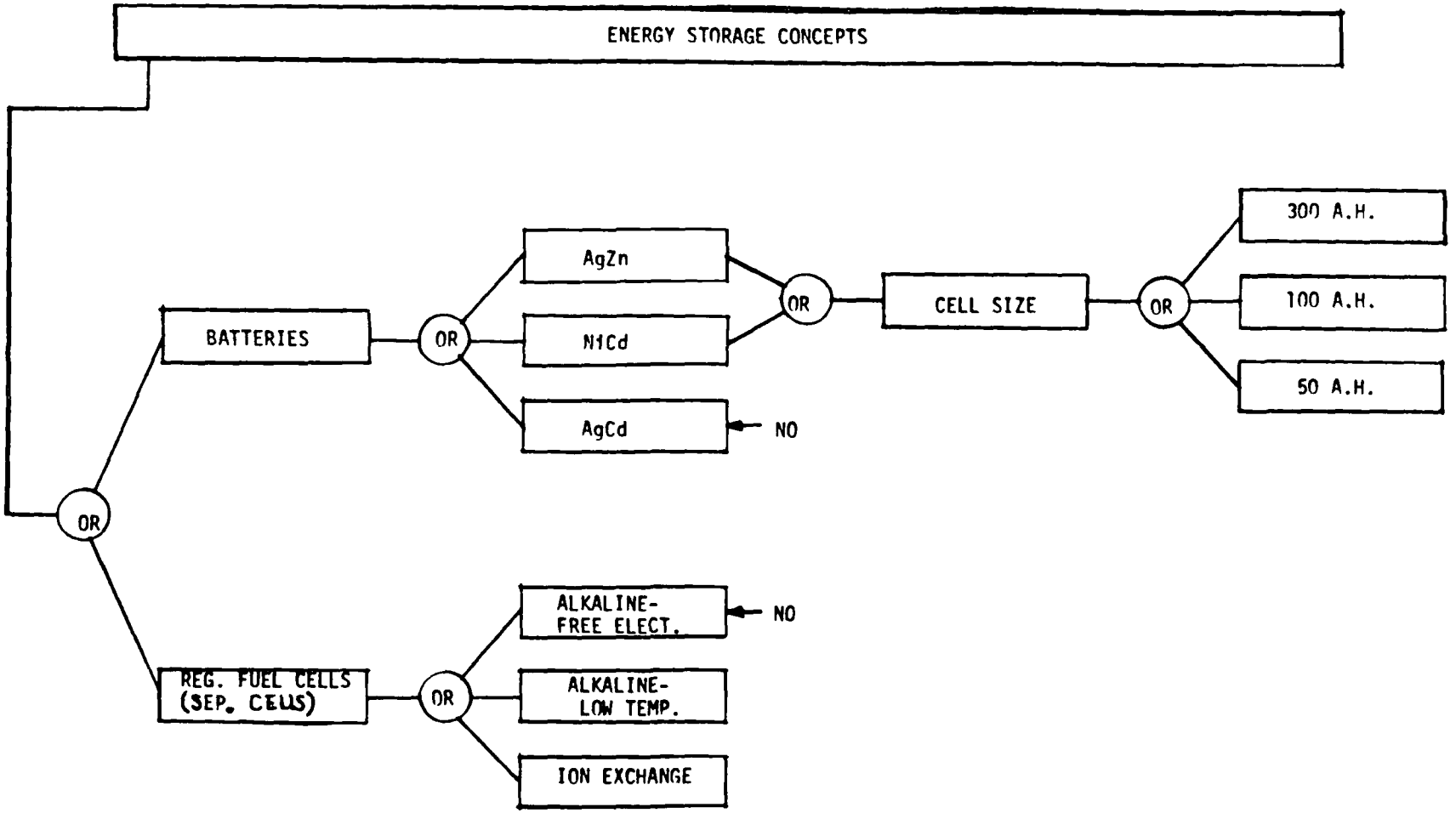


Figure 3.2-3. Energy Storage Concept

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Battery control is based on the per-cell concept, which lends itself to programmed computer control of individual cell performance. Computer control monitors cell status with appropriate readouts. Overpressure is read by the computer as though it were hard-wired to a caution and warning system. The system will detect cell failure and take correction action. It will also control the cell charge technique of the battery charger.

Both battery and charger concepts represent major development items, because no off-the-shelf hardware exists at the present time. The interface with the ISS computer also is a complex and relatively expensive development program but necessary to obtain long battery life. Even with batteries operating at ideal temperatures and using per-cell charging techniques, it is anticipated that the batteries will need to be resupplied 3 to 4 times over a 10-year period.

Weight of the batteries as a function of mission duration is shown in Figure 3.2-4. Battery weights are based on the following:

Orbit	Battery
60-n mi circular (polar) Duration = 1.987 hours Light (worse-case)* = 1.212 hours Dark (worse-case)* = 0.775 hour	Type = Nickel Cadmium DOD $\leq$ 15 percent Power density = 16 wh/lb Resupply = 4 times/10 years

*Does not include eclipses of the sun by the earth.

Regenerative Fuel Cell Candidate

Fuel cells are predicated on hydrogen and oxygen combining in the presence of a suitable catalyst, resulting in a reaction that produces heat, water, and electricity. The elements and materials that provide the environment to sustain the reaction comprise a fuel cell. Under controlled conditions, the reaction can be sustained to provide a constant potential of approximately 0.8 to 1.2 volts per cell, depending upon cell design. Usable electrical power is obtained by electrically connecting multiple cells into a fuel cell assembly (FCA). The FCA produces electrical energy on demand, operating only to the capacity required to satisfy the load. The most promising of these are the ion-exchange and low-temperature alkaline types. The free-electron alkaline types are eliminated because of greater degradation with operating life and requirements for higher temperature (heat) rejection.

Fuel cells operating in the reverse mode, that is electricity + water resulting in heat, hydrogen, and oxygen, are called electrolysis cells. Fuel and electrolysis cell assemblies operating in a closed-loop back-to-back mode can alternatingly produce water and reactants indefinitely with very little loss in reactants. As a result, such a regenerative fuel-cell concept may be employed as an energy-storage concept similar to batteries.

Fuel cell technology has advanced to the stage where predicted life compares favorably with predictions for nickel-cadmium batteries and at approximately 15 percent of the weight required for batteries. Development costs for the fuel cell and the electrolysis cells may not be excessive (considering the OLS 1983 IOC date) as these components may be required at earlier dates for other programs. It is anticipated that the regenerative fuel-cell concept can be obtained at near recurring cost value if the technology is developed for these on going space programs.

Weight of the RFC concept as a function of mission duration also is shown in Figure 3.2-4. RFC weights are based on the use of separate cells and a resupply equal to three times in 10 years.

3.2.6 MAJOR TRADES

The trade trees showing candidate concepts considered in this study are depicted in logic figures. Various major tradeoffs leading to the selections are described in the following paragraphs of the study.

Primary Power Sources

A trade tree of the candidates for the primary EPS is shown in Figure 3.2-5. An analysis of the trade is included in the following paragraphs.

Solar Photovoltaic (Battery/RFC)

Preliminary tradeoff analysis and feasibility studies related to the integration of a solar array primary power source for the manned-unmanned applications have been performed in previous NR studies. Various types of solar-array constructions were investigated to assess the tradeoff factors necessary in the selection of a preferred approach (see paragraph 3.2-4). Based on these considerations, a roll-up array concept was selected as the best approach for the OLS.

A review of solar cells for space applications reduced the number of contenders to single crystal silicon and cadmium sulfide thin film cells. At present the silicon cells are clearly favored. However, despite being about 50 percent heavier, having a lower conversion efficiency (around 7 percent compared to 11 percent for silicon cell), and requiring more development, the cadmium sulfide cell's potential for overall cost reduction (approximately 50 percent of silicon cell systems) makes them attractive provided the orbiting payload costs can be reduced sufficiently. For this study, silicon cells were assumed.

Two array locations considered were body-mounted and remote deployment (boom). Docking constraints and shadowing effects clearly favors a remote deployment concept.

N on P cells degrade considerably less from radiation than P on N cells; thus, they are preferred over P on N cells.

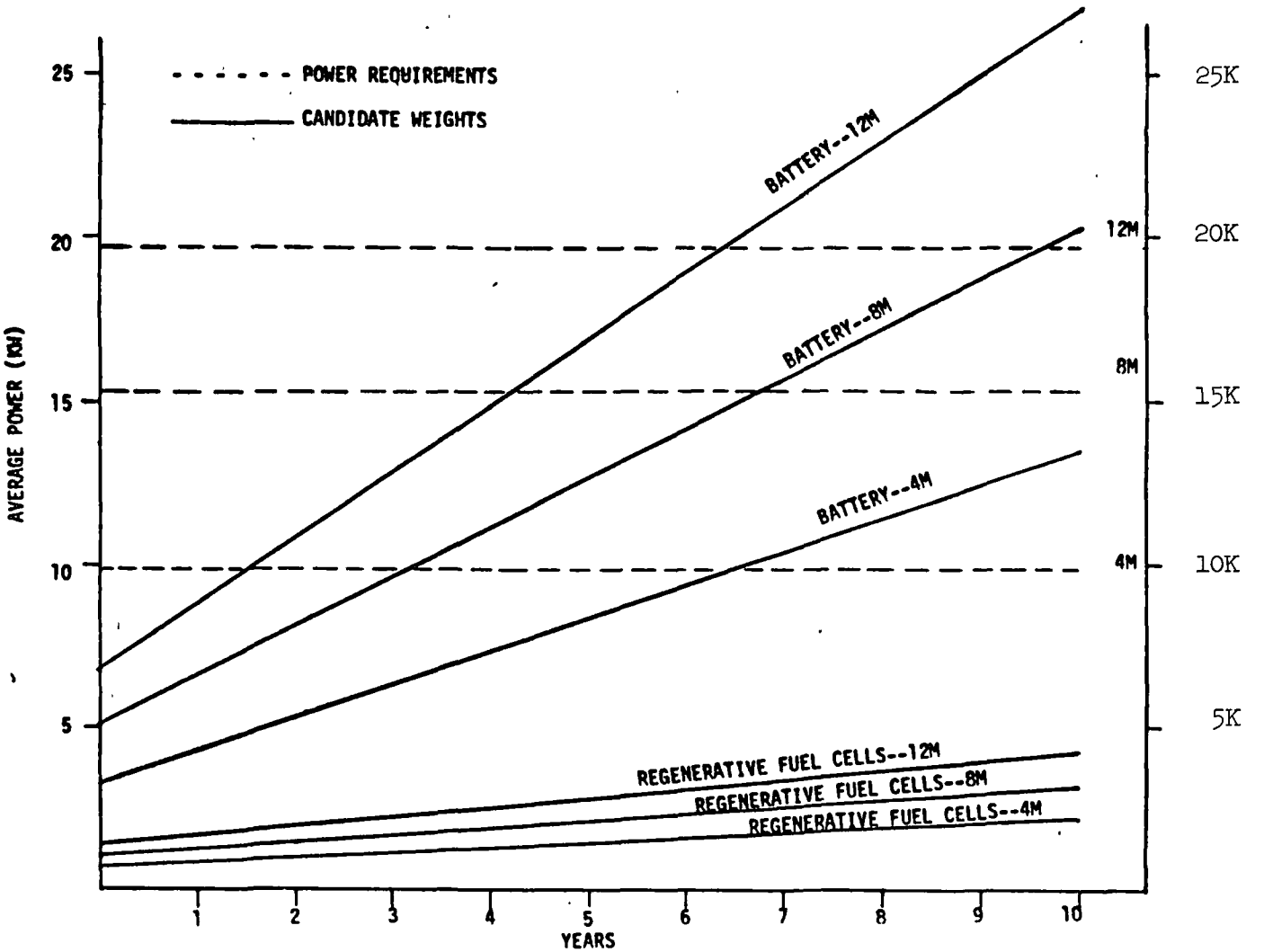


Figure 3.2-4. Energy Storage Candidate Comparison

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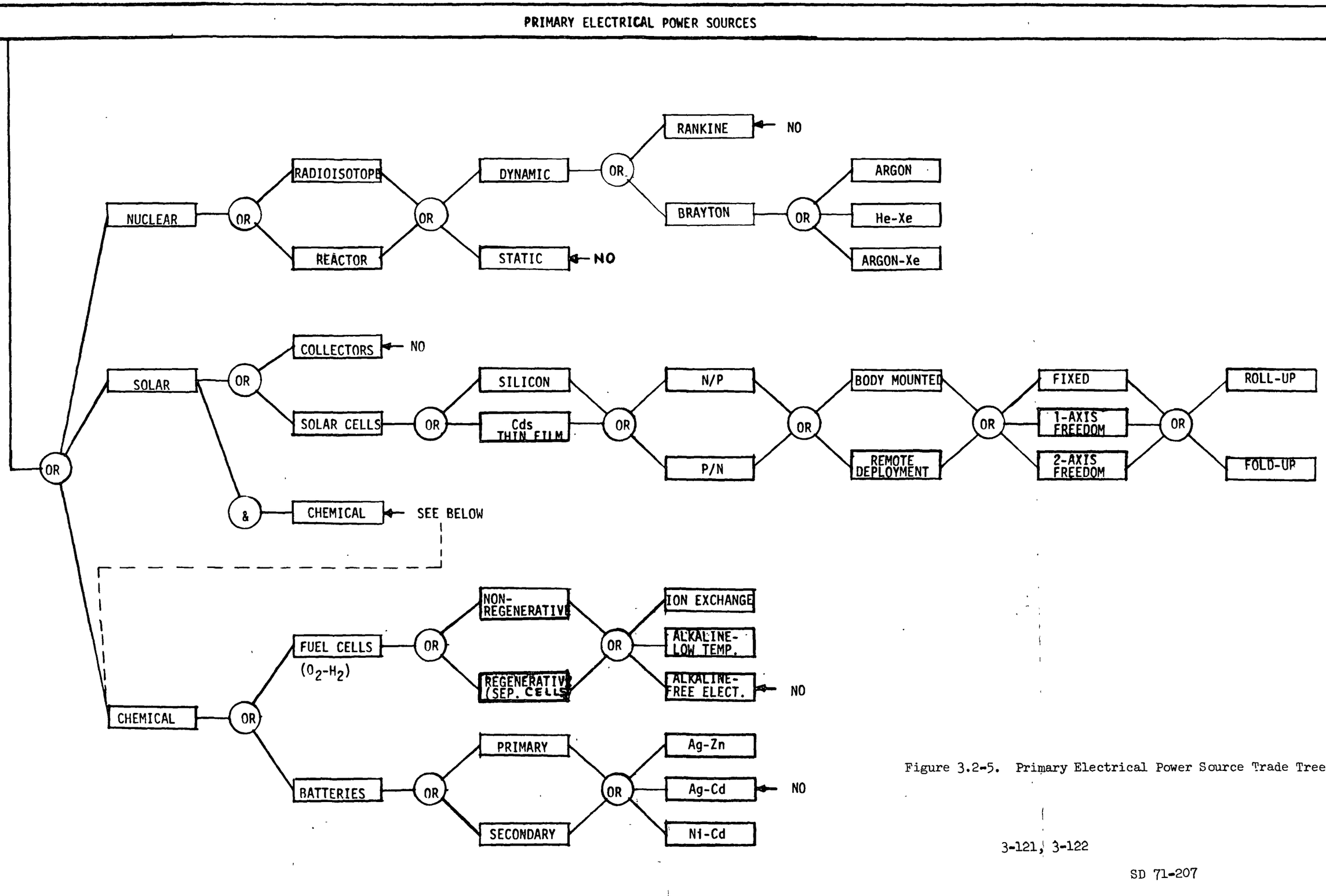


Figure 3.2-5. Primary Electrical Power Source Trade Tree

Fixed solar arrays are not acceptable from EPS considerations (loss of power resulting from misorientations, etc.). Single-degree-of-freedom solar arrays are acceptable from EPS considerations but may impose severe penalties on OLS operations. Two-degree-of-freedom solar arrays minimizes the penalties on OLS operations.

Solar array area requirements up to 14,650 square feet were considered. Figures 3.2-6 and 3.2-7 show area requirements for various missions as a function of crew size. The basic difference in Figures 3.2-6 and 3.2-7 is the degree of solar array panel degradation attributed to solar flare protons. Data on solar flare proton degradation in the lunar environment are quite limited at present. Severe radiation storms may occur about 15 to 20 times over a 10-year period, and each storm lasts for approximately 1 hour. Figure 3.2-7 represents a very conservative estimate of the solar array requirements whereas Figure 3.2-6 assumes the array will be rolled up during the severe storms. It has been concluded that a 10,000 ft^2 panel is adequate for an 8-man mission, hence a solar array of this area was divided into four identical panels and investigated in detail.

Solar array panel characteristics are shown in Figure 3.2-8. Beginning of life (BOL) power was used in this figure.

Figure 3.2-5 shows an option for supplying the eclipse power (batteries or fuel cells)necessary for any solar power subsystem. If the fuel-cell option is selected, solar array requirements can be further reduced by using tug boil-off and also supply the crew water requirements as a byproduct (Figure 3.2-9). For crews of 8 men or less, boil-off can supply all of the water requirements. For larger crews, an additional source of reactants (RCS) will be required. It should be noted that for an 8-man mission, the solar array power requirements can be reduced by almost 10 percent. The use of batteries for eclipse power does not have this advantage.

A computer program was written to perform a transient temperature and power analysis of the deployed array to show time - temperature/power history during the worst-case lunar orbit as a function of altitude. Temperature-time and power-time history of the deployed solar array are shown in Figures 3.2-10 and 3.2-11 respectively. Major characteristics of these curves are summarized as follows:

Orbit Altitude	Power (W/Ft^2)			Temperature o(C)		
	Max.	Min.	Avg.	Max.	Min.	Avg.
20 n mi	22.787	6.680	9.585	114	-131	70
60 n mi	22.558	6.940	9.801	110	-128	67
100 n mi	22.433	7.175	9.914	107	-126	65

The nominal beginning of life performance is degraded by a series of time-dependent factors to account for lunar orbit environment influences. These factors are summarized in Table 3.2-5 and are discussed in the following paragraphs.

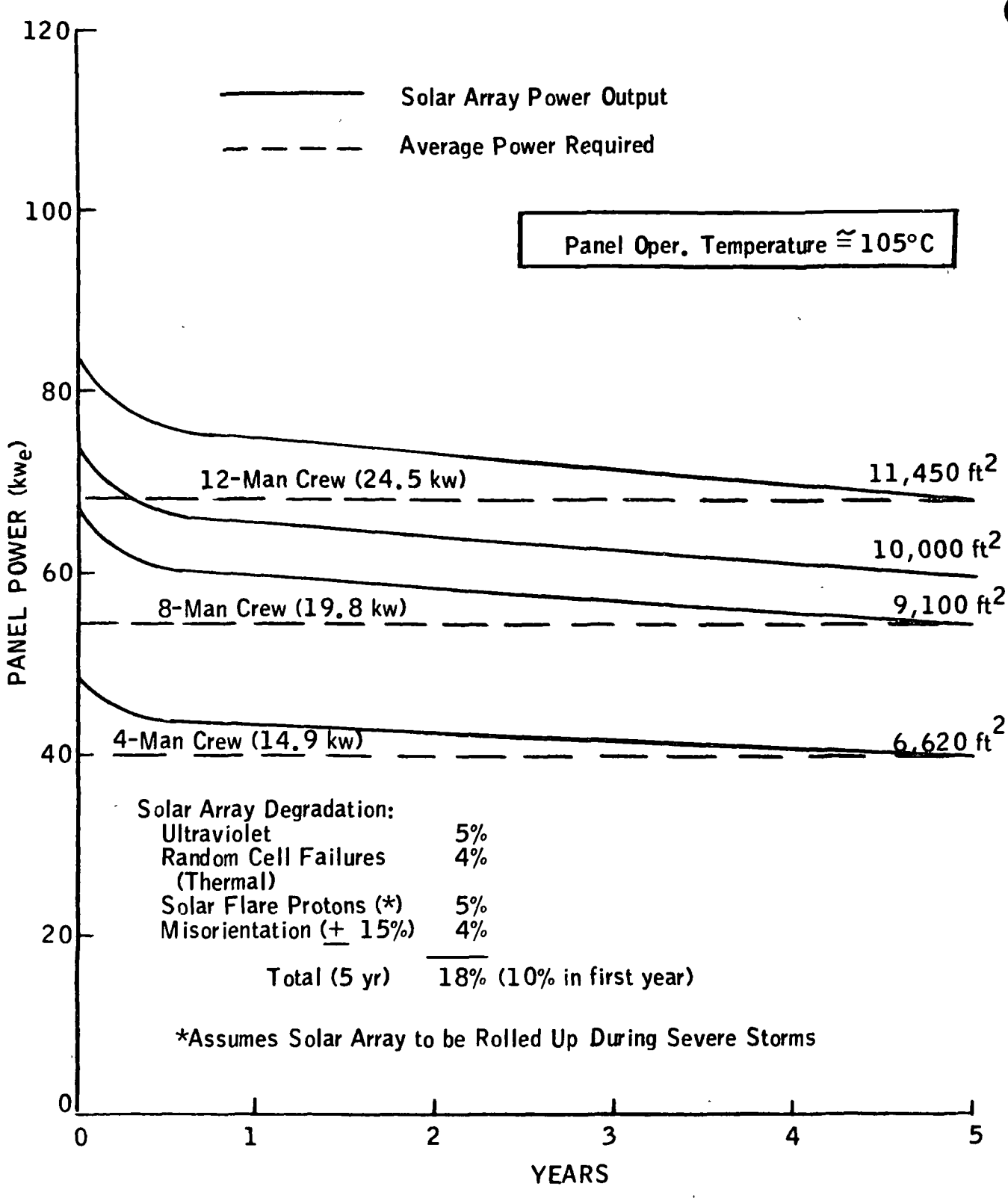


Figure 3.2-6. OLS Solar Array Power Characteristics

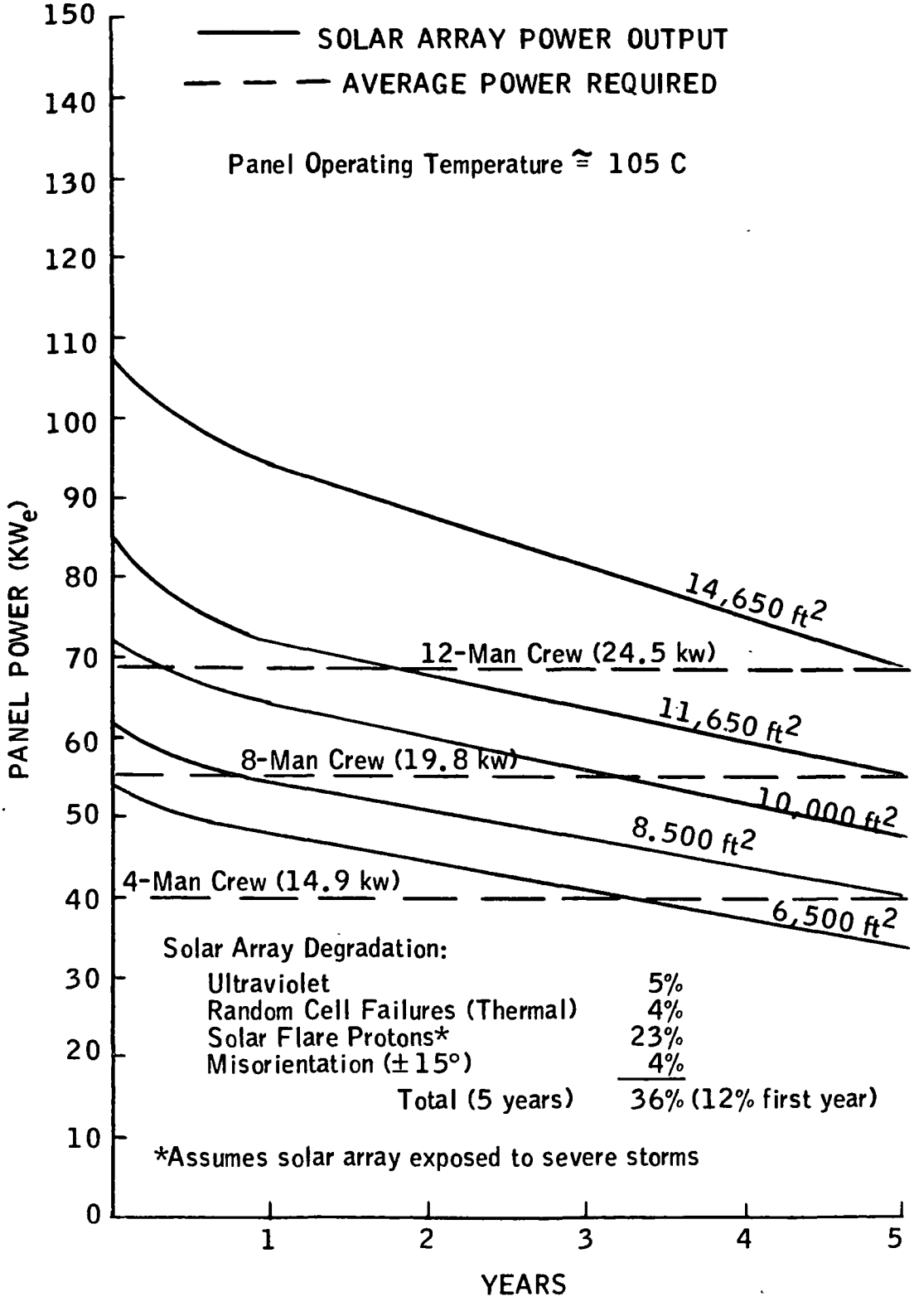


Figure 3.2-7. OLS Solar Array Power Characteristics

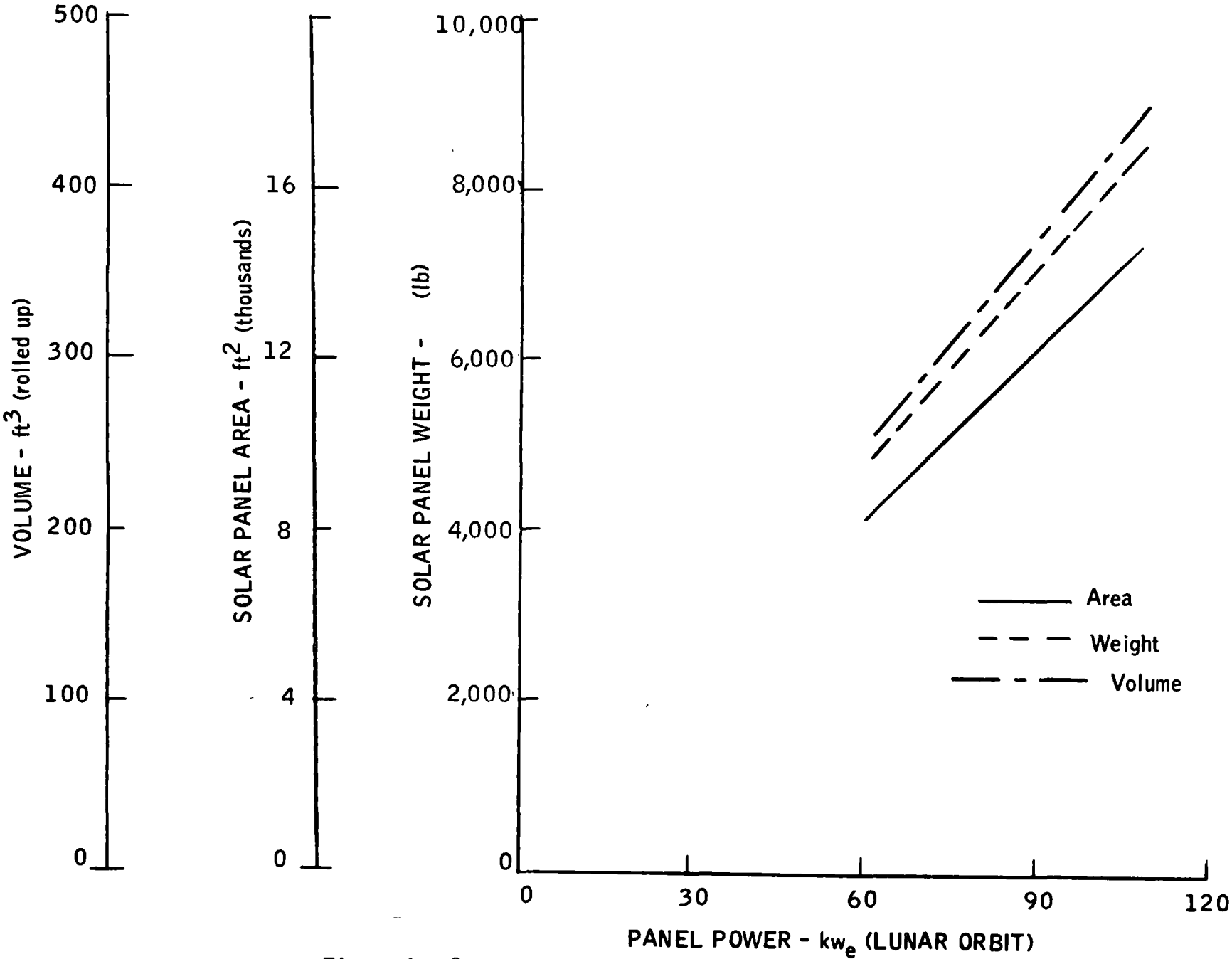


Figure 3.2-8. Solar Array Panel Characteristics

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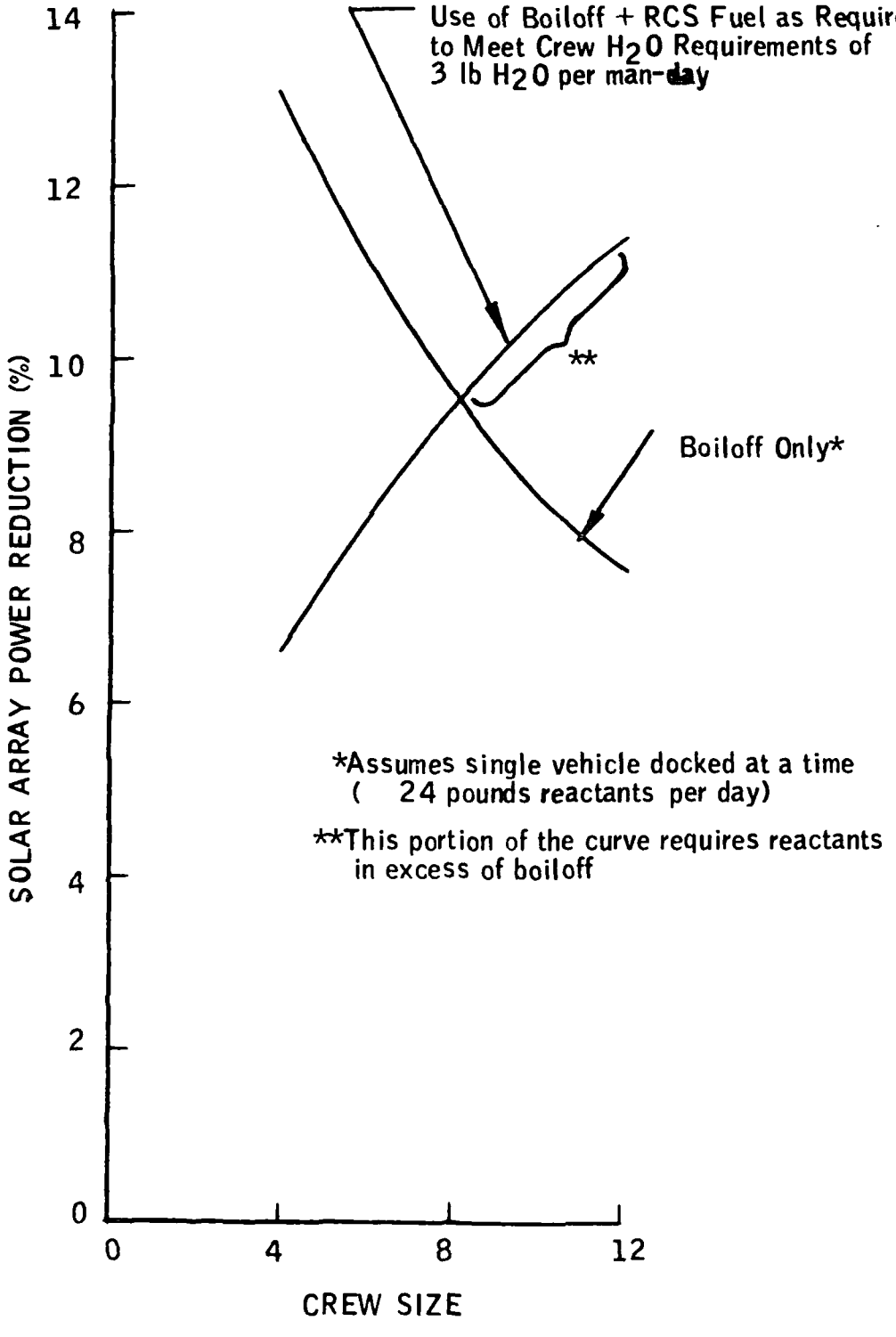
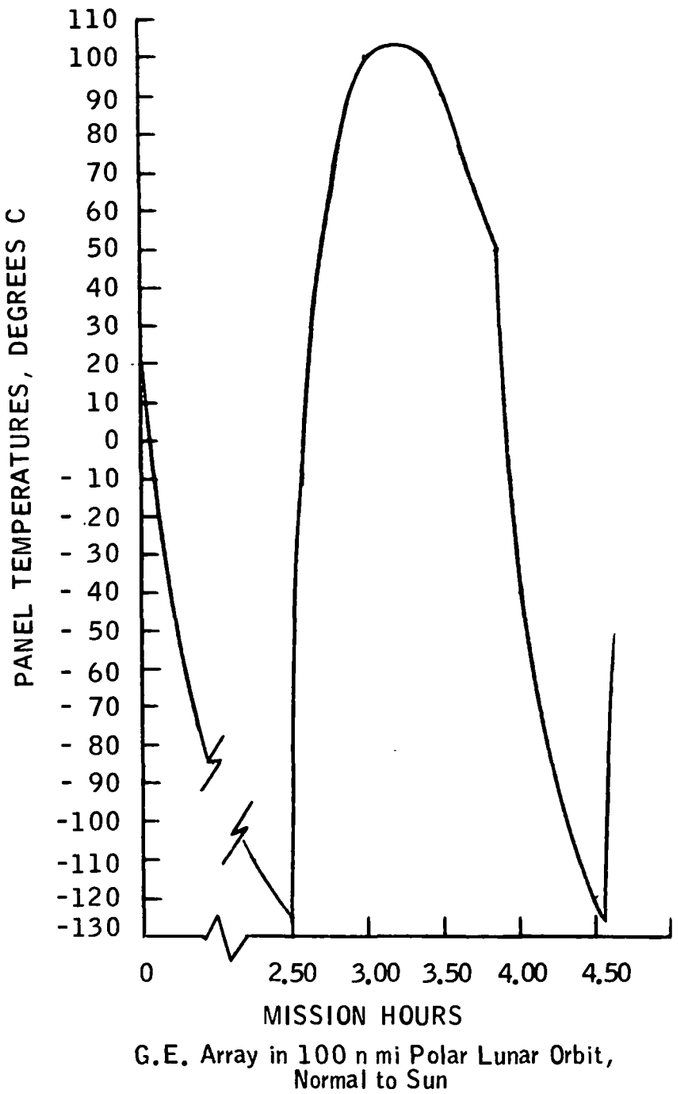
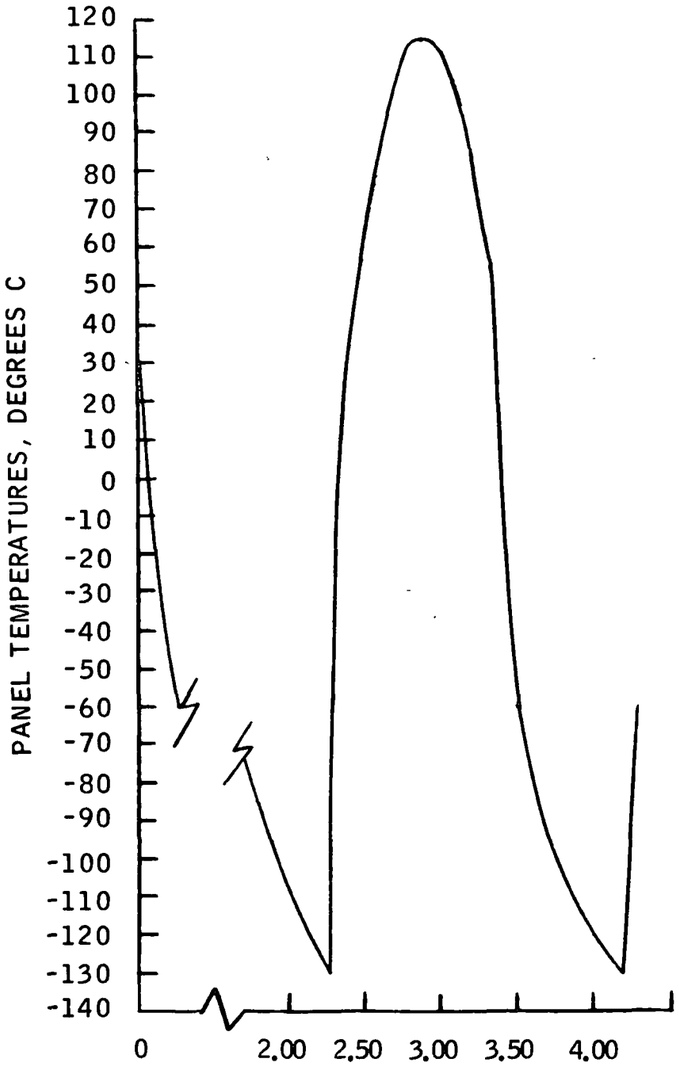


Figure 3.2-9. Effect of Tug Boil-off on Solar Array Requirements

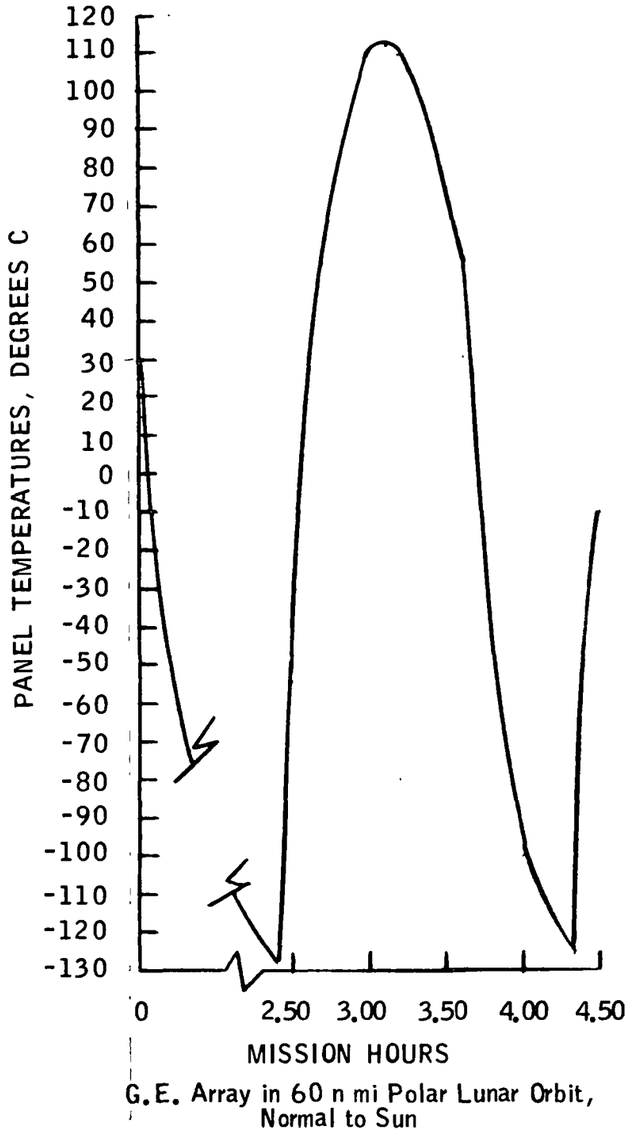
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G.E. Array in 100 n mi Polar Lunar Orbit, Normal to Sun



G.E. Array in 20 n mi Polar Lunar Orbit, Normal to Sun



G.E. Array in 60 n mi Polar Lunar Orbit, Normal to Sun

Figure 3.2-10. Temperature-Time History for Solar Array in Lunar Orbit

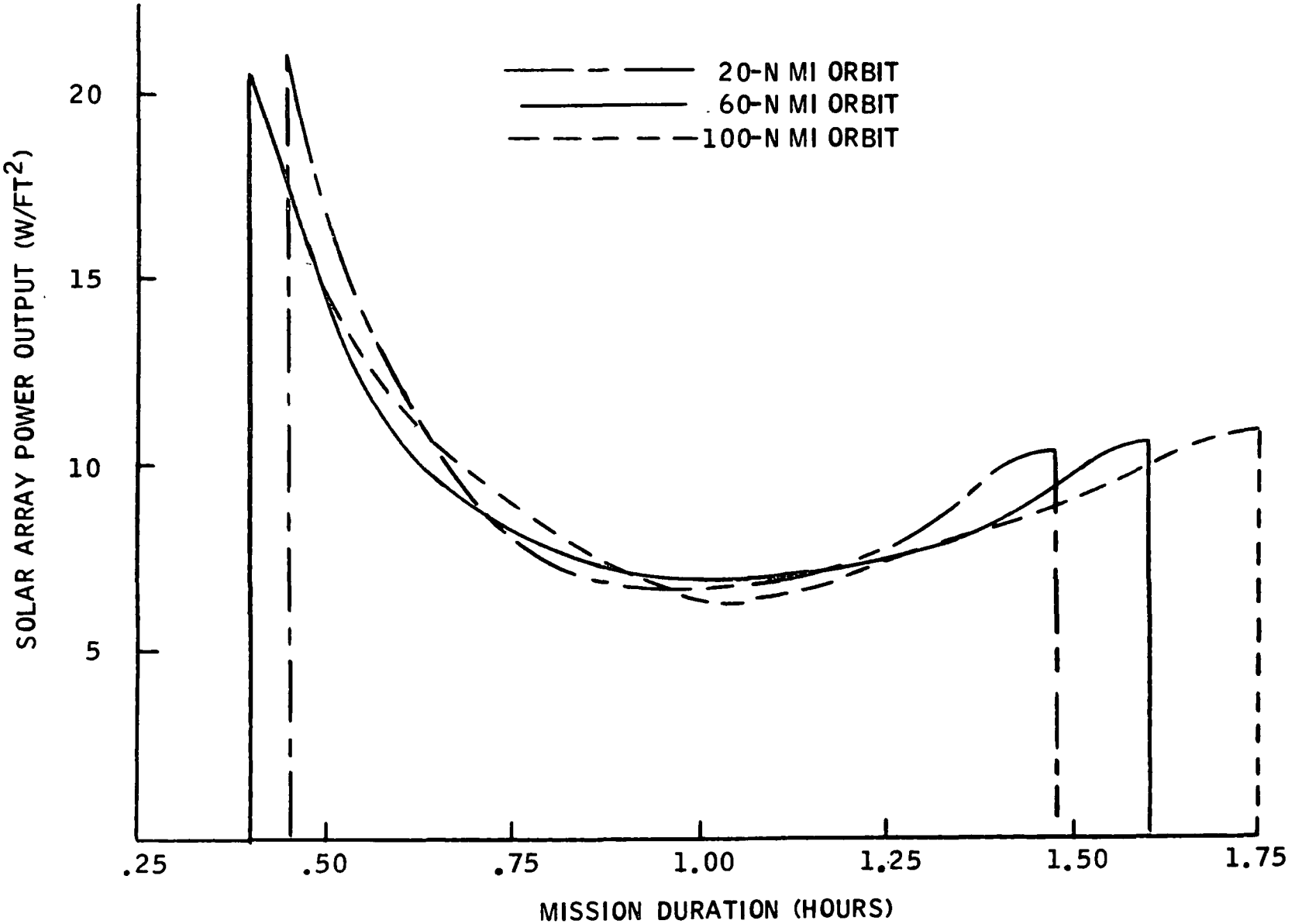


Figure 3.2-11. Power - Time History for Solar Array in Lunar Orbit

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Table 3.2-5. Array Environmental Loss Factors

Degradation Source	Power Loss Factor			$P_{max}/(P_{maxo})$		
	B.O.L	1 Year	2 Years	3 Years	4 Years	5 Years
1. Ultraviolet	1.000	0.95	0.95	0.95	0.95	0.95
2. Random cell failures	1.000	1.00	0.99	0.98	0.97	0.96
3. Solar-flare protons	1.000	0.99	0.98	0.97	0.96	0.95
4. Misorientation (15 degrees)	0.96	0.96	0.96	0.96	0.96	0.96
Environmental loss Factor	0.96	0.89	0.89	0.86	0.84	0.82

Ultraviolet

Extensive subcontractor tests have been made on composites of filters, fused silica, organic adhesives, and silicon solar cells to determine the effect of ultraviolet radiation. During this testing, it was found that there was an initial decrease in the transmittance of the filter because of UV radiation; the loss occurred in the first 20 hours of exposure, either in vacuum or in air. The loss amounts to about 5.5 percent of the short-circuit of a cell. Continued exposure up to a simulated exposure of 4.3 years showed no further increase.

Random Cell Failures

This degradation occurs primarily because of the effects of thermal cycling on cells and cell interconnections. The exact amount to allow for this loss depends upon the confidence in panel life cycle testing and qualification testing programs. Until experience has been gained with this type of array construction (lightweight, flexible substrate), an allowance of 5 percent over five years might not be unreasonable for random cell and thermal cycling failure.

Solar Flare Protons

The solar flare proton damage to the solar array is estimated to be relatively small for the nominal mission provided the panel is rolled up for the very severe storms. An allowance of 5 percent over five years may not be too unreasonable.

Misorientation

As the angle of incidence of sunlight on a panel of solar cells increases, the power output will decrease. The greatest effect is the decrease of the effective light intensity as a function of the cosine of the incidence angle. Also, reflectance from the front surface of the cell coverglass will vary, increasing from about 4 percent between 0 and 45 degrees to 100 percent at 90 degrees (grazing incidence). Figure 3.2-12 shows the short-circuit current versus the incidence angle curve which accounts for the cosine and reflectance effects. Solar cell power output is not simply directly proportional to short-circuit current; the power output decreases more rapidly. The relation between short-circuit current and power at constant voltage was determined for solar cells of typical I-V curve shape by linear interpolation between the dark curve and the I-sun curve. This resulting power output versus incidence angle relationship is also shown in Figure 3.2-12.

For these sizing calculations, a total misorientation angle of 15 degrees has been used. This deviation from normal incidence is from out-of-plane thermal bending of the roll-up array deployable boom and to possible error in the array orientation drive system.

At first glance, the effect of energy storage requirements on solar array size shows batteries as slightly favored over regenerative fuel cells (RFC) based on an overall battery recharging efficiency of about 58 percent and approximately 54 percent for electrolysis. However, some of the adverse efficiency for RFC's could be offset by reduction in transmission and power conditioning losses by means of proper system design and optimum location of equipment. It has been concluded that energy storage requirements (battery versus RFC) will influence the solar array size in approximately the same magnitude.

The light weight roll-up array concept was selected as the best approach to meet the solar array requirements for the OLS. This selection was based on array system weight, stowed volume, and the inherent capability for repeated orbital retractions and extensions. It appears feasible to perform the design, development, fabrication, and qualification of such an array system over a 6-year period. Thus, this development effort should start during calendar year 1977 to meet the specified OLS launch date of 1983.

Although large in physical size, the proposed roll-up array concept requires no technological breakthroughs. The development of the large-diameter deployable boom is an extension of present technology, but no insurmountable development problems are foreseen.

Nuclear

Two nuclear energy sources (reactor and radioisotope) are considered prime candidates for the primary EPS. Both static and dynamic conversion concepts were evaluated with static conversion eliminated because of the inherent low efficiencies, high specific mass and the high technical risk

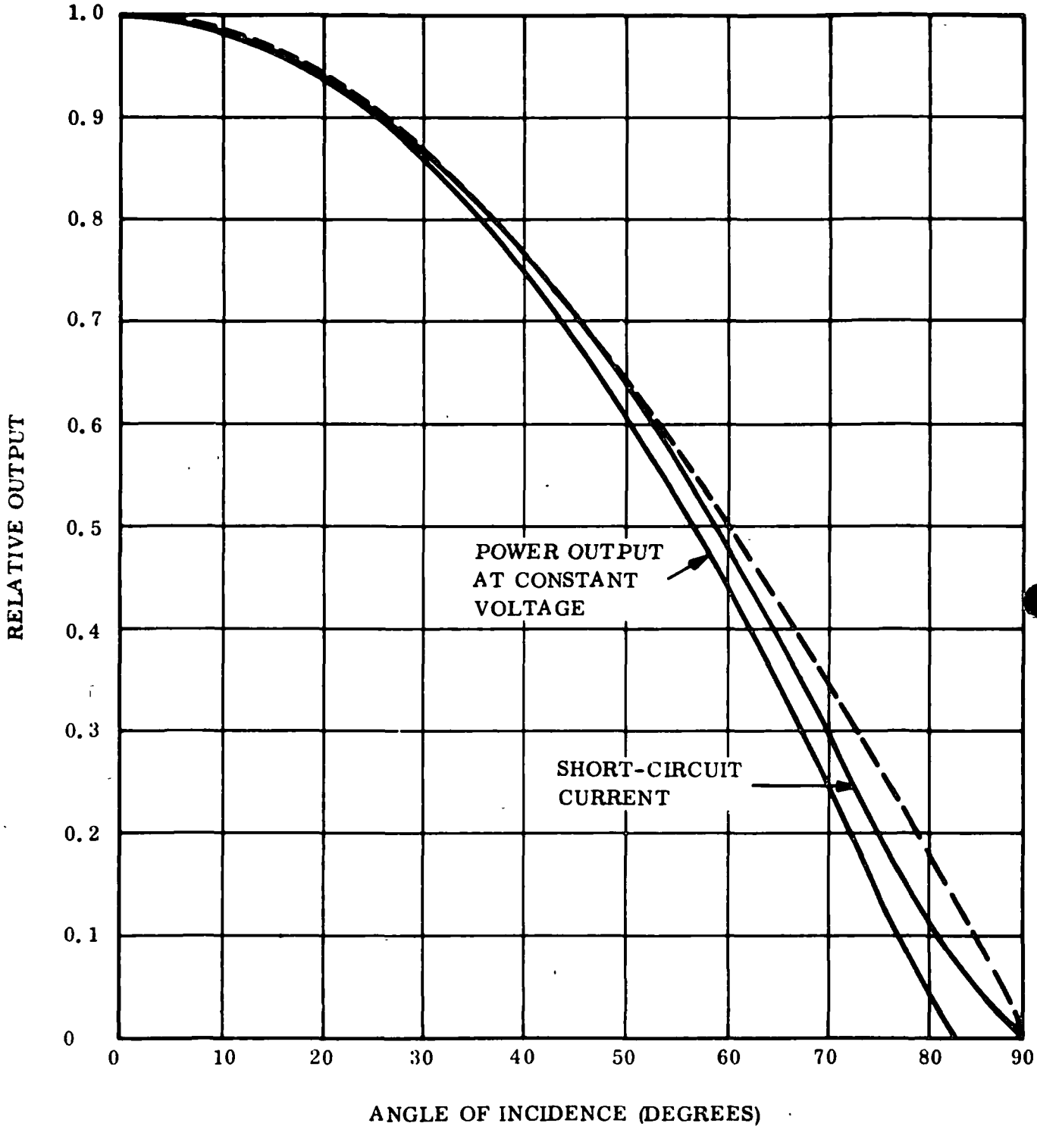


Figure 3.2-12. Dependence of Solar Cell Output on Angle of Incidence

(availability) of advanced conversion devices associated with high temperature nuclear heat sources (i.e., thermionics and MHD). A number of dynamic conversion concepts also were considered. Rankine-cycle was deleted on the grounds of relatively low efficiency resulting in the closed Brayton-cycle concept being the preferred candidate for nuclear power sources. (See paragraph 3.2.4.) Advantages of the Brayton-cycle systems are high system efficiency (smaller heat source and less shielding), simplicity (improves reliability), a nontoxic working fluid (enhances safety), freedom from materials erosion and corrosion by the working fluid (reduces maintenance), and potentially long life with the use of gas bearings. In addition, Brayton-cycle systems are competitive with other candidates for use in the late 1970's and early 1980's time period. The extreme high costs of nuclear heat sources (radioisotope or reactor) make the use of Brayton-cycle systems particularly attractive.

Cycle working fluids for space power applications have been limited to the monatomic gases because of their chemical inertness and the slightly higher performance achievable in turbines and compressors with these gases. A wide range of monatomic gases (i.e., helium, argon, xenon, and mixtures thereof) of varying molecular weight have been considered. The choice of fluid results from a tradeoff of the higher heat transfer rates (thermal conductivities) attainable with low molecular weight gases versus the higher cycle efficiency attainable with the higher molecular weight gases. Figures 3.2-13, 3.2-14, and 3.2-15 show the effect of molecular weight on component efficiency for the candidate monatomic gases and gas mixtures. Figure 3.2-16 shows the increase in heat transfer area required as the molecular weight is increased. Figure 3.2-16 shows a great heat transfer advantage for helium-xenon moistures at any particular molecular weight, while Figure 3.2-15 indicates only a small efficiency disadvantage. For this reason, a helium-xenon mixture was assumed in this study.

Radioisotope-Brayton

The radioisotope Brayton cycle power subsystem data used in this study was developed during NR's EOSS study. Two basic methods for integration of the power source with the spacecraft are:

1. Separately launched power module configuration (SLPM) - This concept involves a self-contained power module that is launched separately from the spacecraft to which the power is to be delivered and assembled in earth orbit prior to injection toward lunar orbit. The module carries its own propulsion, attitude control, and automatic docking subsystems in addition to a controlled intact reentry package, startup system, telemetry, and batteries for auxiliary electrical power. It can be launched either before or subsequent to astronaut occupation of the OLS. Following its docking, hookup and startup, it will deliver specified amounts of power for operating periods measured in years. Following completion of the mission, the power module can be parked in a high orbit for reuse at a later time, sent on a sun termination mission, or the isotope heat source in its reentry package can be separated from the remainder

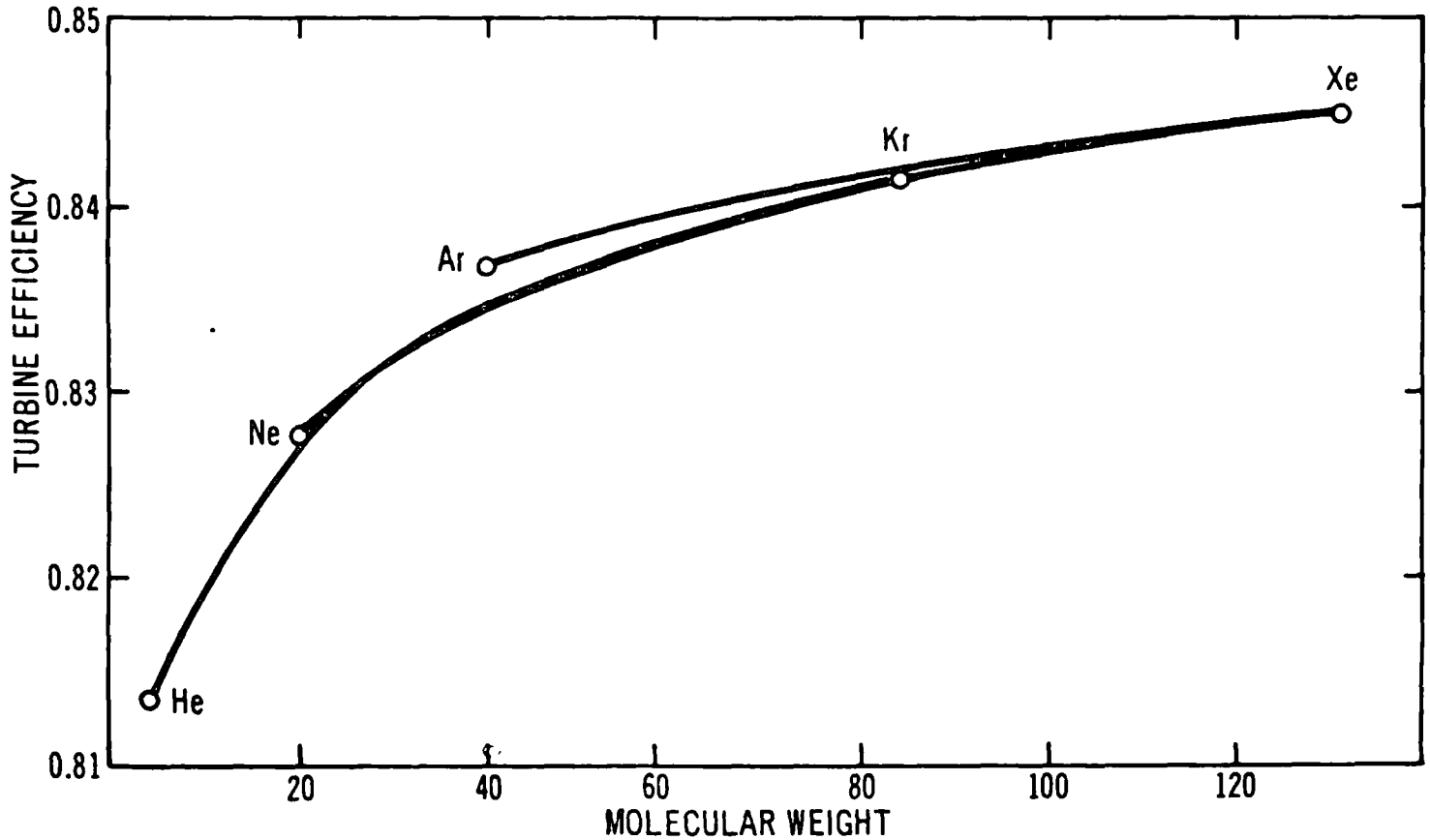


Figure 3.2-13. Turbine Efficiency Versus Molecular Weight

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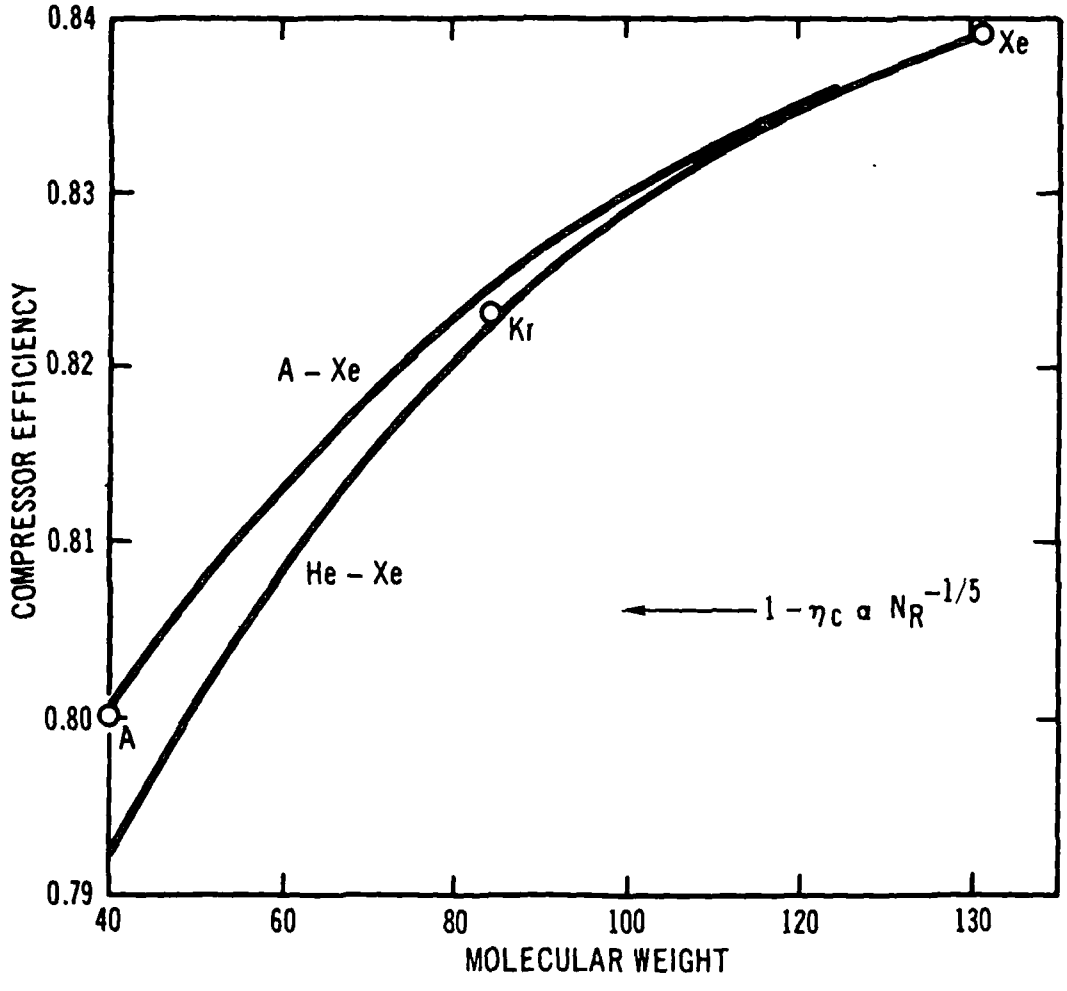


Figure 3.2-14. Compressor Efficiency Versus Molecular Weight

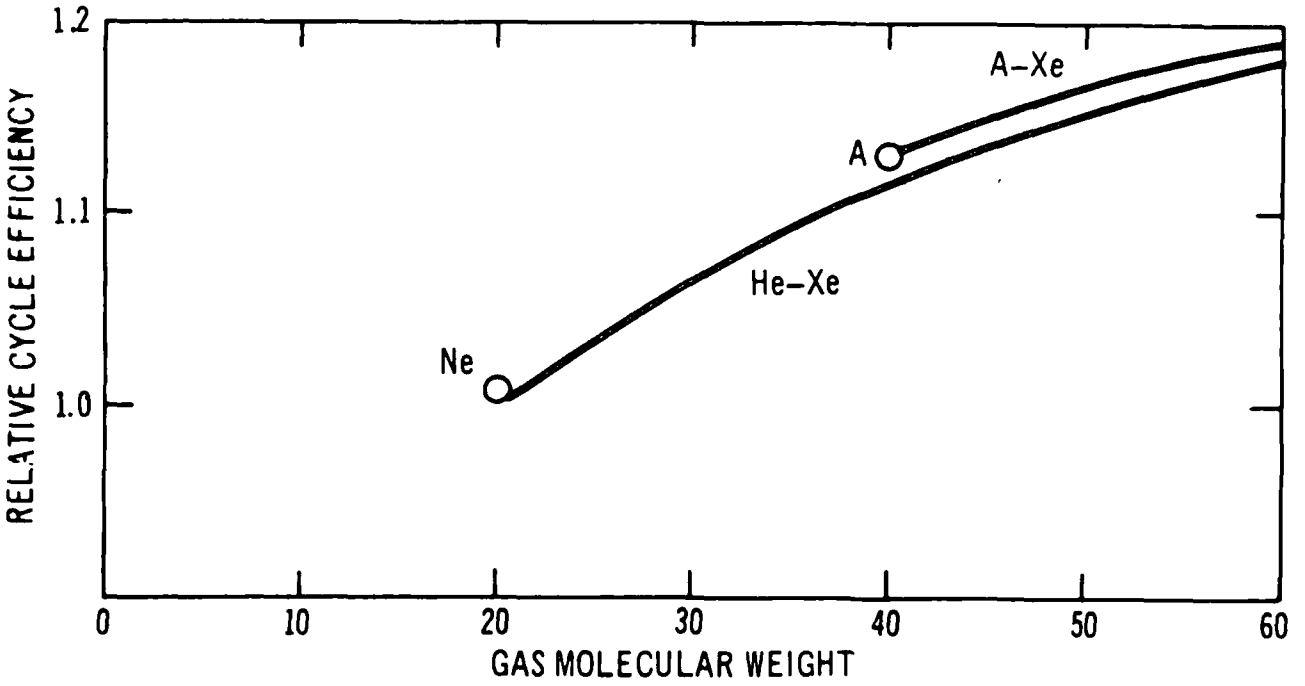


Figure 3.2-15. Influence of Fluid and Molecular Weight on Cycle

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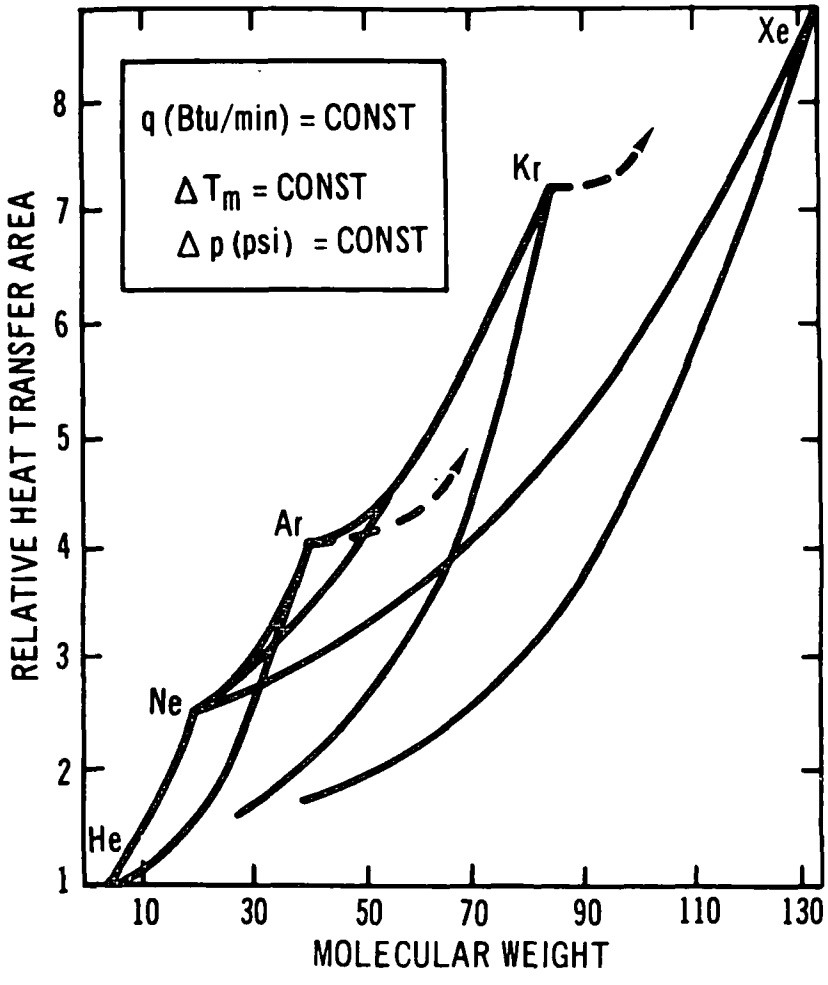


Figure 3.2-16. Relative Heat Transfer Areas for Inert Gas Mixtures

of the module to be reentered and recovered through the controlled intact reentry package and the recovery system. Each SLPM can be configured to deliver from 3 to 15 kw_e and is 9 feet in diameter and about 60 feet long.

2. Integrally launched power module configuration (ILPM) - In this concept, the radioisotope Brayton assemblies are configured so that they may integrate into the basic spacecraft or into structures attached to the core module. The ILPM concept uses a spacecraft engine room for the Brayton cycle power generating units (PGU's). The engine room can be pressurized for equipment maintenance, thus eliminating the need for EVA as in the case of the SLPM.

The primary differences in the two concepts are a result of operational safety considerations. The SLPM appears to be more complex and thus more costly than this ILPM, but ancillary provisions are incorporated for launch safety and recovery disposal operations. The ILPM relys upon the launch vehicle, shuttle, and tug for operational safety. If a similar safety philosophy is acceptable for the SLPM, the delta weight and system complexity associated with the SLPM is reduced significantly. The two major advantages to the SLPM are (1) shielding weights are not a primary constraint on the total OLS configuration and (2) replacement, recovery, or disposal of the radioisotope can be accomplished in modular form.

To illustrate some of the design considerations, a typical radioisotope system including a Brayton power conversion system is shown in Figure 3.2-17.

The system consists of a closed working gas loop and associated subassemblies. The radioisotope heat source assembly adds heat by thermal radiation to an inert working gas (helium-xenon mixture with a molecular weight equal to 83.8) in its heat exchanger. The hot gas (T = 1600 F) flows to a single-stage radial turbine. Expansion of gas through the turbine spins the machinery shaft and produces useful work. Part of the work is absorbed by driving a single-stage, radial compressor which circulates the loop working gas. The remaining shaft work is available to the alternator. The rotating shaft is supported on journal and thrust bearings that are lubricated by the working gas itself (the gas loop can therefore be hermetically sealed).

After expansion through the turbine, the gas (T = 1230 F) flows through a recuperator. The heat exchanger transfers most (about 70 percent) of the energy that is potential waste heat back to the low-temperature gas (T = 278 F) flow leaving the compressor. A gas-to-liquid heat exchanger removes the cycle waste heat. Any portion of this rejected heat is available for exchange to another spacecraft system. The coolant gas is then compressed and ducted back through the recuperator, where it is preheated and returned to the heat source heat exchanger.

The electrical assembly regulates and distributes the output power and controls the power subsystem. Output frequency is sensed and regulated, maintaining the shaft at a constant speed. Alternator output can be designed for the desired voltage and frequency values. A gas management assembly is used for startup, shutdown, and to adjust the working gas inventory.

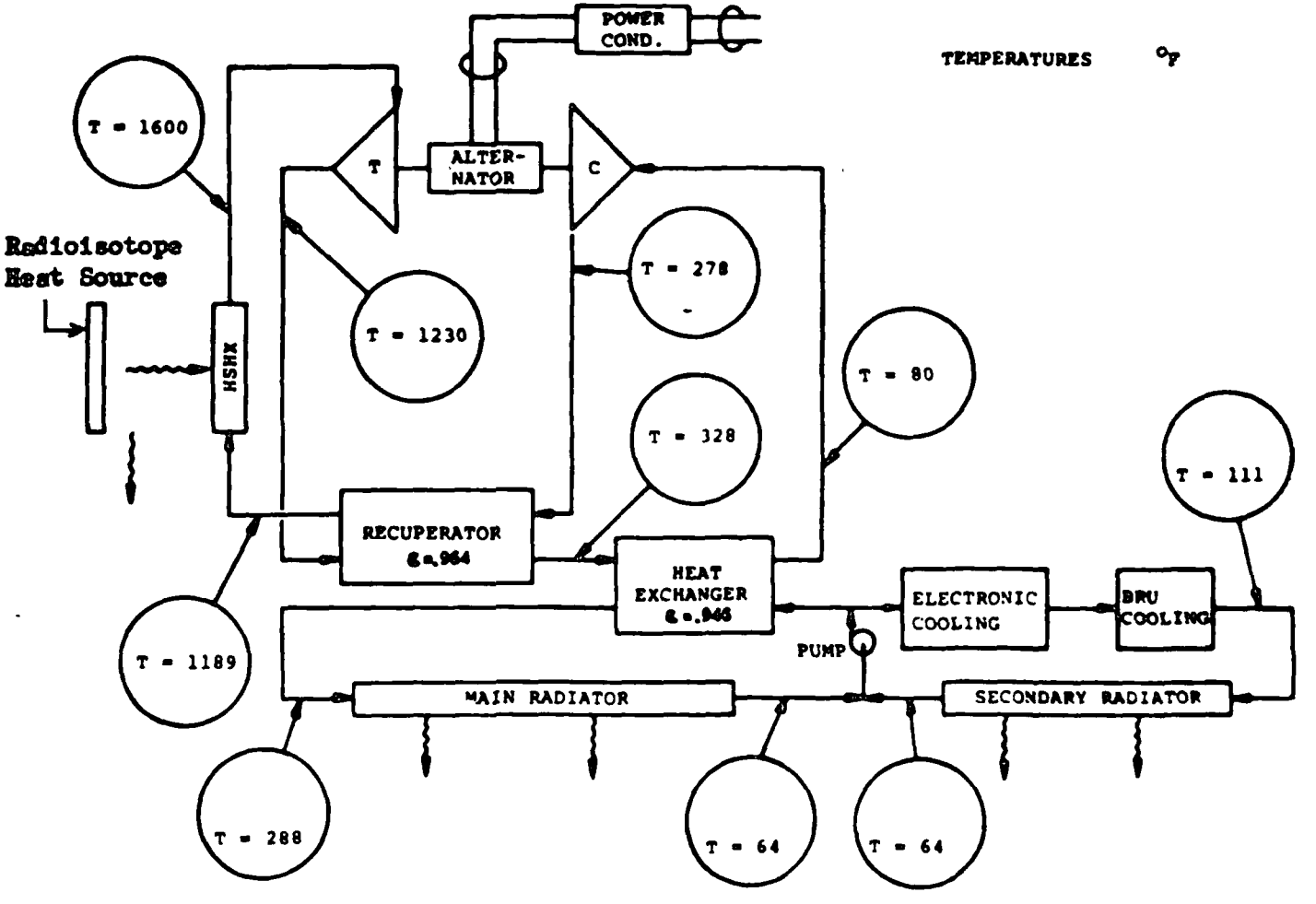


Figure 3.2-17. Brayton Power System Schematic

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The heat rejection subassembly liquid coolant is also used to cool the alternator and electrical assembly components. For added reliability, two identical power conversion loops (PCL) with their associated coolant loops are available for use. During normal power operation, one PCL is active while the other is inactive. The liquid loop couples the Brayton cycle power to its space radiator.

The weight of the shield required to protect equipment and personnel from the radioisotope heat source radiation is the major weight consideration for radioisotope EPS's. The weight of the shield depends upon:

1. Radioisotope source characteristics
2. Form of the radioisotope (i.e., oxide, etc.)
3. Impurities in the radioisotope heat source
4. System geometry
5. Dose criteria

For preliminary shield analysis, Figure 3.2-18 presents the parametric shielding curve for Pu 238. These curves do not include self-shielding, thus, they give conservative shield thickness. These curves present the thickness of uranium (or other shield materials) and lithium hydride as a function of a shielding parameter DS^2/P, and in some cases this parameter is multiplied by F or (1-F) and (1-k). The terms are defined as follows:

F = Fraction of dose rate from gamma rays

D = Average dose rate, i.e., total integrated dose allowed divided by mission time (mrem/hr)

P = Thermal power (watts)

S = Effective separation distance (ft)

$$S = \sqrt{\frac{\sum_i T_i}{\sum_i T_i / S_i^2}}$$

T_i = Residence time at S_i

S_i = Separation distance (ft)

k = Neutron multiplication factor.

The (1-k) term is only used for the Pu 238 neutron shield thickness, and it is the correction for the neutron subcritical multiplication factor. It is not used for the gamma shield thickness, because the dose rate from spontaneous fission and induced fission gamma rays is much less than that of decay gamma rays of Pu 238.

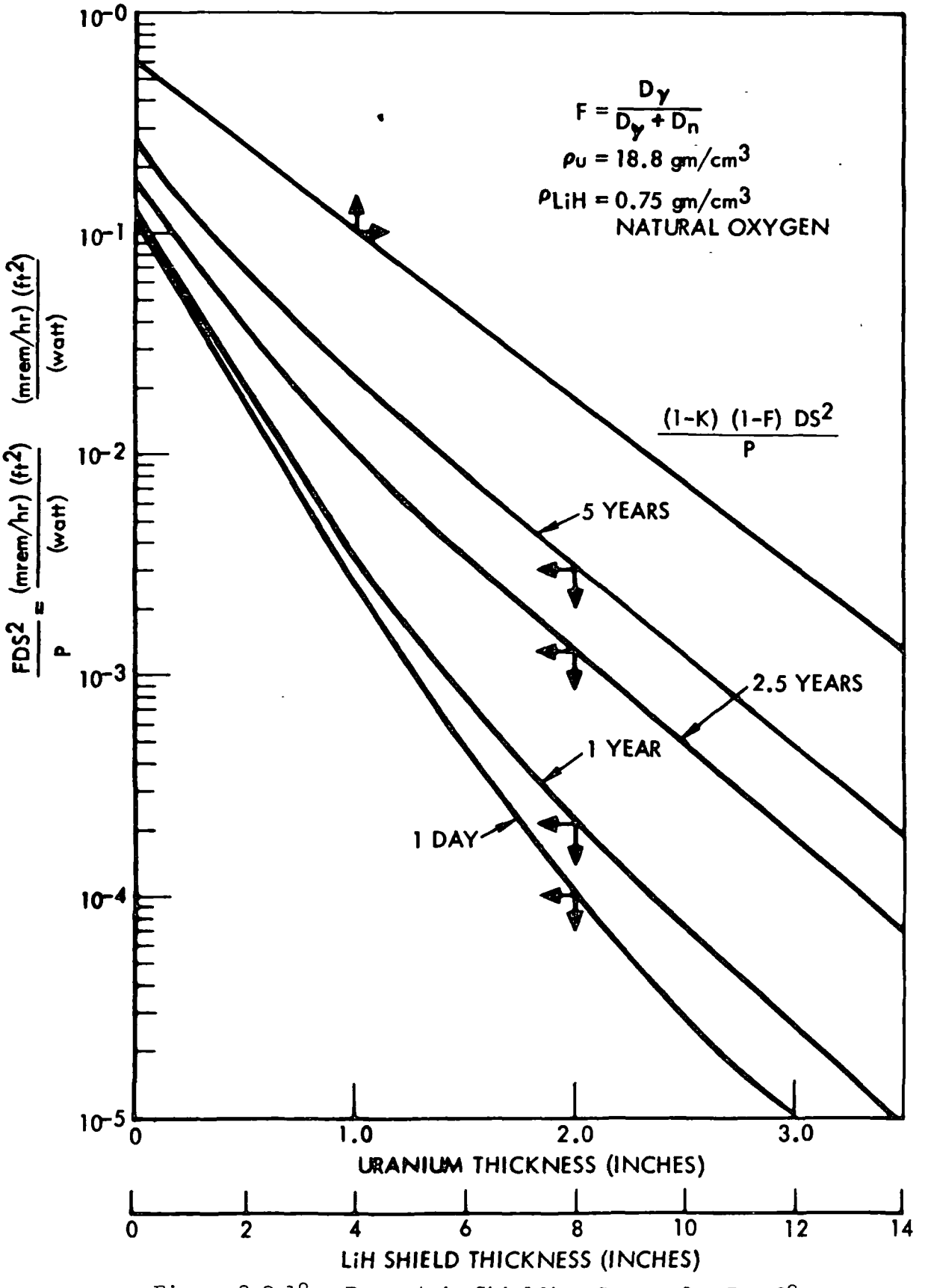


Figure 3.2-18. Parametric Shielding Curves for Pu-238

For Pu 238, using a value of 0.7 for F will result in a minimum weight. This applies for slab geometry and could be used for preliminary work.

Figure 3.2-19 shows a typical parametric example of the shielding requirements for a radioisotope heat source.

Safety considerations play a paramount part in the choice of radioisotopes for space applications. An accidental release of a radioisotope can create a hazardous condition over relatively large areas. Because of this potential danger, an important part of the radioisotope heat source program is to evaluate and solve the nuclear safety problem.

There are three basic general design guidelines:

1. Design for complete containment of the radioisotope under all conditions

2. If complete containment is impossible, design for dispersal and diffusion of the fuel such that it is within the maximum permissible concentration

3. Define an exclusion area where accidental releases may occur (such as launch pads)

For space missions, a nuclear safety analysis has to be performed from the time the radioisotope is produced up to the time capsules are disposed of. This time period can be subdivided into the following:

1. Fuel capsule fabrication
2. Transportation to the launch site
3. Prelaunch activities
4. Mission activities (flight safety)

All the above have to be analyzed with probabilities of malfunctions and fuel released, the quantities and locations of fuels that might be released, and their biological impact on the population. If maximum permissible concentration of that particular radioisotope in air and water has not been exceeded and if the dose to the general population has not exceeded the maximum permissible exposures to external radiation, then the EPS will be acceptable.

Nuclear Reactor-Brayton

The bulk of reactor development for space applications to date is based upon a compact reactor design using zirconium hydride fuel elements. The SNAP-10A and SNAP-8 reactors have been tested extensively. The reactor coolant outlet temperature is limited to approximately 1300 F by the materials selected. The high-temperature, advanced, reactor technology programs (liquid metal cooled and thermionic) have not progressed to the point of integral reactor tests.

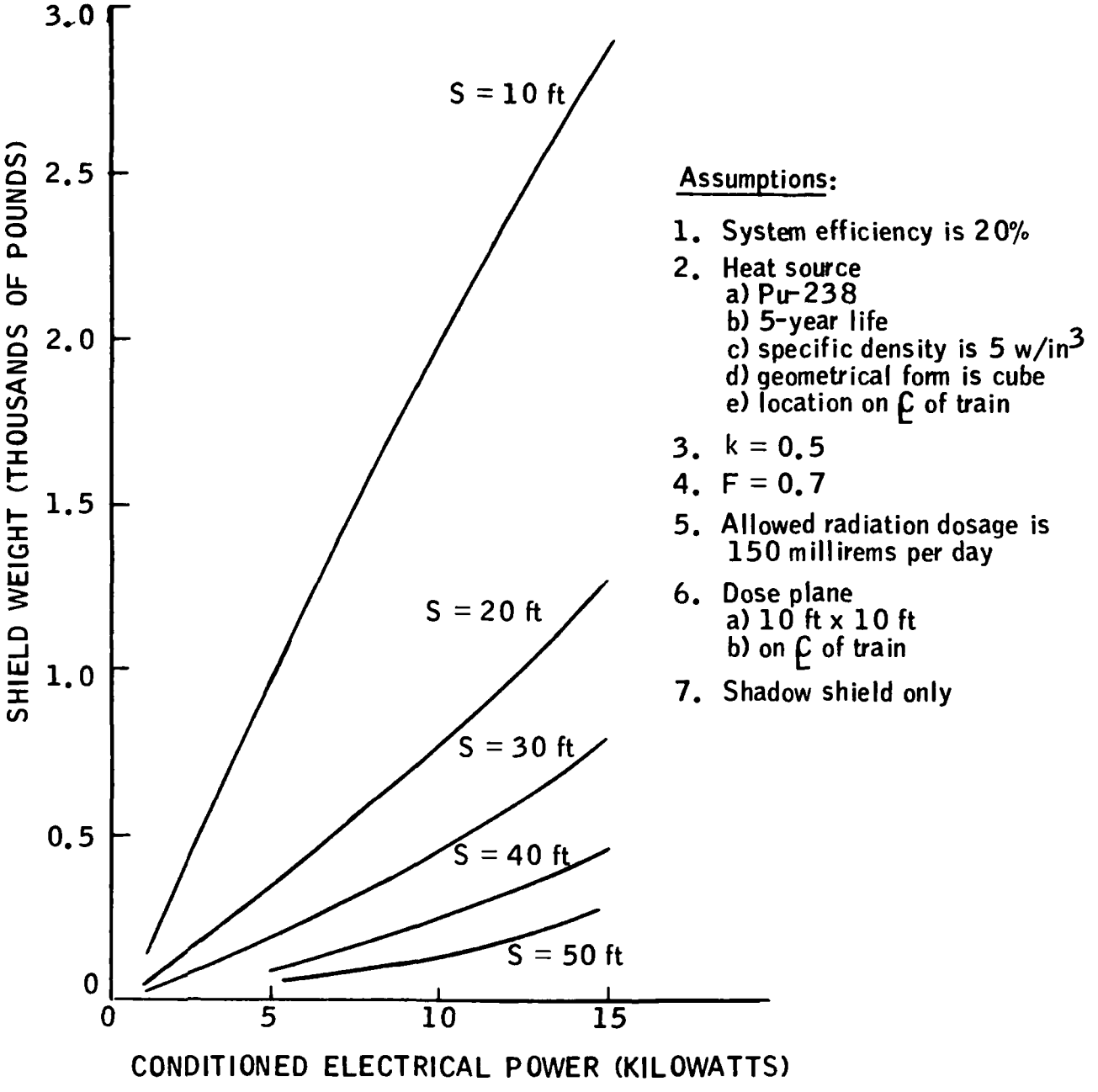


Figure 3.2-19. Radioisotope Shield Weights

The reactor design is similar to SNAP-8 reactor but has a slightly larger core and an internally cooled reflector suitable for operation within an enclosed shield. The reactor design is summarized in Table 3.2-6. Ten cylindrical BeO control drums with neutron-absorbing material on one side are installed in dry wells, the outer surface of which are cooled by the NaK (a eutectic mixture of sodium and potassium), which enters the bottom annular plenum through four inlet nozzles. The NaK flows upward around the dry wells and fixed reflector elements into the upper plenum, downward through the core to the lower plenum, and out through four exit nozzles.

The reactor core consists of a triangular pitch array of 236 fuel elements and has about a 9-inch diameter. The control drums have neutro-reflective material on one side and absorber material on the other. The reactor is controlled by drum rotation, which varies the amount of neutron reflection into the core. Because control-drum cooling is provided internally and the neutron-absorber material eliminates the sensitivity of the reactor to backscattering from outside the control-drum region, the 4π shield can be located directly adjacent to the reactor.

As in the previous SNAP reactors, the fuel element consists of an alloy of 10.5 percent (by weight) fully enriched uranium in zirconium, which is massively hydrided to provide neutron moderation. The hydrogen content of the fuel is 6.3×10^{22} atoms/cc, which is about the same as in cold water. The fuel rod is contained within a 0.015-inch-thick nickel allow cladding tube, which protects it from the NaK coolant and contains the fission products and hydrogen moderator. Because of the significant permeation rate of hydrogen through the bare cladding, a thin glass barrier is fused to the inside of the cladding tube. Small clearances are provided between the fuel rod and cladding to allow for radiation-induced growth of the fuel.

The control drum is 4.5 inches in diameter by 18 inches and consists of BeO reflector material fastened to a neutron-absorbing metal (Ta-10W), which also serves as the main structural member. The drum is supported by self-aligning ball-and-socket-type bearings. The control-drum shaft and the bearing socket are coated with flame-sprayed alumina, which provides a low-friction surface when in contact with the solid graphite ball. Drum rotation is produced by a stepper motor operating through an integral 6:1 gear set. When the drum is in the fully shutdown position, the gear teeth are disengaged by a cam lookout device to prevent drum rotation resulting from launch acceleration.

All parts of the containment vessel and associated structure are of 316 stainless steel. At design conditions, the NaK enters the lower vessel plenum at 1044 F through four 1.5-inch inlet lines. It flows upward in the area between the control-drum dry wells, turns 180 degrees, flows downward through the core, and exits at 1246 F from the outlet plenum. A minor amount of heat is transferred between inlet and outlet coolant through the flow-dividing cylinder. Thus, the NaK mixed-mean outlet temperature from the core is a few degrees hotter than the exiting temperature from the vessel.

Table 3.2-6 Reactor Design Summary

Thermal power (kw_t) (8-man)	480
Outlet temperature (F)	1300
Lifetime (hour) (minimum)	20,000
Temperature rise (F)	200
Pressure drop (psi)	0.65
Number of fuel elements	236
Peak fuel temperature (F)	1420
Fuel material	U-ZrH
Control-drum type	reflector-poison
Number of control drums	10
Reflector material	BeO
Poison material	Ta-10W
Drum radius (inches)	2.25
Peak control-drum temperature (F)	1520
Peak drum bearing temperature (F)	1350
Coolant	NaK-78
Reactor weight (pounds)	1450

The ZrH reactor ultimate performance capabilities are presented in Figure 3.2-20. Estimated reactor lifetimes for various outlet temperatures and power levels are read directly from Figure 3.2-20. It should also be mentioned that a large uncertainty exists in extrapolating these curves beyond 5 to 8 years because no component tests approaching this duration have been performed, and failure modes associated with the control drive assembly, etc., may truncate the reactor lifetime. The effect of lower reactor powers and temperatures on reactor lifetime is significant.

Reactor Radiation Shield

The required reactor shield weight is a function of:

1. Allowable astronaut radiation dosage
2. Reactor-dose plane separation distance
3. Dose plane diameter
4. Shielded reactor volume
5. Reactor power
6. Allowable radiation from top and side of the shield

The reactor shield design shown in Figure 3.2-21 was optimized to meet the radiation dose criteria established for the OLS and will be near minimum weight. The gamma shielding materials used in this design are lead, tungsten, and depleted uranium. Lithium hydride is used for neutron shielding. The shield materials are thickest in the direction of the OLS where the crew will spend most of its time. The side shield thickness and geometry are designed to limit the dose to the tug crew to safe values (Table 3.2-7). The shield cone angle is 36 degrees, which provides a primary shielded zone encompassing the outer tip of a tug docked to the side of an assumed 33-foot-diameter OLS for a reactor-station separation distance of 200 feet (Figure 3.2-22). Reactor shield weights required to obtain the radiation curves presented by Figure 3.2-22 may be obtained from Figures 3.2-23 or 3.2-24. The weight may be read directly from the reactor dose plane separation distance curves at the reactor power required for a specific power system output. Shield weights may be reduced by the weight of the power system components in the gallery.

Although the design shown uses lead for the gamma shield immediately below the reactor, Ta-10W can be substituted at a weight penalty of approximately 300 pounds. The shield directly below the PCS consists of a depleted uranium (U-8Mo) gamma shield followed by a LiH neutron shield. This lower shield may be separately supported, as shown in Figure 3.2-21, so that it can be retained with the OLS when the power system is replaced.

