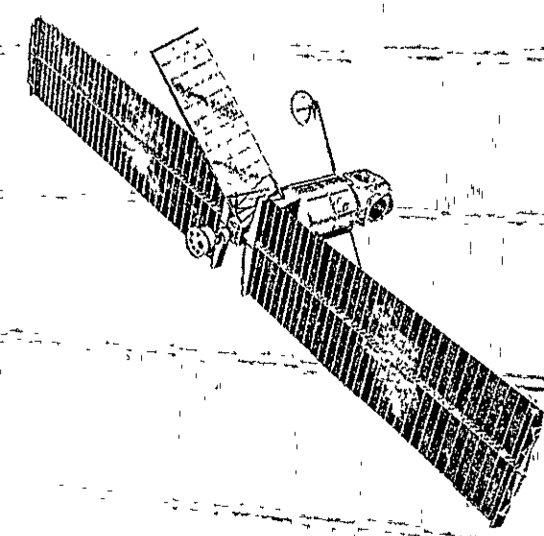
LMSC D614944 –4



(NASA-CR-161148) THE 25 kW POWER MODULE EVOLUTION STUDY. PART 3: CONCEPTUAL DESIGNS FOR POWER MODULE EVOLUTION. VOLUME 4: DESIGN ANALYSES Final Report (Lockheed Missiles and Space Co.) 347 p HC A15/MF A01 G3/15

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NASA
George C. Marshall
Space Flight Center

25 kW POWER MODULE EVOLUTION STUDY

VOLUME 4 DESIGN ANALYSIS

PART III : CONCEPTUAL DESIGNS FOR POWER MODULE EVOLUTION • FINAL REPORT

LOCKHEED MISSILES & SPACE COMPANY, INC.

LMSC-D614944-4

FINAL REPORT

25 kW POWER MODULE EVOLUTION STUDY

PART III: CONCEPTUAL DESIGNS FOR POWER MODULE EVOLUTION

VOLUME 4: DESIGN ANALYSES

27 January, 1979

Submitted to the

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GEORGE C. MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA 35812

CONTRACT NAS8-32928
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DR NO. MA-04

LOCKHEED MISSILES & SPACE COMPANY
SUNNYVALE, CALIFORNIA

FOREWORD

This volume of the Part III Final Report for the 25 kW Power Module Evolution Study was prepared by Lockheed Missiles and Space Company, Inc. for the George C. Marshall Space Flight Center (MSFC), National Aeronautics and Space Administration (NASA), under Contract No. NAS8-32928.

The objective of the study was to define how the 25 kW Power Module can be evolved by the addition of system elements in evolutionary steps to meet the future mission requirements. For each step, conceptual designs were prepared. The level of capability at each step was commensurate with the mission and payload requirements. Emphasis was placed on the near-term steps beyond the 25 kW Power Module.

The study activity comprised the following parts/tasks:

- Part I – Payload Requirements and Growth Scenarios (LMSC, TRW, and Bendix)

This analytical effort was conducted to develop payload application summaries and time-phased requirements that will drive the concepts for the 25 kW Power Module and the supporting systems definitions (for the period 1983-1990). The Part I effort was documented in Final Report LMSC-D614921A, dated 1 August 1978.

- Part II – Payload Support System Evolution (LMSC, IBM, and Bendix)

This effort was devolted to establishing baseline program support elements and candidate evolutionary growth capabilities for final candidate definition (element data, cost, modifications, development sequence, and precursor missions). The Part II effort was documented in Final Report LMSC-D614928A, dated 30 September 1978.

- Part III – Conceptual Designs for Power Module Evolution (LMSC and Bendix)

This effort was conducted to establish design approaches for the evolutionary systems, to develop associated programmatics data, and to assess the evolution scenario and capabilities of the 25 kW Power Module for representative missions.

This report constitutes Volume 4, Design Analyses, of the Part III Final Report. It meets the requirements of Contract No. NAS8-32928 Data Procurement Document, Data Requirement MA-04, Final Study Report.

The volumes comprising the Part III Final Report are:

- Volume 1 – Power Module Evolution
- Volume 2 – Program Plans
- Volume 3 – Cost Estimates
- Volume 4 – Design Analyses
- Volume 5 – Mission Accommodations
- Volume 6 – WBS and Dictionary

ACKNOWLEDGEMENTS

This report is the product of the efforts of an integrated team of specialists representing various technical disciplines. Their contributions are hereby gratefully acknowledged.

Structures	R. W. Goldin, E. J. Loss, S. R. Nichols and A. W. Steele
Power Subsystem	M. G. Gandel and R. W. Given
Thermal Control Subsystem	A. L. Lee, R. Horn and P. E. Schroeder
Attitude Control Subystem	R. A. Barsocchi (Bendix)
C&DH Subsystem	L. R. Phegley
Orbit Reboost Analysis	J. Reitman and C. J. Rudey
Mass Properties	G. J. Strom
Power Module Operations	H. Burkleo and E. W. Waller

This design analyses documentation was integrated by R. W. Goldin, with guidance and direction from J. W. Overall and R. J. Watson.

CONTENTS

SECTION 1
INTRODUCTION

DESIGN ANALYSES REPORT: SCOPE AND OBJECTIVE

- All supporting design analyses completed in Part III, or not reported in Part I and Part II reports, are summarized in this volume.

- Design layouts completed during Part III are included where appropriate in Volume 1 and this volume. Appendix A lists all of the layouts, and provides the volume and page number location.

DESIGN ANALYSES REPORT: SCOPE & OBJECTIVE

PURPOSE & SCOPE:

SUMMARIZE THE DESIGN AND ANALYTICAL STUDIES PERFORMED IN PART III, INCLUDING CHANGES TO PART II STUDY CONCLUSIONS, GROUPED AS FOLLOWS:

- SYSTEM ANALYSES
- SUBSYSTEM ANALYSES
- SYSTEM SUPPORT ELEMENTS
- POWER MODULE SUPPORT EQUIPMENT
- OPERATIONS
- TECHNOLOGY PLANNING

OBJECTIVE OF THE DESIGN/ANALYSES:

PROVIDE THE TECHNICAL BASIS FOR THE EVOLUTIONARY SYSTEM RECOMMENDATIONS

PART III DESIGN ANALYSIS BACKGROUND

- The sources of Part III Study input data are listed on the chart, with the completion date shown for each input package.

- A large volume of output from related Space Transportation System (STS) studies was reviewed, both prior to and during Part III. Most of these have been identified and listed in the Part I and Part II reports.

- Specific references utilized during Part III design/analysis efforts are listed in the Bibliography, Section 8.1. Engineering Memoranda summarizing results of specific design/analysis activities are listed in Section 8.2.

PART III DESIGN ANALYSIS BACKGROUND

INPUT INFORMATION	DATE
• MSFC 25 kW POWER MODULE BASELINE CONCEPT	9/77
• MSFC POWER MODULE SYSTEM DESIGN REQUIREMENTS	3, 5/78
• DESIGN REQUIREMENTS GENERATED IN PART I	7/78
• ITERATIVE INTERACTION WITH MISSION ACCOMMODATIONS ANALYSIS	9/78
• INITIAL TRADE STUDIES AND DESIGN CONCEPTS FROM PART II	9/78
• PART II OUTPUT CRITIQUES/COMMENTS FROM MSFC ENGINEERS	9-10/78
• OUTPUTS FROM RELATED STS STUDIES	1-11/78

SECTION 2
SYSTEM ANALYSES

2.1 GROWTH OPTIONS EVALUATION

Part II of the study (Ref. Pg. 2D-1 Report LMSC-D614928A) provided optional growth configurations for selection (in Part III) of a single evolutionary system and further definition of the conceptual designs of each evolutionary stage. Although detailed quantitative assessments were not possible within the time and budget constraints of the study, cost/benefit evaluations were the basis for the selections of the recommended evolutionary paths. These included considerations of reliability, achievement of the operational life goals for the system, and minimization of operational and maintenance complexity/cost. This subsection summarizes the basic considerations employed in the growth evaluation, and delineates the recommended evolutionary system together with the rationale for its selectior

BASIC GROWTH CONSIDERATIONS

- The chart lists key considerations addressed in making the selection of the recommended evolutionary growth system. The selection was focused explicitly on meeting the Power Module requirements for the Program Scenario I.

- Whereas Part II defined concepts for evolutionary growth to a 250 kW Power Module, the Program Scenario I requires growth only to 50 and 100 kW configurations. Growth to the lower level (100 kW vs 250 kW) enables greater commonality between subsystems of the Power Module. It is also possible to select Power Module configurations which have essentially the same over all arrangement, and which employ very similar operational procedures. These commonalities between the three sizes of Power Modules required by Program Scenario I favorably affect each of the growth considerations identified on the chart for all three Power Modules.

BASIC GROWTH CONSIDERATIONS

MODULARITY

- PRODUCTION ECONOMY
- ON-ORBIT ASSEMBLY SIMPLICITY

RIGIDITY

- COMPATIBLE WITH ATTITUDE CONTROL SYSTEM
- AFFECTED BY ORBITER-BAY PACKAGING DESIGN AND ON-ORBIT ASSEMBLY PROVISION

MECHANICAL SIMPLICITY

- COST AND MAINTENANCE AVOIDANCE
- RELIABILITY ENHANCEMENT

EVA OPERATIONS

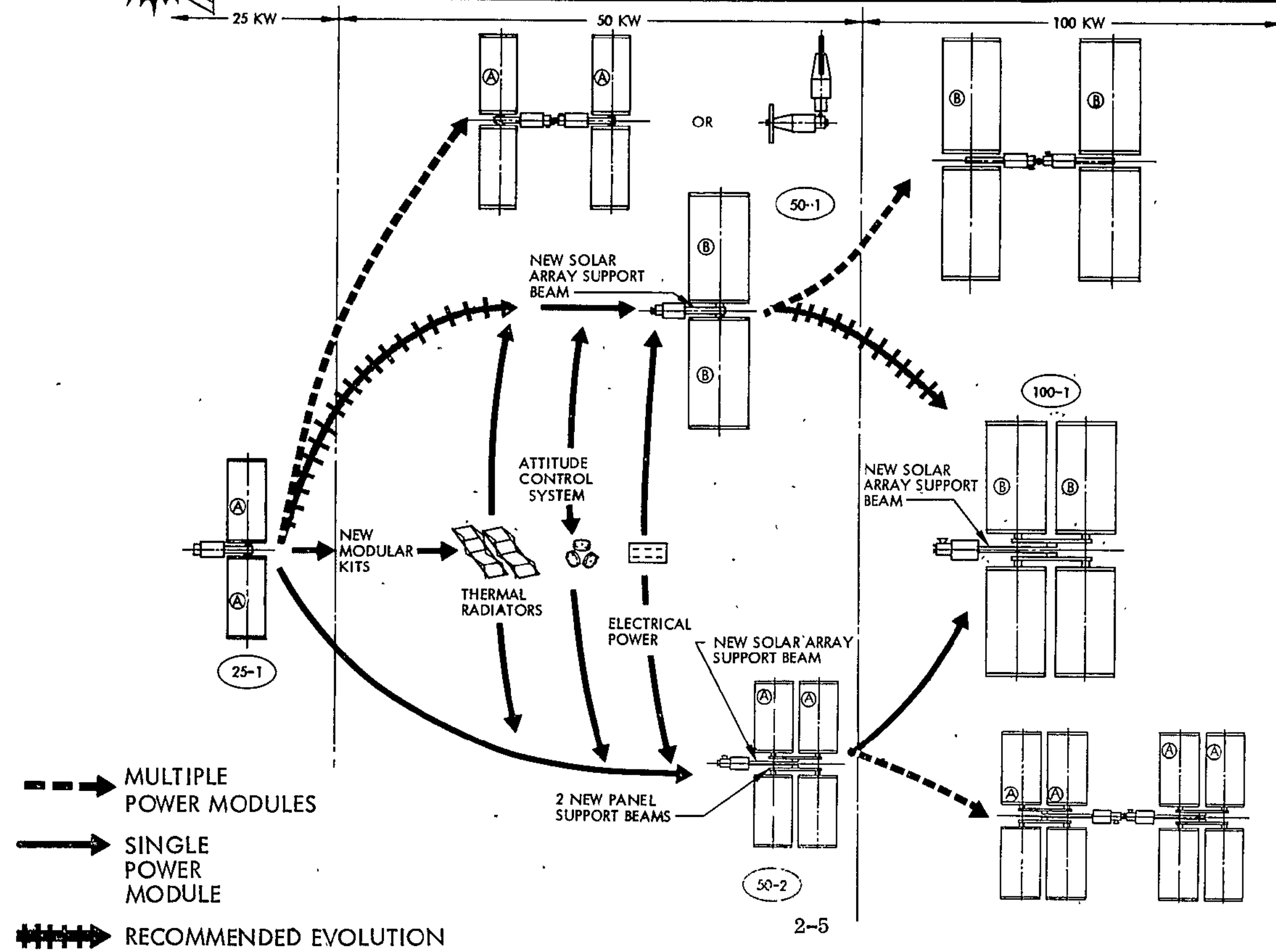
- ON-ORBIT ASSEMBLY GROWTH KIT CONCEPTS
- DESIGN FOR EASE OF MAINTENANCE

POWER MODULE GROWTH CONCEPTS
25 kW TO 100 kW

- Viable candidate configurations for Power Module evolutionary growth, from 25 kW to 100 kW, are illustrated on the chart. The single-vehicle growth paths are identified by the solid line arrows, while growth simply by use of two Power Modules is shown with the dashed line arrows. The configuration growth path recommended to satisfy Program Scenario I is identified by the crosshatched solid line arrows.

- The recommended evolution utilizes two sizes of solar array blankets, the "A" type at 13.2 x 130 feet and the "B" type at 19.8 x 130 feet. The "B" type, which is required in 1986 or later, utilizes technology which provides twice the power-generation output with only 1.5 times the area. The 50-2 configuration, using existing technology with the smaller blanket, will meet initial 50 kW requirements, if facilities for fabrication of the "B" type blanket are not yet available. However, the use of more mechanical elements and more difficult stowage in the Orbiter make this option technically less desirable. The blankets are used in either two or four pairs, with a single deployment mast per pair.

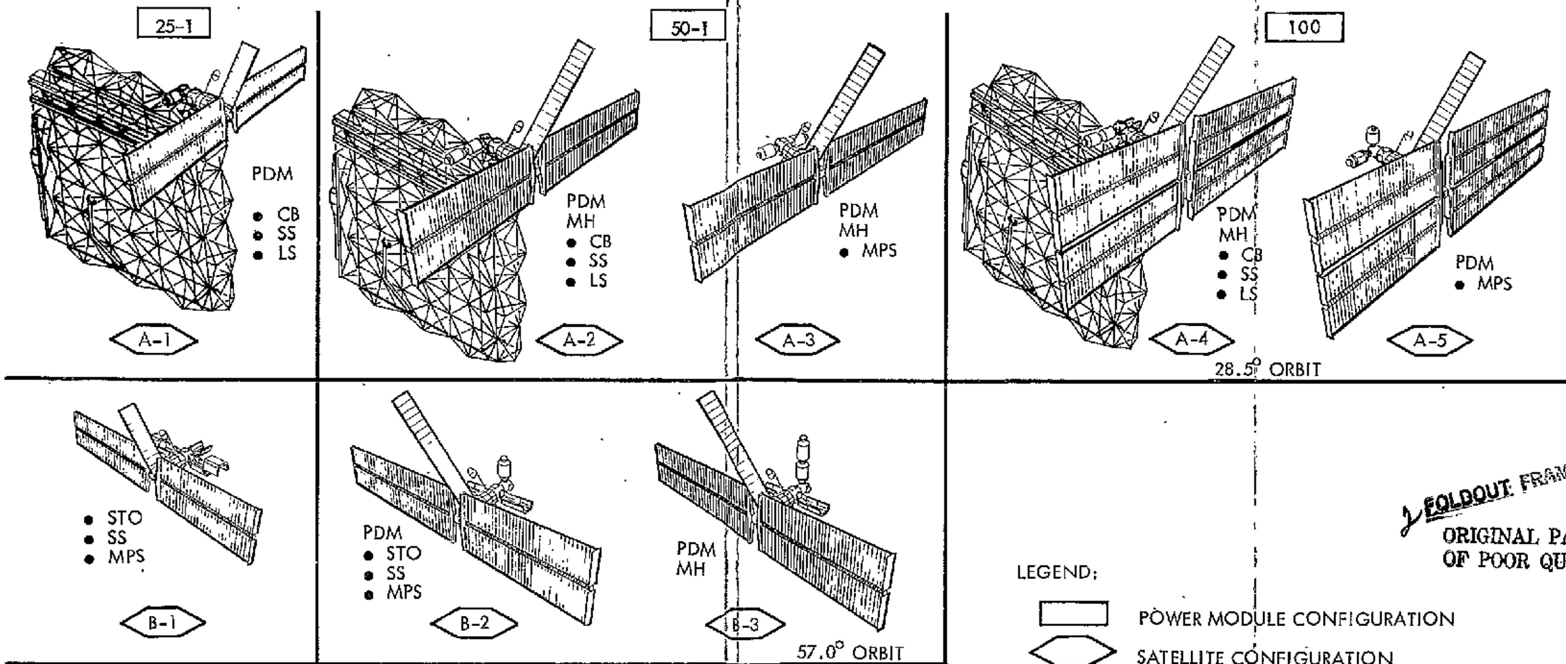
POWER MODULE GROWTH CONCEPTS 25kW TO 100kW

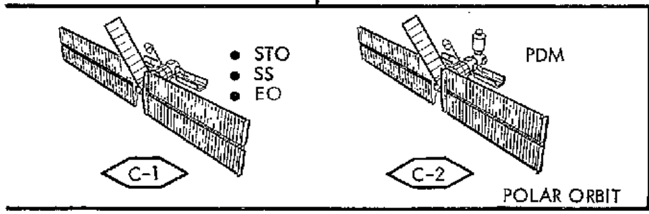


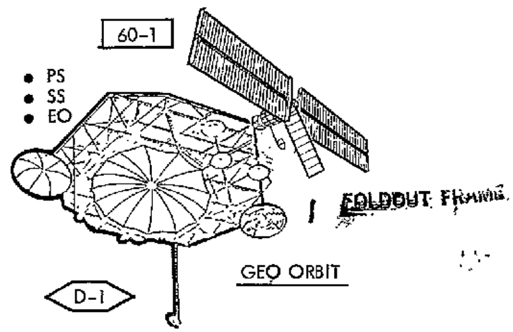
DERIVED SYSTEM CONFIGURATIONS

- The chart illustrates, at equivalent scale, each of the Power Module/ payload satellite configurations analyzed in Part III of the study. Both Power Module and satellite designations are identified.

- The configurations are grouped by orbit: 28.5°, 57.0°, polar and geosynchronous. Payload disciplines carried by each satellite configuration are also identified.







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LEGEND:

▭ POWER MODULE CONFIGURATION

⬡ SATELLITE CONFIGURATION

- PDM – PRESSURIZED DOCKING MODULE
- MH – MANNED HABITAT
- ● MPS – MATERIAL PROCESSING
- ● STO – SOLAR TERRESTRIAL OBSER
- ● PS – PUBLIC SERVICE
- ● SS – SPACE SCIENCE
- ● EO – EARTH OBSERVATION
- ● CB – CONSTRUCTION BASE FOR:
  - ● SPS – SPACE POWER SYSTEM
  - ● GEO PLATFORM
  - ● LEO PLATFORM
  - ● OTHERS
- ● SCD – SPACE CONSTRUCTION DEMO
- ● LS – LIFE SCIENCE

NOT RELEASED

A CLASSIFICATION CHGS 11-9-78 RWS

10-27-78

PICTORIAL SCENARIO 1 POWER MODULE

C164070 A

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CONFIGURATION DESIGNATION MATRIX

- The chart provides a cross-index of configuration designations (for Scenario I) used in the design analysis efforts. The "Flight Vehicle" identifiers are in effect Power Module serial numbers, in the order they are fabricated.
- The "R1" and "R2" identify the first and second refurbishments of FV-1.
- The "4K" identifies a kit modification of FV-4.

CONFIGURATION DESIGNATION MATRIX

FLIGHT VEHICLE	CALENDAR YEAR(S) IN ORBIT	PM CONFIG-URATION	SCENARIO I SATELLITE
FV-1	83-85	25-1	B-1
FV-1R1	86-87	25-1	A-1
FV-1R2	88→	25-1	C-1 & C-2
FV-2	86→	50-1	B-2 & B-3
FV-3	87→	60-1*	D-1
FV-4	88-90	50-1	A-2
FV-5	89→	50-1	A-3
FV-4K	90→	100-1	A-4
FV-6	91→	100-1	A-5

*GEO COMPONENTS, WITH SOLAR ARRAYS LIKE THOSE ON 25-1. THIS WAS NOT SHOWN ON PREVIOUS "GROWTH OPTION" CHARTS

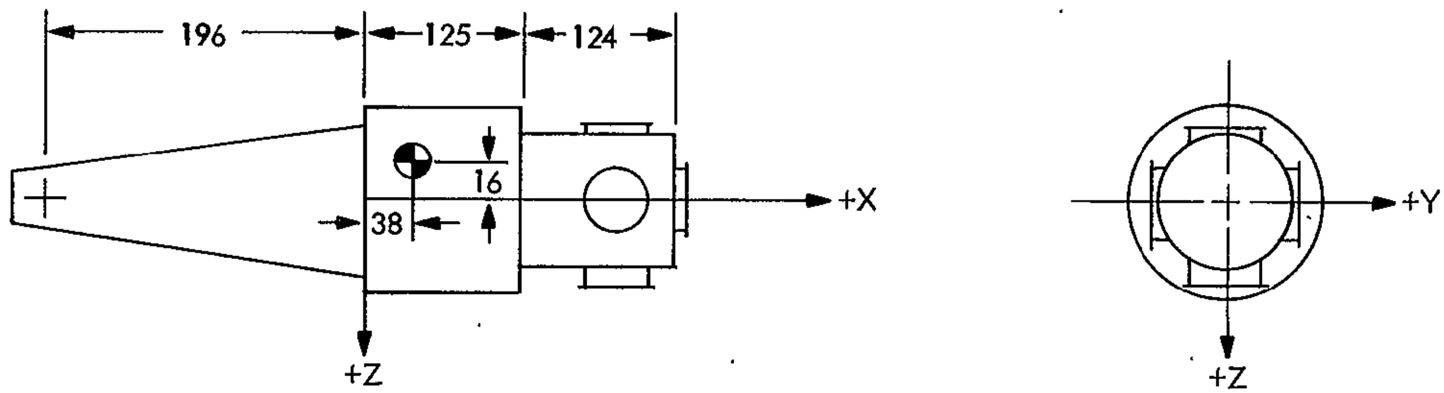
2.2 MASS PROPERTIES

This section contains the basic and growth PM mass properties with and without payloads as well as CG information for the power module.

25 kW POWER MODULE CENTER OF GRAVITY

- This chart shows the weights and centers of gravity of each of the major subassemblies of the 25 kW power module.

25 kW POWER MODULE CENTER OF GRAVITY



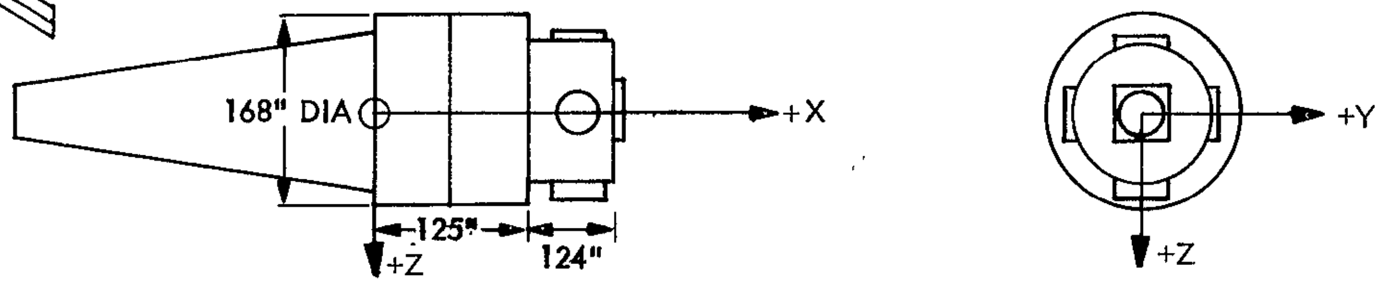
MODULE	WEIGHT LB	CENTER OF GRAVITY		
		X	Y	Z
S/A, DRIVES, DEPLOYMENT	2,900	-196	0	0
RADIATOR, DEPLOYMENT	1,185	-128	0	-330
S/A SUPPORT MODULE	1,625	-78	0	0
EQUIPMENT MODULE	14,082	+62	0	0
BERTHING MODULE	4,783	+190	0	0
SUBTOTAL	24,575	+38	0	-16
CONTINGENCY - 25%	6,144	+38	0	-16
TOTAL	30,719	+38	0	-16

POWER MODULE MASS PROPERTIES

- For use in Power-Module-alone analyses, mass properties of the three basic sizes of Power Modules are provided on the chart. Properties shown are for the on-orbit, fully deployed configuration.

- On-orbit configuration mass properties with typical payloads, and with the Orbiter, are provided in the two following charts.

POWER MODULE MASS PROPERTIES



		25-1	50-1	100-1
WEIGHT – LB		30,719	35,103	42,014
CG (INCHES)	X	38	5	-53
	Y	0	0	0
	Z	-16	-37	-31
MOMENTS OF INERTIA (SLUG-FT^2/10^6)	Ix	0.66	2.06	3.53
	Iy	0.13	0.35	0.72
	Iz	0.72	2.07	3.64

MASS PROPERTIES FOR SCENARIO 1 CONFIGURATIONS: WITH PAYLOADS

- Mass properties for the combined power module (with arrays deployed) and attached payloads are shown for eleven orbital configurations. A common coordinate system for all configurations is defined by the sketch.

- The construction bases were assumed to weigh 25,000 lb, the GEO platform 20,000 lb, and all other small payload modules (as shown on page 2-7) 10,000 lb each.

- The moments of inertia about the principal axes are also shown. These are the axes about which the moments of inertia reach minimum and maximum values.

NOTE: No attempt has been made to optimize the distribution of weights to achieve perfect symmetry.

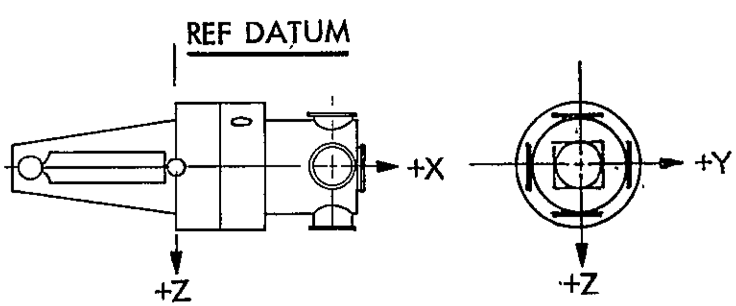
MASS PROPERTIES FOR SCENARIO I CONFIGURATIONS: WITH PAYLOADS

CONFIG*	WEIGHT LB	CENTER OF GRAVITY – IN X	Y	Z	MOMENTS OF INERTIA – SLUG FT^2/10^6 I_x	I_y	I_z
A-1	64,345	841	19	808	22.3	41.1	20.4
A-2	85,763	646	40	596	26.7	46.1	24.4
A-3	60,763	76	-17	-19	2.44	0.52	2.57
A-4	102,339	535	22	502	29.3	49.0	27.7
A-5	67,339	28	11	-35	3.98	1.05	4.29
B-1	59,345	149	-4	-5	0.74	0.31	0.96
B-2	70,763	95	-15	-37	2.48	0.60	2.61
B-3	80,763	110	-13	-70	2.61	0.76	2.64
C-1	59,345	149	-4	-5	0.74	0.31	0.96
C-2	70,763	95	-15	-37	2.48	0.60	2.61
D-1	49,345	923	-5	-6	4.92	16.59	21.22

*REF PAGE 2-7

PRINCIPAL MOMENTS OF INERTIA – SLUG FT^2/10^6

CONFIG*	I_x	I_y	I_z
A-1	34.9	41.1	7.8
A-2	41.4	46.1	9.7
A-3	2.43	0.51	2.59
A-4	45.5	49.0	11.5
A-5	3.98	1.04	4.30
B-1	0.74	0.31	0.96
B-2	2.48	0.59	2.62
B-3	2.57	0.75	2.69
C-1	0.74	0.31	0.96
C-2	2.48	0.59	2.62
D-1	4.92	16.59	21.22



MASS PROPERTIES FOR SCENARIO I CONFIGURATIONS: WITH PAYLOADS & ORBITER

- This chart provides similar mass-property data to that provided on the preceding chart for the eleven satellites with the Orbiter attached in a sortie mode. Orbiter mass properties were taken from Ref. 30

- A detailed study of all sortie configuration arrangements has not been made. For the mass property calculations of the space platform configurations it has been assumed that the Orbiter is berthed to the Power Module at a berthing port 180° from that to which the space platform is berthed. To accomplish this, a berthing adaptor/extension may be needed, or the PM may have to be berthed 180° reversed from what is shown on page 2-7 (with radiator close to the station, rather than positioned away from it).

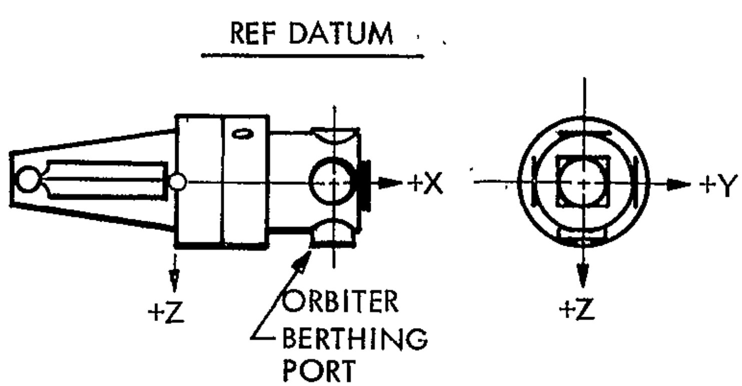
MASS PROPERTIES FOR SCENARIO I CONFIGURATIONS: WITH PAYLOADS AND ORBITER

CONFIG*	WEIGHT LB	CENTER OF GRAVITY – IN			MOMENTS OF INERTIA – SLUG $FT^2/10^6$		
		X	Y	Z	I_x	I_y	I_z
A-1	261,800	764	4	-120	39.0	63.6	27.3
A-2	283,300	711	12	-114	41.0	66.1	31.3
A-3	258,300	583	-3	319	5.3	13.4	13.8
A-4	299,800	669	7	-107	42.6	68.6	35.1
A-5	264,800	558	3	307	7.1	15.3	16.6
B-1	256,800	603	0	324	3.4	12.0	11.2
B-2	268,300	569	-3	302	5.7	14.1	14.1
B-3	278,300	556	-3	280	6.5	15.2	14.3
C-1	256,800	603	0	324	3.4	12.0	11.2
C-2	268,300	569	-3	302	5.7	14.1	14.1
D-1	246,800	776	0	337	7.4	24.9	28.3

PRINCIPAL MOMENTS OF INERTIA – SLUG $FT^2/10^6$

CONFIG*	$I_{x'}$	$I_{y'}$	$I_{z'}$
A-1	48.9	71.1	9.9
A-2	51.5	76.0	10.9
A-3	4.5	13.4	14.5
A-4	53.7	79.6	13.0
A-5	6.1	15.3	17.6
B-1	2.8	12.2	11.6
B-2	4.7	14.1	15.1
B-3	5.2	15.1	15.7
C-1	2.8	12.2	11.6
C-2	4.7	14.1	15.1
D-1	7.3	24.9	28.4

*REF PAGE 2-7



STRUCTURE & MECHANICAL SUBSYSTEM GROWTH WEIGHTS

- The solar array support structure includes provisions for mounting magnetic torquers. For the 100-1' configuration, this structure also includes provisions for mounting the second set of batteries. The weight of the 100-1' structure is less than the weight of the 100-1 structure because its ascent-loads are less, since it is delivered to orbit in two launches.

- The two equipment racks are based on the current design used by Space Telescope.

- The weight for mechanisms includes motors, latches, and connecting structure for primary deployment of the radiator and solar arrays.

- Mechanisms to rotate the arrays on orbit are included in the electrical power subsystem.

STRUCTURE & MECHANICAL SUBSYSTEM GROWTH WEIGHTS

ITEM	CONFIGURATION				
	25-1	50-1	100-1	100-1' *	200-1
WEIGHT (LB)					
S/A SUPPORT STRUCTURE	880	1,250	2,850	2,500	4,000
FWD EQUIP RACK	1,435	1,435	1,435	1,435	1,435
AFT EQUIP RACK	1,465	1,465	1,465	1,465	1,465
BERTHING MODULE	3,200	3,200	3,200	3,200	3,200
MECHANISMS	450	800	1,100	1,300	2,300
SUBTOTAL	7,430	8,150	10,050	9,900	12,400
CONTINGENCY - 25%	1,858	2,038	2,513	2,475	3,100
TOTAL	9,288	10,188	12,563	12,375	15,500

*100-1' = 50 kW CONFIGURATION CONVERTED TO 100 kW CONFIGURATION USING THE 50-100 kW KIT

ELECTRICAL POWER SUBSYSTEM GROWTH WEIGHTS

- The weights of the power subsystem components reflect the "current" technology available at the time of launch.

- The 100-1' configuration has a double set of batteries and duplication of most of the other components since the old components will not be removed when the on-orbit assembly is accomplished.

- Conversion from 50-1 to 100-1 configuration was considered to be more cost-effective (with considerably less EVA) than designing for on-orbit substitution of new for old technology batteries. The other solar array add-on kit items would still require the same amount of EVA, which in the present kit design also effects the battery change.

- The 100-1' configuration solar array weight consists of 4800 lb from the initial 50-1 configuration launch plus 3600 lb (representing newer technology) added by the conversion 50 kW to 100 kW kit.

ELECTRICAL POWER SUBSYSTEM GROWTH WEIGHTS

ITEM	CONFIGURATION				
	25-1	50-1	100-1	100-1'*	200-1
WEIGHT (LB)					
SOLAR ARRAY	2,400	4,800	7,200	8,400	14,400
SOLAR ARRAY DRIVES	200	250	300	550	300
BATTERIES	7,440	6,400	6,400	12,800	12,800
ELECTRONICS	1,395	1,395	2,055	3,450	4,110
PWR. DISTRIBUTION	630	880	1,145	2,025	2,290
WIRE HARNESS	500	500	500	1,000	1,000
SUBTOTAL	12,565	14,225	17,600	28,225	34,900
CONTINGENCY – 25%	3,141	3,556	4,400	7,056	8,725
TOTAL	15,706	17,781	22,000	35,281	43,605

*100-1' = 50KW CONFIGURATION CONVERTED TO 100KW CONFIGURATION USING THE 50-100KW KIT.

THERMAL CONTROL SUBSYSTEM GROWTH WEIGHTS

- The weights of the thermal control subsystem components reflect the "current" technology available at the time of launch.

- In the case of the 100-1' configuration, duplication of the twelve battery cold plates and an oversized payload heat exchanger is shown. The old components will not be removed when the on-orbit modification to 100 kW is made.

THERMAL CONTROL SUBSYSTEM GROWTH WEIGHTS

ITEM	CONFIGURATION				
	25-1	50-1	100-1	100 - 1'*	200-1
WEIGHT (LB)					
RADIATOR	1,035	2,020	1,900	2,170	2,600
COLD PLATES, LINES	565	565	565	825	1,130
HEAT EXCHANGERS AND CONTROLS	137	257	137	257	257
PUMPS, CONTROLS	177	177	177	177	354
MLI**, PAINT, MISC.	100	100	100	120	200
SUBTOTAL	2,014	3,119	2,879	3,549	4,541
CONTINGENCY – 25%	504	780	720	887	1,135
TOTAL	2,518	3,899	3,599	4,436	5,676

*100 -1' = 50KW CONFIGURATION CONVERTED TO 100KW CONFIGURATION USING THE 50 - 100KW KIT.
**MULTI-LAYER INSULATION.

ATTITUDE CONTROL SUBSYSTEM GROWTH WEIGHTS

- The 50-1 configuration shown includes 3 CMGs, which is sufficient for the 50 kW module that is not modified to 100 kW.

- If a 50 kW module is to be upgraded to the 100-1' configuration, a fourth CMG (minimum) must be included in the initial launch.

ATTITUDE CONTROL SUBSYSTEM GROWTH WEIGHTS

ITEM	CONFIGURATION				
	25-1	50-1	100-1	100-1'*	200-1
WEIGHT (LB)					
CMG's & INVERTERS	1,416	1,416	1,888	1,888	2,832
RATE GYROS	104	104	104	104	104
SIG. COND/IF UNITS	90	90	90	90	90
HORIZON SENSORS/ELECTR.	54	54	54	54	54
MAG. TORQUERS/ELECTR.	456	456	456	456	912
MISC.	18	18	18	18	18
SUBTOTAL	2,138	2,138	2,610	2,610	4,010
CONTINGENCY - 25%	535	535	653	653	1,003
TOTAL	2,673	2,673	3,263	3,263	5,013

*100-1' = 50 KW CONFIGURATION CONVERTED TO 100KW CONFIGURATION USING THE 50-100KW KIT.

C & DH SUBSYSTEM GROWTH WEIGHTS

- The only changes required in the C & DH subsystem as power level increases are more remote units, switches, and cabling.
- The antenna/drives weights shown include mechanisms and latches for deployment.

C&DH SUBSYSTEM GROWTH WEIGHTS

ITEM	CONFIGURATION				
	25-1	50-1	100-1	100-1'*	200-1
WEIGHT (LB)					
TRANSPONDERS	31	31	31	31	31
COMPUTERS (NSSC-II)	130	130	130	130	130
CENTRAL & REMOTE UNITS	80	100	120	140	140
ANTENNAS/DRIVES	116	116	116	116	116
STEERING ELECTRONICS	48	48	48	48	48
SWITCHES & CABLING	23	25	27	29	29
SUBTOTAL	428	450	472	494	494
CONTINGENCY - 25%	107	113	118	124	124
TOTAL	535	563	590	618	618

*100-1' = 50KW CONFIGURATION CONVERTED TO 100KW CONFIGURATION USING THE 50-100KW KIT.

TOTAL POWER MODULE GROWTH WEIGHTS

- The subsystem weights shown are taken from the subtotal (without contingency) lines on the subsystem summary weight charts.

- The weights for the 100-1' configuration represent the total assembled weight of the original 50 kW vehicle plus the 50 kW to 100 kW modification kit.

TOTAL POWER MODULE GROWTH WEIGHTS

ITEM	CONFIGURATION				
	25-1	50-1	100-1	100-1'*	200-1
WEIGHT (LB)					
STRUCTURE & MECHANICAL	7,430	8,150	10,050	9,900	12,400
ELECTRICAL POWER	12,565	14,225	17,600	28,225	34,900
THERMAL CONTROL	2,014	3,119	2,879	3,549	4,541
ATTITUDE CONTROL	2,138	2,138	2,610	2,610	4,010
C&DH	428	450	472	494	494
SUBTOTAL	24,575	28,082	33,611	44,778	56,345
CONTINGENCY – 25%	6,144	7,021	8,403	11,195	14,086
TOTAL	30,719	35,103	42,014	55,973	70,431

*100-1' = 50KW CONFIGURATION CONVERTED TO 100KW CONFIGURATION USING THE 50-100KW KIT.

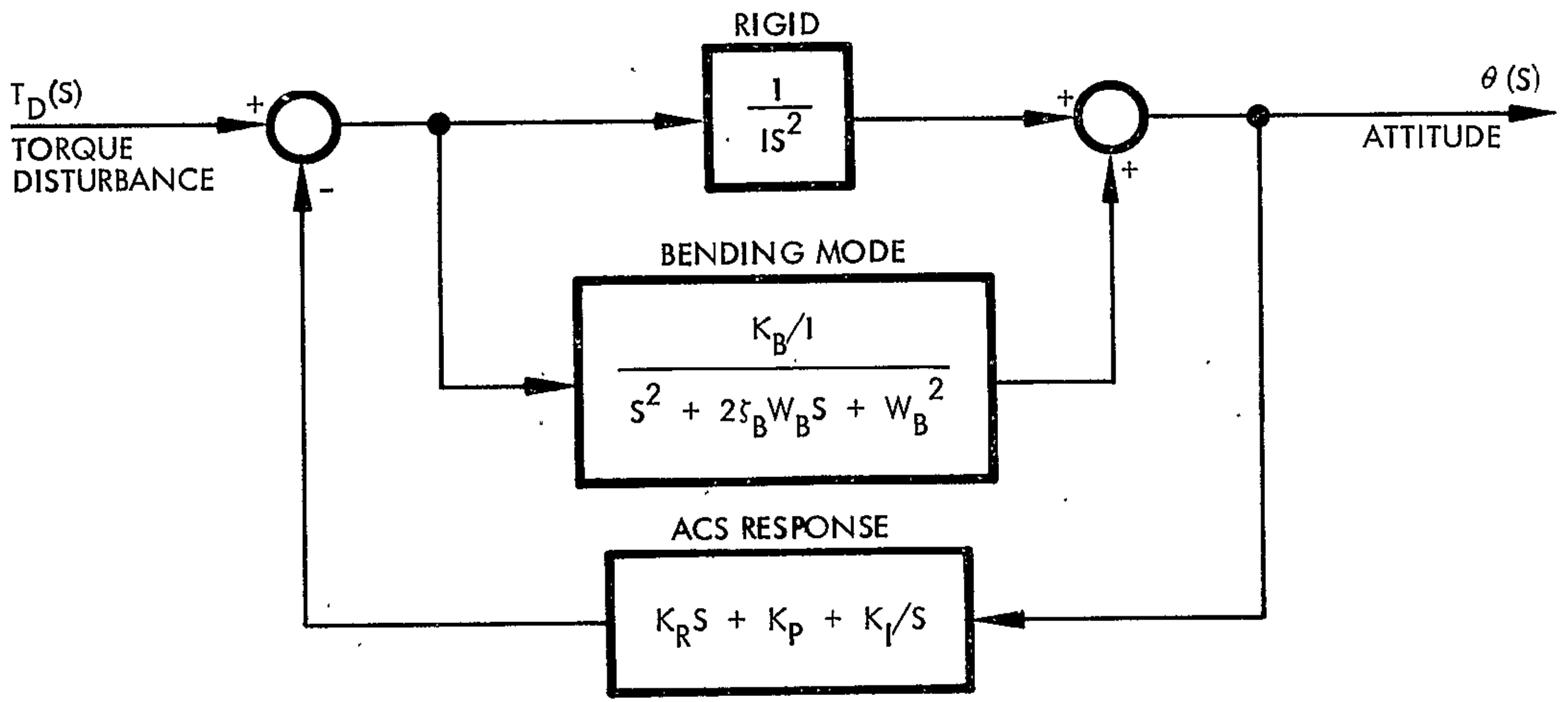
2.3 ATTITUDE CONTROL DYNAMICS

ATTITUDE CONTROL SUBSYSTEM MODEL

- An analysis was performed to evaluate the performance of the attitude control subsystem (ACS) as a function of bandwidth. Two values of bandwidth, 0.01 and 0.001 Hertz, were investigated. The mathematical model of the control system and assumptions used in this analysis are shown on the chart. The model used is a simplified dynamic model which considers the bending mode of the solar array only. The gains used were extrapolated from previous analyses.

- Three disturbances were investigated: (1) the roll response to a disturbance caused by CMG unbalance; (2) a torque of 5 foot pounds resulting from an acceleration at the solar array drive; and (3) a crew motion disturbance of 22.5 pounds at 50 feet from the cluster center of mass. The crew motion disturbance model is based on the model reported in NASA TM X-64972 (Ref. 19).

ATTITUDE CONTROL SUBSYSTEM MODEL



ASSUMPTIONS:

1. IDEALLY INFINITE BANDWIDTH OF SENSORS AND ACTUATORS
2. $K_B = 8$, $f_B = 0.04$ Hz, $W_B = 2\pi f_B$, $\zeta_B = .005$
3. $K_R = 4.74\ I\ f_{BW}$

   $K_P = 8.41\ I\ f_{BW}^2$

   $K_I = 6.64\ I\ f_{BW}^3$
4. ALL OTHER BENDING MODES ARE NEGLIGIBLE
5. CONTROL SYSTEM BANDWIDTH, $f_{BW} = 0.01$ Hz

ROLL AXIS RESPONSE TO CMG UNBALANCE TORQUES

- As shown on the chart, the magnitude of the displacement (θ) and displacement rate ($\dot{\theta}$) in the roll response to the CMG unbalance disturbance was so small for both bandwidths that this response did not aid in the discrimination of bandwidths.

ROLL AXIS RESPONSE TO CMG UNBALANCE TORQUES

- AT 150 Hz (CMG WHEEL FREQUENCY)
- 22 LB FORCE, 2 CMG, EACH AT 1.4 FT ARM
- VIBRATION MOMENT = 61.6 FT-LB

f_{BW}	Hz	0.01	0.001	0.01	0.001
CONFIGURATION	—	PM	PM	PM + ORBITER	PM + ORBITER
I	SLUG-FT^2	286,445	→	1.5×10^6	→
$\theta(S)/T_D(S)$	$\frac{DEG}{FT\text{-}LB}$	2.03×10^{-9}	2.03×10^{-9}	3.87×10^{-10}	3.87×10^{-10}
$\dot{\theta}(S)/T_D(S)$	$\frac{DEG/SEC}{FT\text{-}LB}$	1.91×10^{-6}	1.91×10^{-6}	3.65×10^{-7}	3.65×10^{-7}
θ	DEG	1.25×10^{-7}	1.25×10^{-7}	2.38×10^{-8}	2.38×10^{-8}
$\dot{\theta}$	DEG/SEC	1.18×10^{-4}	1.18×10^{-4}	2.25×10^{-5}	2.25×10^{-5}

TRANSIENT RESPONSES OF POWER MODULE CONFIGURATIONS

- The long-term response to the crew motion disturbance clearly indicates that the 0.001 Hertz bandwidth is unacceptable.

- A structural natural frequency of 0.04 Hz is considered compatible with the $f_{BW} = 0.01$ Hz control system bandwidth.

TRANSIENT RESPONSES OF POWER MODULE CONFIGURATIONS

- CREW MOTION STEP TORQUE DISTURBANCE INPUT = 22.5 LB X 50 FT = 1125 FT-LB IN PITCH (SORTIE MODE)
- SOLAR ARRAY STEP TORQUE DISTRUBANCE INPUT = 5 FT-LB IN PITCH.

		f_{BW} (Hz)					
		0.01	0.001	0.01	0.001	0.01	0.001
CONFIGURATION	---	PM	PM	PM	PM	PM + ORBITER	PM + ORBITER
AXIS	---	ROLL	ROLL	PITCH	PITCH	PITCH	PITCH
I SLUG-FT^2		286,445		96,628		10^7	
θ/T_D	SHORT TERM	NOT APPLICABLE	±.0023 @t = 20 SEC	NOT APPLICABLE	±.014 @t = 20 SEC	NOT APPLICABLE	±.00013 @t = 20 SEC
DEG / FT-LB	LONG TERM	0.191 @t = 110 SEC	18.8 @t = 1100 SEC	0.567 @t = 110 SEC	55.8 @t = 1100 SEC	0.0055 @t = 110 SEC	0.54 @t = 1100 SEC
θ	SHORT TERM	NOT APPLICABLE	NOT APPLICABLE	NOT APPLICABLE	±.07	NOT APPLICABLE	±.00065 ±.146
(DEG)	LONG TERM	NOT APPLICABLE	NOT APPLICABLE	2.84	279	0.0275 6.2	2.7 607

TYPICAL ROTATIONAL DISTURBANCES

- An analysis was performed to investigate the ability of the control system to provide a 10^{-5} g environment in the presence of a number of disturbance torques.
- The system was modeled as described on the previous charts with a band width of 0.01 Hertz.
- Disturbance torques cause an acceleration about the center of mass, which is a function of the distance from the center of mass, R. The maximum distance at which the 10^{-5} g is maintained is given by

$$R = \frac{4.67 \times 10^{-4}}{(f^2)\ (\phi)}$$

where f = the frequency of the disturbance, ϕ = peak amplitude of the disturbance.

- The results of the analysis, shown on the chart, indicate that crew disturbance is the limiting factor on acceleration level. The duration of these disturbances are relatively short, typically less than 1 second.

TYPICAL ROTATIONAL DISTURBANCES

DISTURBANCE	MAXIMUM DISTANCE (FEET) – CLUSTER CENTER OF MASS TO EXPERIMENT FOR 10^{-5} G *
CREW MOTION	27
SOLAR PANEL DRIVE	2432
CMG SYSTEM UNBALANCE	> 27 AFTER STRUCTURAL FILTERING AND ISOLATION (150 Hz)
GRAVITY GRADIENT	
X-LOCAL VERTICAL	79
X-POP	3000
COOLANT LOOP EFFECT	3522

* ROTATIONAL ACCELERATIONS PROPORTIONAL TO DISTANCE FROM CENTER OF MASS.

TYPICAL TRANSLATIONAL ACCELERATION DISTURBANCES

- The linear translation resulting from a wall push-off by a crew member causes a 10^{-4} g acceleration for 0.8 seconds. In the model used for this analysis it was assumed that 1.6 seconds elapsed before an equal and opposite push-off occurred. The total vehicle displacement using this model was 0.0273 inches, which is the magnitude of the sway space required if the entire experiment package were to be levitated.

- Aerodynamic drag was found to cause no violation of the 10^{-5} g requirement.

TYPICAL TRANSLATIONAL ACCELERATIONAL DISTURBANCES

- WALL PUSHOFF CREW DISTURBANCE

  0 TO 100 NEWTONS (22.48 LB) IN 0.8 SECONDS

  ACCELERATION

  $$a = \frac{F}{M} = \frac{22.48 \text{ LB}}{250{,}000 \text{ LB}} \quad g \approx 10^{-4} g \text{ FOR 0.8 SECONDS}$$

  DISPLACEMENT

  ASSUMING 1.6 SECONDS BETWEEN TWO COUNTERACTING PUSHOFFS
  S = 0.0273 INCHES

- AERODYNAMIC DRAG (1959 ARDC MODEL)

SORTIE	FREE-FLYER
$17{,}000 \text{ FT}^2 \longrightarrow 0.17 \text{ LB}$	$11{,}000 \text{ FT}^2 \longrightarrow 0.10 \text{ LB}$
ACCELERATION	
$a = \frac{F}{M} = \frac{0.17}{250{,}000} = 0.7\ 10^{-6} g$	$a = \frac{F}{M} = \frac{0.10}{29{,}000} = 3.4\ 10^{-6} g$

CMG REQUIREMENT ANALYSIS: MOMENTS OF INERTIA

- Preliminary estimates of the principal axis moments of inertia, for use in this analysis, are provided in the chart. The "Satellite Configurations" referenced in the table are illustrated on page 2-7. This chart repeats some of the data on page 2-15.

- In order to establish the number of CMGs required for the vehicle configurations in the nominal scenario, several assumptions were made. It was assumed that Power Module orientations are limited to: (1) a principal axis perpendicular to the orbit plane for inertial orientations, and (2) any principal axis along the local vertical. It was further assumed that one CMG is required for redundancy, and one additional CMG is adequate for control of disturbances (other than gravity gradient) and maneuvering. The driving requirement for the number of CMGs needed is the control of the gravity gradient cyclic torques in the POP orientation.

- From the gravity gradient torque equations and the momentum storage capacity of the CMGs (2300 ft-lb-seconds, each), it can be shown that the number of CMGs required to control the cyclic torques on the axis that is POP is given by 0.3797 (I_1-I_2), where I_1 and I_2 are the vehicle principal axes that are not POP.

CMG REQUIREMENT ANALYSIS: MOMENTS OF INERTIA

SATELLITE CONFIGURATION	PRINCIPAL AXES (SLUG FT^2/10^6)		
	Ix'	Iy'	Iz'
A-1	34.9	41.1	7.8
A-2	41.4	46.1	9.7
A-3	2.43	0.51	2.59
A-4	45.5	49.0	11.5
A-5	3.98	1.04	4.30
B-1	0.74	0.31	0.96
B-2	2.48	0.59	2.62
B-3	2.57	0.75	2.69
C-1	0.74	0.31	0.96
C-2	2.48	0.59	2.62
D-1	4.92	16.59	21.22

INERTIAL ORIENTATIONS FOR SCENARIO I SATELLITE CONFIGURATIONS

- Based on the magnitude of the cyclic gravity gradient torques, the number of CMGs required to control the Scenario I satellite configurations are shown in the chart. The very large satellites in some of their orientations obviously require considerably more control capability than can realistically be supplied by a power-module system.

- The orientations of the satellite which can be controlled by appropriate configurations of the ACS proposed for the power-module evolutionary family are shown in the chart.

INERTIAL ORIENTATIONS FOR SCENARIO I SATELLITE CONFIGURATIONS

CONFIG-URATION	ORIEN-TATION	NUMBER OF CMGs TO CONTROL CYCLIC TORQUES		CMGs REQUIRED FOR MANEUVERING AND REDUNDANCY	TOTAL CMGs RQD	ORIEN-TATION
		CALCULATED	INTEGRAL			
A-1	XPOP	12.64	13	2	15	
	YPOP	10.29	11	2	13	
	ZPOP	2.35	3	2	5	✓
A-2	XPOP	13.82	14	2	16	
	YPOP	12.04	12	2	14	
	ZPOP	1.78	2	2	4	✓
A-3	XPOP	0.79	1	2	3	✓
	YPOP	0.06	1	2	3	✓
	ZPOP	0.73	1	2	3	✓
A-4	XPOP	44.63	45	2	47	
	YPOP	41.13	42	2	44	
	ZPOP	1.32	2	2	4	✓
A-5	XPOP	1.24	2	2	4	✓
	YPOP	0.12	1	2	3	✓
	ZPOP	1.12	2	2	4	✓
B-1	XPOP	0.24	1	2	3	✓
	YPOP	0.08	1	2	3	✓
	ZPOP	0.16	1	2	3	✓
B-2	XPOP	0.77	1	2	3	✓
	YPOP	0.05	1	2	3	✓
	ZPOP	0.72	1	2	3	✓
B-3	XPOP	0.74	1	2	3	✓
	YPOP	0.05	1	2	3	✓
	ZPOP	0.69	1	2	3	✓

2.4 STRUCTURAL DYNAMICS

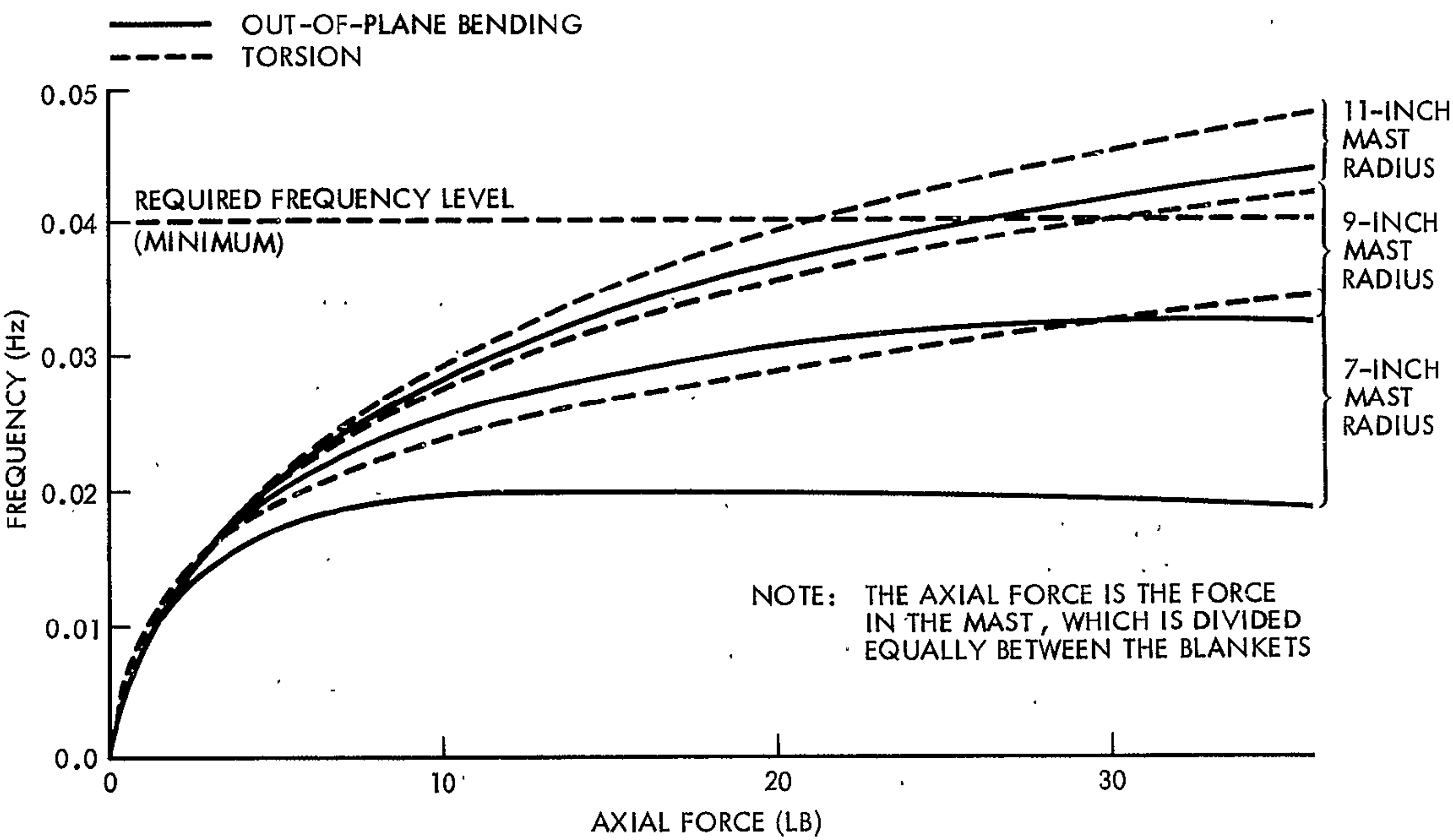
MODE FREQUENCY VS PRELOAD FOR SOLAR ARRAY

- The figure shows the results of a trade study that evaluated the interaction of solar array sheet preload and mast radius on the first bending and torsion modes of the solar array for configuration 25-1 (See page 2-5,-7 and Volume 1). As can be seen, the 11-inch radius mast meets the 0.04 Hz frequency requirement (See page 2-34).

- The studies were performed with the solar array cantilevered at its base. The design-driver basic requirement, stemming from attitude control system characteristics, is that the solar array system must have a natural frequency of more than 0.04 Hz. This is accomplished using an 11.0 inch radius coilable mast.

- When the solar array modes are coupled with the dynamic characteristics of the Power Module, the overall minimum frequencies should increase slightly above the 0.04 Hz requirement.

MODE FREQUENCY VS PRELOAD FOR SOLAR ARRAY



DYNAMICS COMPARISON OF MASTS

- The dynamic response and load-carrying capabilities of the PM solar array were developed as a function of mast strength and stiffness parameters. The accompanying chart shows the dynamic response characteristics as a function of wing width and length (area), and of the mast. Case 4 represents the recommended configuration for the 25 kW Power Module.

- As can be seen for an aspect ratio of a wider and shorter array, (Case 5), a higher natural frequency can be developed using the same same 11.0 inch radius mast.

DYNAMICS COMPARISON OF MASTS

CASE	MAST CHARACTERISTICS RADIUS (IN.) BENDING STIFFNESS (LB-IN.^{2}) TORSIONAL STIFFNESS (LB-IN.^{2})	WING WIDTH (IN.)	WING LENGTH (IN.)	BENDING FREQUENCY f_B (SHEET TENSION)	TORSIONAL FREQUENCY f_T
1	7.0 19×10^6 0.4×10^6	158	1,240	0.043 (26 LB)	0.072
2	7.0 19×10^6 0.4×10^6	316 2-158 BLANKETS	1,535	0.020 (20 LB)	0.029
3	9.0 52×10^6 1×10^6	316 2-158 BLANKETS	1,535	0.033 (50 LB)	0.047
4	11.0 116×10^6 2.3×10^6	316 2-158 BLANKETS	1,535	0.046 (50 LB)	0.055
5	11.0 116×10^6 2.3×10^6	474 2-237 BLANKETS	1,100	0.07 (200 LB)	0.08

DYNAMIC/LOAD CONSIDERATIONS

- Design load conditions which are likely to be design drivers were considered in conjunction with the dynamic analyses referenced with the previous chart. Illustrated are two conditions: (1) sizing to withstand a typical 0.003g acceleration* (during drag makeup); and (2) sizing to satisfy an attitude control constraint of first bending-mode frequency in the deployed array of not less than than 0.04 Hz.

- For these two criteria, the first condition requires a 9.3 inch radius mast, and the second, and governing, condition requires an 11.0 inch radius mast. The 11.0 inch mast provides a bending capability to withstand 0.0045 g.

*See page 2-83. For more severe accelerations, the solar arrays are retracted.

DYNAMIC/LOAD CONSIDERATIONS

- MAST SIZED TO MEET 0.003 G DESIGN CONDITION:

  MOMENT (MAST BENDING) = 1,551 IN.-LB

  1.5 (F.S.) X 1,551 = 2,326 IN.-LB ≈ 200 FT-LB

  $M_{C_R} = 5.284 \times 10^7 \epsilon^4 R^3$

  R = RADIUS OF MAST

  ϵ = LONGERON STRAIN

  $\therefore$ R = <u>9.3 IN.</u>

- MAST SIZED TO MEET 0.04 HZ f_B, f_t

  $\therefore$ R = <u>11.0 IN.</u> (REFERENCE PREVIOUS CHART ON MODE FREQUENCY FOR SOLAR ARRAY)

- WITH LARGER RADIUS REQUIRED FOR STIFFNESS, MORE BENDING STRENGTH IS AVAILABLE

  $\therefore$ 11.0 IN MAST IS CAPABLE OF ≈ <u>0.0045 G</u>

RECOMMENDED STRUCTURAL DYNAMICS ANALYSES & TESTS

- The chart lists several analyses and tests that typically are required for space vehicle configurations of the 25 kW class. As can be seen, the analysis and testing cover primary and secondary structure loads and environments in both the ascent and orbital configurations.

- Attention also is directed to ground transportation and handling as well as major subsystem testing such as the radiator assembly.

RECOMMENDED STRUCTURAL DYNAMICS ANALYSES AND TESTS

DESIGN/ANALYSES	QUALIFICATION TESTS
• FLEXIBILITY MODELS FOR ANALYSIS OF POWER MODULE CONTROL AND ORBITAL LOADS	• MODAL SURVEY TEST OF COMPLETE POWER MODULE ASCENT CONFIGURATION
• EVALUATION OF COMBINED SHUTTLE AND POWER MODULE MODELS FOR ASCENT AND LANDING LOADS	• SYSTEM-LEVEL ASCENT CONFIGURATION ACOUSTIC TEST
• GROUND TRANSPORTATION ANALYSIS TO DETERMINE HANDLING LOADS	• SYSTEM-LEVEL PYROSHOCK TEST
• SHOCK, VIBRATION, AND ACOUSTIC ANALYSIS TO ESTABLISH EQUIPMENT SERVICE ENVIRONMENTS	• RADIATOR DYNAMIC DEVELOPMENT AND QUALIFICATION TESTS
• SECONDARY STRUCTURE DYNAMIC ANALYSIS TO CONTROL EQUIPMENT SERVICE ENVIRONMENTS	• LIMITED SECONDARY EQUIPMENT SUPPORT STRUCTURE STATIC INFLUENCE TESTING

RECOMMENDED DEPLOYMENT DYNAMICS AND MECHANISM ANALYSES/TESTS

- Typical analyses to be performed in the design and qualification cycles are identified. Analyses are performed by synthesizing the deployable items in a rigid body sense to determine: trajectories, time histories, force/torque margins and quasi static loads. Where necessary, elastic body models are to be generated to determine the loads during transients such as release and lock up.

- The functional tests are those typically included as part of the development and qualification master plan. Qualification tests are to be performed where qualification by similarity or analysis is impossible or inadequate.

RECOMMENDED DEPLOYMENT DYNAMICS AND MECHANISM ANALYSES/TESTS

DESIGN/ANALYSIS	TESTS
• DEPLOYMENT/RETRACTION ANALYSIS OF: • SOLAR ARRAY CONTAINERS • RADIATOR • HI-GAIN ANTENNAS • BERTHING SYSTEM TO DETERMINE: TIME HISTORIES, FORCE/TORQUE MARGINS AND LOADS	• MODULE LEVEL DEPLOYMENT/RETRACTION TESTS WITH CRITICAL ENVIRONMENTS SIMULATED.
• ANALYSIS OF RELEASE/LATCHING MECHANISMS TO DETERMINE MECHANICAL ADVANTAGES, FORCE/TORQUE MARGINS AND LOADS	• COMPONENT AND MODULE LEVEL TESTS OF RELEASE/LATCHING MECHANISMS WITH CRITICAL ENVIRONMENTS SIMULATED.
• EXTENSION/RETRACTION ANALYSIS AS REQUIRED, OF SOLAR ARRAY "SHEET". THIS OPERATION WILL BE QUALIFIED PRIMARILY BY TEST.	• MODULE LEVEL EXTENSION/RETRACTION TESTS WITH CRITICAL ENVIRONMENTS SIMULATED.

2.5 CONTAMINATION EVALUATION

This section assesses the effects of contamination on sensitive surfaces of the Power Module. The potentially damaging contamination sources were identified to be the plumes from the Reaction Control Subsystem (RCS). The vernier thrusters are not potential contamination sources to the PM, because the forward vernier jets are pointing downward and away from the PM. The environment induced by deflection of the rear vernier thruster plumes off the Orbiter wings and by other outgassing sources are not likely to produce condensibles. The solar arrays are retracted during RCS operations due to their dynamic constraints, and therefore are not exposed to the plume flowfield. The impingement forces on the radiator panels are expected to be negligible and are not calculated, since the plumes do not impinge on any of the surfaces directly. The contamination evaluation assesses surface property degradation due to contaminant deposition.

CONTAMINATION EVALUATION

- The most detrimental condensibles are the nitrate salts of monomethylhydrazine (MMH-nitrate) (Ref. 20, 21) from the impingement of N_2O_4 -MMH from the RCS plumes. MMH-nitrate is formed as a result of incomplete combustion at low chamber temperature. It is produced during pulsing, starting-up, and shutting-down operations.

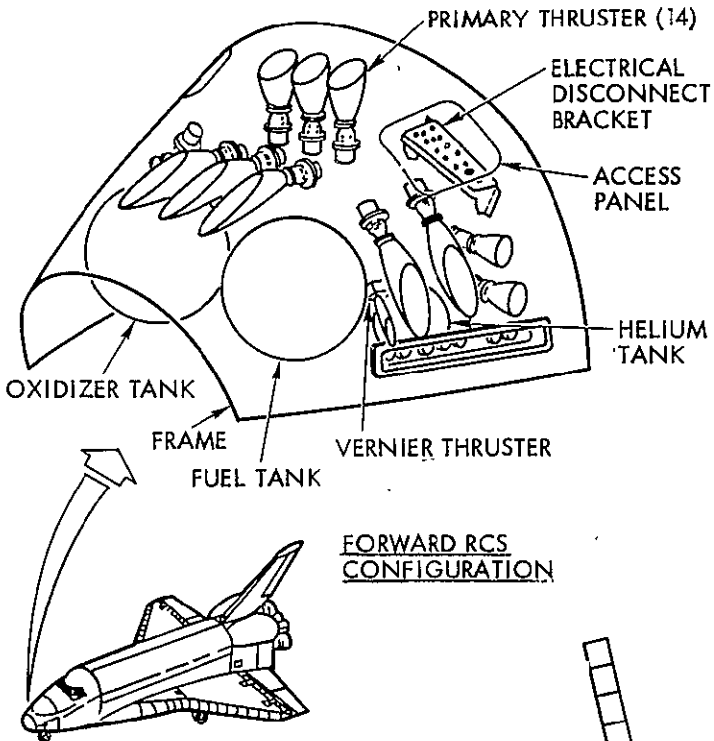
CONTAMINATION EVALUATION

CONTAMINATION SOURCES	CONTAMINANTS	EFFECTS
MAIN THRUSTERS OF FORWARD RCS MODULE RCS VERNIER JETS	MMH-HNO_3, CH_3 NH_2, CO, CH_4, CH_3 N_3, NO_2, H_2O, N_2, NO, H_2, CO_2	• DEPOSITION OF MMH-NITRATE DUE TO DIRECT IMPINGEMENT OF THE PLUMES CAN DEGRADE THE PROPERTIES OF THERMAL CONTROL SURFACES • THE OTHER GASEOUS PRODUCTS IN THE PLUMES HAVE LITTLE EFFECT ON THE PM PERFORMANCE
OUTGASSING, OFFGASSING, EVAPORATORS, CABIN LEAKAGE, AMBIENT REFLECTION	HYDROCARBON CHAIN FRAGMENTS, RTV's VOLATILES, H_2O, N_2, H_2, CO, CO_2, ETC.	• INDUCED ENVIRONMENT HAS LITTLE EFFECT ON THE PM PERFORMANCE

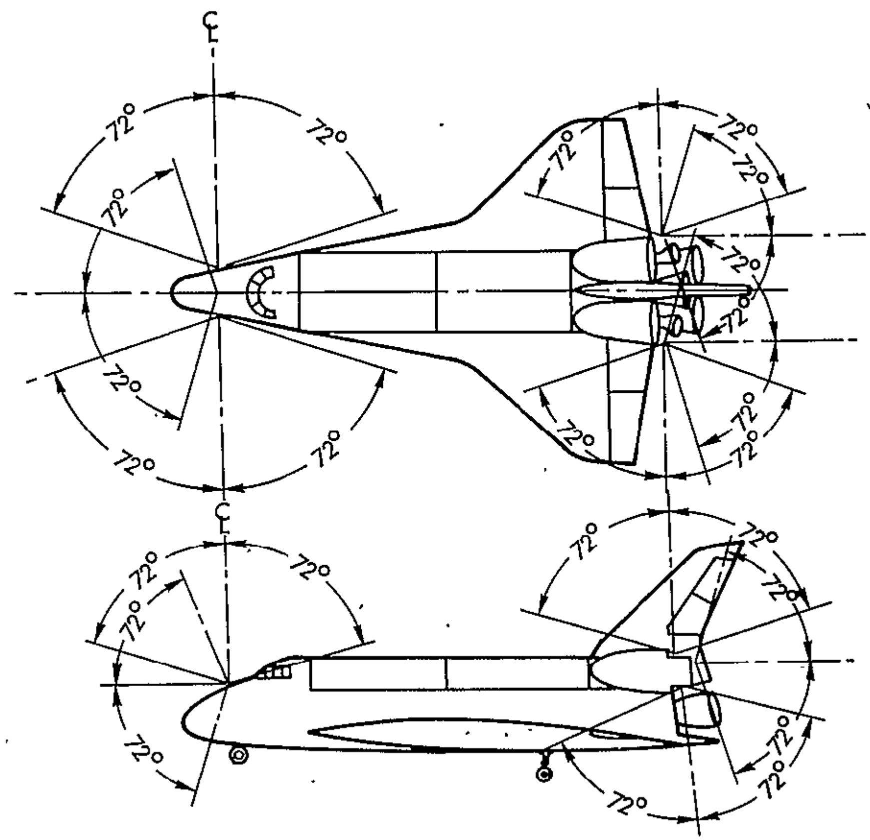
CONTAMINANT FLOW FIELDS

- Parts of the radiator system are in the flowfields of the forward RCS plumes. Induced forces are considered negligible.
- The solar arrays are retracted prior to RCS operation.
- Shuttle interface data are taken from Ref 2, Pages 3-12 and 4-7.

