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RE-ENTRY F PROGRAM

RE-ENTRY F
TURBULENT HEAT EXPERIMENT FAMILIARIZATION MANUAL

Prepared for

LANGLEY RESEARCH CENTER
LANGLEY STATION
HAMPTON, VIRGINIA

NAS1-6039

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RE-ENTRY SYSTEMS DEPARTMENT

GENERAL ELECTRIC

[redacted]

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RE-ENTRY F

TURBULENT HEATING EXPERIMENT

FAMILIARIZATION MANUAL

30 October 1967

E. W. RICHARDSON, Manager
Re-entry F Program

GENERAL ELECTRIC
RE-ENTRY SYSTEMS DEPARTMENT
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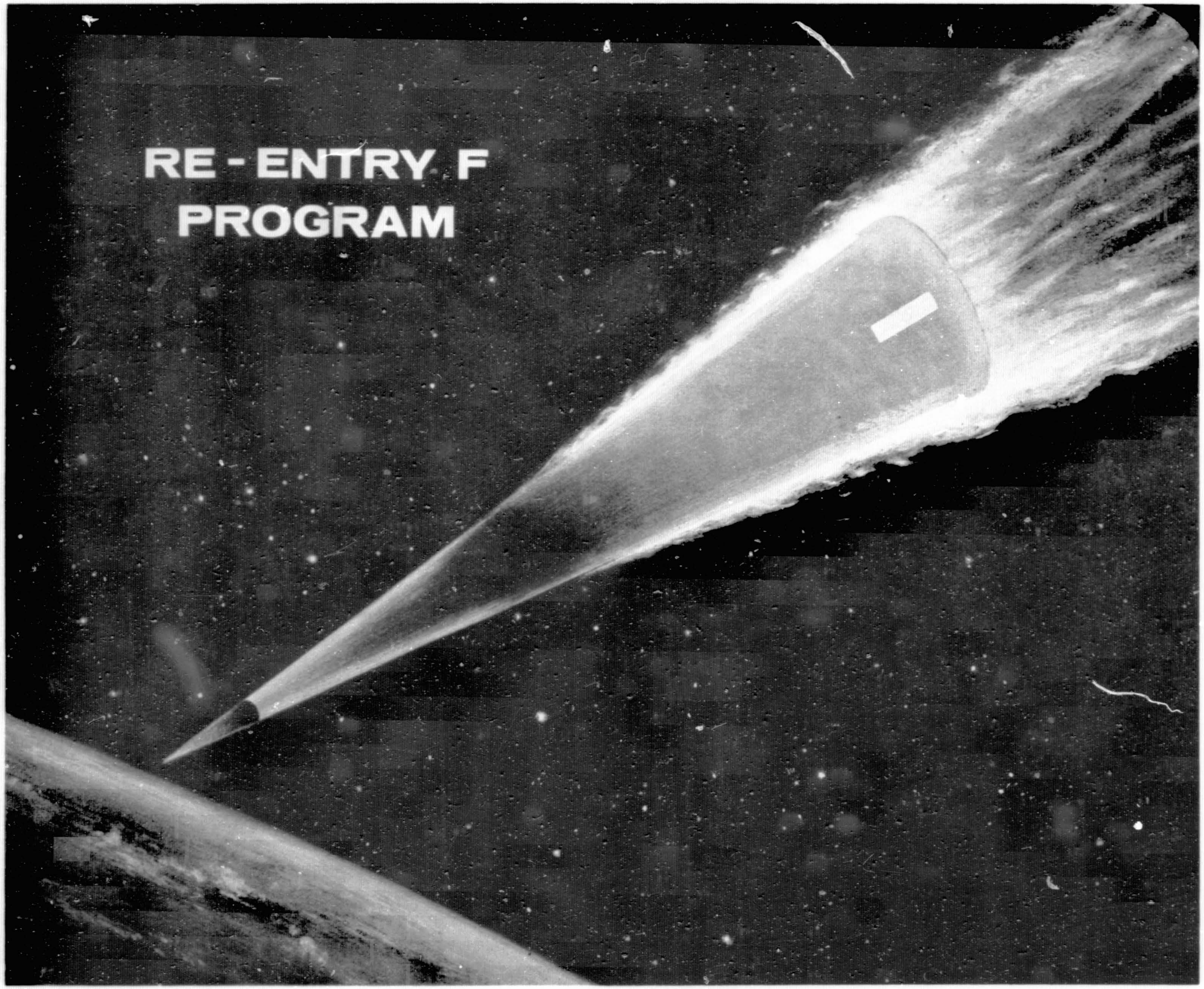
RE-ENTRY F
PROGRAM

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1.0 GENERAL

This Familiarization Manual provides general information on the Re-entry F Turbulent Heating Experiment and Spacecraft. Detailed manuals and other reference documentation are listed in Table 4-2, Documentation and Data Index.

The Re-entry F Turbulent Heating Experiment is a basic research flight experiment for the Langley Research Center (LRC), of the National Aeronautics and Space Administration. The purpose of the experiment is to obtain measurements of fully turbulent convective heating and to determine the transition Reynolds number at conditions representative of advanced re-entry vehicles. The Spacecraft portion of the project is conducted under NASA Contract No. NAS 1-6039 by the General Electric Company, Missile and Space Division, Re-entry Systems Department. The launch of the Re-entry F Spacecraft is scheduled for January 1968. The contract start date was June 1966 and the completion date is February 1968.

1.1 Turbulent Heating Experiment Summary

The NASA Re-entry F Turbulent Heating Experiment is a basic-research experiment designed to obtain real environmental turbulent heating rate data for correlation with ground facility test results and with theoretical methods in an advanced flight regime and to determine the transition Reynolds number on a slender cone at flight conditions typical of advanced vehicles.

The mission profile is shown in Figure 1-1. An instrumented spacecraft will be boosted by a three-stage Scout launch vehicle from the NASA Wallops Island Test Range facility. Re-entry will occur in the broad ocean area off Bermuda. There is no requirement for recovery of the Spacecraft. After first and second stage boost there will be a coast period, during which the spacecraft will go through apogee. The third stage will provide the final re-entry injection velocity increment to the spacecraft. The spacecraft will then be spun-up by the Scout spin table and will separate at 300,000 feet altitude at a re-entry angle of -20°. Data will be telemetered from the Spacecraft throughout the entire flight. The primary data period is from 120,000 feet to calorimeter surface first melt (about 49,000 feet).

The 600 pound spacecraft (Figure 1-2) is a 5-degree half-angle cone, 13 feet long with a 27.3-inch base diameter. The Spacecraft consists mainly of a calorimeter, instrumentation, and an internal structure. With the exception of a graphite nose tip and quartz antenna windows at the aft end, the Spacecraft uses a nonablative heat-sink heat shield made of 0.6-inch thick beryllium. The graphite nose tip is ablatively-cooled. The quartz windows permit radio transmissions from antennae inside the aft section of the Spacecraft. Primary data are obtained from 21 four-element thermal sensors, imbedded in the calorimeter heatshield as shown in Figure 1-3. The thermal sensors provide temperature-profile data through the beryllium. The data will be correlated with other ground and flight test data and analyzed to provide a basis for more accurate heat transfer theory.

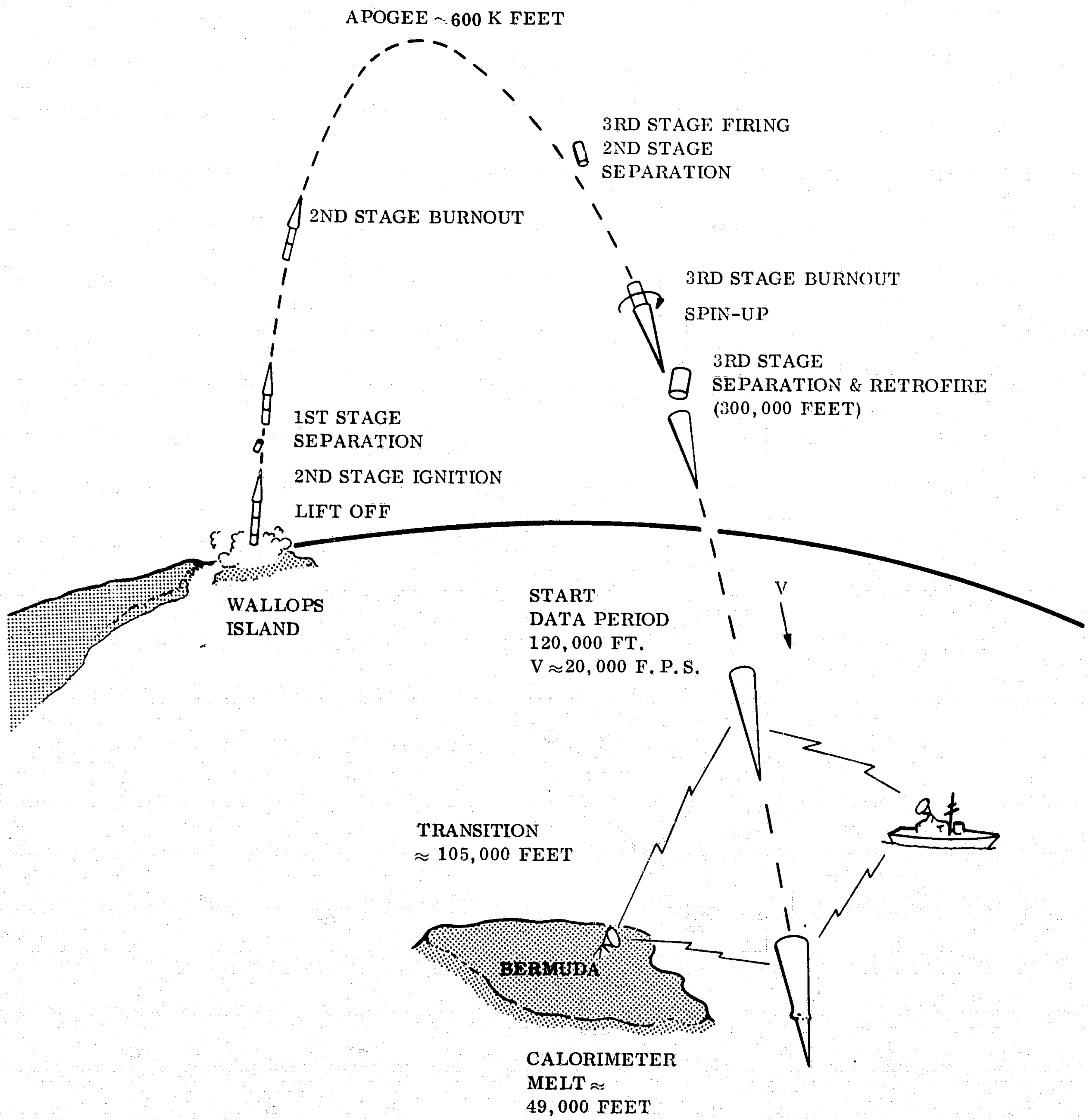


Figure 1-1. Mission Profile

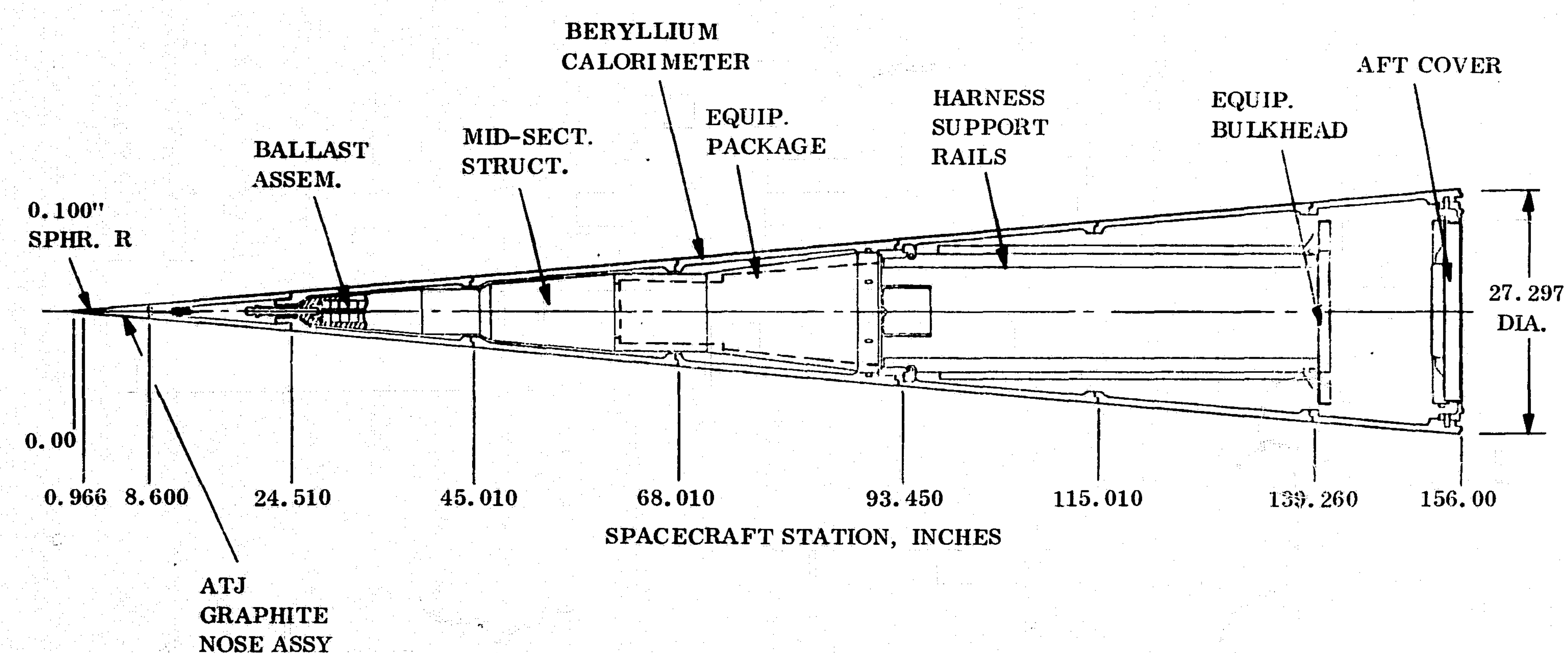


Figure 1-2. General Configuration of the Re-entry F Spacecraft

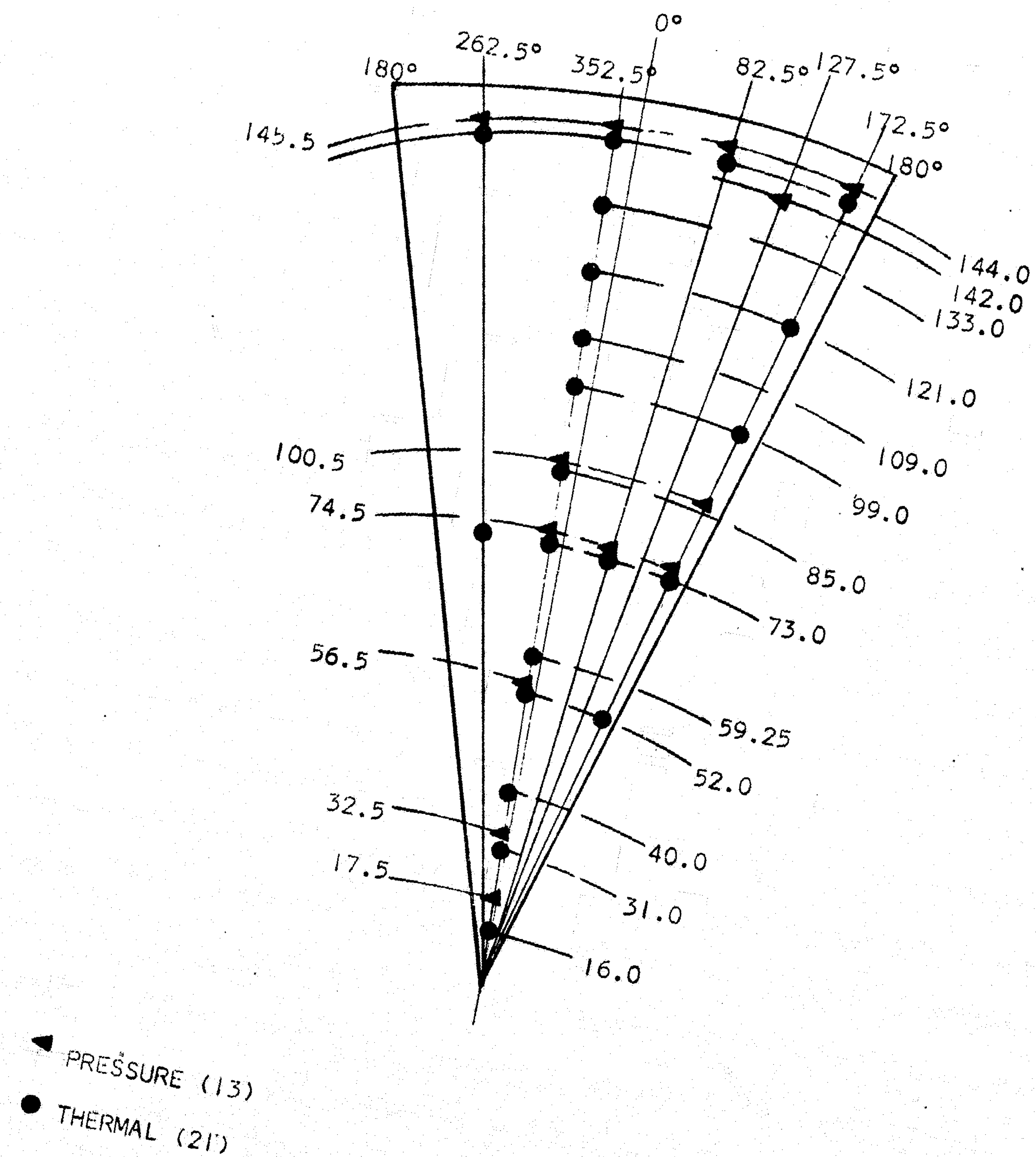


Figure 1-3. Sensor Locations

1.2 Objectives and Data Requirements

1.2.1 Objectives

To obtain measurements of heat transfer in a fully developed turbulent boundary layer with minimal complicating variables.

To determine the Reynolds number at which transition from laminar to turbulent flow occurs.

1.2.2 Data Target Accuracy

Target accuracy goals for data during the primary data period (from 120,000 feet to calorimeter melt) are as follows:

(a)	Heating rates	±15 percent
(b)	Transition Reynolds number	±10 percent
(c)	Body pressure distribution	±0.2 psia
(d)	Base pressure	±0.2 psia
(e)	Resultant angle of attack	±0.25 degree
(f)	Aerodynamic forces	
	(1) Longitudinal	±1g
	(2) Normal and transverse	±0.2g
(g)	Spin rate	±20 degrees per sec.

1.3 Background Information

1.3.1 Re-entry F

In early 1963 NASA Langley made an assessment of experimental and analytical information on the status of aerodynamic heating technology and determined:

(a) the need for establishing confidence at higher velocities in the turbulent heating and transition data determined in ground tests by partial simulation techniques

(b) the need to establish reliable analytical turbulent heating methods for use in the design of advanced vehicles.

This review included ground facility capability and likely progress in ground research, and an evaluation of several flight vehicle systems. As a result, a flight experiment was recommended to obtain reliable turbulent heating data at high boundary layer edge velocities.

Following this recommendation, the LRC Re-entry Science Program Steering Committee reviewed the justification and objectives, and endorsed the experiment which was later approved by LRC Management.

In 1964 LRC received direction to define the overall experiment. During this study technical consultation with industrial organizations having experience in re-entry flight tests and heat transfer was maintained. These results were presented in the Project Development Plan (Revision A) of July 1965 and in April 1966 the Re-entry Systems Department of General Electric Company was awarded the Contract (NAS 1-6039) to provide the hardware for the Re-entry F Turbulent Heating Experiment.

1.3.2 General Electric

The General Electric Re-entry Systems Department (GE/RSD) was organized in 1957 and has had development responsibility for some thirty major re-entry systems involving over 200 flight tests. The above experience has included electronic, optical, and mechanical subsystem design, development, test, and fabrication, and the design and development of associated aerospace ground equipment.

There are four GE/RSD programs of primary significance to the Re-entry F Experiment, because they were concerned with the development of sharp-nose, slender-vehicle, and beryllium heatsink technology. These were the Nike-Zeus Target Program (TVX), Wake Analysis and Control Program (WAC), Graphite Test Vehicle (GTV), and Re-entry Measurements Program (RMP-A).

2.0 MANAGEMENT

The GE management plan for the Re-entry F experiment is presented in RSD Proposal N70543 (25 January 1966), Re-entry F Turbulent Heating Experiment, Volume II, Business Management Proposal (Section B). This document also includes cost-and-incentive formulas, management information, past-performance history, and information about facilities.

The NASA-LRC and General Electric organizations charged with the responsibility for executing the requirements of the Re-entry F (THE) experiment contract with NASA-LRC is given in the subsequent paragraphs.

2.1 Organization and Responsibilities

Four major organizations are involved in the Re-entry F experiment: NASA/Langley Research Center (NASA-LRC) as the sponsoring government agency, under cognizance of NASA/OART; the General Electric/Re-entry Systems Department (GE/RSD) as the spacecraft contractor, the Ling-Temco-Vought Corporation (LTV), as the launch-vehicle contractor, and the NASA/Wallops Island Station as the launch facility.

A Scout Mission Working Group made up of representatives of GE, LTV, NASA-LRC and NASA/Wallops coordinates the entire payload-to-launch vehicle integration and launch support operations. The Mission Working Group is charged with mission responsibility at the working level. It is responsible for the direction of all documentation efforts, the physical integration program, and the operational integration program. The Mission Working Group meets, as required, from time to time, to discuss the interface coordination problems that arise between the Scout launch vehicle contractor and the Re-entry F Spacecraft contractor. These meetings are coordinated and chaired by NASA-LRC.

2.1.1 NASA

The NASA-Re-entry F Project organization is shown in Figure 2-1.

Mr. E.C. Hastings, NASA-LRC Re-entry F Project Manager is responsible to Mr. C.A. Sandahl for the Re-entry F program. Directly responsible to Mr. Hastings is Mr. J.L. Raper, Assistant Project Manager. NASA/LRC Engineers are responsible for monitoring contractor performance in Experiment Planning and Data Analysis, Reliability and Quality Assurance, Project Engineering, and Instrumentation. Mr. Hastings and Mr. Raper coordinate the Re-entry F activities of NASA-LRC Scout Project Office (LRC-SPO), the NASA Wallops Station (launch site), the NASA Bermuda Station (down-range), and the two associate contractors (GE and LTV).

The NASA Wallops Station is the launch site for the experiment, and will provide certain administrative and technical coordination and support in connection with the launch. Launch vehicle operations at NASA Wallops are contracted to LTV under NASA-LRC direction.

The NASA Bermuda Station is the prime data acqusition and tracking site. In addition, a NASA data-acquisition aircraft and ship will be involved.

2.1.2 General Electric Company

The Re-entry F Spacecraft design, development and fabrication is being accomplished by the General Electric Re-entry Systems Department (GE/RSD). GE/RSD is one of six decentralized operating departments of the Missile and Space Division (MSD).

MSD is a division of the Aerospace and Defense Group, one of six groups in the Executive Office of the General Electric Company.

The General Electric organization is shown in Figure 2-2.

The Re-entry Systems Department is headed by Mr. Mark Morton, General Manager. Reporting directly to Mr. Morton is Mr. R.L. Hammond, Manager of Ballistic Systems Programs, including the Re-entry F Program. Immediately responsible to Mr. Hammond is Mr. E.W. Richardson, Re-entry F Program Manager. The GE Re-entry F program team is shown in Figure 2-3.

2.2 Project Implementation Plan

The Re-entry F Turbulent Heating Experiment project includes the design, fabrication, and delivery of hypersonic re-entry spacecraft and field support services, as specified in NASA Document No. L-6345-A Exhibit A (28 April 1966), Statement of Work/Re-entry F Turbulent Heating Experiment.

The period of performance for this design, fabrication, and delivery is approximately 20 months from the date of Contract No. NAS 1-6039 (27 June 1966); i.e., until approximately February 1968. Launch of the flight spacecraft is contemplated for January 1968.

The places of performances are GE facilities in the Philadelphia (Pa.) metropolitan area; the launch-vehicle contractor's (LTV) plant at Dallas, Texas; and two NASA facilities: Wallops Station (Wallops Island, Virginia) and Goddard Space Center (Greenbelt, Maryland).

GE provides the following items:

(a) Four spacecraft: Engineering development model and three identical prime spacecraft (a prototype and 2 flight S/C)

(b) Aerospace Ground Equipment (A.G.E.)

(c) Spares

(d) Documentation and data

2.2.1 Spacecraft

The project requires the fabrication of three instrumented hypersonic re-entry spacecraft: prototype, flight, and backup. The prototype and backup spacecraft are to be retained at a GE facility until after launch of the first flight spacecraft. The flight spacecraft is to be

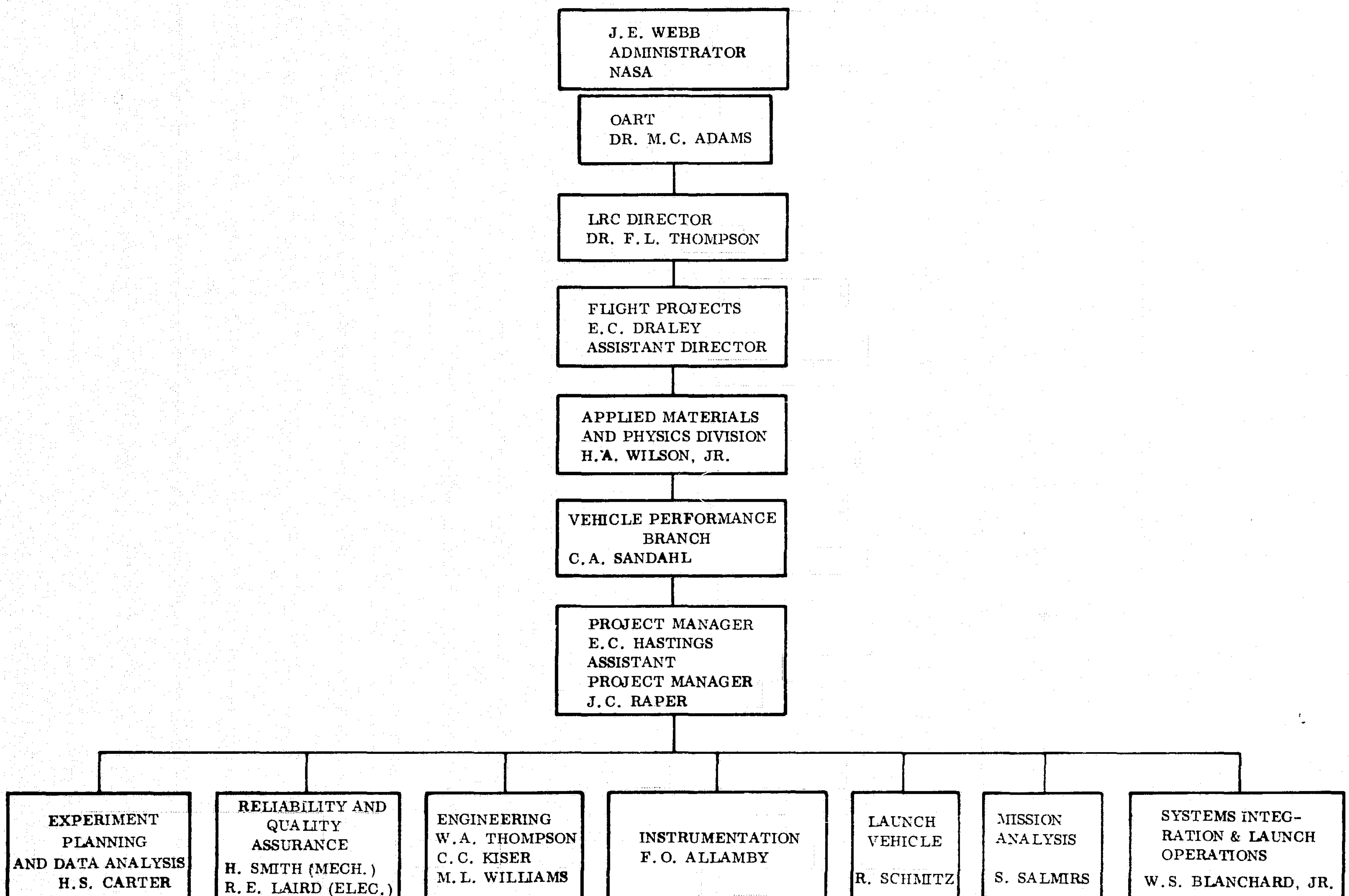


Figure 2-1. Re-entry F-NASA Project Organization

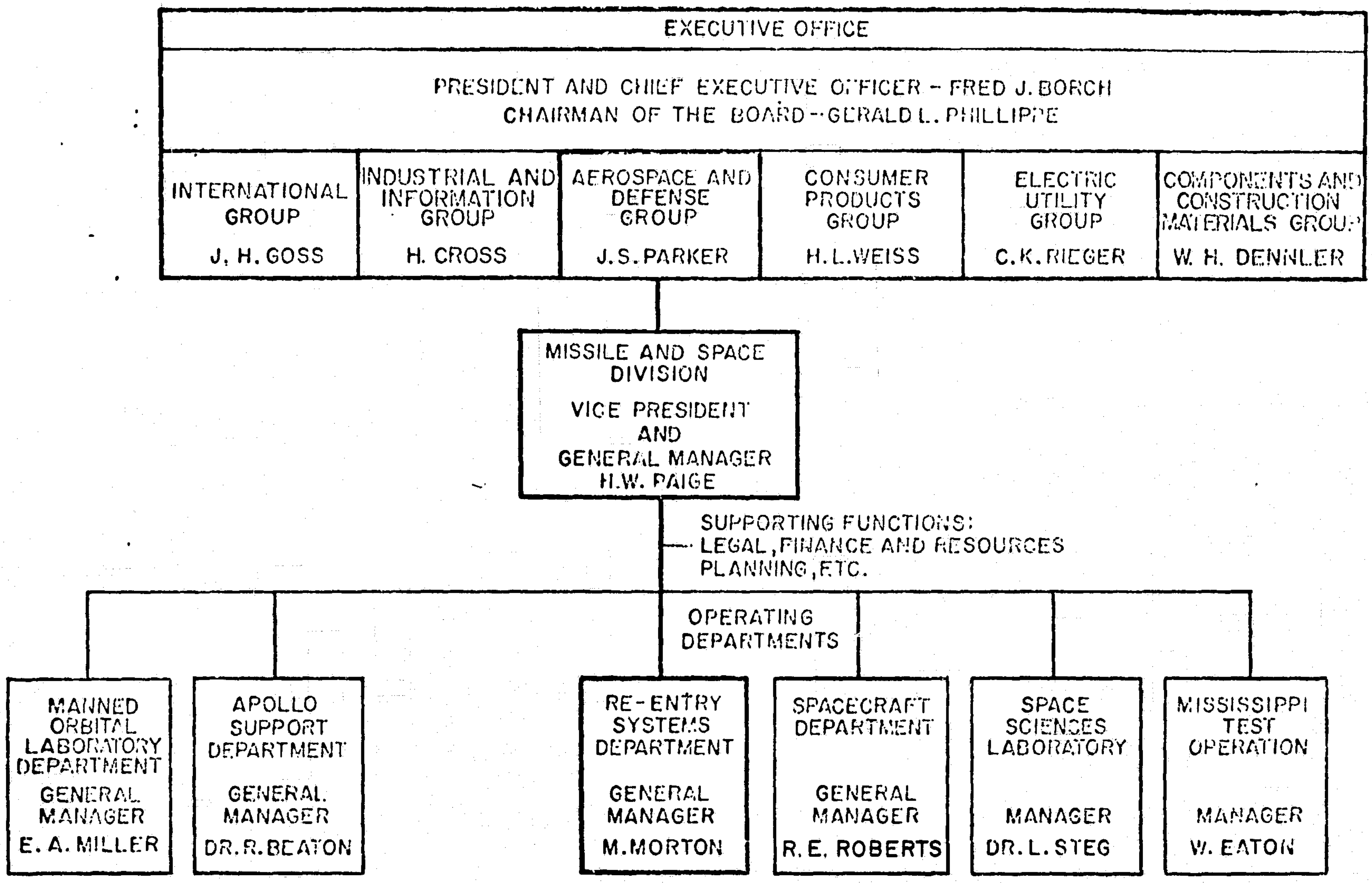


Figure 2-2. GE Organization

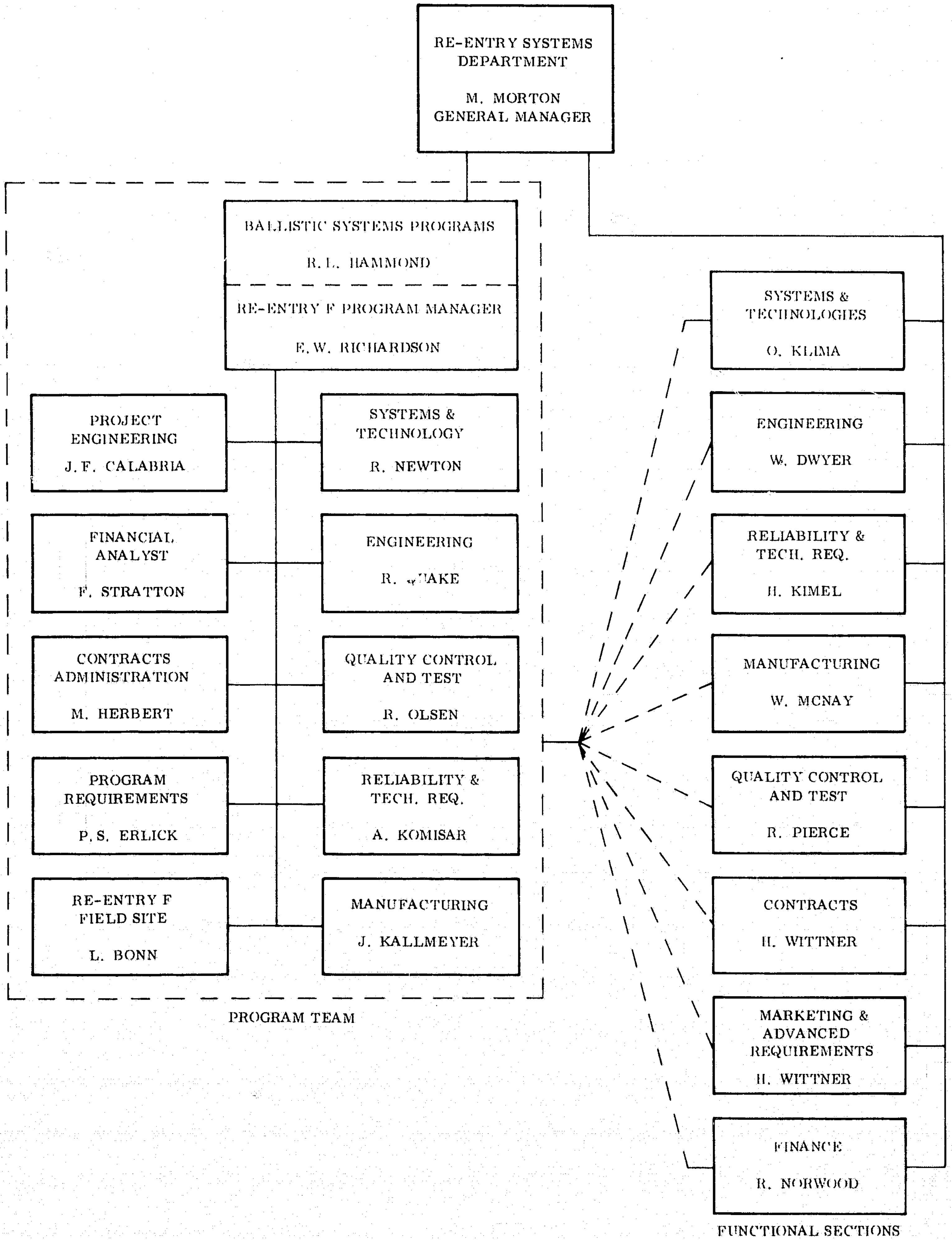


Figure 2-3. RSD Organization, Re-entry F Program

delivered, at the completion of flight assurance tests, to the NASA Wallops Station where it will be launched on top of a 3 stage Scout vehicle. The prototype spacecraft will be ultimately delivered to the NASA Langley Research Center at Langley Station, Hampton, Virginia. The backup spacecraft will be ultimately delivered to the NASA Langley Research Center or to the NASA Wallops Station, according to instructions from the contracting officer. Acceptance of the three spacecraft shall be as follows:

(a)	Prototype Spacecraft	Preliminary acceptance upon satisfactory completion of functional checks following qualification tests. Final acceptance upon completion of radio-frequency interface (RFI) investigations and fit checks.
(b)	Flight Spacecraft	Preliminary acceptance upon completion of flight assurance tests. Final acceptance upon completion of prelaunch checkout tests.
(c)	Backup Spacecraft	Final acceptance upon completion of functional checks following flight assurance tests.

2.2.2 Aerospace Ground Equipment

Aerospace Ground Equipment (AGE) will be provided for handling and checkout of the spacecraft prior to flight. There are two types of AGE: mechanical and electrical. The AGE will be fabricated, tested and delivered, as required, for the support of the spacecraft, and will ultimately be delivered to NASA-LRC.

2.2.3 Spares

Spares will be delivered, as required, for the field support of the spacecraft, and will ultimately be delivered to NASA-LRC. Available spares are defined in the Field Support manual.

2.2.4 Documentation and Data

Documentation and data include reports, design data, documentation, oral briefings, photographs, etc. These are prepared and submitted as specified in NASA Document No. L-6345A, Exhibit A.

Data Requirements List Matrix

The following matrix, Table 2-1, provides a list of contract data requirements and the frequency with which this data is issued.

2.3 Spacecraft Schedules

The Re-entry F Master Schedule is shown in Figure 2-4.

TABLE 2-1. DATA REQUIREMENTS LIST MATRIX

RE-ENTRY F

CONTRACT DATA REQUIREMENTS LIST

CDRL SEQ NO.	DATA	REFERENCE TASK	CUSTOMER ACTION	FREQUENCY: AS REQUIRED	ONE TIME	BI-WEEKLY	MONTHLY	QUARTERLY	REMARKS
001	PRELIMINARY AND FINAL DESIGN REVIEWS	[illegible]	APPROVAL	X					5 WORKING DAYS PRIOR TO REVIEW
002	AGENDA FOR ORAL PRESENTATION	[illegible]	REVIEW	X					7 DAYS PRIOR TO PRESENTATION
003	ORAL PRESENTATION SUMMARY	[illegible]	INFO	X					AT PRESENTATION
004	NASA-PERT NETWORK	6.1.1	INFO			X			45 DAYS AFTER CONTRACT AWARD
005	FINANCIAL MANAGEMENT	6.1.2	REVIEW				X		
006	ENVIRONMENTAL CRITERIA SPECIFICATION	6.2.1	APPROVAL	X					DURING DESIGN PHASE
007	DESIGN CRITERIA SPECIFICATION	6.2.2	APPROVAL	X					DURING DESIGN PHASE
008	MATERIAL, PARTS AND PROCESS	6.2.3	REVIEW	X					DURING DESIGN PHASE
	SPECIFICATION								
009	AEROSPACE GROUND EQUIPMENT (AGE)	6.2.4	REVIEW	X					DURING DESIGN PHASE
	SPECIFICATION								
010	TECHNICAL PROGRESS REPORTS	6.3.1	REVIEW				X	X	
011	EMERGENCY ACTION REPORTS	6.3.2	REVIEW	X					
012	SUPPORT PLAN	6.4.1	REVIEW	X					DURING DESIGN PHASE
013	INTEGRATED TEST PLAN	6.4.2	APPROVAL		X				6 WEEKS AFTER CONTRACT AWARD
014	MANUFACTURING PLAN	6.4.3	REVIEW	X					DURING DESIGN PHASE
015	MAINTENANCE PLAN	6.4.4	REVIEW	X					DURING DESIGN PHASE
016	PHOTOGRAPHY PLAN	6.4.5	APPROVAL	X					DURING DESIGN PHASE
017	ANALYTICAL REPORTS	6.5.1	REVIEW	X					
018	TEST RESULTS AND SUMMARIES	6.5.2	INFO	X					2 WEEKS AFTER TEST AND DATA ANALYSIS
019	PRELIMINARY DRAWINGS	6.6.1	APPROVAL	X					
020	FINALIZED DRAWINGS	6.6.2	INFO	X					
021	FIELD SUPPORT MANUAL, OUTLINE	6.7.1	APPROVAL		X				6 MONTHS AFTER CONTRACT SIGNED
	FIELD SUPPORT MANUAL, FINAL	6.7.1	APPROVAL		X				
022	FAMILIARIZATION MANUAL, OUTLINE	6.7.2	REVIEW		X				6 MONTHS AFTER CONTRACT SIGNED
	FAMILIARIZATION MANUAL, FINAL	6.7.2	REVIEW		X				
023	GROUND SUPPORT EQUIPMENT MANUALS	6.7.3	REVIEW	X					ACCOMPANIES EQUIP'T
024	PAYLOAD DESCRIPTION DOCUMENT	6.8.1	APPROVAL		X				3 MONTHS PRIOR TO LAUNCH
025	GENERAL DOCUMENTATION	6.9	INFO	X					
026	RELIABILITY PROGRAM PLAN	R 2.2	APPROVAL		X				30 DAYS AFTER CONTRACT AWARD
027	RELIABILITY PROGRAM REVIEWS	R 2.3	REVIEW	X					15 DAYS AFTER EACH REVIEW
028	LIST OF SUBCONTRACTORS AND SUPPLIERS	R 2.6.2	REVIEW	X					FIRST SUBMITTAL 15 DAYS PRIOR TO PRELIMINARY DESIGN REVIEW
029	TRADE-OFF STUDIES	R 3.3	REVIEW	X					SUMMARIZED IN MTHLY PROGRESS RPT.
030	FAILURE MODE, EFFECTS AND	R 3.4	REVIEW	X					SUMMARIZED IN MTHLY PROGRESS RPT.
	CRITICALITY ANALYSES								
031	DESIGN REVIEW REPORTS - CONTRACTOR	R 3.6.1	REVIEW	X					15 DAYS AFTER EACH DESIGN REVIEW
032	DESIGN REVIEW REPORTS -	R 3.6.2	REVIEW	X					15 DAYS AFTER EACH DESIGN REVIEW
	SUBCONTRACTOR								
033	FAILURE MALFUNCTION REPORTS	R 3.7	INFO	X					3 DAYS AFTER EACH FAILURE
034	FAILURE ANALYSIS AND CORRECTIVE	R 3.7.4	REVIEW	X					21 DAYS AFTER EACH FAILURE
	ACTION REPORTS								
035	PARTS AND MATERIALS QUALIFICATION	R [illegible]	REVIEW	X					
	TEST SPECIFICATIONS								
036	PARTS AND MATERIALS APPLICATION	R 3.9.6	REVIEW		X				15 DAYS AFTER EACH REVIEW
	REVIEWS								
037	EQUIPMENT LOGS	R 3.10	INFO	X					ACCOMPANY ASSEMBLIES
038	EQUIPMENT LOGS SUMMARY REPORT	R 3.10	REVIEW		X				45 DAYS PRIOR TO EACH LAUNCH
039	TEST SPECIFICATIONS AND PROCEDURES	R 4.3.3	REVIEW	X					21 DAYS PRIOR TO USE
040	RELIABILITY EVALUATION PROGRAM	R 4.5	REVIEW	X					30 DAYS AFTER EACH REVIEW
	REVIEWS								
041	REVIEW TEAM REPORT	R 4.6	REVIEW	X					7 DAYS AFTER EACH REVIEW
042	RELIABILITY PROGRESS REPORTS	R 5.2	REVIEW				X		INCL. IN MTHLY TECH. PROGRESS REPORTS
043	QUALITY PROGRAM PLAN, PRELIMINARY	Q 3.1.1	REVIEW		X				30 DAYS AFTER CONTRACT AWARD
044	QUALITY PROGRAM PLAN	Q 3.1.1	APPROVAL		X				60 DAYS AFTER CONTRACT AWARD
045	QUALIFICATION STATUS LIST	Q 4.3.5	APPROVAL				X		90 DAYS AFTER CONTRACT AWARD
046	PROCUREMENT SPECIFICATIONS	Q 5.3.4	REVIEW	X					
		APPX A							
047	INSPECTION AND TEST PROCEDURES	Q 7.3.1	REVIEW	X					15 DAYS PRIOR TO USE
048	END ITEM TEST PLAN	Q 7.4.2.1	APPROVAL		X				30 DAYS PRIOR TO USE
049	END ITEM TEST AND INSPECTION	Q 7.4.2.2	APPROVAL	X					30 DAYS PRIOR TO USE
	PROCEDURES								
050	PROCESS CONTROL PROCEDURES	Q 7.5.4.1	REVIEW	X					15 DAYS PRIOR TO USE
051	STORAGE PROCEDURES FOR END ITEMS	Q 11.5	REVIEW	X					90 DAYS PRIOR TO USE
052	SPECIAL SAMPLING PLANS	Q 12.3	REVIEW	X					21 DAYS PRIOR TO USE
053	QUALITY STATUS REPORT	Q 14.2.1	REVIEW				X		INCL. IN MTHLY TECH. PROGRESS REPORTS
054	NARRATIVE END-ITEM REPORT	14.2.4	INFO	X					ACCOMPANY ASSEMBLIES
055	QUARTERLY SUMMARIES OF QUALITY	Q 15.2	REVIEW					X	
	AUDITS								

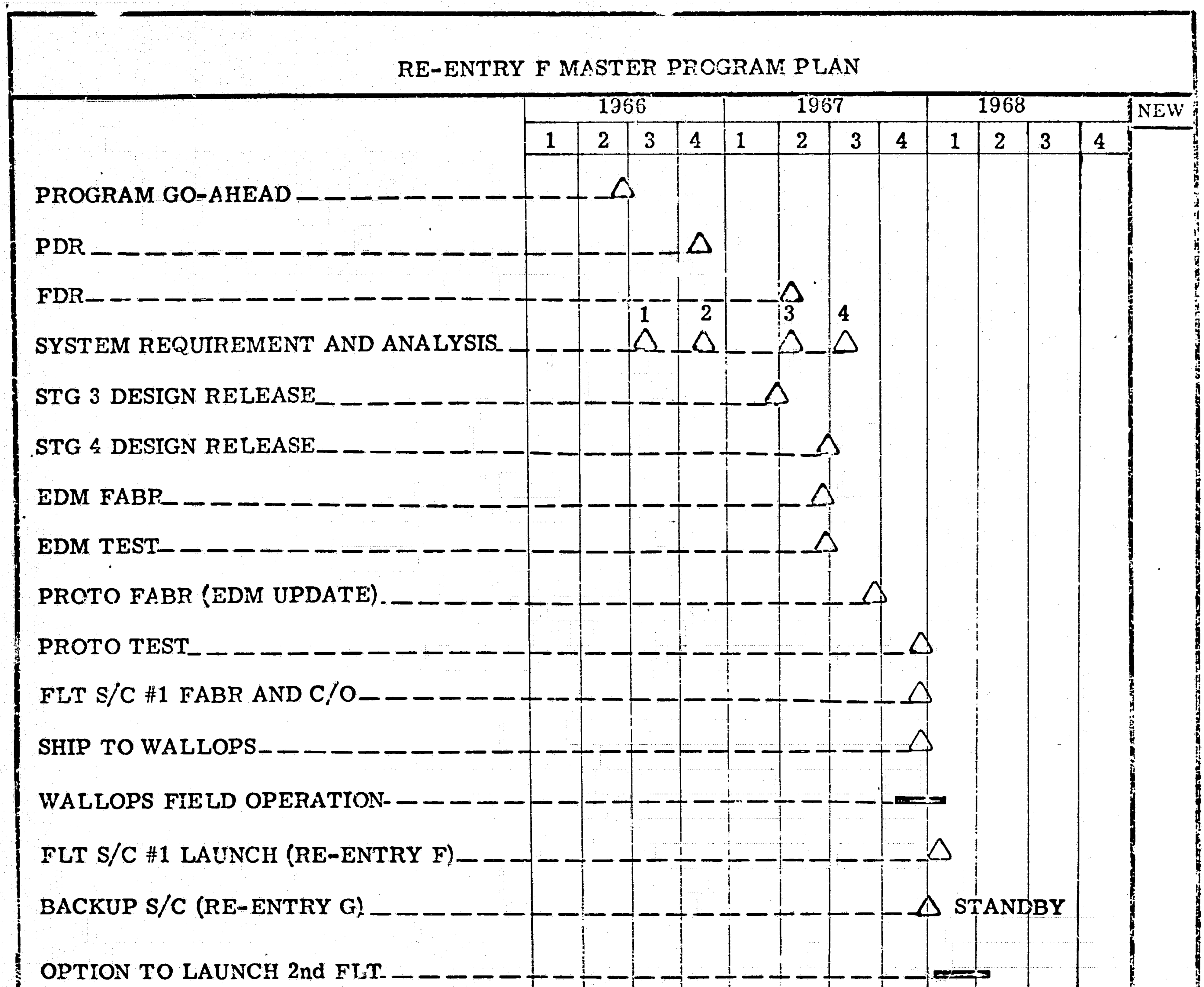


Figure 2-4. ReF Master Program Plan

3.0 TECHNICAL

3.1 Experiment Requirements

Experiment requirements have been defined for the following program objectives:

(a) Obtain measurements of heat transfer in a fully-developed turbulent boundary layer on a simple shape with minimal complicating variables.

(b) Determine the Reynolds number at which transition from laminar to turbulent heat flow occurs.

There are three major categories of experiment requirements: design, data, and data analysis. These requirements are based upon the design objectives given below.

3.1.1 Design Objectives

From a given baseline which is established by the following:

(1) Three-stage Scout launch vehicle

(2) $V - \gamma$ (velocity - re-entry angle) restraints

(3) Spacecraft size, configuration, and material

(4) Nose configuration and material

(5) Spin stabilization

The design requirements of the Re-entry-F experiment are based on achieving the following objectives:

(1) Maximal turbulent-test time

(2) Minimal complicating variables

(3) Maximal edge mach number and Reynolds number

(4) Meeting of data-accuracy goals (paragraph 3.1.2)

(5) Angle of attack, less than 1 degree (below 120,000 ft)

Specific design requirements for the spacecraft are given in Table 3-1.

TABLE 3-1. SPACECRAFT DESIGN REQUIREMENTS

CONFIGURATION	GIVEN OR REQUIRED	DESIGN
General		
Length	13 feet maximum	13 feet
Cone half angle	5 degrees	5 degrees
Weight	600 pounds maximum	600 pounds Ballasted
Calorimeter		
Material	metallic	beryllium
Thickness	--	0.6 inches
Surface discontinuities	minimum	minimum
Nose Tip		
Material	ablatively cooled	ATJ graphite
Tip radius	less than 0.15 inch	0.10 inch
Interface structure and antennae		
(Aft 12 inches of calorimeter)	To include all antennae, umbilical, etc.	Includes all antennae, umbilical, etc.

3.1.2 Data Accuracy and Requirements

The primary data accuracy goals as established by contract, and the data accuracies to be achieved are given in Table 3-2. The type of measurements to be taken are listed in Table 3-3.

