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Phase B Final Report

Expendable Second Stage
Reusable Space Shuttle Booster

Volume II. Technical Summary

Book 2. Expendable Second Stage Vehicle Definition

Contract NAS9-10960, Exhibit B
DRL MSFC-DRL-221, DRL Line Item 6
DRD MA-078-U2
SD 71-140-2
25 June 1971

Space Division • North American Rockwell 12214 Lakewood Boulevard, Downey, California 90241

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REUSABLE SPACE SHUTTLE BOOSTER

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Technical Summary

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Vice President and General Manager
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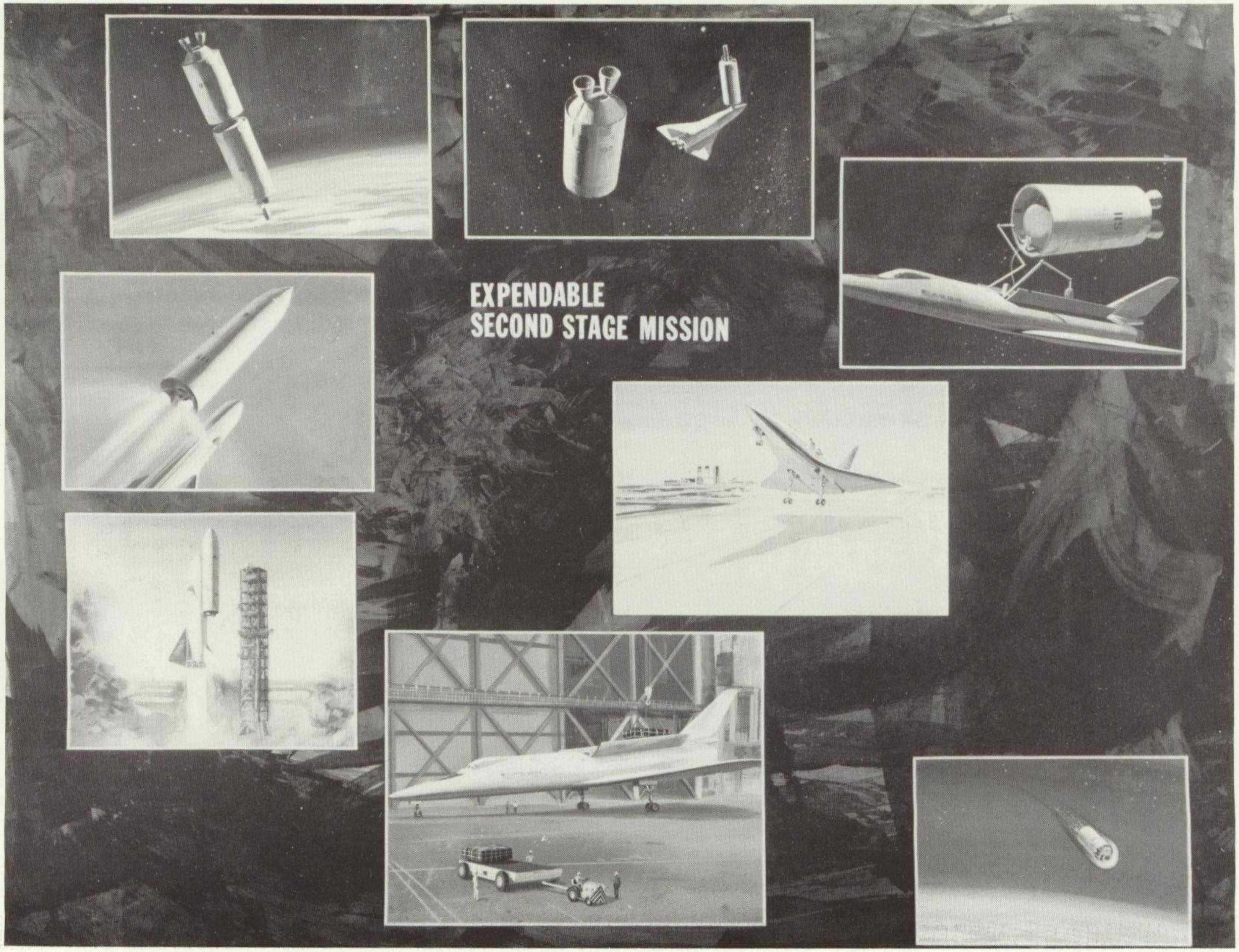
EXPENDABLE
SECOND STAGE MISSION

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FOREWORD

The Space Shuttle Phase B studies are directed toward the definition of an economical space transportation system. In addition to the missions which can be satisfied with the shuttle payload capability, the National Aeronautics and Space Administration has missions planned that require space vehicles to place payloads in excess of 100,000 pounds in earth orbit. To satisfy this requirement, a cost-effective multimission space shuttle system with large lift capability is needed. Such a system would utilize a reusable shuttle booster and an expendable second stage. The expendable second stage would be complementary to the space shuttle system and impose minimum impact on the reusable booster.

To evaluate the expendable second stage concept, a two-phase study was authorized by NASA. Phase A efforts, which ended in December 1970, concentrated on performance, configuration, and basic aerodynamic considerations. Basic trade studies were carried out on a relatively large number of configurations. At the conclusion of Phase A, the contractor proposed a single configuration. Phase B commenced on 1 February 1971, based on the recommended system. Whereas a large number of payload configurations were considered in the initial phase, Phase B was begun with specific emphasis placed on three representative payload configurations. The entire Phase B activity has been directed toward handling the three representative payload configurations in the most acceptable manner with the selected expendable second stage, and toward the design of the subsystems of the expendable second stage. Results of this activity are reported in this 12-volume Phase B final report. This is Volume II, Technical Summary.

Volume I	Executive Summary	SD 71-140-1
Volume II	Technical Summary	SD 71-140-2
Volume III	Wind Tunnel Test Data	SD 71-140-3
Volume IV	Detail Mass Properties Data	SD 71-140-4
Volume V	Operations and Resources	SD 71-140-5
Volume VI	Interface Control Drawings	SD 71-140-6
Volume VII	Preliminary Design Drawings	SD 71-140-7
Volume VIII	Preliminary CEI Specification - Part 1	SD 71-140-8
Volume IX	Preliminary System Specification	SD 71-140-9
Volume X	Technology Requirements	SD 71-140-10
Volume XI	Cost and Schedule Estimates	SD 71-140-11
Volume XII	Design Data Book	SD 71-140-12

Volume II, Technical Summary, is divided into three books:

Book 1 Expendable Second Stage/Reusable Booster System Definition

Book 2 Expendable Second Stage Vehicle Definition

Book 3 Booster Vehicle Modifications and Ground Systems Definition.

This book is intended to be used together with the other books of Volume II. Book 1 contains basic data on mission/system requirements, performance, trajectories, aerodynamics, stability and control, loads, heating, and acoustic environment. Book 2 is devoted to the definition of the selected expendable second stage, its subsystems, and overall ESS operation. Book 3 covers the definition of the ESS/booster separation system, modifications required on the reusable booster for ESS/payload flight, and the ground systems needed to operate the ESS complementary with the space shuttle.

CONTENTS

Section		Page
1.0	INTRODUCTION AND SUMMARY	1-1
	1.1 General Arrangement	1-3
	1.2 ESS Weight and Mass Properties	1-7
	1.3 Payload Attachment	1-7
2.0	ESS DESIGN	2-1
	2.1 Structural Subsystem Group	2-1
	2.1.1 Structural Criteria and Loads	2-1
	2.1.2 Vehicle Structures	2-25
	2.1.3 Thermal Protection System	2-69
	2.2 Propulsion Subsystem Group	2-91
	2.2.1 Main Propulsion Subsystem	2-91
	2.2.2 Auxiliary Propulsion Subsystem	2-201
	2.3 Avionics Subsystem Group	2-223
	2.3.1 Guidance, Navigation, and Control Subsystem	2-239
	2.3.2 Data and Control Management	2-301
	2.3.3 Communications Subsystem	2-328
	2.3.4 Electrical Power and Distribution	2-358
	2.3.5 Electrical Controls	2-376
	2.3.6 Instrumentation	2-434
	2.3.7 Avionics Equipment Installation and Environmental Control	2-442
	2.3.8 Recovery and Deorbit Subsystem	2-453
	2.4 ESS Vehicle Installation, Assembly, and Checkout	2-468
	2.4.1 Major Vehicular Installation	2-468
	2.4.2 Final Assembly and Installations	2-468
	2.4.3 ESS Acceptance Checkout	2-469
	2.4.4 Manufacturing Facilities	2-469
	2.5 ESS System Support	2-471
	2.5.1 Support Equipment	2-471
	2.5.2 Support Software	2-481
	2.6 Wind Tunnel Models	2-482

Section		Page

3.0 RELIABILITY 3-1
3.1 Summary 3-1
3.2 Failure Mode Analysis and Hardware Criticality 3-2
3.2.1 Main Propulsion Subsystem 3-4
3.2.2 Auxiliary Propulsion Subsystem . . . 3-18
3.2.3 Avionics Subsystems 3-23
3.3 Maintainability 3-28
3.4 Manufacturability 3-29

4.0 TRANSPORTATION—SEAL BEACH TO KSC . . . 4-1

APPENDIXES
A. ENGINE ACTUATION TRADE STUDY A-1
B. ORBIT MANEUVERING SUBSYSTEM TRADE STUDY B-1
C. DEORBIT TRADE STUDY C-1
D. RECOVERABLE AVIONICS TRADE STUDY . . D-1

ILLUSTRATIONS

Figure		Page
1-1	Expendable Second-Stage Payloads	1-4
1-2	General Arrangement, ESS	1-5
1-3	Payload Separation Subsystem	1-11
1-4	Space Tug Adapter	1-15
2-1	ESS Structural Configuration	2-2
2-2	ESS Forward Bulkhead Temperature Distribution, LH_2 Loading: 450,000 Pounds (Tug) to 675,000 Pounds (MDAC)	2-6
2-3	ESS/RNS Max $q\alpha$ Pitch Plane Body Loads (Headwind)	2-8
2-4	ESS/RNS Max $q\alpha$ Axial Body Loads (Headwind and Tailwind)	2-9
2-5	ESS/MDAC Max $q\alpha$ Pitch Plane Body Loads	2-10
2-6	ESS/MDAC Max $q\alpha$ Axial Body Loads	2-11
2-7	ESS/RNS Max $q\alpha$ Pitch Plane Body Loads (Tailwind Case)	2-12
2-8	ESS/MDAC Max $q\alpha$ Pitch Plane Body Loads (Tailwind Case)	2-13
2-9	ESS/MDAC Max $q\alpha$ Axial Body Loads (Tailwind Case)	2-14
2-10	ESS/RNS Max $q\alpha$ Trim Yaw Body Loads (Tailwind Case)	2-15
2-11	ESS/RNS Max $q\alpha$ Trim Pitch Body Loads	2-16
2-12	ESS/RNS Max $q\alpha$ Trim Axial Body Loads (Alpha Trim Case)	2-17
2-13	ESS/MDAC Max $q\alpha$ Trim Yaw Body Loads	2-18
2-14	ESS/MDAC Max $q\alpha$ Trim Pitch Body Loads	2-19
2-15	ESS/MDAC Max $q\alpha$ Trim Axial Body Loads	2-20
2-16	ESS/RNS End Boost Normal Body Loads	2-21
2-17	ESS/RNS End Boost Axial Body Loads	2-22
2-18	ESS/MDAC End Boost Normal Body Loads	2-23
2-19	ESS/MDAC End Boost Axial Body Loads	2-24
2-20	S-II Applied Compressive Loading Comparison, ESS With Payloads	2-26
2-21	Summary of Major S-II Stage Structural Modifications	2-27
2-22	Forward Skirt Layout	2-29
2-23	Forward Skirt Frame, Station 831, Internal Loads	2-32
2-24	ESS LH_2 Tank	2-35
2-25	ESS LH_2 Forward Bulkhead Limit Pressure Capability	2-36

Figure		Page

2-26 ESS LH_2 Tank Pressure Capability 2-38
2-27 ESS LO_2 Tank 2-39
2-28 ESS LO_2 Tank Burst Pressure Capability 2-41
2-29 Aft Skirt Layout 2-43
2-30 ESS MDAC Aft Skirt Shell Loading Intensity/MDAC Payload 2-46
2-31 ESS Aft Skirt Shear Flow (MDAC Payload) 2-48
2-32 ESS/Frame, Station 31, Internal Loads 2-50
2-33 ESS/Frame, Station 174.5, Internal Loads 2-53
2-34 Booster Attach Fitting, Aft Skirt 2-55
2-35 Thrust Structure Layout 2-57
2-36 Thrust Structure Shell Loading Intensity 2-60
2-37 Thrust Structure Skin Shear 2-61
2-38 Frame X_B 52 Bending Moment 2-63
2-39 Frame X_B 52 Axial Load 2-64
2-40 Frame X_B 52 Shear Load 2-65
2-41 Frame X_B 105 Bending Moment 2-66
2-42 Frame X_B 105 Axial Load and Shear 2-67
2-43 Thrust Cone Longeron 2-68
2-44 Cross-Tie Beam, X_B 52 and Support Bending Moment . 2-70
2-45 Insulation General Arrangement 2-73
2-46 Heat Shield Layout 2-75
2-47 Skin Temperature Due to Aerodynamic Heating During Boost 2-77
2-48 ESS Temperature Histories With RNS Payload for Region B of Figure 2-47 2-78
2-49 ESS Temperature Histories With MDAC Space Station Payload for Region B of Figure 2-47 2-79
2-50 ESS Temperature Histories With Space Tug Payload for Region B of Figure 2-47 2-80
2-51 ESS Base Heating Temperatures 2-82
2-52 Typical Intersection of Panels 2-87
2-53 Heat Shield Drawing 2-89
2-54 Main Propulsion Subsystem Schematic 2-92
2-55 ESS Space Allotment, General Arrangement 2-97
2-56 ESS Propellant Feed Subsystem Layout 2-99
2-57 LO_2 Residual at Depletion Cutoff, Normal Depletion and Engine Out 2-103
2-58 ESS Engine Servicing Subsystem Layout 2-105
2-59 ESS Pressurization Subsystem Layout 2-107
2-60 ESS LH_2 Tank Ullage Pressure, 450 K Loading . . . 2-136
2-61 ESS LH_2 Tank Ullage Pressure, 675 K Loading . . . 2-137
2-62 ESS LO_2 Tank Ullage Pressure, 450 K Loading . . . 2-138

Figure		Page
2-63	ESS LO_2 Tank Ullage Pressure, 675 K Loading	2-139
2-64	ESS 450 K Propellant Load, LH_2 Ullage Pressure During Shuttle Boost	2-140
2-65	ESS 450 K Propellant Load, LO_2 Ullage Pressure During Shuttle Boost	2-141
2-66	ESS 675 K Propellant Load, LH_2 Ullage Pressure During Shuttle Boost	2-142
2-67	ESS 675 K Propellant Load, LO_2 Ullage Pressure During Shuttle Boost	2-143
2-68	ESS Forward Bulkhead Temperature Distribution, LH_2 Loading	2-145
2-69	ESS 450 K Propellant Load, LH_2 Engine Inlet Conditions (Based on Minimum Ullage Pressure)	2-146
2-70	ESS 450 K Propellant Load, LO_2 Engine Inlet Conditions (Based on Minimum Ullage Pressure)	2-147
2-71	ESS 675 K Propellant Load, LH_2 Engine Inlet Conditions (Based on Minimum Ullage Pressure)	2-148
2-72	ESS 675 K Propellant Load, LO_2 Engine Inlet Conditions (Based on Minimum Ullage Pressure)	2-149
2-73	ESS LH_2 Tank Ullage Mass Prepressure, 34-36 Psia . .	2-151
2-74	ESS LO_2 Tank Ullage Mass Prepressure, 34-36 Psia . .	2-152
2-75	Expendable Second Stage TVC	2-155
2-76	Typical Servoactuator	2-158
2-77	Typical Engine-Driven Pump	2-160
2-78	Typical Accumulator, Reservoir Manifold Assembly (ARMA)	2-161
2-79	ESS Engine Compartment Conditioning System . . .	2-167
2-80	ESS Main Propulsion System Operating	2-169
2-81	ESS Recoverable Engine Stage/Engine Connect Panel Layout	2-173
2-82	Operational Sequence for ESS Docking	2-186
2-83	Proposed Attachment of Manipulator Arm to Orbiter Engine	2-187
2-84	Proposed Attachment of Manipulator Arm to Orbiter Engine	2-189
2-85	ESS Orbiter Engines Stored Vertical	2-192
2-86	ESS Orbiter Engines Stored Horizontal	2-193
2-87	ESS Recoverable Hardware Horizontal/Vertical Engine Orientation	2-195
2-88	Impingement of Separated Flow on ESS Nozzles . . .	2-197
2-89	Nozzle Impingement Geometry	2-199
2-90	Auxiliary Propulsion Subsystem Schematic	2-203

Figure		Page
2-91	ESS Installation, ACPS	2-205
2-92	ACPS Conditioning System Control Sequence	2-210
2-93	ESS OMS and APS Tankage Fuel Lines Layout	2-213
2-94	Orbit Maneuvering System Start Sequence	2-222
2-95	Rationale for Integrated Systems Selection	2-224
2-96	Avionics Hardware Sources	2-226
2-97	ESS Avionics Subsystem Block Diagram	2-229
2-98	Avionics Equipment General Arrangement	2-233
2-99	Recoverable Avionics Equipment	2-237
2-100	ESS/Booster Avionics Interface	2-238
2-101	ESS GN&C Functional Block Diagram (All Phases)	2-252
2-102	Prelaunch Phase GN&C Functional Block Diagram	2-253
2-103	Mated Boost GN&C Functional Block Diagram	2-254
2-104	Separation/ESS Insertion/Booster Return Maneuver	2-255
2-105	Orbit Operations (Including Rendezvous and Stationkeeping)	2-256
2-106	Baseline GN&C Subsystem	2-258
2-107	IMU Block Diagram	2-259
2-108	Rate Sensor Schematic Diagram	2-261
2-109	Typical ACPS/OMS Engine Driver Mechanization (4-Engine Configuration)	2-263
2-110	Typical TVC Gimbal Control Channel	2-265
2-111	Inertial Error Propagation With No Navigation Updates	2-273
2-112	Navigation Position Accuracy Versus Time for Multilateration Update	2-274
2-113	Navigation Position Accuracy Versus Time for MSFN Tracking Updates	2-276
2-114	Direct Ascent From Parking Orbit to Final Orbit Insertion	2-280
2-115	RSS Position Errors for Inertial Navigation	2-283
2-116	Orbital Transfer and Terminal Phase Position Errors Using Multilateration Updates	2-286
2-117	Orbital Transfer and Terminal Phase Position Errors Using Minimum MSFN Tracking	2-287
2-118	Phasing Time Required, Each Launch Opportunity	2-291
2-119	Phasing Time Required, Composite Using Both Launch Opportunities	2-292
2-120	Phasing Time Versus Target Orbit Variations, Baseline Mission	2-293
2-121	Phasing Time Versus Target Orbit Variations, RNS Mission	2-294
2-122	Orbits for Zero-Phase Change	2-295

Figure		Page

Figure		Page
2-123	Orbital Altitude Versus Launch Delay and Phasing Time	2-297
2-124	Synchronous Orbit Trajectory	2-298
2-125	DCM Subsystem Schematic Diagram	2-307
2-126	Data Bus Message Formats	2-310
2-127	Failure Criteria, Electrical and Mechanical Systems	2-318
2-128	Redundancy Management	2-319
2-129	Data Bus Flow Diagram	2-321
2-130	Logic Flow, Close Prevalve Command	2-323
2-131	Thrust Vector Control Redundancy	2-325
2-132	Motor-Driven Switch Schematic Diagram	2-327
2-133	Motor-Driven Switch Source and Load Monitoring	2-327
2-134	Motor-Driven Switch, Load Monitoring	2-329
2-135	Motor-Driven Switch Isolation Technique	2-329
2-136	Motor-Driven Switches, Total Need for Redundancy	2-330
2-137	ESS Communication Block Diagram	2-331
2-138	ESS Communications Operational Timelines	2-332
2-139	Range Safety Command Subsystem	2-335
2-140	Propellant Dispersion System Block Diagram	2-336
2-141	Up-Data Decoder Block Diagram	2-338
2-142	Modulation and Switching Unit	2-340
2-143	USBE Block Diagram	2-342
2-144	S-Band Power Amp Block Diagram	2-344
2-145	Range Safety Command Receiver	2-346
2-146	Range Safety Command Decoder	2-347
2-147	Existing Apollo S-Band Communication System	2-351
2-148	Redundant S-Band Communication System	2-353
2-149	ESS Main Electrical Power System (+28 vdc Main)	2-361
2-150	ESS Main (+28 VDC) Load Profile (Distribution, Three Buses)	2-363
2-151	ESS AC Load Profile (Main Engine Operation)	2-364
2-152	56-vdc Bus Distribution System Schematic	2-367
2-153	115/200 VAC Bus Distribution System Schematic	2-369
2-154	ESS +28 VDC Main Power Grounding	2-372
2-155	ESS +56 VDC Inverter Power Grounding	2-373
2-156	ESS AC Power Grounding	2-374
2-157	Data Bus Command/Response Signal Return	2-375
2-158	Main Propulsion Electrical Controls	2-379
2-159	Main Propulsion Electrical Control, Power	2-381
2-160	Engine Control System Redundancy Checkout Measurement	2-383
2-161	ESS Auxiliary Propulsion Electrical Controls (Triple Redundant Valve Control).	2-386
2-162	Attitude Control Propulsion System Electrical Control Driver Unit Mechanization	3-388

Figure		Page

2-163 Auxiliary Propulsion System Attitude Control Propulsion System Electrical Controls 2-389
2-164 ESS Auxiliary Propulsion System (Attitude Control) . . 2-390
2-165 Typical FO/FS Fill Valve Control Block Diagram . . 2-395
2-166 Typical Prevalve Control Block Diagram 2-397
2-167 Typical Recirculation Valve and Helium Injection Solenoid Control Block Diagram 2-398
2-168 Pogo Accumulator Fill Valve Control Block Diagram . . 2-399
2-169 Pogo Accumulator Bleed Valve Control Block Diagram . 2-400
2-170 Typical Vent Valve Control Block Diagram 2-404
2-171 Typical LO_2 / LH_2 Pressurization Solenoid Control Block Diagram 2-406
2-172 Propellant Loading Electronics 2-408
2-173 Propellant Management System Main LO_2 and LH_2 Tank Sensors 2-411
2-174 APS Propellant Level Monitoring System 2-412
2-175 Point Sensor Level Monitoring 2-413
2-176 Typical Point Sensor Control Block Diagram . . . 2-414
2-177 Separation Electrical Controls 2-418
2-178 Safing Electrical Controls (Main LO_2 Tank) 2-421
2-179 Safing Electrical Controls (Main LH_2 Tank) 2-422
2-180 Main Propulsion System Engines, Thrust Vector Electrical Controls 2-427
2-181 AMS Engines, Thrust Vector Electrical Controls . . . 2-429
2-182 Typical Digital Signal Conditioned Measurement . . . 2-440
2-183 Typical Vibration Measurement 2-441
2-184 Avionics Container Installation, ESS 2-451
2-185 Avionics Container Separation (Ordnance Train) . . . 2-454
2-186 Deorbit Package Block Diagram 2-457
2-187 Typical EBW Separation Control Block Diagram . . . 2-458
2-188 Typical Control Circuit for Main Engine Recovery Separation 2-459
2-189 IMU Container 2-462
2-190 Deorbit/Recovery Schematic 2-465

TABLES

Table		Page
1-1	Systems Mass Properties, ESS/MDAC Space Station	1-8
1-2	Systems Mass Properties, ESS/NR Tug	1-9
1-3	Systems Mass Properties, ESS/RNS	1-10
2-1	Weight and Center-of-Gravity Data for Loads Analysis	2-5
2-2	ESS LH_2 Tank Limit Deflections Summary	2-84
2-3	Space Shuttle Orbiter Engine Characteristics	2-95
2-4	Propellant Feed Assembly Fail-Operational/Fail-Safe Considerations	2-134
2-5	Pressurization System Differences Between S-II and ESS	2-135
2-6	Engine Servicing Assembly Fail-Operational/Fail-Safe Considerations	2-153
2-7	Propellant Dump Conditions	2-164
2-8	Safing Assembly Fail-Operational/Fail-Safe Considerations	2-165
2-9	Impingement Loads and Moments for ESS Nozzle	2-200
2-10	ACPS Thruster Characteristics	2-207
2-11	ACPS Fail-Operational/Fail-Safe Considerations	2-208
2-12	Orbit-Maneuvering Engine Characteristics	2-212
2-13	Propellant Budget, APS Baseline Mission-MDAC Space Station	2-217
2-14	Propellant Margin for Contingency, APS Baseline Mission-MDAC Space Station	2-218
2-15	OMS Fail-Operational/Fail-Safe Considerations	2-220
2-16	Conventional Versus Integrated Avionics Subsystem	2-225
2-17	ESS Avionics Equipment Redundancy Provisions	2-231
2-18	Deorbit Considerations	2-234
2-19	Forecased Hardware Savings	2-235
2-20	GN&C Performance Requirements	2-244
2-21	GN&C Functional Requirements (Nonrendezvous Baseline Mission)	2-245
2-22	GN&C Functional Requirements (Rendezvous Baseline Mission	2-249
2-23	LRU Characteristics Summary	2-268
2-24	MSFN, Multilateration, and TDRS Tracking Coverage for ESS 100-NM Parking Orbit	2-275
2-25	ESS Deletions From Shuttle Total Requirements	2-313
2-26	ESS Development Deltas From Shuttle Summary	2-314

Table		Page
2-27	DMC Memory Requirements	2-315
2-28	DMC Speed Requirements	2-315
2-29	Power Budget Parametric Equations and Constants . .	2-355
2-30	Receiver Noise Power for Subcarriers	2-355
2-31	Communication Modes and Mission Phase	2-356
2-32	Signal to Noise and Signal Margins	2-357
2-33	56-VDC Bus Loading	2-366
2-34	Main 28-VDC Bus Loading	2-371
2-35	Hydraulic Fluid Operating Limits	2-435
2-36	ESS Measurement Requirements	2-437
2-37	ESS Transducer and Signal Conditioning Requirements .	2-438
2-38	ESS Avionics Equipment	2-445
2-39	Surface Coatings Considered for Thermal Control Applications	2-449
2-40	Avionics Systems Installation	2-450

1.0 INTRODUCTION AND SUMMARY

1.0 INTRODUCTION AND STUDY OBJECTIVES

A space shuttle system utilizing a reusable booster and orbiter that can transport personnel and cargo to low earth orbit and return personnel and cargo safely to earth is being evaluated through a preliminary design and development (Phase B) study. Future space exploration and operations will be greatly enhanced by the basic economic efficiency which will evolve through development of such a space logistics system. The shuttle system presently is being designed to handle payloads up to approximately 65,000 pounds. The cargo bay into which these payloads must fit is cylindrical and is 60 feet long and 15 feet in diameter Whereas most of the total projected payload spectrum can be handled by the shuttle, a significant number of payloads for the decade of the 1980's have been defined as much larger and heavier than can be placed in orbit by the booster/orbiter combination.

To supplement the shuttle capability, a space shuttle system using an expendable second stage (ESS) with a reusable space shuttle booster has been under investigation for the past nine months of the Phase B Study. The prime objective of this supplemental study has been to determine the feasibility, cost effectiveness, and preliminary design of a system to be suitable for a wide variety of advanced space missions beginning in the last half of CY 1979. The results of the study will be available for comparison with results of past studies of all-expendable systems for launching the same types of large, heavy payloads

On 14 September 1970, NASA authorized the Phase A/B Study for an Expendable Second Stage on a Reusable Space Shuttle Booster, as Exhibit B to Contract NAS9-10960. Supplementary study during Phase A on a modified Saturn S-II as a candidate ESS was authorized by Change Order 1980 to Contract NAS7-200. During Phase B, which commenced on 1 February 1971, supplementary study was defined by Task Authorization 5 to Contract NAS7-200. Beginning in March 1971, additional supplementary or related technical effort by the IBM Corporation, on guidance, navigation, and control for the ESS was authorized by NASA. Results of this effort are utilized in this report. The entire final report is organized in accordance with Contract NAS9-10960, DRL MSFC-DRL-221, Line Item 6, and DRD MA-078-U2, 28 August 1970. The document is submitted by North American Rockwell Corporation through its Space Division and contains results of design, performance, and resource studies performed during the Phase B portion of the contract.

As previously indicated, the study was divided into two sequential phases, Phase A and Phase B. The Phase A work concentrated upon such aspects as performance, configuration, and basic aerodynamic considerations to provide the foundation for a proposed single configuration for a analysis in the Phase B portion. Phase A required analysis and definition of space shuttle systems with an optimized expendable second stage (1) utilizing existing hardware, (2) space shuttle engines with 400, 000 pounds of thrust*, and (3) new hardware or (4) a combination of existing and new hardware. Further, the definitions of systems with minimum modified S-II stages and minimum modified S-IVB stages were included.

The study depth was to be sufficient to permit a decision by NASA on whether to proceed with a particular approach or to eliminate all concepts from further consideration. To accomplish this objective, consideration was given to the following:

1. The defined payload spectrum.

2. The required operational characteristics.

3. Identification of any modifications and the extent of penalties (if any) in payload and performance required to employ the reusable booster with the selected expendable second stage (but without incorporation into the Phase B space shuttle system study).

4. Research, design, test and evaluation, production, and operational costs.

5. Identification of cost, performance, and mission effectiveness.

The contractor recommended that an ESS-derived from the Saturn S-II be investigated in the Phase B portion of the study. The recommended ESS main propulsion system featured two space shuttle orbiter engines and its liquid hydrogen tank was shortened by 99 inches. The results of the Phase A study were reported in December 1970 in the Interim Final Report (Phase A only), SD 70-607. A summary of these results is included in Volume I, Executive Summary, of this report.

*The thrust level for Phase B was increased to 550,000 pounds at sea level.

This volume of the final report, Volume II, Technical Summary, covers the selected expendable second stage/reusable booster system and consists of the following:

1. Study objectives
2. Mission/System requirements
3. Study Approach
4. System and flight characteristics and environment
5. Analyses and design data generated during the Phase B study

On 1 February 1971, a technical directive was received from NASA that indicated for the remaining portions of the Phase B Study emphasis should be placed on the short S-II stage with two space shuttle engines. Further, to facilitate an in-depth study up to mid-June 1971, baseline payloads should be consolidated into three. The specific payloads are indicated in Figure 1-1, along with the candidate payload spectrum for the Phase B study. Other ground rules to be used in Phase B were defined in an updated study control document dated 1 February 1971 and are included in Section 2.0, Book 1, Volume II.

1.1 GENERAL ARRANGEMENT

The ESS is a modified Saturn V S-II. The primary structure includes the S-II LO_2 tank, the LH_2 tank with one 99-inch cylinder removed, a new thrust structure, a new aft skirt, and a modified forward skirt. The forward and aft skirts contain provisions for mounting the ESS to the B-9U booster. The ESS general arrangement is shown in Figure 1-2.

The main propulsion system employs two space shuttle orbiter engines mounted to the new thrust structure. Air-stream deflectors are attached to the aft skirt to avoid direct airstream impingement on the engines.

An auxiliary propulsion system employing two 10,000-pound-thrust OMS engines is also attached to the thrust structure. This system is required to perform the circularization maneuver after insertion into orbit and to provide deorbit impulse. Fourteen 2,100-pound-thrust ACPS (attitude control propulsion system) thrusters are mounted in the two LH_2 feedline fairings to provide vehicle pitch, yaw, and roll control. These ACPS thrusters and OMS engines are fed by four LH_2 and one LO_2 tank mounted in the thrust structure.

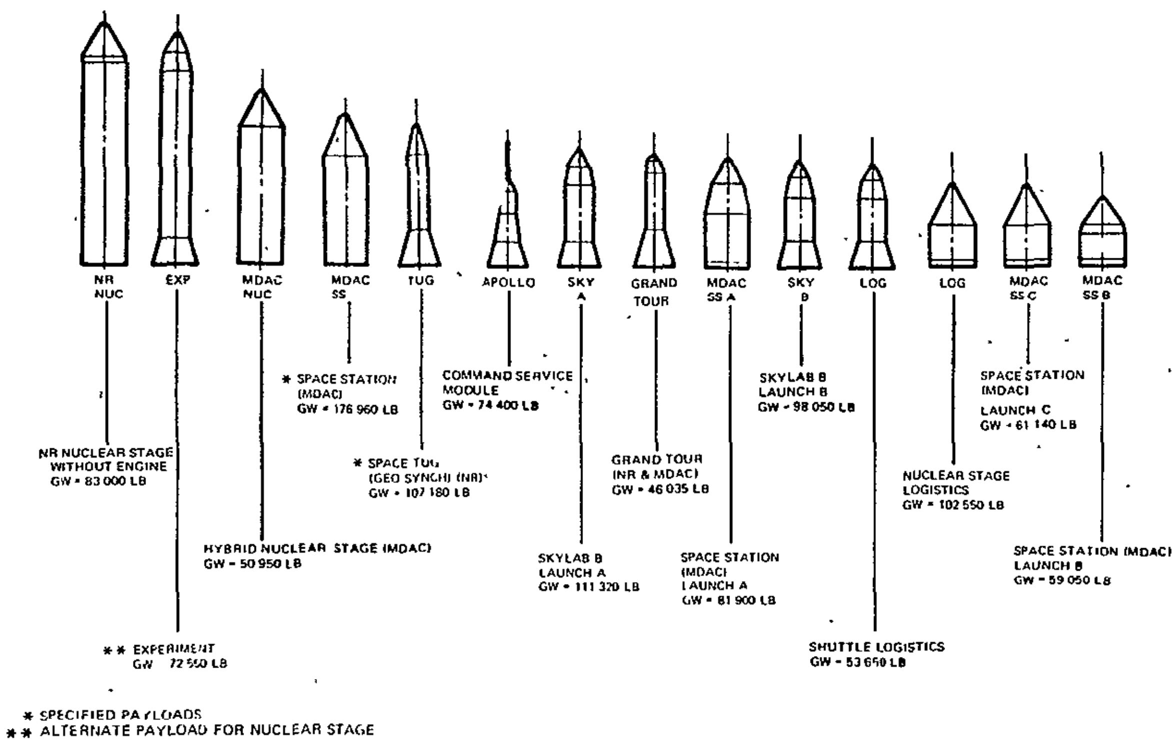


Figure I-1. Expendable Second-Stage Payloads

A docking port is integrated into the LH_2 feedline fairing at Position III. This docking port is provided to mate with the orbiter for equipment retrieval prior to deorbit.

The avionic equipment used for guidance, navigation, and control is located in both the forward and aft skirts and on the thrust structure. Some of the more expensive avionic equipment is located in the aft end of the ESS to facilitate equipment recovery.

1.2 ESS WEIGHT AND MASS PROPERTIES

ESS weight and mass properties are summarized in Tables, 1-1, 1-2 and 1-3. Further weight data and mass properties are delineated in Volume 4 of this report.

1.3 PAYLOAD ATTACHMENT

Two payload attachment structures have been identified for the three payload configurations. The 33-foot-diameter MDAC space station and reusable nuclear stage (RNS) require a separation plane at the ESS/ payload interface. The space tug requires a transition cone to reduce from the ESS 33-foot diameter to the 15-foot diameter of the tug. Separation of the space tug is required at the transition cone and payload interface.

The method of structural separation selected for the MDAC space station and the RNS is a confined explosive charge contained in a short cylindrical segment. This segment would be added between the ESS forward skirt and the payload aft skirt (see Figure 1-3). The separation segment is made up of two sections, one forward and one aft of the separation plane. These sections are 7075-T6 aluminum alloy channels with integrally machined stiffeners. Compression loads are transferred from the ESS forward interface angle through the separation segment to the payload. Tension loads are transmitted from the payload skirt stringers to the separation segment by stringer end fittings. Tension straps are employed to transfer the load across the separation plane. Radial shear loads are transferred through shear pins in the flange at the separation plane.

Separation is made by detonation of confined explosive charges to sever the tension straps. It is proposed to use an explosve charge contained within a continuous thin steel housing. The original cross-sectional shape of the charge assembly is a flat oval. After ignition, the covering sheath expands to a circular cross-section. The deformation of the covering sheath breaks the tension straps.

Table 1-1. Systems Mass Properties, ESS/MDAC Space Station

SYSTEMS MASS PROPERTIES

CONFIGURATION: ESS/MDAC SPACE STATION | BY | DATE | PAGE OF

NO	SYSTEM	WEIGHT LB.	CENTER OF GRAVITY INCHES X	Y	Z	MOMENT OF INERTIA SLUG FT² X 10⁻⁶ I_{x-x}	I_{y-y}	$I_{z\,z}$	PRODUCT OF INERTIA SLUG FT² X 10– I_{xy}	I_{xz}	I_{yz}
1	Wing group										
2	Tail group										
3	Body	57,847	344.3	1.0	3.0	0.40	1.03	1.02			
4	Induc env protect										
5	Landing & docking										
6	Ascent propulsion	25,328	72.0	5.4	- 1.3	0.06	0.17	0.20			
7	Cruise propulsion										
8	Auxiliary propulsion	4710	68.3	23.2	-28.0	0.02	0.02	0.01			
9	Prime power	3400	188.6	0.0	0.0	-	0.01	0.01			
10	Electrical conv & dist	1190	203.4	3.7	14.0	-	0.01	0.01			
11	Pneumatic conv & dist	406	215.7	9.5	3.2	-	-	-			
12	Surface controls										
13	Avionics	2955	323.5	-10.1	-51.4	0.02	0.06	0.06			
14	Enviro control	1100	376.7	0.0	0.0	-	0.03	0.03			
15	Personnel provisions										
16	Range safety										
17	Ballast										
18	Growth										
19											
	Subtotal (dry weight)	96,936	251.8	2.9	- 1.3	0.55	1.65	1.67			
20	Personnel										
21	Cargo/payload	176,900	1355.0	0.0	0.0	0.97	5.05	5.05			
22	Ordnance										
23	Residual fluids	5965	280.0	0.0	0.0	-	0.04	0.04			
24	Payload margin	5330	1355.0	0.0	0.0	0.03	0.15	0.15			
	Subtotal (inert weight)	285,191	957.5	1.0	- 0.4	1.56	24.13	24.15			
25	Reserve fluids	6659									
26	Inflight losses	83	55.6	0.0	0.0	-	-	-			
27	Propellant - ascent	677,150	288.1	0.0	0.0	-	1.90	1.90			
28	Propellant - cruise										
29	Propellant - maneuv/acs	22,917	55.6	0.0	0.0	0.02	0.01	0.01			
30											
	Total (gross wt) lb	992,000	475.2	0.3	- 0.1	1.58	46.40	46.42			

Table 1-2. Systems Mass Properties, ESS/NR Tug

FORM ... NEW 8-70

SYSTEMS MASS PROPERTIES

CONFIGURATION: ESS/NR TUG | BY | DATE | PAGE OF

NO	SYSTEM	WEIGHT LB.	CENTER OF GRAVITY INCHES X	Y	Z	MOMENT OF INERTIA SLUG FT^2 X 10^{-6} I_{xx}	I_{yy}	I_{zz}	PRODUCT OF INERTIA SLUG FT^2 X 10- I_{xy}	I_{xz}	I_{yz}
1	Wing group										
2	Tail group										
3	Body	57,847	344.3	1.0	3.0	0.40	1.03	1.02			
4	Induc env protect										
5	Landing & docking										
6	Ascent propulsion	25,328	72.0	5.4	- 1.3	0.06	0.17	0.20			
7	Cruise propulsion										
8	Auxiliary propulsion	4710	68.3	23.2	-28.0	0.02	0.02	0.01			
9	Prime power	3400	188.6	0.0	0.0	-	0.01	0.01			
10	Electrical conv & dist	1190	203.4	3.7	14.0	-	0.01	0.01			
11	Pneumatic conv & dist	406	215.7	9.5	3.2	-	-	-			
12	Surface controls										
13	Avionics	2955	323.5	-10.1	-51.4	0.02	0.06	0.06			
14	Enviro control	1100	367.7	0.0	0.0	-	0.03	0.03			
15	Personnel provisions										
16	Range safety										
17	Ballast										
18	Growth										
19											
	Subtotal (dry weight)	96,936	251.8	2.9	- 1.3	0.55	1.65	1.67			
20	Personnel										
21	Cargo/payload	107,180	1415.0	0.0	0.0	0.09	0.76	0.76			
22	Payload adapter	10,000	1013.0	0.0	0.0	0.04	0.05	0.05			
23	Residual fluids	5965	280.0	0.0	0.0	-	0.04	0.04			
24	Payload margin	9413	1415.0	0.0	0.0	-	0.07	0.07			
	Subtotal (inert weight)	229,494	876.7	1.2	- 0.5	0.70	18.53	18.55			
25	Reserve fluids	18,487									
26	Inflight losses	83	55.6	0.0	0.0	-	-	-			
27	Propellant - ascent	496,420	269.6	0.0	0.0	-	1.18	1.18			
28	Propellant - cruise										
29	Propellant - Maneuv/acs	10,401	55.6	0.0	0.0	0.01	-	-			
30											
	Total (gross wt) lb	754,885	451.2	0.4	- 0.2	0.70	32.70	32.72			

Table 1-3. Systems Mass Properties, ESS/RNS

FORM [illegible] NEW 8-70

SYSTEMS MASS PROPERTIES

CONFIGURATION ESS/RNS	BY	DATE	PAGE OF

NO.	SYSTEM	WEIGHT LB	CENTER OF GRAVITY INCHES X	Y	Z	MOMENT OF INERTIA SLUG FT² X 10⁻⁶ I_{xx}	I_{yy}	I_{zz}	PRODUCT OF INERTIA SLUG FT² X 10– I_{xy}	I_{xz}	I_{yz}
1	Wing group										
2	Tail group										
3	Body	57,847	344.3	1.0	3.0	0.40	1.03	1.02			
4	Induc env protect										
5	Landing & docking										
6	Ascent propulsion	25,328	72.0	5.4	- 1.3	0.06	0.17	0.20			
7	Cruise propulsion										
8	Auxiliary propulsion	4710	68.3	23.2	-28.0	0.02	0.02	0.01			
9	Prime power	3400	188.6	0.0	0.0	-	0.01	0.01			
10	Electrical conv & dist	1190	203.4	3.7	14.0	-	0.01	0.01			
11	Pneumatic conv & dist	406	215.7	9.5	3.2	-	-	-			
12	Surface controls										
13	Avionics	2955	323.5	-10.1	-51.4	0.02	0.06	0.06			
14	Enviro control	1100	376.7	0.0	0.0	-	0.03	0.03			
15	Personnel provisions										
16	Range safety										
17	Ballast										
18	Growth										
19											
	Subtotal (dry weight)	96,936	251.8	2.9	- 1.3	0.55	1.65	1.67			
20	Personnel										
21	Cargo/payload	83,000	179.0	0.0	0.0	0.70	5.03	5.03			
22	Ordnance										
23	Residual fluids	5965	280.0	0.0	0.0	-	0.04	0.04			
24	Payload mission	9314	179.0	0.0	0.0	0.08	0.55	0.55			
	Subtotal (inert weight)	195,215	980.1	1.4	- 0.6	1.34	32.09	32.11			
25	Reserve fluids	10,480									
26	Inflight losses	83	55.6	0.0	0.0	-	-	-			
27	Propellant - ascent	428,198	260.0	0.0	0.0	-	0.94	0.94			
28	Propellant - cruise										
29	Propellant - maneuv/acs	17,724	55.6	0.0	0.0	0.02	0.01	0.01			
30											
	Total (gross wt) lb	651,700	470.1	0.4	- 0.2	1.35	48.87	48.88			

The transition cone to adapt the space tug payload to the ESS is an all aluminum structure employing skins, stringers, and frames (see Figure 1-4). This structure is similar to the Saturn V cone located between the S-II Stage and the S-IVB Stage. The fifteen-degree cone angle has been retained along with the basic structural design. Additional length has been added to provide the reduction to 15 feet in diameter.

Separation of the tug (payload) will be at ESS Station 1215, the adapter/payload interface. The separation system proposed for this payload is identical to the Space Station system.

The payloads are considered as passive during separation and will be physically separated by the forward facing ACPS thrusters on the ESS.

2.0 ESS DESIGN

2.1 STRUCTURAL SUBSYSTEM GROUP

The ESS structural design has been established to be compatible with loads applied from three payload configurations; the MDAC space station, the NR space tug, and the RNS. Structural attachment to the booster is provided at two points on the forward and aft skirts. Boost loads are transferred from the ESS to the booster by a separation system linkage defined in Volume II, Book 1, Section 5.1.3, of this report.

The structural subsystem is divided into primary vehicle structure and the thermal protection system. Primary structure includes all skirts, tankage, and thrust structure (see Figure 2-1), and the thermal protection system includes the vehicle sidewall insulation and the heat shield.

This study defines the structural configuration required to perform an ESS mission. Maximum use of the current S-II design was considered throughout the study to minimize development and qualification requirements. Where ESS loads exceed current S-II capability, the structural design concept utilizes reinforced S-II structure.

Detailed calculations used in establishing sizes are contained in the Design Data Book, Volume XII.

2.1.1 Structural Criteria and Loads

Criteria and loads considered in this report include primary body loads generated during prelaunch and boost phases by winds, control system response, combined system aerodynamics, and vehicle trajectories. Loads were determined for prelaunch, maximum q alpha, initial maximum acceleration and end boost using these criteria with structural response dynamics and aeroelastic effects. Special trajectories were determined for each of the three payload configurations to minimize loads imparted to the B-9U booster. The ESS body loads (distributed shears and moments) were converted to running loads (N_x) and compared to the structural capability of the S-II-11 design.

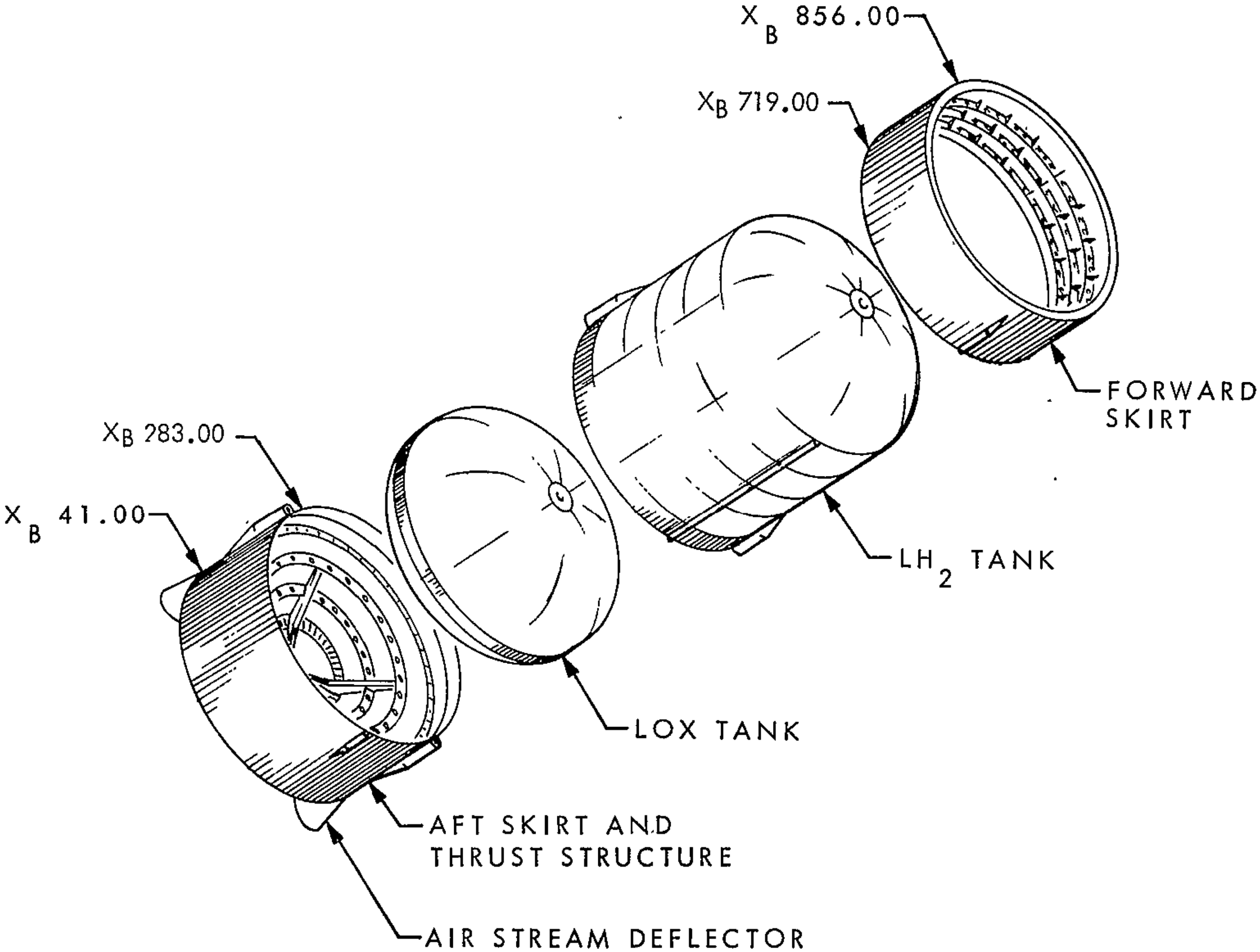


Figure 2-1. ESS Structural Configuration

Structural Criteria

The following criteria were used in establishing the structural design for the ESS vehicle.

1. Baseline Payloads. For this study, load analyses are limited to the following three representative payload configurations;

    a. Nuclear stage without engine (NR): weight, 83,000 pounds

    b. Space tug, geosynchronous mission (NR): weight, 107,180 pounds

    c. Space station (MDAC)—Single Launch Configuration: weight, 176,960 pounds

2. Configuration. The baseline vehicle is the B-9U booster with an ESS/payload. The total vehicle liftoff weight ranges from 4,217,000 to 5,190,000 pounds.

3. Propulsion. The rocket engine used for this study is the space shuttle high performance engine as identified in Drawing 13M15000, space shuttle vehicle and engine 550K(SL) interface control document.

4. Factor of Safety. The factor of safety will be 1.4.

5. Design. Design that would compromise the booster sufficiently to preclude its use with the orbiter will not be considered, and the load-carrying capability of the booster primary structure shall not be exceeded.

6. Minimum Modification. Minimum modification definition is "using present stage tankage."

7. Launch Capability. All-azimuth launch capability.

8. Load Factors. The vehicle trajectory load factors will not exceed 3 g for manned missions and 4 g for unmanned missions after staging.

9. Ground Winds. Ninety-nine-percentile winds for a one-day exposure per NASA Technical Memorandum 53872 are used with the vehicle fueled and pressurized. Ninety-nine-percentile winds for a 14-day exposure are used with the vehicle unfueled and unpressurized.

10. Flight Winds. A 95 percentile wind profile plus a 9 meter per second embedded gust per NASA Technical Memorandum 53872 is used for determining loads. The wind shear and the quasi-square wave discrete gust must be multiplied by a factor of 0.85 when constructing the synthetic wind profile.

11. Trajectory. The trajectory data used for this study are defined in Volume II, Book 2, Paragraph 5.2.2 of this report.

12. Control. The control requirements for the determination of loads will be those required to maintain trimmed flight. In the vicinity of maximum dynamic pressure, the pitch loads will be predicated on the product of dynamic pressure and angle of attack (q alpha) between the limits of +1500 psf-degree and -2900 psf-degree. The yaw loads in the vicinity of maximum dynamic pressure will be predicated upon α angle of ±4 degrees.

13. Weight and Center of Gravity. Table 2-1 defines the weight and center of gravity information used for the loads analysis. These data differ slightly from the final weight data iteration. The effect on loads is estimated to be small.

14. Thermal. The thermal data used for the forward bulkhead structural evaluation are shown in Figure 2-2. The aerodynamic heating rates are based on a MDAC space station payload. It is assumed for this study that the TPS will limit the structural temperatures to those experienced on the Saturn V program.

Loads

The ESS in conjunction with three separate payloads—Reusable Nuclear Stage, MDAC space station, and space tug—was considered in the formulation of the ESS/booster attach fitting loads and the ESS body loads. The body loads presented here do not include the influence of the fittings; therefore, to define a balanced system, the unbalanced shear, bending moment, and axial load must be balanced by the fittings. The loads were so presented to facilitate the optimization of the attachment design; this eliminated the necessity of reconstructing load diagrams each time an iteration in fitting geometry or design took place. The axial loads, as presented, act at the ESS centerline and do not reflect the effects of internal pressure. All loads presented are limit.

Prelaunch Loads Body Loads. Two wind conditions were considered for the prelaunch loads; i.e., a 14-day exposure was used for the unfueled vehicle while a 1-day exposure was used for the fueled vehicle. For both cases, 99 percentile winds were used. The fitting and body loads were

Table 2-1. Weight and Center of Gravity Data for Loads Analysis

MISSION/EVENT	COMBINED			ESS		
	WEIGHT	X_{CG}	Z_{CG}	WEIGHT	X_{CG}	Z_{CG}
B-9U/ESS-MDAC						
Lift-Off	5190350	2173	71	997900	2224	450
Max. $q\alpha$	3934100	2336	94			
Initial Max. g	3163900	2448	117			
End Boost	1810500	2640	217			
B-9U/ESS-RNS						
Lift-Off	3957650	2311	56	662820	2252	450
Max. $q\alpha$	2871300	2480	78			
Initial Max. g	2032300	2665	110			
End Boost	1477750	2753	166			
B-9U/ESS-TUG						
Lift-Off	4534000	2220	51	691616	2246	450
Max. $q\alpha$	3296800	2402	71			
Initial Max. g	2472000	2552	94			
End Boost	1504100	2735	170			

X_{CG} : Given in B-9U Body Stations

Z_{CG} : Given in Inches Above B-9U Centerline

determined for the wind blowing in from ESS to booster (headwind), booster to ESS (tailwind), and a crosswind which is normal to both the headwind and tailwind. The effects of vortex shedding and a gust are included in the analysis. It is recognized that a definition of the dynamic effects due to ground winds must be determined by wind tunnel tests; therefore, it is recommended that the prelaunch loads presented here do not form the basis for design which may penalize vehicle performance. Instead, they should form the basis for defining the need of wind tunnel tests and the necessity to consider ground-handling devices; e.g., dampers to restrict the loads at all times to as late as possible prior to launch. The prelaunch body loads for the three configurations are contained in Volume XII, the Design Data Book.

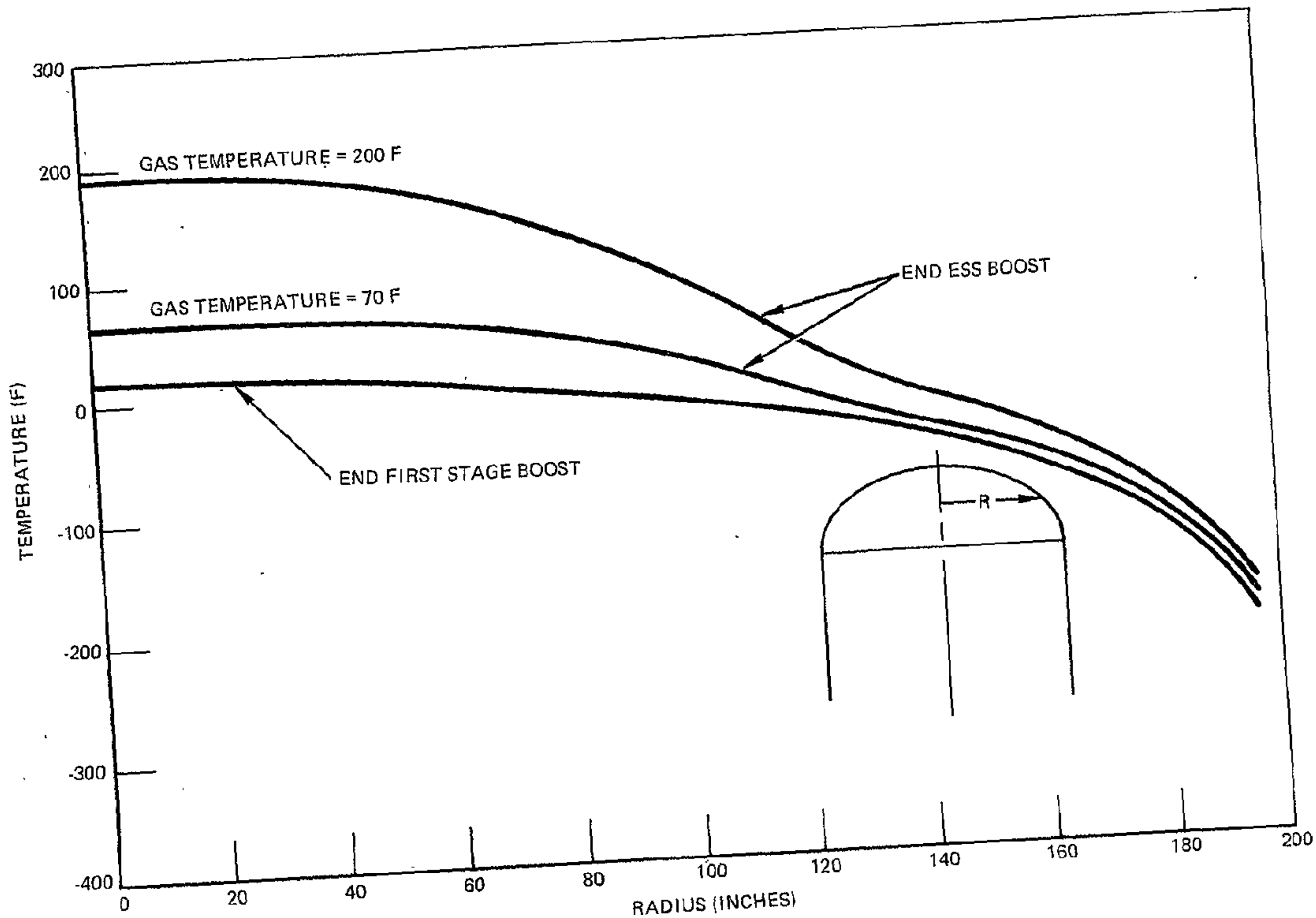


Figure 2-2. ESS Forward Bulkhead Temperature Distribution, LH_2 Loading: 450,000 Pounds (Tug) to 675,000 Pounds (MDAC)

Maximum Dynamic Pressure Body Loads. The maximum pitch loads in the vicinity of maximum dynamic pressure (q) were determined by using a q alpha of +1500 psf-degree (headwind) and -2900 psf-degree (tailwind). The maximum lateral loads in the region of maximum q were determined using a ±4-degree angle of yaw. These limits were the result of trajectory and control system studies which are discussed separately in Volume II, Book 1, Paragraph 5.2. The wind criteria used was 95 percentile scalar windspeed combined with 0.85 of 99 percentile wind shears and a 99 percentile gust. The body loads for the MDAC space station and RNS configurations with a q alpha of +1500 psf-degree (headwind) are presented in Figures 2-3 through 2-6, and for -2900 psf-degree (tailwind) in Figures 2-7 through 2-9. No side loads are to be combined with the maximum headwind and tailwind conditions. The maximum lateral loads for the two configurations are presented in Figures 2-10 through 2-15. The lateral load conditions require the combining of side loads with a set of pitch and axial loads. The lateral loads are due to the crosswind assumed effective for this condition. The pitch loads are those induced by maintaining a trimmed flight condition in pitch with no headwind or tailwind, and the axial loads are those induced by the drag, thrust, and inertia forces. The manner in which the lateral, pitch, and axial loads are to be combined is noted on the applicable figures. Tug loads are not critical for design and are contained in the Design Data Book (Volume XII).

End Boost Body Loads. The end boost body loads were obtained by applying the normal and axial components of acceleration from the trajectory information presented for each configuration. Since the vehicle is essentially out of the sensible atmosphere, the loads are due to inertia effects only. The body loads for the three configurations are shown in Figures 2-16 through 2-19. End boost loads for the space tug payload configuration are contained in the Design Data Book (Volume XII).

Fitting Loads. The ESS/booster attach fitting loads for the three configurations are summarized in Volume II, Book 1, Paragraph 5.2.2. The forward and aft fittings are considered effective in taking out pitch and lateral loads while only the booster forward fitting is considered effective in taking axial loads. The axial fitting load is assumed to act 202.5 inches below the ESS centerline.

Load Intensities

The external distributed loads (moment, axial, and shear) shown in paragraph 2.1.1 under Loads for the three payload configurations were converted to distributed running load (N_x) and compared to current S-II capability. This comparison shows the locus of maximum body loads for all loading conditions and the S-II capability during first stage boost.

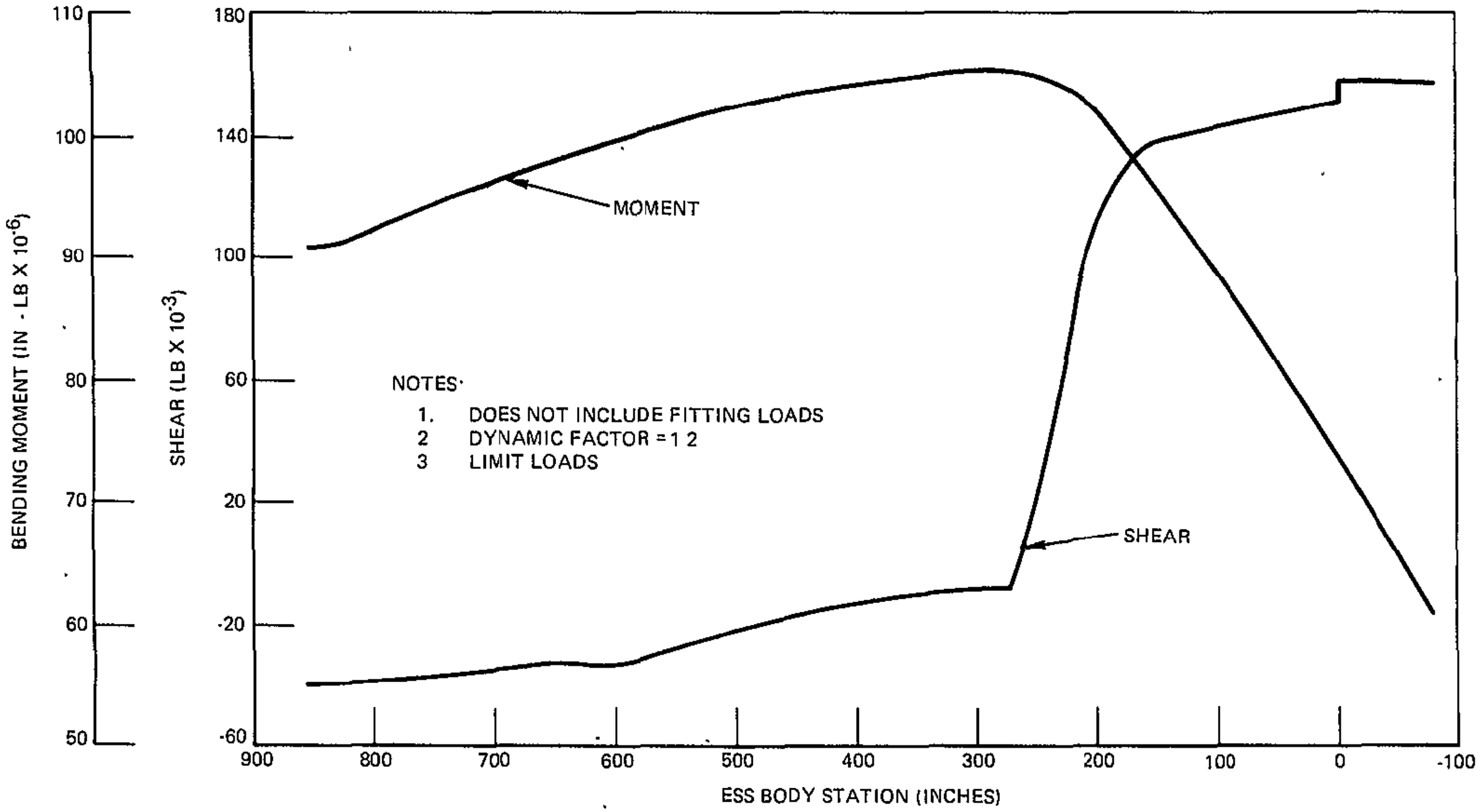


Figure 2-3. ESS/RNS Max $q\alpha$ Pitch Plane Body Loads (Headwind)

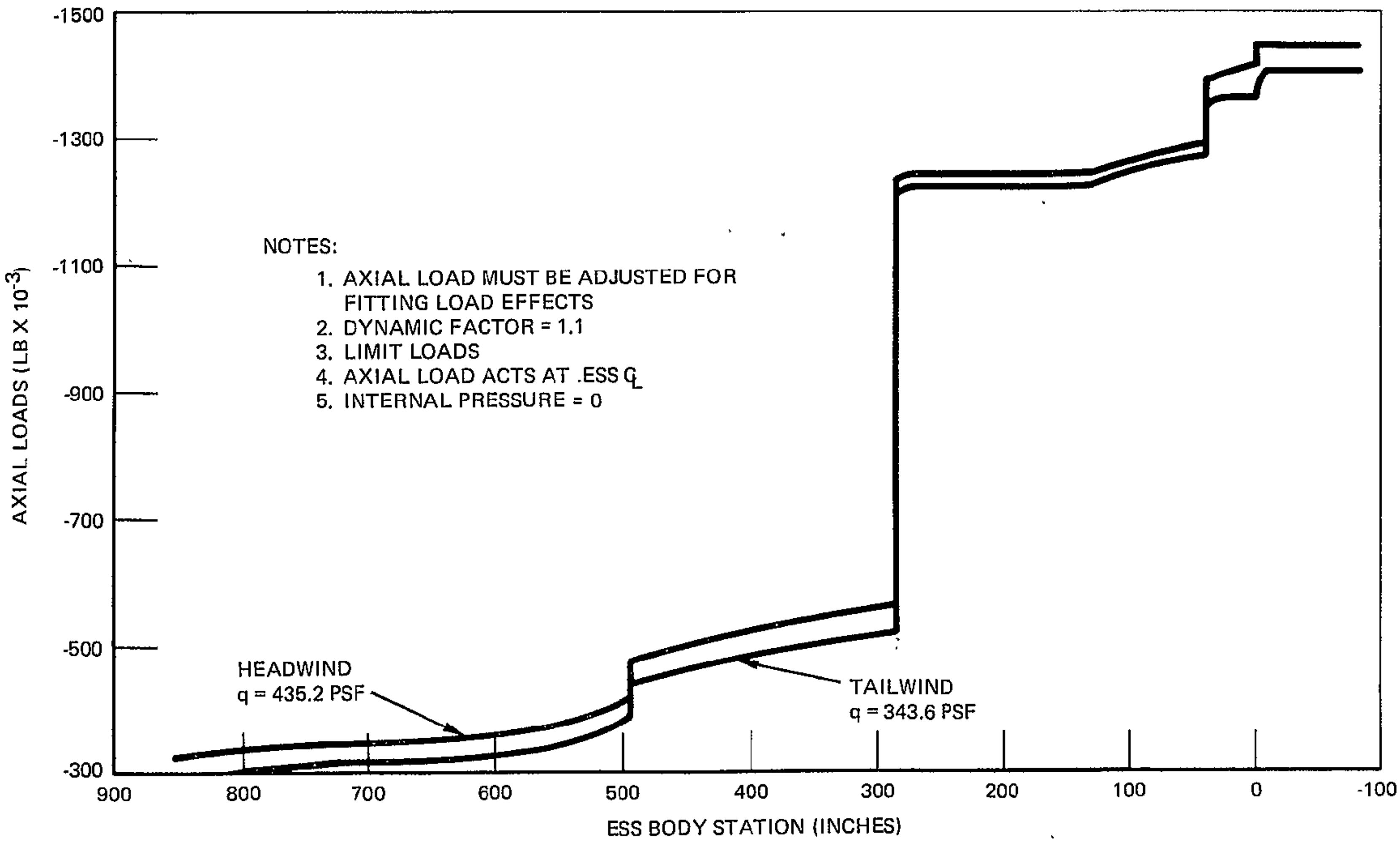


Figure 2-4. ESS/RNS Max qα Axial Body Loads (Headwind and Tailwind)

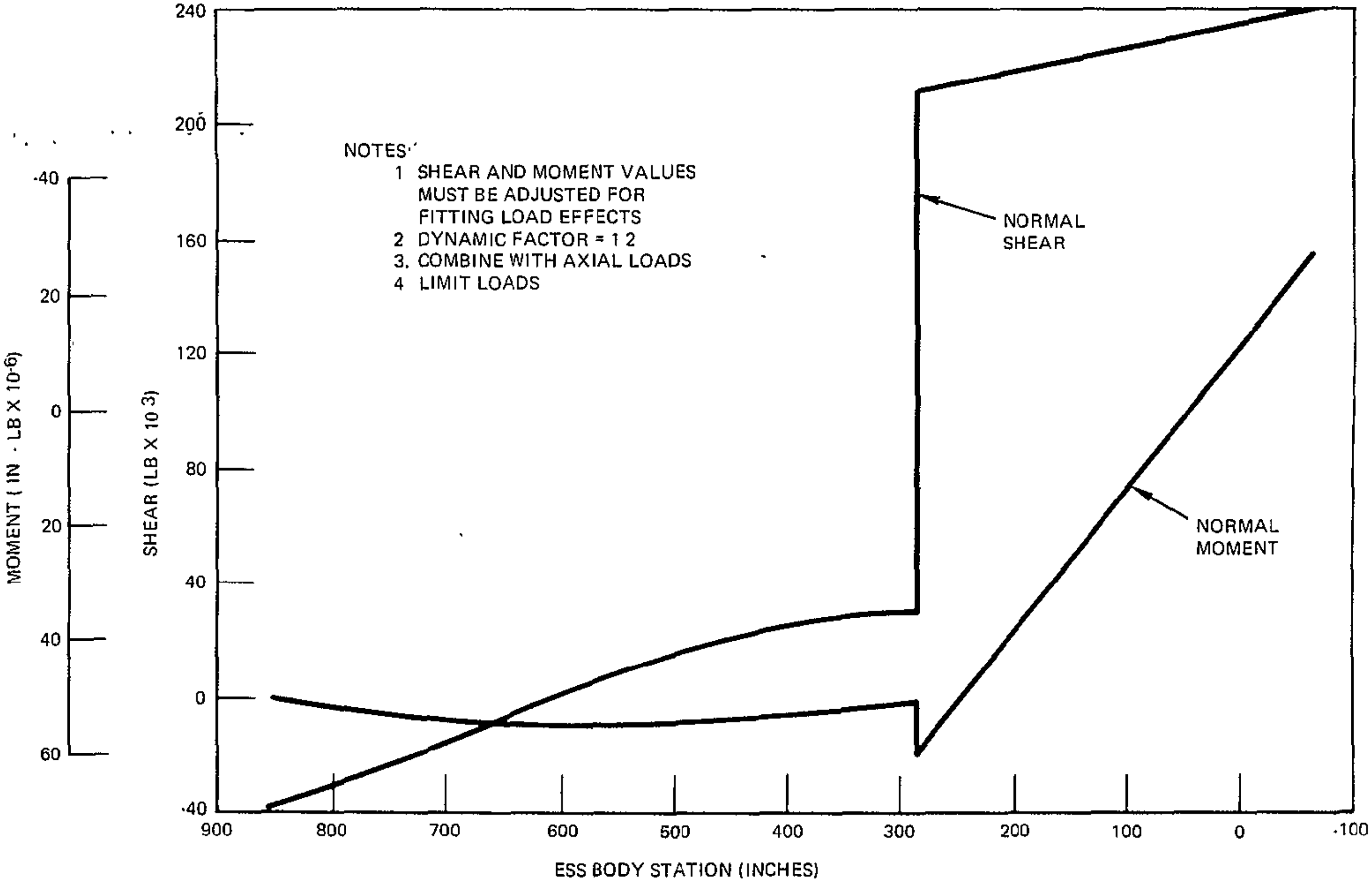


Figure 2-5. ESS/MDAC Max qα Pitch Plane Body Loads

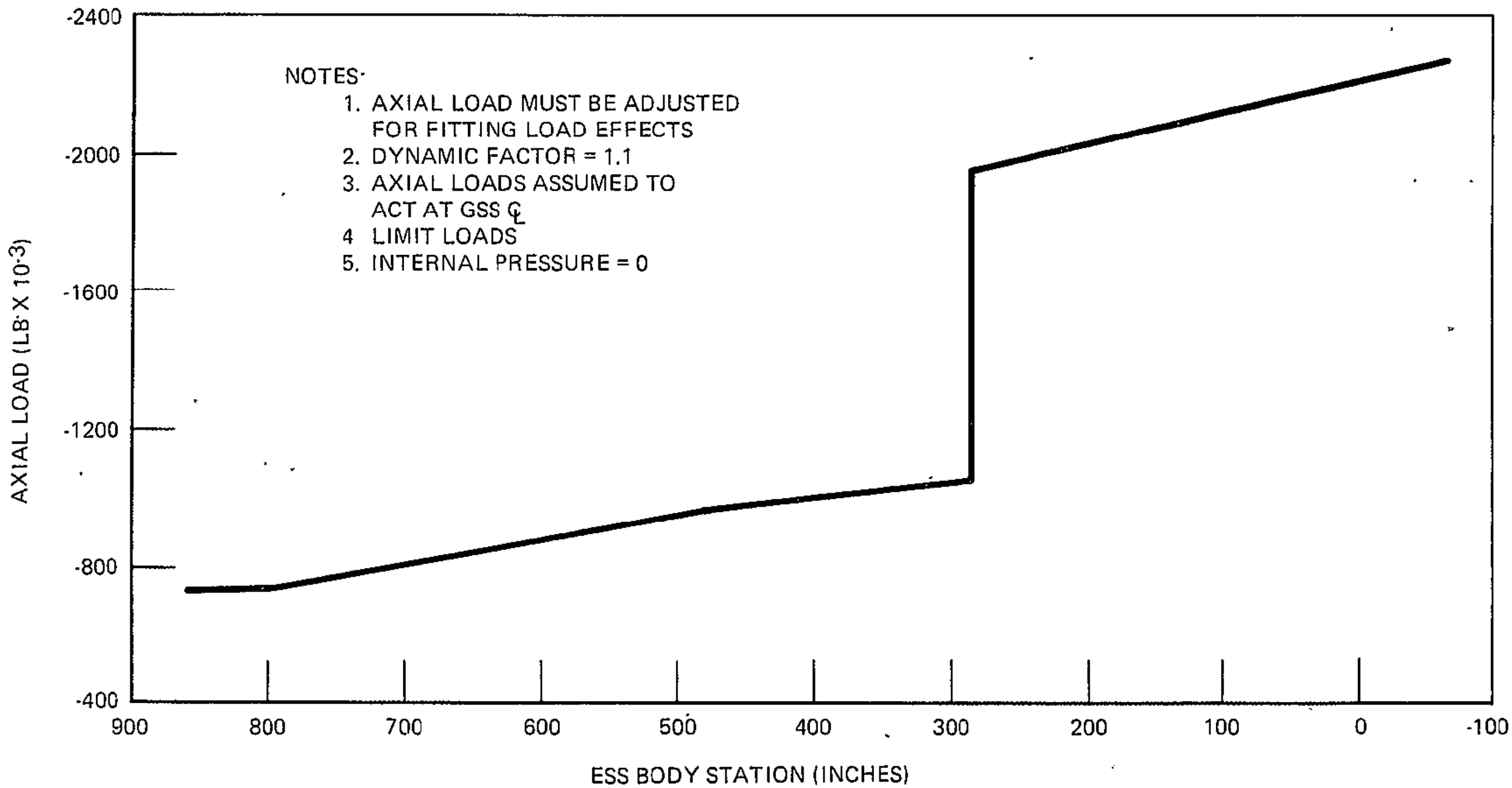


Figure 2-6. ESS/MDAC Max $q\alpha$ Axial Body Loads

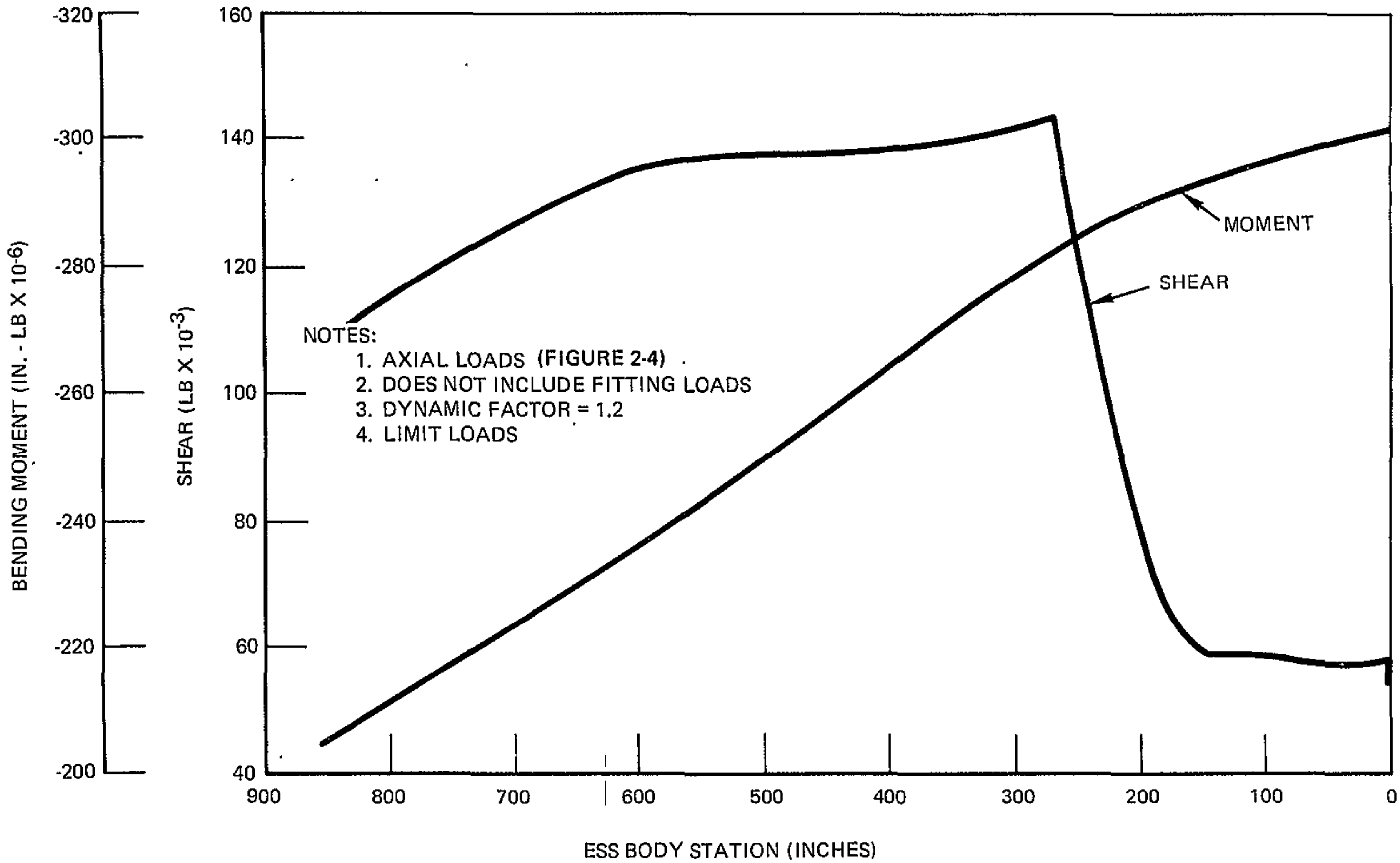


Figure 2-7. ESS/RNS Max $q\alpha$ Pitch Plane Body Loads (Tailwind Case)

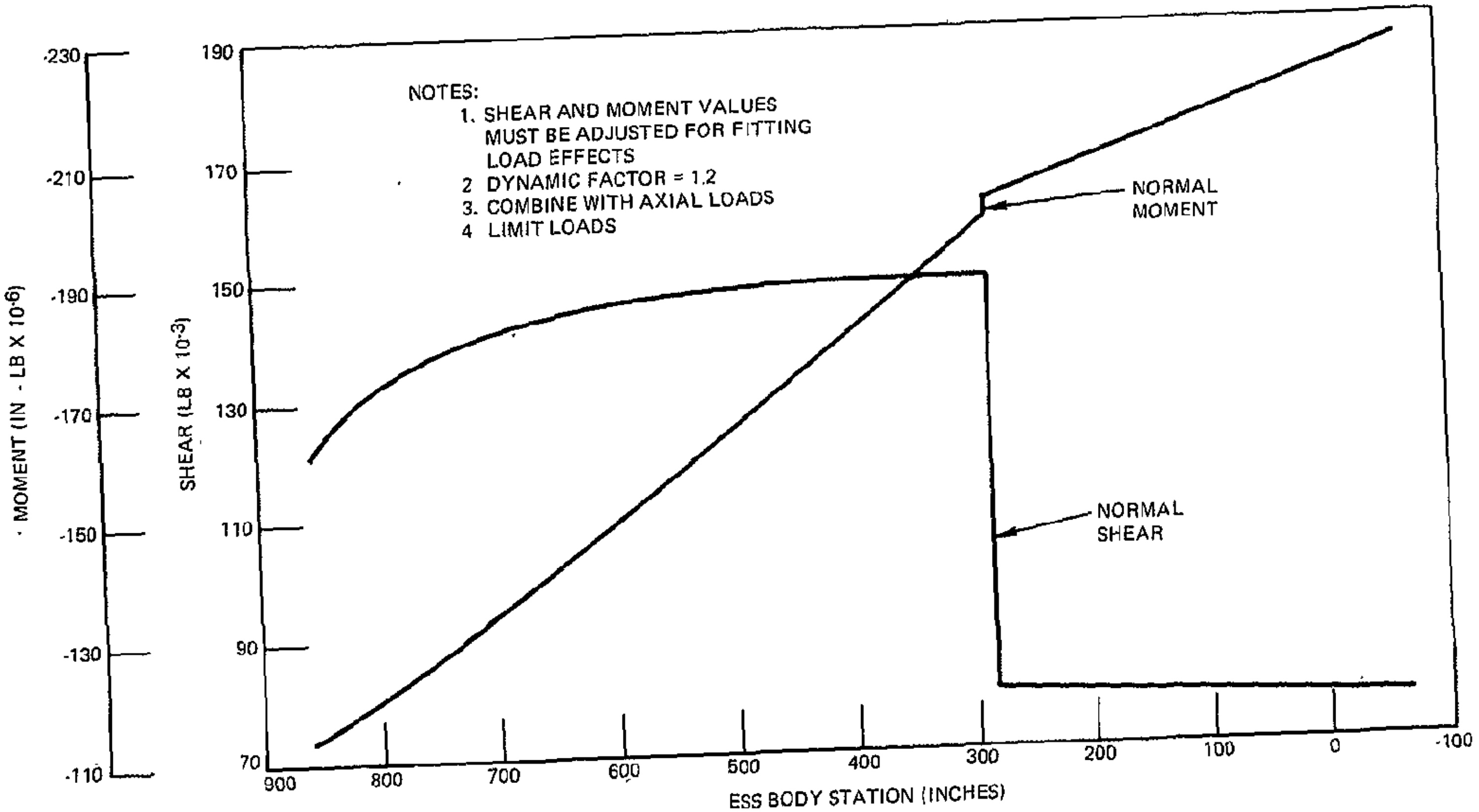


Figure 2-8. ESS/MDAC Max qα Pitch Plane Body Loads (Tailwind Case)