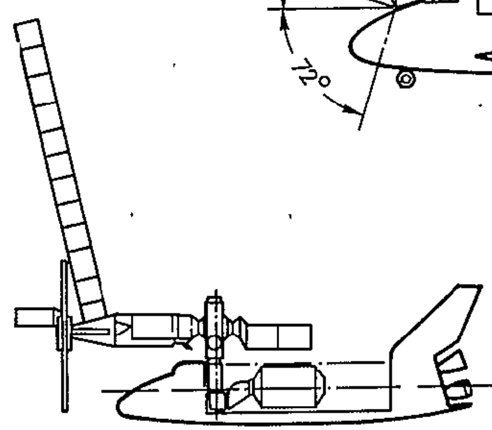
CONTAMINANT FLOW FIELDS



FORWARD RCS CONFIGURATION



FLOW-FIELD CONE ANGLES



TYPICAL SORTIE-MODE ORBITER/PM CONFIGURATION

CONTAMINANT MASS FLUXES

- The mass fraction of the MMH-nitrate can be considered to be 1.2 percent in the boundary layer region, though the amount varies with different thrusters.

- To estimate the contamination effect, it is assumed that the mass fraction of MMH-HNO_3 is 0.1 percent in the total plume efflux.

- The mass fluxes along the center line versus the distances from the RCS thruster are given in the table (Ref. 20, Pages E-1 to E-16).

CONTAMINANT MASS FLUXES

DISTANCE (FT)[1]	MASS FLUX OF MMH-HNO_3 (G/CM^2 – SEC)[2]
25	1.66×10^{-6}
50	4.15×10^{-7}
100	1.04×10^{-7}
500	4.15×10^{-9}
1000	1.04×10^{-9}
2000	2.55×10^{-10}
3000	1.15×10^{-10}

NOTES:

(1) MMH-HNO_3 MASS FLUXES ALONG ORBITER RCS PLUME CENTER LINE.

(2) MASS FRACTION OF MMH-HNO_3 = 0.1% ASSUMED.

CONTAMINATION CONTROL SUMMARY

- Measurements of changes in thermal control surface properties were made at LMSC by deposition of a MMH-HNO_3 layer on the sample surfaces.
- The preliminary result shows that it takes a 0.015 g/cm^3 MMH-HNO_3 layer to increase α and reduce ϵ by 0.1 (Ref. 20). This deposited layer is equivalent to 900 seconds of cumulative transient operation by a RCS main thruster at 25 feet and the MMH-nitrate in the plume is assumed to be completely deposited on the surface.
- The radiator panels are deployed in such a position that direct impingement by RCS main thruster plumes is avoided. The potential threat of performance degradation is therefore minimum. No restraint on RCS operation is required from the standpoint of contamination control for radiator surfaces.
- Although results of a partial assessment of contamination potential on the solar arrays is shown on the chart, the solar arrays will be in the retracted position (due to strength constraints) when RCS thrusters are employed and therefore are not exposed to the RCS plumes.

CONTAMINATION CONTROL SUMMARY

SENSITIVE SURFACES	POTENTIAL DAMAGES	CONCLUSION
RADIATOR SURFACES	INCREASE IN SOLAR ABSORPTANCE, α , AND DECREASE IN EMITTANCE, ϵ , RESULTING IN INADEQUATE HEAT DISSIPATION	NOT CRITICAL
BACKSIDE OF SOLAR ARRAYS	INCREASE IN α AND DECREASE IN ϵ , RESULTING IN HIGHER SOLAR ARRAY TEMPERATURE	NEGLIGIBLE DEGRADATION IN PERFORMANCE
SOLAR ARRAY	REDUCED POWER OUTPUT	SOLAR ARRAY IS NOT IN PLUME FLOWFIELD; THEREFORE NOT A CONCERN

2.6 BERTHING CONCEPTS

BERTHING SYSTEM TRADE STUDY

- A trade study was performed for different conceptual designs of a berthing system (Power Module-to-Orbiter). The system is envisioned as space support equipment installed in the Orbiter bay.

- The basic design requirements and a listing of the design concepts studied are given on the facing chart. The concepts are described, and the evaluations summarized, on the following charts.

BERTHING SYSTEM TRADE STUDY

REQUIREMENTS:

- TO PROVIDE A MATING INTERFACE, POWER MODULE TO ORBITER, WITH POWER MODULE IN SORTIE OR MAINTENANCE MODE. INTERFACE TO BE AT STA X=619, Y=0 AND Z=515.
- STOWABLE WITHIN THE ORBITER PAYLOAD COMPARTMENT DURING LAUNCH AND REENTRY.
- TO BE COMPATIBLE WITH THE PAYLOAD COMPARTMENT ATTACHMENT AND LOADING SYSTEM.

CANDIDATE DESIGN CONCEPTS

A – TELESCOPING TUNNEL WITH DOCKING COLLAR (NASA BASELINE)
B – DOCKING COLLAR WITH TELESCOPING STRUTS
C – DOCKING COLLAR WITH RADIAL ARMS AND SUPPORT STRUT
D – DOCKING COLLAR WITH HINGED CRADLE
E – BERTHING LATCH SYSTEM WITH ELEVATING TABLE AND MAINTENANCE PLATFORM
F – DOCKING COLLAR WITH HINGED REMOVEABLE ADAPTOR PLATFORM

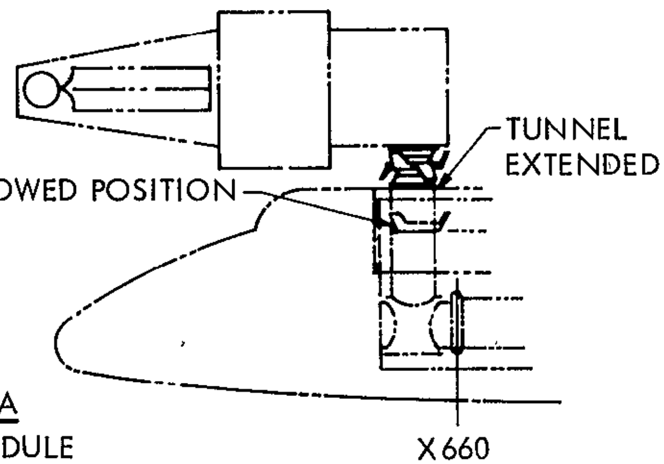
BERTHING SYSTEM DESIGN CONCEPTS A & B

- Design Concept A. This concept, with a docking collar and telescoping tunnel connecting to the Orbiter airlock, was shown as the initial NASA baseline concept (Ref 4, pages 1-43 & 3-9)*. In the course of the 25 kW evolution study the requirement for pressurized IVA access to the Power Module has been shown to be unnecessary.

- Design Concept B. The docking collar, supported on telescoping struts and mounted on a platform-and-frame attached to the Orbiter sill trunnions and keel fitting, was considered as a lower-cost, lighter-weight approach. Locking devices are built into the struts to hold the docking collar in the stowed or deployed position.

* Also see Ref. 2, Pages 9-2 and 9-15.

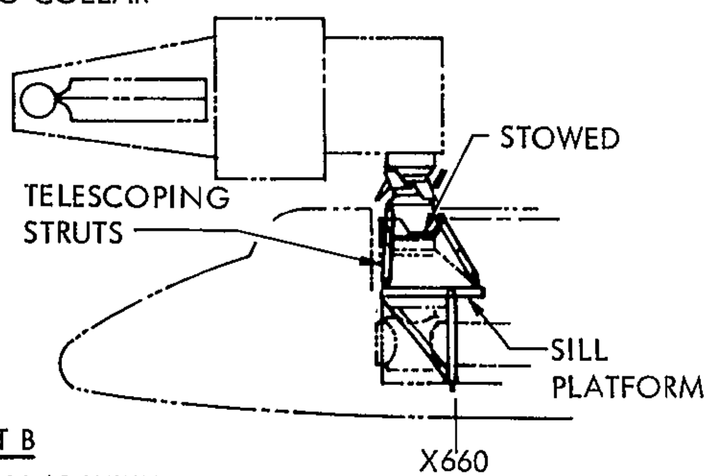
BERTHING SYSTEM DESIGN CONCEPTS A&B



CONCEPT A

DOCKING MODULE TELESCOPING TUNNEL WITH DOCKING COLLAR

ADVANTAGES	DISADVANTAGES
• MOST ECONOMIC USE OF SPACE • MAXIMUM GROWTH POTENTIAL • MINIMUM NUMBER OF PARTS • INTEGRATED INTO AIR LOCK SYSTEM • EVA AND IVA OPERATION	• HIGH COST • HIGH WEIGHT • DOCKING MODULE NOT CURRENTLY PART OF THE SHUTTLE PROGRAM



CONCEPT B

DOCKING COLLAR WITH TELESCOPING STRUTS

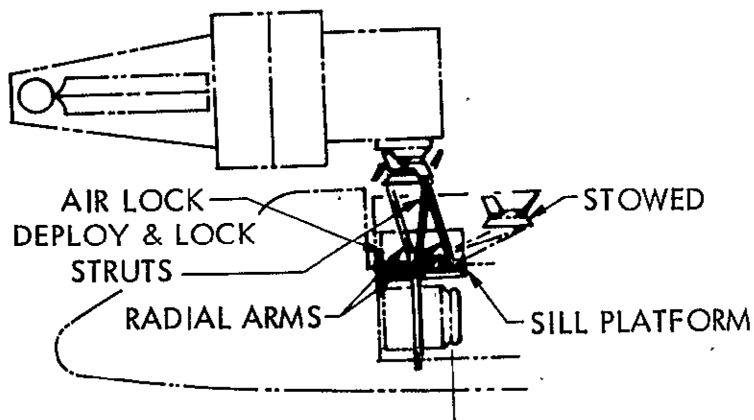
ADVANTAGES	DISADVANTAGES
• ECONOMIC USE OF SPACE • MINOR INTER-FERENCE WITH RESERVED EVA ENVELOPE • SMALL NUMBER OF PARTS	• INTERFERENCE WITH AIRLOCK • COMPLICATED DESIGN OF TELESCOPING/LOCKING SYSTEM • HIGH COST

BERTHING SYSTEM DESIGN CONCEPTS C & D

- Design Concept C. The docking collar is supported by radial arms hinged from a platform attached to the Orbiter sill trunnions, and a frame attached to the keel fitting. A telescoping strut extends to move the docking collar from its stowed position within the payload compartment to the deployed position outside the Orbiter. Locking devices are built into the telescoping strut for both stowed and deployed positions.

- Design Concept D. The docking collar is attached to a cradle which is mounted by hinges on a platform-and-frame attached to the Orbiter sill trunnions and keel fitting. A folding strut, with locking devices, extends to move the collar and cradle from the stowed to deployed position.

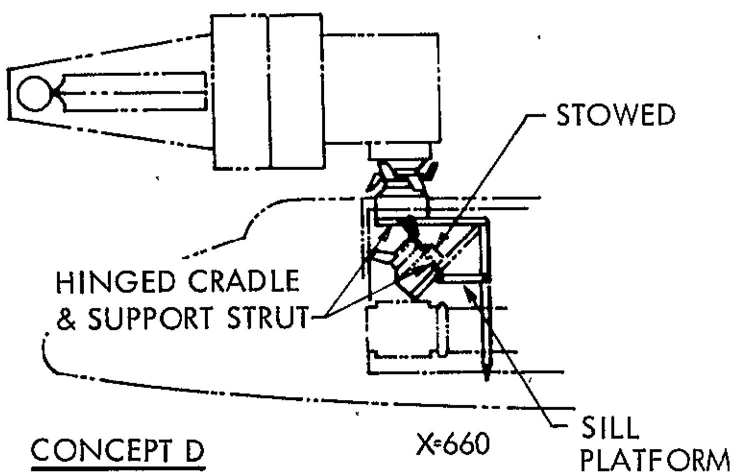
BERTHING SYSTEM DESIGN CONCEPTS C&D



CONCEPT C

DOCKING COLLAR/RADIAL ARM AND SUPPORT STRUT SYSTEM

ADVANTAGES	DISADVANTAGES
• NO INTERFERENCE WITH AIR LOCK	• PROJECTS AFT OF STA 660 WHEN STOWED • NO GROWTH POTENTIAL • NO ROOM FOR MAINTENANCE PLATFORM



CONCEPT D

DOCKING COLLAR/HINGED CRADLE SYSTEM

ADVANTAGES	DISADVANTAGES
• SIMPLICITY OF DESIGN	• INTERFERENCE WITH AIR LOCK • NO GROWTH POTENTIAL • PROJECTS INTO PAYLOAD BAY AFT OF STA 660 • NO ROOM FOR MAINTENANCE • PLATFORM

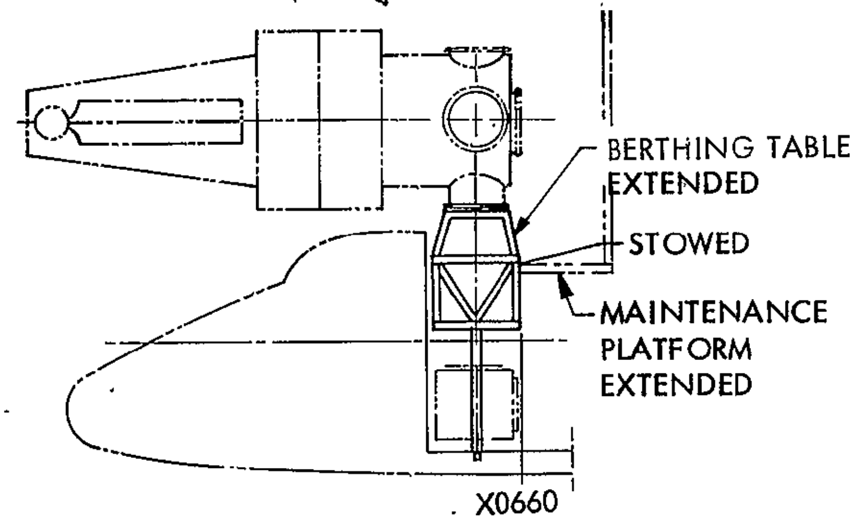
BERTHING SYSTEM DESIGN CONCEPTS E & F

- Design Concept E. Berthing latches and guides are attached to a rotation ring on an elevator table which is mounted on four ball screws and nuts to a sill platform-and-frame attached to the Orbiter sill trunnions and a keel fitting. Synchronized electric motors (one at each ball screw) elevate and lock the berthing latch/table from the stowed position to the deployed position and vice versa. The berthing latch system (with Power Module attached), can be rotated into any desired position by an electric motor/rack and pinion system mounted upon the elevator table. A maintenance platform is stowed under the sill platform. Attached to it is a folded access mast.

- Design Concept F. The system consists of a separate docking adapter platform with a docking collar on one face and manipulator grapple units on its reverse face. This assembly is mounted by hinges on a platform-and-frame which is attached to the Orbiter trunnions and keel fitting. The platform has RMS end effectors mounted upon it, to which the adaptor platform grapple units are mated when the adapter platform and collar are rotated (by electric motor) into the deployed position. This adapter platform, with docking collar, can be removed by the RMS for attachment to other payloads.

BERTHING SYSTEM DESIGN CONCEPTS E&F

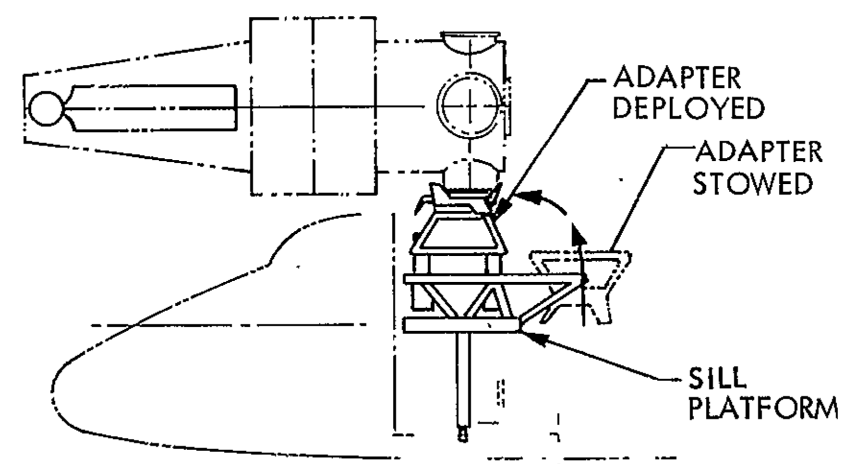
CONCEPT E



BERTHING LATCHES/ELEVATING TABLE AND MAINTENANCE PLATFORM SYSTEM

ADVANTAGES	DISADVANTAGES
• SIMPLICITY AND RELIABILITY OF EXTENSION MECHANISM (BALL SCREWS/ELECTRIC MOTORS) • COMPACT DESIGN-NO INTRUSION AFT OF STA 660 • INCLUDES MAINTENANCE PLATFORM • LOW WEIGHT AND COST	• DEVELOPMENT TESTING OF BERTHING LATCH, SYSTEM.

CONCEPT F



DOCKING COLLAR WITH HINGED AND REMOVEABLE ADAPTOR PLATFORM

ADVANTAGES	DISADVANTAGES
• GROWTH POTENTIAL-REMOVEABLE ADAPTER (BY RMS) CAN BE ATTACHED TO OTHER PAYLOADS • NO INTERFERENCE WITH AIR LOCK	• COMPLICATED MECHANISM • HIGH WEIGHT AND COST • PROJECTS INTO PAYLOAD BAY AFT OF STA 660.0

BERTHING SYSTEM CONCEPTUAL STUDY CONCLUSIONS

- Concept A includes a pressurized, extendable docking module, which is both costly and heavy. In most mission applications IVA through a pressurized docking collar and tunnel to the Power Module is not considered to be necessary. In addition, it can only be used at STA Xo = 619 (at the airlock) which limits its flexibility.

- All other concepts can be installed where required along the Orbiter bay. Of these other concepts, B, C, D, and F utilize the docking collar, which is both costly and heavy. Concept E utilizes a latching system with guides which will be lighter. Furthermore, it is expected to be a common system considered for the Space Telescope Program, thereby reducing the cost.

- Accordingly, Concept E is recommended with the rationale as summarized on the chart. A preliminary weight for this SSE is estimated to be 1,181 lb.

- The two charts which follow illustrate: (1) the stowed and deployed positions with respect to the Orbiter positions while changing the Power Module from stowed to berthed conditions; and (2) a weight summary of the recommended berthing system.

BERTHING SYSTEM
CONCEPTUAL STUDY CONCLUSIONS

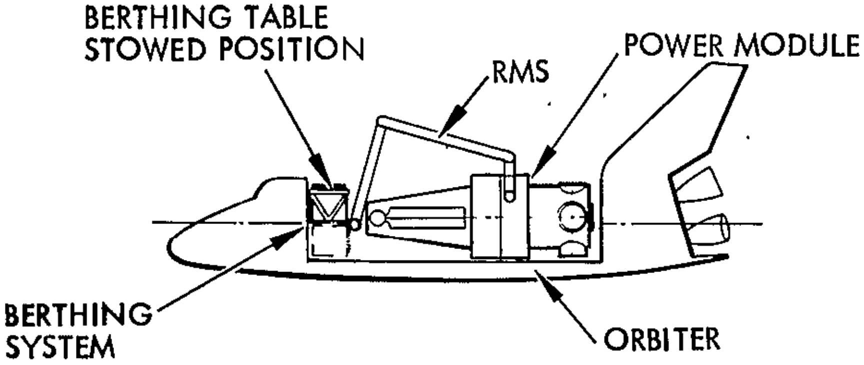
RECOMMENDATION

DEVELOP CONCEPT E – ELEVATING TABLE WITH BERTHING LATCHES AND MAINTENANCE PLATFORM

RATIONALE

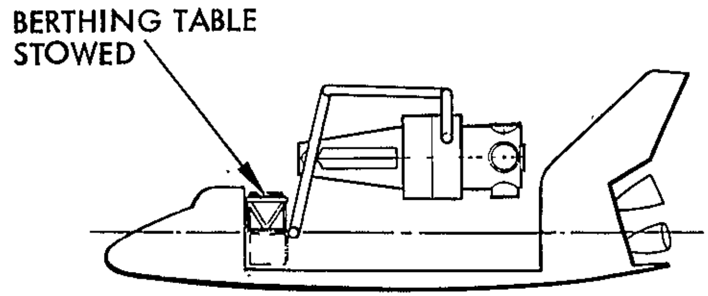
- RELIABILITY, SIMPLICITY AND COMPACTNESS OF DESIGN
- INCLUDES MAINTENANCE PLATFORM IN DESIGN
- NO INTERFERENCE WITH AIR LOCK
- LOW WEIGHT
- LOWCOST: USE OF "OFF THE SHELF" PARTS (SAGINAW BALL SCREWS, MOTORS, ETC.)
- CAN BE SIMILAR, IF NOT IDENTICAL, TO SYSTEM BEING DEVELOPED FOR SPACE TELESCOPE, WITH ATTENDANT ECONOMIES

INITIAL DEPLOYMENT/BERTHING SEQUENCE



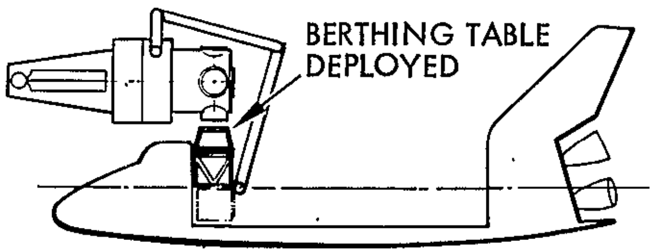
STEP I

RMS ATTACHED TO POWER MODULE

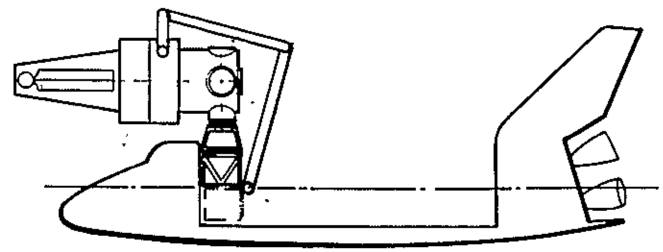


STEP II

INITIAL LIFT FROM ORBITER



POWER MODULE POSITIONED
ABOVE BERTHING TABLE



STEP IV

POWER MODULE LOWERED
ONTO TABLE AND BERTHED

DEPLOYMENT/BERTHING SEQUENCE

BERTHING CONCEPT E-WEIGHT SUMMARY

ITEM	WEIGHT (LB)
BEAMS	450
ORBITER FITTINGS	70
TABLE	125
WORK PLATFORM	150
LATCHES	50
MECHANISM, MOTORS	100
SUBTOTAL	945
CONTINGENCY – 25%	236
TOTAL	1,181

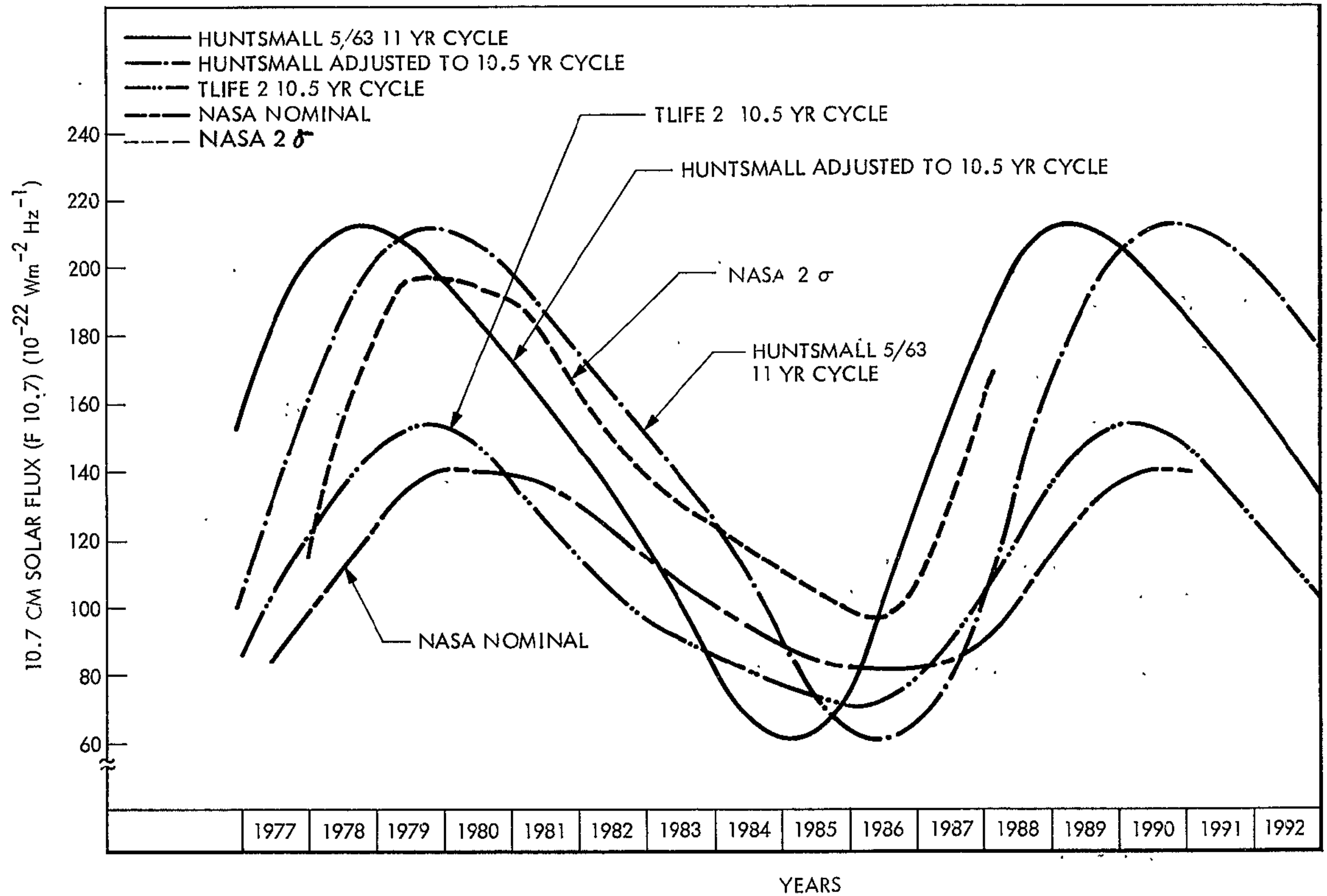
2.7 ORBIT REBOOST OPTIONS

SOLAR CYCLE EFFECT ON UPPER ATMOSPHERE

- Atmospheric density varies as a function of heating from ultraviolet solar radiation. The 10.7-cm solar flux ($F_{10.7}$) is used as an index of solar ultraviolet radiation and varies over a 10 1/2 year cycle (approximately).

- Maximum and minimum atmospheric densities were calculated from the Jacchia '71 log density model based on maximum and minimum predictions of $F_{10.7}$ during vehicle flight life times. (See References 24, 25, 26 and 27.)

SOLAR CYCLE EFFECT ON UPPER ATMOSPHERE



SCENARIO I SATELLITE BALLISTIC CHARACTERISTICS

- Each satellite configuration was considered to have a typical average (between minimum and maximum orbital orientation) drag coefficient, and resultant ballistic coefficient. These coefficients were used in computing aerodynamic drag and orbit decay rate. The configuration descriptions and designations are provided in Section 2.1.

- The flight years and range of solar flux to be expected are also tabulated for each configuration.

- For the dormant (on-orbit storage) mode the first flight vehicle, with arrays retracted to 12 1/2% area and oriented with the edge of the arrays into the wind, will have a drag coefficient of 8 and a ballistic coefficient of 0.33.

SCENARIO I SATELLITE BALLISTIC CHARACTERISTICS

SATELLITE CONFIGURATION (SEE SECTION 2.1)		DRAG COEFFICIENT[1]			WEIGHT (1,000 LB)	BALLISTIC COEFFICIENT ($FT^3/LB\ SEC^2$) (TYP. AVER.)	FLIGHT YEARS	SOLAR FLUX,[2] $F_{10.7}$	
		HIGHEST	LEAST	TYPICAL AVERAGE				MIN	MAX
25 kW	A-1	125	20	73	64.3	2.81	86 - 87	60	180
PM	B-1	106	8	57	59.3	2.38	83 - 85	60	150
	C-1	106	8	57	59.3	2.38	88 - 89	90	210
	C-2	106	8	57	70.8	1.99	90 →	–	210
50 kW	A-2	218	26	122	85.8	3.52	88 - 89	90	210
PM	A-3	207	14	111	60.8	4.52	89 →	120	210
	B-2	207	14	111	70.8	3.88	86 - 88	60	210
	B-3	207	14	111	80.8	3.40	89 →	120	210
100 kW	A-4	419	35	227	102.3	5.50	90 →	–	210
PM	A-5	405	24	215	67.3	7.91	91 →	–	210

(1) REFERENCE AREA = 154 FT^2
(FOR ALL CONFIGURATIONS)

(2) $F_{10.7}$ = 10.7 CM WAVE LENGTH SOLAR FLUX, IN UNITS OF 10^{-22} WATT M^{-2} H_2^{-1} BANDWIDTH. $F_{10.7}$ IS GENERALLY USED INDEX OF SOLAR EXTREME ULTRA-VIOLET RADIATIONS.

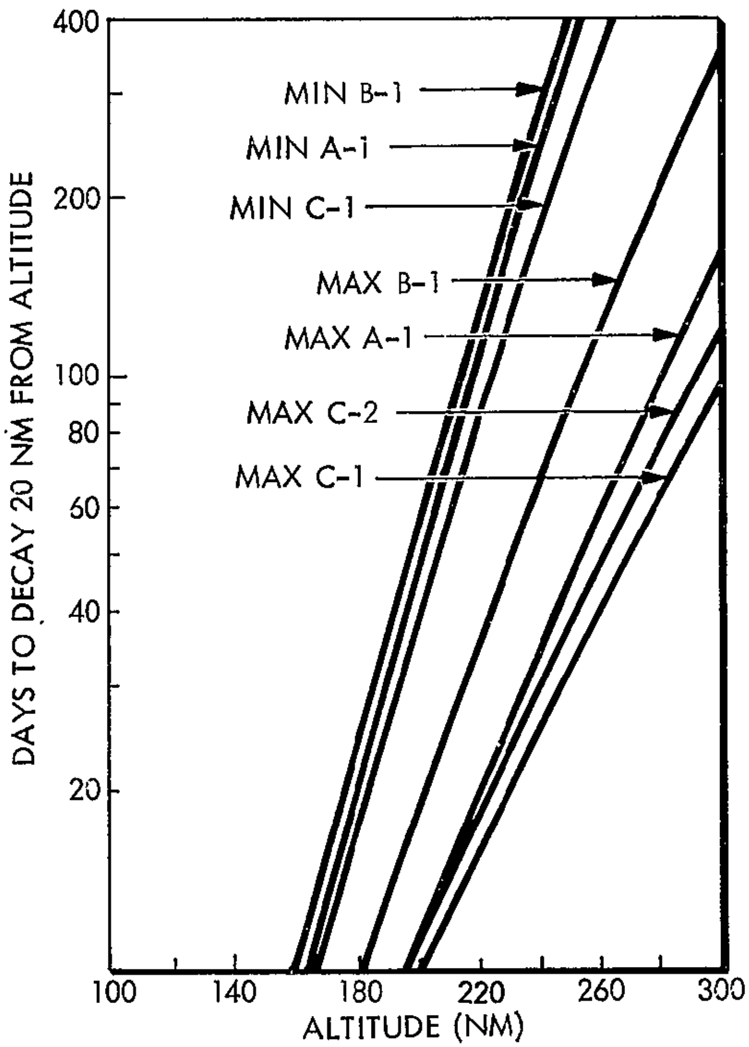
ALTITUDE EFFECTS ON AERO DRAG FOR 25 kW PM

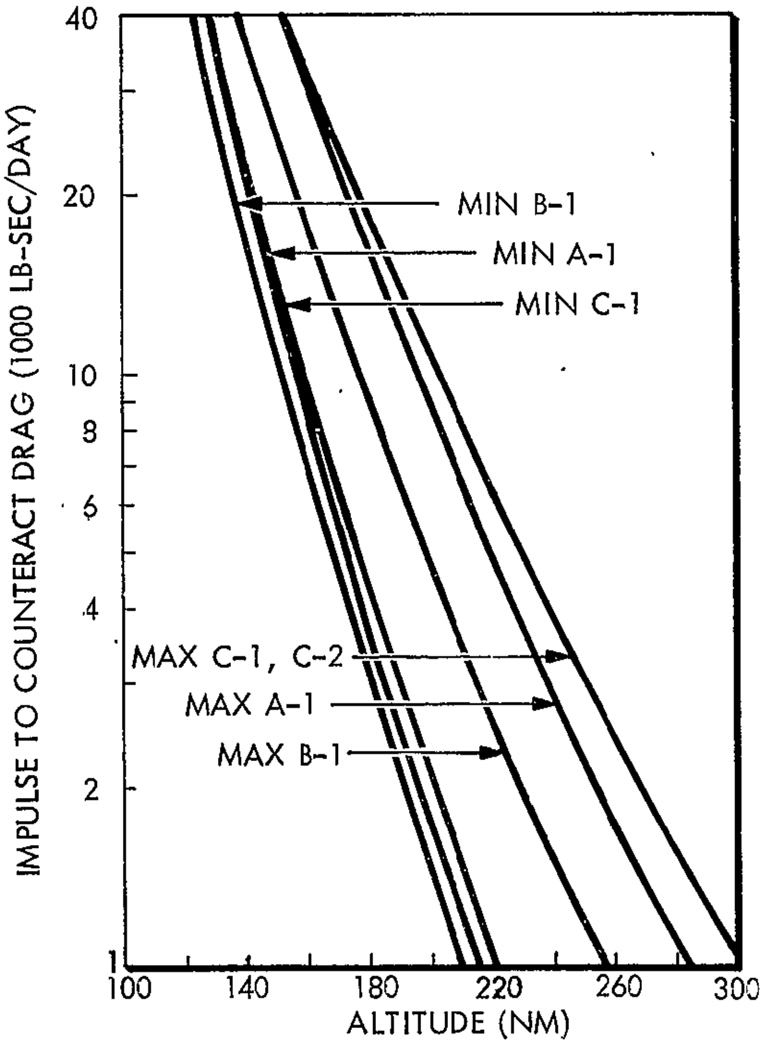
- Orbit decay rates and atmospheric drag were calculated for the minimum and maximum atmospheric densities to be expected during each satellite configuration's lifetime. Typical average ballistic and drag coefficients were assumed.

- The number of days required for the orbit to decay 20 nm due to atmospheric drag, and the impulse required to counteract the atmospheric drag, were computed as a function of orbit altitude for each satellite configuration.

- The 1st flight vehicle, in dormant mode oriented with the edge of the arrays into the wind and flying at an altitude of 240 nm, in 1983 will take between 480 and 1460 days to decay 20 nm in altitude. Due to scale differences, this calculation is not represented on the graph.

ALTITUDE EFFECTS ON AERO DRAG FOR 25 kW PM





NOTE: MIN AND MAX REFER TO THE EXTREMES IN THE ATMOSPHERIC DENSITY MODEL OVER A SATELLITE CONFIGURATION'S LIFETIME.

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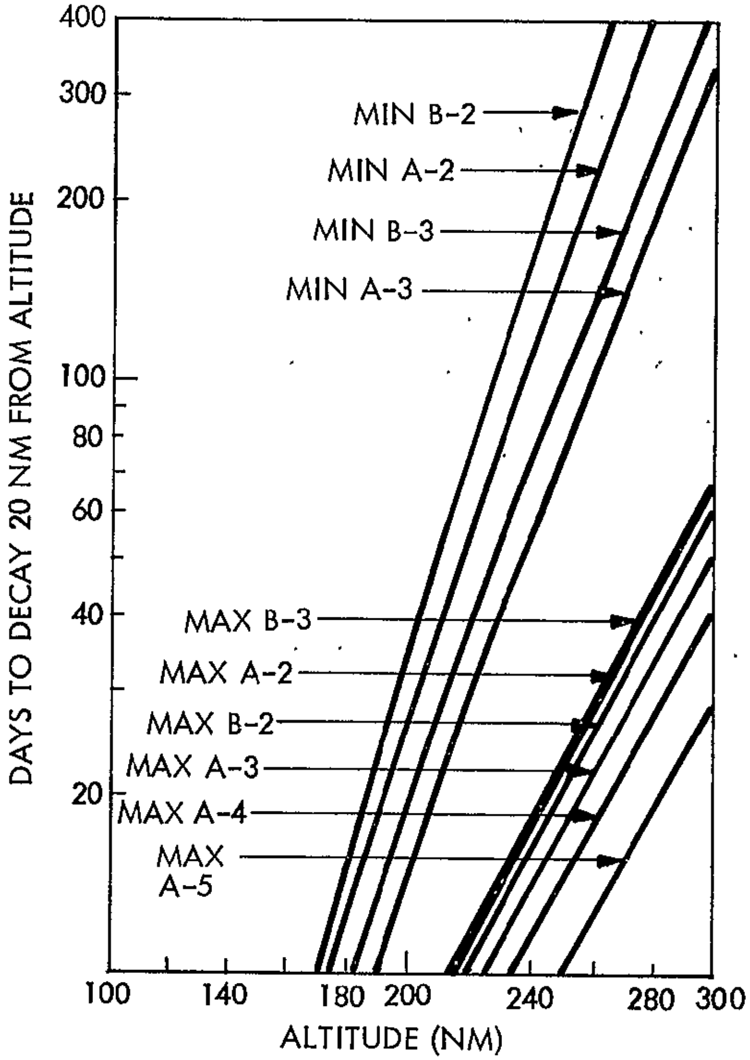
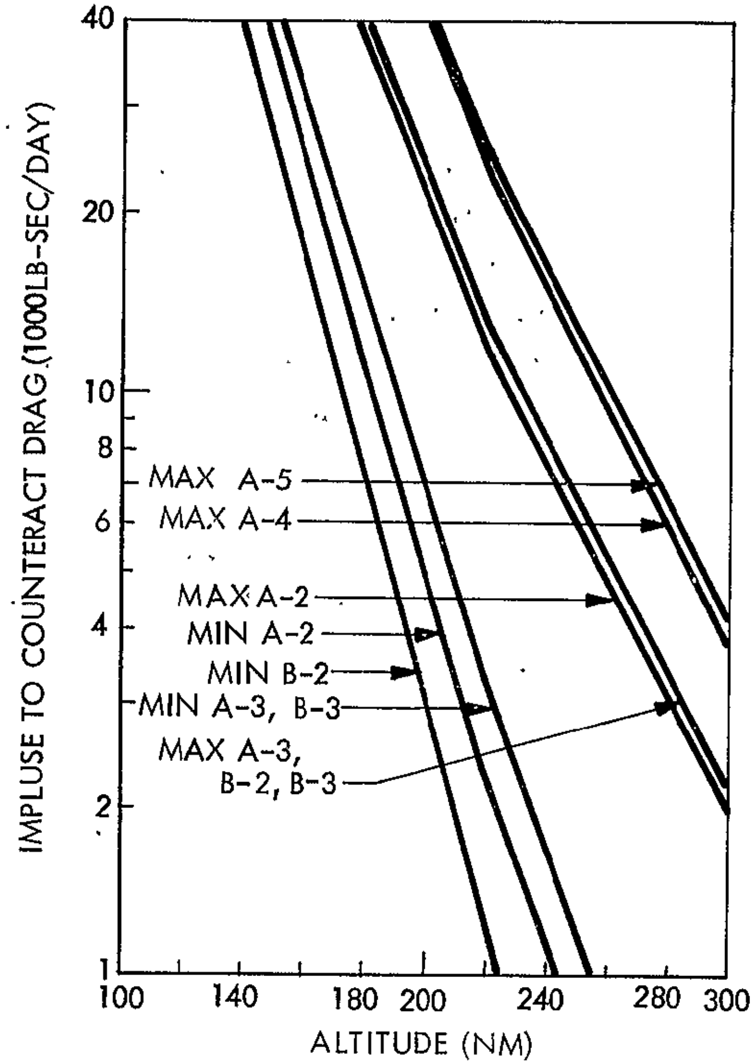
ALTITUDE EFFECTS ON AERO DRAG FOR 50 AND 100 kW





POWER MODULE REBOOST SUMMARY

- The results of a reboost analysis indicate that the first vehicle will require a revisit of the Orbiter at approximately 67 days initially (1983). As the flight continues into 1984 and 1985, revisit-time will continuously increase to a maximum of 270 days. The other vehicle configurations would require Orbiter revisit (for a 20 nm decay) as often as every 9 days during solar maximum periods.

- The use of the Teleoperator (TRS) will result in increasing the Orbiter revisit-interval to 90 days or more for all vehicles evaluated. Based on current predicted capabilities of the TRS thrusters, vehicles A-4 and A-5 would initially require refurbishment (replacement) of the TRS at Orbiter revisit. As the flights of vehicles A-4 and A-5 proceeded towards solar minimum, TRS replacement time would increase. All other vehicles evaluated would have longer periods before TRS replacement is required.

POWER MODULE REBOOST SUMMARY

CONFIG-URATION	OPERATING ALTITUDE (NM)	OPERATING PERIOD	20 NM DECAY (DAYS)	REBOOST PROPULSION SYSTEM	REBOOST PROPELLANT (1)(2)(4) (LBM/90 DAYS)	MAX. ACCEL (G)	NO. OF THRUSTERS FIRING
B-1	240	1983 → 85	67 → 270	TRS	595	0.0032	8
C-1	240	1988 → 89	168 → 27	TRS	1,566	0.0032	8
A-1	240	1986 → 87	226 → 35	TRS	1,193	0.0029	8
A-2	240	1988 → 89	112 → 17	TRS	3,013	0.0033	12
A-3	240	1989 →	56 → 14	TRS	3,330	0.0047	12
A-4	245	1990 →	13 → 119(3)	TRS	5,630	0.0075	32(5)
A-5	240	1991 →	9 → 83(3)	TRS	5,828	0.0113	32(5)
B-2	240	1986 → 88	169 → 16	TRS	3,330	0.0040	12
B-3	240	1989 →	76 → 18	TRS	3,330	0.0035	12
C-2	240	1990 →	32 → 329(3)	TRS	1,566	0.0027	8

(1) MAXIMUM 90-DAY USAGE OVER VEHICLE OPERATING PERIOD
(2) TRS MAX. PROPELLANT IS 6,000 LBM
(3) BASED ON MISSION FLYING INTO SOLAR MINIMUM
(4) TRS SPECIFIC IMPULSE OF 227 SECONDS
(5) REQUIRES TRS REFURBISHMENT

TELEOPERATOR RE-BOOST

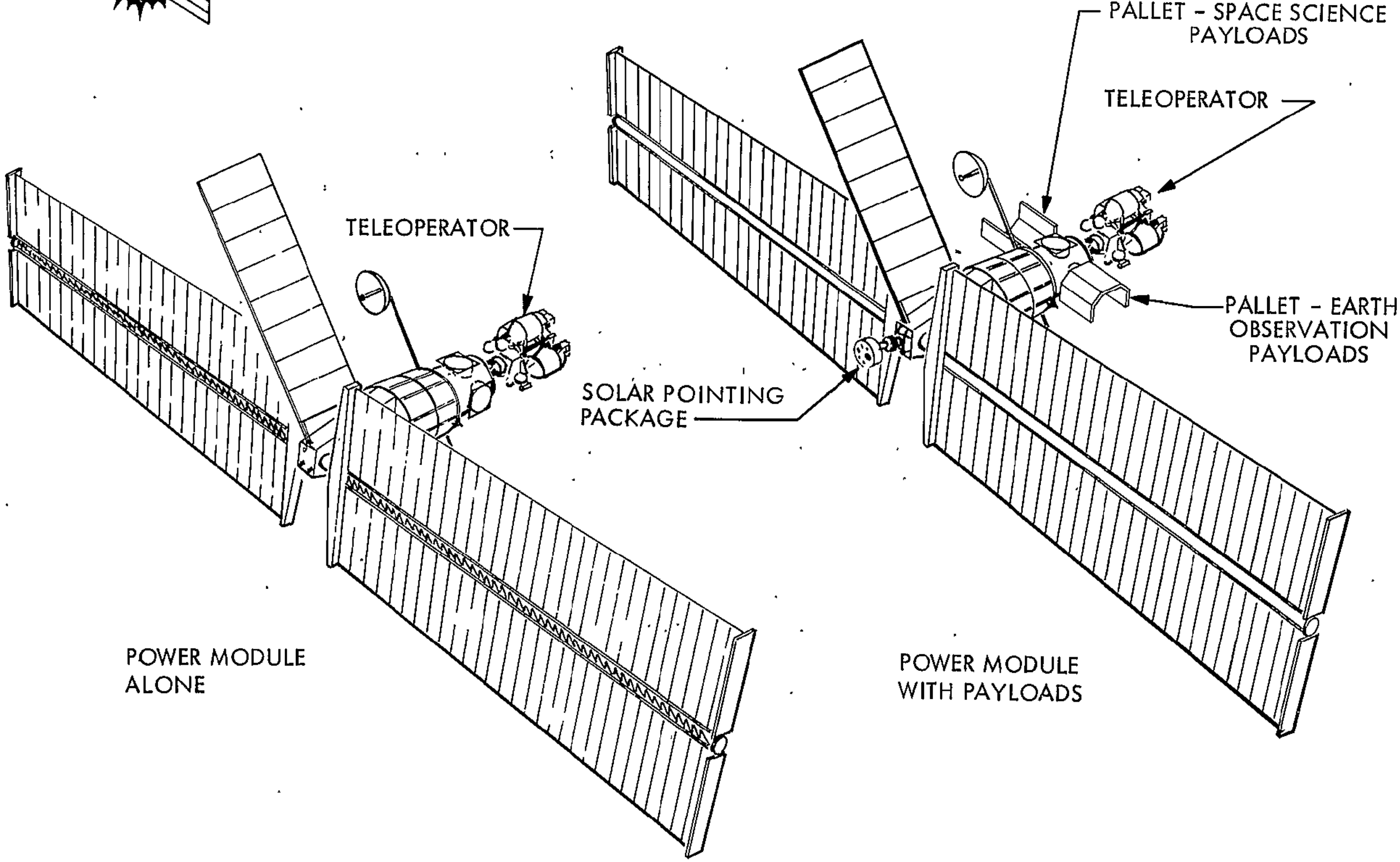
- When Teleoperator (TRS) is used as an attached aero-drag makeup propulsion system, a decreased frequency of Orbiter revisits is achieved.

- The use of the TRS as a reboost stage results in a flexible stage (propellant load up to 6000 lbm) with low accelerations imposed on the Power Module/ payloads (0.01g or less). This provides great latitude in Orbiter revisit time.

- Reboost by the Orbiter to relatively low-altitude orbits is feasible, but probably not cost-effective. Future studies should evaluate trade options between use of Orbiter, a separate propulsive vehicle (such as Teleoperator), and an on-board propulsion subsystem for Power Module reboost and orbit adjust.

C-2

TELEOPERATOR RE-BOOST



2.8 GROWTH KIT CONCEPTS

POWER MODULE GROWTH KIT
CANDIDATE SSE CONFIGURATIONS

- The chart illustrates three candidate space support equipment configurations for assembly of Power Module kits and providing for their support in the Orbiter payload compartment.

- Configuration 1: A standard NASA pallet, or pallets. If this is used, it would be classed as "System Support Elements", since it derives from Spacelab hardware.

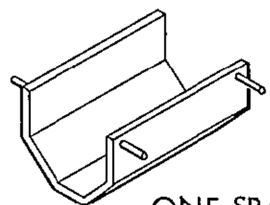
- Configuration 2: A special pallet. The 25 to 50 kW kit and the 50 to 100 kW kit require a separate pallet for each kit.

- Configuration 3: Support truss concept. Consists of a lightweight system of trusses fitted with sill trunnion and keel fittings. Utilizes the solar array beams that are being supported as load carrying members between the trusses, for distributing the loads into the Orbiter payload bay structure. The Power Module solar array beams and support structure are attached to the support trusses by manually operated latches shown on Page 2-93 of this volume. The forward and aft trusses are used with both the 25 to 50 kW kit and the 50 to 100 kW kit.

POWER MODULE GROWTH KIT CANDIDATE SSE CONFIGURATIONS

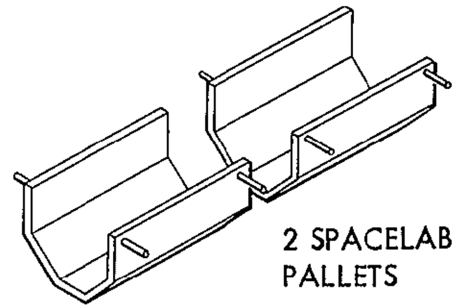
25 TO 50 kW	50 TO 100 kW

(1)



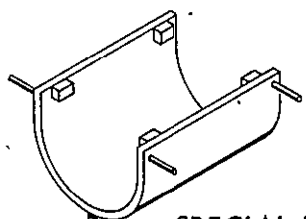
ONE SPACELAB PALLET

(1)



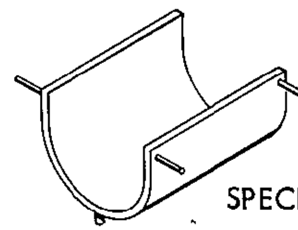
2 SPACELAB PALLETS

(2)



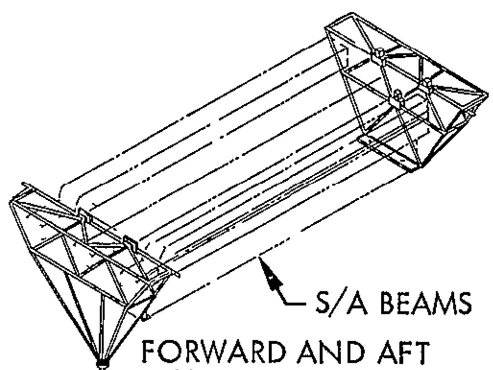
SPECIAL PALLET AND BRACKETS

(2)



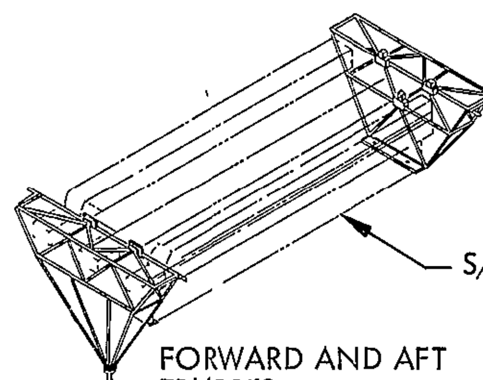
SPECIAL PALLET

(3)



FORWARD AND AFT TRUSSES

(3)



S/A BEAMS

FORWARD AND AFT TRUSSES

GROWTH KIT SSE SELECTION

- The chart depicts the advantages and disadvantages of alternative kit configurations. A kit configuration is selected and the rationale depicted.

- The table is self explanatory.

GROWTH KIT SSE SELECTION

POWER MODULE CONVERSION	ALTERNATE KIT CONFIGURATIONS*	ADVANTAGES	DISADVANTAGES	SELECTION AND RATIONALE
25 TO 50 kW	(1) SPACELAB PALLET	• STANDARD STRUCTURE • COMPATIBLE WITH ORBITER • HARD POINTS TO CARRY LOAD	• WEIGHT PENALTY • COSTLY • NOT COMPATIBLE WITH PAYLOAD	SELECTION: FWD AND AFT TRUSS • SIMPLICITY OF DESIGN AND ANALYSIS • EASE OF HANDLING • LIMITED SPACE IN ORBITER ON ASCENT AND DESCENT • PAYLOAD COMPONENTS IN ORBITER REMOVED INDIVIDUALLY
	(2) SPECIAL PALLET	• COMPATIBLE WITH PAYLOAD • TRANSMITS LOADS THROUGH PALLET	• WEIGHT PENALTY • NEW DESIGN • EXTENSIVE LOAD ANALYSIS	
	(3) FWD & AFT TRUSS	• COMPATIBLE WITH PAYLOAD • SIMPLICITY OF STRUCTURE • LIGHT WEIGHT	• LOADS TRANSMITTED THROUGH PAYLOAD BEAMS • DEVELOP LATCH MECHANISM	
50 TO 100 kW	(1) SPACELAB PALLET	• STANDARD STRUCTURE	• WEIGHT PENALTY • COSTLY • NOT COMPATIBLE WITH PAYLOAD	SELECTION: FWD AND AFT TRUSS • VOLUME AND WEIGHT, KIT TO PAYLOAD IS MINIMUMIZED • LIGHT WEIGHT • SIMPLICITY OF STRUCTURE • PAYLOAD UTILIZES TOTAL CROSS SECTION OF ORBITER
	(2) SPECIAL PALLET	• COMPATIBLE WITH PAYLOAD • TRANSMITS LOADS THROUGH PALLET	• WEIGHT PENALTY • STUDY REQUIRED TO SELECT AN EFFICIENT STRUCTURE	
	(3) FWD & AFT TRUSS	• COMPATIBLE WITH PAYLOAD • SIMPLICITY OF STRUCTURE • EASE OF HANDLING	• LOADS TRANSMITTED THROUGH PAYLOAD BEAMS • ASSEMBLY FIXTURE REQUIRED TO INTERFACE TRUSS/PAYLOAD	

*SEE PREVIOUS CHART

ORBITER PAYLOAD SUPPORT STRUCTURE
50 TO 100 kW KIT

- The chart illustrates packaging, in the Orbiter payload compartment, of the support structure required to support a growth kit for changing a 50 kW to 100 kW Power Module. To secure this kit in the Orbiter Bay compartment, they are supported by a forward and aft truss that attaches to the Orbiter trunnions and keel. The kit and support structure are described below.

- The structural components of the kit consist of a solar array support structure and two solar array beams. One of the two beams has solar arrays integral with it.

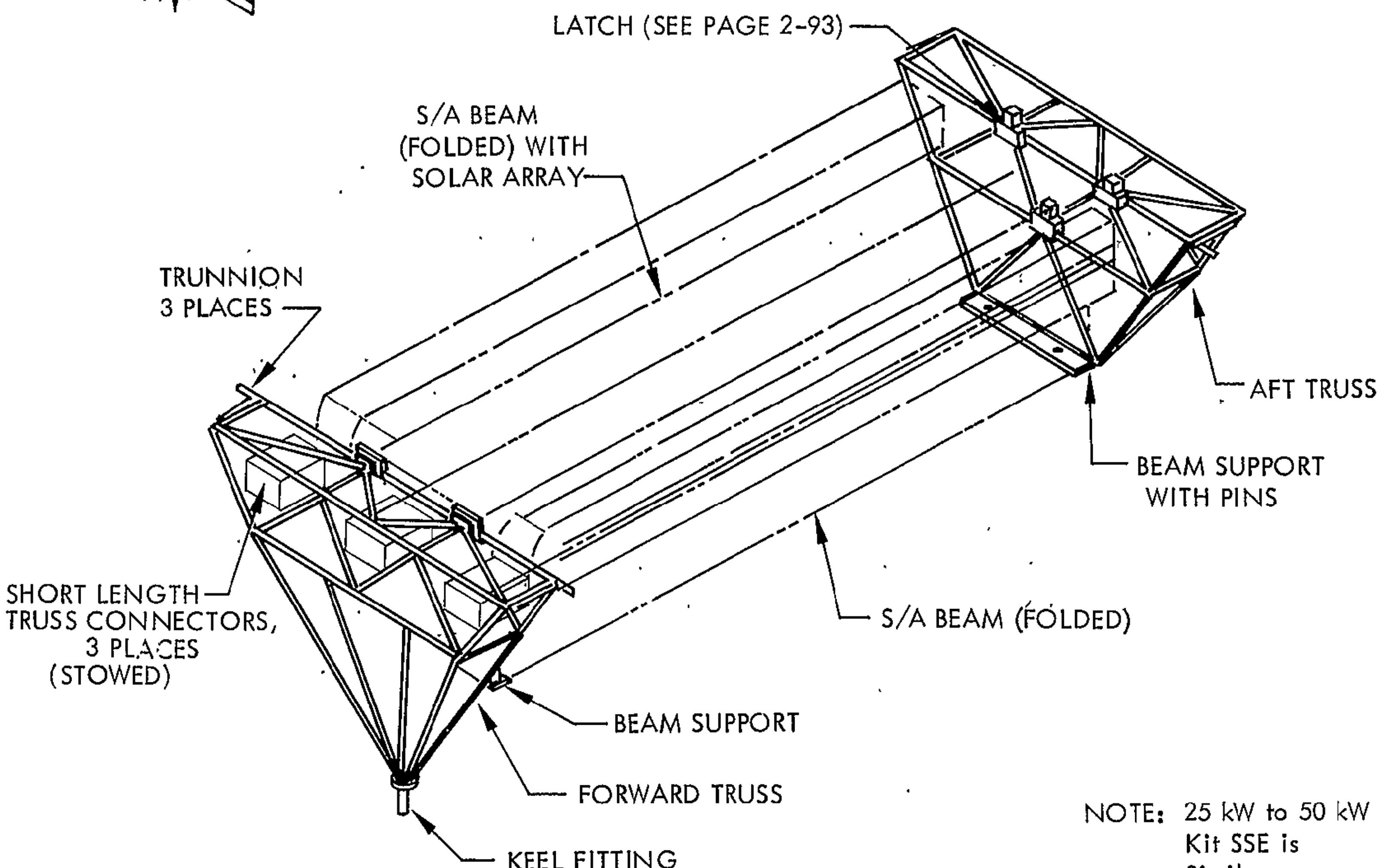
- The support structure consists of forward and aft trusses that, in conjunction with the two solar array beams, form an integral structure. The trusses are fastened to the beams by latches. The ends of the beams are supported and indexed by brackets and pins on the end trusses. The solar array support structure is fastened to the beams with pins and latches

- The latches provide the structural interface between beams and trusses. The latches are described and illustrated on pages 2-92 and 2-93.

- After removal on orbit of the solar array beams, both forward and aft trusses are prepared for return to earth as an integral structure by interconnecting them with the short length truss connectors shown stowed in the forward truss.

- The support structure for the 25 kW Kit is similar to that illustrated and described for the 50 kW to 100 kW Kit.

ORBITER PAYLOAD SUPPORT STRUCTURE
50kW TO 100kW KIT



NOTE: 25 kW to 50 kW
Kit SSE is
Similar

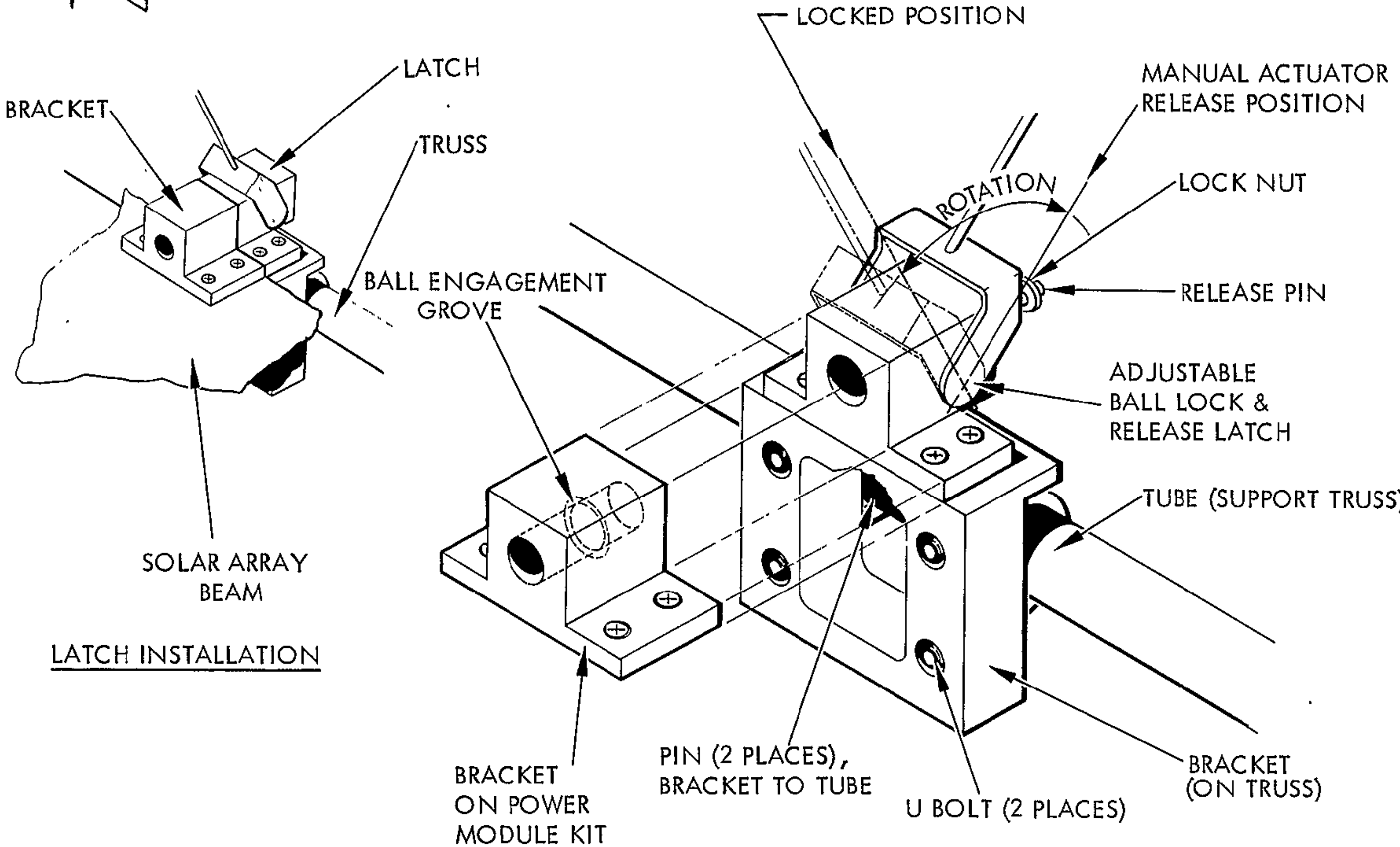
GROWTH KIT LATCH CONCEPT

- This chart illustrates a latch concept that provides a positive connection between the Power Module kit and truss supports. The latch is also used for positive connections between Kit components and for fastening the keel truss to the 100 kW Power Module S/A beam.

- The concept shown illustrates a ball lock mechanism. Other locking devices such as chucks and tongs would be acceptable.

- Manual operation of the latch is shown for attaching/releasing the kit components. Electrical operation of a latching concept is feasible but it would be quite costly.

GROWTH KIT LATCH CONCEPT



ORBITER PAYLOAD SUPPORT STRUCTURE
100 kW POWER MODULE

- The chart illustrates, in the Orbiter payload compartment, the support equipment required to package the solar array components for the 100 kW Power Module. To secure these components in the Orbiter bay compartment, they are supported by forward and aft trusses and a keel truss that attaches to the Orbiter trunnions and keel. The equipment and support structure are described below.

- Two solar array beams are provided; each beam contains a solar array assembly. A solar array support structure with radiators is provided and carried on top of the two solar array beams.

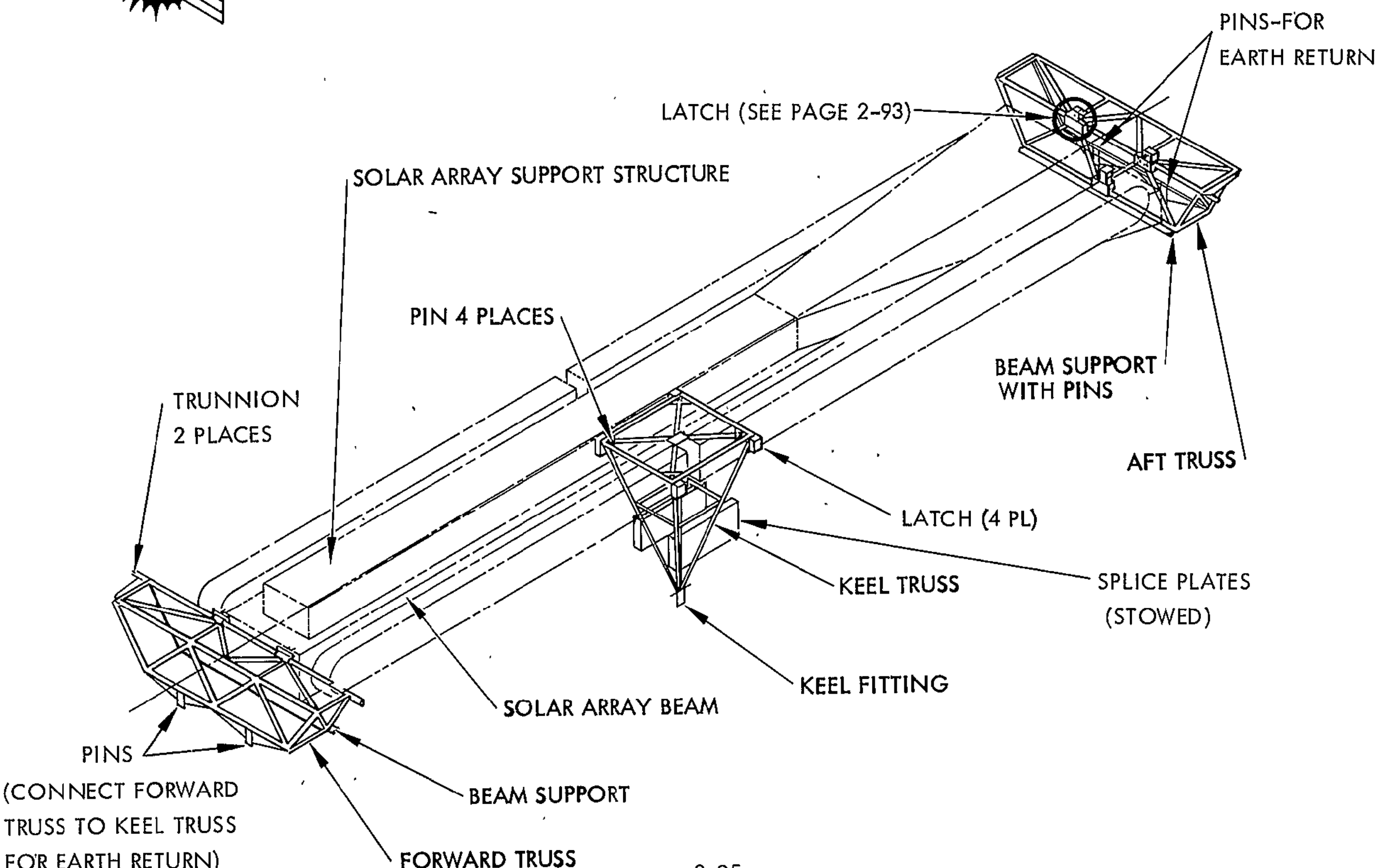
- The support structure consists of forward and aft trusses and a keel truss; in conjunction with the two solar array beams these form an integral structure. The truss structures are fastened to the beams by latches. The end beams are supported on the end trusses by brackets and pins. The solar array support structure is fastened to the beams with pins and latches.

- The latches provide the structural interface between the beams and trusses. The latches are described and illustrated on pages 2-92 and 2-93 of this volume.

- For return of support trusses to earth after removal of solar array beams:
  The forward and AFT trusses are interconnected by splice plates (shown stowed on the keel truss in the chart), and pinned to the existing latches on the keel truss – forming an integral structure for "Tying-In" to the trunnion/keel fitting system of Orbiter payload bay attachments.

- The equipment rack and the berthing assembly are transported as a unit in a second Orbiter. No special support structure is required for this Orbiter package.

ORBITER PAYLOAD SUPPORT STRUCTURE 100kW POWER MODULE



POWER MODULE INSTALLATIONS IN ORBITER PAYLOAD BAY

- Preliminary layouts of Power Module installations in the Orbiter payload bay have been sketched. Such layouts for the selected configurations are included in the Volume 1 of the Part III study.

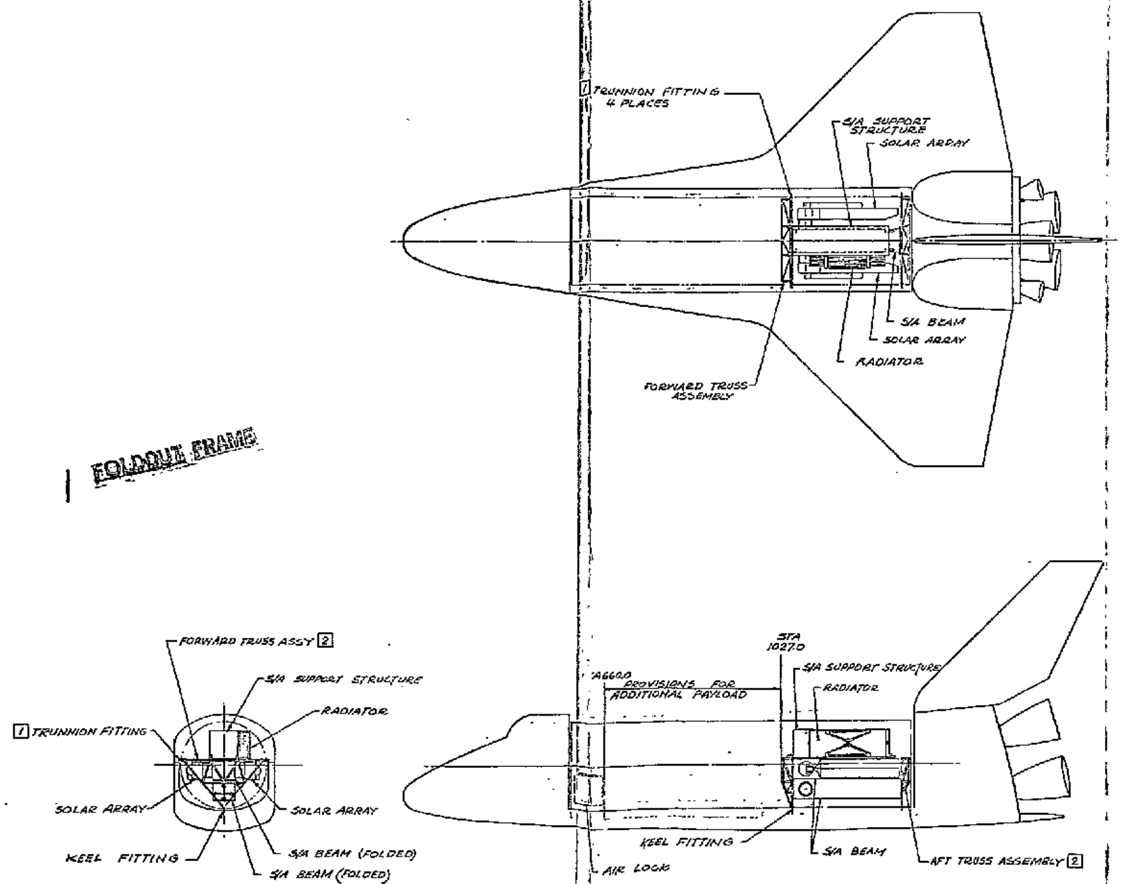
- The two layouts which follow reflect installations of optional Power Module configurations also considered in the study:

  DWG No. 6164064 Power Module Orbital Conversion Kit; Configuration 25-1 to 50-2.

