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(Vol. 2) (Final) (Pt. B)

NUCLEAR SHUTTLE SYSTEMS DEFINITION STUDY
PHASE III

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GEORGE C. MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA
UNDER CONTRACT NAS8-24715
MSCF DRL-197, LINE ITEM 3

LMSC
SPACE SYSTEMS DIVISION

FINAL REPORT

VOLUME II
CONCEPT AND FEASIBILITY ANALYSIS
Part B - Baseline System Definition

LOCKHEED MISSILES & SPACE COMPANY
A GROUP DIVISION OF LOCKHEED AIRCRAFT CORPORATION
SPACE SYSTEMS DIVISION • SUNNYVALE, CALIFORNIA

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FOREWORD

This report presents the results of the Nuclear Shuttle Systems Definition Study, Phase III, performed by the Lockheed Aircraft Corporation under Contract NAS8-24715. This documentation represents the submittals identified under MSFC DRL-197, Line Items 3, 5, 6, and 7. This phase of the subject study was performed under the technical direction of Mr. C. C. Priest, PD-SA-P, George C. Marshall Space Flight Center.

The report is published in a total of eight volumes:

Volume I	Executive Summary
Volume II	Concept and Feasibility Analysis
Part A	System Evaluation and Capability
Part B	Baseline System Definition
Part C	Systems Engineering Documentation
Volume III	Program Support Requirements
Volume IV	Cost Data
Volume V	Schedules, Milestones, and Networks
Volume VI	Reliability and Safety Analysis
Volume VII	RNS Tank Pressurization Analyses
Volume VIII	RNS Test Program Requirements – NRDS

Mc

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Mr. C. E. Vivian	Operating and Development Requirements
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CONTENTS

Section		Page
	FOREWORD	iii
1	INTRODUCTION	1-1
	1.1 Reusable Nuclear Shuttle Concept	1-1
	1.2 Reusable Nuclear Shuttle Definition	1-1
2	STRUCTURAL SYSTEM	2-1
	2.1 Propellant Tank Assembly	2-1
	2.1.1 Propellant Module	2-6
	2.1.2 Propulsion Module	2-11
	2.2 Engine Thrust Structure	2-11
	2.3 Forward and Aft Skirt	2-15
	2.3.1 Propellant Module	2-15
	2.3.2 Propulsion Module	2-15
	2.4 External Fairing	2-16
	2.5 Propellant Containment System	2-16
	2.6 Engine Flyaway Module Structure	2-17
	2.7 Space Frame Assembly	2-17
	2.8 Insulation	2-25
	2.8.1 Propellant Module	2-26
	2.8.2 Propulsion Module	2-26
	2.9 Meteoroid Shielding	2-27
	2.9.1 Propellant Module	2-27
	2.9.2 Propulsion Module	2-27
	2.10 Docking and Tank Support	2-27
	2.10.1 Propellant Module	2-28
	2.10.2 Propulsion Module	2-28
	2.10.3 Spaceframe	2-28
	2.11 Vehicle Assembly Support Equipment	2-29

Section			Page
3	PROPULSION SYSTEM		3-1
	3.1 Introduction		3-1
	3.2 Primary Engine		3-1
	3.3 Engine Flyaway Module		3-5
		3.3.1 Module Requirements	3-5
		3.3.2 Module Description	3-5
		3.3.3 Module Characteristics	3-8
	3.4 Propellant Feed and Storage		3-9
		3.4.1 Subsystem Requirements	3-9
		3.4.2 Subsystem Description	3-9
		3.4.3 Subsystem Characteristics	3-19
	3.5 Pressurization System		3-22
		3.5.1 Subsystem Requirements	3-22
		3.5.2 Subsystem Description	3-22
		3.5.3 Subsystem Characteristics	3-26
	3.6 Reaction Control System		3-26
		3.6.1 Subsystem Requirement	3-26
		3.6.2 Subsystem Description	3-26
		3.6.3 Subsystem Characteristics	3-33
4	ASTRIONIC SYSTEM		4-1
	4.1 Introduction		4-1
	4.2 Data Management System		4-7
		4.2.1 Subsystem Requirements	4-7
		4.2.2 Subsystem Description	4-7
		4.2.3 Subsystem Characteristics	4-8
	4.3 Communications System		4-8
		4.3.1 Subsystem Requirements	4-8
		4.3.2 Subsystem Description	4-9
		4.3.3 Subsystem Characteristics	4-12
	4.4 Navigation, Guidance & Control System		4-14
		4.4.1 Subsystem Requirements	4-14
		4.4.2 Subsystem Description	4-14

Section		Page
	4.4.3 Subsystem Characteristics	4-21
	4.5 Power System	4-21
	4.5.1 Subsystem Requirements	4-21
	4.5.2 Subsystem Description	4-21
	4.5.3 Subsystem Characteristics	4-27
	4.6 Environmental Control Subsystem	4-32
	4.6.1 Subsystem Requirements	4-32
	4.6.2 Subsystem Description	4-32
	4.6.3 Subsystem Characteristics	4-38
5	VEHICLE DEFINITION	5-1
	5.1 Configuration	5-1
	5.1.1 Spaceframe	5-1
	5.1.2 Propulsion Module	5-5
	5.1.3 Propellant Module	5-15
	5.1.4 Baseline RNS	5-25
	5.2 Mass Properties	5-33
	5.2.1 Detailed Weight Statement	5-33
	5.2.2 Module Mass Properties	5-33
	5.2.3 RNS Baseline Mass Properties – Assembly	5-41
	5.2.4 RNS Baseline Mass Properties – Mission	5-41
	5.3 Environment	5-50
	5.3.1 Thermal	5-50
	5.3.2 Radiation	5-54
6	INTERFACE DEFINITION	6-1
	6.1 Engine Interface	6-1
	6.2 Space Shuttle Interface	6-1
	REFERENCES	R-1

BLANK

ILLUSTRATIONS

Figure		Page
1-1	RNS Baseline Configuration	1-3
1-2	RNS Lunar Mission Performance Capability	1-6
1-3	Work Breakdown Structure	1-7
1-4	Engineering Drawing Tree	1-9
1-5	System Schematic Tree	1-11
2-1	Structural System	2-2
2-2	Propellant Module Structure	2-3
2-3	Propulsion Module Structure	2-4
2-4	Control and Assembly Structure (Spaceframe)	2-5
2-5	LH_2 Propellant Tank Pressure Loading	2-8
2-6	Propellant Tank Structural Assembly	2-9
2-7	Propulsion Tank Structural Assembly	2-13
2-8	Engine Flyaway Module Structure	2-18
2-9	Spaceframe Structural Assembly	2-19
2-10	RNS Spaceframe Equipment Arrangement	2-23
2-11	Orbital Assembly Interface	2-31
2-12	Propellant Module Orbital Assembly Alignment	2-33
3-1	Propulsion System	3-2
3-2	Engine System	3-3
3-3	Engine Flyaway Control System	3-6
3-4	Engine Flyaway Control Module	3-11
3-5	Propellant Feed System Schematic	3-12
3-6	Orbital Refueling System	3-15
3-7	Forward Docking Cone/Propellant Transfer Assy	3-17
3-8	Purge System	3-20
3-9	NERVA Turbopump Net Suction Pressure Requirements for Design Saturation Pressure of 26 psia	3-23

Figure		Page
3-10	Pressurization System Schematic	3-25
3-11	Suggested Thruster Orientation for RNS Reaction Control System	3-28
3-12	Reaction Control System Schematic	3-31
4-1	Astrionics System	4-2
4-2	Integrated Avionics System	4-3
4-3	Data Management System	4-5
4-4	Communication System Schematic	4-10
4-5	Modal Control During Transit and Lunar Orbit Phases	4-11
4-6	Navigation, Guidance and Control System	4-16
4-7	Rendezvous and Docking Laser Radar	4-20
4-8	Power Profile – Lunar Mission	4-24
4-9	Solar Array Assembly and Installation	4-29
4-10	Electrical Power System Schematic	4-31
4-11	Typical Earth Orbit External Heating Rates – Equipment Module	4-34
4-12	Typical Lunar Orbit External Heating Rates – Equipment Module	4-34
4-13	Payload/Equipment Module Interface	4-35
4-14	Heat Rejection as a Function of Radiating Surface Temperature	4-36
4-15	Radiating Area Requirements	4-36
5-1	RNS Integration	5-2
5-2	Spaceframe Module Inboard Profile	5-3
5-3	RNS Spaceframe Equipment Arrangement–General Arrangement	5-7
5-4	RNS Spaceframe Equipment Arrangement – Bay Arrangement	5-9
5-5	RNS Spaceframe Equipment Arrangement – Installation Techniques	5-11
5-6	Spaceframe Module Inboard Profile (Module Interface Detail)	5-13
5-7	Propulsion Module	5-17
5-8	Propulsion Module Tank (Inboard Profile)	5-19
5-9	Engine System	5-21

Figure		Page
5-10	Propellant Module Inboard Profile	5-23
5-11	Orbital Assembly Interface	5-27
5-12	RNS Inboard Profile	5-29
5-13	RNS Schematic	5-31
5-14	RNS Vehicle Stations and Definitions	5-37
5-15	RNS/Payload Integration	5-47
5-16	Thermodynamic Model	5-51
5-17	Vehicle Orientation During Mission	5-52
5-18	Total Energy Into Propellant Tanks	5-56
5-19	Total Energy Into Propellant Tank No. 5	5-57
5-20	Total Energy Into Propellant Tank No. 6	5-58
5-21	Total Energy Into Propulsion Module Tank	5-59
5-22	Radiation Analysis Configuration	5-60
5-23	Flyaway Control Module Equipment	5-65
5-24	Radiation Effects for Typical Solar Cell	5-71
6-1	NERVA Engine Interface	6-3
6-2	NERVA Engine Interface (Interface Detail)	6-5
6-3	RNS/Launch Vehicle (Space Shuttle) Interface	6-9

BLANK

TABLES

Table		Page
1-1	RNS Weight Summary	1-5
2-1	Preliminary Space Shuttle Acceleration Loads	2-7
3-1	Engine Flyaway Module – Weight Statement	3-10
3-2	Propellant Feed System Weight Summary	3-21
3-3	Pressurization System Weights	3-26
3-4	Thruster Combinations for Vehicle Motion for Recommended System	3-29
3-5	Reaction Control System Weights	3-34
4-1	Data Management Subsystem Characteristics	4-9
4-2	RNS Communications Subsystem Characteristics	4-13
4-3	Navigation, Guidance and Control Characteristics	4-22
4-4	Power Consumption	4-23
4-5	Electrical Power and Distribution System Weights	4-27
4-6	Equipment Power and Temperature Limits	4-33
5-1	RNS Weight Statement	5-34
5-2	RNS Vehicle Elements Mass Properties	5-39
5-3	RNS Vehicle Elements Mass Properties, Partially Loaded With Propellant at Launch	5-40
5-4	RNS Vehicle Elements Mass Properties, Fully Loaded With Propellant at Launch	5-42
5-5	RNS Sequential Orbital Assembly Mass Properties With Partially Loaded Propellant Tanks	5-43
5-6	RNS Mass Properties Refilling With Partially Loaded Propellant Tanks	5-44
5-7	RNS Sequential Orbital Assembly Mass Properties With Fully Loaded Propellant Tanks	5-45
5-8	RNS Mass Properties, Nominal Lunar Mission With 78,451 lb P/L Out, 20,000 lb In. (Mission Phase Sequence of Events)	5-49
5-9	Vehicle Orientation Timeline	5-53

Table		Page
5-10	External Boundary Conditions	5-55
5-11	Gamma Ray Environment Below Propulsion Tank Module	5-61
5-12	Environment Above Propulsion Tank Module	5-62
5-13	Environment Above Propellant Tank Modules	5-63
5-14	Component Radiation Effects	5-67
6-1	RNS/NERVA Interface Requirements	6-7

Section 1
INTRODUCTION

1.1 REUSABLE NUCLEAR SHUTTLE CONCEPT

The Reusable Nuclear Shuttle is a completely self-contained space transportation system that can operate in an autonomous mode. It is propelled by the 75,000-pound thrust NERVA engine which uses hydrogen fuel. The basic system is of modular design, with all modules launched to Earth orbit by the Space Shuttle. The modules are then assembled in Earth orbit into a Reusable Nuclear Shuttle (RNS). The baseline vehicle, shown in Fig. 1-1, consists of a space frame module, six propellant modules, and one propulsion module. The total propellant load of the vehicle is 269,427 pounds of hydrogen. Each propellant module has a capacity of 38,676 pounds and the propulsion module has a capacity of 37,371 pounds of hydrogen. A simplified weight statement of the RNS is shown in Table 1-1. The weights are allocated both by module and by category. The primary mission for the RNS is the transport of payloads between Earth orbit and lunar orbit. Figure 1-2 indicates the basic performance capability for this mission operating in two different mission modes. The data shows the capability for maximum payload to the moon, maximum payload returned from the moon, and all combinations in between. Mission requirements for these and other missions in the study are defined in Section 2.2 of Vol II, Part A.

1.2 REUSABLE NUCLEAR SHUTTLE DEFINITION

The RNS is defined according to the Work Breakdown Structure (WBS) specified in the study guidelines and illustrated in Fig. 1-3. The RNS is defined at the third level and the modules of the RNS which are the key identification points are at the fourth level. The subsystems are carried in parallel for each module and are at the fifth level. All of the structural systems are defined in Section 2 of this volume; propulsion systems in Section 3; and astrionics systems in Section 4. A synthesis of the completely defined modules and RNS is made in Section 5. This section includes inboard profiles of the modules and vehicle, a schematic of the vehicle, and the mass properties of the vehicle. It also includes the thermal and radiation environment of the RNS. A drawing tree, shown in Fig. 1-4, was developed to provide traceability to the major drawings of the RNS according to the WBS approach. In a similar manner, a schematic tree was developed as shown in Fig. 1-5. The RNS interfaces are defined in Section 6.

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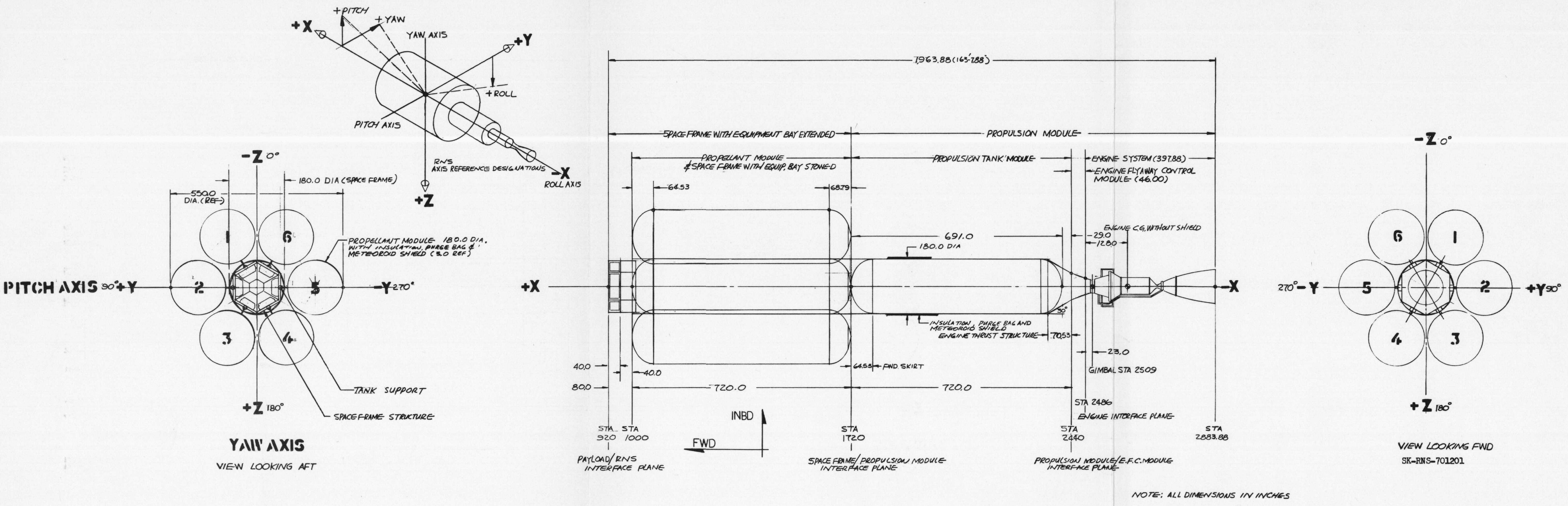

Fig. 1-1 RNS Baseline Configuration

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Table 1-1

RNS WEIGHT SUMMARY

Code	Description	Module Weight, Lb			Vehicle Weight Lb*
		Propellant	Propulsion	Control & Assy/Struct	
2.00	Structure	3,370	4,118	2,293	26,631
3.00	Meteoroid/Thermal Protection	1,141	2,617	–	9,463
4.00	Docking/Tank Support	87.5	122	3,917	3,664
5.00	Main Propulsion	242.5	37,038	667	39,160
6.00	Auxiliary Propulsion		113	548	661
7.00	Astrionics	40	45	1,533	1,818
8.00	Safety/Ordnance System	–	–	–	TBD
9.00	Contingency 5% (Exclude NERVA Engine and Disc Shield)	244	411	403	2,278
	Total Inert Weight	5,125	44,464	8,461	83,675

*6 Propellant Modules
1 Propulsion Module
1 Space Frame

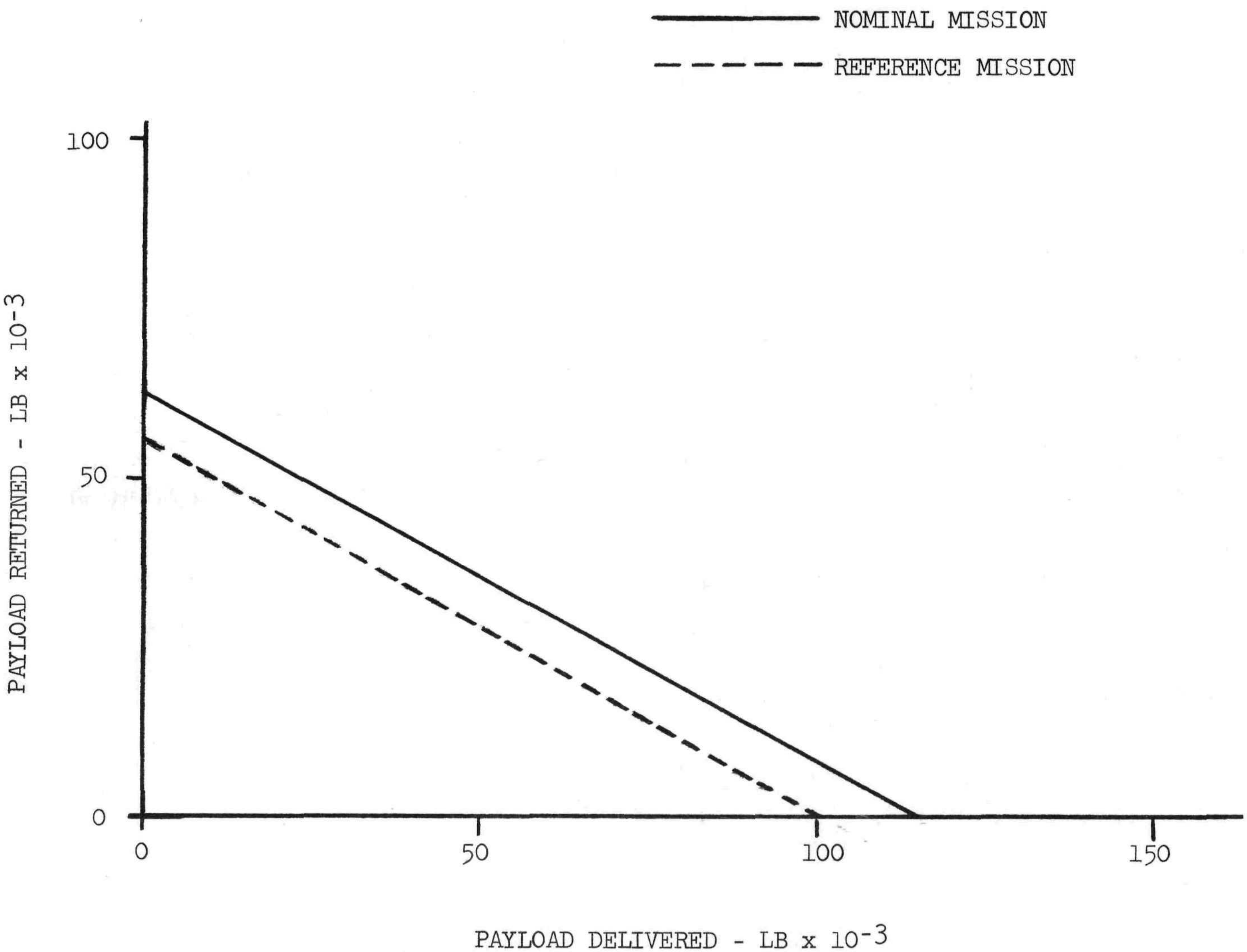

Fig. 1-2 RNS Lunar Mission Performance Capability

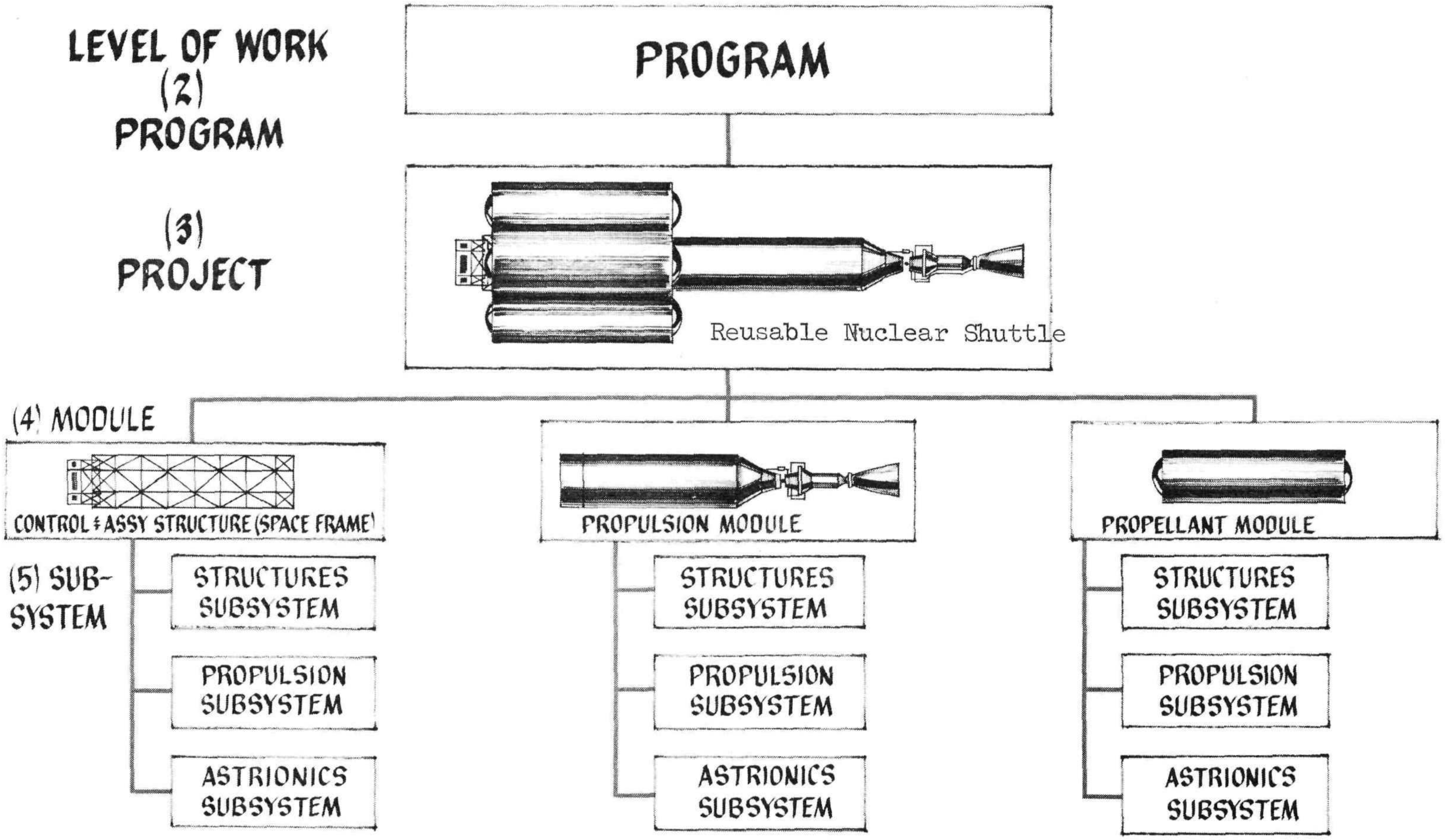

Fig. 1-3 Work Breakdown Structure

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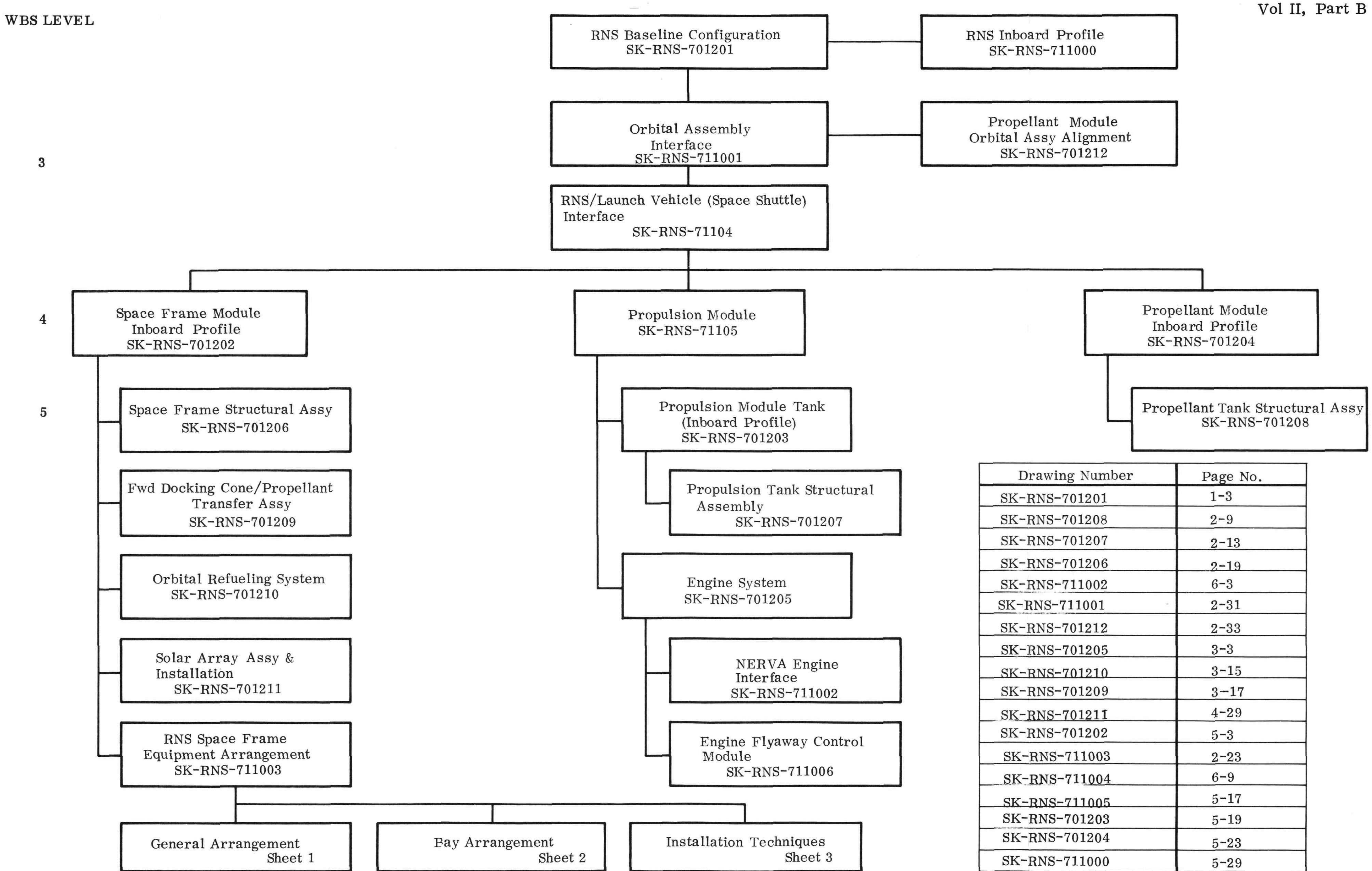

Drawing Number	Page No.
SK-RNS-701201	1-3
SK-RNS-701208	2-9
SK-RNS-701207	2-13
SK-RNS-701206	2-19
SK-RNS-711002	6-3
SK-RNS-711001	2-31
SK-RNS-701212	2-33
SK-RNS-701205	3-3
SK-RNS-701210	3-15
SK-RNS-701209	3-17
SK-RNS-701211	4-29
SK-RNS-701202	5-3
SK-RNS-711003	2-23
SK-RNS-711004	6-9
SK-RNS-711005	5-17
SK-RNS-701203	5-19
SK-RNS-701204	5-23
SK-RNS-711000	5-29
SK-RNS-711006	3-11

Fig. 1-4 Engineering Drawing Tree

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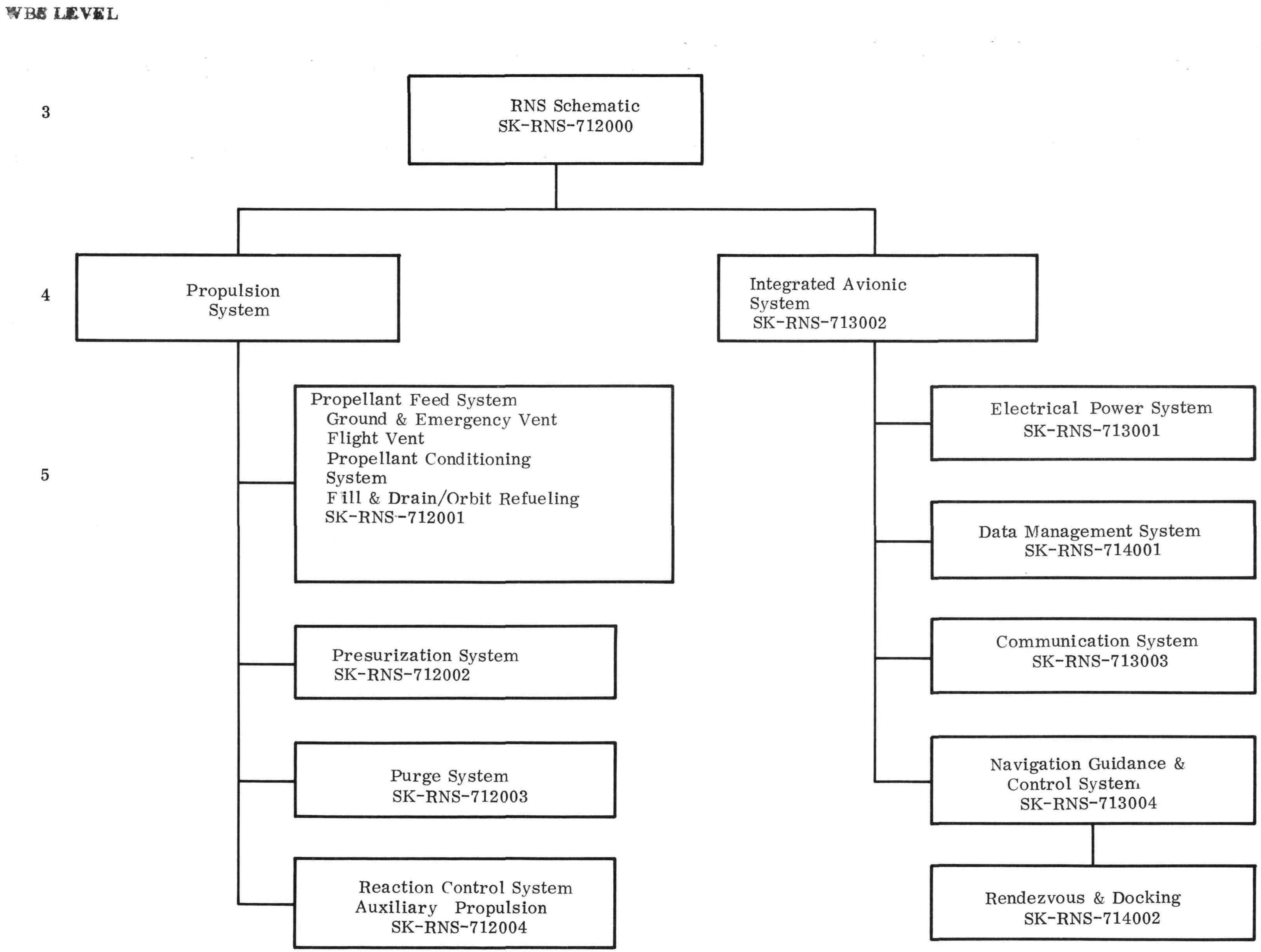

Schematic No.	Page No.
SK-RNS-712001	3-12
SK-RNS-712003	3-20
SK-RNS-712002	3-25
SK-RNS-713004	4-16
SK-RNS-713002	4-3
SK-RNS-712004	3-31
SK-RNS-714001	4-5
SK-RNS-713003	4-10
SK-RNS-713001	4-31
SK-RNS-712000	5-31
SK-RNS-714002	4-20

NOTE: Propulsion System is included in RNS Schematic SK-RNS-712000

Fig. 1-5 System Schematic Tree

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Section 2
STRUCTURAL SYSTEM

The basic structural system of the RNS includes the structural elements of the propellant module, propulsion module, control and assembly module (space frame) as shown in Fig. 2-1. This section provides a general description of the vehicle structural arrangement.

The propellant module structure shown in Fig. 2-2 consists of the propellant tank assembly including forward and aft skirts, and slosh baffles and screens. The equipment support structure is incorporated into the forward and aft skirts. Docking and tank supports are incorporated for attaching the tank to the space frame. The propellant tanks are completely enclosed with insulation. A meteoroid shield completely encloses the propellant tanks except for the 120-degree segment on the cylindrical section which faces inboard and is self-shielding.

The propulsion module structure shown in Fig. 2-3 consists of the propulsion tank assembly including forward and aft skirts, tunnel and fairing, and slosh baffle and screen. Equipment support structure is incorporated into the forward and aft skirts. The propulsion tank is completely enclosed with insulation and a meteoroid shield. The propulsion module also includes an engine thrust structure, engine flyaway module structure, and forward docking and tank support structure to attach the module to the space frame.

The control and assembly module shown in Fig. 2-4 consists of the spaceframe assembly, equipment module assembly, docking structure, equipment support structure, and vehicle assembly support equipment.

2.1 PROPELLANT TANK ASSEMBLY

Propellant tanks have been designed to carry a full propellant load both for the launch environment within the Space Shuttle and for all operational phases. For the re-entry

PROPELLANT MODULE

- Propellant Tank Assy
- Forward Skirt
- Aft Skirt
- Exterior Finish & Sealer
- Propellant Containment System
- Equipment Support
- Insulation
- Meteoroid Protection
- Tank Support

PROPULSION MODULE

- Propulsion Tank Assy
- Engine Thrust Structure
- Fwd Skirt
- Aft Skirt
- Tunnel & Fairing
- Exterior Finish & Sealer
- Propellant Containment System
- Equipment Support
- Engine Flyaway Module Structure
- Insulation
- Meteoroid Protection
- Tank Support

CONTROL & ASSY STRUCTURE (SPACE FRAME)

- Equipment Support
- Equipment Module
- Space Frame Assy
- Docking & Tank Support
- Vehicle Assembly Support Equipment

Fig. 2-1 Structural System

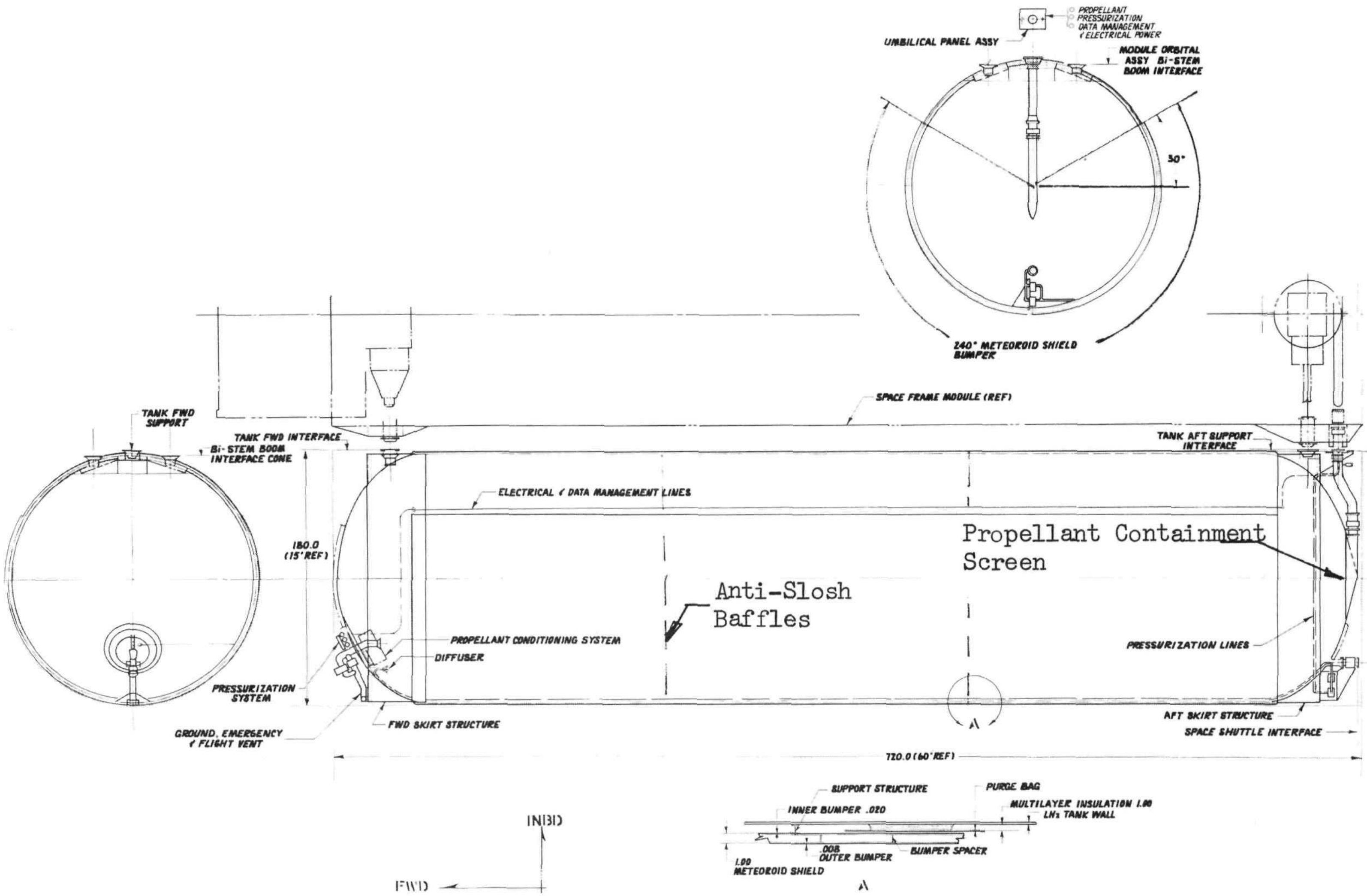

Fig. 2-2 Propellant Module Structure

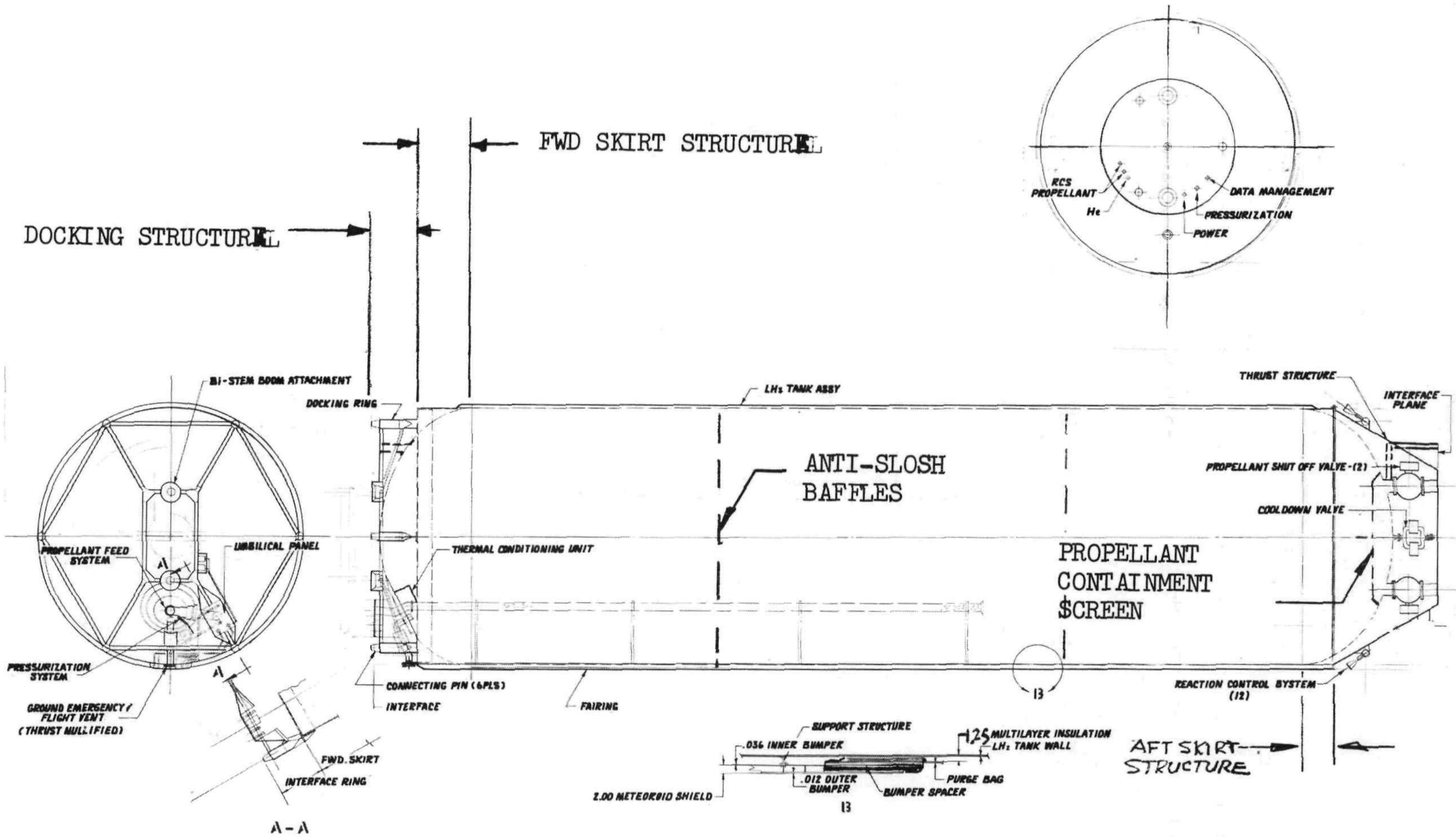

Fig. 2-3 Propulsion Module Structure

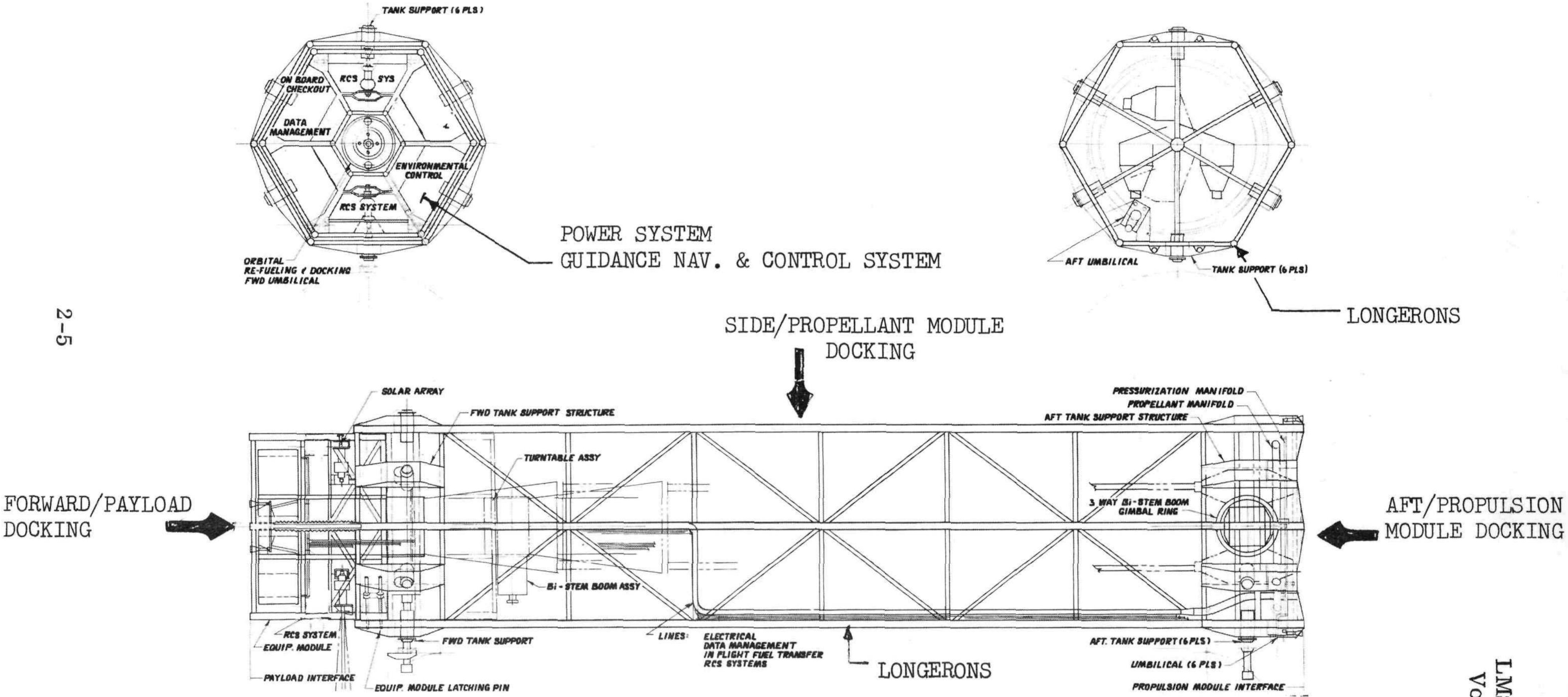

Fig. 2-4 Control and Assembly Structure (Spaceframe)

and landing/taxiing/braking loads, designs are for empty tanks since the hydrogen can be transferred to the Space Shuttle propellant tanks. The launch loads are based on the Space Shuttle loading conditions given in Table 2-1 (see Ref 2-1).

The launch loads dictate the minimum pressures and consequently the minimum weight of the tank. The operating pressure during the mission increases and was optimized at 25.4 psi. This design pressure was used based on mission operation and engine burn requirements. The effective pressures acting on the tank are shown in Fig. 2-5. The minimum tank design pressure of 20 psi is required during the Space Shuttle at maximum q-alpha region due to lateral loads. The fore and aft axial loads impose an effective pressure of 21 and 26 psi fore and aft, respectively. The acceleration during the mission imposes a pressure of 27.4 psi on the aft bulkhead. Aluminum alloy, 2021-T81, was selected as the tank material. A 42,600 psi allowable at $-190^{\circ}F$ was used for tank operating stress in the membranes based on a fracture mechanics approach.

2.1.1 Propellant Module

The propellant tank structural assembly is shown in Fig. 2-6. The tank size is based on the 15 by 60-ft Space Shuttle cargo bay and allows for a 3-inch radial and axial clearance around the tank for insulation and meteoroid shielding. The tank is a cylinder with elliptical $\sqrt{2}$:1 bulkheads. A one-piece bulkhead construction is utilized. The bulkheads are girth-welded to the cylinder joints with Y-rings. The forward bulkhead incorporates an access cover including propellant vent system, pressurization system, and instrumentation connections penetrations. The aft bulkhead incorporates a 75 degree propellant sump. There are two penetrations: (1) a ground propellant fill, drain, and tank purge line which interfaces with the Space Shuttle interface; and (2) a propellant orbital feed system line. The tank has a volume of 9,190 ft^3, which allows a propellant capacity of 38,676 lb of LH_2 with five percent ullage. 2021-T81 aluminum alloy was selected as the tank material. The total weight of the tank structural assembly (2.01)* is 2,904 lb, exterior finish (2.06)* is 25 lb. and equipment support structure (2.08)* is 10 lb.

