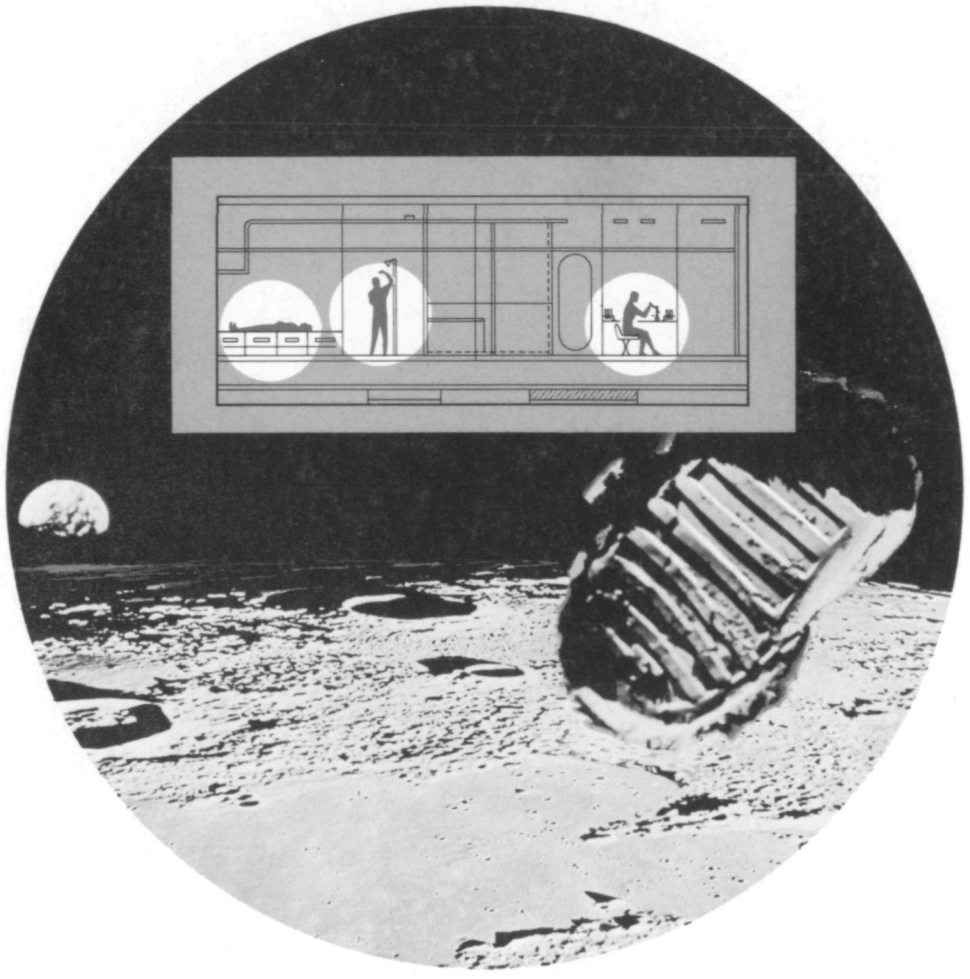
N71-25836
NASA CR-115019

Contract NAS9-10924
DRL Line Item 6

MSC-02687
SD 71-207
VOL V



ORBITING LUNAR STATION (OLS)

PHASE A FEASIBILITY AND DEFINITION STUDY

VOLUME V

OLS CONFIGURATION DEFINITION

FINAL REPORT

CASE FILE COPY

Space Division
North American Rockwell

APRIL 1971
Prepared by Advanced Program Engineering

12214 Lakewood Boulevard, Downey, California 90241

SD 71-207
VOLUME V

ORBITING LUNAR STATION PHASE A FEASIBILITY AND DEFINITION STUDY VOLUME V

OLS CONFIGURATION DEFINITION

(FINAL REPORT)

APRIL 1971

Approved by

L. R. Hogan
Program Manager
Orbiting Lunar Station Study

SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION

Page intentionally left blank

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 or 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 V of the report and contains a definition of the Representative and Derivative OLS configurations and subsystems. Also included is an Artificial G configurational concept that includes the Representative OLS, a suitable counterweight, a non-rotation zero g hub and a multiple docking adaptor (MDA).

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 Configurations

Page intentionally left blank

ACKNOWLEDGMENTS

The following persons have participated in OLS configuration and system analyses and have contributed to the preparation of this report:

L. R. Hogan	Program Manager
J. J. Armstead	Project Engineer, OLS Configuration/Systems Analyses
W. L. Steinwachs	Project Engineer, OLS Operations Analysis
F. H. Daurio	Systems Analysis
J. Ambrose	Electrical Power
J. O. Bartlett	Propulsion
M. Feiner	Thermal Protection
B. Liley	Nuclear Radiation Protection
M. F. Madden	Guidance and Control
C. K. McBaine	Weights Analysis
T. H. Moore	Information Management
S. J. Nalbandian	Electrical Power
T. W. Outten	Thermal Protection
D. W. Peebles	Configuration Analysis
A. J. Richardson	Meteoroid Protection
R. D. Trust	Environmental Control/Life Support
R. G. Vasta	Information Management
M. Vocka	Configuration Design
G. S. Canetti	Safety Analysis
J. R. Giglio	Reliability Analysis

Page intentionally left blank

CONTENTS

Section		Page
1.0	INTRODUCTION AND SUMMARY	1-1
	1.1 REPRESENTATIVE OLS CONFIGURATION DEFINITION .	1-1
	1.2 REPRESENTATIVE OLS SUBSYSTEM DEFINITION . .	1-3
	1.3 DERIVATIVE OLS DEFINITION	1-4
2.0	REPRESENTATIVE OLS CONFIGURATION SUMMARY . .	2-1
	2.1 CONCEPT DEFINITION	2-2
	Representative OLS Core Module . . .	2-2
	Science Experiment Accommodation . . .	2-22
	Docking Adapter	2-33
	2.2 OLS MASS PROPERTIES	2-34
	Representative OLS Core Module Dry Weight .	2-34
	Docking Structure	2-43
	Induced Environment Protection . . .	2-43
	Other Subsystems.	2-44
	OLS Power Module Dry Weight	2-44
	Experiment Module Dry Weight. . . .	2-45
	Group Weight Statement.	2-46
	OLS Weight Statement	2-46
	OLS Mass Moments of Inertia	2-46
	2.3 ARTIFICIAL GRAVITY CONCEPT	2-54
	2.4 DELIVERY TO LUNAR ORBIT	2-57
3.0	REPRESENTATIVE OLS SUBSYSTEM DEFINITION . .	3-1
	3.1 ENVIRONMENTAL CONTROL/LIFE SUPPORT SUBSYSTEM .	3-3
	3.1.1 Concept Definition	3-3
	3.1.2 Selection Rationale	3-9
	3.1.3 Impact of Artificial Gravity . .	3-31
	3.2 ELECTRICAL POWER SUBSYSTEM	3-35
	3.2.1 Concept Definition	3-38
	3.2.2 Selection Rationale	3-61
	3.2.3 Adaptability to Artificial Gravity .	3-72
	3.3 INFORMATION MANAGEMENT.	3-77
	3.3.1 Concept Definition	3-77
	3.3.2 Selection Rationale	3-93
	3.3.3 Adaptability to Artificial Gravity .	3-98
	3.4 GUIDANCE AND CONTROL SUBSYSTEM . . .	3-99
	3.4.1 Concept Definition	3-99
	3.4.2 Selection Rationale	3-123
	3.4.3 Adaptability to Artificial Gravity .	3-143

Section			Page
	3.5	REACTION CONTROL SUBSYSTEM	3-145
		3.5.1 Concept Definition	3-145
		3.5.2 Selection Rationale	3-146
		3.5.3 Adaptability to Artificial Gravity	3-158
	3.6	ENVIRONMENTAL PROTECTION SUBSYSTEM	3-161
		3.6.1 Meteoroid Protection	3-161
		Concept Definition	3-161
		Selection Rationale	3-163
		Adaptability to Artificial Gravity	3-163
		3.6.2 Thermal Protection	3-163
		Concept Definition	3-180
		Adaptability to Artificial Gravity	3-187
		3.6.3 Radiation Protection	3-187
		Concept Definition	3-187
		Selection Rationale	3-189
		Adaptability to Artificial Gravity	3-193
4.0	FUNCTIONAL REQUIREMENTS ANALYSIS		4-1
	4.1	FLOOR AREA REQUIREMENTS.	4-2
	4.2	CORE MODULE DOCKING PORT REQUIREMENTS	4-2
	4.3	RESUPPLY AND STORAGE REQUIREMENTS	4-4
	4.4	SCIENCE SUPPORT.	4-7
5.0	PERFORMANCE REQUIREMENTS COMPARISON		5-1
	5.1	STRUCTURES	5-3
		5.1.1 Orbital Maneuvering Considerations	5-3
		5.1.2 Lunar Orbit Delivery Considerations	5-3
		5.1.3 Configuration - General	5-13
		5.1.4 Core Module 1A and 1B	5-13
		5.1.5 Power Module	5-13
		5.1.6 Crew Quarters Modules	5-13
		5.1.7 Galley Module	5-13
		5.1.8 Control Center Module 1	5-19
		5.1.9 Control Center Module 2	5-19
		5.1.10 Cryogenic Storage Modules	5-19
		5.1.11 Experiment Module	5-19
		5.1.12 Dual-Support Cargo Module	5-19
	5.2	ENVIRONMENTAL CONTROL AND LIFE SUPPORT SUBSYSTEM	5-20
		5.2.1 O_2/N_2 Leakage	5-20
		5.2.2 CO_2 Partial Pressure	5-20
		5.2.3 Heat Rejection Loads	5-20
		5.2.4 Hygiene Flushing	5-20
		5.2.5 Housekeeping and Personal Hygiene	5-20

Section Page

5.2.6 Housekeeping Storage 5-20
5.2.7 Other Differences 5-20
5.3 ELECTRICAL POWER SUBSYSTEM 5-25
5.3.1 Flight Envelope 5-25
5.3.2 Crew Size 5-25
5.3.3 Major Power Modes 5-25
5.3.4 Power Source Applications . . 5-25
5.4 INFORMATION SUBSYSTEM 5-28
5.5 GUIDANCE AND CONTROL SUBSYSTEM . . . 5-28
5.5.1 Routine Operations 5-28
5.5.2 Location of Hardware 5-28
5.6 REACTION CONTROL SUBSYSTEM 5-28
5.7 ENVIRONMENTAL PROTECTION SUBSYSTEM . . 5-36
5.7.1 Meteoroid Protection 5-36
5.7.2 Thermal Protection 5-36
5.7.3 Radiation Protection 5-36

6.0 MSS CONVERSION MODIFICATIONS 6-1
6.1 STRUCTURES 6-3
6.1.1 Required Changes 6-3
6.1.2 Desired Changes 6-5
6.2 ENVIRONMENTAL CONTROL/LIFE SUPPORT SUBSYSTEM 6-7
6.2.1 Required Changes 6-7
6.2.2 Desired Changes 6-8
6.3 ELECTRICAL POWER SUBSYSTEM 6-9
6.3.1 Required Changes 6-9
6.3.2 Desired Changes 6-9
6.4 INFORMATION SUBSYSTEM 6-11
6.4.1 Required Changes 6-11
6.4.2 Desired Changes 6-13
6.5 GUIDANCE AND CONTROL SUBSYSTEM . . . 6-15
6.5.1 Determination of Flight Mode . . 6-15
6.5.2 Selected Configuration . . . 6-23
6.5.3 Required Changes 6-23
6.5.4 Desired Changes 6-25
6.6 REACTION CONTROL SUBSYSTEM 6-27
6.6.1 Required Changes 6-28
6.6.2 Desired Changes 6-30
6.7 ENVIRONEMTAL PROTECTION SUBSYSTEM . . 6-33
6.7.1 Meteoroid Protection 6-33
Required Changes 6-33
Desired Changes 6-33
6.7.2 Thermal Protection 6-33
Required Changes 6-37
Desired Changes 6-40
6.7.3 Radiation Protection 6-40
Desired Changes 6-41

Section			Page
7.0	DERIVATIVE OLS CONFIGURATION		7-1
	7.1	STRUCTURES	7-5
		Core Modules	7-5
		Operational and Living Quarters Modules	7-13
		CQM1 (Crew Quarters Module 1)	7-13
		CQM3 (Crew Quarters Module 3)	7-14
		CCM1 (Control Center Module 1)	7-14
		CCM2 (Control Center Module 2)	
		GM (Galley Module)	7-14
		CSM2 (Cryogenic Storage Module 2)	7-14
		CSM1 (Cryogenic Storage Module 1)	7-15
		PM (Power Module)	7-15
		Dual Support Cargo Module (DSCM)	7-15
		Experiment Module	7-16
		Weight Statement	7-16
		Mass Properties	7-32
	7.2	ENVIRONMENTAL CONTROL/LIFE SUPPORT SUBSYSTEM	7-35
		7.2.1 Required Configuration	7-35
		7.2.2 Desired Configuration	7-38
	7.3	ELECTRICAL POWER SUBSYSTEM	7-39
		7.3.1 Derivative OLS Concept	7-39
		7.3.2 Desired Configuration	7-39
	7.4	INFORMATION SUBSYSTEM	7-47
		7.4.1 Identification of Functions/ Assemblies	7-47
		7.4.2 Major Functional Requirements	7-47
		7.4.3 ISS Characteristics	7-50
	7.5	GUIDANCE AND CONTROL SUBSYSTEM.	7-67
	7.6	REACTION CONTROL SUBSYSTEM	7-69
		7.6.1 Derivative OLS Design With Required Modifications	7-69
		7.6.2 Derivative OLS Design With Desired Modifications	7-72
	7.7	ENVIRONMENTAL PROTECTION SUBSYSTEM	7-75
		7.7.1 Meteoroid Protection	7-75
		7.7.2 Thermal Protection	7-77
		7.7.3 Radiation Protection	7-78

Section		Page
8.0	RECOMMENDED MODULAR SPACE STATION DESIGN MODIFICATIONS	8-1
	8.1 STRUCTURES.	8-1
	8.2 ENVIRONMENTAL CONTROL/LIFE SUPPORT SUBSYSTEM	8-1
	8.3 ELECTRICAL POWER SUBSYSTEM . . .	8-1
	8.4 INFORMATION SUBSYSTEM	8-2
	8.5 GUIDANCE AND CONTROL SUBSYSTEM . .	8-2
	8.6 REACTION CONTROL SUBSYSTEM . . .	8-2
	8.7 ENVIRONMENTAL PROTECTION SUBSYSTEM. .	8-2
	8.7.1 Meteoroid Protection . . .	8-2
	8.7.2 Thermal Protection	8-3
	8.7.3 Radiation Protection . . .	8-3

Page intentionally left blank

ILLUSTRATIONS

Drawings		Page
2282-25	Representative 27-Foot Diameter OLS	2-3
2282-27	Representative OLS Flight Configuration	2-17
2282-26	Experiment Module	2-29
2282-28	OLS Artificial G Configuration	2-55
9953-1011	MSS "Preferred" Module Concept Arrangement	5-15
2282-29	Derivative OLS	7-7

Figures		
2-1	Integrated Experiment Concept (Bottom Viewing)	2-23
2-2	End Docked Experiment Module (Side Viewing)	2-24
2-3	Bottom Docked Experiment Module (End Viewing)	2-25
2-4	Representative OLS Docked - Configuration 1	2-50
2-5	Representative OLS Docked - Configuration 2A	2-51
2-6	Representative OLS Docked - Configuration 2B, C & D	2-52
2-7	Representative OLS Docked - Configuration 3A and B	2-52
2-8	One Step Delivery	2-58
2-9	Two Step Delivery	2-59
3.1-1	Active Thermal Control - Fixed Weight	3-19
3.1-2	Active Thermal Control	3-20
3.1-3	Single Vs. Dual System for Water Reclamation	3-23
3.1-4	Pressure Decay With No N_2 Makeup	3-30
3.2-1	Major EPS Functions.	3-35
3.2-2	Zero-G Mode	3-37
3.2-3	Typical OLS Mission Power Profile (8-Man)	3-40
3.2-4	Typical Orbit Load Profile for OLS Profile (8-Man)	3-41
3.2-5	EPS (Solar Array/RFC) One-Line Schematic	3-42
3.2-6	EPS (Solar Array/RFC) Functional Block Diagram	3-44
3.2-7	EPS Assemblies	3-46
3.2-8	EPS Installation	3-45
3.2-9	Baseline Rollup Solar Array	3-49
3.2-10	Single-Point Grounding System	3-53
3.2-11	Weight Comparison of Energy Storage Candidate	3-68
3.2-12	Summary of EPS Trades	3-74
3.2-13	Artificial G Concept	3-75
3.3-1	ISS Functional Assemblages	3-78
3.3-2	ISS Subsystem Interfaces	3-79
3.3-3	ISS Assemblies	3-80
3.3-4	Information Subsystem Block Diagram	3-81
3.3-5	Mass Memory Configuration.	3-84
3.3-6	Remote Acquisition and Control Unit	3-85
3.3-7	Command and Control Console	3-87
3.3-8	External Communications and Tracking Subassembly.	3-89
3.3-9	Internal Communications Subassembly	3-92
3.3-10	High-Gain Antenna Comparison	3-97

Figures

		Page
3.4-1	Guidance and Control Subassemblies	3-100
3.4-2	G&C Functional Block Diagram	3-101
3.4-3	Navigation Mode.	3-103
3.4-4	OLS Attitude Control Concept	3-104
3.4-5	OLS Translation Control Concept	3-105
3.4-6	Inertial Reference Assembly.	3-107
3.4-7	Optical Navigation/Attitude Reference Assembly	3-109
3.4-8	Control Moment Gyro Assembly	3-116
3.4-9	RCS Electronics Assembly	3-117
3.4-10	Guidance and Control Signal Processing	3-120
3.4-11	Local-Level Flight Modes, Configuration 1	3-124
3.4-12	Vehicle Configurations.	3-130
3.4-13	RCS Only Versus RCS + CMG's Trade.	3-142
3.5-1	RCS Schematic	3-147
3.5-2	Subsatellite Propellant Provisions	3-148
3.5-3	RCS Loaded Mass Comparison	3-149
3.5-4	RCS Propellant Volume Comparison	3-150
3.6-1	Representative OLS Reference Configuration	3-162
3.6-2	Planetary Thermal Environments	3-165
3.6-3	Lunar Thermal Environment in a Sunsolar Orbit.	3-167
3.6-4	Heat Rejection at Subsolar Point	3-168
3.6-5	OLS External Radiator Location	3-169
3.6-6	OLS Thermal Control System Alternatives.	3-171
3.6-7	Heat Rejection Vs. Radiation Sink Temperature.	3-174
3.6-8a	Comparison of Heat Rejection System Performance Characteristics - Lunar Orbit	3-175
3.6-8b	Comparison of Heat Rejection System Performance Characteristics - Lunar Orbit	3-176
3.6-8c	Comparison of Heat Rejection System Performance Characteristics - Lunar Orbit	3-177
3.6-9	Required Radiator Area	3-179
3.6-10	Capacitor Weight for Active Concepts	3-181
3.6-11	Capacitor Weight Versus Radiator Area	3-182
3.6-12	Required Boom Radiator as a Function of α_s/ϵ	3-185
3.6-13	Required Length of Boom Radiator as a Function of α_s/ϵ	3-186
3.6-14	Model Solar Flare "Event" Probability as a Function of OLS Program Duration	3-188
3.6-15	Probability of No Model Solar Flare "Events" with Time	3-190
3.6-16	OLS Radiation Shelter Configuration	3-191
3.6-17	Side View of OLS Radiation Shelter	3-192
5-1	Derivative OLS Configuration - Module Locations	5-9
5-2	Disassembled Delivery Concept	5-10
5-3	Partially Assembled Delivery Concept	5-11

Figures

		Page
6.4-1	Antenna Subsystem Weight Versus Antenna Diameter . .	6-14
6.5-1	Vehicle Configurations	6-16
6.5-2	Momentum History for Configuration 2	6-21
6.5-3	Effect of Principal Axis Misalignment.	6-22
6.7-1	Derivative OLS Heat Rejection from an Unobstructed Module with 293 ft^2 Radiator Area, 70 F Radiator . . .	6-36
6.7-2	Module Rejection Capability as a Function of Location .	6-38
6.7-3	Derivative OLS Total Rejection Capability . . .	6-39
7.3-1	Derivative OLS EPS (Solar Array/Battery) On-Line Schematic	7-40
7.4-1	Information Subsystem Functions.	7-48
7.4-2	ISS Equipment Complement	7-61
7.4-3	Mechanization Diagram, Data Processing Assembly . .	7-62
7.4-4	Communications Assembly Block Diagram. . . .	7-65

Page intentionally left blank

TABLES

Table		Page
2-1	Science Experiment Location	2-26
2-2	Detail Weight Statement, OLS - 27' Dia; 8-Man	2-35
2-3	Group Weight Statement, OLS (8 Crewmen; 27 ft. dia.).	2-47
2-4	Weight Summary - OLS (Maximum in Lunar Orbit)	2-48
2-5	Docked Configurations	2-49
2-6	Mass Properties - 27 Ft. Dia., 8-Man Representative OLS (Less Solar Array) 180-Day On-Board OLS Supply	2-53
3.1-1	Selected ECLSS Concepts	3-4
3.1-2	ECLSS Cryogenic Gas Requirements (Stored by RCS).	3-5
3.1-3	ECLSS High Pressure Gas	3-8
3.1-4	Selection Rationale (Priority Order).	3-9
3.1-5	Oxygen and Nitrogen Densities	3-10
3.1-6	Gas Requirements	3-10
3.1-7	Gases Subject to Pumpdown	3-12
3.1-8	Pumpdown System Weight and Power Requirement.	3-12
3.1-9	Pumpdown System Trades	3-12
3.1-10	CO_2 Removal Concept Comparison, 8 Men, 10 Years, 5 mm Hg $PPCO_2$	3-14
3.1-11	Sunside Water Electrolysis Comparison, 8 Men-10 Years.	3-15
3.1-12	CO_2 Reduction Concept Comparison, 8 Men - 10 Years (18 lb/Day CO_2)	3-16
3.1-13	Typical Flow Rates (lb/hr)	3-17
3.1-14	Trace Gas Control Comparison, 8 Men - 10 Years	3-18
3.1-15	Water Model	3-21
3.1-16	Distillation Process Evaluation, 8 Men - 10 Years	3-22
3.1-17	Single Versus Dual Water System Characteristics, 8 Men - 109-Day Resupply	3-23
3.1-18	Potable Water Accumulator Sizing	3-24
3.1-19	Water Processing Concepts	3-26
3.1-20	Waste Management OLS Waste Model	3-27
3.1-21	Laundry Characteristics	3-28
3.1-22	Food Management Hardware	3-29
3.1-23	ECLSS Weight-Power-Volume Summary	3-32
3.2-1	OLS Electrical Load Analysis (8-Man Crew)	3-39
3.2-2	EPS Functional Descriptions Summary	3-47
3.2-3	Summary of Array Design Parameters	3-50
3.2-4	Array Weight Breakdown	3-50
3.2-5	Electrical Power Subsystem Major Interface Require Requirements.	3-57
3.2-6	Solar Powered (RFC) EPS Weight Summary (8-Man)	3-58
3.2-7	EPS Volume Summary	3-59
3.2-8	EPS Heat Rejection Requirements	3-60
3.2-9	Primary Electrical Power Sources	3-62
3.2-10	Operational Considerations	3-63

Table		Page
3.2-11	Radiation Safety Issues	3-64
3.2-12	EPS Candiate Weight Comparison Matrix	3-66
3.2-13	EPS Candidate Cost Comparison Matrix	3-67
3.2-14	Emergency/Auxiliary Power Source Requirements	3-69
3.2-15	AC Versus DC Power Distribution	3-73
3.3-1	ISS Equipment List	3-94
3.4-1	G&C Physical Characteristics	3-122
3.4-2	Docked Configuration	3-125
3.4-3	Results of Stability and Control Analysis of Vehicle A	3-126
3.4-4	Results of Stability and Control Analysis of Vehicle B	3-127
3.4-5	Results of Stability and Control Analysis of Vehicle C	3-128
3.4-6	Results of Stability and Control Analysis of Vehicle D	3-129
3.4-7	Results of Stability and Control Analysis of Vehicle 1_A	3-131
3.4-8	Results of Stability and Control Analysis of Vehicle 1_C	3-132
3.4-9	Results of Stability and Control Analysis of Vehicle 1_D	3-133
3.4-10	Results of Stability and Control Analysis of Vehicle 2_A	3-134
3.4-11	Results of Stability and Control Analysis of Vehicle 2_C	3-135
3.4-12	Results of Stability and Control Analysis of Vehicle 2_D	3-136
3.4-13	Results of Stability and Control Analysis of Vehicle 3_A	3-138
3.4-14	Results of Stability and Control Analysis of Vehicle 3_C	3-139
3.4-15	Results of Stability and Control Analysis of Vehicle 3_D	3-140
3.4-16	CMG Weight and Propellant Consumption Comparison.	3-141
3.5-1	Propulsion Requirements (180 Days)	3-145
3.5-2	OLS RCS Options	3-148
3.5-3	Subsystem Integration (180 Days).	3-152
3.5-4	Propellant Summary (lbm).	3-152
3.5-5	Cryogenic Fluid Tankage	3-153
3.5-6	RCS Mass Summary.	3-154
3.5-7	RCS Volume and Power Summary.	3-156
3.5-8	Propellants Required on Day 79 of a 109-Day Resupply Period	3-158
3.6-1	Heat Rejection Requirements	3-164
3.6-2	Concept Characteristics Summary	3-183
3.6-3	OLS Shielding Requirements	3-189
4-1	Floor Area Requirements Comparison	4-3
4-2	Docking Provisions	4-5
4-3	Storage Requirements Comparison	4-6

Table		Page
4-4	Consumables Resupply Comparison	4-8
4-5	Experiment Imposed Station Requirements	4-10
5-1	EOSS-MSS Vs. OLS-MSS Performance Requirements Comparison Matrix Structures	5-4
5-2	EOSS-MSS Vs. OLS-MSS Module Characteristics Comparison	5-6
5-3	EOSS-MSS Vs. OLS-MSS Performance Requirements Comparison Matrix ECLSS	5-21
5-4	EOSS-MSS Vs. OLS-MSS Performance Requirements Comparison Matrix EPS	5-26
5-5	EOSS-MSS Vs. OLS-MSS Performance Requirements Comparison Matrix ISS	5-29
5-6	EOSS-MSS Vs. OLS-MSS Performance Requirements Comparison Matrix G&C	5-31
5-7	EOSS-MSS Vs. OLS-MSS Performance Requirements Comparison Matrix RCS	5-33
5-8	EOSS-MSS Vs. OLS-MSS Performance Requirements Comparison Matrix ENPS	5-37
6.4-1	OLS Antenna Weight Characteristics	6-13
6.5-1	Weights of OLS/MSS Modules	6-15
6.5-2	Results of Stability and Control Analysis of Vehicle 1	6-17
6.5-3	Results of Stability and Control Analysis of Vehicle 2	6-18
6.5-4	Results of Stability and Control Analysis of Vehicle 3	6-19
6.5-5	Results of Stability and Control Analysis of Vehicle 4 by Mission Phase	6-24
6.5-6	Results of Stability and Control Analysis of Vehicle 2 by Mission Phase	6-24
6.6-1	Propulsion Requirements Comparison	6-27
6.6-2	Subsystem Integration Comparison	6-28
6.6-3	Derivative OLS Propellant and Nitrogen Storage Summary (lbm)	6-29
6.6-4	Derivative OLS Cryogenic Fluid Tankage - Required Modification	6-30
6.6-5	Derivative OLS Cryogenic Fluid Tankage - Desired Modificiation	6-31
6.7-1	Thermal Control Subsystem Change Options.	6-34
6.7-2	Nonadditive Heat Loads	6-35
7.1-1	Derivative OLS Weight Summary	7-17
7.1-2	Comparison of OLS and MSS Modular Weights	7-18
7.1-3	Mass Properties and Locations of Derivative OLS Modules	7-33
7.1-4	Mass Properties - Derivative OLS (Configuration 4)	7-34
7.2-1	Derivative OLS Configuration - Required Changes - ECLSS Weight and Power	7-36
7.3-1	EPS Weight Characteristics	7-41
7.3-2	EPS Volume Characteristics	7-42
7.3-3	Minimum Change Concept EPS Weight Summary	7-43
7.3-4	Minimum Change Concept, EPS Volume Summary	7-44

Table		Page
7.3-5	Minimum and Extensive Change Desired EPS Cost Evaluation Summary	7-46
7.4-1	Communication Performance Characteristics	7-50
7.4-2	Data Processing Performance Characteristics	7-51
7.4-3	Displays and Controls Performance Characteristics	7-53
7.4-4	Component/Subassembly Equipment Requirements.	7-54
7.4-5	ISS Characteristics (Component/Subassembly Level)	7-55
7.4-6	ISS Characteristics (Module Level)	7-56
7.4-7	ISS Characteristics Summary	7-59
7.6-1	Derivative OLS RCS Mass Summary - Required Modifications	7-70
7.6-2	Derivative OLS RCS Volume and Power Summary - Required Modifications	7-71
7.6-3	Derivative OLS RCS Mass Summary - Desired Modifications	7-73
7.6-4	Derivative OLS RCS Volume and Power Summary - Desired Modifications	7-73
7.7-1	Derivative OLS Elements with Equivalent Meteoroid Protection	7-75
7.7-2	Derivative OLS Meteoroid Strike Data	7-76
7.7-3	Summary of Derivative OLS Meteoroid Protection Data	7-77

1.0 INTRODUCTION AND SUMMARY

Phase III of the Orbiting Lunar Station (OLS) Phase A Feasibility and Definition Study consisted of the derivation of two configurations, a representative OLS and an adaptation of an earth orbital Modular Space Station (MSS) that is identified as the derivative OLS. The performance requirements for the configurations were developed in Phase I of the study and are reported in Volume III, OLS Performance Requirements. The concept selection trade data was developed in Phase II of the study and is reported in Volume IV, Configuration and System Analysis. This report, Volume V, Configuration Definition, presents the synthesis of the two configurations.

1.1 REPRESENTATIVE OLS CONFIGURATION DEFINITION

In Section 2.0 the representative OLS configuration selection is presented. Section 2.1 details the design effort performed during the conceptual definition of the representative OLS with a description and layouts of the principal structural elements: core module, power boom, and experiment module. Section 2.2 presents the mass properties of the representative OLS, the weight statement, and mass moments of inertia. The adaptability of the selected representative OLS configuration to artificial gravity operations is discussed in Section 2.3.

A summary of the representative OLS configuration definition presented in Section 2.1 is given below.

1. The core module is 27 feet in diameter with an overall length of 60.83 feet. The internal arrangement consists of four transverse circular decks with toroidal end pressure bulkheads and two separate pressure volumes. Unpressurized volumes are provided in the upper and lower torus regions for cryogenic storage.

2. A four-element rollout solar array of 10,000 square feet is accommodated on a cylindrical power module. When mated at launch, the overall length of the power and core modules is 94.25 feet.

3. Four passive docking ports are located on the cylindrical portion of the core module plus two active neuter docking cones at each end. Passive side docking ports are employed to eliminate the need for large boost fairings. At anytime after boost to earth orbit these passive ports can be modified by the addition of active/active docking adaptors. Four of these are defined as necessary to satisfy OLS operational docking requirements.

4. An experiment module 15 feet in diameter and 22 feet long (EOS compatible) is docked at the +Z axis port of the experiments deck to physically accommodate those science experiments requiring an unrestricted field of view while body mounted to the OLS. The experiment module also incorporates a separate compartment which functions as an airlock and subsatellite servicing hangar.

A summary of the weight statement presented in Section 2.2 is presented below.

1. The total OLS dry weight is 107,475 pounds, consisting of an 85,155-pound core module, an 11,320-pound power module, an an 11,000-pound experiment module.

2. The total OLS inert weight is 129,375 pounds, including 21,900 pounds of inert fluids.

3. The total OLS maximum in lunar orbit weight is 157,625 pounds including 26,650 pounds of consumables and 1600 pounds for a crew of eight.

A conceptual design of the representative OLS, configured for artificial-gravity operation, is presented in Section 2.3. The basic configuration consists of:

1. The OLS with another OLS or optional counterweight rotating at a rate of 3.5 revolutions per minute. The range of artificial gravity in the four decks with a rotating diameter of 323.3 feet would be 0.55 g to 0.65 g.

2. A zero-g hub with the driving mechanism and docking ports to mate the normal power module, experiment module, tug/landers, and cargo module.

A concept for delivery of the representative OLS to lunar orbit is developed. Single and two stage Chemical Propulsion Stages (CPS-1; CPS-2) and a Reuseable Nuclear Shuttle (RNS) are considered as candidate Cislunar Shuttles (CLS). Also, operation of CPS-1 and the RNS in an "expended" mode (refueled in lunar orbit) is considered. The selected baseline delivery concept is to use the RNS as the CLS and to deliver the total assemblage of OLS modules, consumables, crew, and one space tug by means of two RNS flights.

1.2 REPRESENTATIVE OLS SUBSYSTEM DEFINITION

Section 3.0 contains a concept definition of each representative OLS subsystem. The trade studies and rationale for selection are summarized for each subsystem subassembly. Also, the adaptability of each subsystem of the OLS to artificial gravity operation is evaluated. The more significant assembly and subassembly selections of each OLS subsystem, which are detailed in Sections 3.1 through 3.6, are summarized below.

1. Environmental Control/Life Support Subsystem . A high pressure nitrogen system is included in the emergency supply assembly. The emergency earth based reaction time for the OLS is 30 days as compared to only 2 days for the Earth Orbit Space Station (EOSS). Resupply weights and costs were the primary factors in selecting a laundry for the OLS. All other ECLSS functions are accomplished in a manner similar to the EOSS concepts.

2. Electrical Power Subsystem. A 10,000 square foot solar array is selected as the primary power source. The major influences were weight and cost. The energy storage concept is regenerative fuel cells. Initial and resupply weights were the prime drivers in the selection of this concept. Pre-manning, emergency, and eclipse power is supplied by open-loop fuel cells. This approach provides a completely independent power source which also will produce potable water during emergency periods.

3. Information Subsystem. The only unique concept in the Information Subsystem as compared to the EOSS is the high gain antenna subassemblies. Phased arrays are preferred over parabolic dishes primarily because of the lesser maintenance required and the capability for simultaneous multiple communication links. An antenna subassembly is also included on the power module in order to eliminate communication dead zones that would occur if antennas were restricted solely to the OLS core module.

4. Guidance and Control. Attitude control is provided by the combination of the Reaction Control System and Control Moment Gyros (CMG). The CMG's are comparable to those of the EOSS. The navigation requirements of the OLS require the inclusion of a landmark tracker and a radar altimeter. A strapdown system was selected as the initial reference assembly primarily because of the reliability and, if necessary, ease of maintenance. Operational science experiment considerations require that the OLS be maintained in a local level orientation. The selected assembly of OLS modules was based upon CMG sizing, RCS impulse requirements, thermal constraints, and experiment sensor viewing requirements.

5. Reaction Control System . An H_2/O_2 propulsion system, with cryogenic storage, was selected primarily because of a smaller required mass as compared to the other candidate concepts and for commonality with the propellants of the space tugs.

6. Environmental Protection System

a. Radiation Protection - Based upon the NASA defined solar flare event model (TMX 53865), it is mandatory that a radiation shelter be included in the OLS. An area that includes the secondary control center, backup galley, and a toilet is enclosed primarily with a water jacket that contains 16,000 pounds of water.

b. Thermal Protection - The thermal environment in lunar orbit (in the sub-solar region) is significantly more severe than in earth orbit. An active external radiator system operating at an effective temperature of 70 F was selected. A heat pump is required and also a thermal capacitor (the water of the radiation shield) is used in order to achieve adequate heat rejection with the available effective radiator area and with reasonable α_s/ϵ values for the control coating.

c. Meteoroid Protection - The differences in the meteoroid environments between the OLS and the EOSS are negligible. Therefore, the same concepts are used. The governing factor in selecting the thickness of the micrometeoroid bumper was manufacturing, handling, and maintenance considerations rather than the meteoroid environment.

1.3 DERIVATIVE OLS DEFINITION

Sections 4.0, 5.0, and 6.0 constitute an analysis and definition of the modifications to an earth orbiting Modular Space Station (MSS) required to permit operation in lunar orbit as the derivative OLS. The MSS concept considered is the preferred baseline concept as documented in MSC 02464, Volume 1, the MSS Definition Document, and in PDS-71-1, the MSS 2nd Quarterly Review Briefing Booklet. Section 7.0 presents a composite definition of the derivative OLS with only mandatory changes to the MSS modules incorporated. Recommended changes to the baseline MSS which would either facilitate conversion to OLS use or enhance operations as an OLS without significantly penalizing the MSS are identified in Section 8.0.

1.3.1 Functional Requirements Analysis

Both the earth-orbiting modular space station (MSS) and the derivative OLS are made up of an assembly of 14-foot diameter modules capable of being transported to earth orbit in the cargo bay of the earth orbit shuttle (EOS). Both configurations utilize core modules (C1, etc.) to which are docked a power module (PM), control center modules (CCM's), crew quarters modules (CQM's), cargo storage modules (CM's), galley module (GM), and an experiment module (EM). The derivative OLS utilizes MSS modules modified to meet OLS requirements.

Section 4.0 presents a comparison between the derivative OLS and the MSS for four types of functional requirements, i.e., floor area, docking provisions, resupply and storage, and science support. The baseline MSS used for comparison purposes is the 6-man configuration. Although the primary purpose of this section is to present minimum requirements, the means by which MSS modules must be modified to meet the derivative OLS requirements are briefly discussed where appropriate.

Significant conclusions which can be drawn based on the functional requirements analysis are:

1. Modules must be added to enlarge the 6-man MSS concept to meet the increased OLS floor space requirements.

2. Two MSS core modules No. 1 (C1) docked end-to-end, rather than a C1 and a C2 as used on the 6-man MSS, are required to meet OLS docking provision requirements.

3. The increased cryogenic storage requirements of the OLS require reconfiguration of the MSS Cargo Module (CM).

4. The unique OLS science requirements will require replacement of the MSS Physics/Materials and Fluid/Electronic/Optics laboratories with a geochemistry laboratory.

1.3.2 Performance Requirements Comparison

Section 5.0 presents a performance requirements comparison (e.g., power, consumables, stability, data handling, impulse, etc.) between the OLS subsystems and the 6-man MSS subsystems. Significant results of this analysis are:

1. The OLS requires the larger 10,000 square foot solar arrays utilized by the 12-man growth version of the MSS.

2. OLS requirements are better satisfied through use of the 12-man MSS Galley Module (GM) and Crew Quarters Module (CQM3) rather than the 6-man MSS CQM2.

3. The ECLSS equipment of the 6-man MSS must be resized for the 8-man OLS.

4. A radiation storm shelter with hygiene facilities must be incorporated in the Control Center Module 1 (CCM1) which provides 16.6 gm/cm^2 of shielding for 8 crewman during solar flare events.

5. High pressure nitrogen and increased high pressure oxygen storage for emergency leakage make-up is required uniquely by the OLS.

6. OLS module heat rejection requirements exceed those of MSS modules which, coupled with the high lunar infrared radiation, requires additional radiator area and necessitates use of a thermal capacitor to store heat during subsolar positions of the lunar orbit.

7. More stringent OLS navigational accuracy requirements necessitate addition of a landmark tracker and a radar altimeter to the MSS G&C subsystem.

8. Larger OLS total impulse requirements require resizing of MSS Reaction Control Subsystem (RCS) accumulators and the cryogenic tank storage facility.

9. The side docking provisions on the core module must be strengthened to accommodate the bending loads that will be experienced during translunar and lunar orbit insertion thrust maneuvers.

The requirement for strengthening the primary structure of the core module was a result of development of a concept for delivery of the derivative OLS to lunar orbit, which is also presented in this section. The three candidate Cislunar Shuttles (CLS) considered were CPS-1, CPS-2, and RNS operated in round trip and expended modes. Both assembled and disassembled arrays of modules were considered in the delivery concepts. The selected baseline concept was to use two RNS flights operated in a round trip mode. The RNS payload capability permitted delivery of a partially assembled OLS on one flight, provided the primary structure of the core modules were strengthened. The primary limitation on use of the CPS as the CLS was the thrust level, which induced bending moments six times as great as the RNS. If the CPS could be throttled to a thrust level comparable to the RNS or lower, it would also be a candidate CLS for delivery of an assembled OLS.

1.3.3 MSS Conversion Modifications

Based on analysis of the requirements comparison data presented in Sections 4.0 and 5.0, the changes to the MSS concept which are absolutely required to allow its use in lunar orbit as the derivative OLS are defined in Section 6.0. Absolutely required changes are:

1. Reconfiguration of the MSS Cargo Modules (CM's) to conform to the OLS cryogenic storage requirements

2. Installation of a solar flare nuclear radiation shelter surrounding the secondary control center

3. Incorporation of high pressure nitrogen and oxygen storage tanks in the core modules for emergency leakage make-up plus oxygen metabolic consumption

4. Addition of a laundry

5. Installation of an additional high gain parabolic antenna subassembly boom mounted above the solar array to meet the continuous communications requirements.

6. Addition of a landmark tracker and a radar altimeter to meet OLS navigation requirements

7. Addition of RCS hydrogen accumulators due to the larger OLS impulse requirements

8. Addition of a heat pump, a thermal capacitor, and additional radiator area to satisfy OLS heat rejection requirements

9. Reinforcement of the side docking structure of the core module to withstand the bending moments during translunar and lunar orbit insertion thrust maneuvers

Some additional MSS modifications which would be desirable, but are not absolutely required, are also described in Section 6.0. Major desired changes are:

1. Remove cryogenic storage tanks from the power boom and replace the batteries and battery chargers with a regenerative fuel cell energy storage system

2. Reconfigure the cryogenic storage modules to house all OLS cryogens

3. Change from a double to a single ac voltage distribution concept

4. Remove excess RCS engine clusters from the OLS core modules and replace the high pressure nitrogen tanks with accumulator tanks and resize the cryogenic storage tanks to optimize for OLS use.

1.3.4 Derivative OLS Configuration Definition

The derivative OLS, which results when the MSS is modified for the absolutely required changes defined in Section 6.0, is presented in Section 7.0. Both structural and subsystems definitions of the resulting derivative OLS are presented. Where applicable, derivative OLS subsystems are defined by comparison with either the parent MSS subsystem or the representative OLS subsystem.

In Section 7.1, the derivative OLS flight configuration is defined. The derivative OLS is composed of modified MSS modules together with the OLS dual support cargo module (DSCM) and the OLS experiment module (XM) plus two space tugs. The MSS modules employed are:

1. The 12-man Power Module (PM)

2. Two No. 1 Core Modules (C1)

3. Control Center Module No. 1 (CCM1)

4. The 12-man version of Control Center Module No. 2 (CCM2)

5. The 12-man Galley Module (GM)

6. Crew Quarters Module No. 1 (CQM1)

7. The 12-man Crew Quarters Module No. 3 (CQM3)

8. Two Cargo Modules (CM's) modified to cryogenic storage modules (CSM's)

The structural changes required in each of the parent MSS modules are described and defined both in text and in layout drawings. The derivative OLS weight statement is also presented in Section 7.1. The derivative OLS gross weight is 223,004 pounds. A comparison of the weight of each OLS module with that of its parent MSS module is included. Also presented are derivative OLS mass properties, e.g. center of gravity locations and moments of inertia. These values were generated for guidance and control analysis purposes in the selection of an optimum configuration and flight mode and the determination of RCS impulse requirements.

In Sections 7.2 through 7.7 the derivative OLS subsystems are defined and described. The derivative OLS Structures Subsystem, Electrical Power Subsystem, Information Subsystem, Reaction Control Subsystem, and the micrometeoroid protection portion of the Environmental Protection Subsystem are but little changed conceptually from their parent MSS subsystems. The derivative OLS Environmental Control and Life Support Subsystem, Guidance and Control Subsystem, and the thermal and radiation protection portions of the Environmental Protection Subsystems are, however, modified to more clearly resemble the representative OLS subsystems concepts.

1.3.5 Recommended MSS Design Modifications

In Section 8.0, modifications to the MSS are recommended. These are proposed changes to the basic MSS which facilitate adaptation of the MSS to OLS use without significantly affecting MSS performance or cost. Some of these changes would also enhance the performance of the MSS. Some of the more significant recommended MSS modifications are:

1. Resize MSS ECLSS equipment processing rates for eight rather than six-man use.

2. Substitute regenerative fuel cells for batteries and battery chargers as the MSS EPS energy storage concept. Increase boom radiator area to 425 square feet as required for OLS use of this concept.

3. Add a third parabolic high gain antenna mounted atop the power boom to eliminate signal blockage by the solar arrays.

4. Add a landmark tracker and a radar altimeter to the guidance and control subsystem to improve navigational accuracy.

5. Incorporate a thermal capacitor and a heat pump in the active thermal control system and design segmented radiator panels so that panels located circumferentially around the modules can be removed excepting for those located (in the OLS flight configuration) on anti-lunar facing surfaces ($120^{\circ} \leq \alpha \leq 240^{\circ}$).

6. Strengthen the side docking structures on core module 1 to accommodate the loads that will be encountered during translunar and lunar orbit insertion thrust maneuvers when used as part of the OLS.

2.0 REPRESENTATIVE OLS CONFIGURATION SELECTION

In this section the representative OLS vehicle concept is developed. An overall description, including vehicle dimensions, interior layouts and mass properties is presented. In order to evaluate potential configuration drivers, an artificial gravity concept is also identified.

A description of the selected representative OLS configuration is presented in paragraph 2.1. In as much as the configuration synthesis studies described in Volume IV showed but little preference for diameter in the representative OLS guideline range, 20 to 33 feet, and as copious data are available from Earth Orbiting Space Station (EOSS) studies concerning vehicle diameters of 22 and 33 feet, a core module diameter of 27 feet was selected for the representative OLS. Layout drawings of this core module describing its structural details and functional arrangement are presented in paragraph 2.1. Also presented are drawings of the other principal structural elements of the Representative OLS, the power and equipment modules.

The OLS core module defined in Section 2.1 has four transverse decks, 27 feet in diameter, and an overall length of over 60 feet. Torroidal end pressure bulkheads with tension ties are employed for minimum weight. A centrally located flat internal pressure bulkhead divides the core module into two separate pressure volumes. A combination IVA/EVA airlock connects the two pressure volumes. A total of six docking ports are provided, two axial active ports and four passive side ports, two on deck 1 and two on deck 4. In its assembled orbital flight mode the power module, which employs a 10,000 square foot solar cell array, is docked at the +X axial port, and the experiment module with its longitudinal axis aligned with nadir is docked to one of the side ports on deck 1. Two space tugs are docked to side ports, one off each pressure volume, and a cargo module is docked, a portion of the time, to the remaining deck 4 side port. The -X axial port is reserved as a spare. Deck 1 is devoted to scientific use and contains the data analysis, photo, and geochemistry laboratories. Decks 2 and 3 house the secondary and primary control centers, respectively, and four crew state rooms each. The secondary control center of deck 2 has two important alternate functions: (1) it serves as the scientific experiment control center, and (2) it is equipped with radiation shielding which also encloses a backup galley and a hygiene facility to form the solar flare radiation shelter. Deck 4 contains the galley, dining, recreation, exercise, and medical facilities. Also located on deck 4, adjacent to the side port to which the cargo module is docked, is located a large open storage area. The unpressurized torus areas on each end of the vehicle are utilized for cryogenic storage.

Presented in paragraph 2.2 is a detailed weight statement for the representative OLS. The total OLS dry weight is 107,475 pounds, which includes an 11,320 pound power module and an 11,000 pound experiment module. The total OLS inert weight is 129,375 pounds, and the total OLS on orbit weight is 157,625 pounds, including 26,650 pounds of consumables.

In addition to the basic zero G OLS, an artificial gravity concept is described in paragraph 2.3. The artificial gravity concept consists of an OLS core module attached with an adjustable length boom to another OLS core module or a suitable counterweight. This combination rotates about the CG of the system at a rate of 3.5 revolutions per minute. A zero G hub is located at the CG point between the OLS core module and the counterweight. This hub houses the rotational driving mechanism. The power module and a multiple docking adaptor (MDA), to which are docked the experiment module, cargo module, and the space tugs, are docked to the hub.

Paragraph 2.4 presents the concept for delivery of the representative OLS to lunar orbit. Two RNS flights, or one CPS-2 flight, is required. If a RNS is expended in lunar orbit, one flight is sufficient with almost a 50 percent margin. A single stage CPS expended in lunar orbit can also deliver the total configuration with about a 12 percent margin.

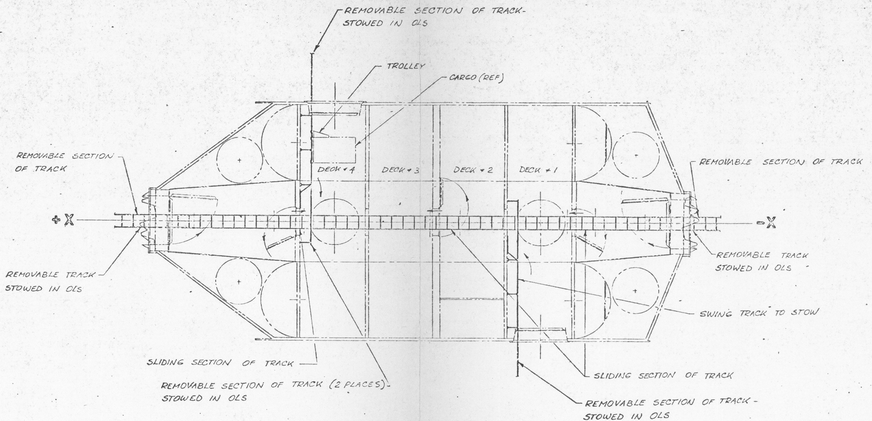
The concept using two RNS flights for delivery of the OLS assemblage delivers the basic OLS module, the power module, and approximately 65 percent of the consumables. The second flight delivers the experiment module, the remainder of the required consumables, the fully loaded space tug, and the OLS crew of eight men.

2.1 CONCEPT DEFINITION

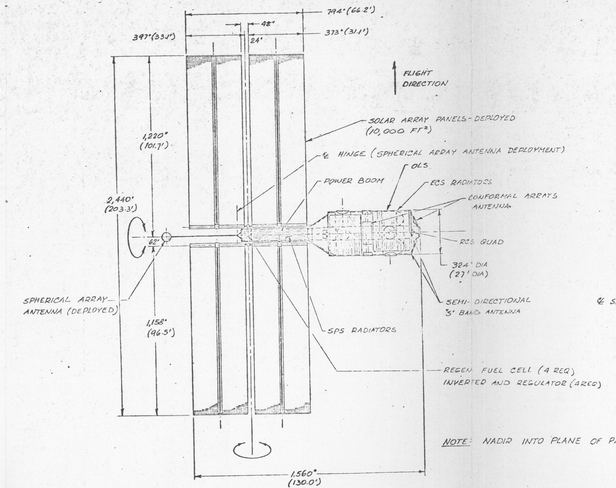
Representative OLS Core Module

This section discusses the design effort performed during the conceptual definition of the representative Orbital Lunar Station (OLS). During Phase 2 of the OLS study a comprehensive parametric conceptual trade study was performed to provide basic weight trend data for modular OLS concepts employing modules having diameters of 12 and 15 feet, and for single OLS core modules of 22 and 33-foot diameters. As a result of this effort, a subsequent assessment indicated that in the range between 22 and 33 feet, the difference in total structural weight was relatively small. Since the current EOSS core module is 33 feet in diameter, and prior design studies of 22-foot diameter EOSS core modules were available, the decision was made, with NASA concurrence, to select a 27-foot diameter for the representative OLS in order to provide another data point between existing studies. A modular concept, utilizing either 12 or 15-foot diameter modules was not selected because one of the tasks of the current study included an investigation, evaluation and delineation of the changes to the 15-foot diameter earth orbiting modular space station (MSS) which is currently under investigation.

The 27-foot diameter representative OLS detailed on Drawing 2282-25, is very similar in external appearance to the 33-foot diameter EOSS. The basic core module consists of four transverse circular decks, with toroidal end pressure bulkheads. Pressure compartmentation has been provided by making the center transverse bulkhead capable of sustaining full compartment pressure from either direction. The forward and aft outer non-pressurized structure are conical in shape to provide structural support for the cryogenic storage



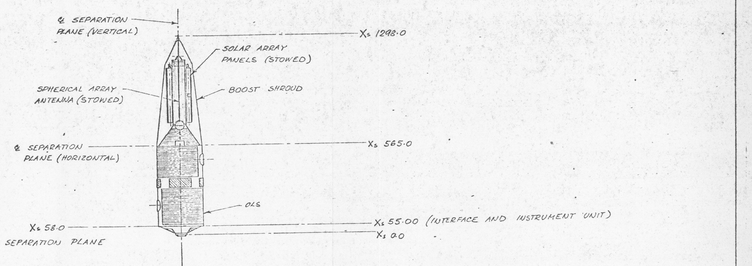
CARGO-PERSONNEL TRANSPORT SYSTEM



NOTE: NADIR INTO PLANE OF PAPER

OLS FLIGHT CONFIGURATION-ZERO G CONCEPT

SCALE: 1/200



LAUNCH CONFIGURATION

SCALE: 1/200

NOTE:
STRUCTURAL LOADING HATCHES FOR DECK = 2 AND = 3 ARE LOCATED ON +Z AXIS

Space Division
North American Rockwell

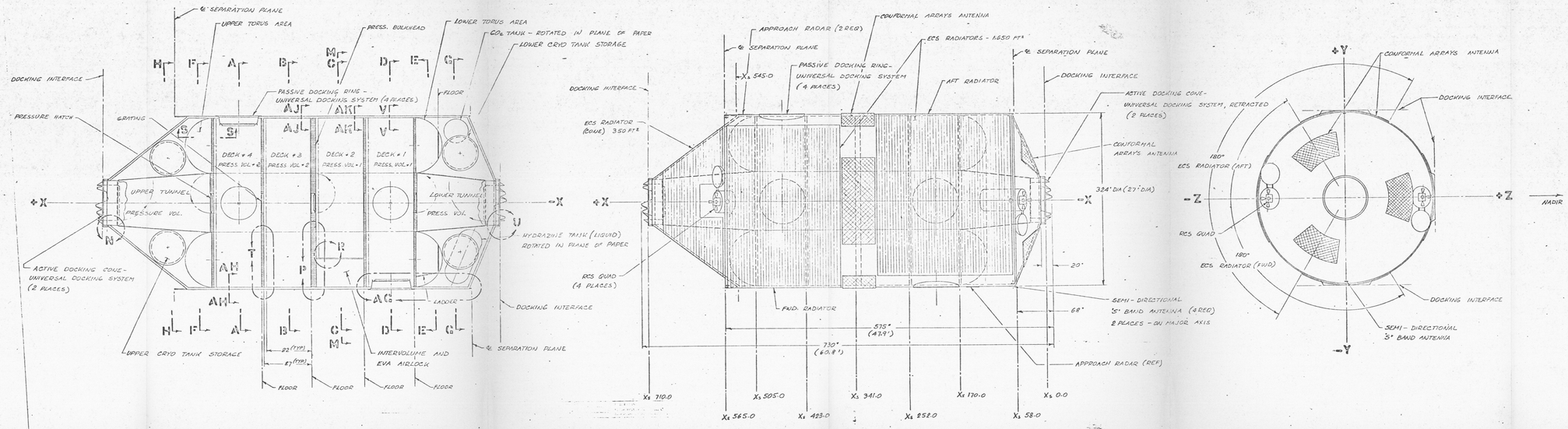
THIS PRINT REDUCED 50%

2-3, 2-4

SD 71-207

DRAWN / CHKD / APPVD	NO. 71-1-1213 DATE 7-31-1971 WHEN	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION DOWNEY/SEAL BEACH/ROCKETDYNE, CANOGA PARK/ATC, HUNTSVILLE/AUTONETICS, ANAHEIM, CALIFORNIA	
REPRESENTATIVE OLS CONFIG.-33 FT DIA, 6 MAN CREW OLS STUDY			2282-25 SHT 1 OF 6

2282-25
SHT 1 OF 6



Space Division
North American Rockwell

THIS PRINT REDUCED 50%

2-5, 2-6 | SD 71-207

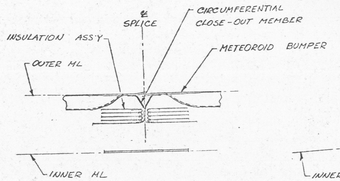
SIZE	NO.	REVISIONS	NORTH AMERICAN ROCKWELL CORPORATION
D			12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA

REPRESENTATIVE OLS CONFIG. - 27 FT. DIA, 8 MAN CREW OLS STUDY

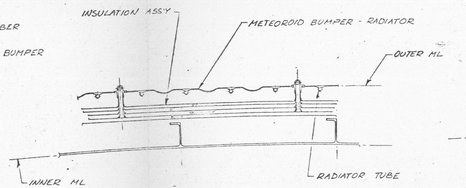
2282-25

SHT 2 OF 6

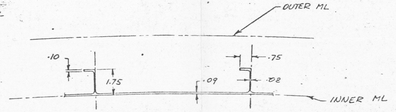
2282-25



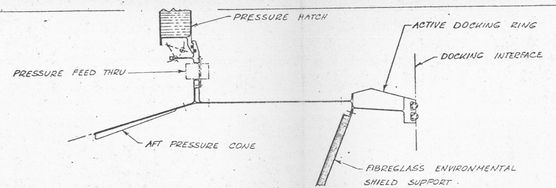
SECTION AK-AK
SCALE : 1/2



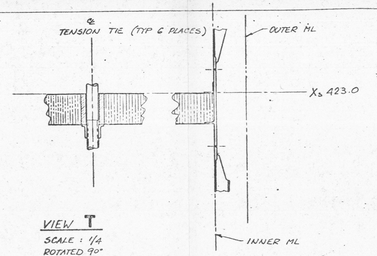
SECTION AN-AN
ROTATED 180°
SCALE : 1/2



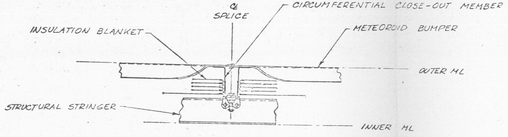
SECTION V-V
SCALE : 1/2
METEOROID BUMPER AND INSULATION OMITTED FOR CLARITY



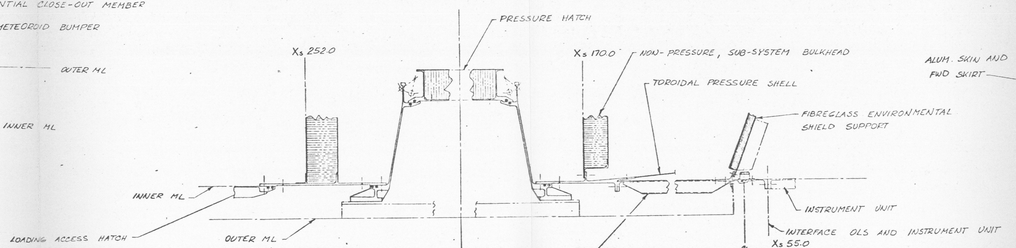
VIEW U
SCALE : 1/4



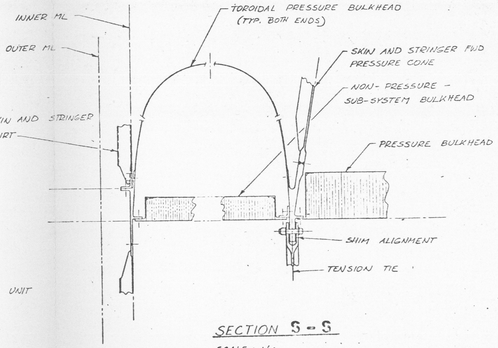
VIEW T
SCALE : 1/4
ROTATED 90°



SECTION AJ-AJ
SCALE : 1/2



VIEW AG
SCALE : 1/4



SECTION S-S
SCALE : 1/4
ROTATED 90°

Space Division
North American Rockwell

THIS PRINT REDUCED 50%

2-7, 2-8 SD 71-207

WBS LEVEL	1	2	3	4	5	6

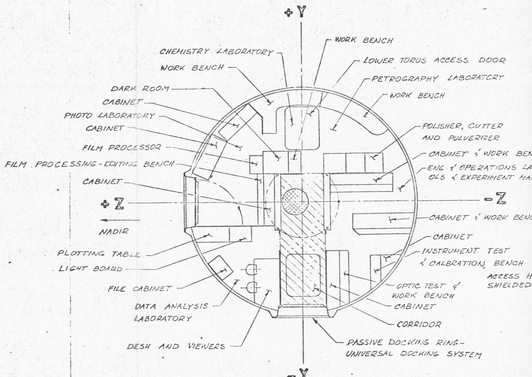
BY	DATE

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

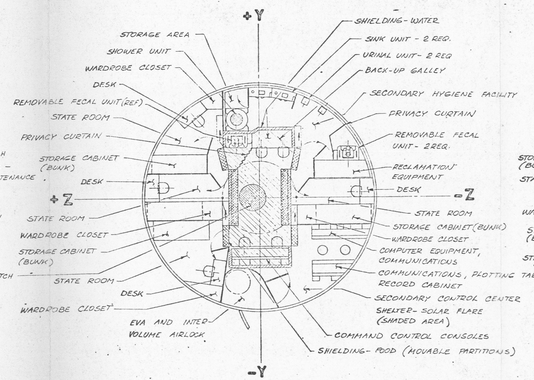
REPRESENTATIVE OLS CONFIG.- 21 FT DIA, 8 MAN CREW, OLS STUDY

2282-25
SHT 3 OF 6

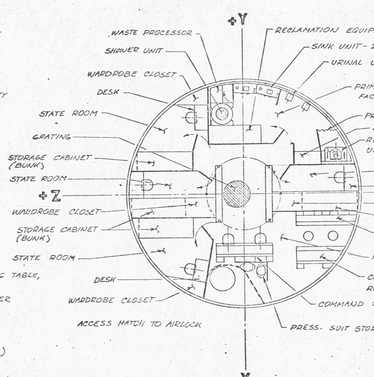
2282-25



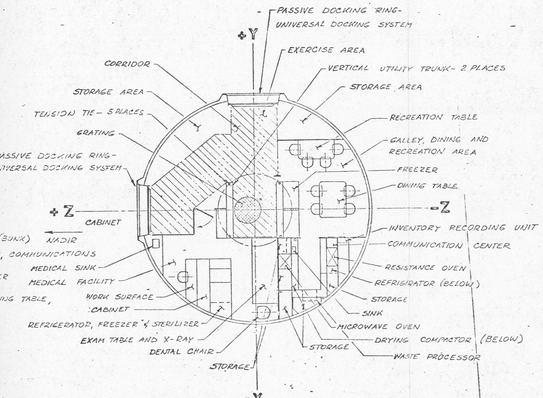
SECTION D-D
DECK = 1
PRESSURE VOLUME = 1



SECTION C-C
DECK = 2
PRESSURE VOLUME = 1



SECTION B-B
DECK = 3
PRESSURE VOLUME = 2



SECTION A-A
DECK = 4
PRESSURE VOLUME = 2

THIS PRINT REVISED 50%

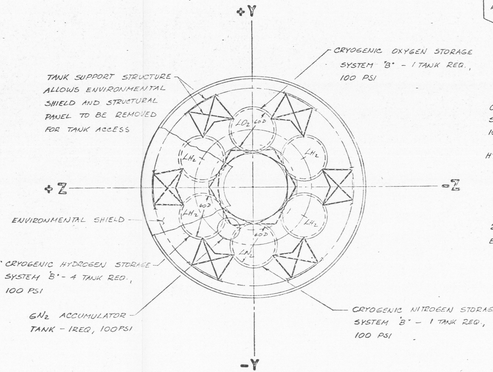
2-9, 2-10

SD 71-207

ISSUE	NO.	SPACE DIVISION	
DATE	DATE	NORTH AMERICAN ROCKWELL CORPORATION	
APPD	APPD.	DOWNEY, CALIFORNIA	

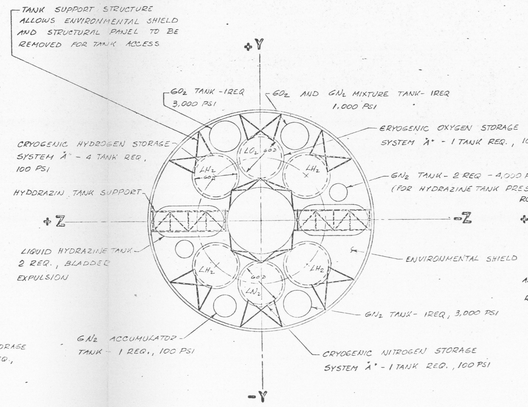
REPRESENTATIVE OLS CONFIG.- 33 FT. DIA., 8 MAN CREW, OLS STUDY

2282-25
SHT. 4 OF 6



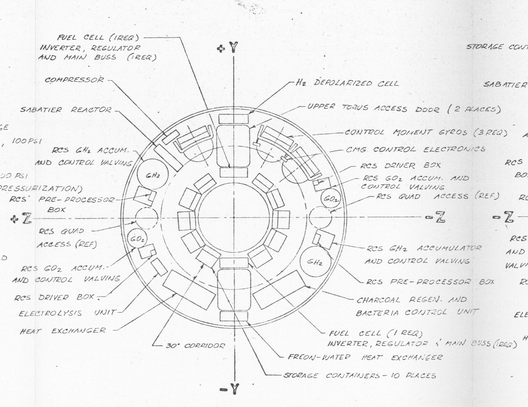
SECTION H-H

UPPER CRYOGENIC STORAGE - SYSTEM 'B'



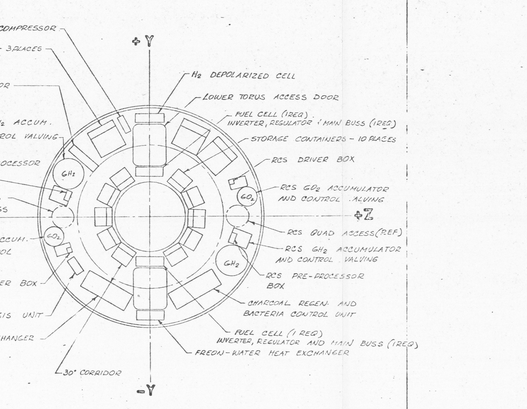
SECTION G-G

LOWER CRYOGENIC STORAGE - SYSTEM 'A'



SECTION E-E

LOWER TORUS AREA



SECTION F-F

UPPER TORUS AREA

Space Division
North American Rockwell

2-11, 2-12 SD 71-207

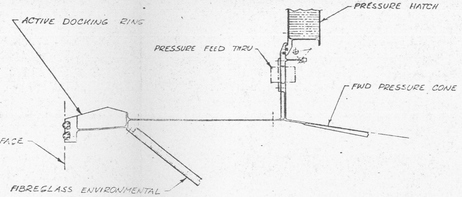
THIS PRINT REDUCED 50%

SCALE 1/20	DR. Wirtley	NORTH AMERICAN ROCKWELL CORPORATION	
NOTED	DATE	12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
	MODEL		

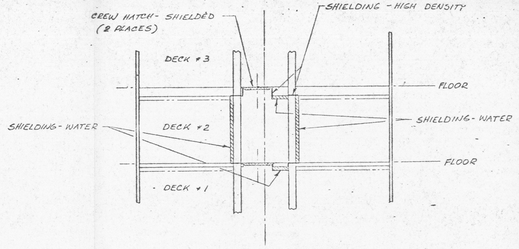
REPRESENTATIVE OLS CONFIG.- 27 FT. DIA. 6 MAN CREW. OLS STUDY

2282-20 SHT 5 OF 6

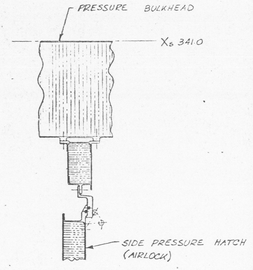
2282-20



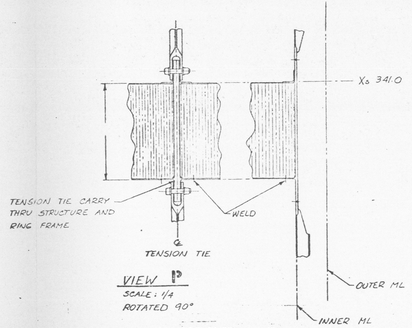
VIEW N
SCALE : 1/4



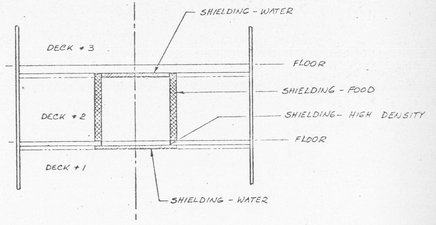
SECTION K-K



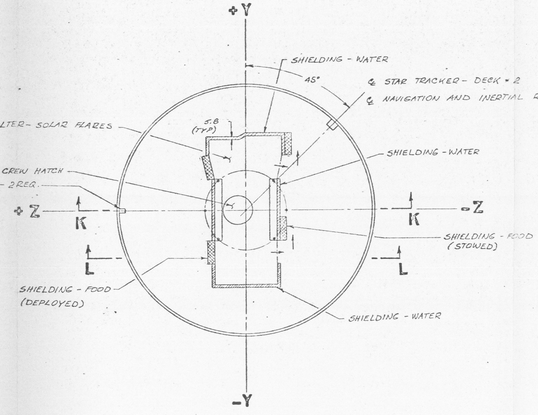
VIEW R
SCALE : 1/4
ROTATED 90°



VIEW P
SCALE : 1/4
ROTATED 90°



SECTION L-L



SECTION M-M

DECK #2
PRESSURE VOLUME #1
NOTE: ALSO HORIZON SCANNER - 2 REQ. ON DECK #3

Space Division
North American Rockwell

2-13, 2-14

SD 71-207

THIS PRINT REDUCED 50%

SIZE	DR	SPACE DIVISION
D/E/C	DATE	NORTH AMERICAN ROCKWELL CORPORATION
NOTED	WORK	12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA

REPRESENTATIVE OLS CONFIG.- 27 FT DIA. 6 MAN CREW, OLS STUDY

2282-25
SHT 6 OF 6

tanks located in these areas. Both structures are covered by thermal insulation and meteoroid protection in addition to portions of the phased array antenna on the aft structure and radiator panels on the forward cone. The selected electrical power system is basically the 10,000-square foot four-element rollout solar array system selected for the EOSS. A cylindrical power boom module, incorporating the four rollout arrays, is attached to the forward conical bulkhead by use of a neuter docking system. The cylindrical body of the OLS has a length of 42.5 feet, an outside mold line diameter of 27 feet, and a pressure shell inside diameter of 26.4 feet. The forward and aft conical non-pressurized structures are 13.75 and 4.58 feet long respectively. In the launch configuration, the OLS has an overall length of 94.25 feet. A jettisonable boost shroud configuration has been shown on the launch configuration. This shroud is biconic in shape and is 59.5 feet in length.

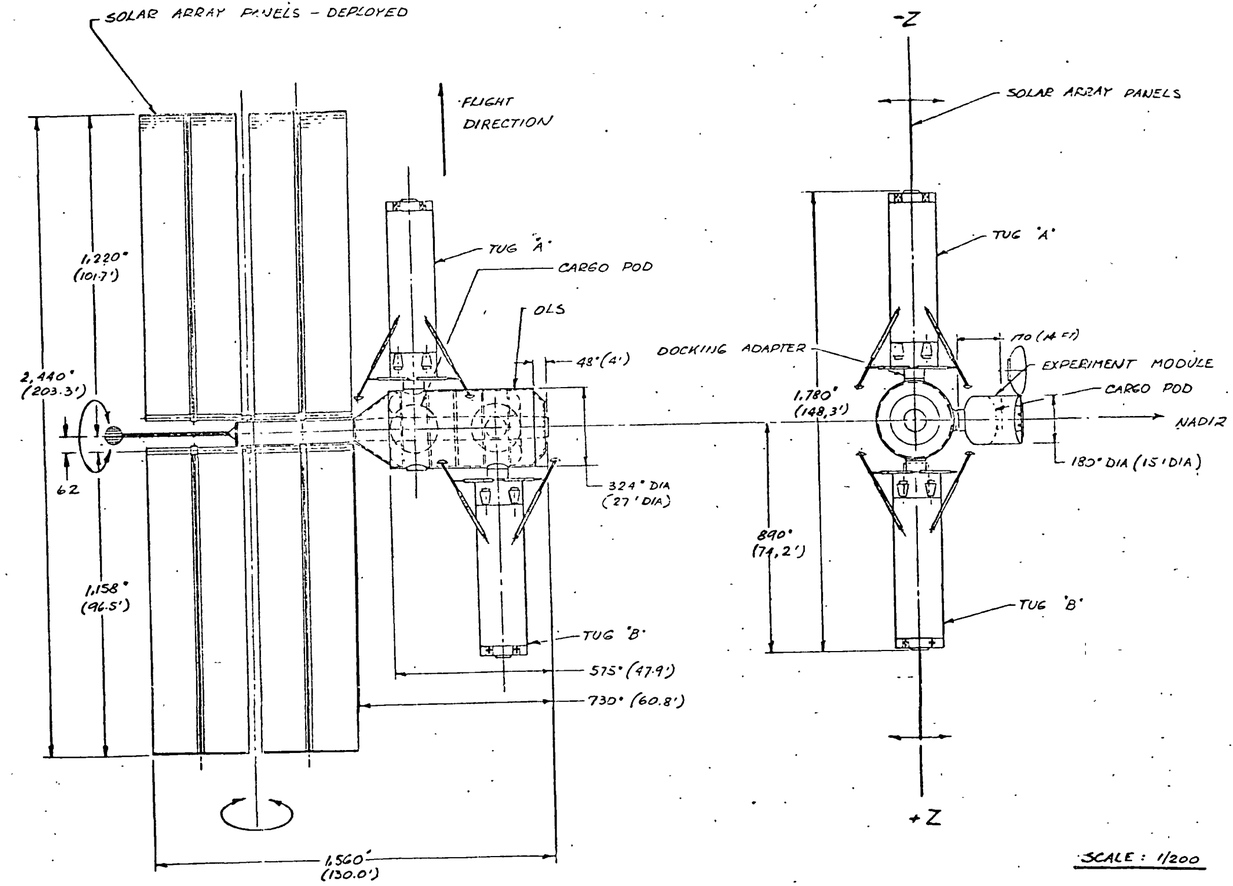
The selected orbital flight configuration, shown in Drawing 2282-27, features the OLS oriented with the X axis normal to the orbit plane, and the +Z axis pointing toward the local vertical. The two tugs are docked in the X-Y plane with one on deck 1 and one on deck 4. The cargo module is docked to the port on deck 4 which is on the +Z side, and the science experiment module is docked to the corresponding port on deck 1.

This orbital flight arrangement was selected after a comprehensive configurational study was performed to determine an optimum arrangement from considerations of attitude control and orbit maintenance propellant requirements, and science experiment accommodation. During this evaluation, 10 alternative module arrangements and flight orientations were analyzed. The quantative results of this study relative to attitude control propellant consumption is detailed in Section 3.4 of this report and will not be delineated here.

The basic core module, with four transverse decks and toroidal end pressure bulkheads has a gross internal volume of 20,020 cubic feet. Four full diameter decks are provided with flat floors. The center floor has been designed to be a pressure bulkhead to effectively compartment the core module into separate pressure volumes. Toroidal end pressure bulkheads have been used because of the lower structural weight. Structural tension rods extend longitudinally through the station and terminate in the inner edges of the toroidal bulkheads as shown in Section S-S on sheet 3 of Drawing 2282-25. The tension rods are judiciously located and do not penalize the spatial arrangements within the station. Interdeck passageways and hatch openings have been maintained at 36 inches to minimize loss of usable floor area. Although the hatch openings in the outer surface of the core module are 5 feet in diameter, the accepted philosophy is that any items initially entering the station through the 5-foot diameter hatches would subsequently be repackaged or partially disassembled to be compatible with the 3-foot interdeck hatches and passageways. Floor to deck clear ceiling heights of 82 inches are provided with an additional 5-inch false ceiling for ducting, cabling, etc.

Page intentionally left blank

Space Division
North American Rockwell



SCALE : 1/200

THIS PRINT REDUCED 50%

2-17, 2-18

SD 71-207

SCALE 1/200	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
REPRESENTATIVE OLS FLIGHT CONFIGURATION		2283-27

As shown on the general arrangement drawing, Drawing 2282-25, the decks are numbered one through four, proceeding forward through the core module. The spatial allocations which have been provided are consistent with those values previously established in the parametric phase of the study, and are based upon the general approach used in the NR EOSS study. In developing the arrangement of equipment and furnishings on each deck, a major effort was made to maximize access to the sidewalls of the pressure shell for detection and repair of any meteoroid penetrations. Whenever possible, equipment was located away from the sidewalls; however, in those cases where this was not feasible, it was planned that such items would be capable of being rotated or rolled away from the sidewall on tracks.

Deck 1, located at the aft end of the core module is primarily arranged as a scientific and experiment deck, as shown in Section D-D of Drawing 2282-25. Two docking ports are provided, one on the -Y axis for docking of a tug, and one on the +Z axis to accommodate the docked experiment module. The principal laboratories accommodated in Deck 1 are photo, chemistry, optical, and petrography. Two rectangular removable floor hatches 4 feet by 5 feet, are provided to permit access to the equipment and subsystems located in the aft torus volume as shown.

The lower torus area, shown in Section E-E on sheet 4 of Drawing 2282-25, has been used to contain a portion of the subsystems, which service the aft pressurized section of the core module. Major subsystems include fuel cells, GO_2 and GH_2, CMG's, environmental control system and RCS quad control and support systems. In addition, a portion of the area is utilized for bulk storage. The subsystems and equipment have been arranged in two concentric rings with a 30-inch wide corridor between them to provide manned access to all of the items in the torus area.

Deck 2 has been configured to accommodate four staterooms, the secondary or backup control center, solar flare "storm cellar", and the secondary hygiene facility, as shown in Section C-C on sheet 4 of Drawing 2282-25. The EVA and intervolume airlock is also located in this compartment. In configuring the storm cellar, maximum utilization of subsystems and major elements available on this deck has resulted in a feasible and efficient concept. The storm cellar is basically an area approximately 7 by 16 feet that will accommodate the entire crew for the duration of a solar flare event. A sanitation facility and the backup galley have been included to eliminate the need for the crew to leave the storm cellar to reach these facilities.

The primary control and display portions of the secondary control center are also included in the storm cellar. By locating the storm cellar centrally, the two deck-to-deck utility runs are utilized and form a portion of the sidewalls. Four doors or movable partitions which contain food are also used for sidewall portions of the storm cellar. To supplement the radiation protection provided by these items, 5.8-inch deep water filled panels are used as sidewalls and in the floor and ceiling of the storm cellar as shown in Sections C, M, L, and K on sheets 4 and 6 of Drawing 2282-25. The total amount of water required for the shields is 16,000 pounds. During occupancy of the storm cellar, and as a result of the limited volume, it is suggested that only one or two of the crew be on duty at a time. The remainder would "sack out"

in sleeping restraints designed to position the crew in an upright position in the center of the compartment to maximize the space for the on-duty crew.

Deck 3, shown in Section B-B on sheet 4 of Drawing 2282-25, provides for accommodation of the other four crew staterooms, the primary hygiene facility, primary control center, and the hatch which permits access thru the pressure bulkhead and into the EVA airlock located in Deck 2. Space for storage of two pressure suits is provided in the area adjacent to the inter-deck hatch.

Deck 4 as shown in Section A-A on sheet 4 of Drawing 2282-25, contains the medical facility, primary galley and dining area, exercise area, and bulk storage area. Two docking ports are also provided, one on the +Z axis to accommodate the logistics cargo module, and one on the +Y axis for the second tug/lander vehicle.

The forward torus area, shown in Section F-F on sheet 5 of Drawing 2282-25, above the ceiling of Deck 4, accommodates the life support subsystems for the forward pressure compartment of the core module, as shown in Section F. Two 4 by 5-foot hatches in the ceiling provide access to this area. Access in the toroidal area is facilitated by a 30-inch wide annular corridor as shown.

Two unpressurized compartments are provided, one at each end of the core module, for accommodation of the cryogenic storage containers. The detail of these compartments is shown in Sections H-H, G-G, E-E, and F-F on sheet 5 of Drawing 2282-25. The aft compartment shown in Section G, accommodates half of the total cryogenic supply in 60-inch diameter tanks. There are four for LH_2, one for LO_2, and one for LN_2. Several high-pressure spheres are provided for gaseous nitrogen and oxygen. In addition, the two hydrazine tanks (for storage of subsatellite propellant), each 45 inches in diameter by 105 inches long, are located in this area. The forward unpressurized area, shown in Section H-H, accommodates the other half of the cryogenics in 60-inch diameter spheres, four for LH_2, one each for LN_2 and LO_2, and a single GN_2 accumulator. All of the cryogenic storage tanks are insulated with high performance insulation blankets and are structurally supported by the use of low thermal loss struts.

The external surface of the core module accommodates the thermal and meteoroid protection system, in addition to the conformal antenna arrays. A total surface area of 2000 square feet of the core module is allocated to radiators. This includes 1650 square feet on the cylindrical body and the remainder on the forward conical structure. The body-mounted radiators are divided into two major segments each encompassing a 180-degree sector and positioned on the upper surface of the core module in the orbital flight orientation. A typical structural detail of a radiator panel is shown in Section AH-AH on sheet 3 of Drawing 2282-25. Meteoroid bumper details are shown in Section AJ-AJ also on sheet 3. The bumper, consisting of longitudinally corrugated fiberglass panels, is supported 3.75 inches away from the core module pressure shell. Thermal insulation for the core module consists of multiple blankets of superinsulation supported on low conduction standoffs, as shown in Sections AH, AJ, and AK on sheet 3 of Drawing 2282-25, in between the pressure shell and the meteoroid bumper and radiators. The conformal antenna system consists of four-phased array panels 5 by 14 feet equally

spaced about the girth of the core module, plus three trapezoidal shaped panels attached to the outer conical surface of the aft unpressurized area. To provide hemispherical coverage forward of the OLS, a 5-foot diameter spherical phased array is mounted on the end of a boom attached to the power boom and deployed to the position shown on the flight configuration on sheet 1 of Drawing 2282-25.

In addition to the four previously identified passive docking ports located on the side of the core module in Decks 1 and 4, active neuter docking cones are positioned on the vehicle centerline at each end. The forward docking cone will be used to receive the solar power boom, and the aft will provide the capability to accept any vehicle equipped with either an active or passive system. Each of these two docking interfaces are connected to the adjacent pressurized deck by a pressurizable tunnel. This tunnel could function as an airlock if required to permit access into an unpressurized module or to perform an EVA.

The RCS nozzle clusters configured for the OLS consist of four groups of four nozzles each, with a thrust level of 10 pounds, located at the forward and aft ends of the core module on the X-Y plane. Each nozzle cluster is provided with an operable cover door, which when closed will permit shirt-sleeve access to the quad via a tunnel interfacing with the toroidal pressure bulkhead at each end of the core module.

Cargo and personnel transfer within the core module during orbital operations is facilitated by the use of an integrated rail transport system as shown on Sheet 1 of drawing 2282-25. This system is the result of a detailed design and operational system performed on the EOSS study, and was adopted for the OLS. The longitudinal rail system is continuous except at the decks, where a section is hinged to fold out of the way to permit opening and closing of the deck hatch. An integral part of the concept is an overhead rail system attached to the ceiling of each deck, which permits lateral movement of cargo from the longitudinal rail system out onto each deck consistent with the aisle space available.

The selected primary eletrical power system for the OLS is a four-element roll-out solar array system providing a total area of 10,000 square feet. This system is basically that selected for use on the EOSS. The solar array modules are attached to the forward end of 90-inch diameter by 450-inch long cylindrical structure which provides structural support of the arrays. The cylindrical structure, called the power boom, is equipped with passive docking interfaces at each end. This will permit the power boom to be docked to the active docking system which is integrated into the forward conical unpressurized structure of the core module. The mechanical and structural interface between the four roll-out arrays and the power boom provides two degrees of freedom to permit optimum orientation of the arrays. To provide thermal control for the solar array power conditioning equipment, a radiator system is attached to the exterior surface of the power boom.

Science Experiment Accommodation

One of the major configurational issues which required an early resolution in order to permit conceptual development of the representative OLS, related to accommodation of the total spectrum of science experiments selected for operation in lunar orbit. A review of the total experiment complement indicated that they could be segregated into two different classes; those that would be mounted on the OLS and those that would be carried on the subsatellites. Table 2-1 identifies the experiments by sensor number and shows their location. Additional scrutiny of the OLS associated experiments also identified certain experiments that would require periodic manned access to retrieve and resupply data storage consumables such as film and tape. A subsequent review of the OLS attitude hold and pointing capabilities indicated that the capability was compatible with the requirements of those same science experiments and it would not be necessary to mount these instruments on hinged gimballed platforms. Therefore, it was decided to accommodate these experiments in an area which would provide shirtsleeve access for film cassette and tape retrieval and resupply. This is in contrast to the NR EOSS experiment accommodation which features these items mounted on hinged and gimballed platforms and only accessible via EVA or by retracting deployed sensors and closing up of the experiment module.

The next major configurational investigation related to the feasibility of accommodating the science experiments in the basic core module versus a separate experiment module. Figure 2-1 presents the configurational characteristics and major changes which result if the experiments were body mounted. In order to satisfy the field-of-view requirements and physically accommodate the 110 square feet of surface area normal to the lunar local vertical that is required to mount the sensors of the experiments, an additional deck and operable doors encompassing 180 degrees of the circumference would be required. The cameras would be mounted in a deployable compartment which would be accessible from within the adjoining pressurized deck. It was concluded that structural complexity and attendant weight penalty would be excessive for this approach.

Subsequently a conceptual study was performed to investigate the accommodation of the science experiments in a separate module. For purposes of this study, the experiment module was constrained to a 15-foot diameter to be shuttle compatible. In concert with the flight orientation investigation previously discussed, two different experiment module concepts were examined. The first concept, Figure 2-2 featured the experiment module docked to the aft centerline port on the core module, with the experiments deployed and viewing normal to the longitudinal axis. This concept required double hinging of some of the large arrays to permit stowage in the module and comply with the operational field-of-view requirements. The cameras would be accommodated in a pressurized compartment adjacent to the full diameter compartment provided for subsatellite servicing.

The second experiment module concept, Figure 2-3 featured the module positioned on the +Z axis and the experiments viewing from the end. This concept resulted in an experiment module less complicated structurally than the previous and one that only requires opening of the end closure. Closing of the experiment module for servicing of the externally mounted experiments does not require retraction of any of the experiments as a necessary and critical first step in closing of the module.

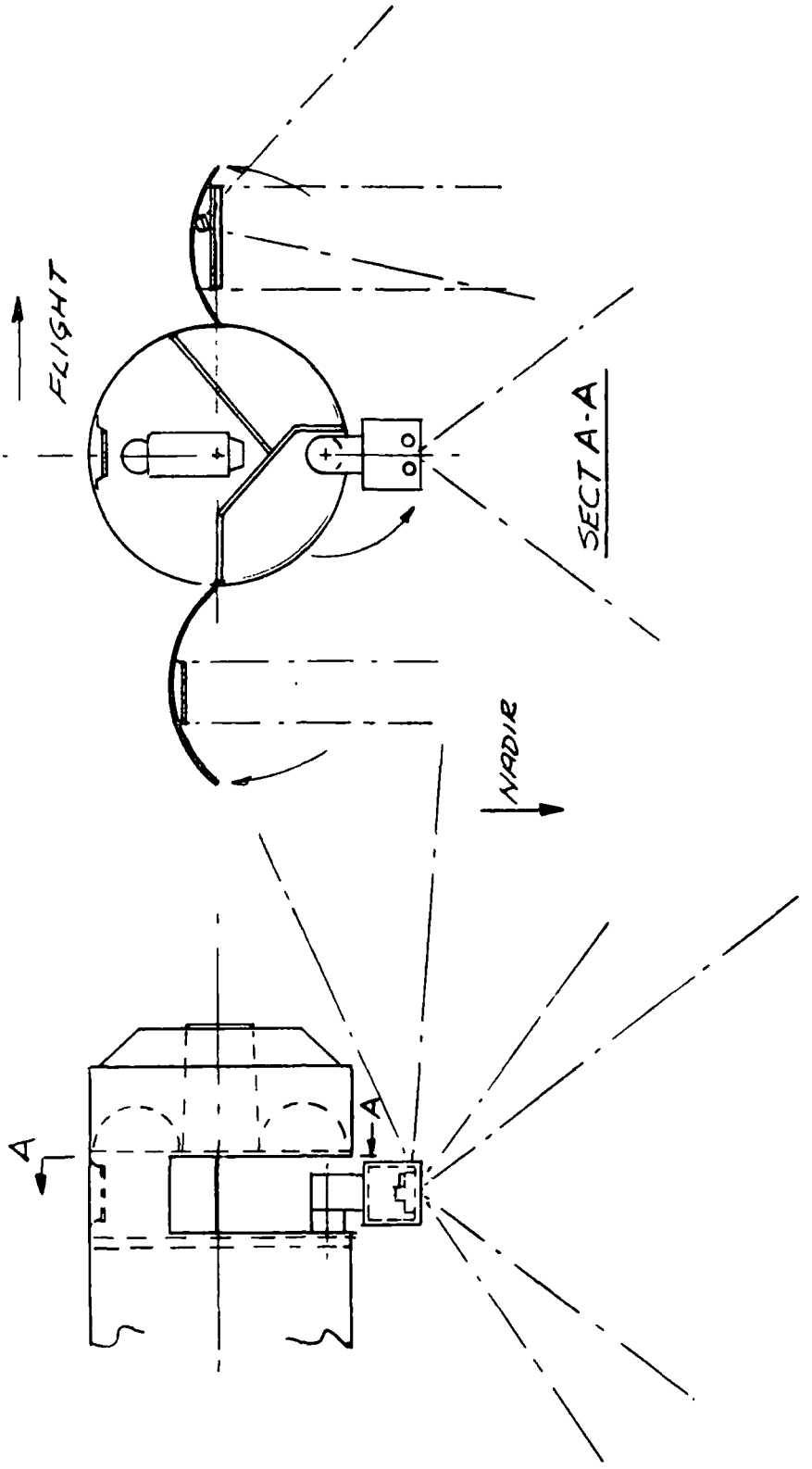


Figure 2-1. Integrated Experiment Concept (Bottom Viewing)

+ Flight direction into plane of paper

Figure 2-2. End Docked Experiment Module
(Side Viewing)

Space Division
North American Rockwell

+ Flight direction into plane of paper

Figure 2-3. Bottom Docked Experiment Module
(End Viewing)

Space Division
North American Rockwell

Table 2-1. Science Experiment Location

OLS Mounted Experiments	
11002	6" metric camera stellar camera (2)
11008	UV imager/spectrometer
11042	High resolution camera
11043	3" metric camera
11099	Laser altimeter
11012	Radar imager
11013	IR radiometer
11015	Passive microwave image
11041	Radar altimeter/scatterometer
11007	Microwave radiometer
Subsatellite Mounted Experiments	
11009*	Radar reflectometer
11010*	EM measurement system
11059*	Antenna (10M Hz - 100G Hz)
11069	X-ray spectrometer
11070	Pulse height analyzer
11098	Cosmic ray spectrometer telescope
12001	Ionization pressurization sensor
14001	Low energy charged particle counter
14003	Alpha proton counter
14005	Mass spectrometer
14018	Gas chromatograph
14020	Charged particle spectrometer
15007	Gravity gradiometer
15009	Electric field meter
15010	Helium magnetometer

*Additions to Lunar Orbiter type satellite

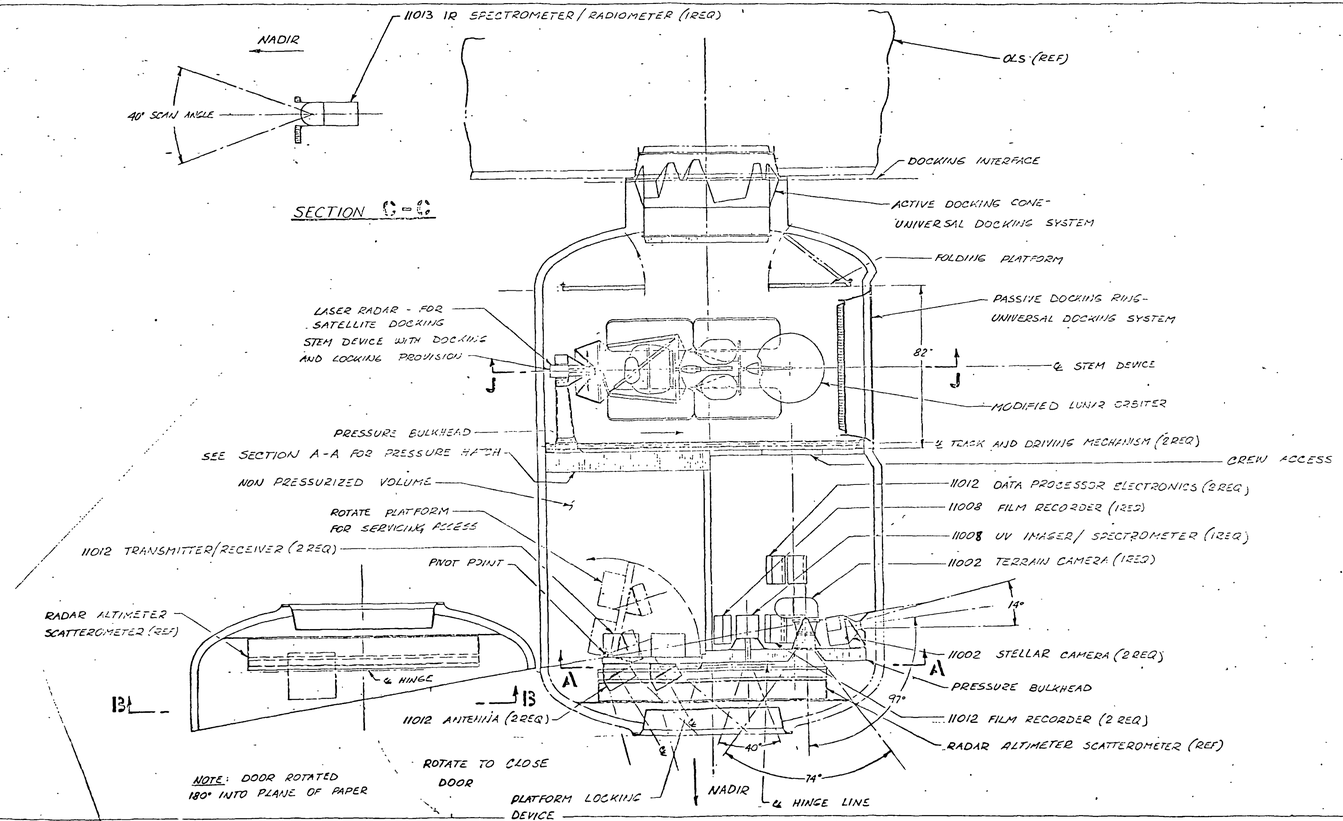
Based upon these advantages, the end viewing experiment module was selected as the preferred concept, and subsequently configurationally developed in greater detail as shown on Drawing 2282-26.

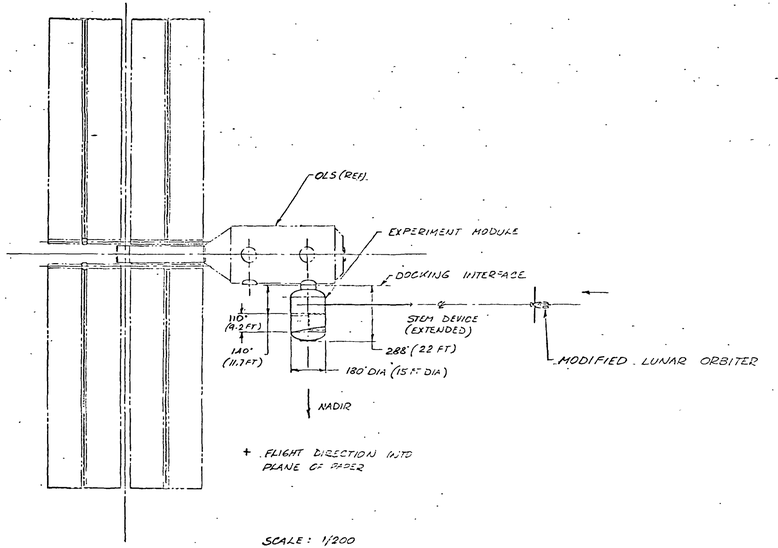
The selected experiment module is basically a 15-foot diameter structure, 22 feet long featuring elliptical end closures. Docking systems are incorporated in each end, an active cone in the end which docks to the OLS, and a passive ring in the end closure which is hinged to permit experiment operation. The basic purpose of the experiment module is twofold. It accommodates the total complement of science experiments which have been located on the subsatellites as a result of specific operational requirements or physical characteristics which are not compatible with the representative OLS configuration and orbital flight orientation. In addition to the science experiments, the experiment module also contains a separate compartment which has been configured to function as an airlock and servicing hanger for the subsatellites.

The subsatellite airlock is configurationally obtained by compartmenting the experiment module with a flat pressure bulkhead as shown on Drawing 2282-26. The compartment is circular in cross section and has a nominal ceiling height of 7 feet. To permit entry into the airlock, a 5-foot diameter hatch, as part of a passive docking ring, is incorporated in the side of the experiment module. A major feature of the airlock is a transverse rail system upon which the subsatellites will be appropriately supported when brought inside of the module. A translating carriage which incorporates an extendable docking probe is provided to permit initial acquisition of the subsatellite at a stand-off distance of 37 feet from the experiment module, and subsequent entry into the experiment module. Control of this operation is performed from within an appropriate control station located in the science experiment deck, Deck 1 of the OLS. The subsatellite servicing deck is adequate for accommodating the representative modified Lunar Orbiter and Pioneer satellites, individually, and providing excellent and complete physical access for servicing the subsatellites.

As previously discussed, the decision was made to incorporate those specific experiments or sensors which require frequent data storage exchange and renewal; i.e., tape and film, in a compartment which would permit shirtsleeve access as required. This compartment is basically a cylindrical module, 7 feet in diameter and 8.3 feet long interfacing with the flat pressure bulkhead provided for the subsatellite airlock. Access to the sensor bay is thru an open passageway thru the flat pressure bulkhead. The sensors and experiments which are located in the compartment include the 6-inch metric camera with integrated stellar index cameras, UV imager/spectrometer, high-resolution camera, 3-inch metric camera with integrated stellar index camera, and laser altimeter. In order to provide optimum system performance of these items, they view through physical openings in the sides and end of the compartment. Conical structural elements extend from the pressure shell structure to close proximity of the lens barrels. A flexible boot capable of satisfying the compartment pressurization requirements provides the adapter interface between the lens barrels and the conical structural elements. This approach permits the compartment to operate in a pressurized mode, yet does not result in any loads or stresses being introduced to the lens barrels as a result of structural and physical deflections between the compartment pressure shell and the individual sensors.

Page intentionally left blank





SCALE: 1/200

NOTE:
STRUCTURE OF OUTER SHELL, INSULATION AND METEOROID BUMPER ARE SAME AS SHOWN ON DWG 2282-25

2-29-, 2-30

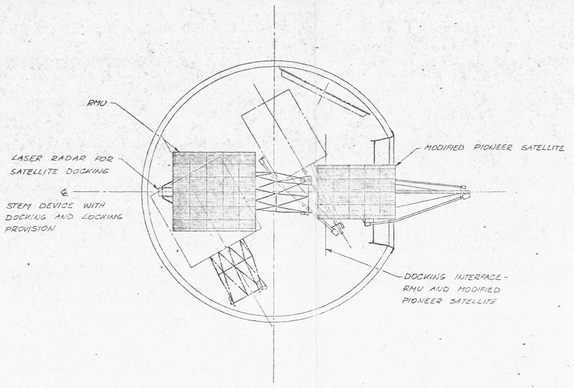
SD. 71-207

THIS PRINT REDUCED 50%

SCALE 1/20 + NONE	DR. [illegible]	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
	CH. [illegible]		
	APP. [illegible]		
	MODEL		

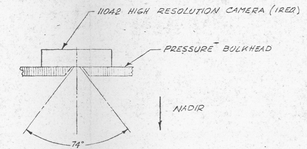
EXPERIMENT MODULE - 15 FT DIA, OLS STUDY

2282-26A
SHT 1 OF 2

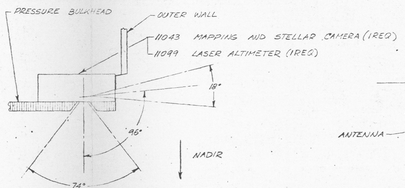


SECTION J - J

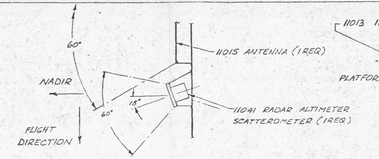
NOTE: RMU DECOUPLED FROM MODIFIED PIONEER SATELLITE AND MOVED APPROX 12" TO CLEAR SATELLITE PRIOR ROTATION OF RMU.



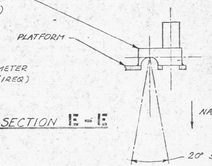
SECTION G - G



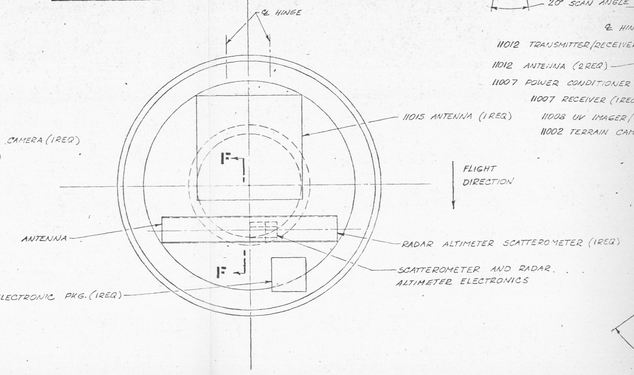
SECTION H - H



SECTION F - F

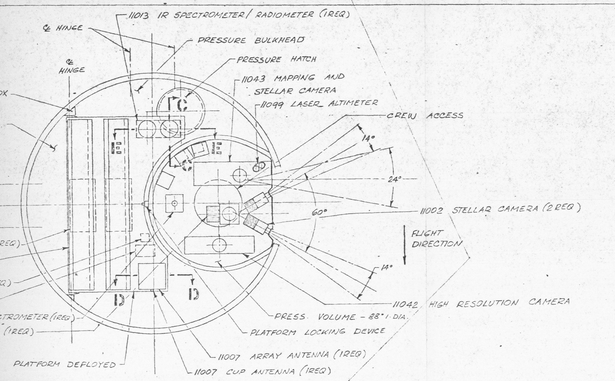


SECTION E - E

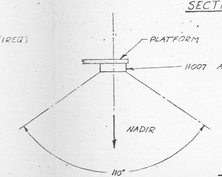


SECTION B - B

ROTATED 180°



SECTION A - A



SECTION D - D

NOTE: SCAN ANGLE ADJUSTABLE TO CLEAR SOLAR ARRAY PANELS

THIS PRINT REDUCED 50%

2-31, 2-32

SD 71-207

SCALE	NONE
DATE	
MODEL	

SPACE DIVISION
NORTH AMERICAN ROCKWELL CORPORATION
SEAL BEACH/LOS ANGELES, CALIFORNIA

EXPERIMENT MODULE - 15 FT DIA, OLS STUDY

2282-26A
SHT 2 OF 2

The remainder of the science experiments are mounted on a movable platform in the unpressurized portion of the experiment module, and on the hinged deployable end closure. The specific location of these instruments has primarily been dictated by their physical characteristics and field of view and pointing requirements. Those items mounted on a platform in the experiment module adjacent to the camera bay include two radar imager antenna, the microwave radiometer antenna, and the IR radiometer. The radar image antenna are oriented with the longitudinal axis parallel to flight direction.

In order to provide physical access to these items for servicing and maintenance, the mounting platform is designed to be hinged, permitting a crew member to have access to each of the antenna assemblies. This would be performed with the experiment module and bulkhead door closed to provide a shirtsleeve environment. The physical size of the radar altimeter and passive microwave imager does not permit mounting on the same platform as the previously identified items. Therefore, they have been mounted on the inner surface of the hinged end pressure bulkhead. The door is rotated 90 degrees to point the antenna toward the local vertical. The physical size of the two antenna fit well within the diameter of the end closure and, therefore, do not require any unfolding or mechanization for operation after the door is fully opened. The electronic modules and systems associated with each of the sensors are located in the unpressurized section of the experiment module behind the mounting platform which supports the various antenna.

During the initial experiment integration studies, it was determined that it would not be considered practical to accommodate the RF reflectrometer and electromagnetic measurement experiment antenna on the OLS. These antennas were basically deployable stem devices of 50, 100 and 750-foot length, clustered in groups of four. In the deployed configuration, the plane of the antenna array was required to be approximately normal to the local vertical and in all of the various orbital flight orientations that were considered for the OLS, the antenna elements were either in the field of view of the cameras, or physically interfering with the rollout solar array panels.

Only one OLS flight configuration was compatible with the antenna and that was with the X-X axis pointing toward the local vertical. This flight attitude was not acceptable from thermal heat rejection considerations (refer to Section 3.6.2). Therefore, the decision was made to place these science experiments and their attendant electronic subsystems on the Lunar Orbiter subsatellite.

Docking Adapter

The docking elements on the cylindrical sides of the basic OLS module are the same as the passive element of the EOSS. This configuration was selected to avoid the incorporation of boost fairings that would be required if active elements were used. Passive elements are essentially flush with the mold line of the module. Active elements extend approximately 2 feet beyond the mold line. This concept requires an adapter to permit docking of the tug and other lunar program elements that may have passive docking elements to dock to the OLS. The adapter consists of two active male docking mechanisms integrated back-to-back into a cylindrical structure. The docking adapter kits would be transported to lunar orbit as part of the cargo during the OLS buildup.

2.2 OLS MASS PROPERTIES

Representative OLS Core Module Dry Weight

The structure subsystem weight includes three basic compartments, the upper and lower nonpressurized compartments and the main pressurized compartments. The detail weights are shown in Table 2-2 and discussed below.

Lower Nonpressurized Compartments

Lower Skirt. The lower skirt is a 27-foot diameter section, 8 feet long acting as the nonpressurized compartment shell and booster attach structure. The lower skirt has a total area of 661 square feet. The material areal density is 2.02 pounds per square foot; therefore, the lower skirt gives a weight of 1335 pounds.

Lower End Cap. The lower end cap serves as a radial closure for the lower nonpressurized compartment. The lower end cap has a total area of 546 square feet. The material areal density is 0.57 pounds per square foot; therefore, the lower end cap weighs 311 pounds.

Attachment Weight. Attachment weight for these two main elements of the lower nonpressurized compartment was estimated at 94 pounds.

Pressurized Compartments

End Pressure Bulkheads. The two end pressure bulkheads of the pressure compartment are semi-torus in section. The total area of these bulkheads is 1879 square feet. The material areal density is 0.83 pounds per square foot, resulting in a total weight of 1560 pounds.

Internal Pressure Bulkhead. The single transverse (circular) pressure bulkhead is of honeycomb construction having an area of 524 square feet. The material areal density is 7.0 pounds, resulting in a total weight of 3668 pounds.

Inertia Bulkheads. The four interia bulkheads, acting as floors, are of honeycomb construction with a total area of 2096 square feet. The material areal density is 2.18 pounds per square foot, resulting in a total weight of 4569 pounds.

Cylindrical Shell. The cylindrical shell forms the main pressure compartment wall and is of skin stringer construction having a length of 31.33 feet. The area of the cylindrical shell is 2588 square feet. The material areal density is 2.27 pounds per square foot, resulting in a weight of 5875 pounds.

Upper Tunnel. The upper tunnel connects the main pressure compartment with the upper docking port. It has an area of 779 square feet and an areal density of 1.85 pounds per square foot, yielding a weight of 779 pounds.