  DWG No. 6164065 50 kW Power Module Configuration 50-2; Orbiter Payload Bay Installation.

ZONE	LTR	REVISIONS DESCRIPTION	DATE	APVD
	A	REVISED & REDRAWN		

FOLDOUT FRAME



2 FOLDOUT FRAME

NOT RELEASED

NOTES:

1. ATTACH TRUNNION FITTING (NON DEPLOYABLE PAYLOAD) PER SHUTTLE ORBITER/CARGO STANDARD INTERFACES I.D.C NO 2-19001, FIGURE 3.3.1.2.1-1 BY NASA- JOHNSON SPACE CENTRE.
2. FORWARD AND AFT TRUSS ASSEMBLIES ARE REUSABLE SHUTTLE/PAYLOAD TRANSPORT EQUIPMENT.

QTY REQD	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	MATERIAL DESCRIPTION OR NOTE	MATERIAL SPECIFICATION	ZONE	ITEM NO
PARTS LIST							

INTERPRET DWG PER

NEXT ASSY | USED ON — APPLICATION

UNLESS OTHERWISE SPECIFIED DIM ARE IN INCHES TOLERANCES ON FRACTIONS = ± 1/16 DECIMALS .X = ± .1 .XX = ± .03 .XXX = ± .010 ANGLES = ± 2 DEG

CONTR

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APVD

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ENGRG

CHKR

APVD

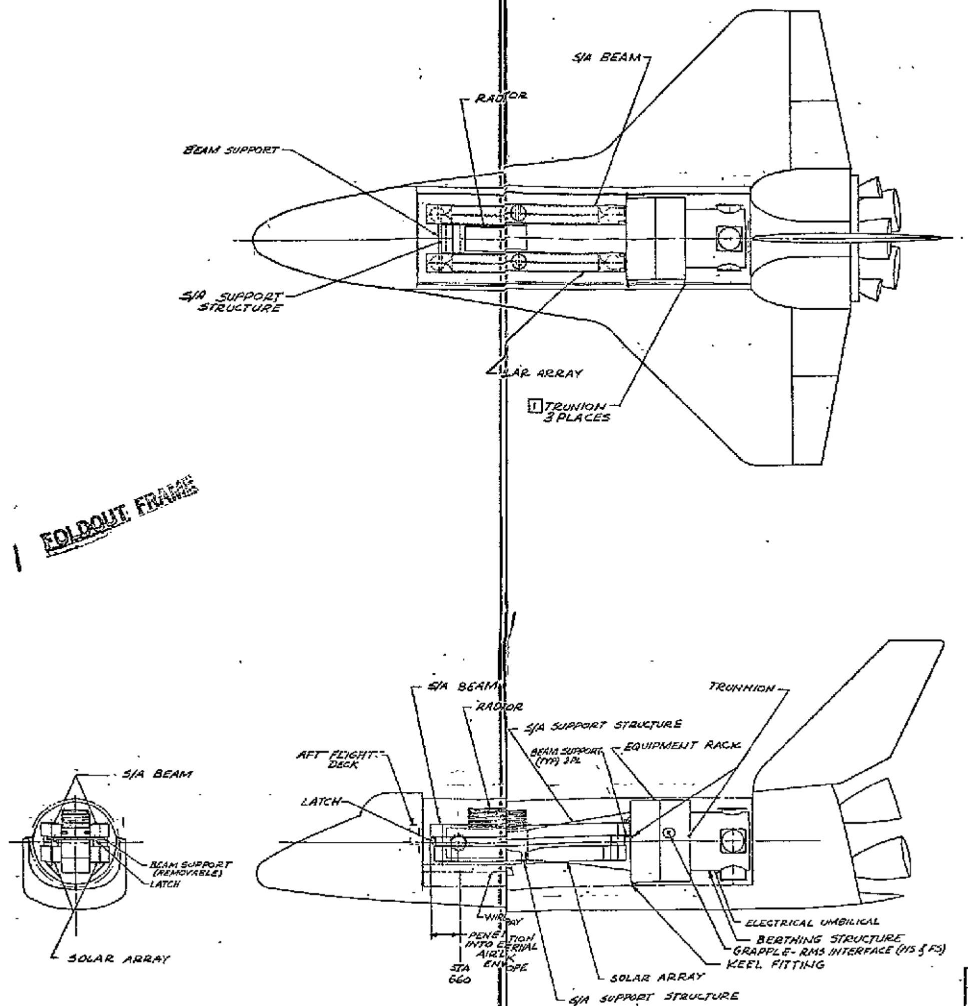
APVD

LOCKHEED MISSILES & SPACE COMPANY INC
A SUBSIDIARY OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

POWER MODULE
ORBITAL CONVERSION KIT
CONFIGURATION 25-1 TO 50-2

SIZE	CODE IDENT	DRAWING NO	REV
E		6164064	A

SCALE | SHEET 1 OF 1



FOLDOUT FRAME 1

FOLDOUT FRAME 2

NOTES:
1. ATTACH TRUNNION FITTING (NON DEPLOYABLE PAYLOAD) PER SHUTTLE ORBITER/CARGO STANDARD INTERFACES I.D.C NO 2-19001, FIGURE 3.3.1.2.1-1 BY NASA- JOHNSON SPACE CENTRE.

NOT RELEASED

QTY REQD	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	MATERIAL DESCRIPTION OR NOTE	MATERIAL SPECIFICATION	ZONE	ITEM NO

PARTS LIST

INTERPRET DWG PER	UNLESS OTHERWISE SPECIFIED DIM ARE IN INCHES, TOLERANCES ON: FRACTIONS = ± 1/16; DECIMALS .X = ± 1, .XX = ± .03, .XXX = ± .010; ANGLES = ± 2 DEG	DATE 78 NOV 6 DWN W Steele APVD APVD ENGRG CHKR APVD APVD	LOCKHEED MISSILES & SPACE COMPANY INC. A SUBSIDIARY OF LOCKHEED AIRCRAFT CORPORATION SUNNYVALE, CALIFORNIA
NEXT ASSY / USED ON — APPLICATION	CONTR QCA/CI/FD		50 KW POWER MODULE CONFIGURATION 50-2 ORBITER PAYLOAD BAY INSTAL. SIZE E · CODE IDENT · DRAWING NO 6164065 · REV - SCALE 1/96 · SHEET 1 OF 1

SECTION 3 SUBSYSTEM ANALYSES

3.1 STRUCTURES

25 KW POWER MODULE STRUCTURE

- The Power Module structure consists of the equipment rack, berthing, and solar array support structures as illustrated on the chart.

- The equipment rack consists of two equipment sections from the Space Telescope program. Some minor modifications will be required to adapt the structure for the 25 kW Power Module.

- The berthing structure is of semi-monocoque construction. It has provisions for berthing with the Orbiter/payloads at five positions. One position is for sortie and four positions are for payloads and/or maintenance. In addition, there are provisions for attitude control equipment installations in this structure.

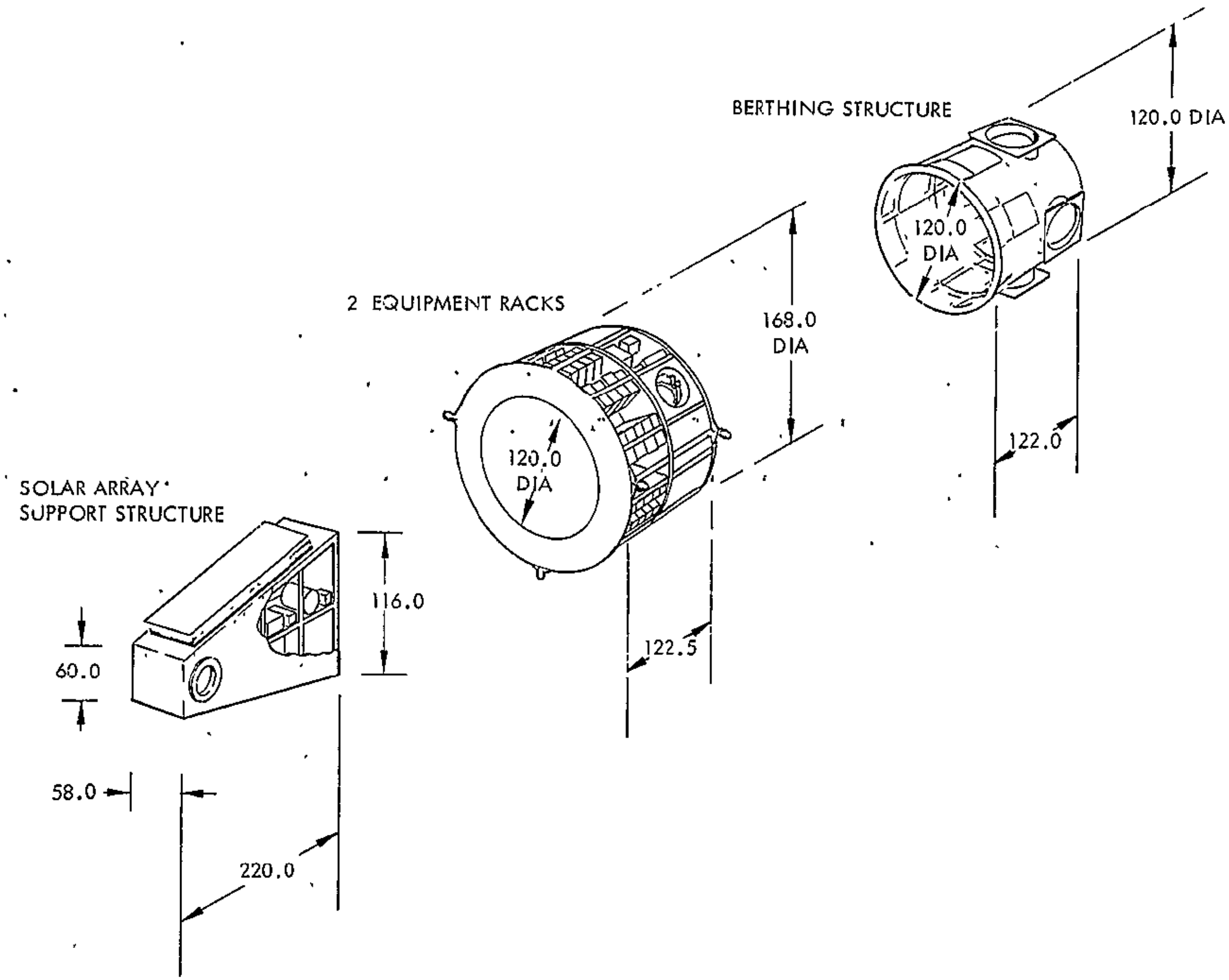
- The solar array support structure is a conventional configuration of structural members and shear panels. It has provisions for the solar array structural/mechanical interface and for thermal control equipment installations. Growth kit concepts, which include additions to solar array support structure required for growth to 50 kW and 100 kW configurations, are discussed in Section 2.8.

25 kW POWER MODULE STRUCTURE

SOLAR ARRAY SUPPORT STRUCTURE
- SHEAR BOX STRUCTURE
- PROVIDES FOR GROWTH
- READILY ACCOMMODATES EQUIPMENT
- APPLICABLE TO 25, 50, AND 100 kW STRUCTURE

EQUIPMENT RACK
- DESIGN AND TOOLING AVAILABLE
- ORBITER ENVIRONMENT QUALIFIED
- SOME REDESIGN REQUIRED
- PROVIDES FOR GROWTH
- SPACE FOR GROWTH EQUIPMENT
- APPLICABLE TO 25, 50, AND 100kW STRUCTURE

BERTHING STRUCTURE
- MONOCOQUE STRUCTURE
- PROVIDES FOR EQUIPMENT GROWTH
- PROVIDES BERTHING AND SORTIE INTERFACE
- APPLICABLE TO 25, 50, AND 100 kW STRUCTURE



DIMENSIONS IN INCHES

MODULAR GROWTH

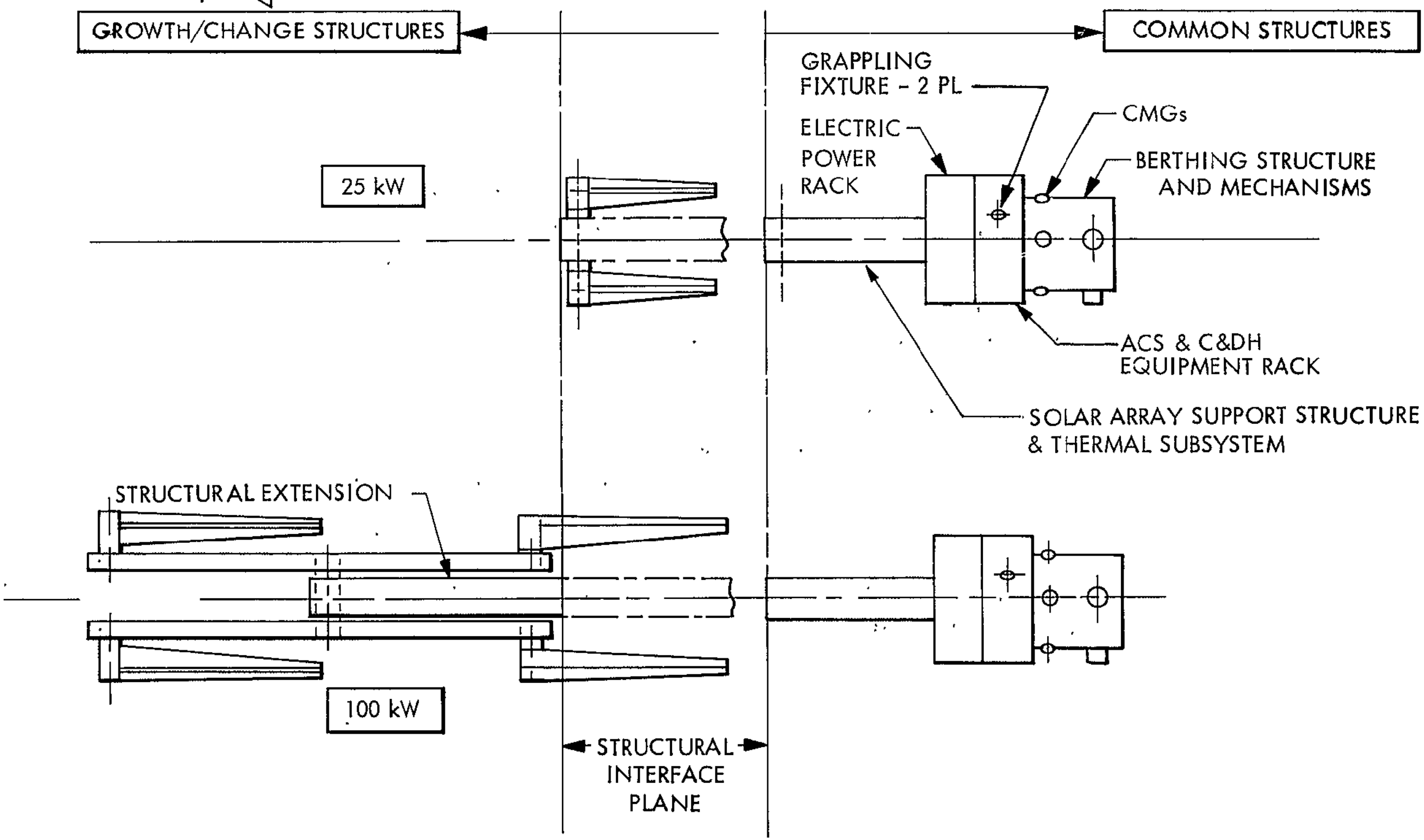
- The right side of the chart displays the structural assemblies which are recommended to be "common" for the 25, 50 and 100 kW Power Modules. Add-on solar array/structural extensions are illustrated on the left side.

- The key factor which supports the feasibility of the "common-structure" growth concept is the utilization of battery technology advances, compatible with growth scenario requirements, which dictates use of the same-size battery installations for all three power module sizes.

- By designing the common structure for the most adverse load conditions for the three power modules, a relatively small weight penalty is incurred in the smallest vehicle. Total vehicle weight changes +14 percent between the 25 and 50 kW power modules, and +20 percent between the 50 and 100 kW vehicles.

MODULAR GROWTH

COMMON ELEMENTS THROUGH 100 kW SIZE



CONFIGURATION 25-1 SHEARS & BENDING MOMENTS

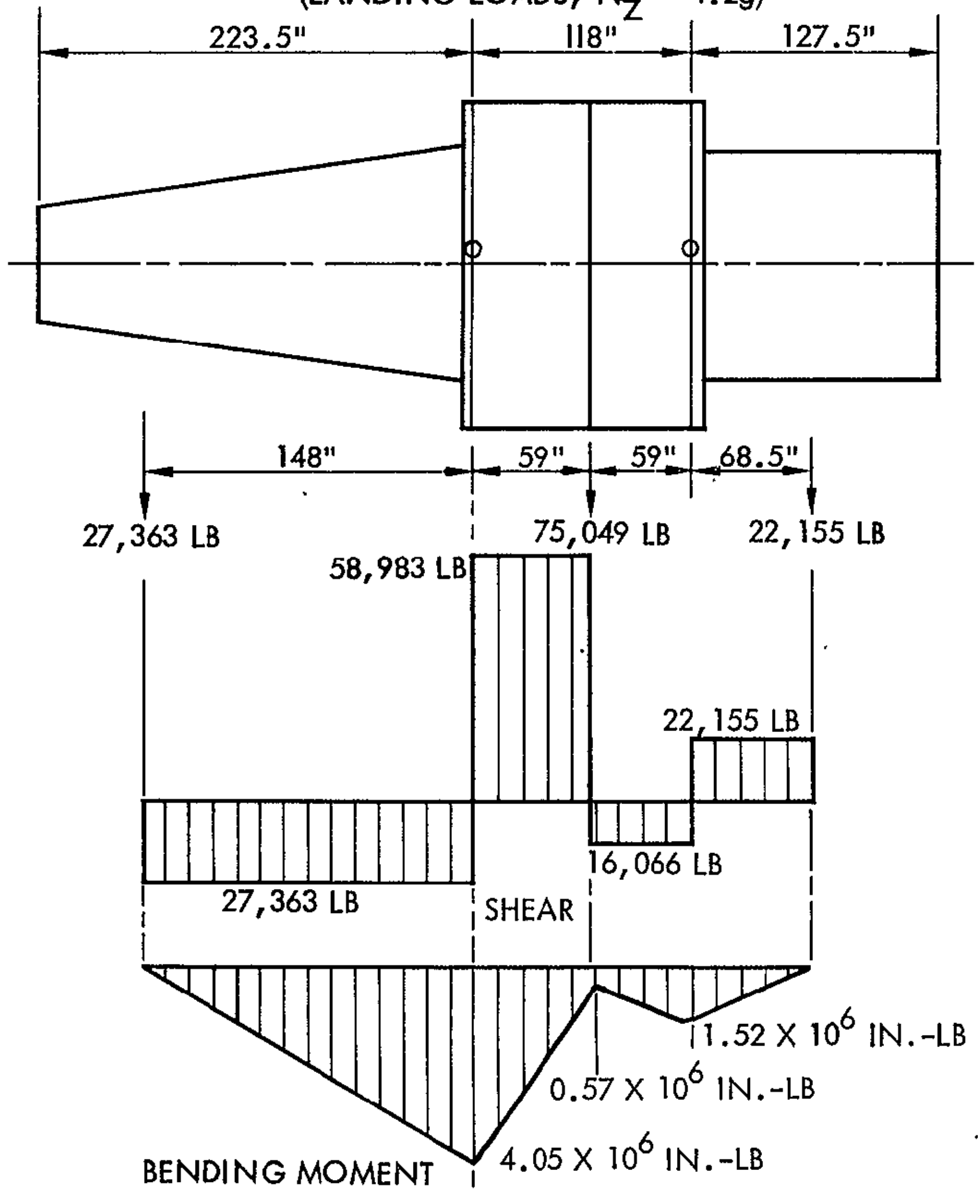
- The evaluation of the Space Shuttle attachment structure of both the 25 kW Power Module and the 50 kW Power Module was accomplished utilizing computerized techniques.

- The most severe design conditions were investigated, involving eight mission phases. These are lift-off, high "q" maximum boost, orbit alone, pitch maneuvers, entry and descent, yaw maneuvers and landing.

- The critical shear and bending moment diagrams shown in the next two charts for the 25 kW Power Module and the 50 kW Power Module are N_z loading associated with 4.2 g landing loading only, and are intended to depict the load distribution to the primary trunnion system and the stabilizing trunnion system for this loading condition.

CONFIGURATION 25-1
SHEARS AND BENDING MOMENTS

(LANDING LOADS, N_Z = 4.2g)

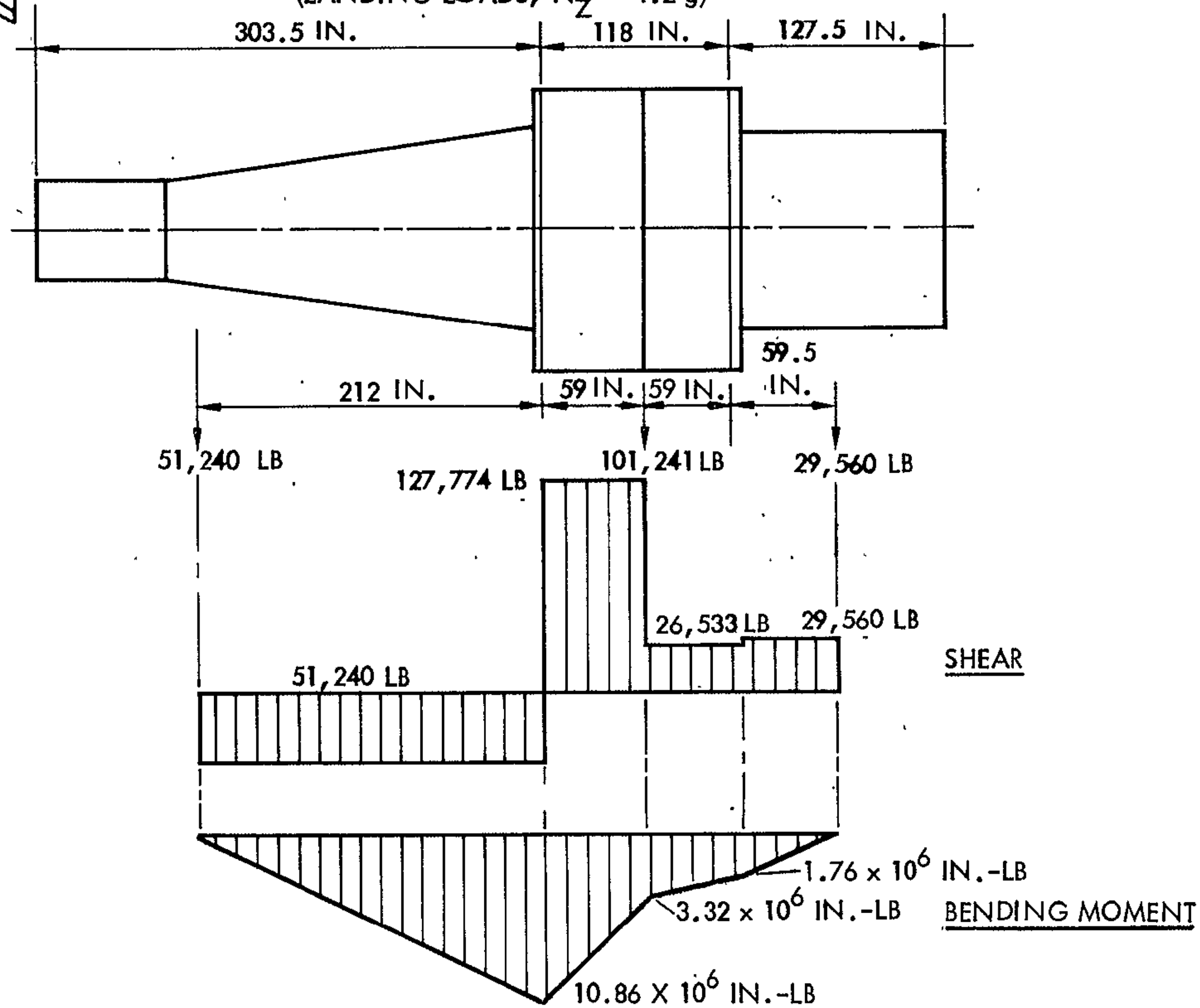


CONFIGURATION 50-1 SHEARS & BENDING MOMENTS

Refer to discussion with preceding chart

CONFIGURATION 50-1
SHEARS AND BENDING MOMENTS

(LANDING LOADS, $N_Z = 4.2$ g)



STRUCTURES SUBSYSTEM TRADES: SUMMARY

- The chart summarizes the primary structure design trades that have been performed. Other related trades were performed for the design of the solar array assembly. Most of these trades were performed and reported in Part II (Ref. 31).
- Design compatability with Orbiter environments, geometry, and loads was a key factor favoring use of the Space Telescope SSM structural hardware. Like Space Telescope, the high safety-factor/ minimal structural testing criteria option was selected in recognition of the criticality of early-year budgets.
- It should be noted that for each major assembly the selected candidate design continues without significant modification into the growth configuration. For example, in the case of the solar array support system, the basic 25 kW structure remains unchanged: only add-on structural extensions are needed for growth.
- For the berthing structure, the recommended 25 kW structure will be designed to provide for optional installation of a maximum of six CMGs.

STRUCTURES SUBSYSTEM TRADES—SUMMARY

ELEMENT OR COMPONENT	NUMBER OF CANDIDATES	CONFIGURATION SELECTED INITIAL 25 kW	CONFIGURATION SELECTED GROWTH	SELECTION RATIONALE
SAFETY FACTOR/STATIC TEST CRITERIA	3 (REF PART II REPORT REF 31)	2.0 YIELD AND 3.0 ULTIMATE/ NO TEST	SAME	• MIN COST DEVELOPMENT PROGRAM • MODULAR STRUCTURAL GROWTH CONCEPT REDUCES WEIGHT CRITICALITY
SOLAR ARRAY SUPPORT STRUCTURE	2 (REF PART II REPORT)	SEMI-MONOCOQUE	SAME PLUS ADD-ON STRUCTURAL EXTENSIONS	• GROWTH POTENTIAL: EQUIPMENT & EXPERMENT INSTALLATION EASE • GROWTH PROVISIONS FOR "ADD-ONS" • ENVIRONMENT PROTECTION
EQUIPMENT RACK STRUCTURE	3 (REF PART II REPORT)	SSM DERIVED HARDWARE	SAME	• COST EFFECTIVENESS • COMPATIBILITY WITH ORBITER SYSTEMS
BERTHING STRUCTURE	2 (REF PART II REPORT)	SEMI-MONOCOQUE	SAME	• COST-EFFECTIVE GROWTH POTENTIAL
BERTHING SYSTEM STRUCTURE	6 (REF PART III VOL I REPORT	BERTHING LATCHES WITH ELEVATING TABLE	SAME	• COST EFFECTIVENESS, SIMPLICITY, RELIABILITY, & COMPACTNESS • NON-INTERFERENCE WITH PAYLOAD BAY AFT OF STA 660 • INCLUDES MAINTENANCE PLATFORM SYSTEM

3.2 ELECTRICAL POWER

LEO POWER SYSTEM EVOLUTIONARY PATHS

- The utilization of space will require low cost, reliable energy generation and storage methods. The accompanying chart indicates the projection of power system technology expected to be available for Power Module use. The component technology advancements represented will require astute scheduling into the Power Module Evolution.

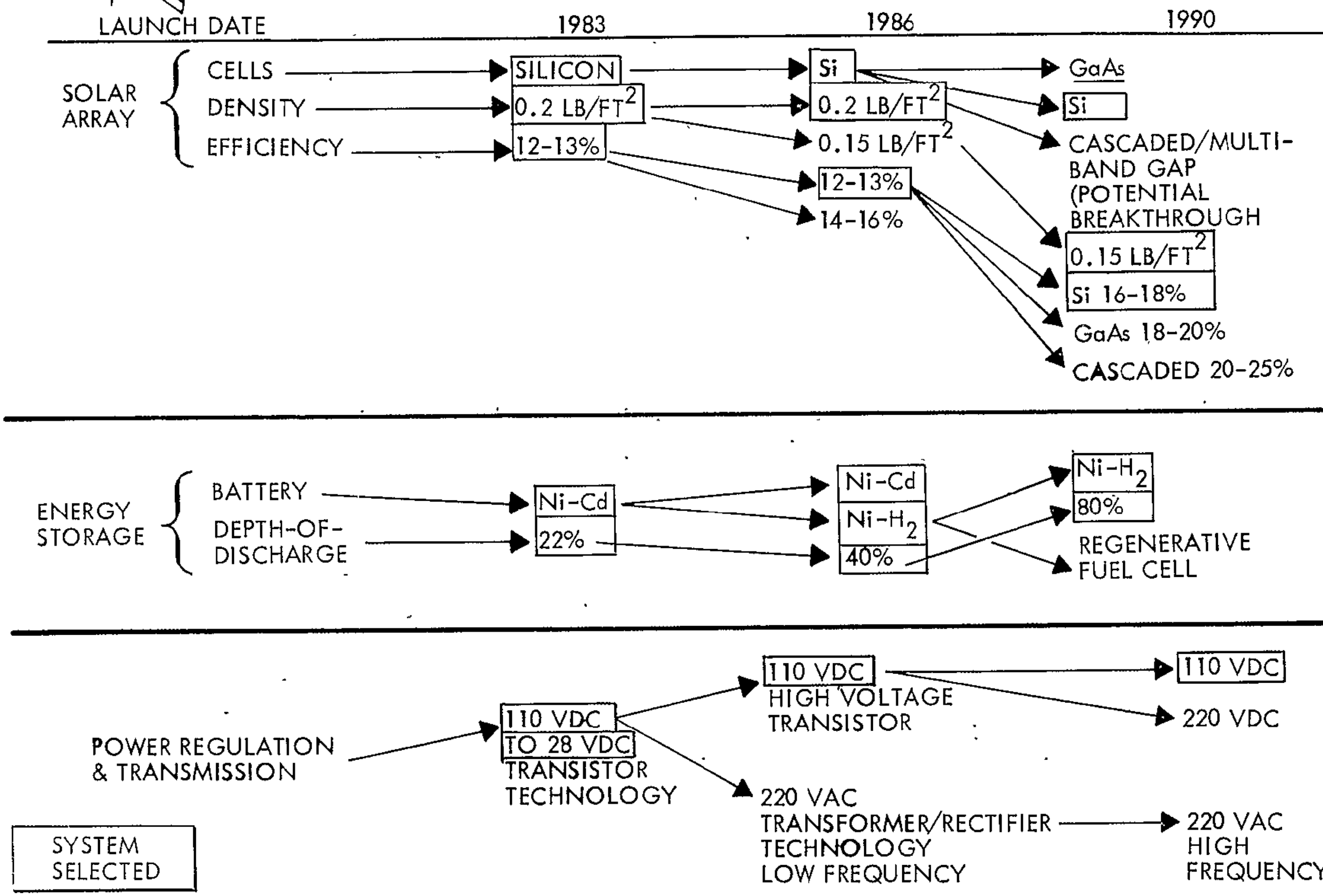
- The advanced components are expected to save substantial dollars by two methods: (1) Substantially increasing the life, thus lowering $/KWH delivered; and (2) weight, volume, and/or efficiency improvements will lower component cost and/or STS charges. These two areas are not mutually independent. For example, replacement of nickel – cadmuim batteries every two to three years on each Power Module is an alternative to using high energy density nickel – hydrogen batteries. The weight for this replacement for Scenario I would require an additional ≈100,000 lb of batteries to be launched for replacement throughout the 1980's (72,000 lb every 2.5 years thereafter, even with no additional Power Module Vehicles being built). This amounts to more than $50M dollars of additional STS charges. Thus, the use of $Ni-H_2$ batteries capable of long life at high depths of discharge will result in substantial savings of millions of dollars.

- Also, the cost of the solar array is dictated mainly by the array area. If, as we project, the cell efficiency goes up and the power system efficiency improves, a 50 percent decrease in array area would result for the later power modules.

- Accordingly, a planned method of incorporating these changes is needed essentially at the start of the program. The method proposed here is to provide nearly identical mechanical elements. Thus, a 50 AH $Ni-H_2$ battery should be designed to replace a 60 AH Ni-Cd battery. Also, a mechanical solar array building block should be able to incorporate higher efficiency solar cells or stronger deployment masts.

- Power Module component design ideally would interchange (fit, form, and function) within a common Power Module System. Added elements will be necessary to meet placement and/or orientation constraints. EPS elements of this nature include scale up as necessary for orientation drive and power transfer assemblies (ODAPT) and positioning booms which provide spacecraft separation.

LEO POWER SYSTEM EVOLUTIONARY PATHS



SCENARIO I VEHICLE ASSIGNMENTS

- The Scenario I evolution of the Power Module concept, with modules in various orbits and inclinations throughout the 1980's, is shown in the accompanying chart. The relationship of the electrical power system elements with this evolution has been studied. The study indicates that power system performance enhancement throughout this evolution provides overall system cost and weight savings by minimizing the STS charges for orbit delivery. Furthermore, these performance improvements in almost all instances will be developed regardless of Power Module involvement. It is evident, however, that the sheer quantity of hardware envisioned by this evolution could have far-reaching effects on the cost of future Solar Array Power Systems, should this system be implemented.

- The accompanying chart shows the use of 6 Power Module vehicles in four different orbits during the 1980's and continuing to be used throughout the 1990's. This "utility bus" concept includes both return-to-earth for refurbishment, and on-orbit performance growth. This concept of both refurbishment and on-orbit assembly (repair, replace, return, reuse) warrants careful electrical power component design.

- Nonetheless, alternative paths are identified so that Power Module evolution is not totally dependent on the anticipated technology advancement (See previous chart).

SCENARIO 1 VEHICLE ASSIGNMENTS

	YEAR			
	1983/5	1986/7	1988/9	1990/91
NUMBER OF SYSTEMS ON ORBIT	FV-1 (25 kW)	FV-2 (50 kW) FV-IRI (25 kW) *FV-3 (60 kW)	→ FV-IR2 (25 kW) → FV-4 (50 kW) FV-5 (50 kW)	→ → → FV-4 K(100 kW) → FV-6 (100 kW)
ORBIT/ INCLINATION (DEGREES	57	57 28 GEO	→ 90 → 28 28	→ → → 28 → 28
REFURBISHMENT/TYPE	———	FV-1 TO FV-IRI (25 kW)	FV-IRI TO FV-IR2 (25 kW)	———
GROWTH ON ORBIT (50 kW TO 100 kW KIT)	———	———	———	FV-4 TO FV-4 K(100 kW)

* GEO SOLAR ARRAY SUBSYSTEM

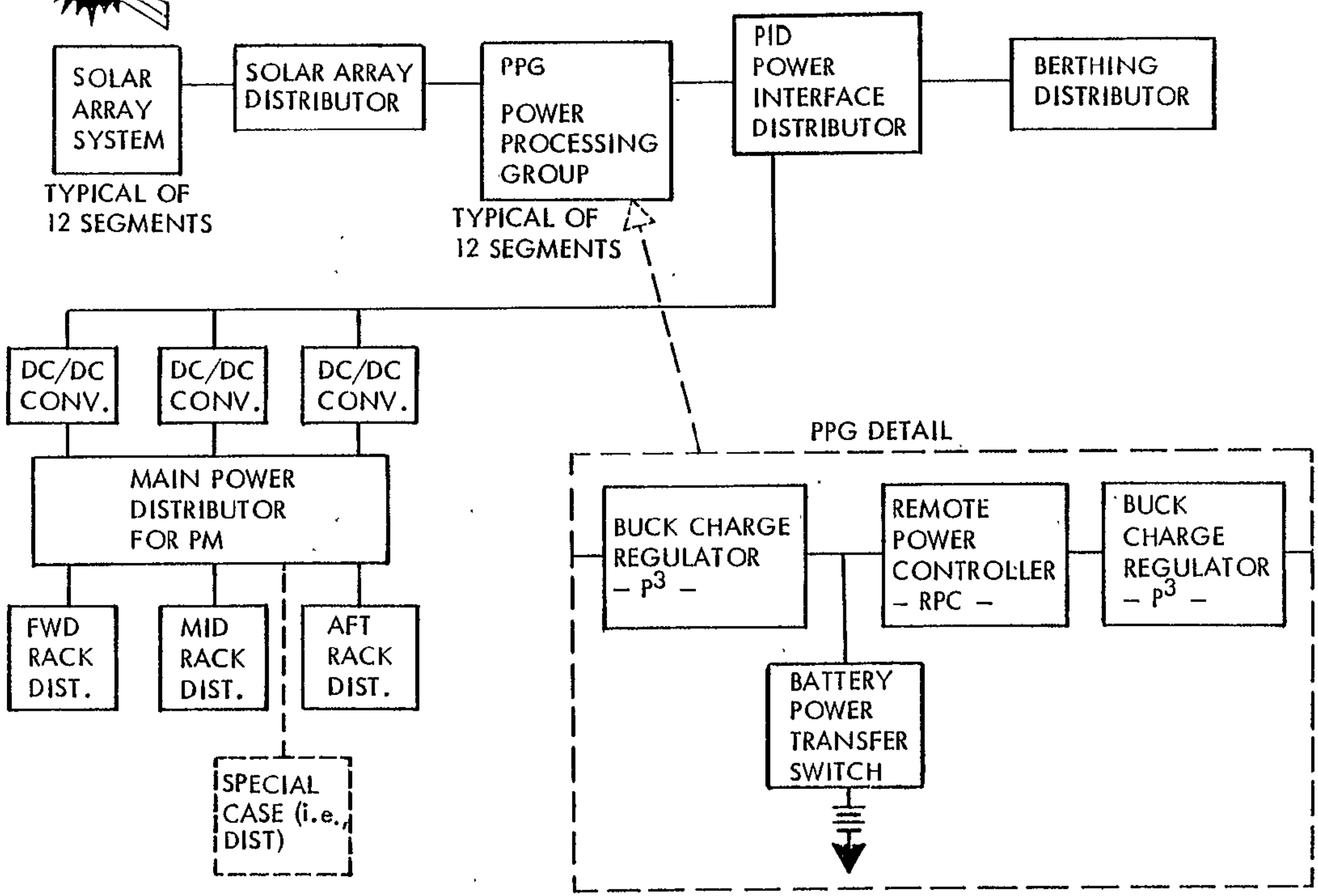
ELECTRICAL POWER SUBSYSTEM BLOCK DIAGRAM

- The Electricl Power System block diagram is shown. The basic power producing, processing, and storage is divided into twelve equal elements. These elements are monitored and controlled by the Power Interface Distributor (PID).

- The PID provides the central management point of the entire distribution system and provides all system management functions. Currents, voltages, and energy are monitored from this point.

- Power for the Power Module is distributed from the PID to DC/DC transformer regulators such that all PM power is isolated from the power distributed to users. A main distribution box provides monitoring and management of the PM power. Auxiliary distribution boxes are used as required throughout the PM spacecraft.

ELECTRICAL POWER SUBSYSTEM BLOCK DIAGRAM



ELECTRICAL POWER SUBSYSTEM EVOLUTION FEATURES

- In Scenario I, the refurbishment of the initial Power Module (FV-1) after approximately 2.5 years of orbital life also is a likely replacement interval for the Ni-Cd batteries. Thus the replacement charges in this scenario would amount to approximately $2.5 M. On the other hand, if batteries were replaced in orbit, the hardware cost of approximately $2.5M is less significant in comparison with transportation ($700/lb x 7500 lb = $5.2M) and on – orbit assembly (EVA/RMS) costs. Thus, return to earth with reusable Power Modules is quite attractive with respect to the EPS.

- Growth on orbit is shown for vehicle FV-4. This growth from 50 kW to 100 kW in power presumes that the technology has evolved to an extent that a long-life, high specific performance energy-storage system is used. The Power Module power processing equipment is also presumed to have evolved with higher efficiency components. The accompanying chart points out some of the salient EPS technology features associated with Scenario I.

ELECTRICAL POWER SUBSYSTEM EVOLUTION FEATURES

- REFURBISHMENT OF TWO 25 kW UNITS AFTER 2.5 YEARS (14,000 CYCLES @ 20% DOD, FOR FV-IRI 1986, AND FV-IR2 1988)
  - REPLACE BATTERIES WITH LONG LIFE NiH_2 TYPES
  - REFURBISH DRIVE SYSTEM AND SOLAR ARRAY MAST
  - REPAIR AND/OR REPLACE
    - POWER ELECTRONICS
    - DISTRIBUTORS
    - SOLAR ARRAY PANELS
- GROWTH ON ORBIT REQUIRED ON 50 kW (FV-4) SYSTEM
  - GROWTH UNITS WILL REQUIRE HIGH POWER DESIGN FEATURES AT INITIAL DESIGN TO ALLOW ADDED POWER AND EPS WASTE HEAT DISSIPATION
    - THERMAL IMPACTS
    - MECHANICAL IMPACTS
  - GROWTH UNITS USE UPDATED SOLAR ARRAY, BATTERY, AND ELECTRONICS
- INITIAL 60 kW UNIT (FV-3) DOES NOT REQUIRE GROWTH BUT DOES REQUIRE LONG LIFE

ELECTRICAL POWER SUBSYSTEM GROWTH

- Power systems have been conceptualized for each of the Scenario I vehicles, utilizing the expected technology available in the applicable time periods. The results of this effort are summarized in the chart.

- As can be seen from the estimated weights of the power system components, power system specific performance is projected to improve to over 4 watts/pounds, from the 1.5 watts/pound produced by the initial power module.

- The main emphasis was placed on power systems for low earth orbit (LEO). However, it can be seen that for a geosynchronous (GEO) derivative power system, specific performance is greatly improved: to a level of approximately 8 watts/pound.

ELECTRICAL POWER SUBSYSTEM GROWTH

ORBIT	LEO							GEO
LAUNCH DATE	1983	1986	1986/88/89	1988	1990	1990 (5)	1991	1987
POWER – kW	25	25	50	25	50-100(KIT)(1)	100-1'	100-1(6)	60
CELL TYPE, #/FT^2	Si, 0.2	Si, 0.2	Si, 0.2	Si, 0.15	Si, 0.15	Si, 0.15	Si, 0.15	Si, 0.15
CELL EFF – %	12-13	12-13	14-16	16-18	16-18	16-18	16-18	12-13
BATTERY	Ni-Cd	Ni-H_2	Ni-H_2	Ni-H_2	Ni-H_2	Ni-H_2	Ni-H_2	H_2-O_2RFC(3)
DoD – %	20	20	40	20	80	80	80	50
VOLTAGE, DC REG	110/28	110/28	110	110	110	110	110	110
WEIGHTS – #								
SOLAR ARRAY	2,400	2,400	4,800	1,800	3,600	8,400	7,200	1,800
ADAPT(2)	200	200	250	200	300	550	300	200
BATTERIES	7,440	6,400	6,400	6,400	6,400	12,800	6,400	3,200(3)
ELECTRONICS	1,395	1,395	1,395	1,395	2,055	3,450	2,055	660
PWR DISTRIBUTION	630	630	880	630	1,145	2,025	1,145	200
CABLING	500	500	500	500	500	1,000	500	300
CONTINGENCY – 25%	3,141	2,881	3,556	2,731	3,500	7,056	4,400	1,590
TOTAL	15,706	14,406	17,781	13,656	17,500	35,281	22,000	7,950
VEHICLE (4) CONFIGURATION	FV-1/ B-1	FV-1R1 A-1	FV-2/B-2 FV-4/A-2 FV-5/A-3	FV-1R2 C-1/C-2	FLIGHT KIT	FV-4/ A-4	FV-6/ A-5	FV-3/ D-1

1. THE 50-100 kW KIT ALLOWS ON-ORBIT BUILDUP TO 100 kW (WEIGHTS SHOWN ARE FOR KIT COMPONENTS ONLY).
2. ORIENTATION DRIVE AND POWER TRANSFER ASSY.
3. REGENERATIVE FUEL CELL (WEIGHT INCLUDES FUEL CELLS, ELECTROLYZER, TANKAGE, REGULATORS, REACTANTS)
4. VEHICLE DESIGNATORS CORRESPOND TO FLIGHT VEHICLES AS SHOWN IN VOLUME 2 OF THIS PART III FINAL REPORT, PAGES 2 - 13. CONFIGURATION DESIGNATORS REPRESENT CONFIGURATIONS SHOWN ON PAGES 3 - 62 IN VOLUME I OF THIS PART III FINAL REPORT.
5. VEHICLE WEIGHT ON-ORBIT AFTER 50-100 kW KIT INSTALLATION.
6. THE 100 kW GROUND LAUNCHED CONFIGURATION.

POWER SERVICES TO USERS

- The electrical power interface from the Power Module berthing module will require either a simple set of redundant power leads or a complex network of multi-buses and multi-voltages. From the power system point of view, the basic power system is comprised of twelve building blocks. These twelve power producing and processing segments can be bused and/or cross-bused in any manner.
- The requirement for three buses for Orbiter/Spacelab interfacing is the only well-defined bus requirement, wherein power management through the berthing port to the Orbiter will require compatibility with Orbiter/payload bay existing buses. If the Orbiter is on-orbit for a short-term sortie mission, compatibility with the PEP/fuel cell busing/electronics would also be required. Remote sensing of fuel cell voltage into the Power Module makes interfacing with the PEP electronics desirable.

POWER SERVICES TO USER

- THREE BASIC ALTERNATIVES

REGULATED HIGH VOLTAGE	110 VDC
UNREGULATED HIGH VOLTAGE	110-165 VDC
REGULATED LOW VOLTAGE	28 VDC

- SUPPLY ALL THREE – TOO COMPLEX
- SUPPLY ONLY ONE VOLTAGE – SIMPLE
- FEEDBACK CONTROL FROM USER IS REQUIRED FOR LOW VOLTAGE APPROACH
- SUPPLY UNREGULATED HIGH VOLTAGE TO PEP ELECTRONICS FOR ORBITER REQUIREMENTS
- RECOMMENDATION: SUPPLY ONLY REGULATED 28 VDC POWER TO ALL USERS. HOWEVER, CONTINUE INVESTIGATION OF MERITS OF SUPPLYING UNREGULATED HIGH VOLTAGE TO USERS IN LATER MODULES.
- RATIONALE: LEAST RISK AND COMPLEXITY, FOR FIRST POWER MODULE.

GROWTH CRITICAL DESIGN DRIVERS

- The growth critical design drivers for the Power Module evolution are shown. The thermal dissipation expansion capability for control of power processor and battery temperatures is of concern. The alternative approach is to have additional parallel elements such that power density and/or losses are no greater in a given box. Thus the thermal inpedances would be identical. Conceptually with a fluid loop system the capability to remove the heat can be expanded; however, the ability to provide sufficiently low thermal inpedances inside the power processor at higher power dissipation levels would impose some weight penalty. The growth of power, and thus current, within the Power Module is a critical design driver.

- For our scenario we have assumed high voltage distribution for all Power Modules after the initial unit. This greatly reduces the wire size with substantial weight savings. The quantity and size of remote power controllers is also greatly reduced. Thus the power management subsystem is not as complex or costly.

- The scenario developed here for growth of orientation drive and power transfer assembly (ODAPT) is that as the solar array grows the ODAPT must be expanded. The basic drive unit can initially be designed to provide for expansion by proper choice of torques and power transfer current densities. This, however, should be determined on the basis of the evolutionary configuration changes.

GROWTH CRITICAL DESIGN DRIVERS

- THERMAL DISSIPATION (ENERGY DENSITY)
- DRIVE SYSTEM EXPANSION
- ELECTRONICS CURRENT CAPABILITY
- BATTERY WASTE HEAT CONTROL
- TYPE OF POWER TRANSMISSION TO USERS
- POWER MANAGEMENT CONTROLLERS

EPS KEY TRADES: SOLAR ARRAY

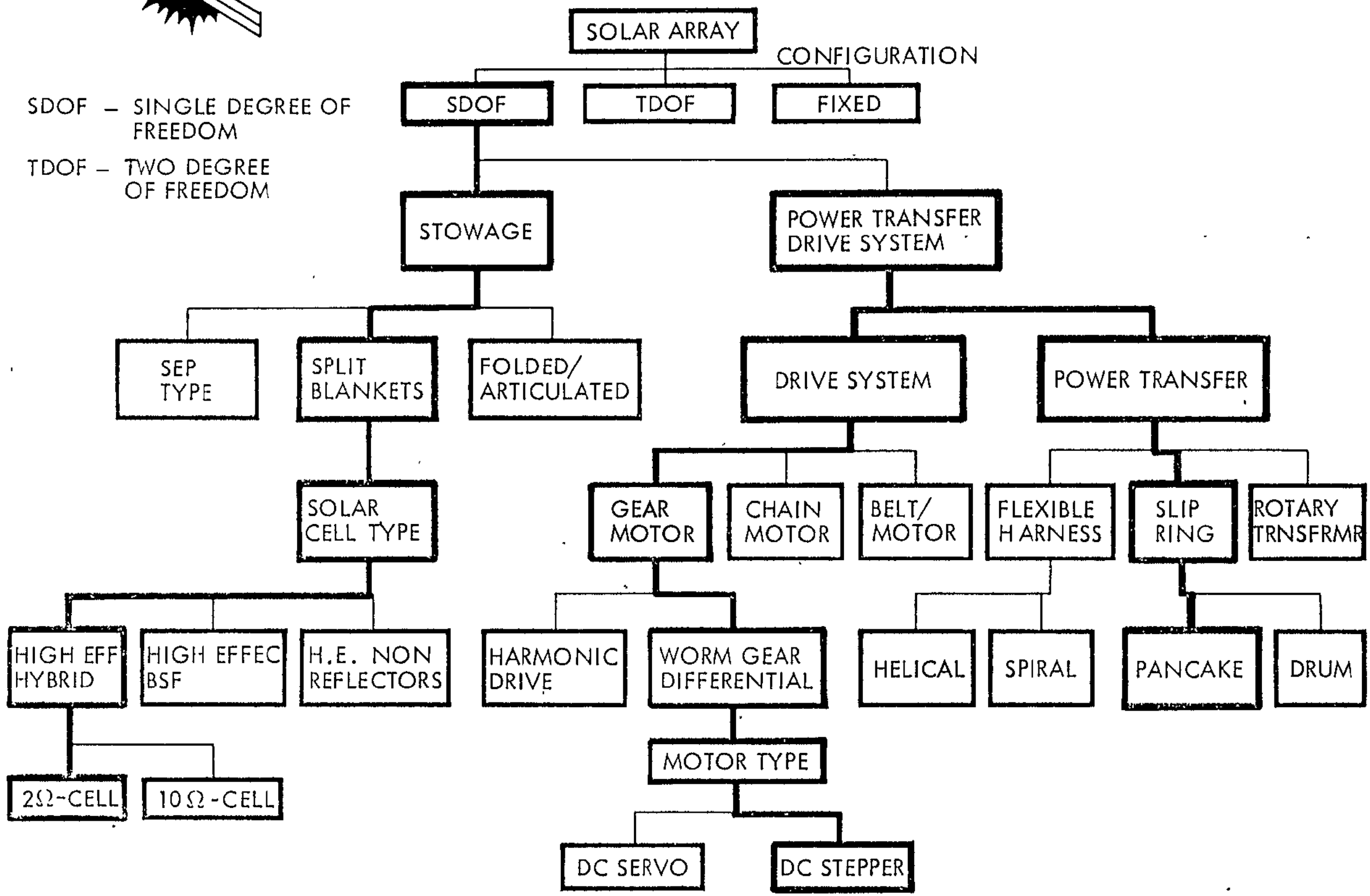
- LMSC performed various trades in support of the PM Evolutionary Study. These trades included: 1) State-of-the-art review of batteries, solar arrays, regulators, and power distribution, 2) Solar Array Configurations, 3) Energy Storage Configurations, 4) Electrical Power System Configurations, 5) Soar Array Deployment Mast Configurations 6) Dynamic Analysis, 7) Orientation Analysis, 8) Drive System Considerations, 9) Power Electronics Considerations, and 10) Orbit Considerations. Typical trade-trees were made initially to depict the recommended selection. These trade-trees are shown in the next three charts.

- The selected system depicted for the S/A is SDOF; Split Blankets, 2 OHM-cm high efficiency hybrid solar cells with a worm gear/differential DC Stepper Motor drive system and pancake slip ring for power transfer.

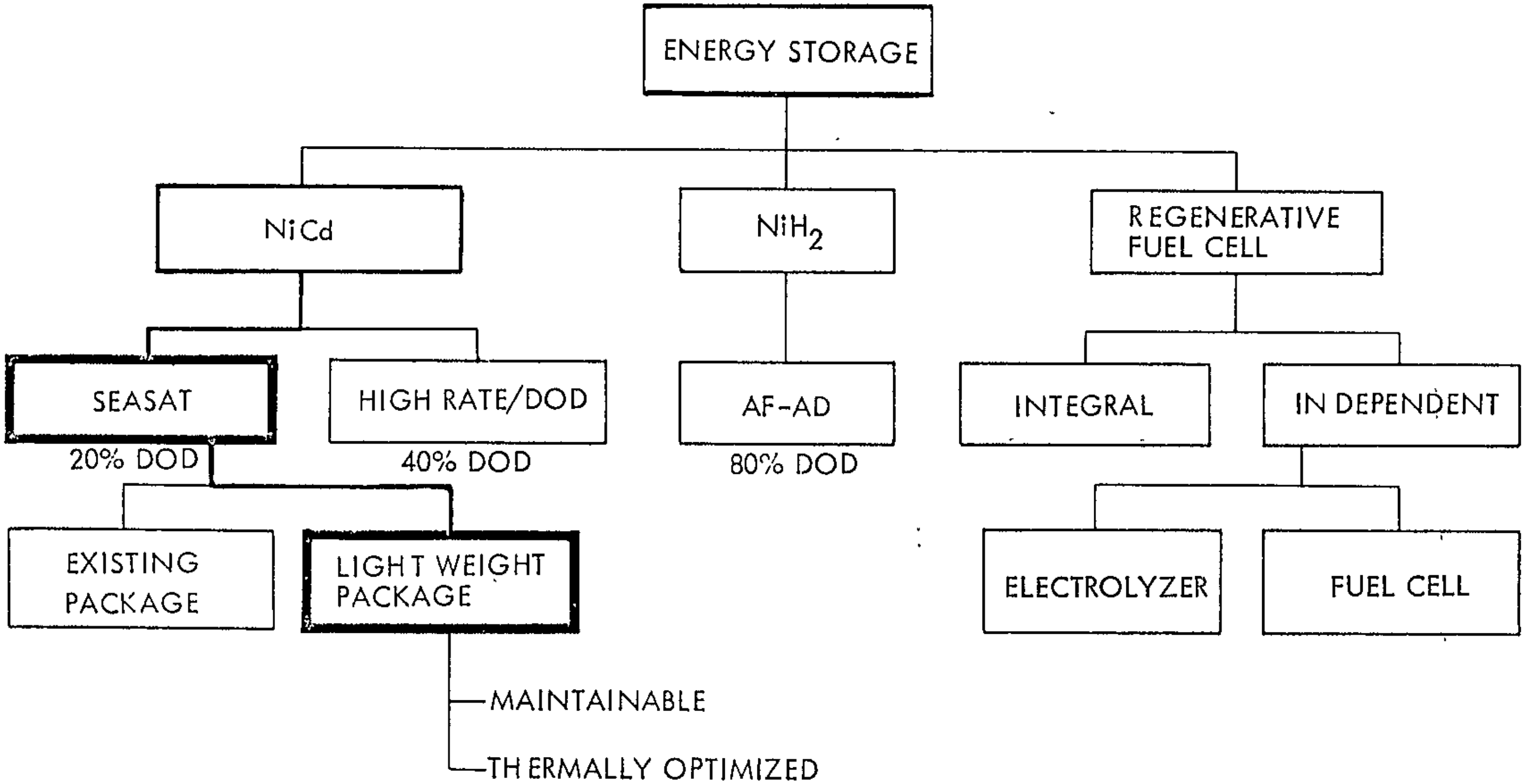
- These trade trees were developed to depict some of the major considerations affecting the 25 kW initial vehicle design selections, and what considerations would be appropriate for Power Module Evolution. The results of some of the major trades were presented on 29 June 1978 to MSFC; these charts are presented in Appendix B. Many of these trades were then presented in the Final Report (Reference 31) of Part II. Areas not covered in the Part II report included drive system layouts and solar array configuration layouts.* Also a basic solar orientation study was made to determine the feasibility and need to provide additional degrees of freedom for the S/A in a sortie mode. Trades performed since those reported in Appendix B and Part II are summarized after the trade-tree charts.

*See Reference 32

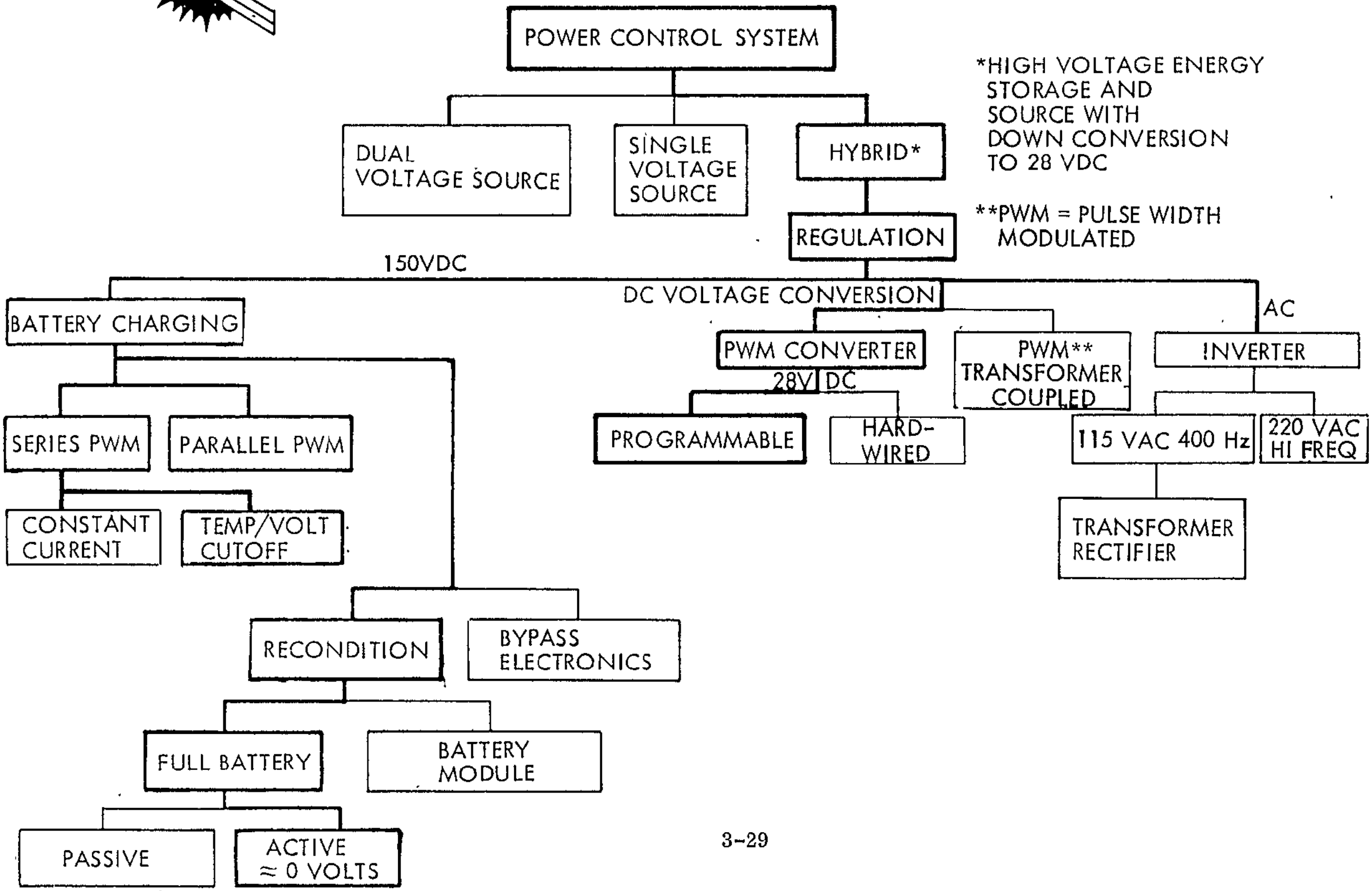
EPS KEY TRADES: SOLAR ARRAY



EPS KEY TRADES: ENERGY STORAGE



EPS KEY TRADES: POWER CONTROL SYSTEM



*HIGH VOLTAGE ENERGY STORAGE AND SOURCE WITH DOWN CONVERSION TO 28 VDC

**PWM = PULSE WIDTH MODULATED

EPS EVOLUTIONARY GROWTH OPTIONS

- During the evolutionary study many EPS options were addressed. The major options, and their impacts, are shown in the preceeding chart.
- The most significant options will be the incorporation of Ni-H_2 batteries into later Power Modules.
- Although these options can provide improved features, no area was uncovered in which new technology, per se, is needed to make the Power Module EPS a success. Many of the options/features are items that will be developed whether they are required by the Power Module or not. The weight and cost savings to the Power Modules alone warrant their development in any case.

EPS EVOLUTIONARY GROWTH OPTIONS

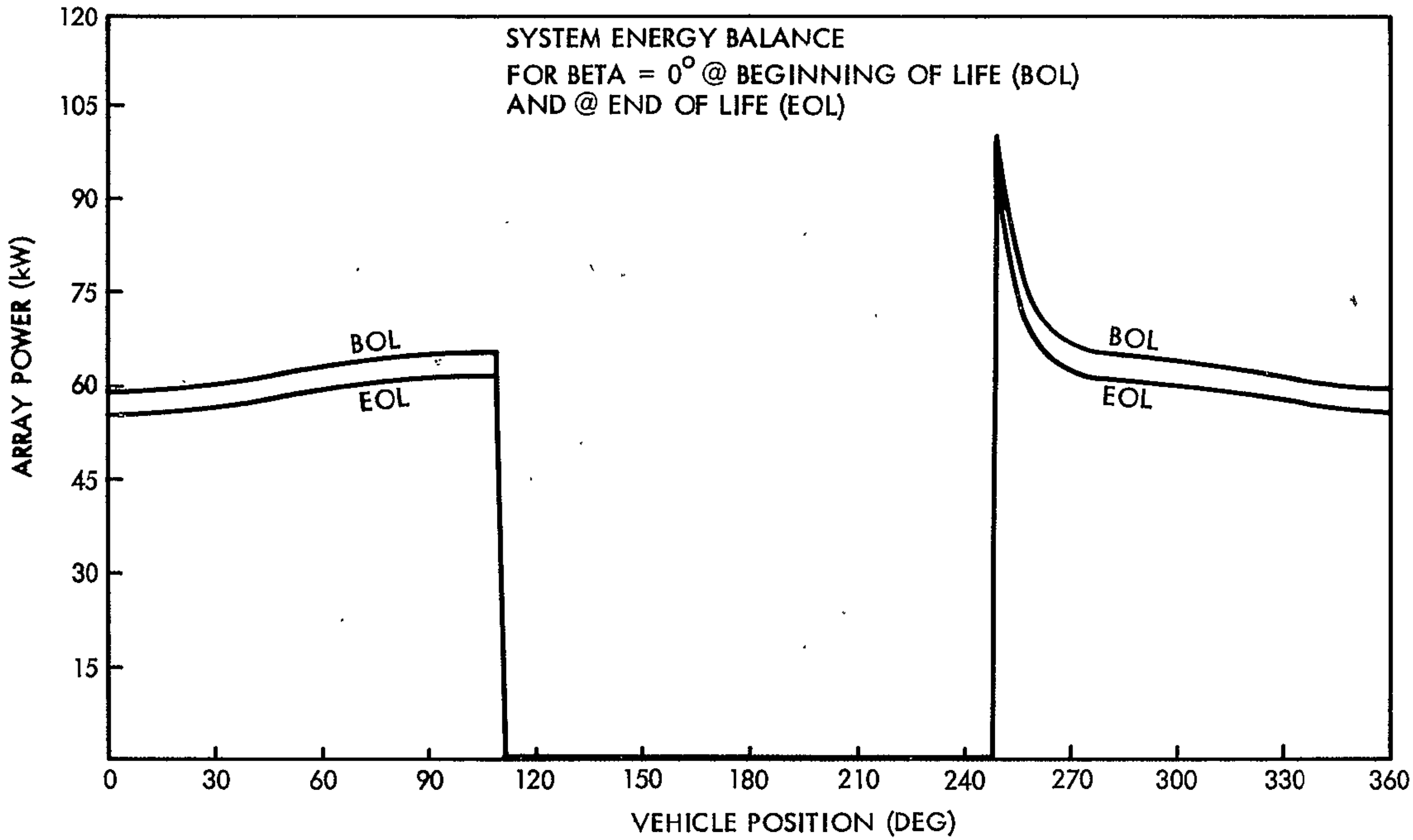
AREA	RATIONALE
(1) NiH_2 BATTERIES	• SUBSTANTIAL LIFE AND PERFORMANCE GAINS EXPECTED
(2) HIGH VOLTAGE DISTRIBUTION	• DECENTRALIZE REGULATION TO USER – IMPROVES SUBSYSTEM AND SYSTEM RELIABILITY • SUBSTANTIAL WEIGHT/COST SAVING • MINIMIZES DISTRIBUTION DUPLICITY
(3) WIDER SOLAR ARRAY BLANKETS	• MINIMIZES NUMBER OF FAB/TEST UNITS • GROWTH ACCOMMODATION IMPROVED WITHIN ORBITER LENGTH CONSTRAINTS • NO INHERENT LIMITATION OF SEPS TECHNOLOGY • 6 METER BLANKETS APPEAR TO BE LARGEST WIDTH DESIRABLE
(4) DEVELOP TRANSFORMER COUPLED REGULATOR (TCC)	• PROVIDES SIGNIFICANT EFFICIENCY AND WEIGHT BENEFITS FOR INITIAL AND/OR FUTURE PM WITH DECENTRAL REGULATION • PROVIDES DC ISOLATION BETWEEN USING ELEMENTS • ELIMINATES HIGH VOLTAGE SHORTING PROBLEM AND THE NEED FOR SHUNT REGULATORS
(5) TWO AXIS ORIENTATION DRIVES FOR LARGER ARRAYS	• IMPROVES FLEXIBILITY OF BASIC PM • MAY MINIMIZE GIMBALS AT P/L INTERFACE • IMPROVES ALL-BETA, ALL-INCLINATION PERFORMANCE

ARRAY POWER VS VEHICLE POSITION

- The performance of the Power Module electrical system is most dependent on the characteristics of the solar array. The solar array operates at unique temperatures for different Beta angles and solar cell types. The array is also degraded by the space environment. Thus the sizing of the solar array is highly dependent on the mission orbit-altitude and inclination. The power available for a Beta = 0° orbit is shown.

- As can be seen the power varies throughout the orbit and can nearly double upon emergence from the dark. This characteristic defines the required capability desired for the P^3 regulator. That is, the regulator should extract the maximum power that is available from the solar array.

ARRAY POWER VS VEHICLE POSITION

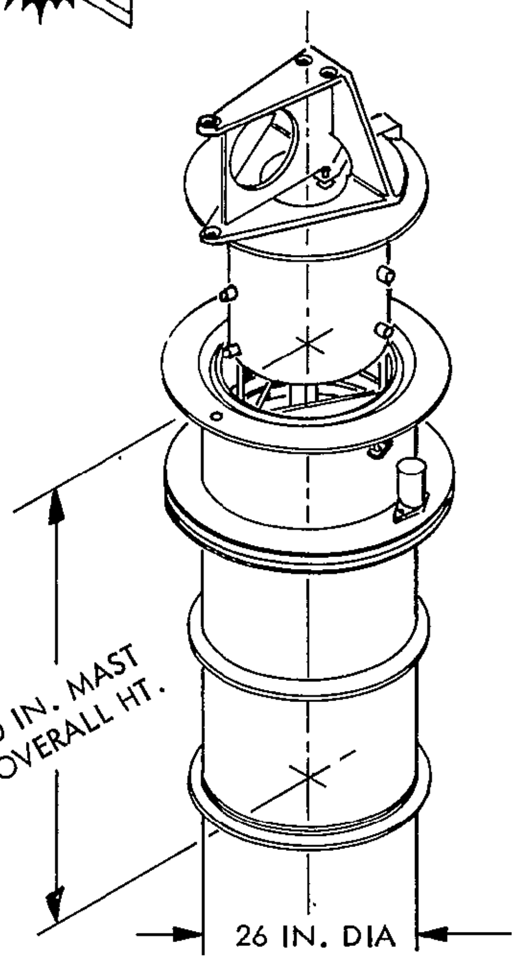


SOLAR ARRAY MAST CAPABILITIES

- The capability of the solar array system to react to the induced on-orbit g loads (when deployed) has been investigated. The SEP-type-technology deployment mast (with continuous fiberglass longeron), which is capable of providing sufficient blanket tension to meet a greater-than 0.04 Hz natural frequency requirement, is presently the baseline design.

- If an articulated steel longeron mast element is used within the same envelope constraint, the loading capability is improved by a factor of ten. The disadvantages of this approach are: (1) weight, (2) a more complex mechanism, and (3) free-play in the mast joints with the attendant effect on natural frequency.