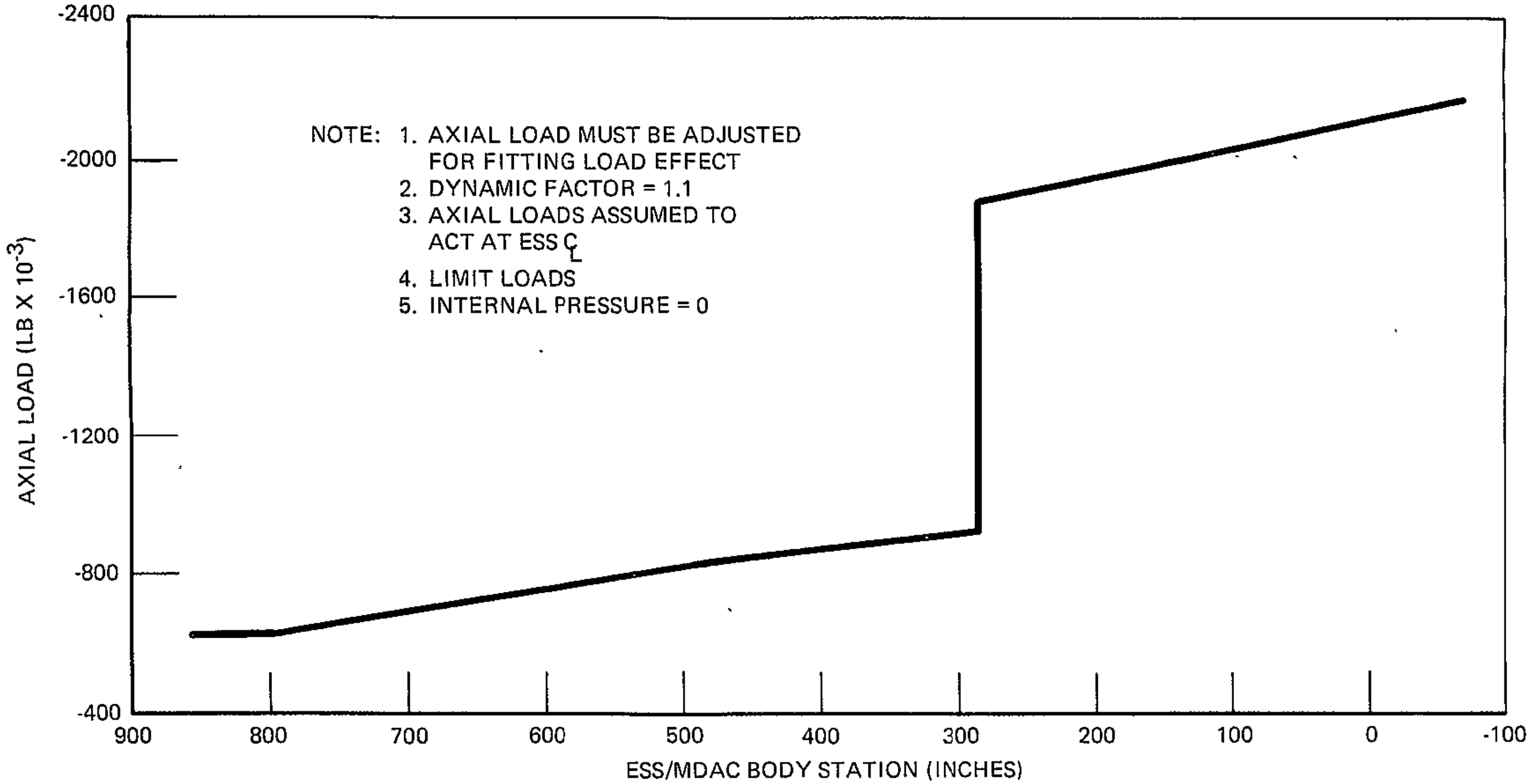


Figure 2-9. ESS/MDAC Max qα Axial Body Loads (Tailwind Case)

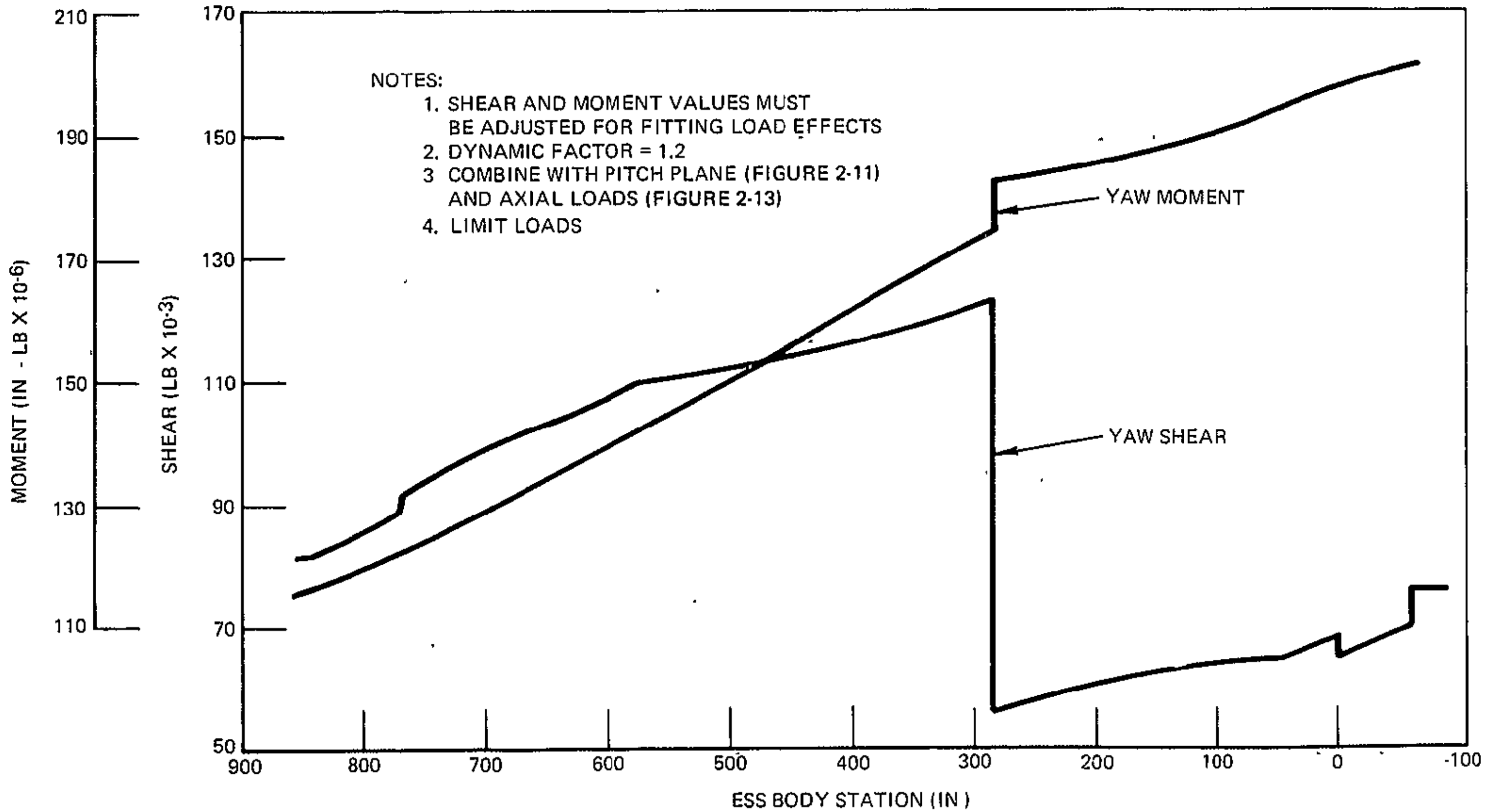


Figure 2-10. ESS/RNS Max qα Trim Yaw Body Loads (Tailwind Case)

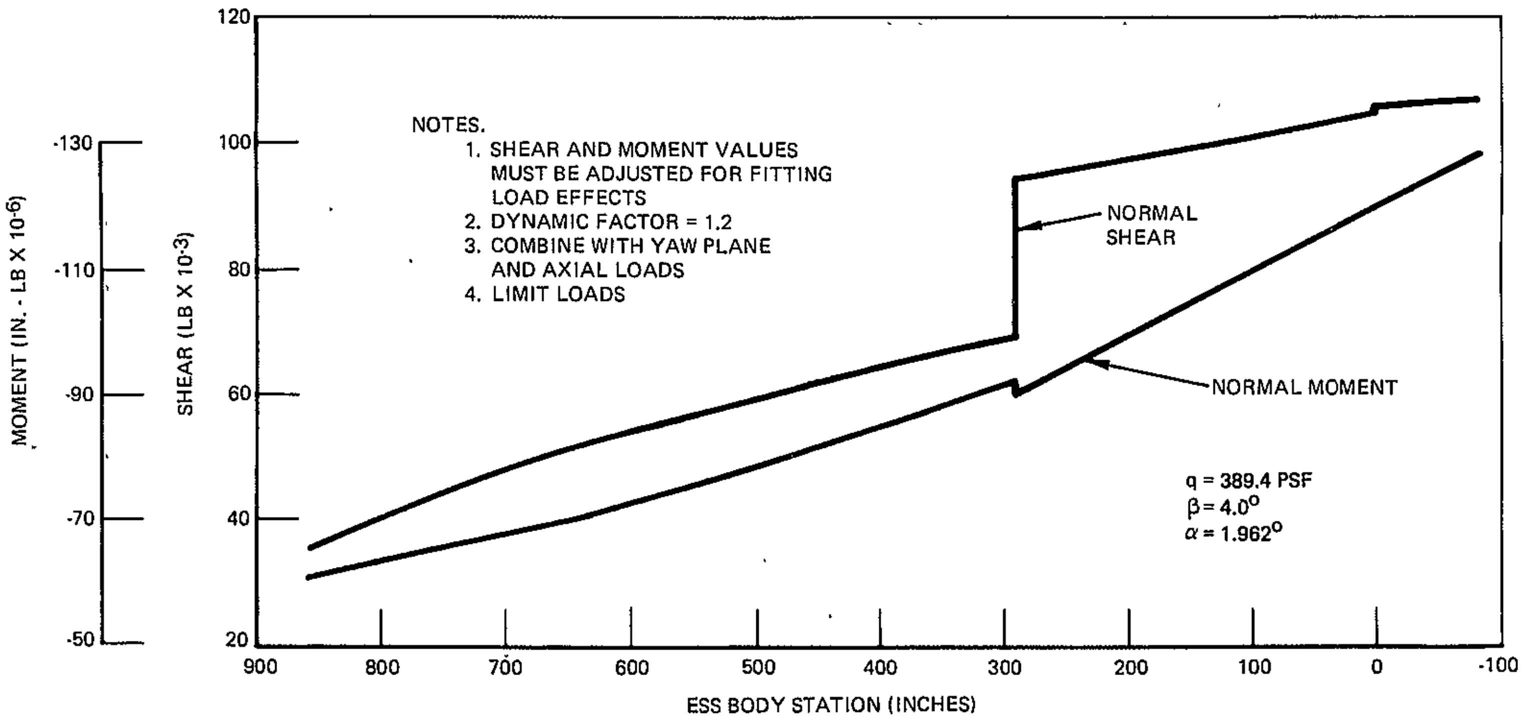


Figure 2-11. ESS/RNS Max qα Trim Pitch Body Loads

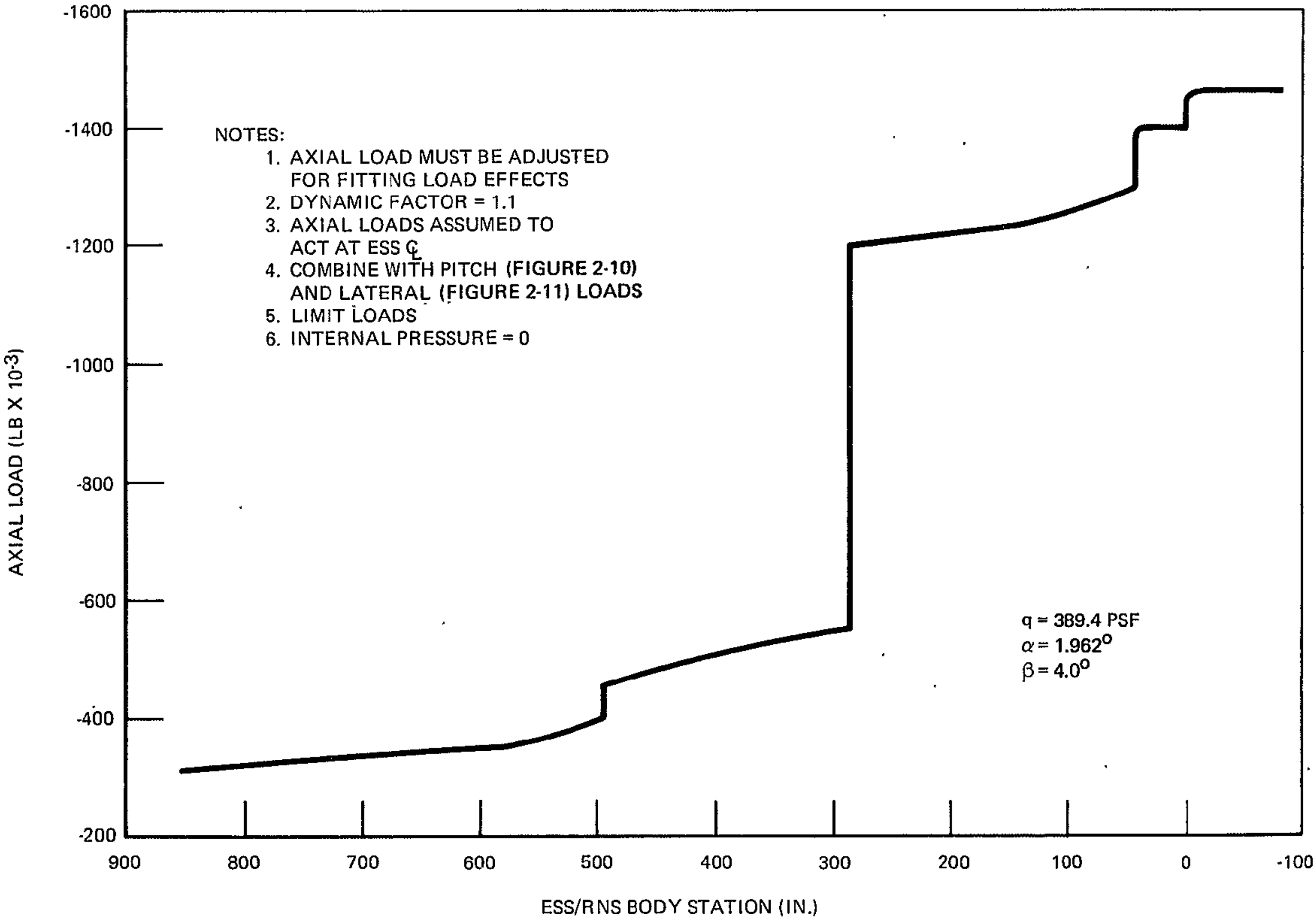


Figure 2-12. ESS/RNS Max $q\alpha$ Trim Axial Body Loads (Alpha Trim Case)

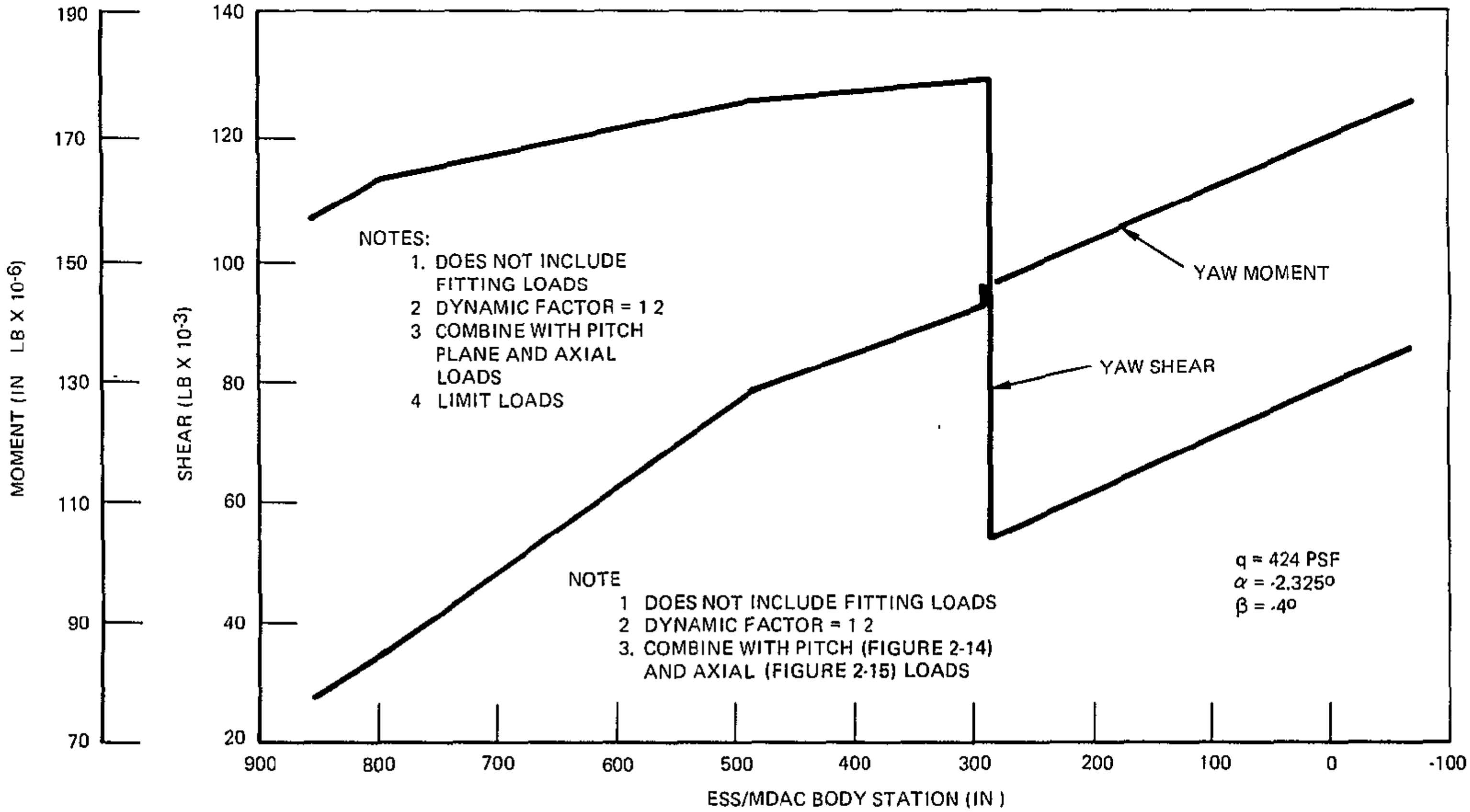


Figure 2-13. ESS/MDAC Max qα Trim Yaw Body Loads

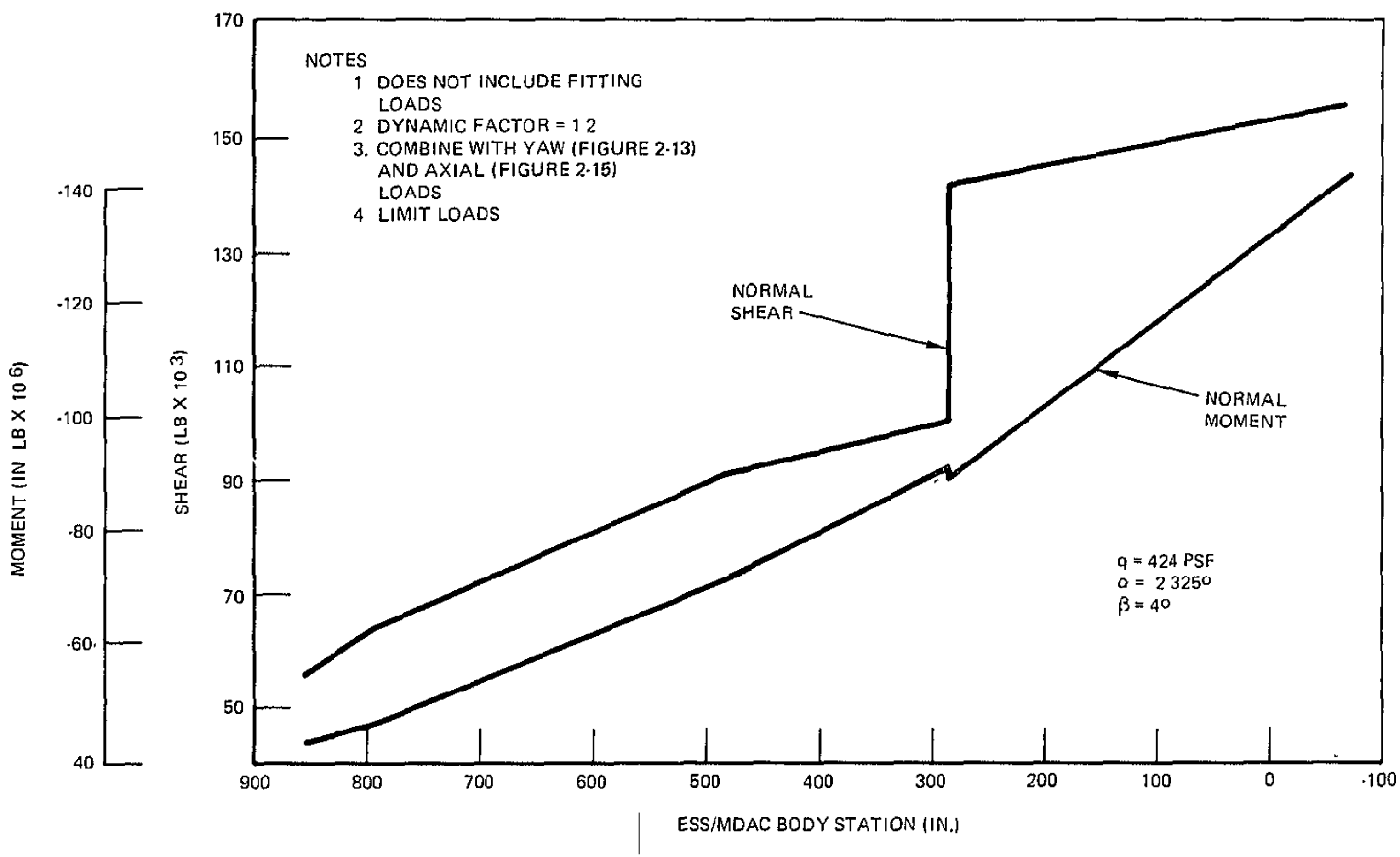


Figure 2-14. ESS/MDAC Max qα Trim Pitch Body Loads

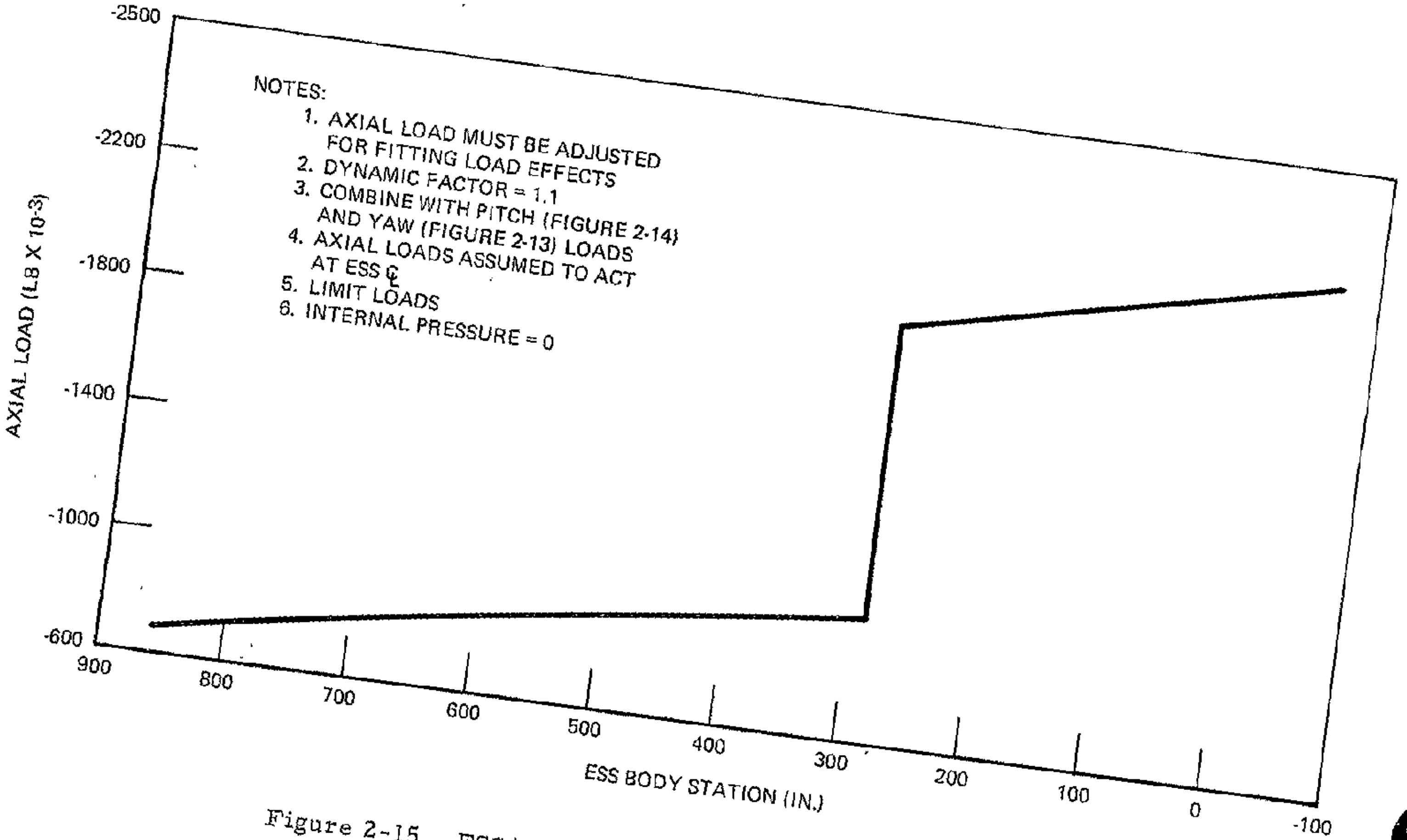


Figure 2-15. ESS/MDAC Max qα Trim Axial Body Loads

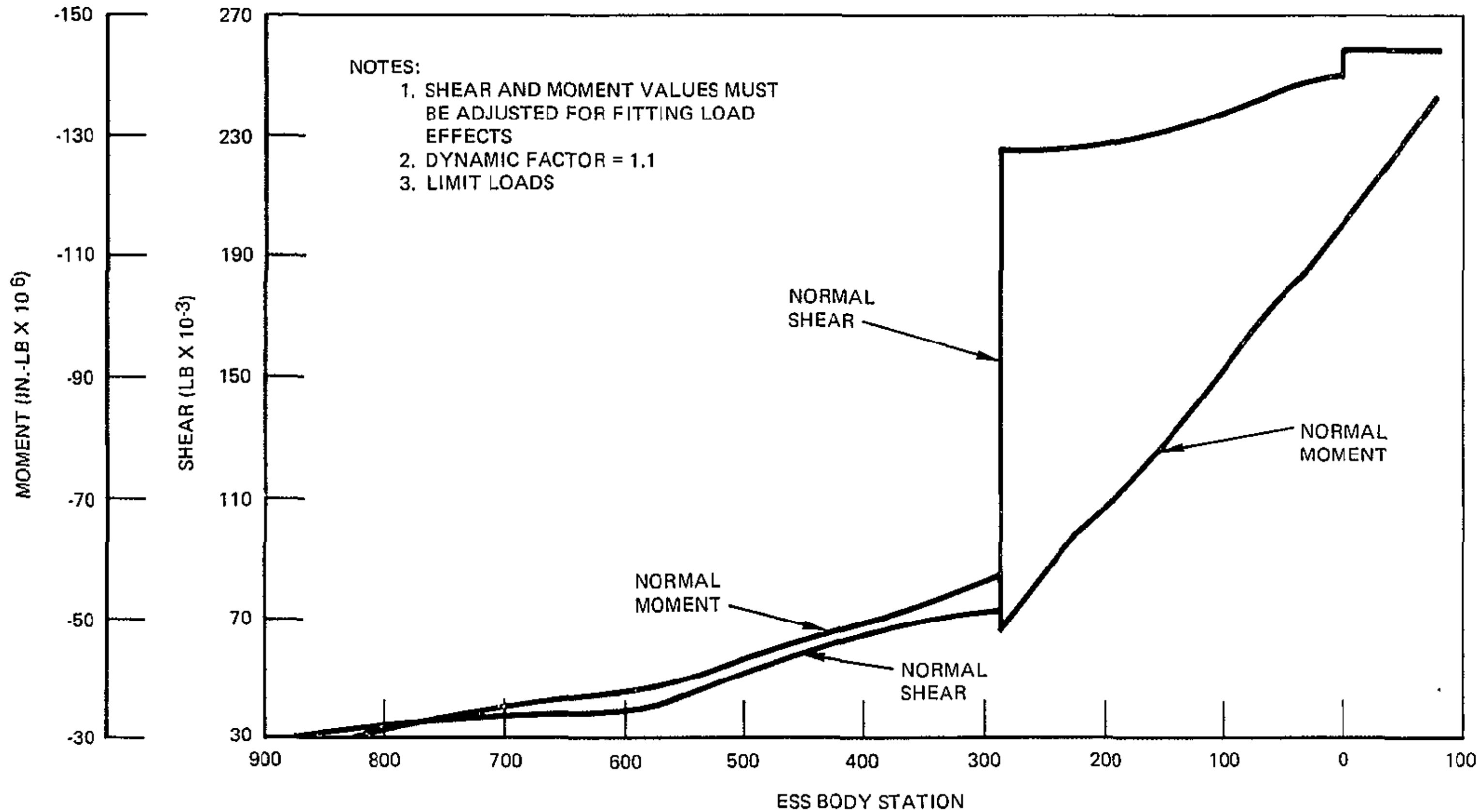


Figure 2-16. ESS/RNS End Boost Normal Body Loads

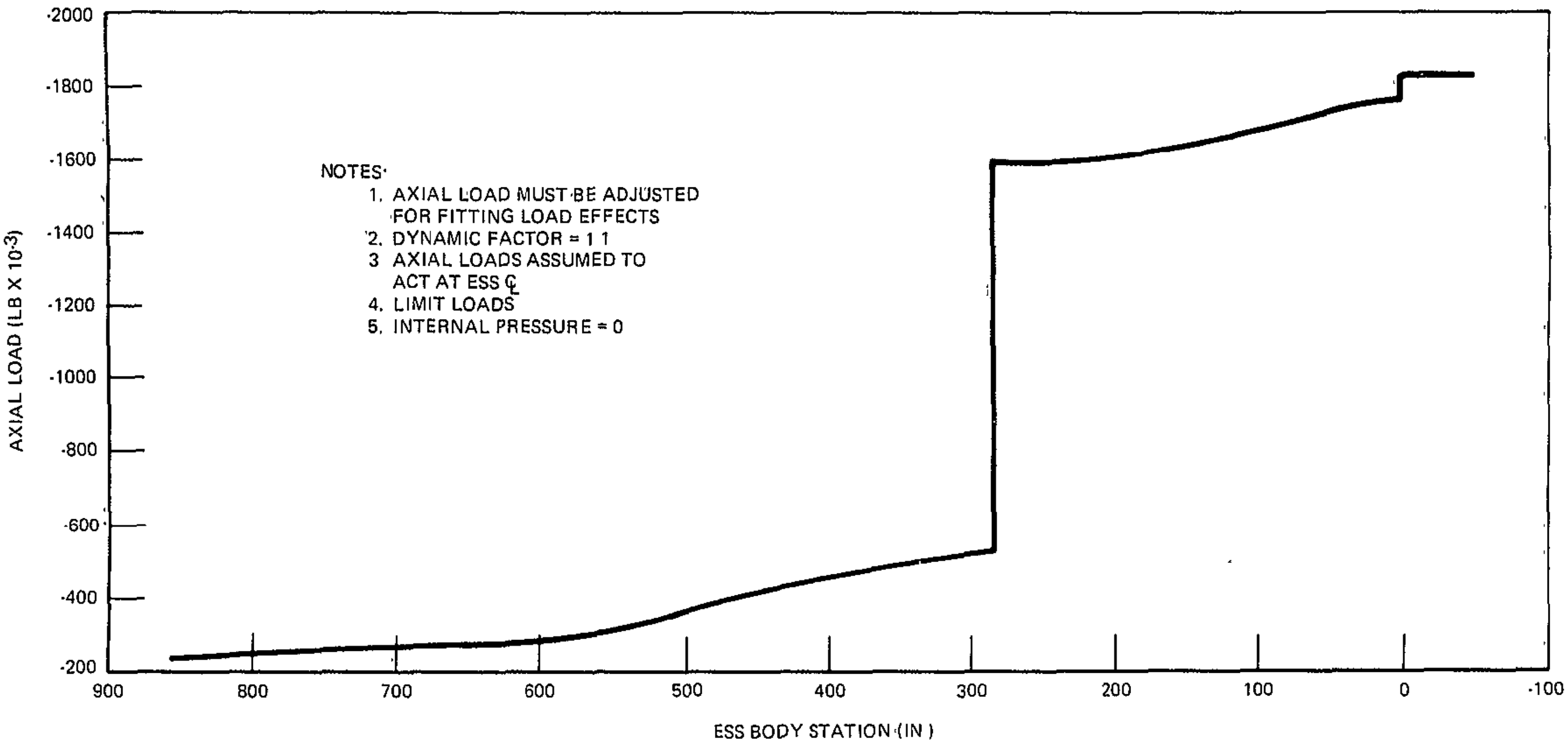


Figure 2-17. ESS/RNS End Boost Axial Body Loads

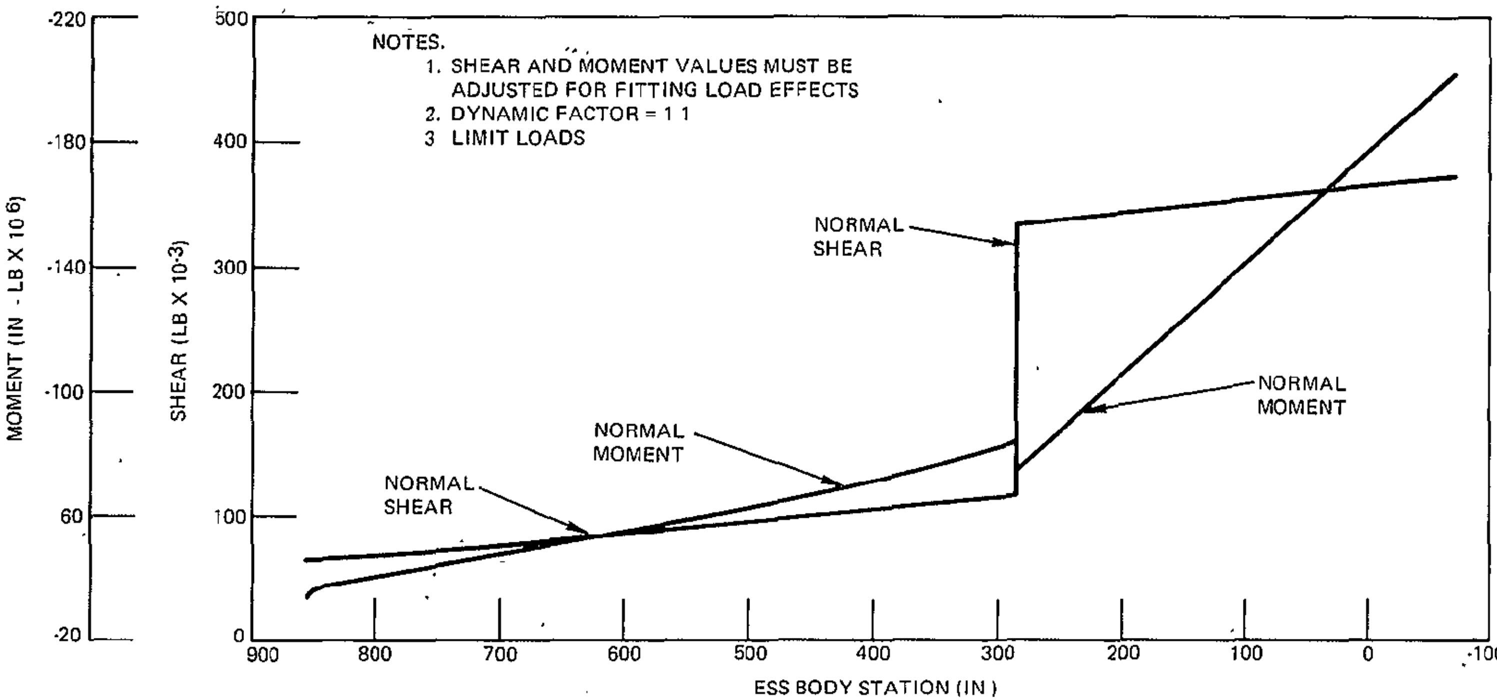


Figure 2-18. ESS/MDAC End Boost Normal Body Loads

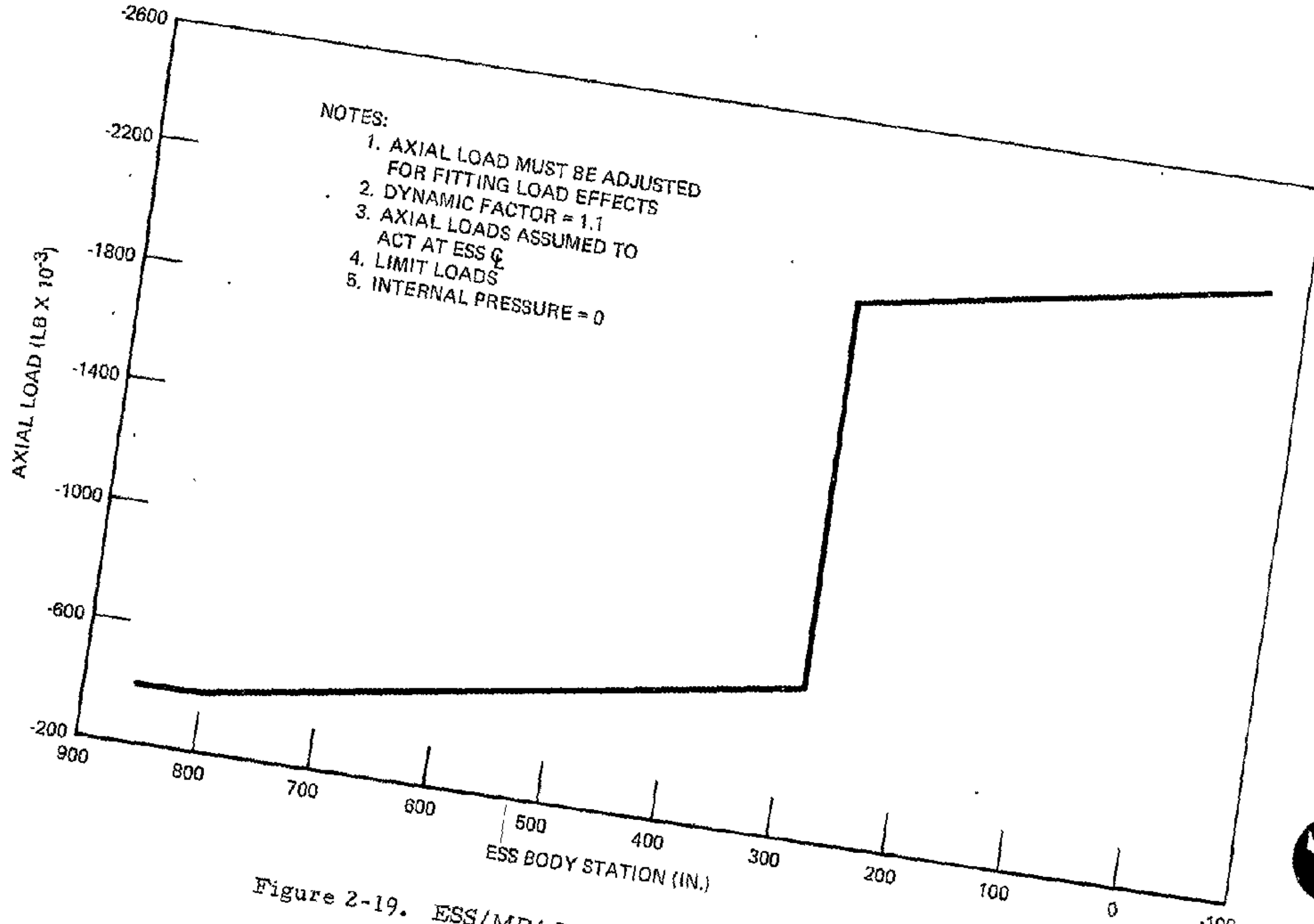


Figure 2-19. ESS/MDAC End Boost Axial Body Loads

All ESS mission tension loads are well within the structural capability of the S-II design and therefore are not included in this report.

Figure 2-20 shows the comparison of compression load intensities. The structural capability of the S-II design is exceeded in the forward skirt region by the RNS payload load intensities. This higher load dictates a small skin-stringer modification to the basic structural design. The minimum ullage pressure of 27.5 psig was used in establishing the compressive capability envelope for the LH_2 tank.

It should be noted that these load intensity figures are based on gross bending moments, axial and shear loads, and do not show the requirement for local reinforcement of the skirt structure to transmit the fitting loads. Local shear loads in these skirts are defined in paragraph 2.1.2.

2.1.2 Vehicle Structures

The ESS vehicle structure has been designed considering maximum use of current S-II structural components. The vehicle structure consists of five major subassemblies; a forward skirt, an LH_2 tank, a LO_2 tank, an aft skirt, and a thrust structure. The structural modifications to the S-II required to support the ESS concept are delineated throughout this section. S-II structural modifications are summarized in Figure 2-21.

The ESS vehicle was analyzed for the booster attachment loads by means of the NARSAMS Structures Analysis Computer program. This program is a modification of ELAS-A, Jet Propulsion Laboratory Technical Report 32-1240. NARSAMS uses a finite element stiffness method for the solution of linear equilibrium problems in structural mechanics. The methods used in this analysis are described in the Design Data Book (Volume XII).

Forward Skirt

The forward skirt of the ESS is a cylindrical semi-monocoque shell that connects the forward end of the LH_2 tank and the ESS payloads. This structure must also provide the forward attachment to the booster. The general arrangement of the forward skirt is shown in Figure 2-22. The length of the structure is identical to the S-II-15 (138 inches). However, with the short LH_2 tank employed on the ESS, the station notation is different. The forward skirt is a simplified design where the intermediate "buffet" stringers were eliminated and constant thickness (0.100 inch) skins incorporated, and the stringer extrusions changed from 1.50 inches high by 1.30 inches wide to 1.875 inches high by 2.00 inches wide.

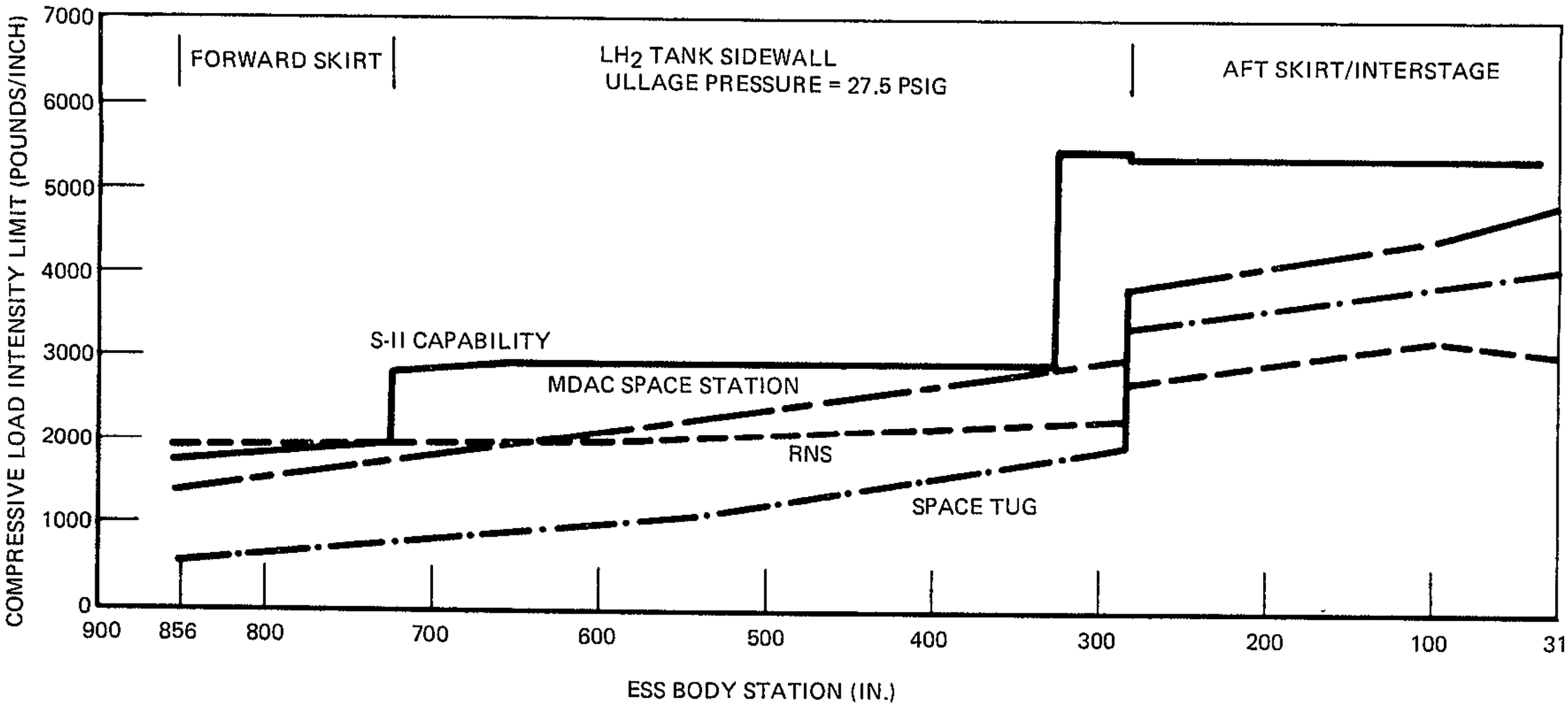
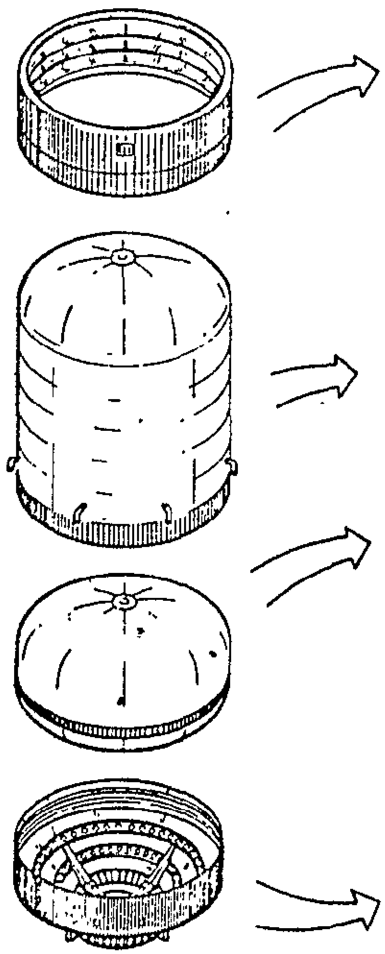


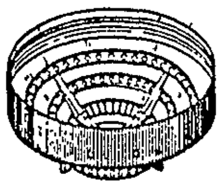
Figure 2-20. S-II Applied Compressive Loading Comparison, ESS With Payloads



STRUCTURAL ASSEMBLY	REQUIRED MODIFICATIONS
FORWARD SKIRT	A. FORWARD BOOSTER ATTACH FRAME (REPLACES GENERAL STABILITY FRAME) B. SKIN THICKNESS INCREASED TO 0.10 IN (INTERMEDIATE STRINGERS DELETED) C. SMALL STRINGER THICKNESS INCREASE D. FORWARD ATTACH FITTING SUPPORTS
LH_2 TANK	A. BOSS PATTERN IN SIDEWALL FOR ERROSION BARRIER INSTALLATION B. TWO FEED-LINE ELBOWS FOR ORBITER ENGINE CYLINDER NO. 5 DELETED
LO_2 TANK	A. SUMP FOR TWO ORBITER ENGINE FEED LINES
AFT SKIRT	A. AFT BOOSTER ATTACHMENT FRAME (REPLACES GENERAL STABILITY FRAME) (STATION 41) B. SKIN DOUBLER AND STRINGER REINFORCEMENT FOR BOOSTER ATTACH LOAD EFFECTS C. AFT BOOSTER ATTACHMENT FITTINGS D. LENGTH ~ FROM 87 INCHES FOR S-II TO 252 FOR ESS E THRUST STRUCTURE ATTACH FRAME AT STATION 174.5 F. FRAME AT STATION 223 CHANGED TO STABILITY FRAME AND MOVED TO STATION 206.5 G. TWO FEED-LINE CUTOUTS H. ATTITUDE CONTROL SUPPORTS I. FRAME REINFORCEMENT (SHEAR DISTRIBUTION)

Figure 2-21. Summary of Major S-II Stage Structural Modifications (Sheet 1 of 2)

STRUCTURAL ASSEMBLY	REQUIRED MODIFICATIONS
AFT SKIRT (CONT)	J. FAIRINGS AT AFT END TO PROTECT ORBITER ENGINES FROM AERO SHEAR K. FRAME AT STATION 240 MOVED TO STATION 238 L. TWO ADDITIONAL GENERAL STABILITY FRAMES. (STATIONS 127 AND 79) M. ATTACHMENT FOR DOCKING ADAPTER SUPPORT STRUCTURE
THRUST STRUCTURE	ALL NEW STRUCTURE INCORPORATING: A. ATTACHMENT FOR TWO SS ORBITER ENGINES B. 125.5 INCH HIGH CONICAL FRUSTUM (S-II IS 111 IN.) C. ATTACHMENT FOR TWO SS ORBITER OMS ENGINES D. REVISED STRINGER EXTRUSION SECTIONS AND SKIN THICKNESSES E. SUPPORTS FOR OMS PROPELLANT CONTAINERS F. SUPPORTS FOR ELECTRICAL EQUIPMENT
INTERSTAGE	NONE REQUIRED



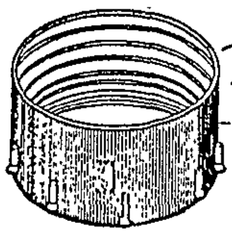


Figure 2-21. Summary of Major S-II Stage Structural Modifications (Sheet 2 of 2)

Skin and Stringer Structural Assessment. The skin and stringer shell is designed by the load intensities shown in Paragraph 2. 1. 1 under Load Intensities. These loads are slightly in excess of the baseline structural capability. Therefore, a small increase in stringer thickness is incorporated. Shear flows from the booster attachment loads are not severe enough to require any increase in the 0. 100-inch skin thickness. However, a skin doubler fourteen stringers wide is required to react the load from the separation strut. This is a large tension load (approximately 400, 000 pounds) that is applied instantaneously upon initiation of the ESS booster separation sequence and diminishes rapidly as the separation sequence progresses. Since this load is reacted by axial tension in the shell and radial frame loads, all of the required reinforcement can be placed into the doubler and the stringers remain unaffected. All allowable stresses are consistent with those of S-II-15, and thermal environment is assumed to be the same or less than S-II-15 and will be controlled by insulation.

Frame Structural Assessment. There are three frames in the forward skirt, located at Stations 767, 795, and 831. No frames are incorporated at either end of the structure. Shell stability is provided by the LH_2 tank at the aft interface and by the payload structure at the forward interface. The frames have the requirement of precluding failure of the shell by general instability plus reacting any direct or indirect internal loads. The frame at Station 767 is a stability frame and as such contains the same sections as the S-II-15 forward skirt frames. The two remaining frames are load frames and have sections greatly in excess of the stability requirement. The frame at Station 831 is the larger of the two. This frame is loaded by the booster vertical attachments approximately 78 inches each side of Position IV, and by side (lateral) loads at the centerline attachment on Position IV. Internal frame loads from the NARSAMS program solution are shown in Figure 2-23. Maximum loading is in the area of Position IV and is minimum at two points approximately 40 degrees each side of Position II. A tapered frame section is employed in order to minimize weight. The load is introduced through external fittings that pick up booster fittings or struts and full depth bathtub frame fittings. Critical loading occurs at maximum q alpha. The frame is constructed of 7075-T651 aluminum alloy.

The frame at Station 795 is loaded by the booster separation strut attachment fitting. This fitting is external to the forward skirt and is connected between the frames at Stations 795 and 831. This frame, like the Station 831 frame, has a tapered section and is constructed of 7075-T651 aluminum alloy.

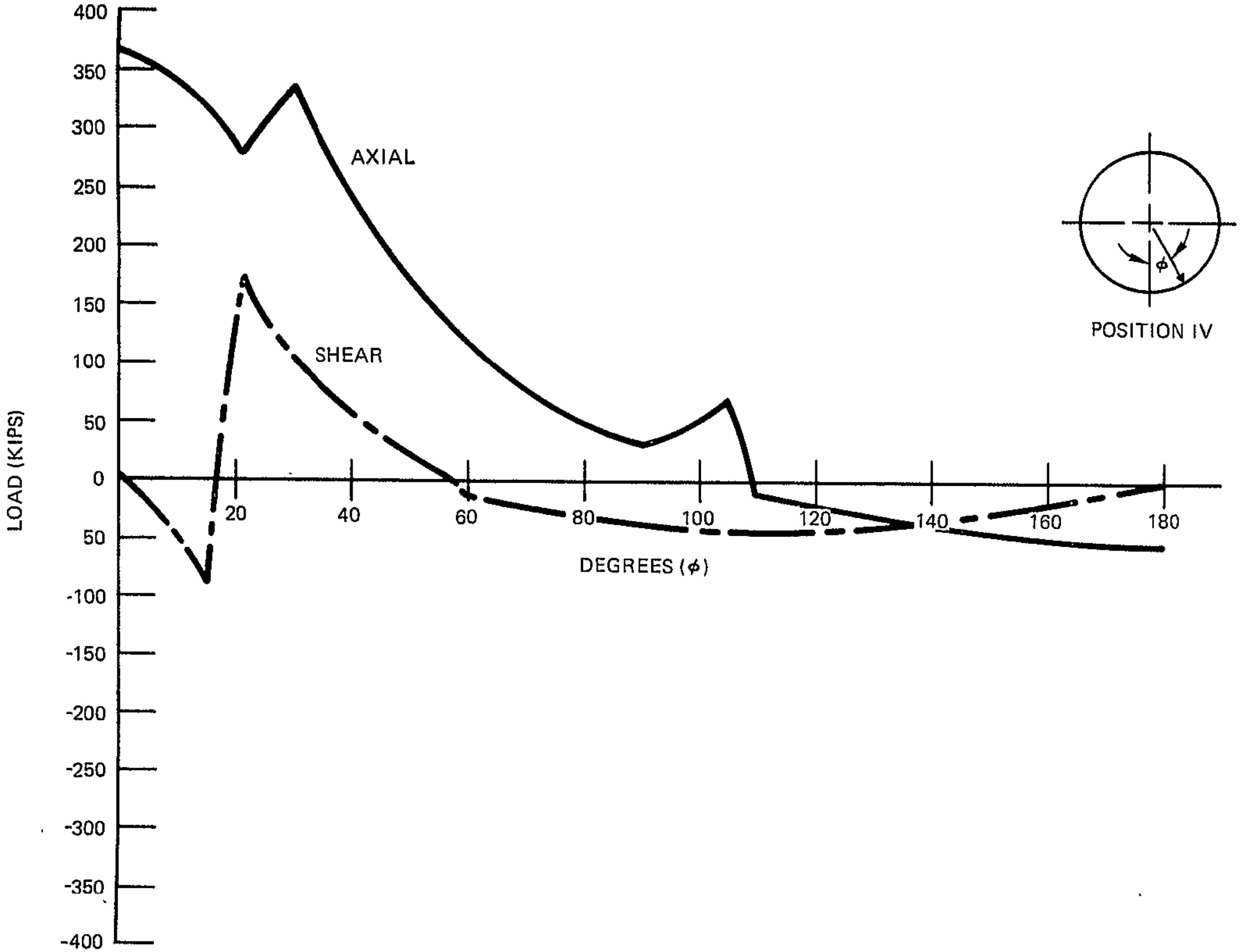


Figure 2-23. Forward Skirt Frame, Station 831, Internal Loads (Sheet 1 of 2)

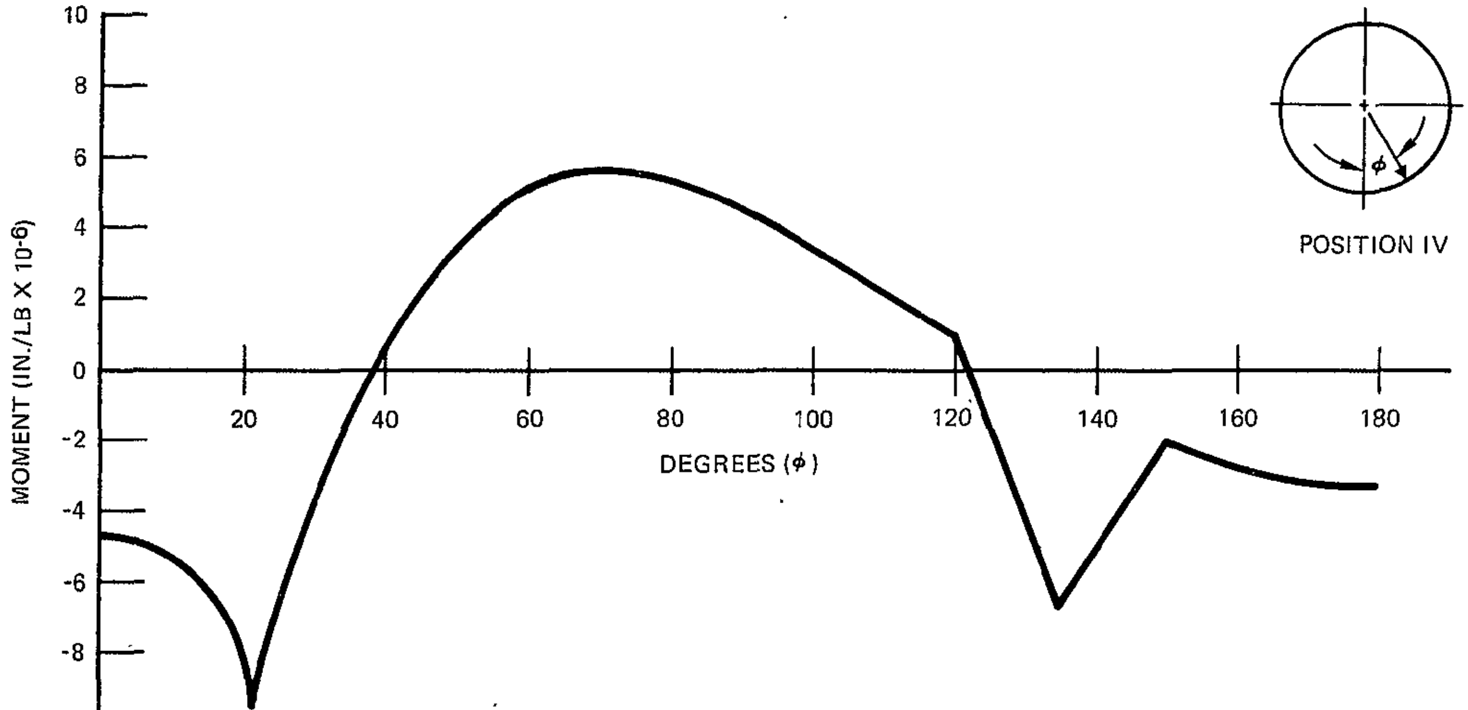


Figure 2-23. Forward Skirt Frame, Station 831, Internal Loads (Sheet 2 of 2)

Tank Structures

LH_2 Tank. The LH_2 tank is an aluminum (Al 2014-T6) cylinder closed at the upper end by the forward bulkhead and at the lower end by the common bulkhead (Figure 2-24).

The cylindrical portion of the LH_2 tank is a semi-monocoque structure consisting of closely spaced integral axial stiffeners and widely spaced circumferential frames riveted to integral rings. Except for a short portion at the bottom of the cylinder, the stiffeners are internal to the tank. The cylindrical section consists of four cylinder assemblies, each 99 inches long, and one cylinder assembly approximately 28 inches long. The tank wall design is identical to the current S-II with the following exceptions: two 14-inch feedline outlets rather than five 8-inch outlets in cylinder 2; new integral bosses in all cylinders to accommodate the insulation erosion barrier and new fairings; deletion of cylinder 5 for the smaller fuel load requirements.

The forward bulkhead is a modified ellipsoid of revolution. It is a monocoque structure consisting of twelve aluminum (Al 2014-T6) gore panels welded together along meridians and closed at the apex by a removable door. The door is bolted to a ring which is circumferentially welded to the gore panels. A circumferential weld attaches the bulkhead to the cylinder just below its equator. This bulkhead is identical to the current S-II.

The common bulkhead is defined as a part of the LO_2 tank and is discussed in the LO_2 tank assessment.

Forward Bulkhead Structural Assessment. The LH_2 tank forward bulkhead is critical during first-stage boost and end ESS boost. Temperature profiles on the bulkhead for these conditions are given in Figure 2-2. Based on these temperature profiles, the allowable differential pressure on the bulkhead can be calculated. Figure 2-25 gives the allowable pressure as a function of bulkhead radius. This allowable pressure is compared to the ESS LH_2 tank pressure requirements. As shown, the pressure requirements are identical to the pressure allowables at a radius of 40 inches. Elsewhere, over the bulkhead, the pressure capability exceeds requirements.

LH_2 Tank Wall Structural Assessment. The LH_2 tank wall is critical for stability during first stage boost. The tank wall compressive loading is compared to the ESS capability in Figure 2-20. As shown the capability exceeds the actual loading in all regions of the LH_2 tank. The tank wall is not critical for tension loading.

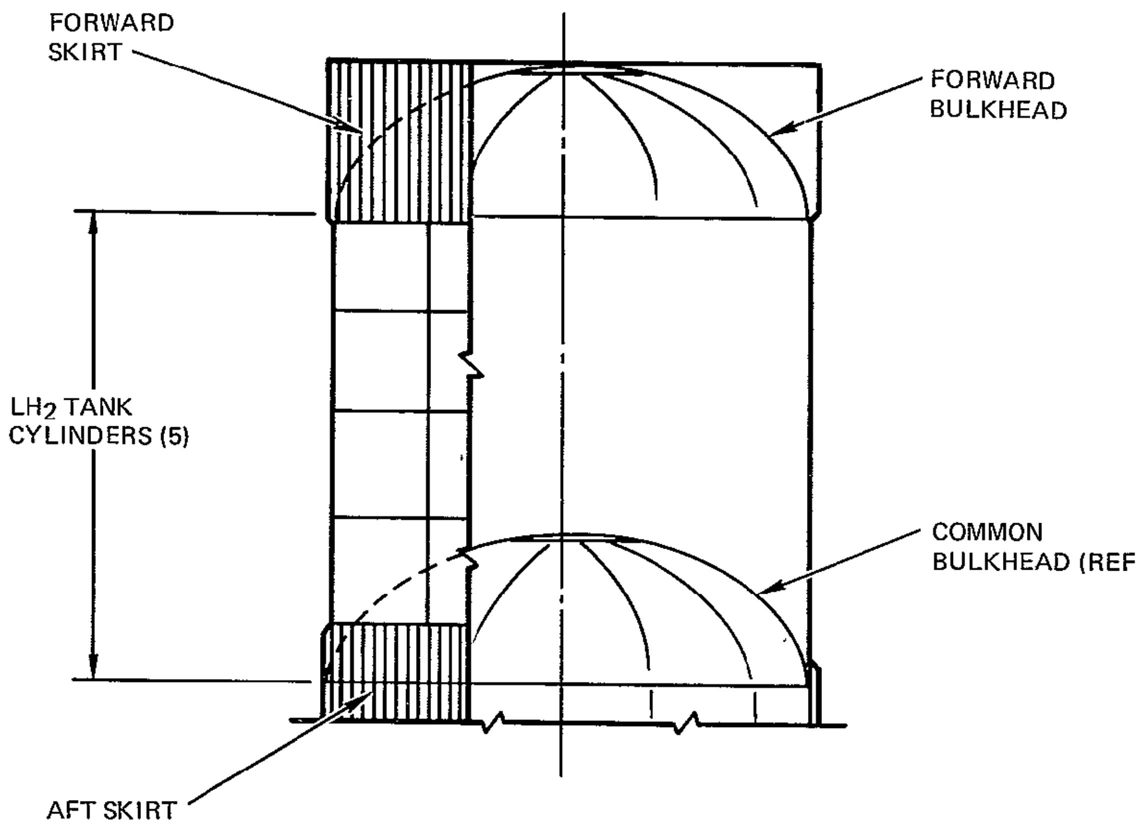


Figure 2-24. ESS LH_2 Tank

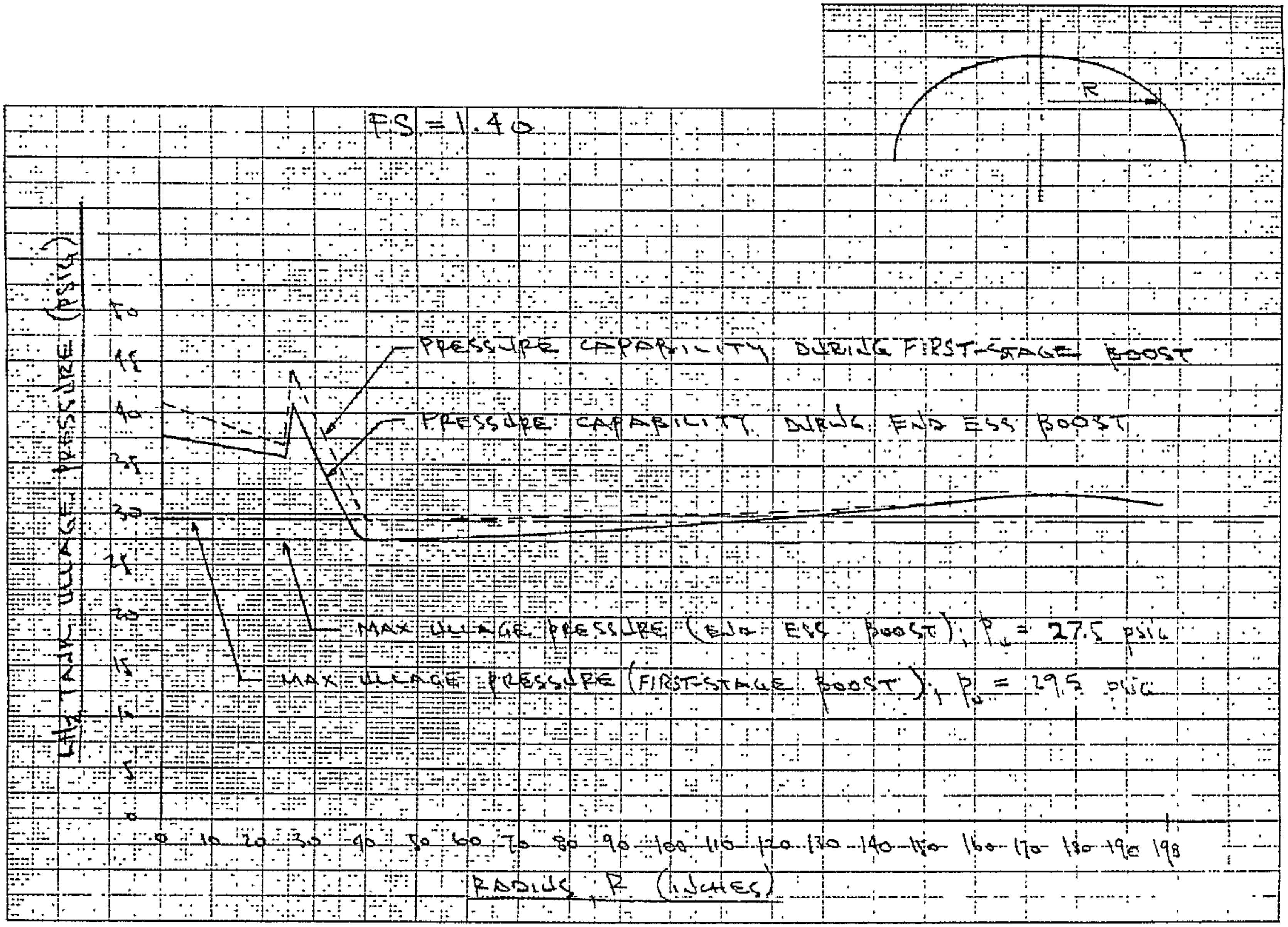


Figure 2-25. ESS LH_2 Forward Bulkhead Limit Pressure Capability

Maximum pressure loading on the tank wall occurs during first stage boost and end ESS boost. As shown in Figure 2-26 the pressure capability of the tank wall exceeds the pressure requirements.

LO_2 Tank. The LO_2 tank is essentially a shell of revolution comprising a forward bulkhead assembly common to the LH_2 tank and an aft bulkhead. Both bulkheads are the same contour as prescribed for the LH_2 tank forward bulkhead. A cutaway view of the LO_2 tank assembly is shown in Figure 2-27.

The common bulkhead is an adhesive-bonded sandwich assembly employing facing sheets of 2014-T6 aluminum alloy and fiberglass/phenolic honeycomb core. The facing sheets of the common bulkhead are fabricated as complete shells by welding twelve formed gore segments and a dollar section. The forward facing sheet terminates in the J-ring section that is welded to the LH_2 tank wall. Waffle-stiffened gore sections are employed in the aft facing to provide shell stability under design collapse pressures.

The LO_2 tank aft bulkhead consists of twelve formed gore segments and a large dollar section of 2014-T6 aluminum alloy. The dollar section incorporates an integral reinforcing ring to provide for attachment of a central sump at the apex of the bulkhead. An access door is provided in one of the gore panels above the dollar section. The gore segments near the equator contain integral waffle stiffening to maintain stability under compression loading.

The LO_2 tank structure is identical to that used on the S-II with the exception of the central sump. This sump is a new design to accommodate the two larger engine feedlines. The fill and drain outlet on the sump remains identical to that of the S-II.

Common Bulkhead Structural Assessment. The common bulkhead is critical in the collapse mode during prelaunch, with both tanks fully loaded, the LH_2 tank at maximum pressure, and the LO_2 tank vent valve open. As shown in the following listing, the factor of safety associated with collapse pressure on the common bulkhead would be slightly less than 1.40.

LH_2 tank ullage pressure	-36.0 psia
LH_2 tank fluid pressure	- 0.6 psia
LH_2 tank total pressure	-36.6 psia
LO_2 tank ullage pressure	+14.7 psia

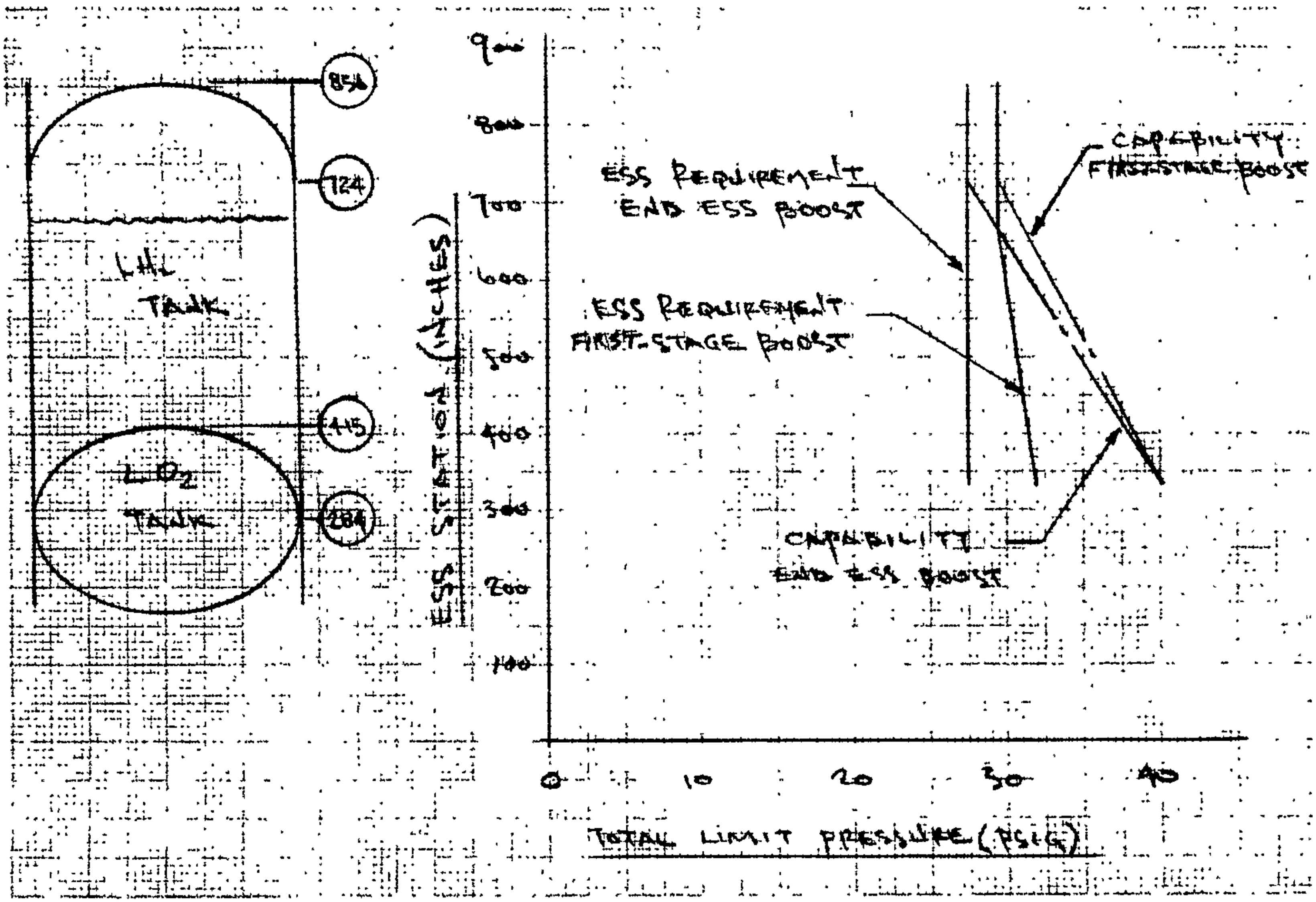


Figure 2-26. ESS LH_2 Tank Pressure Capability

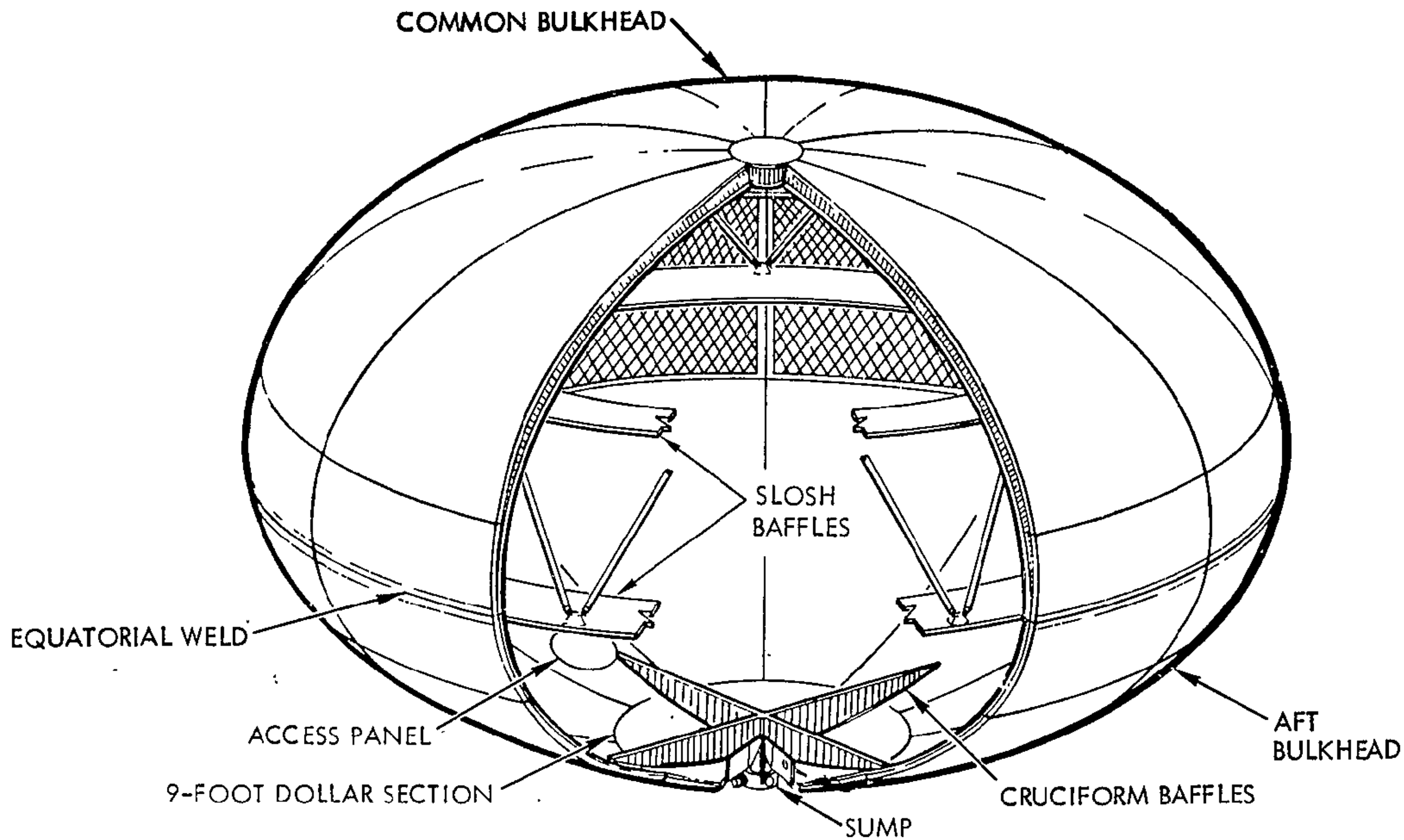


Figure 2-27. ESS LO_2 Tank

Common bulkhead differential limit pressure | -21.9 psid

Common bulkhead capability (ultimate) | -30.1 psid

$$\text{Factor of safety} \quad \frac{-30.1}{-21.9} = 1.37$$

This condition is based on a failure mode where the LO_2 tank has suddenly lost its pressure. This condition presumes a LO_2 vent valve must malfunction in the open position and the console operator fail to react in venting the LH_2 tank. For all other collapse loading conditions, the bulkhead factor of safety is high.

The common bulkhead is also critical during prelaunch for the burst mode with the LO_2 tank at maximum pressure and the LH_2 tank vent valve open. Figure 2-28 compares the maximum differential burst pressure across the common bulkhead to the burst pressure capability.

LO_2 Aft Bulkhead Structural Assessment. The aft bulkhead is critical during first stage boost when maximum acceleration occurs in conjunction with maximum ullage pressure. Figure 2-28 compares the maximum pressure requirement to the bulkhead capability. As shown, the capability of the aft bulkhead well exceeds the pressure requirements.

Aft Skirt

The aft skirt structure for the ESS, while not departing from the S-II-15 structural concepts, does incorporate a portion of the interstage into the aft skirt, resulting in a structure that is 165 inches longer than S-II-15 (252 inches versus 87 inches). This concept is dictated by the all-azimuth launch ground rule that eliminates the possibility of incorporating a jettisonable interstage structure. Therefore, the aft booster attachment structure must be included in the aft skirt. The skirt is a semi-monocoque cylindrical shell extending from ESS Station 31.0 to the attachment to the LO_2 tank at ESS Station 283. The two large booster attachment fittings extend from Station 31.0 to Station 174.5 and are located approximately 140.0 inches each side of Position IV. A small side-load fitting is located just off Position IV at Station 31.0. The aft skirt also provides the attachment for the thrust structure and transmits ESS burn thrust loads. Aerodynamic deflectors are incorporated at the aft end of the skirt at Positions I and III to protect the ESS engines during first stage boost. The general arrangement of the aft skirt is shown in Figure 2-29.