*The numbers in parenthesis represent the RNS weight statement coding system.

Table 2-1

PRELIMINARY SPACE SHUTTLE ACCELERATION LOADS

Mission Phase/Event	Longitudinal (X) Steady (g)	Longitudinal (X) Dynamic (g)	Lateral (Y/Z) Steady (g)	Lateral (Y/Z) Dynamic (g)
Launch Release Transient (Within 2 seconds of release)	+ 1.5	± 2.0	-	± 2.0 (Y & Z)
Lift-off + 5 seconds	+ 1.5	± 0.25	-	± 0.25 (Y & Z)
Max Q-Alpha Region (35% - 55% Booster Burn)	+ 2.0	± 0.40	- 0.75(Z)	± 1.5 (Y & Z)
Maximum Acceleration (80% - 100% Booster Burn)	+ 3.0	± 0.25	- 0.5 (Z)	± 0.25 (Y & Z)
Cut-off/Separation (Within 2 seconds of Booster Cut-off)	-	± 3.0	-	± 2.0 (Y & Z)
Separation + 5 seconds	+ 1.5	± 0.25	-	± 0.1 (Y & Z)
Maximum Acceleration (60% - 100% Orbiter Burn)	+ 3.0	± 0.25	-	± 0.1 (Y & Z)
Re-entry	- 1.0	-	- 4.0 (Z)	± 0.1 (Y & Z)
Landing/Taxiing/Braking	- 1.0	-	- 2.0 (Z) 0.5 (Y)	± 2.0 (Z) ± 0.5 (Y)

(Based on Payload at c.g. location. Sign Conventions Based on Orbitter Reference Data, Level II Requirements)

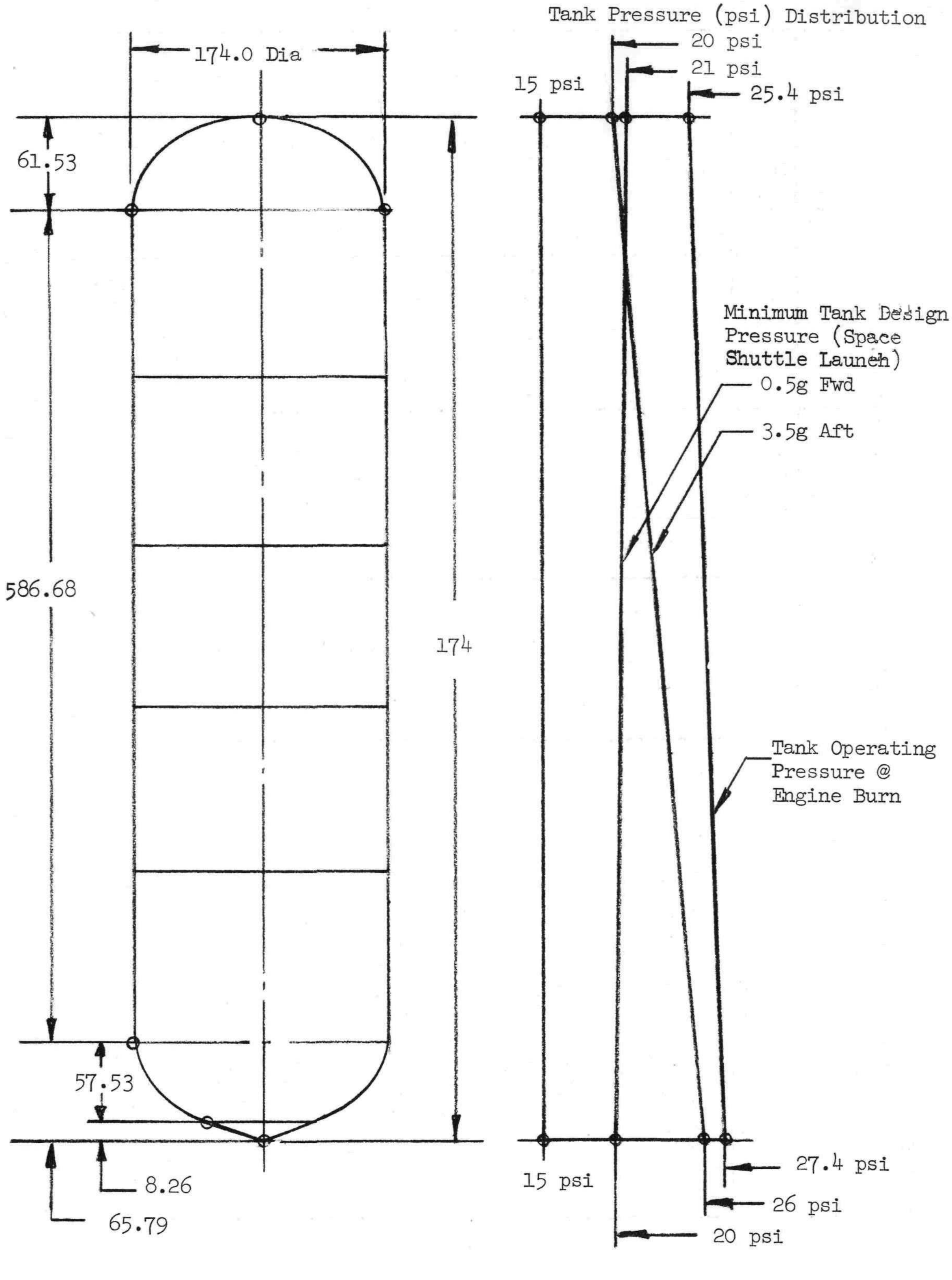

Fig. 2-5 LH_2 Propellant Tank Pressure Loading

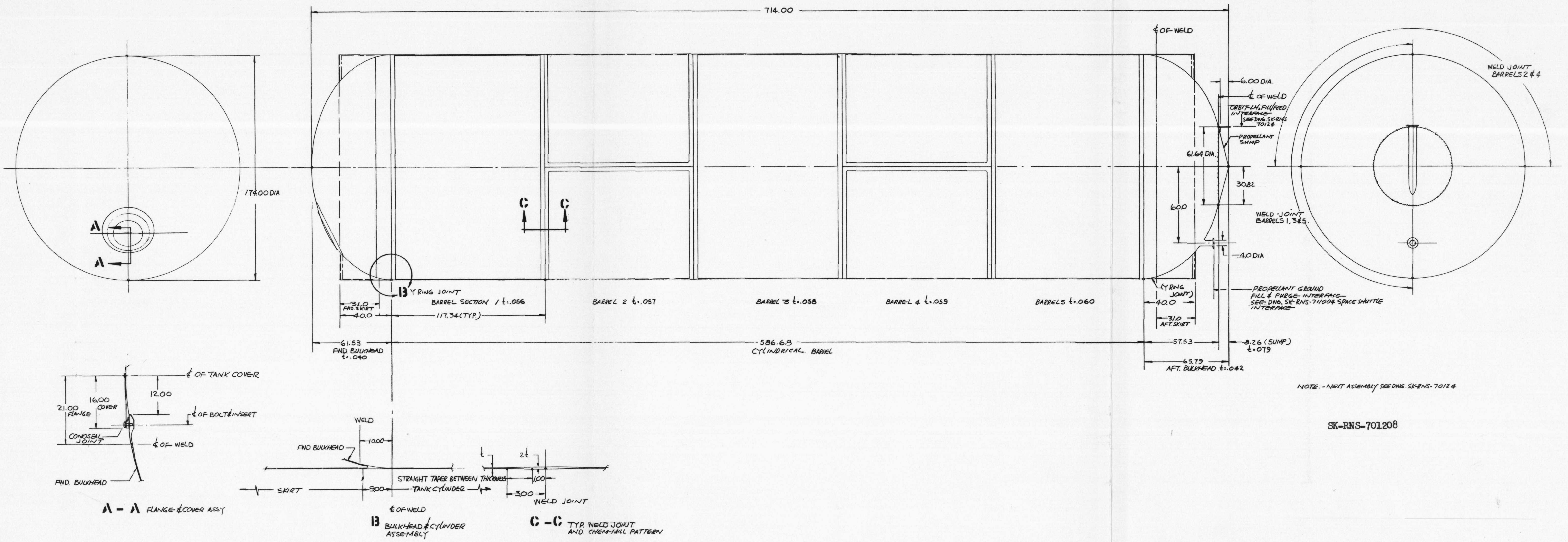

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Fig. 2-6 Propellant Tank Structural Assembly

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2.1.2 Propulsion Module

The propulsion tank structural assembly shown in Fig. 2-7 is sized for the 15 by 60-ft Space Shuttle cargo bay and allows for a 3-in. radial clearance around the tank for insulation and meteoroid shielding and a 29-in. axial clearance at the aft end for the propellant feed system installation and the engine thrust structure interface. The tank is a cylinder with elliptical $\sqrt{2}:1$ bulkheads. It consists of one-piece bulkheads which are girth-welded to the cylinder joints with a Y-ring. The forward bulkhead incorporates two penetrations: (1) an access cover with pressurization feed, in-flight vented and instrumentation connections; and (2) a 4-in. dia. ground propellant fill, drain, and tank purge line that interfaces with the Space Shuttle. The aft bulkhead incorporates three penetrations – the two propellant feedlines and an engine cooldown line. A ground emergency and purge vent line tees into one feed line. The tank has a volume of 8,880 ft^3 which allows a propellant capacity of 37,371 lb of LH_2 with five percent ullage.

The total weight of the tank structural assembly (2.01) is 2,818 lb, exterior finish (2.06) is 25 lb, and equipment support structure (2.08) is 15 lb.

2.2 ENGINE THRUST STRUCTURE

The engine thrust structure provides for the attachment of the NERVA engine system and also transmits all of the engine thrust loads to the propulsion aft skirt. The thrust structure is a 30-degree conical shell stiffened with external longitudinal corrugations and internal angle section rings. Aluminum alloy 7075-T6 is used for the entire structure. A forward closure ring is used for attaching the thrust structure to the propulsion tank module and also as an end closure ring for mounting the NERVA engine in orbital assembly.

For the analysis of the engine thrust structure a gimbal angle of 5 degrees was assumed. The cone is analyzed for axial thrust of 1.4 by 75,000 lb and a bending moment due to the side load of 1.4 by 6,540 lb (shear load). The weight of the engine thrust structure (2.02) is 501 lb.

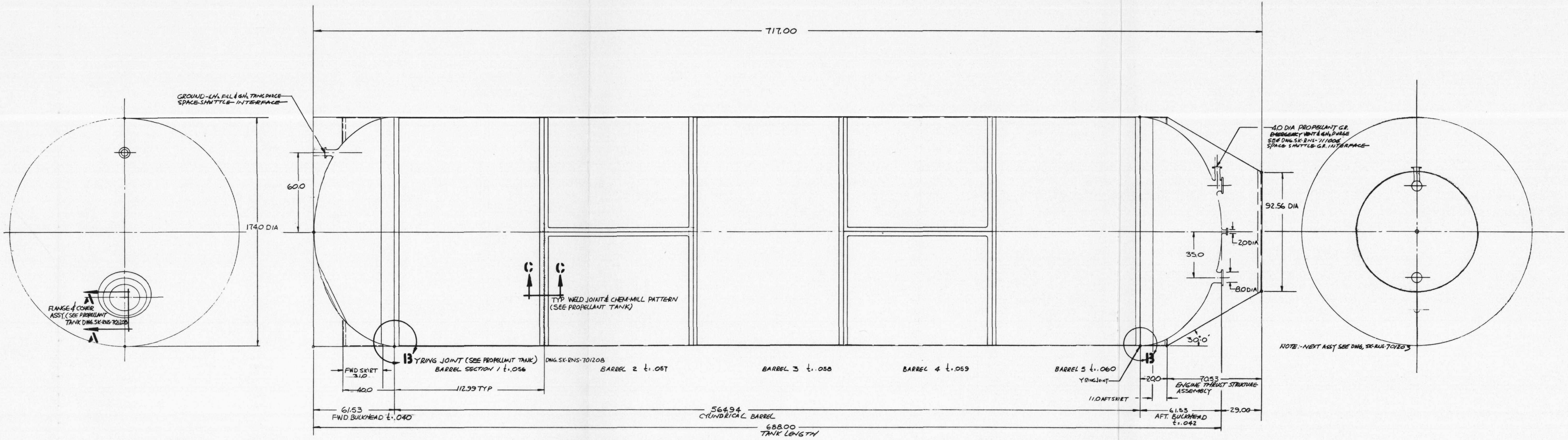

Fig. 2-7 Propulsion Tank Structural Assembly

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2.3 FORWARD AND AFT SKIRT

The forward and aft skirts are cylindrical shells stiffened with external longitudinal corrugations and internal rings providing an efficient unpressurized load carrying structure during ground launch. They are constructed of 7075-T6 aluminum alloy.

2.3.1 Propellant Module

The forward skirt structure is 174 inches in diameter by 31 inches long and is located at the forward end of the propellant tank and joined to the tank forward Y-ring at the common juncture of the cylinder section and tank forward bulkhead. The forward skirt provides the forward support of the propellant tank tiedown to the space frame, the support mounting of the RNS assembly booms, and support with the Space Shuttle during launch. A ground umbilical panel is located in the forward end of the skirt consisting of ground and emergency vent, flight vent, and tank purge interface. The aft skirt structure is similar to the forward skirt structure and is 174 inches in diameter x 31 inches long. It is located at the aft end of the propellant tank and is joined to the tank aft Y-ring at the common juncture between the cylindrical section and tank aft bulkhead. The skirt provides the aft tank support tiedown to the spaceframe, support mounting of the RNS assembly booms, and with the Space Shuttle during launch. An umbilical panel is located in the aft end of the skirt, consisting of electrical, data management, pressurization and LH_2 feed connections from the spaceframe. The forward (2.03) and aft (2.04) skirts weigh 203 lb each.

2.3.2 Propulsion Module

The forward skirt structure is 174 inches in diameter by 31 inches long, located at the forward end of the tank and is joined to the tank forward Y-ring at the common juncture of the cylindrical section and tank forward bulkhead. The forward skirt provides for the mounting of the propulsion module spaceframe module docking interface ring, and mounting of the external fairing for plumbing and electrical line outlets. The critical loading condition is at engine startup. The thrust vector may be offset from the c.g., which would induce vehicle pitching about the c.g. A value of 1 1/2 degrees was used for this offset angle to derive pitching inertia loads in the critical sections, and the skirt is designed so that the stiffeners are capable of carrying the design load with the skin in the post-buckled state.

The aft skirt is 174.0 inches in diameter by 11.0 inches long and joined to the tank at the tank cylindrical shell and the tank aft bulkhead Y-ring juncture. The aft skirt structure assembly provides the mounting surface of the engine thrust structure, and also transmits all the engine thrust loads from thrust structure into the tank shell. The forward skirt (2.03) weighs 215 lb and the aft skirt (2.04) weighs 114 lb.

2.4 EXTERNAL FAIRING

The external fairing provides an enclosure for the supply line of the RCS system, data management bus, and power system bus from the spaceframe module to the aft end of the propulsion system. This also contains the engine GH_2 pressurization system supply to all propellant tanks.

The fairing runs the full length of the propulsion module carrying no loads and is fabricated from fiberglass and is attached to the forward and aft skirt structure with local fiberglass standoff supports on the tank wall. It is fully insulated to minimize heat leaks and provides meteoroid protection for the lines. The fairing (2.05) weighs 30 lb.

2.5 PROPELLANT CONTAINMENT SYSTEM

The propellant sump serves as propellant containment and settling device. There will be a scavenging requirement for the propellant tank to ensure suppression of gas ingestion due to vortexing, slosh, and surface dip as depletion is approached. The device chosen for the control of scavenging is a fine-mesh metal stainless steel wire screen formed into a downward-dished, shallow cone of about one-third the tank diameter. This device is located forward of the propellant outlet. The screen is supported by an upward-dished, shallow, perforated, sheet metal cone of the same diameter as the screen. Two circular aluminum rings are located circumfirentially around the tank inner wall at the 1/3 and 2/3 length to minimum propellant sloshing. The weight of the system (2.07) is 25 lb in all propellant and propulsion module tanks.

2.6 ENGINE FLYAWAY MODULE STRUCTURE

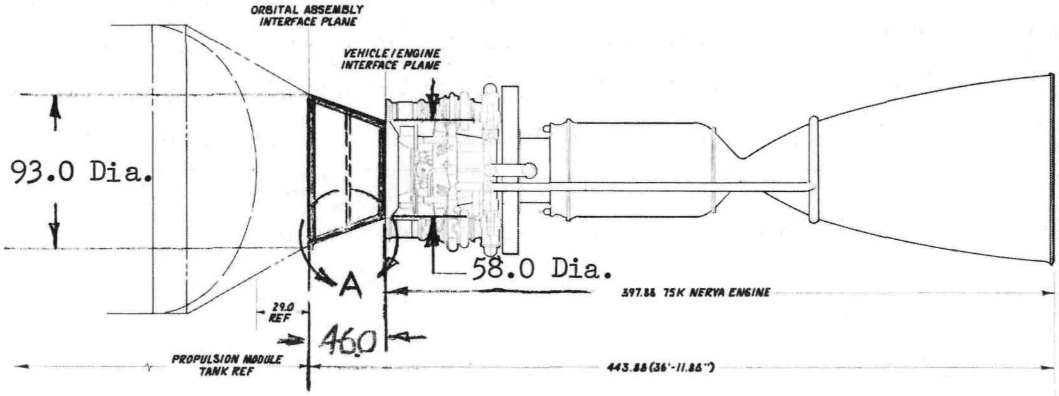
The engine flyaway module structure is shown in Fig. 2-8. It houses all the engine flyaway equipment, and is bolted to the interface of the NERVA engine. The structure of the flyaway module is a conical shell with skin and internal rings of 7075-T6 aluminum alloy. It has a major diameter of 93 in. and minor diameter of 58 in. by 46 in. long. A forward closure ring is used for orbital interface assembly with the RNS propulsion tank thrust structure. The aft closure ring is bolted in 12 places at 56 in. diameter to the NERVA engine upper thrust structure interface. The weight of the structure (2.11) is 375 lb.

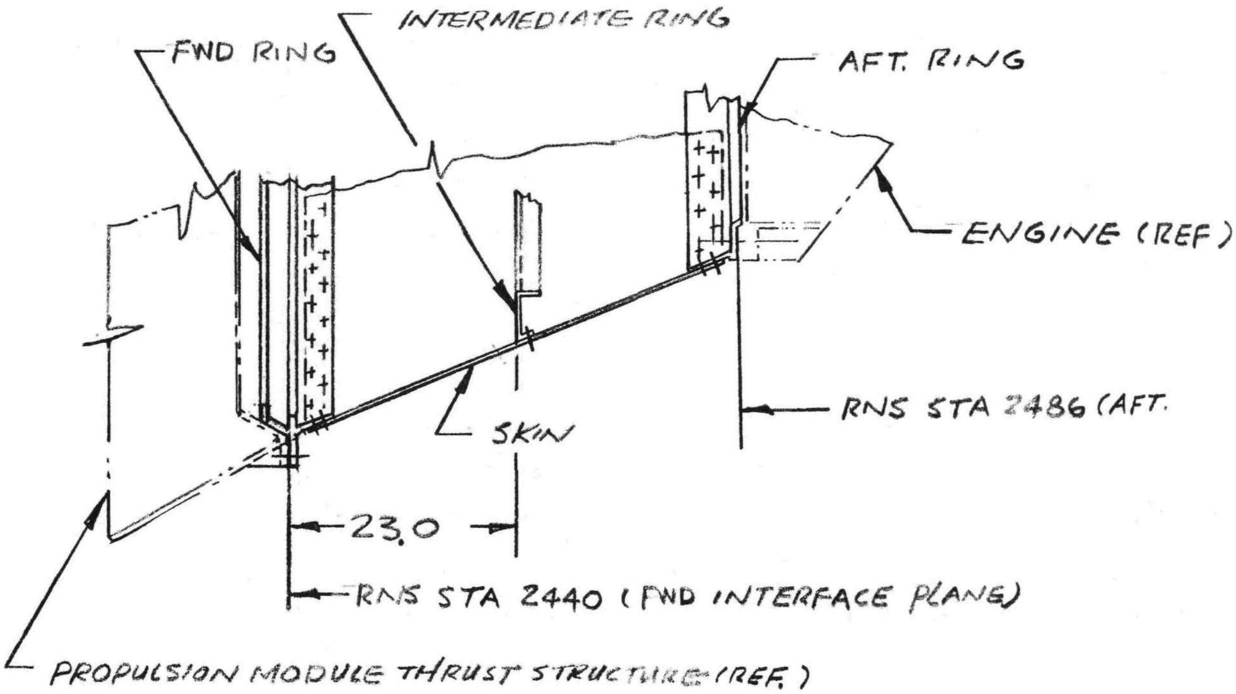
2.7 SPACEFRAME ASSEMBLY

The spaceframe structural assembly is shown in Fig. 2-9. The frame size is based on the 15 by 60 ft Space Shuttle cargo bay. The spaceframe is launched within this cargo bay and then extended to 67 feet in orbit. The hexagonal core of the spaceframe forms the basic load-carrying structure of the RNS. This spaceframe serves as a thrust structure, propellant module support structure, forward docking structure, aft propulsion module docking structure, and vehicle alignment system. An equipment module is located in the forward end of the spaceframe and is stowed inside the spaceframe during launch in the Space Shuttle. The equipment bay is extended forward with the vehicle assembly support system in orbit.

The critical design condition of the structure is compression buckling. The individual sections being considered are welded, tubular, truss structures of aluminum alloy, 6061-T6. This material was chosen for the spaceframe because of the ease and reliability of its weldments. However, the particular alloy and temper are not of great significance, because of the relatively high slenderness ratio of most of the members; i.e., they are designed by column buckling rather than material strength.

The hexagonal equipment module in the forward end of the spaceframe provides six equipment bay areas with each bay having a clear volume of 28 feet^3. This structure also serves to support the solar array system, reaction control system and vehicle support equipment.





DETAIL A ENGINE FLYAWAY MODULE STRUCTURE

Fig. 2-8 Engine Flyaway Module Structure

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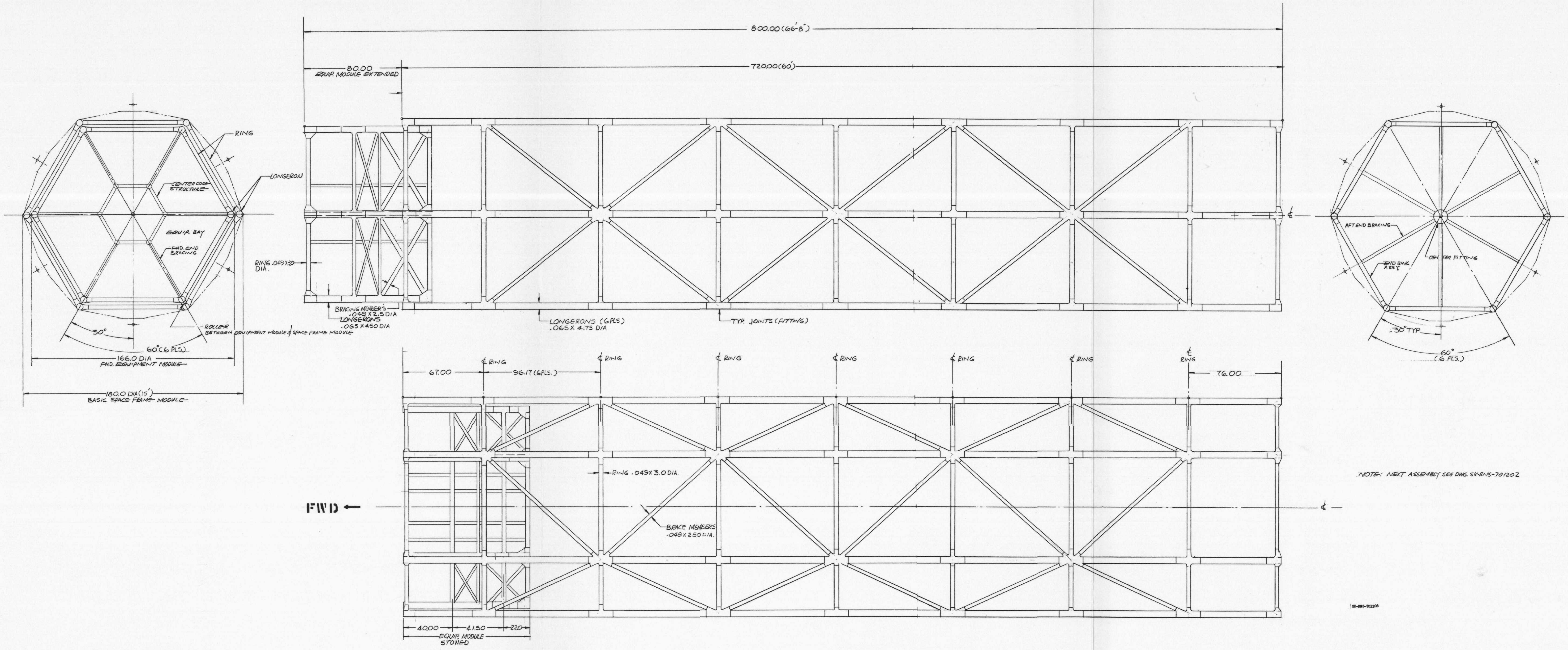

Fig. 2-9 Space Frame Structural Assembly

Figure 2-9 also shows the structural member sizes of the spaceframe. The overall weight of the basic structure (2.10) is 1155 pounds and equipment module (2.09) is 1,078 pounds.

Typical equipment support structure in the spaceframe is shown in Figure 2-10. The equipment support structure is attached to the module with quick release and lock mechanism supported at the top and the bottom with tapered channels. Hinged panel doors or hinged trays provide accessibility to the equipment within the structure. The weight allocation for equipment support (2.08) is 60 lb.

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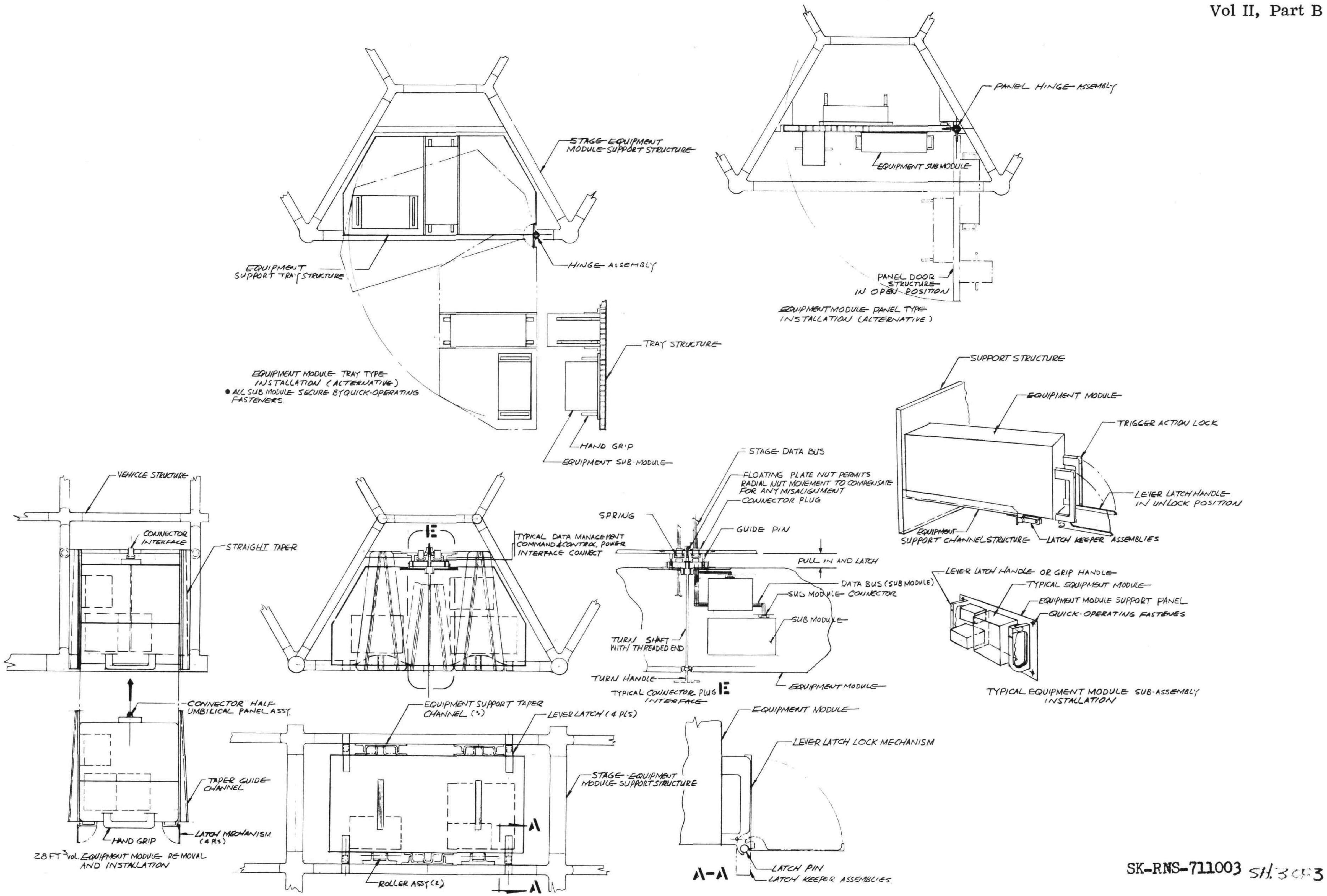

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Fig. 2-10 RNS Spaceframe Equipment Arrangement-Installation Techniques

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2.8 INSULATION

All propellant-carrying tanks are insulated with double aluminized mylar multilayer insulation using tissuglas spacers. The insulation thickness for each tank is uniquely defined based on the mission profile and the tank's specific location on the vehicle. The determination of the specific thickness for each tank is based on a computer optimization which incorporates tank surface temperature, energy injected into the tank by hot pressurization gas, tank operating pressure, and tank depletion schedule. An energy balance is performed prior to and after each burn and the computer program optimizes payload. The insulation thickness is one of the design variables defined. Because of this refined analysis the conductivity of the insulation is a variable throughout the mission since it varies with temperature. The following relationship is used to determine the conductivity.

$$K = C\left[1.83 \times 10^{-12}\,\bar{N}^{2} T_m + \frac{1.7\sigma\left(T_h^{2} + T_c^{2}\right)\left(T_h - T_c\right)t}{N-1 \quad \left(\frac{2}{\epsilon} - 1\right)}\right]$$

where

$C = 5$ (degradation factor)

$E = 0.036$ (emissivity)

$\sigma = 1.73 \times 10^{-8}$ (Stefan-Boltzman Constant)

$N = 100$ (layers per inch)

$$T_m = \frac{\left(T_h + T_c\right)}{2}$$

and

T_h = hot side temperature, oR

T_c = cold side temperature, oR

t = Insulation thickness, feet

The insulation density used was 2.3 lb/ft^3.

2.8.1 Propellant Module

The insulation thickness on propellant tanks 1-4 is 0.5 inch, on tank 5 it is 0.75 inch and on tank 6 it is 1 inch. The insulation is made up of blankets which are attached to the tank with valcro fasteners. Insulation is also laid up around all penetrations and along the propellant lines. Insulation weight (3.01) on tanks 1-4 is 308 lb, on 5 it is 440 lb, and on 6 it is 572 lb.

2.8.2 Propulsion Module

The insulation thickness on the propulsion module is 1.25 inches for the reference lunar mission. The insulation weight (3.01) on the propulsion module is 730 pounds.

2.9 METEOROID SHIELDING

A dual wall aluminum meteoroid bumper is used for all applications. These shields are separated by fiberglas standoffs, and the bumper is supported from the tank forward Y-ring with fiberglas fasteners and standoffs are used sparingly for axial load distribution. The meteoroid shielding provides for a $P_o = 0.995$ for 45 days with no penetration of the tank. The surface of the outer bumper is designed to provide an α/ϵ of $.08/_{.8}$. Because of the primary sun orientation of the vehicle the surface characteristics are not very significant.

2.9.1 Propellant Module

The propellant module bumper consists of a 0.0080-inch outer shield and a 0.0020-inch inner shield spaced one inch apart. The bumper covers the bulkheads and the external 240 degrees of the cylindrical section of each propellant module. The total meteoroid protection system (3.02) weights 767 lb per module.

2.9.2 Propulsion Module

The propulsion module bumper consists of a 0.012-inch thick aluminum outer shield and a 0.036-inch thick aluminum inner shield space two inches apart. Since the two-inch bumper spacing external to a 1.25-inch insulation thickness required for an optimized design would violate the 3-inch clearance in the Space Shuttle bay, the insulation is split and some is between the meteoroid bumper shields which also enhances their effectivity. The meteoroid protection system (3.02) weighs 1,888 lb. It is significantly heavier than for the multiple propellant modules to provide an equal probability of no system failure.

2.10 DOCKING AND TANK SUPPORT

Docking between modules and tank support to the spaceframe is accomplished with specific structural elements and mechanisms providing a maximum of commonalty.

2.10.1 Propellant Module

The propellant module provides two female docking adapter cones which are located in the forward and aft skirts. They provide the attachments for the tank support. The structural tiedown is accomplished by male 10-inch diameter conical pins located on the spaceframe, mating with female adapters on the module. The aft connection is rigid and represents the primary tiedown point. The forward connection provides axial motion for propellant tank thermal expansion compensation. The structural joint is completed by a motor-driven Marman clamp on each of the two connections. The tank support adapter is constructed from 7075-T6 aluminum. The weight of all the docking and tank support structure (4.03) is 87.5 lb.

2.10.2 Propulsion Module

The hexagonal docking ring forms the basic load-carrying forward docking structure and provides the interconnection of the spaceframe module with the propulsion module tank. The critical design condition of the structure is compression buckling. The individual structural sections consist of welded tubular 6061-T6 aluminum alloy elements. The forward end of the interface provides assembly boom attachments and an umbilical connection between the six spaceframe attachment points that engage the six longeron members of the spaceframe. Final alignment and structural joining are achieved by motor-driven bolts mounted in the spaceframe longerons aft end. In the docking structure are six longeron members attached to the forward skirt structure with six machined fittings with local doubler reinforcements to provide equal engine thrust load transfer. The weight of the tank support system (4.03) is 122 lb.

2.10.3 Spaceframe

The forward and aft end of the spaceframe structure provides for payload docking. The spaceframe also provides forward and aft conical pins for propellant module docking and attachment at all six sides. Docking of the payload and propulsion modules is accomplished with six pins that engage the six longeron members of the spaceframe. Rigid structural joints are procured with motor-driven bolts mounted in the spaceframe longerons. Attachment of the propellant modules is by two male 10-inch diameter conical pins located forward and aft on the spaceframe and supported by the box beam fitting structure. These pins mate with female adapters located on the propellant module tank skirts. The aft connection represents the primary tiedown point. The forward connection is installed in the spaceframe to provide axial motion for thermal expansion compensation. The structural joint consists of a motor-driven Marman clamp on each of the connectors. The weight of the forward docking system (4.01) is 54 lb, the aft docking system (4.02) is 105 lb and the propellant tank supports (4.03) is 443 lb.

2.11 VEHICLE ASSEMBLY SUPPORT EQUIPMENT

The vehicle assembly support equipment consists of four bi-stem booms and their assemblies. Two are mounted on a turntable in the forward end of the spaceframe structure and provide 360-deg rotational capability. The other two booms are mounted in the aft end of the spaceframe on an a-ring gimbal device providing forward, aft, and lateral positioning capability. The bi-stem boom system provides for lateral propellant module displacement. The turntable and ring gimbal device are rotated by an electrically operated reduction gear motor. The aft set of the bi-stem boom system rotates forward and aft and provides deployment of the equipment module and propulsion module upon release from the Space Shuttle. The ring gimbal mechanism is rotated by an electrically operated reduction gear motor and internal ring gear and ring channel. Figure 2-11 shows the bi-stem boom system in operation and Fig. 2-12 shows the bi-stem details. The weight of the vehicle assembly support equipment (4.04) is 1498 lb.

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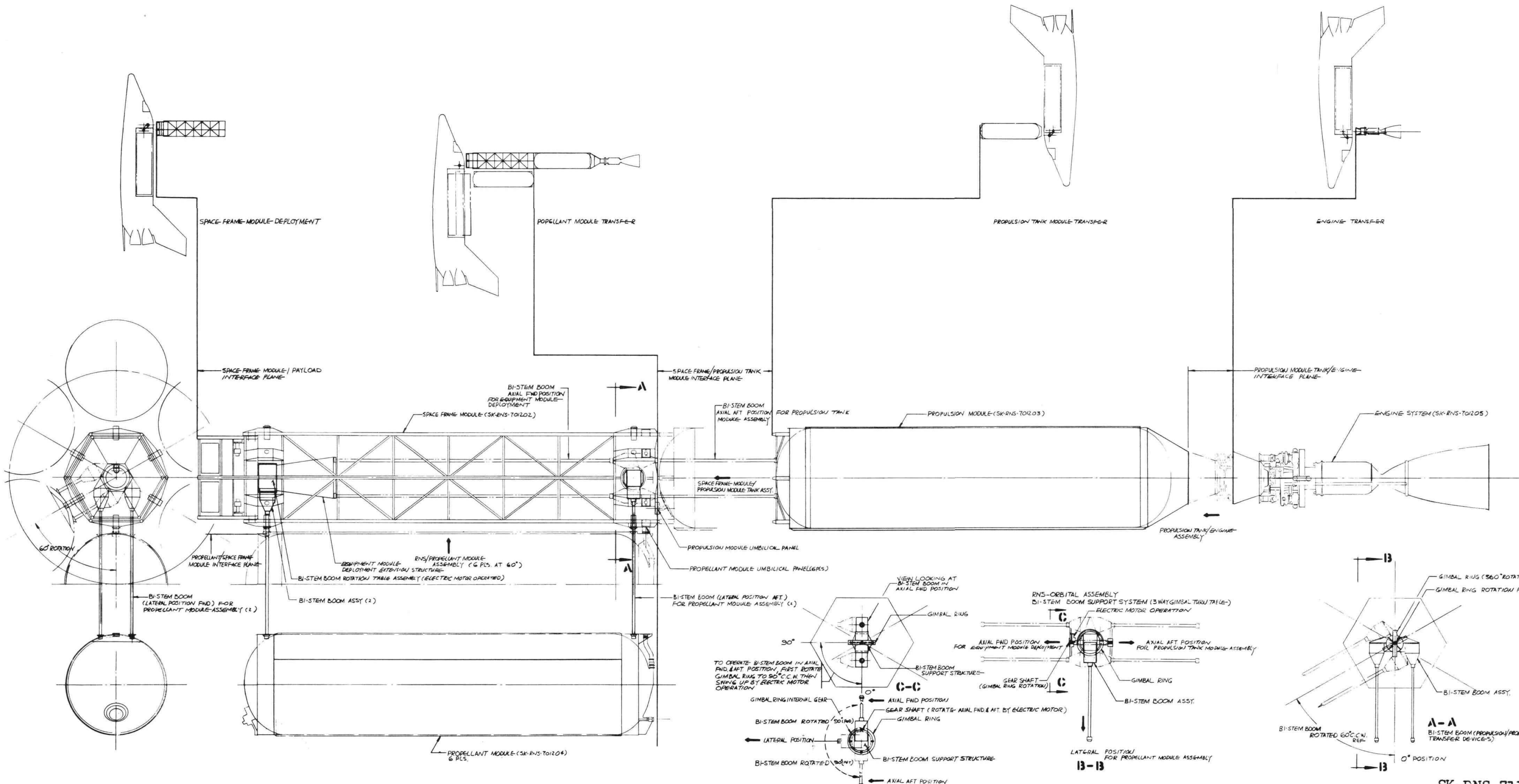

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Fig. 2-11 Orbital Assembly Interface

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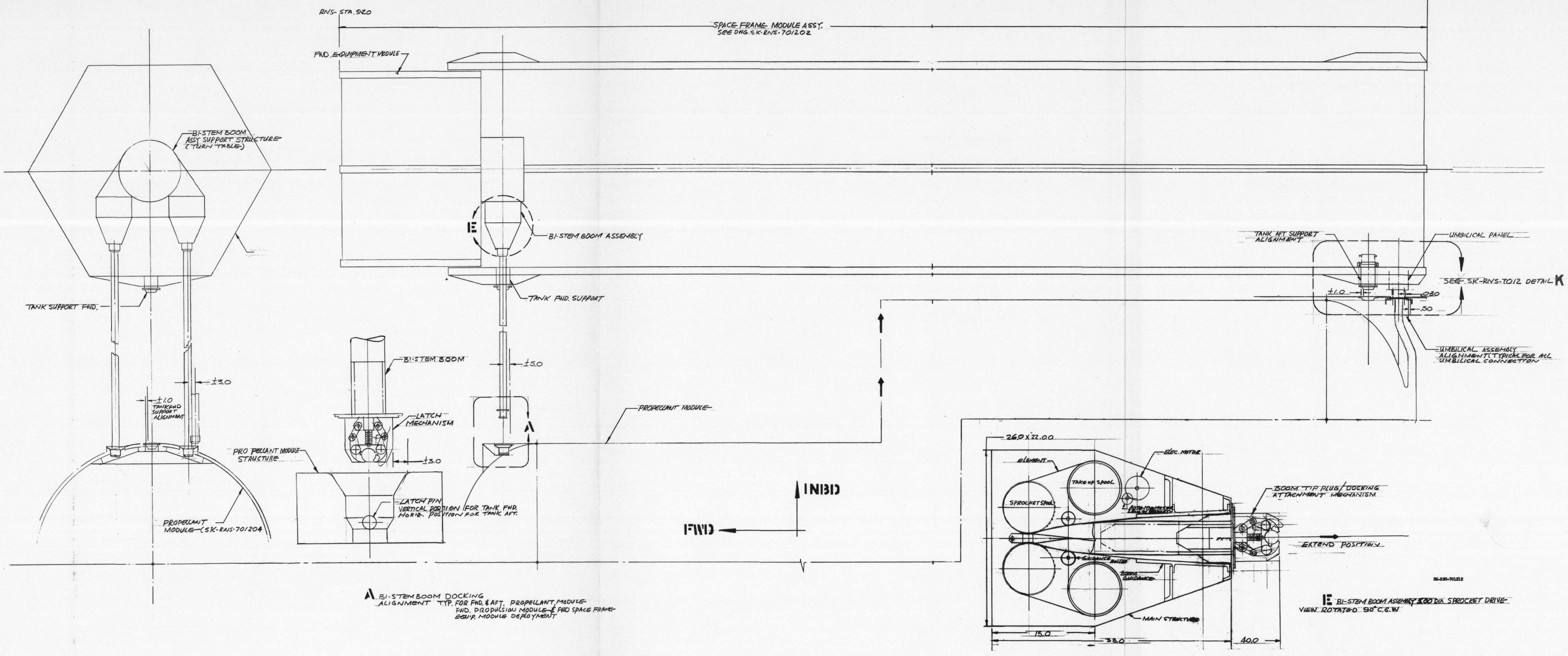

Fig. 2-12 Propellant Module Orbital Assy. Alignment

Section 3
PROPULSION SYSTEM

3.1 INTRODUCTION

The propulsion system provides the RNS with both primary and secondary propulsion. Primary propulsion capability is provided by the 75,000-lb thrust NERVA engine. Hydrogen is the propellant for the primary engine system. The propulsion system also provides for the storing, handling, managing, and pressurization of the propellant during all phases of the mission and engine operation. Secondary propulsion requirements are satisfied with a completely separate propulsion system utilizing N_2O_4/MMH propellants and small thrusters. The definition of the propulsion system is divided into four parts: the primary engine system, propellant feed and storage, propellant expulsion, and auxiliary propulsion. The specific elements which make up the propulsion systems are identified in Fig. 3-1 by location within each module.

3.2 PRIMARY ENGINE SYSTEM

The primary engine system for the RNS consists of the NERVA engine and an engine flyaway control module shown in Fig. 3-2. The engine is completely defined in Reference 3-1. The basic characteristics of the engine are:

Thrust	= 75,000 lb
Specific Impulse	= 825 sec
Propellant Flow Rate	= 90.91 lb/sec
Weight (5.01)	= 27,728 lb

The engine flyaway control module provides the capability to fly the engine away in an autonomous mode. This module is a completely self-contained propulsive system that is attached to the engine on the ground. This module and the engine are launched in the Space Shuttle as a unit and are then connected to the propulsion module tank to complete the propulsion module.

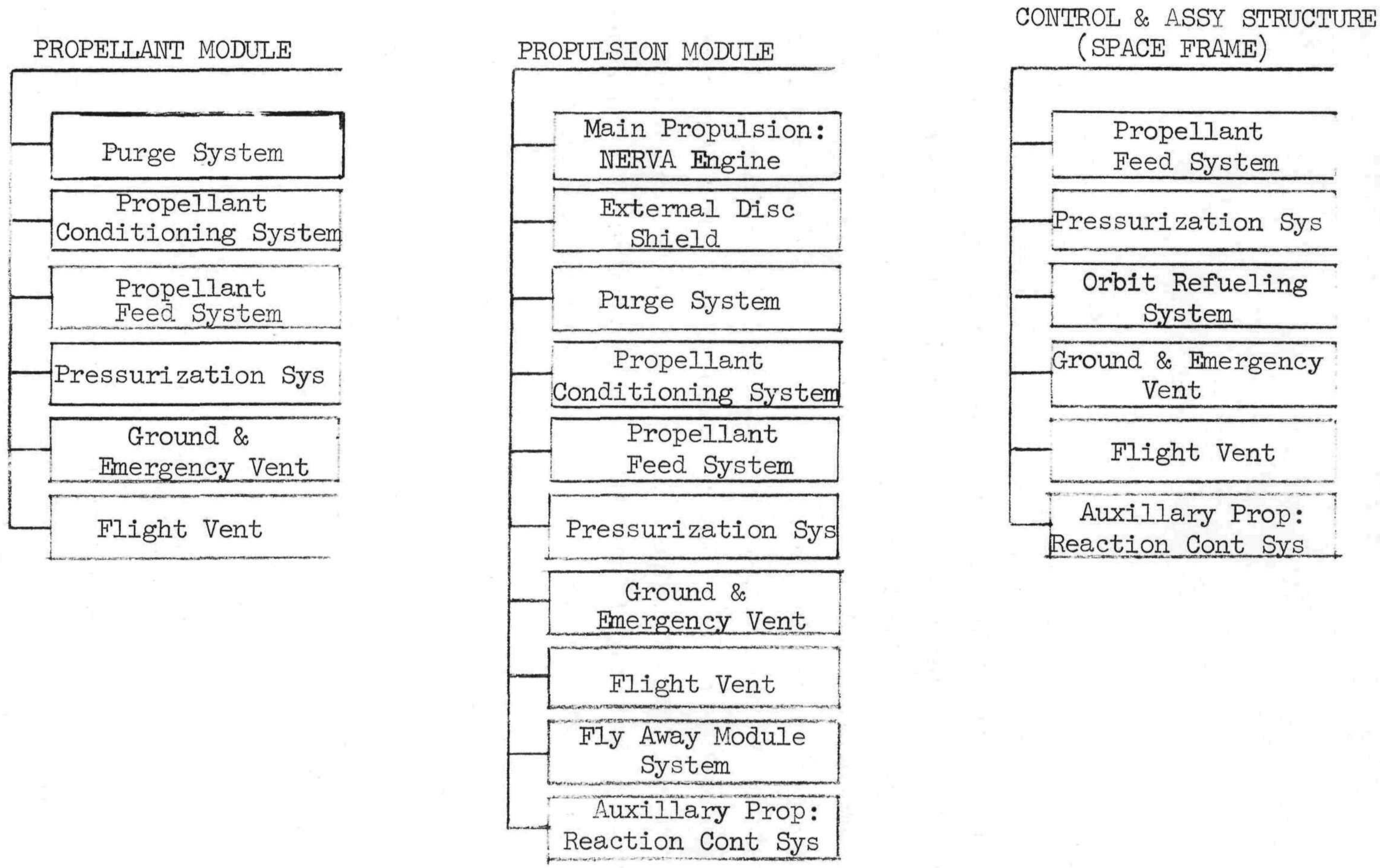

Fig. 3-1 Propulsion System

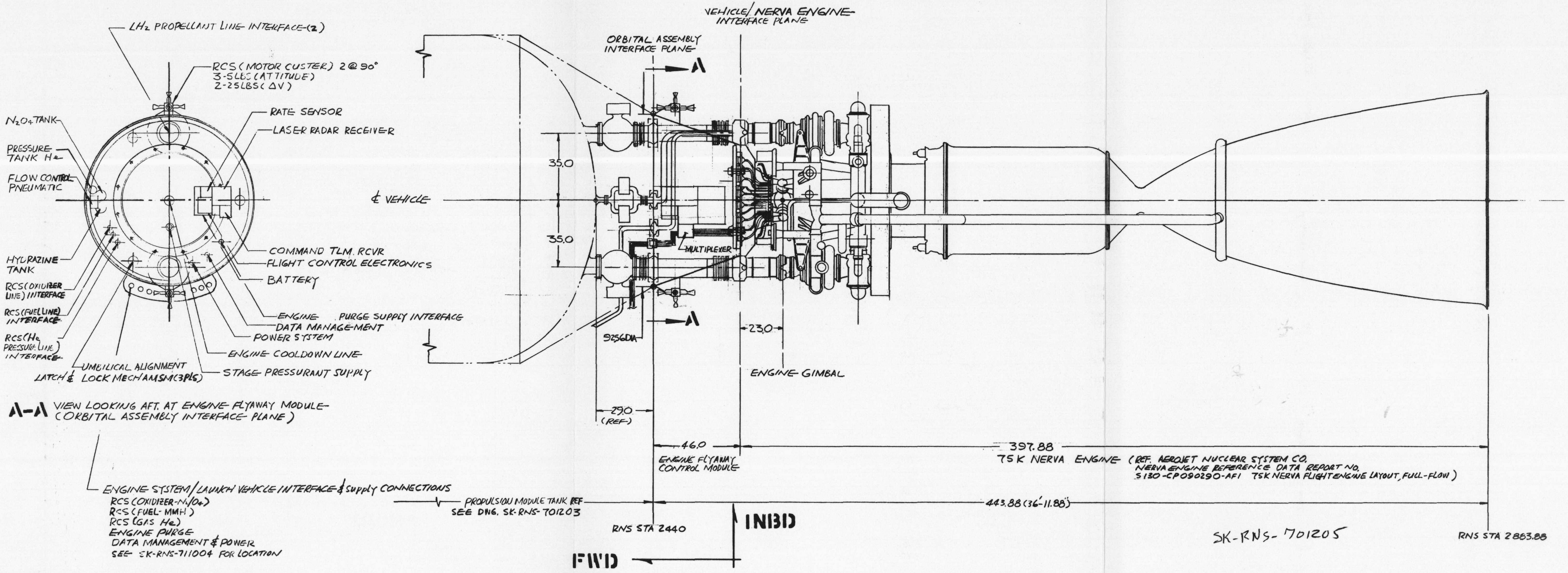

Fig. 3-2 Engine System

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3.3 ENGINE FLYAWAY MODULE

3.3.1 Module Requirements

This self-contained approach for the module requires a separable engine interface adapter as an integral part of the engine thrust cone structure. All power, propulsion, control, and communications subsystems required for engine handling would be included in this module and separated with the engine when removal is required. The following design goals were utilized:

- Totally autonomous operation (i.e., no EVA or manual control)
- Simple maintenance/resupply
- RNS systems commonality
- Power/weight efficiency
- Technical feasibility

3.3.2 Module Description

Initial In-orbit Vehicle Assembly. It is desirable to have the initial assembly point coincide with the remote separation joint. If the RNS structure is powered up and has minimum maneuvering capability, the engine assembly could be released from the Space Shuttle at a stand-off location and the automatic docking system used to fly the assembly in. No EVA should be required.