The reactor shield was optimized using the following ground rules:

1. Dose rate = 6.25 m rem/hr (0.15 rem/day)
2. Separation distance = 200 ft
3. Dose plane diameter = 154 ft
4. Side dose rate = 73 rem/hr (to meet the criteria of a tug or CLS stationkeeping at a 1-mile distance (Table 3.2-7)

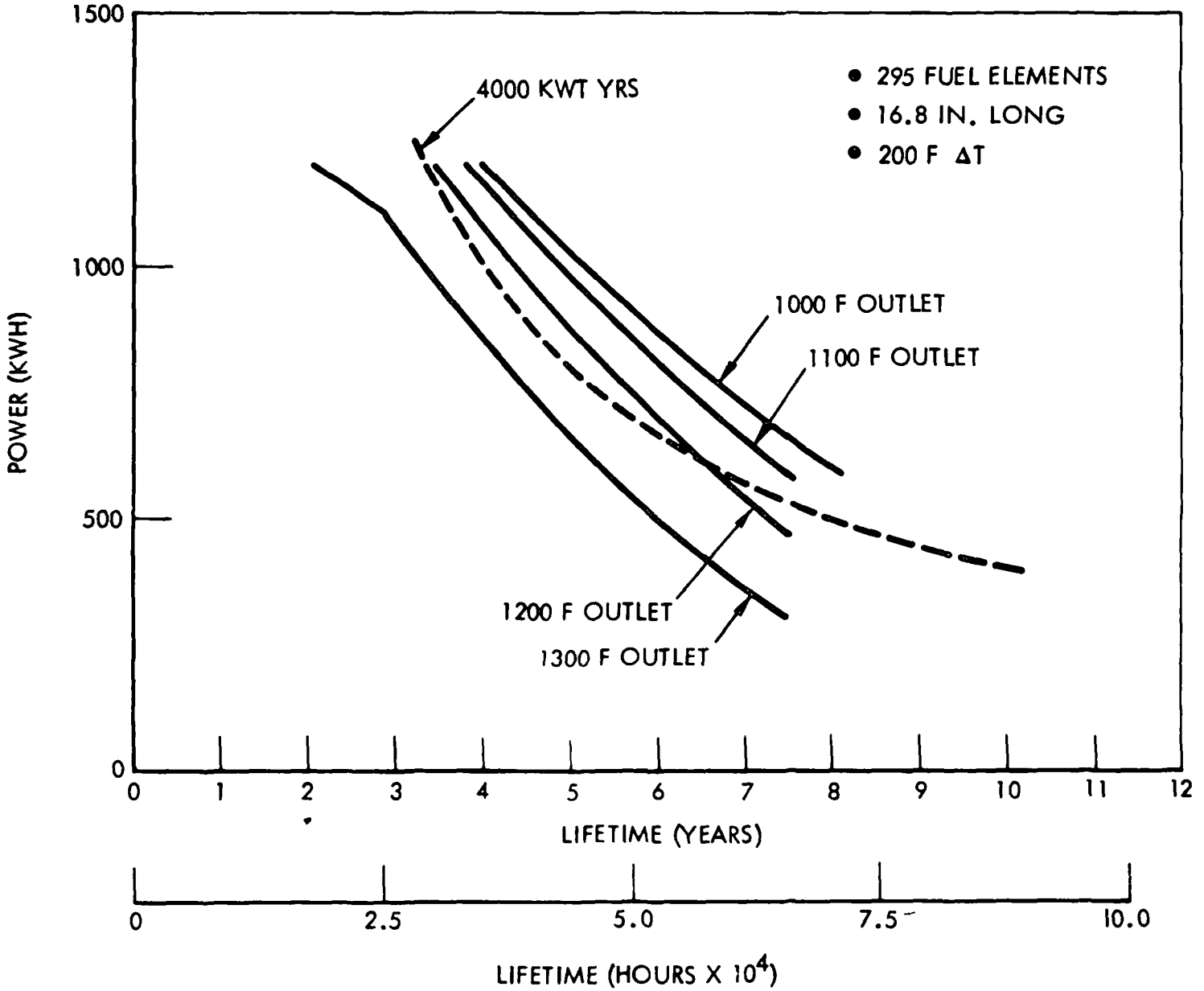


Figure 3.2-20. ZrH Reactor Performance

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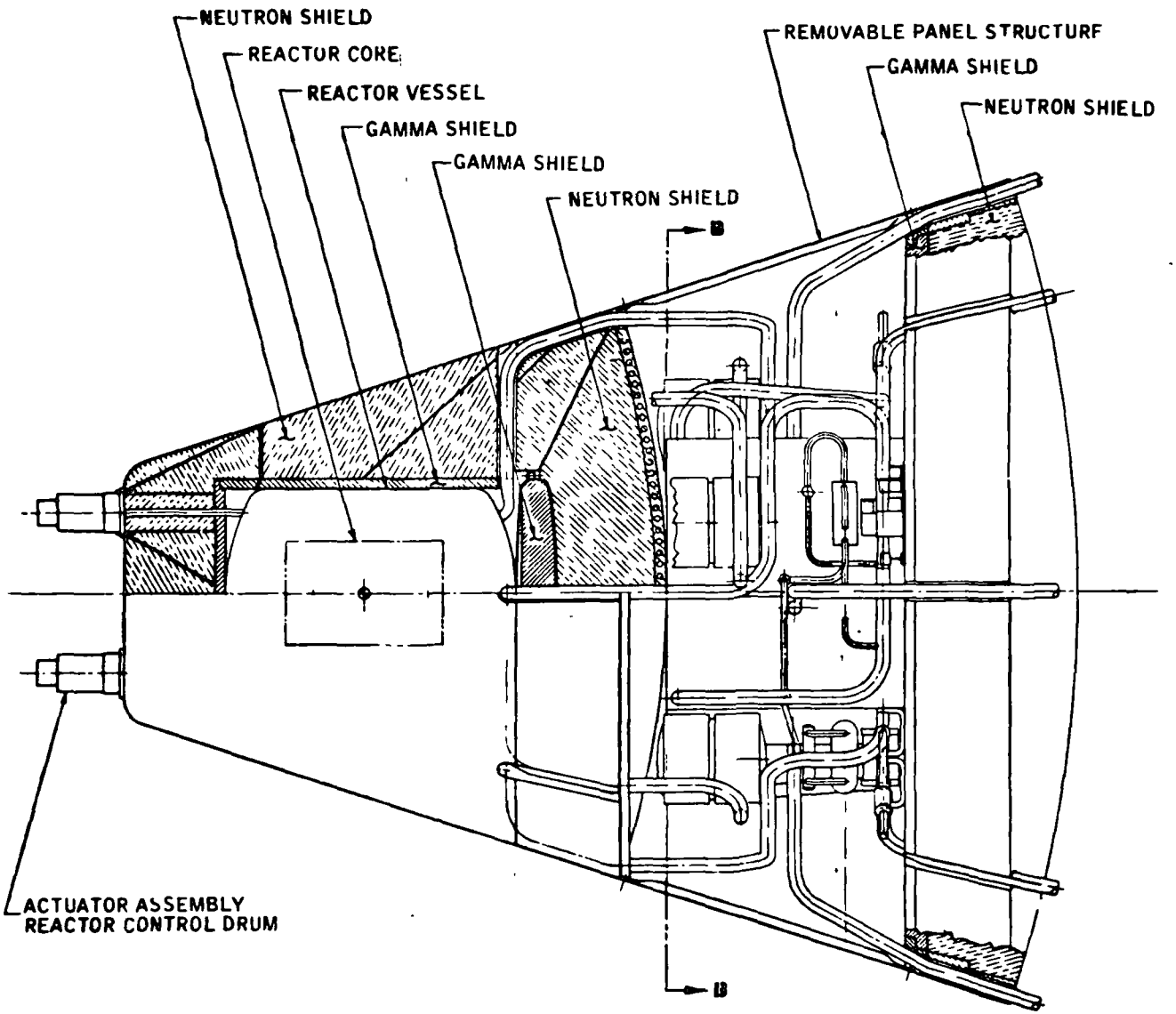
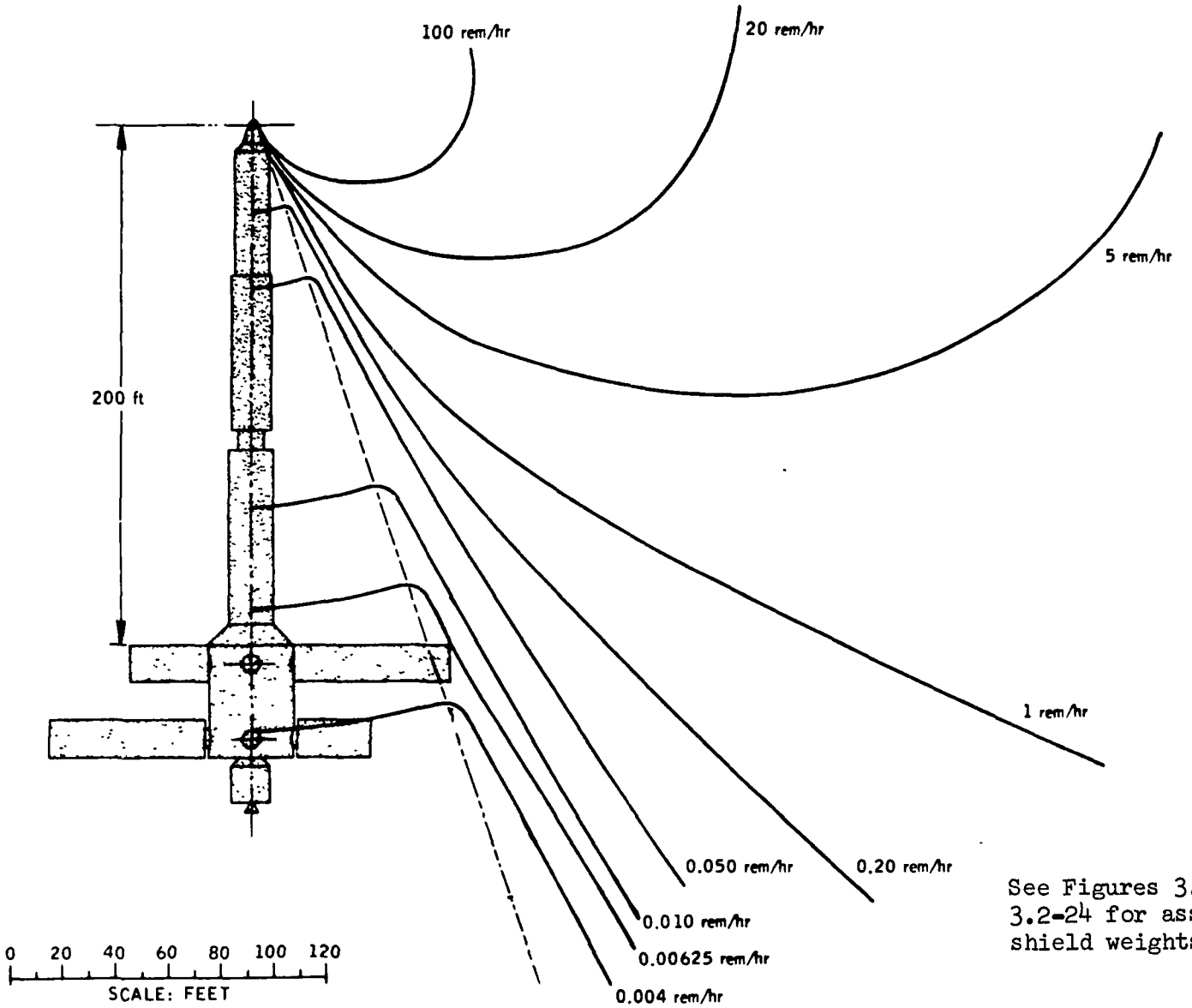


Figure 3.2-21. Reactor Shield Assembly

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See Figures 3.2-23 and 3.2-24 for associated shield weights

Figure 3.2-22. Isodose Plot - OLS Concept

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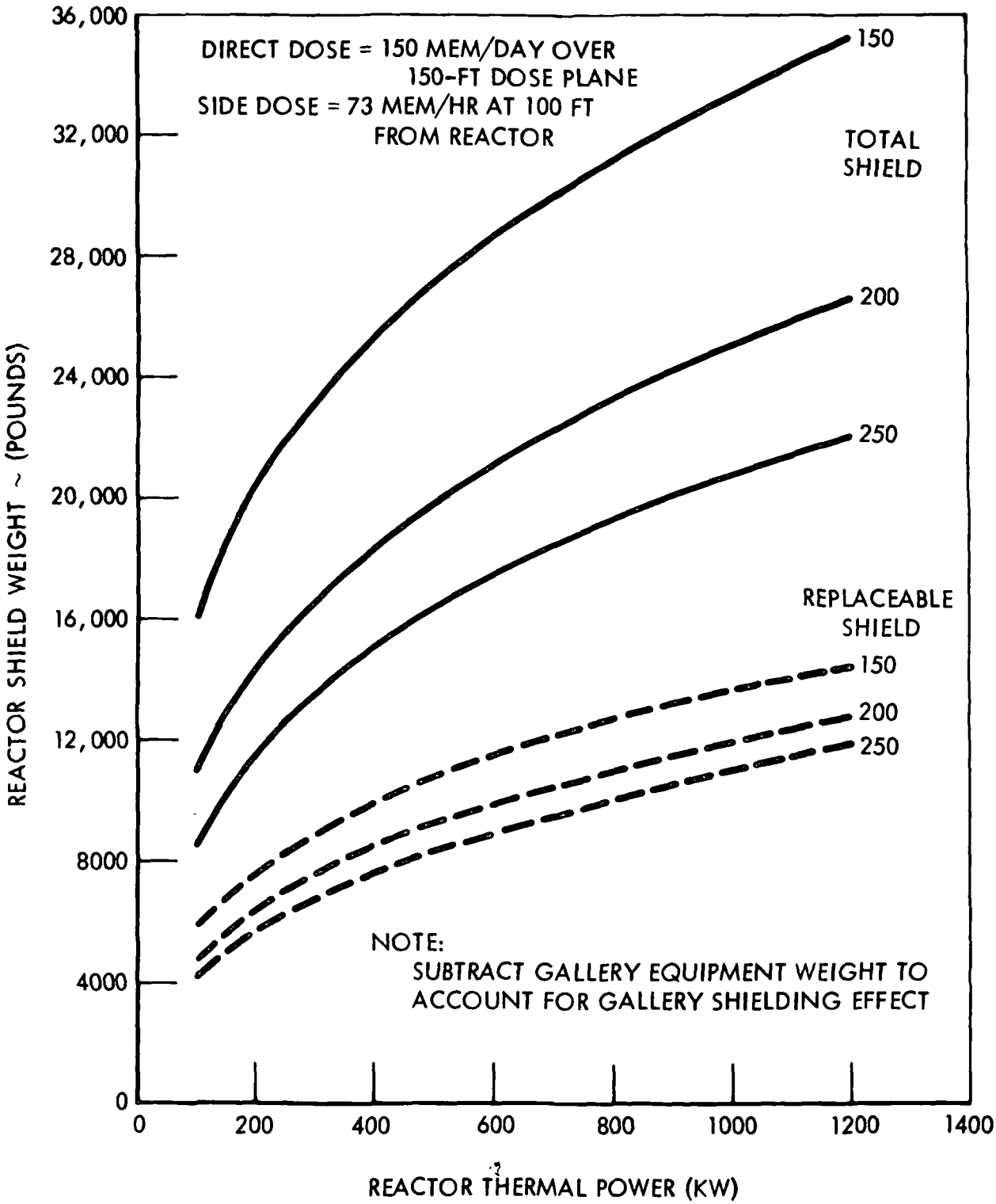


Figure 3.2-23. Shield Weight Versus Thermal Power
(26-Inch Gallery Height)

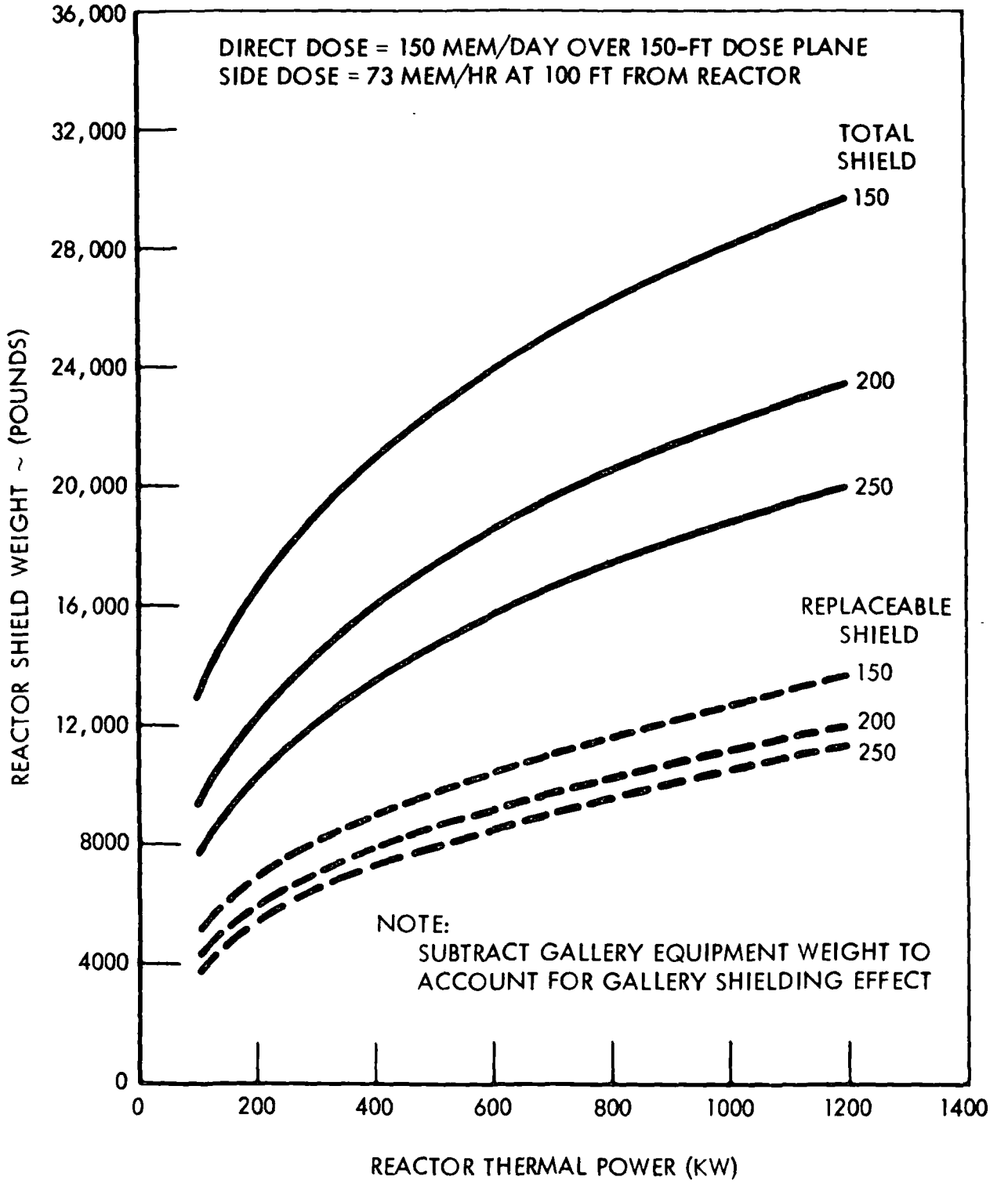


Figure 3.2-24. Shield Weight Versus Thermal Power (12-Inch Gallery Height)

Table 3.2-7. Shielding Criteria and Dose Received

Maximum Dose Allowable	OLS Crew	Tug Crew	CLS Crew	
	11.25 rem/2.5 mo (0.15 rem/day)	11.25 rem/2.5 mo (0.15 rem/day)	25 rem/yr	
Assumptions:			7 trips/year	
1. In OLS	2.5 mo at station mid-height	2.5 mo in forward compartment	0 days/trip	
2. Rendezvous and departures	1	20	-	
3. Loitering 1 mile from OLS			-	16 days/trip
Dose Received (rem)				
1. Reactor source Direct dose in OLS	8.23	8.95	-	-
Tug scattered dose in OLS*	0.12	0.15	-	-
Rendezvous dose	0.02	0.40	-	-
2. Radioisotope source Direct dose in OLS	0.83	0.83	-	-
Total	9.18 rem/2.5 mo	10.33 rem/2.5 mo	0 rem/yr	0 rem/yr

*Tug docked 5 days/month

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Shield weight optimization calculations were performed varying shield angle, direct dose rate, and side dose rate such that all of the ground rules were satisfied. With the upper limit on side dose rate, it was found that the minimum weight occurred when the shield half angle was reduced from 20.6 to 18 degrees, which completely shadows the dose plane, such that the dose at the outer edge of the plane is partially direct and partially from radiation coming from the side of the shield.

Parametric shield weights are shown in Figure 3.2-23 for a 26-inch equipment gallery and in Figure 3.2-24 for a 12-inch equipment gallery. These data are for a direct dose rate of 150 mrem/day and a side dose rate of 73 rem/hr at 100 feet.

For other dose rates and separation distances, the shield weight and size can be determined by using an equivalent reactor power obtained from the following relationships:

$$P_{equiv} = P_o \left(\frac{R_o}{R'}\right)^2 \times \frac{D_o}{D'}$$

R_o = separation distance shown

P_o = actual reactor power

R' = actual separation distance in feet

D_o = 150 mrem/day

D' = desired dose rate

For example, if P_o = 150 kw_t, R' = 300 ft, and D' = 37.5 mrem/day, R_o = 250 ft, then:

$$P_{equiv} = 150 \times \left(\frac{250}{300}\right)^2 \times \frac{150}{37.5} = 416\ kw_t$$

For a 12-inch gallery height and using the 250-foot separation-distance curve, the shield weight for this case (from Figure 3.2-24) would be approximately 13,800 pounds.

The radiation environment created by the reactor is shown in Figure 3.2-22. Results are shown in Table 3.2-7.

The OLS crewmen were assumed to be in the station for 2.5 months, residing on the average at the station mid-point. The tug crewmen were considered in a worst-case condition, assuming 2.5 months of residence in the compartment closest to the reactor and involvement in 20 rendezvous and departures. One case was analyzed for the CLS crewmen. It was assumed they made seven trips a year, spending 30 days per mission trip with the CLS loitering 1 mile from the OLS for sixteen days.

As noted in Table 3.2-7, the shield meets all criteria, with consideration given to scattering from the docked CLS and reduction of the allowable dose from the reactor because of other radioactive sources on the station. Three conditions actually set the shield design:

1. The CLS crew dose limit when loitering 16 days/trip at 1 mile
2. The requirement to shield a docked 60-ft vehicle to 0.15 rem/day (6.25 mrem/hr) over their entire length
3. The worst-case requirement of shielding a tug pilot residing in the forward compartment to an average of 0.51 rem/day

The isodose values in Figure 3.2-22 may seem high. However, the doses received will be small for accidental flyby cases. The expression following presents the maximum dose received in a straight flyby as a function of both constant velocity and minimum distance from the reactor:

$$ID = \frac{638}{VR}$$

where,

ID = Integrated dose during flyby (rem)
V = Constant velocity (fps)
R = Minimum distance from reactor (ft)

For example, if a tug or CLS flies by the reactor at a velocity of 10 fps at a minimum distance of 100 feet, the maximum dose received would be 0.638 rem.

One of the most important variables affecting shield weight is dose plane diameter. Figure 3.2-25 illustrates this point. As required dose plane diameter becomes increasingly larger, for small reactor-dose plane separation distances shield weights comparable to those used for the Earth Orbiting Space Base are approached. Two (Y-configured) reactor type shield weights are shown by Figure 3.2-26. Dose rate versus angle is shown by Figure 3.2-27. These weights are for comparison and were not used for OLS nuclear reactor shield weights.

The specific shielding criteria used was to limit the dose rate to 2.5 rem per reactor at a separation distance of 200 ft and a shield half angle of 30 degrees. Shield weight and size adjustments for other angles and separation distances can be made from the data presented. The dose resulting from scattering from adjacent power systems and the dose received during rendezvous with the tug will be negligible as a result of the relatively thick side shielding provided.

Figure 3.2-27 shows the estimated dose rate profile at a radius of 200 feet for shield angles up to 90 degrees. It will be noted that the radiation dose rate is essentially constant and equal to 1 rem/6 mo (0.23 mrem/hr) for angles up to 20 degrees. At 30 degrees, the estimated dose rate is 2.5 rem/6 mo. The calculated dose rate at 180 degrees ranges from 200 (300 kw_t) to 800 (1200 kw_t) rem/6 mo, depending upon the reactor power level.

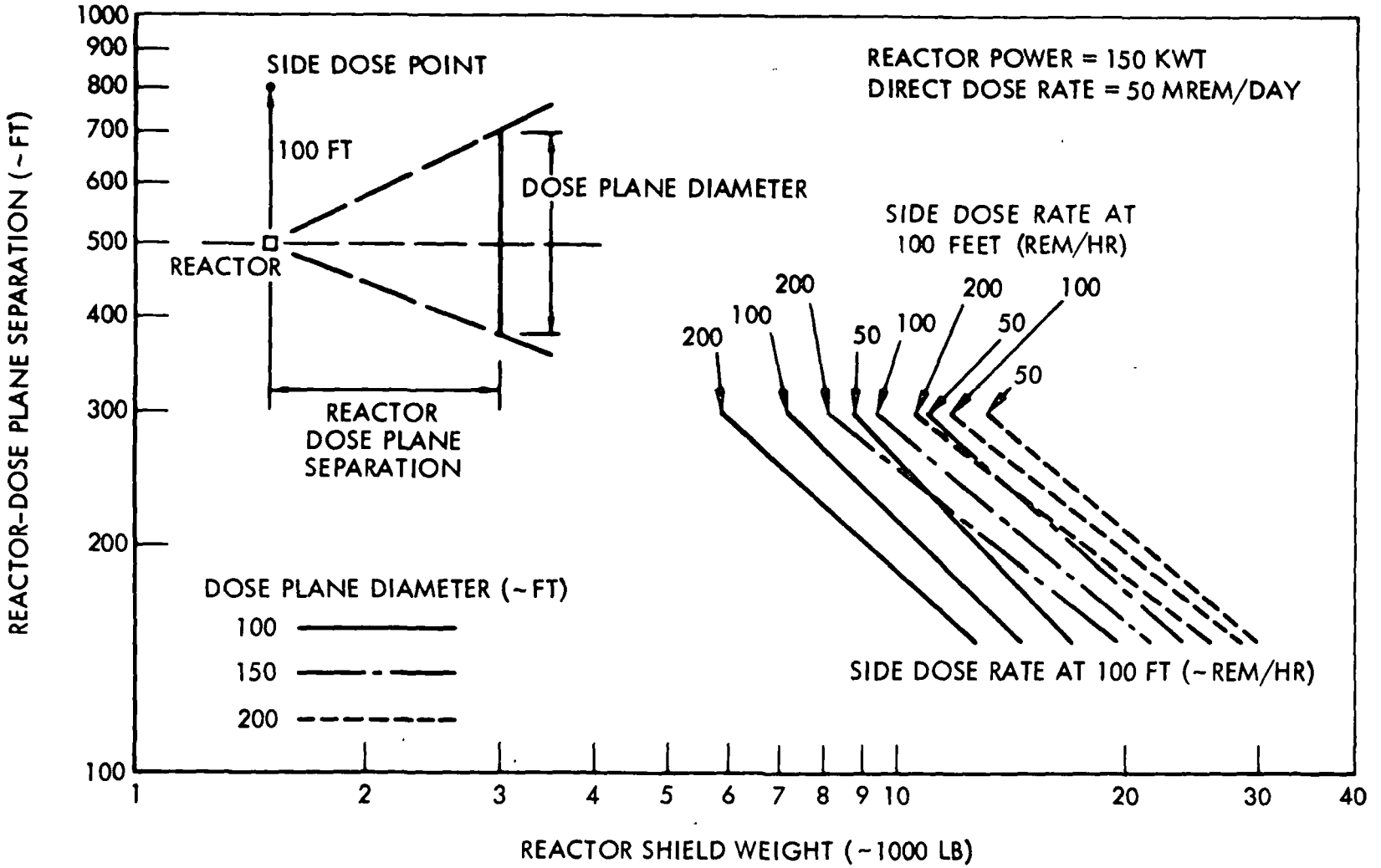


Figure 3.2-25. Effect of Dose Plane Diameter on Reactor Shield Weight

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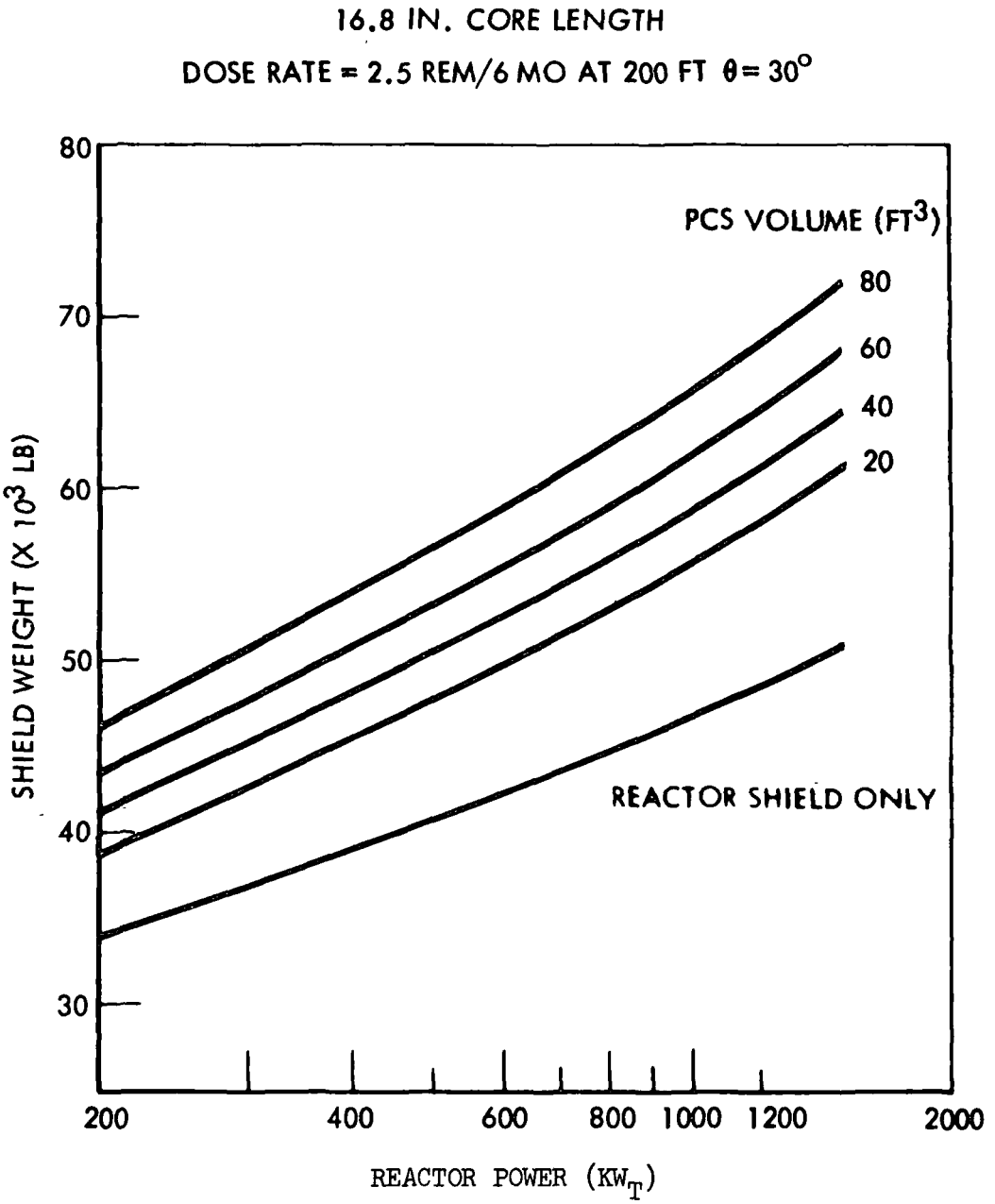


Figure 3.2-26. Y-Configured Type Shield Weight Versus Power

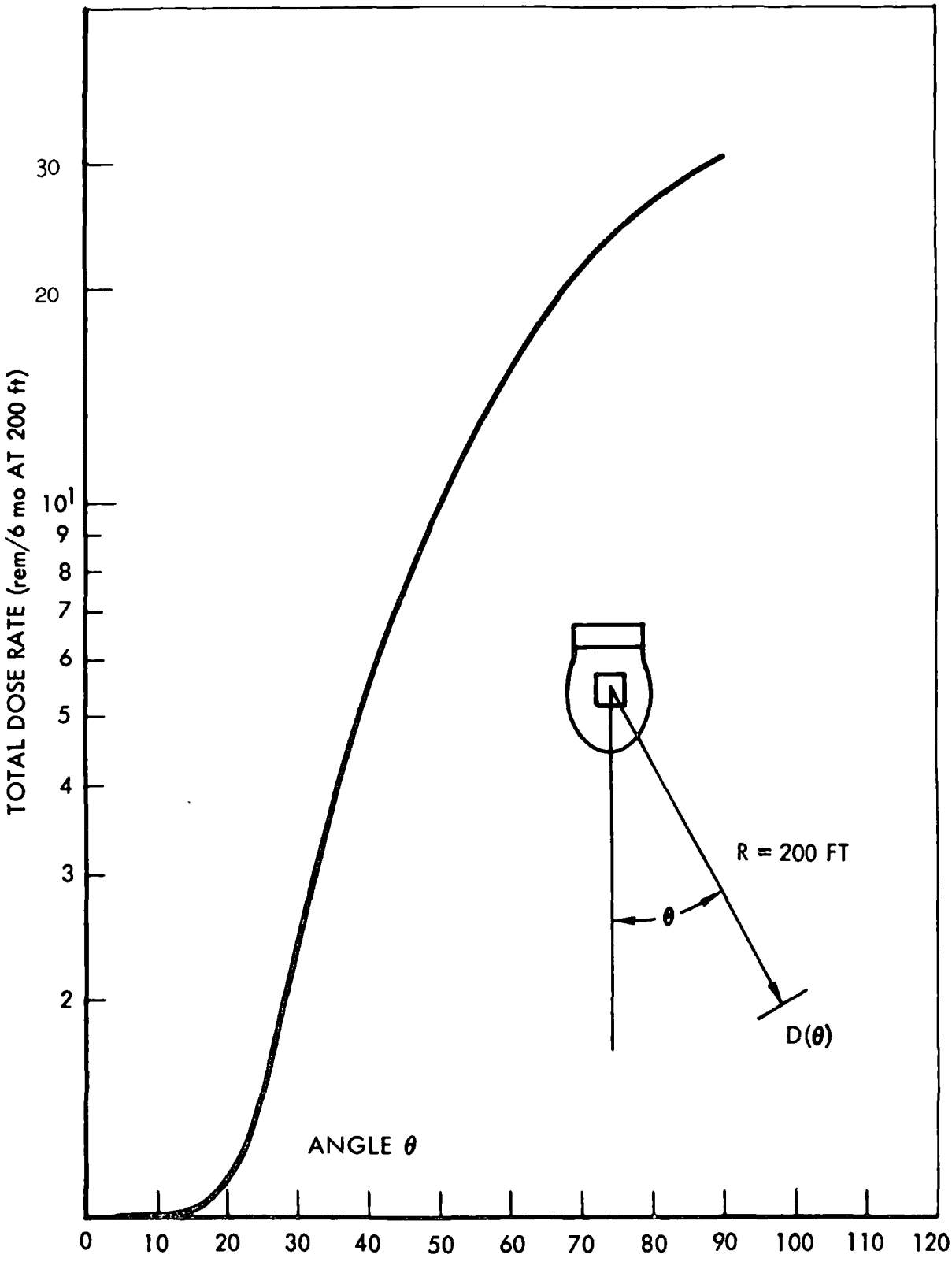


Figure 3.2-27. Y-Configured Type Shield Dose Rate Versus Angle

The equivalent relationship discussed may be used to obtain Y-configured type shield weights for other dose rates and reactor-dose plane separation distances. Figure 3.2-28 illustrates radiation environment from a two (Y-configured) reactor type shield.

Chemical

Two chemical energy sources (batteries and fuel cells) were considered as primary EPS candidates but were eliminated on the basis of weight and resupply requirements. (See paragraph 3.2.4.) However, chemical sources still are a vital part (energy storage) of primary EPS's either for supplying power during dark periods (solar powered EPS's), supplementing peak power demands (nuclear powered EPS's) or for emergency application. Batteries and regenerative fuel cells (separate cells - closed loop) are prime candidates for energy storage.

Primary batteries as energy storage candidates were deleted on the basis of weight and resupply requirements. A large number of charge-discharge cycles and long-life requirements eliminated silver-zinc and silver-cadmium batteries leaving nickel-cadmium batteries as the logical battery candidate for energy storage. Batteries are extremely heavy, have relatively short lifes even with optimum design provisions and will require three major development programs (per-cell-approach concept for battery, charger, and ISS interface). On the other hand, batteries have relatively small volumes and decades of experience. (Refer to paragraph 3.2.5 for additional logic selection rationale.)

The regenerative fuel cell (RFC) candidate is superior to batteries in areas of weight, life, resupply requirements, subsystem integration complexity (ISS), launch/payload costs, and subsystem integration options but is less desirable in the areas of volume, charging/electrolysis efficiency, and experience. The development technical risks of RFC's compared to the three development programs required for the battery candidate, as well as costs are about equal. Radiator requirements for heat rejection are approximately the same, because RFC operation permits heat rejection at a higher efficiency (higher temperatures); therefore, the radiator weight penalty remains about the same. (Refer to paragraph 3.2.5 for cell selection rationale.)

Secondary Electrical Power Sources (Backup Power)

Backup power is required for sustaining operations of the OLS upon loss of primary and/or to sustain operations for repair periods up to 30 days. The main criterion for selection of a backup power subsystem is to maximize the ability to retain primary power levels.

Electrical power must be maintained in the event of any single-point failures to satisfy critical functions and mission continuation. This can be achieved by built-in redundancy and/or including an adequate auxiliary EPS. Various candidates for backup electrical power were evaluated. These included multiple primary EPS's operating at reduced power levels and an auxiliary EPS at both full-power and reduced-power capability (Figure 3.2-29). The evaluation criterion used was to maximize ability to retain primary power levels.

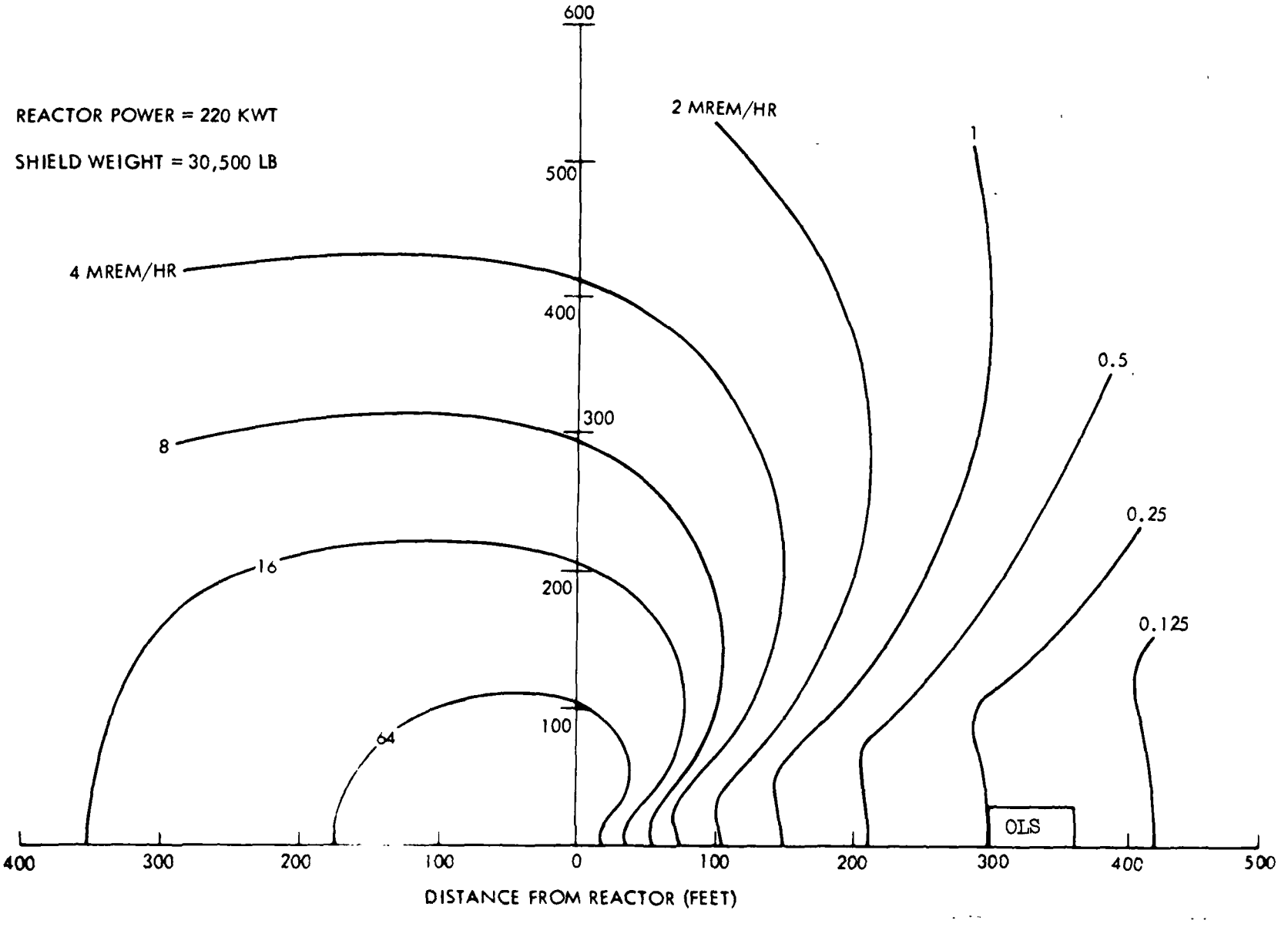


Figure 3.2-28. Two (Y-Configured) Reactors Radiation Environment from One Nuclear Reactor

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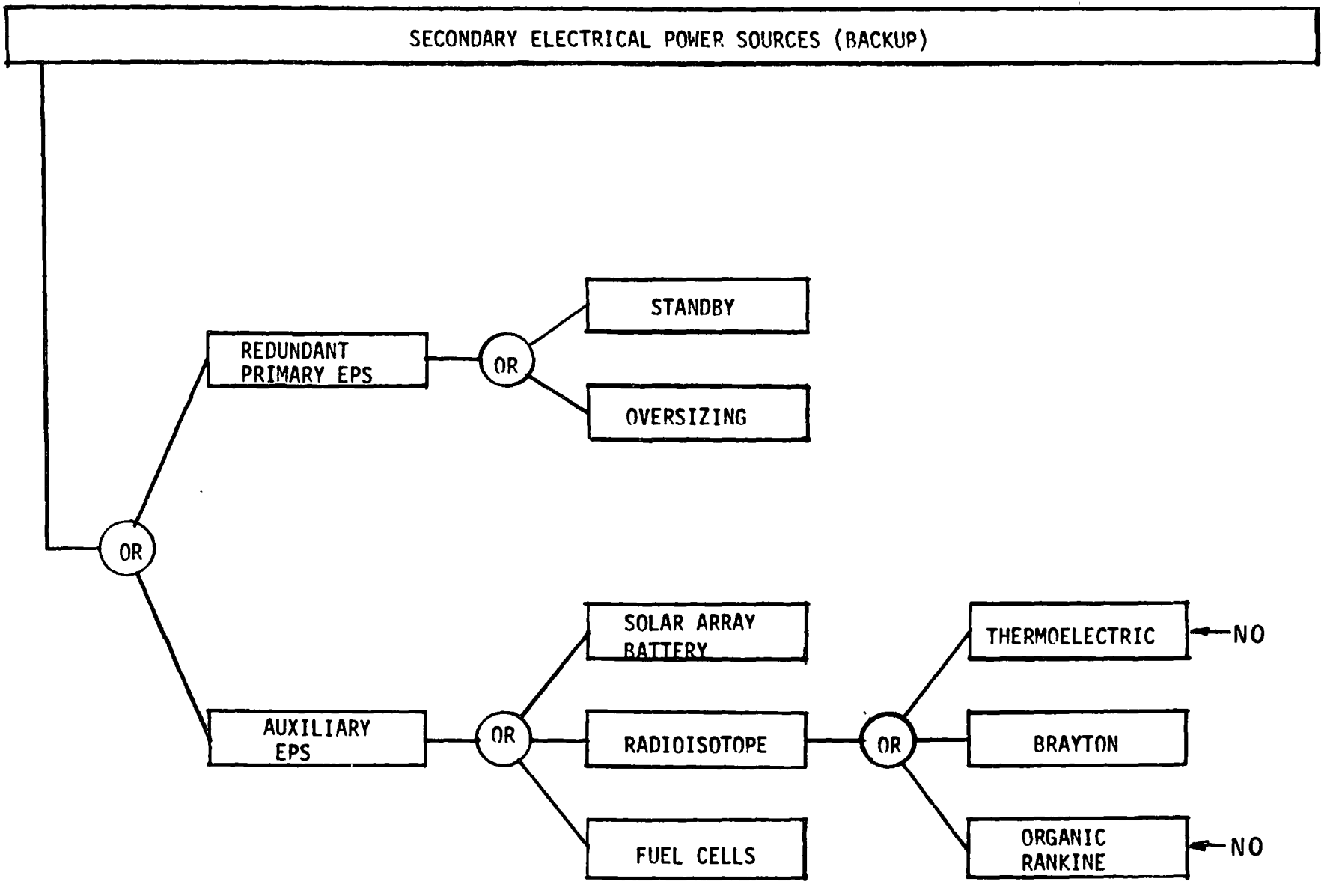


Figure 3.2-29. Secondary Electrical Power Source Trade Tree

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Backup power was considered for six basic primary power configurations:

1. Single reactor source (batteries)
2. Single reactor source (RFC)
3. Solar array - RFC
4. Solar array - battery
5. Isotope - dual heat sources (batteries)
6. Isotope - dual heat sources (RFC)

An EPS tradeoff comparison is summarized in Table 3.2-8. The redundancy inherent in primary power configurations 3 through 6 is considered adequate to meet the requirement of backup power conditions, with the single exception of the period between unmanned launch and premanning and the deployment or operations of the primary sources. All primary sources were considered as being operational up to 5 days after achieving lunar orbit and, once operational, remains so until a failure occurs. The startup cycle of a nuclear reactor is assumed to be 4 hours to full power. Full power is not required in support of the earth/lunar premanning phase of the OLS operations.

The batteries are considered to be a part of the primary system, because their major function is to average demands. These demands vary because of crew work schedule (assumed to be 14-hour day with 10-hour night and variations that occur because of load modulation and crew activity).

Batteries and fuel cells are adequate in configurations 1 through 6 for the anticipated load for premanning operations. The major consideration is, therefore, reduced to backup power requirements for primary power failures.

The EPS options and backup requirements are:

Solar array-battery concept-4 solar array/battery units with 4 FCA's for emergency power.
Backup is not required because emergency power source may be used.

Solar array-RFC concept-4 solar array/RFC units with 4 FCA's for emergency power.
Backup is not required because emergency power source may be used.

Isotope concepts-2 isotope/Brayton cycle units with 4 FCA's for emergency power.
Backup is remaining isotope/Brayton cycle unit and emergency power if required.

Reactor concepts-1 nuclear/Brayton cycle unit with 4 FCA's for emergency power.
Backup is emergency power plus additional backup.

Backup power (reactor concepts) may be provided by a fuel-cell subsystem (Table 3.2-8). The fuel cells feed power through four independent channels. During maintenance or repair, loading may also be reduced to the

Table 3.2-8. Secondary Electrical Power Subsystem Tradeoffs

- Candidate Comparisons
- Interfaces
  - Experiments
  - Habitability
- Suitability with Growth Concepts

Parameter \ Candidates	Thermo-Electric	Organic Rankine	Brayton	Fuel Cell	Solar Array
Maximum utility in conversion processes	Poor	Superior	Superior	Superior	Poor
Minimum development and/or steps for growth	Poor	Good	Good	Superior	Poor
Minimum degradation and resupply/replacement	Poor / Poor	Good / Superior	Superior / Good	Superior / Good	Poor / Good
Maximum ability to retain primary power operation	Superior	Good	Good	Good	Superior
Minimum constraint imposed on operations	Good	Good	Good	Superior	Poor
Minimum dollars per kilowatt hour	Poor	Superior	Superior	Superior	Poor
Maximum crew safety	Poor	Good	Good	Good	Superior

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level of power available from operating channels. The backup power available from the redundant sources will provide a reliable and flexible subsystem that will permit mission continuation in event of failures.

Emergency Electrical Power Sources

In the event of primary subsystem failures electrical power must be maintained to support life on the station for 48 hours (minimum) in earth orbit and 30 days in lunar orbit. Auxiliary electrical power subsystems were considered, including primary silver-zinc batteries (Figure 3.2-30). Size and type of emergency subsystem were determined, with relationship established among primary, backup, and emergency power subsystems. The evaluation criterion used was maximum crew safety.

Maximum emergency power requirement (8 man) is 4 kw for 30 days. Viable candidates were:

1. Primary battery
2. Primary and secondary battery
3. Fuel cells

Related EPS considerations were:

1. Requires power on launch and premanning for up to 5 days before use of prime power
2. Backup power for artificial-gravity period, if required
3. Backup power for a single nuclear configuration that does not restrict flight mode as would solar array backup
4. Strong EOSS/shuttle relationship to reduce total EPS development costs
5. Possibility for common usage of power conditioning hardware with reusable nuclear stage vehicle (RNS baseline power distribution is predominantly dc).

EPS Power Penalty Assessment

The following is the assigned efficiency versus load of EPS components:

Item	Full Load	4-kw Load
Inverter	0.85	0.75
Rectifier-filter	0.95	0.90
Autotransformer	0.97	0.95
Feeder loss		100 watts
EPS control power		250 watts

The calculations for the required electrical energy for emergencies follow:

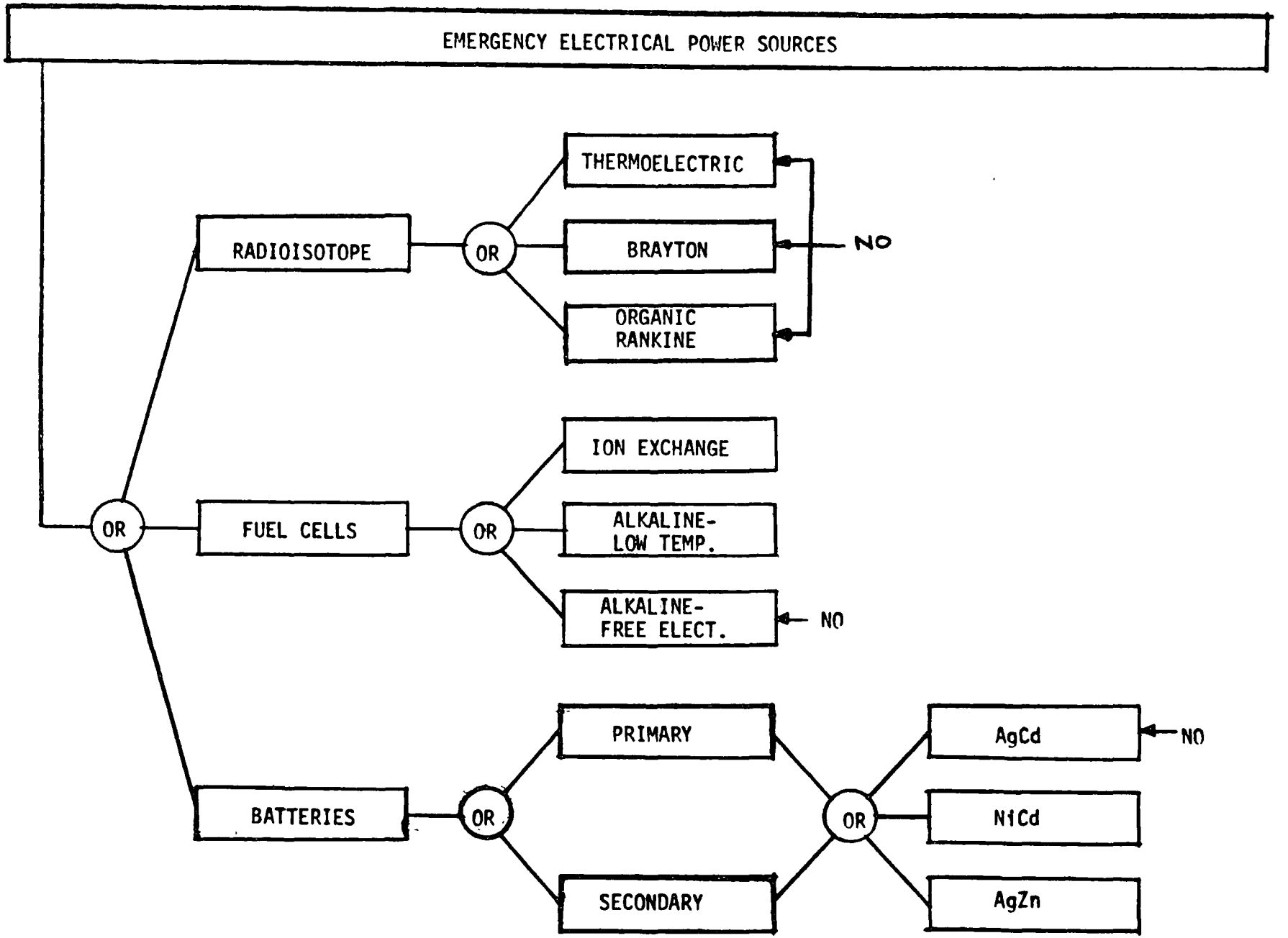


Figure 3.2-30. Emergency Electrical Power Source Trade Tree

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720 hours ⟶ 4 kw load = 3.6 kw ac and 0.4 kw dc

$$\frac{3.6}{(0.75)\times(0.95)} + \frac{0.4}{(0.90)\times(0.75)} = 5.65 \text{ kw}$$

5.65 + 0.350 (feeder losses) = 6.0 kw (total load requirement)

6.0 kw X 720 hours = 4320 kwh's.

Energy Sources

The two electrochemical energy sources considered for this requirement are fuel cells and batteries. In addition to fuel cells, two battery concepts (secondary batteries and a combination of primary and secondary batteries) were considered. Weight calculations for the emergency power sources are included in the following paragraphs.

Primary Battery Considerations for Emergency Power

The load requirement for emergency power is 4320 kw-hours. Primary long life silver-zinc batteries are assumed, although high energy-high density battery technology can change this selection before 1983. The power density assumed for these batteries is 60 watt-hours per pound of batteries = 72,000 pounds (battery weight).

Combination Primary and Secondary Battery Considerations

Available power from secondary batteries used with SAB, RIB or NRB subsystems is somewhat dependent upon the configuration. However, the difference is small between NRB and RIB concepts.

Secondary battery size has been based on nickel-cadmium plates and a 15-percent depth of discharge. The reserve capacity that is usable for emergency power is assumed to be 50 percent of the net capacity left after 15-percent discharge. The best case SAB station requires a total battery capacity of 720 ampere-hours (8-man), and the NRB/RIB station requires only 180 ampere-hours, with both having nominal discharge voltage of 112 vdc.

NRB/RIB Case

180 amp-hr (112 volts) (1 hr)	= 20,150 watts hours
Less 15 percent of 20,150	— 3,020
	17,130
(assume 50 percent avail)	= 8,565 watt-hours

Solar Array Battery (SAB)

720 amp-hr (112 volts) (1 hour) = 80,600 watt-hours
Less 15 percent of 80,600 = − 12,100
68,500

50 percent of above capacity (available) 34,250 watt-hours

4,320,000 watt-hours gross emergency requirement
-34,250 watt-hours(available from secondary NiCd)

4,285,750 watt-hours required from primary batteries

$$\frac{\text{4,285,750 watt-hours net emergency requirement}}{\text{60 watt-hr/lb}} = \text{71,400 pounds primary battery weight}$$

Fuel Cells

All emergency configurations are influenced by the selection of the primary power and distribution subsystem. The efficiency of the emergency subsystem is low because of the large ratings of the power-conditioning equipment that have to be energized. Reactant weight calculation for 4,320,000 watt-hours follows:

(4320) (0.87 lb/kw-hr) = 3760-lb total

(418 pounds of H_2 and 3342 pounds of O_2)

The average voltage per cell is approximately 0.863 under full load. The average voltage for the 30 days at 4 kw is estimated to be above 1 volt. This will reduce the total reactant consumptions in direct proportion to the voltage improvement over the average voltage. Another factor that can reduce reactant consumption is the possibility of turning off one or two a-c busses, providing some efficiency improvement. The weight of the fuel cells must also be considered in comparing candidate weights.

No reactant tankage is required for EPS emergency subsystem based on the following:

1. Reactants are provided by propulsion and environmental control subsystem (assuming that RCS uses cryogenic oxygen and hydrogen).

2. The actual consumption is expected to be less than 3760 pounds of reactant based on the voltage output of the fuel cells being up to 20 percent higher than the average of 0.863 volt, which has been measured on cells designed for a 3000-hour life. The expected usage, including ground test, launch, checkout, post-launch, and four emergency cycles of 24 hours, would be less than 1000 hours.

3. The water byproduct can be returned to the environmental control subsystem for OLS use. This water may be electrolysed into reactant if and when power is available after the emergency. An alternative is to replace reactants on the next CLS resupply trip.

4. No reactant allowance has been made to meet unmanned launch and premanning power for 5 days.

5. No penalty is included for fuel cell cooling, because the fuel cells will not be operated at the same time that all other equipment is operating. The existing radiator is adequate to cool both the emergency electrical load and the fuel cells.

The entire emergency load can be met with two of the four fuel cells operative.

For emergency and use before manned operations of the station, the fuel cell can provide more than 2000 hours of operation on the station, because the MTBF is in excess of 4000 hours and only two out of four units are required to meet this emergency load requirement. Four 7.0-kw units can deliver 40 kw from the fuel cell, which will be greater than 32 kw at the secondary bus. However, actual power delivered to the secondary bus may be limited by the rating of the power-conditioning equipment and the capacity of the cooling subsystem.

A comparison of the candidate weights shows battery candidates may be eliminated as emergency power sources.

The emergency power assembly may consist of four fuel cells, one for each channel of the primary EPS power subsystems. To achieve commonality and reduce development costs, fuel cells developed for the EOSS should be considered for OLS application.

EPS CONCEPT COMPARISONS

The EPS candidate concepts evaluated are:

1. Solar power with batteries supplying power during dark orbital periods.
2. Solar power with regenerative fuel cells (separate cells) supplying eclipse power
3. Nuclear reactor-Brayton cycle with batteries supplying the peak loads.
4. Nuclear reactor-Brayton cycle with RFC's supplying the peak loads.
5. Radioisotope-Brayton cycle with batteries supplying the peak loads.
6. Radioisotope-Brayton cycle with RFC's supplying the peak loads.

One line diagrams for the above six EPS candidates are shown in Figures 3.2-31 through 3.2-36 respectively. EPS weights, based on an 8-man crew, are summarized in Tables 3.2-9 through 3.2-14 respectively. The weights do not include weight of the power booms but do included resupply requirements for a 10-year period. Distribution and control, etc., system weights are assumed for total vehicle EPS. EPS weight as a function of conditioned power is shown in Figure 3.2-37. EPS hardware and in-lunar-orbit costs are shown in Figures 3.2-38 and 3.2-39 respectively.

The relative ranking of each EPS candidate based on weights and costs is summarized in Table 3.2-15.

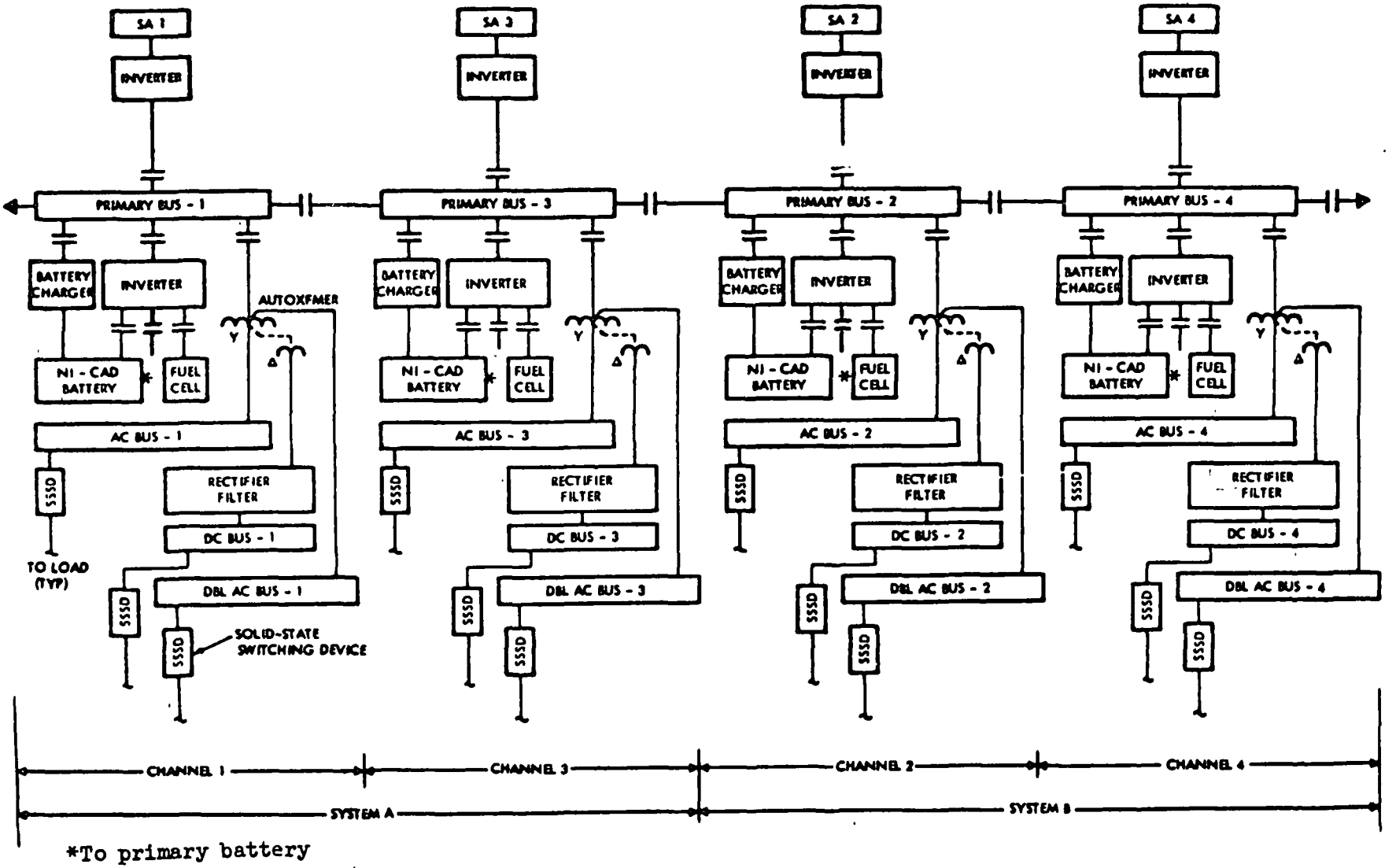


Figure 3.2-31. Derivative OLS EPS (Solar Array/Battery)

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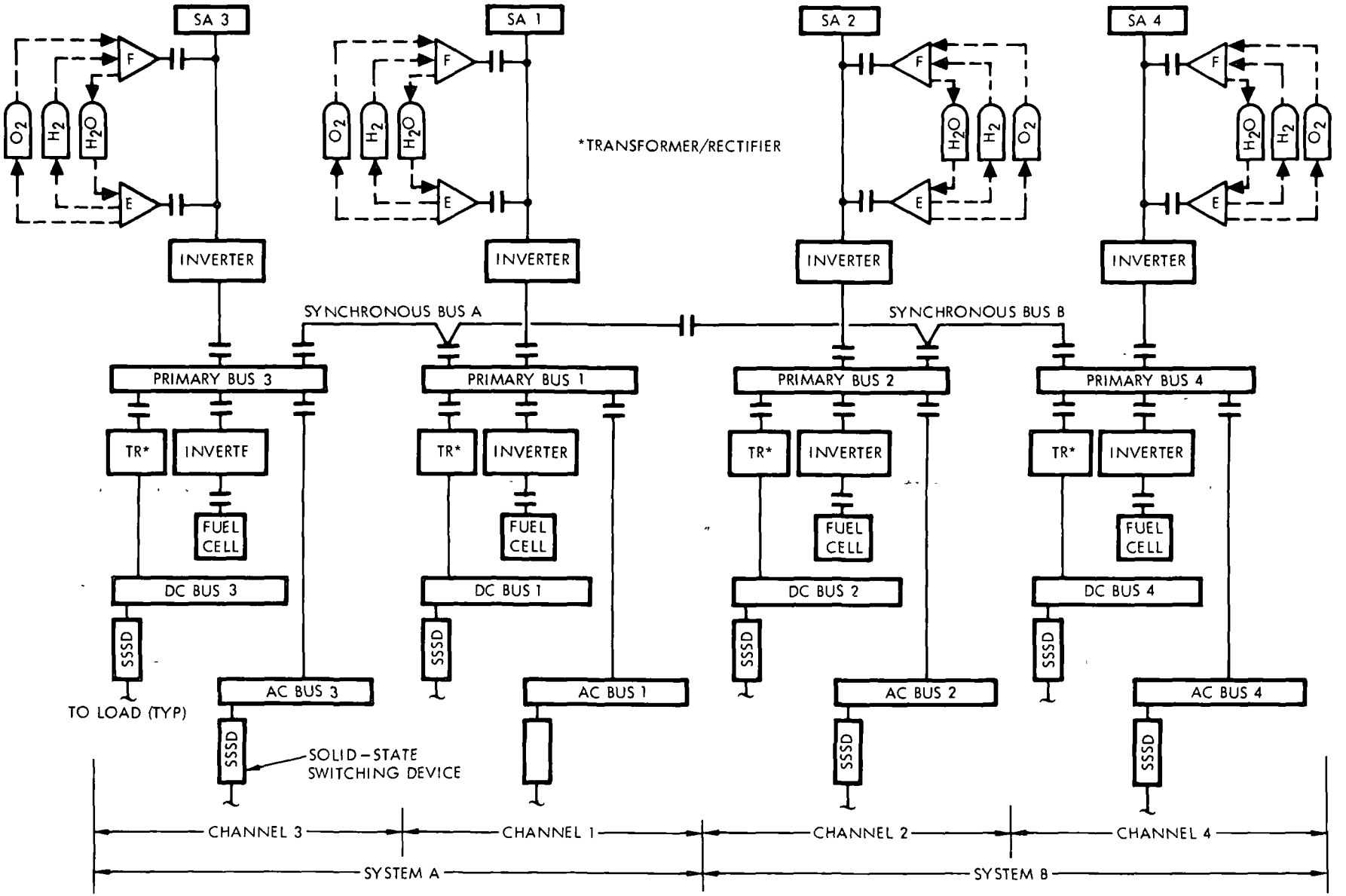


Figure 3.2-32. EPS (Solar Array/RFC) One-Line Schematic

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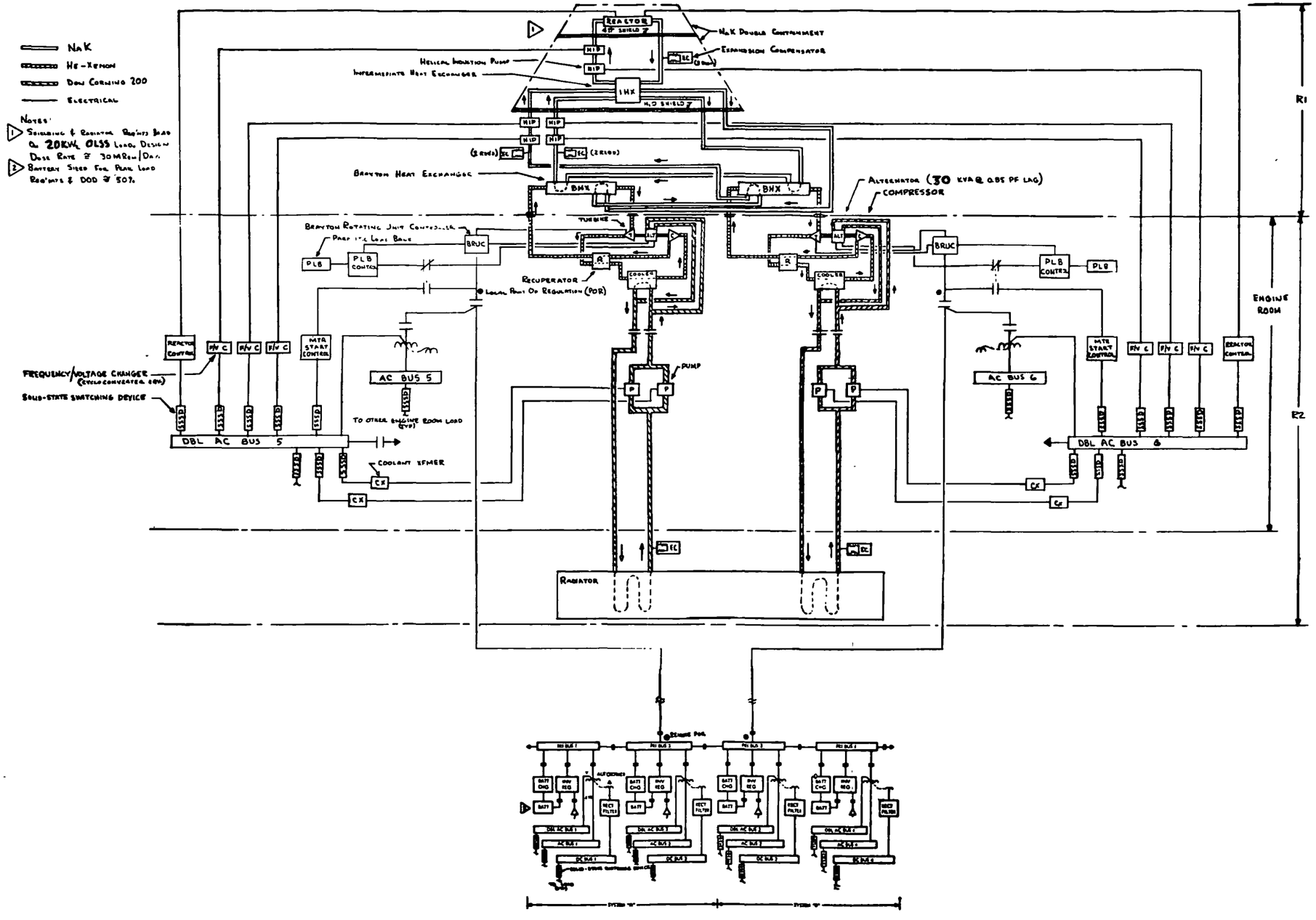


Figure 3.2-33. EPS (Nuclear Reactor-Brayton)

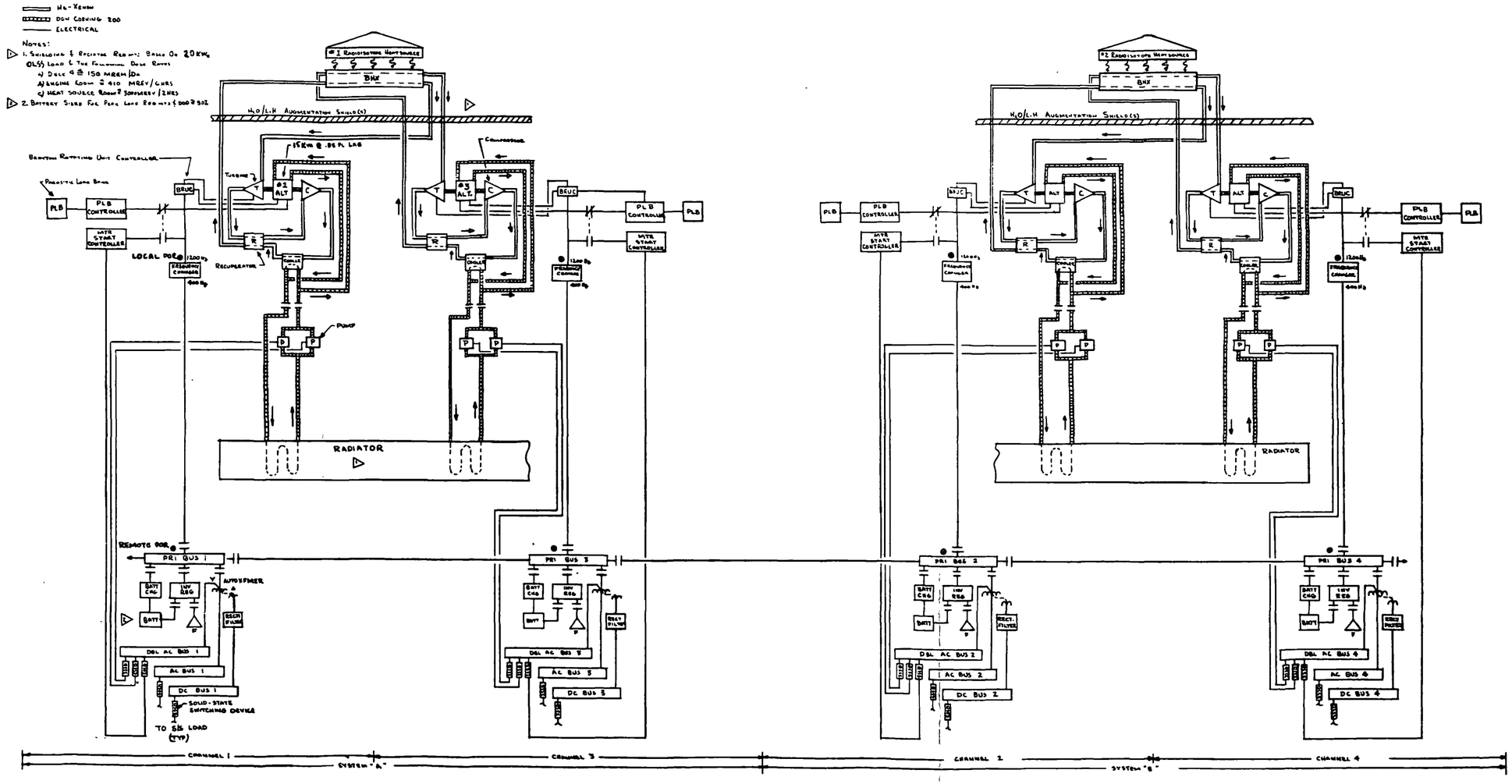


Figure 3.2-34. EPS (Radioisotope-Brayton)

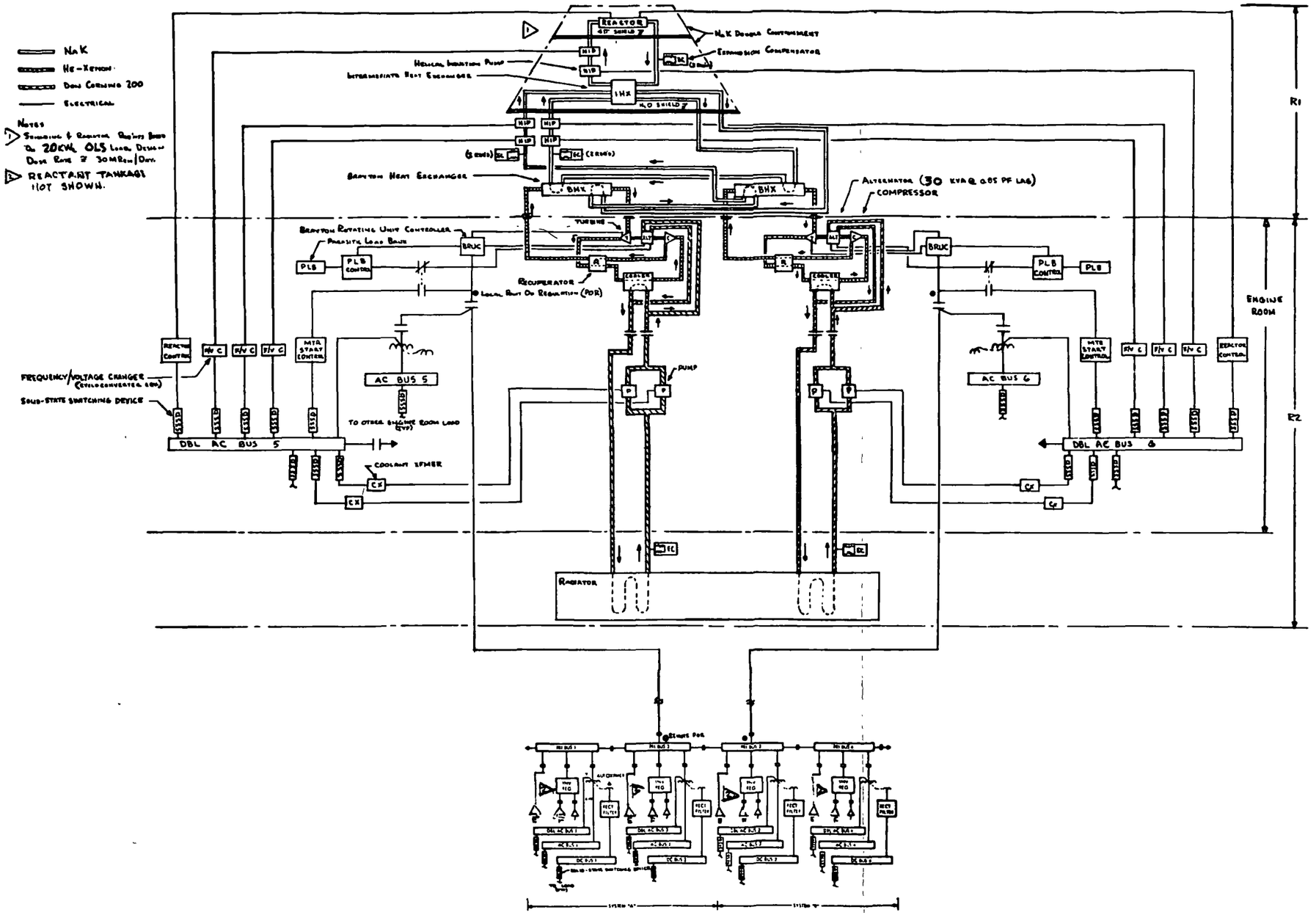


Figure 3.2-35. EPS (Nuclear Reactor-Brayton)
(RFC for Peak Loads)

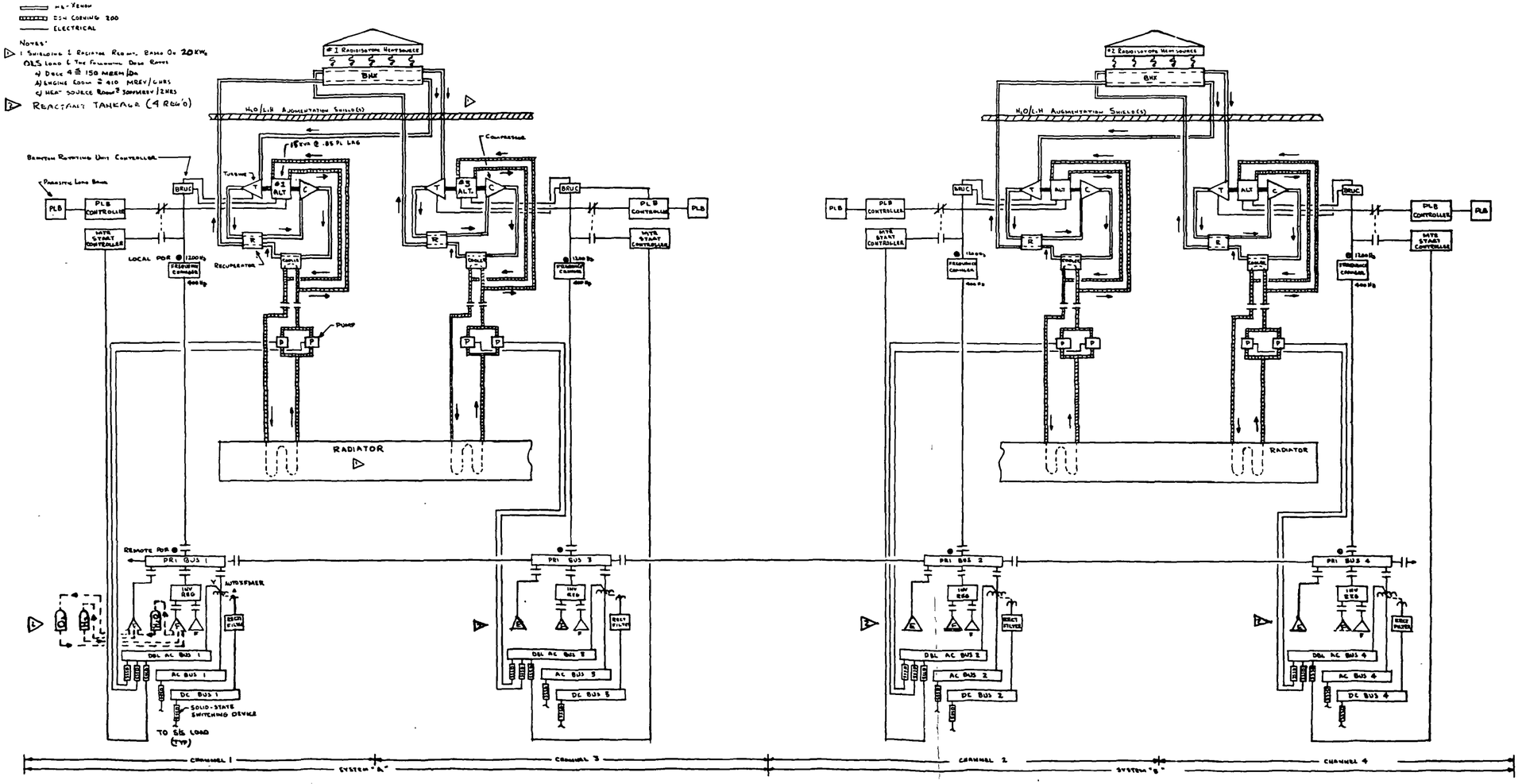


Figure 3.2-36. EPS (Radioisotope-Brayton) (RFC for Peak Loads)

Table 3.2-9. Solar Powered--Battery EPS Weight Summary (8-Man)

			Initial Launch	Orbit	Ten Years
SOLAR ENERGY CONVERSION			5,815	5,815	15,255
Solar Cell Rollup Array ($10{,}000\ ft^2$)		4,720			
Orientation Mech		286			
Array Support (Boom)		374			
Mounts and Supports		434			
FUEL CELLS (EMERGENCY)			607	607	607
Fuel Cells (4)		552			
Mounts and Supports		55			
ENERGY STORAGE			5,569	5,569	20,758
Batteries (4)		5,063			
Mounts and Supports		506			
POWER CONDITIONING			1,735	1,735	4,829
Chargers	Core	160			
Controls	Core	80			
Inverters (4-5 KVA)	Core	252			
Secondary Bus	Core	300			
Mounts and Support	Core	112			
Inverters (4-15 KVA)	Boom	755			
Mounts and Supports	Boom	76			
DISTRIBUTION CONTROL AND WIRING			3,451	3,451	3,567
Wiring	Core	3,000			
Buses, Contactors	Core	58			
Mounts and Supports	Core	369			
Wiring	Boom	20			
Mounts and Supports	Boom	4			
LIGHTING			598	598	2,330
Lights		544			
Mounts and Supports		54			
TOTAL (Pounds)			17,775	17,775	47,346

Table 3.2-10. Solar Powered--FC--Electrolysis
EPS Weight Summary (8-Man)

			Initial Launch	Orbit	Ten Years
SOLAR ENERGY CONVERSION			5,815	5,815	15,255
Solar Cell Rollup Array ($10{,}000\ ft^2$)		4,720			
Orientation Mech		286			
Array Support (Boom)		374			
Mounts and Supports		434			
FUEL CELLS (EMERGENCY)			607	607	607
Fuel Cells (4)		552			
Mounts and Supports		55			
FUEL CELLS--ELECTROLYSIS			1,050	1,050	3,150
Fuel Cells (4-5 KW)		721			
Electrolysis Units (4)		251			
Reactants		38			
Tankage		40			
POWER CONDITIONING			1,242	1,242	3,402
Controls	Core	150			
Inverters (4-2.5 KVA)	Core	125			
Secondary Bus	Core	300			
Mounts and Supports	Core	112			
Inverters (4-10 KVA)	Boom	504			
Mounts and Supports	Boom	50			
DISTRIBUTION CONTROL AND WIRING			3,451	3,451	3,567
Wiring	Core	3,000			
Buses, Contactors	Core	58			
Mounts and Supports	Core	369			
Wiring	Boom	20			
Mounts and Supports	Boom	4			
LIGHTING			598	598	2,330
Lights		544			
Mounts and Supports		54			
TOTAL (Pounds)			12,763	12,763	28,311

Table 3.2-11. Reactor/Brayton--Battery EPS Weight Summary (8-Man)

		Initial Launch	Orbit	Ten Years
R-1 MODULE			14,261	28,522
Reactor	1,450			
Reactor Support Structure and Containment	180			
Primary Loop	--			
Ducting	50			
Expansion Compensators (3)	75			
HIP (2)	300			
Insulation	6			
Intermediate Loop	--			
Ducting	500			
Expansion Compensators (4)	90			
HIP (4)	600			
Insulation	50			
IHX	60			
BHX (2)	90			
Control and Wiring	200			
Radiation Shield (Solid)*	10,600			
R-2 MODULE		5,468	5,468	13,898
PCU's (3)	3,320			
PCU Controls and Auxiliaries (3 sets)	736			
Parasitic Load Resistors (2)	80			
Radiator Fin, Tube, Armor and Fluid	1,205			
Radiator Manifolds	50			
HRL Auxiliaries	77			
FUEL CELLS (EMERGENCY)		607	607	607
Fuel Cells (4)	552			
Mounts and Supports	55			
ENERGY STORAGE		2,244	2,244	8,364
Batteries (4)	2040			
Mounts and Supports	204			

*Shields Dose Plane (204 ft) to 150 mrem/day

Table 3.2-11. Reactor/Brayton--Battery EPS Weight Summary (8-Man) (Continued)

			Initial Launch	Orbit	Ten Years
POWER CONDITIONING			840	840	2,250
Chargers	Core	160			
Controls	Core	80			
Inverters (4-2.5 KVA)	Core	125			
Secondary Bus	Core	320			
Mounts and Support	Core	135			
Primary Bus	Core	20			
DISTRIBUTION CONTROL AND WIRING			3,751	3,751	4,227
Wiring	Core	3,000			
Buses, Contactors	Core	58			
Mounts and Supports	Core	369			
Wiring	Reactor Module	120			
Mounts and Supports	Reactor Module	24			
Secondary Bus	Reactor Module	150			
Buses, Contactors	Reactor Module	20			
Primary Bus	Reactor Module	10			
LIGHTING			598	598	2,330
Lights		544			
Mounts and Supports		54			
TOTAL (Pounds)			13,508	27,769	60,194

Table 3.2-12. Radioisotope/Brayton--Battery
EPS Weight Summary (8-Man)

		Initial Launch	Orbit	Ten Years
HEAT SOURCE		4,898	4,898	4,898
Fuel Capsules (2)	2,290			
Heat Source Plate (2)	1,910			
Support Structure (2)	251			
Insulation (2)	447			
AEROSHELL		1,350	1,350	1,350
Heat-Shield and Bond (2)	900			
Structure (2)	315			
Fence (2)	135			
RECOVERY AIDS		360	360	360
Ballute (2)	280			
Floatation Aids (2)	40			
Location Aids (2)	40			
SHIELD		9,750	9,750	9,750
Augmentation Tank*	--			
Augmentation Water**	9,250			
Augmentation LiH (2)***	250			
HEAT REJECTION		1,519	1,519	1,777
Radiator (1380 ft^2)	1,175			
Pump Assembly, Accumulator and Cooling System Components (4)	200			
Cold Plates (4)	144			
Emergency Cooling (2)****	--			
ENERGY CONVERSION		3,344	3,344	4,573
Brayton Rotating Unit (4)	268			
Heat Exchanger (Heat Source) (4)	588			
Cooler (4)	36			
Recuperator (4)	280			
Interconnecting Ducts (4)	112			
Gas Management (4)	360			
Plumbing and Engine Room Wiring (4)	216			
Controls (4)	416			
Parasitic Load Resistor (4)	180			
Enclosure and Insulation (4)	888			

*Part of Structure
**3550 Pounds Required for Art. G and 5700 Pounds for Heat Source Room Operations
***Required for Heat Source Room Operations
****360 Pounds Added to Boom Structure Subsystem

Table 3.2-12. Radioisotope/Brayton--Battery EPS Weight Summary (8-Man) (Continued)

			Initial Launch	Orbit	Ten Years
FUEL CELLS (EMERGENCY)			607	607	607
Fuel Cells (4)		552			
Mounts and Supports		55			
ENERGY STORAGE			2,244	2,244	8,364
Batteries (4)		2040			
Mounts and Supports		204			
POWER CONDITIONING			1,256	1,256	3,422
Chargers	Core	160			
Controls	Core	80			
Inverters (4-2.5 KVA)	Core	125			
Secondary Bus	Core	320			
Mounts and Support	Core	135			
Frequency Changer (4-7.5 KVA)	Heat Source Module	378			
Mounts and Supports	Heat Source Module	38			
Primary Bus	Heat Source Module	20			
DISTRIBUTION CONTROL AND WIRING			3,551	3,551	3,667
Wiring	Core	3,000			
Buses, Contactors	Core	58			
Mounts and Supports	Core	369			
Wiring	Heat Source Module	120			
Mounts and Supports	Heat Source Module	4			
LIGHTING			598	598	2,330
Lights		544			
Mounts and Supports		54			
TOTAL (Pounds)			29,477	29,477	41,098

Table 3.2-13. Reactor/Brayton-RFC
EPS Weight Summary (8-Man)

		Initial Launch	Orbit	Ten Years
R-1 MODULE			14,261	28,522
Reactor	1,450			
Reactor Support Structure and containment	180			
Primary Loop	--			
Ducting	50			
Expansion Compensators(3)	75			
HIP (2)	300			
Insulation	6			
Intermediate Loop	--			
Ducting	500			
Expansion Compensators (4)	90			
HIP (4)	600			
Insulation	50			
IHX	60			
BHX (2)	90			
Control and Wiring	200			
Radiation shield (Solid)*	10,600			
R-2 MODULE		5,468	5,468	13,898
PCU's (3)	3,320			
PCU Controls and Auxiliaries (3 sets)	736			
Parasitic Load Resistors(2)	80			
Radiator Fin, Tube, Armor and fluid	1,208			
Radiator Manifolds	50			
HRL Auxiliaries	77			
FUEL CELLS (EMERGENCY)		607	607	607
Fuel Cells (4)	552			
Mounts and Supports	55			
ENERGY STORAGE		322	322	966
Fuel Cells (4-2KW)	290			
Electrolysis units(4)	101			
Reactants	16			
Tankage	16			

*Shields Dose Plane (204 ft) to 150 mrem/day

Table 3.2-13. Reactor/Brayton--RFC EPS Weight Summary (8-Man) (Continued)

			Initial Launch	Orbit	Ten Years
POWER CONDITIONING			744	744	2008
Controls	Core	80			
Inverters (4-5 KVA)	Core	252			
Secondary Bus	Core	300			
Mounts and Support	Core	112			
DISTRIBUTION CONTROL AND WIRING			3,741	3,741	4,013
Wiring	Core	3,000			
Buses, Contractors	Core	58			
Mounts and Supports	Core	369			
Wiring	Reactor Module	120			
Mounts and Supports	Reactor Module	24			
Secondary Bus	Reactor Module	150			
Buses, Contactors	Reactor Module	20			
LIGHTING			598	598	2,330
Lights		544			
Mounts and Supports		54			
TOTAL (pounds)			11,480	25,741	52,344

Table 3.2-14. Radioisotope/Brayton--RFC EPS Weight Summary (8-Man)

		Initial Launch	Orbit	Ten Years
HEAT SOURCE		4,898	4,898	4,898
Fuel Capsules (2)	2,290			
Heat Source Plate (2)	1,910			
Support Structure (2)	251			
Insulation (2)	447			
AEROSHELL		1,350	1,350	1,350
Heat-Shield and Bond (2)	900			
Structure (2)	315			
Fence (2)	135			
RECOVERY AIDS		360	360	360
Ballute (2)	280			
Flotation Aids (2)	40			
Location Aids (2)	40			
SHIELD		9,750	9,750	9,750
Augmentation Tank*	--			
Augmentation Water**	9,250			
Augmentation LiH(2)***	250			
HEAT REJECTION		1,519	1,519	1,777
Radiator (1380 Ft^2)	1,175			
Pump Assembly, Accumulator and Cooling System Components(4)	200			
Cold Plates (4)	144			
Emergency Cooling (2)****	--			
ENERGY CONVERSION		3,344	3,344	4,573
Brayton Rotating Unit (4)	268			
Heat Exchanger (Heat Source)(4)	588			
Cooler (4)	36			
Recuperator (4)	280			
Interconnecting Ducts (4)	112			
Gas Mdsnagement (4)	360			
Plumbing & Heat Source Module	216			
Controls (4) Wiring(4)	416			
Parasitic Load Resistor (4)	180			
Enclosure and Insulation (4)	888			

*Part of Structure
**3550 Pounds Required for Art.G and 5700 Pounds for Heat Source Module Operations
*** Required for Heat Source Module Operations
**** 360 Pounds added to boom structure subsystem

Table 3.2-14. Radioisotope/Brayton--RFC EPS Weight Summary (8-Man) (Continued)

		Initial Launch	Orbit	Ten Years
FUEL CELLS (EMERGENCY)		607	607	607
Fuel Cells (4)	552			
Mounts and Supports	55			
ENERGY STORAGE		322	323	966
Fuel Cells (4-2 KW)	290			
Electronics Units (4)	101			
Reactants	16			
Tankage	16			
POWER CONDITIONING		1096	1096	2942
Controls (Core)	80			
Inverters (4-2.5 KVA) (Core)	125			
Secondary Bus (Core)	320			
Mounts and Support (Core)	135			
Frequency Changer(4-7.5 KVA) (Engine Room)	378			
Mounts & Supports (Engine Room)	38			
Primary Bus (Engine Room)	20			
DISTRIBUTION CONTROL AND WIRING		3551	3551	3667
Wiring (Core)	3,000			
Buses,Contactors (Core)	58			
Mounts and Supports (Core)	369			
Wiring (Heat Source Module)	120			
Mounts and Supports (Heat Source Module)	4			
LIGHTING				
Lights	544			
Mounts and Supports	54			
TOTAL (pounds)		27,395	27,395	33,220

Table 3.2-15. EPS Concept Comparison Matrix (8-Man)

EPS Candidate	Weight		Cost		Technical Risk	Space Program Commonality
	Initial	10 Years	First Set	10 Years		
SA-battery	2	4	2	2	1	1
SA-RFC	1	1	1	1	2	2
NRB (battery)	5	6	4	4	3	3
RIB (battery)	6	3	6	6	4	3
NRB (RFC)	3	5	3	3	5	4
RIB (RFC)	4	2	5	5	6	4

Code: 1 = Best
2 = Very good
3 = Good
4 = Fair
5 = Poor
6 = Worst

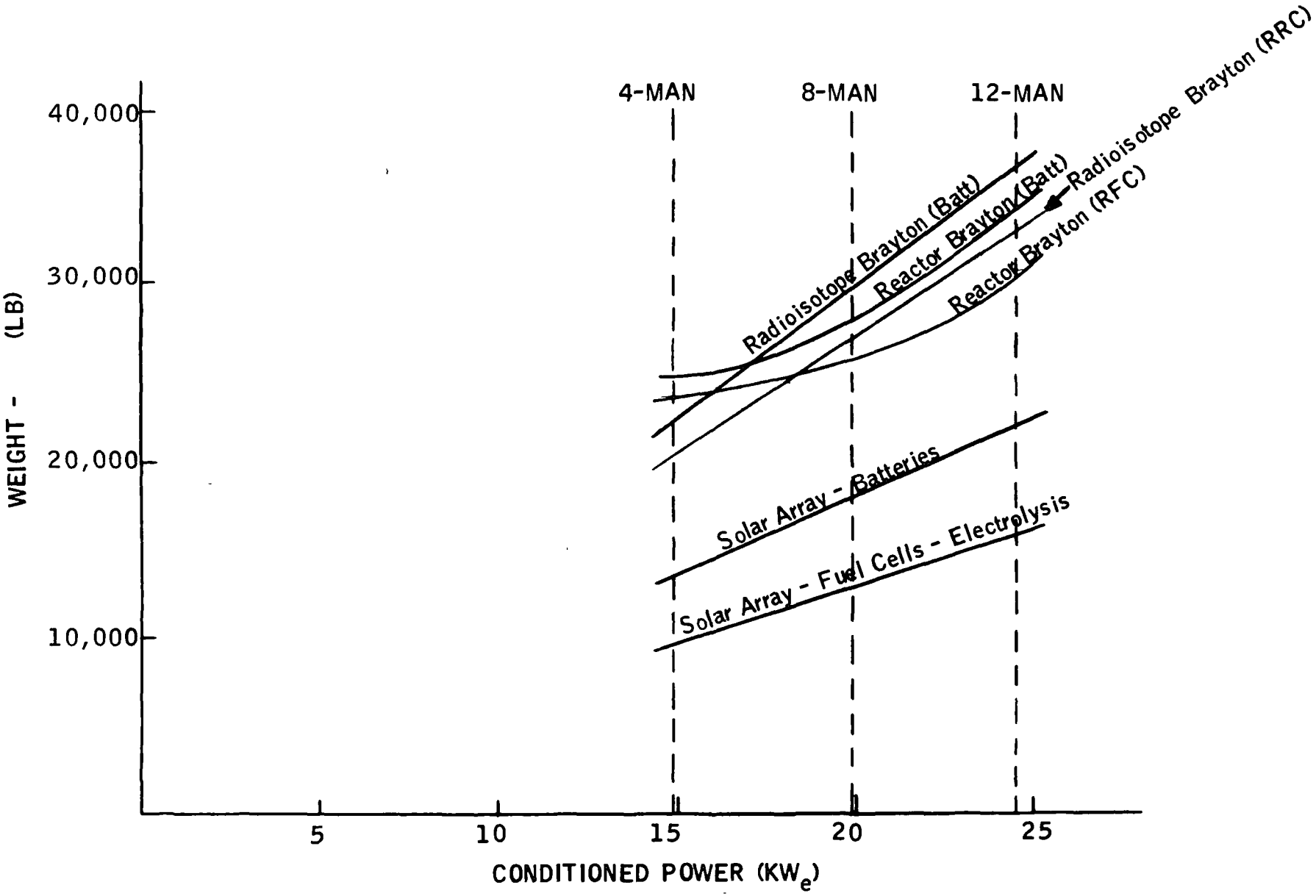


Figure 3.2-37. OLS EPS Candidate Weight Comparison

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20 KW_e AV. POWER

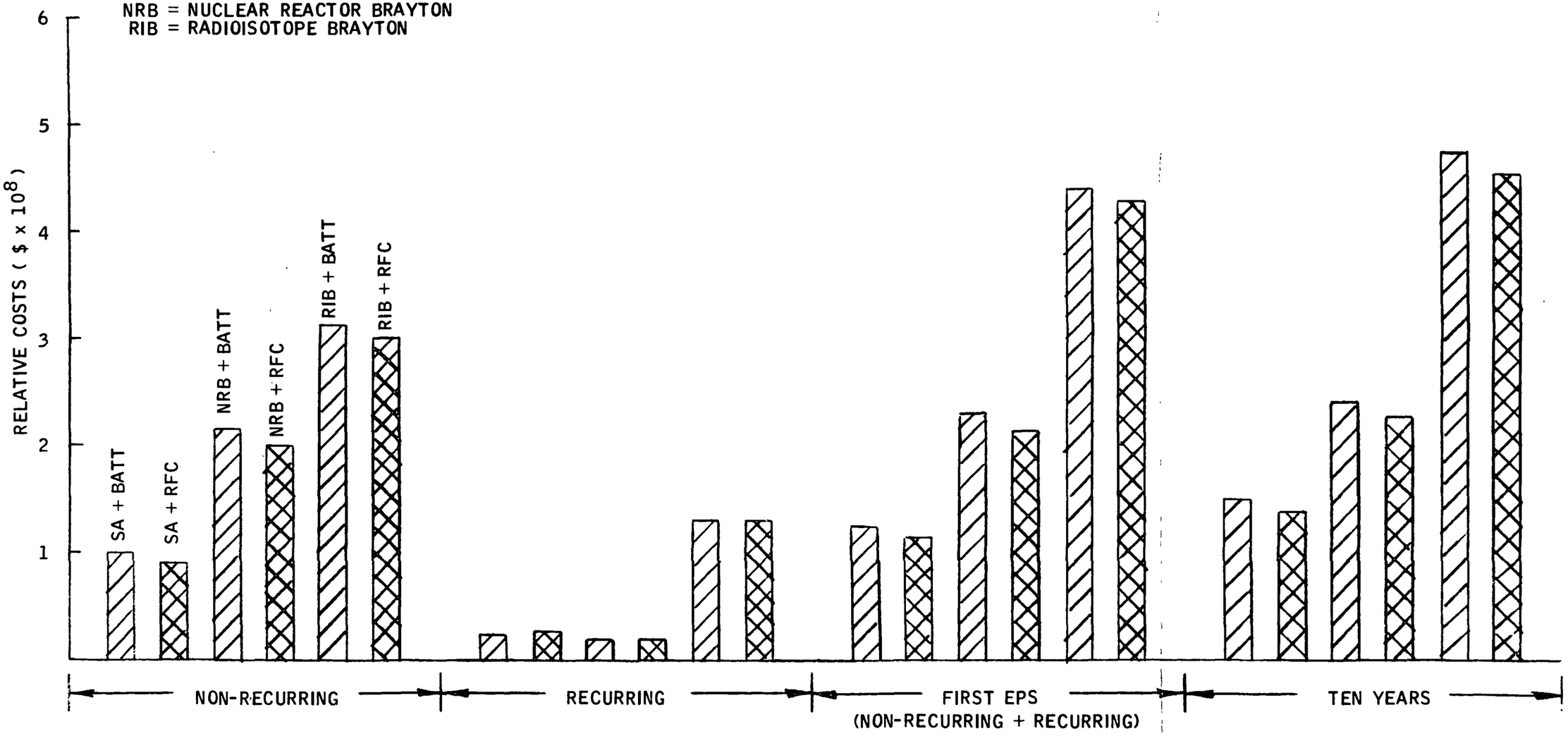


Figure 3.2-38. EPS Comparative Costs (8-Man)

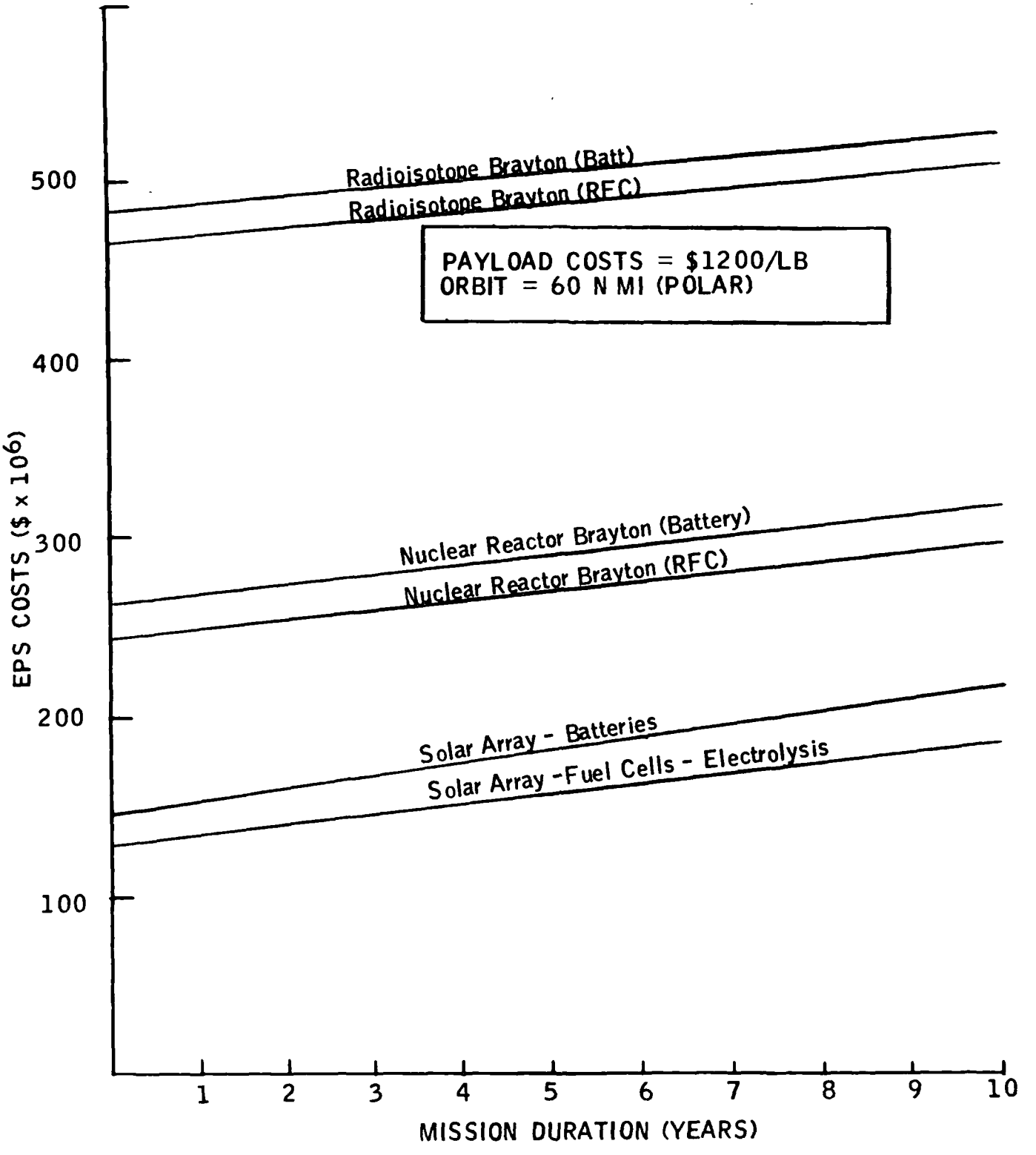


Figure 3.2-39. EPS Lunar Orbit Costs

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3.3 INFORMATION MANAGEMENT SUBSYSTEM (ISS)

The ISS provides the communications and data management capability for the OLS. Trade study data are furnished in this section for each of the four main functional areas of the ISS; data processing (paragraph 3.3.2), communications (paragraph 3.3.3), display and control (paragraph 3.3.4), and software (paragraph 3.3.5).