TABLE 3-2. PRIMARY DATA ACCURACY GOALS

Parameter	Maximum Error	Design Status
Turbulent heating rate	± 15%	less than 3%
Transition Reynolds number	± 10%	± 3% to ± 10%
Resultant angle of attack	1/4 degree	meets

TABLE 3-3. DATA SYSTEM REQUIREMENTS

DATA CATEGORY	TYPE	OBJECTIVE
Primary	Temperature 21 thermal sensors (4 T/C's each)	• turbulent heating rate • transition
Secondary	Body pressure	• flow field • angle of attack
	Base pressure	• transition
	Motion (Rate Gyros)	• angle of attack • diagnostic • transition
	Force (Accelerometers)	• angle of attack • trajectory • transition
	Temperature	• cold junction
Diagnostic	Internal temperature	• system operation
	Voltage and current	• battery monitor
	Voltage and temperature	• beacon monitor
	Incident and reflected power	• telemetry monitor

3.1.3 Data Period Criteria

The experiment is designed to maximize the turbulent boundary layer period. The experiment period is about 10 seconds (from about 120,000 ft to approximately 49,000 ft), with the turbulent time being about 6.5 seconds (105,000 ft to 49,000 ft).

The sampling rate of individual sensing elements is based on data frequency response requirements. As such, data is monitored at 10, 20, 40, or 60 samples per second (SPS), or is telemetered over continuous channels. The specific measurements and their sampling rates are given in Table 3-4.

3.1.4 Re-entry Environment Requirements

No window requirement exists for the launch or re-entry and the basic requirement is that environmental conditions are such that the telemetry and radar ground stations can acquire the spacecraft track and telemetry data.

3.1.5 System Requirements

The objectives of system requirements are to ensure compliance with the purpose and constraints of the Re-entry F Turbulent Heating Experiment. Criteria have been summarized above. System requirements include data requirements, and requirements for launch, re-entry, transition, and angle-of-attack.

3.1.5.1 Data Requirements

Data requirements are based on primary data accuracy goals for the following parameters: turbulent heating rate, transition Reynolds number, and resultant angle of attack. The maximum error and design status for each parameter are listed in Table 3-2. Data system requirements are listed by category, type, and objective in Table 3-3.

3.1.5.2 Re-entry Requirements

The spacecraft is designed to meet its system requirement after being subjected to the following 3 sigma limits of injection errors, at separation from the Scout third stage:

(a) Velocity . . . . Nominal ± 270 fps

(b) Altitude . . . . Nominal ± 36,500 feet

(c) Path Angle . . . . 20° ± 0.75° DFH

(d) Body Orientation

(1) Scout Programmer and Gyro 0.75 degrees

(2) Third Stage Control System Deadband 0.70 degrees

(e) Body Rate

(1) Third Stage Rate Deadband 2.0 degrees/second

(2) Spring Separation System 0.45 feet/pound/second at 11-inch radius

3.1.5.3 Transition Reynolds Number

The transition Reynolds number is defined as the local Reynolds number, based on length from the tip at which the laminar boundary layer first becomes turbulent. The accuracy goal for transition Reynolds number is ± 10%.

3.1.5.4 Turbulent Heating Rate

The fully turbulent heating rate as calculated from the thermal sensor thermocouple data and beryllium property data shall be accurate within ± 15%.

3.1.5.5 Angle-of-Attack

The spacecraft is required to obtain a minimum angle-of-attack from 120,000 feet altitude until the beryllium surface begins to melt (approximately 49,000 feet). The design goal for the maximum absolute value of angle-of-attack at 120,000 feet is 0.5 degrees 3 sigma including aerodynamic trim. The measurement accuracy goal for angle of attack is ± 1/4 degree.

3.1.6 Spacecraft Design Requirements

Spacecraft design requirements in addition to those summarized in Table 3-1 are:

(a) Weight and Balance

(1) Weight 600 pounds maximum (for more detail see section 3.4.10)

(2) Balance (Figure 3-1)

(aa) Tilt angle (maximum) . . . . 0.02 degree for Spacecraft during re-entry

Tilt angle = angle between principal axis and spin axis

$$= \frac{\sqrt{(I_{xy})^2 + (I_{xz})^2}}{{}^{*}I_{(y\ or\ z)} - I_x} \leq 0.02^\circ \text{ during re-entry}$$

where $I_{x,y,z}$ = moment of inertia $I_{xy} + I_{xz}$ = Products of Inertia

x = roll axis

y = pitch axis

z = yaw axis

(bb) Aerodynamic (velocity vector) axis, to be determined by "banana" (surface mapping) test is summarized in subsection 3.8 and is described in detail in Standing Instruction SI-243890.

(cc) Trim angle (maximum) . . . . 0.1 degree

Trim angle = angle between principal axis and aerodynamic axis

= Tilt angle + banana angle (added vectorially)

= 0.1 degree (max) static

= 0.1 degree (max) rolling

(dd) Dynamic Balance (Spacecraft) = 200 oz. in^2 (as a goal)

*Note: Use smaller of I_y or I_z.

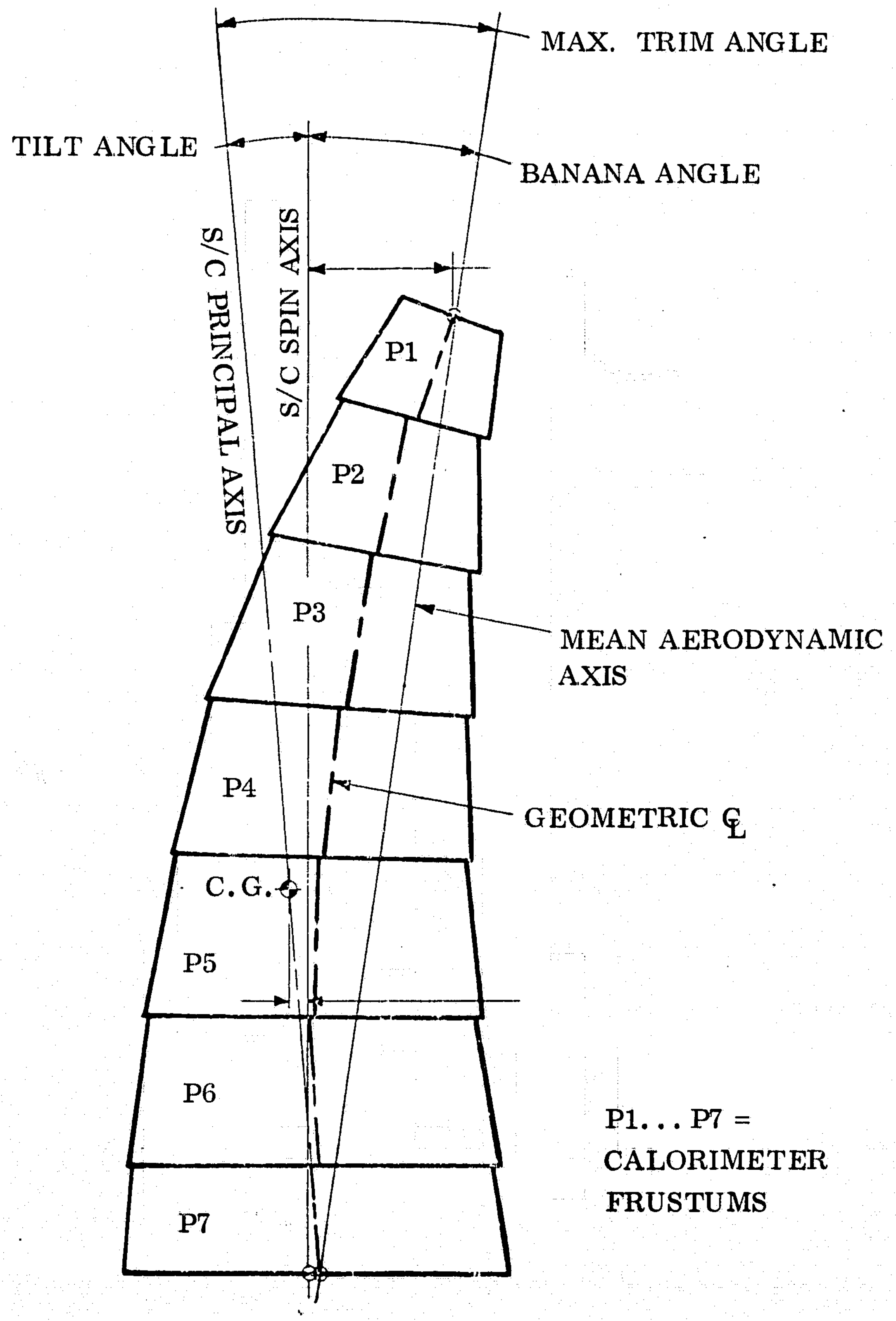


Figure 3-1. Banana Test, Axis Definition

(b) Telecommunications Subsystem (GE/RSD Specification S0150-00-0016)

PDM/FM/FM modulated carrier at VHF frequency of 253.8 MH_Z; transmitter power, 10 dbw. Four telemetry antennae, with longitudinal slotted quartz windows symmetrically located in aftmost 12 inches of the spacecraft; antennae isolated so that failure of one would not preclude transmission by others.

(c) Tracking Subsystem (GE/RSD Specification S0150-00-0016)

The C-Band transponder is compatible with the ground tracking radar and has the following operating characteristics: receiving frequency, 5625 MH_Z; receiving sensitivity -65 dbm; transmitting frequency, 5700 MH_Z; 0.5 μ sec pulses with peak power output in excess of 500 w. Two longitudinal slotted quartz antennae windows 180 degrees apart in aftmost 12 inches of Spacecraft.

(d) Electrical Power and Distribution Subsystem (GE/RSD Specification 0010-08-0008)

There are two 29 vdc batteries (one each for telecommunications and tracking subsystems); each capable of 29 vdc output after 30-day activated wet stand life and capable of providing power for entire flight time plus 30-minute reserve operation. Umbilical cable connection for preflight test and launch checkout.

(e) Performance

Capable of:

(1) obtaining and transmitting real-time data on thermal and pressure performance of calorimeter, and on spacecraft motion, during flight under actual ballistic re-entry thermal and load environments, from liftoff to loss of signal (3-degree radio horizon, approximately 45,000 feet)

(2) continuous performance from T-30 minutes to loss of signal (approximately T+7 minutes), and 45-minute minimum continuous performance

(3) meeting requirements and restraints of Re-entry F experiment, as summarized in section 1 and subsections 3.1 and 3.2

3.1.7 Sensor Design Requirements

Sensors are designed to meet requirements listed in Table 3-4. Accuracy requirements are as follows:

(a) Measurement

Base thermal response . . . Less than 6-BTU/sq ft/sec error

Body Pressure Distribution . . . . ± 0.2 psia

Base Pressure. . . . . ± 0.05 psia

Aerodynamic Forces

Longitudinal . . . . ± 1 g

Transverse . . . . ± 0.2 g

Spin Rate . . . . ± 20°/sec

NOTE: The listed pressure and motion measurement accuracies are to be treated as goals. Larger accuracy limits are permissible where other considerations, such as increased full scale ranges, are considered to be overriding.

(b) Calculation

	Required
Heating Rate	± 15%
Transition Reynolds Number	± 10%
Resultant Angle of Attack	± 1/4°

3.1.8 Nose Tip

The nose tip is an ablatively-cooled ATJ-graphite cone with a tip radius of 0.1 inch. The nose tip is thermally isolated from the calorimeter. Three thermocouples are provided on the nose tip to sense an anomaly resulting in a catastrophic failure. Design details are given in paragraph 3.4.5.

3.1.9 Interface and Internal Structures

The aft 12 inches of the spacecraft constitute the interface structure. The Spacecraft interface ring is required to be mechanically compatible with the interface of the Scout launch vehicle. The internal structure of the spacecraft consists mainly of a substructure of aluminum rings and longerons, enclosed by the calorimeter and separated from it by an adiabatic-boundary air gap. The internal structure encloses and supports the equipment package assembly (instrument package), an electrical harness, and other components. Design details are given in paragraph 3.4.4.

3.1.10 Loads - Aero, Thermo, Mechanical

The spacecraft is designed to meet the following load-limit requirements:

Limit load . . . . anticipated load on structure

Yield load . . . . 1.15 x limit load

Ultimate load . . . . 1.50 x limit load

TABLE 3-4. MEASUREMENT LIST

Type of Sensor	Measurement	Range	Sampling Rate (samples per second)	Quantity
Temperature	Thermal sensors depth			
	Thermocouple No. 1 (.01 inches)	amb to 2350° F	20	21
	No. 2 (.1 inch)	amb to 2350° F	10	21
	No. 3 (.3 inch)	amb to 2350° F	10	21
	No. 4 (.6 inch)	amb to 2350° F	10	21
	Temperature, nose tip	amb to 4200° F	10	1
	Temperature, internal	amb to 200° F	10	4
	Temperature, nose tip	amb to 2200° F	10	2
	Temperature, cold junction reference	20 to 140° F	10	1
	Base heat flux	0 to 150 Btu/ft^2/sec	10	4
	Body pressure	0 to 50 psia	20	1
Pressure	Body pressure	0 to 3 psia	20	12
	Body pressure	0 to 20 psia	40	13
	Base pressure	0 to 0.1 psia	20	2
	Base pressure	0 to 1.0 psia	20	2
Motion	Pitch rate, coarse	0 ± 20° /sec	60	1
	Pitch rate, fine	0 ± 5° /sec	cont	1
	Yaw rate, coarse	0 ± 20° /sec	60	1
	Yaw rate, fine	0 ± 5° /sec	cont	1
	Roll rate, coarse	0 ± 2000° /sec	60	1
	Roll rate, fine	0 to 500° /sec	cont	1
	Pitch Acceleration, coarse	0 ± 10 g's	60	1
	Pitch Acceleration, fine	0 ± 2 g's	cont	1
	Yaw Acceleration, coarse	0 ± 10 g's	60	1
	Yaw Acceleration, fine	0 ± 2 g's	cont	1
	Axial Acceleration, coarse	+2 to −40 g's	cont	1
	Axial Acceleration, medium	+2 to −10 g's	cont	1
	Axial Acceleration, fine	+2 to −3 g's	cont	1

Ground, powered flight, and separation limit loads are given in Table 3-5. Four phases of flight were considered critical for determining the load factors:

Maximum Q (dynamic pressure in lb/ft^2)

Third-stage ignition

Third-stage burnout

Maximum re-entry condition

TABLE 3-5. GROUND, POWERED FLIGHT, AND SEPARATION LIMIT LOADS

Load Condition	Load Factors			
	G_x (g's)	G_n (g's)	G_{np} (g's/inch)	G_{spin} (g's/inch)
Ground Handling	± 3.0	± 3.0		
Spin Test				1.133
Launch	-2.0±3.0	1.5		
Max Q	-4.8±20	± 9.96		
First Stage Burnout	-5.2			
Second Stage Ignition	-3.2±50			
Second Stage Burnout	-10.4	3.0	0.035	
Third Stage Ignition	-5.2±50			
Third Stage Burnout	-12.8			
Spin Up to 70 RPM		0.047		
Re-entry	23.0	+7.0		

The spacecraft is designed to withstand the following conditions during exospheric flight from liftoff to calorimeter melt:

Spin rate. . . . . 70 rpm

Internal Temperature . . . . 20° to 140° F

Time . . . . 7 minutes

The axial acceleration (G_x) during the various phases of flight and other static environment and design conditions are given in Table 3-6.

TABLE 3-6. RE-ENTRY F FLIGHT DESIGN CONDITIONS

Flight Condition	G_x	G_n	G_{np}	h (ft)	Q (lb/A^2)	degrees	M	t (sec.)
Maximum Q	4.8	0	0	35,200	2400	5	2.56	40
1st stage burnout	5.15	0	0	47,000	2130	-	3.2	45
2nd stage ignition	3.20	0	0	145,000	0	0	3.5	80
2nd stage burnout	10.4	3.0	0.035	268,000	0	0	12.2	118
3rd stage ignition	5.2	0	0	545,000	0	0		342
3rd stage burnout	12.8	0	0	400,000	0	0		378

During powered flight there is no significant increase in temperature of either the beryllium calorimeter or the substructures. Re-entry limit loads are as follows:

G_x = 23.0g

G_n = 7.0g

G_{np} = 0.052g/inch

G_{spin} = 0.0139g/inch

Worst-case loading conditions occur during trim flight at an altitude of 40,000 feet. The design conditions are as follows:

Q = 100,000 lb/ft^2

αT = 0.3 degrees

M = 13

G_x = 23 g's

G_n = 7 g's

These conditions are for worst V-γ conditions of

αe = 22 degrees

V_e = 20,400 ft/sec

which are greater than the Scout injection limits.

More detail information on loads can be found in:

(a) Re-entry F Structural Mechanics Stage 3 Release SM-TM 8156-202

(b) Structural Mechanics Technology Loads Release PIR SM 8156-3340

3.1.11 Internal and External Environments

The spacecraft is designed to meet the environmental requirements given in the following GE/RSD specifications:

S0020-02-0015 External Environmental Criteria and System Qualification Test Requirement Specification for Re-entry F Turbulent Heating Experiment

S0020-02-0020 Internal Environmental Criteria and Component Qualification Test Requirements Specification for Re-entry F Turbulent Heating Experiment

Electrical component temperature environment limits under which the components must operate, whether as individual components or as part of the spacecraft system, are defined by the prelaunch phase where the spacecraft and booster are in the erected position on the launcher, and are thus exposed to climatic conditions. However, the Scout Booster will not be exposed to atmospheric temperatures below 30° F or above 100° F; therefore, the spacecraft electrical-components operating-temperature environment extremes for qualification purposes are as follows:

High temperature 140° F

Low temperature 25° F

3.2 Technical Implementation Plan

The following is a summary description of the experiment plan, the program scope, the spacecraft characteristics, and the flight trajectory.

3.2.1 Experiment Plan

(a) Mission . . . . Obtain turbulent heat transfer data during the flight of an instrumented re-entry spacecraft.

(b) Launch Vehicle . . . . Modified three-stage Scout, including a re-entry injection stage (Boosted Re-entry)

(c) Launch site . . . . Wallops Island, Virginia

(d) Impact point . . . . near Bermuda

(e) Launch date. . . . . January 1968

3.2.2 Program Scope

The over-all scope of the Re-entry F Program includes the following technical elements:

(a) Experiment analysis

(b) Sensor analysis

(c) Flight planning

(d) Design

(e) Tests

  (1) Development test

  (2) Qualification test

(f) Engineering development test model

(g) Spacecraft

  (1) Prototype Spacecraft

  (2) Prime flight spacecraft

  (3) Backup flight spacecraft

(h) Aerospace Ground Equipment (AGE)

  (1) Mechanical AGE

  (2) Electrical AGE

(i) Launch operations support - prime spacecraft

(j) Second flight option - backup spacecraft

(k) Data reduction and analysis (not included in present GE contract for Re-entry F)

3.2.3 Spacecraft Characteristics

(a) Shape . . . . . 5-degree half-angle cone with flat base

(b) Length . . . . . 13 feet (maximum)

(c) Base Diameter. . . . 27.3 inches

(d) Weight . . . . 600 pounds (ballasted)

(e) Static Margin. . . . . . . greater than 5 per cent

(f) Ballistic Coefficient 3750

(g) Telemetry . . . . . . PDM/FM/FM (VHF 253.8 MC) - 4 slot antennae

(h) Tracking . . . . . . C-band beacon (Rec. 5.625 KMC) - 2 slot antennae (Trans. 5.700 KMC)

(i) Separation System . . . . LTV Scout Mark II (Modified Standard Scout)

(j) Test Body . . . . . Beryllium, 0.6 inch thick (Calorimeter)

(k) Nose tip . . . . . ATJ Graphite, 0.1 inch radius

3.2.4 Trajectory

(a) Launch site .....NASA facility, Wallops Island, Virginia

(b) Impact ....near Bermuda

(c) Re-entry Velocity .....20,000 feet per second

(d) Re-entry Angle ....20 degrees

(e) Spin rate ...... ≈ 60 rpm (imparted by Injection Stage)

(f) Transition altitude ..... 105,000 feet (estimated)

(g) Melt altitude ...... 49,000 feet (estimated)

(h) Data period .... ≈ 6.4 seconds

The flight trajectory is shown in Figure 3-2.

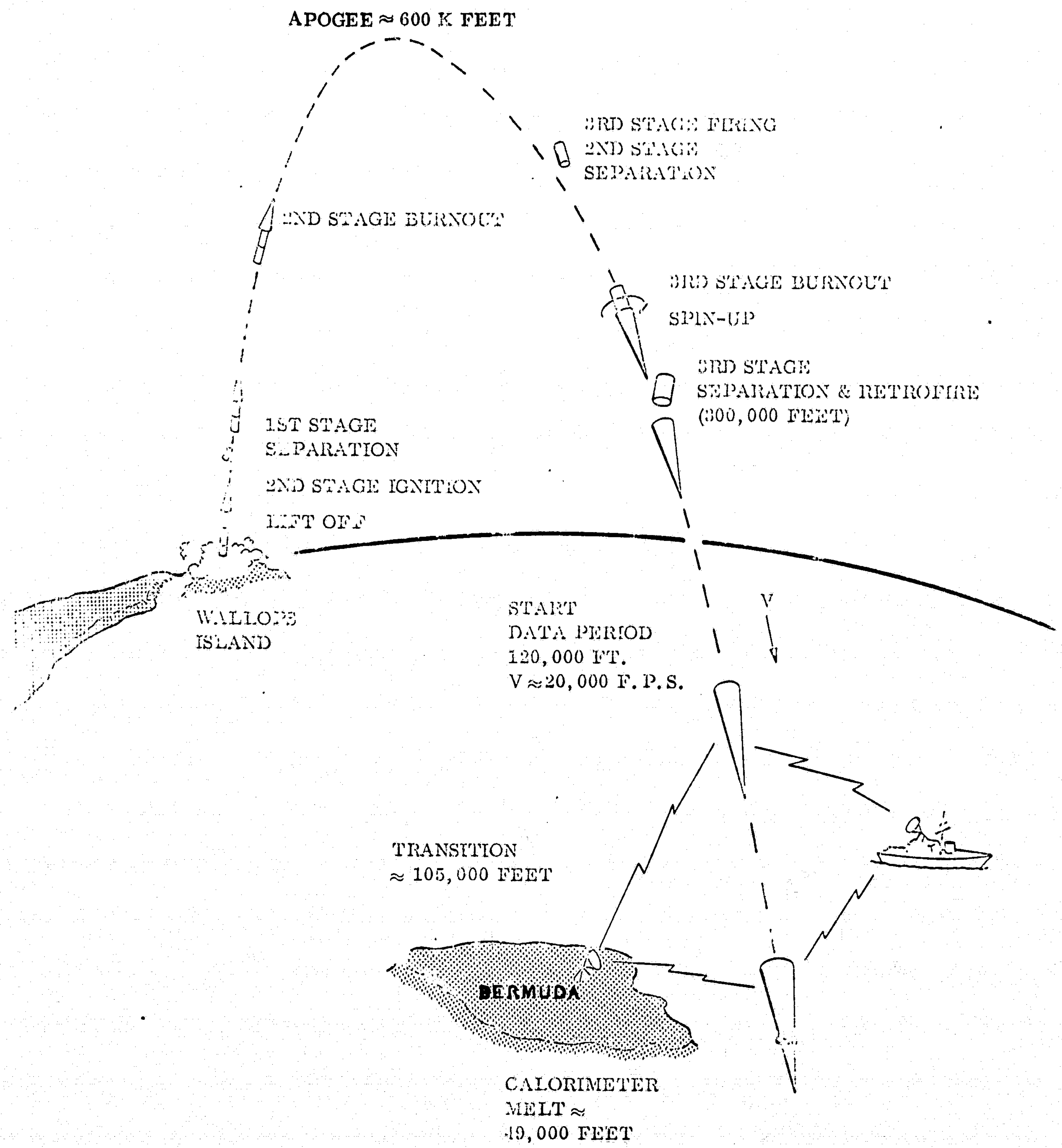


Figure 3-2. Mission Profile

3.2.5 Scout Launch Vehicle

The Scout Launch Vehicle (Figure 3-3) is described in the Scout User's Manual. The Scout is a solid-propellant multistage rocket-powered vehicle equipped with a preprogrammed guidance system whereby each expended stage separates from the vehicle on a timed sequence. The launch vehicle referred to throughout the Re-entry F Familiarization Manual is a modified Scout with Algol IIB first-stage motor, Castor II second-stage motor, and Antares II third-stage motor. Leading particulars of the Scout are summarized in Table 3-7. The spacecraft is mated to the transition "D" section which includes a large-diameter spin bearing. The spacecraft and the Scout are joined by a flange and marmon clamp arrangement. The spinup system (Figure 3-4) includes two spinup motors which provide clockwise spin stabilization to the spacecraft during boosted re-entry prior to separation.

The fiberglass fairing is jettisoned by explosive bolt-marmon clamp action after third stage burnout plus a minimum coast time to permit tailoff. This action unlocks the spin table (Figure 3-5) causing the spin motors fire for 0.6 seconds giving 60 rpm. At this time, the Spacecraft explosive bolt-marmon clamp releases and the third stage retro thrust is applied. Springs and retro provide the separation velocity between the third stage and the spacecraft.

TABLE 3-7. SCOUT LAUNCH VEHICLE, LEADING PARTICULARS

Weight (pounds)	Net	With Spacecraft*
at liftoff (complete vehicle)	38,680	39,280
at spacecraft separation**	775	1,375
Rocket Motor Performance	**Total Impulse (pounds per second)**	**Burn Time (seconds)**
1st stage (Algol IIB)	5,472,350	80.0
2nd stage (Castor II)	2,317,000	39.3
3rd stage (Antares II)	719,540	34.9

* Spacecraft weight, 600 pounds (ballasted).

** First and second stages separated; third stage consumed.

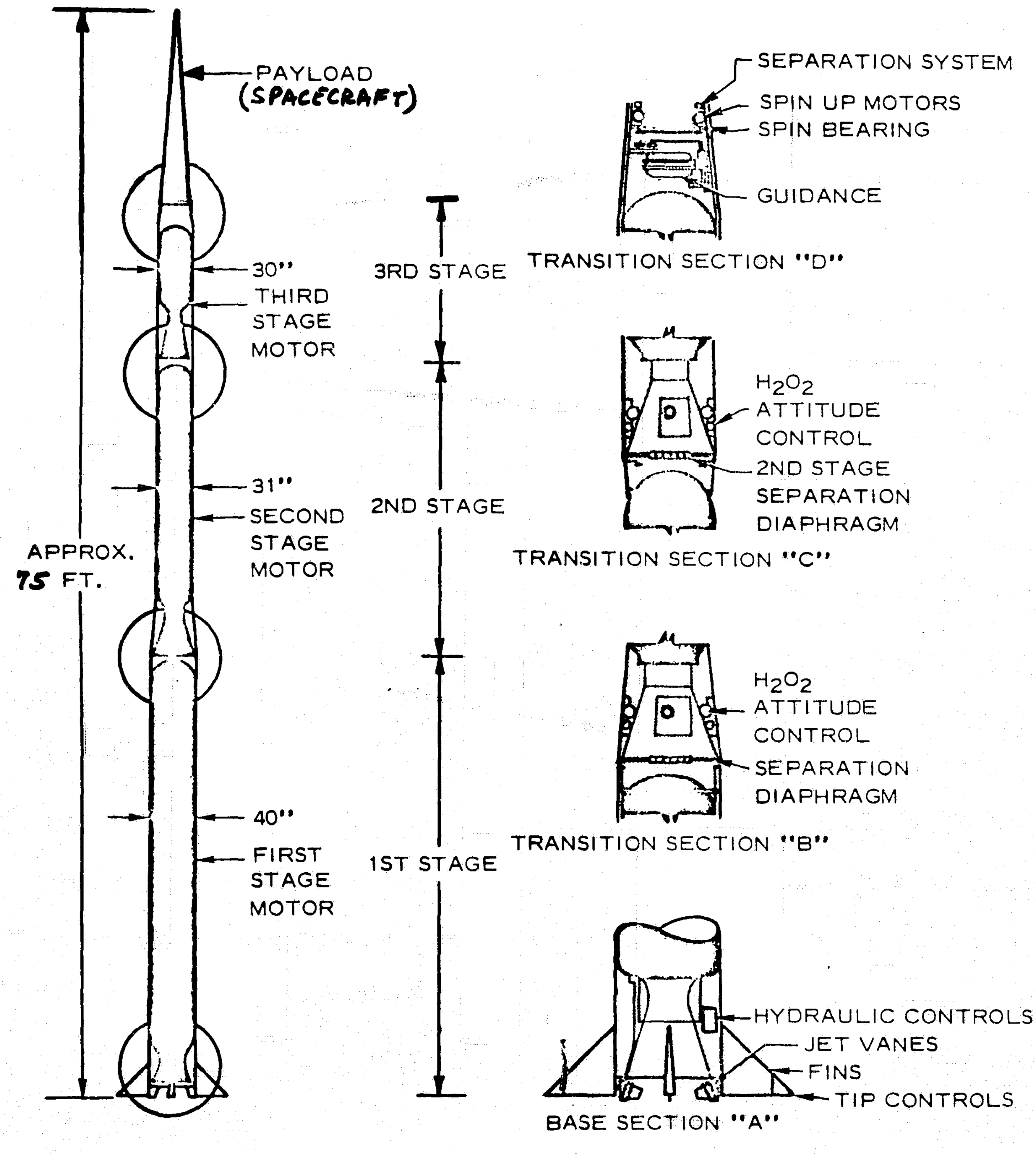


Figure 3-3. Modified Scout Launch Vehicle for Re-entry F

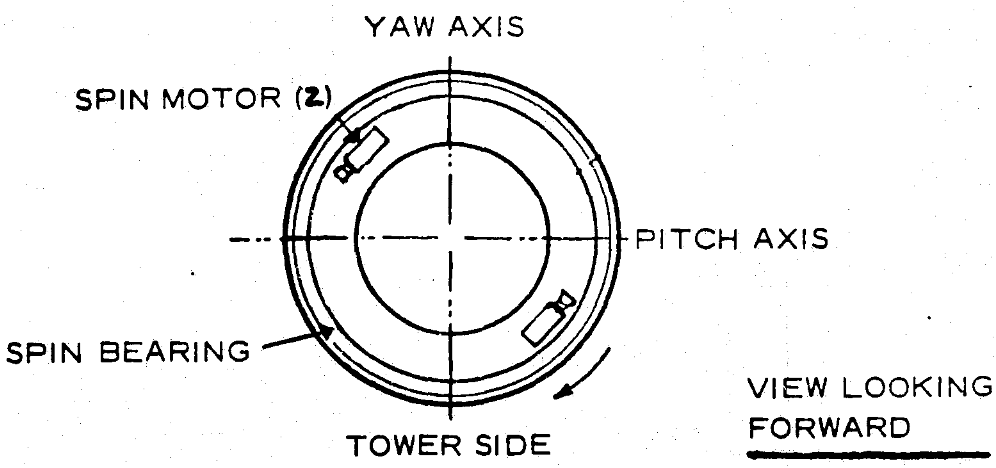


Figure 3-4. Launch Vehicle Spinup System

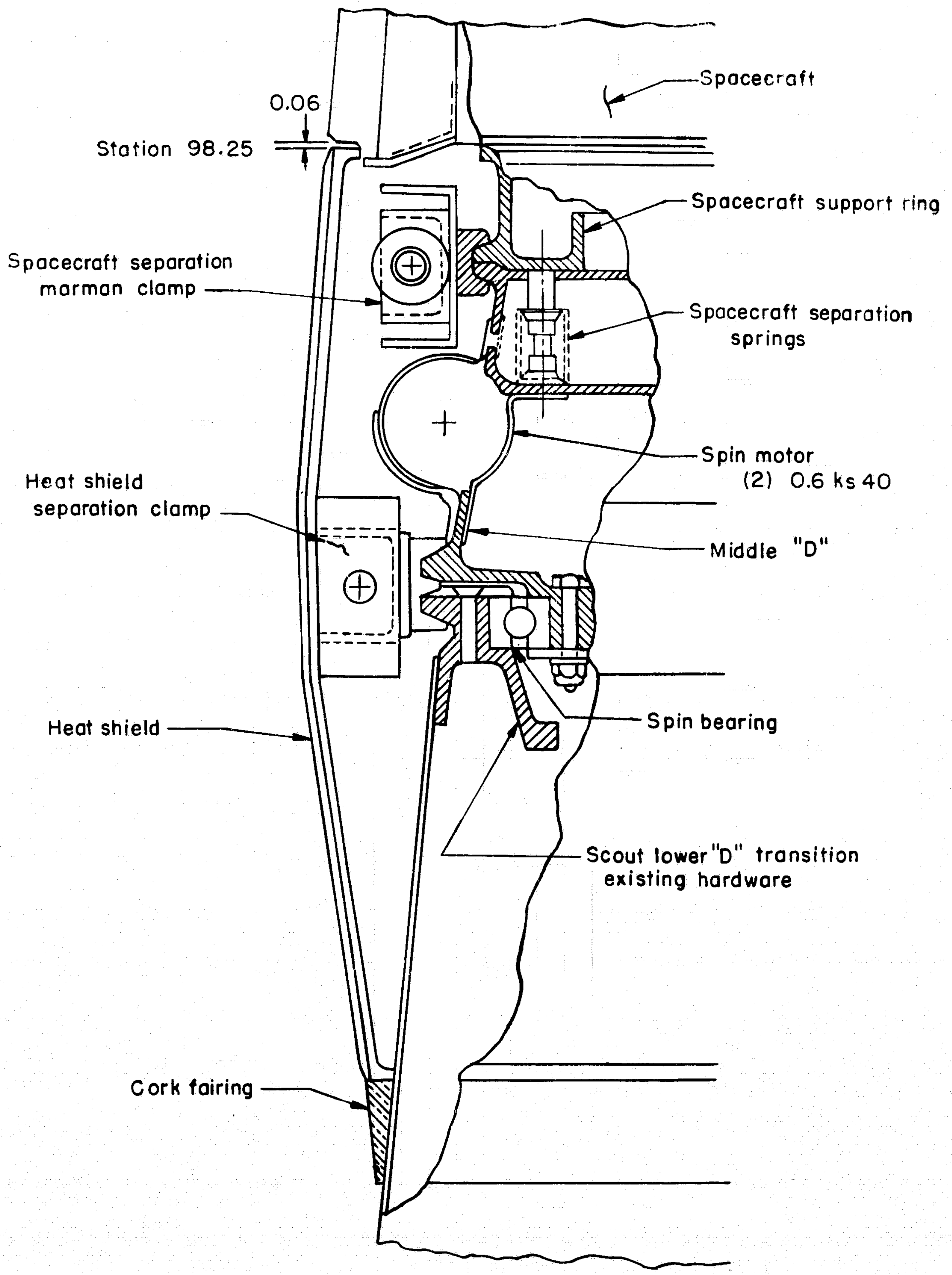


Figure 3-5. Spring Ejection Separation System

3.3 Data Acquisition, Reduction and Analysis

Acquisition, Reduction and Analysis of data are the responsibility of NASA/LRC. Data will be acquired over the entire trajectory with overlapping by the tracking and telemetry facilities of NASA/Wallops and Bermuda Stations. Bermuda will be the primary site for telemetry data acquisition during re-entry, and the site for all re-entry radar tracking. The NASA telemetry ship, Range Recoverer, and two NASA aircraft also will be used for acquiring telemetry data during re-entry. In addition, the two aircraft will acquire optical data (photographic and spectrographic) during re-entry.

The primary data period by definition will begin at an altitude of 120K feet during re-entry and will continue to the altitude of first melt of the beryllium surface which is predicted to occur at 49K feet. Figures 3-6 and 3-7 are simplified block diagrams of data acquisition. All telemetry data will be recorded in real time. Radar tracking will provide continuous real time location and course of the Spacecraft through use of a C band transponder.

A shematic view of the electrical system (figure 3-8) shows there are eight continuous channels and three commutated channels of telemetered data. The arrangement of the Thermal Sensors and the pressure orifices on the Spacecraft is shown in figure 3-9. The spacecraft coordinate System is shown in figure 3-10.

The data will be reduced to heating rates, transition Reynolds numbers, angles of attack, etc. by NASA/LRC and published in the form of a data report. This will provide the data at an early date to those who may wish to make their own analyses. Later NASA/LRC will publish an analysis report in which the data are discussed in relation to theory and other data.

3.4 Spacecraft Design

3.4.1 Spacecraft System Description

The Spacecraft (figure 3-11) is a 5-degree conical body 156 inches (13 feet) long with a 27.3-inch base diameter and a 0.1-inch nose-tip radius. Its weight will be ballasted to 600 pounds. Axial locations are designated by stations, numbered according to the distance in inches from the cone apex, and by degrees of rotation around the cone from a reference line.

The main test body of the spacecraft is the calorimeter, a beryllium shell instrumented with 21 thermal sensors and 13 pressure ports. The calorimeter is the primary load-bearing structure of the spacecraft and consists of seven trepanned frustums of 0.6" thick hot-pressed-block, structural-grade beryllium, mechanically joined to form the outer shell of the spacecraft. This structure forms all of the cone surface except for an ATJ-graphite nose tip and six quartz antenna windows at the aft end of the spacecraft. The base of the cone is closed by a fiberglass aft cover.

The calorimeter encloses a substructure, separated from it by an air gap which provides an adiabatic boundary condition. The substructure is an aluminum ring and skin assembly in the mid section, and is used to support an equipment package, harness, and pressure transducers. The equipment package assembly, designed for rear-access insertion into the substructure, includes most of the components of the telemetry, tracking, and electrical subsystems. Physical and mass properties of the spacecraft are listed in Table 3-8 and electrical characteristics are listed in Table 3-9.

3.4.2 Spacecraft Configuration

The spacecraft consists of the following five major assemblies:

a) Forward section

b) Mid section

c) Aft section

d) Aft cover

e) Equipment package

These assemblies are shown in Figure 3-12, and a drawing tree of all assemblies and sub-assemblies is shown in Figure 3-13.

3.4.2.1 Forward Section Assembly

The forward section assembly extends from the forward tip to station 24.5 and includes a beryllium frustum, and an ATJ Graphite nose tip assembly. The beryllium frustum is fastened to the mid section assembly through a threaded breech joint, and is fastened to the nose tip assembly through a preloaded bolt arrangement. The nose tip assembly is instrumented with three thermocouples.

3.4.2.2 Mid Section Assembly

The mid section assembly extends from station 24.5 to station 93.44 and consists of three mechanically joined beryllium frustums, a ballast assembly, an internal structure (which will contain the ballast and enclose the equipment package), and thermal and pressure sensors with associated wiring. The ballast assembly is designed to permit adjustment of the mass properties. The mid section is bolted to the aft section at station 93.44.

3.4.2.3 Aft Section Assembly

The aft section assembly extends from station 93.44 to the aft end of the spacecraft, and consists of three mechanically joined beryllium frustums, an internal structure, thermal and pressure sensors, and associated harnessing. The aftmost 12 inches constitutes the interface structure, and includes four VHF antenna windows and two C-Band antenna windows.

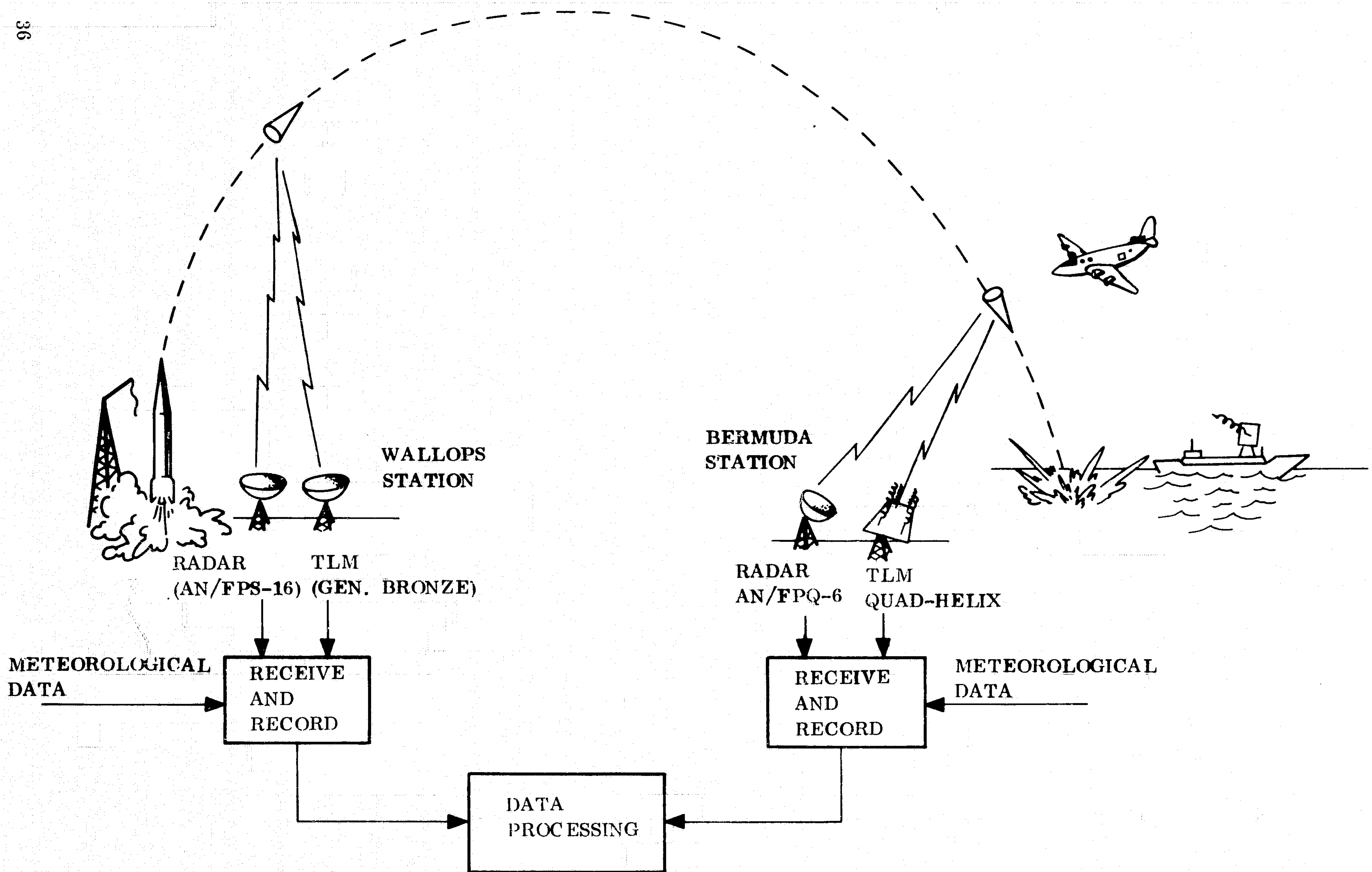


Figure 3-6. Data Acquisition, Block Diagram

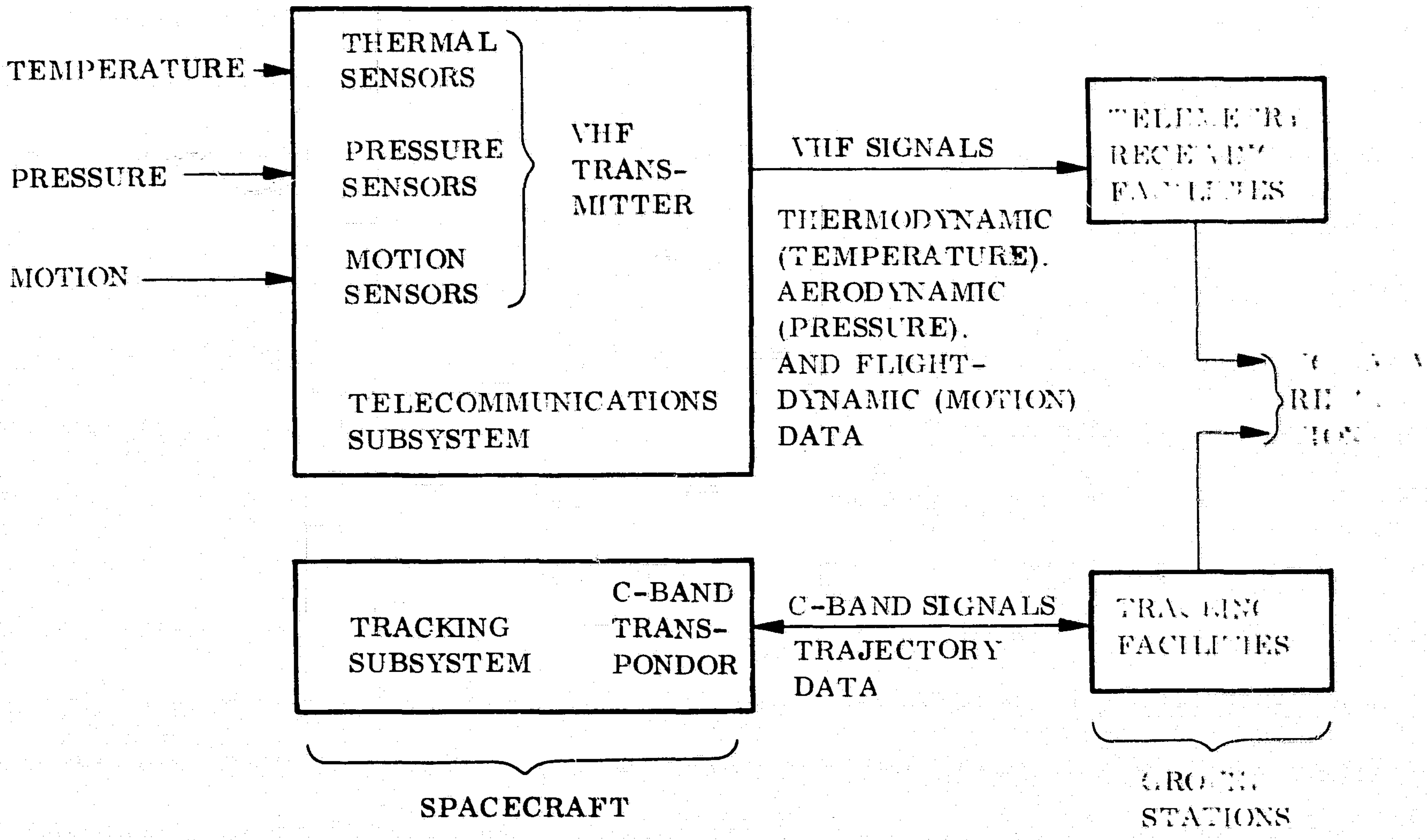


Figure 3-7. Data Acquisition, Simplified Diagram

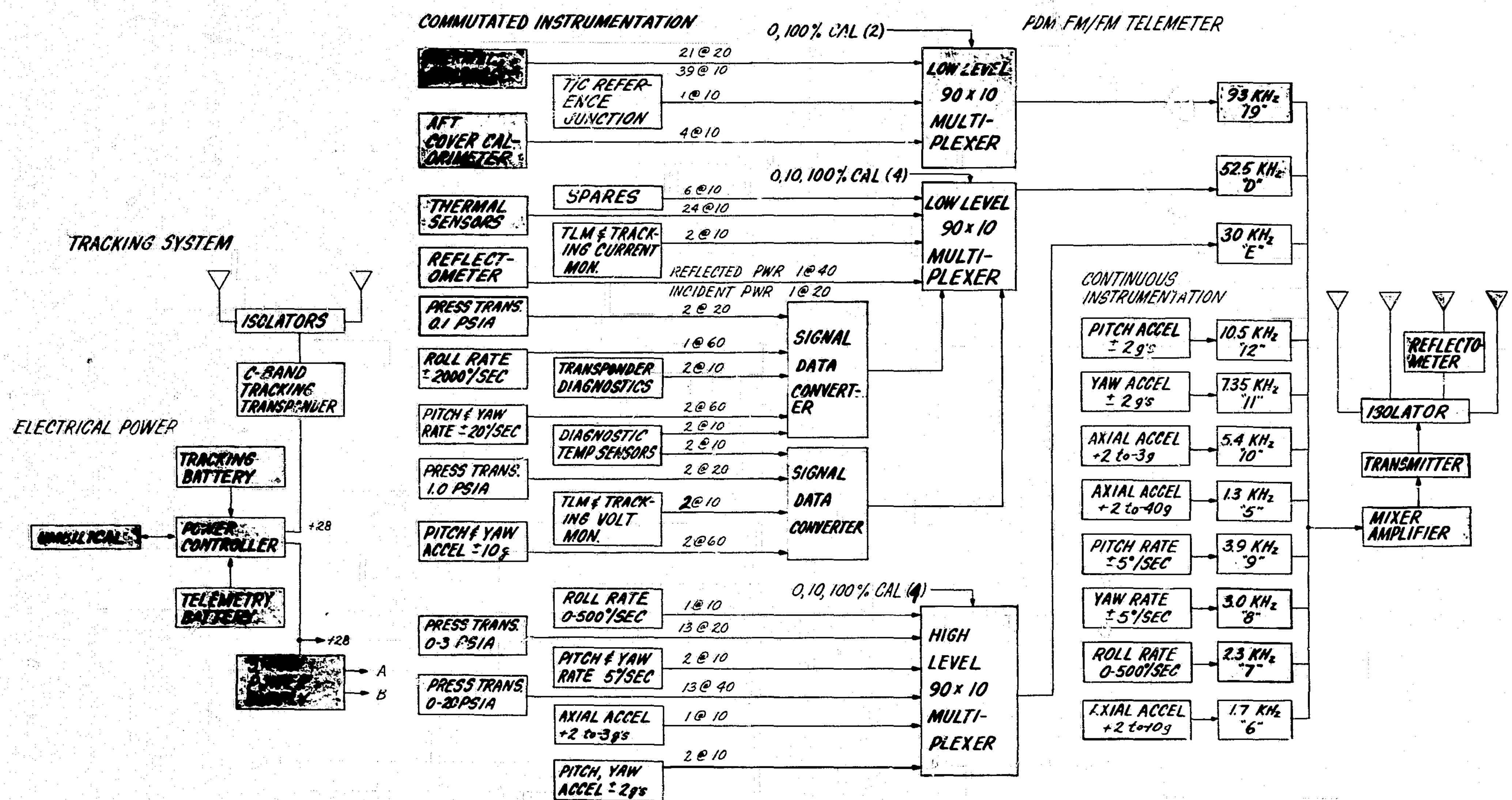


Figure 3-8. Telecommunications and Electrical System

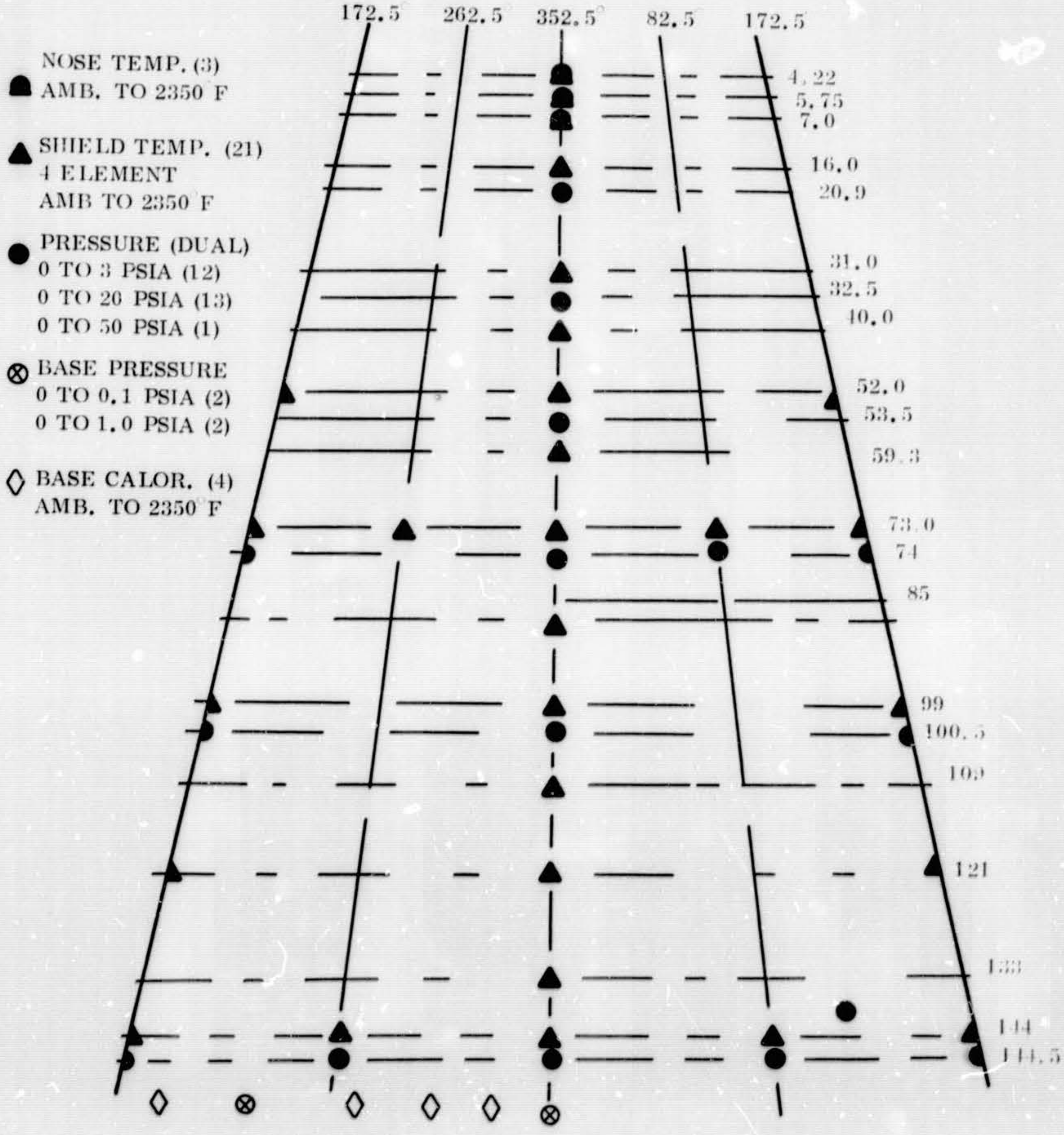
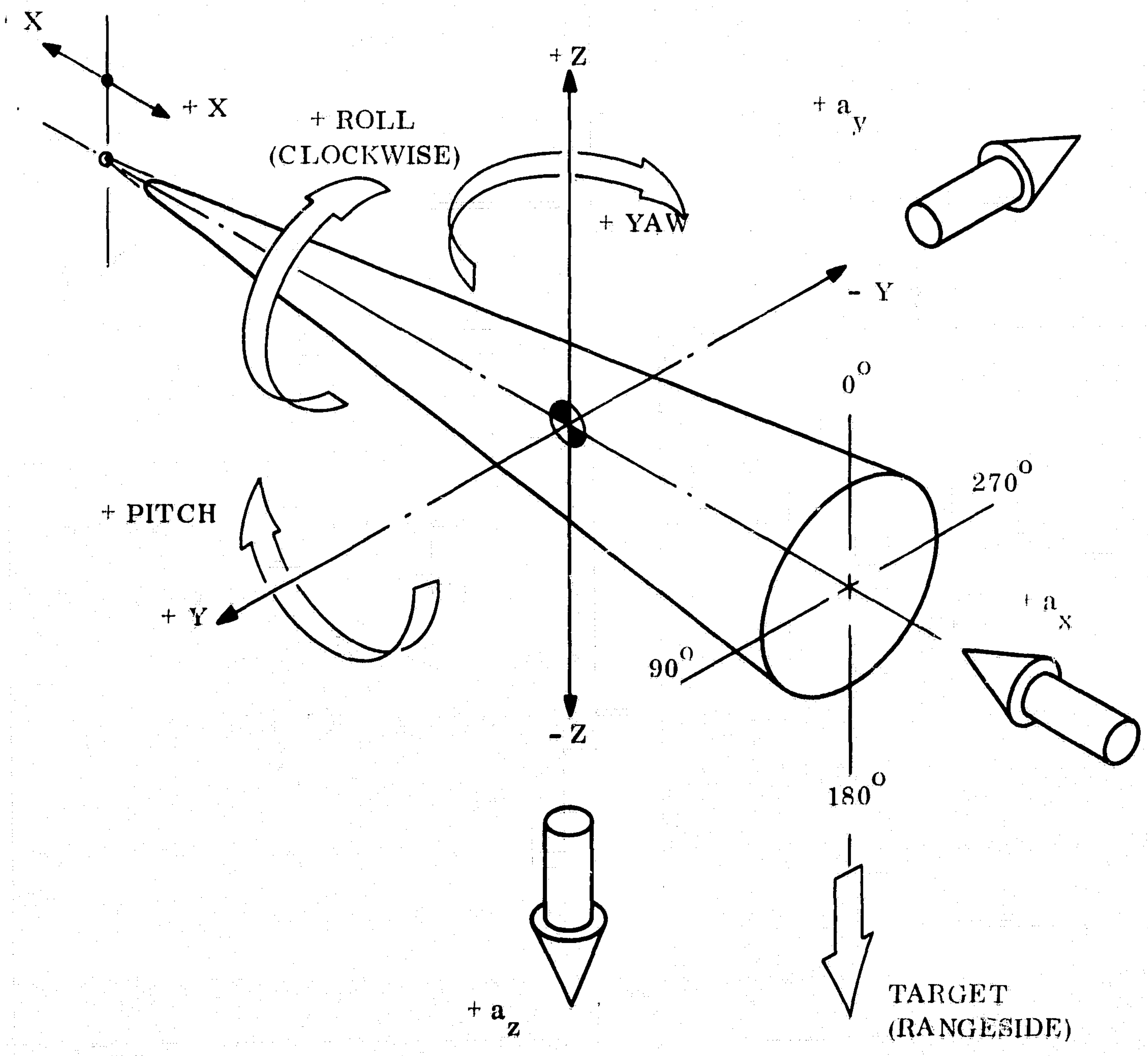