Engine Removal – Docking. The engine should be remotely separable and attitude stabilized (internally) for flyaway operations. Using the candidate control system, Fig. 3-3, as a reference, the flyaway maneuvers could be accomplished in the following manner:

After main engine propellant lines are vented, the flyaway module is switched to internal power and a fault/status check made by the RNS computer. If all systems check, the engine module is armed and the remote separation joint is uncoupled. The RNS laser engine docking radar is slaved to track the receiver and the RNS NG&C computer used

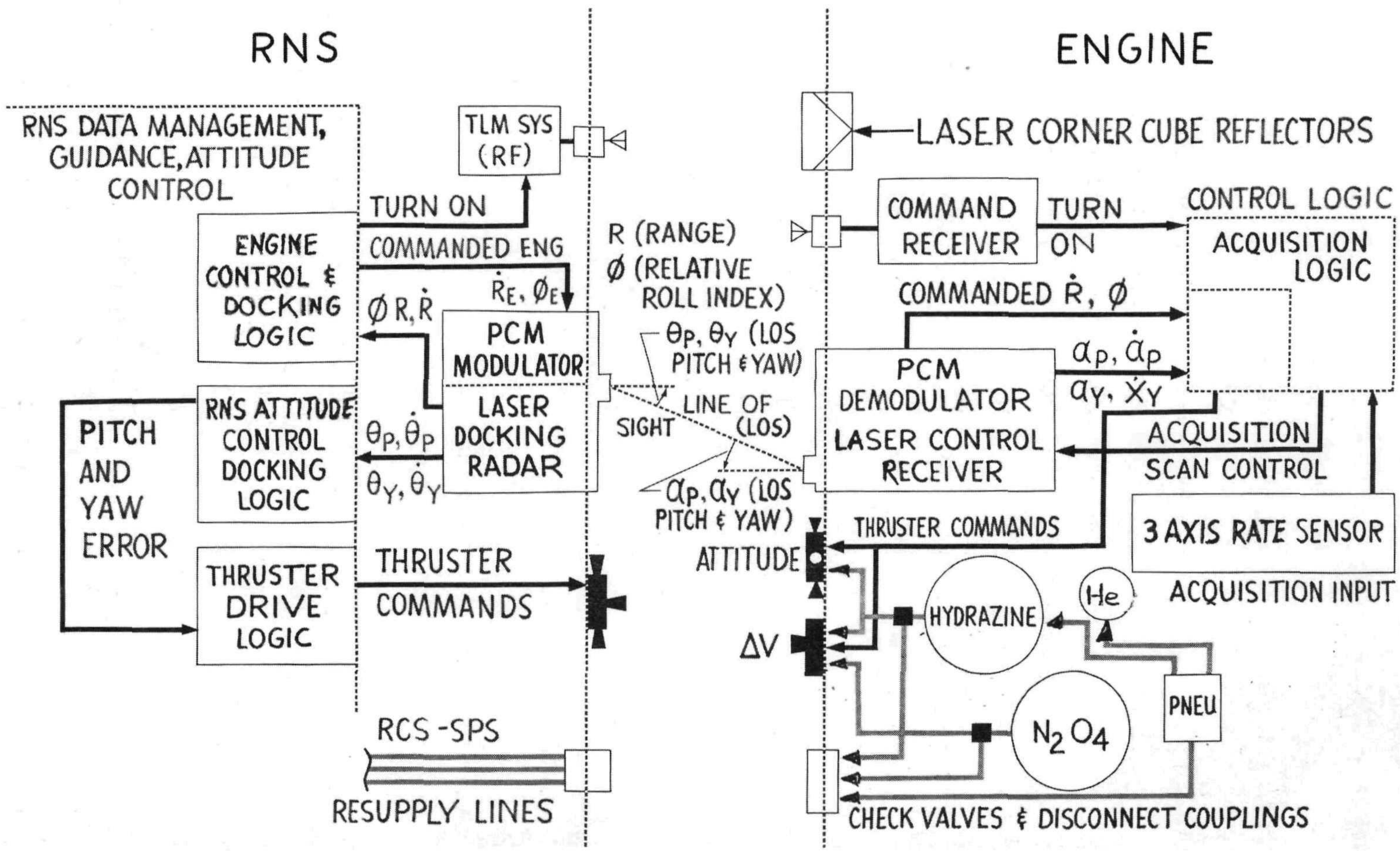


Fig. 3-3 Engine Flyaway Control System

to compute maneuvering commands for the engine. During the separation, the computer will generate error signals (relative velocity, pitch and yaw in terms of LOS) to be coded PCM on the laser signal and demodulated by the module receiver. If two 25-lbF thrusters are used to generate the 2 ft/sec separation rate, the acceleration time will be reduced to under 30 sec, simplifying the control problem.

During the transit time (1.5 hr.) the engine will be attitude stabilized by the LOS angle errors $X_{p1}X_y$) measured by the module laster receiver. After the engine has reached its storage station and has been stopped, all residual translation (relative to the RNS) will be removed to within the accuracy of the laser radar (3 cm/sec). Precise attitude and station will be determined by recursive filtering the position O/P of the laser radar over a time period (10-30 min) at the end of which all remaining residual rates will be removed. It should be possible to execute this maneuver with sufficient accuracy to reduce the residual relative drift rates between the engine and RNS to less than 1000 ft/2 weeks, eliminating the requirement for station-keeping maneuvers during that time span. At this point the engine system can be turned off (with the exception of the command receiver and rate limiting stabilization circuits) to conserve power. During storage the engine would not be attitude stabilized but rate limited to prevent excessive rotation from building up. If analysis indicates these rates for a two-week period would not be excessive (or a rate damper is added, 5-10 lb), the rate sensor could be turned off and power reduced to less than 1 watt, cutting battery weight by two-thirds.

why not bring it back like a fish on a line

Retrieval. When it is desired to return the engine, a two-stage acquisition is required. The RNS laser must first acquire the 360-deg coverage corner reflectors on the engine. This should only take a few seconds. When the engine is acquired, a turn on-acquisition command is transmitted via telemetry to the engine. At that point the acquisition sensors (small solid-state IR detectors) coupled in an analog bridge will point the laser receiver toward the transmitted beam (pulsed mode). When the LOS angle is less than 15 deg, the receiver will acquire the beam and the return maneuver will essentially be a reversal of the flyaway. During terminal docking, the measured roll misalignment (relative roll index) angle will be transmitted to the engine along with rate commands. The control sources (error signals) will be LOS angle of both vehicles; that is, the RNS will be maneuvered in pitch and yaw such that the LOS angle is zeroed, and the engine will be maneuvered in pitch, yaw, roll and longitudinal axis ΔV such that LOS angle and relative roll index are zeroed and the LOS commanded closing rate is maintained. After docking is complete, the propellant is resupplied from the RNS main RCS storage tanks through normally vented plumbing, and the Ni-Cd batteries are recharged. Equipment in the engine module will be located such that maximum radiation shielding will be derived from the battery and propellant tanks.

3.3.3 Module Characteristics

The functional systems required to provide the capabilities of the engine flyaway module are as follows:

<u>Propellant Sizing</u>

Engine Mass $\cong$ 850 slugs

I_{sp} (MMH/N_2O_4) = 250 sec (small thrusters)

Maneuver ΔV = 2 ft/sec max.

Propellant Weight $\cong \dfrac{\Delta V \cdot M}{I_{sp}} = \dfrac{2 \cdot 850}{250} = 7.2$ lb

ΔV – Accel. and decel. 2 ft/sec twice	28.8 lb
+ miscellaneous station keeping (25%)	7.2 lb
Attitude maneuvers (I_{sp} = 120-240)	10.0 lb
	46.0 lb
Contingency (50%)	23.0 lb
Total Prop. Weight	~70.0 lb
At 1:1 vol. mix ratio – 280 psi	
2 tanks at .47 ft^3 (6 in. dia.)	5 lb
2 lb He pressurization gas	2 lb
200 $in.^3$ sphere (3600 psi)	2 lb
Two Thrust Clusters – 3 motors (5 lb F)	10 lb
2 motors (25 lb F)	16 lb

Battery Sizing. Peak power requirements during separation will be about 25-30 watts for 2 to 4 hours. During storage, the power required will be about 3 to 4 watts for 2 weeks. Thus, total power required for the mission (including retrieval maneuvers) will be about 56 amp-hr (28V system), or about 1.6 Kw hours. If the rechargeable battery is sized for a 2.5 margin (to prolong useful life) the Ni-Cd battery will be sized for 4.0 Kw-hr.

Table 3-1 is a weight statement of the flyaway module functional systems, radiation shield and structure.

The arrangement of the components in the module is shown in Figure 3-4.

3.4 PROPELLANT FEED AND STORAGE SYSTEMS

3.4.1 Subsystem Requirements

The primary function of the propellant feed system is the control and transfer of the propellant from the propellant storage tanks to the engine with the orbital and ground refueling being a secondary function. Propellant storage functions such as venting, conditioning, and insulation purging are also included.

3.4.2 Subsystem Description

The propellant feed system is an interconnection of lines and valves that are part of the space frame, propulsion module, and propellant modules. The feed line size optimization is discussed in Vol II, Part A, Section 6. The basic schematic of the feed system is shown both functionally and by location in Fig. 3-5.

The RNS propellant feed system is designed to combine simplicity, high reliability, and minimum pressure drop in controlling the flow of LH_2 to the engine. Plumbing for each of the six propellant modules is identical. Each tank, including the propulsion module, is equipped with an in-flight vent system that can operate with any combination of liquid and gas at the inlet and an emergency vent system to safeguard the tank against

Table 3-1

ENGINE FLYAWAY MODULE – WEIGHT STATEMENT

Component	Weight, lb		
Systems (with NERVA)			
RCS propellant	70		
RCS propellant tanks	5		
He pressurization gas and tanks	4		
RCS thrusters	26		
Valves, couplings, misc. plumbing, mounting	71		
Laser receiver	26		
Corner reflectors (2 – 180° x 180° passive)	5		
Command receiver and antenna	2		
Acquisition sensor	1		
Control electronics	10		
Battery (rechargeable), distribution	125		
RNS pressurization lines	3		
RNS-RCS lines	11		
RNS-propellant feedlines	20		
Engine cooldown line	10		
	389		
Radiation shield	320		
Total System Weight (with NERVA)		709	
Systems (with RNS)			
Laser radar X-mitter	30		
RCS lines, valves, couplings	10		
Misc. elec. harness and connectors	10		
Total System Weight (with RNS)		50	
Total System Weight (5.12)			759
Structural Weight (2.11)			375
Total Engine Flyaway Module Weight			1134

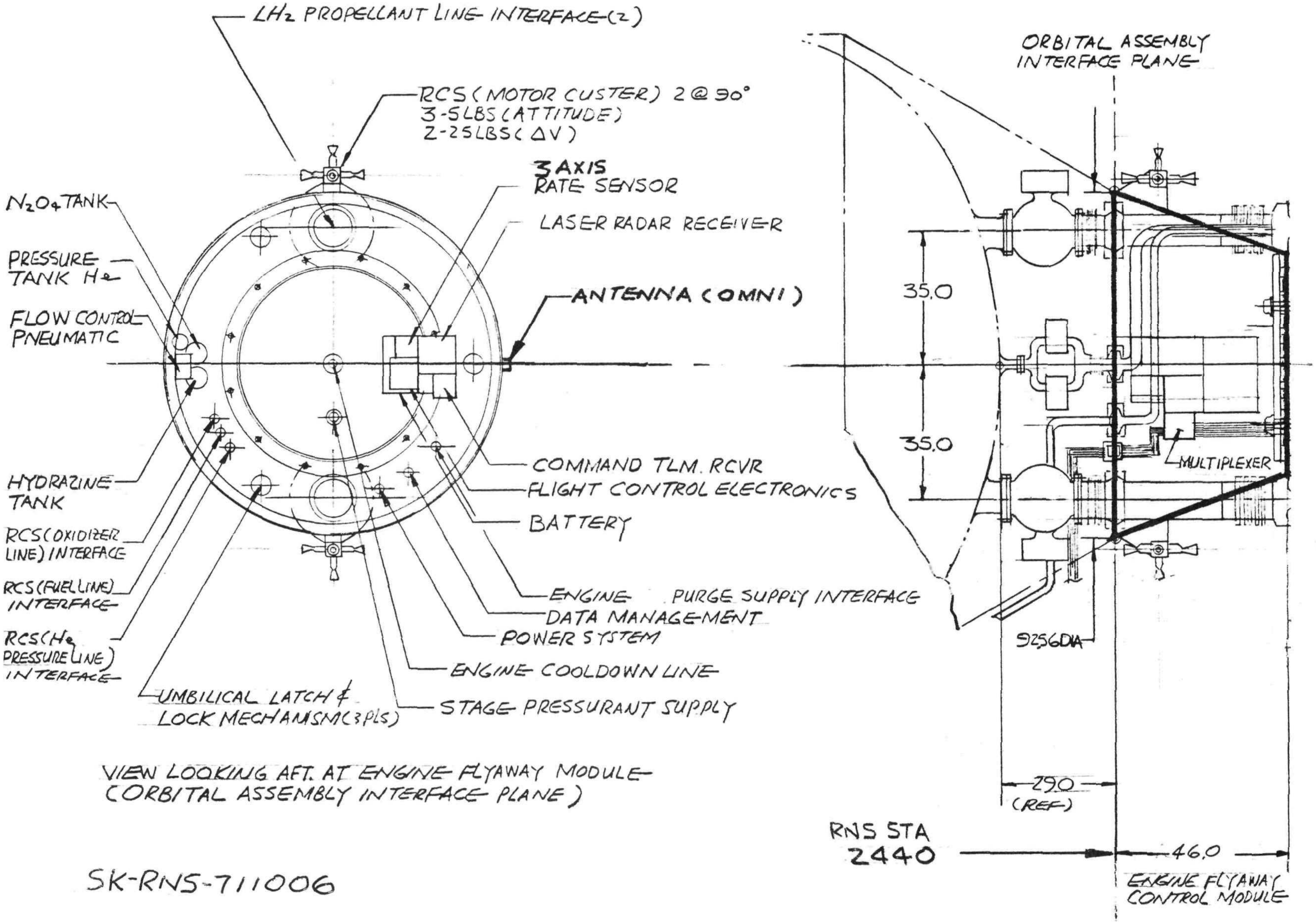
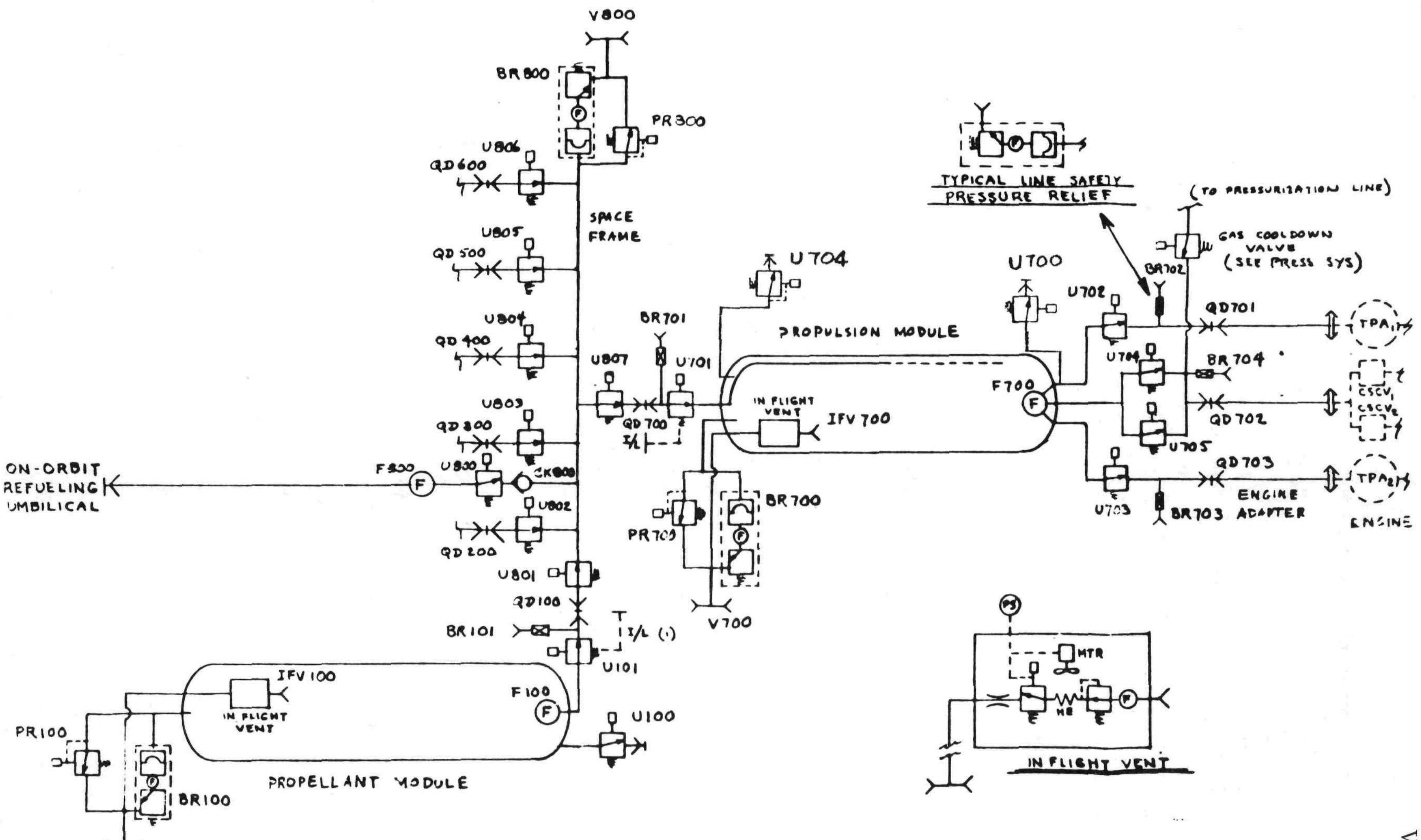


Fig. 3-4 Engine Flyaway Control Module



(1) NORMALLY OPEN TANK SHUTOFF VALVES ARE INTERLOCKED TO PRECLUDE OPENING UNTIL MATED WITH SPACEFRAME.

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Fig. 3-5 Propellant Feed System Schematic

PSOV
Back up
TSOV

abnormal pressure rise. All feed valves, with the exception of the two Tank Shut-Off Valves (TSOV) at the propulsion module outlet, are designed to fail open in order to preclude loss of liquid availability from any tank. The propulsion module TSOV are designed to fail in their operational position, and must then be connected into a closed position in order to be consistent with the NERVA engine redundant feed system and to permit either leg to be isolated in the event of a Turbo Pump Assembly or open line failure.

All tanks are connected to the spaceframe manifold by means of remotely operated disconnects. To avoid undesirable pressure drops, disconnect checks are not used in these lines. Connector leaks can be isolated with the spaceframe isolation valves, normally open during flight. Propellant module shut-off valves are held closed, except when propellant outflow is required.

Ground fill and drain valves on each tank provide a common interface with the Space Shuttle that will not interfere with tank deployment. They are capable of actuation only when the tank is secured in the Space Shuttle cargo bay.

The central propellant distribution system is in the spaceframe. It contains the primary manifold that connects the propellant modules to the propulsion module. The manifold has one branch for each propellant module which is controlled by isolation valves. These isolation valves, U801-U806, are normally in the open position but can serve as control valves should there be a malfunction in the individual propellant tank shut-off valves or if there is a leak in the joint between spaceframe and propellant module. The plumbing connection between the spaceframe and propellant module feed lines is a simple flanged joint in order to minimize the pressure drop in the system. The connection between the spaceframe and propulsion module is very similar and isolation valve U807 operates in a similar manner. The manifold can be vented through valve PR800 if required. In case this normally closed valve will not open an alternate flow path is provided which will rupture a burst disc and then control pressure with regulator BR800. The spaceframe also contains an on-orbit refueling umbilical which extends from the forward end of the spaceframe to the manifold at the aft end, and containing a normally closed valve U800 and a check valve for back flow. The orbit refueling system is shown in Fig. 3-6. The refueling system contains a section of telescope tubing in order to allow for the 80-inch extension of the equipment bay in the spaceframe. Shown in Fig. 3-7 is the latching mechanism and propellant line coupling on the RNS to interface with the Space Shuttle fueling system. Each propellant module is connected to the spaceframe feed line system through a simple flanged disconnect coupling. The propellant flow from each propellant module is controlled through control valves U101-U601. These valves are mechanically interlocked closed while they reside in the Space Shuttle. The line segment between the isolation valve and the control valve in each branch has a pressure relief system which consists of a burst disk followed by a pressure regulator. Each propellant module also has a fill line and fill valve. These fill valves, U100-U600, control the propellant filling into the propellant modules while they are in the Space Shuttle and consequently all have a common interface into the Space Shuttle. Each propellant module also contains an inflight vent and propellant conditioning system. These systems, IFV 100-600, consist of a helium heat exchanger actuated by a valve set which operates when the set vent pressure is reached. Both liquid and gas can be ingested, which is then expended through an orifice and dumped overboard with a set of non-propulsive nozzles. Each system also contains a mixer that prevents stratification of the propellant. Each propellant tank also contains a ground and emergency vent. This vent system is similar to the manifold system in that it consists of a pressure regulated vent valve backed up by a burst disk and pressure regulator. The propulsion module has a similar in-flight vent and ground and emergency vent system. It also contains an inlet control valve, U701, which is interlocked in the closed position while it is in the Space Shuttle. The propellant fill line interfaces with the Space Shuttle in the same manner and in the same place as the propellant modules. Two feed lines connect the propulsion module tank with the engine. Propellant flow is controlled by TSOV U702 and U703. One cooldown line penetrates the tank and provides propellant through two parallel valves, U704 and U705, installed for redundancy. In addition to the liquid propellant used for cooling purposes, residual vapor in the propellant tanks can be utilized. This is transported through the pressurization system lines controlled by a valve, and introduced into the cooldown line downstream of control valves U704 and U705. The engine system is connected to the propulsion module tank with three quick disconnects QD701-703. The engine adapter serves as a connection to the actual engine interface. The interface between the engine

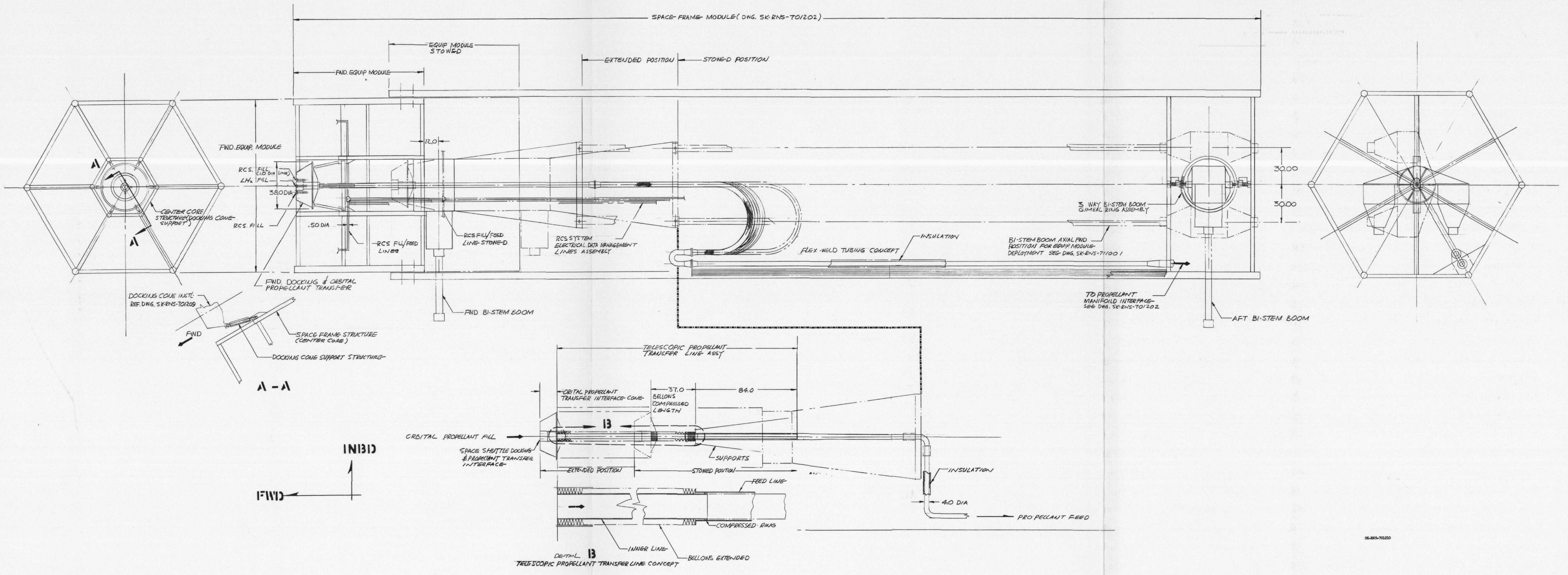

Fig. 3-6 Orbital Refueling System

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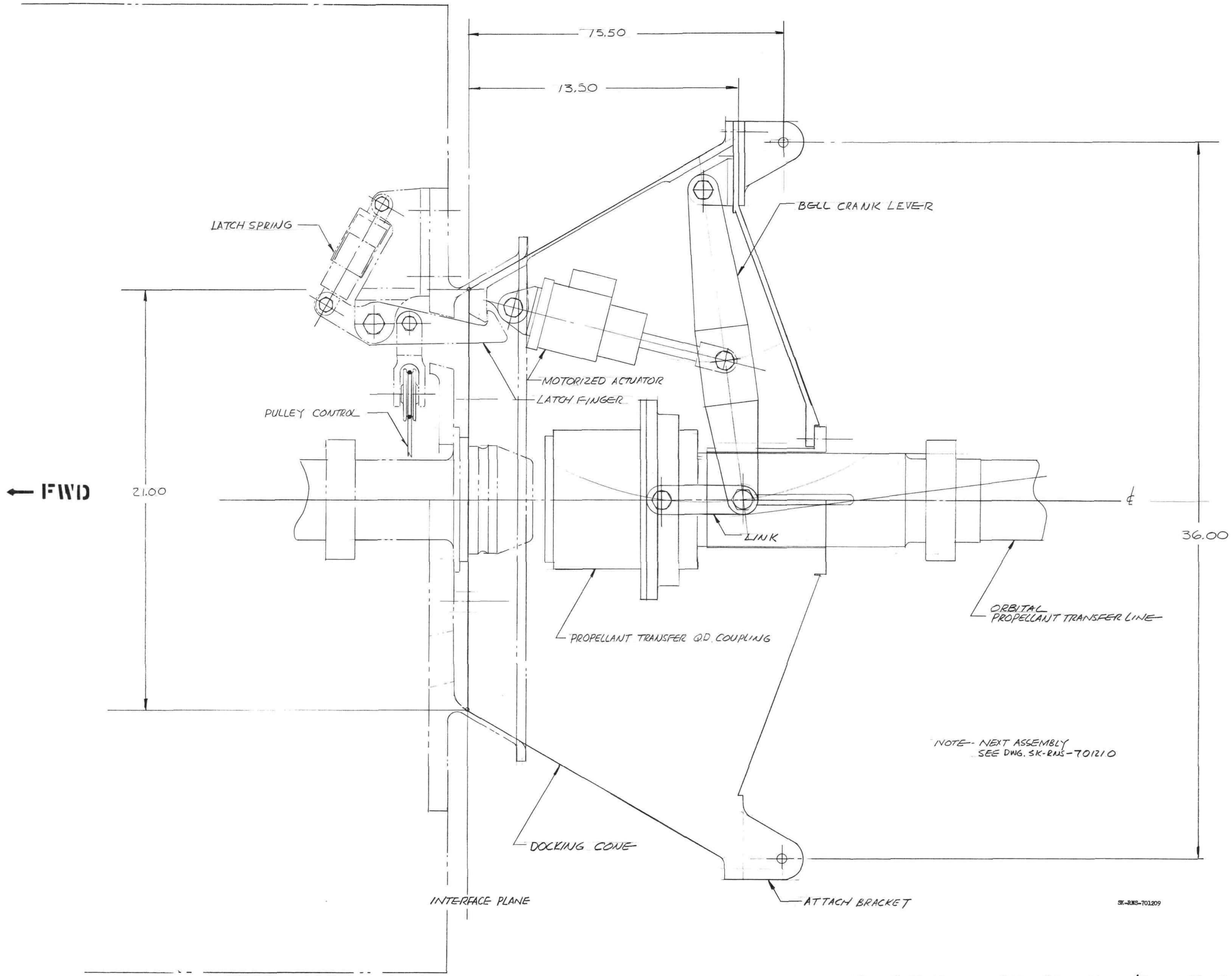

Fig. 3-7 Forward Docking Cone/Propellant Transfer Assembly

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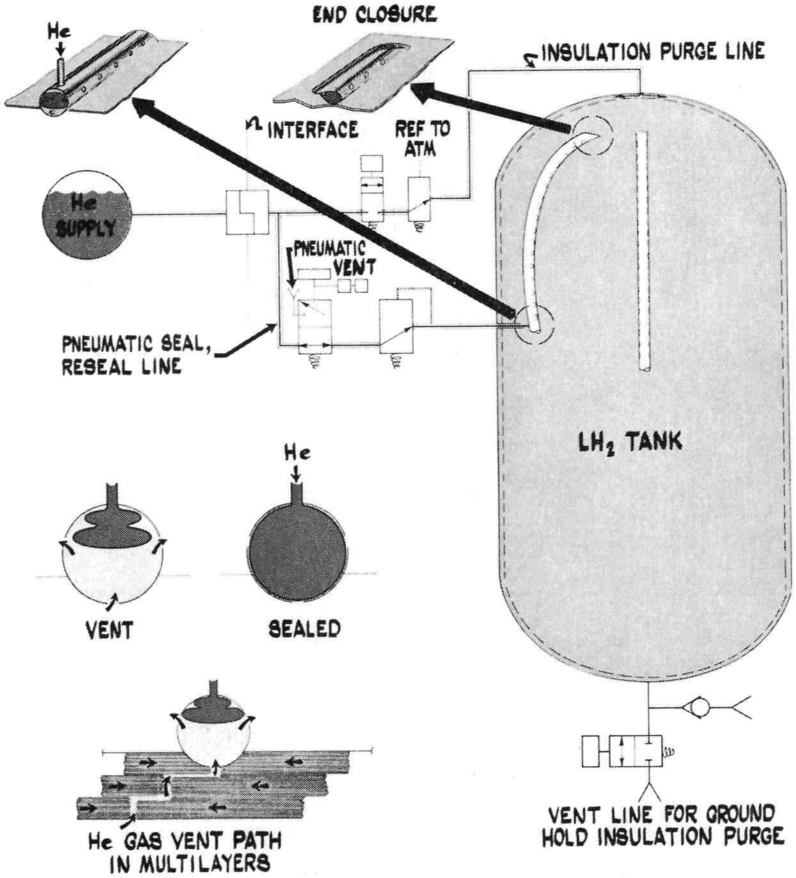
and engine adapter is made on the ground and all other interfaces are made in space. The two feed lines feed their respective turbopump assemblies and the cooldown line feeds the cooldown module.

Many techniques can be used to prevent air liquefaction in the insulation during ground hold. Prior analyses have indicated that helium purged systems are the most promising. In the selected purge system shown in Fig. 3-8, a seal-reseal function is provided on the LH_2 tank purge bag for the ascent (and possibly for reentry cycles). Helium gas is used to remove the air initially from the insulation during ground hold. The purge bag is then vented during ascent so the insulation pressure will reach $< 10^{-5}$ torr in space. If the tank is brought back into the Earth's atmosphere, either the insulation must be repurged if the tank is still cold or the tank must be warmed to prevent air or moisture condensation in the insulation (moisture has been shown to remove aluminized coating but not gold coating). Use of gold instead of aluminum alleviates the moisture problem but drives the insulation material cost up by a factor of about 100.

In the approach developed the vent-path length is reduced, and consequently vent time, by opening the purge bag with tubes located over the insulation joint areas, as shown in Fig. 3-8. Inside each of these tubes is another tube that collapses into a flattened figure-eight shape when not pressurized. When the flattened tube is pressurized, it expands and fills the outer tube, sealing off the vent holes (for ground hold or reentry phases of flight if required). When it is desired to vent the insulation again (ascent), the three-way valve releases the gas pressure in the inner tube. The line sizes have been optimized for specific system requirements. The propellant feedlines to the engine are 8-inch lines, and the cooldown line is 3 inches in diameter. The propellant manifold is 6 inches in diameter, and the propellant fill lines, on-orbit refueling line, and vent lines are 4 inches in diameter.

3.4.3 Subsystem Characteristics

Weights of the various elements of the feed and storage systems are given in Table 3-2.



SK-RNS-712003 Fig. 3-8 Purge System

Table 3-2

PROPELLANT FEED SYSTEM WEIGHT SUMMARY

Subsystem	Weight, lb			
	Space Frame	Propellant Module	Propulsion Module	Vehicle
Propellant Feed System (5.05)				
Lines	87	12	46	205
Valves, Components, etc.	411	72	234	1,077
	498	84	280	1,282
Propellant Conditioning System (5.04)	–	25	25	175
Venting System (5.10, 5.11)	27	33.5	33.5	261.5
Orbit Refueling System (5.09)	90	–	–	90
Purge System (5.03)	–	70	69	489

3.5 PRESSURIZATION SYSTEM

3.5.1 Subsystem Requirements

The primary function of the pressurization system is to pre-pressurize the propellant tanks and provide pressurant during propellant expulsion.

3.5.2 Subsystem Description

Pre-pressurization and pressurization during tank expulsion is accomplished in an autogenous pressurization mode. This system does not, therefore, require separate pre-pressurization for expulsion. The only hardware required is a gaseous hydrogen bleed line from the engine to tank injector and a pressure-modulating valve. The valve is solenoid-operated, pressure-balanced, and is controlled by pressure switch action. Three pressure switches, using a 2-out-of-3 voting logic circuit, narrows the expulsion pressure band to ±1 psi and improves system reliability (as compared to single pressure switch operation).

The tank pressure requirements are a variable throughout the mission. The tank is designed to operate up to 25.4 psia. Propellant is loaded into the tanks so that it is at a saturation pressure of 18 psia at the beginning of the mission. Pressurization requirements vary throughout the mission, since a "tracking mode" of pressurization is utilized. In this mode, the pressure in the tank for each burn is raised up to the level required for engine inlet requirements corresponding to the saturation pressure of the hydrogen. These requirements are shown in Fig. 3-9. Both the normal and malfunction modes are shown in the figure, but the system is designed for the more demanding malfunction mode.

After engine start, pressurization gas is tapped off from the turbine discharge line in the engine which makes available heated gaseous hydrogen for pressurization at 700 psia and 250^{o}R during steady state operation. As pressurant is then injected into the tank, additional speed can be imparted to the pump. Only those tanks that are being emptied are pressurized. Valve actuation times required for tight tank pressure control of

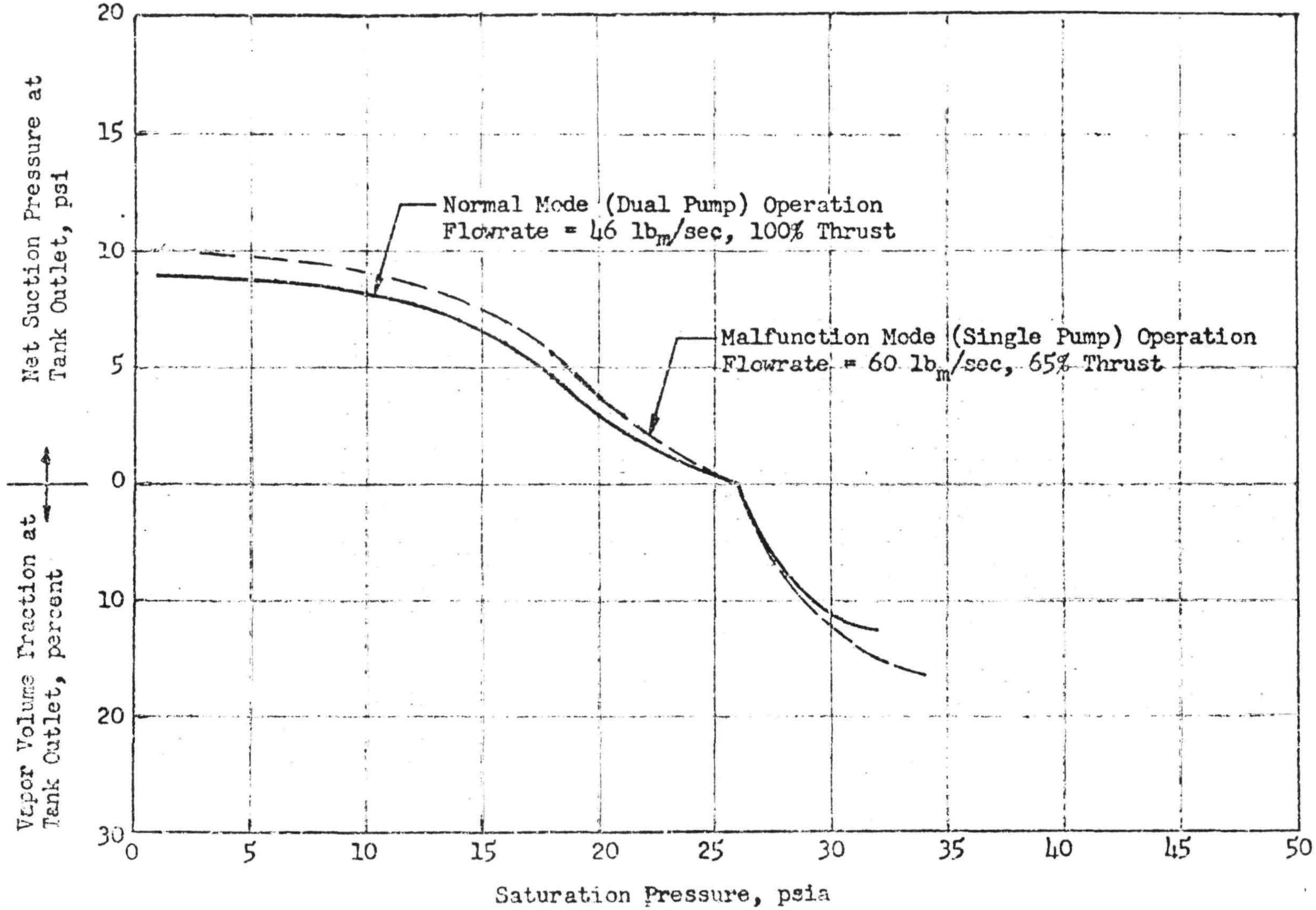


Fig. 3-9 NERVA Turbopump Net Suction Pressure Requirements for Design Saturation Pressure of 26 psia

± 1 psia are well within the state-of-the-art. Following engine shutdown, GH_2 trapped in the pressurization-manifold is dumped to space through a thrust nullifier nozzle. The evacuated lines considerably reduce the plumbing heat leak to the tanks between burns as compared to GH_2 filled lines. When a propellant module is emptied, the residual GH_2 is dumped through this same nozzle to reduce vehicle inert weight.

The maximum electrical power consumption occurs if all of the tanks required for the TLI burn are activated simultaneously. This would require 300 watts to open the valves and approximately 20 percent or less for the duration of the burn to maintain the valves open. Although the power system is sized for this capability only one or two tanks are typically drained simultaneously.

The system schematic shown in Fig. 3-10 identifies the pressure system components and indicates the redundancies in valves and sensors. Pressurization gas coming from the engine enters the propulsion module line through quick-disconnect QD701 and then enters the spaceframe through QD700. The pressurant is then distributed through a manifold to the propellant modules through QD100-600. To provide double redundancy in the pressurization control valves of the propulsion module, three parallel valves, R700-702, are installed. Should one of these normally closed valves inadvertently fail open, orifice OR700 assures that the propellant tanks do not get pressurized.

The multiple pressure sensors PS700 sense the pressure in the tank and the propellant management system then controls the operation of the pressurization control valves. Pressurization of the propellant modules is essentially identical except that only two pressurization control valves are installed on each tank. This is adequate since with the failure of any component in a tank the propellant may be transferred to another tank. After each engine burn the manifold may be vented of the hot pressurant through regulator PR800. If the regulator is inoperative a burst disc and back-up regulator BR800 take over. All venting is through non-propulsive nozzles. Gaseous vapor from any tank may also be used for engine cooldown which is controlled by valve U700.

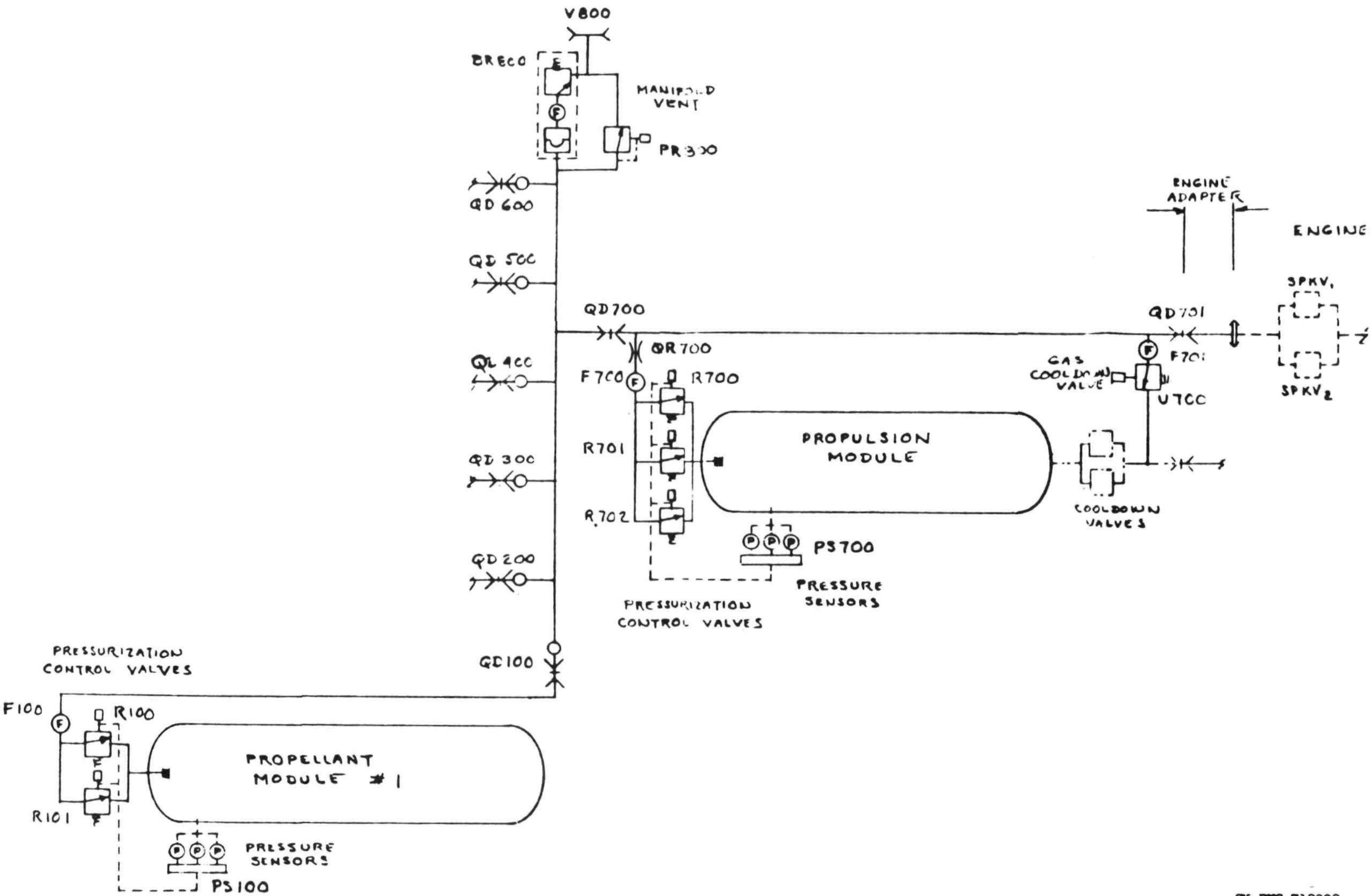


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Fig. 3-10 Pressurization System Schematic

3.5.3 Subsystem Characteristics

Table 3-3 is a weight summary (5.06) of the pressurization system valves and lines.

Table 3-3

PRESSURIZATION SYSTEM WEIGHTS

Components	Weight (lb)			
	Space-Frame	Propulsion Module	Propellant Module (ea.)	Total
Valves, Switches, etc.	29	27.5	16	152.5
Lines	23	16	14	123.0
Total	52	43.5	30	275.5

3.6 REACTION CONTROL SYSTEM

3.6.1 Subsystem Requirements

The Reaction Control System provides the following functions:

- Attitude control
- 3-Axis translation
- Roll control during engine burn

These functions are to be mechanized so that there is dual redundancy for attitude control, i.e., fail operational-fail operational-fail safe. For axial translation, single redundancy is required, i.e., fail operational-fail safe, and for lateral translation, zero redundancy and a fail safe condition.