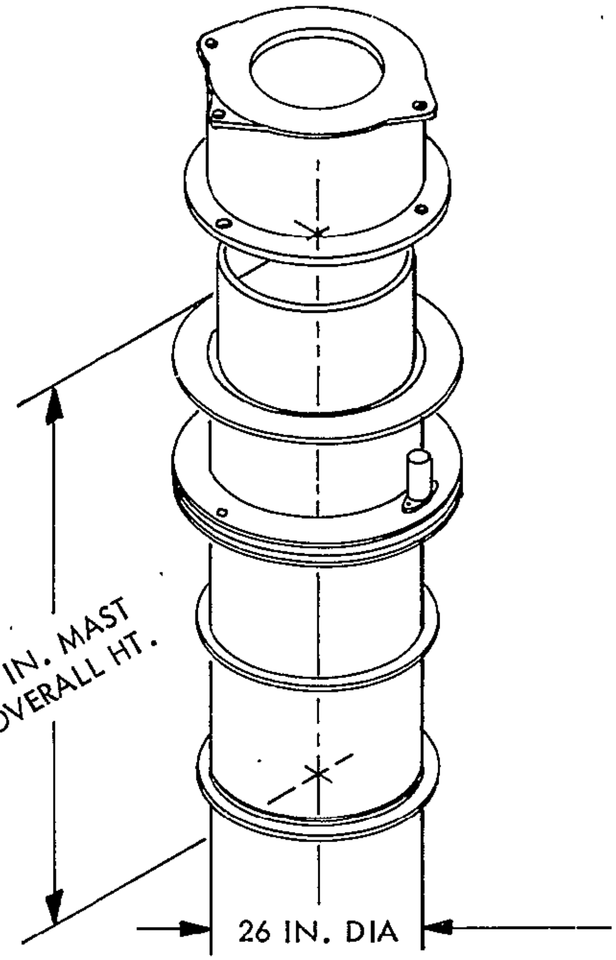
SOLAR ARRAY MAST CAPABILITIES



CONTINUOUS FIBERGLASS LONGERON

R = 11.0 IN.
WT = 180 LB.
G LEVEL = .004

$EI = 100 \times 10^6$ LB-IN^2
M_{cr} = 330 LB-FT



ARTICULATED STEEL LONGERON

R = 11.0 IN.
WT = 590 LB.
G LEVEL = .04

$EI = 300 \times 10^6$ LB-IN^2
M_{cr} = 3300 LB-FT

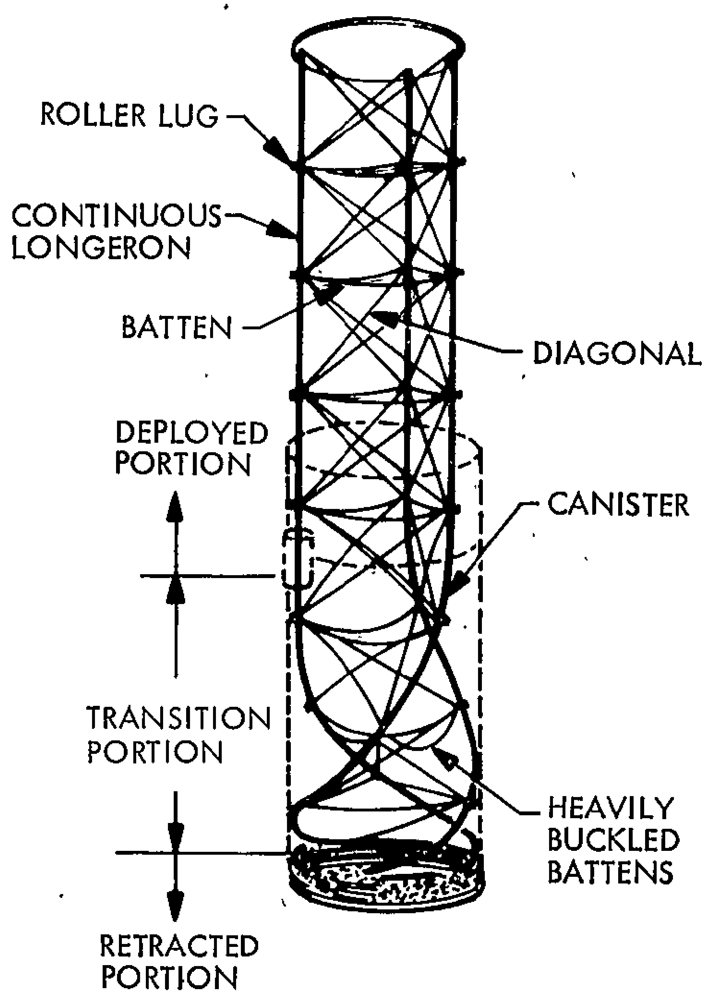
SOLAR ARRAY DEPLOYMENT MAST

- The deployment geometry, and prime component nomenclature, are illustrated for both the mast and the cannister which contains the mast prior to its deployment.

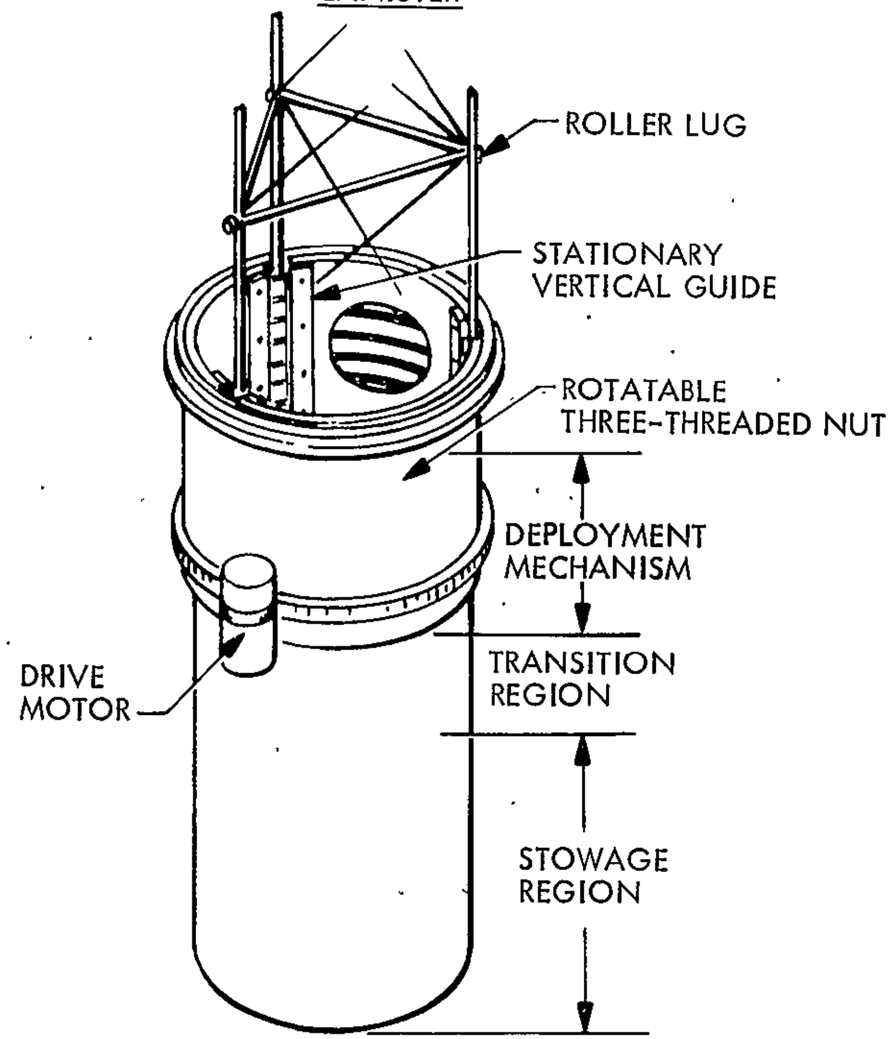
- This mast design is used on the SEP program, where the development program is proceding through on-orbit testing.

SOLAR ARRAY DEPLOYMENT MAST

DEPLOYMENT GEOMETRY AND NOMENCLATURE



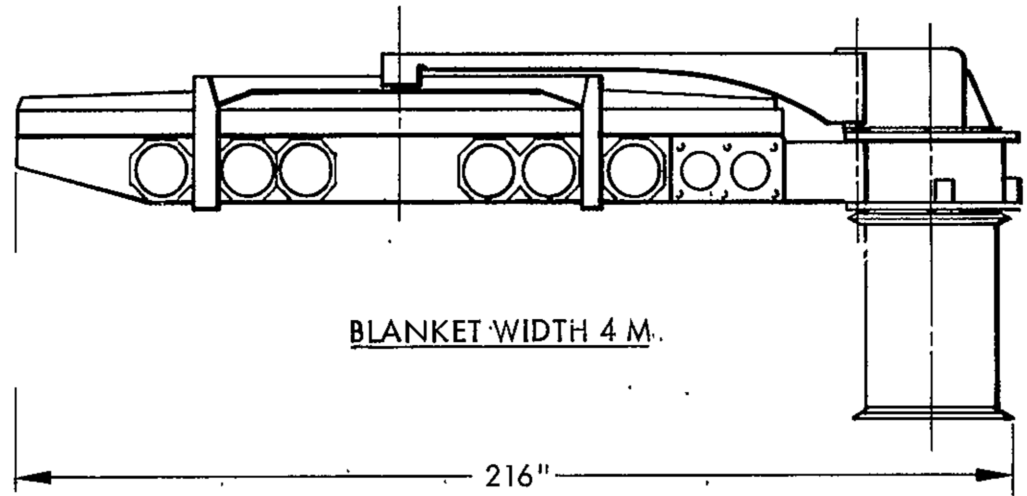
CANISTER

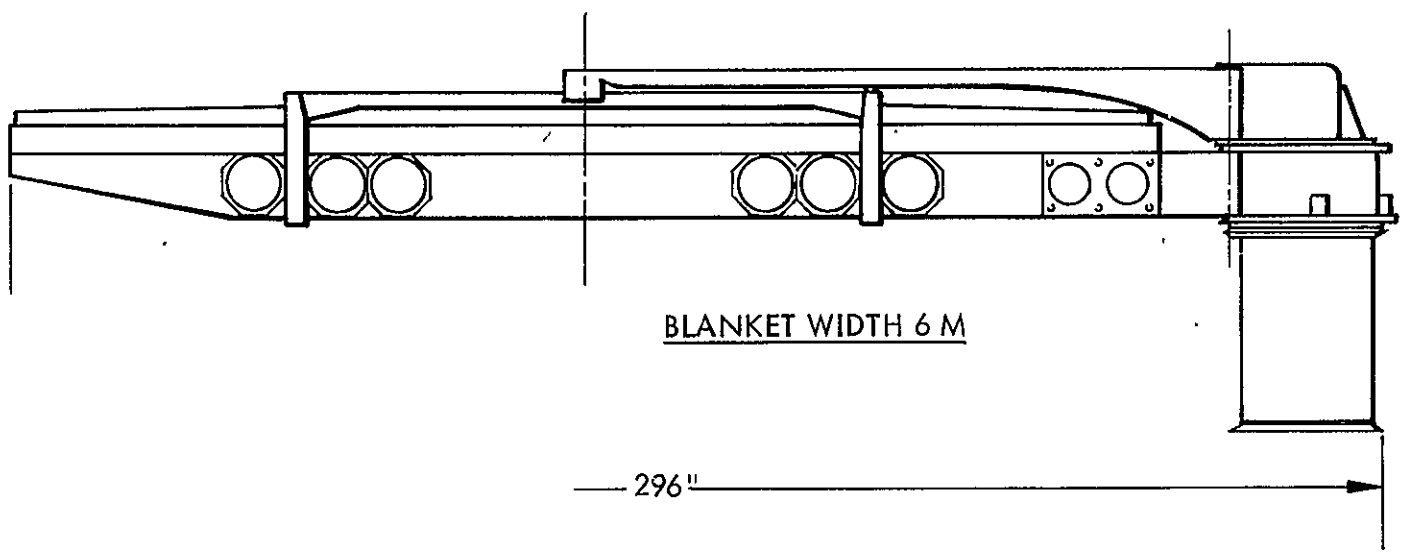


SOLAR ARRAY GROWTH

- LMSC studied various methods of expanding the solar array system to provide increases in areas to meet the higher power levels. The result of this study concluded that a wider solar array blanket would provide more flexibility for growth for spacecraft carried to orbit in the Orbiter Bay. Configurations were studied for solar arrays up to 50,000 square feet (250 kW Power Modules).
- The use of a 6-meter wide solar array building block is shown in comparison to the 4-meter SEP-type system necessary for SEP and PEP because of the storage limitations. As can be seen on the chart, the 6 meter system is conceptually identical to the 4-meter system.

SOLAR ARRAY GROWTH





POWER PROCESSING EQUIPMENT SIZING VS GROWTH

- The power processing equipment for power control and regulation must provide for either parallel operation of units as the Power Module power increases or for increased current capability per individual unit. Since the major driver in increasing voltage is to minimize the number of power processing units (as well as achieving system efficiency), growth in current capability in the power processing units is desired.

- The chart shows how the power level (current) increases in the programmable power processors (P^3). The use of twelve individual power strings results only as a consequence of the energy storage sizing. The projected use of a 50-AH Ni-H_2 battery then determines the number of batteries that make up the energy storage complement for each power level.

- The current carrying capability required in the initial Power Module is determined by the P^3 regulator. This regulator must be designed to regulate voltage at 28 VDC currents approaching 150 amperes. This built-in capability then is acceptable for the P^3 changes up to approximately the 75 kW Power Module level (27 kW charger output). If power is not supplied at higher voltages the output regulators would then have to be paralleled. Thus 24 P^3 regulators would be required at 50 kW output and 48 regulators for 100 kW. This can be avoided by scale-up of the bus regulator or by encouraging use of power at higher voltages.

POWER PROCESSING EQUIPMENT SIZING VS GROWTH

NOTE: SIZING SHOWN FOR EACH OF TWELVE PARALLEL ELECTRICAL POWER STRINGS

DISTRIBUTORS	
28 VDC 110 VDC	25 kW 50,100 kW

POWER MODULE TOTAL OUTPUT	SOLAR ARRAY	P^3 CHARGE CONTROLLER	BATTERY	BUS/REG AND CONTROL
25 kW	5 kW S/A	5 kW P^3 (9)* CHARGE CONTROLLER	8 kWH BATTERIES (60 AH Ni-Cd) 20% DOD	3 kW BUS CONVERTERS AT 28 VDC (TBD)
50 kW	9 kW S/A	10 kW P^3 (18 kW)*	7 kWH BATTERIES (50 AH $Ni-H_2$) 40% DOD	6 kW BUS CONVERTERS AT 110 VDC (TBD)
100 kW	18 kW S/A	20 kW P^3 (36 kW)*	7 kWH BATTERIES (50 AH $Ni-H_2$) 80% DOD	12 kW BUS CONVERTERS AT 110 VDC (TBD)

*PEAK

COMPARISON OF BUCK VS TRANSFORMER CONVERSION
(140 Vdc to 28 Vdc)

- One of the major results of the evolutionary study is the analysis that relates the power system efficiency to the power system cost. The major driver in the EPS cost effectiveness is the cost of the solar array. The major driver in overall EPS efficiency is the main down regulator (120 to 28 vdc). Both the size of the solar array and the battery depend on this regulator efficiency.

- The analysis summarized on the chart shows the cost saving incurred, in relation to solar array size and weight, as well as the savings related to the weight savings for the radiators and regulators. The efficiency of voltage conversion is an important consideration for PM scenarios since many varied service requirements are to be expected.

COMPARISON OF BUCK VS. TRANSFORMER CONVERSION (140vdc TO 28vdc)

•	PREDICTED EFFICIENCY	BUCK REGULATOR TRANSFORMER CONVERTER	0.88 0.92
•	SOLAR ARRAY SIZE SAVING	4110 WATTS @ \$300/WATT 135 LBS @ \$700/LB	= \$1,233,000 = 94,000
•	CONVERTER WEIGHT SAVING	414 LBS @ \$700/LB	= 289,800
•	RADIATOR AFFECT	3600 WATTS LESS DISSIPATION 110 LBS OF RADIATOR @ \$700/LB	= 77,000
•	TOTAL ADVANTAGE/ 25 kW PM	DOLLAR WEIGHT	\$1,700,000 ≈700 LB
	ADVANTAGE FOR 100 kW CONVERSION TO 28VDC		\$7,800,000
•	ESTIMATED COST OF TCC DESIGN, DEVELOPMENT, QUALIFICATION		\$1,200,000
•	NET SAVINGS ON ONE PM		= \$ 500,000

ELECTRICAL POWER SUBSYSTEM TRADES – SUMMARY

- The accompanying chart identifies the scope of primary trades performed to support the evolution of the Power System from 25 kW to 250 kW power levels. These trades were employed to size and select the power system components to meet the requirements generated from the mission scenarios.

- The recommended configurations, and the rationale for their selection, are summarized.

ELECTRICAL POWER SUBSYSTEM TRADES—SUMMARY

ELEMENT OR COMPONENT	NUMBER OF CANDIDATES	CONFIGURATION SELECTED INITIAL 25 kW	CONFIGURATION SELECTED GROWTH	SELECTION RATIONALE
EPS SYSTEM CONFIGURATION	4	28 VDC	110 VDC	• EFFICIENCY, WEIGHT, COST (SERIES BUCK SYSTEM)
SOLAR ARRAY CONFIGURATION	6	FOLDED 4 METER	FOLDED 6 METER	• ORBITER ULTILIZATION
ENERGY STORAGE SYSTEM	3	Ni-Cd (<20% DOD)	$Ni-H_2$ (20 TO 80% DOD)	• MEETS INITIAL REQMTS • $/KWH IS DESIGN DRIVER
BATTERY CONTROL (CHARGE, PROTECTION, CONDITION)	3	SERIES PWM BUCK W/PPT	SAME	• MINIMIZE S/A SIZE & COST; MAXIMIZE BATTERY LIFE
BUS REGULATION	3	TRANSFORMER CONVERTER @ 28 VDC	SERIES PWM BUCK @ 110	• MINIMIZE S/A SIZE AND COST,
BUS VOLTAGE (DISTRIBUTION)	3	28 VDC REG	140 ± 30	• DECENTRALIZE REGULATION TO ENCOURAGE HIGH VOLTAGE USAGE
POWER SYSTEM VOLTAGE	1	110 - 185	110 - 185	• LIMIT OF PRESENT COMPONENT TECHNOLOGY
ORIENTATION DRIVE AND POWER TRANSFER (ODAPT)	4 DIFFERENT AXES	1 AXIS	2 AXES	• 1 AXIS IS ADEQUATE WITH SMALL PMs
SOLAR DEPLOYMENT MAST	2	SEPS TECHNOLOGY COILABLE	SPACE STATION ARTICULATED	• HIGH LOADS REQUIRE UNACCEPTABLE VOLUMES USING COILABLE MAST
SOLAR CELL TYPES	5	2Ω - CM HYBRID	SAME	• HIGHEST WATTS/$

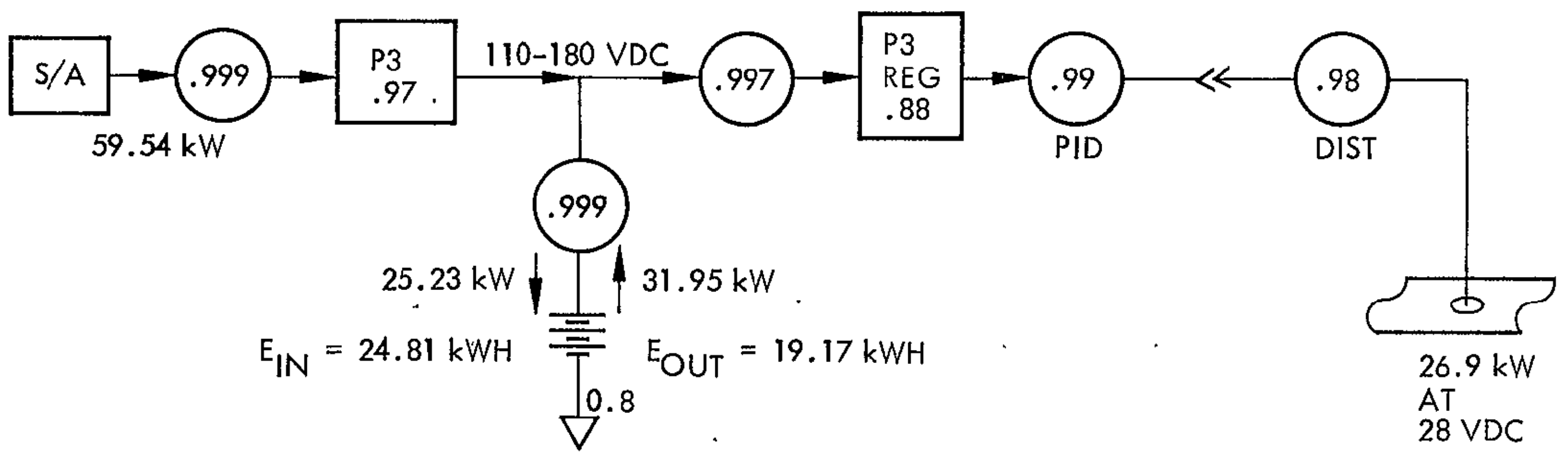
RECOMMENDED POWER SYSTEM CONFIGURATION

- The recommended power system configuration is shown in the accompanying chart. LMSC investigated many alternative power system configurations. The choice of the recommended system is based on the use of MSFC P^3 technology developments. The long range objective is to use the programming capability to encourage use of high voltage.

- The use of 28 VDC distribution requires component designs for handling the switching/control functions in the distributors and will result in the heaviest distribution system. However, this is no different than the Orbiter System and is thus a low technical risk approach.

RECOMMENDED POWER SYSTEM CONFIGURATION

SYSTEM ENERGY BALANCE FOR 25 kW POWER MODULE



SOLAR ARRAY POWER REQUIRED EOL = 59.54kW

BATTERY ENERGY USED EOL = 19.17 kWH

◯ = IR LOSS OR REMOTE POWER CONTROLLER

PID = POWER INTERFACE DISTRIBUTOR

DIST = BERTHING DISTRIBUTOR

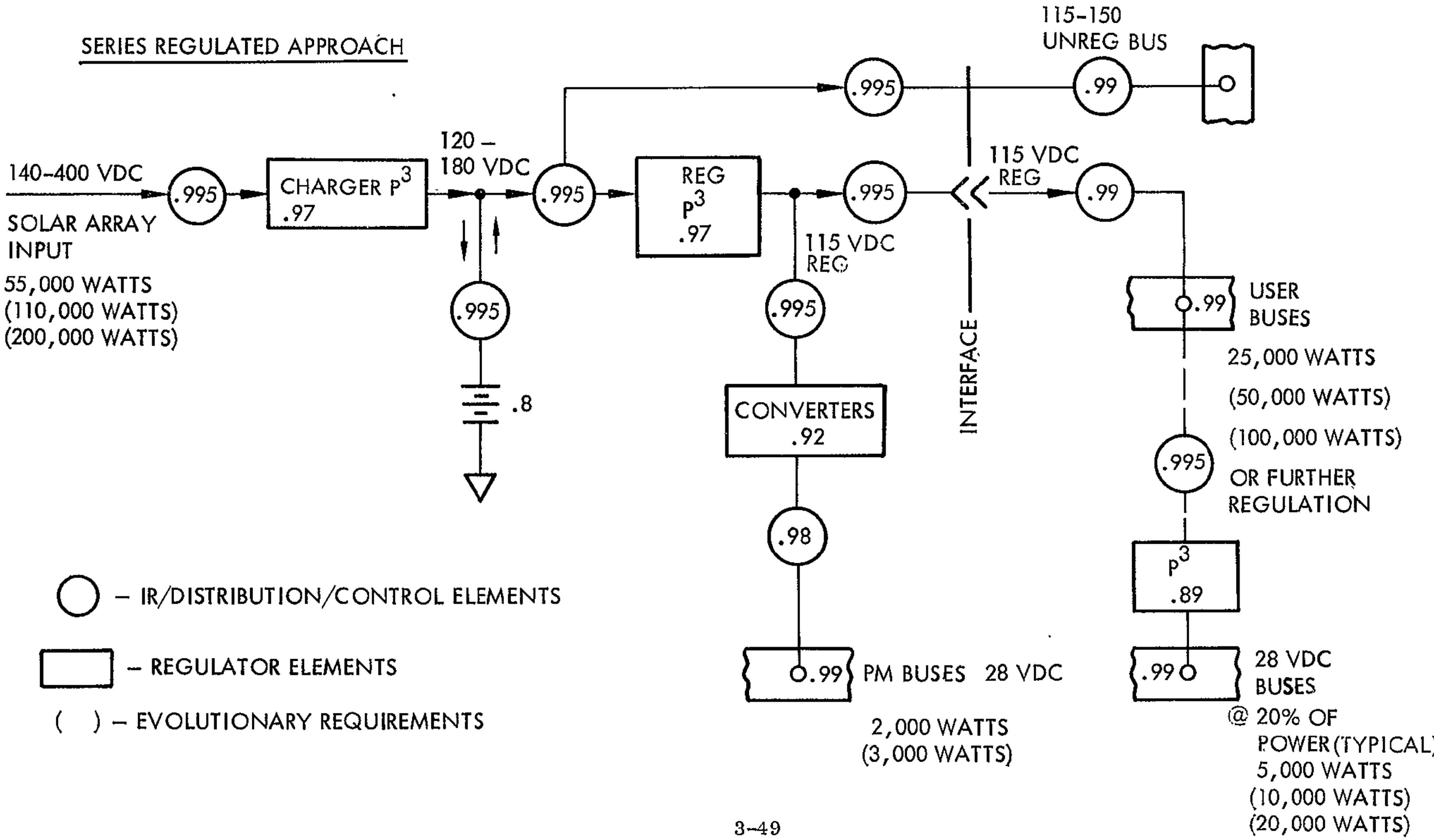
P^3 = PROGRAMMABLE POWER PROCESSOR

ALTERNATIVE POWER MODULE ELECTRICAL SYSTEM

- While the initially recommended system (for the 25 kW Power Module) supplies 28 VDC service only, a later Power Module will very likely supply both regulated and unregulated power, with several user-options. Such a concept is described on the chart.

ALTERNATIVE POWER MODULE ELECTRICAL SYSTEM

SERIES REGULATED APPROACH



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ELECTRICAL POWER EVOLUTIONARY STUDY —RESULTS

- SOLAR ARRAY GROWTH CAN BE ACCOMMODATED WITH SEPS TECHNOLOGY SOLAR ARRAYS AT LEAST UP TO 50,000 FT^2.
- ENERGY STORAGE SUBSYSTEM SHOWS MOST POTENTIAL FOR MAJOR PERFORMANCE IMPROVEMENTS WHICH SHOULD SUBSTANTIALLY LOWER COST ($/kWH AND WEIGHT (kWH/LB).
- *POWER ELECTRONIC* COMPONENT TECHNOLOGY ALLOWS A 5 FOLD INCREASE IN VOLTAGE LEVEL (28 VDC TO 140 VDC) WITH SUBSTANTIAL COST AND WEIGHT SAVINGS, ESPECIALLY IN POWER PROCESSING EQUIPMENT.
- SCALING SOLAR ARRAY TO 6 METER BLANKETS FOR 50 kW AND ABOVE MINIMIZES SOLAR ARRAY COMPLEXITY. (FEWEST MASTS AND IDENTICAL TEST METHODS.)
- DISTRIBUTION OF DC SOURCE AND ENERGY STORAGE HIGH VOLTAGE TO USER RESULTS IN SIMPLEST SERVICE APPROACH AND WOULD ENCOURAGE USE OF HIGHER VOLTAGE POWER THROUGHOUT 1980s (DISTRIBUTION TECHNOLOGY FOR APPROXIMATELY 200 VDC EXISTS).
- SOLAR ARRAY VOLTAGE EXTREMES PRECLUDE DISTRIBUTION OF POWER ABOVE APPROXIMATELY 150 VDC.
- 120 VDC POWER CAN EASILY BE INVERTED OR CONVERTED USING POWER COMPONENT TECHNOLOGY REQUIRED FOR THE BATTERY CHARGE CONTROLLER (P^3).
- VOLTAGE CONVERSION GREATER THAN 4-1 CAN BE ACCOMPLISED MORE EFFICIENTLY WITH A TRANSFORMER COUPLED CONVERTER (TCC).

SUMMARY OF EVOLUTION TECHNOLOGY NEEDS

- Based on the trade-study determinations, and the logical evolutionary development, technology requirements are summarized.

- A recommended schedule for these development efforts is provided in Section 7.

SUMMARY OF EVOLUTION TECHNOLOGY NEEDS

(1) ACCELERATE DESIGN, DEVELOPMENT, AND TESTING OF:

- 120 VDC Ni - H_2 BATTERIES AND CONTROLS
- POWER ELECTRONIC COMPONENTS FOR
  - POWER REGULATION
  - POWER SWITCHING
  - FAULT CONTROL/ISOLATION
  - EMC/EMI CONTROL
- STANDARD CONVERSION UNITS (LRUs) FOR 115 VDC TO 28 VDC – 250, 500, 1000, 2500 WATTS

(2) ENCOURAGE PAYLOAD DEVELOPMENTS TO USE 115 VDC AS PRIME POWER, ESPECIALLY ANY TWTA* TYPE DEVELOPMENTS OR FURNACES/HEATERS

(3) ENCOURAGE/INVEST IN LOW-COST PROCESSING TECHNIQUES FOR SOLAR CELL FABRICATION AND PANEL ASSEMBLY

(4) DEMONSTRATE ADVANCED TYPES OF DEPLOYMENT MASTS

- ARTICULATED – STEEL OR COMPOSITE
- OTHER

(5) CONTINUE TECHNOLOGY EFFORTS FOR:

- SOLAR CELL ASSEMBLY IMPROVEMENTS
  - RADIATION
  - THERMAL
  - EFFICENCY
- BATTERY CELL LIFE CYCLE DEMOSTRATION
- COMPOSITE MATERIAL USE FOR
  - SOLAR ARRAYS
  - MASTS
  - SUPPORT STRUCTURE

*TWTA = TRAVELING WAVE TUBE AMPLIFIER

3.3 THERMAL CONTROL

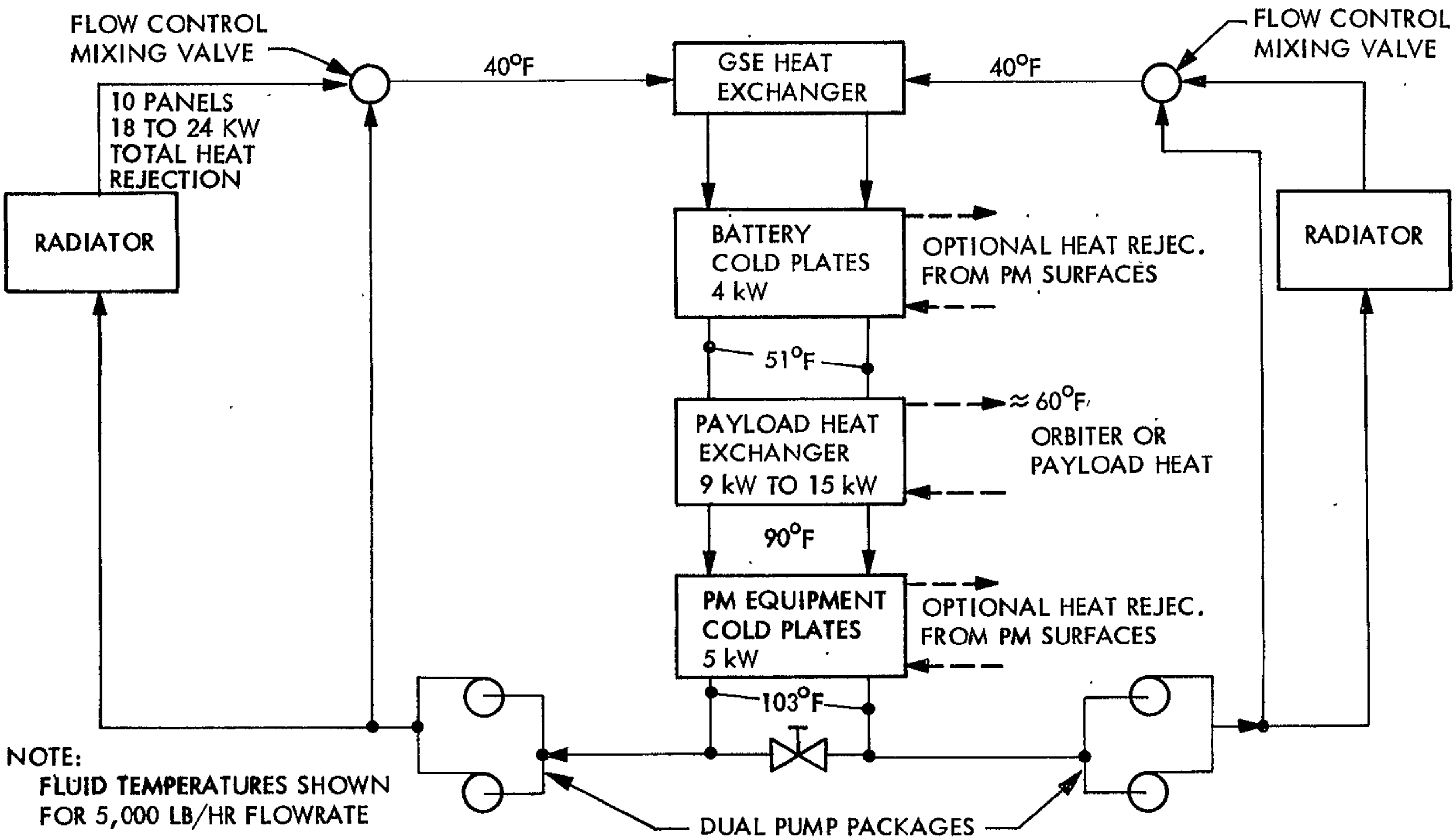
The Thermal Control Subsystem (TCS) discussions present the payload heat exchanger and control concepts, the radiator design, and heat rejection growth options. Some payload heat rejection requirements are unique which may be better served with a TCS optimized to these requirements. Additional capability may be provided in the 25 kW PM baseline TCS at the risk of being under utilized by some payloads.

POWER MODULE COOLANT LOOP

- The Power Module thermal control subsystem is based on a Freon 21 pumped coolant loop and deployable radiators for rejecting heat. The thermal control subsystem (TCS) is designed to maintain Power Module temperatures within limits while simultaneously providing some heat rejection capability for the attached payloads.

- Extensive use of coolant loop components and technology developed for the Shuttle Orbiter have been implemented. A hybrid heat pipe fluid loop header radiator design has been selected to replace the existing Orbiter all-fluid radiators. This substitution was chosen to provide reliability for radiators exposed to the potentially hazardous micro-meteoroid environment.

POWER MODULE COOLANT LOOP



NOTE:
FLUID TEMPERATURES SHOWN FOR 5,000 LB/HR FLOWRATE

THERMAL DESIGN TRADE STUDY SUMMARY

- The facing table summarizes the major thermal design trade studies performed during the Power Module evolutionary study. Items 1 and 3 (flat vs curved panels and heat pipe size) were analyzed during the Part II phase of the study. Flat radiators were found to be more efficient with respect to thermal, manufacturing, maintenance, and growth. Small heat pipes (1/4 inch) offered a weight advantage over larger pipes when used in the radiator panel design.

- The remaining items listed in the table were analyzed in more depth during the Part III study phase and the results are described in the following charts in detail. Engineering memoranda* have been written on the analysis approach and results for the fluid loop and heat pipe radiator design criteria and radiator flow circuitry. Technical assistance has been provided by Vought Corporation in the radiator design trade-offs and Hamilton Standard has provided assistance in the payload heat exchanger design and flow control techniques.

*Refer to Engineering Memoranda, EM 1.2.2 – C-101 thru -105, and -107. See listing in Section 8.2.

THERMAL DESIGN TRADE STUDY SUMMARY

TRADE	SELECTION	ADVANTAGES	DISADVANTAGES
1. RADIATOR SHAPE	FLAT RADIATOR PANELS	HIGH THERMAL EFFICIENCY	NEW DESIGN
2. HEAT PIPE VS. FLUID FLOW	HEAT PIPE	NON-CATASTROPHIC WITH METEOROID PUNCTURE OF A HEAT PIPE	5% LESS EFFICIENT
3. HEAT PIPE SIZE SELECTION	1/4-INCH HEAT PIPE	LESS TOTAL RADIATOR WEIGHT	MORE HEAT PIPES REQUIRED
4. RADIATOR LOCATION	EXTENDED FROM SOLAR ARRAY TRUSS STRUCTURE ALONG Z-AXIS	HIGHER THERMAL EFFICIENCY IMPROVED MAINTENANCE/ REFURBISHMENT	NONE
5. HEAT REJECTION GROWTH	• ADD 10 PANELS FOR 50 kW • ADD 10 PANELS PLUS ROTATION CAPABILITY FOR 100 kW	MAINTAIN THERMAL, MAIN-TENANCE, REFURBISHMENT EFFICIENCY OF THE 25 kW DESIGN	LARGE CENTER-OF-GRAVITY OFFSETS FROM PM CENTERLINE
6. RADIATOR FLUID FLOW CIRCUITRY	FREON CHANNELS IN CENTER OF PANEL	PROVIDES SHORTER HEAT PIPE CONDENSER LENGTHS, ALLOWING LIGHTER RADIATOR WEIGHTS	NONE
7. PAYLOAD HEAT EXCHANGER DESIGN	MODIFY TWO ORBITER PAYLOAD HEAT EX-CHANGERS TO ACCEPT 4 PAYLOADS	MODIFIED RATHER THAN NEW DESIGN AND CAN CONTROL PAYLOADS HEAT REJECTION	WILL REQUIRE ADDI-TIONAL PERFORMANCE TESTING
8. ADDITIONAL PAYLOAD HEAT REJECTION GROWTH	PROVIDE A THERMAL MODULE KIT	CAN BE TAILORED TO PAYLOAD TEMPERATURE REQUIREMENTS	REQUIRES A SEPARATE COOLANT LOOP AND RADIATOR SYSTEM

RADIATOR LOCATION & SIZE TRADE STUDIES

- The thermal analysis of radiator heat rejection capability for the MSFC baseline configuration was performed during Part II tasks and the results were documented in the final report. A new radiator configuration was developed during Part III based on improving both the thermal performance and reducing the impact on Power Module and payload field of view requirements. The modified 25 kW Power Module radiators are shown as configuration 25-1 in the table and the following chart. The recommended radiator location was found to provide 9 kW heat rejection for the internal Power Module components and batteries plus 9 to 16 kW cooling for payloads. The variations in payload cooling capacity results from variations in the solar and earthshine loads associated with spacecraft orbit and attitude changes. The radiator area for configuration 25-1 was selected to provide 70 $\pm 30^{\circ}F$ component temperature levels. The 675 ft^2 is seven (7) percent larger than the MSFC baseline and provides twenty (20) percent more cooling capacity.

- Three 50 kW configurations were evaluated to determine the optimum radiator system growth in capacity. The performance and description of these configurations are shown in the table. The 50-1 system was selected based on minimum modifications to the existing 25 kW TCS design. Configuration 50-2 provides an identical area with an extension and rotation capability to improve performance. The additional weight and cost was not considered to be justified by the twelve percent performance improvement. A third configuration, 50-3 added ten additional panels and support structure to the 25-1 design by extending the panels in the opposite direction. This approach was not chosen because the heat rejection growth was limited, excessive mounting hardware was required, and the new panels contributed to Orbiter docking and sensor field-of-view problems.

- The heat rejection capability of the 1350 ft^2 of radiators on 50-1 was reduced approximately 8 percent with the solar array growth to 100 kW. The internal heat generation within the PM was conservatively estimated to be 36 kW. The net result of the decreased performance and increased PM heat rejection requirements was a 2 kW payload cooling capacity. To provide some payload cooling, three performance improvement designs were analyzed. The 100-1 radiator design shown allows the panels to be rotated so that they remain at the outer edge of the Solar Array blankets regardless of blanket position. This concept maintains the 1350 ft^2 radiator area of the 50-1 kW design and provides 12 kW of payload cooling capacity. This configuration was recommended as it provides thermal control for the manned habitat scheduled for 1988. A 10 kW heat rejection requirement was estimated for a 3-man occupancy level.

- Alternate 100 kW radiator designs listed in the table, included a second degree of rotation to keep the panels edge lit by the sun. Configuration 100-2 incorporates this feature with a resulting 18 kW payload cooling capacity. Adding additional panels to the 20-panel basic arrangement was considered but not accepted based on uncertainties in increasing the fluid line couplings and additional pressure drop effects on the overall performance. Configuration 100-3 and 100-4 shows that approximately 2.5 kW per panel are achievable with additional panels. Configuration 100-5 includes a maximum 10 kW heat rejection from the PM external surfaces. This reduces the load on the Freon loop and radiators thereby allowing a larger P/L budget. PM surface heat rejection increases the thermal control design complexity and was not recommended for the baseline vehicle.

RADIATOR LOCATION AND SIZE TRADE STUDIES

POWER	CONFIG-URATION	FREON-21 MASS FLOW RATE (LB/HR)	HEAT REJECTION (kW)			FREON TEMP (°F)		RADIATOR AREA (FT^2)	DESCRIPTION NO. OF PANELS/ CONFIGURATION
			PM	P/L(1)	TOTAL(2)				
25 kW	25	5,000	9	13.1	22.1 ± 4	58	40/98	675	10
50 kW	50-1	10,000	18	22.8	40.8 ± 5	54	40/94	1,350	20
	50-2	10,000	18	27.7	45.7 ± 3	60	40/100	1,350	20 EXTENSION + ROTATION
	50-3	10,000	18	19.0	37.0 ± 5	49	40/89	1,350	20 2-10 PANEL STRINGS
100 kW	100-0	10,000	36	2.0	38.0 ± 4	50	40/90	1,350	20
	100-1	10,000	36	12.4	48.4 ± 6	64	40/104	1,350	20 EXTENSION + ROTATION
	100-2	10,000	36	18.0	54.0 ± 2	71	40/111	1,350	20 EXTENSION + 2-DEGREE ROTATION
	100-3	10,000	36	12.0	48.0 ± 5	63	40/103	1,620	24
	100-4	10,000	36	22.0	58.0 ± 6	76	40/116	1,890	28
	100-5	10,000	26	12.0	38.0 ± 4	50	40/90	1,350	20 W/10 kW FROM PM SURFACES

NOTES: (1) HEAT REJECTION AVAILABLE TO PAYLOAD MAY BE USED FOR MANNED HABITAT CONTROL. 10 kW ESTIMATED FOR 3-MAN OCCUPANCY.
(2) TOTAL HEAT REJECTION INCLUDES EFFECTS OF BETA ANGLE, SUN INCIDENT UP TO 30°, AND EARTHSHINE.

[ ] INDICATES SELECTED THERMAL CONTROL RADIATOR CONFIGURATIONS

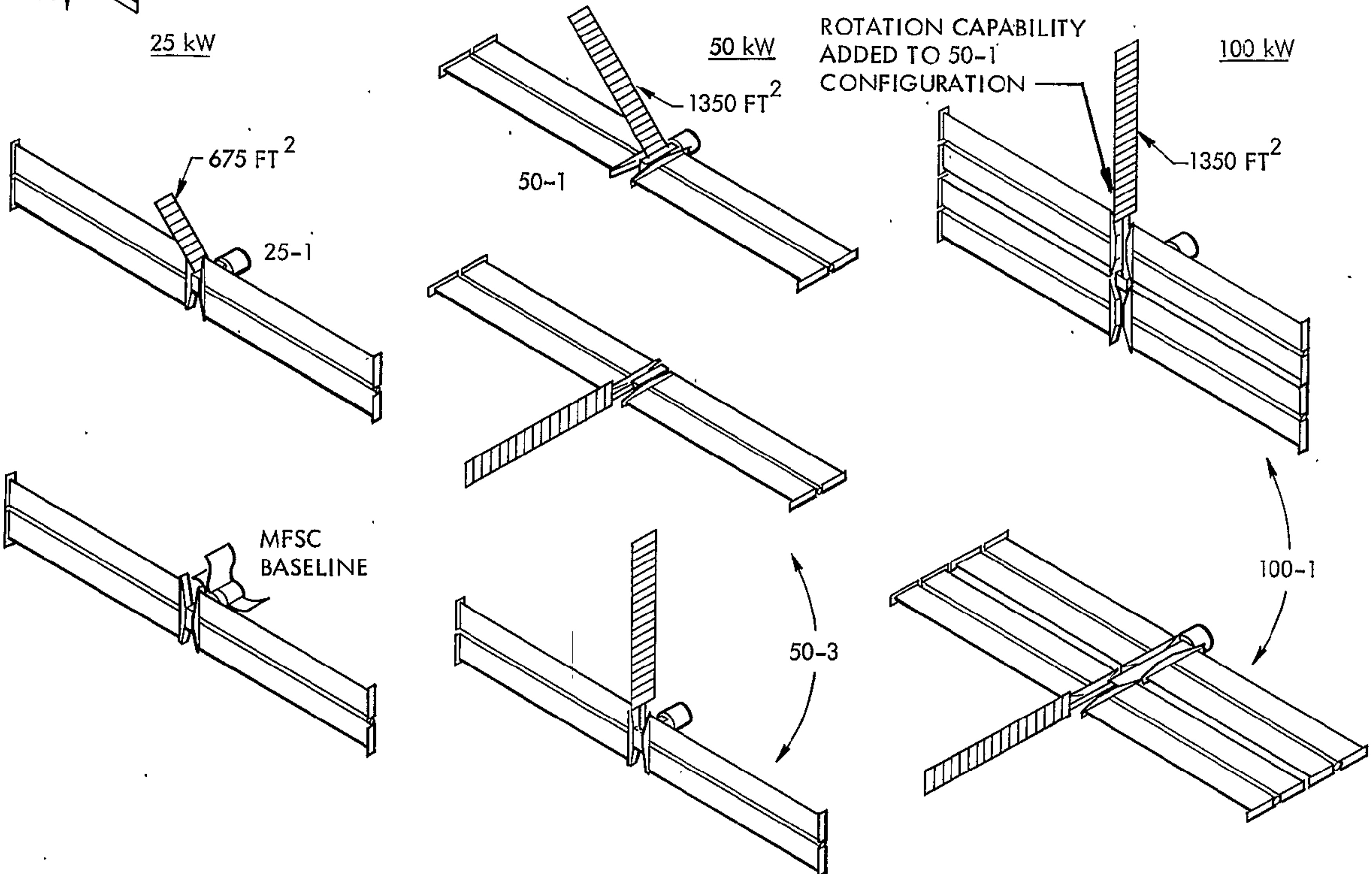
RADIATOR SIZE AND LOCATION SUMMARY

- The figure shows some of the radiator locations which were analyzed during the Phase III tasks of the Evolution Study. The configurations 25-1, 50-1 and 100-1 were recommended locations to provide all of the PM heat rejection requirements and 10 kW or more payload heat rejection. These and other radiator configuration are listed in the previous table and their performance characteristics are compared.

- This study showed that for the 50 kW and 100 kW Power Modules, it would be advantageous to provide payload cooling with a dedicated control system-either payload or Power Module supplied. The reason being the decentralized location of some payloads and widely varying temperature level requirements. Additional radiators could be added to the basic Power Module but the long coolant loop lines and $70^{\circ}F$ coolant temperatures may not provide the optimum control design. For example, a payload which could reject heat at $170^{\circ}F$ would require one-half the radiator area of an equivalent addition to the basic Power Module system.

- The 10 kW excess PM radiator capability available for payloads was considered a minimum level. The manned habitat planned for the 50 and 100 kW configurations would require 10 kW at approximately $70^{\circ}F$ and would be attached to the Power Module spacecraft.

RADIATOR SIZE & LOCATION SUMMARY



ALTERNATE RADIATOR FLOW CIRCUITS

- Some potential heat pipe radiator configurations are shown. The assumption in all cases is that the radiators are initially folded over each other (except configuration D) and deploy away from the spacecraft. In all cases the initial heat delivery to the panel is by circulating liquid, because the heat load is excessive for heat pipes. Details of this trade study are described in EM C-1.2.2-104 (See listing in Section 8.2).

- Configuration A has a heat distributor (manifold) on one side, to which the heat pipes are attached. All the fluid flowing through the radiator array passes through the distributor before leaving for the adjacent radiator (or radiators). The returning fluid flows through the pipe at the left. The designation "distributor" is used to distinguish it from the manifold in the all-liquid radiator. Configuration B has a heat distributor along the "long side".

- Configuration C is similar to Configuration B, except that all the radiator distributor flow paths are in series.

- Configuration D is the same as Configuration B except that the panels are deployed adjacent to each other. The flow pattern is similar to that of the shuttle Orbiter. The baseline PM structure would not be appropriate for this configuration.

- Configuration A has the shortest distributor length. The heat pipes become quite long however, depending upon panel length, maybe beyond the point where they can be used most efficiently. Also, the liquid cools off as it flows away from the spacecraft but on its return flows through progressively hotter zones (unless the return pipe is insulated from the panel). This degrades the heat rejection capability of the system.

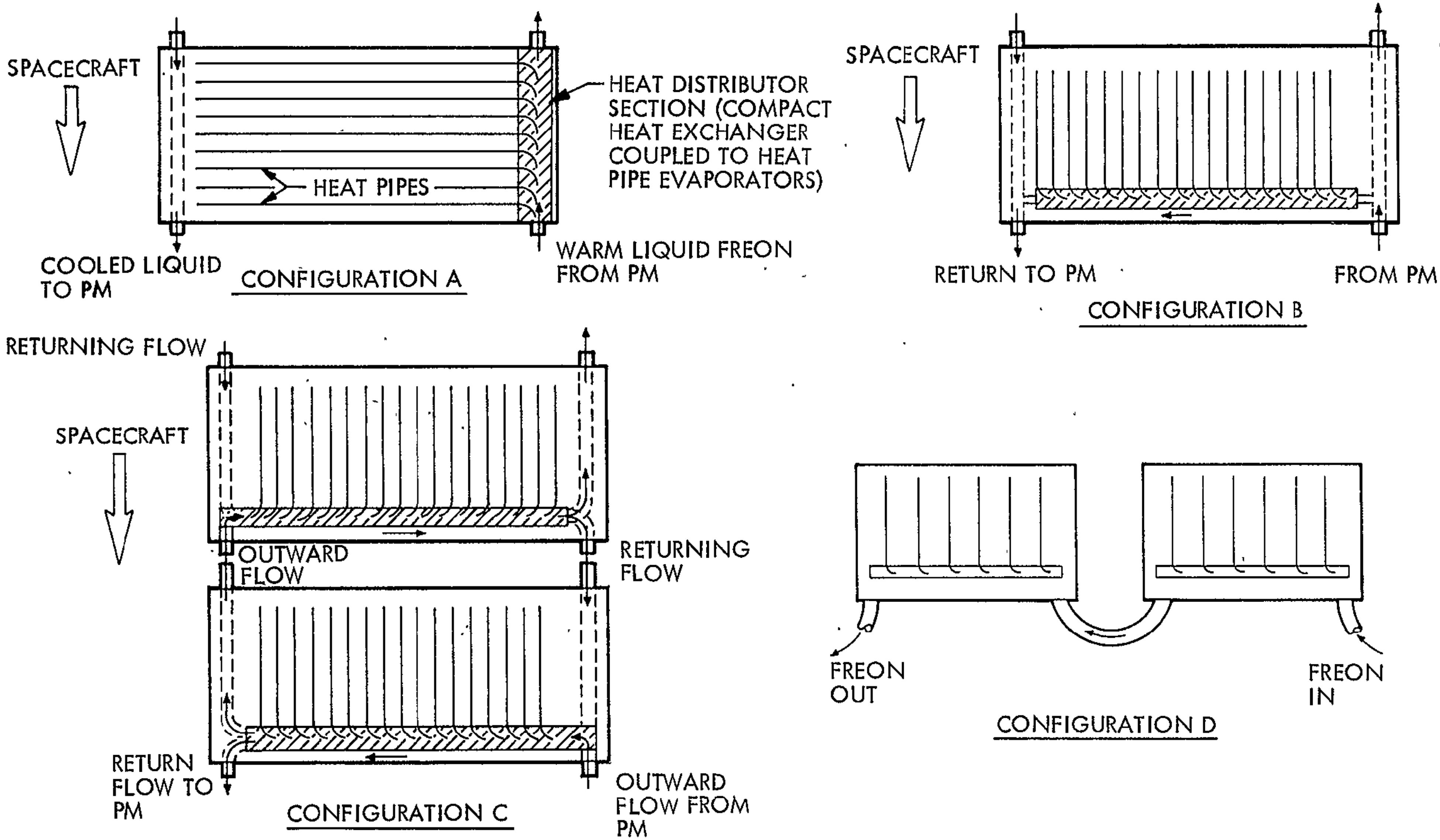
- The heat pipes can, in effect, be made half their actual length by placing the distributor in the middle of the panel. However, this does not avoid the problem of returning fluid being reheated (again, unless it is insulated).

- In Configuration B the temperature decreases from right to left and the fluid returning from the outer panels is at a temperature similar to that at the inner panels. The split fluid flow, however, means that local velocities within each distributor, and therefore heat transfer coefficients, are lower than would be the case if the entire flow passed through the distributor.

- In Configuration C the entire flow goes through each distributor, but since each radiator is progressively colder in the outward direction, the return fluid suffers from the same difficulty as in Configuration A (the return pipe can traverse the center of each panel, rather than as shown, but this does not remove the basic problem of adjacent fluids at different temperatures.

- Configuration D does not have the return temperature problem and does have full fluid flow through the distributor, but requires a different structural layout.

ALTERNATE RADIATOR FLOW CIRCUITS

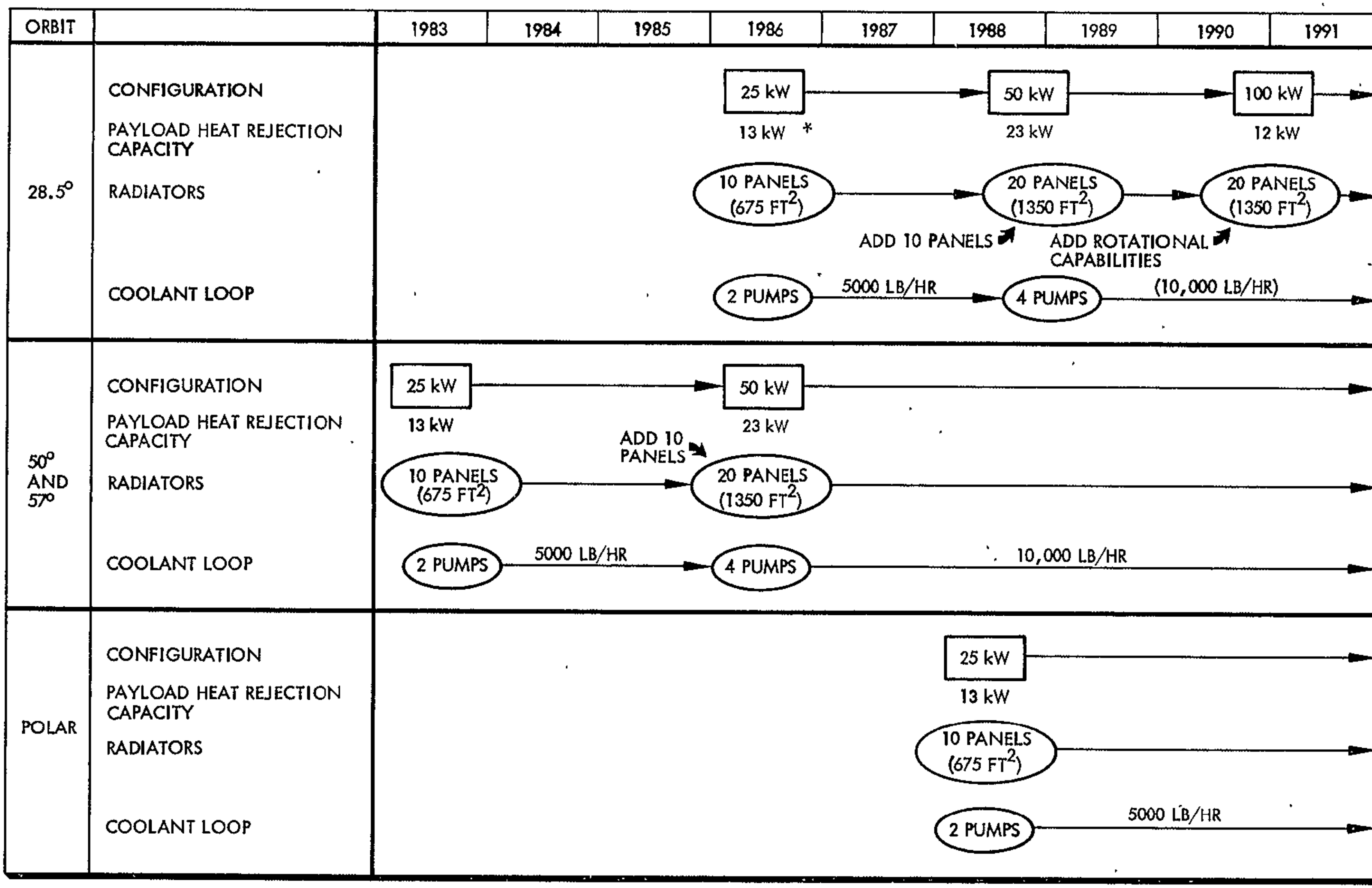


THERMAL CONTROL SUBSYSTEM GROWTH SCENARIO

- The following two figures show the nominal growth scenarios for the Power Module thermal control subsystem. The heat rejection levels which are shown correspond to the average payload capacity at all spacecraft attitudes.

- An optional thermal control module is shown in the second figure with the 50 and 100 kW Power Module configurations. These modules can provide additional cooling for payloads with limited access to adequate heat rejection surfaces. It is anticipated that the user payloads will have widely varying thermal control requirements and desired temperature ranges. Therefore, it is recommended that additional payload cooling be provided by thermal control systems optimized for the user's requirements.

THERMAL CONTROL SUBSYSTEM GROWTH SCENARIO



*FREE-FLYING MODE

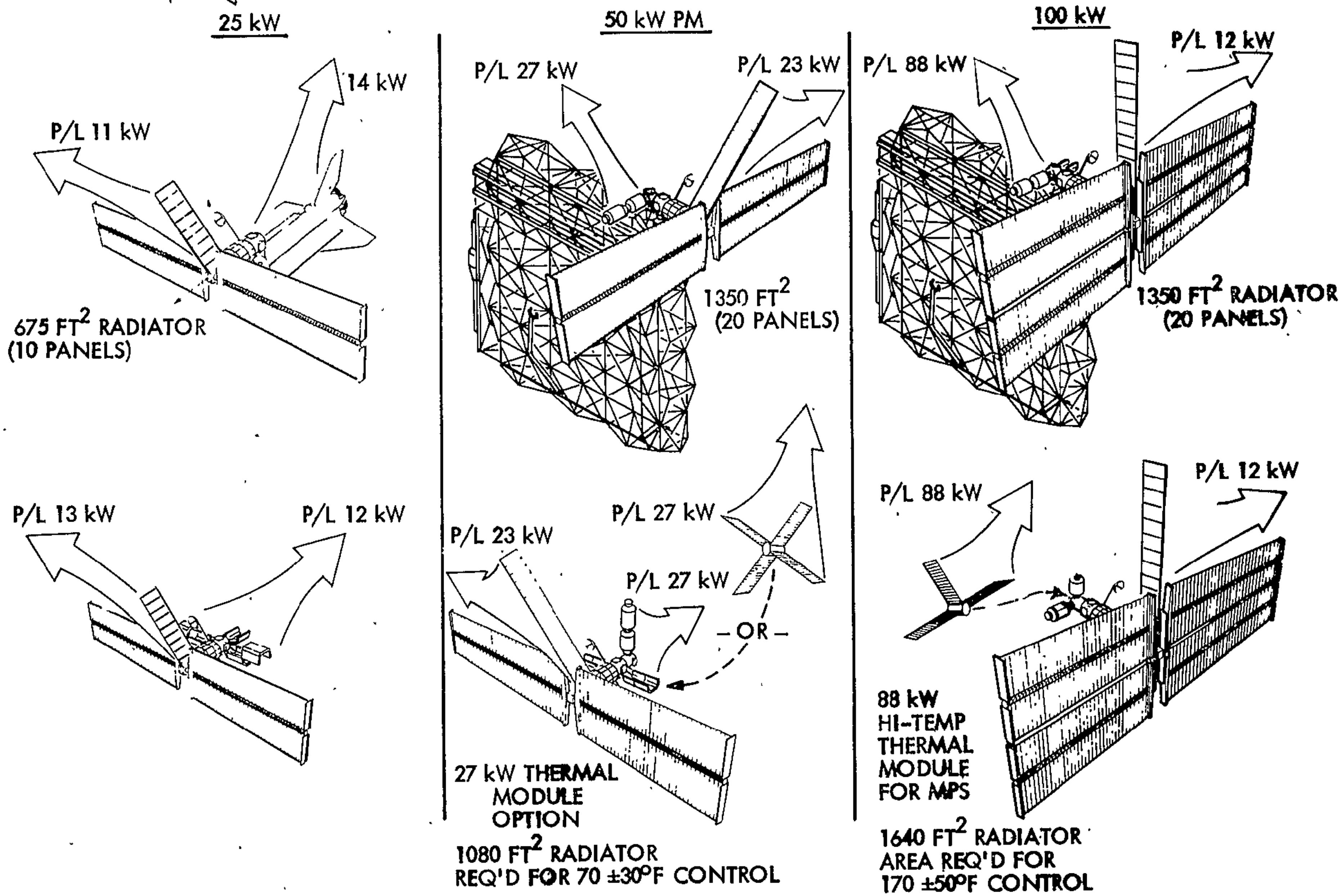
POWER MODULE HEAT REJECTION GROWTH

- This figure illustrates the power module evolutionary growth defined in the previous figure. The heat rejection requirements for the Power Module and the excess capability available for payload cooling are shown.

- As pointed out previously, it was considered advantagious in some cases for additional payload cooling to be provided by individual systems tailored to specific payload operational and temperature requirements. However, the basic Power Module heat rejection capacity could readily be increased by adding more radiator area, rejecting heat at higher temperatures or providing additional radiator positioning control to keep the panels edge-lit by the sun.

- A thermal control module is shown as an optional equipment item. The design of this module would be patterned after the design of the Power Module thermal control system.

POWER MODULE HEAT REJECTION GROWTH



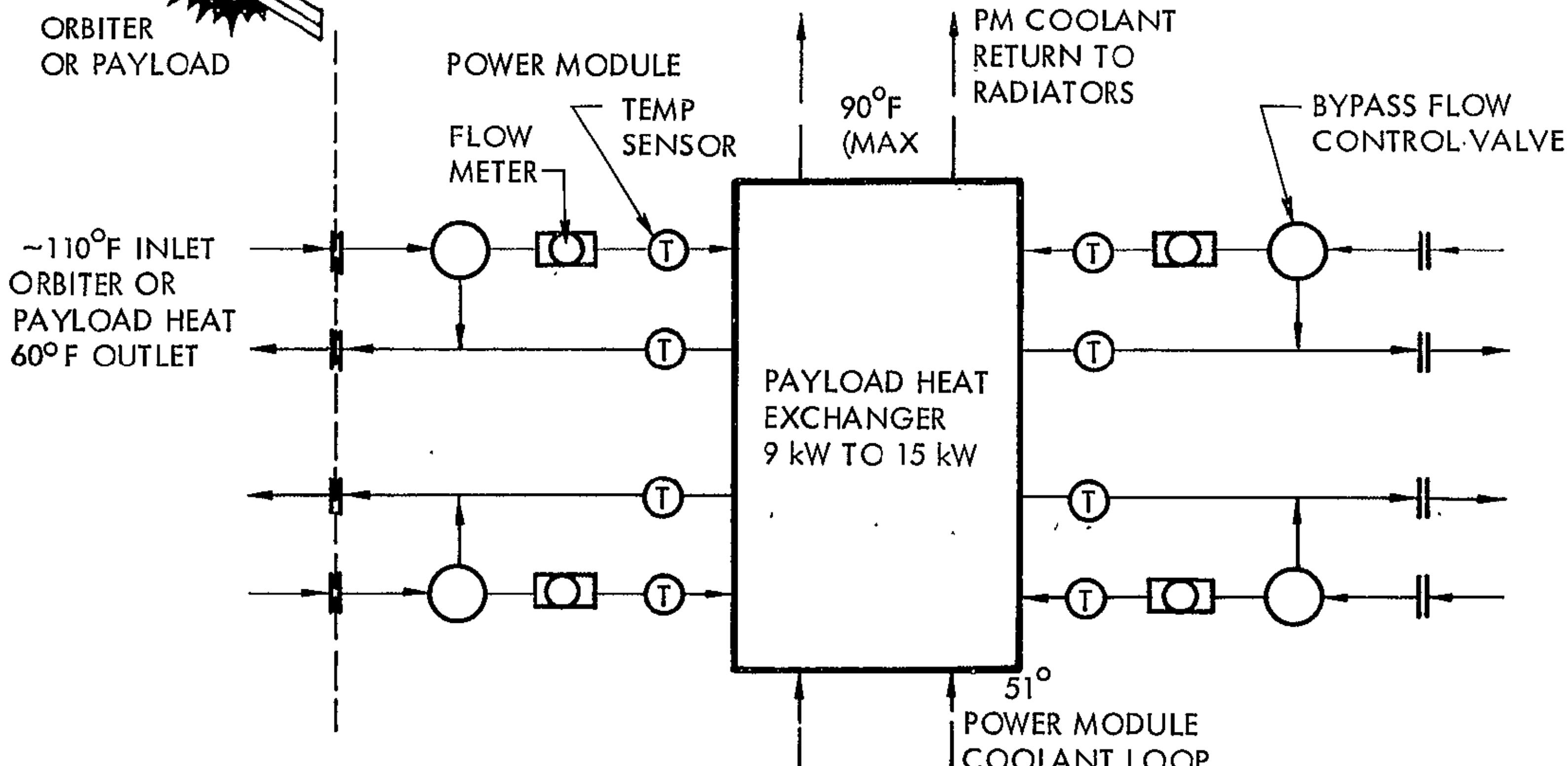
PAYLOAD HEAT EXCHANGER INTERFACE

- The payloads will reject heat from the Power Module through the payload heat exchanger in the PM coolant loop. The payload heat exchanger design is based on the payload heat exchanger developed by Hamilton Standard for the Orbiter. This unit is modified to accept up to four separate payloads with each payload varying in cooling requirements and priority.

- Since each payload may have different cooling requirements, an interface control system designed to prevent one or more payloads from inadvertently exceeding the Power Module payload heat exchanger 15 kW capacity is required. Alternative control concepts have been provided by Hamilton Standard. However, the one shown was selected to provide the greatest flexibility and hardware simplicity. This system is designed to limit each payload to its predetermined heat rejection needs. The amount of heat being extracted from the payload coolant loop is monitored by measuring the coolant loop inlet and outlet temperatures and flow rates. When the heat being extracted exceeds the capacity of the payload heat exchanger, the bypass control valve will divert an appropriate amount of flow back to the payload before encountering the payload heat exchanger. The payload coolant loop(s) to be diverted will be determined on a priority basis.

- The heat exchanger shown combines two orbiter payload heat exchanger units which are modified to interconnect the heat transfer functions. Some new tooling and verification testing would be required; however, manifold connections are identical to existing design. Another concept also used two existing fuel cell heat exchangers which would operate in parallel. No additional qualification testing would be required; however, six additional flow control valves in the PM lines would be required to provide the flexibility of the recommended payload interface shown.

PAYLOAD HEAT EXCHANGER INTERFACE



INTERFACE ALTERNATIVES	ADVANTAGES	DISADVANTAGES
SINGLE HEAT EXCHANGER	MAXIMUM PAYLOAD HEAT LOAD CONTROL FLEXIBILITY	SOME NEW TOOLING AND TESTING REQUIRED
TWO ORBITER FUEL CELL Hx CONNECTED IN PARALLEL	"OFF THE SHELF" Hx HARDWARE	ADDITIONAL VALVES REQUIRED FOR INTERFACE CONTROL

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THERMAL CONTROL SUBSYSTEM CHARACTERISTICS

LAUNCH DATE	1983	1986	1987	1988/89	1990/91
FLIGHT VEHICLE	FV-1	FV-2	FV-3 (5)	FV-4/-5	FV-6
POWER (kW)	25	25	60	50	100
RADIATOR AREA (FT^2)	675	675	675	1,350	1,350
RADIATOR PANELS	10	10	10	20	20
FREON-21 FLOW RATE (LB/HR)	5,000	5,000	5,000	10,000	10,000
HEAT REJECTION CAP.(1)(kW)	22.1 ±4	22.1 ±4	27.0 ±2	40.8 ±5	48.4 ±6
PM REQUIREMENTS (kW)	9	9	5	18	36
AVAIL. FOR PAYLOADS (kW)	13.1	13.1	22.0	22.8(2)	12.4(2)
WEIGHTS – (LB)(4)					
RADIATOR PANELS	945	945	945	1,890	1,620
P/L HEAT EXCHANGER AND CONTROL	120	120	240	240	120
REMAINING COMPONENTS	949	949	949	989	1,139
TOTAL	2,014	2,014	2,134	3,119	2,879

NOTES:

1) HEAT REJECTION INCLUDES EFFECTS OF BETA ANGLE, SUN INCIDENT AT 30° AND EARTHSHINE VARIATIONS.
2) 10 kW COOLING REQUIRED TO SUPPORT 3-MAN HABITAT MODULE.
3) RADIATOR DEPLOYMENT MECHANISM INCLUDES RADIATOR EXTENSION HARDWARE.
4) ESTIMATED WEIGHTS, CONTINGENCY VALUES NOT INCLUDED.
5) THIS MAY BE TAILORED TO THE GEO PLATFORM, AND MAY REQUIRE NO HEAT REJECTION ELEMENTS FROM THE POWER MODULE.

3.4 ATTITUDE CONTROL

The Attitude Control Subsystem utilizes a complement of rate gyros, sun sensors and horizon sensors, modified ATM Control Moment Gyros, in conjunction with computational capability in the Command and Data Handling System to provide primary attitude control. A magnetic torquing system, which uses Space Telescope hardware, provides a contingency stabilization mode for retrievals and also provides additional desaturation capability.

ATTITUDE CONTROL SUBSYSTEM TRADES-SUMMARY

- The chart presents a summary of four attitude control subsystem trades which are described in the following pages.

ATTITUDE CONTROL SUBSYSTEM TRADES—SUMMARY

ELEMENT OR COMPONENT	NUMBER OF CANDIDATES	CONFIGURATION SELECTED		SELECTION RATIONALE
		INITIAL 25 kW	GROWTH	
ATTITUDE SENSOR	3	ITHACO CSA-9530 HORIZON SENSOR	SAME	LOWEST COST; GOOD RELIABILITY; GOOD ACCURACY
DESATURATION TECHNIQUE	3	GRAVITY GRADIENT MANEUVERING	SAME	NO ADDITIONAL HARDWARE REQUIRED
SORTIE MODE CMG REQUIREMENTS	N/A	3 CMGs	6 CMGs FOR 100 kW PM	ONLY 3 CMGs REQUIRED
CONTINGENCY STABILIZATION	3	MAGNETIC TORQUING	SAME	NO CONSUMABLES REQUIRED; COMMONALITY WITH SPACE TELESCOPE HARDWARE

ATTITUDE POINTING SENSOR TRADE STUDY

- The requirement to be satisfied involves providing the free-flying Power Module with ability to point payloads to an accuracy of 0.5 degree. Since the wide angle sun sensing system provides a line of position to the sun with an accuracy of ±0.25 degree, it is necessary to determine another line of position with similar accuracy.