Table 2-2. Detail Weight Statement
OLS - 27' Dia; 8 Man

Core Module (dry)			(85,155)
Structure			28,300
Lower Nonpressurized Comp.		1,740	
Lower Skirt	1,335		
Lower End Cap	311		
Attachments	94		
Pressurized Compartment		24,592	
End Press. Bulkhead (2)	1,560		
Internal Press. Bulkhead	3,668		
Inertia Bulkheads (4)	4,569		
Cylindrical Shell	5,878		
Tunnel - Upper	779		
Tunnel - Lower	896		
Tunnel - Upper RCS	38		
Tunnel - Lower RCS	55		
Tunnel Attachment	91		
Tension Ties (6)	658		
Docking Hatch (6)	1,080		
Ground Access Hatch Prov (2)	186		
Pressure Bulkhead Hatch	160		
Inertia Bulkhead Hatch (7)	203		
Torus Access Hatch (4)	140		
Internal Airlock	350		
Windows & Ports (3)	84		
Partitions - Vertical	740		
False Ceilings	400		
Storage Prov.	900		
Vertical Utility Trunks	800		
Horizontal Utility Trunks	295		
RCS Access Dome Outer Covers (4)	200		
Window Covers	18		
Brackets	150		
Contingency	694		
Upper Nonpressurized Comp.		1,968	
Upper Skirt	490		
Upper End Cap	1,106		
Fairing Adapter	272		
Attachments	100		
Docking Structure			2,480
Lower End Port (1)		580	
Upper End Port (1)		580	
Side Port (4)		1,320	

Table 2-2. Detail Weight Statement (continued)

Induced Environment Protection				8,160
Thermal Protection			690	
Coatings		24		
Insulation		625		
Main Cyl. Shell	294			
Lower Skirt	75			
Lower End Cap	62			
Upper Skirt	34			
Upper End Cap	94			
Misc.	66			
Vent Fittings		41		
Meteoroid Protection			4,470	
Solar Radiation Shelter & Hi Density Shield			3,000	
Reaction Control System				8,625
Main System			5,133	
Tanks (cryo) (12)		3,508		
Tank Supts		464		
Propellant Conditioning Assy (Hx & Blowers)		55		
Plumbing, Valves, Regulators, etc.		582		
Engine Clusters (4)		120		
Accumulator		306		
Spares		98		
Sub-Satellite System			3,492	
Tanks - Hydrozene		2,099		
Tank - Supports		477		
Tank - Pressurant		746		
Plumbing, Valves, Reg. etc.		100		
Compressor		50		
Spares		20		
Electrical Power System				5,445
Fuel Cells (4) - Emergency			552	
Fuel Cell Mounts & Supports			55	
Power Conditioning			722	
Controls		60		
Inverters (4-2.5 KVA)		250		
Secondary Bus		300		
Mounts & Supports		112		
Distribution Control & Wiring			3,427	
Wire Harness		3,000		
Buses, Contactors		58		
Mounts & Supports		369		

Table 2-2. Detail Weight Statement (continued)

Lighting		599	
Lights	544		
Mounts & Supports	55		
Contingency		90	
Guidance and Control			2,255
Telescope/Sextant		88	
Star Tracker (2)		66	
Landmark Tracker		33	
Horizon Tracker (4)		60	
Radar Altimeter		35	
IMU		45	
Docking Laser (2)		56	
RCS Drivers (5)		50	
CMG (3)		1,520	
Pre-Processor (6)		165	
NAV Base		17	
Calibration Equip (4)		120	
Information Management System (ISS)			6,380
Data Processor Assy.		2,259	
Processor (5)	50		
I/O Controller (5)	50		
Operation Memory Module (8)	400		
Mass Memory Module (13)	1,300		
Archival Memory Tape Deck (2)	240		
RACU - 256 (24)	144		
RACU - 128 (8)	24		
RACU - 64 (10)	15		
Central Timing Unit (2)	36		
Display Control Assy.		1,205	
Console Struct. & Ftgs. (3)	249		
Printer (3)	120		
Display Add Keyboard (6)	36		
Function Select (6)	90		
Light Emitting Diode (LED) Display (6)	60		
LED Display electro (6)	120		
CRT (3)	75		
CRT Electronics (3)	60		
Microfilm Viewer (3)	105		
Hand Control Stick (6)	120		
Monitor and Alarm (3)	90		
Remote Terminal Unit (2)	80		
Communication Tracking Assy.		1,060	
Phase Array (7) (370 ft^2 total)	130		
Semidirect (8)	8		
Approach Radar (2)	50		

Table 2-2. Detail Weight Statement (continued)

Transponder Pwr. Amp. & Diplexer (12)	348		
Premodulator Processor (1)	50		
Facsimile Unit (1)	40		
Central Sw. Unit & self-test (2)	240		
Audio/Video Units - A (20)	120		
Audio/Video Units - B (3)	54		
Paging Amplifier (2)	20		
Peripheral Group		755	
Voice Recorder (3)	360		
Audio Recorder (2)	30		
Entertainment Recorder (1)	15		
Video Cameras (Color) (9)	90		
Video Monitors (9" Color) (12)	180		
Video Monitors (14" Color) (2)	80		
ISS Interconnecting Cables		850	
Spares		251	
Environmental Cont. Life Support System			14,920
Atmospheric Storage		1,807	
Oxygen Tank (Hi-Press.)	795		
Nitrogen Tank (Hi-Press.)	618		
Pumpdown System (2)	360		
Nitrogen Accumulators (2)	34		
Atmospheric Control (Incl. 244 lb. spares)		1,764	
CO_2 Management		1,973	
Hydrogen Dept. Cell (2) (Incl. 221 lb. spares)	1,271		
Sabatier (2) (Incl. 50 lb. spares)	150		
Solid Polymer (2) (Incl. 152 lb. spares)	552		
Active Thermal		3,144	
Hardware (Incl. 120 lb. spares)	2,384		
Radiator (2000 ft^2)	760		
Water Management		1,177	
Pot. Recl.-V.C. (2) (Incl. 328 lb. spares)	998		
Pot. H_2O Tank (4)	29		
Wash H_2O Recl - RD (Incl. 49 lb. spares)	92		
Wash H_2O Tank (2)	10		
Purity System (3) (Incl. 12 lb. spares)	48		

Table 2-2. Detail Weight Statement (continued)

Waste Management (2) (Incl. 63 lb. spares)		536	
Hygiene		2,152	
Hardware (Incl. 50 lb. spares)	837		
Crew Pers Eq. (Incl. 45 lb. spares)	895		
Laundry	420		
Food Management (Incl. 150 lb. spares)		1,150	
Spec. Life Support			504
IVA Umb. Loop (3) (Incl. 30 lb. spares)	92		
EVA - PLSS (2)	160		
EVA - LiOH	7		
EVA - OPS (2)	81		
Fire Fight (12)	164		
Contingency		713	
Personnel Provisions			2,390
Accommodations for Personnel		1,670	
Chairs, Bunks & Tables	1,000		
Recreation & Exercise	250		
Medical & Dental Equip.	200		
Mobility Aids & Restraints	120		
Supports	100		
Furnishings		400	
Ladders	230		
Tools - Maint.	150		
Supports	20		
Emerg. Equip.		270	
Cargo Handling		50	
Crew Station Control & Panels			1,080
Control Stands		1,060	
Display & Control Consoles	1,000		
Stateroom Console	60		
Supports		20	
Experimental Laboratories			5,120
Maintenance Lab.		1,000	
Data Analysis Lab.		120	
Photo Lab.		1,500	
Geochemistry Lab.		2,500	

Table 2-2. Detail Weight Statement (continued)

Power Boom (dry)			(11,320)
Structure			1,540
Bulkheads		76	
Outer Walls		1,072	
Hatches		180	
Partitions		114	
Brackets, etc.		57	
Misc.		41	
Induced Environment Protection			450
Thermal Protection		117	
Insulation	110		
Vent Fittings	7		
Meteoroid Protection		333	
Bumper	214		
Reinforcement	52		
Closeouts & Supports	67		
Docking			560
Passive Ring Ports		280	
Electrical Power System			7,440
Solar Cell Rollup Array		4,720	
Orientation Mech.		286	
Array Support (Boom)		374	
Mounts & Supports		432	
Fuel Cells (4-5 KW)		721	
Electrolysis Units (4)		251	
Reactants		38	
Tankage		40	
Inverters (4-10 KVA)		504	
Mounts & Supports		50	
Wiring		20	
Mounts & Supports		4	
Environment Control System			1,080
Atmospheric Control Syst.		40	
Coolant System		1,040	
Equipment (Incl. 30 lb. spares)	570		
Radiator (500 ft^2)	470		
Communications			230
Antenna Instal.		205	
Antenna (5'D Spherical)	35		
Boom (30' Rigid)	130		
Retraction Mech & Systs.	40		
Contingency		25	

Table 2-2. Detail Weight Statement (continued)

Experiment Module (dry)		(11,000)
Structure		4,700
End Press. Bulkheads	400	
Cylindrical Shell	800	
Internal Press. Bulkhead	830	
Press. Can	170	
End Bulkhead	260	
Docking Hatch (3)	540	
Press Bulkhead Hatch	160	
Air Lock	350	
Misc. Partitions	300	
Satellite Retrieval Mech.	200	
Door Mech.	200	
Contingency	490	
Docking Structure (3)		1,130
Instruments (21)		1,920
Sub-Satellites		1,230
No. 1	140	
No. 2	140	
No. 3	950	
Environmental Control System		100
Induced Environment Protection		850
Electrical Power System		840
Contingency		230

Lower Tunnel. The lower tunnel connects the main pressure compartment with the lower docking port. It has an area of 309 square feet and a material areal density of 2.9 pounds per square foot, yielding a weight of 896 pounds.

RCS Tunnels and Attachment. The RCS tunnel for both the upper and lower RCS quad access was estimated to weigh 184 pounds.

Tension Ties. The weight of the six tension ties having a length of 31.33 feet each was estimated to be 658 pounds.

Docking Hatches. The six docking hatches are identical to those used on the EOSS. Based on EOSS weight estimates these would weigh 1080 pounds.

Ground Access Hatch Provision. Two ground access hatches are provided and are identical in weight to those employed on EOSS, a total of 186 pounds.

Pressure Bulkhead Hatch. The internal pressure bulkhead hatch is identical to that employed on the EOSS. It weighs 160 pounds.

Inertia Bulkhead Hatches. The weight of the seven inertia bulkhead hatches based on EOSS estimated weights is 203 pounds.

Torus Access Hatches. The four torus access hatches are assumed identical to those employed on the EOSS, weighing a total of 140 pounds.

Internal Airlock. The internal airlock is identical to that employed on the EOSS; it weighs 350 pounds.

Windows and Ports. A total of three windows and viewing ports weighing a total of 84 pounds are required.

Vertical Partitions. The vertical partition weight was estimated at 82 percent of the EOSS vertical partition weight (ratio of 27 feet/33 feet), resulting in a total weight estimate of 740 pounds.

False Ceilings. Weight estimated as 74 percent of the EOSS false ceiling weight (ratio of diameters squared), resulting in a total weight of 400 pounds.

Storage Provisions. Weight estimated as 74 percent of the EOSS storage provision weight (ratio of diameters squared), resulting in a total weight of 900 pounds.

Vertical Utility Trunks. Vertical utility trunk weight estimated to be 800 pounds.

Horizontal Utility Trunks. Weight estimated as 82 percent of EOSS horizontal utility trunk weight (ratio of diameters), resulting in a total weight of 295 pounds.

RCS Access Dome Outer Covers. The RCS access dome outer covers are estimated to weigh 200 pounds, based on the EOSS weight estimates.

Window Covers. Window covers are assumed the same as those used on the EOSS, which weighed 18 pounds.

Brackets. Bracket weight is estimated to total 150 pounds, approximately 74 percent of EOSS bracket weight.

Contingency. The pressure compartment contingency weight of 694 pounds was estimated to be approximately 3.5 percent of the total pressure compartment weight.

Upper Nonpressurized Compartment

Upper Skirt. The upper skirt is a 27-foot diameter section, 3.7 feet long acting as the nonpressurized compartment shell. The upper skirt has a total area of 303 square feet. The material areal density is 1.62 pounds per square foot, resulting in a total weight of 490 pounds.

Upper End Cap. The upper end cap frustum serves as a closure for the upper nonpressurized compartment. It has an area of 832 square feet and material areal density of 1.33 pounds per square foot, resulting in a total weight of 1106 pounds.

Fairing Adapter. The fairing adapter is a two-foot long section, 27 feet in diameter and acts as the power module fairing attach point. It has an area of 170 square feet and based on an estimated areal density of 1.6 pounds per square foot, weighs 272 pounds.

Attachments. Attachment weight is estimated to be 100 pounds, 82 percent of EOSS attachments weight (ratio of diameters).

Docking Structure

The docking structure consists of two active end ports and four passive side ports. The structure weight of the active docking ports based on EOSS weight estimates is 580 pounds each. The structural weight of the four passive side ports is estimated to be 330 pounds each based on EOSS weight estimates.

Induced Environment Protection

The induced environment protection consists of the items shown in Table 2-2; these are discussed below.

Thermal Protection

Coatings. OLS coating weight is estimated to be 24 pounds based on EOSS data.

Insulation. The OLS insulation weight is based on the complete core module external surface area of 4930 square feet and an areal density of 0.1136 pounds per square foot, plus an estimated attachment provision weight of 66 pounds. These estimates result in a total insulation weight of 625 pounds.

Vent Fitting. Vent fitting weight is estimated to be 41 pounds based on EOSS data.

Meteoroid Protection

The OLS meteoroid protection weight is estimated to be 4470 pounds, based on the total core module area of 4930 square feet assuming an areal density of 0.9 pounds per square foot which includes support and spacing frames.

Solar Flare Radiation Cellar

The OLS solar flare cellar dry is estimated to be 3000 pounds which includes approximately 1000 pounds of high density radiation shield. The remaining 2000 pounds is to provide structural housing for the 16,000 pounds of water required as radiation shielding. This 16,000-pound water weight is included as an inert fluid weight.

Other Subsystems

The other core module subsystem weights shown in Table 2-2 are based on data from other portions of the report as follows:

ECLSS	- Section 3.1	G&C	- Section 3.4
EPS	- Section 3.2	RCS	- Section 3.5
ISS	- Section 3.3		

Personnel provisions, crew station control and panels weight estimates shown in Table 2-2 are based on EOSS data. The experimental laboratory weight estimates shown in Table 2-2 are from Section 4.0 of Volume I, the OLS Objectives Report.

OLS Power Module Dry Weight

Structure

The power module structure weight shown in Table 2-2 is 1540 pounds, identical to those estimated during the EOSS study.

Induced Environment Protection

The power module induced environment protection weight shown in Table 2-2 is identical to the EOSS estimated weight of 450 pounds.

Docking Structure

The power module docking structure weight for two passive docking ports is estimated to be 560 pounds shown in Table 2-2 is based on EOSS data.

Electrical Power System

The electrical power system power module weights shown in Table 2-2 are based on requirements specified in Section 3.2 and include a solar array with four 2500-square foot panels.

Environmental Control System

The environmental control and life support subsystem power module weights shown in Table 2-2 are based on requirements specified in Section 3.1.

Communications

The power module communications weights shown in Table 2-2 consists primarily of a five-foot diameter spherical directive phased array antenna installed on a rigid 30-foot boom. The boom weight is based on a maximum deflection of one degree.

Experiment Module Dry Weight

Structure

The experiment module structure weight shown in Table 2-2 is based on a 15-foot diameter cylindrical section, 17 feet long with ellipsoidal ends having an overall length of 22 feet.

The ellipsoidal end pressure bulkheads weight was estimated based on data in Volume IV, Configuration and Systems Analysis Report, which results in an areal density of approximately 1.0 pounds per square foot.

The cylindrical shell weight of 800 pounds is based on an areal density of 1.0 pounds per square foot.

The centrally located transverse (circular) pressure bulkhead weight of 830 pounds is based on an estimated areal density of 4.7 pounds per square foot.

The remaining items of structural weight were estimated based on EOSS or OLS core module data.

Docking Structure

The structural weight of the three docking ports was estimated to weigh 1130 pounds.

Instruments and Subsatellites

The instruments and subsatellite weights shown in Table 2-2 are as listed in Section 4.0 of Volume I.

Induced Environment Protection

The induced environment protection weight was based on the environment protection weight required for the EOSS earth surveys experiment module (ESEM), since it is dimensionally quite similar to the OLS experiment module.

Electrical Power System

The electrical power system weight was also based on the weight required by the EOSS ESEM.

Environmental Control System

The environmental control system weight estimate was based on an assumed fluid heat transfer loop in the experiment module originating in the core module.

Contingency

The contingency requirement represents approximately 2 percent of the experiment module empty weight.

Group Weight Statement

Table 2-3 presents the dry weight of the three OLS modules at the subsystem level. These are summarized from the detail weight breakdown presented in Table 2-2.

OLS Weight Summary

Table 2-4 presents the OLS maximum weight in lunar orbit, weight build-up based on the module dry weights. Shown is the total OLS dry weight, inert weight, and maximum in lunar orbit gross weight. This maximum gross weight represents a condition existing immediately after resupply. Consumables are for eight men and are based on 180-day nominal storage or 109-day resupply plus contingency requirements, whichever is the greater. Table 2-4 denotes the controlling requirement for each of the consumables.

OLS Mass Moments of Inertia

The mass moments of inertia of various docked configurations of the representative OLS were computed to support stability and control analyses as a function of cargo, tug and experiment module locations and of flight and nadir direction.

Table 2-3. Group Weight Statement - OLS
(8 Crewmen; 27 ft. dia.)

	Dry Weight	
Core Module		(85,155)
Structure	28,300	
Docking structure	2,480	
Induced environment protection	8,160	
Reaction control system	8,625	
Electrical power system	5,445	
Guidance and control	2,255	
Information management system (ISS)	6,380	
Environmental control/life support system	14,920	
Personnel provisions	2,390	
Crew station control & panels	1,080	
Experimental laboratories	5,120	
Power Module		(11,320)
Structure	1,540	
Docking Structure	580	
Induced environment protection	450	
Electrical power system	7,440	
Environmental control system	1,080	
Communications	230	
Experiment Module		(11,000)
Structure	4,700	
Docking structure	1,130	
Induced environment protection	850	
Electrical power system	840	
Environmental control system	100	
Instruments	1,920	
Sub-satellites	1,230	
Contingency	230	

Space Division
North American Rockwell

Table 2-4. Weight Summary - OLS
(Maximum in Lunar Orbit)

Core module (dry)		85,155
Power module (dry)		11,320
Experiment module (dry)		11,000
Total OLS Dry Weight		107,475
Inert Fluids		21,900
Oxygen (hi-press.)	666	
Nitrogen (hi-press.)	515	
Water	389	
Water (solar-radiation shield)	16,000	
Cryogenic H_2 & N_2 (residual)	1,510	
Launch Air	2,250	
Freon	180	
N_2H_4	390	
Total OLS Inert Weight		129,375
Consumables		26,650
Oxygen (cryo) (109 days + contingency)	6,823	
Hydrogen (cryo) (180 days)	1,645	
Nitrogen (cryo) (180 days) (Incl. 455 lb for sub-satellites)	3,551	
N_2H_4 (sub-satellites) (for one sub-satellite mission)	7,811	
Sabatier Expendables (109 days + contingency = 191 days)	453	
Atmospheric Control (109 days + contingency = 191 days)	147	
Water (109 days + contingency = 191 days)	112	
Waste management & purity (109 days + contingency = 191 days)	58	
Personnel hygiene supply (109 days + contingency = 191 days)	200	
Food (109 days + contingency = 191 days)	5850	
Crew (8)		1,600
Total OLS maximum lunar orbit weight		157,625

Space Division
North American Rockwell

Four docked conditions for each of three flight directions and four alternate locations of the modules were studied. These combinations resulted, however, in only four different configurations, each with the four docked conditions. These unique configurations are represented in Figures 2-4, 2-5, 2-6, and 2-7. The docked conditions are shown in Table 2-5.

Table 2-5. Docked Configurations

Docked Config.	Time* Days	Tug A		Tug B		Cargo Mod.	
		Full	Empty	Full	Empty	Full	Partial
A	16	X	-	X	-	X	-
B	31	X	-	X	-	-	X
C	31	X	-	-	X	-	-
D	31	X	-	-	-	-	-

*complete cycle = 109 days

The weights used in this inertia analysis were based upon preliminary estimates and are as follows:

	Weight - lb
Core module (dry)	78,620
Onboard experiments	4,120
Power module (less solar array)	4,305
109-day supply	16,532
+71 day supply	10,768
Crew	1,600
Experiment module	7,050
Tug A (full)	84,615
Tug A (empty)	24,615
Tug B (full)	91,900
Tug B (empty)	31,900
Cargo module (full)	29,470
Cargo module (partially full)	10,945

Table 2-6 presents the moments of inertia of the docked configurations represented in Figures 2-4, 2-5, 2-6, and 2-7. To be noted is that these properties do not include the solar array since it is handled separately in the control analysis. The mass properties of the solar array were identified, however, in Volume IV.

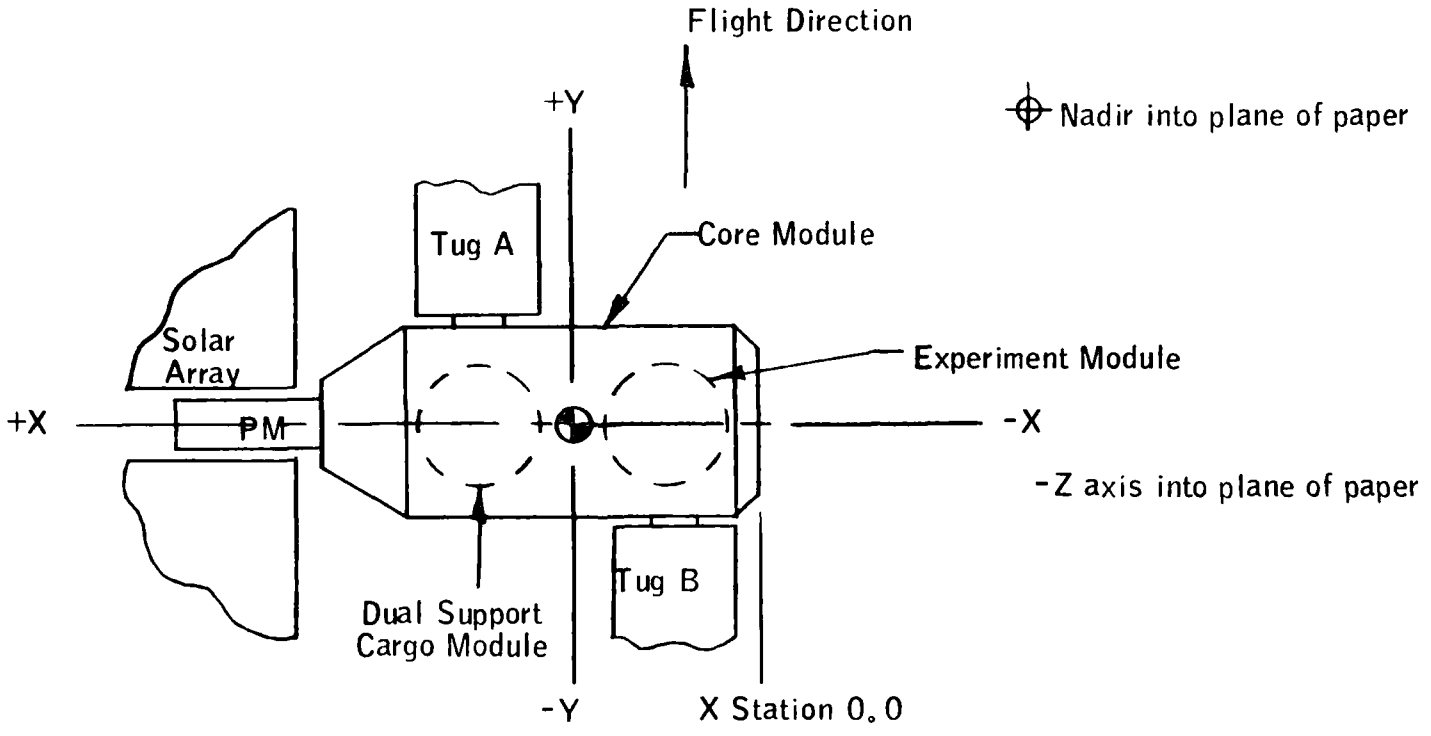


Figure 2-4. Representative OLS Docked - Configuration 1

Space Division
North American Rockwell

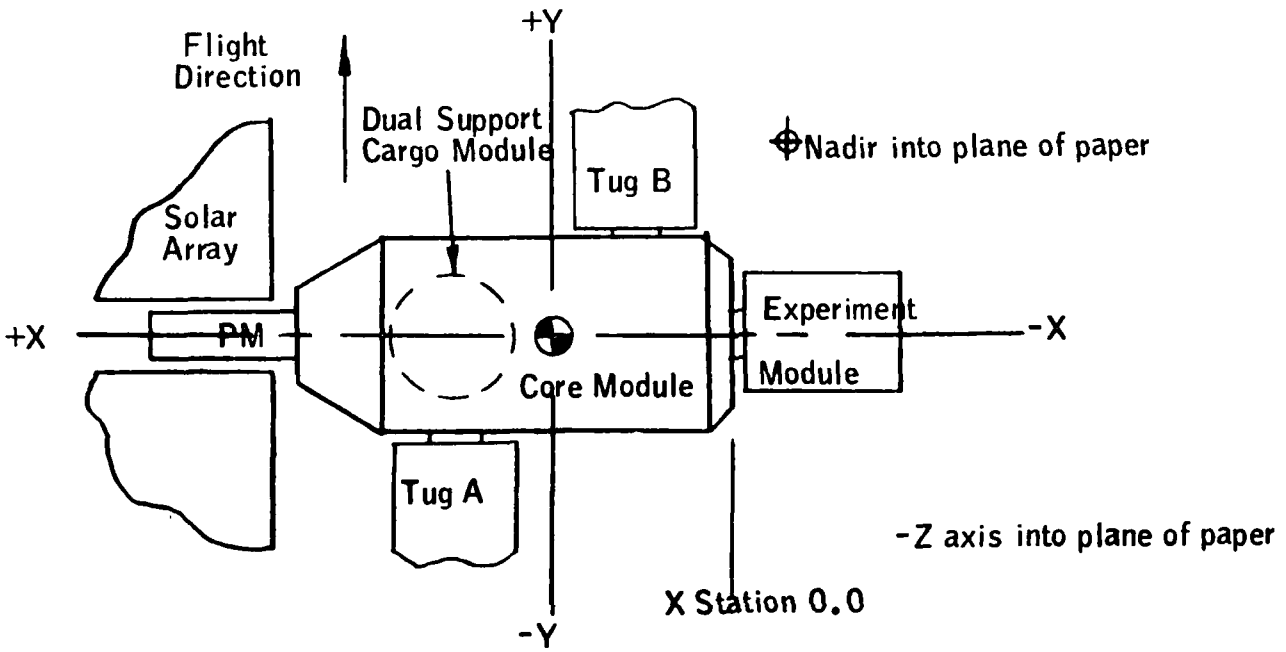


Figure 2-5. Representative OLS Docked - Configuration 2A

Space Division
North American Rockwell

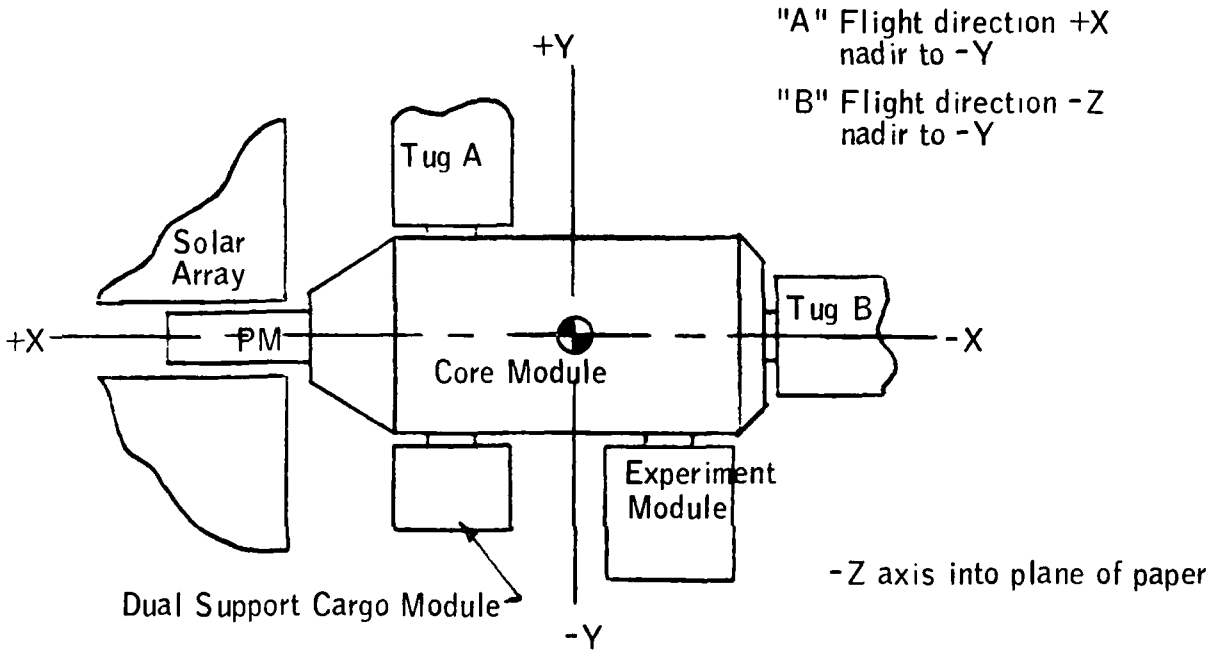


Figure 2-6. Representative OLS Docked-Configuration 2B, C, and D

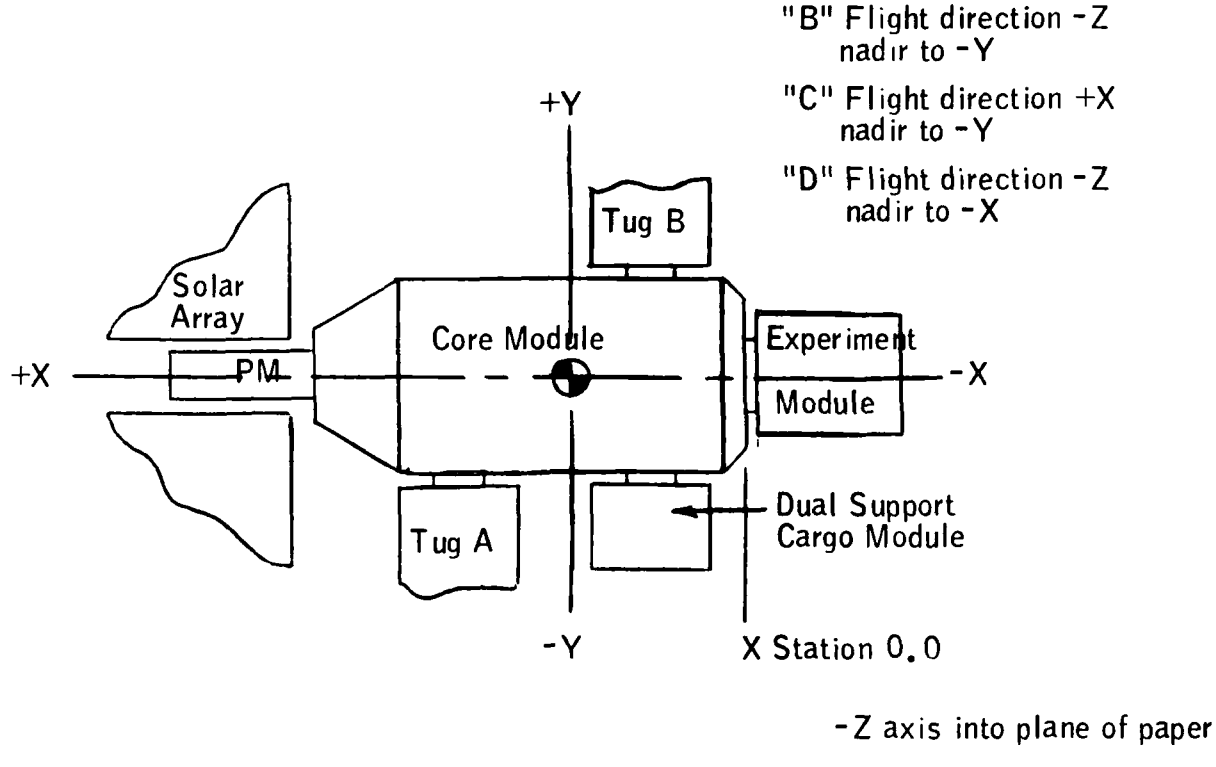


Figure 2-7. Representative OLS Docked - Configuration 3A and B

Table 2-6. Mass Properties - 27 Ft Dia, 8-Man Representative OLS (Less Solar Array) - 180-Day On-Board OLS Supply

Configuration		Weight (lb)	Center of Gravity			Moment of Inertia - Slug Foot^2			Product of Inertia - Slug Foot^2		
			X	Y	Z	I_X	I_Y	I_Z	I_{XY}	I_{XZ}	I_{YZ}
2A	Dock Conf A	328,980	358.4	11.7	-20.6	$.13 \times 10^8$	$.29 \times 10^7$	$.15 \times 10^8$	$-.31 \times 10^7$	$-.20 \times 10^6$	$.21 \times 10^5$
	" " B	310,455	350.0	12.4	- 8.1	$.13 \times 10^8$	$.26 \times 10^7$	$.15 \times 10^8$	$-.31 \times 10^7$	$-.77 \times 10^5$	$.11 \times 10^5$
	" " C	239,510	383.2	-116.6	0.0	$.77 \times 10^7$	$.20 \times 10^7$	$.94 \times 10^7$	$-.18 \times 10^7$	$.4 \times 10^4$	$.4 \times 10^4$
	" " D	207,610	412.9	-216.0	0.0	$.41 \times 10^7$	$.17 \times 10^7$	$.55 \times 10^7$	$-.82 \times 10^6$	$.4 \times 10^4$	$.4 \times 10^4$
2B, C, & D	" " A	328,980	328.9	-8.9	0.0	$.13 \times 10^8$	$.26 \times 10^7$	$.15 \times 10^8$	$-.29 \times 10^7$	$.4 \times 10^4$	$.4 \times 10^4$
	" " B	310,455	338.3	4.3	0.0	$.13 \times 10^8$	$.25 \times 10^7$	$.15 \times 10^8$	$-.30 \times 10^7$	$.4 \times 10^4$	$.4 \times 10^4$
	" " C	239,510	383.2	-116.6	0.0	$.77 \times 10^7$	$.20 \times 10^7$	$.94 \times 10^7$	$-.18 \times 10^7$	$.4 \times 10^4$	$.4 \times 10^4$
	" " D	207,610	412.9	-216.0	0.0	$.4 \times 10^7$	$.17 \times 10^7$	$.55 \times 10^7$	$-.82 \times 10^6$	$.4 \times 10^4$	$.4 \times 10^4$
3A & B	" " A	328,980	211.6	109.8	0.0	$.58 \times 10^7$	$.11 \times 10^8$	$.17 \times 10^8$	$.24 \times 10^7$	$.4 \times 10^4$	$.4 \times 10^4$
	" " B	310,455	194.4	130.0	0.0	$.53 \times 10^7$	$.11 \times 10^8$	$.16 \times 10^8$	$.28 \times 10^7$	$.4 \times 10^4$	$.4 \times 10^4$
	" " C	239,510	319.4	179.1	0.0	$.46 \times 10^7$	$.53 \times 10^7$	$.96 \times 10^7$	$.18 \times 10^7$	$.4 \times 10^4$	$.4 \times 10^4$
	" " D	207,610	423.7	206.6	0.0	$.43 \times 10^7$	$.13 \times 10^7$	$.54 \times 10^7$	$.86 \times 10^6$	$.4 \times 10^4$	$.4 \times 10^4$
1	" " A	328,980	365.3	-11.7	-25.9	$.13 \times 10^8$	$.27 \times 10^7$	$.14 \times 10^8$	$.32 \times 10^7$	$-.13 \times 10^6$	$-.17 \times 10^5$
	" " B	310,455	357.3	-12.4	-13.8	$.13 \times 10^8$	$.24 \times 10^7$	$.14 \times 10^8$	$.30 \times 10^7$	$-.13 \times 10^5$	$-.74 \times 10^4$
	" " C	239,510	392.7	116.6	- 7.3	$.78 \times 10^7$	$.18 \times 10^7$	$.89 \times 10^7$	$.18 \times 10^7$	$.78 \times 10^5$	$.48 \times 10^5$
	" " D	207,610	423.8	216.0	- 8.5	$.41 \times 10^7$	$.14 \times 10^7$	$.50 \times 10^7$	$.76 \times 10^6$	$.90 \times 10^5$	$.86 \times 10^5$

Space Division
North American Rockwell

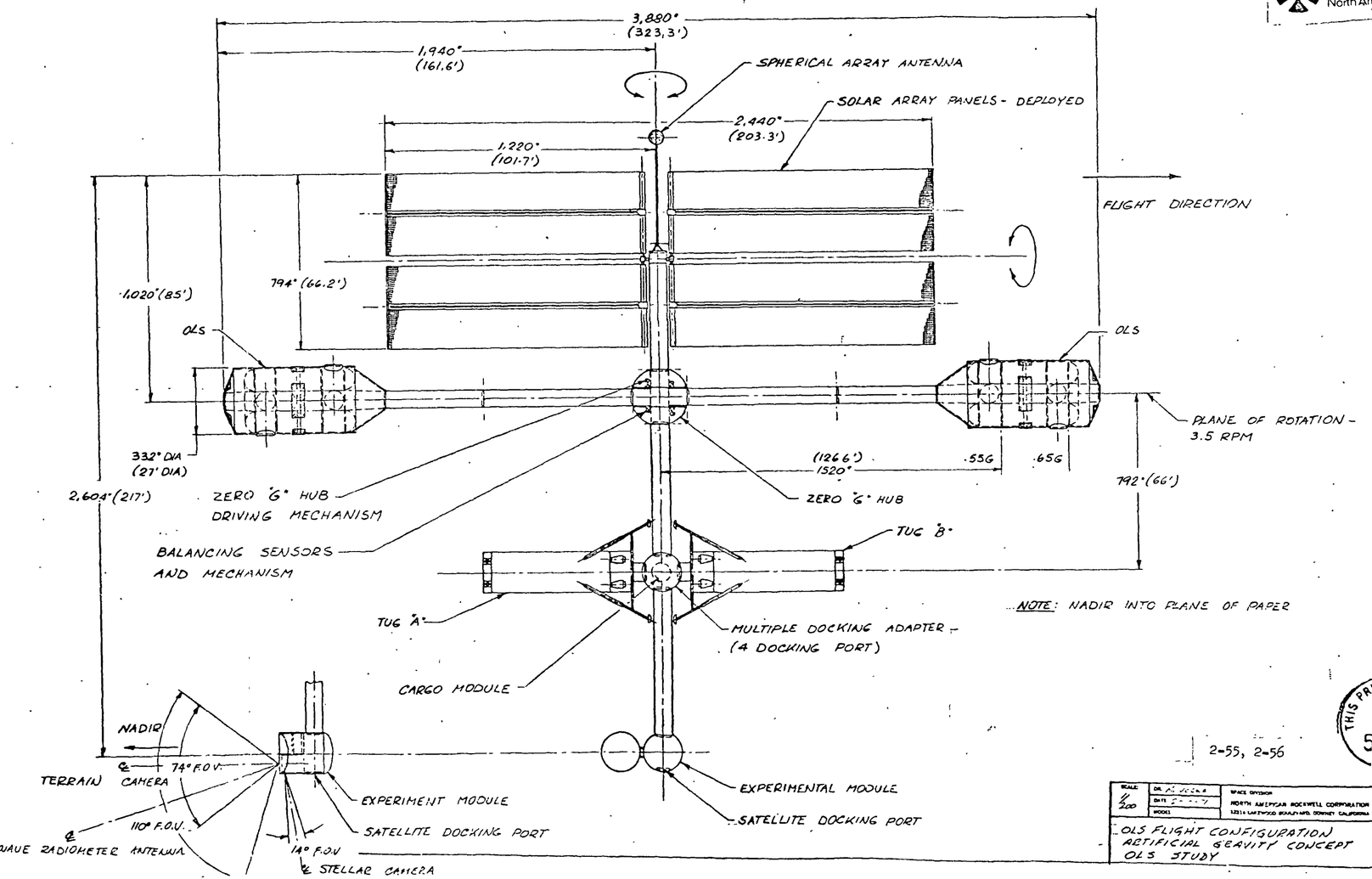
2.3 ARTIFICIAL GRAVITY CONCEPT

A final task performed during the conceptual design synthesis of the 27-foot diameter single module Orbital Lunar Station study was the investigation of a feasible conversion of the baseline zero g system into one providing a low level gravity environment for the crew. Drawing 2282-28 illustrates the major configurational characteristics of the artificial gravity concept. The concept basically involves counter-balancing the OLS with another OLS or some optional counterweight transported to lunar orbit. The OLS, rotating at a rate of 3.5 rpm at a radius distance of 1220 inches will provide 0.55 g at the forward deck level and 0.65 g at the aft deck level. As shown, the artificial-g concept utilizes four rigid fixed length tubes, 6 feet in diameter, each equipped with docking systems, to connect the OLS and its counter balance. This boom could be made to telescope, however, which would allow the g level within the OLS to be varied. A necessary new element in the OLS concept is the zero g hub. This two section module incorporates docking ports, and the large diameter bearing system and driving mechanism to permit rotating the zero g hub modules counter to the rotation of the OLS. One of the docking ports is used by the normal power boom module, to which the rollout solar array panels are attached. It was considered necessary to position the solar array system on the despun axis of rotation because of the lack of structural capability to sustain any significant g loading in the deployed configuration. The second docking port on the zero g hub is used by the other major elements of the OLS. In order to dock the tug/landers to the OLS, it is necessary to provide a non-rotating docking interface. This has been accomplished by providing a multiple docking adapter at a distance of 6 feet from the zero g hub. The MDA is attached to the zero g hub by a rigid, fixed length, 6-foot diameter tube docked into the zero g hub. The MDA docking ports accommodate the two tug/landers and the cargo module as shown.

Positioned further out on the axis of rotation is the baseline OLS end viewing experiment module. The module is structurally supported from the MDA by use of a 53-foot long, 6-foot diameter rigid tube. The positioning of the experiment module 131.5 feet from the zero g hub is necessary to insure that the rotating OLS core module clears the field of view of the terrain camera located in the experiment module.

The orbital flight orientation of this OLS concept features the axis of rotation normal to the orbit plane. This is desirable to minimize stabilization impulse requirements. A review was made of the OLS core module to determine what changes would be necessary for the artificial gravity derivative. Since the OLS core module features transverse floors with normal head/toe internal arrangement of furnishings on the four decks, and during the surface to earth orbit transportation phase, the longitudinal g loading may be in the vicinity of four to five, no structural or internal arrangements are considered necessary to the baseline OLS to permit its operating in the artificial gravity mode. In the fully deployed operational configuration, the artificial gravity OLS station has a maximum rotating diameter of 323.3 feet, and a width of 217 feet along the spin axis.

Space Division
North American Rockwell



THIS PRINT REDUCED 50%

2-55, 2-56

SD 71-207

SCALE 1/200	DR. A. JACKS	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA
	DATE	
	MODEL	

OLS FLIGHT CONFIGURATION
ARTIFICIAL GRAVITY CONCEPT
OLS STUDY

2282-28

2.4 DELIVERY TO LUNAR ORBIT

Table 2-4 indicates a total on orbit weight of the OLS of 157,605 pounds. A fully loaded tug weighs 84,615 pounds. Therefore, the total weight to be delivered to lunar orbit to initiate OLS operations is 242,220 pounds. The three baseline Cislunar Shuttles (CLS) that were considered for delivery of the OLS have the following payload characteristics.

	Payload Out (Pounds)	Payload Back
Reuseable Nuclear Shuttle (RNS)		
Round Trip	147,000	-0-
One Way	330,000	Not Applicable
Single Chemical Propulsion Stage (CPS-1)		
Round Trip	93,000	-0-
One Way	273,000	Not Applicable
Double Chemical Propulsion Stage (CPS-2)		
Round Trip	350,000	-0-

One Step Delivery

Figure 2-8 presents a delivery concept using only one cislunar flight. The total payload of the OLS assembly is well within the round trip capability of CPS-2. However, both the RNS and CPS-1 must be "expended" (refueling required in lunar orbit) to deliver the OLS assembly. A second CLS flight is required for both the RNS and CPS-1 cases to recover them from lunar orbit.

Two Step Delivery

Figure 2-9 presents a two step approach for delivery of the OLS assembly to lunar orbit. On the first CLS flight, the OLS core-module power boom and consumables are delivered. The total weight is 145,025 pounds. On the second CLS flight, the experiment module, 8 man crew, and fully loaded space tug are delivered. The total weight on this flight is 97,215 pounds. The much smaller payload requirement on the second CLS flight permits much greater flexibility in the selection of translunar injection opportunities. Section 2.0 of Volume II showed that opportunities existed approximately every nine days if reduced CLS payloads are acceptable.

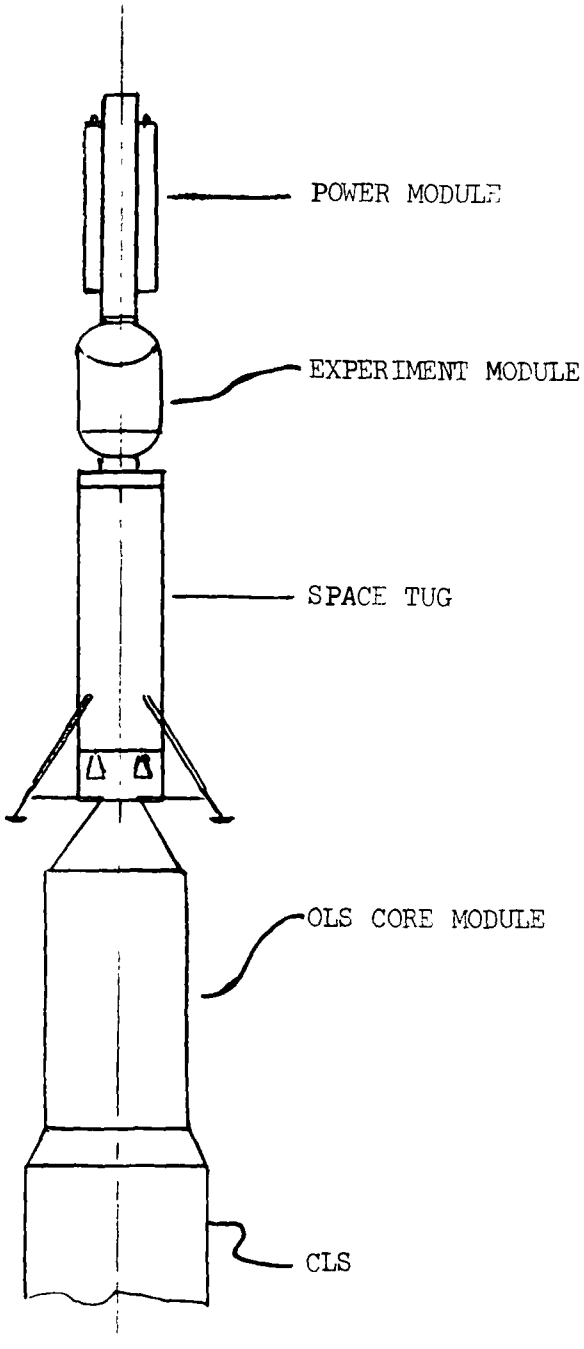
Only the RNS is applicable for the two step delivery approach. CPS-1 payload capability is less than required on each of the flights. CPS-2 payload capability is excessive for either of the flights.

Conclusion

There are four possible techniques to deliver the representative OLS to lunar orbit:

1. One CPS-2 round trip flight

2. One CPS-1 expended flight



(approximately 242,220 pounds payload)

Figure 2-8. One Step Delivery

3. One RNS expended flight

4. Two RNS round trip flights

In this study there has been no attempt to compare or evaluate the development or operational characteristics of the CPS or RNS cislunar shuttles. The only criterion used in selection of a baseline CLS is to minimize logistics payload to earth orbit. Therefore, for purposes of this study the RNS has been constantly used as the baseline CLS, because the payload to propellant ratio is greater for the RNS (See Section 7.0 of Volume II). Option 4 is selected as the baseline delivery concept because it does not require a long quiescent storage operation of the RNS in lunar orbit which may be the case in Option 3.

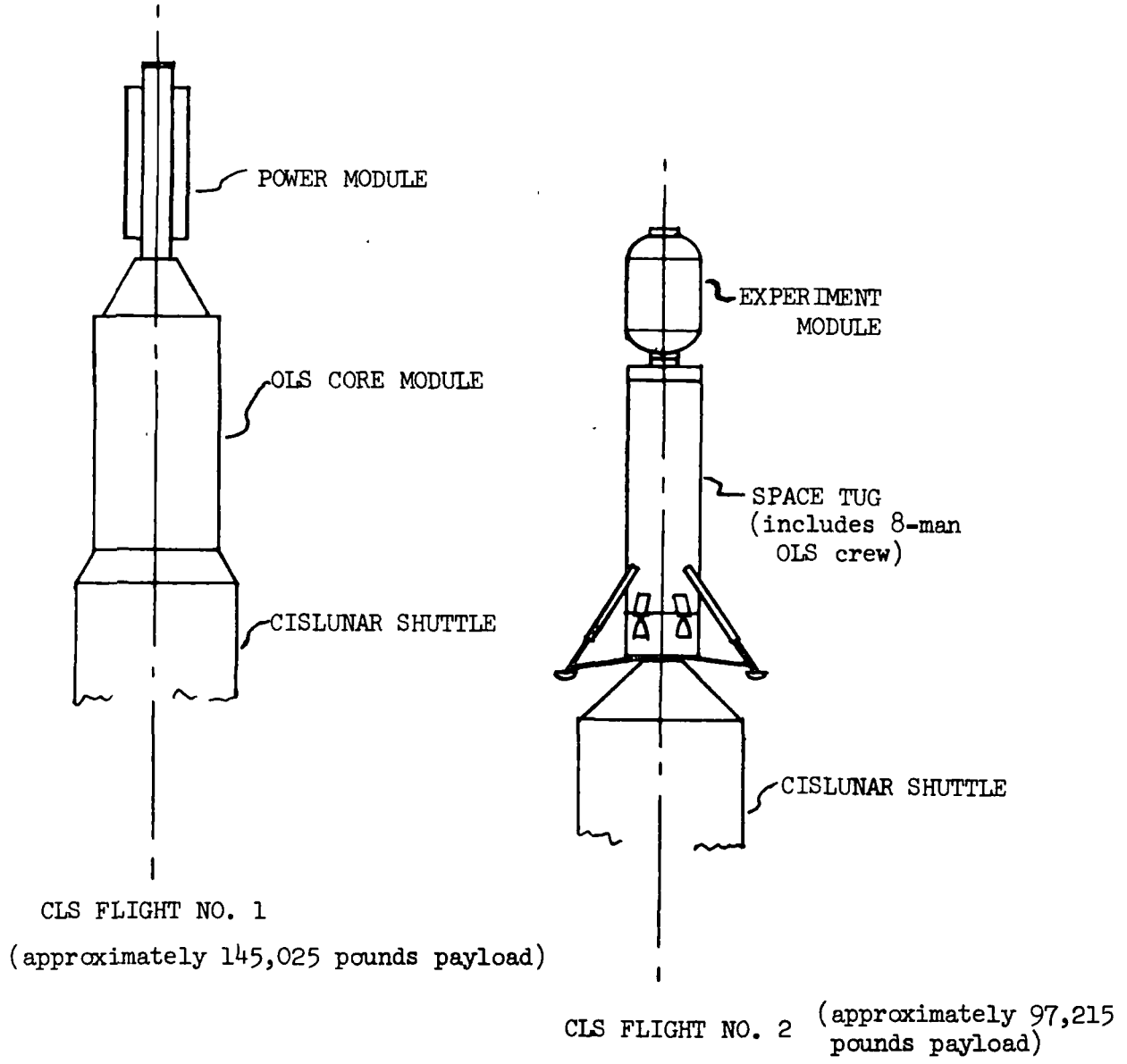


Figure 2-9. Two Step Delivery Concept

3.0 REPRESENTATIVE OLS SUBSYSTEM DEFINITION

Selected representative OLS subsystem concepts are described and defined in this section. These concepts were selected based on the parametric data presented in Volume IV. In some cases where EOSS and OLS requirements are similar and where development risk is not an issue the same subsystem concepts selected for EOSS use are recommended for the OLS.

The selected representative OLS subsystem concepts are described in detail in sections 3.1 through 3.6. A summary and comparison of the more significant representative OLS subsystem concept selections with EOSS subsystem concepts is presented in the following paragraphs.

The selected OLS Environmental Control/Life Support Subsystem (ECLSS) equipment is similar to that selected for the EOSS except that for the OLS a laundry and a high pressure emergency supply nitrogen system are required.

The OLS Electrical Power Subsystem (EPS) is basically identical to that of the EOSS. A 10,000 square foot solar array is used for primary power. However, on the OLS a regenerative fuel cell energy storage concept is employed rather than batteries and battery chargers.

The OLS Information Subsystem (ISS) is very similar to that of the EOSS except that for OLS external communications phased array antennas are employed rather than parabolic dishes.

The OLS Guidance and Control (G&C) subsystem concept, which employs control moment gyro and reaction control jets, is similar to that of the EOSS. The OLS G&C however requires the addition of a landmark tracker and a radar altimeter to satisfy the more demanding lunar navigation requirements. OLS science requirements require maintenance of a local level flight mode. Therefore, the CMG size and Reaction Control System (RCS) impulse requirements are based on this flight mode. Various combinations of experiment module, cargo module and tug placement on the core module were analyzed for the various possible local level flight modes. The following factors were considered in the selection of the prepared flights mode.

1. Conformance with experiment viewing requirements.
2. Provisions for adequate heat rejection capability.
3. Operational considerations which require that tugs be docked to separate pressure volumes.
4. Minimization of CMG size and weight and RCS impulse requirements.

The selected mode is the vehicle x axis perpendicular to the orbital plane, the vehicle y axis is the direction of flight, and the z axis is the direction of nadir. An oxygen-hydrogen RCS was selected for the OLS. The OLS RCS is thus very similar to the EOSS RCS except that OLS cryogenic storage requirements are greater.

The OLS Environmental Protection Subsystem (ENPS) requirements are identical to those for the EOSS. However the difference in earth orbit and lunar orbit natural environments results in some OLS ENPS changes. The nuclear radiation during solar flare events is much higher in lunar orbit necessitating the inclusion of a shielded solar flare shelter on the OLS. This is accomplished primarily with water filled panels supplemented with stored food, equipment placement, and some high density dedicated shielding. The water in the radiation shield (16,000 pounds) also serves as thermal capacitance in the OLS thermal control system. The high lunar infrared radiation makes heat rejection more difficult in lunar orbit than in earth orbit. The worst case is when the OLS is at the subsolar point. For the subsolar case heat rejection requirements are found to exceed heat rejection capability. This necessitates use of thermal capacitance so that heat can be stored and rejected later during the dark portion of the orbit. By employing 2000 square feet of active fluid loop radiators and allowing for thermal coating degradation to an α_s/ϵ value of 0.3 the 16,000 pounds of water in the radiation shield serving as thermal capacitance will not fluctuate more than a few Farenheit degrees from its mean temperature value. Meteroid bumpers of comparable thickness and concept employed on the EOSS are found to be satisfactory for the representative OLS.

3.1 ENVIRONMENTAL CONTROL/LIFE SUPPORT SUBSYSTEM

The environmental control and life support subsystem (ECLSS) is a utility subsystem that provides a habitable environment and the elements necessary to sustain the life and comfort of the crew where required within the OLS.

3.1.1 CONCEPT DEFINITION

The ECLSS consists of nine assemblies (Table 3.1-1) with their subassemblies and subassembly concepts.

Atmospheric Storage

The atmospheric storage assembly is composed of the conditioning and distribution equipment required to supply nitrogen (N_2) and oxygen (O_2) from the subcritical cryogenic storage assemblies which are part of the reaction control subsystem (RCS) cryogenic tank farm. Vehicle repressurization requirements are supplied from the cryogenic stores. In addition, a pumpdown system is provided to minimize gas losses from airlock, power boom, and certain experiment-module operations. The pumpdown system pumps to 1.0 psia and includes a 1000-psia reservoir.

The ECLSS atmospheric storage gas requirements, as cryogenic gases, are tabulated in Table 3.1-2.

Atmospheric storage interfaces with special life support and with RCS as noted above. Since oxygen is recovered by CO_2 management equipment, this assembly interfaces with the atmospheric storage unit. The pumpdown system interfaces with the configuration subsystem.

CO_2 Management

The CO_2 management assembly consists of carbon dioxide (CO_2) removal and CO_2 reduction (to produce oxygen).

The concept selected for CO_2 removal is the hydrogen depolarized cell. This unit is run on the sun side of the orbit to reduce power requirements during the dark side of the orbit. The H_2 depolarized cell consists of a concentration unit and an electrolysis cell. The concentrator utilizes O_2 and H_2 to remove CO_2 from the air. Water is generated from the reaction so that the electrolysis segment is a closed cycle.

CO_2 and a small quantity of oxygen are absorbed from the air passing through the cells into an alkali carbonate electrolyte matrix. Several ionic species occur within the electrolyte, but the overall effects are: (1) absorbed CO_2 is transported across the electrolyte and evolves again as a

Table 3.1-1. Selected ECLSS Concepts

Assembly Subassembly	Selected Concept
Atmospheric storage pumpdown	Two 4-stage compressors and 1000 psia reservoirs
Gas storage	Cryogenic (see RCS, Section 3.5) Emergency supplies are high pressure
CO_2 management	
CO_2 removal	H_2 depolarized cell
CO_2 reduction	Sabatier with CH_4 dump and solid polymer electrolysis
Atmospheric control	Centralized heat exchangers and humidity control Localized fans for comfort Regenerable charcoal
Active thermal control	Cabin water loop Freon radiator loop Freon boom loop
Water management	Multiple system
Potable water	Vapor compression
Wash water	Reverse osmosis
Waste management	Three toilets with urinals Two wall urinals Two drying chamber/compactors
Hygiene	4 sinks 2 showers Laundry
Food management	Freezer, refrigerator Ovens Fresh, frozen, freeze-dried foods
Special life support	
IVA	Umbilical loop - fluid flow
EVA	PLSS
Fire extinguisher	CO_2 gas

Table 3.1-2. ECLSS Cryogenic Gas Requirements
(Stored by RCS)

Oxygen		
Metabolic		1.6 lb/day
Seal leakage		4.9
Venting loss		0.19
Pumpdown loss		0.14
	Total	6.83 lb/day
Repressurization	350 lb (20,000 cubic feet)	
Nitrogen		
Seal leakage		16.1 lb/day
Venting loss		0.6
Pumpdown loss		0.5
	Total	17.2 lb/day
Repressurization	1150 lb (20,000 cubic feet)	
Hydrogen		
Sabatier makeup		1.8 lb/day

gas at the anode, and (2) hydrogen gas (from the water electrolysis segment) sweeps across the anode, where part of it reacts with the oxygen transported across the electrolyte and the remainder mixes with the evolving CO_2. Thus, the concentrate gas stream passing out of the cell modules and into the CO_2 reduction reactor contains hydrogen, CO_2, and a small quantity of water vapor. Generated heat and water vapor are rejected to the process air stream. The water vapor is eventually removed in a condenser and ultimately electrolyzed to produce the oxygen and hydrogen necessary to operate the concentrator.

The concept selected for reduction is Sabatier with CH_4 dump. This unit requires a water electrolysis unit and the solid polymer concept was selected. Since the exhaled CO_2 only contains 89 percent (by weight) of the required metabolic oxygen, atmospheric storage must make up the 11 percent difference as well as oxygen loss through leakage.

The Sabatier reduction unit receives CO_2, and with hydrogen reduces it to CH_4 and H_2O. The H_2O is electrolyzed to hydrogen and oxygen. The hydrogen is returned to the Sabatier reactor while oxygen goes to the cabin.

The H_2 depolarized cell operates on bleed air from the centralized heat exchanger and condenser assembly in atmospheric control. The water electrolysis unit returns oxygen to the vehicle via the pressure control subassembly in atmospheric control.

Atmospheric Control

The atmospheric control assembly consists of subassemblies for control of trace contaminants, humidity, temperature, pressure, and circulation.

The trace contaminant control subassembly removes gaseous trace contaminants, aerosols, particulates, and bacteria from the cabin atmosphere. It consists of a debris trap, filters, ammonia sorbent bed, acid gas sorbent bed, a catalytic oxidizer, and regenerable charcoal beds.

Atmospheric pressure control and composition is maintained by the pressure control subassembly. It consists of a total pressure sensor which controls the delivery of O_2 and N_2 makeup gas from atmospheric storage and an O_2 partial pressure sensor and timer which controls the time during which the electrolysis unit operates.

The humidity control subassembly and the thermal control subassembly control the cabin temperature and the humidity level in the vehicle. Their functions are provided by two identical packages consisting of fans, sensible heat exchanger, condensing heat exchanger, a water separator, a check valve, temperature control valves, and fan bypass valves.

The circulation subassembly consists of additional fans to provide ventilation flows in all cabins.

Vehicle configuration has a strong interface with atmospheric control. Heat leaks through the structure and the ratio of air and coldplate thermal loads also affect atmospheric control.

Active Thermal Control

The active thermal control assembly consists of a radiator heat rejection loop, a cabin thermal loop, and a power boom loop. The radiator loop is a freon circuit that removes heat from the cabin thermal loop in the freon/water interface heat exchangers and rejects the heat to space by means of the vehicle radiator. The cabin thermal loop is a water circuit that removes heat from all equipment requiring heat rejection and transfers this heat to the radiator loop in the interface heat exchangers. The power boom loop transfers heat from the regenerative fuel cells mounted within the power boom to radiators mounted on the exterior of the power boom. The non-regenerative fuel cells mounted in the upper torus also utilize this radiator loop. The radiators are described more fully in Section 3.6.2.

Water Management

The water management assembly consists of subassemblies for processing urine, wash, and experiments water; storage of processed water; and maintenance of sterility.

The wash and condensate recovery subassembly processes wash water from showers and sinks in addition to condensate from the humidity control subassembly in a reverse osmosis unit. Reclaimed water is transferred to the storage subassembly while rejected wash water liquor is sent to the potable recovery subassembly. The potable recovery subassembly processes urine, dishwasher water, and reverse osmosis rejected brine in rotating vapor-compression stills. Reclaimed water is transferred from this subassembly to the storage subassembly. Water from all experiments is processed by the central vapor compression reclamation subassembly.

The vapor compression unit operates by evaporating the waste waters under low pressure and temperature and then recondensing the vapor. The latent heat released by condensation is used in evaporating the waste water. The reverse osmosis unit operates by pushing waste wash water through a porous membrane which physically removes particles larger than the membrane pore size. Reverse osmosis cannot be used to recover urine because of some of the constituent organics.

The clean water storage subassembly consists of separate storage tanks for wash and potable water. This water is maintained at 160 F in all storage tanks for bacteria control through pasteurization. In addition, silver ions are added to the water in the distribution lines to retard bacteria.

Vehicle configuration as to bathroom and sink locations influence the water management assembly in terms of plumbing and processing equipment location. Water processing near the waste sources (mainly bathrooms) is desired since long sewer runs could present toxic problems if the plumbing lines leak.

Waste Management

The waste management assembly consists of subassemblies for collection of urine, collection and processing of solid wastes (feces and trash), and disposal of wastes.

The urine collection subassembly consists of urinals that collect and deliver urine to potable water recovery subassemblies. Feces are collected and vacuum-dried in the fecal collection and processing tanks. All wet trash is sterilized at 250 F and vacuum-dried, compacted for interim storage and then disposed of by removal to the lunar surface or earth.

Hygiene

Hygiene equipment includes the full body shower, hand-and-face-wash sink, provisions for medical and dental support, and equipment to support general station housekeeping. The shower is an air-stream type with water recirculation capability. The sink is an enclosed chamber type with air entrainment. The personal hygiene facility sinks include water supply attachment for dental care and an expectorate tube. Portable vacuum cleaners are used for debris collection. A clothes laundry is also included.

Food Management

The food management assembly consists of subassemblies for food storage, food preparation, serving and cleanup, and inventory control. A compression-cycle food freezer is provided for storage. A refrigerator to aid in preparation of cold foods and for short-term storage is also provided. Food preparation equipment is based on a flexible food supply that ranges from food heating to actual cooking and baking. The primary preparation equipment is a resistance heating oven and a microwave oven. The serving and cleanup subassembly consists of reusable trays and an automatic dishwasher. Inventory control is accomplished by a control panel to the station ISS.

Special Life Support

The special life support assembly consists of subassemblies for high pressure oxygen and nitrogen supply, PLSS and IVA support, and fire detection and control. The high pressure gases are required for emergency reserves storage, for IVA suit support, and for PLSS recharge. The pressure suits and umbilicals are in crew provisions under the hygiene assembly. Fire control consists of CO_2 extinguishers and cabin atmosphere dump provisions.

Emergency oxygen and nitrogen are stored in high pressure tanks at 3000 psia. Table 3.1-3 itemizes the uses of the high pressure storage. The IVA gas is for the umbilical suit loop. The EVA gas is for recharging the PLSS for EVA.

Table 3.1-3. ECLSS High Pressure Gas

Oxygen		
Metabolic	810 lb	
Leakage	287 lb	(all types)
IVA and EVA	66 lb	
Total	1163 lb	
Nitrogen		
Leakage	945 lb	(all types)

3.1.2 SELECTION RATIONALE

The major selection rationale for all systems are listed in Table 3.1-4 in priority order. The major emphasis was placed upon resupply weight, heat rejection, and power. In cases where these three items were not the deciding factor, other rationale were utilized. In most cases durability was not a governing factor because the ratings were very close. Cost comparisons were only relative and are extremely dependent upon research and technology studies either currently in progress or tentatively planned.

Table 3.1-4. Selection Rationale (Priority Order)

Resupply weight Fixed weight Heat rejection Power Durability* (MTBF, scheduled and unscheduled maintenance) Development status* Commonality* Relative cost* Volume* *Priority of these items very close

Atmospheric Storage

Repressurization and Gas Make-Up

The options considered were cryogenic O_2/N_2 storage, high pressure O_2/N_2 storage, and (for nitrogen) ammonia or hydrazine (N_2H_4) storage. From the parametric curves in Section 3.1.2 of Volume IV (Figures 3.1-3 and 3.1-4), it can be seen that the cryogenic N_2 system is slightly lighter and less complex than the hydrazine N_2 system. Also the option of utilizing H_2 boiloff from docked tugs minimizes the need for generating H_2 from N_2H_4 or NH_3 (ECLSS requires H_2 for its CO_2 reduction unit. Some H_2 is stored cryogenically to eliminate total dependence on tug boiloff.) Thus, the ammonia and hydrazine systems for N_2 storage are eliminated. Cryogenic N_2 is selected. This selection also makes OLS cryogenic storage common with EOSS and the space tugs and allows integrated storage with RCS.

Table 3.1-3 of Volume IV presents the penalties involved in the selection of cryogenic and high pressure O_2 and N_2. Table 3.1-5 of this volume presents the densities of these two concepts. Comparison of the data of these two tables shows that for the major portion of gas storage, cryogenic storage is preferrable over high pressure storage on a tank weight and volume basis. Thus, as much gas as possible is stored cryogenically. All makeup gases advantageously can be stored cryogenically due to the continuous usage requirement.

Table 3.1-5. Oxygen and Nitrogen Densities

Condition	High Pressure (3000 psia)		Cryogenic (subcritical)	
Gas	O_2	N_2	O_2	N_2
Density (lb/ft^3)	18	15	57	40

Repressurization gases also can be stored cryogenically in the integrated RCS system. Table 3.1-6 lists the amounts of gas required for leakage and repressurization.

Table 3.1-6. Gas Requirements

Type	Rate (lb/day)	109-Day Requirement (lb)
Oxygen		
Metabolic[1]	1.60	174
Leakage	5.23	570
Repressurization[2]	-	350
Emergency[3]	-	598
IVA/EVA	-	66
Nitrogen		
Leakage	17.20	1865
Repressurization[2]	-	1150
Emergency[3]	-	515
Hydrogen[4]	1.8	196

1 Make-up of O_2 not obtained from CO_2 reduction
2 For 20K cubic feet, 3.1 psia partial pressure O_2, 14.7 psia total pressure
3 For 8 men and 30 days - metabolic and leakage
4 For CO_2 reduction unit

Emergency Gas Supplies

The emergency gas supplies consist of contingency oxygen and nitrogen storage for leakage and metabolic purposes. Also, oxygen for IVA and EVA are included. These gases are stored under 3000 psia pressure at 70 F.

The emergency, or contingency, O_2 and N_2 are stored under high pressure for two major reasons. First, the gas is rapidly available in any desired flow rate. No heater has to be used to force the gas out of the tank as is the case with cryogenic gases (when the desired flow rate exceeds that provided by normal boil-off). Second, the gas normally is not used. To store it cryogenically would incur resupply due to boil-off losses.

The IVA and EVA gases are stored with the contingency O_2 gas because the EVA backpack and IVA umbilical system require a relatively high pressure (approximately 1000 psia) source to fully recharge the EVA O_2 tank and to overcome the IVA system pressure drops. The IVA/EVA gas is only 66 pounds, approximately 10 percent of the total high pressure O_2. By putting the IVA/EVA O_2 with the high pressure oxygen, a compressor is not required.

Pumpdown Systems

Figures 3.1-5 and 3.1-6 in Volume IV indicate that for the EOSS type pumpdown system, approximately 5 to 10 cycles per month are required to justify a pumpdown system. The OLS has a frequency of approximately 5 cycles per month. Table 3.1-7 presents the item and frequency of pumpdown along with the gas loss with and without pumpdown. The weight and power of the pumpdown system is presented in Table 3.1-8. Table 3.1-9 presents the actual trade of no pumpdown versus a pumpdown system. Parametric curves on pumpdown systems are difficult to generate because the pumpdown system is sized by the pressure to which a volume is reduced; the pressure the reservoir is pumped up to, or whether the reservoir is a specific size or infinite; and the volumetric flow rate through the pump. Thus, each specific pumpdown case must be sized separately. The sizing criteria for OLS was 1200 cubic feet pumped from 14.7 psia to 1.0 psia in 24 hours with the reservoir reaching 1000 psia.

From Table 3.1-9 it is seen that the pumpdown system is an obvious choice, especially considering the penalties on resupply to lunar orbit. Therefore, the major rationale for selection of the pumpdown system is resupply weight. The pumpdown system is similar to that on the EOSS and MSS. The system pumps a given volume to 1.0 psia and pumps the gas to a 1000 psia reservoir. The pump is a four-stage water-cooled pump. The OLS pumpdown system consists of two reservoir and two pump units (one per pressure volume). Thus, two 1200-cubic foot volumes can be pumped.

Table 3.1-7. Gases Subject to Pumpdown

Item	Frequency Cycles/Mo	Gas Loss (lb/mo)		Gas Loss (10 yr)	
		Without Pumpdown	With Pumpdown	Without Pumpdown	With Pumpdown
Docking port	2.0	0.574	0.039	68.9	4.7
Boom	0.08	7.20	0.490	864.0	58.9
EVA airlock	0.11	1.40	0.095	168.0	11.4
Experiment module	3.0	269.00	18.30	32,200.0	2,200.0
Total	5.19	278.17	18.92	33,300.0	2,275.0

Table 3.1-8. Pumpdown System Weight and Power Requirement

Pump	55 x 2 = 110
Plumbing	35 x 2 = 70
Reservoir	90 x 2 = 180
Total	360 lb
Power	600 watt

Table 3.1-9. Pumpdown System Trades

10 Years - Weight (1 lb/watt Power Penalty)		
Item	Without Pumpdown	With Pumpdown
Gas weight (lb)	33,300	2,275
Hardware weight (lb)	0	360
Power penalty (lb)	0	600
Total	33,300	3,235

CO_2 Management

CO_2 Removal

The options considered for removal of carbon dioxide were molecular sieve, steam desorbed resin, solid amine, and hydrogen depolarized cell. Utilizing the parametric curves shown in Volume IV, the weights, power, heat rejection, and durability penalties were compared. Table 3.1-10 presents this comparison with an estimated relative cost and present development status. (Figure 3.1-65 in Volume IV, presents the availability of the concepts. All concepts could be available by the OLS flight date of 1983.)

Heat rejection and power criteria eliminate the steam desorbed resin and the solid amine concepts. Heat rejection and durability weigh the scales in favor of the H_2 depolarized cell over the molecular sieve. Also, the H_2 depolarized cell may be used on the Modular Space Station (MSS). Thus, the OLS selection is the hydrogen depolarized cell for CO_2 removal.

CO_2 Reduction

CO_2 reduction cannot be considered without also discussing water electrolysis, since all but one of the reduction concepts require water electrolysis to produce oxygen. Therefore, a short discussion on water electrolysis is presented before moving to the actual CO_2 reduction units. Table 3.1-11 presents weight and power for the water electrolysis concepts. Only metabolic oxygen requirements were used in sizing the units. No make-up water was added to compensate for leakage or for the small portion of metabolic oxygen not contained in the exhaled carbon dioxide. The rationale for this decision will be discussed shortly. Table 3.1-11 illustrates that of all the electrolysis concepts, wick-feed has the least weight, low power, and least heat rejection. However, all of the concepts are very close in these characteristics, except for the cabin air and circulating electrolyte concepts. The remaining three concepts can be said to be approximately equal. The durability ratings are, also, approximately equal though solid polymer leads by two points.

After these water electrolysis concepts were analyzed, the solid polymer concept was selected for the following reasons. First, this concept is slightly better in durability. Second, new data indicates that the solid polymer concept is less sensitive to the environment (such as dissolved gases in the water) than the gas circulation and wick-feed concepts. Third, it may replace wick-feed on the MSS. Therefore, in the comparisons of the CO_2 reduction units, the solid polymer water electrolysis concept will be utilized.

The concepts considered for CO_2 reduction were the Sabatier reactor with methane (CH_4) dump (requiring hydrogen make-up), Sabatier without methane dump (add cracker unit to crack CH_4 to C and H_2), Bosch reactor, and solid electrolyte reactor. All of these CO_2 reduction concepts, except solid electrolyte, require a water electrolysis unit. As the solid electrolyte concept does not require a water electrolysis unit, it cannot be used to make up oxygen leakage. This is why the comparison of electrolysis concepts were based upon generation of metabolic oxygen only. Table 3.1-12 compares the CO_2 reduction concepts. The water electrolysis concept utilized is solid polymer

Table 3.1-10. CO_2 Removal Concept Comparison, 8 Men, 10 Years, 5 mm H_g $PPCO_2$

Characteristic \ Concept		Molecular Sieve	Steam Desorbed Resin	Solid Amine	H_2 Depolarized Cell
Weight (lb)	Fixed and spares	604	432	623	454
	Resupply	0	0	0	0
Power (watt)		550	970	1850	650
Heat rejection (watt)		776	1400	1860	73
Durability*		21	20	21	24
Relative cost		Low	Low	Low	Low
Development status		State-of-art	Lab Prototype	Lab Prototype	Lab Prototype

*The maximum rating is 25 points, and the higher the rating, the better the durability.

Space Division
North American Rockwell

Table 3.1-11. Sunside Water Electrolysis Comparison, 8 Men - 10 Years

Characteristics \ Concept	Wick-Fed	Gas Circulation	Solid Polymere	Cabin Air	Circulating Electrolyte
Weight (lb) Fixed and Spares	293	307	352	468	646
Weight (lb) Resupply	0	0	0	0	0
Power (watt)	4000	4200	4000	6000	4000
Heat rejection (watt)	1410	1410	1525	3280	1700
Durability	18.5	18	20.5	18	20
Relative cost	Medium	Medium	Medium	Medium	Medium
Development status	Prototype	Prototype	Prototype	Prototype	Prototype

Table 3.1-12. CO_2 Reduction Concept Comparison, 8 Men - 10 Years (18 lb/Day CO_2)

Characteristic \ Concept	Sabatier W/CH_4 Dump and Electrolysis	Sabatier W/O CH_4 Dump and Electrolysis	Bosch and Electrolysis	Solid Electrolyte
Weight (lb) Fixed and Spares	452	669	589	670
Weight (lb) Resupply	6720* (260**)	4400	4400	5400
Power (watt)	4050	5100	4250	4000
Heat rejection (watt)	2155	2345	2916	1925
Durability	20	13	15	14.5
Relative cost	Low	High	Medium	Medium
Development status	State-of-art	Feasibility	Lab prototype	Lab prototype

*Resupply weight with H_2 included but no tank penalty
**Resupply weight without H_2

Space Division
North American Rockwell

sized for sun side operation. To compare the Sabatier and Bosh concepts (plus sun side water electrolysis) with the solid electrolyte, the solid electrolyte concept also had to be adjusted to work for the sun side portion of the orbit only.

The data in Table 3.1-12 indicates that the Sabatier with CH_4 dumping is least in weight and cost, and best in durability and development. Note that the resupply weight decreases if a "free" source of hydrogen is available. Such a source of hydrogen could be H_2 boil-off from docked tugs. Only 1.8 pounds per day of H_2 is required by the Sabatier reactor. One tug will boil off 24 pounds per day of H_2. Thus, Sabatier with CH_4 dumping is selected. Some cryogenic H_2 must be stored, however, to allow for times that tug H_2 boil-off is not available.

Atmospheric Control

Circulation, Temperature-Humidity, Pressure Control

The tentative configuration of the OLS is very similar to the EOSS. Because of the similarities in configuration, the analyses performed for the EOSS are applicable to the OLS. Thus, the OLS has local ventilation (circulation) fans for the crew comfort requirement of 40 feet per minute while temperature, humidity, and pressure control are centralized. The rationale for this division is that centralization reduces complexities of plumbing heat exchangers distributed throughout the OLS while the removal of the ventilation requirement from the central system reduces duct size and required fan power. Typical flow rates for the various atmospheric control functions are tabulated in Table 3.1-13.

Table 3.1-13. Typical Flow Rates (lb/hr)

CO_2 removal	200 to 300
Trace gas and odor removal	300 to 500
Latent load (humidity)	700 to 1,400
Sensible load (temperature)	7,000 to 14,000
Ventilation (crew comfort)	30,000 to 70,000

Trace Contaminate Control

Trace contaminate control consists of trace gas and odor removal, bateria detection, and bacteria removal. The method of bacteria detection and removal consists of bacteria filters for removal of bacteria and an agar tape incubator detector.

The only trade performed in the atmospheric control assembly for the OLS was concerned with trace gas removal. The options were: non-regenerable charcoal plus catalytic oxidization, regenerable charcoal plus catalytic oxidization, and sorbent bed plus catalytic oxidization. Table 3.1-14 presents a quantitative comparison of the three concepts. The selection is regenerable charcoal plus catalytic oxidation because of the extensive experience available with charcoal, the small resupply weight, and commonality with the EOSS.

Table 3.1-14. Trace Gas Control Comparison, 8 Men - 10 Years

Characteristic \ Option	Non-Regenerable Charcoal Plus Cat. Oxidation	Regenerable Charcoal Plus Cat. Oxidation	Sorbent Bed Plus Cat. Oxidation
Weight (lb) Fixed and spares	65	182	98
Weight (lb) Resupply	12,400	190	1330
Power (watt)	140	490	180
Heat rejection (watt)	161	468	205
Durability	21.5	19	22
Relative cost	Low	Med. to high	Med. to high
Development status	State-of-art	Concept feasibility	Concept feasibility

Physical Characteristics

Characteristics for the atmospheric control assembly were derived from the parametric data in Volume IV. Table 3.1-23 presented at the end of this section lists the weight, power, and volume for this assembly for a crew of eight men.

Active Thermal Control

The active thermal control is composed of two major segments. The first is the cabin transport loop. The second is the heat rejection loop. The trade studies associated with active thermal control are discussed in detail in Section 3.6.2, as part of the Environmental Protection Subsystem. The results are briefly summarized below.

Cabin Heat Transport Loop

A water loop has been selected to transfer heat from the interior of the vehicle to the heat rejection loop. In addition a thermal capacitance heat sink is included. The heat sink consists of the water shield in the radiation shelter. Water was chosen for the cabin heat loop because it is non-toxic, has good heat transfer characteristics, can readily use the water of the radiation shelter as a heat sink, and is compatible with the heat rejection loop via a heat exchanger. Figures 3.1-1 and 3.1-2 present parametric data for the cabin water loop.

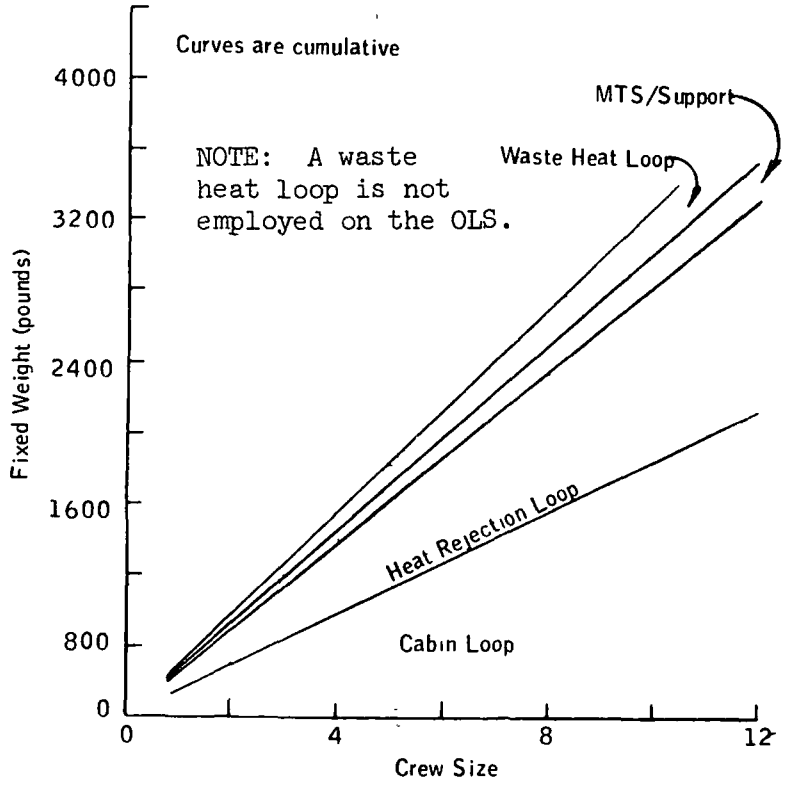


Figure 3.1-1. Active Thermal Control - Fixed Weight

Heat Rejection Loop

An active fluid loop radiation system was the selected concept. Figures 3.1-1 and 3.1-2 present data for that portion of the heat rejection loop which interfaces with the cabin water loop and the radiator. This loop utilizes freon as a working fluid because of its good temperature and viscosity characteristics. A separate freon loop will be used for the power boom radiators.

Water Management

Water Reclamation

OLS water requirements are summarized in Table 3.1-15. As shown, the rate of production of waste water totals 1.10 pounds per man-hour. In Volume IV, the Configuration and Systems Analysis Report, parametric trade data for all distillation units were based on a relative process rate of 0.618 pounds per man-hour. The revised subassembly trade data corresponding to the required OLS waste water process rate of 1.10 pounds per man-hour are shown in Table 3.1-16. Vapor compression is the selected concept based mainly upon its low power and heat rejection characteristics.

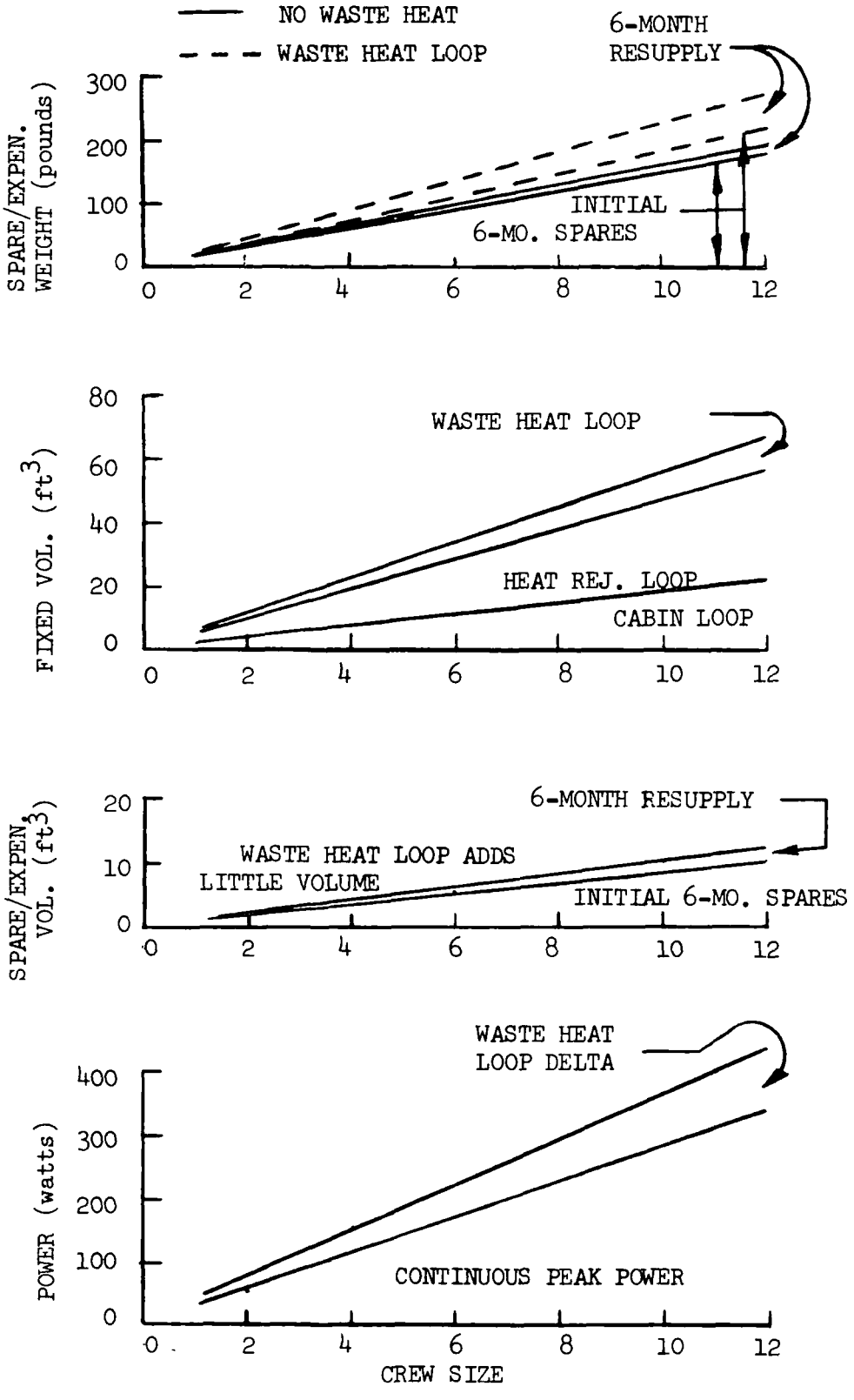


Figure 3.1-2. Active Thermal Control
(Note: Each Curve Set is Cumulative)

Table 3.1-15. Water Model

Fresh Waters Required		Waste Waters Produced	
Food & drink	5.36 lb/m-d	Urine	3.45 lb/m-d
Dishwashing	3.00	Dishwashing	3.00
Experiments	4.38	Experiments	4.38
Urinal flush	3.50	Urinal flush	3.50
Hand washing	4.00	Hand washing	4.00
Shower	4.92	Shower	4.92
		Condensate	3.09
Total	25.16 lb/m-d	Total	26.34 lb/m-d = 1.10 lb/m-hr

Dual System

Potable System		Wash System	
Urine Dishwashing Urinal Flush Experiments R.O. concentrate	17.42 lb/m-d or 0.727 lb/m-hr (24 hr/day)	Handwashing Shower Condensate	12.01 lb/m-d or 0.669 lb/m-hr (18 hr/day)

Figure 3.1-3 presents a schematic of a single and dual water system. As indicated in Table 3.1-15 the water requirements can be broken into potable waters and wash waters. A dual system results in an increase in overall water reclamation efficiency. Wash water can be processed at approximately 100 percent efficiency. An efficiency of 95 percent for the vapor compression unit (a conservative efficiency) was assumed in the comparison. The vapor compression unit processes significantly less water in the dual system concept, and, therefore, loses less total water than the single system concept. As indicated in Table 3.1-17 the dual system actually results in useable water surplus. The single system results in a net deficit of water. (Note that raising the vapor compression unit efficiency to the 97 to 98 percent range would result in water surplus for both concepts.) The selected concept is the dual system. It has lower fixed weight, less resupply weight (when considering water makeup), requires less power, and has increased recovery efficiency (water excess). Although the reverse osmosis unit cannot process urine waters, its light weight, small power requirement (as indicated by the parametric curves in Figure 3.1-50 of Volume IV), and high efficiency greatly enhance the dual system concept.

Table 3.1-16. Distillation Process Evaluation, 8 Men - 10 Years
(1.10 lb/Man-Hour Flow Rate)

Characteristic \ Concept	Air Evaporation	Vapor Compression	Vacuum Distillation	Vapor Diffusion
Weight (lb) Fixed and spares	548	1000	1080	796
Weight (lb) Resupply	3490	1350	640	1210
Power (watt)	2210	624	3290	2670
Heat rejection (watt)	2350	678	3440	3130
Durability	21	17	17	18
Relative cost	Low	Medium	Medium	Low
Development status	Lab prototype	Lab prototype	Lab prototype	Lab prototype

Space Division
North American Rockwell

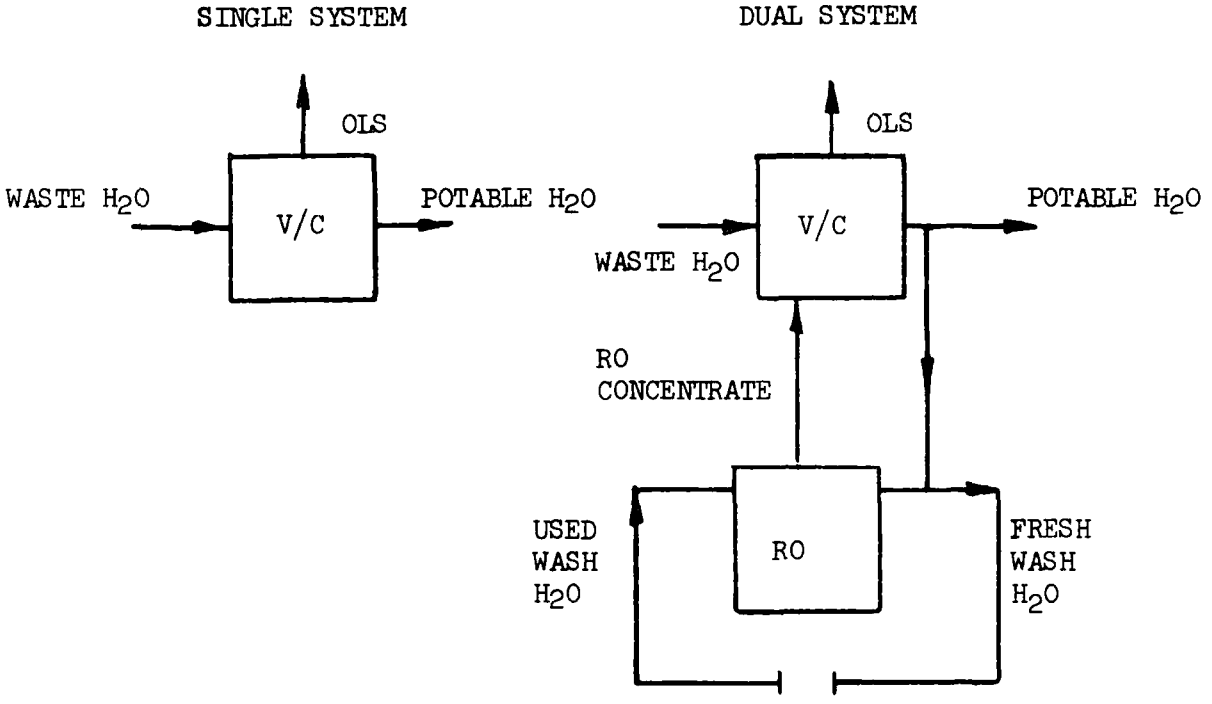


Figure 3.1-3. Single Vs. Dual System for Water Reclamation

Table 3.1-17. Single Versus Dual Water System Characteristics, 8 Men - 109-Day Resupply

Characteristics	Single System	Dual System
Weight (lb) { Fixed and spares	1000	755
Weight (lb) { Resupply	42	55
Power (watt)	624	482
Heat rejection (watt)	678	514
Durability	Good	Good
Relative cost	Medium	Medium
Development status	Lab prototype	Lab prototype
Water balance	1050 lb deficit	262 lb excess

Water Storage and Purity Monitoring System

Waste storage tank characteristics have been included in the trade data but fresh water storage tanks, which are dependent on the selected concept were not included. The design for the OLS will include two vapor compression potable water systems for redundancy and one reverse osmosis wash water system. Each vapor compression system will be able to meet all water requirements for the eight men during any 24 hours. Thus, one vapor compression unit usually is not operating but is on stand-by redundancy. Sizing of the potable water accumulators is based upon the fact that if one vapor compression unit fails, another is able to commence operation immediately. Table 3.1-18 presents the requirements used to size the potable water accumulator. In essence, if both vapor compression units fail, the accumulators can supply the normal water requirements for two days or with emergency water requirements 3.5 days.

Table 3.1-18. Potable Water Accumulator Sizing

Daily water requirement: 16.55 lb/m-d = 132.5 lb/day for 8 men	
Total on-board potable water:	
2-day supply (normal rate):	265.0 lb
EVA water	14.7
IVA water	10.0
Total	289.7 lb
Store in each pressure volume: 145 lb water (2.33 cu. ft)	
Emergency water rate: 8.32 lb/m-d	
Emergency water available in OLS if both vapor compression units fails: 3.5 days (see text on emergency fuel cells)	

If a period greater than 3.5 days is required for repair of at least one of the vapor compression units or if a power failure occurs, the emergency fuel cells can be used. These fuel cells will produce 76.9 pounds of water per day while only 66.5 pounds per day of emergency water is required.

The fresh wash water accumulators are sized at 100 pounds of water since the daily rate is 96.5 pounds (12.01 pounds per man-day). The fresh wash water accumulators contain enough water so that two showers and one-half of the hand wash water can be used in any 12-hour period without draining the fresh wash water accumulators. The average shower cycle is three showers per day based upon the two showers per man per week requirement. Also, the reverse osmosis unit is sized for 18 hours of operation per day, so that if it fails, repair time and water processing makeup time exist.

The purity monitoring system operates by taking a sample of processed water, adding nutrient, incubating the sample, and then testing the conductivity of that sample with a similar sample in which no nutrient (and no incubation period) was added. Before testing for micro-organisms, pH and conductivity tests are used to determine inorganic salt concentration. The purity monitoring system requires isolation of the water being tested until it has passed inspection. This necessitates at least two separate tanks for each water recovery unit. Thus, each vapor compression unit has two 72.5-pound capacity accumulators, and the reverse osmosis unit has two 50-pound capacity accumulators.

To assure that the processed water remains sterile after testing, it is maintained at 160 F in the storage tanks. Silver ion units asssure sterility in the water plumbing.

Waste Management

The waste management system consists of a minimum of two stand-up wall urinals with water flushing and three toilets. The toilets are of the dry-john type (no water flush) and also have separate urinals mounted on them. Two waste processing drying chamber-compactors are included. Figures 3.1-53 to 55 (Volume IV) were used to determine the weight of this system. The weight and volumes of toilets were ratioed by 3/4 and the urinals by 1/2 (since the curves are set for four toilets and four urinals). The resulting characteristics are shown in Table 3.1-23 at the end of this section.

The minimum of two wall urinals evolves from a desire to have one wall urinal in each isolatable pressure volume. However, the toilets do have urinals so the wall urinal requirement is for convenience rather than necessity.

Three toilets were selected because of configuration requirements and crew size. First, the two isolatable pressure volumes imply at least two toilets. The decision to add a third toilet resulted from the 2 to 6-hour drying time required for each toilet. The toilets can be used one to three times before drying, but then the feces must be dried by exposing the toilet fecal container to vacuum. Crew overlap considerations also entered into the decision for three toilets.

Two drying chamber-compactors were selected so that one would be in each pressure volume and for redundancy.

The drying chamber concept for waste processing was selected as an interim concept because several detailed integrated trades are required. For Phase B, a trade of propellant to move trash to the lunar surface (or earth) versus bringing in expendables for changing solid wastes to ventable non-contaminating gases must be performed. Applicable concepts are listed in Table 3.1-19 with their expendables. These concepts will change the trash solids to CO_2, CH_4, and similar type gases. The present low temperature drying (250 F) only dehydrates the trash. Thermal decomposition (800 to 1000 F) cannot be permitted because the volatile hydrocarbons boiled off will contaminate the OLS exterior surface.

Table 3.1-19. Waste Processing Concepts

Candidate Waste Processing Concepts	Required Expendables
Wet oxidation	O_2, H_2O
Steam reformation	H_2O
Incineration	O_2

In this discussion of waste, note that feces and toilet paper are processed separately. The dry john uses vacuum to dehydrate the feces in its own chamber. The reason for separate processing is to eliminate manual transfer of feces. Thus, the Table 3.1-19 concepts are mainly for the non-fecal trash. The waste model utilized for the OLS study is given in Table 3.1-20. Subtracting feces and toilet paper, the model gives 26.4 pounds per day of trash which is approximately two cubic feet per day uncompacted or 0.4 cubic feet per day compacted. The generated trash actually may be as low as 20 pounds per day since in any given day, various items may not be generated. Thus, the cumulative total gives a conservative estimate.

Hygiene

The hygiene assembly contains the required hardware elements for crew grooming (sinks, showers), station cleaning (vacuum cleaners, cleaning cloths), crew personal equipment (clothes, toilet articles, PGA's, etc.) and laundry for washing cloths and linens.

The major trade in this assembly involved the laundry and the wear time of crew clothing. The options considered in the laundry trade are described in Table 3.1-9 (in Volume IV). The options range from disposable clothes at two different rates of use to a partial laundry with washing of undergarments only to full washing of all clothes. Table 3.1-21 presents the laundry model as well as listing other pertinent trade data.

Since the laundry area is in the very early concept definition phase, the data must be taken as approximate order of magnitude.

Table 3.1-21 indicates, as would be expected, that the disposable clothing concepts have high resupply weights and low costs. The disposable cases (options 1 and 2) have very high resupply weights and do not appear to be an optimum situation for a lunar mission and the associated cargo handling. Of the laundry options (3, 4, or 5), the full laundry cases (3 and 4) appear to be best from an overall weight standpoint, with option 3 the better of the two. However, option 3 requires that new crew members use clothing available on the station from previous crews. This could cause some problems with fit and the willingness of the crew to use used clothing. Option 4 is the next lightest concept and does not have the used clothing problem. In option 4, each new crew brings up its own clothing. Using weight and the used clothing problem as the major selection criteria, laundry option 4 is

Table 3.1-20. Waste Management OLS Waste Model

Source	Description	Basic Rate lb/Man-Day	8 Men lb/Day
Crew	Urine solids	0.13	1.04
	Fecal*	0.38	3.02
Food management	Food waste	0.40	3.20
	Food packaging	1.18	9.45
	Utensils, soap, etc.	0.01	0.08
Crew related	Wipes	0.20	1.60
	Hair, nails, skin	0.05	0.40
	Toilet tissue*	0.014	0.11
	Medical supplies	0.02	0.16
	Housecleaning supplies	0.02	0.16
	Soap, hygiene	0.033	0.26
	Dental	--	0.08
	Hair control	--	0.08
ECLSS process	Filters, charcoal	--	3.0
	Wicks, cartridges	0.033	0.26
	Water treatment	0.067	0.54
	Used catalysts	0.033	0.26
	Waste treatment and bags	--	0.28
	Clothing, towels, etc., (worn out)	0.10	0.80
Subsystems	Teletype paper	--	0.80
	Microfilm, magnetic tape	--	--
	Spare part packaging	--	0.50
Experiments	Photo lab	--	0.65
	Contaminated water (non-norm)	--	2.8
	Total (with Feces and Toilet Paper)		29.53
	Total (without Feces and Toilet Paper)		26.40

*Stored in Dry John

Table 3.1-21. Laundry Characteristics

ITEM → / FREQ.(DAYS) ↓	SHIRT	PANTS	UNDER-GARMENTS	SOCKS	SHEETS	TOWELS	WASH CLOTH	CONCEPT
MAXIMUM WEAR	10	14	3	3	10	7	7	1,5 (Outer)
NOMINAL WEAR	4	7	2	2	7	7	2	2,3,4,5 (Inner)

CONCEPTS

1 NO LAUNDRY
2 NO LAUNDRY
3 TOTAL LAUNDRY - NEW CREW USES OLD CREW'S CLOTHES
4 TOTAL LAUNDRY - NEW CREW USES OWN CLOTHES
5 DISPOSABLE OUTER GARMENTS AND LAUNDER UNDERGARMENTS

8 MEN - 10 YEARS

CONCEPT		1	2	3	4	5
WEIGHT (LB)	Fixed	160	250	515	420	460
	Resupply	9200	14,400	1400	3200	6400
POWER (WATT)		0	0	230	230	220
RELATIVE COST*		1	2	3	3	3
CREW TIME (HR/WK)		1.5	1.5	6.0	6.0	3.5

*1 = Lower Cost

Space Division
North American Rockwell

selected as the Phase A concept. It is recognized that a more detailed evaluation is required to select the optimum laundry concept. Consideration in greater depth of crew comfort, flexibility of cargo, and overall program cost is required.

The other hardware elements of the hygiene assembly do not require major trades to select hardware concepts, once the functions, such as a whole body washing by shower, are determined. For this phase of the study, the functions and hardware elements of the hygiene assembly have been selected and defined on the bases of good habitability. An attempt was made to make body washing and house cleaning as earth-like as possible.

Food Management

The food management assembly provides for food storage, preparation, serving, cleanup, and inventory control. The food management hardware must accommodate a large range of food types, cooking operations, and crew use modes. A key goal in the food management design is flexibility--flexibility in terms of what can be permitted, so that not only nutritionally adequate and organoleptically acceptable meals will be consumed, but also that maximum morale boosting and psychological benefits can be achieved. The meals will consist of a mix of meats, vegetables, fruits, cereals, and beverages provided in bulk and individually packaged form from frozen, dried, thermostabilized (canned), and fresh stores. The rationale for selecting the above food mix is, briefly, crew acceptability, earth-like foods, and a requirement that no special preparation of the food be required during manufacturing. The hardware associated with this food mix is listed in Table 3.1-22.

Table 3.1-22. Food Management Hardware

Resistance oven for browning foods
Microwave oven for rapid cooking
Freezer for frozen foods
Refrigerator
Utensils
Reconstitution units
Dishwasher
Reduces utensil resupply

For emergencies, bite size, freeze dried, and ready-to-eat canned foods will be eaten.

Special Life Support

Emergency O_2 and N_2 is stored for 8 men and 30 days. The 30-day emergency requirement is defined in Section 5.0 of Volume III, the OLS Performance Requirements Report. Nitrogen storage is required for emergency leakage in order to maintain the total pressure at or greater than 10.0 psia, and to keep the oxygen percentage fixed. Figure 3.1-4 illustrates the magnitude of pressure decay and O_2 volume percent increase as a function of days with no N_2 leakage makeup.

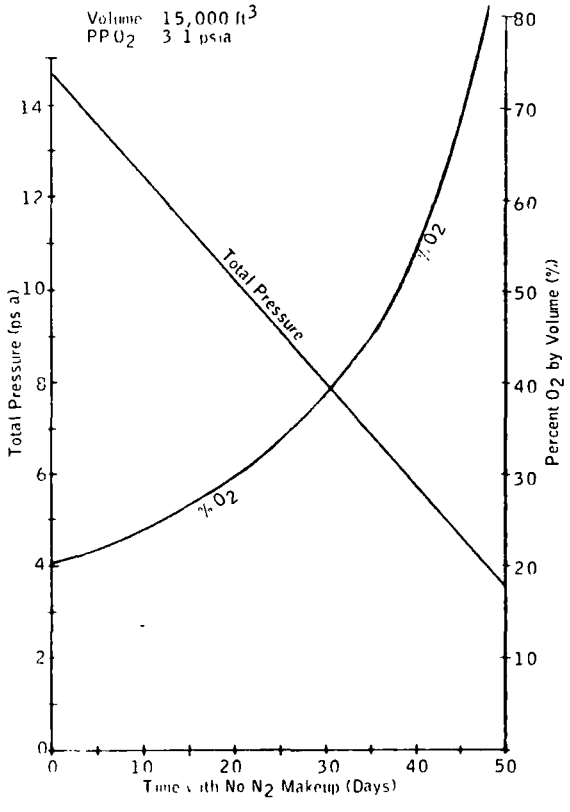


Figure 3.1-4. Pressure Decay With No N_2 Makeup

Although two vapor compression potable water processing systems are included for redundancy of the water supply, these units would not be available if a power emergency occurs. Thus, provisions for emergency water supply are required.

Emergency water, per se, is not stored in the OLS because the emergency EPS fuel cells will generate 2300 pounds of water in 30 days while the crew will only require 2000 pounds based upon the 9.32 pounds per man-day emergency water requirement. This water rate includes 6.32 pounds per man-day food and drinking water plus 2.00 pounds per man-day for hand washing or other uses.

The criteria on IVA suit loop, fire control and other miscellaneous provisions for OLS are identical to EOSS. Therefore, the same concepts were selected.

ECLSS Characteristics

Table 3.1-23 lists the ECLSS characteristics of weight, power, and volume. The values shown are from the parametric curves of Volume IV or calculated separately (e.g., the fresh water storage tanks) as concepts were selected. Items sized by non-parametric calculations are discussed in the appropriate assembly sections.

3.1.3 IMPACT OF ARTIFICIAL GRAVITY

Assuming that the OLS will always have some zero-gravity manned operations, all ECLSS equipment will have to be designed for zero-gravity. Thus, the system will be G-vector sensitive. The "floor" will have to be the "floor" in zero and artificial gravity. When in the artificial-g mode, natural convection air mixing may allow fan power reduction. Fan power requirements can definitely be reduced in fluid and particle migration (as for toilets). Vapor/liquid phase separation is aided by artificial-g. Maintenance is also aided in fluid lines in the sense that fluid spillage is less of a problem and gas bleeding of fluid lines is simplified by the presence of a gravity gradient.

Table 3.1-23. ECLSS Weight-Power-Volume Summary

Subsystem	Weight - Pounds						Power - Watts			Volume - Cubic Feet					Notes
	Fix Module	No. Mod.	Total Fixed	Total Spare (180 D)	Resupply*		Peak	Oper. Time (hr/day)	Avg.	Fix Module	Total Fixed	Total Spare (180 D)	Resupply		
					Expen.	Spare							Expen.	Spare	
Atmos. storage			5236	-	-	-			350		133	-	-	-	Includes gas and launch air
CO_2 management			1550	423	260	-			3090		40.8	17.1	-	-	
Atmos. control			1520	244	84	46			2300		375	18.4	-	-	
Active thermal			2670	150	-	13			335		41.9	8.8	-	1.6	Includes boom loop; excld. rad. and therm. mass
Water mgmt.			788	389	64	3			837		102	44	0.6	0.6	Water fill excld.
Waste mgmt.			473	63	33	-			70		22.7	6	16	-	
Hygiene			2047	95	114	3			180		213	5.5	2.1	-	Includes laundry and crew pers. equipment
Food management			1000	150	3330	-			630		165	30	121.5	-	Exp. - food
Spec. life supp.			474	30	-	-			0		19.3	0.4	-	-	
Total			15,758	1544	3885	65			7792		1113	130	140	2	
Atmos. storage $N_2/O_2/H_2$ cryo	See RCS														
Hi press O_2			664	-	-	-			-		-	-	-	-	8 men-30 days and IVA-EVA
Hi press tank			795	-	-	-			-		37	-	-	-	1.20#TK/#gas
Hi press N_2			515	-	-	-			-		-	-	-	-	30 day
Hi press tank			618	-	-	-			-		41	-	-	-	1.20#TK/#gas
Pumpdown sys.	180	2	360	-	-	-	600		350	19	38	-	-	-	
N_2 accumu.	17	2	34	-	-	-	-		-	8.5	17	-	-	-	
Launch air	-	-	2250	-	-	-	-			-		-	-	-	
Total			5236	-	-	-	-		350	-	133	-	-	-	
CO_2 management															
H_2 dep. cell	525	2	1050	221	-	-	1070	14.6	650	8.6	17.2	9.2	-	-	5mm Hg PPCO_2
Sabatier	50	2	100	50	260	-	50	-	10	4.4	8.8	2.2	-	-	H_2 from RCS
Solid polymer	200	2	400	152	-	-	4000	14.6	2430	7.4	14.8	5.7	-	-	
Total	-		1550	423	260	-	-	-	3090	-	40.8	17.1	-	-	

*Resupply for 8 men and 109 days. For resupply storage of expendables, multiply by 191/109.