3.6.2 Subsystem Description

The selected reaction control system utilizes nitrogen tetroxide ($N_2 0_4$) and mon-methyl-hydrazine (MMH) propellants. The RCS will be functionally directed by the NG&C computer to respond to attitude control requirements. The analytic portion of the RCS (modulator characteristic, gains, vehicle center of rotation, thruster logic, etc.) will be software-generated in the attitude control portion of the computer. From the computer I/O section, discrete signals will enable individual thruster firing in any desired combination and timing sequence. To simplify the thruster-computer interface umbilical, the signals can be multiplexed onto a single harness and individual thruster signals decoded by demultiplexers at distribution points on the line. Solenoid valve power will be drawn from the vehicle bus with valves enabled by low power relays.

The recommended thruster layout is illustrated in Fig. 3-11 and thruster combinations for specific vehicle motion are shown in Table 3-4. All tankage is located in the forward equipment module where environmental conditions are most favorable. The detailed schematic of the system is shown in Fig. 3-12.

Definition. There are 22 thrusters required, each having a 100 lbf steady state thrust. The steady state I_{sp} is 300 seconds and each thruster has a 20 msec minimum on-time. The minimum impulse bit is 1 lb-sec at an I_{sp} of 150 seconds. All fuel and oxidizer valves are series-redundant. The total impulse required for the baseline lunar shuttle mission is 630K lb-sec. This requires a nominal propellant loading of 2200 lb. The system has a growth capability to 1M lb-sec of impulse.

Operation. The operational modes, sequencing, and select logic will be programmable software functions. When point failures are detected (through instrumentation or incorrect vehicle response), an alternate section matrix will be inserted to make optimal use of the remaining system. In this way, the full performance redundancy will be supplemented by many levels of degraded performance operation for a great number of failure modes. The surviving portion of the system is protected from damage (through pressure loss, etc.) from other failed components by an extensive valve isolation network. After a number of critical failures, the probability of saving the mission through a minimum residual capability would still be high.

In normal, no-failure, operation, as many of the redundant elements as possible will be isolated from each other. At set intervals the expendables will be switched and balanced to maintain the minimum required levels for abort level operation. During

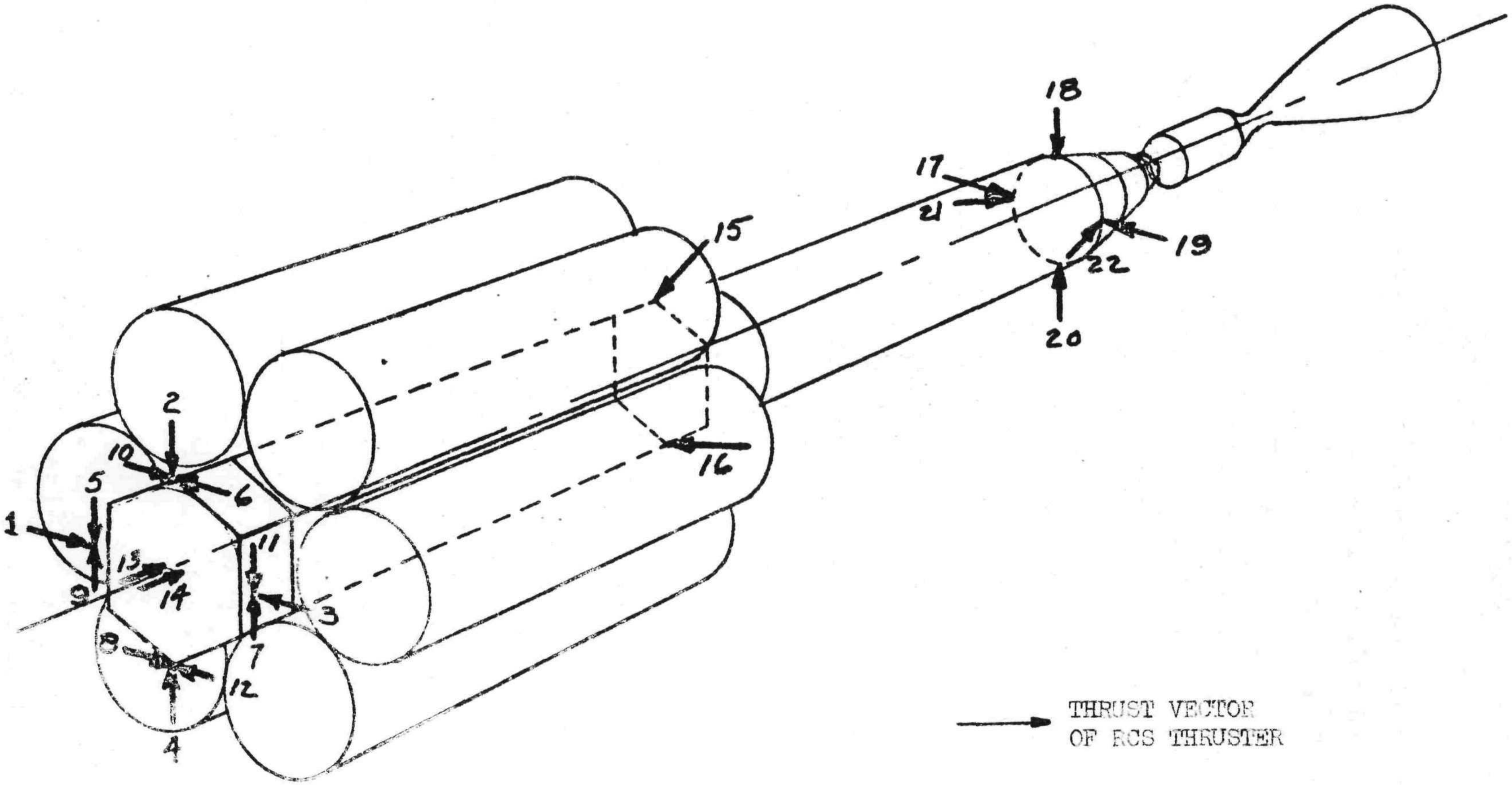

Fig. 3-11 Suggested Thruster Orientation for RNS Reaction Control System

Table 3-4

THRUSTER COMBINATIONS FOR VEHICLE MOTION FOR RECOMMENDED SYSTEM
(22 Thrusters)

Motion	Balanced	Unbalanced (1)
Yaw Left	1 plus 19 8 & 10 " 19	1; 8 & 10; 19
Yaw Right	3 plus 17 6 & 12 " 17	3; 6 & 12; 17
Pitch Up	4 plus 18 7 & 9 " 18	4; 7 & 9; 18
Pitch Down	2 plus 20 5 & 11 " 20	2; 5 & 11; 20
Roll Clk	5 & 7 full torque 6 & 8 full torque 1 & 6 1/2 torque 3 & 8 1/2 torque 2 & 7 1/2 torque 4 & 5 1/2 torque	5; 6; 7; 8
Roll CClk	9 & 11 full torque 10 & 12 full torque 3 & 10 1/2 torque 4 & 11 1/2 torque 1 & 12 1/2 torque 2 & 9 1/2 torque	9; 10; 11; 12
Left	3 plus 19 6 & 12 " 19	6 & 19; 12 & 19 (Use 9 & 11 or 5 & 7 for roll balance)
Right	1 plus 17 8 & 10 " 17	10 & 17; 8 & 17 (use 9 & 11 or 5 & 7 for roll balance)
Up	4 plus 20 7 & 9 " 20	9 & 20; 7 & 20 (use 6 & 8 or 10 & 12 for roll balance)
Down	2 plus 18 5 & 11 " 18	5 & 18; 2 & 18 (use 10 & 12 or 6 & 8 for roll balance)

(1) Only major combinations cited.

Table 3-4 (cont)

Motion	Balanced	Unbalanced
Forward	15 & 16	15; 16 (Pitch up or down depending on CG location)
Aft	21 & 22 (PL attached) 13 & 14 (PL removed) 13 " " 14 " "	21; 22

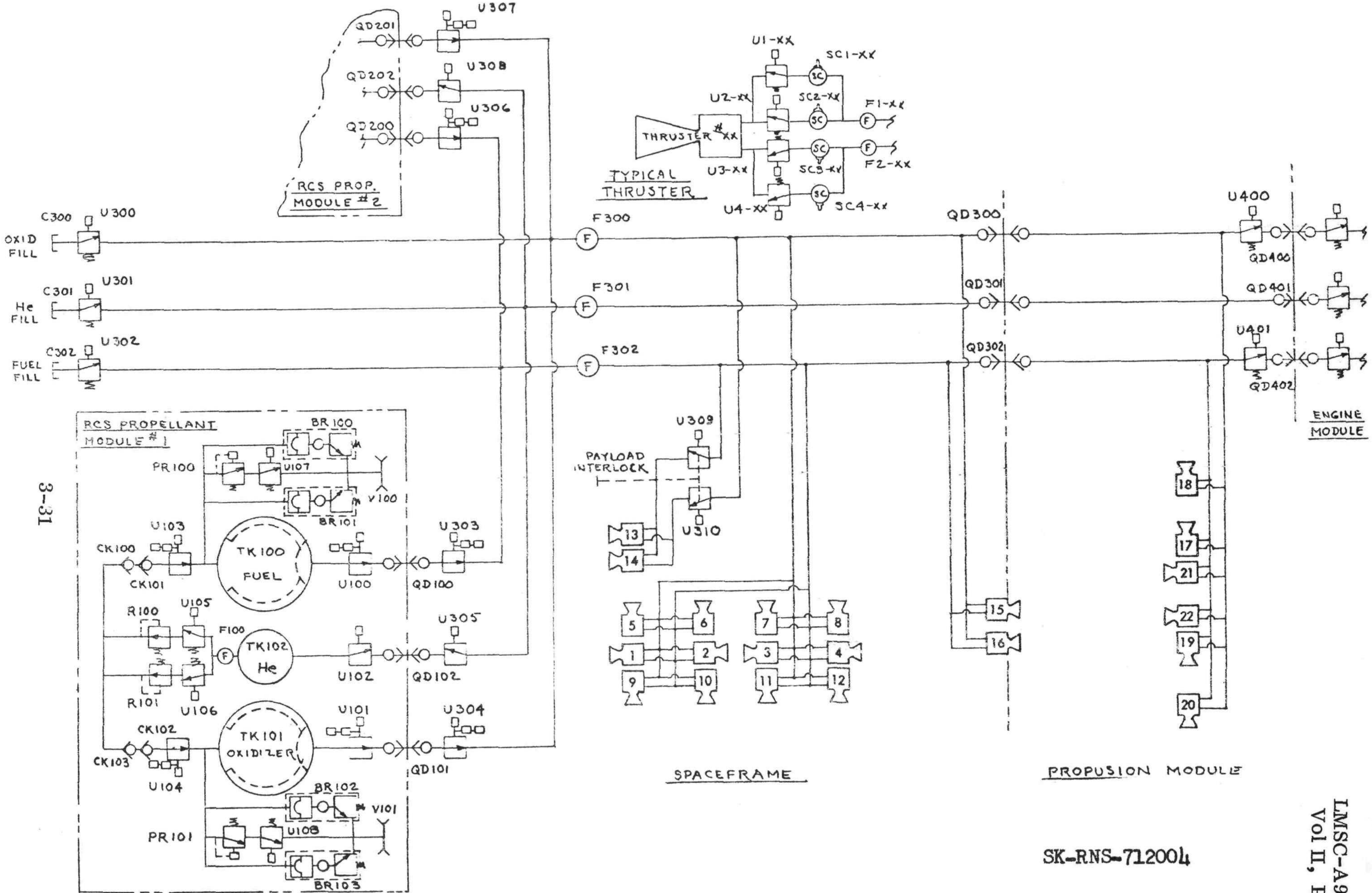

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Fig. 3-12 Reaction Control System Schematic servicing, the system should be recharged from a single location. Various system elements will be charged in sequence, starting with the flyaway engine module, with remote valve operation requiring a minimum (if any) of EVA. Expendables replacement is the favored method of servicing (as opposed to wet tank replacement), since feed line losses are kept to a minimum.

Installation Requirements. In several thruster locations, exact location and blast impingement are critical. All thrusters located on the forward equipment module must not extend beyond the hexagonal bay envelope to allow for extension and retraction of the module. Similar problems exist in the other installation areas (aft core structure and propulsion module skirt). Motor installation must be such that the thrusters do not extend beyond the allowed 15 by 60 ft envelope. At the same time, the blast of the thrusters must not impinge on temperature or erosion sensitive elements. In several cases, this will require less than optimal thrust vector pointing where blast must be angled away from tank walls, etc. Blast shielding will be required in most of these areas.

All tankage and plumbing in each modular area (core, propulsion, etc.) must be brazed and leak checked in ground manufacture. This should essentially limit potential leaks to module interconnect points and fill disconnects. The disconnects and interconnects can be valve isolated from the rest of the system except when needed so that the potential leak duty cycle is kept low. The sealing problems will be no greater than those encountered in the remote connection of the main propulsion system components since the number of joints is considerably less and cryogenic materials are not involved.

Environmental Requirements. The proposed propellant system must be maintained at $-5^{o}C$ or above to avoid slushing (freezing of N_2O_4 at $-10^{o}C$, MMH at $-35^{o}C$). Locating the tanks near some high dissipation electronics in a well insulated area would eliminate the need for active heating. Long propellant feed lines will require insulation, isolation from cryogenic environment, and in some cases, resistive wrap heating elements. Proper design will utilize available heat sources where possible.

Radiation effects are largely unknown at this time. At present, it is not probable that the RCS will be sensitive to location in the aft section. In the past, under certain laboratory conditions, N_2O_4 has exhibited unusual viscosity changes which have been tentatively linked to absorption of metallic byproducts of corrosion. It is not felt that this is a serious problem area since the dependent conditions of these observations are not similar to planned operating methods. This phenomenon is under study at LMSC and further lab simulations are under consideration at this time.

Alternate Layout. For purposes of reliability it may be desirable to locate one complete tank set in the propulsion module aft skirt section and provide isolated interconnection between the two supplies (such that all thrusters in the system have access to either or both supplies). It would be advantageous to be able to service the system from one docking location (on the forward equipment module). Without thorough analysis it is estimated that the inert weight would be increased by about 60 lb and the lost propellant (due to line fill and venting) would increase from 100 lb to ~ 25 lb. This approach would then bring system weight to 3050 lb gross.

3.6.3 Subsystem Characteristics

A weight summary of the Reaction Control System is shown in Table 3-5.

Table 3-5

REACTION CONTROL SYSTEM WEIGHTS

Description	Weight, lb	
Hardware (6.01)		
Tanks (2 fuel; 2 oxid.)	60	
He spheres, gas	55	
Thrusters (22)	121	
Plumbing & Valves	195	
Structure & Thermal Control	130	
Propellant (oper. losses)	100	
Total Hardware		661
Propellant (10.0)		
Impulse		2200
Total System		2861

Section 4
ASTRIONIC SYSTEM

4.1 INTRODUCTION

The astrionic system of the RNS provides the vehicle with data management; guidance, navigation, and control; communication; power; and environmental control. These systems are located throughout the spaceframe, propulsion modules, and propellant modules, as shown in Fig. 4-1.

An integrated avionics system was utilized to integrate all of the astrionic subsystems physically and functionally. The schematic in Fig. 4-2 shows the functional interaction as well as the physical location of the various elements of the system.

The data management subsystem integrates the RNS avionics system, providing for the control of signal flow between subsystems. The data management subsystem consists of a central processor, a high-speed memory, and bulk storage. A data bus is used for signal distribution. Local processors are utilized in each module to reduce the data flow on the bus and perform the maximum number of computations in situ.

The signals are further distributed within each module on a data bus to the respective standard interface units. The indicated input/output boxes may be integrated into the respective components or may act as separate units for interface between the components and the standard interface units. The diagram indicates not only the data management distribution system throughout the RNS but also the interaction of the power system, guidance system, and communications system with the data management system.

The use of this integrated approach with a local processor on each module allows for the monitoring and checkout of all subsystems not only on the RNS, but also when the

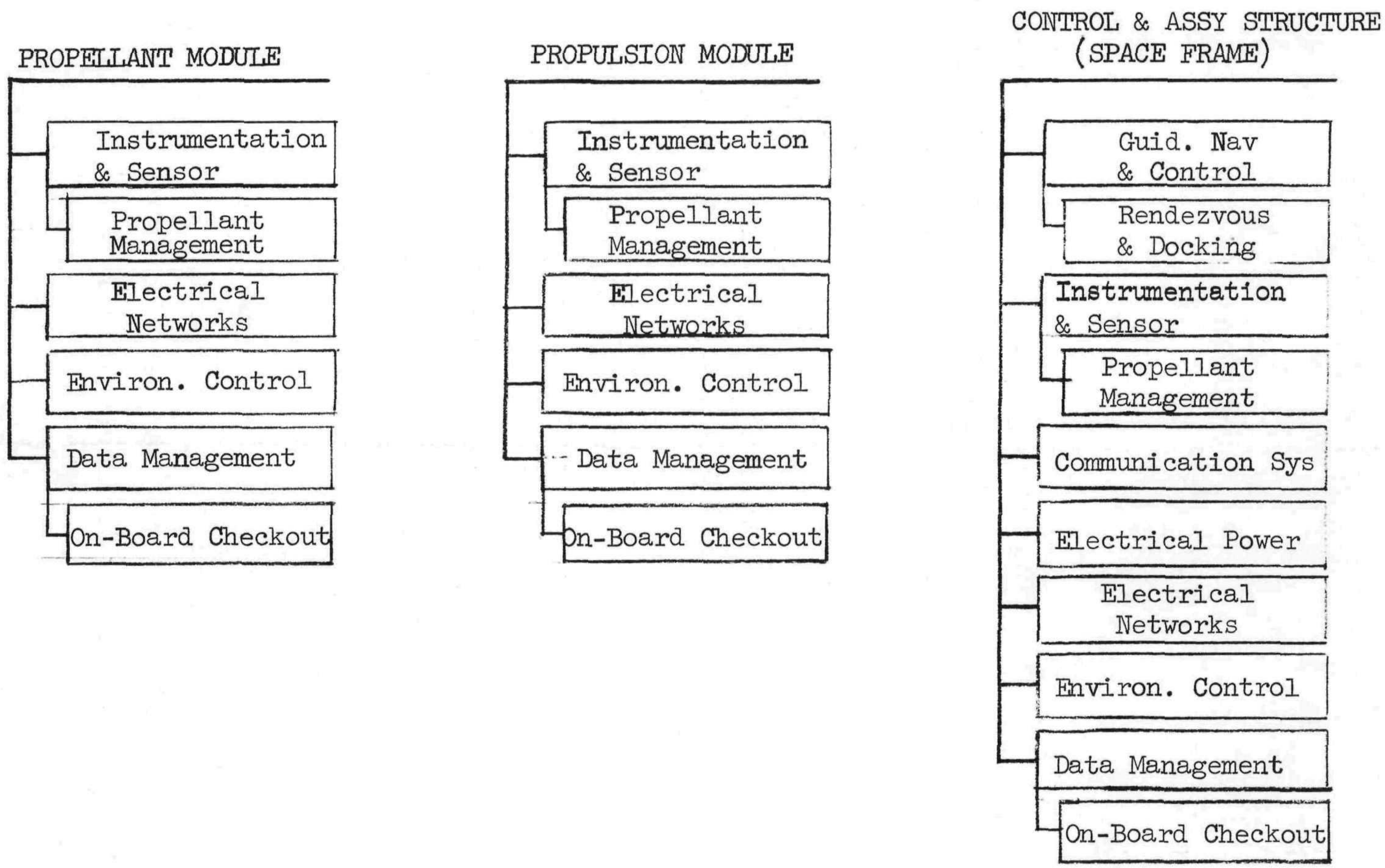

Fig. 4-1 Astrionics System

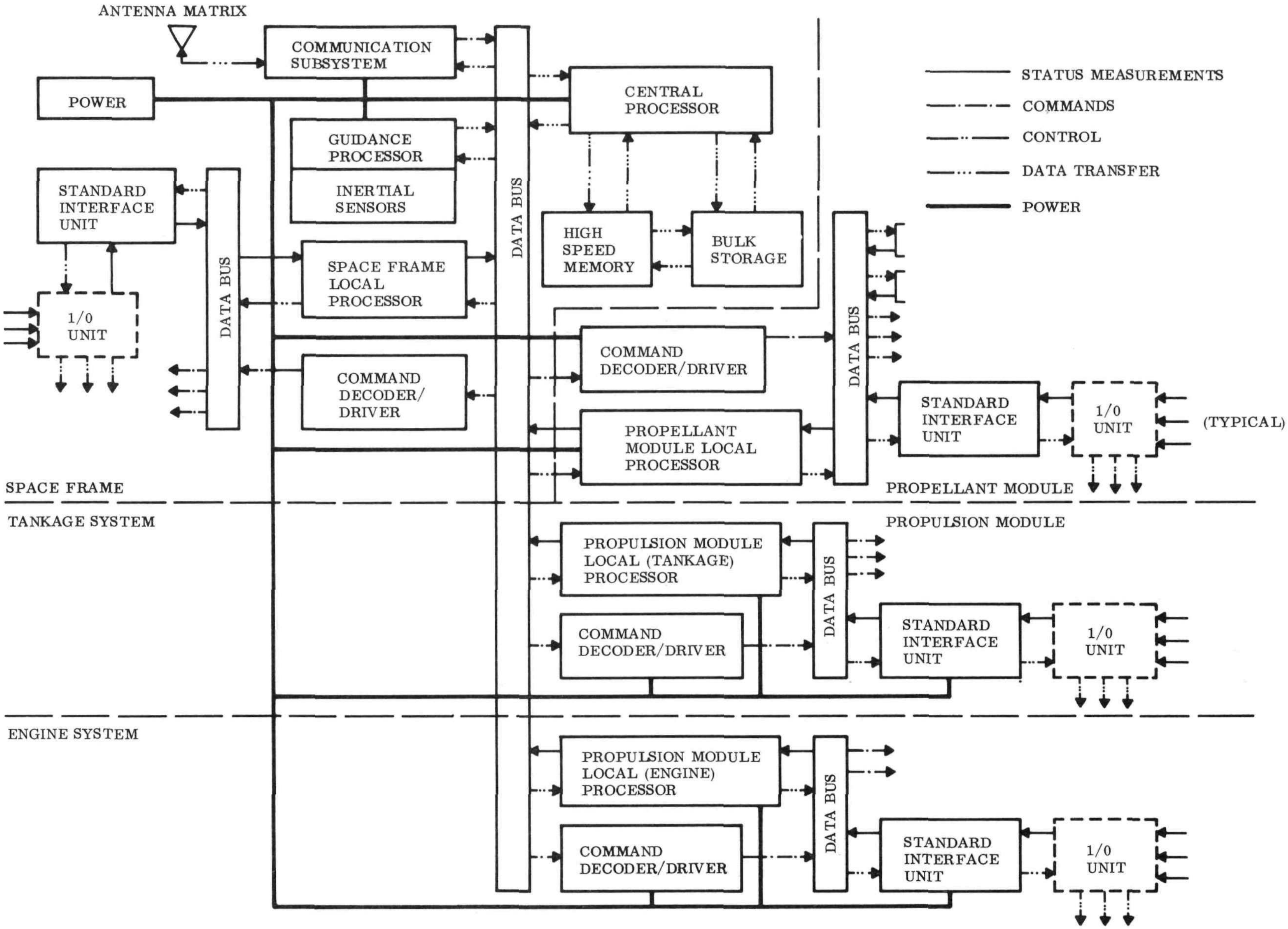

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Fig. 4-2 Integrated Avionics System

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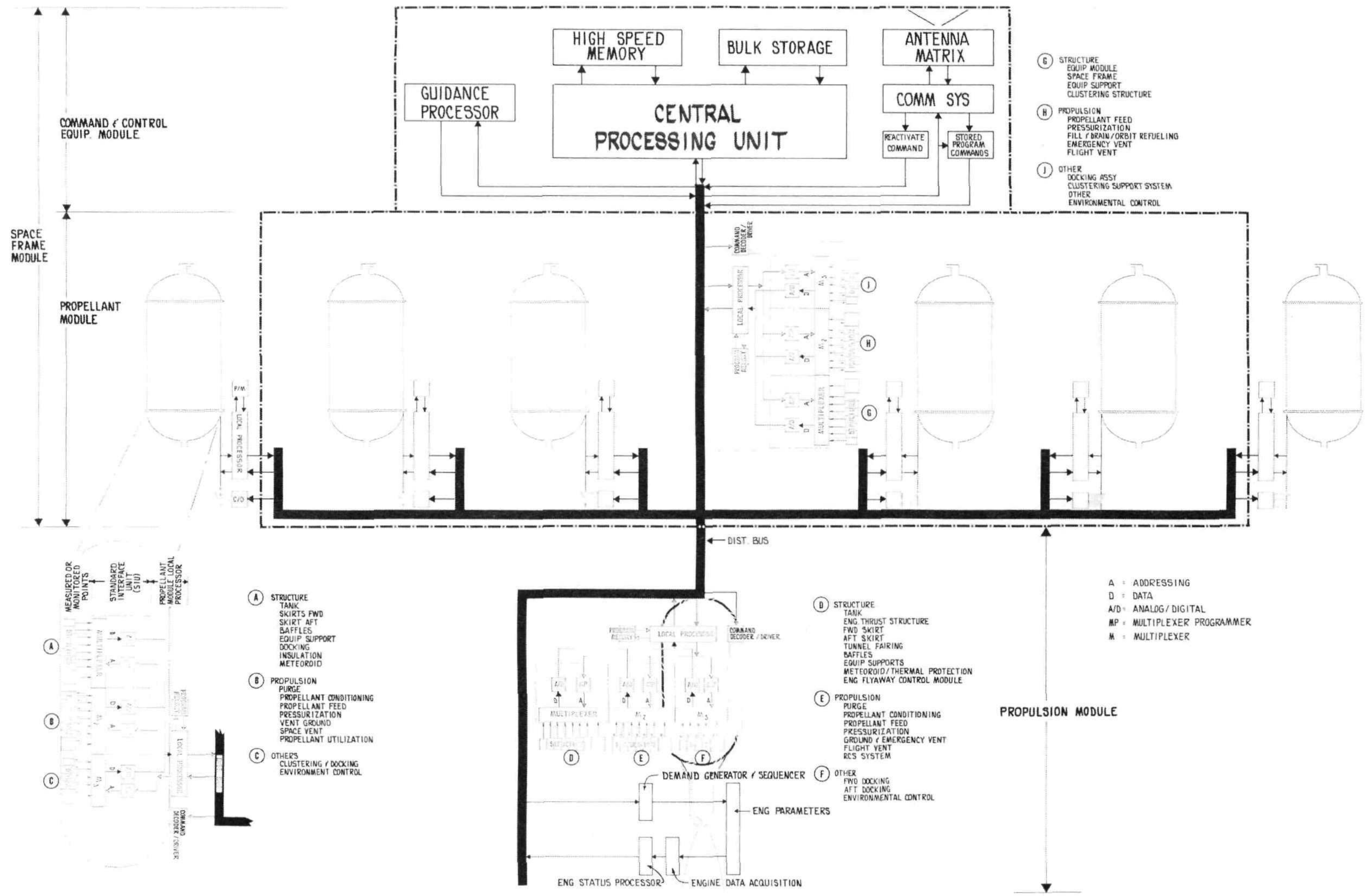

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Fig. 4-3 Data Management System

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module is in the Space Shuttle. Checkout on the ground and during ascent is feasible with minimal software updating by the Space Shuttle data management system.

4.2 DATA MANAGEMENT SYSTEM

4.2.1 Subsystem Requirements

The data management system acquires, conditions, processes, and stores the RNS data. The data management system also integrates all of the avionics systems providing for the control of signals between the elements of the system.

4.2.2 Subsystem Description

The data management system shown in Fig. 4-3 consists of a central processing unit, high speed memory, bulk data storage and a remote data acquisition and control system interfacing with all vehicle subsystems by means of a data bus that runs throughout the vehicle. The data bus is shown both physically and functionally in Fig. 4-2. A built-in test equipment, or BITE, approach to acquiring and controlling test data is utilized wherever possible. This system utilizes multiplexer, converters and command encoders/decoders. In addition a remote (local) processor is utilized which performs a significant number of functions at the module level and thus minimizes the communications with the central processor. This approach also requires a standard interface unit with each functional area and therefore permits a more thorough ground checkout and a replacement readiness prior to orbit launch or assembly.

This system must perform the following functions:

- Warning and Abort. The identification of the incidents which may or may not need immediate attention. Abort evolves from critical situations usually pertinent to safety of life when manned. If unmanned, it could mean impending destruction. Either of these two cases would result in a move to a predetermined contingency plan.
- Malfunction Detection. The determination of an error, whether it be hardware or software. It is primarily a limit testing or error checking procedure.

- Onboard C/O. The logical isolation of failures to the level of a box or unit. This can be on-line, off-line, automatic or manual. Test point data, interface data, and intraface data are required to perform this function satisfactorily.
- Propellant Management. The determination of propellant tank draining sequence and control.
- Operations Support. The provision of pre-orbit launch checkout, countdown of the RNS, launch, and mission planning, including automatic sequencing and configuration control.
- Maintenance Support. The provision for checkout and thorough self-repair technique (i.e., voting logic and switching-in redundancy) in order to establish repair or replacement requirements.

4.2.3 Subsystem Characteristics

The selected data management system has a total weight (7.09) of 393 lb. The computer and memory data storage unit are the largest bulk items, and are located in the equipment module cylinder near the top of the spaceframe. The basic system weight, volume, and power requirements are listed in Table 4-1. Additionally, an allocation of 20 lb per module has been added for intramodule distribution. An allocation of instrumentation and sensor weights (7.02) has also been made. Ten pounds are allocated to each propellant and propulsion module and to the spaceframe module.

Because of the minaturization in other items of the subsystem, physical distribution, remote location, weight, volume, and power there is little special and performance impact on the installation requirements to the RNS. However, emphasis must be given to (1) the location of remote data management cable with respect to electrical power cable due to EMI, (2) the protection of equipment from radiation, and (3) the location of equipment for easy maintenance by replacement.

4.3 COMMUNICATIONS SYSTEM

4.3.1 Subsystem Requirements

The communications system provides for the assembly of signals, conversion to radio frequency, command decoding and generation, and reception and transmission between the RNS and all space communication network elements, including ground communication elements where applicable.

Table 4-1

DATA MANAGEMENT SUBSYSTEM CHARACTERISTICS

Assemblies	WT(lb)	VOL. (ft^3)	POWER (watts)
Computer & Memory	118	1.0	168
Remote I/O	20	.06	60
Cable RI/O	18	.18	
Central Bus Module	18	.08	70
Cable C I/O	22	.1	
Pwr Unit	17	.1	40
Data Storage	40	.2	51
Totals	253	1.72	389

4.3.2 Subsystem Description

The communication system is shown functionally in Fig. 4-4. Primary and secondary communications modes are performed by S-band systems. Four omnidirectional antennas are used and a 4-foot diameter steerable, stowable antenna is used for long-range communication or for high-data-rate transmission. The VHF equipment is the primary voice link from the RNS to the other space vehicles. The VHF system may also be used as a wake-up command. This requires two omni antennas. An emergency transceiver operating at 4-GHz is incorporated and operates in a manner similar to the primary system. Two omnis are also used for this system.

Various network elements (space transportation system, space station, crew, ground or other space communications elements) will be empowered to control the total RNS operations by communication during transit (translunar and transearth) and lunar orbit, as in the earth orbit case. The matrix in Fig. 4-5 shows how the roles change for different elements of the network as the mission phase changes.

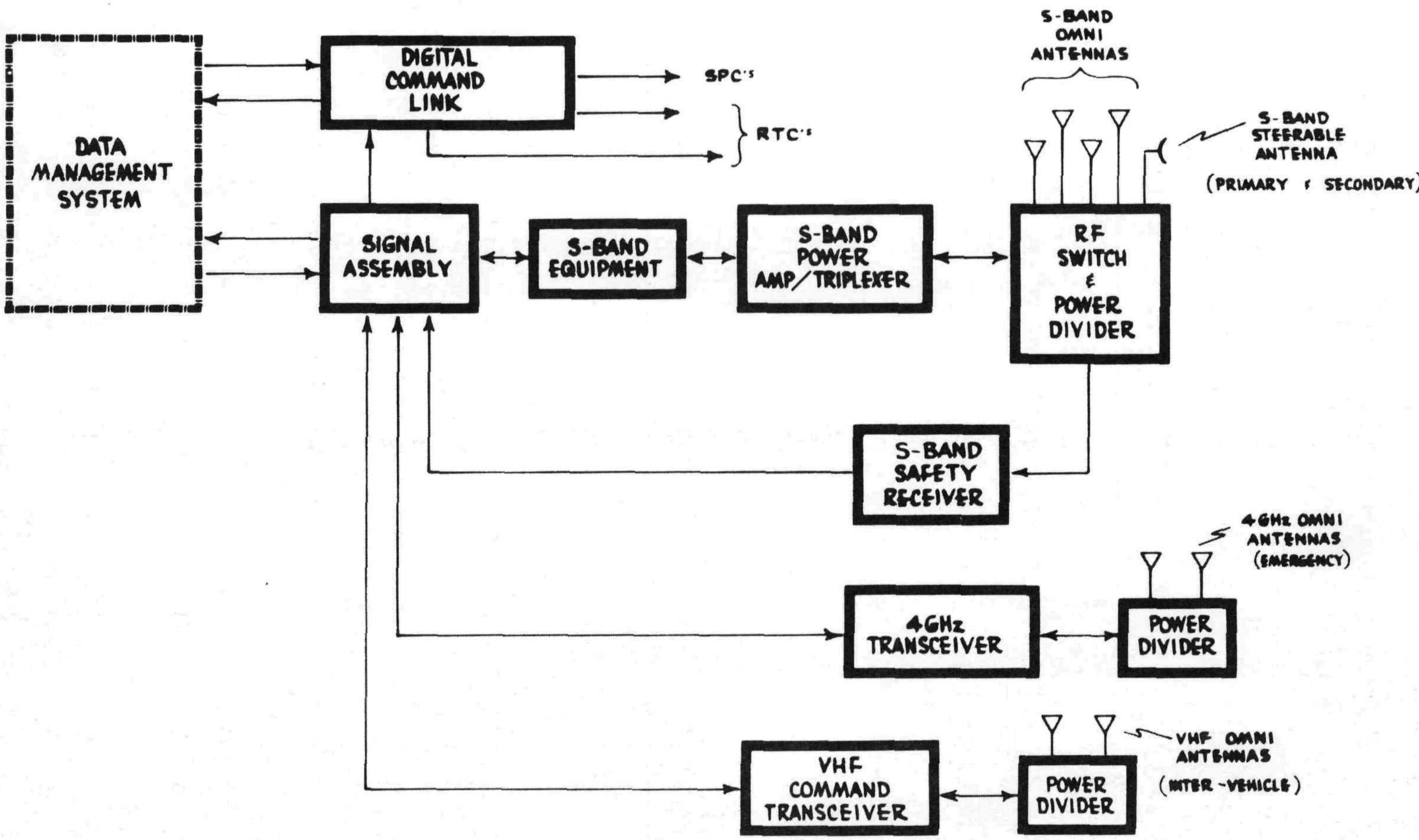

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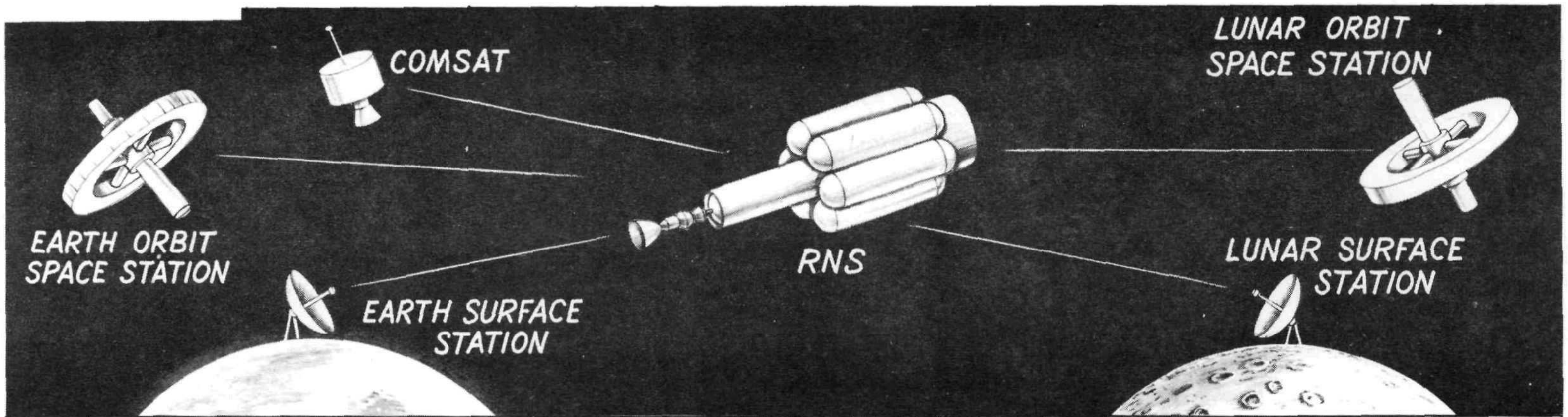
Fig. 4-4 Communication System Schematic



MODAL CONTROL DURING TRANSIT & LUNAR ORBIT PHASES

MISSION PHASE		EARTH SPACE STATION	LUNAR SPACE STATION	LUNAR ORBIT TUG	EARTH SURFACE STATION	LUNAR SURFACE STATION	RNS	EVA	COMSAT
TRANSIT	TRANSLUNAR	SC / SCB	SCB / SC	P	PC / PCB	PCB / PC	T	P	CO
TRANSIT	TRANSEARTH	SCB / SC	SC / SCB	P	PCB / PC	PC / PCB	T	P	CO
LUNAR ORBIT		SCB	SC	T	PCB	PC	T	T	CO

KEY:
- PC PRIMARY CONTROL
- PCB PRIMARY CONTROL/BACKUP
- SC SECONDARY CONTROL
- SCB SECONDARY CONTROL/BACKUP
- T TERTIARY
- P PASSIVE
- CO CARRIER ONLY

Fig. 4-5 Modal Control During Transit and Lunar Orbit Phases

4.3.3 Subsystem Characteristics

The communication system has a total weight (7.03) of 254 lb. Detailed weight, volume, and power requirements are listed in Table 4-2. The communication system is located primarily in the equipment module. It is desirable to locate all communications equipment handling rf as close to the applicable antennas as possible to avoid long rf cables resulting in large rf power losses.

All antennas must be located near the top of the spaceframe as noted below:

- 4 S-band omni antennas 90 deg apart on the periphery of the spaceframe.
- 2 vhf omni antennas 180 deg apart on the periphery of the spaceframe.
- 2 4-GHz omni antennas 180 deg apart on the periphery of the spaceframe.
- Single steerable S-band antenna must be located at the top of the spaceframe and 90 deg perpendicular to the longitudinal axis of the RNS and must be mounted so as to extend and retract when required.

Table 4-2

RNS COMMUNICATIONS SUBSYSTEM CHARACTERISTICS

	WT (lb)	VOL. (ft^3)	POWER (watts)
Digital Command Unit	34	0.3	7
Signal Assembly	11.3	0.2	13
S-Band Equipment	38	0.6	3
S-Band Pwr Amp/Triplexer	32	0.5	90
S-Band RF Switch	5	0.01	2
S-Band Omni Antenna	10	0.05	–
S-Band Steerable Antenna	75	2.0	10
Vhf Omni Antenna	0.5	0.02	–
Vhf Pwr Divider	0.5	0.02	–
Vhf Dual Command Rcvr	10	0.2	40
4 GHz Data Relay Satellite XCVR	30	0.5	25
4 GHz Ant. & Pwr Divider	2	0.02	–
S-Band Safety Receiver	6	0.02	1
Totals	254.3	4.44	191

4.4 NAVIGATION, GUIDANCE & CONTROL SYSTEM

4.4.1 Subsystem Requirements

The primary function of the navigation, guidance, and control system is to (1) determine the vehicle position and velocity from on-board inertial measurements, (2) steer the vehicle in accordance with the guidance equations and (3) execute control commands from the guidance system for the thrust vector control of the main engine and from the autopilot for attitude control and stabilization. In addition, for rendezvous and docking a rendezvous and docking sensor system is required.

4.4.2 Subsystem Description

Function. This system is an explicit inertial navigation and guidance system operating in a closed-loop strapdown configuration. The system provides steering signals, attitude control commands, and vernier translation commands during the mission. In addition, the system generates real-time discrete signals for engine start and cutoff and provides control of the limited variation in occurrence and duration of cooling pulses.

The complete system consists of the inertial sensor assembly (ISA) and the central computer (general purpose). In addition, the system contains the following mission phase-peculiar equipment:

- Transceiver and beacon for DSIF update
- Sun sensor
- Star tracker
- Earth/lunar horizon tracker
- Radar altimeter
- Rendezvous and docking sensor
- Rate gyro package

The baseline strapdown system (ISA and computer) is augmented by the mission phase-peculiar equipment. The inertial system can be updated by DSIF to improve the onboard estimate of the state vector and by the on-board celestial sensors (sun sensor and star tracker) to update the attitude estimate. Figure 4-6 is a functional block diagram of the complete system.

Operational Modes. The basic system recommended for this application is an inertial guidance and navigation system operating in a strapdown configuration. The gyros and accelerometer are, in effect, mounted directly on the vehicle structure in the ISA without gimballing. The system provides steering signals and attitude control commands during the entire mission. In addition, the system generates real-time discrete signals for events such as engine start and docking maneuvers. Operation of specific areas follows:

- Inertial Sensor Assembly. The ISA senses vehicle rotation about the vehicle axes and acceleration along these axes. The sensors are restrained near null by the pulse rebalance electronics. The guidance computer receives the outputs from the sensors and continuously computes attitude, velocity and position. A typical ISA consists of three single-degree-of-freedom integrating gyros, three fluid damped flexure pivot-type accelerometers, pulse rebalance electronics, and precision timing generator
- SDF Integrating Gyros. A typical gyro for this application is an integrating, floated gyro utilizing a hydro-dynamic gas bearing spin motor. This device has no physical spring restraint but the viscous damping time is large. By closing the loop from the signal generator output to the gyro torquer, via the pulse rebalance electronics, an "electrical" spring, equivalent to the spring constant, is introduced. Viscous damping of the gimbal is provided by the shear forces developed by the floatation fluid. Gimbal freedom is restricted to less than ±1 degree, which decreases flexure lead torque and improves the g-insensitive drift stability. The gyro is operated near null and has a very low characteristic time (0.5 msec), which reduces drift caused by angular vibration.
- Accelerometers. A typical accelerometer for this application is a miniature hinged pendulum accelerometer of the flexure pivot type with fluid damping. The pendulous mass is supported on a pair of flexure pivots which deflect freely around one axis (the pivot axis) but are highly resistant to displacement about the orbit axes. Torque rebalance is accomplished by two torquers operating in a "push-pull" mode. The spring restraint is small and the accelerometer operates in a tight closed loop. Viscous damping is provided by the fluid which surrounds the pendulum. Unlike the "floated" gyro, this fluid is not used for support, and temperature control is not as critical.
- Guidance Computer (GC). This computer would typically be part of a general purpose computer programmed to provide a strapdown algorithm. This special algorithm is required to process the incremental ISA output signals to obtain the incremental rotation of the direction cosine (attitude) matrix and the velocity

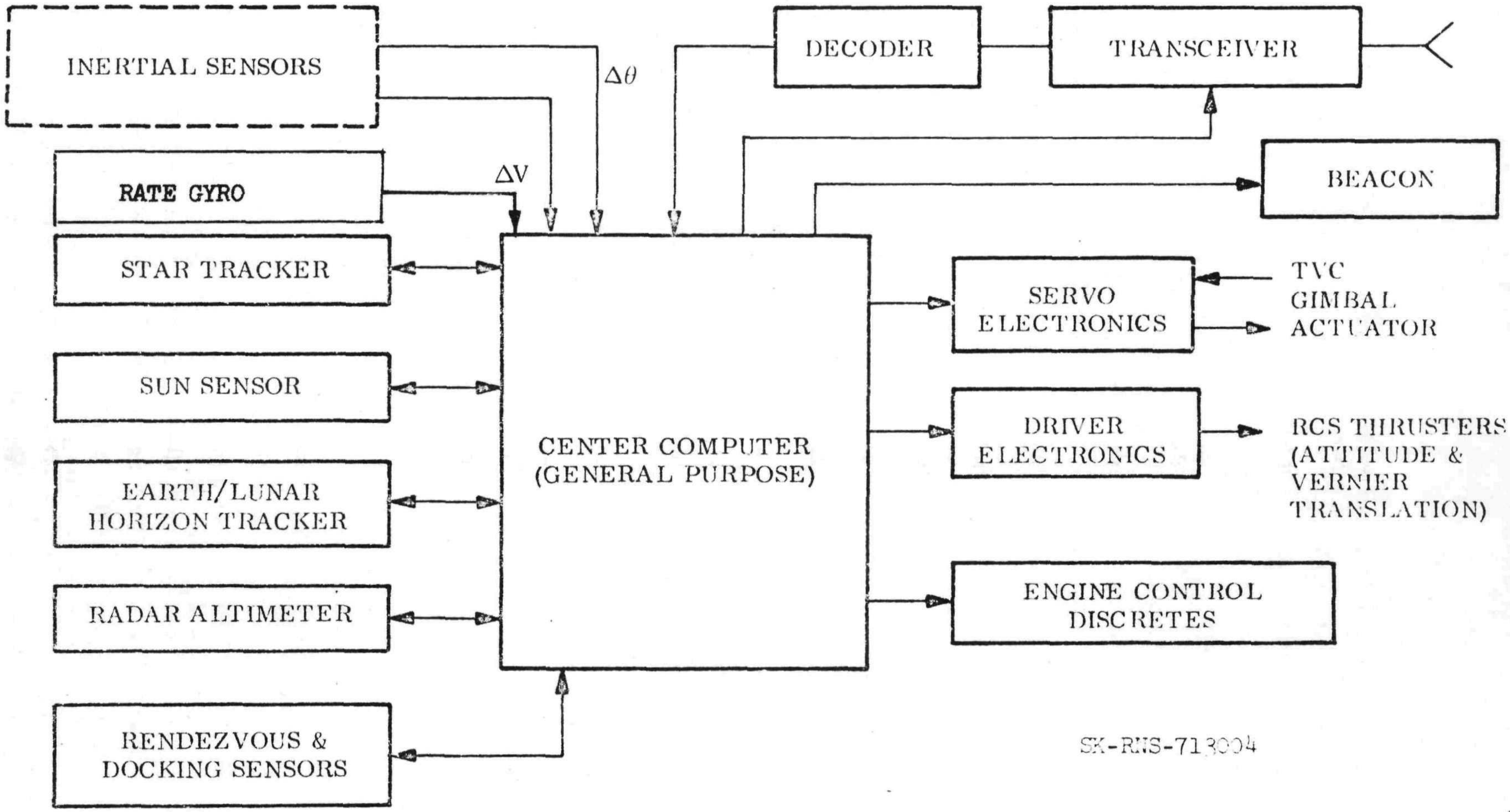


Fig. 4-6 Navigation, Guidance & Control System

vector transformation from the body frame to the desired navigation frame. A first-order algorithm is as follows:

- Input into the algorithm is the output of the sensors.
- Sampling rate is high.
- Pulses are processed as they are received.
- The algorithm is reversible, i.e., alternate $+\Delta\theta$ and $-\Delta\theta$ pulses result in no change in the matrix.

Computation characteristics are as follows

- Equations implemented (three sets of direction cosine rate equation)
- Number of cosine registers (nine)
- Sampling rate about 2000 times/sec
- Register length 20 to 24 bits

The first-order algorithm is a numerical approximation because (1) the angular rate information is processed in discrete numerical fashion and (2) truncation, a result of using truncated series approximations of the sine and cosine function of the incremental angle.

The accuracy with which the transformation matrix can be initialized and continuously updated is of major importance to navigational accuracy. The required accuracy can be achieved with recently developed computers and associated algorithms.