- Three candidates, two horizon scanners and the NASA Standard Fixed Head Star Tracker, were considered as indicated on the facing page. Data used in the trade study are presented in the following pages.

- For the horizon scanners it is necessary to provide a view capability along the x-axis to accommodate local vertical orientations, and perpendicular to the x-axis for inertial orientations.

- The Ithaco system was chosen, principally because of low cost.

ATTITUDE POINTING SENSOR TRADE STUDY

REQUIREMENT: PROVIDE A CAPABILITY TO POINT THE POWER MODULE TO ±0.5 DEGREES.

CANDIDATES	ADVANTAGES	DISADVANTAGES
HORIZON SENSOR (BARNES 13-166)	• EXTENSIVE FLIGHT EXPERIENCE • GOOD ACCURACY	LOW RELIABILITY HIGH COST
HORIZON SENSOR (ITHACO CSA 9530)	• GOOD ACCURACY • GOOD RELIABILITY (WITH REDUNDANCY) • LOW COST	• NEW DESIGN, TO BE AVAILABLE IN 1979
STAR TRACKER (BALL NASA STANDARD FIXED HEAD STAR TRACKER)	• HIGH RELIABILITY • BEST ACCURACY	• ACCURACY EXCEEDS REQUIREMENT • HIGH COST • DIFFICULT TO PHYSICALLY INTEGRATE

RECOMMENDATION: USE ITHACO HORIZON SENSOR

RATIONALE: LOWEST COST: GOOD RELIABILITY: GOOD ACCURACY

ATTITUDE POINTING COMPONENT DATA—WITH NO REDUNDANCY

- The parameters used in the trade study for each of the candidate components are shown in the chart.
- All data provided are for a single-thread system with no redundancy.

ATTITUDE POINTING COMPONENT DATA—WITH NO REDUNDANCY

MANUFACTURER	DESCRIPTION	TYPE	ACCURACY 3Ω	SIZE	WEIGHT	POWER	RELIABILITY	COST RECURRING	REMARKS
BARNES	13-166	CONICAL SCAN HORIZON SENSOR	±0.15°	HEADS: 6 IN DIA X 5.3 IN L ELECTRONICS 3 1/8 IN. X 8 1/2 IN. X 10 7/8 IN.	HEADS: 5.5 LB ELECTRONICS 9.5 LB	12.5 W PER SYSTEM	MTBF = 18245 HR PER SYSTEM	$400K PER SYSTEM	1 SYSTEM = 2 HEADS + 1 ELECTRONIC UNIT
ITHACO	CSA-9530	CONICAL SCAN HORIZON SENSOR	±0.25°	HEADS: 3.6DIA X 5L ELECTRONICS. 6.4 IN. X 6.4 IN. X 5.6 IN.	8.5 LB PER SYSTEM	10 W PER SYSTEM	MTBF = 58000 HR PER SYSTEM	$280K PER SYSTEM	1 SYSTEM = 2 HEADS +1 ELECTRONIC UNIT UNIT CAN ACCOMMODATE BOTH XPOP AND LV
BALL	NASA STD FHST	FIXED HEAD STAR TRACKER	±.0083°	3.5 FT^3 VOL PER FHST & SUN SHADE	2.9 LB PER FHST & SUN SHADE	18 W PER FHST	MTBF = 187000 HR PER SYSTEM	$300K PER FHST & SUN SHADE	3 FHST REQUIRED DUE TO ANTI-SOLAR REQUIREMENT

ATTITUDE POINTING COMPONENT DATA—WITH REDUNDANCY

- The data for the candidate components are presented for a vehicle complement to provide a basis for comparison.
- The Ithaco sensor was chosen because of lowest recurring cost and acceptable reliability.

ATTITUDE POINTING COMPONENT DATA– WITH REDUNDANCY

MANUFACT – URER/MODEL	TYPE	ACCURACY 3Ω	VOLUME/ WEIGHT	POWER	RELIABIL – ITY	COST RECURRING	REMARKS
BARNES (13-166)		TWO SYSTEMS ALONG X-AXIS AND TWO SYSTEMS PERPENDICULAR TO X-AXIS ARE REQUIRED TO ACCOMMODATE XPOP AND LV MODES AND TO IMPROVE RELIABILITY.					
	CONICAL SCAN HORIZON SENSOR	±0.15°	1.36 FT^3 164 LB	12 W (ONLY ONE SYSTEM OPERATING AT ANY GIVEN TIME)	P_S (XPOP) = P_S (LY) = .74 (ASSUMES DUTY CYCLE SHARED EQUALLY	$1600K	AVAILABLE
ITHACO (C5A-9530)		TWO HEADS ALONG X-AXIS AND TWO HEADS PERPENDICULAR TO X-AXIS PLUS TWO ELECTRONICS PACKAGES ARE REQUIRED TO ACCOMMODATE XPOP AND XLV MODES AND TO IMPROVE RELIABILITY.					
	CONICAL SCAN HORIZON SENSOR	±0.25°	0.38 FT^3 17 LB	10 W	P_S = .978	$560K	AVAILABLE 9/1/79
BALL (NASA STD FHST)		THREE STAR TRACKERS (PLUS SUN SHADES) REQUIRED TO PROVIDE TWO LINES OF SIGHT AND ADEQUATE ANTI-SOLAR VIEWING.					
	FIXED HEAD STAR TRACKER	±.0083°	10.5FT^3 87 LB	54 W	P_S = .981	$1000K (INCLUDES 100K FOR SOFTWARE)	AVAILABLE

P_S BASED ON 5-YEAR MISSION WITH 50% DUTY CYCLE

CONTINGENCY STABILITY TRADE STUDY

- Three candidate systems were evaluated for their ability to provide a secondary stabilization system for the Power Module.
- An undamped Power Module will assume a local vertical attitude in approximately eighteen hours, but will still oscillate ± 30 degrees about that vertical, indefinitely.
- The magnetic torquing system was selected primarily because no consumables are required and the commonality of the hardware with that being utilized on Space Telescope.

CONTINGENCY STABILITY TRADE STUDY

REQUIREMENT: PROVIDE A CONTINGENCY STABILIZATION FOR RETRIEVAL

CANDIDATES	ADVANTAGES	DISADVANTAGES
GRAVITY GRADIENT	• NO ADDITIONAL HARDWARE REQUIRED	• >30 DEGREE OSCILLATION AFTER THE LOCAL VERTICAL IS OBTAINED
RCS	• MEETS REQUIREMENT • PROVIDES ADDITIONAL DESATURATION CAPABILITY	• ADDITIONAL HARDWARE REQUIRED • POTENTIAL PAYLOAD CONTAMINATION • CONSUMABLES REQUIRED
MAGNETIC TORQUING	• MEETS REQUIREMENT • PROVIDES ADDITIONAL DESATURATION CAPABILITY • SYSTEM HARDWARE AND SOFTWARE CURRENTLY BEING CONFIGURED FOR SPACE TELESCOPE • NO CONSUMABLES REQUIRED	• ADDITIONAL HARDWARE REQUIRED • POTENTIAL PAYLOAD CONTAMINATION (MAGNETIC) WHEN USED TO PROVIDE ADDITIONAL DESATURATION CAPABILITY

RECOMMENDATION: UTILIZE MAGNETIC TORQUING SYSTEM

RATIONALE: NO CONSUMABLES REQUIRED; COMMONALITY WITH SPACE TELESCOPE HARDWARE

C-3

CMG DESATURATION TRADE STUDY

- In order to provide a CMG desaturation capability in the free-flying mode, three candidate techniques were evaluated as shown on the facing page.

- Making small maneuvers so that gravity gradient torques would be improved on the vehicle was selected as imposing the least complexity, therefore lowest cost and risk.

- A magnetic torquing system has also been added to the vehicle to provide a contingency stabilization system for retrieval. Therefore, the magnetic torquing system will also be used for CMG desaturation in conjunction with the GG technique.

CMG DESATURATION TRADE STUDY

REQUIREMENT: PROVIDE CMG DESATURATION CAPABILITY IN THE FREE-FLYING MODE.

CANDIDATES	ADVANTAGES	DISADVANTAGES
GG MANEUVERING	• NO ADDITONAL HARDWARE REQUIRED	• NO RECOVERY FROM SATURATION
RCS	• SHORT TIME REQUIRED. PROVIDES BACK-UP ACS	• ADDITIONAL HARDWARE REQUIRED • POTENTIAL PAYLOAD CONTAMINATION
MAGNETIC TORQUING	• PROVIDES BACK-UP ACS • SYSTEM HARDWARE AND SOFTWARE CURRENTLY BEING CONFIGURED FOR SPACE TELESCOPE	• ADDITIONAL HARDWARE REQUIRED • SENSITIVE TO ORIENTATION WITH RESPECT TO EARTH'S MAGNETIC FIELD • POTENTIAL PAYLOAD CONTAMINATION (MAGNETIC)

RECOMMENDATION: USE GG MANEUVERING

RATIONALE: NO ADDITIONAL HARDWARE IS REQUIRED

SORTIE MODE CMG REQUIREMENTS

- An analysis was performed to determine the number of CMGs required to control the Power Module/Orbiter in the Sortie Mode for a number of orientations, both inertial and local vertical.
- The results are as shown on the facing page.

SORTIE MODE CMG REQUIREMENTS

OBJECTIVE: DETERMINE NUMBER OF CMGs REQUIRED VS ORIENTATION FOR SORTIE MODE

REQUIREMENT: CAPABILITY TO MANEUVER THE ORBITER/POWER MODULE TO ENHANCE EXPERIMENT VIEWING

RESULTS:

ORIENTATION OPTIONS	NUMBER OF CMGs REQUIRED
X POP, INERTIAL	3
X LOCAL VERTICAL	3
Y LOCAL VERTICAL	3
Z LOCAL VERTICAL	3
X IOP, INERTIAL (Y OR Z, POP)	4
Z POP, Y PERPENDICULAR TO SUN LINE	4
X IOP, INERTIAL (Y OR Z, 45 DEGREES TO ORBIT PLANE)	5
X MORE THAN 30 DEGREES TO ORBIT PLAN (Y OR Z, IOP)	IMPOSSIBLE

RECOMMENDATION: FLY X POP, INERTIAL, OR ANY PRINCIPAL AXIS ALONG THE LOCAL VERTICAL

RATIONALE: ONLY 3 CMGs REQUIRED

3.5 COMMUNICATION & DATA HANDLING

C&DH TRADES/SELECTIONS

- This chart identifies the trade studies undertaken for selecting major C&DH subsystem components.
- Selections are based on providing cost-effective evolutionary growth to accommodate the Power Module configurations to 1991.
- All systems considered are the distributed bus type, to minimize docking interface wiring with multiple docking parts.

C & DH TRADES/SELECTIONS

	CANDIDATE		SELECTION
ANTENNA	BALL BROTHERS, AN 128 (ELECTRONICALLY STEERED)		
	STEERABLE DISH		STEERABLE DISH
	FIXED DISH		
COMPUTER	NSSC-I		
	NSSC-II		NSSC-II
SYSTEM CONFIGURATION	NASA STD (MMS)	CANDIDATE 1	SELECTION TO BE BASED ON AN ANALYSIS OF COMPETITIVE PROPOSALS DURING PHASE B
	SPERRY FMDM*	CANDIDATE 2	
	SCI DACS	CANDIDATE 3	
	HIGH DATA RATE SYSTEM	OPTION	

*FMDM = FLEXIBLE MULTIPLEXER/DE-MULTIPLEXER

C&DH TRADE STUDIES: SUBSYSTEM ADVANTAGES AND DISADVANTAGES

- This chart summarizes the advantages and disadvantages of two major C&DH subsystem component trades.
- Data management alternatives which meet Power Module requirements are identified on the preceding chart.

C&DH TRADE STUDIES: ADVANTAGES AND DISADVANTAGES

EQUIPMENT	CANDIDATE	ADVANTAGES	DISADVANTAGES
ANTENNA	BALL BROTHERS	• NO MOVING PARTS • DESIGNED FOR TDRS	• GAIN TOO LOW NOT MOUNTABLE ON PM
	FIXED DISH	• MECHANICALLY SIMPLE	• VEHICLE ORIENTATION REQUIRED FOR POINTING
	STEERABLE	• OPTIMIZES CONTACT WITH TDRS	• SOFTWARE REQUIRED FOR POINTING • MAY REQUIRE STAR TRACKERS ON PM
COMPUTER	NSSC-I	• LOW POWER • LOW DEVELOPMENT RISK • NON VOLITILE MEMORY	• DOES NOT MEET BASIC REQUIREMENTS PLUS GROWTH CAPABILITY • 64K WORDS MAXIMUM
	NSSC-II	• GOOD GROWTH CAPABILITY • HIGHER SPEED THAN NSSC-I • 56K WORDS BASIC EXPANDABLE TO 512K WORDS	• HIGHER POWER AND WEIGHT THAN NSSC-II • VOLATILE MEMORY • DEVELOPMENT RISK FOR STINT II

COMPUTER SYSTEM COMPARISONS

- This chart shows the major items used in the computer system (computer, storage, and I/O) with a summary of the assumed requirements or goals, and the characteristics of each computer system to meet those requirements. In most cases, the assumed requirements are from the MSFC Pre-Phase A Study for the Power Module and are shown to provide a yardstick in comparing the two computer systems with the ATMDC/WCIU system proposed in the MSFC Pre-Phase A study. The comparisons are made for two computers and two STINTS each, with a total 128K bytes (8 bits) of storage for the NSSC-I system and a total of 224K bytes (8 bits) of storage for the NSSC-II system.

- The cost comparisons were based on the assumed one-time cost for software development, the Software Development Facility and one set of DMS flight hardware. The NSSC-I system costs 30% more than the NSSC-II system costs. The NSSC-II system is heavier and uses more power than the NSSC-I system and the NSSC-I computer does not meet the 100% speed contingency requirements. Some minor risk is assumed in the development of a STINT II to interface the NSSC-II with the central unit.

COMPUTER SYSTEM COMPARISONS

	NSSC-I/STINT I	ASSUMED REQUIREMENT OR GOAL	RECOMMENDED NSSC-II/STINT II
COST	98%	DMS FLIGHT UNIT PLUS SOFTWARE COSTS	100%
WEIGHT	37.1 LB	200 LB (MSFC BASELINE)**	73.2 LB
POWER			
(STANDBY)	46.4 W	165 W (MSFC BASELINE)**	254.3 W
(ACTIVE)	56.4 W		254.3 W
VOLUME	1,437.2 $IN.^3$	8812 $IN.^3$ (MSFC BASELINE) **	1,432.4 $IN.^3$
GROWTH	64 K WORDS (16 BITS)	TBD	512 K WORDS (16 BITS)
	81 KOPS	100 KOPS MINIMUM	122 KOPS
PERF	81 KOPS*	100 KOPS (REQUIREMENTS ANALYSIS)	122 KOPS
RISK	NONE	NONE OR MINIMIZE	STINT II DEV

*DOES NOT MEET 100 PERCENT CONTINGENCY REQUIREMENT

**INITIAL STRAWMAN: NOT A FIRM REQUIREMENT OR GOAL

25 kW POWER MODULE – RECOMMENDED STRUCTURAL SUBSYSTEM

- Three of the recommendations shown on the chart were of major importance, and were developed during the Part III effort: (1) the use of an unpressurized berthing structure, with relatively simple berthing-latch devices (as opposed to standard IVA-type docking rings); (2) the use of "common" structures for 25, 50, and 100 kW Power Modules; and (3) the use of 5 berthing ports with a square-arrangement, 4-point attachment.

- The basic rationale for using the unpressurized concept is predicated on the facts that: (a) the Power Module does not require IVA access; and (b) when IVA is needed, the pressurized payload docking module is also required. Thus, the extra weight, cost, and complexity of an additional pressurized interface through the Power Module is unnecessary.

- While the use of common structures is not required on the first mission, significant total-program cost benefit is anticipated by use of the common-design approach for all Power Modules.

- The four-point attachment is necessary to allow 90° "clocking" of payloads, especially those which are palletmounted, for varied pointing requirements.

- The remaining three recommendations were discussed in the Part II Final Report.

25kW POWER MODULE — RECOMMENDED STRUCTURAL SUBSYSTEM

RECOMMENDATION	RATIONALE	
	FOR FIRST MISSION	FOR GROWTH
USE SPACE TELESCOPE "SSM" EQUIPMENT SECTION	IMPROVE MAINTENANCE/ COST	MODULAR GROWTH, REPLICABILITY
USE UNPRESSURIZED BERTHING STRUCTURE	IMPROVE SIMPLICITY/ COST	IMPROVE SIMPLICITY/ COST
PROVIDE FIVE BERTHING PORTS	ENABLE OPTIONAL PAY-LOAD POINTING/CLOCK-ING	ENABLE OPTIONAL PAY-LOAD POINTING/CLOCK-ING
PROVIDE DETACHABLE SOLAR ARRAYS, THERMAL RADIATORS, AND ANTENNAS	IMPROVE MAINTENANCE AND CREW SAFETY	MODULAR GROWTH
DEVELOP ALL STRUCTURAL MODULES FOR COMMONALITY WITH 25, 50, AND 100 kW CONFIGURATIONS	NOT APPLICABLE	PROVIDES FIELD JOINTS FOR GROWTH AND SINGLE DEVELOPMENT COST

25 kW POWER MODULE – RECOMMENDED ELECTRICAL POWER SUBSYSTEM

- The use of the folding solar array blanket assemblies reduces the overall length requirements for the Power Module and thus provides for more efficient use of Orbiter cargo bay space. Further, this concept can achieve solar array growth to 250 kW (increased blanket size and number of blankets) within the available Orbiter cargo bay space.

- Higher capacity cold plates, sized for the higher depth-of-discharge batteries may be installed on the first mission. A Phase B study should determine whether it is cost-effective to have one design sized for maximum load, or several smaller designs sized as required for the lower cooling loads.

- A similar Phase B study is needed in regard to design of battery installations for either Ni-Cd or Ni-H_2 batteries.

25kW POWER MODULE — RECOMMENDED ELECTRICAL POWER SUBSYSTEM

RECOMMENDATION	RATIONALE	
	FOR FIRST MISSION	FOR GROWTH
FOLDING SOLAR ARRAY BLANKET MODULES	IMPROVE ORBITER UTILIZATION	IMPROVE ORBITER UTILIZATION
SWITCHING WITHIN DISTRIBUTORS FOR UNREGULATED POWER USE	FOR FURNACE APPLICATIONS USE UNREGULATED POWER (EFFICIENT)	DISTRIBUTE HIGH VOLTAGE FOR ALL USERS
SIZE COLD PLATES FOR HEAT DISSIPATION GROWTH	PHASE B STUDY CONSIDERATION	HIGHER POWER LEVELS HANDLED WITHOUT COLD-PLATE REDESIGN
DESIGN BATTERY INSTALLATION FOR EITHER NiCd OR $Ni-H_2$	PHASE B STUDY CONSIDERATION	$Ni-H_2$ BATTERIES FOR LONG-LIFE ENHANCEMENT

25 kW POWER MODULE – RECOMMENDED THERMAL CONTROL SUBSYSTEM

- The recommended Power Module thermal control subsystem is based on the conceptual design which was defined at the beginning of the evolution study. Analysis of the performance characteristics of the Freon-21 coolant loop showed this concept would provide adequate heat rejection while maintaining the flexibility to control wide variations in thermal requirements (both Power Module and payload).

- During the Power Module Evolution Study it was found that some hardware modifications to the MSFC baseline would provide increased heat rejection capability, growth potential, and operational flexibility. In particular, a cost reduction is achieved by substituting flat panels in lieu of using the existing Orbiter radiators.

- Cost impact was estimated to be neglibible for oversizing the coolant loop plumbing hardware on the first vehicle to handle the increased heat loads and flow rates for the 50 and 100 kW configurations.

25kW POWER MODULE — RECOMMENDED THERMAL CONTROL SUBSYSTEM

RECOMMENDATION	RATIONALE	
	FOR FIRST MISSION	FOR GROWTH
FLAT RADIATORS	IMPROVE EFFICIENCY	FACILITATES LARGER ARRAY SUBSTITUTION/ PACKAGING
OVERSIZED FLUID-LOOP COOLING BETWEEN BATTERIES/EQUIPMENT/PAYLOAD	NOT APPLICABLE	TO AT LEAST 50 kW CONFIGURATIONS
MECHANICAL ATTACHMENT/FLUID CONNECTORS SUITABLE FOR EVA REPLACEMENT	PERMIT MAINTENANCE	PERMITS ADDING LARGE RADIATORS

25 kW POWER MODULE– RECOMMENDED ATTITUDE CONTROL SUBSYSTEM

- A wide angle 4π steradian sun sensing system is used to provide a line of position to the sun for attitude determination. These sensors also provide the ability to reacquire the sun in a contingency mode.
- A magnetic torquing system, identical to that used on Space Telescope, provides contingency stabilization for retrieval. This system will also provide additional desaturation capability.
- Horizon sensors provide the additional data required to compute three-axis attitude.
- To accommodate stabilization of the larger sortie-mission configurations, and at the same time augment feasibility of growth on-orbit, provision for 6 CMGs is recommended.
- If attitude determination capability on the order of arc minutes is required, star sensors can be added to future Power Modules.

25kW POWER MODULE — RECOMMENDED ATTITUDE CONTROL SUBSYSTEM

RECOMMENDATION	RATIONALE	
	FOR FIRST MISSION	FOR GROWTH
WIDE ANGLE SUN SENSOR	FOR ATTITUDE DETERMINATION	FOR ATTITUDE DETERMINATION
MAGNETIC TORQUING SYSTEM	• PROVIDE CONTINGENCY RETRIEVAL STABILIZATION • PROVIDE ADDITIONAL DESATURATION CAPABILITY	• PROVIDE CONTINGENCY RETRIEVAL STABILIZATION • PROVIDE ADDITIONAL DESATURATION CAPABILITY
HORIZON SENSOR	PROVIDE ATTITUDE DETERMINATION CAPABILITY	PROVIDE ATTITUDE DETERMINATION CAPABILITY
PROVISION FOR SIX CMGs	PROVIDE ADDITIONAL MISSION ORIENTATION CAPABILITY	FOR GROWTH CONFIGURATIONS
STAR SENSORS	NOT APPLICABLE	PROVIDE IMPROVED ATTITUDE DETERMINATION CAPABILITY

25 kW POWER MODULE RECOMMENDED C&DH SUBSYSTEM

- High gain antennas are required to provide RF link closure for data rates above approximately 4 KBS. For payload high data rate transmission, a Ku band kit is recommended. For cost considerations, a dual S and Ku band antenna feed could be implemented on the first Power Module.

- The recommended 256 KBS data rate capability will support Power Module housekeeping (64 KBS) and payload housekeeping and scientific rates (up to approximately 192 KBS) for 25 kW, 50 kW, and 100 kW PM systems.

- The NSSC-II recommendation for the on-board computer is based on an IBM study (Reference 28) which compared the NSSC-I and NSSC-II computers. Special emphasis was given to the key drivers of memory growth, speed, and general performance (Reference 28).

25kW POWER MODULE — RECOMMENDED C&DH SUBSYSTEM

RECOMMENDATION	RATIONALE	
	FOR FIRST MISSION	FOR GROWTH
HIGH GAIN ANTENNAS, S-BAND (STEERABLE)	REQUIRED FOR SOLAR TERRESTRIAL DATA AND PM DATA > 4 KBS	Ku BAND KIT ALLOWS DATA RATE GROWTH TO 300 MBS
NSSC II COMPUTER	IMPROVED SPEED FOR EARLY PAYLOAD SYSTEM REQUIREMENTS	FOR HANDLING MORE ACS AND MEMORY REQUIREMENTS
256 KBS DATA RATE CAPABILITY	TO SUPPORT EARLY PAYLOAD SYSTEM RQMTS	Ku BAND KIT TO MEET EXPANDED PAYLOAD & PM DATA RATE RQMTS
DISTRIBUTED DATA BUS SYSTEM (REMOTE TELEMETRY & COMMAND UNITS)	MINIMIZES WIRES CROSSING DOCKING INTERFACES	MINIMIZE WIRES CROSSING PAYLOAD/POWER MODULE INTERFACES. GROWTH BY ADDING REMOTE UNITS.

SECTION 4
SYSTEM SUPPORT ELEMENTS

STS ENHANCEMENT SYSTEM SUPPORT ITEMS

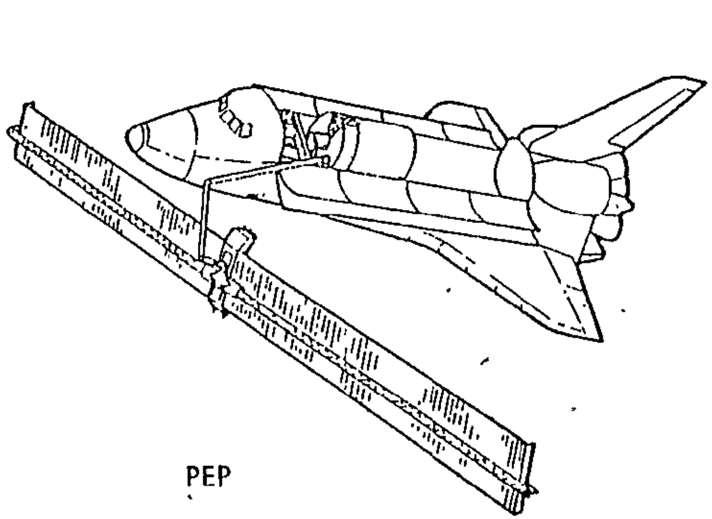
- The chart depicts, in addition to the PEP augmentation to the Orbiter and the Power Module itself, five STS support elements utilized in the scenarios studied.

- The pressurized Payload Docking Adaptor (PDA) is required whenever the missions utilize a Working Module or a Manned Habitat. Also, most payloads for the free-flyer missions will be utilizing one type or more of pallets. In the unmanned missions it is expected that the berthing capability of the Power Module will provide an adequate interface with payloads and other STS elements, and the PDA will not be needed.

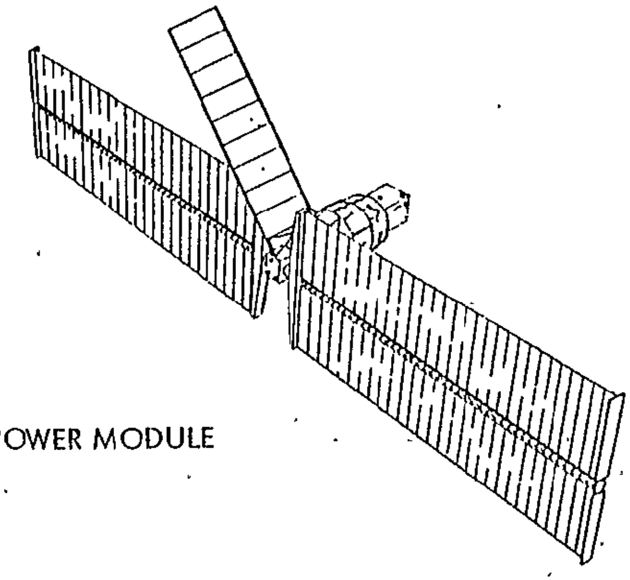
- As discussed in Section 2.7, Teleoperator is expected to satisfy most, if not all, orbit reboost requirements.

- The Working Module, the Manned Habitat, and the Pallets are all projected outgrowths of the Spacelab program. Interface development between them, the Orbiter, the pressurized Payload Docking Adaptor, and the Power Module will require comprehensive coordination.

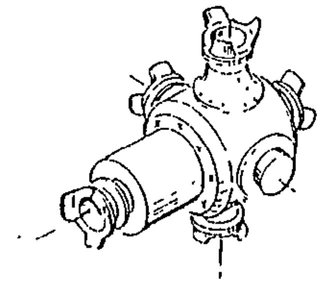
STS ENHANCEMENT SYSTEM SUPPORT ITEMS



PEP



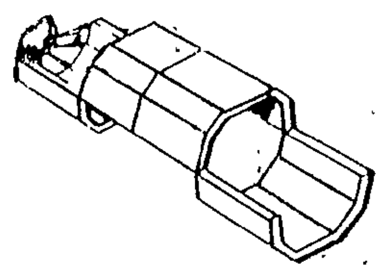
POWER MODULE



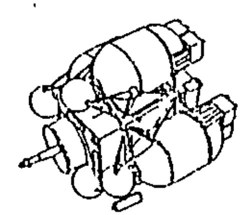
PAYLOAD DOCKING ADAPTOR

WORKING MODULE

MANNED HABITAT



SPACELAB PALLETS
(FREE-FLYER MODE)



TELEOPERATOR

PAYLOAD DOCKING ADAPTOR POTENTIAL

- The Payload Docking Adaptor (PDA), in addition to its usefulness with the Power Module, appears to have a wide variety of applications with other elements of the Space Transportation System. Its basic function is to interconnect the elements.

- Like the Power Module berthing structure, it also provides the capability to assem ble any orthogonal multi-unit space platform configuration. In addition, however, it will provide pressurized interconnect for IVA operations.

- The chart identifies optional features, and associated subelements, likely to be utilized with a PDA.

PAYLOAD DOCKING ADAPTOR POTENTIAL

- TELESCOPING INTERFACE SECTION

  PROVIDES CLEARANCE BETWEEN MATING ELEMENTS, AND ACHIEVES COMPLIANCE WITH ORBITER STA. 660 CONSTRAINTS.

- ROTATABLE INTERFACE ADAPTER

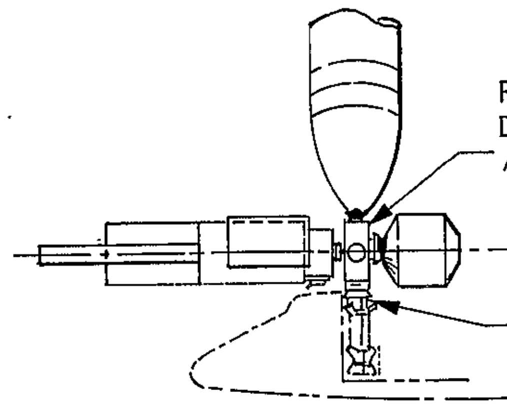
  ALLOWS CLOCKING OF MATING ELEMENTS AND FACILITATES DOCKING AND DEMATING.

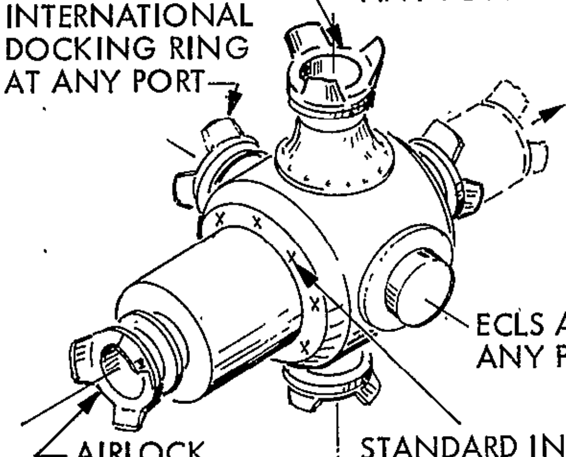
- EMERGENCY ECLS PACK

  PROVIDES AN ENVIRONMENTAL CONTROL AND LIFE SUPPORT (ECLS) MODULE ENABLING USE OF THE PDA AS A SHORT-TERM LIFE-RAFT

- AIRLOCK CHAMBER

  PROVIDES AIRLOCK FOR EVA OR IVA OPERATIONS WITH ANY ELEMENT COMBINATION





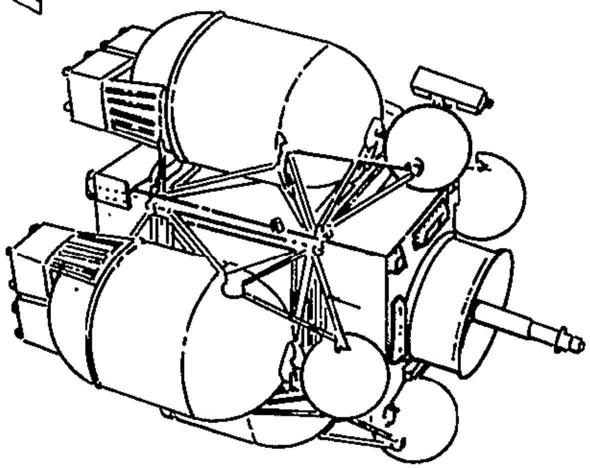
TELEOPERATOR BASELINE CAPABILITY

- The Teleoperator retrieval system consists of a vehicle approximately 130.0 inches dia. x 129.5 inches, capable of accomplishing a variety of useful tasks on orbit. It is controlled either through preprogrammed instructions from its Communication and Data Management Computer, or through manual control by a shuttle crew member using support equipment in the Orbiter. As indicated on the chart, its initial application is in connection with Skylab retrieval (References 6 and 33).

- The basic TRS vehicle contains six subsystems:
  - Structures and Mechanisms
  - Thermal Control
  - Guidance, Navigation and Control
  - Propulsion
  - Communication and Data Management (two TV cameras)
  - Electrical Power and Distribution

- System characteristics and performance data of interest in Power Module applications are summarized on the chart. The capability available with these characteristics satisfies nearly all the requirements for reboost in Scenario I (see discussion in Section 2.6).

TELEOPERATOR BASELINE CAPABILITY



PERFORMANCE DATA

GROSS WEIGHT (WET)	9,900 LBS
BASIC CORE (WET)	2,300 LBS
4 BASIC PROPULSION KITS (WET)	7,600 LBS
DRY WEIGHT	3,440 LBS
BASIC CORE	1,870 LBS
4 PROPULSION KITS	1,570 LBS
PROPELLANT: CORE	25,000 LB. SEC.
(N_2H_4) KITS (4)	1,350,000 LB. SEC
PROPULSION KIT THRUST (EACH)	300 LBS
RF LINK RANGE	760 N. MILES

PLANNED MISSIONS

- IOC DATE 1979
- SKYLAB REBOOST OR DE-ORBIT

SYSTEM CHARACTERISTICS

- 24-NOZZLE GUIDANCE AND ATTITUDE CONTROL SYSTEM, 6 DEGREES OF FREEDOM
- STRAP-ON PROPULSION KITS (4)
- DOCKING PROBE SYSTEM
- COMMUNICATION AND DATA MANAGEMENT
- MANUAL CONTROL CAPABILITY
- RMS GRAPPLING FIXTURE; ASE FITTING
- TV CAMERAS (2); ILLUMINATION SYSTEM
- THERMAL CONTROL SUBSYSTEM

SECTION 5
SPACE SUPPORT EQUIPMENT

SPACE SUPPORT EQUIPMENT

- A listing of primary items of Space Support Equipment (SSE) for use in Power Module flight operations is provided on the chart. Estimated weights are given, as well as type of mission on which each item is used.

- An estimated total weight (at time of Orbiter launch) of SSE required to support each type of mission is also provided. Approximately half of the spreader-bar weight is believed to be chargeable to the Orbiter rather than PM/SSE (to be determined during Phase B).

- Additional data on the Berthing System and Maintenance Platform are given in Section 2.6. Growth kit data are provided in Section 2.8.

- The charts which follow provide data on the berthing system and the spreader bar.

SPACE SUPPORT EQUIPMENT

ITEM	WEIGHT (LBS)	TYPES OF MISSIONS			
		SORTIE	PLACEMENT RECOVERY	MAINTENANCE	ON-ORBIT GROWTH
1 BERTHING PLATFORM ASSEMBLY	956	X	X	X	X
2 UMBILICAL RETRACT ASSY CARGO BAY	30	–	X	–	–
3 UMBILICAL RETRACT ASSY BERTHING PLATFORM	30	X	X	X	X
4 GROWTH KIT CRADLE	700	–	–	–	X
5 MAINTENANCE PLATFORM & MAST ASSY	225	–	–	X	–
6 CONTROL AND DISPLAY PANEL	50	X	X	X	X
7 WIRE HARNESS CONTROL & DISPLAY	20	X	X	X	X
8 WIRE HARNESS CARGO BAY UMBILICAL	40	–	X	–	–
9 WIRE HARNESS BERTH UMBILICAL	40	X	X	X	X
10 SPARES CONTAINER	75	–	–	X	–
11 MAINTENANCE TOOL KIT	50	–	–	X	X
12 REMOTE MANIPULATION SYSTEM (ORBITER KIT)	850	X	X	X	X
13 TRUNNION/KEEL SPREADER BARS (4)	700	X	X	X	X
SSE LAUNCH WEIGHT		2646	2716	2996	3396

POWER MODULE BERTHING SYSTEM

- The system consists of a table supported upon four ball nuts which ride upon vertically oriented "Saginaw" type screws. The screws are attached to a sill platform and frame assembly, which in turn attach to three trunnions and a keel fitting in the Orbiter payload compartment. The sketch shows the berthing system installed over the air lock. It can be installed wherever required along the Payload Bay. Also Refer to Section 2.6.

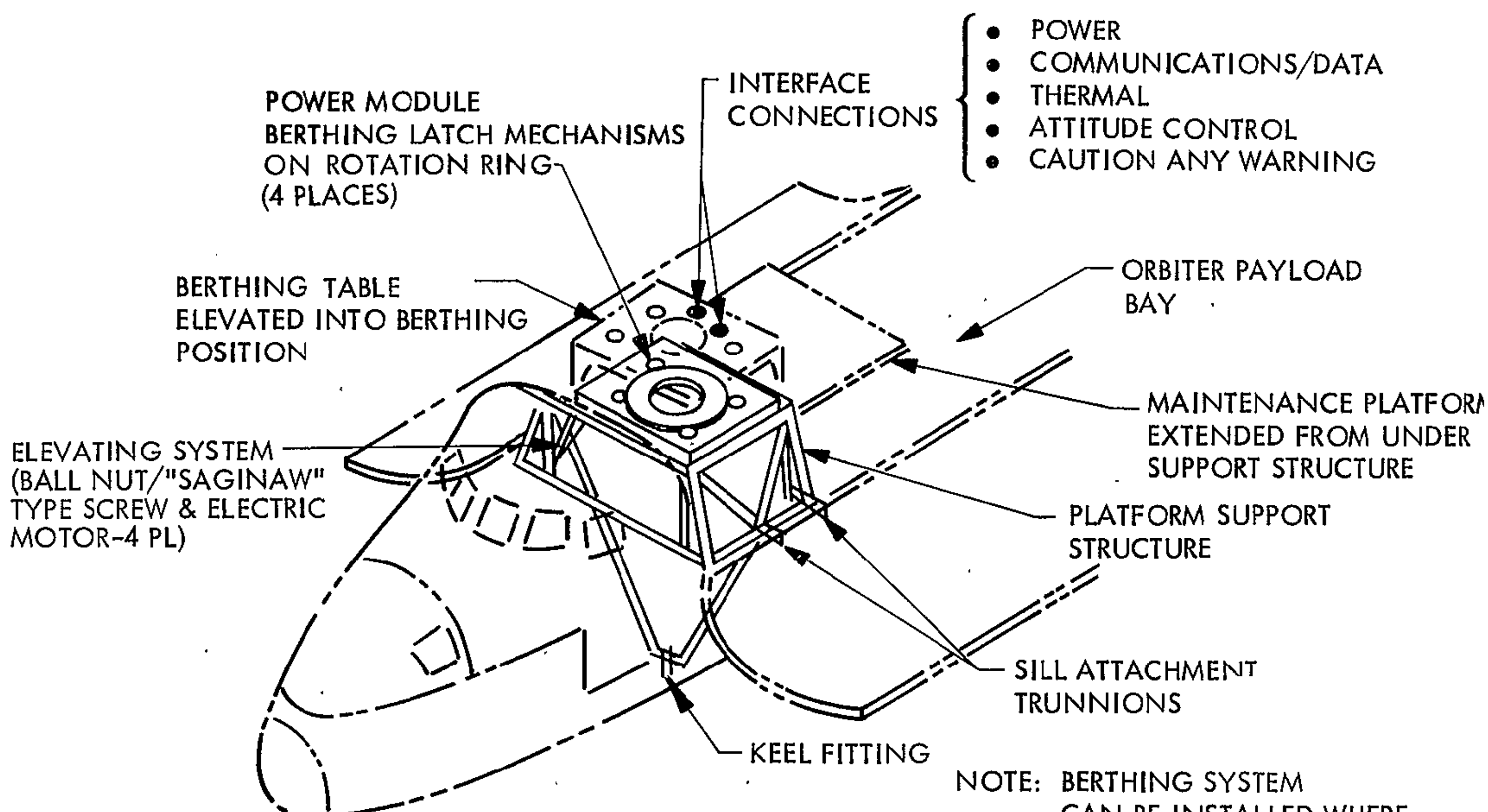
- On the upper surface of the table is mounted a rotation ring to which is attached the latch mechanism and guide system to which the Power Module berths. An electric motor/rack and pinion system mounted upon the table enables rotation of the berthed Power Module into any desired position.

- Interface connections, Power Module to Orbiter (for power, communications, attitude control, thermal caution and warning), are situated on the rotation ring.

- Under the sill platform is stowed a maintenance platform to which is attached a folded access mast.

- Movement of the berthing table into the deployed (or stowed) position is by synchronized electric motors driving the "Saginaw" type screws.

POWER MODULE BERTHING SYSTEM



NOTE: BERTHING SYSTEM CAN BE INSTALLED WHERE REQUIRED ALONG THE LENGTH OF THE ORBITER PAYLOAD BAY

POWER MODULE SPREADER BAR

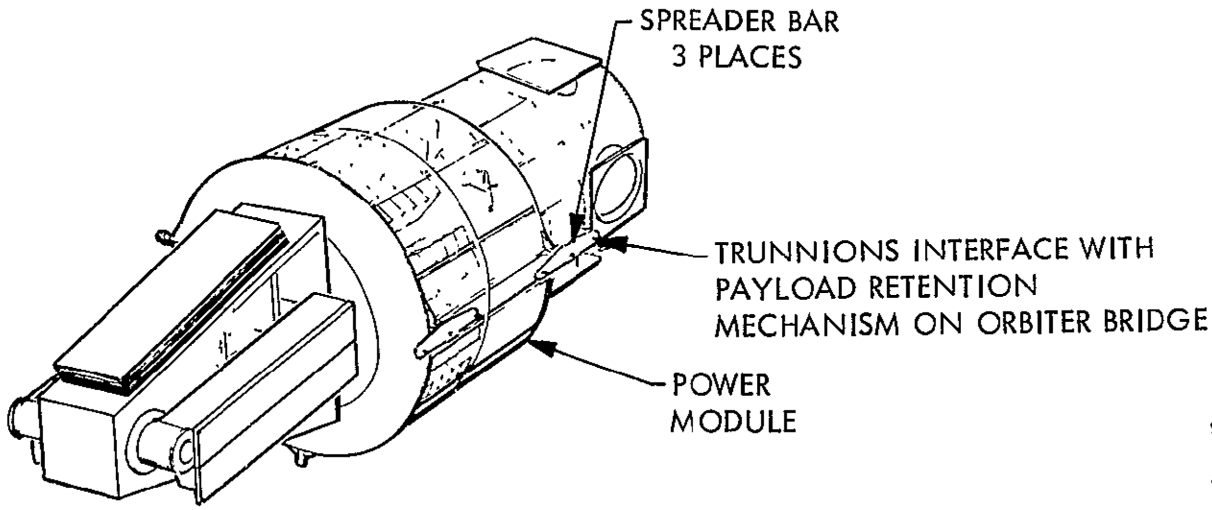
- The spreader bar provides the capability to distribute a payload trunnion load to two points. This is a means of maintaining loads on the Orbiter sill within the load limitations.

- The spreader bar consists of an I beam type structure, a trunnion fitting and two trunnion brackets. The payload trunnion is clamped to the beam by the trunnion fitting. The trunnion brackets are fastened to each end of the I beam. The trunnion brackets are aligned to the Orbiter sill and supported by the Orbiter payload retention mechanism. This design concept is one of several that can be developed.

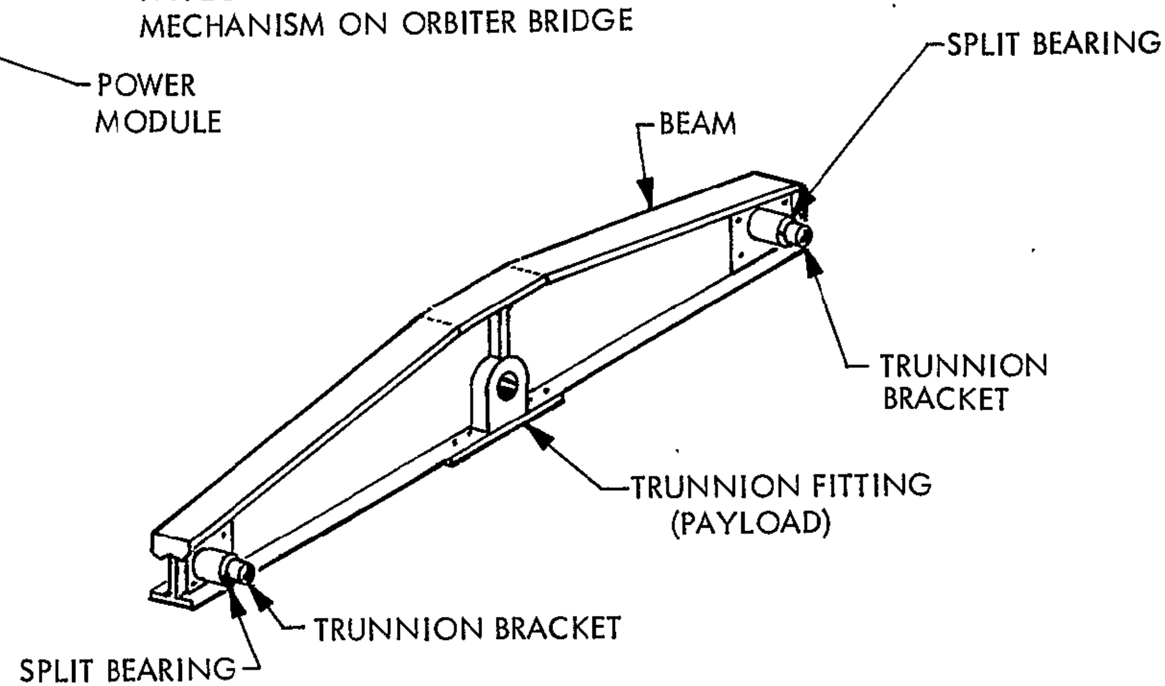
- The Power Module uses three spreader bars mounted on the Orbiter sill. The bar assembly can be removed from the payload by sliding it off the Power Module trunnion.

POWER MODULE SPREADER BAR

FUNCTION: TO SPREAD POWER MODULE ASCENT/DESCENT LOADS TO 2 ORBITER TRUNNION RETENTION MECHANISMS.



SPREADER BAR INSTALLATION



SECTION 6
POWER MODULE OPERATIONS

6.1 TRANSPORT, PRELAUNCH & LAUNCH

POWER MODULE GROUND OPERATIONS

- The ground operations flow sequence for the Power Module interfaces with the STS processing facilities and Orbiter vehicle as shown on the chart. All processing times for the PM are within Orbiter processing timelines. To accommodate the PM size, the NASA Guppy aircraft is used to transport the vehicle from the factory to the KSC landing site.

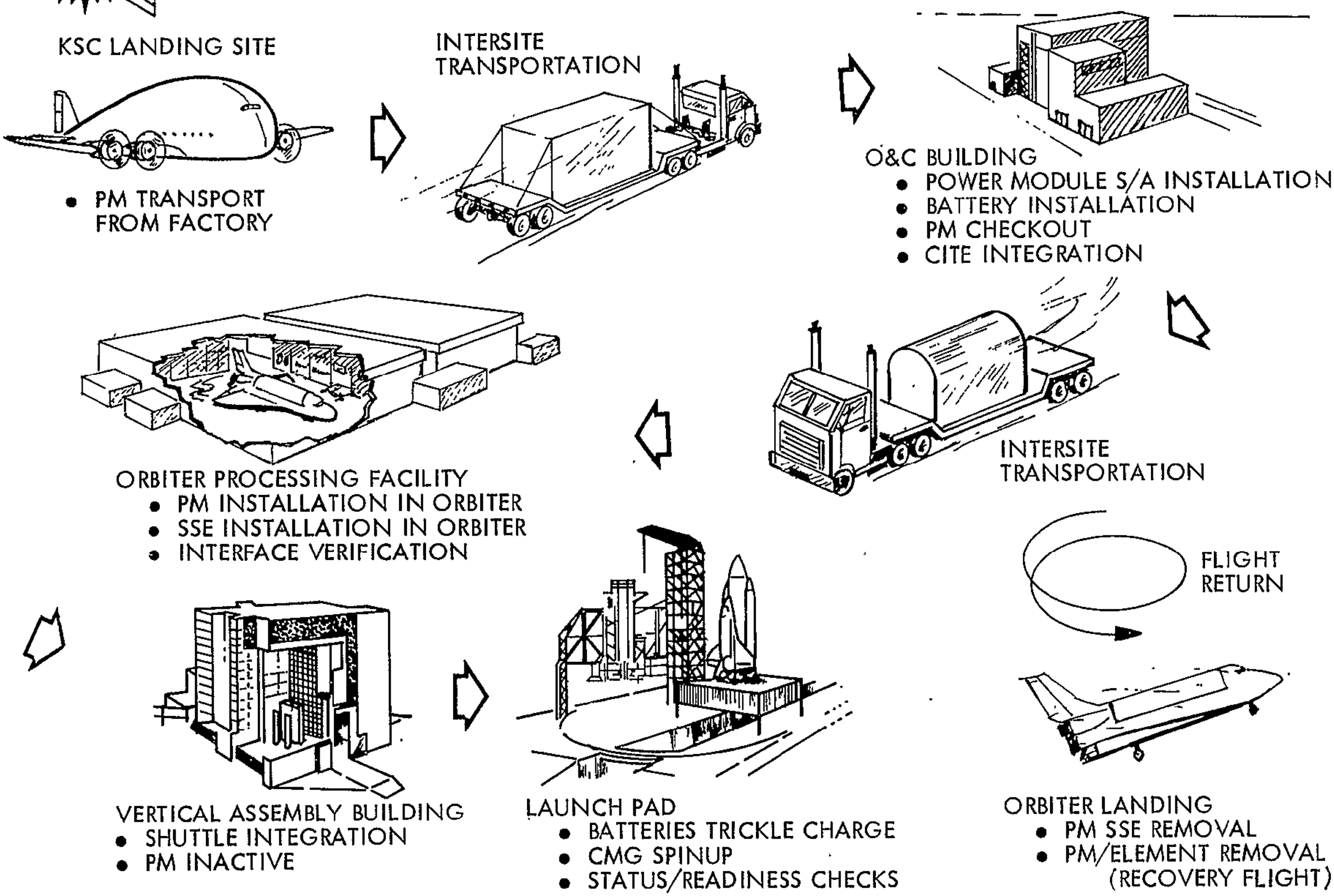
- Upon arrival the PM is transported via ground carrier to the Operations and Checkout building where the vehicle will be prepared for flight. The PM will be processed in a horizontal attitude and will conform to the O&C processing flow line. The solar array containment cannisters and battery pack modules will be installed into the PM vehicle and tests will be performed to verify PM subsystems and interfaces.

- Space support equipment for the PM/Orbiter interfaces will be serviced, updated, maintained and verified in the O&C building. SSE will be installed in the Orbiter at the OPF to support Orbiter/PM operations. After return from orbit flight, the SSE will be removed from the Orbiter at the OPF. The SSE will be returned to the O&C building for inspection, testing, and updating as required to support subsequent PM/Orbiter flight operations.

- For a final verification of the PM/Orbiter interfaces, the PM will be tested in the cargo integration test equipment (CITE). The tests will verify the readiness of the PM to be installed in the Orbiter. The PM will be placed in the payload cannister and transported to the Orbiter Processing Facility for horizontal installation in the Orbiter cargo bay where PM space support equipment will have previously been installed and tested.

- The PM remains inactive at the Vertical Assembly Building during Shuttle build-up activities. At the launch pad the PM will be activated to verify functional interfaces with the Orbiter and verify launch readiness. Prior to launch the batteries may be trickle charged (TBD) and the control moment gyros will be spun up to low speed for launch.

POWER MODULE GROUND OPERATIONS



POWER MODULE ASSEMBLY VERIFICATION SEQUENCE – LAUNCH SITE

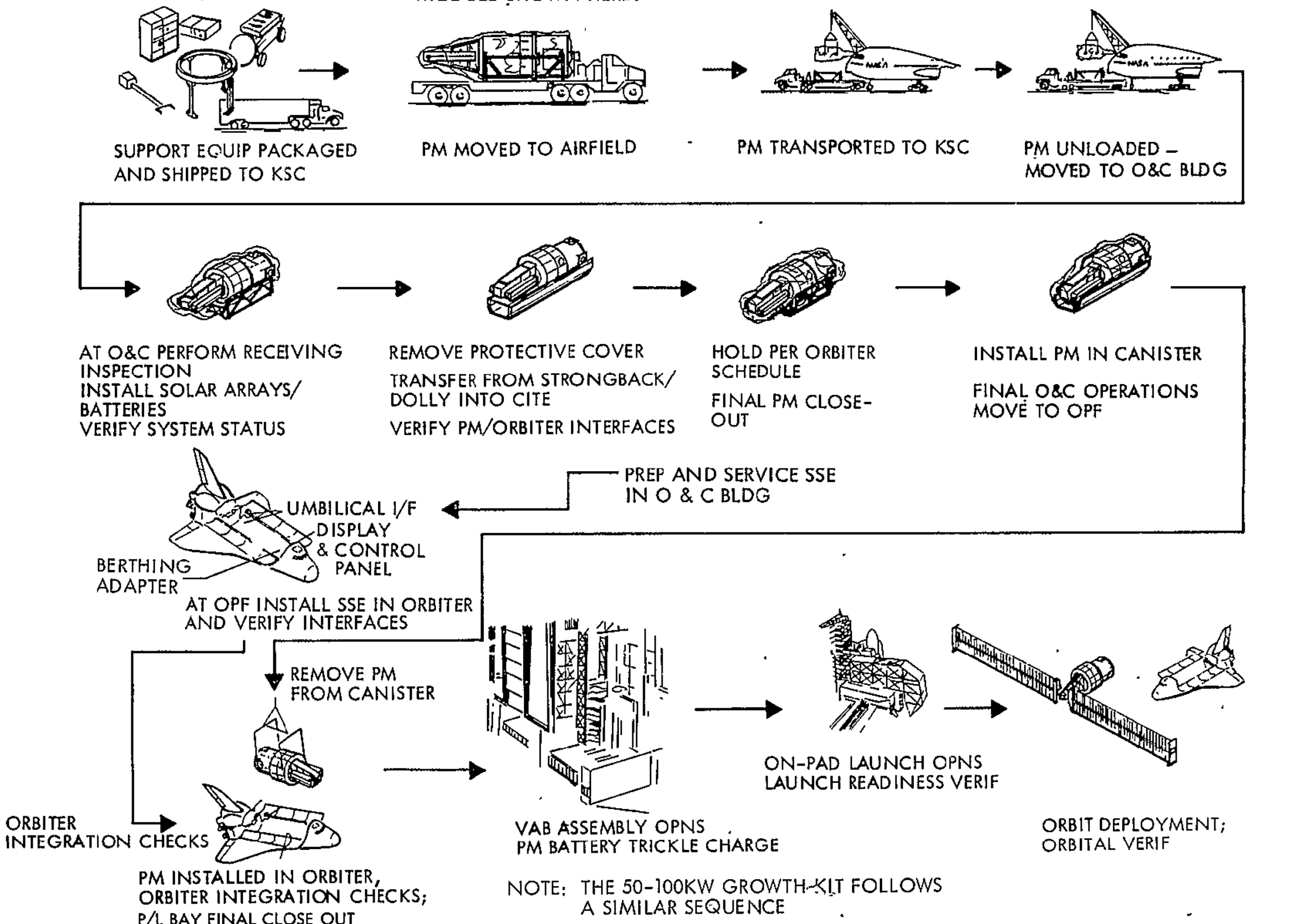
- At the KSC launch site the PM assembly sequence and prelaunch operations are oriented toward processing the vehicle in a horizontal attitude in the Operation and Checkout Building (O & C). Assembly activities on the PM include the following:
  - Installation of solar array wing assemblies
  - Charging and installation of the flight batteries
  - Installation of PM space support equipment in the Orbiter at the Orbiter Processing Facility.

- The PM will be installed in the Orbiter at the OPF and will remain in the payload bay during all subsequent shuttle assembly operations. The only unique requirement for PM support while in the Orbiter may be the maintenance of a trickle charge on the PM batteries. (To be determined)

- Verification tests will be performed during the processing sequence as follows:
  - Visual inspection of flight hardware and review of records on receipt of equipment at the launch site.
  - Solar array continuity test after installation on the PM.
  - Battery performance tests including charge and discharging testing after installation in the PM.
  - PM systems functional performance tests.
  - Interface verification of the PM/Orbiter interface using the cargo integration test equipment in the O & C Building.
  - Integration checks of Orbiter/PM functional interfaces after PM installation in the Orbiter.
  - Pre-launch PM system status verification tests on the launch pad.

POWER MODULE ASSEMBLY VERIFICATION SEQUENCE – LAUNCH SITE

NOTE: THE 50-100KW GROWTH KIT FOLLOWS A SIMILAR SEQUENCE TO THE POWER MODULE SHOWN HERE.



NOTE: THE 50-100KW GROWTH KIT FOLLOWS A SIMILAR SEQUENCE

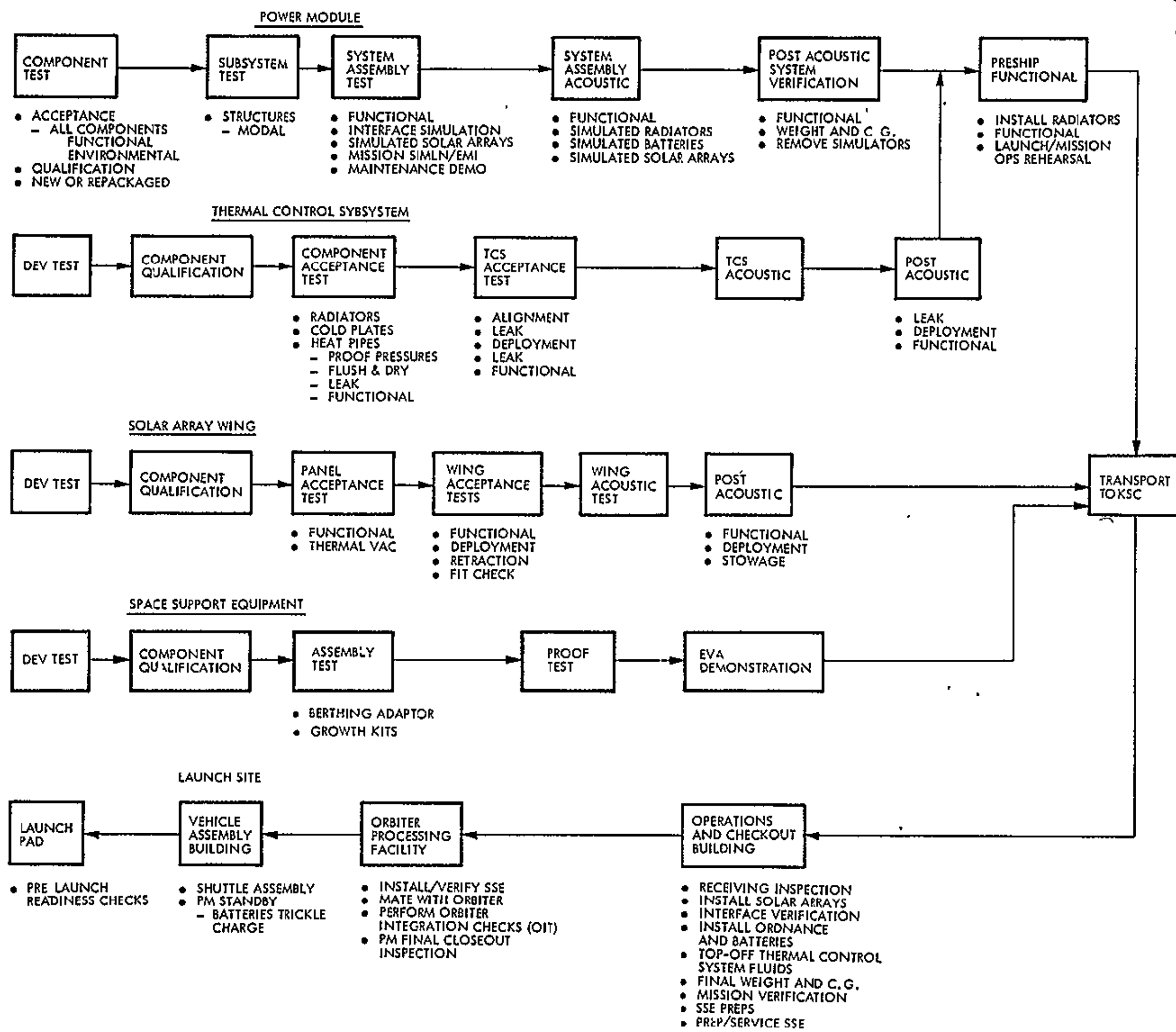
- The 25 kW PM ground test program includes testing at component, subsystem, and system assembly levels at the factory and the launch site. The test program is structured to take advantage of the use of existing hardware and minimizes the extent of ground tests. Where technically feasible, simulators will be used in lieu of flight equipment. Thermal vacuum tests will be conducted at subsystem levels on the thermal control subsystem and the electrical power subsystem. Data from tests previously performed on existing equipment, such as the control moment gyro assemblies, will be used to verify capability of this equipment. Vehicle system level acoustic testing will be performed to verify all system functional performance. Simulators will be used for batteries, radiators, and solar array wings.

- The major objective of factory testing of flight systems will be to verify the functional capability of all flight equipment and to demonstrate functional compatibility of all interfaces through use of interface simulators. Final acceptance of the flight systems at the factory will include an all-systems, sequenced mission simulation test. The solar array wings will be tested as a separate assembly under acoustic environment conditions. With successful completion of these factory tests, the flight equipment will be considered "ready for flight use."

- Pre-launch assembly and testing at the launch site is minimized. The solar array wing assemblies and flight batteries are installed during the ground processing operations at the Operations and Check-out Building at KSC. A simulated mission sequence test is performed to verify readiness of all PM systems. The PM is then installed in the Orbiter payload bay and interface checks are performed to verify Orbiter/PM compatibility. Final pre-launch readiness checks will be conducted on the launch pad to verify PM launch readiness.

- The ground test program for the berthing adaptor is a matter for further study during subsequent program phases to define details of functional testing at the factory and the launch site. From a concept viewpoint, it is assumed that the adaptor will be subjected to structural testing, interface fit and functional demonstration testing at the factory (test facility) to verify functional capability. In addition, EVA demonstration-tests will be conducted using a berthing adaptor demonstration test item to develop and verify procedures for handling the adaptor in Zero "g" simulations. At the launch site the berthing adaptor will be prepared and serviced in the O&C building, along with other Space Support Equipment (SSE). Generally speaking, verification of the adaptor will include inspection (visual and non-destructive testing) to verify integrity of the structure, and interface fit and functional testing using templates to match/mate test interfacing surfaces and latching/release hold downs. The umbilical connect/retract system and hold down/release mechanism will be functionally tested to verify proper operation.

25kW POWER MODULE GROUND TEST PROGRAM



POWER MODULE KSC PROCESSING TIMELINE

- The timeline for performing PM pre-launch processing tasks at the launch site is shown on the facing page. All shuttle times are from Shuttle Turnaround Analyses report (Star 015), reference 29.

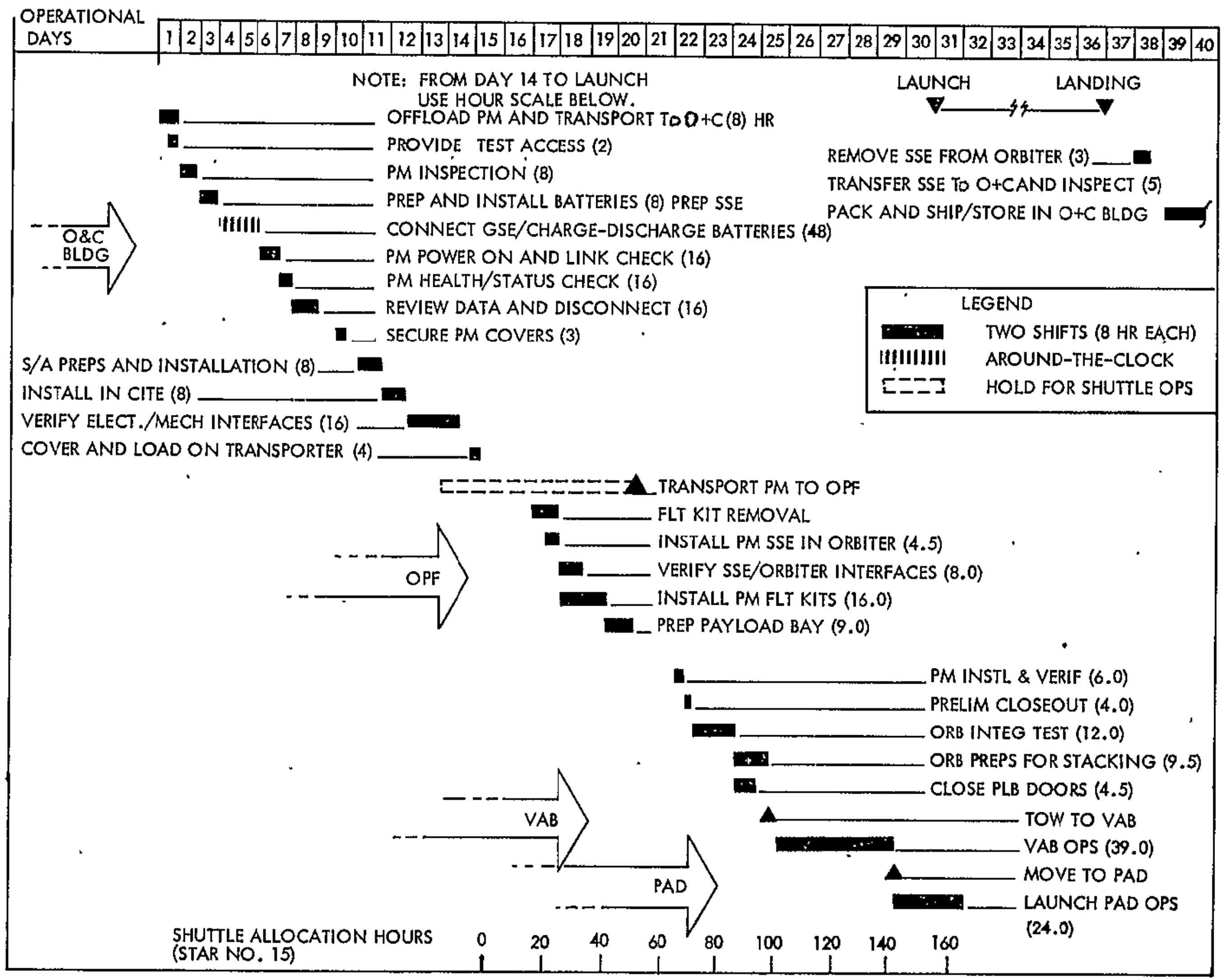
- Major timeline activities are associated with processing of the PM at the Operation and Checkout (O&C) Building. These activities are independent of the Orbiter processing and require approximately 15 operational days (3 weeks).

- Interface with the Orbiter on line processing begins in the Orbiter Processing Facility (OPF) with the installation of PM space support equipment at approximately 130 hours before launch.

- Actual installation of the Power Module in the Orbiter Cargo Bay is planned to be performed at the OPF approximately 90 hours before launch. After installation in the Orbiter Cargo bay, the PM will remain essentially quiescent during Orbiter preparation at the Vertical Assembly Building (VAB) and the launch pad.

- All PM processing activities at the launch site are compatible with the Orbiter processing sequence and schedules.

POWER MODULE KSC PROCESSING TIMELINE



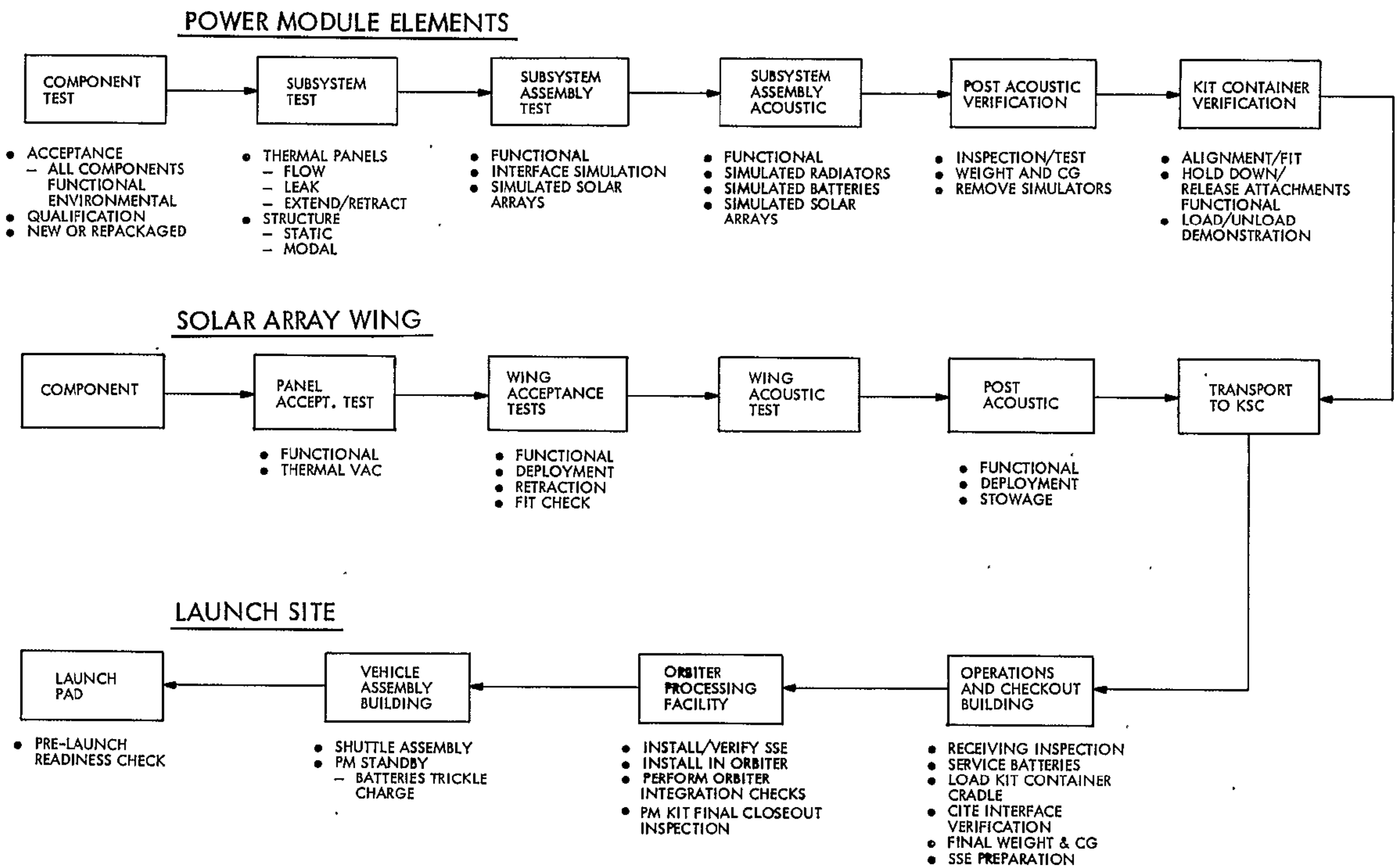
50-100 kW POWER MODULE GROWTH KIT GROUND TEST PROGRAM

- The 50-100 kW PM kit ground test program includes testing at component and subsystem assembly levels at the factory and the launch site. Where technically feasible, simulators will be used in lieu of flight equipment. Thermal vacuum tests will be conducted at subsystem levels on the thermal control subsystem and the electrical power subsystem. Subsystem level acoustic testing will be performed to verify functional performance. Simulators will be used for batteries, radiators and solar array wings.