The majority of the data presented consists of the results of analyses conducted in the Earth Orbit Space Station (EOSS) study because of the similarity of requirements between the OLS and the EOSS in ISS performance. Even the trade data for external communications are primarily the result of EOSS studies; but an extensive analysis of the operational aspects of the two programs to verify the data applicability also is presented. Functional performance requirements and OLS/EOSS comparisons are presented in paragraph 3.3.1.

The ISS is relatively independent of modular size and configuration. Crew size affects quantities of specific assemblies, but does not affect ISS concept selection. In paragraph 3.3.6, a baseline concept is presented, and the complement of equipment as a function of crew size is indicated. Estimates of weight, power, and volume are presented.

3.3.1 FUNCTION PERFORMANCE SUMMARY

ISS functions are shown in Figure 3.3-1. They include data processing, communications tracking, and display and control. These functions are integrated using software (computer programs) and crew participation. These functions are equally applicable to an OLS or an EOSS. As indicated in subsection 7.4 in Volume III of this report, the data processing traffic for the OLS is comparable to EOSS traffic. Table 3.3-1 presents a functional comparison of EOSS and OLS requirements. All internal functions are identical, and concept trade studies are common. External communication functions are essentially the same, but the viable concept options are not because of the transmission distances and interfacing equipment. For example, EOSS use of the manned space flight network (MSFN) is extremely limited by line-of-sight constraints. The OLS is dependent upon the MSFN for communications with earth.

3.3.2 DATA PROCESSING

The total array of trades associated with selection of a data processing system are depicted in Figure 3.3-2. Also indicated are the evaluation factors that should be used in conducting each trade. Those trades that are significant in selecting a data processing concept are presented in the following paragraphs.

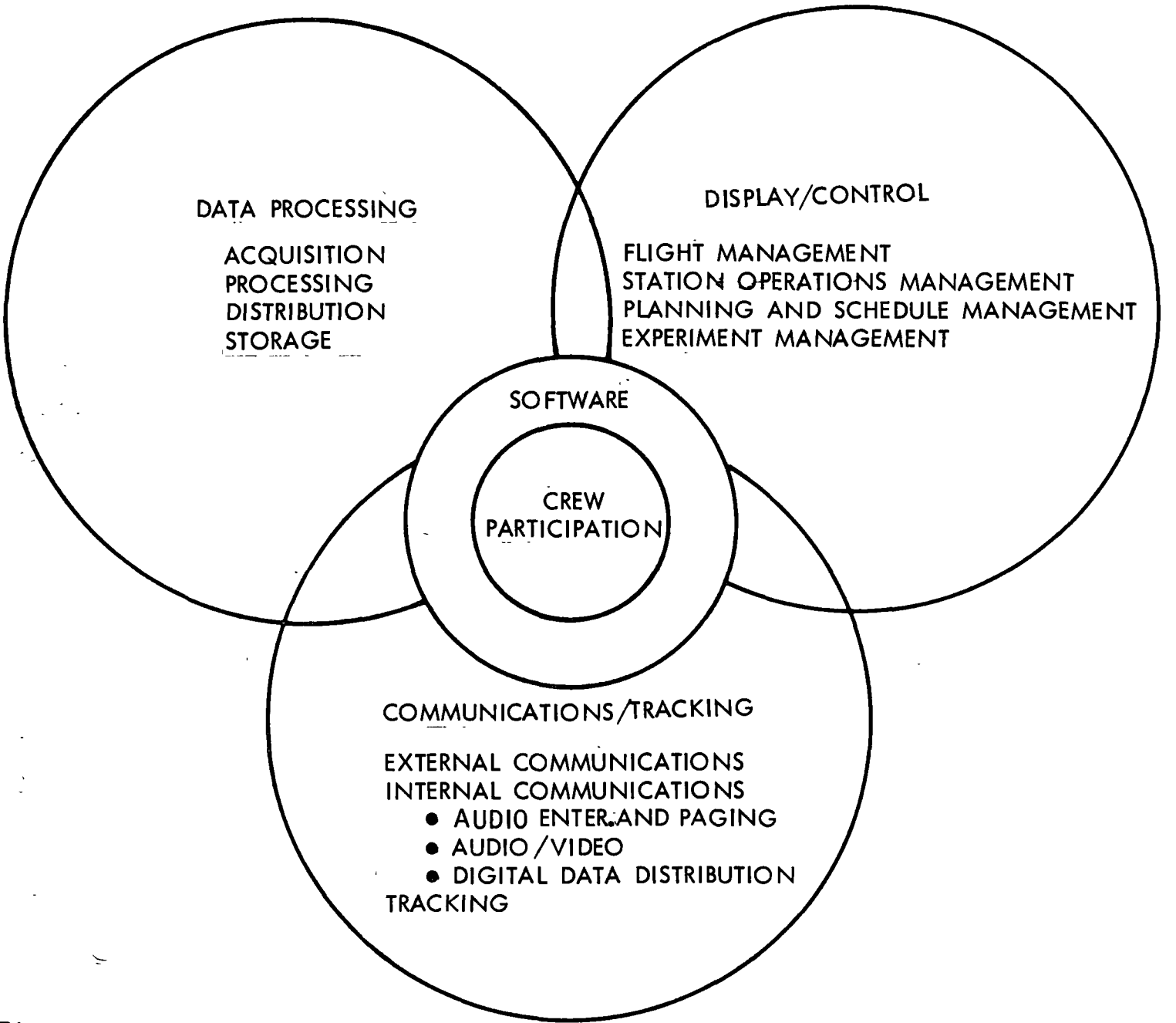


Figure 3.3-1. ISS Functional Assemblages With Software and Crew Participation Integration

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Table 3.3-1. Functional Requirements Comparison EOSS-OLS

Function	EOSS (12-Man Crew)	OLS (4-, 8-, 12-Man Crew)
Data Processing	Acquire Process Distribute Store	Identical functions. Trades at ISS level hold. Variations only if reduced capacity accepted; otherwise independent of crew or station size variations within study limits
Communications	External EOSS/ crew to ground EOSS/crew to subsatellites EOSS/crew to Earth Orbit Shuttle EOSS/crew to DRS EOSS - EVA Internal Information exchange between: Compartments Subsystems Integral experiments IVA Attached modules Control centers External communications Tracking Acquisition Traj. determination Integration Beam pointing	External OLS/crew to lunar surface OLS/crew to subsatellites OLS/crew to cislunar shuttle and space tug OLS/crew to MSFN OLS - EVA Internal Identical functions. Trades hold for variations being considered. Number of crew audio/video terminals a function of crew size and isolation possible in station configuration Tracking Identical requirements
Display and control	Flight management Station operations mgmt Planning and scheduling mgmt Experiment management	Same requirements. Reduced crew size will eliminate capability for some simultaneous operations and therefore reduce total number of displays and controls needed
Software	Computer programs required for station/ISS operation	Essentially the same. Software, a major development item for the EOSS, could be modified for OLS if the same ISS approach is used for data management.

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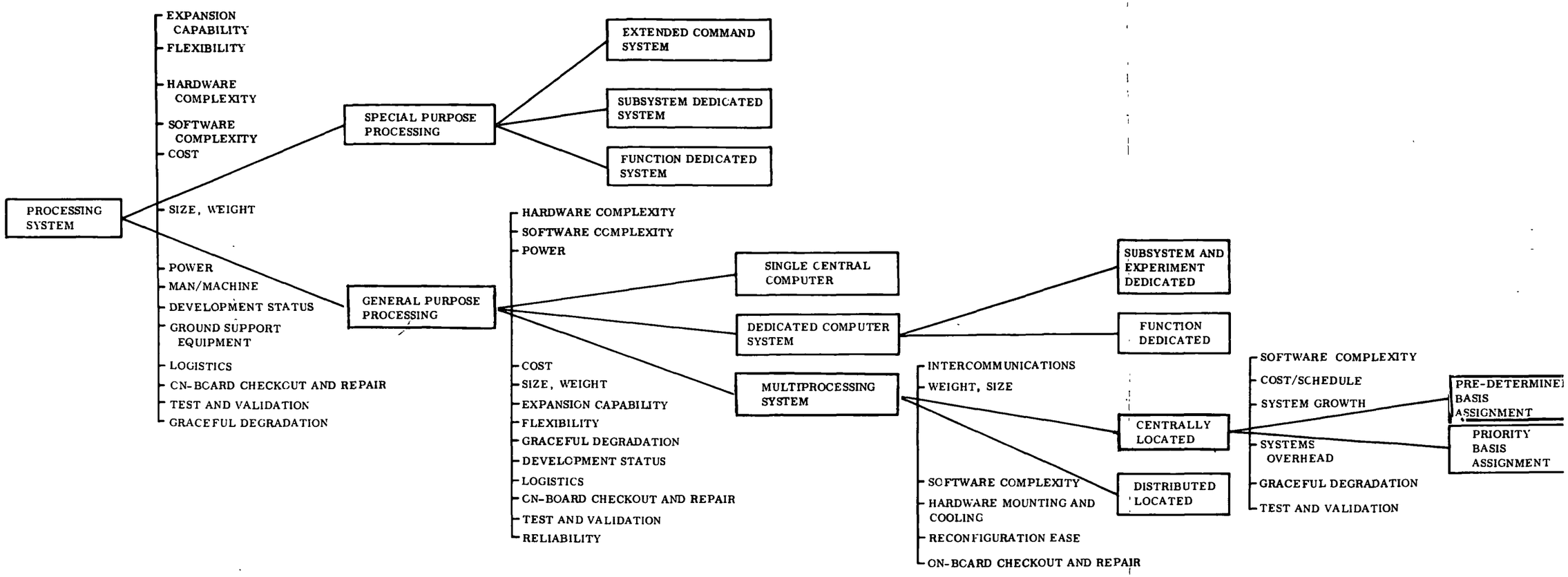


Figure 3.3-2. Processor Subassembly Trade Tree

Table 3.3-2 presents the data processing assembly options. Acquisition, processing, and storage options are considered. Special purpose and general purpose processors are compared in Table 3.3-3. A comparison between a single centralized computer, dedicated computers, and a multiprocessing system is presented in Table 3.3-4. A subtrade in computer selection is the question of centralized-versus-distributed equipment. This comparison is presented in Table 3.3-5. An evaluation of predetermined basis assignment versus priority basis assignment is presented in Table 3.3-6.

Storage Subassembly Study

Studies were performed for the storage methods that would best suit a space station's requirements. These studies were associated with the operating, mass, and archive storage.

For the purposes of this study, the following definitions apply to the classification of the three storage subassemblies (all three storage techniques must have read-and-write capability and be nonvolatile):

Operating storage - A high-speed memory that is integral with the processor from which instruction is executed, the location of programs stored, priority routines, internal fault detection routines, safety routines, temporary storage used during the data reduction and evaluation process, etc. The executive programs would be stored in this memory.

Mass storage - A high-speed memory that is not integral to the processor and is available through the input-output devices. The mass storage stores larger quantities of data, and its usual function is the store of programs and subroutines, data needed for long-term analysis and computation (trend data, guidance and navigation parameters, electrical power profiles, consumables status, subsystem configuration working plans and schedules, etc.), and it frequently calls for data normally in the archive storage.

Archive storage - A memory that stores data after it has been processed and converted to information. The archive storage information is of a historical nature and is not necessarily active at all times. Archive storage units can be removed from their read and write devices and placed in long-term containment or shipped to the ground via the logistics flights. Experiment results, subsystem performance, historical operational and schedules information, and crew status are examples of information that would be stored in this memory.

Operating and Mass Storage Analysis

An analysis of operating and mass storage media showed their requirements to be almost identical as far as access time and read-and-write capability etc., are concerned. These two storage subassemblies were, therefore, studied together, and the results are reported here as operating storage. Figure 3.3-3 shows the study options and factors used to establish a baseline for estimating physical parameters.

Table 3.3-2. Data Processing Assembly Options

Subassembly	Options Considered	Characteristics	Remarks
Processor	General purpose	Handles varying problem types	
	Dedicated	Inflexible	
	Monolithic	No expansion capability	
	Multiple	Growth, maximum redundancy for smallest cost	Can start with minimum capability
Data Storage			
Operating	Core	Not nondestruct readout; slow	
	Plated wire	Nonvolatile	
	Integrated circuit	Volatile; high power	
Mass	Plated wire	Low power	
	Tape	Too slow	
	Disc	Operating problem	
Archive	Tape		
	Film	Compact	Needs development
Data Acquisition	Hardwire to processor	Too heavy	
	Remote subprocessor	Heavy, unique	
	Local commutation/ digitizing (single data system)	Flexible	Control by processor unit

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Table 3.3-3. Trade Study Evaluation - Special-Purpose Processing Versus General-Purpose Processing

Considerations	Special-Purpose	General-Purpose
Significant advantages	1. Processor tailored for task 2. Processor immediately available 3. No software complexity	1. Processor not completely tailored to task 2. Capable of handling varying problem types 3. Adaptable to some changes in requirements 4. Computational flexibility
Significant disadvantages	1. Minimum computational flexibility 2. Not adaptable to changes in requirements 3. Not capable of handling varying problem types 4. Maximum size, weight, power because of off-line back-up redundancy 5. Not capable of growth	1. Software required 2. Less efficient than special-purpose
Criteria evaluation		
Hardware complexity	Processor tailored to 1 task, thus making hardware simple	Processor capable of handling varying tasks, thus more complex
Software complexity	No software involved	Software need for operation of system
Size, weight	Processor tailored to 1 task, thus more black boxes necessary Redundancy requires more hardware	Processor not tailored; therefore, it can handle many tasks. Few black boxes needed Redundancy requirements are less
Power	Average power is equal to peak power	Average power is less than peak power
Man-machine interface	Limited flexibility in communication with crew	Maximum flexibility in communication with crew
Development status	Simple design; does not push state of the art	More complex design; may push state-of-the-art in software area
Ground support equipment	Each black box needs special-purpose test set, both at component and system level	Each processor can use one universal test set
Logistic spares	Larger logistics problem because of many different black boxes	Little logistics problem because processors use common modules
Expansion capability	Not capable of handling any additional processing	Capable of handling additional problems with some software addition
Flexibility	Not capable of changing requirements once they are defined	Capable of handling varying problem types and adapting to requirement changes
On-board checkout and repair	Must have many checkout procedures and schematics because of many different modules. Hardware and software complexity is less	Use of common hardware modules requires less procedures, schematics, and training. Hardware and software more complex
Test and validation	Hardware and software less complex, thus easier to test on ground	Hardware and software more complex, thus more difficult to test on ground
Gradual degradation	None, unless total block redundancy is available	Processor can handle different tasks, thus total block redundancy is not necessary
Cost	Least expensive because of minimal hardware, software, and test complexity	More expensive because of complex hardware, software, and test complex

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Table 3.3-4. Multiprocesssing System

	Single Centralized Computer	Dedicated Computers	Multiprocessing System
Significant advantages	1. One computer runs the entire Space Station 2. Offers some system flexibility	1. Minimizes software complexity because of no complicated system executive program 2. Computer may be tailored to its own task 3. Because computers are tailored to only one task, they are less complex, and therefore, are easier to develop and cost less	1. Minimizes parameters of size, weight, power 2. Offers extreme flexibility in performing any tasks 3. Offers extremely large growth potential with little penalty 4. Allows system to fail gracefully
Significant disadvantages	1. Because of speed and sophisticated capability, it will be harder to develop and more expensive 2. Average power will be equal to peak power 3. Does not offer much expansion capability without adding total new computer system	1. Maximizes size, weight, because of amount of hardware needed 2. Offers no flexibility 3. Offers no growth potential without adding totally new computer system	1. Complicated system executive; maximizes software complexity 2. Switching techniques must be developed 3. Has not been used in aerospace application before
Criteria Evaluation			
Hardware complexity	Because of the high speed imposed on this system, the hardware organization will be very complex	Since each processor is tailored to one task, the processor hardware will be very simple	These processors are not as simple as the tailored processors, but are less complicated than one super computer. The switching technique chosen could add some system complexity
Software complexity	Executive program is more complex than one needed for tailored processor because all tasks must be monitored by one executive program	Executive programs are very simple, because they have to monitor only 1 task. Each processor has its own independent executive program	Each processor has its own executive program as in dedicated system. This system also requires an overall system executive to control switching and task assignment
Size, weight*	670 lb - 14 .0 cu ft	830 lb - 15.0 cu ft	500 lb - 11.0 cu ft
Power	Average power = peak power Average power - 600 watts Peak power - 600 watts	Average power only slightly less than peak power Average power - 600 watts Peak power - 725 watts	Average power much less than peak power Average power = 450 watts Peak power = 600 watts
Expansion capability	Expansion only within the limits of the one computer	Expansion only with the addition of a totally new computer	Because no processor is dedicated, additional tasks can be added with little or no hardware addition
System flexibility	Capable of minor changes in the requirements without impact. A major change would cause redesign	Any change in system requirements would necessitate redesign	Can accept major changes without any changes in the hardware
Graceful degradation	Single super computer is totally block redundant. Two failures will put system out	GN&C computer and Life Support computer are block redundant; experiment computers are not. Single failure in any experiment computer will stop experiment performance	Since no task is dedicated, task can be assigned to any computer; therefore, it will take many failures to put the system down
Development status	Because of high speed (10^7 ops/sec), this super computer is not in the development state	Because of simple design, tailored computers may then be developed and even space-qualified	Commercial multiprocessing systems exist today. These need to be converted to space packages and qualified.
Logistics	Only need one computer type to be kept in stock	Because processors are tailored, many types must be kept in stock	Processors are identical; therefore, only one type need be kept in stock
Onboard — checkout and repair	Has self-check capability. Computer speed, executive program and hardware complexity may make failures hard to detect. Repair will be very complex.	Has self-check capability. Repair will be complex because of tailored processors	Has self-check capability. Executive program, switching techniques may make failures slightly harder to detect. Repair may be slightly complex.
Test and validation	Difficult because of hardware and software complexity	Easiest to test because of least hardware and software complexity. Requires more test equipment to support testing	Most complex to test because of master system executive program and switching technique

*Includes needed on-line redundancy

Table 3.3-5. Trade Study Evaluation - Centralized Location Versus Distributed Location

Considerations	Centrally Located	Distributively Located
Significant advantages	1. If multiprocessing system is used; processors must communication with each other, this allows for better transfer of information 2. Allows for use of shared memories 3. Allows for more efficient distribution of tasks	1. Allows processor to interface directly with equipment it services. 2. Keeps processors from being all in one area 3. Favors a modular space station concept
Significant disadvantages	1. All processors are in one physical location. This may present reliability problems 2. May be difficult to achieve in a modular space station concept	1. Causes more complex system of intercommunication with processing system 2. Is not as efficient as if centrally located 3. Precludes use of shared memories
Criteria evaluation		
Intercommunication	The processors, memories and I/O units can communication with each other through a hardware switching technique. This allows for use of shared memory and increases the transfer speed	Processors, memories, and I/O units can communicate with each other, but it must be done through a data bus system. This does not allow for the use of shared memories and increases the transfer speed
Type of processing system chosen	Easily adaptable to multiprocessing system because of necessity for intercommunications within the system	Not easily adaptable to multiprocessing system because of complexity needed for intercommunications within the system
Weight and size	Lower weight and size because a shared memory system can be adapted and interconnecting cables need not be spread throughout the vehicle	Higher weight and size, because memory cannot be shared and additional cabling needed for intercommunication
Software complexity	Less complex because of simpler intercommunication routines	Most complex, because of complex intercommunication routines
Hardware mounting and cooling	Components can be mounted in one console with coldplate cooling	Components are distributed; therefore, many smaller consoles, taking up more volume, are needed. Coldplate cooling must also be spread throughout the vehicle
Reconfiguration ease	Reconfiguration of the system because of a processor failure can be accomplished within microseconds after the processor failure has been detected	Reconfiguration of the system because of a failure cannot be easily accomplished because of the difficulty in communication between processors. The system reconfiguration could take several seconds, depending on the amount of data needed to be transferred
On-board checkout and repair	Individual processor requires same checkout and repair as in distributed system. Switching slightly harder to check out	Individual processor requires same checkout and repair as in distributed system. Use of dedicated memory is slightly easier to check out

Table 3.3-6. Trade Study Evaluation - Predetermined Basis Assignment Versus Priority Basis Assignment

Considerations	Predetermined Basis Assignment	Priority Basis Assignment
Significant Feature	Available processors can be assigned to any task, but once assigned, are dedicated to that mode of operation. Predetermined, concurrent (multiprogrammed) performance of certain other compatible tasks within given modes can also be performed by a structured, operationally constrained environment	Tasks are assigned to on-line processors as a function of their ability to accommodate this processing within their unutilized (idling) computational capacity. Additional processors are brought on-line as demand requires. This environment is essentially unconstrained, except where specific tasks are at cross-purposes (e.g., simultaneously attempting to monopolize same device)
Criteria Evaluation		
Software complexity	Less executive flexibility required to support structured environment	Very complex executive required for multitask scheduling, processor assignment and control
System overhead	Individual processors will tend to operate at less than 100% of their computational capability	Multitask environment will require higher executive overhead in each processor
Gradual degradation (primarily a function of processor loss)	Degrades at somewhat faster rate due to overhead and mode constraints. Ultimately degrades to same configuration, performing mission critical tasks only at near 100% efficiency	Near-optimal degradation
Test and validation	Structured environment permits exhaustive test and validation of all modes of operation	Virtually impossible to validate due to manifold number of task combinations and permutations that must be exhaustively tested
Cost and schedule	Good visibility and control of cost and schedule. Finite, bounded test cases permit efficient resource allocation and control	Prohibitively expensive in both cost and schedule, due to vast extent and quantity of test software and formal testing that must be generated and performed
System growth	New experiments and tasks will result in the definition of new modes of operation. Flight validation will be concerned primarily with only these new modes	New tasks in this dynamic environment will require repeating a major portion of test and validation activities
Relative Merit	Good	Poor

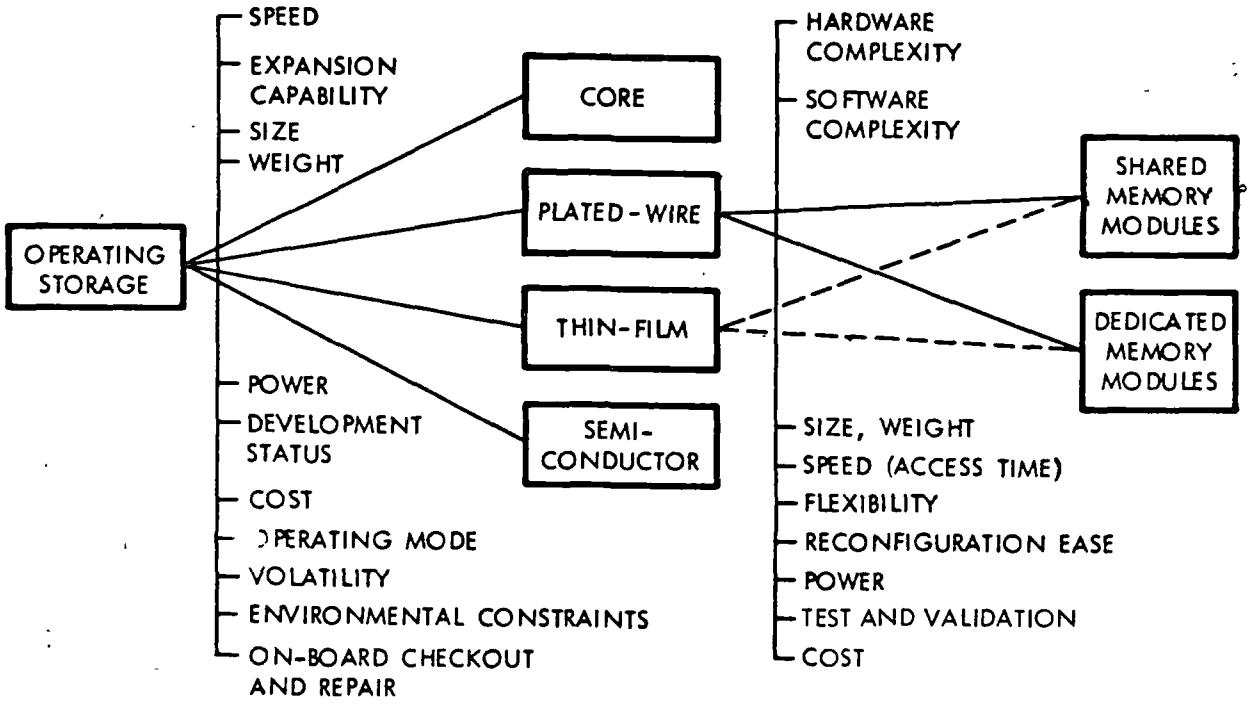


Figure 3.3-3. Operating Storage Trade Options

The results of the trade analysis are summarized in Table 3.3-7.

Plated wire was used to estimate physical parameters for the operating memory because of its excellent parameters. Plated wire is nonvolatile, NDRO with present capabilities having access times of 200 nanoseconds and power consumption of 100 to 150 watts for a 1.024 megabit module.

Semiconductor memory should be used for a processor scratchpad memory (if required) because of the significant speed increase over the magnetic memories. The power consumption per bit is high; however, a scratchpad memory consists of only 4 kilobits.

Shared Memory versus Dedicated Memory. The feasibility of using a shared-memory concept, which is defined as the capability of any processor or input-output unit to address and access any memory module within the multiprocessor system, was analyzed. The shared memory concept was a dedicated memory concept in which only one processor or input-output unit can address or have access to a particular memory. One memory bank is assigned to each processor, and only that processor can access the memory. The results of this analysis are summarized in Table 3.3-8.

Table 3.3.7. Trade Study - Operating Memory Type

Considerations	Cores	Plated-Wire	Thin-Film	Semiconductor
Significant Advantages	1. Space-qualified on other programs 2. Availability in large modules - (131K words) 3. Low cost	1. Low power, small size, and low weight 2. Operate in NDRO mode 3. Extended temperature range	1. Low power, small size, low weight 2. Fast access time	1. High speed 2. Capable of simultaneous read and write
Significant Disadvantages	1. Heavy and large 2. Slow speed 3. Can't operate in NDRO mode without size, weight and power penalties	1. Higher cost than cores 2. Not available in large modules today	1. Higher cost than cores 2. Not capable of operation in NDRO mode without penalties	1. High power 2. Still in the development state 3. Memory is volatile 4. High cost
Evaluation Criteria				
Speed (access time)	1 microsecond	200 nanoseconds	300 nanoseconds	50-100 nanoseconds
Expansion capability	Operating memories up to 131K words	Operating memories up to 32K words	Operating memories up to 32K words	Operating memories up to 8K words
Size (32K words)	1000 cu in.	600 cu in.	576 cu in.	1500-2000 cu in.
Weight (32K words)				
Power (32K words)	300-400 watts	100-150 watts	150-200 watts	500-1000 watts
Development status	Space-qualified	Space-qualified	Not space-qualified	Not space-qualified
Cost	4-6 cents/bit	8-10 cents/bit	8-10 cents/bit	20-25 cents/bit
Operating mode	DRO	NDRO	DRO	DRO
Volatility	Nonvolatile	Nonvolatile	Nonvolatile	Volatile
Environmental constraints	Not radiation resistance low operating temperature range	Radiation resistance and able to operate over a wide temperature range	Not radiation resistance, wide temperature range	Not radiation resistance, wide temperature range
Checkout and repair	Can't repair on-board, must replace module	Can't repair on-board	Can't repair on-board	Can repair on-board

Table 3.3-8. Shared Memory Versus Dedicated Memory

Considerations	Shared Memory	Dedicated Memory
Significant advantages	1. Does not require any memory-to-memory transfer of data 2. Requires less memory capacity because storage of duplicate information is not required	1. Does not require a complex switching network for memory access 2. Lessens the software complexity
Significant disadvantages	1. Requires a switching network for total shared access 2. Requires each memory unit to communicate that it may be busy 3. Requires complex software for switching network	1. Requires more size, weight, and power for additional memory capacity 2. Requires memory-to-memory transfer of data
Evaluation Criteria		
Hardware complexity	Multiport memories require a complex solid-state switching network for processor-to-memory and I/O to memory access	Requires simple intercommunication between memory modules through the respective I/O units
Software complexity	Requires the executive program to operate the switching network	Requires software to load the needed data into the memory module before starting an application program
Size and weight	Has minimum size and weight	The duplicate memory locations needed with this system would increase size and weight by 10 to 15 percent
Speed	Application program can use any data in any memory module without any time loss	Requires more time to execute an application program because of the memory-to-memory transfer of data needed to update the memory
Flexibility	System can adapt easily to change in either program size or data base size	System can't adapt to any changes which may exceed the size of the particular memory module
Reconfiguration ease	System easily reconfigured in the event of failure	Dedicated system is not easily reconfigured in the event of failure
Power	Lower memory capacity needed; therefore, power is lower	More memory capacity is needed; therefore, power is higher
Test and validation	Switching technique causes testing to become quite complex	Straightforward testing required. Only one module need be tested at a time. Memory-to-memory transfer also needs to be tested
Cost	Capacity is less but system needs a switching network. Cost approximately equal to dedicated system	Capacity 10 to 15 percent higher, but does not need a switching network. Cost approximately equal

Archive Storage Analysis

Two basic types of storage devices were analyzed: magnetic and optical. The results of this study are summarized for these two types of storage in Tables 3.3-9 and 3.3-10.

The optical storage devices appear the most attractive because of the high density they can achieve and the high transfer rate they can accommodate. The significant drawback to the optical storage devices is (1) they are in a relatively early stage of development, and (2) they offer read-only memory.

The magnetic storage devices do not offer the required packing density to accommodate the large volume of data generated by the OLS. Work on high-density magnetic tape (40,000 bits per inch) is in progress.

A development program must be undertaken in the area of archive storage to achieve the high-density packing required. Optical storage using the electron beam shows the most promise for successful development during the OLS time period.

3.3.3 COMMUNICATION/TRACKING

This section presents the analyses performed on the communication/tracking functions of the ISS. This function is subdivided into three areas: external communications, tracking, and internal communications. Figure 3.3-4 illustrates the numerous communication links of the OLS. Table 3.3-11 summarizes the options considered.

External Communications

The options for the external communication system are presented in trade tree format in Figure 3.3-5. The evaluation criteria that apply to the overall links and equipment selection as well as specific criteria that pertain to individual equipment performance evaluation are also shown in this figure.

Communication Link Analysis

The summary of the results of the analysis performed on the concept of using RF versus laser links is shown in Table 3.3-12. The subtrade of composite versus individual links is summarized in Table 3.3-12A.

RF Links

There are three major links associated with the external communications function:

Communications with earth
Communications with the lunar surface
Communications with subsatellites, logistic vehicles, space tugs, etc., via direct links

Table 3.3-9. Magnetic Type Archive Storage

Considerations	Magnetic Tape		Rotating Magnetic Media	
	Cartridge Incremental	Continuous Run	Magnetic Disc	Magnetic Drum
Significant Advantages	1. Transfer by word 2. Cartridge loading for ease and simplicity	1. Transfer by record 2. Farthest advanced in space qualifications	1. Access time is in the order of 10-20 milliseconds 2. Removable disk packs for high data quality storage	1. Access time is in the order of 10-20 milliseconds
Significant Disadvantages	1. Very slow search rate 2. Very slow transfer rate	1. Search rate is slow	1. Reliability low because of moving parts	1. Low reliability because of moving parts
Criteria Evaluation				
Capacity	$5.76X10^6$ bits per cartridge. Up to 12 cartridges on line	$2X10^9$ bits per tape. Tape 7,200 feet	$8X10^8$ bits per disc pack	10^8 bits per drum
Data density (per sq in.)	1000 bits/sq in.	$2X10^4$ bits/sq in.	10^6 bits/sq in.	10^5 bits/sq in.
Reliability (MTBF)	5,000 hours	5,000 hours	500-1000 hours if removable, 10,000 hours if non-removable	500-1000 hours if removable, 10,000 hours if non-removable
Access time (average)	10 minutes	190 seconds	8-15 milliseconds	10-20 milliseconds
Transfer rate				
Write	5400 bits/second	320 bits/second	10^6 bits/second	10^6 bits/second
Read	5400 bits/second	10^6 bits/second	10^6 bits/second	10^6 bits/second
Size and weight	Size = 0.25 cu ft Weight = 40 lb	Size = 0.60 cu ft Weight = 20 lb	Size = 0.75 cu ft Weight = 60 lb	Size = 0.75 cu ft Weight = 80 lb
Power (Average)	55 watts	44 watts	150 watts	150 watts
Development status	Tape units have been space-qualified	Tape units have been space-qualified	Units developed but have not been space-qualified. Problem with movable heads	Unit commercially available, but not space-qualified. Problem with movable heads
Cost	Least expensive	Least expensive	Slightly more expensive than tapes	Slightly more expensive than tapes

Rating: 0 - No impact
1 - Poor
2 - Medium
3 - Good

Table 3.3-10. Optical Type Archive Storage

Considerations	Laser	Electron Beam	Holographic
Significant Advantages	1. High density 2. High transfer rate	1. High density 2. High transfer rate	1. High density 2. High transfer rate
Significant Disadvantages	1. Need large-scale development program 2. Very expensive	1. Need large-scale development program 2. Very expensive	1. Need large-scale development program 2. Very expensive
Criteria Evaluation			
Capacity	$2.9X10^9$ bits per strip. Up to 400 strips per system	10^{11} bits per system	10^{12} bits per system
Data density (per sq in.)	$4X10^7$ bits/sq in.	$4X10^7$ bits/sq in.	10^8 bits/sq in.
Reliability (MTBF)	Not known today	Not known today	Not known today
Random access time (average)	200 milliseconds within strip. Five seconds strip to strip	50 microseconds within 10^4 block of data	20 microseconds within 10^4 block of data
Transfer rate			
Write	$4X10^6$ bits/second	$4X10^6$ bits/second	$4X10^6$ bits/second
Read	$4X10^6$ bits/second	$4X10^6$ bits/second	$8X10^6$ bits/second
Size and weight	Note 1	Note 1	Note 1
Power (average)	Note 1	Note 1	Note 1
Development status	Early stages of lab development	Early stages of lab development	Early stages of lab development
Cost	Expensive development program	Expensive development stages	Expensive development program

Note 1 Size, weight, power numbers not known at this time because of early stages of development.

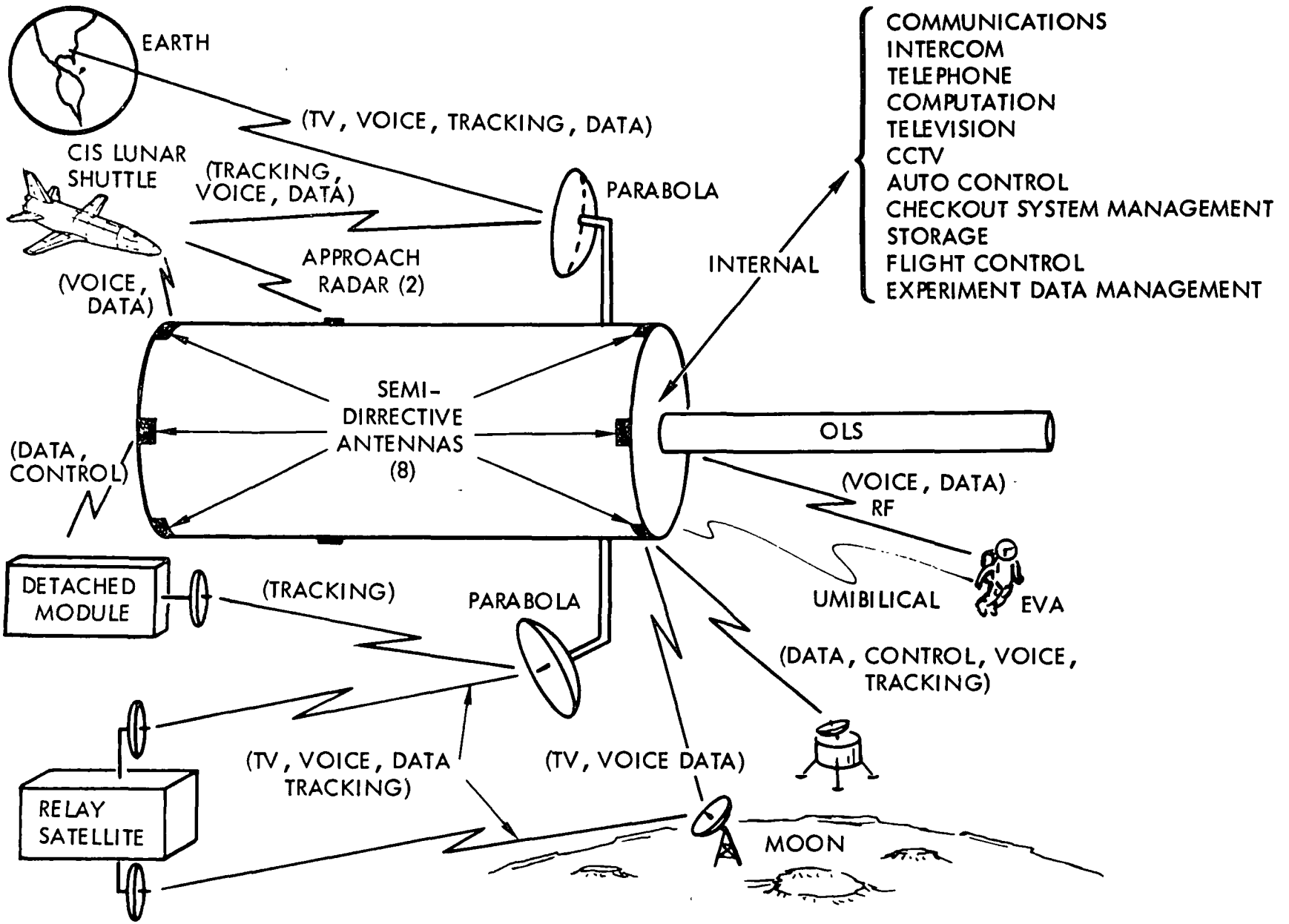


Figure 3.3-4. Communication Requirements (External and Internal)

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Table 3.3-11. Communications Assembly Options

Major Function	Options	Significant Considerations
External		
RF Communication to Earth	S-band	Common with current programs
	X-band	Lacks commonality - requires MSFN modifications
	C-band	Frequency allocation problems
	K-band	Needs development
High-Gain Antenna	Parabola	Being developed (including radar), EVA required for repair
	Phased array	Complex and costly
	Multipurpose, segmented	New concept - development required
Near-Space to Space	S-band, omnidirectional	Common with current programs
CLS, Detached Module, etc.	VHF, omnidirectional	Lacks commonality, low power required
Tracking		
Approach Radar	S-band, directive	Common with current programs
<20 n mi	X-band, directive	Development required
Internal		
Data Distribution		
Video	FDM	Realizable BW and equipment
	TDM	Excessive BW required
Digital	TDM	Data in digitized form (asynchronous timing)
	FDM	Added CONV complexities
Voice	FDM	Combined on video bus
	TDM	

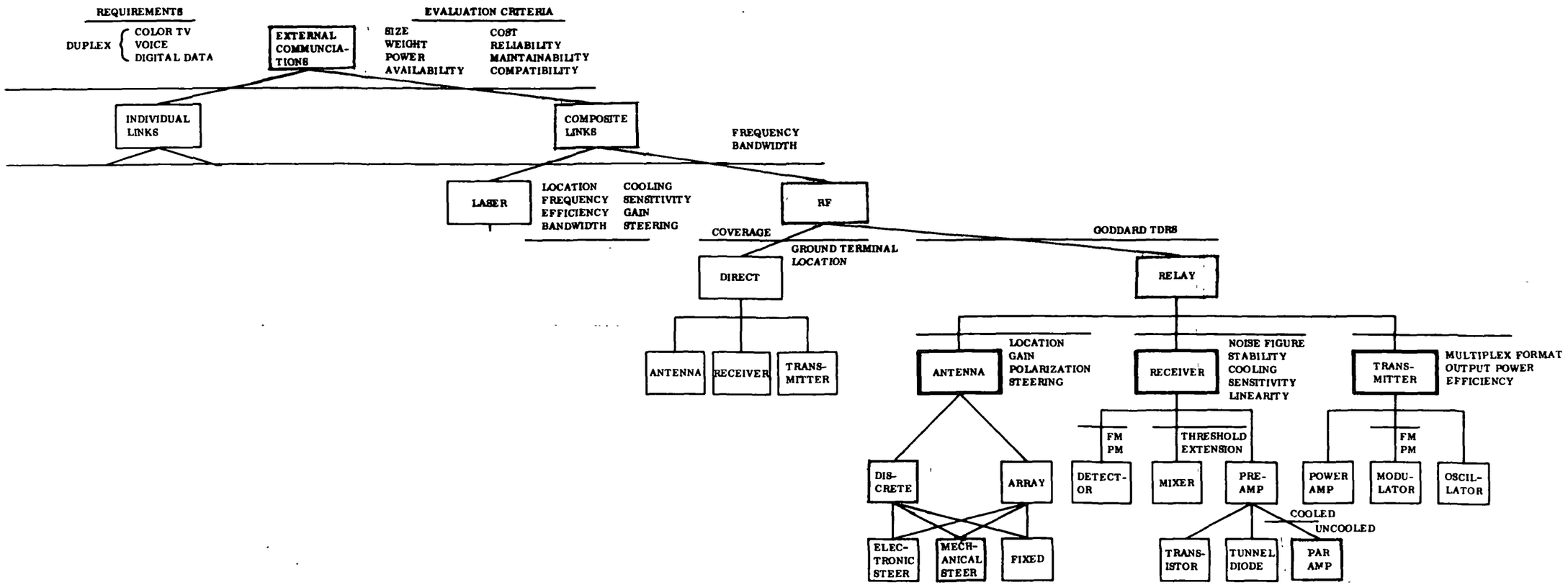


Figure 3.3-5. External Communication Trade

Table 3.3-12. RF and Laser Link Frequency Qualitative Analyses

Consideration	Radio Frequency	Laser
Advantages	Fewer ground stations are needed for orbital coverage. Tracking requirements are less stringent. Existing technology. Has high reliability. Has longer lifetime.	Ground station is less expensive. Antenna gain is higher. Uses uncrowded frequency spectrum.
Disadvantages	Antenna size limitations. Uses crowded frequency spectrum.	Technology is emerging. High prime power is required. Detectors have poor sensitivity. Bandwidth is limited by doppler. Stringent tracking is required. Has low reliability.

Table 3.3-12a. Composite Communication Link Analysis

Consideration	Composite Links	Individual Links
Advantages	Conserves RF bandwidth. Conserves prime power. Minimizes equipment. Conserves RF power. Can accommodate most tracking functions.	Equipment failure impacts only one communications function. Interference and cross-talk problems are fewer. Equipment is simpler.
Disadvantages	Equipment is complicated. Equipment failure affects a number of communications functions. Cross-talk and interference reduction require careful design and production.	Requires many RF carrier assignments. Requires much equipment. Requires sufficient RF power output to establish threshold for many links.

In each link, the information to be transferred, which includes television, voice, and digital data, is generally the same. The quantity and quality of the information depends upon the requirements of the individual categories, and the specific mission functions and options, accordingly the worst-case information transfer loads, are analyzed.

The OLS link differs from the EOSS link in path loss and in the capability of the terminal with which the station is communicating. Path losses, or the loss in signal strength with distance, for the various links are shown in Figures 3.3-6 and 3.3-7. The terminal capability may be identified by a ratio of the power gain of the terminal antenna divided by the noise power density of the receiving system. This is normally identified as G/T, G representing the gain of the terminal and T representing noise power density.

Potential MSFN terminals include a 30-foot-diameter antenna and an 85-foot-diameter antenna.

Communication links and normalized RF path losses for EOSS program elements and for lunar exploration program elements are shown in Figures 3.3-6 and 3.3-7, respectively. A comparison of these links will indicate the applicability of EOSS trade data as well as delta trades required.

Communication with Earth. In the case of the EOSS, the main path for earth communications is through a geosynchronous data relay satellite to MSFN. The OLS link is directly to MSFN. The primary concern, therefore, for the EOSS is its interface with the data relay satellite. Tables 3.3-13 and 3.3-14 summarize the receiving and transmitting characteristics of a NASA-Goddard proposed tracking and data relay satellite (TDRS). Also included in the tables are the performance characteristics of the links and the impact on the EOSS antenna system.

Review of the data indicates that a 30-watt output with a 13-foot antenna may be used in either the FM or PSK mode via the TDRS.

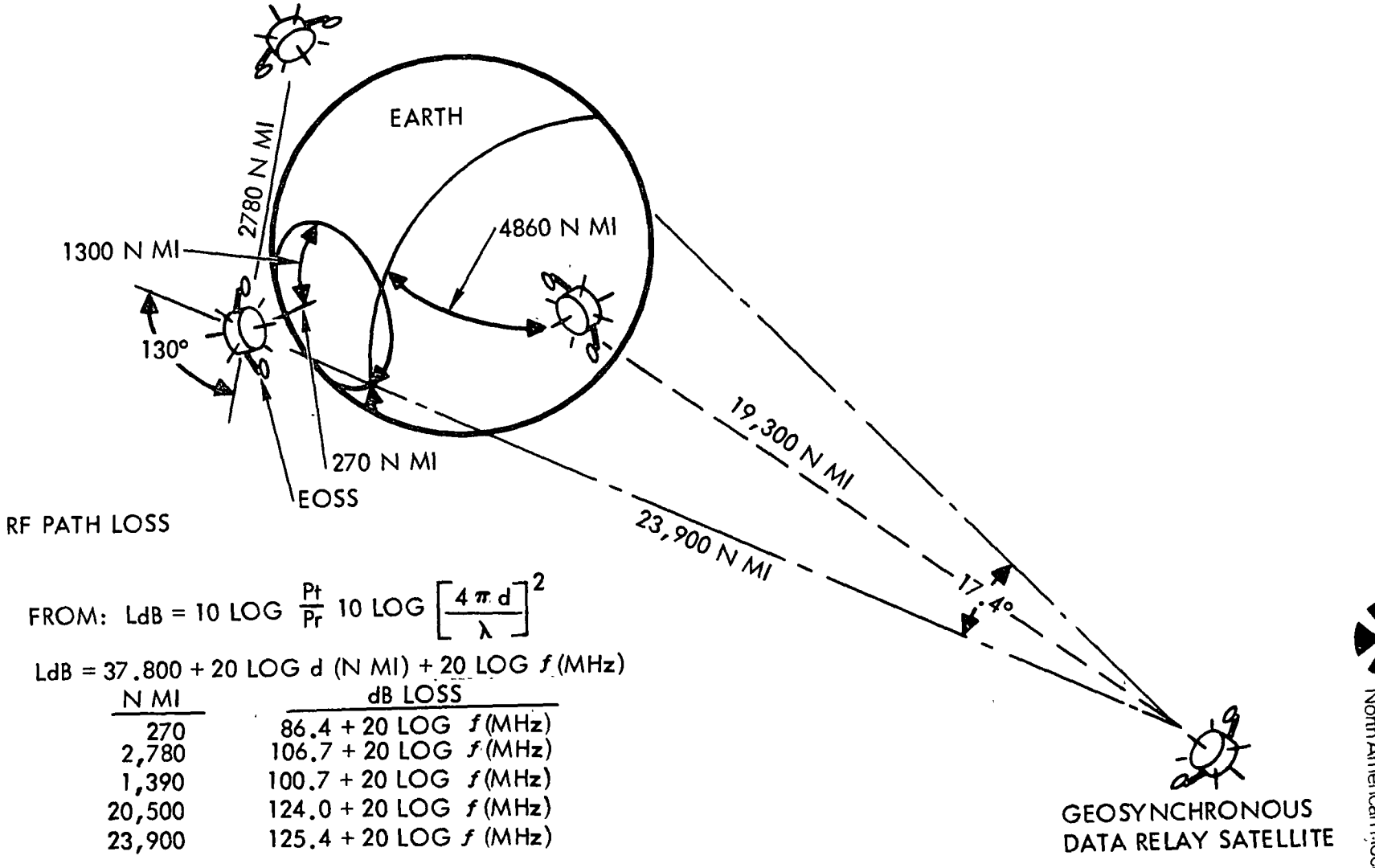
A comparison of the links indicates the following:

EOSS to DRSS Path Loss = 125.4 db + 20 log F

OLS to MSFN Path Loss = 144.2 db + 20 log F

The apparent net degradation for the OLS case is 18.8 db. However, the TDRS G/T was 0.0 db; the MSFN G/T (30-foot antenna) is +19.6 db. Therefore, the resultant for the OLS case is a +0.8 db in performance (If the 85-foot antenna were used, the resultant improvement would be 9.8 db). Thus, the EOSS trade study data associated with the earth link is equally applicable to the OLS-earth link.

To achieve OLS performance just equal to EOSS performance, the OLS antenna diameter could be reduced from 13 feet to 12 feet or transponder bus



RF PATH LOSS

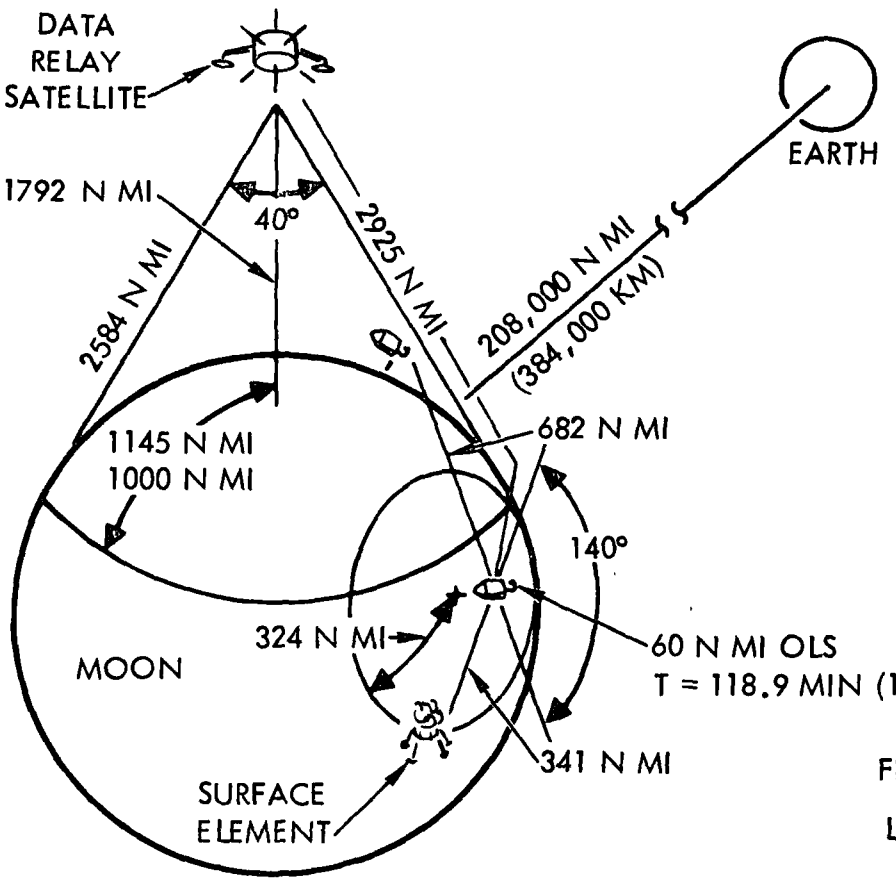
FROM: $LdB = 10 \text{ LOG } \frac{Pt}{Pr} \; 10 \text{ LOG } \left[\frac{4\pi d}{\lambda}\right]^2$

$LdB = 37.800 + 20 \text{ LOG } d \text{ (N MI)} + 20 \text{ LOG } f \text{ (MHz)}$

N MI	dB LOSS
270	86.4 + 20 LOG f (MHz)
2,780	106.7 + 20 LOG f (MHz)
1,390	100.7 + 20 LOG f (MHz)
20,500	124.0 + 20 LOG f (MHz)
23,900	125.4 + 20 LOG f (MHz)

Figure 3.3-6. Link Geometry - Earth Elements

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RF PATH LOSS

FROM: $LdB = 10 \text{ LOG } \frac{Pt}{Pr} = 10 \text{ LOG } \left[\frac{4\pi d}{\lambda}\right]^2$

$LdB = 37.800 + 20 \text{ LOG } d \text{ (N MI)} + 20 \text{ LOG } f \text{ (MHz)}$

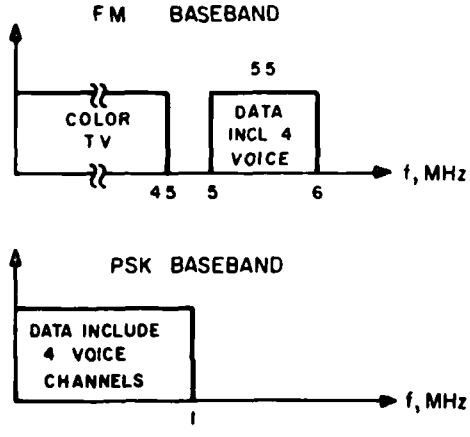
N MI	dB LOSS
60	73.4 + 20 LOG f (MHZ)
341	88.5 +
682	94.5 +
2925	107.1 +
208000	144.2 + 20 LOG f (MHZ)

Figure 3.3-7. Link Geometry - Lunar Elements

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Table 3.3-13. EOSS Receiver Noise Figure for FM and PSK Links from Relay Satellite

OLS Antenna Diameter (feet)	Frequency (GHz)	Noise Figure db (°K)
6.5	1.8	1 (30)
13	1.8	3.2 (320)
28	1.8	7.9 (1480)

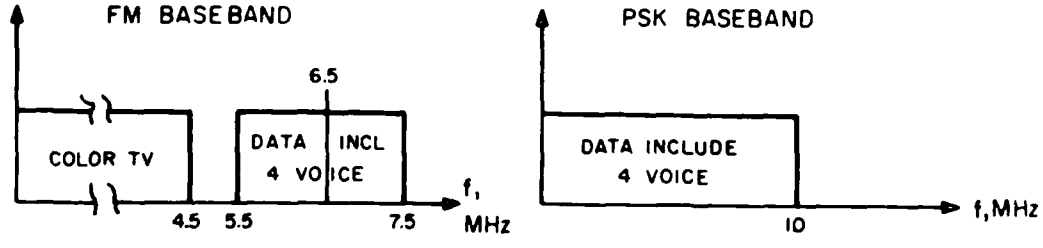


Link Characteristics

Parameter	FM Link	PSK Link
Composite baseband (MHz)	6.0	1.0
Color TV (MHz)	4.5	-
Digital voice, 4 channels (kbps)		
PCM data (Mbps)	0.88	0.88
Data bit-error rate	10^{-6}	10^{-6}
RF bandwidth (MHz)	40.0	1.0
Slant range (n mi)	21000	21000
Relay satellite EIRP	+39.5	+39.5
Received CNR (db)	+6.6	-
Television SNR (db)	+40.9	-
Received E/N_o (db)	-	+22.6

Table 3.3-14. EOSS Transmitter Power Output for FM and PSK Links to Relay Satellite

OLS Antenna Diameter (feet)	Frequency (GHz)	FM Transmitter Output Power (watts)	PSK Transmitter Output Power (watts)
6.5	2.25	120	120
13	2.25	30	30
28	2.25	6.5	6.5



TDRS LINKS

Link Characteristics

Parameter	FM Link	PSK Link
Composite baseband (MHz)	7.5	10.0
Color TV (MHz)	4.5	-
Digital voice, 4-channel (kbps)	120	120
PCM data (Mbps)	1.88	9.88
Data bit-error-rate	10^{-5}	10^{-5}
RF bandwidth (MHz)	40.0	10.0
Relay satellite G/T (db)	0.0	0.0
Slant range (n mi)	21,000	21,000
Relay satellite CNR (db)	+7.0	-
TV SNR on ground (db)	+37.8	-
Relay satellite E/N_o (db)	-	+13.0

power reduced from 120 to 100 watts (30-foot MSFN antenna). If the interface is with the 85-foot MSFN antenna, the OLS antenna could be reduced to 5 feet or the transponder bus power reduced to 12 watts.

Communications directly with the surface (Tables 3.3-15 and 3.3-16) present direct communication characteristics between the EOSS and the earth's surface. Because the same type of data must be transmitted between the OLS and the lunar surface, path loss comparisons are made. From Figures 3.3-6 and 3.3-7, the path losses are:

EOSS to earth surface 100.7 db + 20 log F

OLS to lunar surface 88.5 db + 20 log F

There is a net decrease in OLS path loss of 12.2 db. For comparable performance between the OLS and EOSS links, the lunar surface antenna equivalent of the 85-foot MSFN antenna would be 21 feet. Likewise, a 7-foot lunar surface antenna would be equivalent to the 30-foot MSFN antenna.

Surface RF power at the antenna for composite color TV, voice, and wideband data to an OLS omni antenna would be: 10 watts for a 21-foot antenna; 40 watts for a 7-foot antenna.

These data represent maximum range, bandwidth, and 0 db OLS antenna gain; that is, worst-case conditions. The assumptions for lunar surface equipment are within the range of values being used on the lunar lander tug. Restriction to voice and wide-band data would further reduce the surface antenna from 7 feet to 22 feet. Comparison of the comparable EOSS and OLS links indicate that EOSS trade data are applicable to the OLS.

Communications with Subsatellites, Logistics Vehicles, Space Tug, etc. There are two classes of near-space links associated with OLS operations. The first class considers coorbiting subsatellites, tug rendezvous operations, and cislunar shuttle rendezvous operations. Tables 3.3-17 and 3.3-18 indicate performance characteristics for a nominal slant range of 250 nautical miles, which are well within the characteristics and trade data identified for OLS external communication links discussed previously.

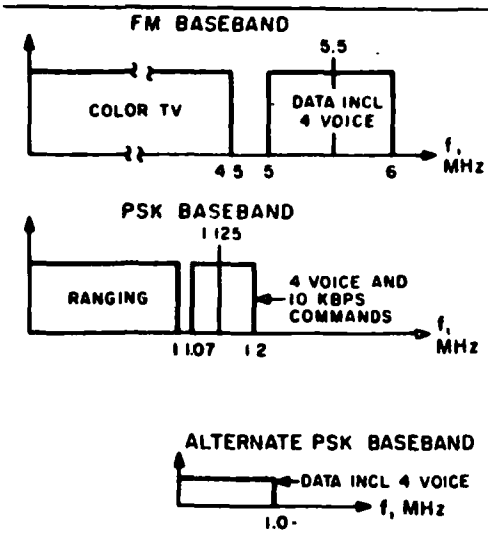
The second class of near-space links is associated with a lunar data relay satellite system (LDRS). This link could be even more significant than the OLS-MSFN link; for example, during lunar far side surface operations. For purposes of this analysis, a satellite system has been selected that will consist of three equally spaced equatorial orbiters at 1792 nautical miles altitude (see Section 2.0 of Volume II of this report). The LDRS-OLS data transmittal requirements are comparable to the OLS-MSFN requirements which, in turn, are comparable to the EOSS-TDRS requirements of Tables 3.3-13 and 3.3-14. From Figure 3.3-7, the RF path loss is 107.1 db + 20 log F. If LDRS performance is comparable to TDRS performance (namely, G/T = 0.0 db), the OLS to LDRS link comparison with the EOSS to TDRS link is:

EOSS to TDRS Path Loss = 125.4 db + 20 log F
OLS to LDRS Path Loss = 107.1 db + 20 log F

Table 3.3-15. Ground Transmitter Power Output for FM and PSK Links Direct to EOSS

Ground Antenna Diameter (feet)	Transmitter Output Power (Watts)	
	FM	PSK
85 (MSFN)	10	1
30 (PI)	40	5

(PI: Principal Investigator Station)



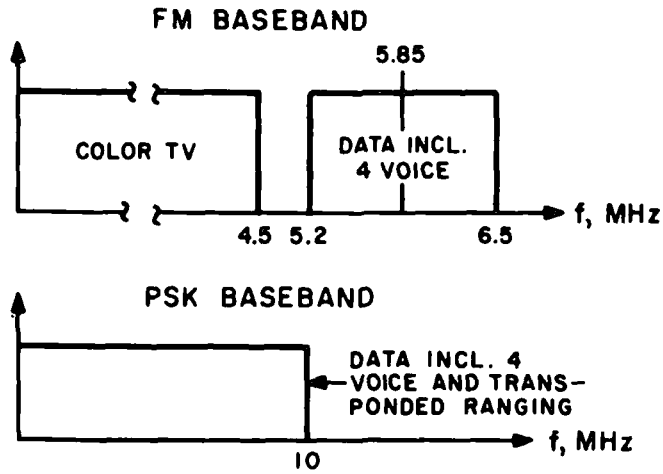
Link Characteristics

Parameter	FM Link	PSK Data Link	PSK Ranging Link
Composite baseband (MHz)	6.0	1.0	1.2
Color TV (MHz)	4.5	-	-
Digital voice, 4 channels (kbps)	120	120	120
PCM data (Mbps)	0.88	0.88	0.01
Data bit-error-rate	10^{-6}	10^{-6}	10^{-6}
RF bandwidth (MHz)	36.0	1.0	4.0
RF carrier (GHz)	2.1	2.1	2.1
Slant range (n mi)	1085	1085	1085
Station antenna	Omni	Omni	Omni
Station receiver noise temperature (°K)	460	460	460
Received CNR (from MSFN) (dB)	+17.5	-	-
Received CNR (from PI) (dB)	+14.5	-	-
Received E/N_o (from MSFN) (dB)	-	+23.1	+30.2
Received E/N_o (from PI) (dB)	-	+21.1	-

Table 3.3-16. EOSS TR Transmitter Power Output for FM and PSK Links Direct to Ground

Ground Antenna Diameter (feet)	Transmitter Output Power (watts)	
	FM	PSK
85 (MSFN)	1	1
30 (PI)	20	20

(PI: Principal Investigator Station)

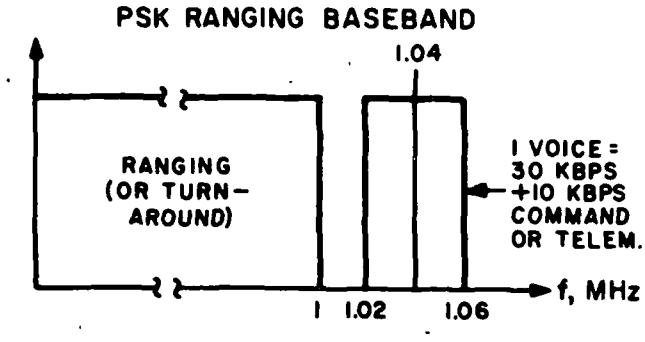


Link Characteristics

Parameter	FM Link	PSK Link
Composite baseband (MHz)	6.5	10.0
Color TV (MHz)	4.5	-
Digital voice, 4 channel (kbps)	120	120
PCM data (Mbps)	1.18	9.88
Data bit-error-rate	10^{-5}	10^{-5}
RF bandwidth (MHz)	36.0	10.0
RF carrier (GHz)	2.3	2.3
Slant range (n mi)	1085	1085
Station antenna	Omni	Omni
MSFN receiver noise temp (°K)	125	125
PI receiver noise temp (°K)	275	275
MSFN receiver CNR (db)	+13.3	-
PI receiver CNR (db)	+13.9	-
MSFN receiver E/N_o (db)	-	+18.9
PI receiver E/N_o (db)	-	+19.5

Table 3.3-17. Transmitter Power Output for Ranging Links Between OLS and Logistics Vehicles, Space Tugs, and Subsatellites

OLS Antenna Diameter	Transmitter Output Power (watts)
1 ft	4
2 ft	1
3 ft	0.45
4 ft	0.25

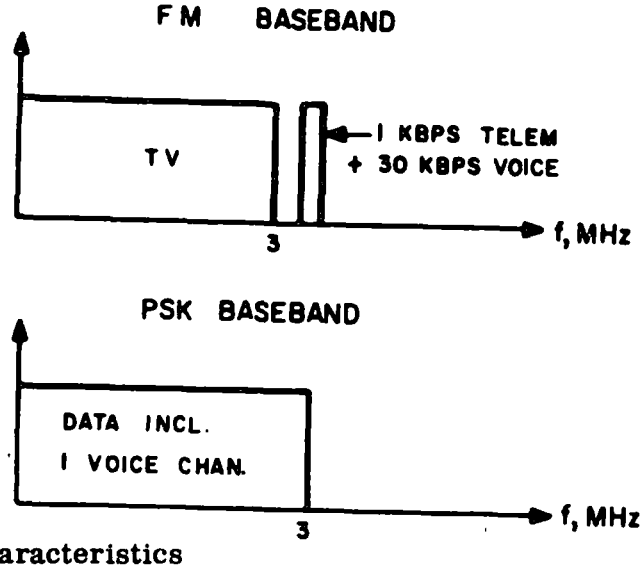


Link Characteristics

Parameter	PSK Ranging Link
Composite baseband (MHz)	1.06
Ranging (Mbps)	1.0
Digital voice, 1 channel (kbps)	30.0
PCM data (kbps)	10.0
Data bit-error-rate	10^{-6}
RF bandwidth (MHz)	4.0
Slant range (n mi)	250
Vehicle antenna	Omni
Receiver noise temperature (°K)	290
Receiver E/N_o, (dB)	+19

Table 3.3-18. Logistic Vehicles, Space Tugs, and Subsatellites Transmitter Power Output for FM and PSK Links to OLS

OLS Antenna Diameter	Transmitter Output Power (watts)	
	FM Link	PSK Link
1 ft	320	200
2 ft	80	50
3 ft	35	22
4 ft	20	12.5



Link Characteristics

Parameter	FM Link	PSK Link
Composite baseband (MHz)	3.05	3.0
TV (MHz)	3.0	-
Digital voice (kbps)	30.0	30.0
PCM data (Mbps)	.001	2.97
Data bit-error-rate	10^{-5}	10^{-7}
RF bandwidth (MHz)	15	3.0
Slant range (n mi)	250	250
Sub satellite antenna	Omni	Omni
OLS receiver noise temperature (^{o}K)	290	290
OLS receiver CNR (db)	+12.9	-
Television SNR (db)	+40.8	-
OLS receiver E/N_o (db)	-	+17.9

or a net gain of 18.3 db. It should be noted that this assumes a 13-foot antenna on the OLS. For comparable performance between the two links, the OLS antenna or power could be reduced significantly; but these factors do not directly affect the trade data. Thus, EOSS trade data are applicable to the OLS.

Antenna Analysis. Basic sizing factors for the high-gain antennas may be expressed in terms of a standard parabolic dish. The characteristics of these antenna types are well documented and are presented in this document only to provide a more complete data package. Phased arrays can also be expressed in terms of equivalent parabolic sizes.

Antenna Size and Power

A minimum antenna size is determined by the receiver characteristics and the signal to be received from a given transmitting station, assuming that the RF power to the antenna is not a limiting factor.

The power requirement range for color TV is shown in Figure 3.3-8.

Weight as a function of antenna size is given in Figure 3.3-9. The curves show weight of the antenna and feed and of the gimbals and drive. The structure required to support (and deploy, if required) the antennas is a function of the station configuration and is not included. Values are based on NASA Technical Report TR R-309, dated April, 1969.

Preamplifier performance with a resultant receiving system temperature has a major effect on the antenna requirement. Table 3.3-19 lists the major choices for the preamplifier design. The effect of system temperature variations may be obtained from:

$$\Delta \text{ db requirement} = 10 \log \frac{T \text{ (new)}}{T \text{ (old)}}$$

Path loss varies with range and frequency. Loss, including the frequency factor, is shown in Figure 3.3-10 as a function of range.

Antenna characteristics are shown in Figure 3.3-11 for a standard parabolic design. Gain is indicated as a function of diameter in feet and frequency. The beamwidth of the antenna is indicated as a function of gain. The angle including the 1-db down (from maximum) is indicated as well as the conventional 3-db down angle.

Accuracy of pointing becomes a factor in narrow-beam antennas or whenever small losses are critical. Figure 3.3-12 indicates the loss from maximum gain of a parabolic antenna as a function of the pointing error expressed as a percentage of the total antenna 3-db beamwidth.

Antenna Location. As a baseline for determining the coverage of a directional antenna in a given location, blockage may be considered to occur at a point where a cylinder the size of the antenna strikes the intercepting structure or where the main beam comes within 5 degrees of the structure. Examples are given in Figure 3.3-13. These examples are only estimates because antenna performance is dependent upon the actual design of the antenna and blocking structure.

As a minimum, the OLS could utilize a single high-gain antenna. The communications requiring a wide-band link with earth would then require a restriction of the OLS attitude to prevent intervention of OLS structure with the antenna beam. Also, long-range angle tracking or exchange of wide-band data with distant spacecraft would not be conducted simultaneously.

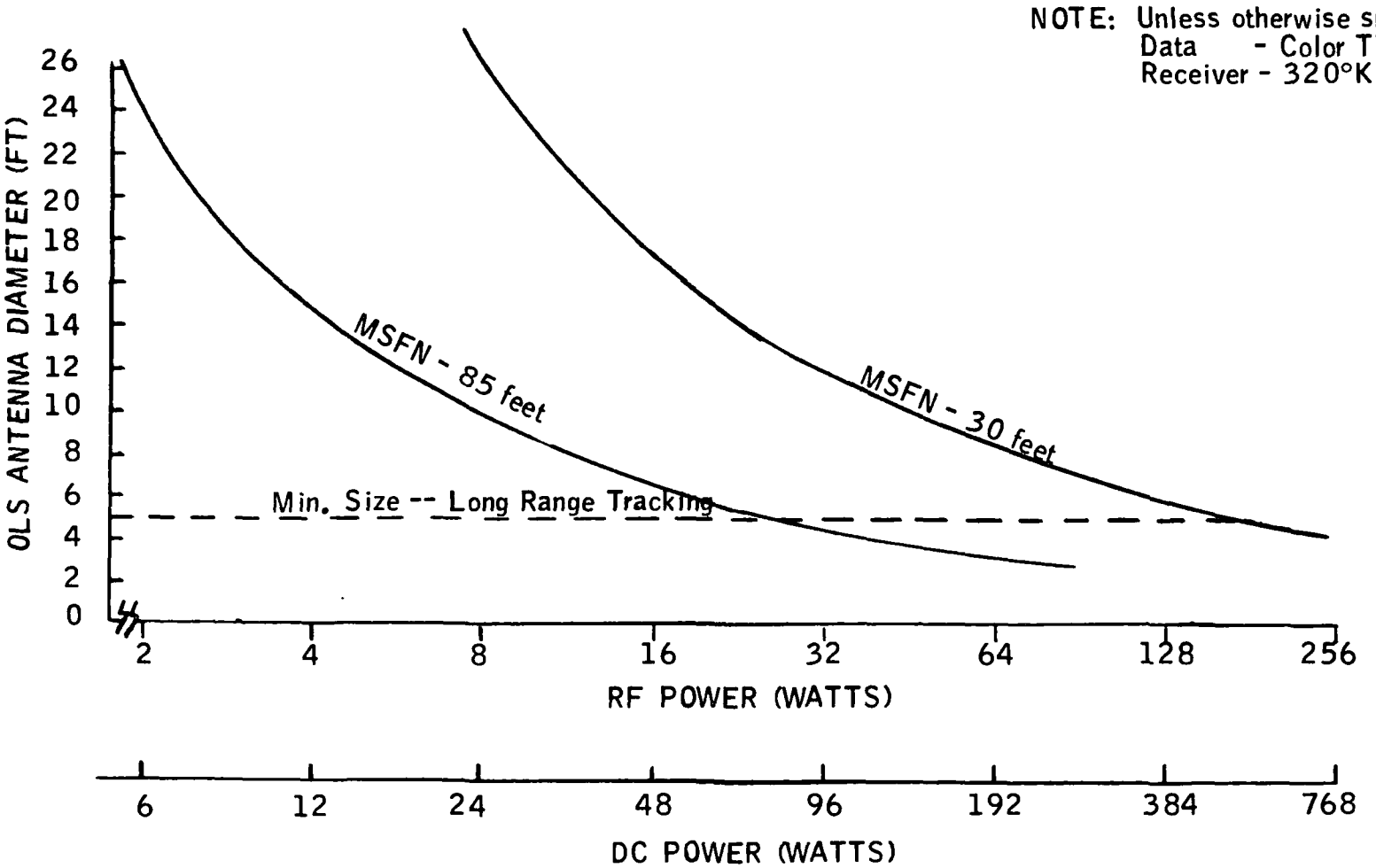


Figure 3.3-8. OLS-MSFN/EOSS Link Characteristics

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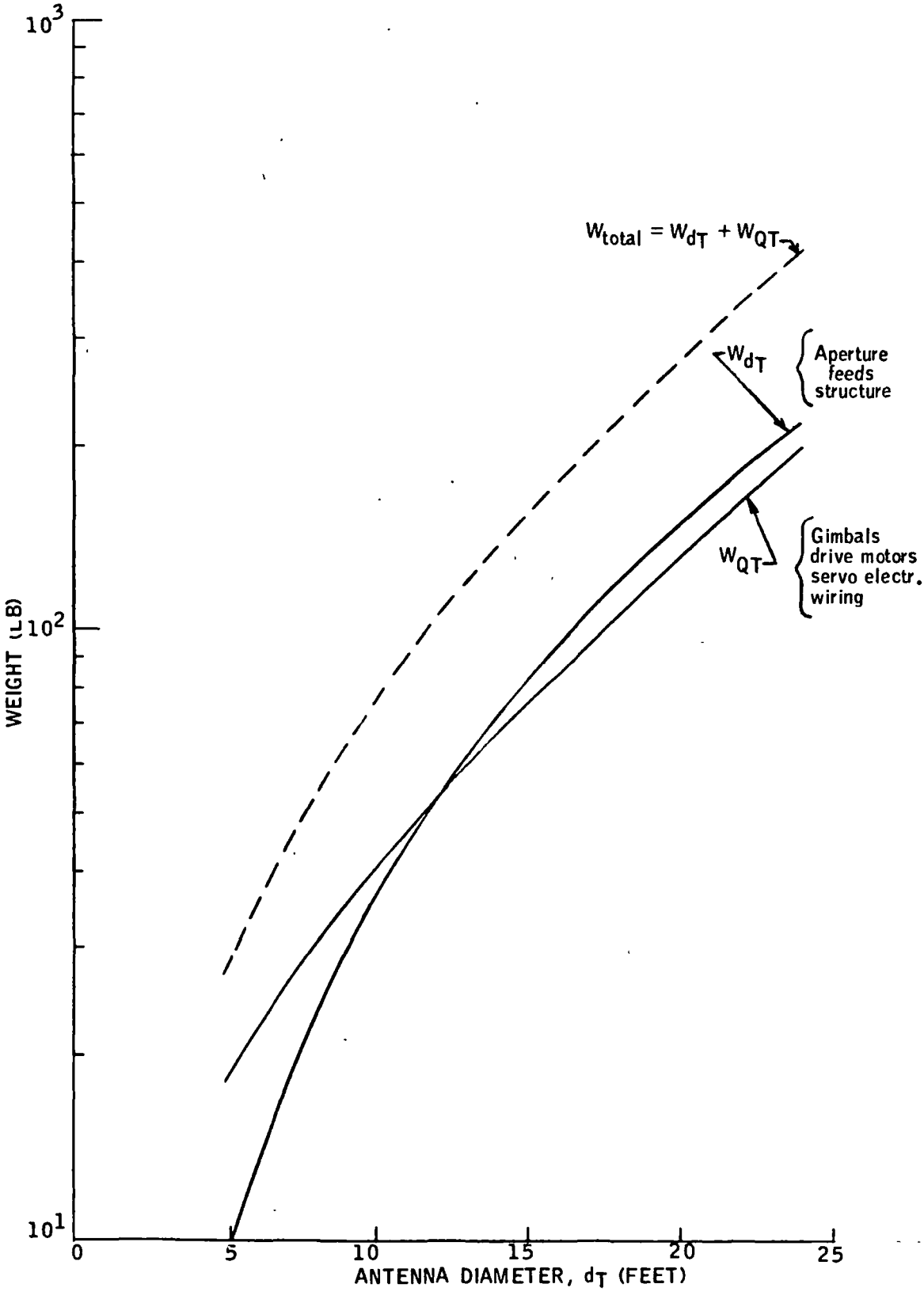


Figure 3.3-9. Antenna Subsystem Weight Versus Antenna Diameter

Table 3.3-19. Long Link Preamplifier Qualitative Tradeoff

Consideration	Uncooled Parametric Amplifier	Cooled Parametric Amplifier	Tunnel Diode	Transistor
Advantages	1-decibel noise figure Existing technology	0.2-decibel noise figure Permits 7-decibel reduction of antenna gain and/or transmitter power over uncooled parametric amplifier	2-decibel noise figure Existing technology	3-decibel noise figure Existing technology
Disadvantages	Not yet space qualified	Requires development for space qualification Cooling development required	Requires 3.5 decibels more antenna gain and/or transmitter power over uncooled parametric amplifier	Requires 6 decibels more antenna gain and/or transmitter power over uncooled parametric amplifier

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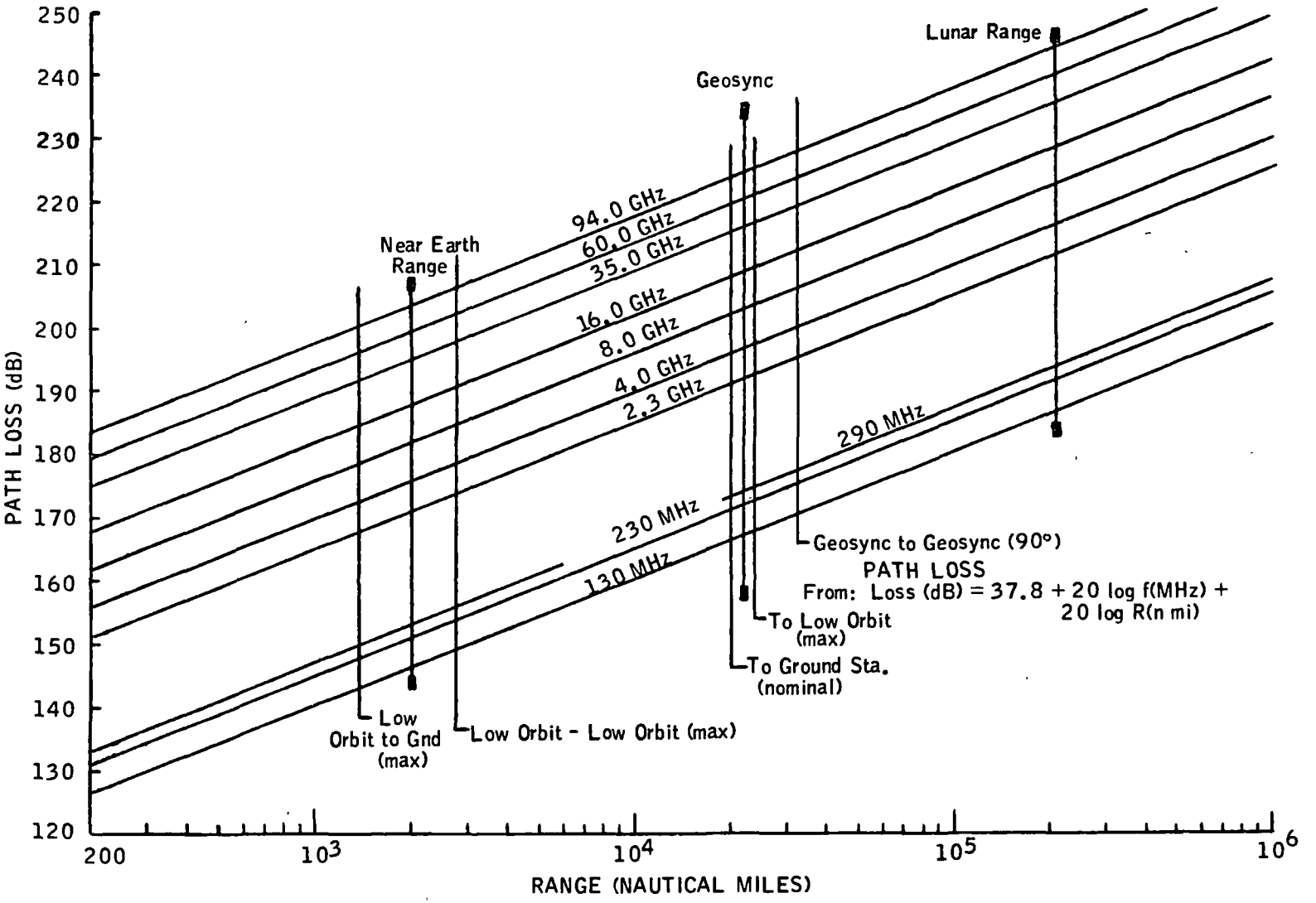


Figure 3.3-10. Path Loss

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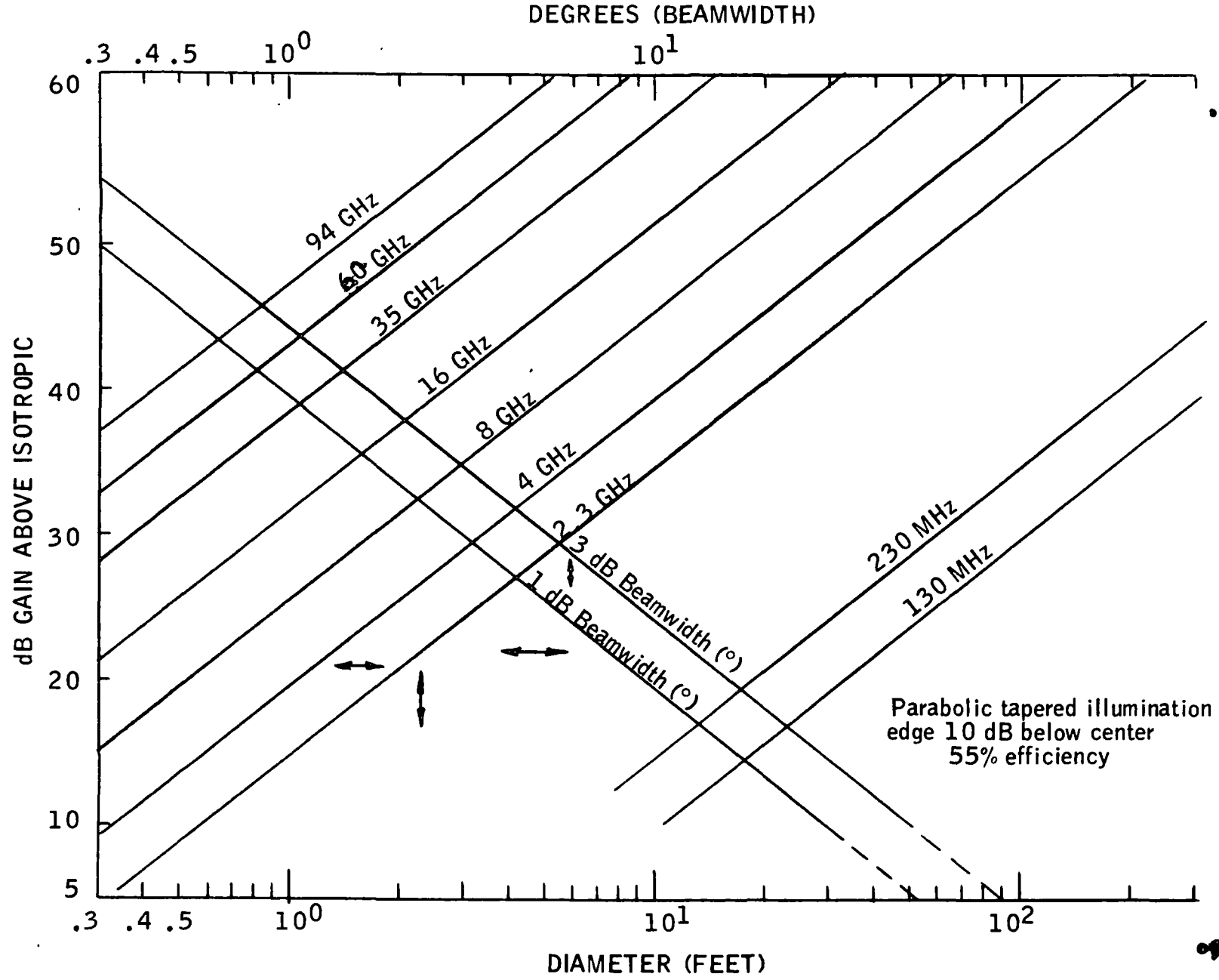


Figure 3.3-11. Parabolic Antenna Characteristics

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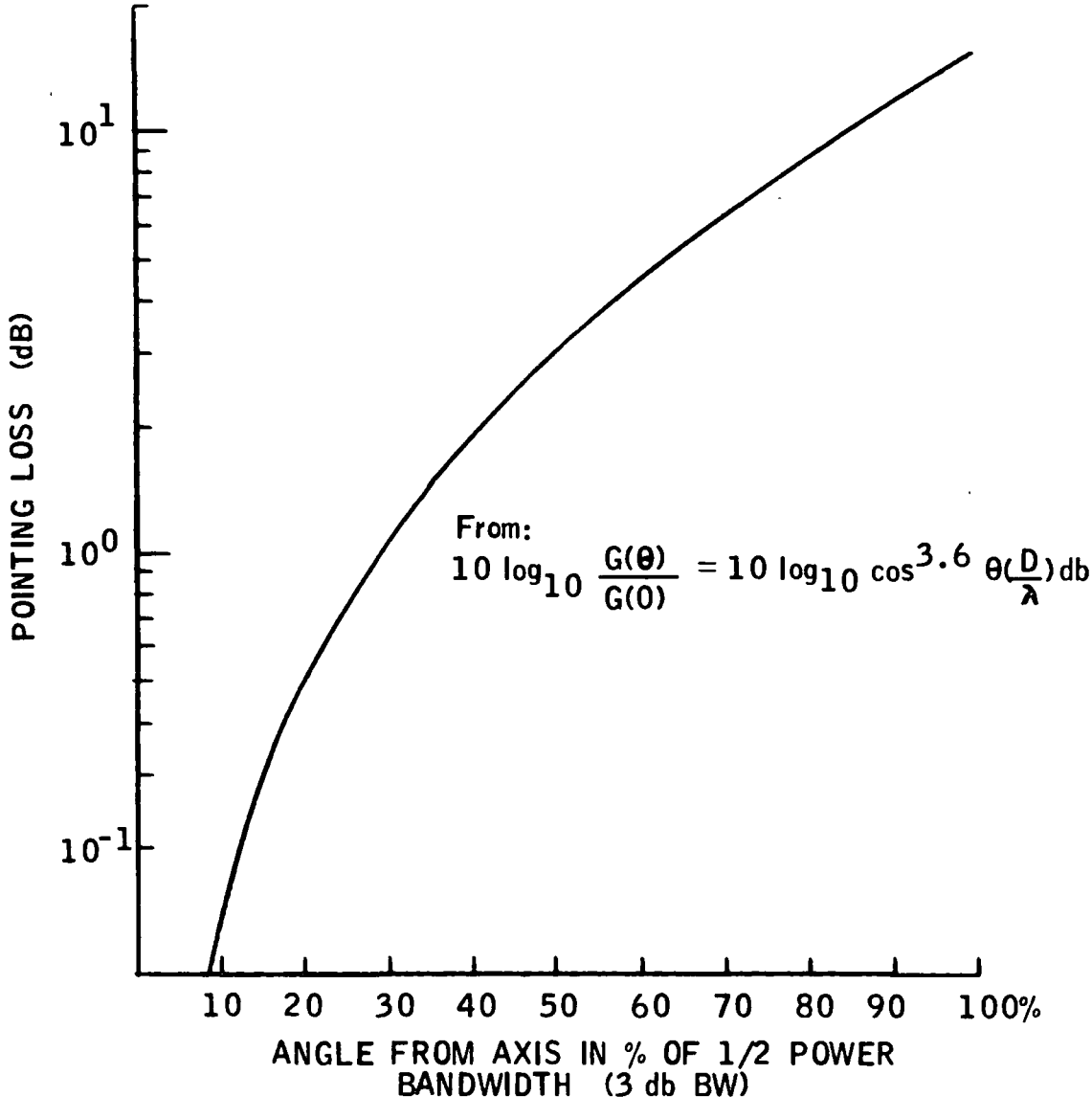


Figure 3.3-12. Parabolic Antenna Pointing Loss

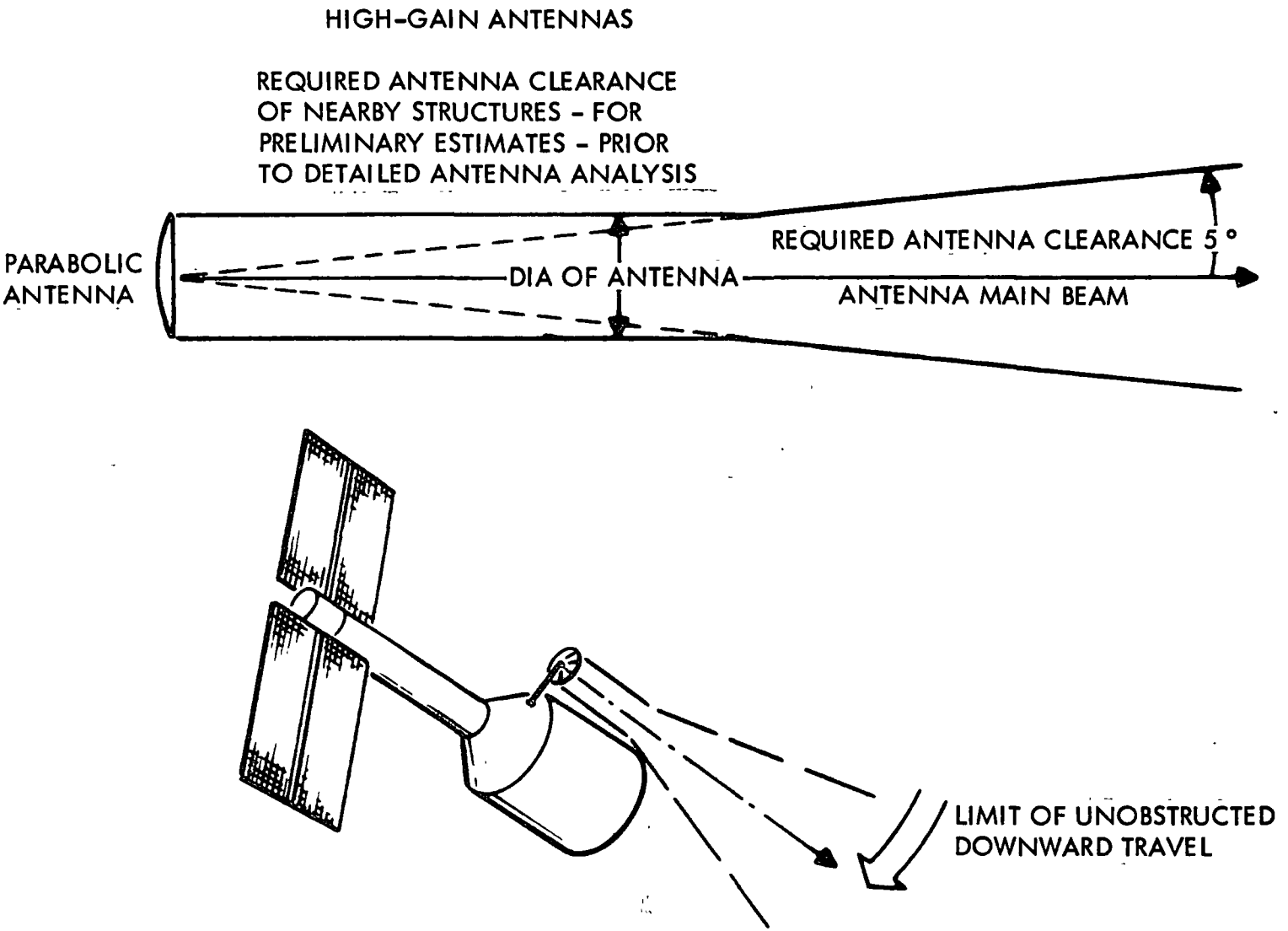


Figure 3.3-13 Parabolic Antenna Blockage (Example)

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Two antennas located such that their coverage overlaps or just meets along the OLS axis would remove the attitude restriction for communications with earth. In addition, long-range tracking and eschange of wide-band data with distant spacecraft could be conducted with restrictions on the approach angles caused by the lack of overlap.