Installation. All NG&C equipment, except possibly the rate gyro package, would be located in the forward equipment module. The field of view, along the longitudinal axis with payload in place, appears to be sufficient to cover the required look angles of the optical sensors during search, acquisition and lock-on modes of operation. The rate gyro package(s), if required for structural mode damping in the flight control system, may have to be located outside the equipment module, possibly on the spaceframe, in order to sense angular rates at locations remote from the equipment module.

Temperature Control. In general, the various equipments which make up the NG&C system are available, or can be designed with built-in thermal control provisions.

This approach is recommended, thus minimizing the need for an environmentally controlled compartment for the NG&C equipment.

Rendezvous and Docking Sensor. For the rendezvous and docking phase a scanning laser radar is selected. This system has the following characteristics:

	Optical-Laser (SLR)
Range accuracy (1 nm)	±4 in.
Range (max)	75 nm
Range (min)	~0 ft
Side lobes	-60 to -70 db
System weight (chaser + target)	90 lb
System power	106 W

The weight and power estimates for the two parts are:

	Weight (lb)	Power (W)
Chaser	50	55
Target	40	51

A summary of the functions performed by the SLR includes:

- Three-axis control of chaser spacecraft from acquisition at ~75 nm to docking.
- Measurement of range, range rate, pitch and yaw angles, and angle rates.
- Station-keeping at programmed range.

The data outputs from the system are:

- Chaser vehicle equipment outputs:

  R – slant range chaser to target

  R – range rate

  θ_p, θ_y, θ_p, θ_y – chaser vehicle pitch and yaw line-of-sight angles and angle rates.

  ϕ – chaser-target relative roll index angle.
- Target vehicle equipment outputs: target vehicle pitch and yaw line-of-sight angles and angle rates.

The major subsystems and the key components associated with these subsystems are as follows:

- Laser Transmitter: Diffraction limited gallium arsenide (GaAs) semiconductor laser
- Beam Steerer: Piezoelectric beam deflector and amplifying optics
- Receiver Optics: High-speed, narrowband, refractive receiver optics
- Scanning Optical Detector: Image dissector
- Electronics: High-speed integrated circuits
- Cooperative Target: Optical corner cube reflector

Figure 4-7 is a system block diagram of the SLR system. The SLR chaser and target equipment can be placed on either vehicle. In this discussion, the chaser vehicle is the maneuvering or active vehicle, and the target vehicle is the cooperative non-maneuvering vehicle equipped with an optical corner cube reflector.

The two basic radar functions of target acquisition and tracking are accomplished by scanning the laser transmitter-receiver. The nominal full raster acquisition scan pattern covers 30 by 30 deg in steps of 0.1 deg by 0.1 deg (the instantaneous field of view). Reduced raster scan can be logically programmed to be used, for example, if the navigation data indicate that the target vehicle is within a smaller ± 15 deg field. The target presence signal is activated only when pulses having an amplitude larger than a preset threshold appear within a designated timing gate or range interval. When this occurs, the scanner switches over to the track mode. In the track mode, the instantaneous 0.1 deg by 0.1 deg field of view is deflected in a cross-scan pattern centered on the target. This pattern is driven to any point in the full 30 deg by 30 deg field of view to maintain track while the vehicles are in motion. The angle and angle rate radar data are obtained from the angle scanner electronics. The range measurement is obtained by a precise measurement of the elapsed time for the radar pulse to go out and come back. The range-rate is calculated from successive range readings over precisely known time intervals. After acquisition the chaser and target vehicles continuously track each other. The boresights may be aligned or offset.

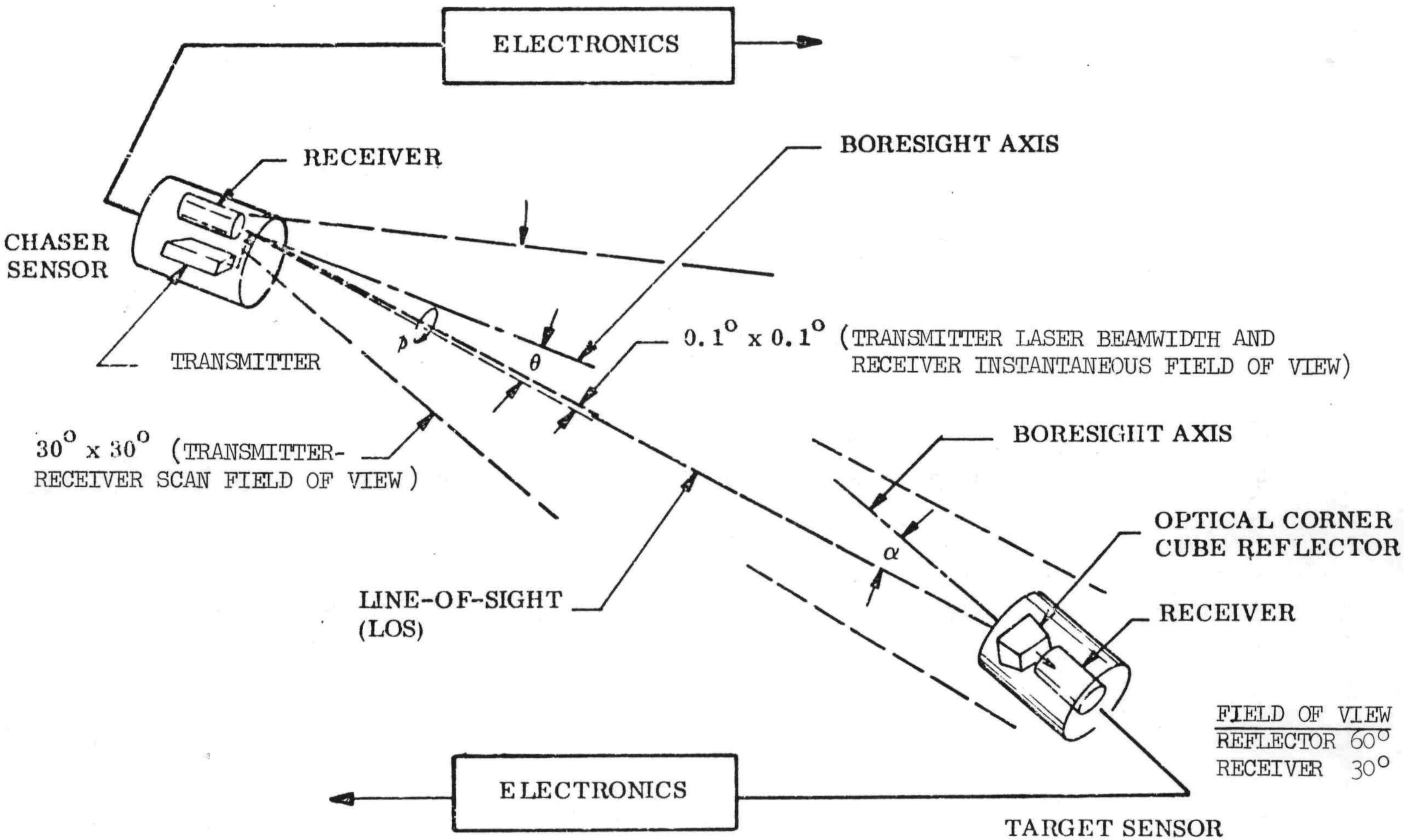


Fig. 4-7 Rendezvous and Docking Laser Radar

4.4.3 Subsystem Characteristics

The weight (7.01), power, and volume characteristics of the NG&C system are shown in Table 4-3.

4.5 POWER SYSTEM

4.5.1 Subsystem Requirements

The power system must provide a power source and distribution system for the RNS, including the NERVA engine. Total power consumption is based on the actual power consumption of each of the subsystems. These values are tabulated in Table 4-4. In addition to peak and average values, the mission phase during which the load occurs is also shown. The actual combination of power demand is shown in a power profile illustrated in Fig. 4-8. This figure specifies the actual power requirements of the RNS, including engine, during vehicle assembly and during the mission. This power profile for the lunar shuttle mission has an average value below 1 KW throughout the mission, with peaks to over 3 KW only during engine burn.

4.5.2 Subsystem Description

The power source consists of solar panels and batteries. These were sized not only according to the power profile but also incorporated battery charging rates, periods between peak power loads, and non-charging periods due to vehicle orientation and occultation of the sun. There are three basic mission phases to consider in establishing the power source size. These are earth and lunar orbit and Earth-Moon flight. The worst case condition from the solar array or battery standpoint is the maximum earth or lunar dark-time. Earth orbit is assumed to be 260 miles circular with a 1.6-hour period, 1 hour in the sun and 0.6 hour in the shade.

Table 4-3

NAVIGATION, GUIDANCE, & CONTROL CHARACTERISTICS

	Weight (lb)	Power (w)	Volume (ft^3)
Inertial Sensor Assembly	40	125	.7
Central Computer *	75	200	1.3
Rate Gyro Package	5	10	0.1
Star Tracker	10	10	0.2
Sun Sensor	15	10	0.2
Earth/Lunar Horizon Tracker	10	5	0.3
Radar Altimeter	31	70	2.2
Optical Radar - Laser (Chaser)	50	55	1.8
Flight Control Electronics	12	10	0.1
Interface Units	48	30	0.1
Total	296	525	7.0

*These characteristics are assignable to NG&C functions.

Table 4-4

POWER CONSUMPTION

	Power Level Watts		Power Profile, Watts				
System	Peak	Average	Assy & C.O.	Burn	Cooldown	Coast	Δ Venting
Engine	3,500	2,200	0	2,200	200 + 400 W pulses	60	
Mechanical	740	100	740*/100**	0	0	0	
Propellant Mgt.	180	120	0	120	30w pulses	0	
Pressurization	300	60	0	60	0	0	
Venting	60	60	0	0	0	0	60
Pneumatics	30	30	0	30	30w pulses	0	
RCS	300	180	90	180	120w pulses		
Power	90	60	90	before and after 60	0	0	
Data Mgt	390	250	250				
Communications	190	100	100				
N.G.& C.	325	175	175				
ECS	50	50	50				
			1315/575	3165	775 + 580 w pulses	635 + 120 w pulses	60

*Four booms transferring a module
**Elect. & Pneu. Interface drive or lock drive

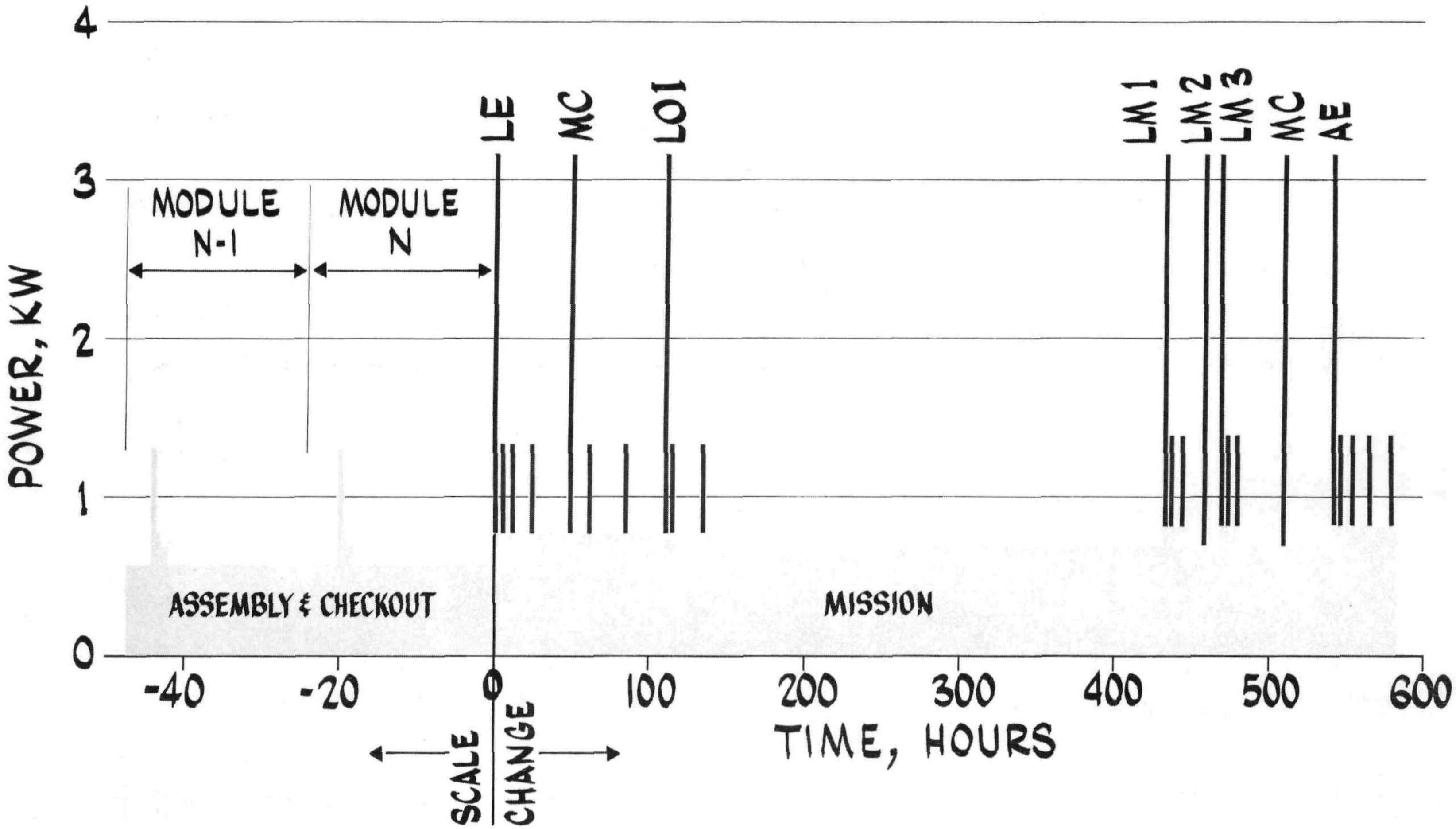


Fig. 4-8 Power Profile – Lunar Mission

The electrical power requirements for the three basic mission phases are as follows:

(1) Lunar orbit:

Energy removed from batteries during shadow	=	620 W-hr
Energy required to charge batteries	=	775 W-hr
Energy required during sunlight	=	920 W-hr
S/A energy per orbit	=	1695 W-hr
S/A orbital average	=	848 W

(2) Earth orbit:

Energy removed from batteries during shade	=	465 W-hr
Energy required to charge batteries	=	580 W-hr
Energy required during sunlight	=	775 W-hr
S/A energy per orbit	=	1355 W-hr
S/A orbital average	=	850 W

(3) Earth-moon flight where the worst case would occur if the first burn coincided with the beginning of an earth shade period.

Energy removed from battery during engine operation coinciding with shade.	=	1,880 W-hr
Energy required for charging batteries	=	2,360 W-hr
Energy required before Burn No. 2	=	55,100 W-hr
S/A average output required	=	805 W

The worst case for the solar array required an orbital average power output of 850 watts. For the earth orbital case, with a S/A effectivity of 0.342 and the LMSC Flex-array capability of 10 W/sq ft, a solar array area of 248 sq ft is required. A total of 1,880 W-hr of capacity could be removed from the batteries in the worst case. With a maximum allowable depth-of-discharge of 50 percent, battery weight = 380 lb.

The selected solar cells are 2b by 2 cm, N/P, 2 SL-cm silicon, 0.008 inch thick. The batteries are nickel-cadmium.

The system must provide the following functions:

- Mechanical. The solar arrays must be of a retractable design and sun oriented.
- Instrumentation. Voltage at critical points of the system; current from each solar array segment (the number to be defined) and to and from each battery; ampere-hours delivered to loads; battery temperatures; and solar array temperatures.
- Command Functions. The most critical command for the system is to extend or retract the solar arrays. It should also be noted that a minimum of ten minutes should be allowed for retraction, while only a few minutes might be required to extend.

Operational Modes and Sequences. The solar arrays may be in a retracted-nonoperational, extended-operational or extended-nonoperational (shaded) node. A battery may be discharging, charging or open-circuited in reserve. The normal sequence of events is conceived as follows:

- As soon as earth orbit is achieved the solar arrays would be extended and the vehicle would be placed in a functional mode with batteries being charged by the arrays; when the vehicle passes into the earth's eclipse the batteries will assume all of the load.
- During earth-moon flight,vehicle orientation and solar array tracking will be used to maximize array electrical output.
- When lunar orbit is achieved, batteries will again have to provide all electrical power during moon's eclipse.

Environmental Requirements. The thermal design will have to provide a range of from 30 to 50^{o}F for the nickel-cadmium batteries. No special requirements are anticipated for the other electrical components.

Installation Requirements. The solar arrays must be mounted off of the avionics equipment rack and contained therein until they can be deployed. Each flexible solar array would be packaged in a flat box approximately 7 by 1.5 by 0.5 ft containing 125 ft^2 of active solar array area, mounted on folded lightweight trusses which would take the array out approximately 17 feet before deployment, to clear the worst case payload interference. Provisions are required for orienting the arrays toward the sun. Based on a preliminary analysis of the vehicle orbital sun relationships a single axis of tracking is adequate to provide effective use of the solar energy. Multiple axis tracking was selected, however, for mission flexibility. In addition, thermal constraints require preferential vehicle/sun relationships. Solar array assembly and installation are shown in Fig. 4-9.

A schematic of the power system is shown in Fig. 4-10. The system includes two solar panels in parallel and two battery packs with their controllers. These sources feed an unregulated bus that traverses the length of the vehicle. There are junction boxes for the astrionics system in the equipment bay of the space frame, and for each of the propellant and propulsion modules. There is also a junction box for the engine.

4.5.3 Subsystem Characteristics

The total weight of the power system and distribution system is shown in Table 4-5. This summary does not include 10 lb in the propulsion module and 5 lb in each propellant module for power distribution.

Table 4-5

ELECTRICAL POWER AND DISTRIBUTION SYSTEM WEIGHTS

System	Weight (lb)	
Power System (7.04)		
Solar arrays, extension and retraction hardware and voltage controllers.	150	
Batteries	380	
Battery charge controllers	30	
	560	
Distribution System (7.05)		
Current sensors, 2 at 1.5 lb	3	
Ampere-hour meter	7	
Junction boxes, 3 at 10 lb	30	
Cabling	100	
	140	
Total Weight		700

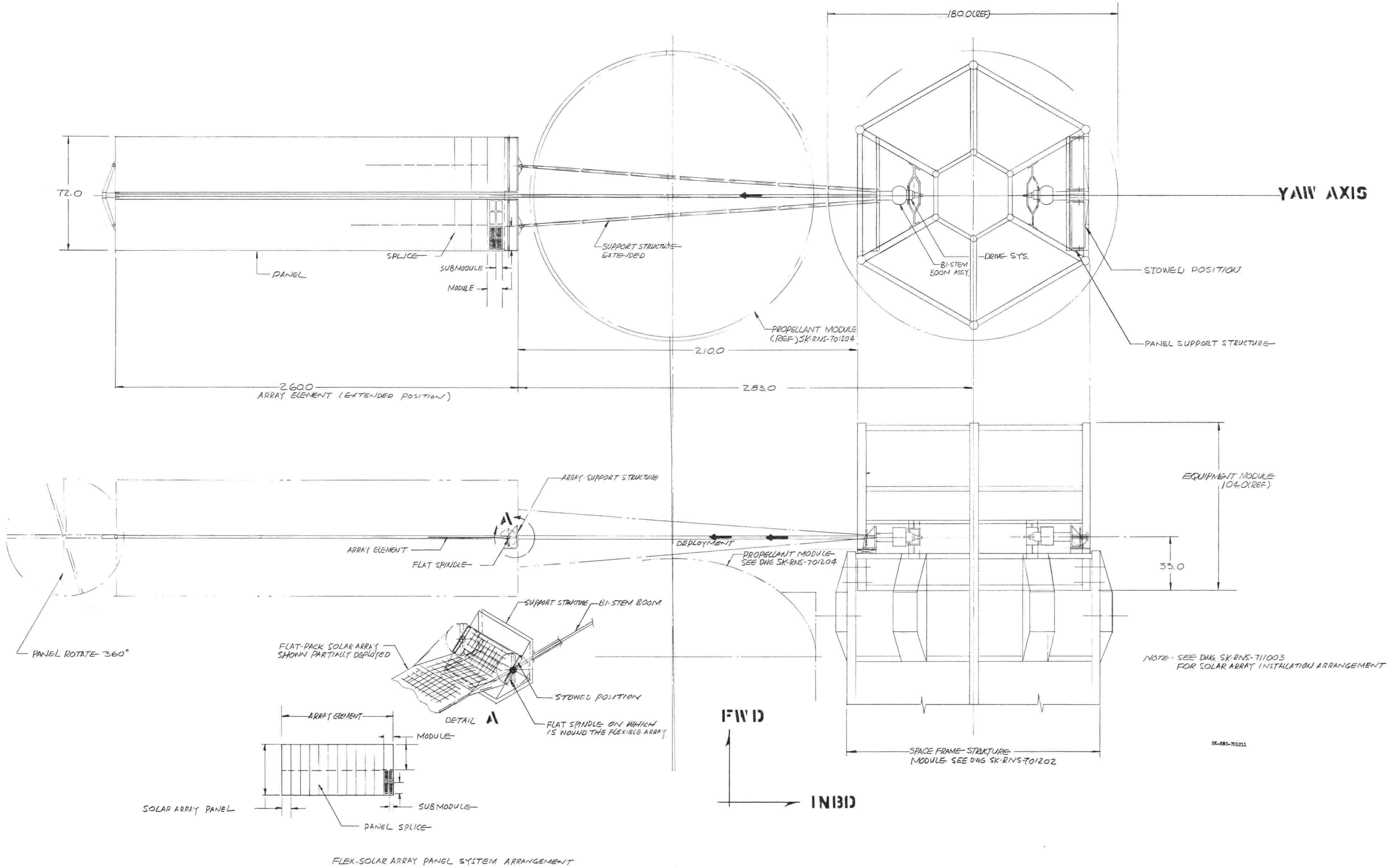


Fig. 4-9 Solar Array Assembly and Installation

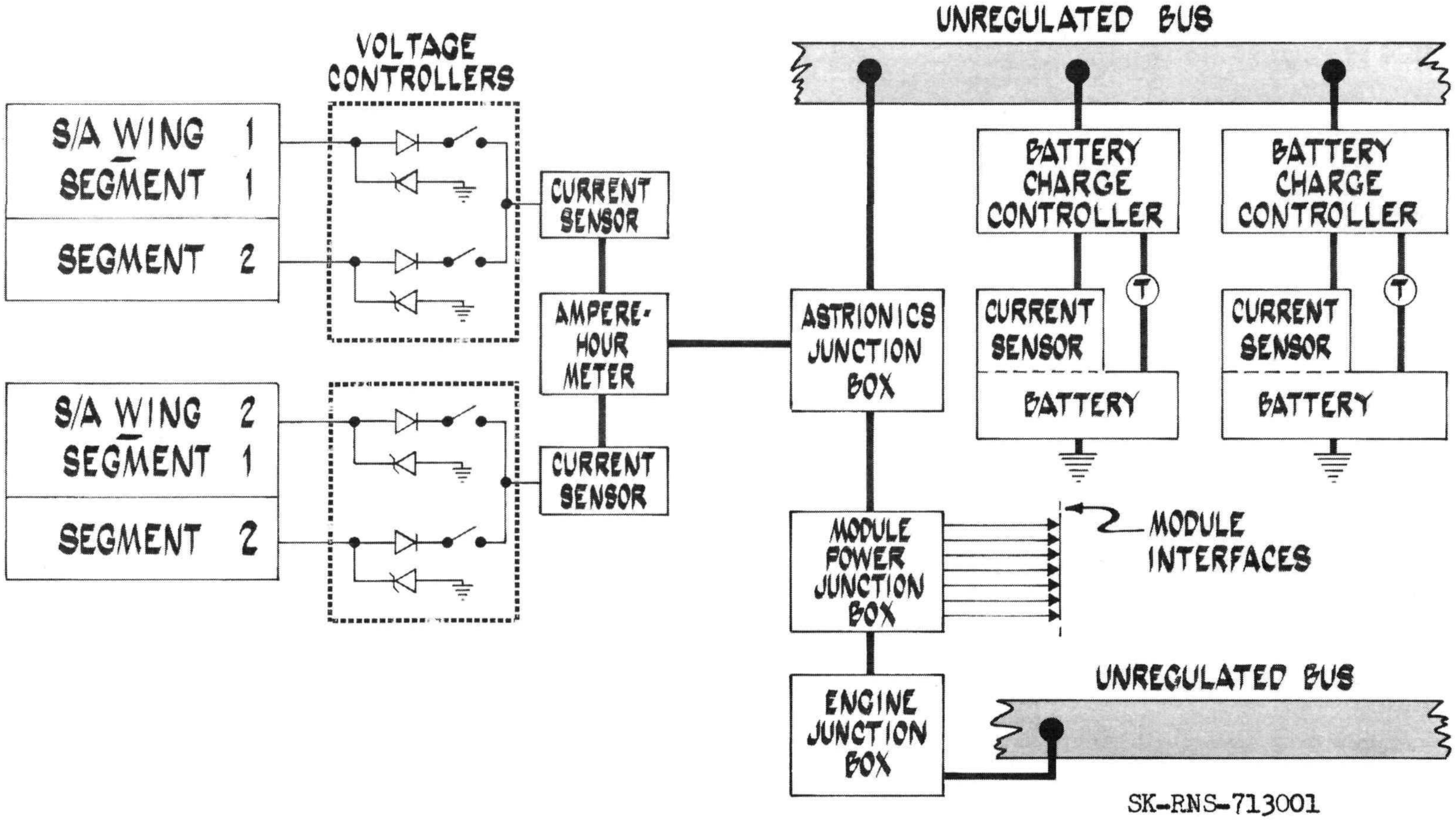


Fig. 4-10 Electrical Power System Schematic

4.6 ENVIRONMENTAL CONTROL SUBSYSTEM

4.6.1 Subsystem Requirements

The environmental control subsystem rejects internally generated heat and maintains equipment temperatures within required limits during all mission phases. Internal heat loads vary due to duty cycle conditions and external environmental heating rates vary due to changing orientation relative to earth, moon,and sun. Temperature limits and internal heat loads are shown in Table 4-6 for all heat generating components in the equipment module.

Since vehicle configuration and orientation have a significant influence on external environmental heating rates, these conditions were examined. Figures 4-11 and 4-12 show estimated external incident heating rates for a large diameter payload configuration shown in Fig. 4-13. These data are based on the assumption that the payload is oriented toward the sun so that there is no direct solar incidence on the equipment module. The equipment module and payload operate at nearly the same temperature level. The large temperature difference between the equipment module and LH_2 tanks necessitates substantial thermal isolation which can be accomplished primarily with an insulation blanket or radiation shield as shown in Fig. 4-13. Structural heat transfer from the equipment to the LH_2 tank area will be limited as required by controlled conductance mountings. The relatively constant equipment duty cycle (750 watts minimum, 1080 watts maximum) minimizes the thermal control problem and makes a nearly passive approach feasible.

4.6.2 Subsystem Description

The passive/heater system was selected because of its simplicity, low cost, ability to meet requirements, and extensive flight history success. In this system the radiating area required to reject internally generated heat when also subjected to external heating (Figs. 4-11 and 4-12) is shown in Figs. 4-14 and 4-15. The external module surface finish assumed was fluorinated ethylene propylene (FEP) Teflon/Aluminum which results in $\alpha_s/\epsilon = 0.2$ and is quite stable under long-term UV exposure. More radiating

Table 4-6

EQUIPMENT POWER AND TEMPERATURE LIMITS

COMPONENT	POWER (Watts)	WEIGHT (Lbs.)	TEMPERATURE LIMITS (of)
Data Management -			
Computer	168	118	30 to 135
Remote I/O	60	20	30 to 135
Central Bus Mod.	70	18	30 to 135
Pwr Unit	40	17	30 to 135
Data Storage	51	40	30 to 135
Communications -			
Digital Command	7	34	30 to 135
Signal Assy	13	11.3	30 to 135
S-Band Eq.	3	38	30 to 135
Pwr Amp/Triplex	90	32	30 to 135
R.F. Switch	2	5	30 to 135
S-Band St. Ant.	10	75	30 to 135
VHF Dual Rcvr	40	10	30 to 135
Data Relay Xcvr	25	30	30 to 135
Safety Rec.	1	6	30 to 135
Nav. Guid & Control -			
Iner. Sen. Assy	125	40	30 to 135
Cent. Computer	200	75	30 to 135
Star Tracker	10	10	30 to 135
Sun Sensor	10	15	30 to 135
Horiz Tracker	5	10	30 to 135
Radar Altim	70	31	30 to 135
Optical Radar	55	50	30 to 135
Flt.Cont.Elec	10	12	30 to 135
Interface Unit	30	48	30 to 135
Power:			
Ni-Cd Battery	50(max)	100(ea)	30 to 60

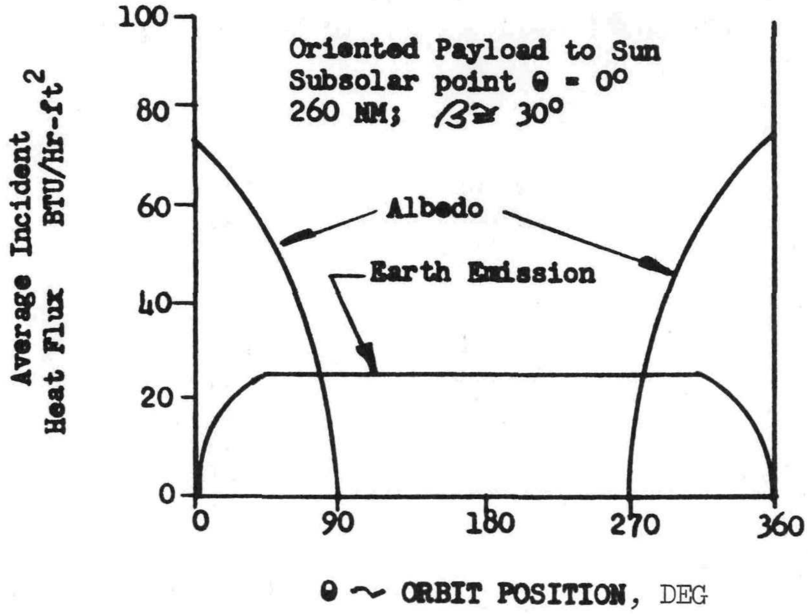


Fig. 4-11 Typical Earth Orbit External Heating Rates – Equipment Module

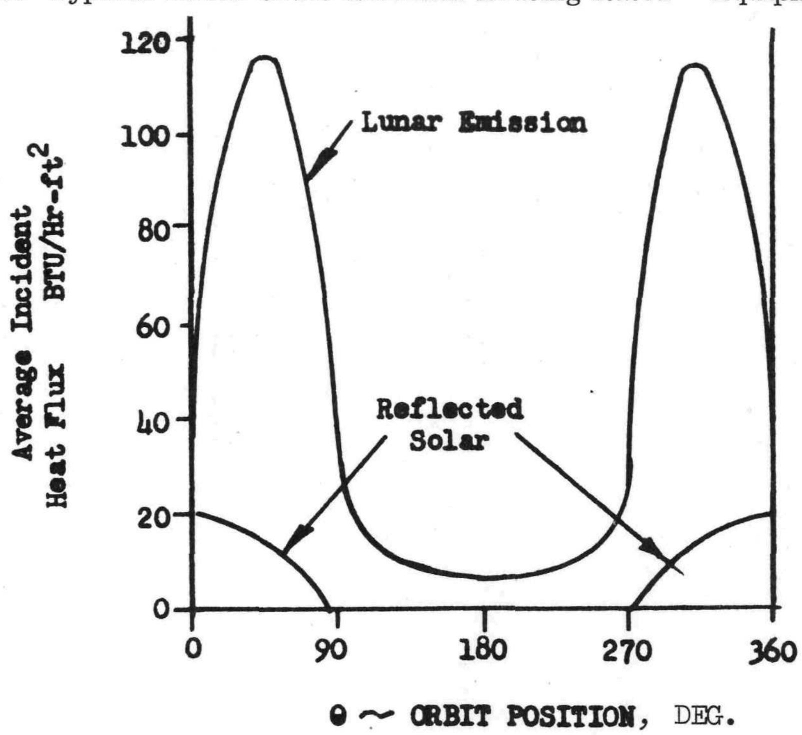


Fig. 4-12 Typical Lunar Orbit External Heating Rates – Equipment Module

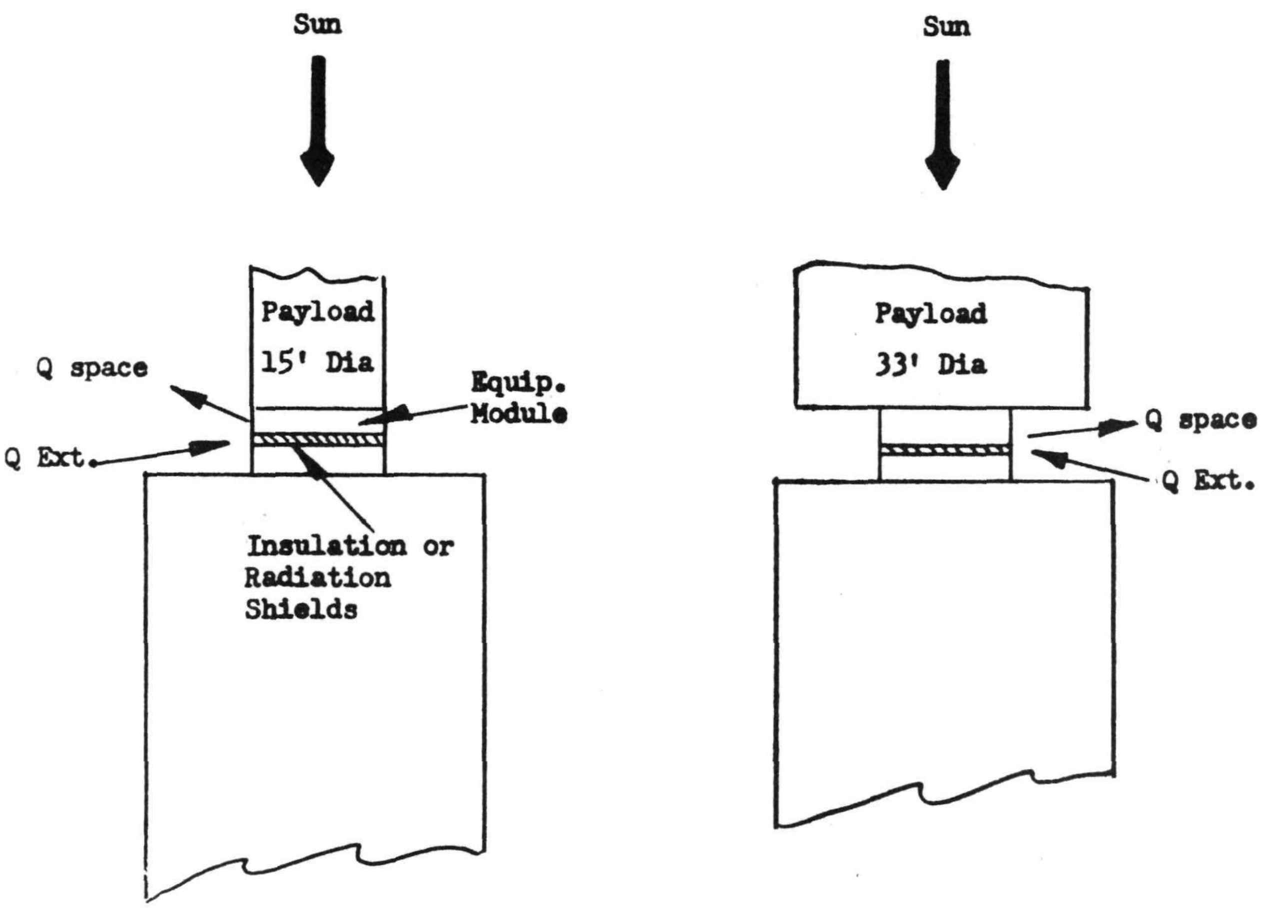


Fig. 4-13 Payload/Equipment Module Interface

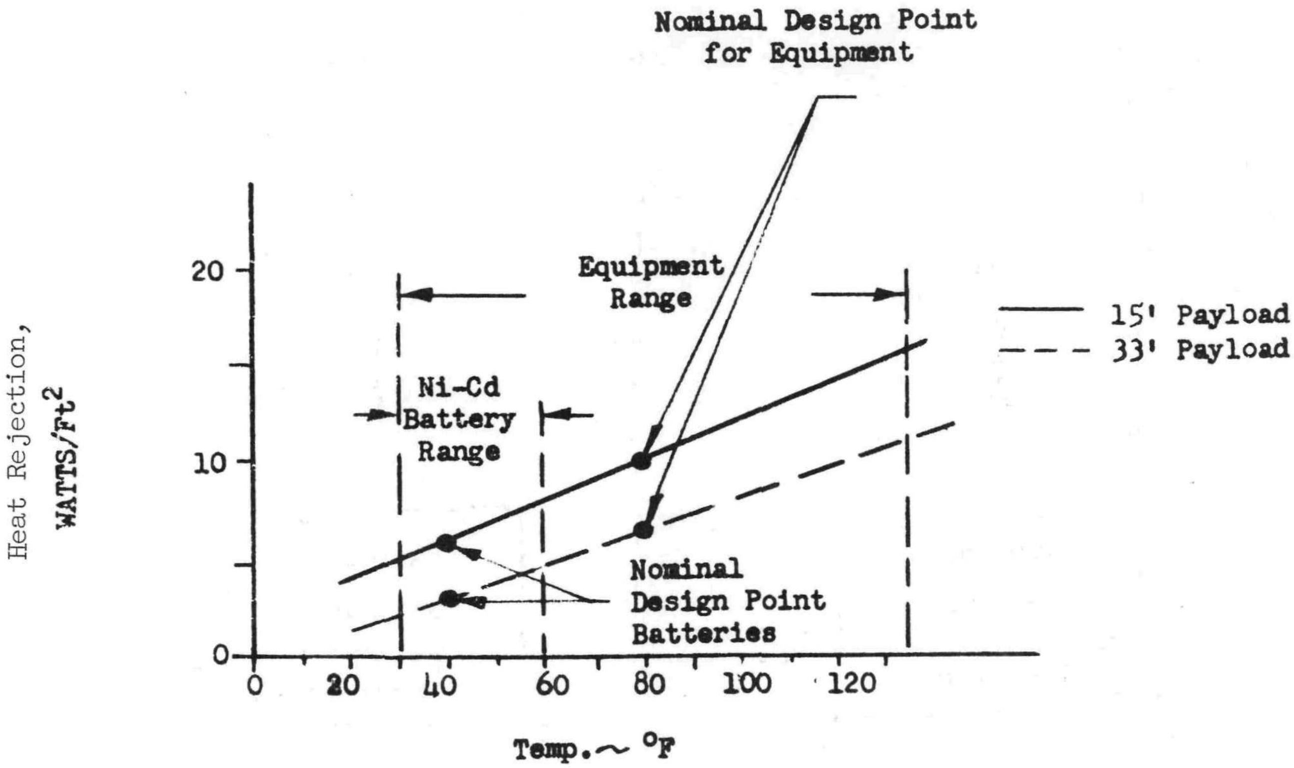


Fig. 4-14 Heat Rejection as a Function of Radiating Surface Temperature

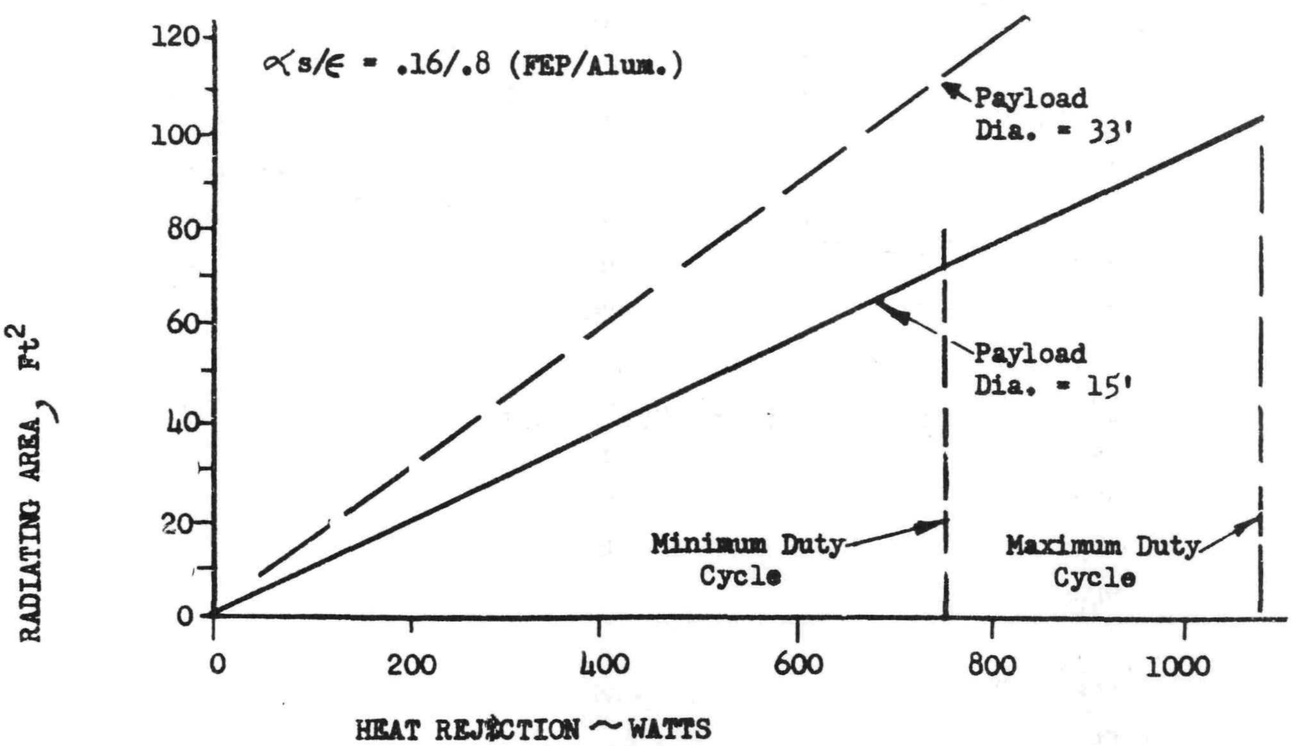


Fig. 4-15 Radiating Area Requirements

area is required with the large diameter payload because the module view to space is reduced. External module heating for the two payload diameters is about the same.

To maintain batteries within 30 to 60^{O}F requires a maximum of 50 watts for heater power. The battery thermal design would provide for adequate radiating area to prevent battery temperatures in excess of 60^{O}F under maximum heat load. The heater power is required to prevent battery temperatures from falling below 30^{O}F under conditions of minimum heating. Heater power also provides a means for compensating for inherent tolerances.

Basically the system provides for heat rejection from external radiating surfaces so that maximum equipment temperatures are not exceeded under conditions of maximum heat loads. Under conditions of minimum heat loads, heat is added via electrical resistance heaters when minimum allowable component temperatures are reached. Sensing of the component temperature and control of heater voltage is accomplished using a thermistor sensing and electronic control device. The system includes no mechanical thermostats. Insulation and materials are selected to control heat transfer between parts of the vehicle and from the vehicle to space within desired limits.

Operational Modes and Sequences. Operation of the environmental control system is continuous, and any time battery temperatures approach their lower limit heat is added. Average equipment component temperatures will be a maximum in lunar orbit, slightly lower in earth orbit and lowest during translunar coast. It is likely that the only time that heater power will be required is during the trans-lunar phase.

Installation Requirements. External thermal control surfaces, including paints, are applied as in conventional spacecraft. After special coatings are applied, appropriate procedures for surface protection must be adhered to during manufacturing, test, and prelaunch checkout. Such procedures have been established and are handled in routine manner.

Multilayer insulation blankets will be manufactured, installed, and attached through use of proven techniques now in use.

Heaters are installed on components or on mounting base plates using conventional bonding techniques (heaters are commonly silicone rubber with imbedded electrical resistors).

Environmental Requirements. The environmental control system senses and controls the module thermal environment. Elements of this subsystem are qualified to withstand launch and orbit induced dynamic environments.

4.6.3 Subsystem Characteristics

The passive/heater system for environmental control has minimal weight requirements. The system consists primarily of heaters, sensors, control elements, and surface coatings. The weight (7.06) allocation for these elements is five pounds for each propellant and propulsion module and twenty pounds for the space frame module.

Section 5
VEHICLE DEFINITION

The baseline RNS, as defined at the third level of the Work Breakdown Structure, consists of one spaceframe module, one propulsion module, and six propellant modules. These modules, in turn, consist of structural elements, propulsion elements, and astrionic elements. The functional integration of these elements into the RNS is shown in Fig. 5-1. The various subsections of Section 5 will define the three modules, their integration into the RNS, the functional interaction of the subsystems across the interfaces, the mass properties of the modules and RNS from launch to mission completion, and the environment the RNS experiences.

5.1 CONFIGURATION

The vehicle configuration is defined first at the module level and then at the RNS level by means of inboard profile drawings. In these drawings the structural, propulsion, and astrionic elements are identified.

5.1.1 Spaceframe

The spaceframe is the core element of the RNS and is completely autonomous after it is deployed. It consists of structural, propulsion, and astrionic system elements that allow it to maneuver, provide station keeping, generate power, communicate, perform checkout procedures, etc. The inboard profile of the spaceframe is shown in Fig. 5-2. The basic module serves as thrust structure, propellant module clustering structure, propellant module fluid distribution system, payload docking structure, orbit refueling docking structure, and astrionic support structure. The module is totally manufactured, assembled, and tested on the ground. The module is then delivered to earth orbit as a unit in the cargo bay of the Space Shuttle. The module is 15 by 60 ft in the stowed position, and 15 by 67 ft with the equipment module extended. The equipment module is extended by means of two internal bi-stem

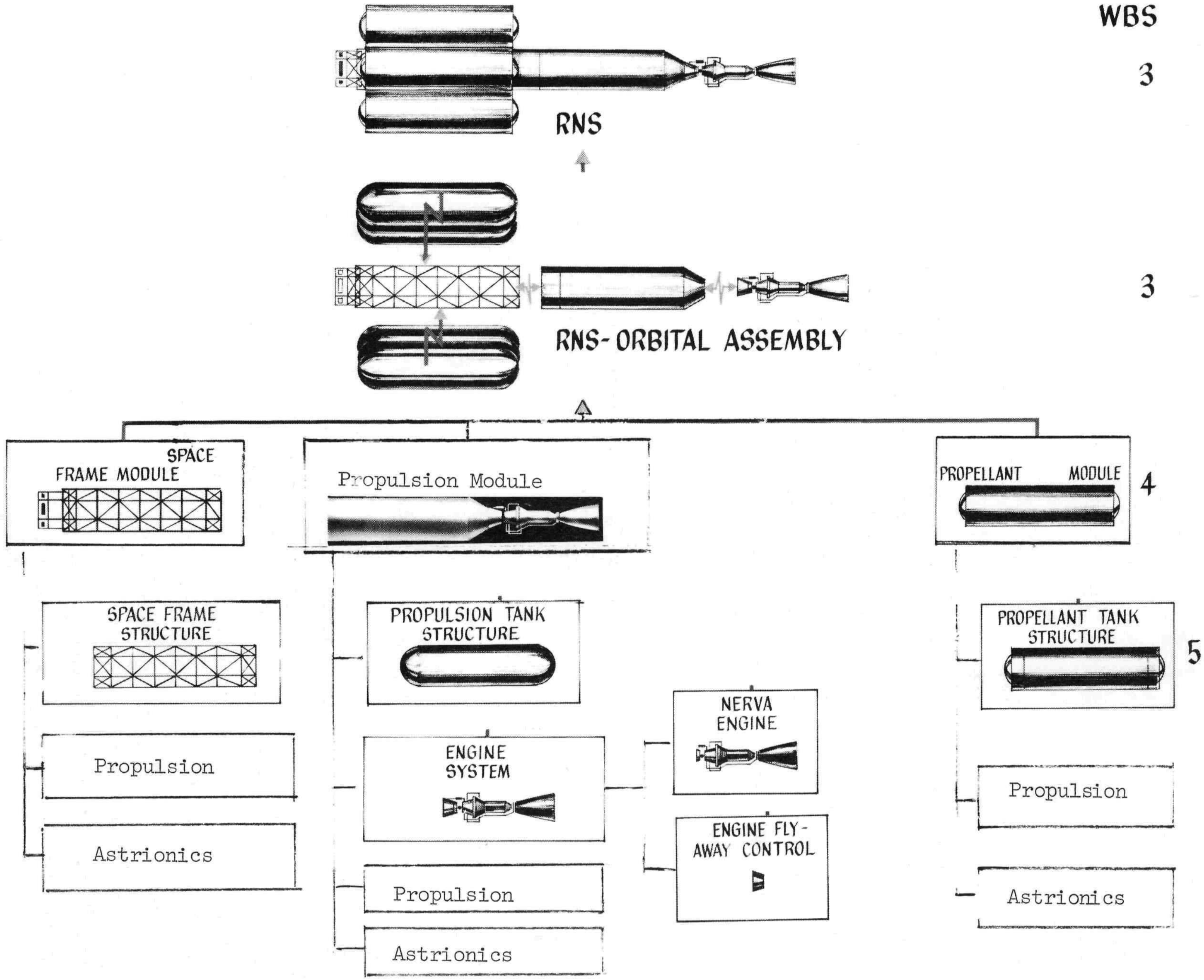


Fig. 5-1 RNS Integration

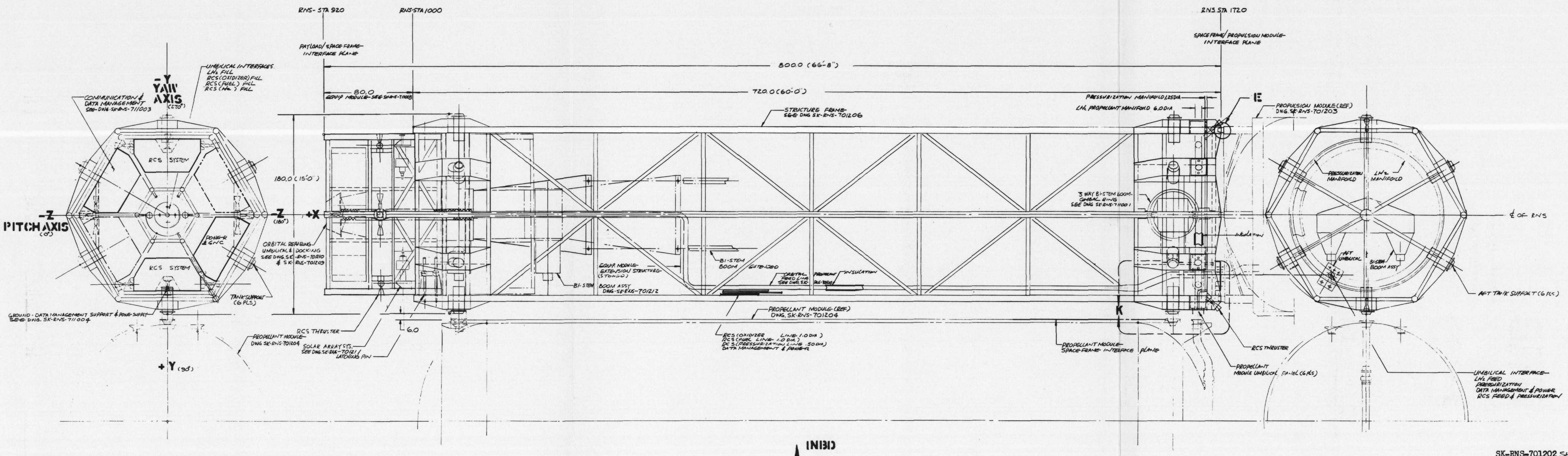


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Fig. 5-2 Spaceframe Module Inboard Profile

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booms which drive the equipment module forward on twelve contoured rollers, two for each pair of longerons.