Space Division
North American Rockwell

Table 3.1-23. ECLSS Weight-Power-Volume Summary (Continued)

Subsystem	Weight - Pounds						Power - Watts			Volume - Cubic Feet					Notes
	Fix Module	No. Mod.	Total Fixed	Total Spare (180 D)	Resupply*		Peak	Oper. Time (hr/day)	Avg.	Fix Module	Total Fixed	Total Spare (180 D)	Resupply		
					Expen.	Spare							Expen.	Spare	
Atmos. control			1520	244	84	46	2300	24	2300		375	18.4	-	-	Reg. char. concept
Active thermal			2100	120	-	6	235	24	235		38	7	-	0.7	Freon, H_2O loops
Boom loop			570	30	-	7	100	24	100		3.9	1.8	-	0.9	only; no rad.
Water mgmt															
Pot.recl.-vc	335	2	670	328	29.4	-	411	24	411	33.5	67.0	37.7	-	-	
Pot. storage															
H_2O	(72.5)	4	(289)	-	-	-	-	-	-	-	-	-	-	-	
Tank	7.3	4	28.9	-	-	-	258	24	258	1.2	4.8	-	-	-	Heat H_2O 80 F to 160 F
Wash recl.-ro.	42.7	1	42.7	49.1	25.9	-	71.4	18	53.5	7.8	7.8	4.2	-	-	
H_2O	(50)	2	(100)	-	-	-	-	-	-	-	-	-	-	-	
Tank	5	2	10	-	-	-	94	24	94	8	16	-	-	-	Heat H_2O 80 F to 160 F
Purity System	12	3	36	12	9	3	150	3	20	2	6	2	0.6	0.6	
Total	-	-	788	389	64	3	-	-	837	-	102	44	0.6	0.6	
Waste mgmt.	236	2	473	63	33	-	1700	-	70	11.8	22.7	6	16	-	
Hygiene															
Hardware			787	50	2	3	1310	-	170		183	4	1.2	-	
Crew pers. eq.			850	45	2	-	-	-	-		30	1.5	0.9	-	
Laundry			420	-	110	-	230	-	10		-	-	-	-	
Total			2047	95	114	3	-	-	180		213	5.5	2.1	-	
Food management															
Hardware			1000	150	-	-	4500	-	630		165	30	-	-	
Food			-	-	3330	-	-	-	-		-	-	121.5	-	
Total			1000	150	3330	-	-	-	630		165	30	121.5		
Spec. life supp**															
IVA umb.loop	20.6	3	61.8	30	0	0	500	0	0		1.2	0.4	0	0	
IVA O_2	Stored in HP tanks														
IVA H_2O	Stored in H_2O tanks														
EVA-PLSS	80	2	160	-	-	-	0	-	0	3.1	6.2	-	-	-	PLSS w/exp. (240 watt-hr battery recharge)
EVA O_2	Stored in HP O_2 tank														
EVA H_2O	Stored in H_2O tanks														
EVA LiOH	-	-	6.7	-	-	-	-	-	-	-	-	-	-	-	PLSS norm. not used
EVA-Ops.	40.5	2	81	-	-	-	0	-	0	1.8	3.5	-	-	-	Charge (5.7# O_2/unit)
Fire fight	13.6	12	164	-	-	-	-	-	-	0.7	8.4	-	-	-	2/deck + toroids
Total	-	-	474	30	-	-	-	-	0	-	19.3	0.4	-	-	

*109 Days
**PGA and umbilicals are part of crew personal equipment

Space Division
North American Rockwell

Page intentionally left blank

3.2 ELECTRICAL POWER SUBSYSTEM

The primary function of the electrical power subsystem (EPS) is to provide continuous electrical service for the 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, docked modules, docked lunar landing vehicles, and docked logistics vehicles. Major EPS functions are summarized in Figure 3.2-1.

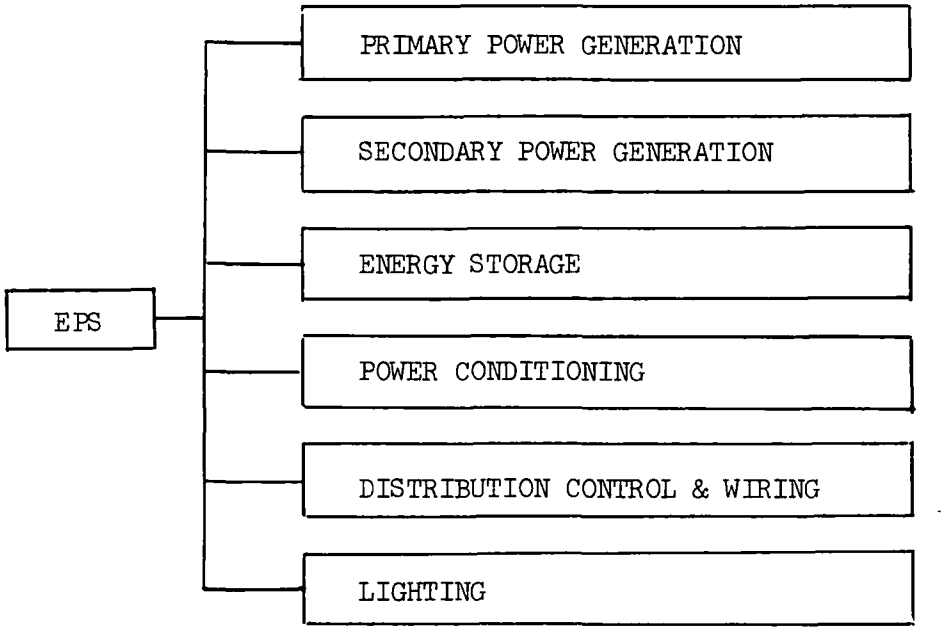


Figure 3.2-1. Major EPS Functions

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 are:

1. Nominal power supplied with one credible failure
2. Power for critical functions but at reduced power level with two failures

3. Separate, independent system for emergency power

4. Safety features - automatic switching on critical loads, isolation of critical life support functions

5. Normal maintenance without primary power shutdown

6. Common electrical interface at docking ports

7. Modes of operation

	Duration	Power Level (kw)
a) Unmanned launch and premanning	5 days	2.0
b) Manned earth orbit (EO) operation	2 days	13.5
c) Unmanned EO, translunar flight and lunar orbit operation	7 days	5.5
d) Lunar orbit (LO) quiescence storage	1 year	1.9
e) Initial manned lunar orbit operations	2 days	15.0
f) Manned lunar orbit operations	Continuous	20.0
g) Fail degraded	90 days	13.5
h) Emergency	30 days	4.0
i) Earth eclipses	3.5 hours	15.0
j) Solar flare storms	72.0 hours	10.0

EPS Performance Requirements

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

1. Operate unmanned for launch and both unmanned 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 (10 year operational life through scheduled maintenance/repair)

8. Inertial attitude hold and earth-, lunar-, and solar-referenced attitude-hold capability

9. Provide solar array orientation hardware to operate from ISS control even when operated from ground uplink command

10. Provide solid-state switching devices, computer operated and monitored, which are used to turn on or off any subsystem. These solid-state switching devices serve the dual function of an OFF-ON switch and wire protection, thereby eliminating the traditional combined requirement of circuit breaker, toggle switch, and power relay for on-off control and protection

11. Provide nonsynchronization of boom deployment/retraction

12. Operate in a zero-g mode (Figure 3.2-2)

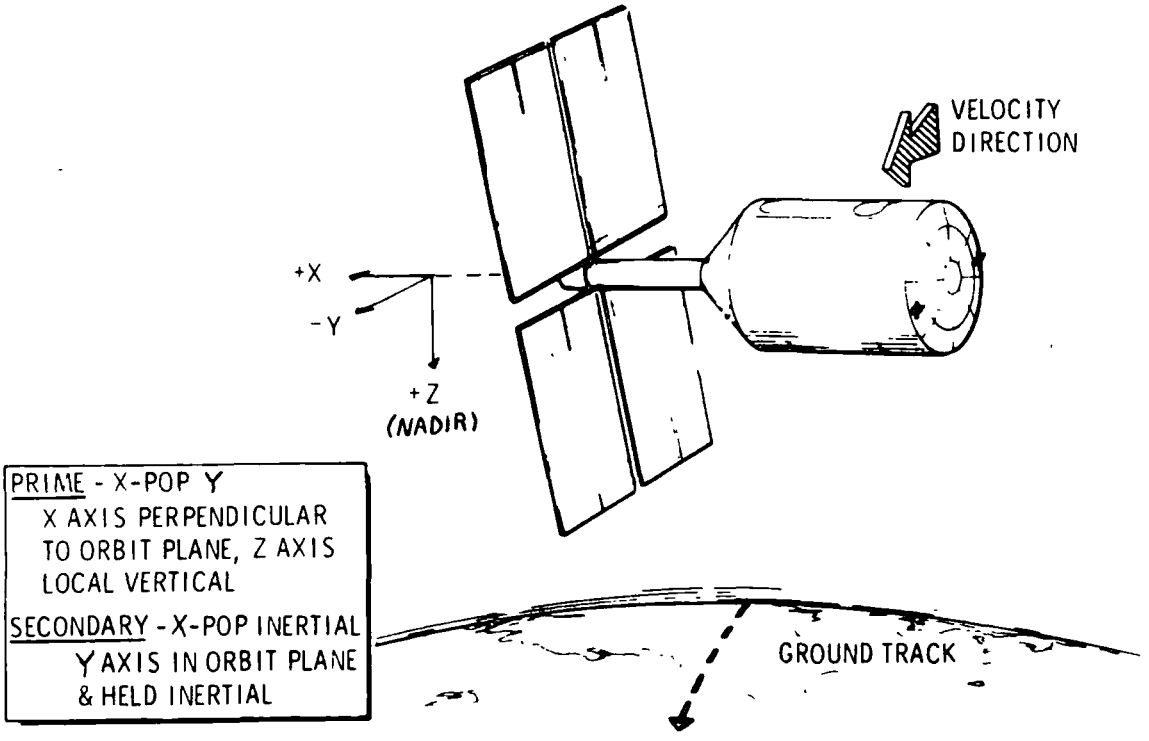


Figure 3.2-2. Zero-G Mode

13. Provide onboard checkout and fault isolation to in-flight replaceable unit (IFRU) levels

14. Provide solar arrays that permit panel deployment as a function of load demand in addition to deploy/retract capabilities.

Power Profile

The EPS must supply a 24-hour average power level of 20 kw. Detailed power requirements based on a 8-man crew are shown in Table 3.2-1. A typical mission power profile is shown in Figure 3.2-3. A typical lunar orbit load profile is illustrated in Figure 3.2-4. The power profile is based on the following assumptions:

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

2. EPS electrolysis will be fully completed each orbit.

3. Weighted system and EPS electrolysis efficiencies are 80.5 and 58.8 percent, respectively.

4. Orbital characteristics are:

Earth	Lunar
258 nautical-mile circular orbit	60 nautical-mile circular orbit
Inclination angle = 31.6 degrees	Inclination angle - 90 degrees (polar)
orbit time = 1.509 hours	Orbit time = 1.987 hours
(worse case = 38 percent dark)*	(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 $\leq$ 234 minutes). For these periods, solar-powered EPS's will be required to revert to the emergency mode of operat ons (fuel cells). The fuel cells have a rated capacity of 10 kw but can be operated at twice rated capacity (20 kw) without sacrificing output power quality. However, such operation must be limited to the contingency periods defined as extensive operation at more than rated capacity would reduce service life.

3.2.1 CONCEPT DEFINITION

A photovoltaic energy conversion concept (SA-RFC) has been selected as the optimum Electrical Power Subsystem (EPS) for the OLS. A one-line schematic of the EPS is shown in Figure 3.2-5. A four-channel design, permitting either isolated, split-parallel or full parallel operation (four small but identical

Table 3.2-1. OLS Electrical Load Analysis (8-Man Crew)

Subsystem/Equipment	EPS Load - (watts) Sustained*	Average	Backup	Emergency	Premanned
Atmospheric storage	600	350	60	0	0
CO_2 management	5,120	3,090	3,090	650	0
Atmosphere control	2,300	2,300	2,300	1,283	210
Active thermal control	335	335	335	235	235
Water management	984	837	837	0	258
Waste management	1,700	70	70	25	0
Hygiene	1,540	180	180	0	0
Food management	4,500	630	630	0	300
Special life support	500	0	0	0	0
Total ECLSS	17,579	7,792	7,502	2,193	1,003
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 tracker	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)		18,951	11,980	3,911	1,910

*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.

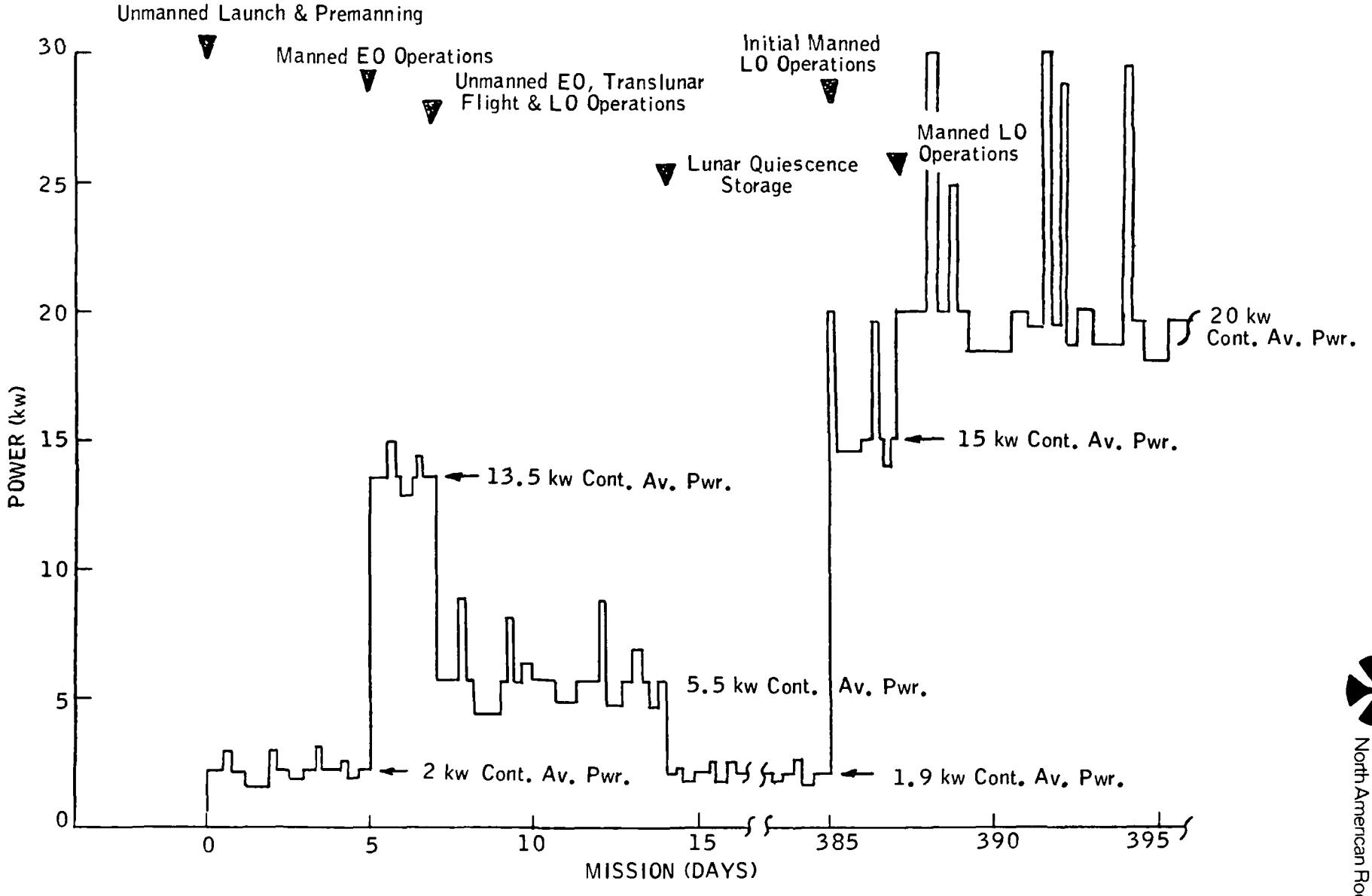


Figure 3.2-3. Typical OLS Mission Power Profile (8-Man)

Space Division
North American Rockwell

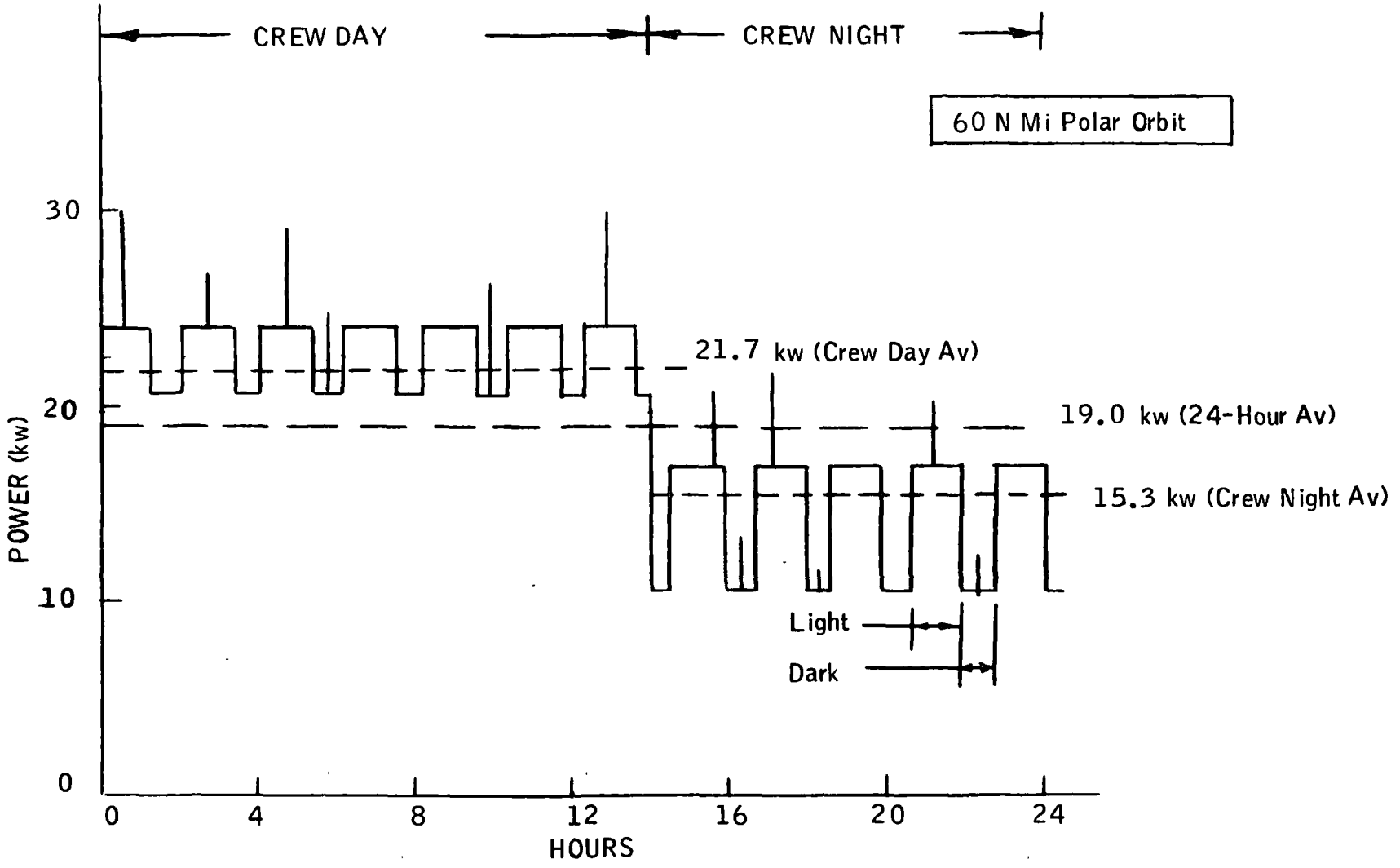


Figure 3.2-4. Typical Orbit Load Profile for OLS Profile (8-Man)

Space Division
North American Rockwell

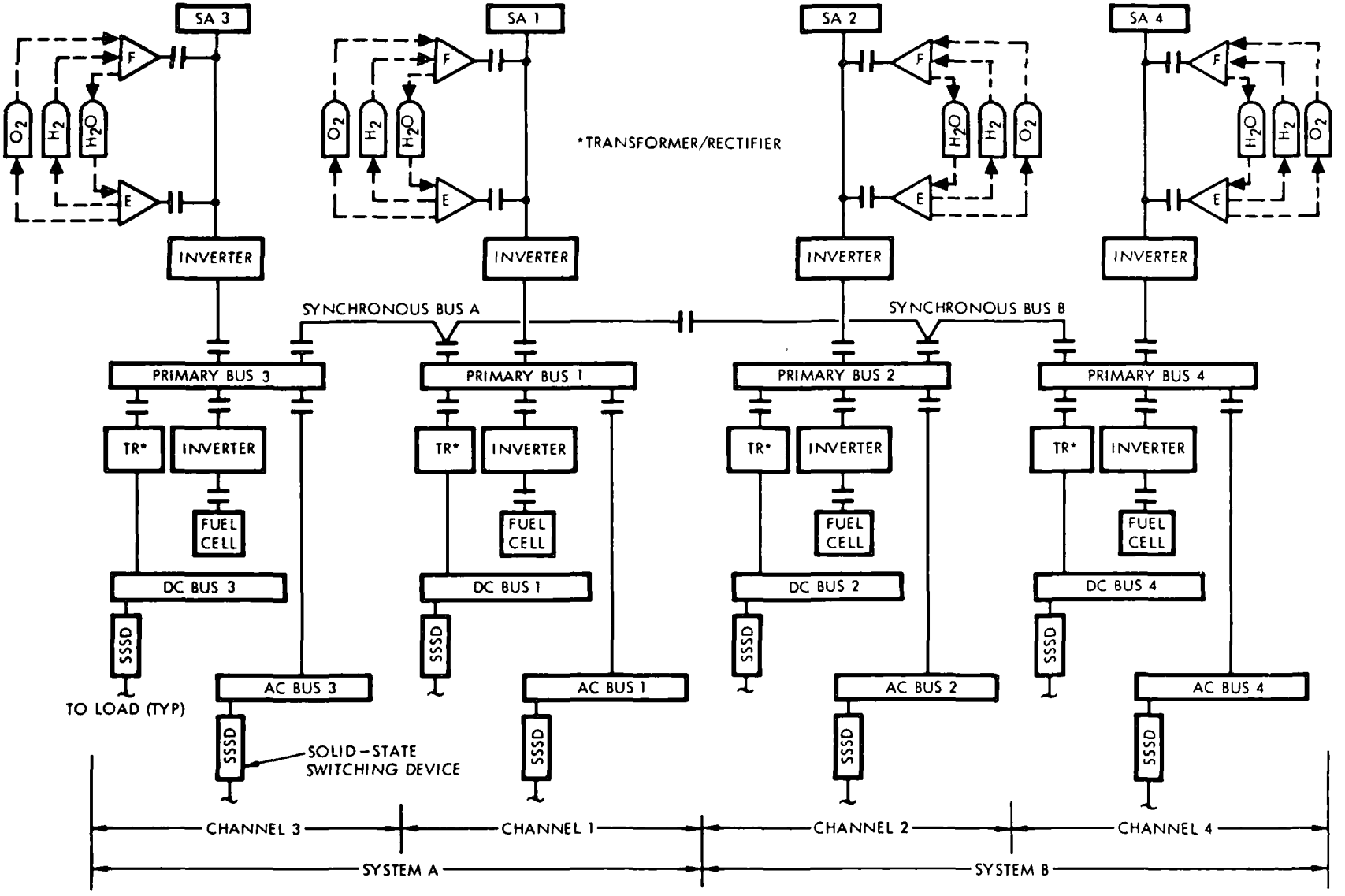


Figure 3.2-5. EPS (Solar Array/RFC) One-Line Schematic

Space Division
North American Rockwell

EPS's) is recommended. The split-parallel mode of operation provides the best combination of operational flexibility and assurance of continuous electrical power for the OLS normal operating mode. The redundancy of this EPS concept plays a paramount role for meeting backup requirements and also simplifies functional description since only one channel requires description. A functional block diagram of this concept is shown in Figure 3.2-6. Power generation is provided by solar photovoltaic conversion utilizing the sun as an energy source. Inverter-regulators provide 120/208 vac, 3ϕ, 400 Hz power for efficient transfer to the OLS core module. Additional power conditioning is provided by transformer-rectifier units to meet the selected 56-volt dc power requirements. Regenerative fuel cells (RFC) are used for energy storage and to supply the dark side period power requirements and peak power demands. Power flow to the various distribution busses completes the major functions. A vehicle single point grounding (SPG) system is used.

The emergency power is provided by four fuel cells (FC). Emergency inverter-regulators provide 120/208 vac, 3ϕ, 400 Hz to the various distribution busses. The fuel cells also provide power during long eclipse periods and when the solar array is rolled up during solar flare storms. The emergency power source can also be used during periods of maintenance on the primary power subsystem. RFC's were considered and eliminated as an emergency power source because of possible adverse effect on cell life (use of propellant grade reactants during emergencies), location of the RFC's (boom) prevents their use during boom replacements and the EPS constraint requiring a separate, independent source for emergencies.

The EPS power characteristics will exceed the requirements defined in MIL-STD-704-A in most cases and represent a balanced combination of system complexity, advanced technology, and inherent redundancy that assures a high degree of reliability and operational flexibility. Although the EPS is basically of contemporary design, a number of unique features have been incorporated. These features include:

1. Optimization of the controls, bus, and feeder configurations to the specific requirements of the OLS.

2. System performance and load monitoring are computer-controlled to assure superior control and protection functions for all operating conditions.

3. Power generation, distribution, and control centers located in areas to optimize system weight, performance, and reliability.

4. An advanced system of multiplexing for point-to-point transmission of selected data and control signals between the EPS and other subsystems (i.e., the information subsystem).

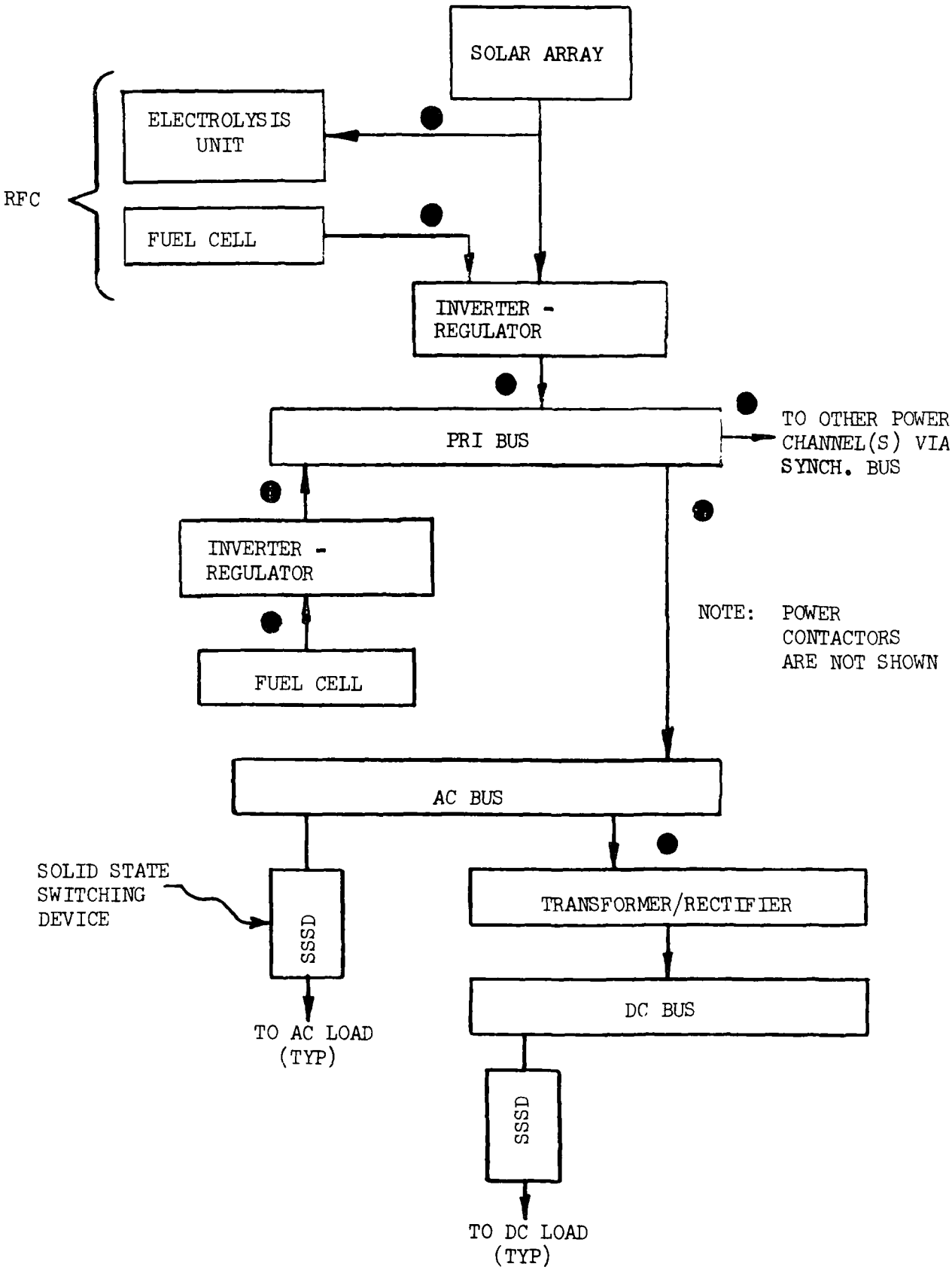


Figure 3.2-6. EPS (Solar Array/RFC) Functional Block Diagram

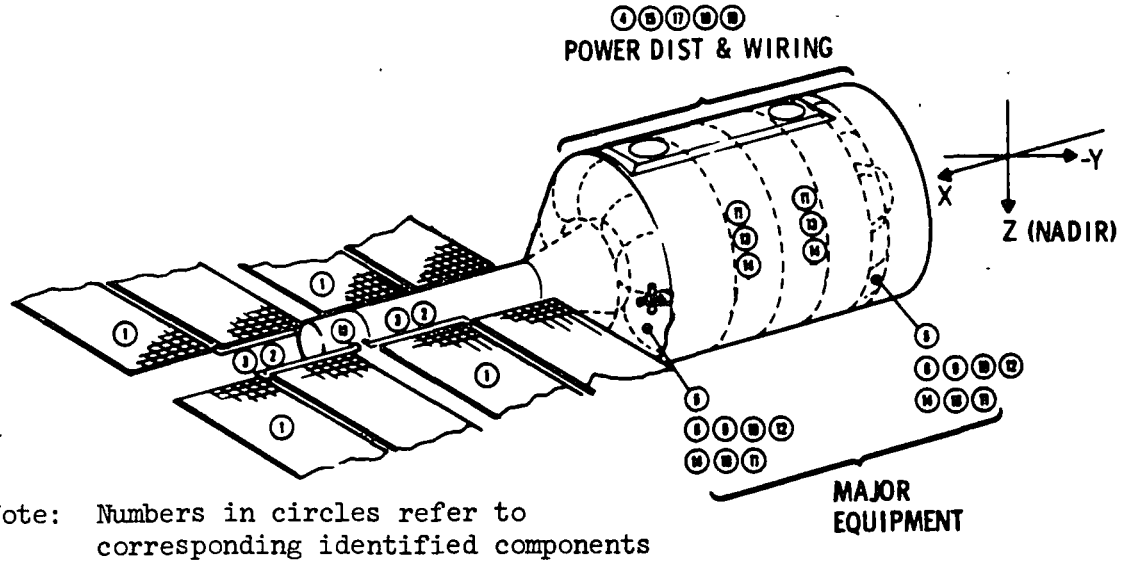
5. Electrical power switching and circuit protection for individual load equipment is accomplished by means of computer-controlled solid-state switching devices.

6. A lightweight deployable rollup solar array panel that permits panel deployment as a function of load demand (deploy/retract).

7. A modular-type inverter whose design permits built-in testing, peak-power tracking and computer control.

8. Major functional units having provisions for built-in testing by computer direction (onboard checkout and fault isolation).

Major assemblies and components are identified in Figure 3.2-7, the functional descriptions are summarized in Table 3.2-2 and the installation locations are shown in Figure 3.2-8.



Note: Numbers in circles refer to corresponding identified components in Figure 3.2-7.

Figure 3.2-8. EPS Installation

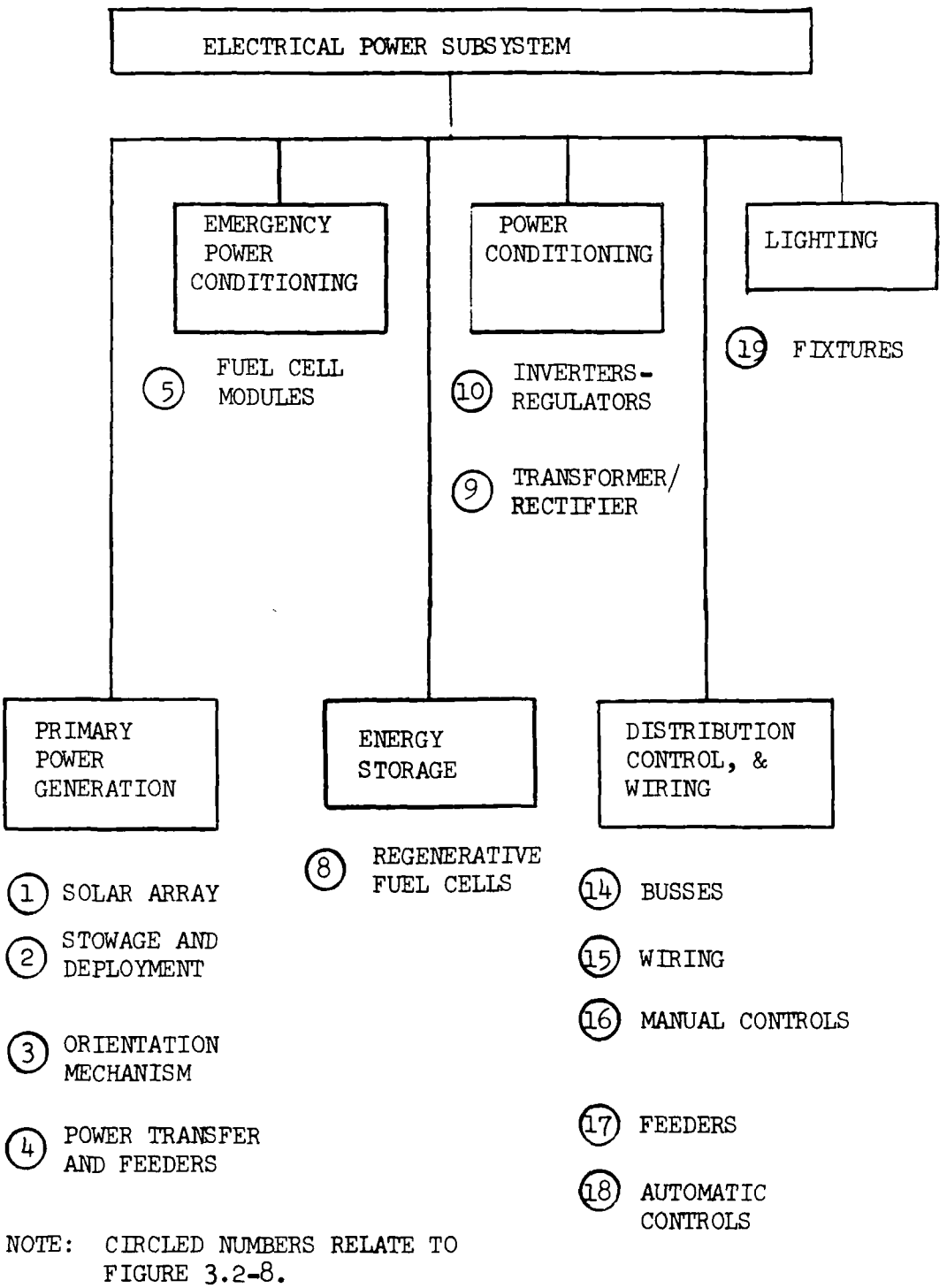


Figure 3.2-7. EPS Assemblies

Table 3.2-2. EPS Functional Descriptions Summary

Function/Assembly	Description
Solar array	10,000 ft^2 divided into four identical panels each of 2,500 ft^2 with 22.8 kw beginning of life (BOL) (70C) power generation, three-wire output of ±112 vdc, 40 circuits/panel. In lunar orbit power output is ≈ 18.25 kw BOL (105C).
Inverter-regulator	Converts power to regulated 120/208 volts, 3Ø 400 Hz ac power, power from transmission feeders to primary bus and establishes OLS ground connection.
Primary bus	Four identical (i.e., one per solar panel) primary busses transmit power to transformer/rectifier units and secondary busses and receives power from regenerative fuel cells, fuel cells, or solar panel. Normally operated in the split-parallel mode.
Transformer-rectifier	Furnishes 56 volts dc to the bus.
Secondary bus	Contains transformer rectifier units and solid-state circuit breakers load/circuit protection equipment.
Control/distribution	Under command of ISS, control and distribution of electrical power to selected loads is performed.
Regenerative fuel cell	Electrical output is connected to solar array inverter-regulators that convert dc voltage to regulated 120/208 volts, 3Ø 400 Hz ac which is delivered to the primary bus and receives its input power (raw) directly from the solar arrays.
Premanning, earth eclipses, emergency and solar flare activities power	Fuel cell output connected to the emergency inverter-regulator.
Control-protection	Absolutely dependent on ISS computer. Minimum control and protection function are: (1) overvoltage; (2) undervoltage; (3) overcurrent; (4) unbalance current; (5) overfrequency; (6) underfrequency; (7) differential feeder, bus external power, and secondary bus; (8) automatic paralleling; (9) open phase; (10) load division.
Ground	Neutrals of each inverter-regulator and transformer/rectifier unit are routed to a single point vehicle ground.
Cooling	ECS furnishes the cooling for the regenerative fuel cells, fuel cells, and power conditioning equipment including the secondary busses.

Power conditioning associated directly with the solar array is located in the power boom (i.e., solar array inverters). This power boom is pressurizable to enable shirtsleeve maintenance. Other power conditioning and primary busses are located in the upper and lower torus areas. Duplicate locations for secondary bus hardware have been chosen utilizing decks 2 and 3.

The EPS employs three basic kinds of power sources for meeting the OLS power requirements: (1) solar arrays for the light orbital periods, (2) RFC's primarily for the eclipse periods, but also for power peaks during light periods, and (3) fuel cells for premanning, earth eclipses, solar flare storms and for emergency requirements.

Primary Power Source

The primary electrical power is obtained from the solar array. The solar array configuration consists of four similar panels. Each panel (negative or positive) has two elements consisting of N/P solar cells. Each element is wrapped on a 25-inch diameter cylindrical storage drum for launch. Each drum is attached to a common center support structure which also mounts the bi-stem deployable boom actuator. A single deployable support boom provides the structural support for all four deployed array elements (Figure 3.2-9).

Significant design parameters associated with the solar panel are summarized in Table 3.2-3 and the corresponding weight breakdown in Table 3.2-4. The panel weight of 1180 pounds assumes the use of thin solder-coated solar cells resulting in a power-to-weight ratio of 22,800/1180 or 19.3 watts per pound nominal at beginning of life (BOL) and 70 C. If solderless solar cells were used, the panel weight would be reduced to 980 pounds or 23.3 watts per pound for the same conditions.

Energy Storage

Regenerative fuel cells (RFC) store energy from the solar array during the light periods of each orbit and supply energy for the dark periods and during power peak periods. Energy storage is accomplished by the electrolysis of water into reactants (hydrogen and oxygen) during the light periods and, by means of fuel cells, converting the reactants to electrical power during the dark periods in a closed-loop system. The requirements for long reliable life dictated separate cells assemblies for electrolysis and power generation. The two cell assemblies, operating in a back-to-back closed-loop mode, use raw power from the solar array during electrolysis and furnish dc power to the solar inverter-regulator during eclipse periods. For reliability, the RFC will have provisions for automatic operational and thermal control, use modular concepts to facilitate maintenance and have extra cells for the substitution of failed cells. Thermal control of the RFC will be by separate circulating cooling loops with the excess heat rejected to the boom radiator.

Cells are packaged in a replaceable cell modular concept to facilitate in-orbit maintenance. Each module has provisions for sensing parameters required for computer monitoring and control of individual cells/modules with

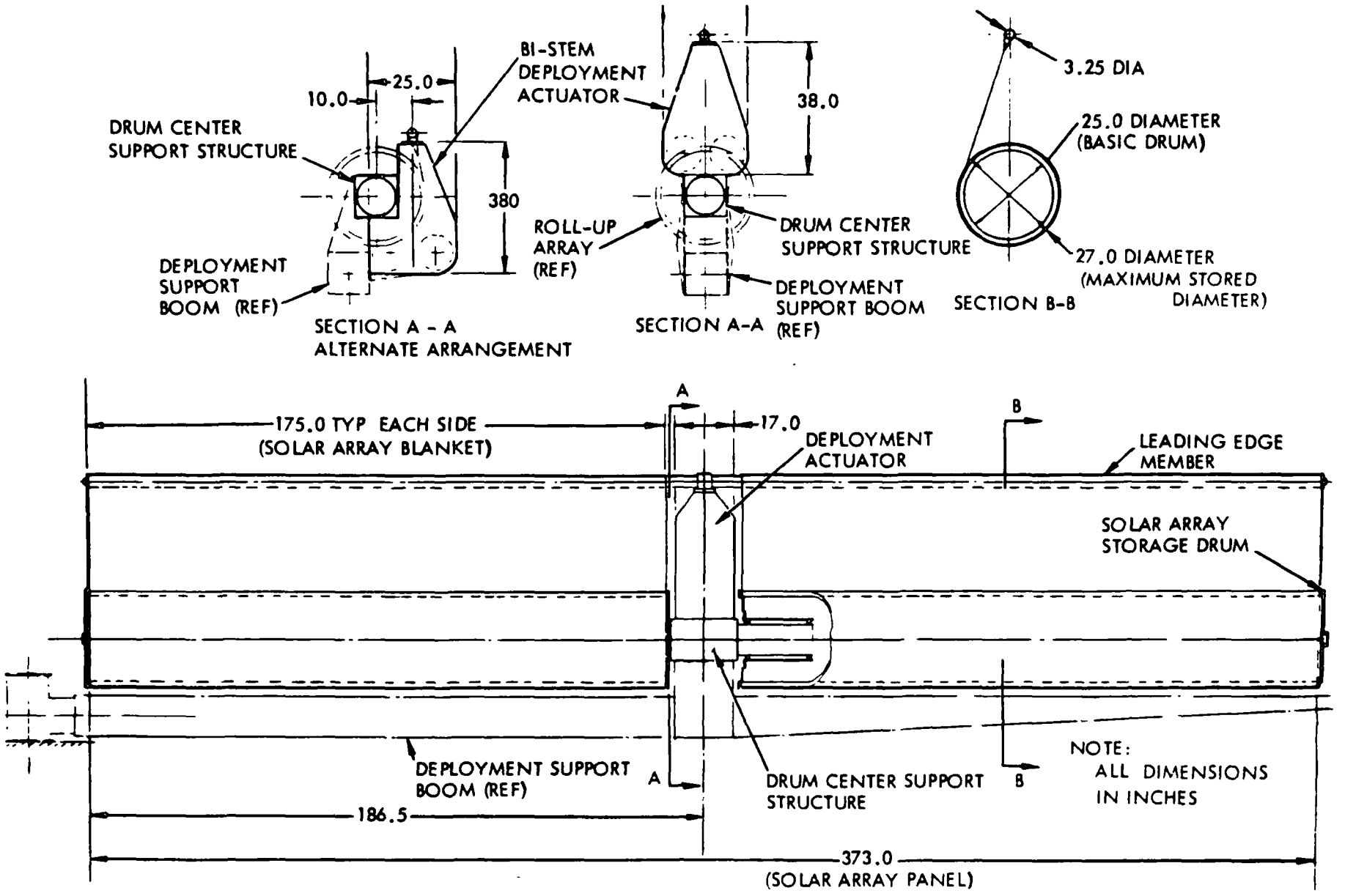


Figure 3.2-9. Baseline Rollup Solar Array

Space Division
North American Rockwell

Table 3.2-3. Summary of Array Design Parameters

1.	Nominal array unit size	2,500 ft^2	
2.	Total solar cell module area	2,489 ft^2	
3.	Max. array power (BOL, 70 C, 15 deg misorientation), 105 C (lunar orbit)	22.8 kw 18.25 kw	
4.	Max. power voltage (BOL, 70 C, 15 deg misorientation)	126 vdc	
5.	Number of cells	184,800	
6.	Total solar cell area	2,388 ft^2	
7.	Number of series cells/circuit	330	
8.	Number of parallel cells/circuit	14	
9.	Fully deployed array length	96.3 ft	
10.	Total blanket width	14.6 ft	
11.	Total array width	31.1 ft	
12.	Artificial-g deployed length	51.0 ft	
13.	Total array area (not including space between blankets)	2812.0 ft	
14.	Leading edge member diameter	3.25 in.	
15.	Storage drum diameter	25.0 in.	
16.	Bi-stem diameter	4.23 in.	
17.	Bi-stem element thickness	0.022 in.	

Table 3.2-4. Array Weight Breakdown

	Weight (lb)
Array blankets	741
Storage drum structure	87
Outboard and support structure	30
Leading edge member	22
Deployable boom	188
Actuator and center support	112
Total array system weight	1,180

Space Division
North American Rockwell

appropriate readouts. The system will be capable of detecting malfunctions with fault isolation to major component levels.

Four completely independent but similar RFC's, one for each channel, are required for the OLS. Each RFC has an output capacity of 6 kw, a electrolysis capacity of 4 lb per hour and weights approximately 260 pounds including reactants and tankage.

Emergency/Auxiliary Power Source

The Emergency/Auxiliary power source for premanning, emergency, earth eclipses and solar flare storm requirements consists of four fuel cells, one for each channel. The OLS power requirements could be met by 3 of 4 planned units. Specific weight of this assembly would be about 60 pounds per kilowatt including mounts and support based upon a 2.5 kilowatt fuel cell rating. Excess heat from the fuel cells are rejected to the environmental control subsystem (ECS) core radiator by means of cooling loops. Reactants for fuel cell operation are planned to be supplied from the reaction control subsystem. Approximately 3760 pounds of reactants are required for emergency, 540 pounds for premanning, 780 pounds for solar flare storms and 58 pounds for earth eclipses.

Power Conditioning

The power conditioning equipment is inherently decentralized due to distribution requirements and redundant services required in support of two pressurized compartments. Power conditioning equipment employs modular design concepts to facilitate in-flight maintenance to the component level and permit a high degree of equipment commonality (i.e., component interchangeability for both like and some unlike units). Coldplates, using two independent cooling loops, are used for thermal control of the power conditioning equipment.

The primary function of the inverter-regulator is to convert unconditioned dc power to conditioned 3-phase ac power. Electrical energy is obtained from two dissimilar dc power sources (3-wire, ±112 volts dc from the solar array, and 2-wire output from the fuel cell), hence the requirement for different input sections of the inverter-regulators. In addition, the controls are somewhat different in that solar array inverter-regulators have provisions for peak-power-tracking. Both inverter-regulator designs utilize selective rating capabilities (efficiency optimizing). Both designs feature similar packaging techniques in that each has input and output modules. However, only the output modules are interchangeable. All modules are replaceable in orbit should a malfunction occur. Offline units are switched on by computer control. All inverter-regulators are clock-synchronized to permit paralleling and assuring the absence of heat frequencies throughout the OLS. The efficiency will remain uniformly high for varying power demands because modular design concepts permit sub-modules to be switched in or out as a function of load requirements, thereby maintaining the efficiency at near an optimum point for all load conditions. Parallel operations will be the normal mode but capability for isolated operations will be provided. Excess heat

from the inverter-regulators is rejected to the core/boom radiators by means of cocling loops passing thru coldplates. Loss of any one cooling loop will not degrade inverter-regulator performance.

The transformer/rectifier (TR) converts power from the secondary bus (120/208 volt ac, 3Ø, 400 Hz) to an unregulated 56 volts direct current power output. The TR consists of a core transformer with a three-phase four-wire (wye connected) primary and a secondary having multiphase winding connected to reactors and series rectifiers as required to maintain the dc output current within the allowable voltage ripple limits. The primary windings of the transformer are provided with taps such that it is possible to achieve three output voltages 1.5 ± 0.2 volts apart. Incorporated into the TR are sensors (i.e., ammeter shunt, temperature detector, voltage sensing points, etc.) required for computer monitoring and control of the TR's. Thermal control of the TR is through a coldplate with two separate cooling loops for rejecting excess heat to the ECS core radiator. Loss of any one cooling loop will in no way degrade the TR's performance.

Four TR's, one for each channel rated at 1 kw, are required for the OLS. It is estimated that the dc power requirements can be met by two of the four units. Each TR will weigh about 7 pounds and have a volume of approximately 0.1 cubic feet.

Power Distribution

Electrical power must be distributed from the generation source tc the user equipment. This design must provide for its own stability under transient and overload conditions and must include capability to adjust for changing load requirements. Based on EOSS experience it is believed that the loads will be approximately 90 percent ac and 10 percent dc.

Fault protection is mechanized on the following definitions: Primary Bus 1 and 2 are located in one pressurized area, Primary Bus 3 and 4 in a second pressurized area. The primary busses are normally operated in split-parallel and the appropriate power channel isolated when faults occur that threaten the stability of the system.

Each primary bus distributes 120/208 volt, 3-phase, 400 Hz, 4-wire ac power to the load areas (secondary busses) for distribution to the loads via the solid-state switching devices at the secondary bus structure (Figure 3.2-6).

Direct current power at 56 volts will be derived from the TR unit. The output from the transformer/rectifier is distributed on a 2-wire system. Fifty-six volts dc was selected as the best compromise between high conductor weight at lower voltages (28 volts) and switching problems at higher voltages (112 volts).

The secondary bus structure consists of the hardware to provide solid-state switching at two different voltage levels (120/208 volts ac, and 56 volts dc) and the TR unit which is integrally mounted on a coldplate.

Any single secondary bus can support the critical loads (i.e., those loads required for personnel safety and mission continuation) of the OLS. Critical loads normally are connected to two or more secondary busses. Computer-operated circuit breakers can transfer any critical load between secondary busses within milliseconds of system malfunctions resulting in loss of electrical power.

General purpose loads (those that can be safely interrupted for load reduction purposes) will be removed from their secondary busses to preserve the critical loads during any fault-clearing action and to limit the demand for power to that of the available power sources under the specific power failures. Extensive use will be made of computer subroutines to dispatch loads with electrolysis, operating, or maintenance requirements. One of the important benefits of computer-EPS integration is the ability to select mode of operations and then to status the required circuit breakers as a set. This is to be accompanied with a plain language verification of mode selected and a direct readout on the cathode ray tube for any circuit breakers or subsystem that did not respond to the execute command. This nearly instant confirmation ability has a direct bearing on cost of operation (e.g., Apollo lost many operating hours due to undetected operating errors).

Electrical power returns and ground will be a wired system isolated from structure except at a single point (Figure 3.2-10). The single-point ground system was chosen to eliminate interference due to ground loops and to eliminate dependence upon structure joints, straps, etc., in the return path.

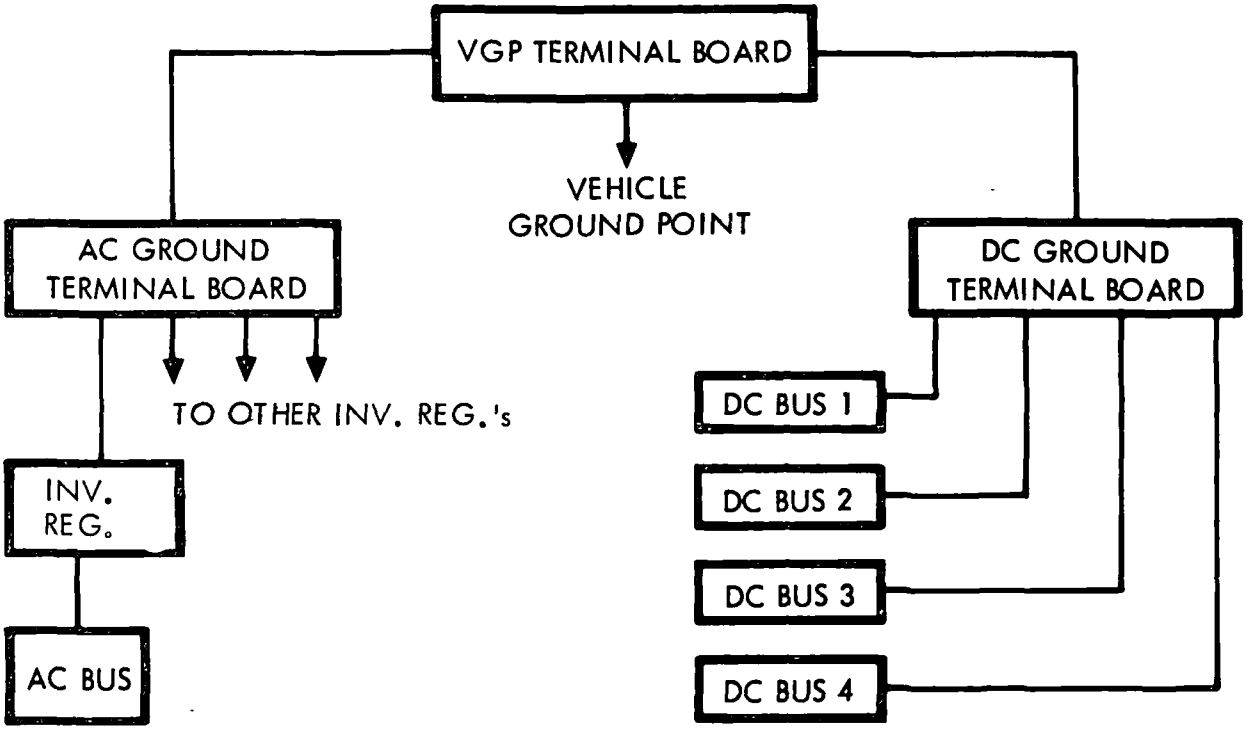


Figure 3.2-10. Single-Point Grounding System

EPS/ISS Interfaces

EPS controls and displays are designed to relieve the crew of the burden of continuous system status monitoring by transferring this function to the ISS computer. Meters as indicators are eliminated in favor of system status indicator (caution and warning) and displays in plain language to monitor all system parameters. Fully automatic protective and corrective controls, including problem-solving (fault diagnosis, load shedding and transfer, etc.) will be the function of the ISS. The complex control-protective functions are absolutely dependent on the ISS computer. Minimum control and protective functions are:

1. Overvoltage protection
2. Undervoltage protection
3. Overcurrent protection
4. Unbalance current protection
5. Overfrequency protection
6. Underfrequency protection
7. Differential feeder, external power, and distribution feeder protection
8. Automatic paralleling
9. Open phase protection
10. Load division protection (real and reactive)

The ISS will monitor the system, evaluate and assess the data, and automatically initiate the appropriate action or commands. Reaction time, particularly in the protective functions, must be kept to an absolute minimum. The concept would provide an integrated system to monitor and control EPS operation, bus loading and transfer, fault and out-of-tolerance operations, and subsystem on-off control for both manned and unmanned operation.

A completely automated system which will provide the startup, check-out, protection, and loading control is necessary to achieve unmanned operations. The program must be capable of accepting overriding commands from ground stations or other spacecraft.

The requirements for automated operation are not reduced during manned operation. The complexity of the systems and the rapid response times necessary make manual control of the systems impractical.

In addition to the protective functions previously listed, the integrated power control systems must provide:

1. Bus transfer commands
2. Load-shedding routines
3. Fault isolation
4. Peak load monitoring and initiating commands for additional power units
5. Load priorities
6. Sequencing of loads to minimize system transients due to load switching

Normal performance of the system will consist of system monitoring and load management. EPS parameters may be "called up" for display at the control center as desired. Malfunctions will be displayed as appropriate in the caution and warning display systems.

During power system emergencies the system would assess the available power capacity, determine essential functions, and energize loads with highest priorities. Manual override or priority changes would be initiated through inputs to the computer.

The major interface would be between the EPS and ISS; however, the integrated system also interfaces with all other subsystems through the computer-controlled solid-state switching devices. The signal interface between EPS and ISS is via the multiplex system. Some essential functions may require hardwiring to assure adequate reliability.

Split-parallel operation of the four channel system will be the normal mode of operation. Provisions will be made to operate in an isolated, full parallel or split-parallel, modes during malfunctions (optional). The regenerative fuel cells will be brought on line to handle peak loads (above solar array system capability) and during the dark (shadow) portion of the orbit. An independent emergency/auxiliary fuel cell system not only supplies emergency power in event of primary source failure but also the power for premanning, earth eclipses and when the solar arrays are rolled up during solar flare storms.

The power distribution system is designed to provide power from multiple sources in a System A and System B primary redundant distribution. Hardware that is expected to experience failure is modular and redundant in the sense that offline spares are turned on automatically when failures occur so that no loss of operation is experienced. Power source feeders are not redundant as the probability of a failure is extremely low and the arrangement of physical isolation nearly precludes multiple failures. The basic design for protection and isolation is reflected in the secondary bus, which includes the transformer-rectifier and the solid-state switching devices for all loads connected to the secondary bus.

Load equipment will be controlled and protected by information subsystem (ISS) through electrically operated solid-state switching devices having the following features:

1. Turn on control (pulse from ISS computer via multiplexer).
2. Turn off control (pulse from ISS computer via multiplexer).
3. Open and close status (pulse from and to ISS computer via multiplexer).
4. Overcurrent protection (pulse to ISS computer via multiplexer).
5. Load transfer (pulse from and to ISS computer via multiplexer).
6. Self-test (pulse from and to ISS computer via multiplexer).

The solid-state switching device (SSSD) will be in the active cooling loop along with other power conditioning equipment (i.e., transformer/rectifiers).

Most failures are expected to occur in the subsystems hardware supplied by each SSSD. All essential loads are simultaneously connected to both an A and a B secondary bus through an independent primary bus. Upon recognition of a fault (loss of bus power), the ISS initiates interruption of the faulted bus and turn-on of the alternate source using the circuit breaker as the switching device. It is possible in the case of some critical loads such as the ISS that power will be delivered from all four secondary busses to each redundant critical load in the ISS. By proper coordination of this EPS fault detection hardware and the ISS load management ability, the reliability of power to essential loads can be of a high order consistent with the requirements of an automated system.

Each solid-state switching device (circuit breaker) can be operated by either of two multiplex channels from the central computer. Each feeder, primary bus, and secondary bus is protected by a differential current protection scheme. Any isolation of a fault in these zones has to be repaired before restoring the affected hardware to service.

An electric shock-resistant design will be incorporated in the wiring that is connected to the circuit breakers. The net current between structure and the vehicle ground point is sensed in a manner to permit high speed turn-off of the affected secondary bus to prevent any fatal shock (i.e., that normally could occur from line-to-ground exposure).

OLS signals will be transmitted via coaxial cables and cables consisting of twisted pairs or twisted shielded pairs. The basic method for communicating between units will be via a multiplex system and the point-to-point wiring and associated hardware required for conventional wiring is drastically reduced. This produces a considerable savings in wiring weight and volume.

Miscellaneous minor components (i.e., electrically operated mechanical relay-type power contactors, connectors, wiring, etc.) will be space-qualified and will consist of off-the-shelf items to the maximum extent possible.

Other EPS Interface Requirements

The electrical power subsystem must provide the general power and electrical distribution required by the various subsystem users and in the form and quality defined.

The reaction control subsystem (RCS) provides storage for fuel cell reactants.

The environmental control and life support subsystem (ECLSS) provides for power conditioning equipment and fuel cell waste heat and water removal.

The EPS provides power to the experiment subsystem for an average demand of 4,000 watts at any one time. Peak power demand for experiments is limited to 7,000 watts (i.e., for durations less than one minute). A maximum sustained peak of 6,000 watts for a duration less than one hour is obtainable. Provisions are included so that any single experiment can use all or any amount of the power allotted; however, the total experiment loads

applied are limited to 4,000 watts including any electrical load demand from docked experiment modules. Power delivered to experiments are 80 percent ac and 20 percent dc with the standard docking adapter delivering only ac.

The major EPS interface requirements are summarized in Table 3.2-5.

Table 3.2-5. Electrical Power Subsystem Major Interface Requirements

Subsystem /Function	Interface
Onboard checkout	Checkout equipment prior to launch and prior to manning
Ground support equipment	External power during countdown
Structure	Heat dissipation for electrical cable runs
Subsystem to subsystem	Electrical distribution
Docking	Umbilical cables
Reaction control	Storage provisions for fuel cell reactants
Environment protection	Temperature control for EPS equipment (including F/C and RFC)
Information	Normal control of circuit breakers (SSSD's) Timing signals for synchronous operation of equipment Trend data on RFC, FC and solar cell degradation Electrical load management Signals for orientation of solar array panels Automatic control of RFC and FC
Guidance and control	General electrical distribution by EPS
All subsystems	Electrical power and distribution 120/208 V, 400 Hz, 3Ø, and 56 vdc
Docking	Standard power and control cables, ac only, 5 kw max

Weight and Volume Characteristics

Weight and volume summaries for the EPS are shown in Tables 3.2-6 and 3.2-7, respectively.

Heat Rejection Requirements

Heat rejection requirements of the EPS components are shown in Table 3.2-8. Temperature ranges for cooling requirements are given along with heat rejection estimates for each assembly.

Table 3.2-6. Solar Powered (RFC) EPS Weight Summary (8-Man)

			Initial Launch	Ten Years
SOLAR ENERGY CONVERSION			5,815	10,535
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
Fuel Cells (4 - 2.5 kw)		552		
Mounts and Supports		55		
FUEL CELLS--ELECTROLYSIS			1,050	3,150
Fuel Cells (4-5 kw)		721		
Electrolysis Units (4)		251		
Reactants		38		
Tankage		40		
POWER CONDITIONING			1,278	3,510
Controls	Core	60		
Inverters (4-5.0 kva)	Core	252		
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	3,567
Wiring		3,000		
Buses, Contactors	Core	58		
Mounts and Supports	Core	369		
Wiring	Boom	20		
Mounts and Supports	Boom	4		
LIGHTING			598	2,230
Lights		544		
Mounts and Supports		54		
TOTAL (Pounds)			12,799	23,599

Table 3.2-7. EPS Volume Summary

Function Assembly	Size (in.)			Volume (ft^3)		Estimated Replacements (10 Years)
	H	W	D	Unit	Initial Launch	
Primary power generation Solar array	25 in. x 31.1 ft	external		dia. 105.5	422	Replace all units 0 to 1 times/ 10 years
EPS hardware						
Fuel cells (emergency/auxiliary/ power source)	12	12	38	4.8	19.3	None
Regenerative fuel cells (RFC)	24	24	36	12.0	48.0	Replace 100% each 3-1/3 years or two times per 10 years
Power conditioners						
Inverters (boom)	8	9	12	0.5	10	
Inverters (core)	8	9	12	0.5	6	
EPS control	9.5	17	21	0.5	2	
Secondary bus structure Transformer/rectifier unit Circuit breakers	9	12	16	1.0	4	
Power distribution Contactors	4	4	5	0.062	0.86	Replace contactors two times/ 10 years
Feeders Wiring						
Lighting					17	Replace fixtures four to six times/10 years
Total					107*	

*Excluding solar array stowage

Space Division
North American Rockwell

Table 3.2-8. EPS Heat Rejection Requirements

Function Assembly	Temp. Range (F)	Heat Rejection		Allocation (%)				Comments Constraints
		Peak (kw)	Avg (2) (kw)	Cold-plate	Fluid Loop	Air Circul.	Struct.	
Power generation								
Solar array								
Array orientation							100	Inboard hardware cooled by structure
Fuel cells (core)	120	6.8	3.4		100			Emergency and premanned launch; daylight; see Note 1
RFC's (boom)	120	17.5	13.7		100			Dark
Power conditioning								
Inverters (boom)	150	5.5	5.0	100				
		4.4	4.0	100				Daylight; see Note 1
Inverters (core)	150	1.7	0.9	100				Dark
EPS control	150	0.6	0.4					
Secondary bus structure								
Transformer/rectifier								
Solid-state switching devices	150	0.7	0.6	100				Daylight; see Note 1
Circuit breakers		0.5	0.4	100				Dark
Power distribution								
Contactors	200						100	Loss included in EPS control
Feeders	400	0.3	0.2				100	
Wiring	(Under fault cond.)					50	50	May require some cooling
Lighting	130	3.8	3.0			90	10	Cooling can be a problem in depressurized volume

Note: 1. All coldplate hardware shall have 100% cooling capability upon one coolant loop failure. Degradation in performance upon second failure must support emergency and backup requirements.

2. Dark/light peak and average for station on worst case day. Load span at one location can be 50% of station.

Maintenance Requirements

Equipment requiring maintenance must be installed so that extra-vehicular activity (EVA) operations can be minimized. Replaceable units must be readily accessible with minimum disturbance of other units. At present, the primary mode of replacing the solar array is by completely replacing the OLS power boom (full replacement); hence the deletion of any requirement for EVA operation (Figure 3.2-8). Maintenance (i.e., single solar panel replacement) is highly desirable. This would involve EVA and modifications to the solar array design. However, the orientation hardware is capable of being maintained since a shirtsleeve environment is provided.

Installation of all internal units is designed to permit direct visual and physical access by pressure-suited crew with connectors and couplings provided for ease of removal and replacement. For installation location of assemblies refer to Figure 3.2-8.

3.2.2 SELECTION RATIONALE

Parametric trade data for candidate EPS concepts is presented in Volume 2, Configuration and Systems Analysis Report. Based on these data, major trades were conducted to select the preferred EPS energy sources, conversion equipment, and storage concepts. Representative EPS candidates were evaluated on the basis of weight, cost, safety, technical risks and operational considerations. In addition, a power distribution trade (ac versus dc) was conducted.

Primary Power Sources

The primary electrical power sources, preferred and rejected candidates with associated rationale, are summarized in Table 3.2-9. Note that the viable candidates have been reduced to three, nuclear reactor, radioisotope and solar.

The candidate finalists were evaluated on the basis of operational considerations (Table 3.2-10). All candidates induce some operational constraints but none of the constraints were found to govern the selection of the source. Hence, operational considerations were reduced to a passive role.

Next, power candidates were evaluated on the safety issues (Table 3.2-11). Since nuclear safety is not an issue with solar power, solar was rated number one, with nuclear reactor second, and radioisotope third.

Although large in physical size, the solar power-source concept requires no technology breakthroughs. Because of its size, solar power does represent an extension of the present state of the art but no real development problems are foreseen. It is believed that the design, development, fabrication and qualification of the required 20 kw solar power source can be accomplished over a six-year period, hence the solar concept is well within the target IOC date (1983). For the above reasons, the solar power

Table 3.2-9. Primary Electrical Power Sources

Preferred Candidates/Rationale	Rejected Candidates/Rationale
Nuclear reactor (ZrH) Brayton ZrH advanced state of development Brayton high performance	Nuclear reactor Fast reactor Thermionics Potassium rankine } Difficult technology (risk) Mercury rankine Thermoelectrics } Low efficiency, High demand on reactor and/or radioisotope
Radioisotope (Pu-238) Brayton Pu-238 long half-life Brayton high performance	Radioisotope (cobalt 60) Safety uncertainties
Solar array (silicon cells) Experience factor	Solar array (cadmium sulfide thin-film cells) High weight Low efficiency More development (increase area)
	Chemical power Weight (resupply)

Space Division
North American Rockwell

Table 3.2-10. Operational Considerations

	Nuclear Reactor	Radioisotope	Solar
Docking	1	1	2
Shadowing (solar)	1	1	2
Radiation (external)	3	2	1
Controls (stabilization)	2	1	3
Radiation Hazards (maintenance)	2	3	1
Crew Safety	2	3	1
Communication Interference	1	1	2

Ranking:
1 = best
2 = 2nd best
3 = worst

Conclusions:
All sources induce some constraints
None govern selection

Space Division
North American Rockwell

Table 3.2-11. Radiation Safety Issues

Radioisotope	Uncertainties exist in Fuel form Atmospheric contamination on launch (fuel release probability) Reentry effects (containment) Note: Considerable work is needed to remove uncertainties and to satisfy safety criteria
Reactor	Relatively safe until start-up Large isotope source from fission product Reentry/disposal at end of life
Solar arrays	Non-nuclear (nuclear safety not involved)

source has been rated one (least risk) as a technical risk. Radioisotope source has been rated number three (highest risk) on the basis that safety aspects present design problems, the solutions of which remain to be resolved, and because there is some doubt as to the assurance of fuel availability in the required amounts by 1983. A reactor power source represents less technical risk than a radioisotope (development further along and less risk of fuel availability), hence was rated number two.

Candidates were evaluated on the basis of weight (Table 3.2-12). Note that the weight parameter was divided into three parts, functional, configuration sensitive and resupply. Solar power was ranked one for functional and configuration sensitive and the radioisotope number one for resupply. Overall weight (10 years) favors the solar power source.

Finally, the candidates were evaluated on the basis of cost (Table 3.2-13). Cost was also divided into three parts (similar to weight). Manpower has the lowest functional and configuration sensitive costs and the radioisotope the lowest resupply cost. Overall cost (10 years) favors the solar power source.

Summarizing, solar power was selected for the primary power source on the basis of weight, cost, safety, and least technical development risk.

Energy Storage

Two candidate concepts for storing energy were evaluated, batteries and RFC's. Weight and cost considerations for the batteries and RFC's are based on resupply each 10 years of three times and two times, respectively. Weight comparisons of the candidate concepts are shown in Figure 3.2-11. Note that both initial and resupply weights favor RFC's by a very substantial margin.

Three parameters considered in the cost evaluation are development, nonrecurring and resupply costs with resupply based on $1000/pound. Based on previous NR studies, development cost for RFC's was approximately 30 percent higher but the nonrecurring cost for RFC's was about 12 percent less. The most significant cost parameter is resupply where the RFC value is approximately 33-1/3 percent of that of the batteries. Overall costs for 10 years favors the RFC's by approximately 35 percent. Costs related to subsystem interface complexity (ISS) are chargeable to batteries and are not included in this evaluation.

Operational considerations, technical risks, and safety were not driving influences on energy storage selection, therefore, the selection was based on weight and costs which clearly favored the RFC's.

Secondary Power Sources

Secondary power source concepts considered were redundant primary EPS and a backup EPS. Backup power sources considered were radioisotope and fuel

Table 3.2-12. EPS Candidate Weight Comparison Matrix

Candidates	Functional (Incl. Shields)	Configuration Sensitive	Resupply	Overall Cost
SA (batt)	0.70	0.07	0.97	1.74
SA (RFC)	0.50	0.07	0.43	1.00
RIB (batt)	1.01	0.27	0.45	1.73
RIB (RFC)	0.93	0.27	0.23	1.43
NRB (batt)	1.48	1.07	1.27	3.82
NRB (RFC)	1.40	1.07	1.04	3.51

- SA (RFC) lowest initial weight
- SA (batt/RFC) lowest configuration sensitive weight
- Radioisotope (RFC) lowest resupply
- SA (RFC) overall 10-year weight advantage

Ranking: Lowest value = lowest weight

Table 3.2-13. EPS Candidate Cost Comparison Matrix

Candidates	Functional	Configuration Sensitive	Resupply (Incl. launch costs)	Overall Cost
SA (batt)	0.62	0.18	0.29	1.09
SA (RFC)	0.62	0.18	0.20	1.00
RIB (batt)	1.60	0.20	0.11	1.91
RIB (RFC)	1.60	0.20	0.09	1.89
NRB (batt)	0.90	0.94	0.25	2.09
NRB (RFC)	0.90	0.94	0.23	2.07

- Solar array lowest functional cost
- NRB most configuration cost sensitive
- RIB lowest resupply
- 10-year overall cost favors solar arrays

Ranking: Lowest number = lowest cost

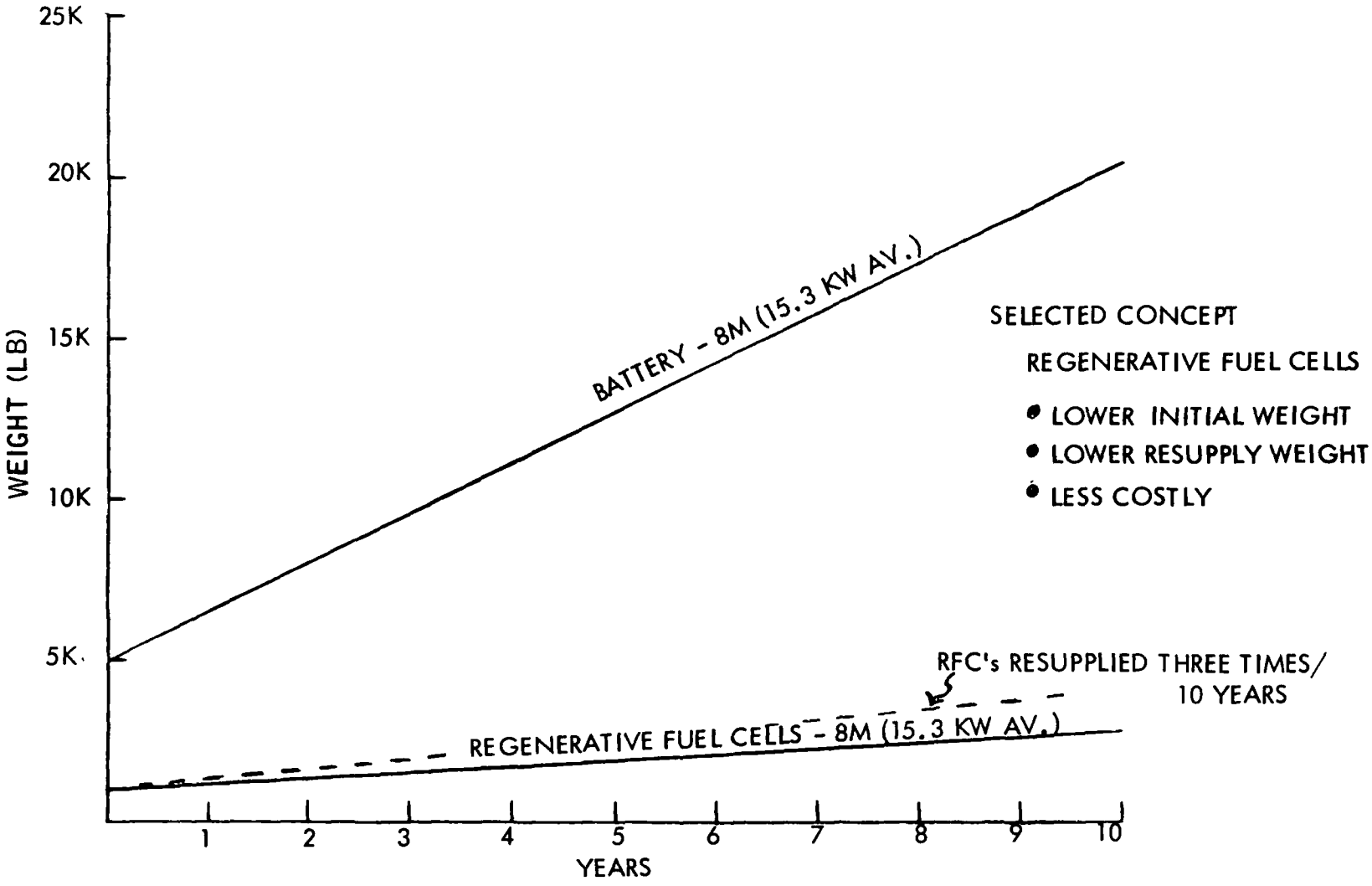


Figure 3.2-11. Weight Comparison of Energy Storage Candidate

Space Division
North American Rockwell

cells. The radioisotope source requires a second development program while fuel cells require excessive reactant storage capacity. Both backup sources would result in a substantial increase in EPS weight and cost. The selected primary EPS concept included four independent arrays and RFC's with initial oversizing for degradation. The inherent redundancy of the primary EPS solar array concept permits secondary power requirements (13.5 kw for 90 days) to be met without requiring a backup EPS source.

Emergency/Auxiliary Power Sources

The emergency/auxiliary power source supplies power not only during emergencies but also for premanning, earth eclipses, during solar flare storms when the solar array is rolled-up and during periods when maintenance of the primary EPS source is required. Power requirements are shown in Table 3.2-14.

Table 3.2-14. Emergency/Auxiliary Power Source Requirements

Mission Phase	Power Requirements	Duration
Premanning	2 kw	5 days
Emergency	4 kw	30 days (48 hours earth orbit)
Solar flare storms	10 kw	72 hours
Earth eclipse	15.2 kw	3.5 hours

Three candidate power sources were evaluted: batteries, fuel cells and radioisotope. Batteries were eliminated on the basis of weight (initial and resupply) and costs. Radioisotope power sources were deleted on the basis of nuclear safety considerations, weight and cost (requires a major development program). Fuel cells (including weight penalties for the required readouts) are light weight (both initially and considering resupply), are low in cost and are ideally flexible (i.e., H_2O byproduct desirable for OLS use, little/no penalty for large variations in load demands). The selection of fuel cells as the emergency/auxiliary power source was primarily based on weight and cost.

Power Distribution

A solar array output can be adjusted to give a wide range of voltage values. The range of voltages considered is narrowed when odd, non-standard values are eliminated. When parameters such as development cost and time and established reliability are considered the choice is further reduced to the standard ac and dc voltage values (i.e., 120/208 volts ac, 28 volts dc, 56 volts dc, and 112 volts dc).

The relatively long distance involved in the OLS distribution drive the design toward higher voltages to reduce distribution losses. This in effect eliminated consideration of 28-volt dc distribution. Compatibility and commonality with OLS docking vehicle requirements also had to be considered in arriving at an optimum design.

One of the decisions driving the power distribution selection was the need for remotely controlled circuit breakers that can be operated by computer command and position verified by computer interrogation. This decision influenced the voltage selection. Solid-state switching devices (circuit breakers) have a high voltage loss (0.35- to 0.5 volts dc and 1.5 volts ac, which is nearly independent of voltage level). This and the relatively long distance between circuit breakers and load calls for the use of the highest practical voltage which in turn permits the use of a smaller wire gauge and a wider voltage regulation allowance. Another factor in selection was that unmanned operation of subsystems, plus the need for fail-operational and fail-safe performance, dictates the need for a power subsystem with built-in test and decision-making ability.

In tradeoff studies, a medium voltage, 112-volt dc, distribution system was used as the basis for dc system calculations. A 112-volt dc system provides an acceptable distribution wiring weight compatible with available system hardware capability and consistent with environmental restrictions. The corona problem during transitions between pressurization and depressurization and EMI effects are considered the primary problems. The lack of power contactors of acceptable weight and life with sufficient interrupting rating may prevent the mechanization of dc voltage for primary distribution. A 56-volt dc, two-wire system was selected for secondary distribution based on the need to use voltages higher than 28 volts dc to accommodate the distances involved between distribution busses and loads.

An additional consideration in the selection of 56 volts dc was the need for the electrically operated solid-state circuit breakers. The loss in these units (between 0.35 and 0.5 volt for the dc case), represents too large a percentage loss in a 28-volt dc system when it is added to the normal one-volt drop used in wiring steady-state dc loads. The use of voltages higher than 56 volts dc was avoided due to the extremely difficult switching problems encountered in higher dc voltages. Toggle switches and relays were not found to be practical for OLS loads at these higher dc voltages. Therefore, the 56 volts is accepted as a reasonable compromise. Fifty-six volt dc relays have been utilized in some flight-type hardware and are available, if required.

One of the major considerations in the selection of an ac distribution concept over a high-voltage (112 volts or higher) dc concept is the problem of providing switching in the dc system. Light weight and long life direct-current contactors do not presently exist which can handle the loading required in the OLS, whereas ac components and the supporting technology are readily available. Alternating current contactors are provided with the advantage of current passage through zero to help extinguish the arc and are inherently able to interrupt more power with a given size and weight than dc contactors. In addition, dc contactor development for aerospace applications has been limited well below the capacity of ac contactors. There-

fore, wherever a choice is possible, ac contactors are desirable to carry the higher power levels during both steady-state and transient interruption conditions. Contactors must be derated well below their maximum capability to provide for high reliability and longer life, as well as to meet fault and transient requirements. There is a direct correlation between the amount of current that is interrupted and the number of cycles before failure through damage to the contacts.

Hermetically sealed connectors are required for OLS pressure bulkhead penetrations. Spacing of pins for ac power is intended to be sufficient to assure safe operation in terms of corona tracking even if a hermetic connector leaks when pressure-cycled through an equivalent altitude of 100,000 to 140,000 feet. The total time of each exposure to this intermediate equivalent altitude pressure is, of course, very limited. Spacing of connector pins to assure safe operation in terms of corona tracking is greater for dc power (larger connectors) and the EMI problems are more severe due to higher induced switching transients. However, power conditioning requirements are somewhat less for dc.

In general, environmental-type connectors will be used within a pressurized volume with the same corona and ac power-pin spacing as described for hermetic connectors. Development problems in connectors center around the following:

1. Need for higher voltages
2. Need for connectors that can be mated by crew wearing pressurized suit.

The need for higher voltages is preferred to the alternative of high current and brings about the need to use voltages up to 208 volts ac on many connectors. The trend toward miniaturization in black box construction and in connectors is not compatible with the need for larger pin spacing or for installation arrangements for gloved-hand connection and disconnection.

Based on switching equipment availability, corona effects, EMI considerations, connector sizes and switching component life, an ac distribution system was selected for the OLS. A 120/208 volt ac distribution system was selected based on the decision that 208 volts can be satisfactorily designed to minimize corona effects during transitions between pressurized and depressurized environment.

The frequency was chosen at 400 Hz rather than 800, 1,600, or 3,200 Hz based on the following:

1. Extensive present usage of 400 Hz
2. Large increased cost for other subsystem developments for higher frequency than 400 Hz
3. Bearings have been developed for fractional horsepower motors for speeds up to 24,000 rpm (2-pole speed at 400 Hz). The OLS will require small motors for such tasks as air circulation and fluid circulation. The suitability has not been demonstrated of motors of higher speeds or multiple-pole designs that would be required to apply higher frequencies with the equivalent reliability.

4. Standardization for subsystem hardware on OLS that is optimum for use on docking vehicles may penalize OLS weight to a small degree but will save the dollar cost of two frequency standards.

The selected ac concept for power conditioning and distribution employs a high degree of modularity primarily because modular equipment with computer operational control provides the benefits of substantial improvement in equipment efficiency. Only those modules required to support the load are utilized at any point in time. This improves system efficiency for all the many hours that loading is less than full load. Utilization of this energy can reduce the propellants required for propulsion, provide reserve electrolysis of water, and add to the flexibility of operations. In addition, heat rejection requirements will be reduced. A comparison summary of ac versus dc power distribution is presented in Table 3.2-15.

Trade Summary

Key EPS trades are summarized in Figure 3.2-12. The selected EPS configuration consists of solar arrays for primary and secondary power, regenerative fuel cells for energy storage and fuel cells for emergency/auxiliary power. Distribution will be basically 120/208 volt ac with some 56 volt dc power available.

3.2.3 ADAPTABILITY TO ARTIFICIAL GRAVITY

The preferred OLS artificial gravity concept is shown in Figure 3.2-13. Key features of this configuration are the elimination of artificial gravity loading on the solar panels by locating the solar array, tugs, and experiment module on a zero-gravity hub. Optimizing the EPS to minimize distribution losses may mean the relocation of the primary busses and associated equipment (emergency fuel cells and inverter-regulators) presently located in the OLS torus areas to the vicinity of the zero-gravity hub and the addition of rotating contacts between the zero-gravity hub and the OLS. Longer feeders are required which reflects in a weight increase for the EPS.

An alternate artificial-gravity concept may be a configuration which produces artificial gravity loading on the solar panels. This concept would require stiffer panels (increased weight of approximately 600 pounds) and/or use of double the number of solar panels coupled with a 50-percent reduction in panel deployment.

Table 3.2-15. AC Versus DC Power Distribution*

Item	AC	DC
Mechanization equipment availability	Excellent	Poor
Corona effects	Good	Poor
Wiring (weight)	Good	Fair
EMI	Good	Poor
Power conditioning **	Good	Excellent**
Switching component life and capability (reliability)	Good	Fair
Connector sizes	Good	Poor

* 120/208 vac, 3 Ø, 400 Hz versus 112 vdc, 2 wire

** Approximately 250 pounds different in power conditioning weights

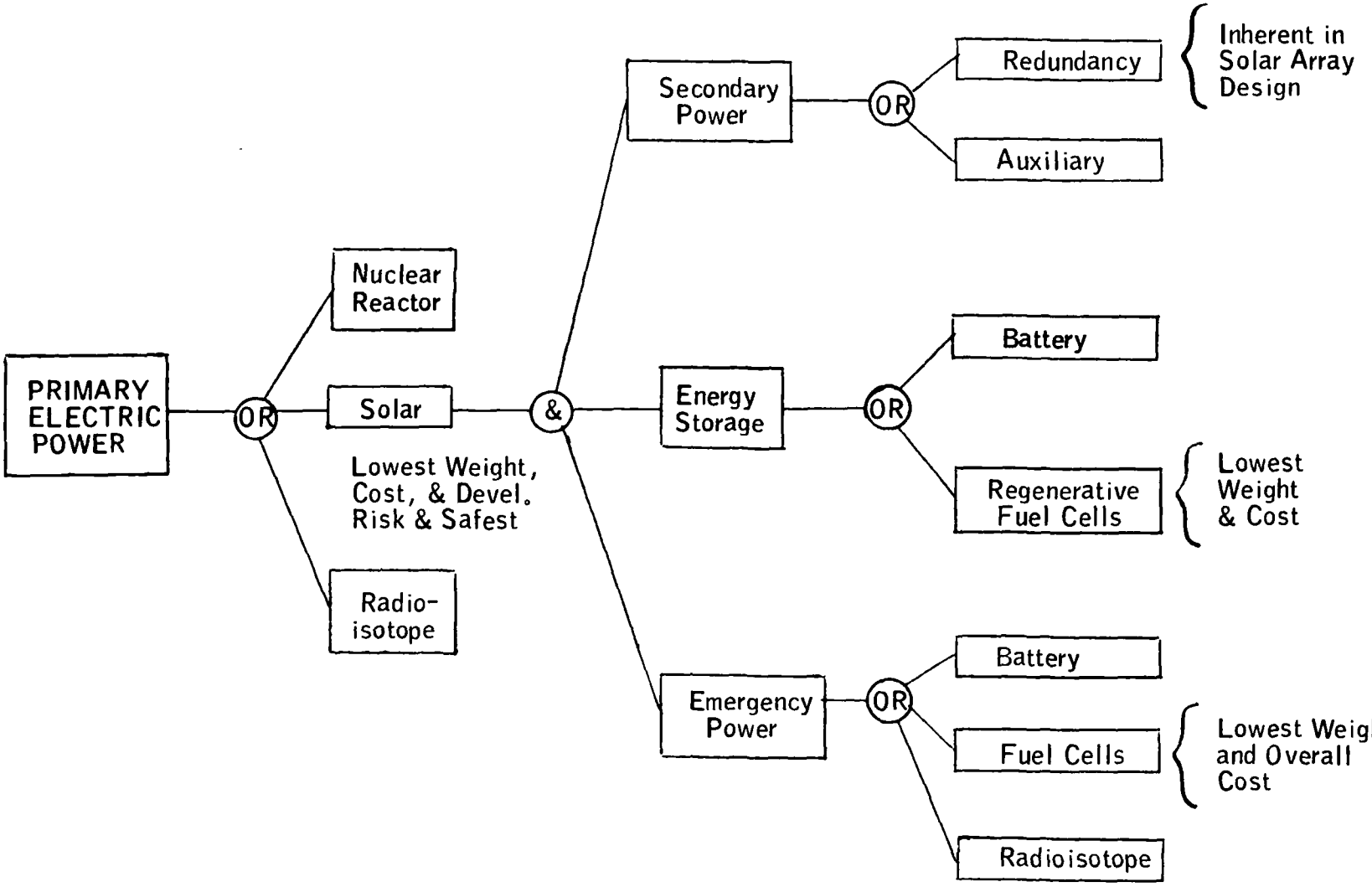


Figure 3.2-12. Summary of EPS Trades

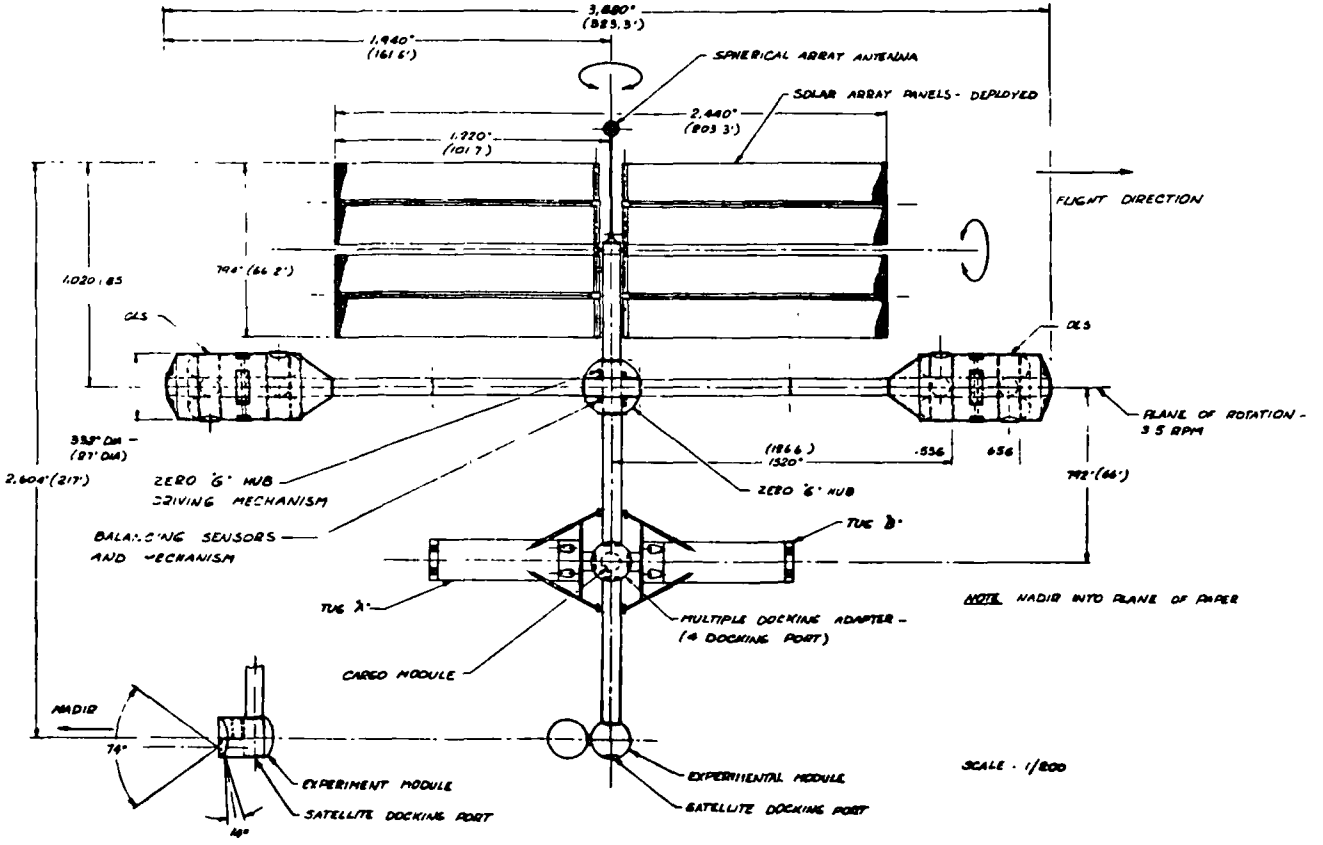


Figure 3.2-13. Artificial G Concept

Space Division
North American Rockwell

Page intentionally left blank

3.3 INFORMATION MANAGEMENT

The OLS information subsystem (ISS) consists of the equipment and installation hardware necessary for providing communications and for the handling, display, and control of data in support of scientific experiments and station operations. Figure 3.3-1 illustrates the functions of three major assemblages of equipment making up the selected concept, and their interface through software with the OLS crew. Specific functions are indicated in the interfaces with other subsystems as shown in Figure 3.3-2.

3.3.1 CONCEPT DEFINITION

The ISS concept consists of a large number of individual components which are grouped into three major assemblages, the Data Processing Assembly, the Displays and Controls Assembly, and the Communications and Tracking Assembly. Figure 3.3-3 shows the major assemblies of the ISS and indicates the types of equipment making up each assembly. In addition, the associated software is shown. The relationship of the components to each other and to other subsystems is shown in the simplified block diagram of Figure 3.3-4. An overall description of the ISS will be presented, followed by more detailed descriptions of each of the assemblages.

Data Processing Assembly

The data processing assembly will acquire soft and hard data which will be collected, conditioned, and combined for subsequent data processing functions. The assembly will acquire command and control data, maintenance and repair data, and experiment support data, which may be analog or digital. Hard data will include photographic film, specimens, and physical data.

The distribution function will control the routing of the data to the appropriate terminal and the data buses.

There will be two basic types of storage of data, non-computer and computer. The non-computer data storage will include storage of texts, microfilm, photographic film, and experiment samples. The three classes of computer data storage will be as follows:

1. Operating storage will be the near-zero access time storage from which instructions are executed.

2. Mass storage will be supplemental to the operating storage and have rapid access time.

3. Archival storage will be the moderate access time off-line storage and will be directly connected to and controlled by the computer.

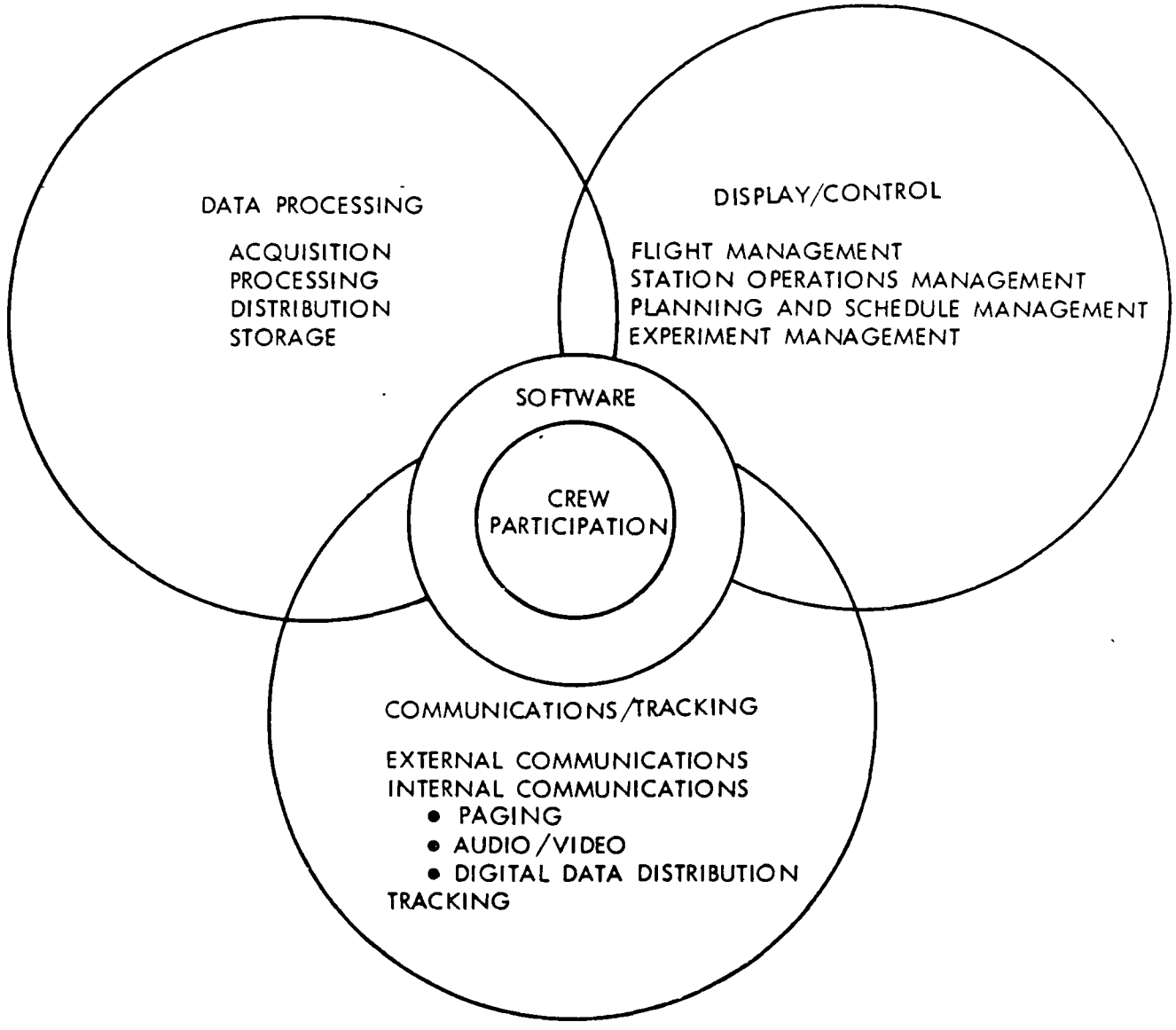


Figure 3.3-1. ISS Functional Assemblages

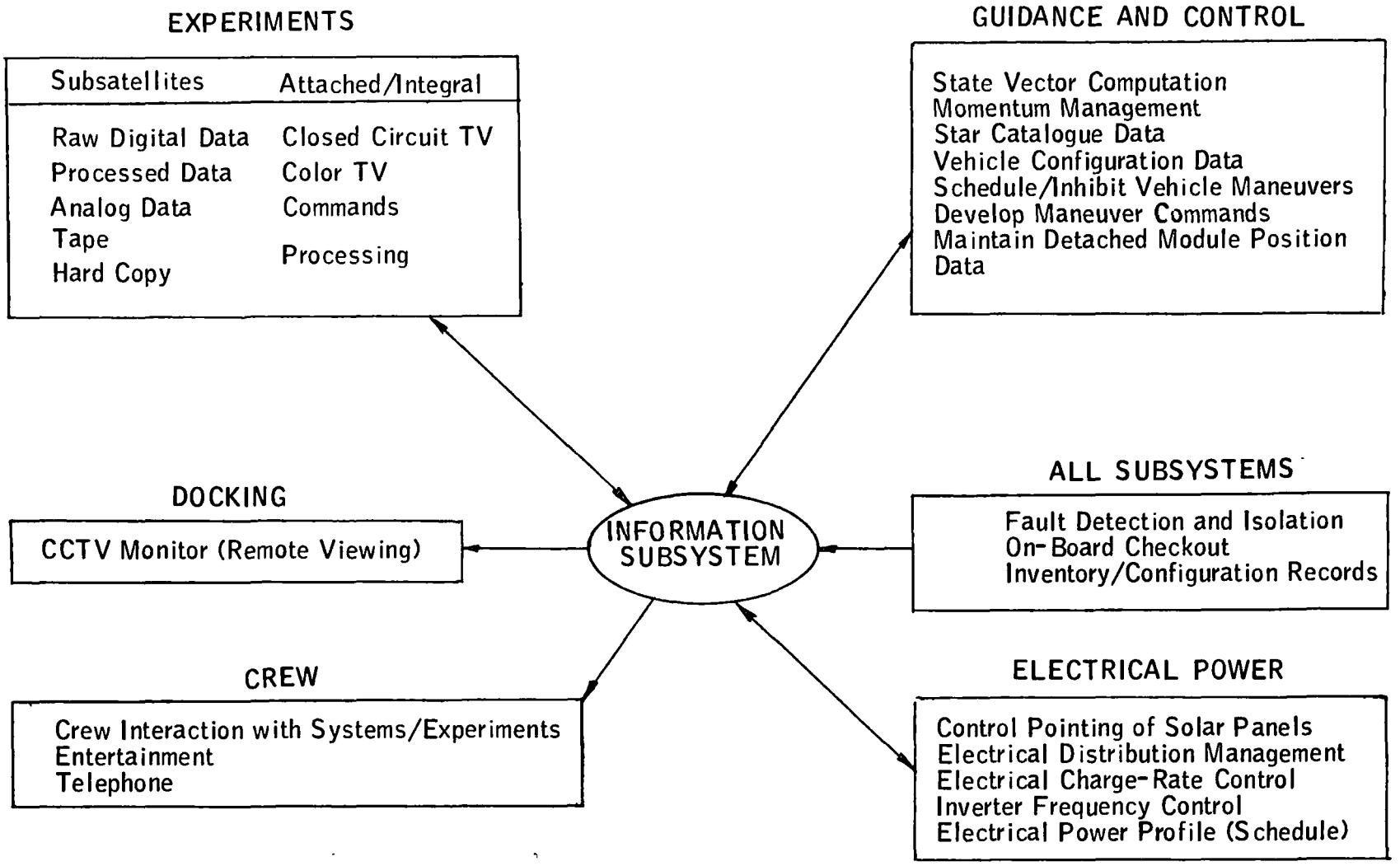
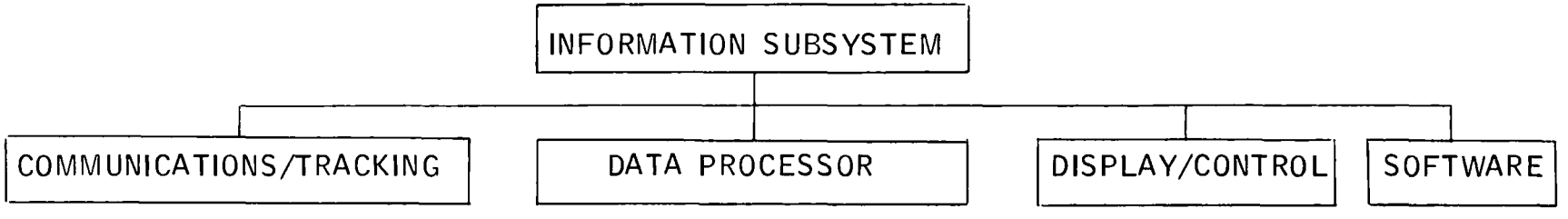


Figure 3.3-2. ISS Subsystem Interfaces



Directive Antenna
Semidirectional Antenna
Approach Radar
Communications Rack
Power Amplifier and Multiplexer
Transponder
Modulation Processor
Central Switch
Telephones
Closed Circuit Television
Data Recorders
Data Bus
Audio/Video Units
Facsimile

Operational Memory
Processor
Input/Output Unit
Mass Memory
Archive Memory
RACU's

CRT
Alphanumeric
Microfilm
Caution/Warn
Keyboards
Hand Control
Printer

Supervisory
Application
Support
Reference
Tape/Film

Figure 3.3-3. ISS Assemblies

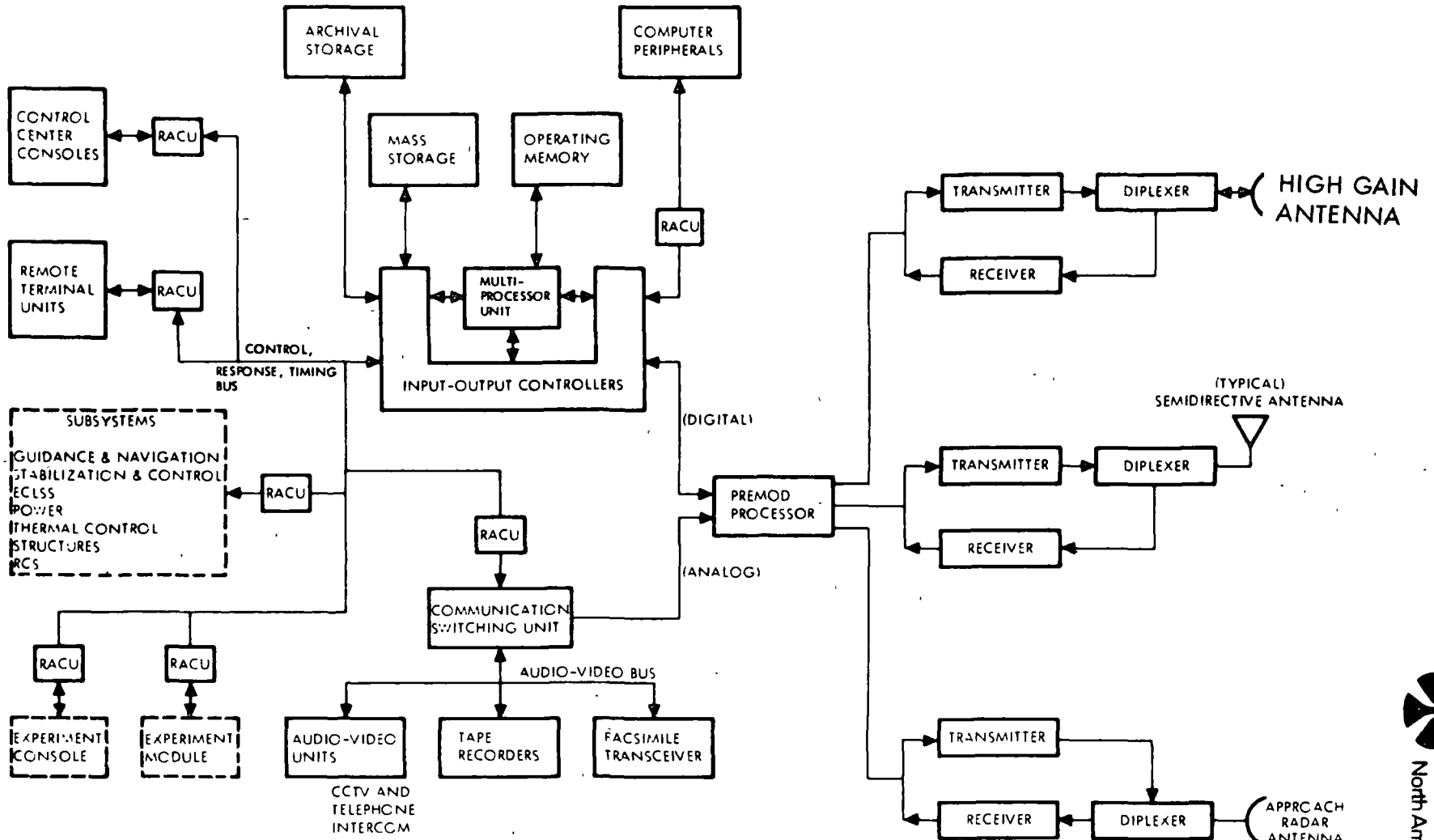


Figure 3.3-4. Information Subsystem Block Diagram

Data processors are configured into two centrally-located multiprocessing assemblies. One is located in the primary control center; the second in the experiment and/or secondary control center. The assembly in the primary control center consists of two Input/Output (I/O) devices and processor units, four operating memories, a mass memory, and an archival storage unit. Since much of the data to be processed is of an experimental nature, three I/O devices and processors are located in the experiment control center to service experiment data. A mass memory to mass memory data transfer is effected via dedicated lines between the mass memory and I/O devices located in the control centers.

Using this concept, processors of differing performance capabilities can be connected to the digital bus. This allows the performance of the system to be modified to meet varying program requirements after the system is operational. Its modular structure enables a precise blend of components to meet the varying speed and capacity requirements of the OLS mission.

Processor Unit

The processing subassembly consists of five general purpose processor units each with an incorporated read only storage (ROS) and a scratch-pad section. The processor unit will have standard instruction repertoire, full arithmetic capability with both fixed and floating point notation, interrupt handling system, byte manipulation and double precision arithmetic operations, and minimal I/O capability to enable it to communicate with other equipment in the subassembly. Each processor is equipped with all functions for execution of instructions including executive control. Included in each processor is its own set of control registers providing multiple accumulators, index registers, and I/O access control registers. Processors will be assigned tasks on a predetermined basis.