Three antennas could be placed for overlap so that an MSFN and a long-range spacecraft link could be maintained essentially independent of the OLS attitude or the approach angle of the distant spacecraft.

Array Antennas. As an alternate approach, multiple-beam-phased arrays may be feasible in the OLS time period. Two gimbaled arrays replacing the two parabolas would then be capable of both coverage and multiple targets. Non-gimbaled flush arrays would not require physical motion, but would require a large surface area on the OLS. Again, exact design would require specific antenna studies. However, initial estimate for the arrays may be made by considering that their projected areas in the direction of the desired beam should be approximately the same as the area of the required parabola. This factor is illustrated in Figure 3.3-14.

The potential reduction of antenna requirements from 13 feet to 8 or 5 feet would make arrays much more attractive. Heavy deployment of airborne and spacecraft arrays is underway (for example, Hughes Aircraft, AIL, NASA, and other industry-sponsored studies). The elimination of complex gimbaling and drive motors for the large tracking volume required would enhance reliability and lower the maintenance problems. The electronic, rather than mechanical, scan would reduce loads on the stabilization system and essentially remove restriction on tracking rates and search modes. The multibeam capability would allow a number of simultaneous links without the need for additional antennas. The only disadvantages, assuming development has evolved to a production level for space arrays, would be those associated with the required surface areas. Again, as development progresses, a number of array types would be suitable for deployment to fixed, extended positions clear of surface structures.

A summary comparison of parabolas and arrays is given in Table 3.3-20.

Tracking

A summary of tracking requirements is given in Table 3.3-21. Evaluation criteria are presented in Table 3.3-22. A trade tree is shown in Figure 3.3-15 outlining the major approaches. Two basic tracking areas are established. These are tracking for ephemeris determination, rendezvous and docking, and tracking in the sense of antenna pointing. The trades among the approaches for these two areas are shown in Tables 3.3-23 and 3.3-24.

Internal Communications Assembly

Internal communications include both the distribution and the processing associated with digital, video and audio information within the OLS. The basic sources of this information and nominal characteristics are shown in Table 3.3-25. The requirements of the OLS, from a functional standpoint, are almost identical to EOSS requirements. The differences do not influence concept trade evaluations. Therefore, EOSS trades are applicable and are presented below.

The primary decision to be made in the internal communications assembly is whether to time-division multiplex (TDM) or frequency-division multiplex (FDM) as illustrated in Figure 3.3-16. The options within each technique are shown in Figures 3.3-17 (FDM) and 3.3-18 (TDM). Dedicated and switched channel advantages and disadvantages are presented in Table 3.3-26. Fixed and variable frequency source comparisons are made in Table 3.3-27. Three types of modulation

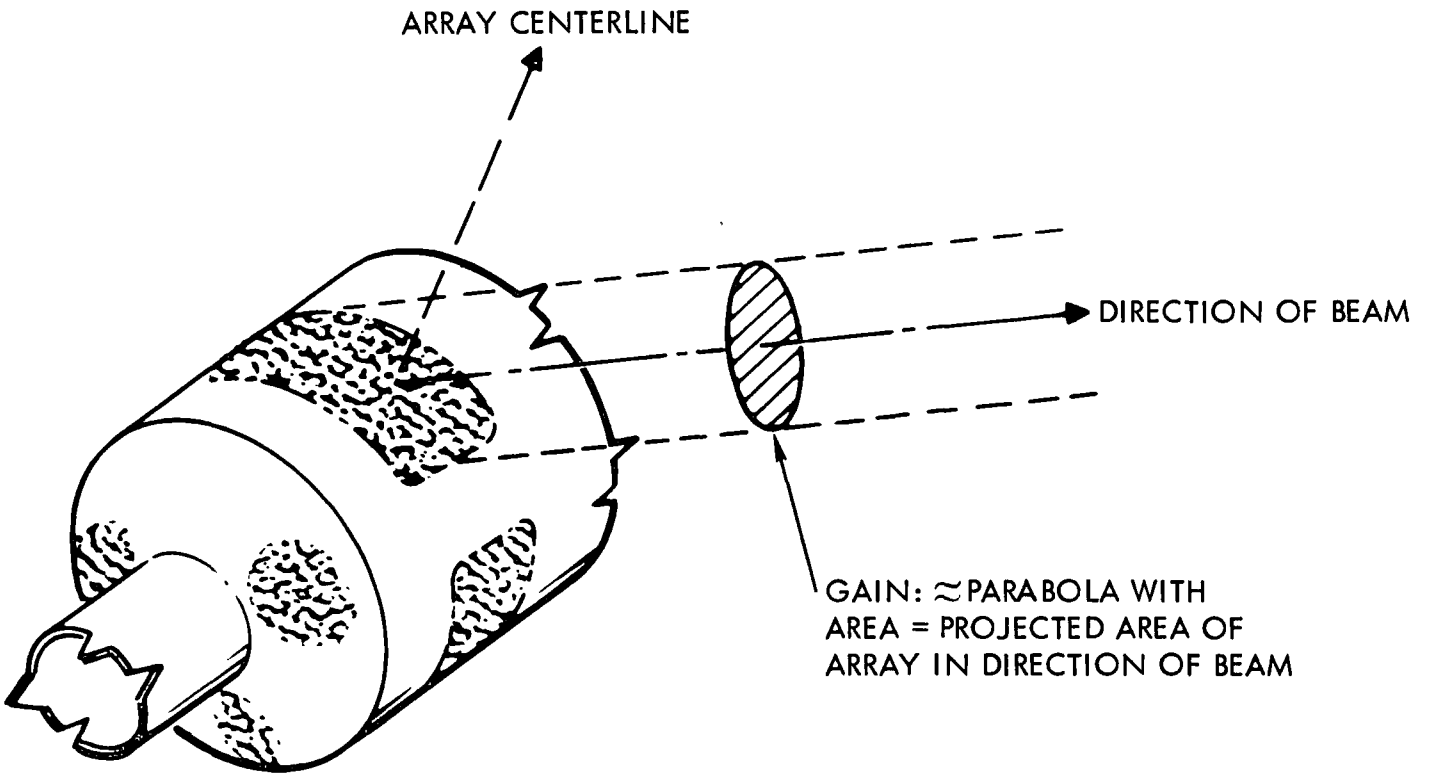


Figure 3.3-14. Approximation of Gain for Conformal Arrays

Table 3.3-20. High Gain Antenna Analysis

Consideration	Parabolic Reflector	Phased Array
Advantages	Has high gain with respect to size and weight. Uses existing unfurlable technology. Uses existing gimbal technology. Feed assembly combining communications and steering elements is existing technology.	Permits multiple lobes. Has no impact on spacecraft stabilization. Is built into spacecraft external surface.
Disadvantages	Requires two antennas, if continuous communication is mandatory. Mechanical steering requires spacecraft motion compensation. Requires deployment on booms subsequent to launch.	High gain requires several thousand radiating elements. Requires continuous supporting software for beam steering. Spacecraft may not provide sufficient surface area for required number of elements. Requires extremely complex feed assembly.

Table 3.3-21. Tracking Requirements

Source of Tracking Signal	Vehicle Being Tracked	Tracking Categories Required	Type Data Required
OLS	Logistics vehicle	a, c	$R, \dot{R}, \theta, \dot{\theta}$
OLS	Experiment module	a, c	R, θ $(\dot{R}, \dot{\theta})$*
OLS	(Lunar relay satellites)	(b)(a)	(θ)
OLS	EVA	N/A	N/A
Ground Station	OLS	a, b	$R, \dot{R}, \theta, \dot{\theta}$
Ground Station	Logistics vehicle	a, b	$R, \dot{R}, \theta, \dot{\theta}$
OLS	Ground	a, b	$R, \dot{R}, \theta, \dot{\theta}$
OLS (optical)	Stars and ground	a, b	$R, \dot{R}, \theta, \dot{\theta}$

*Rendezvous phase only

a = ephemeris determination

b = antenna pointing

c = rendezvous and docking

( ) = additional capability which may be required

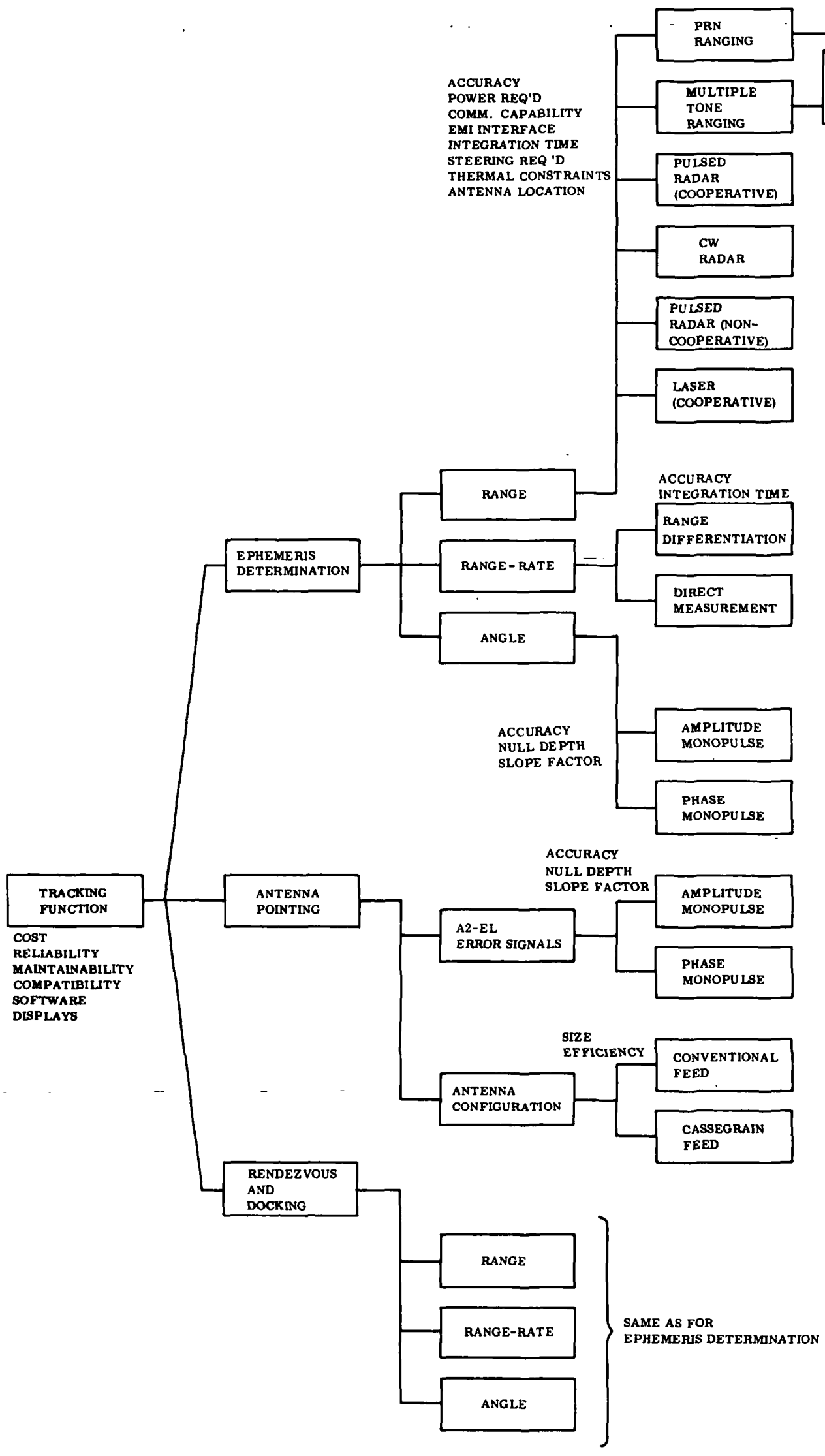


Figure 3.3-15. Tracking Trade Analysis

Table 3.3-22. Tracking Evaluation Criteria

Category	Parameters of Interest
1. Ephemeris determination	Range, range rate Range measurement accuracy 50 ft, random, 50 ft bias Range-rate measurement accuracy $\pm$0.2 ft/sec Maximum range Integrated with communication link when possible Time-shared with wideband link, on short term basis if integration with communication is not applicable Additional power required is negligible Angle measurement is required
2. Antenna pointing	Monopulse feed required on steerable directional antenna Tradeoff typically between amplitude and phase monopulse Monopulse feed results in slight decrease in antenna efficiency Gimbal configuration, X-Y versus az-el mounted Gimbal response and tracking rates
3. Rendezvous and docking	Range, range rate, angle, and angular rate measurements Range accuracy - 5 ft design goal Range rate accuracy Maximum range Angular coverage Required ERP Pulse versus CW; cooperative versus noncooperative radar

Table 3.3-23. Tracking for Ephemeris Determination, Rendezvous, and Docking Tradeoff Chart

Tracking Approach	PRN Ranging	Multiple-Tone Ranging	Pulsed Radar (noncooperative)	"CW" Radar	Pulsed Radar (noncooperative)	Laser (cooperative)
Relative Complexity	Moderate	Low	Moderate	Low	High	High
Power	Low	Low (high)	Low Average, High Peak	Low	Medium Average, High Peak	Medium Average
Weight (pounds)	4 to 8	4 to 8 (TS)	15 to 20	10 to 15	25 to 50	25 to 50
Mechanical Interface	Part of Comm. S/S Four antenna or monopulse feed required	Part of Comm S/S. Four antenna configuration or monopulse feed required	Requires separate antenna	Requires separate antenna	Requires separate steerable antenna	Special optical systems with stringent alignment requirements
EMI Interface	NA	NA	High Peak Power may constrain location of antenna	NA	High peak power will constrain location of antenna	NA
Equipment Availability*	Available	Available	Available	Available	Aircraft type components available	Development stage
Reliability	High	High	Medium	High	Low	Low
Compatibility	Compatible with Communication S/S	Can be made compatible with Communication S/S	NA	NA	NA	NA
Approach Proven on other Systems	Yes	Yes	Yes	No	Yes	No
Acquisition Time (sec)	~ 10	~ 10	< 1	< 1	1 to 5	5 to 10

*Availability refers to the RF equipment; the baseband equipment for the extraction of range and range rate would be a new design.

( ) indicates parameters unique to rendezvous and docking for the selected approach.

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Table 3.3-24. Antenna Pointing Tradeoff Chart

Approach	Amplitude Monopulse	Phase Monopulse	Single-Channel Amplitude Monopulse	Conical Scan
Accuracy	3	1	2	4
Least complex	2	1	3	4
Most reliable	1	2	3	4
Environmental sensitivity	3	2	1	4
Unique problems	Critical balancing of channels	No significant problems	Reliable RF switching required	Requires mechanical motion
Rating: 1 - best 4 - worst				

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Table 3.3-25. Internal Communication Information Sources (Terminal Units) - Maximum Station

Space Station Compartment	Sources and Bandwidth					
	Primary Private Voice Communication, 4.5 kHz	Backup Voice Intercommunication, 4.5 kHz	Video Camera, 4.5 MHz	Video Monitor, 4.5 MHz	Digital Data, IMPS	Public Address Speakers and Intercom, 10 kHz
Experiment provisions	5	5	5	5	5	5
Docking ports	1*	1	1	1	1	1
Airlock laboratory	1*	1	1	1	1	1
Prime control center	2	1	1	3	8	1
Backup control center	1*	1	1	2	4	1
Wardroom	1			1	1	1
Galley and dining	1			1		1
Medical Area	1*		1	1	1	1
Staterooms (12) (Max.)	12			12	1	12

* - Override Capability

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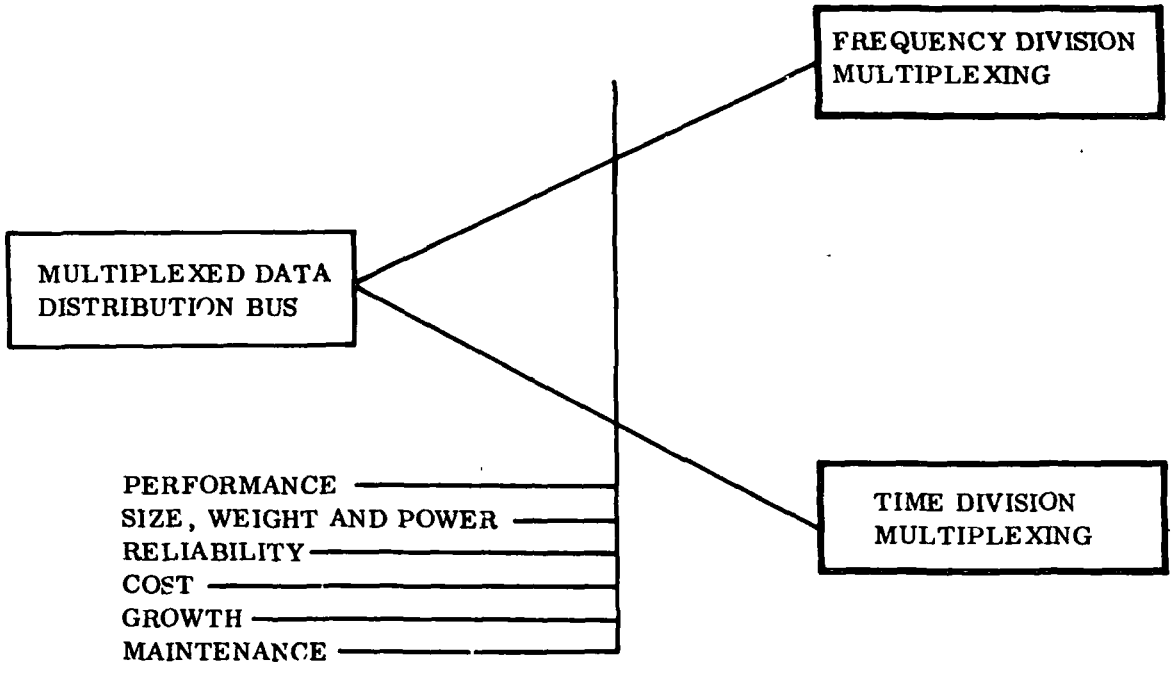


Figure 3.3-16. Internal Communication Trade Tree

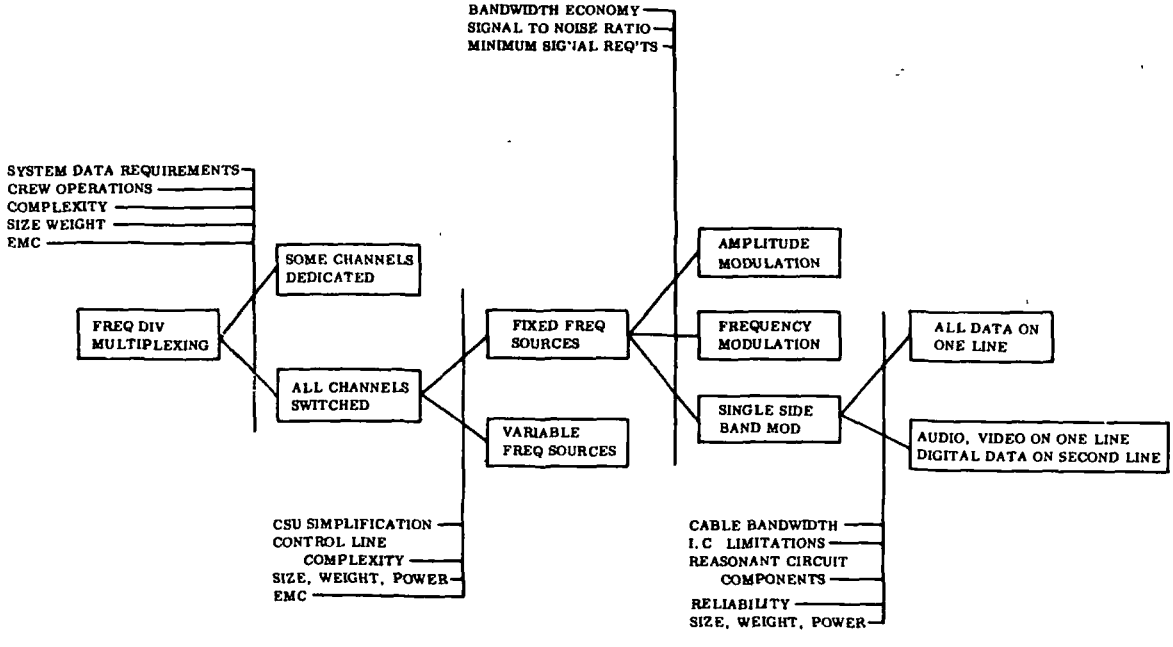


Figure 3.3-17. Frequency-Division Multiplexing Trade Tree

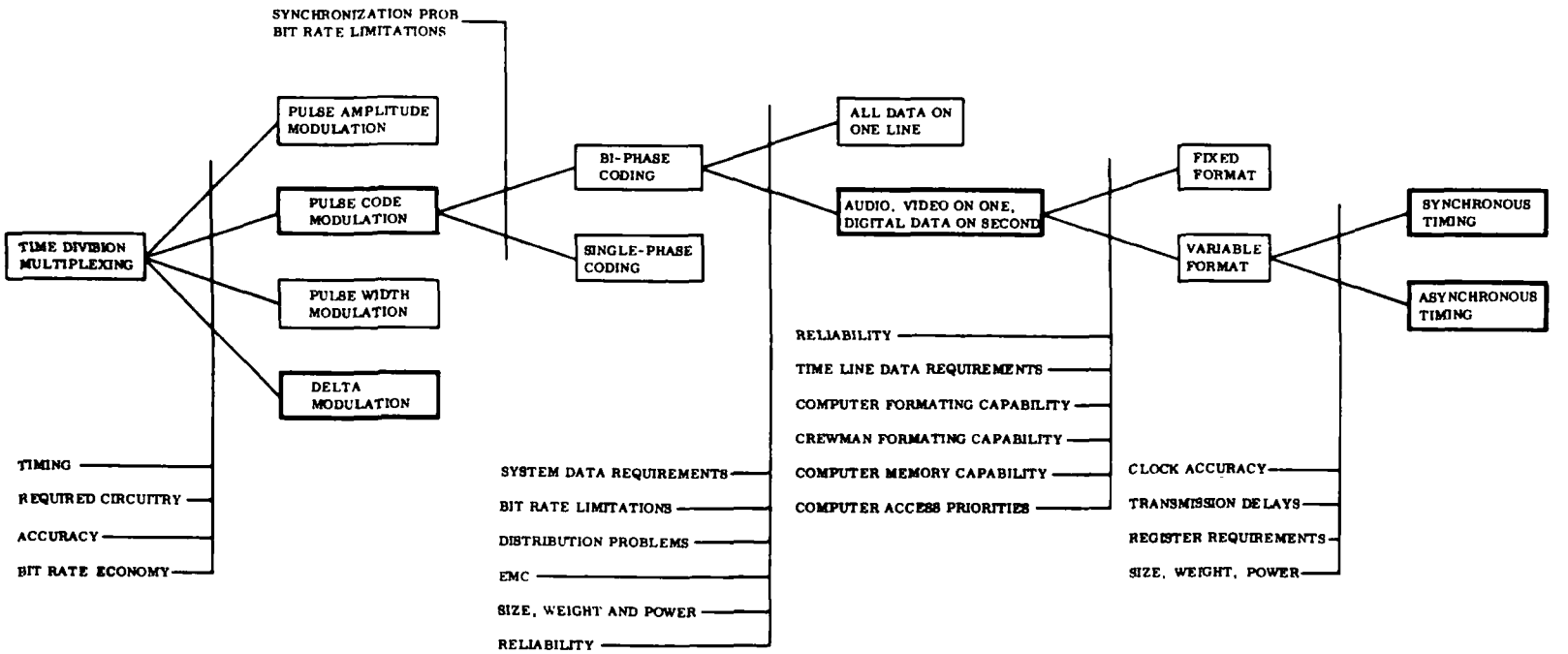


Figure 3.3-18. Time-Division Multiplexing Trade Tree

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Table 3.3-26. Trade Study Evaluation -- Some Channels Dedicated Versus All Channels Switched

Consideration	Some Channels Dedicated	All Channels Switched
Advantages	Dedicated load requires no select switch, CSU requires no load related components for dedicated loads as these loads are not switched.	All stations have flexibility to communicate with all other stations of same function and receive only calls directed at them.
Disadvantages	Dedicated load can communicate only with one source yet receives all information generated by that source even if it is not directed at that load.	Provide selection and switching capability that will be almost always used to select same destination.
Criteria		
Size (cubic feet)	23.75 cu. ft.	24.0 cu. ft.
Weight (pounds)	421 lbs.	425 lbs.
Power (watts)	537 watts	540 watts
Reliability	better	less
Flexibility	None for dedicated load	good
Unnecessary Crew Irritation	Dedicated areas will receive calls not intended for them.	None
EMC	Dedicated channels will be free of switching noise.	Provided for
Onboard checkout and GSE	Operate 106 switching circuits	Operate 115 switching circuits
Software	Logic commands for 104 switching circuits	Logic commands for 115 switching circuits.
Logistics	Stock 106 different modems, 5 different audio-video units	Stock 115 different modems, 8 different audio-video units
Cost	Less	More

(six combinations) applicable to FDM are evaluated in Table 3.3-28. Distribution by single or multiple lines is evaluated in Table 3.3-29. The modulation techniques associated with TDM are evaluated in Table 3.3-30. Biphase and single phase coding are compared in Table 3.3-31. The advantages of fixed and variable formats are presented in Table 3.3-32. Synchronous and asynchronous timing alternatives are evaluated in Table 3.3-33.

3.3.4 DISPLAYS AND CONTROLS

Primary evaluations of displays and controls have been made in console arrangements, types of controls, and types of displays. The trade trees for these three areas are indicated in Figure 3.3-19.

Console Arrangements

An estimate of factors involved in utilizing a functional, subsystem, or hybrid arrangement of consoles is given in Table 3.3-34. A numerical summary of these factors is given in Table 3.3-35 with a weighted summary presented in Table 3.3-36.

Hybrid Arrangements

The command-and-control and engineering consoles are basically the same as those in the functional arrangement; therefore, the crew support hours are the same. The experiment consoles are the same as those in the subsystem arrangement. With this configuration, there is the smallest crew involvement for station service functions and the largest participation in experiment operations.

Types of Displays

Display approach considerations are given in Table 3.3-37. The flight displays are indicated in Table 3.3-38, with assumed preferred types of display indicated for each type of function.

Types of Controls

Basic types of controls considered are discrete switches, control sticks, and keyboards. Detail designs will reveal a large variety of forms under each of these categories. However, general physical parameters can be determined from the basic type. Control approach considerations are summarized in Table 3.3-39. Table 3.3-40 lists control functions and identifies the type of control most suited to that function.

3.3.5 SOFTWARE

The purpose of analysis of software is to establish an organization and design approach guideline that will yield the most cost-efficient, highly qualified and validated software package.

Table 3.3-27. Trade Study Evaluation — Fixed-Frequency Versus Variable-Frequency Sources

Factor	Fixed Frequency Sources	Variable Frequency Sources
Advantages	Audio-video units are kept light and simple with only one modem. Switching is done in CSU on commands from DMA.	Data does not have to be demodulated, switched, and remodulated in CSU.
Disadvantages	Some noise will be introduced in demodulation, switching, and remodulation in CSU.	Audio-video units must contain modems for all stations to which unit may be switched. Switching commands for telephone must clear through CSU and DMA and back to audio-video unit to ascertain if desired line is not busy.
Criteria		
Size (cubic feet)	9.8	11.1
Weight (pounds)	213	287
Power (watts)	78	68
Reliability	Better	Less
EMC	Less	Better
Software	Logic command for 115 switching circuits	Logic commands for 95 switching circuits
Cost	Less	More

Table 3.3-28. Trade Study Evaluation—Types of Modulation

Consideration	Modulation Alternatives[1]					
	AM – AM	AM – PM	AM – AM	FM – PM	SS – AM	SS – PM
Relative Rating	4	3	6	5	2	1
Criteria (in order of importance)						
Bandwidth required (MHz)	287	287	348	348	215	215
Signal-to-noise ratio,						
Audio	0.0344	0.0298	0.0891	0.154	0.137	0.356
Video	0.103	0.090	0.269	0.469	0.416	1.08
Digital (low)	0.063	0.054	0.108	0.187	0.250	0.648
Digital (high)	0.0395	0.0342	0.0684	0.118	0.157	0.205
Minimum signal power required (MW)						
Audio	13.8	18.4	2.07	0.692	0.876	0.130
Video	43.4	66.5	6.49	2.09	2.66	0.396
Digital (low)	12.6	17.2	4.30	1.43	0.791	0 119
Digital (high)	2640	3510	879	297	165	97
Interchannel interference[3]	More subject to fixed pitch beat note interference		More subject to fixed pitch beat note interference		More subject to fixed pitch beat note interference	

Notes:

1. AM - amplitude modulation
   PM - phase modulation
   FM - frequency modulation
   SS - single-side-band modulation

2. With respect to ratio of amplitude modulation of single subcarrier.

3. Except as noted, the ratios of signal-to-interference ratios for various approaches correspond roughly to the ratios of signal-to-noise ratios given above.

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Table 3.3-29. Trade Study Evaluation — One Line Versus Multiple Lines

Consideration	All Data On One Line	Multiple Lines
Advantages	**Saves 50 pounds**	Transmission bandwidth on cable can be kept to realizable values. Do not have total dependency on one line.
Disadvantages	Total bandwidth may not be realizable with practical hardware. Line failure could destroy all communications.	Increased weight. Dual software programs required in computer to control switching.
Criteria		
Bandwidth	215 MHz using single side band modulation,	**Audio-video require 91 MHz, digital 125 MHz. Figures for base would be audio-video, 230 MHz; digital, 350 MHz.**
Hardware Capability	**Present maximum practical frequency is 300 MHz for capability of** Adjustable lumped resonant circuit components Integrated circuits Small coaxial cables.	Two cable division would have Station requirements well within present capabilities.
Weight	**Space Station is approximately 50 pounds, using redundant lines.**	**Space Station is approximately 100 pounds, using redundant line for each line.**
EMC	More problems	Less problems
Software	One program to control all switching	Two programs required

Table 3.3-30. Trade Study Evaluation -- Types of Modulation

Consideration	Pulse-Amplitude Modulation	Pulse-Word Modulation	Pulse-Code Modulation	Delta Modulation
Advantages	**Quantizing error is minimum; bit rate is low**	Quantizing error is minimum. Not subject to amplitude errors. Uses mostly digital circuits. low bit rate.	Quantizing error is adjustable. Noise immunity is good. All digital circuits. High overall accurracy if enough bits are used in coding. Station digital data already in this form. Can be regenerated.	Can tolerate very high transmission error rate. Noise immunity is good. Mostly digital circuits. Can be regenerated.
Disadvantages	Susceptible to noise. Requires accurate, linear analog circuits. Pulse can not be regenerated.	**Jitter and timing errors results in data errors. Wide dynamic range results in very narrow pulses with fast rise and fall time. Requires accurate clock, fast pulse circuits, and wide bandwidth.**	High bit rate, requiring fast circuits. Timing and synchronization can be problems.	Simulation is poor if signal varies very fast or very slowly. Requires accurate differential amplifier.
Criteria Bit Rate	Low, one pulse per sample	Low, one pulse per sample	High, depending on number of bits per sample for coding. Use 8 bit coding for most purposes.	High, depending on simulation and quantizing required.
Quantizing Error	**Minimum, since this is a continuous process.**	**Minimum, since this is a continuous process.**	Depends on number of bits used in coding, **±.25 percent if 8 bits are used.**	Depends on dead zone of differential amplifier.
Accuracy	All noise results in direct error, difficult to filter.	**Amplitude variants cause no error; only timing errors and pulse jitter cause errors**	Only errors are in A/D conversion (can keep below 1/4%). Error due to loss of pulse can be avoided with parity checks. Not susceptible to pulse amplitude error.	Not susceptible to pulse amplitude error, only error would be drift in differential amplifier.
Type of Circuits	Analog circuitry, requiring high accuracy and linearity with minimum drift.	**Amplitude circuits on** digital but timing circuits are analog.	Only need for analog circuits are in A/D and D/A converters. All others are fast digital circuits.	Digital circuitry except for differential amplifier.
Timing Accuracy	Precise timing not required.	Precise timing required as timing errors result in direct data errors.	Precise timing needed for synchronization.	Timing requirements not as severe as PCM.

Table 3.3-31. Trade Study Evaluation - Single-Phase Versus Biphase Coding

Consideration	Single-Phase Coding	Biphase Coding
Advantages	Most digital data originate in this form, and require no conversion. Data are easily converted back to analog form, as a 1 is always one level, 0 is the other level. Can result in lower transmission bit rate during periods of lack of data change.	Level transition occurs every pulse, whether this is a change in signal level or not. Receiving detector can not get out of synchronization with transmitter.
Disadvantages	Periods of lack of change in signal level result in continuous level signals at the detector. Can result in loss of synchronization unless bit clock signals are sent in parallel to data.	Binary data that originates in single phase form must go through another stage and be converted to biphase coding. Bit rate stays at full rate even during periods of lack of data change.
Criteria		
Synchronization	Not certain, unless separate clock line is supplied.	No problem
Circuit speed	Must be fast enough to handle periods of high data-rate change.	Circuit speeds stay constant.
Weight	May have to add bit clock transmitter, line, and detectors.	Have to add converters if data originate in single-phase coding.

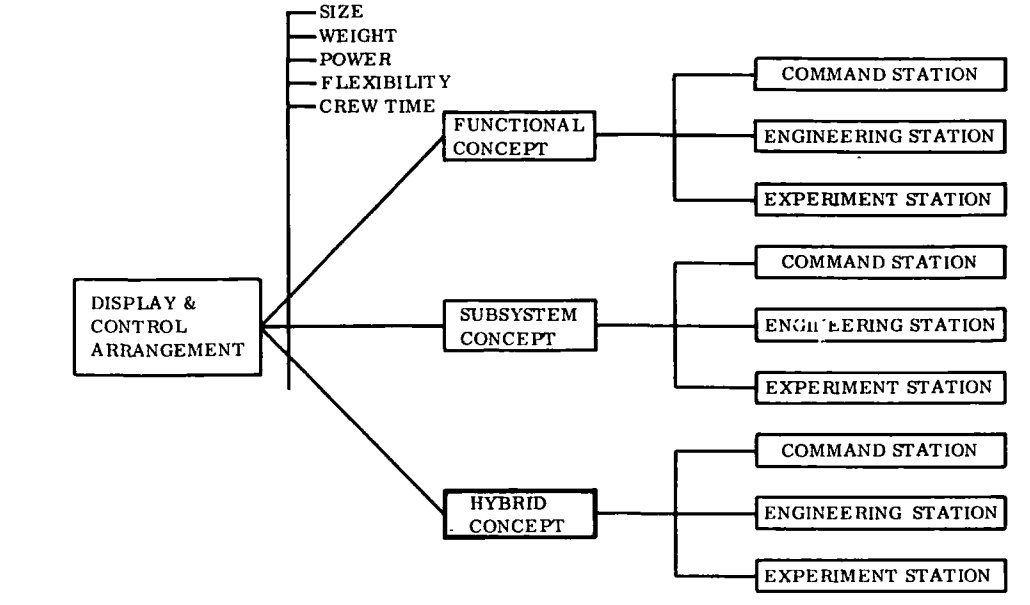
Table 3.3-32. Trade Study Evaluation — Fixed Versus Variable Format

Consideration	Fixed Format	Variable Format
Advantages	Each piece of data has a known place in the format and can be accessed by the computer or by the crew with a minimum amount of computer time and memory. Highly reliable system, since nothing is changed after the system has been thoroughly tested.	The only data being taken and transferred about the vehicle are data that are required at the time. Data telemetered to the ground are kept to a minimum. Emphasis can be placed on specific points as situations arise on the vehicle. Bit rate kept as low as possible at all times.
Disadvantages	Data sources are queried when they have no data. Format will have percentage of empty slots during large portion of operation. High bit rate at all times. No method of increasing emphasis on particular point during an anomalous condition.	Appreciable memory capability must be dedicated to keeping track of data format and changing format. Possibility of error if crew changes format through remote terminal unit.
Criteria		
Time-Line Data Requirements	Can not respond to changing needs	Can respond
Data Bit Rate	Fixed high enough to sample all sources at all times	Only as high as required
Computer Memory	Minimum required	Considerable required
Computer Access to data	Excellent	Can be done but requires extensive memory and software
Crew Access to date	Good	Requires additional software
Crew-variable format	Not possible	Possible
Reliability	Higher	Lower

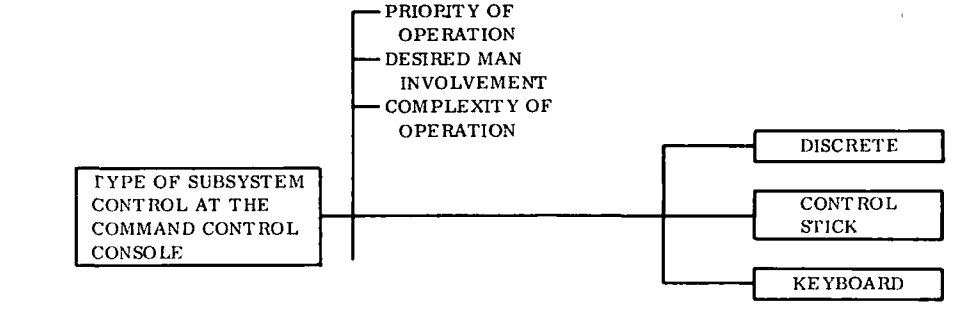
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Table 3.3-33. Trade Study Evaluation — Synchronous Versus Asynchronous Timing

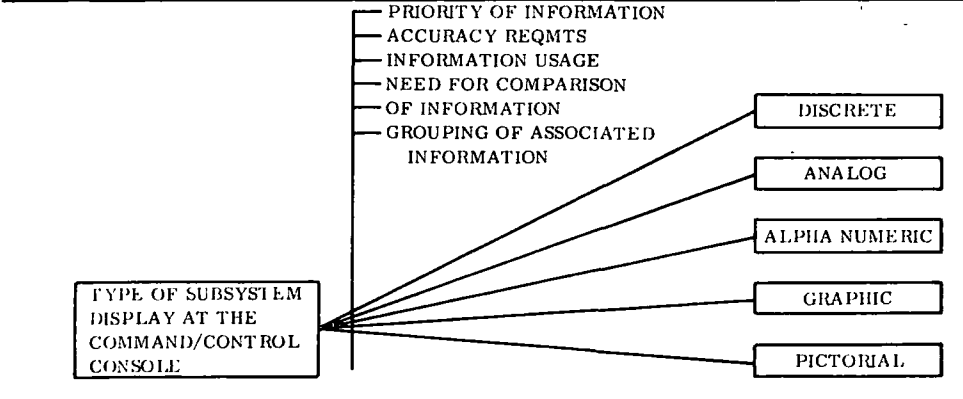
Consideration	Synchronous Timing	Asynchronous Timing
Advantages	Has minimum number of Storage registers. All data are handled in real time; do not require time tag or source identification.	Time delays due to long length are not a problem. Clock timing errors do not cause problems.
Disadvantages	May be impossible to achieve due to transmission delays over long cables. Clock timing errors can cause problems, even over short-line transmissions.	Storage registers to hold data temporarily will impose penalty in size, weight and process. Delays in getting data into format may necessitate addition of source code and time tag to all data resulting in increased bit rate.
Criteria		
Transmission delays	Can make operation impossible	Do not cause problems
Clock variations	Can cause loss of synchronization	Do not cause problems
Data identification	No problem, as data are clocked into format as they are requested.	Transmission delays may effect positioning of data in format, requiring location and timing labels added to data. Increases bit rate.
Hardware	No temporary registers required.	All data received asynchronously must be temporarily stored until they can be used in format, results in increase in size, weight, and power.



CONSOLE ARRANGEMENT TRADE



TYPE OF CONTROL



TYPE OF DISPLAY

Figure 3.3-19. Display and Control Trades

Table 3.3-34. Evaluation Estimates for Console Arrangement Study

Console Arrangement	Volume (ft^3)	Weight (lb)	Power (watts)	Crew Time (hr/day)	Flexibility Ranking
Functional					
Command station	57	1100	450	24	
Engineering station	100	1332	260	4	
Experiment station	57	760	270	16	
Total	214	3192	980	44	5*
Subsystem					
Command station	83	1420	580	48	
Engineering station	118	1900	380	4	
Experiment station	80	1040	340	48	
Total	281	4360	1350	100	10
Hybrid					
Command station	66	1140	410	24	
Engineering station	80	1180	200	4	
Experiment station	80	1040	390	48	
Total	226	3360	1000	76	10

*Doesn't allow for parallel experiment operation.

Table 3.3-35. Numerical Summary

Arrangement	Size (ft^3)	Weight (lb)	Power (watts)	Crew Requirements (hr/day)	Flexibility Ranking	Comments
Functional	214	3192	980	44	5	Doesn't allow parallel operation of experiments.
Subsystem	281	4360	1350	100	10	Allow parallel operation of experiments
Hybrid	226	3360	1000	76	10	

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Table 3.3-36. Weighted Summary

Arrangement	Size, Weight and Power		Crew Hours		Flexibility		Total Relative Weight	Remarks
	Weight Factor 30 percent	Rank (percent)	Weight Factor 30 percent	Rank (percent)	Weight Factor 40 percent	Rank (percent)		
Functional	0.300	100	0.300	100	0.200	50	0.800	
Subsystem	0.222	74	0.132	44	0.400	100	0.754	
Hybrid	0.288	96	0.174	58	0.400	100	0.862	Selected approach

Note: By definition, a rank of 100 percent is the most desireable approach for a particular parameter. The console arrangement with the highest total relative weight is the best approach.

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Table 3.3-37. . Display Approach Considerations

Display Device	Advantages	Disadvantages
Discrete Light	Attracts attention of crew Uses minimal power Uses minimal command data bits Not subject to misinterpretation	Highly specialized display devoted to a single function. Qualitative information (good-bad, Go-No Go)
Discrete alphanumeric matrix	Versatile display Uses less power than CRT alphanumeric. Can be operated directly from logic circuitry. Data rate requirements less than CRT.	Characters per square inch is less than CRT alphanumeric. Presently available in only single color.
CRT alphanumeric	More versatile than above. Same CRT can be used for graphic display. More characters per square inch than discrete alphanumeric	Requires more power for tube illumination and display-generation circuitry. Information data rate requirements are larger than discrete alphanumeric.

Table 3.3-37. Display Approach Considerations (Continued)

Display Device	Advantages	Disadvantages
CRT alphanumeric (continued)	Color coding can be used for special identification.	
CRT Graphic Presentation	Increases comprehension of data by consolidating and presentating in graphic form. Reduces decision-making time.	Requires software programs for collection of data, computation, and formatting of output data to the display.
Analog	Allows visual comprehension of relative magnitude of data presented (meter display). Safe values of data can be easily recognized by color coding zones of the display.	Digital information must be converted to analog form. Display is highly specialized; devoted to single function.

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Table 3.3-38. Flight Display Analysis

Function	Discrete Light	Discrete Alphanumeric	CRT Alphanumeric	CRT Graphic	Analog	Television	Microfilm Viewer	Rationale	
1 0 Flight parameter									
1 1 Orbital position		X						Can be described in short numeric form	
1 2 Orbital velocity		X						Can be described in short numeric form	
1 3 Three-axis altitude				X				Will be used for manually orienting Space Station altitude	
Three-axis rate			X					Need visual presentation	
1 4 Operating status		X						Can be described in short engineering forms	
Operational mode		X							
1 5 Time and timing indications		X						Want continuous read-out of vehicle time	
1 6 G&N computation indications			X					Large quantity of data prohibits use of alphanumeric	
1 7 Propulsion parameters			X					Large quantity of data prohibits use of alphanumeric	
2 0 Experiment parameters									
2 1 Experiment schedule			X					Large quantity of data prohibits use of alphanumeric.	
2 2 Operating status			X					Large quantity of data prohibits use of alphanumeric	
2 3 Operational mode			X					Large quantity of data prohibits use of alphanumeric	
2 4 Experiment and experiment-area display						X			

Table 3.3-38. Flight Display Analysis (Cont)

Function	Discrete Light	Discrete Alphanumeric	CRT Alphanumeric	CRT Graphic	Analog	Television	Microfilm Viewer	Rationale	
2 5 Data Bank							X	Would require unreasonable amount of memory core for magnetic storage	
2 6 Monitor and alarm of experiments	X		X					Flashing light for getting attention Red flashing alphanumeric describing anomaly Discrete light for crew critical functions CRT-alphanumeric for functions affecting station status	
3 0 Personnel status									
3 1 Name Title Location Present assignment Health Duty schedule			X					Large number of characters required to present this data CRT will allow addition of new parameters if desired	
3 2 Station Inter video						X		Per NASA guidelines and constraints	
4 0 Emergency display									
4 1 Identify emergency Type Location			X					Related items 4 1 through 4 3 will be displayed on common CRT Will be presented in word format Large quantity of characters are required	
4 2 Identify problem-solving mode (auto or manual)			X						
4 3 Recommend course of action			X						
5 0 Satellite flight parameters									
5 1 Track satellite Range data Range rate LOS angle LOS angular rate		X						Data required for information only Adjustment of satellite orbit will be via computer program	

Table 3.3-38. **Flight Display Analysis (Cont)**

Function	Discrete Light	Discrete Alphanumeric	CRT Alphanumeric	CRT Graphic	Analog	Television	Microfilm Viewer	Rationale	
5 2 Three-axis attitude				X				Will be used for manually changing satellite attitude	
Three-axis rate			X					Need visual presentation	
5 3 Satellite video data						X		Allow cmd 'control center to observe general progress High resolution data will be monitored in the experiment station	
6 0 Logistics parameters									
6 1 Tracking logistic vehicle Range data Range rate LOS Angle LOS Angular rate		X						Status information required by crew until visual contact is made	
6. 2 Logistic vehicle attitude Three axis attutude Three-axis rates			X	X				Information used for controlling logistic vehicle during docking	
6 3 Logistic vehicle propulsion parameters			X					Large quantity of data	
6 4 Data bank							X	Would require unreasonable amount of memory core for magnetic storage	
6 5 Logistics caution and warning	X		X					Flashing light for getting attention Red flashing alphanumeric describing anomaly Discrete light for crew-critical functions CRT - alphanumeric for functions affecting station status	

Table 3.3-38. Flight Display Analysis (Cont)

Function	Discrete Light	Discrete Alphanumeric	CRT Alphanumeric	CRT Graphic	Analog	Television	Microfilm Viewer	Rationale	
7 0 Service subsystem 7 1 Present status of service subsystems (including consummable status)			X					Telemetry information pertaining to subsystem operations can be addressed for display by crew selection via keyboard input, which selects the information required	
8 0 Operating schedules 8 1 Display the derived and accepted schedule			X					Quantity of data too large for alphanumeric Light-emitting diodes display	

Table 3.3-39. Control Approach Considerations

Control Device	Advantages	Disadvantages
Discrete switch	Allows quick crew response Control device easily recognizable. Crew operations reduced to a minimum.	Highly specialized control devoted to single operation.
Control stick	Provides crew with capability of direct manual control.	
Keyboard	Most versatile Multipurpose	Requires use of special coding for selection of operations. Response time to initiate operation is longer than discrete switch.

The basic software organization trade tree is shown in Figure 3.3-20. Considerations for organization are given in Table 3.3-41. Design considerations are shown in Table 3.3-42, and those for language in Table 3.3-43.

3.3.6 ISS BASELINE CONCEPT

To establish some basic physical parameters for the support of the overall OLS configurations, a baseline subsystem is assumed and extrapolations made from equipment are defined in the EOSS studies. Variations as a function of crew size will be in the areas of data processing, display and control, and internal communications. The external communications variations will be primarily affected by the trade between physical antenna configurations, capability, and power requirements. The baseline configuration assumes two parabolic 13-foot antennas, eight omniantennas, two approach radars, and the associated transponders for all configurations.

Functional Configuration

The functional configuration of the baseline ISS is shown in Figure 3.3-21. This block diagram will be valid for the range of OLS designs under consideration. Variations will be in weight, size, power, and number of elements represented by the blocks.

Table 3.3-40. Control Analysis

Function	Keyboard	Control Stick	Discrete Switch	Paging	Rationale
1.0 Flight control					
1.1 Thrust control	X				Computer reactions faster than man. Can compensate for variations in thrust levels.
1 2 Attitude control	X	X			Active man involvement needed for docking function
1.3 G&M operating mode Selection			X		Relatively high usage rate
2 0 Experiment status					
2.1 Turret TV camera in experiment area		X			Allow crew selection of picture at command control console
2 2 Zoom and focus TV camera		X			Allow crew selection of picture at command control console.
3.0 Personnel status					
3.1 Update personnel data					Magnitude of information precludes discrete switches.
3.2 Select interstation video to be presented at command center		X	X		Provide easy selection of video.
3.3 Turret TV camera		X			Allow crew selection of picture at command control console
3 4 Zoom and focus TV camera		X			Allow crew selection of picture at command control console.
4 0 Emergency control					
4 1 Alert crew				X	Verbal description of problem and action required.

Table 3.3-40. Control Analysis (continued)

Function	Keyboard	Control Stick	Discrete Switch	Paging	Rationale
4. 2 Safe malfunctioning subsystem (if not automatic)				X	Using personnel status information, organize repair party and direct course of action
5. 0 Satellite control					
5. 1 Attitude control	X	X			Attitude changes will be required during satellite experiment operation
5. 2 Thrust control	X				Computer reactions faster than man can compensate for variations in thrust levels.
5. 3 Emergency destruct	X				Coded signal needed to select satellite and perform destruct
6. 0 Flight control, Rendezvous, and Docking					
6 1 Maneuver logistic vehicle, attitude control	X	X			Active man involvement required.
6. 2 Thrust control	X				Computer reactions faster than man can compensate for variations in thrust levels
7. 0 Service subsystem					
7 1 Select subsystem information	X				Quantity of data precludes individual switches for selection
8 0 Operations scheduling					
8. 1 Manually enter information on variables related to operation scheduling.					Quantity of information precludes discrete functional switches
8. 2 Manually modify the derived schedule and update by entering accomplishments.	X				Interface to ISS is through the keyboard.

Table 3.3-40. Control Analysis (continued)

Function	Keyboard	Control Stick	Discrete Switch	Paging	Rationale
9.0 Communications					
9 1 Direct telephone access			X		Secure immediate access to critical areas.
9 2 Dial telephone access	X				Allow calls to all space station phones and to earth via RF link.

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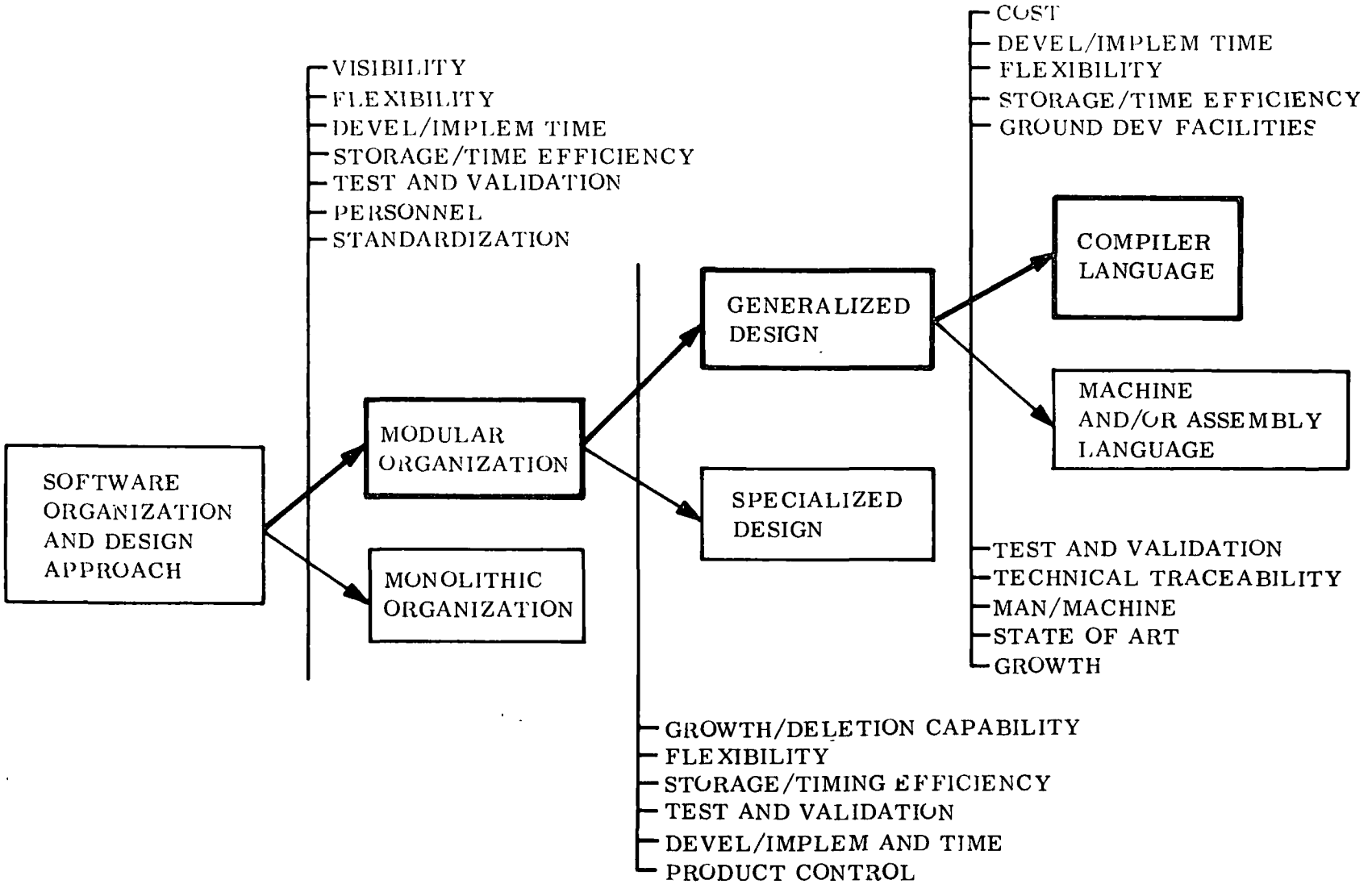


Figure 3.3-20. Modular Versus Monolithic Organization

Table 3.3-41. Modular Versus Monolithic Organization

Considerations	Modular Organization	Monolithic Organization
Significant feature	Large software packages are broken down to individual programmer-size modules having rigorously defined I/O and performing distinct, essentially stand-alone processing functions.	Independent, large software packages are written as continuous script by a group of analyst(s) and programmer(s) interfacing with each other on an informal personal basis. Only package interfaces and overall functional requirements are rigorously defined.
Criteria evaluation		
Management visibility (during development and implementation)	Module unit scheduling facilitates progress monitoring and resource control.	True progress hard to assess, completion date is only firm milestone.
Technical visibility	Program and modules firmly defined as to relationship, interfaces, and processing functions.	Visibility is a direct function of technical-team organization, individual capabilities, and personal relationships.
Change flexibility	Impact usually restricted to only one module.	Impact may require extensive rewriting.
Development and implementation time	Resource-allocation control and problem solution by module parts insures continuous progress and allows deferring modules requiring R&D (more analysis).	Time period dependent on having complete definition of software package. Modifications cause dalays.
Storage and timing efficiency	Packaging and organization reduces efficiency.	Very efficient code can be written.
Test and validation	Detailed debugging and test occurs at each development-level module, module group, and package. High confidence is established.	Lack of formal organization impedes systematic testing approach. Parts of package will tend to be under-exposed — allowing bugs to go undetected.
Personnel adaptability	Revision programmer can readily familiarize himself with module(s) to be modified. Prior knowledge is not an absolute necessity.	Certain modifications will require prior detailed familiarity.
Standard documentation	Basis exists for standard documentation starting with basic element (module) and upward.	Standardization applicable only at a system level.
Relative merit	Good	Poor
Recommendation: Software organization should be modular because of management and technical visibility and control, cost and time economics, standardization aspects, maintenance and change flexibility, and quality assurance.		

Table 3.3-42. Generalized Versus Specialized Design

Considerations	Generalized Design	Specialized Design
Significant feature	Programs are written in a generalized manner with the aim of accommodating changes (constants, variables, etc.). Concept is that changes will be made to data base tables and items and therefore not result in program rewrite or modification. Also, data flow between modules and programs is made via the data base.	Programs are written with the express purpose of satisfying a specific application. Provision for "fine tuning" variables and constants is provided within program structure itself. Minimal interdependence on other programs in system.
Criteria evaluation		
Change flexibility	Change data base quantity.	Change program code.
Storage and timing efficiency	System overhead is low, since common code can be used for several applications.	Overhead for individual application is low.
Test and validation	Test inputs are easily variable via data base; outputs via snapshot dumps of data base.	Special logic is required to provide or dump diagnostic data; cumbersome.
Growth capability (new programs)	There is potential to use existing routines and other system services.	Capability must evolve from scratch.
Deletion capability (obsolete programs)	Shared portions must be considered prior to deletion; may require modifications.	Programs are removable with minimum effort in that no other use of their code exists.
Configuration change control	Documentation of data base and program separately enables tight control and visibility; uses standardization to utmost benefit.	Program and data are intermixed — primarily visible only to programmer.
Development and implementation time	These are facilitated by common use of existing software and test and validation capabilities.	Design requires more special logic and test software.
Relative merit	Good	Medium to poor
Recommendation: Design should be generalized because of overall system flexibility, growth capability, and visibility. Approach promises economies in both cost and schedule.		

Table 3.3-43. **Compilar Language Versus Machine and/or Assembly Language**

Considerations	Compiler Language	Machine and/or Assembly Language
Significant feature	High level programming language automatically convertible, via software processing, to machine code of the object computer (one-for-many correspondence).	Programming performed in terms of machine code of the code of the object computer (one-for-one correspondence).
Criteria evaluation		
Initial costs	Potentially high. Compiler software generation is required.	Low. Assembler and code generators required.
Recurring costs	Low. One-for-many coding during modification and program generation.	High. One-for-one coding during modification and program generation.
Development and implementation time	Minimal time is required; code is essentially error-free, duplication of code is reduced.	Concept is time consuming, tedious, error-prone.
Change flexibility	Recompile	Recording and bookkeeping required.
Storage and timing efficiency	Has flexibility and convenience cost-efficiency. (Compiler code can be optimized where mandatory.)	Efficient. (Varies as a direct function of individual programmers skill.)
Ground development facilities	Virtually machine-independent, provided associated compiler exists.	Machine-dependent; limited to a specific machine or family (same manufacturer).
Test and validation	Compiler generates more error-free code and will improve with use. New programs will be validated quicker and with higher confidence.	Coding and bookkeeping errors commonplace. They increase rapidly as a function of program size and complexity.
Technical traceability	Compiler use for equation and logic designer's programming provides traceability throughout SS Program and insures continuity of technical expertise and support.	Interpretation of assembly and machine code is extremely difficult, resulting in technical isolation.
Man and machine relationship	Language is readily learned, provides large and interchangeable programmer pool for on-board application.	Language is difficult to learn. Individual talents important, restricting pool and interchangeability. Ground application only.
State of development and application	Well in hand, SPL a reality — others are in various development and application phases.	Standard availability with given machine.
Experiment growth	Future contractors will require only a compiler for their machine. Interfaces are well bounded.	Requires access to flight machine and/or ground equivalent.
Relative merit	Good	Poor
Recommendation: Programs should be written in a compiler language because of long-range benefits in cost and schedule, quality assurance, machine independency, personnel flexibility, technical visibility, and space autonomy.		

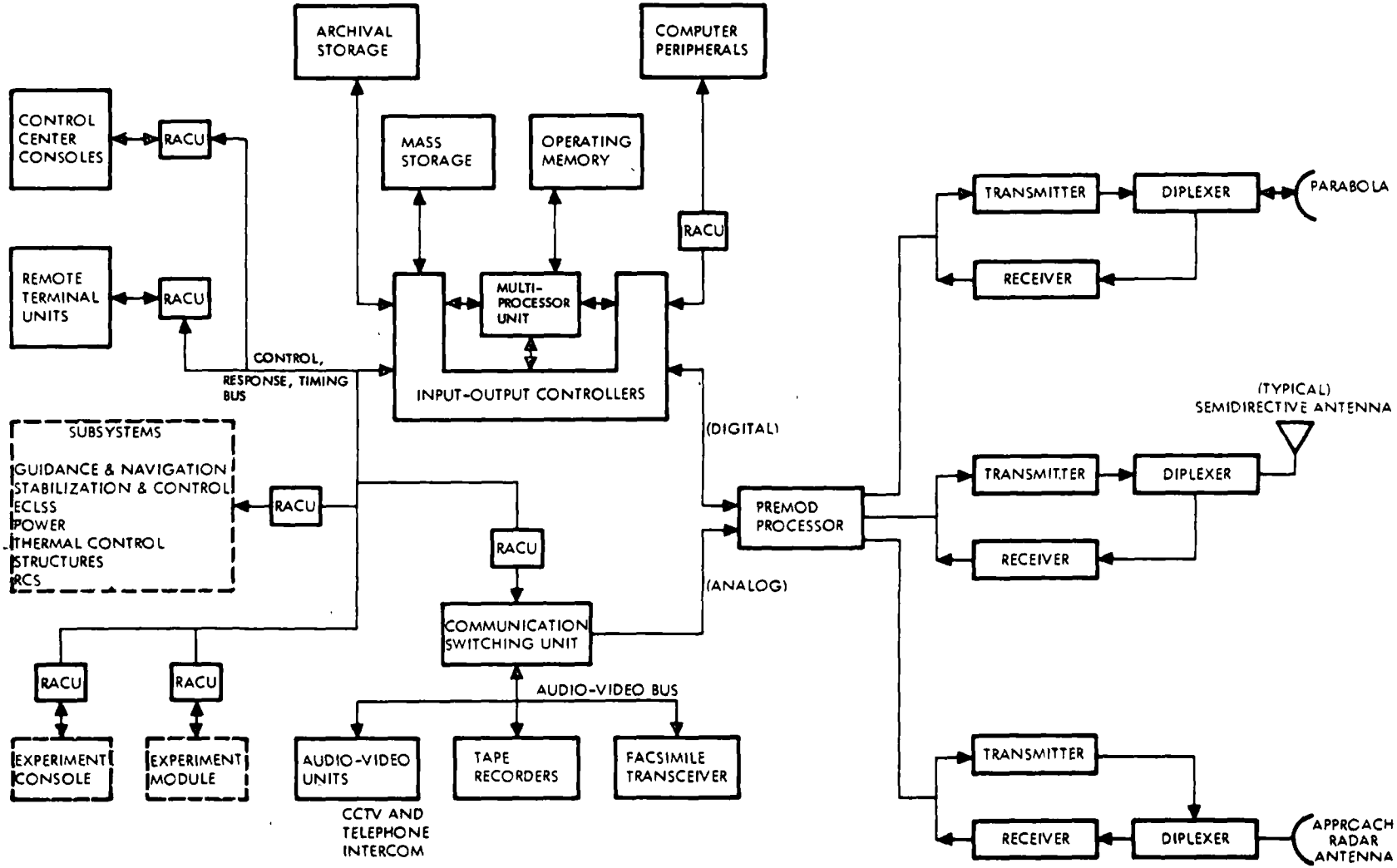


Figure 3.3-21. ISS Block Diagram

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Data Processing

The baseline data processor assembly is physically divided into two multiprocessors; one is located in the primary control center, and the other is in the experiment control center. The total task of station operations management and experiment data management would reside in their individual processors; however, either processor is capable of maintaining OLS operations.

A major sizing factor for the data processing portion of the ISS is the memory requirement. The actual processing equipment represents a small percentage of the overall weight and power. A minimum requirement would be sufficient memory for the basic station functions with no additional provisions for experiment support. The following indicates estimates for the required station operation storage and the additional storage required to support the experiment processing shown in the requirements:

Memory	Station/Subsystems	Experiment	Total
Operational	128K words	128K words	256K words
Mass	256K words	3 M words	3.3 M words
Archival	(2×10^9 bits/day)	(230×10^9 bits/day)	(232×10^9 bits/day)

NOTE: The archival is a function of dump capability to MSFN. The basic station requirement could be met with one 40-minute tape with a 12×10^9 bit capability, providing a dump schedule could be maintained.

Display and Control

The display and control assembly includes the consoles in the primary control center and the experiment control center. Multiple forms of data displays and operator controls provide adequate flexibility and capacity to handle four or more simultaneous tasks in the largest configuration. In addition, experiment-unique consoles designed to utilize the same form and flexibility of display and control, are provided for on the experiment deck. The experiment consoles are coupled to the data processor and communication and tracking assemblies in the same way as is the command console. However, these experiment consoles are not a portion of the ISS.

The minimum display and control capability would be essential station operation functions duplicated in each pressure volume with one set capable of full station operations. In the minimum configuration, experiment support would be performed on a noninterference basis with the maintenance and flight operation. The major areas of variation with crew size would be that associated with the limit on simultaneous operations and an assumed fewer number of possible isolated operating areas.

Internal Communications

Internal communications provide a data bus and an audio-visual bus for internal distribution of information. Processing for the bus interface also is performed. A bus consists of several lines for data, interrogation and control, and timing.

The electrical interface between the ISS and other OLS subsystems will be through standardized remote acquisition and control units (RACU). The RACU's are distributed throughout the OLS and transfer data from the subsystems and transfer commands to the subsystems. The RACU will accept data from the subsystem or experiment sensors and will digitize and/or format the data for transmission to the ISS processing assembly. The ISS will provide capabilities for command control, data acquisition, processing, storage, inventory, displays and alarms, checkout, and fault isolation required to support the subsystems.

The minimum internal communications configuration is estimated by elimination of all areas associated with convenience, entertainment, and division of work into subgroups of the total crew. Support of requirements would be by utilizing units with primary station functions. For example, TV cameras from docking ports could be used elsewhere when the ports are inactive.

Subsystem Summary

The estimated variation in the overall baseline subsystem from a minimum operational configuration to a 12-man crew size is shown in Table 3.3-44, while an overall summary is given in Table 3.3-45. The table assumes full experiment support capability at the 8-man level with some loss of simultaneous operation consistent with the reduced crew size from the 12-man evaluation.

The components generally represent the technological status assumed for the EOSS. As such, the evaluation will be conservative in certain areas. These are expected to be primarily associated with the high-gain antennas, the approach radar, information storage, and possible shifts in communication links to millimeter wave or laser bands.

Table 3.3-44. Information Subsystem Equipment Complement

Item	Unit Weight (lb)	Dimensions (inches)	Power-Watts Peak (Avg) (watts)	Minimum Operation			Crew Size 4			8			12		
				Q	W	PP/PA	Q	W	PP/PA	Q	W	PP/PA	Q	W	PP/PA
Data processor assembly															
Processor	10	9x3.6x14	30(3)	2	20	60/6	3	30	90/9	5	50	150/15	5	50	150/15
I/O controller	10	9x3.6x14	25(15)	2	20	50/30	3	30	75/45	5	50	125/75	5	50	125/75
Operation memory module	50	9x9x9	50(20)	4	200	200/80	4	200	200/80	8	400	400/160	8	400	400/160
Mass memory module	100	9x18x14	30(15)	1	100	30/15	2	200	60/30	13	1300	390/195	13	1300	390/195
Archival memory tape deck	120	18x18x14	135(33.3)	2	240	270/64	2	240	270/67	2	240	270/67	4	480	540/133
RACU - 256	6	3x3x8	8(6)	12	72	96/72	12	72	96/72	24	144	192/144	36	216	288/216
RACU - 128	5	3x3x4	4(3)	4	12	16/12	4	12	16/12	8	24	32/24	10	30	40/30
RACU - 64	1.5	3x3x2	2(1.5)	4	6	8/6	4	6	8/6	10	15	20/15	18	27	36/27
Central timing unit	18	9x6x14	30(30)	2	36	60/60	2	36	60/60	2	36	60/60	2	36	60/60
Display control assembly															
Console structure and fittings	83	48x48x24		2	166		2	166		3	249		4	332	
Printer	40	(1) ↑	100(20)	0	0		2	80	200/40	3	120	300/60	4	160	400/80
Display add keyboard	40	(2) ( )	7(1.75)	3	18	21/5	4	24	28/7	6	36	42/10	8	48	56/14
Function select	15	(2)	5(1.25)	4	60	20/5	4	60	20/5	6	90	30/8	8	120	40/10
Light emitting diode (LED) display	10	(2) indicates the max.	20(10)	4	40	80/40	4	40	80/40	6	60	120/60	8	80	160/80
LED display electro.	20	(2) number	30(15)	4	80	120/60	4	80	120/60	6	120	180/90	8	160	240/120
CRT	25	(1) of units	50(50)	2	50	100/100	2	50	100/100	3	75	150/150	4	100	200/200
CRT electronics	20	(1) in one	10(10)	2	40	20/20	2	40	20/20	3	60	30/30	4	80	40/40
Microfilm viewer	35	(1) console	50(25)	0	0		1	40	50/25	3	105	150/75	4	140	200/100
Hand control stick	20	(2) structure	1(.15)	2	40	2/1	4	80	4/1	6	120	6/1	8	160	8/1.2
Monitor and alarm	30	(1) ↓	89(45)	2	60	178/90	2	60	178/90	3	90	267/135	4	120	356/180
Remote terminal unit	40	12x12x18	250(25)	1	40	250/25	1	40	250/25	2	80	500/50	4	160	1000/100
Microfilm projector	35	15x15x12	50(25)	0	0		0	0		0	0		2	70	100/50
Communications tracking assy.															
Parabolic Antenna*	130	13 feet	100(75)	2	260	200/150	2	260	200/150	2	260	200/150	2	260	200/150
Semidirect	1	6(dia)x8	0	8	8	0/0	8	8	0/0	8	8	0/0	8	8	0/0
Approach radar**	25	12x12x12	125(15)	2	50	250/30	2	50	250/30	2	50	250/30	2	50	250/30
Transponder power amplifier and diplexer	29		120(36)	12	348	1440/440	12	348	1440/440	12	348	1440/440	12	348	1440/440
Premodulator processor	50	12x12x14	50(50)	1	50	50/50	1	50	50/50	1	50	50/50	1	50	50/50
Facsimile unit	40	18x12x14	200(20)	0	0		0	0		1	40	200/20	2	80	400/40
Central switch unit and self-test	120	18x27x14	70(50)	1	120	70/50	1	120	70/50	2	240	140/100	2	240	140/100
Audio/Video units - A	6	6x6x2	9(5)	10	60	90/50	10	60	90/50	20	120	180/100	40	240	360/200
Audio/Video units - B	18	6x10x2	12(12)	2	36	24/24	2	36	24/24	3	54	36/36	4	72	48/48
Paging amplifier	10	6x6x8	100(10)	0	0		0	0		2	20	200/20	2	20	200/20
Peripheral group															
Voice recorder	120		135(67.5)	0	0		1	120	135/68	3	360	405/204	6	720	810/405
Audio recorder	15		20(10)	0	0		1	15	20/10	2	30	40/20	2	30	40/20
Entertainment recorder	15		20(20)	0	0		0	0		1	15	20/20	2	30	40/40
Video cameras (color)	10		50(10)	3	30	150/30	7	70	350/70	9	90	450/90	13	130	650/130
Video monitors (9-inch color)	15		50(10)	3	45	150/30	5	75	250/50	12	180	600/120	20	300	1000/200
Video monitors (14-inch color)	40		75(15)	0	0		0	0		2	80	150/30	4	160	300/60
			Total		2307	4005/1548		2793	4804/1786		5409	7775/2854		7057	10,757/3,820

* Includes gimbal drive electronics - 13-foot baseline

** Includes RF electronics

*** Q = quantity
W = weight (lb)
PP/PA = peak power/average power (watts)

Table 3.3-45. Information Subsystem Summary

Item	Minimum Operation			Crew Size								
				4			8			12		
	Total Weight (lb)	Peak Power Total PPT (watts)	Average Power Total PPT (watts)	WT	PPT	PAT	WT	PPT	PAT	WT	PPT	PAT
Data processor	706	790	348	826	875	381	2259	1639	755	2589	2029	911
Display and control	594	791	346	755	1050	413	1205	1775	669	1730	2800	976
Communication tracking ass	932	2124	794	932	2124	794	1190	2696	946	1368	3088	1078
Peripheral group	75	300	60	280	755	198	755	1665	484	1370	2840	855
ISS Interconnecting cable	600			720			850			920		

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3.4 GUIDANCE AND CONTROL SUBSYSTEM

The guidance and control (G&C) subsystem equipment provides sensing, computation, and torque generation for the OLS. The choices of sensors and computation equipment are relatively insensitive to vehicle configuration and flight modes. The torquer choice, on the other hand, is strongly dependent upon these factors; and, in turn, the choice of flight mode and vehicle configuration is heavily influenced by the impact of torquer selection on subsystem weight and propellant resupply weight. Torquer selection consists of the choice between the reaction control subsystem (RCS) used alone or in conjunction with momentum exchange devices such as control moment gyros (CMG's) or reaction wheels. The G&C subsystem weight, power, and volume and RCS propellant requirements are presented parametrically in this section for several combinations of vehicle configuration and flight mode.

3.4.1 ASSUMPTIONS AND MODELS

A digital computer program developed for the Earth Orbiting Space Station (EOSS) analysis was used to generate parametric data for various OLS configurations and flight modes. The disturbance torques considered were man-motion, gravity gradient, and nonprincipal axes rotations. The program integrated these torques to obtain the impulse or momentum changes to be countered by the vehicle either by an RCS system alone or by the RCS plus momentum exchange devices. Orbit maintenance maneuvers were assumed to occur frequently enough to allow the vehicle to remain in an essentially 60-nautical miles circular lunar orbit. No consideration was given to combining the required rotation and orbit maintenance translation maneuvers. The propellant consumption requirements derived are for parametric comparison only and should not be used for sizing of propellant storage facilities.

Man-motion is assumed to produce a transient disturbance of 750-foot-pound-seconds five times per orbit. (While this disturbance is somewhat large and may be considered highly conservative, it may be readily altered in the parametric data because it represents an additive 3750-foot-pound-seconds of impulse in the RCS-only budget and 750-foot-pound-seconds in CMG sizing.)

Gravity-gradient and nonprincipal axis rotation torques are produced by a misalignment of the vehicle principal axes with the local vertical coordinate system. Part of this misalignment is deterministic and intentional to enable experiment sensors to be designed for rigid mounting with respect to the vehicle geometric axes. The alternative is to attempt to fly with

the vehicle principal axes aligned with the local vertical axes and gimbal the experiment sensors with respect to the vehicle body axes as a function of the mass distribution of attached modules, etc. This alternative approach was not adopted because the alignment of vehicle principal axes with the local vertical employing rigid mounted experiment sensors allows for more flexible design of experiments. In addition to the deterministic misalignments, an uncertainty of 0.5 degree in the principal axis location and control and sensor bias errors is assumed. For a rigid vehicle holding a constant attitude with respect to a local level coordinate system, these torques are constant in body axes. The torques are periodic in body axes, if the vehicle is holding a constant attitude with respect to an inertial coordinate system. For this parametric analysis all of the OLS configurations considered are assumed to utilize 2-degree of freedom solar panels to produce electrical power; these panels are assumed to be held with a fixed orientation in inertial coordinates (with respect to the sun line). These panels have sizable inertias and experience significant gravity-gradient torques. Because it is assumed that the panels are controlled to point toward the sun regardless of disturbances torques, the resultant torques also are experienced by the vehicle and must be countered by RCS propellant or momentum exchange devices on the vehicle. The total vehicle (i.e., main vehicle plus solar panels) must then be considered as a rigid body with respect to gravitational disturbances at any instant of time even though the main vehicle may be holding a constant attitude with respect to local vertical coordinates while the panels are held fixed in inertial space. (Inertias are different for vehicle maneuvers than for gravity-gradient disturbances.)

The RCS engines are assumed to be mounted on the vehicle and are assumed to be used in the RCS-only mode to counter disturbances and keep the vehicle attitude within some deadband (a $\pm$ 0.1-degree deadband is assumed). In the RCS plus momentum exchange configuration, they are used to desaturate the momentum exchange devices. In the RCS-only configuration, RCS impulse is used to counteract all disturbances and limit cycle motion. The man-motion disturbance is 3750 foot-pound-seconds as previously discussed. Because the RCS system alone does not have the capability to store momentum, the absolute value of the disturbances is countered in body axes. In the absence of disturbances, the vehicle will limit cycle within the deadband at a rate determined by the RCS minimum impulse, lever arm, deadband size, and vehicle inertia. For a symmetrical limit cycle (worst-case), the per-axis-average torque produced by the RCS jets is given by

$$T_{LIM} = \frac{p^2 \ell^2}{4 \theta_{DB} I}$$

where p is the minimum impulse (in pound-seconds), ℓ is the lever arm (in feet), $\pm$ θ_{DB} is the deadband (in radians) and I is the axial inertia (in slug-feet^2). The angular impulse expenditure is simply the integral of this average torque. In the presence of disturbance, two cases must be considered: (1) if the disturbances are large enough to cause the vehicle to "hang up" on one side of the deadband, in this case the RCS torque is equal to the average of

the absolute value of the disturbance torques (in body axes); and (2) if the disturbances are not large enough to "hang up" the vehicle on one side of the deadband, in this case the average RCS torque is equal to the average of the absolute value of the disturbance plus the limit cycle torque. The condition for hang up is given by

$$T_{DIST} > \frac{p^2 \ell^2}{16 \theta_{DB} I}$$

where the symbols are the same as previously shown. In the program, p is assumed to be 0.1 pound-seconds, θ_{DB} = 0.1 degree, and ℓ and I are determined by the vehicle configuration. The integral (in body axes) of the average RCS torque previously described divided by the appropriate lever arm over one orbit is denoted in the following data as RCS-only impulse and represents the RCS impulse required per orbit if momentum exchange devices are not used.

Momentum Exchange Concepts

Momentum exchange devices can be of several types: reaction wheels, double gimbal CMG arrays, skewed single gimbal CMG arrays, V pairs, and hybrid arrays. Choice of one of these types is highly dependent upon the performance and optimization to a specific momentum history. A planar array of three double gimbal CMG's is chosen as representative for this study based on experience with the EOSS. If vehicle motion is to be kept constant, disturbance torques acting on the vehicle must be countered by the CMG's or by RCS impulse expenditure. If the CMG momentum storage capacity is large enough, a periodic momentum variation (in inertial coordinates) can be absorbed by the CMG's without expenditure of RCS propellant; however, if secular torques give rise to a momentum buildup, the CMG's will saturate and the eventual expenditure of RCS propellant will be necessary. The amount of impulse required per orbit is denoted as "desaturation impulse" and represents the RCS impulse required per orbit to counter the secular momentum buildup. If continual desaturation of the CMG's by the RCS is not desirable, the CMG's must be sized to store not only the periodic momentum but also the secular momentum buildup for the duration between desaturation maneuvers. For this study, it is assumed that the CMG's will be desaturated at least every third orbit: the CMG's are, therefore, sized for three orbits of momentum history. The program integrates the disturbance torques in inertial coordinates to determine the momentum history. The total momentum storage capacity of the CMG's is the maximum value of momentum vector magnitude, if the CMG's are initially biased to the center of the expected momentum envelope. It is assumed that any two of the three gyros in the array must be capable of storing this momentum to provide design margin and to allow repair of a malfunctional gyro. Thus, each gyro is sized to provide one half of this

momentum. This value is referred to as "individual gyro size" and represents the momentum capacity for each of the three gyros.

CMG Design

Design of CMG's is an involved process and requires consideration of a large number of factors for an individual mission. The following empirical equations and design data were taken as representative of OLS requirements based on EOSS experience. The weight of an individual gyro is given by

$$W = 7.7 \sqrt{H}$$

where H is the momentum in foot-pound-seconds and W is the unit weight in pounds. The total array weight is three times the individual gyro weight plus a structural mounting provision of 10 percent. The weight of electronics required per gyro is given by

$$W_E = 20 \sqrt[4]{\frac{H}{1000}}$$

The total weight is then the sum of gyros plus electronics plus a spares allowance of 20 percent. This value is referred to as "3-gyro weight" and represents the total weight of the CMG system including gyros, mounting, electronics, and spares.

The diameter (D) of the gyro rotor in inches is given by

$$D = 3.2 \sqrt[4]{H}$$

The required volume for each gyro is a box of width W = 1.8D and length L = 2.2D. The total volume is given by

$$V = 3V_I = 3\ (2.2D)\ (1.8D)^2$$

where V_I represents individual gyro volume.

The gyro average power is given by

$$P_I = 7 \sqrt[4]{H}$$

and total power equals three times the individual gyro power.

Because there will always be periodic momentum components, CMG's can always be used to counter these components to save RCS propellant. If the mission duration is long enough, the amount of propellant saved will eventually be greater than the weight of the CMG's. Therefore, a total weight-saving can usually be achieved through the use of CMG's. The mission time at which and equivalent amount of RCS propellant has been expended for both RCS-only and RCS-CMG configuration is given by

$$t_{eq} = \frac{W_G \, I_{SP}}{(I_p - I_{DES}) \, \tau_o}$$

where t_{eq} is the time in days, W_G is the CMG weight in pounds, I_p is the RCS only impulse (pounds-seconds) expenditure per orbit, I_{DES} is the desaturation impulse per orbit, and τ_o is the number of orbits per day. This time is a measure of the efficacy of the CMG's and is given as "time to equivalence."

It is not suggested that CMG's should always be used. Very large momentum requirements probably dictate the use of several smaller CMG's rather than one large one with the attendent increase in weight, power, volume, and complexity. Very large CMG size indicates that the particular vehicle configuration should not be flown in that particular flight mode for long duration. The primary function of the CMG's is to store periodic momentum not to delay desaturation maneuvers. If large momentum storage is required because of large secular torques, consideration should be given to sizing the CMG's to store only periodic momentum and requiring frequent use of the RCS to remove secular momentum.

Seven vehicle configurations were investigated. After examination of the data from the seven vehicle configurations and nine flight modes, it was evident from the spread of data values that it was not feasible to cross-plot the information in terms of simple parameters. Because the optimal design for a chosen vehicle configuration is not likely to be one of the data points generated even if all presently defined vehicle configurations were investigated, it was decided to terminate data generation with the seven vehicles listed.

This type of investigation will be repeated for the chosen configurations in the next phase of the study and will include the effects of mission timeline and tug operations. (The presence or absence of tugs and percent propellant loading of tugs can greatly change vehicle inertias and thus gyro size and RCS propellants.)

The characteristics of the configurations investigated are given in Section 2.0. These configurations consist of: four-man 22-foot-diameter (4-22), four-man 33-foot-diameter (4-33), eight-man 22-foot-diameter (8-22), eight-man 33-foot-diameter (8-33), twelve-man 22-foot-diameter (12-22), twelve-man 33-foot-diameter (12-33), and twelve-man 15-foot-diameter cruciform (CRUC). Each of these configurations was investigated in each of nine flight modes. The local level flight modes are designated as X-POP-Y for

example, meaning X axis perpendicular to the orbit plane, Y axis in the direction of the velocity vector. The local level modes are X-POP-Y, Y-POP-X, Z-POP-X, X-POP-Z, Y-POP-Z, Z-POP-Y. The inertial modes are X-POP, Y-POP, Z-POP.

3.4.2 ILLUSTRATION OF DATA GENERATION AND USAGE

The best explanation of what the data means is to consider a specific example. For this purpose, the eight-man 33-feet-diameter vehicle flying in Z-POP-Y level mode is chosen. The computer generated momentum history for three orbits is shown on Figure 3.4-1. The figure shows that the X and Z components (in plane) are mostly periodic. The X component has a slight secular buildup of 141 foot-pound-seconds per orbit. This is caused by the solar panel torques, which are held inertially fixed - the buildup is almost entirely in the X-axis because of the initial orientation with respect to the sun. The Y-axis component is small and almost all secular with a small periodic component. The secular buildup is -54 foot-pound-seconds per orbit. The maximum vector magnitude occurred at the end of the third orbit with $H_X = 474$, $H_Y = 90$, and $H_Z = -300$. This magnitude is 568 foot-pound-seconds. If on top of this is added a crew motion disturbance of 750 foot-pound-seconds (which in fact can occur in any axis at any time), the result would be 1318 foot-pound-seconds maximum momentum. For two gyros out of three to supply this, each must supply 659 foot-pound-seconds, and adding a 15-percent margin results in 758 foot-pounds. The average jet moment arm is about 20 feet, so the desaturation impulse is 141/20 + 54/20 or 195/20 pound-seconds or 9.75 pound-seconds.

The gyro weight is

$$W_G = 7.7 \sqrt{H} \times 3 = 637.6 \text{ pounds}$$

The electronics weight is

$$W_E = 20 \sqrt[4]{\frac{H}{1000}} = 18.75 \text{ pounds}$$

adding the spares weight of 20 percent results in

$$W_G = 834 \text{ pounds}$$

If an RCS specific impulse of 419 seconds is assumed, the desaturation propellant per day is 0.235 pound. The RCS-only impulse per orbit computed by the program 371 pound-seconds or 8.85 pounds per day. The RCS-only propellant weight would, therefore, equal the CMG weight plus desaturation propellant weight in 97 days.

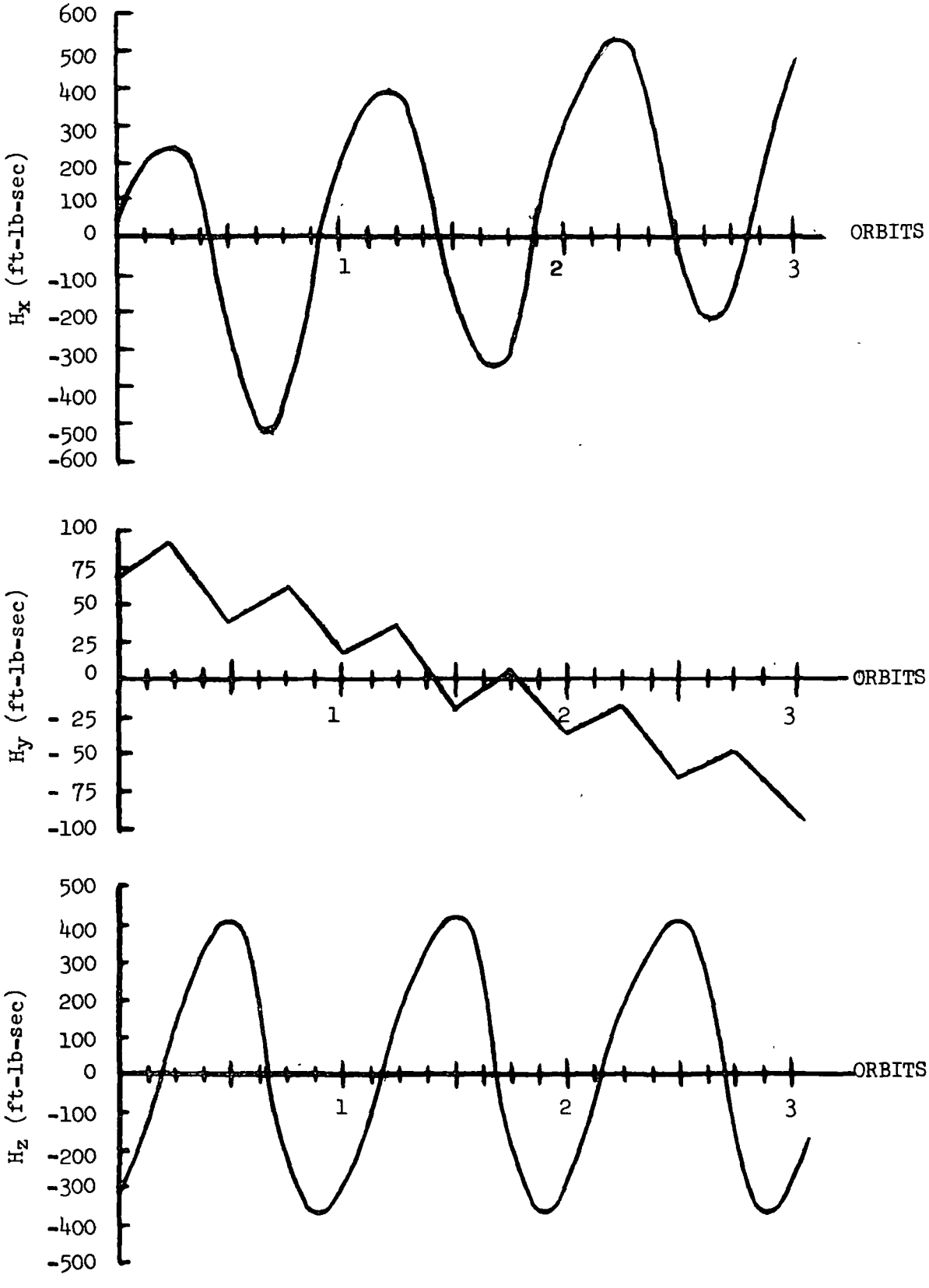


Figure 3.4-1. Momentum History (Inertial Coordinates)
Vehicle 8-33 Z-POP-Y Level Mode

3.4.3 PARAMETRIC DATA

Tables 3.4-1 through 3.4-7 present propellant consumption and gyro sizing data for the seven vehicles discussed above. From Table 3.4-1 it can be seen that the four-man 22-foot-diameter vehicle should be flown in the Z-POP-Y level mode to conserve propellant and minimize gyro size. If an inertial mode is desired, the X-POP mode requires the least propellant and smallest CMG's. The Y-POP mode seems the least desirable if CMG's are used. Table 3.4-2 indicates the same is true for the four-man 33-foot-diameter vehicle. Z-POP-Y is also the best level mode for the eight-man 22-foot-diameter vehicle as can be seen from Table 3.4-3, but Z-POP is the preferred inertial mode in terms of CMG sizing. If CMG's are not used, X-POP is the preferred inertial mode. Table 3.4-4 indicates that Z-POP-Y also is the preferred level mode for the eight-man 33-foot-diameter vehicle, and that X-POP is the preferred inertial mode. Table 3.4-5 indicates that if CMG's are used, X-POP-Z is the preferred level mode for the twelve-man 22-foot-diameter vehicle and X-POP is the preferred inertial mode. (The Z-POP mode that has a large residual impulse requirement should be avoided.) If CMG's are not used, X-POP-Y is the preferred level mode. Table 3.4-6 indicates that for the twelve-man 33-foot-diameter vehicle, X-POP-Y is the preferred level mode if CMG's are not used; however, X-POP-Z may be preferred if CMG's are used. X-POP is the preferred inertial mode. (Z-POP modes should be avoided if CMG's are used.) Table 3.4-7 indicates that Z-POP-Y is the preferred level mode and X-POP the preferred inertial mode for the twelve-man 15-foot-diameter cruciform vehicle.