Figure 3-9. Sensor Locations



AXIS		+	ACCELERATION		-
X	LONGITUDINAL	AFT	a_x	AXIAL	FORWARD
Y	PITCH	LEFT	a_y	YAW	RIGHT
Z	YAW	UP	a_z	PITCH	DOWN

Figure 3-10. Spacecraft Motion Coordinates

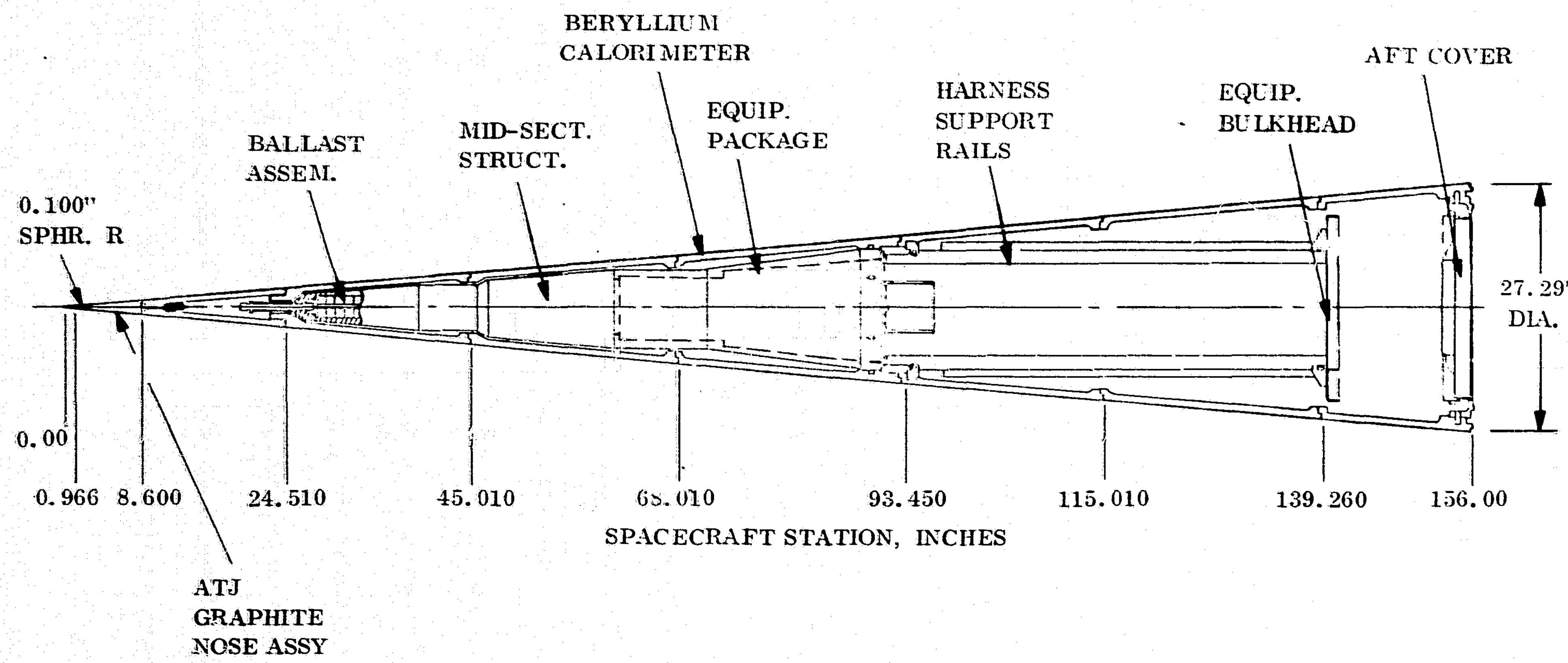


Figure 3-11. General Config of the ReF

TABLE 3-8 SPACECRAFT PROPERTIES

Physical Properties	
Length:	156 inches (max)
Nose Radius:	0.1 inches (nominal)
Cone Half Angle:	5 degrees (nominal)
Base Diameter:	27.3 inches (nominal)
Calorimeter Surface Step Heights: Aft Facing:	.010 ± .009 inches (nominal)
Calorimeter Surface Gap Width:	.002 to .006 inches
Calorimeter Surface Finish:	125 micro inches
cg Location	97.45 station, inches (max)
Angle Between Geometric Axis and Inertial Axis:	Less than 0.1 degree
Weight:	600 pounds (max)
Mass Properties	
c.g. Radial Offset:	0.02 inches (max)
Static Stability Margin:	5 percent (minimum)
Moments of Inertial $I_{pitch} - I_{yaw}$:	± 10 slug ft^2 (max)
Products of Inertia:	± 0.05 slug ft^2*

* The goal shall be to reduce the dynamic unbalance about the spin axis to less than 200 oz. $in.^2$

TABLE 3-9 SPACECRAFT ELECTRICAL CHARACTERISTICS*

Power	
External	
Source	Electrical A.G.E. via umbilical
Requirements	
Voltage	24vdc to 33vdc, adjustable
Current	Telemetry, 10 amp; tracking, 5 amp
Source impedance	Less than 0.5 ohm
Ripple	Less than 1.0v (peak to peak)
Internal	
Source	Two 29vdc batteries
Requirements	
Voltage	29 ± 2vdc
Current	Telemetry, 6 amp; tracking, 1.5 amp
Ripple	Less than 1.0v (peak-to-peak)
Source impedance	Less than 0.5 ohm
Signals	
Telemetry	
Modulation	PDM/FM/FM
Carrier frequency	253.8MHz
Output Rf power	10 watts
Tracking	
(Tracking subsystem; C-Band transponder)	
Receiver frequency.	5625MHz
Output characteristics	
Frequency.	5700MHz
Peak Power.	500 watts
Pulse width.	0.5 micro sec.

* Refer to the following GE/RSD specifications for detailed electrical requirements: S0010-08-0008-A (Electrical System Requirement Specification) and S0150-00-0016-B (Instrumentation and Communications).

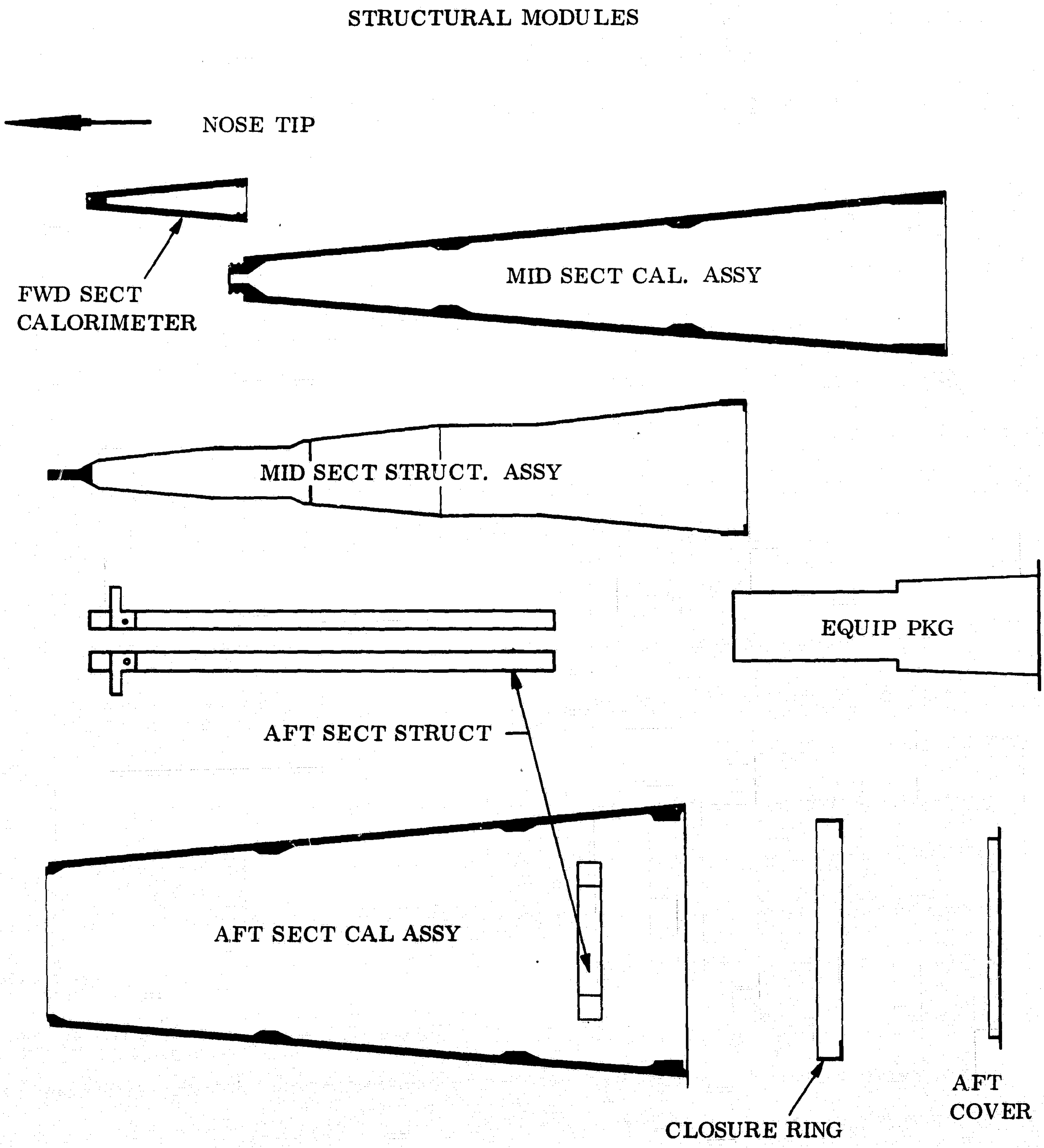
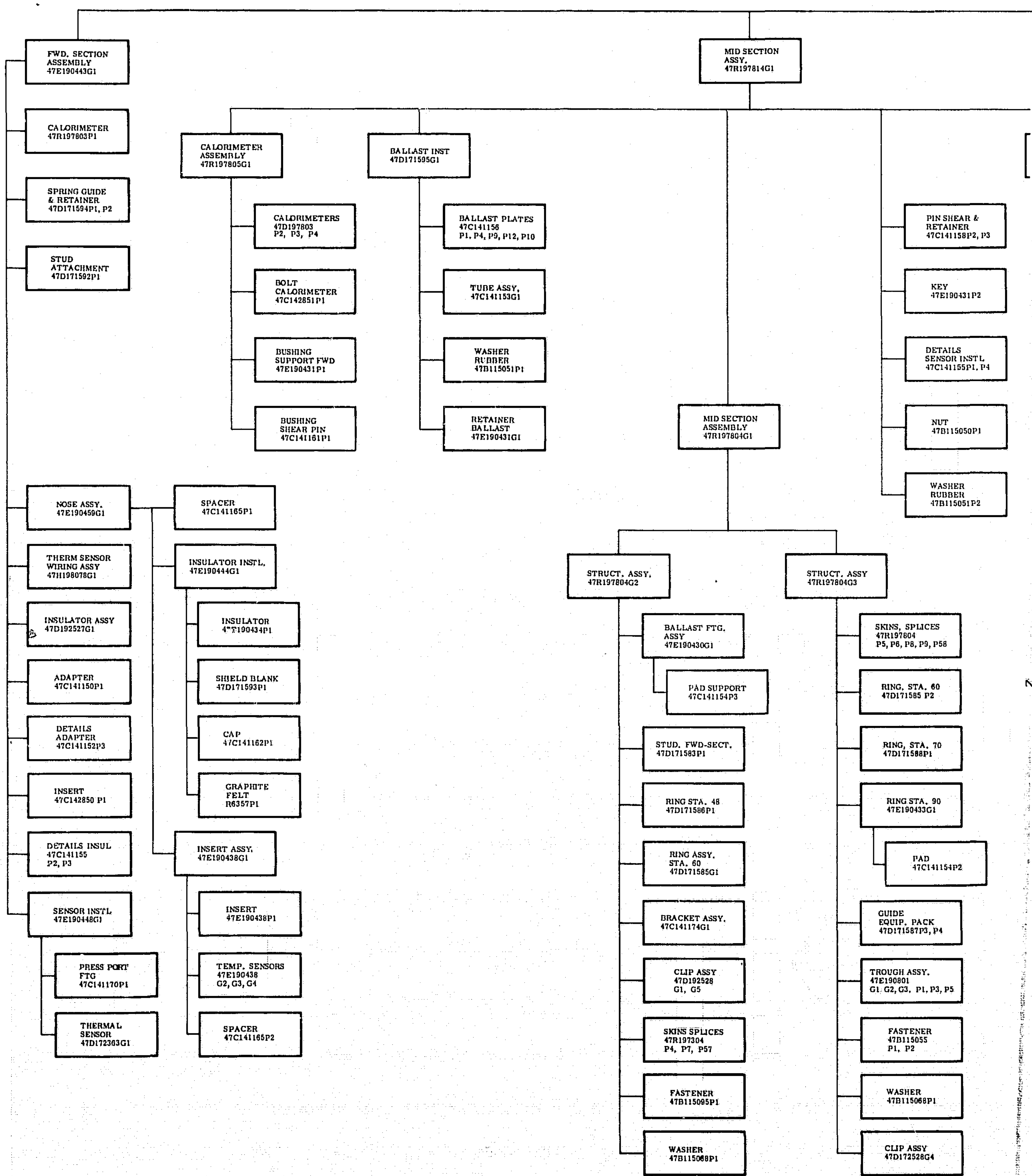
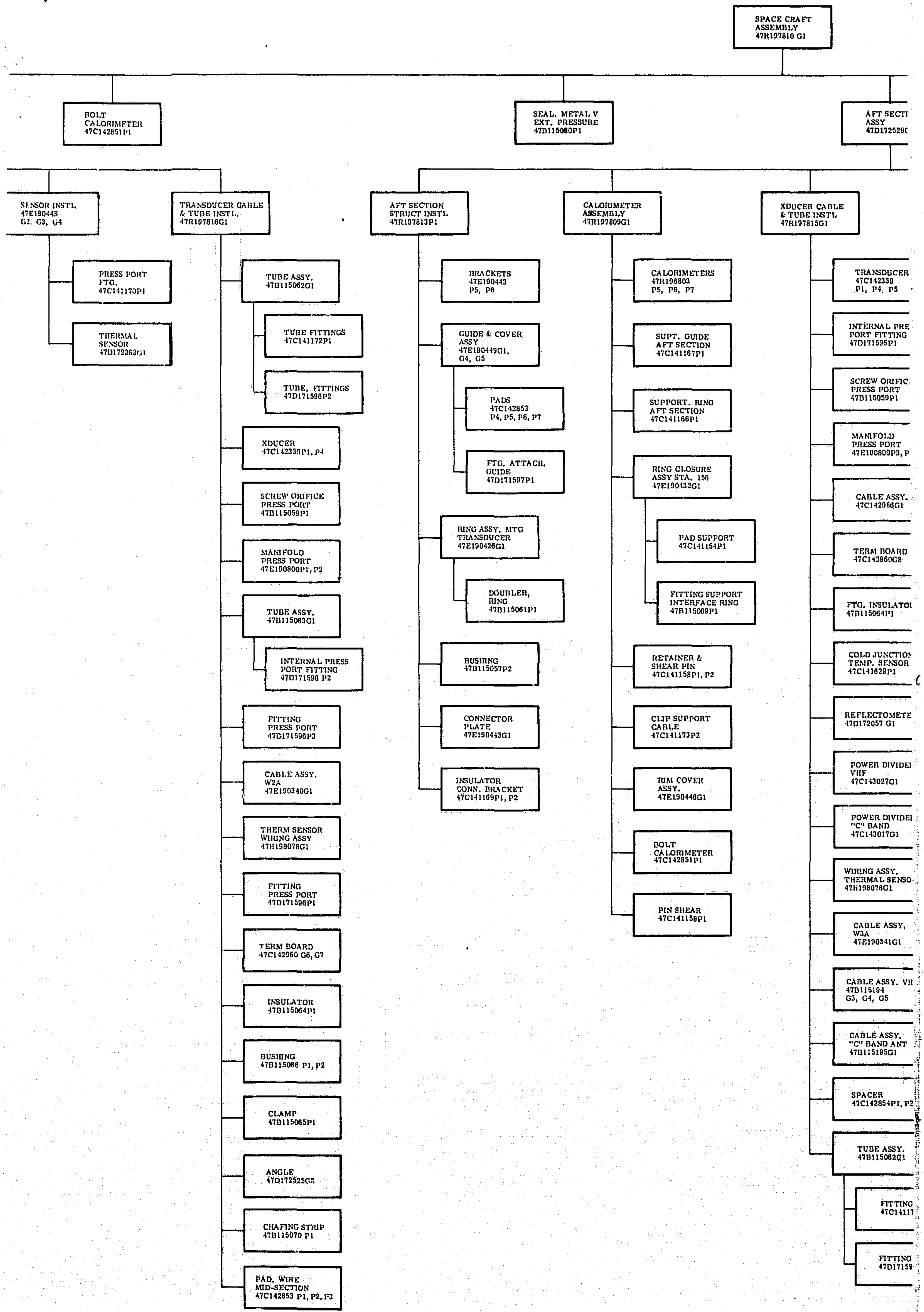


Figure 3-12. Re-entry F Spacecraft Assembly


FWD. SECTION ASSEMBLY 47E190443G1
CALORIMETER 47R197803P1
SPRING GUIDE & RETAINER 47D171594P1, P2
STUD ATTACHMENT 47D171592P1
NOSE ASSY. 47E190459G1
THERM SENSOR WIRING ASSY 47H198078G1
INSULATOR ASSY 47D192527G1
ADAPTER 47C141150P1
DETAILS ADAPTER 47C141152P3
INSERT 47C142850 P1
DETAILS INSUL 47C141155 P2, P3
SENSOR INSTL 47E190448G1
PRESS PORT FTG 47C141170P1
THERMAL SENSOR 47D172303G1
SPACER 47C141165P1
INSULATOR INSTL. 47E190444G1
INSULATOR 47E190434P1
SHIELD BLANK 47D171593P1
CAP 47C141162P1
GRAPHITE FELT R6357P1
INSERT ASSY. 47E190438G1
INSERT 47E190438P1
TEMP. SENSORS 47E190438 G2, G3, G4
SPACER 47C141165P2
CALORIMETER ASSEMBLY 47R197805G1
CALORIMETERS 47D197803 P2, P3, P4
BOLT CALORIMETER 47C142851P1
BUSHING SUPPORT FWD 47E190431P1
BUSHING SHEAR PIN 47C141161P1
BALLAST INST 47D171595G1
BALLAST PLATES 47C141156 P1, P4, P9, P12, P10
TUBE ASSY. 47C141153G1
WASHER RUBBER 47B115051P1
RETAINER BALLAST 47E190431G1
MID SECTION ASSY. 47R197814G1
MID SECTION ASSEMBLY 47R197804G1
PIN SHEAR & RETAINER 47C141158P2, P3
KEY 47E190431P2
DETAILS SENSOR INSTL 47C141155P1, P4
NUT 47B115050P1
WASHER RUBBER 47B115051P2
STRUCT. ASSY. 47R197804G2
BALLAST FTG. ASSY 47E190430G1
PAD SUPPORT 47C141154P3
STUD. FWD-SECT. 47D171583P1
RING STA. 48 47D171586P1
RING ASSY. STA. 60 47D171585G1
BRACKET ASSY. 47C141174G1
CLIP ASSY 47D192528 G1, G5
SKINS SPLICES 47R197304 P4, P7, P57
FASTENER 47B115095P1
WASHER 47B115068P1
STRUCT. ASSY 47R197804G3
SKINS, SPLICES 47R197804 P5, P6, P8, P9, P58
RING, STA. 60 47D171585 P2
RING, STA. 70 47D171588P1
RING STA. 90 47E190433G1
PAD 47C141154P2
GUIDE EQUIP. PACK 47D171587P3, P4
TROUGH ASSY. 47E190801 G1, G2, G3, P1, P3, P5
FASTENER 47B115055 P1, P2
WASHER 47B115068P1
CLIP ASSY 47D172528G4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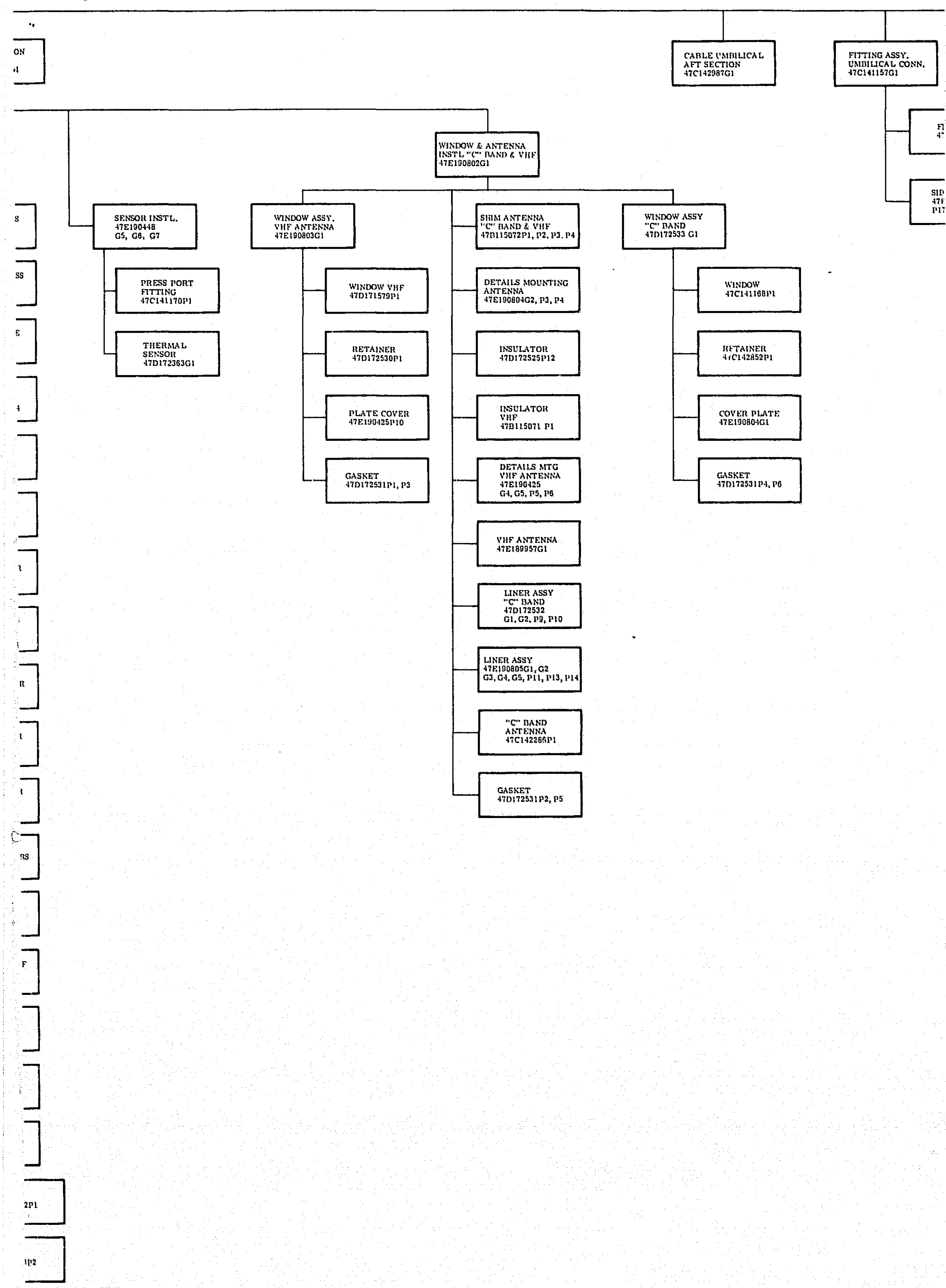
BOLT CALORIMETER 47C142851P1
SEAL, METAL V EXT. PRESSURE 47B115060P1
AFT SECTI ASSY 47D172529C
SENSOR INSTL 47E190443 G2, G3, G4
PRESS PORT FTG. 47C141170P1
THERMAL SENSOR 47D172363G1
TRANSDUCER CABLE & TUBE INSTL. 47R197816G1
TUBE ASSY. 47B115062G1
TUBE FITTINGS 47C141172P1
TUBE, FITTINGS 47D171596P2
XDUCER 47C142339P1, P4
SCREW ORIFICE PRESS PORT 47B115059P1
MANIFOLD PRESS PORT 47E190800P1, P2
TUBE ASSY. 47B115063G1
INTERNAL PRESS PORT FITTING 47D171596 P2
FITTING PRESS PORT 47D171596P3
CABLE ASSY. W2A 47E190340G1
THERM SENSOR WIRING ASSY 47R198078G1
FITTING PRESS PORT 47D171596P1
TERM BOARD 47C142960 G6, G7
INSULATOR 47B115064P1
BUSHING 47B115066 P1, P2
CLAMP 47B115065P1
ANGLE 47D172525G3
CHAFING STRIP 47B115070 P1
PAD, WIRE MID-SECTION 47C142853 P1, P2, P3
AFT SECTION STRUCT INSTL 47R197813P1
BRACKETS 47E190443 P5, P6
GUIDE & COVER ASSY 47E190449G1, G4, G5
PADS 47C142853 P4, P5, P6, P7
FTG. ATTACH. GUIDE 47D171597P1
RING ASSY. MTG TRANSDUCER 47E190426G1
DOUBLER, RING 47B115061P1
BUSHING 47B115057P2
CONNECTOR PLATE 47E190443G1
INSULATOR CONN. BRACKET 47C141169P1, P2
CALORIMETER ASSEMBLY 47R197809G1
CALORIMETERS 47R198803 P5, P6, P7
SUPT. GUIDE AFT SECTION 47C141167P1
SUPPORT, RING AFT SECTION 47C141166P1
RING CLOSURE ASSY STA. 156 47E190432G1
PAD SUPPORT 47C141154P1
FITTING SUPPORT INTERFACE RING 47B115069P1
RETAINER & SHEAR PIN 47C141156P1, P2
CLIP SUPPORT CABLE 47C141173P2
RIM COVER ASSY. 47E190446G1
BOLT CALORIMETER 47C142851P1
PIN SHEAR 47C141158P1
XDUCER CABLE & TUBE INSTL 47R197815G1
TRANSDUCER 47C142339 P1, P4, P5
INTERNAL PRE PORT FITTING 47D171596P1
SCREW ORIFIC PRESS PORT 47B115059P1
MANIFOLD PRESS PORT 47E190800P3, P
CABLE ASSY. 47C142986G1
TERM BOARD 47C142960G8
FTG. INSULATO 47B115064P1
COLD JUNCTION TEMP. SENSOR 47C141629P1
REFLECTOMETE 47D172057 G1
POWER DIVIDE VHF 47C143027G1
POWER DIVIDE "C" BAND 47C143017G1
WIRING ASSY. THERMAL SENSO 47R198078G1
CABLE ASSY. W3A 47E190341G1
CABLE ASSY. VH 47B115194 G3, G4, G5
CABLE ASSY. "C" BAND ANT 47B115195G1
SPACER 47C142854P1, P2
TUBE ASSY. 47B115062G1
FITTING 47C14117
FITTING 47D17159


CABLE UMBILICAL AFT SECTION 47C142987G1
FITTING ASSY. UMBILICAL CONN. 47C141157G1
WINDOW & ANTENNA INSTL "C" BAND & VHF 47E190802G1
SENSOR INSTL. 47E190448 G5, G6, G7
PRESS PORT FITTING 47C141170P1
THERMAL SENSOR 47D172363G1
WINDOW ASSY. VHF ANTENNA 47E190803G1
WINDOW VHF 47D171579P1
RETAINER 47D172530P1
PLATE COVER 47E190425P10
GASKET 47D172531P1, P3
SHIM ANTENNA "C" BAND & VHF 47B115072P1, P2, P3, P4
DETAILS MOUNTING ANTENNA 47E190804G2, P3, P4
INSULATOR 47D172525P12
INSULATOR VHF 47B115071 P1
DETAILS MTG VHF ANTENNA 47E190425 G4, G5, P5, P6
VHF ANTENNA 47E189957G1
LINER ASSY "C" BAND 47D172532 G1, G2, P9, P10
LINER ASSY 47E190805G1, G2 G3, G4, G5, P11, P13, P14
"C" BAND ANTENNA 47C142286P1
GASKET 47D172531P2, P5
WINDOW ASSY "C" BAND 47D172533 G1
WINDOW 47C141168P1
RETAINER 47C142852P1
COVER PLATE 47E190804G1
GASKET 47D172531P4, P6

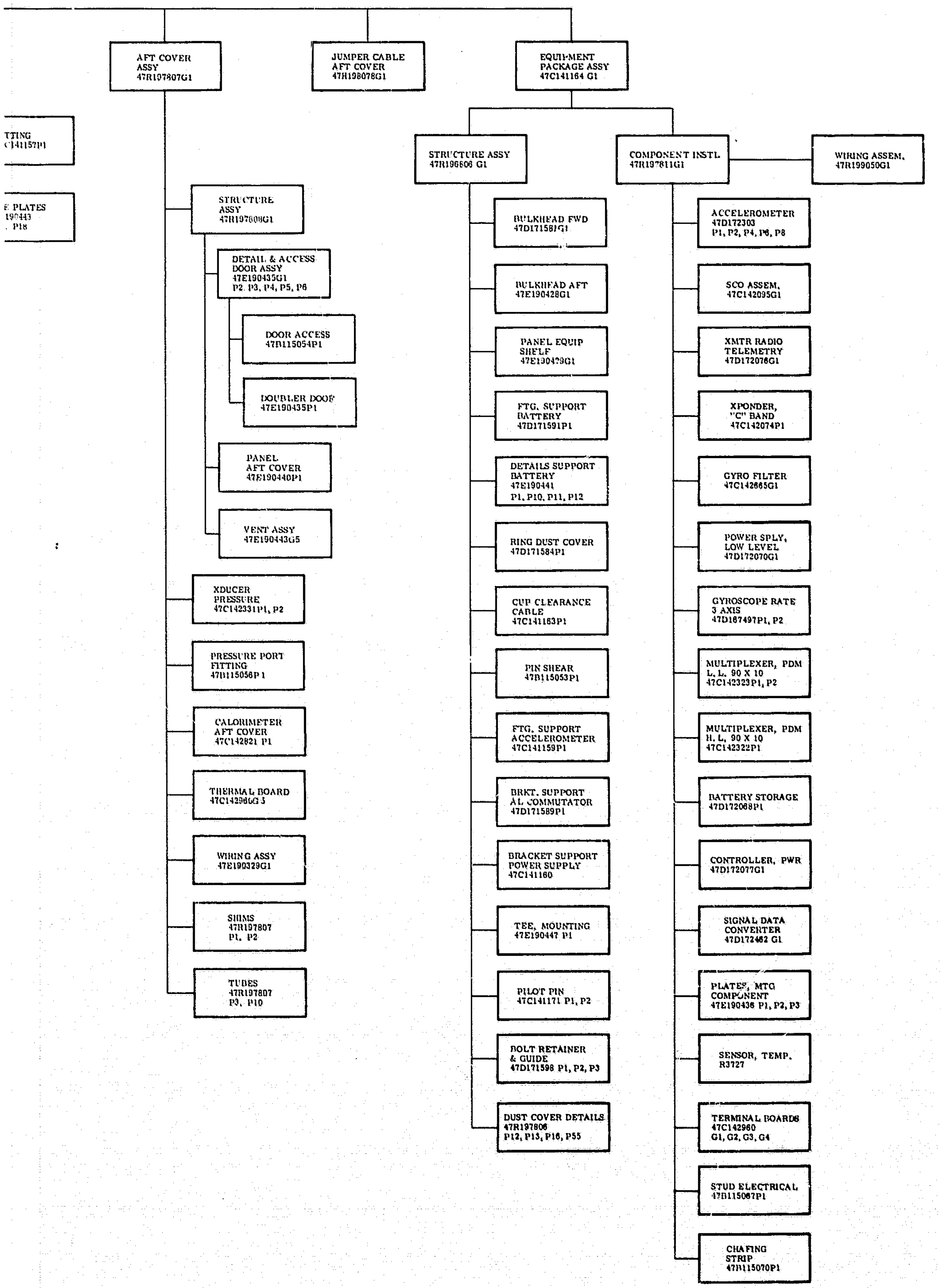


Figure 3-13. Drawing Tree for Re-entry F

The aft end of this assembly is terminated with a closure ring which mates with the booster interface ring.

3.4.2.4 Aft Cover Assembly

The aft cover assembly is a glass-phenolic bulkhead with two aluminum stiffening members. The stiffening members mount four base pressure transducers and four calorimeter-type temperature sensors.

3.4.2.5 Equipment Package Assembly

The equipment package assembly contains the electronic equipment for the telemetry, tracking, and electrical subsystems. The batteries are mounted at the aft end of this assembly and are readily accessible. The spacecraft motion sensors are also located in this assembly. Since the equipment is assembled together in a single package, electrical interfaces are minimal. These are limited to the VHF and C-Band Coax RF Connectors, the calorimeter thermal and pressure sensor harness connectors, and the wiring to the umbilical connector. The complete package is contained within a removable cylindrical shell which provides maximum handling protection.

3.4.3 Calorimeter

The calorimeter consists of seven conical 0.6 inch thick beryllium frustums, 6 of which are together. The seventh is connected by a breech joint. The material is beryllium (2% BeO), and the normal thickness of the shell is 0.600 inch. Figure 3-14 shows the detail of a bolted joint typical of 5 of the 6 joints. The number of bolts and locations are given below. The sixth joint is a breech joint at station 24.5 where frustums P_1 and P_2 (the 2 most forward beryllium shells) are joined together. This breech joint provides for ease in assembling the nose tip, which is a sensitive section of the spacecraft.

Station	Number of Bolts, N	Bolt Circle Radius, r (inches)	Bolt Spacing (inches)
45.00	12	3.02	1.56
68.00	16	5.03	1.96
93.44	24	7.27	1.90
115.00	30	9.14	1.91
139.25	36	11.26	1.96

The calorimeter serves as the primary load bearing structure of the S/C, and consists of three major sub-assemblies: forward section, mid-section, and aft-section. The forward section contains only one beryllium conical shell P_1, while the mid-section and aft-sections contain three beryllium shells each. The antenna windows for both the VHF and C-band antennae are in the aft-most beryllium shell. The windows are a fuzed silica, Corning type 7941M (quartz). Two thicknesses of window are used, 1.19 inch for the VHF and 1.10 inch for the C-Band window.

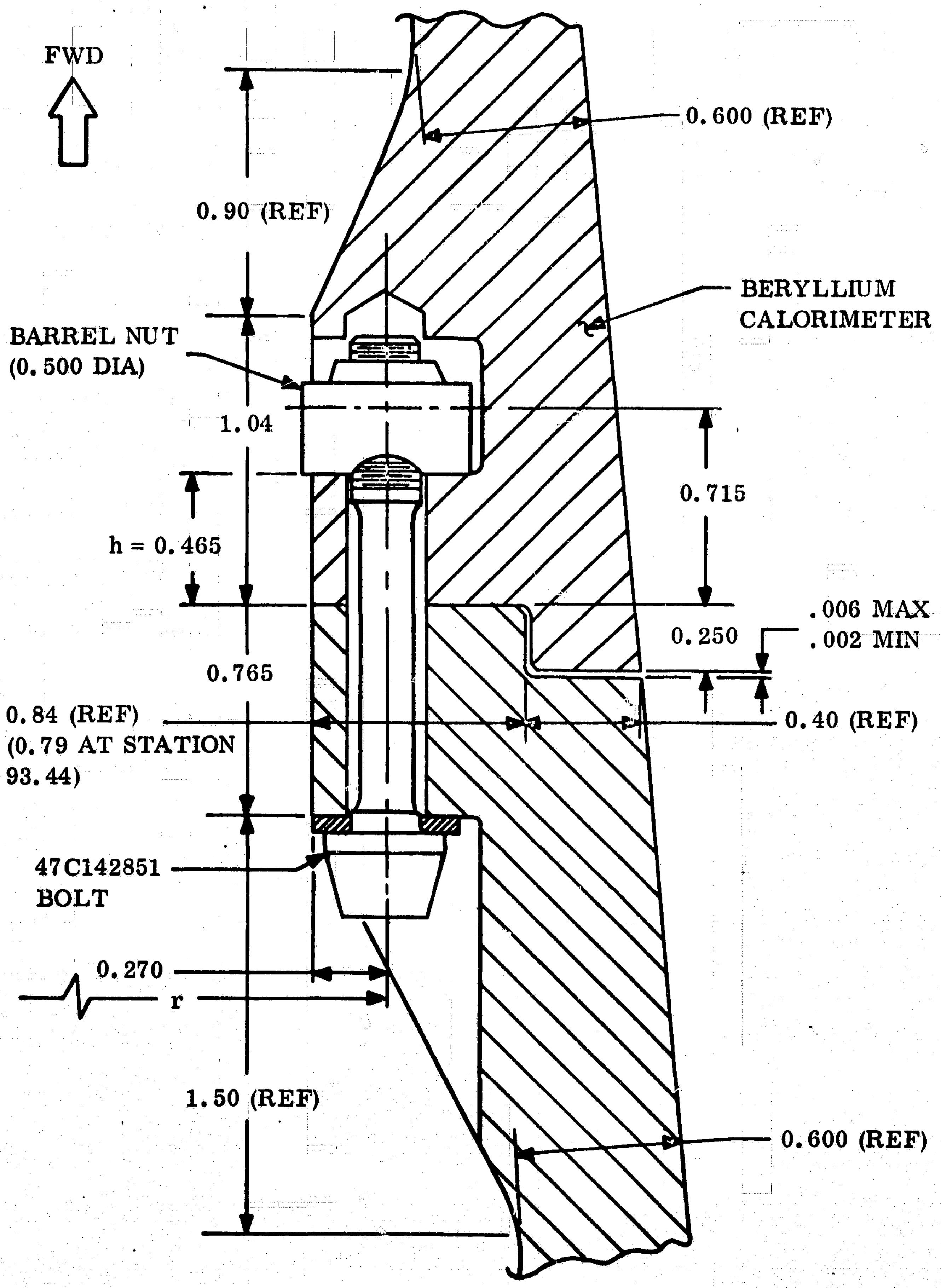


Figure 3-14. Calorimeter Bolted Joint

For the VHF antenna window, four meridianal slots are cut in the beryllium calorimeter measuring 10.5 inches by 2.44 inches each, located at 37.5°, 127.5°, 217.5° and 307.5° between stations 144.0 and 154.529. The windows are secured to the internal surface of the calorimeter by an aluminum framework which is bolted to the calorimeter. Liners of pyrolytic graphite are provided between the quartz and beryllium as thermal transition material. The VHF window frame design is such that large strains are not induced into the framework due to longitudinal and hoop expansion of the calorimeter. The longitudinal strains are relieved by providing pinned expansion joints integral with the angles while the hoop strains are relieved by slotting the angles where the antenna box attaches. The window frame is completely isolated from the calorimeter by a thin glass silicone insulation layer.

The two C-band antenna windows cut-outs are located in the beryllium calorimeter at 82.5° and 262.5° at Station 150.5. The window is held in place by an aluminum framework which is fastened to the calorimeter. Pins and slotted or oversize screw holes have been employed to allow the framework to expand in the meridianal and hoop directions during re-entry. The C-band windows have pyrolytic graphite frames and are 2.83 inches long, 1.64 inches wide, and 1.10 inches thick.

The calorimeter is closed at the rear by an aft cover. The aft cover is a glass-phenolic, aluminum beam reinforced plate and is attached to the closure ring. The cover has a radius of 10.9 inches and a minimum thickness of 0.15 inches.

3.4.4 Internal and Interface Structure

3.4.4.1 Internal Structure

The internal structure in the spacecraft is composed primarily of a mid-section and an aft-section substructure.

As shown in figure 3-15 the forward substructure is a ring-stiffened shell type assembly. The shell consists of three segments all of which are made of 0.063 inch thick 2024-T3 aluminum alloy.

The forward shell segment is conical in shape and extends from station 48.67 to 60.34. The shell has radii of 3.55 and 4.55 inches at its forward and aft ends, respectively.

The mid shell segment is cylindrical in shape and extends from station 61.3 to 69.78 and has a radius of 4.62 inches.

The aft shell segment is conical in shape and extends from station 70.67 to 87.72. The shell has radii of 4.62 and 6.54 inches at its forward and aft ends respectively.

A ballast fitting is attached to the forward end of the forward substructure. This ballast fitting is made of 4130 alloy steel and extends from spacecraft stations 26.281 to 48.03. Variable ballast is held in the steel ballast fitting by a steel screw type retainer located at the aft end. The plug has a maximum diameter of 3.625 inches.

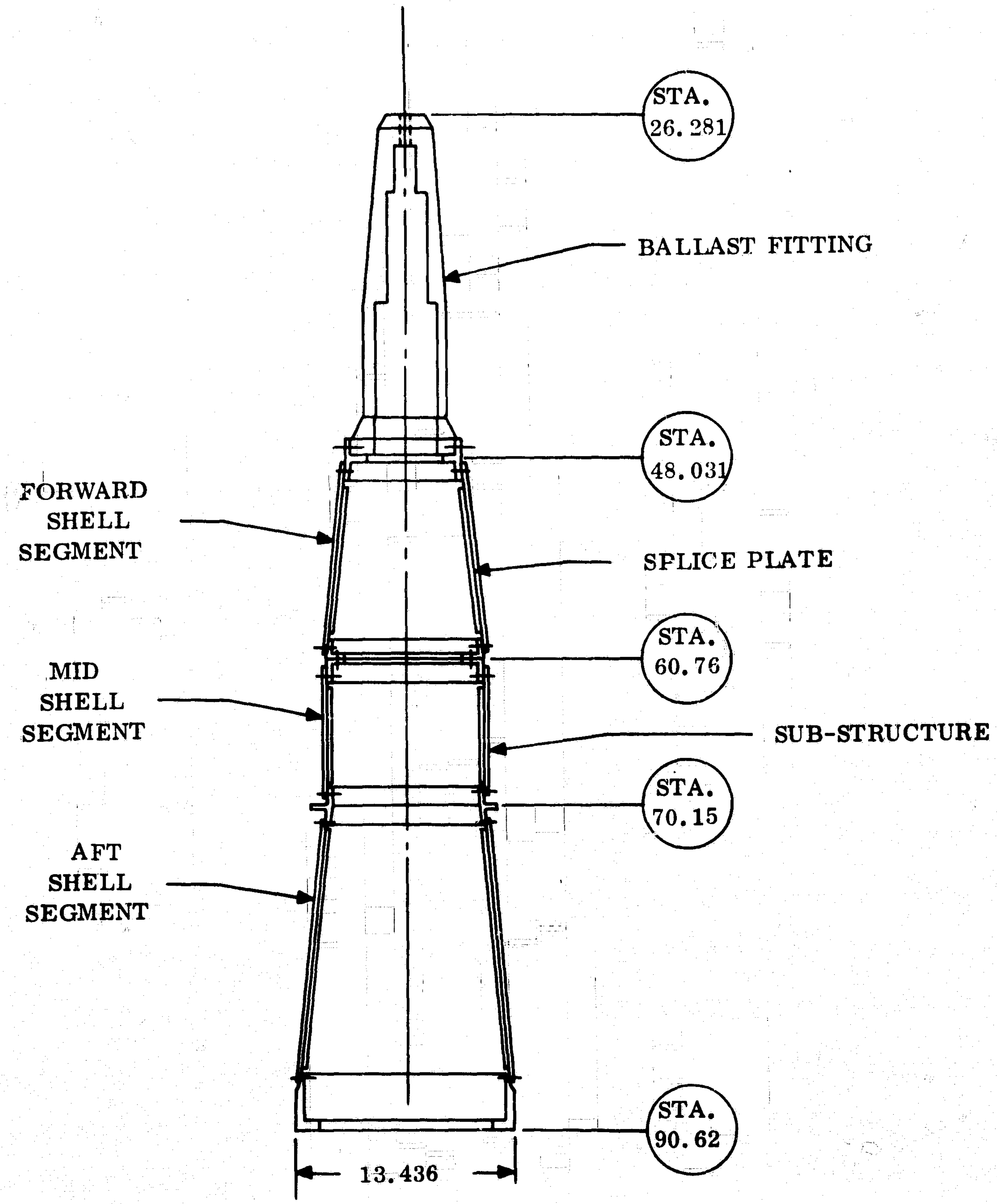


Figure 3-15. Mid-section Substructure and Ballast Fitting

The forward substructure is utilized to mount the equipment package, (see paragraph 3.4.7) and serves as a mounting frame for the electrical and other components.

The aft substructure is located approximately between stations 94 and 139 as shown in Figure 3-16.

The "rail" portion of the structure is made of two 0.050 inch thick 2024-T3 aluminum channel sections riveted together by their flanges. The forward support at station 94.65 reacts with both axial and lateral loads while the aft support at station 139 reacts only with lateral loads since it is slotted to allow for thermal expansion of the calorimeter.

The channel rails house wire bundles which weigh approximately 0.40 pounds per axial inch. After the wires are in place a non-structural type cover is placed over the open side of the rails to retain the wire.

The aft end of the rails tie to a ring bulkhead at 0° and 180°. This ring is attached to the calorimeter at 6 locations, 0°, 60°, 120°, 180°, 240°, and 300°, by brackets fastened to the web of the channel shaped bulkhead. The web portion of the channel supports 6 lbs. of wire harness and 6 lbs. of equipment. A support bracket is used to support the forward end of the rail.

3.4.4.2 Interface Structures

The calorimeter is attached to the aft closure ring by twenty 0.500 diameter steel shear pins at station 154.5 as shown in Figure 3-17. The aft closure ring is attached to the spacecraft interface ring by 24 tension bolts (NAS 1304). The interface ring is designed by Ling-Temco-Vought Corp. The mechanical interface between the Re-entry-F Spacecraft and the Scout Launch Vehicle is shown in Figure 3-18. In addition, the spacecraft umbilical connection at the Re-entry-F/Scout Interface is shown in Figure 3-19.

3.4.5 Nose Tip

The nose tip assembly, (Drawing No. 47E190439) extends from stations 0.966 to 8.600 and consists of an ATJ graphite conical shell, a porous carbon insulator inner shell, and a tungsten alloy insert. The bond between the ATJ graphite skirt and the porous carbon is C-10 and the bond between the porous carbon and the tungsten alloy insert is PD162A.