Input/Output Control Units

The input/output (I/O) control units will provide the interface between the processor units and the digital control and response buses. The I/O control units will relieve the processor unit of the time consuming chore of servicing the relatively slow remote devices. It will perform such services as data formatting buffering, termporary information storage, and may contain special purpose subroutines. The I/O units will be under direct control of the processor unit in communication with it. It will contain pointing registers, access control registers (including external function), normal I/O and communications.

Operating Memory

The operating memory will have both read and write capabilities and serve as a logical extension of the memory located in the processor units. The memories will be addressable by any processour unit under control of the master executive program. The operating memory will be a plated wire random access, non-volatile, non-destruct readout (NDRO) memory having a minimum of 8 ports and capable of operation in a multiprocessing configuration. It will employ small block segmentation and interleaving within each module and among

the modules. The main storage provides a high performance, immediate access area for instructions, data, and I/O communications. Its design supports the basic multiprocessing concepts as well as the design goals of modularity and reliability.

Mass Memory

The mass memory subassembly consists of a multiplicity of plated-wire storage devices. These bulk storage units will provide the required operational programs, the program alternatives, reference information, data base information as well as any special information necessary to operate the OLS or to perform the mission objectives. These units will also be used to store data needed for long term analysis and computations (i.e., trend data, guidance/navigation parameters, electrical power profiles, consumables status, schedules, etc.), and frequently called-for-data which would normally be stored in the archival storage unit.

The mass memory storage units will be modular in nature (Figure 3.3-5) consisting of six modules in the primary control center and seven modules in the experiment control center. The mass memory will be a random access, non-volatile, non-destructive readout magnetic memory of 10^8 bits. Power switching will be employed to reduce average power consumption.

Archival Storage

The archival storage unit consists of two high-speed cartridge tape units. One unit is located in the primary control center and one in the experiment control center. The archival storage information is of a historical nature and not necessarily active at all times. Experiment results, subsystem performance, station log data, crew status, etc., are types of data stored in this memory. This data (cartridge) will be transported to the ground by the logistic vehicle or sent to the MSFN.

Remote Acquisition and Control Unit

The primary function of the Remote Acquisition and Control Unit (RACU) is to accept data from the subsystems, experiment sensors, and the display/control consoles; and digitize and/or format the data for transmission to the processor assembly. The data is addressed to the sensing RACU (command/control) and to the receiving unit (response). The receiving unit could be the input/output unit, display unit or external communications subassembly.

The RACU has three methods of operation, (1) normal data acquisition, (2) service request, and (3) data transfer. In each of these modes, the operation of the RACU is essentially the same. The RACU's receive instructions and release data at the command of the software routines in the processor units. Digital data is transmitted to the RACU on the control (command) bus and from the RACU on the response bus. Normally, each RACU is addressed in a mapped predetermined manner; however, where emergencies require or when directed by the processing unit, the RACU will perform the requested function. A block diagram of the RACU is shown in Figure 3.3-6.

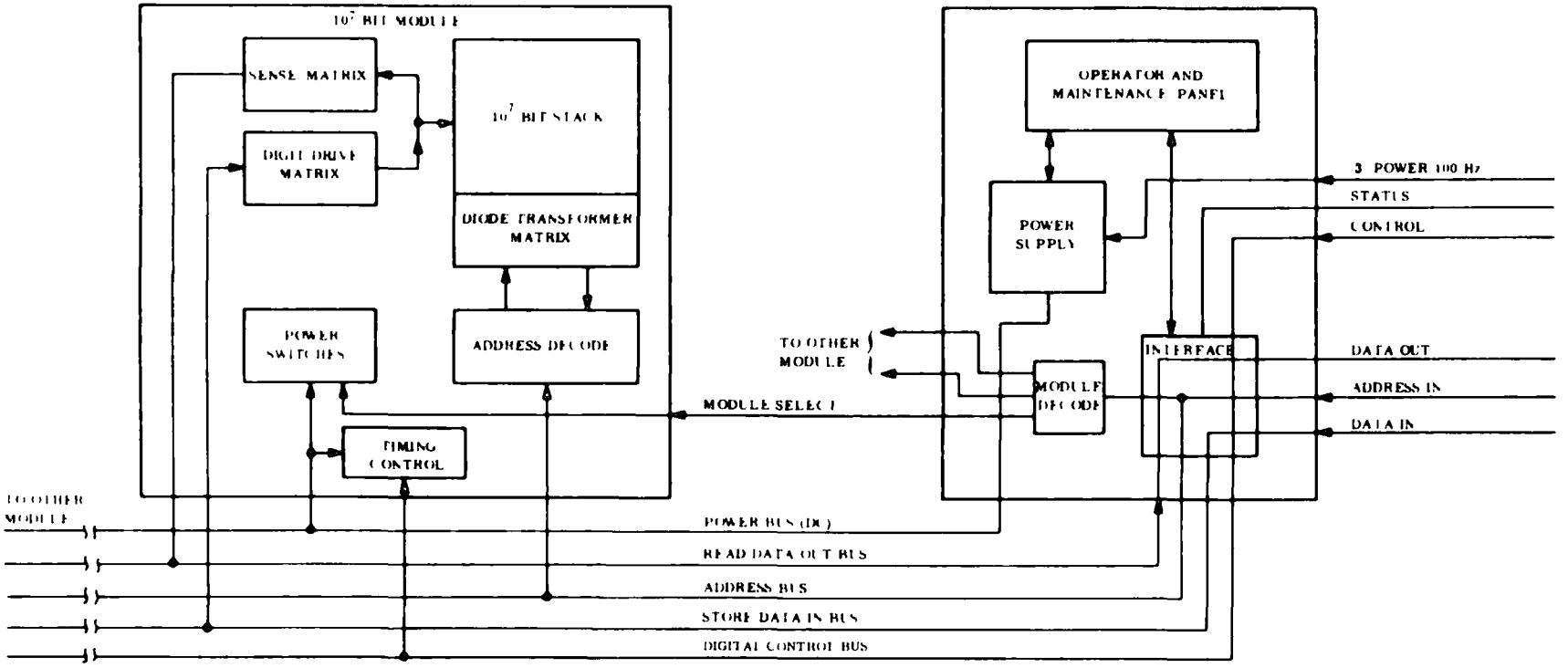


Figure 3.3-5. Mass Memory Configuration

Space Division
North American Rockwell

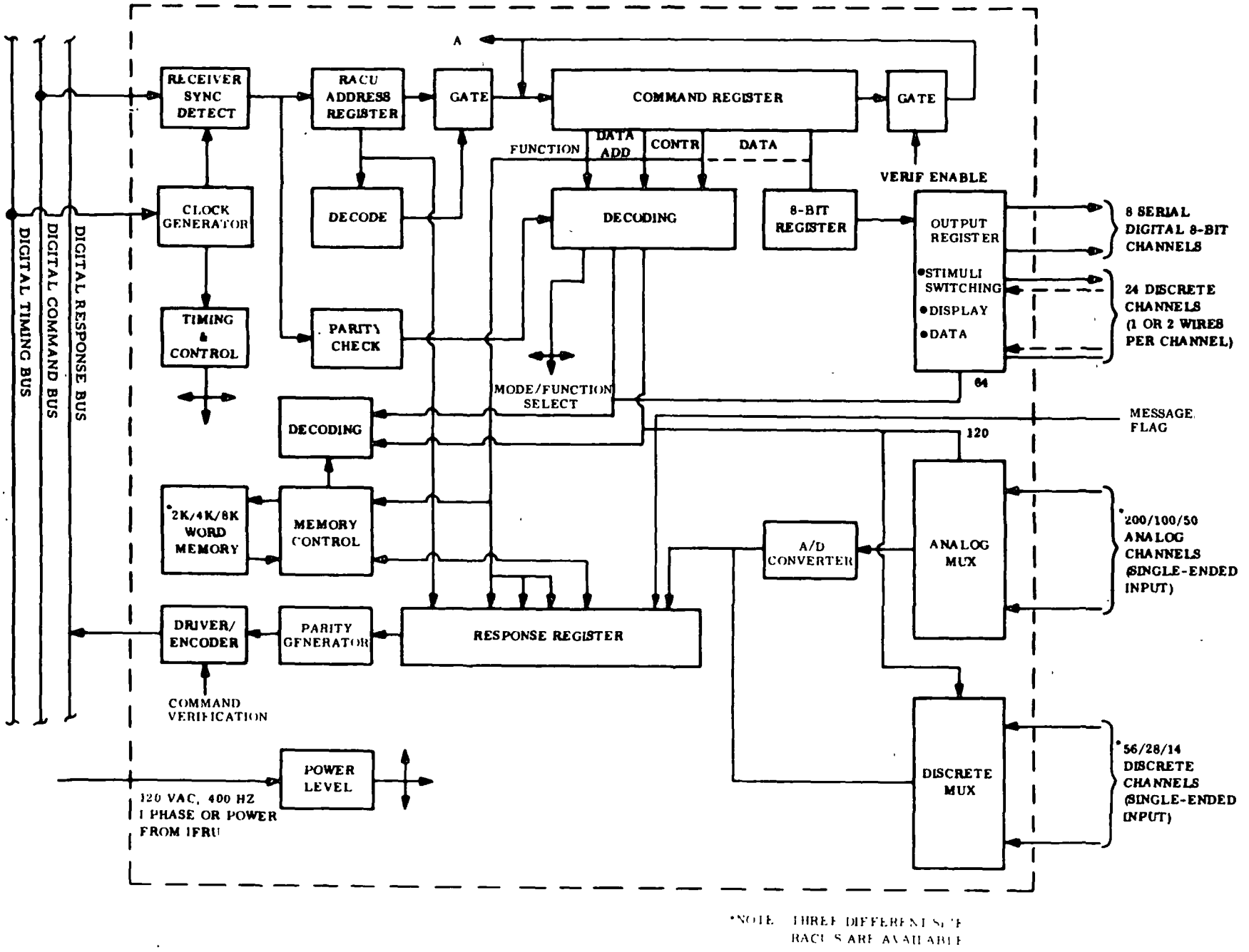


Figure 3.3-6. Remote Acquisition and Control Unit

Space Division
North American Rockwell

Displays and Controls Assembly

Three areas have display and control capability. In the primary control center one position will normally be utilized to perform the flight management function. The other position will be utilized to perform OLS operations management and planning and scheduling management functions. The experiment control center will normally be utilized to perform experiment management functions. The display and control equipment in both control centers will be physically identical in design and configured by software to the desired configuration. The commander will be provided with sufficient displays and controls in his quarters to perform planning and scheduling management. Displays and controls consist of two types of equipment integrated into a single package. The annunciation (message) associated with each display/control will be configured by software. Any information which is in the memory bank can be called for display at the console by the operator's entry of proper address using the keyboard.

The displays and controls assembly consists of three standardized 50-inch wide and 48-inch high consoles (Figure 3.3-7). Two consoles are located in the primary control center, and one in the experiment control center. Normally the consoles in the primary center are utilized for OLS control and subsystems management while the console in the experiment center is utilized for experiment management and control.

The assembly is a hybrid arrangement, allowing experiment and subsystems control from the same consoles. The experiment center console may be used for OLS control when both primary control center consoles are off-line.

Design of all units in the consoles is based on standardized panels which are interchangeable with like panels in other consoles. Each console consists of the following subassemblies:

Alphanumeric displays - 2
Multiformat color CRT display - 1
Microfilm viewer - 1
Function select keyboards - 2
Numeric keyboard - 1
Display address keyboard - 1
Alphabet keyboard - 1
Intercommunications telephone - 1
Hand controller - 1
Printer - 1
Critical digital displays - 2
Monitor/alarm displays - 1

In addition to the display and control consoles there will be two remote terminal units. The RTU's will be portable units which can be carried throughout the station, plugged into the data bus system, and used for local control, adjustment, or checkout of subsystems and experiments.

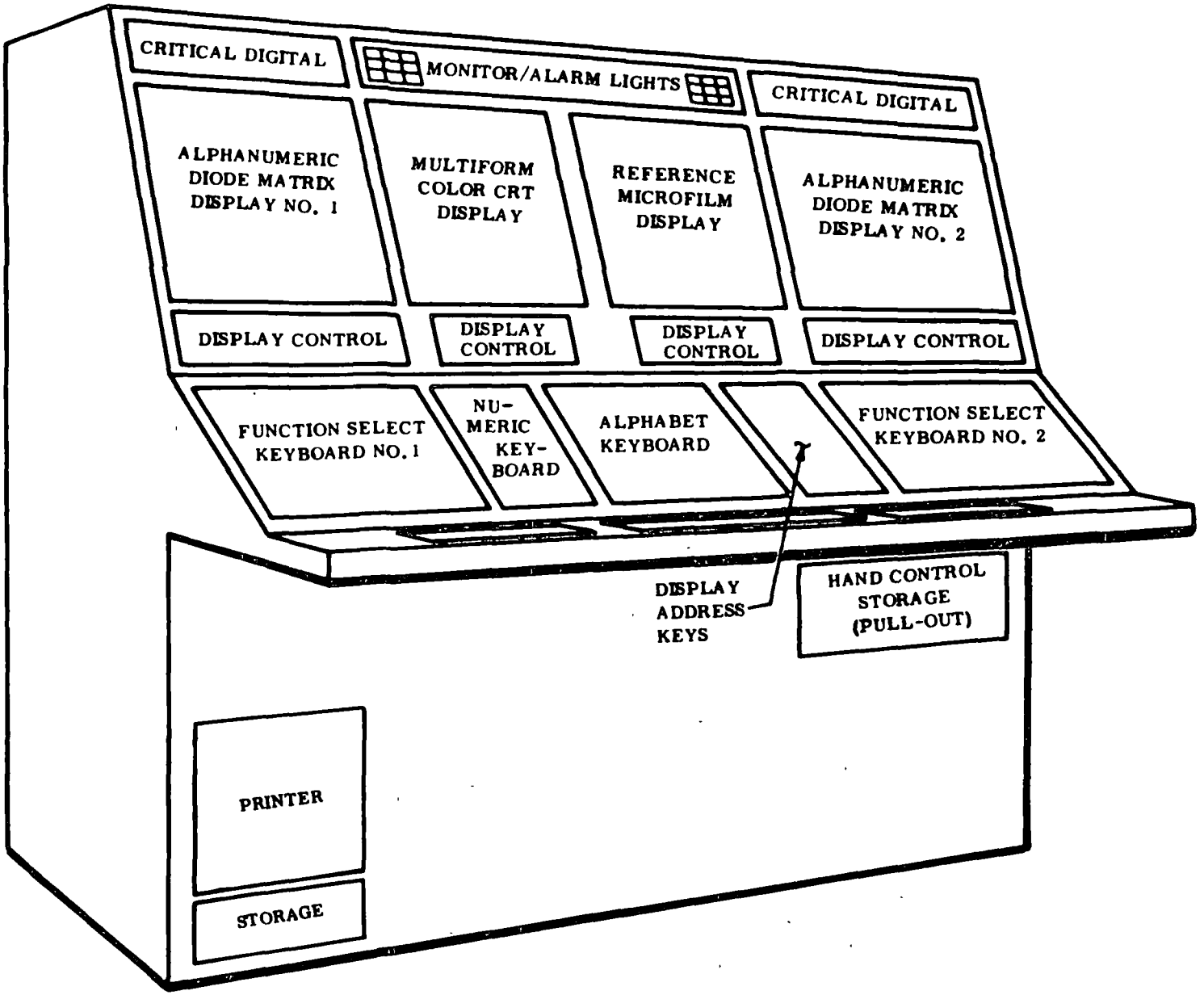


Figure 3.3-7. Command and Control Console

Space Division
North American Rockwell

Communications and Tracking Assembly

The communications and tracking assembly includes both external and internal functions. External communications and tracking provides the capability to transmit intelligence to and track various elements external to the OLS. Internal communications provides for audio entertainment and paging, audio/video distribution network linking crew in all OLS locations, and digital data distribution networks linking all data sources in the OLS.

External Communication and Tracking Subassembly

The design of the external communications and tracking subassembly (Figure 3.3-8) for the OLS assumes utilization of the 85-foot diameter antenna of Manned Space Flight Network (MSFN) for transferring data to earth. In order to provide the capability for continuous communications coverage to any point on the lunar surface, a lunar data relay satellite system with the characteristics of the Tracking and Data Relay Satellite (TDRS) proposed for earth orbit is assumed. In addition, communication and tracking is provided for logistics vehicles and subsatellites.

The communications and tracking assembly consists of the following components:

S-band transponder
Premodulation processor
Duplexers
Seven body-mounted directive phased array antennas
One spherical array antenna boom-mounted above the solar arrays
Eight semidirective antennas
Approach radar and its associated phase-monopulse antenna
Facsimile device
RACU provided at each location to interface with Data Processing Assembly

There are 12 transponders located on the OLS. Two are located at the body arrays, one at the aft arrays, one deployed with the spherical array, and one transponder located at each of the eight semidirective antennas. The transponder includes the power amplifier, exciter, FM and PSK modulator, preamplifier, receiver, range unit, and range code generator. All transponders are identical and have the capability to accommodate the defined operating modes selectable by the computer via the RACU. Selectable crystals are provided to satisfy the RF frequency channels.

The transponder receives a wave train of formatted information from the pre-mod processor which provides the signal at either the FM or PSK modulator. In the modulator, the signal from the psuedo random noise (PRN) range code generator is mixed with the data. The composite signal modulates the carrier which is amplified in the power amplifier and delivered to the antenna.

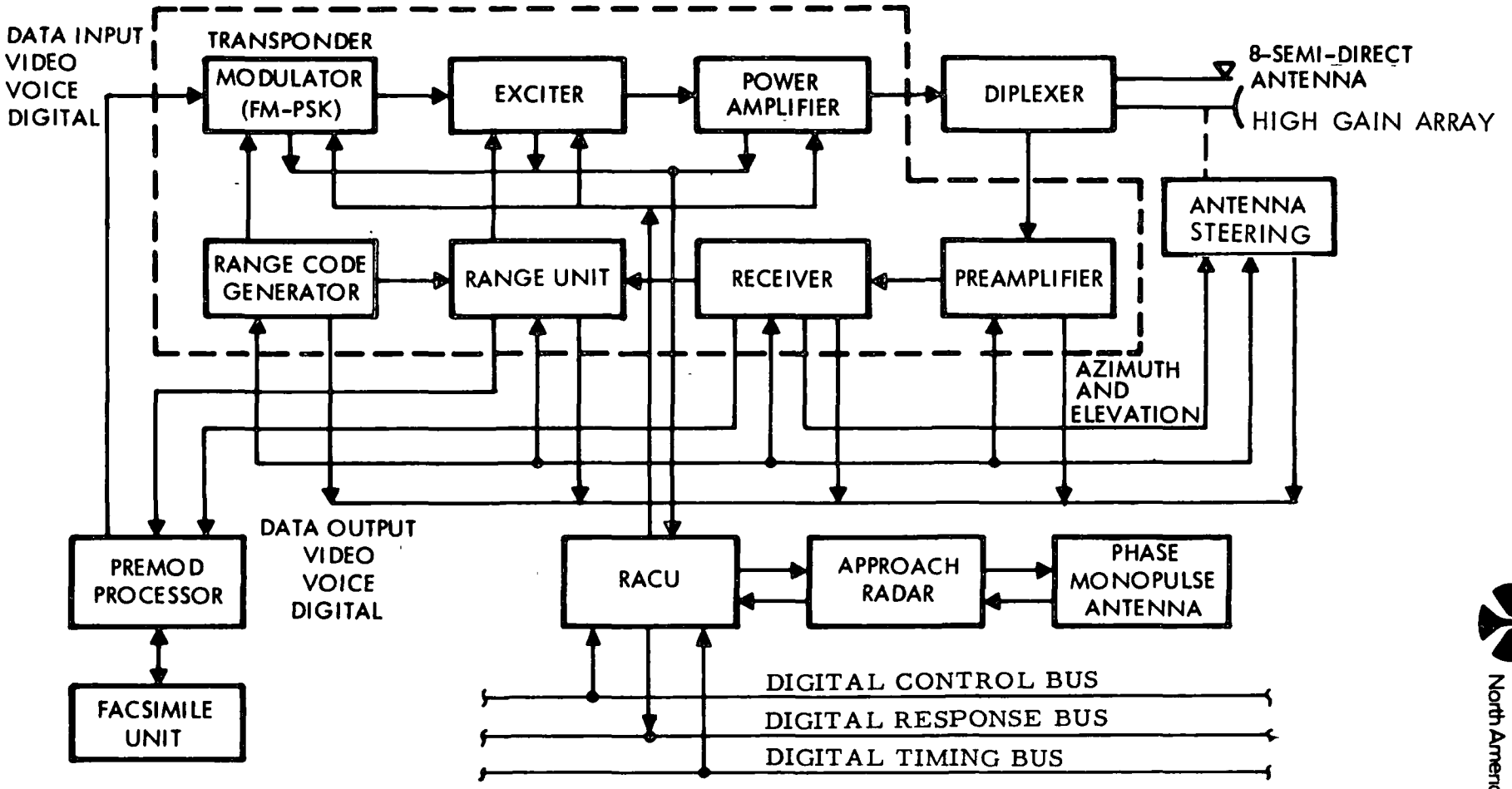


Figure 3.3-8. External Communications and Tracking Subassembly

Space Division
North American Rockwell

A wide-band, low-noise, uncooled, parametric amplifier is planned as the receiver front end. In the receiver the RF signal is demodulated and distributed to either the PRN range unit for range data extraction and/or to the premodulation processor which will demodulate and distribute voice, video, or digital data as required. The premodulation processor located in the primary and experiment control center, formats and transfers the data to the transponder and receives data from the receiver via coax cables. The unit is controlled by the computer via an included RACU. The unit interfaces with the audio and video recorders which provide data to the premodulation processor or receive data from the premodulation processor. The processor contains the subcarrier oscillators which carry voice and data information to be multiplexed on the RF carrier.

A total of eight phased arrays are located on the OLS. One spherical array is located on a 30-foot boom above the solar arrays. Four 5 by 14-foot arrays are located around the body of the OLS, and three 5 by 6-foot arrays are mounted on the aft conical surface of the OLS. These antennas provide long range communications to the earth via MSFN, the lunar surface, lunar data relay satellites, and other space vehicles (including subsatellites) at any relative angle from the station.

The antenna beam will be pointed toward the station being communicated with via auto-track techniques. During the acquisition phase the computer will orient the beam until such time as the target is acquired. After acquisition, beam orientation will be maintained via auto-track capability provided in the receiver, which would provide both azimuth and elevation signals.

Eight semidirective antennas (four located around each end of the station) are provided for short range communications between the program elements including lunar surface stations, and the OLS. Each of these antennas provide 120-degrees beamwidth coverage.

An approach radar is provided for rendezvous with the logistic vehicle and detached modules. This unit takes over from the longer range high-gain antennas at a range of 20 n mi until the target vehicle approaches to a range of 1000 feet. At this range the docking sensor assumes control of the vehicle for the final docking maneuver.

A facsimile unit is provided to transmit high-quality graphic data between the OLS and earth. The unit provides capability to transmit 70 mm film negative, 100 optical lines/millimeter resolution, 32-level gray scale in less than 30 seconds, and record data in 8-1/2 by 11 inch format at 30 lines per inch in less than 30 seconds. These data will be handled by the medium-band data channel.

Internal Communications Subassembly

The internal communication subassembly (Figure 3.3-9) is responsible for distribution of all information within the OLS. The information falls into three categories: (1) entertainment and paging, (2) audio and video, and (3) digital data. The information is transmitted on the entertainment and paging bus, audio/video bus, and digital buses (control, response, and timing). Redundant buses are provided for reliability. Entertainment and paging, and audio and video information is frequency division multiplexed (FDM) onto the audio/video buses. Information on digital buses is time division multiplexed (TDM). Control of switching will be maintained by the ISS computer.

Summary of Information Subsystem

Communication/tracking includes both external (RF) and internal (hard-wire) transfer of data. Directive antenna arrays are used for long-range tracking and data transmission to the MSFN and to subsatellites and other space vehicles. Semidirective antennas at each end of the OLS provide communication and ranging to nearby elements. An approach radar located near the docking ports at each side of the station provides close range radar data. Each antenna has a full complement of RF equipment (power amplifiers, diplexers, S-band transponders with pseudo random noise (PRN) ranging) so that any antenna may be used for any of the communication and tracking tasks.

The data processor assembly is physically divided into two multi-processors; one is located in the primary control center and the other is in the experiment control center. The total task of OLS operations management and experiment data management would reside in their individual processors; however, either processor is capable of maintaining OLS operations.

The display/control assembly includes the consoles in the primary control center and the experiment control center. Two consoles are at the primary center and one at the experiment center. All three consoles are identical in construction and capability. Multiple forms of data displays and operator controls provide adequate flexibility and capacity to handle three or more simultaneous tasks. In addition, experiment unique consoles, designed to utilize the same form and flexibility of display/control, are installed on the experiment deck. The experiment consoles are coupled to the data processor and communication/tracking assemblies in the same way as the command console.

The software incorporates all the supervisory, application, and support program development, plus the reference data (microfilm), hard copies of programs (tape), and the operating supplies (tape and paper) required to operate the OLS computer complex.

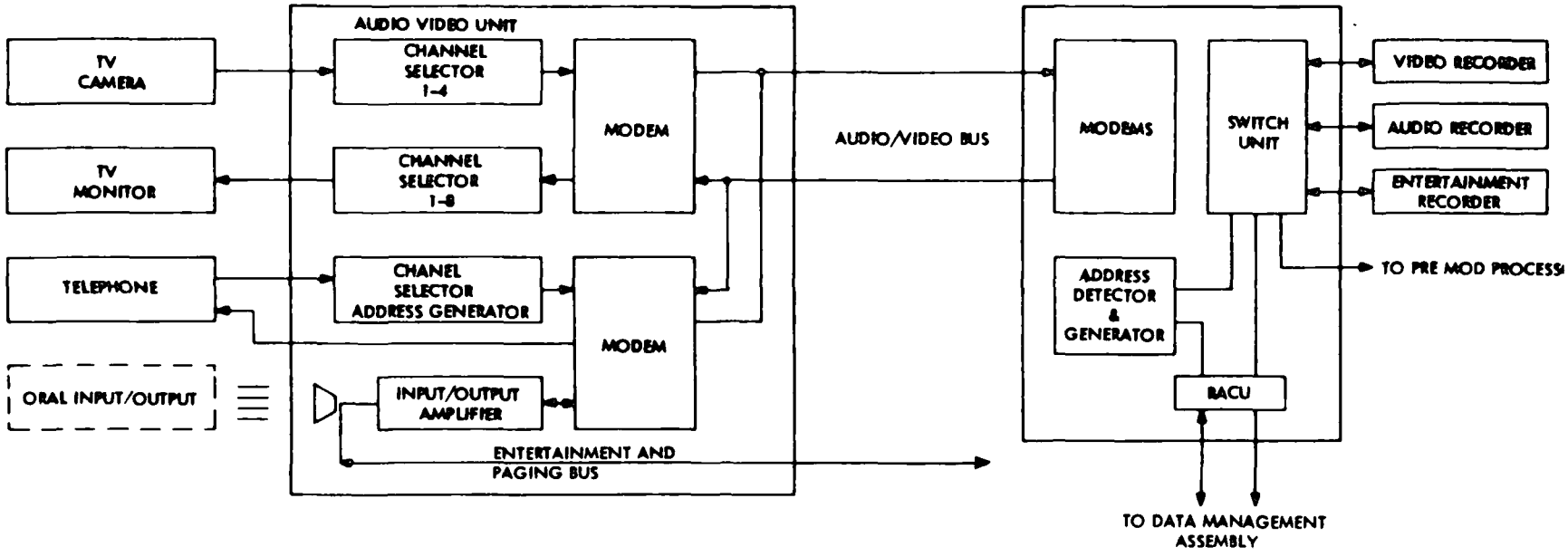


Figure 3.3-9. Internal Communications Subassembly

Space Division
North American Rockwell

The electrical interface between the ISS and the other OLS subsystems will be through the standardized 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, 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 ISS is modular in nature with the in-flight replaceable unit (IFRU) being the lowest level of equipment which is replaceable during routine flight operations. Measurements used to determine operational worthiness, subsystem status, and equipment checkout are utilized to detect and isolate faults to the IFRU level.

The overall compliment of ISS equipment for the OLS is shown in Table 3.3-1. Unit weight, size, and power requirements are indicated as well as the quantities needed.

3.3.2 SELECTION RATIONALE

Investigation of the requirements imposed on the OLS ISS and comparison of these requirements with those used in the EOSS trade studies shows that the trade criteria is identical for a major portion of the ISS subsystem. Volume IV, Configuration and Systems Analysis Report, compares requirements and shows that the major difference between OLS and EOSS requirements exists in external communications.

Further analysis indicates that the external communications links are comparable from a functional standpoint. Major differences are the reduction in gain requirement by 9.8 db for the OLS primary link to earth, and by 18.3 db for a satellite link. The result of the lowered gain requirement, the equivalent of reducing the EOSS 13-foot parabolas to 5 feet in diameter, combined with a need to permit contact with the OLS in any orientation required reevaluation of the high-gain antenna selection.

The following paragraphs reflect the selections and associated rationale from the trade studies in the ISS section of Volume IV of this report.

Data Processing Assembly

Processor

Candidates include special and general purpose processors. The major factors in the selection is the lack of needed flexibility in special purpose processors, and lower power and weight requirements for the general purpose system. Software development impact should be lessened for the OLS due to Earth Orbital Shuttle (EOS) and EOSS development.

Table 3.3-1. ISS Equipment List

Item	Unit Weight (lb)	Dimensions (inches)	Power-Watts Peak (Average) (watts)	Quantity	Weight (lb)	PP/PA ***
Data processor assembly						
Processor	10	9x3.6 x 14	30 (3)	5	50	150/15
I/O controller	10	9x3.6 x 14	25 (15)	5	50	125/75
Operation memory module	50	9 x 9 x 9	50 (20)	8	400	400/160
Mass memory module	100	9 x 18 x 14	30 (15)	13	1300	390/195
Archival memory tape deck	120	18 x 18 x 14	135 (33.3)	2	240	270/67
RACU - 256	6	3 x 3 x 8	8 (6)	24	144	192/144
RACU - 128	5	3 x 3 x 4	4 (3)	8	24	32/24
RACU - 64	1.5	3 x 3 x 2	2 (1.5)	10	15	20/15
Central timing unit	18	9 x 6 x 14	30 (30)	2	36	60/60
Display control assembly						
Console structure and fittings	83	48 x 48 x 24		3	249	
Printer	40	(1)	100 (20)	3	120	300/60
Display add keyboard	40	(2)	7 (1.75)	6	36	42/10
Function select	15	(2)	5 (1.25)	6	90	30/8
Light emitting diode (LED) display	10	(2)	20 (10)	6	60	120/60
LED display electronics	20	(2)	30 (15)	6	120	180/90
CRT	25	(1)	50 (50)	3	75	150/150
CRT electronics	20	(1)	10 (10)	3	60	30/30
Microfilm viewer	35	(1)	50 (25)	3	105	150/75
Hand control stick	20	(2)	1 (.15)	6	120	6/1
Monitor and alarm	30	(1)	89 (45)	3	90	267/135
Remote terminal unit	40	12 x 12 x 18	250 (25)	2	80	500/50
Communications tracking assembly						
Body array set	25	5 x 14 feet		4	100	
Aft array	10	3 x 6 feet		3	30	
Sphere array*	205	5 ft sphere 30 ft boom		1	205	
Semidirect	1	6 (dia.) x 8	0	8	8	0/0
Approach radar**	25	12 x 12 x 12	125 (15)	2	50	250/30
Transponder power amplifier and diplexer	29		120 (36)	12	348	1440/440
Premodulator processor	50	12 x 12 x 14	50 (50)	1	50	50/50
Facsimile unit	40	18 x 12 x 14	200 (20)	1	40	200/20
Central switch unit and self-test	120	18 x 27 x 14	70 (50)	2	240	140/100
Audio/video units - A	6	6 x 6 x 2	9 (5)	20	120	180/100
Audio/video units - B	18	6 x 10 x 2	12 (12)	3	54	36/36
Paging amplifier	10	6 x 6 x 8	100 (10)	2	20	200/20
Peripheral group						
Video recorder	120		135 (67.5)	3	360	405/204
Audio recorder	15		20 (10)	2	30	40/20
Entertainment recorder	15		20 (20)	1	15	20/20
Video cameras (color)	10		50 (10)	9	90	450/90
Video monitors (9-inch color)	15		50 (10)	12	180	600/120
Video monitors (14-inch color)	40		75 (15)	2	80	150/30
			Total		5449	7525/2679

(+) Includes boom

** Includes RF electronics

*** PP/PA = peak power/average power (watts)

Computer System

Candidates include the single centralized computer, dedicated computers, and multiprocessing systems. The multiprocessing systems offer flexibility, redundancy advantages, and growth potential greater than the alternates. Additionally, it requires the least power and weight with small development risk.

Processor arrangements considered distributed and central locations. The centrally located processors are generally more efficient, and avoid complexities in intercommunications between processors which counter to some extent the advantages of direct interfaces with serviced equipment. The use of two groups of processors, one in the control center and one in the experiment center alleviates the concentrated physical location.

Processor assignment candidates include priority and predetermined assignments. Test and software requirements prohibit the use of priority assignments.

Data Storage

Operating Memory. Candidates include core, integrated circuit, and plated wire systems. Plated wire was selected due to good overall characteristics, and lower (order of magnitude) power requirement for high speed operation. Some development risk was assumed in the EOSS selection of plated wire that should not exist in the OLS time frame.

Mass Memory. Candidates include tape, disc, and plated wire. The selection was made on the basis of low power requirements for the plated wire, and access time problems with tape and disc units.

Archival Memory. Candidates include film and tape. Tape must be selected at this time since the film types of storage are not adequately developed for predictability. However, the orders of magnitude in increased storage density make film approaches attractive if future development warrants reconsideration.

Communications and Tracking Assembly

External Communications

Link candidates include individual and composite links. The primary reason for selection of composite links include increases in the variety and quantity of RF equipment installed at antenna locations, and the additional frequency allocations required to allow the simultaneous operations required if individual links are used.

Equipment candidates include laser and RF systems. The specialized use of laser links is undergoing consideration by a number of agencies. However, no potential is seen for the development of a MSFN-type capability with lasers by the OLS time period.

High-gain antenna candidates include parabolic high-gain antennas and phased arrays. Since this selection is at variance with the EOSS study, some further elaboration is required. The conformal array concept has been attractive for a number of years due to the lack of mechanical motion, multiple beam capability, and fewer deployment problems. The development required for arrays is associated with the complex feed, maintenance of polarization off the array axis, and the large structure surface areas required by the arrays.

The OLS arrays (Figure 3.3-10) would each require 30 to 70 square feet depending on the coverage angles. The EOSS arrays would require 200 to 470 square feet for similar coverage. The reduction in required area and in the number of elements needed in each array makes phased-array antennas more feasible for the OLS. A summary of the basic selection rationale is given in Figure 3.3-10.

Tracking candidates include PRN ranging and phase monopulse. PRN ranging is slightly more complex than some alternate choices. However, it is most likely to be compatible with the overall space program elements and presents no undue problems. The phase monopulse is expected to provide the greatest accuracy with good reliability. Actual development of array designs could affect the latter choice.

Internal Communications

Candidates for the internal communications system include video, voice, and digital. These are essentially direct choices compatible with the types of signals being distributed.

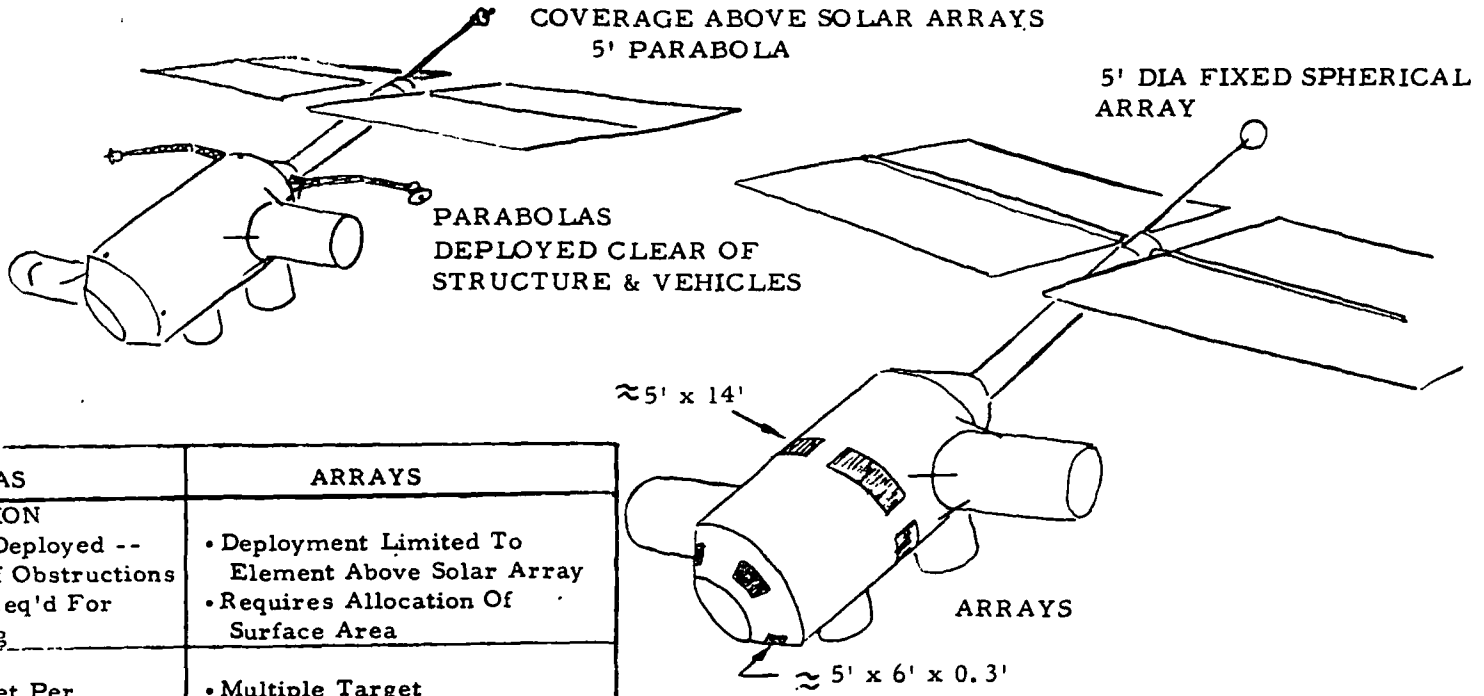
Displays and Controls Assembly

Console arrangement candidates are functional, subsystem, and hybrid. The hybrid arrangement requires less size, weight, and power than the subsystem concept. A functional arrangement limits the experiment capability.

Controls and displays is a trade area that is primarily to evaluate and define applicability of the various types of controls and display features to the functional requirements. Essentially following a preliminary design approach based on these data, the console approach represented in 3.3.1 evolved during the EOSS studies.

Software

The EOSS trades resulted in modular organized, generalized design, compiler language software. These recommendations and rationale are included in Volume IV. In the time period of the OLS, it is anticipated that considerable software will have been developed for EOS and EOSS types of spacecraft. Existing designs will heavily influence the actual software and development on the OLS.



PARABOLAS	ARRAYS
INSTALLATION • Must Be Deployed -- Clear Of Obstructions • Rigidity Req'd For Tracking	• Deployment Limited To Element Above Solar Array • Requires Allocation Of Surface Area
OPERATION • One Target Per Antenna • Mechanical Movem't Affects SCS • Gimbals Require Periodic Maintenance • Software Minimal For Tracking	• Multiple Target Capability • No Mechanical Movement • No Periodic-Graceful Degradation • Continuous Software Support For Track
WEIGHT • Light Antennas-Heavy Deployment Structure (≈680 lb)	• Weight Less Than Parabola-Boom Combination (≈335 lb)
COST • Moderate	• Moderate to High
DEVELOPMENT RISK • Low	• Moderate

SELECTION
ARRAYS • No Mechanical Operation/Maint. • Avoids Deployment -- Boom Problems • Avoids Operational Restrictions With Multiple Links Overall Weight Comparable Initial Development/Cost Balanced By Increased Perform.

Figure 3.3-10. High-Gain Antenna Comparison

Space Division
North American Rockwell

3.3.3 ADAPTABILITY TO ARTIFICIAL GRAVITY OPERATION

Spin induced artificial gravity will not effect the internal ISS except for modifications to software to reflect procedural changes. The external effect will be severe as regards antenna design.

The effect on antennas may be evaluated by considering the entire volume of rotation as a blocked area. In general, links interrupting each cycle of spin will be unsatisfactory for communications or tracking. Also, continuous despin of the tracking antenna in unblocked areas should be considered an additional requirement to that for normal tracking.

Both omni or semi-directional, and directional antenna location need to be considered for the required rates of antenna switching. The switching will interrupt data and tracking channels and could be a severe development problem if required at or near spin rates.

The most likely alternate to antenna switching would be to install antennas on the despun power boom. For example, the antenna deployed above the solar array would not suffer degradation for targets above the panels. Additional deployed antennas could be utilized to clear the spinning assembly, with semidirectional installations on the spun sections as backup to avoid loss of communications in the event of an RF rotary joint failure.

3.4 GUIDANCE AND CONTROL SUBSYSTEM

3.4.1 CONCEPT DEFINITION

The guidance and control (G&C) subsystem consists of five major assemblies (Figure 3.4-1). The inertial reference assembly contains the initial measurement unit (IMU) and associated preprocessor (the inertial reference trade is documented in Volume IV, Configuration and Systems Analysis, of this report). The optical reference assembly contains the horizon trackers, star trackers, landmark tracker, telescope/sextant, radar altimeter and docking sensor with their associated electronics together with the optical reference preprocessor. The momentum exchange assembly consists of the control moment gyroscopes (CMG's), associated electronics, and CMG preprocessor. The reaction control subsystem (RCS) electronics assembly contains the jet driver electronics and the RCS preprocessor. The fifth assembly contains no hardware but consists of the software associated with the G&C functions. It is resident in the information subsystem (ISS) and is identified as a separate assembly because of its size and complexity and because it could be mechanized in a separate G&C computer (to accommodate space or computational requirements). The interrelationship of these assemblies in performing G&C tasks is shown on the functional block diagram of Figure 3.4-2. Where required, redundancy provides reliability for the critical attitude control functions.

Operational Modes

The G&C subsystem performs sensing, computation, and torque command functions in the areas of navigation, attitude control, and translation control. Figure 3.4-3 is a diagram of the navigation mode. The star trackers and horizon trackers are used to perform the basic star-horizon navigation measurement to determine vehicle attitude and position in orbit. The landmark tracker is used to improve navigation accuracy by automatically acquiring, identifying, and tracking known lunar landmarks.

The landmark tracker is an optical device somewhat similar to a star tracker. It uses pattern correlation techniques to identify and track predetermined landmarks on the illuminated side of the moon. The radar altimeter is used to supplement these measurements, especially on the dark side of the moon where optical landmarks are not available (Figure 3.4-4 is a diagram of the attitude control mode). The same sensors are employed as in the navigation concept. Vehicle attitude is controlled by the RCS jets in conjunction with the CMG's (Figure 3.4-5 is a diagram of the translational control concept). Docking translation maneuvers are normally accomplished by the docking vehicle; however, the OLS has a translational capability to accomplish docking if required. The OLS crew will monitor all dockings using the laser docking sensor and closed circuit TV with a manual override in case of failure.

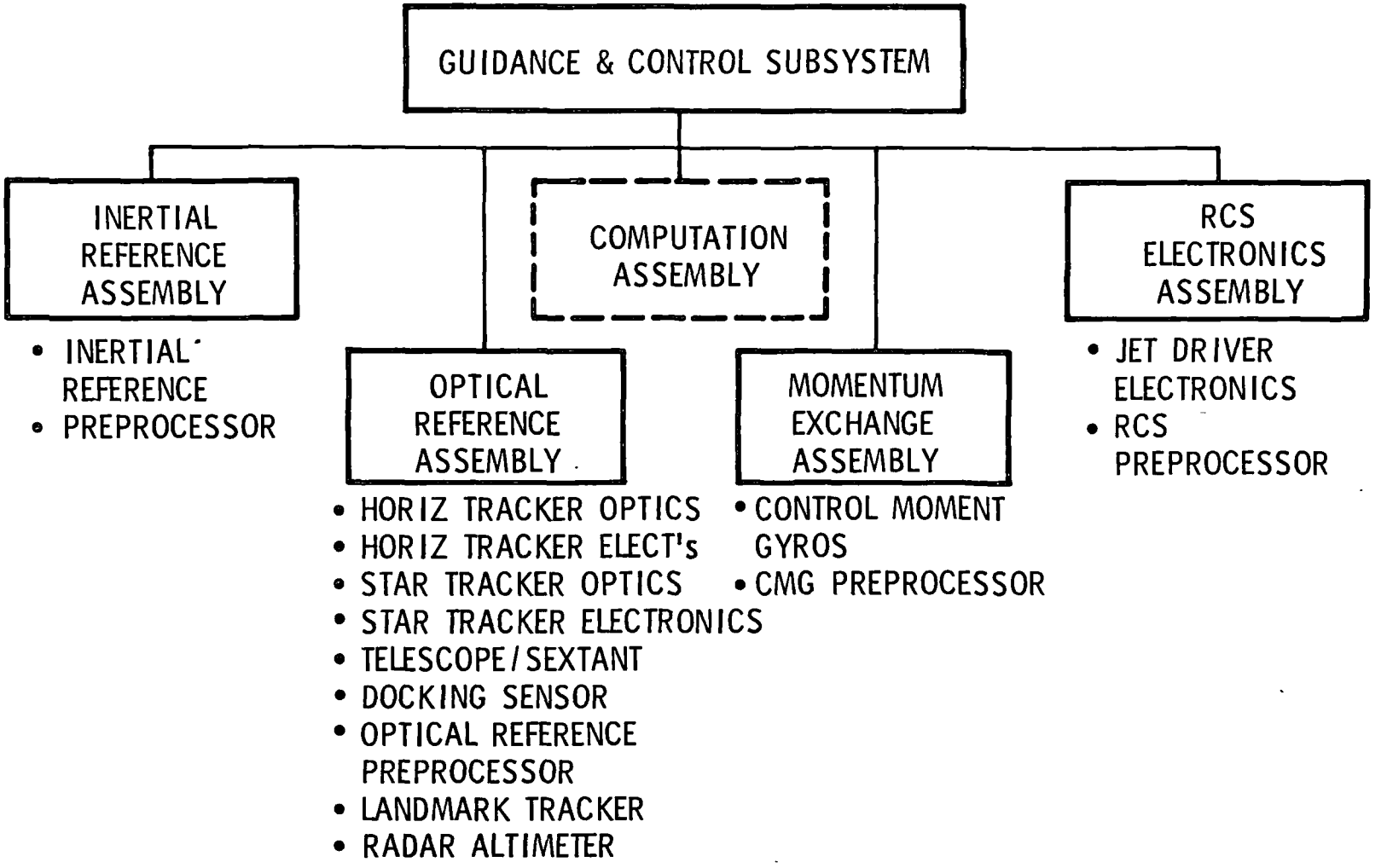


Figure 3.4-1. Guidance and Control Subassemblies

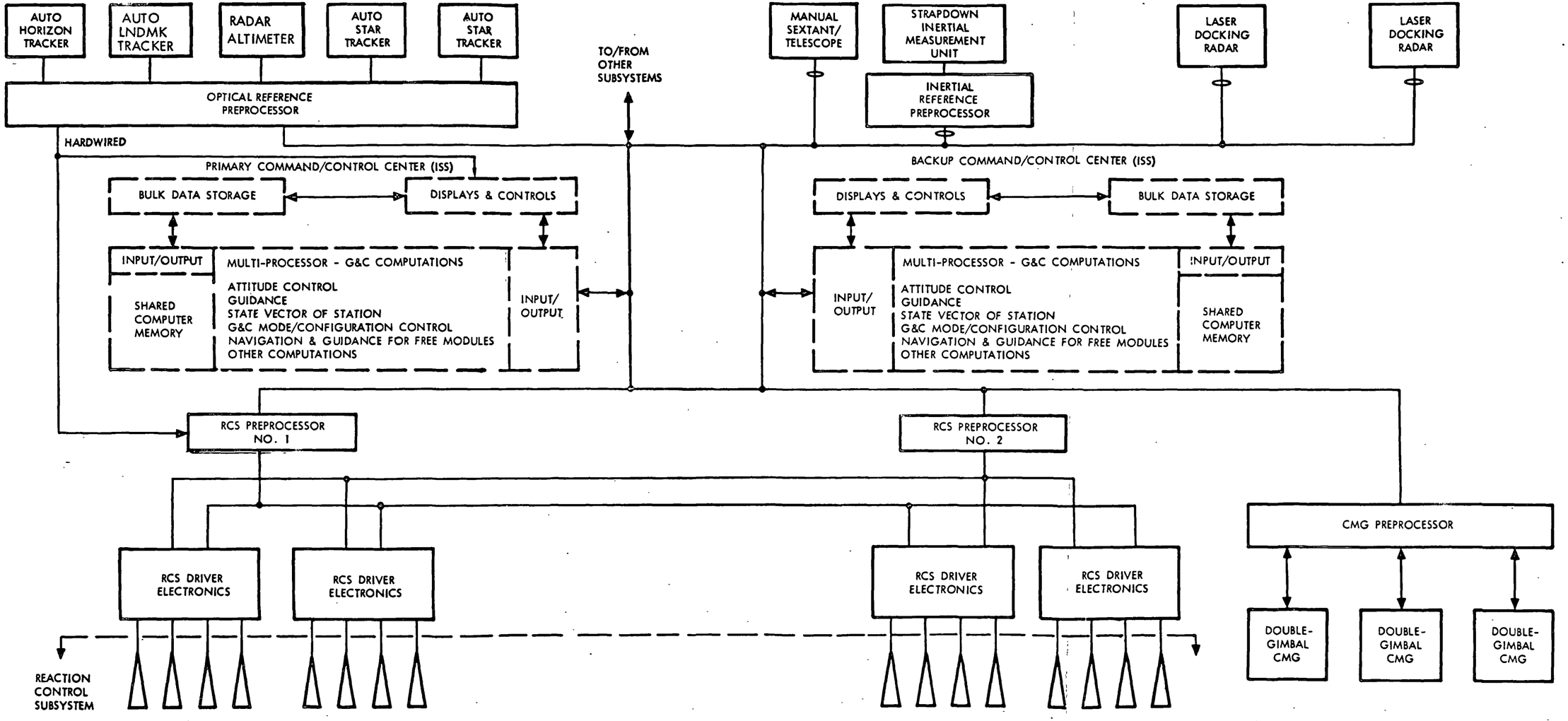


Figure 3.4-2. G&C Functional Block Diagram

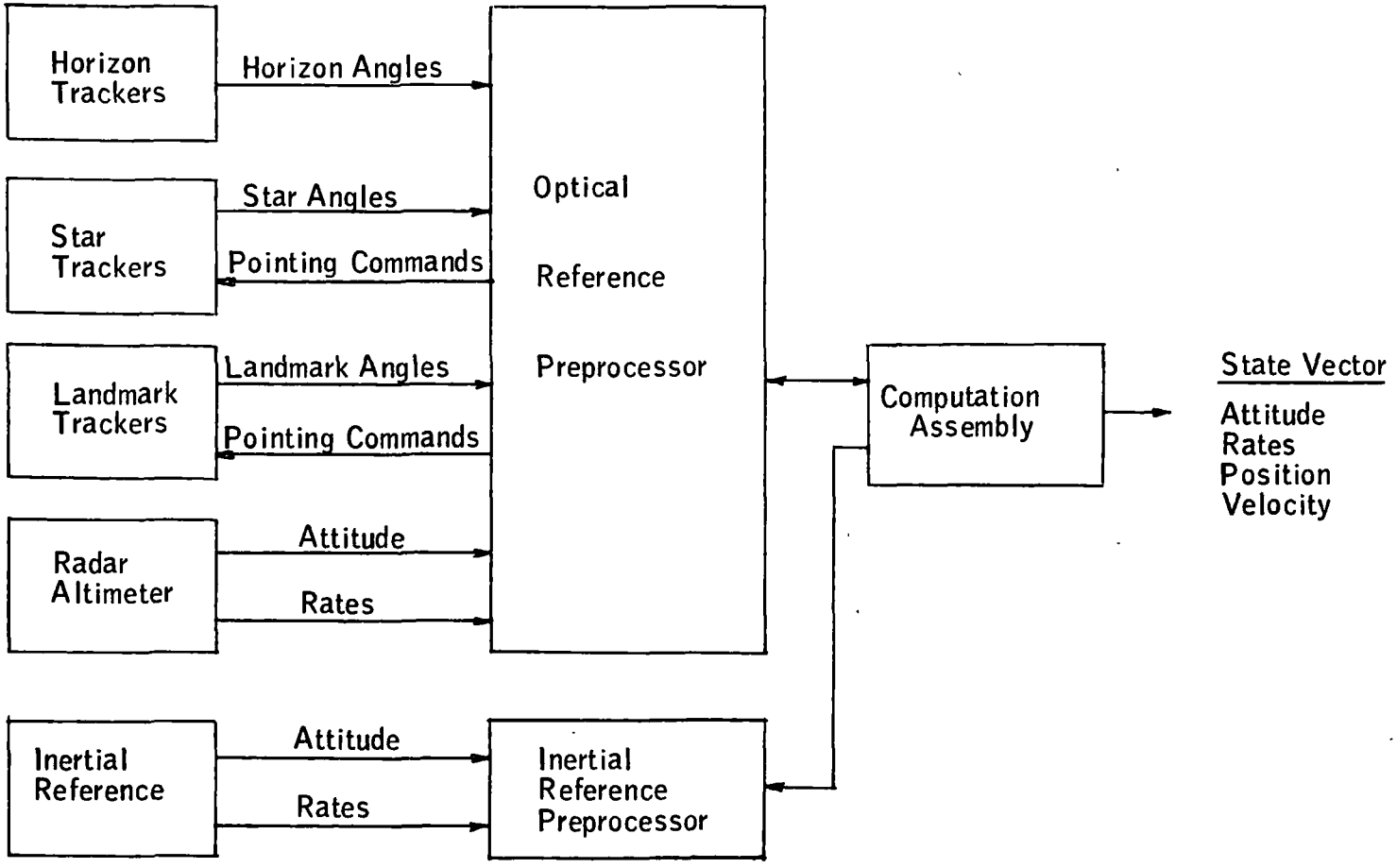


Figure 3.4-3. Navigation Mode

Space Division
North American Rockwell

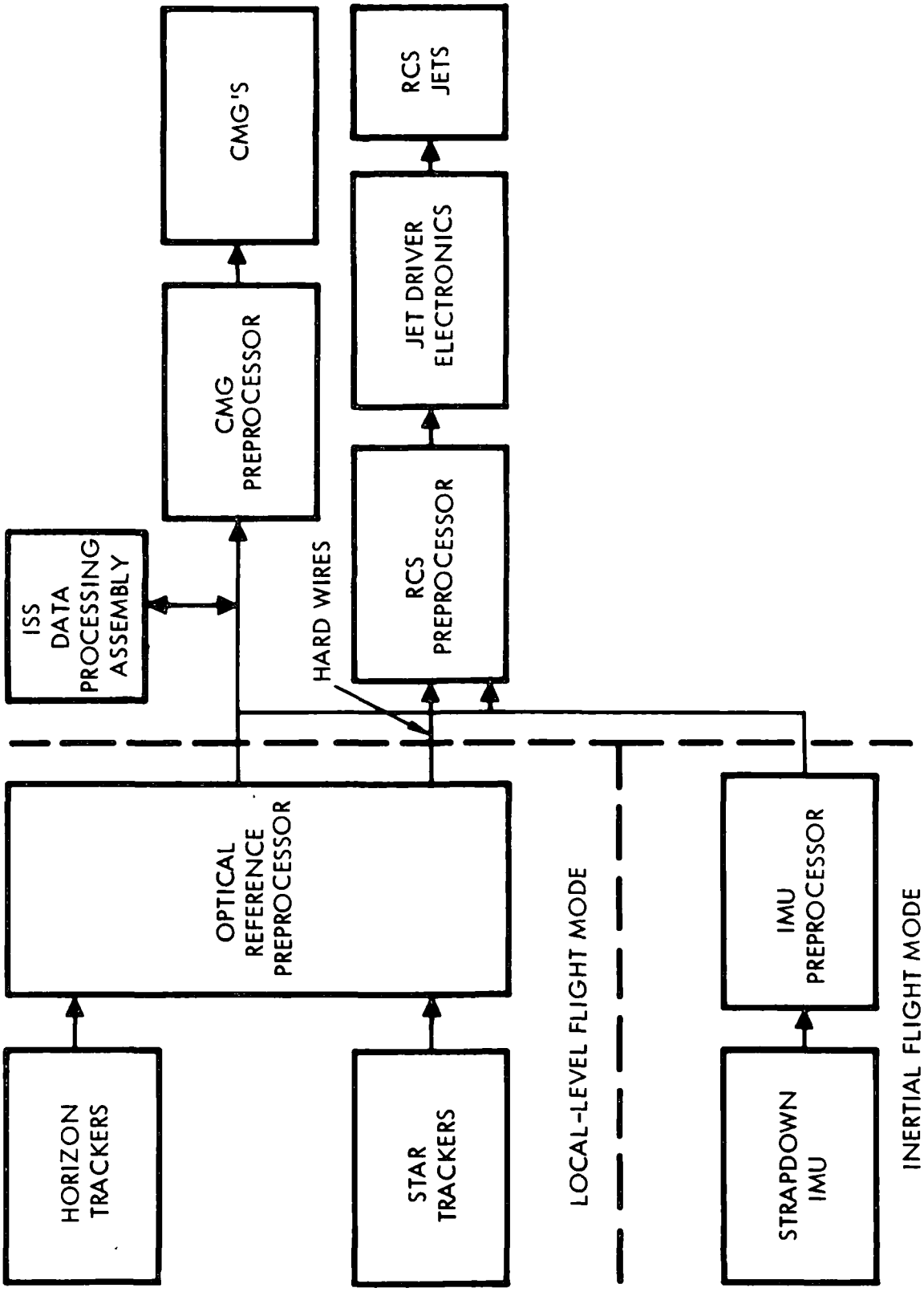


Figure 3.4-4. OLS Attitude Control Concept

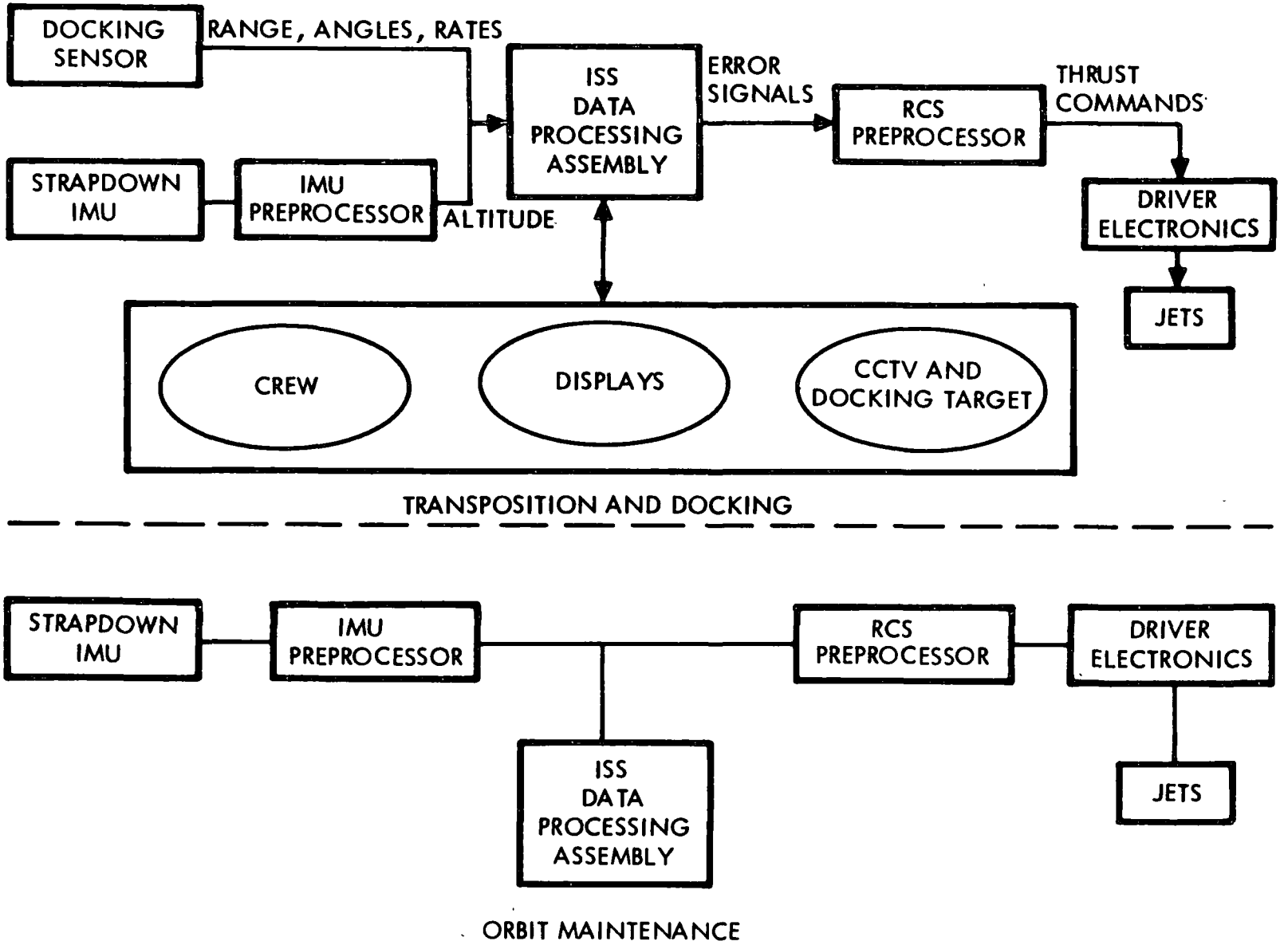


Figure 3.4-5. OLS Translation Control Concept

Space Division
North American Rockwell

Inertial Reference Assembly

Figure 3.4-6 is a block diagram of the inertial reference assembly. When all gyros are active, the nonorthogonal orientation of the six gyros provides quadruple-redundant measurements of angular rates about any of the three orthogonal vehicle axes. Increased accuracy and the detection of degradation-type failures (such as increased gyro drift) are obtainable by processing the gyro outputs. The Inertial Measurement Unit (IMU) has the following physical characteristics:

1. Weight - 45 pounds

2. Size - 14-inch diameter x 12 inches; volume - 1.07 cubic feet

3. Power

   a. All gyros on - 160 watts; this will occur from launch through initial manning, and during dockings

   b. 4 gyros on - 140 watts; this will be the normal strapdown IMU (SDIMU) active configuration and is estimated at 24 hours per day.

4. Heat Dissipation - cold plates. Temperature range of cold plates is +80 to 130 F (Individual gyro temperature-control servos can only maintain gyro fluid at the +160 F required for specification performance with this cold plate range).

IMU Preprocessor

The preprocessor accomplishes skew-to-body transformation, body rate and inertial attitude computations, and local vertical mode transformations (Figure 3.4-6). It controls operations of the SDIMU as instructed by the central computer, monitors the gross failure detection outputs of the SDIMU, performs statistical analysis on sensor outputs to maximize accuracy and detect degradation failures of individual sensors, and performs its own internal monitoring.

This equipment is one of a family of small peripheral computers or preprocessors used in the G&C subsystem, and the following general discussion applies to the optical and RCS preprocessors also. The CMG preprocessor appears (at this time) to be better suited to a hybrid rather than an all-digital design. Detailed analysis, related to the CMG control laws eventually selected, and to the commonality tradeoff discussed below, is required to finally decide the digital versus hybrid question for this preprocessor. Future in-depth study may show identical machines to be the most cost-effective for the G&C even though there would be various degrees of excess capability in three of the four usages. Advantages would be: (1) common (interchangeable preprocessor) IFRU and (2) reduced development, design, and procurement costs of increased numbers of a single item may offset excess capacity costs.

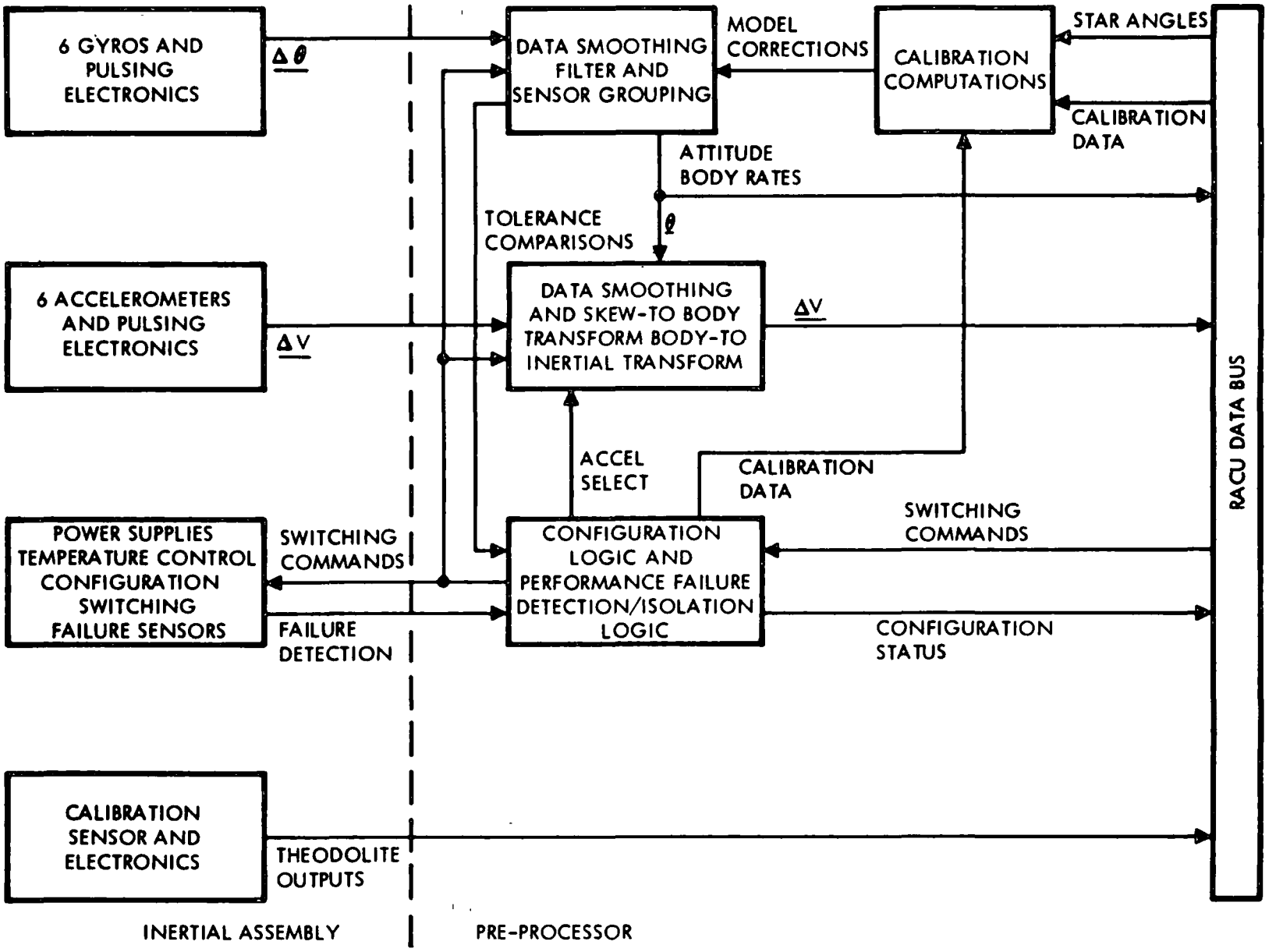


Figure 3.4-6. Inertial Reference Assembly

Space Division
North American Rockwell

The present baseline approach uses differing combinations of identical modules to provide the requirements of each preprocessor. For example, one preprocessor may require two memory modules versus one for another. A single-size maximum modular capacity chassis has been assumed. The IMU preprocessor's physical characteristics follow.

1. Weight - 28 pounds

2. Size

   a. Dimensions - 8 x 5 x 17.5 inches
   b. Volume - 0.41 cubic feet

3. Power

   a. Average - 38 watts peak

   b. Duty factor - 24 hours/day

4. Heat Rejection - by coldplate; temperature range +30 to 130 F

Optical Reference Assembly

The optical reference assembly is composed of star trackers, horizon scanners, telescope and sextant, and laser docking sensor along with a landmark tracker, altimeter, and preprocessor as shown in Figure 3.4-7. The equipment is used variously for attitude reference, attitude reference alignment, optical navigation, unknown target tracking, and docking.

The separate equipments which make up the optical reference assembly are located at various places in the vehicle depending on the function each performs. Knowledge of the relative orientations of several of these devices is therefore required to perform certain functions such as navigation and experiment pointing. An optical link consisting of a three-axis autocollimator is provided for the purpose of determining deviations to the nominal relative orientations. The orientation of the star tracker with respect to the horizon tracker is calibrated in this fashion as is the orientation of the sextant with respect to the experiment module. In addition, three-axis autocollimators are used to calibrate the alignment of the landmark tracker to the star trackers and horizon scanners. The two star trackers are calibrated against each other and the sextant by sighting on the same star. Thus, all of the optical reference equipments can be aligned with each other and with the experiment module, except for the docking sensor. The alignment computations are performed in the optical reference preprocessor. The optical alignment link will be active whenever necessary to insure the required alignment accuracy. In the case of the star tracker/horizon scanner link, this may be continuous during navigation depending on the structural flexing characteristics of the vehicle. Other alignments are performed periodically as required (e.g., after change of experiment module orientation).

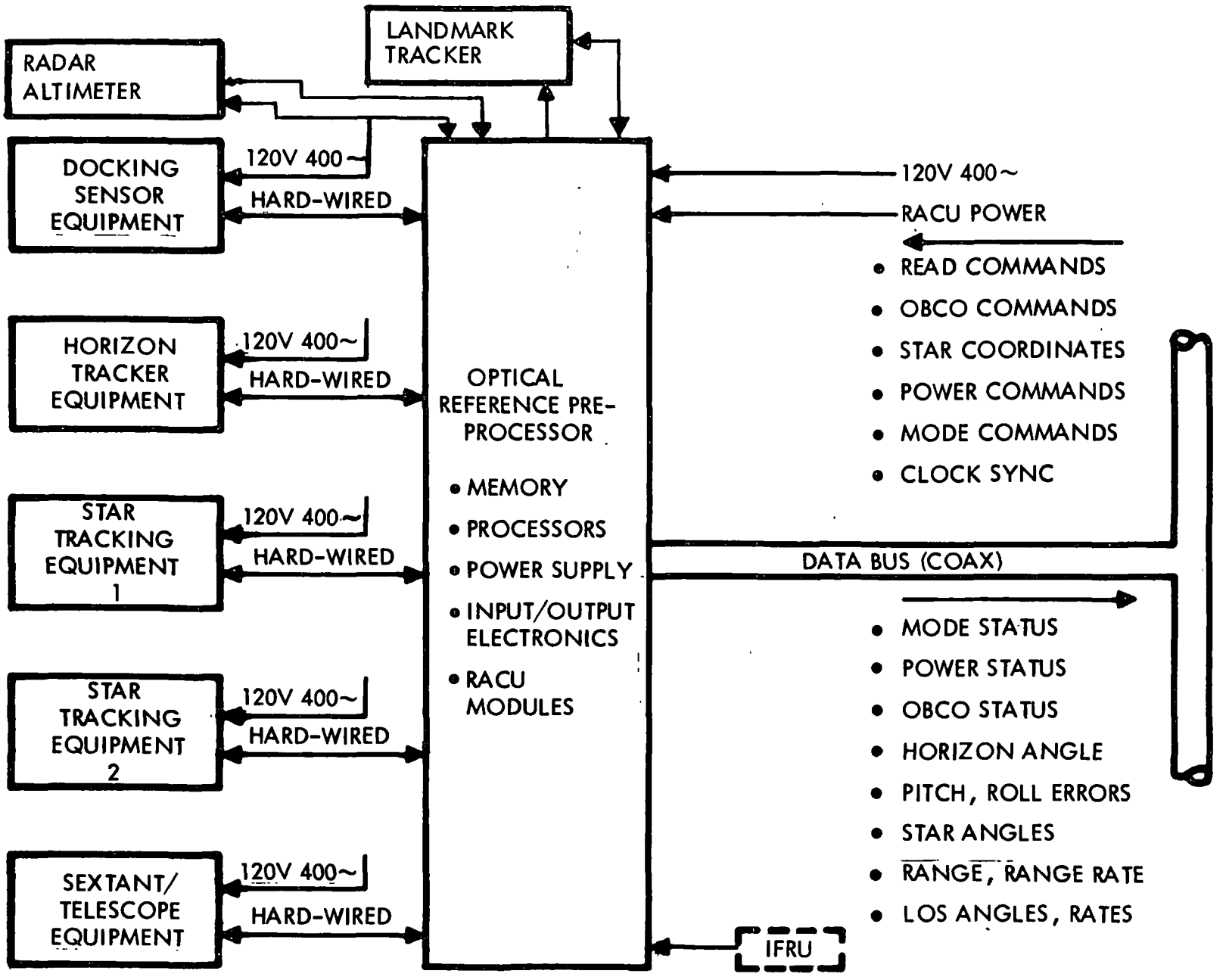


Figure 3.4-7. Optical Navigation/Attitude Reference Assembly

Space Division
North American Rockwell

Horizon Tracker

The physical characteristics of the horizon tracker are as follows:

1. Weight
   a. Each sensor - 7.75 pounds
   b. Four sensors - 31.0 pounds
   c. Electronics unit - 11.0 pounds
   d. Total subassembly - 42 pounds
2. Size
   a. Each sensor - 3.75 x 5.25 x 6.0 inches; volume, 0.07 cubic feet
   b. Four sensors - volume, 0.28 cubic feet
   c. Electronics - 6.0 x 9.0 x 8.5 inches; volume, 0.27 cubic feet
3. Power
   a. Each sensor - 3 watts peak and 3 watts average since tracker is on continuously
   b. Four sensors - 12 watts peak and 12 watts average
   c. Electronics - 22 watts peak and 22 watts average
   d. Total - 34 watts peak, 34 watts average
4. Heat Dissipation
   a. Sensors - conduction to structure and radiation to/from space and earth; only 3 watts required to maintain temperature 33 F thermal-vacuum environment
   b. Electronics - Designed for an environmental range of -65 to +165 F; cooling may be by coldplate or structure/air to transfer the 22-watt average dissipation

Landmark Tracker

The landmark tracker is mounted in the vicinity of the horizon trackers for maximum landmark coverage. Due to size limitations imposed by maintenance considerations, it is not possible to mount this device on a common base with horizon trackers. Calibration equipment is provided to keep the alignment among star trackers, horizon scanners, and landmark tracker as well determined as possible.

The landmark tracker physical characteristics are as follows:

1. Weight (including electronics) - 33 pounds
2. Size
   a. Tracker - 0.82 cubic feet
   b. Electronics - 0.28 cubic feet
   c. Total per unit - 1.1 cubic feet
3. Power - 30 watts peak/average
4. Heat Dissipation - neither tracker nor electronics unit requires coldplating.

Calibration Equipment

Three-axis autocollimators are used to continuously measure the alignment between the star trackers, horizon scanners, landmark trackers and the experiments. Three-axis equipment is chosen because of its versatility; however, subsequent analysis may show that two-axis equipment is adequate.

The physical characteristics are:

1. Weight (per unit) - 30 pounds
2. Power (per unit) - 8.3 watts average
3. Size (per unit) - 0.25 cubic feet
4. Heat Dissipation - coldplate is required. In general, the autocollimators are mounted on the same base as other equipment and can use the same coldplate.

Radar Altimeter

The radar altimeter may be the same piece of equipment as used by some of the experiments obviating the necessity for the experiments to provide their own altitude measurements. However, the G&C equipment is not required to be as accurate and has different operating requirements; thus a separate piece of equipment is included.

The physical characteristics are:

1. Weight - 35 pounds
2. Size - 2.0 cubic feet
3. Power - 54 watts average
4. Heat Dissipation - coldplating is required

Star Tracker

Two identical star-tracking subassemblies are included in the optical reference assembly. They are mounted adjacently and have overlapping fields of view. Use of two trackers provides a celestial three-axis attitude reference mode as an alternate capability to the inertial attitude reference mode (provided by the strap-down gyro/processor subassembly) and provides a means of checking for changes in bias (drift shifts) of the gyros. The two trackers are also used to converge the state vector estimation rapidly to smaller errors than can be easily achieved by a single tracker. Subsequently, one may be turned off and used as a standby redundant unit since the computational process can be designed to prevent use of erroneous data for sufficient time to automatically activate it again.

1. Weight

   a. Per tracker unit - 22 pounds

   b. Per electronics unit - 11 pounds

   c. Total per equipment - 33 pounds

   d. Total for two equipments - 66 pounds

2. Size

   a. Tracker unit - 10.0 x 15.0 x 9.5 inches

   b. Volume per unit - 0.82 cubic feet

   c. Electronics unit - 7.0 x 8.0 x 8.5 inches

   d. Volume per unit - 0.28 cubic feet

   e. Total volume per assembly - 1.1 cubic feet

   f. Total volume - two assemblies - 2.2 cubic feet

3. Power

   a. Per tracker unit - 14 watts peak/average

   b. Per electronics unit - 16 watts peak/average

   c. Total per assembly - 30 watts peak/average
   (average power is essentially the same as peak when equipment is operating since torquing duty cycles are small)

   d. Total for initial optical navigation or for optical attitude hold (two equipments operating) - 60 watts

   e. Total for normal optical navigation - 30 watts

4. Heat Dissipation

   Trackers, because of requirements for external viewing, establish a radiation/conduction thermal equilibrium with space and structure and are designed for operation over the hot/cold space extremes.

The electronics unit may be coldplated to reduce the load on the air heat-exchange equipment of the environmental control system, but has been designed to operate with convection/conduction dissipation at space extremes.

Sextant/Telescope

The sextant/scanning telescope unit is an Apollo optical unit assembly, with design modifications to allow separate in-flight retraction and replacement of either the sextant or the scanning telescope using the portable airlock design which is standardized to be used with all hull-penetrating G&C optical sensors.

The unit may be used to make the following optical (planet and celestial) measurements:

1. Bearing (known landmark)
2. Bearing (unknown landmark)
3. Star occultation
4. Star-to-horizon
5. Star-to-known landmark
6. Star-to-star

From these measurements, the star catalog and ephemerides in the onboard data storage, and control timing (clock), the position and orientation of the station can be determined.

The sextant/telescope is used to:

1. Initialize the automatic (star tracker/horizon tracker) optical navigation mode
2. Periodically check the performance of the automatic optical mode
3. Initialize/align the inertial reference mode
4. Calibrate the drift of the inertial reference and periodically check for changes in the drift
5. Obtain bearings of earth landmarks. The sextant/telescope can be shared between G&C and experiments

The physical characteristics of the sextent/telescope are:

1. Weight - 88 pounds

2. Size - 8 x 18 x 31 inches; volume, 2.6 cubic feet

3. Power consumption - 125 watts; expected usage - 3.8 hours/day of which 3.0 is an allowance for experiments usage, resulting in average (daily) wattage of 20.0

4. Heat dissipation - by conduction to the mounting base, thence to a coldplate

Laser Docking Equipment

Measurements obtained by this equipment, either alone or in conjunction with data transmitted between the vehicles by radio link, enable automatically controlled docking of these vehicles. A suitable display of selected parameters facilitates manually controlled docking as an alternative. Physical characteristics of this equipment are:

1. Weight

   a. Per transceiver unit - 20 pounds

   b. Per electronics unit - 8 pounds

   c. Per equipment - 28 pounds

   d. Total initial launch - 56 pounds

2. Size

   a. Per transceiver unit - 8-inch diameter x 17 inches; volume, 0.63 cubic feet

   b. Per electronics unit - 8 x 8 x 12 inches; volume, 0.67 cubic feet

   c. Volume per equipment - 1.3 cubic feet

3. Power Consumption

   a. Transceiver - 20 watts peak/average

   b. Electronics - 10 watts peak/average

   c. Equipment - 30 watts peak/average

   Since only one laser will be used per docking port, and there is no reason to activate the other laser, peak station load will also equal 30 watts. For the initial manning, both lasers would be off

and activation of one would only be by computer command in event of a shuttle laser failure, pre-manning peak load is also 30 watts. Long-term average is negligible because of infrequent docking operations.

4. Heat Dissipation - to air/structure, since low average wattage plus docking port location of the transceiver makes coldplate cooling unattractive.

Control Moment Gyro Assembly

A block diagram of the CMG assembly is shown in Figure 3.4-8. The purpose of this assembly is to provide control torque to the vehicle and provide a means of momentum exchange. The assembly consists of a preprocessor and three two-degrees-of-freedom CMG's. The assembly is commanded from the computation assembly in the form of attitude error signals. These signals are relayed via the data bus and are processed in the preprocessor. The attitude error signals are converted from body axis signals to CMG torque commands via a control law algorithm. Other modes of the preprocessor are momentum dumping and wobble damping. The CMG's physical characteristics are based on a planar array of three double-gimbal gyros of 2700 ft-lb-sec each. The weight and volume figures include allowances for electronics, mounting, spares, acoustical padding, and safety covers. The CMG's physical characteristics are:

1. Weight - 1520 pounds total

2. Size - 170 cubic feet (rectangular dimensions of 3-3/4 x 3-3/4 x 4 feet per gyro)

3. Power - total peak operating power is 500 watts

   a. During spinup, the peak is 800 watts
   b. Average power is 150 watts

4. Heat Rejection - The gyros will be enclosed in a dead air enclosure with a coldplate in the enclosure. It should be noted that the CMG's would operate satisfactorily in a vacuum. The coldplate is provided to reduce loads on the thermal control equipment for the cabin atmosphere.

RCS Electronics Assembly

A block diagram of the RCS electronics assembly is shown in Figure 3.4-9. The assembly includes two preprocessors and four quad driver units. The equipment is divided between the two torus sections of the vehicle. Each torus contains a preprocessor and two driver units (one for each quad). The preprocessor is commanded from the computation assembly via the data bus in a manner similar to that used for the CMG assembly. All four quads can be commanded from each of the preprocessors.

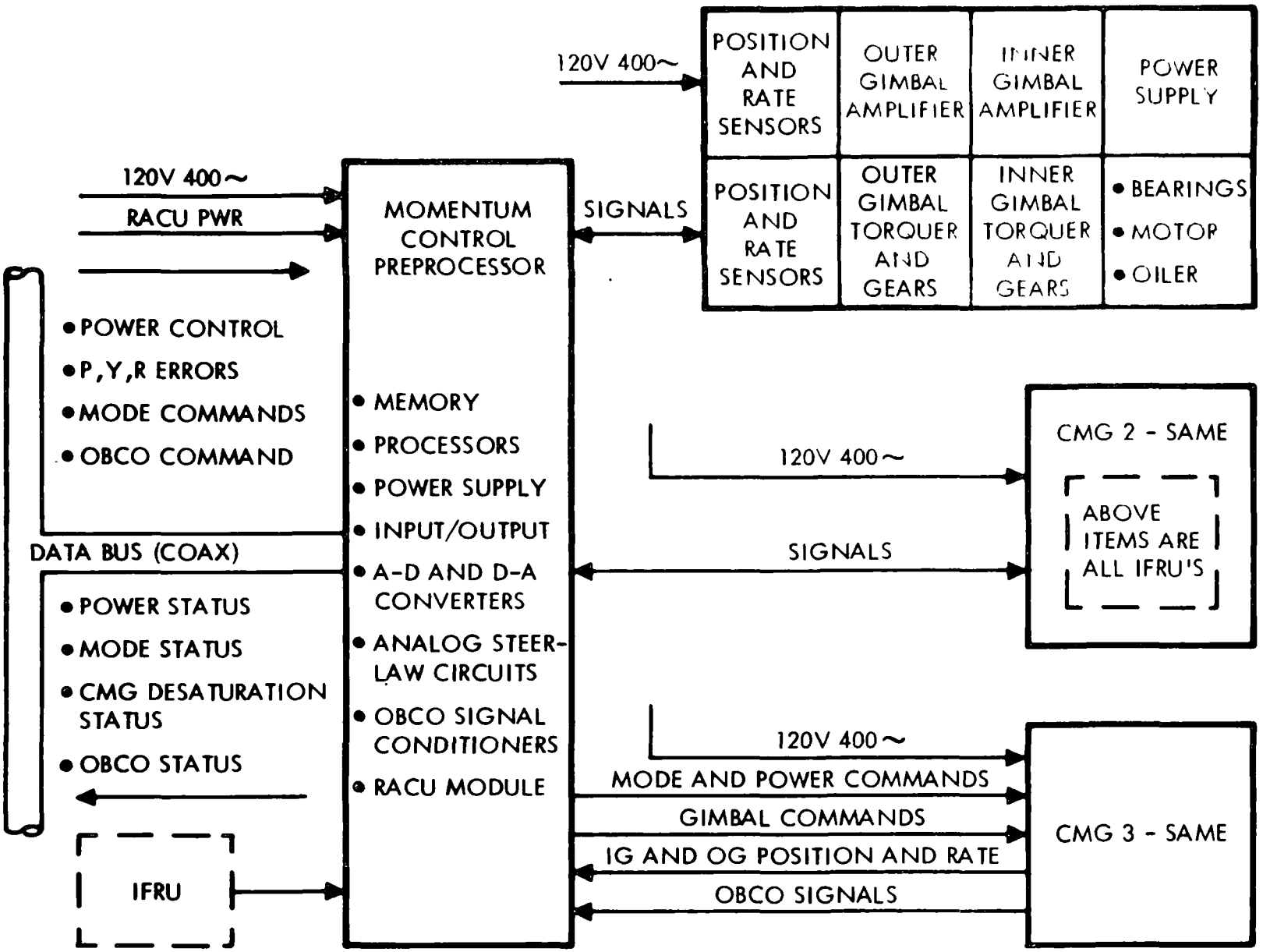


Figure 3.4-8. Control Moment Gyro Assembly

Space Division
North American Rockwell

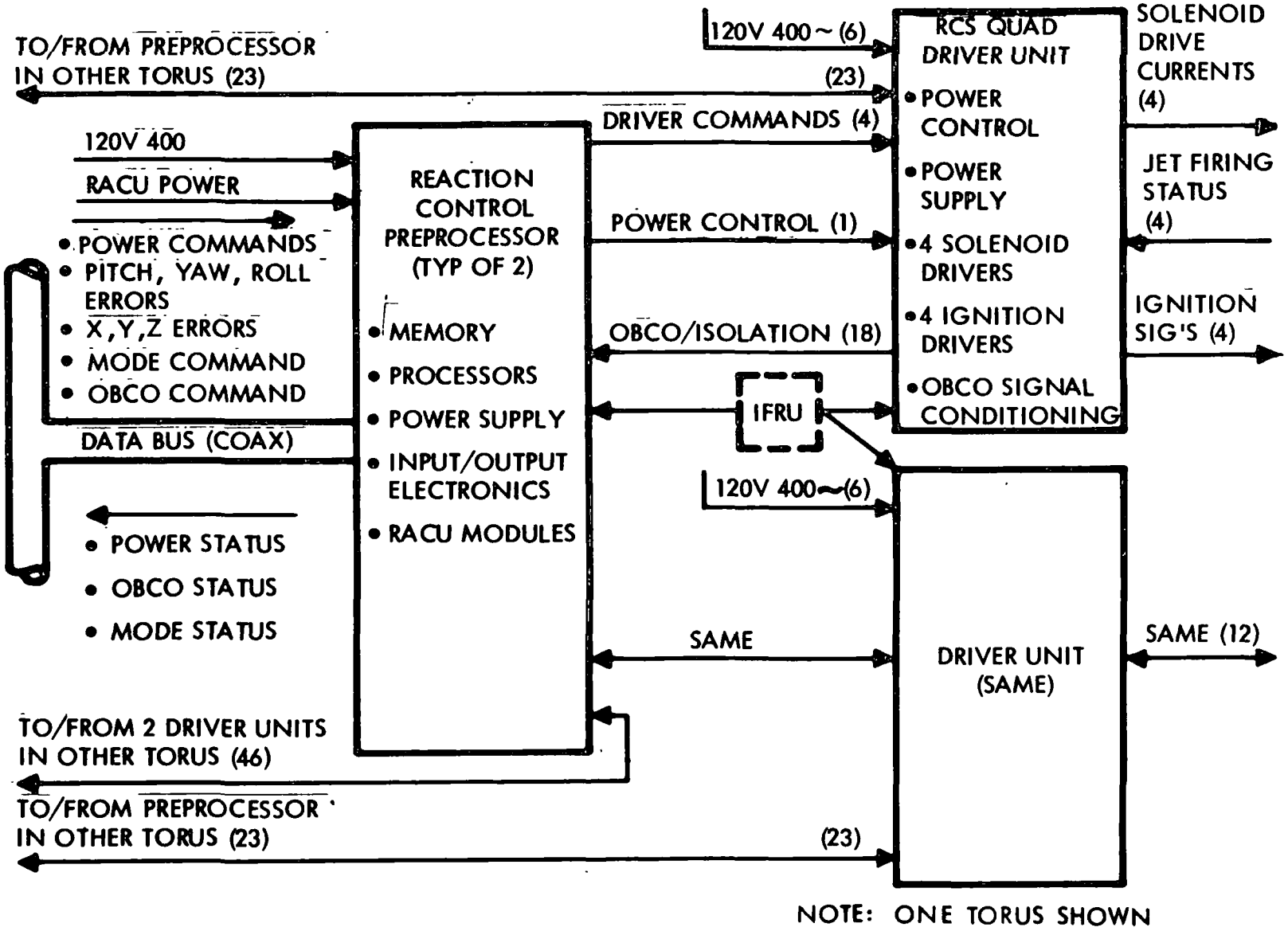


Figure 3.4-9. RCS Electronics Assembly

Space Division
North American Rockwell

The preprocessor incorporates the phase-plane switching logic and the jet select logic. It is assumed that the jet select logic will be reconfigurable. The preprocessor will be capable of operating directly with the optical reference system and with direct manual commands. This will allow the system to be configured into a simple but reliable backup failure mode.

The RCS preprocessor scales and sums the outer loop attitude and rate signals and sums these with any attitude maneuver and/or translation commands. It includes the phase-plane logic with deadbands as selected by the computer assembly and incorporates jet selection logic. It also accomplishes monitoring of OBCO signals from the jets and driver units. It can automatically disable a malfunctioning jet, or signal to central control via the monitor and alarm of the ISS for crew action.

A general discussion of preprocessor features is in the inertial reference preprocessor section.

The following are the physical characteristics of the jet driver electronics:

1. Weight - driver unit, 10 pounds; total, 40 pounds

2. Size

   a. Driver unit - 8 x 5 x 7 inches
   b. Volume per driver unit - 0.16 cubic feet; total, 0.64 cubic feet

3. Power Consumption

   a. Ten watts per driver unit when unit is on, has been estimated for bias. This would disappear if bias is found unnecessary in final design. As noted for the RCS preprocessor, the unit may be off most of the time. Transient power during firing of jets is estimated at 3 watts per jet driver. This does not include power consumed in the RCS solenoids and igniters themselves during firing; and is only that due to resistance in the driver switch, current sensor, and in the rectifier-filter.

   b. The duty factor of this transient power is low (estimated at less than 2 percent) for attitude holding and also for orbit maintenance, which only requires thrusting for a few minutes at a time. Simultaneous firing of more than 3 of the 4 jets of a quad is highly improbable.

4. Heat Rejection - conduction/convection to structure/air is the planned heat rejection concept

Computation Assembly

The computation assembly is composed of all G&C software not in the preprocessors. This software is in and receives executive control from the information subsystem (ISS). The computation assembly interfaces only with the ISS and with the G&C preprocessors as shown in Figure 3.4-10. The following G&C data are computed within the computation assembly:

1. CMG desaturation requirements (time-to-saturation prediction)
2. Current station attitude and rate and reference attitude alignment
3. Position vector of targets of opportunity in ECI coordinates (tracked by astronaut using G/C sextant/telescope)
4. Current station estimated state vector (position and velocity, orbital elements)
5. Shuttle return-to-earth guidance parameters
6. Experiment to G&C reference calibration data
7. Guidance targeting and delta-V commands for rendezvous, docking, deployment and stationkeeping of free-flying subsatellites, shuttle vehicles, and space tugs
8. Reaction jet commands and delta-V prediction for station orbit maintenance
9. G&C configuration status (real time)
10. G&C operation status (mode)
11. Real time failure identification and maintenance/replacement requirements
12. Energy management computations associated with jet firings
13. Control modeling, parameter estimation and adaptation
14. Landmark tracker and radar altimeter computations
15. Star tracker pointing commands
16. State vector propagation and update for free-flying modules, stationkeeping, and collision avoidance computations
17. FPE ground track and pointing angle

In support of these computations, the ISS will make the following information available to the G&C.

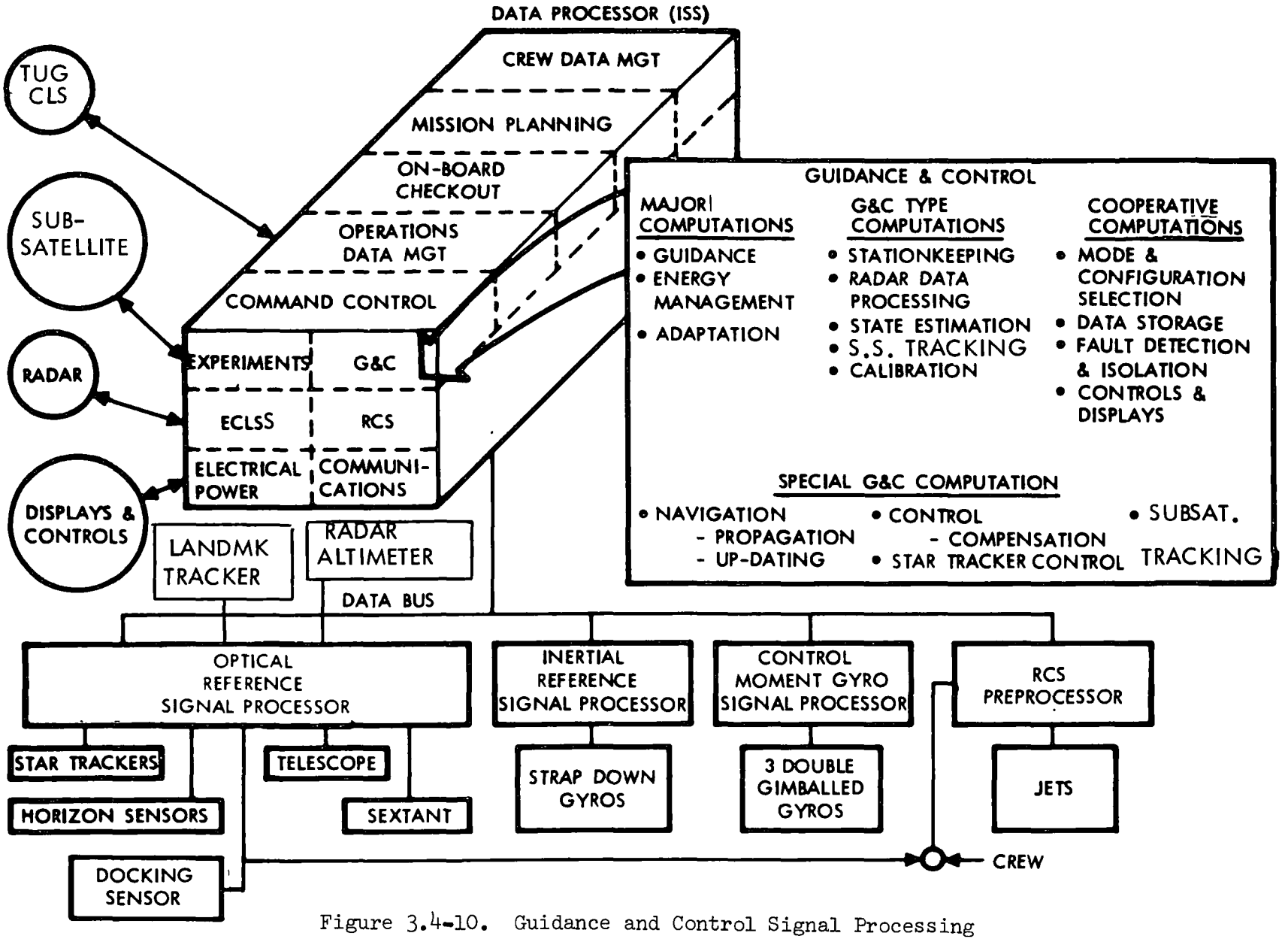


Figure 3.4-10. Guidance and Control Signal Processing

Space Division
North American Rockwell

1. Vehicle configuration and solar panel attitude
2. Solar panel orientation commands
3. Sun and moon ephemerides
4. Star catalog
5. Ground up-link data
6. Range, range rate, line of sight and line-of-sight rates from approach and tracking radars (for ranges > 1000 feet)
7. Scheduled initiation of station delta V's and CMG desaturation
8. Reaction jet attitude control inhibits and jet failure data
9. Crew interface - manual navigation sightings, operation mode commands, configuration commands, maintenance-in-progress/ accomplished data
10. Subroutines and bulk storage data loads including landmark identification and correlation data
11. Permanent and temporary data storage
12. Experiment reference alignment
13. Maneuver schedule

The physical characteristics of the total G&C subsystem are presented in Table 3.4-1. Weight and volume numbers include spares.

Table 3.4-1. G&C Physical Characteristics

Subassembly	No.	Weight (lb)	Avg. Power (watts)	Volume (cu ft)
Telescope/sextant	1	88	0	2.6
Star tracker	2	66	30	2.2
Landmark tracker	1	33	30	1.1
Horizon tracker	4	61	34	0.6
Radar altimeter	1	35	54	2.0
IMU	1	45	140	1.07
Docking laser	2	56	0	2.6
RCS drivers	5	50	40	0.8
CMG's	3	1520	151	167.0
Preprocessor	6	168	84	2.4
Navigation base	1	17	0	2.2
Calibration equip.	4	120	25	1.0
Total		2259	588	185.5

3.4.2 SELECTION RATIONALE

The two major G&C trade studies were the selection of an inertial reference assembly and the method of torquing or stabilizing the vehicle. A strapdown gyro system rather than a multi-gimballed platform was selected as the inertial reference assembly because of increased reliability, ease of maintenance, and lower power requirements. The total evaluation was reported in Volume IV. The torquing options are: RCS only, or RCS in conjunction with momentum exchange devices. In general, the selection of the control concept is a function of both vehicle configuration and flight mode. Conversely, propellant consumption and/or control moment gyro size and weight are a prime consideration in the selection of the vehicle configuration and the flight mode. The paragraphs below discuss the iterative process used to select both the vehicle configuration and flight mode and include consideration of heat rejection and experiment viewing requirements.

Vehicle Configuration

The total OLS vehicle configuration is made up of a power boom assembly with solar panels, a core module, two docked lunar landing tugs, a cargo module, and an experiments module. One tug is fully fueled and normally remains docked to the core module to provide rescue capability for personnel on the lunar surface or operating in the lunar vicinity. The second tug periodically conducts lunar landing and/or logistics missions.

Preferred Flight Mode

The only flight mode that is required by the experiments that are integral with the OLS is the local-level flight mode. Therefore, all possible local-level flight modes were considered. These modes are distinguished by which vehicle axis is oriented perpendicular to the orbit plane and which axis is oriented along the velocity vector. Thus, X-POP Y designates that the X axis is perpendicular to the orbit plane and the Y axis is in the direction of the velocity vector. The six possible local-level flight modes are illustrated in Figure 3.4-11. Inertial flight modes X-POP, Y-POP, or Z-POP were also evaluated.

In order to evaluate propellant and CMG requirements, a baseline profile of OLS configurations over a 109-day resupply cycle was developed. The mass properties and moments of inertia of these configurations (see Section 2.2) were approximated for use in this evaluation. The configurations during this period are summarized in Table 3.4-2. For example, during the first 16 days of each cycle, both tugs are full and docked to the core module. The cargo module is attached and full (Configuration A). Actually, the cargo module is gradually off-loaded into the core module during this 16-day period but was considered full for the analysis. At the end of this period, the cargo module is approximately half-full (with tug cargo) and remains so for the next 31 days (Configuration B). At the end of this period, Tug B departs with the cargo module and remains away for 31 days (Configuration D). The tug then returns without the cargo module and, with its fuel tanks empty, remains docked to the core module for 31 days (Configuration C).

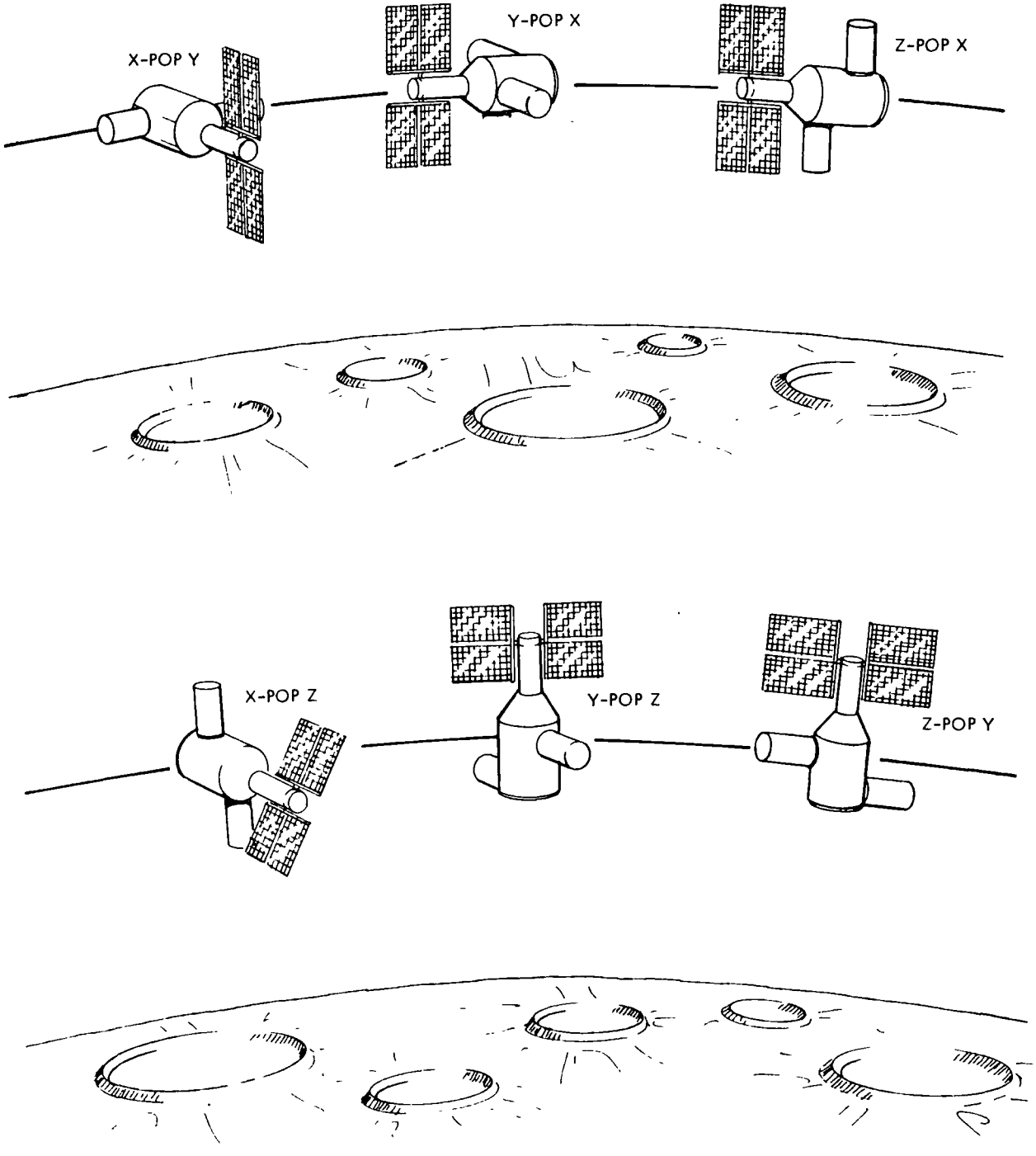


Figure 3.4-11. Local-Level Flight Modes Configuration 1'

Table 3.4-2. Docked Configuration

Configuration	Time Days*	Tug A		Tug B		Cargo Module	
		Full	Empty	Full	Empty	Full	Partial
A	16	X		X		X	
B	31	X		X			X
C	31	X			X		
D	31	X					

*Complete cycle = 109 days

There are four identified vehicle configurations to be considered in nine different flight modes. Tables 3.4-3 through 3.4-6 present the results of these analyses. Included in the data are:

1. Impulse required per orbit if CMG's are used
2. Desaturation impulse required per orbit if CMG's are used
3. CMG size in foot-pound-seconds
4. CMG weight
5. CMG power requirements
6. CMG volume requirements
7. Number of days to weight equivalence between CMG's plus desaturation propellant and RCS propellant if only an RCS is used

Evaluation of the data indicates that configurations A and B are very similar. Therefore, only configuration A is considered in later analyses. Also, the preferred local-level flight mode for the baseline configuration is X-POP Y. This flight mode requires the least amount of propellant for the RCS only case and also the smallest CMG's for the RCS-CMG case.

Figure 3.4-12 illustrates three different assemblages of the vehicles and modules of the OLS. Configuration 1 is the same as the baseline configuration except for the location of the cargo module. The relocation of the cargo module did not alter the preferred local-level flight mode for this configuration; therefore, only X-POP Y local-level data are presented for configuration 1 in Tables 3.4-7, 3.4-8, and 3.4-9.

The control system characteristics for configuration 2 are presented in Tables 3.4-10, 3.4-11, and 3.4-12. The similarity of configurations 1 and 2 permits deletion of some of the candidate modes. X-POP Y is similar to but preferred over Y-POP X, thus Y-POP X is deleted. The stabilization impulse and CMG weight requirements for the Z-POP Y mode are a magnitude greater than for the X-POP Y mode, therefore the Z POP Y mode was eliminated. The Z-POP X mode is included to illustrate the sensitivity to configuration orientation.