3.4.4 MECHANIZATION CONSIDERATIONS

Figure 3.4-2 illustrates the makeup of the G&C Subsystem required to perform the necessary functions of guidance, navigation and control. Although the figure shows a functional partitioning into two areas--guidance and navigation and attitude control, the subsystem itself is not partioned in this manner by mechanization (e.g., the same sensors are used for navigation as for control).

Trade studies were conducted in the areas of inertial reference equipment and torquers. The computation and sensor mechanizations will be chosen based on commonality of equipment.

Inertial Reference

The guidance and navigation functions necessitate an inertial reference that is basically a system of accelerometers and sensing gyroscopes that serve as the basis for measuring angular and linear velocity changes. The trade study discussed in the following paragraphs was conducted in the EOSS study. The data is equally applicable to the OLS. It compares a strapdown reference with a conventional gimbaled platform reference. The strapdown concept utilizes single-degree-of-freedom gyroscopes and accelerometers mounted directly to the vehicle, so that their measurements are performed in a "body-fixed" reference frame. Transformations to an inertial frame, for example, are performed by a computer. The strapdown concept evaluated in this study employs a nonorthogonal sensor alignment in which the input

Table 3.4-1. RCS and CMG Data, Vehicle 4-22

Mode	RCS Only Impulse (lb-sec)	Desaturation Impulse (lb-sec)	Individual Gyro Size (ft-lb-sec)	3-Gyro Weight (lb)	3-Gyro Power (watts)	3-Gyro Volume (ft^3)	Days to Equivalence (days)
X-POP-Y	564	142	1923	1302	139	132	130
Y-POP-X	588	216	2700	1535	151	169	173
Z-POP-X	509	105	1445	1334	129	107	118
X-POP-Z	516	130	1935	1306	139	133	142
Y-POP-Z	501	206	2698	1534	151	169	219
Z-POP-Y	510	211	751	828	109	66	71
X-POP Inertial	513	198	2562	1496	149	163	200
Y-POP Inertial	870	497	5981	2259	185	304	254
Z-POP Inertial	608	206	2587	1503	150	164	157

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Table 3.4-2. RCS and CMG Data, Vehicle 4-33

Mode	RCS Only Impulse (lb-sec)	Desaturation Impulse (lb-sec)	Individual Gyro Size (ft-lb-sec)	3-Gyro Weight (lb)	3-Gyro Power (watts)	3-Gyro Volume (ft^3)	Days to Equivalence (days)
X-POP-Y	613	156	2084	1354	142	140	124
Y-POP-X	619	256	3173	1660	158	190	192
Z-POP-X	559	133	1762	1248	136	124	123
X-POP-Z	563	144	2101	1359	142	141	137
Y-POP-Z	542	247	3175	1660	158	191	237
Z-POP-Y	545	22	783	845	111	68	68
X-POP Inertial	542	219	2801	1562	153	173	203
Y-POP Inertial	990	599	7172	2468	193	347	265
Z-POP Inertial	699	275	3387	1713	160	200	169

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Table 3.4-3. RCS and CMG Data, Vehicle 8-22

Mode	RCS Only Impulse (lb-sec)	Desaturation Impulse (lb-sec)	Individual Gyro Size (ft-lb-sec)	3-Gyro Weight (lb)	3-Gyro Power (watts)	3-Gyro Volume (ft^3)	Days to Equivalence (days)
X-POP-Y	258	84	2590	1504	150	164	362
Y-POP-X	290	142	3478	1735	161	204	494
Z-POP-X	240	69	1996	1326	140	136	325
X-POP-Z	235	77	2613	1511	150	165	402
Y-POP-Z	233	111	3176	1660	158	191	568
Z-POP-Y	234	60	763	834	110	67	202
X-POP Inertial	244	120	3619	1769	163	210	602
Y-POP Inertial	426	279	7189	2470	193	348	706
Z-POP Inertial	267	97	2423	1456	147	156	361

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Table 3.4-4. RCS and CMG Data, Vehicle 8-33

Mode	RCS Only Impulse (lb-sec)	Desaturation Impulse (lb-sec)	Individual Gyro Size (ft-lb-sec)	3-Gyro Weight (lb)	3-Gyro Power (watts)	3-Gyro Volume (ft^3)	Days to Equivalence (days)
X-POP-Y	419	116	2267	1410	145	149	195
X-POP-X	487	243	3718	1792	164	214	308
Z-POP-X	421	144	2299	1420	145	150	215
X-POP-Z	386	105	2280	1414	145	149	211
Y-POP-Z	389	192	3444	1727	161	202	369
Z-POP-Y	371	10	758	832	110	67	97
X-POP Inertial	373	165	3126	1648	157	188	332
Y-POP Inertial	727	483	7842	2578	198	371	445
Z-POP Inertial	513	239	3636	1773	163	211	272

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Table 3.4-5. RCS and CMG Data, Vehicle 12-22

Mode	RCS Only Impulse (lb-sec)	Desaturation Impulse (lb-sec)	Individual Gyro Size (ft-lb-sec)	3-Gyro Weight (lb)	3-Gyro Power (watts)	3-Gyro Volume (ft^3)	Days to Equivalence (days)
X-POP-Y	777	86	4096	1879	168	230	114
Y-POP-X	843	166	5962	2255	185	303	140
Z-POP-X	1897	1749	60436	7020	329	1679	2003
X-POP-Z	1332	79	5493	2167	181	285	73
Y-POP-Z	1315	113	5751	2216	183	295	77
Z-POP-Y	1842	1671	60112	7001	329	1673	1715
X-POP Inertial	787	128	5317	2133	179	278	136
Y-POP Inertial	969	294	9286	2800	206	420	174
Z-POP Inertial	1905	1760	60786	7040	330	1687	2039

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Table 3.4-6. RCS and CMG Data, Vehicle 12-33

Mode	RCS Only Impulse (lb-sec)	Desaturation Impulse (lb-sec)	Individual Gyro Size (ft-lb-sec)	3-Gyro Weight (lb)	3-Gyro Power (watts)	3-Gyro Volume (ft^3)	Days to Equivalence (days)
X-POP-Y	776	97	2855	1577	154	176	98
Y-POP-X	883	260	5194	2109	178	274	142
Z-POP-X	1660	1427	30916	5048	278	1021	910
X-POP-Z	1126	87	3237	1676	158	193	68
Y-POP-Z	1154	174	4301	1924	170	238	82
Z-POP-Y	1525	1268	29635	4944	276	990	807
X-POP Inertial	738	144	3744	1799	164	215	127
Y-POP Inertial	1080	460	8951	2750	204	409	186
Z-POP Inertial	1730	1500	32595	5180	282	1062	947

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Table 3.4-7. RCS and CMG Data, 12-Man 15-foot-diameter cruciform

Mode	RCS Only Impulse (lb-sec)	Desaturation Impulse (lb-sec)	Individual Gyro Size (ft-lb-sec)	3-Gyro Weight (lb)	3-Gyro Power (watts)	3-Gyro Volume (ft^3)	Days to Equivalence (days)
X-POP-Y	257	71	2439	1461	148	157	331
Y-POP-X	325	174	4349	1934	171	240	539
Z-POP-X	285	115	3563	1756	162	207	433
X-POP-Z	236	64	2445	1463	148	157	356
Y-POP-Z	224	107	3052	1629	156	185	586
Z-POP-Y	205	17	1043	970	119	84	216
X-POP Inertial	228	101	3296	1690	159	196	559
Y-POP Inertial	438	288	7159	2466	193	347	691
Z-Pop Inertial	309	140	3790	1809	165	217	452

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axes of six gyros are aligned along the normals to the faces of a regular dodecahedron. This arrangement provides satisfactory performance with any three gyros working and is in fact more reliable than an orthogonal arrangement of nine gyros.

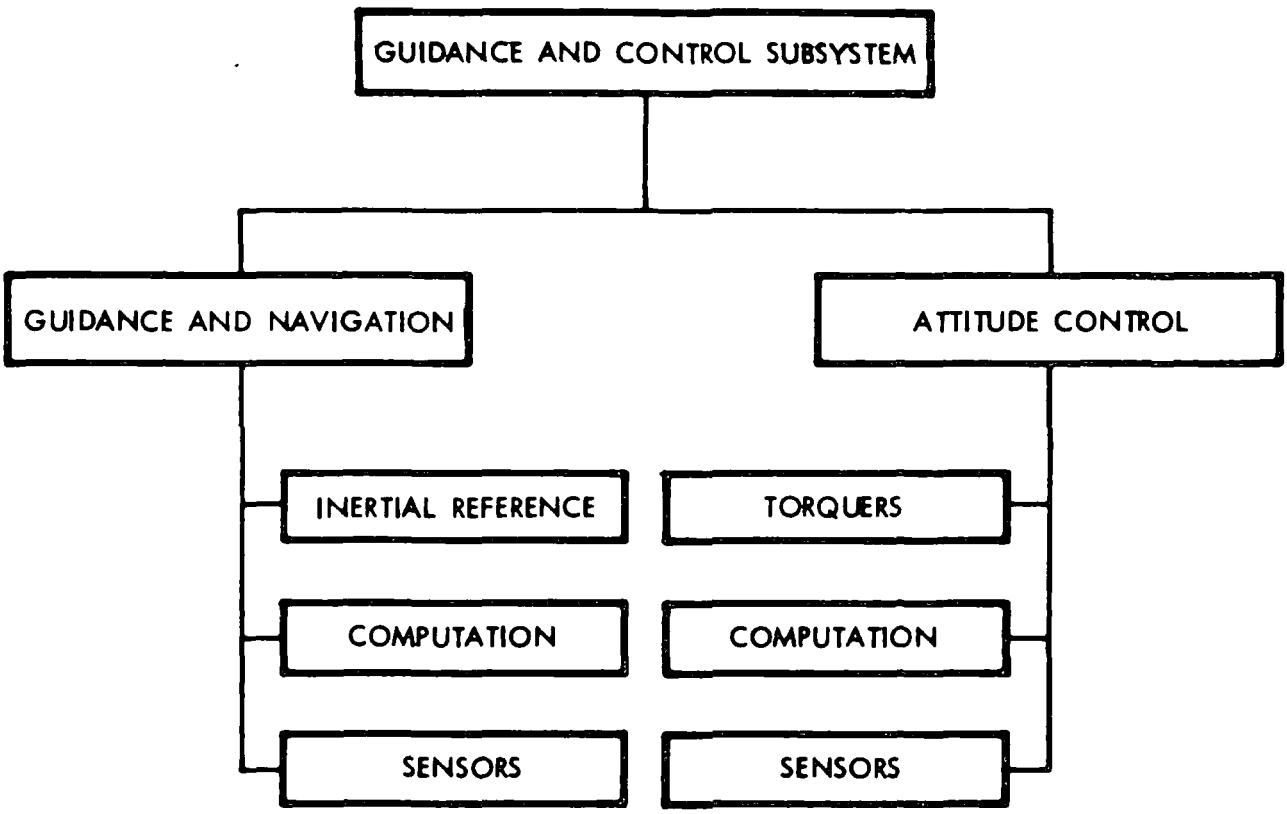


Figure 3.4-2 Guidance and Control Subsystem Elements

A four-gimbaled platform concept also was considered to provide an inertial reference. In this concept, the sensors (gyros and accelerometers) are mounted on a stable member, which is isolated from the vehicle by a set of gimbals. The stable member is then held fixed with respect to an inertial frame of reference. This concept is used as the basis for aircraft navigation systems; the Apollo command module, for example, contains a three-gimbal platform.

Computation

As indicated in Figure 3.4-2, there are computation functions both for guidance and navigation and attitude control. There are three approaches possible for mechanizing these computations. First, a dedicated approach in which a computer is used exclusively for G&C computations. Second, an integrated approach in which the G&C computations are performed by the information subsystem. Third, a hybrid approach in which some computations are performed within the G&C equipment in a dedicated fashion and others are performed by the information subsystem. There is no G&C preference to which approach is chosen. The computer mechanization trades are presented in the Information Management Subsystem section (Section 3.3).

Sensors

To perform the state vector estimation function autonomously, the G&C subsystem must include equipment to perform navigation measurements. The basic navigation measurement applicable to lunar orbit is the star-horizon included angle measurement. The accuracy of this technique is improved considerably by the addition of landmark tracking. A sextant/telescope is provided as part of the G&C equipment for manual landmark measurements, and an automatic landmark tracker is included for automatic measurements. Star trackers and horizon scanners are used both for the navigation measurement and for attitude control sensing. The star trackers are the same as those used for Earth Orbit Space Station (EOSS). The horizon scanners are also the same as those used for earth orbit, but they are designed to sense infrared radiation in the 40-micron region rather than the 14-micron region (because of the lack of lunar atmosphere, the spectral composition of the infrared horizon is different than for the earth). A radar altimeter also is used to provide navigation data on the dark side of the moon, where optical land marks are not available.

The navigation equipment is essentially the same as for earth orbit navigation with the addition of the automatic landmark tracker and the radar altimeter. The additional equipment is required to provide equivalent pointing accuracy for experiments at the lower orbital altitude (60 n mi, as opposed to 240 n mi). The only feasible alternate scheme is to plant well surveyed radio beacons on the lunar surface for OLS tracking. Maintenance of these beacons, however, would be a very burdensome and expensive operation. The automatic landmark tracking technique is well-suited to lunar orbit operation because of the clarity and definition of landmarks and the lack of obscuration by clouds. The tracker is essentially similar to a star tracker, and the software can be mechanized to track either known or unknown landmarks.

In lunar orbit there is no lack of a sufficient number of landmarks (although only day side tracking is possible). The major problem is the acquisition and identification of known landmarks. This problem is mitigated by using the star horizon measurements for gross navigation. Automatic acquisition then can be accomplished by comparing the scan of a relatively small area of uncertainty with prestored landmark data-using pattern correlation techniques. Shadow patterns and viewing aspect angle changes

must of course be accounted for during acquisition and identification. The landmark tracker can also be mechanized to acquire an unidentified landmark with desirable geometric properties, but this should not be necessary because of the abundance of known landmarks. Once acquisition of the landmark has been accomplished, tracking can be performed using scan-to-scan correlation techniques to minimize the computation burden.

The only other sensor used is a scanning laser radar for docking, which is identical to that used on the EOSS.

Torquers

The torquer mechanization trade consists of a choice between RCS only and RCS plus momentum exchange devices. This choice is discussed parametrically in the previous paragraphs. Momentum exchange devices may be either CMG's or reaction wheels. The parametric data are based on CMG's because they offer greater flexibility and quicker response time. Reaction wheels would be designed to a specific mission and vehicle and would probably not be acceptable for artificial-gravitation operation; they would also not supply the response time required to absorb crew motion disturbances without causing RCS firings. CMG's would also require less power and would weigh less than reaction wheels.

Inertial Reference Trade

An inertial reference trade study was performed for the EOSS. The details of this trade are applicable for the most part to the OLS and are repeated here for completeness. The major difference between the trades for the EOSS and the OLS lies in the difference in transportation cost to lunar orbit rather than to earth orbit. (A cost of $150 per pound was assumed for earth orbit, whereas a cost of $1000 per pound is assumed for lunar orbit.)

Although actual hardware characteristics are used in the trade, it is not meant to imply that this hardware should be chosen. Particular hardware is used as a basis of comparison of mechanization types; the actual hardware should be chosen based on the state of the art, the relative development of candidate systems, and commonality with other vehicles at the time the choice is made.

Tradeoff Criteria

Both mechanizations considered were required to meet the absolute criteria of required mission performance and safety. These mechanizations were then compared against each other on the basis of relative criteria. The relative criteria were weighted 70 percent for program cost factors and 30 percent for mission performance factors. These categories were divided into subcategories, each of which was assigned a maximum number of points in that subcategory and the other mechanizations assigned a number of points proportionate to its relative "goodness" in that category. For example, in the subcategory weight valued at a maximum of 10 points, mechanizations A and B--weighing one pound, and two pounds, respectively would be valued

at 10 points (A) and 5 points (b).

The program cost factor was divided into the following subcategories and weights:

1.	Equipment cost	15
2.	Ease of maintenance	20
3.	Development risk	10
4.	External interfaces	5
5.	Size	5
6.	Weight	10
7.	Power	5
	Subtotal for program cost	70

Mission performance was weighted as follows:

1.	Reliability	10
2.	Performance accuracy	5
3.	Versatility	5
4.	Crew workload	10
	Subtotal for mission performance	30

Equipment Cost

The recurring, nonrecurring, and sustaining costs are considered in this category. The nonrecurring and recurring costs are evaluated for a 10-year mission based on the assembly cost to orbit, component cost to orbit, expected wearout rate, long life, reliability, and performance requirements. Sustaining costs include those of ground checkout, repair and replacement, and could equal or exceed the recurring and nonrecurring costs, depending upon the method of implementation.

For comparison, each assembly is considered to be made up of substantially similar subassemblies. In order to determine the costs of spares and replacements, the expected assembly costs have to be broken down into subassembly costs as follows:

Subassemblies	Strapdown	Platform
Gyro	$ 20,000	$ 10,000
Gyro electronics	$ 10,000	$ 5,000
Accelerometer	$ 4,000	$ 4,000
Accelerometer electronics	$ 10,000	$ 10,000
Power supply and temperature control	$ 5,000	$ 5,000
Mounting base	$ 5,000	--
Gimbal unit	--	$ 15,000
Gimbal replacement computer	$ 50,000	--
A-D converter		$ 10,000

Neglecting qualification and development costs, the assembly cost can be computed as shown in Table 3.4-8.

Table 3.4-8. Assembly Cost Breakdown

Subassembly	Strapdown		Platform	
	No. Required	Total Cost	No. Required	Total Cost
Gyros	6	$120,000	3	$ 30,000
Gyro electronics	6	60,000	3	15,000
Accelerometer	6	24,000	3	12,000
Accelerometer electronics	6	60,000	3	15,000
Power supply	1	5,000	1	5,000
Mounting base	1	5,000	0	
Gimbal unit	0		1	15,000
GRC	1	45,000	0	
A-D converter	0		1	10,000
Assembly and checkout		8,000		8,000
Subtotal		$327,000		$110,000
10% Fee		32,700		11,000
Total		$359,700		$121,000

To determine the amount of equipment on board, the criticality of the item must be established. In the case of the inertial reference, which is used to support docking operations, this equipment must fail operational. Thus one strapdown will suffice whereas two platforms are required.

The spares requirement is based on the fact that the element with the highest wearout rate is the gyro. In the case of the strapdown, individual gyros can be replace on board. A gyro failure on a platform, however, would require replacement of the entire platform. The platform could then be shipped to the ground for replacement of the failed gyro.

Assuming that all operating gyros fail on the average at their expected time to wear out, a relative wearout ratio can be computed based on the number of gyros normally operating in each mechanization. Four gyros normally would be running in the strapdown. Considering the two platforms, six gyros would be running. This leads to an expected platform to strapdown wearout ratio of 3:2. Allowing an expected time to wearout of a gyro to be 3-1/3 years for a 90 percent confidence level, the strapdown requires 8 spare gyros and the platform 12 spare gyros. In addition, the platform mechanization requires a spare platform on the ground, which can be sent up in the same shuttle that brings back the onboard failed platform.

Equipment cost can now be evaluated. Because development and qualification costs are not considered, nonrecurring costs are represented by the cost of onboard equipment (some spares also are carried onboard and are therefore considered part of the nonrecurring cost by this method, but this should present no difficulty because equal weighting will be applied to nonrecurring and recurring costs). Allowance is made for transportation costs of $1000 per pound to orbit (note that any gyro failure requires an entire platform to be shipped to orbit). The nonrecurring costs (onboard equipment costs) are computed as follows (including onboard spares).

Strapdown

Assembly	Cost	Weight (lb)
1 unit	$360,000	58
1 power supply and temp. control	5,000	2
2 gyros	40,000	3
2 gyro electronics	20,000	3
	425,000	66
Cost to orbit	66,000	
	$491,000	

Platform

Assembly	Cost	Weight (lb)
2 units	$240,000	80
2 power supplies and temp. control	10,000	4
	$250,000	84
Cost to orbit	84,000	
	$334,000	

The recurring costs (on the ground spares) are:

Strapdown

Assembly	Cost	Weight (lb)
6 gyros	$120,000	9
6 gyro electronics	60,000	9
6 accelerometers	24,000	4
6 accelerometer electronics	60,000	9
	$264,000	31
Cost to orbit	31,000	
	$295,000	

Platform

Assembly	Cost	Weight (lb)
1 unit	$120,000	*
12 gyros	120,000	
12 gyro electronics	60,000	
	$300,000	
Cost to orbit	160,000	
	$460,000	

*4 platforms = 160 pounds

Sustaining costs are more difficult to evaluate. Because platforms are used commercially in aircraft, facilities and personnel for their maintenance and checkout will be available throughout the duration of the Space Station Program; this may not be available for strapdowns. On the other hand, whether this facility will suffice for space-rated hardware and whether strapdown will achieve the same production quantities and status are questions, which cannot be evaluated with any certainty at this time. It is concluded that the platform may involve somewhat lower sustaining cost than the strapdown.

Based on this discussion, the 15 points for equipment cost are allocated as follows:

Equipment Cost	Maximum	Platform	Strapdown
Nonrecurring	4	4	3
Recurring	4	3	4
Sustaining	7	7	6
Subtotal	15	14	13

Ease of Maintenance

Both scheduled and unscheduled maintenance frequency and size of in-flight replaceable unit (IFRU) are considered in this category. The 20 points for ease of maintenance are distributed to unscheduled maintenance, scheduled maintenance, and IFRU size as 10 points, 5 points, and 5 points, respectively. The heaviest weighting is given to unscheduled maintenance because of both the inconvenience and the cost associated with untimely maintenance requirements (for example, to repair a failure which occurs shortly after departure of a shuttle vehicle presents a significantly different problem than a failure occurring just before its arrival or one whose maintenance can be delayed until the arrival of the next vehicle).

The strapdown can be reconfigured by the computer in case of a failure (or several failures for a six-gyro strapdown) and maintenance can be delayed for a long time. Redundant platforms can tolerate a single failure, but if maintenance is delayed for a long period the reliability may become questionable. Scheduled maintenance (if any) is a function of procedure and expected time to wearout. The actual maintenance of the strapdown may be slightly more difficult than for the platform, because only one component is being replaced rather than the entire unit and access may be restricted; however, this may be entirely offset by the masses involved.

Unscheduled maintenance requires the crew to replace a platform each time a gyro fails, whereas crew action is only required for every third gyro failure on the strapdown. This would indicate (on the average) three times as much crew effort for each platform gyro failure compared to the strapdown.

Scheduled maintenance is based on the platform-to-strapdown wearout ratio of 3:2 described previously.

In-flight replaceable unit size is based on the unit volume. For a strapdown, this is the volume of one gyro plus its associated electronics, which is estimated to be 52 cubic inches. The volume of a platform is estimated to be approximately 1152 cubic inches. Thus the IFRU size of the platform is more than 20 times greater than that of the strapdown. Based on the above, the 20 points for ease of maintenance are allocated as follows:

	Maximum	Platform	Strapdown
Unscheduled	10	3	10
Scheduled	5	3	5
IFRU size	5	0	5
Subtotal	20	6	20

Development Risk

Both high- and low-performance risk are considered; the 10 points for this category are divided equally between the two. The platform has the lowest risk for both categories. The strapdown is considered to have negligible low-performance risk (present demonstrated capability more than exceeds

present defined mission requirements). To attain much higher performance, however, would require a large reduction in some of the gyro error parameters which would entail considerable development risk. The 10 points for development risk are therefore allocated as follows:

Development Risk	Maximum	Platform	Strapdown
High performance	5	5	0
Low performance	5	5	5
Subtotal	10	10	5

External Interfaces

Complexity and amount of external equipment are considered in this category. Because the cost of a gimbal replacement computer for the strapdown is included in the penalties associated with all the other categories, no difference is seen between it and a platform. The 5 points for external interfaces are therefore allocated as follows:

Maximum	Platform	Strapdown
5	5	5

Size

This category considers the in-orbit volume for two platforms, or one six-gyro strapdown and spares. The volume for two platforms is 3456 cubic inches. The basic strapdown volume is 1066 cubic inches. To this 321 cubic inches is added for the spares (assumes a packing factor of 3), giving a total platform volume of 1387 cubic inches. The 5 points for size are allocated as follows:

Maximum	Platform	Strapdown
5	2	5

Weight

This category considers the onboard weight of the systems mentioned.

Strapdown Weight (lb)

1 unit	58
Spare power supply	2
2 spare gyros	3
2 gyro electronics	3
	66
Packing for spares and tools	12
	78

Platform Weight (lb)

2 units	80
2 power supplies and temperature control	4
	84

The 10 points are allocated as follows:

Maximum	Platform	Strapdown
10	10	10

Power

This category considers the operating power for two platforms or one strapdown with four gyros operating. (The accelerometers are shut down.)

Strapdown

Temperature control for 12 instruments	18 watts
Suspension, signal generation, and demodulator excitation for 4 gyros	8 watts
Pulse torquing loop for 4 gyros	60 watts
Gimbal replacement computer	50 watts
	136 watts

Platform

2 units 185 watts	370 watts
-6 accelerometers	-12 watts
-6 accelerometer electronics	-90 watts
	268 watts

The 5 points are allocated as follows:

Maximum	Platform	Strapdown
5	3	5

Reliability

Comparison of the reliability of these devices requires either historical data from similar programs or a theoretical model. The second approach was chosen because of lack of adequate data - the approach consists of comparing theoretical models of the two mechanizations constructed using similar components. Considering each gyro has a probability of successful operation R, the probability of success of two 3-gyro platforms operating redundantly can be computed as $R^P = R^3 (2-R^3)$. Similarly, for a six-gyro strapdown in which any three failures can be tolerated (with operational redundancy of six gyros), the probability of success may be computed as $R^S = R^3 (20-45R + 36R^2 - 10R^3)$. Comparison is further confounded by the fact the R is a function of operating time. Thus, a new gyro operating for a short time may have a probability of success very close to unity (say 0.99), whereas a gyro operating to wearout (say MTBF minus 0.44 sigma for a normal wearout distribution) may have a probability of success of only about 2/3. In addition, the two redundant platforms may be in a condition (after maintenance) where the gyros on one platform are new and the gyros on the other are close to wearout. A comparison of the strapdown and the platform with new gyros for short operating time would indicate little difference in the two. This would be unrealistic since the six-gyro strapdown is more reliable than even three redundant platforms. On the other hand, to compare them with gyros near wearout (or for long operating times) would be to ignore the heavy emphasis already placed on maintenance. Therefore, in the absence of mission timeline information and numerical reliability criteria, the approach adopted was to average the short-time and long-time success probabilities by dividing the 10 points into 5 points for short-term success and 5 points for long-term success.

Thus, the short-term success probability ($R \approx 1$) of the two mechanizations is essentially the same and both are awarded 5 points. The long-term success probability ($R \approx 2/3$) is computed as follows:

$$R^S_{2/3} = \left(\frac{2}{3}\right)^3 \left(\frac{82}{27}\right), \quad R^P_{2/3} = \left(\frac{2}{3}\right)^3 \left(\frac{46}{27}\right)$$

Thus, the redundant platform success probability is in the ratio of 46:82 with the strapdown.

The 10 points for reliability are apportioned as follows:

	Maximum	Platform	Strapdown
Short term	5	5	5
Long term	5	3	5
Total	10	8	10

Performance Accuracy

This category considers accuracy of alignment and drift. The alignment accuracy of both mechanizations is essentially the same. The platform and strapdown are both subject to gyro drift, and, in addition, the strapdown drift is greater in a local vertical mode because of gyro scale factor error. The 5 points for performance are allocated as follows:

Maximum	Platform	Strapdown
5	5	3

Versatility

This category considers the ability of the mechanizations to perform adequately for missions other than the nominal design mission. The platform is considered the most versatile. The strapdown would not have as high a performance capability for thrusting modes such as boost monitor. The 5 points for versatility are allocated as follows:

Maximum	Platform	Strapdown
5	5	4

Crew Workload

There is substantially no difference in the operational crew work load between the mechanizations. Both are therefore awarded the maximum of 10 points for this category.

Summary

The comparison matrix is shown in Table 3.4-9. The major difference is seen to be the ease of maintenance factor which favors the strapdown. The development risk factor is the major difference in favor of the platform concept.

Table 3.4-9 Inertial Reference Comparison Matrix

Factor	Maximum	Platform	Strapdown
Equipment Cost	15	14	13
Ease of Maintenance	20	6	20
Development Risk	10	10	5
External Interfaces	5	5	5
Size	5	2	5
Weight	10	10	10
Power	5	2	5
Reliability	10	8	10
Performance Accuracy	5	5	3
Versatility	5	5	4
Crew Workload	10	10	10
Total	100	77	90

3.5 REACTION CONTROL SUBSYSTEM (RCS)

The OLS RCS must provide thrust for four functions in zero gravity. These are control moment gyro (CMG) backup and desaturation (or stabilization and attitude control if CMG units are not used), maneuvering, orbit maintenance, and docking torque control. If artificial gravity is used at any point in the OLS mission, the RCS would be designed to provide spinup, despin, and momentum vector control.

A trade tree has been prepared (Figure 3.5-1) to illustrate the more probable RCS design options to be considered for the OLS. Past studies by G&C on similar vehicles such as the Earth Orbiting Space Station (EOSS) have indicated that the medium thrust RCS engines can perform all of the required functions (for the purposes of this discussion, medium thrust refers to a thrust range of 1 to 100 pounds). Low-thrust engines (those having a thrust of less than 1 pound) can perform orbit maintenance and CMG desaturation efficiently. Therefore, the trade tree shows that there is a choice between a pure medium thrust subsystem and a combined medium and low-thrust subsystem.

In the medium thrust portion of the study, several options are available. The subsystem may be designed around earth storable liquid propellants such as the bipropellant combination of nitrogen tetroxide and monomethyl hydrazine or the monopropellant hydrazine concepts. The concept also may be based on gaseous propellant engines burning oxygen with hydrogen, methane, or ammonia. The characteristics of these concepts are discussed subsequently.

The low-thrust concepts considered feasible for the OLS use non-combustion, heated gas engines. In particular resistojets, arcjets, and isojets have been examined. The propellants usable in these designs may be obtained from RCS storage tanks or they may be wholly or partially waste gases available from ECLSS.

3.5.1 MEDIUM THRUST

It is necessary to consider several factors in selecting an RCS concept. Concepts that seem to have favorable performance factors or storage characteristics may be quite undesirable from other standpoints. Factors that must be considered in relation to the OLS mission life include available power, crew safety, maintenance criteria, and plume criteria including contamination and impingement. Other factors of importance are propellant resupply intervals, interface with other systems and subsystems, and OLS quiescent mode requirements. Tables 3.5-1, 3.5-2, 3.5-3, and 3.5-4 summarize some of the major factors to be considered in relation to long life and crew safety. Notice that the factors of explosiveness, corrosiveness, and toxicity presume propellant leakage into the spacecraft living areas. The reason for this is that while the propellants must be stored outside of the pressurized OLS components, it will probably be necessary for the crew to perform RCS

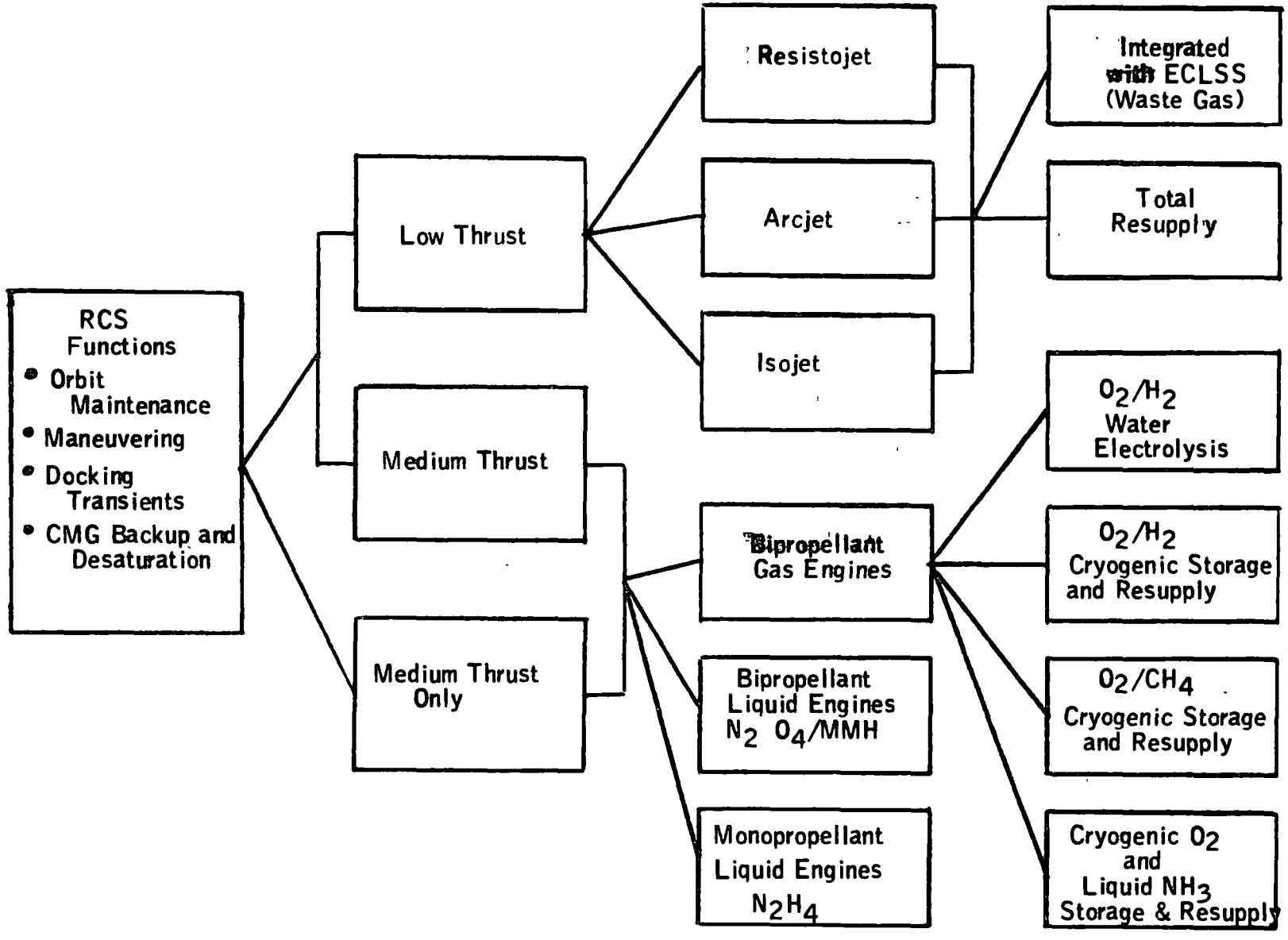


Figure 3.5-1. RCS Trade Tree

Table 3.5-1. Explosion Hazards

Propellant	Explosion Hazards
Nitrogen tetroxide, N_2O_4	• No significant problems
Hydrazine, N_2H_4	• Has sufficient vapor pressure to support an explosive atmosphere from a spill at temperature of 104 F or more • Explosive atmosphere contains 4.7 to 100% N_2H_4 by volume
Monomethyl hydrazine, $CH_3N_2H_3$ (MMH)	• Forms an explosive atmosphere • Easily catalyzed by rust to ignition point
Oxygen, O_2	• None in a clean area • Reacts vigorously with flammable and many normally nonflammable materials • Liquid forms shock-sensitive, detonable mixtures with many flammable materials (oil, grease, asphalt, etc.)
Hydrogen, H_2	• Extremely flammable in air in concentrations between 4.0 and 74.2% by volume
Ammonia, NH_3	• May be ignited in air at concentrations of 16.2 to 27% by volume • Forms explosive compounds on contact with various chemicals including Hg, Cl, I, Br, Ag_2O, $HClO_2$

maintenance during the mission. The decision factors that indicate the probable need for RCS maintenance are even more significant for the OLS than they were for the EOSS. It is impractical to replace the OLS in the event of a failure. Further, it is impractical to build 10 years of reliability into a subsystem when the alternative is maintenance. Any RCS selection must consider this maintenance factor.

Table 3.5-2. Corrosion

Propellant	Corrosive Properties
Nitrogen tetroxide, N_2O_4	• Corrosive to many common materials • Combines with H_2O to form HNO_3, which increases corrosivity to include several common structural and conductive metals
Hydrazine, N_2H_4	• Corrosive to many common metals such as Pb, Zn, Cu, Fe, and alloys containing 0.5% or more of Mo by weight • Corrosive to many organic materials such as natural rubber and cork
Monomethyl hydrazine, $CH_3N_2H_3$ (MMH)	• Corrosive properties similar to N_2H_4 • Somewhat more corrosive to organic materials than N_2H_4
Oxygen, O_2	• No serious problem
Hydrogen, H_2	• None
Ammonia, NH_3	• Corrosive to Cu, Zn, and alloys of these such as brass, bronze, monel (in the presence of moisture) • Corrosive to alkali and alkalene earth metals, Mg, Be, etc.

Table 3.5-3. Exhaust Plume Contamination

	Propellants		
	$N_2O_4/CH_3N_2H_3$ (MMH)	O_2/H_2	N_2H_4
Primary exhaust products (steady-state)	H_2O, CO, N_2, CO_2	H_2O, H_2	N_2H_4, NH_3, N_2, H_2
Deleterious exhaust products	N_2O_4, HNO_3, $CH_3N_2H_3$ $CH_3N_2H_4NO_3$, tramp metals*	None	NH_3, NH_4NO_3* N_2H_4, NH_4Cl*
Deleterious decomposition characteristics effects	Reaction, deposition, abrasion (tramp metals)	Deposition	Reaction, deposition, abrasion (Shell 405 catalyst)
*From trace impurities			

Table 3.5-4. Propellant Toxicity

Propellant	Propellant Toxicity Characteristics
Nitrogen tetroxide, N_2O_4	• Maximum safe concentration in air 5 ppm • Mild distinct odor at 5 ppm • Forms HNO_3 in lungs causing fluid exudate; noticeable effects may be delayed 12 hours or more • Liquid spill on skin is similar to 65% HNO_3
Hydrazine, N_2H_4	• Maximum safe concentration in air 1 ppm • Distinct amine odor at 3 to 4 ppm • Corrosive to body tissue (alkali burn) • Mild overexposure; dizziness, nausea, hoarseness come gradually over several hours of exposure • Causes lower blood sugar and anemia
Monomethyl hydrazine, $CH_3N_2H_3$ (MMH)	• Maximum safe concentration in air 0.2 ppm • Distinct amine odor at 1 to 3 ppm • Biological effects similar to N_2H_4
Oxygen, O_2	• Maximum safe concentration in air 60% (volume) • Respiratory damage at pressures up to 2 atmospheres of pure O_2 (bronchitis, pneumonia, and lung collapse) • Central nervous system adversely affected at pressures above 2 atmospheres pure O_2
Hydrogen, H_2	• Not toxic
Ammonia, NH_3	• Maximum safe concentration in air 50 ppm • Least detectable concentration about 50 ppm • Corrosive to body tissue (alkali burns) • Irritation to eyes and respiratory track

Through the use of Tables 3.5-1 through 3.5-4, some of the RCS trade tree options of Figure 3.5-1 may be eliminated. The combination of nitrogen tetroxide and monomethylhydrazine may be eliminated for two reasons. First, it has a very dirty exhaust plume which is undesirable because of the

deleterious effects on thermal control coatings, solar cells, optics, and some experiments. The quantitative data available on exhaust plume contamination are sparse and somewhat conflicting. Table 3.5-3 does not present quantitative data for this reason. Nevertheless, from a qualitative viewpoint, the combination of nitrogen tetroxide and monomethylhydrazine is much dirtier than either monopropellant hydrazine or oxygen and hydrogen, and the data available do support this statement. Secondly, both chemicals are highly corrosive and toxic.

Another propellant combination also was eliminated because of its plume. Oxygen and methane (not listed in Table 3.5-3) can produce particulate carbon. This byproduct will most likely occur at the low mixture ratio required to maintain cool, long-lived combustion chambers, and during engine start and stop operations. Furthermore, this propellant combination showed no significant advantages over the other chemicals considered.

The water electrolysis concept was dropped because it is a heavy consumer of electrical power, a reason that does not appear on the foregoing tables. Figure 3.5-2 shows the expected average power consumption for this concept over 180 days as a function of total impulse. The figure shows that a total impulse of 1×10^6 pound-seconds over 180 days requires an average power of 2000 watts. Experience gained on the EOSS indicates that power levels on this order would put an undue additional burden on the electrical power subsystem. Electrolysis is an entirely undesirable option unless the RCS requirement (per 180 days) drops to the 100,000-pound-second range, which is considered unlikely.

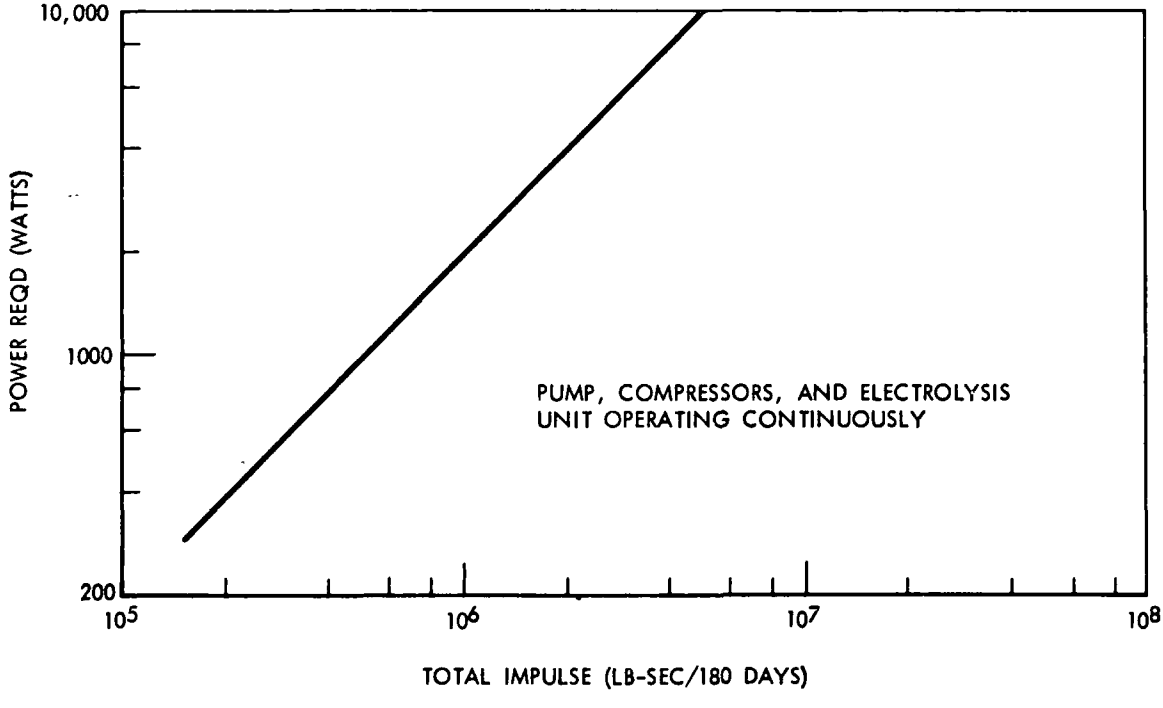


Figure 3.5-2. Average Electrical Requirements for Water Electrolysis System

In general, any chemical chosen for a propellant that exhibits a corrosion, explosion, or toxicity problem must be given special treatment to minimize the hazard. To minimize the danger of explosion, fire, and toxicity because of leakage, RCS components controlling these fluids that are accessible from inside the crew compartment must have inert or nitrogen gas padding inside of air-tight compartments. In addition, the use of vulnerable materials must be avoided in any place where contact with the propellant is likely leakage or maintenance activities.

The three remaining medium thrust choices are monopropellant hydrazine, cryogenically stored oxygen and hydrogen, and cryogenically stored oxygen with ammonia stored at ambient temperatures. These three concepts have been parametrically analyzed for comparison. The analysis covers a subsystem design total impulse capacity of 2×10^5 to 2×10^6 pound-seconds. For this comparative analysis, four engine clusters of four engines each were assumed. It was assumed that the propellant storage tanks and pressurant bottles (if any) were located outside of the pressurized compartment (but shielded), and that there would be two propellant storage areas. It was also assumed that inflight maintenance would be performed.

The monopropellent hydrazine concept is presented in Figures 3.5-3, 3.5-4, and 3.5-5. The calculations presented in these data assume nitrogen pressurant and bladder expulsion propellant tanks. The delivered specific impulse is assumed to be 230/lb-sec/lb_m. Figure 3.5-3 presents the total loaded subsystem mass and the total subsystem mass without any consumables. Figure 3.5-4 presents the consumable masses, and Figure 3.5-5 presents the propellant volume. For an estimate of the total propellant tankage volume, add five percent to the values in Figure 3.5-5.

Tables 3.5-1 through 3.5-4 indicate that hydrazine can readily form an explosive mixture in air. It also is corrossive and toxic but remains a viable RCS option primarily because it is simple and only one propellant is present to cause a hazard. The components of a hydrazine subsystem that are serviceable from within the living quarters of the OLS must be housed in nitrogen chambers (as previously mentioned) to reduce the risks involved. Of all the candidate concepts considered, the hydrazine plume is clean by comparison. Only the oxygen and hydrogen plume is cleaner. Again, this is a qualitative comment because the available quantitative data are conflicting.

If hydrazine were selected for the RCS, it is possible that hydrazine also would be selected as a source of nitrogen for the ECLSS atmospheric makeup requirement. In addition, the hydrogen from the hydrazine could be used to supply the ECLSS Sabatier reactor (if one is used). In this case, it would be practical to use a common hydrazine storage provision for both the RCS and the ECLSS requirements, achieving overall OLS simplification through subsystems integration.

Figures 3.5-6, 3.5-7, 3.5-8, and 3.5-9 cover the parametric design of the oxygen and hydrogen concept. The calculations presented in these data assume vapor pressure pressurization and waste heat propellant conditioning. If the propellants are stored at 100 psia, there is sufficient vapor pressure to operate the entire RCS. Nominally, the vapor pressure would be maintained by heat leak through the propellant tank insulation. The propellants leave

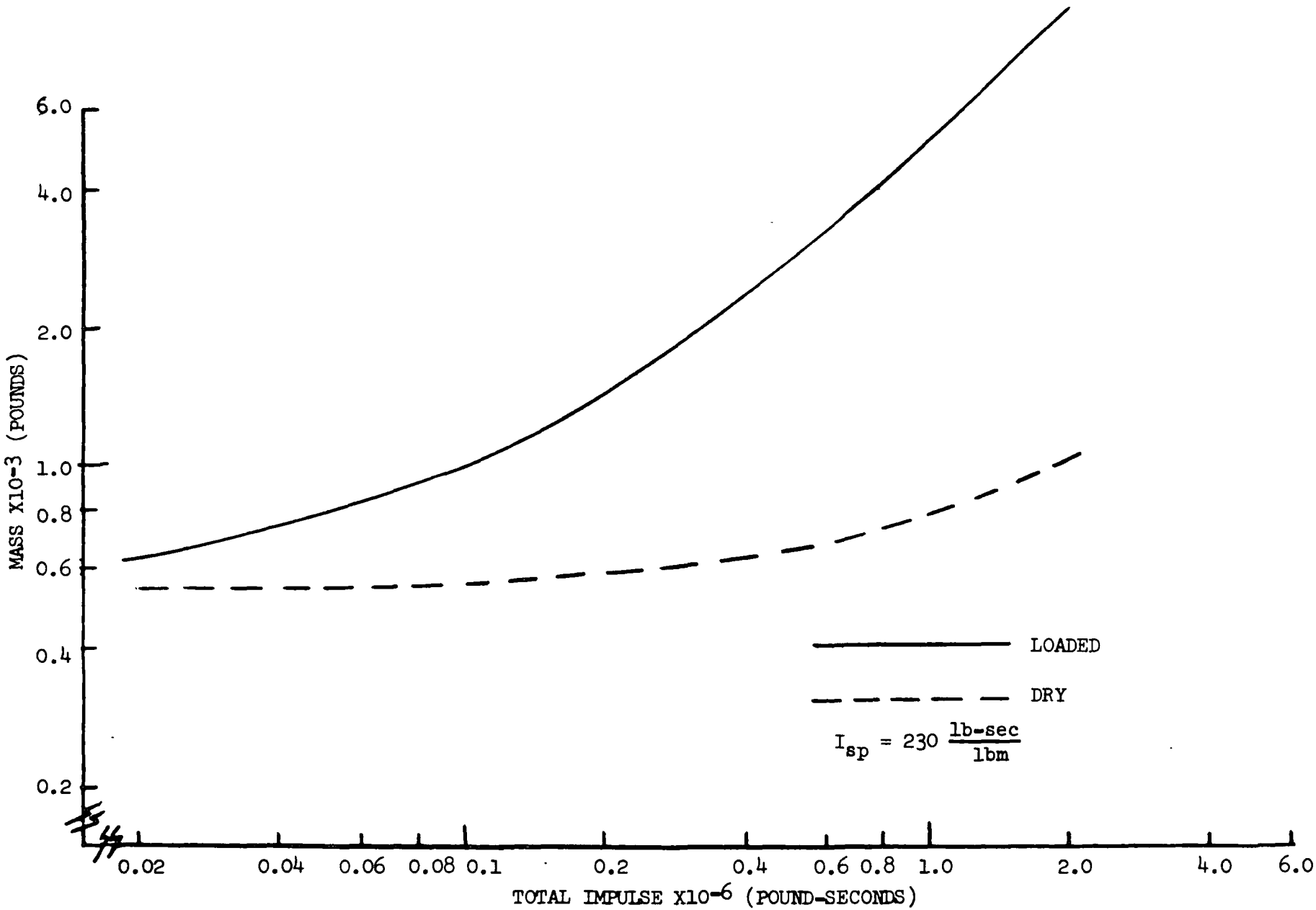


Figure 3.5-3. Reaction Control Subsystem Dry and Loaded Mass Monopropellant Hydrazine

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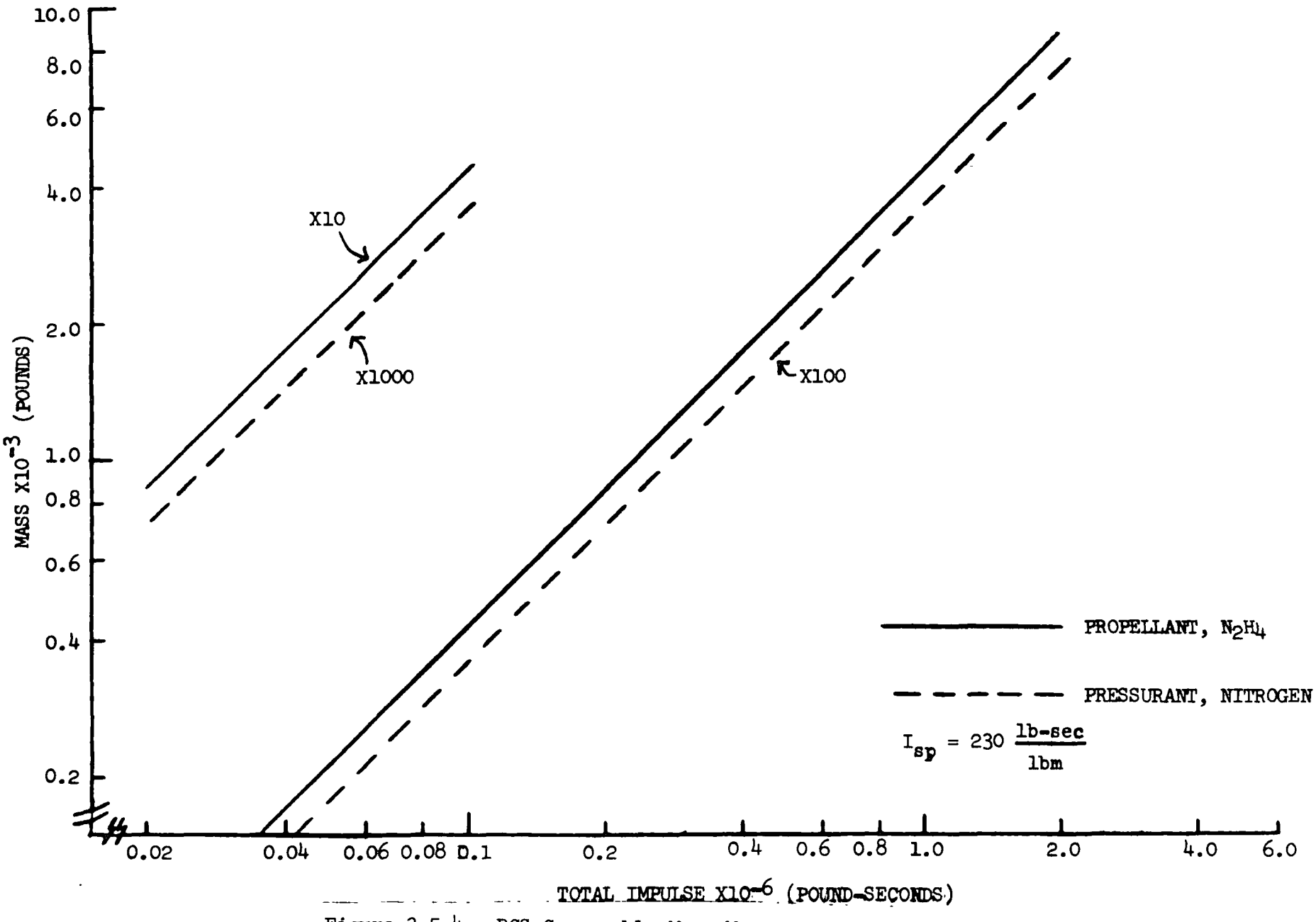


Figure 3.5-4. RCS Consumable Mass Monopropellant Hydrazine

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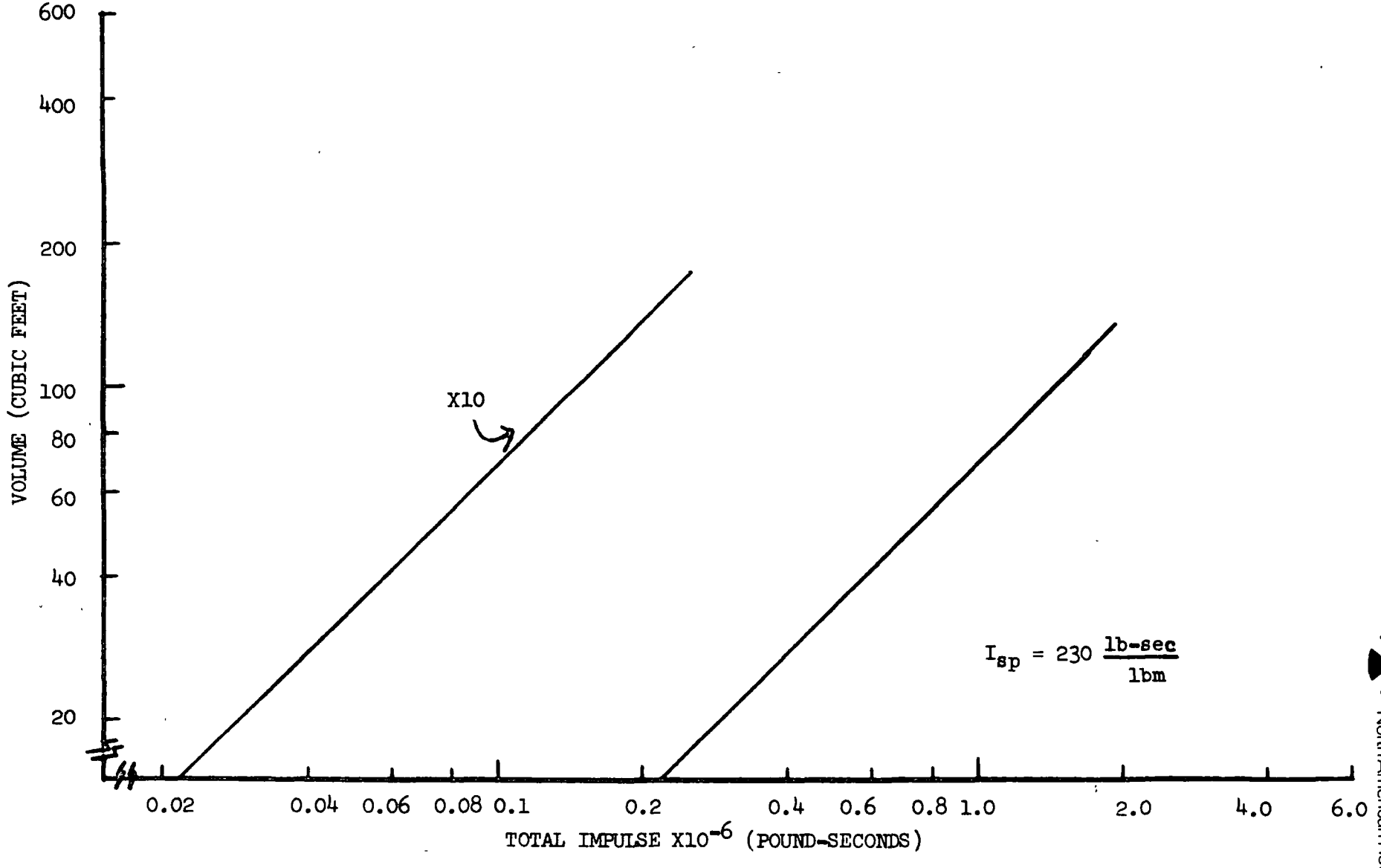


Figure 3.5-5. RCS Propellant Volume Monopropellant Hydrazine

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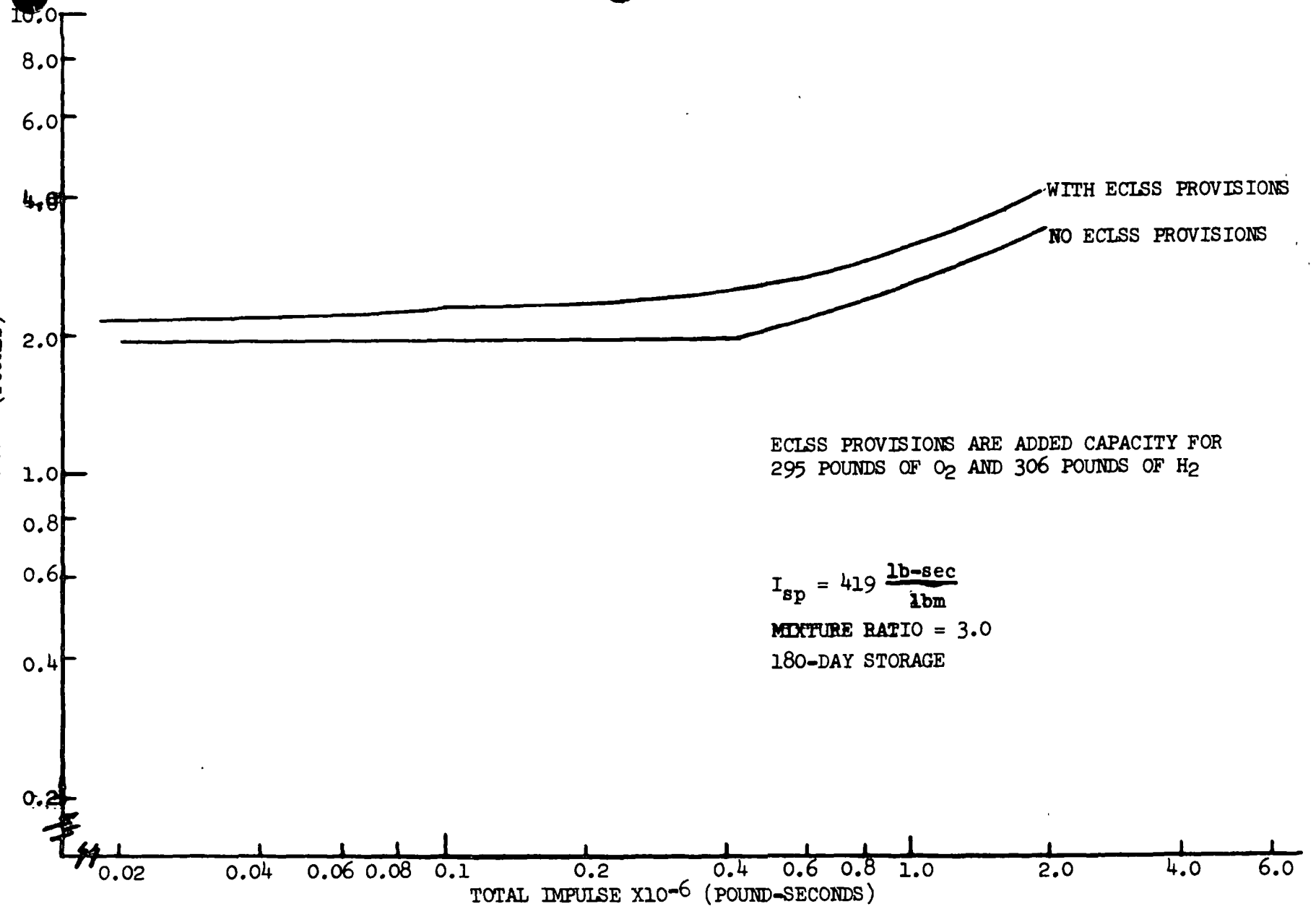


Figure 3.5-6. RCS Dry Mass Cryogenic Oxygen and Hydrogen

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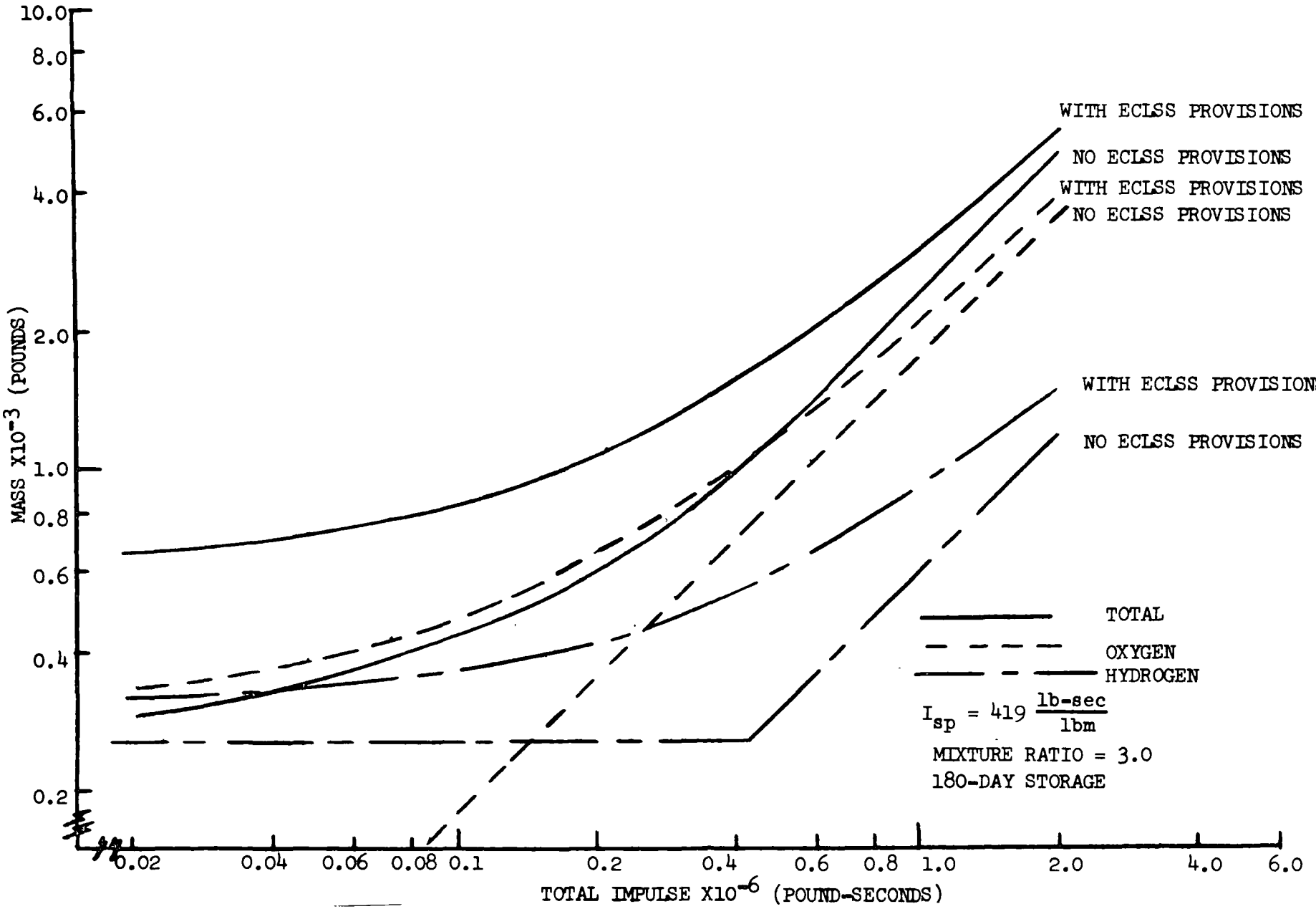


Figure 3.5-7. RCS Propellant Mass Cryogenic Oxygen and Hydrogen

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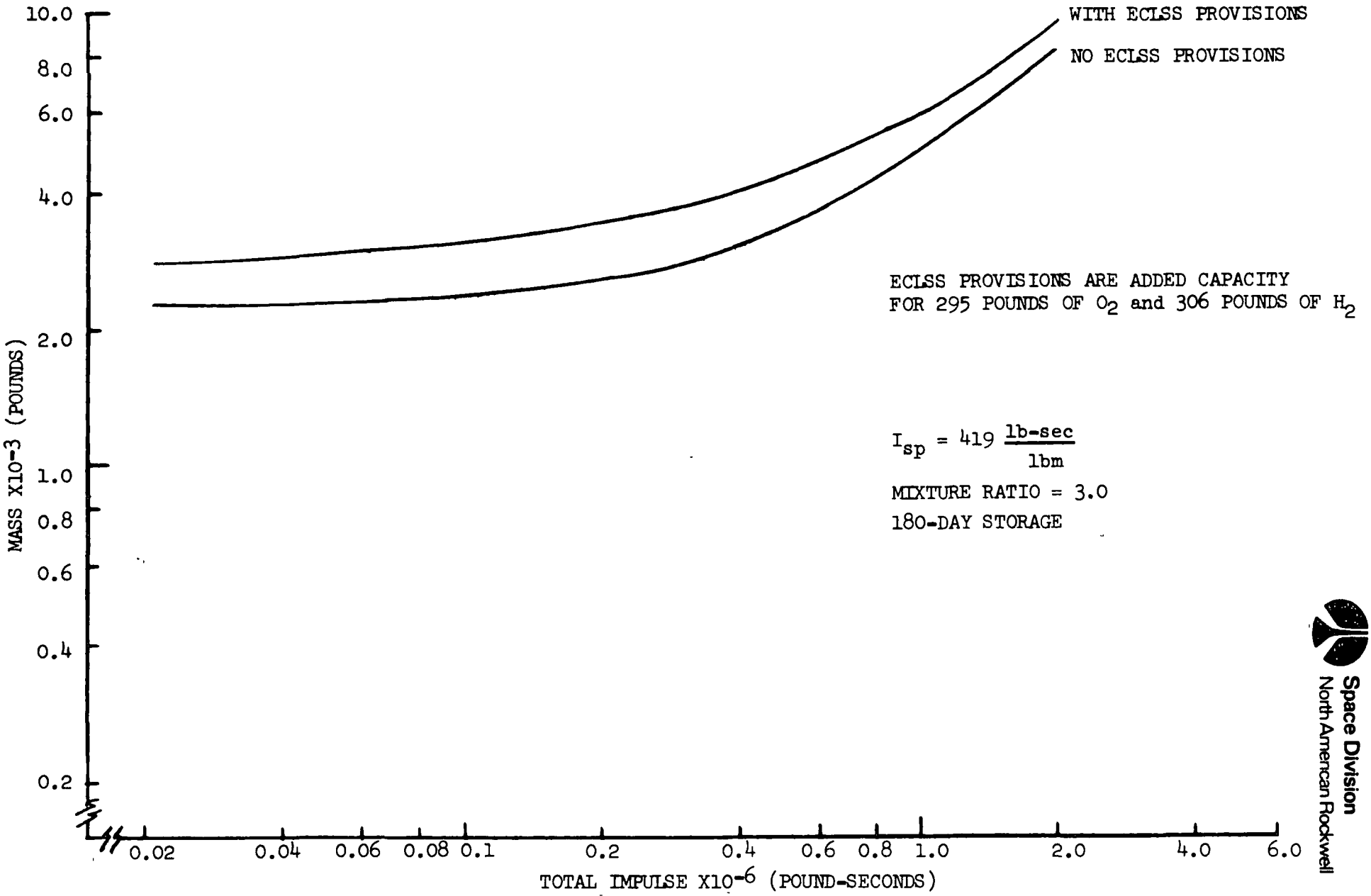


Figure 3.5-8. RCS Loaded Mass Cryogenic Oxygen and Hydrogen

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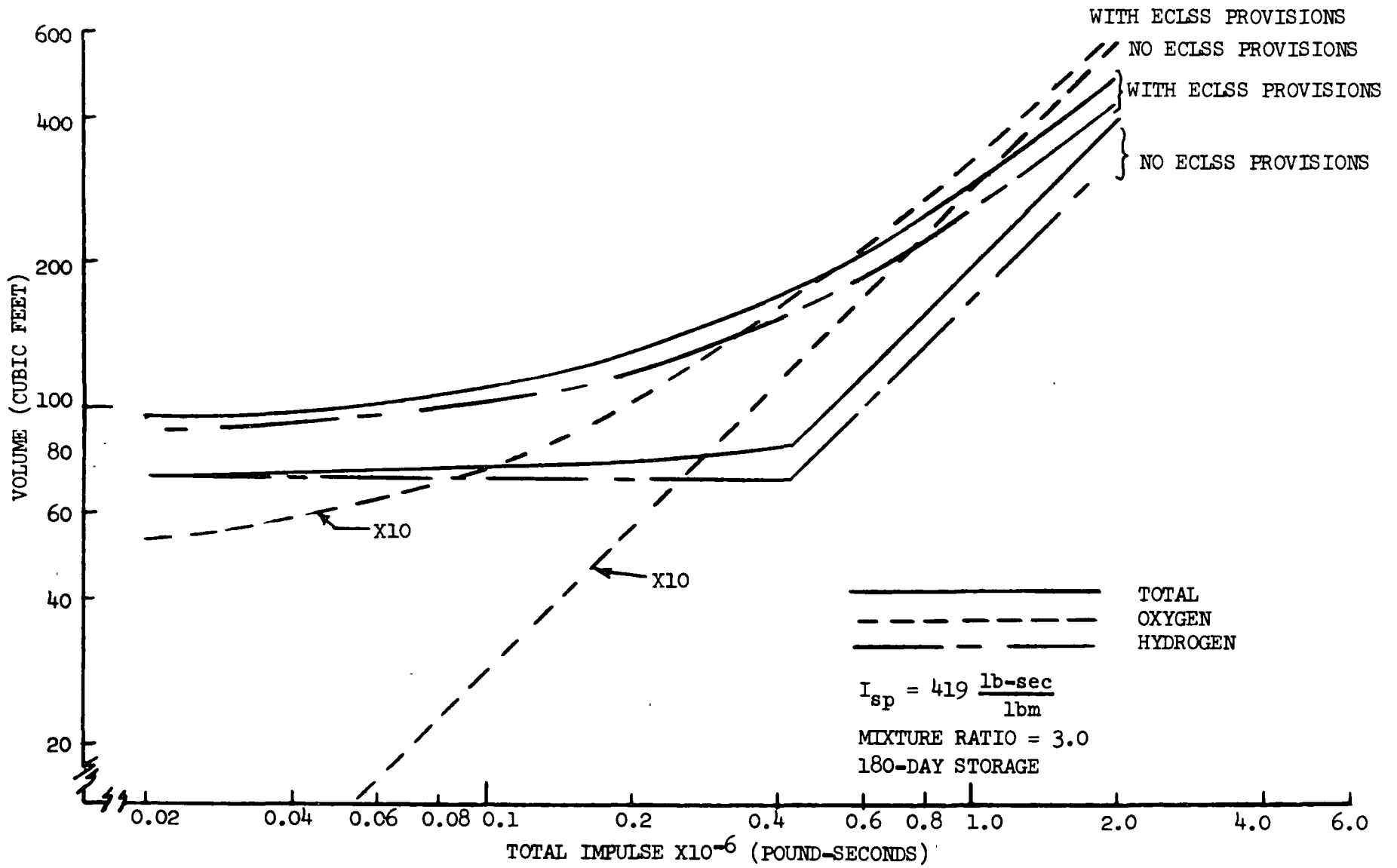


Figure 3.5-9. RCS Propellant Volume Cryogenic Oxygen and Hydrogen

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the storage tanks on demand in either the liquid or gaseous phase. The propellants can then be used in a propellant tank vapor shield to improve the insulation effectiveness. Next, the propellants pass through a heat exchanger and are warmed using ECLSS waste heat to become ambient temperature gases. If necessary, some of these warm propellants may be recirculated to the propellant tanks to supplement the heat leakage into the tanks during periods of high propellant usage and to break up stratification. From the heat exchangers, the propellants pass to accumulators to be used on demand by the engines. The delivered specific impulse assumed for this concept was 419 lb-sec/lb_m at a mixture ratio of 3.0 to 1. A significantly lower specific impulse of about 356 lb-sec/lb_m would have to be assumed if waste heat were not available for conditioning. In this case, a portion of the propellants would be burned in gas generators to produce the required conditioning heat.

The mixture ratio of 3.0 to 1 was selected as being a good tradeoff between long engine life (using hydrogen film cooling) and tankage sizing. A higher mixture ratio reduces the total tankage volume but significantly reduces either engine life, specific impulse (because of increased film cooling to maintain engine life), or both. A lower mixture ratio lowers the combustion temperature and the film cooling requirements, but it increases the propellant tankage volume requirements.

Figure 3.5-6 presents the dry mass of the oxygen and hydrogen RCS concept. Figure 3.5-7 presents the propellant masses, Figure 3.5-8 presents the total loaded mass, and Figure 3.5-9 presents the required propellant volumes. For a first estimate of tankage volume, add 10 percent to the values in Figure 3.5-9. Figures 3.5-7 and 3.5-9 show that the hydrogen requirement has a clearly defined minimum of 256 pounds or 72 cubic feet. These values are based on a 180-day design storage period and a minimum practical heat transfer coefficient for cryogenic hydrogen tanks of 0.07 Btu/hr ft^2. This means that at the end of 180 days, 256 pounds of hydrogen will be gone, even if it is not used. It will boil off and be released through the safety valves. There is a minimum for oxygen also, but it is about 6 pounds and does not affect these data.

As with hydrazine, the cryogenic storage oxygen and hydrogen concept is compatible with other subsystems. The EPS may incorporate fuel cells as backup power units. The reactants for these cells are oxygen and hydrogen. It would be a simple matter to tap into the RCS propellant supply to run these reactors. Likewise, previous studies on the EOSS have indicated that repressurization atmospheric oxygen may be stored with the RCS oxygen. If a Sabatier reactor is used, the RCS hydrogen may be used for supplementary hydrogen. In the EOSS study, 295 pounds of oxygen were required for repressurization, and 306 pounds of hydrogen were required for the Sabatier units (180 days). These values were selected as examples to show the effect of combining the ECLSS storage with the RCS storage, and these effects are shown in Figures 3.5-6 through 3.5-9. Not shown in the figures is the effect of also storing the ECLSS nitrogen as a cryogenic as proposed for the EOSS.

Hydrogen easily forms an explosive mixture in air. For this reason, all of the hydrogen components in the pressurized section of the OLS will probably be isolated in nitrogen-filled cabinets as previously discussed.

Oxygen, on the other hand, supports combustion and is present in air. It will not require a nitrogen pad. Neither propellant is particularly corrosive or toxic. Also, oxygen and hydrogen engines produce the cleanest exhaust plume of all of the propellants considered. These factors are illustrated in Tables 3.5-1 through 3.5-4.

The oxygen and ammonia concept is presented in Figures 3.5-10, 3.5-11, and 3.5-12. It was assumed that the oxygen would be stored at 100 psia (saturation temperature of 204 R), as was done in the cryogenic storage oxygen and hydrogen concept. However, ammonia would be stored at ambient temperature (530 R). At this temperature, the vapor pressure of liquid ammonia is 129.5 psia. Vapor pressure would be used to expel the propellants from the tanks in a two-phase flow. These propellants would pass through a conditioning system similar to that discussed for oxygen and hydrogen and would also be stored in accumulators for use by the engines on demand. Again, waste heat was assumed for conditioning. Under these circumstances, a delivered specific impulse of 292 lb-sec/lb_m was assumed.

Figure 3.5-10 presents the total loaded subsystem mass and the total subsystem mass without any consumables. Figure 3.5-11 presents the consumable masses, and Figure 3.5-12 presents the propellant volume. For a first estimate of the total propellant tankage volume, 10 percent is added to the values in Figure 3.5-12.

As previously mentioned, oxygen shows no undue problems in explosion, corrosion, or toxicity. Tables 3.5-1, 3.5-2, and 3.5-4 show a few problems with ammonia. Ammonia will form an explosive atmosphere in air over a restricted range. It also forms explosive compounds on contact with various materials. It is corrosive to a few common materials including skin. Ammonia is toxic; but it is only a tenth as toxic (on a parts per million basis) as hydrazine, and less is required for any given level of performance. Exhaust plume data are not listed in Table 3.5-3, but a few undesirable exhaust materials are likely to occur under certain operating conditions. Two of these materials are unreacted ammonia and nitrous oxide. In general, however, the exhaust plume should be expected to be reasonably clean and comparable to the cleanliness of monopropellant hydrazine.

As in the hydrazine and the oxygen and hydrogen concepts, the oxygen and ammonia concept may be integrated with the ECLSS. The RCS oxygen tanks can be used to provide repressurization gas. Ammonia may be used as a source of ECLSS nitrogen and Sabatier unit hydrogen (if a Sabatier unit is used). The ECLSS ammonia would be stored in the same tanks as the RCS fuel.

The data plotted on Figures 3.5-3 through 3.5-11 show some interesting comparisons between the three candidate concepts. The hydrazine concept has the lowest dry mass because of its simple design and relatively dense propellant. The greater the density of the propellants, the smaller the storage tanks may be. Also, because hydrazine is stored at ambient temperatures, no insulation is required. The dry mass of the oxygen and ammonia concept is higher because of the increased subsystem complexity, lower propellant density, and the need for insulation. The bulk propellant density for the oxygen and

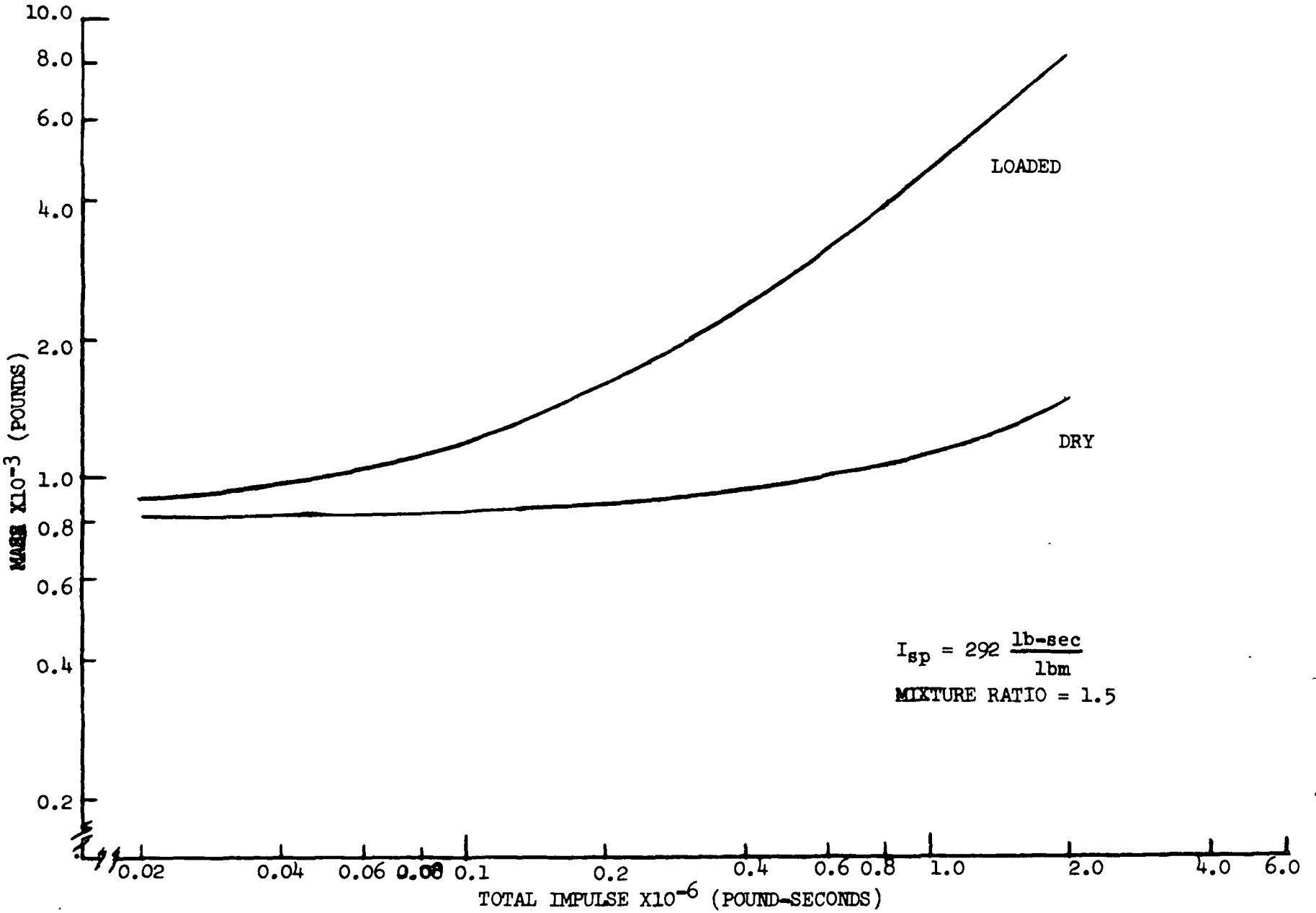


Figure 3.5-10. RCS Loaded and Dry Mass Liquid Oxygen and Ammonia

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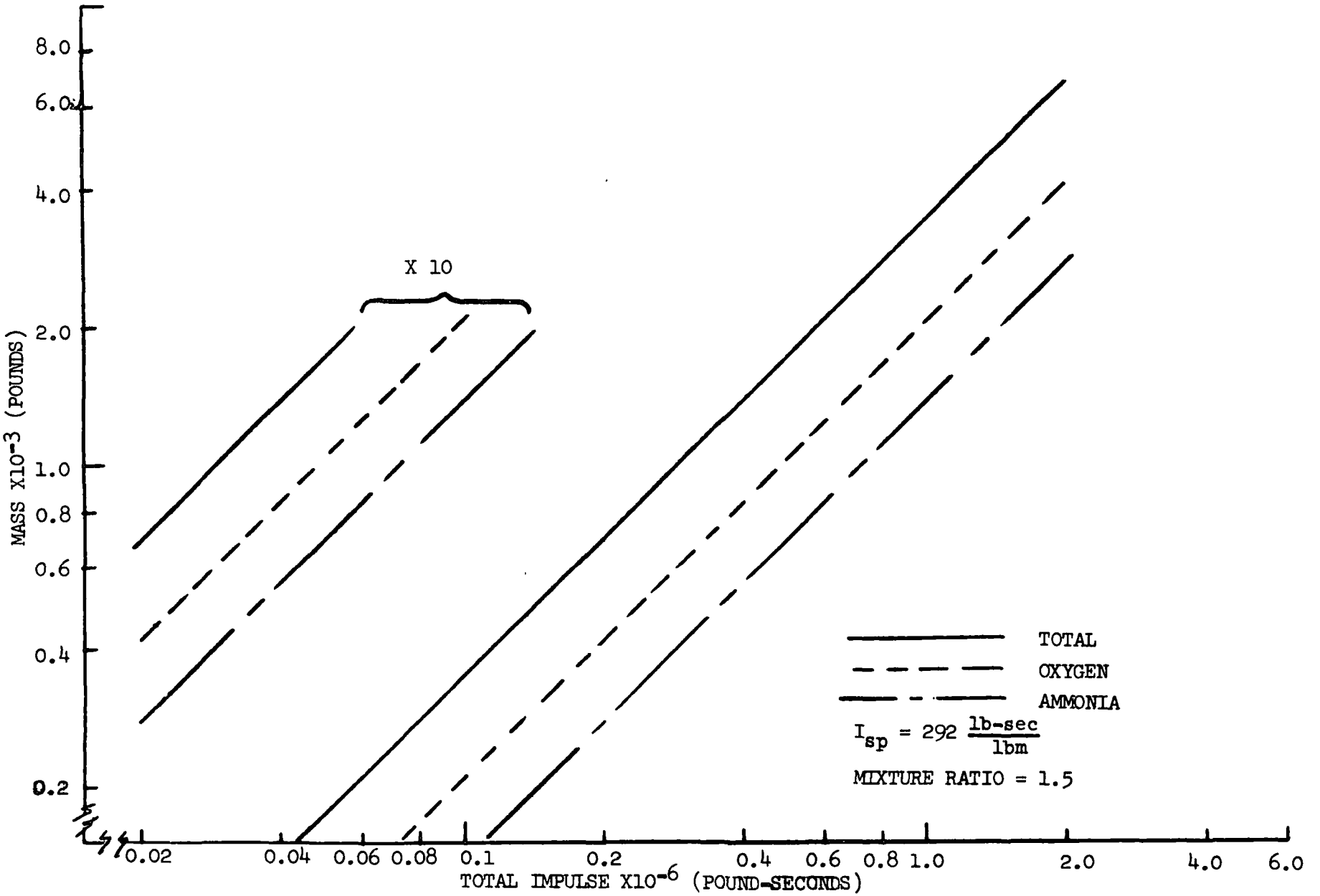


Figure 3.5-11. RCS Propellant Mass Liquid Oxygen and Ammonia

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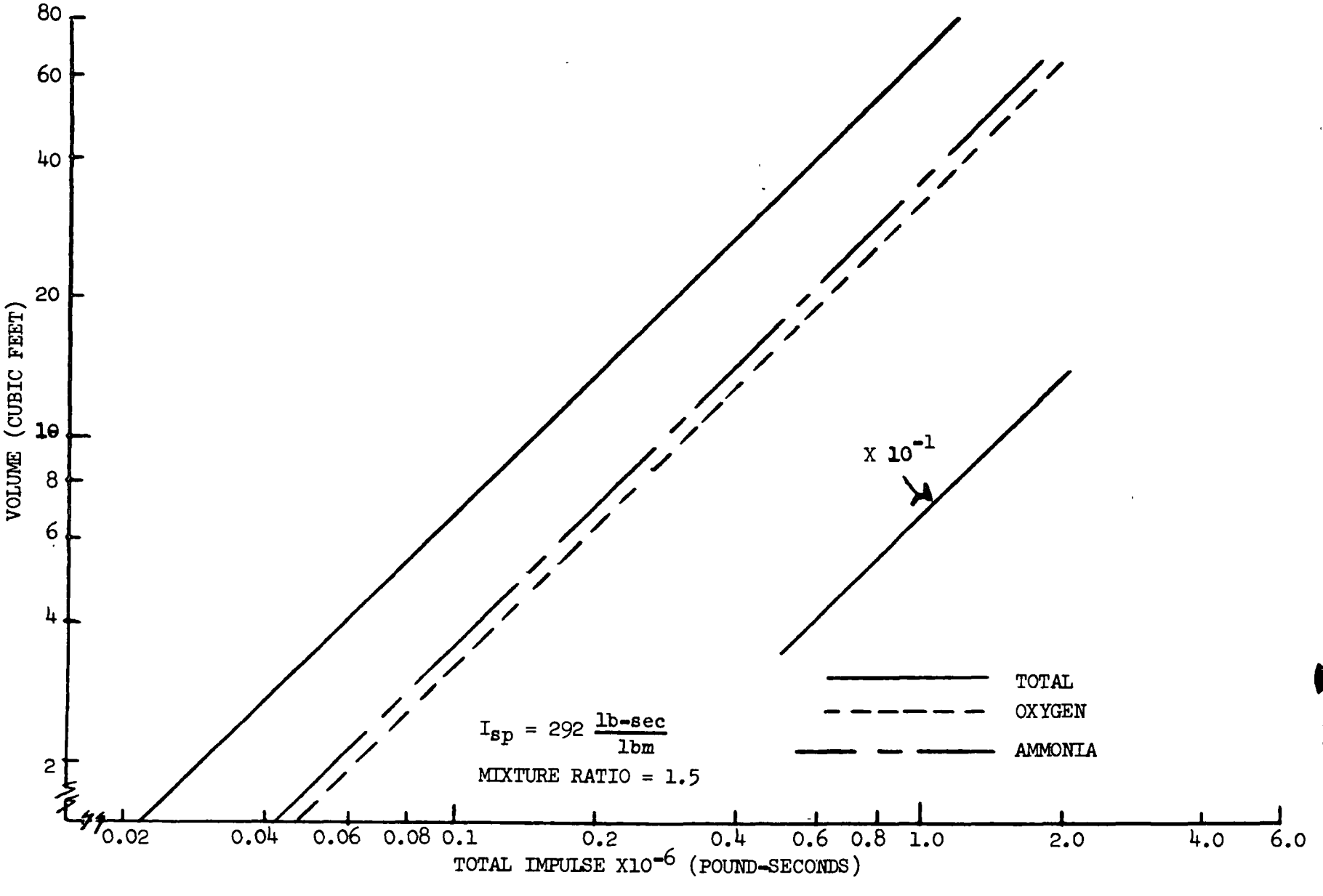


Figure 3.5-12. RCS Propellant Volume Liquid Oxygen and Ammonia

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hydrogen concept is much lower than that of the others, and insulation is required on all the storage tanks. For these reasons, the dry mass of the oxygen and hydrogen concept is highest.

Because of its high specific impulse, the oxygen and hydrogen concept has the lowest resupply mass for (180-day) impulse requirements in excess of 150,000 pound-seconds. For less than 150,000 pound-seconds, the minimum permissible hydrogen tank loading requirement allows the oxygen and ammonia concept to have the lowest resupply mass. Because of its low specific impulse, the hydrazine concept has the highest resupply requirement of the three concepts.

Combining the dry and consumable masses, the three options become quite close in total mass. Below a total impulse requirement of about 315,000 pound-seconds, the hydrazine concept is lightest. For impulse requirements between 315,000 and 1,800,000 pound-seconds, the oxygen and ammonia concept has the least total mass. The high specific impulse of the oxygen and hydrogen concept finally becomes effective at impulse requirements greater than 1,800,000 pound-seconds, and this concept is lightest for all total impulses over this level.

One final comparison of the consumables volume shows that the oxygen and ammonia and the hydrazine concepts require nearly equal total propellant volumes. However, because of the low density of hydrogen, the oxygen and hydrogen concept requires much more space.

3.5.2 LOW THRUST

Thrusters

Three low-thrust thruster concepts were considered; arcjets, isojets, and resistojets. All are characterized by heat addition to the propellant rather than by chemical reaction. Therefore, the specific impulse is primarily a function of the temperature of the exhaust gases.

The arcjet converts electric energy to thermal energy by discharging an arc through a gas. The thermal energy in the arc is produced by Joule heating, which is transferred to the propellant gas by conduction and convection. The heated gas is expanded in a nozzle to convert thermal energy into directed kinetic energy.

Very high gas temperatures and corresponding high specific impulse can be obtained in the arcjet. This high performance is achieved at the expense of electric power. A specific impulse of 1000 lb_f-sec/lb_m requires approximately 70 kilowatts of electric power per pound of thrust. The high gas temperatures result in dissociation of most gases, and the use of propellants containing carbon, such as methane or carbon dioxide, would result in carbon particles in the exhaust stream with significant performance loss and possible degradation of external surface coatings.

The OLS requires relatively long-life components (greater than 1 year) to minimize the crew time required for maintenance tasks. There has been

little development of arcjets in recent years, and the failure rates are not defined. However, the discharge of a high-energy electric arc may present problems in designing cathodes and anodes to meet the engine life requirements.

Isojets use encapsulated radioisotope fuel as an energy source for heating propellant gases. The propellant gas flows through tubes built into the fuel capsule and is exhausted through a jet nozzle to provide thrust. Electric power is required only for the operation of control valves. Specific impulses of 200 to 700 lb_f-sec/lb_m can be obtained, depending primarily upon the propellant gas used and the design temperature.

The radioisotope fuel should have sufficient half life so as not to be the controlling mode for engine replacement. The fuel form should be chosen to minimize nuclear radiation and thus minimize shielding weight. Plutonium 238 appears to be the best candidate fuel. Because a radioisotope emits energy continuously, special cooling provisions are required when the engine is in a pressurized atmosphere environment (ground handling, maintenance, onboard storage, etc.). The weight of the thruster, including shielding (more than 50 pounds per thruster), and the need for special cooling requirements make the isojet unattractive for manned vehicle applications.

The resistojet is similar to the isojet, except that the heat energy is obtained from electric energy. The propellant gas flows through passages heated by electric resistance elements. The specific impulse varies with the propellant gas used and the electric energy available. Approximately 200 to 700 lb_f-sec/lb_m can be obtained with 3 to 26 kilowatts per pound of thrust.

The resistojet concept is well developed and has an expected long-life capability. The resistojet was selected as the low-thrust concept to be used for additional evaluation of low-thrust options. The reasons for this selection are as follows:

1. The arcjet has a low development status.
2. The high specific impulse of the arcjet is offset by the high power requirement (relative to the resistojet).
3. The life problems of the arcjet are unknown.
4. The isojet has special handling requirements and hazards.
5. The excessive isojet thruster weight offsets the advantage of no electric power requirement.