A graphite plug is placed between the tungsten alloy insert and the ATJ nose tip. It has the configuration shown in Figure 3-20. The plug is composed of 0.480 inch of ATJ and 0.173 inch of PG graphite. On its forward end, the plug is surrounded by graphite felt, originally 0.060 ± 0.020 thick but compressed to 0.20 inch during assembly. To preload the nose tip assembly to the beryllium calorimeter, a belleville spring arrangement is used. Twenty springs are compressed an amount 0.100 ± 0.030 to create a 0.890 inch spring height and a resultant preload of 470 ± 175 pounds.

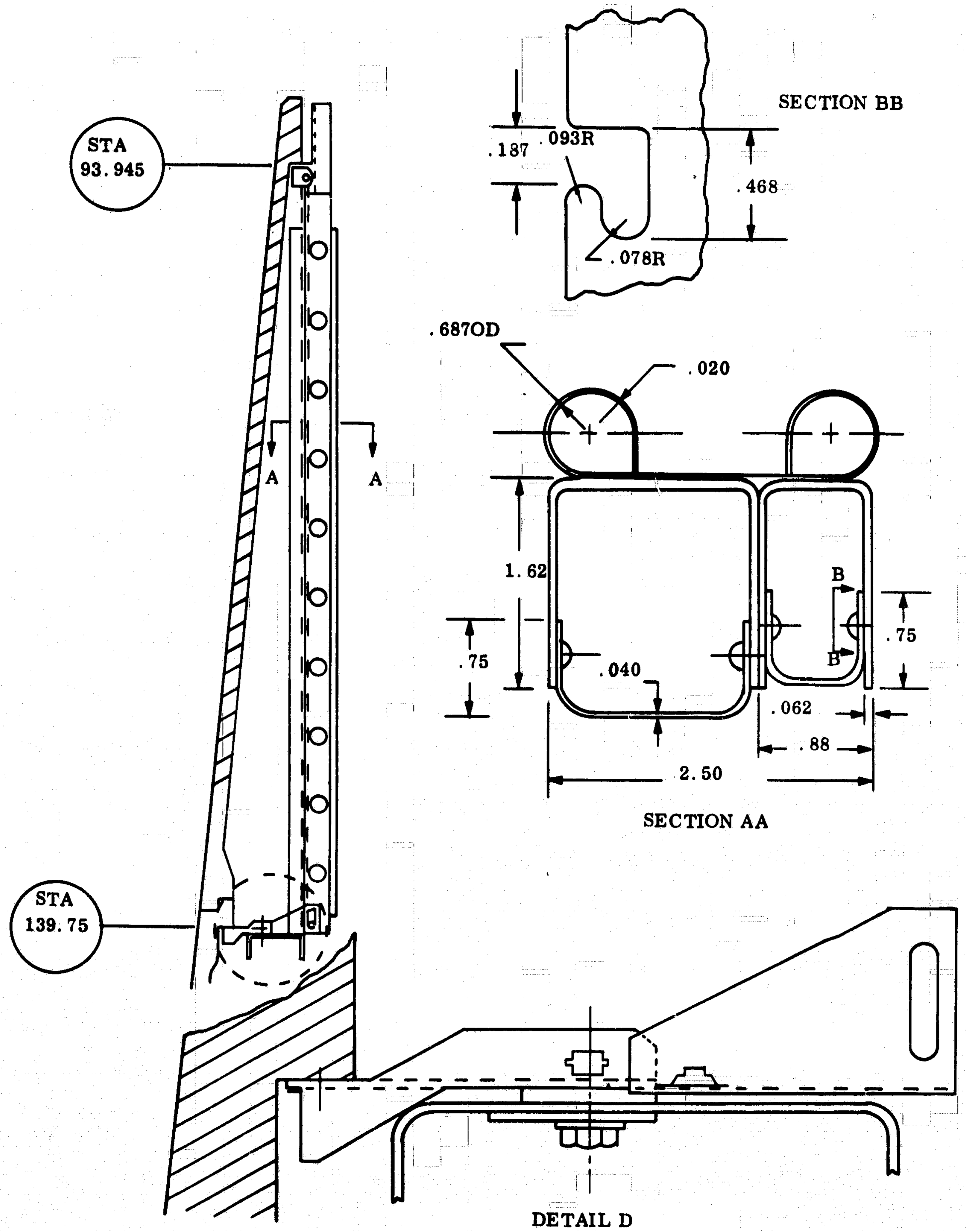


Figure 3-16. Aft Substructure

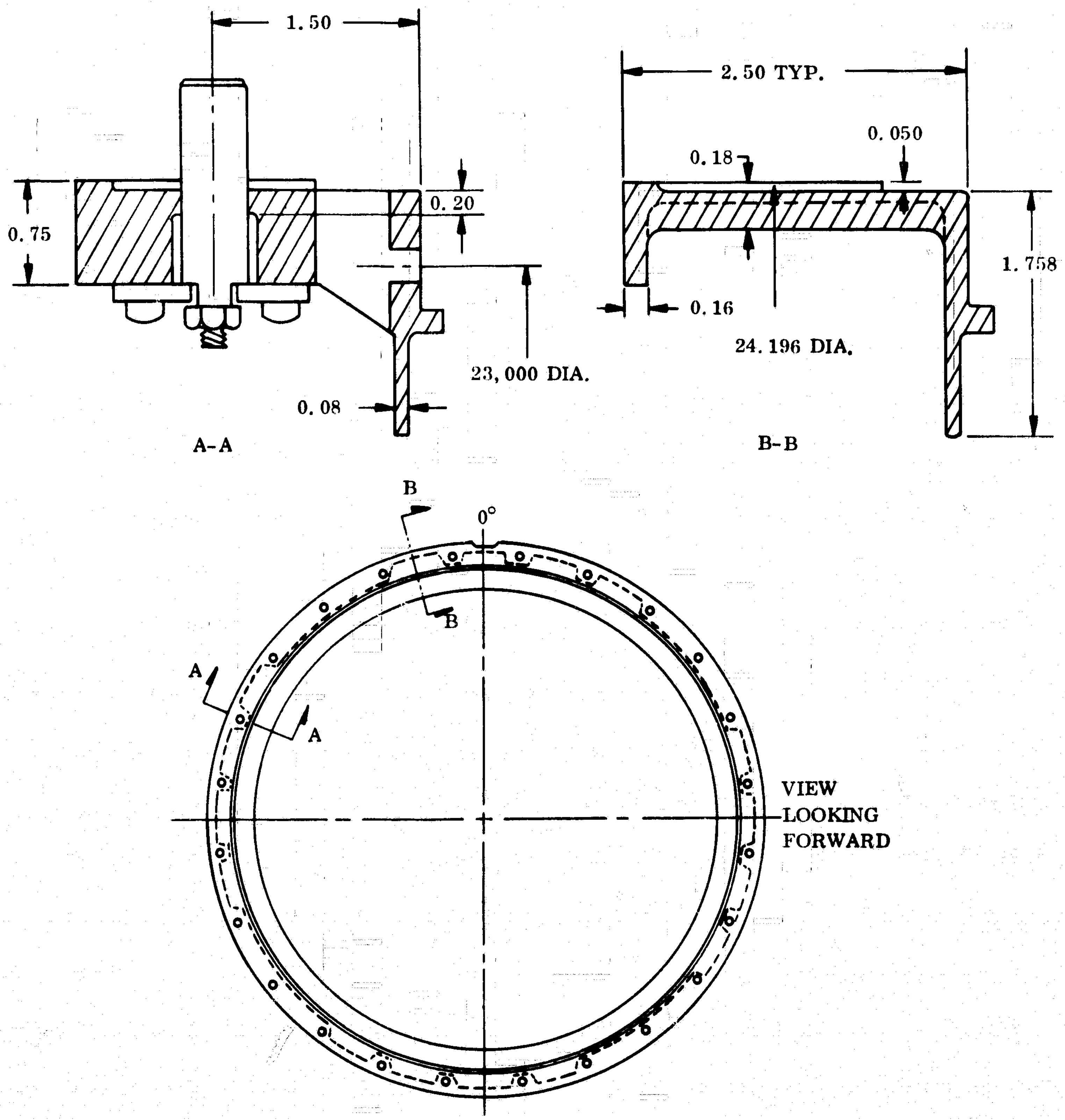


Figure 3-17. Closure Ring Design

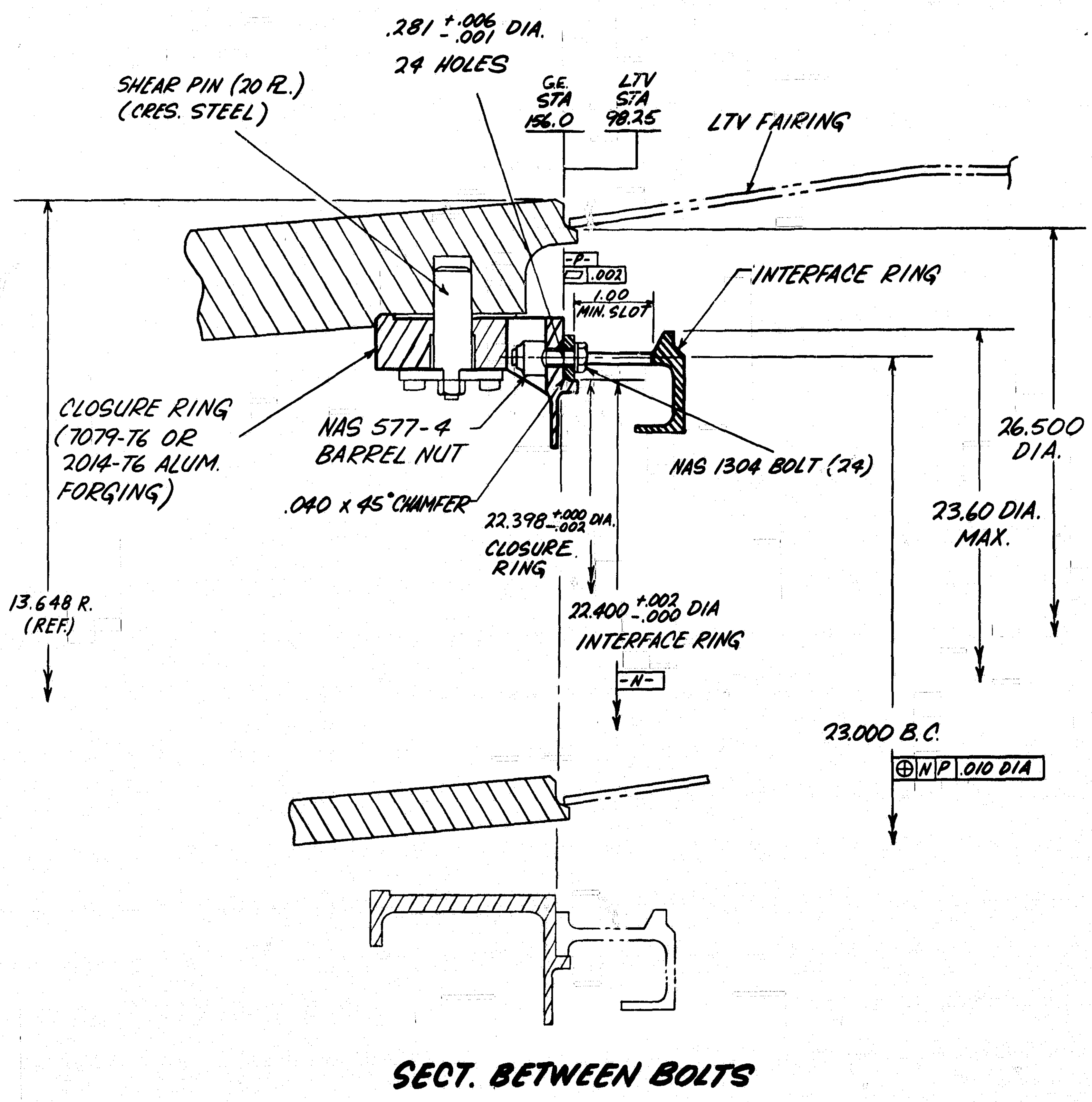


Figure 3-18. Re-entry F/Scout Mechanical Interface

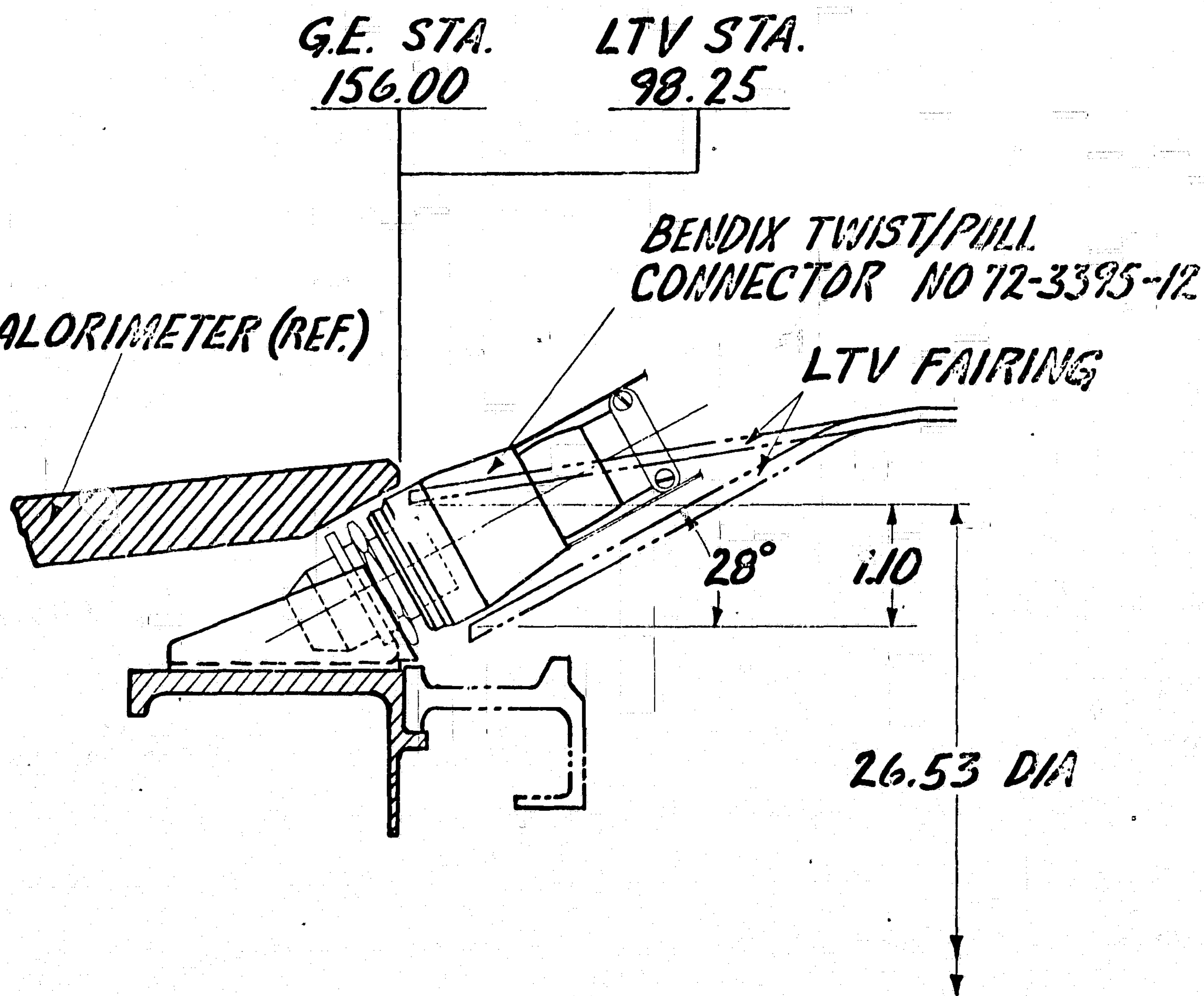
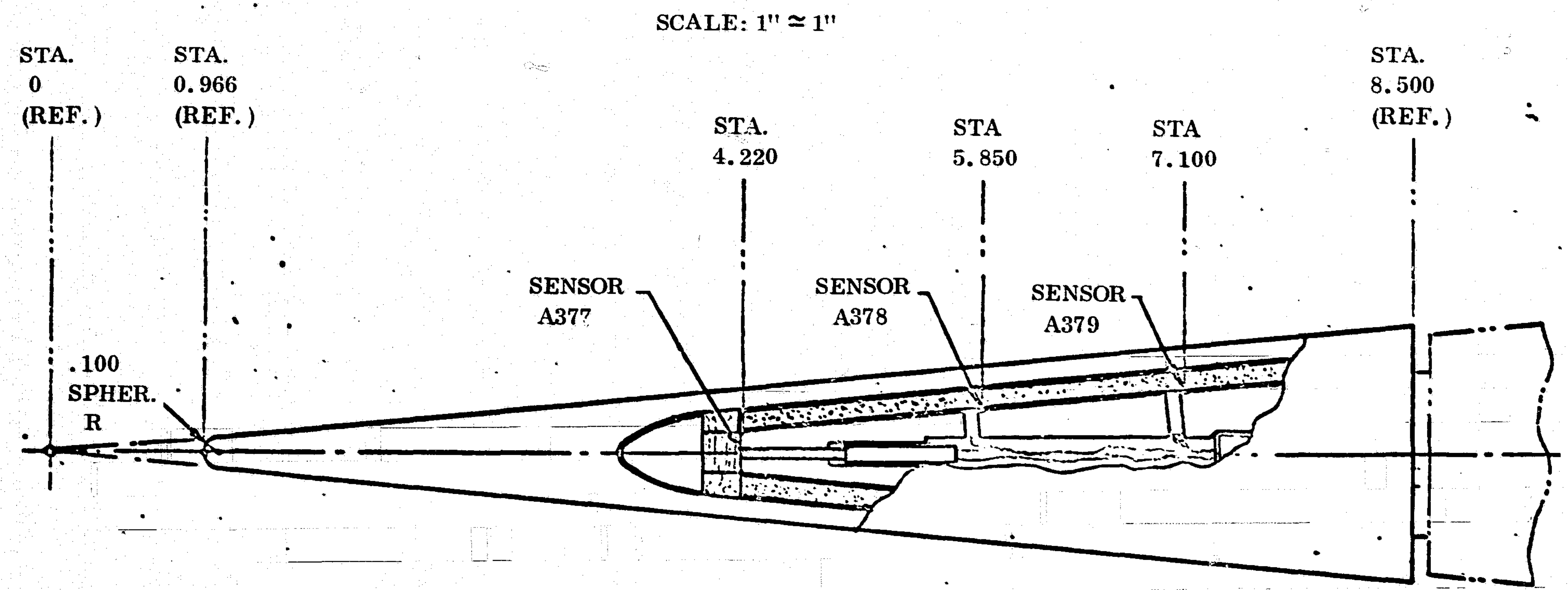
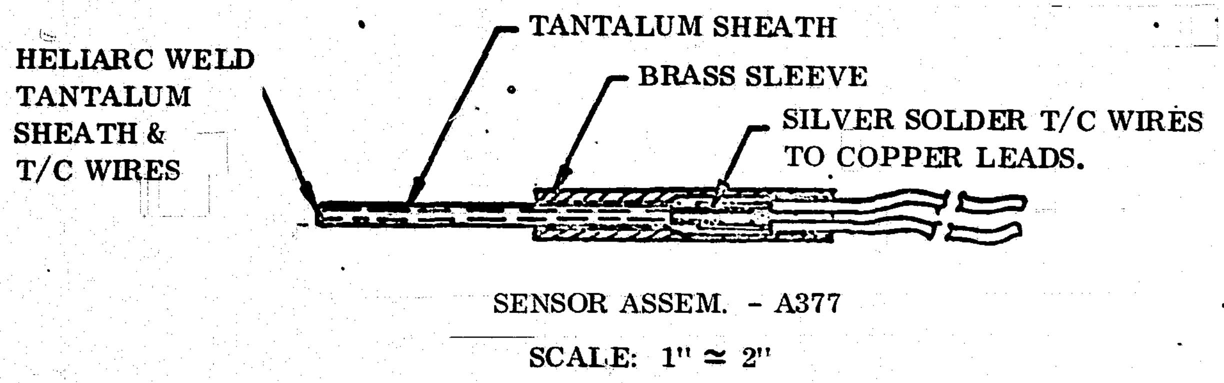
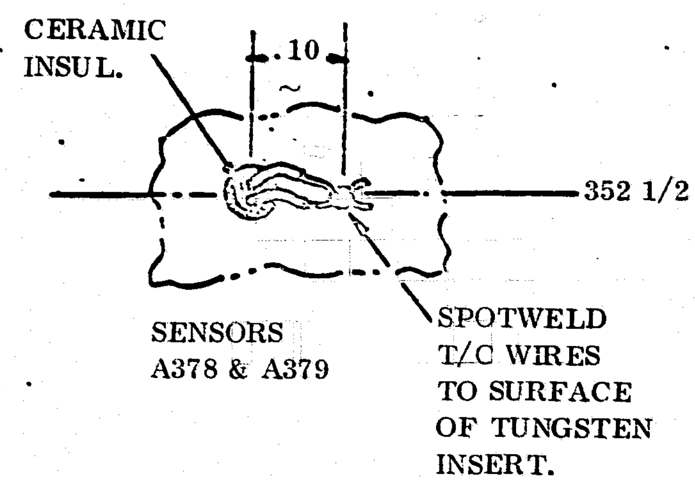


Figure 3-19. Re-entry F Umbilical Connection







SENSOR IDENT. NO.	(+)	(-)	MAX. RANGE
A377	W-5% Re.	W-26% Re	4400°F
A378	CHROMEL	ALUMEL	2300°F
A379	CHROMEL	ALUMEL	2300°F

Figure 3-20. Nose Tip Sensor Installation

The bonded composite nose tip structure of the Re-entry F nose is a follow-on to similar designs in previous programs and has these common design features:

1. The heat shield is free from cut-outs or stress raisers

2. Upon bond failure (stress-relief), the ATJ is a free-standing shell without mechanical constraint

3. The tungsten ballast extends the length of the skirt section to provide a lateral load transfer surface

The details of this design are discussed in Document SM-TM 8156-202 Re-entry F Spacecraft Structural Mechanics Stage 3 Release dated March 17, 1967.

3.4.5.1 The nose tip has the following design configuration:

a. ATJ Graphite Shell

b. 0.1 Inch Nose Radius

c. 5° Half Angle

d. 2.5 Inch Overhang

e. 7.453 Inch Length

f. 0.12 Inch Skirt Thickness

3.4.5.2 The Nose Tip environment and response are as follows:

- Maximum cold wall heat flux of 20,000 BTU/SEC-Ft^2 at 49,000 Ft
- Stagnation Pressure at 49,000 Ft is 60 ATM
- Total Recession of 0.77 inches at 49,000 Ft
- Nose Radius of 0.171 at 49,000 Ft
- 0.008 Inches Sidewall Ablation Average
- 0.005 Inches Recession at End of Skirt
- Tungsten Attachment Maintained at 1200° R

3.4.5.3 The nose tip has a manufactured gap of 0.100 inches and a rearward facing step of 0.042 inches when attached to the P1 frustum.

3.4.6 Sensors

There are basically four types of sensors utilized in the spacecraft:

a. Temperature

b. Pressure

c. Motion

d. Diagnostic

These sensors provide data on the thermal and pressure performance of the main body calorimeter and aft cover, and the spacecraft motion during the primary data period. Table 3-10 provides a list of thermal, pressure and motion sensors, and of diagnostic monitors.

3.4.6.1 Thermal Sensors

21 thermal sensors, each having 4 thermocouples located at nominal depths of 0.010, 0.10, 0.30, and 0.6 inches from the surface, are distributed over the calorimeter in four orthogonal rays as shown in Figure 3-9. Additional sensors on the aft cover (Figure 3-21) aid in the determination of time of transition of the base flow.

The prime thermal sensor shown on Figures 3-22 and 3-23 consists of a core with slots milled to the proper depths and distances to accommodate the four thermocouples comprising the sensor. A mandrel is provided to facilitate the assembly of the plug into the sleeve. The core is shrink fitted by cooling with liquid nitrogen and pressing it into the sleeve. The sensor assembly in turn is shrink fitted by the same process into a hole machined in the calorimeter. The mandrel is machined off after assembly and the surface is blended to form no protrusions in the finished vehicle.

3.4.6.2 Pressure Sensors

Body surface pressures are used to provide verification of the predicted pressure distribution used in the thermodynamic heating analysis, to provide a backup technique for determining the vehicle angle of attack and to detect any asymmetric condition. Base pressure sensors on the aft cover are used to detect the time of wake attachment at the base of the vehicle. Each pressure tap feeds two pressure sensors - a fine range (0 - 3 psi) and a coarse range (0 - 20 psi) for body pressure, and 0.1 and 1.0 psi for the base pressure.

The locations for the body pressure taps are consistent with the thermal sensor locations (about 1 1/2 inch behind the thermal sensor). There are six ports along the main heat transfer data ray (352.5 degree ray) two ports along the 82.5 ray; one along the 262.5 degree ray, and three ports along the 172.5 degree ray. At the most rearward station, an additional port is added to improve the angle-of-attack determination from the circumferential pressure distributions. The location of these surface pressure ports is shown in Figure 3-21.

TABLE 3-10 SENSOR INSTRUMENTATION

MEASUREMENT	RANGE	SAMPLING RATE (sps)	NUMBER
Pitch rate, coarse	0 ± 20°/sec.	60	1
Pitch rate, fine	0 ± 5°/sec.	cont.	1
Yaw rate, coarse	0 ± 20°/sec.	60	1
Yaw rate, fine	0 ± 5°/sec.	cont.	1
Roll rate, coarse	0 ± 2000°/sec.	60	1
Roll rate, fine	0 to 500°/sec.	cont.	1
Pitch acceleration, coarse	0 ± 10 g's	60	1
Pitch acceleration, fine	0 ± 2 g's	cont.	1
Yaw acceleration, coarse	0 ± 10 g's	60	1
Yaw acceleration, fine	0 ± 2 g's	cont.	1
Axial acceleration, coarse	+2 to -40 g's	cont.	1
Axial acceleration, medium	+2 to -10 g's	cont.	1
Axial acceleration, fine	+2 to -3 g's	cont.	1
Body pressure	0 to 3 psia	20	12
Body pressure	0 to 20 psia	40	13
Base pressure	0 to 0.1 psia	20	2
Base pressure	0 to 1.0 psia	20	2
Body pressure	0 to 50.0 psia	20	1
Temperature, Thermal Sensor			
Thermocouple #1	amb. to 2350° F	20	21
#2	amb. to 2350° F	10	21
#3	amb. to 2350° F	10	21
#4	amb. to 2350° F	10	21
Temperature, Nose	amb. to 2200° F	10	2
Temperature, Nose	amb. to 4200° F	10	1
TLM Incident Power	0 to 3.0 watts	20	1
TLM Reflected Power	0 to 2.5 watts	40	1
TLM Battery Monitor	0 to 34 volts	10	1
5 VDC P.S. 0% Ref. (A)	170 - 180 sec.	10	1
5 VDC P.S. 100% Ref. (A)	760 - 790 sec.	10	1
5 VDC P.S. 10% Ref. (A)	0.48 to 0.52 VDC	10	1
5 VDC P.S. 100% Ref. (B)	4.8 to 5.2 VDC	10	1
TLM Current Monitor	0 to 10 amps	10	1
Temp. Monitor, Commutators	32 to 200° F	10	1
Temp. Monitor, XMTR	32 to 200° F	10	1
Xponder Signal Perf. Monitor	3.5 to 5 VDC	10	1
Xponder Temp. & Volt. Monitor	50 to 180° F; 4.5 VDC	10	1
Tracking Voltage Monitor	0 to 34 VDC	10	1
Tracking Current Monitor	0 to 4 amps	10	1
T/C Cold Junction Temp. Ref.	20 to 140° F	10	1
Temp. Monitor (Mid Sect.)	32 to 200° F	10	1
Temp. Monitor (Aft Comp.)	32 to 200° F	10	1

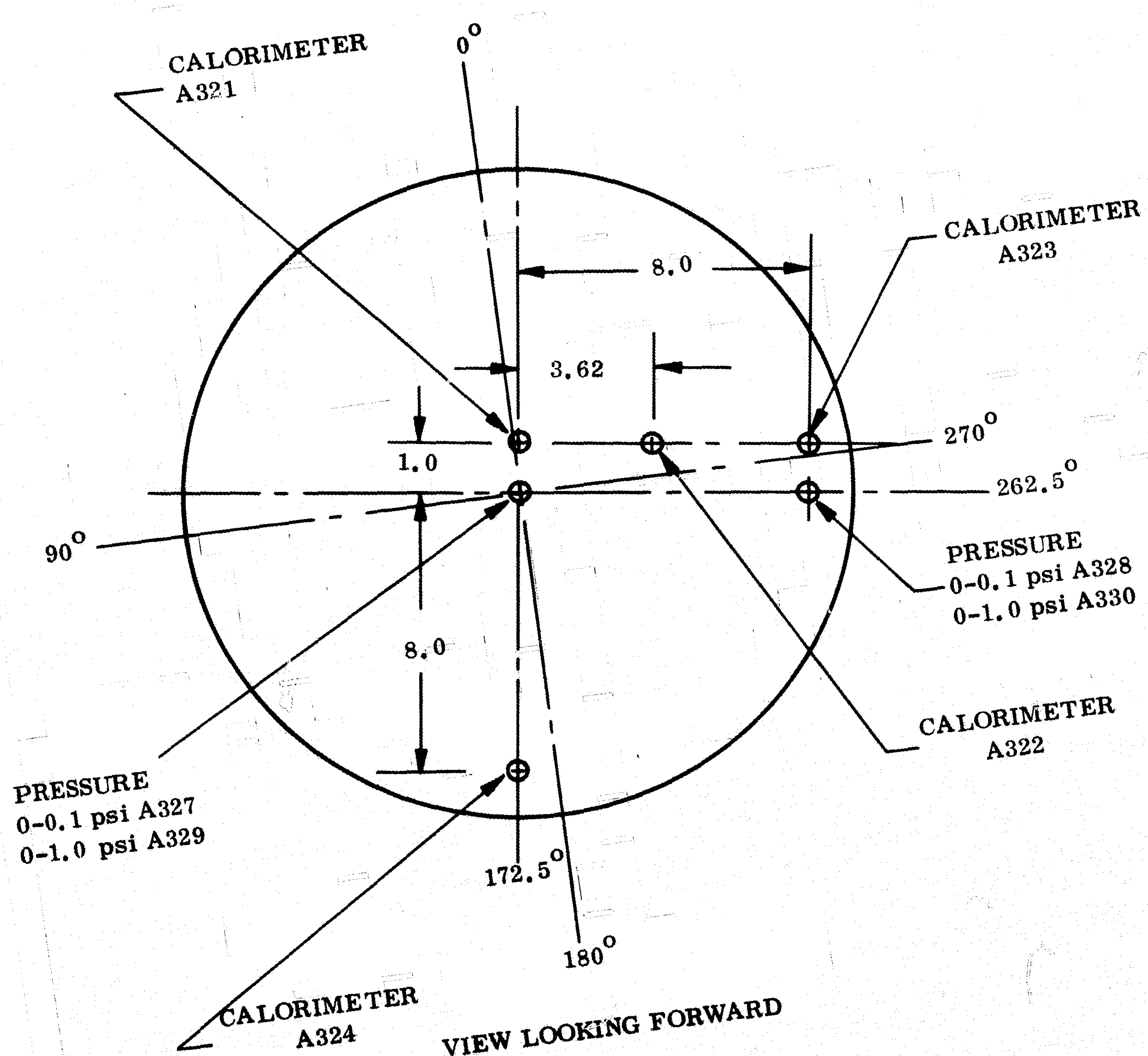


Figure 3-21. Aft Cover Sensors

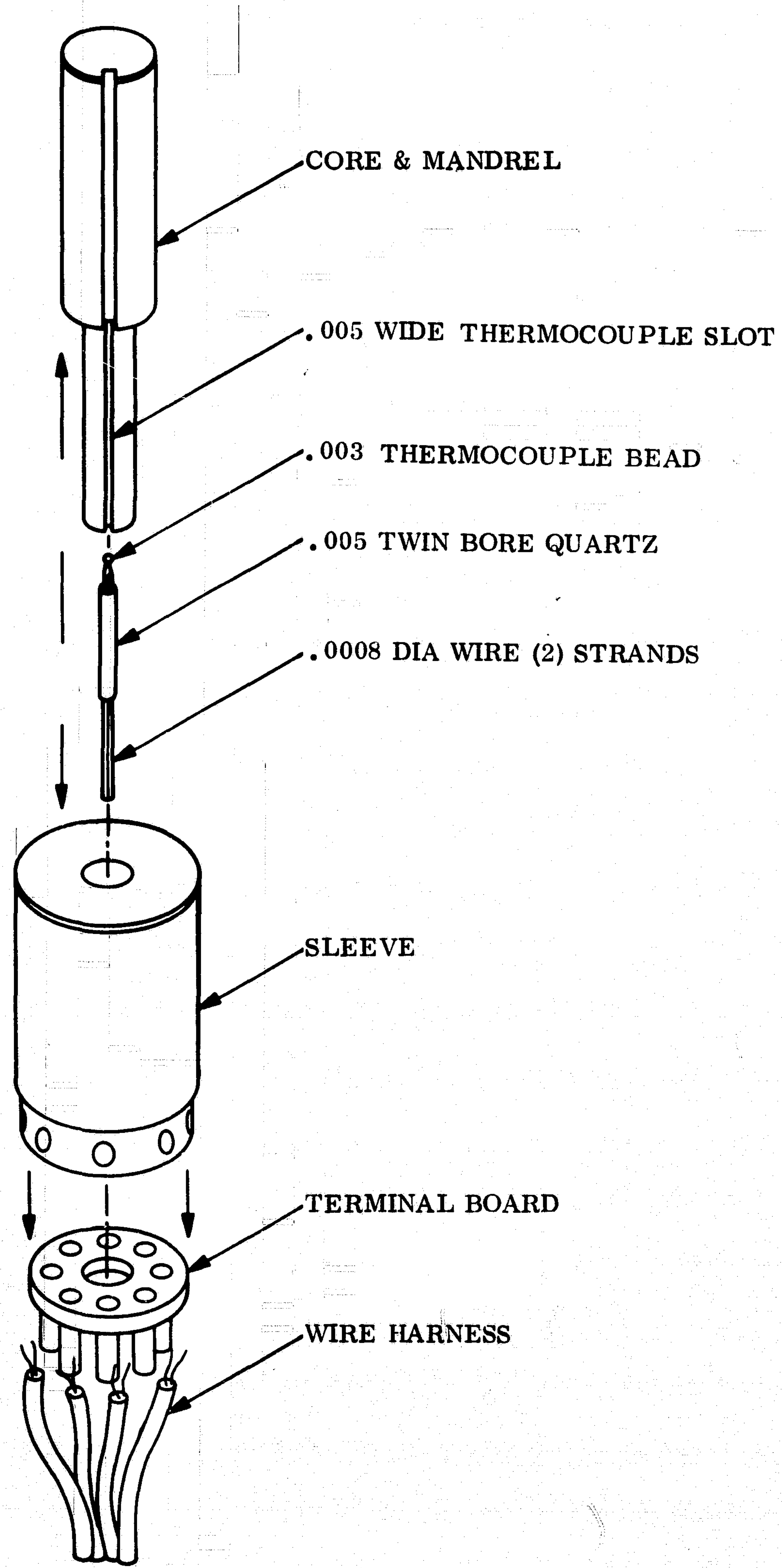


Figure 3-22. Thermal Sensor Assembly (47D172363)

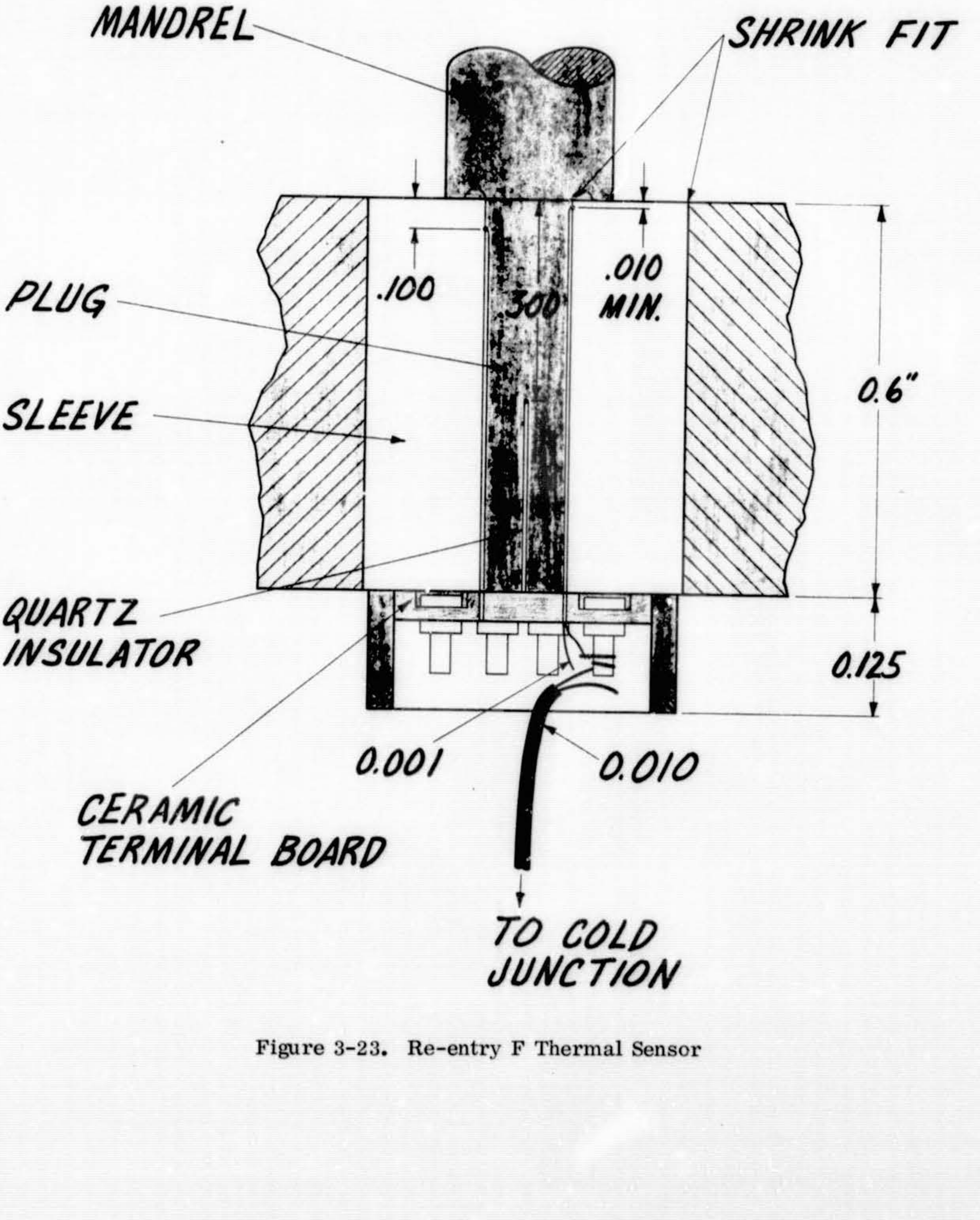


Figure 3-23. Re-entry F Thermal Sensor

3.4.6.3 Motion Sensors

Angular velocities about the Spacecraft pitch, yaw, and roll axis are measured by two tri-axial rate gyros. These rate sensing devices provide 0 to 5 vdc output signals proportioned to pitch, yaw and roll rates of the Spacecraft.

Accelerations along each axis are measured by non-pendulous force balance servo type linear accelerometers. These units provide a 0v to 5v output signal proportional to the sensed pitch, yaw and axial accelerations of the spacecraft.

3.4.6.4 Diagnostic Monitors

Various diagnostic sensors are included to monitor S/C electrical and other system performance.

3.4.7 Equipment Package

The equipment package assembly Figures 3-24 and 3-25 (Drawing 47R-197811) consists of a 28 inch beam supported by bulkheads at stations 60.38 and 90.00 and enclosed by a non-structural conical shell dust cover. Two bathtub type fittings attached to the aft bulkhead support two batteries mounted "piggy back". The entire assembly is made of 2024-T4 aluminum alloy. The assembly is supported laterally at the forward and aft bulkheads and axially at the aft bulkhead.

The components on the Equipment Package and their weights and applicable specifications are shown below:

Unit	Weight/Unit	Applicable Spec.
1. Power Controller	0.65 lbs.	6110-75-0033
2. Transmitter	2.25	5821-95-0011
3. SCO	1.51	5821-71-0007
4. C-Band Transponder	2.50	5821-96-0005
5. Signal Data Converter (2)	2.50	5895-35-0016
6. High Level Commutator	0.81	5895-65-0009
7. 5 Volt Power Supply	0.89	6130-15-0015
8. Low Level Commutator (2)	2.70	5895-65-0008
9. Accelerometers (7)	2.68	6680-01-0025
10. Filter Assembly (2)		5915-41-0005
11. Batteries (2)	10.0	6140-61-0016
12. Battery Support Brackets (2)	1.0 lb each	
13. Aft Bulkhead	3.73 lbs	
14. Fwd Bulkhead	1.65 lbs	
15. Rate Gyros (2)	2.20 lbs each	6615-30-0014
16. Dust Cover	3.0 lbs	

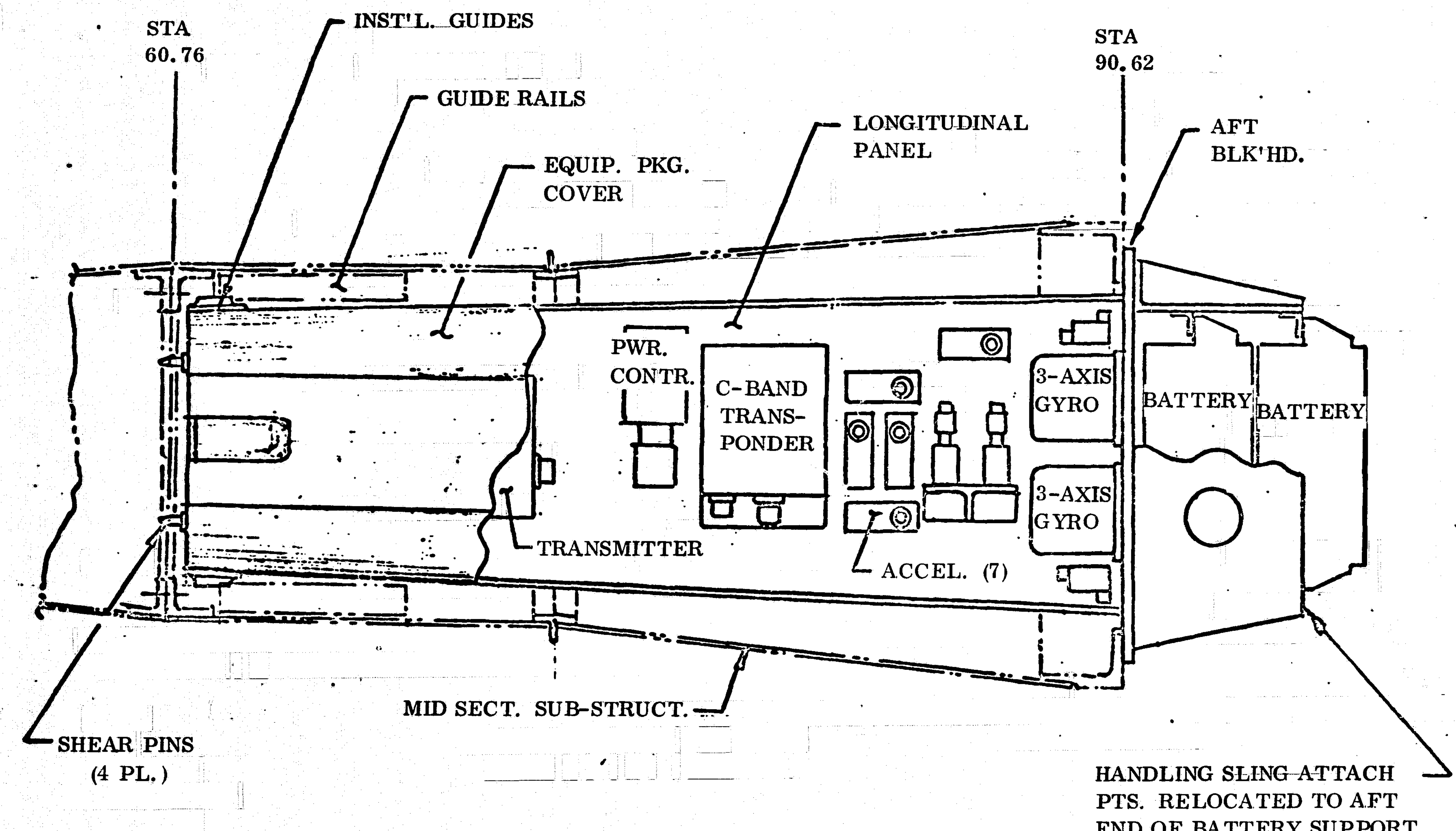


Figure 3-24. Equipment Package

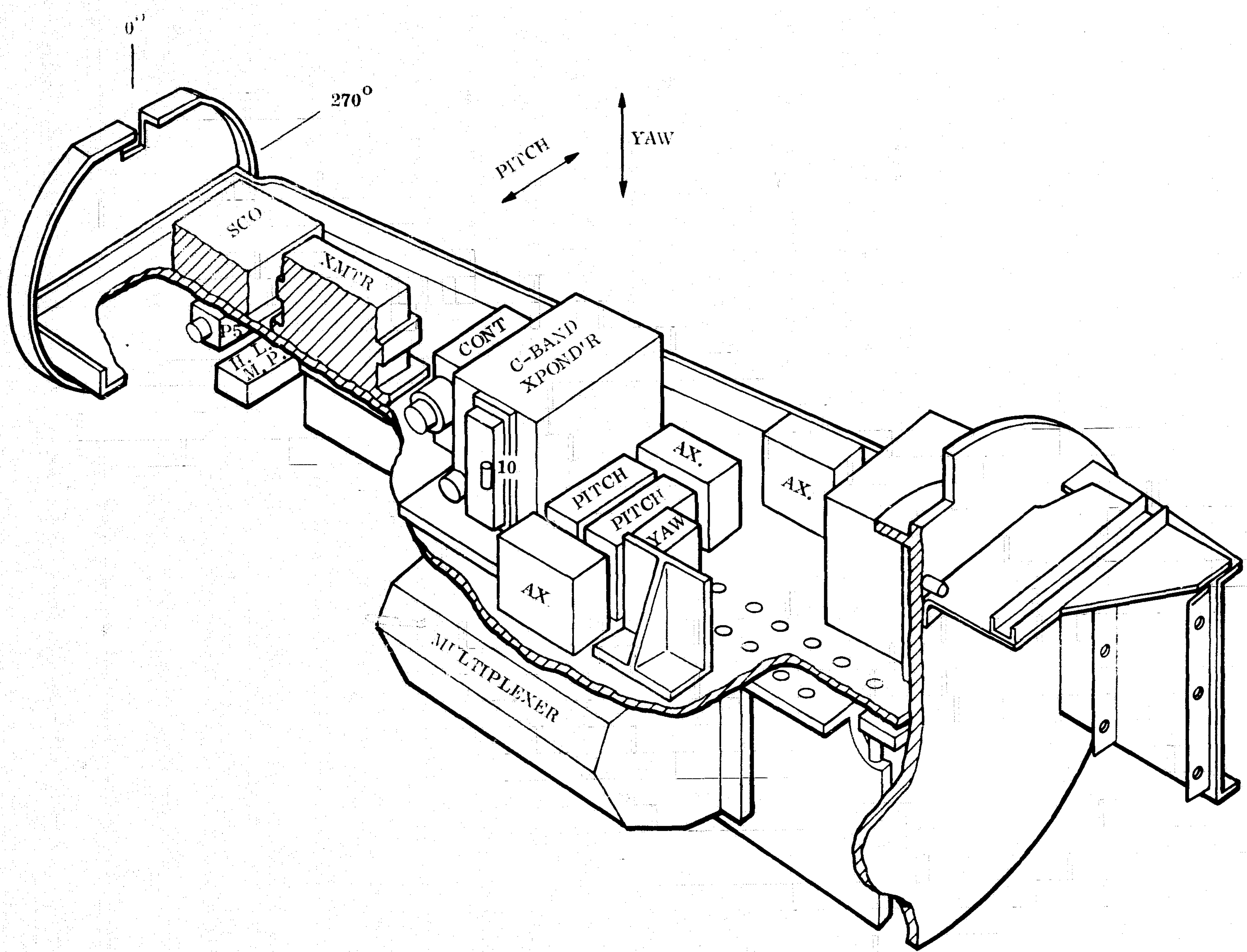


Figure 3-25. Equipment Package

3.4.8 Telemetry and Tracking

The Telecommunications Subsystem Figure 3-27 provides the means for transmitting data from the S/C in flight to the ground stations. The aerodynamic shape of the S/C and the construction of the S/C calorimeter permit continuous transmission of telemetry data throughout the flight until S/C melt. The Tracking Subsystem utilizes a transponder aboard the S/C to permit tracking of the S/C during re-entry. The tracking transmission is also continuous.

3.4.8.1 Telemetry Subsystem

This subsystem consists of a PDM/FM/FM link in the standard VHF band, operating on a carrier frequency of 253.8 MH_Z with a power output of 10 watts radiating from four VHF slot antennae through the VHF antennae windows, a section of which is shown in Figure 3-26. The VHF antenna system consists of an isolated power divider feeding four longitudinally-oriented slot-type VHF antenna elements located 90 degrees apart at the aft end of the calorimeter. A reflectometer in series with one of the antennae elements monitors the incident and reflected power.

The telemetry data is processed through eight subcarrier oscillators (IRIG channels 5 through 12) which are frequency-modulated by continuous high-level data (0 to 5v), and three multiplexed subcarrier oscillators (IRIG channels 19, B, and D) each of which is frequency-modulated by a pulse-duration-modulated (PDM) wavetrain provided by three electromechanical 90 x 10 multiplexers (commutators). Two of these three multiplexers accept low-level inputs (0 to 50 mv), while the third accepts high-level inputs (0 to 5v). Each 90 x 10 multiplexer sequentially samples up to 88 separate differential inputs 10 times per second, and provides a nominal 5-volt serial PDM output wavetrain to its associated SCO. The 90 x 10 high-level multiplexer samples high-level 0v to 5v single-ended inputs, and also provides a nominal 5-volt serial PDM output wavetrain to its SCO (Subcarrier Oscillator) unit.

The output of the eight continuous sub-carrier oscillators and three multiplexed SCO's are combined in a linear mixer-amplifier whose output is the composite FM signal used to modulate the 10-watt transmitter.

A dual 5v power supply provides two independent 5v regulated sources for instrumentation. In addition, one source provides calibration reference voltages at 10 per cent and 100 per cent levels for the high level commutator, and a differential 100 per cent calibration output (50 mv) for the low-level commutators. Each supply is short-circuit protected.