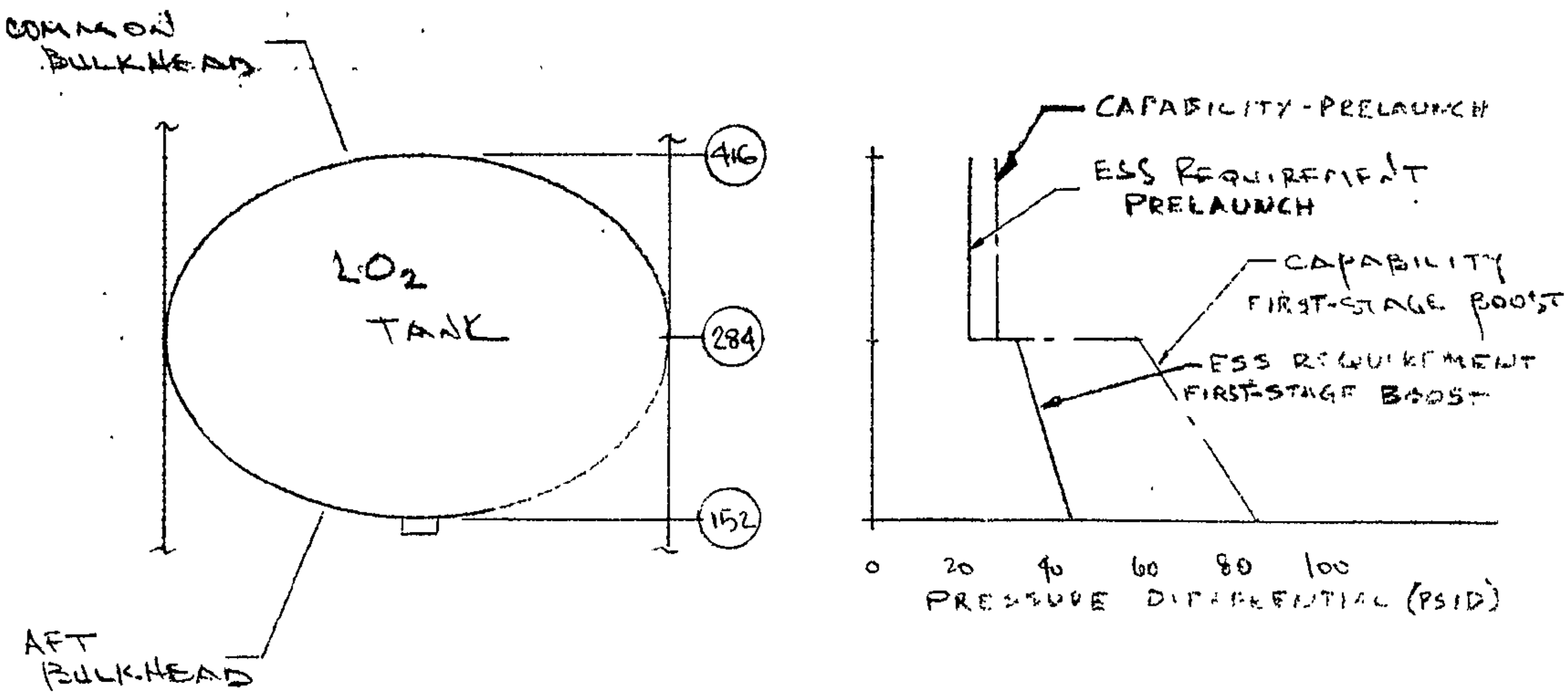


Figure 2-28. ESS LO2 Tank Burst Pressure Capability

Skin-Stringer Structural Assessment. Two sets of internal loads are used to design the ESS aft skirt; the load intensity curves of Paragraph 2.1.1 under Load Intensities and loads from the NARSAMS computer solution. The shell structure was first sized using the general load intensity curves, and since the maximum load intensity values are within the S-II-15 capability, S-II-15 structure is used directly. The only change is the elimination of the stringer thickness increase between Stations 283.0 and 238.0. This is possible because of better definition of the Station 283.0 discontinuity moment and the lower load intensity. This structure along with the booster attachment structure developed in the rough sizing analysis was then input to NARSAMS computer program. External loads representing boost of the MDAC space station and RNS payloads at maximum q alpha and end boost were run in the program. The critical condition for the shell structure as well as the frames was determined to be boost of the MDAC Space Station at maximum q alpha. Figures 2-30 and 2-31 present the resulting ultimate internal shell loads. Plotted are the skin shear flows and axial load intensity at mid bay between each of the aft skirt frames. In the undisturbed areas, the skin-stringer sizing is based on column stability requirements using stringer loads generated from the compressive load intensity curves. Skin gages are set using the shear flow curves from the computer solution, using a 40,000-psi panel shear allowable stress. Skin gages taper from 0.400 inch near the SS booster attach fittings to a minimum of 0.071 inch in the undisturbed areas. The skin is a constant 0.071 inch in the upper three bays (Stations 174.5 to 283).

The S-II-15 stringers, with a 0.125 inch crown, 0.090 inch side walls, and 5.76 inch spacing, are adequate for most of the aft skirt. The lower load intensity allows the use of these stringers with larger frame spacing on ESS. The stringers in the vicinity of the booster attachment fittings are reinforced by the addition of straps on the crowns. Stringer and skin material is 7075 aluminum alloy. S-II-15 thermal environment is assumed for ESS design. Local insulation will be incorporated at any hot spot in order to keep maximum structural temperatures within the S-II-15 limits.

Frame Structural Assessment. The aft skirt contains six frames which fall into two general categories: stability frames and load frames. The frames at Stations 206.5 and 238.0 are sized by their "EI" requirement to preclude failure of the shell by general instability. This requirement is set by the shell compressive load intensity and the spacing of the frames. These two frames have an overall depth of eight inches.

The load frames are at Stations 31.0, 82.5, 128.0, and 174.5. Their loading is taken from the NARSAMS computer solution. Figure 2-32 presents the internal loads for the frame at Station 31.0. This frame is the aft booster attachment frame, and is sized as a fifteen-inch-deep box section. The box section is employed to react torsional loads generated by

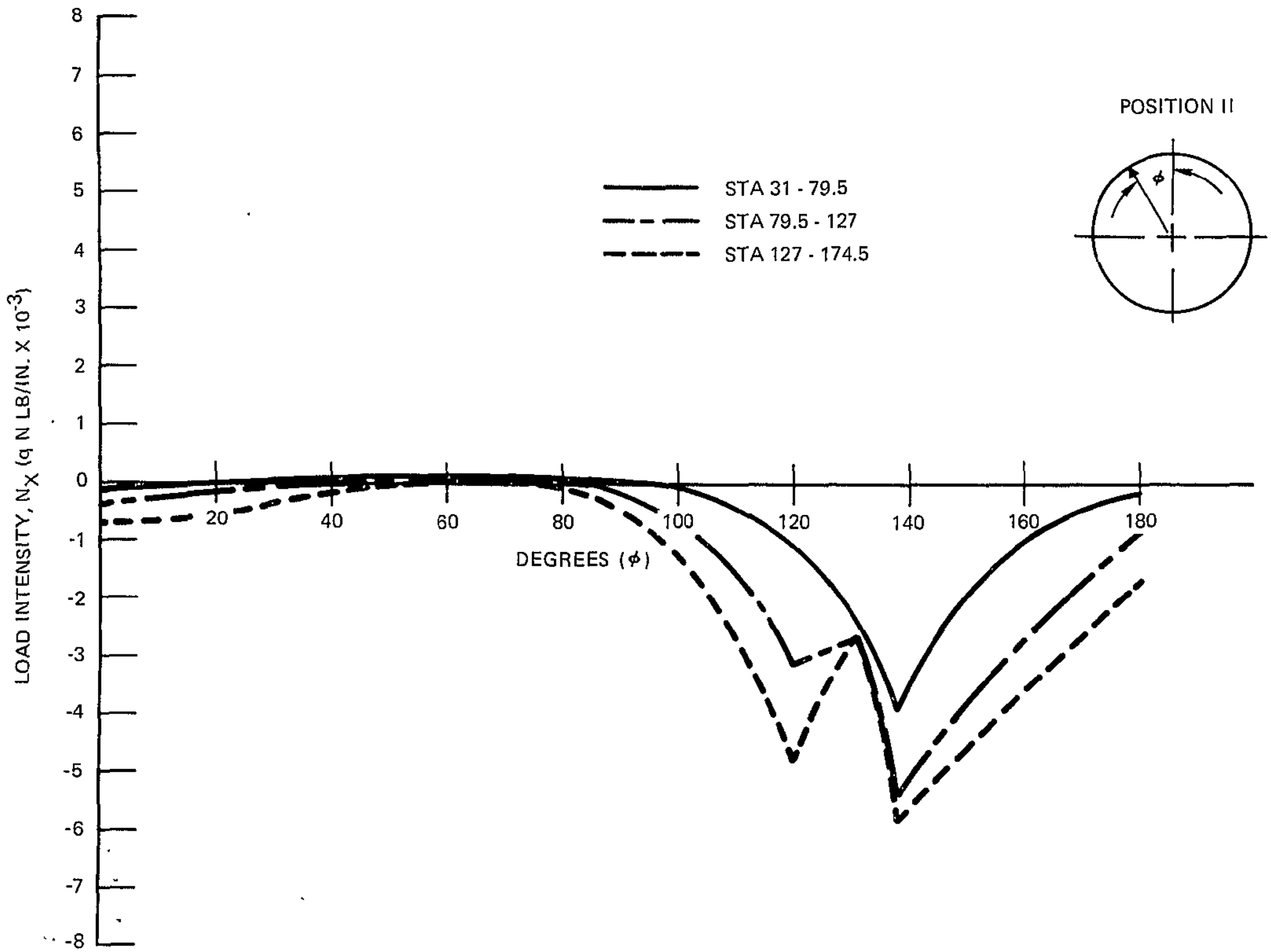


Figure 2-30. ESS MDAC Aft Skirt Shell Loading Intensity/MDAC Payload (Sheet 1 of 2)

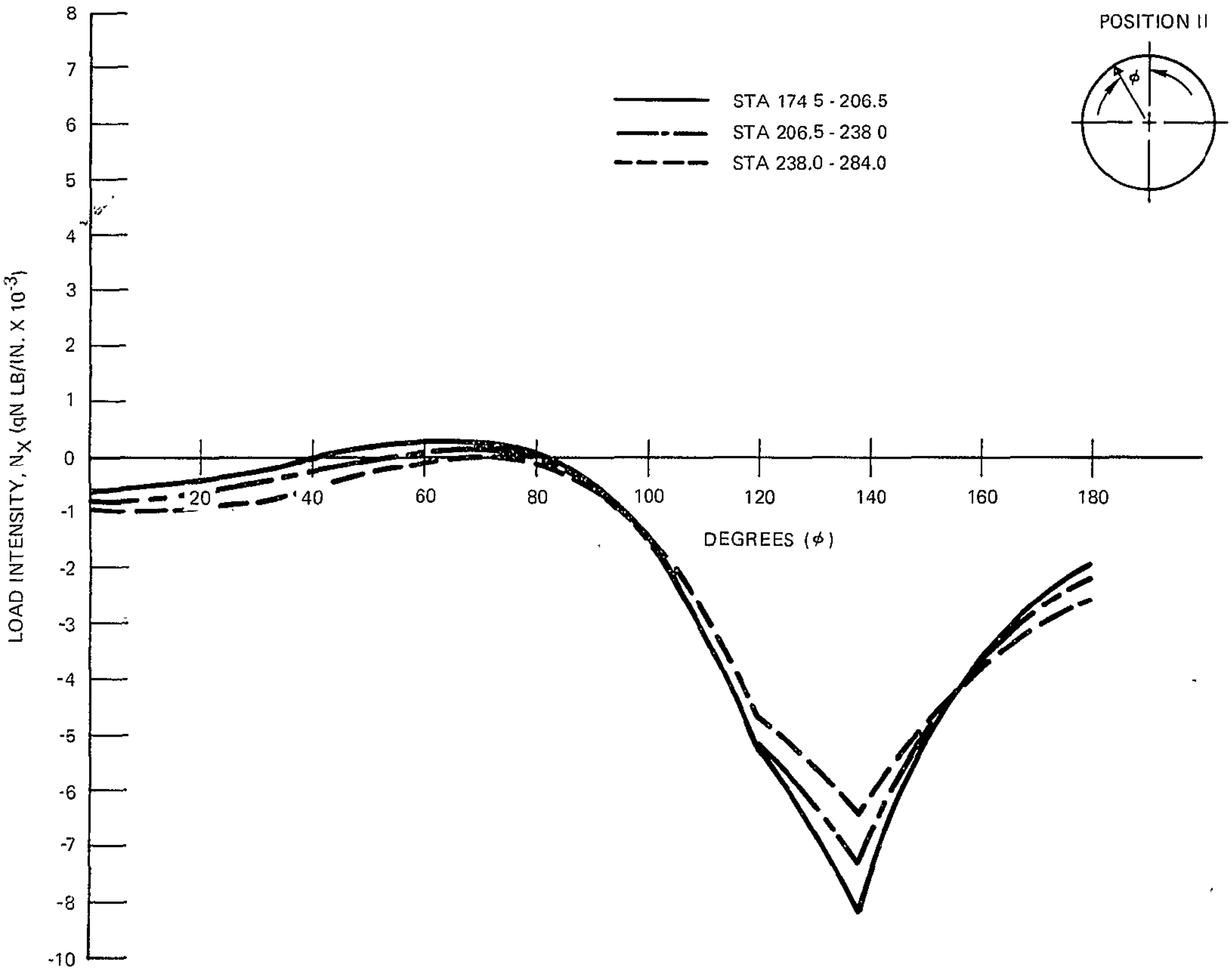


Figure 2-30. ESS MDAC Aft Skirt Shell Loading Intensity/MDAC Payload (Sheet 2 of 2)

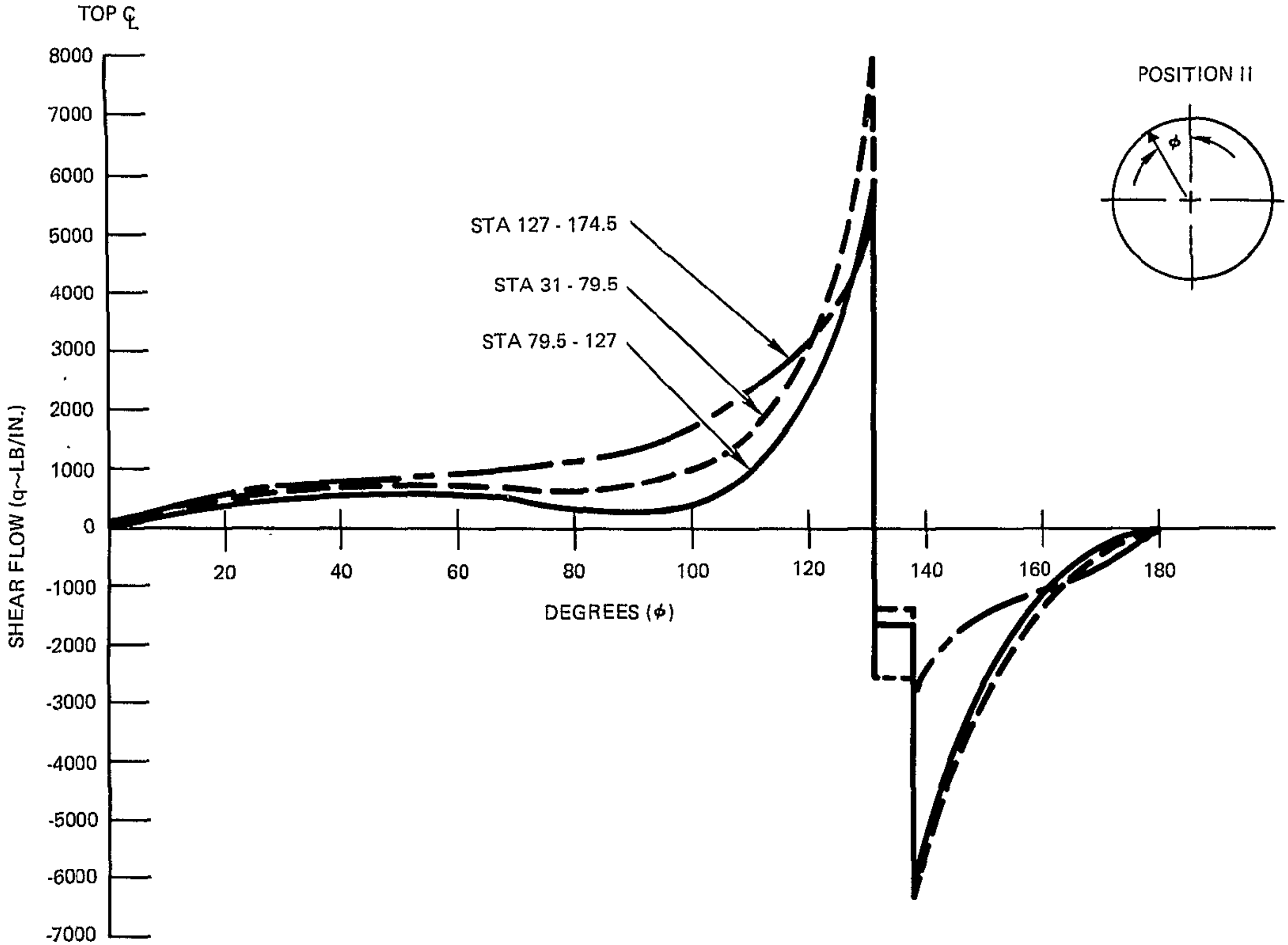


Figure 2-31. ESS Aft Skirt Shear Flow (MDAC Payload) (Sheet 1 of 2)

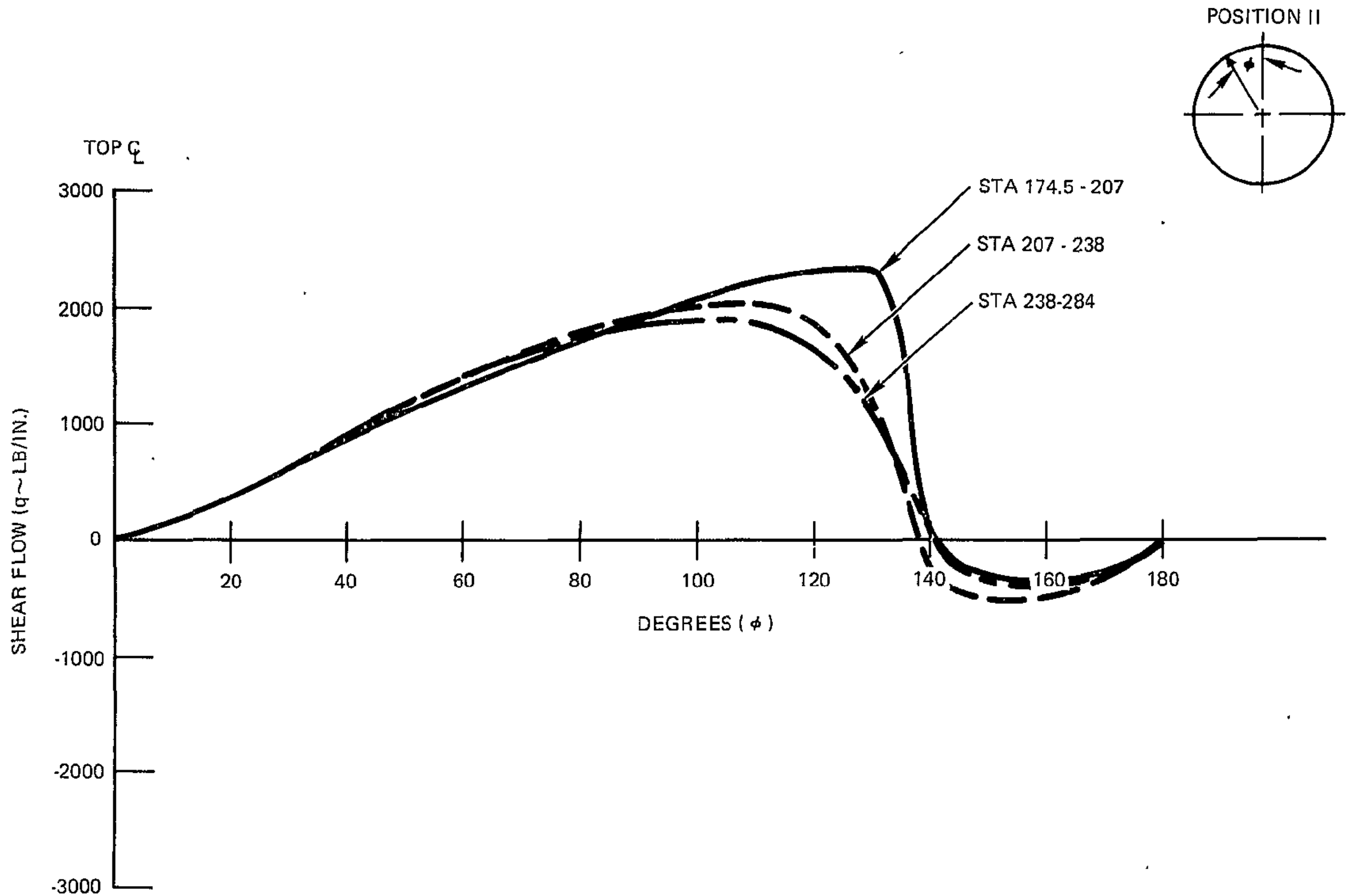


Figure 2-31. ESS Aft Skirt Shear Flow (MDAC Payload) (Sheet 2 of 2)

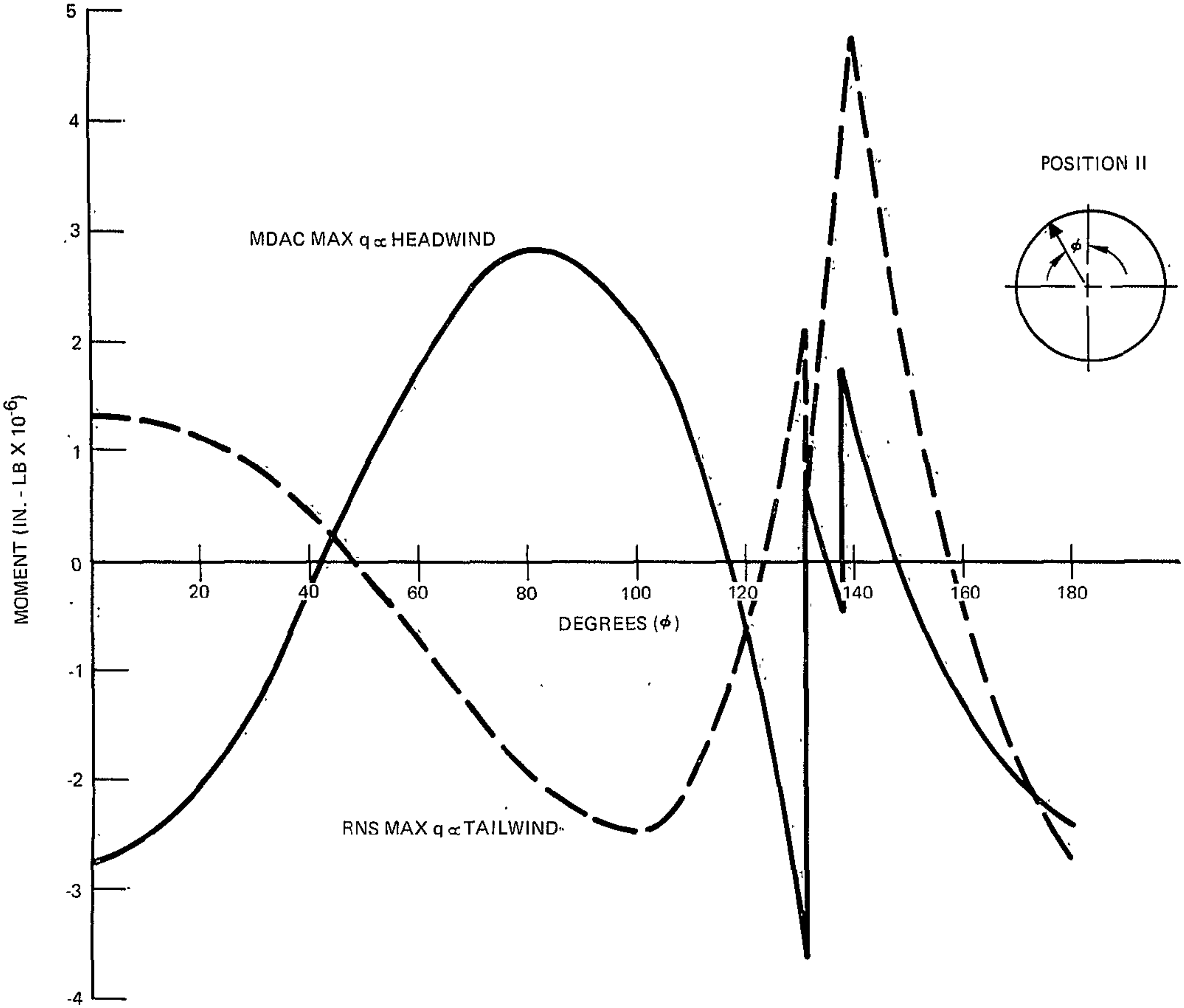


Figure 2-32. ESS/Frame, Station 31, Internal Loads (Sheet 1 of 2)

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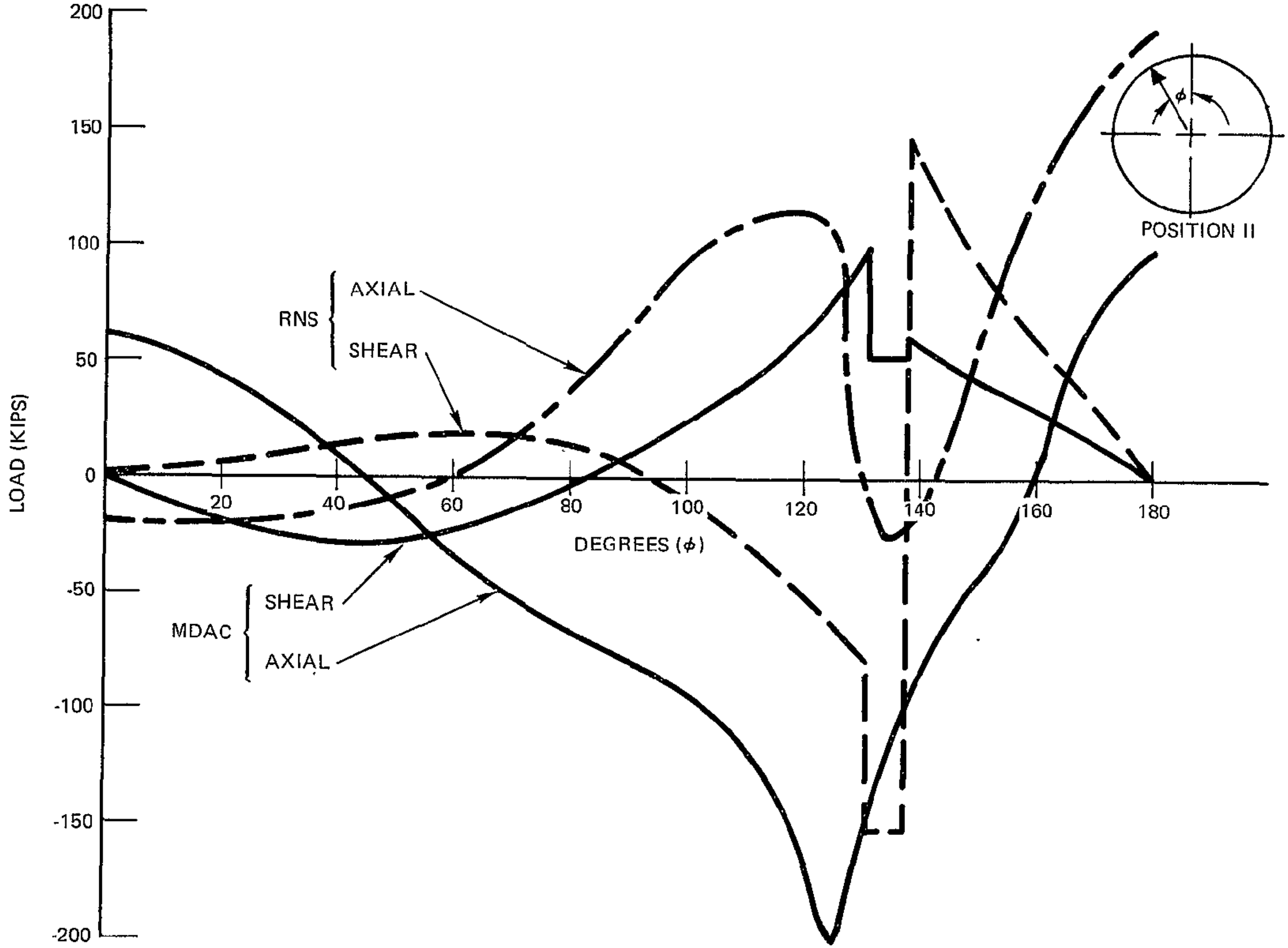


Figure 2-32. ESS/Frame, Station 31, Internal Loads (Sheet 2 of 2)

the booster attachment fittings. The internal loads for the frames at Stations 82.5, 128.0, and 174.5 are closely comparable with the loads for 174.5 being maximum. Therefore, the 174.5 loads are used to size all three of these frames. These loads are presented in Figure 2-33. The frame at Station 174.5 is also subjected to thrust cone load conditions during ESS burn. However, these loads are not critical. All of the load frames are sized using 70,000 psi allowable cap stress and 40,000 psi allowable shear web stress. All of the frames are constructed of 7075 aluminum alloy.

Fitting Structural Assessment. Shown in Figure 2-34 is a diagram of the ESS/booster attach fittings and the ultimate external applied loads for the critical condition (MDAC space station at maximum q alpha). These fittings are required to take the aft booster attach loads at two points and then distribute the loads into the aft skirt shell structure. The horizontal cross tie connecting the two fittings (shown in View A-A) is required to react the side loads (P_Y) generated by the angularity of the booster attachment struts. Sizing of the fittings is determined by the drag loads (P_x) and the moment caused by its offset from the aft skirt shell. The fitting is constructed of 7075 and 7079 aluminum alloys, with allowable cap stress of 70,000 psi and allowable shear web stresses of 40,000 psi. A relatively small side load fitting is attached to the skirt just off Position IV at Station 31.0. Small local reinforcements of the frame at Station 31.0 are required to accommodate this loading.

Thrust Structure

The ESS thrust structure is sized to support two orbiter engines and two OMS engines. The vacuum thrust for the main engine at the emergency power level used for analysis is 690,000 pounds. The OMS engine thrust used is 10,000 pounds.

The two-orbiter engine concept employs a new conical frustum for the thrust structure. This cone, like the S-II-15 cone, is stabilized by a crossbeam. Differences in geometry of the larger engine require a greater distance between engine interfaces and the LO_2 tank sump. This generates extra space inside the thrust cone which allows for the LO_2 and LH_2 APS tanks to be located inside the cone. The basic structural elements of the S-II-15 thrust cone are maintained. Engine thrust and gimbal loads are reacted directly by thrust longerons. The skin is stiffenered by hat-section stringers and the shell is stabilized by ring frames and a crossbeam at the aft end to avoid adding an excessively large thrust frame. This beam provides an ideal mounting for the APS LO_2 tank.

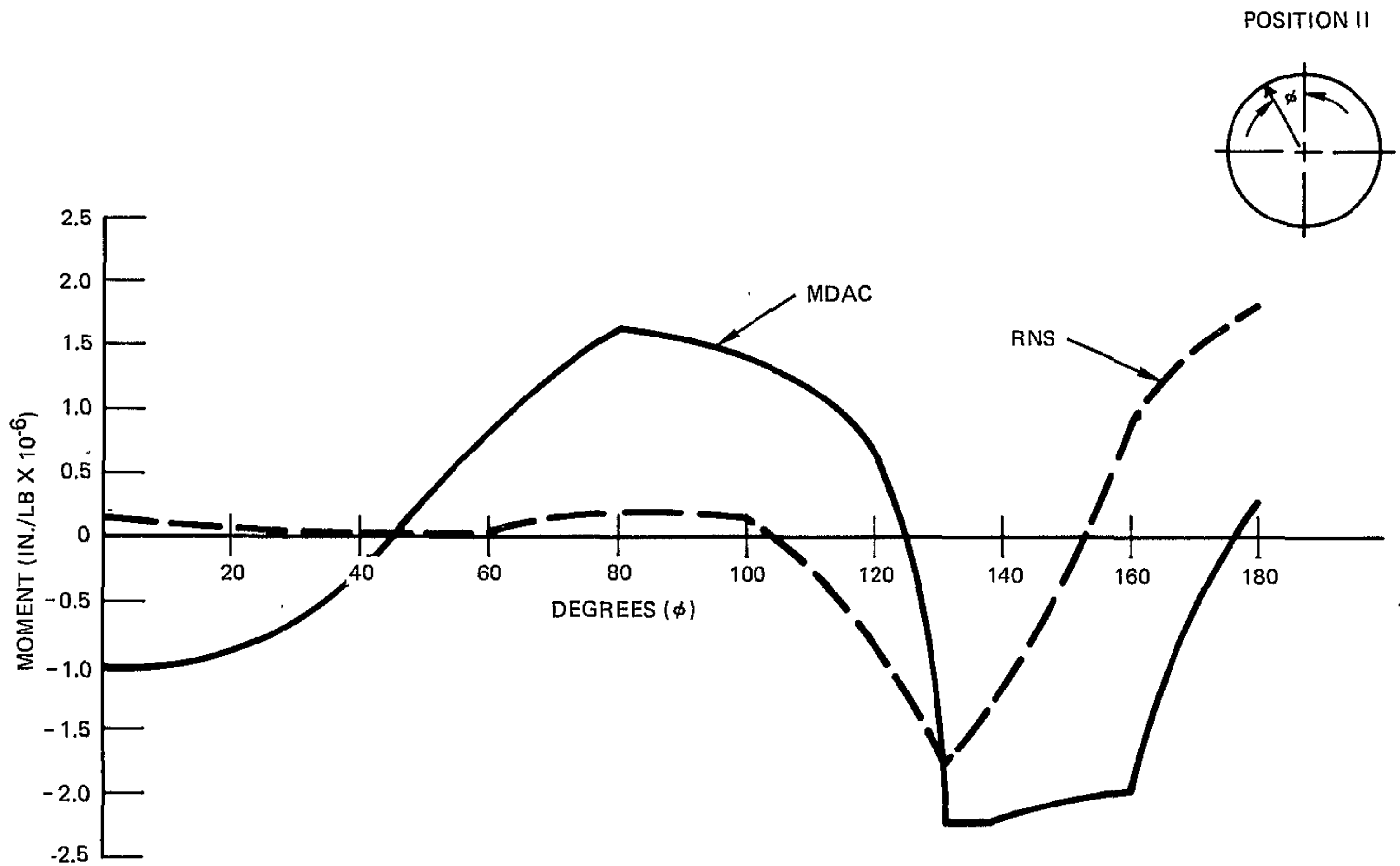


Figure 2-33. ESS/Frame, Station 174.5, Internal Loads (Sheet 1 of 2)

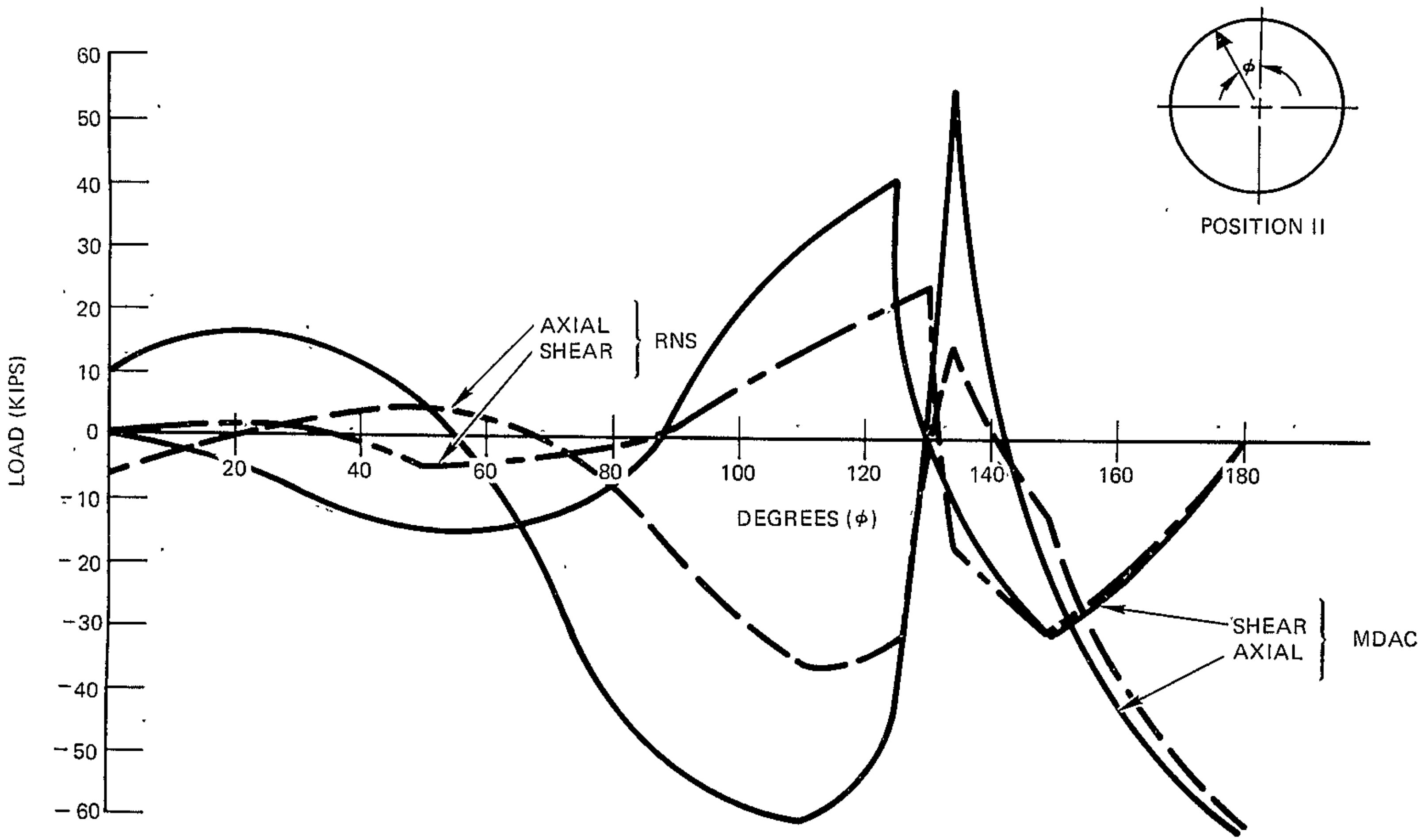


Figure 2-33. ESS/Frame, Station 174.5, Internal Loads (Sheet 2 of 2)

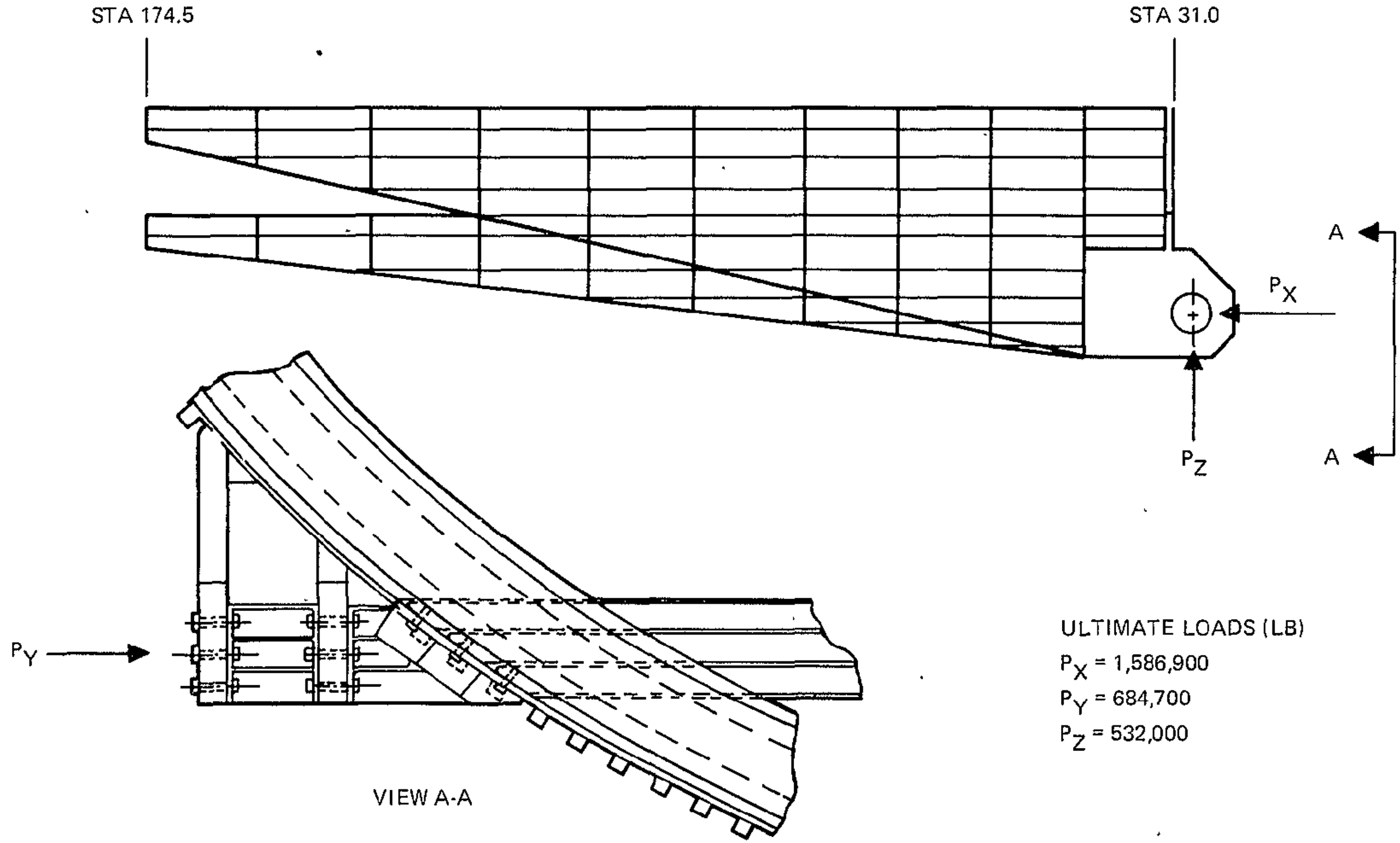


Figure 2-34. Booster Attach Fitting, Aft Skirt

Incorporation of two orbiter OMS engines as the OMS requires the addition of support fittings and primary structure to the basic ESS two-orbiter engine thrust cone. Structural modifications are required to support the OMS engines, provide any needed thermal protection, and to support miscellaneous mechanical and electrical systems components.

The concept is a skin stringer shell structure with internal ring frames. The aft frame is used to react kick loads from engine thrust. Crossbeam ties are incorporated with this frame to assure adequate stiffness to prevent engine compliance problems. Thrust longerons are included to react engine thrust, eccentric moments, and distribute engine loads into the shell by shear lag. The forward frame provides the attachment to the aft skirt and reacts the kick loads produced when thrust structure skin-stringer loads are reacted by the aft skirt. This frame must also work as a stability frame for both the thrust structure and the aft skirt.

Although structural similarities between the S-II-15 thrust structure and the ESS concept may be used to establish the validity of employing common analysis methods, such practice cannot be extended to the structural test requirements. The greater thrust of the orbiter engines (1,380,000 pounds versus 1,200,000 pounds) and the less uniform application to the conical structure (two places versus five places) dictates that full-scale verification tests be performed on the new thrust structure for the ESS. This testing will be similar to the testing performed on the S-II thrust structure to verify integrity under the various thermal and loading conditions.

OMS engine loads are small in comparison with the engine thrust loads to which the thrust structure is designed. However, the OMS engine loads will be incorporated into the test program to verify the integrity of the engine attach structure and local fittings.

The ESS thrust structure configuration and structural diagram is shown in Figure 2-35. The geometry is set to maintain growth potential to a chemical interorbital shuttle (CIS) application. Recent studies on the CIS require an aft docking adapter compatible with the space tug which makes the large base frame diameter necessary.

Design Assumptions. These assumptions are as follows:

1. Two orbiter engines will be employed as main ESS engines

2. Vacuum thrust is 690,000 pounds (limit) and 966,000 pounds (ultimate)

3. Engine precant is 13 degrees outboard (1-degree compliance)

4. Engine gimbal is ±7 degrees along 45 degree lines and ±10.5 degrees (pitch and yaw)

5. X_B 52 and X_B 105 frames react kick loads

Load Conditions. The following load conditions are used to determine the maximum loading on the various structural elements.

Condition I. Symmetrical 12 degree precant

Condition II. Hard-over gimbal

one engine 12° + 10-1/2° = 22-1/2° outboard; other engine 12° - 10-1/2° = 1-1/2° outboard

Condition III. Hard-over gimbal; both engines in null radial position (12° outboard precant) but gimbaled 10-1/2° in pitch plane

Condition IV. One engine out; other engine 12° + 10-1/2° = 22-1/2°

Structural Assessment. The method of analysis used to size the thrust structure was to use the load conditions shown above to establish the "first cut" sizes of the structural members. The structure and critical load conditions (Condition I and Condition II) were then input to the NARSAMS computer program. The NARSAMS structural model consisted of half of the thrust structure and half of the aft skirt from one bay aft of the thrust structure attachment to the joint with the LO_2 tank. Half of the structure is modeled because the structure is symmetrical. The program solves for the internal loads in the structural elements. These internal loads were then used to redefine the sizes of the structural members. A summary of the internal loads and structural sizing is presented in the remainder of this section. All sizing shown is based on the use of 7075-T6 and 7079-T651 aluminum alloys.

Skin and Stringers. Figures 2-36 and 2-37 show the thrust structure shell compressive load intensity and skin shear flow. The loading is plotted at mid-bay between frames and spanwise for one-quarter of the shell circumference. Figure 2-35 delineates the resulting skin and stringer thicknesses. Tapered skins and stringers are employed in order to achieve a more efficient "shear lag" load distribution and reduce shell weight. The thrust cone shell is designed by the critical condition of "two engines on 0° gimbal" (Condition I). Allowable stresses used for design are consistent with S-II-15 values. These are an average of 15,000-psi compression stability (column buckling), and 27,500-psi shear (skin panels).

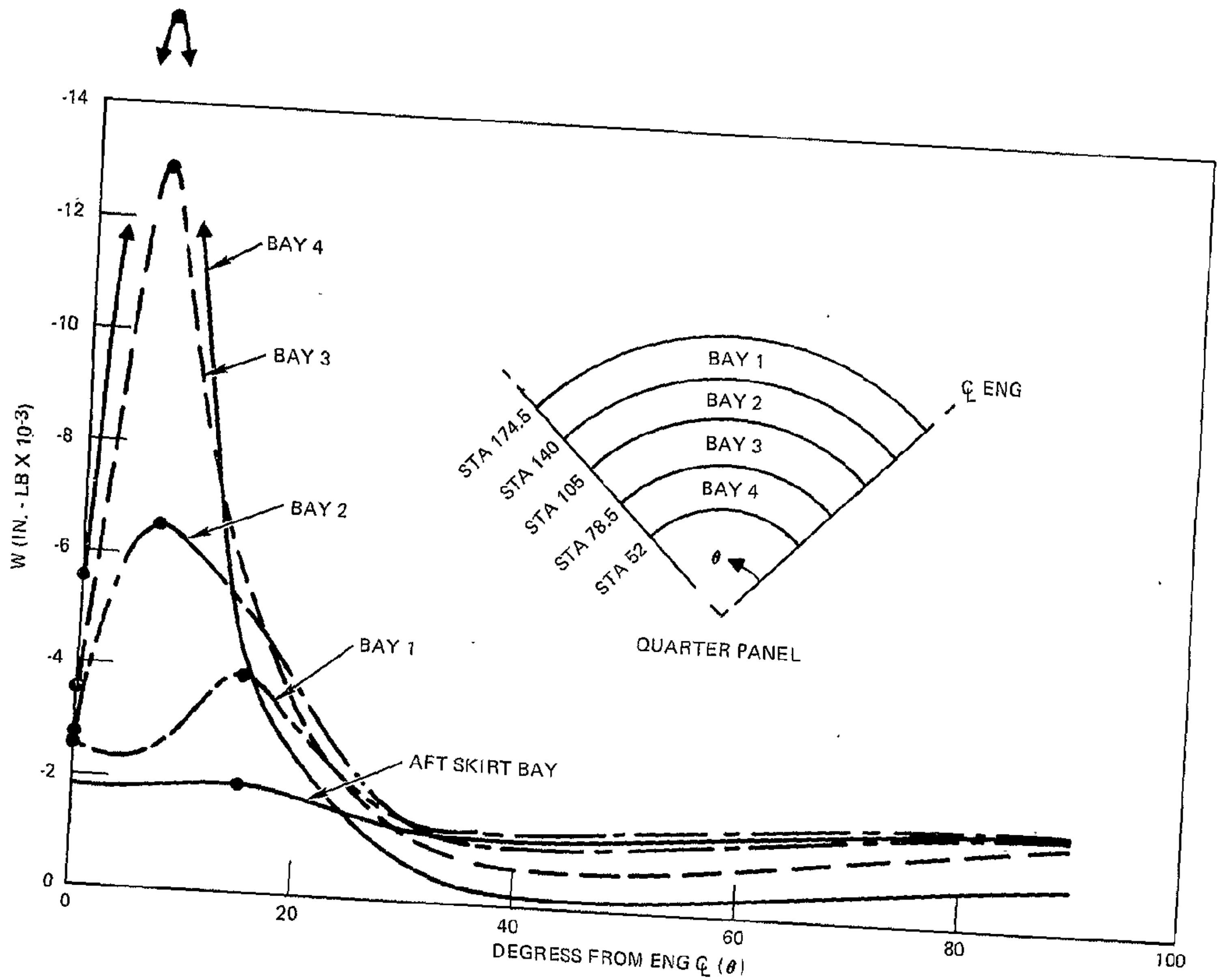


Figure 2-36. Thrust Structure Shell Loading Intensity

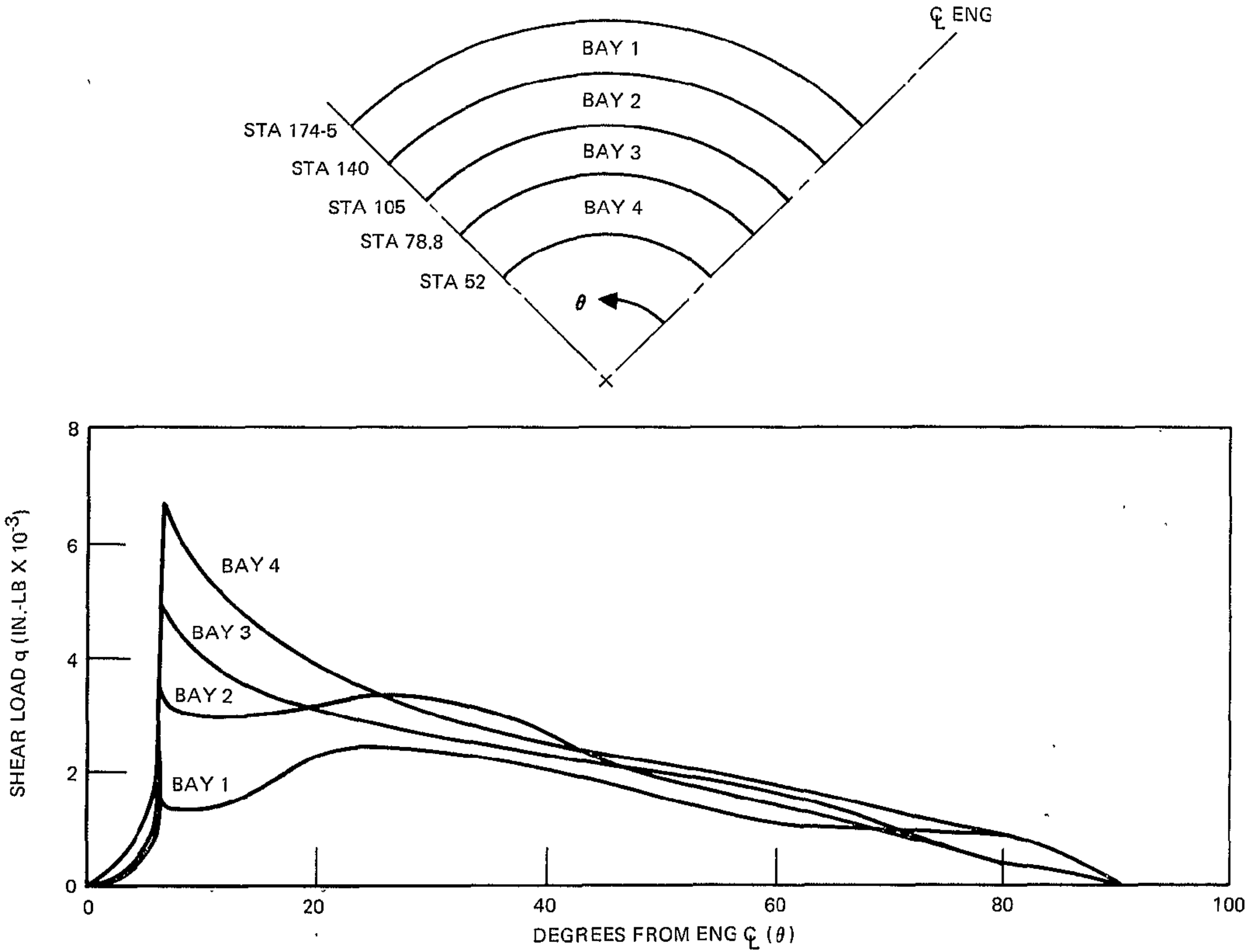


Figure 2-37. Thrust Structure Skin Shear

Frames. There are five frames in the thrust structure assembly. All must serve to preclude failure of the skin stringer shell by general instability. The frame sections necessary to react the applied loads, however, are in excess of the moment of inertia requirement for general stability. Three of the frames (at Station 52, Station 105, and Station 174.5) are heavily loaded, and the other two (at Station 78.5 and Station 140) are lightly loaded and approach stability frames. The frame at Station 52 functions as a kick frame to react loads from longeron bending and cone reactions to engine thrust. This frame must also react engine thrust components that arise from the engines being gimbaled tangentially to the frame. The frame at Station 105 functions as a kick frame to react loads from longeron bending. The frame at Station 174.0 is the cone forward end frame and provides the attachment to the aft skirt. During ESS burn it is loaded by loads generated when the thrust structure skin-stringer loads are reacted by the aft skirt. However, loads produced by the aft skirt booster attach fittings comprise the critical loading for this frame and sizing is presented in the aft skirt discussion.

Figures 2-38 through 2-42 summarize the internal moment, shear, and axial load of the frames at Stations 52 and 105. Figure 2-35 defines the resulting frame sections required.

The heavily loaded base frame at Station 52 is designed by Condition I, "Two engines on 0° gimbal." Stress allowables of 70,000-psi tension, 60,000-psi compression, and 41,000-psi shear were employed in sizing this frame. The forward kick frame at Station 105 is designed by Condition II, "One engine out hard over gimbal." Working stresses are consistent with those used for the frame at Station 52. Frame depths shown were chosen by a preliminary design evaluation and are considered a point for optimization in subsequent design refinement.

Internal loads are not presented for the two lightly loaded frames since they are small, and only insignificant differences between load conditions exist. The frame at Station 78.5 is interrupted at four places by large support fittings for the APS LH_2 tanks, and at two places by the main engine longerons. Therefore, this is referred to as a partial frame. Both of these frames (at Stations 78.5 and 140) provide support points for the APS LH_2 tanks. Consequently, these frames are designed for a combination of APS tank loads and cone shell secondary loads. Because of the light sections, the allowable cap stress is limited to 50,000 psi in their design.

Longerons. The longeron design is similar to the S-II-15 with a heavy section for bending plus axial loading from the base frame to the forward kick frame at Station 105, and a light section for axial loading from the forward kick frame to the top of the cone. Internal loads from the computer program solution for the longerons are presented in Figure 2-43.

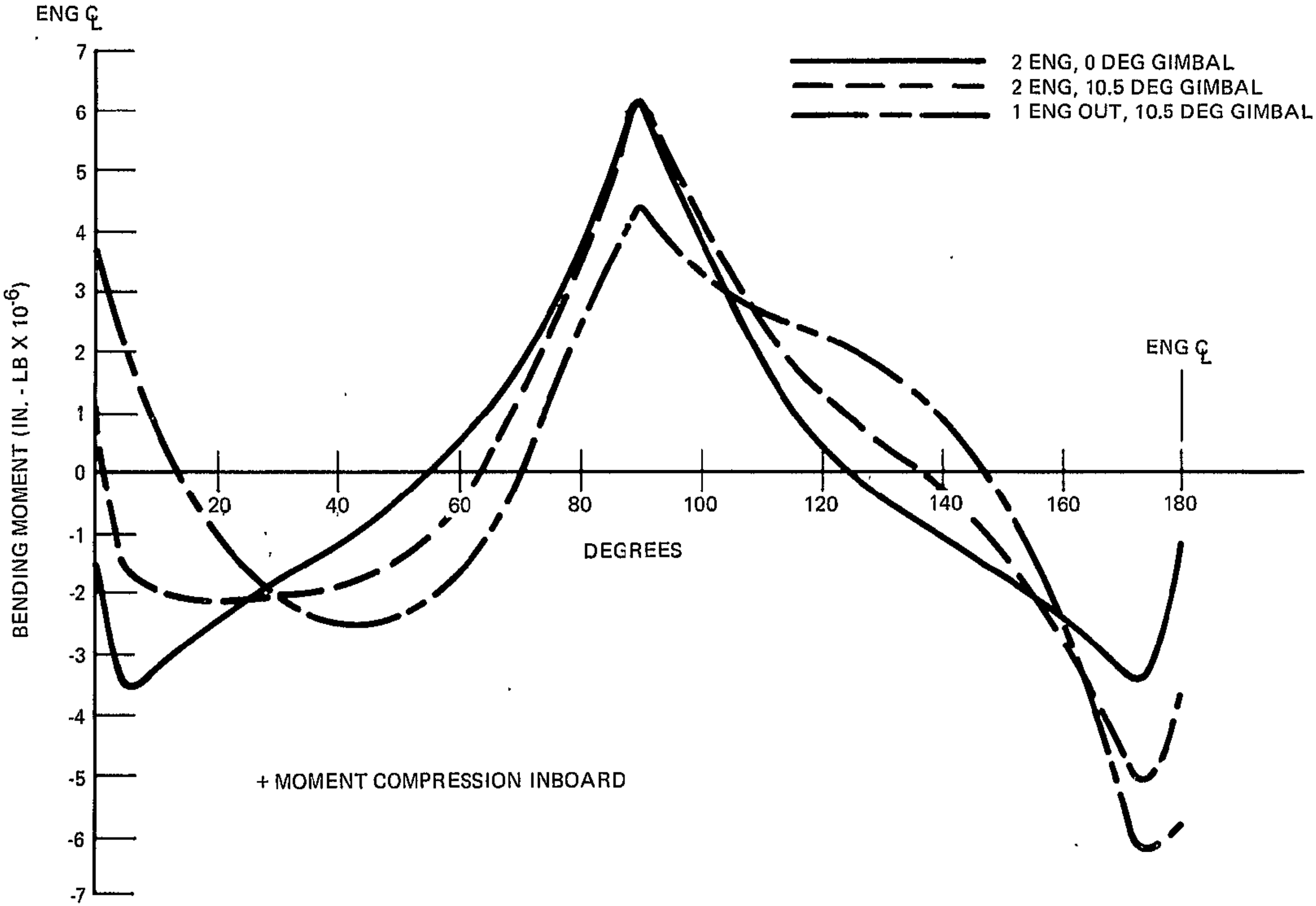


Figure 2-38. Frame X_B 52 Bending Moment

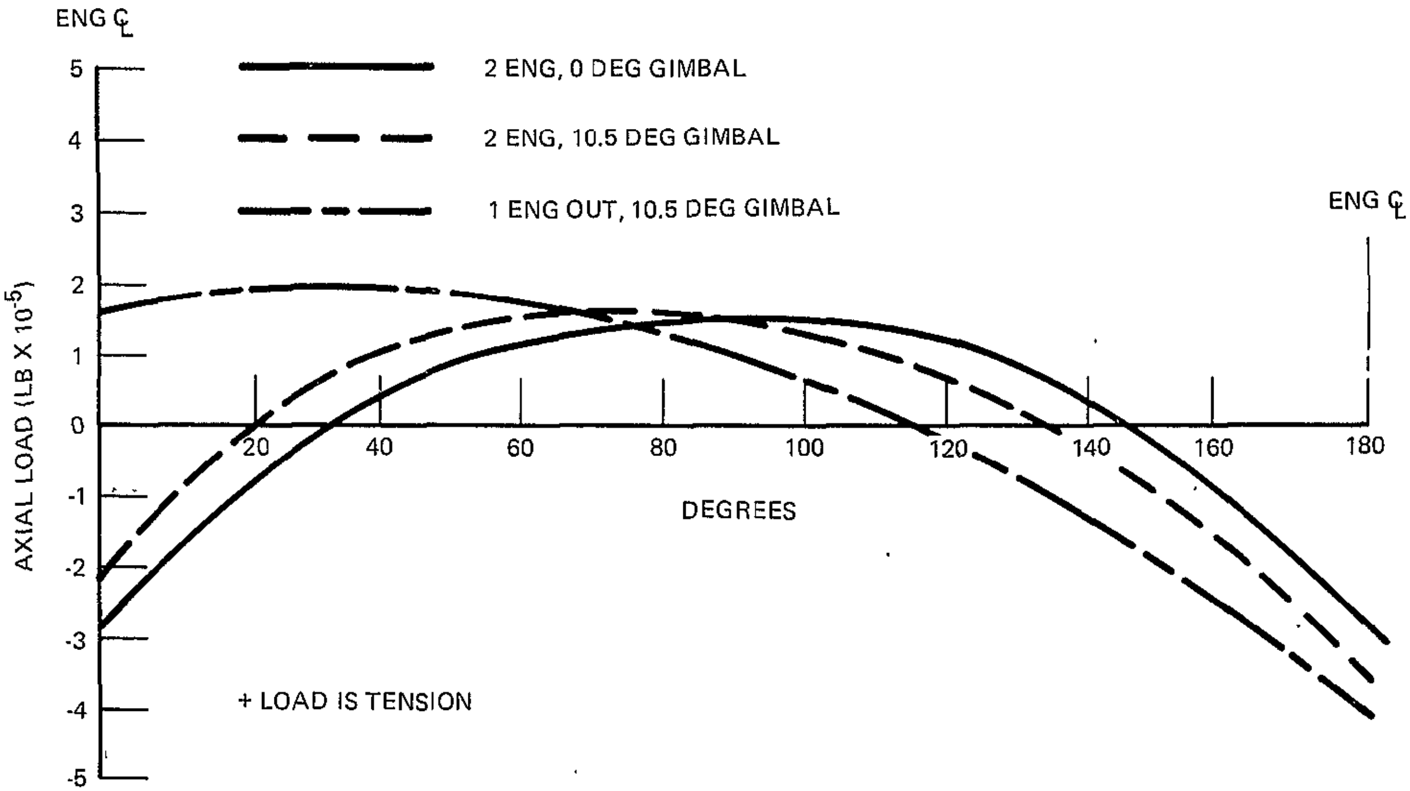


Figure 2-39. Frame X_B 52 Axial Load

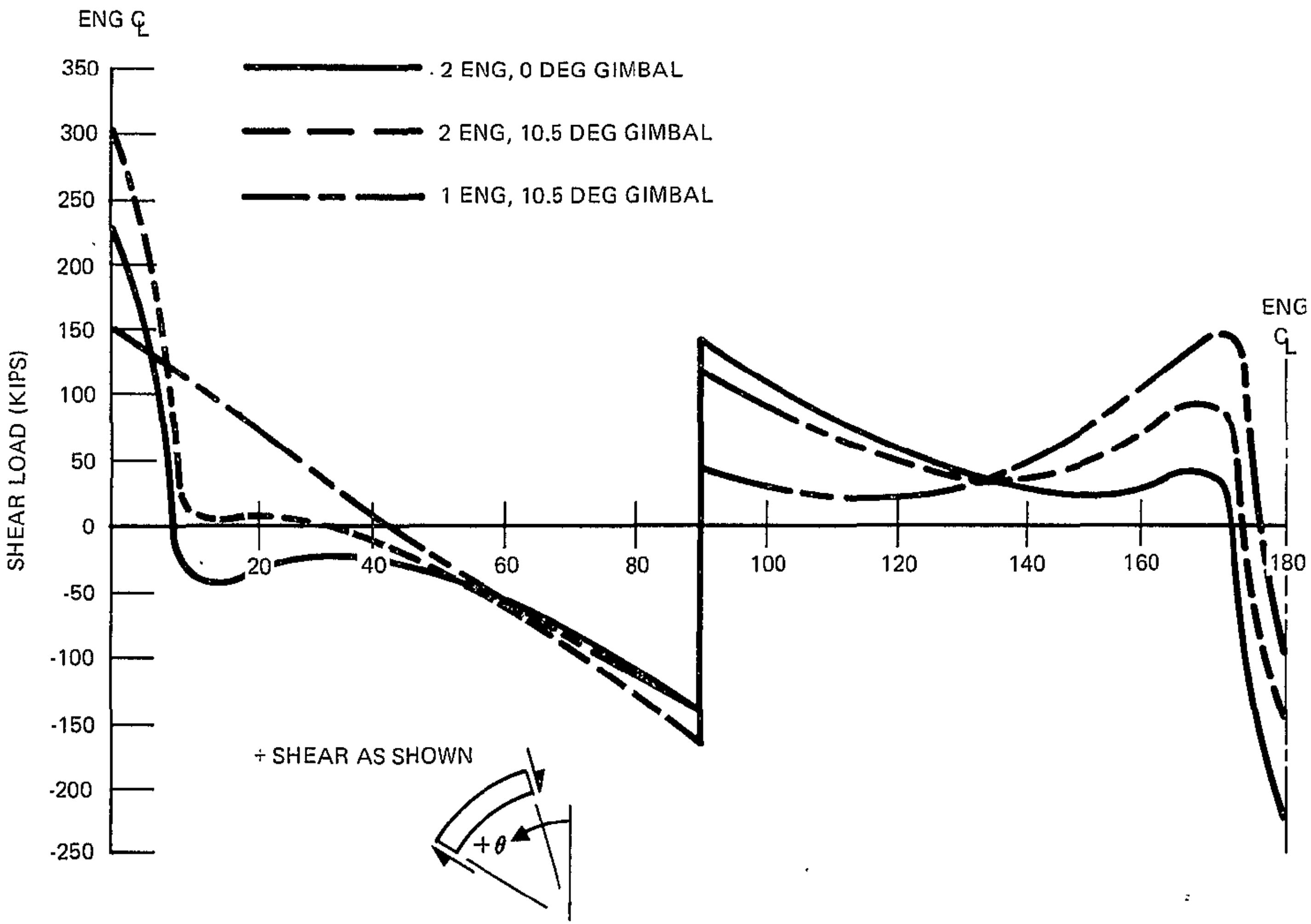


Figure 2-40. Frame X_B 52 Shear Load

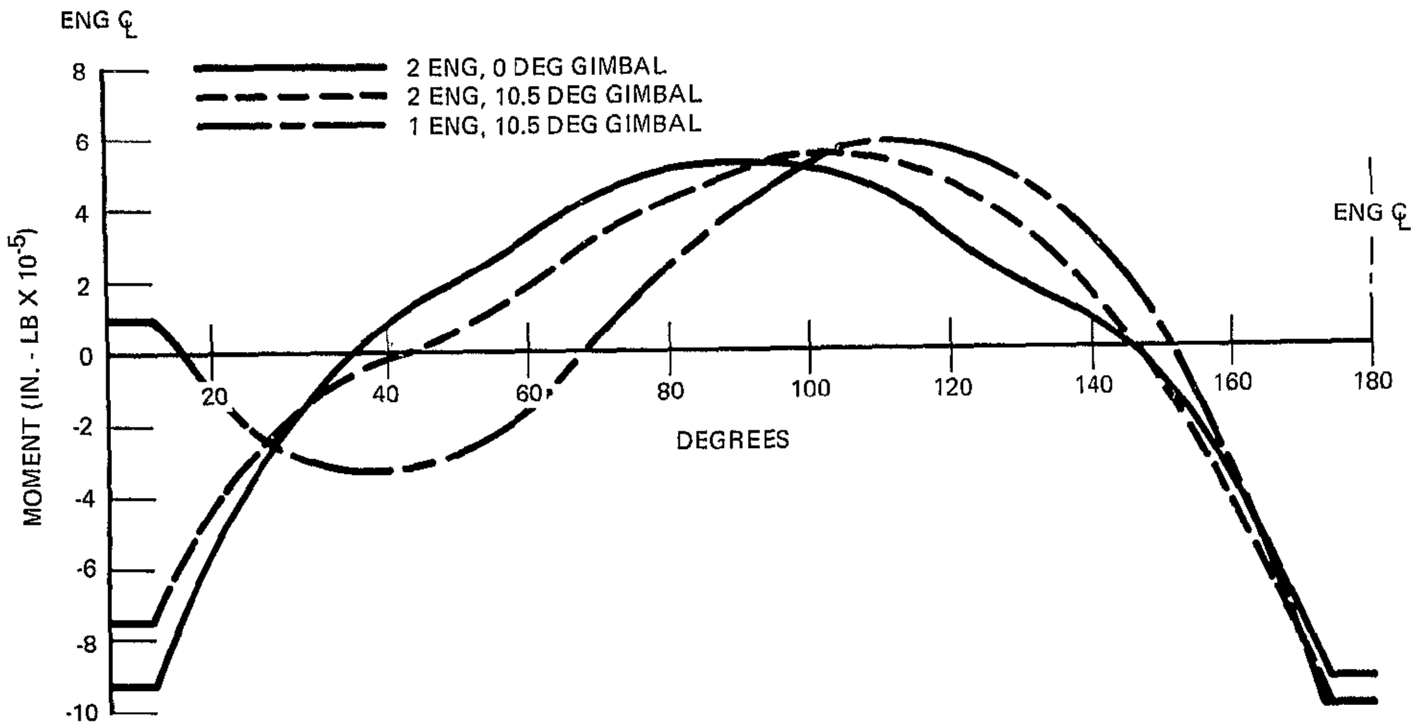


Figure 2-41. Frame X_B 105 Bending Moment

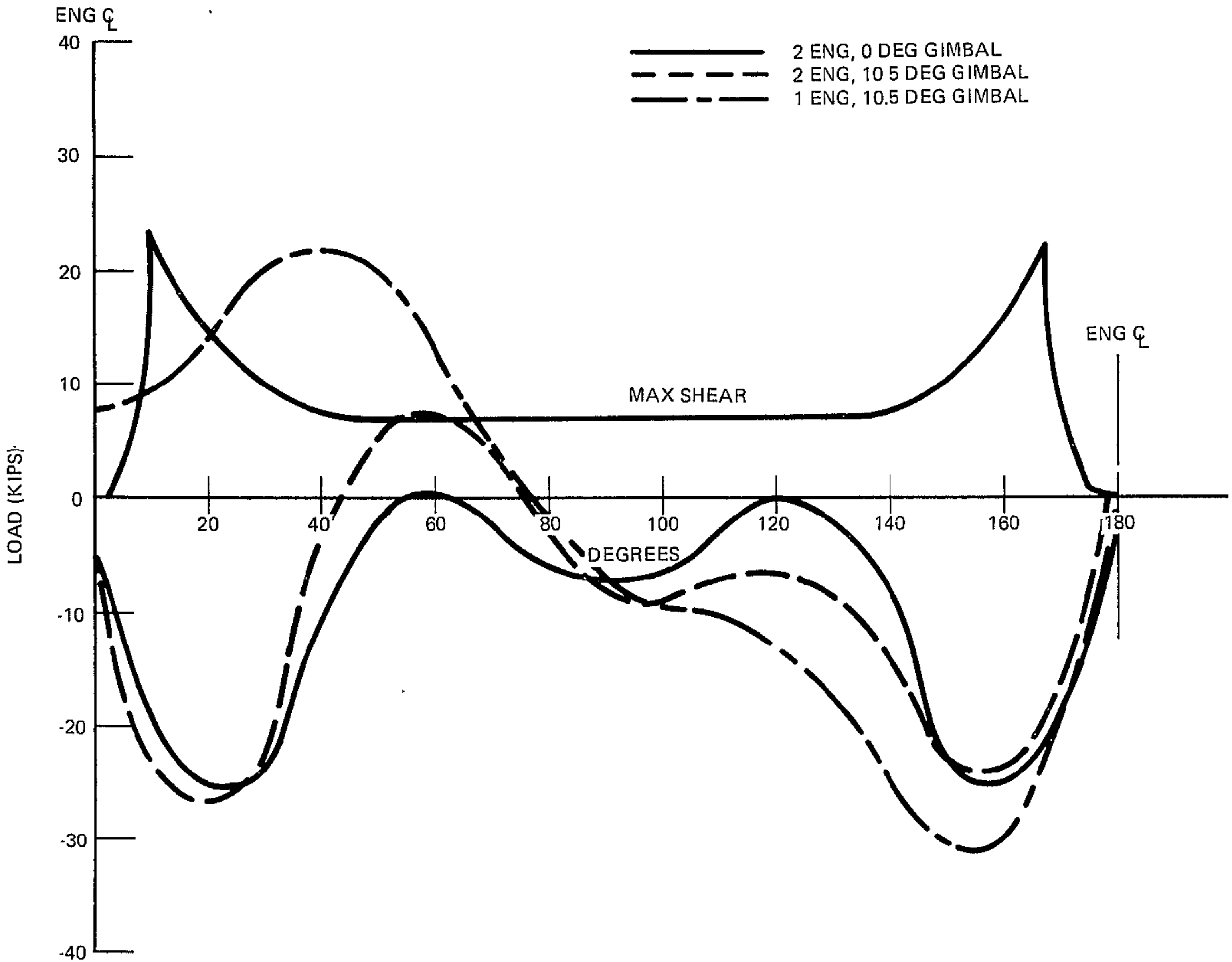
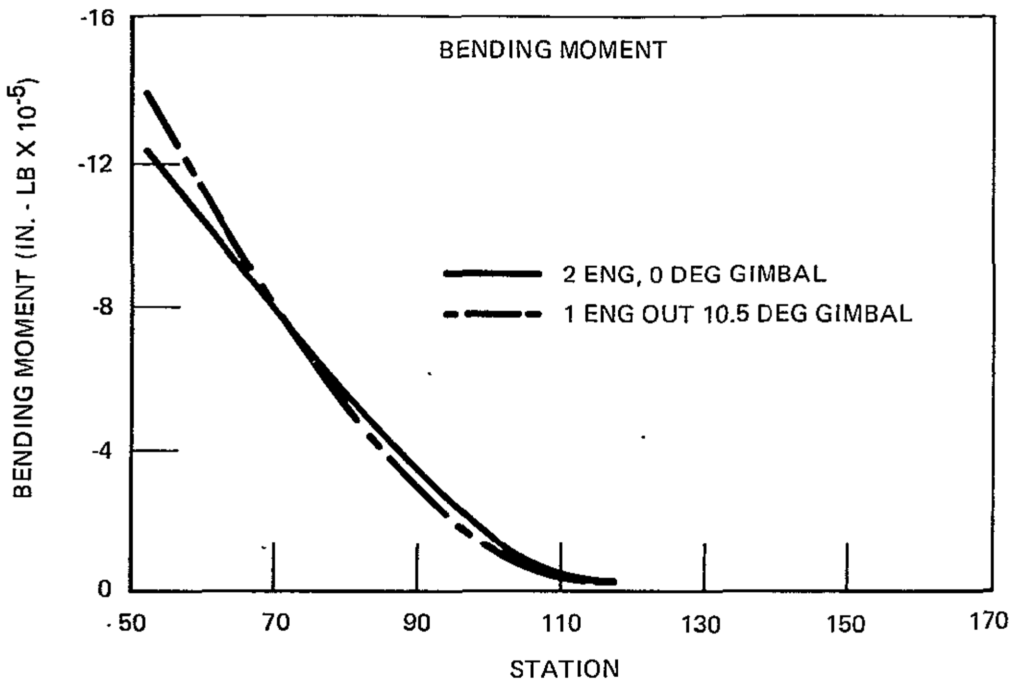


Figure 2-42. Frame X_B 105 Axial Load and Shear



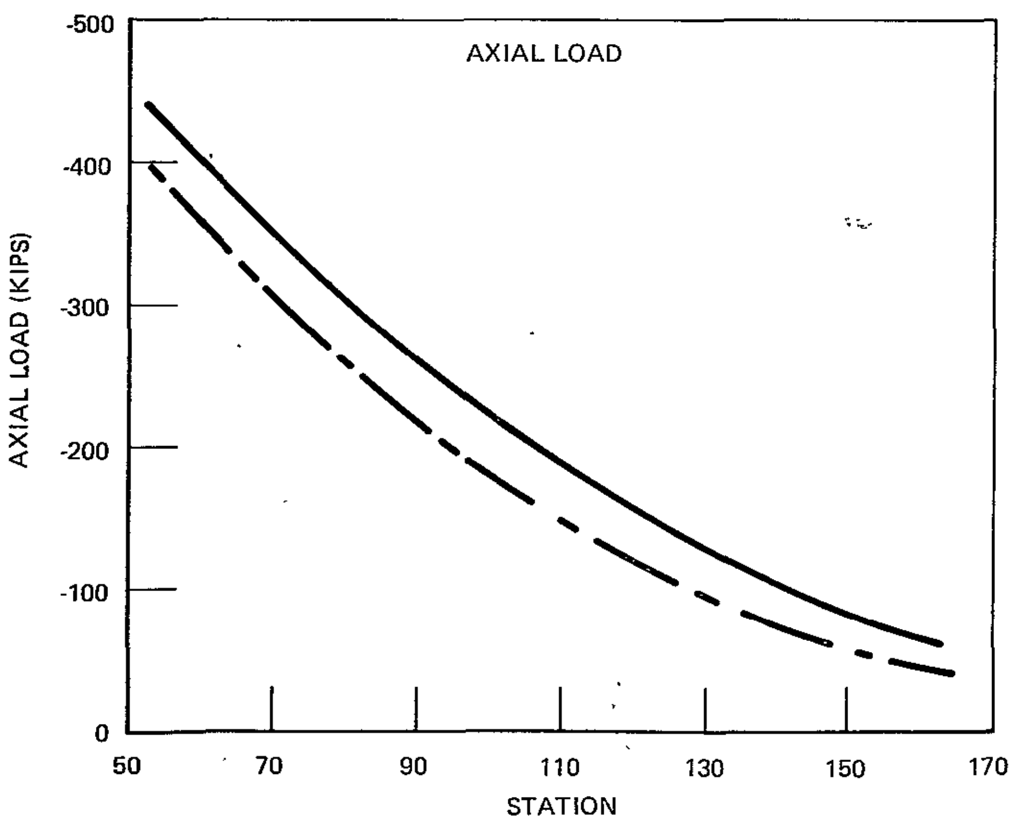
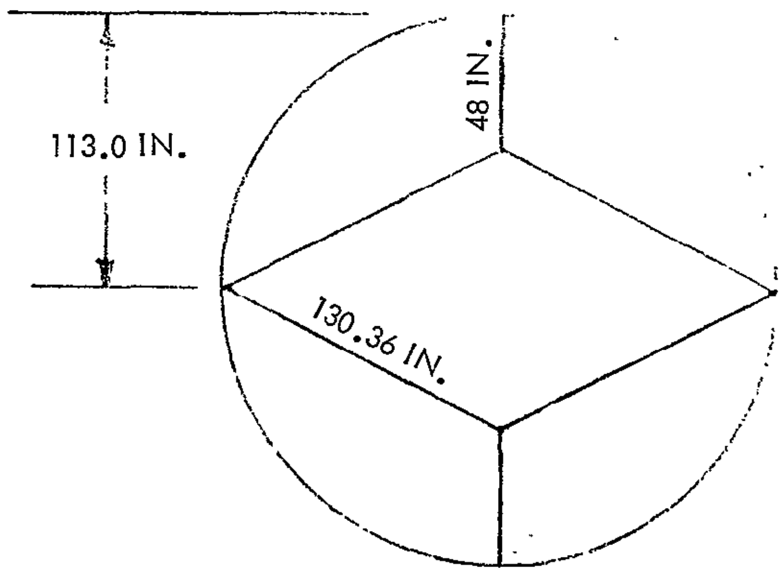


Figure 2-43. Thrust Cone Longeron

The critical design condition is "Two engines on 0° gimbal," (Load Condition I). Since the longeron will be machined from a forging, the allowable stress of 60,000 psi is used for sizing. Section depth for the frame can be optimized in subsequent design refinement.

Cross-tie Beam and Supports.



The general arrangement of the cross-tie beam and supports is indicated in the sketch above. This beam system provides the support and attachment for the APS LO_2 tank and stiffeners, the base frame and the aft end of the shell. The system is similar to the center engine support beam on the S-II-15 thrust structure, however, in this case the beam ends are fixed to reduce weight. Beam bending moments from the program solution are shown in Figure 2-44. Indicated on each moment curve is the corresponding axial load. The critical design condition for the system is hard-over gimbal (Condition II). Loads from the APS LO_2 tank were used in conjunction with these loads to establish the required sections. The allowable stresses used in the beam and support sizing are 60,000-psi tension or compression for the caps and 27,500-psi shear for the webs.

2.1.3 Thermal Protection System (TPS)

TPS Design Requirements

This insulation study effort was undertaken to establish a feasible insulation system, satisfying the requirements imposed by ESS missions.