Waste Gas Concept

Figure 3.5-1 shows that a low-thrust subsystem may be supplied with waste gas propellant or it can use stored propellant that is resupplied periodically.

The first propellant source to be examined is waste gas. The OLS ECLSS oxygen recovery cycle would be the primary source of waste gases. Carbon dioxide produced by respiration is combined with hydrogen in a Sabatier (if ECLSS uses a closed-loop concept) unit to produce methane and water. The water is then electrolyzed to recover the oxygen. The methane from the Sabatier unit must be disposed of and would be available as an RCS propellant. If the Sabatier unit is operated hydrogen rich, some hydrogen would also be available as a propellant. If the Sabatier unit is operated hydrogen poor, some carbon dioxide would be available as a propellant. The amount of available waste gas will be in direct proportion to the crew size. A waste gas low-thrust subsystem would compress the waste gases as they are made available (continual flow) and would store them in an accumulator. When thrust is required, the accumulator would blow down to a lower pressure as the gases are used.

Waste gas from the Sabatier unit would be filtered and passed through regulators before being compressed and stored in the accumulator. The regulators would maintain the constant back pressure required by the Sabatier unit. Vents would be provided to dump any excess waste gas when the accumulators are at maximum pressure.

The specific impulse and power requirements of resistojet thrusters are a function of both the operating temperature and the composition of the propellant. The specific impulse and power requirements for resistojets operating on CH_4-H_2 and CH_4-CO_2 mixtures at various temperatures are shown in Figures 3.5-13 through 3.5-16. Mixtures containing methane should not be operated at temperatures above 2000 R to avoid dissociation of the methane.

Resupply Concept

The other low-thrust propellant option from Figure 3.5-1 is the use of propellant resupply. This propellant could be of sufficient quantity to satisfy all low-thrust requirements or could supplement the waste gases available onboard the OLS. The propellants considered for this operation are hydrazine, ammonia, and hydrogen. They are selected for their commonality with fluids considered by the ECLSS and/or medium-thrust RCS.

The hydrazine low-thrust subsystem would be pressurized by a high-pressure nitrogen bottle. Hydrazine would be expelled from the tanks in liquid form and, consequently, would require an expulsion device. A surface tension device was assumed for this study because of its light weight and recycle capability. When the engine valve is opened, liquid hydrazine flows into a gas generator built into each thruster. The hydrazine reacts in the presence of a catalyst to form a mixture of nitrogen, hydrogen, and ammonia gases.

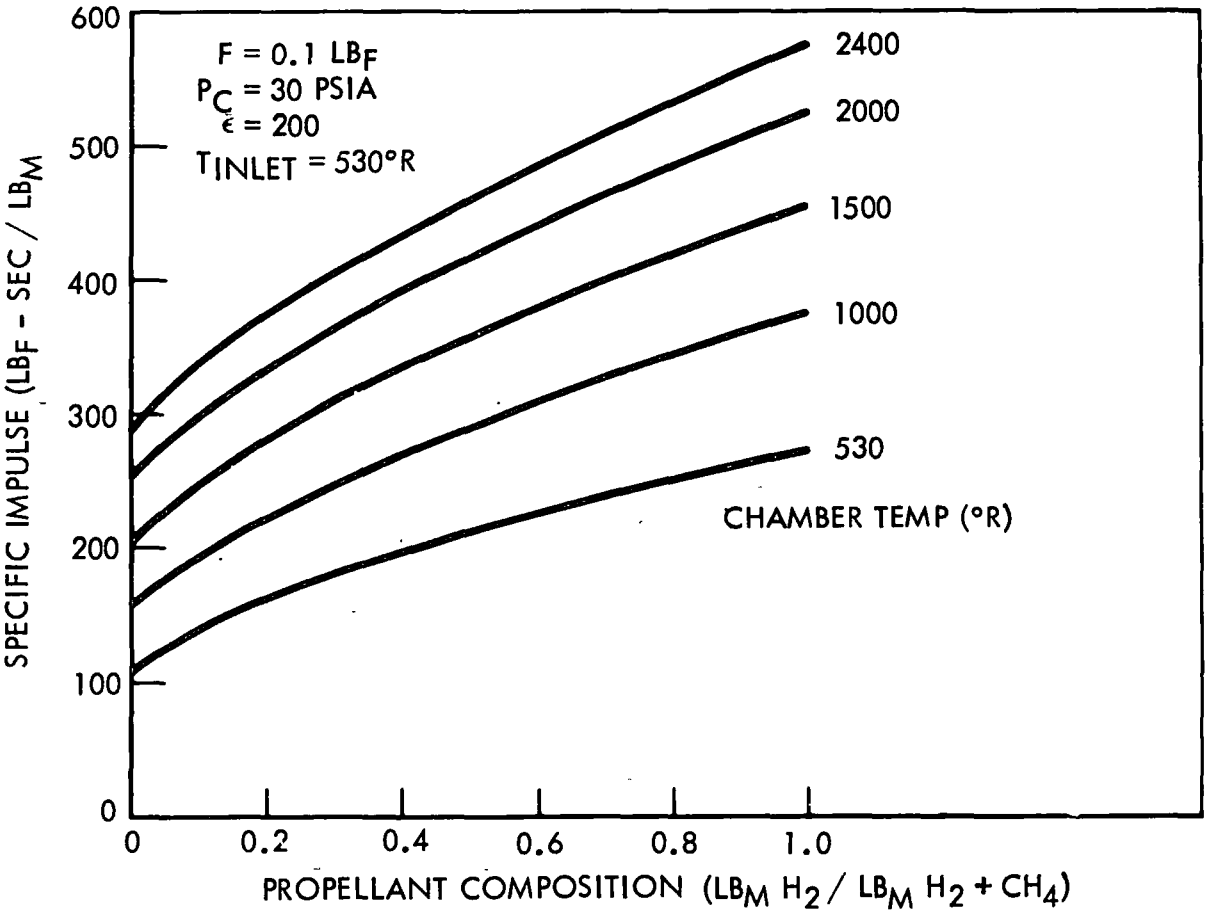


Figure 3.5-13. Specific Impulse of H_2-CH_4 Resistojet

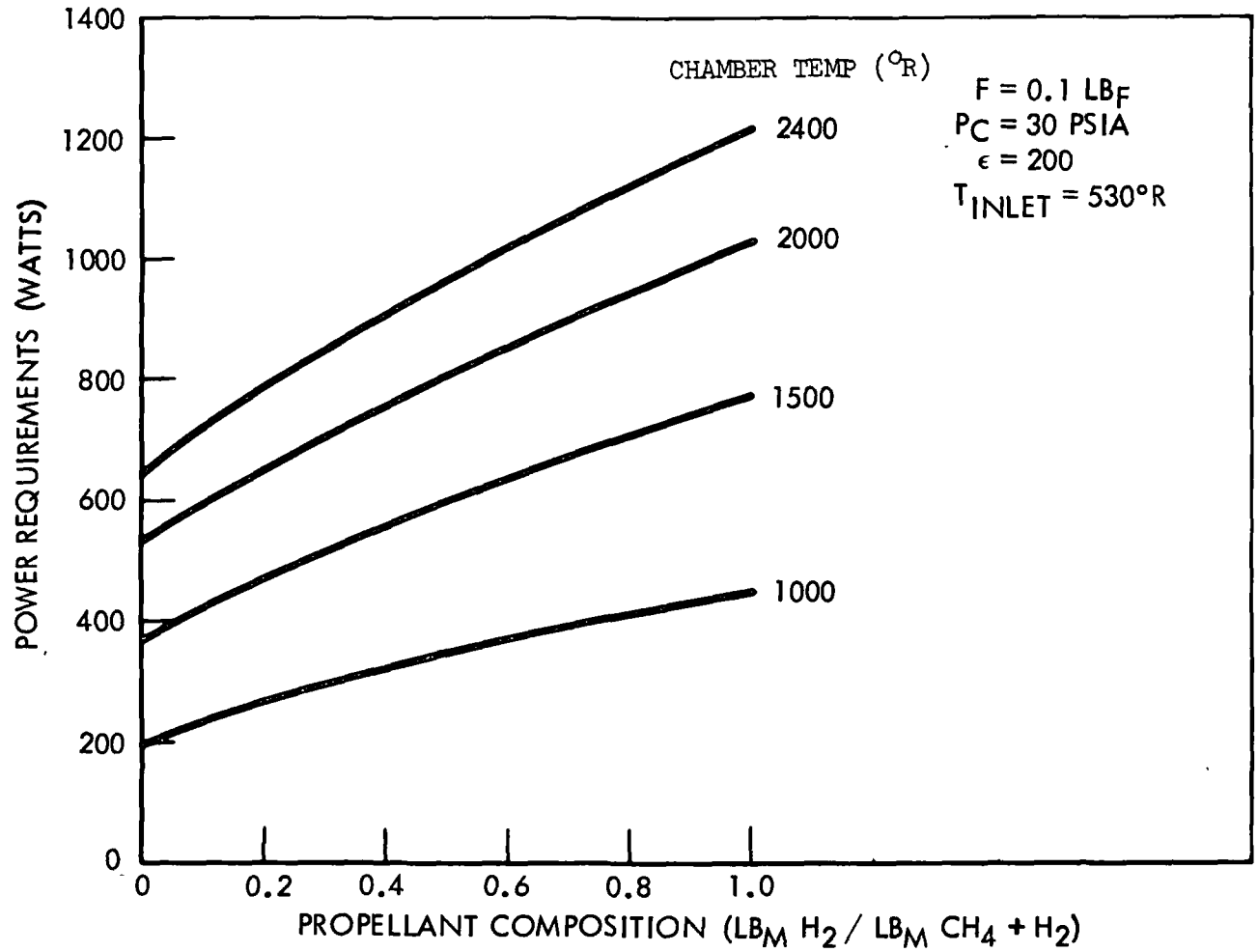


Figure 3.5-14. Power Requirement for H_2-CH_4 Resistojet

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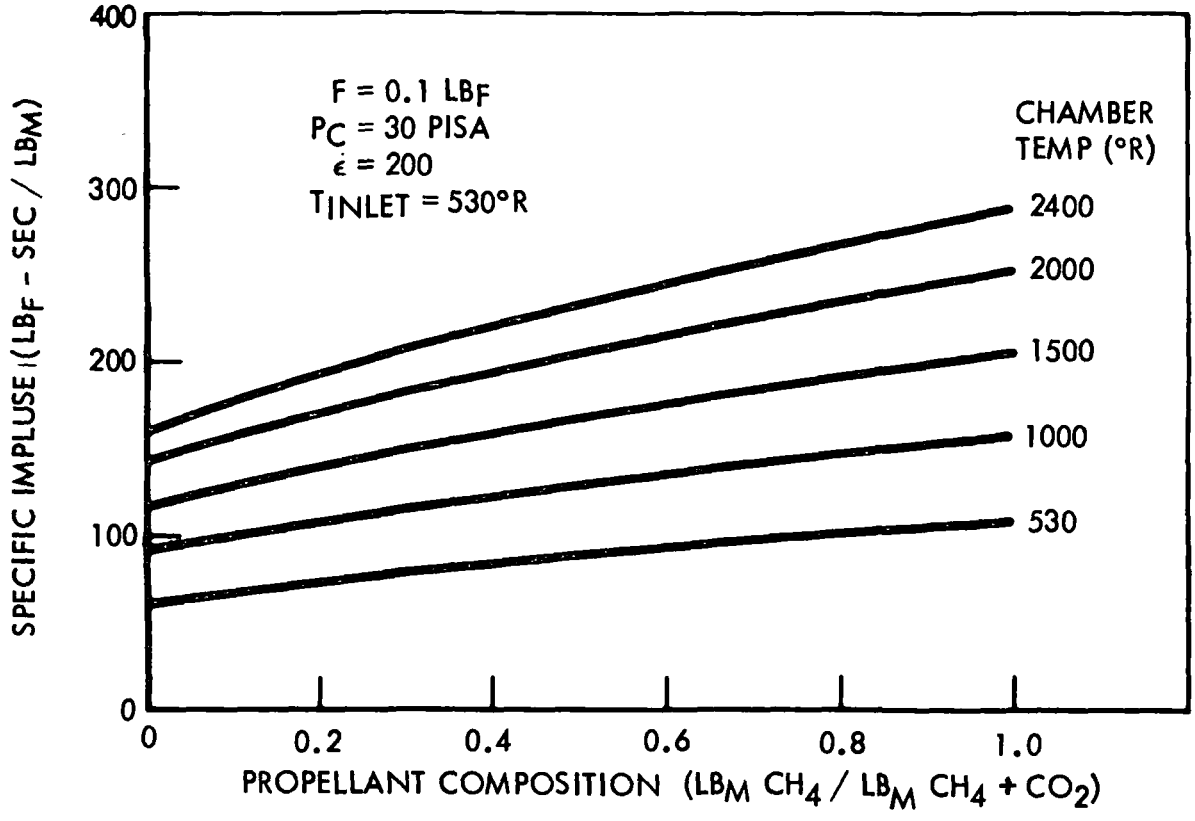


Figure 3.5-15. Performance of CH_4-CO_2 Resistojet

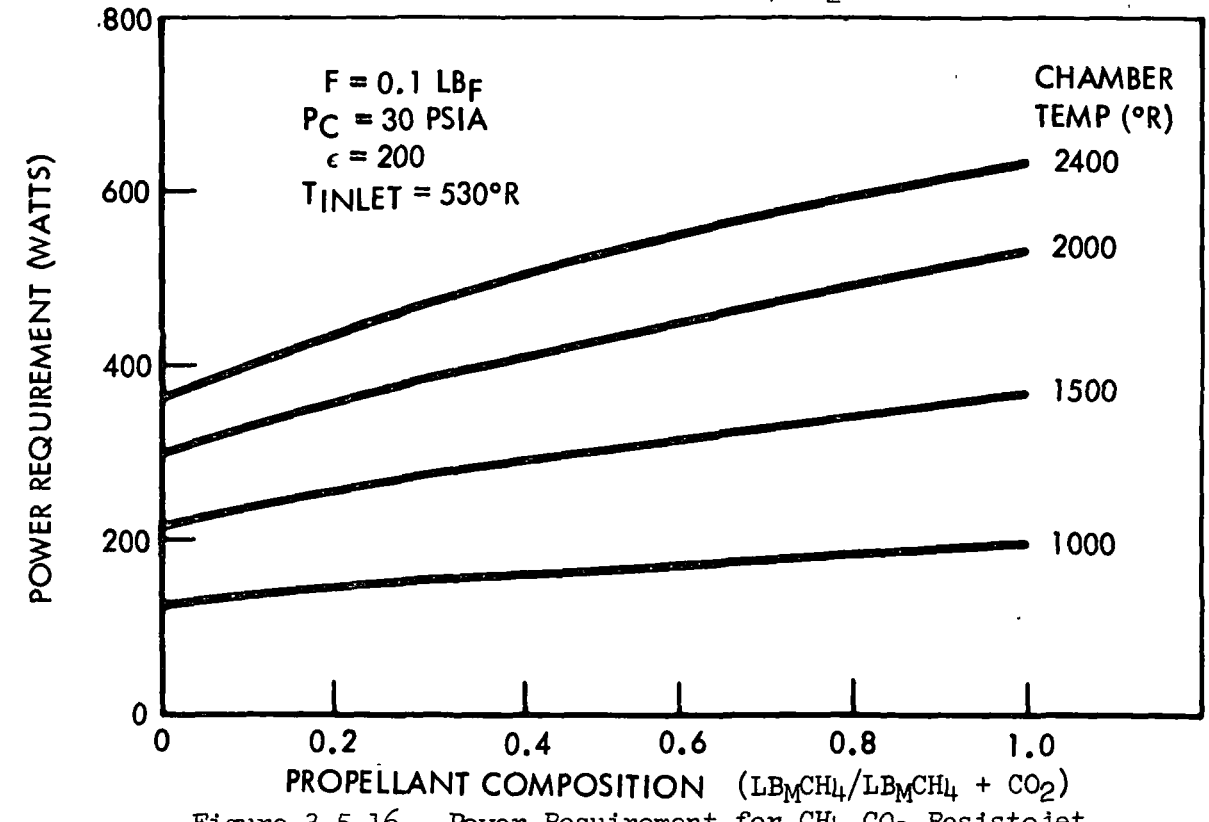


Figure 3.5-16. Power Requirement for CH_4-CO_2 Resistojet

The gas would pass into the resistojet and be expanded through a nozzle to produce thrust. Solenoid valves would be provided to isolate all major components. This subsystem would be essentially the same as the monopropellant medium thrust subsystem discussed previously and could be combined with a hydrazine medium thrust subsystem. The primary difference is that the gases are heated in the resistojet after decomposition by the catalyst.

Thruster power requirements and chamber temperatures for a 0.1-pound-thrust hydrazine resistojet are shown in Figure 3.5-17. Gas temperatures above 3000 R would require anhydrous hydrazine to prevent oxidation of heater elements. Figure 3.5-18 shows the subsystem dry weight as a function of total impulse. Figure 3.5-19 shows the resupply weight, and Figure 3.5-20 shows the average power requirement. Hydrazine low-thrust subsystems require less power than subsystems using other propellants because the overall gas generator reaction is exothermic. The inlet temperature of a hydrazine resistojet is at least 1560 R if the gas generator is part of the thruster.

There has been little or no development on a hydrazine resistojet as described. However, both hydrazine and resistojet technology is well developed. Integration of these two concepts poses no apparent problems.

An ammonia low-thrust subsystem would be pressurized by the ammonia vapor pressure of approximately 129.5 psia at the Orbiting Lunar Station ambient temperature of 530 R. Sufficient heat would be transferred through the tank walls to vaporize the ammonia and maintain tank pressure. The propellant supply is provided in the same way as previously proposed for the ammonia in the oxygen and ammonia medium-thrust concept. This subsystem could use common storage tanks with an oxygen and ammonia medium-thrust subsystem. Thrust is produced by heating the gaseous ammonia in a resistojet and expanding it through a nozzle.

Power requirements and chamber temperature for a 0.1-pound-thrust resistojet operating on ammonia are shown in Figure 3.5-21. Figure 3.5-22 shows subsystem dry weight as a function of total impulse. Figures 3.5-23 and 3.5-24 show resupply weight and power, respectively, as a function of total impulse.

A hydrogen low-thrust subsystem would be pressurized by the hydrogen's vapor pressure of 100 psia at the storage temperature of 52 R. For both boiloff and insulation weight to be minimized, insulation is sized to allow operation on heat leak for the lowest duty cycle. Additional heat is provided by a tank heater during high duty cycle periods. A heated accumulator assures that all propellant is vaporized and that a supply of gas is always available. Again, the propellant storage and even the accumulators could be common with an oxygen and hydrogen medium-thrust subsystem. Thrust is produced by heating the gaseous hydrogen in a resistojet and expanding it through a nozzle.

Power requirements and chamber temperatures for a 0.1-pound-thrust resistojet operating on hydrogen are shown in Figure 3.5-25. Figure 3.5-26 shows the subsystem dry weight as a function of total impulse. Figure 3.5-27 gives specific impulse and peak power as a function of chamber temperature.

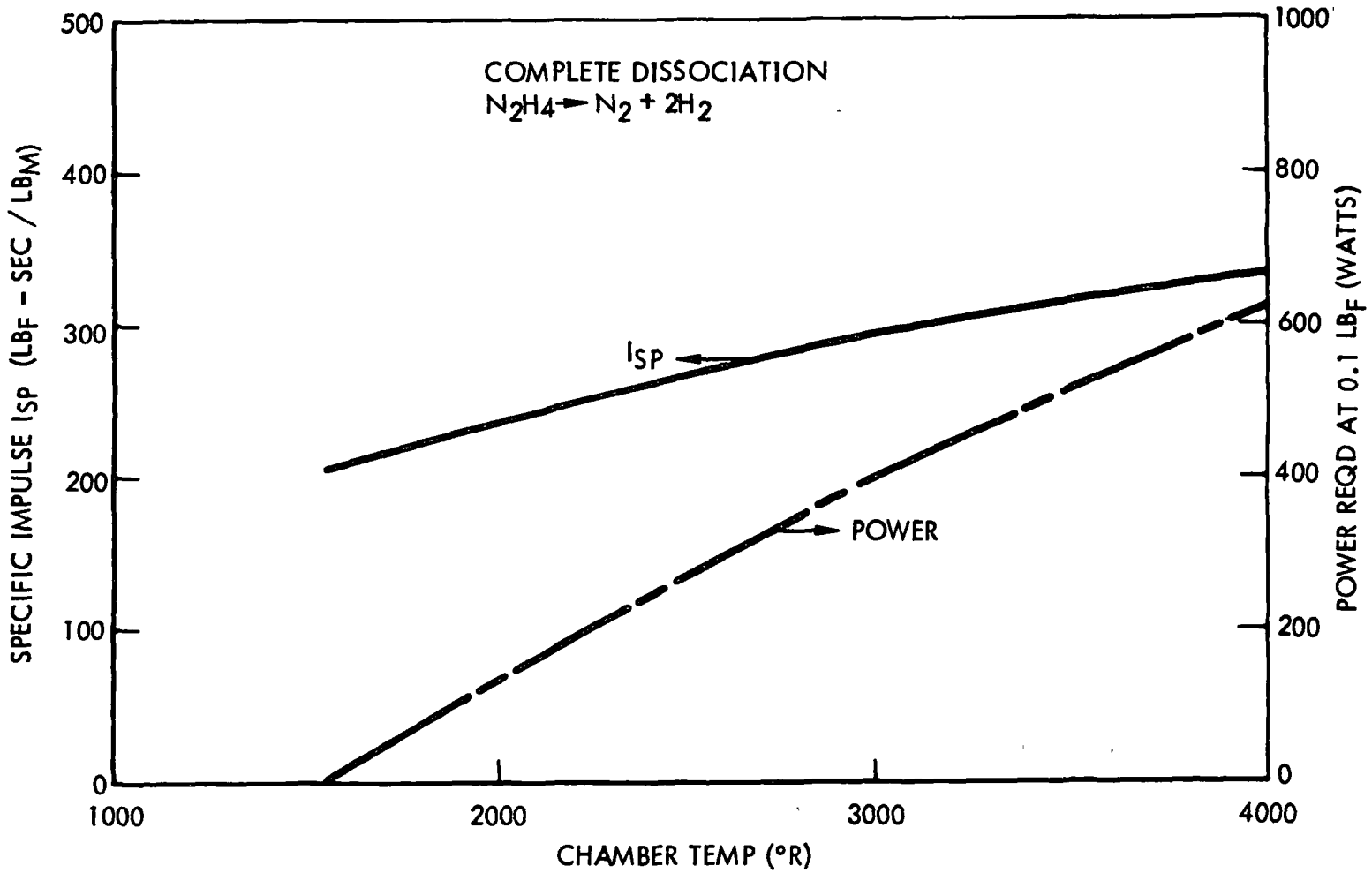


Figure 3.5-17. Characteristics for N_2H_4 Resistojet

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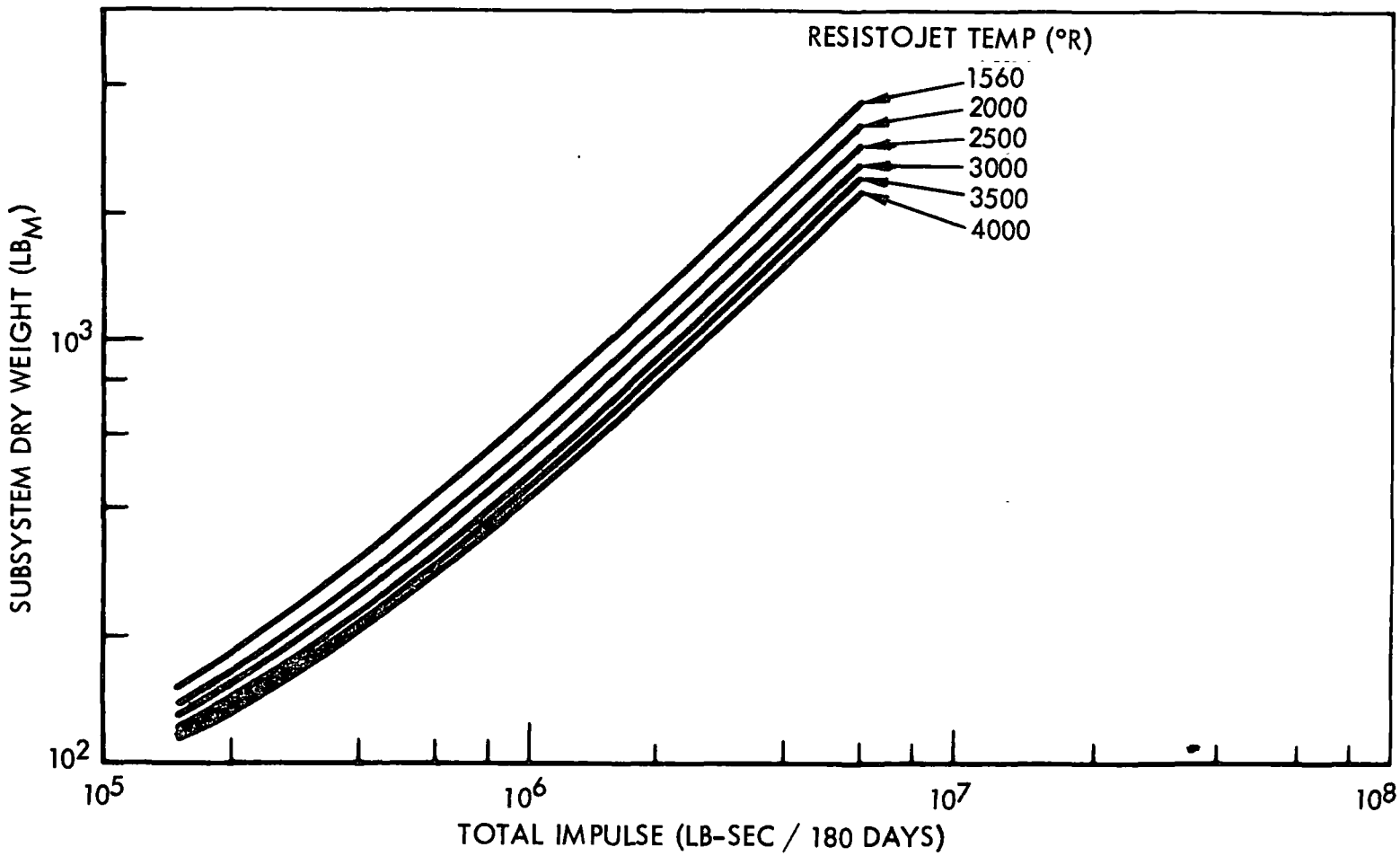


Figure 3.5-18. Dry Weight for N_2H_4 Resistojet Subsystem

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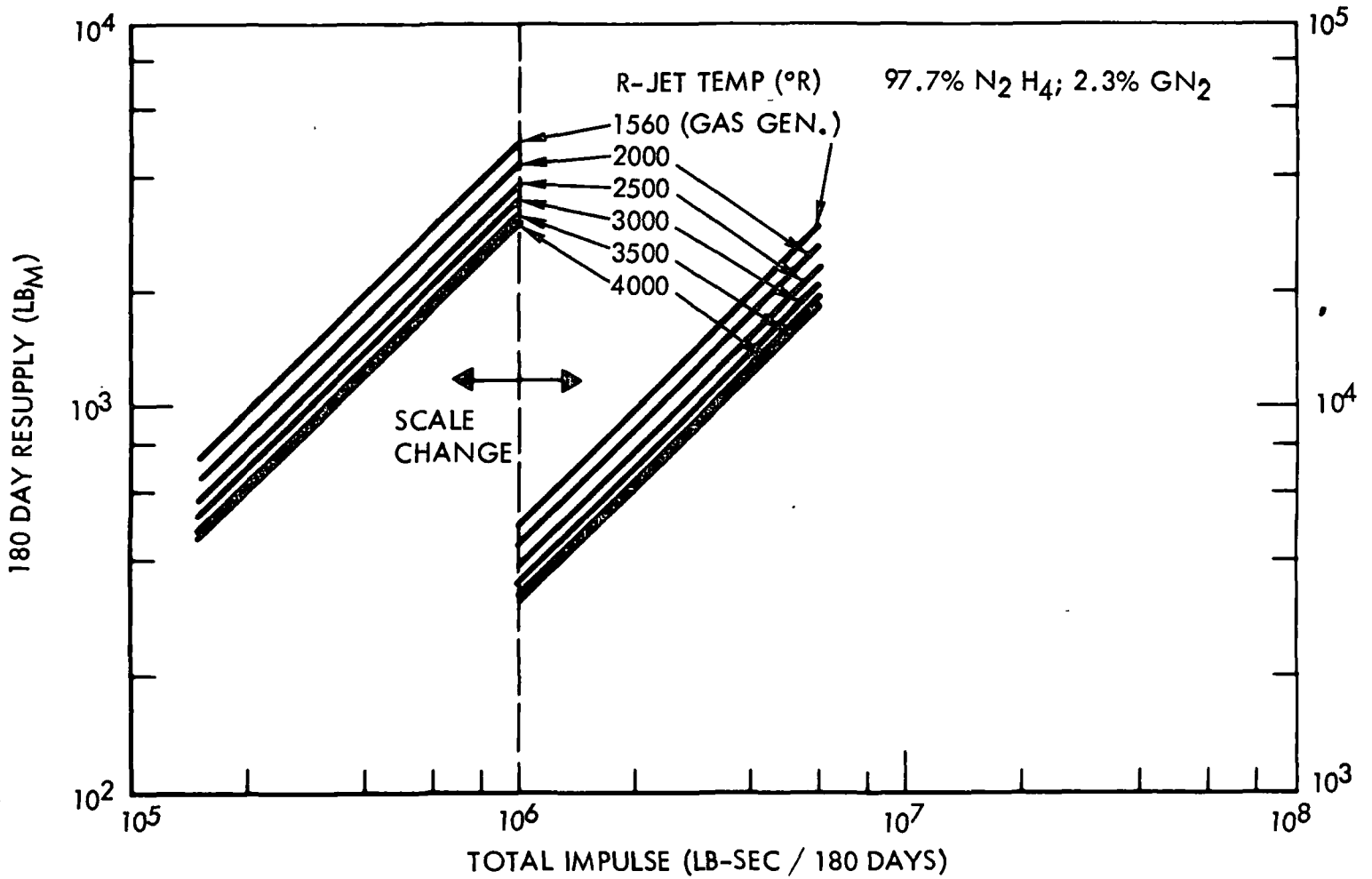


Figure 3.5-19. Resupply for N_2H_4 Resistojet Subsystem

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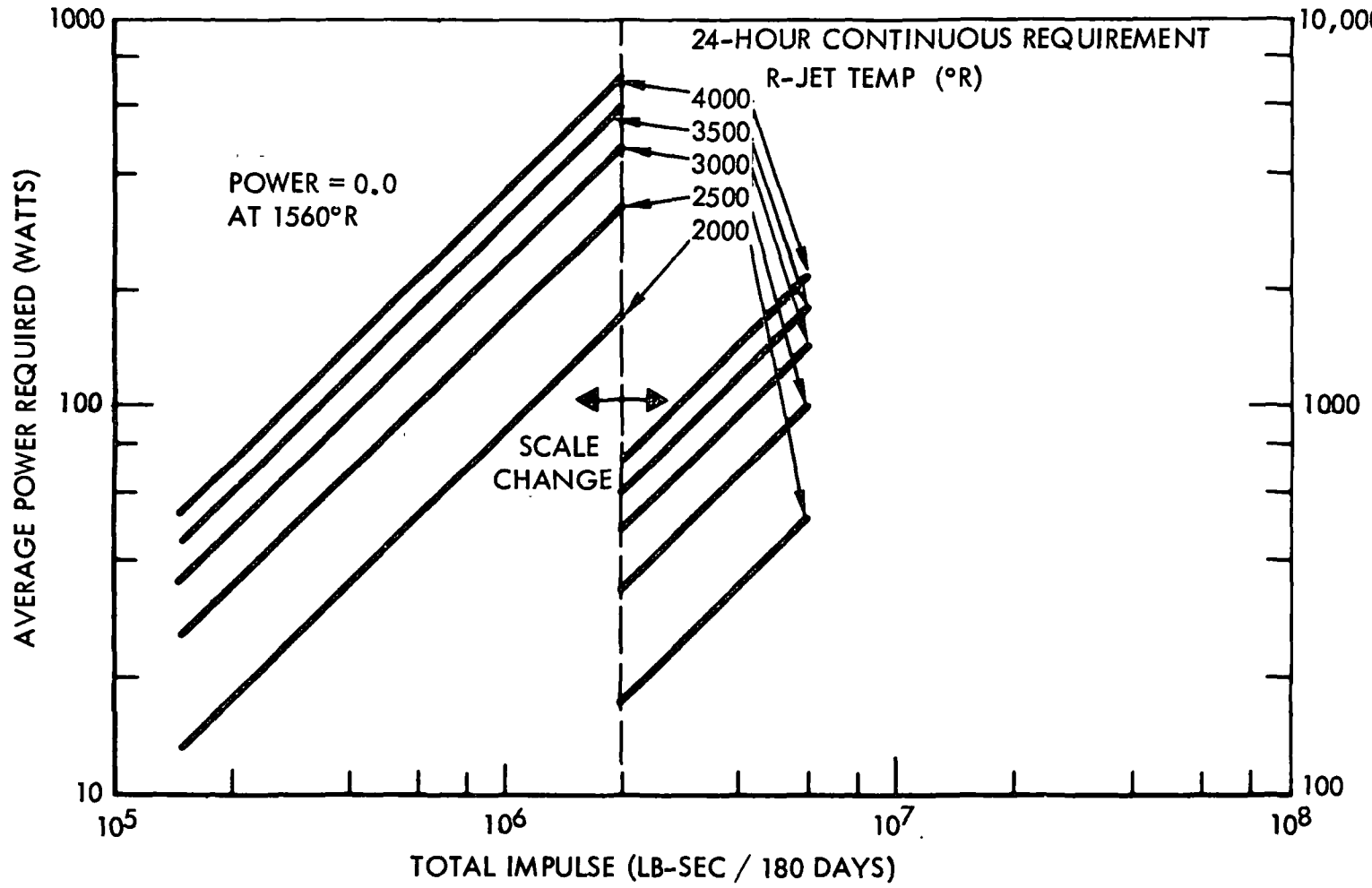


Figure 3.5-20. Average Power Requirement for N_2H_4 Resistojet Subsystem

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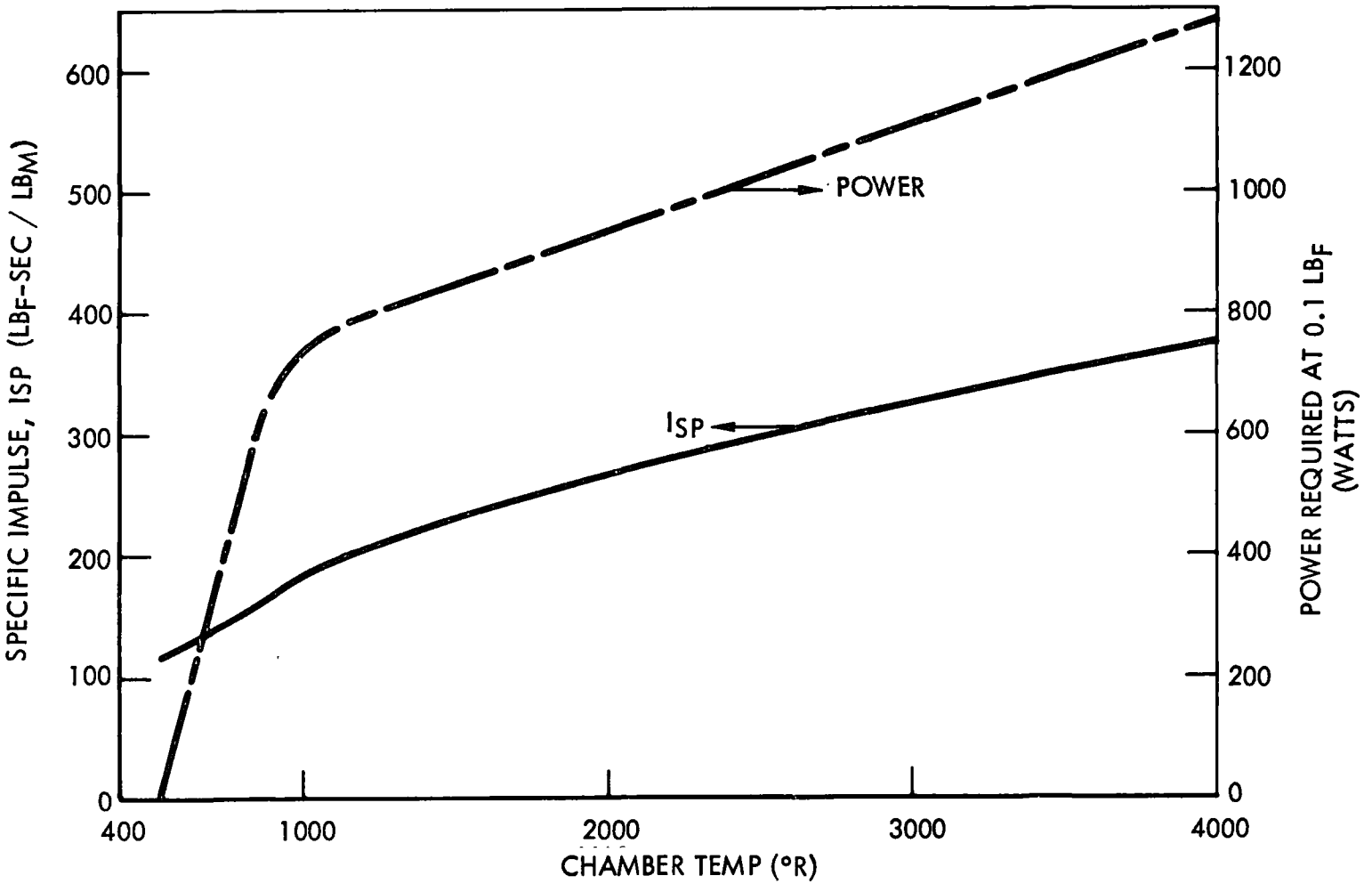


Figure 3.5-21. Characteristics of NH_3 Resistojet

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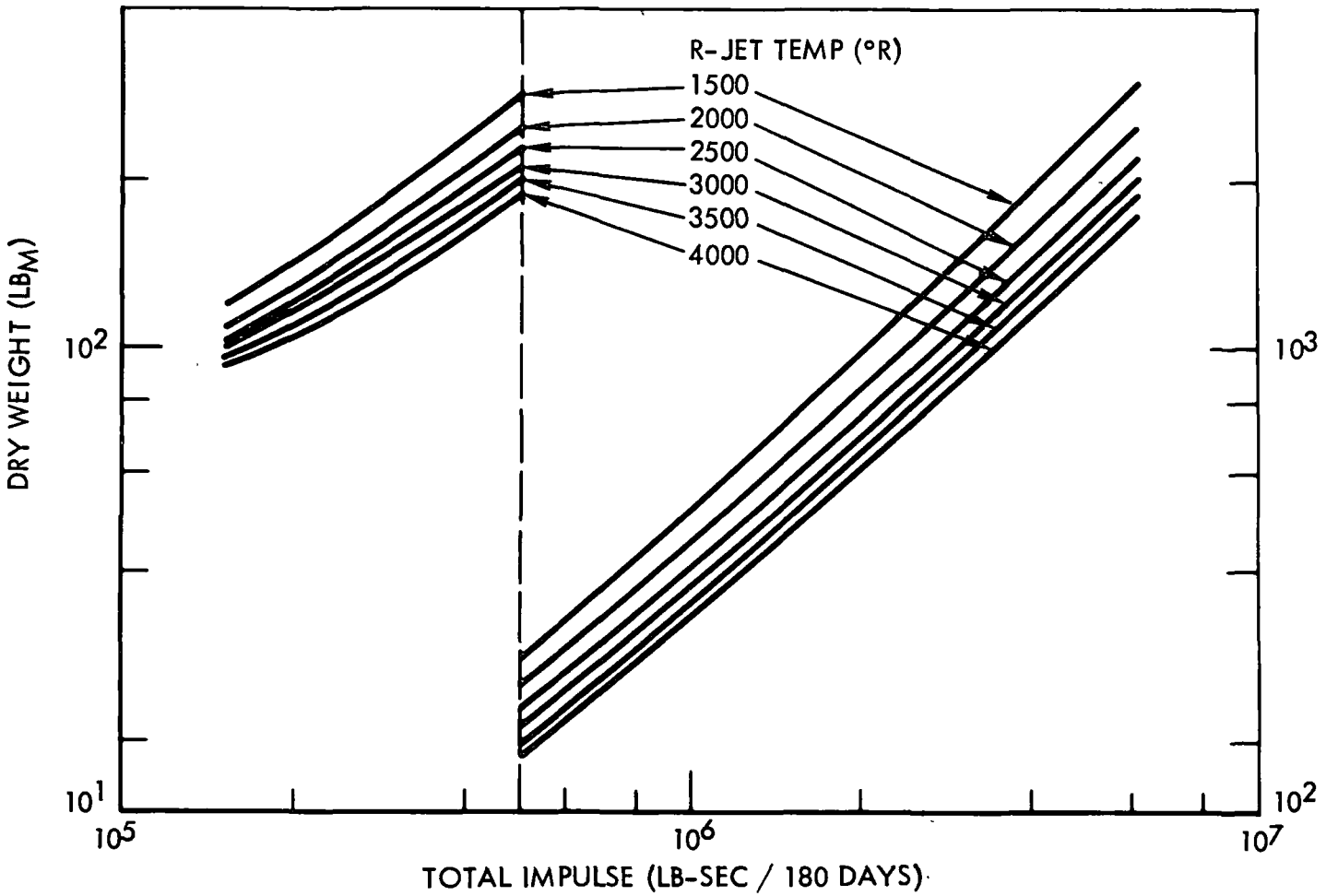


Figure 3.5-22. Dry Weight for NH_3 Subsystem

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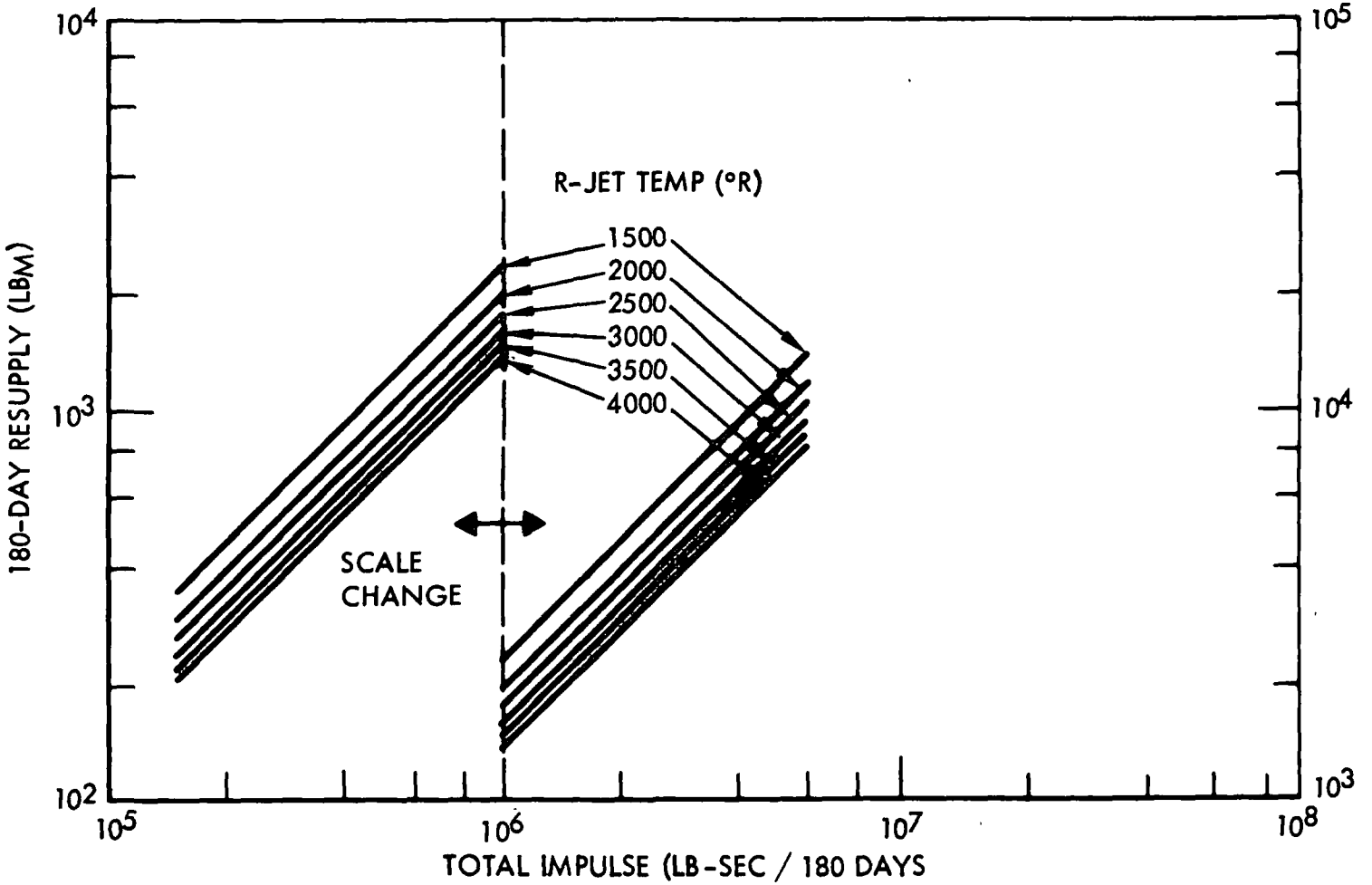


Figure 3.5-23. Resupply for NH_3 Resistojet Subsystem

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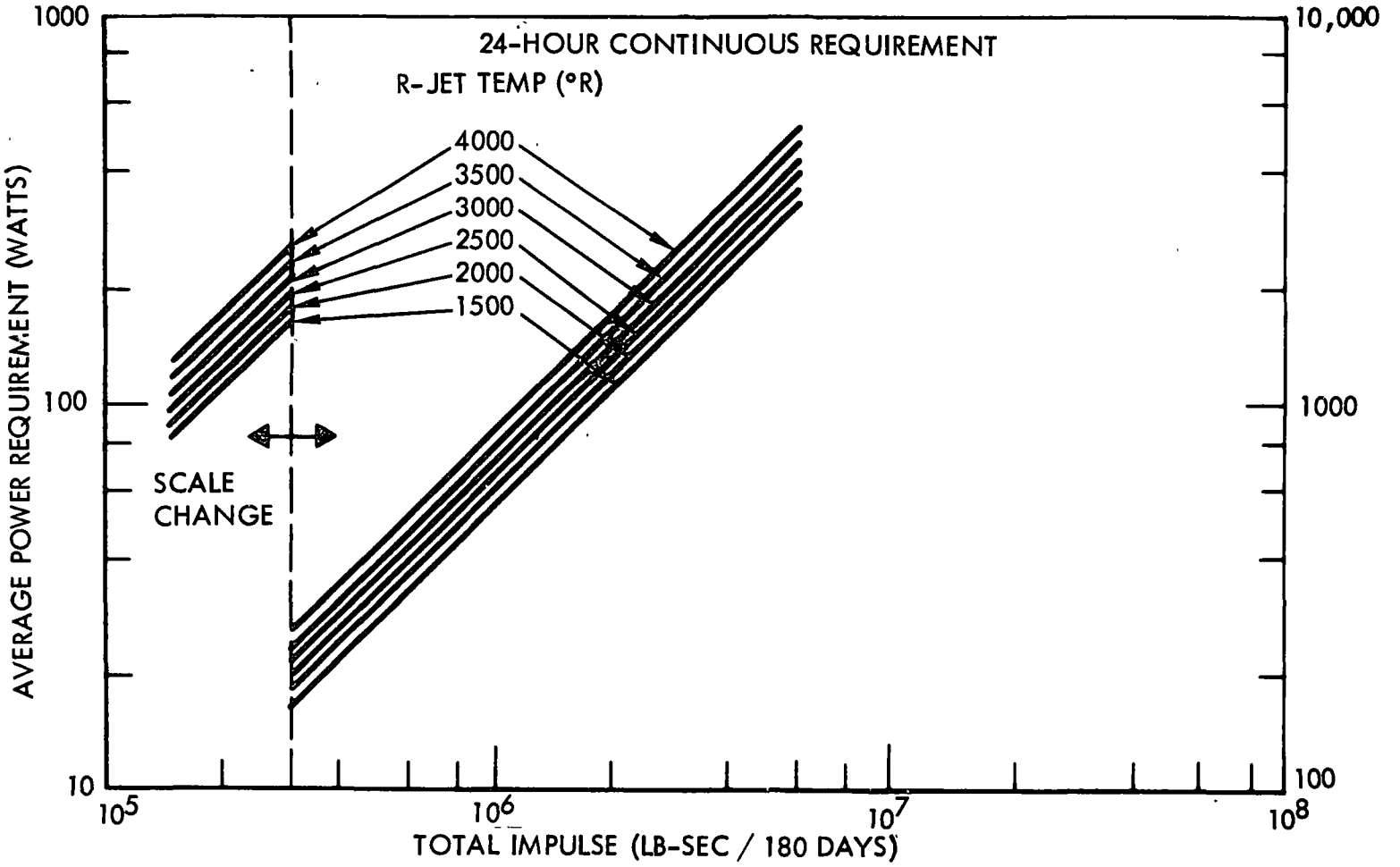


Figure 3.5-24. Average Power for NH_3 Resistojet Subsystem

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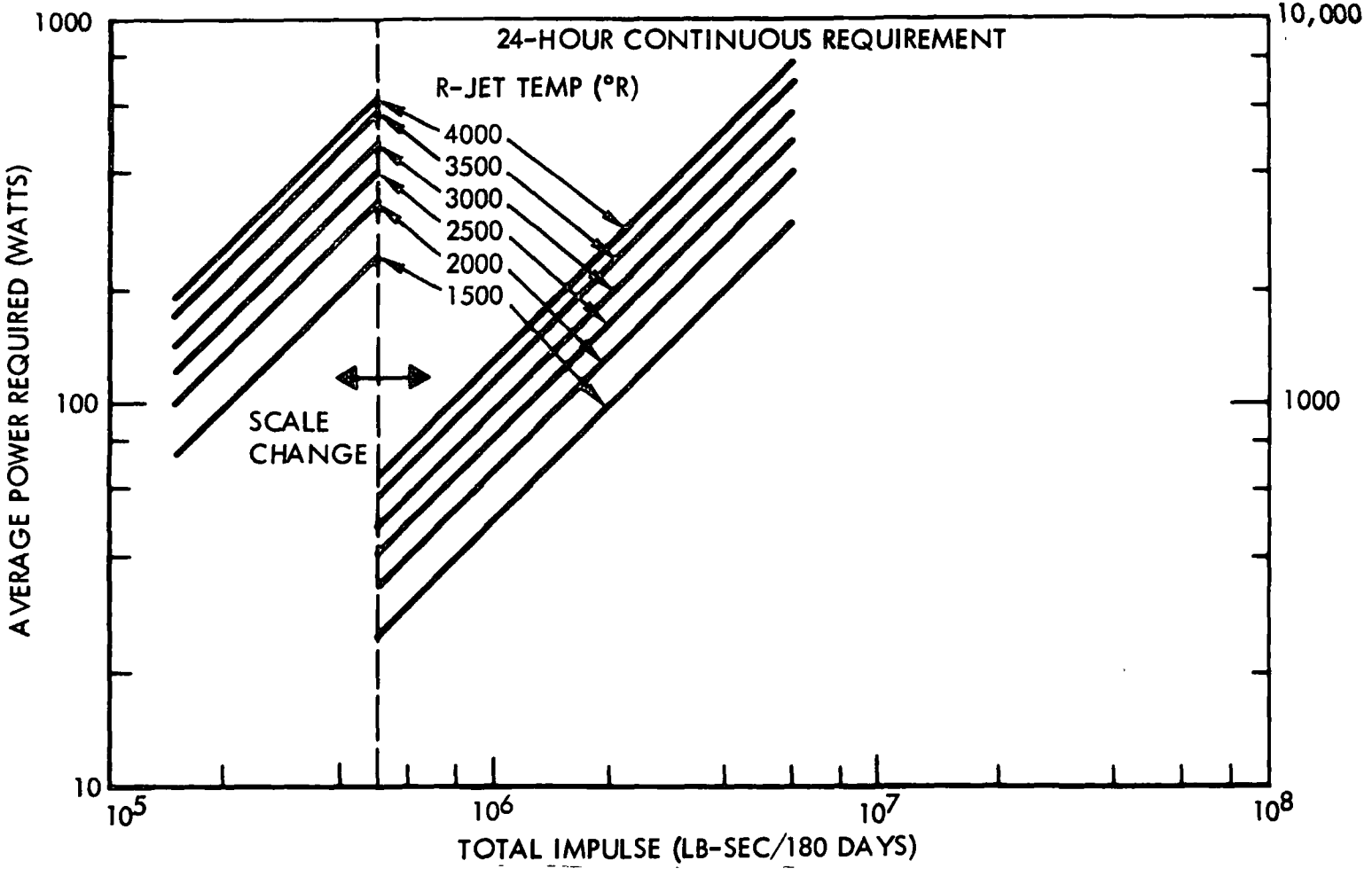


Figure 3.5-25. H_2 Resistojet Power Requirements

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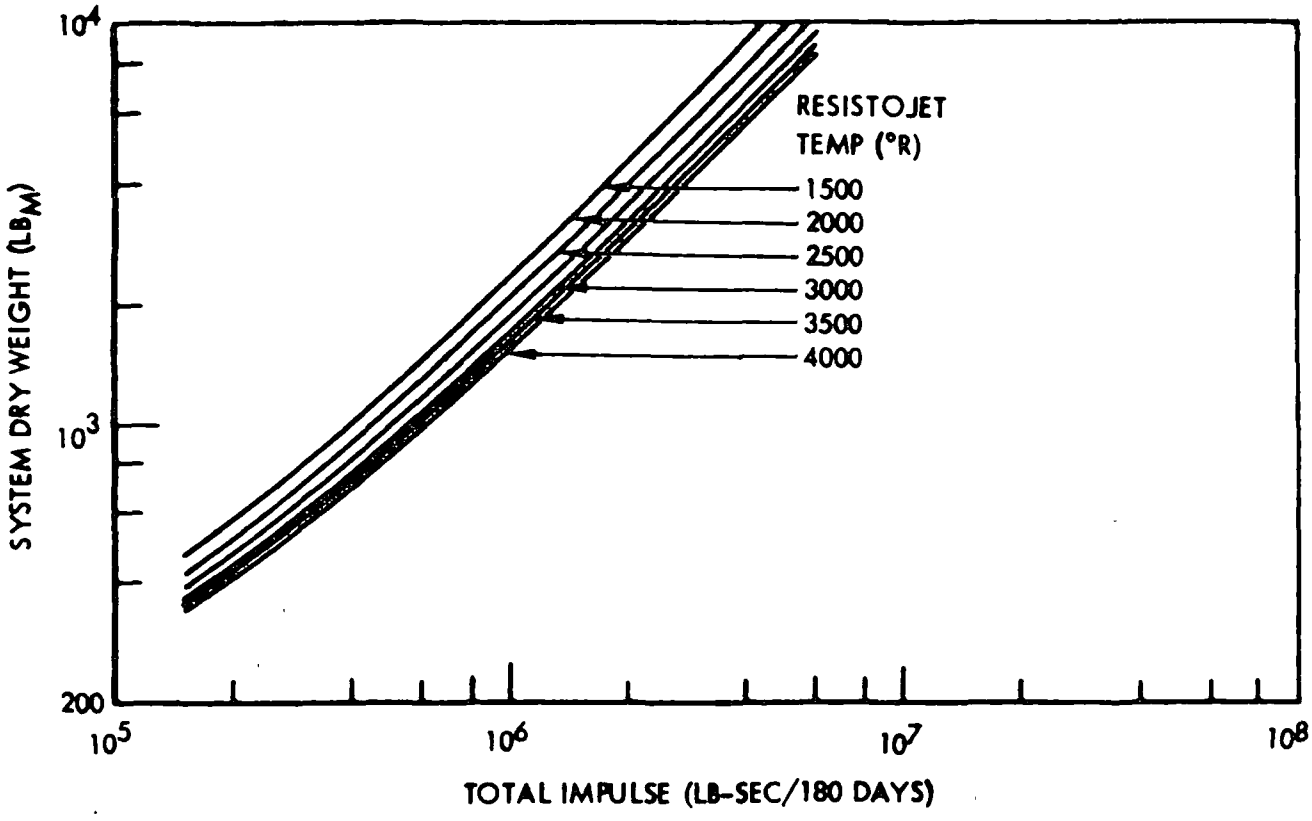


Figure 3.5-26. H_2 Resistojet Dry Weight

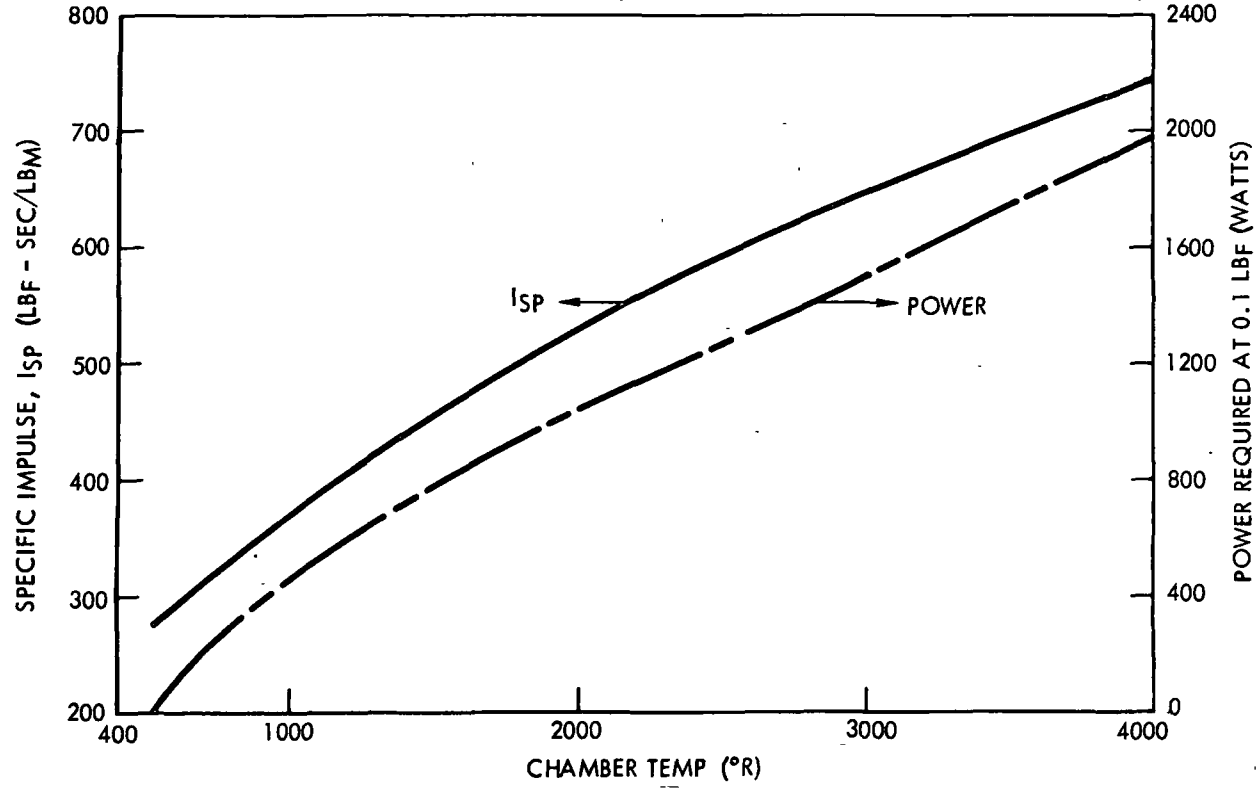


Figure 3.5-27. Characteristics of H_2 Resistojet

A low-thrust resistojet with resupply concept does not appear to have any great advantages over a medium-thrust only concept. There is usually a resupply savings, but this is not great. If the low-thrust and medium-thrust portions of the RCS use integrated propellant storage, there may be a dry weight savings, but this also is not great. Further, the ammonia and hydrogen concepts require rather high power. In view of these factors, the low-thrust resistojet using resupply does not appear to be a viable option for the OLS. The minor savings do not justify the complication and expense of an additional subsystem with significant power requirements.

3.5.3 GENERAL RCS CONCEPTS

Quiescent Mode

For some of the concepts considered for the OLS RCS, the imposition of a quiescent mode requirement is of little consequence. For other options, this mode is quite significant. The monopropellant hydrazine concept is not sensitive to quiescent mode shutdown. It may be shut down for a year or more without ill effect. The oxygen and ammonia concept is sensitive, but it can be designed to take up to a year of nonoperation without excessive difficulty. There will be a measurable loss of propellant through the relief valves, however. The oxygen and hydrogen concept is the most sensitive to quiescent modes. The problem is that there will be excessive loss of hydrogen. There is no reason, however, why the system could not be shut down for six months or longer. The hydrogen storage provisions would be designed to allow the OLS RCS to perform a stabilization and docking operation at the end of the quiescent period. This simply means that there is sufficient hydrogen capacity to allow continuous boiloff during the quiescent period plus sufficient capacity to perform the required maneuvers with a margin of safety.

There is another aspect to quiescent mode considerations, however. If the quiescent period is to last more than one or two weeks, it will probably be necessary to perform orbit maintenance. This requirement has little affect on the medium thrust options, but it will reduce the amount of hydrogen wasted in the oxygen and hydrogen concept. If a waste gas, low-thrust resistojet subsystem is included in the RCS for orbit maintenance and CMG desaturation, it will be unusable in an unmanned quiescent mode because there will be no waste gases. Orbit maintenance would then be accomplished with the medium-thrust subsystem.

Low Thrust - Medium Thrust Integration

The waste gas, low-thrust resistojet concept will save propellant resupply and total dry weight when used with a hydrazine or, to a lesser extent, with an oxygen and ammonia medium-thrust subsystem. When used in conjunction with an oxygen and hydrogen medium-thrust concept, the low-thrust concept will not save any resupply weight. This concept will increase the overall dry weight, the total subsystem complexity, and the subsystem cost. Again, the reason for this is the difficulty in storing liquid hydrogen for extended periods. If the medium-thrust engines are used for day-to-day operations of orbit maintenance and CMG desaturation, the hydrogen would be in constant use and would not be wasted. If, however, a separate subsystem

performed these functions, the oxygen and hydrogen subsystem would be dormant for extended periods. During these periods, heat leak to the hydrogen will continue, and the hydrogen will boil off through the relief valves. Heat leak will continue to the liquid oxygen, but the loss of oxygen will be very small by comparison. A low-thrust subsystem would not be used with an oxygen and hydrogen medium-thrust subsystem. The use of a low-thrust subsystem in conjunction with either a hydrazine or an oxygen and ammonia subsystem would have to be determined after the RCS requirements are more clearly defined. These requirements would be defined in Phase B of this program and preliminarily in the Phase A baseline selection.

Resupply Technique

The propellant resupply technique to be used on the OLS RCS is dependent partly upon which propellants are selected. Hydrazine and ammonia are relatively easy to handle. They may be transported with or in a cargo module. The cargo module would be equipped with positive expulsion propellant storage tanks and high-pressure nitrogen tanks. When the cargo module is docked to the OLS, propellant lines and pressurant lines in the docking hatches would be connected by an astronaut. The hydrazine or ammonia then would be forced into the RCS tanks by positive expulsion. For a monopropellant hydrazine design where high-pressure nitrogen also is needed by the RCS, the nitrogen tanks in the cargo module would be of a very high-pressure design. Before the nitrogen is used to transfer propellant, it would be used to resupply the RCS nitrogen tanks. This would be a slow process, probably taking several hours. If the propellants are transferred rapidly, the resupply tank temperature would drop sharply and would reduce the supply pressure. At the same time, the RCS tank would increase sharply in temperature and thus would increase the RCS tank pressure. The total effect of these factors is that the RCS tank does not get fully loaded. A slow nitrogen transfer process allows thermal conduction and thus maintains reasonably steady nitrogen tank temperatures and complete nitrogen resupply.

An alternative resupply method for the hydrazine propellant storage assembly would be to transport a separable propellant storage module as part of the cargo module. The tug would first dock the cargo module to the existing propellant storage module on the OLS. The existing module would be disconnected from the OLS and removed (and disposed of or stored) by the tug. The tug (still carrying the cargo module) would then return to the OLS and dock the cargo module. Other cargo would be transported through the new propellant module to the OLS. At the end of the cargo transfer operation, the main portion of the cargo module would be disconnected from the new propellant module and removed by the tug. There would be a sealable hatch on both ends of the propellant module to facilitate this operation.

Either of the above propellant transfer concepts may be used with the cryogenic RCS propellants. They may be transferred by positive expulsion (using helium rather than nitrogen) or by propellant module replacement; but there is another option. A large propellant module containing cryogenic oxygen and hydrogen is to be brought up on a regular basis to resupply the tug modules. If this module were designed to transfer propellants from both ends, it could be removed from the cislunar shuttle (CLS), using a tug, and

docked to the OLS long enough to supply the RCS. The amount of propellant required by the RCS will be very small in relation to the oxygen and hydrogen propellant module capacity of about 77,000 pounds. The method of RCS resupply will have a small impact on the propellant module design. The capability of having the propellant transferable from either end of the propellant module permits the tug to resupply its tanks without switching ends on the propellant module.

The oxygen in the oxygen and ammonia concept probably could be obtained from the large oxygen and hydrogen propellant module. However, this would mean that the ammonia would be obtained from a separate source, or that a small ammonia supply would be added to the large oxygen and hydrogen module. The first consequence is unnecessarily complicated in that two separate propellant supplies must be separately docked with the OLS. The second consequence is impractical. The necessary amount of ammonia is so small in relation to the large propellant module capacity that it would be more practical to make a complete, separate supply associated with the cargo module to resupply the OLS RCS.

The RCS resupply options are outlined in Table 3.5-5.

Table 3.5-5. Candidate OLS RCS Propellant Resupply Concepts

Propellant	Replaceable Propellant Module	Cargo Module Positive Expulsion	Tug Oxygen-Hydrogen Module
N_2H_4	X	X	
O_2/NH_3	X	X	
O_2/H_2	X	X	X

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3.6 ENVIRONMENTAL PROTECTION SUBSYSTEM (ENPS)

The OLS ENPS provides passive protection devices to protect the crew, structure, and subsystems equipment from thermal, meteoroid, and nuclear radiation environmental hazards experienced by the OLS during all flight phases.

During the boost-to-earth orbit phase, the ENPS provides protection to OLS structure from aeroheating. No parametric studies were performed on this protection, however, because it is considered primarily a function of the booster vehicle concept and only secondarily a function of the OLS configuration. Selection of the OLS aeroheating protection concept is believed to be a Phase B task.

During OLS earth orbit operations (preceding transport to lunar orbit via a cislunar shuttle), the ENPS, which is designed for lunar orbit application, is expected to provide more than adequate protection in all respects. Therefore, no further discussion is required here of the earth orbit phase of flight. The foregoing statements apply equally to the earth orbit to lunar orbit transport phase of flight. Thus, the remainder of this section is devoted to OLS ENPS requirements in lunar orbit. The ENPS requirements in lunar orbit are defined in Section 7.7 of Volume III. Consideration of concepts and parametric data to satisfy these requirements as a function of OLS configuration and crew size is presented in the following subsections.

3.6.1 THERMAL ENVIRONMENT

Planetary Orbit Heating

While in planetary orbit, the thermal flux incident on an elemental surface is the summation of the following sources: direct solar energy (insolation), reflected solar energy (albedo), and planetary emission.

$$\frac{\dot{q}_{incident}}{A} = \frac{\dot{q}_{solar}}{A} + \frac{\dot{q}_{albedo}}{A} + \frac{\dot{q}_{emission}}{A}$$

The following presents a simplified technique for representing these fractions of the orbital thermal environment in a manner that expedites analyses and provides correlation of total thermal environments through an artifice called "radiant sink temperature". Although the utilized heating rate calculation is not exact, it provides sufficient accuracy for most applications.

Direct incident solar energy flux is simply the solar constant ($S = 442$ Btu/hr-ft^2) multiplied by the cosine of the angle ϕ between the surface unit normal vector and the solar vector.

$$\frac{\dot{q}_{solar}}{A} = \hat{S} \cos \phi$$

The reflected solar energy flux reaching the surface is a function of the solar constant, the planetary albedo (a), the view factor from the surface to the planet (F), and the angle (θ) representing the surface angular location from the subsolar point.

$$\frac{\dot{q}_{albedo}}{A} = \hat{S}\, aF \cos \theta,\ 0 \leq \theta \leq 90$$

If θ is greater than 90 degrees, there is no reflected solar energy, Earth albedo is approximately 0.35 compared to lunar albedo of only 0.053.

The portion of solar energy incident on a planet that is absorbed (1-a) is emitted as infrared radiation (planetary emission). The emission flux incident on the surface of a vehicle orbiting a body with a long day and no atmosphere (such as the moon) depends upon the solar constant, the planetary albedo, the view factor of the surface to the planet, and the angular location from the subsolar point. For a planet with a short day and an atmosphere (such as the earth), planetary emission is approximately constant over the planet surface especially for low-inclination orbits. An expression accounting for these factors on the sunlit side of the planet is

$$\frac{\dot{q}_{emission}}{A} = \hat{S}\ (1-a)\ F\ \frac{\cos(b\theta)}{(4-3b)}$$

where b is an artifice to convert between the two types of planetary bodies. Set b=0 for earth orbit and b=1 for lunar orbit. For θ greater than 90 degrees, the lunar emission is a constant 2.2 Btu/hr-ft^2, and the earth emission is as calculated from the planetary emission equation (approximately 73 Btu/hr-ft^2).

The total energy flux incident on a surface at any point in orbit over a sunlit planetary surface (defined by θ) is

$$\frac{\dot{q}_{incident}}{A} = \hat{S}\cos\phi + \hat{S}aF\cos\theta + \hat{S}(1-a)F\ \frac{\cos(b\theta)}{(4-3b)}$$

The portion of the incident energy flux absorbed by a surface depends upon its absorbtivity to the thermal fluxes involved. A term called solar absorbtivity (α_s) is used to determine the fraction of direct solar and albedo incident energy absorbed. The absorbtivity to planetary emission is the infrared emittance, ϵ . The preceding equation then becomes

$$\frac{\dot{q}_{absorbed}}{A} = \alpha_s \hat{S}\cos\phi + \alpha_s \hat{S}aF\cos\theta + \epsilon \hat{S}(1-a)\frac{F\cos(b\theta)}{(4-3b)}$$

On the dark side, the only heat source is the planetary emission.

The view factor from an incremental area to the planetary surface is obtained from Figure 3.6-1.

The steady-state energy equation for a surface in planetary orbit is a balance between the energy generated, energy absorbed, and the energy emitted by surface.

$$\frac{\dot{q}_{emitted}}{A} = \frac{\dot{q}_{absorbed}}{A} + \frac{\dot{q}_{generated}}{A}$$

The energy flux a body emits is a function of its absolute temperature (T) to the fourth power, emissivity, and the Stephan-Boltzmann constant.

$$\frac{\dot{q}_{emitted}}{A} = \epsilon \sigma T^4$$

It is convenient here to represent the heat flux absorbed with an equivalent radiation sink temperature, T_{rs}.

$$\frac{\dot{q}_{absorbed}}{A} = \epsilon \sigma T_{rs}^4$$

where, from the expression for the total heat flux absorbed:

$$\sigma T_{rs}^4 = \frac{\dot{q}_{absorbed}}{A} = \hat{S}\left[\frac{\alpha_s}{\epsilon}(\cos\phi + aF\cos\theta) + (1-a)\,F\,\frac{\cos(b\theta)}{(4-3b)}\right]$$

The energy equation is simplified when expressed in terms of the radiation sink temperature,

$$\epsilon \sigma \left(T^4 - T_{rs}^4\right) = \frac{\dot{q}_{gen}}{A}$$

and $\dot{q}_{gen}$ can now be considered as the total required heat rejection.

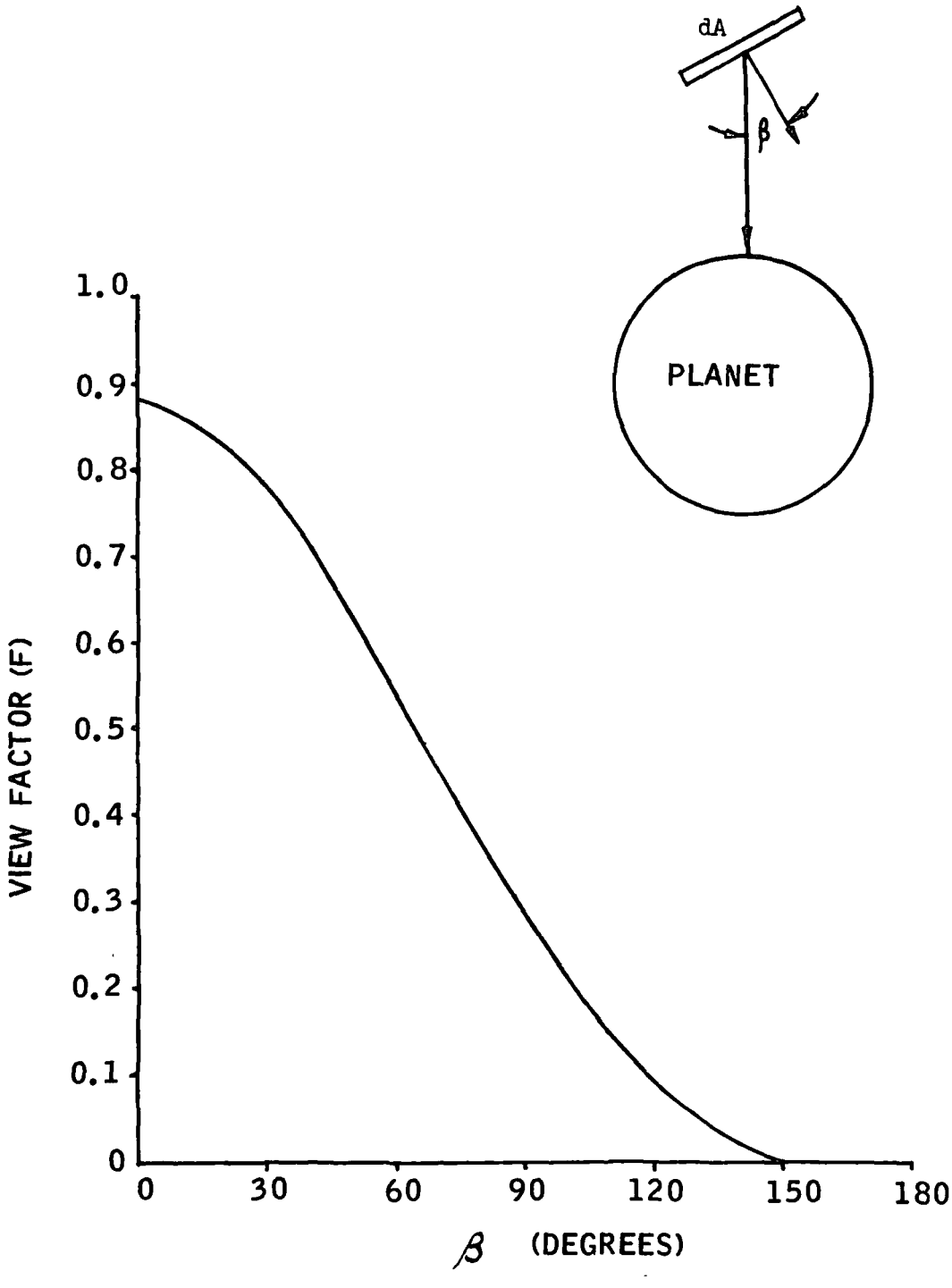


Figure 3.6-1. View Factor From Plate to Planet

The radiation sink temperature concept is useful for comparing the α_s and ϵ temperature effect on thermal environments of different planets. A nomograph (Figure 3.6-2) is supplied to aid in the calculation of the radiation sink temperature from the equation for total energy absorbed. Figure 3.6-2 can be used to define (1) the thermal environment at a specific location on a spacecraft at a certain instant in an orbit, (2) the average thermal environment over the spacecraft surface (when isothermalized by the use of heat pipes, for instance) at any location in an orbit, or (3) to obtain the mean radiation sink temperature (average over an orbit) of surface by first summing or integrating the heat flux with proper time limits and then using the average heat flux ($\bar{q} / \epsilon A$) to find an equivalent radiation sink temperature. The following example illustrates an application of the nomograph solution in a situation of the second type (an isothermal cylinder in lunar orbit):

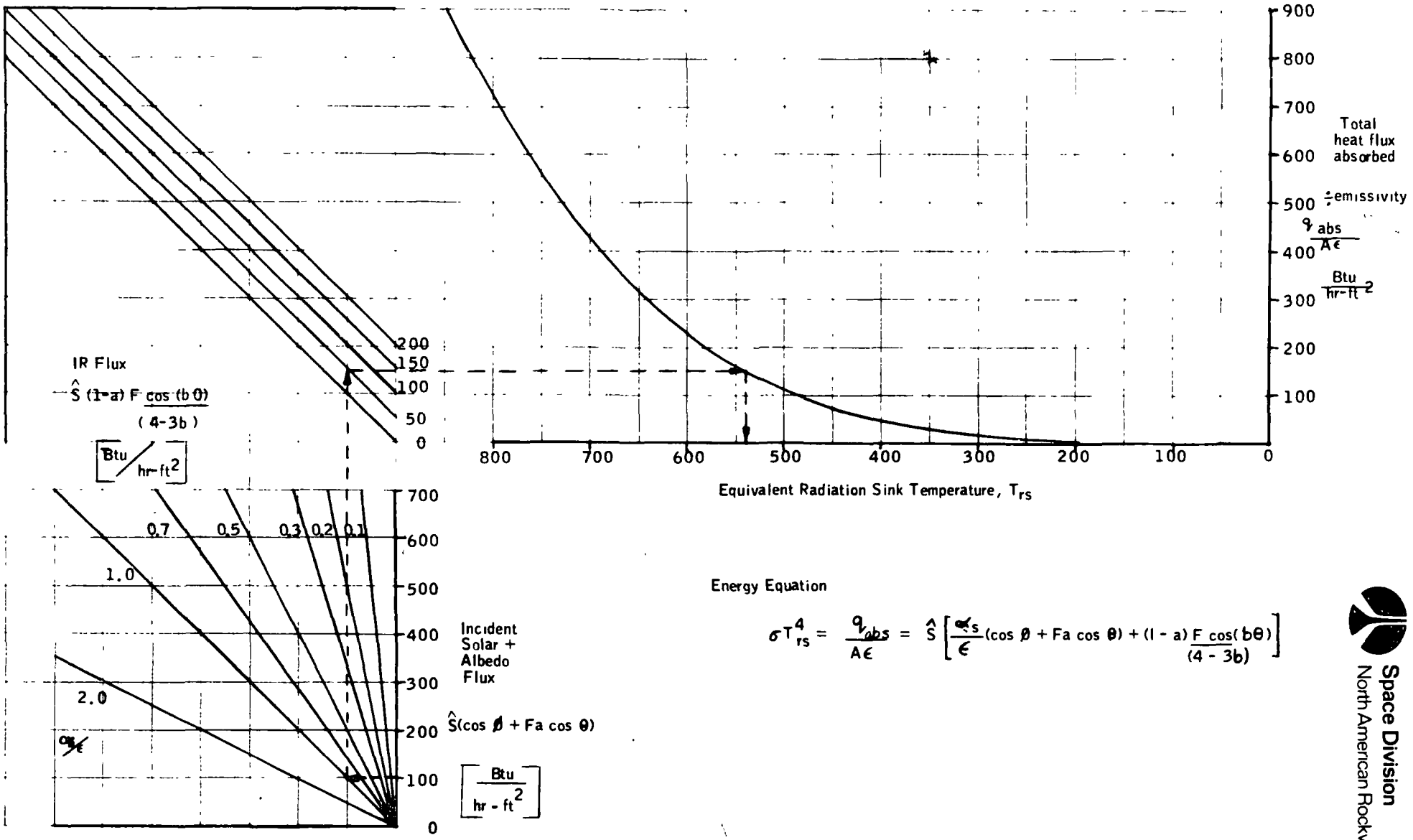
> When averaging heating rates over a nonplanar area, certain modifications to the energy equation are required. For instance, the view factor to the planet must be integrated to obtain an area average value. For a cylinder in a 60-n mi subsolar lunar orbit, in which the cylinder axis remains perpendicular to the solar vector, the average view factor to the moon is 0.378. A further modification to the energy equation is needed to reflect the difference between the area receiving radiation (DL) and the total area emitting radiation (π DL). Hence, for the cylinder the solar constant is divided by π . Other bodies of nonplanar shape require similar modifications to the energy equation. The highest radiant sink temperatures for the cylinder occur at the subsolar point. Radiant sink temperature varitations with the angle from the subsolar point are shown in Figure 3.6-3 for an isothermal cylinder with adiabatic ends. For a nonisothermal cylinder the radiation sink temperatures in the regions facing the sun and moon will be higher than the average isothermal cylinder values shown in the curve. The impact of low absorptivity/emissivity (α_s/ϵ) coatings on reducing the radiation sink temperature is illustrated by the curve.

Thermal Control Concepts

EOSS Type Active Fluid-Loop System

The primary functions of a thermal control system are to reject heat, transport heat, and control temperature. The OLS could employ an EOSS-like thermal control system that would utilize active fluid-loop, body-mounted radiators for heat rejection and active fluid-loops for heat transport within the pressure volume. In this event, design of passive thermal control elements is guided in a manner which:

1. Minimizes the environmental influences to the active system
2. Thermally couples external assemblies and components to the temperature stabilized volume where possible, if appropriate



Energy Equation

$$\sigma T_{rs}^4 = \frac{q_{abs}}{A\epsilon} = \hat{S}\left[\frac{\alpha_s}{\epsilon}(\cos\phi + Fa\cos\theta) + (1-a)\frac{F\cos(b\theta)}{(4-3b)}\right]$$

Figure 3.6-2. Nomograph to Calculate Equivalent Radiation Sink Temperature From Energy Equation

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- Isothermal cylinder
- Adiabatic ends
- Subsolar orbit
- Cylinder - to solar vector

$$\sigma T_{rs}^4 = \frac{q_{abs.}}{A\epsilon} = \frac{\alpha_s}{\epsilon} \left(\hat{S} \cos \phi + \hat{S}Fa \cos \theta\right) + SF(1-a)\frac{\cos(b\theta)}{(4-3b)}$$

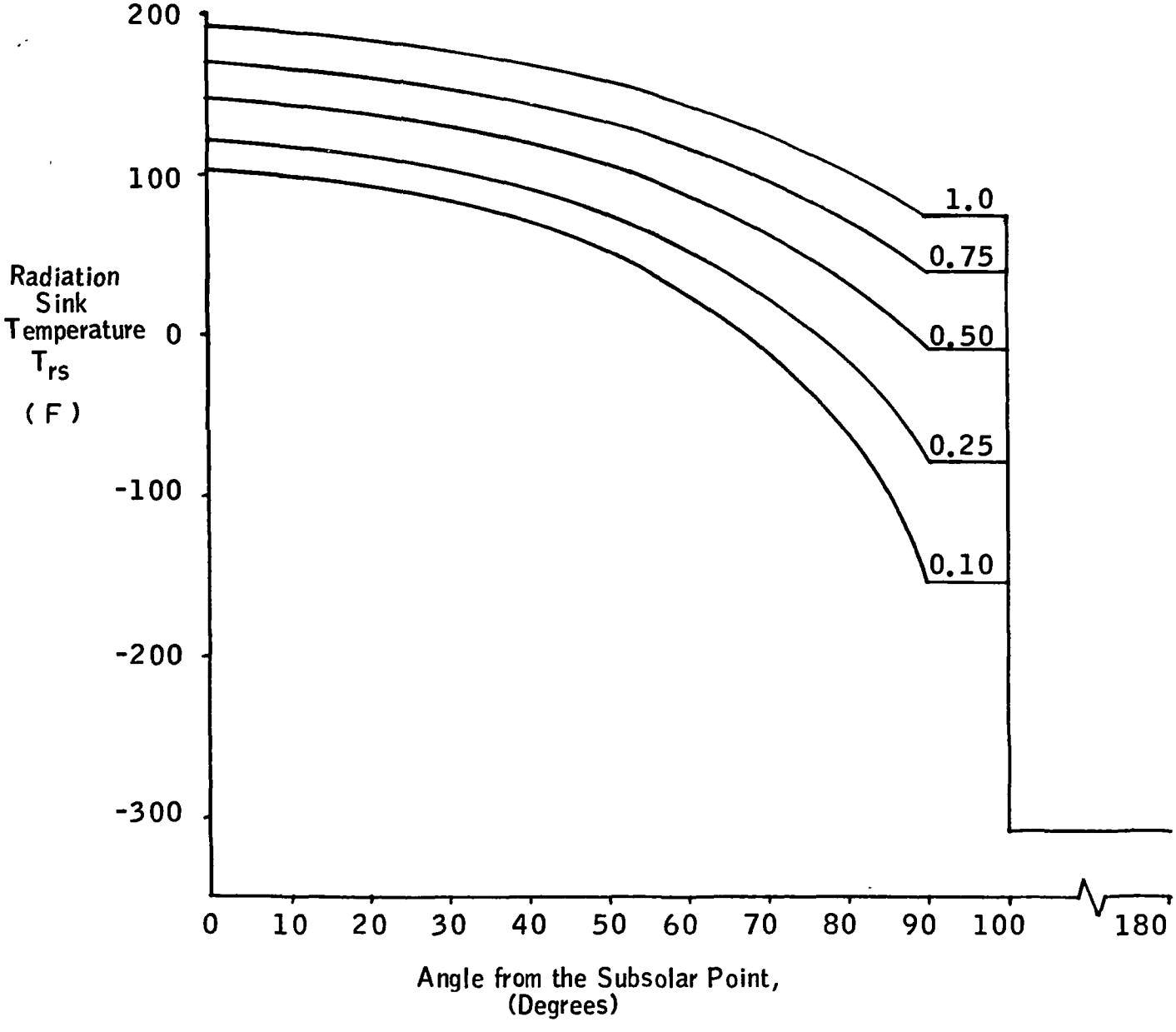


Figure 3.6-3. Radiation Sink Temperature Versus Position in Orbit for Various α_s/ϵ Ratio Coatings

3. Provides thermal control to other components which required protection from the hostile thermal environment but could not, or should not, be coupled to the pressure volume ambient thermal conditions.

Accomplishment of the first two of these objectives was centered about a primary passive thermal control element, which thermally isolates the active elements from the external ambient environment. The element, a high-performance insulation consists of multiple layers of a metalized polymeric film such as aluminized Mylar or Kapton, through which structural penetrations and heat shorts have been minimized. It is attached to the micrometeoroid bumper panels, which provide handling and boost protection. High-performance insulations require very low interstitial pressures (less than 10^{-4} torr). Therefore, the insulation must be isolated from pressure vessel leakage by a polymeric film sealed to the bumper.

Alternate Thermal Control Concept

Another thermal control concept worthy of consideration would rely heavily, if not exclusively, on passive thermal control elements. The following paragraphs describe a candidate OLS heat rejection control concept based on louvers.

Louvers consist of a set of parallel highly polished thin blades. Each blade is controlled by a bimetallic sensor, which rotates the louver blade when a temperature change is sensed. The louver regulates thermal radiation by varying the effective emittance of the louvered surface as the blades open and close. A typical range of emittance is 0.07 (fully closed) to 0.8 (fully open). Because the louvers operate independently, the failure of one or several louver blades will not effect the performance of the remaining blades and the louver system will continue to function at a slightly degraded but nonetheless operational level.

An entirely self-actuating louver system requiring no computer logic or control can insulate to prevent both heat loss from the spacecraft in times of reduced activity and heat gain to the spacecraft in severe thermal environments. A moment later the same area, with the louver blades open, can operate as a space radiator to reject excess heat. The louver concept is not unproven. Louver platforms weighing less than 1 lb/ft^2 have been utilized successfully on numerous unmanned spacecraft.

Two louver system configurations are discussed here. A louver system consisting of a single platform of louvers can be located between the pressure wall and the micrometeoroid bumper. The pressure wall temperature controls the louver actuator. A double louver system would have two louver platforms between the pressure wall and micrometeoroid bumper, one controlled by the pressure wall temperature and the other by the micrometeoroid bumper temperature. The application of the louver systems are illustrated herein.

Figures 3.6-4, 3.6-5, and 3.6-6 illustrate heat rejection capability as a function of radiant sink temperature for a number of heat rejection control system configurations and pressure wall temperatures. Curve 1 in these figures is included primarily to demonstrate the heat rejection capability of an unobstructed radiator. Curve 2 demonstrates the decrease in heat rejection from the first configuration caused exclusively by the presence of a passive wall (required for protection from micrometeoroid penetration). Curves 3 through 6 show heat rejection for both open and closed single- and double-louver assemblies. It should be noted that the heat rejection from an unobstructed surface (Curve 1) is reduced approximately 50 percent by the addition of the micrometeoroid protection shield (Curve 2). The single-louver system is capable through complete self-actuation to adjust to heat rejection rates within the region bounded by Curves 3 and 5. Likewise, the double-louver system is through self-adjustment able to operate in the range between Curves 4 and 6. A comparison of the maximum heat rejection rates for the louver systems shown the single louver system (Curve 3) slightly higher than the double system (Curve 4), but slightly lower than high emissivity ($\epsilon = 0.95$) plain walls (Curve 2). The double-louver system has a minimum heat rejection rate about the same as the single louver because the outer louver blandes controlled by the micrometeoroid bumper temperature would be in the open position.

To provide for maximum heat rejection, the micrometeoroid bumper outer surface must have a high emissivity. In an environment where there is only solar radiation, low α_s/ϵ coatings can be used to maintain low micrometeoroid bumper temperatures and prevent unwanted heat gain to the pressure wall. However, in lunar orbit there is also a considerable amount of infrared radiation. In this situation, a low α_s will not reduce micrometeoroid bumper temperatures and the accompanying heat gain to the pressure wall. It is likely that this heat gain could occur at a time when maximum heat rejection is needed. A double-louver system with the outer set of blades designed to close when the micrometeoroid bumper temperature exceeds that of the pressure wall will act to insulate the pressure wall and reduce heat gain rates. Reduction in the heat gain with the double louver system is illustrated for several pressure wall temperatures in Figures 3.6-7, 3.6-8, and 3.6-9.

By insulating the back surface of the louver blades, it may be possible to reduce the louver platform effective emissivity from 0.07 to 0.01 in the closed position. The effective emissivity for the open position would also be reduced somewhat from the uninsulated value of 0.8. Although the maximum heat rejection would be reduced slightly, the minimum heat rejection with single- and double-louver systems and the heat gain with the double-louver system would be significantly reduced (by approximately an order of magnitude).