Two identical signal data converter units, each containing nine resistor divider networks, convert high-level (0v to 5v) single-ended data inputs to two-wire differential low-level (0 to 50 mv) signal outputs. This provides the capability to utilize low-level commutator channels,

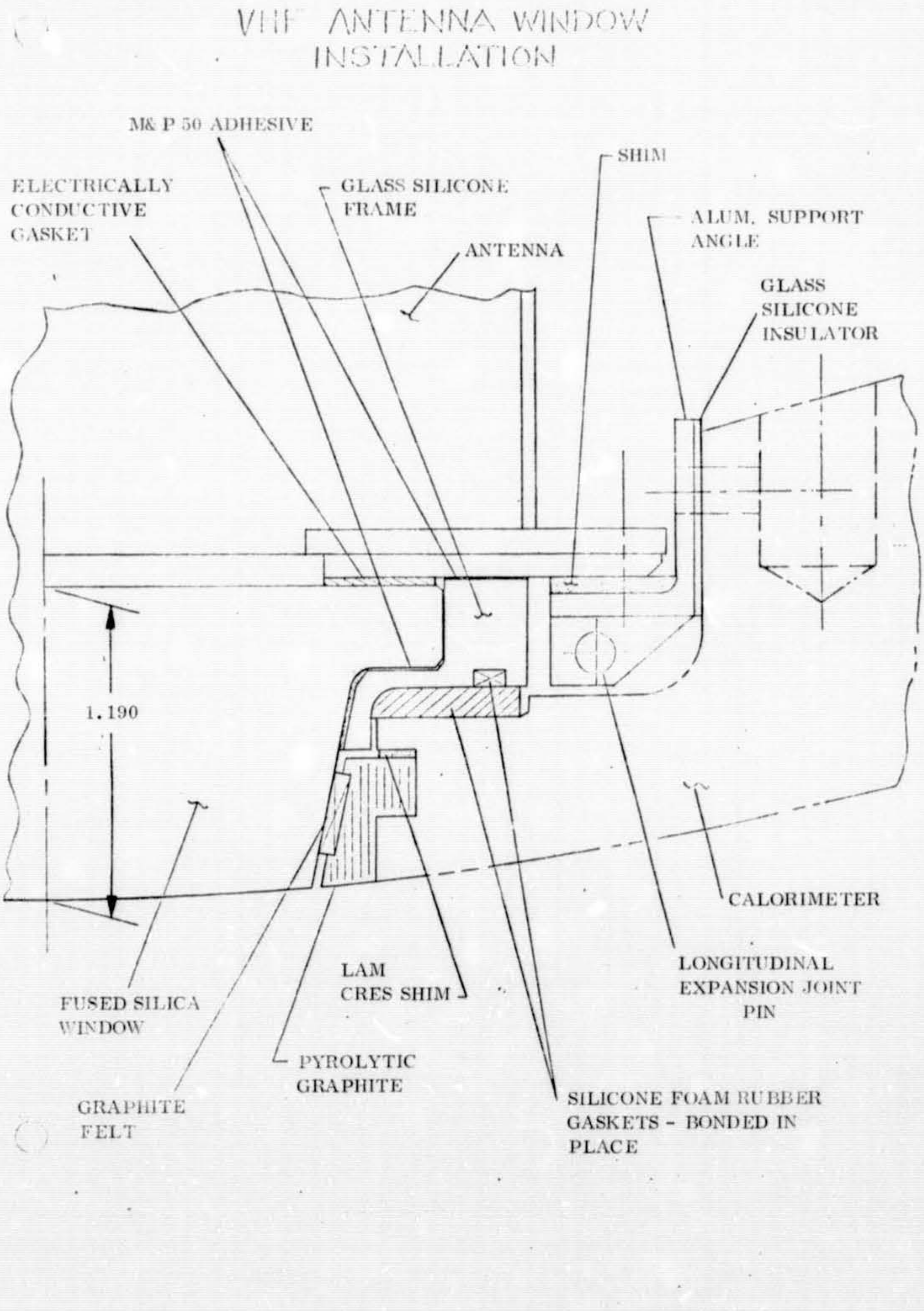


Figure 3-26. VHF Antenna Window Inst'l.

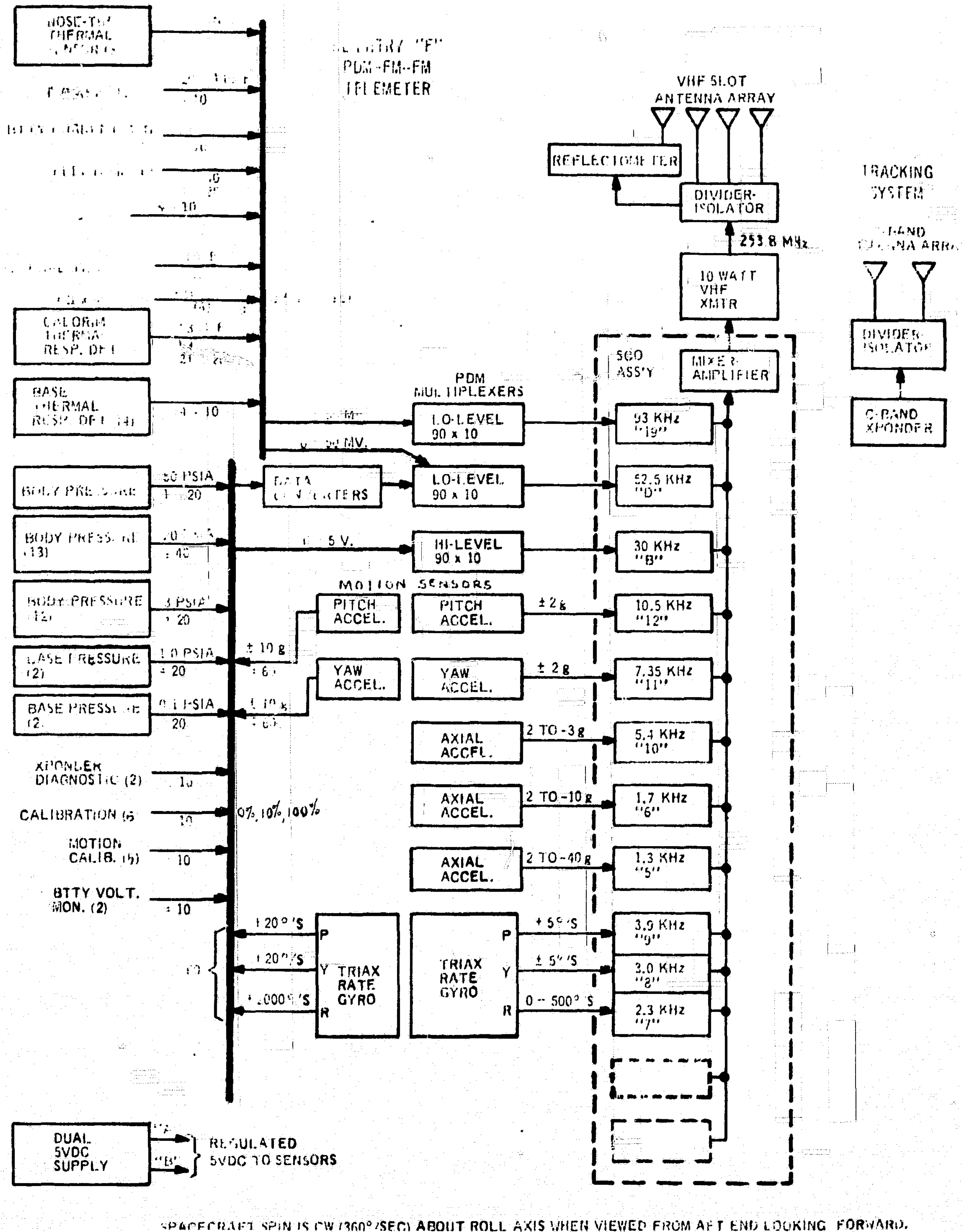


Figure 3-27. Telemetry and Tracking Subsystems

as required, for certain high-level instrumentation. The telecommunication subsystem current requirement will not exceed 6.0 amperes with the telemetry link and all instrumentation operative. The instrumentation consists primarily of temperature, pressure, and motion sensors. All telemetered measurements, their ranges and assigned channels are shown in the Instrumentation and Communications Subsystem Specification S0150-00-0016.

3.4.8.2 Tracking Subsystem

The tracking subsystem permits radar tracking of the Spacecraft during flight, utilizing a C-Band transponder and a C-Band antenna. The C-Band transponder provides a 500-watt 0.5 μsec output pulse, and a receiving sensitivity of 67-dbm. The C-Band antenna system consists of an isolated power divider feeding coax-to-waveguide transitions which radiate through two quartz windows located 180 degrees apart at the aft end of the spacecraft calorimeter. The tracking subsystem current requirement will not exceed 1.5 amperes with the C-Band transponder operative and being interrogated at a pulse repetition rate of 100pps. Transponder temperature is monitored through the telemetry system.

3.4.9 Electrical Power and Distribution

The Electrical System encompasses the power sources for the two subsystems (telemetry and tracking) a power control function, and the Aerospace Ground Equipment (AGE) interface as well as the electrical system interconnection harnesses.

The Electrical System mission profile may be broken into three major events:

(1) Pre-Flight Checkout

(2) Pre-Launch Condition

(3) Flight

Preflight checkout is normally performed on external power supplied from the launch complex. Approximately four minutes prior to lift-off the electrical system is commanded on internal power. As the S/C lifts off, its connection with the launch umbilical is severed by a pull-away action. During the flight the S/C is self-reliant. No electrical interface exists between it and the Scout booster.

Power is supplied by two manually-activated Silver Zinc batteries.

The Telemetry Subsystem battery is capable of supplying 5.0 ± 1 amperes for a period up to 30 minutes. The Tracking Subsystem battery supplies 2.0 (+2.0 and -.5) amperes for the same period of time. A Power Controller is provided to allow control of the electrical system from the launch pad control complex. It includes the capability for switching and controlling up to 6 amperes reliably.

A common ground reference is provided throughout the spacecraft in order to establish an equipotential level about which all EMF's are generated and measured. The ground reference is common with the return side of all power and signal paths, and at one point only.

3.4.10 Weight and Balance

3.4.10.1 A typical weight and balance summary of the Re-entry F Spacecraft and its various components is given in Table 3-11.

3.4.10.2 The values for moments of inertia and static margin for the Re-entry F Spacecraft are approximately as shown below:

a. Weight 600 lbs

b. Roll Moment of Inertia (Ir) = 7.26 slug ft^2

c. Pitch & Yaw Moments of Inertia (I_{p-y}) = 241.04 slug ft^2

d. Static Margin 7.5% (CG location 93.54)

TABLE 3-11. WEIGHT AND BALANCE SUMMARY

Item	Weight (lb.)		C.G. Sta. (in.)	
Nose	2.03		9.49	
Calorimeter	292.68		106.11	
Structure	121.47		74.58	
Ballast Fitting		69.27		38.92
Fwd Substructure		3.32		54.91
Mid Substructure		14.32		80.29
Aft Substructure		11.44		125.03
Interface Structure		15.36		155.47
Aft Cover		6.10		156.37
Rim Cover		1.66		156.09
Equipment Package	80.09		86.62	
Structure		20.51		81.66
Equipment		50.81		84.43
Harness		8.77		110.87
Antennas	22.70		149.26	
C-Band		3.34		150.48
VHF		19.36		149.05
Miscellaneous	21.00		123.80	
Fwd/Mid Equipment		3.44		69.96
Aft Equipment		4.14		130.57
Aft Cover Equipment		4.40		154.11
Transducer Wiring		2.25		116.62
Antenna Cables		2.57		142.26
Sensor Wiring		1.53		107.03
Connectors, etc.		2.67		130.63
Ballast	60.03		51.04	
Dynamic Allowance*		7.40		155.30
Weight Trim (Fwd Cavity)		37.63		35.91
Adjustment		15.00		37.57
Totals	600.00		93.54	

*Spacecraft Weight prior to dynamic trim is 592.60 pounds. The location of final ballast masses depends on measured mass properties.

3.5 Manufacturing

Manufacturing includes the functions of procurement, fabrication, assembly and shipping. Figure 3-28 is a flow plan of the manufacturing process for the R-entry F spacecraft.

3.5.1 Procurement

The Manufacturing operation administers the procurement of all material on the Re-entry F Program. Each item to be procured is placed into a Make or Buy category.

3.5.1.1 Buy Items

A Material Request (MR) is prepared with applicable drawings, specs and vendor instructions and QAP's (Quality Assurance Provisions) where applicable. A request for quote is sent to a minimum of 3 approved vendors if the value of the MR exceeds $500. The order is awarded to the bidder selected by the award group after the bids have been reviewed. On items under $500 the order is placed with an approved vendor who can meet the schedule.

3.5.1.2 Make Items

For make items, a material list is prepared and MR's written for parts not carried as standard stock. Specifications and drawings are attached to the MR and forwarded to Purchasing who follow the same procedure as with the Buy items.

Table 3-12 gives a list of components for the spacecraft together with the Make or Buy status and the quantities required.

3.5.2 Fabrication

3.5.2.1 Nose Assembly

The ATJ graphite is received in large blocks and is machined with the outer surface 1/4-inch oversize. The porous graphite is machined into a solid tapered cone. The porous carbon, ATJ plug and graphite felt are assembled, and the porous carbon is pre-bonded with C-10 cement to the ATJ shield with a $500^{\circ}F$ heat treatment. The assembly is then put through a final high temperature cure after which it is X-rayed and the internal diameter of the porous carbon is machined. Temperature sensors are welded to a tungsten insert and the insert assembly is bonded to the shield assembly with a silicone bonding agent, PD 162A.

After PD 162A cure, the aft face is machined to drawing tolerances and a pyrolytic graphite washer is bonded. The nose assembly is then bolted to the P1 calorimeter

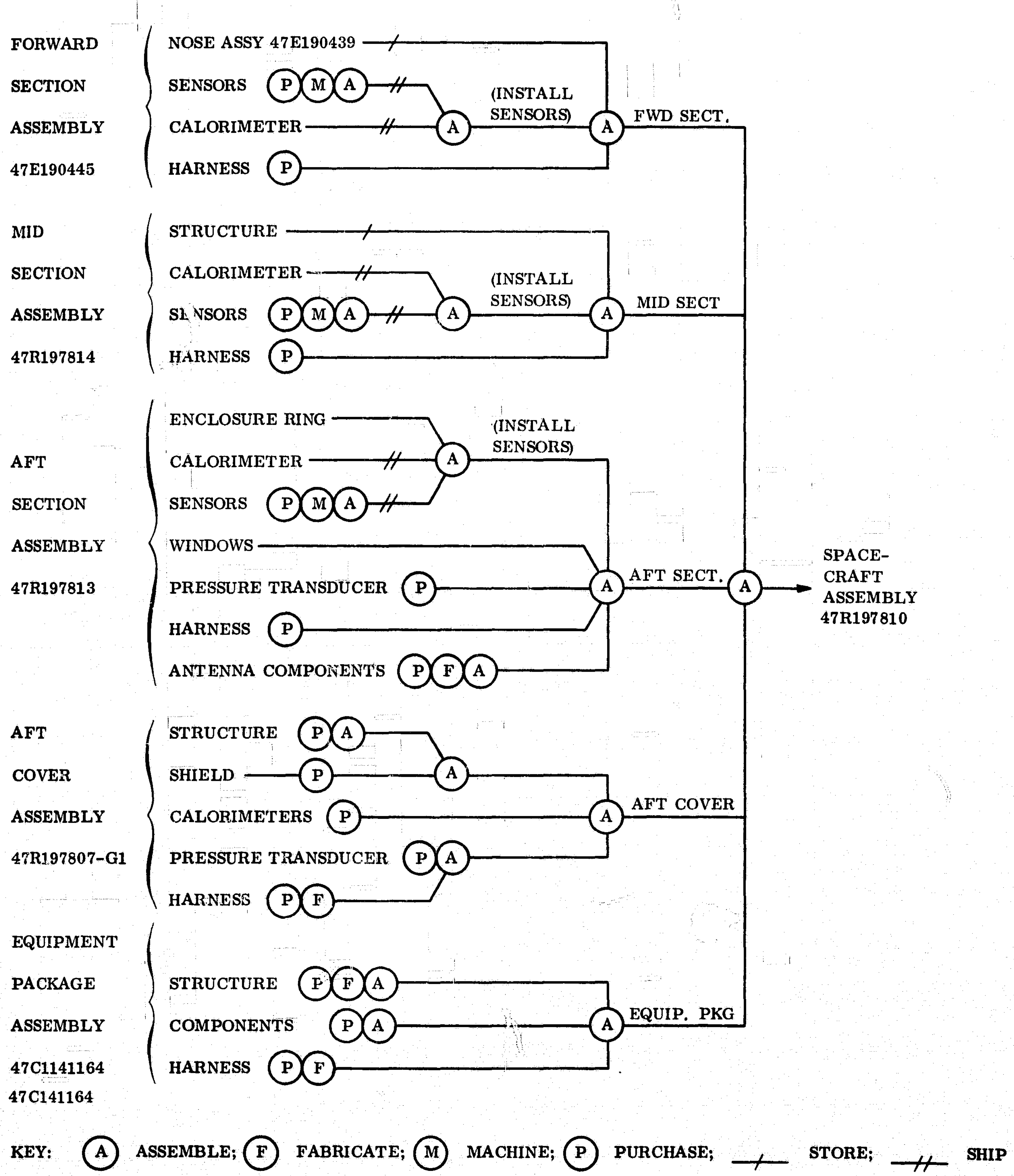


Figure 3-28. Manufacturing Flow Plan

TABLE 3-12. COMPONENT MAKE OR BUY

COMPONENT SHELF	Make or Buy	Qty Per S/C
Transmitter	Make	1
Data Converter	Make	2
Power Controller	Make	1
5 Volt Power Supply	Make	1
Filter Assy.	Make	2
"C" Band Transponder	Buy	1
Battery	Buy	2
Gyro	Buy	1
Gyro	Buy	1
SCO Assembly	Buy	1
Commutator (Low)	Buy	1
Commutator (Low)	Buy	1
Commutator (High)	Buy	1
Accel. (+2 to -3g)	Buy	1
Accel. (+2 to -10g)	Buy	1
Accel. (+3 to -30g)	Buy	1
Accel. (± 2g)	Buy	2
Accel. (± 10g)	Buy	2
VEHICLE		
Reflectometer	Make	1
VHF Antenna	Make	4
VHF Power Divider	Make	1
"C" Band Power Divider	Make	1
"C" Band Adptr.	Buy	2
Pressure Xdcr. 50#	Buy	1
Pressure Xdcr. 1#	Buy	2
Pressure Xdcr. 1#	Buy	2
Pressure Xdcr. 3#	Buy	12
Pressure Xdcr. 20#	Buy	13
Thermal Sensor	Buy	21
Temperature Sensor	Buy	1
Calorimeter	Buy	4
Temperature Sensor Assy.	Buy	4

frustum. The unit is machined to size in this configuration to maintain the tight T. I. R. requirements, and is X-rayed to ascertain that the assembly is acceptable.

3.5.2.2 Calorimeters

The calorimeters of all prime spacecraft and sensors are obtained from beryllium billets manufactured from a common beryllium powder lot in order to maintain close control of material properties for all pieces. The calorimeter blanks P2 through P7 for each spacecraft are trepanned from one billet. The P1 frustums for all spacecraft and the thermal sensors for all spacecraft are made from separate billets. Thermal characterization, density and tensile tests are performed on samples of each billet.

The beryllium billets and frustum blanks are produced by Brush Beryllium Co. and then shipped to American Beryllium Co. (ABC) for final machining and the sensor installation.

Following the completion of the match-machining of the first five frustums (P1 to P5), frustum P6 is match-machined to P5, and P7 is then match-machined to P6 to assure that the T. I. R. (Total Indicated Runout) is held to acceptable limits. The aft enclosure ring is assembled and installed into the P7 frustum just prior to match-machining it to P6.

After completion of the match-machining operation, the frustums are disassembled and the sensor holes are drilled and lapped for pre-selected sensors. The pressure ports and temperature sensors are installed into the Calorimeter at ABC by using liquid nitrogen (about-350°F) to shrink the sensors before pressing them into the calorimeter frustums.

3.5.3 Assembly

3.5.3.1 Sub-assembly

Following the completion of the blending of the pressure and thermal sensors, the sub-assemblies are installed in the frustums. The C-band and VHF arrays are installed in the P-7 frustum followed by the ring assembly containing the pressure transducers, reflectometer, isolator power dividers, and the harness.

The frustums are then assembled to form the forward assembly (nose and P-1 frustum), the mid section assembly (P2, P3 and P4 frustums), and the aft assembly (P5, P6 and P7 frustums).

The harnesses are routed through troughs and terminated in the rear assembly ring.

3.5.3.2 Final Assembly

The assembly sequence for the completed spacecraft is shown in Figure 3-29.

3.5.4 Shipping

The spacecraft is assembled into the spacecraft shipping container for delivery. A polypropylene protective cover is placed over the entire vehicle during final test and shipping to insure the quality of the finish on the spacecraft. In addition a Synthetic Plastic cover is included with the shipping container for added protection.

3.6 Reliability

The reliability effort is described in GE/RSD Document No. 66SD948A. Reliability Program Plan/Re-entry F Turbulent Heating Experiment, which is in accordance with NPC 250-1, "Reliability Provisions for Space System Contractors".

3.6.1 Reliability Activities

The major Reliability activities are indicated in the Reliability Management Matrix, Figure 3-30.

In addition, the Reliability Program encompasses the following elements:

a. Reliability Design Analysis

b. Component Parts

c. Specifications and Standards

d. Design Review

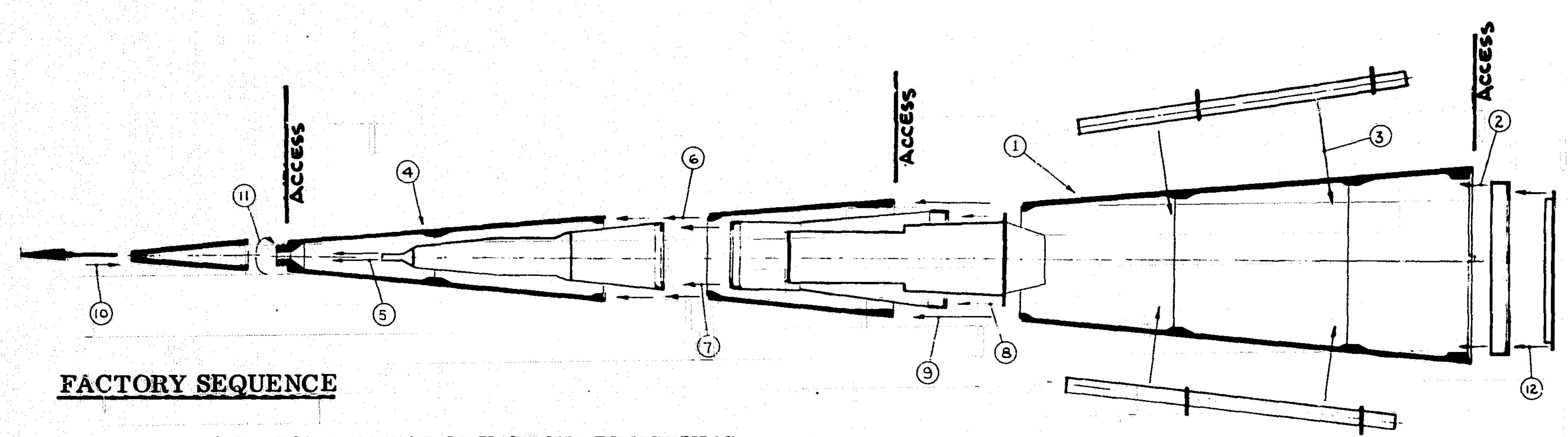
e. Design Change Control

f. Failure Reporting and Analysis

3.6.2 Design Specifications

The System Requirements Specification is the technical interpretation of the Work Statement and indicates what is to be done. From this document the subsystem and component design and performance requirements are generated.

The component Qualification Specification indicates the test program required for component qualification and acceptance testing. Test data recording, failure reporting, etc., are included in the Qualification Specification. This Specification is compatible with the System Acceptance and Test Specification generated by Engineering.



FACTORY SEQUENCE

1. ASSEMBLE AFT CALORIMETER FRUSTUMS
2. INSTALL CLOSURE RING
3. INSTALL AFT STRUCT. MEMBERS
4. ASSEMBLE FIRST TWO FRUSTUMS OF MID CALORIMETER
5. INSTALL FWD. PORTION OF MID STRUCT.
6. ATTACH AFT FRUSTUM OF MID CALORIMETER
7. INSTALL AFT PORTION OF MID STRUCTURE
8. INSTALL EQUIP. PK'G.
9. JOIN MID & AFT SECT. ASSEMBLIES
10. ATTACH NOSE TO FWD. CALORIMETER FRUSTUM
11. JOIN FWD. SECT. TO S/C
12. ATTACH AFT COVER

SEQUENCE (NORMAL MAINT.)

A. EQUIP. PK'G. REMOVAL: PERFORM STEPS (8), (9), & (12) IN REVERSE ORDER.

B. NOSE TIP REMOVAL: PERFORM STEPS (10) & (11) IN REVERSE ORDER.

A1. REMOVE AFT COVER, ATTACH EQUIPMENT PACKAGE SLING, REMOVE BOLTS IN EQUIPMENT PACKAGE RING, REMOVE EQUIPMENT PACKAGE.

Figure 3-29. S/C Assembly Sequence

[redacted]

TOP LEVEL PROGRAM DOCUMENTATION	NUMBER	REVISION	ISSUE DATE
SYSTEM DESIGN SPECIFICATION	0010-02-0026	A	FW 12
EXTERNAL ENVIRONMENT SPECIFICATION	0020-01-0015		FW 4
INTERNAL ENVIRONMENT SPECIFICATION	0020-02-0020	A	FW 6
EMI CONTROL PLAN	0010-09-0019		FW 5
ELECTRICAL SYSTEM REQUIREMENT SPEC.	0010-08-0008	A	FW 7
INTEGRATED TEST PLAN	66SD961		8 AUG 66
SYSTEM ACCEPTANCE TEST SPECS.	0040-01-0035		22 / 22
SELECTED PARTS LIST AVE	0030-03-0016	A	FW 7
SELECTED PARTS LIST AGE	0060-04-0005	A	FW 7
RELIABILITY PROGRAM PLAN	66SD948	A	30 AUG 66
SELECTED MATERIALS & PROCESSES LIST REF	0060-07-0004		13 DEC 66

SYSTEM DEFINITION / MONITORING EVENT					RELIABILITY ESTIMATE ANALYSIS		
		SPECIFICATION					
NOMENCLATURE	DRAWING NUMBER	NUMBER	ITPB APPROV.	SPEC ISSUE	DOCUMENT	DATE	REL. EST.
TELEMETRY TRACKING & COMMAND SUBSYSTEM		0150-00-0016	NA		REA 54B	10	.9993
THERMAL SENSOR	47D172363	6685-51-0016	7 / 7	7	REA 54B	10	.999999
TRANSDUCER, PRESSURE	47C142331	6685-46-0028	8 / 8	8	REA 54B	10	.999999
CALORIMETER, AFT COVER	47C142821	6630-10-0010	15 / 15	16	REA 54B	10	.999999
ACCELEROMETER (PITCH, YAW)	47D172303	6680-01-0025	9 / 9	9	REA 54B	10	.999995
SCO ASSEMBLY	47C142095	5821-71-0007	9 / 9	9	REA 54B	10	.99988
SIGNAL DATA CONVERTER	47D172462	5895-35-0016	11 / 9	9	REA 54B	10	.99999
MULTIPLEXER, LOW LEVEL, 90 X 10	47C142323	5895-65-0008	11 / 9	9	REA 54B	10	.99998
TRANSMITTER	47D172076	5821-95-0011	9 / 9	9	REA 54B	10	.99996
C-BAND TRANSPONDER	47C142074	5821-96-0005	7 / 8	9	REA 54B	10	.99998
REFLECTOMETER	47D172057	6625-55-0003	11 / 16	16	REA 54B	10	.999997
ANTENNA, VHF	47E189957	5985-31-0019	8 / 8	9	REA 54B	10	.999999-
ANTENNA, C BAND	47C142286	5985-18-0002	8 / 8	8	REA 54B	10	.999999-
POWER SUPPLY, 5 VOLTS	47D172070	6130-15-0015	9 / 9	9	REA 54B	10	.999991
RATE GYRO, THREE AXIS	47D167497	6615-30-0014	10 / 9	9	REA 54B	10	.99998
MULTIPLEXER, HIGH LEVEL, 90 X 10	47C142322	5895-65-0009	11 / 9	9	REA 54B	10	.99998
TRANSDUCER, PRESSURE, 0-3, 0-20	47C142339	6685-46-0029	8 / 8	8	REA 54B	10	.999999
SENSOR, TEMP.	47C141629	6885-66-0001	11 / 11	12	REA 54B	10	.999999
FILTER ASSEMBLY	47C142665	5915-41-0005	14 / 14	14	REA 54B	10	.99998
EP&P SUBSYSTEM			NA		REA 54B	10	.9993
UMBILICAL							
BATTERY STORAGE	47C172063	6140-61-0016	10 / 10	10	REA 54B	10	.99999
CONTROLLER, POWER	47D172077	6110-75-0033	10 / 8	8	REA 54B	10	.999999-
POWER DIVIDER, C-BAND	47C143017	5985-03-0008	11 / 16	16	REA 54B	10	.999999-
POWER DIVIDER, VHF FOUR WAY	47C143027	5985-03-0007	11 / 16	16	REA 54B	10	.999999-

Figure 3-30. Reliability Mgmt. Matrix

ISSUE DATE 31 JULY 1967

FAILURE ANALYSIS & CORRECTIVE ACTION		FAILURE MODE EFFECT & CRITICALITY ANALYSIS		DESIGN REVIEW		QUALIFICIATION	
SIG. FAIL. / TOT. FAIL.	FAR	DOCUMENT	DATE	DOCUMENT	DATE	DOCUMENT	DATE
		S0010-02-0026A	10 / 10	R-619	/ 49		/
	A-2212	S0010-02-0026A	10 / 10	R-619	/ 49		31 /
		S0010-02-0026A	10 / 10	R-427	JAN. 65		32 /
		S0010-02-0026A	10 / 10		/		34 /
		S0610-02-0026A	10 / 10	R-619	/ 49		34 /
		S0010-02-0026A	10 / 10	R-414	DEC 64	QCR 47C142095	29 / 29
		S0010-02-0026A	10 / 10	R-619	/ 49		28 / 29
	A-2217	S0010-02-0026A	10 / 10	R-232	APR. 63		36 /
		S0010-02-0026A	10 / 10	R-579	JUN 66	QCR 47D172076	16 / 16
	A-2169	S0010-02-0026A	10 / 10	R-492	MAY 65		28 / 29
		S0010-02-0026A	10 / 10	R-619	/ 49		28 / 29
		S0010-02-0026A	10 / 10	R-619	/ 49		45 /
		S0010-02-0026A	10 / 10	R-619	/ 49		45 /
		S0010-02-0026A	10 / 10	R-619	/ 49	QCR 47D172070	29 / 29
	A-2091 A-2196	S0010-02-0026A	10 / 10	R-240	APR. 63		31 /
		S0010-02-0026A	10 / 10	R-233	APR. 63		36 /
		S0010-02-0026A	10 / 10	R-619	/ 49		33 /
		S0010-02-0026A	10 / 10	R-619	/ 49		34 /
		S0010-02-0026A	10 / 10				32 /
		S0010-02-0026A	10 / 10	R-619	/ 49		
				R-619	/ 49		
		S0010-02-0026A	10 / 10	R-619	/ 49		39 /
		S0010-02-0026A	10 / 10	R-380	SEP. 64		31 /
		S0010-02-0026A	10 / 10				45 /
		S0010-02-0026A	10 / 10				45 /

Figure 3-30. Reliability Mgmt. Matrix (Cont'd)

TOP LEVEL PROGRAM DOCUMENTATION	NUMBER	REVISION	ISSUE DATE

ISSUE DATE 31 JULY 1967

SYSTEM DEFINITION / MONITORING EVENT				FAILURE ANALYSIS & CORRECTIVE ACTION	
NOMENCLATURE	DRAWING NUMBER	SPECIFICATION NUMBER	SPEC. ISSUE	SIG. FAIL / TOT F...	FA...
MECHANICAL AGE DRAWING TREE	47D172409				
SLING, EQUIPMENT PACKAGE	47E190399	3940-11-0021	4		
STAND, EQUIPMENT PACKAGE	47E190396	3990-03-0016	4		
SHIPPING CONTAINER	47R187608	8110-02-0014	13		
SLING, POSITIONING	47E190397	3940-11-0022	4		
SLING, ROTATIONAL	47E190398	3940-11-0023	4		
PROTECTOR, NOSE	47E190393	8110-03-0005	7		
ASSEMBLY TOOL	47C1426[illegible]6	5120-76-0001	9		
MECHANICAL AGE-SPACECRAFT INTERFACE	47R197601				
MECHANICAL AGE-FIELD FLOW PLAN	47R196749				
ELECTRICAL AGE					
TEST SET, SPACECRAFT	47E191243	4935-04-0027	13		
SIMULATOR, SPACECRAFT	47C142354	4935-02-0018	12		
TEST SET, LOW LEVEL SENSOR SIMULATOR	47D172736	4935-04-0028	12		

3.6.3 Reliability Estimate Analysis

A reliability design analysis has been performed on all components of the Re-entry-F Spacecraft using data available on currently developed components of the system.

3.6.4 Failure Mode, Effect, and Criticality Analysis

A Failure Mode, Effect and Criticality Analysis (PIR 8444 REF 587 and PIR 8434-VR-539) has been performed utilizing the Reliability Design Analysis.

3.6.5 Design Preview Program

Two general types of Design Reviews are conducted: formal and informal. The formal review involves considerable advance preparation plus a design review meeting. For this type of review, a checklist is employed to assure that every aspect of the design is considered. The informal design review is usually conducted by a single design review engineer in conjunction with the design engineer. These "desk-side" reviews are conducted on relatively non-complex items and fractions of more complex designs.

3.6.6 Failure Analysis Board

Formal Failure Analyses are conducted on all significant and repetitive failures by a Failure Analysis Board. The Board members are the Reliability Engineer (Chairman), Design and Quality Control Engineers, and other specialists as required. The Failure Analysis Board reviews the history of the test failure, determines the mode and cause of failure, arrives at conclusions and makes recommendations, all of which are included in the Failure Analysis Report.

3.6.7 Standards Program

A Standards Activity provides for integrated technical decisions and mutually acceptable definition and documentation for all parts, materials, and processes selected as standards and a system for the rapid retrieval of standards information.

3.6.8 Configuration Management

The Configuration Management Program is directed to meet the specific needs of the Re-entry F Program in accordance with the Statement of Work.

The specific configuration elements to be managed are as follows:

a. Interface Definition

b. Design Definition

c. Design Changes

d. Non-conforming Material

e. Hardware Verification

These items are defined in more detail in the "Reliability Program Plan."

3.6.8.1 Configuration Control

All changes to and exceptions from approved drawings and specifications are integrated, reviewed, evaluated, and approved/disapproved by an authorized Design Change Board which consists of representatives from Engineering, Manufacturing, Quality Control, and Reliability and is chaired by a Program Configuration Control Engineer representing the Program Manager.

The Design Change Board reviews all proposed changes for the following:

a. Compliance with overall program direction.

b. Completeness and adequacy of change integration.

c. Proper change introduction point and disposition of existing material, if any.

d. Accuracy, clarity, completeness, and understandability of the change.

3.6.9 Parts and Material Program

The component parts program applicable to the Re-entry-F program includes the following elements:

a. The selection of those component parts currently available which are best suited to the successful design of the equipment.

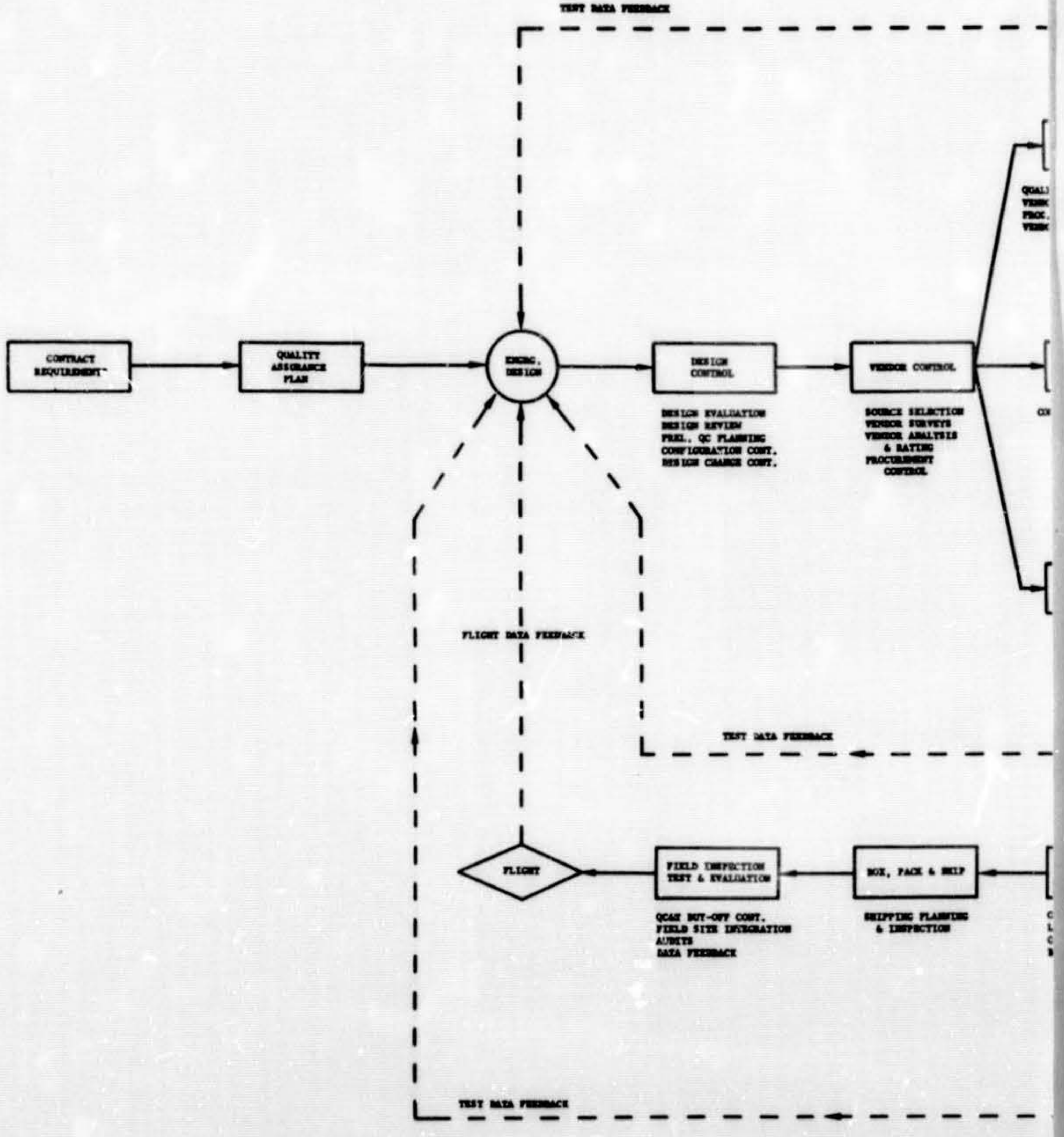
b. The development, dissemination and implementation of monitoring techniques to assure the proper application of component parts with respect to optimum usage, derating, and reliability.

c. The development and issuance of part drawings to assure proper definition of parts.

d. The analysis of failed parts to determine failure causes and to provide feedback information.

3.7 Quality Assurance

The Quality Assurance program is described in GE document No. 66SD797, "Quality Assurance Program Plan for Re-entry F Turbulent Heating Experiment." It is in compliance with the NASA definition of quality assurance as a planned and systematic pattern (Figure 3-31) of all actions necessary to provide adequate confidence that end items will perform satisfactorily in actual operations.


TEST DATA FEEDBACK
CONTRACT REQUIREMENTS
QUALITY ASSURANCE PLAN
ENGRG. DESIGN
DESIGN CONTROL
DESIGN EVALUATION
DESIGN REVIEW
PREL. QC PLANNING
CONFIGURATION CONT.
DESIGN CHANGE CONT.
VENDOR CONTROL
SOURCE SELECTION
VENDOR SURVEYS
VENDOR ANALYSIS & RATING
PROCUREMENT CONTROL
FLIGHT DATA FEEDBACK
TEST DATA FEEDBACK
FLIGHT
FIELD INSPECTION TEST & EVALUATION
QCAS BUY-OFF CONT.
FIELD SITE INTEGRATION
AUDITS
DATA FEEDBACK
BOX, PACK & SHIP
SHIPPING PLANNING & INSPECTION
TEST DATA FEEDBACK

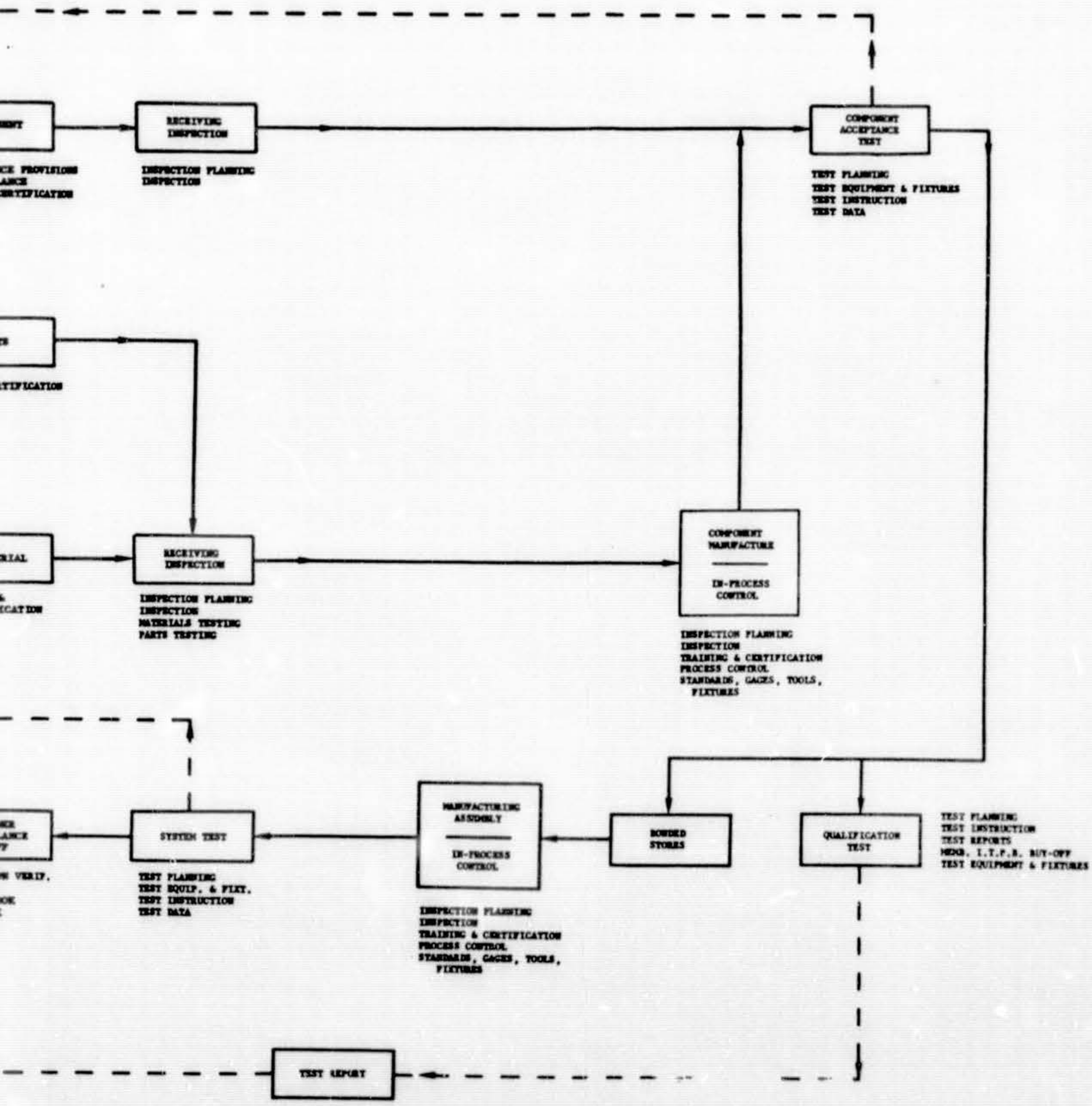


Figure 3-31. Quality Control and Test Work Flow

Objectives of the plan are:

a. To provide a quality control system that assures the conformance of supplies and services with contractual requirements as interpreted by the Statement of Work L-6345-A. The Re-entry F Quality Program will satisfy the basic requirements of NPC 250-1, NPC 200-3, and NPC 200-4 as modified by Appendix A of the Statement of Work, and Waiver No. 1 as granted.

b. To collect and analyze data to derive information concerning prevention, appraisal, corrective action, scrap, rework, failures, and training. Periodic reviews and independent audits will be performed to generate and verify the data forming bases for decisions and control over policies, procedures, organization, staffing, training, processes, records, and products as they apply to, or are influenced by, quality program activities.

3.7.1 Functions

The Quality Assurance program has the following functions:

a. Control of quality program changes

b. Drawing and specification review

c. Control of procured material

d. Inspection of Government-furnished property

e. In-process control of "Make" items and of final assembly

f. Control of nonconforming material

g. Control of measuring equipment and test equipment

h. Statistical planning, analysis, and quality control

i. Quality data management and audits

j. Materials quality control and test

k. Component and System acceptance and test

l. Field test planning and support

m. Quality-information equipment provisioning and maintenance

n. Detailed quality assurance and test planning

Figure 3-32 is a test flow diagram for a Re-entry F Spacecraft. Figure 3-33 shows the work flow plan for materials quality control acceptance tests. Figure 3-34 is a typical procedure and test flow diagram for component quality control. Figure 3-35 is a summary of Re-entry F hardware and procedural flow.

The work/hardware flow plan for the Quality Assurance program can be seen in more detail in the Quality Assurance Plan Document Number 66SD979.

3.7.2 Operational Procedures

Procedures for performing the tests summarized in section 3.8 of this manual are described in detail in the following documents: "Integrated Test Plan for Re-entry F Turbulent Heating Experiment", Document Number 67SD961 and, "Systems Acceptance Plan for Re-entry F Turbulent Heating Experiment" Document Number 67SD781, and in the appropriate SI's and QAP's (Standing Instructions and Quality Assurance Procedures), which are issued for the purposes as described below:

3.7.2.1 Quality Assurance Procedures

Quality Assurance Procedures (QAP) are used for purchased components and materials, and provide details of vendor/subcontract or quality/reliability requirements. QAP's are issued under one or more of the following circumstances:

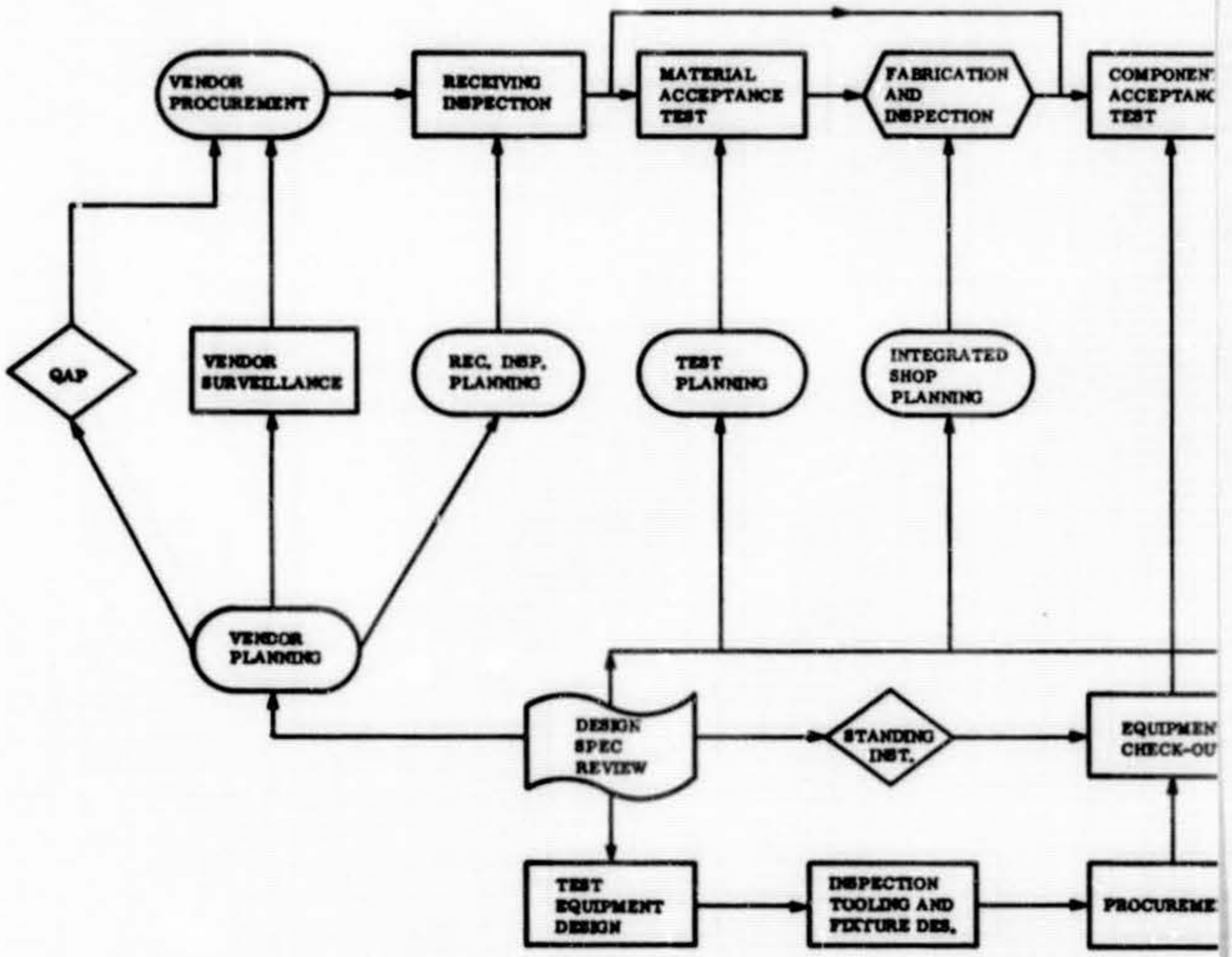
a. The applicable specification or purchase order is inadequate, and additional testing, controls, or data are required.