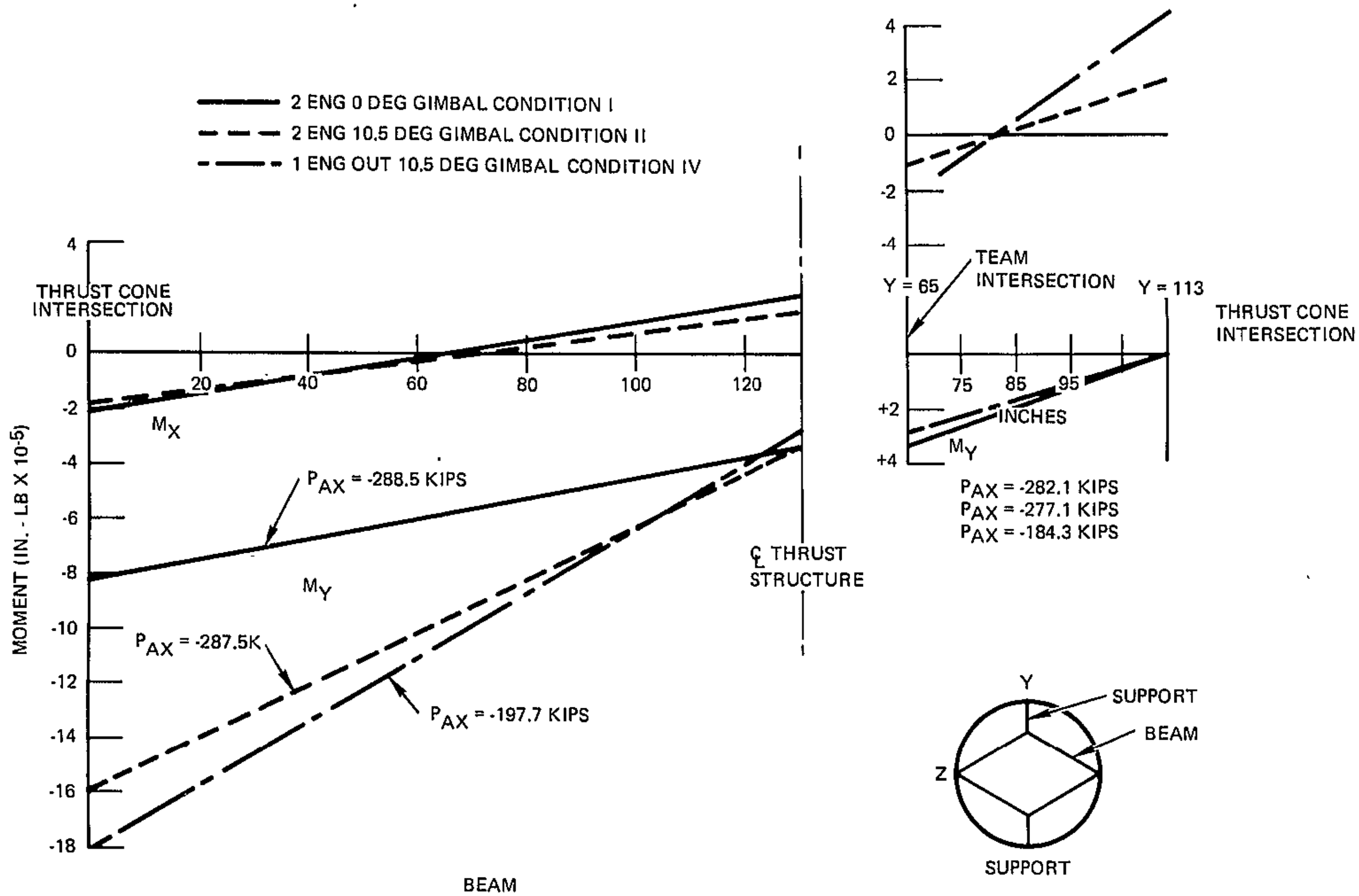


Figure 2-44. Cross-Tie Beam, X_B-52 and Support Bending Moment

These missions involve boosting space station, reusable nuclear shuttle and space tug payloads into low earth orbit. In this application, the requirements for the insulation system are:

1. Protection of propellant tanks from excessive aerodynamic heating, induced during first stage boost, to maintain proper propellant conditioning for engine start.

2. Protection of engine compartment systems and structure from excessive heating.

Four areas have been identified that require insulation and thermal protection to satisfy the above requirements:

1. LH_2 tank sidewall

2. Forward skirt

3. Aft skirt

4. Engine compartment

General Description

For all surfaces of the LH_2 sidewall and the lower part of the forward skirt, the thermal protection system consists of spray-on foam insulation covered with a polyimide, honeycomb erosion barrier (Figure 2-45). In local hot spot areas the erosion barrier is protected by an ablative material. Ablative material is also required in local hot spot regions of the forward and aft skirts.

The ESS base area is completely closed out with a heat shield to protect structure and equipment from the back flow of hot plume gases during main engine burn. A rigid heat shield covers all of the area except for a flexible curtain around each of the two engines to permit gimbaling. S-II base heat shield materials are used in both the rigid panels and flexible curtains and provide adequate temperature margins. Attachment of the curtain to the rigid heat shield includes provisions for release to permit engine removal while in orbit. Access for removal of electronic packages mounted on the thrust cone structure is accomplished through hinged rigid panels.

The heat shield is attached to the aft skirt frame at Station 37 and supported inboard by the thrust cone through stand-off attachments, (see Figure 2-46).

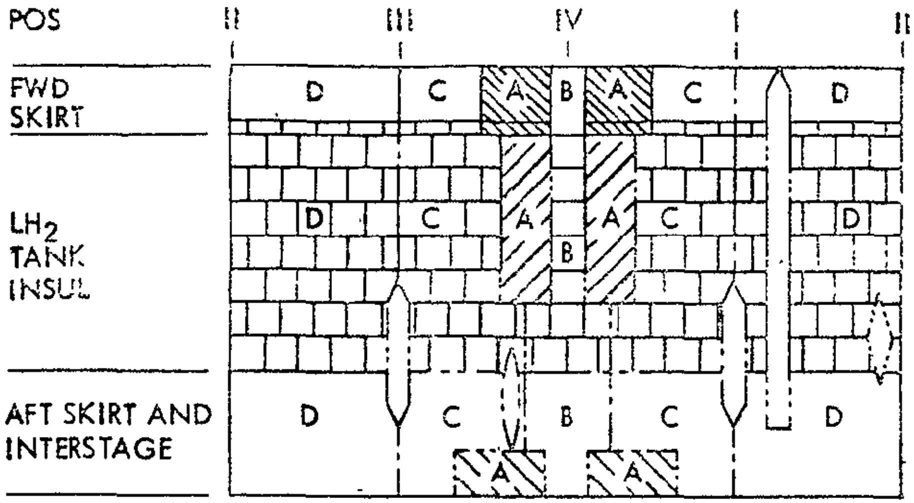
Thermal Environment

The thermal environment of the ESS consists of aerodynamic heating during booster ascent, orbital heating, ACPS engine plume impingement heating, and base region heating due to flow reversal resulting from the main propulsion system (MPS).

Aerodynamic Heating. Temperatures resulting from aerodynamic heating were calculated for the forward skirt, LH_2 tank sidewall insulation, and aft skirt/interstage areas in order to determine critical areas where additional thermal protection is required. The forward and aft skirt configurations analyzed were similar to the aluminum skin-stringer designs used on the S-II. The LH_2 tank sidewall insulation design differed from that of the S-II in that the spray-on foam insulation was protected by a polyimide honeycomb erosion barrier. Peak temperatures, shown in Figure 2-47, were calculated for three aerodynamic heating environments representing effects of RNS, MDAC space station, and space tug payloads. The aerodynamic environments are shown in Volume II, Book 1, Paragraph 5.2.2, and include basic heating plus booster interference heating, payload protuberance heating and local protuberance heating where applicable.

Figure 2-47 identifies the type of thermal protection required on various areas of the vehicle. The shaded areas will require an ablator and the slashed areas will need insulation to maintain allowable temperatures. Ablative material is recommended for use in higher aero heating protuberance regions in the vicinity of the booster-to-ESS attach fittings. A light weight insulation material will be used in other areas where thermal protection is required. Temperature histories for Region B of Figure 2-47 are shown in Figures 2-48 through 2-50 for RNS, MDAC Space Station and Space Tug payloads, respectively.

ESS base region temperatures were calculated for the external and internal surfaces of the thrust structure, rigid base heat shield, flexible curtains, APS LO_2 tank heat shield and OMS engine supports. The temperatures were based on engine exhaust heat rates, shown in Volume II, Book 1, Paragraph 5.2.2, for an aft skirt which is retained for the entire ESS mission. Initial calculations for the external and internal thrust structure surfaces yielded peak temperatures of 550 and 650 F, respectively, as shown in Figure 2-47. Both of these peak temperatures exceed the allowable temperature of 350 F requiring that the entire base region be protected from heating by engine exhaust gases.



ESS FLAT PATTERN

BOOST HEATING PEAK TEMP		
FWD SKIRT	A	2400
	B	1030
	C	610
LH_2 TANK INSUL	A	2220
	B	1135
	C	950
	D	830
AFT SKIRT & INTER-STAGE	A	2050
	B	680
	C	580
	D	510

Figure 2-47. Skin Temperature Due to Aerodynamic Heating During Boost

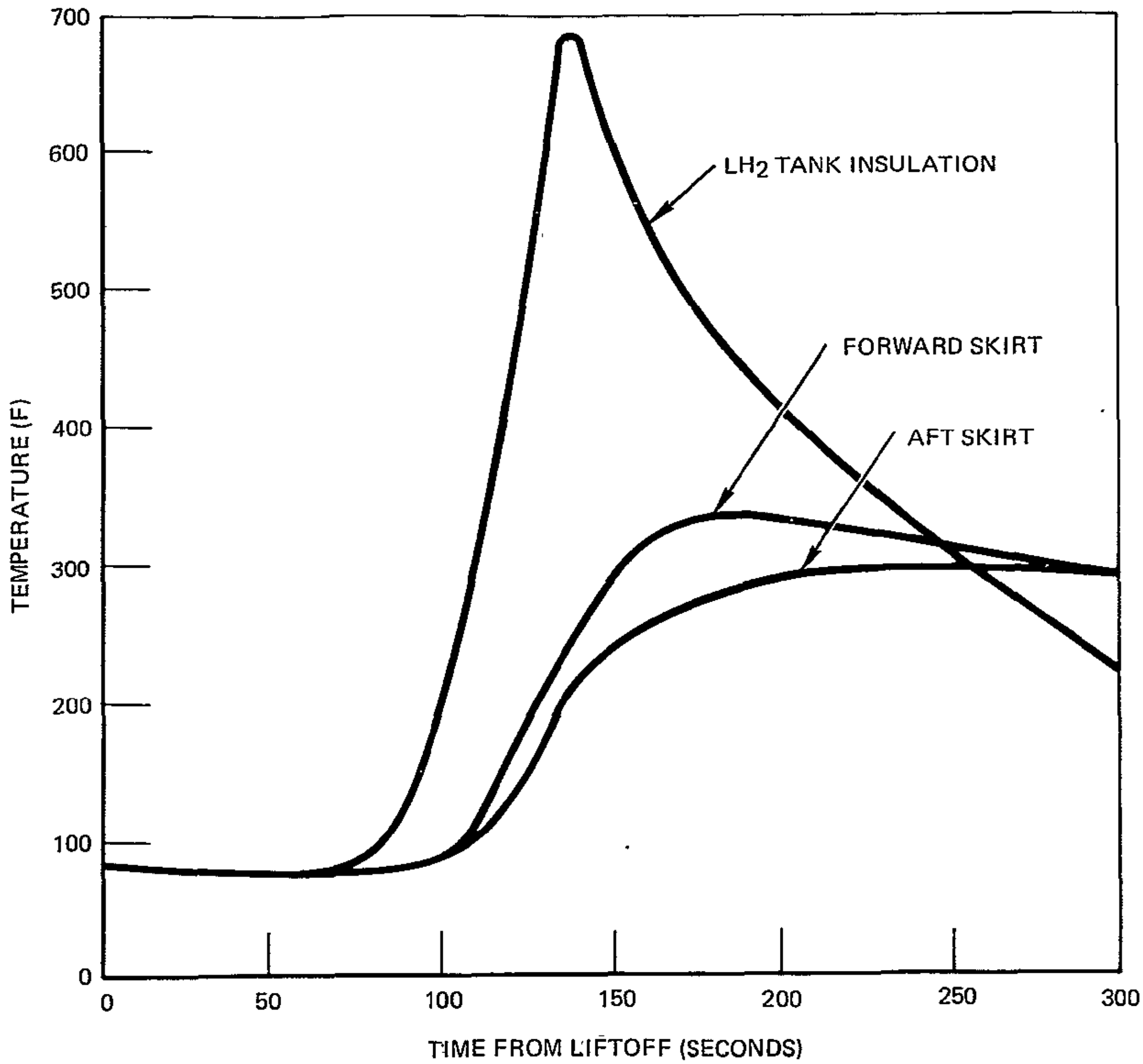


Figure 2-48. ESS Temperature Histories With RNS Payload for Region B of Figure 2-47

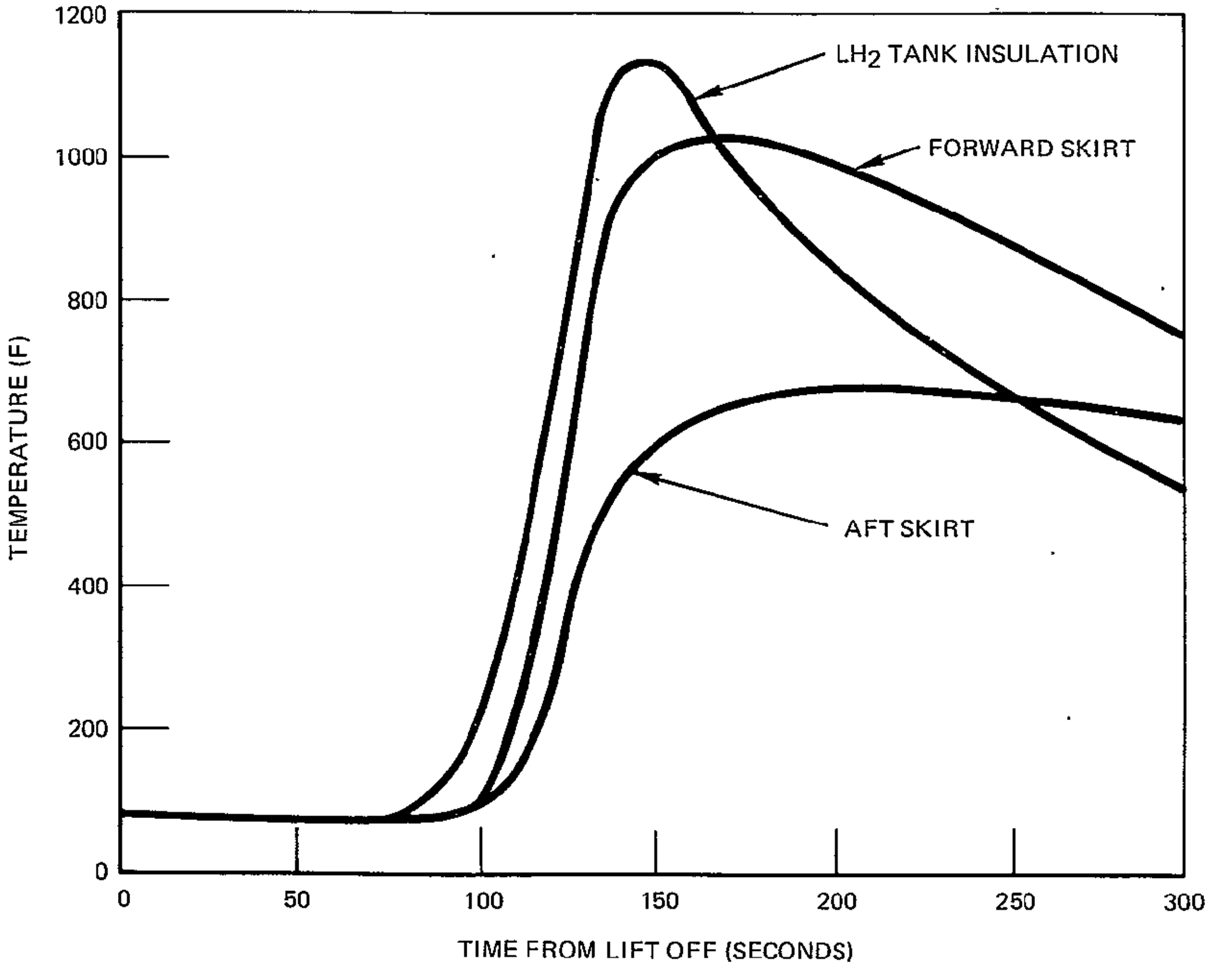


Figure 2-49. ESS Temperature Histories With MDAC Space Station Payload for Region B of Figure 2-47

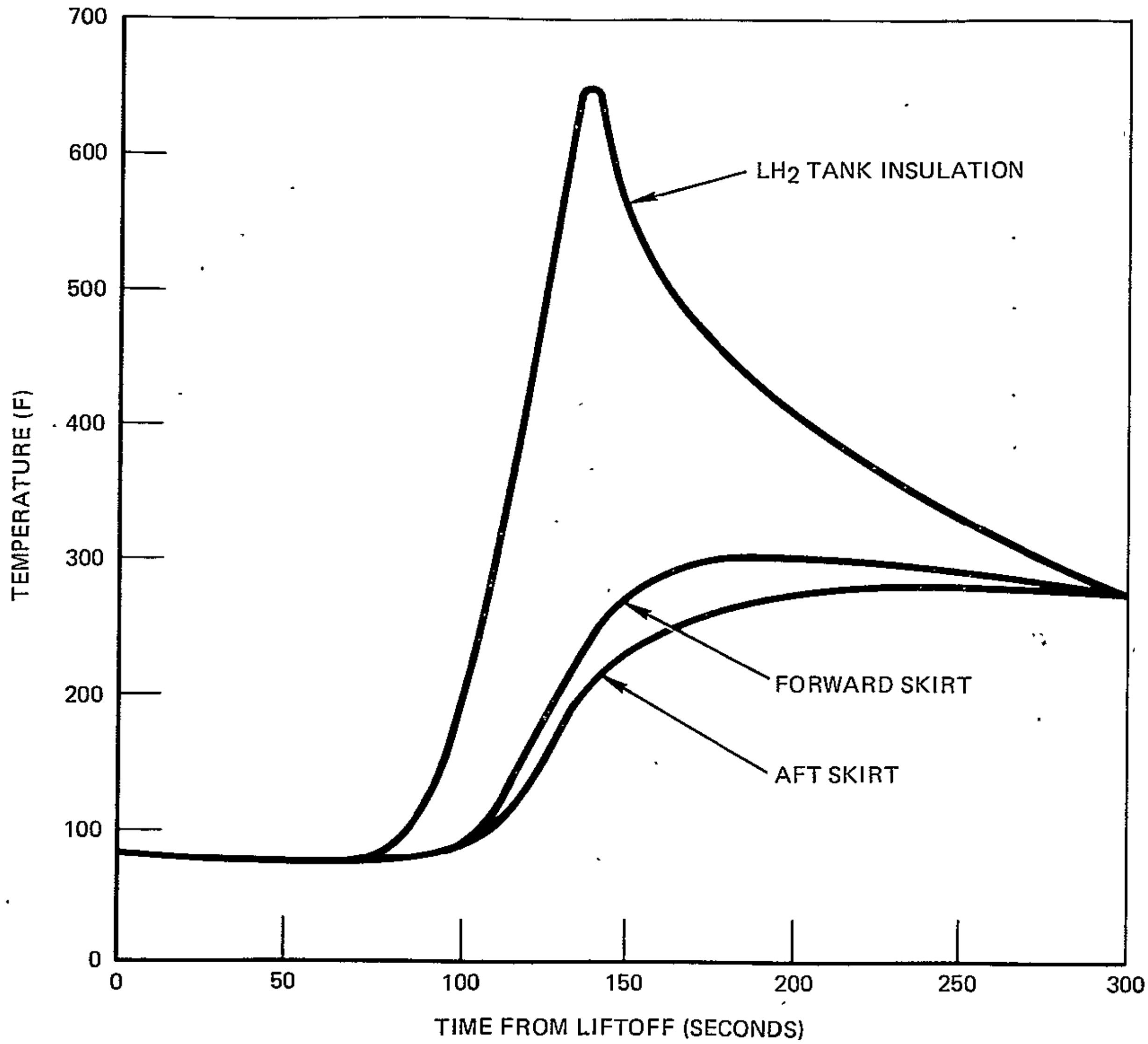


Figure 2-50. ESS Temperature Histories With Space Tug Payload for Region B of Figure 2-47

Base Heating. A rigid heat shield, similar to the S-II rigid heat shield, was designed to close out the entire base area at the aft end of the interstage as shown in Figure 2-47. A heat shield of the same design is used to protect the APS LO_2 tank which extends below the thrust structure. Flexible curtains, of the design used on the S-II, close out the areas adjacent to the orbiter engines. Figure 2-51 presents peak temperatures predicted for the base region utilizing the proposed thermal protection system. The temperatures for the external and internal thrust structure surfaces will be reduced to well below their allowable temperatures. The peak temperature of 1370 F predicted for the flexible curtain is well below the S-II flexible curtain allowable temperature of 1650 F. The rigid heat shield and APS LO_2 tank heat shield predicted peak temperatures are 790 F and 850 F, respectively. These are also well below the allowable temperature of 1550 F for the S-II heat shield. The predicted peak temperature for the OMS engines supports was 1230 F, which is above the allowable temperature of 900 F for stainless steel and therefore a protective shroud will be required around the supports.

ACPS Plume Impingement Temperatures. A peak temperature of 2300 F was calculated for the ESS aft skirt when using the ACPS plume impingement heat rates presented in Volume II, Book 1, Paragraph 5.2.2. In order to reduce the predicted structural temperatures to the allowable temperature of 250 F, a thermal protection system will be required on the aft skirt and air stream deflector and on the tank wall under the forward facing jet. The optimization of the thermal protection system has not been included and will depend on duration and number of times of engine firing as to the type of system such as insulation, ablators or deflectors.

Design Loads and Criteria

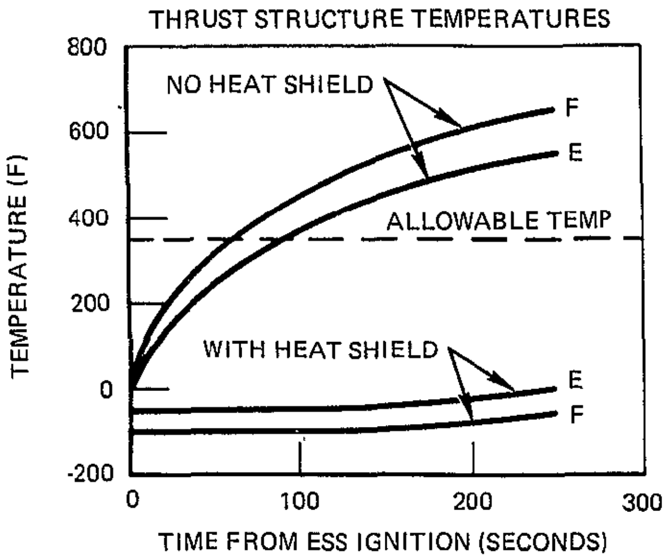
The following loads and criteria were used as a basis for development of the thermal protection system.

1. Ground-hold boil-off of LH_2 shall be limited to 6 percent per hour.

2. Boost heat leak shall be limited to 209,000 Btu per mission.

3. Sidewall insulation system shall be capable of resisting associated acoustic loadings in localized areas.

4. The erosion barrier shall be designed to permit expansions and contracting of the LH_2 tank defined in Table 2-2.



LOCATION AND DESCRIPTION	PEAK TEMPS - F
A-FLEXIBLE CURTAIN	1370
B-RIGID HEAT SHIELD	515 - 790
C-APS LOX TANK H/S	850
D-OMS ENGINE SUPPORTS	1230
E-THRUST STRUCTURE EXTERNAL	LESS THAN 0
F-THRUST STRUCTURE INTERNAL	LESS THAN -50

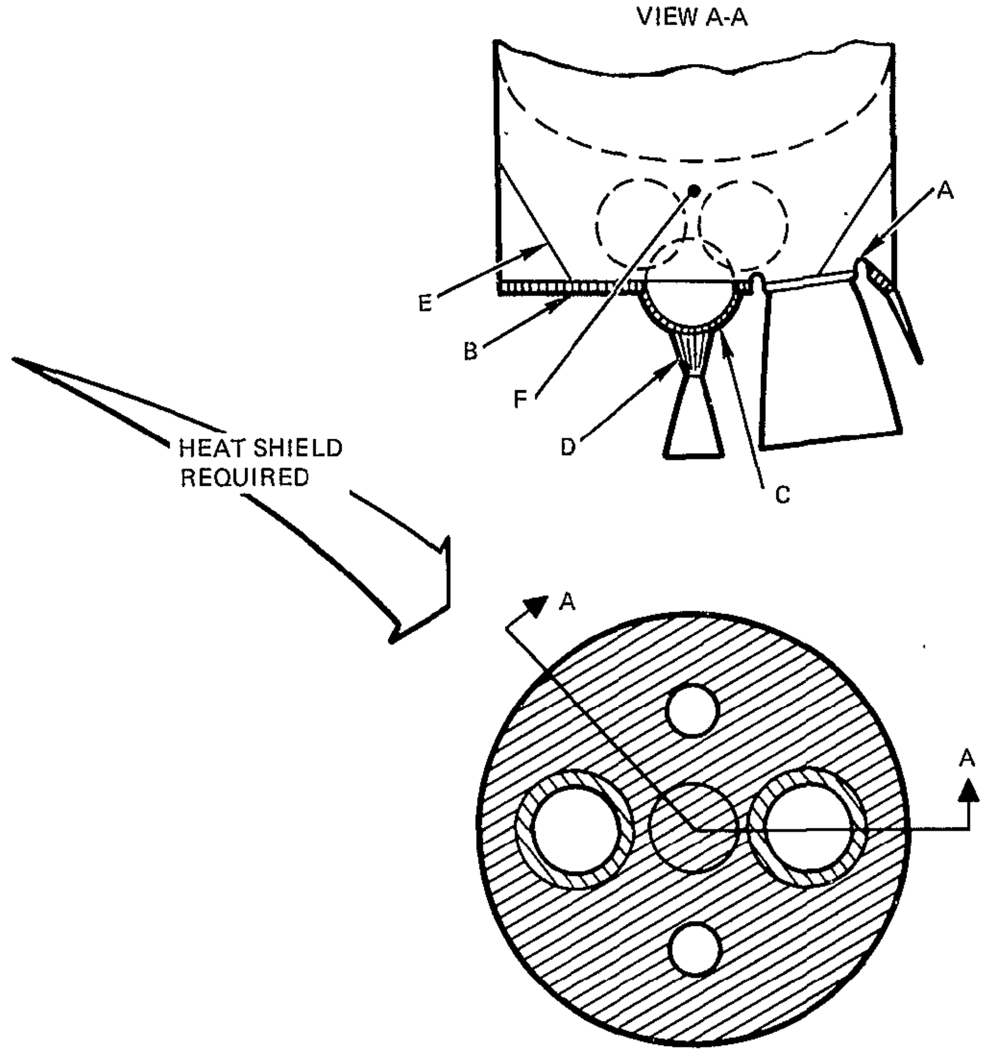


Figure 2-51. ESS Base Heating Temperatures

Specific requirements for the base region are as follows:

1. Closeout skirt base area from high-temperature exhaust gases to maintain structure below 350 F.

2. Provide for maximum engine gimbal of ±10.5 degrees in the pitch and yaw planes with the nozzle retracted or extended.

3. Accommodate relative structural and thermal deflections between thrust cone attachment and skirt frame.

4. Withstand a pressure gradient of 0.5 psi acting aft during launch along with an acoustic environment of approximately 150 db.

5. Perform heat shield functions during main engine burn with a pressure gradient of 0.1 psi acting forward.

6. Provide for main engine removal while in orbit.

7. Provide access for electronic package removal while in orbit.

Functional Interface and Support Requirements

No TPS functional or support requirements have been identified for the ESS vehicle.

Maintenance

Special maintenance is not needed to support the ESS vehicle because of the simple installation requirements. Deficiencies in the spray foam insulation can be corrected by removing the existing foam and respraying with new foam in the affected area. The polyimide sandwich erosion barrier panels are relatively inexpensive and easily replaced when defects appear.

Thermal Structural Concepts Trade

Thermal analysis methods are described in Volume II, Book 1, and in Volume XII. Thermal environment on the ESS in general is similar to conditions investigated in past studies. (Refer to SD 70-684, Saturn INT-21 Launch Vehicle, Task 10). ESS/Reusable Booster interference and local heating effects have been the area of primary investigation. Material selection for thermal protection was based on trade studies performed on previous S-II programs and on S-II experience. Selections based on these previous programs include.

Table 2-2. ESS LH_2 Tank Limit Deflections Summary

Condition	Pressure (psi)	Crit Sta	Temperature (°F)	N_x (limit)	Radial Deflection (in.)			Long Strain (in./in.)		
					Pressure $+N_x$	Temperature	Net	Pressure $+N_x$	Temperature	Net
Fueled	0	380 to 700	-423	0	0	-0.82	-0.82	0	-0.0041	-0.0041
Mated ascent	29.5	380	-423	-2800	+0.81	-0.82	-0.01	+0.00294	-0.0041	-0.00116
End ESS Boost	27.5	700	-220	--	+0.61	-0.64	-0.03	+0.00183	-0.0032	-0.00137

Maximums			
Contraction		Expansion	
Δ_r	ϵ_ϕ	Δ_r	ϵ_ϕ
-0.82 in.	-0.0041 in./in.	-0.01 in.	-0.0012 in./in.

1. NOPCO BX-250 spray foam insulation was selected by virtue of its proven performance on Saturn V lunar orbital missions.

2. Polyimide sandwich erosion barrier was selected for its top rating on a system evaluation trade. The table included factors such as performance, cost, weight, and ease of manufacture.

Design Analysis

Insulation. The basic sidewall and forward skirt insulation consists of spray-on foam, covered with an erosion barrier. From S-II programs and follow-on studies, it has been determined that NOPCO BX-250 spray foam will adequately function as a thermal protection system provided it is protected from aerodynamic erosion during atmospheric boost. For the ESS program, it is proposed that the shingled type ship-lapped aerodynamic erosion barrier be used to protect the fragile insulation systems (Figure 2-45).

The erosion barrier is composed of 60- by 66-inch overlapping panels of sandwich construction. The core of these panels is heat resistant phenolic honeycomb. Both top and bottom facing sheets are of polyimide composite; they are bonded to the core during the curing process by an adhesive contained within the sheets themselves.

Considerable analytical evaluation and studies have been performed on this configuration, all leading to a feasible design concept; however, precise acoustic, flutter and shock wave analysis are presently beyond the state-of-the-art and require development testing. It is anticiapted that there will be design modifications after wind tunnel and acoustic testing; however, the only foreseeable basic problems, however, are possible panel stiffening with the attendant weight increase. This is not considered to affect design feasibility.

The erosion barrier panels are attached to the sidewall with phenolic and fiberglass spacers, which are threaded into the sidewall bosses. The spacers are provided with threaded inserts, and the erosion barrier contains a hardspot at each point of attachment to the spacers. Attachment is accomplished using standard screws and washers.

Each erosion barrier panel has a rectangular, eight-hold bolt pattern (33 by 31 inches), which is located off center in order to rigidly fix one point of the panel. It is believed that the remaining three corners can obtain flutter resistance through the clamping action of the surrounding panels.

Because of severe thermal gradients established by the LH_2 tank loading operation and during the boost phase of the flight, the erosion barrier must be capable of accommodating expansion-contraction behavior of the tank sidewall. A system of slotted and oversize holes at the erosion barrier attach points will permit the panels to slide with respect to the sidewall and with respect to each other. Thus, strains due to expansion and contraction are relieved. In addition, a space must be left between adjacent panels to avoid expansion-contraction interference. These spaces are covered by the overlapping of the close-out edge of one panel onto a joggle in the adjacent panel (Figure 2-52).

Through aero-shear testing it has been determined that the maximum temperature to which the erosion barrier can be exposed is 900 F. On the ESS, however, the effects of multiple protuberances and the attachment fittings to the shuttle booster cause localized areas of excessive heating (hot spots) which are identified in Figure 2-47. After a study of hot spot problems, it was elected to cover these regions of the erosion barrier with a spray-on ablative coating. Follow-on analysis and testing will be needed to specify an ablative material; however, some leading candidates are:

Firex 10-035 filled modified epoxy

Firex 250 filled polyurethane

D.C. 93-027 silicone elastomer

Korotherm 790-005 silica filled isocyanate

Rigid Heat Shield. The rigid heat shield is fabricated of two 1-inch-thick phenolic impregnated glass honeycomb panels bonded together to form a 2-inch section. The aft half is filled with silica spheres for greater insulation. The forward half, at a lower temperature, provides the necessary strength. Two-ply glass fabric is used for the face and middle laminates. Inserts for structural attachments are potted in a thixotropic adhesive.

A major portion of the base area is made up of flat panels forming an in-station plane bulkhead at Station 37. A rigid hemispherical dome covers the protruding APS LO_2 tank located on the body centerline.

Thirty-degree truncated cones, extend forward from the flat panels along the engine null centerline to provide for the flexible curtain attachment.

Figure 2-53 defines the preliminary design concept for the ESS vehicle.

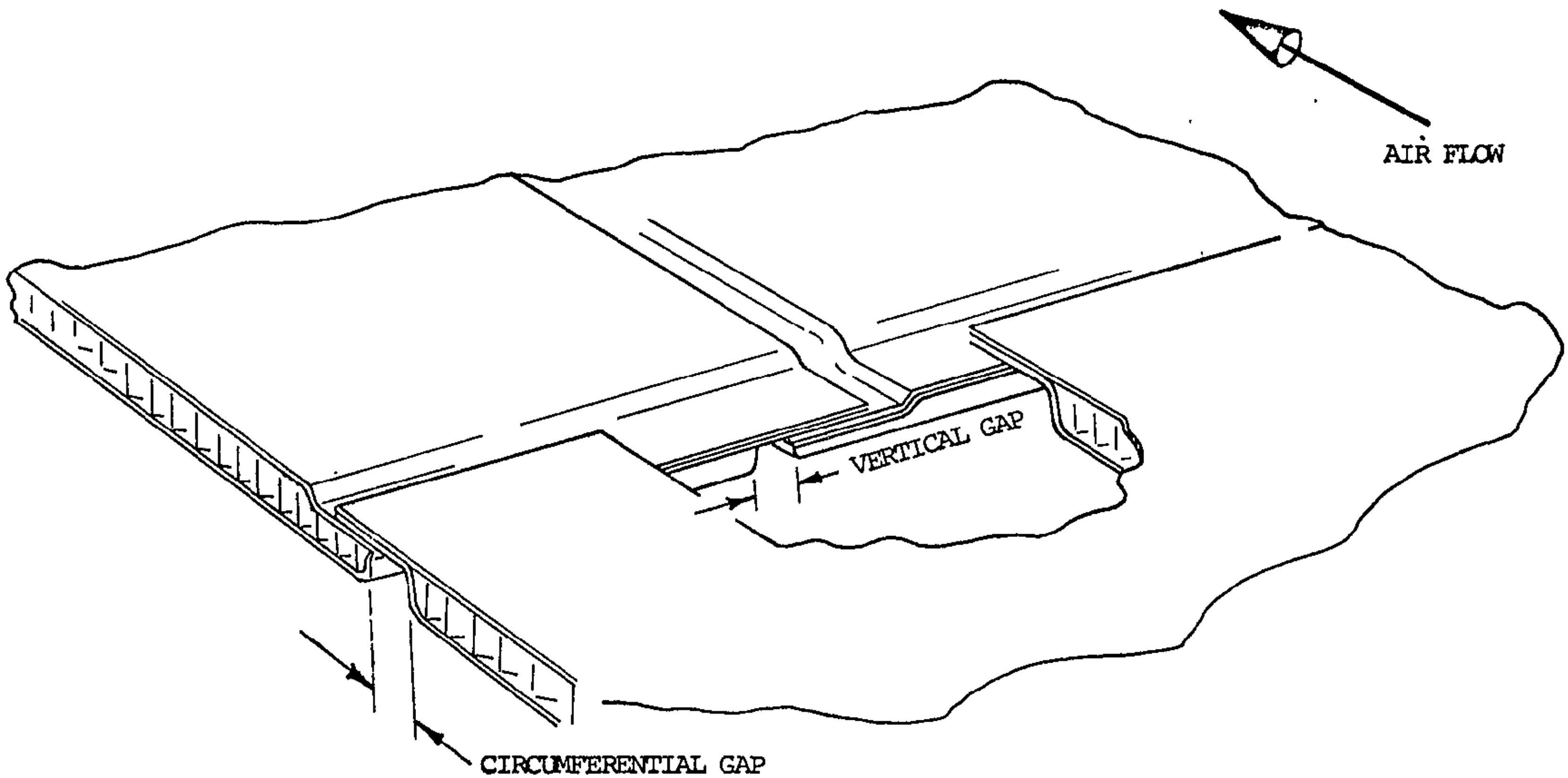


Figure 2-52. Typical Intersection of Panels

Flexible Curtain. The material and design of the flexible heat shield curtain is essentially the same as for S-II. S994 glass fabric facings enclose a 0.25-inch high silica batt. The curtain forms a gas and heat barrier between the engine heat shield and the rigid heat shield cone. The attachment to the engine may be either by a tension band or a clamping ring secured by bolts.

To provide for engine removal while in orbit, a spring-loaded ring clamps the curtain to the heat shield cone. A tension band preloads the ring to a configuration which will clamp and hold a glass rope welding fabricated in the edge of the curtain. Captive explosive bolts separate the tension band, allowing the ring to release the curtain. All other parts are held captive. Testing will be required to verify the curtain separation system.

The configuration in which the curtain drapes inward toward the engine gimbal point was selected because of limited radial clearance between the engine envelope and the structural frame. It is believed that adequate displacement can be attained; however, a simulated mock-up will be required to optimize and determine flexure limits.

The heat shield panels nestle inside the inboard cap of the aft skirt frame and butts against a flange, attached through a ring flush to the forward face of the frame. The thrust structure provides inboard panel support through short struts or formed sheet metal stand-offs. The relative deflection between thrust cone and aft skirt frame must be considered for these attachments.

The OMS engine thrust structure consists of a truss with three struts each which penetrates the base heat shield and attach to the thrust cone. Cutouts in the honeycomb heat shield are required for installation. Flexible curtain material will be used to close out the area around each strut. Some form of thermal protection may be required for the OMS engine thrust structure protruding aft of the heat shield.

2.2 PROPULSION SUBSYSTEM GROUP

The propulsion subsystems consist of the main propulsion subsystem and the auxiliary propulsion subsystem. The main propulsion subsystem provides the expendable second stage velocity increment necessary to place the stage in initial earth orbit. The auxiliary propulsion subsystem is utilized for changing orbits, deorbit, attitude control, and for rendezvous if it is performed.

2.2.1 Main Propulsion Subsystem

The main propulsion subsystem utilizes two space shuttle orbiter engines to provide the required impulse to place the ESS in initial orbit. The engines are assumed to be GFE and are in accordance with ICD 13M15000B except where noted. The main propulsion subsystem includes all assemblies required to support main engine burn. The subsystem does not include the propellant tanks, which are structural, or controls, instrumentation, etc., which are part of the avionics subsystem groups. The main propulsion subsystem is shown schematically in Figure 2-54.

Design Requirements

The design requirements of the main propulsion subsystem are set by a number of considerations including the ESS general arrangement, trajectory, stability and control, the study control document, and the engine interface requirements. The main propulsion subsystem must satisfy the engine interface requirements documented in "Space Shuttle Engine (SSE)/ Expendable Second Stage (ESS) Interface Requirements," Appendix B of Volume VI of this report. The main propulsion subsystem must fulfill the interface requirements for the time duration dictated by the mission trajectory.

The major design requirements for the main propulsion subsystem assemblies are as follows:

1. Space Shuttle Orbiter Engines. The SSEO's provide the velocity increment necessary to place the ESS in initial earth orbit. The engines must be installed so that the vehicle is controllable and the engines serviced per ICD 13M15000B.

2. Propellant Feed, Recirculation, Valve Actuation, Pogo Suppression, and Fill and Drain. Propellants must be applied to the engine at the required flowrates; adequate preconditioning must be performed to ensure properly subcooled propellants at engine start; the interaction between structure, feedline, and engine must not be of enough magnitude to cause hardware damage

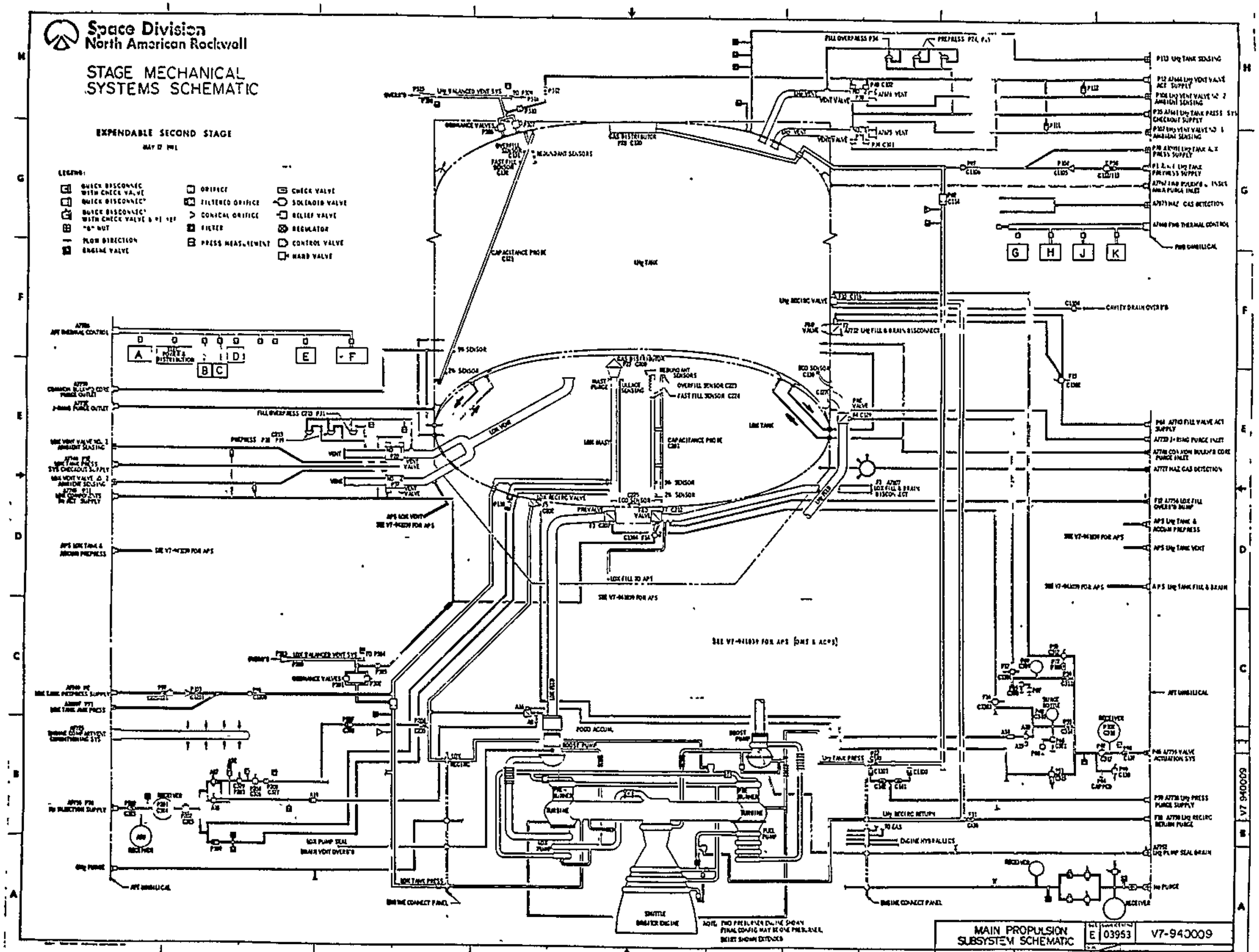


Figure 2-54. Main Propulsion Subsystem Schematic

or major performance degradation; the propellant fill of the ESS must be accomplished within the two-hour servicing time set by the space shuttle; and a rapid propellant drain is required in the event of an emergency condition.

3. Pressurization and Vent. The propellant tanks must be adequately pressurized on the ground to assure ESS structural integrity during first stage boost; proper tank pressure at engine start and throughout engine burn to assure adequate NPSH must be provided; and the tanks must be protected by adequate vent and relief capability.

4. Engine Servicing. Nitrogen purge gas and helium must be supplied to the engines at proper conditions and an adequate fuel vent must be provided.

5. Thrust Vector Control. The main engines must be gimballed as required by the stability and control analysis and hydraulic fluid must be supplied to the engine at proper conditions.

6. Safing. The main propellant tanks must be safed subsequent to main engine burn and residual propellants must be dumped without significant mission effect.

7. Engine Compartment Conditioning. The engine compartment must be conditioned and inerted on the launch pad.

In addition to the specific requirements for each assembly, a number of general requirements must be observed. These include:

1. The main propulsion subsystem must be capable of withstanding the predicted environment—acoustic, vibration, and thermal.

2. Standard operational requirements such as purging, leak checking, functional testing, and component replacement will be performed.

3. The subsystem will be designed to fail operational after the first failure of a component and to fail safe for booster crew survival after the second failure (FO/FS). This rule is applicable only when mated with the shuttle booster and is not applicable to structure or pressure vessels. Existing assemblies that are not FO/FS will be identified.

4. The subsystem will provide for safe mission termination.

5. The ESS will be controllable (continue to fly safely) with one main engine out.

These requirements are discussed in the description and operations section of this report when they have an effect on the ESS main propulsion subsystem.

Subsystem Description

The main propulsion subsystem is composed of the following significant assemblies:

1. Space shuttle orbiter engines
2. Propellant feed
3. Pressurization
4. Engine servicing
5. Thrust vector control
6. Safing
7. Engine compartment conditioning

Space Shuttle Orbiter Engines. The basic characteristics of the space shuttle orbiter engine (SSEO) are shown in Table 2-3. The installation of the engine is shown in Figure 2-55, the ESS general arrangement drawing.

The engines are mounted on the Position I-III line on a 114.8-inch radius. The mounting on the position line minimizes ESS plume impingement on the shuttle booster. The 114.8-inch radius permits the ESS to utilize the same thrust cone as the chemical interorbital shuttle (CIS). The CIS has provisions for a seven-foot diameter aft docking adapter for docking with the space tug. Therefore, to utilize the same thrust cone the engines must be located on a larger radius than they would be if only the ESS needs were considered.

To provide for engine-out control capability, the engines are canted at a 12-degree angle to permit the thrust vector to pass as nearly as possible through the center of gravity. In addition to the 12-degree cant, an additional 1-percent precant is provided for thrust structure compliance. To minimize aero loads on the engine during first-stage boost, an airstream deflector is provided and the engines are toed into a 3.46-degree angle.

Table 2-3. Space Shuttle Orbiter Engine Characteristics

Parameter	Nozzle Extended	Nozzle Retracted
Overall length	277.0 +0/-0.5 in.	165.5 ± 0.250 in.
Exit diameter	147.0 ± 0.125 in.	147.0 ± 0.125 in.
Propellant inlet diameter	15 in. max	15 in. max
Gimbal capability	±7 deg square	±7 deg square
Dry weight (excluding insulation)	8810 lb	8810 lb
Burnout weight (excluding insulation)	9530 lb	9530 lb
Mixture ratio capability	5.5 to 6.5 F	5.5 to 6.5 F
Thrust range capability	50 to 100 percent	50 to 100 percent
Emergency power level	109 percent at 6.0 EMR	109 percent at 6.0 EMR
Nominal performance at 6.0 EMR		
a. Vacuum thrust	632,000 lb	619,000 lb
b. Vacuum I_{sp}	459 sec	447 sec
c. Sea level thrust	N/A	488,000 lb
d. Sea level I_{sp}	N/A	352 sec

Engine-out capability also dictates a gimbal capability of greater than 7 degrees for yaw and pitch. This has been provided by orienting the engine so that the 10-degree components of the ± 7-degree square pattern are in pitch and yaw.

Propellant Feed. A number of subassemblies are included in the propellant feed assembly. In addition to the basic propellant feed these include recirculation, valve actuation, pogo suppression, and fill and drain.

The propellant feed ducts are shown in Figure 2-56. The ESS will utilize 13-inch vacuum jacketed ducts for both the LO_2 and LH_2 feed. With the SSEO, the existing (S-II) LH_2 tank outlet elbows must be replaced to eliminate local cavitation in the elbow. Cavitation would occur in the existing elbow because of the high-velocity pressure resulting from the small size of the elbow (9.58 in.). The high-velocity pressure would drop the static pressure in the elbow below the vapor pressure and cavitation would result.

As a result of experience gained on the S-II program, a 90-degree tank outlet elbow configuration has been selected for ease of manufacture. A new sump is required on the LO_2 tank to allow for mating with a 13-inch prevalve. The space shuttle prevalves will be utilized for ESS prevalves. The current planning is to design the shuttle prevalve so that it may be actuated either electrically or pneumatically. The ESS will actuate the valve pneumatically. A straight run of as much as 56 inches (the exact length dependent upon the engine contractor) is required upstream of the engine inlet. With the selected ESS configuration, a straight run of approximately 80 inches results. Four primary propellant depletion sensors will be provided in each propellant tank and a two-of-four voting logic provided as a backup to the primary velocity cutoff planned for the ESS. In the event of an engine-out, different sensors are used for depletion because of the ESS tilt angle which results from engine-out. The reason for this procedure may be seen in Figure 2-57. By providing different cutoff sensors for the engine-out case, the payload loss from increased LO_2 residual with an engine out is limited to 1140 pounds.

Prestart propellant conditioning is accomplished by recirculating propellants through the feed ducts and engines and then back to the propellant tanks. The recirculation subassemblies are shown on the engine servicing design layout (Figure 2-58). Liquid hydrogen recirculation is accomplished by electrically motoring the engine LH_2 boost pump prior to engine start. This procedure provides the delta pressure necessary for recirculation flow through the engine and back to the LH_2 tank via a manifolded vacuum-jacketed return line. A purge line is provided to the return line for system purging. Liquid oxygen recirculation flow is accomplished with a natural convection system augmented with helium injection. The LO_2 is returned to

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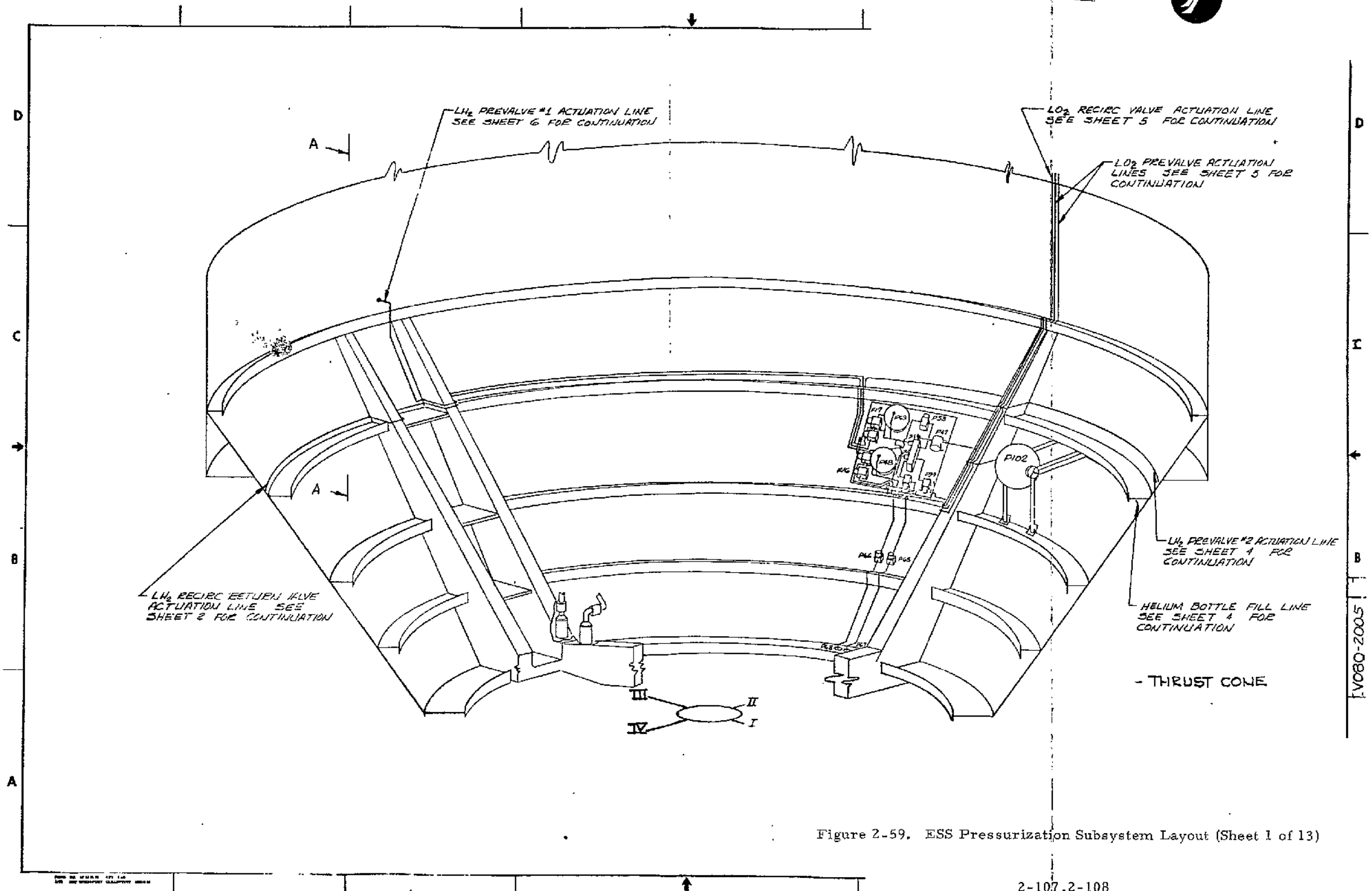

Figure 2-59. ESS Pressurization Subsystem Layout (Sheet 1 of 13)

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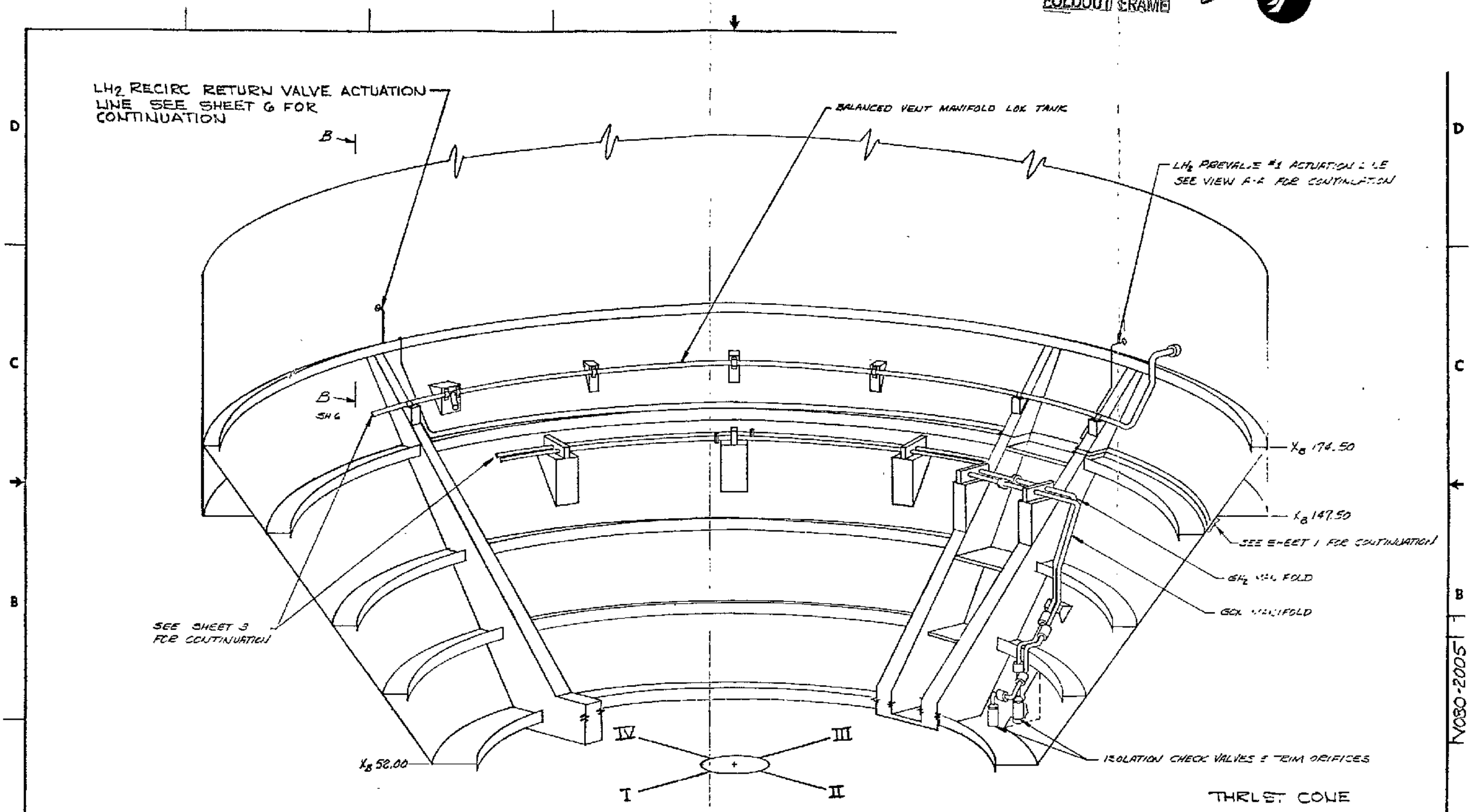

Figure 2-59. ESS Pressurization Subsystem Layout (Sheet 2 of 13)

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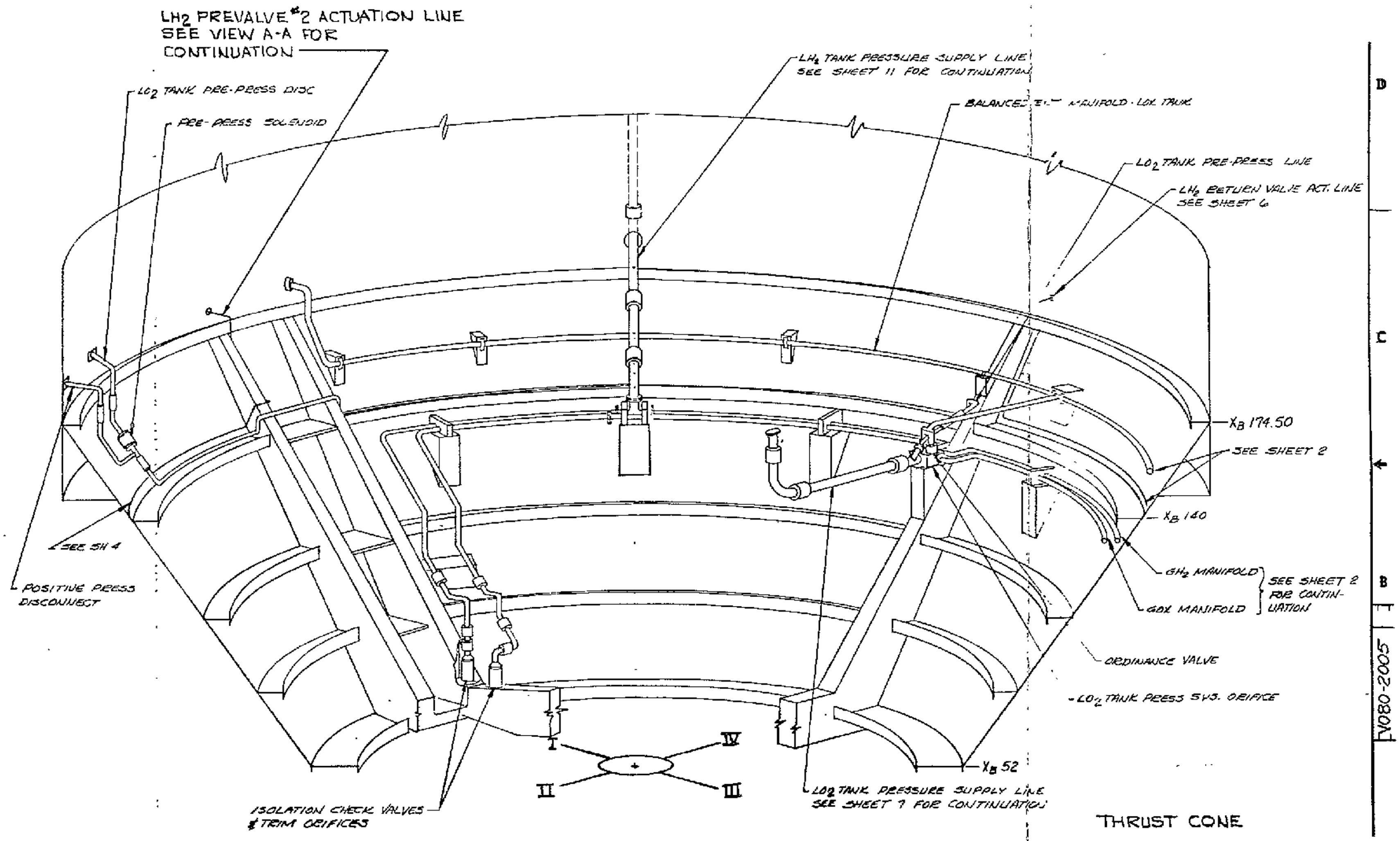

Figure 2-59. ESS Pressurization Subsystem Layout (Sheet 3 of 13)

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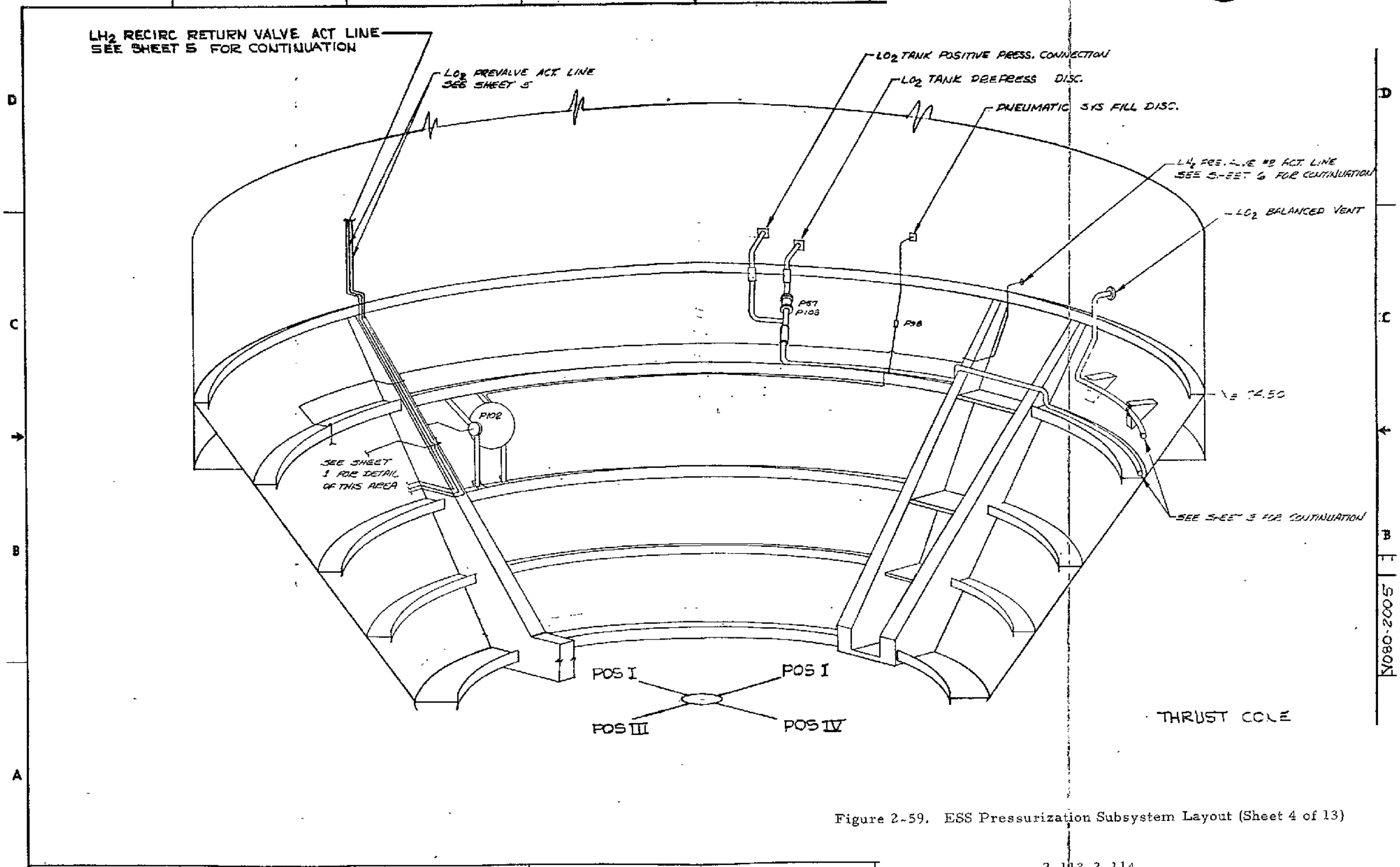

Figure 2-59. ESS Pressurization Subsystem Layout (Sheet 4 of 13)

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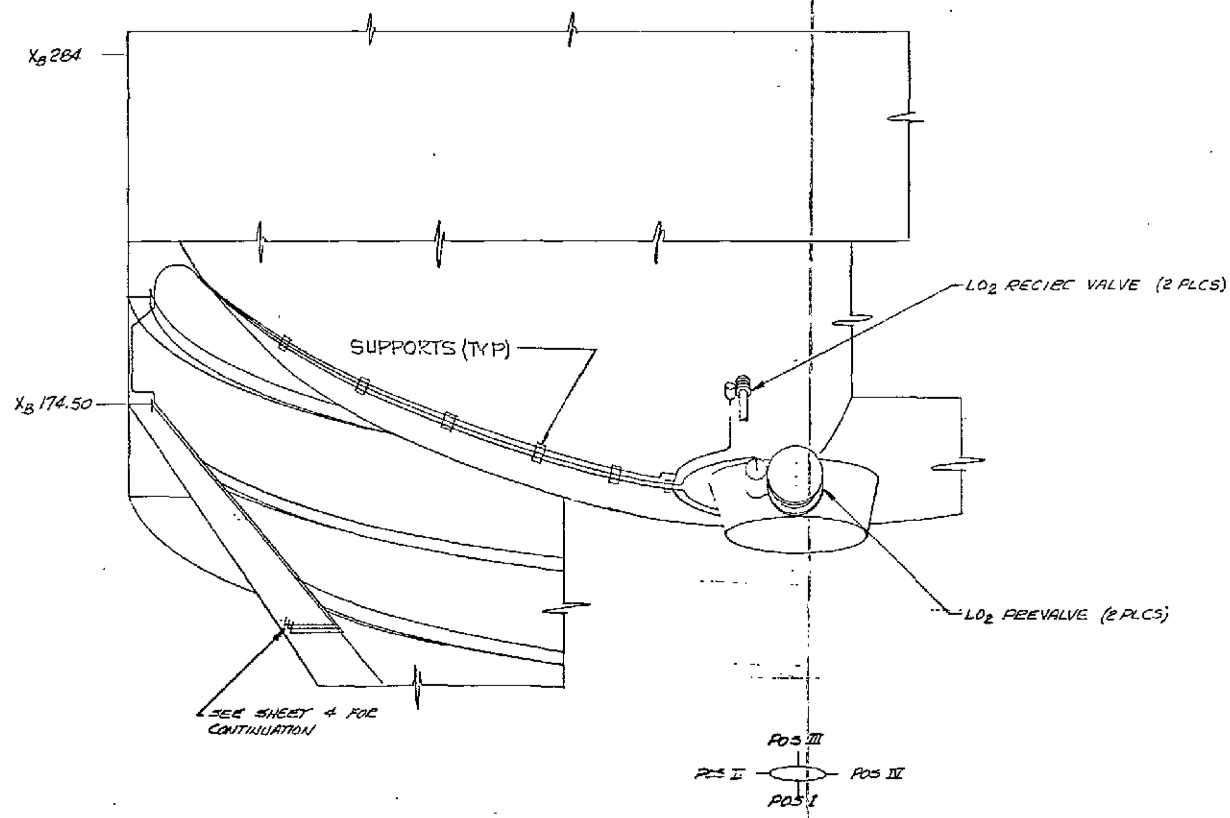

Figure 2-59. ESS Pressurization Subsystem Layout (Sheet 5 of 13)

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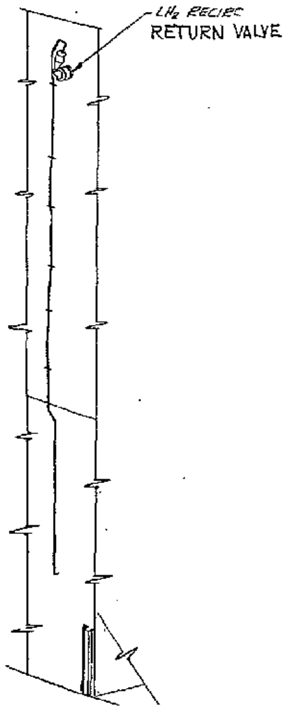

VIEW B-B
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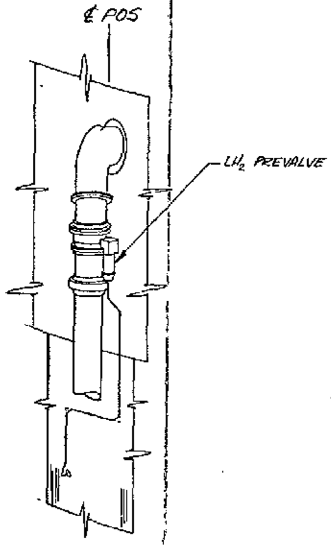

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Figure 2-59. ESS Pressurization Subsystem Layout (Sheet 6 of 13)

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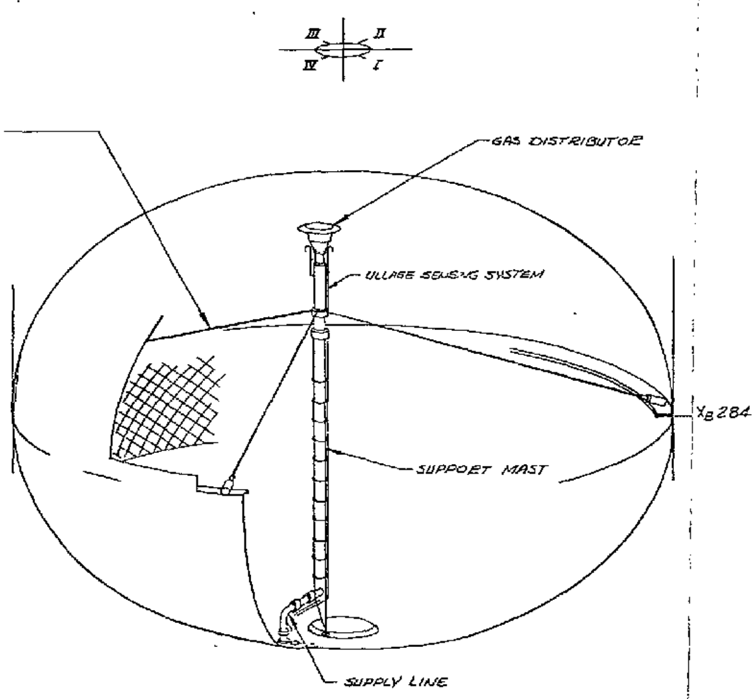

Figure 2-59. ESS Pressurization Subsystem Layout (Sheet 7 of 13)

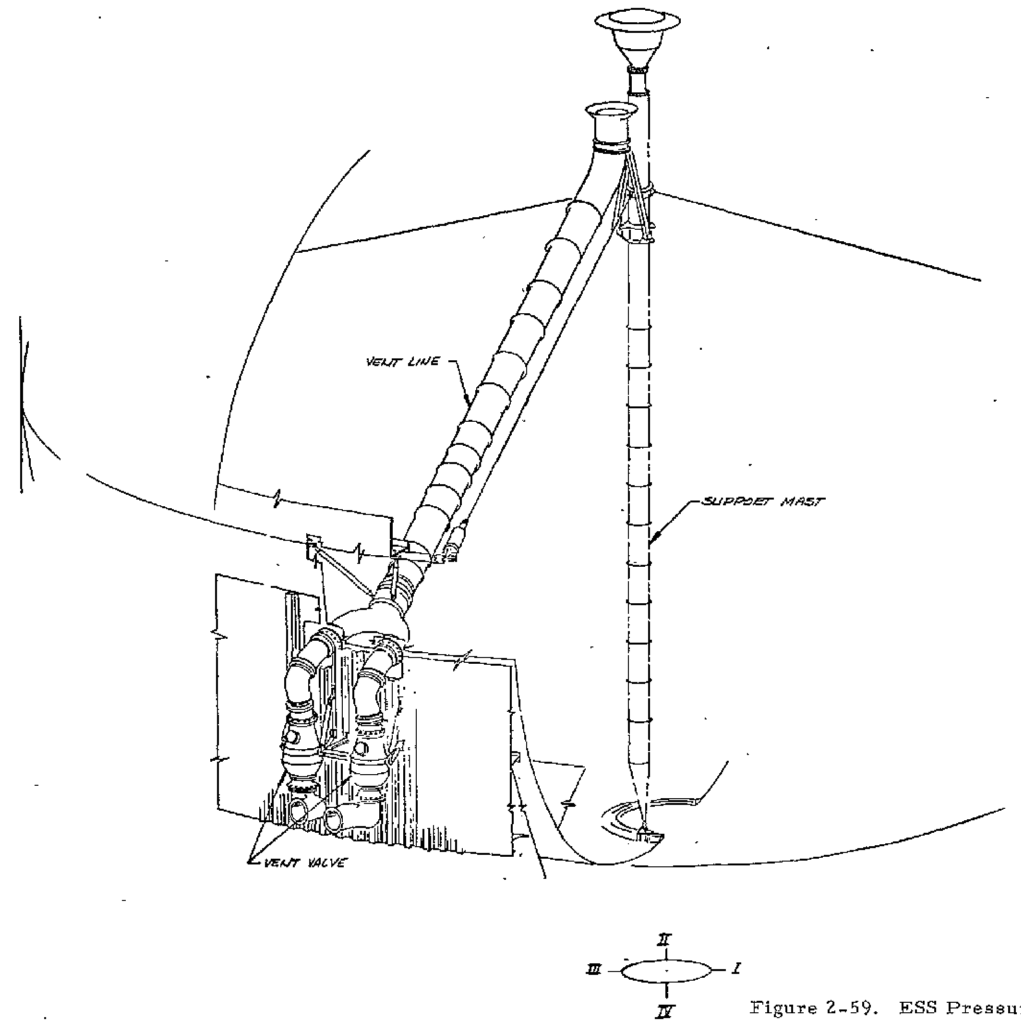

Figure 2-59. ESS Pressurization Subsystem Layout (Sheet 8 of 13)

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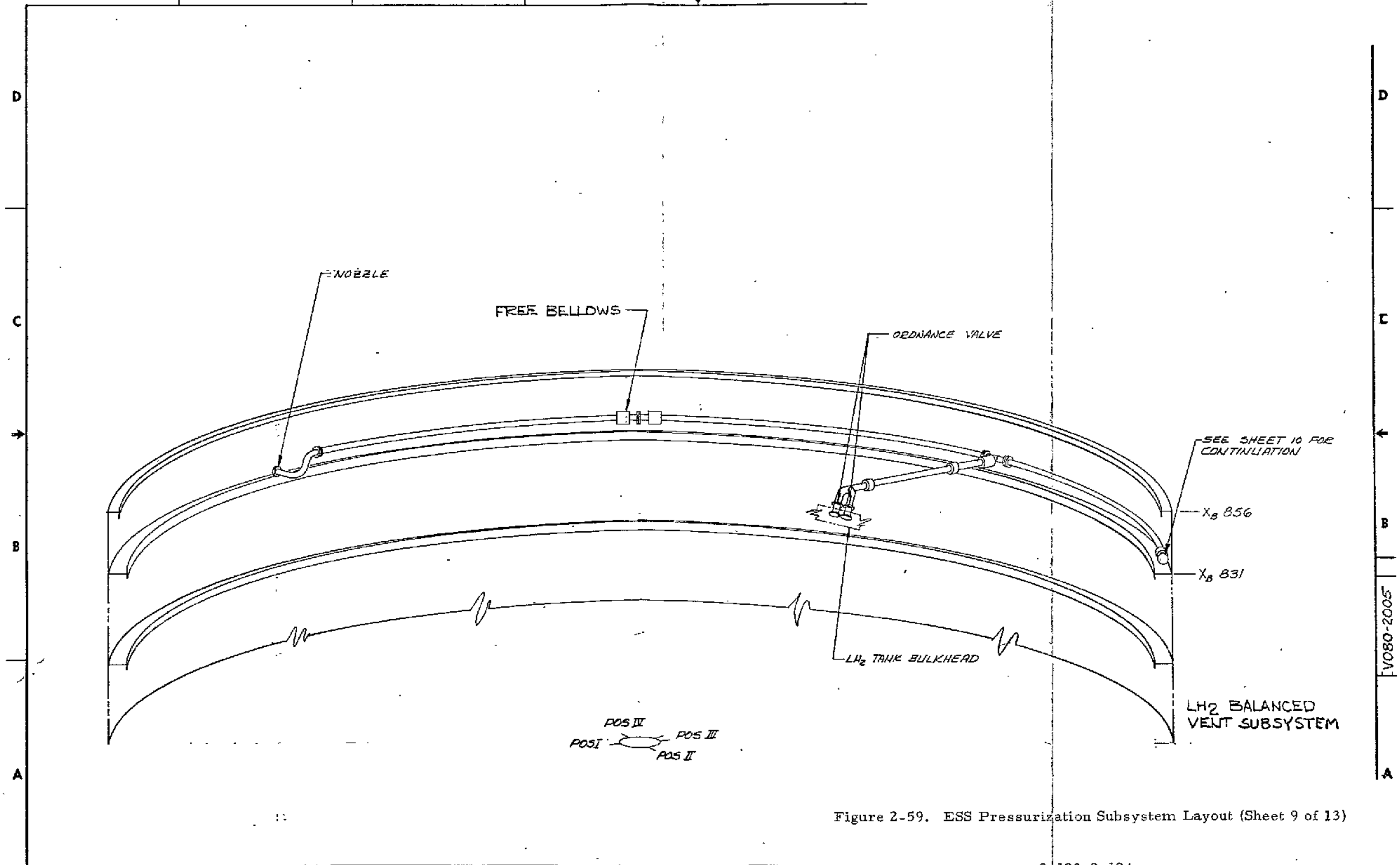

Figure 2-59. ESS Pressurization Subsystem Layout (Sheet 9 of 13)

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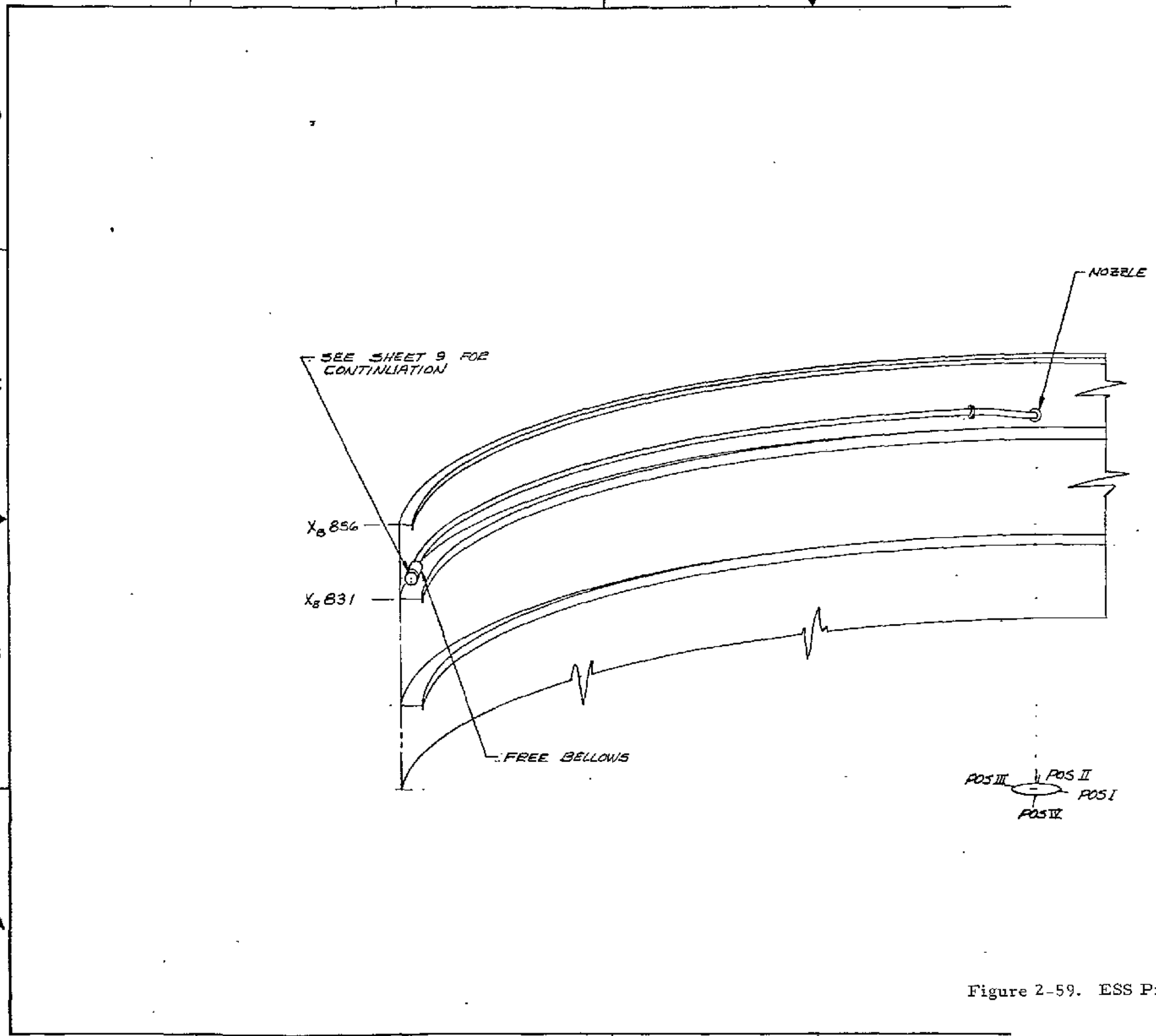

LH2 BALANCED VENT SUBSYSTEM

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Figure 2-59. ESS Pressurization Subsystem Layout (Sheet 10 of 13)

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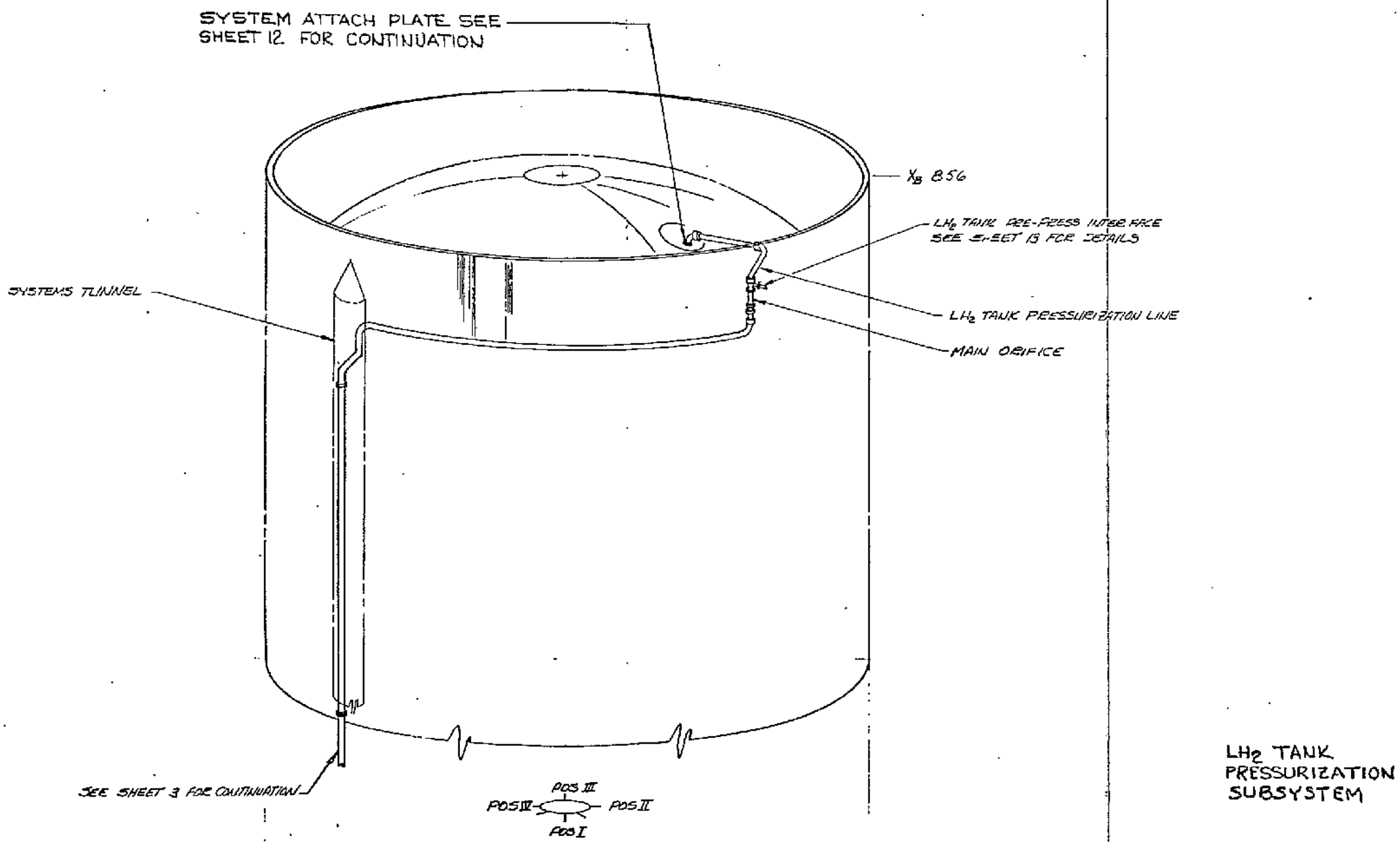

Figure 2-59. ESS Pressurization Subsystem Layout (Sheet 11 of 13)

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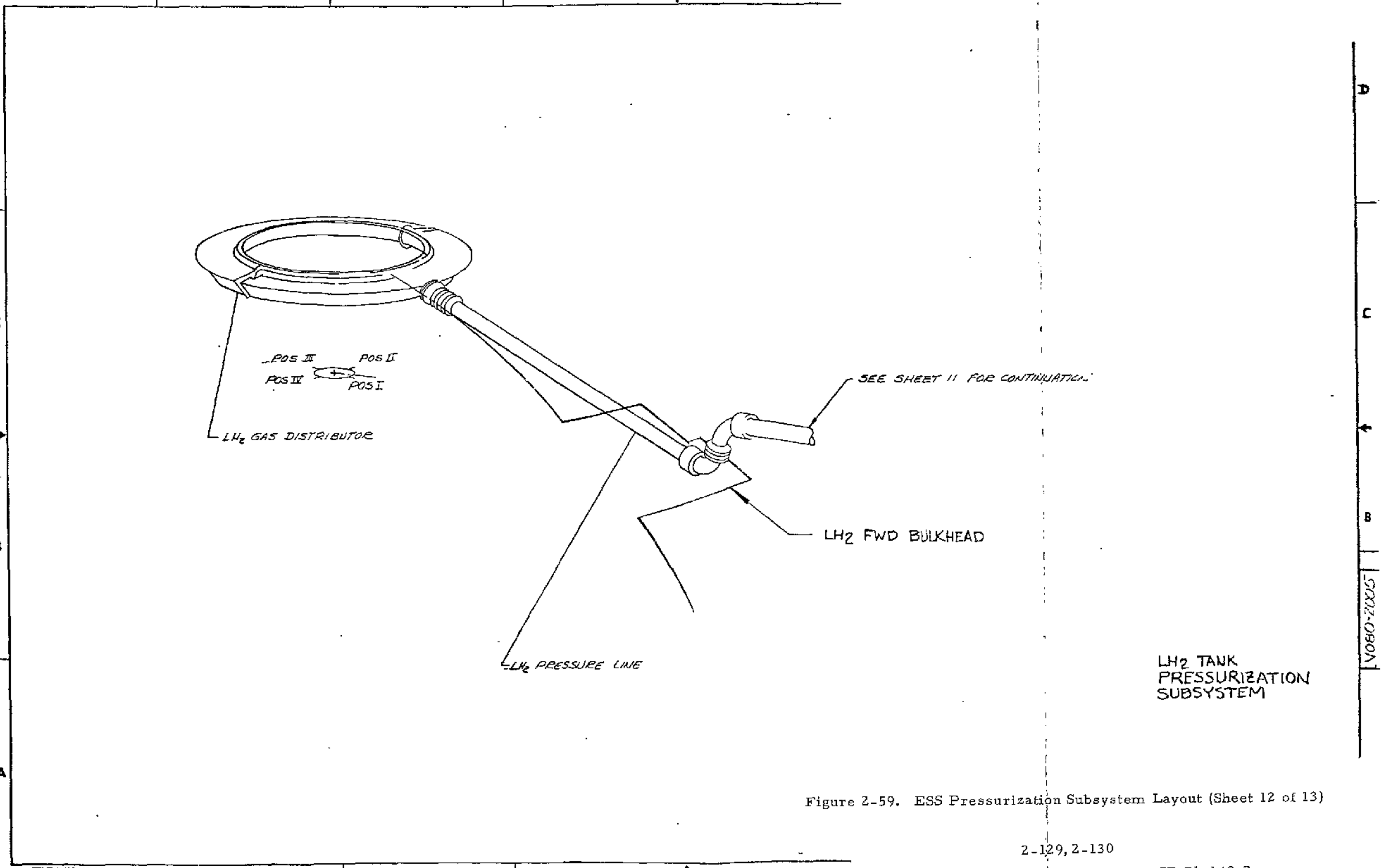


Figure 2-59. ESS Pressurization Subsystem Layout (Sheet 12 of 13)

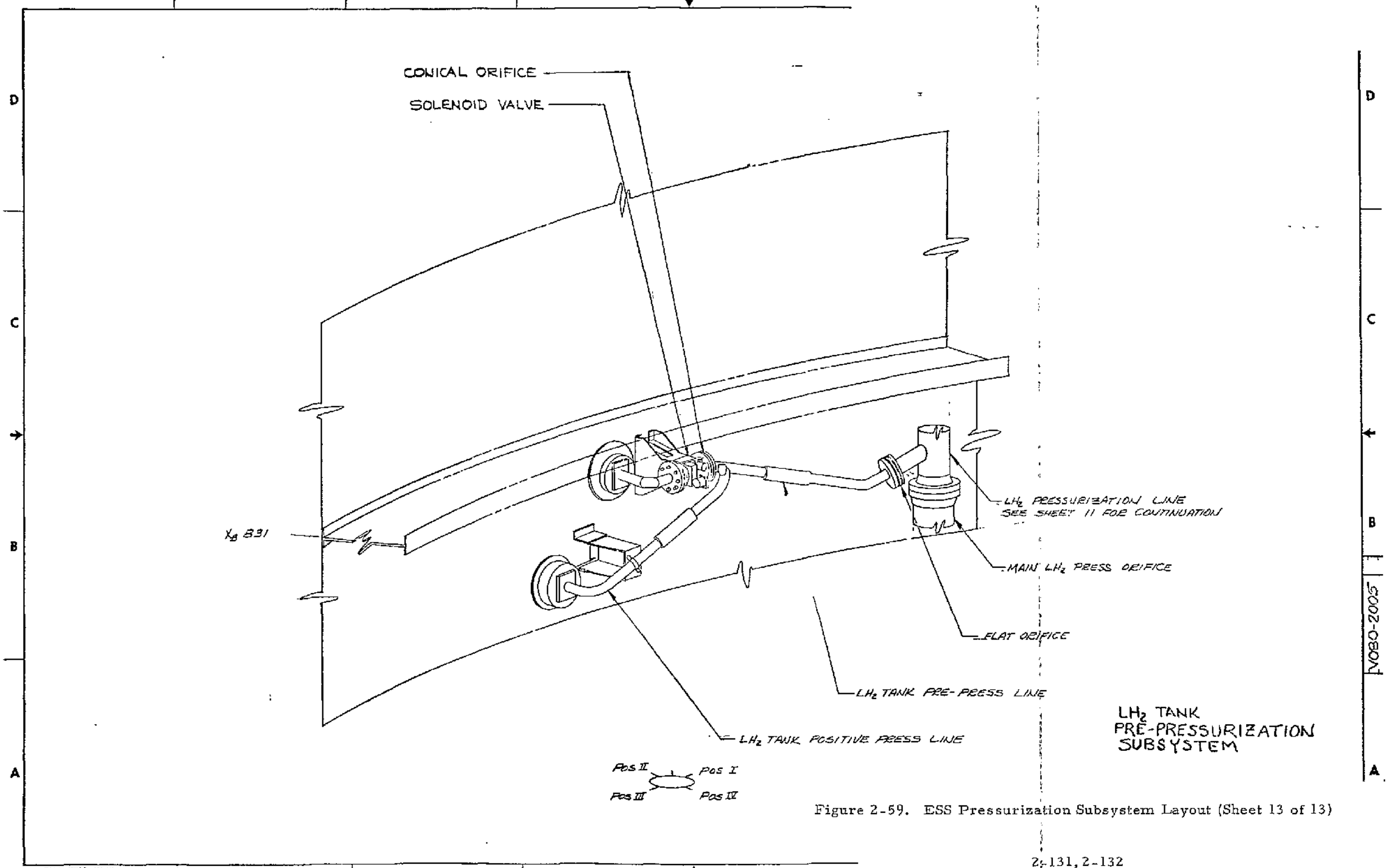


Figure 2-59. ESS Pressurization Subsystem Layout (Sheet 13 of 13)

The propellant feed assembly fail-operational/fail-safe (FO/FS) considerations while the ESS and booster are mated are shown in Table 2-4. Lines are considered to be "pressure vessels" and exempt from FO/FS requirements. No assembly design changes were made as a result of the FO/FS requirement. The LO_2 recirculation helium injection system is identified as an existing system which does not satisfy FO/FS criteria, in accordance with the ground rules of the study control document.