Table 3.4-3. Results of Stability and Control Analysis of Vehicle A

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 (ft3)	Days to Equivalence (I_{sp} = 220 sec)
X-POP Y	776	83	2,639	1,518	151	166	46
Y-POP X	849	195	3,347	1,703	160	198	54
Z-POP X	1,730	1,488	38,004	5,587	293	1,190	485
X-POP Z	1,239	71	3,557	1,754	162	207	31
Y-POP Z	1,164	119	3,900	1,835	166	222	38
Z-POP Y	1,599	1,348	36,887	5,505	291	1,164	460
X-POP Inertial	864	122	3,514	1,744	162	205	52
Y-POP Inertial	1,008	357	7,021	2,442	192	342	83
Z-POP Inertial	1,787	1,547	39,593	5,700	296	1,227	522

Table 3.4-4. Results of Stability and Control Analysis of Vehicle B

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 (I_{sp} = 220 sec)
X-POP Y	783	83	2,639	1,518	151	166	45
Y-POP X	855	199	3,404	1,717	160	200	55
Z-POP X	1,733	1,493	38,125	5,595	293	1,193	489
X-POP Z	1,239	70	3,558	1,755	162	207	31
Y-POP Z	1,171	124	3,991	1,855	167	225	38
Z-POP Y	1,609	1,352	37,002	5,514	291	1,167	451
X-POP Inertial Y-POP Inertial Z-POP Inertial	Essentially the same as Vehicle A						

Space Division
North American Rockwell

Table 3.4-5. Results of Stability and Control Analysis of Vehicle C

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 (I_{sp} = 220 sec)
X POP Y	596	83	2,374	1,442	147	154	59
Y-POP X	646	152	2,573	1,499	150	163	63
Z-POP X	1,211	984	24,210	4,476	262	852	414
X-POP Z	903	70	2,876	1,582	154	177	41
Y-POP Z	792	74	2,429	1,458	147	157	42
Z-POP Y	1,081	861	23,558	4,417	260	835	421
X-POP Inertial	602	67	1,948	1,310	140	133	54
Y-POP Inertial	745	255	4,308	1,925	170	238	86
Z-POP Inertial	1,242	1,017	25,090	4,555	264	875	444

Table 3.4-6. Results of Stability and Control Analysis of Vehicle D

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 (I_{sp} = 220 sec)
X-POP Y	471	86	2,350	1,435	146	153	78
Y-POP X	477	119	2,049	1,343	141	138	78
Z-POP X	762	536	11,993	3,173	220	507	294
X-POP Z	609	72	2,494	1,477	148	160	59
Y-POP Z	470	40	1,468	1,143	130	108	55
Z-POP Y	657	425	11,639	3,127	218	496	297
X-POP Inertial	438	23	1,072	982	120	86	52
Y-POP Inertial	531	179	2,558	1,495	149	163	93
Z-POP Inertial	776	550	12,372	3,221	221	519	314

Space Division
North American Rockwell

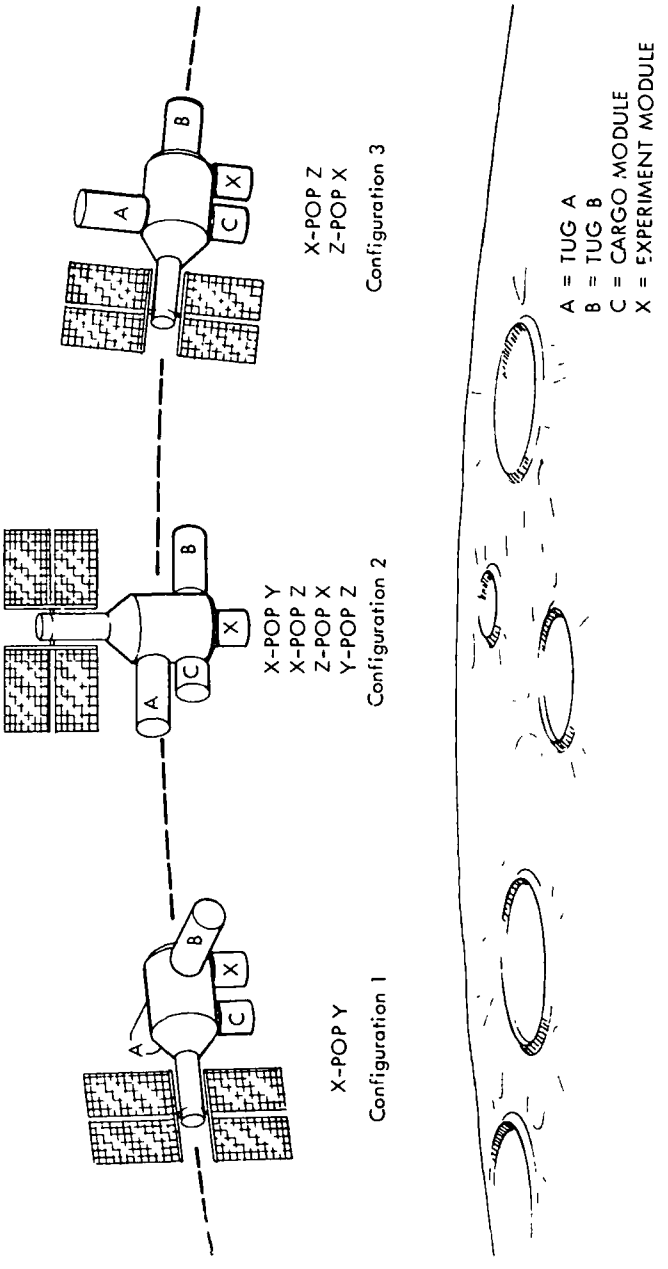


Figure 3.4-12. Vehicle Configurations

Table 3.4-7. Results of Stability and Control Analysis of Vehicle 1_A

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 (I_{sp} = 220 sec)
X-POP Y	927	76	2,648	1,520	151	167	39
X-POP Inertial	1,006	132	3,821	1,817	165	218	46
Y-POP Inertial	1,134	454	9,649	2,853	208	432	92
Z-POP Inertial	1,854	1,641	42,167	5,880	301	1,286	608

Space Division
North American Rockwell

Table 3.4-8. Results of Stability and Control Analysis of Vehicle 1C

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 (I_{sp} = 220 sec)
X-POP Y	597	84	2,392	1,447	147	155	62
X-POP Inertial	600	66	1,925	1,303	139	132	54
Y-POP Inertial	746	256	4,332	1,931	170	239	87
Z-POP Inertial	1,242	1,065	25,090	4,555	264	875	445

Space Division
North American Rockwell

Table 3.4-9. Results of Stability and Control Analysis of Vehicle 1_D

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 (I_{sp} = 220 sec)
X-POP Y	471	86	2,350	1,435	146	153	82
X-POP Inertial	438	23	1,072	982	120	86	52
Y-POP Inertial	531	179	2,558	1,495	149	163	93
Z-POP Inertial	776	550	12,372	3,221	221	519	314

Space Division
North American Rockwell

Table 3.4-10. Results of Stability and Control Analysis of Vehicle 2_A

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 (I_{sp} = 220 sec)
X-POP Y	815	90	2,760	1,551	152	172	47
Z-POP X	1,686	1,451	36,976	5,512	291	1,167	516
X POP Z	1,223	78	3,627	1,771	163	210	34
Y POP Z	1,171	151	4,476	1,962	172	245	42

Space Division
North American Rockwell

Table 3.4-11. Results of Stability and Control Analysis of Vehicle 2_C

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 (I_{sp} = 220 sec)
X-POP Y	616	71	2,235	1,400	144	147	57
Z-POP X	1,172	940	23,027	4,368	259	821	415
X-POP Z	875	59	2,718	1,540	152	170	42
Y-POP Z	827	108	3,067	1,633	156	186	50

Space Division
North American Rockwell

Table 3.4-12. Results of Stability and Control Analysis of Vehicle 2_D

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 (I_{sp} = 220 sec)
X-POP Y	442	54	1,605	1,193	133	116	68
Z-POP X	763	546	12,263	3,207	221	515	325
X-POP Z	574	41	1,808	1,264	137	126	52
Y-POP Z	525	78	1,997	1,326	140	136	65

Configuration 3 control system characteristics are presented in Tables 3.4-13, 3.4-14, and 3.4-15. The only two modes of interest in this configuration are X-POP Z and Z-POP X. All other modes have larger control system requirements than the comparable modes of the other configurations. These two modes were evaluated because of the view aspect of the experiments module. The analyses in Section 2.1 of this report indicated that the preferred experiments module orientation was along the local vertical (end-viewing).

The data from Tables 3.4-6 through 3.4-15 are summarized in Table 3.4-16. Vehicle 2 in the X-POP Z mode has the lowest desaturation impulse requirement. Thus, from a propellant resupply standpoint, this is the optimum (minimum propellant) configuration. The right-hand column of the table indicates the 190-day resupply weight penalty for choosing a less optimum configuration. The Z-POP X modes of Vehicles 2 and 3 are ruled out, based on this propellant penalty. The Y-POP Z mode of Vehicle 2 is ruled out, based on the radiator heat rejection requirements discussed in Section 3.6.2. Only X-POP modes are left and the relative propellant penalty among these modes are small. Because the weight penalties associated with designing the ECLSS radiators are greater than the associated propellant penalty, the X-POP Z mode for Vehicle 2 is ruled out. Thus, the choice is between Vehicle 1 and Vehicle 2 in the X-POP Y mode and Vehicle 3 in the X-POP Z mode; the propellant penalties are 52 pounds, 32 pounds, and 60 pounds, respectively over a 109-day period. The penalties for designing the experiment module for side-viewing rules out Vehicle 2 in the X-POP Y mode, and Vehicle 1 in X-POP Y is seen to require less propellant and lighter CMG's than Vehicle 3 in X-POP Z. Vehicle 1 is therefore the chosen vehicle, and is flown in the X-POP Y level mode. Examination of the inertial mode data in the tables indicates that the preferred inertial mode is X-POP. The CMG's are sized for the level mode but can accommodate the inertial mode through the expenditure of additional RCS propellant.

Momentum Exchange Trade

A planar array of three double gimbal CMG's was chosen as the momentum exchange device to be considered, based on EOSS experience. From the tables given above, the CMG weight, power, and volume required for each configuration can be determined as discussed in Volume IV, Configuration and Systems Analysis Report. For the chosen configuration, Vehicle 1 in the X-POP Y level mode, the RCS-only propellant consumption is compared to the CMG weight plus the desaturation propellant consumption (Figure 3.4-13). The two extreme configurations are plotted: Configuration 1 with full tugs and cargo module and Configuration 2 with tug B and without the cargo module. The numbers in the figure are obtained from the tables above, but with the I_{sp} converted from 220 to 419 seconds. The intersection of the RCS-only propellant line with the CMG weight and desaturation propellant line represents the days to equivalence (note that the time on the tables is therefore lowered by the ratio 220/419). It can be seen from the figure that even in the worst case, the RCS-only propellant exceeds the CMG weight plus desaturation propellant after 164 days and continues to increase at a rate of about 10 pounds per day thereafter. Thus, since the OLS mission duration is far in excess of 164 days, it

Table 3.4-13. Results of Stability and Control Analysis of Vehicle 3_A

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 (I_{sp} = 200 sec)
Z-POP X	1,459	1,215	30,523	5,016	278	1,012	452
X-POP Z	1,236	121	4,361	1,937	171	241	38

Space Division
North American Rockwell

Table 3.4-14. Results of Stability and Control Analysis of Vehicle 3C

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 (Isp = 220 sec)
X-POP Z	1,203	984	24,229	4,478	262	853	451
X-POP Z	923	69	2,915	1,593	154	179	41

Space Division
North American Rockwell

Table 3.4-15. Results of Stability and Control Analysis of Vehicle 3D

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 (I_{sp} = 220 sec)
Z-POP X	760	550	12,250	3,200	220	510	320
X-POP Z	570	45	1,800	1,260	135	1,250	50

Space Division
North American Rockwell

Table 3.4-16. CMG Weight and Propellant Consumption Comparison

Vehicle Configuration	Flight Mode (local level)	Desaturation Impulse (1) (pound-seconds)	CMG Weight (pounds)	Propellant Penalty (1), (2) (pounds)
1	X-POP Y	91,000	1,520	52
2	X-POP Y	83,000	1,550	32
	X-POP Z	69,500	1,770	0
	Z-POP X	1,430,000	5,512	3,200
	Y-POP Z	132,000	1,960	150
3	Z-POP X	1,330,000	5,020	3,000
	X POP Z	93,500	1,937	60

(1) Per 109-day cycle resupply

(2) I_{sp} = 419 seconds, penalty above minimum propellant

Space Division
North American Rockwell

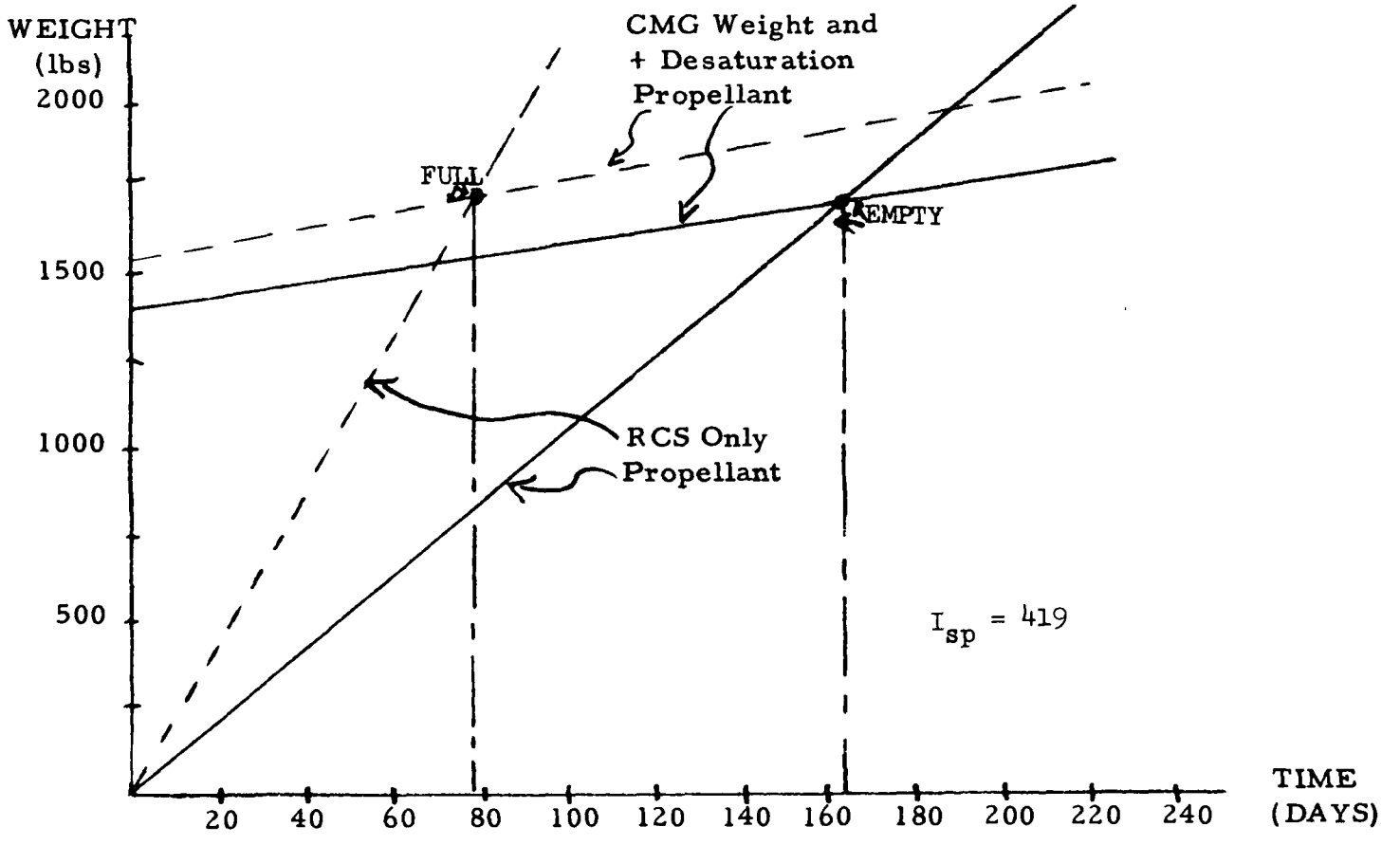


Figure 3.4-13. RCS Only vs. RCS + CMG's Trade

Space Division
North American Rockwell

is obvious that substantial propellant savings can be achieved by using CMG's. The CMG size required is approximately the same as that chosen for the EOSS (1600 pounds) and to minimize cost, this size would probably be used on the OLS.

3.4.3 ADAPTABILITY TO ARTIFICIAL-G OPERATION

Assuming that solar panel orientation requirements do not constrain vehicle orientation (probably accomplished by relocating the panels to the non-rotating hub), the vehicle would be rotated about a major inertia axis with the momentum vector perpendicular to the orbit plane. This orientation allows an experiment module located on the hub to maintain a local level attitude without requiring precession of the momentum vector. If the RCS engines are properly located, orbit maintenance can be accomplished without maneuvering the vehicle or spinning down. The G&C subsystem would then be capable of meeting all pointing and stabilization requirements. The only changes necessary would be to relocate the G&C sensors to the hub, add a tachometer to the hub assembly for spin rate control, and add a mass balancing system. The CMG's would provide wobble damping and would have to be resized.

Page intentionally left blank

3.5 REACTION CONTROL SUBSYSTEM

3.5.1 CONCEPT DEFINITION

The reaction control subsystem (RCS) must provide thrust for three functions in zero gravity. These functions are control moment gyro (CMG) backup and desaturation, maneuvering, and orbit maintenance. If artificial gravity is used in the OLS mission, the RCS also will provide thrust for spin-up, despin, and momentum vector control.

The nominal design storage period for the RCS is 180 days and the RCS impulse requirement for this time period has been determined to be 2,214,000 pounds-seconds plus emergency impulse. The emergency impulse would be used to counter any thrust developed by cabin atmosphere gases escaping through a pressure hull puncture. The impulse requirements are summarized in Table 3.5-1 along with the subsatellite requirements (discussed later in this section).

Table 3.5-1. Propulsion Requirements (180 Days)

RCS Requirements	
CMG desaturation	153,000 lb-sec
Maneuvers	12,000 lb-sec
Orbit maintenance	2,049,000 lb-sec
Total	2,214,000 lb-sec
Emergency	82,000 lb-sec
Total	2,296,000 lb-sec
Subsatellite Requirements	
N_2 storage	455 lbm
N_2H_4 storage	7811 lbm

The RCS concept selected to meet the Table 3.5-1 criteria is an oxygen and hydrogen gas design that uses cryogenic liquid storage and vapor pressure propellant feed. A schematic of this subsystem is shown in Figure 3.5-1. The propellants are stored in insulated tanks in unpressurized sections of the OLS. These feed to the propellant conditioning units where any entrained liquid is vaporized and the propellant temperature is increased to 530 R. From this point, the gases are sent to accumulators for temporary storage. If the propellant tanks have cooled, allowing the vapor pressure to drop, a portion of the conditioned gases will be recirculated to the storage tanks, warming them and restoring the vapor pressure. The conditioning units obtain heating energy from the ECLSS Freon cooling loop. Finally, propellant is fed on demand from the accumulators to the RCS engines through pressure regulators. The engines are located in four engine clusters (on the plus and minus Z-axis) on the forward and aft environmental shields.

The RCS also will provide oxygen and nitrogen to the ECLSS for metabolic use, leakage makeup, and emergency repressurization, and will provide oxygen and hydrogen to the EPS for emergency power. ECLSS supplies the conditioned nitrogen for the subsatellite provisions.

The subsatellite provisions include bellows expulsion hydrazine tanks, high-pressure nitrogen tanks (for expulsion), nitrogen compressors, and associated lines, valves, and fittings. The compressors supply nitrogen to both the high-pressure nitrogen tanks and similar tanks on the subsatellites. A schematic of the subsatellite provision hardware is shown in Figure 3.5-2.

3.5.2 SELECTION RATIONALE

The primary RCS requirement is to provide the total impulse required during the 180-day design period. In Volume IV of this report, the RCS mass and volume characteristics were presented as functions of total impulse. Important drivers in the selection of RCS propellants and the resultant size of the RCS are specific impulse, toxicity, corrosivity, explosivity, reliability, electrical power requirements, and plume cleanliness. These factors were considered in Volume IV and the RCS options remaining are listed in Table 3.5-2. The table show that two of the options include low-thrust (less than one pound) engines. As explained in Volume IV, the low-thrust engines would be used for orbit makeup and possibly for CMG desaturation. Medium-thrust engines could be used for these functions, but in the options presented, the low-thrust engine would be more efficient. By using waste ECLSS gas, the resupply requirements are reduced. The penalties for using a low-thrust system include the problem of having an additional RCS and the fact that the engines must operate for much longer periods (perhaps 100 times as long). The long engine operating period is not as serious as expected because plumes formed are "clean" and much less material is expelled (perhaps a hundredths as much) per unit time.

As indicated by the impulse requirements listed in Table 3.5-1, most of the total impulse requirement for the OLS RCS is for orbit maintenance (2,049,000 out of 2,214,000 pound-seconds). Any flight plan that significantly changes the orbit maintenance requirements will directly affect the RCS sizing and possibly the concept selections. Figure 3.5-3 is a composite of

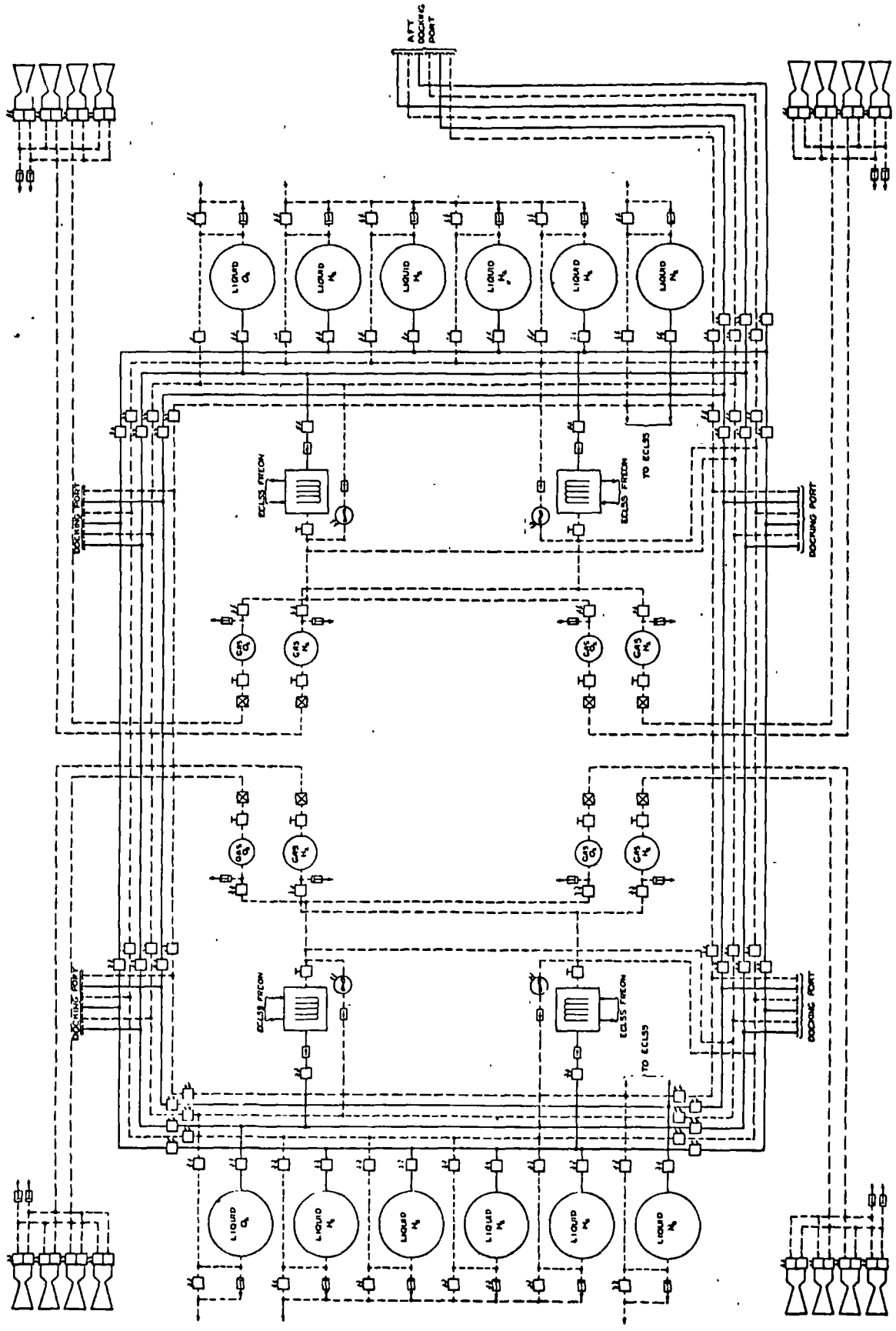


Figure 3.5-1. RCS Schematic

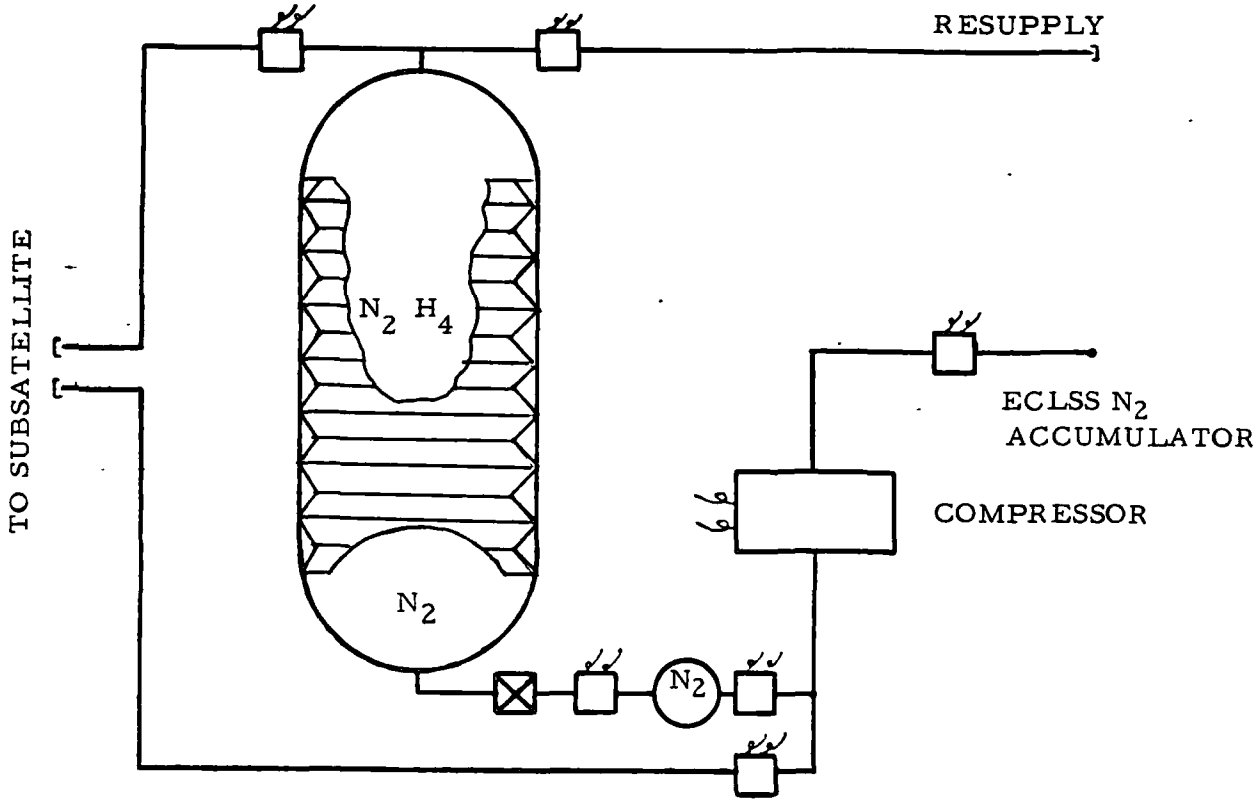


Figure 3.5-2. Subsatellite Propellant Provisions

Table 3.5-2. OLS RCS Options

Options	Medium Thrust Only	Medium Thrust with Low Thrust, Waste Gas, Resistojets
N_2H_4 (earth storable)	X	X
O_2/NH_3 (cryogenic/earth storable)	X	X
O_2/H_2 (cryogenic storable)	X	

three figures from Volume IV and shows the effect of impulse requirements on the comparative masses of the candidate propellant concepts. Figure 3.5-3 shows that for the total impulse requirement in question, the total masses of both the oxygen/ammonia and the oxygen/hydrogen concepts are very close, with the oxygen/hydrogen concept slightly lighter at about 8450 pounds. Figure 3.5-4 also is a composite of three figures from Volume IV and shows the relative propellant volumes as a function of required impulse for the candidate medium-thrust concepts. The oxygen/hydrogen concept requires a rather large propellant volume of 420 cubic feet, compared with approximately 146 cubic feet for the other two concepts. However, there would be little difficulty in accommodating 420 cubic feet of propellant in the unpressurized areas of the representative OLS; hence, this is not an important driver. It should also be noted that because of its high specific impulse, the oxygen and hydrogen concept has the lowest resupply mass; hydrazine has the greatest resupply mass.

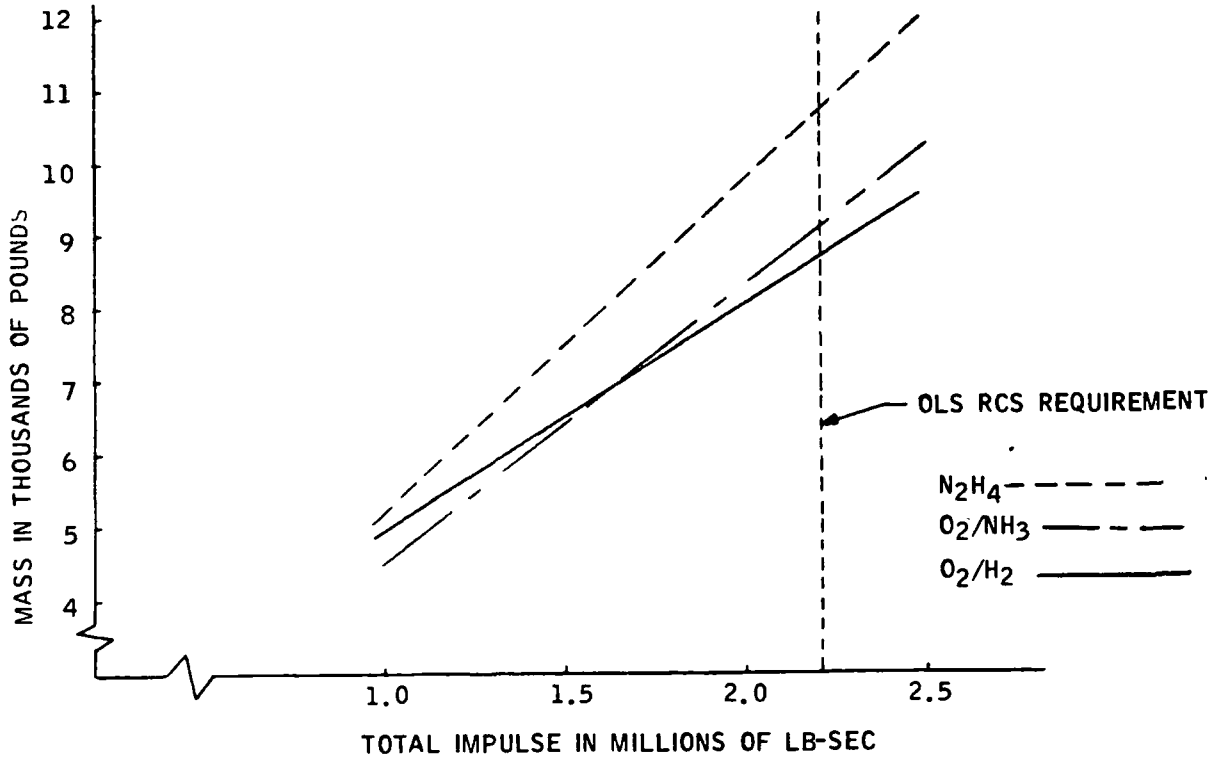


Figure 3.5-3. RCS Loaded Mass Comparison

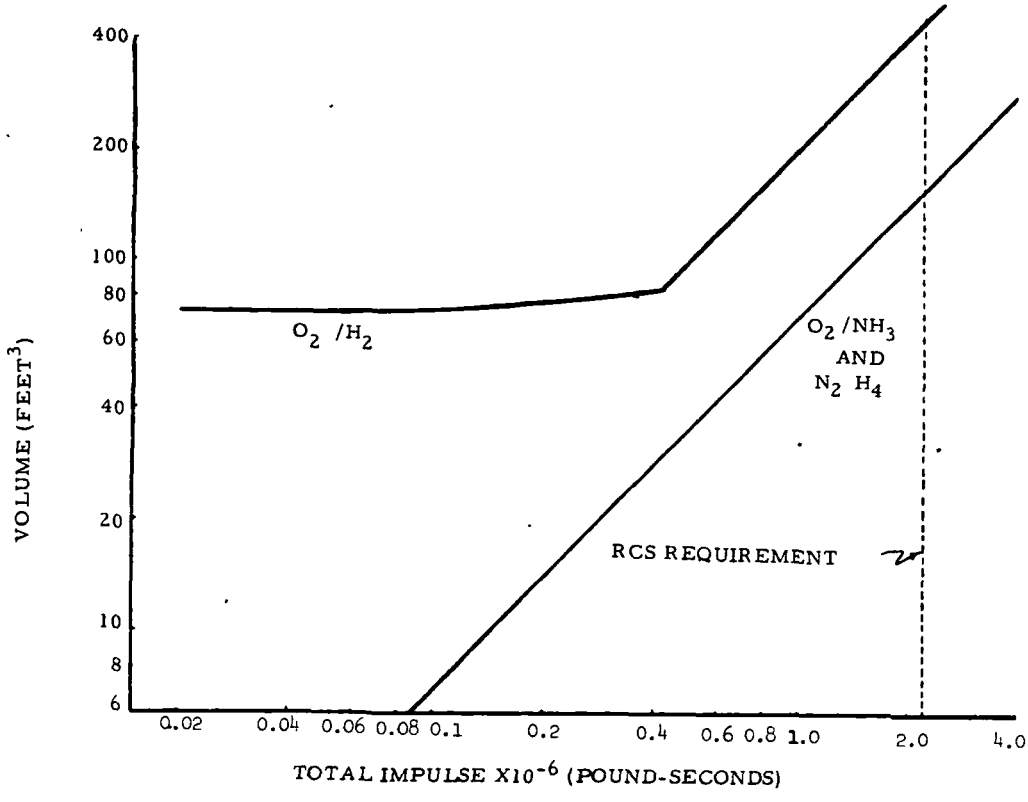


Figure 3.5-4. RCS Propellant Volume Comparison

Each of the three medium thrust level concepts could be integrated with other subsystems to provide an overall weight savings. The hydrazine concept could be integrated with the EPS to provide emergency power through the use of a gas generator operating a turbine driven generator. This is an inefficient and heavy concept. However, the hydrazine also could be used to provide nitrogen to the ECLSS through a dissociator. The cryogenic oxygen and ambiently stored ammonia concept could be used to supply oxygen to the EPS. The oxygen could be used in conjunction with high-pressure hydrogen for producing emergency electrical power via fuel cells. High-pressure rather than cryogenically stored contingency hydrogen is recommended because of the boiloff losses associated with cryogenic storage. Emergency power also could be supplied through an oxygen and ammonia gas turbine at a cost of about a fifty percent increase in propellant consumption over an oxygen and hydrogen fuel cell design. Oxygen and ammonia also can be used to supply oxygen and nitrogen to the ECLSS. The nitrogen would be supplied through electrolysis or extra cryogenic storage tanks.

The oxygen and hydrogen RCS concept is the most flexible of the three choices for subsystem integration. Oxygen and hydrogen may be used directly (after conditioning) by the EPS for emergency power using fuel cells. In addition, oxygen may be used directly by the ECLSS for metabolic consumption, leakage makeup, and repressurizations. The addition of cryogenic nitrogen tanks permits the storage of nitrogen for leakage makeup and repressurization. The tank design for cryogenic nitrogen storage could be identical to the oxygen tanks.

Another reason for including cryogenic nitrogen in the two concepts that use cryogenic oxygen is for the subsatellite provisions. Also required for the subsatellites is hydrazine. This, of course, would integrate well with a hydrazine medium-thrust RCS concept, but the advantage is not great.

Considering all of the previous factors, the cryogenically stored oxygen and hydrogen concept was selected for the representative OLS. The hydrazine concept would be heavier and would not integrate as efficiently with the ECLSS and the EPS when compared with the other options. The oxygen/ammonia concept is comparable in mass to the oxygen and hydrogen concept and has advantages over the oxygen/hydrogen concept in that it takes less volume, and liquid ammonia is much less difficult to store than liquid hydrogen. However, storage difficulty is not a significant driver because of the expected heavy, continuous propellant use. The hydrogen will be used as rapidly as it boils off. The primary disadvantage to the oxygen and ammonia concept is that the ammonia constitutes an additional fluid to be carried to the OLS on a regular basis. On the other hand, liquid hydrogen and oxygen must be brought up regularly for the space tugs. The option of utilizing tug boiloff also favors selection of the oxygen/hydrogen RCS concept. This option is discussed later in this section. The tradeoff between using hydrogen or ammonia fuel is rather close, but hydrogen would appear to be the best choice at this time.

After oxygen and hydrogen were selected as the RCS propellants, interface requirements with EPS and ECLSS were established. These requirements are outlined in Table 3.5-3. ECLSS leakage, metabolic makeup, and Sabatier reactor consumptions are day-to-day functions and these quantities of oxygen and hydrogen should be added to the normal RCS requirements. The ECLSS repressurization requirement results from the contingency condition of a pressure structure puncture. The RCS emergency propulsion requirement of 82,000 pound-seconds (Table 3.5-1) is also a puncture contingency in which the RCS is used to counter torques and translations caused by the loss of the crew compartment atmosphere through a hole in the hull. The RCS puncture contingency propulsion requirement is 147 pounds of oxygen and 49 pounds of hydrogen. The EPS emergency power requirement is a separate contingency not connected with the puncture problem. These two contingency requirements establish the amount of contingency oxygen and hydrogen required. The emergency power contingency is the worst failure as far as oxygen and hydrogen are concerned. The nitrogen contingency is determined by the repressurization requirement. The total propellant requirements including these contingency requirements are summarized in Table 3.5-4. The subsatellite provisions shown in the table will be discussed later in this section.

Table 3.5-3. Subsystem Integration (180 Days)

ECLSS	
Leakage	O_2 - 1247 lbm
	N_2 - 3096 lbm
Repressurization	O_2 - 350 lbm
	N_2 - 1150 lbm
Sabatier unit	H_2 - 324 lbm
EPS	
Emergency power	O_2 - 3697 lbm
	H_2 - 533 lbm

Table 3.5-4. Propellant Summary (lbm)

	O_2	H_2	N_2	N_2H_4
RCS - 180 days	3963	1321	-	-
ECLSS - 180 days	1247+	324	3096	-
Total	5210	1645*	3096*	
RCS - 109 days	2400	800	-	-
ECLSS - 109 days	726+	196	1875	-
Contingencies	3697	533	1150	-
Total	6823*	1529	3025	
Subsatellite provisions	-	-	455*	7811*
Required loading (sum of * items)	6823	1645	3551	7811

+Includes extra 18 lbm metabolic load for crew overlap

There are two criteria considered in determining the totals shown in Table 3.5-4. The nominal sizing criteria for the RCS is the 180-day without resupply, operational guideline requirement. The normal resupply period is 109 days. Therefore, the cryogenic storage requirement is based upon 180 days normal usage or 109 days normal resupply plus contingencies, whichever is larger. Table 3.5-4 shows that the 109 days plus contingencies requirement sized the oxygen loading while the 180-day normal usage criteria sized the hydrogen and nitrogen loading. The resultant required loadings are 6823 pounds of oxygen, 1645 pounds of hydrogen, and 3551 pounds of nitrogen.

Data from Beech Aircraft Company indicated that if only one cryogenic storage tank is designed rather than two, there would be a 30-percent savings in development cost and a six-month reduction in development time. An even greater savings would be possible in employing only one tank design rather than three. Using this criteria as a guide, a common 64.4-cubic foot tank was selected. The tank would provide for a nominal pressure of 100 psia and have at least 10 percent ullage. Table 3.5-5 summarizes the resultant tank volumes and the potential excess cryogenic loading.

Table 3.5-5. Cryogenic Fluid Tankage

Cryogenic Fluid	Required Load (lbm)	Number of Tanks	Possible Load (lbm)	Possible Excess (lbm)
O_2	6823	2	7297	474
H_2	1645	8	1645	0
N_2	3551	2	5062	1511

Note: Assumes 64.4 ft^3 tanks using 10 percent ullage

The cryogenic concept (Figure 3.5-1) is to be designed to operate on its own vapor pressure of 100 psia. The nominal temperature of the tanks will correspond to this vapor pressure. The nominal temperature of the oxygen will be 204 R, the temperature of the hydrogen will be 52 R, and the temperature of the nitrogen will be 177 R. The vapor pressure will cause the fluids to flow from the tanks on demand. The flow to the propellant conditioning units may be either liquid or gas. The nitrogen flow to the ECLSS nitrogen conditioning assembly may also be either liquid or gas. The propellant conditioners are heat exchangers designed to vaporize any entrained liquid and warm the propellants to 530 R. The heat source will be waste heat from the ECLSS Freon coolant.

Downstream of the propellant conditioners the H_2 and O_2 propellant lines branch. One branch will go to the propellant accumulators. The other branch will have a recirculation fan used to blow conditioned propellant back into the propellant tanks. During periods of heavy use, the propellant tanks will cool, causing a drop in vapor pressure. The recirculation fans will be used to rewarm the tanks and maintain the vapor pressure.

The accumulators store the conditioned propellants until they are required by the engines, at which time the propellants flow through pressure regulators to the engines where they are burned at 30 psia. The engines will be located at the outer radius of the forward and aft micrometeoroid shields on the plus and minus Z-axis as shown in Drawing 2282-25, Sheet 2. Each RCS quad will contain a cluster of four engines, permitting three-axis translation, pitch, yaw, and roll. Each engine will produce 10 pounds of thrust. This thrust level is adequate for all the proposed RCS operations.

The estimated mass of the proposed RCS is summarized in Table 3.5-6. The loaded mass (including 455 pounds of nitrogen for the subsatellites) is 17,154 pounds. If the excess cryogenic storage capability is added, the loaded mass becomes 19,139 pounds.

Table 3.5-6. RCS Mass Summary

Dry mass	5,135 lbm
Cryogenics*	12,019 lbm
	17,154 lbm
Excess cryogenics	1,985 lbm
Total	19,139 lbm
Subsatellite Provisions	
Dry mass	3,487 lbm
N_2H_4	7,811 lbm
Total	11,298 lbm

*Includes 455 lbm of N_2 for the subsatellite provisions

As indicated in Table 3.5-1, the RCS must provide propellants for the subsatellites. Two types of subsatellites were identified in Volume I. One type, similar to the lunar orbiter, has a relatively large total impulse requirement and uses cold gas (nitrogen) thrusters. These fluids would be stored on the OLS until they are required and then transferred to the subsatellite. Normally, in an 18-month period, it will be necessary to resupply low total impulse subsatellites four times and high total impulse subsatellites twice.

Nitrogen already is stored cryogenically onboard the OLS. The nitrogen supplied by RCS to ECLSS is conditioned by ECLSS and stored in ECLSS accumulators. This stored nitrogen gas may be tapped and run through compressors for use by the subsatellites. Assuming that three stage compressors will transfer 50 pounds per hour from 100 psia to a maximum of 4000 psia, the peak power required will be 3740 watts. This requirement is very high, but it may be scheduled for low OLS power usage periods and should present no particular difficulty. The operating time for the compressors will average about one hour per month. The compressed gas could be piped directly to the subsatellites or to the pressurant tanks on the OLS for use in expelling the stored hyrazine.

The hydrazine storage requirement of 7811 pounds includes an assumed expulsion efficiency of 95 percent with bellows tanks. The total dry mass of the subsatellite provisions is shown in Table 3.5-6 as 3487 pounds. When added to the hydrazine the mass becomes 11,298 pounds. However, it should be noted that 455 pounds of subsatellite nitrogen is included in the required cryogenics for the RCS. A schematic of the subsatellite provisions is shown in Figure 3.5-2.

The final aspects of the RCS and subsatellite provisions to be considered are the volume and power estimates. These values are presented in Table 3.5-7. As indicated in the table, the major volume requirements in the RCS are for tankage and accumulators. This is caused primarily by the low density of liquid and gaseous hydrogen. The subsatellite provisions are quite compact by comparison because of the relatively high density of hydrazine.

The average power loads presented in Table 3.5-7 are very low due to the short duration of the expected electrical power requirements. However, some of the peak loads are notable. The peak engine load assumes the simultaneous ignition of four engines and the activation of the associated eight injector valves. This event would not occur often, and the peak load would be momentary. The high power associated with the subsatellites provision feed control includes the nitrogen compressors previously mentioned. It is a rare event of one to two hours duration. The power builds up steadily from about 100 or 200 watts and reaches a maximum just prior to shutdown. As mentioned previously, this would be scheduled for low power usage periods.

Table 3.5-7. RCS Volume and Power Summary

Item	Volume (ft^3)	Power (watts)	
		Peak	Average
RCS			
Tankage	773	0	0
Conditioning	1	104	1
Feed control	100	336	1
Engines	16	1008	2
Accumulators	294	0	0
Total	1184		4
Subsatellite Provisions			
Tankage	149	0	0
Feed control	30	3970	3
Total	179		3

Utilization of Tug Boiloff

It may be possible to obtain oxygen and hydrogen from the docked space tug vehicles and, thereby, reduce the RCS resupply requirements. The boiloff from the tugs is currently estimated to be 24 pounds per day of oxygen and 24 pounds per day of hydrogen for each vehicle. At least one of these vehicles may be assumed to be docked to the OLS for at least 107 days out of each 109-day resupply period. An analysis has been performed to determine how this tug boiloff could be effectively utilized and how the OLS cryogenic storage tanks shall be resized if this option is selected.

For this analysis, the following assumptions were made for liquid oxygen and hydrogen at their 100 psia boiling points.

Heat of vaporization $\left(\frac{\text{Btu}}{\text{lbm}}\right)$:

$O_2 = 78$

$H_2 = 142$

Density $\left(\frac{\text{lbm}}{\text{ft}^3}\right)$:

$O_2 = 63$

$H_2 = 3.55$

OLS cryogenic tank insulation capabilities $\left(\frac{\text{Btu}}{\text{hr ft}^2}\right)$:

$O_2 = 0.10$

$H_2 = 0.07$

Using these values it may be shown that:

O_2 : $\dot{b} = 0.218\ V^{2/3}$

H_2 : $\dot{b} = 0.084\ V^{2/3}$

where:

$\dot{b}$ = boiloff rate for each OLS storage tank $\left(\frac{\text{lbm}}{\text{day}}\right)$

V = volume of each OLS storage tank (ft^3)

Beacuse of the low density of hydrogen, the OLS RCS storage tank sizing was based on hydrogen and verified for oxygen.

In sizing for the worst case, the propellant required onboard the OLS on day 79 (at which time a 30-day power failure could occur) would be 3733 pounds of oxygen and 551 pounds of hydrogen (as shown in Table 3.5-8). Assuming 10 percent tank ullage and four hydrogen tanks, the required volume of each tank is found to be 82 cubic feet. Two of these size tanks (for redundancy) meet the oxygen storage requirements by a wide margin. The boiloff rate for oxygen is found to be 8.28 pounds per day, while normal consumption (using the values in Table 3.5-4) is 28.84 pounds per day. Likewise, the hydrogen boiloff rate is found to be 6.34 pounds per day and normal consumption is 9.14 pounds per day. Assuming that the tugs supply the additional propellant over the normal usage rate for 107 out of 109 days, the resupply requirements may be reduced by 2200 pounds of oxygen and 300 pounds of hydrogen.

Table 3.5-8. Propellants Required on Day 79 of a 109-Day Resupply Period

Item	O_2 (lbm)	H_2 (lbm)
Emergency power (30 days)	3697	533
2 days of no tugs	58	18
Crew overlap O_2	18	-
Totals	3773	551

The RCS would have a design similar to that used for the baseline OLS, but with only four (82 cubic foot) rather than eight (64 cubic foot) hydrogen tanks. It would be necessary to add "cold accumulators" to store a one-day normal usage of tug boiloff, because the RCS usage is intermittent. The net result is that the total subsystem loaded mass (assuming the excess cryogenic capacity is not used in each case) is only slightly less.

A separate approach would be to add tug boiloff gas to the baseline OLS concept without resizing the storage tanks. In this case, no tug hydrogen can be utilized because the total daily requirement is met by OLS boiloff. The oxygen resupply, however, may be reduced by 2330 pounds per 109 days. In this case, the oxygen boiloff rate would be 7.04 pounds per day. Because of the nonsteady operation of the RCS, it would be necessary to add cold oxygen gas accumulators. The subsystem dry mass increase for this modification would be about 30 pounds.

3.5.3 ADAPTABILITY TO ARTIFICIAL GRAVITY OPERATION

In an artificial gravity mode, the RCS will be responsible for three additional functions, spinup, despin, and momentum vector control. Considering the projected artificial gravity configuration, it would be desirable to locate the main RCS in the nonrotating hub. The hub would be designed with sufficient length and diameter to provide room for the RCS components and suitable operating moment arms for the engines.

The spinup and despin operations would require separate units, however. One approach would be to use packaged earth storable nitrogen tetroxide and monomethyl hydrazine thrust units. These would be mounted externally on the outer radii of the OLS rotating arms. The toxicity and corrosivity problems of these chemicals previously mentioned (Volume IV of this report) are not as pertinent here because these units would be mounted completely external to the habitable volumes of the station, and they would not be required to be maintainable. They would be sufficiently redundant to ensure a safe despin on demand. Further, they would be used for only one or perhaps two spinup

and despin cycles and then replaced. The replacement operations could be performed using a tug. The plume contamination problems, also mentioned previously, would not be a problem for two reasons: first, the units would rarely be used; second, when in use the engines would be burning steadily. Steady burning produce a much cleaner exhaust than the short burst or pulse operations usually used in RCS operations.

Because the nitrogen tetroxide and monomethyl hydrazine concept is well developed and the main RCS concept is the same as that discussed for the zero gravity OLS, no particular difficulty is anticipated in accommodating the artificial gravity OLS requirements.

Page Intentionally Left Blank

3.6 ENVIRONMENTAL PROTECTION SUBSYSTEM

The environmental protection subsystem (ENPS) of the representative OLS provides protection to the crew and critical subsystems equipment from natural environmental hazards. These environmental hazards include micrometeoroids, thermal radiation, and nuclear radiation. The characteristics of these environments and their effects on system design are described and discussed in Volume III, OLS Performance Requirements Report, and Volume IV, Configuration and Systems Analysis Report.

3.6.1 METEOROID PROTECTION

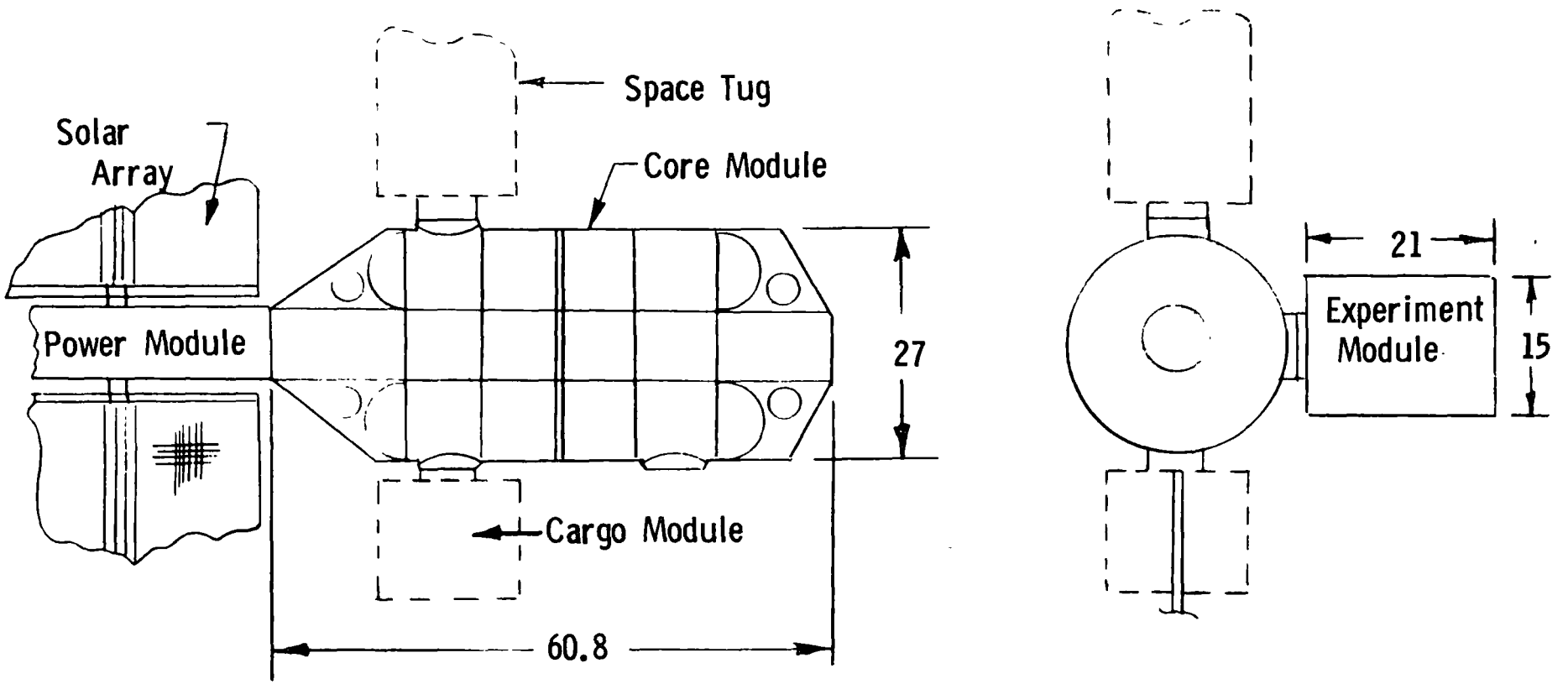
Concept Definition

The basic design requirements for the meteoroid protection concept for the OLS are:

1. Insure a 10-year operational life
2. Provide protection in the meteoroid environment of a 60-n mi lunar orbit altitude
3. Ensure a probability of no puncture of 0.9, of the pressure wall, tanks, and plumbing over the 10-year period

Parametric data for the meteoroid environment and shielding concepts are presented in Section 3.6.2 of Volume IV. It is pointed out in that analysis that the three most significant parameters affecting concept selection are the probability of no failure (P_{nf}), the pressure wall thickness (t_r), and the area-time product (AT). The probability of no failure (P_{nf}) is specified and t_r is determined based on structural considerations.

Figure 3.6-1 shows the representative OLS configuration. The modules that require meteoroid protection are the core module, power module, and experiment module. The power module requires protection because it is pressurizable and must be capable of shirtsleeve entry for maintenance and repair of EPS equipment. The experiment module may or may not remain attached to the OLS for the full 10 years, but meteoroid protection equivalent to that provided for the core and power module is provided. It is assumed that the space tugs are provided with their own meteoroid protection and that the dual-support cargo modules need not be provided with equivalent protection since they are expended and replaced every 109-day resupply period.



Core Module	Power Module	Experiment Module	Surface Area Requiring Shielding
500 m^2	80 m^2	100 m^2	680 m^2

$AT = 2.14 \times 10^{11}$ m^2 sec

$m_p = 0.0096$ grams

Figure 3.6-1. Representative OLS Reference Configuration

The total surface area of the modules that require meteoroid protection is found to be about 680 m^2. For a 10-year operational life the AT product is then 2.14×10^{11} m^2-sec. From the meteoroid flux-mass model presented in Section 3.6.2 of Volume IV, the particle mass that the OLS meteoroid protection must be capable of stopping without penetration of the pressure wall can be found. The design particle (m_p) is 0.0096 grams. The meteoroid protection is designed to allow penetration of the pressure wall by the design particle to within about 0.555 t_r. All particles less than 0.0096 grams striking the OLS will make a penetration $\leq$ 0.555 t_r.

Figure 3.6-30 of Section 3.6.2 of Volume IV illustrates the two meteoroid protection concepts (single bumper and double bumper) and gives the meteoroid protection weight (lb/ft^2) as a function of AT with t_r as a parameter. The curves show that for the OLS area-time product, there is no weight advantage for a double bumper concept. In addition to the weight saving, the simplicity of the single bumper concept makes it desirable. Figure 3.6-29 of Section 3.6.2 of Volume IV shows the bumper thickness (t_1) versus AT with P_{nf} as a parameter for the two bumper materials (aluminum and glass epoxy). For AT = 2.14×10^{11} m^2-sec and P_{nf} = 0.9, the minimum bumper thicknesses for glass epoxy and aluminum are found to be 0.017 inch and 0.009 inch, respectively.

Selection Rationale

Since the representative OLS uses phased arrays, any array meteoroid protection material must be non-conducting, so as not to interfere with external communications. Hence, glass epoxy is selected over aluminum for phased array meteoroid protection. Aluminum shielding (t_s) to thicken the pressure wall is not required because the design particle (m_p) is small. The single bumper and insulation shown in Figure 3.6-30 of Volume IV provides the required protection. It is recommended that the meteoroid protection concept consist of approximately 0.020 inch aluminum bumpers except for 0.020 inch glass epoxy on the phased arrays. The bumper thickness is chosen above the minimum requirement to facilitate ease of manufacturing, handling and maintenance and to reduce the support structure points and heat shorts.

Artificial-G Adaptability

An artificial-g mode of operation for the OLS will have no effect upon the meteoroid protection concept.

3.6.2 THERMAL PROTECTION

One of the most unique aspects of an OLS, as compared to an Earth Orbit Space Station, is the thermal operating environment. In the following paragraphs a comparison of the earth orbit environment to the lunar orbit environment is made. Alternate concepts are discussed and their limitations as applied to the OLS identified.

Heat Rejection Requirements

Heat rejection requirements are established by subsystem electrical power requirements, electrical power generation efficiencies, electrical power conditioning and distribution losses, the crew metabolic load, and the energy exchange between the external environment and the vehicle volume internal to the thermal protection scheme (insulation and/or heat rejection control assembly). Electrical power for sunlit portions of the orbit is generated by solar cells to satisfy OLS subsystem requirements and to electrolyze water for the regenerative fuel cells, which are the power source in non-sunlit portions of the OLS orbit. The rejection requirements are listed in Table 3.6-1.

Table 3.6-1. Heat Rejection Requirements

	Light Side (Solar Cell)		Dark Side (Fuel Cell)	
	Core	Boom	Core	Boom
Subsystems	24.0 kw	0 kw	21.0 kw	0 kw
EPS Efficiency				
Distribution	1.5	4.5	1.3	4.0
Fuel cell	0	0	0	17.5
Electrolysis	0	1.3	0	0
Metabolic	1.1	0	1.1	0
Total	26.6 kw	5.8 kw	23.4 kw	21.5 kw
	(90,000 Btu/hr)	(19,800 Btu/hr)	(80,000 Btu/hr)	(73,500 Btu/hr)

The external environmental heat loads to the internal volume were assumed to be zero for this evaluation because their actual magnitude is small by comparison to the OLS generated loads.

Thermal Environment

The lunar thermal environment is considerably different from the earth thermal environment. Figure 3.6-2 compares the components which comprise the total. Albedo flux, solar energy reflected from a planet, is very high in earth orbit due to the atmosphere, clouds, and large bodies of water. (The spectral distribution of energy in the albedo flux is assumed equivalent to the distribution in the incident solar energy.) Energy absorbed by the earth (not reflected) is emitted in considerably longer wavelength (infrared). Because the earth spins comparatively rapidly and has an atmosphere which aids in distributing the absorbed heat, the emitted infrared flux is almost uniform around the planet for low inclination earth orbits.

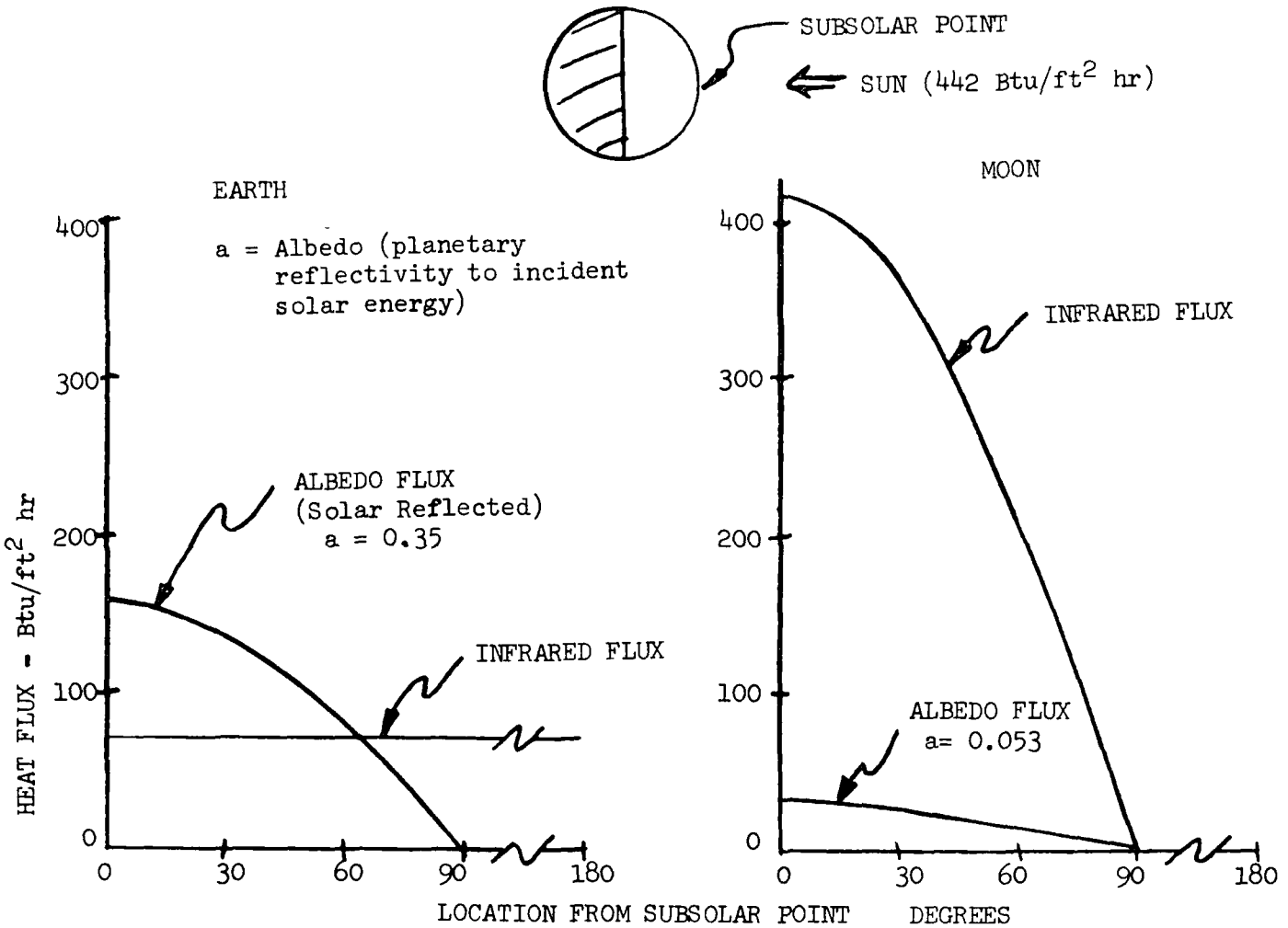


Figure 3.6-2. Planetary Thermal Environments

Space Division
North American Rockwell

The moon has no atmosphere and the absorptivity of its surface to solar energy is very high. Accordingly, the emitted flux (infrared) is very high and reaches a local equilibrium condition such that it equals the solar energy absorbed at different locations on the lunar surface because its spin rate is slow. The lunar surface temperature at the subsolar point, for example, reaches 250 F.

Radiator Location

Heat rejection is essentially the difference between the energy a surface emits at the desired temperature and that which it absorbs. The integrated rejection capability must, therefore, be greater than or equal to the heat generation to preclude overheating the vehicle.

The considerations described in Thermal Environment are very significant in selecting surfaces on the OLS that are capable of heat rejection. To facilitate discussion of this selection, an artifice called radiant sink temperature, developed in Section 3.6.1 of Volume IV, will be used to define total planetary thermal environments. Significant in this definition are two surface property characteristics; solar absorptivity (α_s) and emissivity (ϵ). The solar absorptivity is the fraction of incident solar energy (direct or albedo) which is absorbed by a surface. The emissivity is the ratio of the energy the surface can emit at the desired surface temperature to the energy a black body (perfect emitter) can emit at the same temperature. In addition, the emissivity also identifies the fraction of energy the surface (the radiator) absorbs from incident energy (infrared) which was emitted by a body (the moon) at approximately the same temperature as the surface of interest.

Figure 3.6-3 demonstrates the thermal environment for a cylinder in a subsolar lunar orbit with the cylinder axis of revolution perpendicular to the orbit plane. This attitude was selected for the OLS to satisfy non-thermal requirements. Locations around the cylinder are described by the angular position, γ , measured from the nadir location. Locations in lunar orbit are described by the angular position, θ, measured from the subsolar point. A comparison of data for α_s/ϵ ratios of 0.1 and 0.2, ϵ = 0.9, illustrates that the α_s/ϵ control effect is limited to those locations in direct sunlight (γ = 180 near the subsolar point and γ = 270 near the lunar terminator).

Figure 3.6-4 shows the net heat rejected from a 70 F perfect emitter (ϵ = 1.0) at the subsolar point as a function of location on the cylinder. For high α_s/ϵ , rejection is possible from sunward locations only ($90^\circ \leq \gamma \leq 180^\circ$). External radiators should be located to take advantage of available area that receives only small amounts of lunar emission. Such an OLS configuration is depicted in Figure 3.6-5. The two 60-degree radiator sections are useful for rejection at locations near either lunar terminator.

Figure 3.6-3. Lunar Thermal Environment in a Subsolar Orbit

Space Division
North American Rockwell

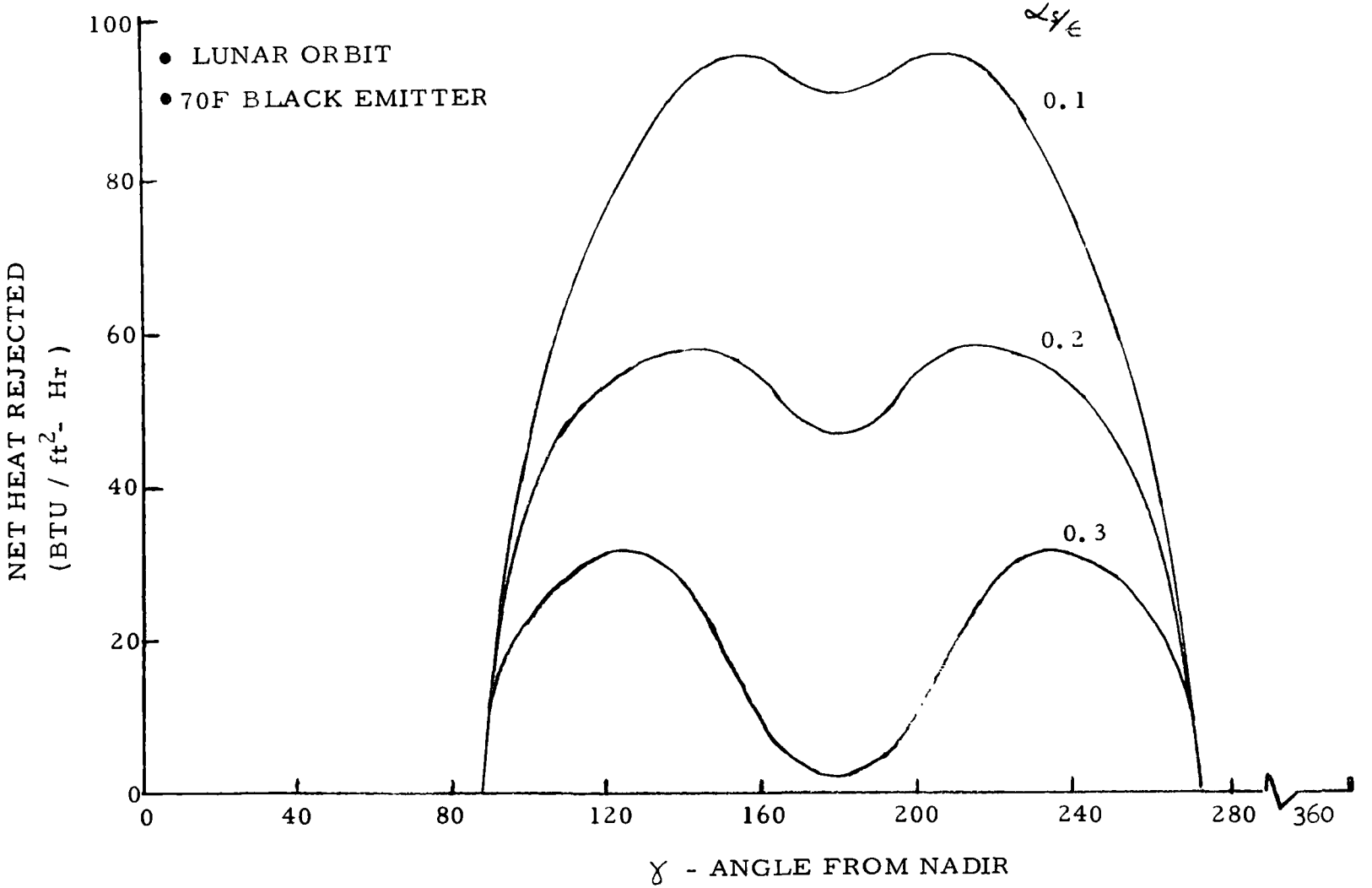


Figure 3.6-4. Heat Rejection at Subsolar Point

Space Division
North American Rockwell

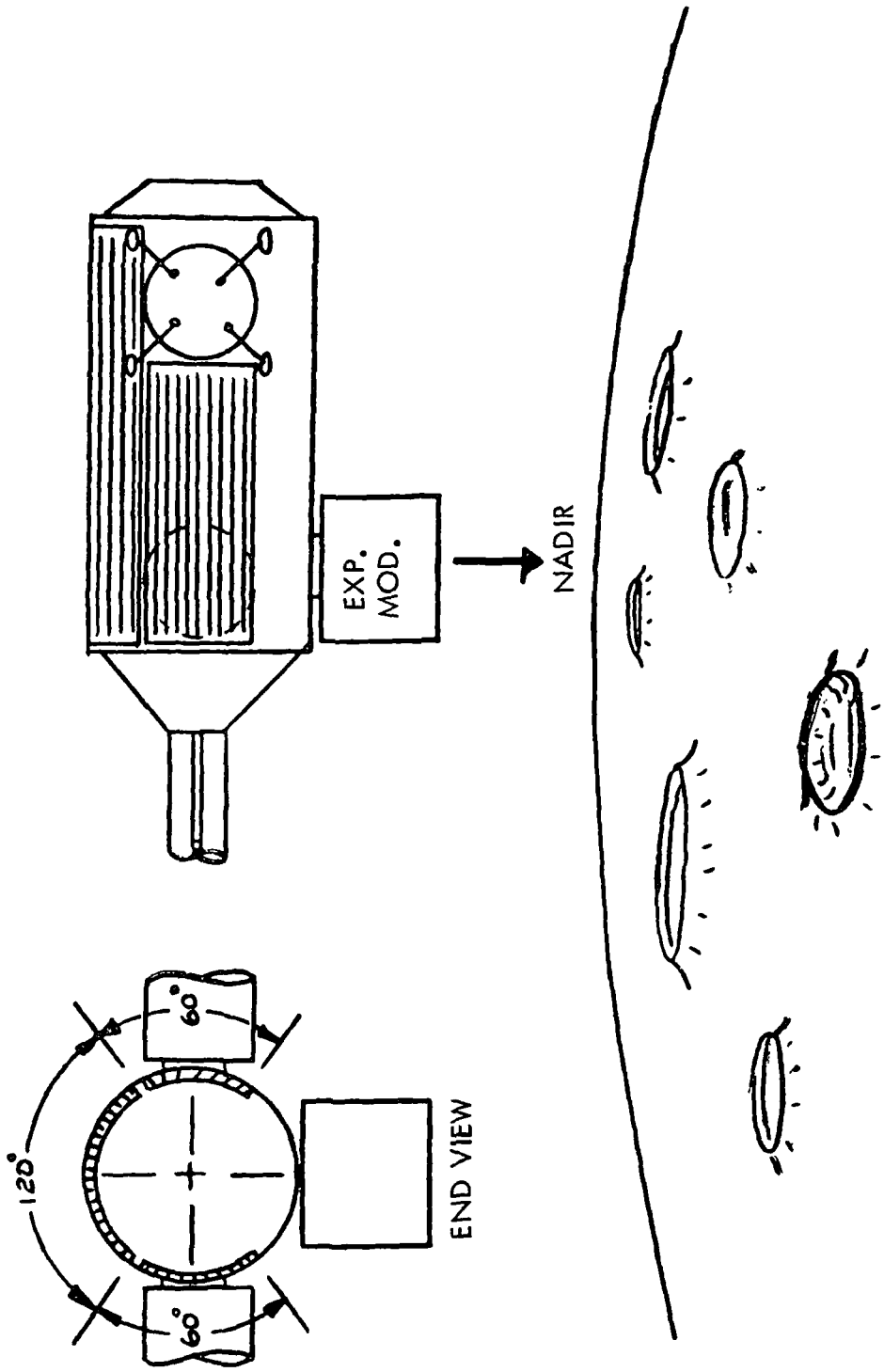


Figure 3.6-5. OLS External Radiator Location

Thermal Control System Alternatives

OLS thermal control system alternatives include combinations of an active external radiator system, a passive double louver system, and refrigeration subassembly for humidity control.

Active External Radiator Without Refrigeration Unit

The active external radiator (Figure 3.6-6) operates on an active fluid loop with radiator fluid tubes attached to the micrometeoroid bumper. Super-insulation is installed between the micrometeoroid bumper and the pressure wall to thermally isolate the active elements from the vehicle. Internal heat transport is also accomplished with active fluid loops, which circulate through coldplates to which heat generating components are mounted. Heat is transported from the internal loop to the radiator loop through a heat exchanger. Humidity control is accomplished by providing a cold surface (approximately 40 F) in the vehicle across which the cabin air is circulated. Moisture is removed by condensation onto this surface in the cabin air heat exchanger. The cold surface was considered to be provided in either of two ways. The first required obtaining cold fluid from the radiator ($\leq$ 35 F) to provide a sufficiently low internal fluid loop temperature to the cabin air heat exchanger. The effective radiator temperature for this approach is approximately 50 F.

Active External Radiator With Refrigeration Unit

The other humidity control technique with the active external radiator relied upon a refrigeration unit to provide the cold surface in the cabin air heat exchanger. This resulted in a cold fluid radiator temperature requirement of approximately 60 F, enabling an increase in the effective radiator temperature to about 70F. The removed heat is assumed rejected from the vehicle at 80 F resulting in a refrigerator coefficient of performance of approximately six. An attendant power penalty (and heat rejection requirement increase) of 0.35 kw (1200 Btu/hr) is experienced.

Double Louver System With Refrigeration Unit

The louver system alternative consists of two louver platforms (Figure 3.6-6) between the pressure wall and the micrometeoroid bumper; one controlled by the pressure wall temperature and the other by the micrometeoroid bumper temperature. The louver blades on both platforms have low emittance surfaces on both sides to reduce radiation heat transfer between the pressure wall and micrometeoroid bumper when required. Bimetallic strips (thermostats) provide the control such that the pressure wall assembly is fully open at a temperature above approximately 70 F, allowing transfer of the thermal radiation emanating from the pressure wall surface ($\epsilon \geq 0.9$). The micrometeoroid bumper activated louver assembly is designed to open at a temperature below 70 F allowing the thermal radiation to reach the micrometeoroid bumper where it is absorbed ($\epsilon \geq 0.9$). The primary purpose of the double louver assembly is to compensate for the case when the local radiant sink temperature (which will drive local micrometeoroid bumper temperatures) exceeds the desirable pressure wall temperature, which would result in heat gain by a hot pressure wall

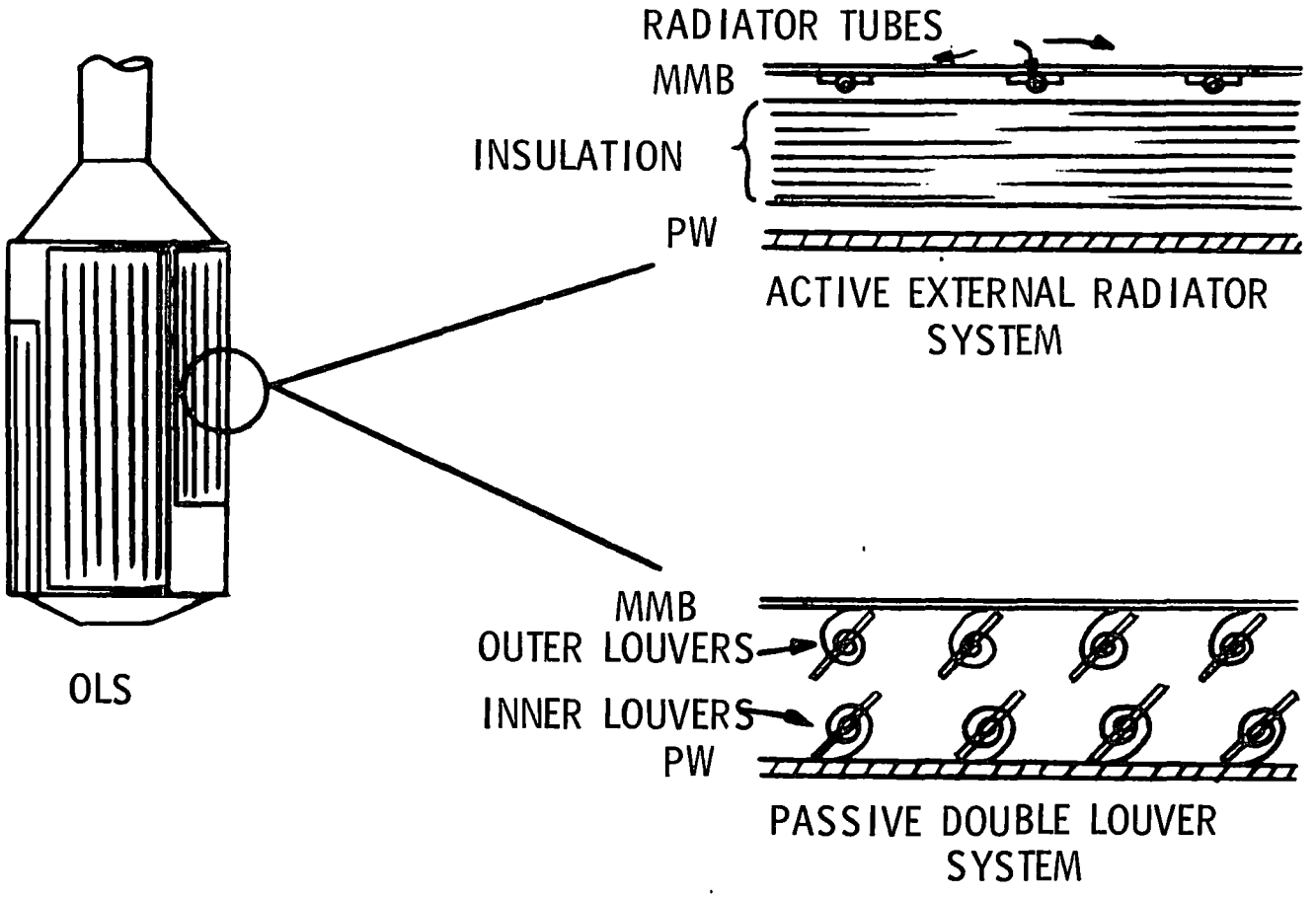


Figure 3.6-6. OLS Thermal Control System Alternatives

Space Division
North American Rockwell

(pressure wall louvers open) from a hotter bumper. The hot micrometeoroid bumper activates the bimetallic strips to close the louvers, significantly reducing the heat transferred from the bumper to the pressure wall. This double control technique allows net heat transfer from the pressure wall to the bumper, but not the reverse. Should internal heat generation be less than the heat rejection capability the vehicle will cool. The control system precludes over-cooling by closing the pressure wall controlled assembly. Because the pressure wall is now the radiator, it is necessary to maintain pressure wall temperatures above the cabin air low point of 57 F. Therefore, a refrigeration unit for providing cold fluid for humidity control is required for this concept.

Lower heat rejection rates per unit area are characteristic of the double louver system. However, their advantages include passive control techniques, very low information subsystem involvement, high reliability, simplified maintenance/repair, and resistance to failure propogation. In fact, a negligible decrease in thermal control capability would result from the loss of a louver blade due either to random failure or micrometeoroid damage. Accordingly, louver blade replacement would probably not be required. If replacement is required, it could be accomplished relatively easily for the louver controlled concept. Only mechanical disassembly/assembly operations are required. Replacement of an active external radiator would be more difficult, requiring decoupling fluid transport lines and recoupling them again.

The double louver system also eliminates the need for the heat exchanger used between the external fluid loops.

Non-Radiator Areas

In non-radiator areas, insulation will be used in the cavity between the micrometeoroid bumper and the pressure wall. It will consist of a multi-layered assembly of a metalized polymeric film; typically, forty layers of aluminized mylar. Insulation would be designed to vent during boost and in space. It requires interstitial pressures of less than 10^{-4} torr to function effectively.

Docking port covers will be required to enable sufficient thermal control when no modules are docked. They will limit heat gain and loss and preclude the occurance of condensation.

Window covers should be provided to eliminate high heat loads which could be transmitted by sunshine through uncovered windows.

Thermal Capacitance Effects to Heat Rejection

The result of insufficient heat rejection in an active fluid loop radiator is that the fluid exiting the radiator will be too warm. This evenutally causes the internal heat transport loop fluid exiting the heat exchanger to be too warm, thereby causing over-heating of the components on the internal loop. Thermal capacitance can be used to eliminate or minimize the temperature rise.

Solid to liquid (e.g., ice to water) phase change materials exist for and can be used at the temperature of interest by placing such an assembly in the internal heat transport loop downstream of the heat exchanger exit. At locations in orbit of insufficient rejection, the fluid, which is too warm, would be cooled by the phase change material which would undergo melting with effectively no change in temperature. The energy which could not be rejected has, therefore, been transferred to and stored in the phase change assembly. When regions of insufficient rejection have been passed, and the radiator can now reject more than the heat load being simultaneously generated, the radiator fluid exit temperature is low resulting in a lower internal loop fluid exit temperature from the heat exchanger. Therefore, the heat which was stored is now given up to the fluid loop for rejection. The portion of the phase change material which had melted is refrozen to repeat the cycle during the next orbit.

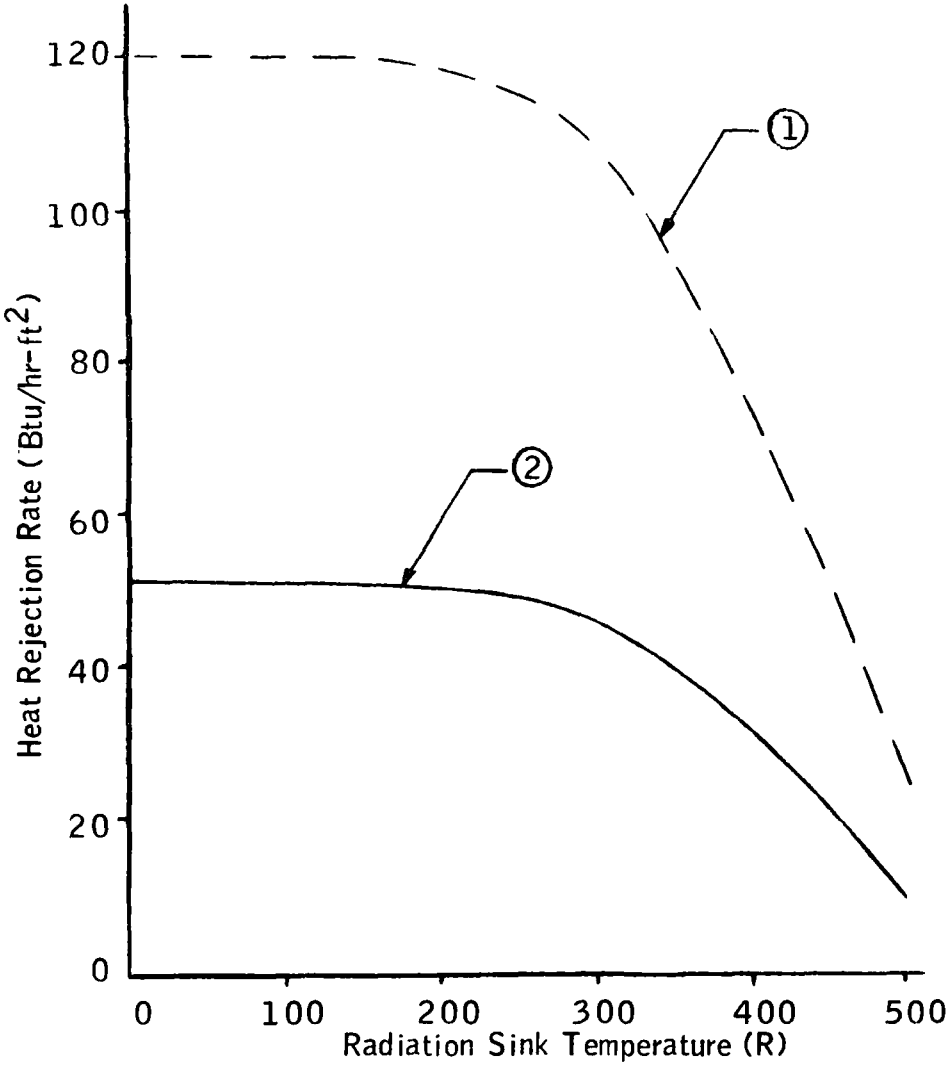
Vehicle thermal capacitance can be used in place of a phase change material (if a refrigeration unit is used to provide humidity control) by plumbing the vehicle structure into the internal heat transport loop. This would result in a slight temperature cycling of the overall vehicle every orbit.

A third technique would rely on a fluid reservoir for the internal transport loop in place of the phase change assembly. With a sufficiently large reservoir, the too warm fluid exiting the heat exchanger is mixed into and replaced by the cooler stored fluid. Cooling the reservoir is accomplished in a manner similar to that with the phase change assembly.

A similar effect can be achieved by circulating fluid through tubes in a stirred reservoir and obtain almost as effective an energy transfer as that described above.

Heat Rejection Capability

The heat rejection capabilities at 70 F for the external radiator and the double louver system are compared in Figure 3.2-7. The double louver system rejects at a lower rate since the micrometeoroid bumper acts as a radiation barrier to reduce heat transfer. With the radiator configuration outlined previously in Figure 3.6-5, 2000 ft^2 of radiator area are available (1300 ft^2 in the 120° antilunar bank and 350 ft^2 in each side bank). Based on radiator area, surface property, and location, net heat rejection can be determined as a function of position in orbit. Figures 3.6-8 a, b, and c show a comparison of heat rejection performance characteristics. The rejection requirement is the dashed line. Higher values of α_s/ϵ result in lower



PW TEMP. = 70 F

1 External radiator $\epsilon = 0.9$ Radiator operating at PW temp

2 Double-louver system blades open

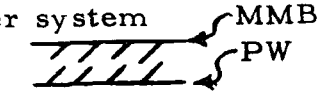


Figure 3.6-7. Heat Rejection vs. Radiation Sink Temperature

Space Division
North American Rockwell

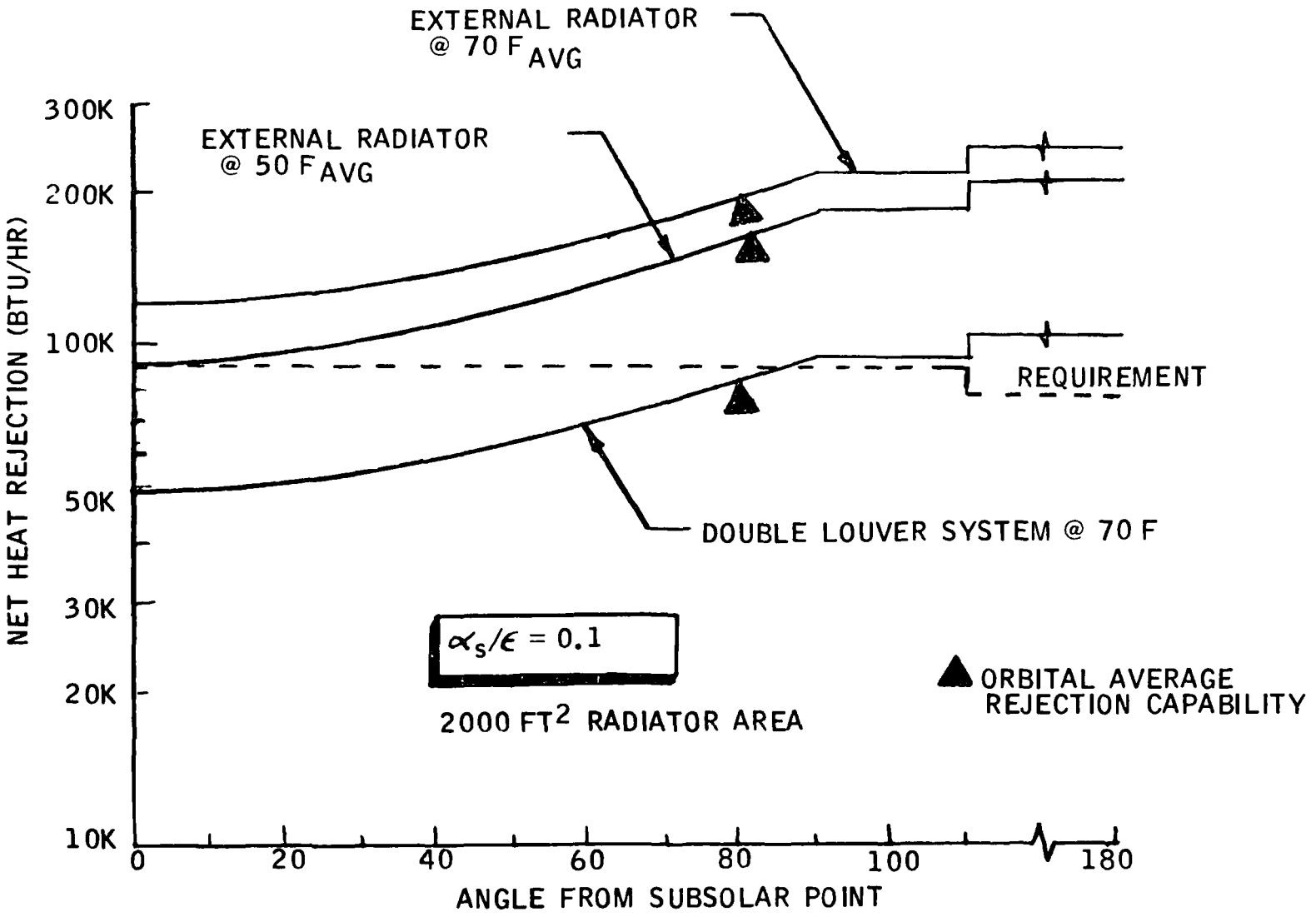


Figure 3.6-8a. Comparison of Heat Rejection System Performance Characteristics - Lunar Orbit

Space Division
North American Rockwell

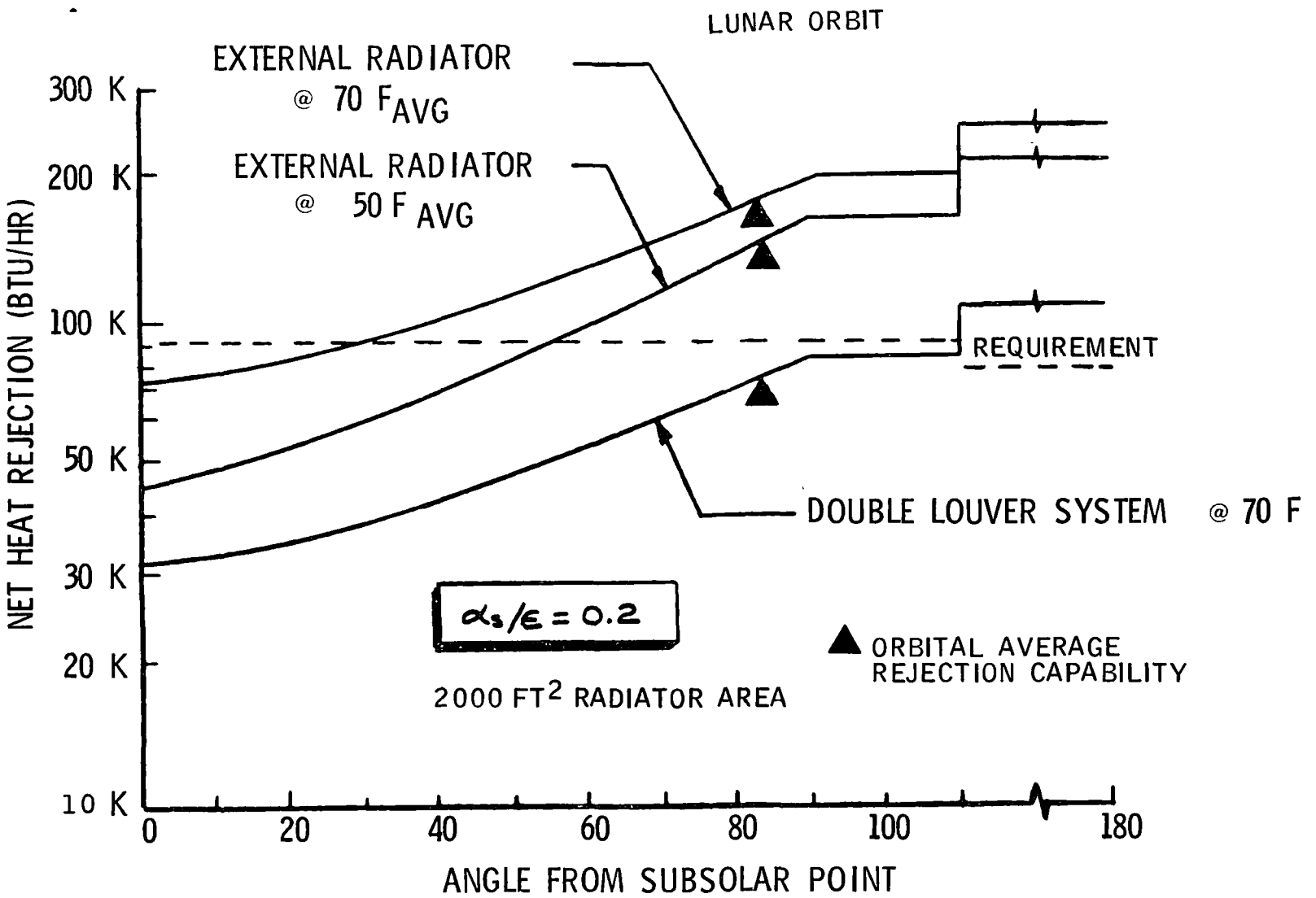


Figure 3.6-8b. Comparison of Heat Rejection System Performance Characteristics

Space Division
North American Rockwell

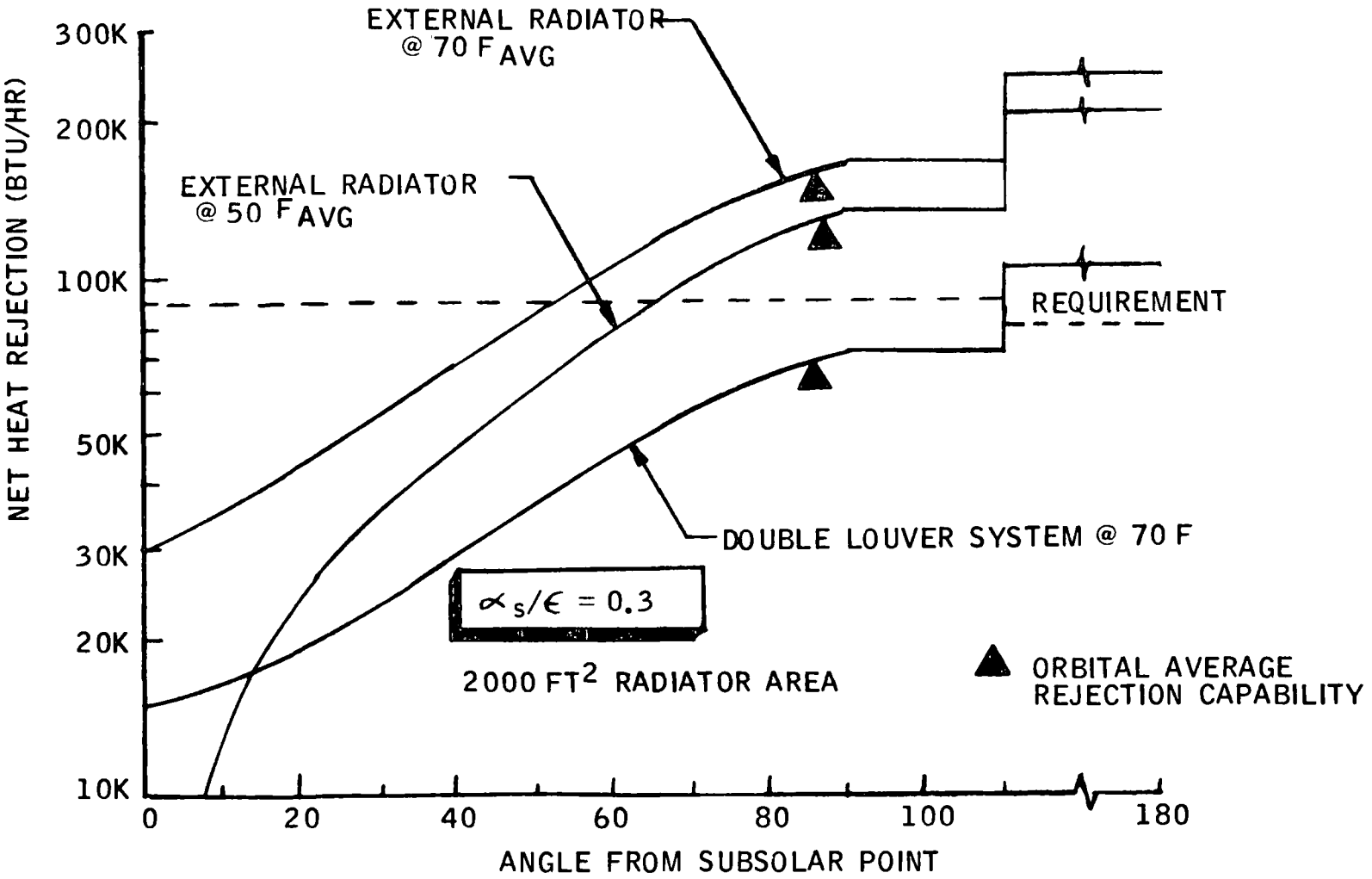


Figure 3.6-8c. Comparison of Heat Rejection System Performance Characteristics - Lunar Orbit

Space Division
North American Rockwell

rejection rates over the sunlit portion of the orbit. The systems compared include a passive double louver system operating with a pressure wall temperature of 70 F, and active external radiators with average temperatures of 50 F and 70 F. A refrigerator is required for humidity control with both 70 F concepts.

As expected, the active systems are capable of rejecting more heat than the louver system. But the curves also indicate that none of the concepts can meet the instantaneous heat rejection requirements throughout the orbit with only 2000 square feet of radiator area. All three require either more radiator area or a heat sink. (It should be noted that the curves for the louver system are of an academic nature because the system inherently includes at least the vehicle pressure wall as a heat sink.)

In order to determine the radiator area needed to meet the required rejection at the subsolar point, the following concept of radiator configurations was established. Radiator areas below 1300 ft^2 utilize a length of a 120-degree sector located antilunar. Area requirements between 1300 ft^2 and 2000 ft^2 use additional lengths of 60-degree sectors on either side of the vehicle as shown in Figures 3.6-5 and 3.6-6. Areas above 2000 ft^2 are obtained by increasing the vehicle length including all three sectors (240 degrees total).

Radiator area, based upon the preceding design concept, is plotted in Figure 3.6-9 as a function of α_s/ϵ . For the radiator concepts which utilize no thermal capacitance (the upper curves) note that the 70 F external active radiator requires approximately half the area required by the 50 F radiator. For example, with 2000 ft^2, the 70 F active external radiator would require an α_s/ϵ of approximately 0.18. The 50 F radiator would require an area of almost 3700 ft^2 to accomplish the same rejection with the same surface temperature control coating. Louver concept data is plotted hypothetically for comparison purposes only.

Note also from Figure 3.6-9 how rapidly the area required increases with higher values of α_s/ϵ . For example with α_s/ϵ = 0.3 more than three times the available area will be required for rejection. Due to the nature of these coatings, degradation to this level is possible. The coatings, used on Apollo ECS radiators, Z-93, increased from α_s/ϵ = 0.18 when it was applied to α_s/ϵ = 0.2 prior to liftoff and $\alpha_s/\epsilon \cong$ 0.3 during a two-week mission. On space station missions of longer duration, considerably more degradation can be expected. Therefore, radiator concepts with no or even low thermal capacitance are not a good design approach and place unreasonably stringent requirements on a temperature control coating.

It is possible to reduce radiator area and sensitivity to α_s/ϵ by designing the system to take advantage of the vehicle thermal capacitance (27,600 Btu/°F). Heat is stored near the subsolar point, where rejection is not possible with a reasonable size radiator, by allowing a slight increase in the structure temperature. Once in the lunar shadow, the very high rejection capability enables rejection of the stored heat in addition to that generated on the dark side, causing the vehicle temperature to drop to its previous value. The expected fluctuation in vehicle temperature over an

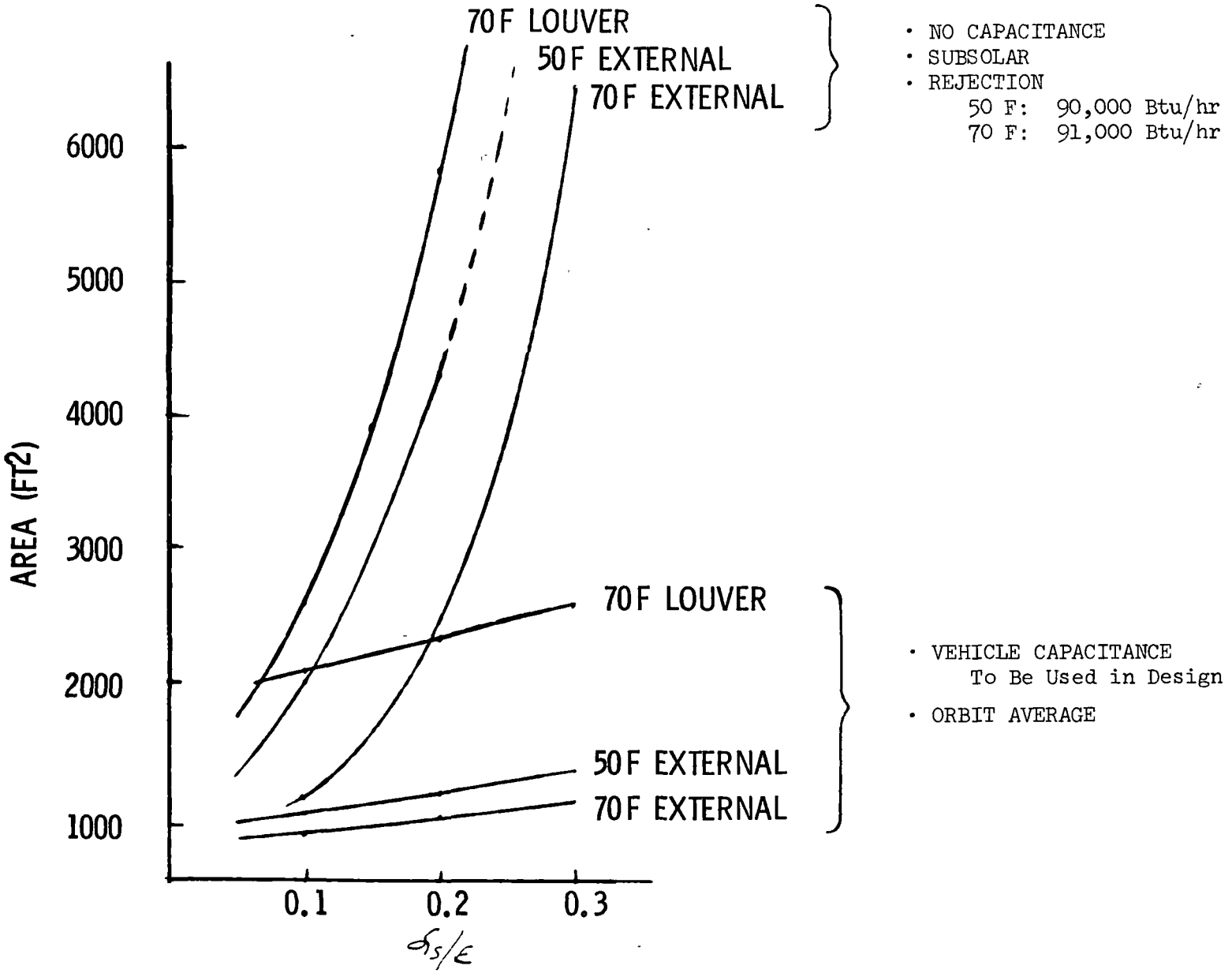


Figure 3.6-9. Required Radiator Area

Space Division
North American Rockwell

orbit is on the order of one degree Fahrenheit. The value of a system designed to utilize vehicle capacitance is demonstrated by the lower set of curves in Figure 3.6-9. The use of thermal capacitance significantly reduces both required area and sensitivity to temperature control coatings.

By nature of its design the passive thermal control system uses the vehicle capacitance. The active system design on the other hand is such that an additional design procedure is required to include thermal capacitance. It was conservatively estimated that fluid loops passing through a phase change material can become a thermal capacitor with an effective capacitance of 25 Btu/lbm at the transition temperature. Figure 3.6-10 shows the weight of such a capacitor required for 2000 ft^2 active radiators at 50 and 70 F as a function of α_s/ϵ ratio. Thus, a 70 F radiator can reject at the subsolar point with α_s/ϵ = 0.18. However, the same radiator with α_s/ϵ degraded to 0.3 requires 900 pounds of thermal capacitor to meet rejection requirements.

The radiator area can be reduced by adding capacitor to the limit where orbit average rejection capability from the area no longer exceeds orbit average rejection requirement. Beyond this point no amount of capacitor addition can facilitate the required rejection.

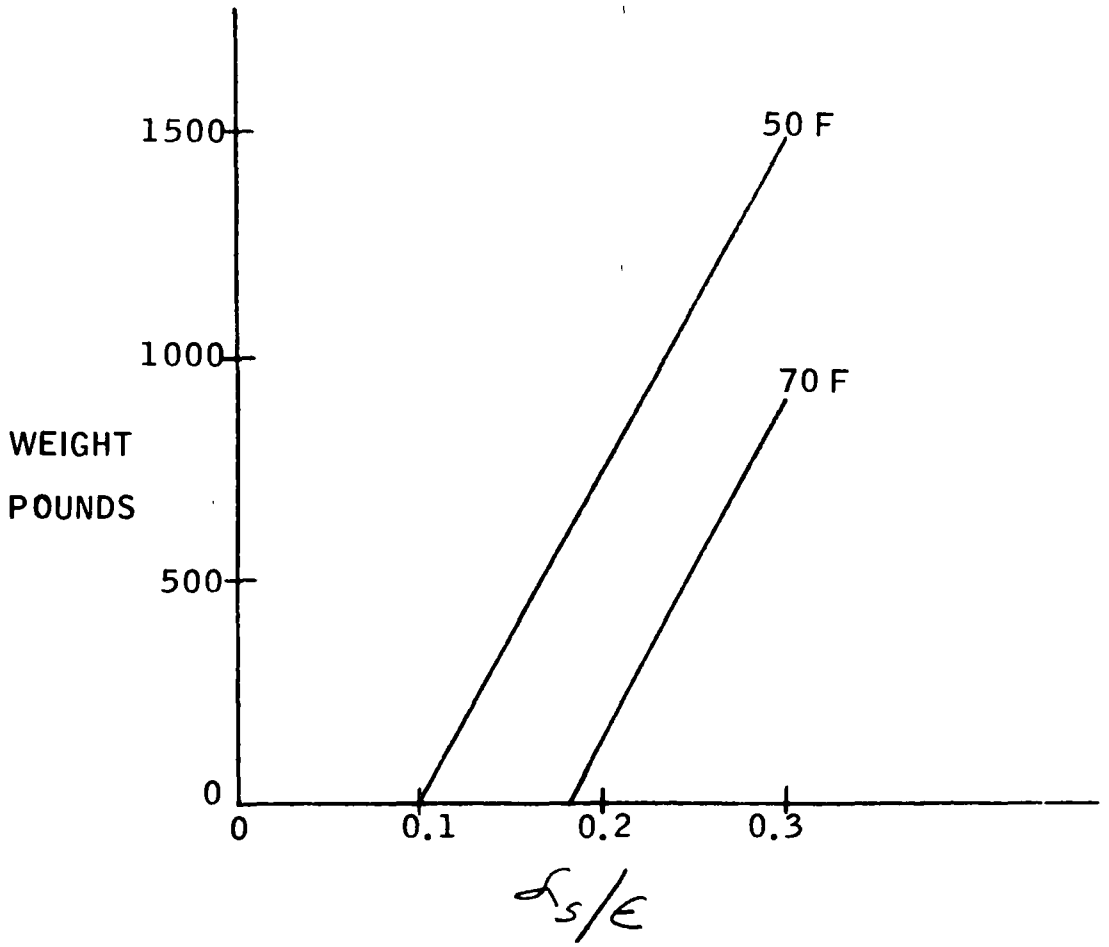
Figure 3.6-11 compares capacitor weight and radiator area necessary to satisfy rejection requirements with an α_s/ϵ degraded to 0.3. The data shows, for example, assuming that an active external radiator weighs one pound per square foot, that the sum of capacitor and radiator weight would be 2900 pounds (2000 pounds for the 2000 ft^2 radiator and 900 pounds for the capacitor). Reducing the radiator area to 1500 ft^2 results in a combined radiator and capacitor weight of 2700 pounds (1500 pounds for the radiator and 1200 pounds for the capacitor). It is apparent that a weight savings of 200 pounds would occur. In fact, from the slope of the curves, trading radiator area for capacitor weight is a weight effective trade down to 1300 ft^2 for both the 50 F and 70 F concepts.