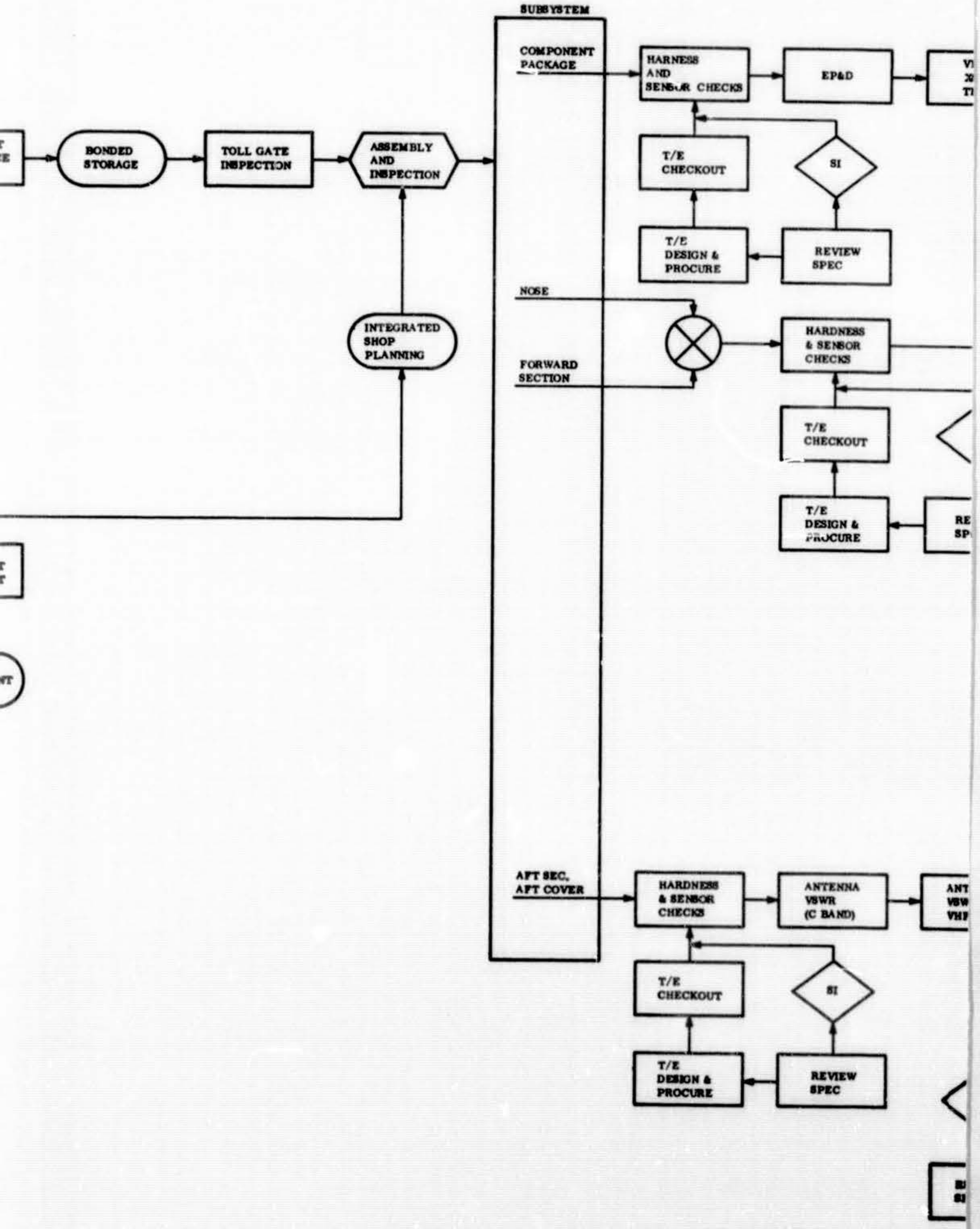
b. Operability assurance tests are performed at the vendor's facility.

c. The purchased item is shipped direct to the field.

d. Additional tests cannot be performed in house.

QAP's are submitted to NASA/Langley for review in accordance with the terms of the contract and NPC 200-2. QAP's issued for the Re-entry F program are listed under CDRL Sequence No. 046 in section 4.2 of this document.


VENDOR PROCUREMENT
RECEIVING INSPECTION
MATERIAL ACCEPTANCE TEST
FABRICATION AND INSPECTION
COMPONENT ACCEPTANC TEST
QAP
VENDOR SURVEILLANCE
REC. INSP. PLANNING
TEST PLANNING
INTEGRATED SHOP PLANNING
VENDOR PLANNING
DESIGN SPEC REVIEW
STANDING INST.
EQUIPMEN CHECK-OU
TEST EQUIPMENT DESIGN
INSPECTION TOOLING AND FIXTURE DES.
PROCUREME



BONDED STORAGE
TOLL GATE INSPECTION
ASSEMBLY AND INSPECTION
INTEGRATED SHOP PLANNING
SUBSYSTEM
COMPONENT PACKAGE
HARNESS AND SENSOR CHECKS
EP&D
T/E CHECKOUT
SI
T/E DESIGN & PROCURE
REVIEW SPEC
NOSE
FORWARD SECTION
HARDNESS & SENSOR CHECKS
T/E CHECKOUT
T/E DESIGN & PROCURE
AFT SEC, AFT COVER
HARDNESS & SENSOR CHECKS
ANTENNA VSWR (C BAND)
T/E CHECKOUT
SI
T/E DESIGN & PROCURE
REVIEW SPEC

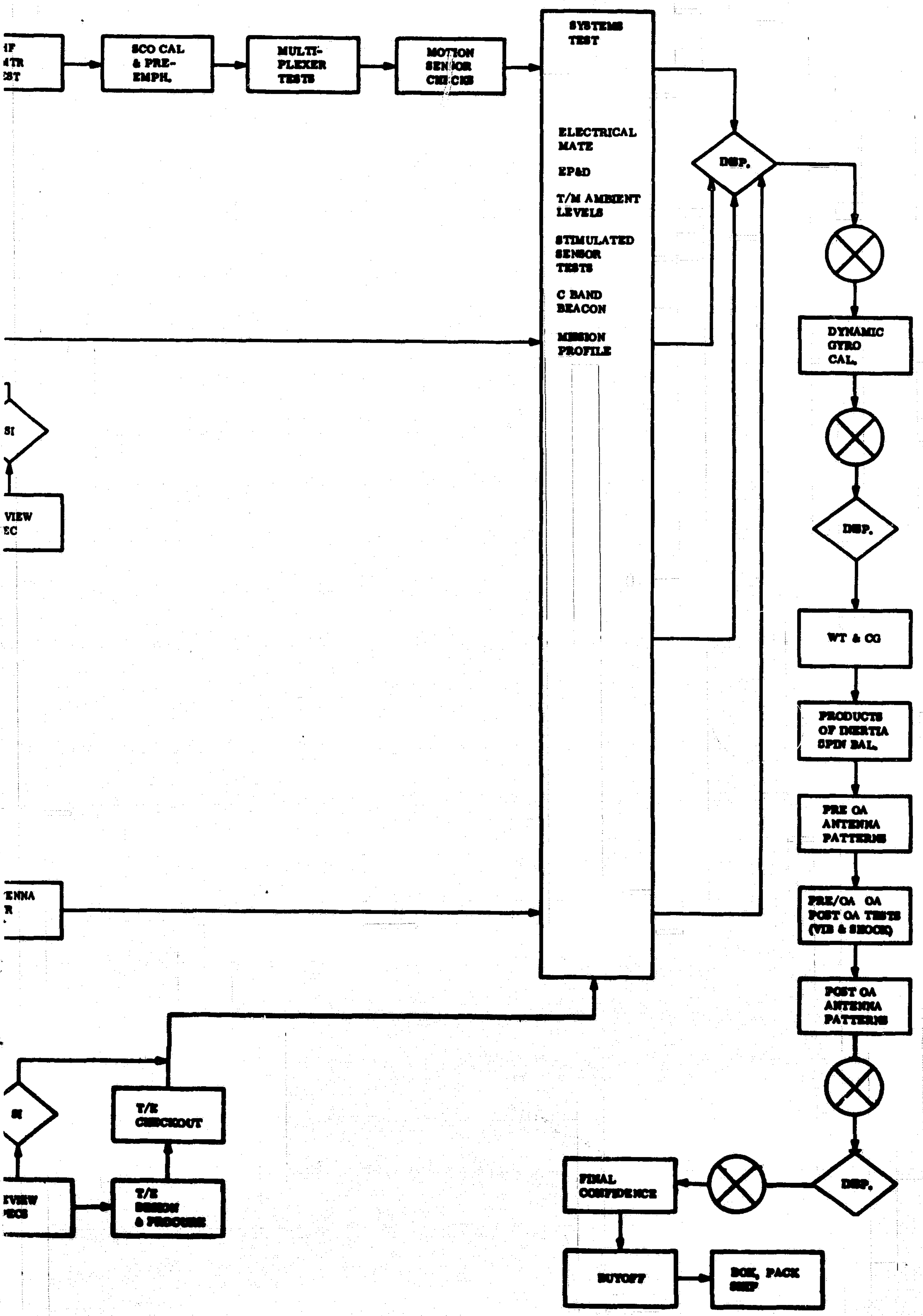


Figure 3-32. Re-entry F Typical Vehicle Test Flow Diagram

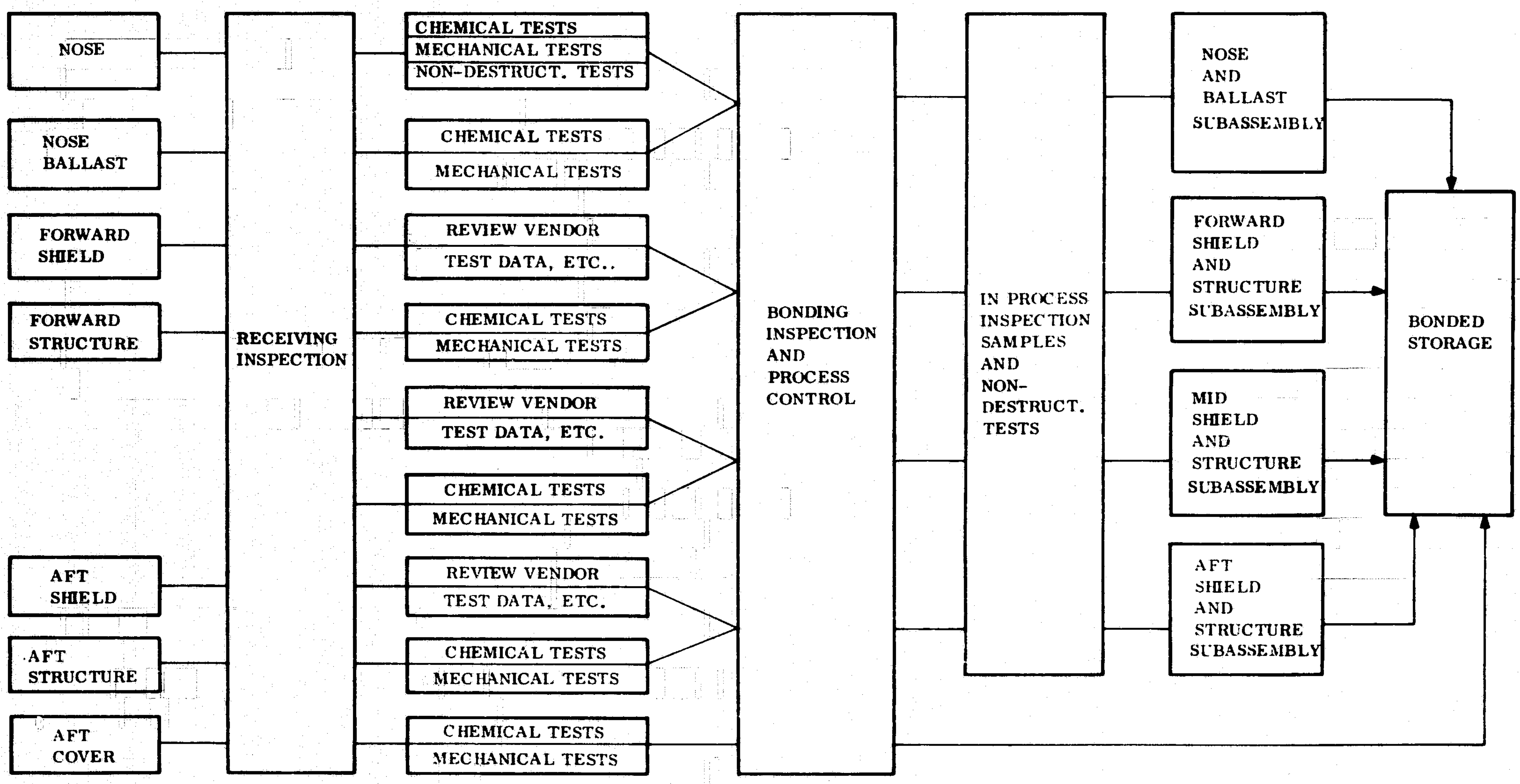


Figure 3-33. Materials Quality Control Acceptance Tests, Process Control and In-Process In-Process Inspection Work-Flow Plan

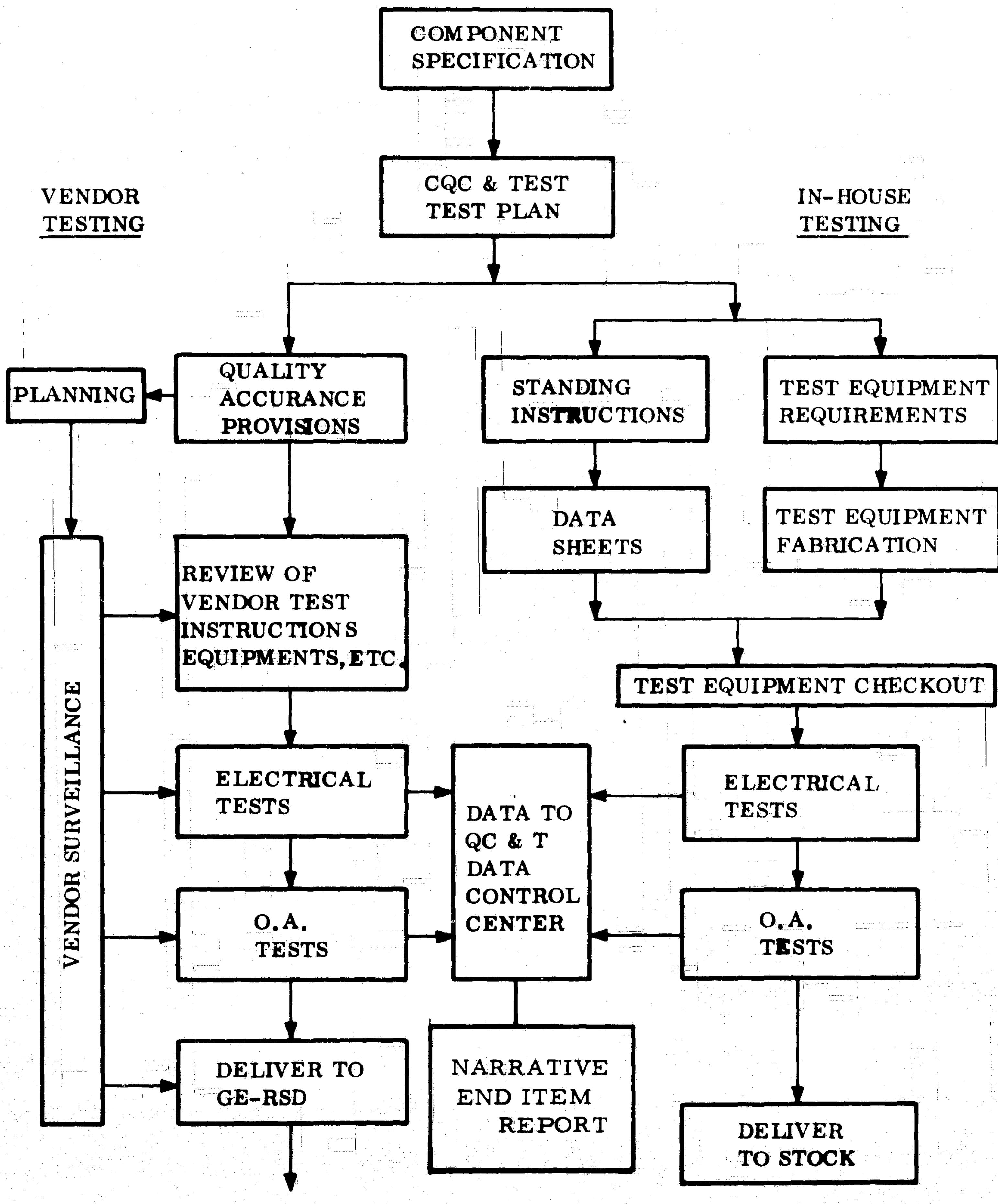


Figure 3-34. Typical Component Quality Control and Test Procedures and Test Flow

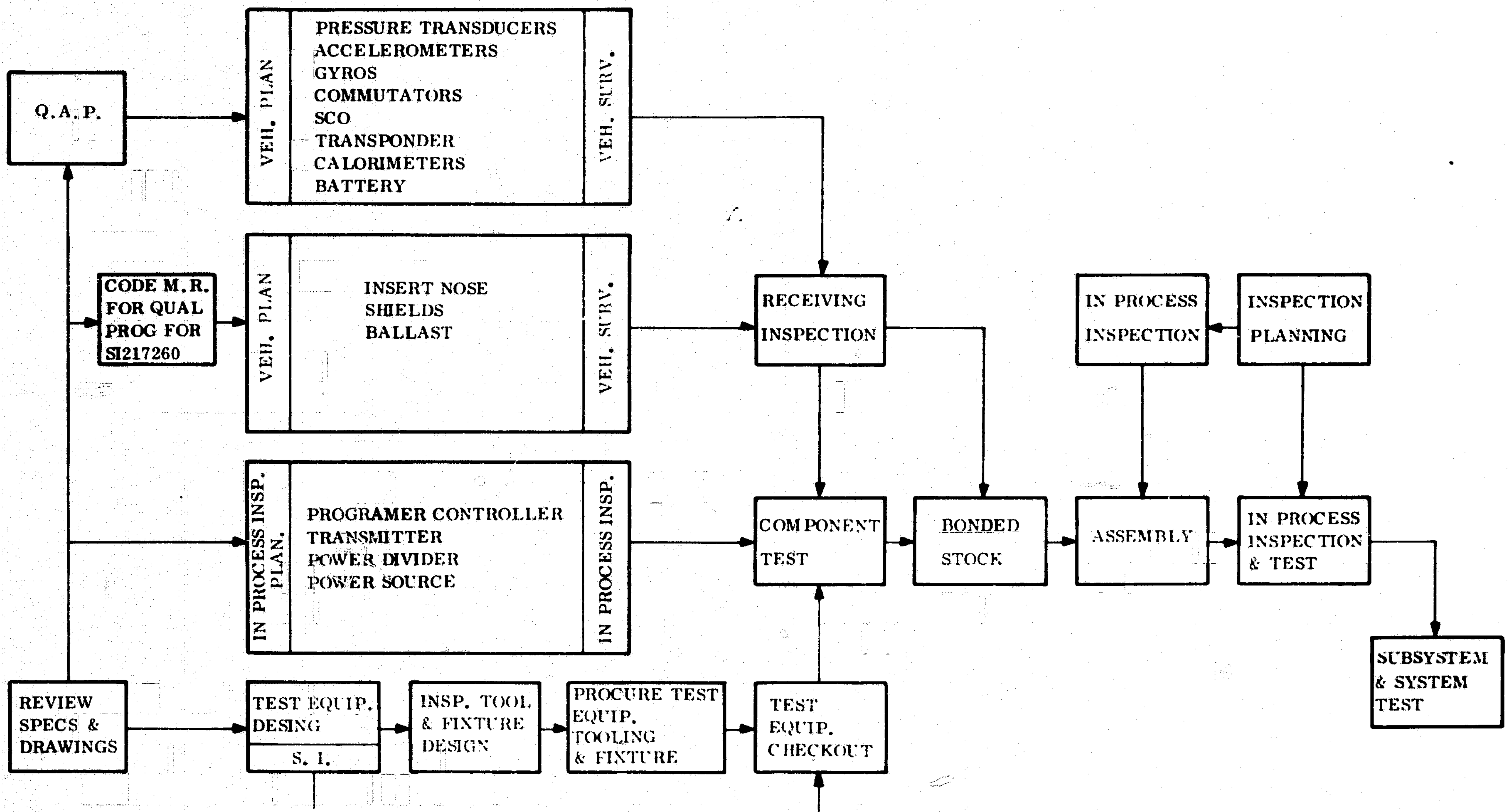


Figure 3-35. Summary Re-entry F Hardware and Procedural Flow

3.7.2.2 Standing Instructions

Standing Instructions (SI) provide detailed information and procedures for inspection or test, as applicable. Standing Instructions (SI) cover procedures for performance of Acceptance Tests and Field Tests, and include the following:

a. Identification of the article to be tested (nomenclature, serial no., Drawing No., etc.)

b. Test Objectives

c. Test and measuring equipment to be used including

   (1) type, range, and accuracy

   (2) scale, dial and device to be observed

   (3) for recording type equipment, the details of tape, chart paper, sensitized paper or film to be used

d. Detailed operations to be performed by the test operator including operational checks or preliminary calibration of test setup

e. Exact method of measuring including necessary manipulation of measuring including necessary manipulation of controls both on the test item and on the measuring and test equipment

f. Laboratory conditions that must be maintained during test including ambient or environmental conditions, and precautions to be observed to prevent damage to the item under test or instruments involved.

g. Test data sheets for recording parameters measured

h. Criteria for passing or failing the test. Each parameter to be measured shall be defined and acceptable tolerance conditions specified on the test data sheets.

3.7.3 Responsibilities

Work elements to be performed by the GE-RSD Quality Control and Test Section are apportioned to the following responsible subsections:

a. Research re-entry vehicle engineering and evaluation

b. Product quality assurance

c. Quality systems engineering

d. Component quality control and test

e. Quality information equipment engineering

f. Quality control and test field integration support

g. System quality control and test

h. Materials quality control and test

The responsibilities for subsection "component quality control and test" are further subdivided to component quality control engineering, component acceptance and environmental testing, vendor surveillance, inspection, and in-process control. The responsibilities for subsection "System Quality Control and Test" are further subdivided to systems quality control engineering, and to system quality reports. The subsection "Materials Quality Control and Test" is comprised of quality control engineering, material acceptance testing and inspection, and failure analysis.

3.8 Testing Program

The testing program consists of an Integrated Test Plan, a Systems Acceptance Test Plan, and a number of component tests.

3.8.1 Integrated Test Plan

The Integrated Test Plan is described in GE Document No. 66SD961. This plan is designed to demonstrate the ability of the spacecraft to complete its mission successfully.

The objective of the plan is the establishment of a formalized, technically balanced program to:

a. Establish the capability and compatibility of the spacecraft components, subsystems and system.

b. Provide a high degree of confidence that the spacecraft will survive the environments imposed during the mission.

c. Determine spacecraft operating and performance characteristics during simulated mission environmental testing.

d. Assure that all tests are planned, conducted, and reported in a sound technical manner consistent with NASA Langley Research Center and GE-RSD objectives, concepts, policies and procedure.

The program provides structural, component, subsystem, and system tests to provide data for design evaluation and qualification of flight hardware, to provide acceptance testing and control of the deliverable hardware, and to support the flight program by providing prelaunch spacecraft and launch compatibility checkouts.

3.8.1.1 Summary Description

The Integrated Test Plan consists of four major categories:

a. Development Tests

b. Special Prototype Spacecraft and Qualification Tests

c. Flight and Backup Spacecraft Tests

d. Component Tests

3.8.1.1.1 Development Tests - The intent of Development Tests is to obtain design data, verify design concepts, and to confirm final design criteria for the spacecraft and its associated support equipment. The data resulting from these tests are used to establish acceptance criteria and tolerances, subsystem and system compatibility, and assessment of performance in the environments.

Many of the components for the Re-entry F Spacecraft have been developed by GE-RSD on similar re-entry programs. Figure 3-36 shows the Development Testing flow.

There are six types of Development Tests:

a. Structural Component and Element Tests

b. Antenna Development and Subscale Antenna Patterns

c. Nose Development Tests

d. Thermal Sensor Development Tests

e. Electrical System Development Tests

f. Engineering Development Model (E. D. M.) Tests

These tests are described in GE-RSD Document No. 66SD961.

3.8.1.1.2 Special Prototype Spacecraft and Qualification Tests - A prototype spacecraft of the flight configuration is subjected to a comprehensive test program designed to demonstrate the successful performance of the spacecraft system under all modes of operation. The prototype spacecraft undergoes essentially four types of testing:

a. System Acceptance Tests

b. Special Design Verification Tests

c. System Qualification Tests

d. Launch Vehicle Compatibility Tests

The first three tests are to be performed at a facility of the General Electric Re-entry Systems Department and, after completion, the spacecraft is shipped to the launch-vehicle contractor for the fourth test. Figure 3-37 shows the prototype-spacecraft test flow.

3.8.1.1.3 Flight and Backup Spacecraft Tests - The flight and backup spacecraft undergo system and subsystem acceptance testing including flight assurance tests of vibration, shock, and temperature/ altitude (See Figure 3-38). The dynamic flight assurance tests are conducted to the anticipated flight levels. Prior to and upon completion of the Flight Assurance Testing, antenna patterns are to be taken to verify that the flight assurance environments have not degraded the antenna patterns of the flight and backup spacecrafts.

There are two types of Flight and Backup Spacecraft Tests:

a. Acceptance Testing

b. Flight Unit Launch Support Tests

Figure 3-39 shows the System Acceptance Test Flow, and Figure 3-40 shows the Flight Spacecraft Field Flow.

3.8.1.1.4 Component Tests - All components are required to pass a component acceptance test consisting of an initial bench test to check operation and performance, and then exposure to the specified environments. Levels of exposure are specified in the applicable component specification which is based on the Internal Environmental Specification S0020-02-0020.

In some cases, the acceptance test is performed at the vendor facilities. In these cases, a test instruction is written as a part of the Quality Assurance Provision.

3.8.2 Systems Acceptance Test Plan

The Systems Acceptance Test Plan is described in GE Document No. 67SD781. The plan includes systems acceptance testing of the Prototype Spacecraft, the Flight Spacecraft, and associated Aerospace Ground Equipment (AGE) as follows:

a. Technical description of the Spacecraft

b. Test/Hardware Flow Diagrams describing the test sequence

c. A detailed description of the Acceptance and Flight Assurance tests to be performed on the Spacecraft, its Subsystems, and associated AGE.

d. A description of the Field Checkout and Pre-Launch tests to be accomplished at the Launch Site Facility

e. The requirements for Test Documentation, Standing Instructions, and Data Reporting

f. A summary of the facilities, ground support equipment, and fixture to support the test program.

3.8.2.1 Test Objectives and Reference Documents

a. Verify through systematic tests that the end item S/C meets specified performance.

b. Demonstrate compatibility between spacecraft subsystems.

c. Demonstrate compatibility between spacecraft and associated AGE.

d. Provide quantitative test data for the spacecraft logbook and calibration book.

The specified requirements (item a) are defined in the System and Subsystem Specifications listed in CDRL Sequence No. 039 of Section 4.2.

3.8.2.2 Test Sequence and Test/Hardware Flow Diagrams

Acceptance testing begins with acceptance tests of Electrical and Mechanical Aerospace Ground Equipment (AGE), followed by subsystem testing of the major spacecraft assemblies. These subsystems tests will verify the performance and completeness of each major assembly before proceeding with final assembly of the spacecraft for systems level testing. Acceptance tests at the systems level will include Mission Profile tests, Flight Assurance tests, Mass Properties tests and Spacecraft Confidence tests.

A flow diagram of the End Item Systems Acceptance Test Plan (Figure 3-39) defines the test sequence and summarizes each acceptance test to be performed on the prototype and flight spacecrafts.

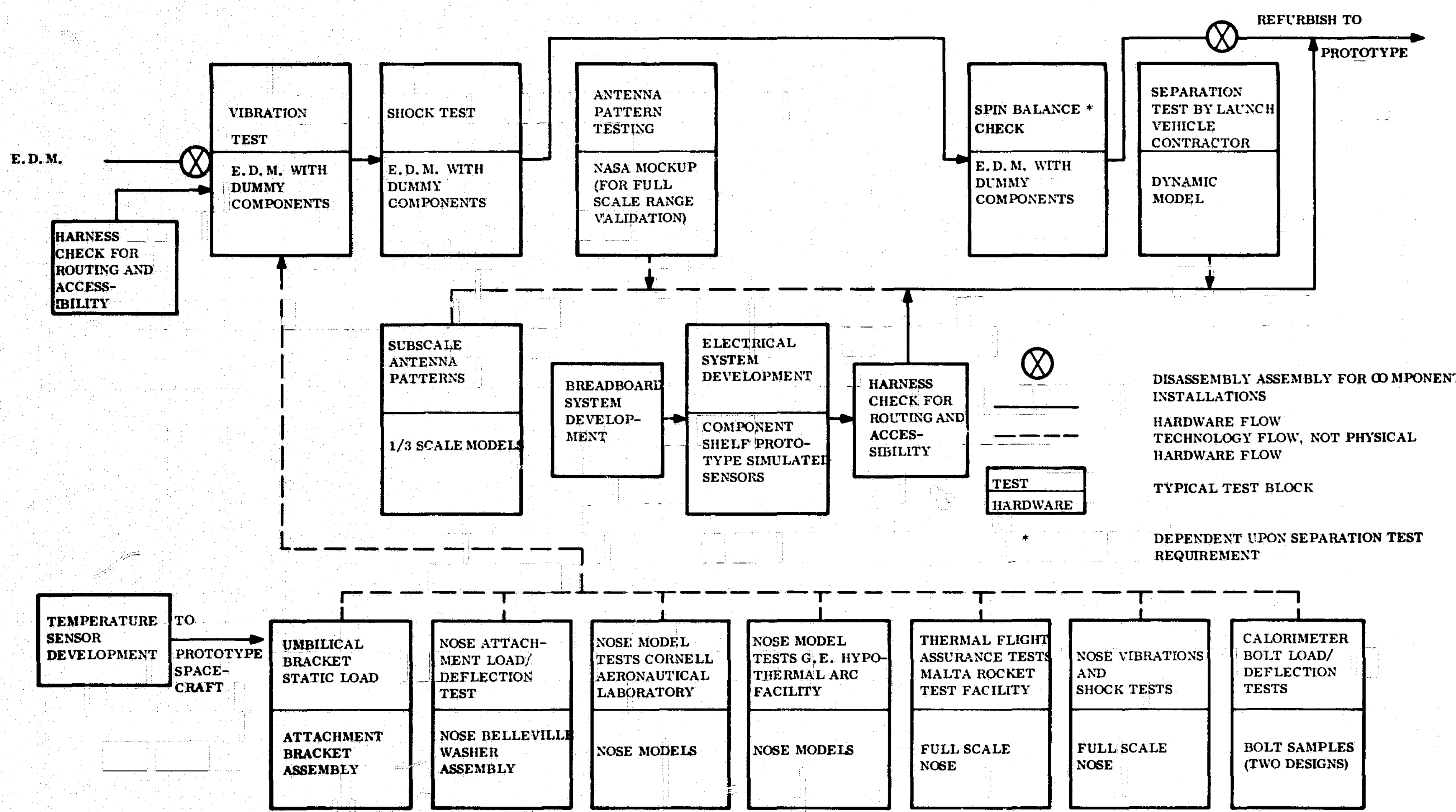


Figure 3-36 Development Test Flow

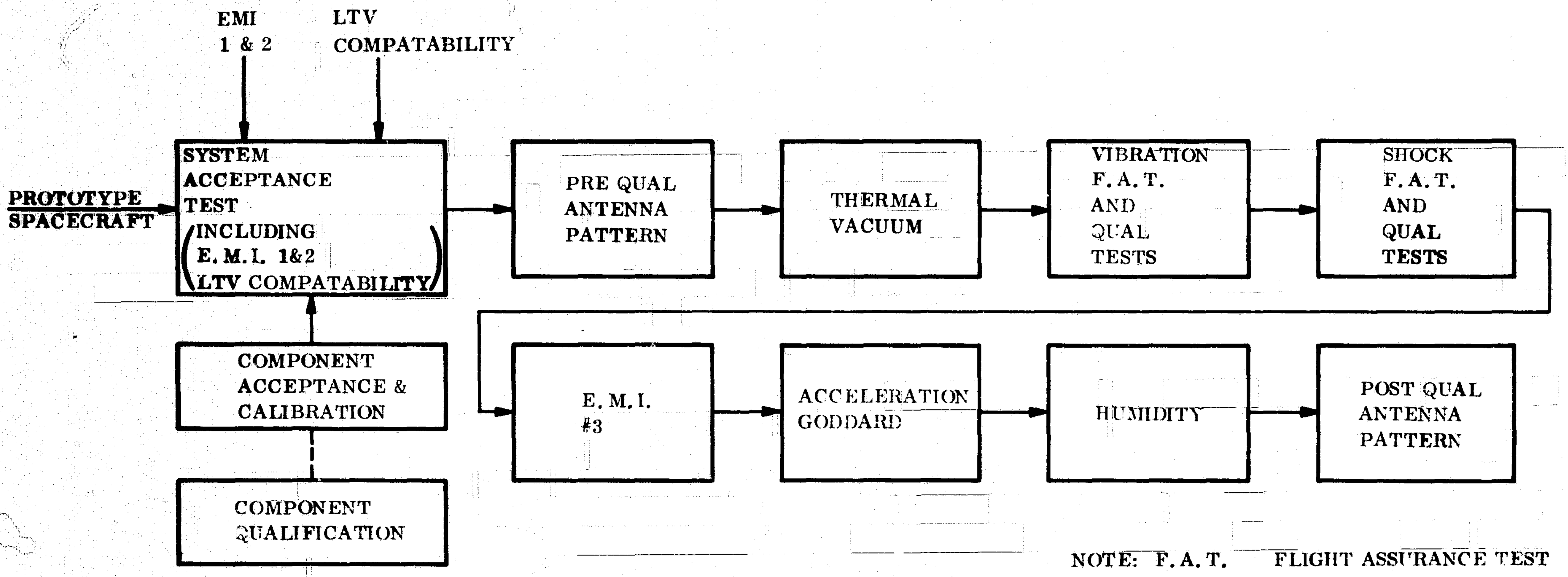


Figure 3-37. Prototype Spacecraft Test Flow

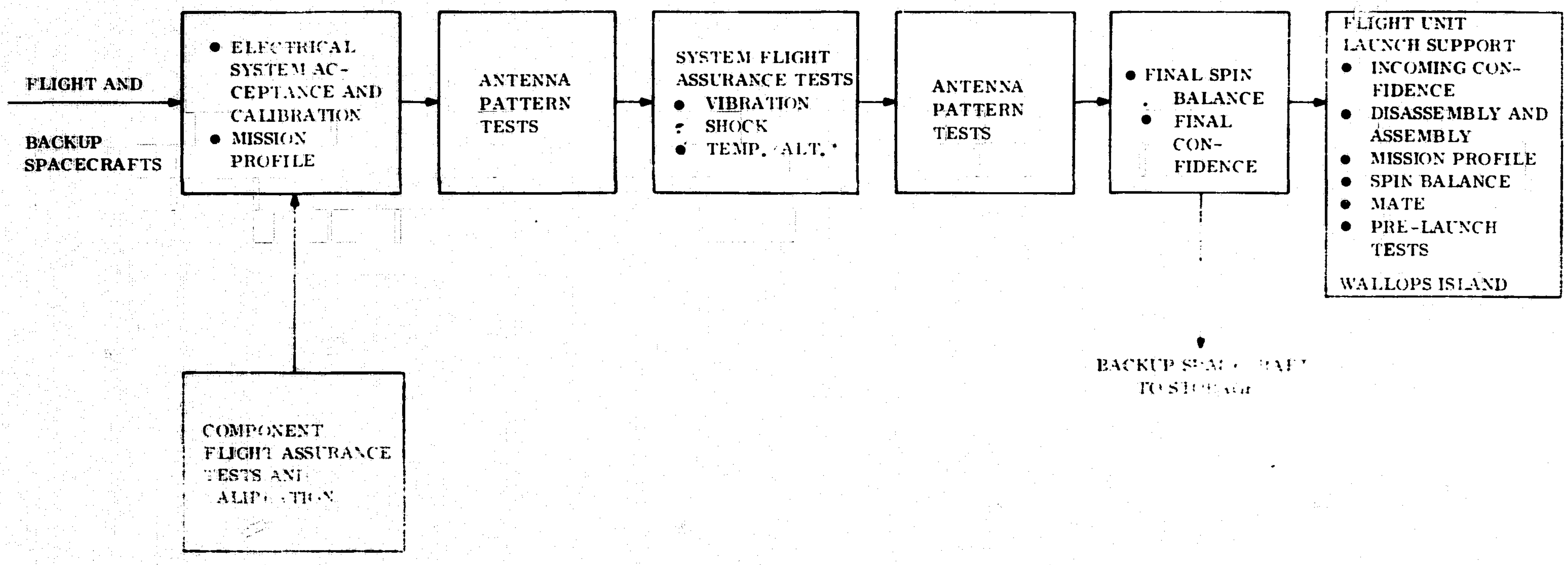
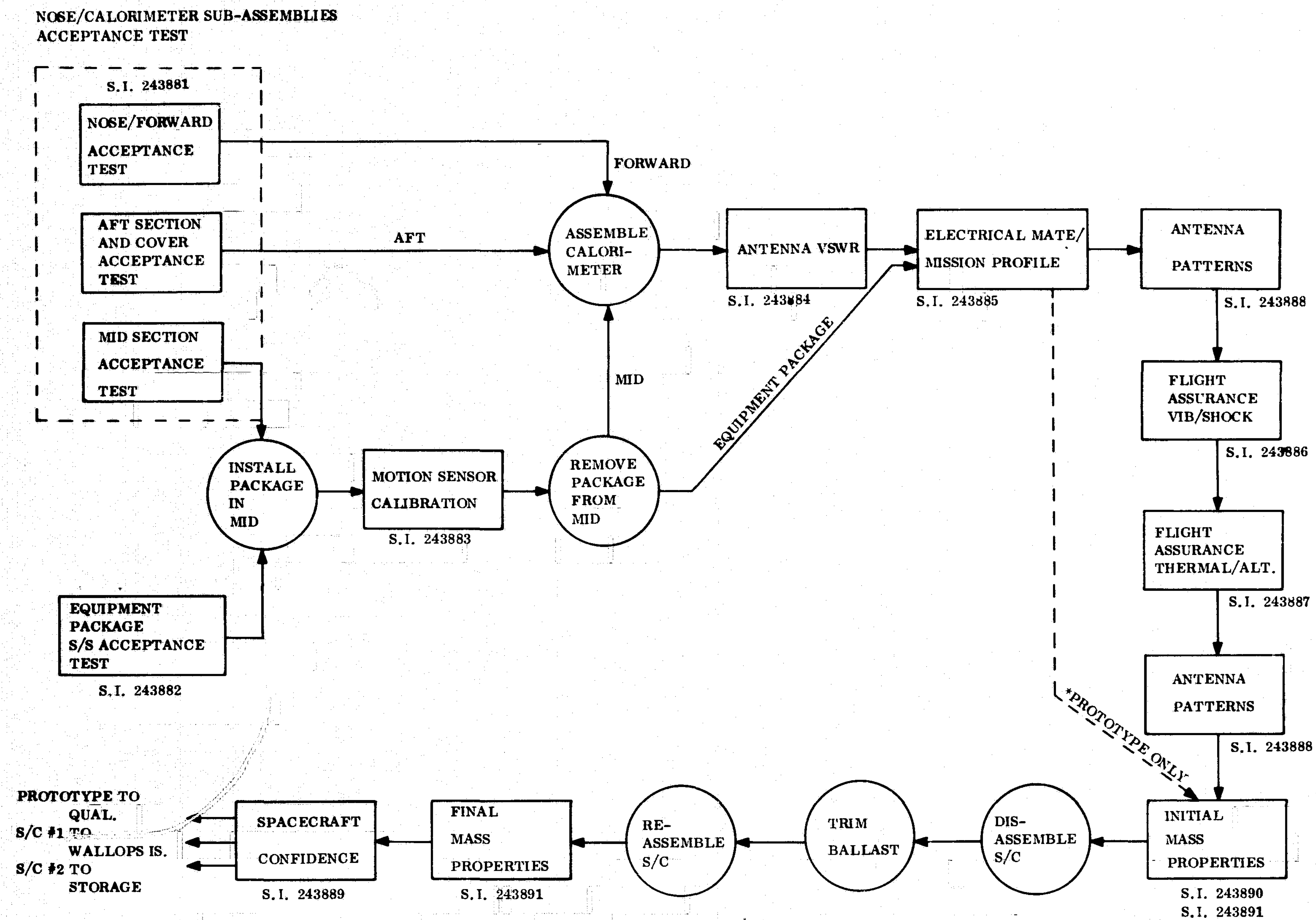


Figure 3-38. Flight and Backup Spacecraft Test Flow



*PROTOTYPE FAT TESTING TO BE PART OF QUALIFICATION TEST CYCLE.

Figure 3-39. System Acceptance Test Flow

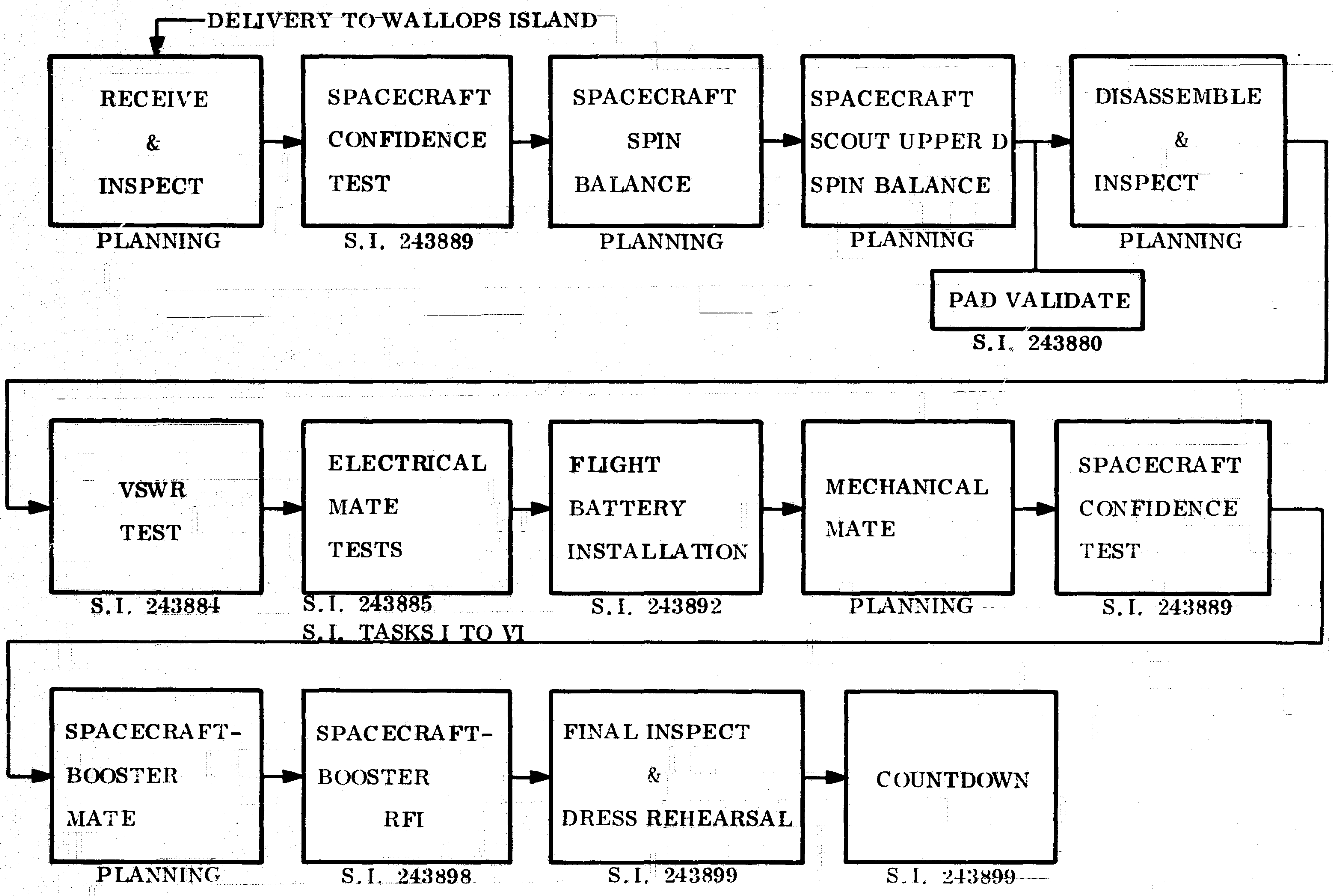


Figure 3-40. Flight Spacecraft Field Flow

3.8.2.3 Acceptance Tests

Subsystems and System Acceptance Tests are performed on the Spacecraft, its subsystems, and associated AGE to verify that the Spacecraft meets the minimum performance criteria specified in Systems Acceptance Test Specification S0040-01-0035. All tests are conducted and performed in accordance with the applicable Standing Instructions (SI). These tests include:

a. AGE Acceptance Tests

b. Nose/Calorimeter Subassemblies Acceptance Tests

c. Instrument Package Subsystem Acceptance Tests

d. Motion Sensor Calibration

e. Voltage Standing-Wave Ration (VHF and C-Band) Test

f. Electrical Mate/Mission Profile Tests

g. Antenna Patterns Test, Pre-flight and Post-flight Assurance

h. Flight Assurance Vibration Shock Test

i. Flight Assurance Thermal/Altitude Test

j. Surface Mapping (Banana) Test

k. Mass Properties (Weight and Balance)

3.8.2.4 Field Tests

The spacecraft and AGE, upon delivery to the Field site, will be subjected to tests and inspections in order to ensure the following:

a. No shipping damage has occurred.

b. The Spacecraft is electrically and mechanically compatible with the booster.

c. Final calibrations are performed as close to the flight date as possible and to verify the S/C calibration book.

d. The S/C is flightworthy at the time of launching.

e. The S/C AGE is compatible with the blockhouse facilities.

f. The flight batteries have been properly activated, tested, and installed.

g. The S/C and Scout upper "D" section are properly balanced dynamically.

Many of the electrical tests will be identical and run to the same Standing Instructions used during factory acceptance.

Following are summary descriptions of field organization and responsibilities, field flow, and field testing.

3.8.2.4.1 Field Organization and Responsibilities - In general, the field activities will be conducted as an extension of the in-house systems assembly and test function with the addition of a Field Support and Logistics task. The organization is shown in Figure 3-41 and Table 3-13.

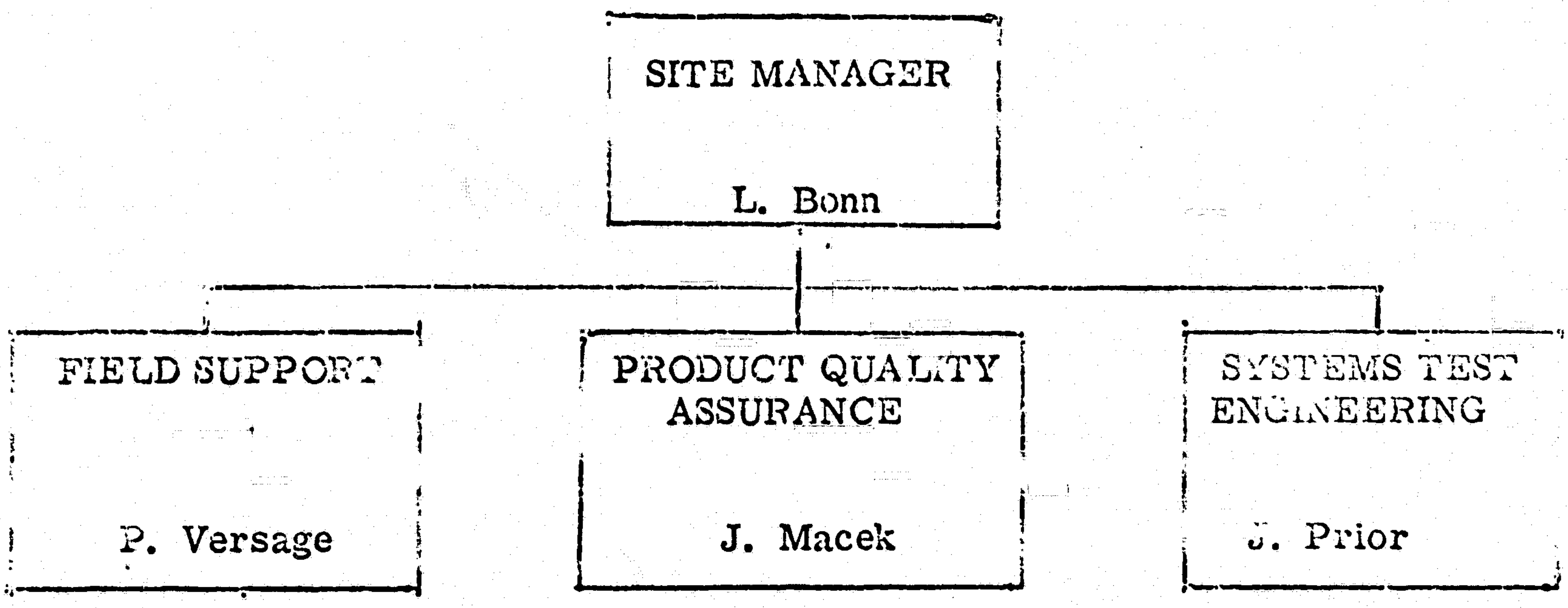


Figure 3-41. Re-entry F Field Organization

3.8.2.4.2 Field Flow - The Spacecraft processing in the Field is described in Section 3.11. It is closely tied to the booster processing because of several mandatory tests requiring booster/spacecraft combinations. The booster schedule dictates the order of testing; i.e., spin balance must be performed before the booster assembly can be completed and the booster/spacecraft EMI test must be performed before going to the pad.

3.8.2.5 Test Documentation - Test documentation includes standing instructions, test data, a summary test report, and the Spacecraft TM Calibration Book.

TABLE 3-13. RE-ENTRY F FIELD ORGANIZATION

Unit	Responsibility
Site Manager	Customer and program interface, definition, and direction of all Field activities.
Field Support	All details of transport, quartering, and support of Field Site. Spares provisioning, facilities communication, station reporting, etc. Checkout of Field Site telemetry van.
Product Quality Assurance	Inspection, inspection planning, test monitoring. Maintenance of Re-entry Systems quality system.
Systems Test Engineering	All testing activities at Field Site, including provision of test SI's and data, integration with Customer and Co-contractor technical personnel as required, status reporting and acting as GE Test Conductor during countdown and launch.

3.8.2.5.1 Standing Instructions - Those SI's covering Systems testing are of the 243,800 series and are listed in Section 4.2.

3.8.2.6 Facilities, Ground Support Equipment, and Fixtures.

3.8.2.6.1 Facilities - Facilities include an Electrical Systems Test Laboratory and five mechanical test facilities: a Rate Table Facility, a Vibration Test Laboratory, a Temperature-Altitude Facility, a Mass Properties Test Facility, and a Mass Properties Measuring Facility.

3.8.2.6.2 Ground Support Equipment - Ground support equipment includes Aerospace Ground Equipment, test-support equipment, and central ground station equipment. The central ground station equipment includes five sections: RF, FM, PDM, display, and monitor.

3.9 INTERFACES

The following paragraphs provide summary descriptions of the interfaces between the spacecraft and the aerospace ground equipment, launch vehicle, and launcher, and includes summaries of requirements and procedures for spin balance and assembly.

3.9.1 Spacecraft to Aerospace Ground Equipment

There are three major interface configurations between the spacecraft and the Aerospace Ground Equipment (AGE): one with the mechanical AGE and two with the electrical AGE. The electrical AGE interfaces are for subsystem and system-level testing at the launch site.

The AGE is defined and described in subsection 3.10 of this manual. Use of the mechanical AGE is described in GE/RSD Document No. 67SD649A (Revised 8 August 1967), Turbulent Heating Experiment Program/Mechanical Aerospace Ground Equipment/Operating Instruction Manual. A technical description of the electrical AGE is provided in GE/RSD Document No. 67SD650, Technical Description/ Re-entry F AGE.

Use of the AGE is summarized in section 3.8 of this manual, and described in GE/RSD Document No. 67SD781, Systems Acceptance Test Plan for Re-entry F Turbulent Heating Experiment. Test procedures are covered in Standing Instructions (S.I.) listed in Section 3.8 and GE/RSD Document No. 67SD781.

3.9.2 Spacecraft to Launch Vehicle

The interface of the spacecraft to the Scout launch vehicle (booster) is designed to meet the requirements of the Scout User's Manual and the Payload Description Document.

There is no electrical interface between the spacecraft and the Scout.

The mechanical interface is between the interface structure of the spacecraft (subsection 3.4) and transition section "D" of the Scout. The interface structure is the aftmost 12 inches of the aft section of the spacecraft, and includes an interface joint. Transition section "D" is at the forward end of the third stage of the Scout. An interface ring supplied by the launch vehicle contractor mates the spacecraft to a large-diameter spin bearing attached to the forward end of transition section "D".

The spin bearing is part of the fourth-stage spinup system, and includes a spring arrangement as part of the ejection separation system.

The fourth-stage spinup system includes two spinup motors which provide clockwise spin stabilization to the spacecraft just after third stage burnout during boosted re-entry before separation.