Pressurization. The ESS main propellant tank pressurization system is shown in Figure 2-59. It utilizes the S-II design and components as much as possible. The system design requirements to meet the structural and engine needs were based on the ESS propellant loads associated with the MDAC space station and space tug payloads. The propellant loads corresponding to these payloads are 450, 000 pounds for the space tug and 675,000 pounds for the MDAC space station.

Because of differences in the launch configurations and associated trajectories between the S-II and ESS, structural considerations imposed limitations which affected the LH_2 tank ullage pressure. Table 2-5 is a list of pressurization system differences between the S-II and the ESS. The ESS LO_2 and LH_2 vent valves will be interchangeable and will have a high-mode band, 27.5 to 29.5 psig, and a low mode band, 25.0 to 27.0 psig, to meet the structural requirements primarily and to avoid the cost of the two-vent-valve designs. Meeting the engine inlet requirements is not critical for the vent valve bands selected since large margins are available for all conditions.

The loads resulting from the dynamic pressure occurring at maximum q require a minimum LH_2 tank pressure of 27.5 psid. As can be seen in Figures 2-60 and 2-61, this situation corresponds to an LH_2 tank pressure of approximately 30.6 psia. Figures 2-62 and 2-63 present a similar ullage pressure profile for the LO_2 tanks. This condition was brought about by employing the S-II pressure switches for controlling pre-pressurization. The 34 to 36 psia pressure switch band allows the LO_2 and LH_2 tanks to be pressurized with ambient helium. The pressurization time is longer for the ESS as compared with the S-II because of the larger ullages. For the 675K propellant load, the pre-pressurization times are 190 seconds for LO_2 and 110 seconds for LH_2 as compared with 20 and 60 seconds, respectively. The pre-pressurization band is high enough to permit pressure decay resulting from the cooling of the ullage by the LH_2 and to assure that tank pressure is within the vent valve band at max Q. Figures 2-64 through 2-67 show that this condition is provided for the 450K-pound and 675K-pound propellant loading. The earliest venting in the high mode for the vent valves would occur at approximately 16,000 feet which is about 60 seconds after lift-off. This altitude is well above the minimum of 10,000 feet considered acceptable for venting. From pre-pressurization

Table 2-4. Propellant Feed Assembly Fail-Operational/ Fail-Safe Considerations

Subassembly	Critical to Separation	Remarks and Rationale
Basic engine propellant feed	Prevalve fail-closed	Failure of a normally open valve which is open on the ground and throughout first stage boost to the closed position is judged so remote that redundancy is not considered necessary.
Recirculation	Recirculation valve fail-closed	Same rationale as for prevalve.
	LH_2 recirculation motor fails	Integral part of engine—FO/FS is assumed in engine design.
	Helium injection supply lost	Existing system—redesign for FO/FS not required. Single low-pressure regulator does not meet FO/FS requirements.
Valve actuation	None	
Pogo suppression	Accumulator bleed fail-closed	Same rationale as for prevalve.
Fill and drain	Fill valve fails open	Existing system—redesign for FO/FS not required. Same rationale as for prevalve (except that valve is normally closed versus normally open for prevalve).

Table 2-5. Pressurization Differences Between S-II and ESS

Item	S-II Requirement	ESS Requirement	Reason for Change
LH_2 tank vent valve	Low-mode crack and reseat, 27.5 to 29.5 psig. High-mode (step vent) crack and reseat, 30.5 to 33.0 psig	Low-mode (step vent) crack and reseat, 25.0 to 27.0 psig. High-mode crack and reseat, 27.5 to 29.5 psig	To meet structural requirements for maximum, 27.5 psid in LH_2 tank at end of ESS burn. See Figures 2-61, 2-63, 2-64, and 2-66.
LO_2 tank vent valve	Crack and reseat, 39 to 42 psia	Low-mode (step vent) crack and reseat, 25.0 to 27.0 psig. High-mode crack and reseat, 27.5 to 29.5 psig	Same configuration as LH_2 vent valve to minimize costs and ullage mass. No structural or engine requirement for change. See Figures 2-61, 2-63, 2-65, and 2-67.
Pre-pressurization LH_2 tank	20 seconds with ambient helium at 0.8 lb/sec	450K propellant load, 362 seconds 675K propellant load, 190 seconds Both with ambient helium at 0.8 lb/sec	Longer pre-pressure times result from using same pre-pressure system as S-II for larger ESS ullages. Helium temperature was changed to ambient for ESS LO_2 tank pre-pressure to reduce cost and simplify procedure. Larger ESS LO_2 tank ullage permits change without resulting in excessive pressure decay prior to ESS start. See Figures 2-61 through 2-64.
Pre-pressurization LO_2 tank	60 seconds with cold helium at 1.6 lb/sec	450K propellant load, 178 seconds 675K propellant load, 110 seconds Both with ambient helium at 0.8 lb/sec	
LO_2 tank GO_2 flow control and orifice	10 lb/sec GO_2 at 520 R	7.66 lb/sec GO_2 at 700 R	To provide required flow rate to re-pressurize LO_2 tank to maintain 25.0 to 27.0 psia.
LH_2 tank GH_2 flow control orifice	4 lb/sec GH_2 at 200 R	1.71 lb/sec GH_2 at 530 R	To provide required flow rate to re-pressurize LH_2 tank to maintain 25.0 to 27.0 psia.

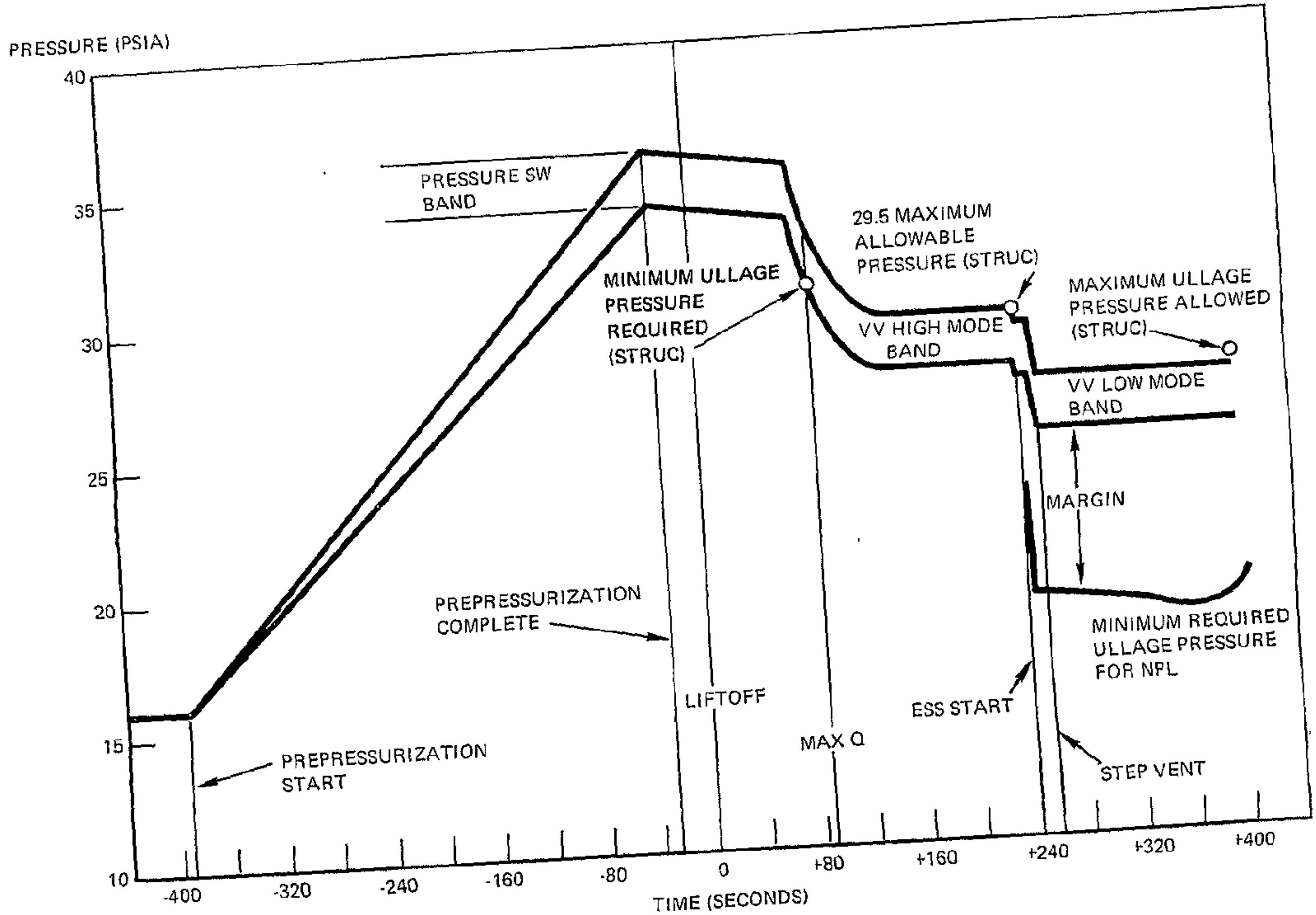


Figure 2-60. ESS LH2 Tank Ullage Pressure, 450K Loading

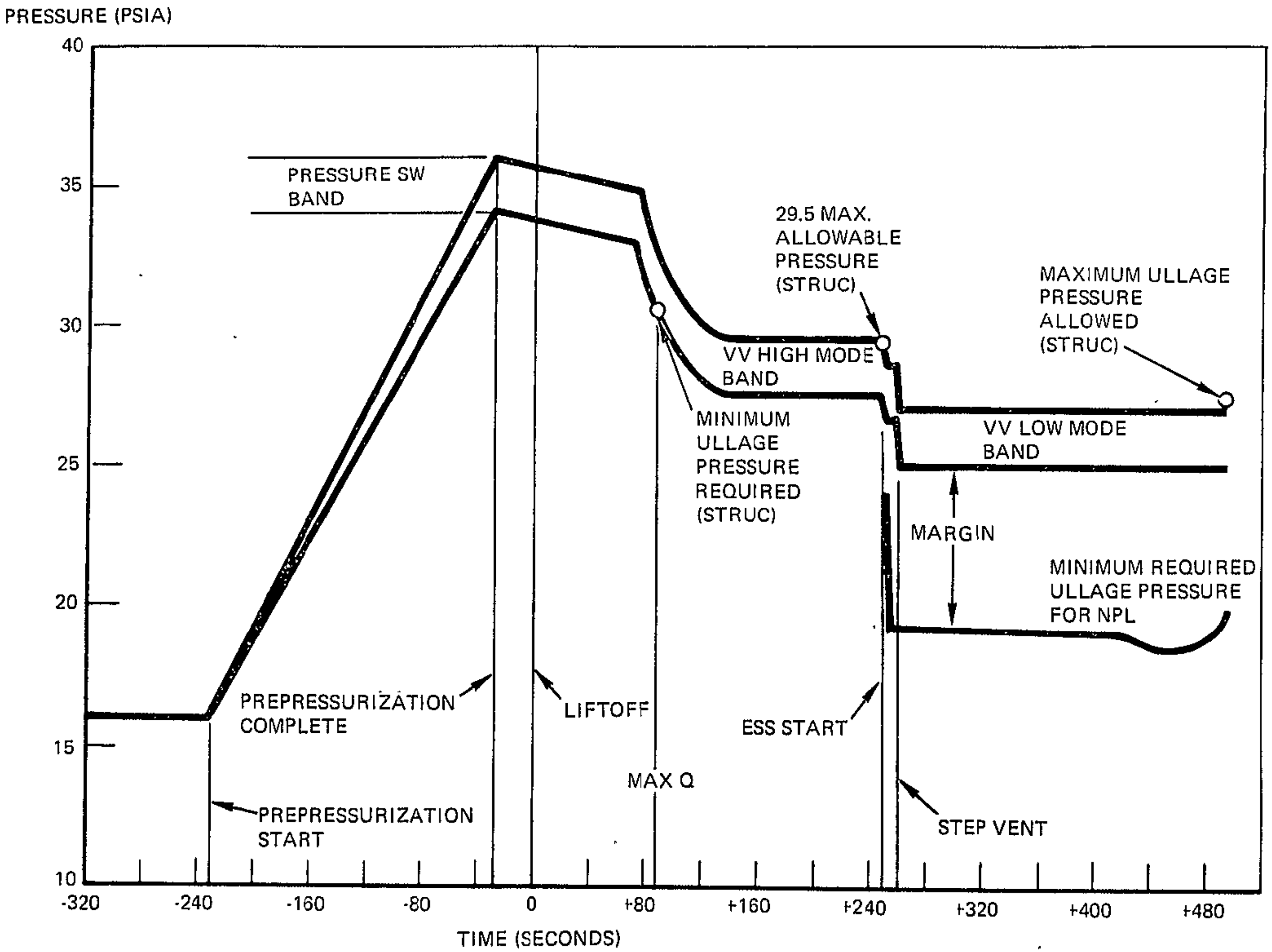


Figure 2-61. ESS LH_2 Tank Ullage Pressure, 675K Loading

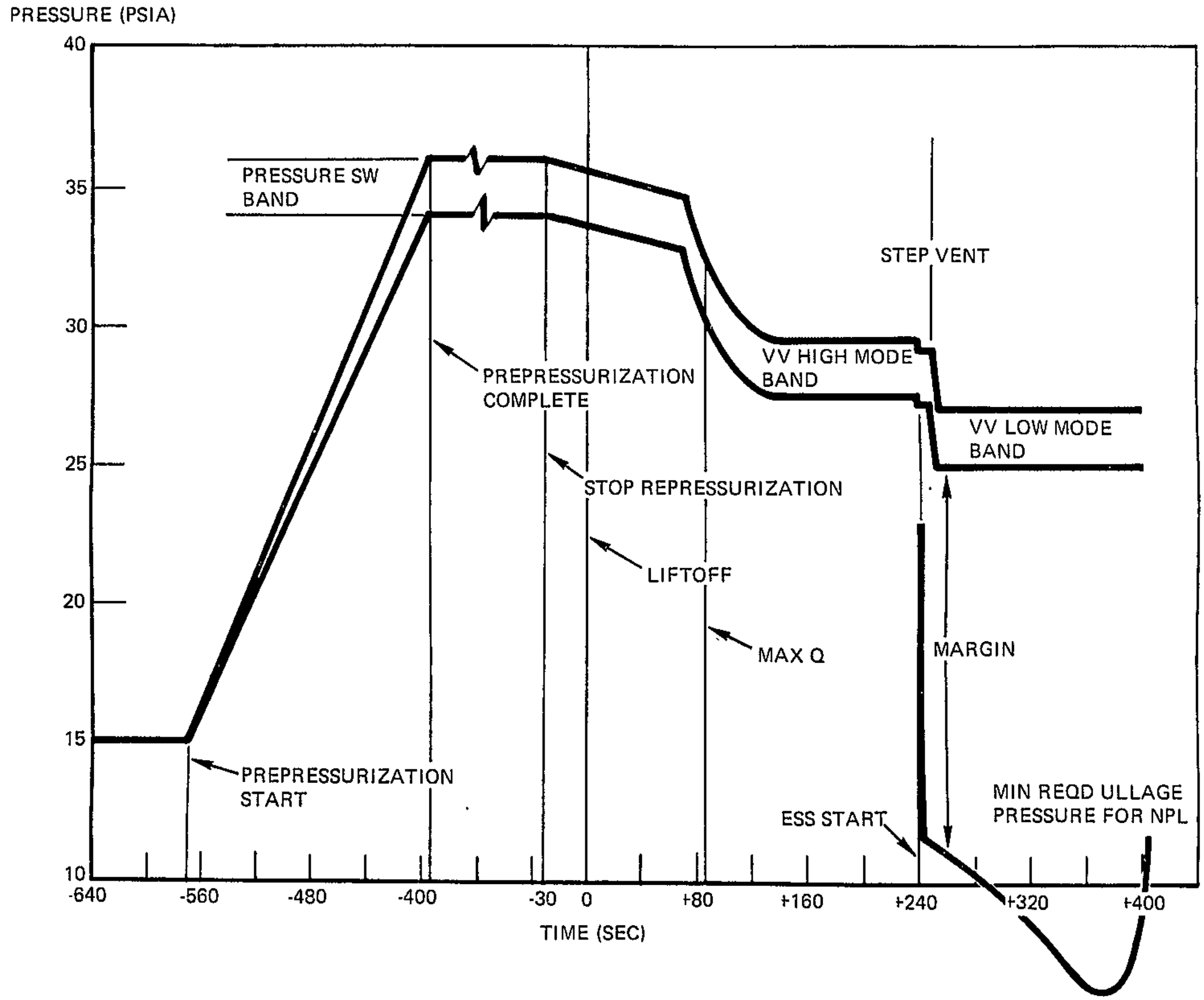


Figure 2-62. ESS LO_2 Tank Ullage Pressure, 450K Loading

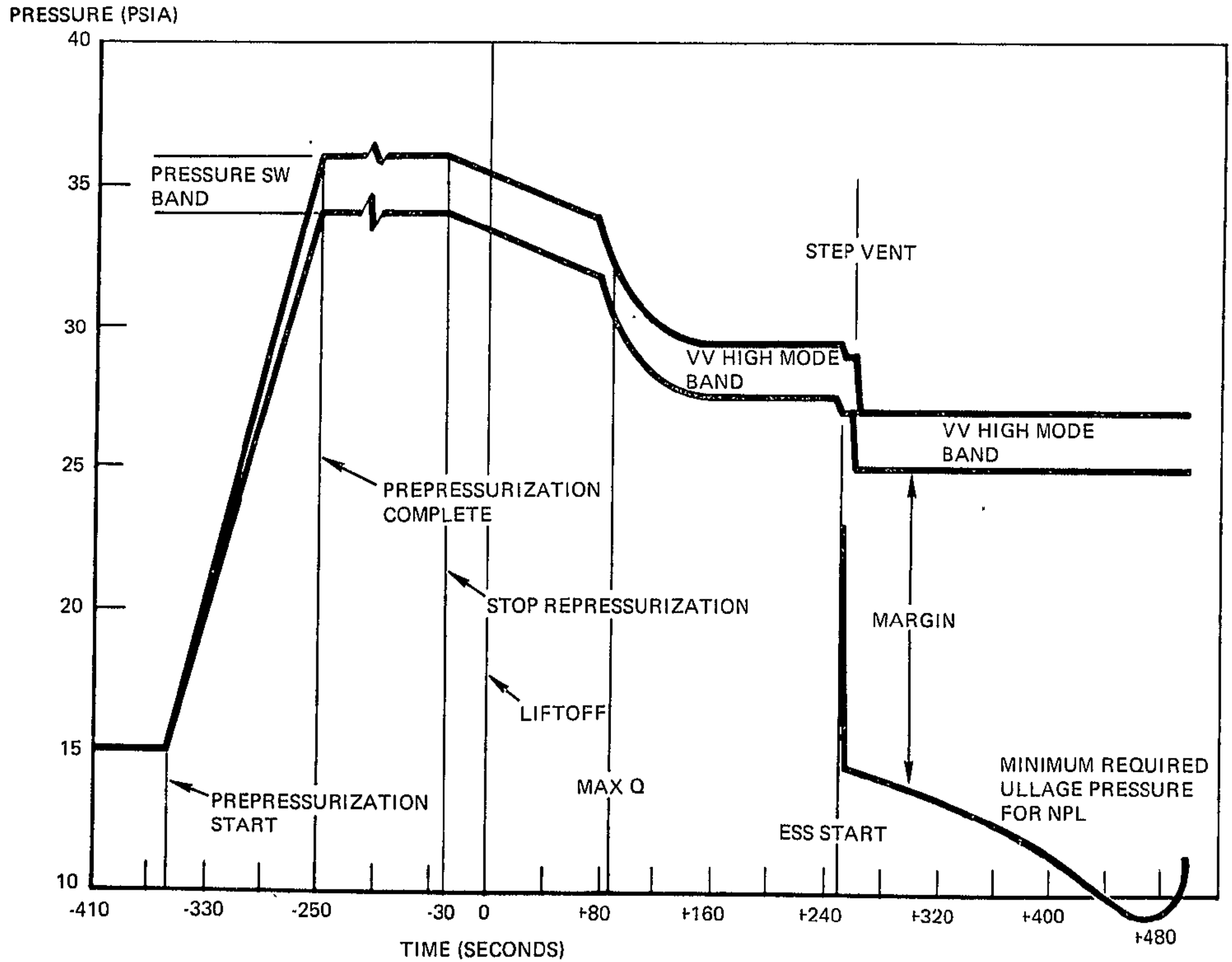


Figure 2-63. ESS LO_2 Tank Ullage Pressure, 675K Loading

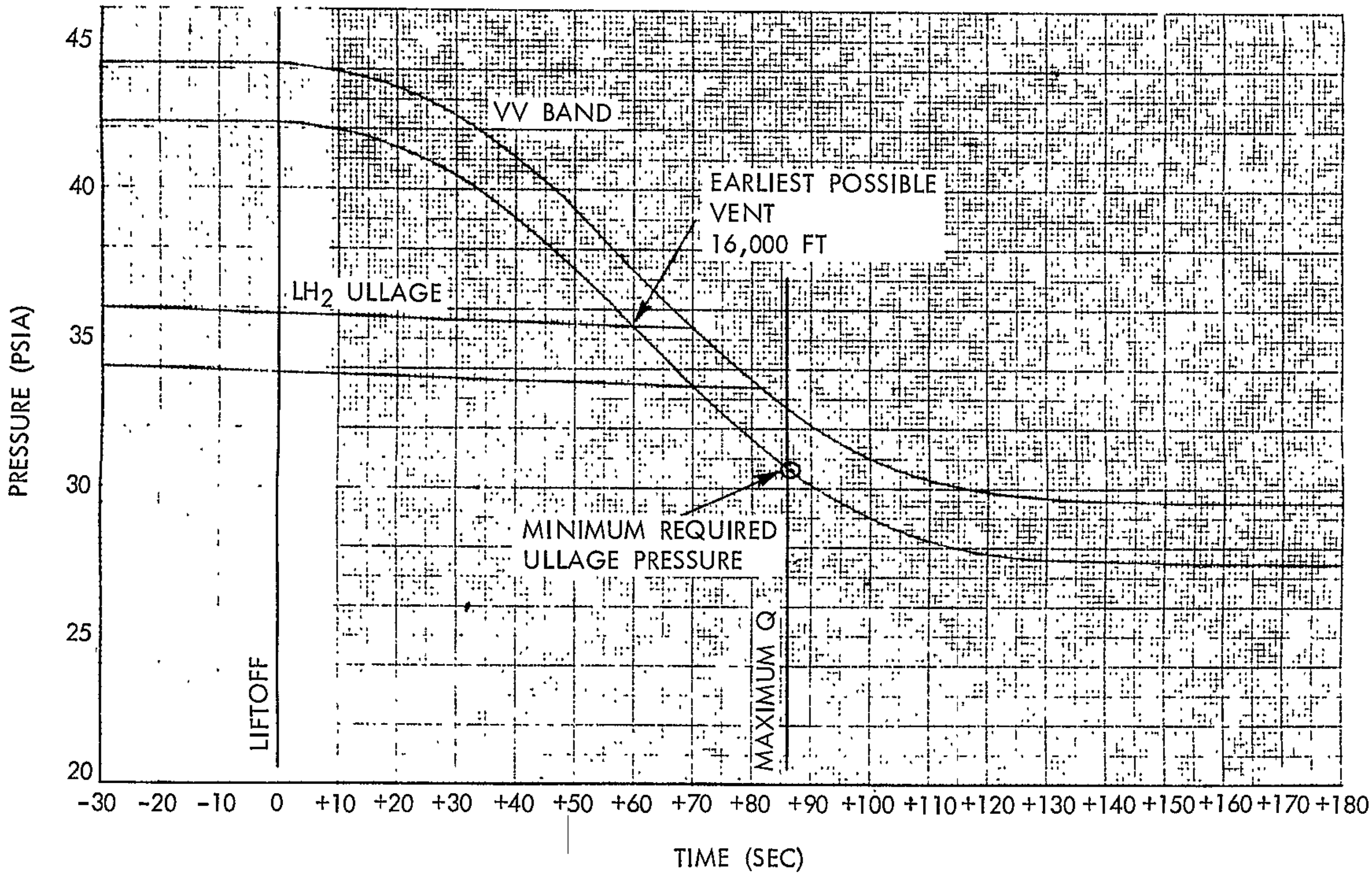


Figure 2-64. ESS 450K Propellant Load, LH_2 Ullage Pressure During Shuttle Boost

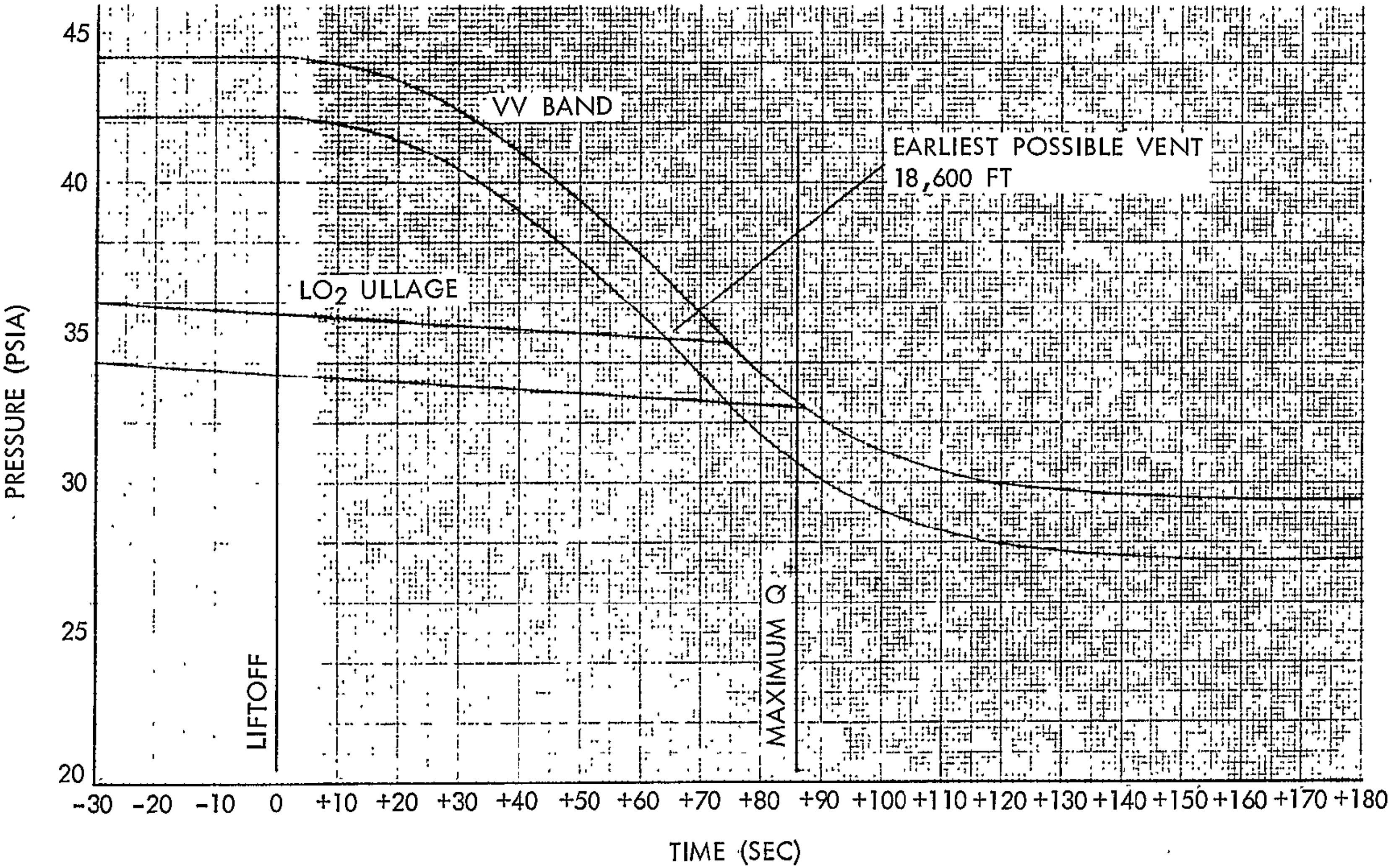


Figure 2-65. ESS 450K Propellant Load, LO_2 Ullage Pressure During Shuttle Boost

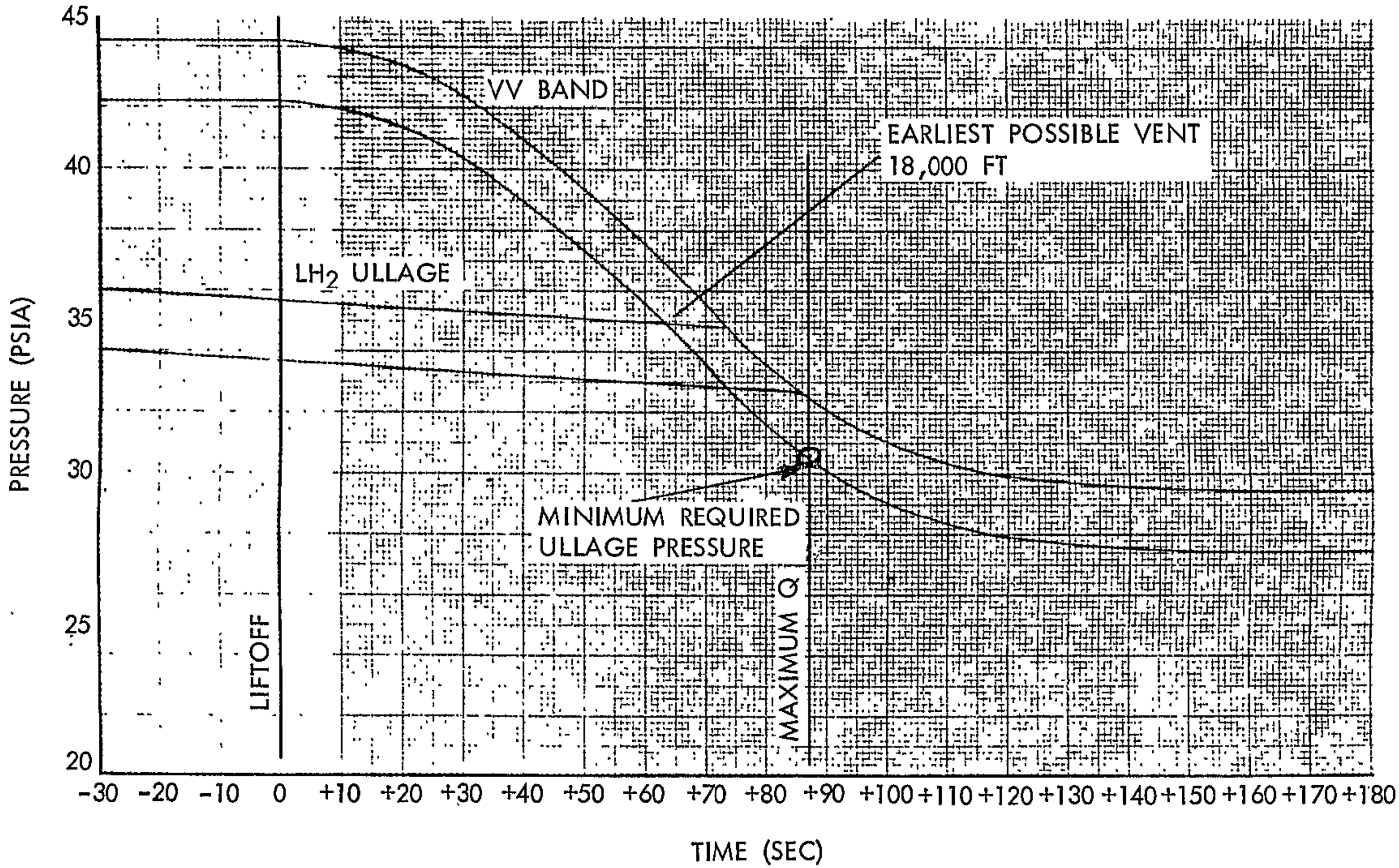


Figure 2-66. ESS 675K Propellant Load, LH_2 Ullage Pressure During Shuttle Boost

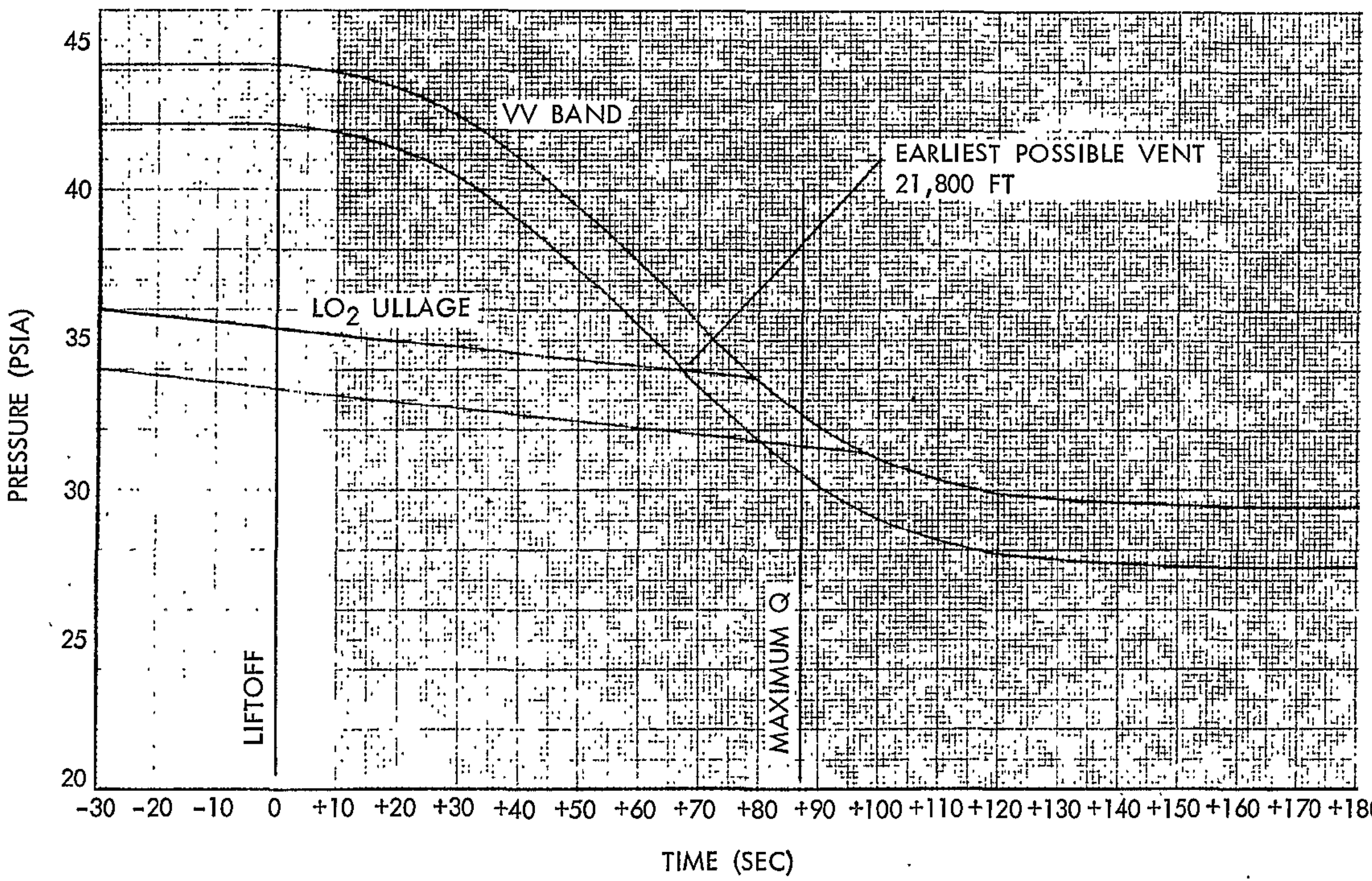


Figure 2-67. ESS 675K Propellant Load, LO_2 Ullage Pressure During Shuttle Boost

until approximately ten seconds after ESS start, the LH_2 and LO_2 vent valves will be in the high mode. This condition assures that the LH_2 tank pressure does not exceed 29.5 psia, which is the maximum allowed because of structural loads at separation of the ESS from the shuttle booster (the same as for the S-II), and that the minimum ullage pressure will not be less than 27.5 psia for ESS start. This assurance will provide an ullage pressure margin of approximately 4.4 and 3.6 psia for the LH_2 and LO_2 tanks over the minimum required.

The GH_2 pressurizing the LH_2 tank during ESS burn enters the tank at 530 ± 130 R. The gas continually heats the LH_2 tank forward bulkhead so that it is warmest at end of ESS burn. Figure 2-68 shows the predicted bulkhead temperatures. Consequently, structural integrity requires that the LH_2 tank pressure does not exceed 27.5 psia at this time. To avoid exceeding this pressure and because of the large inlet margins to the engines, the vent valves were designed to have a low mode of 25.0 to 27.0 psig. Ten seconds after ESS start command, the LO_2 and LH_2 vent valves will be commanded to the low mode for the remainder of the burn.

The LO_2 tank does not have any similar requirements because of its being chilled by the LH_2 tank aft bulkhead (the pressurizing gas enters the LO_2 tank at 700 R) and because the LO_2 tank forward bulkhead has a larger margin of safety than the LH_2 tank forward bulkhead. Figures 2-69 through 2-72 present the available NPSP (net positive suction pressure) and margin for NPL (normal power level) operation of the engines. Minimum inlet pressure margins of 5.8 and 4.2 psi for the LH_2 and LO_2, respectively, are shown for the engine start transient. The margin during run at NPL is least at cutoff. The following is a comparison of the inlet pressure margin at cutoff for NPL (two engines) and EPL (emergency power level) (one engine only).

Inlet Pressure Margin at Cutoff

Propellant Load	NPL (two engines)	EPL (one engine)
450K LO_2	13	2.5
450K LH_2	5.2	2.8
675K LO_2	13.4	3.0
675K LH_2	5.0	2.4

The S-II flow control orifices will have to be changed to provide the flow rates of 1.71 lb/sec and 7.66 lb/sec of GH_2 and GO_2 for pressurizing

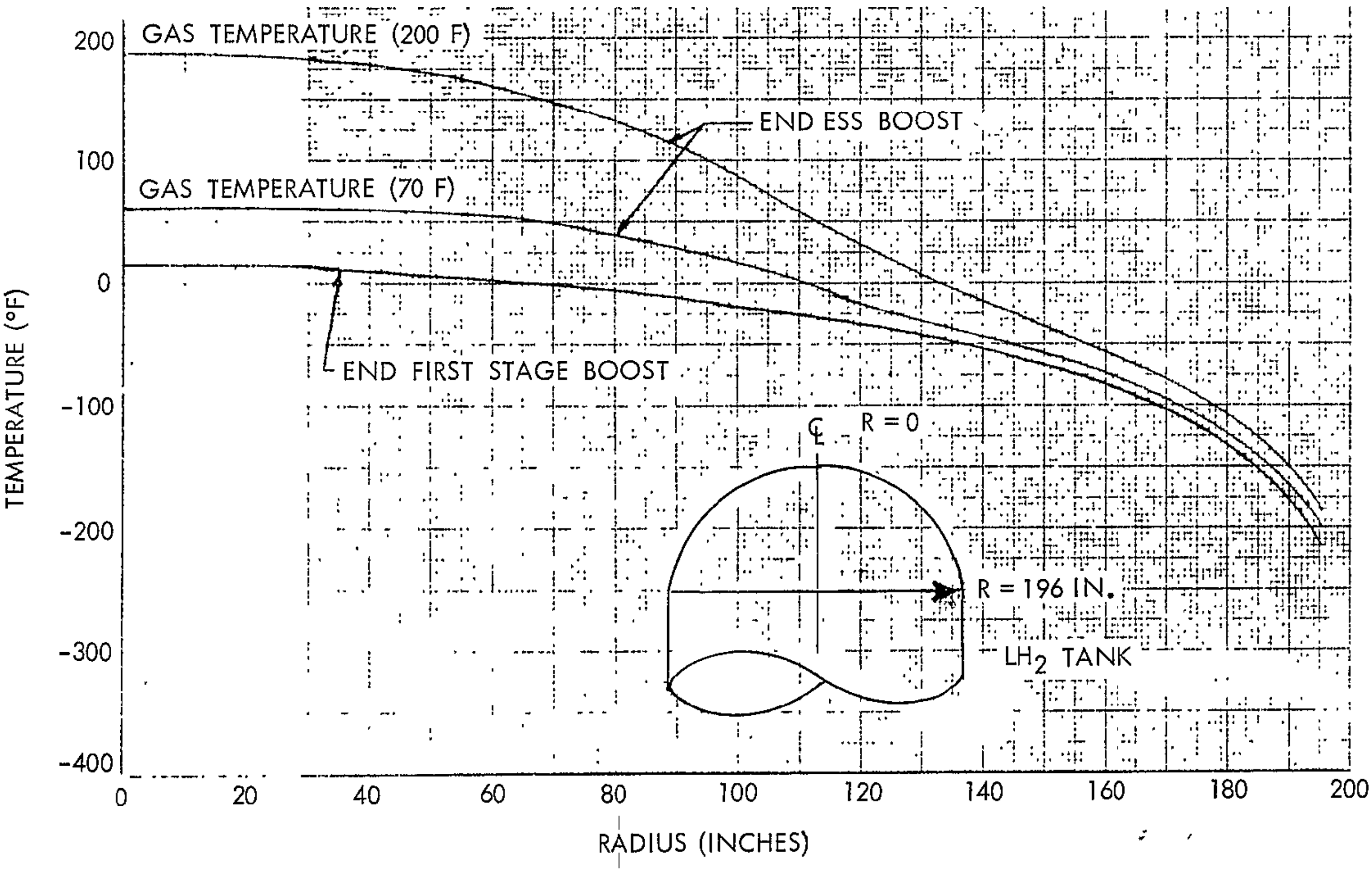


Figure 2-68. ESS Forward Bulkhead Temperature Distribution, LH_2 Loading

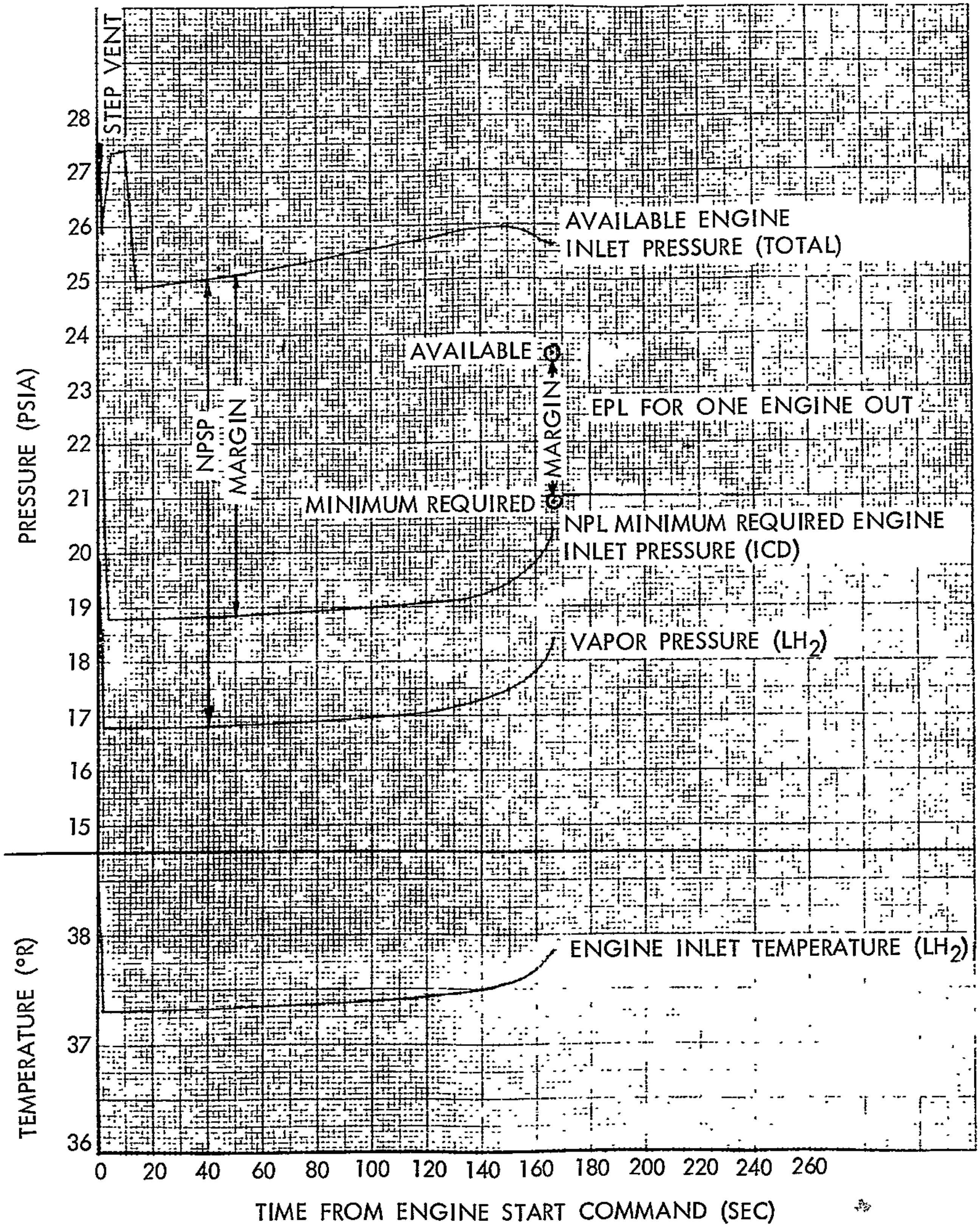


Figure 2-69. ESS 450K Propellant Load, LH_2 Engine Inlet Conditions (Based on Minimum Ullage Pressure)

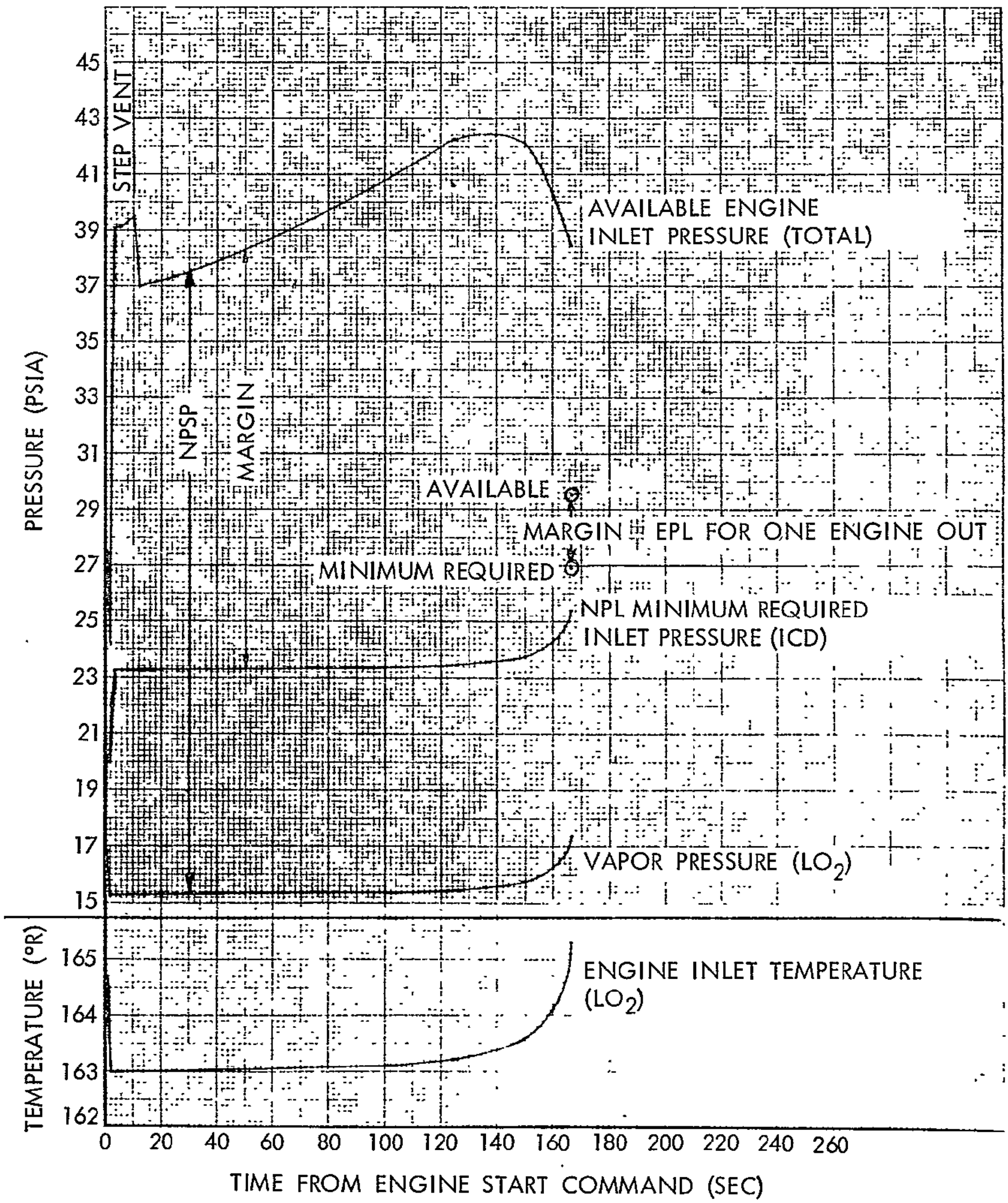


Figure 2-70. ESS 450K Propellant Load, LO_2 Engine Inlet Conditions (Based on Minimum Ullage Pressure)

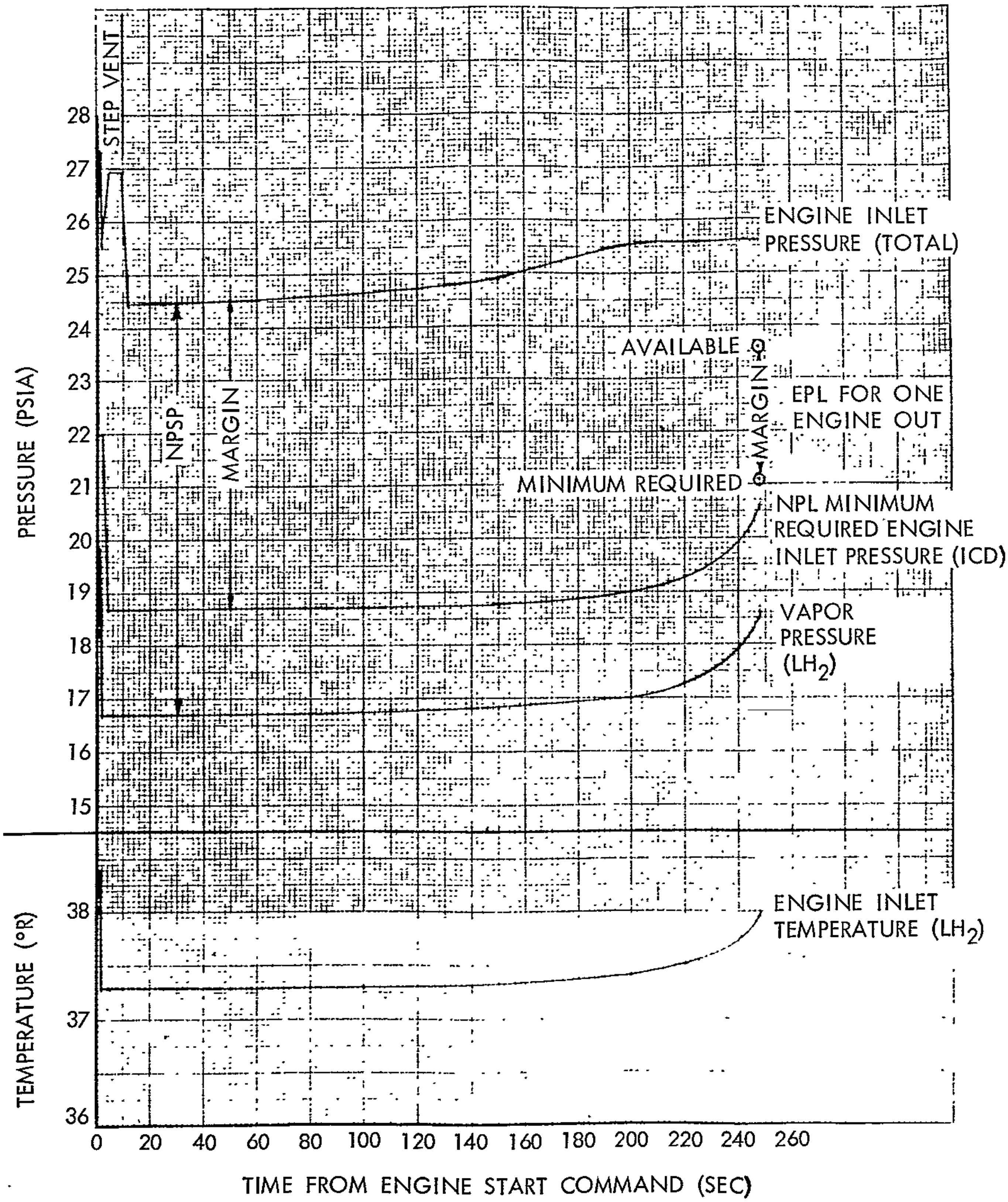


Figure 2-71. ESS 675K Propellant Load, LH2 Engine Inlet Conditions (Based on Minimum Ullage Pressure)

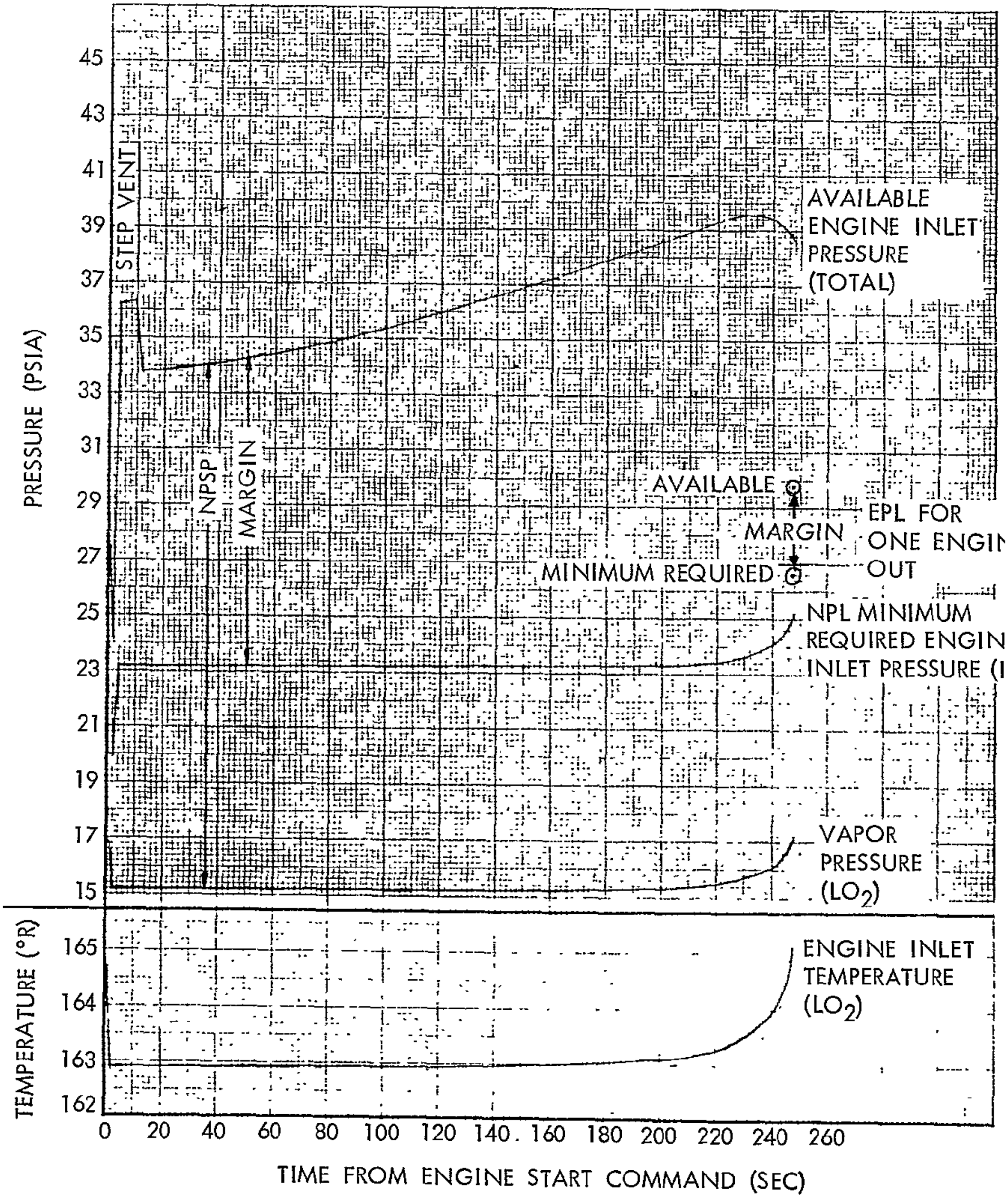


Figure 2-72. ESS 675K Propellant Load, LO_2 Engine Inlet Conditions (Based on Minimum Ullage Pressure)

the LH_2 and LO_2 tanks, respectively. The total ullage mass for each propellant tank and propellant load from pre-pressurization through ESS burn is shown in Figures 2-73 and 2-74.

Evaluation of the pressurization system for its capability of meeting the FO/FS requirements during shuttle boost was made. Although the ESS main propulsion system will not operate until after separation from the booster, loss of the LH_2 ullage pressure to less than 27.5 psid at max Q could result in LH_2 tank structural failure. Outside of line or fitting failure, the primary causes for loss of ullage pressure could occur because a vent valve fails open or because the GH_2 isolation check valve fails open during boost. Similar failures in the LO_2 tank pressurization system could result in collapse of the common bulkhead.

A single order failure mode exists in the GO_2 and GH_2 pressurization lines. If the check valve in the LO_2 line to the engine heat exchanger or the isolation check valve in the GH_2 pressurization line should fail open, the respective tank ullage pressure would decrease because the ullage gas would escape through the engine. This failure mode exists in the S-II and has been considered an acceptable risk. The usual check valve failure mode would not be in the full-open condition since closing of the check valves is assisted by the ullage pressure. The vent valves also present a single order failure for both tanks. Although the vent valves are normally closed, a single failure could open the vent valve. The failures could reduce the LH_2 ullage pressure below the minimum of 27.5 psid required at max Q, or if the LO_2 tank pressure were completely lost, collapse of the common bulkhead could occur. The LH_2 and LO_2 pressurization systems are therefore identified, in accordance with the study control document ground rules, as existing subsystems which do not meet the FO/FS criteria.

Engine Servicing. Three subassemblies make up the engine servicing assembly—the engine helium supply, the engine nitrogen supply, and the engine fuel vent.

Helium must be supplied to the SSEo on the ground, during first stage boost and during ESS burn. The helium is required for purging and for engine valve actuation if the engine utilizes pneumatic actuation. Based on the requirements of the SSEo ICD, a 15 cubic foot supply bottle at 4500 psi is needed to provide an adequate helium supply. A flight regulator is used to reduce the pressure to the 1600 ± 200 psia specified at the engine connect. Because of the large, sudden demands, a 1500-cubic-inch accumulator is required downstream of the regulator. The engine helium supply subassembly is shown in Figure 2-58. S-II experience indicates that the stipulated helium requirement is much larger than should be necessary. Contact with all three potential engine contractors points out that a smaller amount of helium at lower engine connect pressure will meet engine requirements.

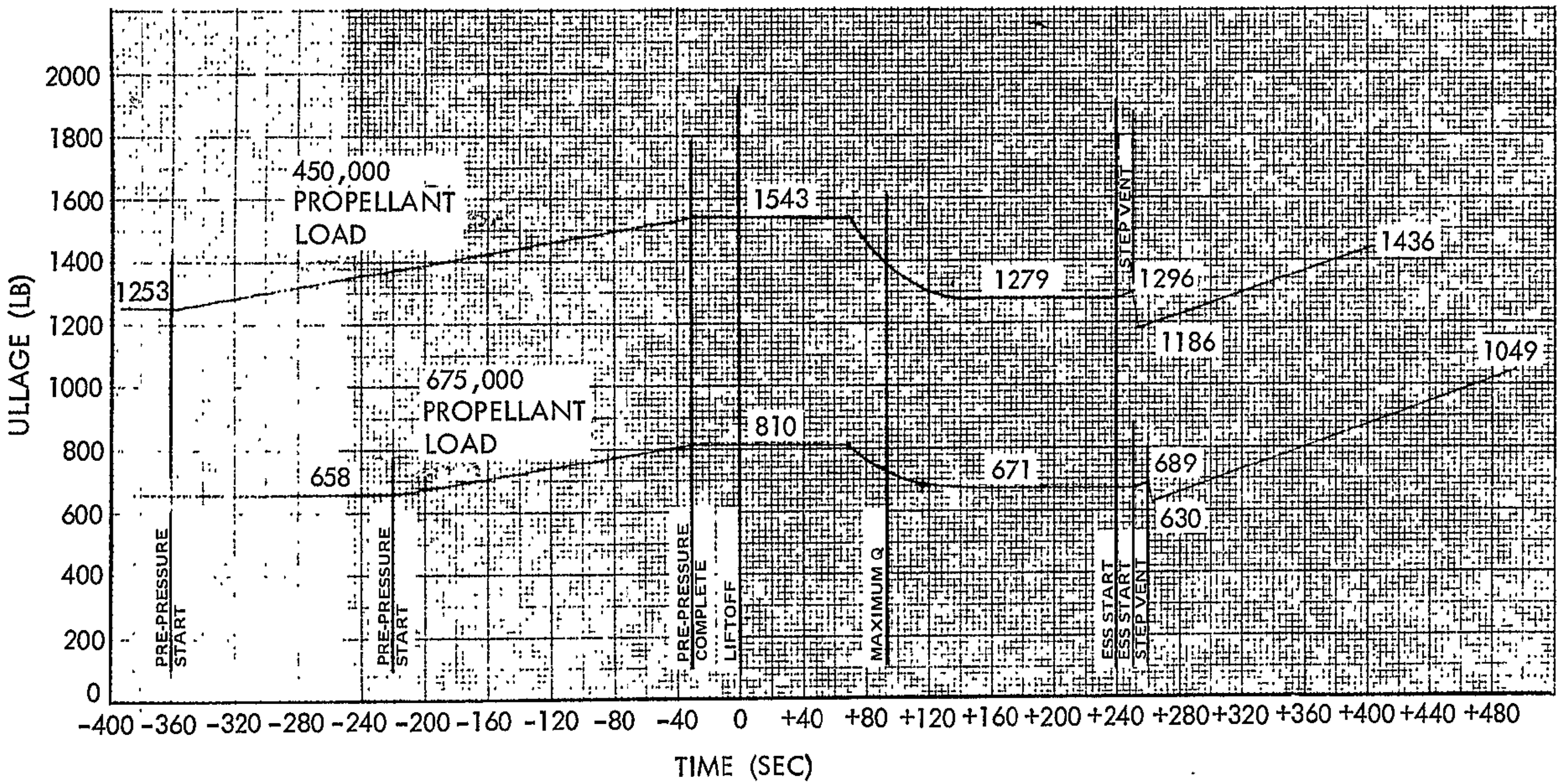


Figure 2-73. ESS LH_2 Tank Ullage Mass Prepressure, 34-36 Psia

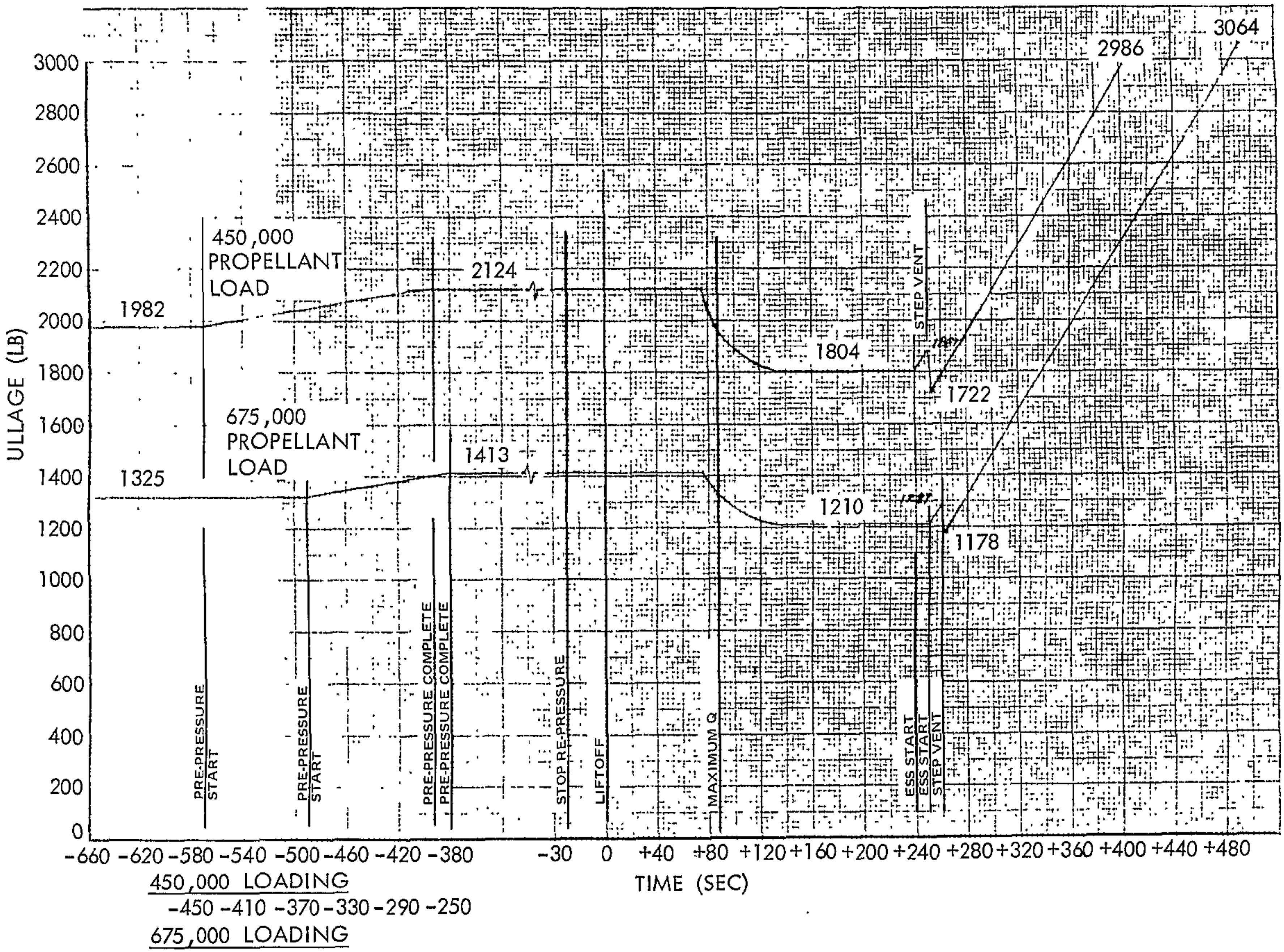


Figure 2-74. ESS LO_2 Tank Ullage Mass Prepressure, 34-36 Psia

The engine nitrogen supply is required for ground purging: the subassembly configuration as shown in Figure 2-58 is merely a tubing manifold from the aft umbilical to each engine-connect panel.

The engine fuel vent as shown in Figure 2-58 consists of a manifold from each engine connect (in order to vent the engine pump seals) to the aft umbilical where it is connected to a vent line for ground operations. For flight operations, the vent is open at the stage skin, but no flow is anticipated until ESS engine start, at which time the ESS altitude is great enough to preclude any fire hazard.

The engine servicing assembly FO/FS considerations are shown in Table 2-6. As a result of the FO/FS requirement, a check valve will be provided in the engine helium supply disconnect and a redundant pressure regulator and control valve added in the engine helium supply subassembly.

Table 2-6. Engine Servicing Assembly Fail-Operational/Fail-Safe Considerations

Subassembly	Critical to Separation	Remarks and Rationale
Engine helium supply	Supply control valves fail open	Redundant control valve supplied in disconnect (F/O). Assumes that the engine will start with loss of supply in flight (F/S).
	Regulator fails (open or closed)	Redundant regulator and control valve supplied (F/O). Assumes that engine will start with loss of supply in flight (F/S).
	Dump valve fails open	Failure of normally closed valve to energized position is judged so remote as not to require redundancy.
Engine nitrogen supply	None	
Engine fuel vent	None	

Thrust Vector Control. An independent hydraulic system, essentially conforming to MIL-H-25475-type II, will be installed at each orbiter engine position. The systems will provide the forces to position and gimbal the engines in response to the vehicle flight control commands.

Each system will employ two linear-acting actuators to gimbal the engine about each of two nearly-perpendicular axes. Each actuator will be capable of deflecting the engine through the maximum rated gimbal angle (±7 degrees). The gimbal axes will be oriented so that they nominally bisect the vehicle pitch and yaw axes. Therefore, pitch, yaw, and roll deflections will be achieved by the combined motion of the two actuators. The actuator orientation is chosen to provide the largest possible engine deflection in the pitch and yaw planes (approximately 10 degrees). The actuator orientations are shown on Figure 2-75. The angular excursions shown on the drawing are slightly larger than the capability of the actuator to allow for snubbing, overtravel, and misalignment.

With the actuators at null, the engines will be aligned to point each thrust vector toward the vehicle's nominal center-of-gravity. The alignment results in the engines being canted radially outboard by 12 degrees when at full thrust and 13 degrees at zero-thrust. The result creates wasted thrust components normal to the line-of-flight; therefore, to reduce the losses and resultant payload penalty, the flight control system will apply 6-degree inboard commands to position the thrust vectors more nearly parallel to the line-of-flight. The flight control system will sense if only one engine of a pair is firing and will remove the bias commands: it thus causes the thrust vector of the firing engine to point toward the nominal center of gravity. The biased commands will not be applied until the engine start transients have subsided and will be removed shortly before engine shutdown.

NASA is currently evaluating the potential addition of an engine driven accessory pad to the space shuttle engine. If approved, the ICD would be changed accordingly. The contractor has assumed that this capability will be provided and has described a hydraulic system based on this assumption. In the event that this capability is not provided, the type of system selected would be a pneumatically driven hydraulic system.

A simplified schematic of the system to be used for the orbiter engine is shown on Figure 2-75. Each engine system will have identical engine-driven pumps, servo actuators, accumulator/reservoir/manifold assemblies (ARMA's), and electric motor-driven auxiliary pumps.