Concept Definition

A summary of the concept characteristics is presented in Table 3.6-2. Briefly, in order to operate either type of radiator system at 70 F, a heat pump must be added to supply the low temperature fluid required for humidity control. One advantage of the 50 F system is that no heat pump is required. Both external radiators require the use of an internal/external heat exchanger while the louver system does not.

Radiator area requirements for any system at a reasonable α_s/ϵ that does not use a thermal capacitor, greatly exceeds that available.

Relying on use of the vehicle thermal capacitance to store energy, which cannot be rejected at the subsolar point, for rejection to cooler environments elsewhere in orbit results in low area requirements for both active concepts. The louver concept requires more area than was considered available. Note the comparative insensitivity to α_s/ϵ which occurs by use of the vehicle thermal capacitance. Providing thermal capacitance for



- EFFECTIVE H_F = 25 BTU/LB
- 2000 FT^2 RADIATOR

Figure 3.6-10. Capacitor Weight for Active Concepts

Space Division
North American Rockwell

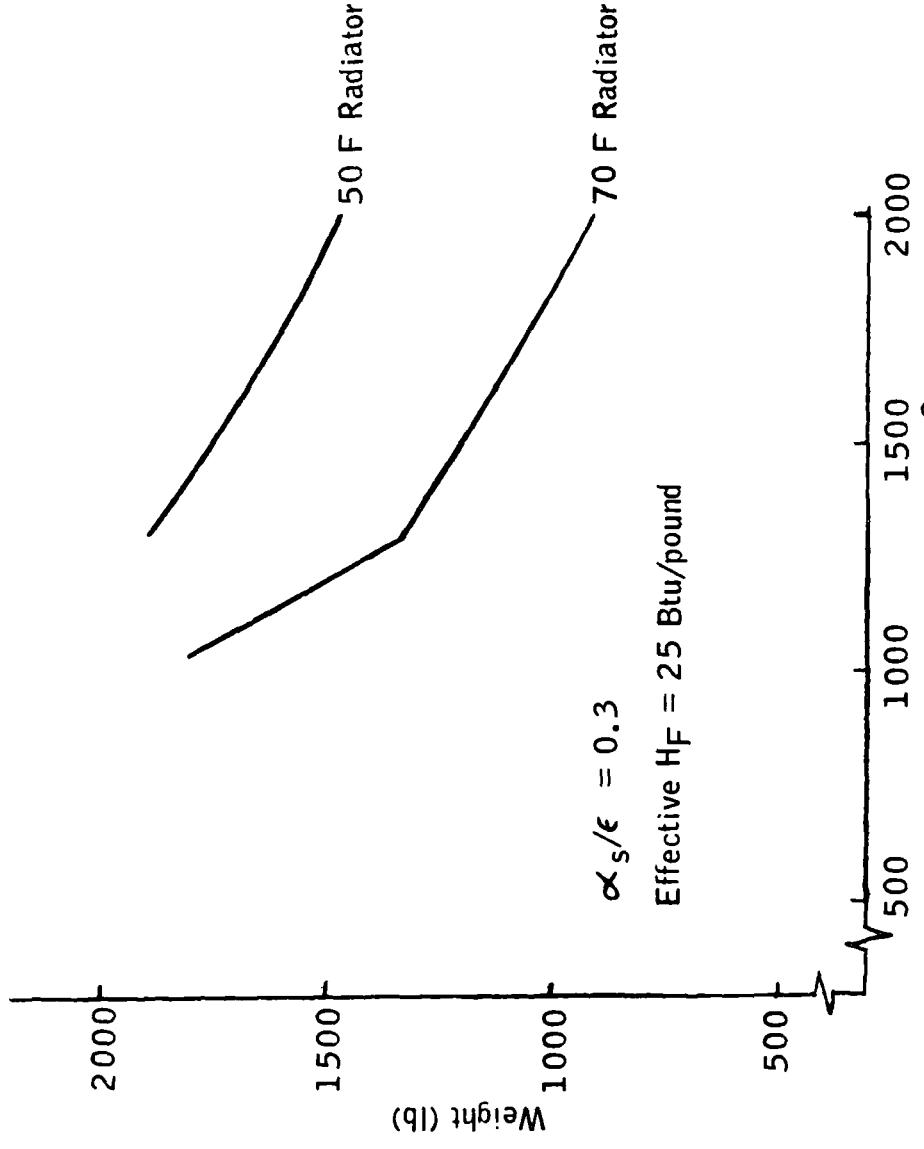


Figure 3.6-11. Capacitor Weight Versus Radiator Area

Table 3.6-2. Concept Characteristics Summary

Control System	Active 50 F	Active 70 F	Louver 70 F
Heat pump	No	Yes	Yes
Heat exchanger	Yes	Yes	No
Radiator area (no vehicle capacitance)			
$\alpha/\epsilon = 0.2$	4300 ft^2	2500 ft^2	Not applicable
$\alpha/\epsilon = 0.3$	Very high	6400 ft^2	Not applicable
Radiator area (with vehicle capacitance) (orbital average)			
$\alpha/\epsilon = 0.2$	1150 ft^2	1000 ft^2	2300 ft^2
$\alpha/\epsilon = 0.3$	1300 ft^2	1100 ft^2	2600 ft^2
Heat sink weight (lb) ($\alpha/\epsilon = 0.3$; area = 1300 ft^2)			
H_F = 25 Btu/pound	1900	1300	Not applicable
Water = 2 F delta T	23,800	16,000	Not applicable
Durability	Nominal	Slightly less	Most
Reliability	Nominal	Slightly less	Most
Maintenance	Nominal	Slightly more	Least
Resupply	Nominal	Nominal	Very low
Development risk	Low	Slightly higher	High
Cost	Low	Slightly higher	High

Selected Concept:

Active 70 F - Heat pump, 1500 ft^2 radiator, 15,000 lb water (radiation shield)

Cost

Weight

Design margin

Flexibility - Inclusion of phase change material

the active systems with a phase change material can be accomplished. With a 1300 ft^2 radiator and α_s/ϵ = 0.3, 1900 pounds of a phase change material would be required for a 50 F system and 1300 pounds would be required for a 70 F system.

Another approach to including thermal capacitance would be to use the storm shelter water. Arbitarily selecting a 2 F rise in water temperature as acceptable, the weights of water required are 23,800 pounds for the 50 F system and 16,000 pounds for the 70 F system. Because the water for the 50 F system would have to be stored at approximately 35 F, it does not lend itself as readily for use in the manner described. Note that 16,000 pounds of water are available (see Section 3.6.3).

Based on the preceding, a thermal control subsystem using a refrigerator for humidity control, storm shelter water for thermal capacitance and an external 70 F radiator is recommended.

Boom Module Thermal Control

Boom module thermal control was separated from the core module thermal control because of the considerably higher temperature allowables. Although an effective radiator temperature of 120 F was used in these analyses, it may have been possible to use an even higher temperature.

The rejection requirements are 19,800 Btu/hr on the light side of the moon and 73,500 Btu/hr on the dark side (Table 3.6-1). Rejection capability is high on the lunar dark side where boom rejection requirements are high due to fuel cell operation. Rejection capability on the lunar dark side is independent of α_s/ϵ variations (ϵ = 0.9, constant). Rejection requirements for the lunar light side result from solar array power conditioning losses and electrolysis unit operation. Rejection capability is lowest at the subsolar point while rejection from surfaces on the lunar side of the boom is impossible. Rejection capability on the sunward side of the boom depends on a low α_s/ϵ value. These various effects are reflected in the radiator area necessary to satisfy rejection requirements as shown in Figure 3.6-12. On the lunar dark side rejection is almost independent of radiator location and the same curve represents full circumferential radiators and 180-degree radiators centered about the -Z axis. At the subsolar point, however, radiator location is important and the two considerations appear as separate curves. At the subsolar point the boom axis lies in the plane of the solar arrays and the view factor from the boom to the array is essentially zero. Hence, the heating to the boom from the hot arrays is negligible.

Figure 3.6-13 shows the length of boom radiator required for rejection. From these figures at α_s/ϵ = 0.3., 425 ft^2 of radiator (21 feet of boom, 360-degree radiator) are required for dark side rejection and 280 ft^2 (28 feet of boom, 180-degree radiator) are required for subsolar rejection. Since the surface area available on the boom for radiators is more than 700 ft^2, it will be no problem to arrange radiators of 425 ft^2 total area of which 280 ft^2 are located on -Z side of the boom.

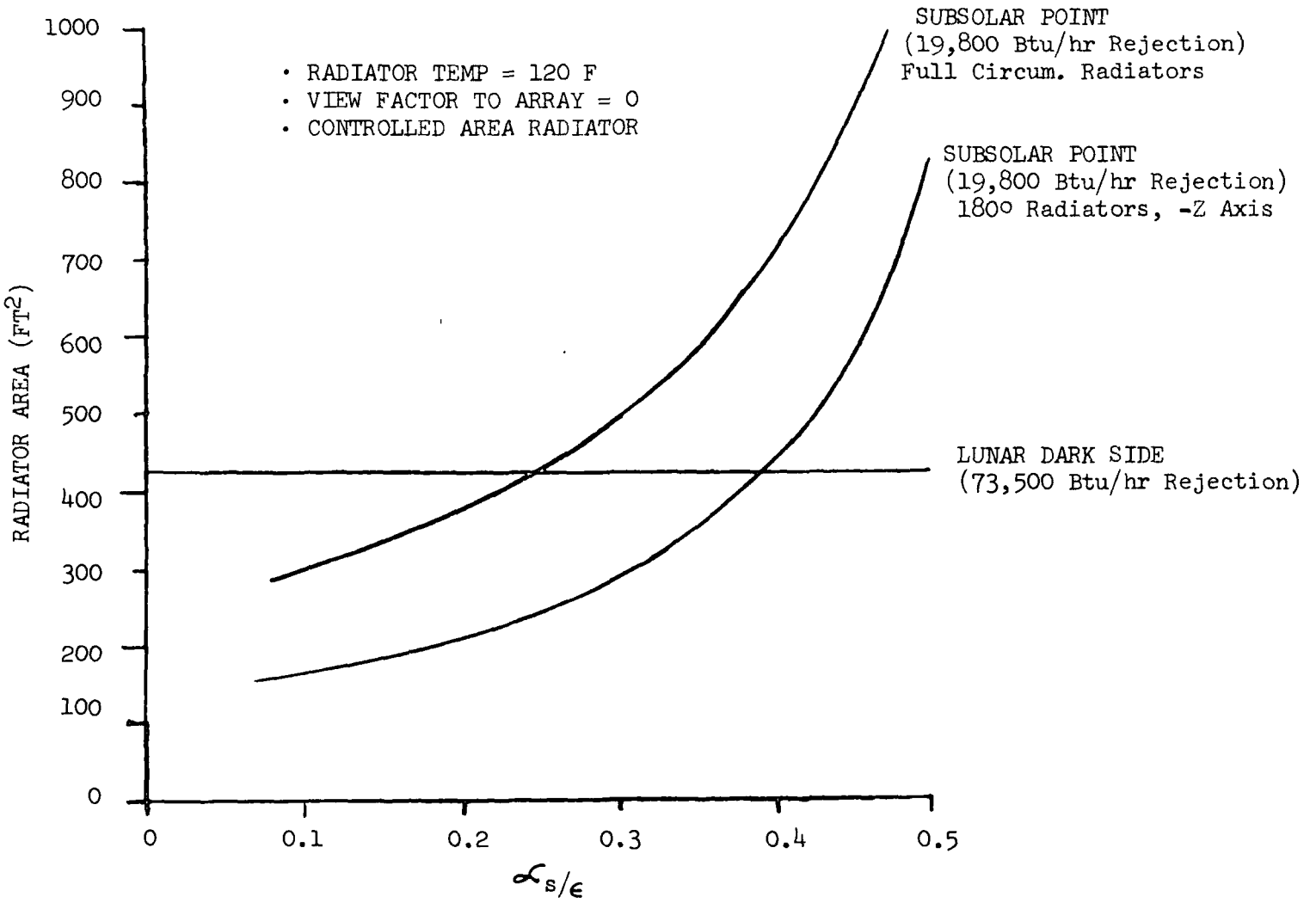


Figure 3.6-12. Required Boom Radiator As A Function of $\alpha_{s/\epsilon}$

Space Division
North American Rockwell

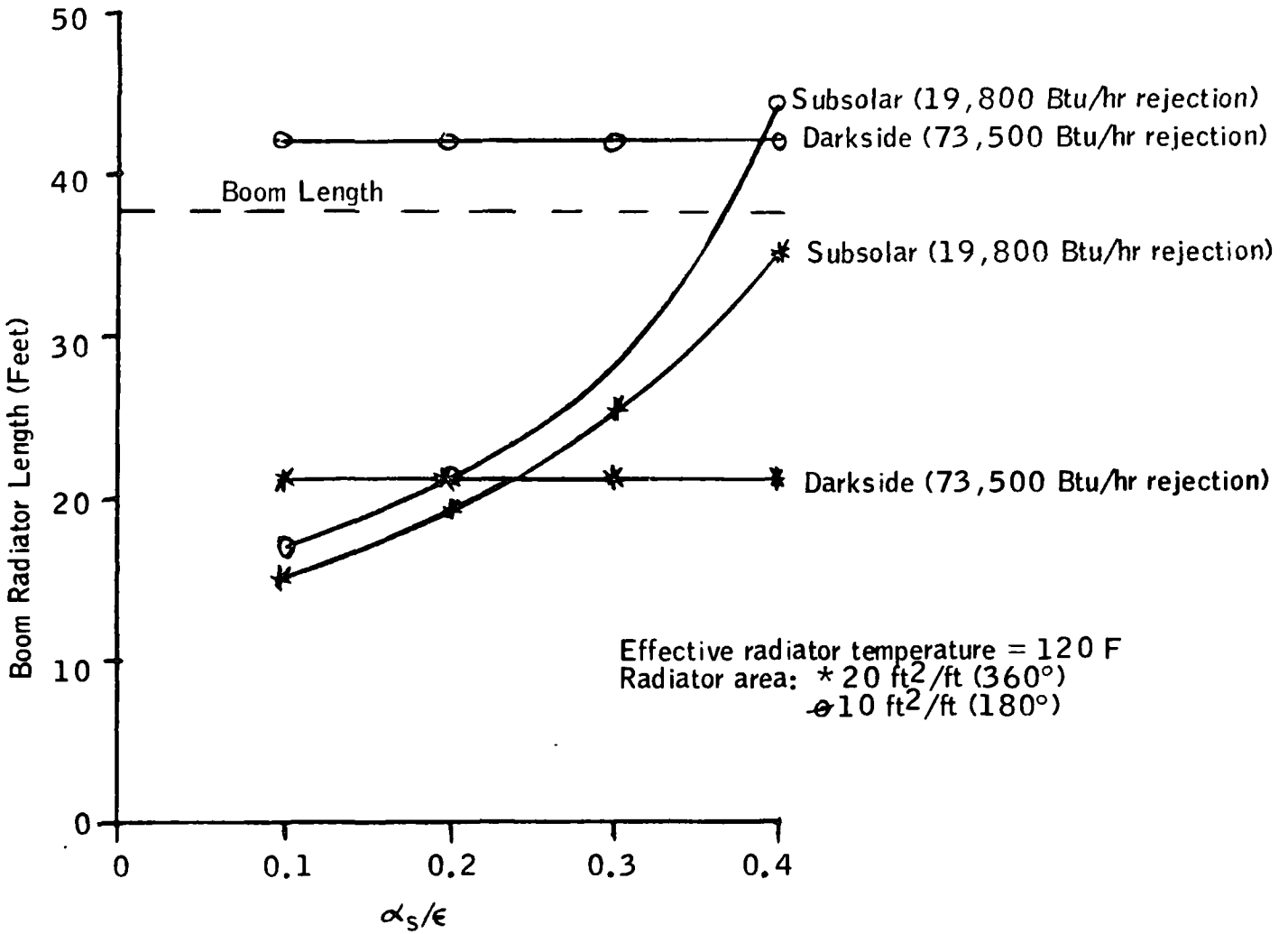


Figure 3.6-13. Required Length of Boom Radiator as a Function of α_s/ϵ

Space Division
North American Rockwell

Adaptability to Artificial Gravity

There is no impact to an active loop heat transport system. However, artificial-G design with heat pipes would be more difficult due to forces acting on the working fluid. Heat rejection becomes a problem near the subsolar point due to vehicle orientation for artificial G. With the sun perpendicular to the spin plane, rejection capability is reduced approximately 50 percent. With the sun in the spin plane, rejection at the subsolar point is almost impossible for either the spin plane in the orbit plane or the spin plane perpendicular to the orbit plane. When not at the subsolar point, the spin plane perpendicular to the orbit plane is the preferable mode. Therefore, the major impact of the artificial-G mode is to reduce rejection capability resulting from unfavorable vehicle orientations.

3.6.3 RADIATION PROTECTION

Concept Definition

Section 3.6.3 of Volume IV presents an analysis of the solar flare "event" model specified by NASA in TMX-53865 and the shielding requirements for limiting the crew radiation dose below the allowable levels to a 2-sigma level of probability. The probability of the OLS witnessing at least one of the NASA solar flare "events" during its 10-year operational life is very close to unity (Figure 3.6-14). Therefore, the OLS must incorporate a solar flare shelter capable of protecting the crew from the effects of at least one solar flare event. For a 234-day OLS crew staytime, it is also determined. that a 2-sigma confidence level of not exceeding the allowable dose limits could be assured by providing enough shielding to protect against a single solar flare "event". A summary of OLS shielding requirement is given in Table 3.6-3. The limiting factor is the dose rate to the blood forming organs of 40 rems per year. The radiation dosage is from three sources and is allocated as follows:

1. Galactic - 5 rems per year
2. Cislunar shuttle (nuclear shuttle) - 10 rems per trip
3. Solar flare - 25 rems per year

Operational considerations were also made in defining the radiation protection concept. From Figure 3.6-15 it can be determined that if a solar flare "event" were to occur, the crew would have to be returned to earth within five weeks before the probability of not seeing another "event" dropped below 95 percent.

The potential duration of a solar storm is three days. Therefore, it is required that the crew be able to have sufficient command and control capability to monitor systems and maintain orbit. Consequently, the solar storm shelter would have to include a control center as well as a galley and hygiene facility in order to permit the crew to remain within the confines of the shelter while still maintaining the minimum required operations.

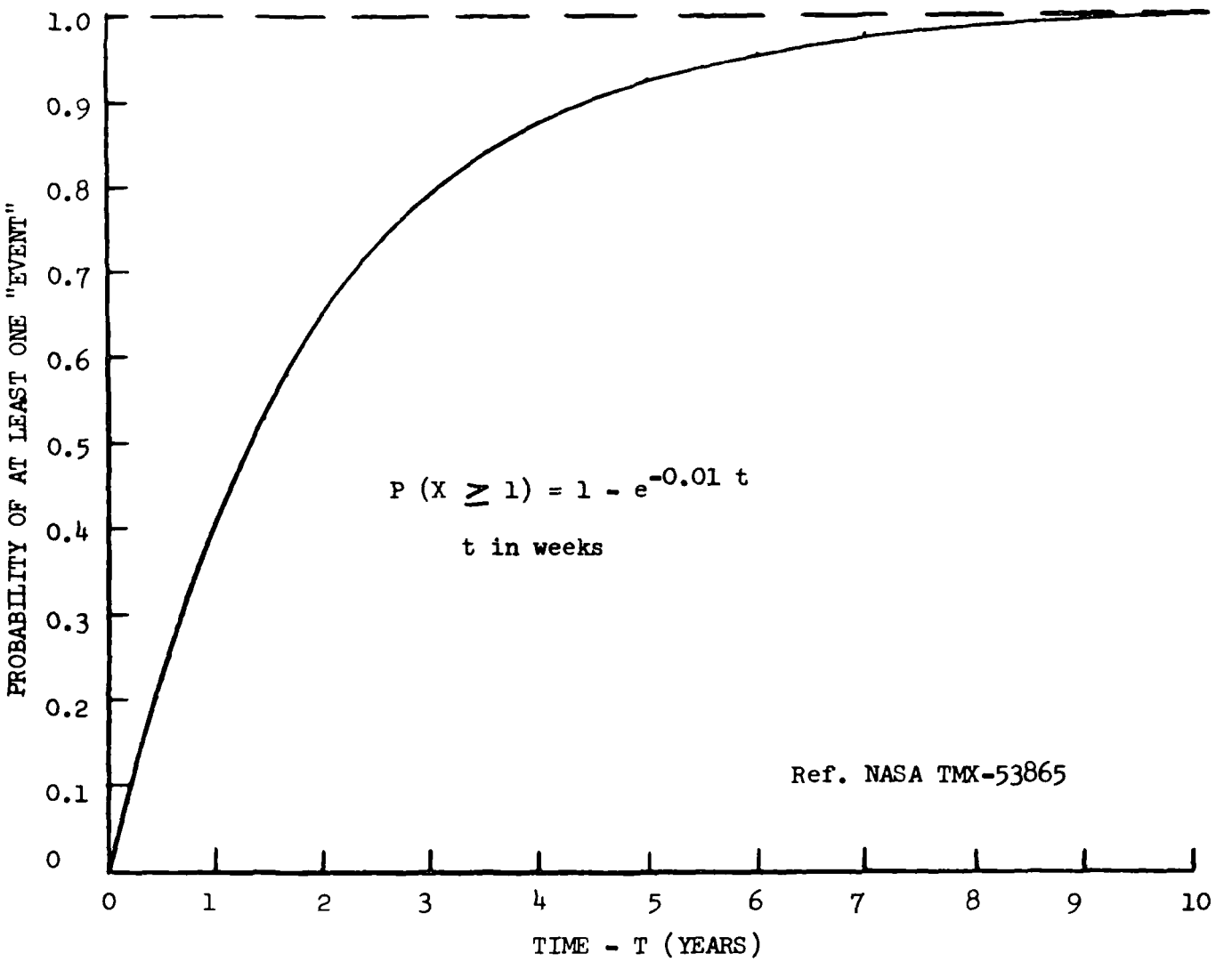


Figure 3.6-14. Model Solar Flare "Event" Probability as A Function of OLS Program Duration

Space Division
North American Rockwell

Table 3.6-3. OLS Shielding Requirements

OLS Crew Staytime - 234 days

OLS shield thickness sized for one "event"

Annual Allowable Dose

Skin - 240 rem
Eyes - 120 rem
Blood forming organs (BFO) - 40 rem
Blood forming organs (BFO) - 80 rem (optional)

Required OLS Shield Thickness* (gm/cm^2)

Skin - 5.0
Eyes - 8.0
Blood forming organs (BFO) - 16.6
Blood forming organs (BFO) - 6.5 (optional)

Selected Concept

Provide 16.6 gm/cm^2 of shielding (structure provides $\approx$2 gm/cm^2)
5.8 inches of H_2O provides 14.6 gm/cm^2

Option

Provide 6.5 gm/cm^2 of shielding 5K pounds H_2O
Provide eye shields, and ground crew after mission

*Assumes RNS contributes 10 rem and galactic 5 rem

Selection Rationale

The OLS shielding thickness required to limit the dosage to 25 rems per year is 16.6 gm/cm^2. This is set by the blood forming organs dose limit of 40 rems in one year. Since the OLS structure provides an inherent shielding of about 2 gm/cm^2, it is recommended that the other 14.6 gm/cm^2 be provided by a water barrier of approximately 16,000 pounds and 5.8 inches thickness. As discussed in Section 3.6.2 of this volume, the 16,000 pounds of water also serves the purpose of a heat sink to lower the radiator area requirements for thermal protection to an acceptable level.

The configuration of the OLS solar storm shelter is shown in Figures 3.6-16 and 3.6-17. The water shield surrounds backup command and control consoles, the backup galley, and a toilet facility. Approximately 2000 pounds of food is also used as shielding; some of it stowed near the galley and some of it is movable to allow crew ingress/egress. Figure 3.6-17 shows that tne

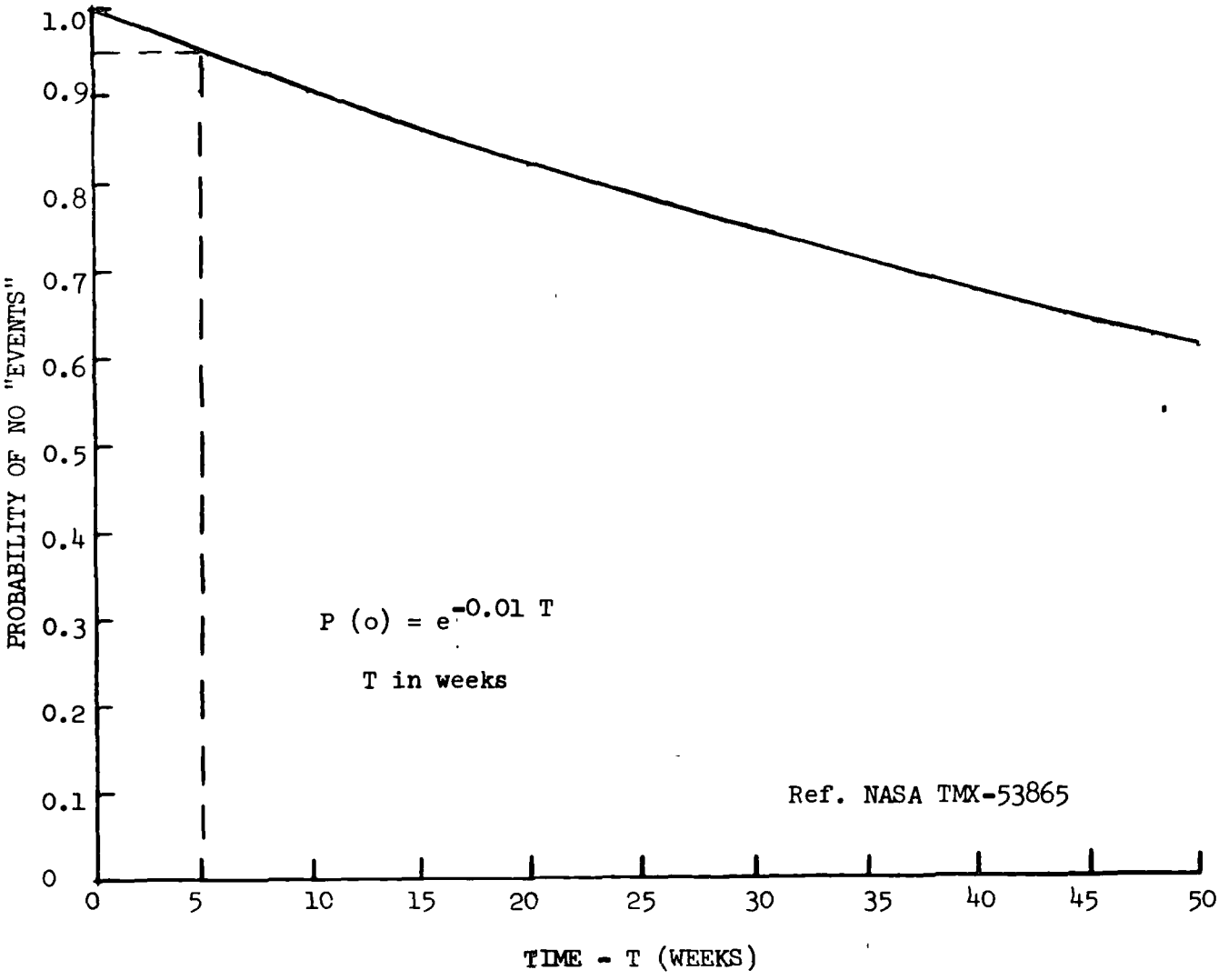


Figure 3.6-15. Probability of No Model Solar Flare "Events" With Time

Space Division
North American Rockwell

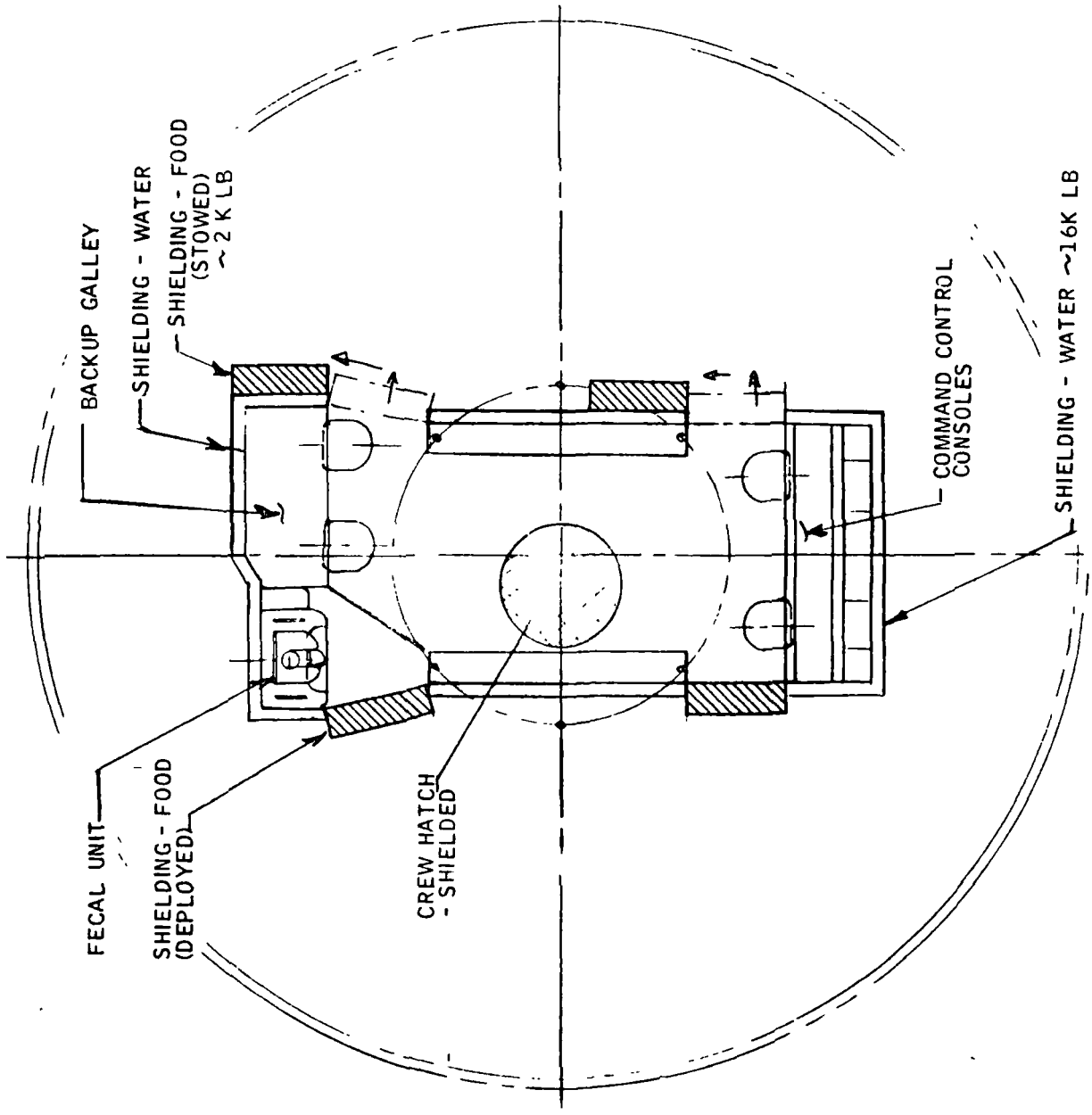


Figure 3.6-16. OLS Radiation Shelter Configuration

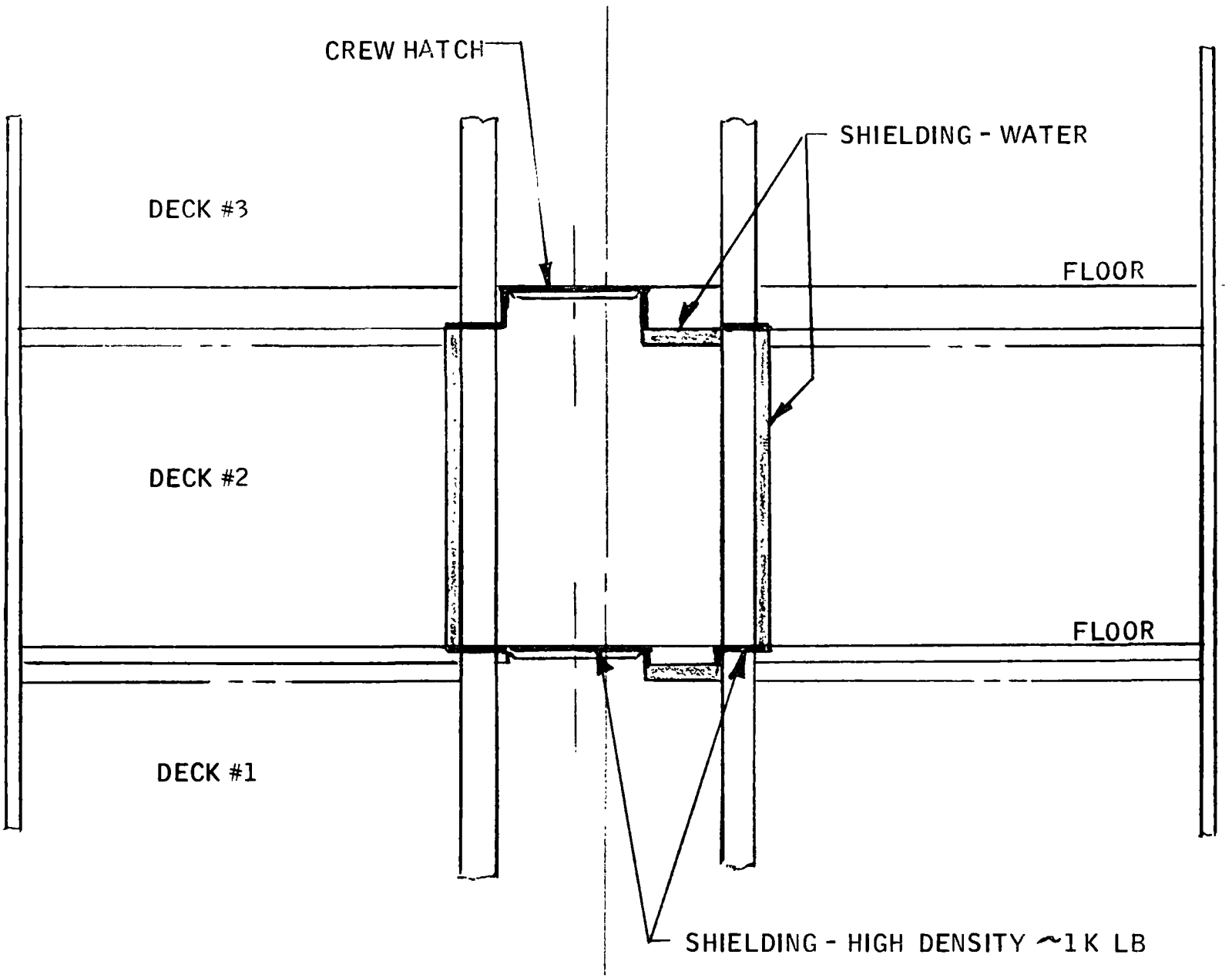


Figure 3.6-17. Side View of OLS Radiation Shelter

Space Division
North American Rockwell

shelter is located on Deck 2 and the hatches connecting the shelter with Decks 1 and 3 are high density shielding of an additional 1000 pounds.

Adaptability to Artificial Gravity

An artificial-G mode of operation for the OLS has no effect upon the radiation protection concept except that if the shelter is in the rotating section, floor space would have to be allotted for fold-away cots for crew sleeping. In a zero-G storm shelter the crew could sleep in "cocoons" suspended from the walls in any suitable orientation.

4.0 FUNCTIONAL REQUIREMENTS ANALYSIS

Both the earth-orbiting modular space station (MSS) and the derivative OLS are made up of a collection of 14-foot diameter modules capable of being transported to earth orbit in the cargo bay of the earth orbit shuttle (EOS). Both configurations utilize core modules (C1, etc.) to which are docked a power module (PM), control center modules (CCM's) crew quarters modules (CQM's), cargo storage modules (CM's), galley module (GM), and an experiment module (XM). The derivative OLS utilizes MSS modules modified to meet OLS requirements.

This section presents a comparison between the 8-man derivative OLS and the baseline 6-man MSS for our types of functional requirements, i.e., floor area, docking provisions, resupply and storage, and science support. Although the primary purpose of this section is to present minimum requirements, the means by which MSS modules must be modified to meet the derivative OLS requirements are briefly discussed where appropriate.

The floor area requirements for the derivative OLS are equal to, or less than, the requirements for the MSS 12-man growth station in all common categories (common to both stations) except recreation area. The derivative OLS has unique requirements for a laundry and a geochemistry laboratory, whereas the MSS has unique requirements for a physics/materials laboratory, fluid/electrical/optics laboratory, and for medical integral experiments. The total floor area requirements are 1436 square feet and 2204 square feet for the derivative OLS and the 12-man MSS respectively.

A total of 13 core module docking ports are required by the derivative OLS; the same number required by the MSS 12-man growth station. Two of the larger size MSS core modules (core module 1's having 10 ports each) are required for the derivative OLS in order to provide 7 docking ports in the same plane.

The total storage weight requirement for the derivative OLS are approximately 2-1/2 times the requirement for the MSS initial station. The biggest factor accounting for this ratio is the relatively large cryogenic storage requirement for the derivative OLS plus hydrazine storage for subsatellites, necessitating the utilization of two permanently docked cryogenic storage modules (i.e., modified MSS cargo modules). The derivative OLS 109-day resupply weight requirements are twice the 120-day resupply requirements for the MSS initial station. Expendable propellant modules and dual-support cargo modules provide resupply to the OLS whereas reusable cargo modules provide resupply to the MSS.

The experiment imposed station requirements have a major impact upon the information subsystem (ISS), and the guidance and control subsystem (G&C) of both stations. The most significant difference in requirements is the OLS

required in-track accuracy of ±850 feet compared to 3800 feet for the MSS and is, by far, the most difficult for the OLS to achieve.

4.1 FLOOR AREA REQUIREMENTS

A comparison between the floor area requirements imposed by the OLS program on the derivative OLS configuration, and the EOSS program requirements imposed upon the MSS is presented in Table 4-1. Requirements are shown for the 6-man baseline MSS configuration and also for the 12-man MSS growth configuration. The latter configuration requirements were added after it was observed that several of the derivative OLS requirements could not be met by the 6-man MSS configuration. All of the OLS floor area requirements are met by the 6-man MSS with the exception of the following items: crew staterooms, hygiene and storage, medical/exercise, laundry, primary control center, and recreation. The subsequent paragraphs discuss how the requirements for each of these six items are met in the derivative OLS.

The 6-man MSS provides four staterooms (50 square feet each) in crew module CQM1 and two staterooms in crew module CQM2. The OLS requirements for 8 staterooms are met by substituting CQM3 (which is used on the 12-man MSS and has 4 staterooms) in place of CQM2.

Hygiene and storage area totaling 194 square feet is provided in crew module CQM1 (77 square feet - with no modification), crew module CQM3 (100 square feet - with no modification), and by the addition of a toilet area in control center module CCM1 (17 square feet), thus meeting the minimum OLS requirement of 192 square feet.

The control center module configured for the 12-man MSS (CCM2) meets the OLS primary control center requirement of 100 square feet including two consoles. CCM2 also includes 250 square feet for a fluid, electrical and optics laboratory which is not required for the OLS. This laboratory area can be redesignated as the medical/exercise area to meet the OLS minimum requirement of 204 square feet.

The 12-man MSS galley module GM contains a galley and dining area of 200 square feet, a recreation area of 60 square feet, and a hygiene and storage area of 77 square feet. For the derivative OLS, the galley and dining area is reduced to 145 square feet, the hygiene and storage area is replaced by a laundry and storage area, and the recreation area enlarged to 80 square feet. These modifications provide for OLS minimum area requirements for laundry, recreation, and primary galley/dining.

4.2 CORE MODULE DOCKING PORT REQUIREMENTS

The OLS docking port requirements are necessarily sensitive to, and affected by, the station configuration concept. Previously defined OLS docking port requirements (Section 9.0, Volume II) apply to the non-modular representative OLS configuration. These requirements apply equally to the derivative OLS configuration, with the addition of several requirements that are unique to the latter configuration. Additional docking provisions requirements appear in Section 6.0 of Volume III.

Table 4-1. Floor Area Requirements Comparison

	Area (feet^2)		
	OLS	MSS	
	8-Man	6-Man	12-Man
Crew staterooms (50-foot^2/man)	400	300	600
Primary galley and dining area	100	110	200
Laundry	35	--	--
Backup galley and crew care	40	40	40
Hygiene and storage	192	154	254
Medical/exercise	204	150	220
Control center - primary ***	100**	50*	100**
Control center - backup	45*	45*	45*
Recreation	80	40	60
Photo lab	70	75	75
Data analysis lab	70	90	90
Physics materials lab	--	140	140
Geochemistry lab	100	--	--
Fluid, electrical, and optics lab	--	150	250
Medical integral experiments	--	--	130
Total	1436	1344	2204

* Including 1 console (48 x 31 x 50 in.)
** Including 2 consoles (48 x 31 x 50 in. each)
*** Additional OLS space required for accommodation of 7 experiment mass-memory modules (12 x 18 x 14 in. each)

Table 4-2 presents a comparison of the 6-man MSS and derivative OLS docking port requirements. The upper portion of the table presents the category of requirements which were previously defined for the OLS representative configuration, as discussed above. The lower portion of the table includes the additional requirements that are uniquely related to the MSS configuration. The location of docking ports, as well as the number of ports, must be specified for the OLS in order to adequately define the total docking port requirements. The bottom, top, and side designations are relative to the lunar local-level flight mode.

The OLS has unique docking port requirements imposed by space tugs and by the expendable dual-support cargo module. Conversely, the MSS has unique requirements imposed by a medical/exercise module (MEM), and multiple experiment modules such as the research application module (RAM). To assure adequate effective radiator cooling area on the various docked modules for the derivative OLS, all of the modules (with the exception of the power boom and the OLS experiment module) must be located at side ports, i.e., in a plane parallel to local horizontal flight mode (Refer to Sections 3.6.2 and 7.7 for further explanation concerning the lunar heat-rejection problem).

To provide a minimum of eight side docking ports, two MSS ten-port core modules (C1's) are required. The core modules (C1 and C2) included in the 12-man MSS configuration will not meet the OLS requirement because they provide a total of only 6 side docking ports. The OLS experiment module must be located at a bottom port to provide end viewing along lunar nadir (see Section 2.1). The tugs can be docked to top and bottom ports, but not at adjacent ports due to interference between landing gears.

4.3 RESUPPLY AND STORAGE REQUIREMENTS

A comparison of the storage requirements between the MSS 6-man initial configuration and the 8-man derivative OLS is presented in Table 4-3. The MSS storage requirements are based upon the requirement that the space station be capable of independent operation for 120 days without resupply. Each OLS requirement figure is based upon the program requirement for 180 days independent operational capability or upon normal 109-day resupply plus worst case contingency allowance, whichever is greater.

For most of the categories listed in Table 4-3, the difference in magnitude between the MSS and the derivative OLS is primarily due to the difference in number of crew (for crew sensitive items) and difference in the time duration requirement for independent operation. A major difference in experiment programs (hardware and objectives) between the MSS and OLS results in greatly different experiment consumable requirements. The storage requirements for MSS experiment supplies are 5380 pounds, or approximately 35 percent of the total MSS storage requirement. Storage for 595 pounds of experiment supplies are required for the OLS. The much larger LH_2 and LO_2 requirements for the derivative OLS are attributable to both the time duration requirement difference, and to the larger orbital maintenance delta V/month requirement of the OLS in lunar polar orbit. The N_2/O_2 leakage at docking ports is considerably higher for the derivative OLS which has 19 docking ports compared to 10 for the MSS 6-man initial station (Refer to Section 6.2 for further discussion).

Table 4-2. Core Module Docking Port Requirements

OLS Representative Configuration, Related Docking Port Requirements	8-Man OLS	6-Man MSS
Power boom	1 (A)	1
Dual-support cargo module	1 (S)	-
Tug	*2 (B/T)	-
Experiment module (RAM for MSS)	1 (B)	2
Spare	1 (B/T/A)	1
Subtotal	6	4
Additional MSS Derivative, Related Docking Port Requirements		
Crew quarters module	*2 (S)	*2
Control center	*2 (S)	*2
Galley and recreation	1 (S)	0
Cryogenic storage module	2 (S)	1
Subtotal	7	5
Total	13	9

*One each in separate pressure volumes

S - Side

A - Axial

B/T - Bottom and/or Top

Table 4-3. Storage Requirements Comparison

Criteria:

Derivative OLS

180-day nominal or resupply plus contingency*

MSS

120-day nominal operations

	Density **	8-Man OLS		6-Man MSS	
	(lb/ft^3)	(lb)	(ft^3)	(lb)	(ft^3)
Food	23.1	5850	253	2795	121
ECLSS expendables	20.6	970	47	460	22
Personnel supplies (stowed in staterooms)	20.7	1050	51	565	27
Station spares	15.0	5920	395	1111	74
Experimental spares and supplies	33.8	595	17	5380	159
Cryogenics and hydrazine					
LH_2	3.23	1691	524	454	141
LO_2	57.5	8042	140	1606	28
LN_2	39.8	7349	185	3000	75
LN_2H_4	50.3	7811	155	-	-
Total		39,278		15,371	

* Whichever is greater

** Includes: 67% stowage efficiency for bulk items; 10% ullage for cryo; 25% allowance for positive expulsion device

The largest difference in storage requirements is due to cryogenics and hydrazine. The 12-man MSS provides no increase in cryogenic storage over the 6-man configuration. The total storage weight requirement for the derivative OLS is approximately 2-1/2 times that of the 6-man MSS.

A comparison of resupply requirements is presented in Table 4-4. The OLS figures are based upon the cislunar shuttle resupply flight frequency of every 109 days used in the operations sequence model developed in Section 7.0 of Volume II. The MSS figures are based upon a resupply frequency of every 90 days which has been determined to be compatible with maintaining the MSS initial configuration level of support. Most of the differences in requirements between the two programs can be attributed to differences in crew size and resupply interval. However, as with the storage requirements discussed previously, the resupply requirements to support the MSS experiments program are much higher than for the OLS experiments program. However, the total 109-day resupply requirements for the OLS are approximately twice the 90-day resupply requirements for the MSS.

The MSS program utilizes the cargo module (CM) for resupply of crew, cryogenics, and bulk cargo. The CM remains docked to the core module providing additional storage space (pantry concept), until a replacement CM arrives three months later. The replaced CM, with down payload, is then returned to earth for replenishing. The OLS cargo resupply concept (as discussed in Section 5.0, Volume IV) utilizes an expendable dual-support cargo module that is docked directly to the OLS for a relatively short time during which the OLS cargo is off-loaded. The module is subsequently expended on the lunar surface during a sortie mission. Resupply cryogenics are transported to lunar orbit in an OLS propellant module (discussed in Section 4.0 of Volume IV), and the crew are transported to lunar orbit in a crew bus on the cislunar shuttle (discussed in Section 7.0 of Volume II). The derivative OLS will utilize the MSS CM as a permanently docked module to provide required cryogenics storage space.

4.4 SCIENCE SUPPORT

The experiment program for the MSS in earth orbit differs from the OLS experiment program in lunar orbit in terms of experiment objectives, experiment equipments, and supporting modules and laboratories.

The MSS NASA Blue Book total experiments program consists of 26 FPE's (functional program elements), comprising 162 experiments to be conducted in on-board laboratories, in an attached airlock (contained in module CCM1), in attached and detached RAM's (total of 16), and in a medical/exercise module (MEM). Several buildup options are being considered in connection with the MSS experiments program schedule; including the 4-man plateau, the 6-man initial station, the 8-man plateau, and the 12-man growth station. Implementation of the total experiments program requires the growth station configuration with a crew of 8 to 12 men.

The OLS lunar orbital experiments program consists of a total of 16 experiments requiring 24 equipment items including three subsatellites. The

Table 4-4. Consumables Resupply Comparison

Criteria:

Derivative OLS

109-day resupply interval

MSS

90-day resupply interval

	8-Man OLS (lb)	6-Man MSS (lb)
Food	3,335	2,100
ECLSS expendables	555	243
Station spares	3,580	411
Station cryogenics		
LH_2	1,158	282
LO_2	4,345	1,490
LN_2	4,175	2,060
Experiment cryogenics		
LO_2	-	78
LN_2	125	246
LN_2H_4	3,150	-
Experiment spares and supplies	360	4,040
Total	20,793	10,950

experiments are to be conducted in the attached OLS experiments module (which contains an airlock) and in the subsatellites at various lunar orbital attitudes and inclinations.

Table 4-5 presents a comparison of experiment imposed station requirements for the derivative OLS and the MSS 12-man growth station. The OLS ISS requirements presented are the same as, or less severe than, the MSS requirements. Conversely, the OLS G&C requirements are the same as, or more severe than, the MSS requirements. The tighter requirements for the OLS are a direct result of the lower nominal orbital altitude of the OLS (60 n mi in lunar orbit) compared to the MSS (270 n mi in earth orbit).

Table 4-5. Experiment Imposed Station Requirements

	8-Man OLS	6-Man MSS
INFORMATION SUBSYSTEM		
Data rate/quantity		
Digital - instant. peak (bps)	6×10^6	6×10^6
daily average (bps)	0.1×10^6	1×10^6
Television - sta. CCTV (channels)	3	3
sta. to earth (channels)	1	1
Storage capability		
Operating memory (words)	18K	18K
Mass storage (megawords)	1	1
Command and control		
Discrete commands	1500	2000
Variable commands (8-bit)	80	100
Maximum distance (n mi)	1100	2800
Data processing (bpd)	1×10^{10}	15.8×10^{10}
GUIDANCE AND CONTROL SUBSYSTEM		
Ephemeris accuracy (1σ)		
Altitude (ft)	±330	±1500
In-track (ft)	±850	±3800
Cross-track (ft)	±490	±2200
Orbital velocity (fps)	±25	±100
Angular rate control		
Continuously (deg/sec)	±0.05	±0.05
Fine pointing mode (deg/sec)	±0.01	±0.01
Attitude control		
Inertial (deg)	±0.25	±0.25
Relative, continuous (deg)	±0.25	±0.25
Relative, fine pointing (deg)	±0.1	±0.1
ON-BOARD LABORATORIES		
Photographic	Yes	Yes
Data analysis	Yes	Yes
Physics/materials	No	Yes
Geochemistry lab	Yes	No
Fluid/Electronics/Optics	No	Yes

NOTE: Refer to Table 4-1 for lab area requirements.

* These requirements are based on previously defined requirements for the 33-foot diameter, 12-man, EOSS concept as neither 6-man nor 12-man MSS science oriented ISS requirements has been identified.

5.0 PERFORMANCE REQUIREMENTS COMPARISON

This section presents performance requirements comparison matrices which aid in identifying the MSS modifications required to permit its use in lunar orbit as the derivative OLS. The data sources used in the development of the matrices are as follows:

1. Volume III, OLS Performance Requirements
2. Volume IV, OLS Configuration and Systems Analysis
3. Sections 2.0 and 3.0 of this volume, Volume V, Configuration Definition
4. MSC-02464, Volume 1, Modular Space Station Definition Document, 6-Man Operational Capability Plateau

A summary of the results of these performance requirements comparisons are presented below.

1. Consideration of feasible methods of delivering the unmanned derivative OLS to lunar orbit following checkout in earth orbit resulted in a unique OLS structural requirement. In order to transport the derivative OLS from earth orbit to lunar orbit in a partially assembled state, with some of the modules docked at side docking ports of core modules C1A and C1B, it is necessary to strengthen the primary structure surrounding these side docking ports. This is required because a moment of 1,890,000 inch-pounds (ultimate load), could be induced into this structure by the docked modules due to RNS thrusting during lunar orbit insertion. MSS core module structural design requirements at these ports are for bending loads of only 170,000 inch-pounds. No other structural or subsystems requirements due to delivery considerations were identified. All other structural requirements differences affect secondary structure only and are primarily the result of consideration of other subsystem requirements or the functional requirements differences discussed in Section 4.0.

2. Significant rearrangements affecting secondary structure were found necessary in:

   a. Control Center Module 1 (CCM1): OLS ENPS requirements necessitated modification to incorporate a solar flare radiation shelter enclosing the secondary control shelter and a toilet. Also the OLS laboratory requirements in this module differ from MSS, necessitating some structural rearrangement.

b. The Galley Module (GM): The hygiene facility is not required by the OLS. These facilities can therefore be removed and laundry facilities, uniquely required by the OLS, are substituted.

c. Control Center Module 2 (CCM2): The Fluid/Electronics/Optics laboratory of the CCM2 is not required. The OLS exercise/medical facilities are substituted for this laboratory.

d. Cargo Modules: Rather extensive internal structural modification and the addition of cryo storage tanks are necessitated by the larger OLS cryogenic storage requirements. The modules are redesignated as Cryo Storage Modules 1 and 2 (CSM1 and CSM2).

3. ECLSS requirements differences of note are:

a. The OLS emergency resupply period of 30 days, compared to the MSS period of 48 hours, coupled with increased OLS seal leakage, results in a unique N_2 and an increased O_2 emergency and leakage makeup storage and resupply requirement.

b. The OLS requirement for a laundry results in a slight increase in water requirements and a decrease in clothing storage requirements as compared to the 6-man MSS.

c. The 8-man OLS with a 180 day resupply period naturally requires larger atmospheric processing units and increased onboard consumables storage and resupply requirements as contrasted to the 6-man MSS with a 120 day resupply period.

4. Electrical Power Subsystem (EPS) requirements comparisons are made between the 8-man OLS and the 12-man MSS, as the 6-man EPS (8,000 square foot solar array) is inadequate for OLS use. The 12-man MSS EPS, however, is found to be adequate or slightly over capacity in all respects.

5. Information Subsystem (ISS) requirements comparisons indicate no differences in required ISS performance. As discussed in Section 6.0, some modifications or relocations of the MSS ISS equipment is required for OLS use even though the performance requirements are the same. An operational requirement difference does result in the addition of one antenna system that extends beyond the solar arrays. The OLS is required to have continuous communications capability in all directions; the MSS does not.

6. Guidance and Control (G&C) Subsystem requirements are very similar for the OLS and MSS. For routine operations, however, the OLS navigational accuracy requirements, and in particular the In-Track uncertainty of $\pm$ 850 ft. (1σ), are more stringent than those required of the MSS in earth orbit.

7. OLS Reaction Control System (RCS) total impulse and cryogenic storage requirements exceed those of the 6-man MSS concept. Also, the OLS propellant resupply logistics requirements are unique.

8. Environmental Protection Subsystem (ENPS) requirements comparisons show that OLS module heat rejection requirements exceed 6-man MSS module heat rejection requirements. This fact, coupled with the high infrared radiation from the lunar surface, necessitate changes in the thermal protection portion of OLS ENPS described in Sections 6.0 and 7.0.

   Micrometeoroid protection requirements are identical and, as described in Section 6.0, MSS provisions are adequate for OLS use.

   Radiation protection requirements are essentially the same for MSS and OLS. However, the nuclear radiation emitted during solar flare events is of critical concern in lunar orbit, while in earth orbit much of this radiation is deflected by the earth's geomagnetic field. Therefore, the OLS will require a solar flare radiation shelter.

5.1 STRUCTURES

In Table 5-1 the structures subsystem requirements of the MSS and derivative OLS are compared. Table 5-2 presents a comparison of the characteristics of the individual modules of the initial (6-man) MSS and the derivative OLS; this comparison is made to show the basic changes in both external and internal configuration of the MSS modules when adapted for lunar orbital operations.

Some differences which exist between structural performance requirements of the MSS and derivative OLS are detailed in the following paragraphs.

5.1.1 Orbital Maneuvering Considerations

During buildup in either earth or lunar orbit, a tug may be used to supply the delta-V for orbital maneuvers. For these operations, a tug would be docked to the X-axis docking port on core module 1B with the thrust applied axially. Hence, this docking port must be designed to withstand the 38,000-pound thrust load of the tug. The aft docking port of the MSS core module must withstand the thrust loads from two RL-10 engines on the EOS (17,300 pounds each) and three RCS engines (1,500 pounds each)

5.1.2 Lunar Orbit Delivery Considerations

Delivery of the unmanned derivative OLS from earth orbit to lunar orbit via a Cislunar Shuttle (CLS) following checkout in earth orbit can be accomplished in either an assembled or disassembled mode. A second consideration is if the CLS is to be "expended" in lunar orbit (must be refueled in lunar orbit) or shall be capable of self-return to earth orbit.

Table 5-1. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix
Structures

Performance Category	8-Man OLS-MSS	EOSS-MSS
Launch and Ascent	See Vol. III, Sect. 7.1, para. 7.1.2	Same
Natural and Induced Environment	Withstand imposed loads without excessive deflection or failure	
Safety Factors	Ultimate = 1.50 Yield = 1.20	
Earth Orbital	See Vol. III, Sect. 7.1, para. 7.1.2	Same
Natural and Induced Environment	See above	
Safety Factors		
Manned Operations	Ultimate = 2.00 Yield = 1.50	
Transient/Short Duration	Ultimate = 1.75 Yield = 1.30	
TLI and LOI	See Vol. III, Sect. 7.1, para. 7.1.2	NA
Natural and Induced Environment	See above	
Safety Factors	Ultimate = 1.50 Yield = 1.20	
Lunar Orbit	See Vol. III, Sect. 2.2, para. 2.2.4	NA
Natural and Induced Environment	See above	
Safety Factors		
Manned Operations	Ultimate = 2.00 Yield = 1.50	
Transient/Short Duration	Ultimate = 1.75 Yield = 1.30	

NA = Not Applicable

Space Division
North American Rockwell

Table 5-1. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix (Continued)
Structures

Performance Category	8-Man OLS-MSS	6-Man EOSS-MSS
Structure - General	See Vol. III, Sect. 7.1, para 7.1.2	
X-axis docking port in core module 1	Withstand thrust loads of orbital maneuvers, TLI, attitude control applied by tug or shuttle vehicle 38K thrust load of tug	Same (where applicable) 39.1K on core module by EOSS
Structure and docking ports	Withstand forces induced by dock/undock ops.	Same
Docking ports	At least 5 ft dia. clear opening for cargo transfer	Same
Primary structure and side docking ports in core module 1	Withstand moments of 1,890,000 in-lb (ult.) induced by side docked modules during delivery to lunar orbit via RNS	Withstand moments of 170,000 in-lb (ult.)
Pressure isolation	Minimum of 2 separate press. compartments	Same
IVA/EVA airlock	At least 5 ft dia. x 7 ft long At least one	At least 10 ft dia. x 6 ft long 1 in each core module
Configuration - General	See Vol. IV, Section 2.0	
Crew habitability interface	For detailed reqmts, see Vol. III, Section 7.1, para. 7.1.4	Basic reqmts. same
Floor arrangement	No requirement	Longitudinal
Aisles	30 inches wide	32 in. - 36 in. wide
Cargo transfer hatches	5 ft dia. hatches	Same
Internal passageway	No requirement	36 in. square hatchway
Module Characteristics	See "EOSS versus OLS Module Characteristics Comparison" Table 5-2	
Artificial-G Adaptability	See Vol. IV, Sect. 2.5 for criteria	
Operations	Zero-G with Artificial-G adaptability	Zero-G only

Space Division
North American Rockwell

Table 5-2. EOSS-MSS Versus OLS-MSS Module Characteristics Comparison

Module Designation	8-Man OLS Characteristics	6-Man MSS Characteristics
C1A - Core Module #1 (OLS) (MSS)	12 ft dia x 41.5 ft long One end expanded to 14 ft dia, but G&N gyros in C1B 10 docking ports (passive) 2 separate pressure volumes Main utility distribution EPS and G&C RCS quads one end only IVA/EVA airlock	12 ft dia x 41.5 ft long 3 G&N gyros at one end which is expanded to 14 ft diameter Same Same Same Same Same Same
C1B - Core Module #1 (OLS) (MSS)	Same as above with noted exceptions 10 docking ports (2 active plus 4 active/active adapters 3 G&N CMG's at one end	Same as above
PM - Power Module	~14 ft dia x 42 ft long Same (10,000 ft^2 - 8 men) 2 docking ports (1 active) Cryo storage - 6 tanks Pressurizable IVA/EVA airlock capability	~14 ft dia x 42 ft long Solar array (8000 ft^2 - 6 men) (10,000 ft^2 - 12 men) Same Same Same Same
CQM1 - Crew Quarters	14 ft dia x 31.75 ft long 4 staterooms Hygiene (1 urinal, sink, shower, Dry John) 2 docking ports (1 active) ECLSS sized for 8 men Backup galley Storage below longitudinal floor	14 ft dia x 31.75 ft long Same Same Same Sized for 6 men Same Same
CQM2 - Crew Quarters (GM - Galley Module on OLS)	14 ft dia x 31.75 ft long No staterooms 2 docking ports (1 active) Hygiene (laundry and sink only) Galley and dining ECLSS not required Storage Same as GM of 12-man MSS	14 ft dia x 31.75 ft long 2 staterooms Same Hygiene (1 urinal, sink, shower, Dry John) Galley and dining ECLSS Same
CQM3 - Crew Quarters	14 ft dia x 31.75 ft long Same as CQM1 but contains commander's stateroom 8-man ECLSS air revitalization and thermal control	14 ft dia x 31.75 ft long Sized for 6 men
CCM1 - Control Center	14 ft dia x 31.75 ft long No airlock required 3 lab areas Storm cellar with toilet Backup control center 2 docking ports (1 active)	14 ft dia x 31.75 ft long plus 88 in. dia x 10 ft long airlock Airlock lab Same No storm cellar Control Center #1 2 docking ports (1 active) - Same

Table 5-2. EOSS-MSS Versus OLS-MSS Module Characteristics Comparison (Continued)

Module Designation	8-Man OLS Characteristics	6-Man OLS Characteristics
CCM2 - Control Center	14 ft dia x 31.75 ft long Exercise/medical (add a sink) Primary control center No labs 2 docking ports (1 active)	14 ft dia x 31.75 ft long Same (no sink) Control Center #2 Lab and medical Same
Cryo Storage Module (CSM1 and 2)	~14 ft dia x 31.75 ft long Function performed by tug Cargo storage and transfer, pantry operations Cryo storage in unpressurized area, replenish PM cryo Pressurizable	~14 ft dia x 31.75 ft long Crew transport (4 men) Same Same Same
RAM - Research/Appl. Experiment Module	None ~15 ft dia x 22 ft long Experiments airlock Subsatellite M&R Cameras, sensors, etc.	EOSS oriented operations None
Dual Support CM (Resupply) (DSCM)	~15 ft dia x 11 ft long Performs same function as on represent. OLS Stores sortie cargo Stores OLS cargo Replaced every 109 days Pressurizable Expended on L.S. 2 docking ports - passive	None
Flex Ports	3 ft ID (≃4 ft OD) Provide auxiliary access route between normally habitable modules Pressurized Connects CCM2 to GM, GM to CCM1, CQM3 to CQM1	3 ft ID (≃4 ft OD) Same Same

Figure 5-1 presents an assembled derivative OLS configuration. It consists of seven modules of identical dimensions, two core modules, a power module, experiment module, two space tugs, and a Dual Support Cargo Module (DSCM). Details of all the modules are discussed in other sections of this report. In this section only the implications of the assembly on delivery are discussed.

In order to activate and initiate OLS operations, only one space tug is required and the DSCM is not required. The second tug is not required until sortie missions are initiated, and the DSCM is not required until subsequent logistics resupply.

Disassembled Delivery

Figure 5-2 presents a possible configuration for delivery of a disassembled OLS to lunar orbit. The total weight of the CLS payload, depicted in Figure 5-2, is approximately 310,000 pounds, not including the mounting adapters between the tiers of modules. This payload is within the capability of the baseline Reuseable Nuclear Shuttle (RNS), when operated in the expended mode (330,000 pounds) and the two stage Chemical Propulsion Stage (CPS-2), even in the self return mode of operation (350,000 pounds out; -0- back). The single stage CPS (CPS-1) model payload capability (223,000 pounds expended) is inadequate.

Delivery by this mode implies assembly in earth orbit, checkout, disassembly, unique assembly on the cislunar shuttle, disassembly of the payload in lunar orbit and then reassembly of the OLS in lunar orbit. In the case of the RNS, a second CLS flight is required to refuel the first RNS. Also, the adapters between tiers are an additional development program and are not useable on the OLS in lunar orbit. The total complexity and cost of this mode of operation indicates delivery of the OLS to lunar orbit in an assembled configuration is preferred. Delivery in an assembled or partially assembled mode is also believed to enhance crew safety by delivering the OLS to lunar orbit in as near an operational state of readiness as possible.

Assembled Delivery

Figure 5-3 presents a concept for delivery of a partially assembled OLS with two CLS flights. The assemblage of modules shown in Figure 5-3 for the first CLS flight was selected to minimize orbital assembly operations in lunar orbit, to provide a favorable CG location on the CLS, and to be approximately equal to the RNS payload capability when operated in a self return mode (147,000 pounds out; -0- back).

The total payload on the first flight is approximately 158,100 pounds. The payload on the second flight is approximately 149,400 pounds. Section 7.0 of Volume II, Operation Sequencing, indicated that the payload capability of the RNS model used in this study only marginally supported the logistics model. It was suggested that RNS payload capability be increased by 30,000 to 50,000 pounds. Such an increase would accommodate the payloads of the delivery concept of Figure 5-3. CPS-1 payload capability (93,500 pounds out; -0- back) would have to be increased significantly. CPS-2 capability is excessive.

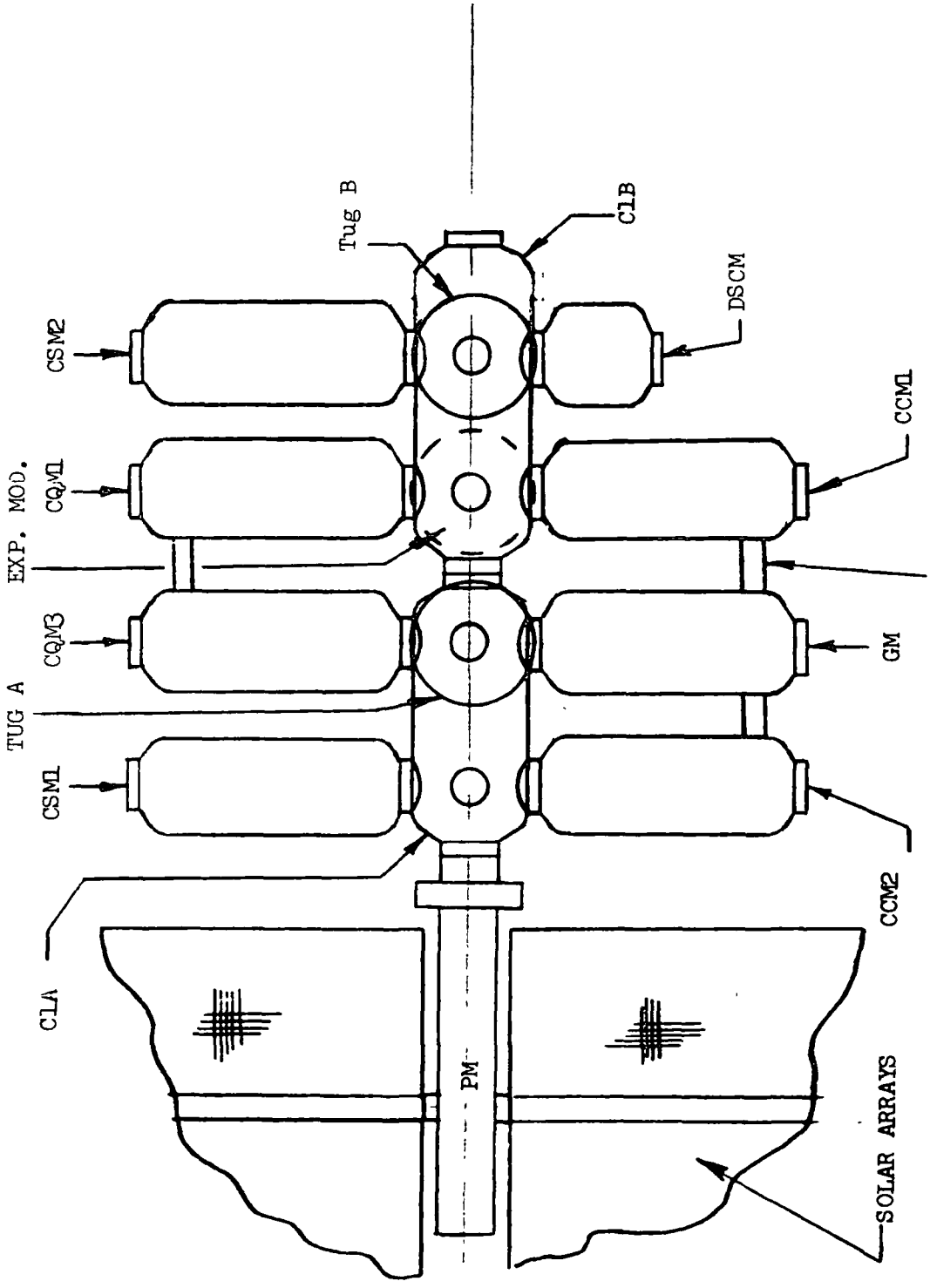


Figure 5-1. Derivative OLS Configuration - Module Locations

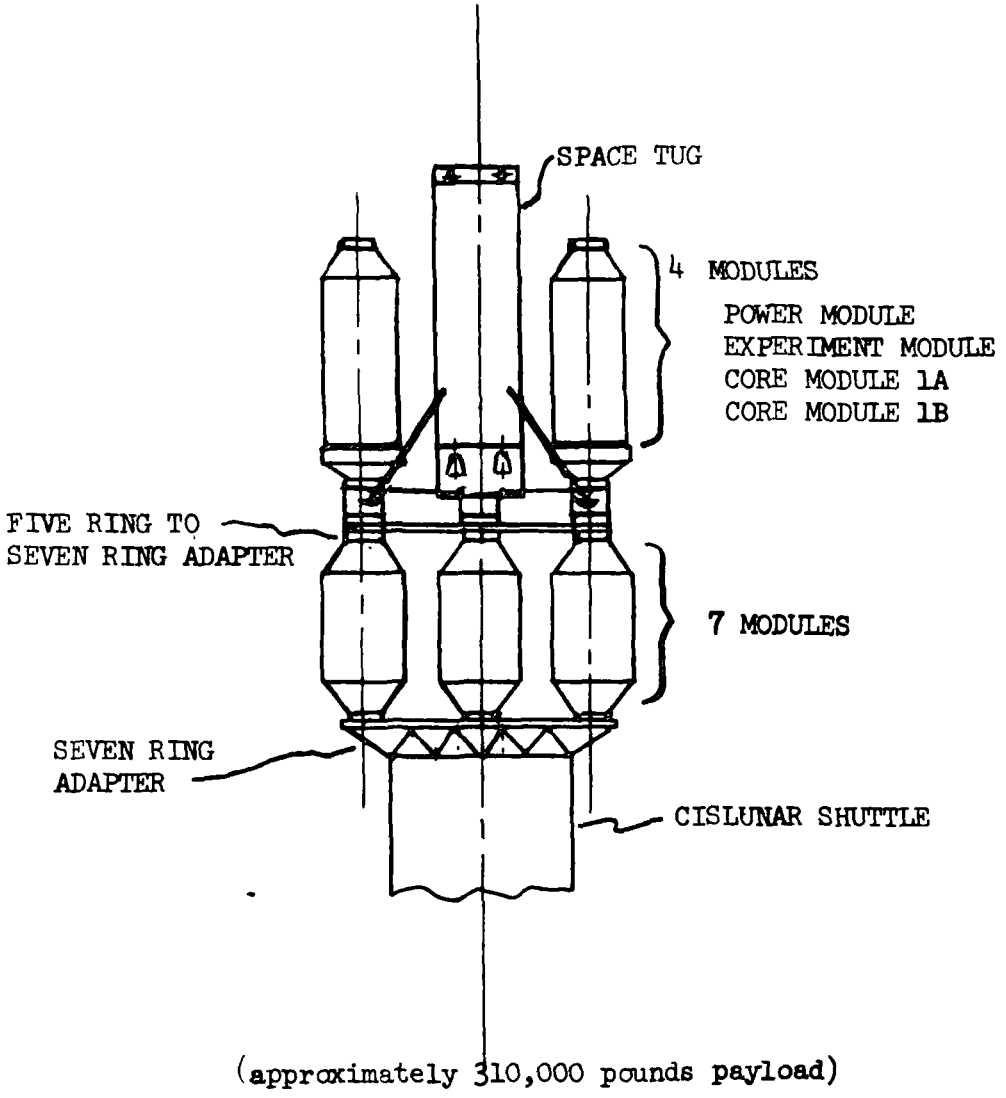
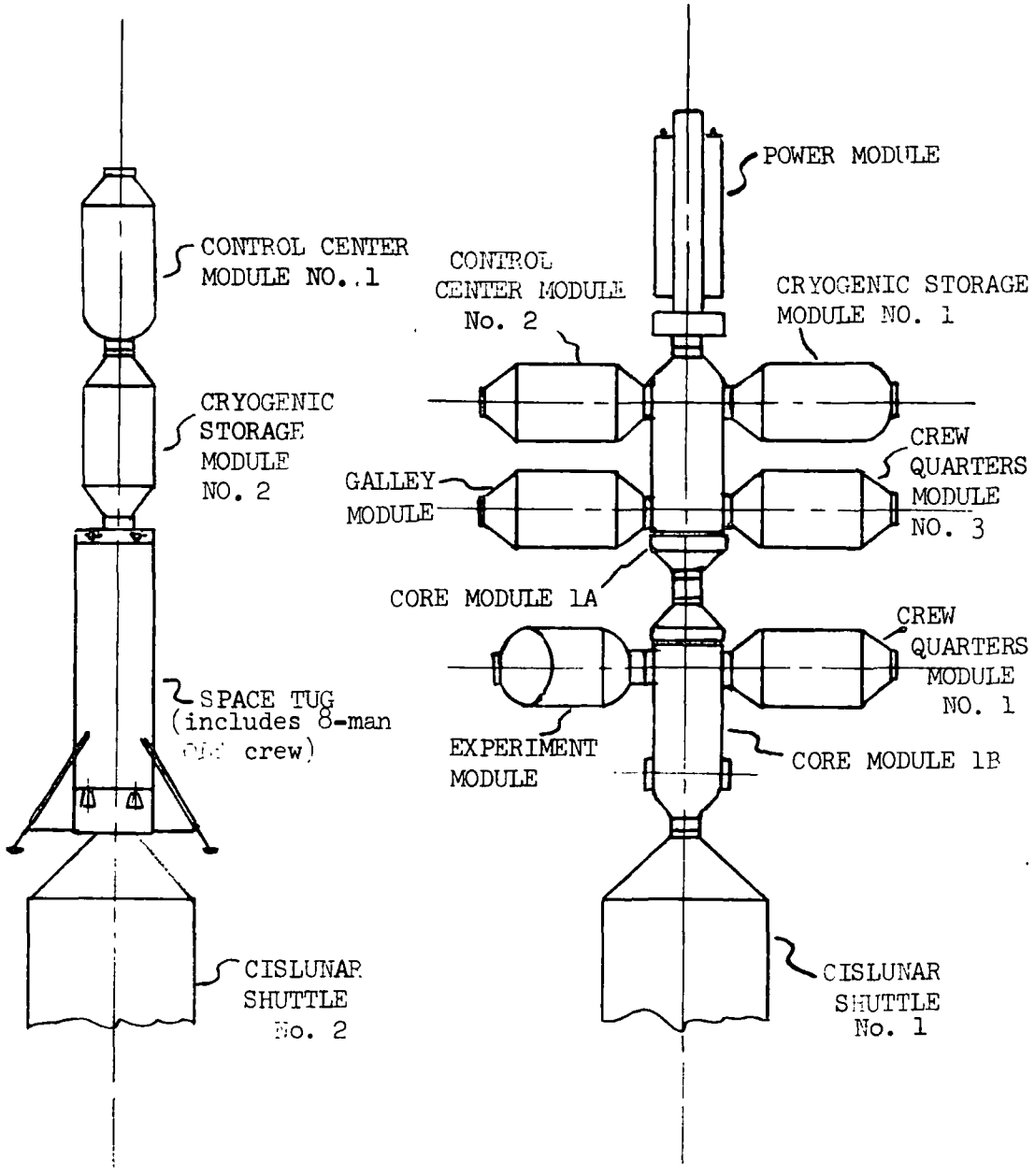


Figure 5-2. Disassembled Delivery Concept



(approximately 149,400 pounds payload)

(approximately 158,100 pounds payload)

Figure 5-3. Partially Assembled Delivery Concept

The primary problem associated with the delivery of the derivative OLS in an assembled configuration is the bending moment induced into the docking ports and surrounding core module structure by side docked OLS modules during CLS thrust maneuvers. The thrust level of the RNS is 75,000 pounds, while the CPS thrust level is 960,000 pounds. Accelerations during Lunar Orbit Insertion (LOI) are 0.22 g's and 1.375 g's for the RNS and CPS, respectively. The worst case bending moment that occurs during RNS LOI is 1,890,000 inch-pounds ultimate load (based on a dynamic amplification factor of 2 and a factor of safety of 1.5) at the junction of the galley module and core module 1A. The baseline CPS loads would be six times larger. MSS core module 1 is designed to withstand only 170,000 inch-pounds bending moment.

Two approaches were considered to circumvent the problem: (1) bracing either by cables or rigid structures between the ends of the modules and the cislunar shuttle, or (2) strengthening the structure surrounding the affected docking ports of the core module. Strengthening this structure to withstand the moment induced by RNS LOI would result in an increase in weight of 250 pounds per affected docking port, or 1,000 pounds per module assuming that both OLS core modules C1A and C1B would be identically modified. A complete major redesign would be required if CPS-2 acceleration levels are assumed. A rigidizing system of support frames for the RNS case was estimated to also weigh approximately 2,000 pounds, but the assembly, disassembly operations appeared to be rather complex. Rigidizing for CPS-2 acceleration levels was not evaluated, but would be heavier and at least equally complex.

Conclusion

Based upon the above rationale, the baseline concept for delivery of the derivative OLS to lunar orbit is as shown in Figure 5-3 using two RNS flights and strengthening the structure at each of the docking ports in core modules 1A and 1B. The only limitation on the use of CPS-2 for delivery purposes is the high acceleration levels. If the CPS engines are throttleable and can have a resultant acceleration comparable to the RNS, the CPS is also a candidate delivery vehicle for the derivative OLS concept.

The linear arrangement of the OLS modules and the tug which make up the payload for the second CLS flight, depicted in Figure 5-3, present no load problems assuming that column body bending effects, not analyzed in this study, are tolerable. The tug and the OLS modules involved are capable of withstanding axial loads greater than the 0.22 g or 1.375 g limits induced by the RNS or CPS, respectively.

5.1.3 Configuration - General

The MSS modules employ a longitudinal floor arrangement. As the OLS has no restrictions on floor orientation, it would utilize longitudinal floors to minimize modification of the existing MSS design. Likewise, internal passageways for cargo transfer and crew mobility on the derivative OLS need not differ in size from those on the MSS.

Figure 5-1 shows an assemblage of modules for the derivative OLS configuration. A comparison of the characteristics of these modules with their MSS counterparts shown in Drawing 9953-1011 is presented in Table 5-2. A discussion of the rationale for the required MSS module conversion modifications is given below.

5.1.4 Core Modules 1A and 1B

The external dimensions of the derivative OLS core modules are identical to those of MSS core module 1. The CMG's and G&C equipment essential to experiment operations have been moved to OLS core module 1B. (For rationale see paragraph 5.5).

As a result of docking port requirements, the second core module of the MSS is replaced by a core module identical to C1A for the derivative OLS and is called C1B. The second core module of the derivative OLS must dock four modules in the sample plane as those docked to C1A. This cannot be done with the second core module of the MSS. In addition, C1B is required to have two active end docking ports: one that mates with C1A and the other which can mate with the passive port of a tug for orbital operations, or a cislunar shuttle for translunar delivery. The X-axis docking port of C1B must withstand the 38,000 pound thrust load of a tug during orbital maneuvers. Whereas, all MSS RCS quads are located on C1, half the OLS RCS quads are located on C1B. The structure and side docking ports of both C1A and C1B must withstand a moment of 1,890,000 inch-pounds (ultimate load) induced by side docked modules during lunar delivery in a partially assembled OLS configuration.

5.1.5 Power Module

The MSS cryogenic storage tanks in the power module are inadequate for the OLS. Additional storage capacity and redundancy is provided in the two cryogenic storage modules (CSM1 and CSM2). The option exists to use the cryogenic tanks in the power module or leave them empty.

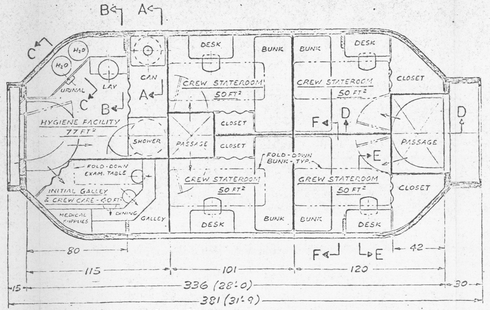
5.1.6 Crew Quarters Modules

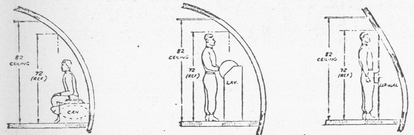
These modules, CQM1 and CQM3, are identical to those of the MSS.

5.1.7 Galley Module

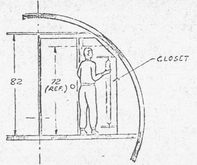
On the growth version of the MSS (12-man), CQM2 is reconfigured to a galley module (GM) by removing the two staterooms and expanding the dining and recreation area. It is this reconfigured CQM2 which is used by the derivative OLS. The primary difference between the MSS GM and the OLS GM exists in the hygiene area, where the MSS urinal, shower, and dry-john are replaced by a laundry. In addition, the MSS ECLSS in GM is not required for the OLS because redundancy is provided in CQM1 and CQM3.

Page Intentionally Left Blank

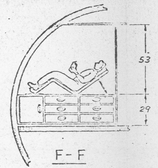




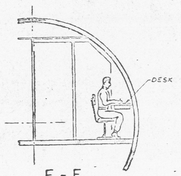
PERSONAL HYGIENE FACILITY UTILIZATION



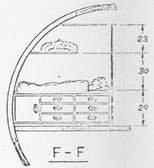
E-E
CLOSET ACCESS



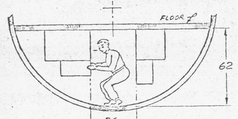
F-F
RECLINING



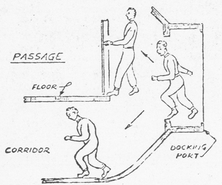
E-E
DESK UTILIZATION



F-F
SLEEPING



LOWER STORAGE



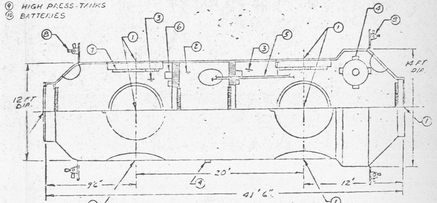
CREW QUARTERS MODULE No.1 (CQM1)
FLOOR PLAN & SECTIONS
(TOTAL FLOOR AREA = 350 SQ. FT.)

9953-1011 SHEET 2

- DESIGN FEATURES
  1. 10 DOCKING PORTS (MASSIVE)
  2. EVA/EVA AIRLOCK
  3. 2 SEPERATE VOLUMES
  4. C&W EQUIPMENT
  5. MAIN UTILITY DISTRIBUTION
  6. ELECTRICAL POWER SYSTEM
  7. OVAL HATCH UTILIZATION
  8. ECS GUARDS
  9. HIGH PRESS. TANKS
  10. BATTERIES

- SPECIAL DESIGN

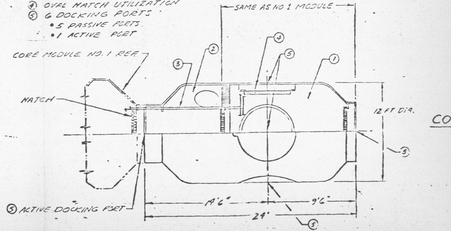
CORE MODULE	CONFIGURATION MODULE
LENGTH 41'6"	LENGTH 31'3"
DIAMETER 12'	DIAMETER 14'
DOCKING PORTS 10	DOCKING PORTS 2



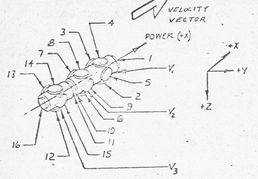
CORE MODULE No.1 (LOI1G) C1

- DESIGN FEATURES
  2. INDEPENDENT VOLUME (1/2)
  3. EVA/EVA AIRLOCK
  4. MAIN UTILITY DISTRIBUTION
  5. OVAL HATCH UTILIZATION
  6. 6 DOCKING PORTS
     - 5 PASSIVE PORTS
     - 1 ACTIVE PORT

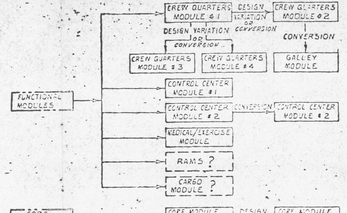
- ADAPTATION OF CORE MODULE NO. 1



CORE MODULE No. 2 (GROWTH) C2



NUMBERING SYSTEM
CORE MODULES DOCKING PORTS

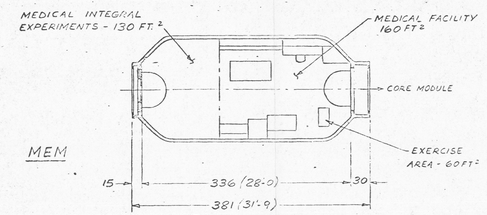


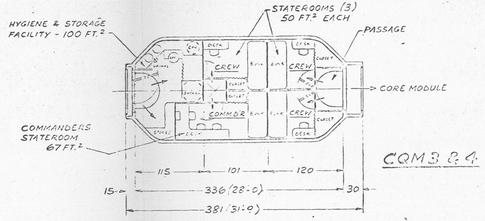
MODULAR SPACE STATION
TITLES AND ABBREVIATIONS

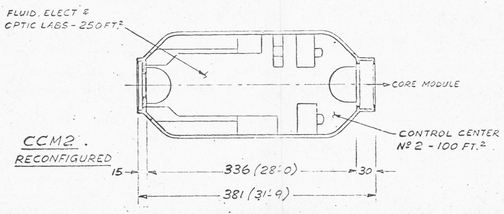
TITLE	ABBREVIATION
MODULAR SPACE STATION	MSS
POWER MODULE	PM
CORE MODULE NO.1	C1
CORE MODULE NO.2	C2
CREW QUARTERS MODULE NO.1	CQM1
CREW QUARTERS MODULE NO.2	CQM2
CREW QUARTERS MODULE NO.3	CQM3
CREW QUARTERS MODULE NO.4	CQM4
CONTROL CENTER MODULE NO.1	CCM1
CONTROL CENTER MODULE NO.2	CCM2
MEDICAL/EXERCIZE MODULE	MEM
GALLEY MODULE	GM
CARGO MODULE	CM
RESEARCH AND APPLICATION MODULE	RAM

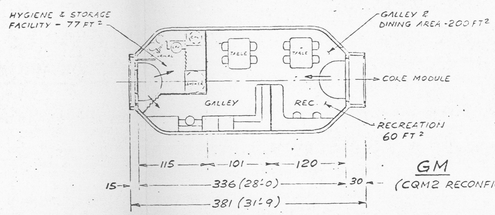
SD 71-207 5-15, 5-16

APPROVED BY			
		MODULAR SPACE STATION 'PREFERRED' MODULE CONCEPT ARRANGEMENT	9953-1011 SHEET 1 OF 2



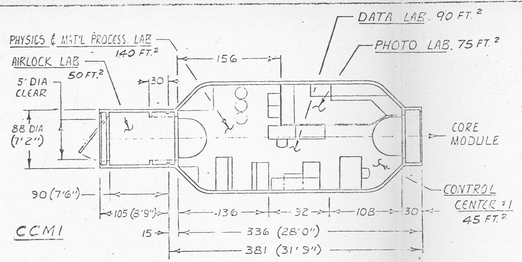


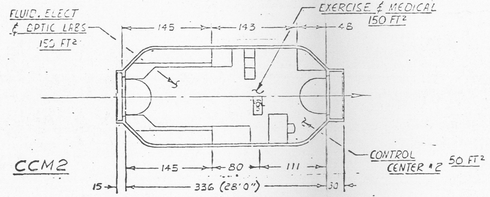




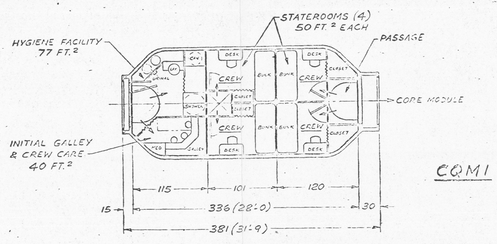
GROWTH CONFIGURATION

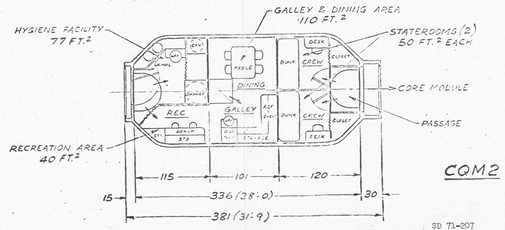
9953-1011 SHEET 4





INITIAL CONFIGURATION





SD 71-207

9-17, 9-18

BLOCK	BY, DATE	WHAT WAS IT
1	M. BECK 6/7/71	NORTH-SOUTH ROTATION REVERSED
	DATE	

MODULAR SPACE STATION 'PREFERRED' MODULE CONCEPT ARRANGEMENTS

9953-1011 SHEET 3 OF 6

5.1.8 Control Center Module 1

The experiments airlock of the MSS CCM1 is not required on the derivative OLS because one exists in the OLS experiment module. The OLS solar storm shelter is incorporated in this module and includes a toilet and the secondary control center of CCM1. The Physics Materials Lab of the MSS CCM1 is replaced by a Geochemistry Lab for the OLS.

5.1.9 Control Center Module 2

Because there is no requirement for laboratry space in CCM2 of the OLS, the exercise/medical area of the MSS CCM2 can be expanded to meet the spatial requirements for OLS medical facilities. (On the MSS the medical/exercise area requirements are fulfilled by the combination of space allocation in CCM2 and a Medical Exercise Module.) A sink is added to the exercise/medical area of the CCM2 for OLS use.

5.1.10 Cryogenic Storage Modules

The cryogenic storage requirements of the OLS are much greater than those of the MSS. Separate storage facilities are required for safety reasons. Therefore, two cryogenic storage modules (CSM1 and CSM2) are required. The MSS cargo module (CM) is used for crew transfer and cargo transfer and storage as well as cryogenic storage. As OLS crew transfer is accomplished by a crew module on the cislunar shuttle and the tug, and cargo transfer is accomplished by means of a dual-support cargo module (DSCM), the MSS CM can be modified to provide adequate cryogenic storage for the OLS. In addition, the hydrazine for subsatellite support is also stored in one of the CSM's. Although the CSM's are normally unpressurized, they can be pressurized for crew maintenance functions.

5.1.11 Experiment Module

This module is identical to that described for the representative OLS in Section 2.1. Although MSS and OLS experiments each have unique requirements, it may be practical to convert an MSS Research Applications Module (RAM) for OLS use. Lack of definition of RAM's at this time precludes this determination.

5.1.12 Dual-Support Cargo Module

This module is identical to that described in Section 9.0 of Volume 2 of this report.

5.2 ENVIRONMENTAL CONTROL AND LIFE SUPPORT SUBSYSTEM

The differences between the OLS and MSS environmental control and life support subsystem (ECLSS) performance requirements (Table 5-3) are discussed below.

5.2.1 O_2/N_2 Leakage Makeup

The derivative OLS uses two MSS C1 core modules to obtain the required number of docking ports. The additional docking ports and thus more leakage paths coupled with the longer emergency resupply period of the OLS (30-day OLS; 48 hours MSS) will result in larger resupply and storage requirements for O_2/N_2 leakage makeup in the OLS.

5.2.2 CO_2 Partial Pressure

The MSS nominal CO_2 partial pressure constraint of 3.0 mm Hg is more stringent that the 5.0 mm Hg constraint of the OLS. Therefore, it would appear that the MSS has excess capability. However, the OLS preferred concept is the same as the MSS concept, an H_2 depolorized cell. Therefore, no changes or modifications are required.

5.2.3 Heat Rejection Loads

Heat rejection loads are discussed in Section 5.7, Environmental Protection Subsystem, under Thermal Protection.

5.2.4 Hygiene Flushing

The 3.5-pounds of water per man-day requirement is in the process of reevaluation. Initial results from tests in a prototype program indicate that the requirement will decrease to 2.0 pounds per man-day. This water requirement decrease would simply decrease the water process rate for the vapor compression unit (and thus, its size).

5.2.5 Housekeeping and Personal Hygiene

There are minor difference in shower and housekeeping water usage rates but in both cases the OLS is more conservative. The difference in laundry water rates is due to the different laundry concepts of the OLS and MSS. The OLS employs a total laundry concept (see Section 3.1) whereas the MSS has disposable clothing.

5.2.6 Housekeeping Storage

With the non-reusable laundry concept on the MSS, there is a larger requirement for clothing storage. The minimum area requirement for the total laundry concept on the OLS is approximately 35 cubic feet.

5.2.7 Other Differences

In addition to the above noted differences, there are others which exist because of the onboard stowage requirements for the OLS and MSS due to the differences in nominal resupply periods (180 days for the OLS, 120 days

Table 5-3. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix
ECLSS

Performance Category	8-Man OLS-MSS	6-Man EOSS-MSS
Atmospheric Storage and Supply	See Vol. III, Sect. 7.2, para. 7.2.2	
O_2/N_2 Quantities		
Crew	1.84 lb/M-D nominal	Same
Crew Overlap	1.84 lb/M-D for 16M x 16D maximum	Same
Crew Emergency	8M x 30D either pressure vol. w/1 tug	2D reserve/pressure volume (No N_2)
LSB/RNS Crew Rescue	20M x 55D max. w/2 docked tugs	NA
Leakage Losses (N_2, O_2)	52.0 lb/day	20.0 lb/day - 6 men
Pressurization Uses		
Exp. Airlock	200 ft^3 (subsatellites), $\simeq$ 4/year	147 ft^3, 5 times/month
Docking Ports	116 ft^3/port, 2 times/month	116 ft^3/port, 4 times/month
IVA/EVA Airlock	$\simeq$175 ft^3, 0 times/month	175 ft^3, 1 time/month
Exp. Module	1200 ft^3, 3 times/year	2400 ft^3, 3 times/month
Repressurization (O_2, N_2)	Provision for emergency repressurization of 1 pressure volume, 20K ft^3	20K ft^3, once per 120 days
CO_2 Management	See Vol. III, Sect. 7.2, para. 7.2.2	
CO_2 Generation	2.25 lb/M-D nom., 3.0 lb/M-D max.	Same
Partial Pressure (mmHg)	5.0 nom. 7.6 emerg. max., 15.0 for 2 hr.	Same, but 3.0 nominal
Atmospheric Control	See Vol. III, Sect. 7.2, para. 7.2.2	
Temperature	60-75 F exercise, 65-80 F hygiene 65-75 F all other areas	Same
Pressure (N_2/O_2 atmos.)	$3.1 \leq PPO_2 \leq 3.5$ psia, 14.7 psia total to 10.0 minimum	Same
Surface Temperatures	105 F maximum, 57 F minimum	Same
Dew Point	57 F maximum	Same
Air Velocity	15 fpm min., 100 fpm max., 40 fpm, nominal	Same

NA - Not Applicable

Space Division
North American Rockwell

Table 5-3. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix (Continued)

ECLSS

Performance Category	8-Man OLS-MSS	6-Man EOSS-MSS
H_2O Partial Pressure	$8 \leq PPH_2O \leq 12$ mmHg, 30-50% rel. hum.	Same
Trace Contaminants	0.1 of threshold limit value/constituent	Same
Bacteria Concentration	Monitor and control	Same
Condensation	No condensation on internal surfaces	Same
Active Thermal Control	See Vol. III, Sec. 7.2, para. 7.2.2	
Heat Rejection Loads	Range among modules; 0.4 kw in CM to 8.0 kw in GM	Range among modules; 0.3 kw in CM to 6.0kw in GM
Equipment Temp. Control	40 to 100 F heat sink	Same
Metabolic Heat Load	Total 11,200 Btu/M-D Latent 4110 Btu/M-D, Sensible 7020 Btu/M-D	Same
Water Management	See Vol. III, Sect. 7.2, para. 7.2.2	
Potable Water		
Food (Freeze Dry)	1.44 lb/M-D	Same
Drink	4.14 lb/M-D	Same
In Food (Wet)	0.96 lb/M-D	Same
Residual Loss	0.22 lb/M-D	Same
Total	6.32 lb/M-D	Same
Latent Loss	3.09 lb/M-D (persp.)	
Water Temperature	50 ± 5 F cold, 155 ± 5 F hot	Same
Hygienic Flushing	3.5 lb/M-D	Same
Experiment Use	35 lb/day, potable	Not defined
Closed Loop	Reclamation, reuse, storage	Same
Housekeeping and Personal Hygiene		
Washing	4.0 lb/M-D	Same
Shower	17.2 lb/man-shower	16.6 lb/man-shower
Housekeeping	1.0 lb/M-D	0.4 lb/M-D
Laundry	4 lb/M-D	2.7 lb/M-D (non-reusable)
Dishwasher	2.0 lb/M-D	Same

Table 5-3. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix (Continued)

ECLSS

Performance Category	8-Man OLS-MSS	6-Man EOSS-MSS
Waste Management	See Vol. III, Sect. 7.2, para. 7.2.2	
Urine		
Water	3.45 lb/M-D nom., 4.48 lb/M-D max.	Same
Solids	0.13 lb/M-D	
Fecal		
Water	0.25 lb/M-D nom., 0.33 lb/M-D max.	Same
Solids	0.13 lb/M-D	
Toilet Facilities	2 fecal and urine collection devices, 1 standup urinal /pressure volume	1 standup urinal and dry john per 4 crew (2 ea. for 6 men)
Food	0.4 lb/M-D wet, 0.6 lb/M-D pack	Same
Miscellaneous	Max. 29.5 lb/day, 0.4 ft^3 compacted, 2.0 ft^3 uncompacted	20 lb/day (8 men)
Contaminated Waste	Disinfect as close to source as possible prior to storage/disposal	Same
Hygiene	See Vol. III, Sect. 7.2, para. 7.2.2	
Housekeeping Storage	35 ft^3 (with laundry)	105 ft^3 (with no laundry)
Shower Facilities	1/pressure vol., 2-3 crewmen in any 12 hr period	1 shower/4 crewmen
Washing Sinks	6	Same
Food Management	See Vol. III, Sect. 7.2, para. 7.2.2	
Galley	Galley module of growth MSS , 180-day storage	Same, 120-day storage
Backup Galley	30D x 8M supply	7D x 6M supply, 48 hr reserve
Freezer (Interior)	70 ft^3 freezer, 10 ft^3 refrigerator	50 ft^3 freezer (6 men)
Food Consumption (Dry)	1.68 lb/M-D supply, 1.50 lb/M-D intake	Same

Space Division
North American Rockwell

Table 5-3. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix (Continued)

Performance Category	8-Man OLS-MSS	6-Man EOSS-MSS
Special Life Support	See Vol. III, Sect. 7.2, para. 7.2.2	
O_2 High Pressure Storage	IVA/EVA O_2 supply	Same
IVA Support		
O_2 nom. max. flow	8 lb/M-hr	Same
O_2 emergency flow	22 lb/M-hr for 30 minutes	Same
O_2 inlet temperature	40 F - 64 F	Same
Heat load	2000 Btu/M-hr for 3 hours	Same
Suit pressure	Source regulate to 110 psig	Same
LCG water flow	240 lb/hr @ 43 F	Same
Design capacity	4 hr operation	Same
PLSS Charging		
O_2/recharge	1.6 lb	Same
H_2O/recharge	11.0 lb	Same
Recharge press.	1410 ±3 psia	Same
Temperature	70 F - 90 F	Same
Design freq.	No use anticipated	Same
Quantity	4 required, 2/pressure volume	Same
Emergency Reserves	O_2 for 8M x 30D + leakage N_2 for leakage, H_2O from 180D storage	2D reserve/volume O_2
PGA	4 required, 2/pressure volume	Same
Fire Detection and Control	CO_2 extinguishers, atmospheric dump for large fires	Same

Space Division
North American Rockwell

for the MSS) and emergency resupply periods (30 days for the OLS, 48 hours for the MSS).

In general, all OLS processing equipment will have increased weight, power, and volume compared to MSS 6-man units because of the increased flow rates required for the 8-man OLS. CO_2 management, atmospheric storage and supply, and water management equipment are affected. Waste management, hygiene, food management, and special life support (except for emergency cases) are only slightly affected by the increased crew size.

5.3 ELECTRICAL POWER SUBSYSTEM

The major differences in electrical power requirements (Table 5-4) are essentially the result of OLS and MSS mission operational differences and crew size. They are discussed below.

5.3.1 Flight Envelope

The differences in MSS and OLS flight envelopes affect the orbit load profiles, the light-dark operations power requirements, and solar array degradation.

5.3.2 Crew Size

The differences between OLS and MSS crew sizes account for the differences in power requirements. The 6-man MSS requires an 8000-square foot solar array to meet primary power requirements. The 8-man OLS requires a 10,000-square foot solar array to meet its primary power requirements.

5.3.3 Major Power Modes

The OLS has several major power modes peculiar to its mission which influence the power requirements at the various phases of operation.

5.3.4 Power Source Applications

The manner in which the power requirements of the above modes are met is essentially the same for both the OLS and the MSS. Prior to the initial cislunar shuttle flight to lunar orbit, the power module and core module 1A will be mated in earth orbit. During unmanned earth orbit and translunar flight operations, power is obtained from partially deployed solar arrays. The primary batteries in core module 1A which supply power during MSS build-up prior to solar array deployment, perform the same function for the OLS in earth orbit. During buildup and system checkout in lunar orbit, power will be supplied by the solar arrays.

Table 5-4. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix

EPS

Performance Category	8-Man OLS-MSS	6-Man EOSS-MSS*
Flight Envelope	See Vol. III, Sect. 7.3, para. 7.3.2	
Earth Orbit	258 n mi circular, 31.6 deg. inclination	240-270 n mi, 55 deg. incl.
Translunar Flight	E.O. to L.O. via cislunar shuttle	NA
Lunar Orbit	60 n mi, circular polar	NA
Major Power Modes	See Vol. III, Sect. 7.3, para. 7.3.2	
Unmanned Launch and E.O. Premanning	100 w 5 days (Core 1) 2 kw for 20 min. (Core 1/Power Mod)	100w, 60 days, Core 1 2.5kw for 20 min PM and 2 kw for 12 hr C/O
Manned E.O. Operations	Zero g = 2.5 kw for 1 day	Zero g = 15 kw (25 kw)
Emergency	4 kw for 30 days in L.O. (48 hours in E.O.)	(4.2 kw for 48 hours) 3.0 kw for 48 hours
Backup	13.5 kw for 90 days	(15.0 kw for 30 days) 7.0 kw
Unmanned E.O., TLF	2.5 kw for 7 days (incl. 3 days C/O)	NA
L.O. Quiescent Storage	1.9 kw for 1 year	NA
Initial Manned L.O. Opns. (Buildup & Checkout)	15 kw for 16 days	NA
Manned L.O. Operations	20 kw continuous	NA
Earth Eclipse	15 kw for 3.5 hr	NA
Solar Flare Events	10 kw for 72 hr	NA
Subsystem Electrical Load Reqmts.	See Vol. III, Sect. 7.3	
Power Dist., AC	120/208 vac, 3Ø, 400 Hz, 4w	120/208 vac, 3Ø, 400Hz, 4w & 240/416 vac, 3Ø, 400Hz, 4w
Power Dist., DC	56 vdc, 2w	Same
Power Quality	per MIL-STD-704	Same

NA = Not Applicable

*12-Man MSS requirements are shown in parenthesis

Table 5-4. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix (Continued)
EPS

Performance Category	8-Man OLS-MSS	6-Man EOSS-MSS
Power Source Applications	See Table 7-7, Vol. III	
Unmanned Launch and Premanning	Primary (batteries)	Same (RFC NA)
	Auxiliary (batteries or RFC)	
Manned E.O. Operations	Primary	Same
Emergency	Emergency	Same
Backup	Primary	Same
Unmanned E.O. and Trans-lunar Flight	Primary*	NA
L.O. Quiescent Storage	Primary	NA
Initial Manned L.O. Opns.	Primary (Buildup and Checkout)	NA
Manned L.O. Operations	Primary	NA
Earth Eclipse	Auxiliary	NA
Solar Flare Events	Auxiliary	NA
Lighting	See Table 7-8, Volume III	
General	30 - 70 ft candles	30 - 50 ft candles
Emergency	5 - 10 ft candles	5 ft candles
Low Level	0.5 ft candles	Same
	*Partially deployed solar arrays on TLF and buildup in L.O.	

NA = Not Applicable

Space Division
North American Rockwell

5.4 INFORMATION SUBSYSTEM

There are no functional differences between the derivative OLS ISS and MSS ISS concepts. Those entries in column three of Table 5-5 marked with an asterisk, were not defined specifically in the MSS Definition Document, MSC 02464. Since the Station A performance specification is the basis for both the OLS and MSS, any differences in requirements are the result of sensitivity to crew size, scientific operations peculiar to the OLS, and Lunar Exploration Program mission support.

5.5 GUIDANCE AND CONTROL SUBSYSTEM

There are very few differences between OLS and MSS performance requirements (Table 5-6). The only significant difference is the tighter OLS ephemeris accuracy requirements during routine on-orbit operations.

5.5.1 Routine Operations

The more stringent OLS orbital position uncertainty requirements reflect the differences in the orbital altitude of the two stations

5.5.2 Location of Hardware

To ensure that the OLS Experiment Operating Plan meets its stability requirements, G&C equipment essential to meet these requirements must be located in core module 1B (where the experiment module is docked). Thus, for example, the OLS CMG's would be located in C1B rather than C1A.

5.6 REACTION CONTROL SUBSYSTEM

The three major factors which account for the differences between the derivative OLS and MSS RCS requirements (Table 5-7) are:

1. The design period for normal operations (directly affecting storage requirements)

2. The orbit parameters and environment (affecting impulse requirements)

3. The configuration and flight mode(s) (affecting impulse requirements)

Much of the large difference between derivative OLS and MSS total O_2 and H_2 storage requirements is due to the RCS impulse required for orbit maintenance. The gravitational perturbations in lunar orbit are much more severe than in earth orbit. Of course, the 180-day OLS design storage period (as compared to 120 days on the MSS) also accounts for part of the difference. More N_2 is needed onboard the OLS primarily to make up for the higher leakage rate. There is a higher leakage rate on the derivative OLS because it has almost twice as many docking ports as the MSS.