3.9.3 Spacecraft to Launcher

The spacecraft is connected to the Mark II Scout launcher, a utility building, and the blockhouse via an umbilical cable. The Spacecraft is equipped with a standard Bendix pigmy-type (PT) connector located at the aft end of the spacecraft. This connector is mated to a Bendix twist/pull fly-a-way plug connector (Bendix twist/pull-type connector No. 72-3395-12).

Additional information may be found in the Payload Description Document and the Scout User's Manual.

3.9.4 Spin Balance

Spin balance, or dynamic balancing, is performed according to NASA supplied documentation, to ensure that the spacecraft and the Scout launch vehicle are aligned so that the combination of all masses atop the spin table on transition section "D" of the Scout are balanced. Additional information is included in the Payload Description Document.

Dynamic balancing equipment at the Wallops Station includes two vertical balancing machines: Gisholt and Trebel. These machines are located at the Dynamic Balancing Facility Figure 3-42. The Gisholt machine (Figure 3-43) is located in Test Building No. 1, and the Trebel machine (Figure 3-44) is located in Test Building No. 2.

The Dynamic Balancing Facility is described in the Scout User's Manual. Mechanical AGE are used in support of spin balance, as described in GE/RSD Document No. 67SD649A (Revised 8 August 1967), Turbulent Heating Experiment Program/Mechanical Aerospace Ground Equipment/Operating Instruction Manual.

3.9.5 Assembly

The spacecraft is assembled in a Payload Work Facility and mated to the Scout launch vehicle in Assembly Building No. 3. General information is included in the Scout User's Manual. Specific information may be found in the Payload Description Document and Wallops Station Handbook.

3.9.6 Sequence of Wallops Operations

The operational activities that take place during the checkout and launch phase of Re-entry F are shown in section 3.11.

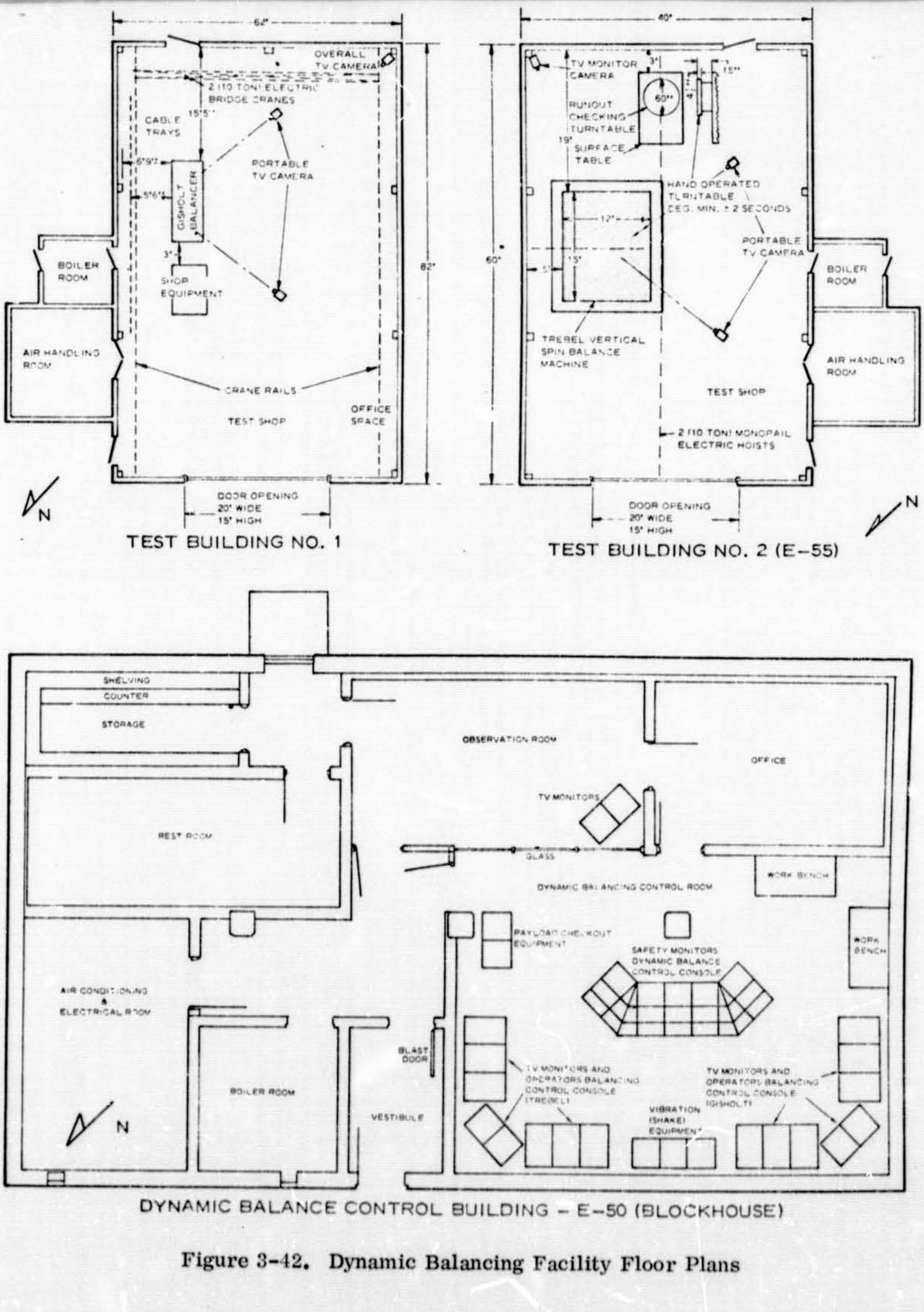


Figure 3-42. Dynamic Balancing Facility Floor Plans

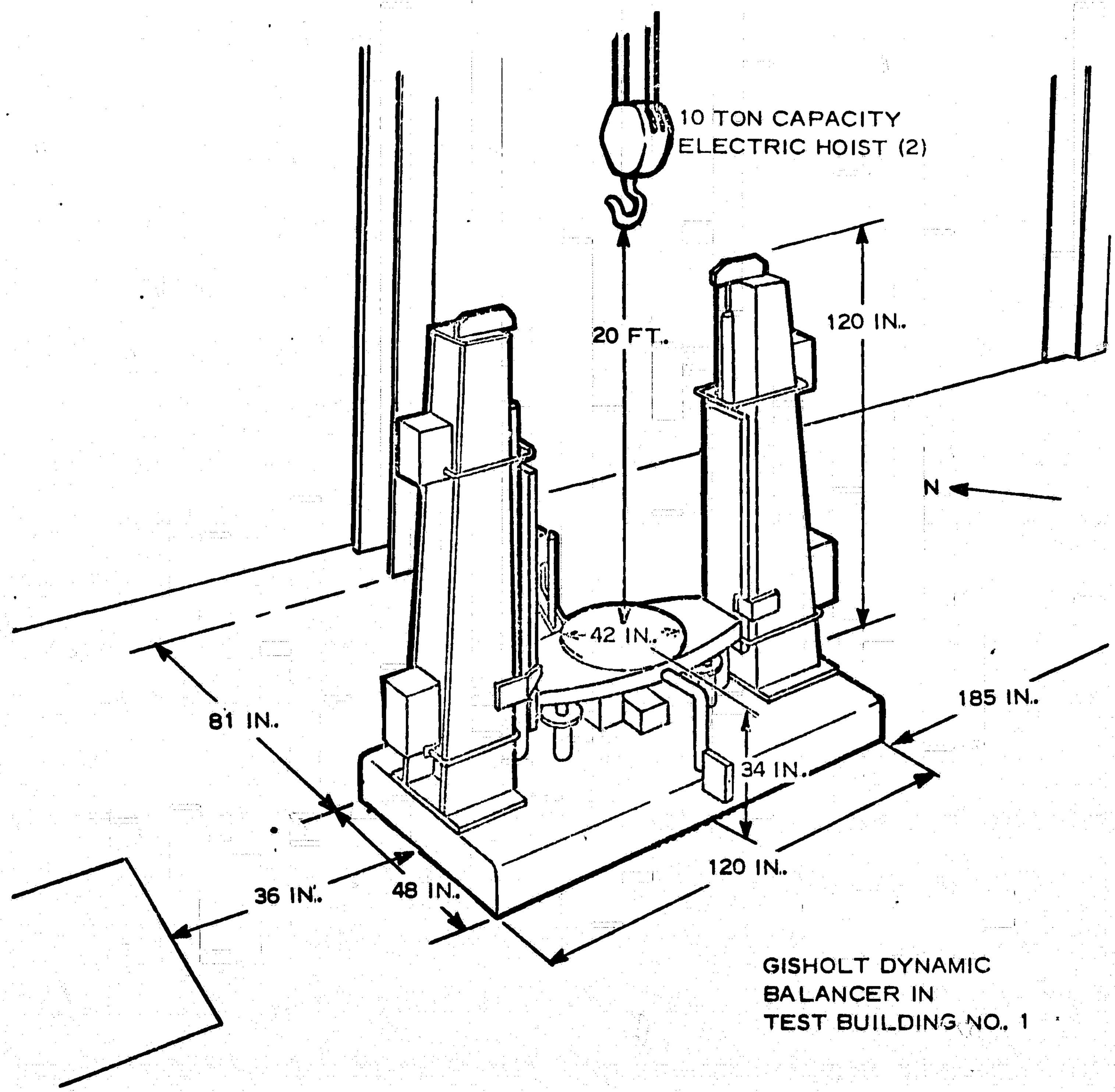


Figure 3-43. Gisholt Dynamic Balancer (T. B. #1)

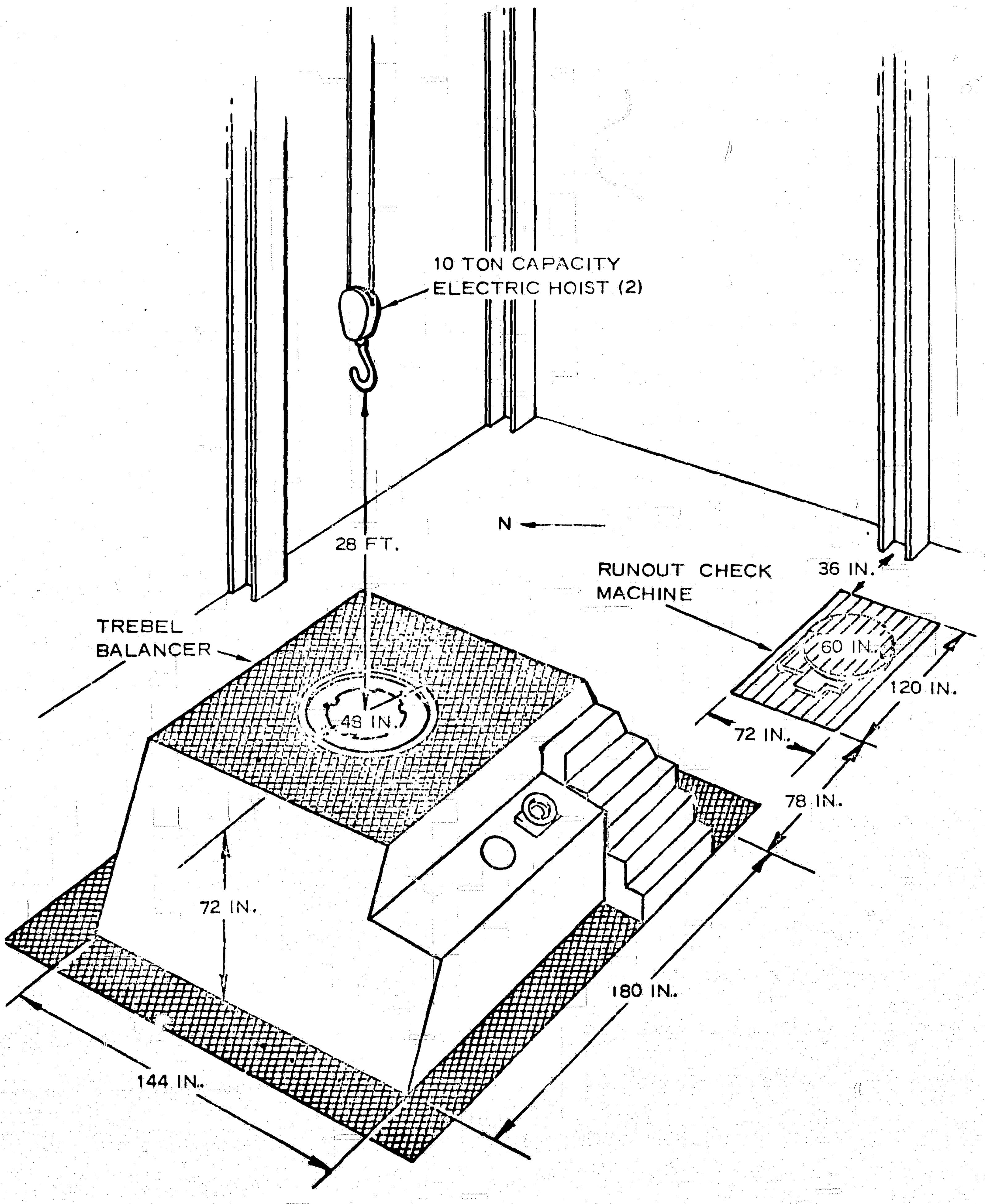


Figure 3-44. Runout Checking Machine and Trebel Dynamic Balancer (T. B. No. 2).

3.10 Aerospace Ground Equipment (AGE)

Aerospace Ground Equipment is defined as the equipment necessary to support the spacecraft during all of its ground handling, test, and checkout operations up to the time of launch.

3.10.1 Mechanical AGE

The mechanical AGE is used during shipment, handling, and orientation of the spacecraft from factory through field checkout to the mating of the spacecraft to the launch vehicle. The interface of the mechanical AGE with the spacecraft is shown in Figure 3-45. Use of the mechanical AGE is summarized in Figure 3-46 and described in 67SD649A (Revised 8 August 1967), Turbulent Heating Experiment Program/Mechanical Aerospace Ground Equipment/Operating Instruction Manual.

The mechanical AGE includes the following items:

(a) Shipping container

(b) Nose protector

(c) Hydraset

(d) Equipment Package Sling

(e) Assembly Tool

(f) Equipment Package Stand

(g) Rotational Sling

(h) Positioning Sling

3.10.1.1 Shipping Container - The shipping container (Figure 3-47) is used as a handling dolly for in-house processing and field handling of the spacecraft, and for transportation of the spacecraft to the field.

3.10.1.2 Nose Protector - The nose protector (Figure 3-48) protects the nose tip of the spacecraft during assembly, test, and transport.

3.10.1.3 Hydraset - The hydraset (Figure 3-48) is a hydraulic/pneumatic device for precision positioning, and is used between an overhead hoist and a load-bearing sling. It is used with the equipment package sling for removal of the equipment package assembly from the spacecraft and for reinstallation of the assembly into the spacecraft, and is used with the positioning sling for mating and demating of the spacecraft and the launch vehicle.

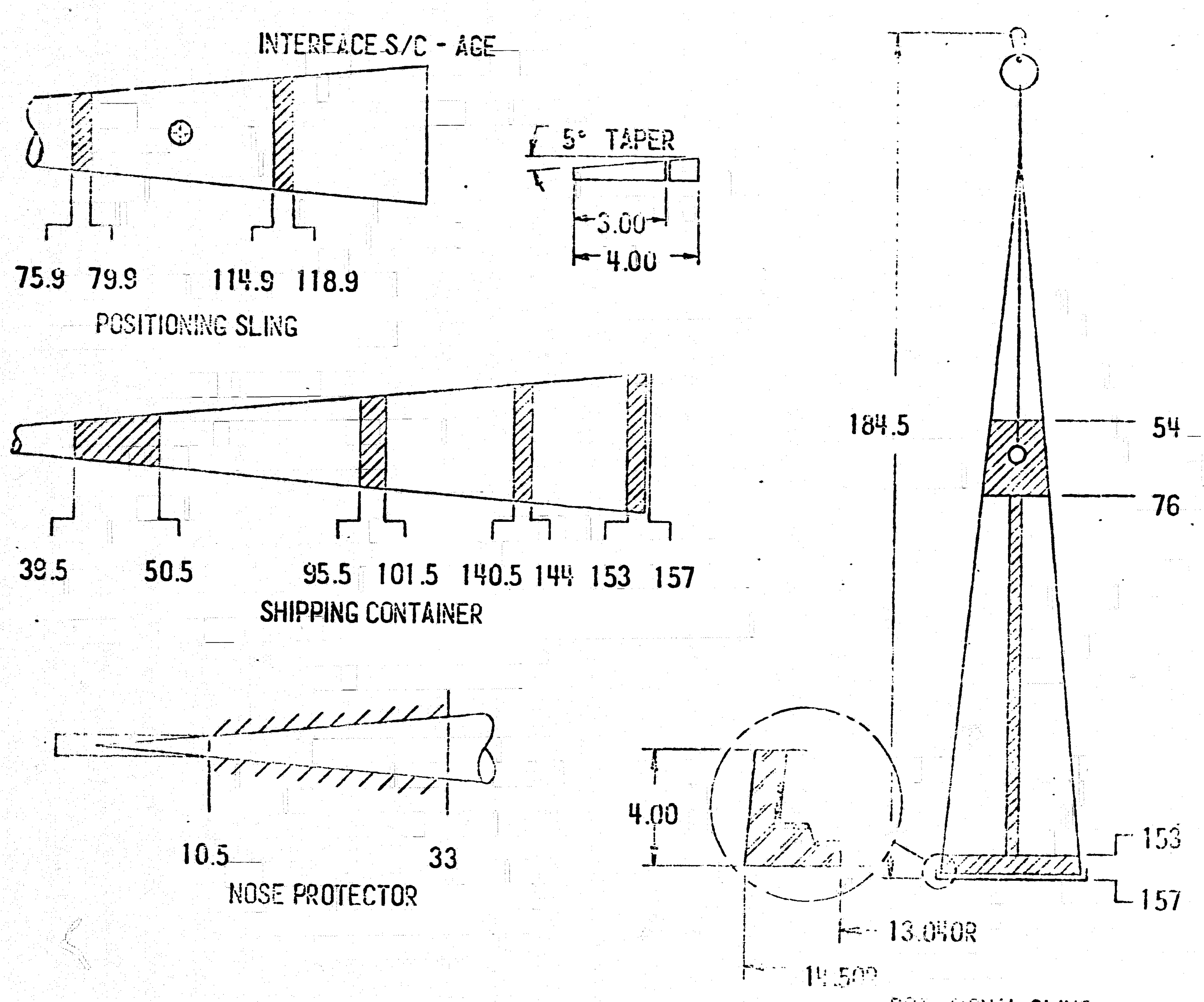
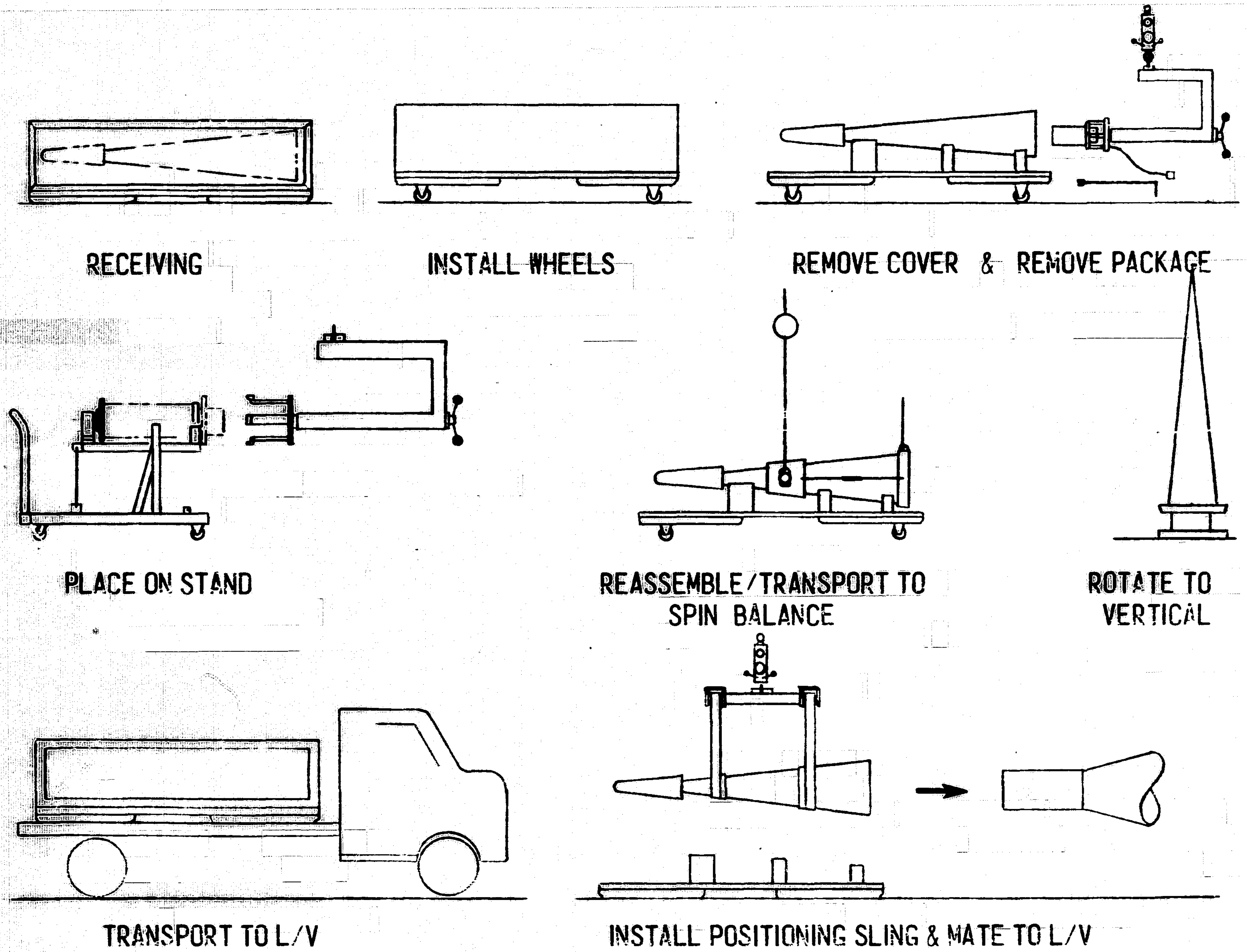


Figure 3-45. Mechanical AGE Interface with Spacecraft



NOTE: See subsequent figures for AGE identification

Figure 3-46. Use of Mechanical AGE

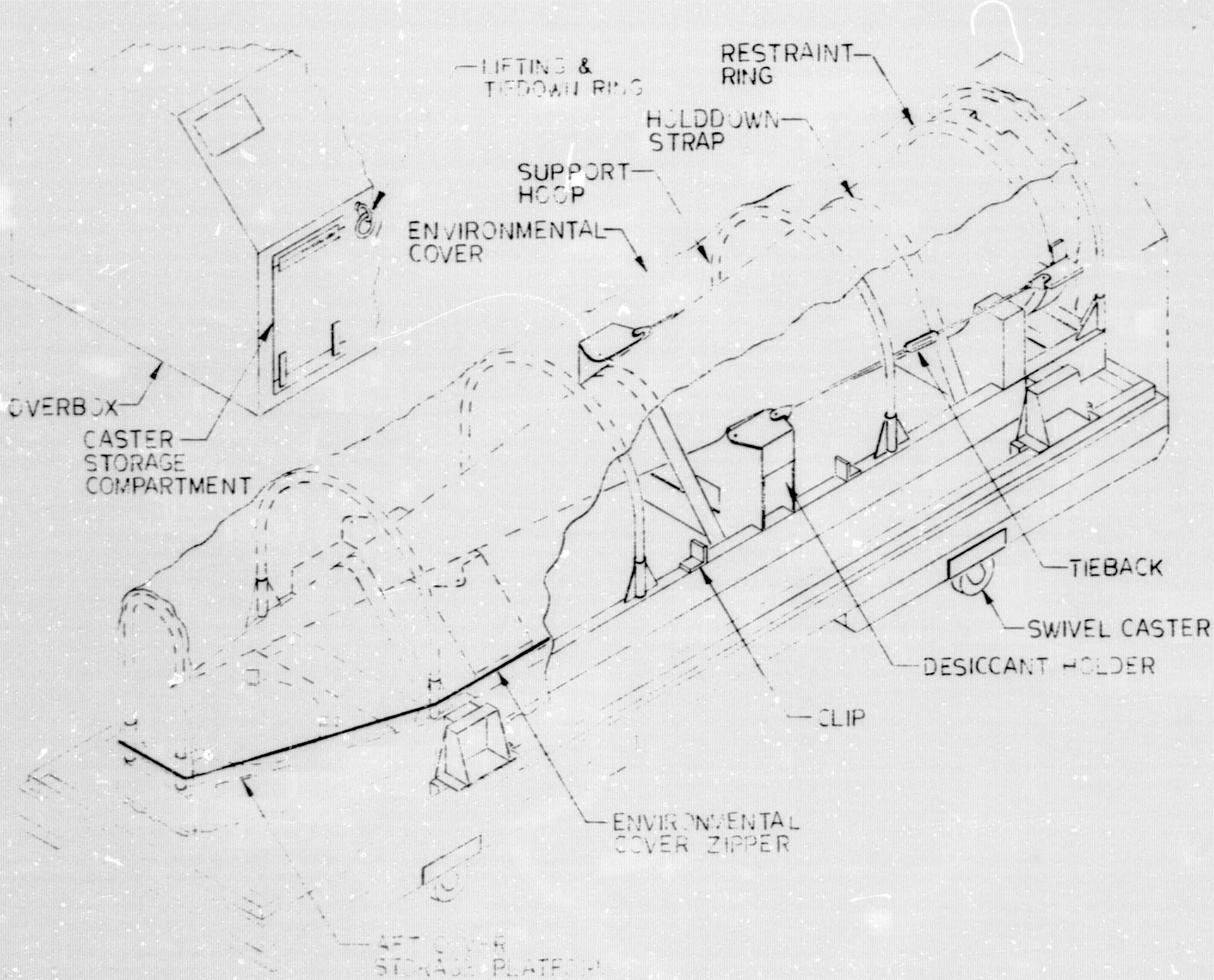


Figure 3-47. Shipping Container

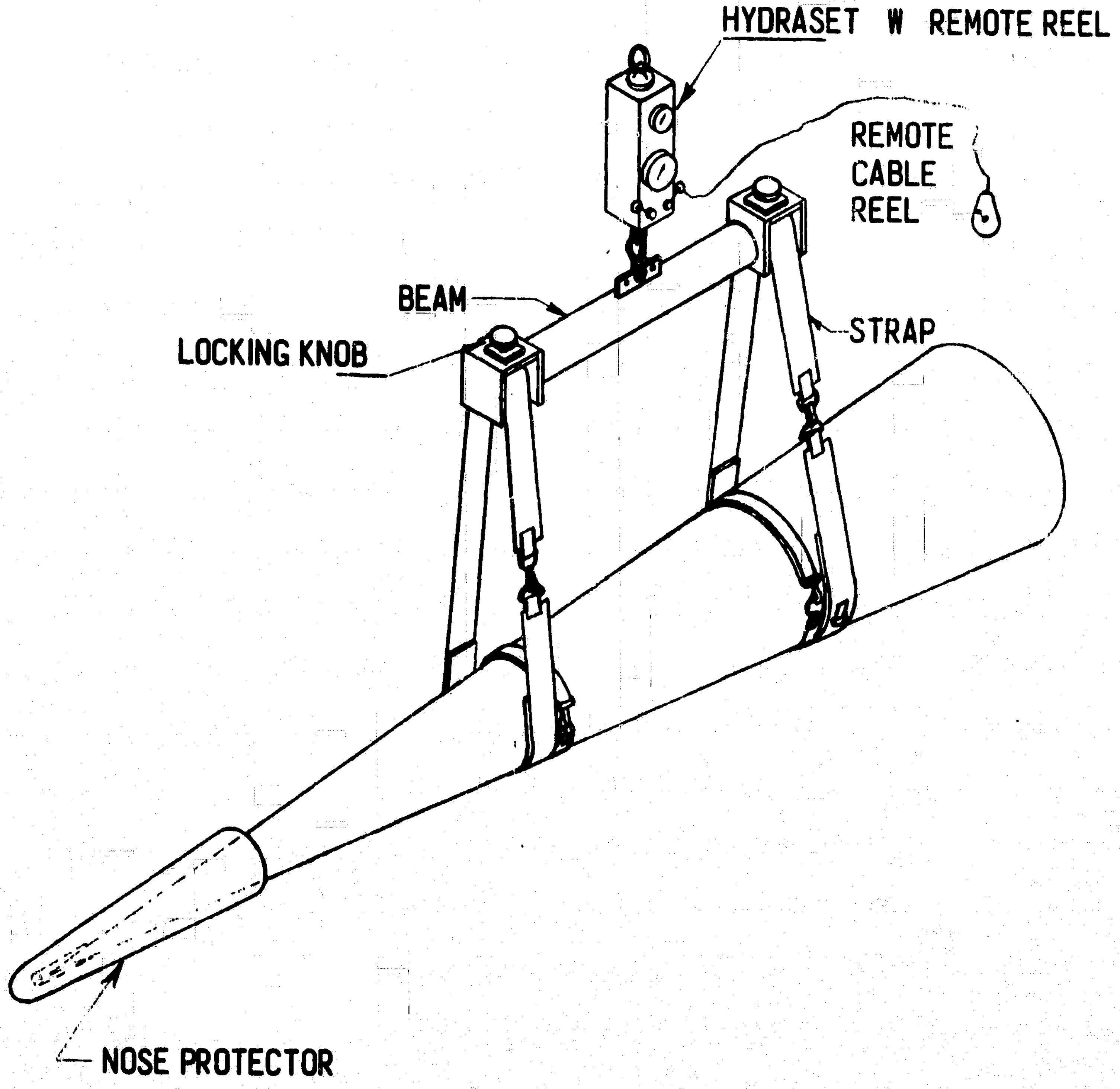


Figure 3-48. Positioning Sling, Nose Protector and Hydroset

The hydraset includes lifting rings for connection to the hoist and the sling, controls for raising and lowering the load, and a strain gage with a scale dial for indicating the weight of the load.

3.10.1.4 Equipment Package Sling - The equipment package sling (Figure 3-49) is used for installation and removal of the instrument package (equipment package assembly) during assembly, test, and checkout.

3.10.1.5 Assembly Tool - The assembly tool (Figure 3-49) is used during removal and installation of the equipment package assembly from/into the spacecraft. It is used to secure and disconnect fasteners in the equipment package sling, and to disconnect and secure fasteners that attach the equipment package assembly to the spacecraft.

3.10.1.6 Equipment Package Stand - The equipment package stand (Figure 3-50) is used for support, handling, and transport of the instrumentation package (equipment package assembly) during assembly, test, and checkout.

3.10.1.7 Rotational Sling - The rotational sling (Figure 3-51) is used to position the spacecraft vertically during in-house tests and field-site dynamic balancing.

3.10.1.8 Positioning Sling - The positioning sling (Figure 3-48) is used for lifting and handling of the spacecraft during assembly, test, and checkout, and for mechanical mating of the spacecraft to the launch vehicle.

3.10.2 Electrical AGE

The electrical AGE (Figure 3-52) is used for control, monitoring, and simulation. A technical description is provided in GE/RSD Document Number 67SD650, Technical Descriptions Re-entry AGE. An engineering test plan is provided in GE/RSD Document Number 67SD706, Engineering Test Plan for the Re-entry F Electrical Aerospace Ground Equipment. The electrical AGE consists of a Spacecraft Test Set, a Spacecraft Simulator, and a Low-Level Sensor Simulator.

3.10.2.1 Functional Configurations - The two major functional configurations are for the launch site, and for subsystem-level testing. In each of these major configurations, there are two subconfigurations: one in which the Spacecraft Test Set is connected to the spacecraft, the other in which the Spacecraft Test Set is connected to the Spacecraft Simulator.

3.10.2.1.1 Launch Site - The launch-site configuration is shown in the Launch Site Block Diagram (Figure 3-53) and the Launch-Site Interface Drawing. In this configuration, the Spacecraft Test Set is located in the blockhouse and the Spacecraft Simulator is located in the launcher area. The Spacecraft Test Set interfaces

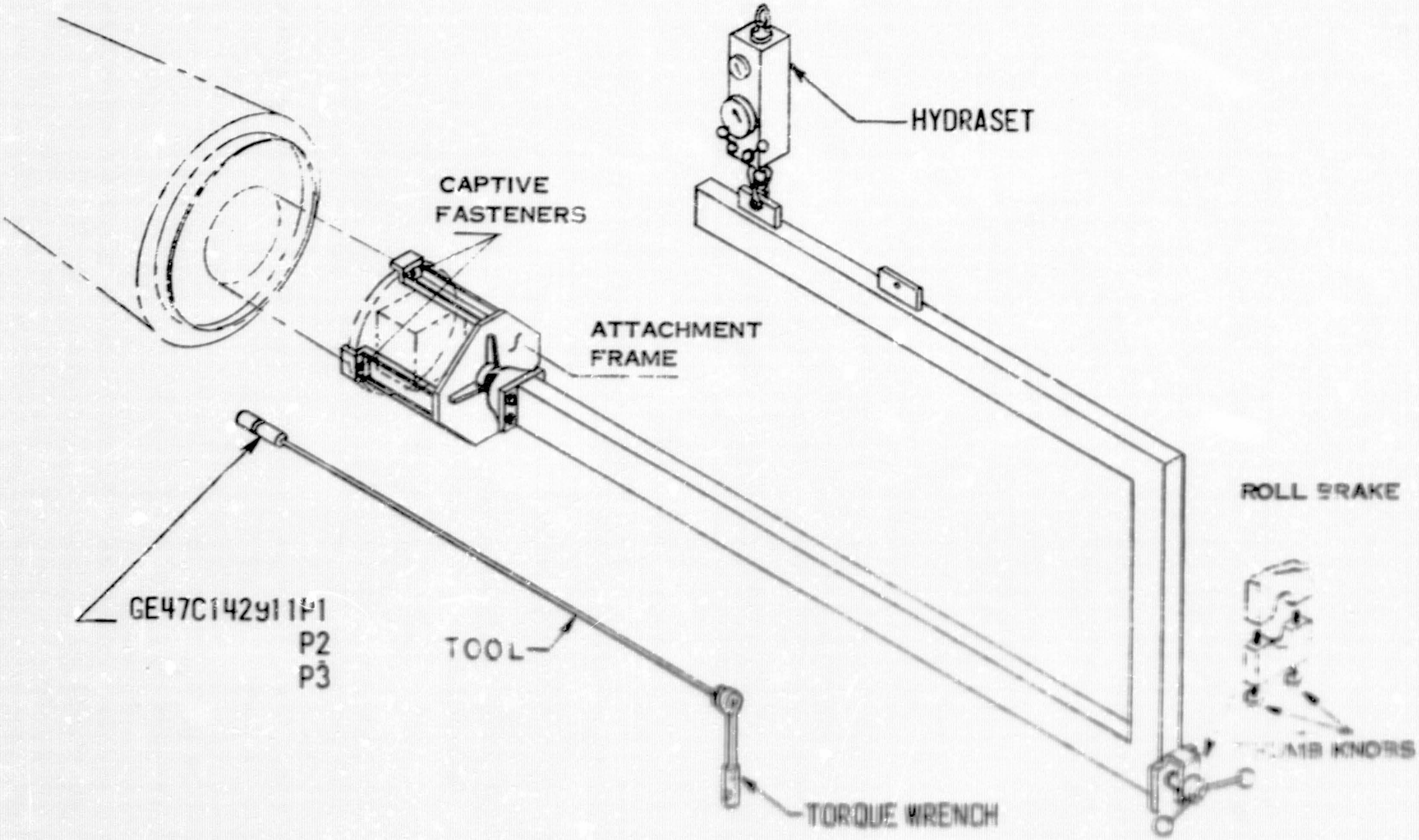


Figure 3-49. Equipment Package Sling and Assembly Tool

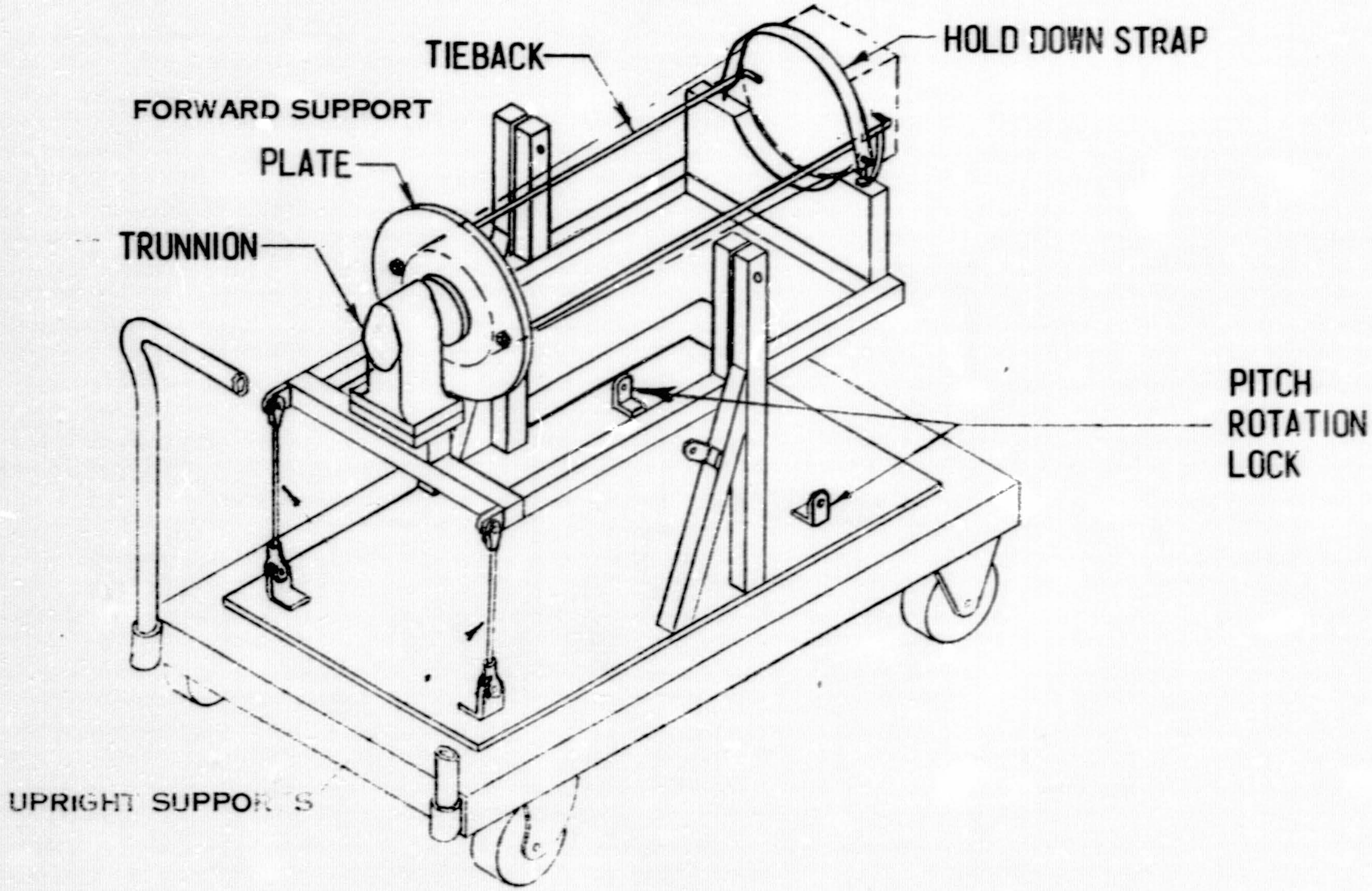


Figure 3-50. Equipment Package Stand

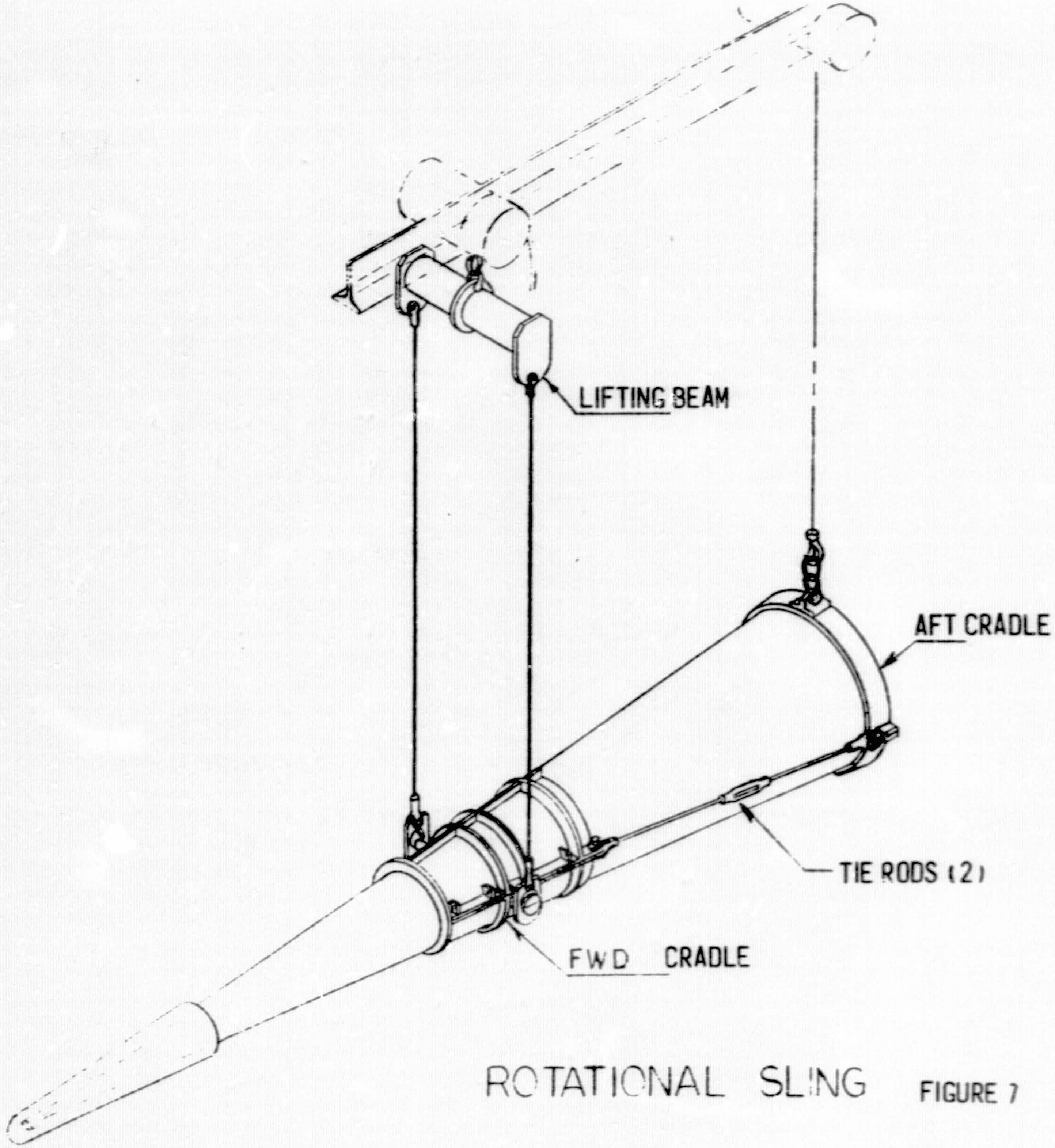


Figure 3-51. Rotational Sling

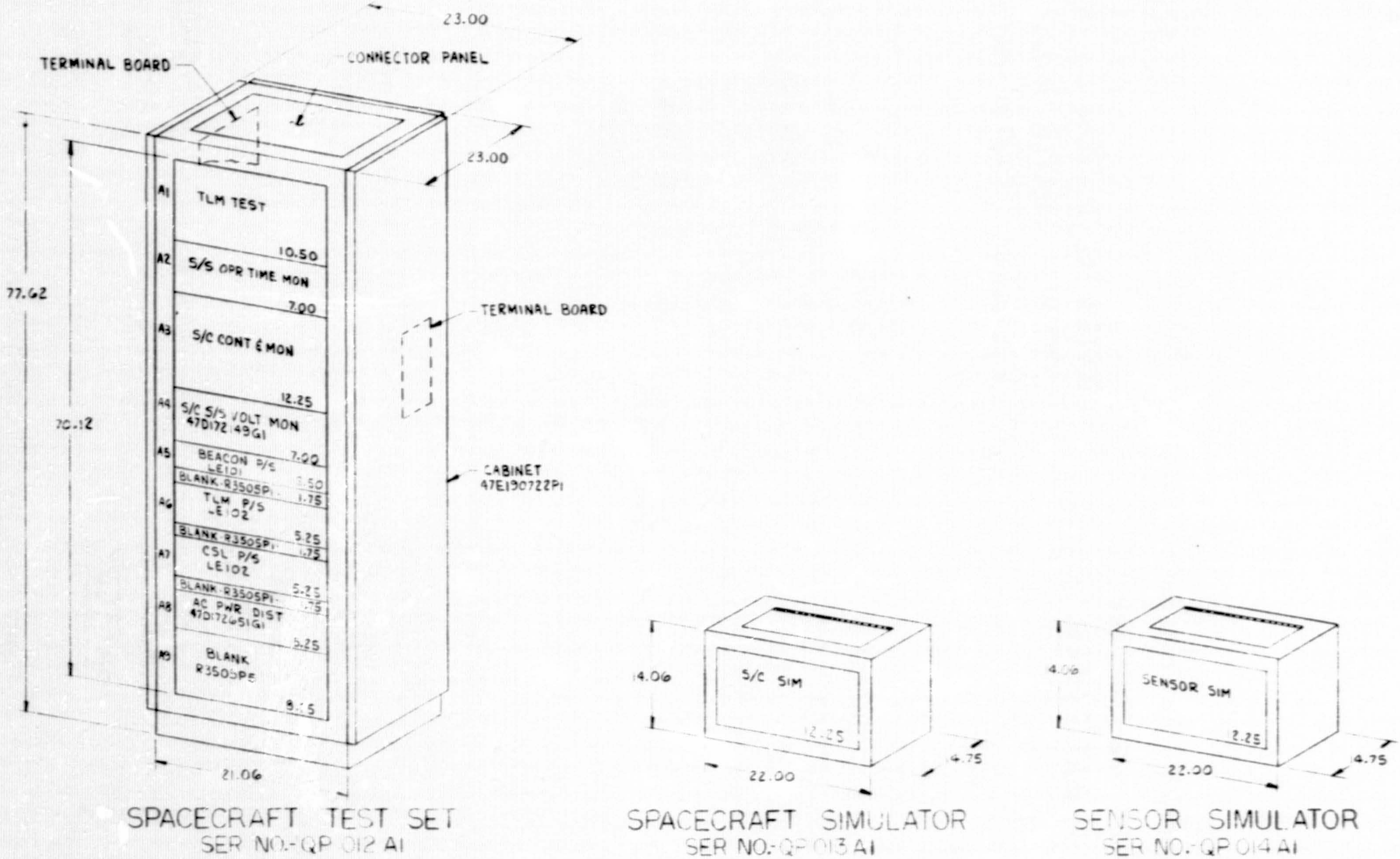


Figure 3-52. Electrical AGE

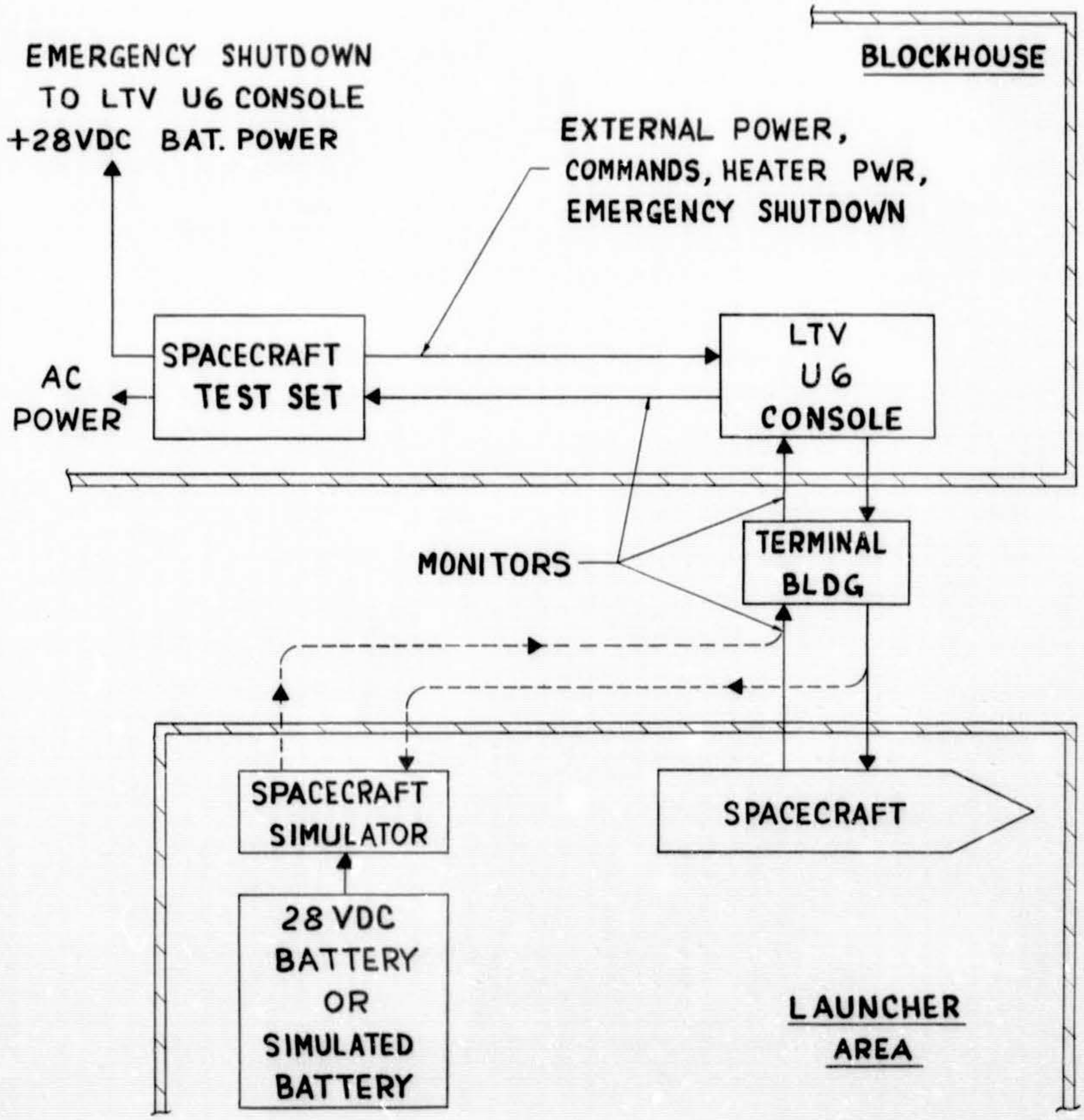


Figure 3-53. Launch Site Block Diagram

with the LTV U6 console in the blockhouse. Land lines are provided by LTV to transmit both power and signals to the launcher junction box. GE provides the umbilical cable from the junction box to the spacecraft. Launch-site functions include control and monitoring of power and the provision of spacecraft simulation.

3.10.2.1.1.1 Spacecraft Simulation - The Spacecraft Simulator is used to test the land line system between the Spacecraft Test Set (in the blockhouse) and the Spacecraft umbilical connector at the launch site.

3.10.2.1.2 Subsystem-Level Testing - The subsystem-level testing configuration is shown in the Subsystem-Level Testing Block Diagram (Figure 3-54). In this configuration, both the Spacecraft Test Set and the Low-Level Sensor Simulator are used. The instrument package is removed from the spacecraft and placed on its stand. The electronic components can now be mated electrically to both the Spacecraft and the electrical AGE through adapter cables. This configuration permits electrical test and checkout of the Telemetry Subsystem by inserting simulated high and low level signals from the Spacecraft Test Set and the low level simulator.