The following are descriptions of the hydraulic system components:

Actuators. Figure 2-76 is a schematic of the S-II servo actuator, which is quite similar to the one which will be used on the ESS. The significant components and features of the actuator are the following:

1. An integral servo valve with mechanical feedback of piston position
2. A dynamic pressure feedback (to attenuate load resonance)
3. A complete filtration of all incoming fluid with additional filtration for servo valve first-stage flow
4. A pre-filtration valve and a bleed valve for flushing and bleeding
5. A hydraulic lock valve to restrain piston motion in the absence of applied hydraulic power
6. Redundant seals on piston rod
7. A cylinder bypass valve to make possible manual positioning of the actuator (the solenoid operator, as shown for S-II, was required because of the J-2 engine-start characteristics in static firing; the need for this feature on ESS has not yet been established).

The S-II actuator has no redundant features other than piston position potentiometers and rod seals. It is proposed that the ESS actuators should have three servo valve first stages arranged in parallel to produce a majority voting system such as was developed for (but never implemented on) the Saturn S-IVB actuator. This arrangement does not rigorously provide fail-operational/fail-safe (FO-FS) capability with respect to the actuator but does allow operation after one failure (or after two failures in some instances) of the weakest elements in the actuator. It should be noted here, however, that overall FO/FS (or operational) is achieved for many failure cases by virtue of two engines (and associated independent TVC systems), in conjunction with the capability of completing the mission with one engine out.

The preceding paragraph describes what are currently judged to be typical actuators for the ESS. It is possible, however, that the actuators which will be developed for the orbiter, or booster, will be directly applicable to the ESS; if applicable, they will be used.

Actuator sizing will depend on final selection of the engine since each of the proposed engines has different actuator moment arms.

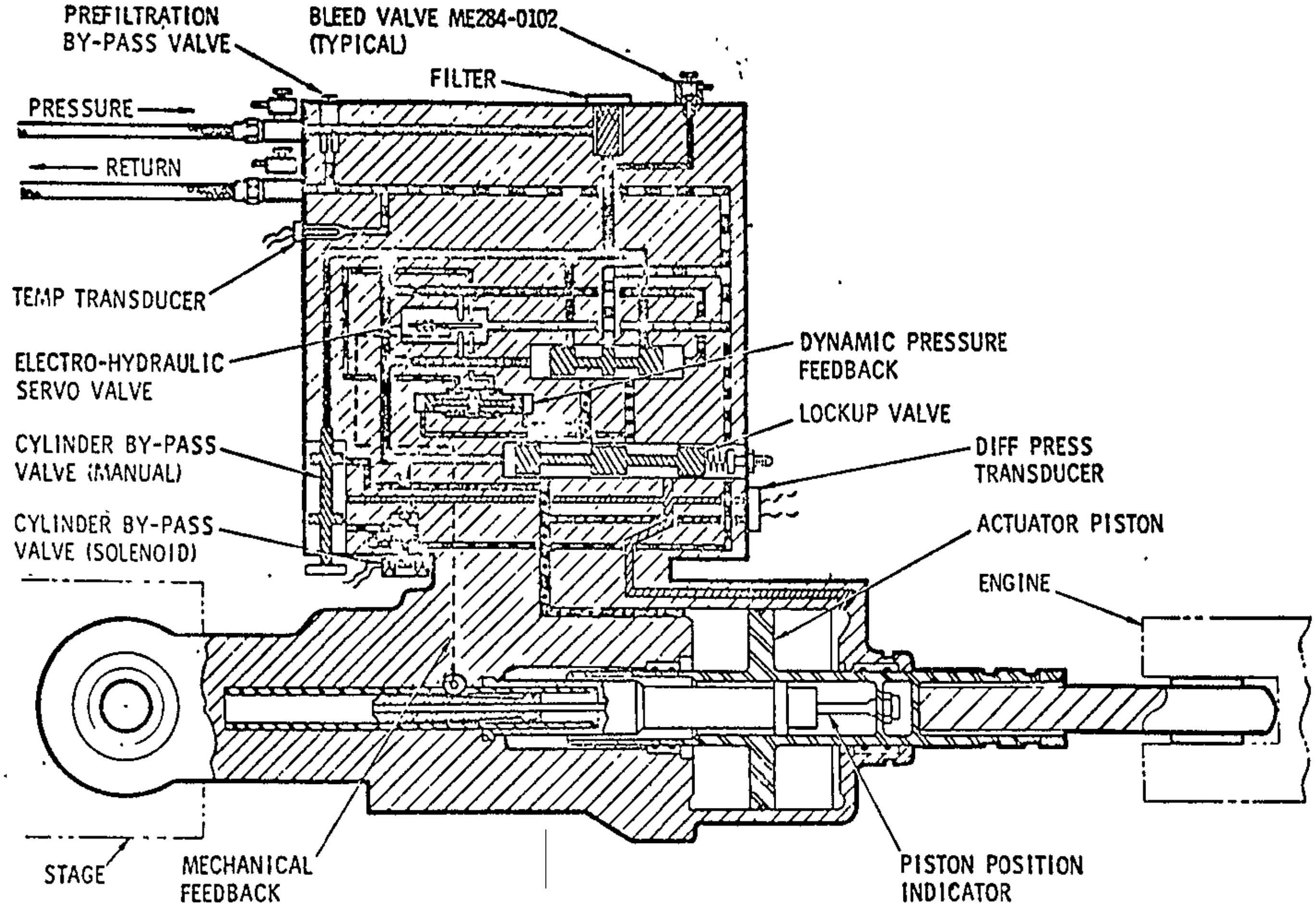


Figure 2-76. Typical Servoactuator

Engine-Driven Pump. The engine-driven pump will be similar to the S-II/J-2 engine-driven pump shown in Figure 2-77. The required pump capacity is approximately 25 gpm. This flow rate must be obtainable at the minimum pump speed (TBD) consistent with the minimum thrust level utilized.

The pump shown in the figure is of the variable-delivery, constant-pressure (adjustable) type, such as is commonly used in commercial and military aircraft. Shown in the figure, but not required for ESS, are the solenoid valve, filter, and orifice arrangement.

Figure 2-76 shows a normally closed solenoid valve interconnecting the pump inlet and outlet ports. The function of this valve would be to minimize the torque applied to the turbine during engine start if required on the selected engine. If not required, the valve will be deleted.

Electric-Driven Pump. An electric-driven pump will be provided to accomplish ground checkout and to provide hydraulic power for the engine control system if required by the engine. If hydraulic power is not required, the pump motor will be ground-powered only.

The pump elements will be similar to those of the engine-driven pump previously discussed but will not incorporate the thermal barrier or provisions for thermal control bypass. Also, it is probable that the case drain will be routed back to the reservoir instead of to the pump inlet as shown in the figure.

Accumulator/Reservoir/Manifold Assembly (ARMA). The ARMA will be similar to that of the S-II which is shown in Figure 2-78. The basic functions of this unit are to provide as follows:

1. Storage of high-pressure fluid in order to supply short-term demands in excess of pump capability (including the time during the engine start before the main pump is up to speed)

2. Damping for hydraulic transients

3. A low-pressure reservoir in order to make up for external leakage, thermal contraction, and the volume accepted or discharged from the accumulator

4. Overpressure relief

5. Filtration of all incoming and outgoing fluid

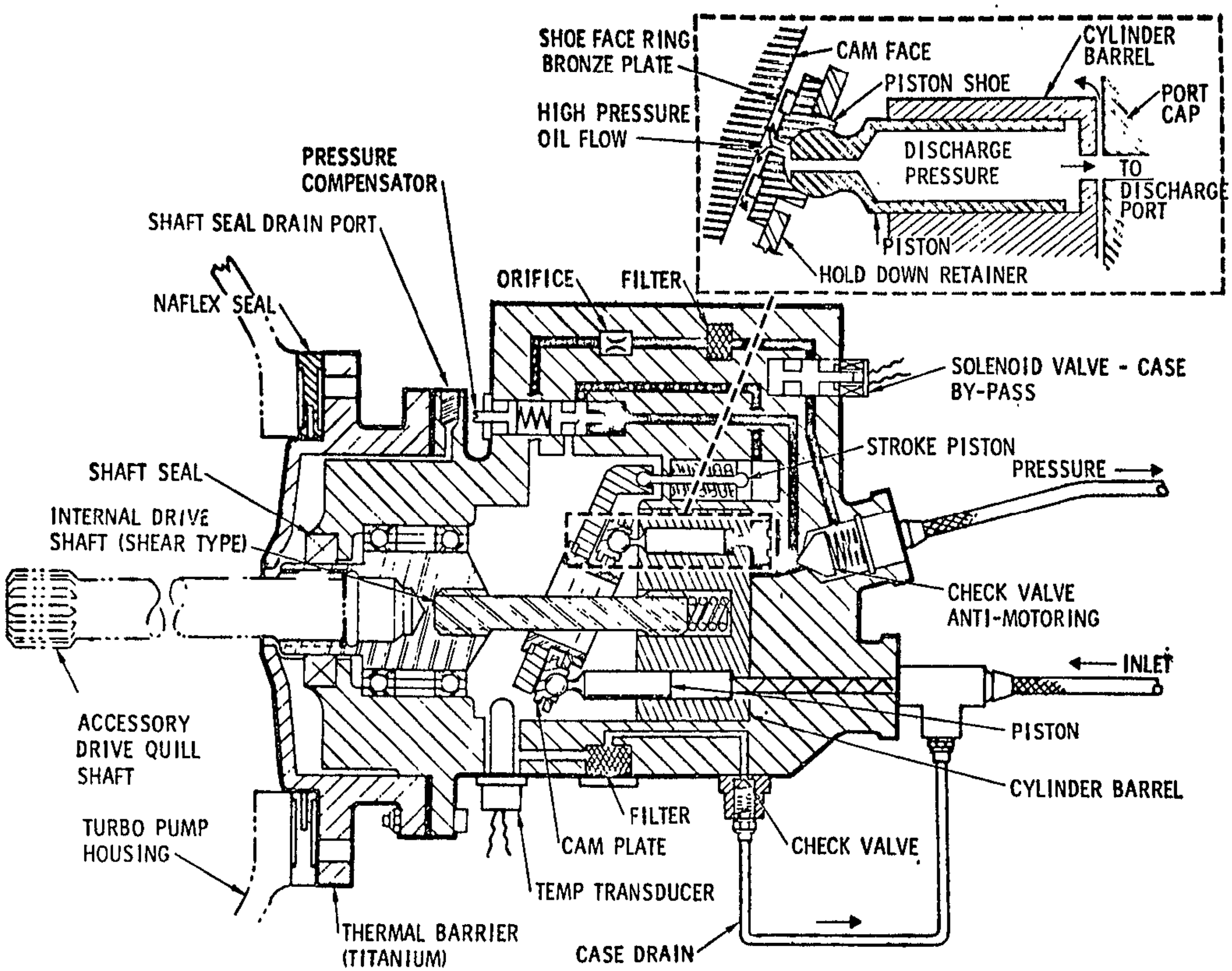


Figure 2-77. Typical Engine-Driven Pump

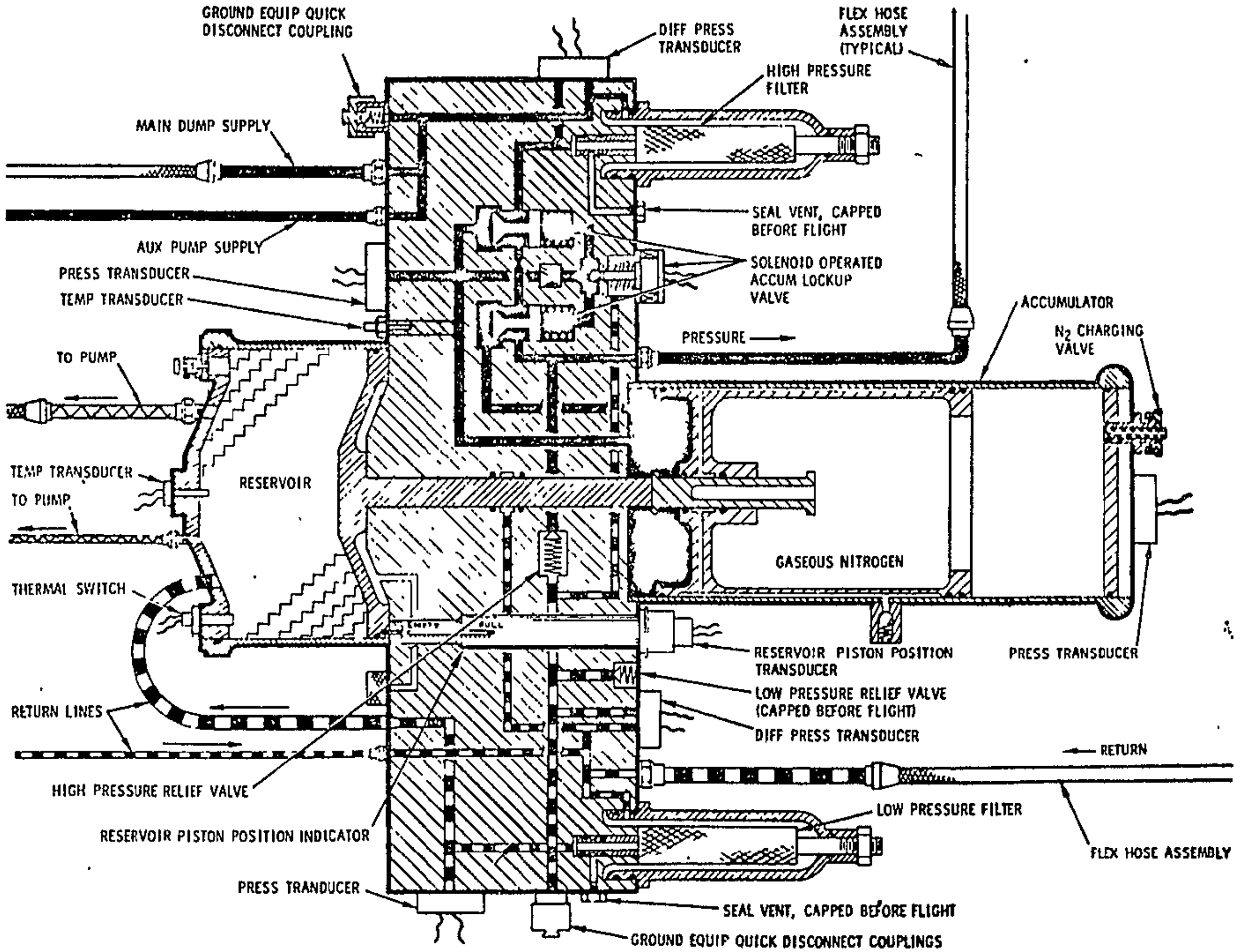


Figure 2-78. Typical Accumulator, Reservoir Manifold Assembly (ARMA)

6. Proper inlet pressure for the pumps

7. A means connecting to external hydraulic power for servicing and checkout

Additional analysis of the transient hydraulic demands during the engine start sequence (and the engine nozzle extension if applicable) will be required to establish the accumulator requirements. The accumulator size will, in turn, partially govern the reservoir size. However, to judge from a cursory evaluation—the accumulator, reservoir, and gas charge volumes for a system which does not have to supply the engine with hydraulic fluid will be approximately as follows:

Item	Volume (cu. in.)
Reservoir	300
Accumulator oil	100
Gas charge	300

Safing. After the velocity cutoff of the ESS, the residual propellants in the main tanks will be dumped through the engines, and the tank will be safed.

The propellant dump is required only if engines are to be recovered in orbit. If engines are not recovered, tank safing by venting will be accomplished immediately after main stage cutoff.

Propellant dump is accomplished by opening the engine propellant valves. The result is a propulsive force applied to the ESS. The propellants are not dumped simultaneously—a precaution to eliminate any safety hazard. The planned sequence is to dump LO_2 first, hold for five minutes following the completion of the LO_2 pump, and then to initiate the LH_2 dump.

The amount of propellant remaining depends upon the mission, whether propellant ballast is used because of load considerations, and the flight performance reserves used to meet the required performance. Dumping the propellant residuals through the engines produces propulsive thrust during the coast period which will affect the nominal 66 x 100-nm orbit.

In order to determine the effect of dumping the propellants, a brief performance analysis was conducted for two configurations. The first configuration had only the propellant and reserves required to perform the mission, and the second had additional propellant loaded as ballast.

Each configuration was analyzed for the minimum and maximum propellant residual cases. The configurations were first checked to determine the maximum effect of dumping by assuming the total propellant was dumped in the posigrade direction. The propellant dump conditions considered for this analysis are presented in Table 2-7.

Case 1A, which has nominal propellant loading and minimum residuals, dumps a total of 6700 pounds of propellant in a period of 656 seconds after main stage cutoff. With a posigrade propellant dump, the orbit was changed so that the apogee increased from 100 nm to 115 nm. If the orbit is circularized when the vehicle reaches an altitude of 100 nm (1700 seconds), a ΔV of 165 fps is required as compared with 63 fps for the nominal case with no propellant dump (66 x 100 nm-orbit). If the orbit is circularized at 115 nm, the ΔV required is 88 fps.

Case 1B, which has nominal loading and maximum residuals, dumps 19,500 pounds of residual propellant in a period of 1227 seconds. In this case, the apogee is increased from 100 nm to 135 nm, and a ΔV of 237 fps is required to circularize the orbit when an altitude of 100 nm is attained. If the orbit is circularized at 135 nm the ΔV required is 120 ft./sec. In Case 1B as in Case 1A, the propellant dump has been completed before the vehicle reaches apogee.

For Cases 1A and 1B, it is felt that if the propellant dump can be predicted fairly accurately, then the effect of dumping can be minimized by dumping during part of the coast in the posigrade direction and part in the retrograde direction. Another way to minimize dumping effects would be to change the boost phase performance to compensate for the effects of propellant dumping. However, if the propellant dump cannot be predicted accurately enough, then posigrade dumping can be used; in this case either the higher ΔV's just quoted will be required or circularization at a higher orbit can be considered.

Configuration 2 has considerably more propellant residuals and, consequently, has a much greater effect on the coast orbit. For example, Case 2A which has additional propellant for ballast dumps 58,000 pounds in a period of 3800 seconds. If the propellant is all dumped in the posigrade direction for this case, the altitude has exceeded 175 nm by the time all the propellant has been dumped. Also, the perigee exceeds 100 nm, and no second opportunity to circularize at 100 nm occurs. Similarly, for Case 2B, a propellant residual of 71,000 pounds is dumped in a period of 4383 seconds, and the deviation from the nominal orbit is even greater. Since dumping continues long after the distance of 100 nm is attained, if the orbit is circularized at 100 nm, the circular orbit will then be affected by the remainder of the propellant dump.

Table 2-7. Propellant Dump Conditions

Condition	Amount Dumped (lb)	Thrust During Dump* (lb)	I_{sp} of Dump* (sec)	Time to Dump (sec)	I (lb/sec)	Proposed Sequence of Dump
Case 1. No propellant ballast						
A. Residuals only						
LO_2	3000	720	30	125	90,000	LO_2 first, hold 5 min, then dump LH_2 and safe.
LH_2	3700	480	30	231	111,000	
Σ least condition	6700			356	201,000	
B. Residuals + 2 (FPR)						
LO_2	14,000	720	30	583	420,000	LO_2 first, hold 5 min, then dump LH_2 and safe.
LH_2	5500	480	30	344	165,000	
Σ maximum condition	19,500			927	585,000	
Case 2. Propellant ballast						
A. Residual + ballast						
LO_2	6000	720	30	250	180,000	LO_2 first, hold 5 min, then dump LH_2 and safe.
LH_2	52,000	480	30	3250	1,560,000	
Σ least condition	58,000			3500	1,740,000	
B. Residual + ballast + 2 (FPR)						
LO_2	17,000	720	30	708	510,000	LO_2 first, hold 5 min, then dump LH_2 and safe.
LH_2	54,000	480	30	3375	1,620,000	
Σ maximum condition	71,000			4083	2,130,000	

*Accurate to ±20%

It is desirable to keep propellant residuals below the amount that can be dumped before apogee of the coasting orbit is achieved. The current load trajectories for all three missions do not use propellant as ballast. Consequently, the propellant residual is considerably less than in Cases 2A and 2B, and propellant dumping will be completed before the apogee of the coasting orbit is achieved.

Subsequent to the propellant dump, the main propellant tanks will be safed by actuating redundant (in parallel) ordnance vent valves mounted on each propellant tank. A nonpropulsive manifold is provided for each tank safing vent.

The safing assembly FO/FS considerations are shown in Table 2-8. No design changes were made to the safing system as a result of the FO/FS evaluation.

Engine Compartment Conditioning. The engine compartment conditioning assembly is utilized for purging the aft boattail. The purge conditions and inerts the aft boattail with GN_2 prior to launch. Air is used for conditioning until the start of loading. The assembly is shown in Figure 2-79. A thermal control purge is also provided to the electrical containers in both the forward and aft skirts. Like the engine compartment purge, both air and GN_2 are used.

Table 2-8. Safing Assembly FO/FS Considerations

Subassembly	Critical to Separation	Remarks and Rationale
Propellant dump	None	Only critical to engine recovery.
Safing vent	Vent fails open	Open failure of normally closed ordnance valve judged so remote as not to require redundancy. Existing system.
	Vent fails closed	Not critical to separation but is critical to rendezvous operations with orbiter. If first vent fails, redundant vent is provided (F/O). If second vent fails, normal tank relief will protect tank (F/S).

Subsystem Operation

The main propulsion subsystem (MPS) operation is shown by Figure 2-80 which depicts the major MPS events from the initiation of launch countdown at lift-off minus 2 hours to the completion of propellant dump and safing; after this time the MPS is not active. Figure 2-80 is based upon a MDAC space station payload. For other payloads the propellant loading will vary so that fill time, prepressurization time, engine burn time, etc., will change accordingly.

The main propulsion system operation is essentially the same as for S-II with the following exceptions:

1. Pad servicing time for launch has been reduced to two hours to be consistent with the space shuttle program.

2. The engines are toed in to a 3.46-degree angle from the normal 13-degree cant angle during first stage boost, returned to a 13-degree angle for ESS engine start, and then toed in to a 6-degree cant for mainstage operation.

3. The engine nozzle is extended prior to engine start and retracted after cutoff.

4. The auxiliary hydraulic pump operates in flight (as well as on the ground) to provide hydraulic fluid to the engine.

5. LH_2 recirculation is accomplished by electrically motoring the engine LH_2 boost pump.

All other operational changes are minor and are dictated by configuration changes because of the employment of a different type of engine and the number of engines.

MPS Engine Removal

Subsequent to ESS separation from the payload, the MPS engines will be recovered by using a space shuttle orbiter. The recovery will be accomplished within the 24-hour service life of the ESS.

In order to perform the on-orbit removal of the ESS main engines. The ESS/engine connects had to be designed so that they are easily separated. This has been accomplished as shown by the engine connect layout, Figure 2-81. Separable nuts are used to effect the separation. The basic sequence of events for engine recovery are orbiter/ESS docking, ESS engine separation, engine stowage and orbiter/ESS separation.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

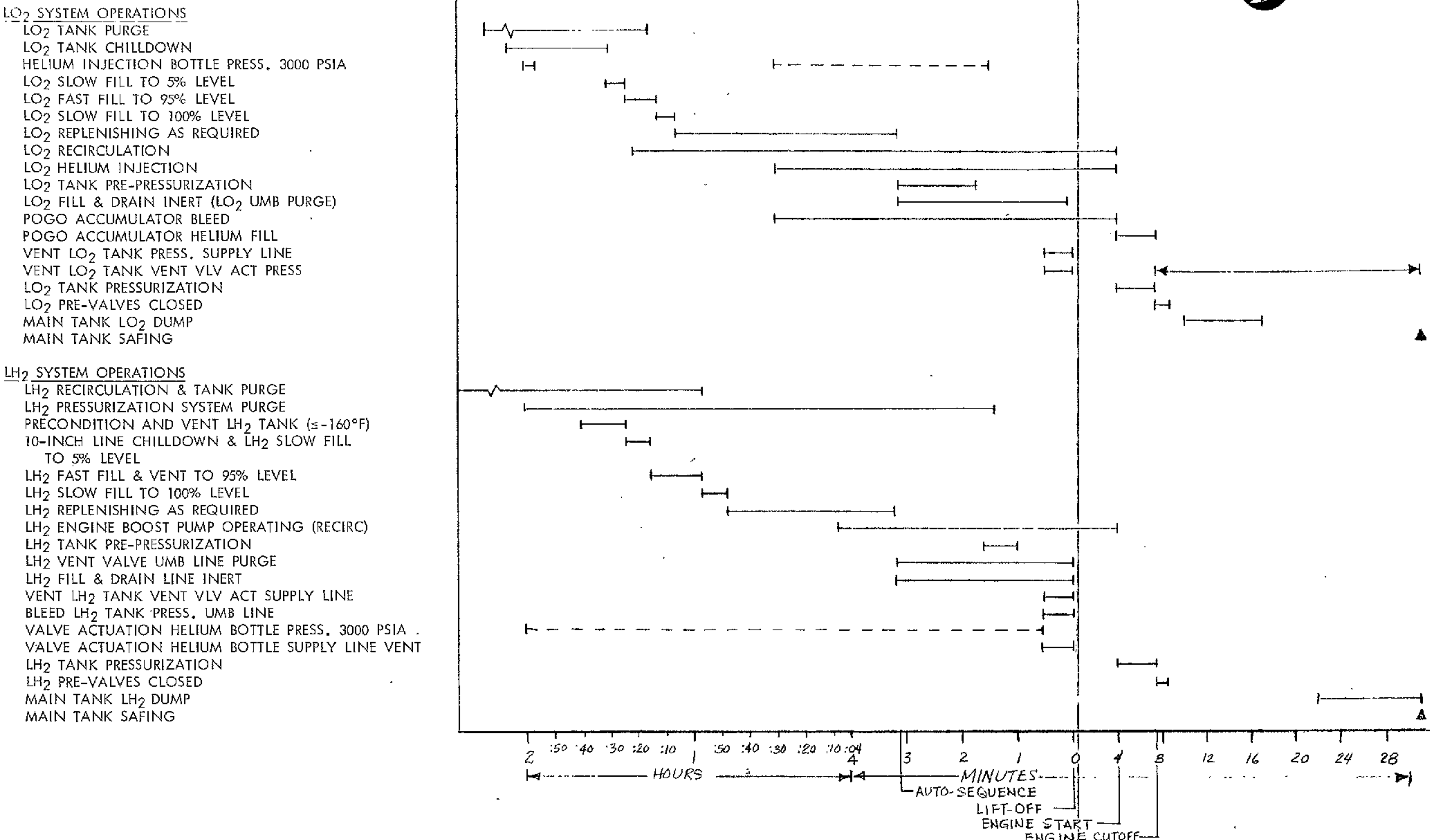


Figure 2-80. ESS Main Propulsion System Operating (Sheet 1 of 2)

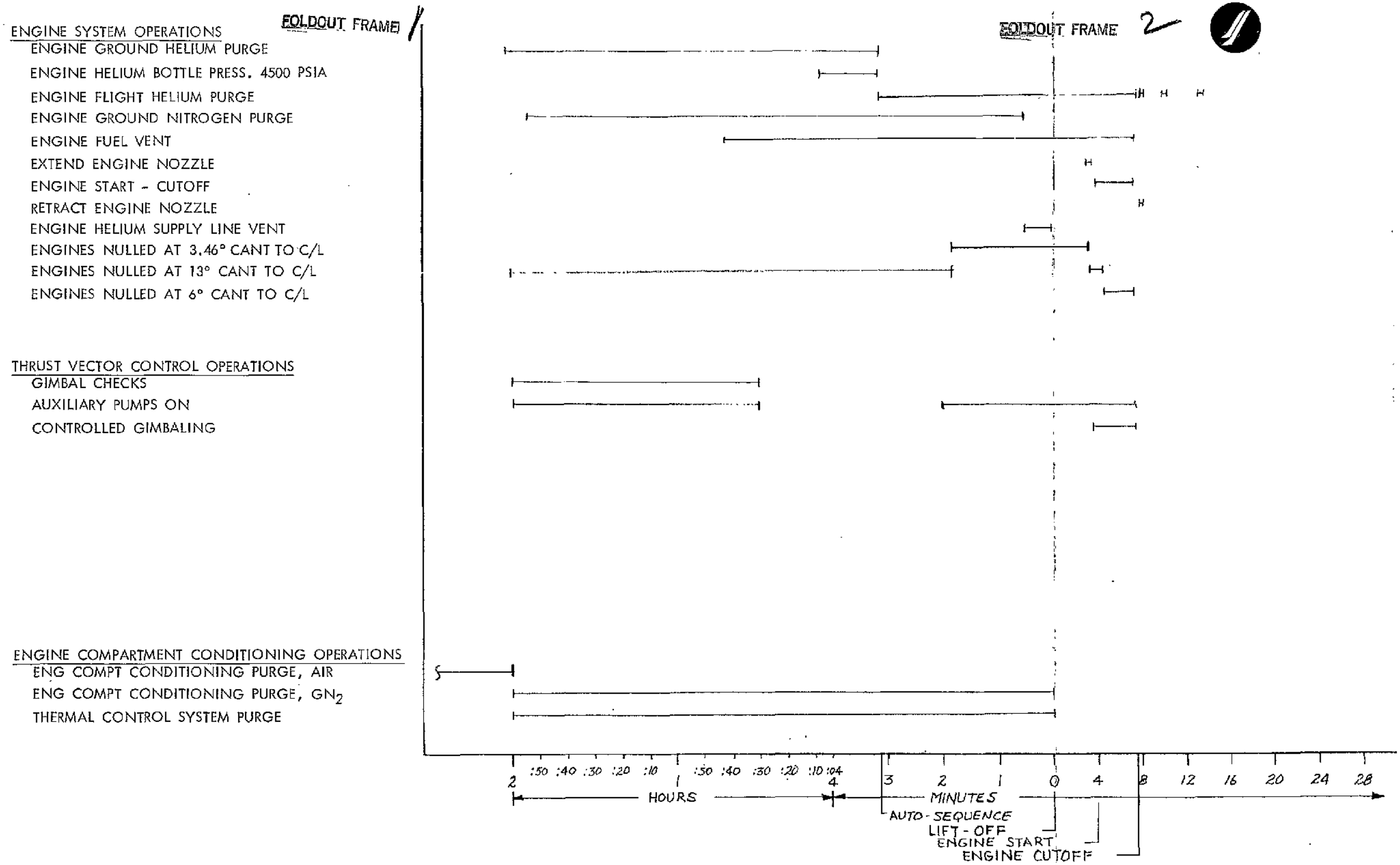


Figure 2-80. ESS Main Propulsion System Operating (Sheet 2 of 2)

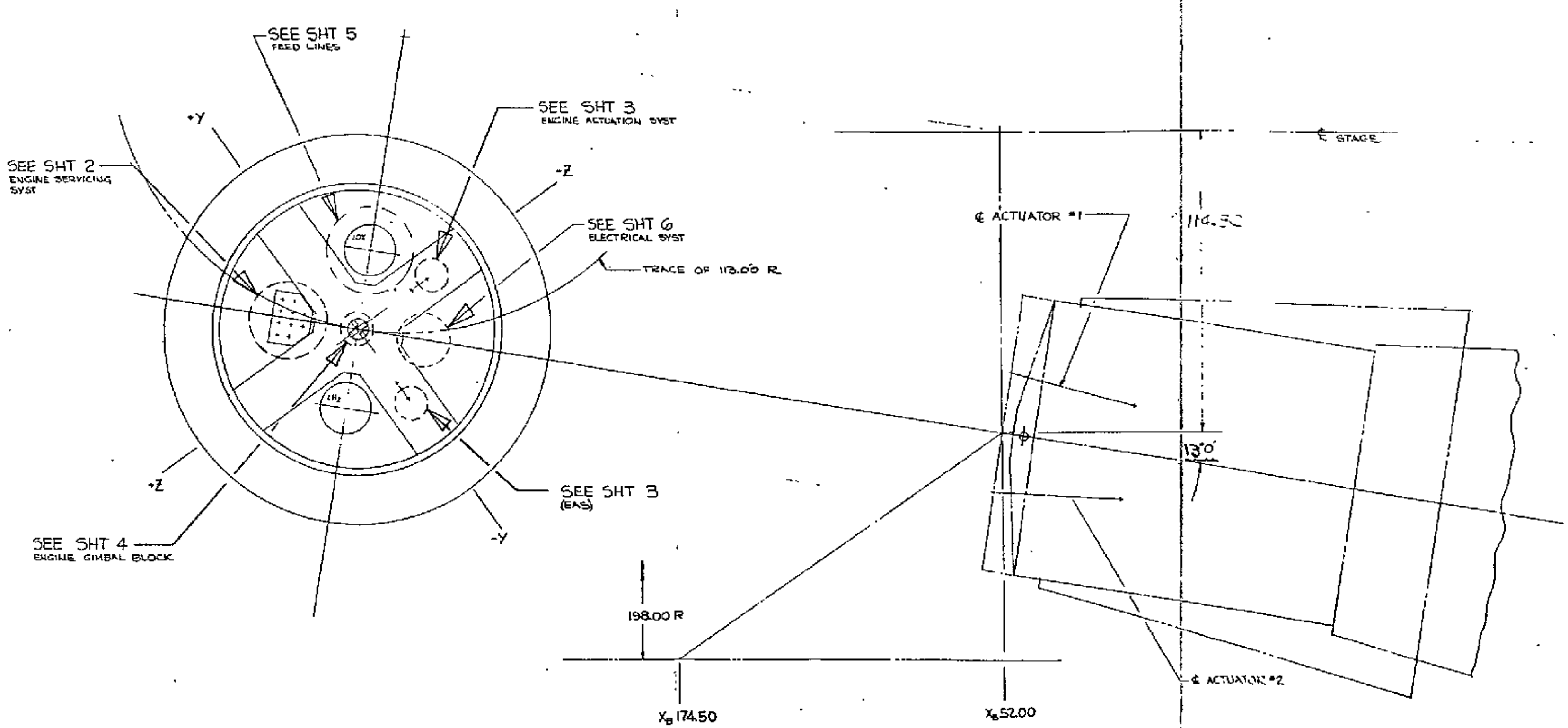


Figure 2-81. ESS Recoverable Engine Stage/Engine Connect Panel Layout
(Sheet 1 of 6)

FOLDOUT FRAME 1

FOLDOUT FRAME 2

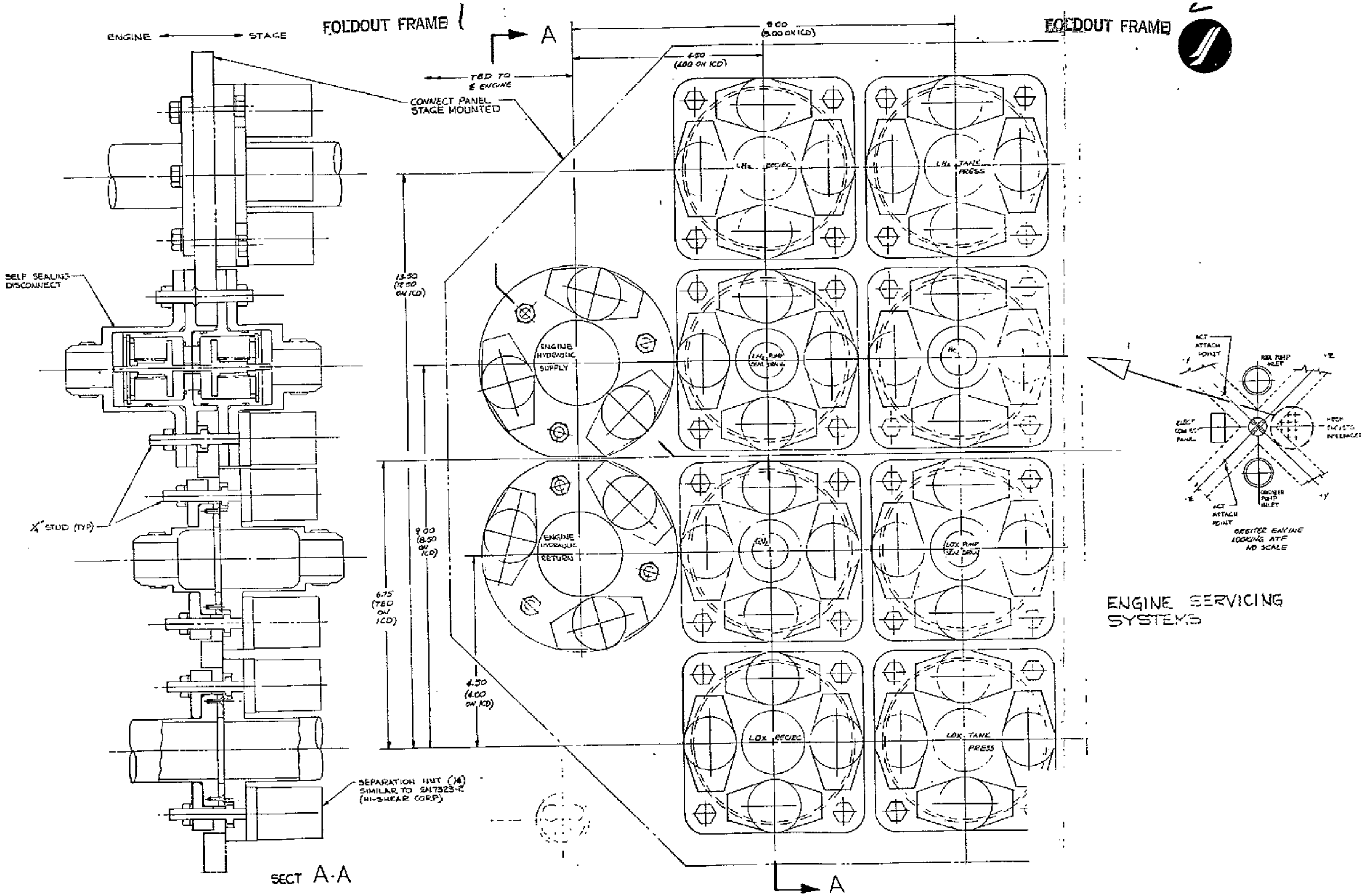
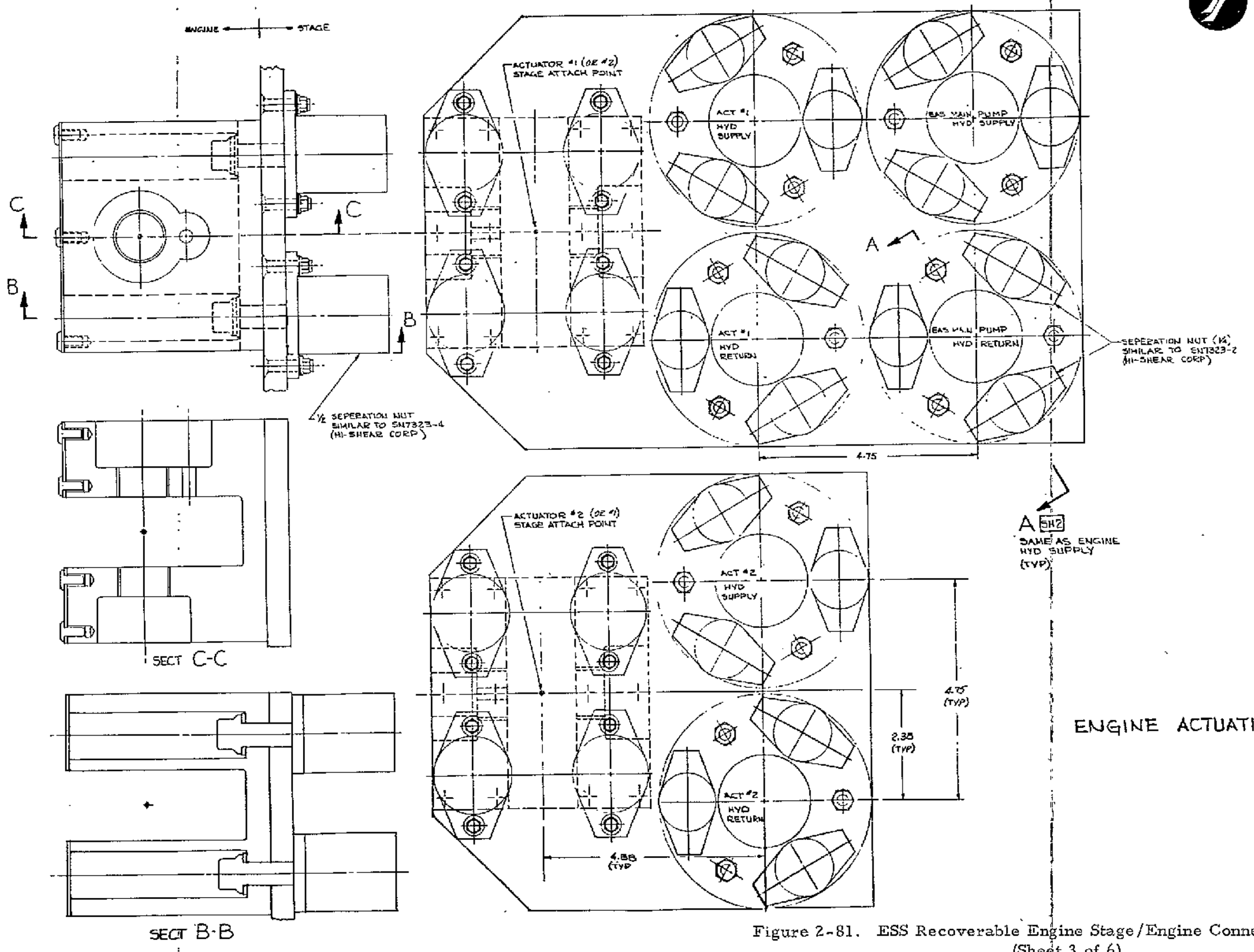


Figure 2-81. ESS Recoverable Engine Stage/Engine Connect Panel Layout (Sheet 2 of 6)



ENGINE ACTUATION SYSTEM

Figure 2-81. ESS Recoverable Engine Stage/Engine Connect Panel Layout (Sheet 3 of 6)

FOLDOUT FRAME

FOLDOUT FRAME

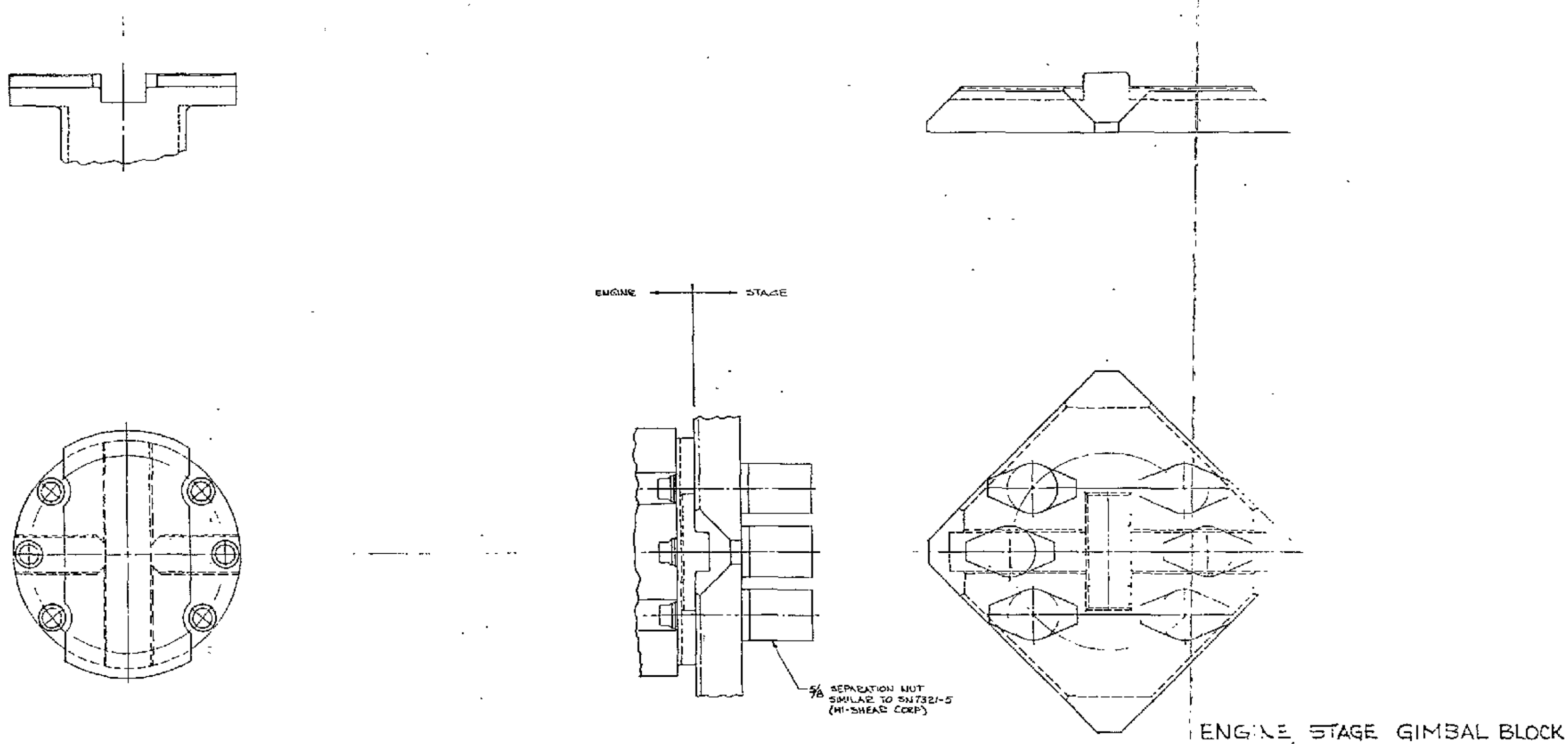


Figure 2-81. ESS Recoverable Engine Stage/Engine Connect Panel Layout
(Sheet 4 of 6)

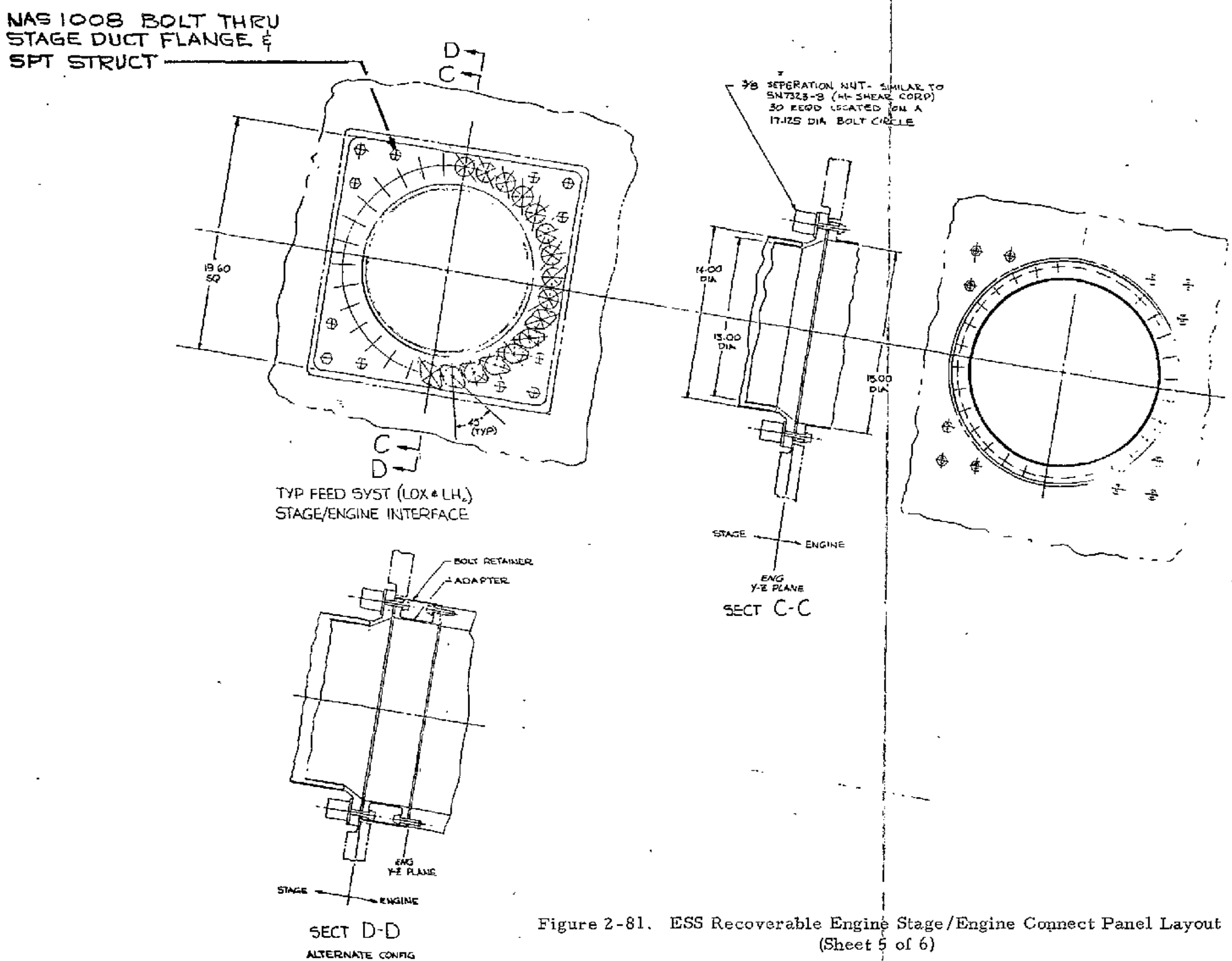


Figure 2-81. ESS Recoverable Engine Stage/Engine Connect Panel Layout (Sheet 5 of 6)

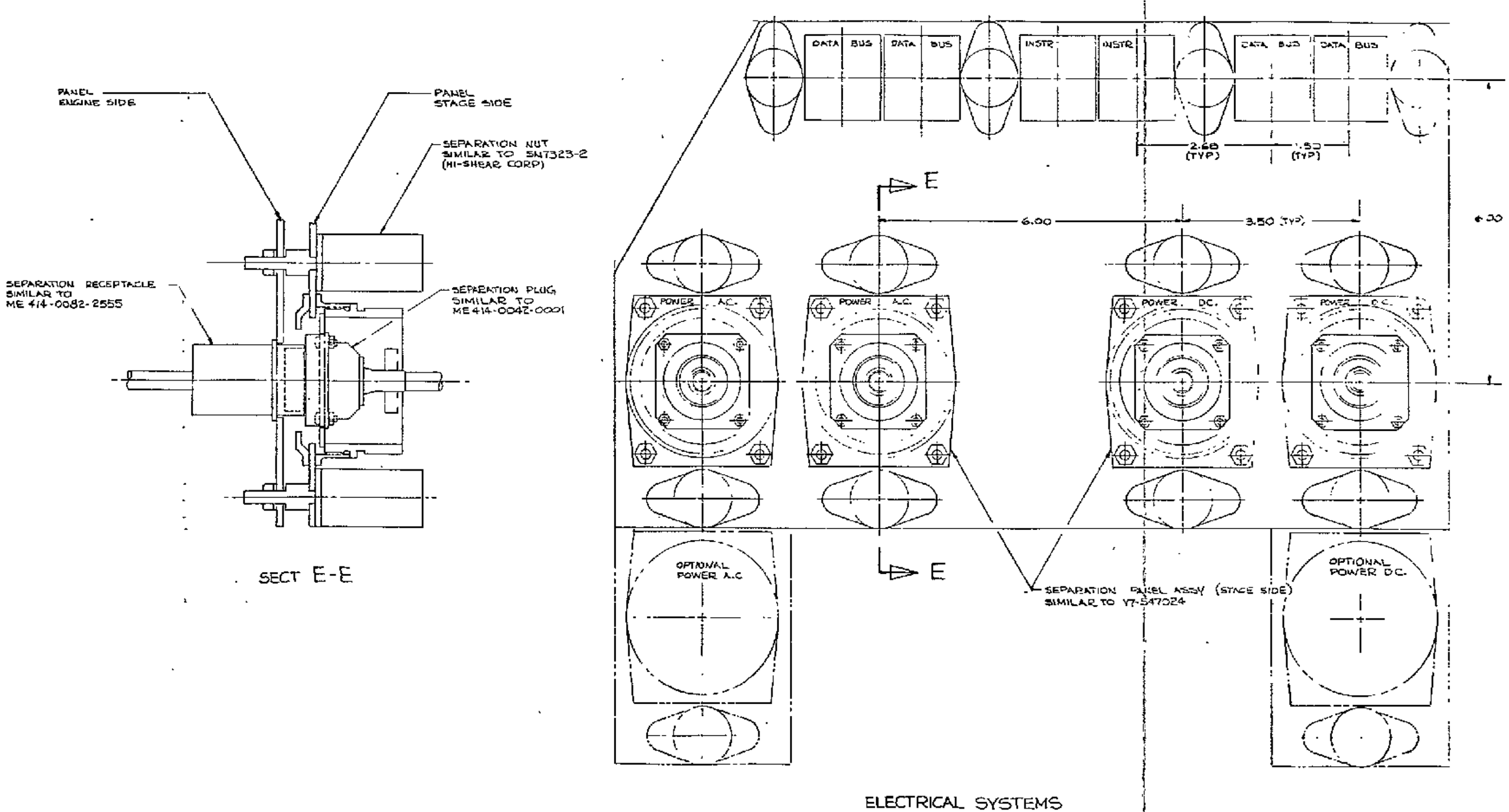


Figure 2-81. ESS Recoverable Engine Stage/Engine Connect Panel Layout (Sheet 6 of 6)

The space shuttle orbiter performs the rendezvous and docking maneuvers for orbiter/ESS docking. During the docking maneuver, the ESS will maintain attitude. A docking port is provided on the ESS at Position III. The port has the same dimensions as the port on the space station so that the orbiter-to-space-station adapter can be used for orbiter/ESS docking. The same sequence as is used for the orbiter/space station docking will be employed for the orbiter/ESS docking. The docking is shown in Figure 2-82. The sequence of docking is as follows:

1. Space Shuttle Orbiter performs near rendezvous with the ESS, the ESS maintains attitude.

2. A majority of the separable nuts connecting the engines to the ESS are actuated. The engine fluid connect, electrical connect, propellant feed ducts, hydraulic actuators, and heat shield are separated; separation is verified. Figure 2-81 shows the engine connect details.

3. The cargo doors of the orbiter are opened.

4. The docking adapter is removed from the orbiter carbo bay by using the orbiter manipulator arms. The manipulator arms are part of the baseline shuttle orbiter and are operated by the orbiter crew.

5. The docking adapter is connected to the ESS docking port.

6. The manipulator arms pull the ESS and orbiter together until the docking adapter mates with the orbiter. The vehicles are now hard-docked and attitude control will be maintained by the orbiter.

7. The manipulator arms tool is changed, and the arms are attached to the ESS engine after the heat shield has been partially removed as required for access. To attach to the engine, an attaching point on the engine is required. No attaching point is presently specified in the engine ICD. Figures 2-83 and 2-84 show proposed attaching points. Figure 2-83 shows the attaching point on the engine powerhead; Figure 2-84 shows a cantilevered bracket which puts the handling point over the engine center of gravity. Either point is acceptable for orbital recovery (zero g) of the engine but, since it is also necessary to handle the engine in earth gravity, a handling point over the center of gravity is preferable. The manipulator arm tool for engine attachment is shown in Figures 2-83 and 2-84, also.

8. The separable nuts holding the engine gimbal block are actuated.

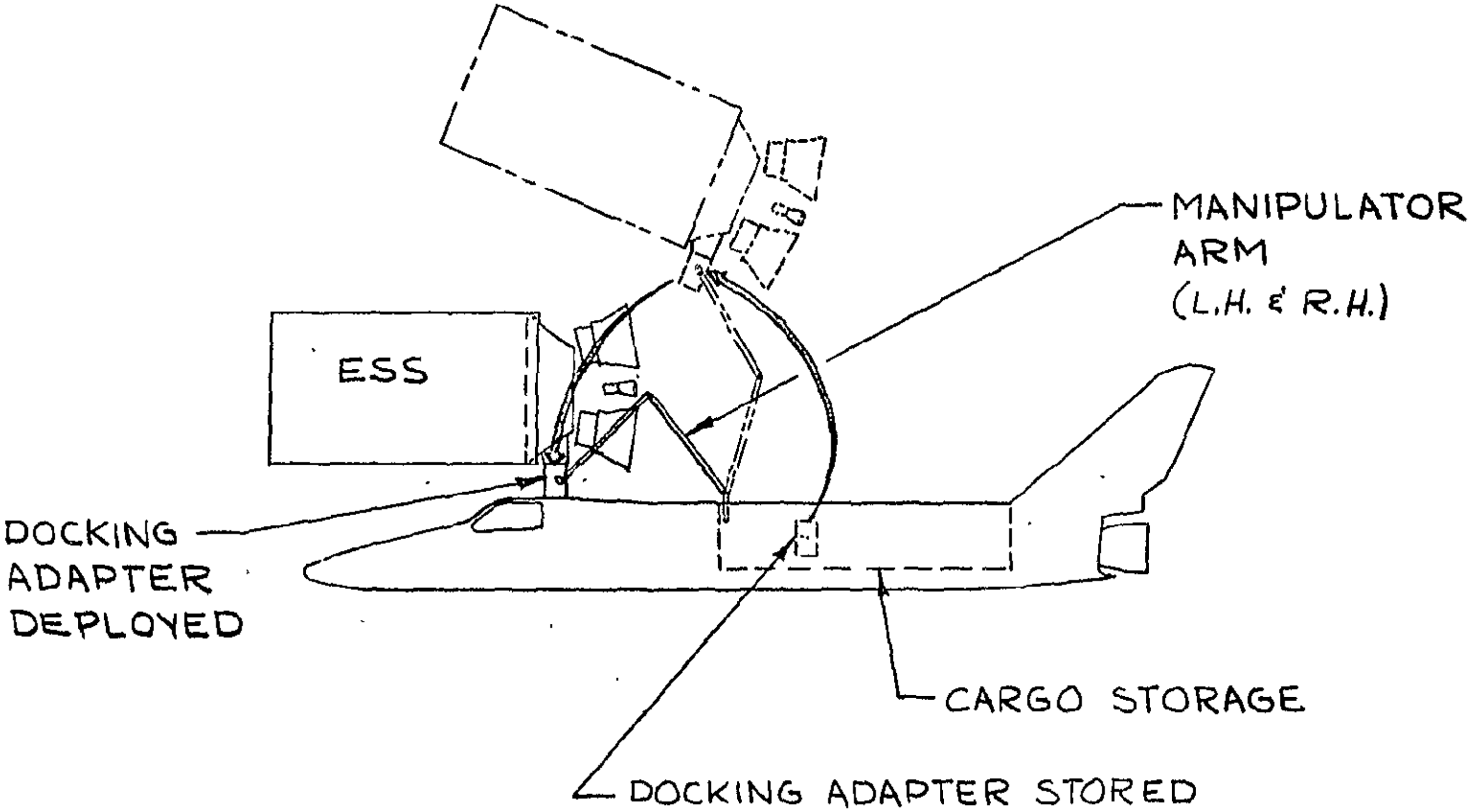


Figure 2-82. Operational Sequence for ESS Docking

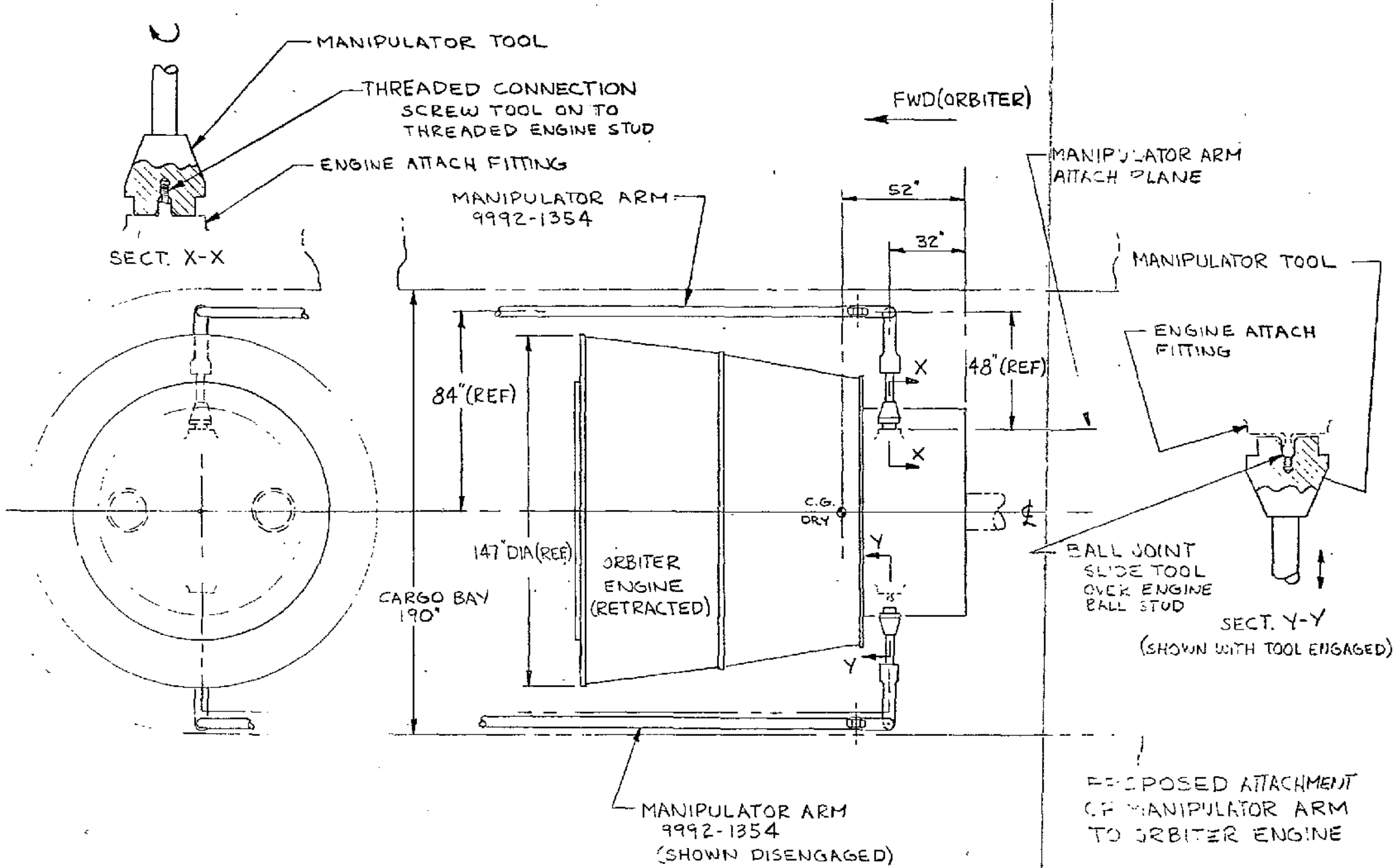


Figure 2-83. Proposed Attachment of Manipulator Arm to Orbiter Engine

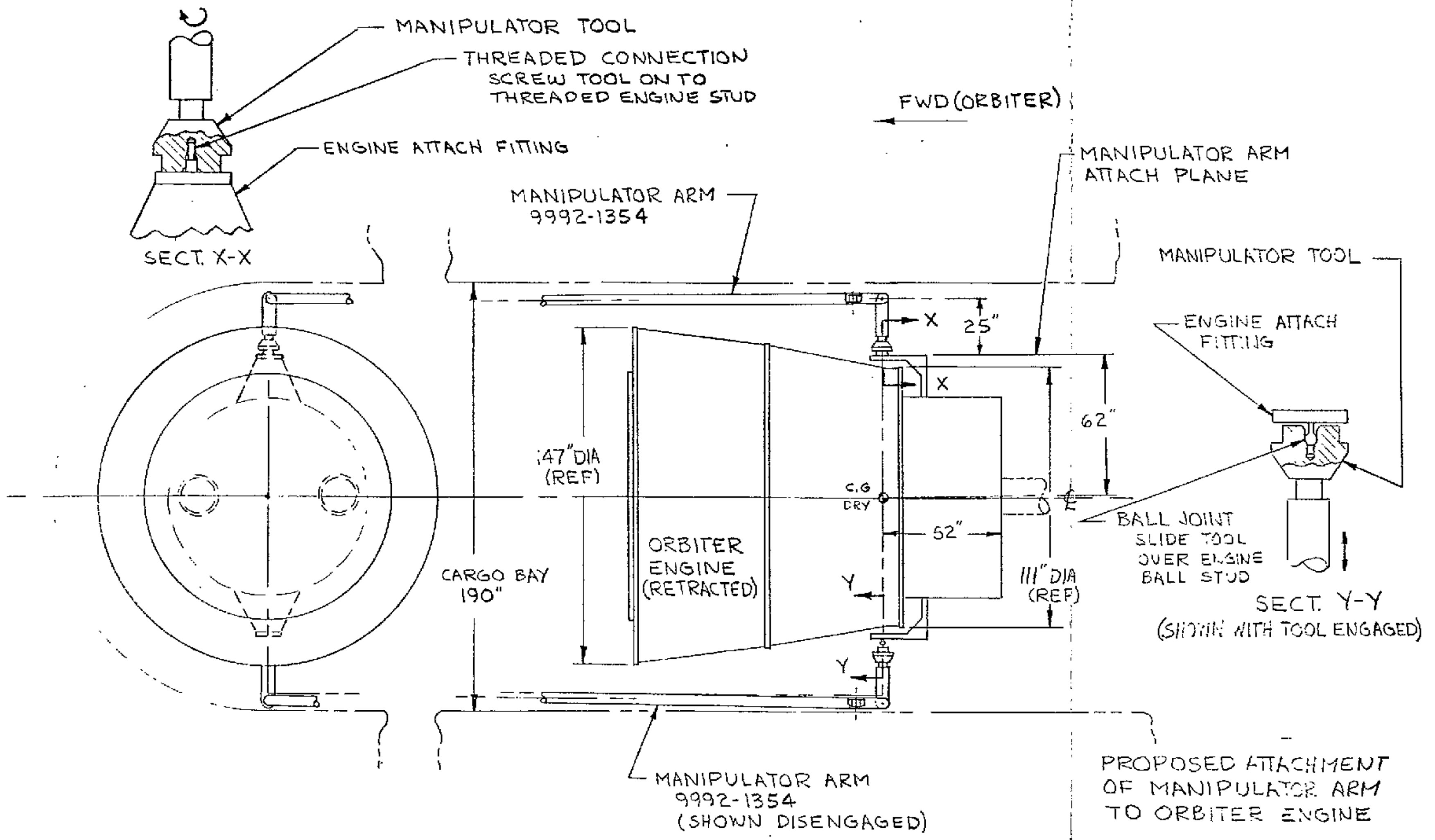


Figure 2-84. Proposed Attachment of Manipulator Arm to Orbiter Engine

The engine is now separated from the ESS and is held by the manipulator arms. The engine recovery operation is completed as follows:

1. The engine is placed in the orbiter cargo bay. Figure 2-85 pictures an SSEo vertical in the orbiter cargo bay. As can be seen, an interference results. Therefore, the SSEo will be stowed horizontally (Figure 2-86). A storage rack will be provided.

2. The manipulator arms tool is changed and the arms are attached to the docking adapter.

3. The ESS and orbiter are separated. The ESS will now once again provide its own attitude control.

4. The docking adapter is placed in the orbiter cargo bay, the manipulator arms are stowed, and the orbiter cargo bay doors are closed.

The engine recovery operation is now complete except for orbiter return utilizing the standard space shuttle orbiter flyback. Figure 2-87 shows the proposed recoverable hardware. The orientation shown in zones 11 through 15 of sheet 2 is the selected recovery mode. As can be seen, the existing shuttle orbiter manipulator arms (600-inch) are satisfactory for the recovery operation; more than 27 feet of the orbiter cargo bay is available for other shuttle cargo after the engines and docking adapter are stowed. (For details on avionics recovery, refer to Section 2.3.8 of this report. Avionics recovery is accomplished at the same time as engine recovery.)

Air Loads on ESS Engines During Boost

The two main propulsion engines on the ESS configuration are carried with nozzle extensions in retracted position during boost. In addition, the nozzles are protected from impingement by airstream deflectors which extend 71 inches aft of the base plane with a flare angle of 15 degrees. (Figure 2-88).

An analysis has been made to predict the occurrence or nonoccurrence of impingement during boost and to estimate the magnitudes of nozzle forces and moments about the gimbal point.

The amount of nozzle surface exposed to impingement depends upon vehicle angle of attack and the detailed geometry of the airstream deflectors and nozzle. The magnitude of the pressure loads will be affected by the turning angles associated with separation and reattachment of the flow. An analytical method has been developed by which numerical estimates of nozzle

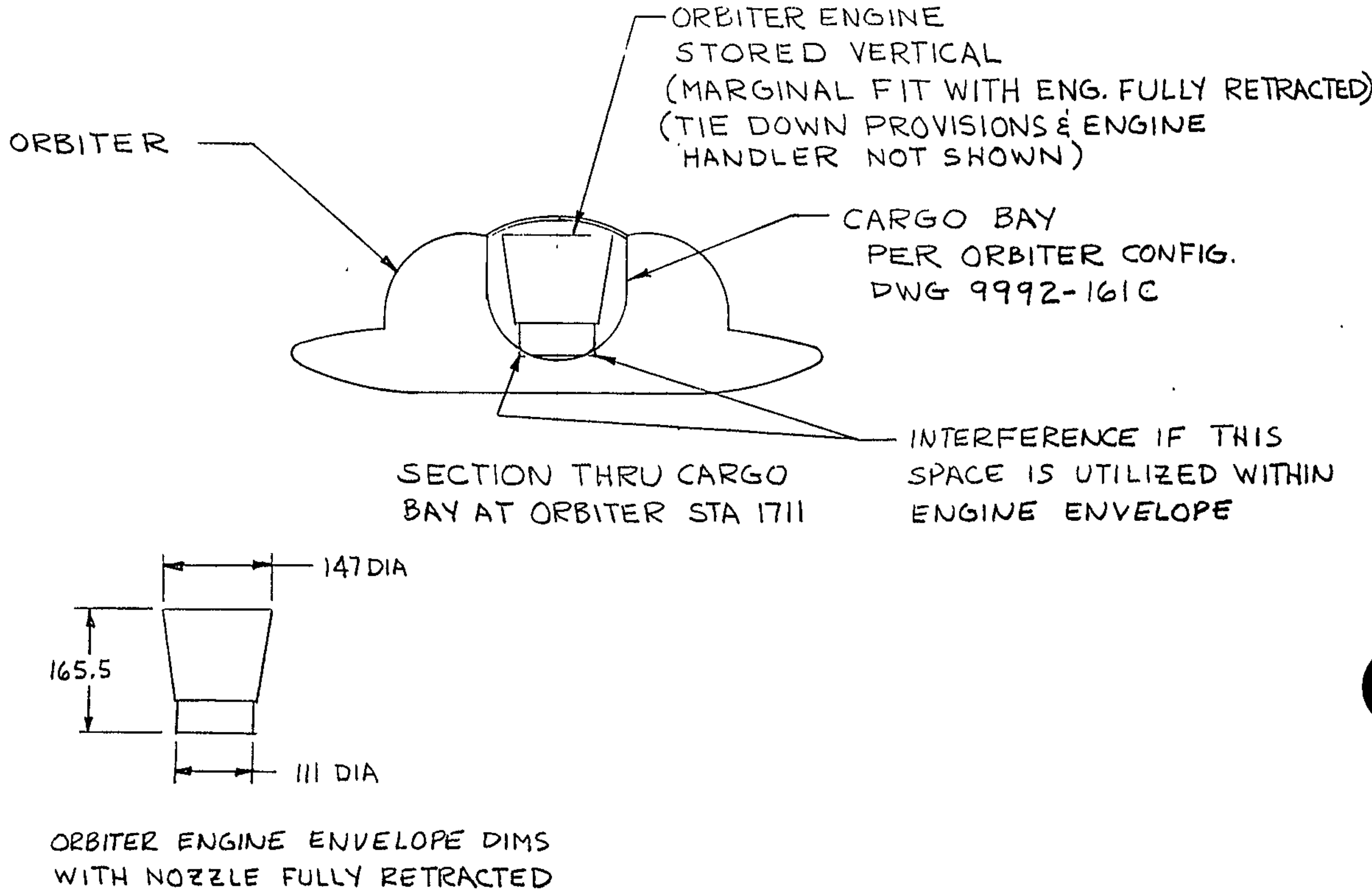


Figure 2-85. ESS Orbiter Engines Stored Vertical

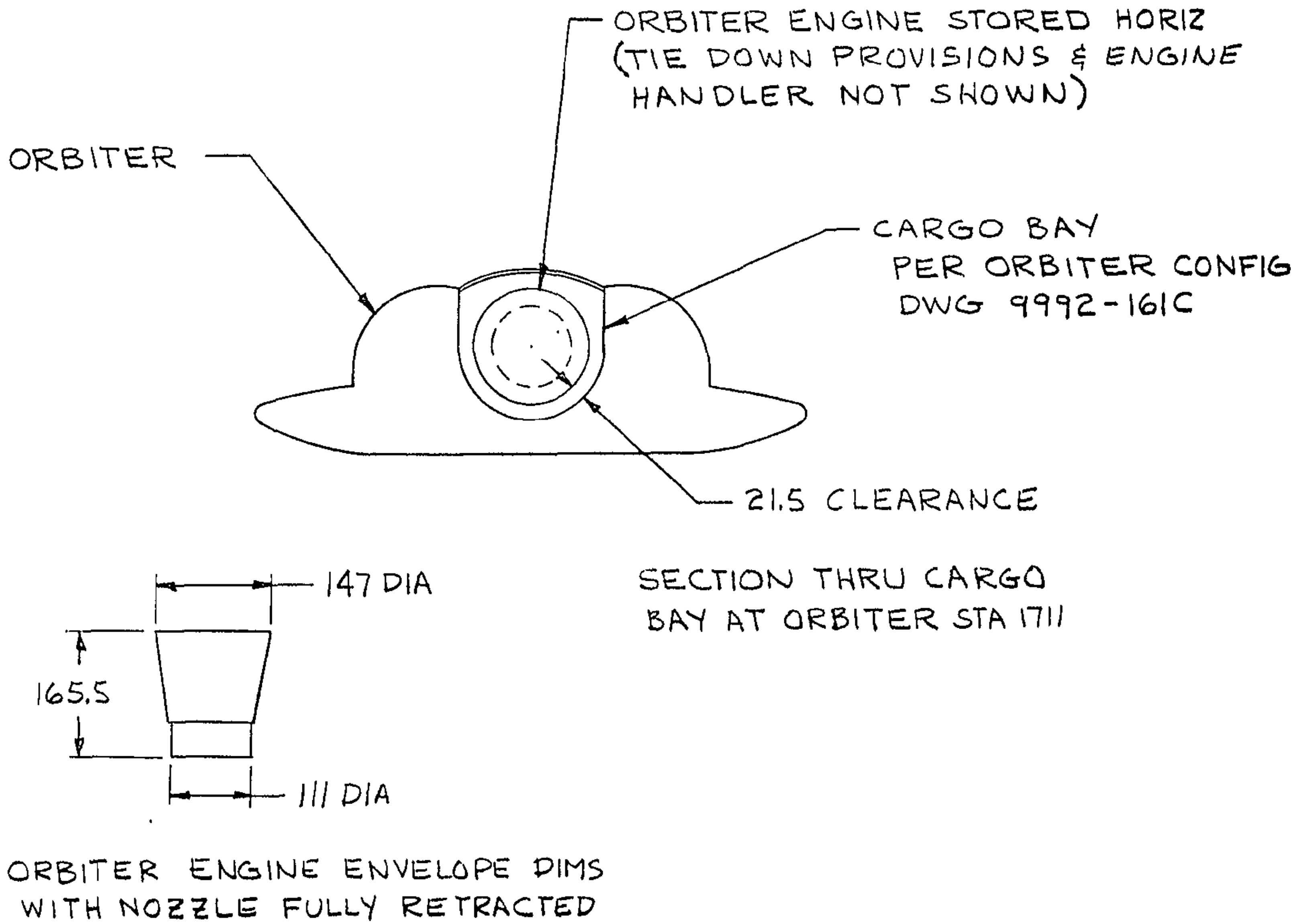


Figure 2-86. ESS Orbiter Engines Stored Horizontal

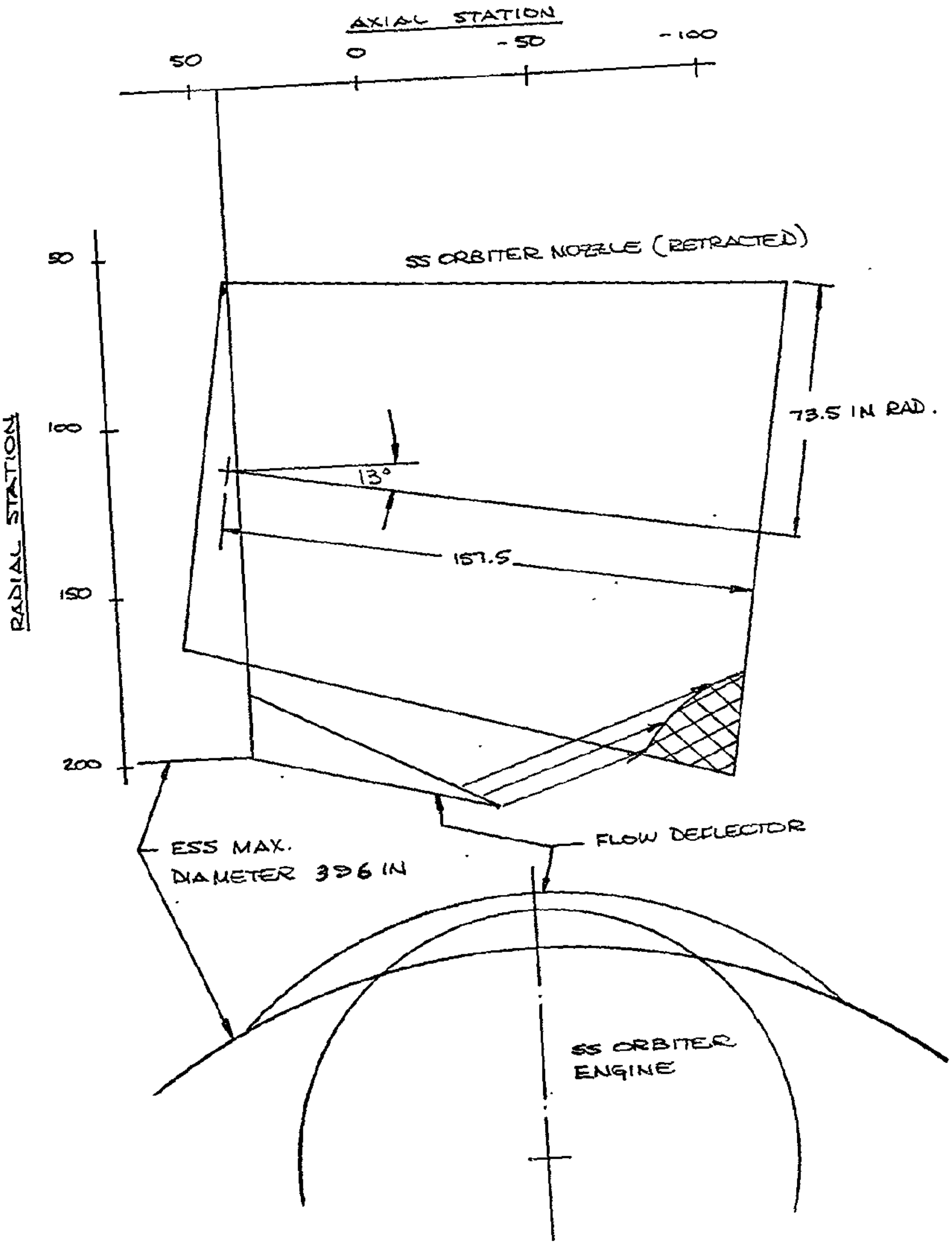


Figure 2-88. Impingement of Separated Flow on ESS Nozzles

loads and moments can be calculated for engine configurations with airstream deflectors of the type described. The principal restriction is that vehicle angle of attack and engine deflection angle must lie in the same plane (the pitch plane).

The base pressure and the Mach number of the separated flow are computed from the Prandtl-Meyer expansion associated with the turning angle ϕ (Figure 2-89). The turning angle is evaluated as a function of free-stream Mach number. The differential pressure acting over the impingement zone is evaluated in terms of a pressure coefficient based on linearized theory, as follows:

$$C_p = \frac{P_I - P_B}{1/2\ \gamma\ P_B M_B^{\ 2}} = \frac{2(\lambda + \theta)}{\sqrt{M_B^{\ 2} - 1}}$$

Here the angle $(\lambda + \theta)$ is the turning angle of reattachment expressed in radians, and P_I is the absolute pressure after impingement.

The same procedure was applied to the prediction of loads on the ESS nozzles in the retracted position. The results for a representative boost trajectory are listed in Table 2-9.

The potential engine contractors indicated that a maximum engine nozzle force of 300 pounds is an acceptable limit without redesign. To alleviate any design changes, three configurational changes have been considered which would prevent any aerodynamic impingement:

1. Extending the length of the airstream deflector 23.3 inches, keeping the flare angle constant at 15 degrees.

2. Terminating the airstream deflector at the same axial station but increasing the flare angle to 23.7 degrees.

3. "Toeing-in" the engines to a 3.46-degree angle to the centerline.

Item 3 was selected because of the minimum impact on the proposed ESS vehicle. The "toe-in" of engines would be accomplished by an electrical signal prior to lift-off.