The equipment module contains the power system; RCS system; environmental control system; navigation guidance, and control system; communication system; and data management system. The allocation of space within the equipment module is accomplished by bay assignments. Bay numbers 1 and 4 contain the RCS hardware; bay 3 communication and data management; and bay 6 power and navigation, guidance, and control. These allocations, shown in Fig. 5-3, also leave bays 2 and 5 unfilled. The availability of these bays is part of the maintainability philosophy, since all of the equipment in each module is replaceable as a package. The interfaces between the equipment in modules 2 and 5 and the spaceframe are identical to 3 and 6, respectively, so that the replacement unit can be installed prior to removal of the used unit and thereby maintain continuity of function. The specific location of equipment within each bay is shown in Fig. 5-4. The replacement of the equipment can be accomplished by various means as shown in Fig. 5-5. The module is also equipped with docking devices as well as propellant and RCS propellant transfer coupling devices. Vehicle orbital assembly equipment (bi-stem boom) is installed below the equipment module structure. Propellant module supports are located in forward and aft ends of the spaceframe. The six umbilical panels provide connections for the data management, propellant feed, pressurization and power from the tank modules to the spaceframe. The propellant feed manifold interconnects to the propulsion module through an aft umbilical plate. Details of the propulsion module and propellant module interfaces with the spaceframe are shown in Fig. 5-6.

5.1.2 Propulsion Module

The propulsion module consists of the NERVA engine, flyaway engine control module, a propellant tank and supporting systems, propellant, and meteoroid and thermal protection systems. It also incorporates ground and emergency venting, flight venting, pressurization system, propellant feed system, propellant conditioning system, purge system, data management system, environmental control system, electrical power, and instrumentation. The propulsion module also includes forward and aft skirts and an external fairing for the subsystem control wires. The forward docking ring

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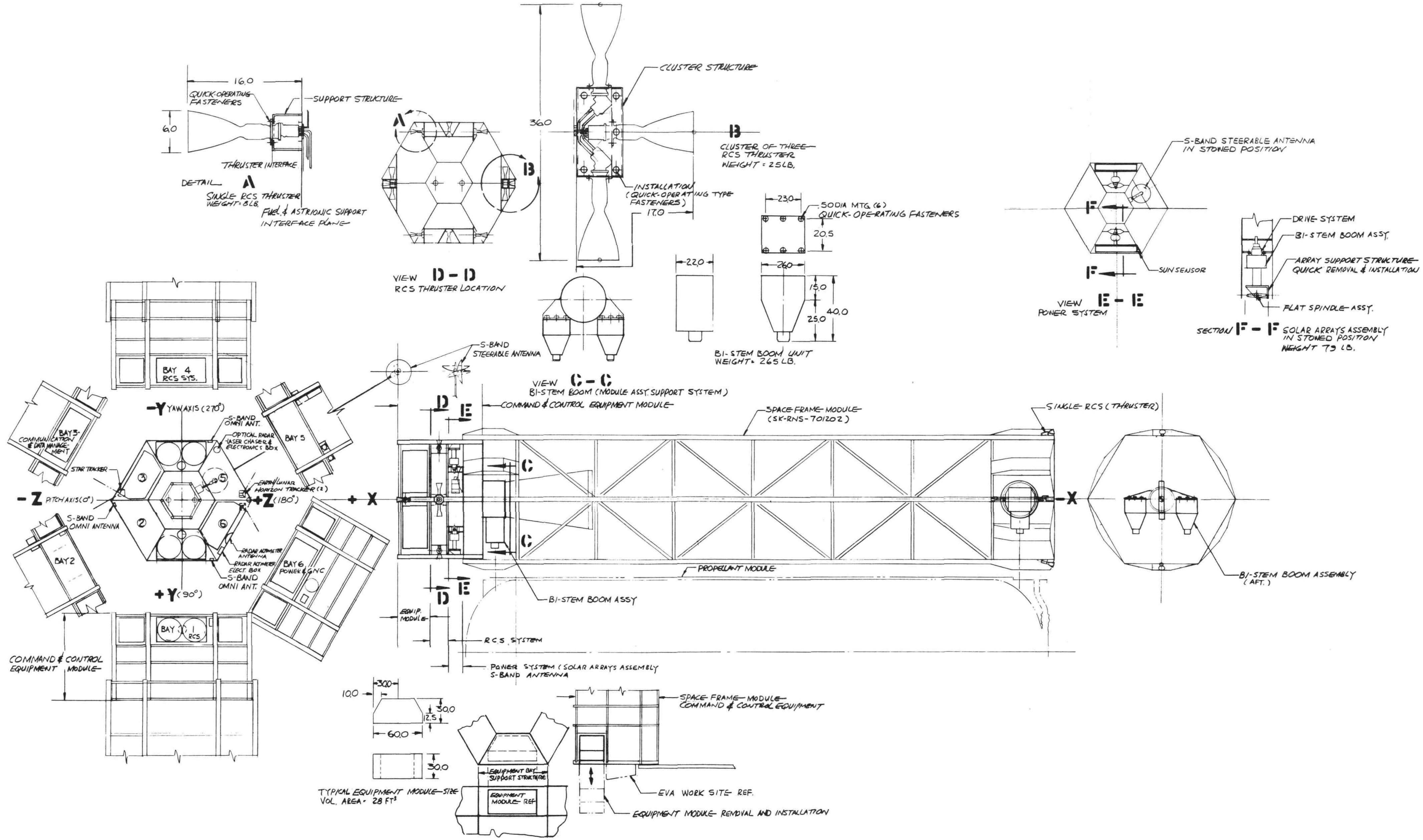

Fig. 5-3 RNS Spaceframe Equipment Arrangement – General Arrangement

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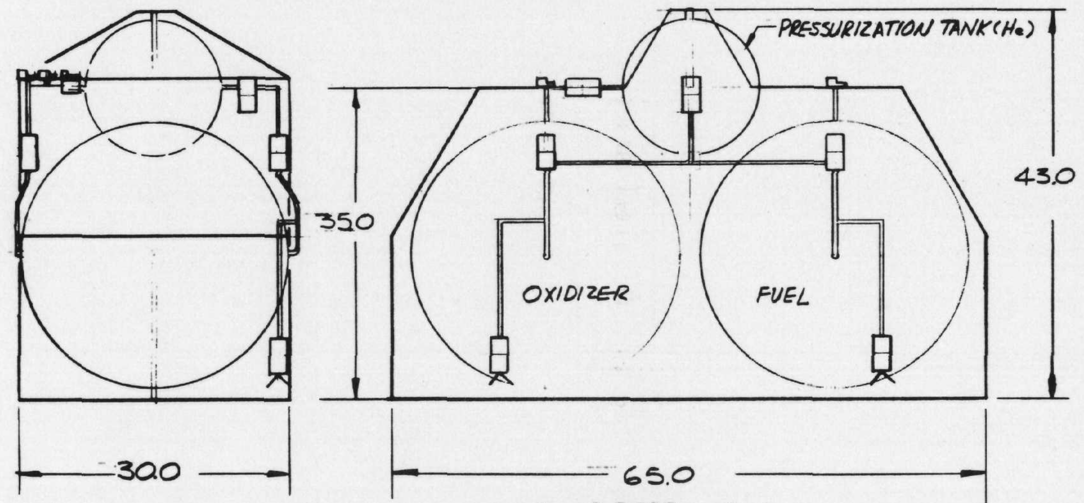
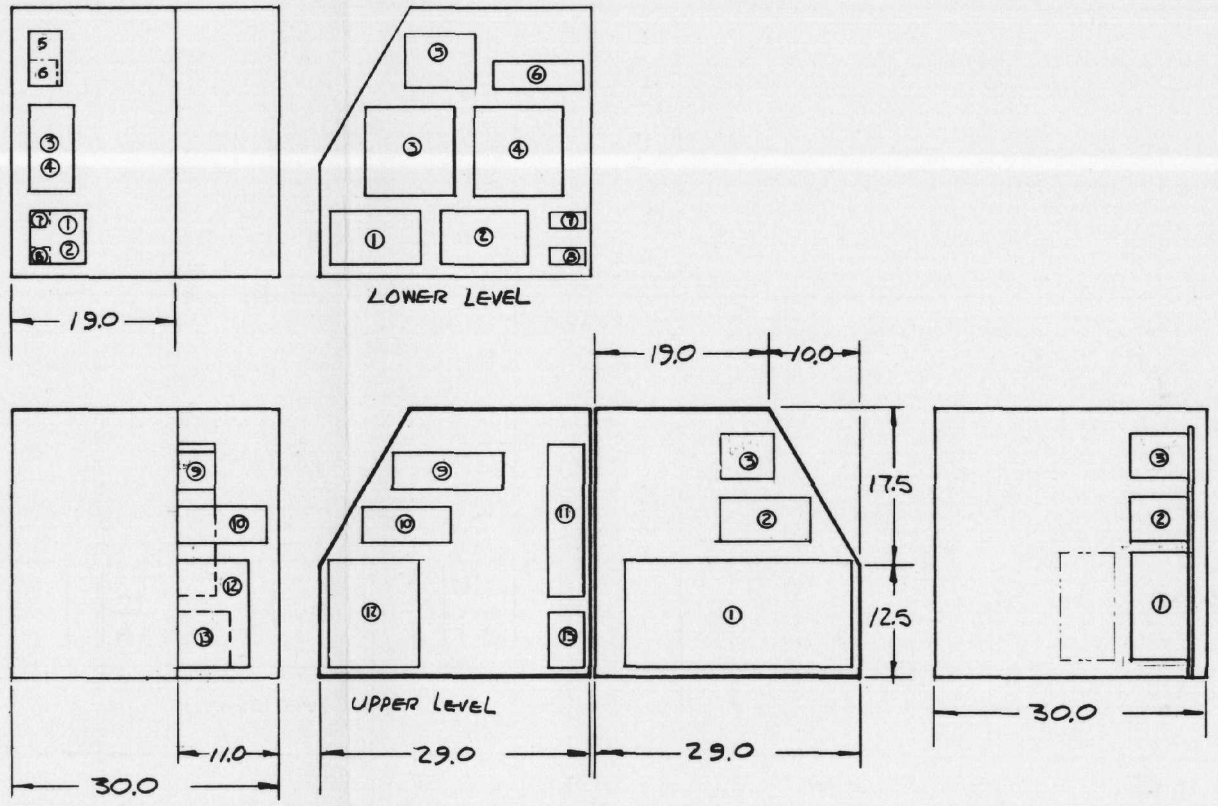


BAY 1 AND 4
RCS MODULE

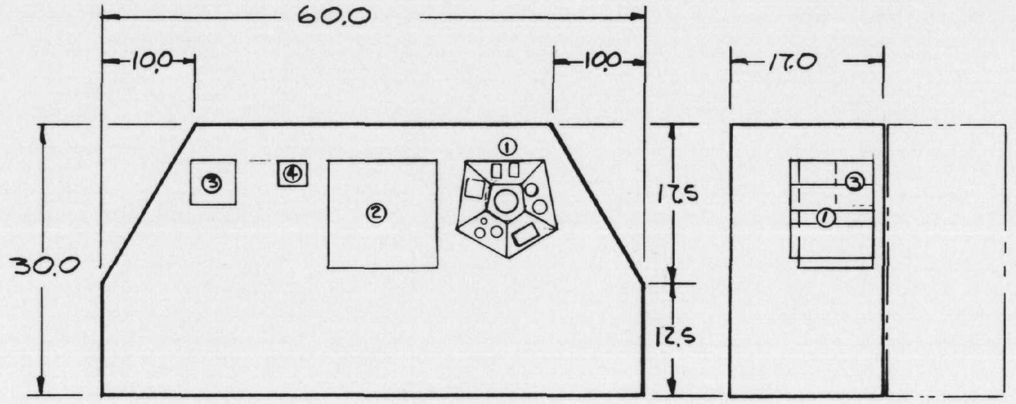
RCS MODULE	WEIGHT (LB)	ESTIMATED VOLUME
FUEL TANK	25	7.5 FT³
OXIDIZER TANK	25	7.5 FT³
PRESSURIZATION TANK ASSY	50	1.0 FT³
PLUMBING ASSY	45	
MODULE ASSY STRUCTURE	90	
TOTAL INERT WEIGHT	235	



BAY 3
COMMUNICATION & DATA MANAGEMENT

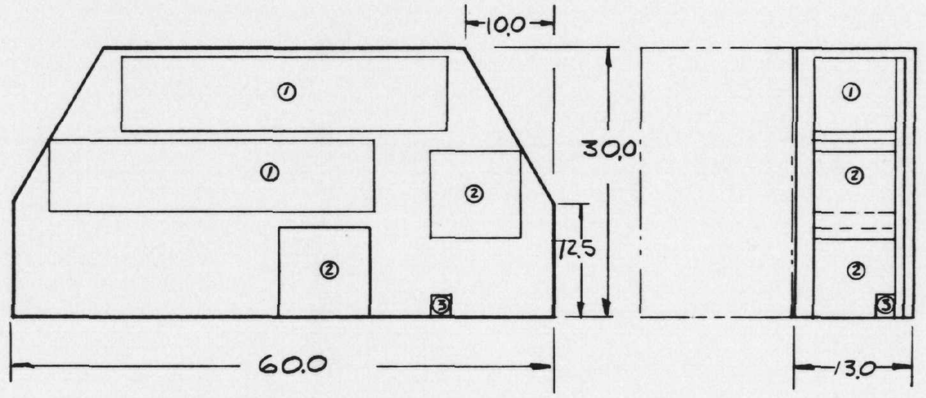
ITEM NO	COMMUNICATION SYSTEM MODULE	WEIGHT (LB)	ESTIMATED SIZE (IN)
1	DIGITAL COMMAND LINK	34	6X6X10
2	SIGNAL ASSEMBLY	11	6X6X10
3	S-BAND EQUIPMENT	19	10X10X5
4	S-BAND EQUIP.	19	10X10X5
5	S-BAND POWER AMP.	16	4X6X8
6	TRIPLEXER	16	3X3X10
7	RF SWITCH	2.5	2X2X4
8	POWER DIVIDER (S-BAND)	2.5	2X2X4
9	S-BAND SAFTY RCVR.	6	4X4X12
10	4 GHZ TRANSCEIVER	30	4X10X10
11	VHF COMMAND-XCVR	10	8X10X12
12	POWER DIVIDER (4 GHZ)	2	4X4X16
13	POWER DIVIDER (VHF)	.5	4X6X6
	MODULE STRUCTURE	45	
	TOTAL INERT WT	213.5	

ITEM NO	DATA MANAGEMENT SYSTEM MODULE	WEIGHT (LB)	ESTIMATED SIZE (IN)
1	CENTRAL PROCESSING UNIT & HIGH SPEED MEMORY	118	6X12X24
2	BULK STORAGE	40	6X6X10
3	POWER UNIT	17	5X6X6
	MODULE STRUCTURE	45	
	TOTAL INERT WT	220	



ITEM NO	GUIDANCE NAVIGATION & CONTROL SYS. MODULE	WEIGHT (LB)	ESTIMATED SIZE (IN)
1	INERTIAL SENSOR ASSY	40	13DIA×9
2	GUIDANCE COMPUTER	75	12X12X8
3	FLIGHT CONTROL ELECT.	12	4X5X5
4	RATE GYRO PACKAGE	5	3×3×6
	MODULE STRUCTURE	45	
	TOTAL INERT WT	177	

BAY 6 UPPER
GUIDANCE NAVIGATION & CONTROL



BAY 6 LOWER
POWER

ITEM NO.	POWER & ENVIRONMENTAL SYSTEM MODULE	WEIGHT (LB)	ESTIMATED SIZE (IN)
1	BATTERY ASSY	380	9X8X36
2	BATTERY CHARGE CONTROLLERS	30	12X12X12
3	ENVIR. CONTROL BOX	2	2X2X2
	MODULE STRUCTURE	45	
	TOTAL INERT WT	457	

SK-RNS-711003 SH. 2 OF 3

Fig. 5-4 RNS Spaceframe Equipment Arrangement – Bay Arrangement

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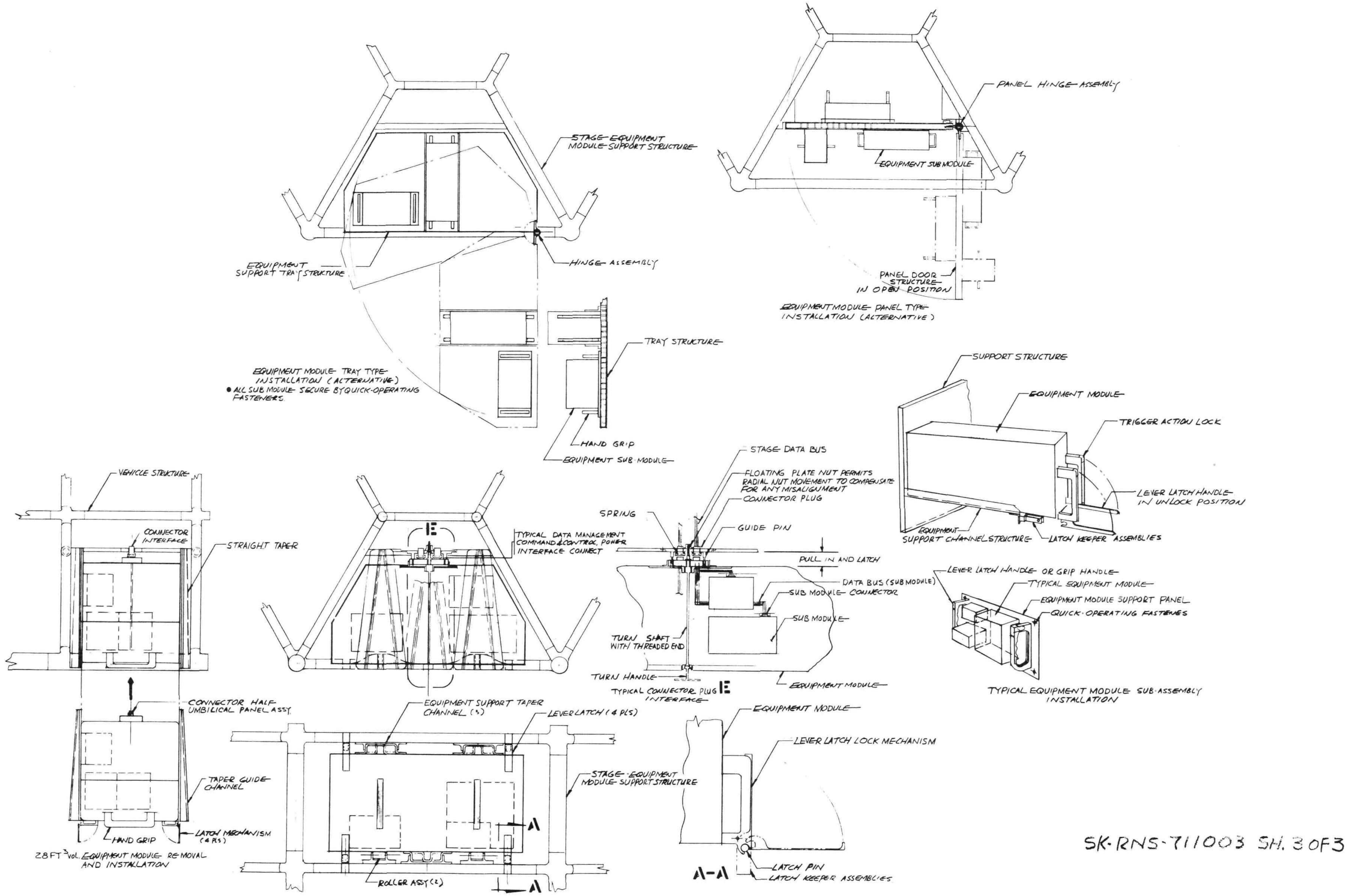

Fig. 5-5 RNS Spaceframe Equipment Arrangement – Installation Techniques

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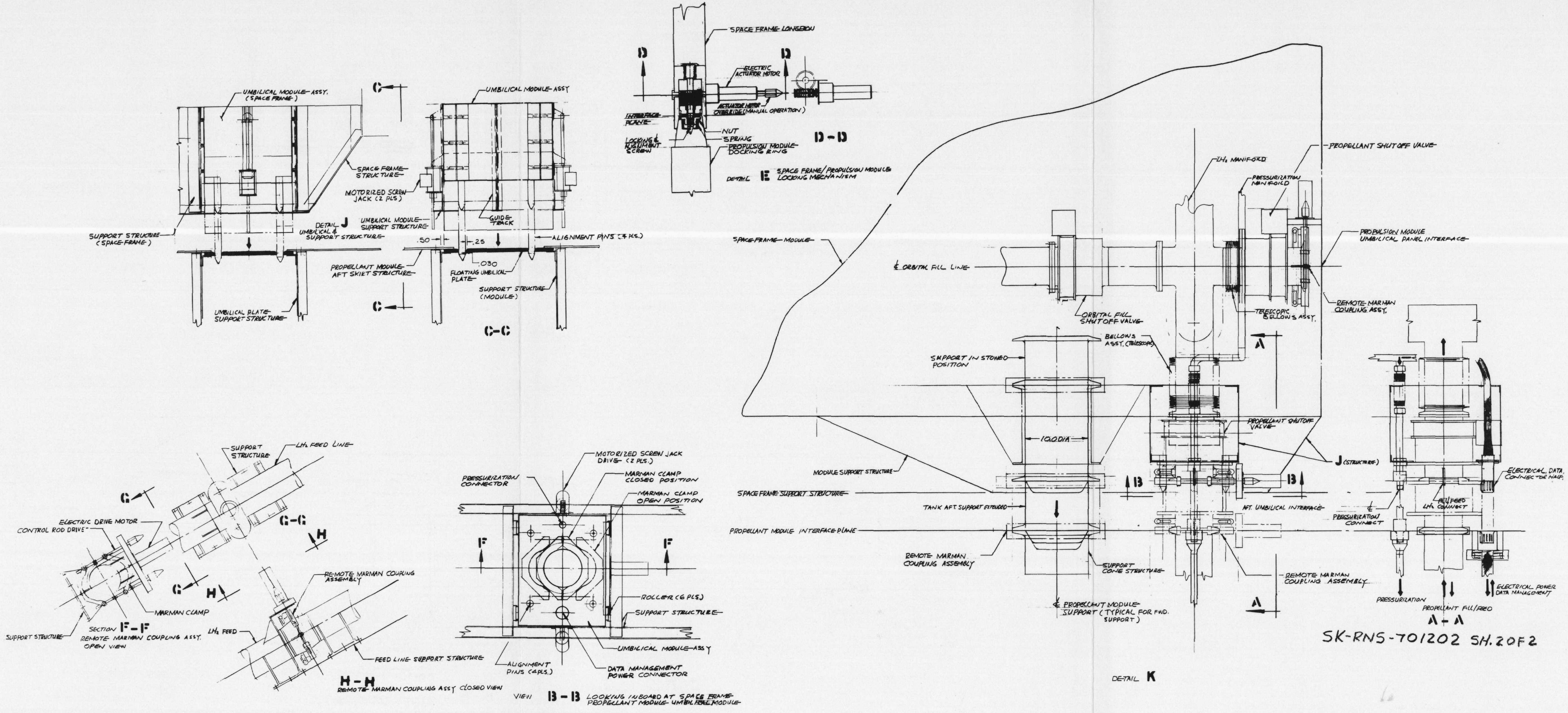

Fig. 5-6 Spaceframe Module Inboard Profile (Module Interface Detail)

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transmits all of the loads through six connecting pins to the spaceframe module, and is structurally mounted to the forward skirt. The engine thrust structure provides for attachment of the engine flyaway control module to the propulsion module, and also transmits all of the engine thrust loads to the propulsion module aft skirt. The thrust structure is a 30-deg cone with skin- and stringer- stiffened construction. Aluminum alloy (7075-T6) is used for the entire thrust structure. Figure 5-7 shows the propulsion module and the two main assemblies, the propulsion tank and engine, each requiring one Space Shuttle launch.

The NERVA/RNS interface is a single plane that will permit installation of the engine on an adapter or thrust structure with standard couplings at the ground launch facility. This arrangement of the propulsion module requires that the module is launched in two parts. The tank assembly, Fig. 5-8, is launched first. The engine system, Fig. 5-9, consisting of the NERVA engine and flyaway module make up the second payload. Assembly in orbit is accomplished by attaching these two submodules at the orbital assembly interface plane.

5.1.3 Propellant Module

The propellant module consists of a propellant tank, propellant, insulation system, meteoroid protection system, and forward and aft skirt structure. It also incorporates tank supports, orbital assembly supports, equipment supports, and an umbilical interface plate. This module also incorporates pressurization and propellant feed lines as well as electrical power distribution system and data management and instrumentation sensors. A propellant management system, purge system, ground emergency vent, flight vent, thermal conditioning unit and anti-slosh baffles are also included.

The module is structurally supported at the forward and aft ends. The aft structural attachment is effected by means of a 10-in.-diameter rigidly mounted cone with a spaceframe interface pin, and the forward attachment is effected by means of a 10-in.-diameter cone with an attachment pin that can slide axially on the spaceframe to allow for tank thermal effects at cryogenic temperatures. The umbilical panel located at the aft end interfaces with the previously described umbilical on the space frame. Figure 5-10 shows an inboard profile propellant module.

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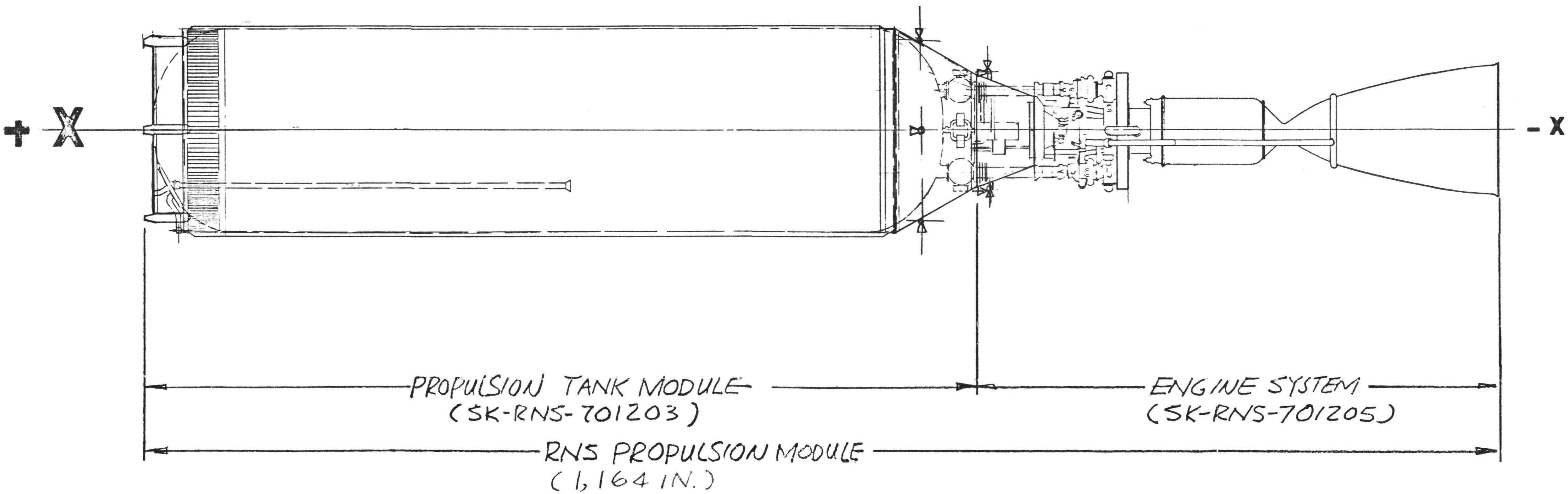

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Fig. 5-7 Propulsion Module

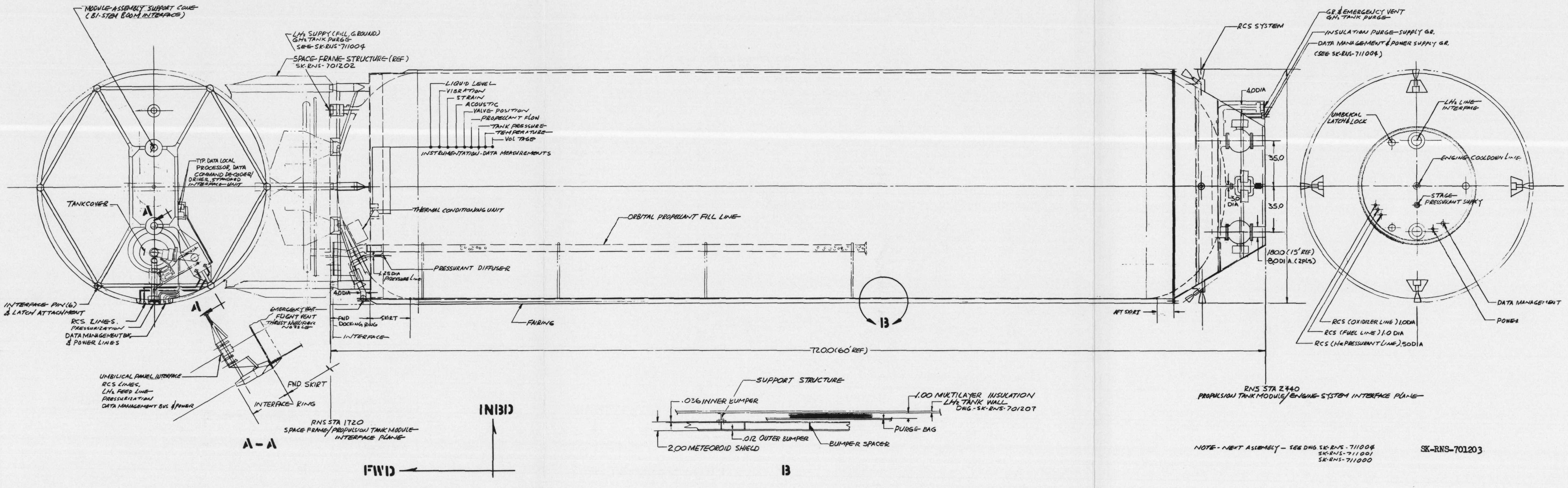

Fig. 5-8 Propulsion Module Tank (Inboard Profile)

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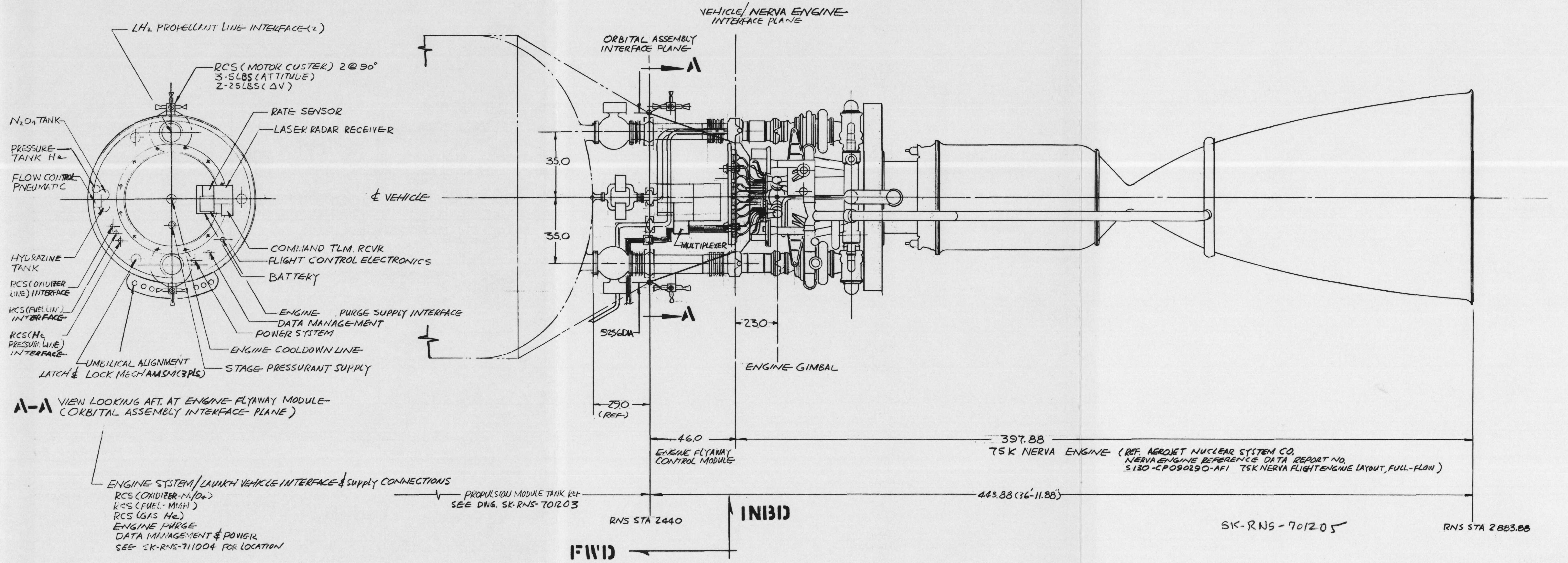

Fig. 5-9 Engine System

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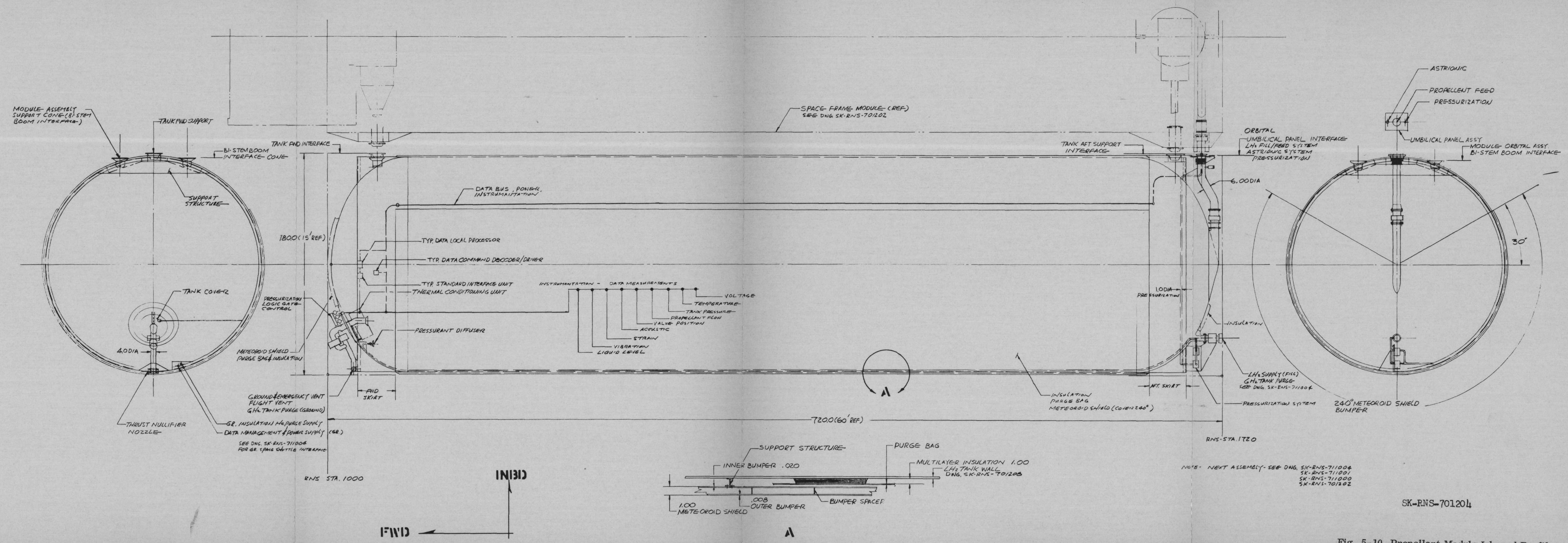

Fig. 5-10 Propellant Module Inboard Profile

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5.1.4 Baseline RNS

The baseline RNS is composed of three types of interchangeable modules launched by the Space Shuttle and discussed in the previous sections. The assembly of this vehicle is discussed in Part A of Volume II and only highlighted in this section. Figure 5-11 portrays the interaction of the modules in the assembly of the RNS. The deployment sequence is as follows: spaceframe, propulsion module tank, propulsion module engine system, and six propellant modules. The spaceframe, which is completely autonomous, is launched first and receives the propulsion module tank at its aft end. The tank is drawn to the spaceframe by two bi-stem booms and them mechanically attached and functionally connected to the spaceframe. The engine system is launched next and it is flown from the Space Shuttle to the propulsion module tank by means of the engine flyaway module. The mechanical and functional connections are then made. Finally, the six propellant modules are launched. Each module is drawn to the spaceframe by four bi-stem booms, two aft and two forward. After the propellant modules are in position they are mechanically latched and functionally connected through the umbilical plate. For the remaining propellant modules the procedure is repeated. The bi-stem boom assemblies are on a turntable and are rotated through 60 degrees to receive another module at the next position. The aft boom assembly also has a three-way gimbal assembly so that the booms can be positioned aft for the propulsion module assembly, and forward for the equipment module deployment.

The completely assembled vehicle is shown in Fig. 5-12. The drawing shows the RNS in a fully deployed configuration with solar panels and antenna extended. The physical location of modules and equipment is indicated on the drawing.

Functionally the RNS is shown in Fig. 5-13. The power source is shown, communications and guidance systems indicated, and the data management central processor located. The power and data management distribution systems are shown throughout the vehicle and the interfaces between modules are indicated. Only the left-hand propellant module is detailed, the others are all identical. The total layout of the RCS system including propellant, pressurization, propellant lines, and thruster locations are shown. The hydrogen propellant feed lines, pressurization lines,

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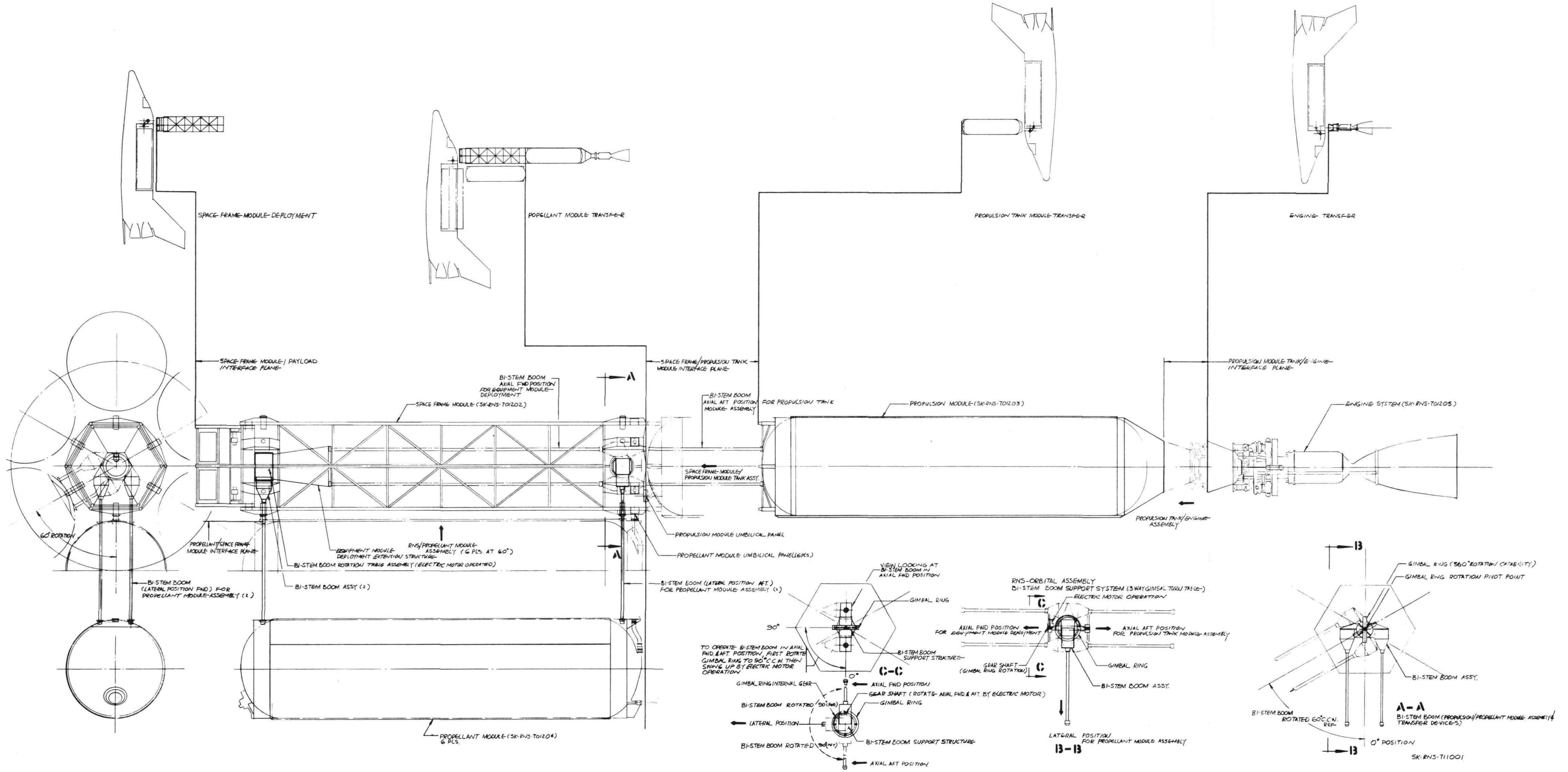

Fig. 5-11 Orbital Assembly Interface

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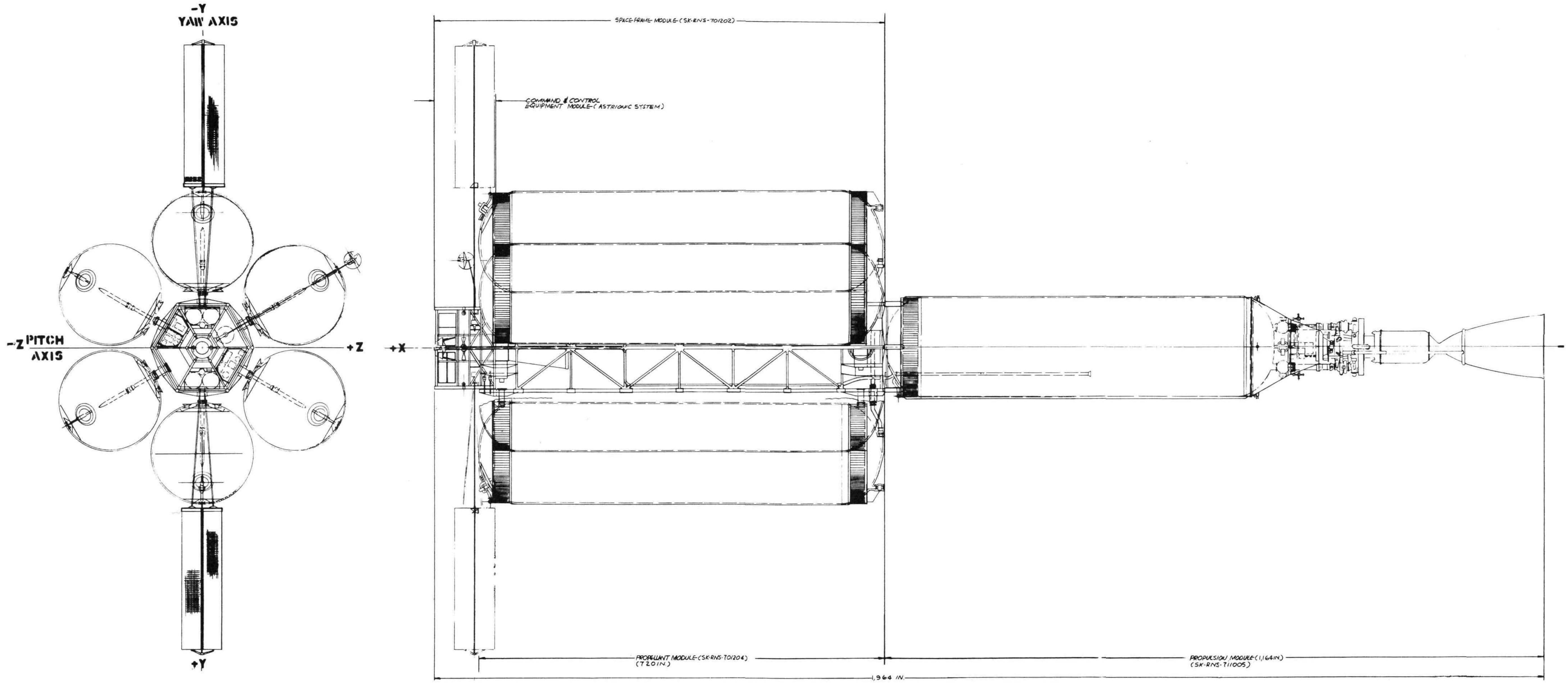

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Fig. 5-12 RNS Inboard Profile

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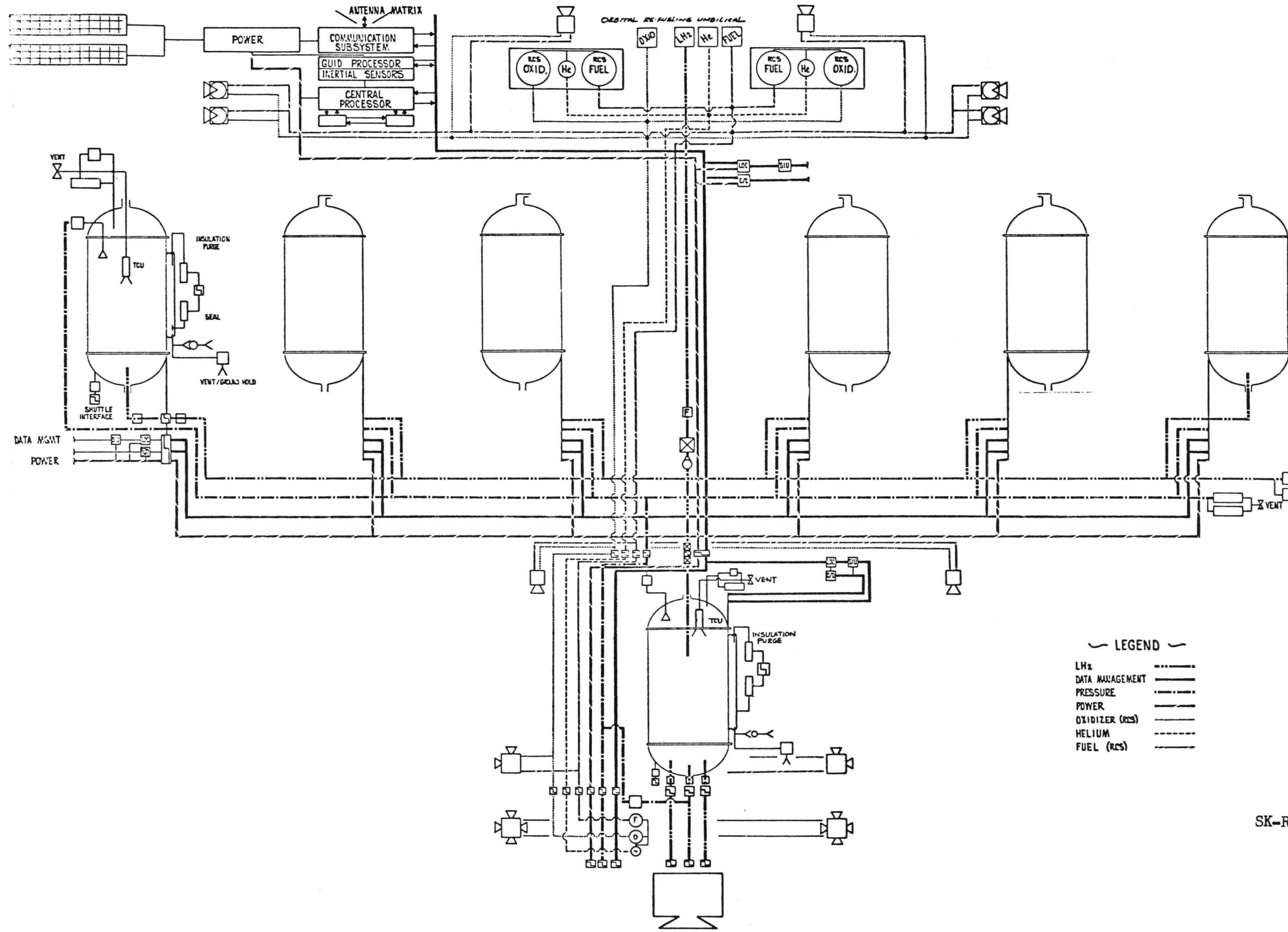

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Fig. 5-13 RNS Schematic

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thermal conditioning units, vent systems, and insulation purge systems are all shown throughout the vehicle again indicating all of the interfaces. The functional interfaces with the engine are also shown.