Sidewall Heat Loss

Thermal protection will be required to limit OLS sidewall heat loss to preclude condensation on the pressure wall and minimize the external environment influence on the internal active thermal control loop. This has been historically accomplished by using high-performance insulations and limiting the thermal conductance of insulation penetrations. Assigning a portion of the total allowable heat transfer to insulation penetrations results in a net

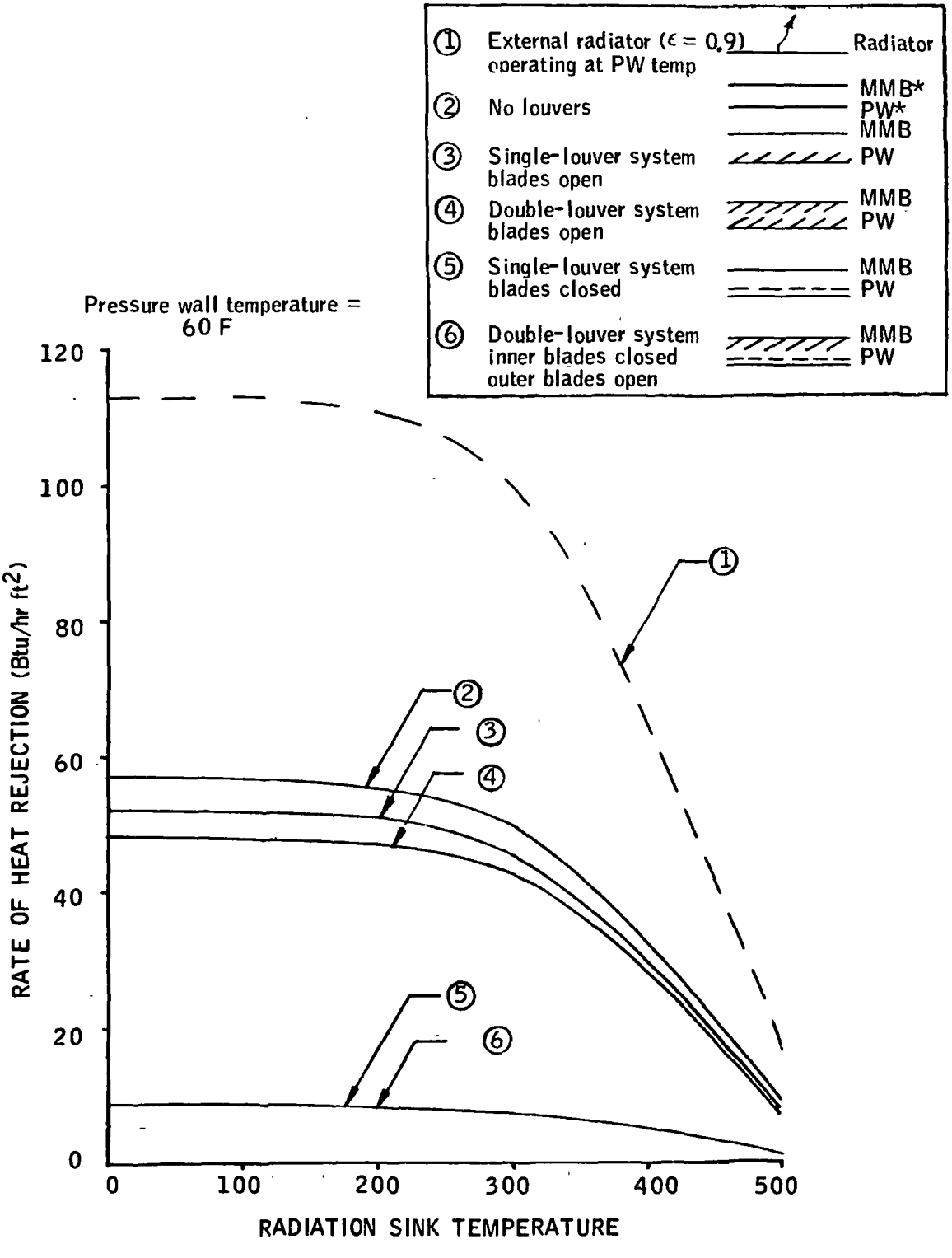


Figure 3.6-4. Effect of Louver System on Controlling Heat Rejection - Pressure Wall Temperature = 60 F

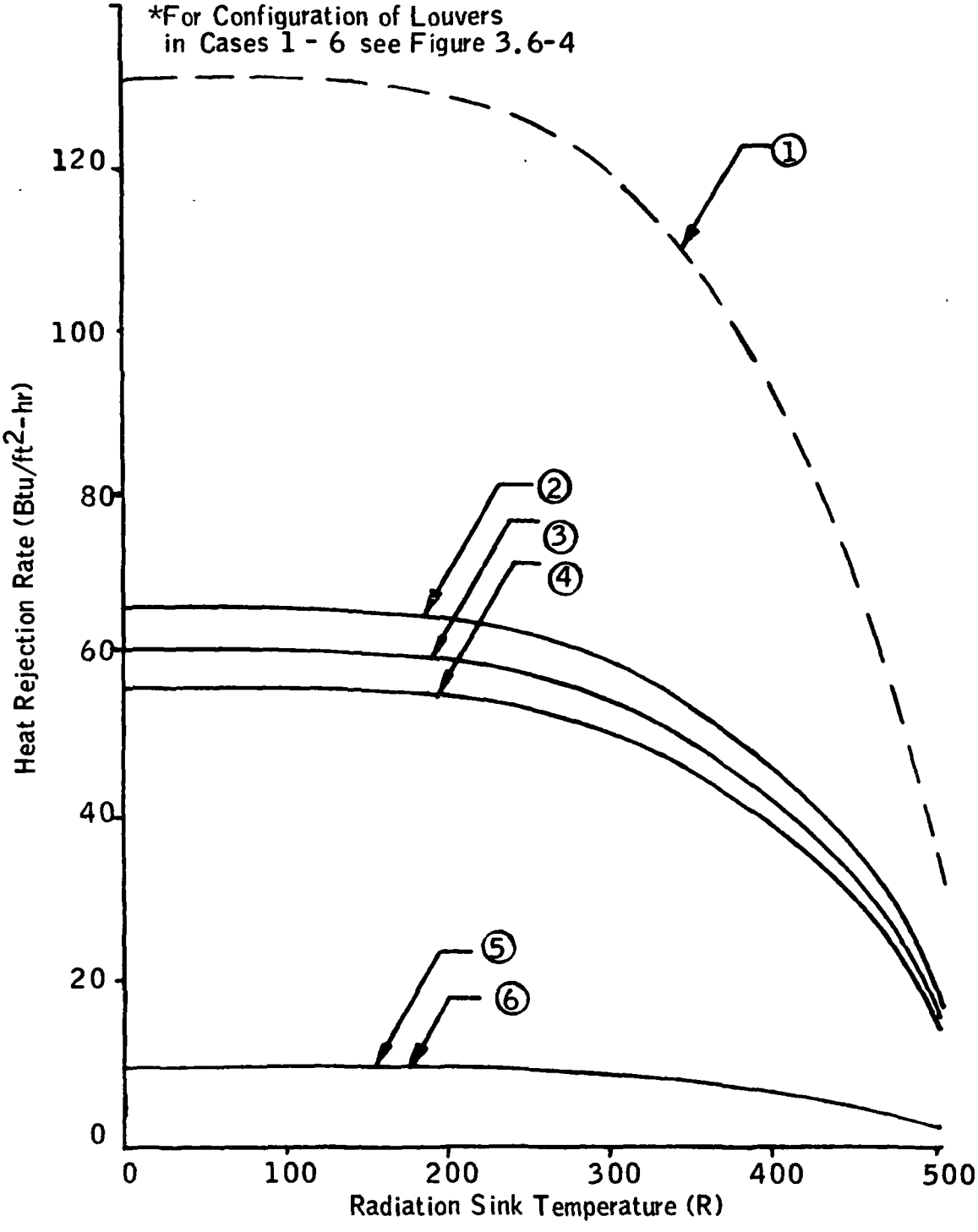


Figure 3.6-5. Effect of Louver Systems on Controlling Heat Rejection - Pressure Wall Temperature = 70 F

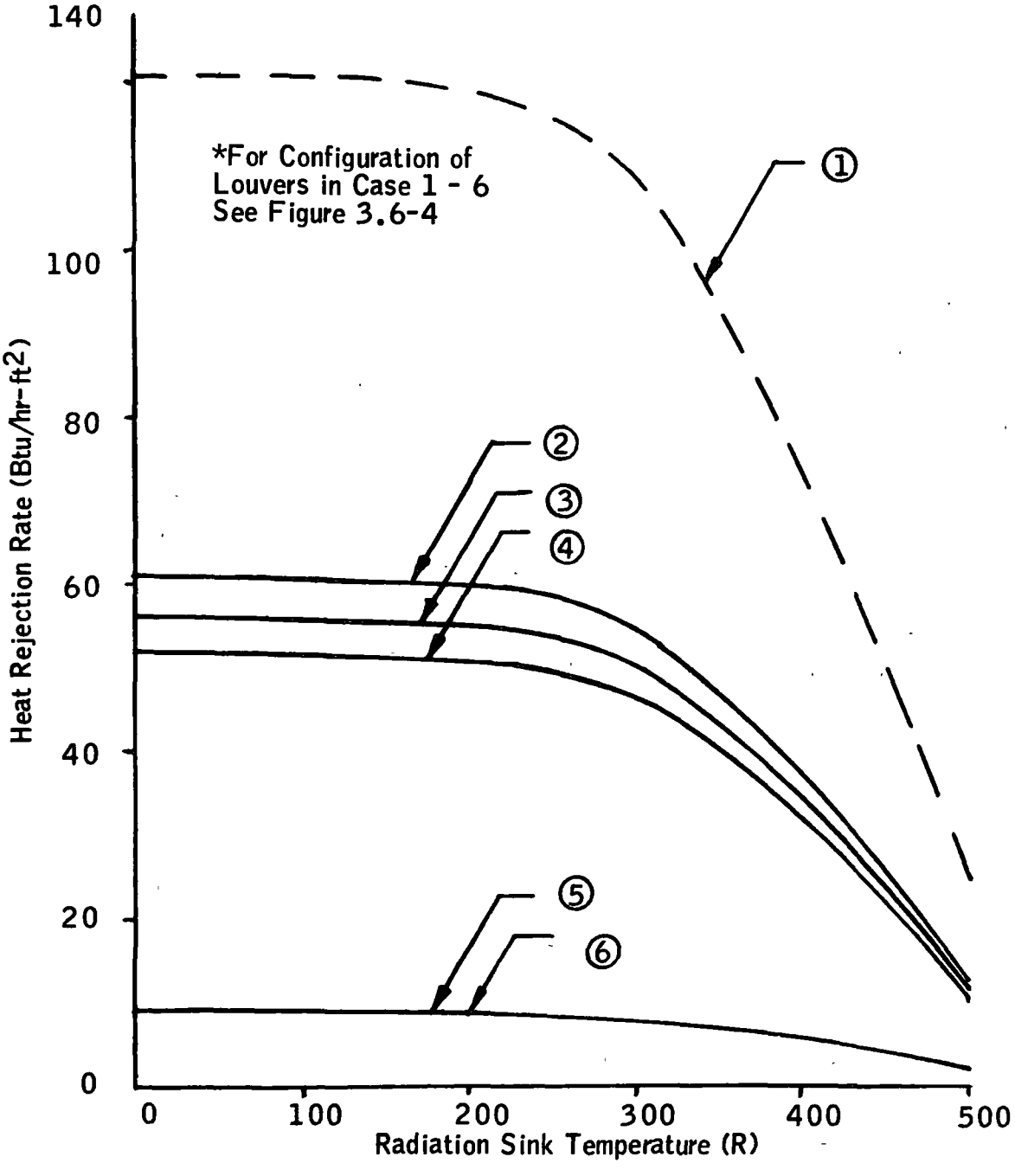


Figure 3.6-6. Effect of Louver Systems on Controlling Heat Rejection - Pressure Wall Temperature = 80 F

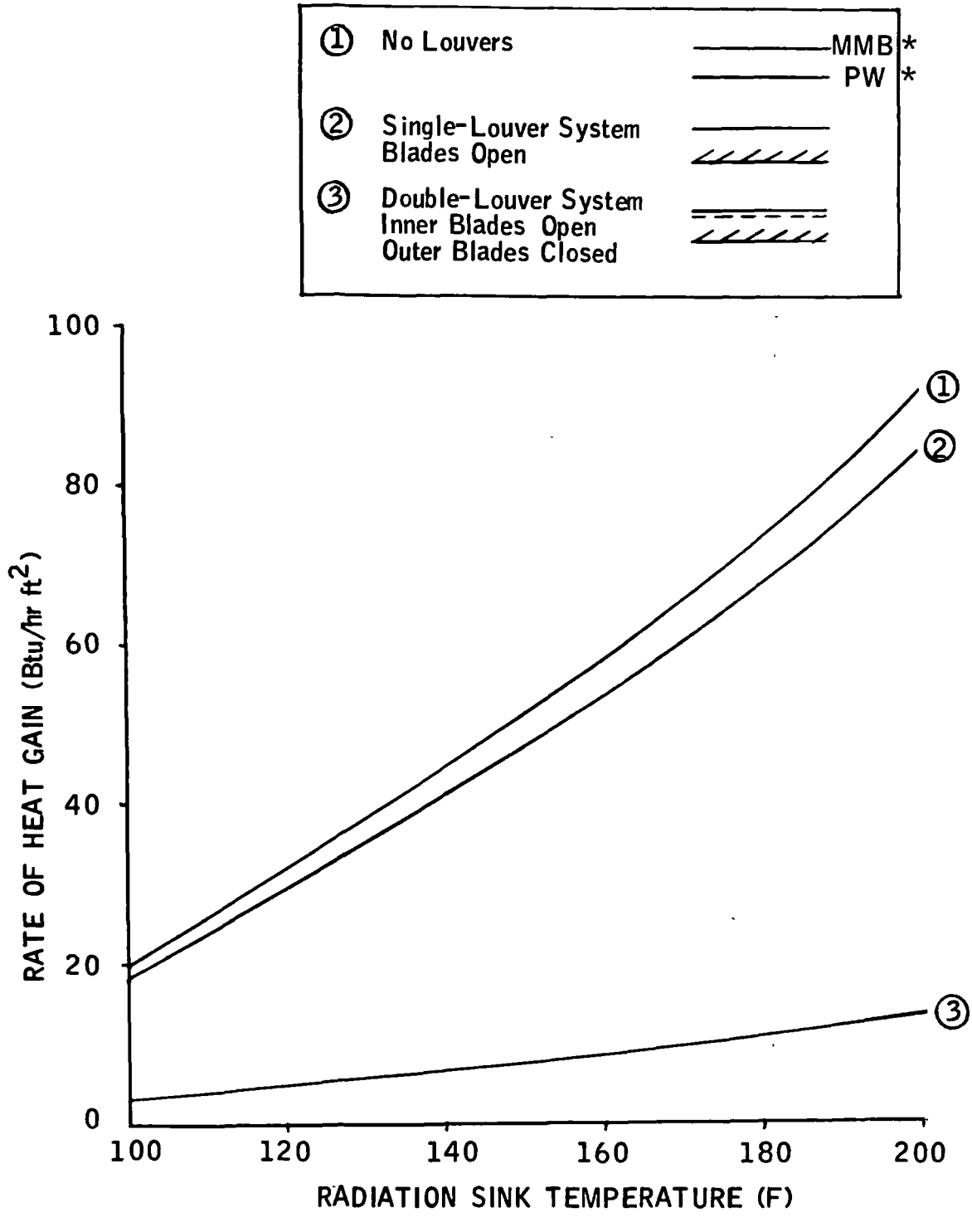


Figure 3.6-7. Effect of Louver Systems on Controlling Heat Gain - Pressure Wall Temperature = 60 F

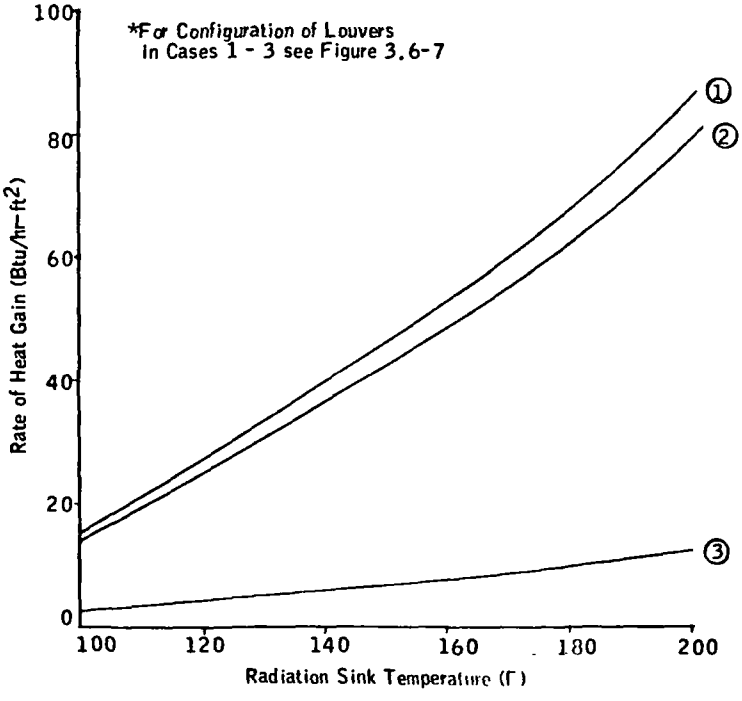


Figure 3.6-8. Effect of Louver Systems on Controlling Heat Gain - Pressure Wall Temperature = 70 F

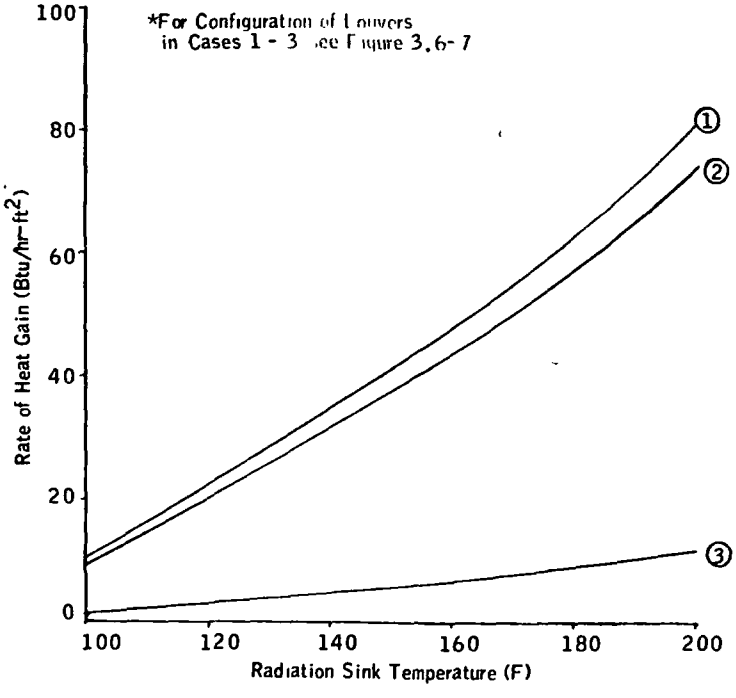


Figure 3.6-9. Effect of Louver Systems on Controlling Heat Gain - Pressure Wall Temperature = 80 F

allowable heat transfer through the insulation assembly. Figures 3.6-10 and 3.6-11 demonstrate the relationship of heat loss through insulation assemblies as a function of percent area covered by radiator with the ratio of allowable heat loss through insulation to the total surface area ($\dot{q}/A_t$). For the purpose of this analysis, pressure wall regions subtended by radiators are considered adiabatic. Horizontal lines shown on Figures 3.6-10 and 3.6-11 correspond to pressure wall temperatures resulting from the internal radiator boundary condition of 65 F (assumed as the minimum allowable air temperature and the minimum expected electronics temperature). Passive walls are assumed between the 65 F radiant heat source and the pressure wall. In Figure 3.6-10, one passive wall is assumed, whereas two passive walls were assumed in Figure 3.6-11.

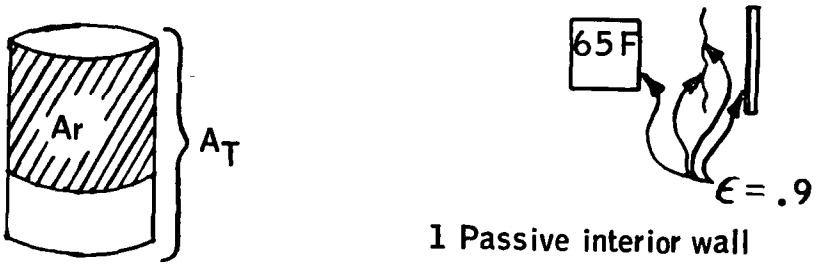
The following example illustrates the use of the figures to determine the allowable heat loss through the insulation assembly:

> Example - Given a total allowable heat loss of 5000 Btu/hr, total cylinder area of 5000 ft^2, and radiator area of 4000 ft^2, find the allowable heat loss through the insulation.
>
> Solution - Calculate the allowable heat loss per total area ($\dot{q}/A_t$= 1 Btu/hr ft^2). Locate the percent of total area covered by radiator (Ar/A_t = 80%) on the abscissae, and refer to the proper ($\dot{q}/A_t$) curve. The corresponding point on the ordinate, 5 Btu/hr-ft^2, is the allowable heat loss through the insulation. However, because of the cabin air dew point constraint, this allowable heat loss will be reduced. Assuming a dew point of 55 F, the maximum allowable heat loss through the insulation is 3.95 Btu/hr ft^2 with one passive interior wall and 2.63 Btu/hr ft^2 with two such walls. Hence, to relax the allowable heat loss requirement through the insulation, it is desirable to minimize the number of interior walls acting as radiator baffles between any spacecraft interior location and the pressure wall.

Solar Array Analysis

Solar array electrical power output decreases as its temperature rises. Figure 3.6-12 illustrates this effect. (The figure is not intended to imply performance characteristics of the cells being considered for the OLS). It is desirable, therefore, to reduce the operating temperature, thereby potentially reducing weight, size, dynamic characteristics, cost, and rate of degradation.

The solar array equivalent radiation sink temperature is plotted against position in orbit in Figure 3.6-13. Thus, the solar array is exposed to an environment that is likely to produce undesirably high temperatures for a significant portion of the sunlit (productive) phase of its orbit.



1 Passive interior wall

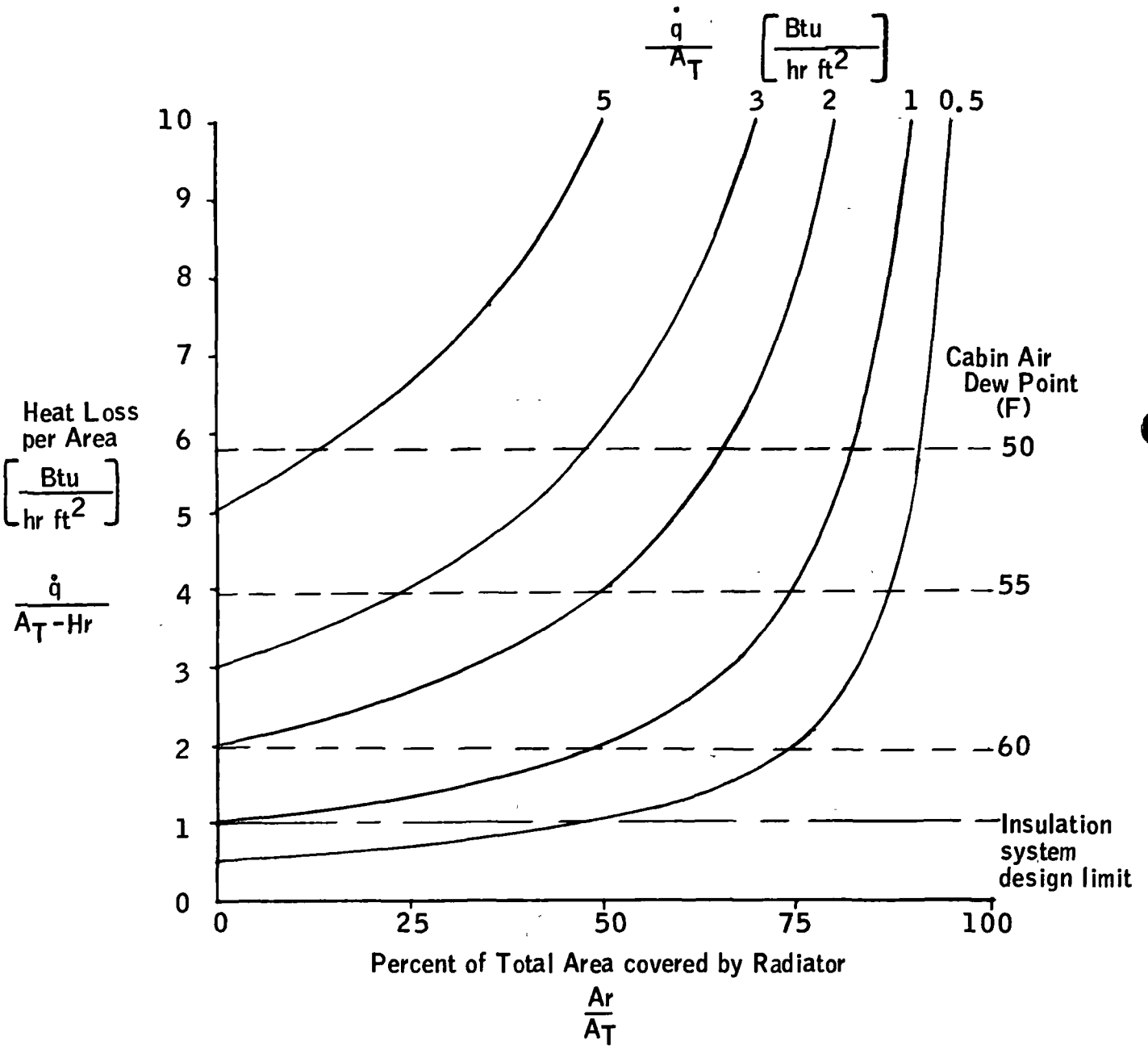


Figure 3.6-10. Allowable Heat Loss Through Insulation - One Passive Interior Wall

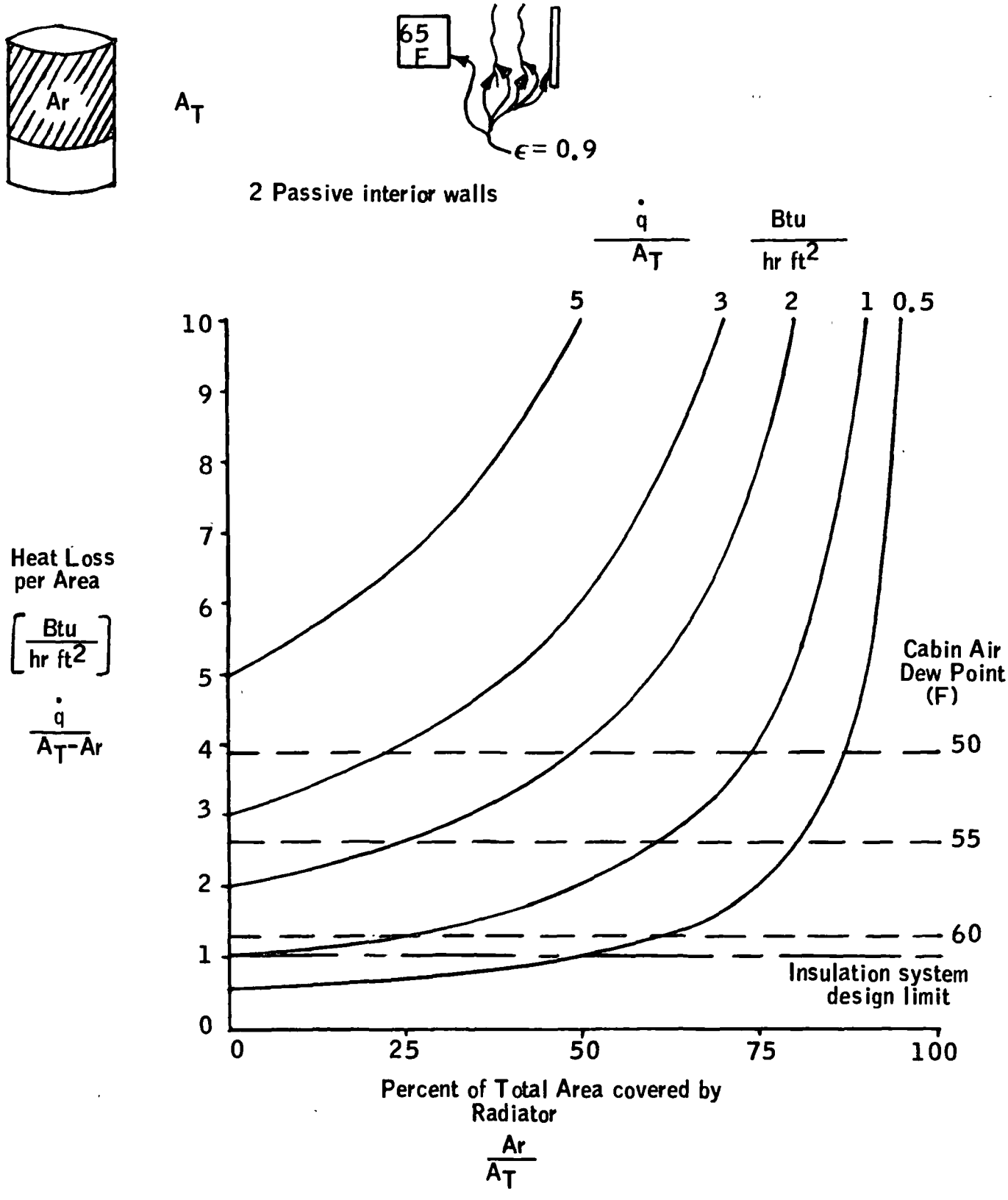


Figure 3.6-11. Allowable Heat Loss Through Insulation - Two Passive Interior Walls

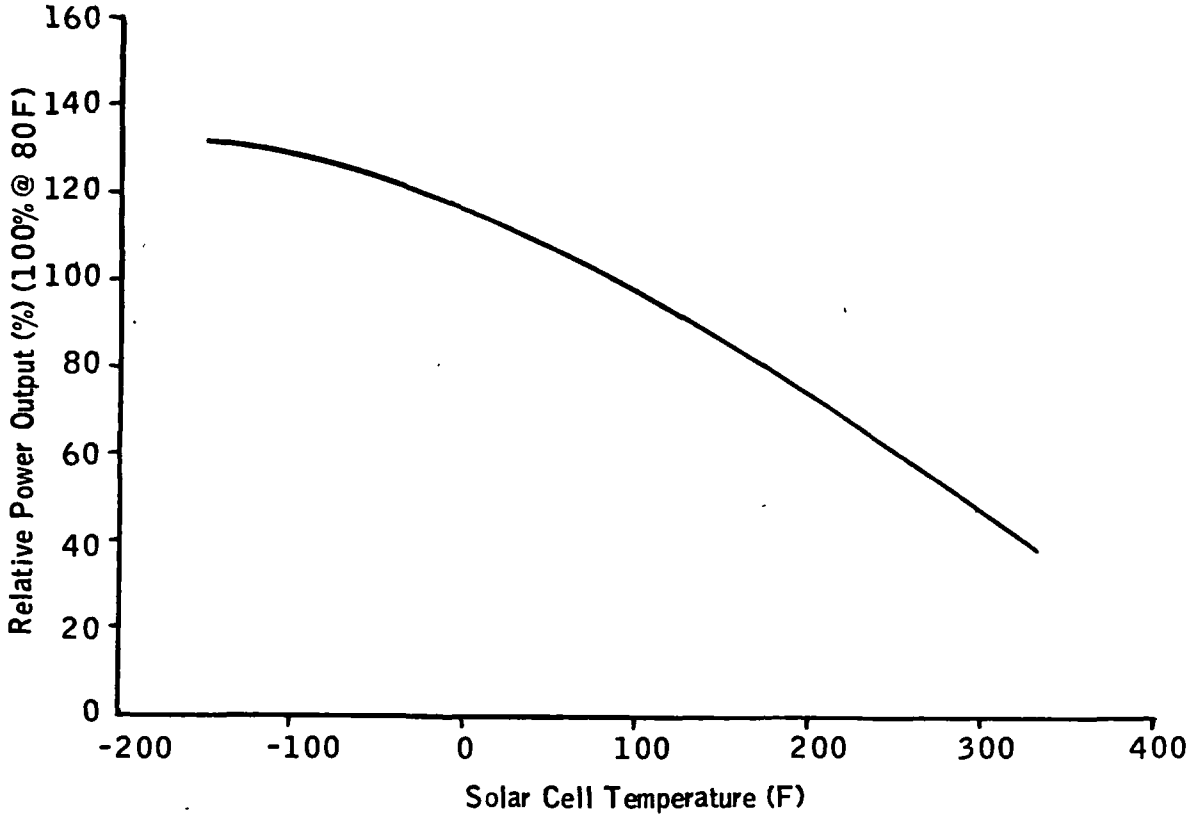


Figure 3.6-12. Relative Power Output for Typical Solar Cell

High solar array temperatures might be avoided in lunar orbit by utilizing phase change materials behind the array. Two categories of materials exist: that which undergoes a solid to liquid change in which the latent heat of fusion provides the required thermal capacitance and that which undergoes a solid to solid transition in which the required thermal capacitance is provided by the change in crystaline state. The latter type would be more readily applicable, and a material may exist that could be used as the solar cell substrate or be imbedded in existing substrate candidates. Upon heating in the transition temperature, heat is absorbed without a temperature rise until all the material has transformed to the new state, thus avoiding a deleterious peak temperature if properly sized. Upon subsequent exposure of the array to a radiant sink temperature lower than the transition temperature, heat is released without temperature change until all of the material has transformed back to the preheating state.

A transient analysis of a typical solar cell section of unit surface area has been performed to determine the effect of a phase change material on the solar cell assembly temperature. Representative material properties assumed were a latent heat of transformation of 100 Btu/lb, a specific heat of 0.5 Btu/lb F, and a transition temperature of 160 F. (The maximum desirable earth orbit operating temperature is approximately 160 F). Resulting lunar orbit solar array temperature histories are shown in Figure 3.6-14 for zero, 0.25, and 0.50 lbs/ft^2 of phase-change material. The phase-change

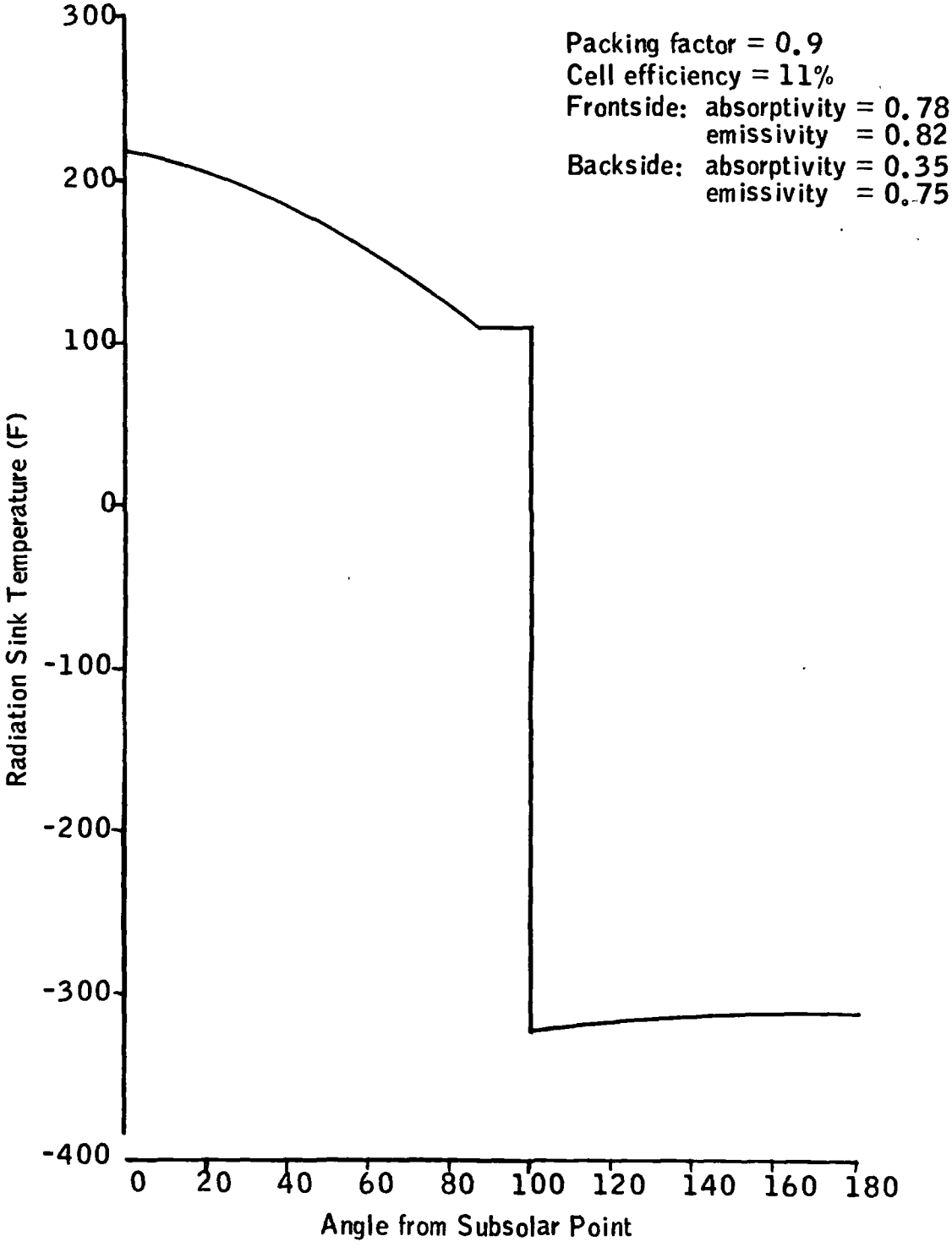


Figure 3.6-13. Solar Array Equivalent Radiation Sink Temperature Versus Angle From Subsolar Point

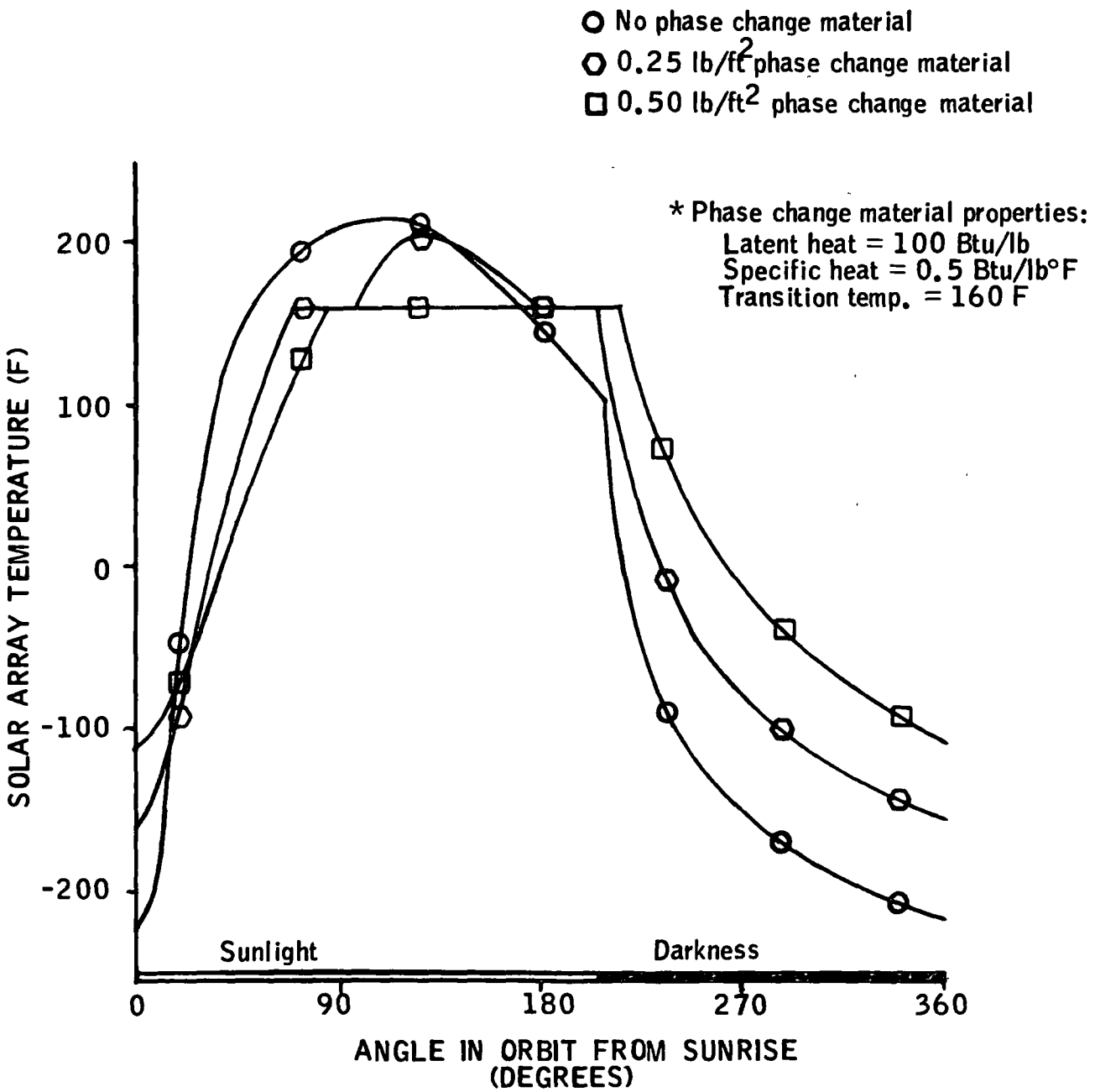


Figure 3.6-14. Effect of Phase Change on Solar Array Temperatures

material is effective in shortening the time the solar array experiences elevated temperatures. However, the added weight may be prohibitive, unless a material with a high latent heat of transformation at a useful transformation temperature can be found.

The solar array temperature also will be influenced by its positioning with respect to the other surfaces. The extent of this influence will depend upon the view factor from the solar array element to the other surface, the temperature of the surface, the thermal environment, and whether one or both sides of the array see the surface. The effect of surfaces at 100 F and 150 F to local solar array temperatures is shown in Figure 3.6-15. Solar array element to surface view factors are shown for four possible solar array element orientations in Figure 3.6-16. The solar array must be located so that this effect will not result in temperatures so high that solar-cell operating efficiency is excessively reduced.

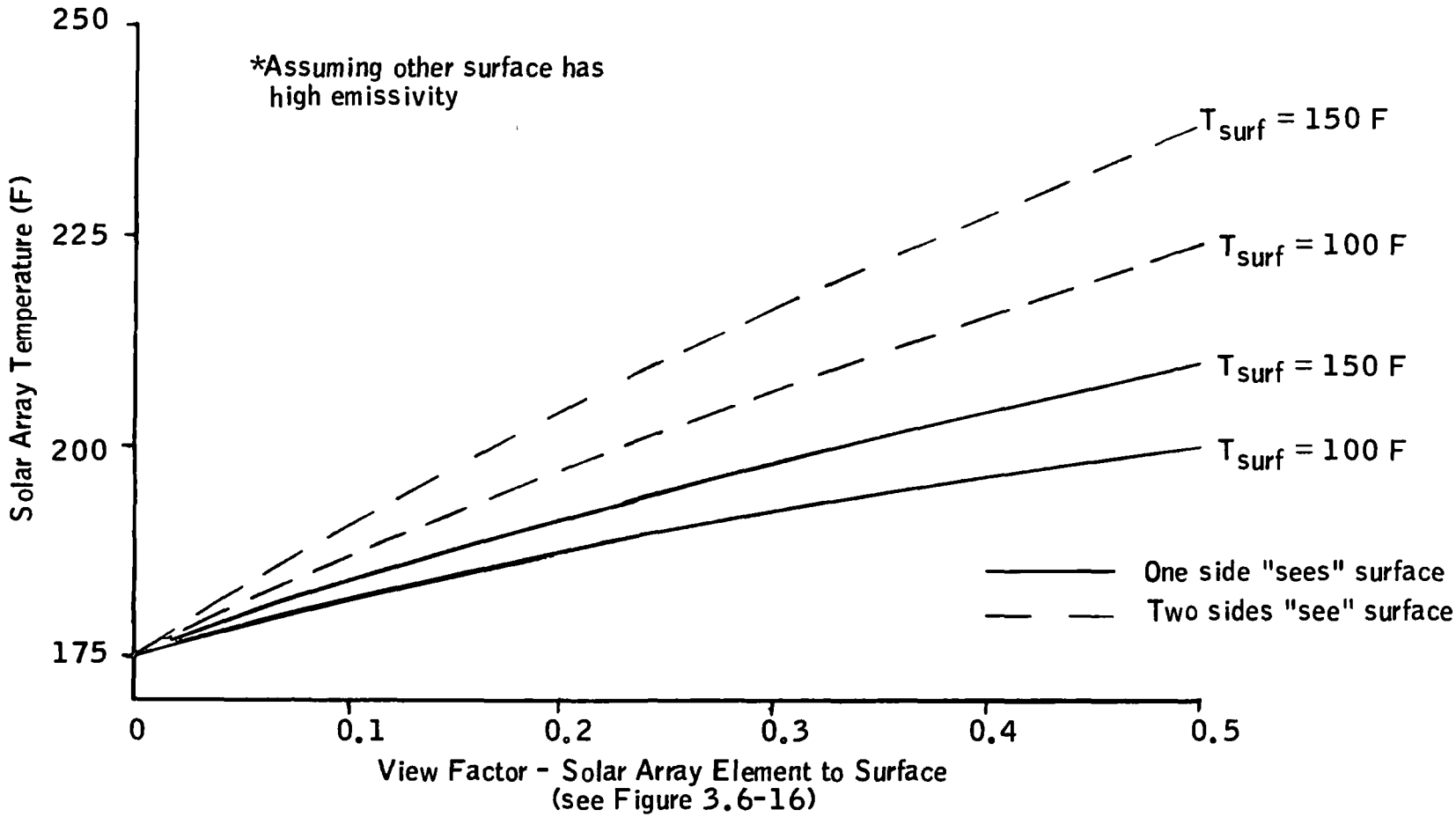


Figure 3.6-15. Influence of Solar Array View Factor to Other Surfaces on Solar Array Temperature

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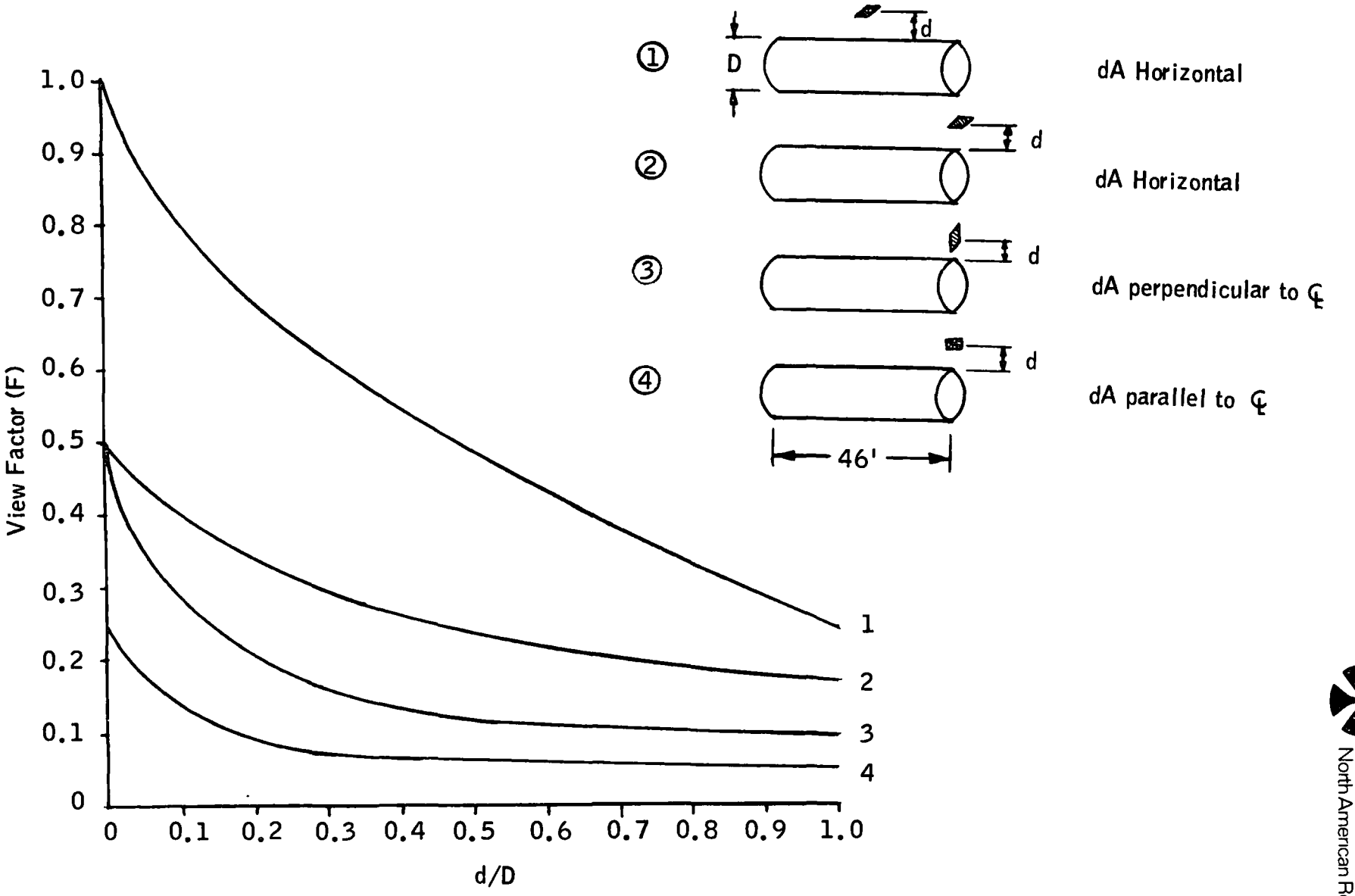


Figure 3.6-16. View From Solar Array Element to Surface

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3.6.2 METEOROID ENVIRONMENT

Meteoroid Strike Probability Analysis

Subsection 2.3 of Volume III, defines as a credible accident penetration of the primary OLS structure by a meteoroid of 1.0 gram mass, 0.615 inch in diameter. Penetration by this 1-gram particle would produce sufficient kinetic energy to explosively release 50 Btu of energy equivalent to 0.025 pound of TNT. An explosion of this magnitude would result in catastrophic failure of the OLS and possible loss of crew in the affected area. Subsection 7.7.2 of Volume III calls for protection of the primary OLS structure from penetration to a probability of 0.9 for 10 years.

To determine the probability of the OLS being struck by a meteoroid of 1.0 gram mass or larger and the amount of meteoroid shielding of crew and critical systems compartments required to prevent meteoroid penetration with the required degree of reliability, the following studies have been conducted.

The average annual cumulative meteoroid flux-mass model on the lunar surface is described mathematically in NASA Report, TMX-53865 by:

for $10^{-6} \leq m \leq 10^{0}$,

$$\log_{10} N_{t_m} = -14.597 - 1.213 \log_{10} m \qquad (1)$$

for $10^{-12} \leq m \leq 10^{-6}$,

$$\log N_{t_m} = -14.566 - 1.584 \log_{10} m - 0.063 (\log_{10} m)^2 \qquad (2)$$

where

N_{t_m} = numbers of particles/m^2/sec of mass m or greater

m = mass in grams.

The gravitationally focused, unshielded flux N_{t_m} is corrected for lunar defocusing and shielding effects. The results for three lunar altitudes are as follows:

$$G_m = 0.966 + \frac{0.034}{r} \qquad (3)$$

where

G_m = lunar defocusing factor

h = lunar altitude

R_m = lunar radius, 1738 km

$$r = \frac{h + R_m}{R_m}$$

$$\xi = \frac{1 + \cos \theta}{2} \qquad (4)$$

where

ξ = body shielding factor

$$\sin \theta = \frac{1}{r} = \frac{R_m}{h + R_m}$$

h (n mi)	r	G_m	sin θ	cos θ	ξ	$G_m \xi$
20	1.022	0.9992	0.979	0.204	0.6022	0.602
60	1.064	0.9979	0.940	0.341	0.6705	0.670
100	1.108	0.9967	0.903	0.430	0.715	0.7125

The density of all meteoroids is given in TMX-53865 as 0.5 gm/cc. Assuming spherical particles, the range of size for $10^{-12} \leq m \leq 10^{6}$ is $.615 \times 10^{-4} \leq d \leq 0.615$, for diameters in inches.

The average annual cumulative number of meteoroid strikes on the OLS can be found for various sizes of meteoroids within the range $10^{-12} \leq m \leq 10^{0}$. The number of strikes for any one mass at orbital altitude is given by

$$N_m = G_m \xi N_{t_m} A T \qquad (5)$$

where

$G_m \xi$ = f (orbital altitude), defined by equations (3) and (4)

N_{t_m} = meteoroid flux of mass m or greater, defined by equations (1) or (2)

A = surface area in m^2 assumed to be 1835 m^2, which is believed to be a conservative estimate (i.e., higher than required); thus increasing the number of strikes

T = time period of exposure, 10 years = 3.155×10^{8} sec

The total cumulative number of strikes, N_T, in the 10-year lifetime of the OLS at orbital altitude is found by solving equation (5) with N_{t_m} determined from equation (2) with $\log_m$ equal to -12.

Solving equation (2) with log m = -12:

$$N_{t_{m \geq 10^{-12}}} = 2.333 \times 10^{-5} \frac{\text{particles of mass} \geq 10^{-12} \text{ grams}}{m^2 - \text{sec}}$$

$$\therefore N_T = 2.333 \times 10^{-5} G_m \xi AT \qquad (6)$$

At h = 20 n mi, $N_T = 8.11 \times 10^6$ strikes in 10 years. At h = 60 n mi, $N_T = 9.05 \times 10^6$; and at h = 100 n mi, $N_T = 9.61 \times 10^6$.

The NASA meteoroid flux-mass model indicates that the number of strikes by meteoroids is a function of meteoroid mass, decreasing with increasing mass. The probability or frequency of occurrence of a strike by a meteoroid of mass m or larger is the ratio of the cumulative number of meteoroid strikes of mass equal to or greater than 10^{-12} grams in a given time period (N_T).

$$f_m = \frac{N_m}{N_T} \qquad (7)$$

The cumulative total number of strikes in a 10-year period of the OLS is plotted as a function of mass in Figure 3.6-17. The cumulative strike frequency is plotted as a function of mass in Figure 3.6-18. Nearly 99 percent of the total cumulative meteoroid strikes upon the OLS in a 10-year period are caused by particles of mass 10^{-7} grams or less. One strike out of every 10 billion is by a meteoroid larger than 0.615 inch in diameter.

The probability of zero impacts by meteoroids of a certain mass or greater is given approximately by Poisson's equation in NASA Report SP-80XX:

$$p_o = e^{-N_m} \qquad (8)$$

where

p_o = probability of zero impacts

N_m = cumulative number of strikes by mass $\geq$ m

This equation is valid as long as $f_m \leq 0.1$ and $N_T \geq 100$. Equation (8) is Poisson's approximation of the exact binomial probability distribution which, for zero impact probability, becomes:

$$p_o = (1 - f_m)^{N_T} \qquad (9)$$

where f_m and N_T have been defined previously by equations (7) and (6), respectively. Equation (8) used to calculate the probabilities quoted in this study, is valid for $p_o \geq 0.9$, which is an OLS design requirement. The results are plotted in Figure 3.6-19 as a function of mass. The inaccuracy

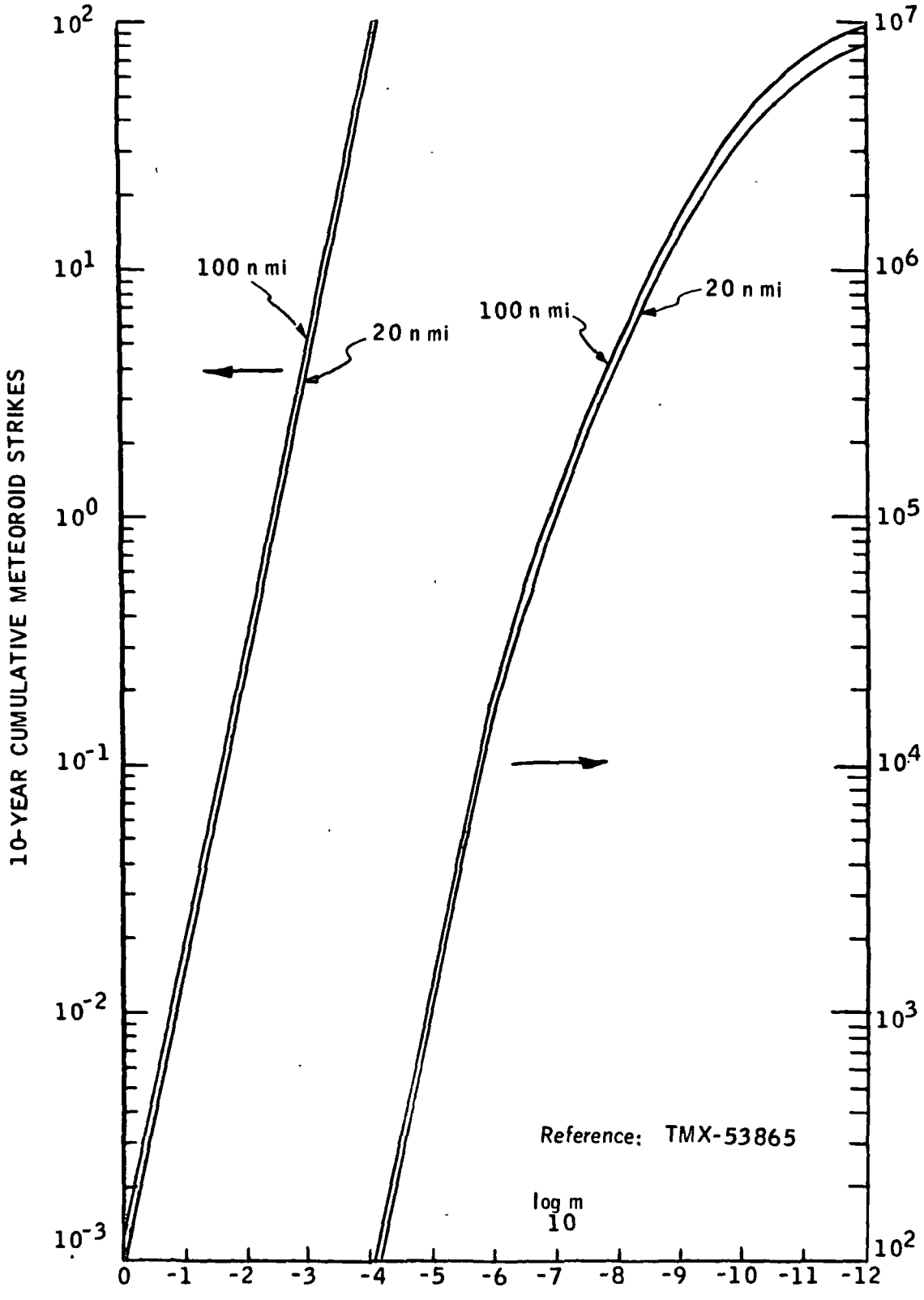


Figure 3.6-17. Average 10-Year Cumulative Meteoroid Strikes on OLS (Stream and Sporadic)

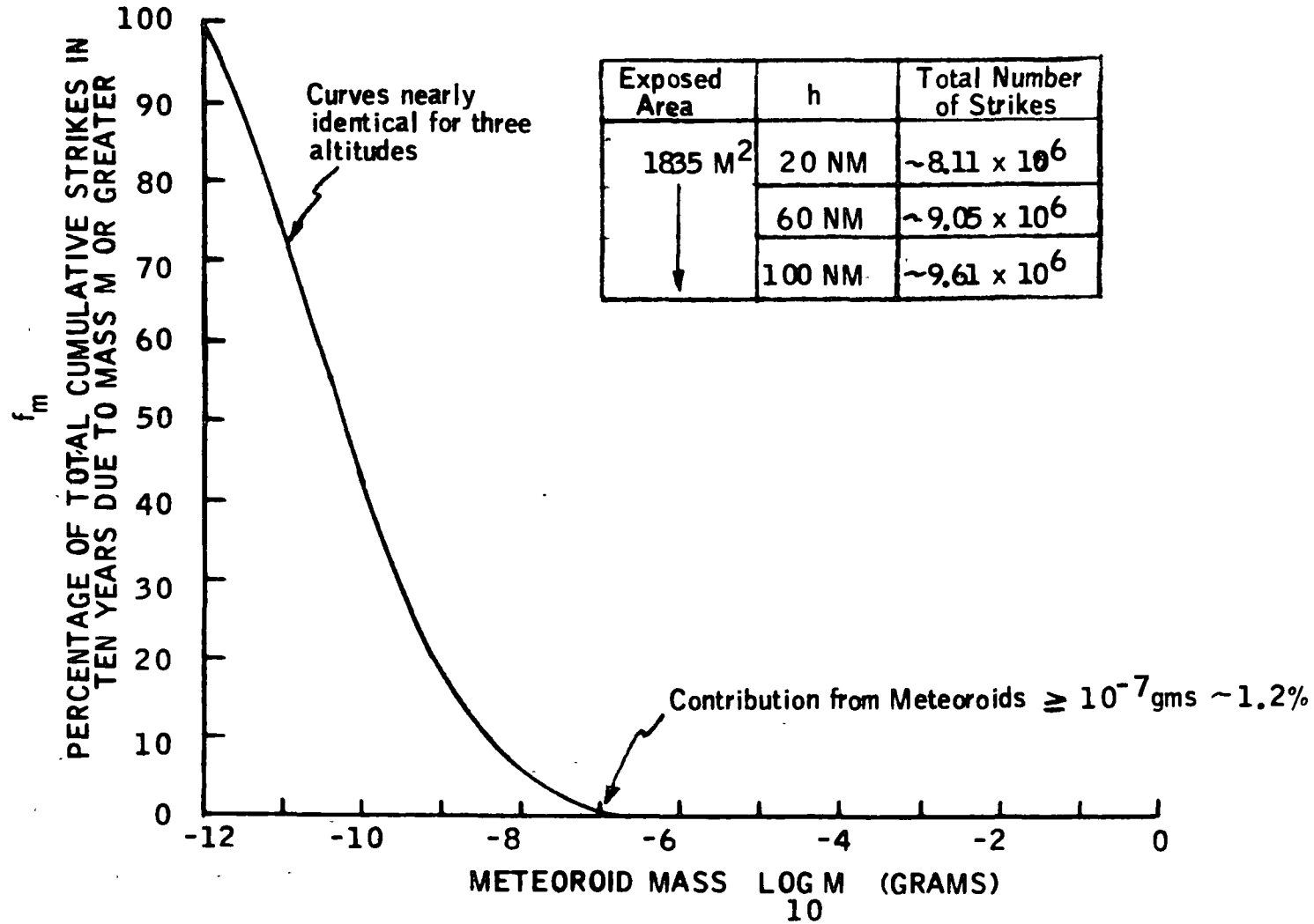


Figure 3.6-18. Contribution to Total Cumulative Meteoroid Strikes on OLS in Lunar Orbit as Function of Mass

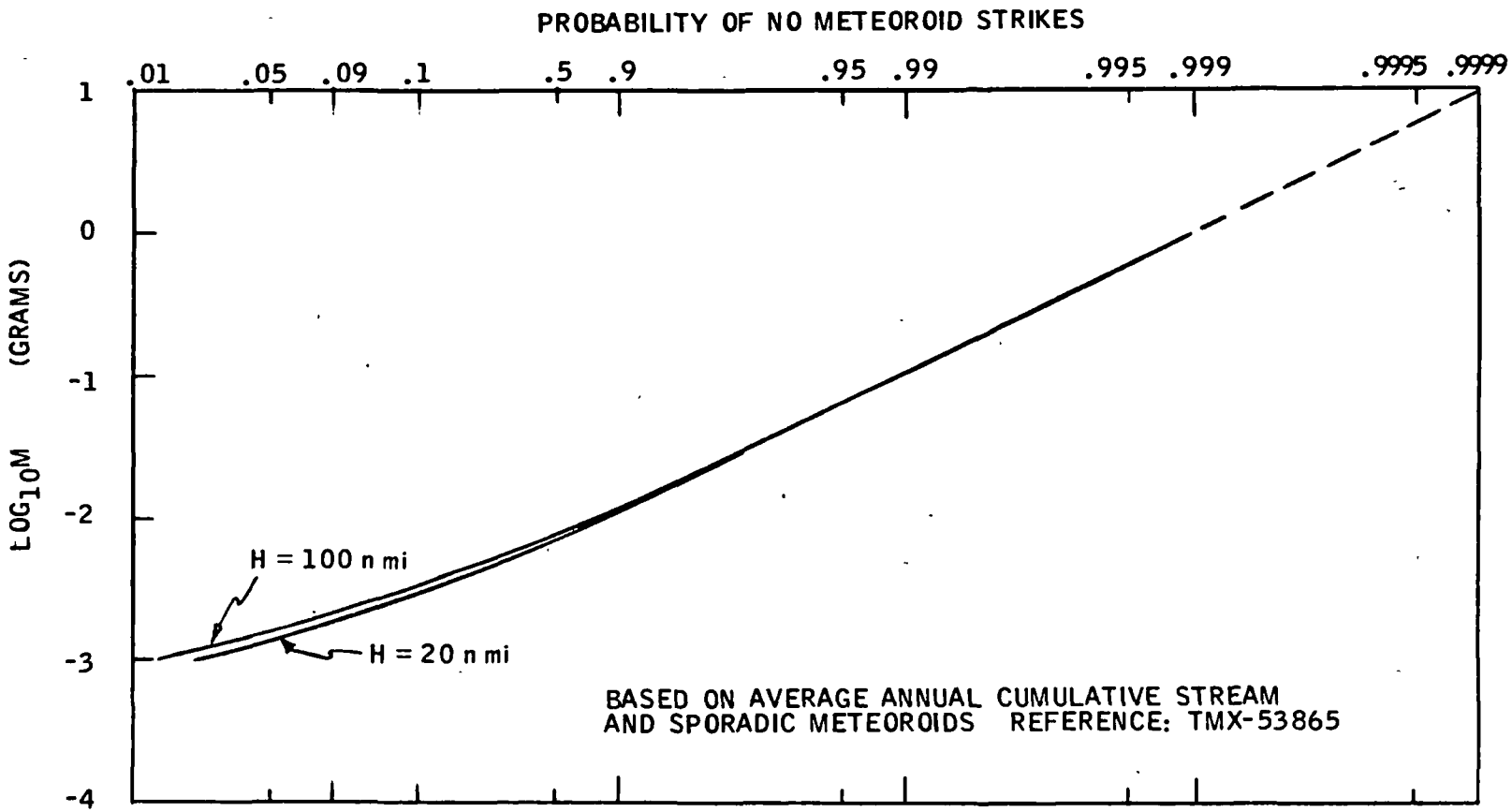


Figure 3.6-19. Meteoroid Strike Probability for OLS in Lunar Orbit 10 Years

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of the curve for $p_o < 0.9$, resulting from the use of equation (8) rather than from the more exact form of equation (9), is of no concern because this is below the OLS design requirement range. From Figure 3.6-19, determination may be made of the particle mass that a meteoroid bumper must be designed for to achieve a specified reliability of zero penetration over the OLS 10-year life span. The results are summarized as follows:

Meteoroid Protection Reliability	Particle Mass to Design For
0.9999	~10 grams
0.999	~ 1.0 gram
0.995	~ 0.6 gram
0.99	~ 0.10 gram
0.90	~0.01 gram

It should be noted that, as for the OLS, the EOSS meteoroid bumper also is to be designed to prevent meteoroid penetration of crew or systems compartments for 10 years with a reliability of 0.9. The meteoroid flux environment N_{t_m} in the vicinity of the earth is only slightly different from that at the moon according to NASA TMX-53865. The product $G_e \xi$ for the altitudes of interest to the EOSS range from 0.647 at 200 n mi to 0.672 at 300 n mi, while the OLS operating in lunar orbits of from 20- to 100-n mi altitude, the product is $0.602 \leq G_m \xi \leq 0.7125$.

Solving equations (6), (7), and (8) indicates the EOSS must incorporate a meteoroid bumper to protect against a 0.02 gram (0.167-inch diameter) meteoroid for a probability of 0.9 of no penetration in 10 years. Similarly, the OLS must incorporate a meteoroid bumper that will preclude penetration of 0.011 gram (.137-inch diameter) for a comparable reliability.

OLS Decompression Due to Meteoroid Punctures

To determine the time the OLS crew would have to take corrective action in the event of a meteoroid penetrating the pressure wall, the time to decompress to the minimum allowable atmospheric pressure must be calculated. The minimum allowable initial total atmospheric pressure in a two-gas N_2/O_2 atmosphere is specified in Section 7.2 of Volume III, as 10 psia. The O_2 partial pressure in this cabin atmosphere is to be between 3.1 and 3.5 psia. The minimum partial pressure of O_2, adequate to support body physiological functions (alveolar limit), is 1.9 psia in a dual-gas N_2/O_2 atmosphere initially at 14.7 psia and 70 F. The alveolar limit is reached when the total cabin pressure falls to 9.2 psia.

To arrive at the expression for decompression time, the following assumptions are made:

1. There is critical flow at the puncture.

2. The hole is idealized as a sharp-edged orifice, justifying the use of a discharge coefficient equal to 0.6.

3. The relative composition of the cabin atmosphere remains constant and thus so do the gas properties (i.e., molecular weight, isentropic exponent, gas constant).

4. For isothermal decompression, the initial temperature remains constant. A nominal value of 70 F is used.

Analysis

For critical flow through a sharp-edged orifice, the mass flowrate is given by

$$\dot{m} = pA_H C_d \sqrt{\frac{\gamma}{RT_o} \left(\frac{2}{\gamma + 1}\right)^{\gamma + 1/\gamma - 1}} \qquad (10)$$

where

$\dot{m}$ = mass flowrate, slugs/sec

p = cabin pressure, lb_f/ft^2

A_H = hole area, meteoroid cross-sectional area, ft^2

C_d = discharge coefficient, 0.6

γ = isentropic exponent, 1.4

R = gas constant, 1715 ft-lb_f/slug^oR for air, 1555 ft-lb_f/slug^oR for O_2

T_o = initial temperature, 70 F = 530 R

The number of gas particles escaping through the orifice per second is

$$\dot{N} = \frac{\dot{m} N_o}{M} \qquad (11)$$

where

N_o = Avogadro's number, 6.025×10^{23} particles/gm-mole
There are $(454\ gm/lb_m)(32.2\ lb_m/slug)$

M = Molecular weight of the gas, 29 gm/gm-mole for air, 32 gm/gm-mole for O_2

The number of gas particles leaving the cabin per unit volume per second is just

$$\dot{n} = \frac{\dot{N}}{V} \qquad (12)$$

where V = cabin volume, ft^3

And finally, the rate of change of pressure with time ($\dot{p}$) becomes

$$\dot{p} = -\dot{n}KT_o = \frac{-\dot{N}KT_o}{V} = -\frac{\dot{m}\,N_oKT_o}{MV} \qquad (13)$$

$$= -\frac{N_oKpA_HC_d}{V}\sqrt{\frac{T_o}{M\bar{R}}\,\gamma\left(\frac{2}{\gamma+1}\right)^{\gamma+1/\gamma-1}}$$

$$= -p\alpha$$

where

$\dot{p}$ = pressure loss rate, psf/sec

K = Boltzmann's constant, 1.38×10^{-23} joule/°K
There are 0.7376 ft-lb_f/joule and $T_o = 294°K$

$\bar{R}$ = Universal gas constant, 1545 ft-lb_f/lb_m-mole°R

$$\alpha = \sqrt{\frac{N_oK}{\bar{R}}}\;\frac{A_HC_d}{V}\sqrt{\frac{T_o}{M}}\;f(\gamma) \qquad (14)$$

$$f(\gamma) = \sqrt{\gamma\left(\frac{2}{\gamma+1}\right)^{\gamma+1/\gamma-1}} = 0.685 \text{ for } \gamma = 1.4 \qquad (15)$$

By using elementary calculus, the isothermal decompression time may be found. From equation (13),

$$\dot{p}/p = d/dt\ (\ell np) = -\alpha \qquad (16)$$

Integrating equation (15) gives

$$\ell np = -\alpha t + C \qquad (17)$$

$$\text{at } t = 0,\ p = p_o \implies C = \ell n\ p_o$$

$$\therefore \ln(p/p_o) = -\alpha t$$

$$t_{ISO} = \frac{\ell n(p_o/p)}{\alpha} \qquad (18)$$

Substituting the appropriate numbers for the constants in equation (14) gives

$$\alpha = 223\ A_H C_d \sqrt{\frac{T_o}{M}}\ f(\gamma) = 151.5\ \frac{A_H C_d}{V}\sqrt{\frac{T_o}{M}} \qquad (19)$$

for the case of air (i.e., the OLS dual-gas atmosphere of O_2/N_2).

The units for equations (18) and (19) are:

t = sec

α = sec^{-1}

A_H = ft^2

V = ft^3

T_o = ^oR

M = lb_m

p = any units of pressure, because it appears in a ratio

The worst possible case (i.e., fastest decompression rate) would be that of an adiabatic decompression; no heat transfer to the environment. Using the isentropic relation between pressure and temperature, it can be shown that the adiabatic decompression time is given by the following expression,

$$t_{AD} = \frac{2\left[(p_o/p)^{\gamma - 1/2\gamma} - 1\right]}{\propto\ (\gamma - 1)} \qquad (20)$$

$$= \frac{5\left[(p_o/p)^{0.143} - 1\right]}{\propto}, \text{ when } \quad \gamma = 1.4$$

It should be noted that both equations (18) and (20) give the decompression time when there is no gas makeup in the evacuating pressure volume. Obviously, if there is gas makeup, the decompression time is lengthened. This is accounted for by introducing the factor $(1 - \phi)$ in the denominators of equations (18) and (20); ϕ being a makeup function such that $0 \leq \phi \leq 1$ with $\phi = 0$ for no gas makeup and $\phi = 1.0$ for complete gas makeup. With $\phi = 1.0$, t_{ISO} and t_{AD} approach infinity, which just means that there is no decompression.

For the OLS study, parametric pressure-time data was generated for cabin volumes between 2500 ft^3 and 30,000 ft^3 filled with a two-gas mixture of N_2/O_2 initially at 14.7 psia and 70 F (P_i and T_o, respectively). Curves illustrating these data for cabin volumes of 24,000 ft^3 and 18,000 ft^3 are shown in Figures 3.6-20 and 3.6-21, respectively. In addition, parametric data was generated to show decompression time to the alveolar limit (9.2 psia) in an N_2/O_2 dual-gas atmosphere for cabin volumes between 2500 ft^3 and 30,000 ft^3. These data are shown in Figure 3.6-22. The results are summarized in Table 3.6-1.

Table 3.6-1. OLS Cabin Decompression caused by Meteoroid Puncture

Cabin Volume (ft^3)	2,500	10,000	15,000	18,000	24,000
Hole Size (in)	t_{AD} to 9.2 psia	t_{AD} to 9.2 psia	t_{AD} to 9.2 psia	t_{AD} to 9.2 psia	t_{AD} to 9.2 psia
2.25	1.4 min	5.6 min	8.4 min	10.1 min	13.45 min
1.0	6.8	27.2	40.8	49.0	65.2
0.5	27.2	108.0	162.5	194.5	259.0
0.25	108.0	432.0	650.0	777.0	1034.0

The table indicates that any puncture by a meteoroid large enough to create a 2-1/4-inch diameter puncture hole would pose a time-critical problem for the OLS crew in cabin pressure volumes smaller than 10,000 ft^3. It was pointed out in subsection 2.3 of Volume III that holes larger than 2-1/4 inches would begin to cause enough damage to endanger crew from factors other than pressure loss (for example, explosion and fire). It is not possible to predict with any confidence the puncture hole size resulting from an impact by a meteoroid of given size because of the number of factors involved: meteoroid size and density, impact angle and velocity, meteoroid shield density and thickness, and bumper spacing.

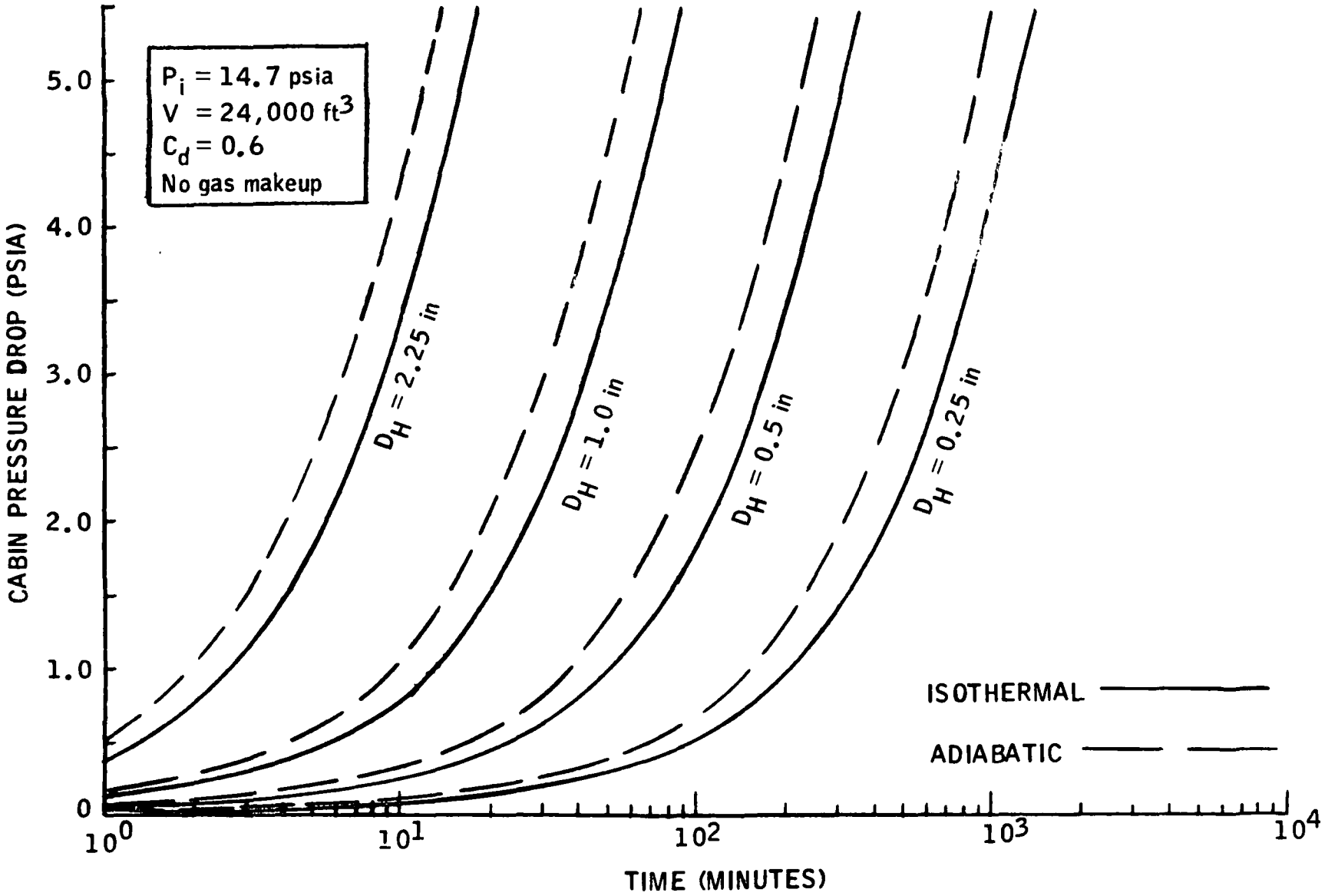


Figure 3.6-20. Decompression Time Due to Meteoroid Puncture of OLS Pressure Volume

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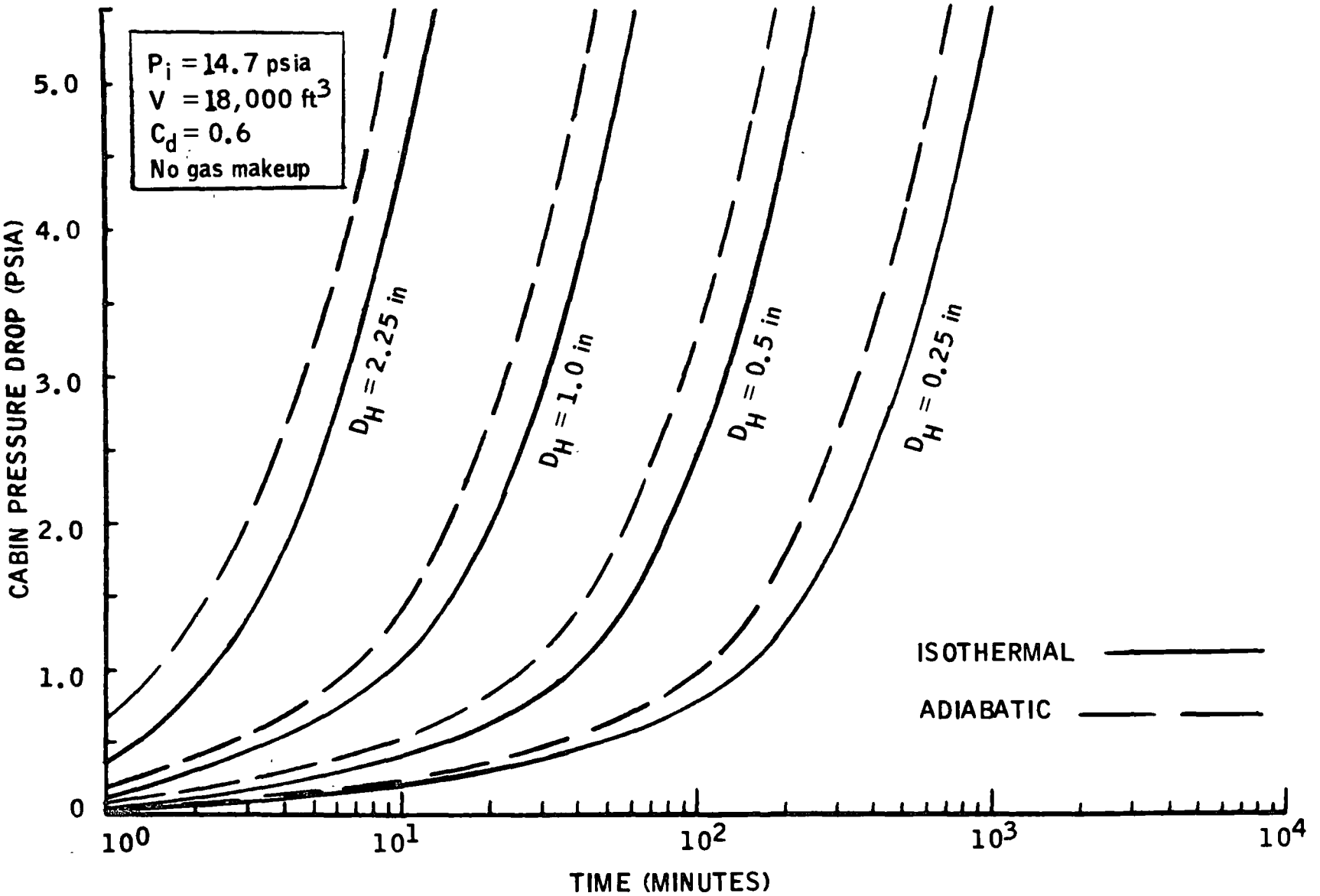


Figure 3.6-21. Decompression Time Due to Meteoroid Puncture of OLS Pressure Volume

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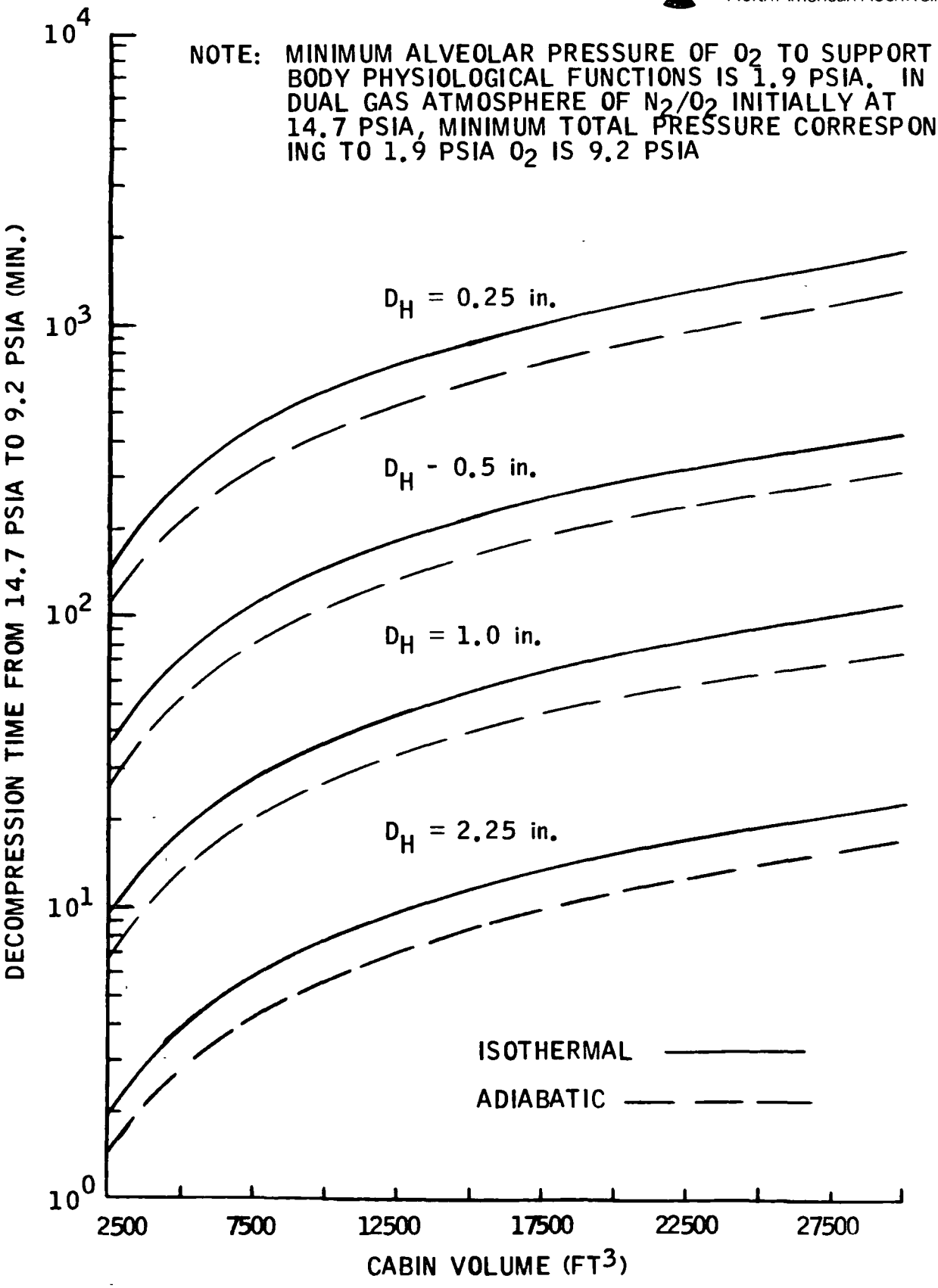


Figure 3.6-22. Decompression Time as Function of OLS Cabin Volume for Meteoroid Punctures

However, for meteoroids of low enough energy, the puncture hole size is not greater than the bumper spacing. Subsection 2.3 of Volume III states that a 1-gram meteoroid can be expected to cause about a $2\frac{1}{4}$-inch-diameter hole. The rationale for this statement is that an impact by a 1-gram meteoroid has a large probability of nonoccurrence ($p_o = 0.999$) and is small enough (0.615 inch in diameter) to expect that the hole size would be about equal to the bumper spacing, which is about $2\frac{1}{4}$ inches. Impacts by meteoroids smaller than 1 gram could result in punctures smaller than $2\frac{1}{4}$ inches, depending upon what size particle the meteoroid bumper is designed to stop. The meteoroid strike probability analysis of Subsection 3.6.2 showed that a meteoroid shield designed to give a 0.9 probability of no failure could stop penetration by meteoroids $\lesssim$ 0.011 grams (diameters $\leq$ 0.137 inch). Obviously then, impacts by meteoroids $\lesssim$ 0.011 grams would result in zero puncture hole size.

It is of interest to know what effect a switch from the N_2/O_2 cabin atmosphere to a pure O_2 atmosphere would have upon decompression time in the OLS. The nominal operating pressure would be 5.0 psia in this case. Figure 3.6-23 shows parametric decompression curves for an 18,000 ft^3 cabin volume. The minimum allowable pressure level is 1.9 psia (the alveolar limit). From equations (18) and (20), the ratios of adiabatic and isothermal decompression times to the minimum level in dual-gas at pure O_2 cabin atmospheres can be determined. The larger pressure ratio for the pure O_2 cabin atmosphere can be determined. The larger pressure ratio for the pure O_2 atmosphere (5/1.9 as opposed to 14.7/9.2) results in a doubling of the decompression time for both isothermal and adiabatic decompression.

It is important to know decompression time for the tug because it is an integral component of the lunar exploration program, and this information would be needed to plan time-critical maneuvers in the event of a meteoroid penetration while the tug is in the vicinity of or docked to the OLS.

The nominal cylindrical dimensions of the tug crew module are about 7.75 feet in length and 15 feet in diameter. When arbitrarily reduced by about 12 percent and 27 percent for equipment storage, the two cabin volumes used to generate the pressure-time curves of Figures 3.6-24 and 3.6-25 are 1000 ft^3 and 1200 ft^2, respectively. The initial pressure and temperature in the tug crew module is 5.0 psia pure O_2 and 70 F, respectively. The results are summarized in Table 3.6-2.

Table 3.6-2. Tug Crew Module Decompression due to Meteoroid Puncture

Hole Size (in)	Cabin Vol. (ft^3)	t_{AD} to 1.9 psia	Cabin Vol. (ft^3)	t_{AD} to 1.9 psia
2.25	1200	1.8 min.	1000	1.5 min.
1.0	↓	8.5	↓	7.1
0.5		32.0		26.6
0.25		120.0		100.0
0.125		485.0		404.0

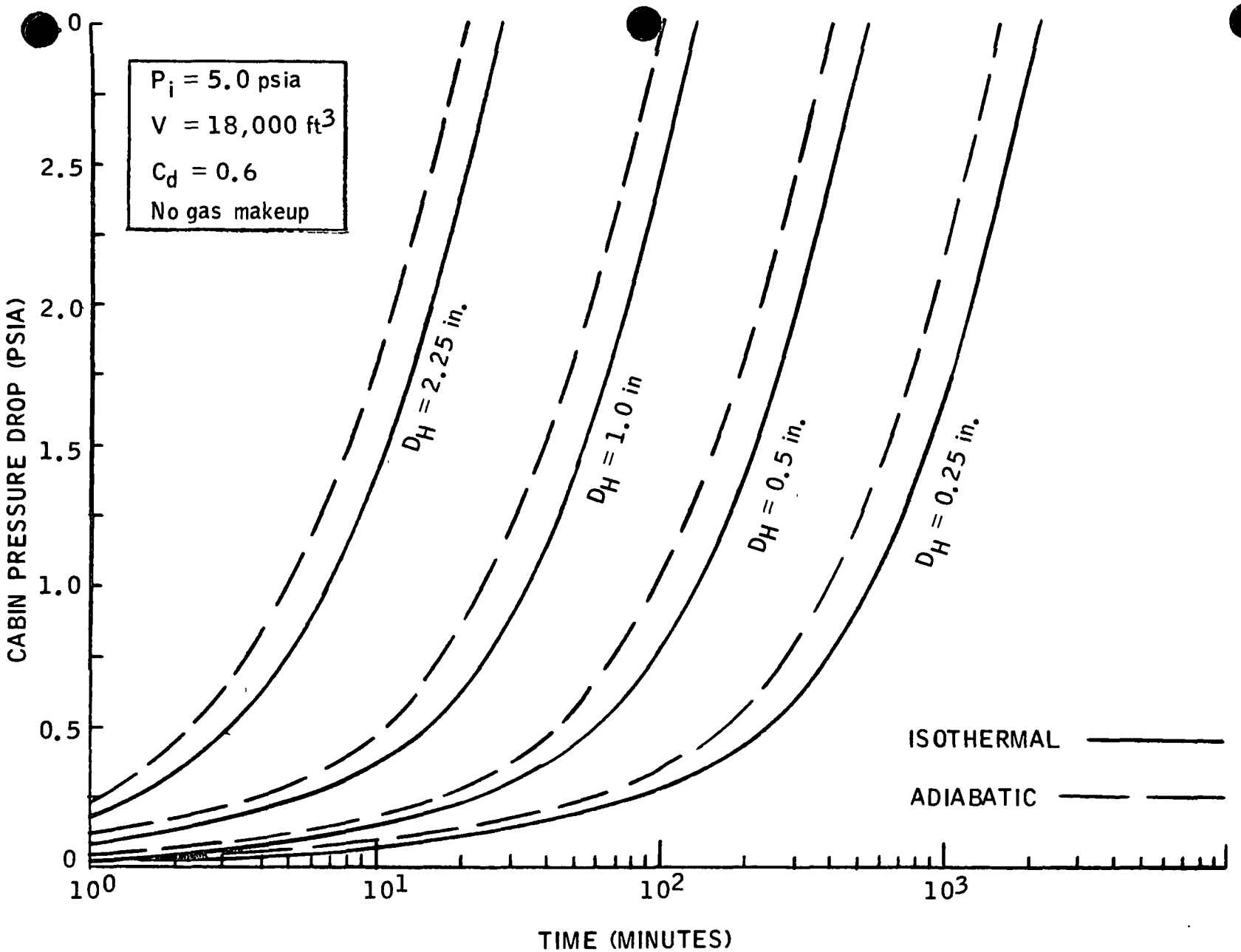


Figure 3.6-23. Decompression Time Due to Meteoroid Puncture of OLS Pressure Volume

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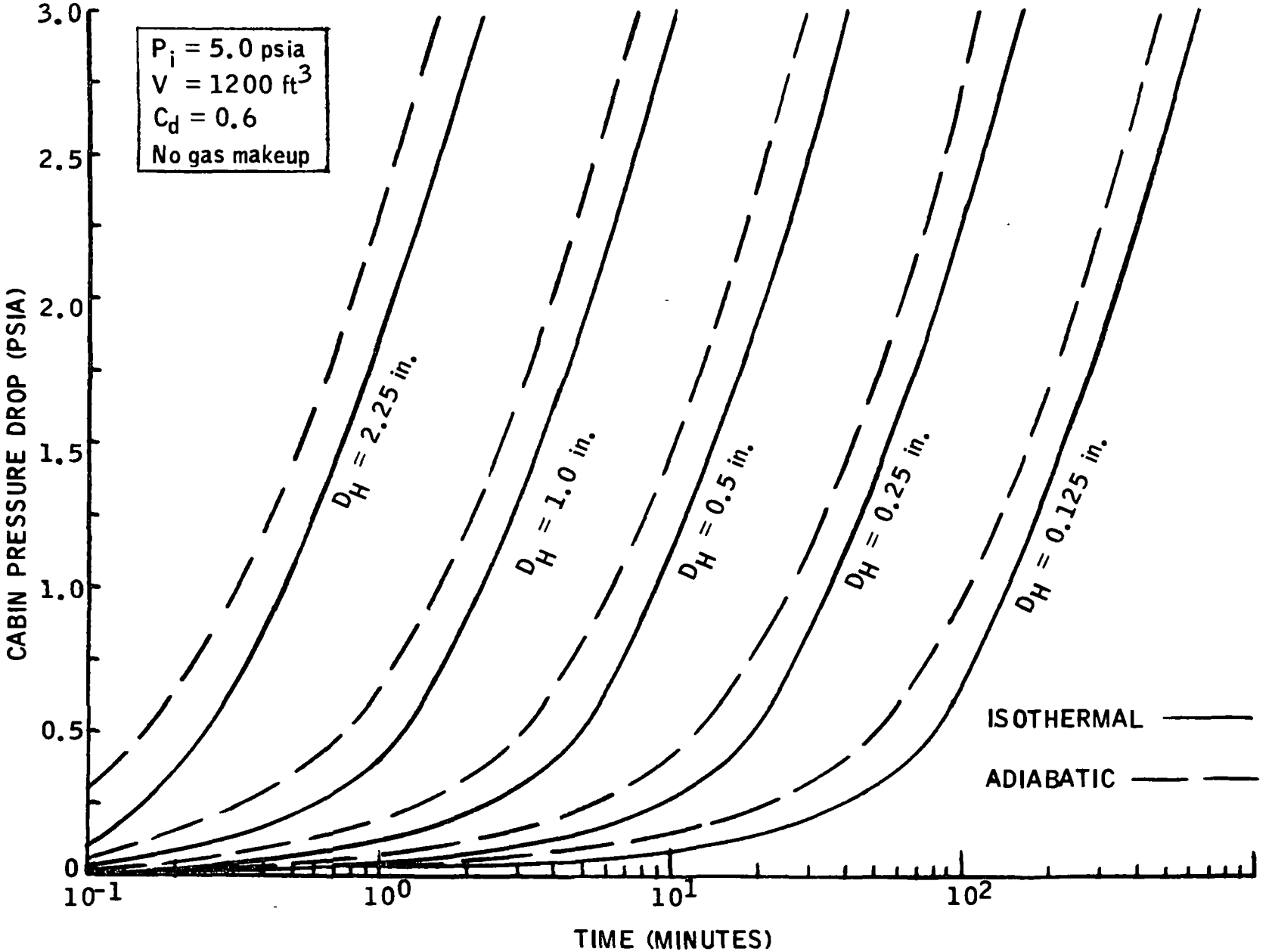


Figure 3.6-24. Decompression Time Due to Meteoroid Punctures of Tug Crew Module Pressure Volume (1200 ft^3)

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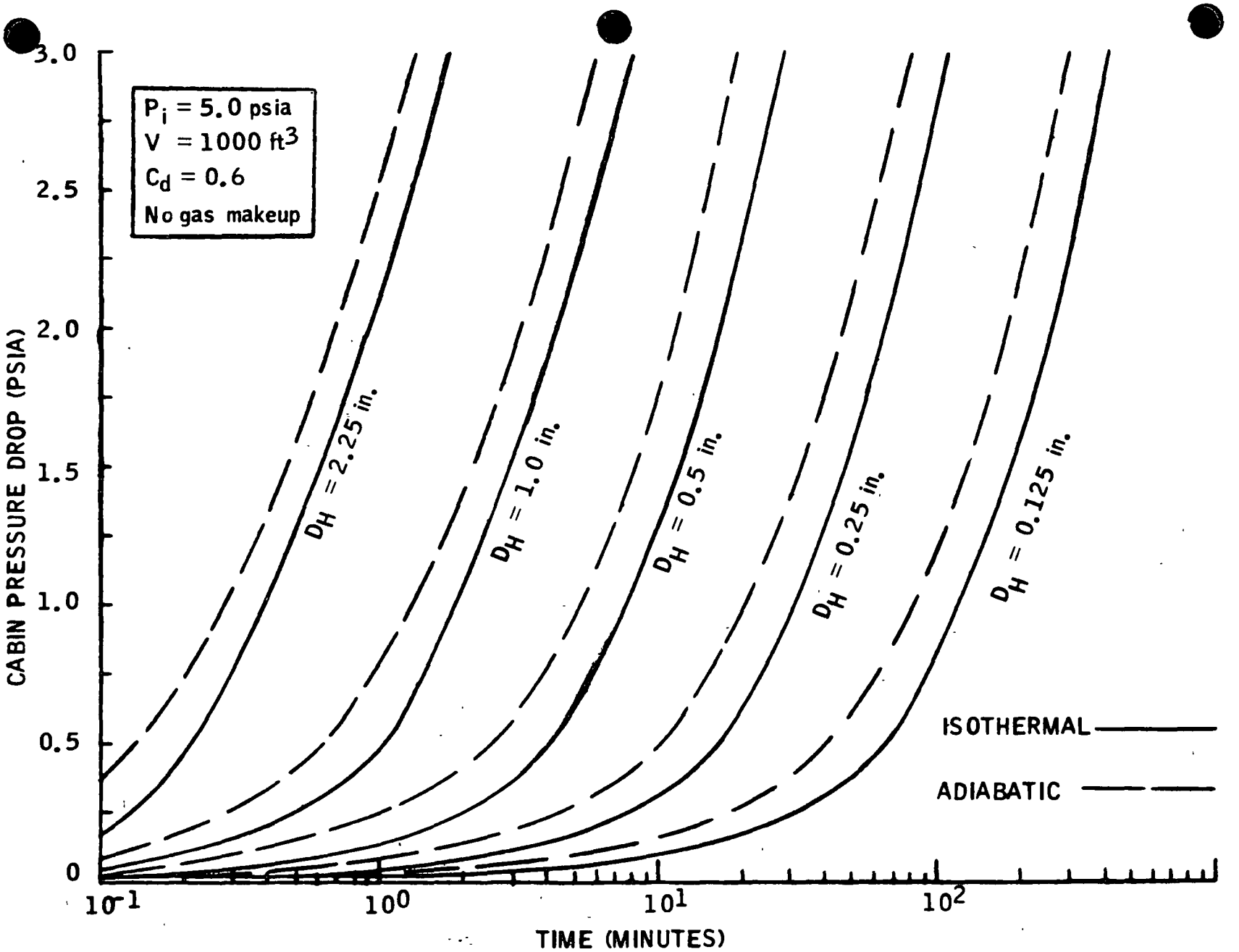


Figure 3.6-25. Decompression Time Due to Meteoroid Punctures of Tug Crew Module Pressure Volume (1000 ft^3)

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If the tug is in the vicinity of the OLS to rendezvous and dock when the crew module is punctured by a meteoroid, it appears that a puncture hole larger than 0.5 inch in diameter would pose a time-critical problem for the tug crew. However, because the surface area of the tug crew module is more than an order of magnitude less than a 33-foot diameter OLS surface area and the tug lifespan is 3 years (as compared to 10 years for the OLS), the number of meteoroid strikes upon the tug by a meteoroid of mass m or larger would be about 1/100 the number of corresponding meteoroid strikes upon the OLS. For example, at 60 n mi above the lunar surface, the OLS can expect about 0.25 impact by meteoroids $\geq$ 0.01 gram in a 10-year period. If the tug stays at the same altitude for 3 years, the crew module might expect about 0.0025 impact by meteoroids of mass $\geq$ 0.01 gram. As a result, the probability of zero impacts upon the tug crew module by meteoroids large enough to cause a time-critical decompression is approximately the same as that for meteoroids large enough to cause a time-critical decompression in the OLS.