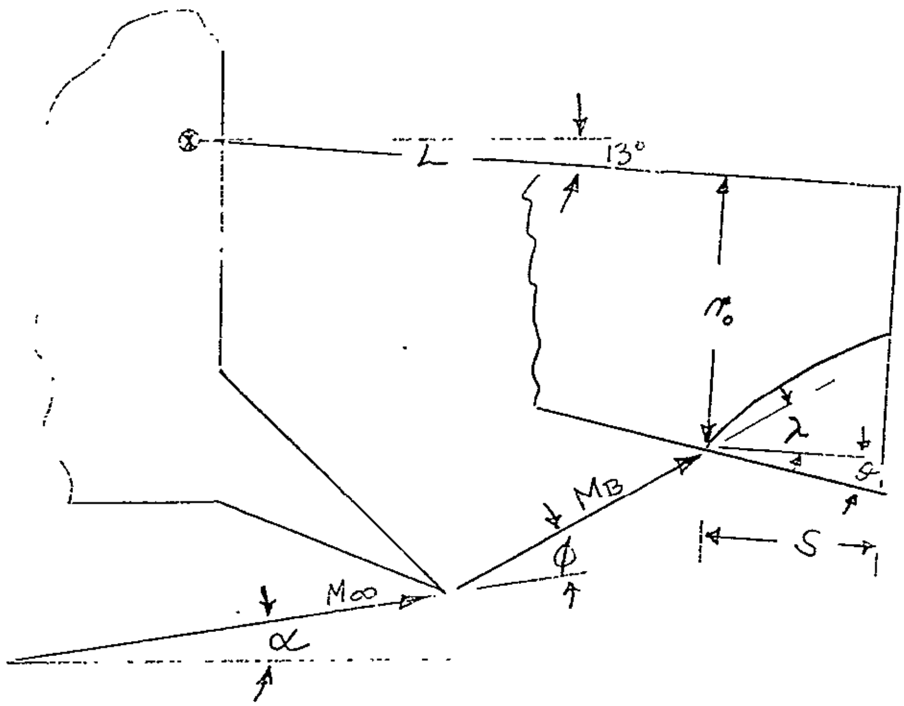


Figure 2-89. Nozzle Impingement Geometry

Table 2-9. Impingement Loads and Moments for the ESS Nozzle

Loads	Trajectory Time (sec)					
	80	90	100	110	130	150
M_{00}	1.372	1.773	2.216	2.666	3.771	5.000
q_{∞} (psf)	649.2	660.4	582.6	463.3	277.9	153.1
α (deg)	0	0.125	0.25	0.375	0.5833	0.75
$P_I - P_B$ (psi)	2.758	1.818	1.083	0.600	0.162	0.0437
S (in.)	6.76	14.29	18.69	21.08	23.75	23.69
F_N (lb-ft)	489.0	1050.0	973.0	660.0	219.0	58.8
M_G (in.-lb-ft)	75.7 $\times 10^3$	159.3 $\times 10^3$	145.9 $\times 10^3$	98.4 $\times 10^3$	32.4 $\times 10^3$	8.7 $\times 10^3$

The integrated pressure force acting on the nozzle in a direction perpendicular to the axis and the corresponding moment about the gimbal plane have been found to be

$$F_N = \frac{4}{3} \left(P_I - P_B\right) \sqrt{2 \tau_o (\tan \theta + \tan \lambda)} \; S^{3/2}$$

and

$$M_G = \left(L - \frac{2}{5} S\right) F_N \text{ , respectively.}$$

The equations have been verified by applying them to S-IC model test data. The moments were found to agree well with model test data over the Mach number range 1.6 to 2.4 and for angles of attack as great as 10 degrees.

2.2.2 Auxiliary Propulsion Subsystem (APS)

The auxiliary propulsion subsystem is divided into two basic elements—orbit maneuvering and attitude control. Two 10,000-pound thrust hydrogen/oxygen engines are utilized for orbit maneuvering. The engines will be developed for the space shuttle orbiter. Attitude control is accomplished with 14 2100-lb thrust hydrogen/oxygen thrusters. The thrusters will be developed for the space shuttle orbiter and booster. The two elements are integrated in that they have common propellant tankage and pumps. The auxiliary propulsion subsystem is shown schematically in Figure 2-90.

Attitude Control Propulsion Subsystem (ACPS)

ACPS Design Requirements. The design requirements for the ACPS are set primarily by flight stability and control requirements. The basic requirements may be summarized as follows:

1. The subsystem must provide pitch, yaw, and roll attitude control during the 24 hour ESS on-orbit service life. Initial estimates indicate that 300,000 pound-seconds of impulse are required.

2. The subsystem must provide roll control in the event of a one main engine or OMS engine out and flight control for a one OMS engine out condition during deorbit. It is this requirement that dictates a total of eight roll control nozzles.

3. The subsystem is to provide translation thrust during rendezvous. A delta velocity of 24 feet per second (braking) is necessary.

4. The system must be designed to meet fail-operational/fail-safe criteria since attitude control is required when the shuttle orbiter is docking with the ESS.

ACPS General Description. The ACPS utilizes a total of 14 thrusters to fulfill its design requirements. The thrusters are mounted seven at each position (I and III). The exact mounting locations are shown by Figure 2-91. The thrusters perform the following primary functions:

1. Eight thrusters provide roll and pitch control: thrusters 1, 2, 4, and 7 at Position I and 8, 9, 11, and 14 at Position III.

2. Two thrusters provide yaw control: thruster 5 at Position I and thruster 12 at Position III.

3. Two thrusters provide braking thrust: thruster 3 at Position I and thruster 10 at Position III.

4. Two thrusters provide longitudinal translation: thruster 6 at Position I and thruster 13 at Position III.

The thrusters are mounted under the main engine LH_2 feed line fairing. The thruster exit planes are essentially flush with the fairing for those thrusters located aft of the feed line. Thruster skirt extensions will be provided wherever needed. However, the braking thrusters (3 and 10), which have the exit forward, must be completely covered during first-stage boost. A thruster door, which will be opened once the ESS is on orbit, is provided, therefore. The door will be designed so that, if it does not actuate open when commanded, the exhaust from the thruster will open the door.

The 2100-pound thrust shuttle attitude control thrusters were selected for ESS because of the large thrust required for roll control when one main engine is out.

The characteristics of the thrusters are shown in Table 2-10. The overall subsystem performance differs from the basic thruster performance because it includes the propellant for the gas generators which drive the pumps and condition the propellants for ACPS use. The thrusters burn high-pressure gaseous oxygen and hydrogen while the propellant is stored as liquid at low pressure. Factoring in the GG propellant causes the system specific impulse and mixture ratio to be lower than the basic thruster values.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

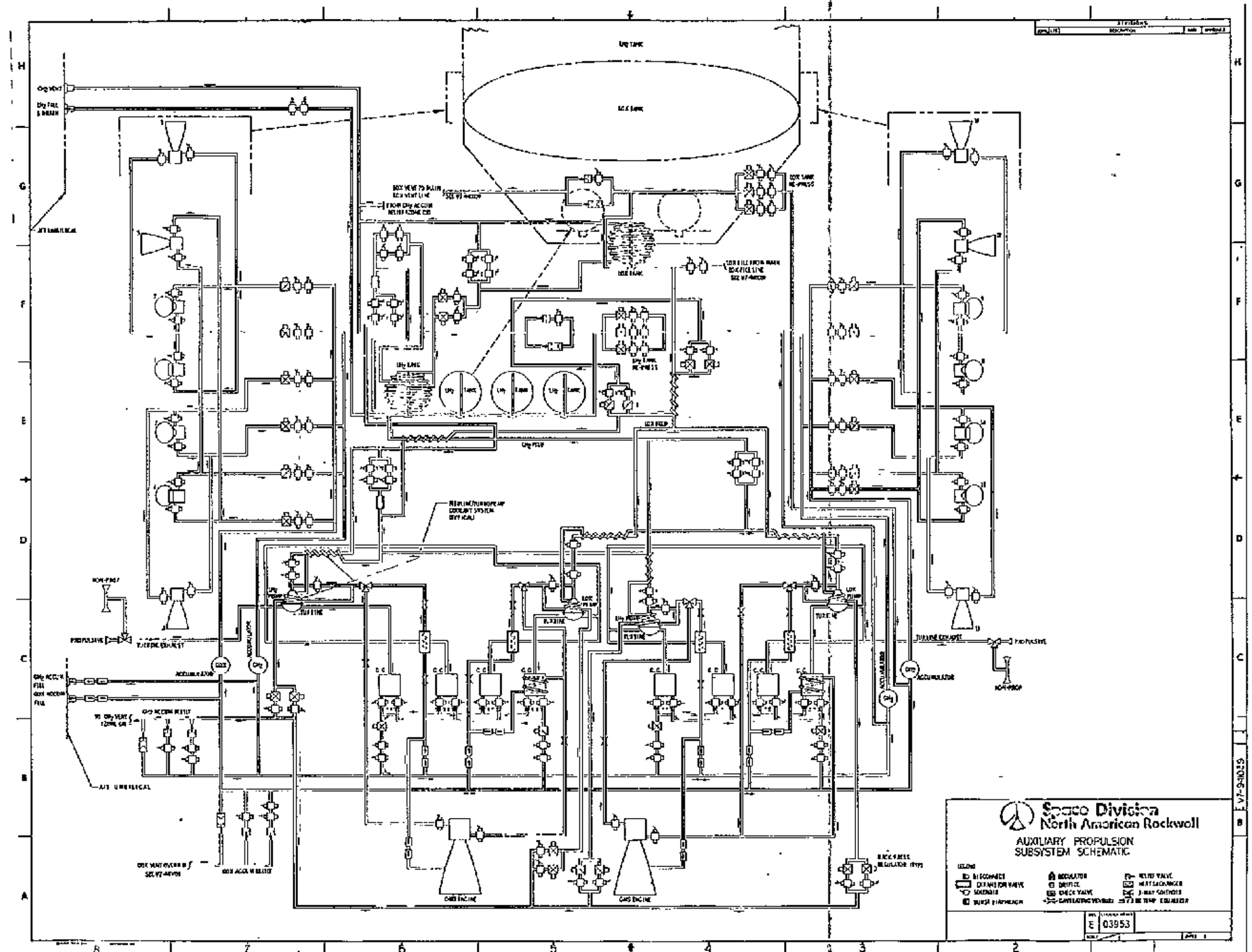


Figure 2-90. Auxiliary Propulsion Subsystem Schematic

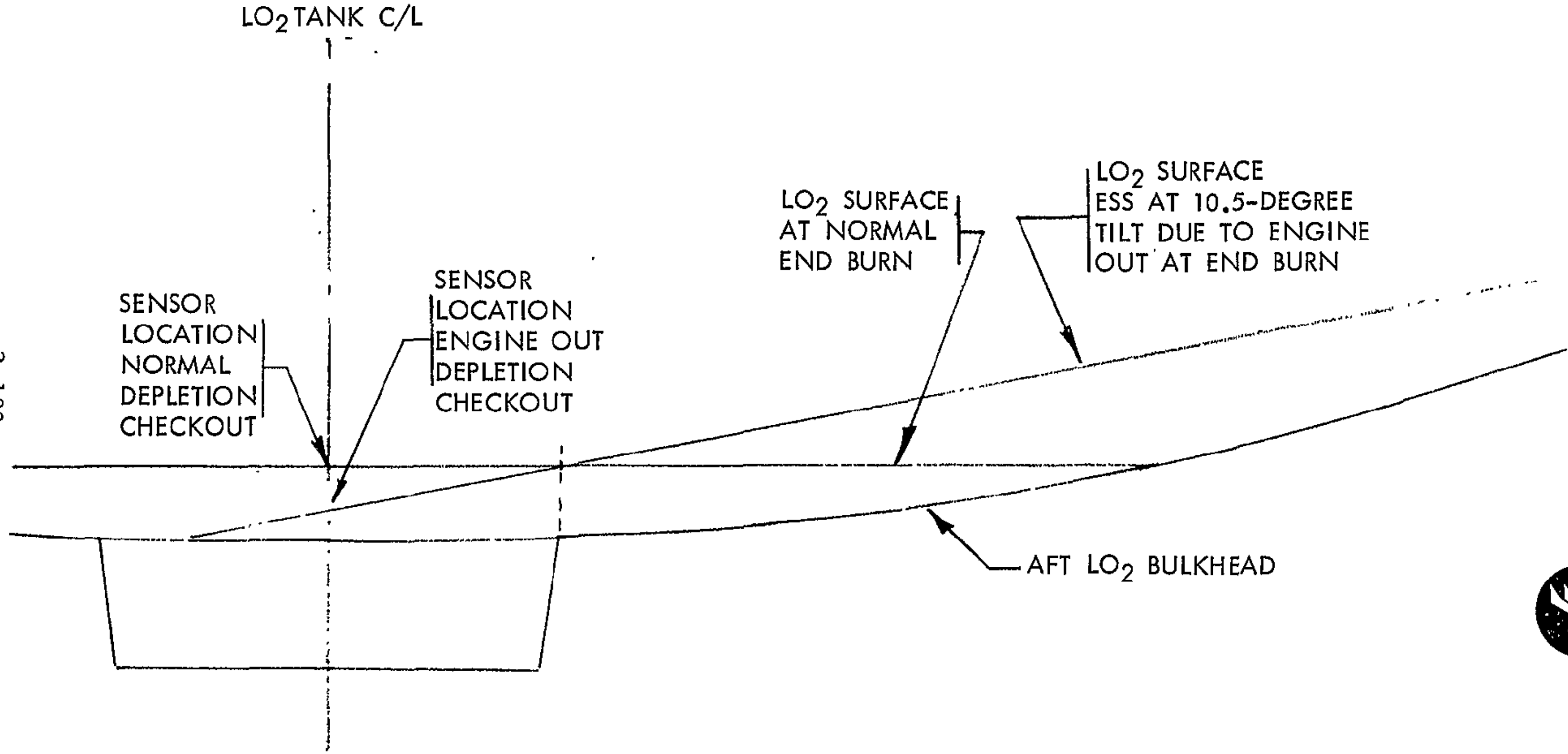


Figure 2-57. LO_2 Residual at Depletion Cutoff, Normal Depletion and Engine Out

the LO_2 tank via separate uninsulated return lines. Shutoff valves are provided at the propellant tanks in both the LO_2 and LH_2 recirculation return lines to provide emergency tank isolation. The S-II recirculation valves will be used for this application. LH_2 recirculation will be initiated approximately ten minutes prior to engine start; LO_2 recirculation commences with LO_2 tanking. Helium injection to augment LO_2 recirculation begins 30 minutes prior to liftoff.

Valve actuation is required for actuating the propellant valves for such ground operations as checkout and purging, and in flight for providing tank isolation and a propellant dump. All the stage propellant valves are pneumatically actuated, and the actuation pressure is supplied by a stage helium bottle and regulator. The valve actuation subassembly is shown in Figure 2-59, the pressurization layout.

Pogo-suppression is provided by supplying accumulators at the engine LO_2 pump inlets. The accumulator decouples the feedline and limits the interaction of structure, feedline, and engine. Based on the assumption that the ESS structural modes are about the same as on the S-II (6 to 10 Hz), a preliminary analysis indicates that a 430-cubic-inch accumulator at the LO_2 pump inlet will minimize pogo susceptibility by reducing the first feedline mode to a point below 6 Hz. It is felt that the probability of a local high gain mode similar to the 16-Hz oscillations that occurred on S-II is quite small because of the absence of the center engine beam. The proposed accumulator is shown in Figure 2-56 along with growth potential to increase the accumulator size to 600 cubic inches if subsequent analysis shows this to be necessary. The accumulator size cannot be considered definite because of the preliminary nature of the ESS design, but it may be concluded that pogo susceptibility may be eliminated by installing feedline accumulators which will be filled from the same helium supply employed for LO_2 recirculation helium injection and bled through the LO_2 recirculation return lines. No analysis was attempted of pogo on the hydrogen side, the decision based primarily on the fact that pogo has never been a problem with LH_2. Further, ESS pogo analysis will be accomplished by Task Authorization 13, Change Order 2026 to Contract NAS7-200.

Main propellant tank fill and drain will be accomplished by utilizing the S-II fill and drain subassembly. Since the subassembly is essentially identical with the S-II subassembly, no layout has been made. The capability of the fill and drain is such that with the maximum planned ESS loading the tanks can be filled within the required two-hour servicing time. The fill will be controlled by the tank capacitance probes, with point sensors as backup. Some new point sensors will be required because of the different propellant loadings. The tanks can be drained in 30 minutes with the maximum propellant load.

Table 2-10. ACPS Thruster Characteristics

Parameter	Basic Thruster	Subsystem
1. Thrust	2100 lb	2100 lb
2. Mixture ratio, O/F	4 to 1	3 to 1
3. Specific impulse	425 sec	370 sec
4. Chamber pressure	300 psia	300 psia
5. Expansion ratio	20	N/A
6. Exit diameter	11.4 in.	N/A
7. Overall length	26 in.	N/A
8. Weight	44 lb	N/A
9. Propellant inlet diameter	1 in.	N/A
10. Minimum impulse bit	210 lb-sec	210 lb-sec

To minimize heating of the ESS skin, the thrusters are mounted at a 20-degree angle. Propellant for the ACPS is stored in the same tanks as is the orbit maneuvering subsystem propellant. The discussion of the ACPS propellant budgeting is included in the OMS discussion.

A fail-operational/fail-safe (FO/FS) analysis was made of the ACPS. The results are documented in Table 2-11. The subsystem has been designed to FO/FS criteria with one exception. This provision is due to having only two turbo pump and gas generator packages when actually three are required to meet the FO/FS standard. For a discussion of the rationale for not meeting the FO/FS standard completely, refer to the orbit maneuvering subsystem. For engine-out conditions, the FO/FS considerations are:

1. One main engine out. For an engine-out condition, the engine loss is the first failure. The vehicle remains operational as the four ACPS thrusters available for roll control provide complete control. If a roll thruster goes out the vehicle is still safe because the effect will be for the vehicle to roll a few degrees and then roll back.

Table 2-11. ACPS Fail-Operational/Fail-Safe Considerations

Failure	Remarks and Rationale
Pitch/roll thrusters (1, 2, 4, 7, 8, 9, 11, and 14)	Four thrusters are provided for pitch/roll in each direction. Two thrusters will provide adequate control.
Yaw thruster (5 and 12)	If thruster fails, use braking thruster for yaw (FO). If braking thruster fails, roll vehicle and use pitch thruster (FS).
Braking thruster (3 and 10)	If first thruster fails, one braking thruster will provide adequate braking. If second thruster fails, stage can be rotated and OMS or translation thrusters can be used to provide fail-safe mode.
Translation thruster (6 and 13)	If first thruster fails, one translation thruster is adequate (FO). If second thruster fails, rotate stage and use braking thrusters.
Thruster manifold inlet regulator	After regulator fails, the manifold inlet valve is closed. If operational valve fails, a redundant valve is provided (FO). Thrusters are manifolded so that the backup thrusters and the primary thruster are on different manifolds.
Accumulator fill check valve	If first check valve fails, a redundant check valve is provided. If second check valve fails, disconnect provides shutoff (FO).
Accumulator relief valve	When the primary relief valve fails, switch to backup relief (FO). If the second failure is a primary relief isolation, a redundant isolation is provided. If the second failure is a backup relief or its isolation valve, a burst diaphragm and relief valve are provided and the system is still operational.
Thermal conditioning gas generator (GG)	If first GG fails, a second GG maintains the system operational (two complete pump and GG sets are provided). If first failure is GOX-conditioning GG and second is GH_2-conditioning GG (or vice versa), system is still operational. If both GOX-conditioning GG's or both GH_2-conditioning GG's fail, the ACPS is inoperative. <u>This condition does not meet FO/FS requirements since attitude control is required for fail-safe</u>.
Gas generator inlet valves, regulators, select valves, check valves	The various system valves are installed (with applicable redundancy) as required to support the thermal conditioning GG's. For this analysis, they may be considered a part of the GG's, and the same considerations apply. For instance, if the oxygen inlet valve to both GH_2 conditioning GG's fails, the system is inoperative.
Pumps, pump drive GG's, etc.	Refer to orbit maneuvering subsystem FO/FS considerations.

2. One orbit-maneuvering engine out. With an OMS (orbit-maneuvering subsystem) engine out, two roll control thrusters are required for the most severe case (deorbit). Full redundancy is available since there are four thrusters for roll control.

ACPS Operation. The attitude control propulsion subsystem for the ESS will operate essentially the same as the space shuttle attitude control propulsion subsystem.

Prior to lift-off, liquid propellants are contained in the auxiliary propulsion subsystem tanks and the ACPS accumulators are charged with GOX and GH_2, which also pressurizes the propellant tanks.

The attitude-control thrusters burn gas from the accumulators. Redundant spark plugs provide ignition. The accumulators are initially pressurized to 1000 psi. As the thrusters burn gas from the accumulators, the accumulator pressure decreases until a pressure switch drops out signaling the pumps to start to recharge the accumulators. A typical accumulator recharge is shown in Figure 2-92. When the accumulators are being charged, all eight gas generators shown in Figure 2-90 are operative. Four of the gas generators drive the turbo pumps and four provide hot gas to the heat exchangers which condition the propellants from the pump discharge prior to pressurization of the accumulators. The exhaust gas from the heat exchangers and pump turbines is vented overboard via a nonpropulsive vent valve. The three-position valve downstream of each pump is placed so that the pump discharge flow is passed through the heat exchangers—not to the OMS thrust chamber. The APS propellant tank pressure is maintained, while the pumps are running, by gas from the accumulators.

The thrusters are capable of operating in any combination commanded by the GN&C. The gas accumulators are sized by two considerations—capability of maintaining the thrusters in continuous operation (if required) and of restricting the number of pressure cycles on the accumulators to less than 100. For a one main-engine out condition, four thrusters will be operated continuously, and it is this condition which sizes the accumulators. Two GOX accumulators of 13 cubic feet each and two GH_2 accumulators of 68 cubic feet each are used.

The attitude control propulsion subsystem is active for the total time the ESS is on-orbit in order to maintain attitude (pulse mode), except when the OMS engines are burning. The subsystem is also active in providing roll control, with a main or OMS engine out, and for braking during rendezvous (continuous mode).

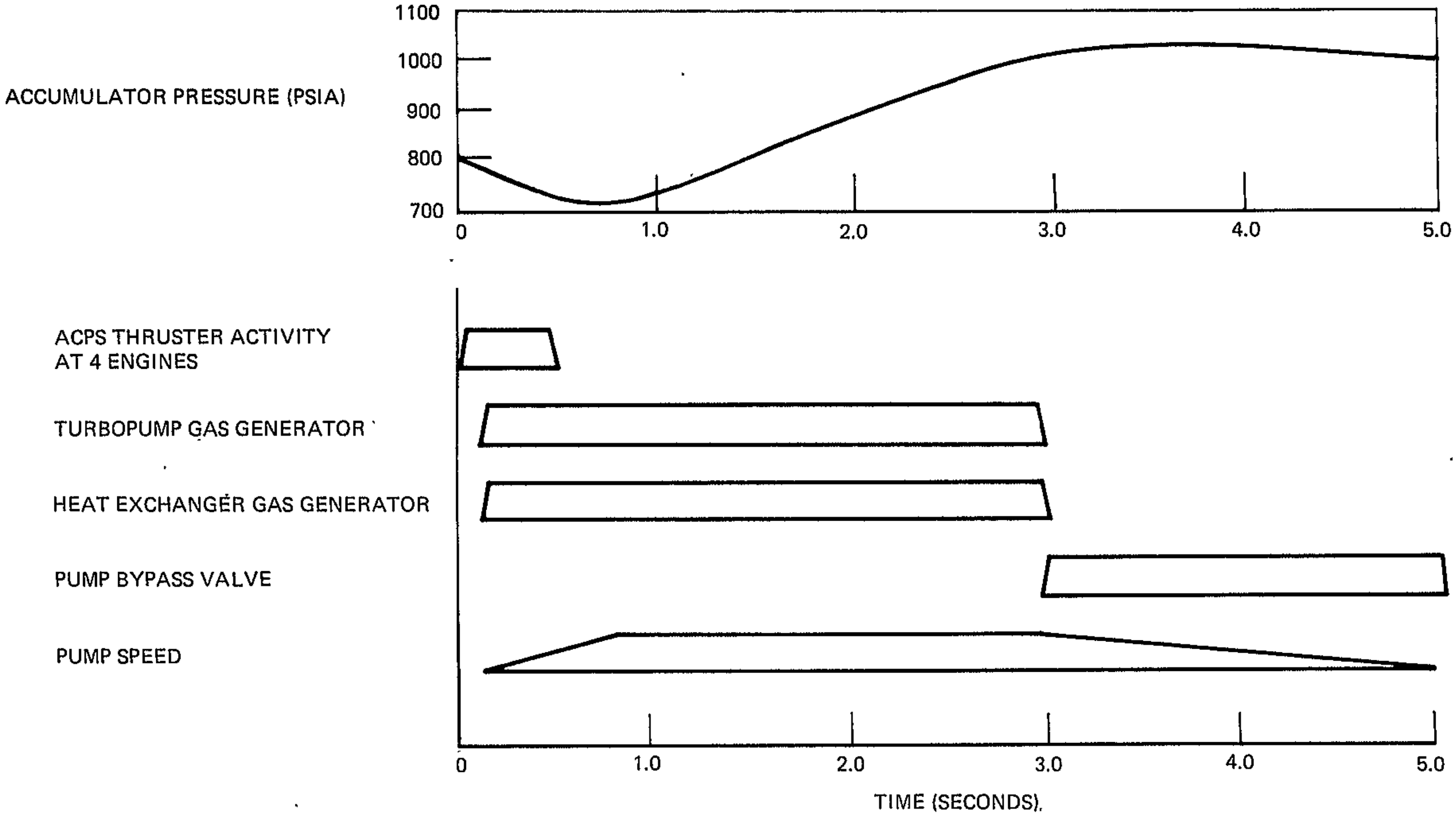


Figure 2-92. ACPS Conditioning System Control Sequence

Orbit-Maneuvering Subsystem (OMS)

OMS Design Requirements. The preliminary design requirements for the OMS are set by the design reference mission. The OMS must provide the velocity increments to change orbits, to rendezvous if required, and to deorbit. The detailed orbit maneuvers are presented in the mission time-line. The basic requirements may be summarized as follows:

1. The subsystem must provide controlled velocity increments to perform orbit maneuvers totaling 621 fps while the ESS and payload are mated. Additional Δ velocity may be required as a result of propellant dump.

2. The subsystem must be capable of providing 25 fps Δ velocity for rendezvous.

3. The subsystem must be capable of providing a Δ velocity of 550 fps to the ESS (without payload) for deorbit.

4. The OMS must have a service life of 24 hours.

5. The vehicle must be controllable with one OMS engine out.

As a result of meeting the basic requirements, a number of other design requirements result, which include withstanding orbit environment and zero-g start.

OMS General Description. Two 10,000-pound thrust engines are utilized for orbit-maneuvering. The engines used are the orbit-maneuvering engines being developed for the space shuttle orbiter. They will be mounted at Positions II and IV with the thrust chamber exit plane at the same station as the main engine with its nozzle retracted. The engine burns LO_2 and LH_2. The characteristics of the engine are shown in Table 2-12.

Propellants are stored in four series-fed LH_2 tanks and a LOX tank inside the main engine thrust cone. All tanks are HPI-protected, Dewar type tanks, the feed tanks employing a thermodynamic vent system to minimize boil-off and ullage venting. The LH_2 tank volume is 208 cubic feet each and the LO_2 tank volume is 280 cubic feet. Liquid propellant feedout under zero g and in adverse load environment during orbital maneuvers is achieved by means of a system of capillary barriers (screens) and collectors (Figure 2-93). The bottom screen in both the LO_2 and LH_2 tanks is used to retain at the tank bottom the volume of propellants required for the deorbit burn.

Table 2-12. Orbit-Maneuvering Engine Characteristics

Parameter	Value
1. Thrust	10,000 lb
2. Mixture ratio, O/F	6 to 1
3. Specific impulse	454 sec
4. Chamber pressure	800 psia
5. Expansion ratio	255
6. Exit diameter	48 in.
7. Length, gimbal to exit	75 in.
8. Weight, thrust chamber	236 lb
9. Weight, pump set	100 lb
10. LO_2 inlet diameter	2.6 in.
11. LH_2 inlet diameter	2 in.

An additional screen is provided in the LO_2 and LH_2 tanks to compartmentize the tanks and thus maintain propellant position control and prevent condensation of tank ullage. A series of collector tubes is provided at the tank walls to provide a flow path between compartments during negative g burns (up to -0.030g). A start basket and a supplemental collector manifold in the bottom compartment of each tank is provided to feed vapor-free propellant to the tank outlet during low-g OMS engine start or during -X axis or lateral accelerations for any ACPS burn.

The propellant budget for the design reference mission is shown in Table 2-13. As can be seen, a margin exists with the planned ESS tankage. Table 2-14 converts the margin to Δ velocity, assuming the heavy MDAC space station payload. For lighter payloads, even more margin exists. The design reference mission also represents a worst-case for the orbital maneuvers required.

Table 2-13. Propellant Budget, APS Baseline Mission—MDAC Space Station

Maneuver/Item	Propulsion	ΔV (fps)	I_{TOT} (lb-sec)	Pounds of Propellant (TOT)	LOX	LH_2
Tank capacity	N/A	N/A	N/A	21,460	17,890	3570
Basic Mission						
1. Circularize at 100 nm	OMS	63	0.59×10^6	1307	1122	187
2. Transfer to 260 nm	OMS	558	5.10×10^6	11,309	9696	1616
3. Terminal phase initiation	OMS	25	0.23×10^6	499	428	71
4. Terminal phase final	ACPS	24	0.22×10^6	580	435	145
5. Attitude control	ACPS	N/A	0.30×10^6	820	615	205
6. Deorbit	OMS	550	1.37×10^6	3035	2604	433
7. Boil-off	N/A	N/A	N/A	50	0	50
8. Chill-down	N/A	N/A	N/A	170	0	163
9. Unusable residual	N/A	N/A	N/A	120	100	20
		1220	7.81×10^6	17,890	15,000	2890
Margin				3570	2890	680

Table 2-14. Propellant Margin for Contingency, APS Baseline Mission-MDAC Space Station

Maneuver Item	Propulsion	ΔV (fps)	I_{TOT} (lb-sec)	Pounds of Propellant (TOT)	LO_2	LH_2
Total margin available	N/A	N/A	N/A	3570	2890	680
Margin used for orbit maneuver	OMS	171	1.52×10^6	3372	2890	482
or						
Margin used for attitude control	ACPS	113	1.01×10^6	2720	2040	680
NOTE Shown is the propellant margin utilized for OMS or ACPS.						

The TVC system concept for the OMS will be essentially the same as that for the main engines. Each engine will be gimbaled by two electro-hydraulic servo actuators. However, unlike the situation in the main engine systems, the sole pumping source in each system will be an electric powered pump. Also, since the OMS/TVC must remain operational throughout the complete mission, the pump will be periodically operated during orbital coast periods to maintain adequate system temperature and thus assure that the TVC will be operable in subsequent OMS burns. A temperature sensor located in the cold point of each system will control operation of the applicable pump during orbital coast periods.

If the final design of the engine and supporting stage structure will allow suitable envelopes for the actuators, all the system components will be packaged in the actuators. If this packaging is possible, the plumbing will be reduced to just two lines that interconnect the two actuators.

The two gimbal axes will be offset 45 degrees from the vehicle pitch and yaw axes in the same manner as those of the main engines. Each actuator will be capable of gimbaling the engine ± 6 degrees and thus to effect pitch, yaw, and roll deflections of ± 8.4 degrees. Each engine will be canted radially outboard 9 degrees (nonfiring) and 8 degrees (firing) in order to point the thrust vector toward the vehicle center of gravity and make possible vehicle control with only one engine firing. During deorbit (no payload), with one engine out, the remaining engine cannot point through the center of gravity; so the ACPS is used for flight control.

The actuator orientation and a simplified TVC hydraulic system schematic are pictured in Figure 2-93.

Although it is not a requirement that the ESS be designed to conform to fail operational/fail safe (FO/FS) criteria for orbital maneuvers, the FO/FS criterion has been applied for the following two reasons:

1. The OMS is integrated with the ACPS which is required to be FO/FS since attitude control affects orbiter crew safety during the ESS engine and avionics recovery operation. With the exception of the OMS engine thrust chamber assemblies, the OMS must therefore be FO/FS.

2. The OMS is utilized for deorbit. Although this fact does not affect orbiter crew safety, it can have an effect on human life. For this reason the FO/FS criterion has been applied.

The FO/FS analysis of the OMS is presented in Table 2-15. Two conditions were identified in which the subsystem is not FO/FS. They are the following:

1. There are only two sets of OMS pumps on the ESS. Since the pumps are required for both OMS and ACPS operation, any failure that results in a condition of both LO_2 pumps or both LH_2 pumps being inoperative will disable the ESS APS. The failures could be to the pumps themselves or to equipment required to operate the pumps (GG's, valves, etc.).

2. If both LO_2 fill valves fail open, the APS LO_2 will be bled overboard.

These two conditions have been waived based on the following rationale:

1. Two pump sets are considered adequate since the ESS design life is only 24 hours. The pumps are being developed for the space shuttle orbiter and will be designed to operate for much longer missions. Thus, the probability of two failures is considered remote enough to preclude the addition of a third set of pumps. It should also be noted that the failures must be of both LO_2 or of both LH_2 pumps to render the APS inoperative. Failure, for example, of a LO_2 pump and of an LH_2 pump will not render the subsystem inoperative.

Table 2-15. OMS Fail-Operational/Fail-Safe Considerations*

Failure	Remarks and Rationale
OMS engine	After first engine failure, the second engine provides operational capability. After the second engine failure, the translation ACPS thrusters provide fail-safe backup.
LO_2 pump	After the first LO_2 pump failure, the second LO_2 pump provides flow to keep the second OMS engine on line, and the subsystem is operational. After the second failure, neither the OMS engine nor the ACPS thrusters are operative, so that the subsystem cannot be considered fail-safe. The FO/FS requirement for the second failure is not met.
LH_2 pump	Same as for LO_2 pump. The FO/FS requirement for the second failure is not met.
Pump drive GG	After the first failure, the subsystem remains operational. If the first failure is a LO_2 pump drive GG and the second is an LH_2 drive GG (or vice versa), the system is fail-safe, inasmuch as the ACPS provides backup. If the two failures are both LO_2 pump drive GG's or LH_2 pump drive GG's, the subsystem is inoperative and does not meet FO/FS requirements.
Tank repressurization regulator	Three regulators with appropriate isolation valves are provided so that the subsystem remains operational after the second failure.
Propellant tank vent	Since the system is designed so that the APS tank never relieves in flight, the first failure mode is that relief is required. If the relief valve fails (second mode), a backup burst diaphragm and relief valve are provided.
Fill valves	Two fill valves in series are provided to assure fail-operational capability. If both LH_2 fill valves fail open, fail-safe is provided by sealing in the disconnect. If both LO_2 fill valves fail open, LO_2 will drain overboard as the APS LO_2 fill tees off the main LO_2 fill, and there is no sealing disconnect. The LO_2 fill is therefore not FO/FS.
Prevalves	The first failure mode is that the prevalves be required to close since they will normally be left open for the entire flight. Two prevalves in series are provided so that the subsystem is FO/FS. A closed failure of the prevalve is not considered. If a closed failure is considered, the failure of two prevalves could result in the subsystem's being inoperative.
Thermodynamic vent	The thermodynamic vent is designed with redundant components to provide fail-operational capability. With the second failure, the vent becomes inoperative but this is considered a fail-safe condition since adequate time is available between the failure of the vent and any detrimental effects to terminate the mission and safely deorbit.
Feedline/turbopump coolant	The coolant circuit is designed with redundant components to provide fail-operational capability. With the second failure, the coolant circuit becomes inoperative, but this condition is considered a fail-safe condition since the OMS engine inlet valves can be opened and the pump inlet conditioned by bleed as a backup fail-safe mode.
GG inlet valves, regulators, select valves, check valves, etc.	The various system components are installed (with applicable redundancy) as required to support the prime component (GG, pump) operation. For this analysis, they may be considered a part of the prime component and the same considerations will apply. For instance, if the oxygen inlet valve to both LO_2 pump drive GG's fail, the system is inoperative.

*It should be noted that not meeting FO/FS usually does not involve crew survival (safety), since when the OMS engines are operating, the vehicle is unmanned.

2. The LO_2 fill valves will be closed on the ground, and the position will be verified. The open failure of two valves which do not have inflight control (inadvertent opening is impossible) is so remote as to warrant not applying the FO/FS requirement.

OMS Operation. The orbit maneuvering subsystem for the ESS will operate in the same manner as the space shuttle OMS.

Prior to lift-off, liquid oxygen and hydrogen are contained in the APS tanks, the propellant tanks are pressurized, and the feedline and turbo pump cooling is initiated. The prevalves are open and liquid propellants are present in the feed and pump system throughout the mission.

The turbo pumps are powered by individual gas generators and supply liquid oxygen and hydrogen to the regeneratively cooled engine thrust chamber. Ignition is provided by redundant spark plugs. The turbine exhaust is vented overboard via a propulsive vent (for each engine) which provides 100 pounds of the total engine thrust of 10,000 pounds.

Pressurization of the propellant tanks is accomplished by tapping hydrogen gas from the thrust chamber and tapping oxygen from the LO_2 pump discharge and passing it through a coil around the LO_2 pump turbine drive gas generator.

A typical start sequence of the OMS engine is shown in Figure 2-94.

The OMS engines operate simultaneously, and thrust vector control is provided by gimballing the thrust chambers.

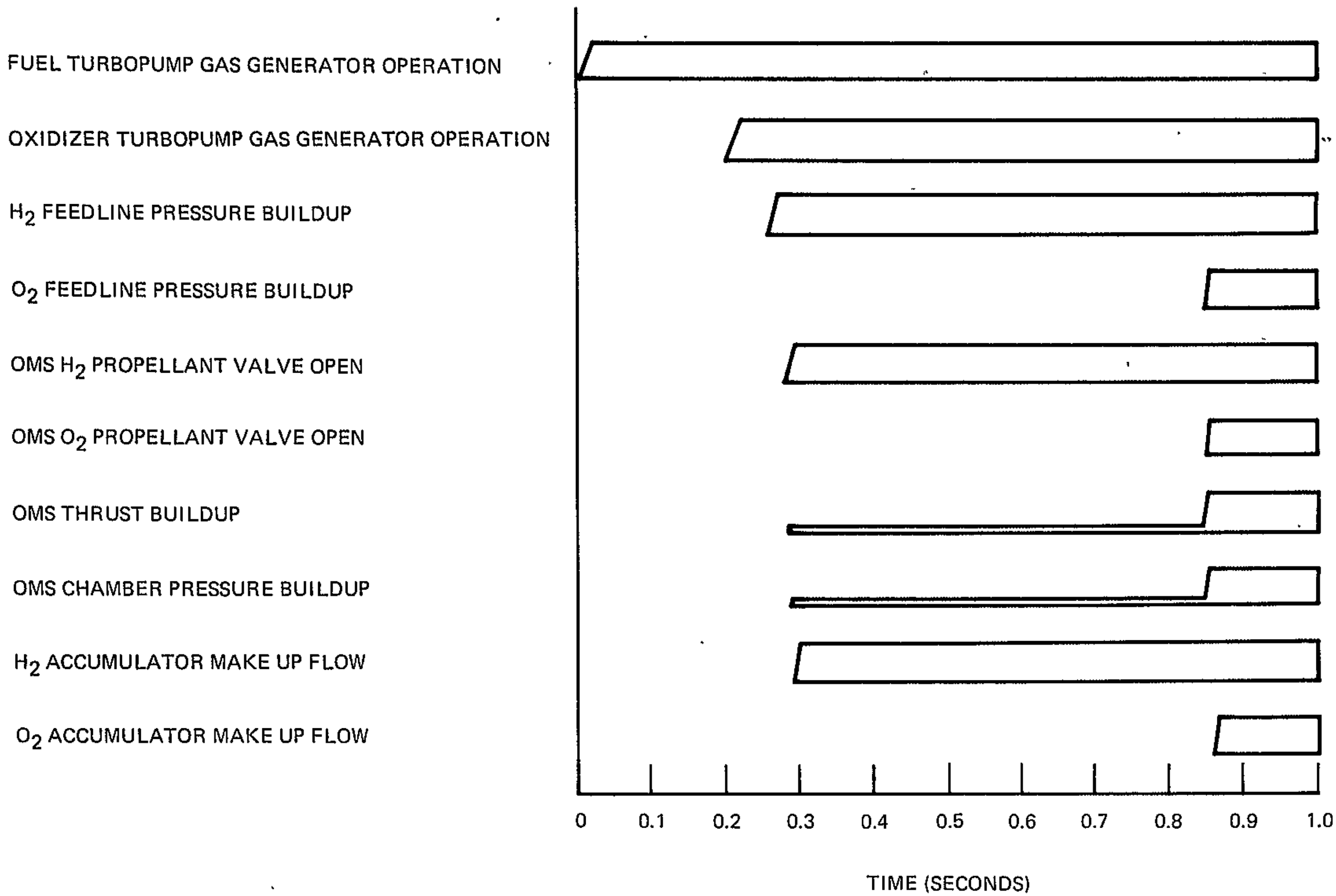


Figure 2-94. Orbit Maneuvering System Start Sequence

2.3 AVIONICS SUBSYSTEM GROUP

The following prime issues were derived from negotiations after the submittal of the Phase A report and taken as items to be addressed and resolved during the Phase B study:

1. The use of a 33-foot-diameter instrument unit (IU) as defined by Configuration II, Task 5 Change Order 1928, (INT-21) or, alternately, the incorporation of the guidance, navigation, and control (GN&C) functions into the expendable second stage (ESS).

2. The use of conventional (relay-logic) type equipment or the advancement to state-of-the-art, solid-state, integrated, central-control-type equipment.

3. Investigation of the feasibility and practicality of recovering high value avionic equipment from orbit.

The Phase B study activities have produced a subsystems approach which satisfies study objectives of performance, low cost, and minimum development risk. Figure 2-95 illustrates the selection process of the avionics approach and the transition from the Phase A recommendation.

As shown in Figure 2-95, a 33-foot-diameter instrument unit, modified to incorporate redundancy, was excessive in cost and complexity. Hardware growth and environmental control demands for the 24-hour baseline mission could be accommodated by conventional avionics equipment, but many peripheral items were added when the Phase B baseline configuration was established.

The initial decision to integrate the GN&C functions into the ESS to minimize costs provided the basis from which to reevaluate the systems approach. Table 2-16 presents the selection criteria. This, together with the selection of a vehicle configuration, provides a baseline for building the rational network from which final decisions were made.

As shown in Figure 2-95, the Phase B approach for avionics subsystem selection is to utilize shuttle-developed equipment and software, to utilize Apollo/Saturn hardware where cost and functional requirements are satisfied, and to develop new hardware only if existing equipments do not satisfy the need.

Figure 2-96, shows the derivation of equipment to be used on the ESS. This figure illustrates a minimum demand for new components for the ESS.

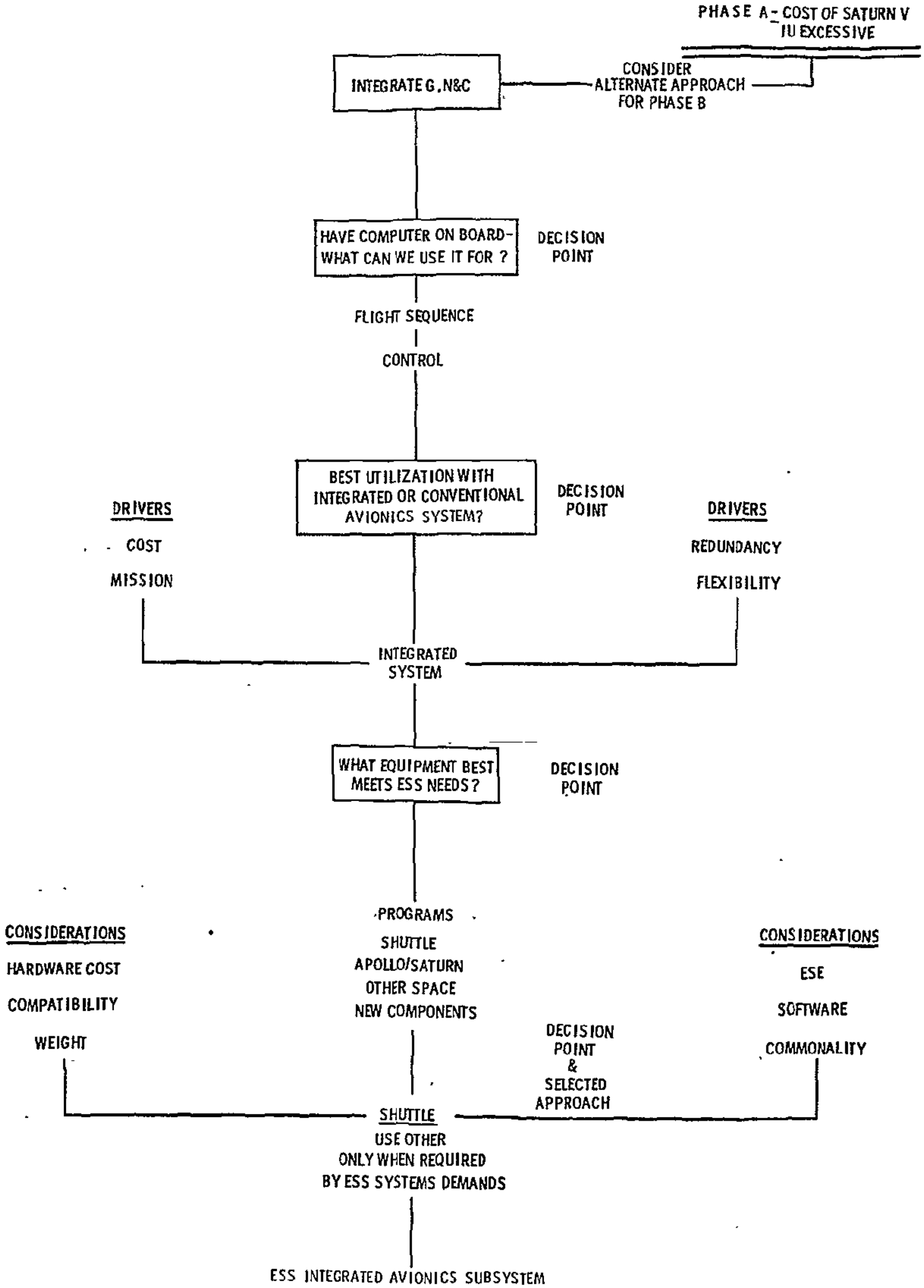


Figure 2-95. Rationale for Integrated Systems Selection

Table 2-16. Conventional Versus Integrated Avionics Subsystems

S-II DERIVATIVES SYSTEM EVALUATION TRADES

DATE: 5/13/71

MAX RATINGS / TRADE FACTORS (CANDIDATES)	CONV. ASTRIONICS	INTEG. ASTRIONICS	REMARKS
(300) TECHNICAL	I	II	
(50) PERFORMANCE	50	50	
(50) RELIABILITY	50	50	
(50) SAFETY	50	50	
(30) STATE-OF-THE ART	10	30	II UTILIZES 1972 EQUIP BASELINE
(10) WEIGHT/SIZE	2	10	II LESSER QTY FOR MORE COMPLEX STS
(15) QUAL/VERIF TESTING	10	15	II SHUTTLE PROVIDED TESTING - I Δ FOR NEW ITEMS
(25) IMPACT ON OTHER SYS	10	25	I NEEDS MUCH TO BE COMPAT. W/ENG & NG&C
(25) GROWTH POTENTIAL	10	25	II BUILT IN HARDWARE FLEXIBILITY
(45) ANTICIPATED PROBLEMS	45	35	II NEW APPROACH
( )			
( )			
SUB-TOTAL	237	290	
(150) PRODUCIBILITY			
(50) EASE OF MANUFACTURE	30	50	
(40) FAB STATE-OF-THE-ART	30	40	II MINIMIZES INTERCONNECTING CABLES
(20) INSPECTION CAPABILITY	20	20	II UTILIZE 1972 BASELINE EQUIP
(20) FACILITIES IMPACT	5	20	
(20) HARDWARE AVAILABILITY	10	20	II BASED ON SHUTTLE PROCUREMENT
( ) KITABILITY			
( )			
SUB-TOTAL	95	150	
(150) OPERATIONS			
(50) GSE IMPACT	20	50	II USES SHUTTLE EQUIP
(30) MAINTAINABILITY	20	30	II LESS NUMBER OF ITEMS
(30) C/O IMPACT	10	30	II HIGH RATE AUTO C/O
(40) LAUNCH FACILITIES IMPACT	5	40	II USES SHUTTLE EQUIP
( )			
SUB-TOTAL	55	150	
(400) COST/SCHEDULE			
(150) NON-RECURRING COSTS	50	150	II USES OFF SHELF SHUTTLE EQUIP
(150) RECURRING COSTS	50	150	II USES OFFSHELF SHUTTLE EQUIP
( )			
(100) SCHEDULE COMPATIBILITY	80	100	II SHUTTLE SCHEDULE COMPATIBILE
SUB-TOTAL	180	400	
(1000) TOTAL SCORE	567	990	

MOST DESIRABLE - HIGH RATING
LEAST DESIRABLE - LOW RATING

NOT ACCEPTABLE - X
NOT APPLICABLE - N/A

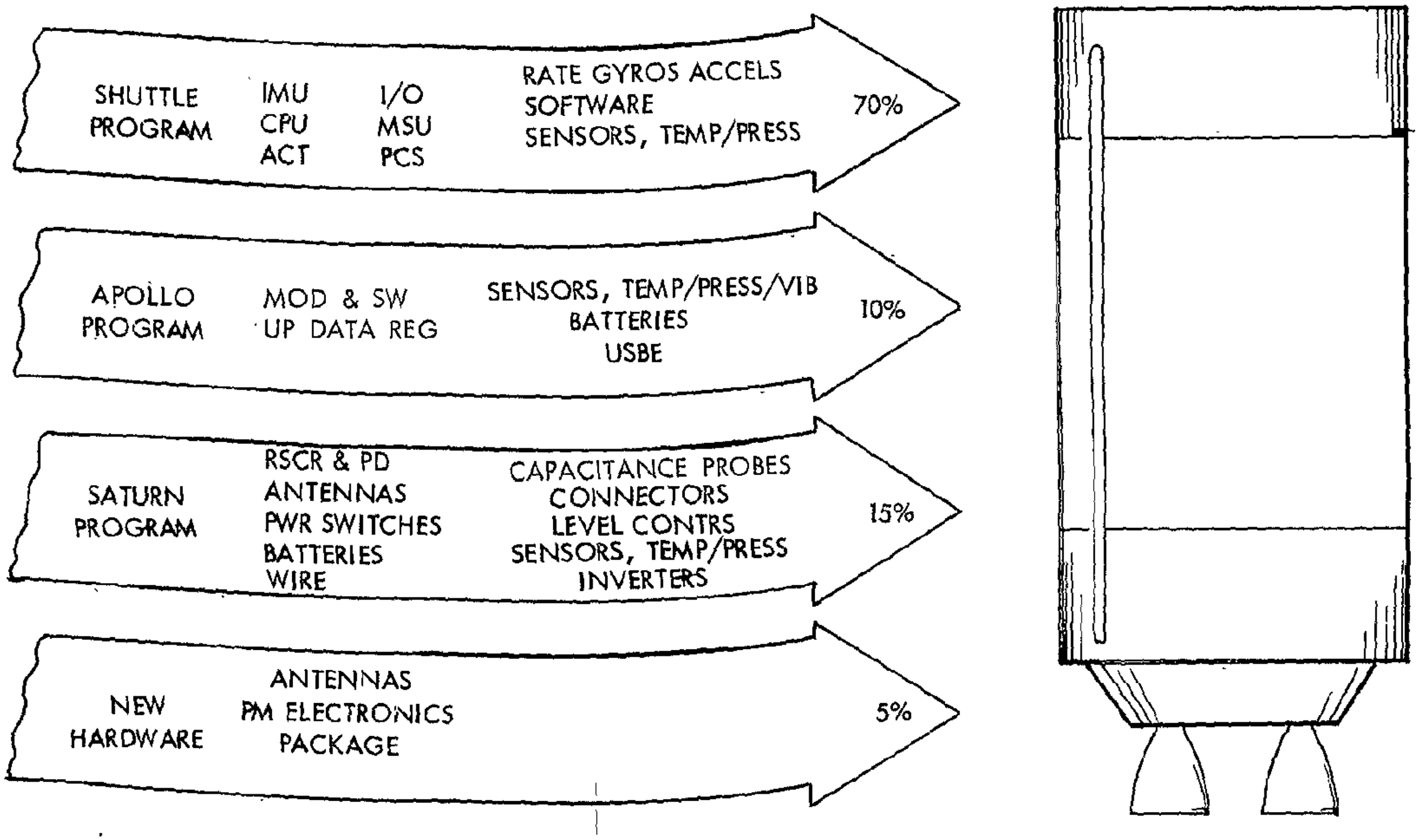


Figure 2-96. Avionics Hardware Sources

Figure 2-97, shows the avionics configuration proposed for the ESS. Table 2-17 lists the components and fault tolerance allowed.

The following is a brief description of the major subsystem elements and their function in the ESS mission:

1. Data and Control Management (DCM) Subsystem. The DCM subsystem is the brain and nerves of the ESS avionics subsystem and as such is the central integrating agent for all ESS functions. The subsystem consists of the hardware and software to address storage, fetch, and store information, process arithmetic and logical data, execute sequential instructions, process interrupts, and initiate communications between storage and the input/output devices.

2. Guidance, Navigation, and Control (GN&C) Subsystem. The function of the GN&C subsystem is to determine the position, velocity, attitude, and attitude rate of the vehicle, to compute, via the DCM subsystem, desired changes to any or all these vehicle states in accordance with internally stored or alterable computer programs, and to provide control signals to the propulsion systems for accomplishing those changes.

3. Communications (COMM) Subsystem. The COMM subsystem provides the capability to transmit and receive all RF information necessary to accomplish the basic ESS mission by providing telemetry data, turnaround ranging data, receiving up-data, and range safety commands.

4. Electrical Controls (EC) Subsystem. The EC subsystem is the interfacing agent between the avionics subsystem and the non-avionics subsystem. Requirements for this subsystem are primarily established by the nonavionic subsystems. Demands upon the DCM subsystem for operation are established by the EC subsystem and the redundancy and operations criteria for the following subsystem elements: main engine, auxiliary propulsion, engine actuation, pressurization propellant feed, propellant management, separation, safing, propellant dispersion, and equipment recovery and subsystem elements.

5. Instrumentation Subsystem. The instrumentation subsystem provides for all measurements not involved in systems monitoring. These include tank temperatures, insulation temperature, vibration, and noncritical pressures. The requirements for signal conditioning, multiplexing, and bridges are included in this system.

6. Electrical Power and Distribution (EPD) Subsystem. The electrical power and distribution subsystem consists of the energy sources and the distribution and conversion of electrical power for all avionic system loads. Two dc voltage levels of 28 and 56 vdc have been established for the basic power buses. The 28-vdc buses will supply all 28-vdc vehicle loads. The 56-vdc bus will be used for conversion to all ac-voltage levels required.

Batteries have been selected as the energy source for the 24-hour mission. Approximately 2500 pounds of batteries are required to support the ESS mission.

The equipment required to provide fault tolerance levels are shown on the block diagram, Figure 2-97. The numbers utilized for ESS are noted in Table 2-17. Where Apollo/Saturn equipment is used to fulfill redundancy requirements, the usage is so noted. The configuration established by the above is in concert with the ESS redundancy guideline which is:

> "From mated ascent through booster/ESS separation, all items that may affect booster safety shall be FO/FS. During ESS boost and orbital operations, redundancy for subsystems and components shall be determined on an individual basis, considering criticality and cost effectiveness. Equipment used for rendezvous and/or deorbit shall be FO/FS."

The avionics equipment installations have been arranged to minimize demands on the environmental control system, thereby minimizing demands on the vehicle power system. Cold areas of the vehicle are used for high-heat-producing equipment. This is supplemented by high-performance insulation and coatings for the containers. This design approach has enabled the environmental control system to be limited to use of heaters and heater control equipment. Coldplates or heat pumps will not be required based upon the analysis completed to date. Figure 2-98 reflects the installation arrangement for the avionics equipment.

One prime method of reducing total program costs would be to recover high-value components from orbit. Items under consideration are the main propulsion system engines and aft-mounted avionics equipment.

Analysis has determined that maximum recoverability of avionics equipment is necessary to gain a minimal cost advantage. Three options were analyzed in an attempt to fully explore the potentials of recovery: Option I, no recovery; Option II, partial recovery; and Option III, maximum recovery. Table 2-18 shows some of the considerations given to the recovery

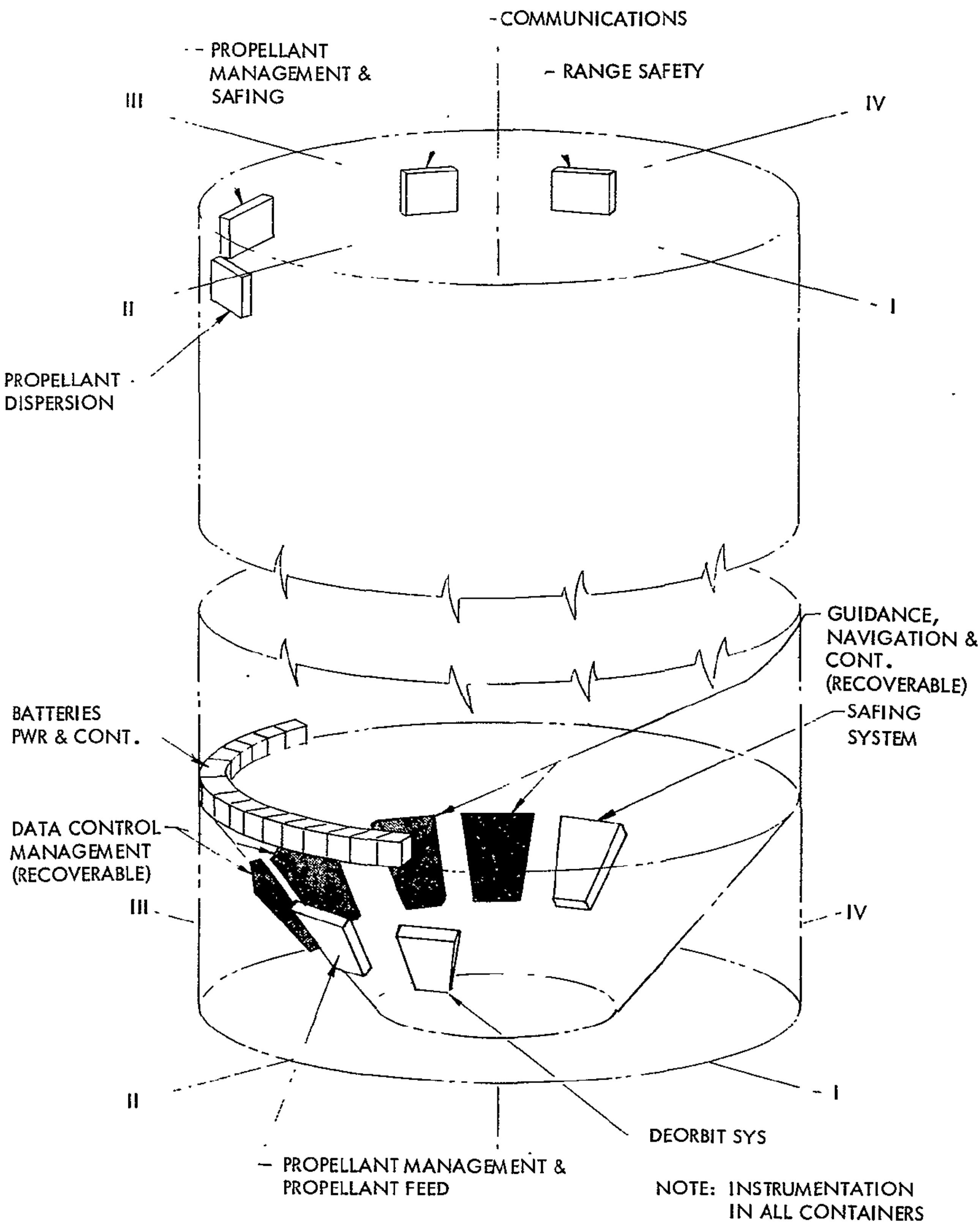


Figure 2-98. Avionics Equipment General Arrangement

Table 2-18. Deorbit Considerations

Consideration	Option I No Recovery	Option II Partial Recovery	Option III Maximum Recovery
Mission time line impact	No	Yes	Yes
Software changes for recoverability	No	Yes	Yes
Development of recovery methods	No	Yes	Yes
Environmental effect on removed hardware	No	Yes	Yes
Added failure modes due to recovery	No	Yes	Yes
Added checkout requirements	No	Yes	Yes
Refurbishment costs	No	Yes	Yes
Cost of new hardware recoverable — installation, packaging, safing	No	Yes	Yes
Shuttle support for deorbit	No	Yes	Yes
Highest initial hardware cost	No	No	Yes
Highest total program cost	Yes	No	No

analysis. Each "yes" must be considered as a negative or undesired characteristic. Table 2-19 reflects pure hardware cost savings which could be realized.

Because of the ramifications of this approach, an in-depth analysis is recommended to determine the optimum selection, as the impact of the following unknowns has not been established:

1. Software changes for safing and deorbit
2. Development of recovery techniques
3. Mission time-line constraints

Table 2-19. Forecasted Hardware Savings (Option I Versus Option III)

COMPONENT	QTY	OPTION I (NONE)	OPTION II (PARTIAL)	OPTION III (MAX)
IMU	3	-	$900K	$900K
CPU/MAIN MEMORY	4	-	-	1320K
			$900K	$2220K
OTHER COSTS				
REFURBISHMENT		-	$360K	$ 888K
INSTL & CHECKOUT		-	150K	387K
NEW DEORBIT KIT		-	-	600K
DELTA PACKAGING COSTS		-	240K	300K
			$750K	$2175K

BASED UPON THE ABOVE:

OPTION I NO RECOVERY, BUILD 20 SETS ($2,220K EA) = $44.4M

OPTION III RECOVERY, BUILD 5 SETS & REUSE

BUILD 5 SETS	11.1 M
INSTALL & C/O (15 SETS)	1.6 M
REFURBISHMENT (15 SETS)	11.1 M
DELTA PACKAGING COST	6.0 M
DEORBIT KIT (20 SETS)	12.0 M
	$ 41.8 M

−41.8 M

FORECASTED HARDWARE SAVINGS (OPTION I vs OPTION III) $2.6 M

(BASED ON IMU COST OF $300K EA AND CPU/MEMORY UNIT COST OF $330K EA)

4. Effect of thermal shock on removed equipment

5. Added failure modes

6. Demands upon shuttle orbiter for attitude orientation for deorbit

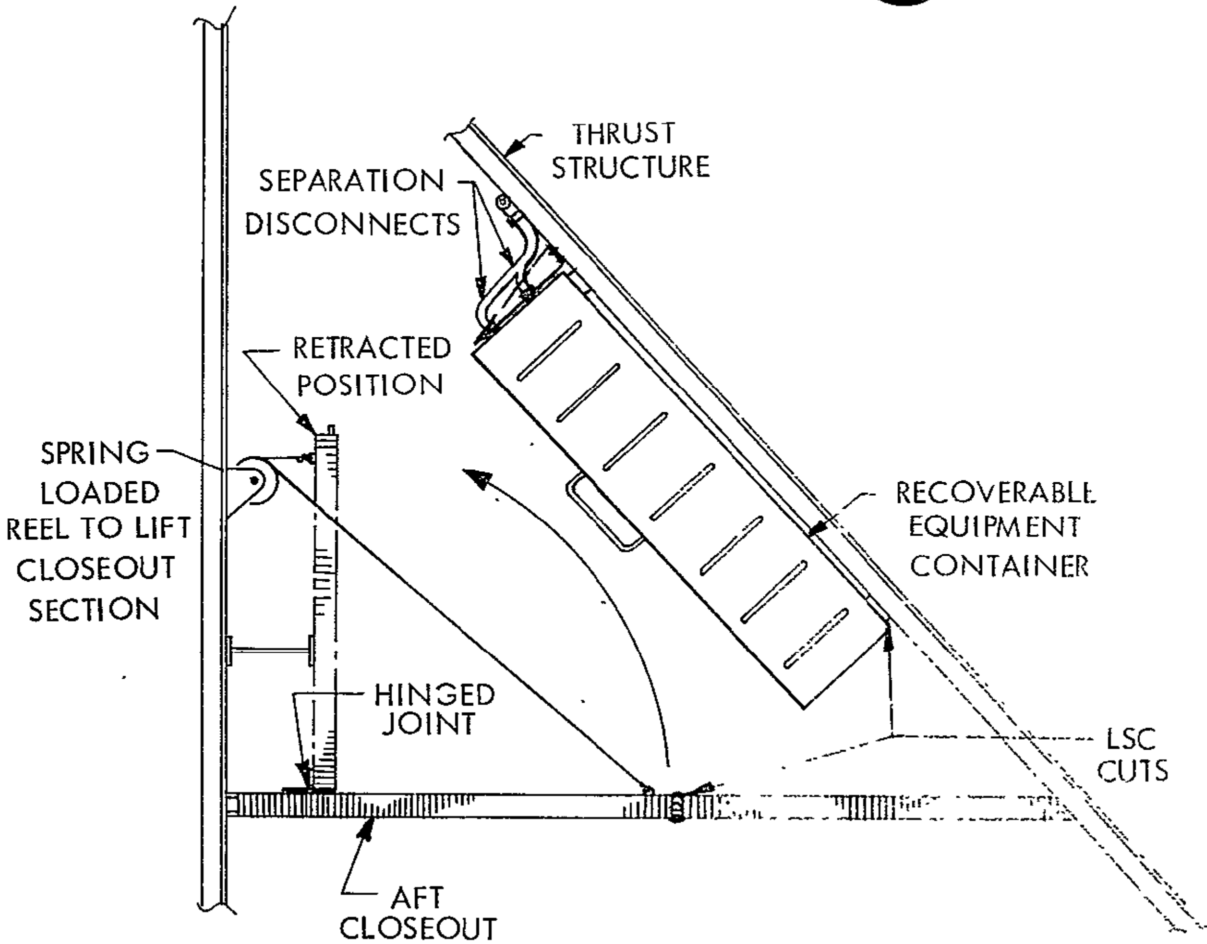
Alternately, if the assumed cost of refurbishment can be lowered or the number of equipment sets reduced, additional savings may be possible. Further evaluation appears to be warranted.

Techniques have been explored in accomplishing separation of the equipment containers with shaped charges similar to the method developed for the Saturn program. These devices, as well as separation-type electrical connectors, are well within existing technology. Shaped charges would also be utilized to cut away the aft closeout to provide access to the equipment containers. Figure 2-99 depicts one proposed method of providing access for a recoverable equipment installation. (See Section 2.3.7 for additional details on equipment recovery installations, and Section 2.3.8 for deorbit avionics description.)

A hardwire interface will be provided to enable the booster crew to monitor critical ESS functions during the ascent phase and, if necessary, to initiate the separation phase for abort purposes. Although not required, the capability to update the ESS navigation equipment is inherently provided by this interface. The physical interface will be accomplished in a similar manner as proposed for the booster/orbiter interface to minimize special devices for the booster/ESS.

Acquisition, control, and test (ACT) select buffers will be interposed between the ESS and the booster data buses to isolate and provide the means of coupling the systems together. Each ACT/select buffer is capable of withstanding short circuits on the input without outputting spurious signals to either data bus. Redundant paths will be provided on each side of the vehicle. The ESS ACT/select buffers will be installed in the forward skirt. The hardware cables will be routed in each of the booster forward struts. The proposed installation is shown in Figure 2-100.

A hardware interface will be provided to enable the electrical support equipment (ESE) to check out and monitor all ESS functions. The prime interface is with the universal test console (UTC) developed for the shuttle program. This piece of equipment will utilize common software routines to fully check out and support the launch of the ESS. The interface with the UTC will be through ACT/select buffer units to assure isolation of both the ESS and UTC subsystems. Dual interfaces will be provided to maintain redundancy requirements. For checkout, additional ESE support is required to provide 28-vdc and 56-vdc power for battery simulation.



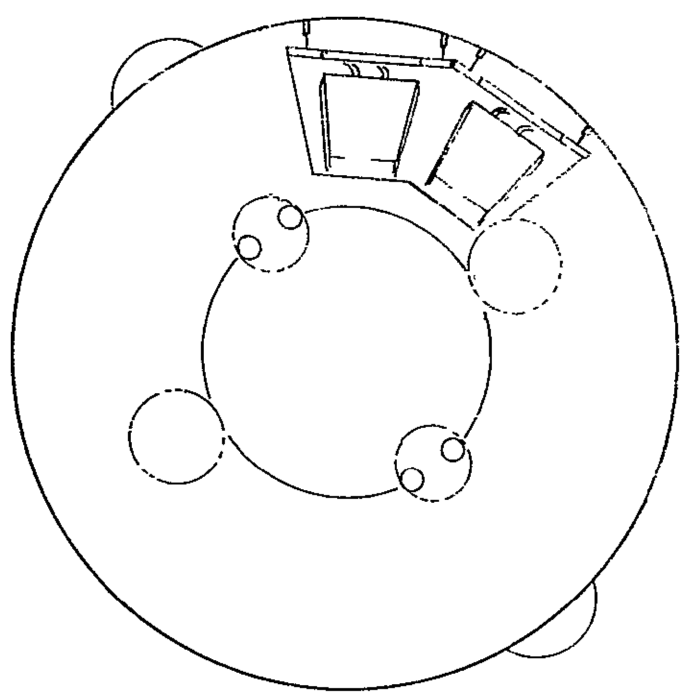


Figure 2-99. Recoverable Avionics Equipment

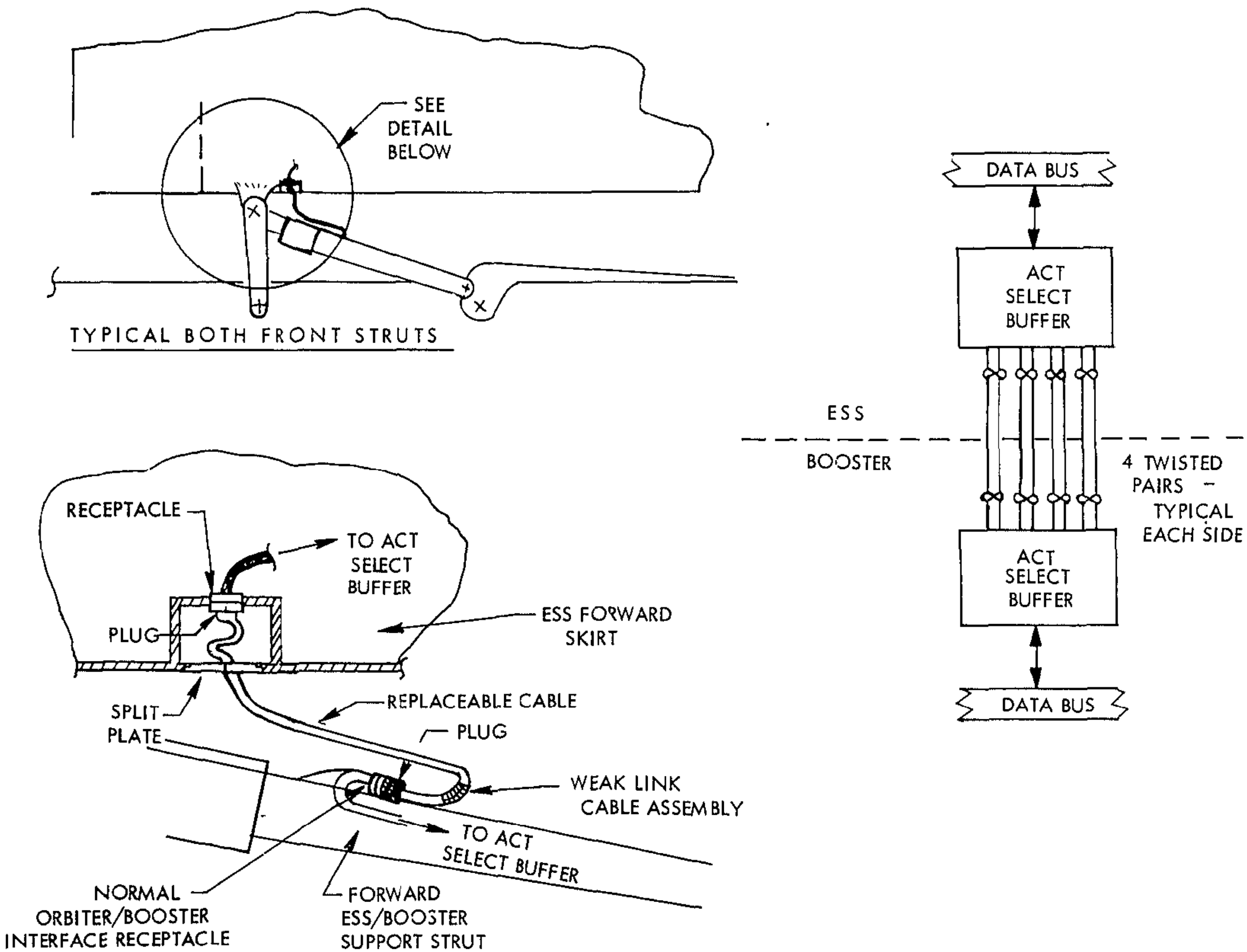


Figure 2-100. ESS/Booster Avionics Interface

The Phase B study activities have achieved the major objectives of the study plan and have produced the definition of a viable avionics subsystem. However, during this study, several areas were identified where additional effort should be expended to develop the system hardware and software. These items are applicable to several on-going programs, and certain of these items are directly concerned with the Shuttle Program. The following are deemed desirable for further development:

1. Computer software programs represent single-point failures. Develop efficient means for program development and transmission via the up-link command system of programs and/or subroutines for vehicle safing and anomaly correction for which on-board programs have not been programmed. Develop time-saving methods for programming, coding, verification, and preparation for transmission methods from time-of-anomaly to time-of-transmission.

2. Develop all aspects of recovery of avionics equipment from orbit, i.e., mechanical release, interface connectors, handling aids, ordnance devices, as well as recovery procedures.

3. Analyze use of TDRS/MSFN as concerns data transmission, up-link command.

4. Develop an integrated package which could be dedicated to the deorbit task. This package is to provide all attitude control, and sequencing functions required to deorbit large vehicles from orbit.

5. Develop a solid-state power control switch which is compatible with the data bus/ACT Units being developed for the Shuttle Program. The switches must meet short-circuit trip and possess no single-failure modes.

6. Analyze and develop requirements for static inverters in the 5-kw to 10-kw rating range.

7. Develop a technique to utilize the shuttle orbiter to align the ESS in preparation for ESS deorbit.

2.3.1 Guidance, Navigation, and Control (GN&C) Subsystem

This section presents the results of preliminary trade studies, performance analyses, and equipment evaluations performed to define a cost-effective, reliable GN&C subsystem which will satisfy the requirements imposed by all ESS reference missions.

A coordinated ESS GN&C study has been performed by IBM and documented in IBM Study Report MSFC-DRL-008A, Astronics Systems Study for Saturn S-II Expendable Second Stage, dated (30 June 1971). Applicable material has been incorporated into this report.

GN&C Requirements

The following sections summarize the background, assumptions, and interpretations from which the GN&C subsystem requirements were derived, followed by a definition of the requirements, and an explanation of the rationale and analyses used in arriving at these subsystem requirements.

Mission Requirements. The primary ESS GN&C requirement will be to inject the baseline payloads into appropriate low-earth orbit without a phasing or rendezvous requirement. Rendezvous with a passive target vehicle in a 270-nm circular orbit is considered to be a secondary requirement. The baseline payloads and associated GN&C related mission parameters are as follows:

1. Inject a space tug payload into a 100-nm circular orbit at 28.5 degrees inclination. This will be accomplished by first-stage boost to approximately 38 nm; separation from the booster; an ESS main engine burn into a 66-nm by 100-nm eliptical orbit; coast to near apogee; and finally, an OMS burn to circularize at 100 nm. The ESS will release the payload, move to a safe distance, and wait for the orbiter rendezvous. The orbiter will dock with the ESS, and remove and stow main engines and selected avionics equipment. The orbiter will then enable the deorbit control system and maneuver the ESS into the inertial attitude required for deorbit to establish the attitude reference for deorbit. The orbiter will undock, move to a safe distance from the ESS, activate ESS propulsion systems, wait for ESS to reacquire deorbit attitude, and command the deorbit burn sequence. The trajectory and significant mission events for this mission are depicted in Volume II, Book 1, Paragraph 5.2.2.

2. Inject a MDAC space station payload into a 270-nm circular orbit at 55 degrees inclination. This will be accomplished by injecting the ESS/payload combination into a 100-nm circular orbit as described in the space tug mission (Item 1 above) and remaining in the 100-nm orbit long enough to make at least two navigation updates to reduce position and velocity errors to an acceptable level. An OMS burn will then be initiated to achieve a 100 by 270-nm eliptical orbit. The ESS will coast to near apogee where an OMS circularization burn will be made to achieve a 270-nm circular orbit.

The separation from the payload, docking with orbiter, removal of main engines selection of avionics equipment, and the deorbit sequence will be identical to those described for the space tug mission. The trajectory and significant mission events are depicted in Volume II, Book 1, paragraph 5.2.2.

3. Inject a reusable nuclear shuttle (RNS) into a 260 nm circular orbit at 31.5 degrees inclination. This will be accomplished in an identical manner to that for the MDAC space station mission described in Item 2 above except for the differences in altitude and inclination.

4. Rendezvous a MDAC space station payload with a passive target in a 270-nm, 55-degree inclination circular orbit. This is accomplished by injecting the ESS/payload combination into a 100-nm circular orbit (as described in the space tug mission, Item 1 above). The ESS will remain in the 100-nm orbit long enough to phase and to make sufficient navigation updates to reduce the errors to acceptable levels. An OMS burn will then be initiated to inject into a 100 by 260-nm eliptical orbit. The ESS will coast to near apogee where an OMS circularization burn will be made to achieve a 260-nm circular orbit. A minimum of one navigation update will be made during the ascent. Fine phasing with the target vehicle will be accomplished in the 260-nm circular orbit, followed by a transfer to a 270-nm circular orbit in the proximity of the target. Multiple navigation updates will be accomplished during phasing and station-keeping. Separation from the payload, docking with the orbiter, removal of main engines and avionics equipment, and deorbit sequences will be identical with those described for the space tug mission. The trajectory and significant mission events are depicted in Volume II, Book 1, Paragraph 5.2.2.

Ground Rules and Guidelines. The following ground rules and assumptions form the background for this study:

1. The baseline GN&C will be configured to meet the requirements of the basic (nonrendezvous) missions described in the previous section.

2. Subsystem performance capabilities for rendezvous with a passive target in a 270-nm circular orbit, using only the baseline GN&C, will be defined, including launch constraints.

3. Subsystem refinements and operational requirements required to effect close rendezvous (2 ± 1 nm as a goal) with passive targets in 270-nm circular orbits will be discussed.

4. Subsystem refinements and operational requirements required to accomplish a synchronous orbit will be discussed.

5. Fail safe/fail operational (FO/FS) configuration at lift-off of all subsystem elements which might affect safety of the crew in the booster, the target vehicle, or the shuttle orbiter, is considered mandatory. Disposition of malfunctions and associate decisions to continue or abort the mission subsequent to lift-off will be considered a function of mission rules.

6. Reliability goals will be such as to be consistent with less than 0.5-percent probability of one unsuccessful mission to the GN&C failures in 20 missions.

7. Deorbit of the expended ESS to an acceptable impact area with an acceptable footprint is mandatory.

8. Mission duration will be limited to 24 hours (launch through deorbit).

9. A tracking and data relay satellite (TDRS) system is assumed to be available with no cost to the ESS program.

10. Navigation updates will be accomplished without any requirement for special vehicle maneuvers or positioning and with no changes required to existing manned space flight network (MSFN) stations.

11. Shuttle-developed hardware and software will be utilized to the greatest extent possible to minimize new development and evaluation and qualification costs, and to maximize use of shuttle GSE and development tools.

12. Engine on-off drivers for the OMS and ACPS, and engine gimbaling servo-amplifiers for the main engines and OMS engines are considered to be part of the GN&C subsystem.

13. The main engine computers, the data control and management (DCM) subsystem, the engine gimbaling servo-actuators, the OMS and ACPS propellant valves, and the OMS and ACPS igniters are considered to be significant interfaces.

Functional and Performance Requirements. The GN&C, together with appropriate interfacing subsystem elements (DCM, main engines, OMS, and ACPS) and available ground or satellite aids (MSFN and/or TDRS), will provide the capability to determine the position, velocity, and inertial attitude of the ESS/payload from launch through deorbit. It will also provide attitude control from booster separation, through ascent, on-orbit operations, on-station operations, and deorbit. This includes thrusting modes (main engine and OMS velocity changes) and nonthrusting modes (angular maneuvers and attitude holds using ACPS).

The GN&C will perform the above functions by providing individual equipment with accuracies and drift characteristics compatible with the performance requirements defined in Table 2-20.

The GN&C will be compatible with the functional requirements by mission phase listed in Table 2-21 for the nonrendezvous baseline mission and in Table 2-22 for the rendezvous mission.

The GN&C will be capable of self-alignment during prelaunch and require no attitude updates throughout the mission.

GN&C Subsystem Description

The following sections describe the baseline GN&C subsystem selected for ESS.

Functional Description. A functional block diagram of the ESS GN&C is shown in Figure 2-101. The diagram depicts a system which utilizes:

1. Inertial navigation data for GN&C operations

2. No attitude reference update from launch through deorbit

3. Navigation updates through MSFN and/or TDRS tracking and uplink

4. Blending/filtering of navigation data to provide an optimum state vector

5. Separate rate sensors for stability augmentation

6. Automatic operation with checkout and fault isolation used in redundancy switching

7. Digitally computed guidance steering laws for all phases

Table 2-20. GN&C Performance Requirements

PERFORMANCE PARAMETER	MISSION		
	100 N.M.	270 N.M.	270 N.M. W/REND.
Injection Errors (3σ):			
• Radial position	±3 N.M.	±5 N.M.	--
• Radial velocity	±20 fps	±50 fps	--
• Tangential velocity	±15 fps	±15 fps	--
• Inclination	±0.1°	±0.1°	
Rendezvous Errors (3σ):			
• Position	--	--	11 ± 10 N.M.
• Velocity	--	--	±15 fps
De-orbit ΔV Errors:			
• Direction (pitch & yaw)	±5°	±5°	±5°
• Magnitude	±4%	±4%	±4%

The major functions-sensor-control relationship employed by the GN&C system is described in the following paragraphs.