- Power Module elements will be fit checked in the growth kit container to verify fit, hold down, and release capability. Loading and unloading will be demonstrated to verify functional interfaces and procedures. The solar array wings will be tested as a separate assembly under acoustic environment conditions. With successful completion of these factory tests, the flight equipment will be considered "ready for flight use".

- Pre-launch servicing and testing at the launch site is minimal. The flight batteries are serviced during the ground processing operations at the Operations and Checkout Building at KSC. The kit is loaded in the flight container cradle and then installed in the cargo integration test equipment to verify compatibility with the Orbiter cargo bay. Then the kit/container is installed in the Orbiter payload bay and interface checks are performed to verify Orbiter-kit compatibility. Final pre-launch status determination will be conducted on the launch pad to verify launch readiness.

50-100 kW POWER MODULE GROWTH KIT GROUND TEST PROGRAM



50-100 kW POWER MODULE KIT KSC PROCESSING TIMELINE

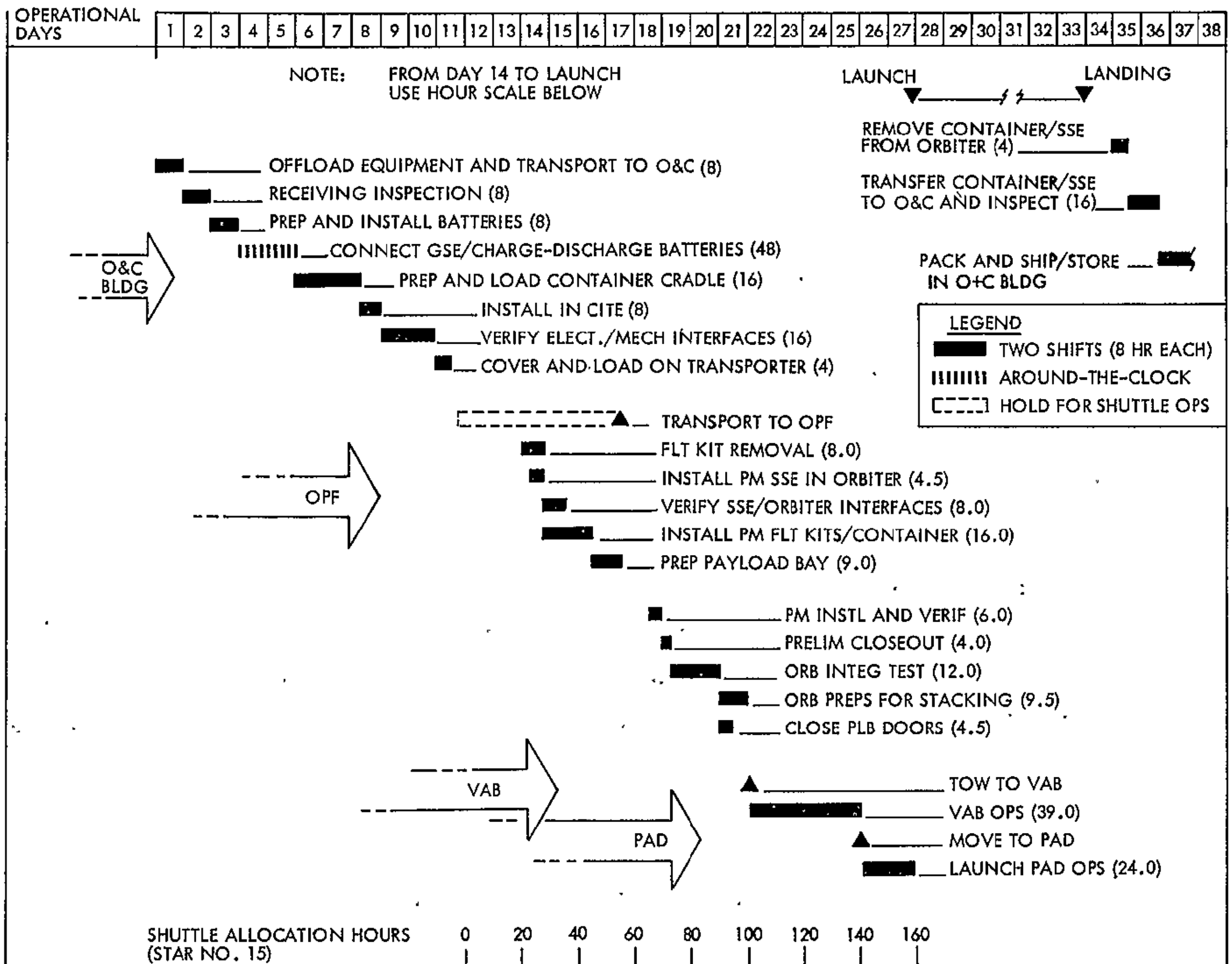
- The timeline for performing ground operations prelaunch tasks for the 50-100 kW PM kit is shown on this page. All shuttle times are from STAR 015.

- Approximately eleven work days are scheduled to prepare, verify and condition the kit/ equipment in the Operations and Checkout Building. The kit may then be held in readiness for the Orbiter vehicle.

- Interface with the Orbiter on-line processing begins in the Orbiter Processing Facility (OPF) with the installation of the space support equipment for PM assembly on orbit, at approximately 130 hours before launch. Installation of the 50-100 kW PM growth kit can be performed at the OPF approximately 90 hours before launch. Since the kit occupies only a part of the cargo bay, the payloads may be installed into the cargo bay at this time. After installation in the cargo bay, the PM growth kit will remain quiescent during Orbiter preparations at the Vertical Assembly Building (VAB), and on the launch pad.

- All 50-100 kW PM orbital growth kit activities at the launch site are compatible with the Orbiter processing sequence and schedules.

50-100 kW POWER MODULE KIT KSC PROCESSING TIMELINE



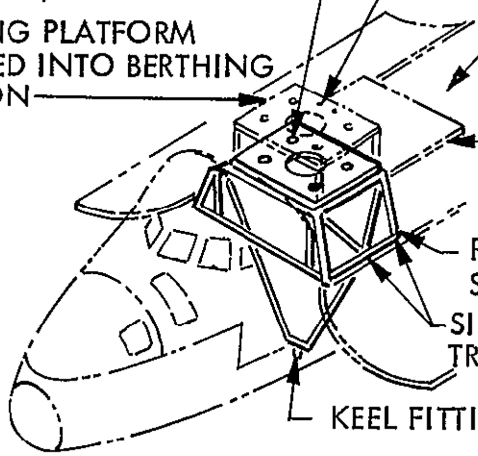
6.2 BERTHING DEPLOYMENT & CHECKOUT

BERTHING OPERATION CONCEPT

- Berthing features for the Power Module/Orbiter are shown here. The berthing system consists of a table supported on a sill platform and frame assembly. It is attached to three trunnions and a keel fitting in the Orbiter payload compartment. The berthing system can be positioned where required along the payload bay. A rotation ring with latch mechanisms and a guide system is mounted on the table. A maintenance platform with a folded access mast is stowed under the sill platform.

- Berthing operation will be utilized to attach the Power Module to the Orbiter for the sortie mode and maintenance mode type missions. The sortie mode berthing orientation is "over the nose" of the Orbiter while the maintenance mode orientation is vertical "tail down" to the Orbiter.

BERTHING OPERATION CONCEPT

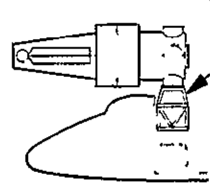


DESIGN

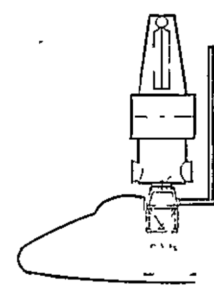
- ORBITER COMPATIBLE
- POSITION FLEXIBILITY ALONG LENGTH OF PAYLOAD BAY
- LIGHTWEIGHT GUIDE MATING CONES AND LATCH MECHANISMS
- PLATFORM OFFSET OVER ORBITER FOR CLEARANCE AND RMS KINEMATICS
- AUTOMATED UMBLICAL CONNECT AND DISCONNECT
- ROTATIONAL PLATFORM FOR MAINTENANCE POSITIONING
- INCORPORATES MAINTENANCE PLATFORM AND ACCESS MAST

OPERATIONS

- DUAL MODE POSITIONING
- RMS FOR CAPTURE (GRAPPLE), TRANSLATION, POSITION ALIGNMENT, BERTHING MATE
- RELEASE VIA RMS
- EVA USE AFTER BERTHING
- NO ACS PERFORMANCE DEGRADATION



SORTIE MODE

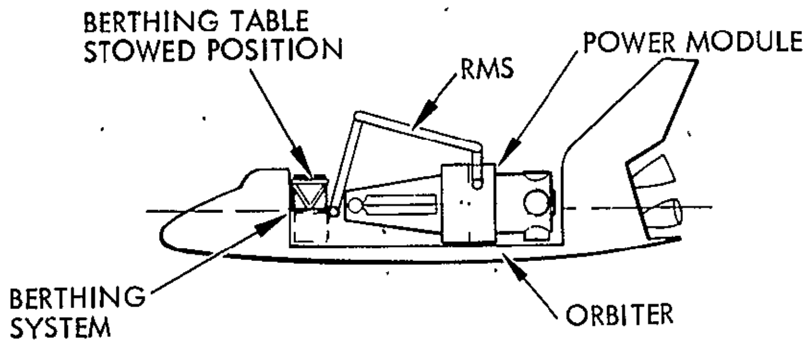


MAINTENANCE MODE

INITIAL DEPLOYMENT/BERTHING SEQUENCE

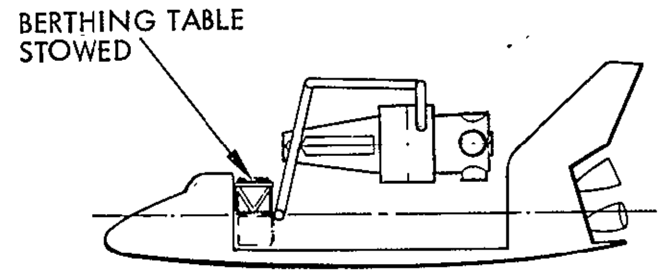
- The four phases of the initial unloading of the Power Module (PM) from the Orbiter, and berthing the PM to it, are illustrated on the chart.

INITIAL DEPLOYMENT/BERTHING SEQUENCE



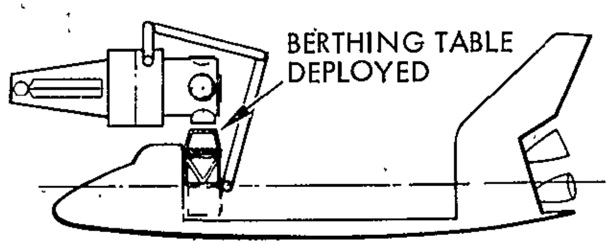
STEP I

RMS ATTACHED TO POWER MODULE



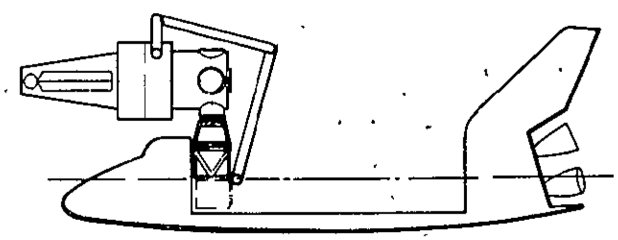
STEP II

INITIAL LIFT FROM ORBITER



STEP III

POWER MODULE POSITIONED
ABOVE BERTHING TABLE



STEP IV

POWER MODULE LOWERED
ONTO TABLE AND BERTHED

INITIAL DEPLOYMENT/BERTHING CHECKOUT OPERATIONS

- The sequence and tasks associated with the initial 25 kW deployment and checkout occur in three phases, as shown on this chart.

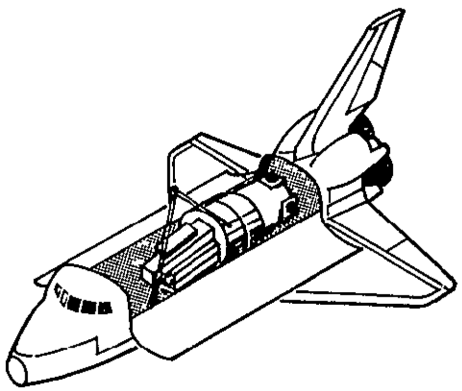
- Checkout operations will be performed in each phase. The significant checkout tasks will be performed during the post docking phase. During this phase, after berthing is completed, all systems of the PM will be command operational. Performance of all PM systems and PM/Orbiter power, heat transfer, and attitude stabilization interface performance will be evaluated for varying power-heat load conditions and attitude orientations. Primary checkout control and monitoring can be from the Orbiter aft crew station using PM display/control instrumentation. Performance data will be relayed to ground stations via RF for determining detail subsystem performance.

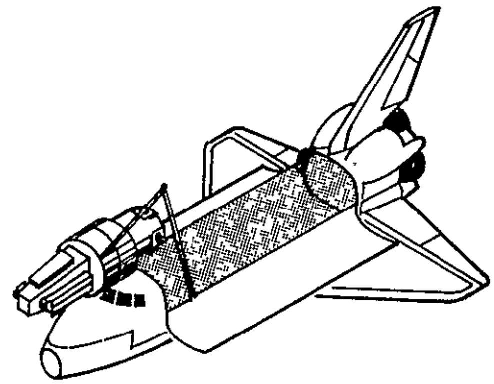
- The significant time-critical item during this sequence is the spinup time for PM CMG reaction wheels. Normal spinup time to achieve 9,000 rpm operating speed is 12 hours. CMG spinup will be initiated during the pre-deployment checkout phase. Estimated sequence timelines are as follows:

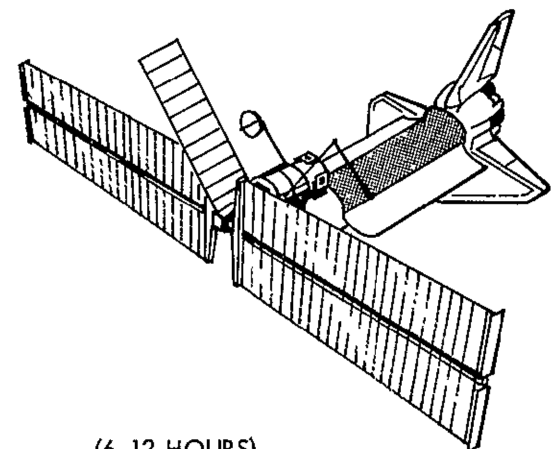
Pre-deployment	12 hours
Deploy and dock	30 to 60 minutes
Post-docking checkout	4 to 6 hours

- During deployment and berthing operations, the Orbiter maintains RF link control of the PM and in the post-docking phase the command and control of the Orbiter is by hardline link to the Orbiter. All disconnect/reconnect mechanisms are actuated by Orbiter control. RMS attach-disconnect and manipulation will be by an astronaut operator at the Orbiter RMS station. The Power Module radiators and solar array will be in stowed position during pre-deployment, deployment, and docking sequences. During PM positioning and docking, the PM CMGs will stabilize the PM. All active guidance and maneuvering will be done by the Orbiter.

INITIAL DEPLOYMENT/BERTHING CHECKOUT OPERATIONS







(12 HOURS)

PRE-DEPLOYMENT CHECKOUT

- INITIATE CMG SPIN UP
- WITH PAYLOAD BAY DOORS OPEN, POSITION ORBITER BERTHING ADAPTER
- ATTACH RMS TO PM
- VERIFY PM C AND W
- VERIFY COMPUTER LOAD
- VERIFY PM CMG ROTATION SPEED
- VERIFY PM ACS/ORBITER INTERFACE
- CONFIRM PM HEALTH AND STATUS

(30-60 MINUTES)

DEPLOY AND BERTH TO ORBITER

- INITIATE DEPLOYMENT SEQUENCE
- SWITCH PM TO INTERNAL POWER
- DISCONNECT UMBILICAL
- RELEASE RETENTION LATCHES
- ROTATE AND POSITION PM
- VERIFY RF LINK TO ORBITER
- CONFIRM PM STATUS AND HEALTH
- VERIFY BERTHING ADAPTER READINESS
- TRANSLATE AND BERTH PM
- SECURE BERTHING MECHANISMS
- CONNECT PM-ORBITER BERTHING UMBILICALS
- CONFIRM BERTHING INTERFACES

(6-12 HOURS)

POST BERTHING CHECKOUT

- VERIFY COMMAND LINK
- COMMAND PM SYSTEMS ON
- VERIFY PM ACS STATUS
- VERIFY PM-ORBITER ACS STABILIZATION
- EXTEND SOLAR ARRAY WINGS
- VERIFY PM POWER SYSTEM STATUS
- VERIFY PM-TO-ORBITER POWER TRANSFER
- EXTEND PM THERMAL RADIATORS
- VERIFY PM THERMAL SYSTEM STATUS
- VERIFY ORBITER-TO-PM HEAT REJECTION
- VERIFY PM-TO-ORBITER FUNCTIONAL INTERFACES
- CONFIRM BERTHED OPERATIONS READINESS

6.3 MISSION OPERATIONS

POWER MODULE FLIGHT OPERATIONS

- The Power Module flight operations consist of two major phases. Phase I is the initial placement Verification Mission leading to IOC. The objective of this phase is to demonstrate capabilities of the PM system to operate in conjunction with the Orbiter vehicle (sortie) and as a free-flyer vehicle.

- The shuttle will transport the PM to low earth orbit (235 nm). Using the Orbiter remote manipulator system (RMS), the PM will be removed from the cargo bay and positioned and berthed to the Orbiter vehicle. All functional capabilities of the PM will be demonstrated for sortie mode operations with the Orbiter. Following successful performance verification, the PM will be separated from the Orbiter and will be operated in a free-flyer mode to demonstrate all systems performance. Following PM successful operations any attached and complete payloads must demonstrate successful operation and be left in an operational mode. The PM will be conditioned for on-orbit storage if no payloads exist. Following the successful completion of these operations, the shuttle Orbiter will return to earth.

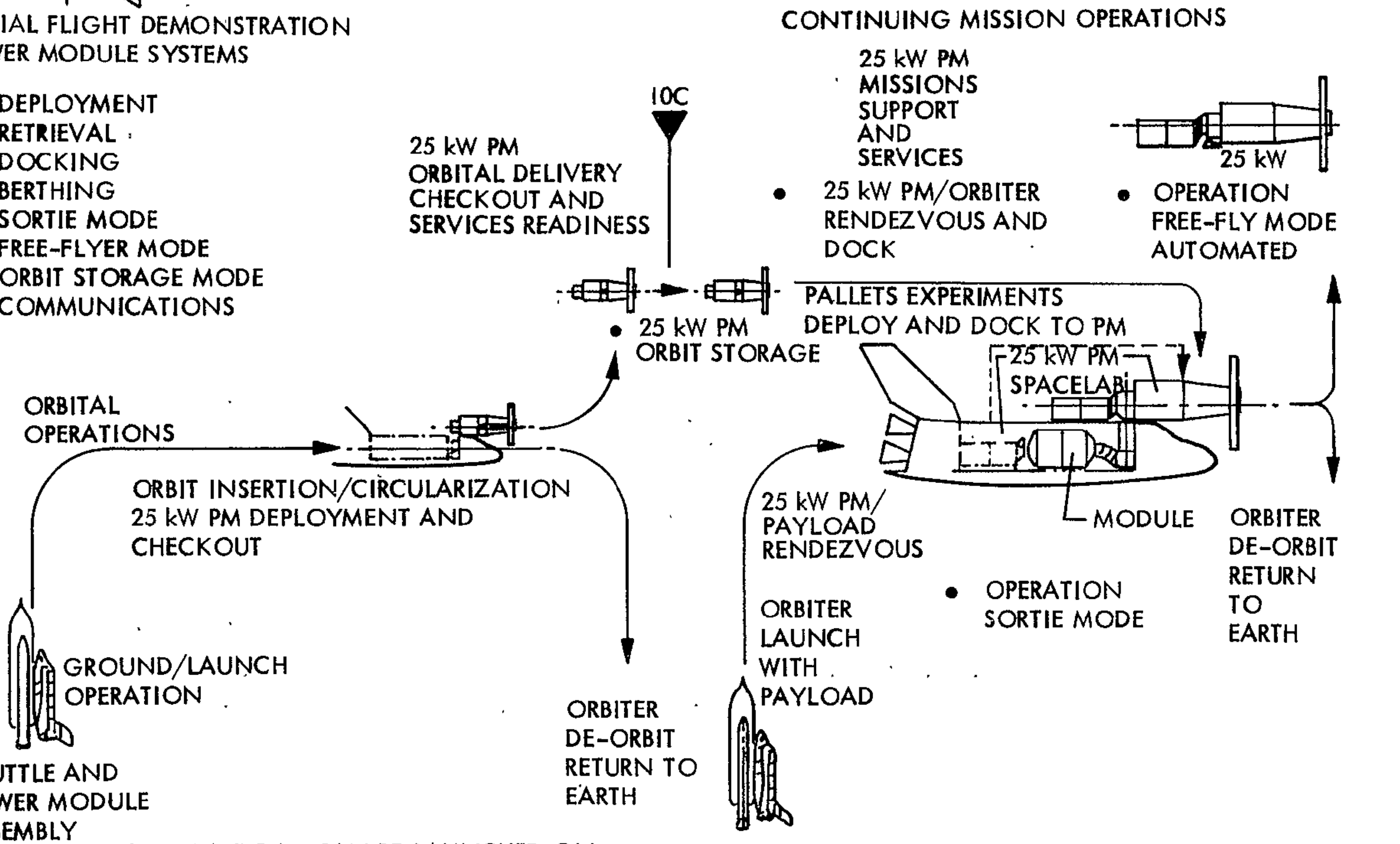
- Following a review and favorable assessment of the PM demonstration, the PM will have attained IOC status, and Phase II flight mission operations will begin.

- Phase II will consist of active payload support missions. These will be conducted in both Orbiter sortie mode and in the free-flying mode. The Orbiter will perform rendezvous and berth to the PM for sortie mode missions. For free-flyer missions, the Orbiter will transfer payloads and through RMS and E VA attach and recover payloads from the PM. Maintenance will be performed on the PM in orbit by Orbiter docking and astronaut E VA activities.

POWER MODULE FLIGHT OPERATIONS

INITIAL FLIGHT DEMONSTRATION
POWER MODULE SYSTEMS

- DEPLOYMENT
- RETRIEVAL
- DOCKING
- BERTHING
- SORTIE MODE
- FREE-FLYER MODE
- ORBIT STORAGE MODE
- COMMUNICATIONS



NOTE: PAYLOAD CAN BE LAUNCHED ON INITIAL PLACEMENT FLIGHT

VERIFICATION MISSION ACTIVITIES

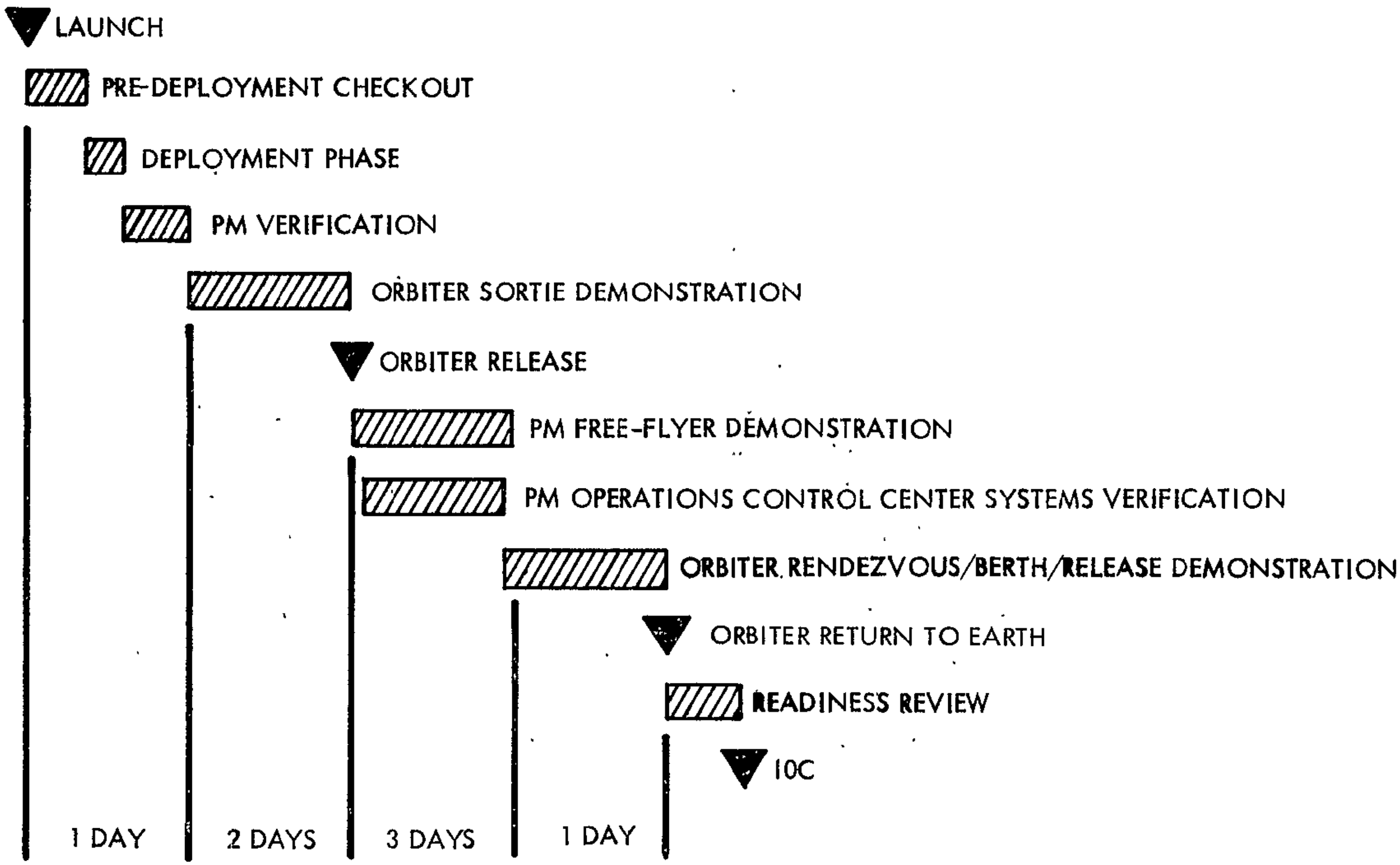
- The Power Module System verification activities are shown in this chart. Predeployment, deployment, and PM verification checkout can be performed in one day.

- Demonstration of Orbiter/PM sortie mode operations is estimated to require two days. During this time active interfaces with the PM electrical power subsystem, thermal control subsystem, and attitude control subsystem shall be exercised in various orientation modes to demonstrate system capability. Communication interfaces and procedures between the Orbiter Ground Operations Control Center and the PM Operations Control Center will be demonstrated.

- A key aspect of the PM free-flyer demonstration will be to verify the capability of the PM Operations Control Center to control and monitor the PM flight operations. These activities are estimated to require three days.

- Capabilities of the Orbiter to rendezvous with the free-flying PM will be demonstrated. The Orbiter shall then demonstrate capability to precondition the PM for berthing by commanding retraction of Solar Arrays, thermal radiators, and positioning of antennas. Recapture by the RMS will be executed, followed by a berthing and subsequent release demonstration. These operations are estimated to require one day. The Orbiter will then return to earth following this 7-day Verification Mission.

VERIFICATION MISSION ACTIVITIES

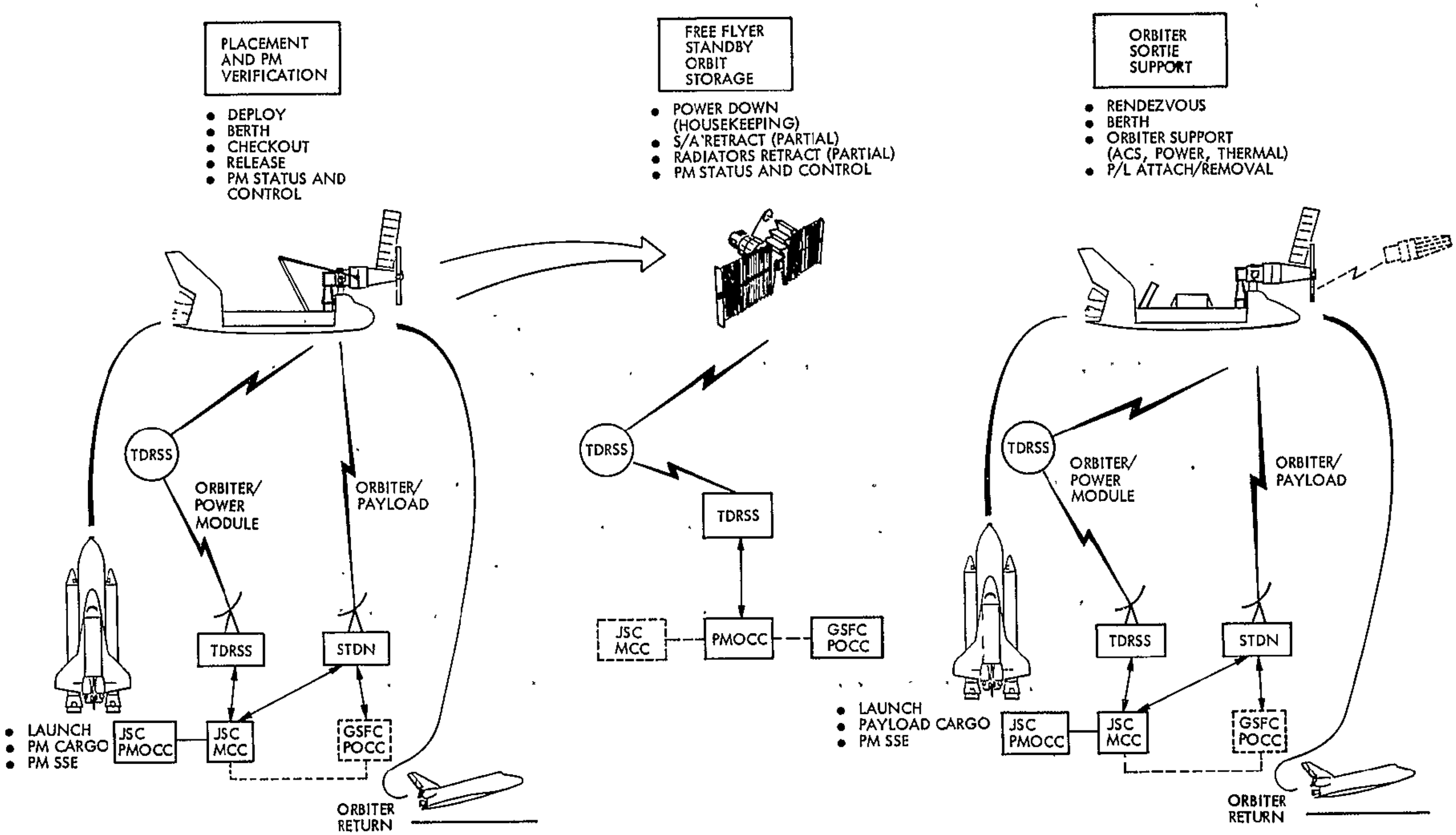


MISSION OPERATIONS: ORBITER SORTIE

- The scope of the Power Module System Operations is shown on the next two charts. Seven different flight operating modes are depicted indicating the major flight performance functions and operating interfaces. Operational elements of the Space Transportation System, communications links with the TDRS network, and Orbiter, Power Module, and payload ground control centers will be required to support these mission phases as shown. During those flights when the Orbiter is involved (PM placement and verification, Orbiter sortie, PM maintenance/growth and PM recovery), primary mission control is under Shuttle authority. Communications and command control shall be exercised by the Shuttle Mission Control Center (MCC) at the Johnson Space Center. For PM ground control it is assumed that a Power Module Operations Control Center (PMOCC) will be operating and supportive to MCC. A Payload Operations Control Center (POCC) may be supporting MCC depending on the complement of the flight payloads.

- When the PM is operating in the free-flyer orbit storage (inactive) mode, it is assumed that command and control functions will be directed through a PMOCC. General status and flight performance/planning data for the PM will be furnished to Shuttle MCC and the Goddard Space Flight Center POCC as required.

MISSION OPERATIONS: ORBITER SORTIE

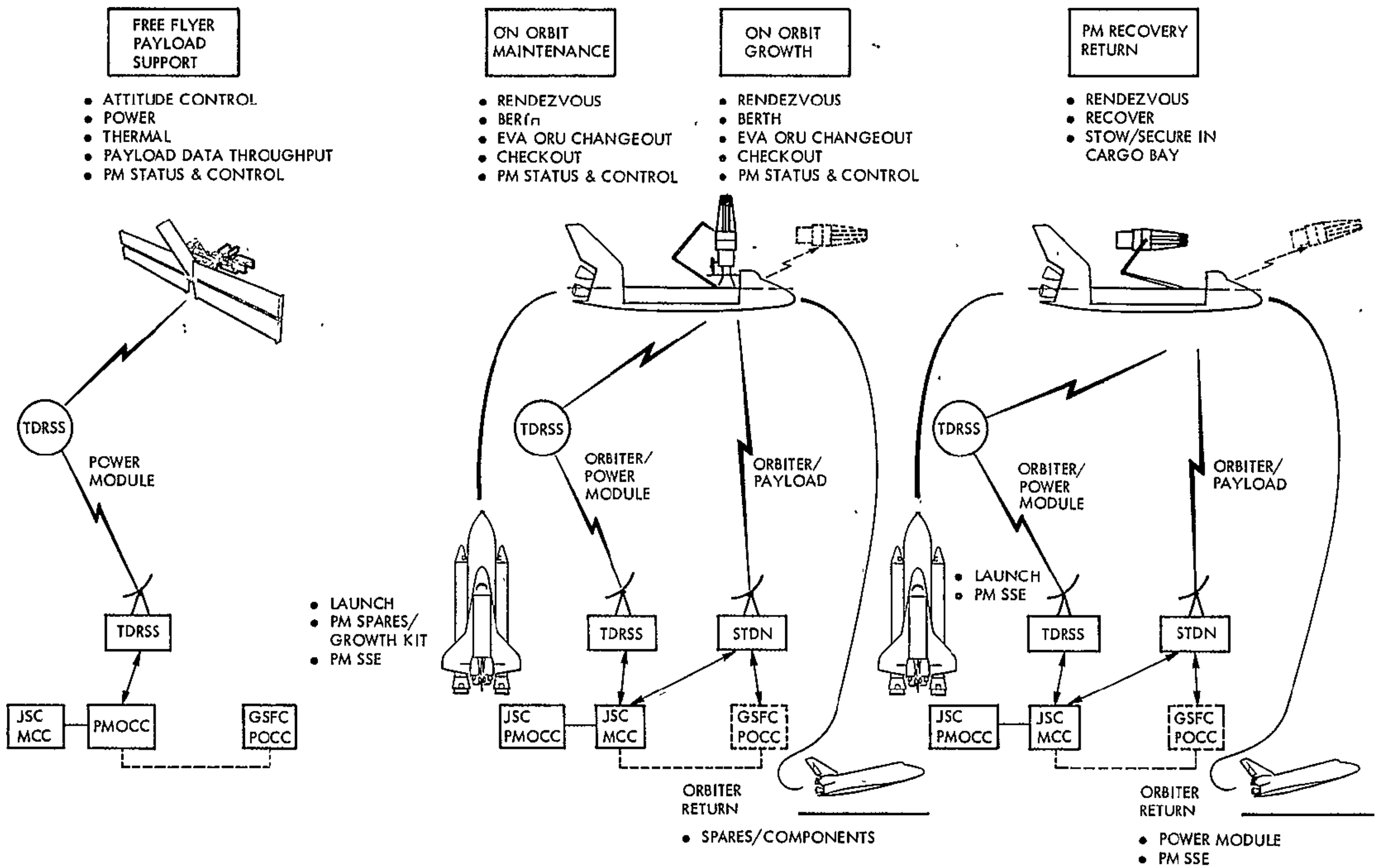


MISSION OPERATIONS: FREE-FLYER AND MAINTENANCE

- The active free-flyer mode, as shown on the chart, reflects Power Module (PM) support to attached payloads. For this case it is assumed that direct PM operations are controlled by the PMOCC. Control of attached payloads will be the responsibility of the POCC working through the PMOCC.

- Also illustrated on the chart are on-orbit maintenance and growth, and PM recovery and return operations.

MISSION OPERATION: FREE FLYER & MAINTENANCE



ON-ORBIT CAPTURE/BERTHING

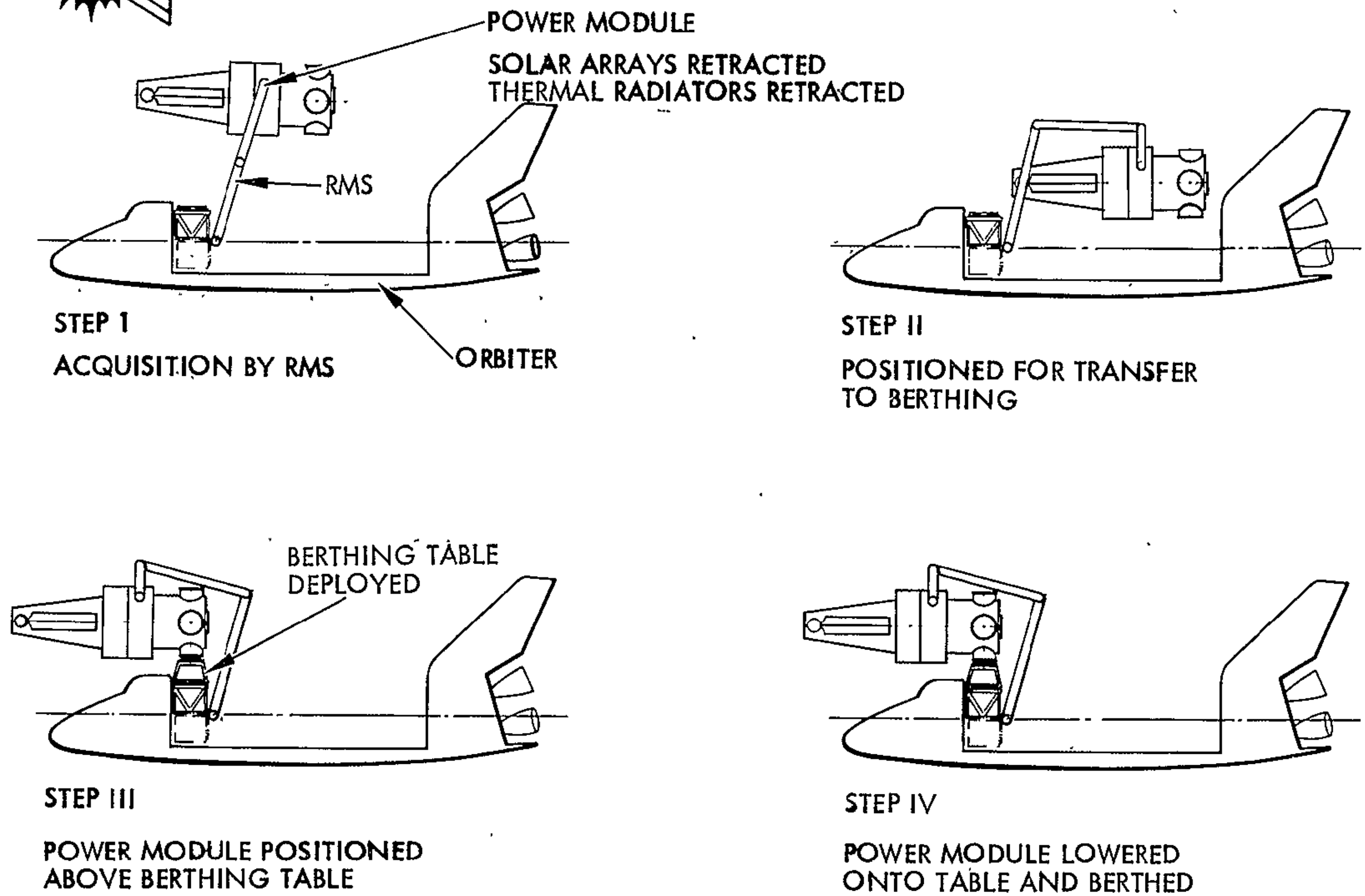
- In a typical operations sequence the Orbiter will arrive in the vicinity of the PM, which will be in a free-flying mode, and will verify and/or condition the PM for capture and berthing. All PM radiators, solar arrays, and antennas will be retracted; the CMGs will stabilize the vehicle during the berthing maneuvers. All active maneuvering will be performed by the Orbiter.

- The Orbiter will rendezvous with the PM and will use the RMS to grapple and capture the PM. The RMS will then be used to translate the PM and guide it into an alignment position over the mating-attachment guide cones. Then the RMS will position the PM on the berthing ring and the latch mechanisms will be engaged, thus securing the PM.

- After berthing, the interface umbilicals will be automatically connected. The RMS can then be released and stowed or used for other tasks as required during maintenance activities.

- Berthing release will be performed in a reverse sequence.

ON-ORBIT CAPTURE/BERTHING



ORBITER FLIGHT UTILIZATION ANALYSIS

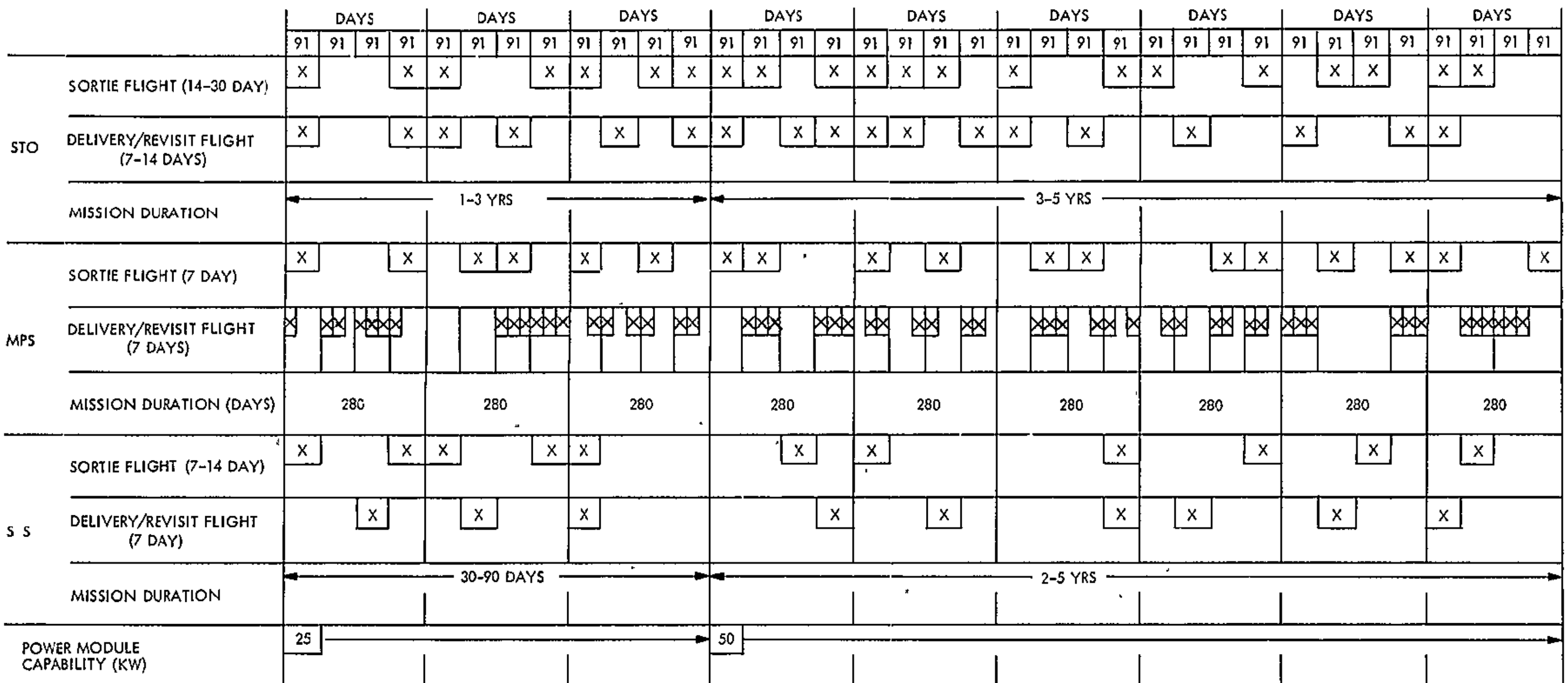
- There are two key systems required in the conduct of on-orbit space power operations – the Orbiter and the Power Module.

- Each system will require long range, near term, and on-line mission flight planning. The next chart delineates the Orbiter utilization based on Scenario I (1983-1991) in orbit as an example (57^o). All mission planning for the Orbiter-PM must include flight utilization analyses similar to these presented here.

- Assuming two types of Orbiter missions (sorties and delivery/revisit) the analysis indicates a modest schedule frequency. The Orbiter utilization rate data (synonymous with PM hookups/berthings) ranges from 9 to 29 flights/year (for all 3 orbits). This number is within the STS model capability of 60 flights/year. The utilization rate stated above is not NASA approved and no attempt was made to redesignate or modify numbers of planned flights/year appearing in the model.

ORBITER FLIGHT UTILIZATION ANALYSIS

SCENARIO I, 57° ORBIT



YEAR	POTENTIALLY SHARED SORTIE FLT* TOTALS/YR △ (per ≈91 day span)				POTENTIALLY SHARED D/R FLT* TOTALS/YR △ (per ≈91 day span)				NO. OF HOOKUPS/YR
83	1	0	1	1	2	2	2	1	9
84	1	1	1	1	1	1	3	3	12
85	2	0	2	2	1	2	1	2	12
86	2	2	1	1	1	3	1	3	14
87	1	1	2	0	2	1	1	2	10
88	1	1	1	1	1	3	1	2	11
89	1	0	1	1	1	1	2	2	9
90	0	1	1	1	3	1	0	3	10
91	1	1	0	1	1	3	3	0	10

▭ = ≈91 DAY SPAN

▭ = 1 FLIGHT

▯ = 1 FLT IN MPS ONLY

△ ASSUMES PAYLOAD FLIGHT SHARING

6.4 GROUND SUPPORT LOGISTICS & REFURBISHMENT

SCENARIO I GROUND OPERATIONS SUPPORT REQUIREMENTS

- Ground operations activity centers around the preparation of Power Module vehicles for launch at Kennedy Space Center (KSC). In addition, Power Module space support equipment is serviced and installed in the Orbiter launch vehicle to directly interface and support Power Module launch delivery placement, Orbiter sortie and payload delivery/revisit flight operations.

- The launch and flight support activities are shown for the 1983-1991 era, conforming to the Scenario I Power Module evolution sequence (see Part III Volume 1, page 2-13).

SCENARIO I GROUND OPERATIONS SUPPORT REQUIREMENTS

GROUND PROCESSING

- BASIC PM LAUNCH PROCESSING (25 kW/50 kW)
- PM GROUND REFURBISHMENT PROCESSING
- PM GROWTH KIT LAUNCH PROCESSING (50-100 kW)
- SPACE SUPPORT EQUIPMENT

LAUNCH SCHEDULE

CALENDAR YEAR

TASK/TYPE	83	84	85	86	87	88	89	90	91
25 kW	ΔFV-1			ΔFV-1R$_1$		ΔFV-1R$_2$			
50 kW				ΔFV-2		ΔFV-4	ΔFV-5		
60 kW *					ΔFV-3 *				
100 kW									ΔFV-6
(50→100) KIT								ΔFV-4 KIT	
FLIGHT RECOVERY				∇		∇			
SORTIE	2	4	6	9	6	9	12	10	11
P/L DELIVERY/REVISIT	7	8	6	14	10	14	17	19	18

* PART OF GEO PLATFORM FOR REFERENCE ONLY

GROUND OPERATIONS SUPPORT SUMMARY – KSC

- The types of ground operations support activities for the Power Module and the support facilities are summarized on the facing chart.

- Major ground operations will focus on processing of the flight vehicles. There are eight vehicles scheduled for processing during the 1983-1991 period per Scenario I. Of the eight, seven are placement flights and one is a growth kit addition flight. Activities for flight vehicles and kits will include prelaunch preparation and checkout of the vehicles/kits and then installation/integration in the Orbiter vehicle. Space support equipment will be integrated in to the Orbiter before launch and removed after landing return of the Orbiter.

- For a recovery return flight, both SSE and the Power Module/elements will be removed after landing return of the Orbiter.

- The primary support for Orbiter sortie and payload addition/removal flights will be to install the SSE and pallets in the Orbiter before launch and to remove the SSE and pallets from the Orbiter after landing return.

- The processing of payload pallets will not be directly related to PM ground operations. It is assumed this responsibility will be handled by other cognizant NASA authority.

GROUND OPERATIONS SUPPORT SUMMARY-KSC

SUPPORT TASK \ OPERATION MODE	POWER MODULE PLACEMENT	POWER MODULE GROWTH KIT ORBIT EXCHANGE	POWER MODULE RECOVERY	SORTIE SUPPORT TO ORBITER	PAYLOAD ADDITION/ REMOVAL
FREQUENCY	PLACEMENT	GROWTH KIT ADDITION	REFURBISHMENT RETURN	AS SCHEDULED	AS SCHEDULED
LAUNCH FACILITIES					
LANDING FIELD	X	X	X	X	X
PROCESSING	O AND C	O AND C			O AND C
INSTALL IN ORBITER	OPF	OPF		OPF	OPF
REMOVE FROM ORBITER	OPF	OPF	OPF	OPF	OPF
ORBITER					
PRE LAUNCH	INSTALL-RMS (1) PM SSE PM	INSTALL-RMS (2) PM SSE KIT CONTAINER	INSTALL-RMS (1) PM SSE	INSTALL-RMS (1) PM PSSE	INSTALL RMS (1) SSE PAYLOAD PALLET
FLIGHT	• ORBIT PLACEMENT • REMOVE & BERTH PM • CHECKOUT	• RENDEZVOUS-BERTH • EVA CHANGEOUT EQUIPMENT • RECOVER EXCHANGE EQUIP IN CARGO BAY	• RENDEZVOUS/ BERTH • RECOVER PM-RMS/EVA • STOW IN CARGO BAY	• RENDEZVOUS/ BERTH • FLY SORTIE MISSION	• RENDEZVOUS/BERTH • REMOVE P/L PALLET AND INSTALL ON PM • REMOVE P/L PAYLOAD PALLET FROM PM AND INSTALL IN CARGO BAY
POST LANDING	REMOVE SSE	• REMOVE SSE • REMOVE EQUIPMENT CONTAINER • REMOVE RMS	• REMOVE SSE • REMOVE PM • REMOVE RMS	• REMOVE SSE • REMOVE RMS	• REMOVE SSE • REMOVE PALLET

LOGISTICS SUPPORT CONCEPT

- The logistics support concept for the PM will encompass the four major areas as shown on this chart.

- Data items assigned to Logistics are logged and accounted for by monitoring the due date list and requests are made for the stored masters in time to permit preparation of revisions as necessary. Thus, Logistics develops detailed task, schedule, and budget planning for all accountable items. All modules for installation, ground or on-orbit repairs will have a top-down breakdown appropriately identified. This provides a uniform method for tracking to the next higher assembly.

- Transportation, packaging, storage and handling requirements for the PM, S/A, GSE, and spares and spare parts will be the Logistics responsibility. From the factory verification sequence to on-site delivery (airfield), PM, Pre-installation preparation (O/C), Orbiter integration (OPF), appropriate modules/spares and GSE (cranes, transporter, slings, etc.) will be made available. Subcontractors will support as required.

- In addition to ensuring ORUs are available as required, the Logistics function is to define the simulation (semi-hard mockups, neutral buoyancy) required to establish effective on-orbit repair. Data will be used to clarify on-orbit procedures and establish time-to-accomplish the on-orbit tasks.

LOGISTICS SUPPORT CONCEPT

LOGISTICS SUPPORT CONCEPT

INVENTORY MANAGEMENT, SPARES SELECTION/ PROVISIONING

- LOG, ACCOUNT, MAINTAIN DATA LIST
- TOP-DOWN IDENTIFICATION FOR TRACKING

TRANSPORTATION, PACKAGING AND HANDLING

- GROUND OPS FLOW AND GROUND SUPPORT EQUIPMENT ALTERNATIVES
- FACTORY AND PAD ACTIVITIES DEFINITION

TRAINING AND PROCEDURES

- BASELINE NASA PROGRAMS & CAPABILITIES
- OPERATIONS AND MAINTENANCE (O & M)

ON-ORBIT REPAIR

- SUPPLY/SPARES SUPPORT
- TASK/TIMELINES
- SIMULATION

LAUNCH

REFURBISH

ON-ORBIT

- FACTORY TO LAUNCH
- SPARES TO FACTORY
- SPARES TO ORBITER PROCESSING

- O & M PROCEDURES
- TRAINING CLASSES

- ON-ORBIT PROCEDURES
- MOCKUP CONSTRUCTION

- TRAINED PERSONNEL
- DELIVER PECULIAR SUPPORT EQUIPMENT
- DELIVER SPARES
- TRANSPORT PM
- MANAGE INVENTORY
- PRODUCE O & M PROCEDURES
- DIRECT REPAIR

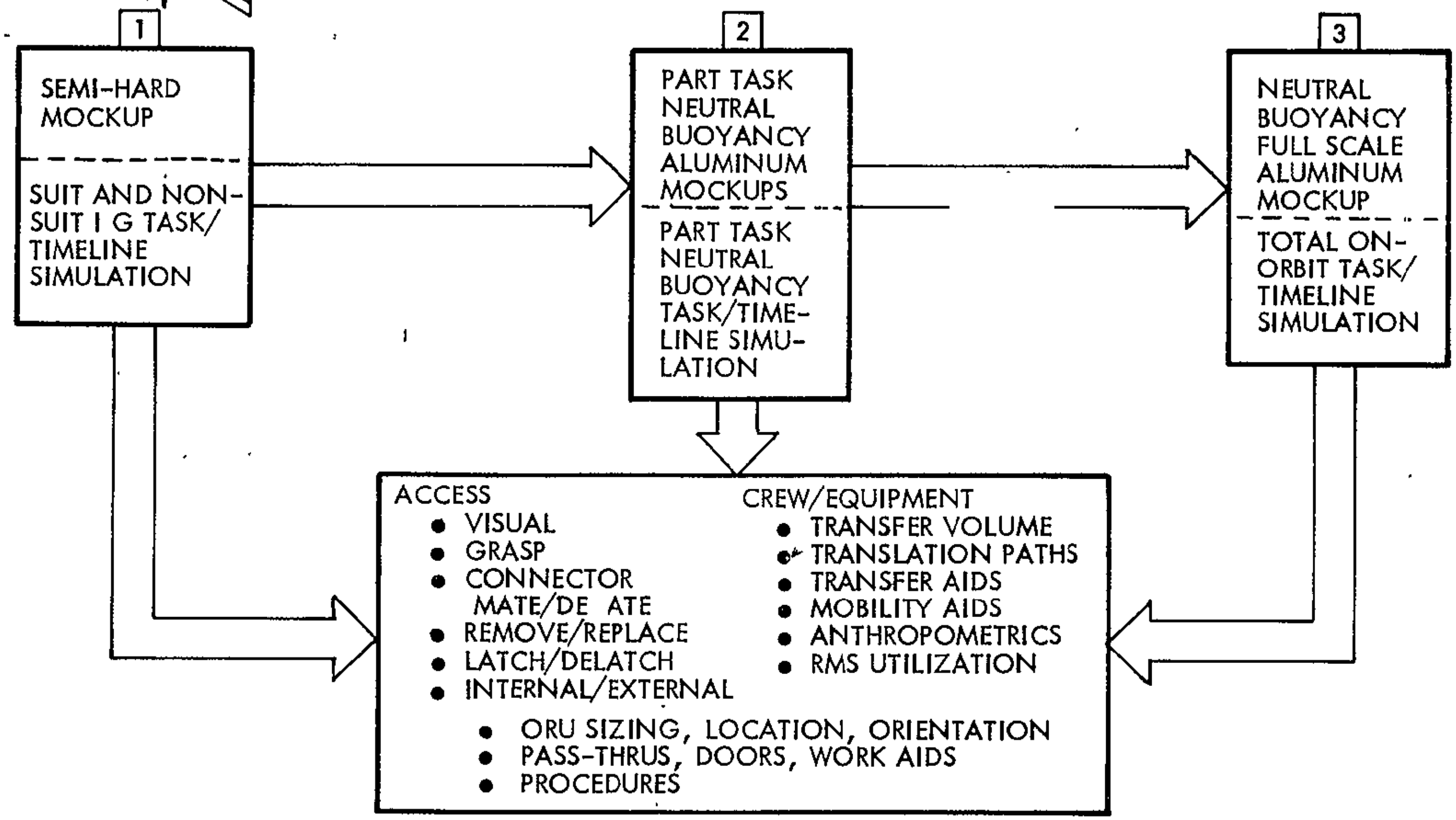
POWER MODULE OPERATIONS SIMULATION AND TRAINING

- As part of logistics planning, training equipment, materials, facilities, and services are specified for all phases of the PM program. Training includes preparations for operations (P/L C & D), on-orbit maintenance and ground maintenance. Course lists and training aids or devices for each course will be developed.

- Training by simulation is a key technique for conducting and developing on-orbit maintenance/growth concepts and procedures. This chart illustrates simulation techniques applicable to manned (both shirtsleeve and suited) PM ground and on-orbit tasks. Neutral buoyancy simulation (2, 3) is expensive and requires metal (or suitable substitute) mockups, must be carefully planned, and requires substantial test support equipment and personnel. However, 1-g testing (1) can be initially accomplished with soft mockups for preliminary layout and interface analysis. Furthermore, soft mockups (foamcore, wood, etc.) permit simple and rapid reconfiguration at minimum expense.

- The three simulation techniques recommended for the PM program are shown on the chart. NASA/MSFC has neutral buoyancy facilities ideal for simulation types 2 and 3. All the simulation types support each other and in actuality are a constant iterative process. The fall-out results assist in solving the ground and on-orbit repair/growth problems related to man. The schedule shows present estimates of when the various types of simulation will be required.

POWER MODULE OPERATIONS SIMULATION AND TRAINING



	83	84	85	86	87	88	89	90	91
25 kW	① △			① △		△			
50 kW			①	△		△	△		
60 kW*				①	△				
100 kW									△
(50-100) KIT						①②③		△	

*SHOWN FOR REFERENCE ONLY

1-2-3 TYPE TRAINING/SIMULATION

(1-2-3) △ START TIMES AND TYPE TRAINING/SIMULATION

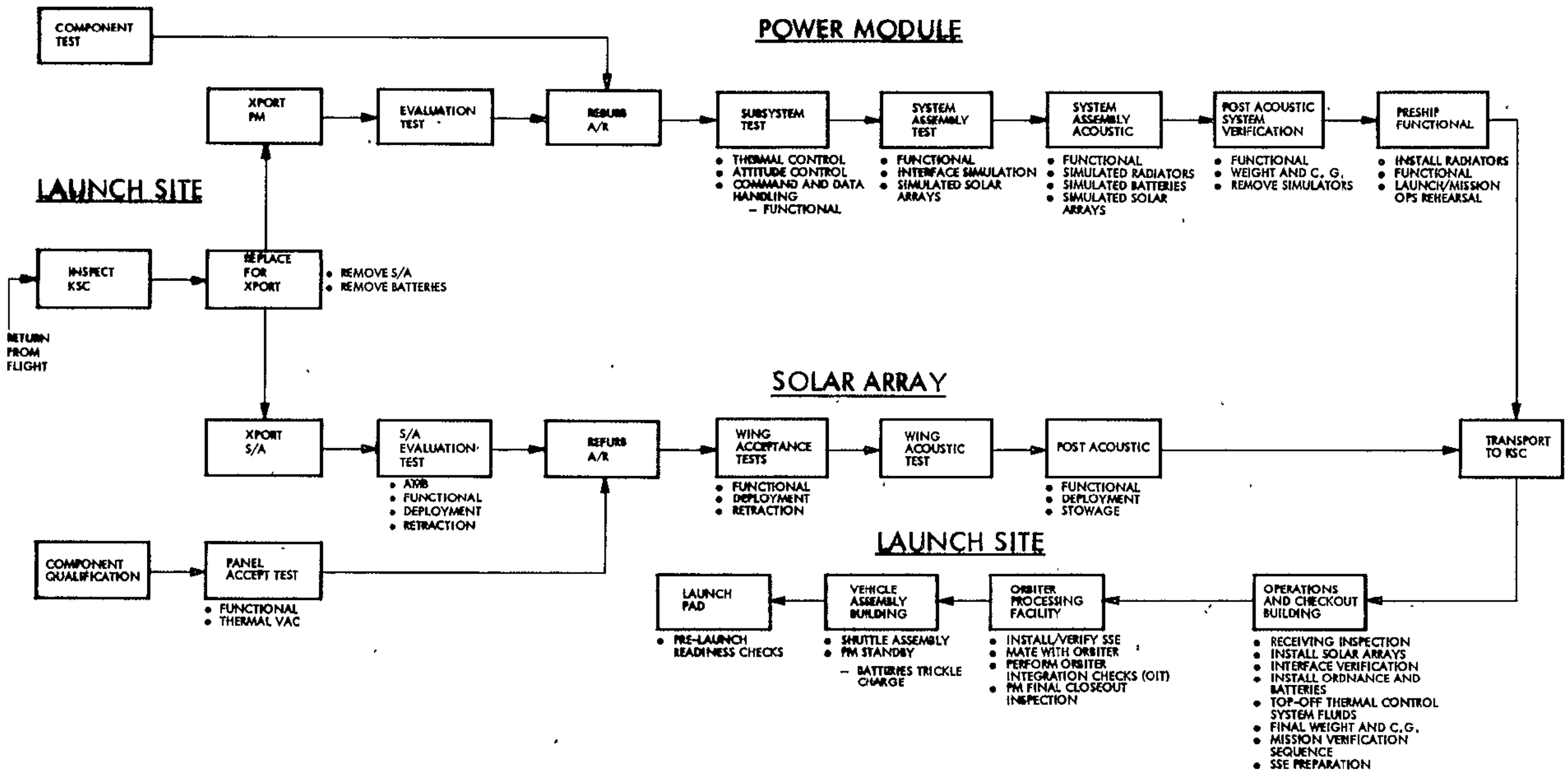
GROUND-BASED REFURBISHMENT AND TEST PROGRAM

- The objective of ground-based refurbishment will be to recondition the Power Module (PM) with capability for a five-year orbit life. Refurbishment will be conducted at the factory to make use of the component and subsystems assembly and test capability which will exist there.

- The refurbishment test program is shown in this chart. Refurbishment philosophy will be to minimize disassembly and to use flight performance records, visual inspection, and initial evaluation testing to establish rework/replacement requirements. After return flight in the Orbiter, the PM will be removed from the payload bay in the Orbiter processing facility and transported to the Operations and Checkout building for inspection. The solar arrays and battery modules will be removed and the PM, solar arrays, and battery elements will be shipped to the factory via air carrier (Guppy). At the factory, a systems functions evaluation test will be performed on the PM and a subsystem functional evaluation test will be performed on the solar arrays. Subsystem components will be replaced as required. Subsystem and system tests will be performed and a system assembly acoustic test of the PM will be conducted utilizing simulators for solar arrays, batteries, and radiators. Final acceptance of the refurbished PM at the factory will include an all-systems sequenced mission simulation test. With successful completion of these tests at the factory, the flight equipment will be considered "ready for flight".

- Ground operations processing at the launch site will be identical to the sequence and operations used for an initial PM placement flight. The refurbished vehicle will be prepared at the Operation and Checkout building. Space support equipment and the PM will be installed in the Orbiter at the Orbiter Processing Facility and prelaunch readiness checks will be performed at the launch pad.

GROUND-BASED REFURBISHMENT AND TEST PROGRAM

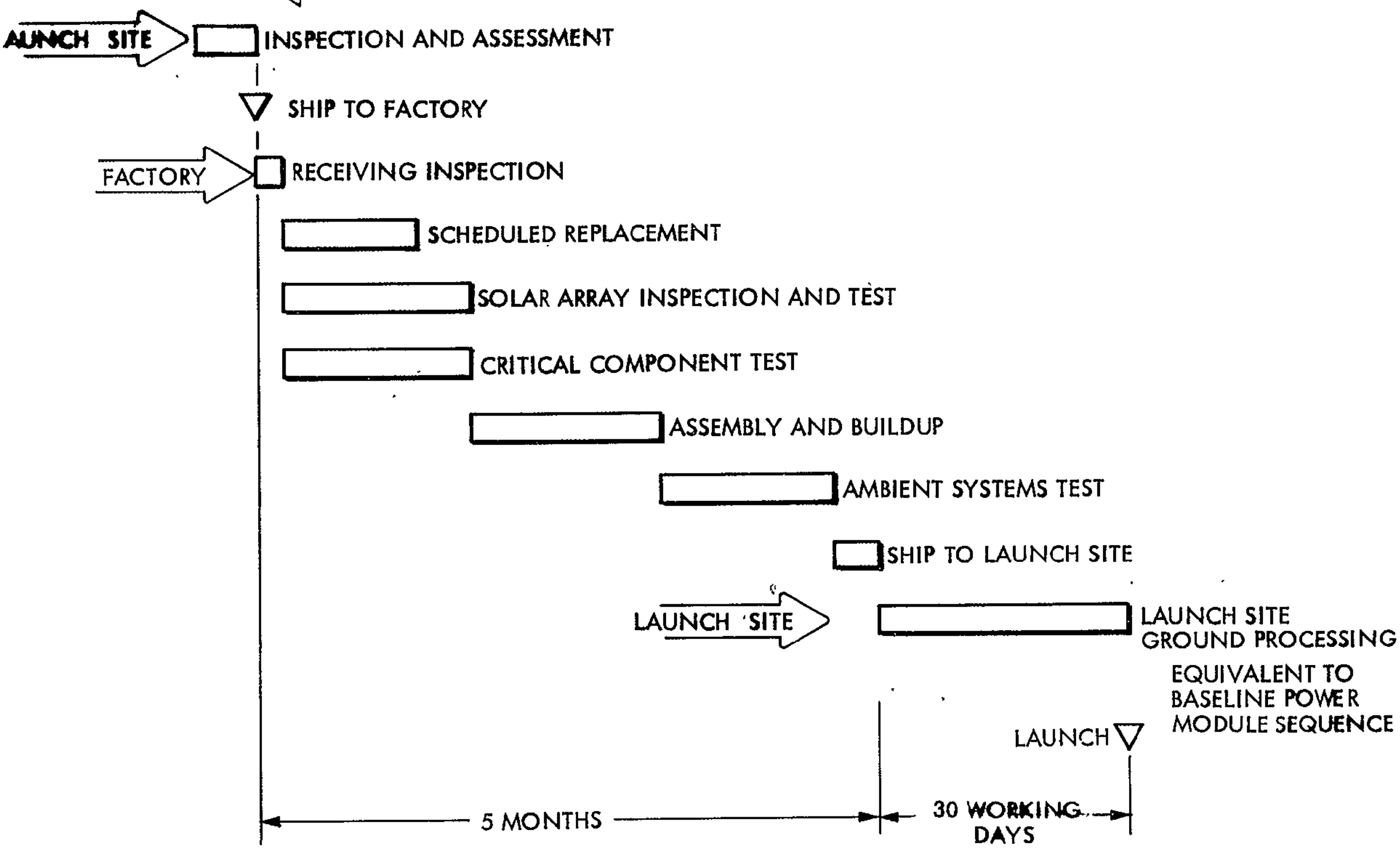


GROUND REFURBISHMENT SEQUENCE

- A typical Power Module refurbishment sequence and timeline is shown on this chart. This sequence is oriented to the test program discussed on the previous chart and shows a relatively rapid rework/turnaround capability.

- The five months allocated at the factory is estimated as realistically achievable in the 1986-1988 time period since this will be four years into the PM program. The thirty operational days at the launch site is identical to the time span for processing of the first flight vehicle.

GROUND REFURBISHMENT SEQUENCE



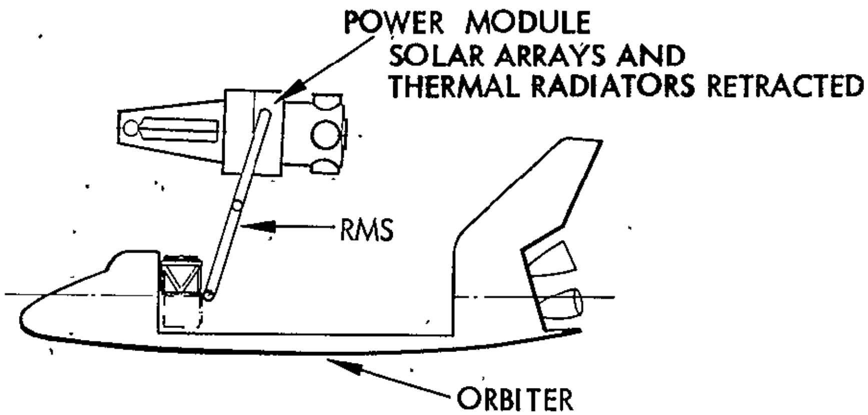
BERTHING FOR ON-ORBIT MAINTENANCE

6.5 ON-ORBIT MAINTENANCE/GROWTH

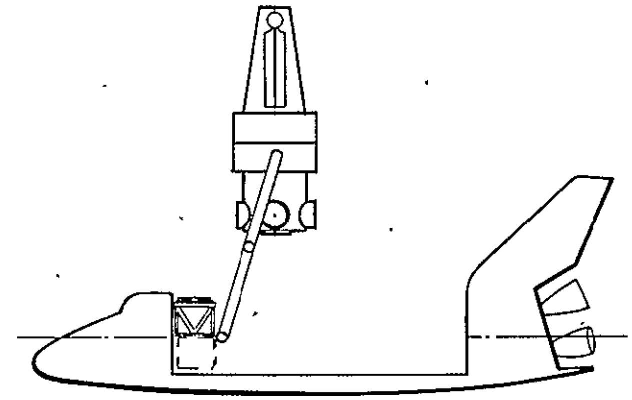
- The four phases of berthing for on-orbit maintenance operations are illustrated on the chart.