Table 5-5. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix
ISS

Performance Category	8-Man OLS-MSS	6-Man EOSS-MSS
External Communications	See Vol. III, Sect. 7.4, para. 7.4.2	
Channels	5 coherent duplex	Same
Mission phase utilization (transmit and receive)	See Table 7-10	*
Internal Communications	See Vol. III, Sect. 7.4, para. 7.4.2	
Int. comm. and info. sources	See Table	*
Entertainment and paging distribution	Carried on single channel, controlled from primary control center	Same
Audio-video distribution	Private voice; intercom between critical areas; closed circuit video; terminals for docked vehicles and attached modules	Same
Digital data distribution	All but certain dedicated sources linked	Same
Mission phase utilization (data distribution)	See Table 7-12	*
Tracking	See Vol. III, Sect. 7.4, para. 7.4.2	
Beam pointing	Within 3 db beamwidths	*
Ranging	See Table 7-13	Same during "terminal rendezvous" (1000 ft - 20 n mi)
	Medium range tracking (1000 ft - 450 n mi)	Provide data for RAM's and shuttles when in RF line of sight distances of 20 - 400 n mi
	Short range tracking by G&C	
Display and Control	See Vol. III, Sect. 7.4, para. 7.4.2	
Flight management	Active control capabilities in primary control center with backup	Same
Operations management	Subsystems, maintenance, emergency, personnel, EVA, inventory	Same

*Not defined in MSS documentation; OLS requirement adapted from Sect. 6.8 of SD 70-510-1, "Solar Powered Space Station Preliminary Performance Specification"

Table 5-5. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix (Continued)

ISS

Performance Category	8-Man OLS-MSS	6-Man EOSS-MSS
Planning and scheduling	Provide flexibility with backup capability	Same
Experiment management	Normally in experiments area	Same
Display types	Universal multiformat, alpha fixed format, hard data viewer, numeric, hard text printer	See Sections 6.4 and 7.4 of this volume
Control types	Discrete and alpha keyboard, mode keys, special purpose control, display address control	
Data Processing	See Vol. III, Sect. 7.4, para. 7.4.2	
Acquisition	1×10^7 bps peak rate	Same
Processing	Minimum capability: command/control, maintenance, emergency management in each pressure volume	Same
	1×10^7 bps peak rate	Same
Distribution	1×10^7 bps peak rate	Same
Storage: Operating memory storage	32K words/module (8 memory modules)	See Sections 6.4 and 7.4 of this volume
Storage: Baseline (excl. arch.)	2.5×10^6 words min.	
Storage: Arch. transfer rate	1×10^6 bps min.	
Software	See Vol. III, Sect. 7.4, para. 7.4.2	
Support operations	OLS flight operations	MSS flight op's.
	OLS vehicle interfaces	MSS veh. operations
	LEP mission support	NA
	Exper. oper./scientific investigation	RAM oper.
	In-orbit program modification	Same
Operating memory utilization	See Table 7-14	*

*Not defined in MSS documentation; OLS requirement adapted from Sect. 6.8 of SD 70-510-1, "Solar Powered Space Station Preliminary Performance Specification"

NA = Not Applicable

Table 5-6. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix
G&C

Performance Category	8-Man OLS-MSS	6-Man EOSS-MSS
	See Vol. III, Sect. 7.5, para. 7.5.2	
Orbit Insertion	Monitor booster	Same
Premanning		Same
Position state vector update	Ground supplied	
3-axis attitude hold	±5 degrees	
Manning		
State vector update	Ground to tug	Ground to EOS
3-axis attitude hold	±1.0 degree	Same
3-axis angular rates	±0.05 deg/sec	Same
Unmanned checkout	To assembly level	Same
Initial Manned Operations		
Initial manned checkout	To IFRU level	Same
Routine Operations	In L.O.	In E.O.
Altitude uncertainty	±330 ft, 1σ	±1500 ft, 1σ
In-track uncertainty	±850 ft, 1σ	±3800 ft, 1σ
Cross-track uncertainty	±490 ft, 1σ	±2200 ft, 1σ
Orbital velocity uncertainty	±25 fps, RMS	±100 fps, 1σ
Control of RCS	On-off signals	Same
State vector	Within 1 spherical n mi, 3σ	Same
Orbit makeup attitude hold	±0.1 n mi	Same
Reorientation rates (3-axis)	$\leq$ ±0.25 deg/sec	Same
Command for Free-Flying Exper.		
Docking range uncertainty	1000 ft to 100 ft $\Rightarrow$ ±1 ft, 1σ	Same
	100 ft to contact $\Rightarrow$ ±0.5 ft, 1σ	Same

Space Division
North American Rockwell

Table 5-6. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix (Continued)
G&C

Performance Category	8-Man OLS-MSS	6-Man EOSS-MSS
Docking contact conditions		
Position deviation	Lateral ± 0.5 ft	Same
Velocity deviation	Lateral ± 0.3 fps	Same
Velocity deviation	Axial ± 0.5 fps	Same
Initial docking rate damped	90 seconds	Same
Recovery	160 seconds later	Same
Max. docking transient	0.001 g acceleration	Same
Angular uncertainty	Not defined	± 0.4 degree, 1σ
Exper. Operations Stability		
3-axis attitude hold	(continuous) ±0.25 deg.	Same
3-axis attitude hold	(30 minutes) ±0.10 deg.	Same
3-axis attitude rate	(continuous) ±0.05 deg/sec	Same
3-axis attitude rate	(30 minutes) ±0.01 deg/sec	Same
Acc. transients	(≤ 5% of time) ≤ 0.001 g	≤ 0.0001 g
Maximum acc.	0.01 g	Same
Logistics Support	See "Command for Free-Flying Experiments"	Same
Docking range uncertainty		When EOS docked to MSS, fuel minimization is attitude control req. rather than attitude hold for experiments. Typical impulse req. for fuel mode is 1390 lb-sec/day.
Docking contact conditions		
Initial docking rate damped		
Recovery		
Max. docking transients		
Location of Hardware	Location of experiment module at C1B requires CMG's and other G&C hardware essential to experiment operations in C1B. Some redundancy in C1A for safety.	In C1, redundancy in both pressure volumes

Space Division
North American Rockwell

Table 5-7. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix
RCS

Performance Category	8-Man OLS-MSS	6-Man EOSS-MSS
Design to	See Vol. III, Sect. 7.6, para. 7.6.2	
Normal Operations Duration	180 days	120 days
Orbit	60 n mi, lunar circular polar	244-270 n mi, 55 deg. incl.
Environment	Defined by NASA TMX-53865	
Configuration	See attached sketch	External dimensions of functional modules same; number of functional modules differ
Operate to	See Vol. III, Sect. 7.6, para. 7.6.2	
Orbit	Any inclination 45 - 80 n mi	28.5 to 55 deg. incl., 200-300 n mi
Flight Modes	X-POPY level, X-POP inertial	Y-POP level (initial MSS) X-POP level (gross MSS)
Design and Sizing Requirements	(180 days)	(120 days)
CMG Desaturation	92,000 lb-sec	50,800 lb-sec
Maneuvers	12,000 lb-sec	8,000 lb-sec
Orbit Maintenance	2,559,000 lb-sec	447,200 lb-sec
Docking	--	14,000 lb-sec
Emergency	81,000 lb-sec	81,000 lb-sec
Cryogenic Storage	See Vol. III, Sect. 7.6, para. 7.6.2	
Design Period	109D + contingencies for H_2 and O_2 180D for N_2 and H_2	120 days storage for all cryo Consumable reserve one month beyond scheduled resupply. Shuttle launch frequency $\leq$ 30 days
Capacity		
Subsatellite Provisions	N_2 455 lb_m N_2H_4 7811 lb_m	NA
ECLSS		
Leakage + Meta.	O_2 = 2394 lb_m, N_2 = 6894 lb_m	O_2 = 0, N_2 = 1850 lbm
Sabatier Unit	H_2 = 324 lb_m	H_2 = 72 lbm
Repressurization	O_2 = 350 lb_m, N_2 = 1150 lb_m	O_2 = 350 lbm, N_2 = 1150 lbm

N/A = Not Applicable

Space Division
North American Rockwell

Table 5-7. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix (Continued)
RCS

Performance Category	8-Man OLS-MSS	6-Man EOSS-MSS
EPS Emergency Power	O_2 = 3697 lb_m, H_2 = 533 lb_m (30 days)	O_2 = 176 lbm, H_2 = 22 lbm
RCS	O_2 = 3033 lb_m, H_2 = 1589 lb_m	O_2 = 1080 lbm, H_2 = 360 lbm
Total Storage Required	O_2 = 8042 lb_m, H_2 = 1913 lb_m	O_2 = 1606 lb_m, H_2 = 454 lb_m
	N_2 = 7349 lb_m, N_2H_4 = 7811 lb_m	N_2 = 3000 lb_m
Location of RCS Ouads	On both core modules	On core module #1 only
Propellant Resupply Logistics	See Vol. III, Sect. 6.0 and Vol. II, Sect. 8.0	
CLS Tug Prop. P/L	55% of RNS P/L	NA
	Provide additional full prop. module for storage in L.O. with CPS-2	
Sortie Tug Mission	Every 109 days w/56,500 lb propellant consumption	NA
Rescue Vehicle	Fully fueled tug docked to OLS	EOS shuttle arrives when needed
Boiloff	48 lb/day per tug	
Subsatellite Provisions	Per 180 days: 455 lb_m N_2 7811 lb_m N_2H_4	NA
Resupply Procedure	Transport in prop. module via CLS. Propellant module expended on the lunar surface	Transport in EOS cargo bay
Need for Orbital Prop. Depot	See Vol. II, Sect. 8.0, para. 4.2	
w/RNS Shuttle Vehicle	None required	NA
w/2-Stage CPS	Full 77,000 lb capacity prop. module	NA
	Extra tug used for rescue capability	
Propellant Module	See Vol. II, Sect. 8.0, para. 4.3	NA
Design Requirements	77,000 lb prop. capacity	
	Positive expulsion device	
	External dimensions compatible w/EOS	
	Neuter docking port at each end	

NA = Not Applicable

Table 5-7. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix (Continued)
RCS

Performance Category	8-Man OLS-MSS	6-Man EOSS-MSS
Configuration	LH_2 capacity, 16,450 lb LOX capacity, 58,550 lb LN_2 capacity 2,000 lb LH_2 volume, 4,570 ft^3 LOX volume, 930 ft^3 LN_2 volume, 46 ft^3 15 ft dia. x 45 ft long 11,000 lb empty weight	NA

NA = Not Applicable

Space Division
North American Rockwell

On the MSS, RCS quads are located on the outer radius of each end of core module 1. These locations allow engine redundancy and minimize plume impingement on other components. There is no requirement to locate RCS quads on MSS core module 2. Because core module 1B on the derivative OLS is essentially identical to core module 1A (41.5 feet long compared to the 24-foot long C2 on MSS), it is to the advantage of the OLS RCS to locate quads on the far end of core module 1B because of the longer moment arm.

Propellant resupply concepts have been discussed in Sections 7.0 and 8.0 of Volume II of this report. The propellant consumption of the tug on surface sorties and Lunar Surface Base (LSB) visitations is the major criteria for the sizing of the OLS propellant module, which is also used to resupply the OLS cryogenic storage tanks located in CSM1, CSM2, and the power boom.

5.7 ENVIRONMENTAL PROTECTION SUBSYSTEM

5.7.1 Meteoroid Protection

The requirements for the MSS and OLS are identical (P_{nf} = 0.9 in 10 years).

5.7.2 Thermal Protection

The heat loads identified in Table 5-8 vary from module to module depending upon the module functions. The difference between OLS and MSS module heat loads is primarily due to the resizing of equipment that is sensitive to crew size (e.g., ECLSS, ISS and experiments, etc.) for the two additional crewmen of the OLS. These numbers are intended to be a conservative thermal model for the derivative OLS.

Heat loads in the OLS core modules should be identical because there are approximately the same number of active modules docked to each. This is not the case with the MSS.

The 6.0 kw load on the MSS power module is for the initial station 6-man crew. An 8.3 kw load on the MSS power module is for a 12-man crew (25 kw average electrical power requirement). For the same solar array battery concept, the OLS power module heat load has been reduced by the ratio of the OLS total power requirement to the 12-man MSS total power requirement.

5.7.3 Radiation Protection

The method by which the crew is protected from solar flare storms on the OLS and MSS is entirely different. In the orbit of the MSS, the earth's magnetic field deflects the charged particles towards the poles. By suitable arrangement of storage and equipment on the MSS, adequate inherent shielding exists without a requirement for a storm shelter. Because the moon has little or no magnetic field, radiation protection must be provided by 16.6 gm/cm^2 of shielding in a solar storm shelter onboard the OLS.

Table 5-8. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix
ENPS

Performance Category	8-Man OLS-MSS	6-Man EOSS-MSS
Thermal Protection	See Vol. III, Sect. 7.7, para. 7.7.2	
Wall temperatures (pressurized volume)	57 F min., 105 F max.	Same
Heat Loads	Varies with each module	
Cargo module	0.4 kw	0.3 kw
Core 1	4.8 kw	2.9 kw
Core 2	4.3 kw	3.5 kw (12-man only)
CCM 1	4.3 kw	3.5 kw
CCM 2	4.3 kw	3.5 kw
GM (CQM 2)	8.0 kw (2 hr max. during cooking)	6.0 kw
CQM 1	7.2 kw	6.0 kw
CQM 3	7.2 kw	6.0 kw (12-man only)
PM	~6.6 solar array battery (SAB) concept	6.0 kw (SAB concept)
Micrometeoroid Protection	See Vol. III, Sect. 7.7, para. 7.7.2	
Penetration	Probability of no penetration of pressure wall or systems compartment = 0.9	Same
Program Duration	10 years	Same
Pressurized Storage Vessels	Same reliability as above	Same
Environment	NASA specified in TMX-53865	Same
Radiation Protection	See Vol. III, Sect. 7.7, para. 7.7.2 and Vol. IV Sect. 3.6, para. 3.6.3	
Environment	NASA specified in NASA TMX-53865	Same
Crew Dose Level Limit	Career / Year / Month	Same
Skin	2400 / 240 / 150	
Eyes	1200 / 120 / 75	
BFO	400 / 40 / 25	
BFO (Opt)	80	
Testes *		200 / 38 / 13

*This is a recent addition to MSS Guidelines. It has no impact on OLS radiation protection.

Space Division
North American Rockwell

Table 5-8. EOSS-MSS Versus OLS-MSS Performance Requirements Comparison Matrix (Continued)
ENPS

Performance Category	8-Man OLS-MSS	6-Man EOSS-MSS
Storm Shelter	Shielding to protect 8 crewmen against radiation from one solar flare "event" as defined by above document. Shield weight = 16.6 gm/cm^2	No storm shelter but protect crew at emergency level during solar storms.
Aerothermodynamic Protection Boost Phase	See Vol.III, Sect. 7.7, para. 7.7.2 Provide protection from aerothermodynamic effects.	Same

Space Division
North American Rockwell

6.0 MSS CONVERSION MODIFICATIONS

Based on the performance requirement comparison matrices presented in Sections 4.0 and 5.0 of the MSS subsystem concepts as described in MSC 02464, the MSS definition document, were modified to allow the resulting vehicle, the derivative OLS, to operate in lunar orbit. Both required and desired MSS subsystem modifications are described in Sections 6.1 through 6.7 together with the rationale supporting the need for the suggested changes.

Required changes are defined and those MSS subsystem modifications which are absolutely necessary to meet the OLS requirements that were documented in Volume III. The derivative OLS that results when only required changes are implemented is, by definition, fully capable of performing all OLS mission requirements. This vehicle is however not optimized for OLS use. Where an MSS subsystem had excess capability (usually resulting in extra weight as well) it was considered a desired rather than a required change, to trim such excesses or to otherwise optimize the subsystem for OLS use. All of the suggested desirable changes are effective, however some of these changes would incur development costs not otherwise present in the derivative OLS program.

Major required changes which were identified are:

1. Reconfiguration of the MSS Cargo Modules (CM's) to conform to the OLS cryogenic storage requirements

2. Installation of a solar flare nuclear radiation shelter surrounding the secondary control center

3. Need for high pressure nitrogen and oxygen storage tanks in the core modules for emergency leakage makeup plus oxygen metabolic consumption

4. Addition of a laundry

5. Requirement for an additional high gain parabolic antenna subassembly boom mounted above the solar array to meet the continuous communications requirements

6. Addition of a landmark tracker and a radar altimeter to meet OLS navigation requirements

7. Addition of RCS hydrogen accumulators due to the larger OLS impulse requirements

8. Addition of a heat pump, a thermal capacitor, and additional radiator area required to satisfy OLS heat rejection requirements

9. Modification of the primary structure surrounding the side docking ports in MSS core module 1 (C1) to meet the unique OLS load requirement resulting from lunar delivery considerations discussed in Section 5.0.

Major desired changes which are described more fully in Section 6.1 through 6.7 are:

1. Remove cryogenic storage tanks from the power boom and replace the batteries and battery chargers with a regenerative fuel cell energy storage system

2. Reconfigure the cryogenic storage modules to house all OLS cryogens and add two GN_2 pressurization tanks to CSM 2

3. Change from a double to a single ac voltage distribution concept

4. Remove excess RCS engine clusters from the OLS core modules and replace the high pressure nitrogen tanks with accumulator tanks and resize the cryogenic storage tanks to optimize for OLS use.

6.1 STRUCTURES

The conceptual definition of the derivative OLS is based upon a thorough review of the current baseline modular earth orbital space station which has been defined in that study contract. The derivation of the derivative OLS concept has been performed in two phases. The first phase, discussed in Section 6.1.1 identifies the minimum changes to the total complement of modules that would be necessary to satisfy the basic OLS requirements. The second phase, discussed in Section 6.1.2 identifies those additional changes that are considered desirable to enhance the operational and configurational characteristics of the OLS.

In delineating the required structural or arrangement changes, each module will be individually discussed. The parent MSS modules are described in Drawing 9953-1011 presented in Section 5.0. For reference to the physical location of each module, Drawing 2282-29 illustrates the OLS orbital flight configuration.

6.1.1 REQUIRED CHANGES

C1A (Core Module 1A) (Derivative of C1)

1. Modify and strengthen primary structure surrounding side docking ports (the ports to which OLS modules CSM1, CQM3, CCM2, and GM are shown docked in Figure 5-1) to meet the moment requirements of 1,890,000 inch-pounds induced by side docked modules during delivery to lunar orbit in a partially assembled configuration as described in Section 5.0.

2. Add two high pressure (3000 psia) GN_2 tanks (4-foot diameter) for emergency leakage makeup and two high pressure GO_2 tanks (3.5-foot diameter) for emergency leakage makeup plus metabolic consumption.

3. The secondary support structure for the CMG's will be deleted since the CMG's are located in C1B.

C1B (Core Module 1B) Derivative of C1)

1. Modify and strengthen primary structure surrounding side docking ports, identically as for C1A above.

2. Add two high pressure GN_2 and GO_2 tanks as in C1A above.

3. The docking interface on each end of the module changed from passive ring to active cone.

CQM1 (Crew Quarters Module 1)

1. There are no structural changes made to the baseline MSS module.

CQM3 (Crew Quarters Module 3)

1. There are no structural changes required to the baseline MSS module for use in the OLS derivative.

CCM1 (Control Center Module 1)

1. Delete airlock laboratory and substitute passive docking ring.
2. Modify and relocate backup control center.
3. Add shielding (water tanks and food panels, and high density shielding) around control center to permit use as storm cellar.
4. Add sanitation facility within shielded boundary of control center.
5. Add structural supports for sleeping restraints to accommodate quiescent off-duty crew members.
6. Delete physics/materials laboratory furnishings, equipment, secondary support structures and replace with geochemistry laboratory furnishings, equipment, and secondary support structures.

CCM2 (Control Center Module 2)*

1. The internal furnishings and secondary support structure for the fluid, electronics, and optics laboratory are deleted.
2. Internal furnishings and secondary structure for the exercise/medical facility is added to this module.
3. An additional sink is added to the above facility.

GM (Galley Module)

1. Convert the 77-square foot hygiene facility to laundry (35 square feet) and storage area.

*Reference MSS growth configuration

2. Delete furnishings and secondary structure to reduce galley and dining area from 200 to 145 square feet.

3. Add furnishings and secondary support structure to increase recreation area from 60 to 80 square feet.

PM (Power Module)*

1. Utilize two MSS tanks for LO_2 and four for LN_2 with appropriate modifications to distribution system.

CSM-2 (Cryo Storage Module - Aft) (Derivative of CM)

1. Reduce length of pressurizable compartment to provide greater unpressurized compartment volume for accommodation of cryogenic and storable propellant tanks.

2. Add structural support for cryogenic and storable propellant tanks.

3. Add five 70-inch diameter cryogenic storage tanks; three LH_2, one LO_2, and one LN_2.

4. Add two cylindrical hydrazine tanks 45 inches in diameter by 105 inches long.

CSM-1 (Cryo Storage Module - Fwd) (Derivative of CM)

1. Reduce length of pressurizable compartment to provide greater unpressurized compartment volume for accommodation of large cryogenic storage tanks.

2. Add structural support for cryogenic storage tanks.

3. Add three 70-inch diameter cryogenic storage tanks for LH_2 and one 48-inch diameter tank for LO_2.

6.1.2 DESIRED CHANGES

This section delineates the structural changes which, from the OLS point of view, are considered desirable, but not absolutely essential.

C1A (Core Module 1A)

1. The deployable RCS nozzle clusters located at the extended diameter (14-foot) end are deleted. The opening in the conical end closure would be closed out on the outer surface.

2. Delete 4000 psia N_2 accumulators and replace with 100 psia accumulators.

*Reference MSS growth configuration

C1B (Core Module 1B)

1. The deployable RCS nozzle cluster located at the extended 14-foot diameter end are deleted exactly as in C1A.

2. Delete 4000 psia N_2 accumulators and replace with 100 psia accumulators as in C1A.

PM (Power Module)

1. Delete cryogenic storage tanks, secondary support structure and distribution system.

2. Remove boom structure (14-foot diameter section) provided for cryogenic tank accommodation. Maintain constant diameter full length of power boom.

CSM-2 (Cryo Storage Module (Derivative of MSS Cargo Module)

1. Increase length of unpressurized compartment to accommodate five cryogenic tanks, 71.5 inches in diameter, and two storable monopropellant tanks. The cryogenic tanks include three LH_2, two LO_2, and one LN_2.

2. Add new secondary support structure for cryogenic tanks.

3. Add two 25-inch diameter GN_2 tanks for N_2H_4 expulsion.

CSM-1 (Cryogenic Storage Module) (Derivative of MSS Cargo Module)

1. Increase length of unpressurized compartment to accommodate five cryogenic tanks, 71.5 inches in diameter. The cryogenic tanks include three LH_2, one LO_2, and one LN_2.

2. Add new secondary support structure for the cryogenic tanks.

6.2 ENVIRONMENTAL CONTROL/LIFE SUPPORT SUBSYSTEM

6.2.1 REQUIRED CHANGES

Required changes to the MSS involve the following assemblies: atmospheric storage, atmospheric control, thermal control, hygiene, and food management.

In atmospheric storage, high pressure nitrogen and oxygen tanks must be added for emergency leakage plus oxygen metabolic consumption. Rescue is a minimum of 30 days for the OLS. The MSS does not require this nitrogen because rescue or resupply can be accomplished in two days before the nitrogen loss noticeably affects the total pressure. Also, because OLS requirements differ, the high pressure oxygen tanks, cryogenic oxygen tanks, and cryogenic nitrogen tanks must be resized. The six-man Earth Orbital MSS has ten 5-foot docking ports which make up the major portion of the MSS leakage. The MSS leakage is in the 20 to 30 pounds per day range. The derivative OLS leakage is 52 pounds per day which is caused in large part by the nineteen 5-foot docking ports. Thus, the major change in tank sizing is caused by the higher leakage rates.

A slight change in oxygen recovery exists. Leakage oxygen will be made up by RCS storage and not water electrolysis. The reason for this is that the availability of the slight amount of excess water in the water balance is not certain. Efficiency of the laundry machine is uncertain. Thus, being conservative, it was assumed that excess water was not available for routing to the water electrolysis unit for production of leakage oxygen.

Regeneration units for regenerating the trace contaminant control charcoal must be added in the atmospheric control assembly. The MSS has non-regenerable units in order to hold development costs down. However, for lunar orbit, the resupply of charcoal would cost more over ten years than the investment in the regeneration equipment.

In the lunar thermal environment, a redesign of the radiators is required. ENPS - thermal analysis covers this area (see Section 6.7).

The hygiene assembly requires addition of a laundry for clothes washing. The resupply of clothing to lunar orbit incurs a greater cost than for earth orbit. A laundry concept incurs less resupply and is more flexible than a disposal clothes concept. In food management, a dishwasher is required to decrease utensil resupply. Again, the distance to lunar orbit incurs larger costs for resupply. Also, the major reason for disposable utensils on the MSS was development costs of the dishwasher.

In addition to the above equipment additions, storage areas for clothing and utensils will decrease slightly while storage for food will increase.

All equipment based upon six men processing rates must be resized for eight men.

6.2.2 DESIRED CHANGES

No additional assembly changes, other than those recommended in Section 6.2.1, are desired.

6.3 ELECTRICAL POWER SUBSYSTEM

The electrical power subsystem's (EPS) performance requirements for the earth orbiting MSS and the OLS are compared in Table 5-4 in Section 5.3. In the table, most EPS requirements are similar although the average power requirement of the MSS is 15 kilowatts (6-man) or 25 kilowatts (12-man) compared to 20 kilowatts (8-man) for the OLS. Another significant difference is that the OLS emergency/auxiliary power source is required to provide power during eclipses of the moon by the earth (a period of high scientific activity) and during solar flare storms. In lunar orbit, both of these events require an auxiliary power source as the solar-array primary power source is (1) ineffective during eclipses and (2) must be rolled up to protect the array during solar flares.

6.3.1 REQUIRED CHANGES

In spite of the difference between MSS and OLS performance requirements, the 25-kilowatt capacity 12-man MSS EPS, although not ideally suited for OLS use, could be employed as the derivative OLS EPS without modification.

6.3.2 DESIRED CHANGES

Because of the differences between the MSS and OLS requirements, and the increased resupply costs associated with lunar orbit operations, it can be advantageous to modify the MSS EPS in several ways.

Minimum Change Concept

Because the 12-man MSS EPS capacity (25 kilowatts) is oversized in certain areas for the OLS (20 kilowatts), the following low cost changes are recommended.

1. Reduce the number of solar array and battery/fuel cell inverter-regulator modules from 60 kilovolt-amperes to 48 kilovolt-amperes and 40 kilovolt-amperes to 20 kilovolt-amperes, respectively

2. Reduce the capacity (weight) of the primary batteries by ~63 percent or 2375 pounds of batteries

3. Reduce the capacity of the emergency fuel cells from 7.5 kilowatts to 2.5 kilowatts

4. Reduce the capacity of the secondary batteries and chargers by approximately 45 percent (compared to MSS - 25 kilowatts EPS configuration requiring 10,165 pounds of batteries)

Extensive Change Concept

The minimum change concept described above would, in comparison to the representative OLS EPS described in paragraph 3.3, have high initial weight, cost, maintenance and resupply weight. A more ambitious and desirable MSS EPS modification concept involves major vehicle rewiring, component substitution and possible further reduction in power conditioning modules.

The extensive change concept involves replacement of the battery energy storage concept and removal of the excess capability of the MSS. The steps to be taken are as follows:

1. Replace the battery-battery charger energy storage concept with regenerative fuel cells (RFC).

2. Eliminate the double ac voltage distribution concept (transformer-rectifier to replace the auto-transformer and rectifier-filter).

3. Reduce the capacity of the primary batteries by approximately 60 percent.

4. Reduce the number of solar array inverter-regulator modules from 60 kilovolt-amperes to 40 kilovolt-amperes.

5. Reduce the capacity of the emergency fuel cells from 7.5 kilowatts to 2.5 kilowatts.

6. Reduce the number of emergency power system inverter-regulator modules from 40 kilovolt-amperes to 20 kilovolt-amperes.

7. Reduce feeder sizes and change runs.

These changes would result in a significant reduction in weight, program cost, maintenance, and resupply. As indicated in paragraph 3.2, the preferred concept for the OLS was RFC's. Weight, volume, resupply, heat rejection, maintenance, and cost all are less for RFC's. In addition, the modificiations to the thermal protection system of the power module (see paragraph 6.7.2) could be eliminated. The power module for the derivative OLS would be identical to the power module for the representative OLS if RFC's were incorporated.

6.4 INFORMATION SUBSYSTEM

The Information Subsystem (ISS) of the MSS is initially sized for a 6-man crew with the capability of growing to a 12-man baseline concept by adding identical assemblies. The ISS is modular so that capabilities can be incrementally added or deleted as mission operations dictate, without changing the baseline concept or implementation approach. This modular approach assists in the conversion from MSS to OLS but there are some modifications that are required or desired and are discussed in the following paragraphs.

6.4.1 REQUIRED CHANGES

Data Processing Assembly

The size and speed of data processing assembly (which strongly influences cost) is largely a function of experiment support requirements. The major component affected is the mass memory capacity. If the memory capacity needed for experiment-unique data is subtracted from the total requirement, the remainder can be accommodated in two mass memory modules (512 K words). All of the remainder relates to the experiment data base and experiment operations. The location of mass memory modules is not critical, and the quantity can be increased as needed; therefore, for the MSS the mass memory modules needed for experiment-unique data can be located within, and become part of the Research Applications Modules (RAM's) concerned. The RAM's however are not utilized by the derivative OLS. Therefore the OLS ISS mass memory modules required for experiment support must all be accommodated within the OLS core, control, or crew quarters modules.

In all, 13 mass memory modules, 11 more than are accommodated within the MSS core, control center, or crew quarters modules, are required for OLS station, subsystem, and experiment operation and control. Each mass memory module can store 256 K words of data. As in the MSS, location of the mass memory modules is not critical, however, for ease of general maintenance and repair, the modules should be installed in a pressurizable area that is relatively accessible.

Display/Control

The control center display/control consoles can be reduced to one console in each control center by recognizing that any mission, vehicle, and most experiment operations can be controlled from any one console. Four consoles are provided in the MSS 12-man concept to support the multiplicity of operations a 12-man crew could handle. For the 6-man MSS, the number of simultaneous multiple operations possible is correspondingly less and only two consoles are provided.

The OLS 8-man baseline specifies that 3 command consoles are required. Since space is allocated for an additional console adjacent to the command console in MSS Control Center Module 2 in the 6-man configuration, the impact of providing these consoles for the derivative OLS does not appear significant.

Communications

Incorporation of the parabolic antenna subassembly and a data relay satellite system for MSS can be postponed by accepting the following operational constraints:

a. short periods of TV/voice/data communications to ground,
b. ground tracking of free-flying modules and shuttle vehicles, and
c. communication interruptions exceeding 4-1/2 hours.

The parabolic antenna subassembly (13 foot dish, boom, electronics, drive, and gimbals) may be added to the MSS any time prior to operations with free-flying RAM's. The antennas will be located on the MSS Crew Quarters Modules 3 and 4.

For the OLS, however, communications via high-gain antenna is a necessity, beginning with initial station operation. Sizing factors for the required OLS high-gain antenna may be expressed in terms of a standard parabolic dish. Calculations presented in Volume IV, the Configuration and Systems Analysis Report, show that the 13-foot diameter MSS parabolic dish antennas would be more than adequate for OLS use. In fact communications from the OLS with earth through MSFN 85-foot antennas will have performance equal to MSS operations with only a 5-foot parabolic dish mounted on the OLS. If the MSS 13-foot antennas were used in the OLS the standard 30-foot MSFN antennas would be adequate.

Antenna location is primarily a function of station configuration and desirability to minimize blockage due to the solar arrays. Regardless of the placement of the two MSS antennas, some shadowing is unavoidable in the forward looking direction. Shadows are minimized by placing the antennas as far from the solar arrays as practical, yet with only two antennas there will exist communication blackouts. This problem can be rectified by extending a third parabolic antenna forward from the Power Module. This would afford complete spherical communications coverage and eliminate communications dead zones.

The OLS configuration does not include a Crew Quarters Module 4 and the location of Crew Quarters Module 3 on the OLS is not compatible with the desired communications coverage for one antenna. Therefore, high gain antennas must be added to OLS modules. The preferred location of these antennas are on Crew Quarters Module 1 and Control Center Module 1.

As stated above only a 5-foot dish is required for OLS operation. As the 5-foot and 13-foot dishes are both state-of-the-art, development consists of integration effort only. A significant weight savings can also be

affected. Based on the data in Figure 6.4-1, replacing the 13-foot diameter MSS dishes with 5-foot diameter dishes reduces the weight for a single antenna subassembly from 130 pounds to 30 pounds. Figure 6.4-1 defines antenna weight as a function of antenna diameter. The total diameter weight includes the weight of the antenna, feed, drive, gimbals, and electronics. The structure required to support the antenna is a function of required length and is not included in the figure. Values are based on NASA Technical Report TR R-309, dated April, 1969. The weight saving achieved by equipping the OLS with a 5-foot antenna subsystem rather than with the MSS 13-foot system is therefore 200 pounds, not including the weight of a supporting boom structure. Table 6.4-1 summarizes the characteristic weights for deployable 13-foot and 5-foot parabolic antennae with 30-foot booms. Note that a 5-foot antenna subassembly weighs significantly less than a 13-foot subassembly. Based on these data a total weight saving of 334 pounds could be achieved by equipping the OLS with a 5-foot rather than a 13-foot high gain antenna assembly.

Table 6.4-1 OLS Antenna Weight Characteristics

13' Parabola		5' Parabola	
• Dish + Feed + Electronics + Drive + Gimbals	130#	• Dish + Feed + Electronics + Drive + Gimbals	30#
• Boom		• Boom	
30' length (folding)	167#	30' length (folding)	130#
folding mech.	50	folding mech.	30
activation mech.	60	activation mech.	50
Subtotal	277	Subtotal	210#
TOTAL	407#	TOTAL	240#

Based upon the above considerations, the selected antennas for the derivative OLS are 5-foot diameter parabolic dishes.

6.4.2 DESIRED CHANGES

There are no desired modifications to the information subsystem of the MSS.

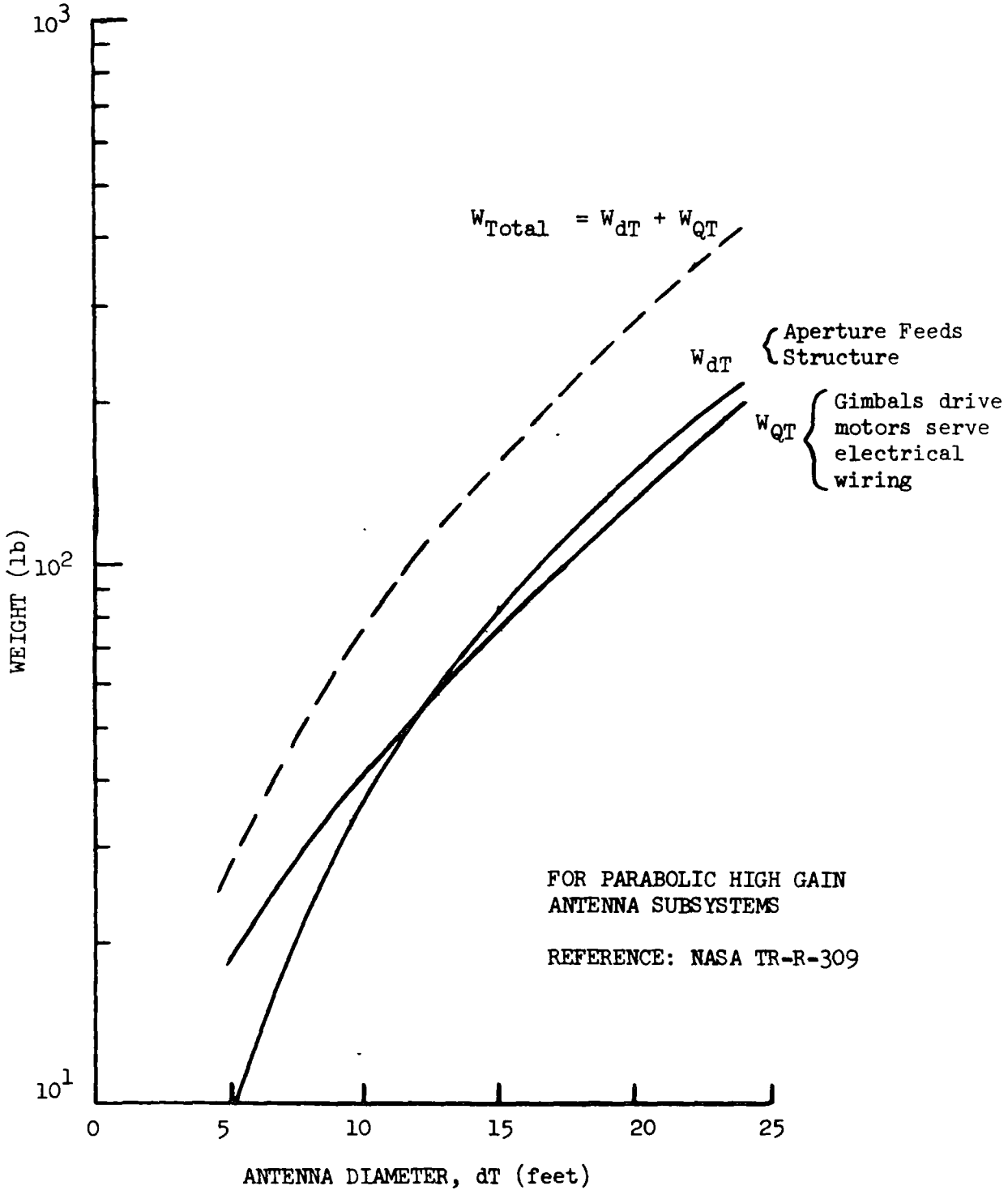


Figure 6.4-1. Antenna Subsystem Weight Versus Antenna Diameter

6.5 GUIDANCE AND CONTROL SUBSYSTEM

6.5.1 DETERMINATION OF FLIGHT MODE

In order to determine the required modifications to the MSS guidance and control subsystem (G&C) it was first necessary to determine the derivative OLS flight mode and vehicle configuration characteristics in the same manner as for the representative OLS discussed in Section 3.4. Figure 6.5-1 illustrates the vehicle configurations considered. Table 6.5-1 lists the weights of the modules used for this analysis. The first three configurations were investigated in all the possible local level modes. The results are shown in Tables 6.5-2, 6.5-3, and 6.5-4. It was determined that the required effective radiator area could not be provided (utilizing body mounted radiators) in the Z-POP X and X-POP Z flight modes and these were eliminated from further consideration. The heat rejection characteristics of the Y-POP Z and Z-POP Y configurations were not analyzed in detail, but preliminary examination indicated that due to module to module view factor heat rejection capability would be quite marginal. Therefore, these two modes were also excluded. Examination of the two remaining modes, X-POP Y and Y-POP X, from a desaturation impulse and CMG size standpoint, indicates that X-POP Y is the preferred flight mode and the preferred configuration is Vehicle No. 2.

Table 6.5-1. Weights of OLS/MSS Modules

	Weight (pounds)
Core module (CM1) / (C1A)	28,220
Core module (CM2) / (C1B)	25,000
Crew quarters module (CQM3)	12,570
Crew quarters module (CQM1)	12,950
Cryogenic storage modules (CM) / (CSM 1 & CSM 2)	20,000
Control center module (CCM2)	19,270
Control center module (CCM1)	35,000
Galley module (GM)	15,590
Power module (PM) less solar array	6,600
Experiment module (XM)	11,000
Tug A - full	84,615
Tug B - full	91,900
Dual support cargo module (DSCM) - full	23,550
Tug B - empty	31,900
Dual support cargo module (DSCM) - off-loaded	8,835

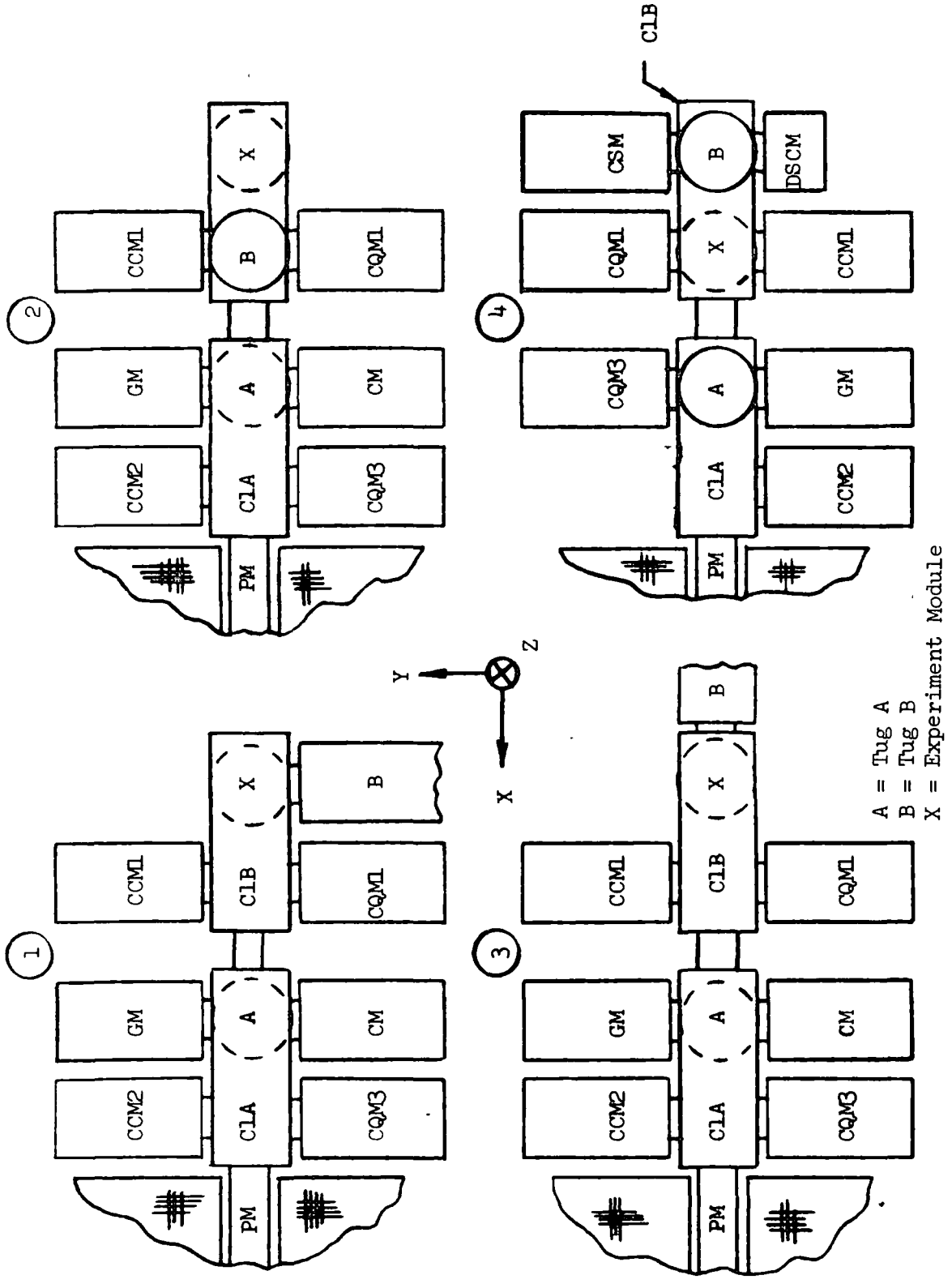


Figure 6.5-1. Vehicle Configurations

Table 6.5-2. Results of Stability and Control Analysis of Vehicle 1

Mode	RCS only Impulse (lb-secs)	Desaturation Impulse (lb-secs)	Individual gyro size (ft-lb-secs)	3-gyro weight (lbs)	3-gyro power (watts)	3-gyro volume (ft^3)	Days to Equivalence (Isp = 220)
X-POP Y	3923	466	16640	3724	238	646	24
Y-POP X	4412	625	14113	3436	228	572	20
Z-POP X	4341	1318	39900	5722	297	1234	42
X-POP Z	3989	430	16492	3708	238	642	23
Y-POP Z	1948	444	14227	3449	229	575	50
Z-POP Y	2836	1088	39752	5712	297	1231	72

Space Division
North American Rockwell

Table 6.5-3. Results of Stability and Control Analysis of Vehicle 2

Mode	RCS only Impulse (lb-secs)	Desaturation Impulse (lb-secs)	Individual gyro size (ft-lb-secs)	3-gyro weight (lbs)	3-gyro power (watts)	3-gyro volume (ft^3)	Days to Equivalence (I_{sp} = 220)
X-POP Y	1521	108	5088	2088	177	270	32
Y-POP X	1233	958	26448	4675	268	910	374
Z-POP X	795	276	2731	1543	152	171	65
X-POP Z	843	66	2863	1579	154	177	45
Y-POP Z	927	778	26405	4671	268	909	689
Z-POP Y	1405	38	4452	1957	172	244	31

Space Division
North American Rockwell

Table 6.5-4. Results of Stability and Control Analysis of Vehicle 3

Mode	RCS only Impulse (lb-secs)	Desaturation Impulse (lb-secs)	Individual gyro size (ft-lb-secs)	3-gyro weight (lbs)	3-gyro power (watts)	3-gyro volume (ft^3)	Days to Equivalence (I_{sp} = 220)
X-POP Y	2116	169	7115	2458	193	345	28
Y-POP X	1456	1072	30713	5031	278	1017	289
Z-POP X	1206	269	2957	1604	155	181	38
X-POP Z	1376	126	5065	2083	177	269	37
Y-POP Z	1062	892	30667	5028	278	1016	649
Z-POP Y	1925	.62	5210	2112	178	274	25

Space Division
North American Rockwell

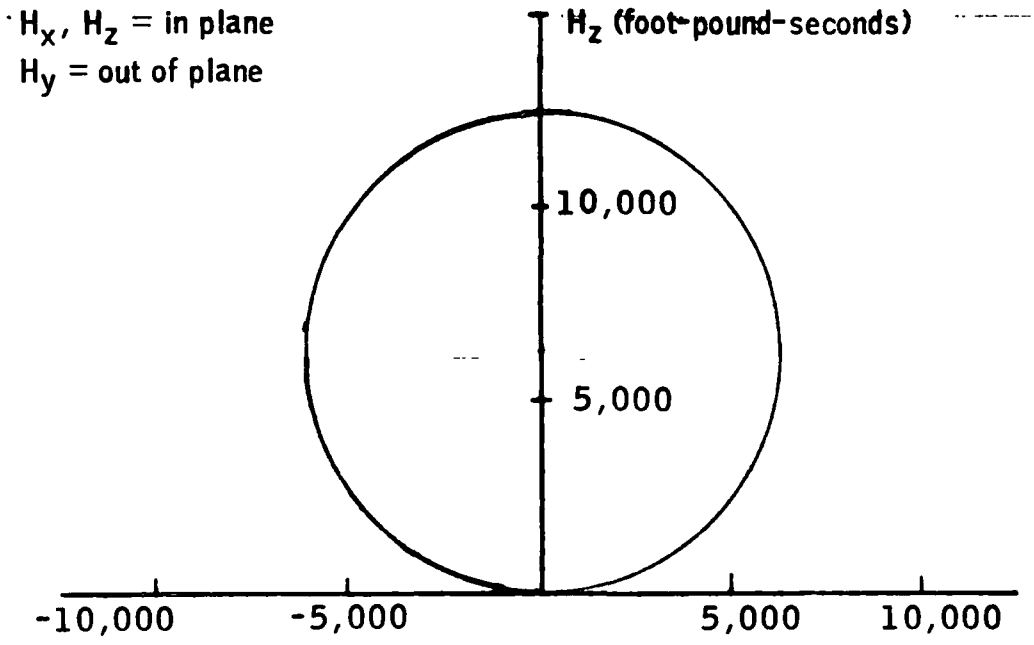
The preferred combination from Tables 6.5-2, 6.5-3, and 6.5-4, requires a very large CMG size (5088 foot-pound-seconds); therefore, further consideration was given to methods for decreasing the momentum requirements. The first consideration was to reduce momentum storage by frequent or continuous desaturation of the CMG's. Figure 6.5-2 is a diagram of the momentum history associated with this vehicle configuration. The top curve is a phase plane plot of the in-plane momentum history and the bottom curve is a plot of the out-of-plane momentum for one orbit. The in-plane momentum is seen to be almost purely periodic, whereas the out-of-plane momentum is mostly secular. Continuous CMG desaturation would not affect the in-plane components but could maintain the out-of-plane component close to zero throughout the orbit. By applying the CMG sizing criteria discussed in a previous section but maintaining the out-of-plane momentum near zero (by using continuous RCS desaturation), it can be seen that the individual CMG size can be reduced to about 4150 foot-pound-seconds (as compared to the 5088-foot-pound-seconds required for the case of desaturating every third orbit). While this represents a substantial reduction in CMG size, it is not a desirable mode of operation and may in fact, interfere with the performance of critical experiments. Further investigation of the problem, therefore, was carried out.

Noting that disturbances are mainly a result of gravity gradient torques caused by a misalignment of the principal inertia axes with respect to the local level coordinates, an investigation was made of the effects of canting the vehicle geometric axes to align the principal axes with respect to the vertical. The effect of tug placement on this vehicle was to cause a substantial inertial cross product (I_{XZ}) and, therefore, rotation about the Y-axis was investigated. Figure 6.5-3 illustrates the effect of canting the vehicle about the Y-axis. Gyro momentum, desaturation impulse, and required CMG weights are plotted as a function of the rotation angle θ. CMG weight is computed based on use of the MSS gyros (3000 foot-pound-seconds) adding gyros as required. This is considered preferable to developing larger gyros from the standpoint of cost (although not optimal from the standpoint of weight). Thus, for gyro moments less than 3000 foot-pound-seconds per gyro the three-gyro MSS array is used; for momenta greater than 3000 but less than 4500 a four-gyro array is used; for momenta greater than 4500 but less than 6000 a five-gyro array is used. The figure indicates that a gyro momentum minimum occurs at about a 14-degree offset and that the desaturation impulse is relatively flat over the range from zero to 14 degrees. Thus, the curves show an optimum cant angle of 9 to 10 degrees, which would save about 1200 pounds of CMG weight over the zero degree case. The difficulty with flying an offset angle is that the experiments module would have to be offset or gimballed to view along nadir.

In order to avoid this difficulty and still reduce CMG size, a fourth vehicle configuration was investigated. This is configuration 4 as shown in Figure 6.5-1. In this configuration both tugs are placed on top of the vehicle to reduce the inertia cross product. For this vehicle both X-POP Y and Y-POP X were investigated for each of the four cases of the 109-day cycle:

A. Both tugs fully fueled and the cargo module full

B. Both tugs fully fueled and the cargo module half full



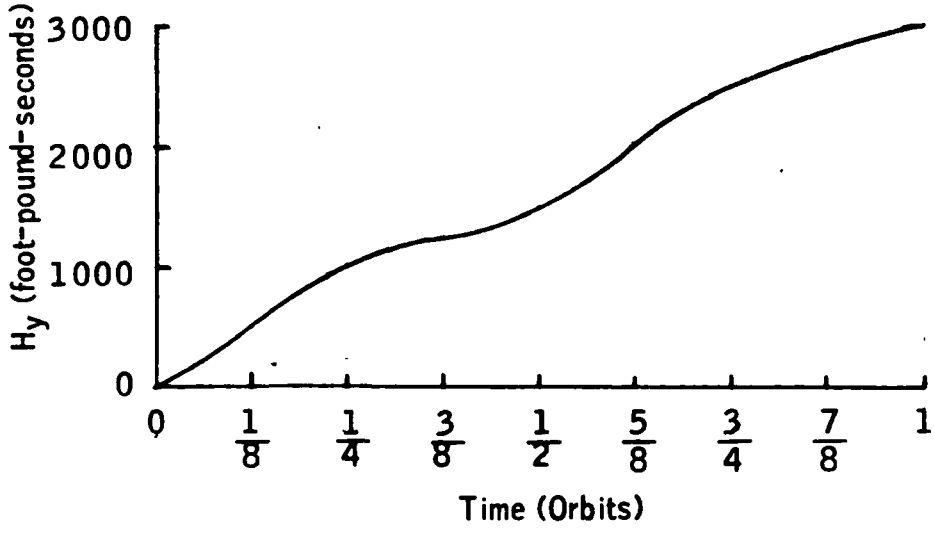


Figure 6.5-2. Momentum History for Configuration 2

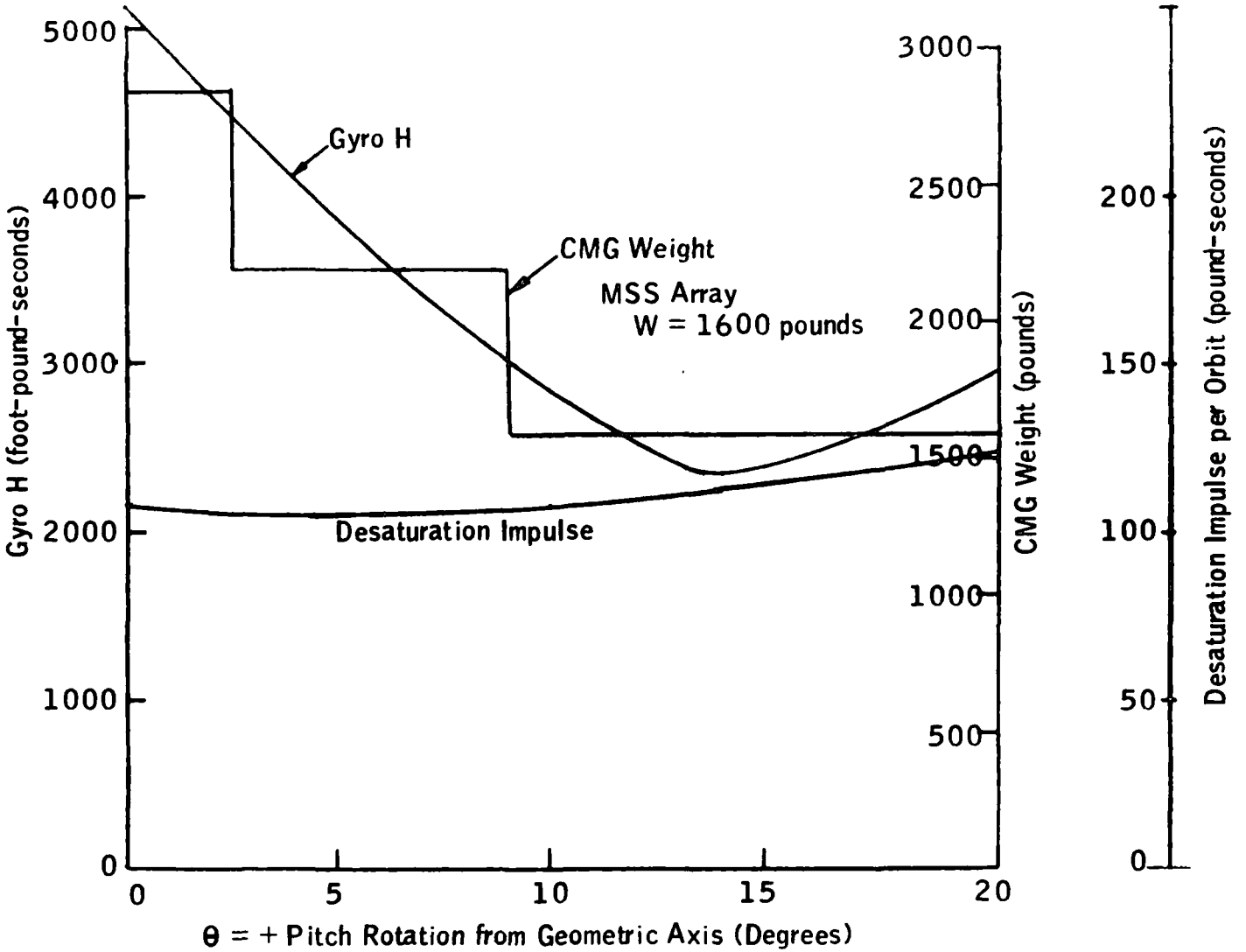


Figure 6.5-3. Effect of Principal Axis Misalignment

Space Division
North American Rockwell

C. Tug A fully fueled, Tug B empty, and the cargo module absent

D. Tug A fully fueled, Tug B, and the cargo module absent

The results for these cases are shown in Table 6.5-5. Clearly, X-POP Y is the preferred mode from the standpoint of both desaturation propellant and CMG size. Vehicle 2 was reinvestigated for the X-POP Y mode for each of the four cases as shown in Table 6.5-6, which indicates by comparison with Table 6.5-5, that Vehicle 4 is superior to Vehicle 2 in all four cases (the slight difference in numbers between Case A of Table 6.5-6 and the X-POP Y mode of Vehicle 2 shown in Table 6.5-3 is due to a slight rearrangement of modules).

6.5.2 SELECTED CONFIGURATION

Vehicle 4 flown in an X-POP Y mode is preferred in terms of both desaturation propellant and CMG size. The CMG size of 2600 foot-pound-seconds is sufficiently close to the MSS CMG's (3000 foot-pound-seconds) that the MSS CMG array can be used directly. In this configuration both tugs are placed on top of the vehicle, the geometric Z-axis is aligned with the local vertical and the geometric Y-axis points down range.

6.5.3 REQUIRED G&C CHANGES

From the above discussion it is concluded that no change is required to the MSS CMG array. The only other required changes are inclusion in MSS core module 1B of a landmark tracker and radar altimeter together with modification of the horizon tracker operating frequency and the removal and relocation of some G&C equipment in OLS core modules 1A and 1B.

The landmark tracker and radar altimeter are required to provide the tighter navigation accuracy caused by the lower orbital altitude. The radar altimeter could be the same device as utilized by the experiments but the landmark tracker with its associated software is a new development item. The horizon tracker modification involves only a change of operating frequency. The horizon tracker used in earth orbit operates most efficiently in the 14 to 16 micron infrared band, whereas the lunar horizon tracker operates more efficiently at around 40 microns because of the lack of atmosphere. This is considered a minor conversion. Equipment relocation is necessary because of the different vehicle configuration and different location of the experiment module. Thus, all G&C equipment (with the exception of one set of RCS driver electronics) is located in core module 1B near the experiment module (see Section 5.5 and Figure 5-1). This is permissible because, unlike the MSS, there is no operational requirement for the OLS G&C to stabilize core module 1A when it is not attached to core module 1B. An additional requirement is that the landmark tracker and inertial reference assembly be mounted on a common navigation base with a continuously active optical calibration link with the star trackers and horizon scanners. This necessitates mounting all these equipments at approximately the same X coordinate. All G&C equipment except one set of RCS electronics (and preprocessor) should, therefore, be removed or not installed in core module 1A.

Table 6.5-5. Results of Stability and Control Analysis of Vehicle 4 by Mission Phase

Case	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 (I_{sp} = 220 sec)
A	X-POP Y	755	73	2,189	1,386	144	145	45
	Y-POP X	1054	414	6,341	2,324	187	317	80
B	X-POP Y	799	40	1,932	1,305	139	132	38
	Y-POP X	869	480	8,710	2,713	203	400	153
C	X-POP Y	597	53	1,448	1,136	130	107	46
	Y-POP X	728	377	5,006	2,071	177	266	130
D	X-POP Y	948	50	2,604	1,508	150	165	37
	Y-POP X	988	642	14,684	3,503	231	589	223

Table 6.5-6. Results of Stability and Control Analysis of Vehicle 2 by Mission Phase

Case	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 (I_{sp} = 220 sec)
A	X-POP Y	1,619	112	5,135	2,097	178	271	31
B	X-POP Y	1,668	116	5,237	2,117	179	275	30
C	X-POP Y	1,382	124	4,441	1,954	171	244	34
D	X-POP Y	1,174	131	4,178	1,897	169	233	40

6.5.4 DESIRED G&C CHANGES

There are no desired modifications to the MSS other than the required changes mentioned above.

Page intentionally left blank

6.6 REACTION CONTROL SUBSYSTEM

The Modular Space Station (MSS) reaction control subsystem (RCS) is designed to provide a total impulse of 601,000 pound-seconds (see Table 6.6-1) over 120 days, while the derivative OLS in lunar orbit must provide 2,744,000 pound-seconds over 180 days. In addition, both station concepts include the cryogenic storage for the ECLSS and the EPS. The OLS requires a much greater cryogenic storage capacity for ECLSS and EPS as indicated in Table 6.6-2. All of these factors add up to one thing, the cryogenic storage capability of MSS is not adequate for the derivative OLS.

Table 6.6-1. Propulsion Requirements Comparison

	MSS (120 days)	Derivative OLS (180 Days)
CMG desaturation	50,800 lb-sec	92,000 lb-sec
Maneuvers	8,000 lb-sec	12,000 lb-sec
Orbit maintenance	447,200 lb-sec	2,559,000 lb-sec
Docking	14,000 lb-sec	0
Total	520,000 lb-sec	2,663,000 lb-sec
Emergency	81,000 lb-sec	81,000 lb-sec
Total	601,000 lb-sec	2,744,000 lb-sec
For Subsatellites		
N_2 Storage		455 lbm
N_2H_4 Storage		7811 lbm

The actual design loading requirements for the derivative OLS were determined in the same way as those for the representative OLS in Section 3.5. There must be sufficient cryogenic fluids for 180 days of normal usage or 109 days plus the contingencies. The contingencies considered were those requiring the greatest amount of fluid in each case. Two types of contingencies were considered. One was a puncture failure. This combined the requirements of repressurization and the reaction control emergency (to counter torques and displacements caused by the atmospheric gases escaping through the hole). The other was a power failure. The power failure set the contingency requirements for the oxygen and hydrogen. The puncture (repressurization) contingency established the nitrogen requirement. These design criteria are outlined in

Table 6.6-2. Subsystem Integration Comparison

		MSS (120 Days)	Derivative OLS (180 Days)
ECLSS			
Leakage and metabolic	O_2	0	2394 lbm
	N_2	1850 lbm	6894 lbm
Sabatier unit	H_2	72 lbm	324 lbm
Repressurization	O_2	350 lbm	350 lbm
	N_2	1150 lbm	1150 lbm
EPS			
Emergency power*	O_2	176 lbm	3697 lbm
	H_2	22 lbm	533 lbm

*Emergency power by fuel cells is required for two days in the MSS and 30 days for the derivative OLS

in Table 6.6-3. As may be seen in the table, the 109 days plus the contingency requirement sized the oxygen loading at 8042 pounds. The 180 days of normal usage requirement sized the nitrogen loading at 7349 pounds, and the hydrogen loading at 1913 pounds.

6.6.1 REQUIRED CHANGES

Assuming that the cryogenics are stored equally in two locations (for safety), there would be an even number of each type of tank. As discussed in Section 3.5, the tanks for oxygen, hydrogen, and nitrogen would optimally all be of the same design to save development cost and time. The vapor shield tanks used in the MSS are not adequate for hydrogen, however, because of their small size (25.5 cubic feet each). In small tanks, the surface area to volume ratio is too great, and the cryogenic hydrogen would boil off before 180 days have elapsed. On the other hand, the six cryogenic tanks in the power boom are adequate for oxygen and nitrogen storage. The MSS power boom tanks thus are utilized as part of the RCS general cryogenic storage.

Table 6.6-3. Derivative OLS Propellant and Nitrogen Storage Summary (lbm)

	O_2	H_2	N_2	N_2H_4
RCS - 180 days	4767	1589	-	-
ECLSS - 180 days	2394 +	324	6894	-
Total	7161	1913*	6894 *	
RCS - 109 days	2888	962	-	-
ECLSS - 109 days	1457 +	196	4175	-
Contingencies	3697	533	1150	-
Total	8042 *	1691	5325	-
Subsatellite provisions			455 *	7811 *
Required loading (sum of * items)	8042	1913	7349	7811

+Includes extra 18 lbm metabolic load for crew ovelap

Considering first the the aft storage area, it is possible to size new storage tanks that are optimum for OLS use. An analysis shows that for common size tankage, the required volume of the new tanks will be 89.8 cubic feet each. This design is based upon the assumption that four of the 25.5-cubic foot tanks in the boom assembly are used for nitrogen. The remaining nitrogen storage would be provided by one tank in the aft storage area. Using this larger size there will be one oxygen tank, three hydrogen tanks, and one nitrogen tank in the aft storage area. There will also be three of these same tanks for hydrogen storage in the forward cryogenic storage area. The two remaining storage tanks in the power boom will be used for oxygen. It will be necessary to install one additional 25.5-cubic foot tank for oxygen in the forward storage area. The tank sizing is summarized in Table 6.6-4. As indicated in the table, the tank sizing is based on the required volume for hydrogen. Therefore, there is a resultant excess capacity of 1698 pounds of oxygen and 193 pounds of nitrogen.

The method of implementation will be to use half of the cryogenic storage in the MSS cargo module and convert it to provide an unpressurized storage area for the OLS cryogenics. The Dewar storage tanks will be removed and replaced with five 89.8-cubic foot vapor shield tanks. This, then, will be the aft cryogenic fluid storage module (CSM 2). A similar module (CSM 1) will be docked to the forward end of the OLS core module 1A, refer to Figure 5-1. The CSM 1 along with the power boom cryogenic storage area

Table 6.6-4. Derivative OLS Cryogenic Fluid Tankage - Required Modification

Cryogenic Fluid	Required Load (lbm)	Number of Tanks*	Possible Load (lbm)	Possible Excess (lbm)
O_2	8042	3 small, 1 large	9740	1968
H_2	1913	6 large	1913	0
N_2	7349	4 small, 1 large	7542	193

*Small tanks are 25.5 cubic feet from MSS configuration.
Large tanks are 89.8 cubic feet based on zero excess of hydrogen.
One additional small tank is added for oxygen in the forward, permanently docked cargo storage module.

that has been replumbed for oxygen and nitrogen storage only, will constitute the forward cryogenic fluid storage area. The aft cryogenic storage area, CSM 2, will also be the location of the subsatellite hydrazine storage tanks.

Both of the core modules, the solar arrays, and at least one complete cryogenic storage module can be brought to the moon on the first RNS. For this reason, the cold gas nitrogen system used by the MSS for stabilization during incremental buildup stage will not be needed. It is not necessary to remove it, however, and it can be used by the derivative OLS. The nitrogen compressors will be in core module 1B and four high pressure tanks are in each core module. All of the high pressure lines must be shielded for safety. Two of the four high pressure nitrogen tanks in core module number 1B of the MSS can be used in conjunction with the subsatellite hydrazine for positive expulsion.

The accumulator requirements are also changed. Based on two orbit maintenance burns a day (both in the same orbit), with two engines thrusting at a time, each pair of engines used must be capable of burning for 5.9 minutes at 10 pounds of thrust each. The remaining six high pressure MSS nitrogen tanks would be used as oxygen accumulators to meet this requirement. Four 100 psia hydrogen accumulators must be added to complete the accumulator requirements. They will be located in core modules 1A and 1B.

6.6.2 DESIRED CHANGES

Core module 1B is basically a core module 1A mounted backward to the aft end of core module 1A. This results in four excess engine clusters in the center of the two core modules. The deletion of these four clusters will provide a mass savings of at least 120 pounds plus the associated propellant feed controls, plus the cost of this hardware.

The eight high pressure nitrogen accumulators are each larger than is required for the hydrazine expulsion requirements. They also are designed to hold 40 times the pressure required for propellant accumulators. The deletion of these tanks will provide a weight savings on the order of 2000 pounds and a reduction in volume of approximately 38 cubic feet. In their place will be added a pair of 25-inch diameter high pressure tanks in the CSM 2 to expel the hydrazine, and four sets of 100 psia accumulators in core modules 1A and 1B adjacent to the four remaining engine clusters.

Since there must be at least two permanently docked cryogenic storage modules on the derivative OLS, and since larger vapor shield storage tanks must be built, the power boom storage area constitutes an inefficient use of space and excessive mass. It would be desirable to remove these assemblies and place all of the storage facilities in the two docked modules. In doing this, there would be one oxygen, three hydrogen, and one nitrogen tank in each module (see Table 6.6-5). Each tank would have an internal volume of 99.8 cubic feet. There would be an excess capacity of 3277 pounds of oxygen and 499 pounds of nitrogen.

Table 6.6-5. Derivative OLS Cryogenic Fluid Tankage - Desired Modification

Cryogenic Fluid	Required Load (lbm)	Number of Tanks*	Possible Load (lbm)	Possible Excess (lbm)
O_2	8042	2	11,319	3277
H_2	1913	6	1,913	0
N_2	7349	2	7,848	

*Based on 99.8-cubic foot tanks with 10% ullage

Page Intentionally Left Blank

6.7 ENVIRONMENTAL PROTECTION SUBSYSTEM

6.7.1 METEOROID PROTECTION

Required Changes

The meteoroid protection requirements for the MSS and derivative OLS are identical (see Table 5-8). While the MSS meteoroid protection design concept has not been selected, it is presumed that the meteoroid protection concept provided by the MSS would be adequate for the OLS. Optimum sizing of the meteoroid bumper(s) for the OLS could be lighter than the 12-man MSS because of the small difference in meteoroid environment and the smaller area-time product of the OLS, but it appears likely that manufacturing and handling constraints will be the governing factor in selecting the minimum thickness. Therefore, there are no required changes to the MSS meteoroid bumper concept.

Desired Changes

As stated in the preceding paragraph, MSS meteoroid protection is compatible with OLS requirements and therefore no desired changes are identified.

6.7.2 THERMAL PROTECTION

Earth and lunar thermal environments were compared in Section 3.6.2. Based on the lunar environment, desirable radiator locations were determined to be on antilunar facing surfaces ($90^\circ \leq \gamma \leq 270^\circ$). The MSS designed for earth orbit employs an active 50 F fluid loop radiator system utilizing full 360 degrees circumferential radiators on four modules. The MSS does not require a heat pump or an auxiliary thermal capacitor. The total radiator area required with this configuration is 3520 square feet which allows for α_s/ϵ degradation to 0.4 before heat rejection requirements exceed radiator capability.

Due to severe infrared heating from the lunar surface, radiators on nadir oriented surfaces are useless. It is recommended that radiators be located in the 120-degree sector centered opposite to the nadir location (The effectiveness of the area between $90^\circ \leq \gamma \leq 120^\circ$ and $240^\circ \leq \gamma \leq 270^\circ$ is reduced considerably by radiation from the adjacent module). The five MSS radiator-equipped modules contain a total 1463 square feet of radiator in the recommended location. This area is insufficient to satisfy OLS heat rejection requirements in lunar orbit at the subsolar point for α_s/ϵ degraded to a value of 0.3.

Three techniques (previously discussed in Section 3.6.2) can be employed separately or in combination to augment the MSS thermal control subsystem (TCS) to satisfy heat rejection requirements: (1) radiator

area addition in effective locations ($120^{\circ} \leq \gamma \leq 240^{\circ}$); (2) employ thermal capacitance to store heat while near the subsolar point; and (3) employ a heat pump so that a 70 F radiator can be used. Four viable combinations of these techniques are listed in Table 6.7-1.

Table 6.7-1. Thermal Control Subsystem Change Options

TCS Option Number	Radiator Area Increase	Effective Radiator Temperature (Degrees F)	Thermal Capacitance Required	Heat Pump Required
1	No	50	Yes	No
2	No	70	Yes	Yes
3	Yes	50	Yes	No
4	Yes	70	Yes	Yes

Based on studies of the representative OLS documented in Section 3.6.2, the following factors appeared as significant drivers to the conclusion that Option No. 4 in Table 6.7-1 be identified as the required derivative OLS TCS.

1. In the location of $120^{\circ} \leq \gamma \leq 240^{\circ}$, where effective radiator area can be added, substituting radiator area for thermal capacitance weight is effective based on total weight change (refer to Figure 3.6-11).

2. Thermal capacitance is required for all the combinations because heat rejection capability at the subsolar point will be considerably less than the heat rejection requirement. Note from Figure 3.6-8 that 2000 square feet of 70 F radiator ($60^{\circ} \leq \gamma \leq 300^{\circ}$) can reject only 29,000 Btu/hour. For simplicity, all rejection can be optimistically attributed to the 1300 square feet between $120^{\circ} \leq \gamma \leq 240^{\circ}$. Therefore, more than three times as much radiator area, >3900 square feet, will be required because the 90,000 Btu per hour being generated is more than three times the rejection capability.

3. A heat pump will allow the capacitor to assume the nominal module temperature (approximately 65 F to 70 F), thereby considerably reducing the complexity of the thermal design.

4. A heat pump, by essentially increasing the effective radiator temperature, increases the heat rejection capability.

Some performance characteristics of Option 4 are discussed in the following paragraphs. The effect of adjacent modules and the attendant view factors which reduces usable surface area for radiators, rejection requirements, rejection capability per module, and the interconnection of module radiator assemblies are covered.

The total heat rejection requirement for the derivative OLS is 27.6 kilowatts, instead of the 26.6 kilowatts of the representative OLS, in addition to a 1.1-kilowatt load generated by the batteries in the power module which is transported to the manned module TCS. To accommodate this and other battery loads from manned module housed batteries, the heat pump will be sized to a 4.3 kilowatts conversion.

The OLS-MSS modular configuration is depicted in Figure 5-1, which shows a view of the side of the OLS away from the lunar surface. The maximum heat loads on a per module basis are shown in Table 6.7-2. Since the requirements represent maximums for each module and do not occur at the same time, the loads are not additive.

Table 6.7-2. Nonadditive Heat Loads

Module	Heat Load (kw)
CQM1	7.2
CQM3	7.2
GM	8.0*
CCM1	4.3
CCM2	4.3
Core 1	4.8
Core 2	4.8
Cargo Module	0.4

*During cooking only (2 hours maximum)

With the modular configuration, more surface area is available for radiators than with the representative OLS. Hence, there is more area available in the prime rejection locations which do not receive lunar emission. However, the rejection capability of a radiator surface is reduced somewhat by the thermal radiation it receives from adjacent modules, the core module, and docked tugs. The view factor (F) from an incremental area on the surface of one module to an adjacent parallel module is highest, F = 0.53, at the point of closest proximity, or γ = 90 degrees (γ is the angle from the nadir location). The view factor is 0.37 at γ = 120 degrees and decreases rapidly as γ approaches 180 degrees. Since at $\gamma < 120$ degrees, the radiator area is at best about 60 percent effective due to the view factor to an adjacent module at the same termperature, the radiators are located between γ = 120 degrees and γ = 240 degrees.. The heat rejection capability per module with 120 degrees sector radiators along its entire length (293 square feet total area per module) is shown in Figure 6.7-1 for a 70 F external active radiator. The curves bracket the rejection capability with the values at the subsolar point and the average rejection over an orbit. The curves of Figure 6.7-1 show maximum heat rejection rates since influence of other spacecraft surfaces has been ignored.

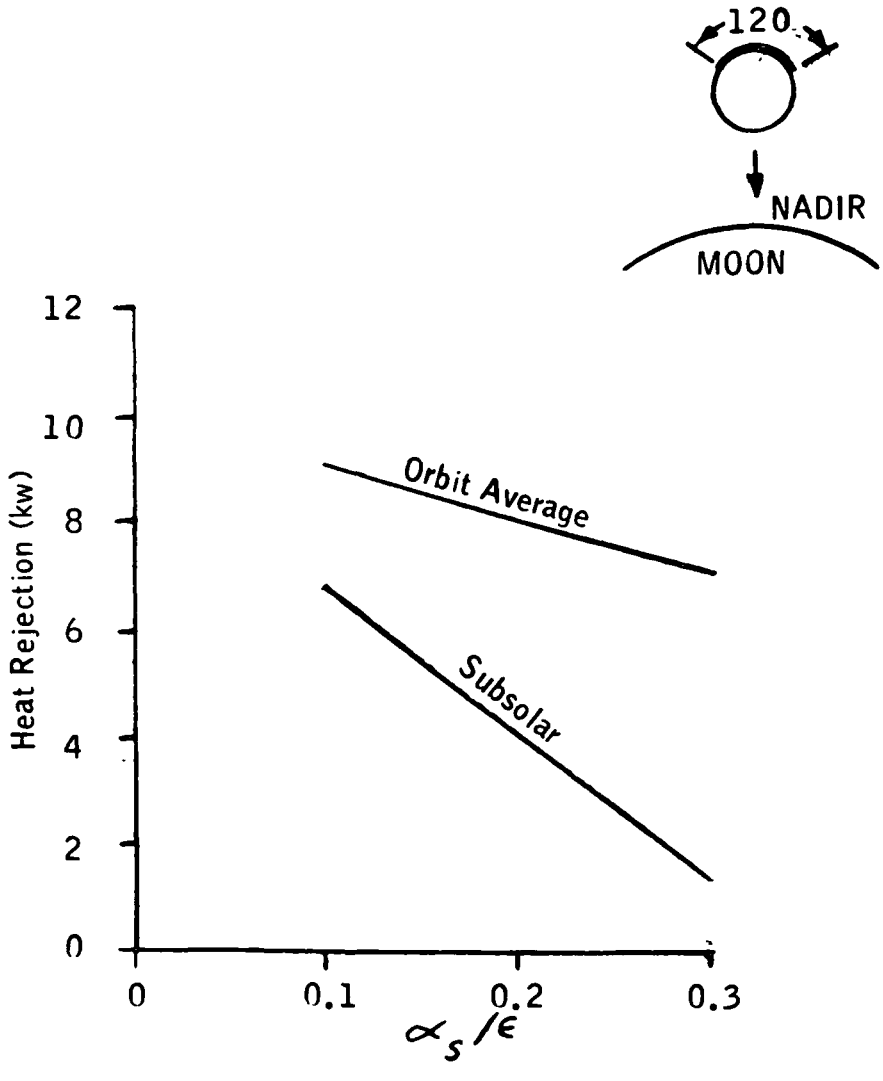


Figure 6.7-1. Derivative/OLS Heat Rejection from an Unobstructed Module with 293 ft^2 Radiator Area 70 F Radiator

The module rejection capability is affected by its location with respect to other modules. Figure 6.7-2 illustrates this effect for a 70 F radiator system. Subsolar and orbit average rejection rates are shown as a function of α_s/ϵ for: (1) a module on the end, (2) a module in a center location or on the end near a tug (the effect of a tug and an adjacent module are nearly identical), and (3) a module in the center (two adjacent modules) and near a tug. Both crew modules have rejection requirements above even their orbit average heat rejection capabilities. The core modules have heat loads of 4.8 kilowatts which must be rejected by other module radiators. For these reasons, the radiators on all modules must be interconnected.

There is no requirement for the station to be manned during the build-up phase. Therefore, the heat rejection system need not be operational until the final module is installed and the proposed interconnected radiator system is completed. Figure 6.7-3 shows the total combined rejection capability of the derivative OLS.

The power module heat rejection requirements total 11.3 kilowatts during the sunlit portion of the orbit. Of this total, 10.2 kilowatts can be rejected at a high temperature (greater than 120 F). However, 1.1 kilowatts must be removed from the batteries which must be maintained at approximately 50 F. The high temperature load can be marginally rejected by the antilunar power boom radiator surfaces (approximately 308 square feet) with an effective radiator temperature of 150 F at the subsolar point. The low temperature load (1.1 kilowatts) must be transported with an active fluid loop to the TCS of the other derivative OLS modules for rejection.

Required Changes

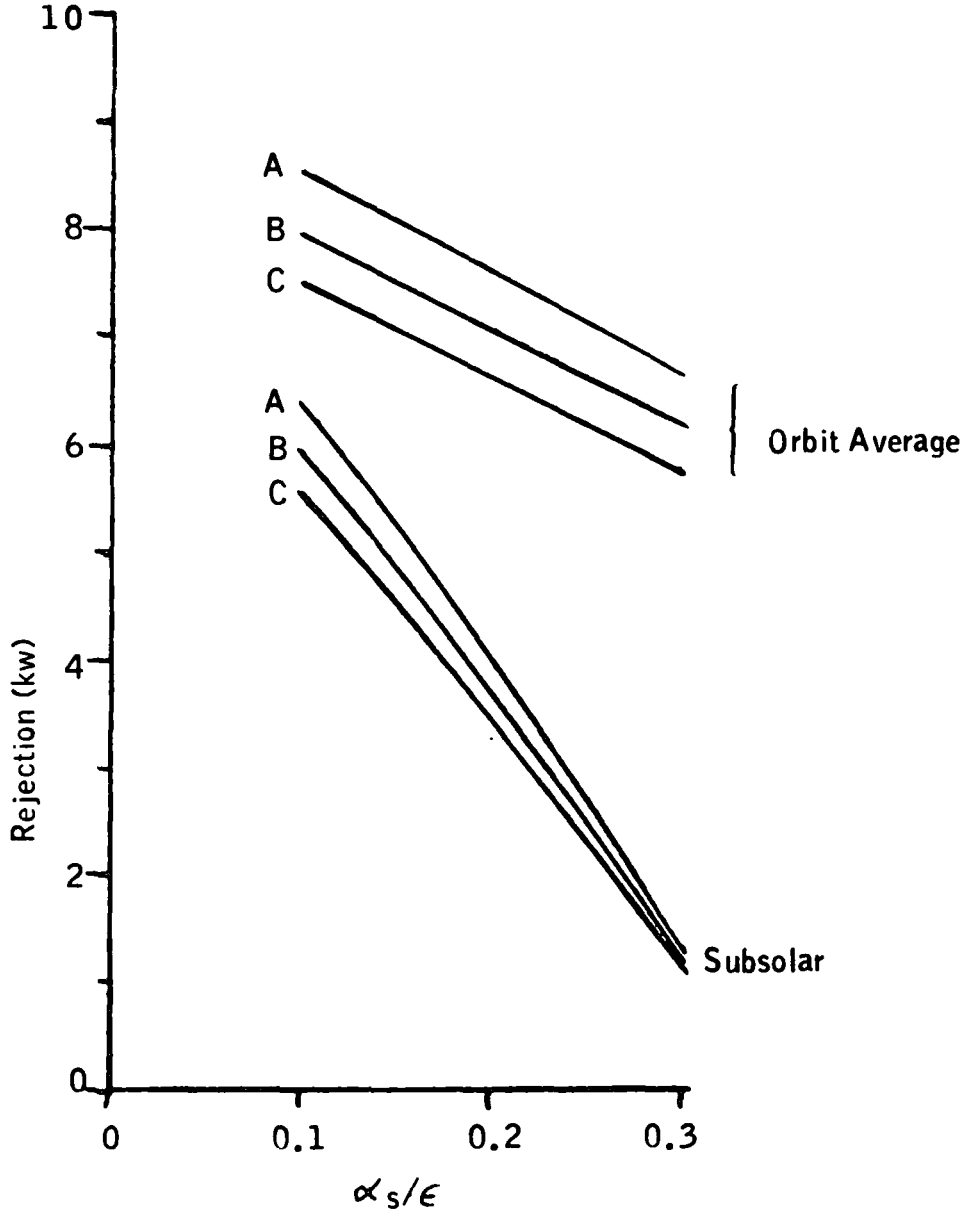
To convert the MSS TCS to the required derivative OLS TCS, Option 4 in Table 6.7-1, the following MSS changes are required.

Additional Radiator Area

Referring to the sketch of the derivative OLS, shown as Figure 5-1, modules CCM1, CCM2, CQM1, CQM3, and GM will (per MSS requirements) have radiators located 360 degrees circumferentially around the module. In addition to the effective portion (1463 square feet) of these radiators, it will be necessary to locate radiator panels (293 square feet per module) the length of the module on the antilunar facing surface ($120^\circ \leq \delta \leq 240^\circ$) of modules CSM1 and CSM2. This will result in a total effective derivative OLS radiator area of $\sim$2000 square feet, the same as for the representative OLS.

Addition of a Heat Pump

Based on the data presented earlier in this section, the requirement for an active 70 F radiator system was established. To employ this system, a heat pump is required. For redundancy, two heat pumps, one in CQM1 and one in CQM3 are required.



A Module on end
B Module in center or end by tug
C Module in center by tug

NOTE: Based on 293 square feet of 70 F external active radiator area

Figure 6.7-2. Module Rejection Capability as a Function of Location

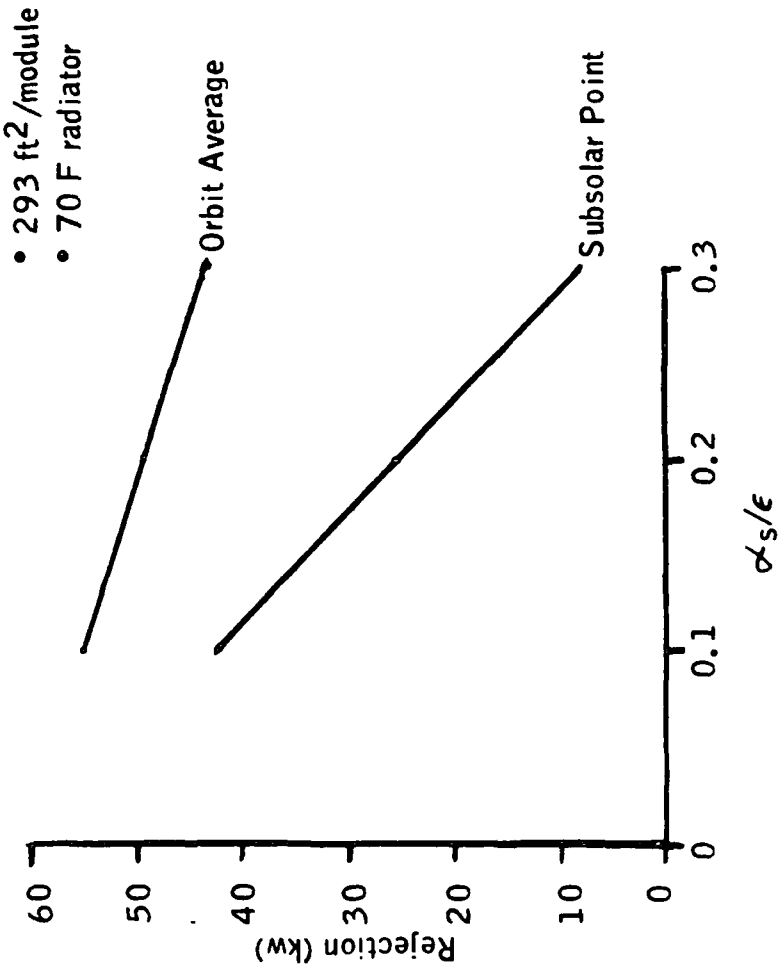


Figure 6.7-3. Derivative/OLS Total Rejection Capability

Thermal Capacitance

For the OLS subsolar case, heat rejection capability of even 70 F active fluid radiators is less than rejection requirements. Therefore, some thermal capacitance is required so that heat can be stored at the subsolar point and rejected later when rejection capability exceeds requirements. Utilization of the 16,000 pounds of water, used as radiation shielding in the solar flare shelter located in CCM1 as a thermal heat sink, is proposed. This was also required for the representative OLS as was discussed in Section 3.6.2. The temperature fluctuation of the water in the heat sink, due to storing heat at the subsolar point, would be minimal (approximately 2 F).

Power Boom Thermal Control Assembly

Because of the required low operating temperature of the batteries (50 F) and the limited effective radiator area on the power module, the coolant loop must be divided between the batteries and the power conditioning equipment. The battery coolant loop must be connected to the loop of the seven OLS modules through a heat pump. The power conditioning coolant loop will be connected directly to the boom radiators that operate at an effective temperature of 150 F.

Desirable Changes

The driving factor in radiator design is rejection near the subsolar point. The orbit average rejection capability of the seven 120-degree radiators is well about the rejection requirement. The radiators which exist between $0^{\circ} \leq \gamma \leq 120^{\circ}$ and $240^{\circ} \leq \gamma \leq 360^{\circ}$ on the five MSS radiator modules add little to subsolar rejection. It is therefore desirable to remove radiators from these locations, leaving only radiators located between $120^{\circ} \leq \gamma \leq 240^{\circ}$, thus saving the additional cost and weight of the unnecessary lunar surface facing radiators.

The thermal protection system could be significantly simplified if the batteries and battery chargers were replaced with regenerative fuel cells (see Section 6.3). The power boom could be identical to the boom for the representative OLS. The heat pump in the manned modules could be reduced, a single coolant loop would suffice on the power boom, and the effective operating temperatures of the radiators could be reduced from 150 F to 120 F.

6.7.3 RADIATION PROTECTION

Radiation protection requirements of the OLS and MSS are identical. However, due to the increased severity of the natural radiation solar flare environment in lunar orbit, the derivative OLS is required to have a storm shelter. This storm shelter must provide 16.6 gm/cm^2 of shielding encompassing the secondary control center, a food reconstitution unit, and a toilet facility. Sufficient food storage must also be provided in the storm shelter to provide for the crew for a solar flare activity of up to three days duration (see Section 3.6.3 for further discussion of the storm shelter design). As a result, Control Center Module 1 of the MSS, which includes the secondary control center, must be modified to accommodate the storm shelter design.

Desired Changes

The required changes to the MSS for radiation protection do not include provisions for a galley within the shelter. Adequate space is available to incorporate a microwave oven in the enclosed area. It is desirable that the microwave oven in the backup galley of the MSS be so configured that it can be moved to the storm shelter area (portable) for use during a solar flare.

7.0 DERIVATIVE OLS CONFIGURATION

In this section, the derivative OLS and its subsystems are described and defined. The modules which make up the derivative OLS configuration are, except for the dual support cargo module (DSCM) and the experiment module (XM), all derived from MSS modules. The DSCM and the XM, which were developed in connection with the representative OLS, are necessary to satisfy unique lunar science support requirements and are therefore also required as part of the derivative OLS. The other modules which are employed in the derivative OLS, as defined in Section 5.0 and depicted in Figure 5-1 and OLS Drawing 2282-29, are all derived from parent MSS module by implementation of the required changes defined in Section 6.0. Note that for purposes of derivative OLS definition, no desired changes are included.

Sections 7.1 through 7.7 define the details of the derivative OLS, a brief summary of each of these sections is presented in the following paragraphs.

STRUCTURES

Section 7.1 describes the structural detail of each derivative OLS module. The primary structure of each of these modules is indentical to its parent MSS module with the exception of the strengthened side docking port structure in the two OLS core modules. This modification was required to meet the moments induced by side docked modules during delivery to lunar orbits as discussed in Section 5.0. Considerable modification of the interior arrangement and secondary structure was necessary in some modules however to satisfy the OLS functional requirements of Section 4.0. Prime examples were the cryogenic storage modules CSM 1 and 2 and control center module 1 which was modified to incorporate the solar flare shelter. This 100 square foot shelter encompasses the secondary control center and includes food preparation facilities and a toilet.

ENVIRONMENTAL CONTROL/LIFE SUPPORT

Section 7.2 describes the ECLSS. The concepts employed to satisfy the ECLSS functions are identical to those of the representative OLS documented in Section 3.2. Major ECLSS equipment is located in the two crew quarters modules, including processing equipment for CO_2 removal and reduction, trace contaminant control, humidity control, pressure control, heat exchangers, heat pumps, water processing and hygiene control. Temperature control and circulation equipment are located in each module. Waste processing units are located in the galley module, and medical areas in control center module 2.

ELECTRICAL POWER SUBSYSTEM

Section 7.3 describes the concepts employed in the derivative OLS for primary, secondary, and emergency/auxiliary electrical power generation and energy storage. The required EPS subsystem is identical to the parent MSS concept, basically a solar array - battery system.

INFORMATION SUBSYSTEM

Section 7.4 describes the ISS functions and the concepts employed on the derivative OLS to satisfy these functions. Except for (1) the replacement of two boom-mounted MSS, 13-foot diameter parabolic antenna subassemblies and (2) the relocation of some components from one module to another, the OLS ISS equipment and concept is identical to the MSS. This concept differs from that employed for the representative OLS in only two ways; it uses parabolic dish antennas rather than phased arrays and it uses modular data processors located in various modules rather than one large centralized multiprocessor.

GUIDANCE AND CONTROL SUBSYSTEM

Section 7.5 presents the derivative OLS G&C subsystem definition. The derivative OLS G&C subsystem configuration is identical to the representative OLS described in Section 3.5 except that the CMG array employed is the lighter 1600 pound MSS array.

REACTION CONTROL SUBSYSTEM

Section 7.6 describes the derivative OLS RCS. The derivative RCS is very similar to the MSS RCS and to the representative OLS RCS described in Section 3.5 The only significant difference between the two OLS concepts is in the physical layout of the engine quads, accumulators, and cryogenic storage tanks.

ENVIRONMENTAL PROTECTION SUBSYSTEM

Section 7.7 describes the devices employed by the derivative OLS to protect against the natural environmental hazards of micrometeoroids, thermal radiation, and nuclear radiation. The meteoroid protection concept discussed in Section 7.7.1 is identical to that employed on the representative OLS which is described in Section 3.6.1 except that for the derivative OLS a double meteoroid bumper concept is found to be weight effective.

The thermal protection concept is described in Section 7.7.2. The only effective radiator area on the OLS is on the anti-lunar side of the modules but the proximity of adjacent modules (view factor effect) further reduces the usable area. To meet heat rejection requirements, all permanent modules of the derivative OLS configuration, excepting the experiment module, require radiators. The concept is basically the same as the representative OLS. An active thermal loop with an effective operating temperature of 70 F and a heat pump are used. The heat pump is an addition to the MSS concept. The power boom thermal protection system incorporates a split coolant loop. The power conditioning equipment is on one loop that uses the radiators on the boom at an effective operating temperature of 150 F. The second loop is for the batteries and is connected to the manned module coolant loop through a heat pump and has an effective operating temperature of 50 F.

The nuclear radiation protection concept for the derivative OLS is discussed in Section 7.7.3. The requirements for crew protection against nuclear radiation, principally from solar flares, is the same for both OLS concepts. Therefore the general concept, that of employing a shield composed of water, food, and high density shielding to enclose the secondary control center, is also employed for the derivative OLS. As in the representative OLS, the water in the radiation shield is also employed as a thermal capacitor in the thermal control system.

Page intentionally left blank

7.1 STRUCTURES

The design effort performed in support of this task has been directed toward assessing the baseline earth orbiting Modular Space Station (MSS) vehicle concept, identifying those modifications which are considered necessary to comply with the OLS operational and mission requirements, and delineating the configurational characteristics of the resultant derivative OLS.

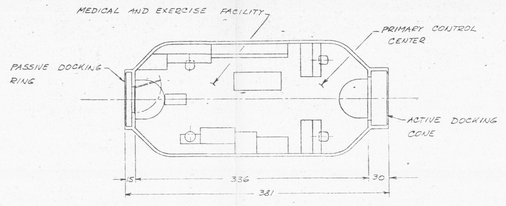
The orbital flight configuration of the derivative OLS is shown on Drawing 2282-29. The OLS consists of a pair of equal sized core modules (MSS CM-1 modules), docked end to end, which serves as the basic "backbone" of the station, and to which are docked all of the other modular elements of the complete station. The power module, identical to that used in the 12-man MSS is docked to the outboard end of OLS core module #1A. As a result of comprehensive analyses of the orbit attitude control energy requirements for a series of alternative clustered arrangements, the selected flight mode features the OLS flying with the X-axis perpendicular to the orbit plane, and with the +Z axis pointing to the local vertical. The flight configuration features most of the permanently docked modules positioned in the X-Y plane, perpendicular to the local vertical. Included in this category are the following: CCM1, CCM2, CQM1, CQM3, GM, Cryogenic Storage Modules (CSM1 and CSM2) and the Dual Support Cargo Module (DSCM). The modular elements docked to the core module and positioned in the X-Z plane include both tug/landers, and the Experiment Module (XM). Tugs A and B are both docked on the -Z axis, and the XM is on the +Z axis.

Although there are a total of 12 modules used in the derivative OLS, most of the modules are structurally similar to their parent MSS modules and in the following documentation, the details of the commonality will be delineated. In brief, the OLS core modules, command & control, and crew quarters modules are very similar to their MSS counterparts. The parent MSS modules are described in Drawing 9953-1011 presented in Section 5.0.

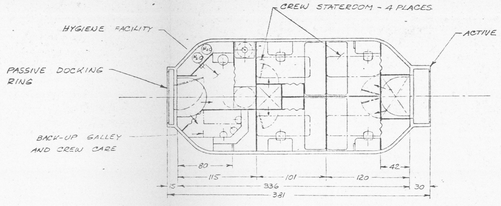
CORE MODULES

The two core modules C1A and C1B are structurally similar to each other and to the single large core module CM1 used on the MSS. The basic purpose of the module is to provide a central "backbone" or "keel" for the space station modules to dock to, and also accommodate various system interconnects between the docked modules. Structurally, the core module is basically a 12-foot diameter cylindrical pressure compartment with conical end bulkheads. The basic structure is of skin stringer construction with the exterior surface covered by insulation and meteoroid shield. The nominal 12-foot diameter is expanded at one end to a diameter of 14 feet to provide a biconic section with increased internal volume for accommodation of the Control Moment Gyros. The end closures of the core module are fitted with the 5-foot diameter EOSS developed docking systems. In the case of the OLS, C1A is equipped with passive rings, while C1B is equipped with active

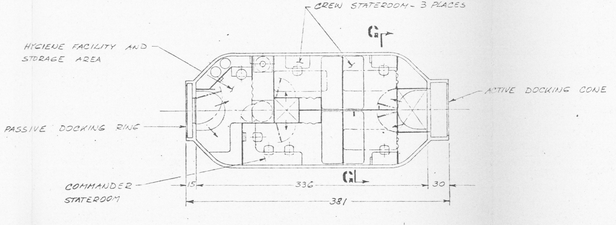
Page intentionally left blank



CCM2 MODULE



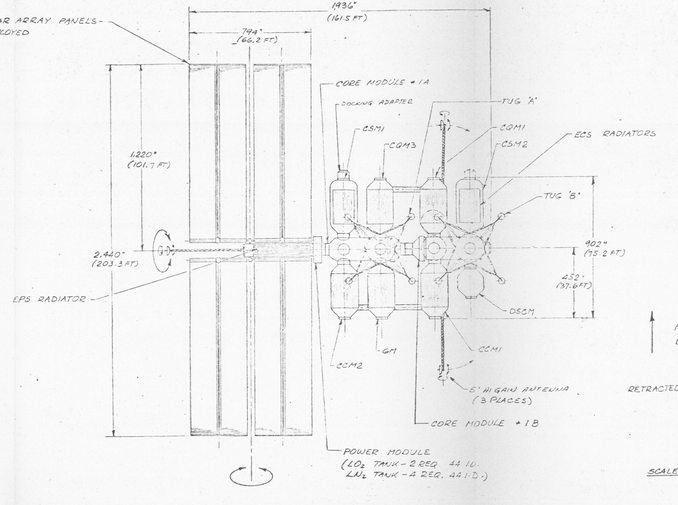
CCM1 MODULE



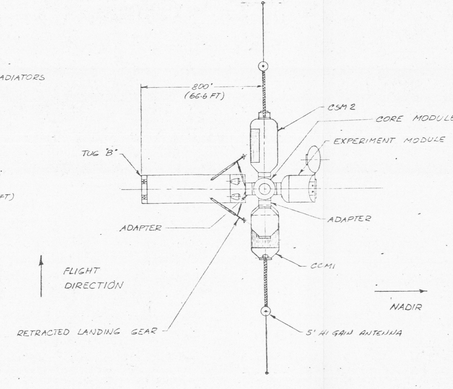
CCM3 MODULE

MODULE GLOSSARY

C1A	CORE MODULE #1
C1B	CORE MODULE #2
CCM1	CONTROL CENTER MODULE #1
CCM2	CONTROL CENTER MODULE #2
CQM1	CREW QUARTERS MODULE #1
CQM3	CREW QUARTERS MODULE #3
CSM1	CRYO SUPPORT MODULE #1
CSM2	CRYO SUPPORT MODULE #2
GM	GALLEY MODULE
DSCM	DUAL SUPPORT CARGO MODULE



DERIVATIVE OLS FLIGHT CONFIGURATION



NOTES:
1. WATER IS UTILIZED IN SOLAR FLARE SHELTER AS HEAT SINK. HEAT PUMPS ARE LOCATED IN CQM1 & CSM3.
2. EOS LOCATED ON 2" AXIS.
3. PRESSURE VOLUME OF CORE MODULE = 1 & 2: 3,900 ft³
4. PRESSURE VOLUME OF CSM1, CQM3, CCM1, CCM2 & GM = 3,300 ft³

SCALE: 1/200

7-7, 7-8

SD 71-207

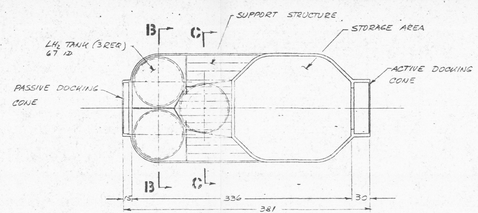
THIS PRINT REDUCED 50%

SIZE D	DWG NO.	SPACE DIVISION	
1/200	SCALE	NORTH AMERICAN ROCKWELL CORPORATION	
NO. STRUCT.	WEIGHT	12214 LAKEWOOD BOULEVARD · DOWNEY, CALIFORNIA	

DERIVATIVE OLS (MODULAR) CONFIGURATION OLS STUDY

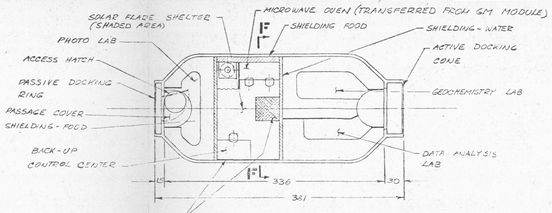
2282-29B
SHT 1 OF 3

2282-29

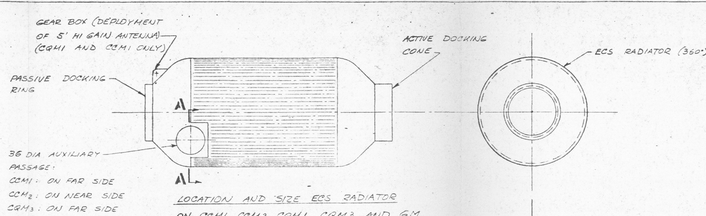


CSM1 MODULE - PRESS. VOLUME 1,650 FT³

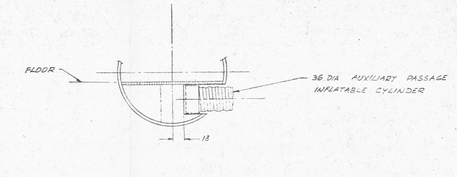
LH_2 TANK - 3 REQ. 67 I.D.
LO_2 TANK - 1 REQ. 84 I.D.



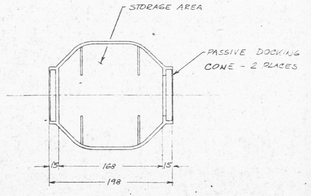
CSM1 MODULE



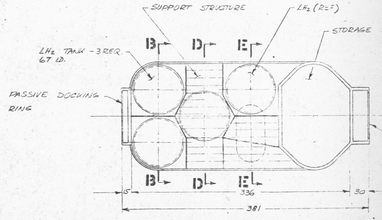
LOCATION AND SIZE ECS RADIATOR ON CSM1, CSM2, CSM1, CSM3 AND GM



SECTION A-A

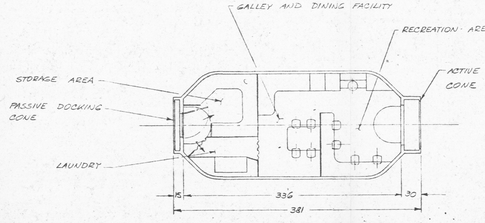


DSCM MODULE
PRESS VOLUME 1,500 FT³

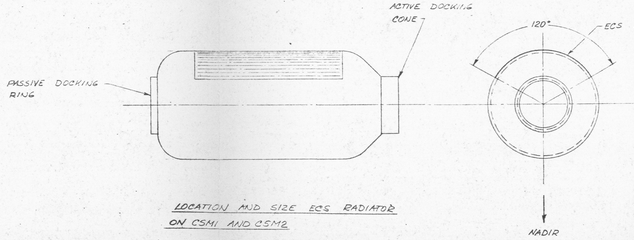


CSM2 MODULE - PRESS. VOLUME 1,002 FT³

LH_2 TANK - 3 REQ. 67 I.D.
LO_2 TANK - 1 REQ. 67 I.D.
LN_2 TANK - 1 REQ. 67 I.D.
HYDRAZINE TANK - 2 REQ.



GM MODULE



LOCATION AND SIZE ECS RADIATOR ON CSM1 AND CSM2

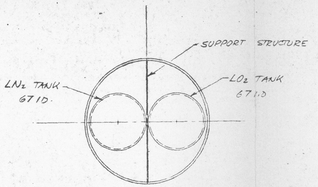
NOTE:
MODULE CG'S SHOWN ON SHT 2

7-9, 7-10

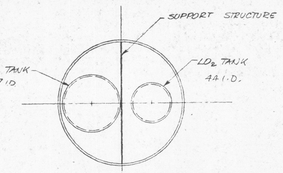
SD 71-207

THIS PRINT REDUCED 50%

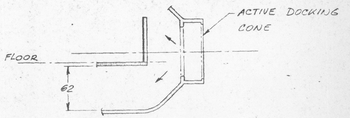
DESIGN DEPT.	NO. 71-9114-1	SPACE DIVISION	
APPROVALS	DATE 2 DEC '71	NORTH AMERICAN ROCKWELL CORPORATION	
	MODEL	12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
DERIVATIVE S/C'S (MODULAR) CONFIGURATION OLS STUDY			2282 - 290 SHT 2 OF 3



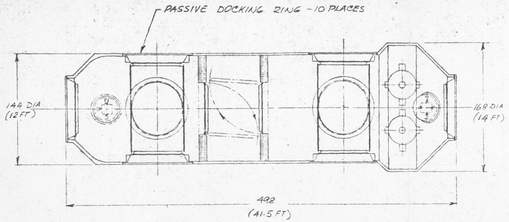
SECTION D - D



SECTION C - C

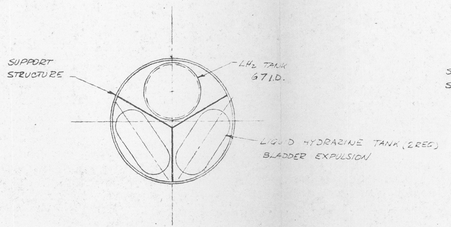


TYP SECTION THRU COM1, COM2, GM, COM1 AND COM3

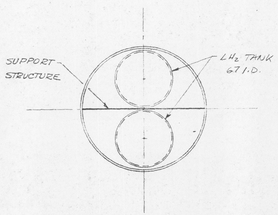


CORE MODULE - 1A

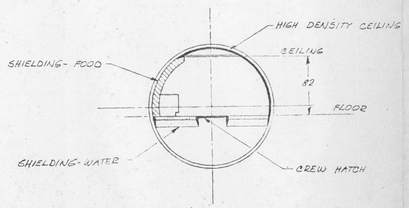
NOTE: CORE MODULE 1B SAME, EXCEPT CONTROL MOMENT GYRO (3 REQ) IS REMOVED AND ACTIVE DOCKING SYSTEM IS ADDED ON BOTH ENDS



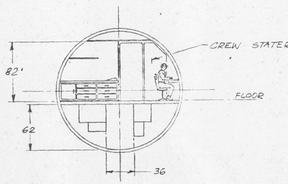
SECTION E - E



SECTION B - B



SECTION F - F
ROTATED 90°



SECTION G - G
ROTATED 90°

NOTE: MODULE GLOSSARY ON SHT 1

7-11, 7-12

SD 71-207

THIS PRINT REDUCED 50%

DESIGN Chan F. 4-25-72	NO.	SPACE DIVISION NORTH AMERICAN ROCKWELL CORPORATION 12214 LAKEWOOD BOULEVARD, DOWNEY, CALIFORNIA	
	DATE 5-25-72		
	MODEL		

DERIVATIVE OLS (MODULAR) CONFIGURATION OLS STUDY

2282-290
SHT 3 OF 3

cone docking mechanisms. This will permit the core modules to be docked to each other and provide a docking interface on the outboard end of C1B that will be compatible with either active or passive systems on another vehicle. The basic core module features 8 docking ports arranged in two bands of 4 each, approximately 10 to 12 feet from the ends of the module. For OLS use, the primary structure surrounding the side docking ports is strengthened to meet the more stringent OLS loads which occur during delivery to lunar orbit in a partially assembled configurations as described and discussed in Section 5.0. These docking ports are passive, and each is equipped with a 5-foot diameter removable pressure hatch. Each core module is compartmented in the middle by a pair of flat pressure bulkheads which provide an IVA/EVA airlock. Deployable attitude control nozzle cluster assemblies are located at each conical end bulkhead on the core modules. The nozzle clusters are integrated into the same end of C1A and C1B. When the core modules are docked together, the nozzle clusters are at the extreme ends of the assembly. Cryogenic gas storage tanks for the RCS nozzles are provided in each end of the core modules adjacent to the nozzle clusters.

The control moment gyros are located in C1B although C1A is physically able to accommodate them. Specific G/N equipment and sensors are located in the C1B including star trackers and horizon sensors. The fuel cell electrical power system is also located in C1B. Both modules contain cryogenic transfer and distribution systems since part of the LO_2 is located in existing tanks in the power module, and the remainder of the LO_2 and all of the LH_2 tanks are located in two of the storage cargo modules. Two high pressure GN_2 tanks are located in C1B for expulsion with the N_2H_4 tanks in CSM_2. High pressure GN_2 and GO_2 tanks are incorporated in both core modules for emergency leakage plus O_2 metabolic consumption.