3.10.2.2 Equipment - The equipment includes the Spacecraft Test Set, the Spacecraft Simulator, and the Low-Level Sensor Simulator.

3.10.2.2.1 Spacecraft Test Set. - The Spacecraft Test Set performs the functions mentioned in paragraph 3.10.2.1.1 except spacecraft simulation and low-level sensor simulation. It is located in the blockhouse and consists of a single bay cabinet with nine chassis, four control-and-indicator panels (A1 through A4), four power supplies (A5 through A8), and one blank panel (A9). The physical arrangement of the nine chassis in the cabinet is shown in Figure 3-52.

3.10.2.2.2 Spacecraft Simulator - The Spacecraft Simulator (Figure 3-55) is housed in a table-model console. The console includes a control-and-indicator panel with appropriate push buttons, lamp-type indicators, and test-point jacks.

3.10.2.2.3 Low-Level Sensor Simulator - The Low-Level Sensor Simulator (Figure 3-56) is a test set that provides low-level simulation voltages (0 to 50 millivolts) to low-level commutators, which are located on the instrumentation shelf and harnessed to the spacecraft. The Low-Level Sensor Simulator is housed in a table-model console located in the blockhouse.

3.11 Flight Operations

The flight operation activities are described in detail in the Re-entry F Payload Description document. The following briefly summarizes range support and Wallops operations.

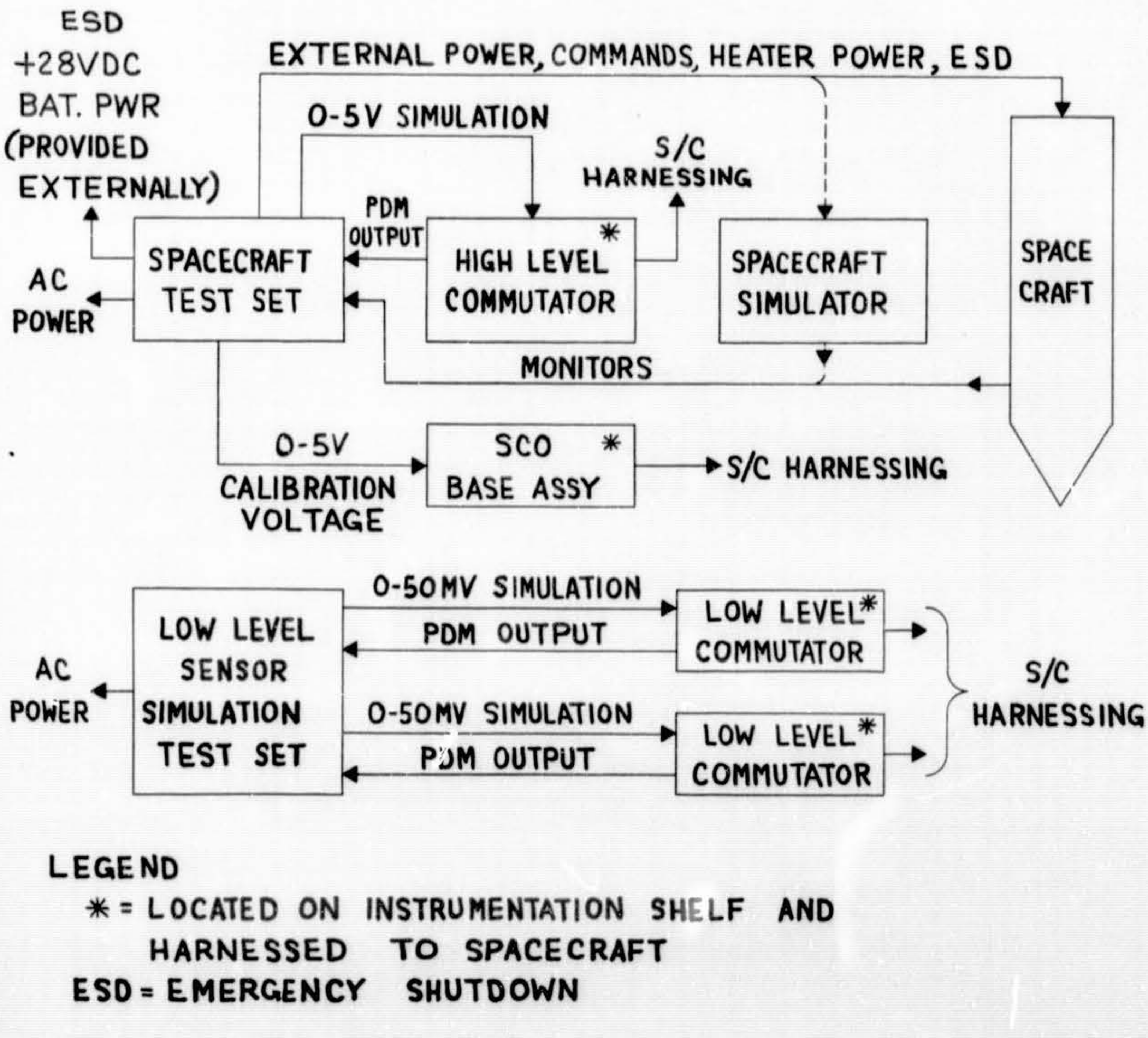


Figure 3-54. Subsystem Level Testing Block Diagram

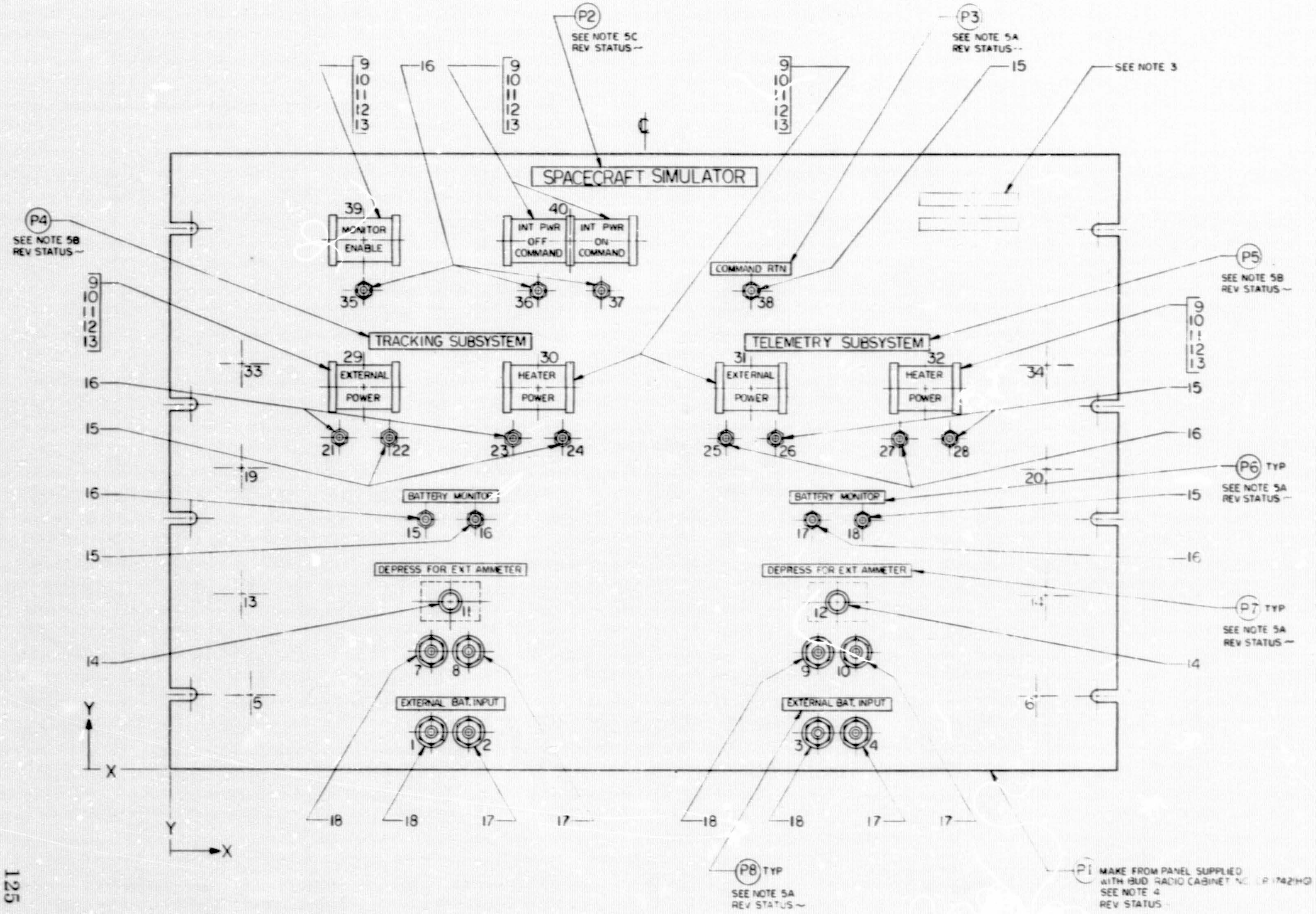


Figure 3-55. Spacecraft Simulator

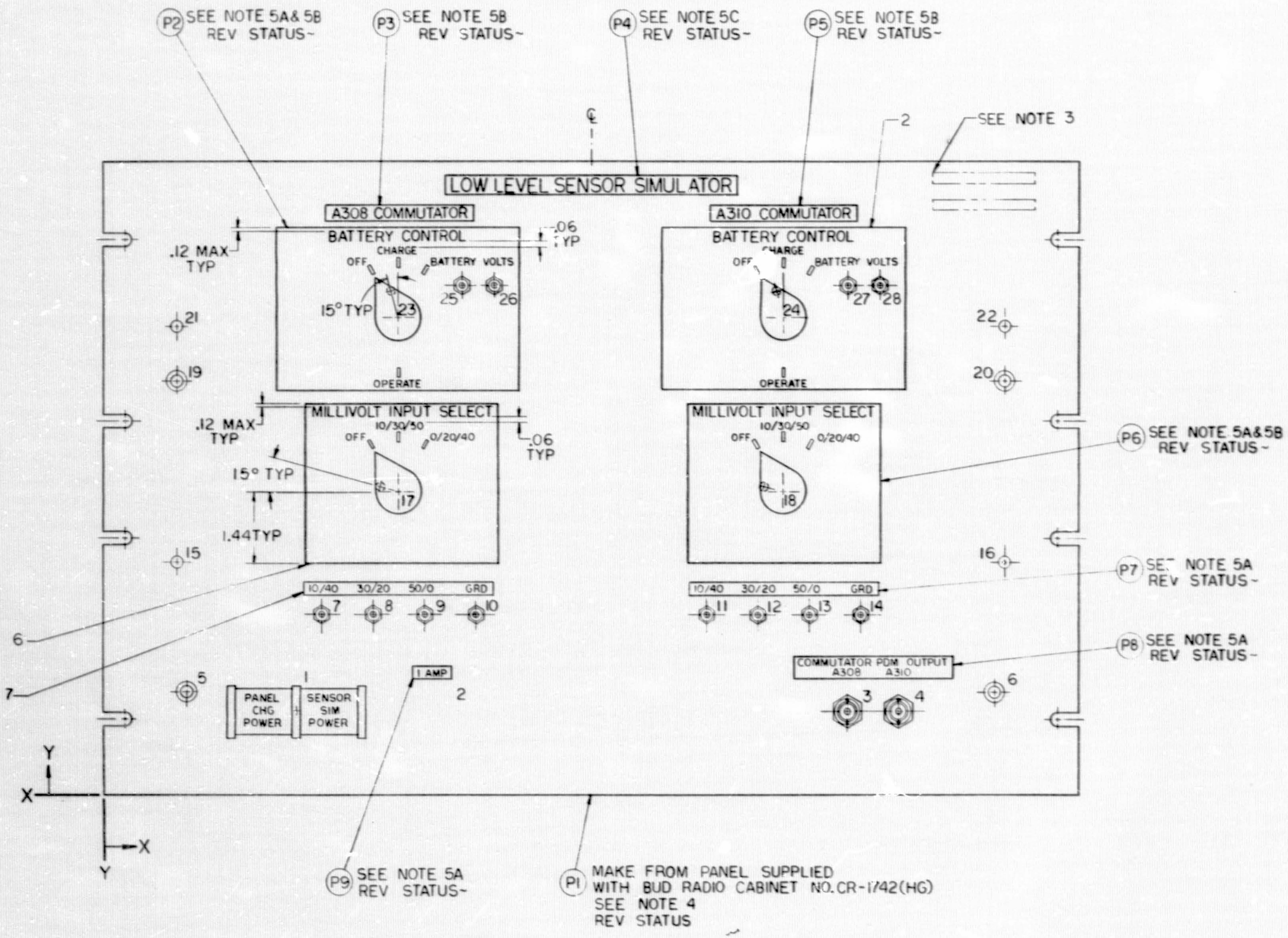


Figure 3-56. Low Level Sensor Simulator

3.11.1 Range Support

Range Support will include:

(1) Telemetry data acquisition at Wallops, at Bermuda, aboard the ship "Range Recoverer", and aboard two down-range NASA aircraft.

(2) Radar data tracking acquisition data at Wallops and at Bermuda.

(3) Optical data acquisition (Spacecraft re-entry) aboard the two down-range NASA aircraft.

(4) Meteorological Data acquisition at Wallops and at Bermuda. Both balloons and rockets will be utilized for this purpose.

(5) Adequate communications between all elements of the range.

(6) Timing for telemetry will be 100 PPS and 2 PPS binary-coded-decimal time-of-day signals or in 100 PPS and 1 PPS 17 digit AFETR time codes.

3.11.2 Wallops Operations

Following is a summary description of the organization and facilities of the NASA Wallops Station, and of the sequence of activities in support of the Re-entry F Program.

3.11.2.1 Organization and Facilities - The organization of the NASA Wallops Stations is described in the Wallops Station Handbook, a NASA Document. The Wallops organization for support of the Re-entry F Program is responsible to the NASA Langley Research Center. Launch operations at NASA/Wallops are contracted to LTV under NASA direction.

The Wallops Station consists of the main base on the mainland and the Wallops Island facilities connected by a causeway. The station headquarters is on the main base, and the launch site is on the island. Re-entry F will use Launch Area No. 3. The assembly building for Re-entry F is shown in Figure 3-57.

The Dynamic Balancing facility and its floor plans of the facility are shown in Figure 3-42. Dynamic balancing equipment includes the Gisholt Dynamic Balancer (Figure 3-43) in Test Building No. 1, and the Runout Check Machine and Trebel Balancer (Figure 3-44) in Test Building No. 2.

The launcher is the Mark II Scout Launcher. Blockhouse-to-payload cabling is shown in Figure 3-58.

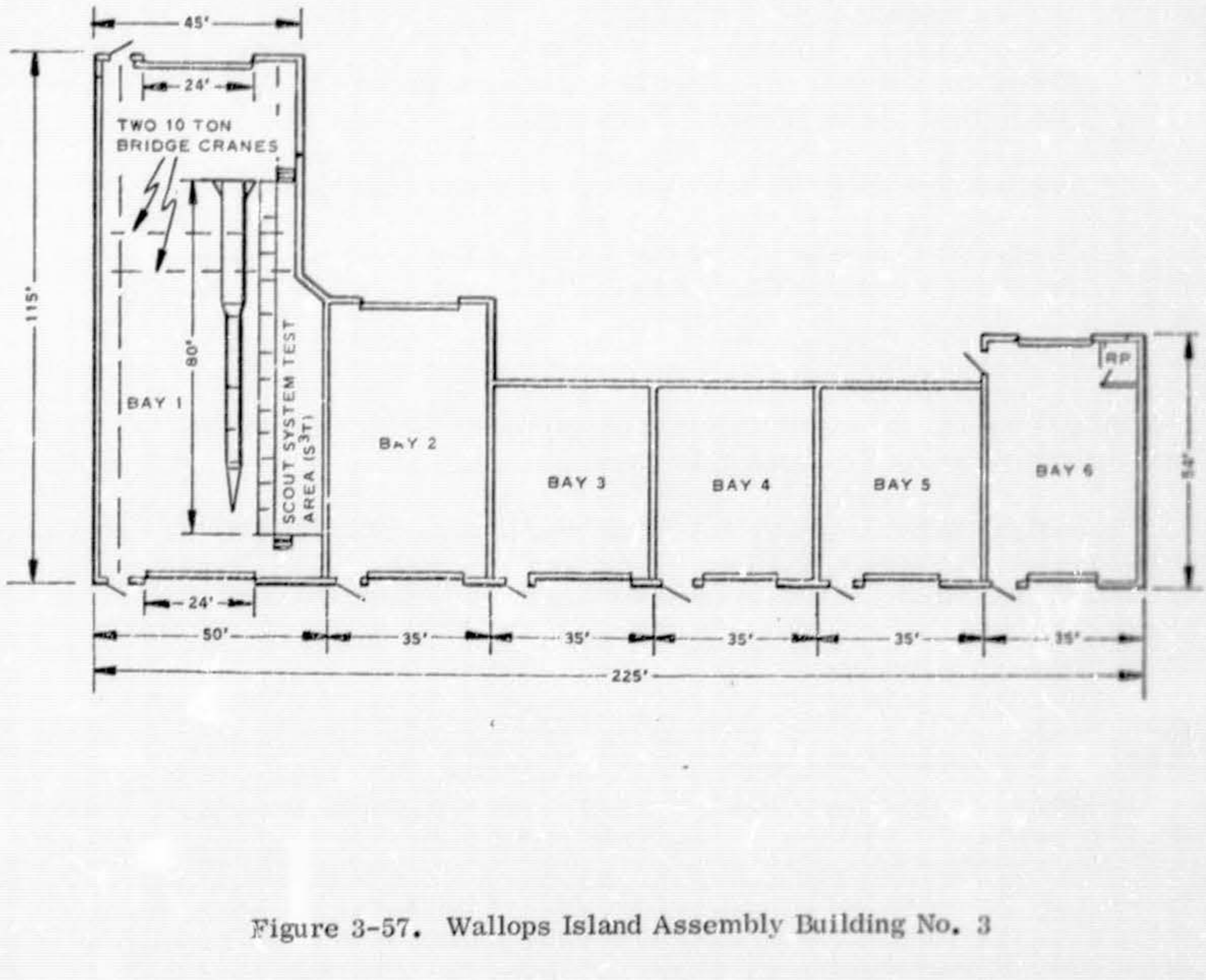


Figure 3-57. Wallops Island Assembly Building No. 3

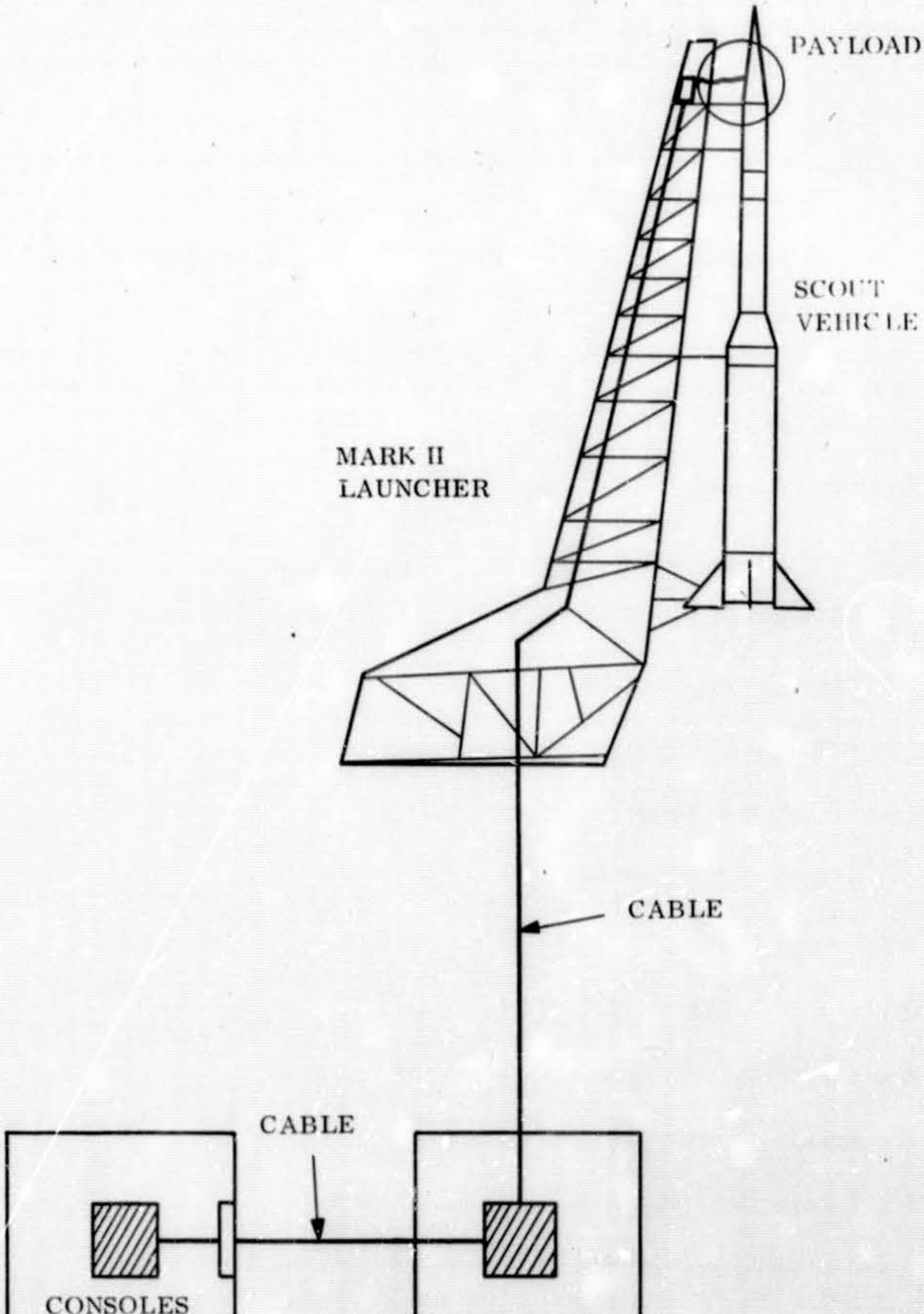


Figure 3-58. Blockhouse to Payload Cabling, Simplified Diagram

3.11.2.2 Activities Sequence - The activities sequence for the Re-entry F spacecraft at NASA/Wallops is described in the Payload Description Document and in GE/RSD Document No. 67SD781, Systems Acceptance Test Plan for Re-entry F Turbulent Heating Experiment. The activities sequence for the Scout launch vehicle and the integration of the spacecraft (payload) with the Scout is described in the Scout User's Manual.

The spacecraft activities sequence at NASA/Wallops is summarized in Figure 3-59.

The sequence has three portions: checkout to Countdown, Countdown to liftoff, and liftoff.

3.11.2.2.1 Checkout to Countdown - The sequency during checkout to countdown begins with the delivery of the spacecraft to NASA/Wallops and its clearance through receiving inspection. Spacecraft activities are as follows:

a) Spacecraft Confidence Test (SI 243889)

b) Spin Balance (as planned)

c) Pad Validation (SI 243880)

d) Disassembly and inspection (as planned)

e) VSWR Test (SI 243884)

f) Instrumentation Tests (SI 243882)

g) Flight Battery Installation (SI 243892)

h) Mechanical Mate (as planned)

i) Spacecraft Confidence Test (SI 243889)

j) Spacecraft- Booster Mate (as planned)

k) Spacecraft-Booster RFI Test (SI 243898)

l) Final inspection and dress rehearsal (SI 243899)

Spacecraft countdown preparation begins at T-540 minutes (9 hours before scheduled launch). The Standing Instruction for Final Inspection and Dress Rehearsal is the same as for Countdown.

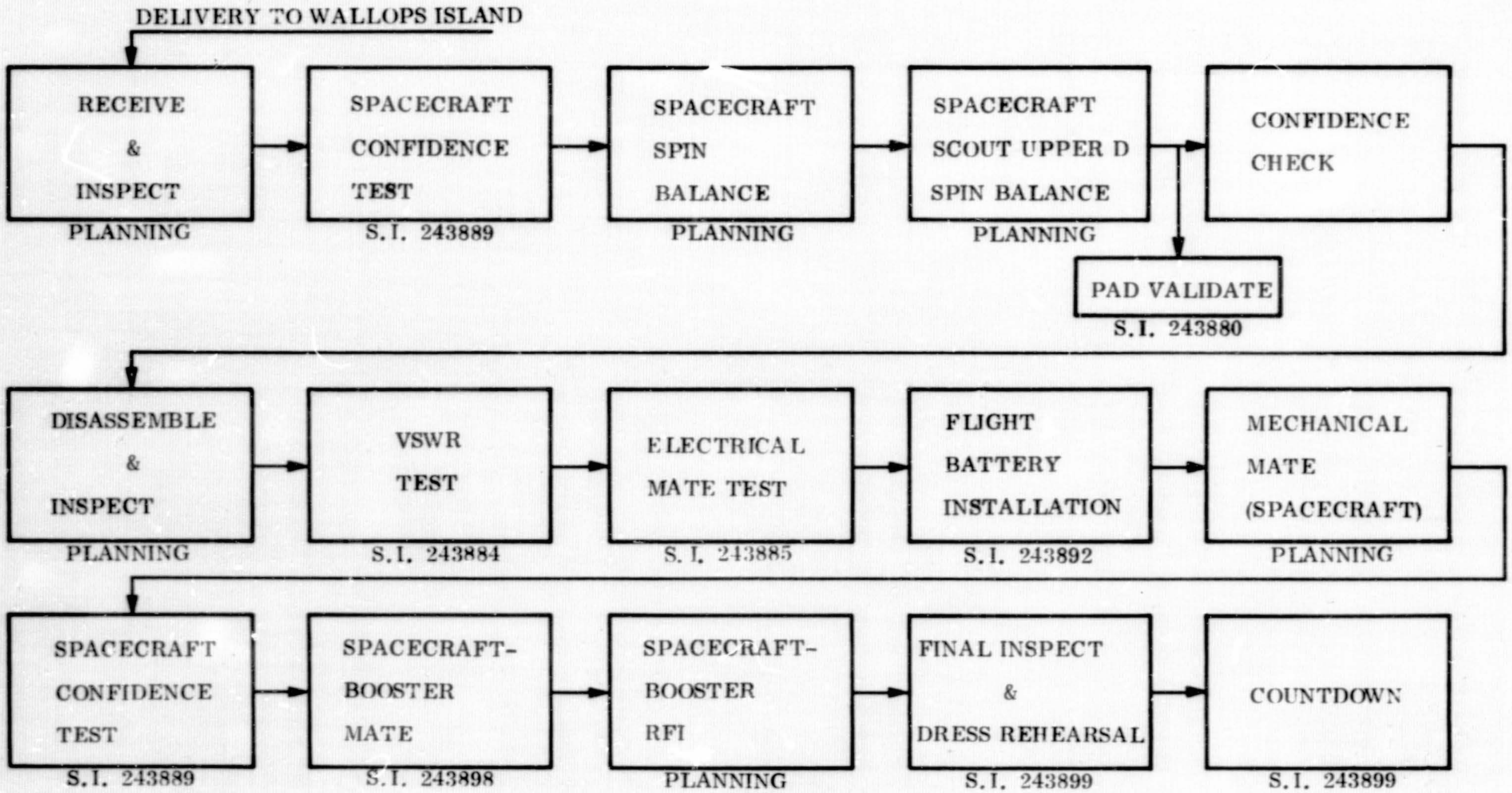


Figure 3-59. Spacecraft Activities Sequence at NASA/Wallops

3.11.2.2.2 Countdown - The Countdown procedure is described in SI 243889 and summarized in Table 3-14. Coundown begins at T-540 minutes for the spacecraft and T-330 minutes for the Scout.

The spacecraft telecommunications (VHF telemetry) and tracking (C-Band beacon) subsystems will be turned on during Scout electronics checkout (beginning at T-300 minutes), to verify critical launch parameters and provide assurance of the absence of radio-frequency interference (RFI).

Beginning with terminal countdown (T-30 minutes), the telecommunications and tracking subsystems will be monitored by Wallops instrumentation. This instrumentation includes magnetic tape recorders and five 8-channel strip chart recorders, for monitoring the VHF telemetry and an FPS-16 tracking radar for interrogating and tracking the C-band beacon of the spacecraft. A description of the Wallops instrumentation is provided in the Payload Description Document.

3.11.2.2.3 Liftoff - At liftoff (T-0 minutes), the umbilical is pulled away as the spacecraft is launched. VHF telemetry monitoring and C-band interrogation and tracking at NASA/Wallops continue as begun at T-30 minutes during Countdown (paragraph 3.11.2.2) until loss of signal (approximately T+7 minutes).

3.11.2.3 Launch Restraints

Weather. . . . . same as for Scout vehicle

3.11.2.4 Technical Difficulties

Mandatory holds if the following are not satisfactory:

Communications (between Wallops Island and downrange)

Spacecraft telemetry (all channels)

Ground telemetry (all stations)

Radar beacon (C-Band transponder)

Radar (Bermuda FPQ-6)

TABLE 3-14. COUNTDOWN

Countdown Time (minutes)	Task Title	Time (minutes)
T-540 (9.0 hours)	Spacecraft Countdown Preparation	20
T-520	Spacecraft Power Control Checks	20
T-500	Spacecraft Ambient Telemetry Verification	120
T-380	Spacecraft C-Band Beacon Checkout	40
T-340	Secure Spacecraft Operations	--
T-330	Scout Countdown Preparation and Initiation	30
T-300 (5.0 hours)	a) Scout Electronics Checkout	120
	b) Spacecraft Telemetry and beacon will be turned on during the period to verify launch critical parameters and to provide RFI assurance	
T-180 (3.0 hours)	Hydrogen Peroxide and Nitrogen Fueling	90
T-90	Closeout and Erection	60
T-30	Terminal Countdown	30
	The Spacecraft portion of the Terminal Countdown will include the following:	
	1) Verification of launch critical parameters	
	2) Assurance of GO beacon and telemetry from range support	
	3) Transfer of spacecraft to internal power (approximately T-5 minutes)	
	4) Upon verification of no changes in launch critical parameters after the switch to internal power, the Spacecraft Test Conductor will signify GO to the LTV Launch Test Conductor.	
T-0	Umbilical disconnect; liftoff	

4.0 Documentation and Data

This section includes a listing of the contract documentation as well as other pertinent data for the Re-entry F Turbulent Heating Experiment.

4.1 Contract Documentation

The contract documentation is provided in Table II of NASA Document No. L-6345-A, Exhibit A (28 April 1966), Statement of Work/Re-entry F Turbulent Heating Experiment, an attachment to the contract (NAS 1-6039).

The schedule is organized in three parts: Part A, as specified in the statement of work (L-6345-A Exhibit A); Part B, as specified in Document No. NPC 250-1; and Part C, as specified in Document No. NPC 200-2. Each part is organized in sequence of paragraph reference to the corresponding basic document (work statement, NPC 250-1, or NPC 200-2) and includes the following information: paragraph reference, document title, date of submittal to NASA, contractor updating requirements, quantity of copies required by NASA, and NASA action.

4.2 Documentation and Index

The Documentation referenced below is indexed according to the CDRL Sequence Number and the documents presently issued and supporting further detail for the familiarization manual are listed below.

A complete listing of the CDRL sequence numbers are included in this document, on page 13.

DESCRIPTION	DOCUMENT NO.
CDRL No. 006 Environmental Criteria Specification	
INTERNAL ENVIRON CRITERIA-REV1	S0020-02-0020
EXTERNAL ENVIRON CRITERIA	S0020-01-0015
CDRL No. 007 Design Criteria Specification	
PROGRAM DEFINITION, REENTRY F	PIR-8252-5512
SYSTEM DESIGN REQUIREMENTS	S0010-02-00268
NOSE TIP DEVLPMT PROGRAM	PIR-8157-872
RE-ENTRY F INTERFACE DATA	PIR-8225-1658
CDRL No. 008 Material, Parts and Process Specification	
SPEC-PRIME BERYLLIUM PARTS	PIR-8157-F763
MECH PROP-ATJ GRAPHITE	PIR 8157-165
POCO GRAPHITE GRADES REENTRY F	PIR-8157-804
POROUS CARBON, POROUS GRAPHITE	S0060-00-0008
PYROLYTIC GRAPHITE CYLINDERS	SPEC-NCS3389A
ZIRCONIUM OXIDE COAT-METAL-NONMET	SPEC-RSD4055
GRAPHITE SPEC	SPEC-NCS3146B
PYROLYTIC GRAPHITE PLATE MATL	S0060-02-0011
POCO GRAPHITE	PIR-8157-827
BERYLLIUM ETCHING PROCESSES	PIR-8157-906
MATERIALS STAGE RELEASE 3	PIR-TDM-005
SELECTED MATLS-PROCESSES LIST	S0060-07-00048
SURFACE CLEANING OF BERYLLIUM	PIR-8631-494
GRAPHITE BONDING (NON-METALLIC)	S0060-02-0015
CONDENSE BILLET RQMTS-COMPONENTS	PIR-8157-933
LINING-STRUCT GRAPHITE NOSE TIPS	PIR-8157-960
POCO GRAPHITE	PIR-8157-901
GRAPHITE FELT	PIR-8157-897
SPEC-BERYLLIUM FOR STRUCT APPLIC	171A4428-A

DESCRIPTION	DOCUMENT NO.
CDRL No. 009 Aerospace Ground Equipment (AGE) Specification	
REQMTS-SYSTEM TEST FOR AGE	PIR-8363-004
PRELIM MECH AGE-SPCRFT INTRFCE	PIR-8262-4624
AGE SPEC-ROTATIONAL SLING	S3940-11-0023
AGE SPEC-EQUIPT PACKAGE STAND	S3990-03-0016
AGE SPEC-POSITIONING SLING	S3940-11-0022
AGE SPEC-NOSE PROTECTOR	S8140-03-0005
AGE SPEC-EQUIPT PACKAGE SLING	S3940-11-0021
AGE SPEC-TEST SET, SPACECRAFT	ERS4935-04-0027
ELECTRICAL AGE AND TSE	PIR-8363-011
MECH AGE INTRFCE-BE CALORIMETER	PIR-8262-4665
MECH AGE USAGE PLAN-REENTRY F	PIR-8363-022
ELECT AGE SPEC-SPCRFT SIMULATOR	S4935-02-0018
ELECT AGE SPEC-LL SENSOR SIMULATR	S4935-04-0028
DWG TREE REENTRY F MECH AGE	ER47C172409
AGE SPEC-ASSEMBLY TOOL	S5120-76-0001
AGE SPEC-SPCRFT SHIP CONTAINR	ERS8140-02-0014
RE-ENTRY F AGE-TECH DESCRIPTION	67SD650
MECH AGE OPERATING INSTR MANUAL	67SD649
CDRL No. 012 Support Plan	
SUPPORT PLAN	67SD709
CDRL No. 013 Integrated Test Plan	
INTEGRATED TEST PLAN	66SD961
CDRL No. 017 Analytical Reports	
BASE FLOW CONDITIONS REENTRY F	PIR-8152-2024
SYSTEM TLM CONSIDERATIONS	PIR-8233-1989
ANTENNA PATTERN REQUIREMENTS	PIR-8174-690
UMBILICAL DISCONNECT LOCATION	PIR-82NO-9656
PRELIMINARY ANTENNA PATTERN	PIR-8235-F680
EXPRMT ERROR ANALYSIS SIMULATN	PIR-8153-1693
TELECOMM TRADE-OFF CONCLUSIONS	PIR-8233-2003

DESCRIPTION	DOCUMENT NO.
CDRL No. 017 Analytical Reports (Continued)	
PRELIM DEFNTN EQUIPT PKG ENVLP	PIR-8225-1621
SPACECRFT REF COORDS, GEOMETRY	PIR-8225-1618
PRESSURE SENSOR REQUIREMENTS	PIR-8152-2029
THERMAL, PRESSURE SENSOR REQMTS	PIR-8159-1483
PRELIM BOOST TRAJ-600LB PAYLD	PIR-8153-1709
MASS PROPS BRKDN-AIR GAP DESGN	PIR-8226-6252
ADDTL MASS-THERMAL SENSTEMP	PIR-8151-718
THERMAL PRESSURE SENSOR REQMTS	PIR-8151-623
STRUCT MECH TECHNLGY RELEASE6	PIR-8156-3079
RTO-EFV TLM SYS ERROR ANALYSIS	PIR-8174-646
CONVECTIVE HEAT FLUX-BNDRY LYR	PIR-8159-1512
REENTRY F ERROR ANALYSIS	PIR-8193-311
EFFECTS CONVECTIVE HEAT XFER	PIR-8159-1508
TELECOMM SYSTEM ANALYSIS	PIR-8174-709
CONSTRAINED RANGE PERFORMANCE	PIR-8153-1719
STRUCT MECH TECH RELEASE 8	PIR-8156-3183
TEMP PROFILES-ANT WINDOW,CALRM	PIR-8151-728
AFT COVER THERMAL ANALYSIS	PIR-8225-1628
TEMP DISTRIB THRU FIELD JOINT	PIR-8151-730
THERMAL SENS-PRESS PORT LOCATN	PIR-8225-1627
LOCAL FLOW PROPERTIES	PIR-8152-2080
TOLER ANALYSIS, CALORMTR ASSY	PIR-8225-1630
ELECT SYS REQMTS SPEC STAGE 3	FRS0010-08-0008
PRESS SENSOR LOCAT JUSTIFICATN	PIR-8152-2098
AEROELASTIC CHARACTERISTICS	PIR-8152-3003
FLT MEASMT SYS ERROR ANALYSIS	PIR-8174-112
REENTRY F ERROR SIMULATION	PIR-8174-726
THERMODYNAMIC INPUT REENTRY F	PIR-8151-524
REQMT-SINGLE BERYL POWDER LOT	PIR-8157-855
HI TEMP INSERT MATL-ATJ-PG	PIR-8157-844
THERMAL SENS-PRESS PORT LOCAT	PIR-8225-1035
PG PLATE-USE IN NOSE DESIGNS	PIR-8157-613
TRANSITION, LAMINAR-TURB FLOW	PIR-8237-349

DESCRIPTION	DOCUMENT NO.
CDRL No. 017 Analytical Reports (Continued)	
AFT COVER THERMODYNAM ANALYSIS	PIR-8151-665
CALCU-TURB HEAT TRANSFER RATES	PIR-8151-656
FLOW FIELD CALCU-5 DEGREE CONE	PIR-8152-2143
DSCS PROGRAM-HEAT XFER, FLUX	PIR-8151-658
VEHICLE LOADS, CRITERIA, STG 3	PIR-8156-3340
DESIGN OF AERODYNAMIC SHAPES . . .	PIR-8152-2113
CALIRIMETER AIR LEAKAGE GAPS	PRI-8156-3375
FLT MEASMT SYS ERROR ANALYSIS	PIR-8174-112
THERM SENSR-FLT EXPRMT SIMULATN	PIR-8159-1616
RATE GYRO MISALIGN ERRORS	PIR-8174-742
VEHICLE LOADS, CRITERIA, STG 3	PIR-8156-3340
GRAPHITE NOSE TIP DESIGN	PIR-8159-1618
AEROELASTIC CHARACTRS REVISED	PIR-8152-2156
SENSOR BACKFACE-DERIVED HEAT FLUX	HTT-8151-675
FLIGHT DYNAMICS STG 3 RELEASE	PIR-8153-1831
THERM PROPS-BERYLLIUM BILLETS	PIR-8155-1394A
BIBL-BOUNDARY LAYER TRANSITION	66SD9267
THERMAL SENS-PRESS PORT LOCAT	PIR-8225-1640
CALORMTR PRESSR XDUCER MOUNTG	PIR-8225-1642
INTRFCE THERM CONTACT CONDUCT	65SD4395
REENTRY F NOSE PLUG ANALYSIS	PIR-8151-674
THERM PROPS-BERYLLIUM BILLETS	PIR-8155-1394A
COMPONENT HEAT DISSIPATION	PIR-8103-1886
AERODYNAMICS STAGE 3 RELEASE	ATDM67-09
STAGE 3 THERMOSTRUCTURAL ANALYSIS	PIR-8156-3448
THERMOSTRUCTURAL ANALYSIS-NOSE TIP	PIR-8156-3515
DYNAMIC ANALYSIS-STAGE 3	PIR-8156-4368
RE-ENTRY F TECHNOLOGY TESTS	PIR-8151-099
STRUCT MECHANICS RELEASE,STAGE 3	PIR-8156-202
THERMODYNAMICS STAGE 3 RELEASE	TTFM-8151-054
REENTRY F DATA EVALUATION PLAN	67SD5213
THERMAL SENSOR LOCATIONS	PIR-8814-084
RECOMMEND-THERM SENSR SLEEVE DIA.	PIR-8103-1806
REENTRY F ATJ SHELL NOSE	PIR-8225-1634

DESCRIPTION	DOCUMENT NO.
CDRL No. 017 Analytical Reports (Continued)	
AERODYNAMIC AXIS-MISALIGNED VEHS	PIR-8152-2077
REENTRY F TESTS IN 5MW ARC	PIR-8151-324
BILLET-THERMAL, PRESSR SENSORS	SK8634-001C
NOSE TIP, ANT WINDW, CALRMTR BOLT	PIR-8156-3575
FLT TEST DATA PROCESSING PLAN	PIR-8177-1331
THERMODYNAMICS STAGE 3, PART III	TIRM-8151-036
MEASUREMENT LIST STAGE 4	PIR-8174-6970
VEH NOSE TIP THERMODYNAMIC RPT	PIR-8151-666
PLASMA ATTENUA-ELECTRON DENSITIES	PIR-8121-1289
DYNAMIC LOADS STAGE 4 SPACECRAFT	PIR-8156-3576
STAGE 4-SHOCK ENVIRON ON COMPS	PIR-8156-3581
FLT MEASMT SYS ERROR ANALYSIS	PIR-8174-744A
PRELIM DRAFT-INSTRUMENTATION VAN	PIR-83A1-120
THERMODYNAMICS STAGE 3PART III	PIR-8151-703
ASSEMBLY OF REENTRY F NOSE TIPS	PIR-8225-1655
NOSE CAP INSTABILITY EFFECTS	PIR-8152-2290
EFFECTS ON TURB CONVECT HEAT XFER	PIR-8159-1752
SCOUT INTRFCE ALIGNMT, INDEX PROV	PIR-8225-1656
RVSD PLAN-INSTALL PRESS PORT, SENSR	PIR-ABC-8701
TLM-ACOUSTIC NOISE XITION DETECTR	PIR-8174-769
TLM IMPLEMENT-TRANSITION XDUCERS	PIR-8233-2280
CDRL No. 018 Test Results and Summaries	
DESIGN VERIF.-ANTENNA TESTS	PIR-8235-F882
PYROLTIC GRAPHITE TESTS-CAL	PIR-8156-3290
QUAL TEST-TEMP LOCATN SENSORS	SM8156-VR-3321
PRETEST RPT-REENTRY F TEST PGM	67SD419
FLT ASSURANCE TEST-MALTA PIT 1	67SD537
HEAT FLUXES-MALTA TEST FACILITY	62SD201
ACCEPTANCE OF MALTA NOSE TIP	PIR-8103-1976
FLT ASSURANCE TEST-MALTA PIT 1	67SD537ADDENDUM
CDRL No. 020 Finalized Drawings	
FINAL DRAWINGS - SEE DWG. TREE	ER47R197812

DESCRIPTION	DOCUMENT NO.
CDRL No. 021 Field Support Manual	
LAUNCH GO-NO GO CRITERIA	PIR-8173-1004
CDRL No. 026 Reliability Program Plan	
RELIABILITY PROGRAM PLAN	66SD948A
CDRL No. 030 Failure Mode, Effects and Criticality Analyses	
FAIL MODE ANALYSIS-ELECT AGE	PIR-8261-3487
FAIL MODES-EFFECTS-CRITICALITY	PIR-8444-587
FDR-RELIAB ESTIMATES-ANALYSIS-548	PIR-8444-586
CDRL No. 035 Parts and Materials Qualification Test Specifications	
SELECTED PARTS LIST-AVE	S0060-03-0016
SELECTED PARTS LIST-AGE	ERS0060-04-0005
CDRL No. 036 Parts and Materials Application Reviews	
LAB RPT-THERMAL SENSOR 47D172353	PIR-8442-1978
GRAPHITE-AEROTHERMOCHEMICAL BEHAVIOR	R63SD89
GRAPHITE FELT, NON-WOVEN	R6357
POROUS CARBON, POROUS GRAPHITE	R6365
CDRL No. 039 Test Specifications and Procedures	
COMP SPEC-5 VOLT PWR SUPPLY	S6130-15-0015
RE-ENTRY F TEST PROGRAM	66SD9234
ENG TEST PLAN-NOSE VIB. TEST	67SD420
ENG TEST PLAN-EDM-VIB, SHOCK	66SD9348
COMP SPEC-THERMAL SHIELD SENSOR	S6685-51-0016
SPEC-THERMAL TEST SPECIMEN	SK8634-TPE-001
SYSTEM VIBRATION TEST REQMTS	PIR-8156-3254
COMP SPEC-90X10 LL MULTIPLEXER	S5895-65-0007
COMP SPEC-THERMAL SHIELD SENSOR	S6685-51-0016
SYSTEM SI-ANTENNA TEST REENTRY F	SI244414-A
COMP SPEC-SIGNAL DATA CONVERTR	S5895-35-0016
COMP SPEC-TEMP REFERENCE SENSOR	S6685-66-0001

DESCRIPTION	DOCUMENT NO.
CDRL No. 039 Test Specifications and Procedures (Continued)	
SUBSYS SPEC-INSTRUM,COMMUNICATN	S0150-00-0016
SYSTEM SPEC-ELECTRICAL REQMTS	S0010-08-0008
COMP SPEC-90X10 HL PDM COMM	S5895-65-0009
COMP SPEC-0-20 PRESSURE XDUCER	S6685-46-0029
COMP SPEC-VHF SLOT ANT NNA	S5985-31-0019
COMP SPEC-COAX WAVEGDE ADAPTER	S5985-18-0002
COMP SPEC-TLM XMTTR-PWR SUPPLY	S5821-95-0011
COMP SPEC-RATE GYRO, 3 AXIS	S6615-30-0014
COMP SPEC-ACCELEROMETER	S6680-01-0025
COMP SPEC-C-BAND TRANSPONDER	S5821-96-00[illegible]5
COMP SPEC-0-3 PRESSURE XDUCER	S6685-46-0029
COMP SPEC LO PASS RATE GYRO FILTR	S5915-41-0005
COMP SPEC-STORAGE BATTERY 28V	S6140-61-0016
COMP SPEC-SCO ASSY	S5821-71-0007
COMP SPEC-POWER CONTROLLER	S6610-75-0033
COMP SPEC-THERMAL SENSR,AFT COVER	S6630-10-0010
COMP SPEC-VHF POWER DIVIDER	S5985-03-007
COMP SPEC-REFLECTOMETER	S6625-55-0003
COMP SPEC-VHF 4WAY POWER DIVIDER	S5985-03-0007
COMP SPEC-C-BAND POWER DIVIDER	S5985-03-0008
COMP SPEC-0-.1 PRESSURE XDUCER	S6685-46-0028
TEST REQMTS-PRIME REPLCMT BILLETS	PIR-8157-1076
TEST REQMTS-PRIME REPLCMT BILLETS	PIR-8155-1585
CDRL No. 044 Quality Program Plan	
QUALITY PROGRAM PLAN	66SD979
CDRL No. 046 Procurement Specifications	
QAP-RATE GYRO,3 AXIS	QAP-10,578
QAP-PRESSURE TRANSUCER, POT	QAP-10,594
QAP-90X10 MULTICODER HI LVL	QAP-10,585
QAP-PRESSURE TRANSDUCER	QAP-10,593
QAP-90X10 LO LVL MULTIPLEXER	QAP-10,586

DESCRIPTION	DOCUMENT NO.
CDRL No. 046 Procurement Specifications (Continued)	
QAP-SUBCARRIER OSCILLATOR ASSY	QAP-10,582
QAP-BATTERY, STORAGE 28V-5AH	QAP-10-588
QAP-ACCELEROMETER,FORCE BAL	QAP-10,579
QAP-C-BAND TRANSPONDER	QAP-10,580
QAP-THERMAL SENSOR	QAP-10,608
QAP-CALORIMETER	QAP-10,581
QAP-CALORIMETER DETAILS	QAP-10,611
QAP-CALORIMETER	QAP-10,581
CDRL No. 047 Inspection and Test Procedures	
ENG-QC STD, ELECTRONIC EQUIPT	S30042
SI-VENDOR QC INSTRUCTIONS	SI217260
COMP SI PWR CONTROLLER	SI243962
COMP SI TLM XMTTR-PWER SUPPLY	SI243969
COMP SI 0-20 PRESSURE XDUCER	SI243960
COMP SI 0-.1 PRESSURE XDUCER	SI243961
COMP SI VHF SLOT ANTENNA	SI243964
COMP SI 5 VOLT PWER SUPPLY	SI243968
COMP SI REF TEMP SENSOR	SI243980
COMP SI VHF POWER DIVIDER	SI243979
COMP SI LOW PASS RATE GYRO FILTER	SI243987
COMP SI C-BAND TRANSPONDER	SI243970
COMP SI SIGNAL DATA CONVERTER	SI243986
COMP SI COAX WAGEGUIDE ADAPTER	SI243966
COMP SI C-BAND POWER DIVIDER	SI243983
COMP SI THERMAL SHLD SENSOR	SI243981
COMP SI-THERMAL SENSOR,AFT COVER	SI243985
COMP SI REFLECTOMETER	SI243998
CDRL No. 048 End Item Test Plan	
SYSTEMS ACCEPTANCE TEST PLAN	67SD781

DESCRIPTION	DOCUMENT NO.
CDRL No. 049 End Item Test and Inspection Procedures	
SYSTEM SI MASS PROPERTIES WT-C.G.	SI243891
SYSTEM SI MISSION PROFILE	SI243885
SYSTEM SI ANT PATTERN PRE-POST FLT	SI243888
SYSTEM SI ANTENNA VSWR	SI243884
SYSTEM SI SURFACE MAPPING-BANANA	SI243890
SYSTEM SI COUNTDOWN REENTRY F	SI243899
SYSTEM SI FLT ASSUR THERM-ALT	SI243887
SYSTEM SI FLT ASSUR, VIB-SHOCK	SI243886
SUBSYS SI EQUIPT PKG ACC TEST	SI243882
SYSTEM SI SPCRFT-BOOSTER RFI TEST	SI243898
SYSTEM SI MOTION SENSOR CALIB	SI243883
SYSTEM SI DYNAM BAL SPCRFT-SCOUT D	SI243897
SYSTEM SI FINAL CONFIDENCE	SI243889
REENTRY F SYSTEM ACCEPTANCE TESTS	PIR-8363-027B
REENTRY F ACCEPTANCE TEST FLOW	PIR-8363-014037
MECH AGE ACCEPTANCE TEST	SI243893
ELECT AGE ACCEPT TEST, PAD VALIDTN	SI243880
REENTRY F COUNTDOWN HOLDS	PIR-8363-040
SURFACE MEASUREMENT (BANANA TEST)	PIR-8225-2118A
SYSTEM SI BATTERY PREPARE-ACTIVATE	SI243892
NOSE-CALORMTR SUBASSYS ACC TEST	SI243881
CDRL No. 050 Process Control Procedures	
PREP-BOND FWD SECT ASSY OF TVX	MS1237548
BOND VARIOUS MATLS USING PD-162A	MS1237582
BONDING PYROLYTIC GRAPHITE SHIELDS	MS1242379
BOND-PYROLYTIC GRAPHITE-TITANIUM	MS1242384
BOND VARIOUS SUBSTRATES-EPON 934	MS1242392
SHRINK FIT OF SPLINE FTG-CALRMTR	PIR-8634-0002