- In the event a payload pallet is attached to the aft berthing port of the power module, prior to this maintenance berthing operation the pallet must be detached and stowed in the Orbiter payload bay.

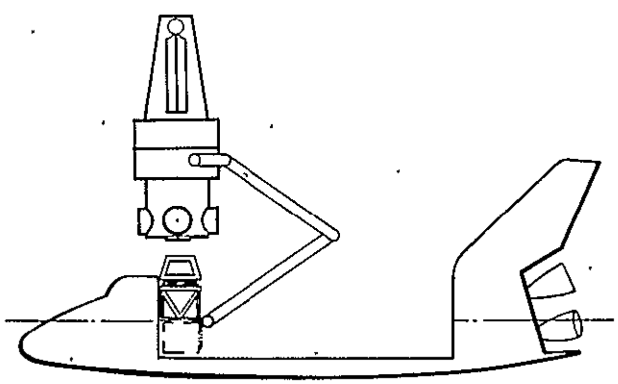
BERTHING FOR ON-ORBIT MAINTENANCE



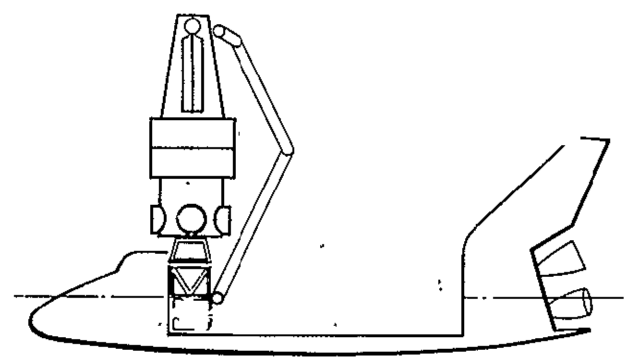
STEP 1
ACQUISITION BY RMS



STEP II
POWER MODULE ROTATED



STEP III
POWER MODULE TRANSLATED FORWARD
ABOVE BERTHING TABLE.
BERTHING TABLE EXTENDED.



STEP IV
POWER MODULE BERTHED
MAINTENANCE PLATFORM & ACCESS
MAST EXTENDED

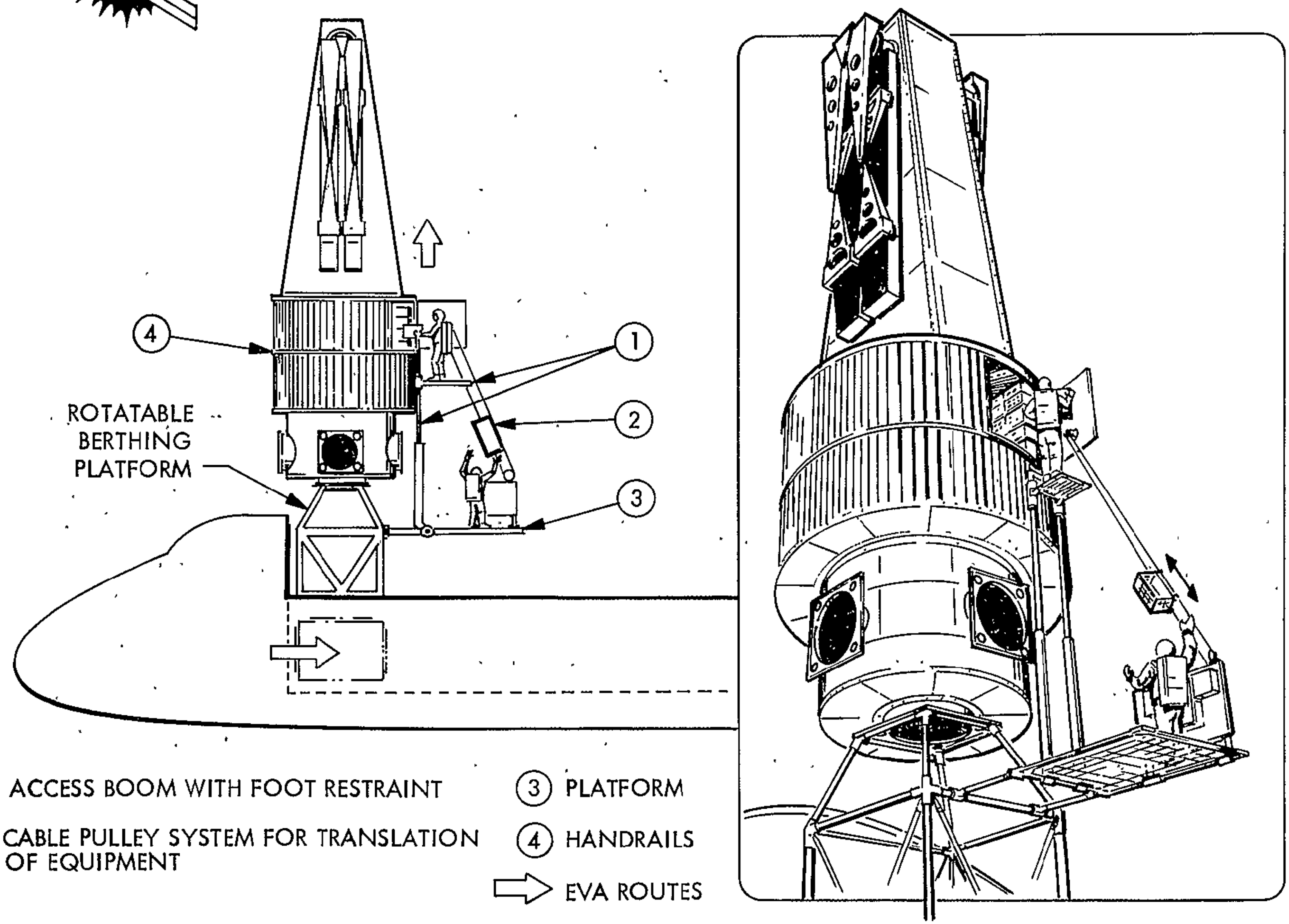
ON-ORBIT MAINTENANCE

- Maintenance features have been incorporated into the basic PM design concept. These features will also enhance on-orbit growth. Major elements of subsystem equipment are grouped within one location in the core vehicle and located with consideration for accessibility via EVA. Equipment/module packages are designed as replaceable assemblies, Orbital Replaceable Units (ORUs). EVA-assist hand rails will be incorporated into the PM design, located along major access areas. The concept design for replacement module ORUs will be for one crewman removal/replacement.

- The operations concept for on-orbit maintenance is shown in this facing chart. The RMS will be used for translation of large equipment modules as required. Two crewmen will perform EVA maintenance tasks for the PM. Standard NASA support equipment, such as tools, tethers, restraints, work stands, and lighting will be considered as PM design baseline. The utilization of universal or multimission ancillary equipment will minimize crew training, simplify operational requirements, and provide the most cost-effective approach. One crewman operates at the work-station position which is moved to the desired location by an extendable boom attached to a support pallet in the Orbiter payload bay. The work station has foot restraints, tool caddy, tether attach points, controllable lighting, and support areas for units to be replaced. The second crewman remains on the pallet and transports packages to the work station on a cable pulley. Details regarding payloads, procedures and equipments will require further study.

- Time estimates for ORU changeouts range from 64 minutes (solar array drive assembly) to 11 minutes (sun sensor).

ON-ORBIT POWER MODULE EVA OPERATIONS— PLANNED AND CONTINGENCY MAINTENANCE



(1) ACCESS BOOM WITH FOOT RESTRAINT
(2) CABLE PULLEY SYSTEM FOR TRANSLATION OF EQUIPMENT
(3) PLATFORM
(4) HANDRAILS
EVA ROUTES

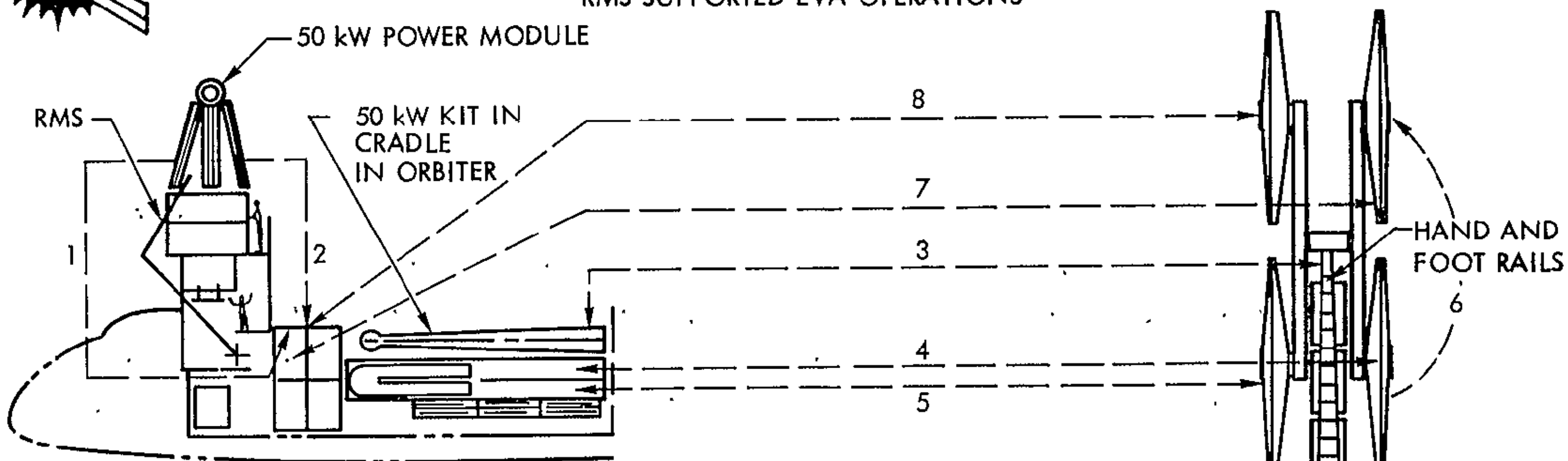
ON-ORBIT GROWTH: 50 kW TO 100 kW

- The 50 kW flight vehicle can be evolved to a 100 kW flight vehicle on-orbit by the addition of a 50 kW kit carried aloft by the Orbiter. The sequence of RMS-supported EVA operations to remove elements of the 50 kW flight vehicle and install new elements in orbit is shown on the facing chart.
- The 50 kW kit is delivered in orbit as a partial cargo load by the shuttle. After rendezvous and berthing with the Power Module, the changeout operation begins using two astronauts who egress into the cargo bay from the airlock.
- On-orbit growth changeout of 50 kW to 100 kW can be performed with one EVA (2 crewmen and an RMS operator) in an estimated 3-hour time span. The sequence consists of eight functional operations as shown on this chart. These changeout activities can be performed with one RMS, assuming the use of two holding attachment fixtures which will be located in the cargo bay for the purpose of temporarily holding the two removed solar array wings while the RMS is used for other sequences.
- The detailed times associated with the tasks to remove, replace, test and ingress into the airlock are given in Appendix C of this volume. A time-task summary is given below:

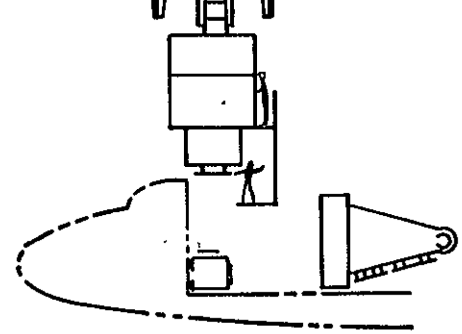
Egress A/L, set up work station	21.5 minutes
Remove/Stow "old" S/A (2)	17.0
Remove/install "new" structure, S/A's (2)	28.5
Rest period	5.0
Unlatch/install "old" S/A's (2)	26.0
Remove/stow work station, RMS, camera	21.0
Perform operational checkout	15.0
Safety check P/L bay, ingress A/L	30.0
Subtotal	164.0
10% Contingency	16.0
Total EVA Time	180.0

ON-ORBIT GROWTH: 50 kW TO 100 kW

RMS SUPPORTED EVA OPERATIONS



50 kW POWER MODULE INSTALLED UPON BERTHING PLATFORM IN ORBITER



100 kW POWER MODULE FULLY ASSEMBLED

SEQ 1 REMOVE 1/2 "OLD" 50 kW S/A TO HOLD* (8.5 MIN)	SEQ 2 REMOVE 1/2 "OLD" 50 kW S/A TO HOLD* (8.5 MIN)	SEQ 3 REMOVE/INSTALL S/A STRUCTURE EXTENSION (12.5 MIN)	SEQ 4 REMOVE/INSTALL 1/2 "NEW" 50 kW S/A (14 MIN)
SEQ 5 REMOVE/INSTALL 1/2 "NEW" 50 kW S/A (14 MIN)	SEQ 6 ROTATE S/A SUPPORT BOOM TOP TO BOTTOM (12 MIN)	SEQ 7 DETACH/INSTALL 1/2 "OLD" 50 kW S/A (13 MIN)	SEQ 8 DETACH/INSTALL 1/2 "OLD" 50 kW S/A (13 MIN)

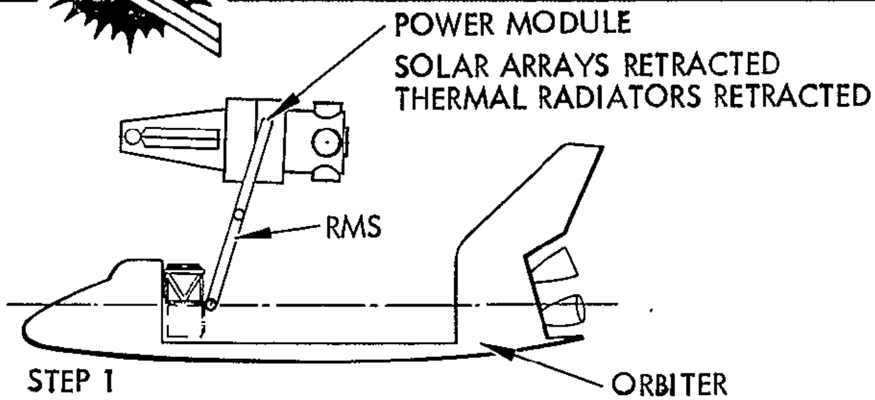
ON-ORBIT CAPTURE/STOWAGE

- In the event that major refurbishment (or growth-modification) is required, the Power Module (PM) will be recaptured and returned to earth. Ground-based refurbishment is described in Section 6.4.

- In a typical operational sequence the Orbiter will arrive in the vicinity of the PM, which will be in a free-flying mode, and will verify and/or condition the PM for capture and stowage. All PM radiators, solar arrays, and antennas will be retracted; the CMGs will stabilize the vehicle during the capture/stowage maneuvers. All active maneuvering will be performed by the Orbiter.

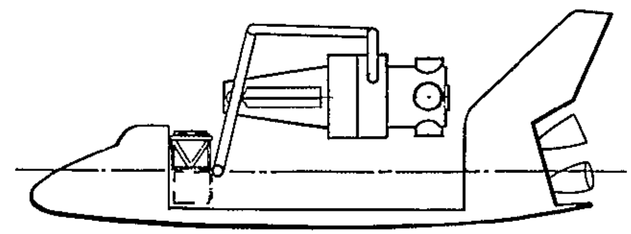
- The Orbiter will rendezvous with the PM and will use the RMS to grapple and capture the PM. The RMS will then be used to translate the PM and guide it into alignment position over the payload bay attachment fittings. Then the RMS will position the PM into the fittings and the latch mechanisms will be engaged, thus securing the PM for re-entry and landing.

ON-ORBIT CAPTURE/STOWAGE



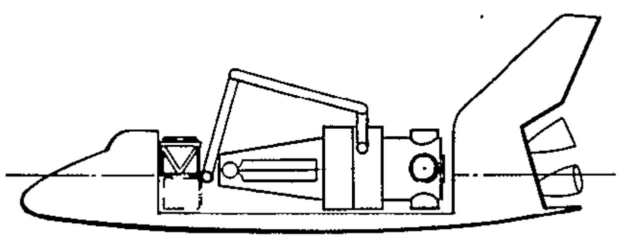
STEP I

ACQUISITION BY RMS



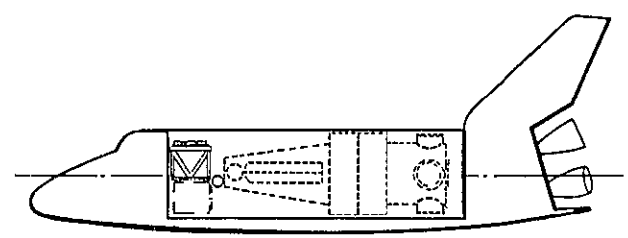
STEP II

POSITIONED TO TRANSFER
INTO ORBITER



STEP III

TRANSFERRED INTO ORBITER



STEP IV

POWER MODULE SECURED
FOR RETURN

SECTION 7
TECHNOLOGY PLANNING

STRUCTURES TECHNOLOGY DEVELOPMENT

- Major Structural/Material advances have been made through the 1970's and will continue through the 1980's. Although the technology exists now for application of composite structures, the manufacturing processes for any particular application has limited availability.

- The chart illustrates the availability of new materials processes, useful in accomplishing both weight reduction and increased rigidity for the larger growth power modules.

- The availability of graphite/metal composites are 2 to 4 years behind the graphite/epoxy composites. The improved metal composites would result in a 15% weight reduction.

- Developments in the early 1980's are expected to enable 15% structural weight reductions from conventional aluminum construction recommended for the first Power Module. In the mid and late 1980's an additional 15% weight reduction is envisioned.

STRUCTURES TECHNOLOGY DEVELOPMENT

DEVELOPMENT ITEM	PLANNED UTILIZATION	SCHEDULE 80 82 84 86
TITANIUM FASTENERS BONDING	FIXED SHEAR PANELS ON ALL STRUCTURES	
THORNELL FABRIC	EQUIPMENT PANELS; THERMAL ISOLATION	
KELVAR 49/T300/HMS WITH ALUMINUM HONEYCOMB CORE	EQUIPMENT PANELS; TRAYS FOR REMOVABLE EQUIPMENT	
GRAPHITE/EPOXY BY: • PROTRUSION • TUBE WINDING MACHINE	STRUCTURAL MEMBERS FOR 50 & 100 kW; SOLAR ARRAY SUPPORT STRUCTURE; SOLAR ARRAY BEAMS; BERTHING STRUCTURE	
GRAPHITE/ALUMINUM	STRUCTURAL MEMBERS; SHEAR PANELS, FOR 200 kW	
GRAPHITE/MAGNESIUM	STRUCTURAL MEMBERS; SHEAR PANELS, FOR 100 kW AND LARGER	

ELECTRICAL POWER SUBSYSTEM TECHNOLOGY DEVELOPMENT

- The Evolutionary Study results indicate that several technology improvements would enhance the life and/or performance characteristics of the PM Electrical Power Subsystem. The areas identified are primarily those that would significantly lower the cost ($/KWH).

- Two important areas are process development for low-cost solar cells and the incorporation of long-life, high energy-density batteries. Since the development of efficient regulators and power management devices directly affects the size of both the solar array and the battery complement, these, too, become high-payoff areas.

ELECTRICAL POWER SUBSYSTEM TECHNOLOGY DEVELOPMENT

DEVELOPMENT ITEM	PLANNED UTILIZATION	SCHEDULE 80 | 82 | 84 | 86 | 88
120 VDC (105-110 CELL) 50 AH BATTERY DEVELOPMENT AND EVOLUTION OF TECHNOLOGY	SHOULD BE USED ON ALL LEO POWER MODULES TO IMPROVE ENERGY DENSITY AND LIFE ($/KWH WOULD BE SUBSTANTIALLY REDUCED)	DEV/TEST; EVOL; PRODUCTION
CONTINUE DEVELOPMENT OF 100-200 VDC POWER MANAGEMENT EQUIPMENT AND COMPONENTS – REMOTE POWER CONTROLLERS – RELAY FUNCTIONS	100-200 AMPERE DEVICES WILL BE NEEDED AT 100-200 VDC 150 AMPERES DEVICES ARE NEEDED AT 28 VDC	DESIGN/FAB; EVOL; LOW VLTG; PROD; HIGH VLTG; PROD
HIGH POWER REGULATION TECHNIQUES (10 kW TO 40 kW)	DEVELOP COMPONENTS, TECHNIQUES, AND EMI/EMC ELEMENTS FOR MORE EFFICIENT REGULATION AND CONVERTERS	DESIGN/TEST; EVOL; PRODUCTION
MATERIAL AND DEPLOYMENT MAST DEVELOPMENT	COMPOSITE MATERIALS FOR SOLAR ARRAY CONTAINER AND ARTICULATED MAST ELEMENTS	DESIGN/TEST; EVOLUTION; PRODUCTION
SOLAR CELL ASSEMBLY AUTOMATION	DEVELOP REQUIRED PROCESSES FOR IMPLEMENTATION OF AUTOMATION FOR COST REDUCTION AS SOON AS POSSIBLE	PROCESS; EQUIPMENT; PRODUCTION

THERMAL CONTROL SYSTEM TECHNOLOGY DEVELOPMENT

- The chart shows the technology development requirements which have been identified for the PM thermal control system. The proposed thermal control design is based on existing design concepts which have been developed for other spacecraft and the Shuttle Orbiter. However, the PM thermal control design is unique with respect to size, payload interfaces, radiator design and deployment, maintenance, refurbishment and growth requirements, and operational flexibility.

- Questions relating to the methods and driving parameters for selecting and optimizing radiator designs have been raised during the 25 kW PM Evolution Study. It is recommended that the analysis effort initiated at LMSC and documented in EM C-1.2.2-104* be continued and expanded to develop the analytical procedures for supporting the selection, design, fabrication, and test of PM radiators.

- The remaining hardware technology which should be initiated includes the development of multiple payload interface design and control in support of the first flight vehicle design. Performance increases in the payload interface will be required for the 50 kW configuration which offers 25 kW of payload cooling capacity.

- Development of new radiator design, manufacturing techniques, and installation methods are required to support the projected weight reductions envisioned for the 1988 time period.

*See section 8.2

THERMAL CONTROL SYSTEM TECHNOLOGY DEVELOPMENT

DEVELOPMENT ITEM	PLANNED UTILIZATION	SCHEDULE 80 82 84 86
ANALYTICAL TECHNOLOGY TO IDENTIFY OPTIMIZATION CRITERIA FOR HEAT PIPE VS FLUID RADIATOR DESIGN	SUPPORT SELECTION OF 25 kW RADIATOR DETAIL DESIGN, AND IDENTIFY POTENTIAL IMPROVEMENTS	
MULTIPLE PAYLOAD INTERFACE DESIGN AND CONTROL	APPLY DESIGN CONCEPTS TO THE FIRST 25 kW FLIGHT VEHICLE	
RADIATOR PANEL DESIGN IMPROVEMENTS TO REDUCE WEIGHT & EXTEND LIFE	UTILIZE DESIGN, MANUFACTURING, AND ASSEMBLY IMPROVEMENTS TO REDUCE THE WEIGHT OF THE 50 AND 100 kW RADIATOR SYSTEMS	

ATTITUDE CONTROL TECHNOLOGY DEVELOPMENT

- Since there are only enough existing ATM Rate Gyros to equip the first Power Module, an alternate system, such as the NASA Standard Inertial Reference Unit, will be required on subsequent Power Modules.

- After the first nine CMG's are used, additional CMGs will be manufactured to meet program requirements. At this time both the CMGs and associated electronics can be redesigned to improve reliability and reduce size, weight, and power by using the current state-of-the-art components. A 10-year life is the ultimate goal.

- The schedule shown for both of the above components assumes a 2-year span from initiation of procurement to delivery by the manufacturer. The delivery schedule is consistent with the requirements of Scenario I.

- With future large flexible system payloads, distributed active elements and/or sensors may be required. Present control system approaches are adequate for Power Modules at least to the 100 kW size.

- In addition to Teleoperator or Orbiter reboost for drag makeup, on-board propulsion subsystem options warrant consideration. If selected, development of long-life high-reliability components, designed for modular EVA replacement, is required. In particular, electro-thermal monopropellant hydrazine thrusters will be considered, eliminating the life limitations of present catalytic beds. Also, development of superheated MMH thrusters to enable $I_{sp} \cong 300$ seconds is potentially needed.

ATTITUDE CONTROL TECHNOLOGY DEVELOPMENT

DEVELOPMENT ITEM	PLANNED UTILIZATION	80 82 84 86 88 90
NASA STANDARD INERTIAL REFERENCE UNIT (PRESENTLY UNDER DEVELOPMENT)	• AFTER THE FIRST POWER MODULE	INITIAL PROCUREMENT △ 1 1 2 1 1 1
CMG ELECTRONICS & CMGs	• AFTER EXISTING NINE CMGs ARE USED • ESPECIALLY FOR CONFIGURATIONS REQUIRING 4 TO 6 CMG'S	INITIATE △ PROCUREMENT DELIVERY 3 3 3 6 6

COMMAND & DATA HANDLING TECHNOLOGY DEVELOPMENT

- This chart identifies anticipated development items for future PM growth and utilization.
- Long-life items include Hi-Rel microprocessors that also will incorporate radiation hardening.
- Data Compression, or limit checking devices which transmit only out-of-limit data (thereby substantially increasing effective data rates) are under development. These are expected to be available by 1986 and concurrent development of the Power Module application would be feasible.

COMMAND & DATA HANDLING TECHNOLOGY DEVELOPMENT

DEVELOPMENT ITEM	PLANNED UTILIZATION	SCHEDULE				
		80	82	84	86	88
HIGH-EFFICIENCY, HIGH-RELIABILITY/LONG LIFE (10+ YEARS) COMPONENTS FOR KU BAND DATA RATES OF APPROXIMATELY 300 MBS	SCIENTIFIC DATA TRANSMISSION VIA TDRS		DEV/TEST		PRODUCTION	
DATA PROCESSING & COMPRESSION EQUIPMENTS COMPATIBLE WITH ABOVE DATA RELAY CAPABILITY	POWER MODULES DEPLOYED AFTER 1986		SAME AS ABOVE			

SECTION 8
REFERENCES

8.1 BIBLIOGRAPHY FOR PART III

The following lists the primary published document references utilized in conjunction with Part III of the study.

Ref No.	Document No.	Title	Author/Source/Contact	Date
1	STAR 15	Shuttle Turnaround Analysis Report	KSC	9 Dec 1977
2	JSC 07700 Vol XIV	Space Shuttle System--Payload Accommodations (Revision F, Change 27)	NASA (JSC)	11 Nov 1978
3	NASA/Langley Memo 78668	An Introduction to Shuttle/LDEF Retrieval Operations: The R-Bar Approach Option	NASA	1 Feb 1978
4		25 kW Power Module Preliminary Definition	MSFC	Sep 1977
5		Power Module Data Management System (OMS) Study (IBM-FSD Huntsville)	IBM	9 June 1978
6		Teleoperator Retrieval System	Hethcoat (MSFC)	16 Mar 1978
7		Orientation Briefing for Power Module Evolution Study, Skylab	Rutland (MSFC)	16 Mar 1978
8		Space Shuttle External Tank Briefing	MSFC	16 Mar 1978
9	MSFC-SPEC-582A	Power Module System Design Requirements Document	NASA/MSFC	1 May 1978
10		System Capabilities	Beasley (MSFC)	17 Mar 1978

Ref No.	Document No.	Title	Author/Source/Contact	Date
11		25 kW Attitude Control System Trade Studies	MSFC	Feb 1978
12		Power Module CMG Status	MSFC	Mar 1978
13		Solar Electric Propulsion	MSFC (Austin)	May 1978
14		25 kW Power Module Mass Properties (Concept IV)	MSFC (Collins)	1 Feb 1978
15	K-STSM-09 Vol VI	Launch Site Accommodations Handbook for STS Payloads	NASA (KSC)	14 Mar 1978
16	SAI No. SAI-79-602-HU	Space Industrialization		15 Apr 1978
17		Study of the Use of Spacelab Derived Elements	ERNO	Jan 1978
18	ICD 2-19001 CH 1	Shuttle Orbiter/Cargo Standard Interfaces	NASA (JSC)	24 Apr 1978
19	TMX-64972	A Miniaturized Pointing Mount for Spacelab Missions	C. G. Fritz, J. T. Howell, P. D. Nicaise, J. R. Parker	11 Nov 1975
20	LMSC-D569577	Contamination Control of Long-Life Shuttle Payload	M. C. Fong, C. K. Liu	10 Dec 1976
21	Combustion and Flames, Vol 14, No. 13, pp 397-404	Simultaneous Mass Spectrometric Differential Thermal Analyses of Nitrate Salts of MMH and Methylamine	P. Breisacher, H. H. Takimoto, G. C. Denault, W. A. Hicks	Jun 1970

Ref No.	Document No.	Title	Author/Source/Contact	Date
22	MCR-75-13, Martin Marietta Aerospace, Denver	Payload/Orbiter Contamination Control Assessment Support	R.O. Rantanen, E.B. Ross	27 Jun 1975
23	NASA TM X-68212	Exhaust Plume and Contamination Characteristics of a Bipropellant (MMH/N_2O_4) RCS Thruster	E.W. Spisz, R. L. Bowman, J.R. Jack	1973
24	Special Report 332	Special Report	L. J. Jacchia, Smithsonian Observatory	5 May 1971
25	ES 84	Solar Activity Indices and Predictions	NASA/MSFC	5 Jan 1978
26	Viewgraph Briefing	1978 Predicted Solar Flux	W.D. McFadden/ MSFC	23 Oct 1978
27	LMSC-A374573	A High Speed Computer Program for Predicting the Decay of Earth Satellites	H.W. Small, R.C. Johnson/LMSC	13 May 1963
28	IBM K42-78-001	Power Module Data Management System (DMS) Study--Final Report	IBM, Federal Systems Div., Huntsville	30 Nov 1978
29	STAR 15	Shuttle Turnaround Analysis Report	Rockwell International	5 June 1978
30	SLP 2104, Issue No. 1	Spacelab Payload Accommodation Handbook	NASA/MSFC	30 June 1977
31	LMSC-D614928A	25 kW Power Module Evolution Study: Part II Payload Support System Evolution	LMSC/J.W. Overall	30 Sep 1978

Ref No.	Document No.	Title	Author/Source/Contact	Date
32	LMSC-665411	25 kW Power Module Solar Array Preliminary Design	LMSC/J. F. Milton	29 Nov 1978
33	NASA Fact Sheet No. 78-49	Teleoperator Retrieval System	NASA	31 Mar 1978

8.2 LIST OF ENGINEERING MEMORANDA

EM No.	Title	Author	Date
C-1.1.0-101	Docking Module Systems Comparative Evaluation	S. R. Nichols	22 May 1978
C-1.1.2-100	25 kW Power Module--LMSC Recommended Candidate Definition	E. Waller	23 Oct 1978
C-1.1.2-103A	Power Module Family--Design Integration Drawings	R. W. Goldin	10 Nov 1978
C-1.2.0-100	25 kW Power Module Study--Baseline Data in Support of Cost Evaluation	B. G. Wong	27 Mar 1978
C-1.2.0-101	25 kW Power Module Study--Comparison Strawman 1 (NASA Concept, Sep 1977) and Strawman 2 (NASA Concept, Jan 1978)	B. G. Wong	28 Mar 1978
C-1.2.0-102	Preliminary Weight Comparison Summary	B. G. Wong	5 Apr 1978
C-1.2.0-103	(1) Interface-Space Shuttle/Strawman 1 (2) RMS Deployment-Strawman 1/Shuttle	B. G. Wong	7 Apr 1978
C-1.2.0-104	Update of the Equipment Arrangement for the 25 kW Power Module	B. G. Wong	14 Apr 1978
C-1.2.0-105	Updating of the 25 kW Power Module Configuration	B. G. Wong	14 Apr 1978
C-1.2.0-106	25 kW PM Docking System Definition	S. Nichols, B. G. Wong	20 Apr 1978
C-1.2.0-107	25 kW Power Module Update	B. G. Wong	15 May 1978
C-1.2.1-101	Structural and Mechanical Design Activity - 25 kW Power Module	B. G. Wong	14 Apr 1978

EM No.	Title	Author	Date
C-1.2.1-102	Structural Subassembly Trade Studies	S. Nichols	15 June 1978
C-1.2.1-103	Solar Array Support Structure Trade Study	B. G. Wong	11 May 1978
C-1.2.1-104	Solar Array Support Structure Trade Study	B. G. Wong, S. Nichols	26 May 1978
C-1.2.2-101	Comparison of Flat vs Curved Radiator Panel Configurations	W. Hutchins	19 May 1978
C-1.2.2-102	Thermal Model Heat Rate Files	A. L. Lee	10 Jul 1978
C-1.2.2-103	25 kW Power Module Heat Rejection Capability	A. L. Lee R. A. Horn	15 Dec 1978
C-1.2.2-104	Comparisons of Fluid-Flow and Heat Pipe Radiators for 25 kW Power Module Application	L. Fried	12 Dec 1978
C-1.2.2-105	Comparison of Flat vs Curved Power Module Radiator Systems	R. A. Horn	10 Dec 1978
C-1.2.2-106	25 kW PM Cooling System: Meteoroid Effects	L. Fried	12 Dec 1978
C-1.2.3-101	25 kW Power Module Retrieval	R. Barsocchi	17 Apr 1978
C-1.2.3-102	Attitude Control System Component Input/Output Lists	R. Barsocchi, J. Kolvek	19 May 1978
C-1.2.5-101	Energy Storage Subsystem Trade and Growth Analysis	M G. Gandel	None
C-2.0-101	Payload Power Requirements for Multi-Discipline Platform--Power Module	W. Miller	20 Jul 1978

APPENDICES

A DESIGN LAYOUTS

B POWER SUBSYSTEM TRADES

C MAINTENANCE TIME ESTIMATES

APPENDIX A DESIGN LAYOUTS

LIST OF POWER MODULE DESIGN
INTEGRATION DRAWINGS

DWG. NO.	TITLE	DATE (1978)	DRAWN BY	REPORT LOCATION VOL	PAGE
6164050B	POWER MODULE FAMILY – BASIC DIMENSIONS	11/15	W. HUTCHINS	1	3-57
051A	POWER MODULE – SOLAR ARRAY SUPPORT STRUCTURES	11/6	W. STEELE	1	4-23
052	POWER MODULE – EQUIPMENT RACK STRUCTURES	10/26	W. STEELE	1	4-27
053A	BERTHING STRUCTURE – POWER MODULE	11/9	W. HUTCHINS	1	4-31
054	POWER MODULE – INBOARD PROFILE, EQUIPMENT INSTALLATION SHEET 1A	11/6	S. NICHOLS	1	3-77
	SHEET 1A	11/6	W. HUTCHINS	1	3-79
055	POWER MODULE – 25 kW INBOARD PROFILE, EQUIPMENT INSTALLATION	11/9	S. NICHOLS	1	4-15
056	POWER MODULE – 50 kW INBOARD PROFILE, EQUIPMENT INSTALLATION	11/9	S. NICHOLS	1	5-15
057	POWER MODULE – 100 kW INBOARD PROFILE, EQUIPMENT INSTALLATION	11/9	S. NICHOLS	1	6-21

APPENDIX A (CONT.)

DWG. NO.	TITLE	DATE (1978)	DRAWN BY	REPORT LOCATION VOL	PAGE
6164060A	POWER MODULE/ORBITER BERTHING SYSTEM	11/6	S. NICHOLS	1	4-33
061A	25 kW POWER MODULE: CONFIGURATION 25-1 ORBITER PAYLOAD BAY INSTALLATION	11/7	W. HUTCHINS	1	4-7
062B	50 kW POWER MODULE: CONFIGURATION 50-1 ORBITER PAYLOAD BAY INSTALLATION	11/14	W. HUTCHINS	1	5-7
063	100 kW POWER MODULE: CONFIGURATION 100-1 ORBITER PAYLOAD BAY INSTALLATION SHEET 1B	11/15	W. STEELE	1	6-11
	SHEET 1A	11/2	W. STEELE	1	6-13
064A	POWER MODULE ORBITAL CONVERSION KIT – CONFIGURATION 25-1 TO CONFIGURATION 50-2	11/6	W. STEELE	4	2-97
065	50 kW POWER MODULE: CONFIGURATION 50-2 ORBITER PAYLOAD BAY INSTALLATION	11/6	W. STEELE	4	2-99
070A	POWER MODULE EVOLUTION SCENARIO I – TYPICAL SATELLITE CONFIGURATIONS	11/9	W. HUTCHINS, C. COFFIELD	1 4	3-63 2-7

Appendix B
POWER SUBSYSTEM TRADES

This appendix contains electrical power subsystem (EPS) growth options to 250 kW for the major components of the subsystem. Also treated are the packaging (containment) concepts and the deployment options, as were presented on 29 June 1978 to MSFC. Subsequent to that meeting solar array growth and containment concepts have changed. However, the energy storage and mast characteristics analyses are valid.

The data contained herein are provided to complement EPS trade study material in the main body of the report, and to reflect in this final report the total scope of subsystem trades actually accomplished.

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EPS — INTRODUCTION

TOPICS OF DISCUSSION

- ELECTRICAL POWER SYSTEM GROWTH SCENARIOS
- ELECTRICAL POWER SYSTEM CONFIGURATIONS
- ELECTRICAL POWER SYSTEM GROWTH CONFIGURATIONS
- WEIGHT PROJECTIONS FOR LEO SYSTEMS
- ENERGY STORAGE GROWTH TRADE STUDIES
- DEPLOYMENT MAST CAPABILITIES
- SOLAR ARRAY GROWTH TRADES
- SOLAR ARRAY DRIVE STUDIES
- SOLAR ARRAY DEGREE OF FREEDOM STUDY
- SOLAR ARRAY CONFIGURATION STUDY
- ELECTRONIC DOWN CONVERSION TRADE STUDY
- SUMMARY

ELECTRIC POWER SYSTEM
ELEMENT GROWTH SCENARIO

As power level is programmed to grow to 200-250 kW by 1989 to 1991, the increased demands can be met by increasing system size and utilizing advances in technology.

Three means are projected for improving power density and packaging efficiency for solar arrays; they are:

- Improvement of cell efficiency
- Replacement of silicon by higher efficiency, up to 20%, gallium arsenide
- Decrease panel density from 0.2 to 0.1 pounds/sq ft

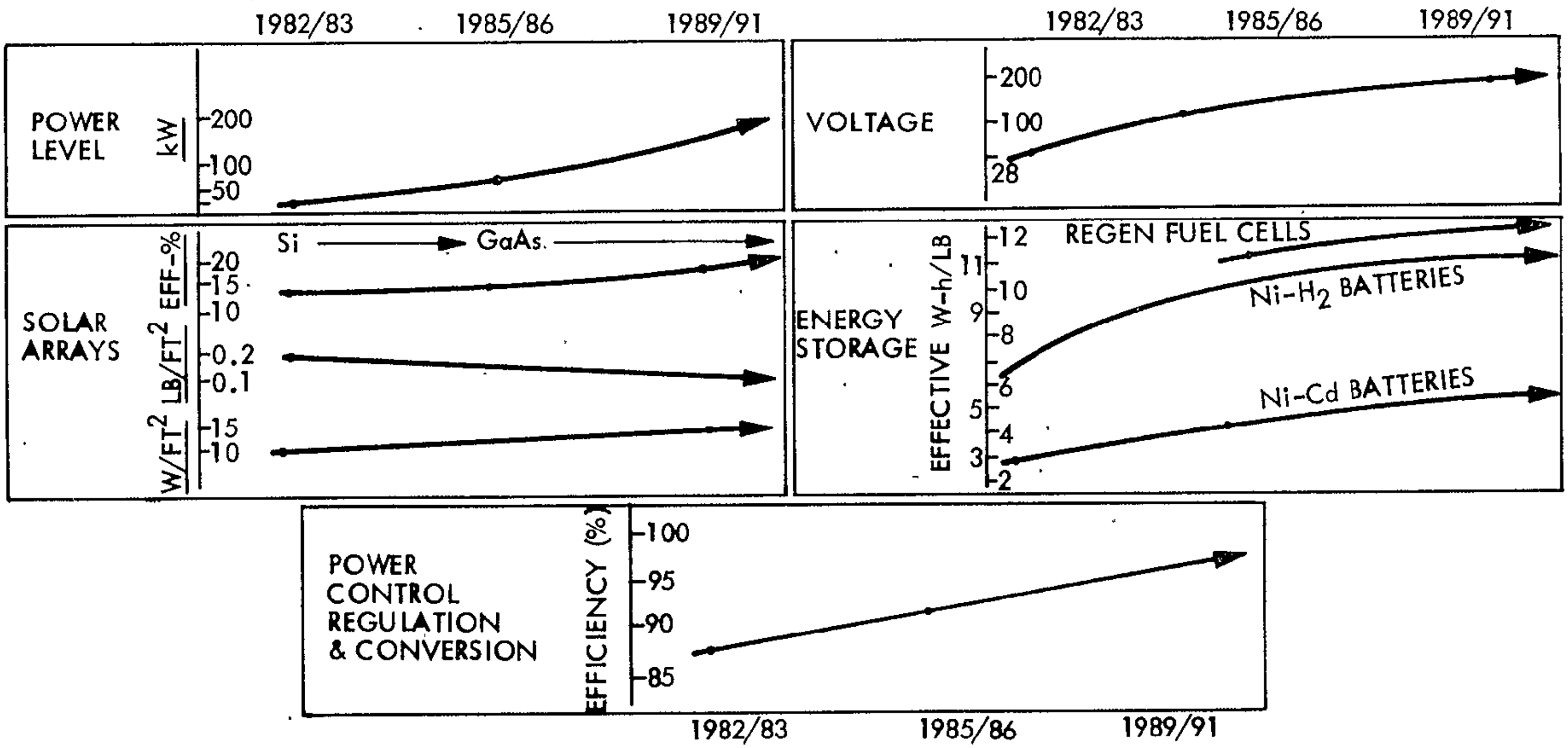
Energy storage effective density is seen to gain significantly in going to $Ni\text{-}H_2$ batteries or regenerative fuel cells from Ni-Cd batteries. The improvement with time is due to both increasing packaging density and DoD.

Regenerative fuel cells are shown with a small weight advantage over $Ni\text{-}H_2$ batteries, however, a slight increase in battery DoD would cancel this difference.

The power control and conditioning equipment efficiency is shown to increase with time. This is attributed to operation at higher voltage levels, advancement in component technology and improved circuit design.

It is projected that for the 1986 and beyond time frame extensive use of graphite composites will be used for structural members, resulting in substantial weight reductions.

ELECTRICAL POWER SYSTEM ELEMENT GROWTH SCENARIO

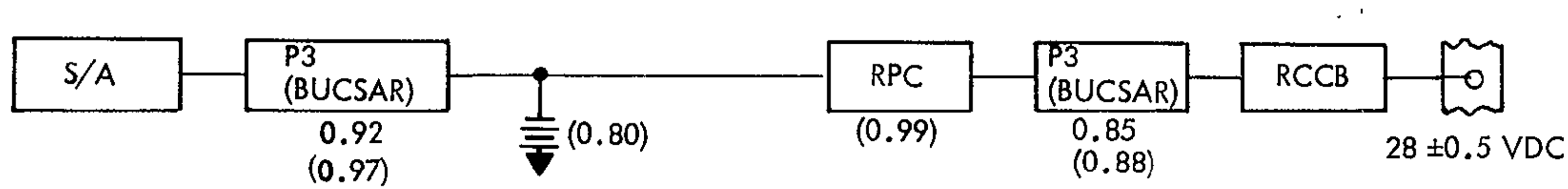


ELECTRICAL POWER SYSTEM CONFIGURATIONS

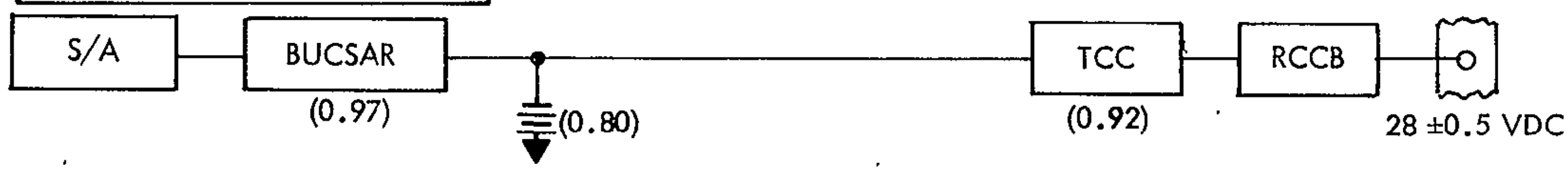
Four concepts are considered for the EPS configuration trade. These represent the combination of the transformer coupled converter (TCC) vs the buck regulator (P^3/BUCSAR) and cascaded power stages (charger and output regulator) vs direct transfer (regulation) of solar array power to the bus. The efficiency values for this trade are based on actual test results in the case of the BUCSAR and a detailed analytical model for the TCC.

ELECTRICAL POWER SYSTEM CONFIGURATIONS

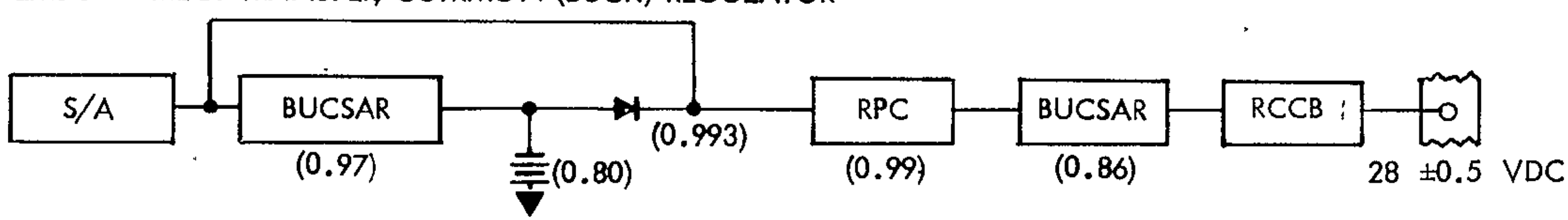
1 MSFC - LABS BASELINE (LMSC COMMON REGULATOR)



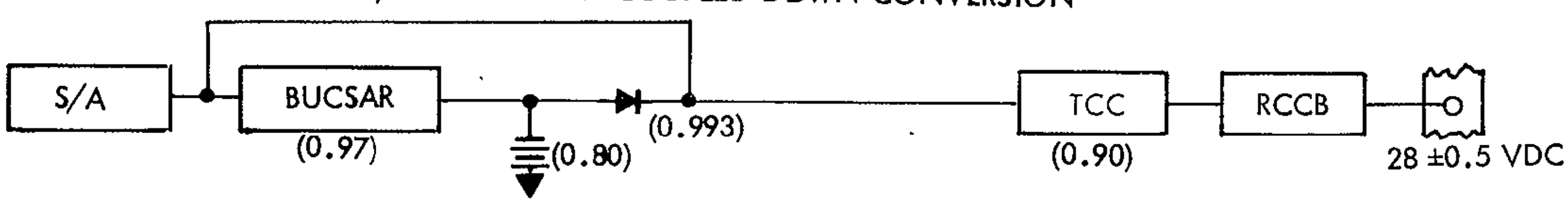
2 LMSC - SELECTED BASELINE BUCK CHARGER/REG WITH TRANSFORMER COUPLED DOWN CONVERSION



3 LMSC - DIRECT TRANSFER/COMMON (BUCK) REGULATOR



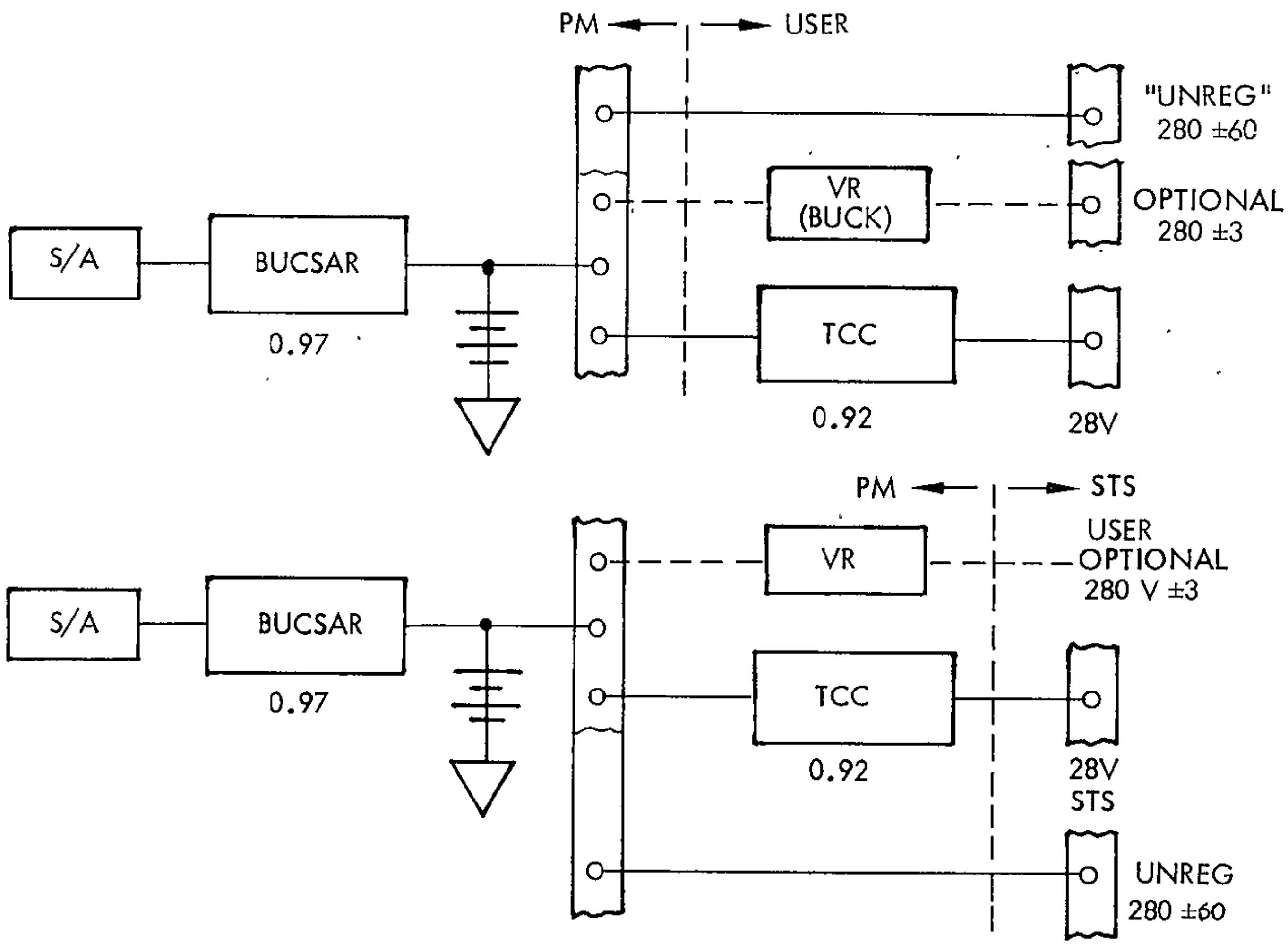
4 LMSC - DIRECT TRANSFER/TRANSFORMER COUPLED DOWN CONVERSION



ELECTRICAL POWER SYSTEM GROWTH CONFIGURATION

The 140 vdc approach, identified by several agencies as the best approach for 25 to 35 kW power systems, is scaleable at reasonable efficiencies to ten times that level or more (300 kW). It is not apparent that higher control efficiency can be obtained at higher voltage for a large space power system of the multihundred kilowatt scale. The efficiency of thyristor based power electronics will not match that of the 140 vdc system below several kilovolts of bus voltage level although distribution weight improvements may be sufficient to warrant still higher voltages. It is projected that the efficiency of the regulator concepts will improve by doubling the 140 vdc level between now and 1990 as a result of component improvements and low IR losses. This may be the practical limit for transistor systems.

ELECTRICAL POWER SYSTEM GROWTH CONFIGURATION



WEIGHT PROJECTIONS FOR LEO SYSTEMS

Advances in technology will allow for significant power system growth within present Shuttle weight and volume constraints. In 1983, 50 kW capability can be provided using present baseline equipment, with all power provided to 28 volt regulated buses.

By 1986, lighter-weight and efficient solar arrays are projected with nickel-hydrogen batteries operating to 53% DoD. The Ni-H_2 technology is advancing rapidly, therefore, early initiation of a development and life test program should yield high confidence in this battery before commitment to flight. Supplying power at 120V provides significant economy in all aspects of power management. The dc/dc converters are sized to maximum current, therefore, higher voltage allows a higher power rating per unit as well as higher efficiency. Power distribution and cabling also benefit from higher voltage; weights at 125 kW are not greater than for the 50 kW system, which are based on ATM estimates.

Projections for 1990 call for going to higher efficiency GaAs solar cells built into a light-weight, 0.1 lb/ft^2, solar array. Present test programs for Ni-H_2 battery cells show 80% DoD capability at LEO. By 1990 it is expected that lighter-weight Ni-H_2 cells will have demonstrated high reliability at 80% DoD. Increasing voltage to 240V will permit weight savings in electronics, power distribution, and cabling. Gains in regulator and converter efficiency are reflected in lighter electronics weight and in reduced solar array area.

WEIGHT PROJECTIONS FOR LEO SYSTEMS

LAUNCH DATE	1983		1986		1990	
POWER	50 kW		125 kW		250 kW	
SOLAR ARRAY	Si-0.2 LB/FT^2	4,850 LB	Si-0.15 LB/FT^2	7,000 LB	GaAs-0.1 LB/FT^2	10,000 LB
BATTERIES	NiCd BASELINE	14,880 LB	NiH_2 53% DOD	10,000 LB	NiH_2 80% DOD	13,300 LB
ELECTRONICS	28 V	2,640 LB	120 V	3,000 LB	240 V	2,200 LB
POWER DIST	BASED ON ATM	470 LB		500 LB		1,500 LB
CABLING	BASED ON ATM	830 LB		1,000 LB		1,000 LB
TOTAL		23,690 LB		21,500 LB		27,000 LB

SYSTEM EFFECTIVE GROWTH ALTERNATIVES FOR ENERGY STORAGE

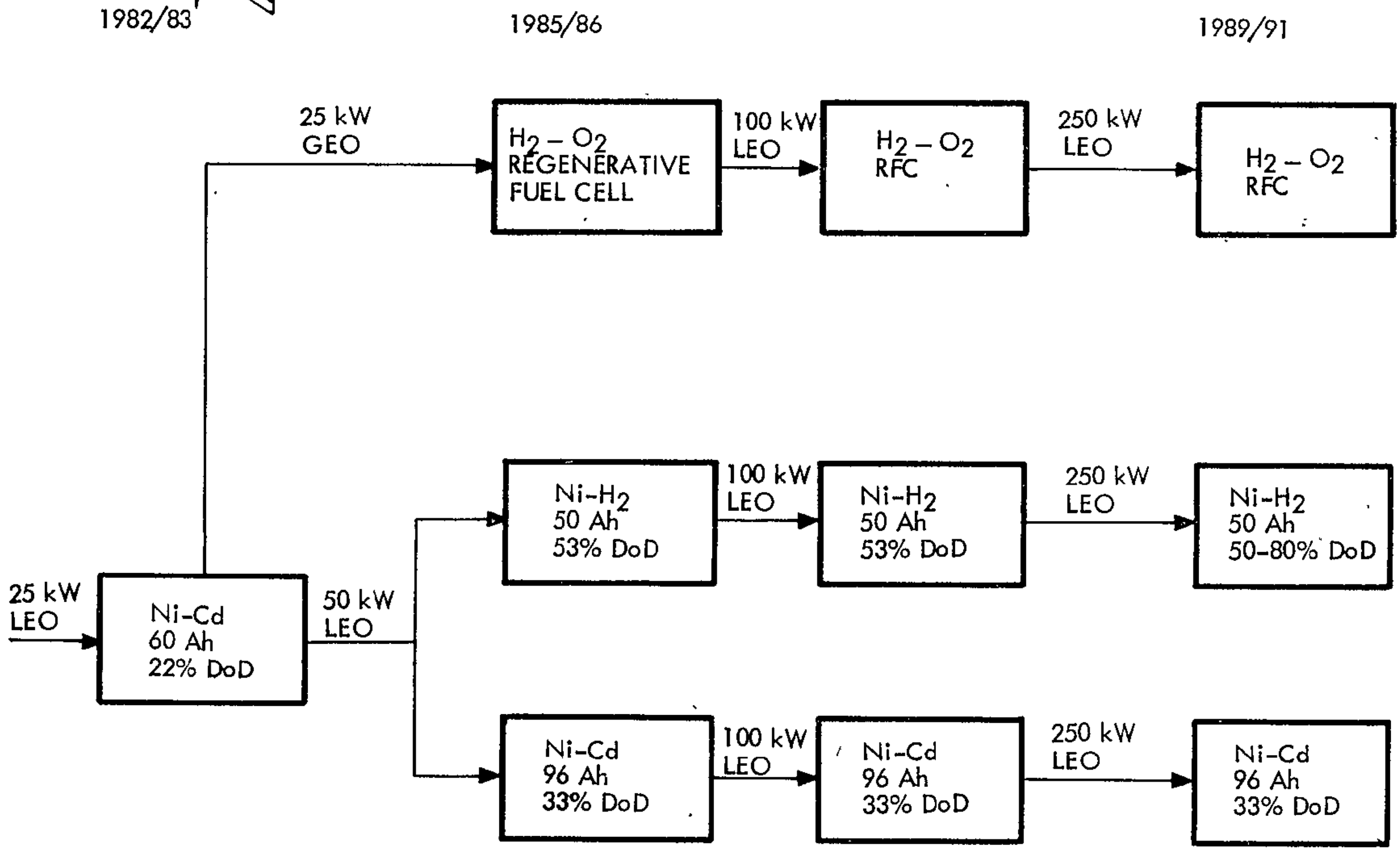
The present baseline Ni-Cd battery system using 12 - 110 cell, 60 AH batteries operated to 22% DoD, is cost effective and reliable for the first PM, regardless of subsequent energy storage system selection.

Early requirements for geosynchronous missions would prompt the development of a regenerative fuel cell system, because of its light weight and the delivery cost to high orbit. Once the non-recurring costs have been assimilated, the recurring costs for regenerative fuel cells are approximately the same as for nickel-hydrogen batteries operated to 53% DoD.

This diagram indicates that if the needs are restricted to LEO, the choice remains between 53% DoD Ni-H_2 and 33% DoD, 96 AH (nominal 100 AH) Ni-Cd batteries. Ni-H_2 is favored because as this technology matures, even higher DoD capability is expected.

The material used in the trade analysis of the energy storage system is treated in detail in LMSC EM No. C-1.2.5-101.

SYSTEM EFFECTIVE GROWTH ALTERNATIVES FOR ENERGY STORAGE



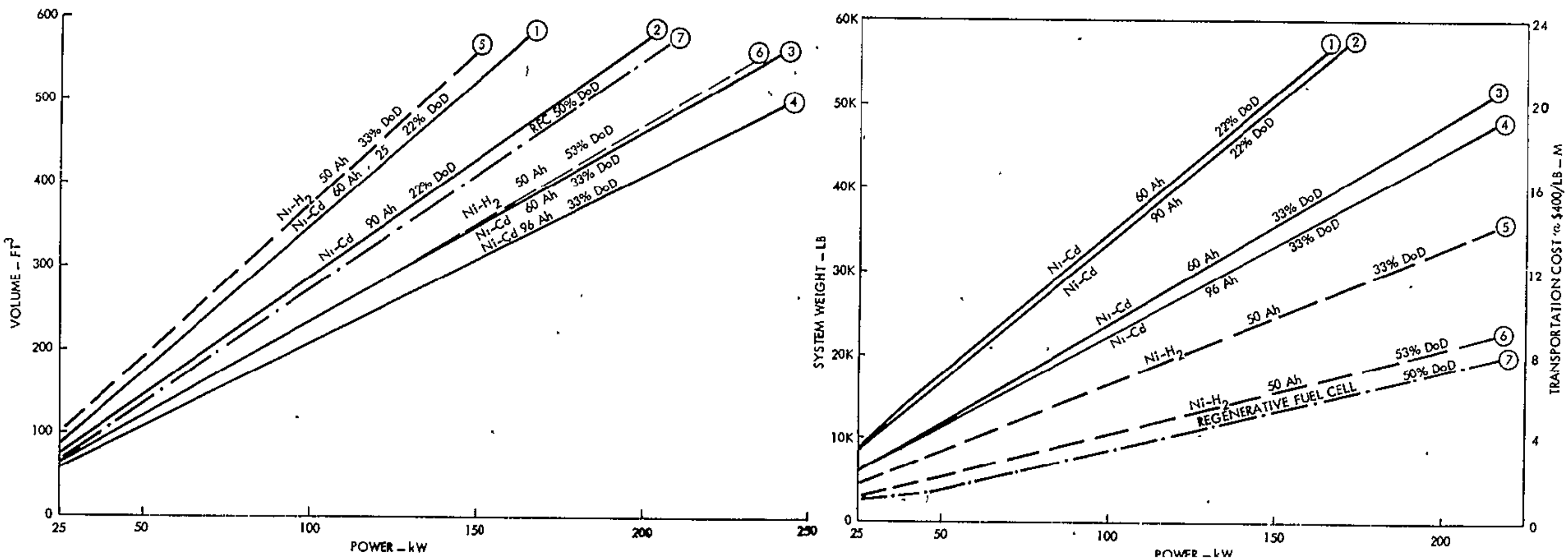
EPS VOLUME VS POWER LEVEL

The graph indicates for like capacity, Ni-H_2 occupies more volume than Ni-Cd batteries and some volume is saved by going to larger cells. But the biggest gain develops from going to greater DoD. Since nickel-hydrogen batteries indicate higher DoD capacity than Ni-Cd, the Ni-Cd volumetric advantage is marginal. The regenerative fuel cell system volume could be made smaller by increasing reactant storage tank pressure from 400 psi, but that would increase electrolyzer operating pressure and weight. Volume requirements for energy storage remain a small percentage of orbiter cargo volume, at 100 kW all systems fall between 2 and 4 percent of orbiter cargo volume.

EPS WEIGHT VS POWER

Each alternative system is assumed linear in growth with power level. Major weight savings may be affected by either increasing DoD or changing electrochemical couples. Smaller weight savings may be gained by developing battery cells of larger capacity. Nickel-hydrogen batteries at 53 percent DoD, which is believed conservative for the long term, and regenerative fuel cell systems offer significant weight savings. When transport costs to LEO are considered at $400/lb, weight becomes a significant cost element.

EPS VOLUME AND WEIGHT VS POWER LEVEL



ENERGY STORAGE SUBSYSTEM COSTS

COST FOR GROWTH SUBSYSTEMS

BASIS: The non-recurring costs are added to the recurring costs for one 25 kW system. The 50 kW point is determined by adding the recurring cost of one 50 kW system to the first 25 kW PM costs. The 100 kW points add the recurring cost of one 100 kW system to the foregoing summation, and so on for the 200 kW point.

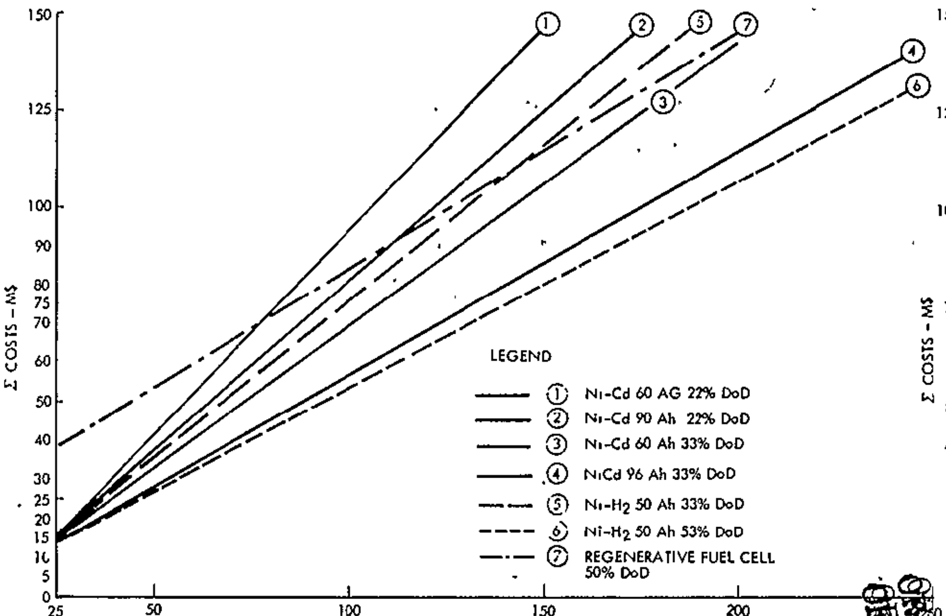
ANALYSIS: The recurring cost slopes for 53% DoD Ni-H_2 and the RFC are approximately equal, and the 96 AH 33% DoD Ni-Cd slope is only slightly higher. This would indicate a first choice of Ni-H_2 followed by Ni-Cd, unless the high RFC non-recurring costs can be amortized over more units.

COSTS FOR BASELINE FOLLOWED BY GROWTH SUBSYSTEM

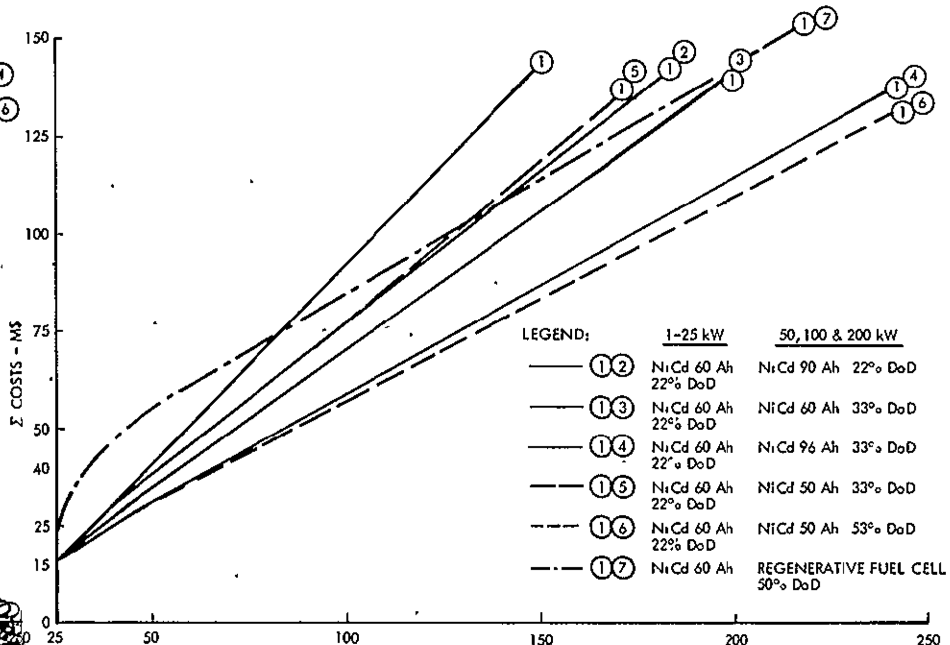
BASIS: All alternative curves begin by using the same baseline Ni-Cd AH 22% DoD energy storage system for one 25 kW PM plus non-recurring and recurring costs for one of each alternative system at 50, 100 and 200 kW.

ANALYSIS: This set of curves does not differ significantly from the preceding case. There is a small penalty in accepting the baseline energy storage system for usage on the first 25 kW PM, and then developing a more cost-effective system for subsequent PMs.

ENERGY STORAGE SUBSYSTEM COSTS



COSTS FOR 25, 50, 100 & 200 kW POWER MODULES



COSTS FOR BASELINE 25 kW FOLLOWED BY ALT 50, 100 & 200 kW SYSTEMS

ENERGY STORAGE TRADE TREE FOR
25 kW POWER MODULE – GEO

Two factors make the regenerative fuel cell system especially attractive. First, it is approximately one-half the weight of the Ni-H_2 system and one-third the weight of the Ni-Cd system. Secondly, the high cost of transportation to GEO gives the RFC the lowest recurring cost. The higher non-recurring, development cost of the RFC would be recovered in two or three flights.

For the GEO, because of the low cycle life required, allowable DoD for the batteries was increased to 60 and 80%, respectively, for Ni-Cd and Ni-H_2 batteries, based on a maximum eclipse of 1.2 hours. The long recharge time reduces electrolyzer requirements, therefore, only two 28-volt modules are required.

If the RFC is developed for GEO, its recurring costs are competitive with the Ni-H_2 battery for LEO applications.