5.2 MASS PROPERTIES

RNS weights have been established to the fifth level of the Work Breadown Structure. In addition to establishing detailed weights; the mass properties have been established for the modules, the RNS during orbital assembly and during the mission phase. The mass properties of the modules and the orbital assembly sequence have been established for varying Space Shuttle capabilities.

5.2.1 Detailed Weight Statement

The detailed weight statement of the RNS is presented in Table 5-1. Weights are classified by structural, propulsion, and astrionic subsystems. In addition the weight for each classified item is also allocated to the propellant module, propulsion module, and spaceframe as well as to the total vehicle. No weight has been allocated to safety/ordnance. A five-percent contingency is carried on all the structural inert weight except for the NERVA engine and disc shield. The weight statement also specifies hydrogen propellant and RCS propellant loads.

5.2.2 Module Mass Properties

The mass properties of the modules have been established for the dry modules; filled with propellant, where applicable; and loaded with propellant so that the launch weight does not exceed 33,000 pounds. Figure 5-14 defines the vehicle stations of the RNS and Table 5-2 lists the mass properties of the propellant module, the propulsion module tank system, propulsion module engine system, and spaceframe. The mass properties listed are weight, three axis c.g. location, and moment of inertia about all three orthogonal axes. Table 5-3 presents the mass properties of the four modules filled with propellant, where applicable, so that the maximum payload for each Space Shuttle launch does not exceed 33,000 pounds. The 33,000 pound payload limit is based on Space Shuttle capability to the baseline 260 nautical mile 31.5 degree earth

Table 5-1

RNS WEIGHT STATEMENT

CODE	DESCRIPTION	Module (Weight lb)			Vehicle Weight *
		Propellant	Propulsion	Control & AssyStruct	
2.00	Structure	(3,370)	(4,118)	(2,293)	(26,631)
2.01	Propellant Tank Assembly **	2,904	2,818	N/A	20,242
2.02	Engine Thrust Structure	N/A	501	N/A	501
2.03	Forward Skirt	203	215	N/A	1,433
2.04	Aft Skirt	203	114	N/A	1,332
2.05	Tunnel & Fairings	N/A	30	N/A	30
2.06	Exterior Finish & Sealer	25	25	TBD	175
2.07	Propellant Containment System	25	25	N/A	175
2.08	Equipment Support Structure	10	15	60	135
2.09	Equipment Module Structure	N/A	N/A	1,078	1,078
2.10	Space Frame Structure	N/A	N/A	1,155	1,155
2.11	Engine Flyaway Module	N/A	375	N/A	375
3.00	Meteoroid/Thermal Protection	(1,141)	(2,617)	N/A	(9,463)
3.01	Insulation	374	729	N/A	2,973
3.02	Meteoroid Protection	767	1,887	N/A	6,490
4.00	Docking/Tank Support	(87.5)	(122)	(3,917)	(3,664)
4.01	Fwd Docking Propellant Trans. Cone	N/A	N/A	54	54
4.02	Aft Docking Mechanism	N/A	N/A	105	105
4.03	Tank Support & Mechanism	87.5	122	1,498	2,145
4.04	Vehicle Assembly Equipment	N/A	N/A	1,360	1,360
5.00	Main Propulsion	(242.5)	(37,038)	(667)	(39,160)
5.01	Nerva Engine	N/A	27,728	N/A	27,728
5.02	External Disc Shield for NERVA	N/A	8,100	N/A	8,100
5.03	Purge System	70	69	N/A	489
5.04	Propellant Condition System	25	25	N/A	175
5.05	Propellant Feed System	84	280	498	1,282
5.06	Pressurization System	30	43.5	52	275.5
5.07	Prepressurization System	N/A	N/A	N/A	N/A
5.08	Pneumatic System	N/A	N/A	N/A	N/A
5.09	Fill & Drain/Orbit Refueling Line	N/A	N/A	90	90
5.10	Ground & Emergency Vent	31.5	31.5	25	245.5
5.11	Flight Vent	2	2	2	16
5.12	Engine Flyaway Control Module	N/A	759+	N/A	759
6.00	Auxiliary Propulsion	N/A	(113)	(548)	(661)
6.01	Reaction Control System	N/A	113	548	661
6.02	Retro System	N/A	N/A	N/A	N/A
6.03	Ullage System	N/A	N/A	N/A	N/A

* 6 PROPELLANT MODULE
1 PROPULSION MODULE
1 SPACE FRAME

** TANK DESIGN PRESSURE 25.4 PSI
+ All subsystems included except structures

Table 5-1 (Cont)

CODE	DESCRIPTION	Modules (Weight LB)			Vehicle Weight*
		Propellant	Propulsion	Control & AssyStruct	
7.00	Astrionics	(40)	(45)	(1,533)	(1,818)
7.01	Guidance Navigation & Control	N/A	N/A	296	296
7.02	Instrumentation & Sensors	10	10	10	80
7.03	Communication	N/A	N/A	254	254
7.04	Electrical Power	N/A	N/A	560	560
7.05	Electrical Networks	5	10	140	180
7.06	Environmental Control	5	5	20	55
7.07	Propellant Management	N/A	N/A	N/A	N/A
7.08	On-Board Checkout	N/A	N/A	N/A	N/A
7.09	Data Management	20	20	253	393
8.00	Safety/Ordnance System	TBD	TBD	TBD	TBD
8.01	Safety System				
8.02	Ordnance System				
9.00	Contingency 5% (Exclude NERVA Engine & Disc Shield)	(244)	(411)	(403)	(2,278)
	Subtotal Inert	5,125	44,464	8,461	83,675
10.00	RCS Propellant	N/A	N/A	(2,200)	(2,200)
14.00	LH_2 Propellant	(38,676)	(37,371)	N/A	(269,427)
	TOTAL WEIGHT	43,801	81,835	10,661	355,302.0

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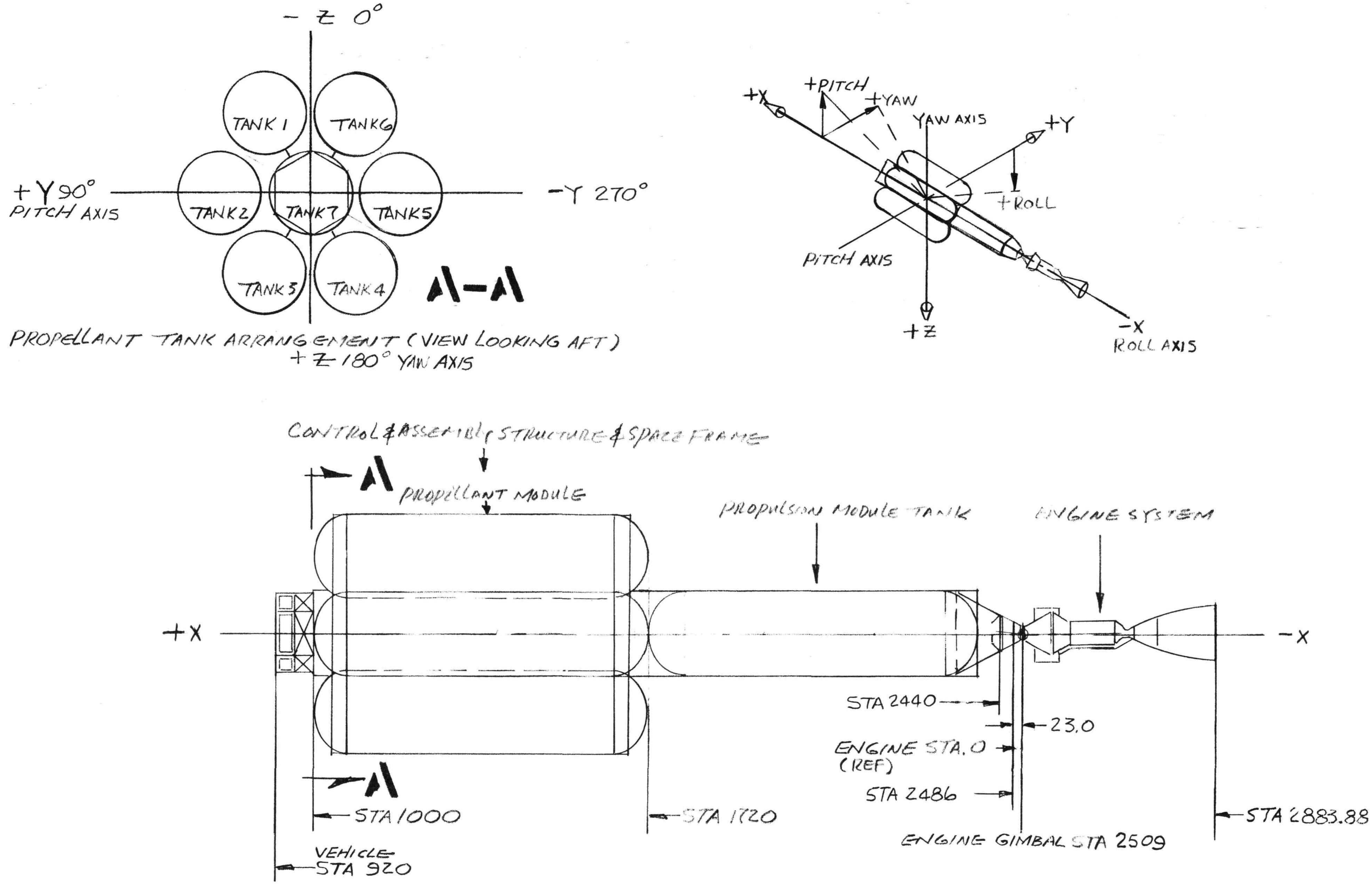

Fig. 5-14 RNS Vehicle Stations & Definitions

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Table 5-2

RNS VEHICLE ELEMENTS MASS PROPERTIES

Item	Vehicle Elements	WEIGHT Lbs.	Arm, Veh. Sta. (X) LONG.	(Y) LAT.	(Z) NORM.	Moment of Inertia (SLUG-Ft^2) (X) ROLL	(Y) PITCH	(Z) YAW
1)	Propellant Module - Empty **Tank**	5124.8	1360.	95.	-164.5	7373.	53416.	53416.
2)	Propulsion Module Tank - Empty **Tank**	7525.	20719	.0	.7	9749.	80706.	80612.
3)	Control & Assembly Structure & Space Frame	10661.	1265.4	.0	.0	9005.	173472.	173472.
4)	Engine System	36940.	25972	.0	.0	7000.	128209.	128109.

LMSC-A984555
Vol II, Part B

Table 5-3

RNS VEHICLE ELEMENTS MASS PROPERTIES PARTIALLY LOADED WITH PROPELLANT AT LAUNCH

REF. Space Shuttle 33K P/L Capability

Item	Vehicle Elements	WEIGHT Lbs.	Arm. Veh. Sta. (X) LONG.	(Y) LAT.	(Z) NORM.	Moment of Inertia (SLUG-Ft^2) (X) ROLL	(Y) PITCH	(Z) YAW
1)	Propellant Module - Empty (Tank 1)	5124.8	1360	95.	-164.5	7373.	53416.	53416.
	Partial Loaded Propellant							
	Propellant Module At Launch	32999.8	1447.8	95.	-164.5	7373.	169017.	169016.
2)	Propulsion Module Tank Empty	7,525.	2071	.0	.7	9749.	80706.	80612.
	Partial Load Propellant							
	Propulsion Module Tank At Launch	33,000.	2154.3	.0	.2	9750.	175672.	175578.
3)	Control & Assy Structure (Space Frame)	10661.	1265.4	.0	.0	9005.	173472.	173472.
4)	Engine System	36940.	2597.2	.0	.0	7000.	128209.	128109.

assembly orbit. Table 5-4 presents the same data with all propellant carrying-modules completely filled. These mass properties then dictate the maximum requirements imposed on the Space Shuttle for RNS delivery to orbit.

5.2.3 RNS Baseline Mass Properties – Assembly

The mass properties of the RNS during the orbital assembly sequence are shown in Table 5-5. The mass properties are developed from the first Space Shuttle launch delivering the spaceframe to the last Space Shuttle delivering the final propellant module. All of the propellant carrying modules in this assembly sequence are partially loaded and consistent with the 33,000 pound Space Shuttle capability. The engine system weighs 36,940 pounds and exceeds the guidelines. Since the propellant-carrying modules are only partially loaded, three additional Space Shuttle launches carrying propellant are required to complete fueling of the vehicle. The mass properties of the RNS during the propellant topping operation is shown in Table 5-6. Data are presented for the RNS with sequential propellant filling by module.

For Space Shuttle payload capabilities in excess of 44,896 pounds all of the modules can be launched completely filled with propellant so that only 9 Space Shuttle launches are required. The mass properties of the RNS orbital assembly with completely filled modules is shown in Table 5-7.

5.2.4 RNS Baseline Mass Properties – Mission

Subsequent to the assembly and fueling phase the RNS performs its Lunar Shuttle mission. As shown in Fig. 5-15, payloads can be attached to the payload support structure forward of the RNS. In addition, manned payloads can be carried centrally on the payload support structure maximizing the distance from the engine and minimizing the radiation exposure. The mass properties of the RNS during the mission are shown in Table 5-8. The data are given for the Nominal Lunar Shuttle mission. In addition to the mass properties presented previously, the NERVA engine gimbal angle required to fly the RNS with asymmetrically loaded propellant tanks is also presented.

Table 5-4

RNS VEHICLE ELEMENTS MASS PROPERTIES, FULLY LOADED WITH PROPELLANT AT LAUNCH

Item	Vehicle Elements	WEIGHT Lbs.	Arm, Veh. Sta. (X) LONG.	(Y) LAT.	(Z) NORM.	Moment of Inertia ($SLUG\text{-}Ft^2$) (X) ROLL	(Y) PITCH	(Z) YAW
1)	Propellant Module - Empty **(Tank 1)**	5124.8	1360.	95.	-164.5	7373.	53416.	53416.
	Full Loaded Propellant							
	Propellant Module at Launch	43800.8	1373.5	95.	-164.5	7373.	334463.	334462.
2)	Propulsion Module Tank - Empty	7525.	2071.9	.0	.7	9749.	80706.	80612.
	Full Loaded Propellant							
	Propulsion Tank Module at Launch	44896.	2079.5	.0	.1	9750.	334271.	334177.
3)	Control & Assy Structure (Space Frame)	10661.	1265.4	.0	.0	9005.	173472.	173472.
4)	Engine System	36940.	2597.2	.0	.0	7000.	128209.	128109.

Table 5-5

RNS SEQUENTIAL ORBITAL ASSEMBLY MASS PROPERTIES WITH PARTIALLY LOADED PROPELLANT TANKS

Item	Assembly Sequence	WEIGHT Lbs.	Arm, Veh. Sta. (X) LONG.	(Y) LAT.	(Z) NORM.	Moment of Inertia ($SLUG-Ft^2$) (X) ROLL	(Y) PITCH	(Z) YAW
1	Control & Assy Structure (Space Frame) with Equipment Section Extended	10661.	1228.5	.0	.0	9005.	210345.	210345.
2	Item 1 Plus Propulsion Module Tank (33,000#)	43661.	1928.2	.0	.1	18754.	1876635.	1876540.
3	Item 1 2 Plus Engine System (36,940#)	80601.	2234.8	.0	.1	25754.	3937786.	3937592.
4	Item 1 2 3 Plus Propellant Module No. 1 (33,000#)	113600.8	2006.2	27.6	-47.7	215592.	7374307.	7282865.
5	Item 1 2 3 4 Plus Propellant Module No. 4 (33,000#)	146600.6	1880.5	.0	.0	554548.	9513180.	9256067.
6	Item 1 2 3 4 5 Plus Propellant Module No. 2 (33,000#)	179600.4	1801	34.9	.0	771805.	10770912.	10723683.
7	Item 1 2 3 4 5 6 Plus Propellant Module No. 5 (33,000#)	212600.2	1746.2	.0	.0	1083552.	11690659.	11947804.
8	Item 1 2 3 4 5 6 7 Plus Propellant Module No. 3 (33,000#)	245600.	1706.1	12.8	22.1	1313367.	12575464.	12721457.
9	Item 1 2 3 4 5 6 7 8 Plus Propellant Module No. 6 (33,000#)	278599.8	1675.5	.0	.0	1612346.	13382102.	13382327.

Note: # = lb

Table 5-6

RNS MASS PROPERTIES REFILLING WITH PARTIALLY LOADED PROPELLANT TANKS

Item	Assembly Sequence	WEIGHT Lbs.	Arm, Veh. Sta. (X) LONG.	Arm, Veh. Sta. (Y) LAT.	Arm, Veh. Sta. (Z) NORM.	Moment of Inertia ($SLUG\text{-}Ft^2$) (X) ROLL	Moment of Inertia ($SLUG\text{-}Ft^2$) (Y) PITCH	Moment of Inertia ($SLUG\text{-}Ft^2$) (Z) YAW
1	RNS Full Assembly with Partial Loaded Propellant (192,726#)	278599.8	1675.5	.0	.0	1612346.	13382102.	13382327.
2	Item 1 Plus Propulsion Module Tank Refilled (LH_2 11,896#)	290495.8	1683.5	.0	.0	1612346.	13485244.	13485468.
3	Item 1 2 Plus Propellant Module 1 Refilled (LH_2 10,800.85#)	301296.8	1664.3	3.4	-5.9	1693468.	14200292.	14159965.
4	Item 1 2 3 Plus Propellant Module 4 Refilled (LH_2 10,800.83#)	312097.8	1646.3	.0	.0	1780596.	14874974.	14791110.
5	Item 1 2 3 4 Plus Propellant Module 2 Refilled (LH_2 10,800.83#)	322898.8	1629.6	6.4	.0	1861941.	15443965.	15441445.
6	Item 1 2 3 4 5 Plus Propellant Module 5 Refilled (LH_2 10,800.83#)	333699.8	1614.0	.0	.0	1948916.	15976514.	16060969.
7	Item 1 2 3 4 5 6 Plus Propellant Module 3 Refilled (LH_2 10,800.83#)	344500.8	1599.3	3.0	5.2	2030392.	16537149.	16580885.
8	Item 1 2 3 4 5 6 7 Plus Propellant Module 6 Refilled (LH_2 10,800.83#) RNS Orbital Assembly Completed (Orbit Gross)	355301.8	1585.6	.0	.0	2117166.	17071757.	17072120.

Note: # = lb

Table 5-7

RNS SEQUENTIAL ORBITAL ASSEMBLY MASS PROPERTIES WITH FULLY LOADED PROPELLANT TANKS

		WEIGHT	Arm, Veh. Sta.			Moment of Inertia (SLUG-Ft^2)		
		Lbs.	(X) LONG.	(Y) LAT.	(Z) NORM.	(X) ROLL	(Y) PITCH	(Z) YAW
Item	Assembly Sequence							
1	Control & Assy Structure (Space Frame) With Equipment Section Extended	10661.	1228.5	.0	0	9005.	210345.	210345.
2	Item 1 Plus Propulsion Module Tank (44,896#)	55557.	1916.2	.0	.1	18754.	1891150.	1891055.
3	Item 1 2 Plus Engine System (36,940#)	92497.	2188.2	.0	.1	25754.	4240596.	4240402.
4	Item 1 2 3 Plus Propellant Module No. 1 (43,800.83#)	136297.8	1926.4	30.5	-52.8	264758.	9007031.	3891011.
5	Item 1 2 3 4 Plus Propellant Module No. 4 (43,800.83#)	180098.6	1791.9	.0	.0	722799.	11866413.	11525209.
6	Item 1 2 3 4 5 Plus Propellant Module No. 2 (43,800.83#)	223899.4	1710.	37.2	.0	1004695.	13532198.	13465516.
7	Item 1 2 3 4 5 6 Plus Propellant Module No. 5 (43,800.83#)	267700.2	1655.	.0	.0	1420122.	14762324.	15103696.
8	Item 1 2 3 4 5 6 7 Plus Propellant Module No. 3 (43,800.83#)	311501.	1615.4	13.4	23.1	1720624.	15960371.	16155263.
9	Item 1 2 3 4 5 6 7 8 Plus Propellant Module No. 6 (43,800.83#) RNS Orbital Assembly Completed (Orbit Gross)	355301.8	1585.6	.0	.0	2117166.	17071736.	17072099.
	Note: # = lb							

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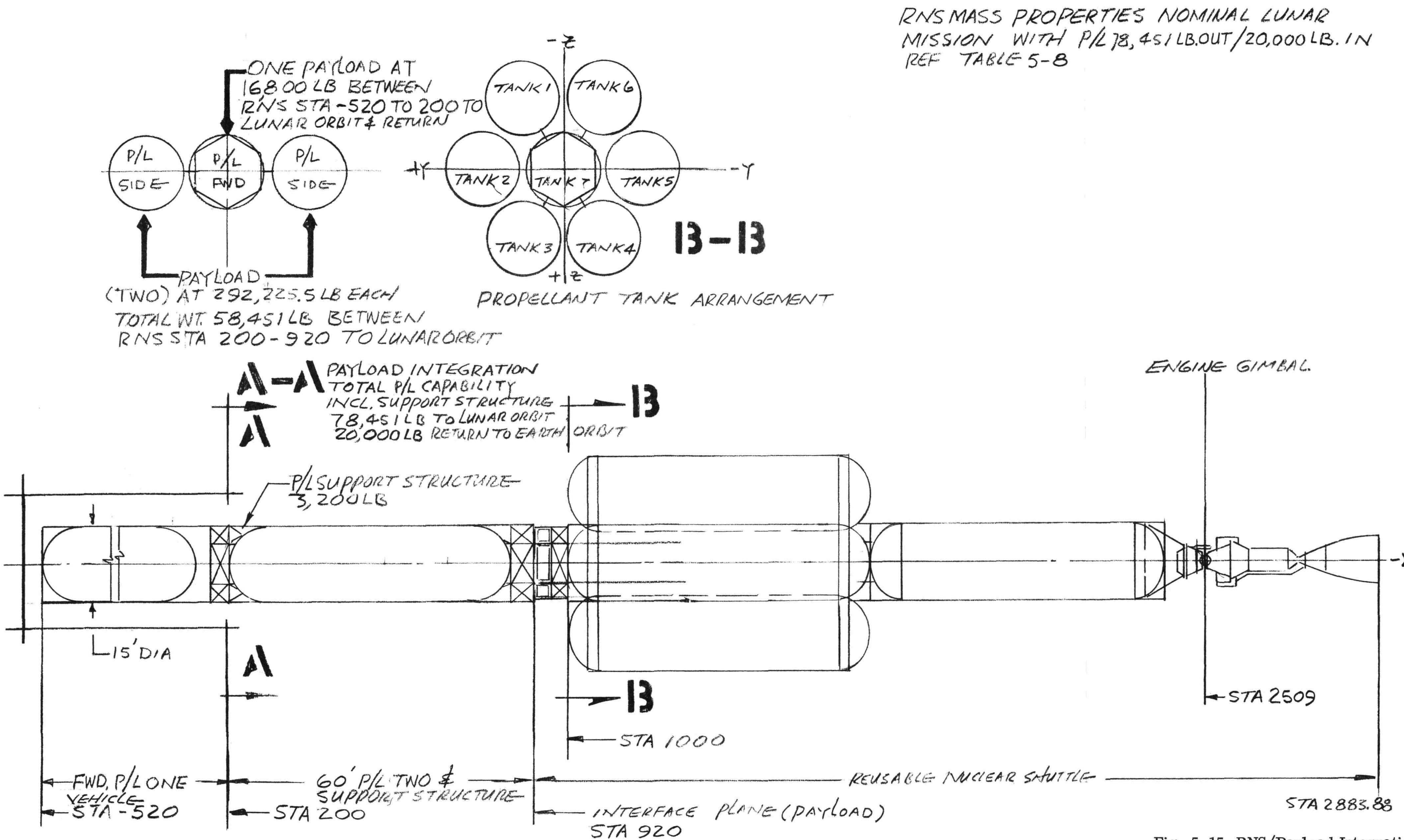

Fig. 5-15 RNS/Payload Integration

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Table 5-8

RNS MASS PROPERTIES, NOMINAL LUNAR MISSION WITH 74,451 LB P/L OUT – 20,000 LB IN (MISSION PHASE SEQUENCE OF EVENTS)

		Engine Gimbal Angle (Deg)	WEIGHT Lbs.	Arm, Veh. Sta. (X) LONG.	(Y) LAT.	(Z) NORM.	Moment of Inertia (SLUG-Ft^2) (X) ROLL	(Y) PITCH	(Z) YAW
1)	RNS Gross Weight	.001	355301.8	1585.6	.0	.0	2117166.	17071736.	17072099.
	P/L Support Structure 3,200#	.001	358501.8	1576.4	.0	.0	2122761.	17824346.	17824709.
	P/L Module Fwd 16,800#	.001	375301.8	1498.7	.0	.0	2137447.	28432138.	28432501.
	P/L Module Side 1 @ 29,225.5# L/H	.729	404527.3	1430.9	-13.7	.0	2374122.	33872477.	34084109.
	P/L Module Side 1 @ 29,225.5# R/H	.001	433752.8	1372.2	.0	.0	2643701.	38617929.	39073734.
	Total Gross								
2)	Leave Earth	1.034	286882.8	1377.5	-20.4	.0	1473726.	37154721.	37344128.
	- Tank 1 Propellant Wt. - 38,676#								
	- Tank 3 Propellant Wt. - 38,676#								
	- Tank 5 Propellant Wt. - 38,676#								
	- Tank 2 Propellant Wt. - 30,842#								
3)	Lunar Orbit Insertion	1.306	252402.8	1381.3	-19.3	-17.	1199613.	36716789.	36968020.
	- Tank 2 Propellant Wt. - 7,834#								
	- Tank 4 Propellant Wt. - 26,096#								
	- RCS Propellant Wt. 550#								
4)	Lunar Stay								
	- 2 Payload Total 58,451# SIDE	2.177	193951.8	1628.5	-25.1	-22.1	682510.	25069038.	24863455.
	- 1 Payload 16,800# FWD	2.951	177151.8	1798.5	-27.5	-24.2	663381.	12199012.	11992867.
	- Tank 4 Propellant Wt. - 2,783#								
	- RCS Propellant Wt. 550#	3.095	173818.8	1805.5	-26.5	-27.3	639040.	12062183.	11874994.
	+ 1 Payload Fwd. 16,800#								
	Leave Moon Burn 1 @ S.U.	2.266	190618.8	1632.2	-24.2	-24.9	658511.	25001768.	24814435.
5)	Leave Moon Burn 1	2.451	176449.8	1646.7	-18.5	-32.0	545791.	24618270.	24510495.
	- Tank 4 Propellant Wt. 9,797#								
	- Tank 6 Propellant Wt. 4,372#								
6)	Earth Orbit Insertion	.003	106976.8	1592.0	.0	.0	330827.	22275120.	22274990.
	- Tank 6 Propellant Wt. 34,304#								
	- Tank 7 Propellant Wt. 34,619#								
	- RCS Propellant Wt. 550#								
7)	Earth Orbit	.003	104224.8	1571.5	.0	.0	330827.	21908386.	21908257.
	- Tank 7 Propellant Wt. 2,752#								
	Note: # = lb								

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5.3 ENVIRONMENT

5.3.1 Thermal

The external environment is the most significant thermal boundary condition imposed on the RNS. The external surface temperature in turn directly affect the insulation thickness and vent pressure of the propellant tanks for an optimized design.

The external boundary condition was established in the following manner. An analytical model was developed of the RNS. The model contained all of the external surfaces of the RNS so that all external radiation and reflections could be determined. The vehicle required 17 surfaces divided into 1277 elements in order to be adequately defined. The internal structure was also defined and modeled into the thermal network. The thermodynamic model surfaces are shown in Fig. 5-16. The temperatures of these surfaces define the external boundary conditions throughout the mission. The vehicle orientation varies significantly throughout the mission as shown in Fig. 5-17. Solar orientation is of course the primary driver for boundary temperatures. During the main engine burns and impulse-utilized cooldown phases, the RNS longitudinal axis is along the flight path. During the impulse-spoiled phase, the RNS is oriented in such a manner that the payload alternately faces the sun and backs to the sun. During the lunar orbit phase and during coast, the payload faces the sun. The detailed time intervals for each of the phases and orientation is shown in Table 5-9. A heat rate program was utilized to obtain external heating rates for the vehicle for each phase of the mission. During translunar and trans-earth portions of the mission, a single instantaneous heat flux from the sun is obtained and assumed to be constant for each of the three positions during their respective mission phases. In lunar orbit, heat rates were obtained as a function of orbit position and then averaged over the entire orbit. All heat rates take into account vehicle and lunar shading, as well as reflections of solar and lunar heat fluxes from one surface to another and solar energy reflected by the moon. The actual boundary conditions consisting of the external surface temperatures of each tank as a function of insulation thickness and time were established. These values were defined with a representative degraded optical surface reflector coating having an α/ϵ of .08/.8. The conduction heat leak boundary temperatures are the outer temperatures for the structural and line penetration connections that go to the hydrogen tanks.

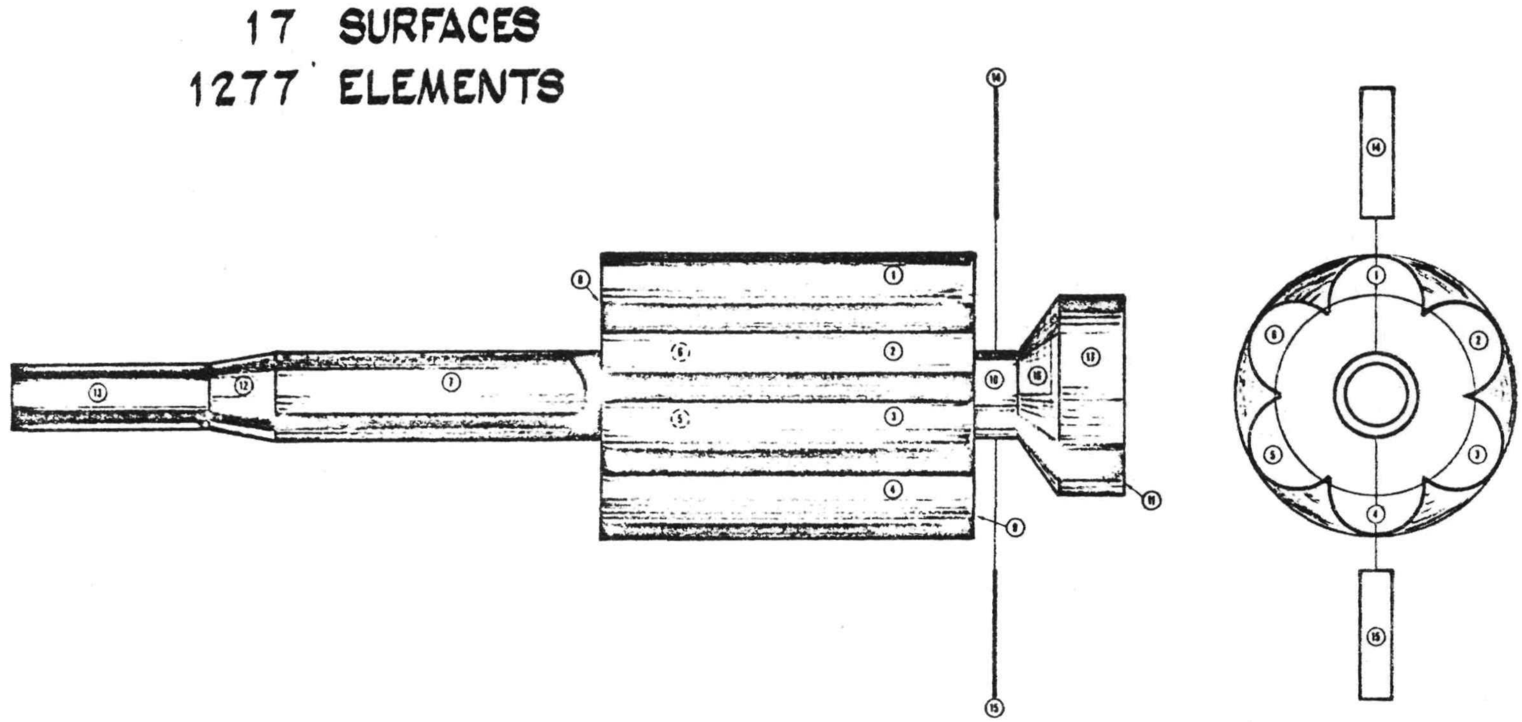

Fig. 5-16 Thermodynamic Model

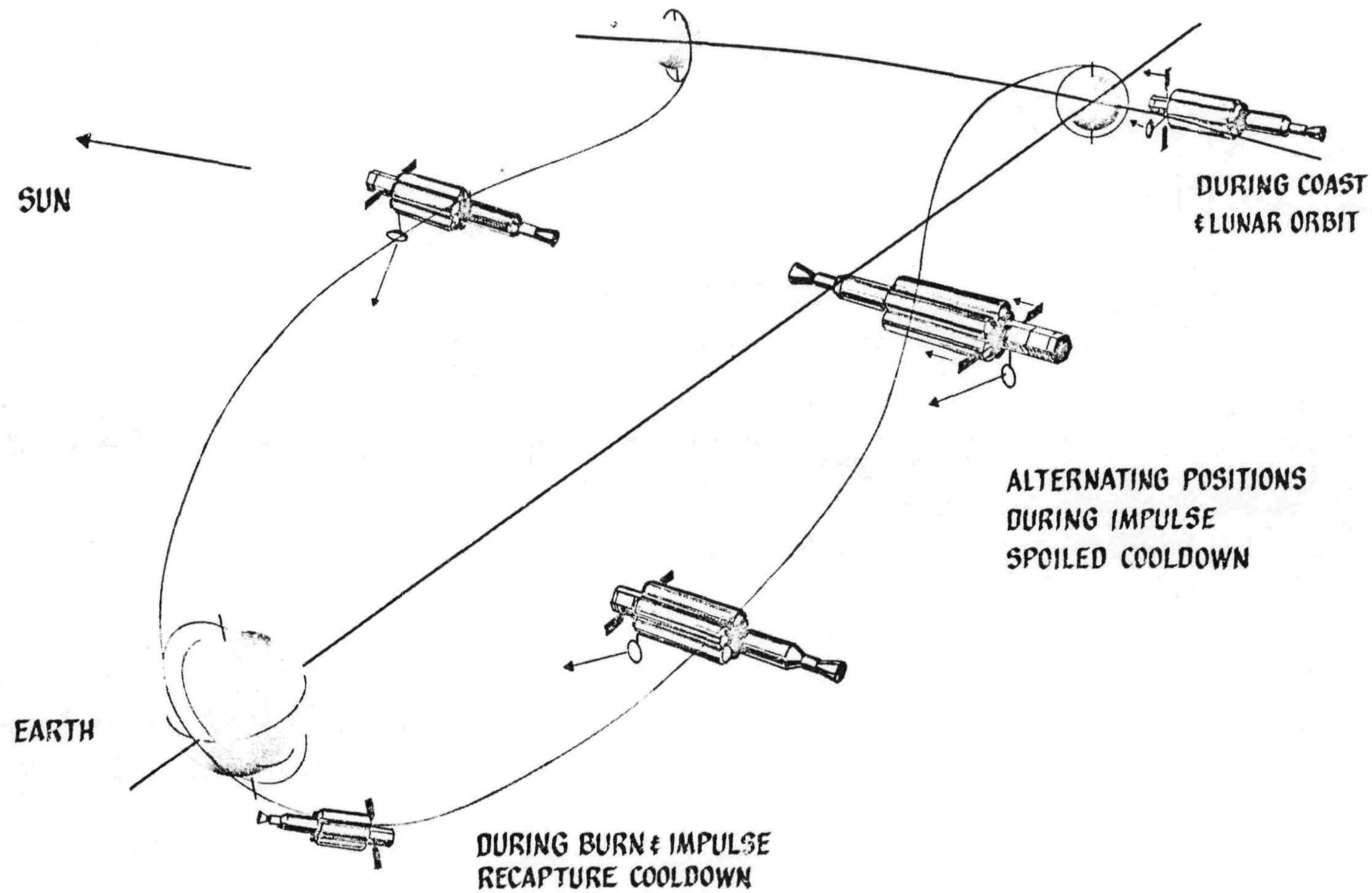

Fig. 5-17 Vehicle Orientation During Mission

Table 5-9

VEHICLE ORIENTATION TIMELINE

PHASE	TIME INTERVAL, HOURS	ORIENTATION TO SUN
Translunar	8	Broadside, fixed.
	100	Alternately 10 hrs. payload to sun and then 10 hrs. engine to sun.
Lunar Orbit	2½	Broadside, fixed. Longitudinal axis perpendicular to radius vector from moon.
	25½	Alternately 6+ hrs. payload to sun and then 6+ hrs. engine to sun.
	332	Payload to sun.
Transearth	2½	Broadside, fixed.
	14½	Alternately 3+ hrs. payload to sun and then 3+ hrs. engine to sun.
	55	Payload to sun.
Earth Orbit	4	Broadside, fixed. Longitudinal axis perpendicular to radius vector from moon.
	40	Alternately 5 hrs. payload to sun and 5 hrs. engine to sun.

The temperatures calculated are shown in Table 5-10. The NUTOP computer program is used to simulate the entire mission involving thermodynamic energy balances for coast phases, engine burns, cooldown, and afterburn equilibrium. The mission simulation is part of an optimization routine that defines insulation thickness and vent pressure. This directly affects the amount of energy that goes into each propellant tank. Figure 5-18 shows the total energy deposited into each tank throughout the mission. This includes the external environmental induced energy as well as the energy injected internally through the hot pressurization gas. The internal and external heating is separated in the following series of figures. Figure 5-19 shows the internal and external heating for propellant tank 5. This tank is first activated (pressurized) at lunar injection, 108 hours mission time. The comparable data for propellant tank 6 is shown in Fig. 5-20. This tank is not activated until the first trans-earth injection burn is made. The characteristics of the propulsion module are shown in Fig. 5-21. This module is of course activated for each burn. The external heating is directly affected by the thickness of the insulation as can be seen from the steeper slopes for the thinner insulation thicknesses. Also the internal heating is minimized in the modular approach due to the frequency of tank depletion during the mission.

5.3.2 Radiation

The radiation environment has been established for the RNS. Both the inflight environment and the component environment were established for the RNS without an external shield. This provides the most severe design conditions and also the environment for the unmanned system. In manned missions an 8100-pound external disk shield is added to limit the crew dose to 10 REM at the interface of the crew module.