OPERATIONAL AND LIVING QUARTERS MODULES

The 5 OLS modules which constitute the primary living and operational quarters of the OLS are structurally similar to their parent MSS modules. The basic module has a diameter of 14 feet, and an overall length of 31 feet, 9 inches, and a pressure volume of approximately 3500 cubic feet. End pressure bulkheads are conical in chape with a cassinian intersection at the bulkhead/sidewall intersection. An integral skin/stringer construction is featured on the cylindrical sidewall, with bonded external stringers attached to the conical end bulkheads. Each module is provided with a passive docking port at one end and an active at the other. A longitudinal primary floor 4 inches thick is featured in each module positioned slightly off-center which results in a floor-to-ceiling height of 94 inches on the normal occupancy side, and 62 inches below the floor for crew passageway and storage. A false ceiling of 12 inches is provided for utility runs. The gross floor area available in each module is 337 square feet. A basic feature of the module is that at each end of the module, the floor is cut away to permit crew access to a 36-inch wide full module length passageway below the floor. This concept results in excellent access to equipment and storage areas below the primary floor in addition to permitting desired compartmentation of the habitable area above the floor. Access to closed-out areas, for example, crew staterooms, is through an appropriately located floor opening. As a result, the aisle required for normal mobility through and into a module are not required, thus allowing full utilization of the floor area for habitability purposes.

CQM1 (CREW QUARTERS MODULE 1)

This module is physically identical to that used in the MSS configuration. The internal arrangement provides four individual staterooms for four crew members. Access to two of the staterooms is from the end of the module through doors near the below-deck passage. A 36-inch square passage in the floor provides entry into the two centrally positioned staterooms. The remainder of the module floor is utilized by a hygiene facility and a backup galley.

CQM3 (CREW QUARTERS MODULE 3)

This module is very similar to CQM1 previously described and is structurally identical to MSS CQM3.

CCM1 (CONTROL CENTER MODULE 1)

The internal arrangement of this module in the OLS configuration is different than when used in the MSS. This module contains the secondary or backup control center with the radiation shielding required to permit its operation as the storm cellar. Within the confines of the protected area, a sanitation facility (16 square feet) is provided. The radiation shielding consists of high density shielding, movable panels filled with ambient storable food, and water-filled side walls, and floor panels. High density shielding is used on the portion of the enclosure formed by the module pressure shield. The total enclosed area is 100 square feet. The 3 major laboratories occupy the remainder of the floor area in this module; included are the Data Analysis Lab, the Photographic Lab, and Geochemistry Lab.

CCM2 (CONTROL CENTER MODULE 2)

The primary control center is located in this module, occupying a floor area of approximately 100 square feet. The remainder of the available floor area, approximately 235 square feet is utilized by the medical/exercise facility. This area was obtained by eliminating the MSS fluid/electrical/optics laboratory.

GM (GALLEY MODULE)

This module has been internally configured to contain a laundry facility, primary galley and dining area, and the recreation area. The MSS hygiene facility area of 77 square feet has been converted to a laundry and storage area. The recreation facility has been expanded from the 40 square feet MSS arrangement configured for 6 men to 80 square feet for the 8-man OLS crew. The remainder of the floor area, approximately 145 square feet, has been configured for the primary galley and dining area.

CSM2 (CRYOGENIC STORAGE MODULE 2)

The primary function of the cargo module is to accommodate a portion of the cryogenic storage system. The cryogenics are located in three places in the modular OLS, the Power Module, CSM1, and CSM2. Five 66-inch diameter cryogenic tanks are integrated into the unpressurized section of CSM2. Three of the tanks are used for the LH_2, one tank for LO_2 and one tank for LN_2. These tanks are encapsulated in approximately two inches of high performance insulation, and suspended from the primary structure on low

thermal conductive cables. In addition to the cryogenic tanks, a pair of cylindrical tanks are also located in the same area to contain the hyrazine supply used for resupplying the subsatellites. These tanks are 45 inches in diameter by 105 inches long. The length of the pressurized compartment in the CM is approximately 100 inches, and the total pressurized volume is 730 cubic feet. This compartment is not dedicated for any particular use.

CSM1 (CRYOGENIC STORAGE MODULE 1)

This module is similar in configuration and construction to CSM2. The primary function is the same, storage for a portion of the cryogenic supply. In this case, there are three 66-inch diameter tanks for LH_2, and one 44-inch diameter tank for LO_2. The tanks are encapsulated in a two-inch thick covering of high performance insulation, and supported on low thermal conductive cables. The remainder of the module is occupied by an undedicated pressurized compartment with a depth of 170 inches and a pressurized volume of 1200 cubic feet.

PM (POWER MODULE)

The power module consists of a 76-inch diameter structural module with an overall length of 536 inches which provides structural support for the four rollout solar arrays. A drive system is incorporated near the outboard end of the module which provides for rotation of the solar array system about the longitudinal axis of the power module. Each end of the power module is equipped with docking interfaces; a passive ring at the outboard end, and an active system at the inboard end which docks to Core Module 1A. The exterior surface of the power module cylindrical structure is covered with insulation and an active radiator system used to reject heat produced by the power conditioning equipment.

The inboard end of the basic cylinder has a circumferential section in which the basic diameter is increased to 14 feet to provide space for accommodation of six 44-inch diameter LO_2 storage tanks. These tanks are encapsulated in two inches of high-performance insulation and are supported on low thermal conductive cables.

DUAL SUPPORT CARGO MODULE (DSCM)

The module is designed to perform a dual function; accommodate OLS cargo and lunar surface sortie non-return cargo. Based upon the allocated respective cargo weights for the OLS and surface base, and reasonable densities and packaging efficiencies, the cargo module has been sized for approximately 1500 cubic feet volume. The module, configurationally, is the same as the previously described habitable modules. It has an outside diameter of 14 feet, a length of 119 inches and features the cassinian/conical end pressure bulkheads. Transverse bulkheads and other secondary internal structure would be provided to support the cargo. Each end of the module is equipped with a passive docking ring. In order for the module to be docked to the OLS, it is necessary

to use a double active/active docking adapter. Since the module is eventually left on the lunar surface, considerable cost and weight savings could be realized by not providing an active docking system on one end. In accommodating the cargo internally in the module, the cargo destined for the OLS use would be located so that it could be removed first without disturbing surface sortie cargo.

EXPERIMENT MODULE

This module is identical to that conceptually developed and described in Section 2.1 of this report. The docked position of the experiment module relative to the other modules on the derivative OLS is shown on Drawing 2282-29. Configurational details are shown on Drawing 2282-26 (see Section 2.1).

WEIGHT STATEMENT

The derivative OLS weight statement (Table 7.1-1) was derived by determining the delta changes from the MSS weight statement. The changes are identified in the following paragraphs and the weights for each module are tabulated in Table 7.1-2. The experiment module for the derivative OLS is identical to that for the representative OLS and its weight characteristics are tabulated in Table 2-2 of Section 2.0.

Core Module - 1A

All subsystem weights are the same as the MSS Core Module 1 with the following exceptions:

Structures. Added 250 pounds of structural support weight at each of the four side docking ports to allow for moments induced during cislunar flight.

Guidance and Navigation. Most of this system is relocated in Core Module 1B.

Instrumentation. The complete MSS system is removed and replaced with 1070 pounds of data processing equipment and 40 pounds of display and controls.

Communications. Replaced the MSS system with new system and therefore, new system weight.

Environmental Control. Completely replaced the MSS system. New weight includes 254 pounds of spares.

In-Flight Losses. 1000 pounds of cold gas N_2 was deleted.

Core Module - 1B

MSS Core Module 1 was used for the base of the derivative OLS Core Module 1B rather than the MSS Core Module 2 since there was more similarity. The following system weights changed.

Structures. Added 1000 pounds of docking port support structure, as in C1A.

Primary Power Source. The complete system was removed and replaced with a portion of the system in MSS Core Module 2.

Power Conversion and Distribution. The complete system was removed and replaced with a portion of the system in MSS Core Module 2.

Table 7.1-1 Derivative OLS Weight Summary

Step	Items	Lunar Orbit Modular Space Station
1	Core Module 1A	31,685
	Core Module Dry Weight	29,395
	Core Module Consumables	2,290
2	Core Module 1B	25,295
	Core Module Dry Weight	23,005
	Core Module Consumables	2,290
3	Power Module	24,215
	Power Module Dry Weight	17,313
	Power Module Consumables	6,902
4	Cryo Module 1	10,258
	Cryo Module Dry Weight	7,689
	Cryo Module Consumables	2,369
5	Cryo Module 2	26,716
	Cryo Module Dry Weight	10,545
	Cryo Module Consumables	16,176
6	Control Center Module 1	36,600
	Control Center Module Dry Weight	18,290
	Control Center Module Consumables	2,310
	Water - Radiation Shelter	16,000
7	Control Center Module 2	11,720
	Control Center Module Dry Weight	11,340
	Control Center Module Consumables	380
8	Crew Quarters Module 1	15,325
	Crew Quarters Module Dry Weight	13,840
	Crew Quarters Module Consumables	1,485
9	Crew Quarters Module 3	12,915
	Crew Quarters Module Dry Weight	12,365
	Crew Quarters Consumables	550
10	Galley Module	15,775
	Galley Module Dry Weight	10,781
	Galley Module Consumables	4,994
11	Experiment Module	11,000
	Experiment Module Dry Weight	11,000
	Experiment Module Consumables	0
	Crew 8-man	1,600
	Derivative OLS Gross Weight	223,004

Table 7.1-2. Comparison of OLS and MSS Modular Weights

Module	MSS Core 1	Delta Weight	OLS Core 1A
Structure	6,780	+1,000	7,780
Induced Environment Protection	1,310		1,310
Launch Recov. & Docking	2,580		2,580
Orientation Controls Separation & Ullage	2,390		2,390
Prime Power Source	8,245		8,245
Power Conversion & Distribution	1,305		1,305
Guidance & Navigation	1,730	-1,695	35
Instrumentation (Data Proc. D&C, Software)	220	+890	1,110
Communications	145	-30	115
Environmental Control	950	+2,360	3,310
Personnel Provisions	1,170		1,170
Crew Sta. Control & Panels	45		45
Subtotal (Dry Weight)	26,870	+1,525	29,395
High Pressure O_2 and N_2			1,940
Personnel	50		50
Residual Prop. & Serv. Items	300		300
Subtotal (Inert Weight)	27,220	+4,360	31,685
Inflight Losses	1,000	-1,000	0
Total (Gross Weight)	28,220	+3,360	31,685

Table 7.1-2. Comparison of OLS and MSS Modular Weights (continued)

Module	MSS Core 1	Delta Weight	OLS Core 1B
Structure	6,780	+1,000	7,780
Induced Environment Protection	1,310		1,310
Launch Recovery & Docking	2,580	+290	2,870
Orientation Controls Separation & Ullage	2,390		2,390
Prime Power Source	8,245	-6,210	2,035
Power Conversion & Distribution	1,305	-450	855
Guidance & Navigation	1,730	+175	1,905
Instrumentation (Data Proc. D&D, Software)	220	-100	120
Communication	145	-30	115
Environmental Control	950	+2,360	3,310
Personnel Provisions	1,170	-900	270
Crew Sta. Control & Panels	45		45
Subtotal (Dry Weight)	26,870	-4,865	23,005
High Pressure O_2 and N_2			1,940
Personnel	50		50
Residual Prop. & Serv. Items	300		300
Subtotal (Inert Weight)	27,220	-2,100	25,295
Inflight Losses	1,000	-1,000	0
Subtotal (Gross Weight)	28,220	-3,100	25,295

Table 7.1-2. Comparison of OLS and MSS Modular Weights (continued)

Module	MSS Pwr.Mod.#2	Delta Weight	OLS Pwr.Mod.
Structure	2,735		2,735
Induced Environment Protection	815	+208	1,023
Launch Recovery & Docking	670		670
Orientation Controls Separation & Ullage	645		645
Prime Power Source	10,325		10,325
Power Conversion & Distribution	795		795
Guidance & Navigation	35	-35	0
Instrumentation (Data Proc. Software)	215	+83	298
Communications	30	+265	295
Environmental Control	600	-208	392
Personnel Provisions	135		135
Total (Dry Weight)	17,000	+313	17,313
Consum. & Resid. (Cryo)	0	+3,011	3,011
Total (Inert Weight)	17,000	+3,359	20,324
Expendable (Overboard)	0	+3,891	3,891
Total (Gross Weight)	17,000	+7,250	24,215

Table 7.1-2. Comparison of OLS and MSS Modular Weights (continued)

Module	MSS CM	Delta Weight	OLS Cryo M-1
Structure	3,520		3,520
Induced Environment Protection	760	+376	1,136
Launch Recovery & Docking	800		800
Orientation Controls Separation & Ullage	2,580	-906	1,674
Prime Power Source	120		120
Power Conversion & Distribution	85		85
Guidance & Navigation	35		35
Communications	60		60
Environmental Control	260	-149	111
Personnel Provisions	180	-50	130
Instrumentation	0	+18	18
Total (Dry Weight)	8,400	-711	7,689
Cargo (Logistic Payload)	11,350	-11,350	0
Residual Prop. & Service Items	190	+ 1,019	1,209
Total (Inert Weight)	19,940	-11,529	8,898
Inflight Losses (Cryo)	60	+1,300	1,360
Total (Gross Weight)	20,000	-10,229	10,258

Table 7.1-2. Comparison of OLS and MSS Modular Weights (continued)

Module	MSS CM	Delta weight	OLS Cryo M-2
Structure	3,520		3,520
Induced Environment Protection	760	+376	1,136
Launch Recovery & Docking	800		800
Orientation Controls Separation & Ullage	2,580	+2,010	4,590
Prime Power Source	120		120
Power Conversion & Distribution	85		85
Guidance & Navigation	35		35
Communications	60	-42	18
Environmental Control	260	-149	111
Personnel Provisions	180	-50	130
Total (Dry Weight)	8,400	+2,145	10,545
Cargo (Logistic Payload)	11,350	-11,350	0
Reserve Prop. & Service Items	190	+3,651	3,841
Total (Inert Weight)	19,940	-6,059	14,386
Inflight Losses	60	+12,270	12,330
Total (Gross Weight)	20,000	+6,211	26,716

Table 7.1-2. Comparison of OLS and MSS Modular Weights (continued)

Module	MSS CCM-1	Delta weight	OLS CCM-1
Structure	3,995		3,995
Induced Environment Protection	505	+596	1,136
Launch Recovery & Docking	870		870
Power Conversion & Distribution	1,000		1,000
Guidance & Navigation	35	-35	0
Instrumentation	655	+401	1,056
Communication	690	+232	922
Environmental Control	1,210	-324	886
Personnel Provisions	350	-45	305
Crew Sta. Cont. & Panels	460	-460	0
Experiment Labs	0	+5,120	5,120
Shelter-Radiation	0	+3,000	3,000
Total (Dry Weight)	9,770	+8,520	18,290
Cargo (Experiments)	7,200	-7,200	0
Residual Prop. & Service Items	310	+2,000	2,310
Water - Radiation Shelter	0	+16,000	16,000
Total (Inert Weight)	17,280	+19,320	36,600
Total (Gross Weight)	17,280	+19,320	36,600

Table 7.1-2. Comparison of OLS and MSS Modular Weights (continued)

Module	MSS CCM-2	Delta weight	OLS CCM-2
Structure	3,800		3,800
Induced Environment Protection	540	+596	1,136
Launch Recovery & Docking	870		870
Power Controls & Distribution	1,000		1,000
Guidance & Navigation	35	-35	0
Instrumentation	655	+964	1,619
Communications	660	-57	603
Environmental Control	1,220	+712	1,932
Personnel Provisions	1,250	-870	380
Crew Sta. Cont. & Panels	460	-460	0
Total (Dry Weight)	10,490	+850	11,340
Personnel	100		100
Cargo (Experiments)	8,400	-8,400	0
Resid. Prop. & Serv. Items	270		270
Total (Inert Weight)	19,260	-7,550	11,710
Inflight Losses	10		10
Total (Gross Weight)	19,270	-7,550	11,720

Table 7.1-2. Comparison of OLS and MSS Modular Weights (continued)

Module	MSS CQM-1	Delta weight	OLS CQM-1
Structure	3,860		3,860
Induced Environment Protection	540	+696	1,236
Launch Rec. & Docking	870		870
Power Conversion & Distribution	1,000		1,000
Guidance & Navigation	35	-35	0
Instrumentation	205	+84	289
Communication	90	+266	356
Environmental Control	3,195	2,194	5,389
Personnel Provisions	1,930	-1,135	795
Crew Sta. Cont. & Panels	45		45
Total (Dry Weight)	11,770	+2,070	13,840
Personnel	470		470
Resid. Prop. & Serv. Items	270	+745	1,015
Total (Inert Weight)	12,510	+2,815	15,325
Inflight Losses	440	-440	0
Total (Gross Weight)	12,950	+2,375	15,325

Table 7.1-2. Comparison of OLS and MSS Modular Weights (continued)

Module	MSS CQM-3	Delta weight	OLS CQM-3
Structure	3,860		3,860
Induced Environment Protection	540	+696	1,236
Launch Recov. & Docking	870		870
Power Conv. & Dist.	1,000		1,000
Guidance & Navigation	35	-35	0
Instrumentation	205	+44	249
Communication	80	+6	86
Environmental Control	3,195	+1,124	4,319
Personnel Provisions	1,815	-1,070	745
Total (Dry Weight)	11,600	+765	12,365
Personnel	280		280
Resid. Prop. & Service Items	270		270
Total (Inert Weight)	12,150	+765	12,915
Inflight Losses	420	-420	0
Total (Gross Weight)	12,570	+345	12,915

Table 7.1-2. Comparison of OLS and MSS Modular Weights (continued)

Module	MSS CQM-2 (Mod)	Delta weight	OLS GM
Structure	3,950		3,950
Induced Environment Protection	540	+596	1,136
Launch Recovery & Docking	870		870
Power Control & Distribution	1,000		1,000
Guidance & Navigation	35	-35	
Instrumentation	205	+44	249
Communications	60	+6	66
Environmental Control	2,875	-315	2,560
Personnel Provisions	2,705	-1,755	950
Total (Dry Weight)	12,240	-1,459	10,781
Personnel (Food)	2,600	+1,050	3,650
Other Expenditures	0	+119	119
Resid. Prop. & Service Items	270		270
Total (Inert Weight)	15,110	-290	14,820
Inflight Losses	480	+475	955
Total (Gross Weight)	15,590	+185	15,775

Guidance and Navigation. The same system as in MSS Core Module 1 is incorporated plus the addition of 175 pounds: landmark tracker (35 pounds), radar altimeter (35 pounds), and three laser docking radar systems (35 pounds each).

Instrumentation. The system was completely replaced with a new system consisting of 30 pounds of data processing, 40 pounds of display and control, and 50 pounds of software.

Communications. The same system as in OLS-MSS Core Module 1A is incorporated resulting in a weight reduction of 30 pounds from the MSS Core Module 1.

Environmental Control. The weight is the same as in OLS Core Module 1A which results in a weight increase over the MSS Core Module 1 of 2360 pounds.

Personnel Provisions. The provisions of MSS Core Module 2 plus 100 pounds for items such as ladders, which are a function of the added length of the OLS Core Module 1B as compared to the MSS Core Module 2 still results in a net reduction of 900 pounds.

In-Flight Losses. 1000 pounds of cold gas N_2 is removed.

Power Module

Orientation Controls Separation and Ullage. No change.

Instrumentation. A new system which is 83 pounds greater than on the MSS is incorporated.

Guidance and Navigation. The laser docking radar system (35 pounds) is deleted.

Communications. A high-gain antenna is installed at the end of the boom.

Environmental Control and Induced Environment Protection. The boom radiator weight is apportioned partially to ENPS, as was done in the representative OLS weight statement; the net weight change from MSS in both subsystems is zero.

Expendables. This is required cryogenic for RCS which is expended overboard.

Cryo Storage Module 1

The cryo storage module 1 is based on the MSS Cargo Module with the following changes.

Orientation Controls Separation and Ullage. The MSS system is completely replaced with a new system resulting in a 906-pound reduction.

Environmental Control. The MSS ECLSS system is completely deleted. Added 293 square feet of radiator to this module; 111 pounds of radiator weight is apportioned to ECLSS; 376 pounds of radiator weight was apportioned to the ENPS as part of the micrometeoroid protection.

Personnel Provisions. The ECLSS portion of 50 pounds deleted.

Instrumentation. 18 pounds of instrumentation requirements are added.

Cargo. (Logistic Payload) deleted.

Residual Propellant and Service Items. The weight of 1209 pounds includes 150 pounds of atmosphere plus 1059 pounds of residual cryogenics.

In-Flight Losses. This increase is in cryogenics expended.

Cryo Module 2

The OLS cryo module 2 is also based on the MSS cargo module, with the following changes.

Orientation Controls Separation and Ullage. A complete new system is incorporated which results in an increase of 2010 pounds as this module serves as the major cryogenic storage facility.

Communications. Deleted except for 18 pounds of instrumentation.

Environmental Control. MSS system deleted, 293 square feet or radiator added, weight apportioned between ECLSS and ENPS as on cryo module 1.

Personnel Provisions. ECLSS items of 50 pounds are deleted.

Cargo. Deleted.

Reserve Propellant and Service Items. A weight of 3841 pounds represents the reserve and residual cryogenics. This results in a weight increase of 3651 pounds over the MSS cargo module.

In-Flight Losses. The 12,330 pounds is the expended cryogenics and subsatellite hydrazine.

Control Center Module 1

The module weight is derived from the counterpart module of the MSS. The follow weight changes result.

Instrumentation. A complete new system results in a 401-pound increase over the MSS module and consists of 695 pounds data processing, 825 pounds display and controls, and 100 pounds software.

Guidance and Navigation. The laser docking radar system (35 pounds) is deleted.

Communications. A new system of 922 pounds results in a weight increase of 232 pounds.

Environmental Control. The weight consists of 552 pounds of system plus 334 pounds of apportioned radiator weight, resulting in a weight decrease of 324 pounds over the MSS. 596 pounds of radiator weight is apportioned to the ENPS.

Personnel Provisions. 45 pounds of ECLSS items are deleted since they are included in ECLSS.

Crew Station Control and Panels. This item is deleted; it is included with instrumentation.

Experiments Laboratories. This is a completely new item and consists of the same labs as on the representative OLS.

Shelter-Radiation. This is also a new item consisting of 3000 pounds of high density shield and water shield container.

Cargo. (Experiments) Deleted.

Residual Propellant and Service Items. 2000 pounds of food added.

Water-Radiation Shelter. This is a new addition of 16,000 pounds of water.

Control Center Module 2

Instrumentation. A new system results in a weight increase of 964 pounds in this module over the MSS.

Guidance and Navigation. The laser docking radar system (35 pounds) is deleted.

Communications. The new system results in a weight decrease of 57 pounds.

Environmental Control. A complete new system results in a weight increase of 712 pounds in this module and includes 334 pounds of apportioned radiator weight. 596 pounds of radiator weight is apportioned to the ENPS for micrometeoroid protection.

Personnel Provisions. 735 pounds of ECLSS items are included in the ECLSS weight, thus the total here is 380 pounds.

Crew Station Control and Panels. This weight is deleted since it is included with the instrumentation weight.

Cargo. Deleted.

Crew Quarters Module 1

This weight is based on adjustments to the MSS CQM1 with the following changes.

Instrumentation. A new system results in an 84-pound weight increase.

Guidance and Navigation. The laser docking radar system (35 pounds) is deleted.

Communications. A new system results in a 266-pound weight increase.

Environmental Control System. A weight increase of 2194 pounds results from the new system. Included are 334 pounds of radiator weight apportioned to ECLSS.

Personnel Provisions. A weight reduction of 1135 pounds is due to the deletion of ECLSS items now included in ECS.

Residual Propellant and Service Item. The weight increase of 745 pounds is in ECLSS items and includes 200 pounds of food.

In-Flight Losses. This items is deleted for CQM1. Losses are included in other modules.

Induced Environment Protection. A 100-pound heat pump is added plus 596 pounds of apportioned radiator weight.

Crew Quarters Module 3

The weight of the OLS CQM3 is based on the MSS CQM3 with the following weight changes.

Instrumentation. The new instrumentation system results in a 44-pound increase.

Guidance and Navigation. The laser docking radar system (35 pounds) is deleted.

Communication. The new communication system results in a 6-pound increase.

Environmental Control System. The complete new system, including 334 pounds of apportioned radiator weight results in a 1124-pound increase to this module.

Personnel Provisions. 1070 pounds of ECLSS items are included in the ECS weight.

Induced Environment Protection. A 100-pound heat pump is added plus 596 pounds of apportioned radiator weight.

In-Flight Losses. Deleted, included in other modules.

Galley Module

The weight derivation of the OLS galley module is derived from the galley module of the MSS.

Instrumentation. The new instrumentation system weight results in a weight increase of 44 pounds.

Guidance and Navigation. The laser docking radar system (35 pounds) is deleted.

Communications. The new communication system weight results in a 6-pound increase.

Environmental Control System. The complete new system including 334 pounds of apportioned radiator weight, results in a 351-pound increase. 596 pounds of radiator weight are apportioned to ENPS as micrometeoroid protection.

Personnel Provisions. The deletion of the hygiene facilities and addition of a laundry facility decreases the weight 1755 pounds.

Personnel (Life Support). Addition of 1050 pounds of food, increases total food to 3650 pounds.

MASS PROPERTIES

The mass, center of gravity, and moments of inertia of each of the modules used in the selection of the preferred flight mode and module assemblage are presented in Table 7.1-3. The data include consideration of the lunar lander tug fully loaded as well as empty and also the dual support cargo module fully loaded and partially loaded. The composite mass properties (including products of inertia) of the configuration are presented in Table 7.1-4. The data for all four docked configurations that result from the degree of loading of the lunar lander tug and the cargo module are included.

The assemblage of modules that was evaluated in configuration 4 of Figure 6.5-1 of Section 6.0. This assemblage does not include the cryogenic storage module normally docked to CIA of the selected derivative OLS configuration. However, an analysis indicated that the overall results of the evaluation would not be altered if this module were included in the detailed mass property calculations.

The mass data used in the evaluation were preliminary estimates based on MSS modular weights. Table 7.1-2 indicates that in some cases there were significant changes in the OLS version of the MSS modules. However, an analysis was conducted to determine the sensitivity of the selection to module weight variations. It was determined that the selection was insensitive to the differences in weight between the initial assumed module weights and the final tabulated module weights.

Table 7.1-3. Mass Properties and Locations of Derivative OLS Modules

	Weight lb.	C.G.			Moment of Inertia - Slug ft^2		
		x	y	z	I_{ox}	I_{oy}	I_{oz}
Vehicle Configuration No. 4							
Core Module (CM1B)	28220	720.5	0.0	0.0	$.014 \times 10^6$	$.452 \times 10^6$	$.452 \times 10^6$
Core Module (CM1A)	25000	238.0	0.0	0.0	$.013 \times 10^6$	$.116 \times 10^6$	$.116 \times 10^6$
Crew Quarters Module (CQM3)	12570	634.2	277.0	0.0	$.106 \times 10^6$	$.010 \times 10^6$	$.106 \times 10^6$
Crew Quarters Module (CQM1)	12950	352.0	279.8	0.0	$.110 \times 10^6$	$.010 \times 10^5$	$.110 \times 10^6$
Cryogenic Storage Module	20000	112.0	462.0	0.0	$.80 \times 10^6$	$.80 \times 10^5$	$.80 \times 10^6$
Control Center Module (CCM2)	19270	872.0	-285.5	0.0	$.163 \times 10^6$	$.013 \times 10^6$	$.163 \times 10^6$
Control Center (CCM1)	35000	352.0	-290.0	0.0	$.29 \times 10^6$	$.024 \times 10^6$	$.29 \times 10^6$
Galley Module (GM)	15590	634.0	-277.0	0.0	$.132 \times 10^6$	$.011 \times 10^6$	$.132 \times 10^6$
Power Boom (PB) less solar array	6600	1240.0	0.0	0.0	$.12 \times 10^4$	$.26 \times 10^5$	$.26 \times 10^5$
Experiment Module (XM)	11000	352.0	0.0	208.0	$.112 \times 10^5$	$.112 \times 10^5$	$.967 \times 10^4$
Tug A - full	84615	634.2	0.0	-462.0	$.74 \times 10^6$	$.74 \times 10^6$	$.74 \times 10^5$
Tug B - full	91900	112.0	0.0	-462.0	$.80 \times 10^6$	$.80 \times 10^5$	$.80 \times 10^5$
Cargo Mod. - full	23550	112.0	-180.0	0.0	$.16 \times 10^5$	$.18 \times 10^5$	$.16 \times 10^5$
Tug B - empty	31900	112.0	0.0	-462.0	$.30 \times 10^6$	$.30 \times 10^6$	$.30 \times 10^5$
Dual Support Cargo Module (DSCM) off-loaded	8835	112.0	-180.0	0.0	$.61 \times 10^4$	$.68 \times 10^4$	$.61 \times 10^4$

Space Division
North American Rockwell

Table 7.1-4 Mass Properties - Derivative OLS (Configuration 4)

(4) Docked Configurations	Weight Pounds	C.G.			Moment of Inertia - Slug ft^2			Product of Inertia - Slug ft^2		
		x	y	z	I_{xx}	I_{yy}	I_{zz}	I_{xy}	I_{xz}	I_{yz}
A	390,065	411.7	-20.4	-203.1	$.104 \times 10^8$	$.136 \times 10^8$	$.117 \times 10^8$	$-.792 \times 10^6$	$.840 \times 10^6$	$-.348 \times 10^6$
B	375,350	423.4	-14.1	-211.0	$.101 \times 10^8$	$.132 \times 10^8$	$.113 \times 10^8$	$-.950 \times 10^6$	$.104 \times 10^7$	$-.241 \times 10^6$
C	306,515	493.4	-12.1	-168.0	$.855 \times 10^7$	$.991 \times 10^7$	$.947 \times 10^7$	$-.100 \times 10^7$	$-.444 \times 10^5$	$-.134 \times 10^6$
D	274,615	537.7	-13.5	-133.8	$.758 \times 10^7$	$.783 \times 10^7$	$.832 \times 10^7$	$-.966 \times 10^6$	$-.905 \times 10^6$	$-.107 \times 10^6$

Docked Configurations

Docked Configuration	Time Days	Basic Module weight pounds	Tug A		Tug B		Cargo Mod.	
			Full	Empty	Full	Empty	Full	Partial
A	15	190,000	84,615	-	91,900	-	23,550	-
B	31			-	91,900	-	-	8,835
C	31			-	-	31,900	-	-
D	31	190,000	84,615	-	-	-	-	-

Space Division
North American Rockwell

7.2 ENVIRONMENTAL CONTROL/LIFE SUPPORT SUBSYSTEM

7.2.1 REQUIRED CONFIGURATION

The following modules of the OLS will have environmental control/life support subsystem (ECLSS) equipment and/or expendables:

1. Core module, C1A
2. Core module, C1B
3. Power module, PM
4. Crew quarters module, CQM1
5. Crew quarters module, CQM3
6. Control center module, CCM1
7. Control center module, CCM2
8. Galley module, GM
9. Cryogenic storage modules, CSM 1 and CSM 2

The concepts employed to satisfy the ECLSS functions are identical to those of the representative OLS documented in Section 3.2. General description of and deviations from the MSS modules is discussed below.

In general, major ECLSS functions are centralized in two redundant crew quarters modules. Various functions, such as heating, cooling, and circulation are located in each module. The centralized functions save weight since units are not repeated. Smaller repeated units whem summed always weigh more than a few large units because of mounts, supports, etc. The local fans and sensible heat exchangers were localized in order to keep duct size small (especially at the docking port interfaces).

Waste processing units are located in the galley and medical areas since these generally produce large amounts of waste. Also, it is desirable to keep waste processing away from the crew quarters in case of failure.

Table 7.2-1 presents the ECLSS weight and power for an 8-man derivative OLS. The total weights are higher than for a single 27-foot diameter OLS because of additional units for local fans, plumbing, etc.

Summary descriptions of each module are given below.

Table 7.2-1. Derivative OLS Configuration - Required Changes - ECLSS Weight and Power

Module	Weight - pounds			Power-watts 24-Hour Average	Food Included in Expendables
	Total Fixed	Initial 180-Day Spares	Expendables		
Core module, C1A	5,924	265	-	350	
Core module, C1B	5,924	265	-	-	
Power module, PM	271	11	-	150	
Crew quarters module, CQM1	3,956	1,041	750	5,084	200 pounds in backup galley
Crew quarters module, CQM3	3,820	62	-	480	
Control center module, CCM1	532	36	2,000	245	2000 pounds in solar storm shelter
Control center module, CCM2	1,540	66	305	680	
Galley module, GM	1,930	223	3,765	860	3650 pounds
Total*	23,897	1,969	6,820	7,849	

*Above weights have no radiator weight penalty

Space Division
North American Rockwell

Core Modules

The ECLSS equipment in the two core modules consists of air, water, freon, and gas plumbing; a pumpdown system per core; high pressure O_2/N_2; and PGA, umbilicals, and connectors for IVA and EVA. Also, an IVA-EVA airlock is in each of the cores.

Power Module

The power equipment conditioned by the ECLSS is located in part of the core attached to the power boom as well as in the boom. Local fans, ducting, sensible heat exchangers, coldplates, and a separate freon radiator loop are provided for the power conditioning equipment. The boom radiator furnishes the appropriate heat rejection area.

Crew Quarters Module, CQM1

Quarters for four crewmen are provided. This module is one of the major ECLSS modules. Processing equipment for CO_2 removal and reduction, central trace contaminant control, central humidity control, pressure control, central freon-water heat exchangers, water processing, and hygiene equipment are provided. Main sensible heat exchanger and local ventilation fans are provided. Also, a backup galley consisting of microwave oven and reconstitution unit is provided.

Crew Quarters Module, CQM3

Quarters for four crewmen are provided. It is almost identical to CQM1 except that it has no backup galley and has the larger commander's stateroom.

Control Center Module, CCM1

This module contains a heat exchanger package, local ventilation fans, air ducts, fire extinguishers and coldplate loop. A toilet and food reconstitution unit are located in the radiation shelter area of this module.

Control Center Module, CCM2

This module contains a heat exchanger package, circulation fans, air ducts, fire extinguishers, and coldplate loop. In addition, a waste processing unit and sink are in the medical area. Some charcoal for local air purification in the medical area is also included.

Galley Module

This module contains the major ECLSS hardware for food preparation and the laundry hardware. This module contains approximately one-half of the total food storage and includes all of the frozen food storage.

Cryo Storage Modules

These modules serve as a storage area for cryogenics and hydrazine. No ECLSS hardware is located in these modules.

General Requirements

The above module descriptions indicate relatively minor configuration differences between the MSS and derivative OLS. All six-man processing hardware for the OLS must be sized for eight men.

7.2.2 DESIRED CONFIGURATION

The desired ECLSS configuration is no different from the required configuration described in Section 7.2.1 since all desired changes are also categorized as required changes.

7.3 ELECTRICAL POWER SUBSYSTEM

7.3.1 DERIVATIVE OLS CONCEPT

EPS primary and secondary power generation and energy storage is achieved by a combination of solar arrays and secondary nickel-cadmium batteries (SAB). The arrays (total area of 10,000 feet^2) will provide an average power of 20 kilowatts, the 8-man OLS requirement. Approximately 10,165 pounds of secondary batteries (100 ampere-hour cell size) are utilized. Fuel cells and primary AgZn batteries are utilized for emergency/auxiliary power generation.

Primary AgZn batteries provide for the premanning function and fuel cells satisfy emergency and auxiliary power requirements. Backup requirements are satisfied by adequate redundancy design in the primary SAB with the exception of solar-array replacement operations when it will be necessary to operate the auxiliary fuel cells. Power conditioning includes inverters for primary power conditioning and regulation; autotransformers for transforming primary power to standard voltages; and computer-controlled battery charging equipment. Battery charging will be either a modified voltage limited or a regulated pulse technique. Power distribution consists of four identical primary buses (i.e., one per solar array). The primary buses accommodate power transfer to battery charging and secondary bus equipment. Battery and fuel cell power feeds, via inverters, to the primary bus.

A one-line schematic of the EPS is shown in Figure 7.3-1. Its weight and volume are summarized in Tables 7.3-1 and 7.3-2.

7.3.2 DESIRED CONFIGURATION

The required derivative OLS concept is identical to the 12-man MSS concept. This concept can be lightened considerably and still meet OLS requirements (see Section 6.3.2, Minimum Change Concept). The weight and volume of the minimum change concept are summarized in Tables 7.3-3 and 7.3-4. However, the minimum change concept is relatively heavy compared to the representative OLS EPS concept described in Section 3.3. The derivative OLS can be further improved by making the additional modifications described in Section 6.3.2, Extensive Change Concept.

The extensive change EPS concept suggested in Section 6.3.2 is similar to the representative EPS described in Section 3.2 except for the addition of primary batteries. Primary batteries are required for unmanned launch and premanning and are optional for deployment of the solar arrays. Primary batteries will weight approximately 300 pounds. If earth orbit checkout is to be accomplished without the primary EPS, then an additional 1250 pounds of primary batteries will be required.

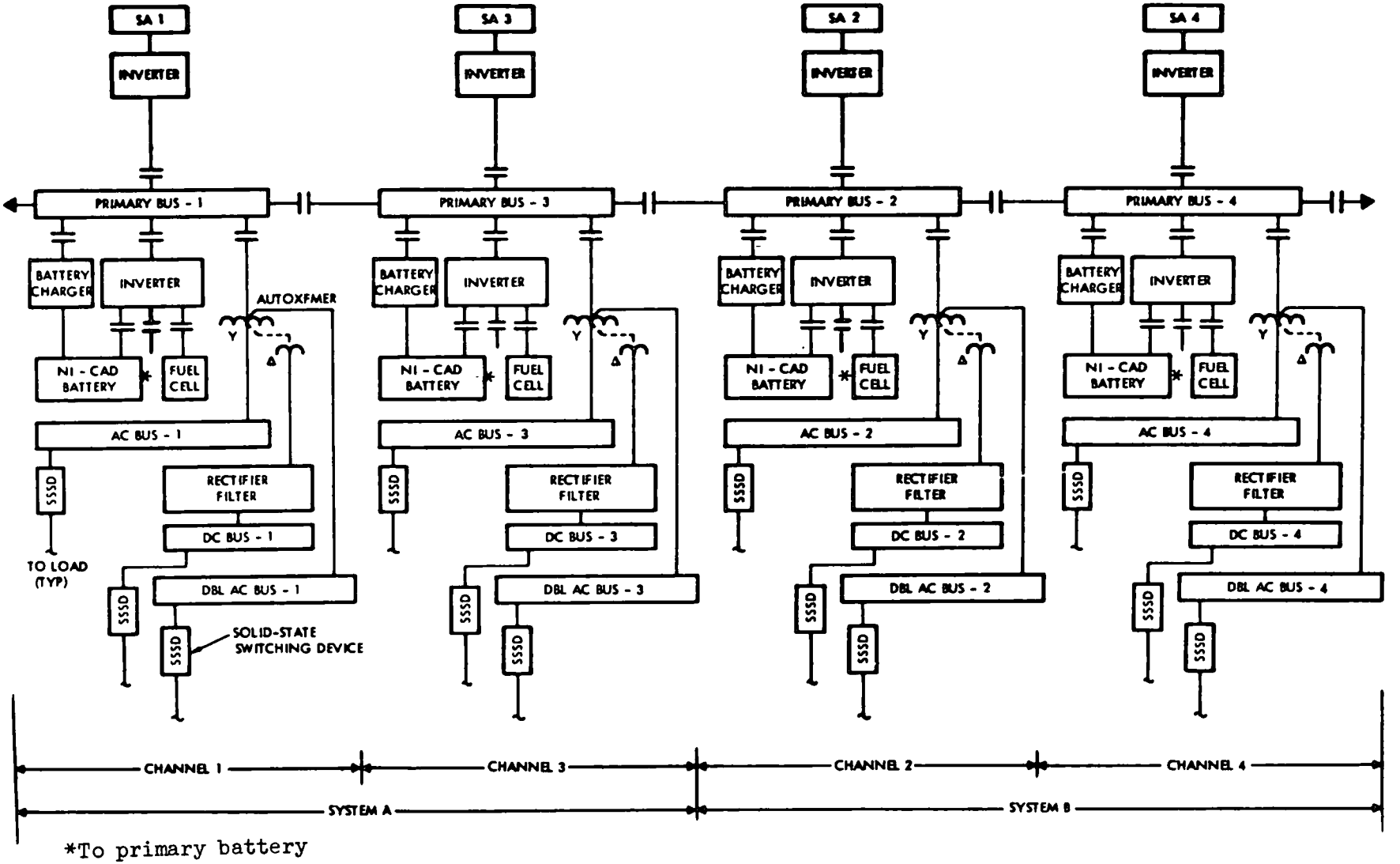


Figure 7.3-1. Derivative OLS EPS (Solar Array/Battery) On-Line Schematic

Space Division
North American Rockwell

Table 7.3-1. EPS Weight Characteristics

Assembly	Unit Weight (lb)	Total Weight 12-Man (lb)
Solar energy conversion		
Solar array	1,180	4,700
Orientation mechanics		287
Array support		375
Mounts and supports		435
Solar array inverters*	30	430
Power conditioning		
Chargers	20	160
Controls	40	160
Inverters	30	240
Secondary bus equipment	60	1,180
Mounts and supports		169
Distribution control and wiring		
Feeders		15
Bus structure and contactors		90
Wiring		4,450
Mounts and supports		557
Energy storage		
Primary batteries*	567	3,740
Secondary batteries*	1,017	10,165
Secondary power (fuel cells)*	160	704
Lighting*	4	839
Total	---	28,701

*Weight includes installation penalty

Table 7.3-2. EPS Volume Characteristics

Assembly	Size (inches)			Units	Volume (ft^3)
	Height	Width	Depth		
Solar array	25 in. D x 31.1 ft			4	422
Orientation mechanism					2.9
Array support					3.7
Mounts and supports					4.4
Solar array inverters	12	12	9	8	7.5
Power conditioning					
Chargers				8	2.0
Controls				4	2.0
Inverters	12	12	9	8	6.0
Secondary bus equipment	9	12	10	20	11.8
Distribution and control					
Feeders					.3
Bus structure/contact					.6
Wiring					43.5
Lighting				210	42.9
Primary batteries				6	30.0
Secondary batteries				10	97.5
Fuel cells	12	12	38	4	12.0
Mounts and supports					5.6
Total (cubic feet)					695.

Table 7.3-3. Minimum Change Concept EPS Weight Summary

			Initial Launch	Ten Years
Solar energy conversion			5,815	10,535
Solar cell rollup array ($10{,}000\ ft^2$)		4,720		
Orientation mechanism		286		
Array support (boom)		374		
Mounts and supports		434		
Fuel cells (emergency)			607	607
Fuell cells (4)		552		
Mounts and supports		55		
Energy storage*			5,569	20,758
Batteries (4)		5,063		
Mounts and supports		506		
Power conditioning			1,735	4,829
Chargers	Core	160		
Controls	Core	80		
Inverters (4 - 7.5 kva)	Core	378		
Secondary bus	Core	300		
Mounts and supports	Core	112		
Inverters (4-12.5 kva)	Boom	629		
Mounts and supports	Boom	76		
Distribution control and wiring			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	2,230
Lights		544		
Mounts and supports		54		
Total (pounds)			17,775	42,526

*Primary batteries are not included. If only unmanned launch and premanning power and power for the deployment of the solar arrays are required, then weight of the primary batteries will be about 300 pounds. If earth orbit checkout power is also required, then an additional 1250 pounds of batteries will be required. It is assumed that the primary batteries will be returned to earth after their use in earth orbit.

Table 7.3-4. Minimum Change Concept, EPS Volume Summary

	Size (inches)			Volume (feet)^3		Estimated Replacements (10 Years)
	Height	Width	Depth	Unit	Initial Launch	
Power generation						
Solar array	25 in. dia. x 31.1 feet external			105.5	422	Replace all units 0 to 1 time/10 years
Orientation hardware						
Fuel cells	12	12	38	4.8	19.3	None
Batteries**					46.0	Replace 100% each 2-1/2 years or 3 times per 10 years
IFRU cells	9	9	3.75	0.17	-	
Power conditioners						
Inverters (boom)	8	9	12	.5	10	Replace 100% each 3-1/3 years or 2 times per 10 years
Inverters (core)	8	9	12	.5	6	
EPS/battery control	9.5	17	21	.5	2	
Battery charger	9.5	17	21	.3	5	
Secondary bus structure						
Auto transformers Rectifier-filters Circuit breakers	9	12	16	1.0	4	
Power distribution						
Contactors	4	4	5	0.062	8	Replace contactors 2 times/10 years
Feeders						
Wiring						
Lighting					17	Replace fixtures 4 to 6 times/10 years
Total					110*	

*Excluding solar array stowage
**Excluding primary batteries (1550 lb of primary batteries have a volume of ~15 ft^3)

Space Division
North American Rockwell

Cost comparisons of the minimum and extensive change derivative EPS concepts are summarized in Tables 7.3-5. Table 7.3-5 is based on secondary batteries replaced three and the RFC's two times per ten years. The cost comparison indicates that due to the high resupply cost associated with the minimum change concept which utilizes batteries for energy storage, the overall relative cost of the extensive change concept is considerably lower ($\approx$ 30 percent).

Table 7.3-5. Minimum and Extensive Change Desired EPS Cost Evaluation Summary

Item	MSS Minimum Change Concept	Extensive Change Concept
Initial Cost	0.590	0.635
10-Year Resupply Cost	0.745	0.365
Overall Cost	1.335	1.000

Initial
- Nonrecurring
- Recurring
- Launch penalty @ $1000/lb

} Minimum change derivative lowest initial cost

Resupply
- Recurring*
- Launch penalty

} Extensive change derivative lowest resupply cost

Overall
- 10-year life
- Initial plus resupply

} Extensive change derivative has overall 10-year cost advantage

*Replace batteries 3 times/10 years and RFC's 2 times/10 years

7.4 INFORMATION SUBSYSTEM

7.4.1 IDENTIFICATION OF FUNCTIONS/ASSEMBLIES

The Information Subsystem (ISS) is composed of the Data Processor, Display/Control, Software and Communications Assemblies and provides support to all other subsystems for automation, performance monitoring and configuration control. It further provides the facilities with on-board crew monitoring, management and vehicle and mission operations control as well as "housekeeping" support to integral or attached experiment module/s (or decks).

The Communications Assembly includes the RF, internal Audio/Video distribution, and RF Radar subassemblies which provide links for audio, video, data and command transmission and reception, the intercom, television and closed-circuit television facilities, and back-up for shuttle terminal guidance maneuvers.

7.4.2 MAJOR FUNCTIONAL REQUIREMENTS

The derivative OLS ISS has the same major functional requirements as identified for the representative OLS configuration (See Figure 7.4-1) and the same total performance requirements listed below:

STATION/SUBSYSTEMS

Memory	Required Capacity
Operational	154 K words
Mass	256 K words
Archival	2 x 10^9 bits per day

EXPERIMENT CONTROL AND/OR OPERATIONS

Memory	Required Capacity
Operational	30 K words
Mass	3 M words
Archival	230 x 10^9 bits per day

TOTAL REQUIRED CAPACITY

Memory	Required Capacity
Operational	184 K words
Mass	3.3 M words
Archival	232 x 10^9 bits per day

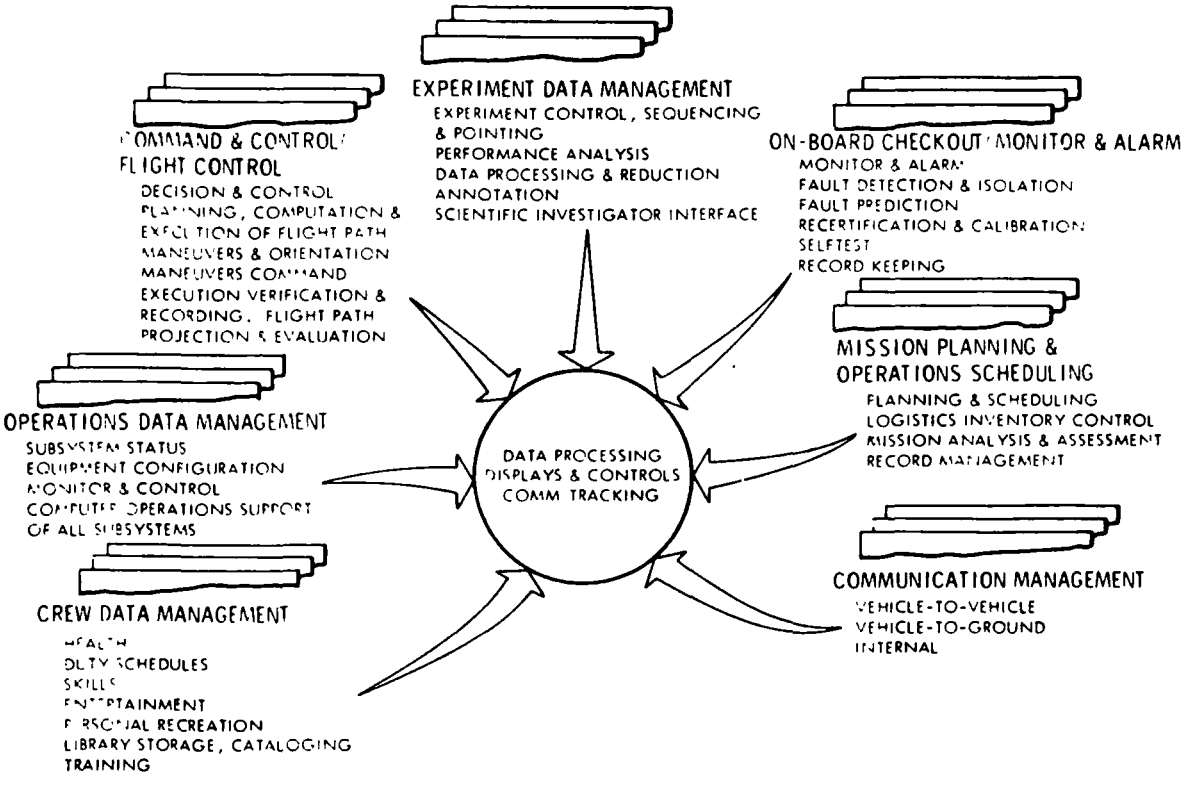


Figure 7.4-1. Information Subsystem Functions

There is, however, a significant difference in the method in which the requirements are implemented. The modular concept of the derivative OLS does not lend itself to the monolithic central computer (multiprocessor) concept proposed for an integral-launched vehicle. Instead the centralized computer configuration has been divided into a number of decentralized modular processors which effectively introduces a heirarchy of processing.

The operating memory requirements of 184K words were satisfied in the OLS by a multiprocessor memory capacity of 256K words. This capacity is retained for the derivative concept and is divided into the three categoric processing levels of central, local, and preprocessor. The preprocessors each have 4K, 16 bit words of internal memory equivalent to 2K, 32 bit words. It is estimated that up to 37 preprocessors will be needed, distributed by subsystem subassembly location, providing the equivalent of 74K words. The local processors have 16K, 16 bit words, or the equivalent of 8K, 32 bit words each. Five local processors are estimated for the derivative OLS providing 40K words capacity. The remaining delta capacity requirement is retained in the central processor. The operating memory of the central processor is also subdivided into two control centers, only one of which will be active (128K words total). Central processor - 1 will retain the planning, maintenance, and supervisor programs while central processor - 2 will be used for backup and experiment operations or experiment data processing activities.

The mass memory in the central processor was sized by assigning to it the flight control, subsystems control, plan/maintenance, supervision and library programs. Some capacity is available for experiment control and operations, but the majority of the experiment-related programs are allocated to the dedicated experiment mass memory located in Core Module 1A.

The preceeding subdivision of the ISS was chosen over the centralized computer configuration of the representative OLS in order to decentralize the more routine functions to a lower computer level and provide commonality of component design. This implementation retains the advantages of the integrated Information Subsystem and affords the following programmatic aspects:

a) The total weight and power characteristics are increased slightly, however, there is no serious impact since only small increments appear at each launch while power is related to duty schedule adjustment of simultaneous operations.

b) The equipment and computer program assembly complexity is significantly reduced providing a reduction in overall development costs since 1) each modular computer is the same except for operating memory capacity and the peripheral equipments connected to the Input/Output (IO) units, and 2) each local processor contains only the computer programs needed for that module, subsystem or experiment, therefore, simplifying the executive or supervisor program design.

c) The computer program components (modules) can be associated with the vehicle or experiment element that it supports.

The following Tables, 7.4-1 through 7.4-3, summarize the ISS performance characteristics at the subsystem level while Table 7.4-4 lists the equipment requirement at the component or subassembly level.

Table 7.4-1. Communication Performance Characteristics

RF Comm. Signals		Transmit Receive
Voice	3 simultaneous, 400Hz to 5KHz	T/R
Video	1 Commercial Standard, 525 TV line color	T/R
Data	1-50Kbps, 1-500Kbps, 1-5000 Kbps	T
Command	1-200bps	R
Text	1-1000bps	R
Ranging	1-500KHz PSK	T/R
Facsimile	1-500KHz	T/R
Internal Communication Signals		
Voice	6 simultaneous intercom 4 simultaneous CCTV 1 entertainment (recorded) TV	
Public Address - 1 channel		
Radar Capability		
20 n-mile, ± 500-foot, radial, resolution, 0.05 ft/sec range rate resolution, minimum range of 1000 feet. Coverage: 120° cone on sides with Tug docking ports.		

7.4.3 ISS CHARACTERISTICS

Characteristics of the per unit ISS components/subassemblies are shown in Table 7.4-5. Identified for each subsystem are the component/subassembly weight volume and peak power requirements. Table 7.4-6 tabularizes the ISS characteristics at a station module level. Component/subassembly quantities defined in Table 7.4-4 were used in conjunction with the per unit estimates of Table 7.4-5 to achieve these results. An overall summary of the ISS characteristics are given in Table 7.4-7. Typically, weight, volume, and power are listed for the four major ISS assemblies, 1) data processing, 2) display and controls, 3) communications, and 4) software. In general, assembly weight and power requirements are somewhat greater than those previously identified for the representative OLS at the eight man level. This is not expected to be of significant impact since only small increments of the total station weight are launched at a given time.

Table 7.4-2. Data Processing Performance Characteristics

Data Processing Assembly (DP) Capability
1. Acquisition: Access to all data through data buses Peak Data acquisition rate - 10 MBPS 2. Processing: Data processing is performed in one of three processing levels Data accepted at a rate of 10 MBPS 3. Distribution: Delivers commands and data, via digital data bus system, to using peripheral equipment. Commands decoded through RACU and distributed to the user. Maximum output data rate - 10 MBPS Central timing provided
Central Processor
Functions: Primary interface to crew Supervises and controls interactions between local processors Maintains subassembly operation in event of local processor failure Processor Unit Type: Multiprocessor (2-CPU) Instruction execution time: Register to operations memory - 500 nanoseconds Add or subtract - less than 1 microsecond Multiply - less than 3 seconds Divide - less than 10 microseconds I/O Unit Transfer rate: 10 MBPS Channels: 64 RACU: 1 mass memory Memory

	Operating (2)	Mass (2)	Archive (2)
Type:	Plated Wire	Plated Wire	Tape
Word Length:	32 Bits	32 Bits	32 Bits
Capacity:	32K words/mod.	128K words/mod.	2.4×10^9 bits/cart.

Table 7.4-2. Data Processing Performance Characteristics (Cont'd)

Local Processor

Functions:

Flight and subsystems control of module
Executes fault isolation instructions
Interacts (through central processor) with other local processors)

Processor Unit

Type: General Purpose
Instruction execution time: Same as preprocessor

I/O Unit

Channels: Same as preprocessor

Memory

Operating (1 or more modules)
Plated wire
16 bits/word
16K words/module

Terminal Unit: Display/Checkout

Preprocessor

Functions:

Controls subsystem or assembly
Performs fault detection of subsystem or assembly

Processor Unit

Type: General Purpose
Instruction execute time: Add or subtract - 2.4 microseconds
Multiply - 30.4 microseconds
Divide - 10.4 microseconds

I/O Unit

Channels: 16-bit parallel party line
1 serial channel

Memory

Internal
MOS
16 bits/word
4K words

Table 7.4-3. Displays and Controls Performance Characteristics

Command Console
Color CRT: 5 mHz commercial quality color TV 3 color graphics Color alphanumerics 50 characters/line 30 lines 30 Hz refresh rate
Alphanumeric displays: 50 characters/line 15 lines/display 5 x 7 DOT matrix
Keyboards: Analog and digital control Solid stage switching
Hand controller: 3-axis manual control
M/A displays: Parameters displayed in numerical decimal values - expressed in engineering units and English words
Remote Terminal Unit
Color CRT: 1000-character display 30 Hz refresh rate Compatible with commercial standard for color and B-W TV
Keyboards: Alphanumeric
M/A displays: Parameters displayed in numerical decimal values - expressed in engineering units and English words
Hard Copy Printer
Controlled from console keyboard or computer Prints 600 characters/second; 6 lines/inch
Microfilm Projector
Displays microfilm data for procedures, drawings, schematics, and other information stored in data bank. Frame selection provided by numeric keyboard.
Commander's Stateroom Console
Unit combines a Remote Terminal Unit, M/A Display Unit, an Audio Video Unit, and Alphanumeric keyboards.
Local M/A Displays
Displays distributed throughout space station. Parameters displayed in numerical decimal values and expressed in engineering units and English words.
Local Clock Displays
Units distributed throughout space station displaying month, date, and time in hours and minutes in one-inch characters. Unit synchronized by Central Timing Unit

Table 7.4-4. Component/Subassembly Equipment Requirements

	Power Module	Core Module #1	Core Module #2	Crew Quarters Module 1	Crew Quarters Module 3	Galley Module	Control Center Module 1	Control Center Module 2	Cargo Module	TOTAL (Units)
Data Processing										
Central Processor (CP)							1	1		2
Local Processor (LP)	1	1		1	1	1				5
Preprocessor (PP)	8	4	4	7	7	7				37
Archive Memory	1	1					2			4
Experiment Related Mass Memory Modules		7						2		9
Central Timing Unit (CTU		1						1		2
RACU	8	5	5	6	6	6	12	12	4	64
Displays and Controls										
Console							1	2		3
Microfilm Unit							1	1		2
Remote Term. Unit Cmdr. Console (RTU)		1	1	1			1	1		5
Monitor/Alarm Unit									1	1
Local M/A Displays	1	2	2	4	4	2	2	2	2	21
Local Clock Displays	1	2	2	4	4	2	2	2	2	21
Communications										
Semidirective Antenna		2	2				3	3		10
Approach Radar		1	1							2
Trandsponder (semidirective)		2	2				3	3		10
Transponder (parabolic)	1			1			1			3
Modulation Processor								1	1	2
Central Sw. Unit - Self-Test								1	1	2
Hard Wire Intercom.	1	1	1	1	1	1	1	1	1	9
AVU	2	2	2	4	4	4	5	5	4	32
Rack							1	1		2
Peripheral Group										
Video Recorder							1	1		2
Audio							1	1		2
Facsimile Unit							1	1		2
TV Camera							1	1	1	3
TV Monitor				4	4	2	1	1		12
Paging Amplifier							1	1		
Interface Unit (cargo module)							1	1		2
High-Gain Antenna (w/deploy.boom)	1			1			1			3
Software										
Tapes	10	5	5	10	10	10	10	10	10	80

Table 7.4-5. ISS Characteristics (Component/Subassembly Level)

Item	Dimensions (in)	Unit Weight (lb)	Volume (ft^3)	Peak Power (watts)
DATA PROCESSING				
Central Processor	25 x 15 x 14	390	3.03	288
2 - Processor Unit (10 lb each)	9 x 3.6 x 14			30/each
2 - I/O Unit (10 lb each)	9 x 3.6 x 14			25/each
2 - Operating Memory (50 lb each)	9 x 9 x 14			50/each
2 - Mass Memory (125 lb each)	12 x 18 x 14			39/each
Local Processor	16 x 18 x 14	110	2.33	355
1 - Processor Unit (10 lb)	9 x 3.6 x 14			30
1 - I/O Unit (10 lb)	9 x 3.6 x 14			25
1 - Operating Memory (50 lb)	9 x 9 x 14			50
1 - Terminal Unit (40 lb)	12 x 12 x 18			250
Preprocessor	4 x 4 x 2.5	3	.02	10
Archive Memory	14 x 14 x 16	40	.68	45
CTU	9 x 6 x 14	18	.43	30
RACU	3 x 3 x 4	3	.02	4
DISPLAY AND CONTROLS				
Console	48 x 31 x 50	375	43.05	425
Microfilm Unit	15 x 15 x 12	35	1.56	50
RTU (cmdr. console)	12 x 12 x 18	40	1.5	250
M/A Unit	10 x 3.5 x 45	6	.09	6
COMMUNICATIONS				
Parabolic Antenna with Deployable Boom	5 feet diameter	240	20.41	100
Semidirective Antenna	6 in. dia. x 8 in.	1	.13	---
Transponder (parabolic)	12 x 16 x 5	29	.55	100
Transponder (semidirective)	12 x 16 x 5	29	.55	100
Modulation Processor	12 x 12 x 14	50	1.17	50
Central Switch Unit	18 x 27 x 14	120	3.94	70
Hard Wire Intercom.	6 x 6 x 2	10	.41	10
AVU	6 x 6 x 2	9	.17	6
Racks	72 x 36 x 17	200	25.56	---
Video Recorder	14 x 18 x 9	60	1.31	45
Audio Recorder	12 x 12 x 6	15	.51	20
Facsimile Unit	18 x 12 x 14	40	1.75	200
TV Camera	8 x 6 x 12	3	.33	50
TV Monitor	9 x 9 x 10	10	.47	50
Paging Amp.	6 x 6 x 8	10	.17	50
Interface Unit (cargo module)	12 x 12 x 12	20	1.0	20
Approach Radar	24 x 24 x 4	25	1.34	125
SOFTWARE				
Tapes				
6 - Video	14 x 14 x 3	10	.34	---
4 - Audio	14 x 18 x 4	10	.58	---

Table 7.4-6. ISS Characteristics (Module Level)

Item	Quantity	Weight (lb)	Volume (ft^3)	Peak Power (watts)	Module
DATA PROCESSING					
LP	1	110	2.33	355	Power
PP	8	24	.16	80	
AM	1	40	.68	45	
RACU	8	24	.16	32	
		198	3.33	512	
Mass Memory Module	7	875	12.25	273	Core #1
LP	1	110	2.33	355	
PP	4	12	.08	40	
AM	1	40	.68	45	
CTU	1	18	.43	30	
RACU	5	15	.10	20	
		1070	15.87	763	
PP	4	12	.1	40	Core #2
RACU	5	15	.1	20	
		27	.2	60	
LP	1	110	2.33	355	Crew Quarters 1, 3 and Galley
PP	7	21	.14	70	
RACU	6	18	.12	24	
		149	2.59	449	
CP	1	390	3.03	288	Control Center 1
AM	2	80	1.36	90	
RACU	12	36	.24	48	
		506	4.63	426	
Mass Memory Module	2	250	Installed in 2nd Console	78	Control Center 2
CP	1	390	3.03	288	
CTU	1	18	.43	30	
RACU	12	36	.24	48	
		694	3.7	444	
RACU	4	12	.08	16	Cargo
DISPLAY AND CONTROLS					
RTU	1	40	1.5	250	Core #1
RTU	1	40	1.5	250	Core #2
RTU	1	40	1.5	250	Crew Quarters 1
Console	1	375	43.05	425	Control Center 1
Microfilm	1	35	1.56	50	
RTU	1	40	1.5	250	
		450	46.11	725	
M/A Unit	1	6	.09	6	Cargo
Console	2	750	86.1	850	Control Center 2
Microfilm	1	35	1.56	50	
RTU	1	40	1.5	250	
		825	89.16	1150	

Table 7.4-6. ISS Characteristics (Module Level) - Continued

Item	Quantity	Weight (lb)	Volume (ft^3)	Peak Power (watts)	Module
COMMUNICATIONS					
Hardwire Intercom	1	10	.41	10	
AVU	2	18	.34	12	
High-Gain Antenna	1	240	20.41	100	Power
Transponder (high-gain)	1	29	.55	100	
		297	21.71	222	
Semidirective Antenna	2	2	.26	---	
Approach Radar	1	25	1.34	125	
Transponder (semidirective)	2	58	1.10	200	Core 1 and 2
Hardwire Intercom	1	10	.41	10	
AVU	2	18	.34	12	
		113	3.45	347	
High Gain Antenna	1	240	20.41	100	
Transponder (high gain)	1	29	.55	100	
Intercom	1	10	.41	10	Crew Quarters 1
AVU	4	36	.68	24	
TV Monitor	4	40	1.88	200	
		355	23.94	434	
Intercom	1	10	.41	10	
AVU	4	36	.68	24	Crew Quarters 3
TV Monitor	4	40	1.88	200	
		86	2.97	234	
Intercom	1	10	.41	10	
AVU	4	36	.68	24	Galley
TV Monitor	2	20	.94	100	
		66	2.03	134	
Semidirective	3	3	.39	---	
Transponder (semidir.)	3	87	1.65	300	
Transponder (high gain)	1	29	.55	100	
High Gain Antenna	1	240	20.41	100	
Modulation Processor	1	50	1.17	50	
Central Switch Unit	1	120	3.94	70	
Intercom	1	10	.41	10	Control Center 1
AVU	5	45	.85	30	
Rack	1	200	25.56	---	
Video Rec.	1	60	1.31	45	
Audio Rec.	1	15	.51	20	
Facsimile Unit	1	40	1.75	200	
TV Camera	1	3	.33	50	
TV Monitor	1	10	.47	50	
Paging Amplifier	1	10	.17	50	
		922	59.26	1075	
Semidirective	3	3	.39	---	
Transponder (semi)	3	87	1.65	300	
Modulation Processor	1	50	1.17	50	
Central Switch Unit	1	120	3.94	70	
Intercom	1	10	.41	10	
AVU	5	45	.85	30	
Rack	1	200	25.56	---	Control Center 2
Video Rec.	1	60	1.31	45	
Audio Rec.	1	15	.51	20	
Facsimile Unit	1	40	1.75	200	
TV Camera	1	3	.33	50	
TV Monitor	1	10	.47	50	
Paging Amp.	1	10	.17	50	
		603	38.51	875	

Table 7.4-6. ISS Characteristics (Module Level) - Continued

Item	Quantity	Weight (lb)	Volume (ft^3)	Peak Power (watts)	Module
COMMUNICATIONS (CONT'D)					
Intercom	1	10	.41	10	
AVU	4	36	.68	24	Cargo
TV Camera	1	3	.33	50	
Paging Amp.	1	10	.17	50	
		59	1.59	134	

Table 7.4-7. ISS Characteristics Summary

Module	Assembly				Total
	Data Processing	Display and Controls	Communica-tions	Software	
POWER MODULE:					
Weight (lb)	198		297	100	595
Volume (ft^3) Typical	3.33		21.71	4.4	29.44
Peak power (watts)	512		222		734
CORE MODULE #1:					
Weight (lb)	1070	40	113	50	1273
Volume (ft^3)	15.87	1.5	3.45	2.2	23.02
Peak power (watts)	763	250	347		1360
CORE MODULE #2:					
Weight (lb)	27	40	113	50	230
Volume (ft^3)	.2	1.5	3.45	2.2	7.35
Peak power (watts)	60	250	347		657
CREW QUARTERS MODULE 1:					
Weight (lb)	149	40	355	100	644
Volume (ft^3)	2.59	1.5	23.94	4.4	32.43
Peak power (watts)	449	250	434		1133
CREW QUARTERS MODULE 3:					
Weight (lb)	149		86	100	335
Volume (ft^3)	2.59		2.97	4.4	9.96
Peak power (watts)	449		234		683
GALLEY MODULE:					
Weight (lb)	149		66	100	315
Volume (ft^3)	2.59		2.03	4.4	9.02
Peak power (watts)	449		134		583
CONTROL CENTER MOD. 1:					
Weight (lb)	506	450	922	100	1978
Volume (ft^3)	4.63	46.11	59.26	4.4	114.4
Peak power (watts)	426	725	1075		2226
CONTROL CENTER MOD. 2:					
Weight (lb)	694	825	603	100	2222
Volume (ft^3)	3.7	89.16	38.51	4.4	135.41
Peak power (watts)	444	1150	875		2469
CARGO MODULE:					
Weight (lb)	12	6	59	100	177
Volume (ft^3)	.08	.09	1.59	4.4	6.16
Peak power (watts)	16	6	134		156
TOTAL Weight (lb)	2954	1401	2614	800	7769
Volume (ft^3)	35.58	139.86	156.91	35.2	367.55
Peak power (watts)	3568	3642	3802		11012

The components generally represent the technological status assumed for the earth orbiting Modular Space Station (MSS). As such, the evaluation will be conservative in certain areas. These are expected to be primarily associated with the high-gain antennae, the approach radar, and information storage.

However, all the features of the representative OLS in terms of safety (two control centers) maintenance, commonality, flexibility (common data bus, interchangeability of components) and operational availability have been retained with the additional feature of less implementation complexity.

Figure 7.4-2 is a listing of the ISS at the Assembly/Unit level. Figure 7.4-3 is a partial mechanization block diagram of the Data Processing Assembly (DPA) to illustrate the hierarchy of processing capability. All levels, including the Remote Acquisition and Control Unit (RACU) and preprocessor are within the DPA, and are similar components.

Two new derivative OLS components which do not appear in the representative OLS multiprocessor configuration have been identified. They are the preprocessor and the local processor. The preprocessor is conceived as a small integral mini-computer, standardized, with internal memory, and display and control features. The preprocessor is used in a dedicated manner to control a subsystem or subassembly. The following general data is considered typical of the preprocessor characteristics:

Weight	3 pounds 4" x 4" x 2-1/2"
Power	10 to 13 watts No cooling required
Environment	MIL-E-5400K, Class 2
MTBF	25000 hours
Integral display and entry keyboard	
Execute times:	Add 2.4 microsec Divide 30.4 microsec Multiply 10.4 microsec
Internal Memory	4K, 16-bit words, MOS 1.6 microsec cycle time
In/Out	16-bit parallel, party line and one serial channel

The local processor is made up of the same components as the preprocessor; at a minimum it would have one each of the processing unit, operating memory unit, the input/output unit, and a remote terminal unit. (It also has a complimentary software assembly assumed to be one or more tape cartridges and a tape deck). The characteristics of a typical local

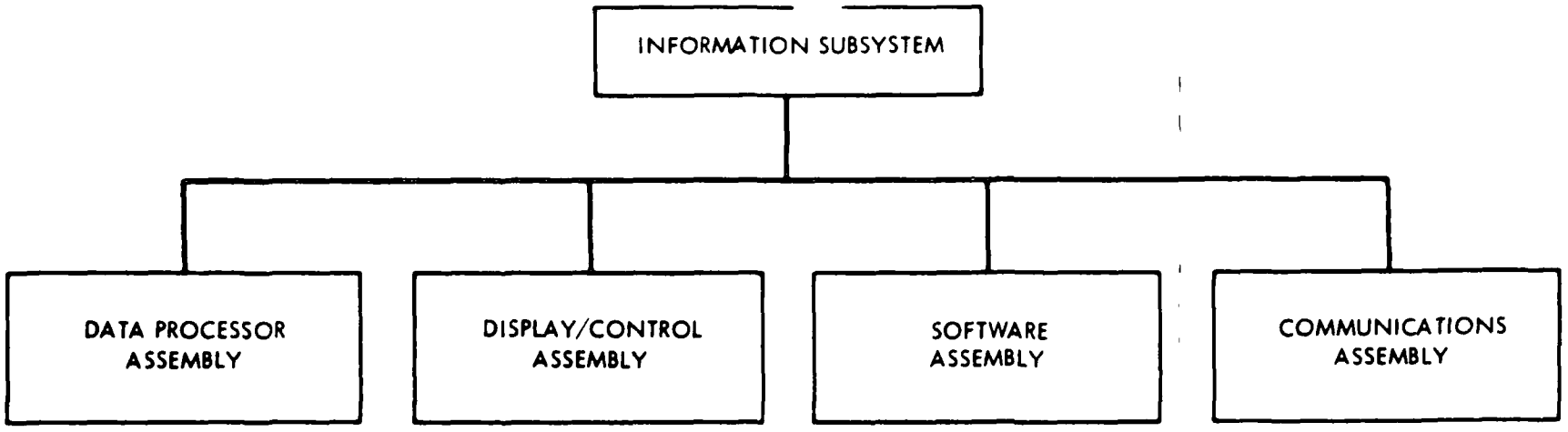


Figure 7.4-2. ISS Equipment Complement

Space Division
North American Rockwell

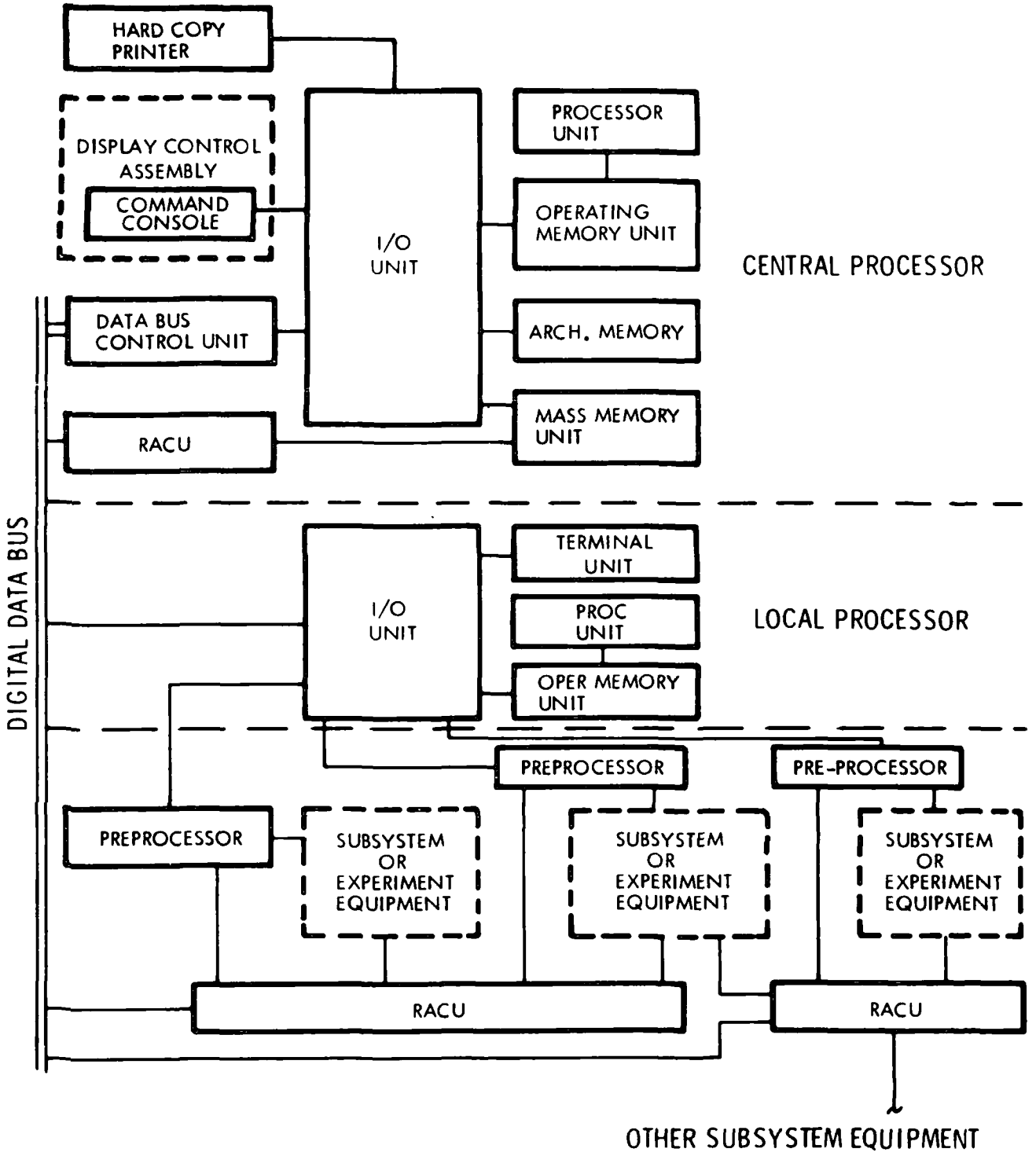


Figure 7.4-3. Mechanization Diagram, Data Processing Assembly

processor are synthesized by adding an extended-plated-wire memory component to the preprocessor. The extended memory capacity is 16 x 10^3 words of 16 bits each, weighs 12 pounds and requires an average power of 5 watts. The local processor is, in general, dedicated to a module, and controls the subsystems and/or experiment operations within that module. It also interacts, through the central processor, with other module local processors. Local processors may vary in the quantity of operating memory and the peripheral attachments (tape decks, printers, display and entry devices) depending upon the application.

The central processor (there are two, one in each control center) uses the same component units as in the representative OLS and in addition has part of the total mass memory capability for station keeping and experiment control and operation. The mass memory (two modules per central processor) is installed in the lower half of the main display consoles, in Control Center Modules 1 and 2. The majority of the experiment related mass memory (7 modules) however, is installed in an area allocated within Core Module 1 as specified in Table 7.4-4. The two additional mass memory modules dedicated to experiment control and operation, completing the 13 mass memory module requirements, is installed within the second console in the Control Center Module 2. In general, the central computer supervises the local processors, and controls interactions between them, and to the outside world. It also supplies the capability to provide computation and data functions that support spacecraft and mission management operations. Both central and local processors have access to the mass memory through the common data bus. The central processor has access to the local processors and, via the RACU, to measurement and control points within the affected subassemblies. Thus, in the event of a local processor failure, the central processor could override, or turn off the local processor and maintain the subassembly operation until a replacement is made.

The total control center operating memory size was reduced by allocating some of the functions to the preprocessors and local processors. What remains is reduced to functions that need not be simultaneous; i.e., the fault isolation/recertification, the periodic navigation calculations, the periodic planning and scheduling functions, and the overall DPA supervisor, or systems monitor. Note that by reducing the size of the central processor the multiprocessor supervisor program is far less complex and less expensive to prepare. All of the listed functions can time-share 64K words of operating memory; also, if two functions must be simultaneous, the secondary control center central processor can be used by displacing the experiment processing needs for that relatively short time.

Note that either central processor is still a multi-processor (in the broad sense of that term) in that either processing unit has access to either of the Operating Memory (O/M) Units. However, the Processing Units (PU) are not operating simultaneously; either one can be "boss", but not concurrently. The second PU is an on-line standby unit, ready to take over if the operating unit fails. The two central processors are independent and share only the mass memory, not I/O or O/M units.

Figure 7.4-4 is a block diagram of the communication assembly. A number of semi-directive antenna subassemblies, each including the antenna, S-band transponder, power amplifier, diplexer and Pseudo Random Noise (PRN) range code generator are installed at locations on the periphery of the vehicle. The approach radar is similar to the semi-directive subassembly with the cosine electronics added. Each subassembly is capable of handling the complete RF communication requirements (see Table 7.4.2-1) on any one of 5 duplex S-band frequencies. As with all other vehicle equipment, monitoring and control is through a RACU.

The modulation processor assembles and disassembles the several data signals, multiplexes and connects them to the selected transponder under Central control. The modulation processor accepts and provides signals to the audio-video bus, the entertainment-paging bus and, through a RACU, to the central processor which formats the digital data signals.

The high gain antenna subassemblies with deployable boom are located on the power module, crew quarters module 1, and control center module 1, and include the same equipment as the semi-directive antennae.

The various recorders, monitors, intercom, telephone and facsimile signals are interconnected by the central-switching unit, similar to a telephone switchboard. Central timing is provided along the digital data bus.

The high gain antenna system consists of three 5-foot diameter parabolic dishes. One is mounted on crew quarters module 1, one on control center module 1, and one on the power module.

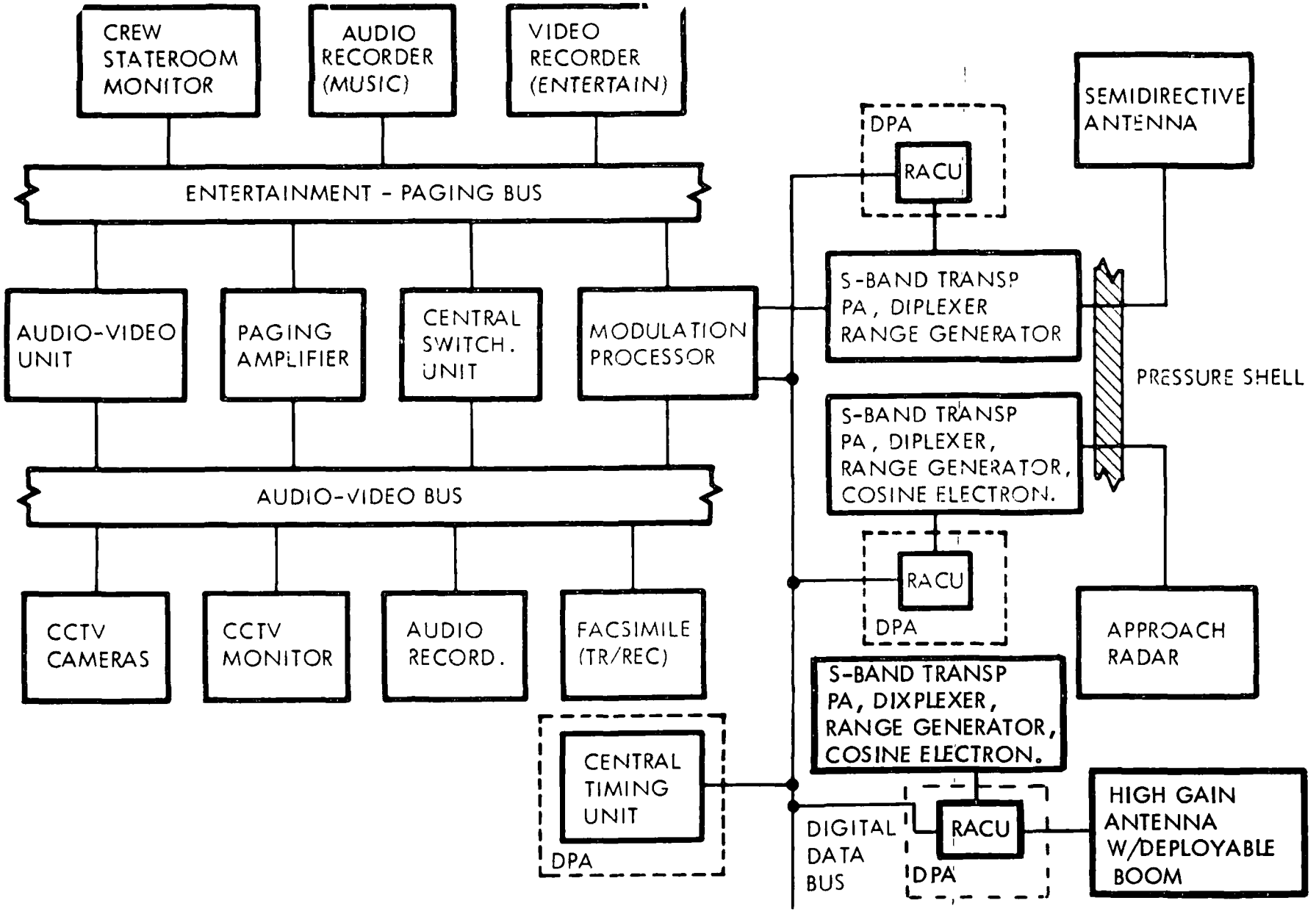


Figure 7.4-4. Communications Assembly Block Diagram

Space Division
North American Rockwell

Page Intentionally Left Blank

7.5 GUIDANCE AND CONTROL SUBSYSTEM

The configuration for the guidance and control (G&C) subsystem of the derivative OLS is identical to the representative OLS configuration described in Section 3.4 with the exception that the MSS CMG array is used rather than the somewhat smaller CMG's required for the representative OLS. The MSS uses 3000-foot-pound-seconds CMG's weighing about 1600 pounds whereas the derivative OLS requires only 2600-foot-pound-seconds CMG's weighing about 1500 pounds. The cost of using the heavier MSS CMG's on the derivative OLS is more than compensated for by the avoidance of the additional development and qualification costs associated with using the slightly smaller size. The derivative OLS G&C physical characteristics are summarized in Table 7.5-1.

Table 7.5-1. G&C Physical Characteristics

Subassembly	No.	Weight (lb)	Avg Power (watts)	Volume (cu ft)	Physical Location Per Fig. 6.5-4
Telescope/sextant	1	88	0	2.6	C1B
Star tracker	2	66	30	2.2	C1B
Landmark tracker	1	33	30	1.1	C1B
Horizon tracker	4	61	34	0.6	C1B
Radar altimeter	1	35	54	2.0	C1B
IMU	1	45	140	1.07	C1B
Docking laser	2	56	0	2.6	C1B
RCS drivers	5	50	40	0.8	C1A & C1B
CMG's	3	1610	155	185.0	C1B
Preprocessor	6	168	84	2.4	C1A & C1B
Nav base	1	17	0	2.2	C1B
Calibration equipment	4	120	25	1.0	C1B
Total		2349	592	203.5	

Page Intentionally Left Blank

7.6 REACTION CONTROL SUBSYSTEM

The derivative OLS reaction control subsystem (RCS) will be very similar to the representative OLS RCS discussed in Section 3.5. The primary difference is in the physical layout. This is necessitated by the modular arrangement of the derivative concept.

7.6.1 DERIVATIVE OLS DESIGN WITH REQUIRED MODIFICIATIONS

The derivative concept for cryogenic storage will be designed to operate at a vapor pressure of 100 psia. The installation of the vapor shield propellant tanks will be divided into two separate areas for safety. The forward area will consist of two parts. There will be six 25.5-cubic foot vapor shield tanks in a storage module in the power boom. Four of these will contain nitrogen and two will contain oxygen. In addition, there will be a permanently docked cryo storage module (CSM 1) on a forward docking port of core module 1A. In an unpressurized section of CSM 1 will be three 89.8-cubic foot hydrogen tanks and one additional 25.5-cubic foot oxygen tank. On a docking port at the aft end of core module 1B is an additional cryogenic storage module (CSM 2). In an unpressurized section of CSM 2 there are five additional 89.8-cubic foot tanks; one for oxygen, three for hydrogen, and one for nitrogen. The nominal temperature of the tanks will correspond to the nominal cryogenic vapor pressure. The nominal temperature of the oxygen will be 204 R, the temperature of the hydrogen will be 52 R, and the temperature of the nitrogen will be 177 R. The vapor pressure will cause the fluids to flow from the tanks on demand. The flow will be either liquid or gas to the propellant conditioners.

The nitrogen will flow to the ECLSS (ECLSS has the nitrogen conditioning assembly). The propellant conditioners are to be heat exchangers designed to vaporize any entering liquid and warm the propellants to 530 R. The heat source will be waste heat from the ECLSS Freon coolant.

Downstream of the heat exchangers there will be a branch in the propellant lines. One branch will go to the propellant accumulators. The other branch will have a fan, and will be used to carry conditioned propellant back into the propellant tanks. During periods of heavy use, the propellant tanks will cool, causing a drop in vapor pressure. The recirculation fans will be used to rewarm the tanks and maintain the vapor pressure.

The accumulators store the conditioned propellants until they are required by the engines, at which time the propellants flow out through pressure regulators to the engines where they are burned at 30 psia. The engines will be located on the outer radii of the forward and aft cones of both core modules on the plus and minus Z-axis. There will be a total of eight engine clusters. Each cluster of engines will contain four engines, permitting three axis translation, pitch, yaw, and roll. Each engine will produce 10 pounds of thrust. This thrust level is adequate for all the proposed RCS operations.

There will be only four sets of accumulators, two sets in each core module. This is due to the fact that the four engine clusters near the center of the two core modules are excess (they will be on-line, however, for backup use). The four high pressure nitrogen tanks located in core module 1A will be used as oxygen accumulators along with two additional 100 psia accumulators for hydrogen. Two of the high pressure nitrogen tanks in core module 1B will be used as oxygen accumulators also, along with two 100 psia hydrogen accumulators.

The estimated mass of the proposed RCS is summarized in Table 7.6-1. The loaded mass (including 455 pounds of nitrogen for the subsatellites) is 25,638 pounds. If the excess cryogenic storage capability is added, the loaded mass becomes 27,529 pounds.

Table 7.6-1. Derivative OLS RCS Mass Summary - Required Modifications

Dry mass	8,334	lbm
Cryogenics	17,304	lbm*
Total	25,638	lbm
Excess cryogenics	1,891	lbm
Total	27,529	
Subsatellite Provisions		
Dry mass	3,638	lbm
N_2H_4	7,811	lbm*
Total	11,449	lbm

*There are 455 lbm of N_2 for the subsatellite provisions included in the RCS cryogenics.

As indicated in Table 6.6-1, the RCS must provide propellants for the subsatellites. These are of two types (from a propulsion standpoint). One type has a relatively large total impulse requirement and uses hydrazine with nitrogen pressurization. The other type has a low total impulse requirement and uses cold gas (nitrogen) thrusters. These fluids would be stored on the OLS until they are required, and then transferred to the subsatellite. Normally, in an 18-month period, it will be necessary to resupply low total impulse subsatellites four times and high total impulse subsatellites twice.

A ready source of nitrogen is already onboard the OLS. The nitrogen supplied by the RCS to the ECLSS is conditioned by the ECLSS and stored in ECLSS accumulators. This stored nitrogen may be tapped and run through compressors in core module 1B for use by the subsatellites. Assuming three stage compressors, transferring 50 pounds per hour from 100 psia to a maximum of 4000 psia, the peak power required will be 3740 watts. This is very high, but it can be scheduled for low OLS power usage periods, and will present no particular difficulty. The operating time for the compressors will average about one hour per month.

The compressed gas will be piped directly to the subsatellites through the experiment module, or to two of the high pressure nitrogen tanks in core module 1B for eventual expulsion use with the hydrazine tanks located in CSM 2, the aft cryogenic storage module.

The hydrazine storage requirement of 7811 pounds includes an assumed expulsion efficiency of 95 percent with bellows tanks. The total dry mass of the subsatellite provisions is shown in Table 7.6-1 as 3638 pounds. When added to the hydrazine the mass becomes 11,449 pounds. It must be remembered, however, that 455 pounds of subsatellite nitrogen is included in the required cryogenics for the RCS. A schematic of the subsatellite provisions is shown in Figure 3.5-2.

The final aspects of the RCS and subsatellite provisions to be considered are the volume and power estimates. These values are presented in Table 7.6-2.

Table 7.6-2. Derivative OLS RCS Volume and Power Summary - Required Modifications

	Volume (ft^3)	Power (watts)	
		Peak	Average
RCS			
Tankage	1174	0	0
Conditioning	1	104	1
Feed control	100	336	1
Engines	32	1008	2
Accumulators	384	0	0
Total	1691		4
Subsatellite Provisions			
Tankage	166	0	0
Feed control	30	3970	3
Total	196		3

As indicated in the table, the major volume requirements in the RCS are for tankage and accumulators. This is due, primarily, to the low density of liquid and gaseous hydrogen. The subsatellite provisions are quite compact by comparison, due to the relatively high density of hydrazine.

The average power loads presented in Table 7.6-2 are very low due to the short duration of the expected electrical requirements. However, some of the peak loads are notable. The peak engine load assumes the simultaneous ignition of four engines and the activation of the associated eight injector valves. This event would not occur often, and the peak load would be momentary. The high power associated with the subsatellite provision feed control includes the nitrogen compressors previously mentioned. It is an infrequent event of one to two hours duration. The power builds up steadily from about 100 or 200 watts and reaches a maximum just prior to shutdown. As mentioned, this would be scheduled for low power usage periods.

7.6.2 DERIVATIVE OLS DESIGN WITH DESIRED MODIFICATIONS

If the desired OLS modifications are made in the derivative concept, there would be only four engine clusters located on the extreme forward and aft ends of the combined core modules. Each of these engine clusters would have added a set of associated 100 psia oxygen and hydrogen accumulators. The high pressure nitrogen tanks of the MSS would be deleted. There would be a pair of smaller (25-inch) high pressure nitrogen tanks adjacent to the subsatellite hydrazine tanks in the aft cryogenic storage module (CSM 2), however.

The power boom cryogenic storage area would be removed. All the cryogenic fluids would be stored in ten 99.8-cubic foot tanks, five in an evacuated section of the forward cryogenic storage module (CSM 1) and five in the aft cryogenic storage module (CSM 2) as is to be done with the required change derivative OLS of Section 7.6.1. In each storage module, there will be one oxygen, three hydrogen, and one nitrogen tank.

If the desired changes are made, the subsystem mass will be reduced by 2043 pounds to a total of 23,595 pounds (or 27,371 pounds including the excess cryogenics) as shown in Table 7.6-3. The dry mass of the subsatellite provisions is reduced by 101 pounds to a total of 3537 pounds. The resulting power and volume estimates are shown in Table 7.6-4.

Table 7.6-3. Derivative OLS RCS Mass Summary - Desired Modifications

Dry mass	6,291 lbm
Cryogenics	17,304 lbm*
Total	23,595 lbm
Excess cryogenics	3,776 lbm
Total	27,371 lbm
Subsatellite Provisions	
Dry mass	3,537 lbm
N_2H_4	7,811 lbm
Total	11,348 lbm

*There are 455 lbm of N_2 for the subsatellite provisions included in the RCS cryogenics.

Table 7.6-4. Derivative OLS RCS Volume and Power Summary - Desired Modifications

	Volume (ft^3)	Power (watts)	
		Peak	Average
RCS			
Tankage	1170	0	0
Conditioning	1	104	1
Feed control	100	336	1
Engines	16	1008	2
Accumulators	363	0	0
Total	1650		4
Subsatellite Provisions			
Tankage	149	0	0
Feed control	30	3970	3
Total	179		3

Page Intentionally Left Blank

7.7 ENVIRONMENTAL PROTECTION SUBSYSTEM

7.7.1 METEOROID PROTECTION

Those structural elements comprising the derivative OLS configuration, which are assumed to require equivalent meteoroid protection are listed in Table 7.7-1. The word "equivalent" is meant to imply that each element must have the same operational life (10 years) and the same probability of no failure due to meteoroid penetration over this period ($P_{nf} = 0.9$). Any conservatism built into the design by assuming the same 10-year life for the checked elements in Table 7.7-1 will provide an inherent margin of safety.

Table 7.7-1. Derivative OLS Elements with Equivalent Meteoroid Protection

Elements	Equivalent Protection
Power Module	X
Core Modules 1A and 1B	X
Control Center Modules 1 and 2	X
Crew Quarters Modules 1 and 2	X
Experiment Module	X
Cryo Storage Modules 1 and 2	X
Galley Module	X
Flex Ports	X
Tugs A and B	No
Dual-Support CM	No
For OLS Derivative Configuration refer to Figure 5-1	

The exposed surface area of the derivative OLS requiring equivalent meteoroid protection was calculated by idealizing each element in Table 7.7-1 as a cylinder with the overall dimensions specified in MSC 02464, "The Modular Space Station Definition Document". The total surface area is found to be 1411 m^2. The area-time product of the derivative OLS is then $AT = 4.45 \times 10^{11}$ m^2-sec.

With the meteoroid flux-mass model specified in NASA TMX-53865 (see also Section 3.6.2 of Volume IV, the Configuration and Systems Analysis Report). a summary of meteoroid strike data for the derivative OLS at 60-n mi altitude is compiled in Table 7.7-2. The design particle (m_p) for the derivative OLS is calculated directly from the meteoroid flux-mass model once AT and P_{nf} have been specified. The second column in Table 7.7-2 gives the cumulative number of meteoroid strikes upon the derivative OLS in 10 years of mass m

or greater, these values include the effects of lunar gravitational de-focusing and body shielding. The third column gives the probability of no failure that would be provided by a meteoroid protection concept designing for the particle in the first column.

Table 7.7-2. Derivative OLS Meteoroid Strike Data

Sizing Data: $A = 1411\ m^2$, $T = 10$ years, $P_{nf} = 0.9$

m (grams)	NAT (strikes $\geq$ m)	P_{nf}
10^{0}	0.000745	$\simeq .999255$
10^{-1}	0.0122	.988
10^{-2}	0.1985	.820
10^{-3}	3.24	.039
10^{-4}	53.3	6.3×10^{-24}
10^{-12}	7.3×10^{6}	$\simeq 0$

$$m_p \doteq \left[\frac{10^{-14.776} \times AT}{1 - P_{nf}} \right]^{1/1.213} = 0.0176$$

The derivative OLS meteoroid protection concept and sizing is determined in the same manner as that for the representative OLS. Use is made of Figures 3.6-29 and 3.6-30 of Section 3.6.2 of Volume IV, to decide between a single or double bumper concept and to choose bumper thickness. From Figure 3.6-30 of Volume IV, with $AT = 4.45 \times 10^{11}\ m^2$-sec, it is evident that a double bumper concept would have a significant weight advantage over the single bumper concept using glass-epoxy bumpers. The minimum bumper thicknesses are found from Figure 3.6-29 of Volume IV. This information is summarized in Table 7.7-3 below. The minimum aluminum bumper weight is approximately 73 percent of the minimum glass epoxy bumper weight. Total meteoroid protection weight includes the meteoroid bumper, glass-epoxy support structure, and aluminum shielding sandwiched to the pressure wall. It is presumed that aluminum would also be chosen for the MSS bumpers and hence used by the derivative OLS. It is likely that the MSS bumper(s) thickness and hence weight would exceed OLS requirements with the result that the derivative OLS bumper weight will exceed the required minimum values in Table 7.7-3.

Table 7.7-3. Summary of Derivative OLS Meteoroid Protection Data

($A = 1411\ m^2 = 1.52 \times 10^4\ ft^2$)

Concept	Minimum Bumper Thickness				Minimum Bumper Weight (W_B)	
	Aluminum		Glass Epoxy			
	t_1	t_2	t_1	t_2	Alum.	Glass-Epoxy
D/B	0.011"	0.00215"	0.021"	0.0042"	2870 lb	3960 lb
$\rho_{G.E.} = 0.072\ lb/in^3; W_B'' = \rho(t_1 + t_2) = 0.261\ lb/ft^2;\ W_B = W_B'' A$						
$\rho_{Al.} = 0.1\ lb/in^3; W_B'' = \rho(t_1 + t_2) = 0.1895\ lb/ft^2;\ W_B = W_B'' A$						

7.7.2 THERMAL PROTECTION

In general, the thermal protection concept for the derivative OLS is the same as for the representative OLS. An active external thermal loop, which uses freon as the fluid, operates at an effective temperature of 70 F. All available effective anti-lunar facing ($120^\circ \leq \gamma \leq 240^\circ$) area ($\approx$ 2000 square feet) is required for body-mounted radiators. A heat pump is required for humidity control, and also a thermal capacitor is required for energy storage during subsolar phases of the lunar orbit. The capacitor consists of the 16,000 pounds of water in the shield of the radiation shelter and is integrated into the internal heat transport loop. The transport loop interconnects eight modules of the derivative OLS including the battery cooling loop of the power boom. The remainder of the power boom heat load (power conditioning equipment) is rejected by the radiators on the boom.

The insulation concept consists of multiple layers of an aluminized polymeric material sandwiched between the pressure wall and the micrometeoroid bumper in non-radiator areas. Similar insulation is installed between the pressure wall and the radiators. The insulation assemblies are vented to space and isolated from pressure vessel leakage. Thermal covers are provided for windows and docking ports when they are not in use.

The power boom thermal protection subassembly of the derivative OLS consists of two fluid loops. A low temperature (50 F) loop is used to transport heat from the battery coldplates through a heat pump to the active 70 F radiators mounted on the other OLS modules. A high temperature, 150 F loop is used to reject the other heat generated within the power boom through 308 square feet of radiators mounted on the anti-lunar facing surface of the power boom.

The only significant difference between the other derivative OLS modules and the representative OLS thermal protection concept is in the placement of the radiators. Due to the shadowing of adjacent modules in the derivative configuration, effective radiators are limited to $120^{\circ} \leq \gamma \leq 240^{\circ}$ (The constraint on the representative OLS was the view angle to the lunar surface and radiators were limited to $90^{\circ} \leq \gamma \leq 270^{\circ}$). Total radiator area on the derivative OLS is 4987 square feet (of which 2000 square feet is effective).

7.7.3 RADIATION PROTECTION

The radiation protection requirements imposed upon the derivative OLS are no different than those imposed upon the representative OLS. Consequently, the discussion in Section 3.6.3, except for configurational details, applies to the derivative OLS also.

Because of the smaller diameter of the modules in the derivative configuration, the outer pressure shell forms one of the boundaries as shown on sheet 2 of Drawing 2282-29, for the storm shelter. In this area, the pressure shell must be supplemented with high density shielding. A water jacket located adjacent to the pressure wall would hamper the location and repair of meteoroid punctures. The internal partitions or walls of the storm shelter will consist of water jackets. Approximately 16,000 pounds of water, which is also used as a thermal heat sink (refer to Section 7.7.2) will be used in these jackets. In addition, half the non-frozen food (2000 pounds) together with convenient structures (cabinets, consoles, etc.) are utilized as shielding. The resulting configuration will provide a minimum of 16.6 grams per square centimeter of shielding which will limit the radiation dosage to the blood forming organs to 25 rems per year. This would allow OLS crewmen to not only survive a major solar flare event, of the magnitude specified in NASA TMX-53865, during an OLS mission, but would not require that these crewmen be grounded following the mission. Refer to Section 3.6.3 and Volume IV for further detail regarding radiation shielding criteria.

The storm shelter in the derivative OLS includes the backup control center of control center module 1. A toilet and a food reconstitution unit are added to the module within the radiation protected enclosure. Provisions are also included for installation of a microwave oven, which can be "borrowed" from the backup galley when a solar flare is imminent. (The complete backup galley was included in the representative OLS storm shelter because the layout options in a single, large diameter module permitted its incorporation with a negligible increase in weight. It was a convenience to incorporate a galley in the storm shelter, not a necessity.) The crew can either rely upon reconstituted Apollo-type food during a solar event or move the oven from the backup galley in crew quarters module 1 to the shelter area when a storm is imminent. Incorporation of a complete galley within the shelter was not considered to be a requirement and did not warrant the relocation of the backup galley.

8.0 RECOMMENDED MODULAR SPACE STATION DESIGN MODIFICATIONS

In this section recommended modifications to the design of the MSS, which will facilitate the adaptation of the MSS to OLS use without unduly penalizing the earth orbit MSS are made.

8.1 STRUCTURES

Based upon the baseline lunar orbit delivery concept established in section 5.0 the MSS core module 1 primary structure must be strengthened in the area of the side docking ports. If additional analyses in future studies of cislunar shuttles and orbital assembly operations determine that the proposed delivery concept of this study is optimum, it is recommended that the necessary changes in the primary structure of core module 1 be incorporated in the basic MSS design. Modification at a later date could be comparable in cost to a new design. The weight penalty to the MSS would be approximately 1000 pounds but the increase in docking flexibility, structural design margin, and operational safety in addition to adaptability to OLS use may adequately justify the weight increase.

8.2 ENVIRONMENTAL CONTROL/LIFE SUPPORT SUBSYSTEM

It is recommended that the regenerable charcoal concept be used in the trace contaminant control system if the technology supports the initiation of MSS development. A significant program weight reduction could be realized. In the case of the OLS, a weight reduction of the order of 12,000 pounds, over a ten-year period, can be achieved.

A dishwasher is also recommended for the MSS. Resupply and trash weight considerations make a dishwasher highly desirable for the OLS. It is believed that to a lesser degree, a total program weight reduction could also be realized with a dishwasher on the MSS.

The current MSS design is sized for processing rates for a six-man crew. If the equipment were resized for an eight-man crew, conversion to OLS use would be greatly enhanced. The resized MSS equipment could be operated at either a reduced rate or only 18 hours per day.

8.3 ELECTRICAL POWER SUBSYSTEM

Substitution of Regenerative Fuel Cells (RFC) for batteries and battery chargers on the MSS are recommended. Section 3.2 detailed the comparison of the two energy storage concepts. The most significant factor in favor of RFC was the large decrease in total program weight that can be realized. The primary constraint on incorporation of RFC in the MSS is the technology status at initiation of the MSS development.

Currently a double ac voltage (240/416 vac) distribution concept is incorporated in the MSS. Potentially long feeder line distances were the driving factor in the selection of this concept. The preferred MSS and OLS configurations require shorter distribution lines. Use of a single, standard voltage system (120/208 vac) will reduce susceptibility to corona effects and switching transients and may result in an overall weight reduction by simplifying the inverter-regulator design. It is recommended that the MSS incorporate a single ac standard voltage distribution concept.

8.4 INFORMATION SUBSYSTEM

It is recommended that a third parabolic antenna be mounted to the power boom. Although continuous communication capability is not a requirement on the MSS, this added antenna would eliminate the signal blockage caused by the shadow effect of the solar arrays. This concept would enhance MSS operation and facilitate conversion to OLS use which requires the third antenna.

8.5 GUIDANCE AND CONTROL SUBSYSTEM

The only additional guidance and control equipment required for the derivative OLS over that required for the MSS is the landmark tracker and the radar altimeter. This equipment is required to satisfy the tighter OLS navigation accuracy requirements. It is recommended that this equipment be incorporated into the MSS for qualification of hardware and (partially) software. Even more important is that inclusion of this equipment would serve to improve the navigation accuracy of the MSS, allow for greater accuracy of presently planned experiments, and provide flexibility and increase performance capability which could be utilized by future experiments.

8.6 REACTION CONTROL SUBSYSTEM

There are no recommended changes to the Reaction Control Subsystem of the MSS. The modified use of MSS modules results in redundant locations of engines but removal of redundant engines is not warranted. Propellant storage requirements for the OLS are so much larger than for the MSS it would be an unrealistic penalty on the MSS to incorporate storage facilities adequate for OLS use.

8.7 ENVIRONMENTAL PROTECTION SUBSYSTEM

8.7.1 Meteoroid Protection

Because a detailed analysis and design of the MSS meteoroid protection concept has not been documented to date, an evaluation of the concept for possible modification recommendations cannot be made. However, it is anticipated that no modifications of the MSS meteoroid protection concept would be suggested because both the derivative OLS and MSS are in the same range of area-time products where the double bumper concept is desirable.

8.7.2 Thermal Protection

The MSS has no requirement for a thermal capacitor or a heat pump. However, these additions would reduce the required radiator area and decrease the sensitivity to α_s / ϵ degradation. These changes are recommended to enhance the MSS operation, provide an operational evaluation of the OLS concept, and facilitate adaptation to OLS use.

In Section 6.7.2 it was indicated that the only useful radiator area for heat rejection on the OLS modules was between $120° \leq Y \leq 240°$. It is recommended that the radiators on the MSS modules be segmented such that the radiator between 120 and 240 degrees can be isolated from the remainder of the circumferential radiator system.

The radiators on the power boom are sized for batteries and battery chargers. In Section 8.3 it was recommended that this energy storage concept be replaced with regenerative fuel cells. Consequently the power boom radiators would also have to be resized. The requirements for boom radiators were derived in Section 3.7.2 and are 425 square feet total with at least 280 square feet on the antilunar side of the boom. The present MSS concept has only 300 square feet of radiator area.

8.7.3 Radiation Protection

There are no recommended MSS modifications to facilitate the adaptation to OLS use. Incorporation of the storm shelter provisions that are a unique requirement of the OLS would unduly penalize the MSS and are not warranted.

8.7.2 Thermal Protection

The MSS has no requirement for a thermal capacitor or a heat pump. However, these additions would reduce the required radiator area and decrease the sensitivity to α_s / ϵ degradation. These changes are recommended to enhance the MSS operation, provide an operational evaluation of the OLS concept, and facilitate adaptation to OLS use.

In Section 6.7.2 it was indicated that the only useful radiator area for heat rejection on the OLS modules was between $120° \leq Y \leq 240°$. It is recommended that the radiators on the MSS modules be segmented such that the radiator between 120 and 240 degrees can be isolated from the remainder of the circumferential radiator system.

The radiators on the power boom are sized for batteries and battery chargers. In Section 8.3 it was recommended that this energy storage concept be replaced with regenerative fuel cells. Consequently the power boom radiators would also have to be resized. The requirements for boom radiators were derived in Section 3.7.2 and are 425 square feet total with at least 280 square feet on the antilunar side of the boom. The present MSS concept has only 300 square feet of radiator area.

8.7.3 Radiation Protection

There are no recommended MSS modifications to facilitate the adaptation to OLS use. Incorporation of the storm shelter provisions that are a unique requirement of the OLS would unduly penalize the MSS and are not warranted.