Provided that suitable leakage detection and warning devices and a meteoroid shield are incorporated into the design of the OLS, the foregoing analysis does not indicate the possibility of a serious decompression caused by meteoroid puncture as long as cabin volumes are greater than about 10,000 ft^3. The situation might cause concern in the case of the tug because of its smaller volume and lack of a second pressurized volume (as in the case of the OLS). But the much smaller surface area and lifespan of the tug crew module compensates for the smaller volume by reducing the number of meteoroid hits that could result in catastrophe.

As far as decompression caused by meteoroid punctures is concerned, there is a definite advantage to operating the OLS cabins in a pure O_2 atmosphere. However, the hazard of fires is increased. Whereas, a puncture by a meteoroid of a certain size in a dual-gas atmosphere may not be a fire hazard, a puncture by a meteoroid of the same size may be a fire hazard in the pure O_2 atmosphere. The increase in potential of a fire hazard with an O_2 cabin atmosphere outweighs the advantages associated with decompression. Only a two gas (N_2/O_2) system will be considered in this study.

OLS Meteoroid Shielding

Meteoroid protection was sized for an OLS spacecraft concept in support of the overall study of this type vehicle. Point design data were available for the Earth Orbit Space Station (EOSS), which from the meteoroid protection standpoint is almost identical to the lunar orbit case. However, for the OLS study, parametric data were desired that would provide a firm basis for selection of a wall concept and which would be applicable to a range of vehicle sizes and mission durations.

Wall Candidates

Candidate wall sections considered are shown in Figure 3.6-26. Each candidate incorporates a meteoroid bumper or bumpers mounted outboard of the main pressure wall of the cabin. Radiator panels also serve as an outer bumper. The bumpers are supported by circumferential rings on 20- to 30-inch centers, which are constructed of low thermal conductivity material to

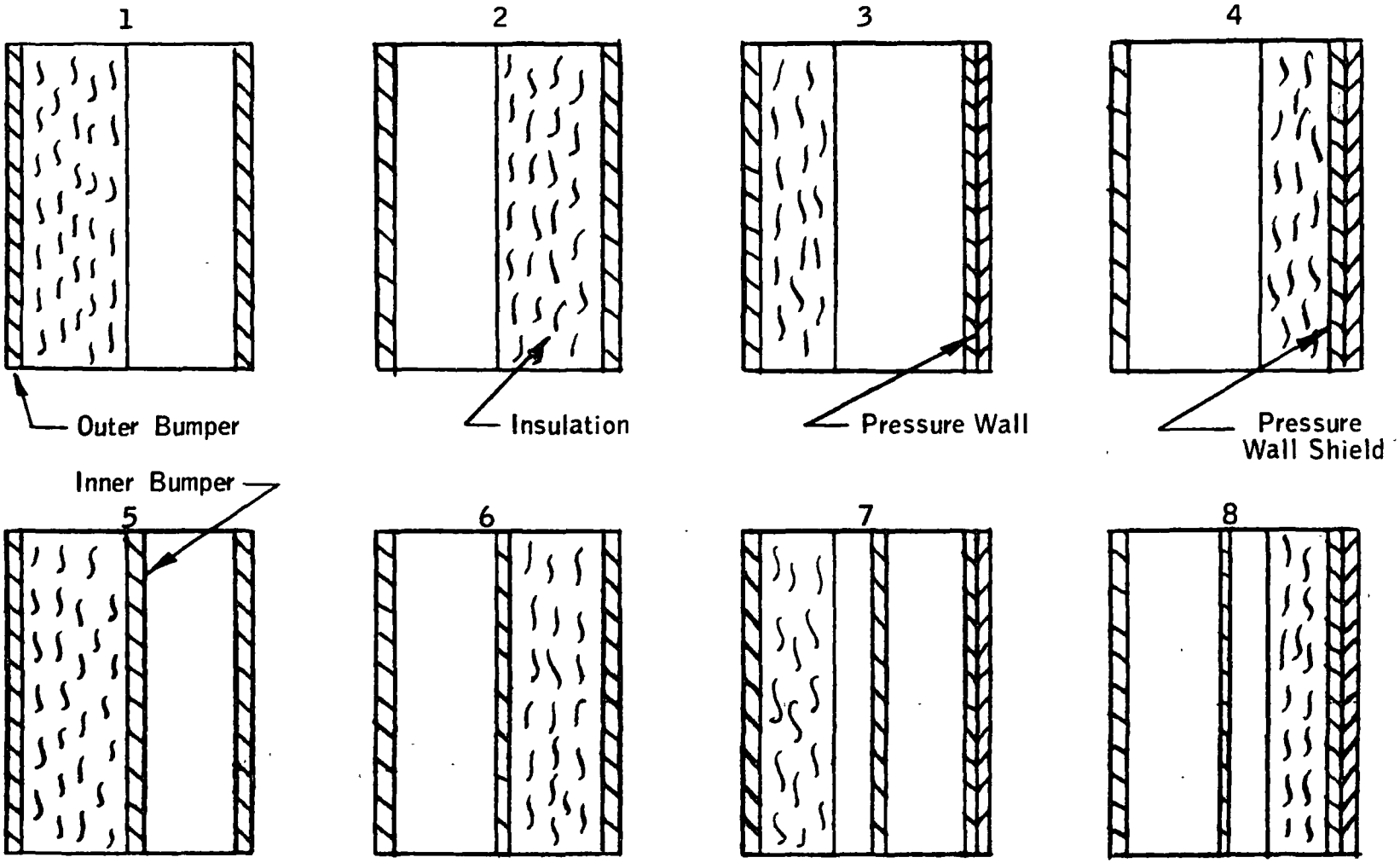


Figure 3.6-26. Wall Construction Candidates for OLS

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reduce heat transfer to and from the cabin. Multilayer insulation is located in the space between the outer bumper and the pressure wall. The first wall candidate has the insulation located on the bumper, and insulation density and/or thickness are increased above the minimum required for thermal reasons as necessary for meteoroid protection. Candidate 2 is similar to candidate 1, except the insulation is located on the pressure wall. Candidates 3 and 4 have minimum insulation, and aluminum shielding is added as required to the pressure wall. This can be wall thickening or a separate sheet. Candidates 5 through 8 correspond to candidates 1 through 4, except that a second meteoroid bumper has been added. Candidate 7 was selected for analysis because this is the reference wall concept for the NR-SD EOSS design. Candidate 3 also was selected for analysis and represents the simpler, single bumper counterpart to candidate 7. In all cases, the pressure wall was assumed to be 2219-T81 alloy.

Analysis

Conventional Poisson probability analysis was applied to relate area-time product and probability of encounter with a design meteoroid. The penetration mechanics used in the analysis were taken from Reference 1, along with data from the Meteoroid Handbook (Reference 2). Credit was taken for the protection afforded by the insulation, and wall components were sized to limit the depth of penetration of the design meteoroid into the pressure wall to an amount that would not produce leakage. Allowable penetration was taken as 0.555 times the pressure wall thickness. The meteoroid environment model used was that described previously, where flux at 60 n mi is given by

$$\log N = -14.776 - 1.213 \log m,\ m^{-2}\text{-sec}^{-1},\ 10^{-6} \leq m \leq 10^{0}$$

The data of Figures 3.6-27 and 3.6-28 were generated to establish the optimum meteoroid bumper thickness for the two selected candidates. It is convenient to do this in terms of the parameter β, which is the ratio of the mass of the impacting particle to the mass of wall material blown out by the meteoroid impact. The results of Figure 3.6-27 for the single bumper wall indicate that: (1) for a 0.5 gram/cm^3 cometary meteoroid, the optimum β value is approximately 1.0 and is relatively insensitive to the area-time (AT) product; and (2) for a more dense 3.5 gram/cm^3 asteroidal type meteoroid, the optimum β value drops to 0.5. The results of Figure 3.6-28 for the dual bumper wall indicate that: (1) the optimum β value is approximately 0.7, and (2) the optimum bumper thickness ration (t_1/t_2) is in the 0.2 to 0.3 range. Based on these findings, values of $\beta = 0.8$ at $t_2/t_1 = 0.2$ appear to be appropriate design values, and the bumper sizing curves of Figure 3.6-29 are based on these values.

The data of Figures 3.6-30 through 3.6-32 were generated to facilitate shield concept selection and to provide weight data. Parameters treated are the AT product, the probability of no failure (P_{nf}), and the pressure wall thickness (t_r). Minimum gage limits were set for the bumpers and, for the lower AT values, this material plus the pressure wall provides sufficient protection. As the value of AT is increased, a point is reached where thicker bumpers are required and shielding must be added to the pressure wall ($t_s > 0$). For the higher AT values, the dual bumper wall concept holds a

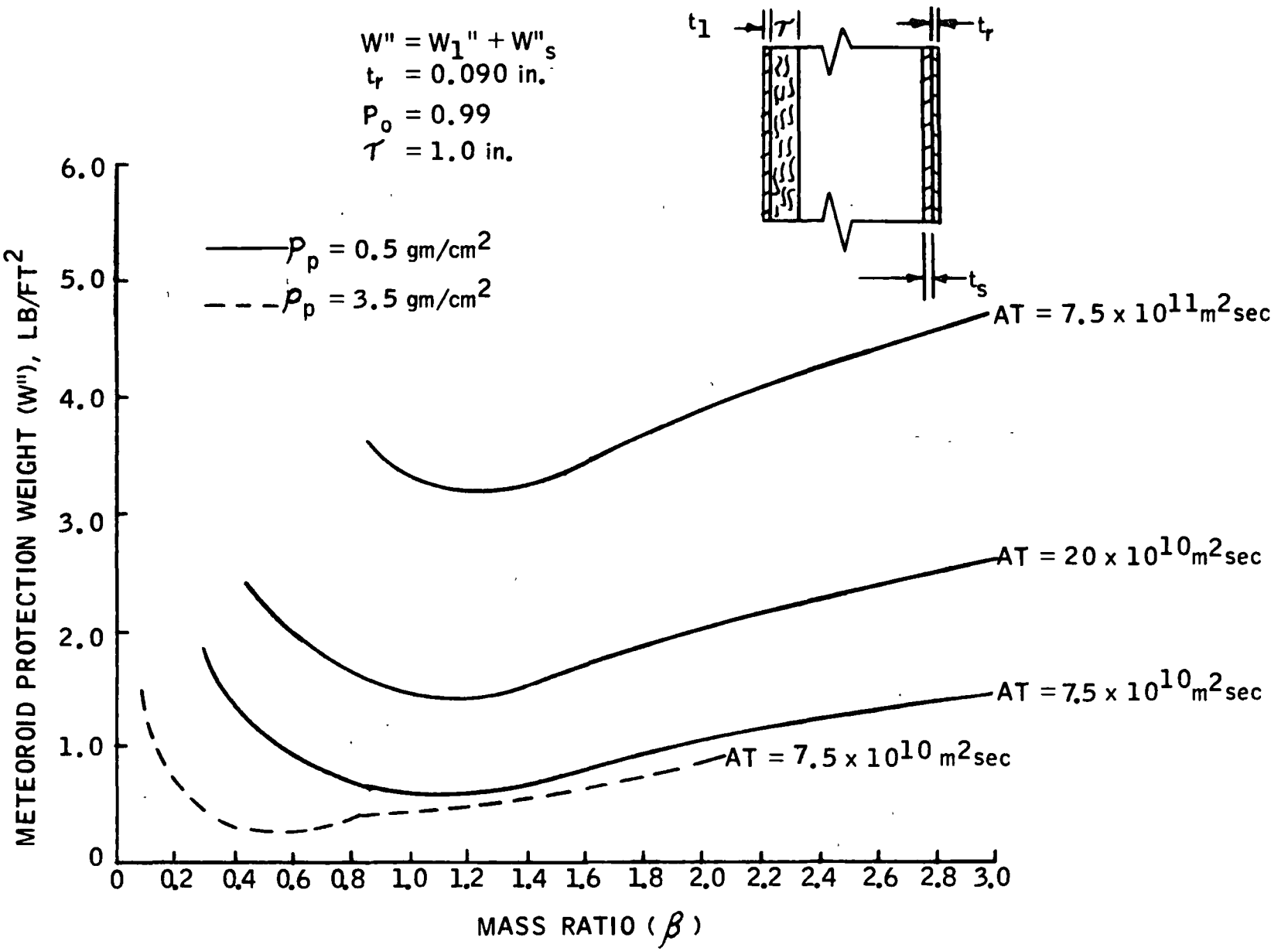


Figure 3.6-27. Effect of Mass Ratio on Meteoroid Protection Weight - Single Bumper Concept

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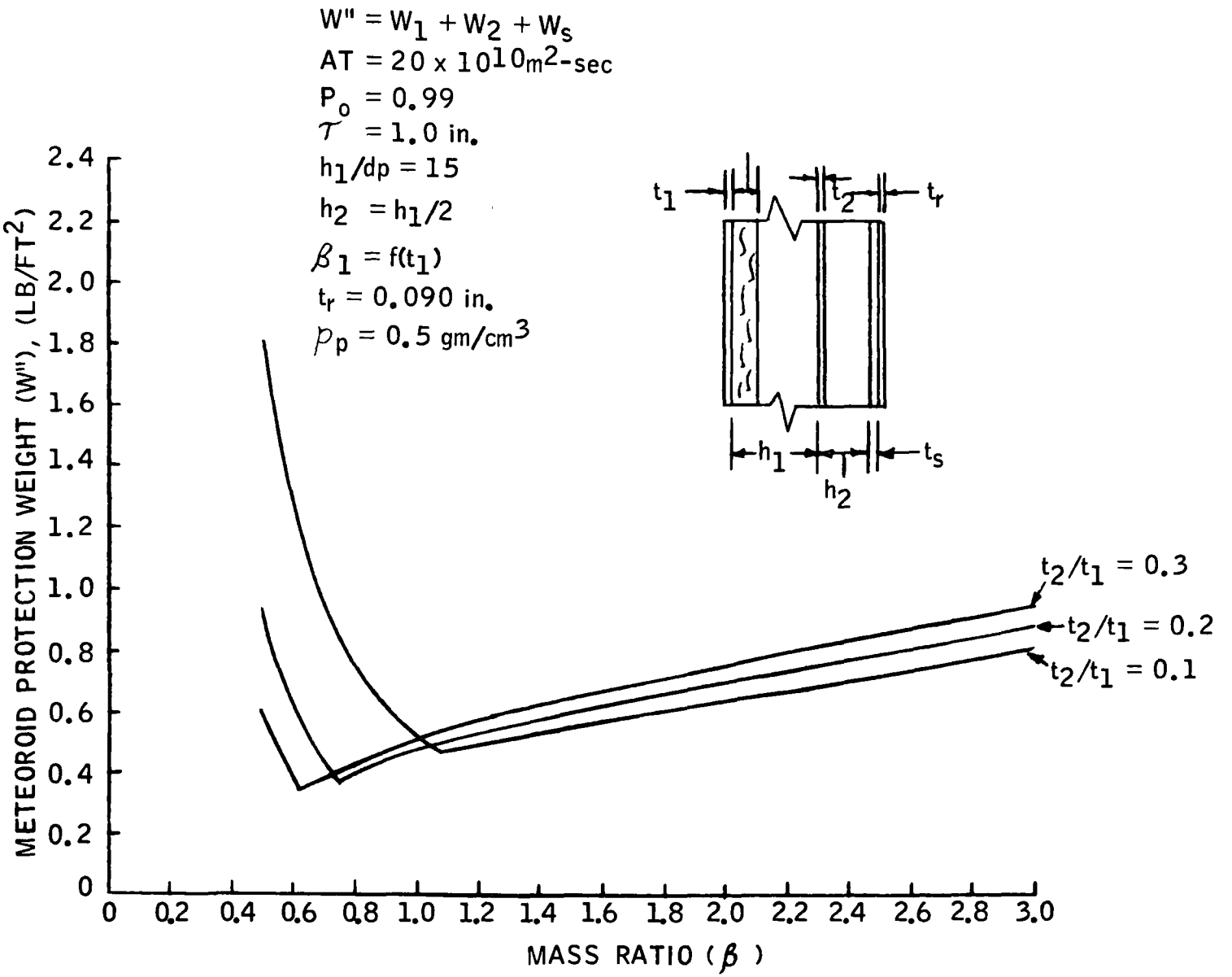


Figure 3.6-28. Effect of Mass Ratio on Meteoroid Protection Weight - Double Bumper Concept

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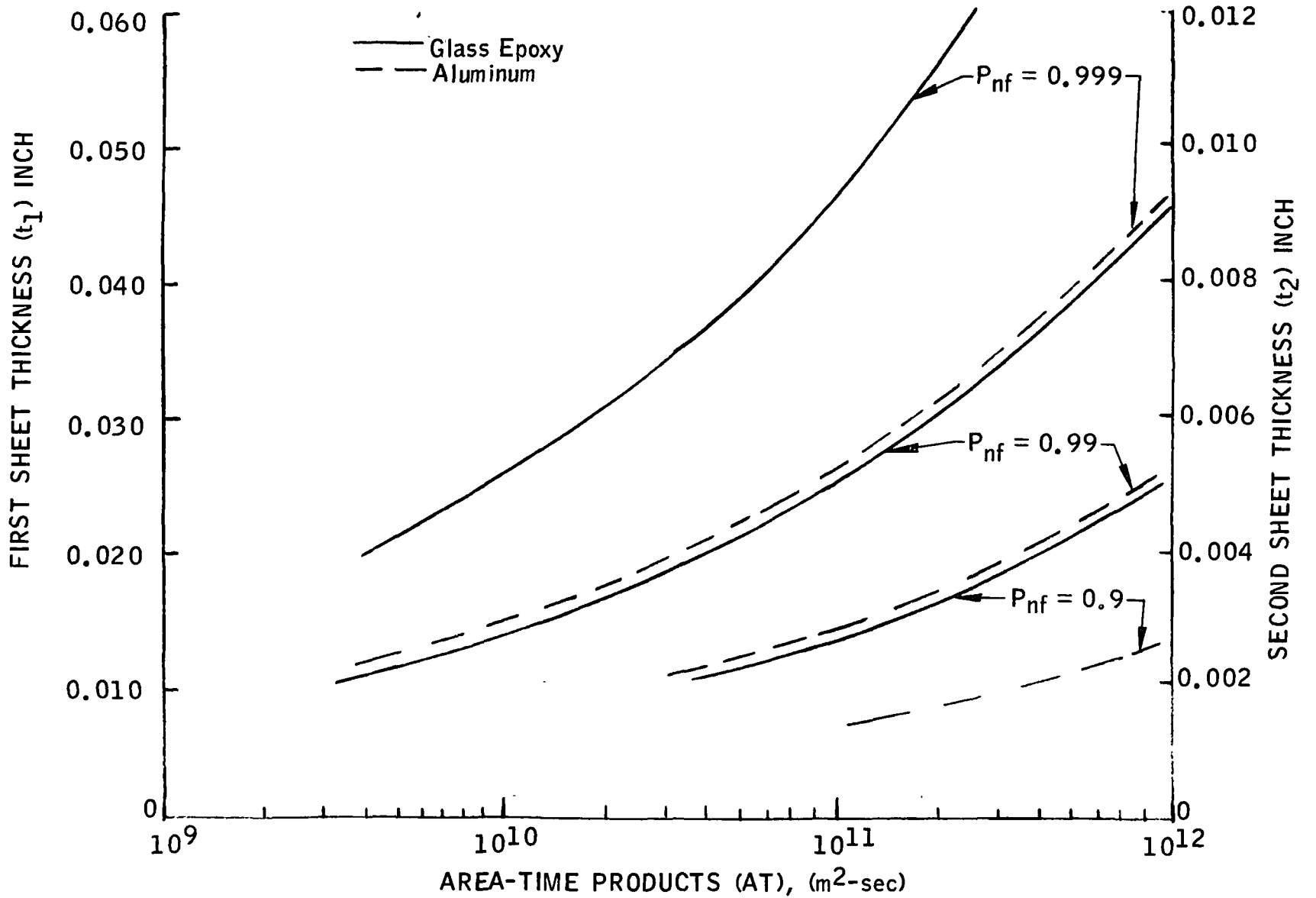


Figure 3.6-29. Bumper Sizing, Single or Double Bumper Concept

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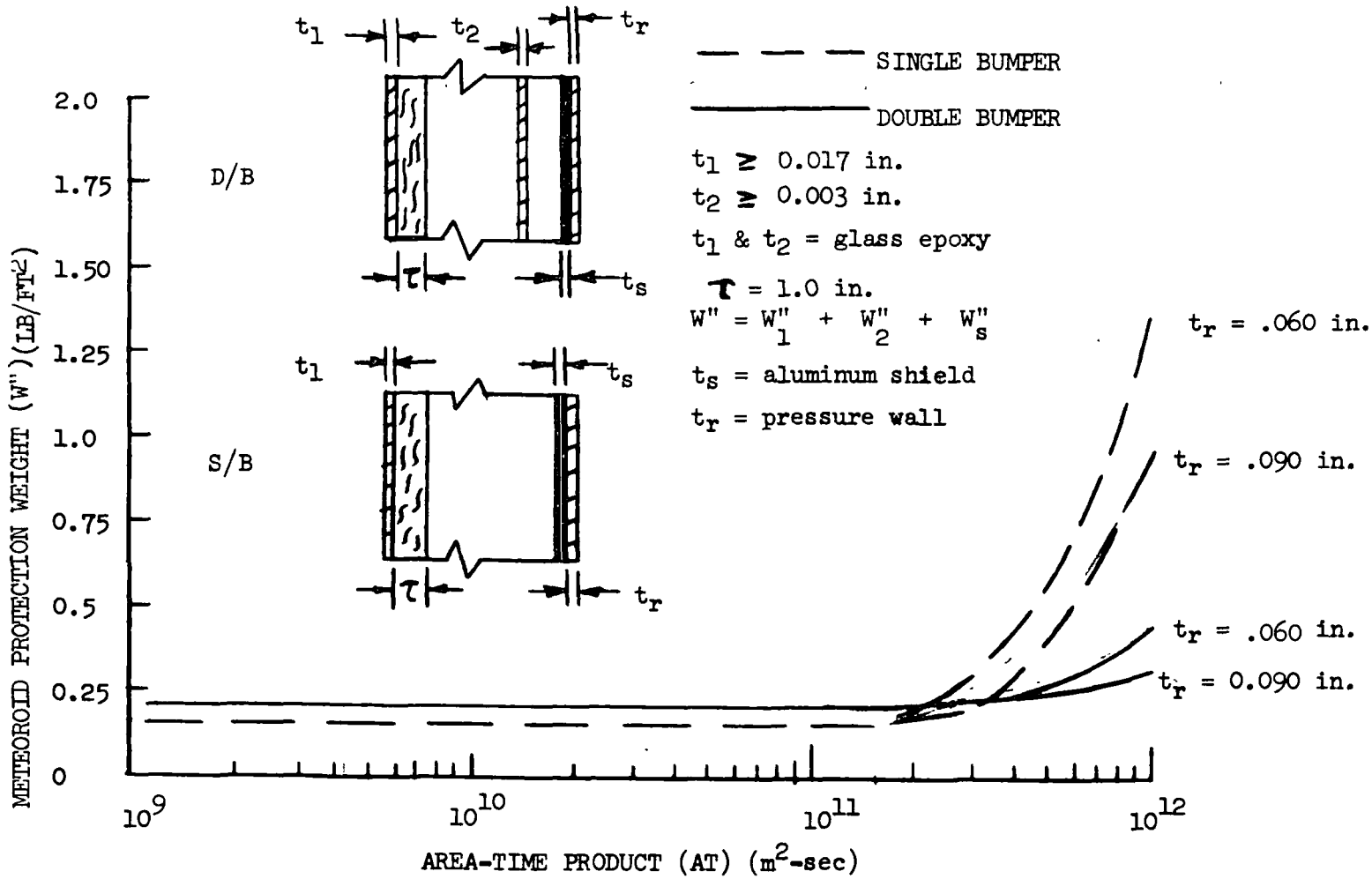


Figure 3.6-30. Comparison of Meteoroid Shield Types for OLS - P_{nf} = 0.9

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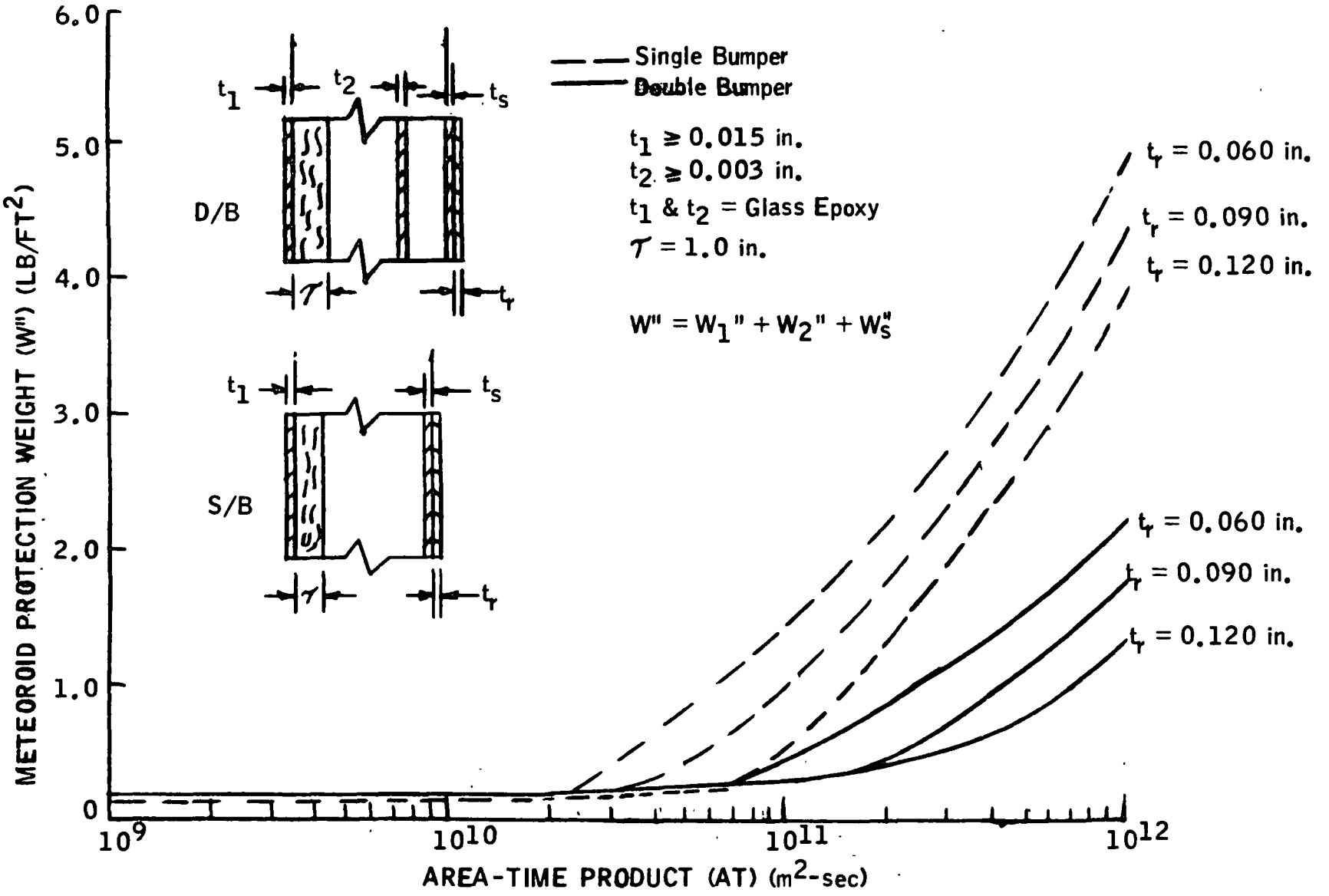


Figure 3.6-31. Comparison of Meteoroid Shield Types for OLS - P_{nf} = 0.99

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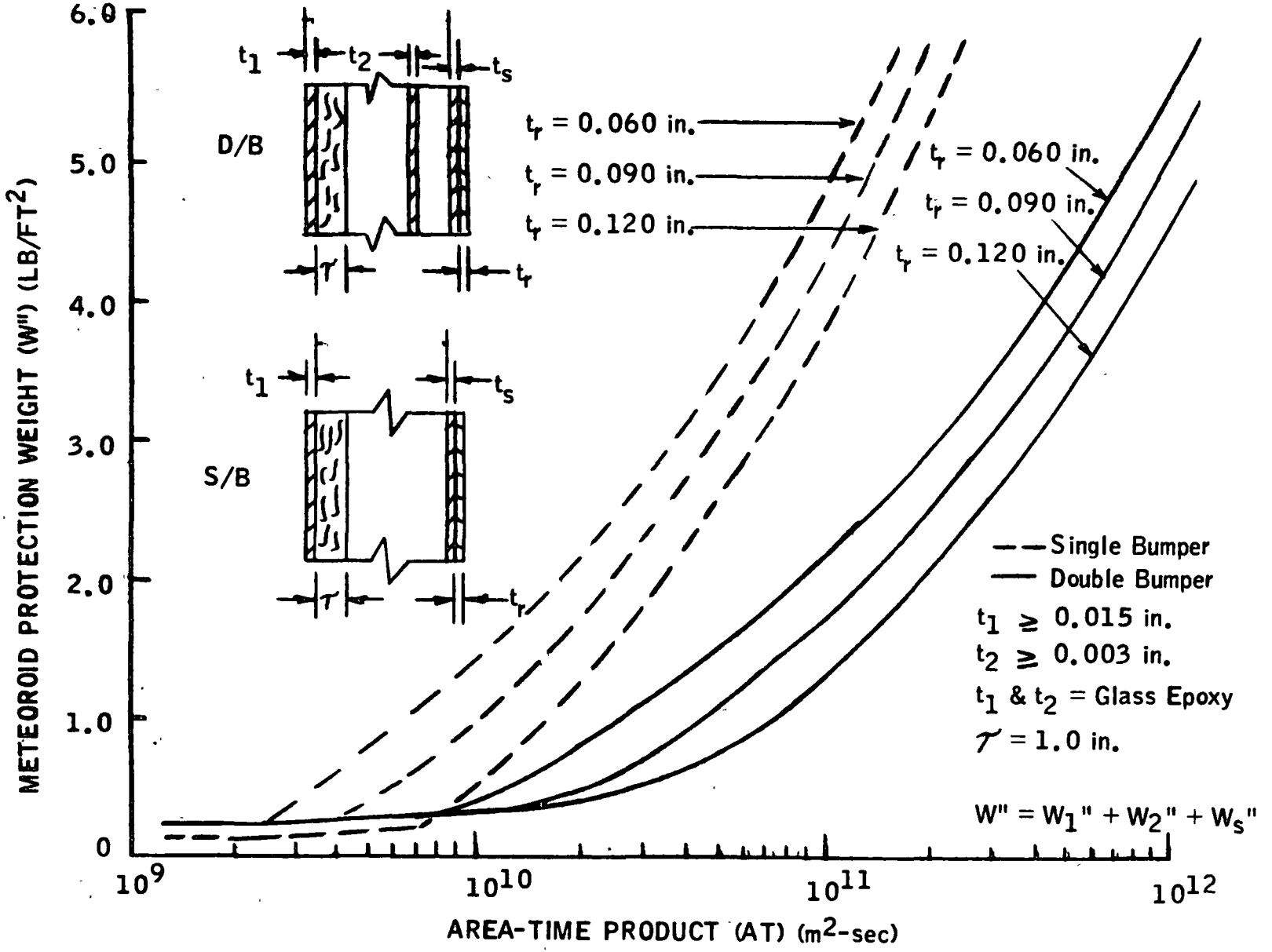


Figure 3.6-32. Comparison of Meteoroid Shield Types for OLS - P_{nf} = 0.999

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large weight advantage. For the lower AT values, the figures do not show a clear choice; however, the reduced bumper spacing associated with the dual bumper concept makes it the best choice for most cases.

References

1. Richardson, A. J., and Sanders, J. P., Penetration Damage to Multisheet Structures Based on Debris Particles of Two Materials, NR-SD Report, SD 70-463 (October 1970).

2. Richardson, A. J., and Treppo, V. S., Sizing Single and Double Sheet Structures for Meteoroid Protection, NR-SD Report STR 242 (June 1970).

3. Weidner, D. K., Natural Environment Criteria for NASA Space Station Program (Second Edition), NASA TMX-53865 (20 August 1970).

3.6.3 NUCLEAR RADIATION ENVIRONMENT

The natural nuclear radiation environment in lunar orbits 60 to 100 nautical miles altitude will be essentially half the free space environment at 1 AU. This environment consists of the following components:

solar wind particles
galactic (cosmic) particles
solar flare particles

The solar wind consists of a high flux (10^8 - 10^9 protons/cm^2-sec) of kev protons and ev electrons blowing radially outward from the sun (300 to 800 km/sec directed velocity). These particles are continuously present, but their flux and velocity vary within the observed limits given previously. The solar wind particles are stopped by $<$ 1 mil of material so they pose a threat only to exposed surfaces (Reference 1). However, surface thermal control coatings (white paints) will be vulnerable -- Lunar Orbiter I exceeded its specified temperature limits within 1 week because of thermal control coating degradation (the white paint turned yellowish, then brown). Metallic surfaces are expected to stand up for periods of months to years in the lunar orbit environment, and inorganic surfaces are expected to perform well for 1 to 6 months. However, organic surfaces often degrade appreciably within $\lesssim$ 1 month (Reference 2).

A second effect of the solar wind is to prevent exposed surfaces from acquiring and holding appreciable voltages ($\lesssim$ 10 volts, normally). Because both electrons and protons are present, any exposed surface with high voltage will attract charges of one sign and repel charges of the other. The effective cross-section for charge attraction becomes much larger than the surface itself as the voltage increases. For high voltages (kilovolts to megavolts) this effective cross-section increases orders of magnitude above the physical size of the surface (especially for positive voltages). If it is desired to maintain a high voltage exposed surface, the required electrical current to maintain the high voltage surface can be orders of magnitude above the

10^{-11} amp/cm^2 of the solar wind or the 10^{-8} amp/cm^2 because of the photoelectric effect (solar ultraviolet radiation on exposed metal surfaces (Reference 3).

The galactic (cosmic) radiation consists of a low flux 2 to 4 particles/cm^2-sec) of very high energy ($\gtrsim$ 100 Mev) protons, alpha particles, and heavier nuclei. The flux of these particles (especially below 1 Bev) decreases when the sun is active. Because of their energies, shielding against these particles is impractical -- thicknesses in excess of 100 gm/cm^2 are required (Reference 1). Fortunately, the low flux of particles makes shielding unnecessary. In lunar orbit, the dose rate caused by galactic particles is less than 10 rem/year (independent of shielding).

One effect of galactic radiation is to provide a background that must be reckoned with. Not only do galactic particles reduce the amount of radiation caused by other sources (e.g., solar flares) that men can tolerate, but they also limit the use of many radiation-sensitive instruments designed to measure other quantities (e.g., ultraviolet radiation, X-rays, etc.). In general, charged particle experiments can be protected against galactic particles by anticoincidence techniques, but uncharged radiation (photon, neutron) experiments are vulnerable because anticoincidence techniques cannot be used for them (Reference 4).

In particular, photographic film can at most be stored a few months in lunar orbit, and any exposed film should be developed immediately after use. The problem of storing photographic film in lunar orbit is shown quantitatively in Figures 3.6-33 and 3.6-34. Figure 3.6-33 shows that shield thicknesses of $\lesssim$ 80 gm/cm^2 ($\lesssim$ 160 lb/ft^2) actually increase the galactic dose rate. A shield thickness of several hundred gm/cm^2 would be required to appreciably reduce film fogging.

The degree of fogging expected as a function of film speed (ASA number) and radiation dose are shown in Figure 3.6-34. For lunar surface photography, a fairly fast film will probably be used to minimize blurring caused by spacecraft motion in orbit. For 6 months in lunar orbit (galactic dose $\lesssim$ 10 rad), ASA-10 film will suffer a fogging optical density of $<$ 0.1 (which will probably not cause problems), but ASA-80 film and faster will suffer a fogging optical density of $\geq$ 0.8 (which will make the film useless for most purposes). Experiments requiring sensitive film for long exposure (e.g., gegenschein light studies and meteor photography) should be accomplished with fresh film to obtain maximum information.

Solar flares are events that sometimes produce appreciable fluxes ($\gtrsim 10^8$ protons/cm^2) of high energy ($\gtrsim$ 10 Mev) particles at 1 AU. Large bright flares near the west limb of the sun have fairly high probabilities ($\sim$30 percent) of producing such particle fluxes, other flares have much lower probabilities. Therefore, even though there are several hundred flares per year, only a few ($\lesssim$ 20 in active years, and less in quiet years) produce large proton fluxes at the earth. Sunspots have been observed since 1742, and we are currently in cycle 20 of the 11-year sunspot cycle. The annual solar flare proton fluxes for the previous sunspot cycle are listed in Table 3.6-3, and the corresponding tissue doses behind 1 gm/cm^2 are shown

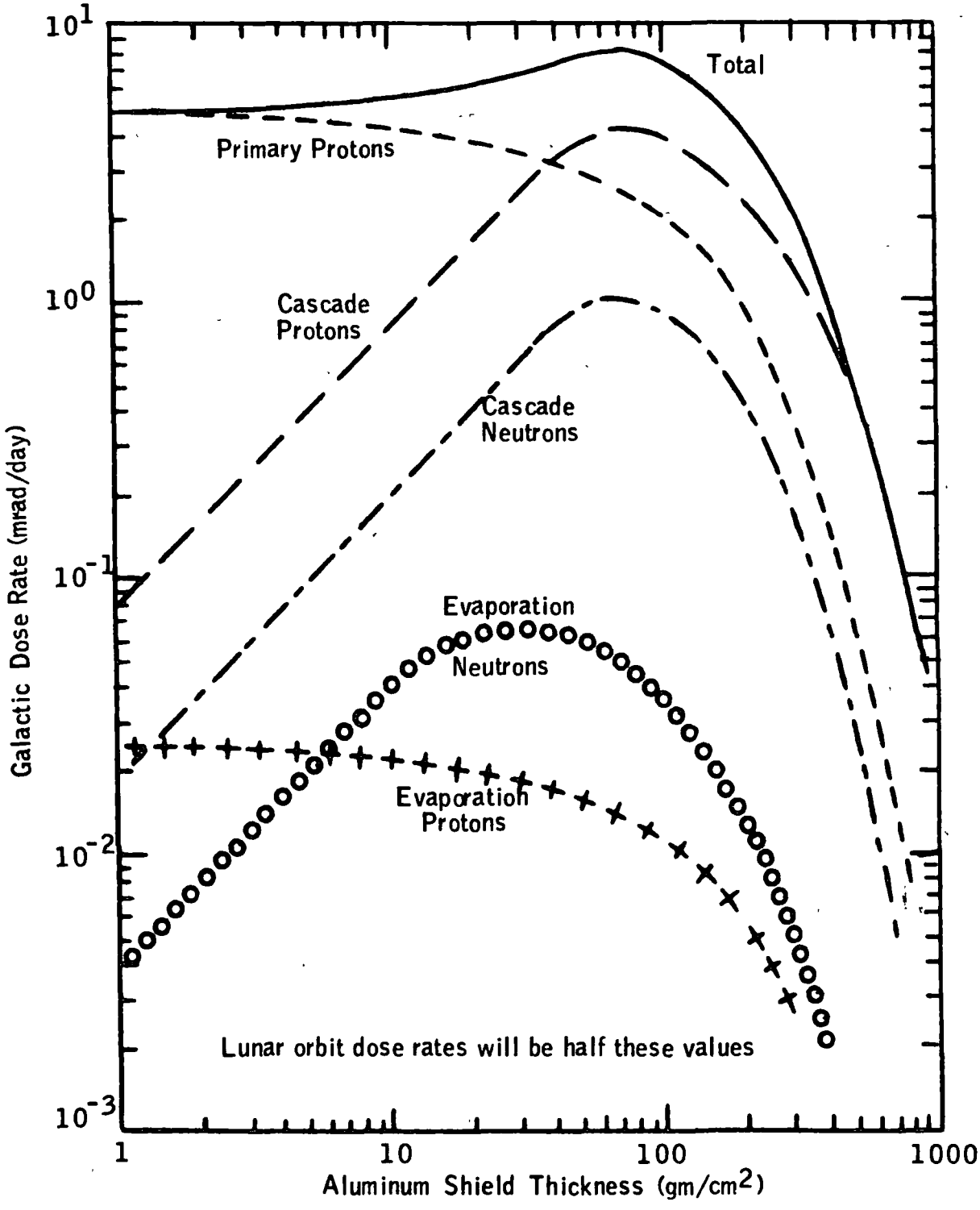


Figure 3.6-33. Galactic Dose Rate Versus Aluminum Shield Thickness

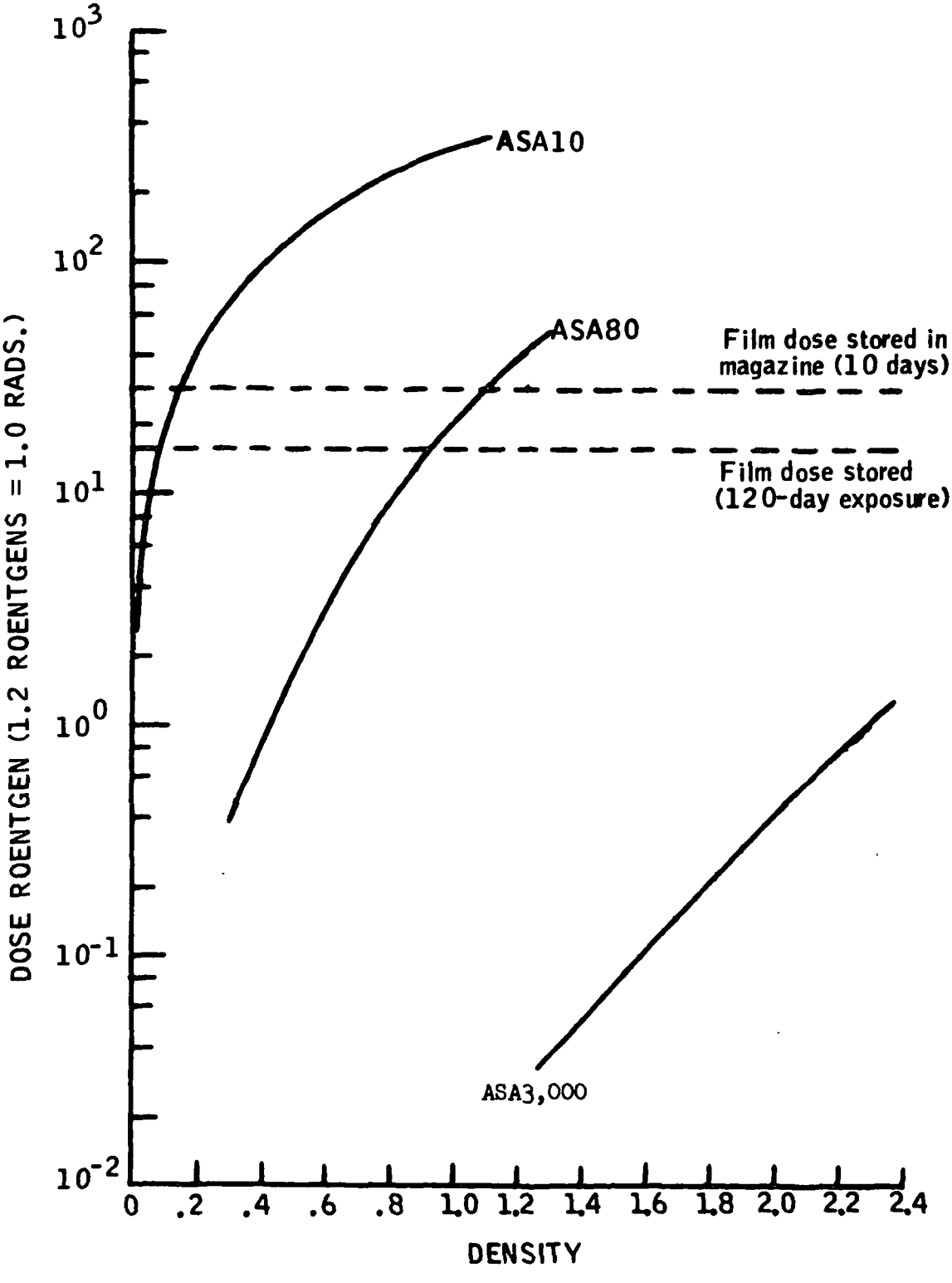


Figure 3.6-34. Film Density Versus Radiation Exposure for Various ASA Speed Films

Table 3.6-3.

Observed Annual Smoothed Sunspot Numbers (R) and Solar Flare Annual Particle Fluxes ($\int \phi$) for the Years 1956 to 1968 (protons/cm^2)

Year	R_{Y-1}	$\int \phi_Y$ >10 Mev	$\int \phi_Y$ >30 Mev	$\int \phi_Y$ >100 Mev
1956	41.5	2.7×10^9	1.1×10^9	3.7×10^8
1957	133.8	1.0×10^{10}	8.5×10^8	5.9×10^7
1958	187.9	1.1×10^{10}	1.4×10^9	9.6×10^7
1959	189.5	2.2×10^{10}	4.3×10^9	4.2×10^8
1960	157.5	1.3×10^{10}	2.7×10^9	3.9×10^8
1961	108.0	2.1×10^9	4.2×10^8	4.9×10^7
1962	~60	1.6×10^6	2×10^5	1.4×10^4
1963	~30	5.6×10^8	6.5×10^7	6.4×10^6
1964	~20	1.2×10^8	1.2×10^7	1.1×10^6
1965	~10	2.2×10^8	2.2×10^7	2.0×10^6
1966	16.3	8.1×10^8	1.0×10^8	1.0×10^7
1967	49.7	2.2×10^9	2.7×10^8	2.8×10^7
1968	89.7	2.1×10^9	2.4×10^8	2.3×10^7

in Figure 3.6-35. The 11-year solar activity cycle had its last maximum in 1969 and will have its next maximum in ~ 1980. The next minimum will be in 1975-1976 (Reference 5).

Each solar flare event is usually preceded by calcium plages and sunspots (in that order). Each significant solar event is accompanied by various types of radio noise and X-rays in addition to the ultraviolet light they produce (only the largest, brightest flares can be observed in white light). Even after the flare has taken place on the sun, it usually takes $\gtrsim 4$ hours for the charged particles it accelerated to reach 1 AU. Consequently, the crew of the lunar orbiter will have at least a few days warning of an impending large solar flare (in approximately 80 percent of the cases) and a few hours after the event before the charged particle radiation becomes significant. The flare-produced radiation typically lasts for 1 to 3 days.

A model solar flare event is given in NASA TMX-53865 (1970 edition). The proton flux for this model event is:

$$\int \phi > E = \begin{cases} 7.25 \times 10^{11} E^{-1.2} & , \quad 1 \leq E \leq 10 \text{ Mev} \\ 3.54 \times 10^{11} e^{-P(E)/67} & , \quad 10 \leq E \leq 30 \text{ Mev} \\ 2.64 \times 10^{11} e^{-P(E)/73} & , \quad E \geq 30 \text{ Mev} \end{cases}$$

where

$$P(E) = \sqrt{E(E + 1876)}$$

This model is shown in Figure 3.6-36, and three date points are given below:

$$\int \phi > 10 \text{ Mev} = 4.5 \times 10^{10} \text{ P/cm}^2$$

$$\int \phi > 30 \text{ Mev} = 1.0 \times 10^{10} \text{ P/cm}^2$$

$$\int \phi > 100 \text{ Mev} = 6.5 \times 10^{8} \text{ P/cm}^2$$

The word "event" is not to be associated with the conventional sense of the word. A typical solar flare event is a discrete outburst of energetic protons and alpha particles whose radiation can last for one to three days. The model solar flare "event" of NASA TMX-53865 is actually an integrated particle flux of several solar flare events which occurred over a short period of time in 1960. The NASA solar flare model has idealized this solar activity into one discrete event in time. The probability of such an event (or events) occurring in time is given by a Poisson probability distribution.

The NASA solar flare "event" model is approximately twice the annual solar flare proton flux observed for the year 1959 (the most active solar flare particle event year on record). In reality, the probability of

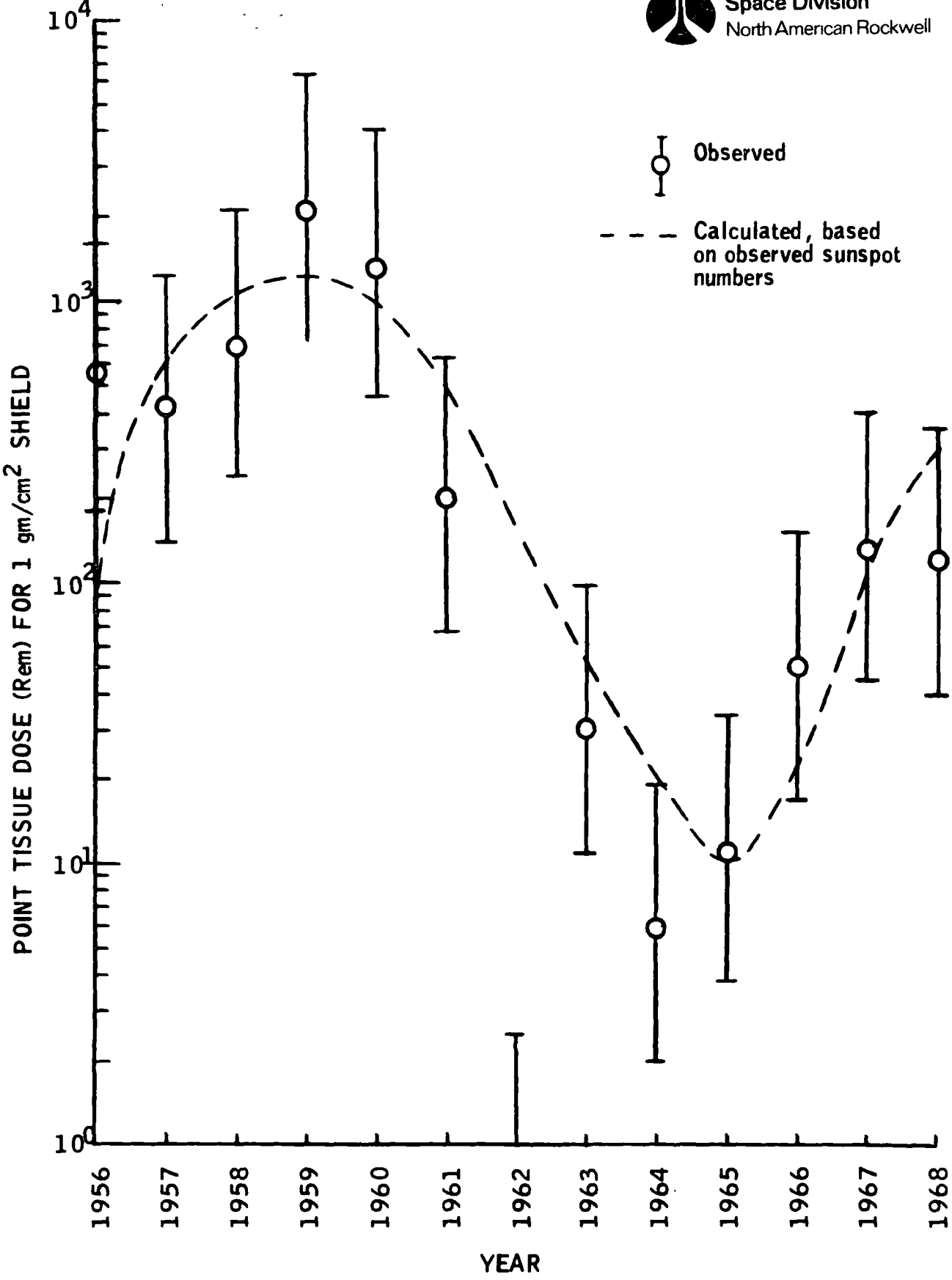


Figure 3.6-35. Point Tissue Dose Rates at 1 AU - Distance Solar Flare Particles (1 gm/cm^2 Shielding)

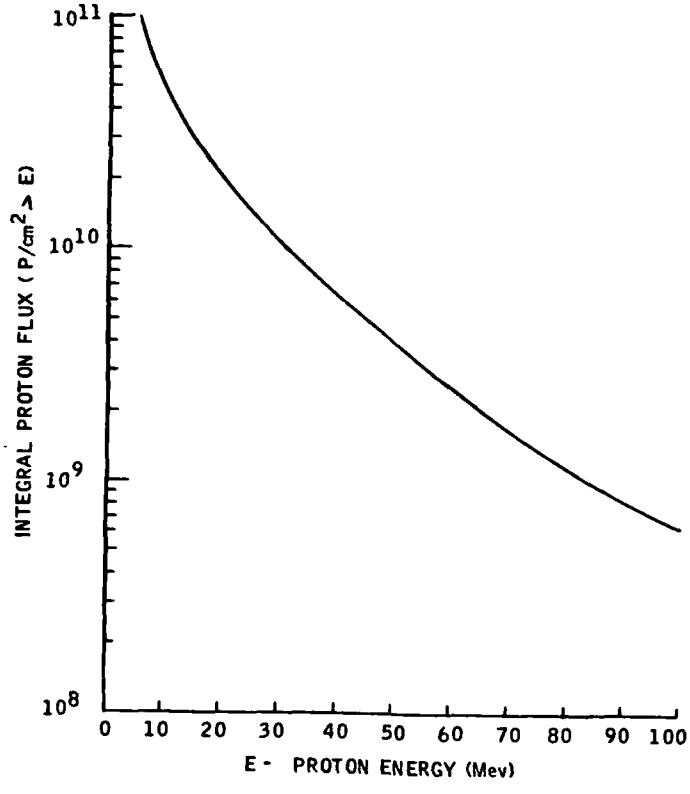


Figure 3.6-36. Model Solar Flare Proton Event

occurrence of a discrete solar flare event having the magnitude of the NASA model is quite small. The inherent conservatism of the NASA model is balanced somewhat by the fact that the time history of the biological radiation dose absorbed by an OLS crew - as predicted by the model - is described by a step function. This is because the Poisson probability distribution given for the model predicts the occurrence of discrete "events" rather than continuous integrated particle flux levels. In any given year, there will be many solar flare events, a few of which can be classified as major. The integrated particle flux levels and the subsequent cumulative biological radiation dose are then, in reality, continuous functions of time.

The Poisson probability distribution used to predict the occurrence of a NASA solar flare event is given as

$$P(X) = \frac{e^{-0.01T}(0.01T)^{X}}{X!}$$

where X is the number of solar flare events, and T is the mission duration

in weeks. Figure 3.6-37 plots the probability of a crewman witnessing exactly X solar flare "events" as a function of his mission duration. Data points for mission durations of interest are tabulated below. The first column of Table 3.6-4 gives the probability of witnessing no solar flare events in mission durations of 90, 180, and 234 days. The nominal crew stay time in lunar orbit is 234 days (see Section 7.0 of Volume II). The last column gives the probability of at least three NASA solar flare events occurring in mission durations of 90, 180, and 234 days, which from elementary probability is the sum (P(1) + P(2) + P(3).

Table 3.6-4. Solar Flare Occurrence Probability for Several Mission Durations

Mission Duration (Days) \ X	0	1	2	3	
	P(0)	P(1)	P(2)	P(3)	$P(X \leq 3)$
90	.8793	.113	.00725	.000311	.1206
180	.773	.1985	.0253	.00217	.336
234	.716	.239	.0399	.00445	.2833

Shielding Requirements

The nominal program duration of the OLS is ten years. Figure 3.6-38 shows the probability of the OLS witnessing at least one of the NASA solar flare events as a function of the OLS program duration. The probability of the OLS witnessing at least one event in ten years is a near certainty ($P(X \geq 1) = 0.995$ in 10 years). Therefore, it is evident that the OLS must provide enough shielding to protect any crewman from the radiation of at least one NASA solar flare event. NASA document TMX-53865 stipulates that OLS crewmen must be protected from solar flare radiation to a 2-sigma confidence level. Thus, given a crew mission duration, the radiation protection design criteria is to provide enough shielding to protect against that number of NASA solar flare events whose probability of non-occurrence is 95 percent or greater (i.e., $P(X) \leq 0.05$).

To determine X above, Figure 3.6-37 is replotted with probability of non-occurrence (synonymous with confidence level P_c) as a parameter instead of solar flare event. The steps on Figure 3.6-39 occur at that time T for which shielding must be provided for an additional solar flare event to maintain the same confidence level. Obviously, the longer an OLS crewman stays in lunar orbit, the more shielding is required to provide him with the same confidence level (P_c) of non-exposure to solar flare radiation. Selected data points from Figure 3.6-39 are summarized in Table 3.6-5. It is seen that for crew mission durations up to 38 weeks in length, a 2-sigma confidence level is ensured by providing shielding to protect against one NASA solar flare event. The blanks in Table 3.6-5 imply that the mission duration for two and three "design-to" solar flare events is off the time scale of Figure 3.6-39 for the indicated confidence levels.

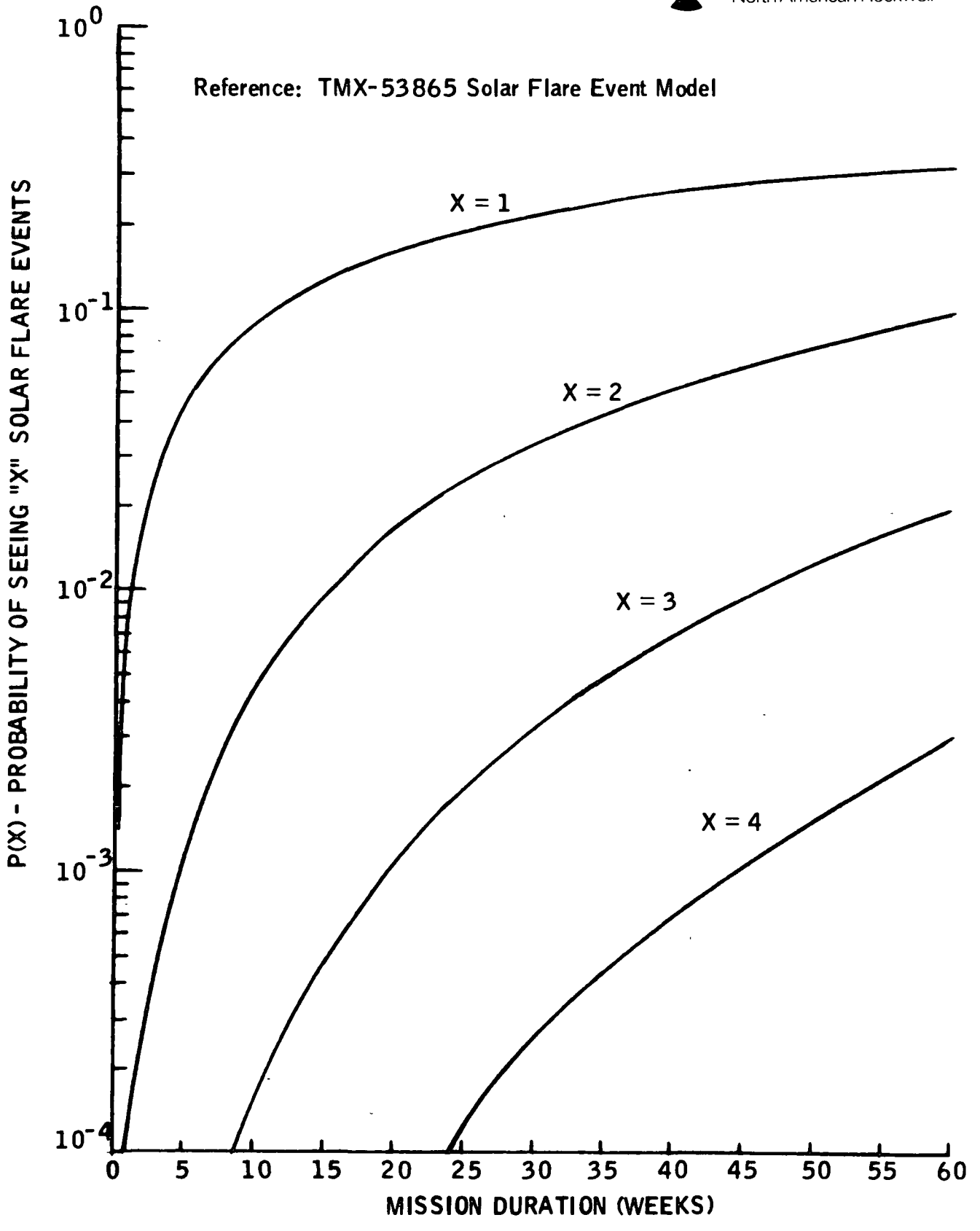


Figure 3.6-37. Solar Flare Event Probability as Function of Mission Duration

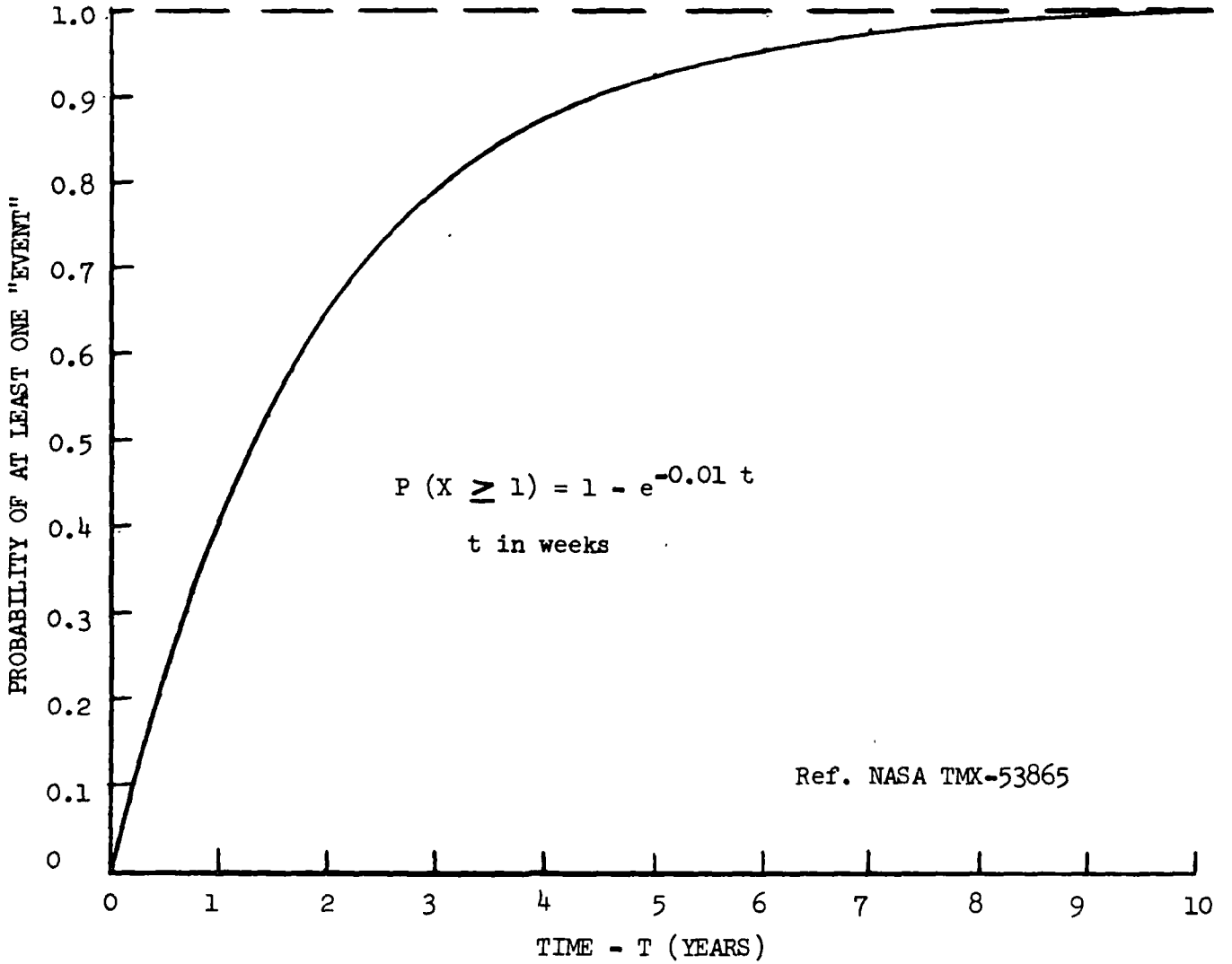


Figure 3.6-38. Model Solar Flare "Event" Probability as A Function of OLS Program Duration

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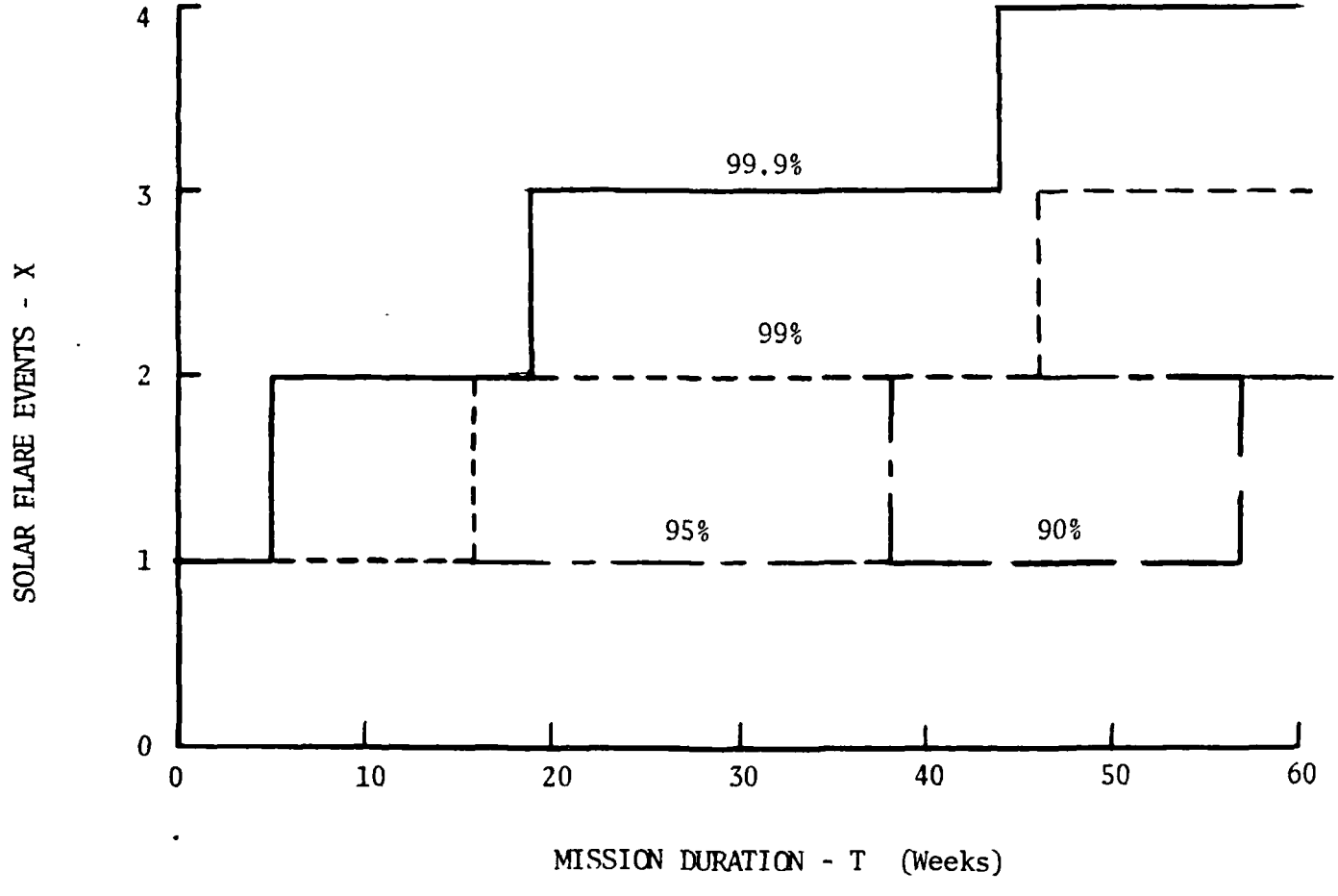


Figure 3.6-39. "Design-To" Solar Flare Events Versus Mission Duration (Probability of Non-occurrence as a Parameter)

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Table 3.6-5 "Design-To" Solar Flare Events for Several Confidence Levels (T in Weeks)

P_c \ X	1	2	3
99.9%	T < 5	T < 19	T < 44
99%	T < 16	T < 46	--
95%	T < 38	--	--
90%	T < 57	--	--

The relationship between particle flux and biological dose is a function of particle type and particle energy. For solar flare particles (largely protons), a simple formula has been developed:

$$D_X(\text{rem}) = 5 \times 10^{-7} X^{-1.5} \int \phi > 30 \text{ Mev}$$

where X is the shield thickness (gm/cm^2) and $\int \phi > 30$ Mev is the proton flux above 30 Mev ($protons/cm^2$). Based upon this relationship, biological doses expected in lunar orbit are one-half the free space numbers to account for the approximate 2π shielding of the moon in lunar orbits of 60 to 100 n mi.

$$D_X = \frac{2500}{X^{1.5}} \qquad \text{(1 "event")}$$

$$D_X = \frac{5000}{X^{1.5}} \qquad \text{(2 "events")}$$

$$D_X = \frac{7500}{X^{1.5}} \qquad \text{(3 "events")}$$

The allowable annual doses for the three critical organs (skin, blood-forming organs, and eyes) are 240 rem, 40 rem (80 rem if no exposure for the following year), and 120 rem, respectively. The effective depths for these critical organs are: skin ($\simeq$ 0 cm), blood-forming organs (bfo) (5 cm), and eyes (0.3 cm). Thus, the equations for shield thickness become:

$$D_{skin} = \frac{A}{X^{1.5}}$$

$$D_{bfo} = \frac{A}{(X+5)^{1.5}}$$

$$D_{eyes} = \frac{A}{(X+0.3)^{1.5}}$$

where A is 2500, 5000, or 7500 for one, two, and three "events", respectively. To determine the shielding requirements on board the OLS it will be assumed that 10 rem are absorbed by the crew from the engines of the Reusable Nuclear Stage cislunar shuttle during translunar and transearth flight, and 5 rem are absorbed by the crew from high energy galactic radiation. The previous equations are plotted in Figures 3.6-40, 3.6-41, and 3.6-42 for one, two and three solar flare events, respectively. The shield thickness requirements obtained from these graphs are listed in Table 3.6-6.

Table 3.6-6. Calculated Shield Requirements in Lunar Orbit

Organ → (Annual Allowable Dose) / Solar Flare Events	Skin (240-15) rem	Blood-Forming Organs (40-15) rem	Blood-Forming Organs (80-15) rem	Eyes (120-15) rem
1	4.98 gm/cm^2	16.6 gm/cm^2	6.4 gm/cm^2	8.0 gm/cm^2
2	7.9 gm/cm^2	29.2 gm/cm^2	13.1 gm/cm^2	12.8 gm/cm^2
3	10.35 gm/cm^2	40.0 gm/cm^2	18.7 gm/cm^2	16.96 gm/cm^2

The blood-forming organ dose limit determines the shield thickness, unless the optional dose limit of 80 rem/year is used; in which case, the eye dose limit determines the shield thickness. Table 3.6-5 together with Table 3.6-6 can be used to select the shielding requirements for various mission durations and desired confidence levels (P_c). It should be noted that 16.6 gm/cm^2 of shielding would be required to provide the crew with protection from one solar flare event in a 234-day mission to a 2-sigma basis.

If it is desired to keep the shielding weight under 10 gm/cm^2, there is the option of providing 6.4 gm/cm^2 of shielding and eye goggles for the crew and immediately returning the crew to earth if they have been exposed to one solar flare event before the completion of a 234-day mission. Upon return to earth, the crew would have to be grounded for one year to limit the blood forming organs dose to 80 rem in two successive years. Operational considerations must be made to completely define this option. Figure 3.6-43 plots the probability of witnessing no NASA solar flare events as a function of time. Assuming that the OLS crew is exposed to one solar flare event at time T=0, it is seen that the crew must be returned to earth within five weeks before the probability of not seeing another solar flare event drops below 95 percent. Five weeks should be sufficient time to prepare a cislunar shuttle for an emergency OLS crew transfer to earth. If not, there is always a fully fueled tug docked to the OLS which can deliver the eight crewmen to a safe earth orbit (see Section 7.0 of Volume II).

Storm Shelter Concept

A shield thickness requirement for a storm shelter concept of 16.6 gm/cm^2 is large by spacecraft standards. Typical spacecraft have

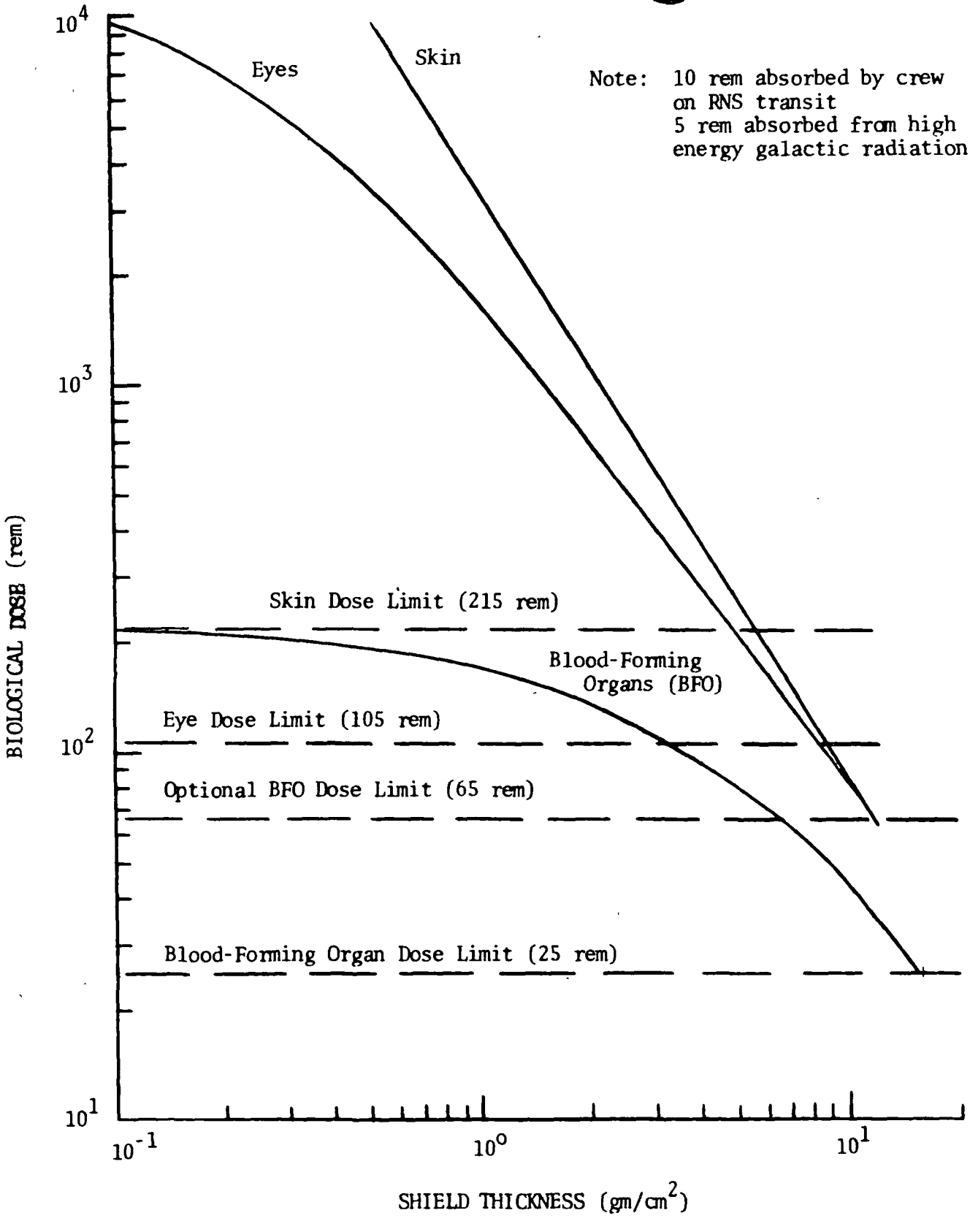


Figure 3.6-40. Critical Organs Doses for one NASA solar Flare Event in Lunar Orbit

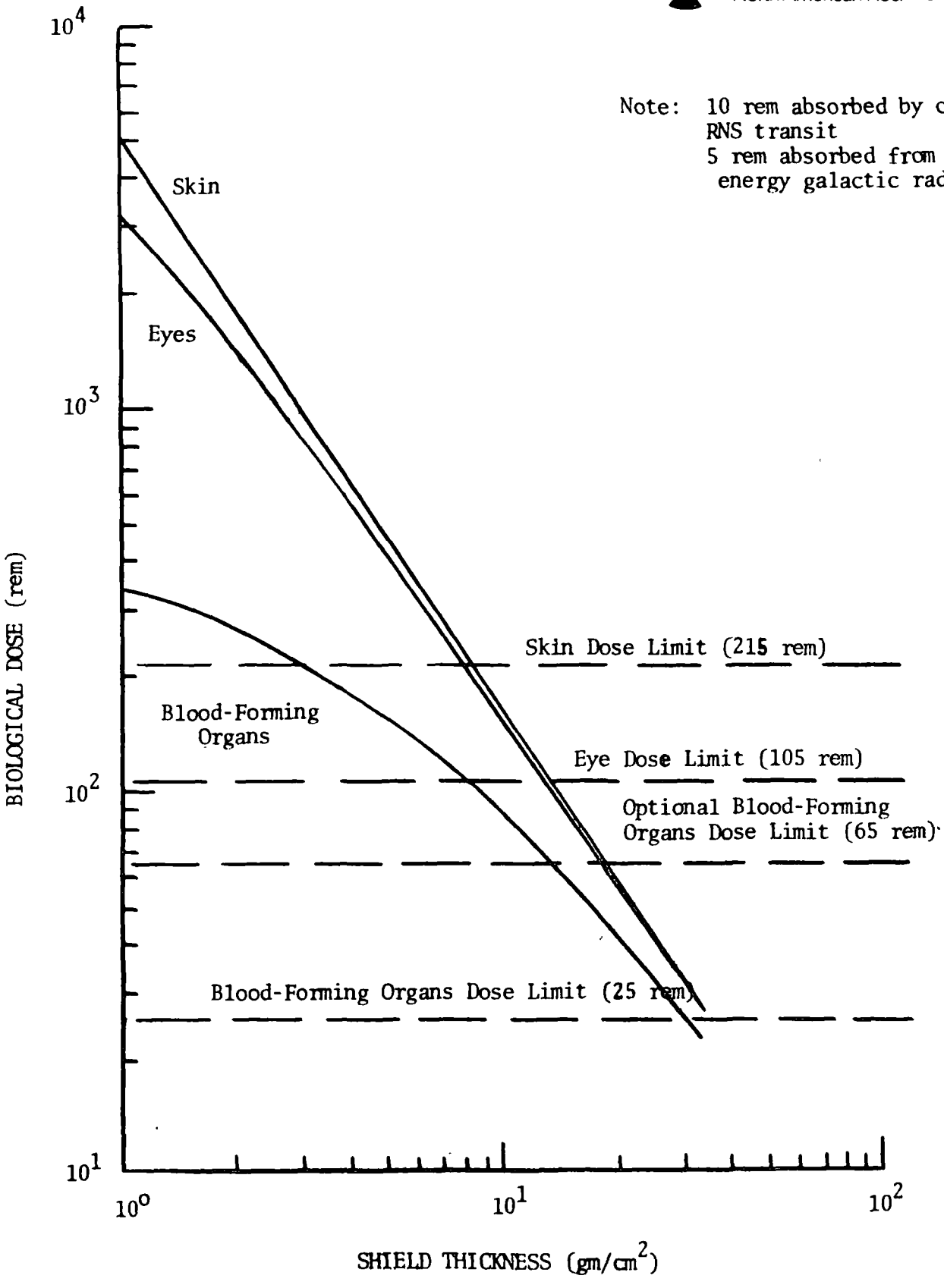


Figure 3.6-41. Critical Organ Dose for two NASA Solar Flare Events in Lunar Orbit

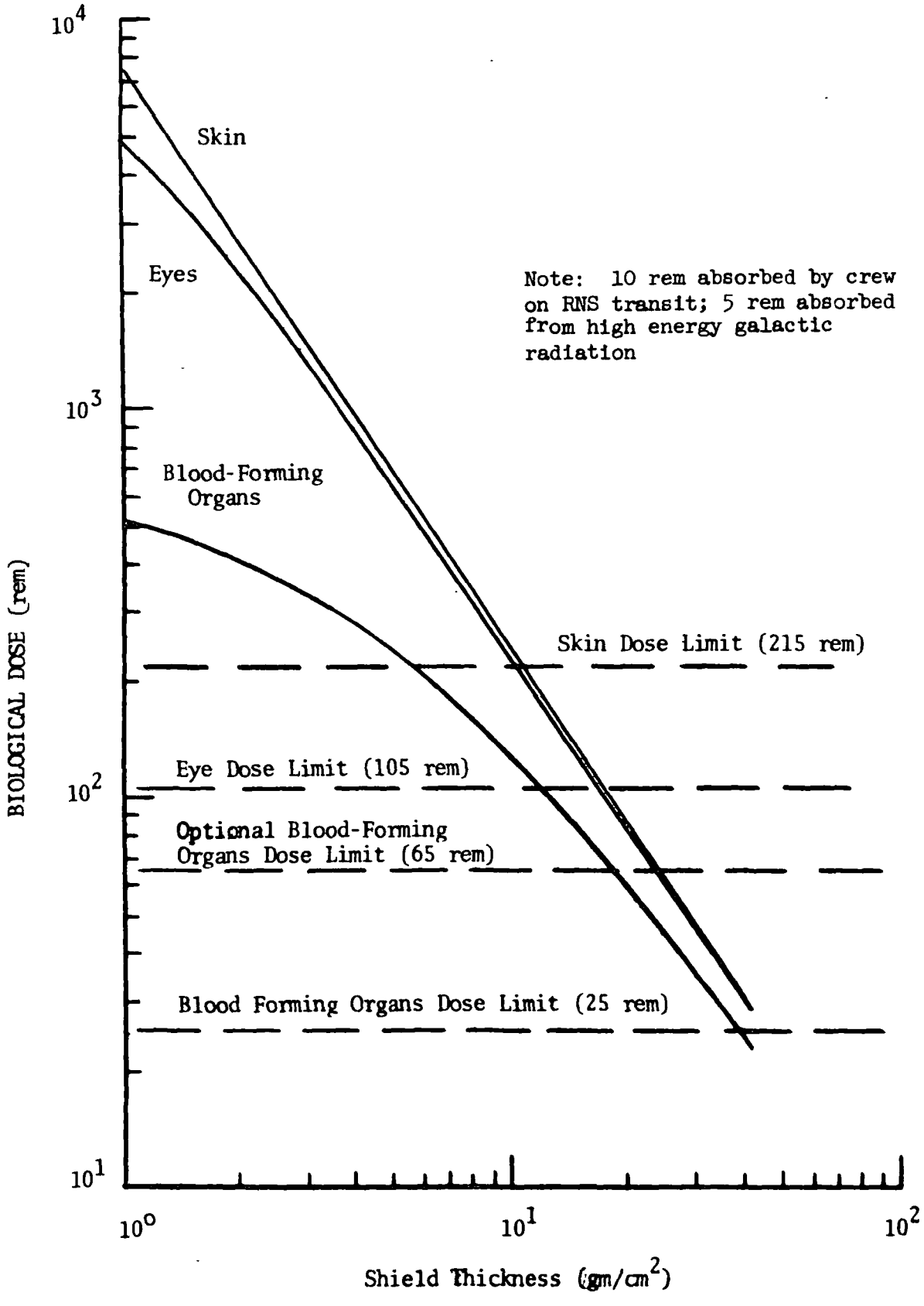


Figure 3.6-42. Critical Organ Doses for Three NASA Solar Flare Events in Lunar Orbit

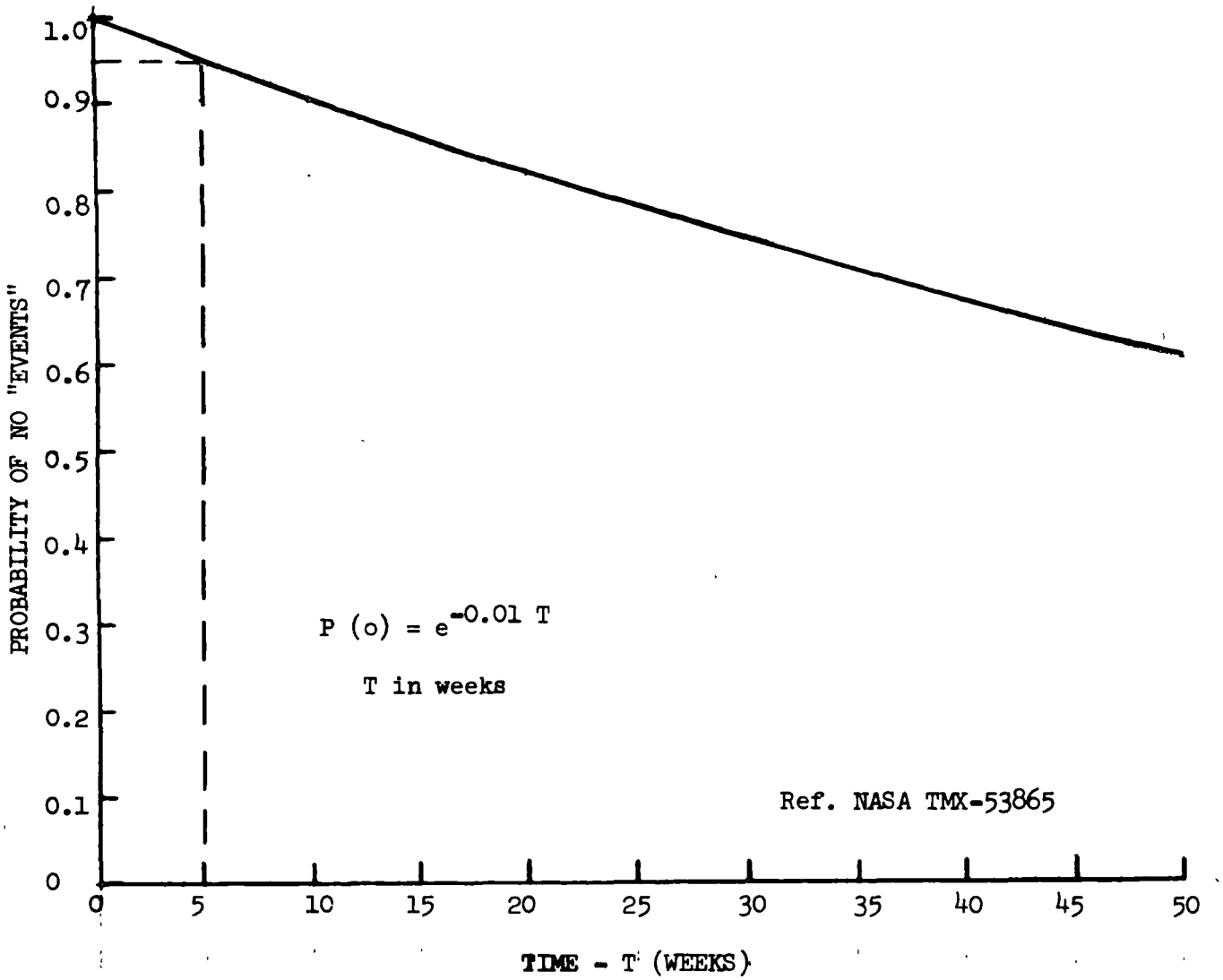


Figure 3.6-43. Probability of No Model Solar Flare "Events" With Time

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effective thicknesses of 2 to 5 gm/cm^2 for the radiation protection of their crews. However, the situation is not as bad as it may appear. The radiation that requires the shielding (solar flare particles) is intermittent in nature, and lasts for $\leq$ 3 days for each event (only a few large-dangerous-events are expected each year). Consequently, the storm shelter concept can be used. For periods $\leq$ 3 days, the crew of the OLS could be confined in a relatively small space, except for short ($<$ 1 hour per day) excursions to eat, attend to personal hygiene, etc., if it is desirable to minimize the volume of the storm shelter. The men could possibly spend much of their time sleeping, perhaps with the aid of sleep-inducing drugs. Under these conditions, a man requires only 50 to 100 ft^3 of living space. Such a storm shelter would consist of individual cylinders 6 feet long by 3 to 4 feet in diameter, stacked like cordwood. This storm shelter concept was investigated and upon evaluation rejected as a viable option for the following reasons:

1. The volume occupied by the storm shelter would be totally useless during normal operations.

2. There would be no access to a command/control console within a shielded region of the OLS during the solar flare activity. Consequently, even the minimum required station operations would involve crew excursions into a radiation environment.

3. Lack of crew access to hygiene or food preparation facilities would also require excursions into a radiation environment.

4. The use of sleep-inducing drugs for up to three days is not desirable from a crew habitability standpoint.

A storm shelter concept which avoids the above problems would have the following characteristics:

(1) A command/control console, hygiene facility and food preparation facility within the shielded volume to allow the minimum required crew operations during solar flare activity;

(2) Sleeping harnesses which could be suspended from any suitable location within the storm shelter;

(3) A water shield approximately 5.8 inches thick to provide 14.6 gm/cm^2 of shield weight (the OLS structure has an inherent shielding of about 2 gm/cm^2); the water could also serve as a thermal heat sink for the thermal protection concept described in Section 3.6.1;

(4) Judicious arrangement of available on-board storage (such as food supplies) to incorporate into the storm shelter shielding.

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