5.3.2.1 In-Flight Environment. Radiation environment during reactor operation was estimated at 27 locations in the baseline clustered tank configuration shown in Fig. 5-22. Gamma ray environment below the propulsion tank is shown in Table 5-11. These data represent dose accumulated during the expenditure of 270,000 pounds of LH_2 at a rate of 90.9 lb/sec. Environments above the propulsion and propellant tanks, respectively, are shown in Tables 5-12 and 5-13. The propellant tanks are assumed to be emptied sequentially, with vapor venting immediately following LH_2 expenditure in each tank. Accumulated dose in the payload above the propellant modules (Detector Locations 1-9)

Table 5-10

EXTERNAL BOUNDARY CONDITIONS

	Average Temperature, °R				
	Equipment Bay	Propellant Tank #3	Propellant Tank #4	Propulsion Module	Engine
Earth Orbit	360	408	410	400	395
Trans-Lunar Coast					
Cooldown	352	390	346	303	537
Facing Sun	372	147	164	35	111*
Away From Sun	167	147	163	253	180
Lunar Orbit	375	421	422	417	406

*Nozzle is 295°R

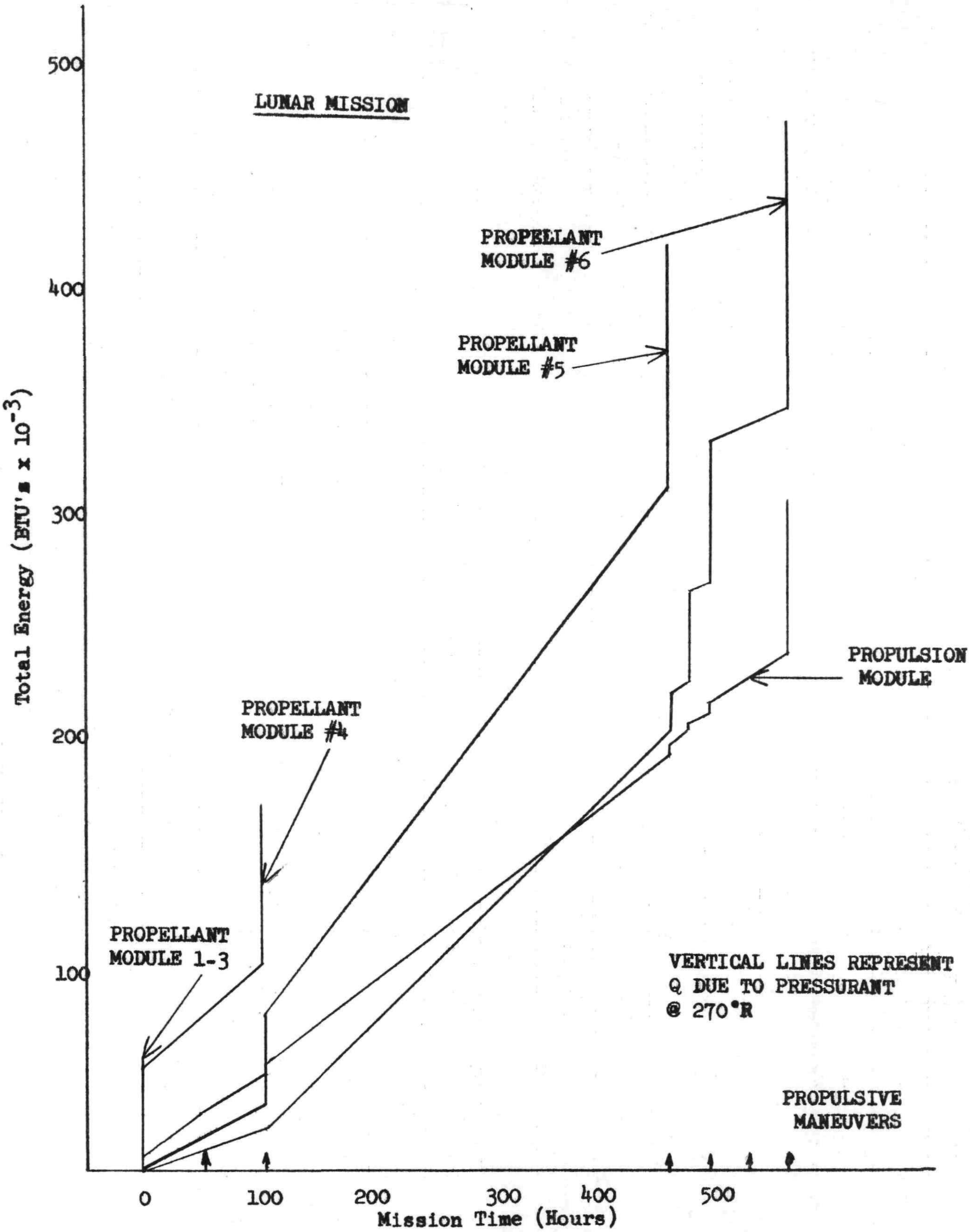

Fig. 5-18 Total Energy Into Propellant Tanks

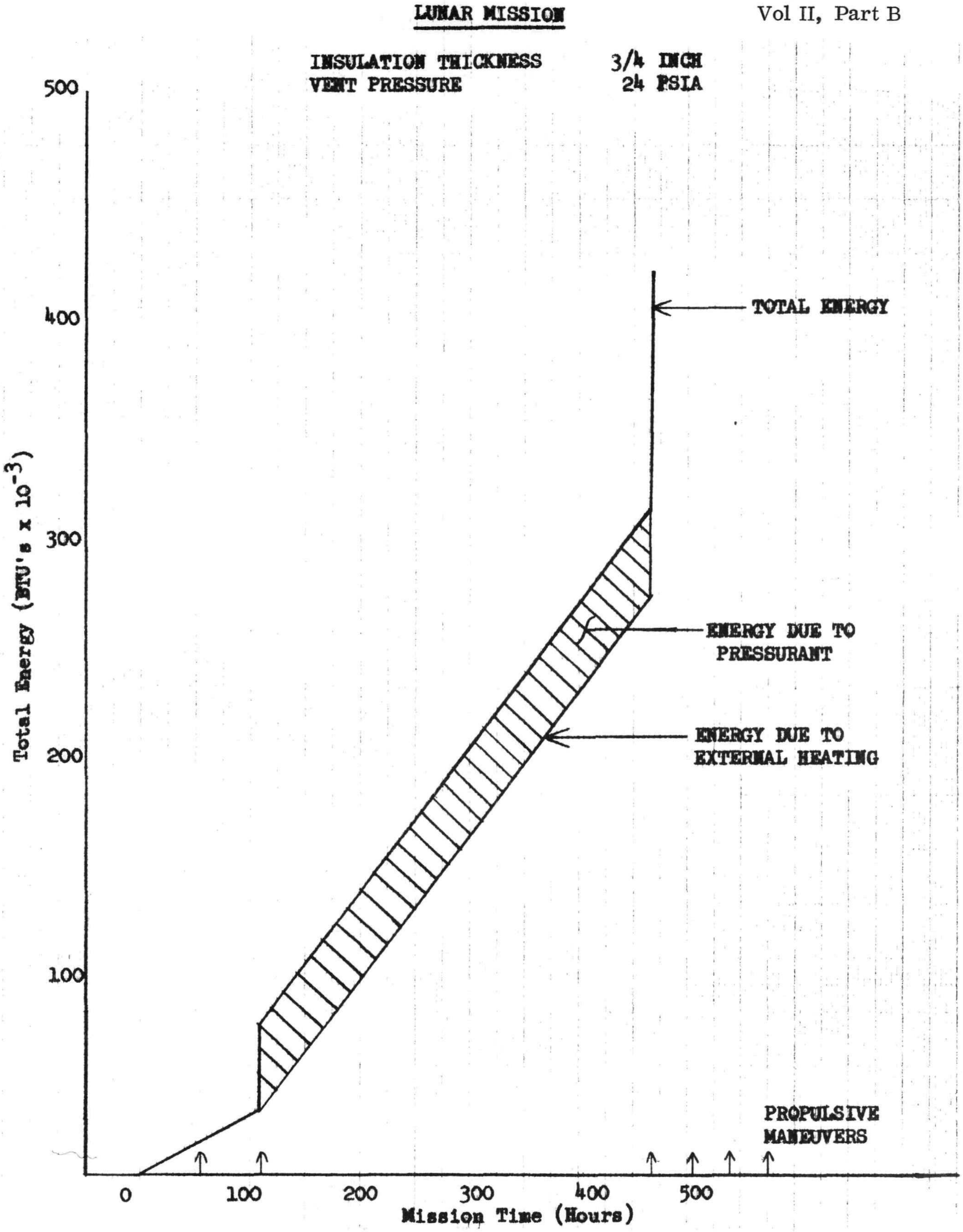

Fig. 5-19 Total Energy Into Propellant Tank No. 5

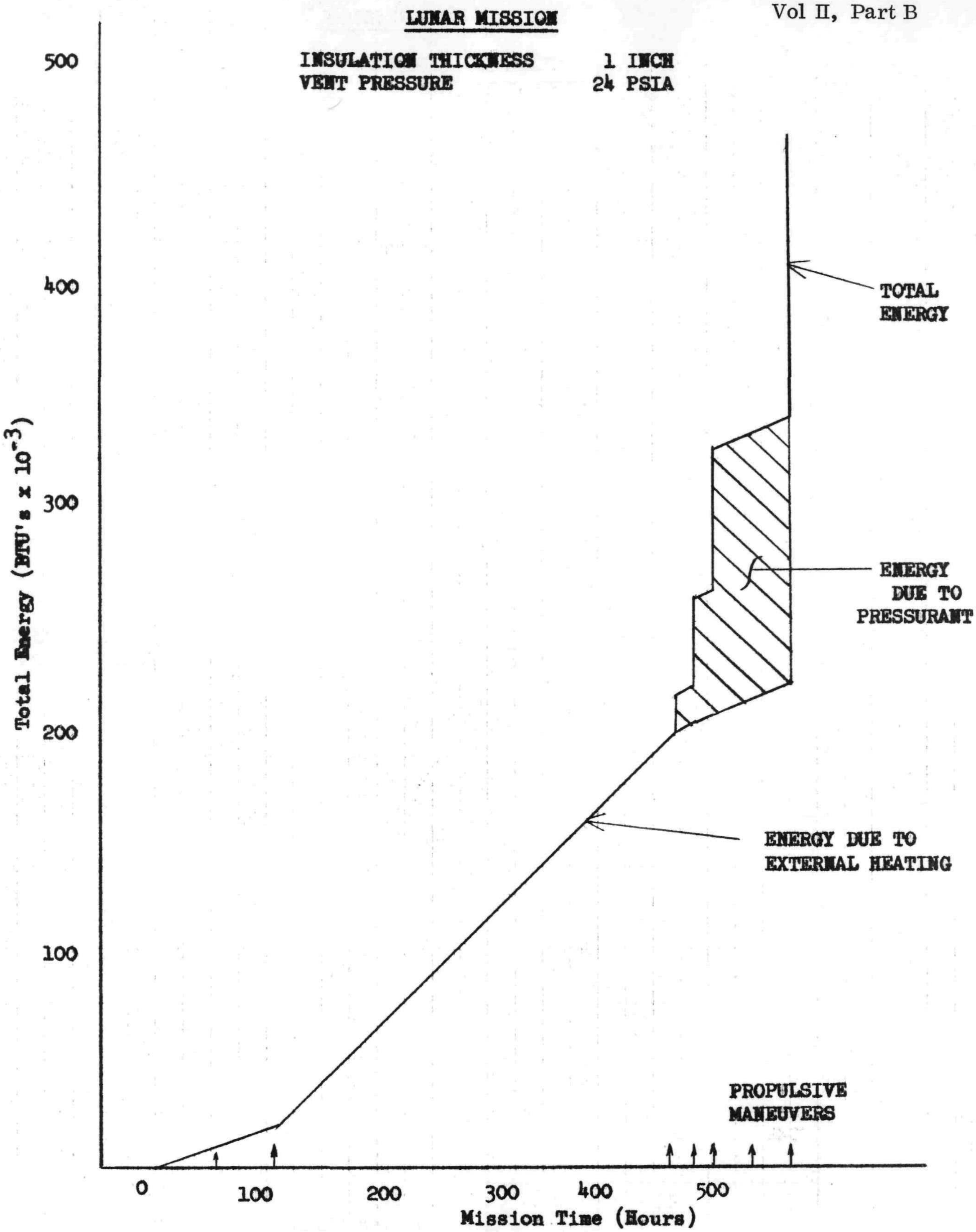

Fig. 5-20 Total Energy Into Propellant Tank No. 6

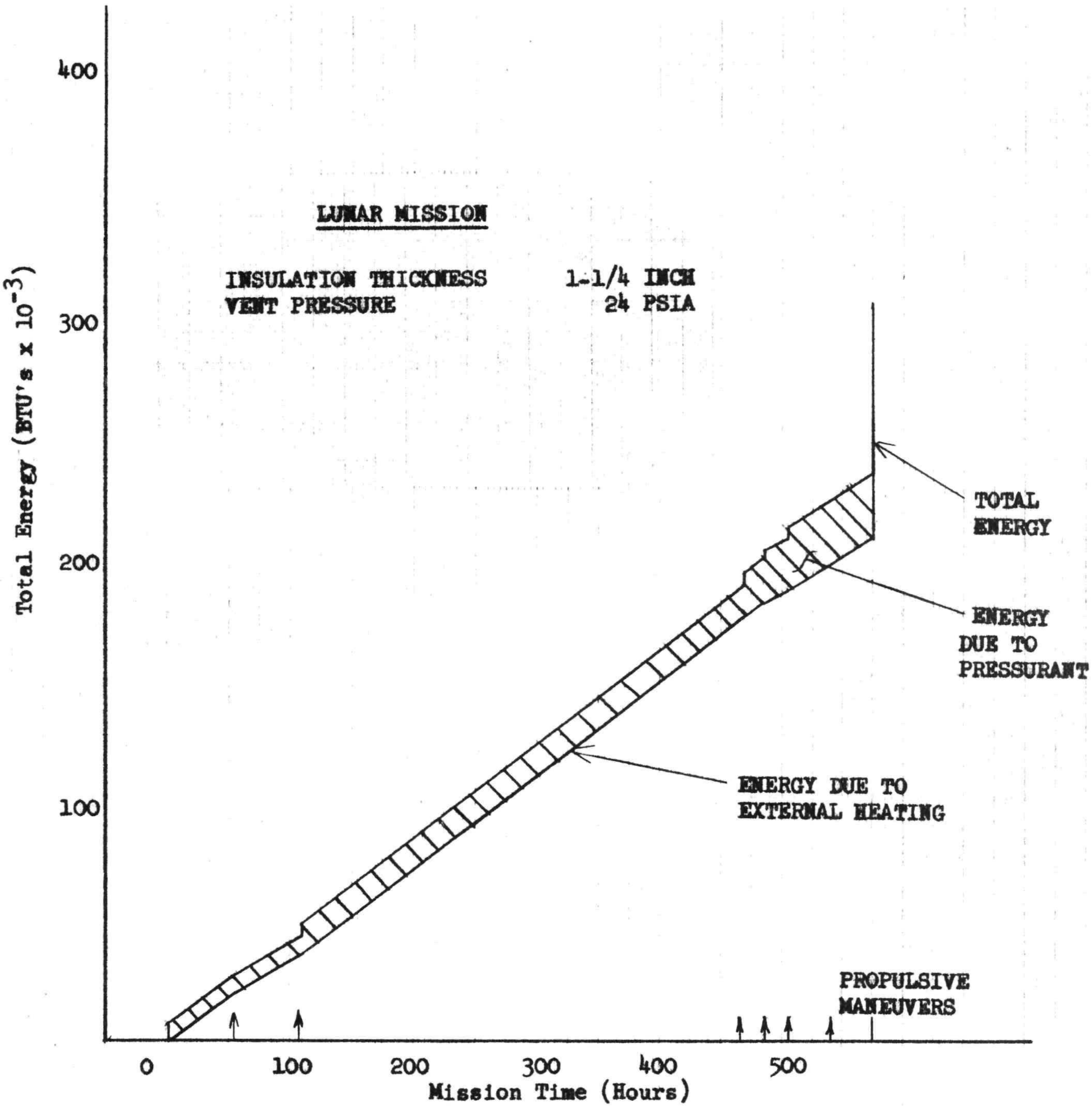

Fig. 5-21 Total Energy Into Propulsion Module Tank

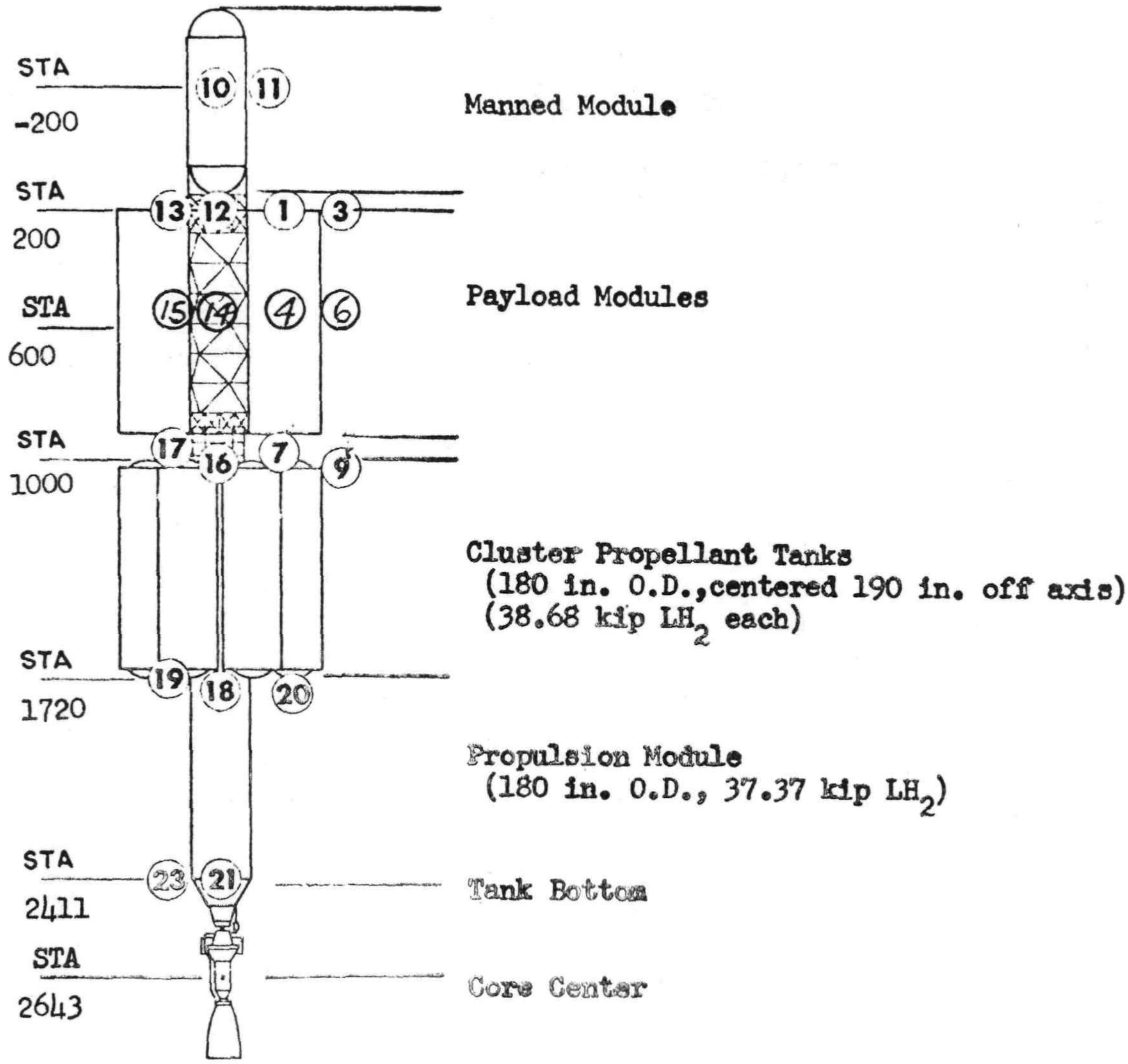

Fig. 5-22 Radiation Analysis Configuration

Table 5-11

GAMMA RAY ENVIRONMENT BELOW PROPULSION TANK MODULE

Detector Number	Location (in.)		Dose ($rad_T x 10^{-5}$)
	Height	Radius	
21	0*	0*	1.70
22	0	45	2.0
23	0	90	14.0
24	23.6	20	1.6
25	28.6	40	1.7
26	38.6	60	2.0
27	56.5	80	1.8

Points 24-27 are on tank surface.

*Reference location is 7 inches below tank bottom (Station 2411) on centerline.

Table 5-12

ENVIRONMENT ABOVE PROPULSION TANK MODULE

Detector Number	Location (in.)		Accumulated Dose (rad_T) 3000 lb Residual
	Height	Radius	
10	1920	0	35
11	1920	90	40
12	1520	0	50
13	1520	90	55
14	1120	0	80
15	1120	90	90
16	720	0	160
17	720	90	180
18	0*	0*	260
19	0	90	310
20	0	190	9800

*Reference location is at bottom of spaceframe (Station 1720) on centerline.

Table 5-13

ENVIRONMENT ABOVE PROPELLANT TANK MODULES

Residual LH_2 (lb)	Detector Number	Location			Accumulated Dose (Rad_T) Above Respective Module Module Drain Sequence					
		Height (in.)	Radius (in.)	Azimuth (deg)	1	2	3	4	5	6
3000	1	800	190	0	530	460	380	300	220	140
	2	800	190	20	550	470	390	310	220	140
	3	800	280	0	710	600	490	380	270	160
	4	400	190	0	600	510	430	340	260	180
	5	400	190	20	620	540	450	360	270	180
	6	400	280	0	1300	1100	870	670	480	280
	7	0	190	0	830	720	610	500	390	280
	8	0	190	20	870	750	630	520	400	280
	9	0	280	0	2600	2200	1800	1400	990	580

Reference point on vehicle centerline at cluster tank top (Station 1000); zero azimuth intersects cluster tank axis. Calculated with 3000 lb of LH_2 residuals).

is strongly dependent on the time at which propellant in the cluster tank directly below is expended, as illustrated in the last six columns of Table 5-13.

The present dose estimates represent an extrapolation of previously published Monte Carlo data (Ref 5-1). In particular, Monte Carlo dose rates above and below a 180 inch propellant tank were used to evaluate a set of e-folding lengths and equivalent point-source strengths for each of four radiation components. The components treated are direct radiation emitted from the Pressure Vessel And Reactor Assembly (PVARA) top, direct radiation from the PVARA side, direct radiation from sources external to the PVARA, and radiation scattered within the propulsion tank. Extrapolation to the propellant modules consisted of applying the derived e-folding lengths and source strengths to a point kernel model of radiation transport.

For the three direct components, the arrays of equivalent source points were located on the system centerline at distances between 0 and 10^4 inches below the PVARA top. A black disk is located above the source points simulating PVARA side leakage, and above some of the points simulating external sources. Sources simulating LH_2 scatter were located in a plane 67 inches above the bottom of the propulsion tank. The simulation procedure proved capable of reproducing the original Monte Carlo dose rates for each component to within ± 5 percent.

The equivalent sources representing scattering in the propellant tanks were derived from the propulsion-tank scatter sources by assuming that scattering density is proportional to the radiation current incident on the tank bottom. Scattering in tank walls and vehicle structure is not included in the present estimates. The Monte Carlo data, and by implication the present results, are derived from the PVARA model defined by the May 1969 Common Radiation Analysis Model (CRAM). Sources external to the PVARA are consistent with the March 1970 CRAM.

5.3.2.2 Component Radiation Environment. A radiation effects analysis of the RNS insulation and instrumentation has indicated that no radiation effects problems could be expected in these areas. The equipment in the flyaway control module, shown in Fig. 5-23, has been analyzed and the environment defined. The analysis was based

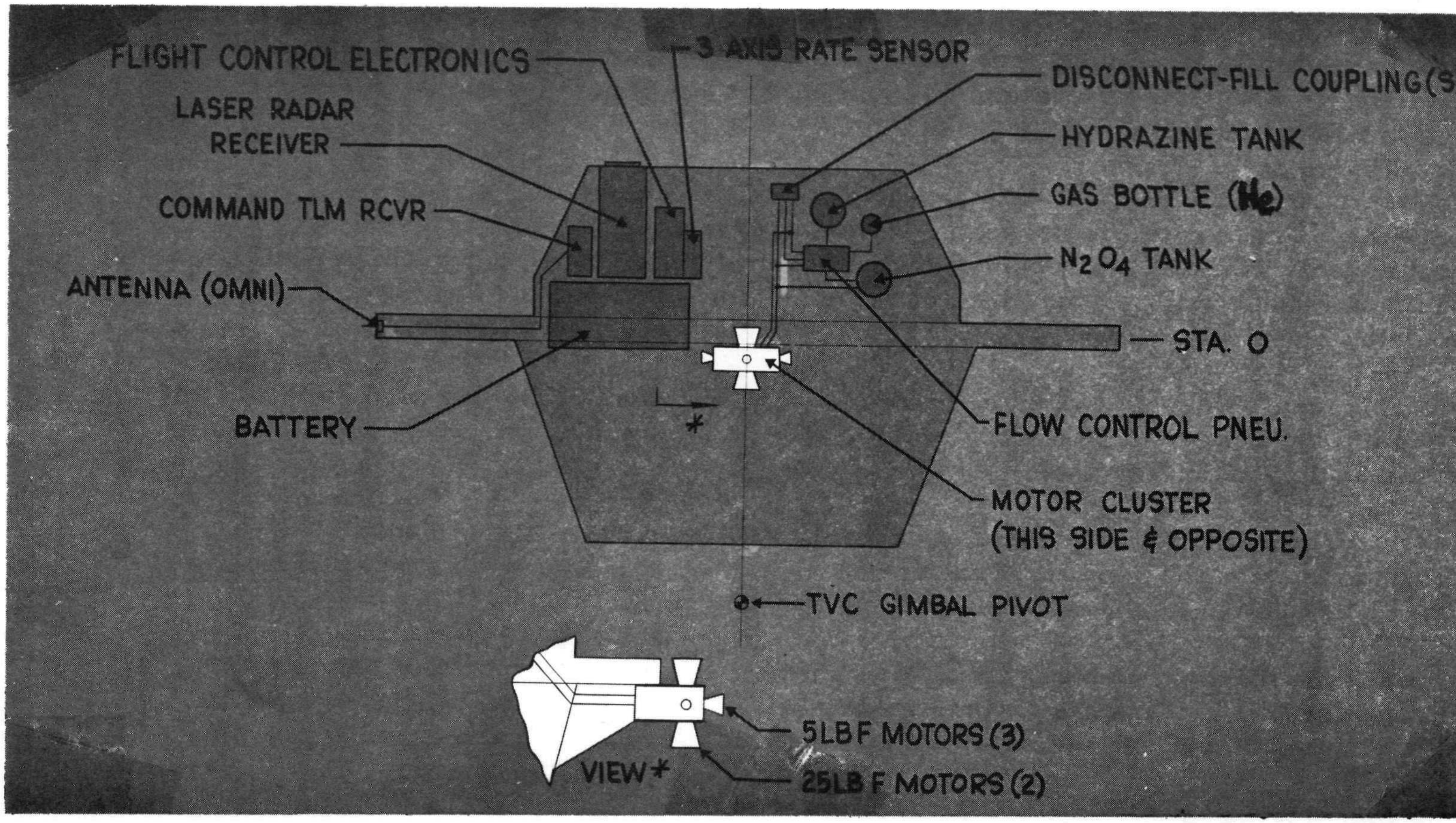

Fig. 5-23 Flyaway Control Module Equipment

on a 10 mission lifetime. The results of the analysis are summarized in Table 5-14, giving dose, damage threshold, and shielding requirements when required. No significant radiation effects from gamma radiation is predicted for the components. However, the flight electronics (semi-conductor devices) will be affected by the neutron environment. These components are therefore protected by a polyethylene shield of the dimensions and weight shown in Table 5-14. The following components were analyzed for the radiation environment.

Batteries. Batteries exist which can withstand the radiation environment expected in the nuclear shuttle. Alkaline batteries have been exposed to gamma doses as high as 10^8 rads without producing any discernible effects. Nickel-cadmium batteries have been shown to retain 90 percent of their capacity when exposed to the environment of a nuclear reactor. See References 5-2 and 5-3.

Silver electrodes have withstood gamma radiation doses as high as 10^9 rads (H_2O) with little if any effect on the silver electrode capacity. Some electrodes may lose material when exposed to a dose of 10^7 rads or more but no significant problem is encountered because battery designs include retentive separators which act to contain residual solids; these separators are materials like irradiated polyethylene which will not be damaged by the 10-mission environments.

Storable Propellants. Storable propellants such as nitrogen tetroxide (N_2O_4) and hydrazine (MMH) are essentially unaffected by nuclear radiation doses of 1×10^7 r gamma and 1×10^{15} nvt fast-neutrons ($E > 0.5$ MEV), except for the normal temperature/pressure rise due to energy deposition. See References 5-4 and 5-5. There are no significant chemical changes nor any effect on ignition time lag.

In the selection of storage tanks or bladders, materials are available that will withstand the stresses of temperature and pressure, that are chemically inert, and that will resist radiation damage.

Seals, Sealants. A rather large selection of seals and sealant materials exist for use in various environments. Nitrite and neoprene elastomers have been tested and shown to be resistant to radiation exposure doses of 1×10^{10} and 1×10^9 ergs $g^{-1(C)}$,

Table 5-14

COMPONENT RADIATION EFFECTS

ADAPTER SECTION

	RADIATION ENVIRONMENT RADS/HR	INTEGRATED DOSE/MISSION RADS	DAMAGE THRESHOLD RADS	MAXIMUM ALLOWABLE RADS	SHIELDING REQUIREMENTS 10 MISSIONS	COMMENTS
BATTERIES	1×10^6	6×10^5	7×10^7	10^8	—	
STORABLE PROPELLANTS	1×10^6	6×10^5	$>10^7$	$>10^7$	—	2×10^{-3} BTU/LB/SEC
METAL SEALS	5×10^5	3×10^5	$>10^8$	$\gg10^8$	—	
VALVE SEALS	5×10^5	3×10^5	$>10^7$	$\gg10^8$	—	ASSUMING URETHANE OR ESTHANE TYPE SEAL
WIRING INSULATION	1×10^6	6×10^5	$>5\times10^7$	$>10^8$	—	ASSUMING ORGANIC INSULATION
TRANSDUCERS	1.5×10^6 2×10^{11} *	1×10^6 5×10^{14} **	$>2\times10^7$ $>10^{15}$ **	$>2\times10^7$ $>10^{16}$ *	—	
INTEGRATED CIRCUITS	2×10^{11} *	5×10^{14} **		2×10^{14} **	BOTTOM, 9.6" SIDE, 3.8" TOP, 0.67"	TO SHIELD RNS COMPONENTS WOULD REQUIRE ~320 LBS

* NEUTRON FLUX $N/CM^2/SEC$

** NEUTRON INTEGRATED FLUX NVT

respectively. Some fluorocarbon and silicone elastomers which showed some softening at 6×10^8 ergs $g^{-1(C)}$ in air were able to survive 1×10^{10} ergs $g^{-1(C)}$ when irradiated in other environments. With judicious selection of suitable materials, no problems should be encountered in the use of sealant materials in the RNS. See Reference 5-6.

Wire and Cable Insulation. Organic and inorganic materials for wire and cable insulation are available that will survive radiation doses of 10^8 rads (C) and 10^{11} rads (C), respectively. Tests have shown that: Teflon is one of the most sensitive materials to radiation and has shown damage as low as 10^3 rads and is obviously not a suitable materials for this application: Polyethylene has little degradation as an insulation for total doses up to about 10^8 rads (C): Irradiated polyolefin has resisted damage when exposed to a total dose of 5×10^8 rads, and ceramic enamel insulated wire has tolerated doses up to 1.5×10^8 rads and $4 \times 10^{17} n/cm^2$ (E > 0.1 MEV). See Reference 5-7.

Available insulation material offers the opprtunity for selecting several materials that will perform satisfactorily in the extreme temperature and vacuum environments combined with the high radiation levels.

Tranducers. Transducer radiation effects problems usually involve the mechanical properties of constituent materials. With the exception of Teflon and certain other organics most space-oriented materials tolerate the nuclear environment for the 10 mission period as can be seen from Table 5-14. In each case, however, the transducers must be selected with due consideration to the radiation-altered properties of their materials; if this is done properly, no significant transducer problems are foreseen for the 10 mission life of the RNS vehicle.

Many types of transducers are suitable for use in radiation environments. By selecting those with only inorganic parts, radiation effects can be tolerated from 10^{15} to 10^{18} n/cm^2 and about 10^{10} erg $g^{-1(C)}$ gamma radiation doses. In the case of thermocouples, considerably higher fluences can be tolerated. Transducers containing organic materials, and particularly semi-conducting devices, must be given special consideration because of lower tolerances to nuclear radiation. Degradation can be expected to occur within a gamma exposure range from 10^7 to 10^{12} ergs $g^{-1(C)}$ and neutron exposure between 10^{13} n/cm^{-2} and 10^{17} n/cm^{-2}. See References 5-8 and 5-9.

Semi-conductor Devices and Circuits. It can be anticipated that the flight control electronics shown in Fig. 5-23 will depend entirely upon semi-conductor devices to perform its function. Both the Laser radar receiver and the command TLM receiver may also utilize semi-conductor devices. These devices are, by several orders of magnitude, the weakest link in the electronics chain with respect to radiation tolerance. It can be expected that gamma exposures of 1×10^4 to 1×10^{10} erg $g^{-1(C)}$ and gamma neutron fluences of 5×10^{11} to 5×10^{15} n/cm^{-2} (E > .1 MEV) may produce significant degradation of performance of transistors depending on type and application. See Reference 5-10.

The use of these solid state devices may be divided into analog (continuous) circuitry and digital (step function) circuitry. The present analog circuitry may be expected to deteriorate at neutron fluence levels of about 10^9 to 10^{11} fast n/cm^2. State-of-the-art in radiation hardened circuitry depends upon power levels required of the particular circuit but good designs may be expected to tolerate levels of 10^{12} to 10^{14} n/cm^2 without failure.

Digital circuitry, i.e., integrated circuits, will likely comprise the majority of the electronics. These devices are inherently more tolerant of the nuclear environment since their "step function" operation allows for large ratios of actual gain versus design gain. The gain or β of transistors is the property which severely deteriorates in the radiation environment. For the RNS application present day integrated circuits can be expected to tolerate levels of 10^{13} n/cm^2. Levels of 2×10^{14} n/cm^2 are feasible with radiation-hardened devices. The primary problem in decreasing radiation activity of IC is one of controlling deposited resistance values rather than one involving the basic semi-conductor device.

Reviewing the levels which can reasonably be tolerated in the light of the adapter section environment, the necessity for shielding becomes evident. Table 5-14 shows thickness and weight for a suitable shield enclosure to protect the electronic equipment in the adapter section. These parameters assume the use of a hydrogenous material, such as water (H_2O) or polyethylene $(CH_2)_n$, of density 1.0. It is reasonable to assume that the rapidly developing semi-conductor industry will be able to supply off-the-shelf radiation hardened devices capable of tolerating (within the proper circuitry) a 2×10^{14} n/cm^2 neutron environment. A tolerance of 5×10^{14} n/cm^2 could be anticipated for some circuitry allowing shielding requirements to be somewhat less stringent.

Solar Cells. Solar cells used as space power sources are subject to long term degradation as a result of exposure to natural and/or artifical radiation such as neutrons and gamma rays that are added to the natural environment by NERVA. The least hazardous of these environment would be the gamma rays followed very closely by electrons. This grouping is quite reasonable since the gamma ray produces compton and photo-electrons which cause subsequent damage. Conversion efficiency for a typical present-day-flight quality solar cell as a function of 1 MEV electron fluence is shown in Fig. 5-24. Electrons generally produce a point defect. Protons generally produce a localized disorder caused by a limited cascade of three to eight displacements per collision. The environment most damaging to solar cells is the neutron which causes a cluster defect of greater magnitude than is caused by a proton.

The major effect of radiation is to produce a defect (point or cluster) which changes the minority carrier lifetime in the base region of the solar cell. The change in lifetime is due to the introduction of trapping or recombination centers. The threshold for the introduction of defects by electrons is 145 Kev and protons 90 ev (see Reference 5-11). Below these energies the particles cannot dislodge the silicon atom by a simple collision. Based on mass-momentum similarities the defects introduced by neutrons should be somewhat similar to those produced by protons.

Gamma dose for 10 missions to the exposed solar panels on the RNS will be less than 5×10^5 rads (C). As Fig. 5-24 indicates essentially no effects will occur at this level. Integrated neutron flux for 10 missions will be about 6×10^{13} n/cm^{-2}. This level must be reduced by one order of magnitude to preclude damage. With normal operation, the solar cells are retracted during engine operation, and consequently those levels are not reached.

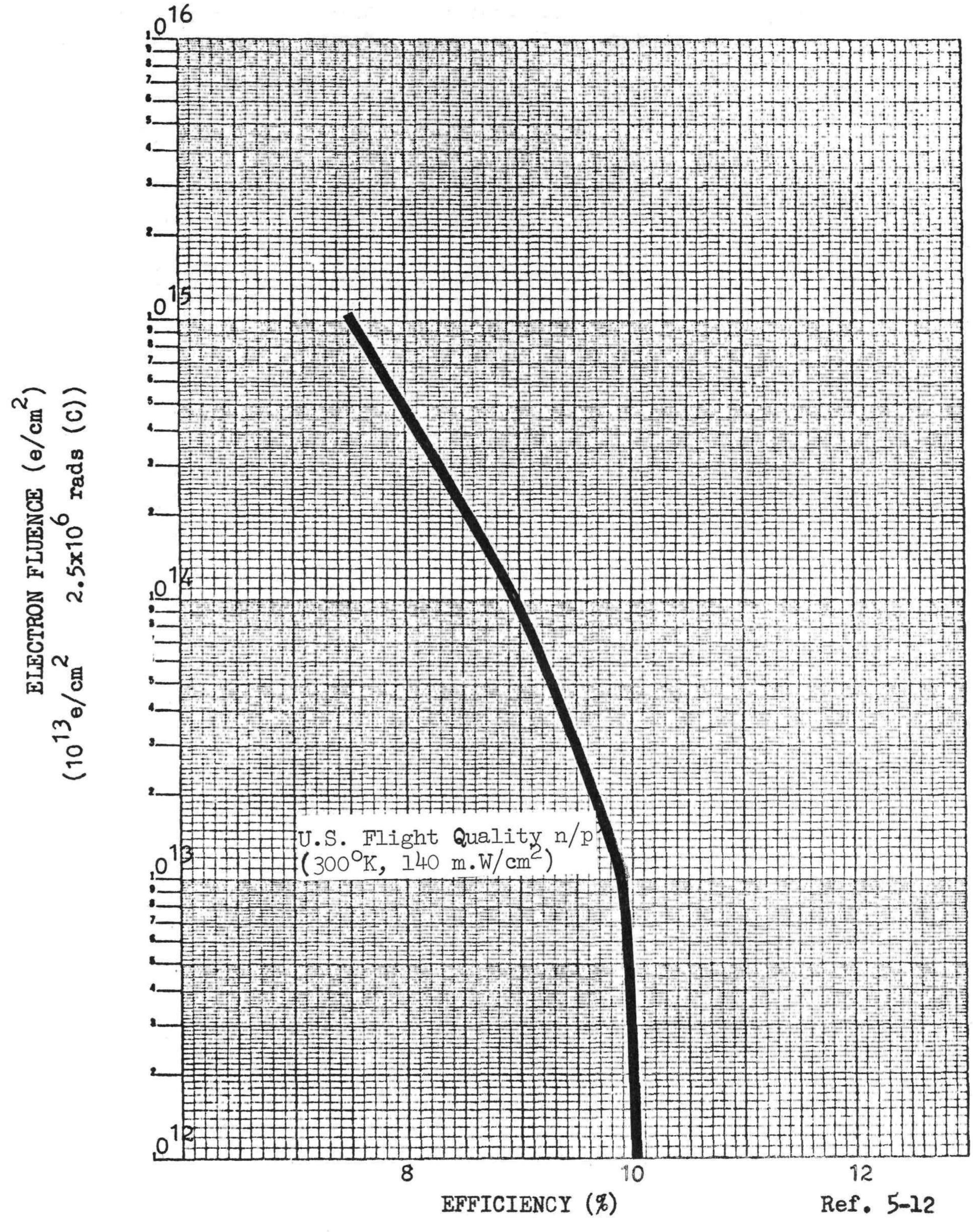

Fig. 5-24 Radiation Effects For Typical Solar Cell

One significant item affecting the use of solar cell power systems is the development of the lithium-doped silicon solar cell. These cells are showing very promising self-annealing characteristics of exposure to various types of radiation. Reference 5-12 indicates 90 to 95 percent recovery of short circuit current after exposure of groups of these developmental cells to 3×10^{14} electrons/cm^2 (1 MEV) (approximately equivalent to 8×10^7 rads (C) and 2×10^{12} fast n/cm^{-2}.

The use of shielding for individual components and/or entire systems should be considered as a trade-off against further development toward enhancing radiation tolerance of some materials and components.

Section 6.0
INTERFACE DEFINITION

The RNS interfaces with many elements. The primary interfaces are with the NERVA engine and with the Space Shuttle. These interfaces are discussed in the subsequent subsections.

6.1 ENGINE INTERFACE

The primary engine interface is shown in Fig. 6-1. The interface plane which is located at engine station 0 is at RNS station 2486. The operational or orbital interface is at RNS station 2440. The 46 inches between these planes contain the engine flyaway control module. This module provides the capability for the engine to separate from the RNS in an autonomous mode and station keep until needed. The engine flyaway module is mated to the engine on the ground and launched with the engine as a single unit in the Space Shuttle. The orbital interface is shown on the left-hand side of Fig. 6-1. Details of the functional elements across the engine interface are shown in Fig. 6-2. Functional requirements across the interface are shown in Table 6-1.

6.2 SPACE SHUTTLE INTERFACE

The Space Shuttle interfaces with each one of the RNS modules. The control and assembly structure module, propulsion module tank system, propulsion module engine system, and the propellant modules are all launched within the 15 by 60 cargo bay of the Space Shuttle. The modules, their deployment, and their interfaces with the Space Shuttle are shown in Fig. 6-3. All of the propellant carrying modules are fueled within the Space Shuttle through a LH_2 line integral to the module and interface at a common location in the Space Shuttle as shown in the figure. In a like manner, all of the modules are configured with an integrated avionics system utilizing a data bus distribution system and local (module) processor that allows for the data checking and status checking in situ during prelaunch, launch, and orbital assembly. In order to provide requirements during launch for the Space Shuttle, the mass properties of the modules as presented in Table 5-2 listed the mass properties of the four payload elements without propellant. Table 5-3 listed the mass properties of the payload elements with propellant loaded so that no propellant carrying module exceeds the Space Shuttle capability of 33,000 pounds. The engine system weighs 36,940 pounds, including shield, and exceeds the current Space System guideline capability. Table 5-4 listed the mass properties of the payload elements fully loaded with propellant.

Engine Weight too high?

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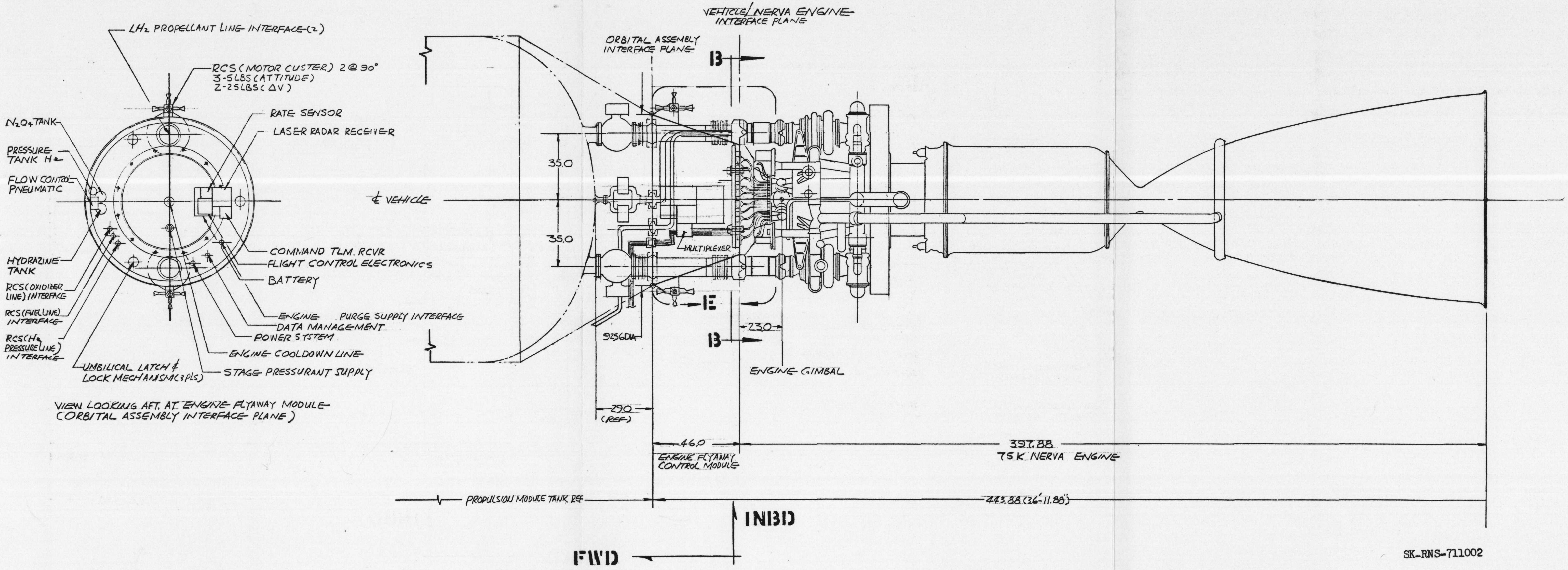

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Fig. 6-1 NERVA Engine Interface

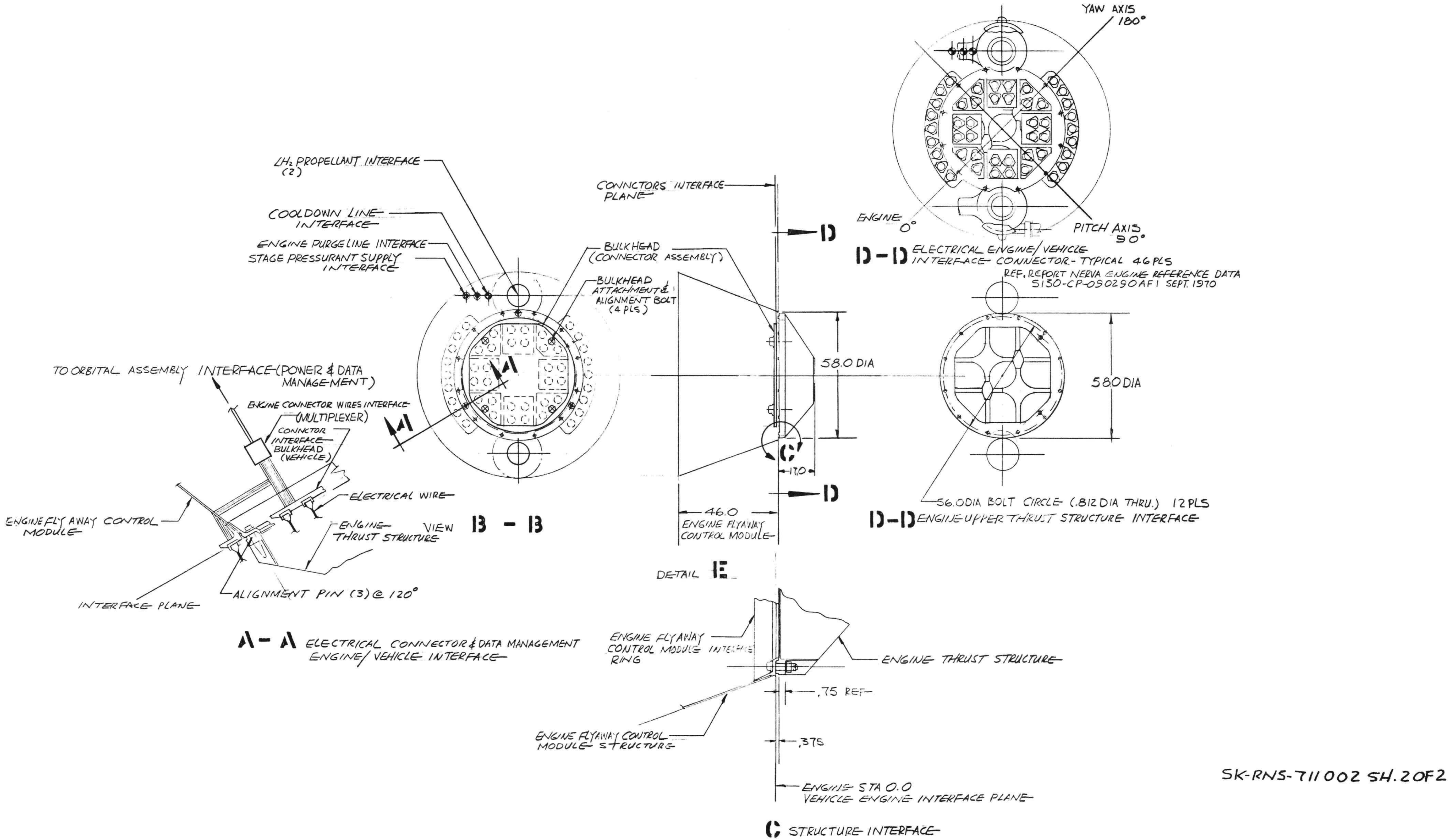

Fig. 6-2 NERVA Engine Interface (Interface Detail)

Table 6-1

RNS/NERVA INTERFACE REQUIREMENTS

FEED SYSTEM

- Pressure – 18 to 26 psia LH_2 (saturated)
- Mass flow rate, $\dot{W} = 91.7$ lb/sec

PRESSURIZATION SYSTEM

- Pressure – 200-750 psia
- Temperature (pressurant gas) – 175^{o} to 25^{o}R
- Mass flow rate, $\dot{W} = 0\text{-}1.5$ lb/sec

ENGINE COOLDOWN LINE

- Mass flow rate, $\dot{W} = 1.7$ lb/sec LH_2
- Pressure – 18-26 psia (saturated)

THRUST VECTOR CONTROL

- Deflection – 5^{o}
- Gimbal rate – 0.25^{o}/sec
- Acceleration – $0.50^{o}/sec^2$

POWER SYSTEM

- Peak – 3.5 Kilowatts
- Average – 2.25 Kilowatts (during engine operation)

TELEMETRY

- Digital Sample rates – 10-100 samples/sec
- Analog frequency range – 0-3000 Hz
- Measurement uncertainties – 5 to 10%

SAFETY

A system to prevent accidental criticality during ground handling, launch, and orbital operation shall be incorporated into the NERVA design.

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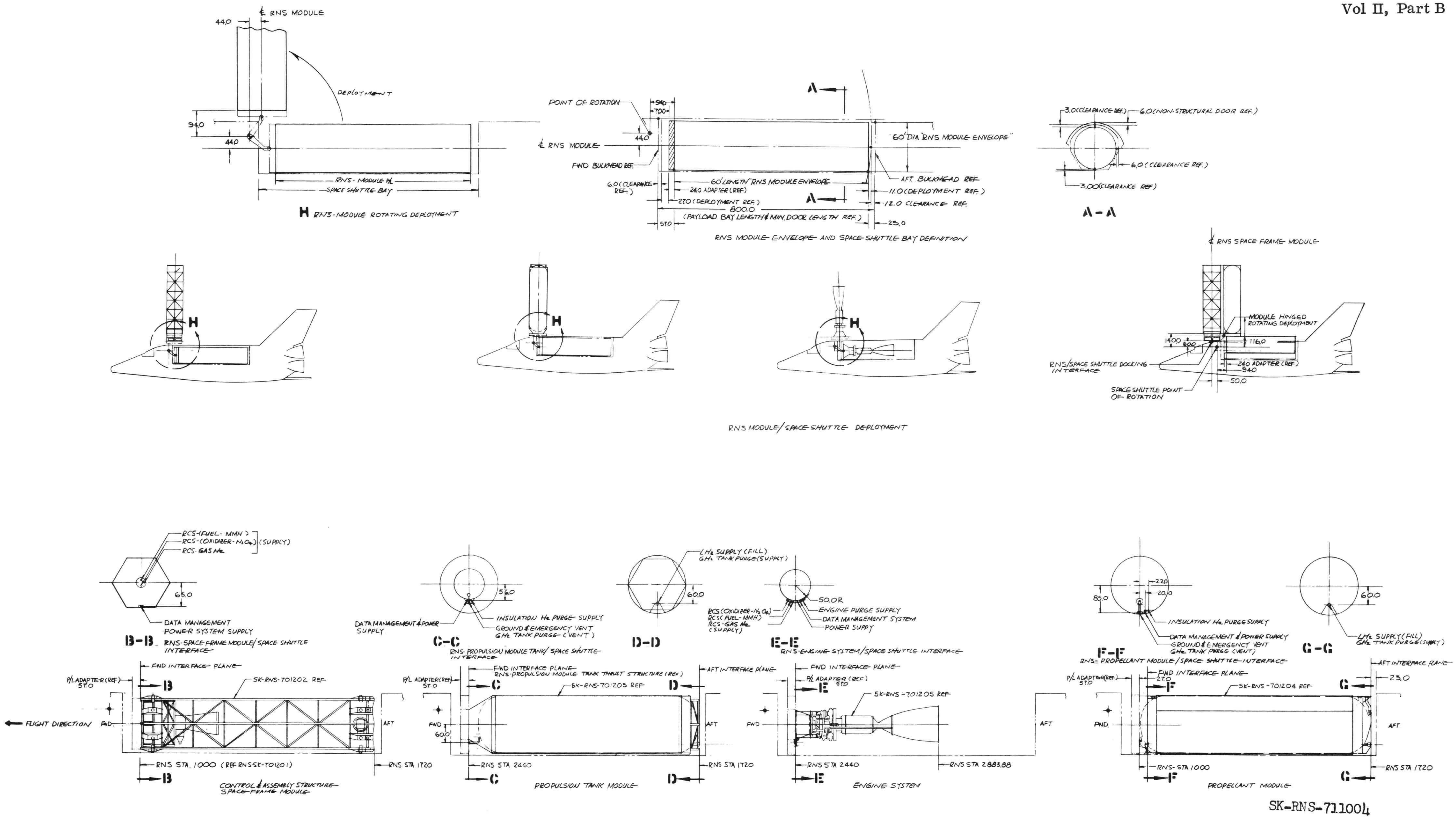

Fig. 6-3 RNS/Launch Vehicle (Space Shuttle) Interface

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