ENERGY STORAGE TRADE TREE FOR 25 kW POWER MODULE - GEO

	NiCD BATTERIES	NiH_2 BATTERIES	REGENERATIVE FUEL CELLS	
			FUEL CELLS	ELECTROLYZERS
	60% DOD	80% DOD	15 kW	3 kW
	90 Ah	50 Ah		
	6-110 CELL BATTERIES	8-105 CELL BATTERIES	2-17 CELL MODULES	3-128 CELL MODULES
SYSTEM				
WEIGHT – LB	6570	4529	2379	
VOLUME – FT^3	66	99	112 (2)	
COST – N.R. – \$M	3.26	3.76	27.06	
RECURRING	5.83	8.97	9.95	
TRANSPORT (1)	32.85	22.65	11.90	
TOTAL	41.94	35.38	48.91	
TOTAL W/O N.R.	38.68	31.62	21.85	

NOTES: (1) \$5,000/LB

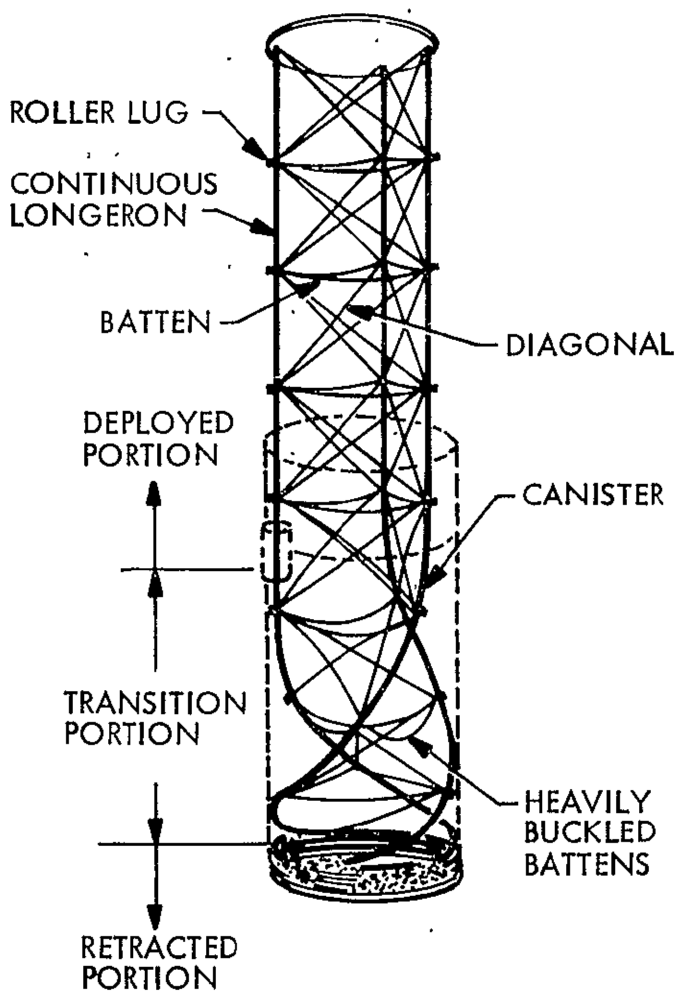
(2) 400 PSI GAS STORAGE FOR 50% DOD

SOLAR ARRAY DEPLOYMENT MAST EVALUATIONS

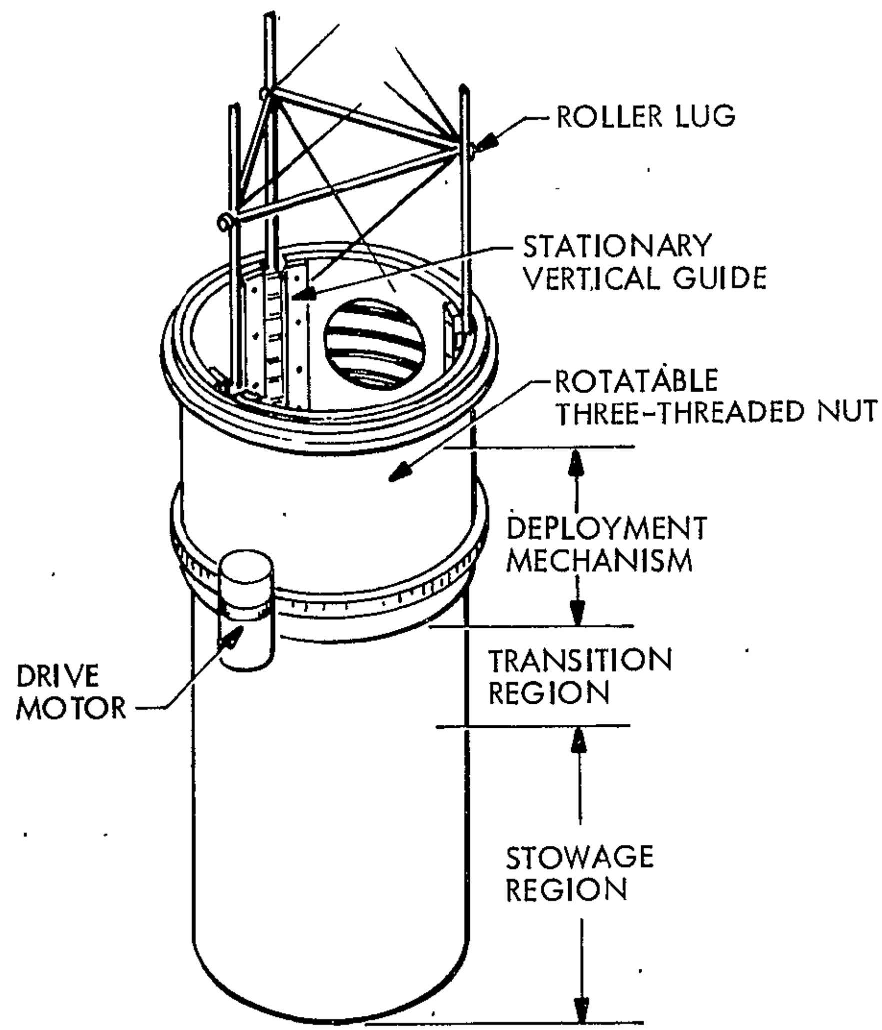
- In order to determine the characteristics of the solar array system with respect to its dynamic response, LMSC has investigated the deployment mast design parameters. This effort has been completed in conjunction with Mr. R. Crawford of AEC-Able Engineering. The following charts present some of this parameter evaluation. LMSC has used this data to investigate the feasibility of a common building block concept for growth to higher power levels. The prime driver in this investigation is how can these large deployment masts be stowed and what solar array capabilities can be achieved given the volume limitation that we have within the Orbiter cargo bay. As a result of this study, it appears feasible to use a common MAST envelope for growth from 25 kW to 250 kW using a common physical blocking solar array system.

SOLAR ARRAY DEPLOYMENT MAST EVALUATIONS

DEPLOYMENT GEOMETRY AND NOMENCLATURE FOR CONTINUOUS-LONGERON LATTICE BOOMS

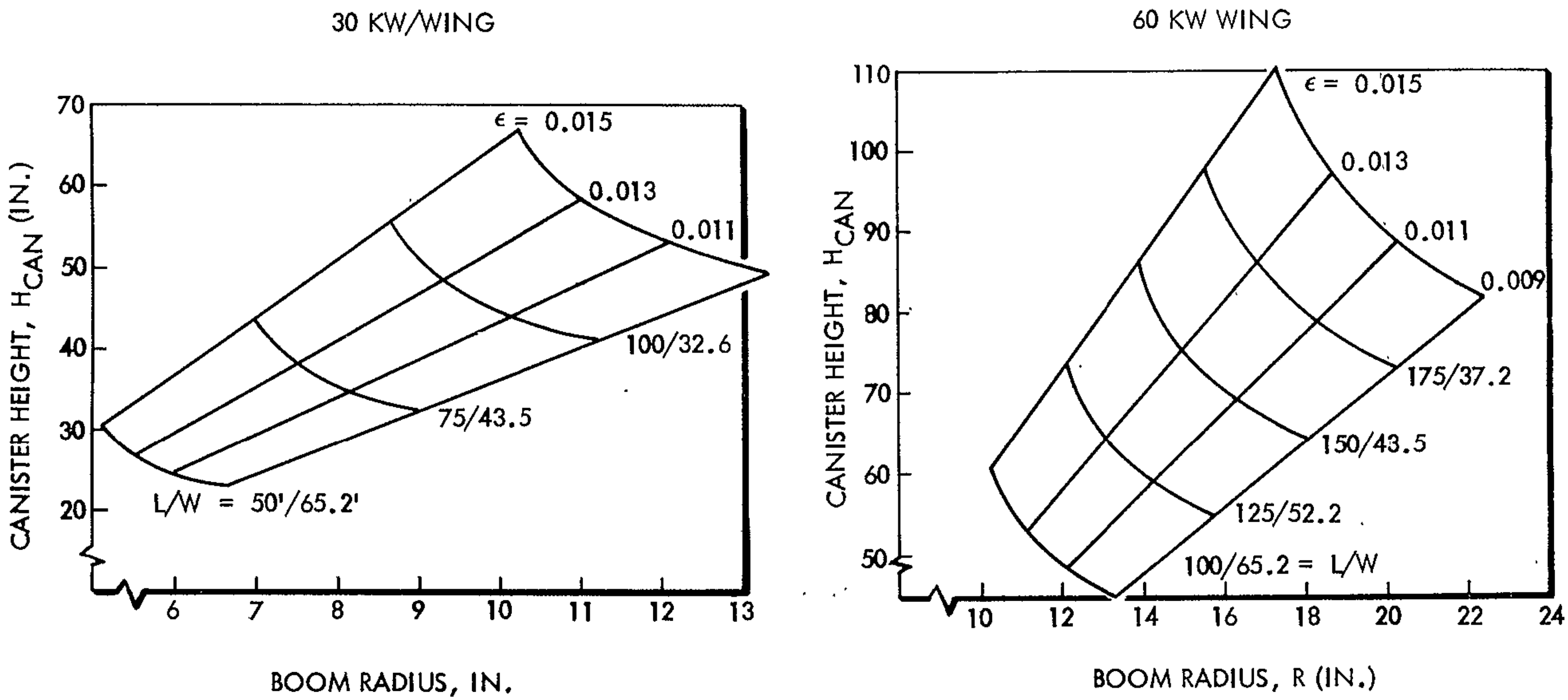


CANISTER FOR DEPLOYING AND SUPPORTING CONTINUOUS-LONGERON LATTICE BOOMS



PM (25kW) DEPLOYMENT MAST CHARACTERISTICS

PM (25 - 50 kW)

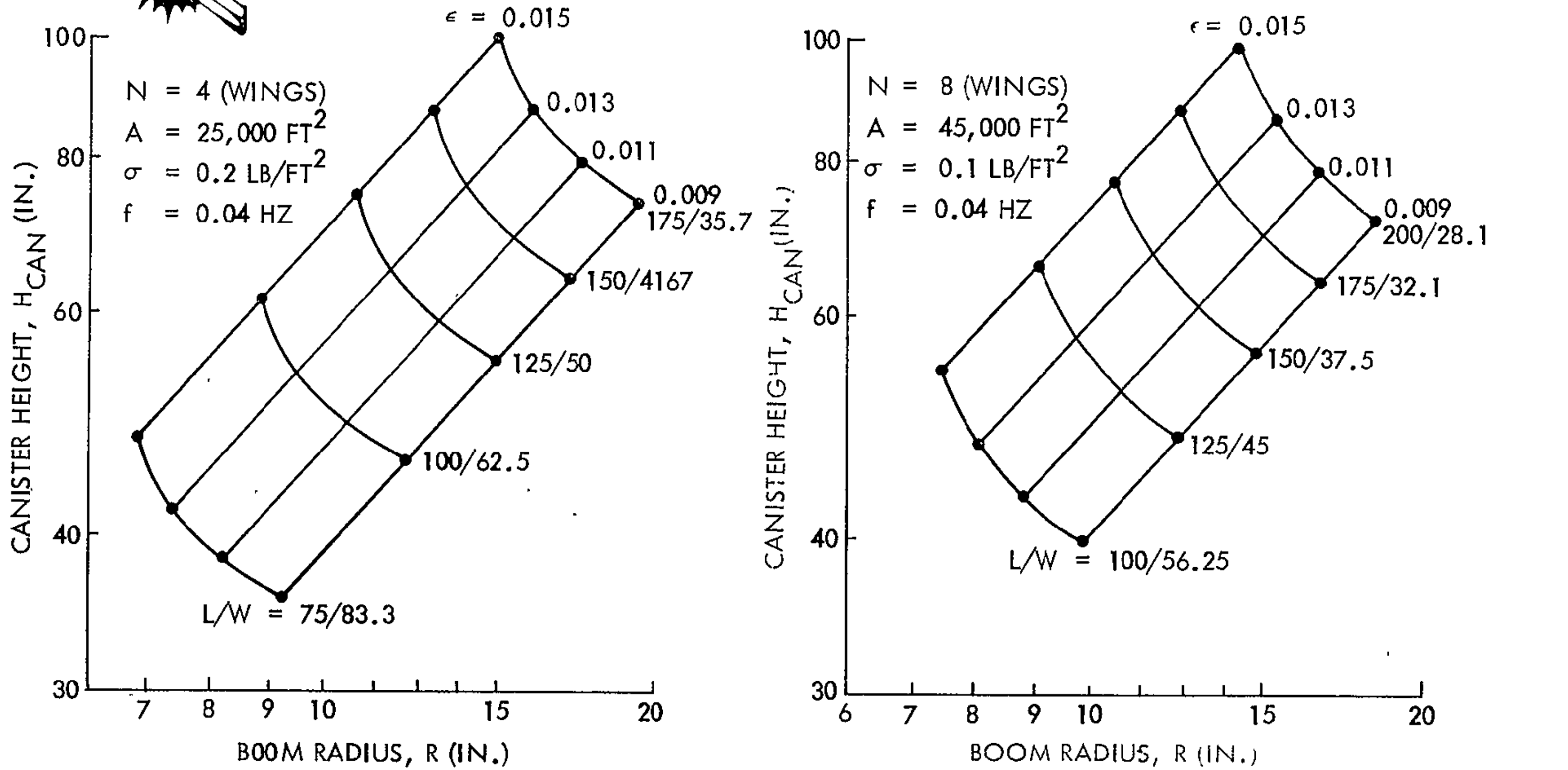


CANISTER HEIGHT AS A FUNCTION OF BOOM RADIUS

ρ = 0.2 LB/FT^2

f = 0.04 Hz

MAST CHARACTERISTICS FOR GROWTH



CANISTER HEIGHT VERSUS BOOM RADIUS

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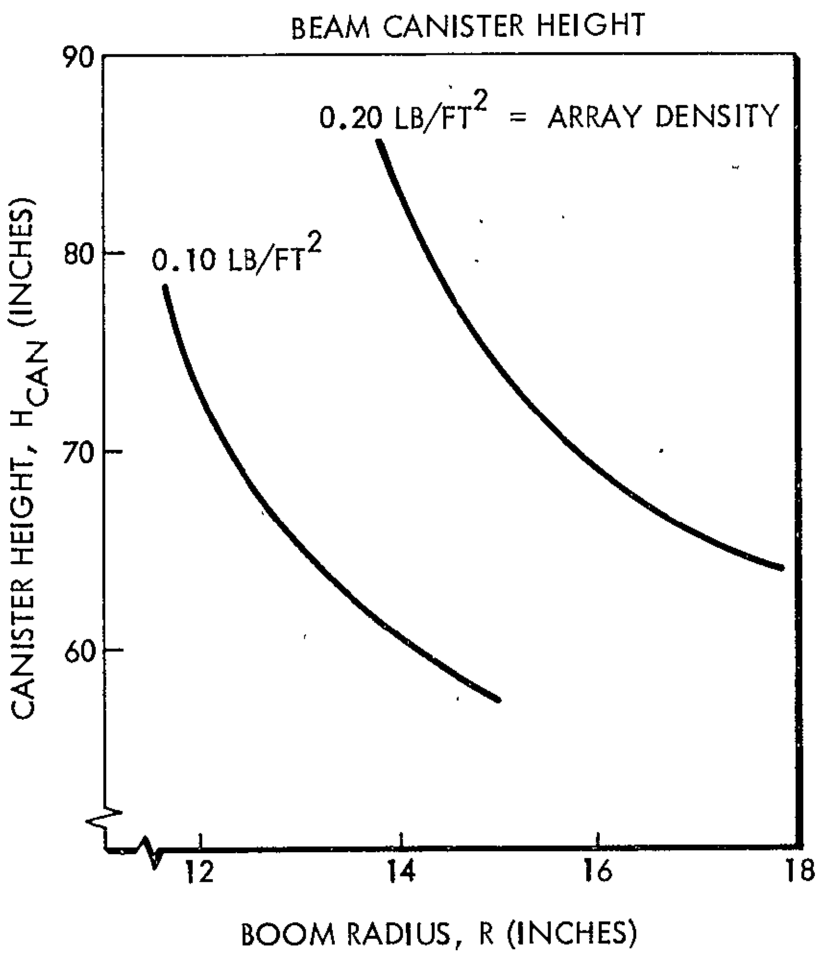
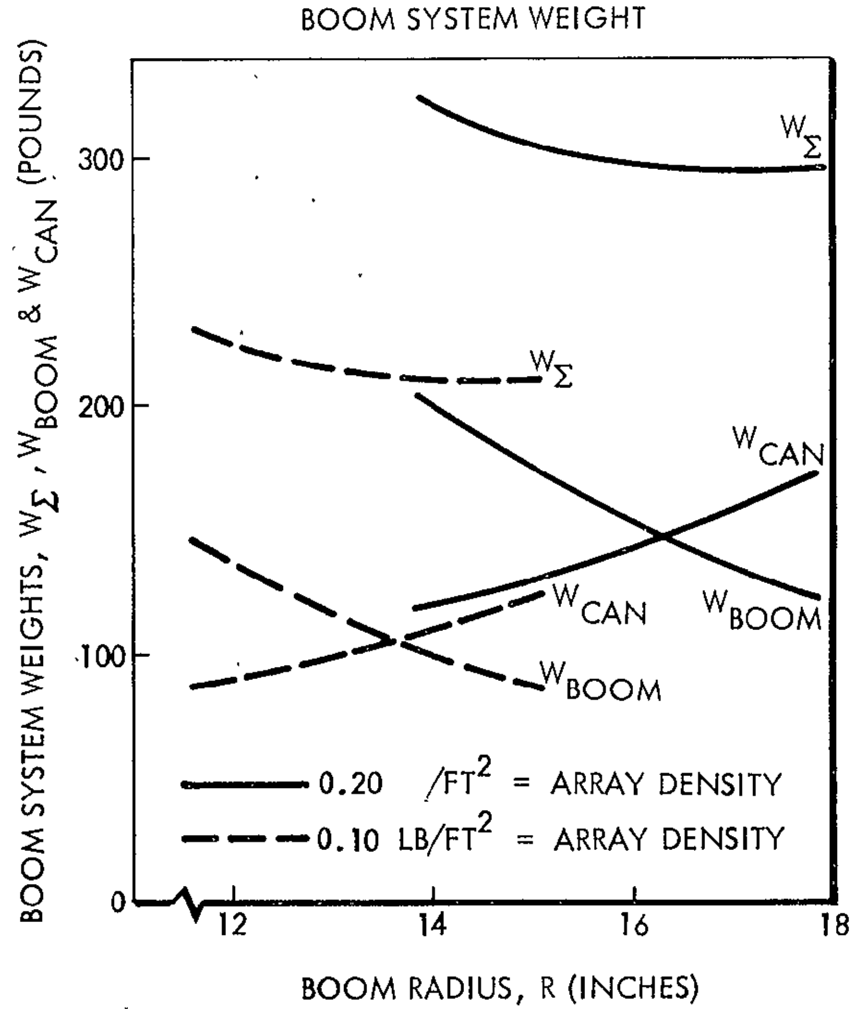
MAST CHARACTERISTICS FOR GROWTH EVALUATION





S/A = 25,000 FT² (4 WINGS)

f_n = 0.04 HZ

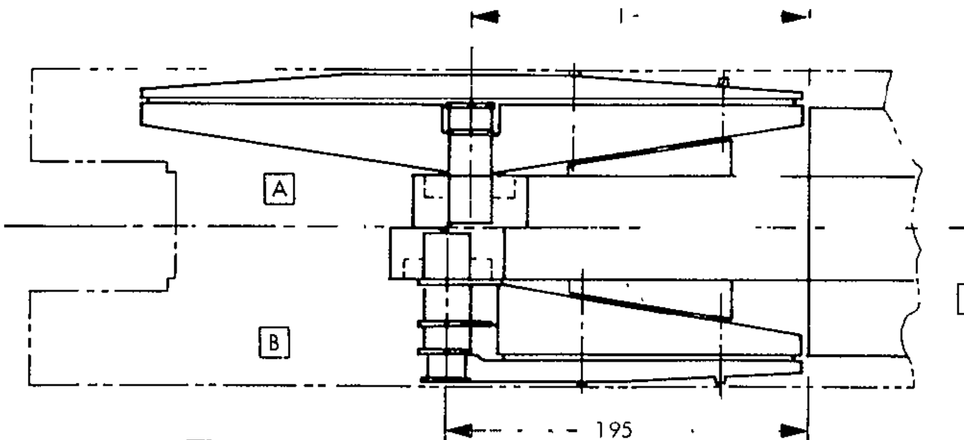
L = 150

W = 41.67 FT

SOLAR ARRAY CONFIGURATIONS STUDY

- Solar array installation and design layout was studied to develop comparisons between alternative arrangements. Six arrangements, as shown on LMSC Drawing SK 58700, were conceived and studied. These configurations ranged from the MSFC baseline fixed solar array to those that are folded and capable of growth to 65 kW power modules. This study assumed that the S/A must have a first mode bending frequency close to 0.04 Hz. In addition, the largest feasible MAST configuration was investigated which would provide for slightly greater stiffness. The folded solar array was estimated to be lighter than the MSFC baseline because of the structural efficiency, particularly when caged for launch. This configuration also minimizes protrusion into the Airlock/MMU regions. Based on this study, LMSC prefers the folded configuration over the fixed arrangement. The fixed versus folded S/A system are shown along with the largest MAST investigated.

SOLAR ARRAY CONFIGURATION STUDY



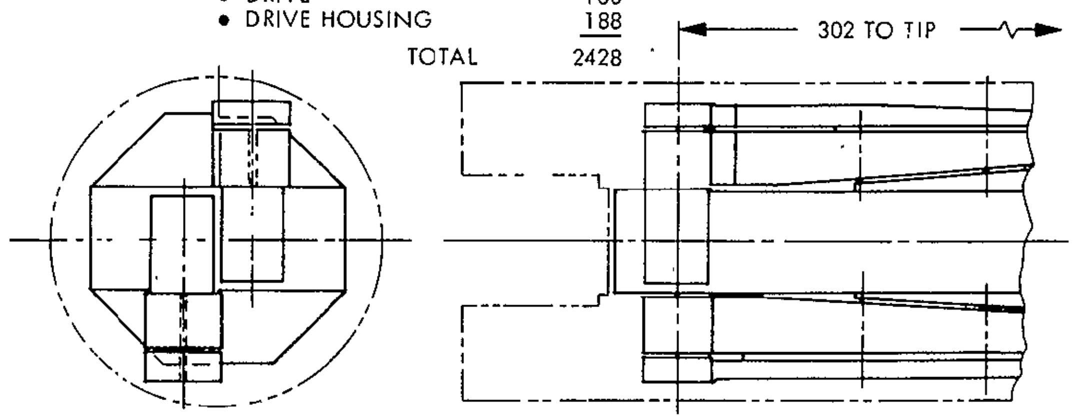
[A] ESTIMATED WEIGHTS (LB)

RIGID CONTAINER DESIGN

• BLANKETS	1380
• MASTS (11R X 71LG CAN)	400
• CONTAINER	440
• DRIVE	130
• DRIVE HOUSING	188
TOTAL	2538

[B] FOLDED CONTAINER DESIGN

• BLANKETS	1380
• MASTS (11 R X 71 LG CAN)	400
• CONTAINER	330
• DRIVE	130
• DRIVE HOUSING	188
TOTAL	2428

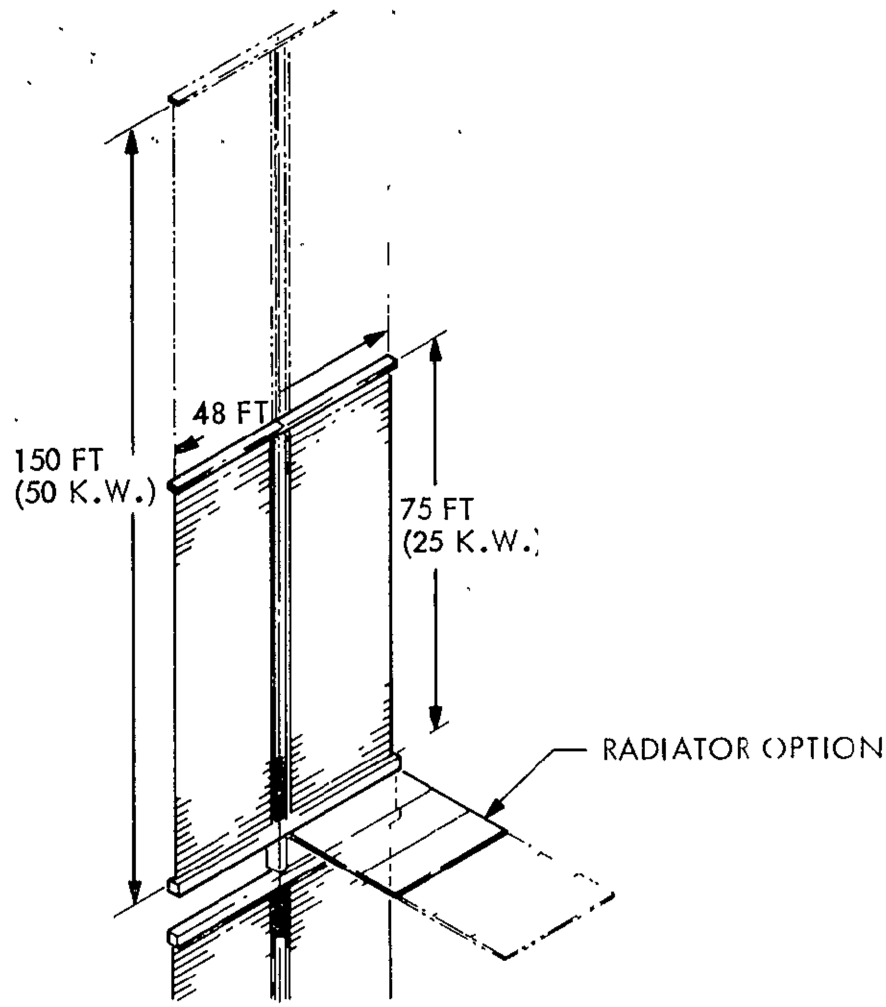


3850 SQ FT BLANKET WITH 15 INCH RADIUS MAST

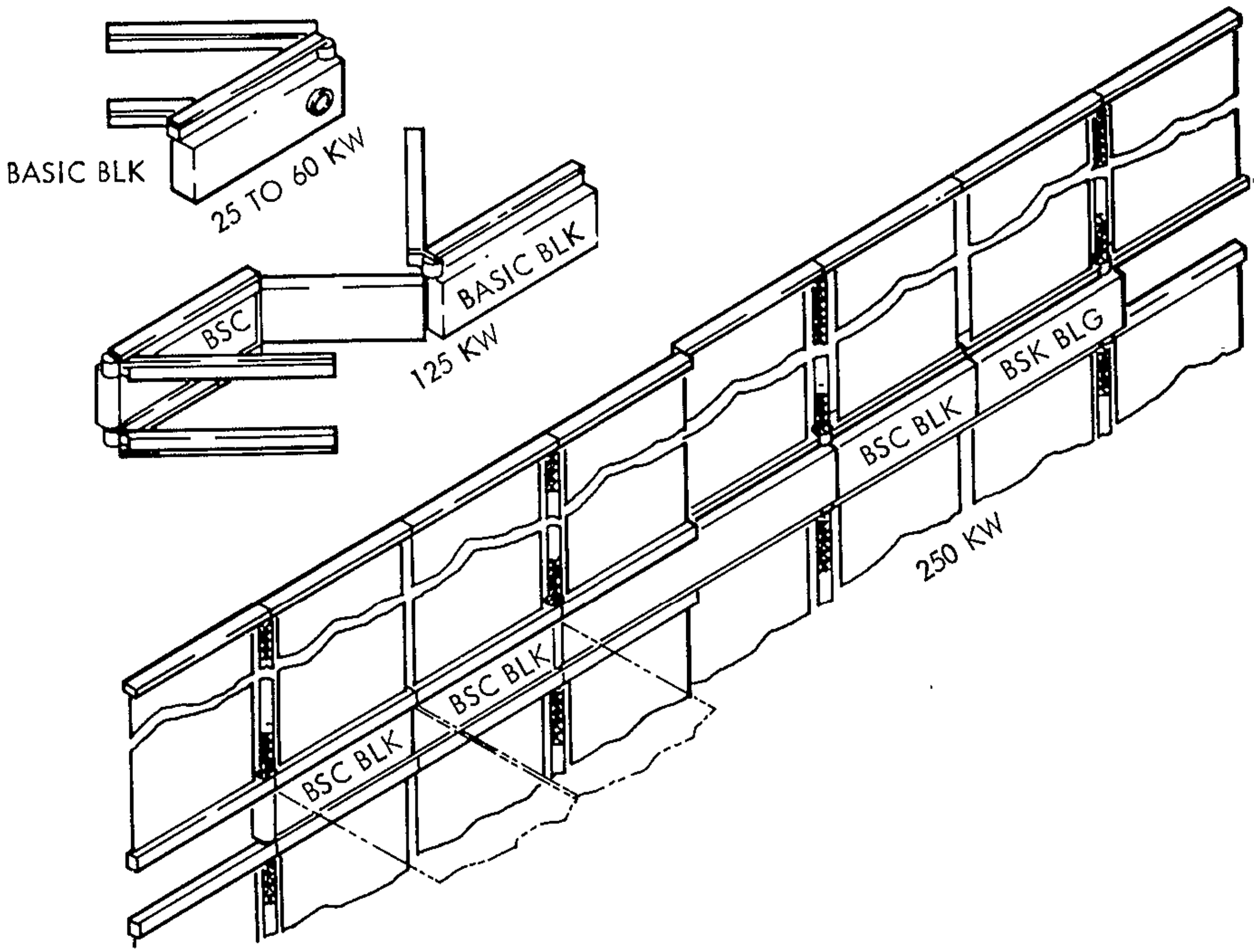
BASIC BUILDING BLOCK OF POWER MODULE FOR GROWTH COMMONALITY

- During the last reporting period LMSC has used the baseline solar array configuration to study and investigate which solar array concepts would provide growth. As a result of this study it has been determined that there is a preferred solar array configuration that can be used for growth up to 250 kW power level (50,000 ft^2 of Solar Array). This approach has developed into a modular solar array building block. This common mechanical element would then be used as power levels increased by use of 2, 4, or 8 units. The initial power module would use two of these units with eight required at the 250 kW power level. These charts illustrate the basic building blocks and its growth to 250 kW.

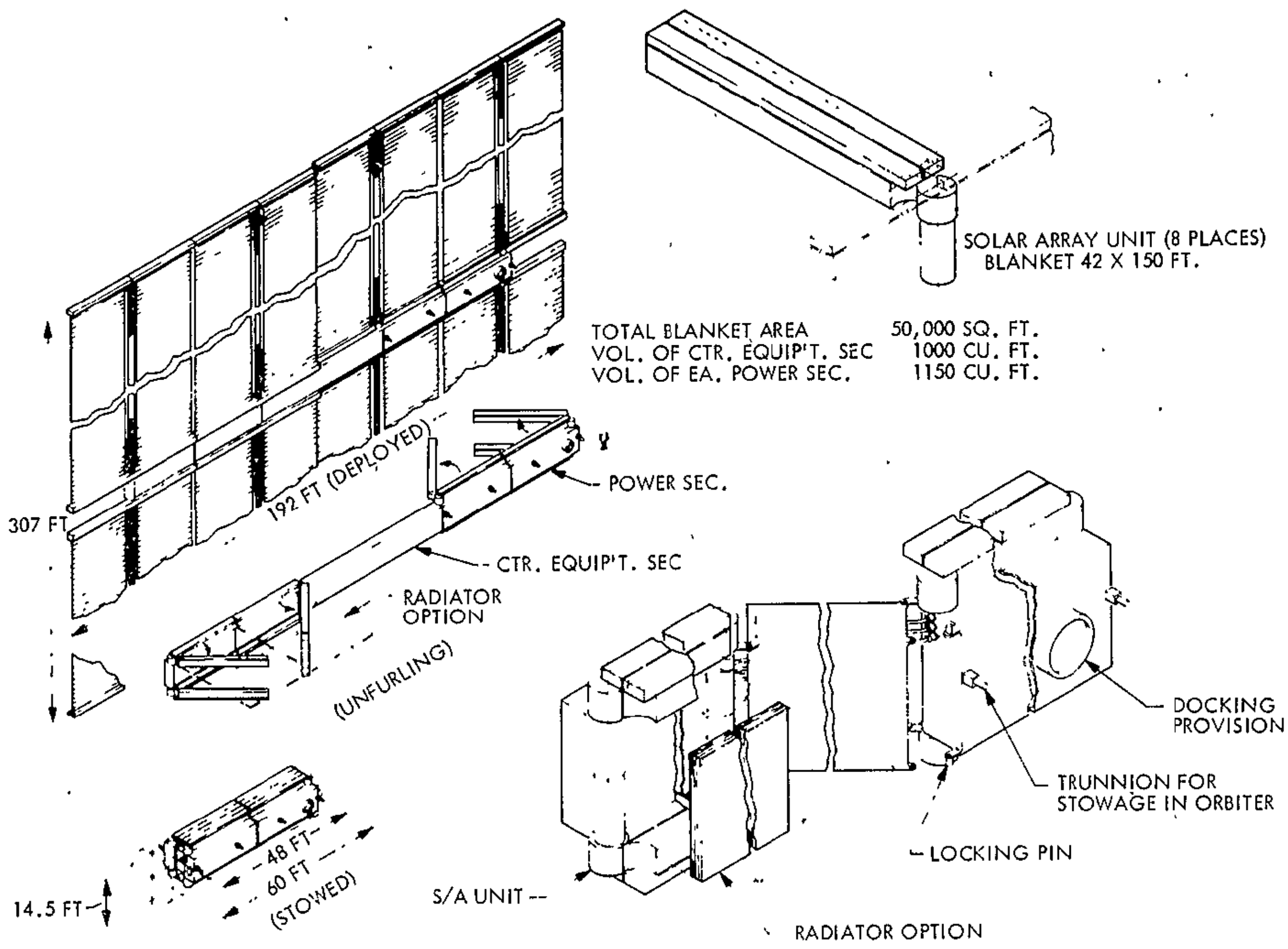
BASIC BUILDING BLOCK OF POWER MODULE FOR GROWTH COMMONALITY



EVOLUTIONARY BUILDING BLOCK CONCEPT



250kW S/A SYSTEMS CONCEPT

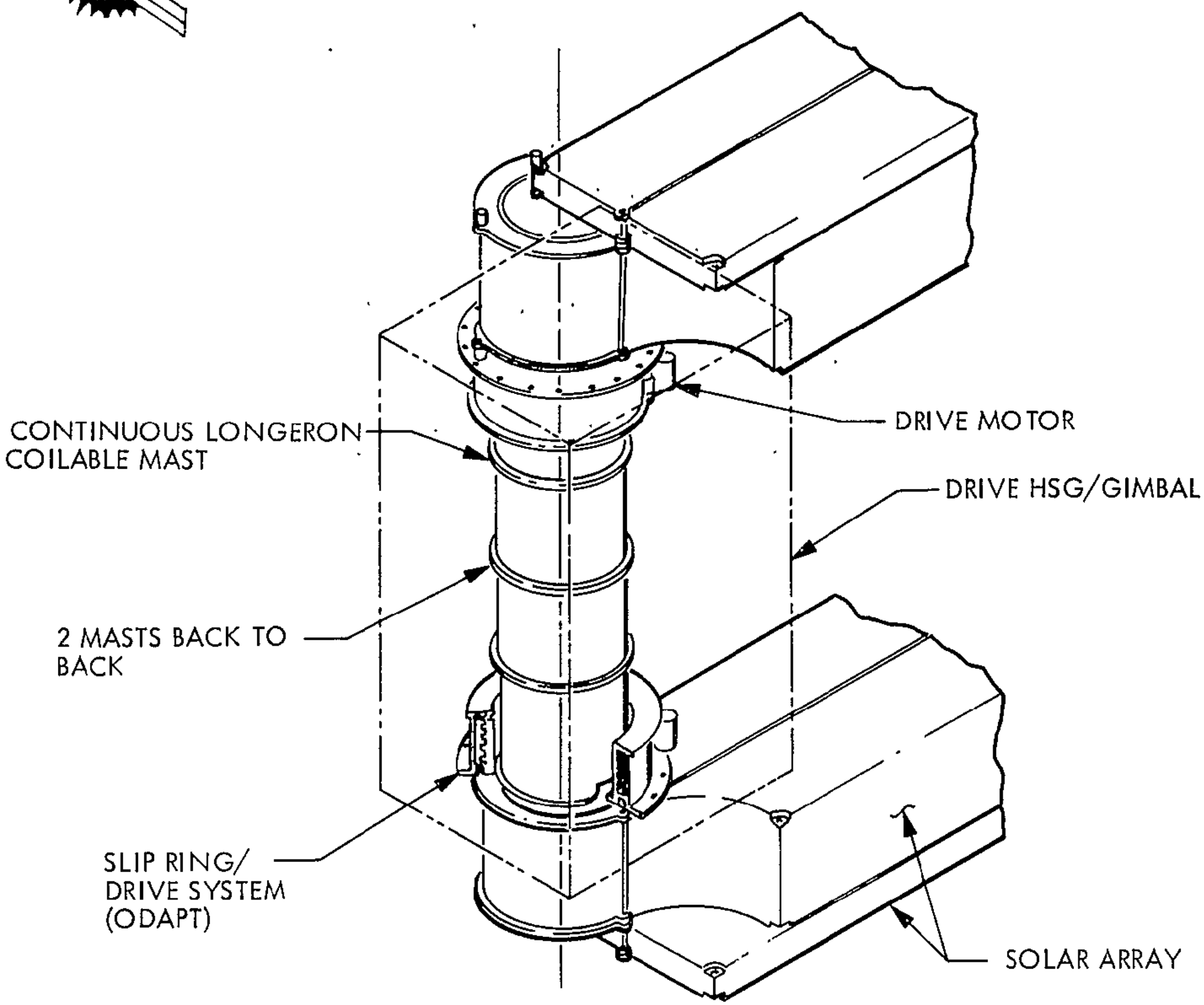


POWER MODULE ODAPT INSTALLATION

- LMSC has developed a set of baseline requirements to determine the drive system characteristics. Using these requirements and the basic installation concepts Ball Aerospace System Division has provided LMSC with a baseline design and supporting comparative component analysis. The basic drive system and power transfer assembly for both solar array sides is estimated to weigh only 300 to 400 lbs, depending on redundancy and built-in growth capabilities. This effort is a direct off-shoot of the Orientation Drive and Power Transfer Assembly technology BASD developed for NASA under subcontract to LMSC. In fact, the outer gimbal of the Space Station Solar Array is nearly identical in size to the drive required for the PM mast axis drive. Therefore, considerable knowledge has been developed on this size ODAPT and is directly applicable to minimize PM effort.

- The following charts show the basic arrangement of the drive to solar array and provide details on the current trades which have led to the baseline power module ODAPT configuration. These charts qualitatively illustrate some considerations that have been used.

POWER MODULE ODAPT INSTALLATION



ORIENTATION DRIVE AND POWER TRANSFER REQUIREMENTS

POWER MODULE
ORIENTATION DRIVE AND POWER TRANSFER
CONSIDERATIONS AND REQUIREMENTS

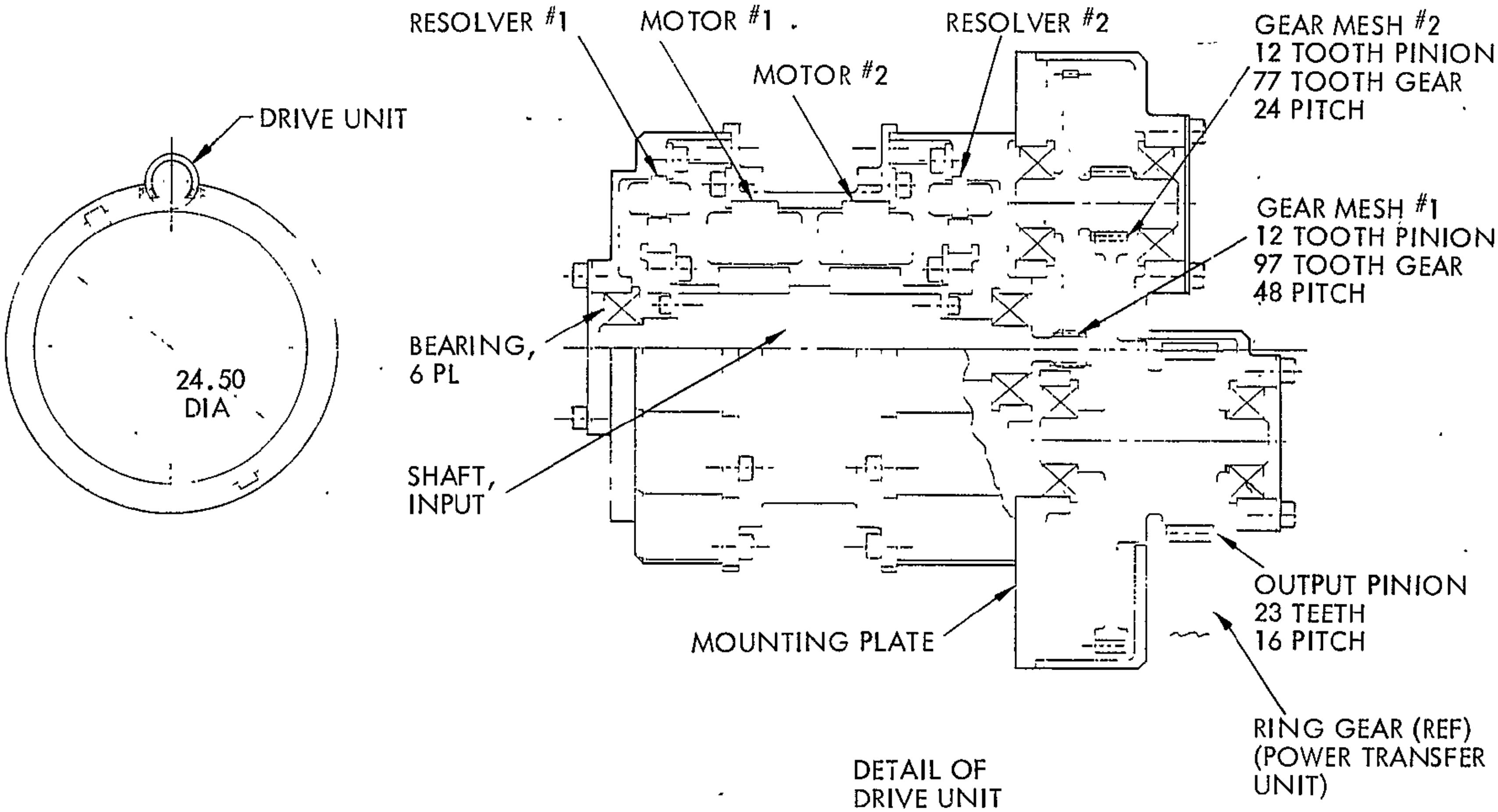
- LIFE – 5 YEARS (MINIMUM)*
- ANGULAR VELOCITY*

ORBIT	-	0.06°/SEC (0.01 RPM)
SLEW	-	0.50°/SEC (0.08 RPM)

- ARRAY GROWTH POTENTIAL - 25 kW TO >100 kW*
- ARRAY INERTIA - 2000 SLUG-FT^2*
- FLEXIBLE ARRAY SUPPORT STRUCTURE ($fn_\theta \approx 0.04$ HZ)*
- SINGLE-DEGREE-OF-FREEDOM (±180°)
- RELIABILITY - MAXIMUM → REDUNDANCY
- WEIGHT - MINIMUM
- COST - MINIMUM
- ANGULAR ACCELERATION (0.06°/SEC → 0.5°/SEC) - 0.04°/SEC^2 +

+BASD IMPOSED SPECIFICATION
*DESIGN DRIVER REQUIREMENTS

ODAPT-BASELINE CONFIGURATION FOR 25-75 kW



ODAPT* COMPONENT CONSIDERATIONS - MOTOR AND POWER TRANSFER

MOTOR CHOICES

		ADVANTAGES	DISADVANTAGES
I	ALTERNATING CURRENT (115 V, 400 HZ)		
	A INDUCTION (ALTERNATE MOTOR)	LOW COMPONENT COST	• PULSED APPLICATIONS (CAN CONTROL SPEED WITH VOLTAGE)
	B SYNC. HYSTERESIS	LIGHT WEIGHT	• HIGHER SPEED • SPEED CONTROL WITH FREQUENCY CHANGE • SPEED CONTROL WITH MECHANICAL CLUTCH AND GEAR TRAIN • HIGH WEIGHT DRIVE DUE TO GEAR TRAIN WEIGHT
II	DIRECT CURRENT (28 VDC) A STEPPER		• PULSED APPLICATIONS • GEAR MESH SHOCK FACTOR (X2)
	B BRUSH TYPE	• FLIGHT PROVEN (OSD, 30 RPM, 5 YEARS) • LOW SYSTEM COST	• POTENTIAL BRUSH LIFE LIMITATION
	C BRUSHLESS TYPE	• FLIGHT PROVEN (DESPIN DRIVES 60 RPM, 5 YEARS)	• ELECTRONIC RELIABILITY FOR COMMUTATION
	COMMUTATION – RESOLVER (BASELINE) – CAPACITOR – HALL EFFECT		

POWER TRANSFER CHOICES

		ADVANTAGES	DISADVANTAGES
I	FLEX CABLES	• LOW COMPONENT COST (EQUAL SYSTEM COST TO DRUM TYPE) • FLIGHT PROVEN	• VARIABLE SYSTEM TORQUE • MODEST GROWTH POTENTIAL TORQUE RELATED • LIFE RELATED TO FLEXURE OF CABLES (QUESTIONABLE FOR SIZE) • UNKNOWN CONTAINMENT FOR LAUNCH LOADS AND VIBRATION
II	PANCAKE SLIP RING	• FLIGHT PROVEN • CONSTANT SYSTEM TORQUE	• MARGINAL GROWTH (RADIAL SIZE LIMIT) • HIGHER SYSTEM COST
III	DRUM SLIP RING	• FLIGHT PROVEN • GOOD GROWTH POTENTIAL (SIZE) • CONSTANT SYSTEM TORQUE	• HIGH COMPONENT COST (EQUAL SYSTEM COST TO FLEX CABLE)

*BALL AEROSPACE SYSTEMS DIVISION DRIVE DESIGN/ANALYSIS EFFORT

ODAPT* COMPONENT CONSIDERATIONS , GEAR, BEARINGS & LUBRICATION

BEARING ARRANGEMENT

	ADVANTAGES	DISADVANTAGES
I DUPLEX PAIR + RADIAL DEEP GROOVE (6 BRGS PER POWER MODULE)	• LARGE STRUCTURAL CAPACITY • THERMAL STABILITY WITHIN EACH DRIVE • EXCELLENT GRD TEST W/O FIXTURE	• WEIGHT • FRICTION TORQUE • HIGHER COST
II 4 POINT SINGLE BEARING	• LIGHT WEIGHT • INTERMEDIATE COST • INTERMEDIATE FRICTION TORQUE	• LARGE FRICTION TORQUE VARIATION (THERMAL) • MARGINAL GROUND - TEST CAPABILITY W/O FIXTURE
III DUPLEX PAIR	• INTERMEDIATE WEIGHT • INTERMEDIATE COST • INTERMEDIATE FRICTION TORQUE • INTERMEDIATE GROUND TEST CAPABILITY W/O FIXTURE	• POOR ALIGNMENT (BEARING MODULE MUST BE PROVIDED FOR AXIAL AND RADIAL ALIGNMENT CONTROL)
IV SINGLE BEARING	• LIGHT WEIGHT • LOW COST • LOW FRICTION TORQUE • GOOD MISALIGNMENT CONTROL	• POOR GROUND TEST W/O FIXTURE • DIFFICULT PRELOAD

GEAR CHOICES

	ADVANTAGES	DISADVANTAGES
I PLANETARY	• HIGH STRENGTH	• LOW EFFICIENCY (WITH HIGH RATIO) • HIGH COMPONENT COSTS
II SPUR OR BEVEL	• HIGH STRENGTH • HIGH EFFICIENCY • FLIGHT PROVEN	• SLIGHTLY HEAVIER
III HARMONIC DRIVE	• LIGHT WEIGHT • LARGE GEAR RATIO	• HIGH COST • LOW EFFICIENCY • REQUIRES LIQUID LUBE FOR LONG LIFE (RESERVOIR DESIGN DIFFICULT)

LUBRICATION CHOICES

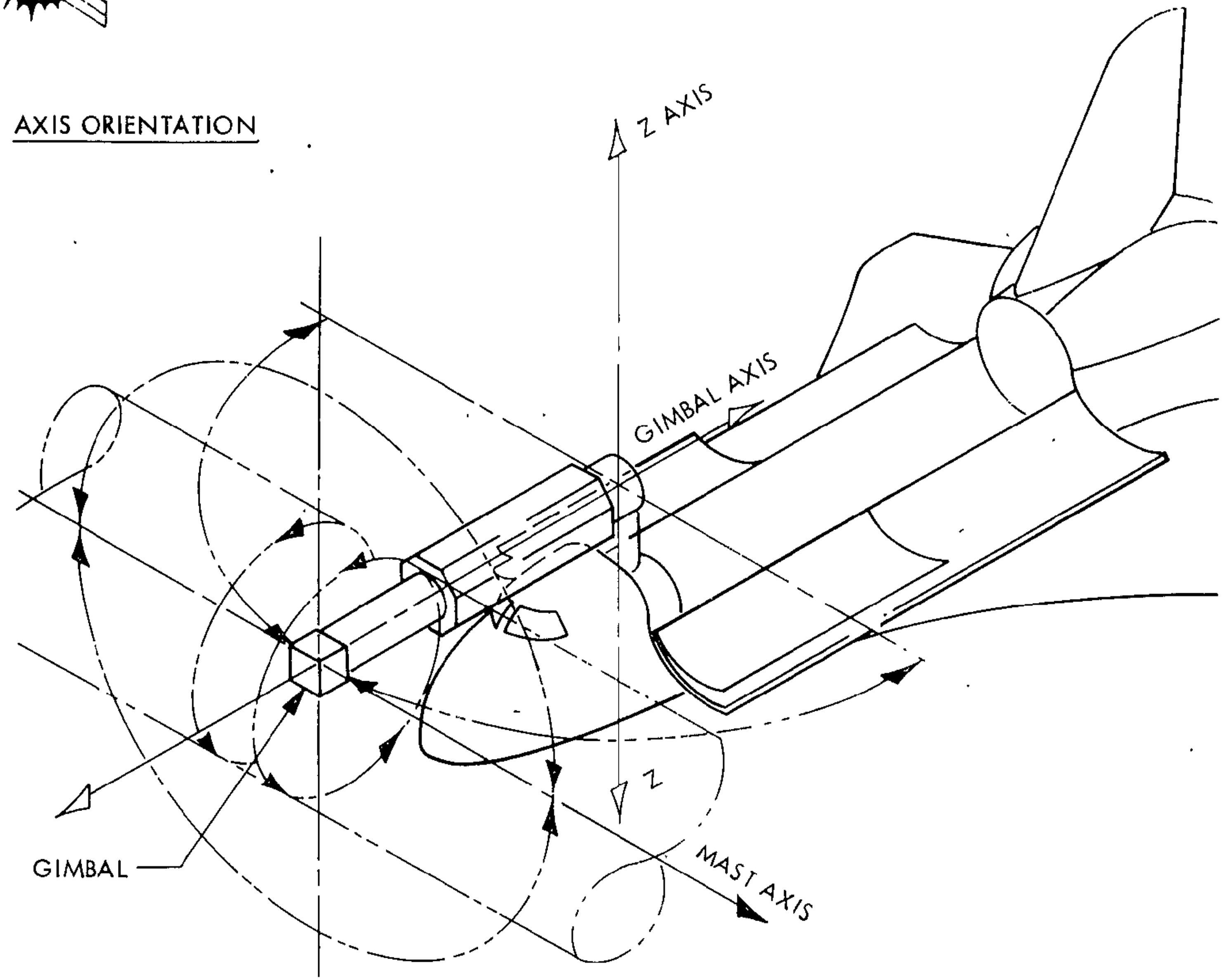
		ADVANTAGES	DISADVANTAGES
WET (LIQUID) VAC KOTE			
BEARINGS	BALL PROCESS SPEC 14.01 (OIL)	• SELF LUBRICATING • DEBPIS CONTAINMENT • FLIGHT PROVEN	• HIGHER TORQUE • RESERVOIRS REQUIRED
GEARS	23560 (GREASE)		
SLIP RINGS	BPS 13.10		
DRY VAC KOTE			
BEARINGS	21207	• LOW FRICTION TORQUE	• NO SELF LUBRICATION ON GEARS
GEARS	23561	• FLIGHT PROVEN	
SLIP RINGS	SM 473 MATERIAL		• HIGHER COST

SOLAR ARRAY ORIENTATION
TWO DEGREES OF FREEDOM

- The ability of the Power Module to provide power for all Beta regimes and control spacecraft orientation is a function of the solar array position. If the solar array could be oriented throughout the mission, almost full output could be assured regardless of the vehicle attitude. Therefore, the study of gimballed solar arrays was made to illustrate that the solar array could be positioned to provide improved output. This study is shown in LMSC Drawing SK 525780. There are up to two degrees of freedom depicted which can be used to re-position the solar array and can provide close to 100% normal operation to the sun. Simpler orientation with allowances for Beta adjustment with a single orbital tracking axis are also possible. The geometry of the Orbiter/PM however, does not allow two degrees of freedom without affecting the baseline design. All the arrangements require minor modification to provide added gimballing.

SOLAR ARRAY ORIENTATION — TWO DEGREES OF FREEDOM

AXIS ORIENTATION



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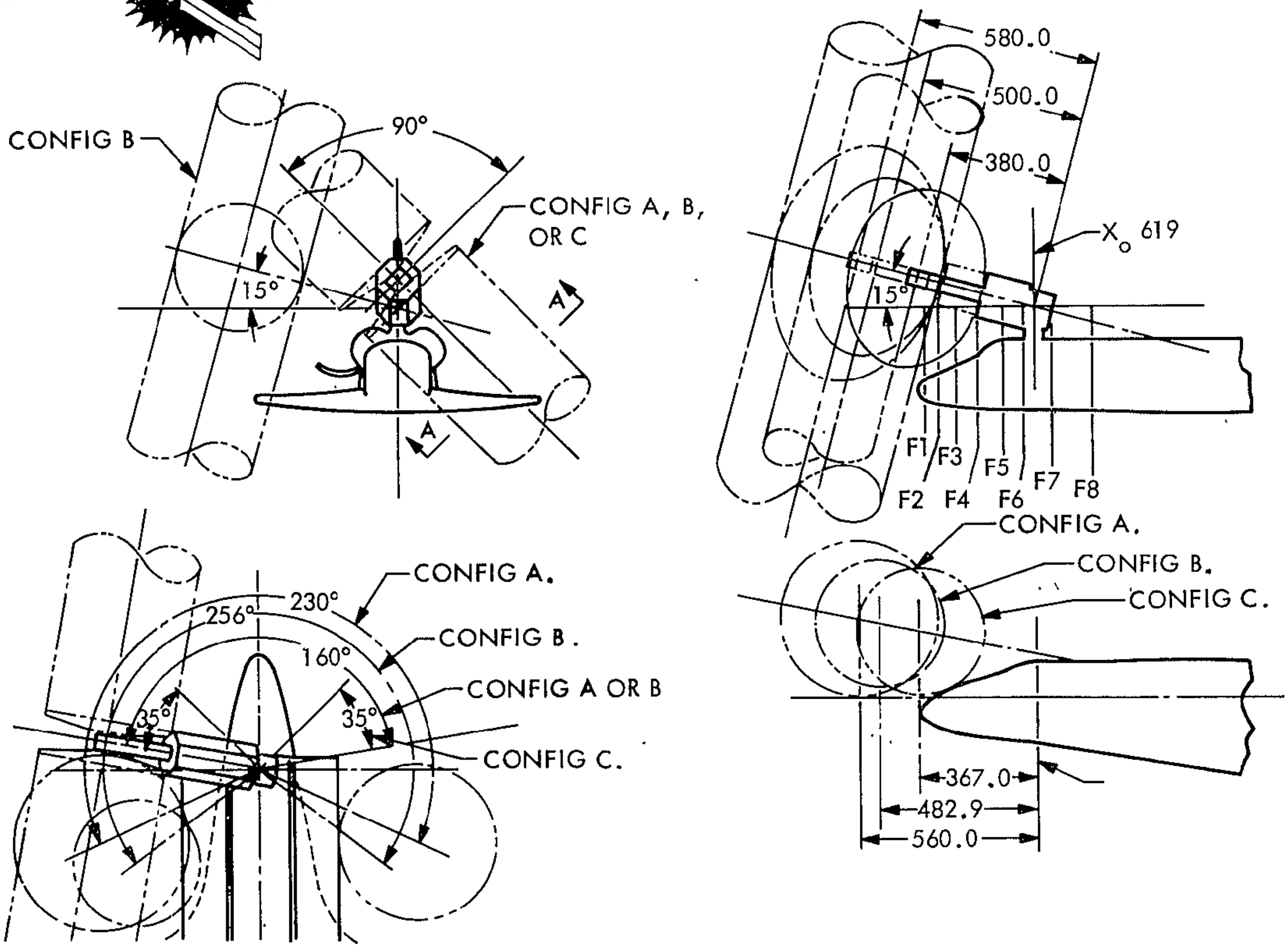
SOLAR ARRAY/PM ORIENTATION CONFIGURATION STUDY



POWER ELECTRONICS CONFIGURATION TRADES

- EPS Concept 2 has been selected as the best overall power system for the 25 kW power module because of its high efficiency, best system performance, and ease of integration. This concept uses a buck regulator for the battery charger and a transformer coupled converter for the output regulator.

- With a power system of this scale (≃ 55 kW in and 27 kW out) efficiency drives the concept selection because of the cost and weight impact on the overall system. For example, although the selected system control concept is 207 pounds heavier than the lightest weight system, this difference is negated by the 1 percent efficiency difference.

- Compared to the MSFC Labs baseline, the BUCSAR/TCC approach saves 1370W of solar array and 838 pounds of system weight. When the cost of the additional controller development is included, there is still a net cost saving of $543K with that development amortized over one PM flight.

POWER ELECTRONICS CONFIGURATION TRADES

CONCEPT	EFFICIENCY(2)	CHARGER+ OUTPUT REG. WEIGHT (LB)	EMC/GROUNDING	REMARKS
1 MSFC-LABS BASELINE (LMSC)(1) (BUCK/BUCK)	0.669 (0.730)	410 656 1066	ORBITER NOT ISOLATED	HIGH TRANSFORM RATIO REDUCES EFFICIENCY
2 LMSC-SELECTED BASELINE (BUCK/TCC)	0.771	388 264 652	LOWEST RIPPLE CONFIGURATION ORBITER GROUND ISOLATED	RPC ELIMINATED
3 DIRECT/BUCK W. COMMON BUCK REGULATOR	0.723	193 656 849	ORBITER NOT ISOLATED	WIDE-RANGE INPUT TRANSFORM.RATIO, REDUCES EFFICIENCY PEAK-POWER TRACKING MORE DIFFICULT
4 DIRECT/TCC	0.761	181 264 445	ORBITER GROUND ISOLATED	WIDE-RANGE INPUT REDUCES EFFICIENCY PEAK POWER TRACKING MORE DIFFICULT

(1) BASED ON LMSC BUCSAR ACTUAL EFFICIENCY

(2) ASSUMES DISTRIBUTION LOSSES OF 2% IN ORBITER AND 2% IN POWER MODULE

3000 WATT DC/DC
BUCK VS TRANSFORMER COUPLED CONVERTERS

Buck Circuit

Advantages

- One less magnetic component
- Possibly fewer power transistors
- Only 1 switching diode
- Lowest transistor voltages

Disadvantages

- Higher current through switching transistors
- Higher input filter capacitor RMS current
- Large output inductor
- No input/output isolation

Transformer Coupled Converter Circuit

Advantages

- Isolation: Input short will not short output and input power and input ground can be isolated
- Fewer input capacitors due to lower RMS capacitor current
- Smaller output inductor
- Reduced current through switching elements
- Higher efficiency than Buck circuit at condition analyzed
- Higher conversion power per unit weight

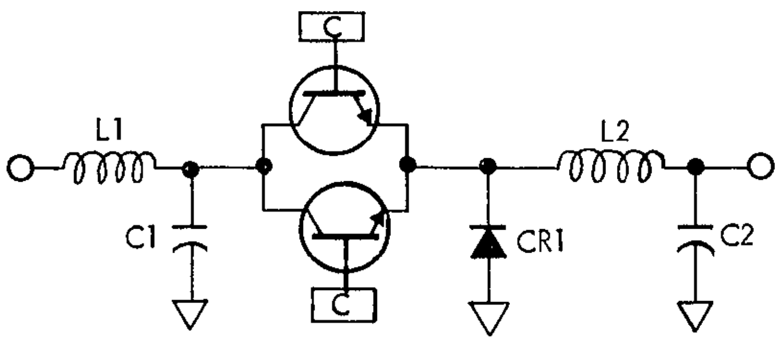
Disadvantages

- More magnetic components
- Possibly high transistor voltages

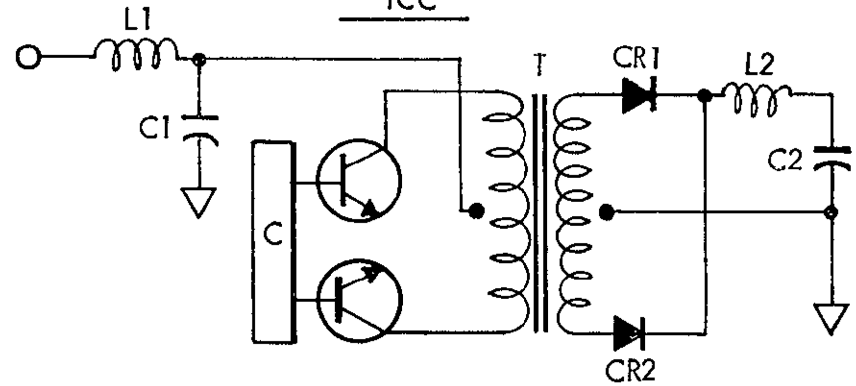
3000 WATT DC/DC BUCK VS TRANSFORMER COUPLED CONVERTERS

BASIC CIRCUIT

BUCK



TCC



COMPONENT ANALYSIS

	% LOSS	WEIGHT (LB)	% LOSS	WEIGHT (LB)
INDUCTOR	2.6	13.5	1.4	5.0
TRANSFORMER	–	–	1.2	1.8
TRANSISTORS	5.7	0.6	1.7	.14
DIODES	2.8	0.16	3.1	.16
CAPACITORS	0.2	2.5	.05	.68
CONTROLS	0.15	1.0	.15	1

EFFICIENCY/POWER DISSIPATION

ANALYTICAL	88	92
PREDICTED	85-87 (MFSC)	–
ACTUAL (LMSC)	87.5-88.5	AVAILABLE 1 AUG 78
POWER DISSIPATION	410 WATTS	260 WATTS
SPECIFIC PERFORMANCE	48 WATTS/LB	114 WATTS/LB

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EPS — SUMMARY

- SOLAR ARRAY CAN BE SCALED TO 250 kW POWER LEVEL WITH BUILDING BLOCK CONCEPT(S)
- NiH_2 BATTERIES PROVIDE SUFFICIENT WEIGHT SAVINGS TO MERIT IMMEDIATE DEVELOPMENT FOR NASA HIGH-POWER LEO MISSIONS AS EARLY AS 1986
- PROJECTIONS FOR GAINS IN POWER ELECTRONICS EFFICIENCY AND USE AT HIGHER VOLTAGES ALLOWS EFFECTIVE GROWTH WITHOUT SACRIFICE OF WEIGHT AND THERMAL DISSIPATION
- INITIAL BUILDING BLOCK CONCEPT MINIMIZES RDT&E TO ACCOMMODATE SOLAR ARRAY SYSTEM GROWTH
- ADVANTAGES OF TCC OVER BUCK REGULATOR WARRANTS ITS USE FOR POWER MODULE
- SOLAR ARRAY DEPLOYMENT MAST CAN PROVIDE SUFFICIENT STIFFNESS TO MEET AT LEAST A 0.04 HZ FREQUENCY REQUIREMENT AT LENGTHS TO 150 FEET
- ORIENTATION OF THE SOLAR ARRAY ABOVE 20,000 FT^2 MAY REQUIRE AN INERTIAL SOLAR ARRAY WITH INDEPENDENT ORIENTATION FOR THE PAYLOAD
- ADDITIONAL DEGREES OF FREEDOM FOR S/A SYSTEM ON THE INITIAL POWER MODULE CAN PROVIDE ADDED MISSION FLEXIBILITY

APPENDIX C
MAINTENANCE TIME ESTIMATE

The Appendix contains time estimates for 25/50 kW Orbital Replaceable Unit (ORU) changeouts, and a step-by-step sequence for performing the EVA 50 kW to 100 kW conversion.

These data are provided as backup information for the summaries presented in Volumes 2 and 4, representative of the scope of study accomplished.

EVA TIMELINE: 50 kW-100 kW CONVERSION

EVA	CREWMAN EV A	EV B	RMS C	TIME (MINUTES)	
1. EGRESS A/L, TRANSLATE TO WORK SITE, UNSTOW WORK PLATFORM, BEGIN SET-UP	X			10.0	
2. EGRESS A/L, UNSTOW RMS, ATTACH CAMERA TO RMS		X	X	6.0	
3. COMPLETE WORK STATION SET-UP, EXTEND EQUIPMENT MAST, ADJUST UPPER PLATFORM	X			5.5	
4. ROTATE PM, UNLATCH 1/2 "OLD S/A"	X			3.5	
5. POSITION RMS NEAR GRAPPLE FIXTURE, PREPARE ORBITAL CRADLE LATCH		X	X		2.0
6. GRAPPLE 1/2 "OLD S/A", VERIFY	X		X	1.5	
7. RMS PARTS S/A FROM PM, POSITION AT STOW SITE A, LATCH "OLD S/A"		X	X	3.5	
8. REPEAT 2ND 1/2 "OLD S/A", LATCH AT STOW SITE B	X	X	X	8.5	
9. RMS UNGRAPPLES FROM "OLD S/A", TRANSLATES AND GRAPPLES 50 kW STRUCTURE BEAM			X	2.0	
10. UNLATCH S/A STRUCTURE EXTENSION, POSITION AT PM		X	X	2.5	
11. ALIGN, ATTACH STRUCTURE EXTENSION TO PM	X			8.0	
12. RETURN RMS TO P/L BAY, UNLATCH, GRAPPLE 1/2 "NEW S/A"		X	X		6.0
13. RMS PARTS 1/2 S/A FROM CRADLE, TRANSLATE, POSITION AT STRUCTURE BEAM			X	2.0	
14. ALIGN, ATTACH S/A TO STRUCTURE BEAM	X			6.0	
15. REPEAT STEPS FOR SECOND HALF S/A	X	X	X	8.0	
REST PERIOD	X	X	X	5.0	
16. POSITION RMS AT STOW SITE A, GRAPPLE "OLD S/A", UNLATCH		X	X	3.0	
17. RMS PARTS S/A, POSITIONS AT STRUCTURE EXTENSION				2.0	
18. ALIGN, ATTACH 1/2 "OLD S/A" TO STRUCTURE EXTENSION	X			8.0	
19. RMS POSITIONS AT STOW SITE B, UNLATCH, GRAPPLE		X	X		3.0
20. REPEAT STEPS FOR SECOND 1/2 "OLD S/A"	X	X	X	13.0	
21. REMOVE, STOW WORKSTATION, STOW RMS, REMOVE CAMERA	X	X	X	15.0	
22. MATE UMBILICAL, TRANSFER TO "SAFE" POSITION, OBSERVE CHECKOUT	X	X		6.0	
23. PSS/MS PERFORM, OPERATIONAL CHECKOUT				15.0	
• VERIFY COMMAND LINK, COMMAND PM SYSTEMS ON					
• VERIFY PM ACS STATUS, PM-ORBITER ACS STABILIZATION					
• EXTEND S/A WINGS, EXTEND RADIATOR BOOM					
• VERIFY PM-TO-ORBITER POWER TRANSFER, THERMAL SYSTEM STATUS					
• VERIFY ORBITER-TO-PM HEAT REJECTION					
• VERIFY PM-TO-ORBITER FUNCTIONAL INTERFACES					
• CONFIRM OPERATIONAL READINESS					
24. SAFETY CHECK ORBITER P/L BAY, INGRESS A/L	X	X		30	
SUBTOTAL				164.0	
CONTINGENCY 10%				16	
TOTAL				180.0	

25/50 kW ORBITAL REPLACEMENT UNIT CHANGEOUT TIME ESTIMATES

ORU	NO. REQ	NO. ORU	TIME (MINS) REMOVE	PER ORU REPLACE	EST WEIGHT PER ORU (LBS)
ELECTRICAL POWER					
BATTERY MODULE ($Ni-H_2$ TYPE) (5 PER MODULE)	60	12	19.0	14.5	633.0
BATTERY CONTROL ASSY	60	12	16.5	14.0	55.0
BUS REGULATOR	60	12	12.0	2.0	55.0
SOLAR ARRAY WING	2	2	31.5	29.5	1400.0
SOLAR ARRAY DRIVE ASSY	2	2	33.0	31.0	125.0
POWER TRANSFER ASSY	2	2	15.5	11.0	125.0
POWER INTERFACE DISTRIBUTOR	1	1	29.5	24.5	300.0
MAIN POWER DISTRIBUTOR	1	1	16.5	14.5	50.0
SOLAR ARRAY DISTRIBUTOR	1	1	14.5	12.5	30.0
BERTHING DISTRIBUTOR	1	1	11.5	10.0	150.0
RACK DISTRIBUTORS	3	3	12.0	9.5	30.0
ATTITUDE CONTROL SUBSYSTEM					
SUN SENSOR	2	2	6.0	5.0	3.7
SUN SENSOR ELECTRICAL ASSEMBLY	2	2	6.5	6.0	1.33
ATM RATE GYRO	9	9	9.5	7.0	11.5
CMG	3	3	24.0	21.0	420.0
CMG INVERTER ASSEMBLIES	3	3	16.0	14.0	52.0
CONICAL SCAN HORIZON SENSORS (4) AND ELECTRONICS (2)	6	2	16.0	11.5	8.5
MAGNETOMETER AND ELECTRONICS	1	1	18.0	14.0	4.0
MAGNETIC TORQUERS	4	4	17.5	15.5	110.0
MAGNETIC TORQUERS ELECTRONICS	1	1	21.0	17.5	16.0
WIDE ANGLE SUN SENSORS AND ELECTRONICS	1	1	16.0	11.5	2.0
SIGNAL CONDITIONER AND INTERFACE UNIT	1	1	19.5	16.5	45.0
THERMAL SUBSYSTEM					
RADIATOR	27	1	31.0	29.0	2565.0
PUMP & ACCUMULATOR & CHECK VALVE & INVERTER	4	2	19.5	16.5	45.0
PRESSURE TRANSDUCERS	2	2	7.5	6.0	10.0
FLOW CONTROL ASSEMBLY (MIXER VALUE)	3	3	11.0	9.5	25.5
INTERFACE H_X	2	2	31.5	29.0	15.0
GSE H_X	2	2	17.5	15.0	17.0
HEATERS	1	1	13.5	11.0	TBD
RADIATOR EXTENSION ROTATION MECHANISM	2	2	31.0	29.0	15.0
RADIATOR DEPLOYMENT MOTORS	52	–	TBD	–	2.0
C & DH SUBSYSTEM					
TRANSPONDER	2	2	17.0	13.0	15.6
S BAND OMNI ANTENNA	2	2	9.5	6.0	0.45
S BAND TRACKING ANTENNA	2	2	21.0	19.5	20.0
ANTENNA DRIVE ASSY AND SUPPORT	2	2	23.5	21.0	49.2
ANTENNA STEERING ASSEMBLY	2	2	19.5	17.0	48.0
COMPUTER	2	2	18.0	14.0	130.0
DATA HANDLING UNITS	2	2	29.5	27.0	60.0
REDUNDANCY MANAGEMENT UNITS	2	2	19.5	16.5	TBD

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TRW