Prelaunch. All ESS GN&C system elements are operational during this phase (to facilitate checkout). There are two major system categories involved in the prelaunch phase: (1) ground operations and (2) vehicle functions. The ground operations relate to total checkout, mission planning, and developing data necessary to booster and ESS GN&C subsystem initialization. The vehicle functions relate to ESS initialization, checkout, and basic navigations. The basic navigation function is effected near launch phase terminations just before liftoff with an inertial measurement unit and computational equipment. Checkout is accomplished primarily with automatic on-board computational equipment with ground controllers as backup. Initialization is strictly a computational function except for alignment, which

Table 2-21. GN&C Functional Requirements
(Nonrendezvous Baseline Mission)

Mission Phase	Functional Requirements
1. Prelaunch	
Ground operations	Check out avionics system (combination of on-board and ESE).
	Load and verify flight program.
	Load targeting data and other parameters.
	Align IMU (level and gyro compass).
On-board operations	Provide on-board targeting refinement to time of launch.
	Provide navigation (after ground release).
2. Ascent	
Mated ascent	Perform powered flight navigation.
	Provide abort capability.
	Provide for main propulsion system (MPS) startups.
Separation	MPS startup commands.
	Provide attitude control and stabilization.
Boost to insertion (66 by 100 nm orbit)	Provide powered flight navigation.
	Provide ascent phase guidance.
	Provide attitude control and stabilization.

Table 2-21. GN&C Functional Requirements (Nonrendezvous Baseline Mission) (Cont)

Mission Phase	Functional Requirements
Coast to apogee	Perform coast phase navigation.
	Provide attitude control and stabilization.
	Provide for OMS startup.
Circularize orbit (100 by 100 nm)	Provide powered flight navigation.
	Provide ascent phase guidance.
	Provide attitude control and stabilization.
3. Orbital operations	
Coast in parking orbit	Provide orbital navigation.
	Provide attitude control and stabilization.
	Perform navigation updates.
	Compute transfer burn parameters.
	Compute and count time to transfer maneuver.
Boost into coelliptic transfer orbit (100 by 270 nm)	Provide powered flight navigation.
	Provide transfer phase guidance.
	Provide attitude control and stabilization.
Coast to apogee*	Provide coast phase navigation.
	Provide attitude control and stabilization.

*Midcourse correction may be needed during the ascent phase to correct out-of-plane conditions.

Table 2-21. GN&C Functional Requirements (Nonrendezvous Baseline Mission) (Cont)

Mission Phase	Functional Requirements
	Compute time to circularization burn.
	Perform navigation updates (when coverage is available).
Circularize orbit (270 by 270 nm)	Provide powered flight navigations.
	Provide transfer phase guidance.
	Provide attitude control and stabilization.
4. On-station operations	
Stationkeeping	Provide coast phase navigations.
	Provide attitude control and stabilization.
	Perform navigation updates.
	Compute burn parameters for station corrections.
	Execute burns (APCS engines)
Jettison payload	Provide coast phase navigation.
	Provide attitude control and stabilization.
	Initiate payload jettison.
Provision for cooperation docking (for shuttle docking)	Provide attitude for docking.
	Provide rate hold mode. Initiate recovery ordnance Disable attitude control at contact.
Salvage components	Initiate last release recovery ordnance and recover engines and avionic packages

Table 2-21. GN&C Functional Requirements
(Nonrendezvous Baseline Mission) (Cont)

Mission Phase	Functional Requirements
Preparation for de-orbit	Initialize deorbit system (data and positioning by shuttle).
Shuttle-ESS undocking	Provide rate damping and attitude hold mode after separation.
5. Deorbit	
Coast in orbit	Establish and maintain deorbit attitude (based on shuttle attitude initialization).
	Count time to deorbit burn (or await command through radio command).
Performance of retro burn	Initiate burn.
	Provide attitude control and stabilization.
	Measure burn time (or velocity change).
	Terminate burn.

involves IMU leveling and gyrocompassing. The prelaunch functional block diagram is shown in Figure 2-102.

Mated Boost. The boost functional block diagram is presented in Figure 2-103. The booster provides the primary mated vehicle guidance, navigation, and control functions, which are summarized as follows:

1. Navigation/guidance. Accomplished by a four-gimbal IMU with digital computation of state vector and steering commands.

2. Control. Accomplished with attitude, rate, and acceleration sensing, digital computation of control law, and electronic drive of TVC servos.

Table 2-22. GN&C Functional Requirements
(Rendezvous Baseline Mission)

Mission Phase	Functional Requirements
1. Prelaunch	(Same as for nonrendezvous mission; see Table 2-21)
2. Ascent	(Same as for nonrendezvous mission; see Table 2-21)
3. Orbital operations	
Coast in phasing orbit	Provide orbital navigation.
	Provide attitude control and stabilization.
	Perform navigation updates.
	Compute transfer burn parameters.
	Compute and count time to transfer maneuver.
Boost into coelliptic transfer orbit (100 by 260 nm)	Provide powered flight navigation.
	Provide transfer phase guidance.
	Provide attitude control and stabilization.
Coast to apogee*	Provide coast phase navigation.
	Provide attitude control and stabilization.
	Count time to circularization burn.
	*Perform navigation updates.

*Navigation updates required during ascent.

Table 2-22. GN&C Functional Requirements (Rendezvous Baseline Mission) (Cont)

Mission Phase	Functional Requirements
Circularize orbit (260 by 260 nm)	Provide powered flight navigation.
	Provide transfer guidance.
	Provide attitude control and stabilization.
Coast in parking orbit	Provide orbital navigation.
	Provide attitude control and stabilization.
	Perform navigation updates.
	Compute rendezvous burn parameters.
	Compute and count time to start rendezvous maneuvers.
Rendezvous with target	Provide rendezvous phase guidance.
	Provide powered/coast navigation.
	Provide attitude control and stabilization.
	Perform navigation updates.
4. On-station operations	
Stationkeeping	Provide coast phase navigation.
	Provide attitude control and stabilization.
	Perform navigation updates.
	Compute burn parameters for station corrections.
	Execute burns (APCS engines).

Table 2-22. GN&C Functional Requirements
(Rendezvous Baseline Mission) (Cont)

Mission Phase	Functional Requirements
Jettison of payload	
Provision for cooperative docking (for shuttle docking)	(Same as for nonrendezvous mission, see Table 2-21)
Salvage components	
Preparation for deorbit	
Shuttle-ESS undocking	
5. Deorbit	(Same as for nonrendezvous mission, see Table 2-21)

The ESS GN&C subsystem is fully powered-up and performing its own navigation function in preparation for separation.

Separation/ESS Insertion. Figure 2-104 presents the block diagram for the functions performed by the ESS during the separation/ESS-insertion mission phase and the booster during separation and return maneuver. These functions are summarized as follows:

1. Guidance/navigation. Accomplished with a four-gimbal inertial platform and digital computation of navigation equations and steering commands.

2. Control. Accomplished with electronic drive of main engine gimbal actuators and engine throttle, rate sensing, and digital computation of control laws.

Orbit Operations. Figure 2-105 presents the block diagram for the functions performed by the ESS GN&C subsystem during orbit operations, including circularization, rendezvous maneuvering, docking, and undocking, and deorbit.

1. Navigation. Accomplished with a four-axis inertial platform and digital computation. The inertial system is augmented by MSFN and/or TDRS state vector update entered via uplink.

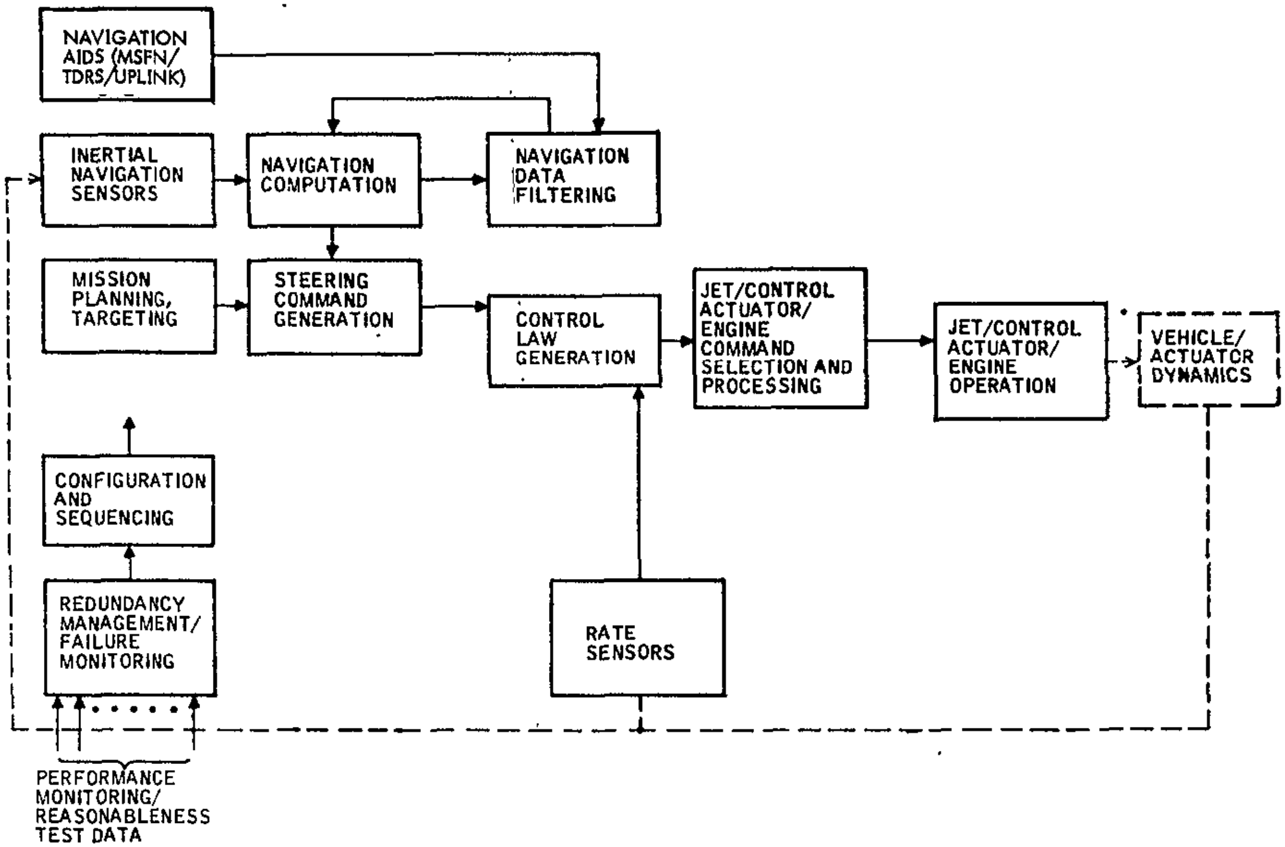


Figure 2-101. ESS GN&C Functional Block Diagram (All Phases)

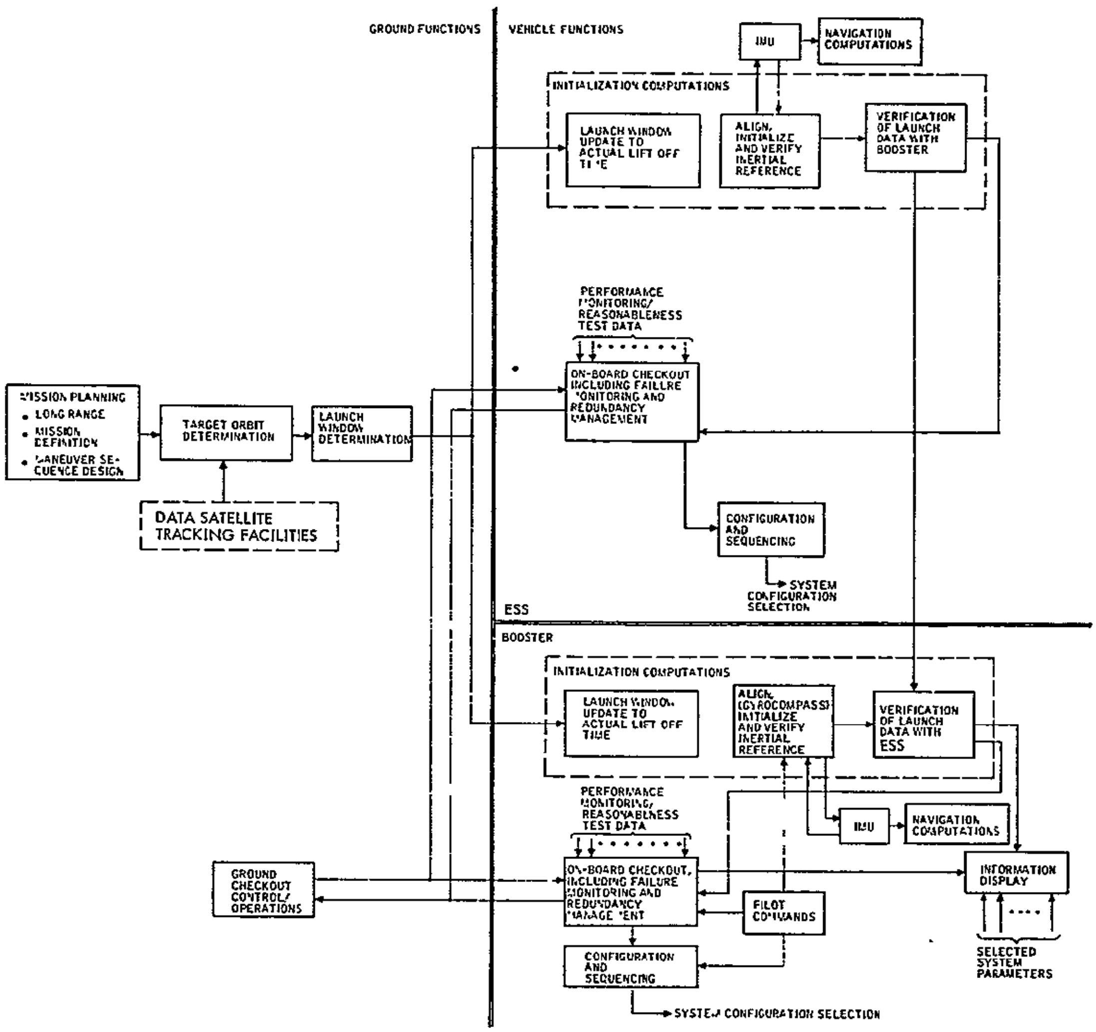


Figure 2-102. Prelaunch Phase GN&C Functional Block Diagram

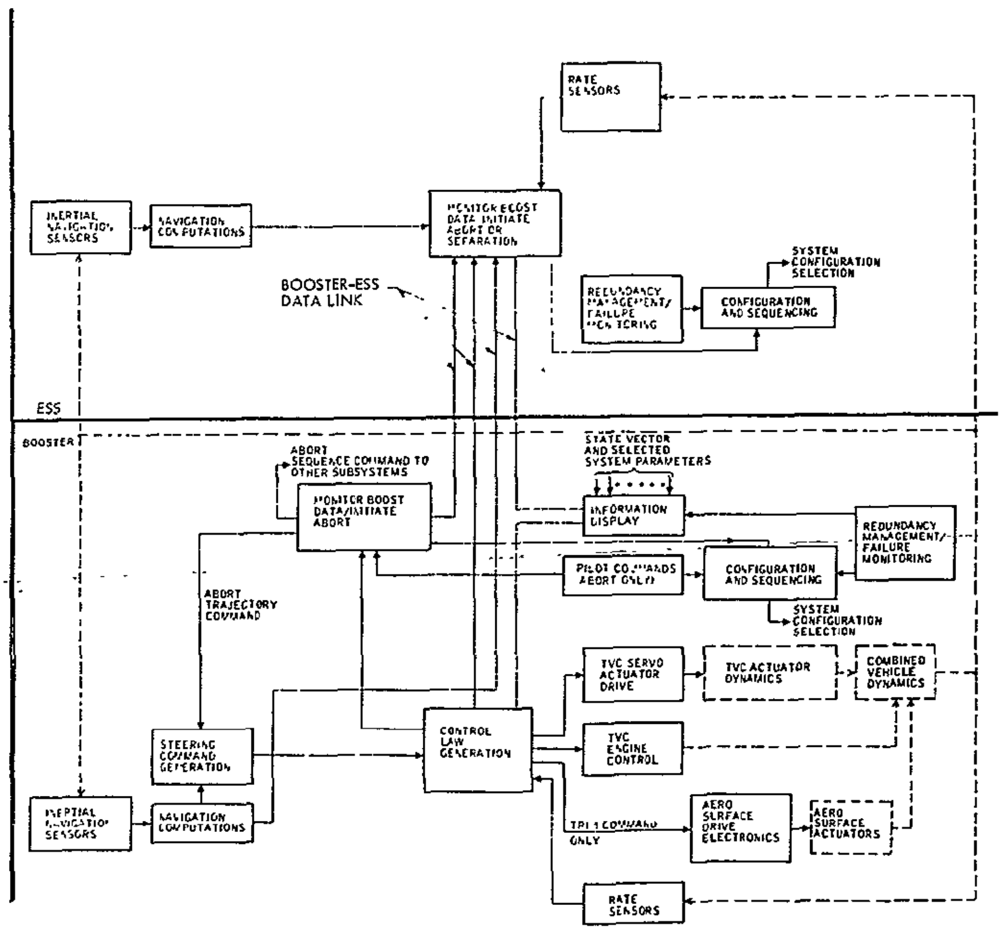


Figure 2-103. Mated Boost GN&C Functional Block Diagram

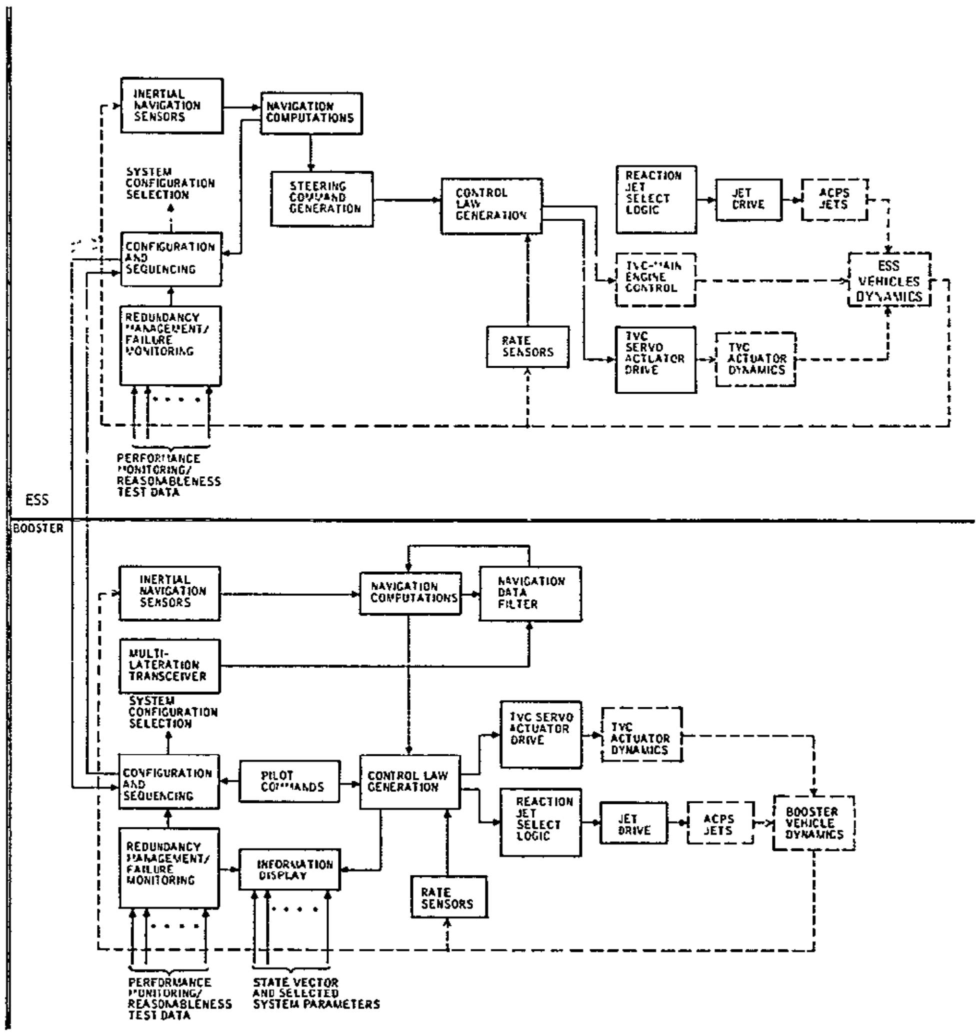


Figure 2-104. Separation/ESS Insertion/Booster Return Maneuver

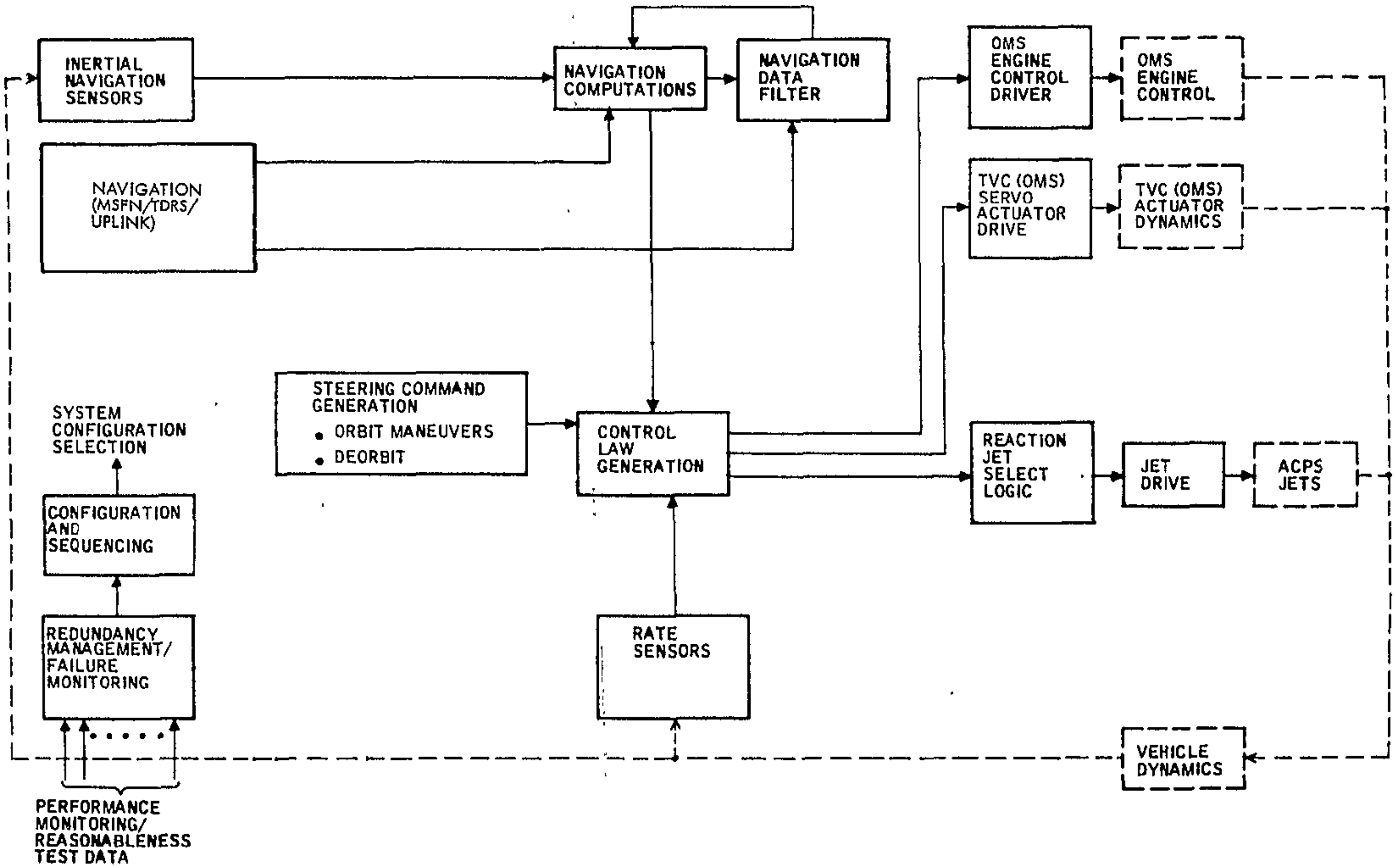


Figure 2-105. Orbit Operations (Including Rendezvous and Stationkeeping)

2. Guidance. Accomplished with digital computation.

3. Control. Accomplished with digital control law computation, rate sensing, and electronic drive of attitude control jets (non-thrusting periods) and OMS gimbal actuators (thrusting periods).

Baseline GN&C Subsystem Hardware Description. The GN&C subsystem is comprised of the following assemblies:

Inertial measuring unit (IMU)

Rate sensor assemblies (RSA)

Attitude/translation driver units (ATDU)

TVC gimbal servo amplifier (TGSA)

Significant GN&C interfaces are as follows:

Data and control management subsystem (DCM), including computers and data bus

OMS and ACPS H_2/O_2 valve actuators and ignition exciters

Main and OMS engine gimbal actuators

MSFN and/or TDRS tracking

Uplink (state vector update)

A GN&C functional block diagram, including significant interfaces, is shown in Figure 2-106. Installation design and thermal considerations are provided in Paragraph 2.3.7 of this section of the report. The following paragraphs describe the hardware elements that comprise the ESS GN&C subsystem. All GN&C hardware is identical with space shuttle hardware. Minor modifications to engine gimbaling and driver units may be required to interface with a separate deorbit subsystem discussed in Section 2.3.8.

Inertial Measuring Unit. The IMU consists of two line replaceable units (LRU's); the platform and the power supply are shown in Figure 2-107. The platform has four gimbals with appropriate synchros, resolvers, and torque motors for each gimbal. The angular sequence, starting with the inner gimbal, is pitch, roll, and yaw; the fourth gimbal provides redundant roll. The stable element (inner member) contains two 2-degree-of-freedom (DOF) gyros with spin rotation axes directed along the platform pitch and yaw gimbal axes, respectively. One gyro controls the roll and yaw platform

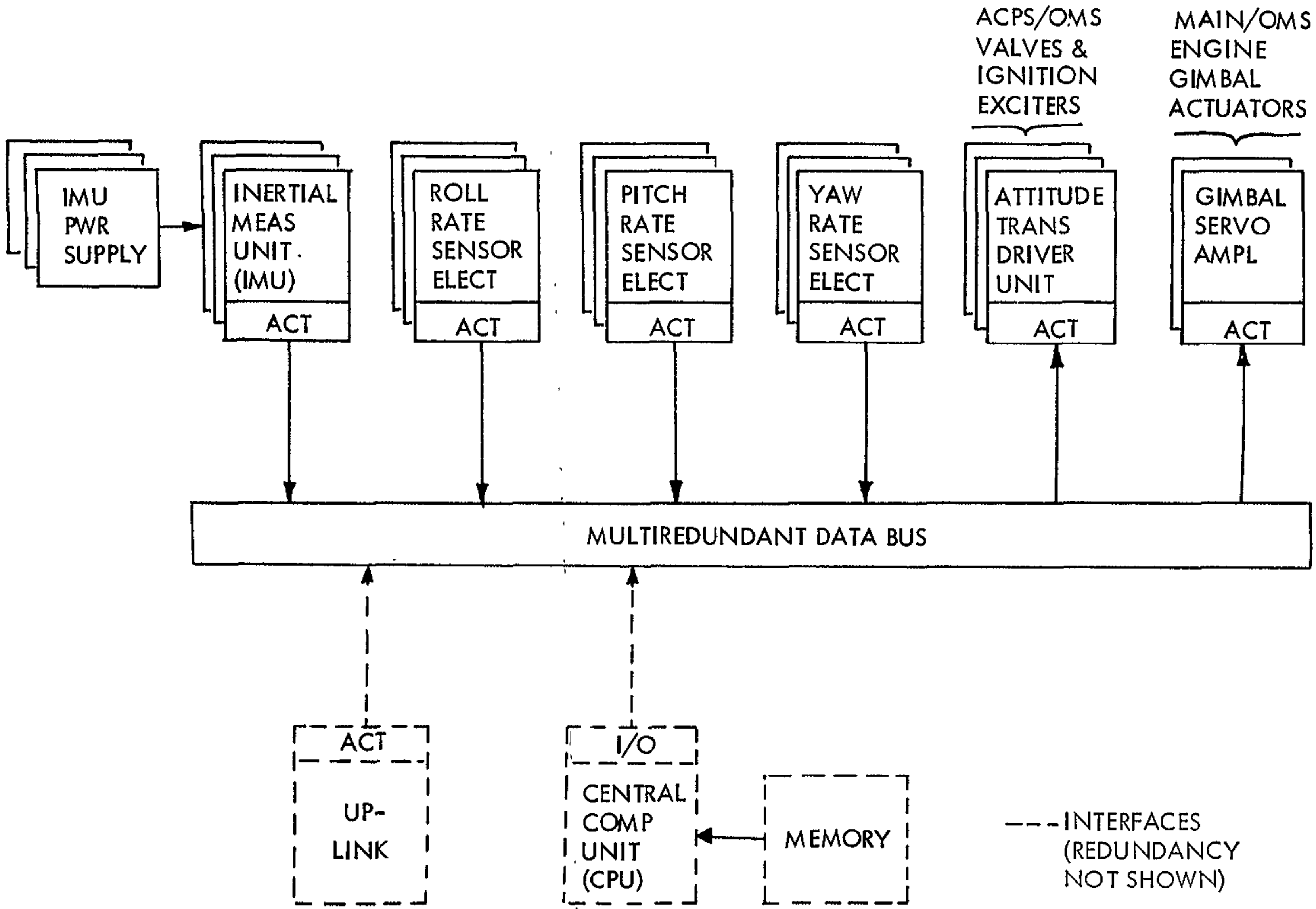


Figure 2-106. Baseline GN&C Subsystem

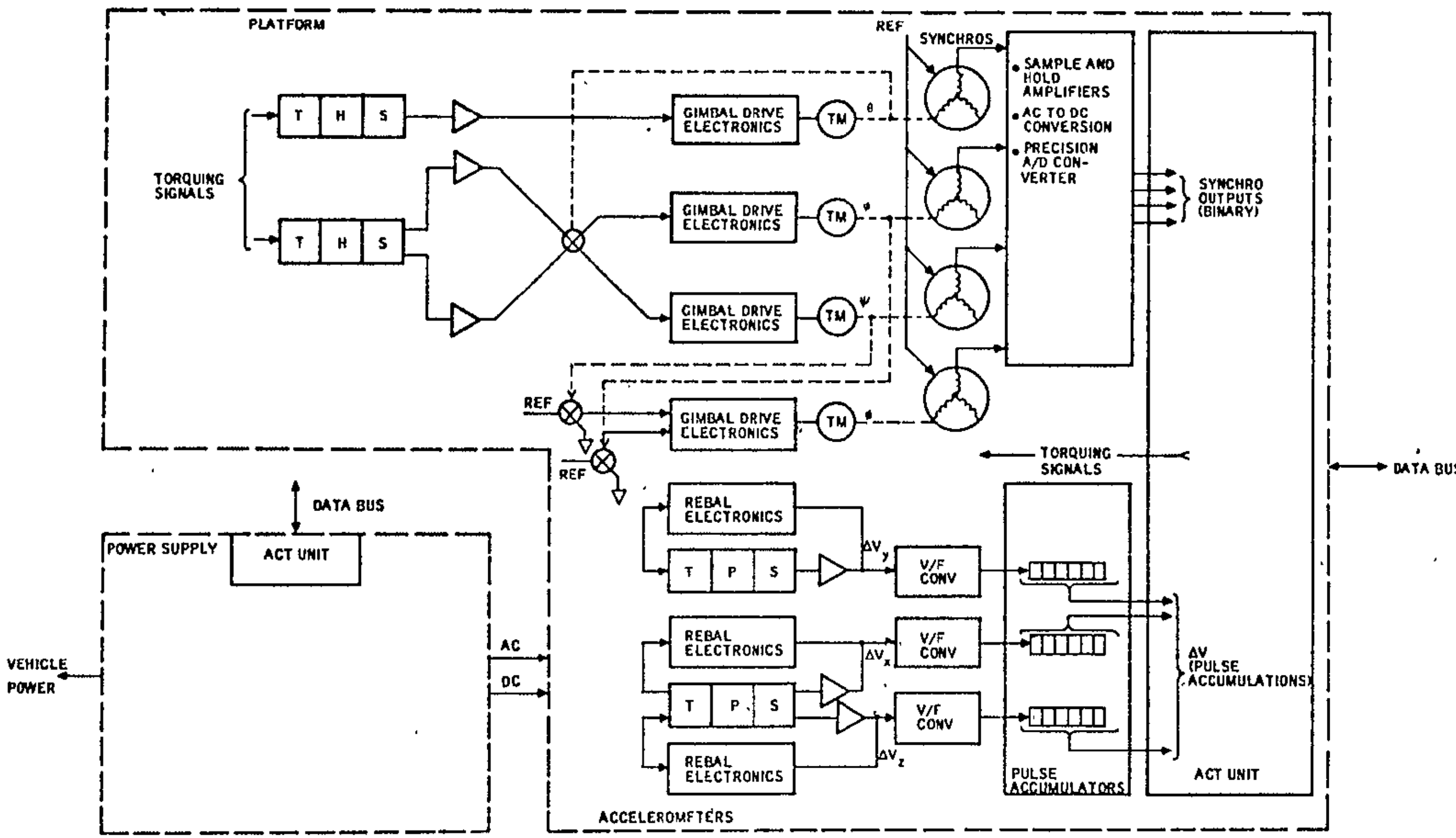


Figure 2-107. IMU Block Diagram

gimbals while the other controls the pitch gimbal. The inertial platform baseline is typified by a Kearfott KT-70A7. (Refer to Singer-Kearfott Division Document ETO-1075, "KT-70 Inertial Measurement Unit for Fighter Aircraft Applications", 30 July 1970.)

The accelerometers, also mounted on the stable member, consist of a two-axis accelerometer, measuring accelerations in the X and Z axis, and a single-axis accelerometer measuring Y-axis accelerations. This definition applies when the gimbal angles are driven to zero in all axes with respect to the vehicle body axis system. The coordinate system X, Y, Z, defined in the conventional sense, correspond to the roll, pitch, and yaw axes of the vehicle. The gimbal torquing electronics and the accelerometer rebalance electronics are located in the platform assembly.

The platform includes a pulse accumulator circuit to accumulate the accelerometer pulses and convert to a binary format, a precision A/D converter to convert synchro outputs to a binary format and an ACT unit to provide multiplexing and input/output control. The platform ACT unit transfers the synchro angles and accumulated delta-V pulses onto the data bus with the proper format. It also accepts gyro torquing commands from the data bus for realigning the platform.

The power supply accepts the 28-vdc and 115-vdc, 400 Hz ac power from the vehicle power to dc and ac levels compatible with the sensor spin motors, electronic circuits, and synchro reference requirements.

Rate Sensors. The rate sensor proposed for ESS is the same as baselined for both shuttle vehicles and is the existing Honeywell GG1102 vibrating-wire sensor. This choice was dictated by its very low failure rate (1.3 percent/1000 hours) as well as life in excess of 10,000 hours. Other unique features include undetectable hysteresis and mass unbalance, tight output linearity and flat response to 100 cycles.

A block diagram of the rate sensor is shown in Figure 2-108. An enabling signal applies 28-vdc line voltage to the dc power supply, providing bias voltages and oscillator excitation. The oscillator provides a high-frequency current exciting both input and output wires. A rate of turn ω on the input axis induces a coriolis force on the output wire causing it to vibrate elliptically. This motion change in a permanent field induces a signal proportional to ω. The rate signal is preamplified and demodulated, and dc amplification is added to provide an appropriate scale factor.

An external rate simulator signal can be applied for checkout, and checkout/fault isolation (COFI) signals fed back include oscillator and power supply monitors. One ACT unit is integrally packaged with each rate sensor unit for interface between sensor electronics and data bus.

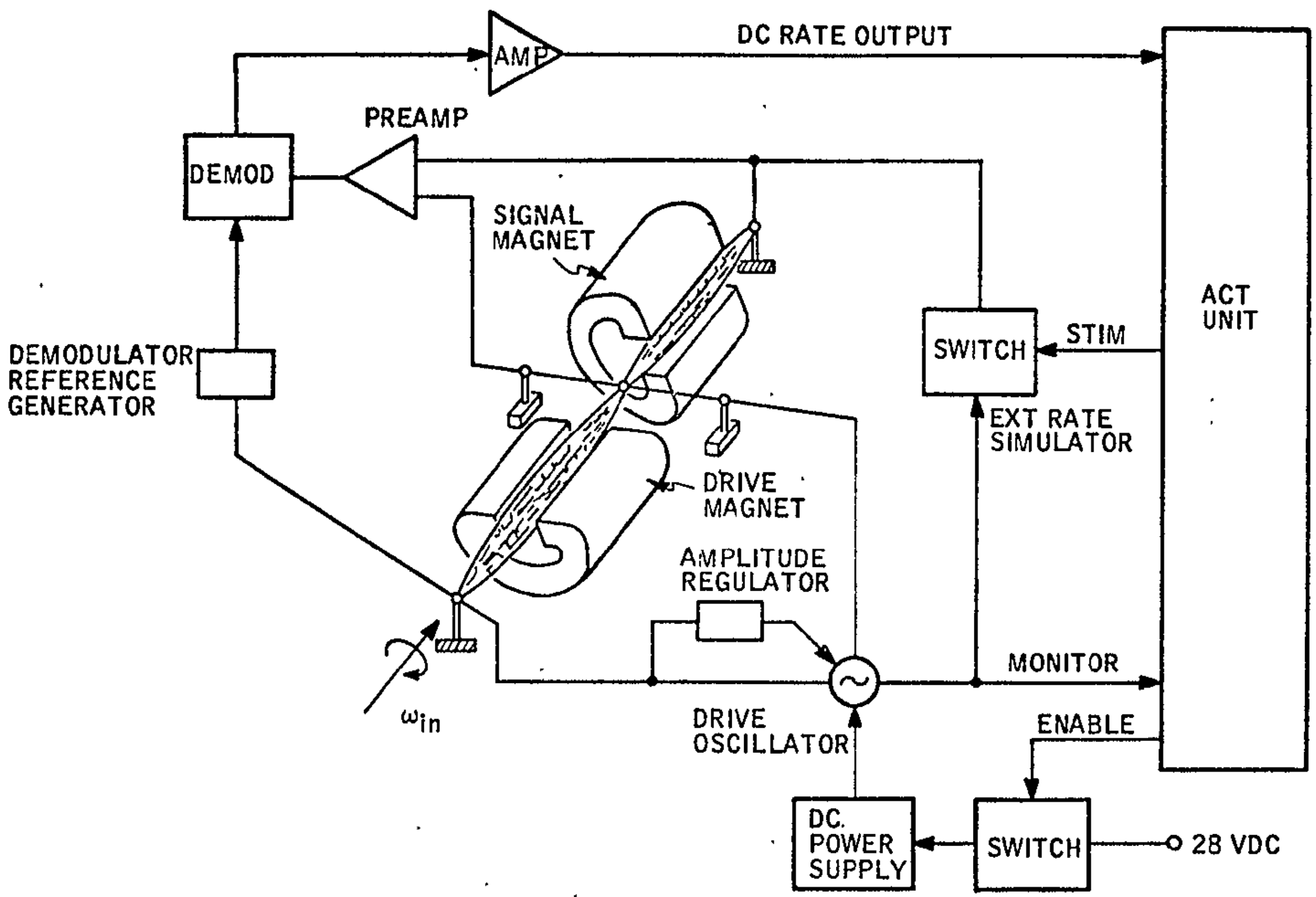


Figure 2-108. Rate Sensor Schematic Diagram

Nine such devices are required for each vehicle, three each on the vehicle pitch, yaw, and rate axes. Only minor scaling differences are anticipated between the three line replaceable unit (LRU) types. The rate sensors are presently located in the thrust cone location with the bulk of avionics equipment. Flexible body considerations may dictate another location when model analyses are available.

Flexible body control analyses may show that separate rate sensors may not be required for stability. Should separate rate sensors be required, less expensive rate sensors might be selected, based upon the shorter length of the mission (24 hours) and the fact that triple redundancy is dictated by the FO/FS criteria.

The physical design of the LRU will be characterized by sealed, rigid construction, and since these are heaterless sensors, heat sink limiting of device temperature is required. Breakup into three separate LRU's per axis was dictated by the shuttle spatial diversity guideline, but they may be combined into one LRU per axis as an optional configuration for ESS application.

Attitude/Translation Control Driver Units. The mechanization of a typical N-engine ACPS/OMS driver LRU is illustrated in Figure 2-109. In the current system mechanization, these LRU's are available in two configurations, driving two or four engines. An "enable" signal for each engine control channel, provides ac and dc power to the channel activated. Subsequent discrete thrust commands provide either direct input to the predriver, or minimum impulse input via a one-shot circuit. A predriver circuit includes input gating and power gain to the PNP driver transistor, which provides on-off control of current to the H_2 and O_2 engine valve solenoids. Separate hard-wire connections are made to the two valve solenoids to facilitate COFI.

Transient suppression circuitry is provided in each channel to reduce to a manageable level the inductive kick generated by a coil current turnoff. A separate ignition signal feeds a second 28-vdc driver circuit to provide a low-power discrete signal to the ignition exciter, which is part of the engine system.

Interface with the data bus is provided by ACT units; three for the four-engine LRU and two for the two-engine configuration. Each ACT unit provides four discrete signals to each driver channel:

Channel enable

Valve command — normal

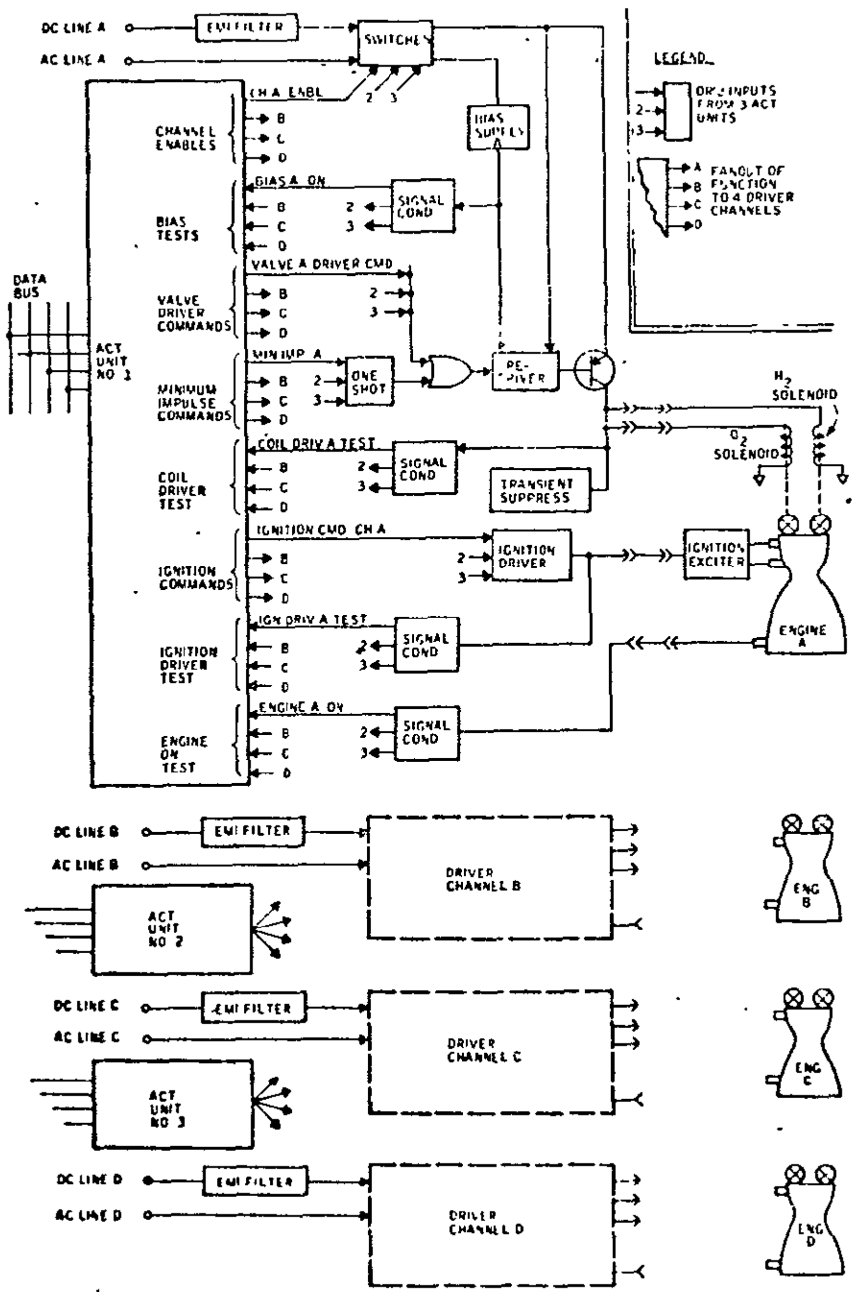


Figure 2-109. Typical ACPS/OMS Engine Driver Mechanization (4-Engine Configuration)

Valve commad — minimum impulse (ACPS only)

Ignition on command

In addition, each ACT unit accommodates four discrete COFI signals from each engine control channel to confirm the following:

Channel bias voltage

Coil driver on

Ignition signal on

Engine on

Growth capability permits addition of at least four more COFI functions per engine control channel.

Multiple isolated dc and ac power lines are required for redundancy management and fault tolerance. Each ac line serves a separate bias supply, and each dc line is provided with an electromagnetic interference (EMI) suppression filter. Four lines are required into each four-engine LRU and two each into a two-engine LRU.

Since the loss of any one of the ACPS or OMS engines allows continued mission operations, and since a second engine failure would not cause an unsafe condition, only one driver channel per engine is required for FO/FS operation. Because of the location of the 14 ACPS engines in one group of seven on each side of the vehicle, two four-channel units will be used with each seven-engine group and one two-channel unit will be used for the two OMS engines.

TVC Gimbal Servo Amplifiers (TGSA). The mechanization of the proposed main engine and OMS gimbal servo amplifiers is shown in Figure 2-110. It is identical to those proposed on shuttle. Each control servo uses a torque motor input and linear variable differential transformers (LVDT) provide loop feedback from the actuator. Each of three ACT units interfaces with each of eight gimbal servo channels. A "TVC enable" function provides amplifier bias and transducer excitation. Gimbal position commands are summed with gimbal position feedback to produce an amplified gimbal error signal to the actuator torque motor. An active network is presumed to be required for loop stabilization in the position feedback path. Conditioned test signals are provided for error amplifier output, actuator position, actuator temperature, and differential pressure.

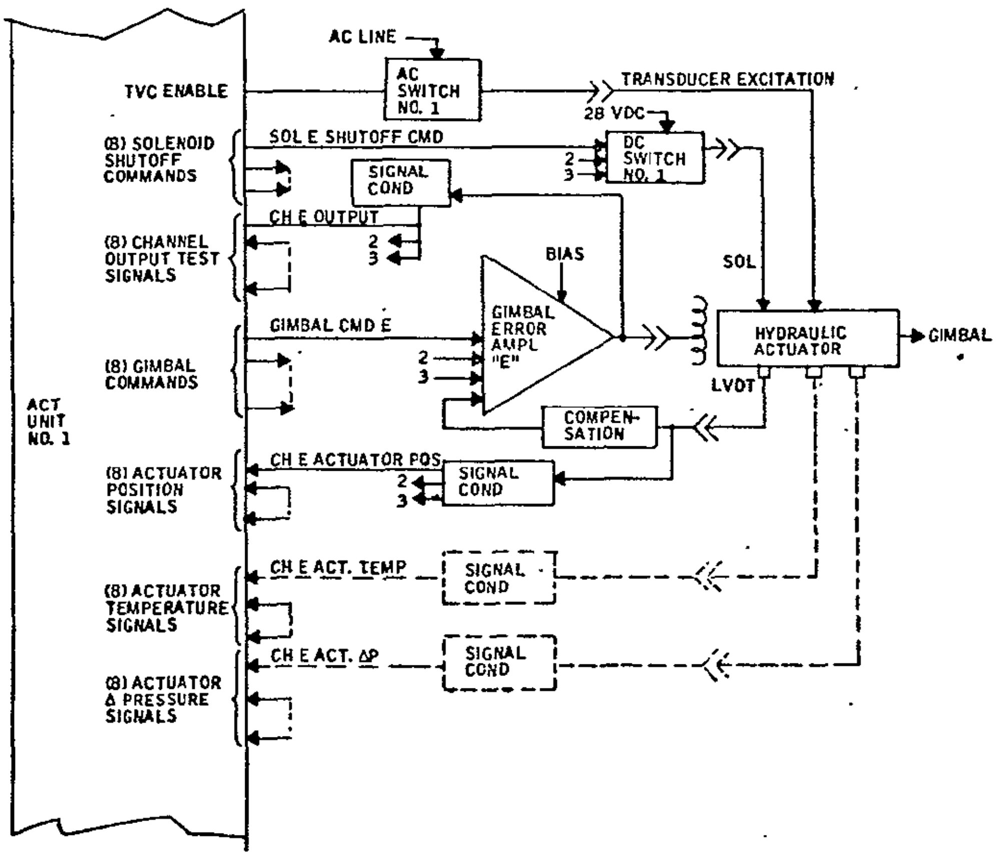


Figure 2-110. Typical TVC Gimbal Control Channel

For gimbal control, a dual-section position servo actuator is used. Each section consists of two channels, one active and a dead-ended monitor. In the section carrying load, a difference sensed between its two channel outputs is regarded as a failure, diverting control to the second section. A difference detected between its monitor and active channels is regarded as a second failure and causes the gimbal to center for fail-safe. Each of the four channels requires an electrical servoamplifier controlling its torque motor input. Two 28-vdc solenoids provide an initial activation function in providing supply pressure to the diverting and bypass valves. Position feedback is provided by LVDT's operating off the output piston shaft. Two main engines and two OMS engines require 16 channels of electronics, if dual-section actuators are used for both.

Two shuttle orbiter LRU's containing eight thrust vector control channels each will be used on ESS. The LRU design is characterized by sealed cases, rigid construction and conduction thermal paths which facilitate hard mounting to a heat sink to allow remote locations subjected to severe temperature, vibration, and vacuum environments.

GN&C LRU Characteristics Summary. All GN&C LRU's and their significant characteristics are summarized in Table 2-23.

GN&C Trade Studies and Performance Analysis

This section contains detailed analyses in the following areas:

Guidance and navigation error analysis

Phasing analysis

Other missions

Synchronous orbit considerations

Close rendezvous considerations

These analyses were performed by the IBM Corporation in support of the ESS GN&C subsystem.

Supporting control analyses and studies are covered earlier in Section 2.2. Recovery of GN&C equipment and associated GN&C deorbit considerations are covered in Paragraph 2.38 of this report. Deorbit analysis relative to GN&C requirements is covered in Paragraph 2.3.3. The ESS GN&C subsystem recommended configuration and requirements are based on these analyses.

Guidance and Navigation (G&N) Error Analysis. A G&N error analysis was performed during this study that encompassed both non-rendezvous baseline and rendezvous missions. The key issues were to determine the navigation and attitude update requirements.

The first task was to determine if navigation and/or attitude updates were necessary and, if so, to determine the best equipment and technique to recommend.

The following data were used in conducting the analysis:

1. ESS Trajectory

   a. Updated nominal ESS ascent trajectory (MDAC space station payload)

   b. Launch azimuth of 40 degrees from KSC

2. Tracking Coverage

   a. Vehicle state vector updated once every ten seconds for the total station coverage time (station acquisition at 5 degrees elevation).

   b. Manned space flight network (MSFN)

      (1) Existing stations

      (2) Four station networks expected to be operational in the 1980's:

         Kennedy Space Center

         Madrid

         Honeysuckle

         Goldstone

   c. Shuttle selected multilateration beacon locations are as follows:

      Kennedy Space Center, Florida

      Columbus, Mississippi

Table 2-23. LRU Characteristics Summary

LRU	No. per Vehicles	Functions	Content of Each LRU	Redundancy	Functional Interfaces	Weight (lb per LRU)	Volume (in.^{3} per LRU)	Power (watts per LRU)	Remarks
Four gimbal inertial measurement unit (IMU) platform	3	Provides incremental measurement of vehicle velocity Provides vehicle attitude information.	2 2 DOF gyros 2 accelerometers (one 2-axis, one 1-axis) 3 accelerometer rebalance electronics 4 gimbal drive electronics 1 A/D converter 2 temperature control electronics 1 ACT unit	FO/FS	Vehicle motion Data management processor IMU power supply	25	785	70	Requires alignment to within 30 arc-minutes of vehicle axes.
IMU power supply	3	Accepts ac and dc power from vehicle bus and converts it to type and levels required by IMU platform	2 heater power supply 1 synchrd power supply 1 fan power supply 3 regulated dc power supplies	FO/FS	IMU platform Vehicle power bus	22	600	50	
Body rate sensor (pitch)	3	Provides body rate data at selected vehicle station.	1 rate sensor 1 exciter/oscillator 1 signal conditioner 1 ±15 vdc regulated power supply 1 ACT unit	FO/FS	Vehicle motion Data management processor	2.2	45	3.9	Utilizes GG1102 vibrating wire rate sensors
Body rate sensor (yaw)	3	-	-	-	-	-	-	-	-
Body rate sensor (roll)	3	-	-	-	-	-	-	-	-
TVC gimbal servo driver unit	2	Provides proportional control of electro-hydraulic actuator valves for main/OMS engine gimbal actuators.	8 TVC servo loop electronic channels 1 bias supply 1 ACT unit		Data management processor Main/OMS engine gimbal drive and feedback	6.5	188	18	Drives two actuators per axis on each of two LSS main or OMS engines.
Attitude/translation driver unit, four jet	4	Provides control of valve solenoids (H_2/O_2) and provides ignition control.	4 ON-OFF driver stages 4 transient suppression circuits 4 ignition circuits 4 bias supplies 3 ACT units	FO/FS driver stage and ignition are single.	Jet valve solenoids Jet ignition exciters OMS engine valve solenoids OMS engine ignition exciters Data management processor	14	390	12	Standby power, actual firing higher, 180 watts per jet.
Attitude/translation driver unit, two jet	1	Provides control of valve solenoids (H_2/O_2) and provides ignition control.	2 ON-OFF driver stages 2 transient suppression circuits 2 ignition circuits 2 bias supplies 2 ACT units	F/O dual, except driver stage and ignition are single.	Jet valve solenoids Jet ignition exciters Data management processor	8.6	238	7	

Bergstrom AFB, Texas

Biggs AFB, Texas

Darwin, Australia

N Djili, Congo

Honolulu AFB, Hawaii

Nandi Intl., Fiji Island

La Lontana, New Caldonia

Ramey AFB, Puerto Rico

Kadena AFB, Ryoko Island (Okinawa)

San Nicolas Island Colf., California

d. Tracking data relay satellites (TDRS). Three satellites at synchronous altitude over longitudes 16°W, 88°W, and 180°. This configuration allows all three satellites to communicate directly with ground stations in the continental USA.

3. On-Board Navigation Hardware Considered

a. IMU (Kearfott KT-70). Selected reference platform for NR Space Shuttle Phase B studies.

b. Star Tracker (ITT Aerobee 150A). Selected reference star tracker for NR Space Shuttle Phase B studies.

c. RF Beacon (CUBIC DM 100). Selected reference tracker for NR Space Shuttle Phase B studies.

4. Analytical Tools

a. ANS. The autonomous navigation simulation program simulates autonomous navigation along a Keplerian orbit as performed by a Kalman filter, using data selected combinations of horizon sensor, star tracker, landmark tracking telescope, radar altimeter, range finder (laser), and ground station tracking measurements. This program effectively models the shuttle navigation routine.

b. IPEP. The inertial platform error program utilizes the normalized integral technique developed by George R. Pitman, (author of Inertial Guidance) to propagate platform hardware errors during boost phases of flight.

c. MIRCUS. This program is a six-demensional orbital simulator that generates orbital trajectories very accurately out to 24 hours ground elapsed time. The simulation is in single precision and utilizes a Runge Cutta third-order integration routine, a PRA 63 reference atmospheric model, and a gravational potential model of an oblate spheroid. The program calculates and prints the orbital trajectory parameters, tracking station look angles, and station acquisition and loss times.

Nonrendezvous Mission G&N Error Analysis. The following missions have been considered:

Baseline mission. Ascent to 66- by 100-nm orbit, parking orbit circularization burn (100-nm circular parking orbit phasing and navigation updates, 100- by 270-nm transfer burn, and 270-nm circular injection burn).

Direct ascent. Ascent to 66- by 270-nm orbit and 270 nm circular injection burn.

Baseline Mission. The organization of this section is as follows:

Boost to 66- by 100-nm orbit

Parking orbit insertion (100-nm circular)

Parking orbit coast phase analysis

Multilateration updates

MSFN updates

TDRS updates

Transfer to 270-nm orbit

a. Ascent to 66- by 100-Nautical Mile Orbit. The ESS ascent trajectory obtained from NR was used as an input to the IPEP program along with the reference platform errors (Kearfott KT-70) in order to conduct the boost error analysis. The platform errors used were (1) azimuth alignment — 360 seconds (1 σ) and (2) drift — 0.02 degree/hour (1 σ).

The results of the ascent navigation error analysis, which used as inertial platform only, is as follows (with shuttle requirements shown for comparison):

Parameter		Simulation Results (N) at 0.66 by 100- nm Orbit Insertion (1 σ)	Shuttle Requirement (1 σ)
Position	Radial (R)	0.21 nm	0.5 nm
	Tangential(T)	0.3 nm	Not critical
	Normal (N)	1.35 nm	2.0 nm
Velocity	R	10.0 fps	25 fps
	T	4.26 fps	6 fps
	N	47.5 fps	75 fps

The large normal (cross-range) position and velocity errors are due principally to the IMU azimuth alignment technique commonly referred to as "gyro compassing." This technique is the same one proposed for the shuttle.

b. Parking Orbit Insertion (100-Nautical Mile Circular). The inertial platform errors at 66- by 100-nm orbit were propagated to parking orbit insertion and the radial (R), tangential (T), and normal (N) errors were as follows:

Parameter		Simulation Results (1 σ)
Position Error	R	1.47 nm
	T	6.5 nm
	N	1.35 nm
Velocity Error	R	39.4 fps
	T	7.78 fps
	N	47.00 fps

c. Parking Orbit Coast Phase (100-Nautical Miles Circular). The initial state vector uncertainties for the Kalman filter in the ANS program. The errors were then propagated through 15 parking orbit revolutions which encompass the maximum possible phasing. Figure 2-111 is a plot of the RSS navigation position error when inertial navigation only during parking orbit phasing is used.

Figure 2-111 illustrates that using only inertial navigation during parking orbit results in very large navigational errors. In addition, if the orbit transfer burns were initiated at the most opportune time (parking orbit insertion), the propagated navigation errors at insertion into 270-nm circular would be in excess of 150 nm. Therefore, the conclusion is reached that navigation updates are necessary in parking orbit to reduce the inertial navigation error at insertion to an acceptable magnitude.

Space Shuttle trade studies have investigated numerous autonomous and ground-assisted navigation update techniques. Among these are RF beacons (multilateration), MSFN tracking, horizon/star tracker sightings, and manned landmark tracking. The trade study results for the shuttle selected the RF beacon approach, and it will be considered here along with MSFN and TDRS.

The ANS program was used to investigate the navigation update accuracies with the following results:

(1) Multilateration navigation updates (100-nautical mile circular). An analysis of parking orbit navigation accuracy was performed by using RF beacons for various navigation updates rates. Figure 2-112 depicts the results of that analysis, showing that the initial navigation uncertainties can be reduced to 0.05 nm. The rate of navigation uncertainty reduction is dependent on the station coverage available. The coverage available during parking orbit is shown in Table 2-24. The multilateration coverage indicates that the steady-state navigation position error of 0.05 nm can be achieved in less than two orbits.

(2) MSFN navigation updates (100-nautical mile circular). An analysis of parking orbit navigation accuracy was performed by using MSFN tracking. Figure 2-113 depicts the results of that analysis.

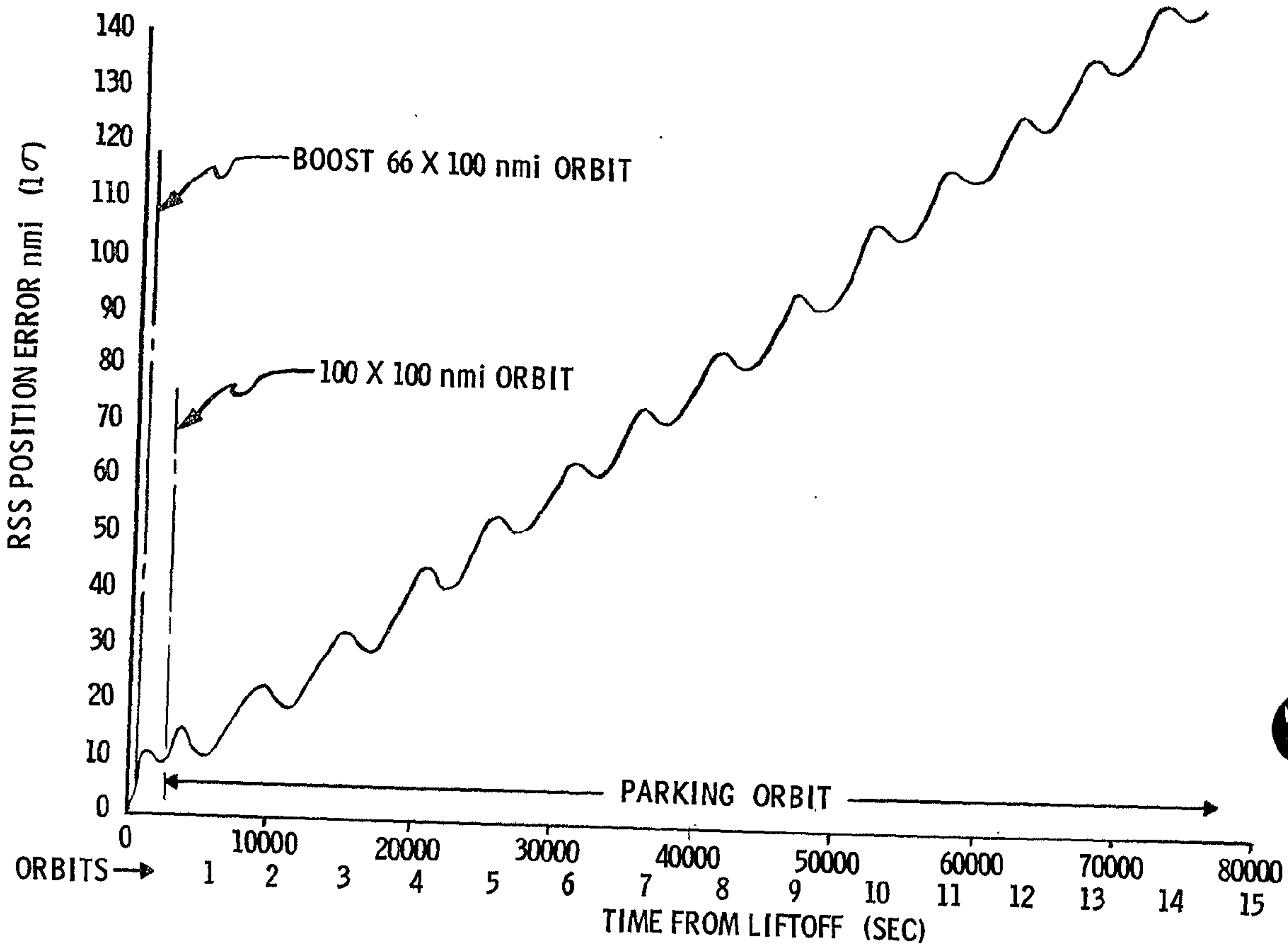


Figure 2-111. Inertial Error Propagation With No Navigation Updates

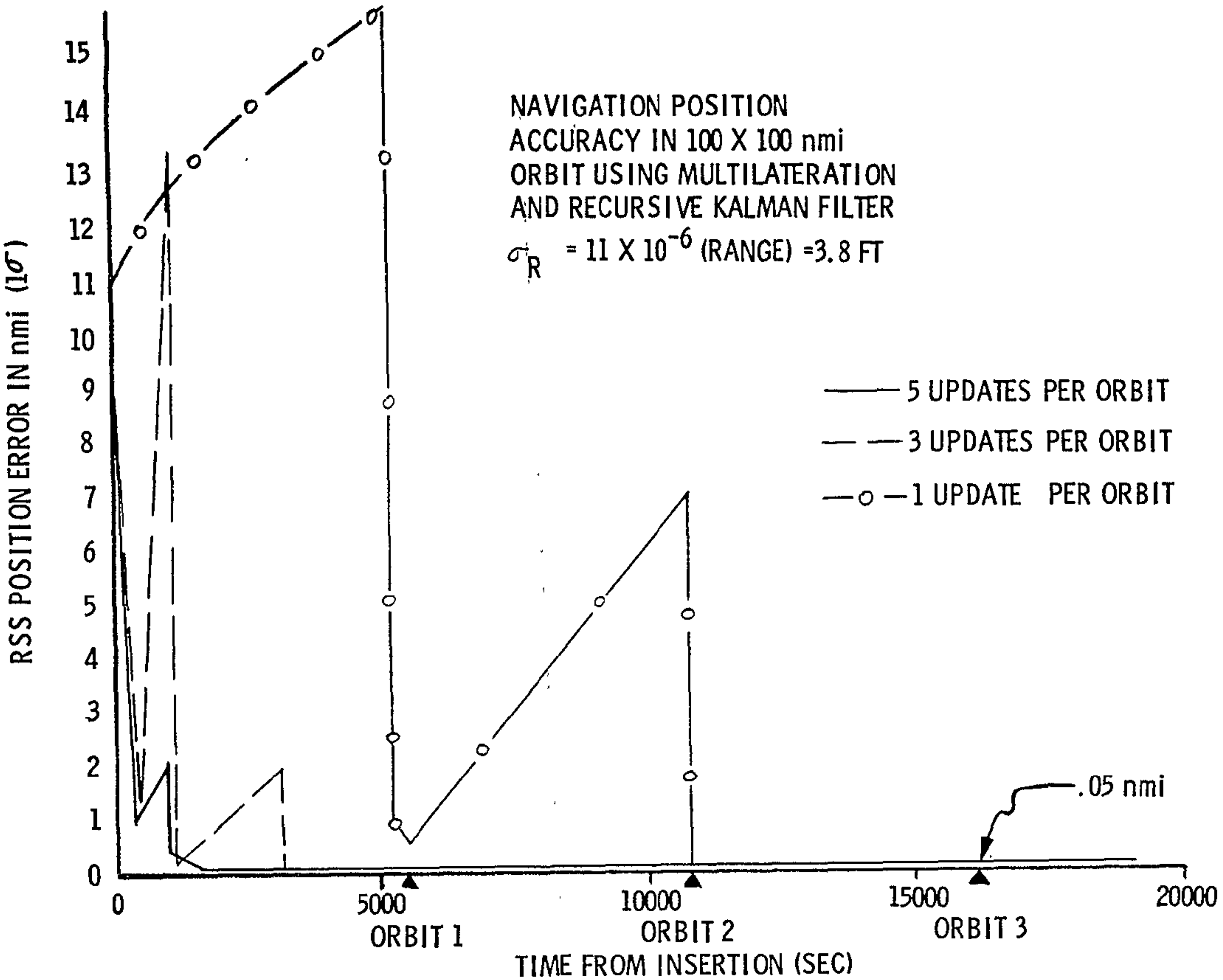


Figure 2-112. Navigation Position Accuracy Versus Time for Multilateration Update

Table 2-24. MSFN, Multilateration, and TDRS Tracking Coverage for ESS 100-NM Parking Orbit (Inclination = 55 Degrees, Azimuth = 40 Degrees; Launch Site = KSC)

Orbit	*MSFN (Eight Stations)	**MSFN (Four Stations)	Multilateration	TDRS (percent)
1	2	1	2	3-91.7
2	2	2	2	3-91.7
3	3	1	4	3-91.7
4	3	1	3	3-91.7
5	4	1	2	3-91.7
6	4	2	6	3-91.7
7	2	1	4	3-91.7
8	0	0	1	3-91.7
9	1	0	1	3-91.7
10	1	0	1	3-91.7
11	1	0	2	3-91.7
12	2	0	3	3-91.7
13	3	2	3	3-91.7
14	3	2	1	3-91.7

*Existing MSFN stations
**MSFN stations projected for the 1980 time period

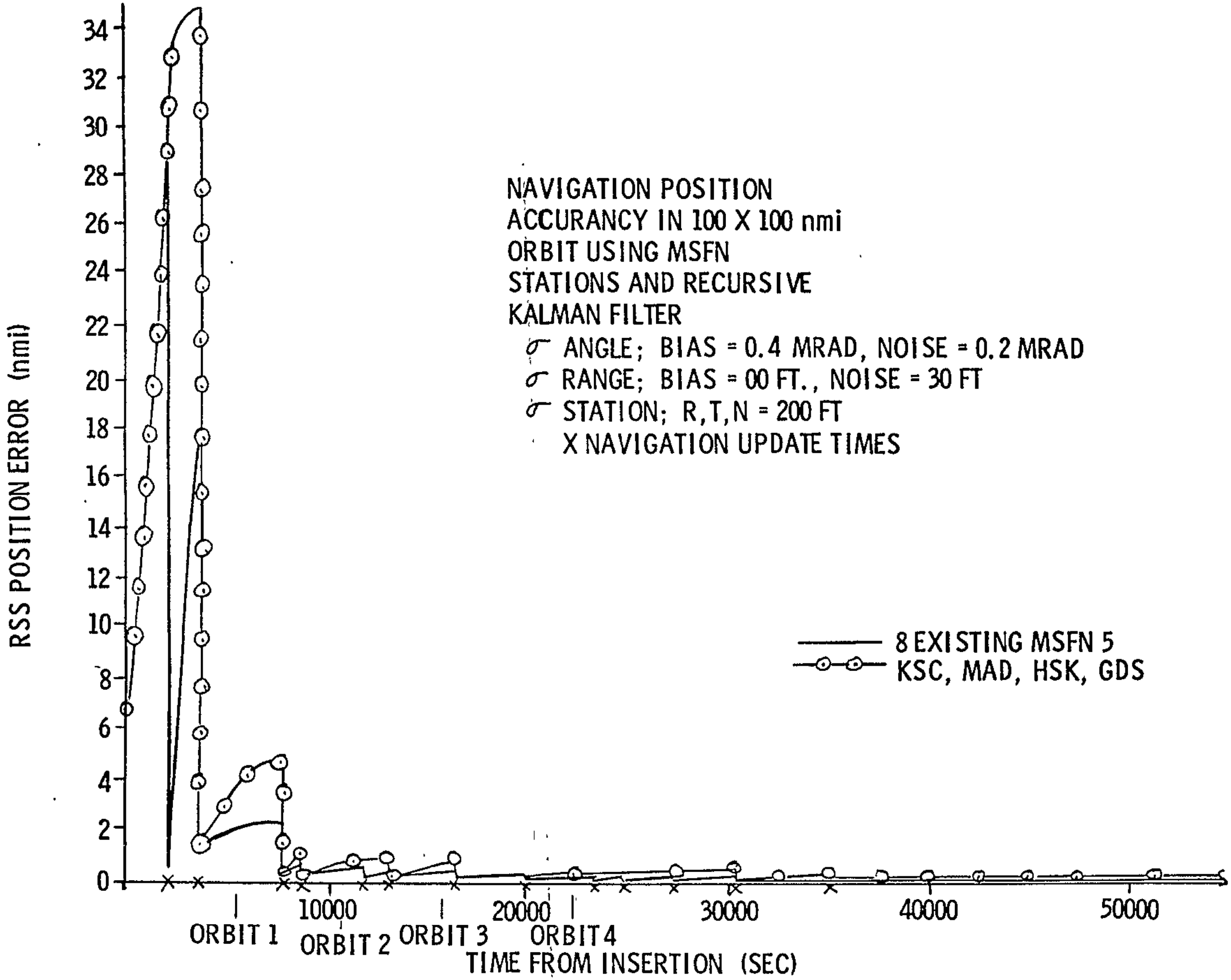


Figure 2-113. Navigation Position Accuracy Versus Time for MSFN Tracking Updates

The figure indicates that a navigational error of 1 nm can be achieved in less than two orbits by using all the available MSFN tracking stations. For the reduced tracking facilities expected to be available for ESS missions, about three orbits would be required to achieve the same result. By appropriate mission planning, the transfer burn to 270 nm can be made immediately after a navigation update with increased accuracy and in a shorter time.

(3) Navigation by using tracking and data relay satellite system (TDRS) (100-nautical mile circular). Concepts of utilizing synchronous satellites for tracking and data relay purposes have been suggested by Dr. F. O. Vonbun of the Goddard Space Flight Center. This TDRS system would employ geostationary synchronous spacecraft to take over the functions of tracking, command, and relay of data from multiple low earth-orbiting satellites to a few centrally located ground stations. The satellites would be located to allow all three TDRS satellites to communicate directly with ground stations in the continental USA.

The principal advantage of synchronous satellite tracking is the increased coverage of low earth-orbiting spacecraft obtainable.

An error analysis study was made by Goddard Space Flight Center to determine the ability of a network of three synchronous satellites to track vehicles in low earth orbit over long arcs. Only one TDRS was allowed to track the vehicle at a time, switching being done between satellites to give better geometry. One proposed system configuration has the tracking link origination at the main ground control center. The signal from the main ground station is transmitted to a TDRS, relayed to the vehicle, and then returned to the ground station via the same link.

The assumptions made in the performance of the study were present tracking uncertainties, present station location uncertainties, and one measurement every ten seconds.

The 100-nm parking orbit determination uncertainties associated with a system of this type are shown as follows:

Tracking Time (hr)	Radial Component of Position (ft)	Position (ft)	Velocity (fps)
2	1000	2000	2.0
4	130	650	0.55
8	115	650	0.7
12	82	850	0.8
16	210	1800	1.4
20	230	1500	1.1
24	380	1500	1.1

The accuracy of this system is comparable to MSFN ground station tracking accuracies, but has the added benefit of increased coverage.

If the orbits of the low altitude satellite and the three TDRS's are solved simultaneously by using a Kalman filter, the improved estimates of their orbits will result in an even greater TDRS tracking accuracy potential.

d. Transfer-Burn Phase (100 by 270-Nautical Miles and Circularization at 270-Nautical Miles). The transfer burn from 100-nm parking orbit to 270-nm orbit was assumed to occur after approximately 1-1/2 orbits at 100-nm by using MSFN navigation updates.

The selection of one and one-half orbits and MSFN update techniques was based on the following rationale:

(1) "Look ahead" analysis indicated that MSFN navigation updates would allow the ESS to meet mission requirements and appears to result in minimum onboard equipment for ESS.

(2) One and one-half orbits result in a minimum instantaneous navigation error. (A greater number of orbits results in a lower average error but this average is offset by the larger inertial alignment errors encountered).

The navigation errors before the transfer burn under the above conditions are:

Position	Navigation Errors (1 σ)
R	320 ft
T	370 ft
N	360 ft
Velocity	
R	0.53 fps
T	0.37 fps
N	0.42 fps

These errors were used as initial conditions for the 100- to 270-nm orbital transfer analysis. The results of this analysis are shown in Figure 2-114, and the navigation insertion errors at 270 nm are shown as follows:

Parameter	Navigation Errors (1 σ) Without Update	Navigation Errors (1 σ)	Tentative Navigation Error Requirements (1 σ)*
Position			
R	0.45 nm	0.125 nm	1.7 nm
T	1.17 nm	0.13 nm	---
N	0.06 nm	0.055 nm	---
Velocity			
R	7.67 fps	1.15 fps	17 fps
T	2.2 fps	0.62 fps	5 fps
N	0.75 fps	0.62 fps	---
Orbital characteristics:			
Apogee/perigee	2.15 nm	0.58 nm	---
Inclination	0.0009 deg	0.00085 deg	0.033 deg

*Refer to NR ESS Phase A/B Technical Review Brochure 21 May 1971.

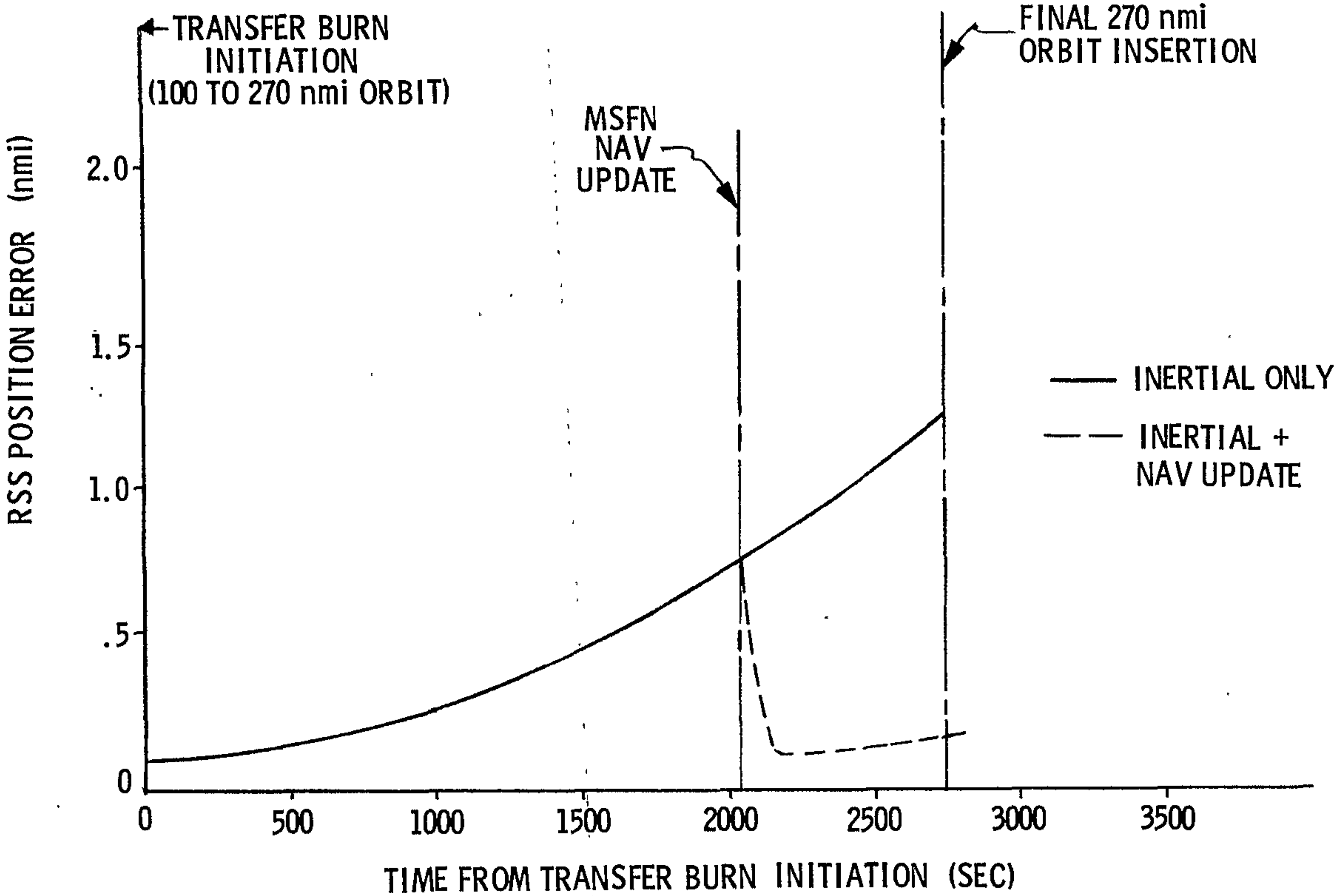


Figure 2-114. Direct Ascent From Parking Orbit to Final Orbit Insertion

e. Attitude Update Analysis. IMU misalignment results from initial gyrocompassing launch pad alignment errors and from constant gyro drift during the mission. The reference IMU has the following accuracy (1σ) requirements:

Azimuth Alignment - 360 seconds

Level Alignment - 40 seconds

Drift - 0.02 degree per hour.

The baseline nonrendezvous mission duration depends upon the time spent in parking orbit. The parking orbit mission duration is primarily dependent on navigation update time and is assured to be approximately one and one-half orbits. The IMU misalignment at this time is in the order of 500 seconds for azimuth and 200 seconds for leveling. These misalignment errors would result in the 270-nm transfer orbit velocity cutoff errors of less than 0.5 fps. As soon as a navigation update is achieved, the effect of this error is minimized since, if necessary, a correction burn could be performed. After the transfer has been completed, any burns would involve minimal total impulse and the platform misalignments would essentially introduce no error. For these reasons, it was concluded that an attitude realignment device such as a star tracker or horizon sensor would not be necessary for the baseline mission.

The direct ascent mission likewise indicates no requirement for an attitude update. The mission is short in duration, and only the initial misalignment terms dominate (AZIMUTH - 360 seconds, LEVELING - 40 seconds). These alignment errors propagated to the final orbit insertion burn would result in velocity cutoff errors of approximately 0.05 fps, which is an order of magnitude less than the baseline mission.

f. Guidance and Targeting. Analysis has indicated no hardware or implementation tradeoffs which should be made to adapt the shuttle orbiter guidance and targeting scheme to ESS. Software impact is discussed in Paragraph 2.3.2, DCM Subsystem.

2. Direct Ascent Mission to 270 Nautical Miles. The baseline mission ESS ascent trajectory was modified to burn eight seconds longer, thus achieving a 66- by 270-nm ascent orbit.

The IPEP program was used to propagate the platform errors to ascent orbit insertion. The resultant ascent orbit inertial navigation errors were then propagated to the 270-nm circular orbit insertion by using the ANS program. The RSS position errors for inertial navigation only are presented in Figure 2-115. Also seen in the figure is the effect of a navigation update just prior to the circularization burn at the 270-nm altitude. The navigation errors not making use of navigation updates are prohibitive. However, if a navigation update using MSFN is performed, then the position error immediately is reduced to approximately 1 nm. The following listing contains the 1 σ position and velocity navigation error components at circularization burn cutoff, using a MSFN update. Also listed are the orbital characteristics associated with these errors. The resultant orbit with the update has a 1-sigma apogee and perigee uncertainty of 2.2 nm (1 σ). Direct ascent navigation errors with and without a navigation update are as follows:

Parameter	Navigation Error (1 σ) Without a Navigation Update	Navigation Error (1 σ) With a Navigation Update	Tentative Navigation Error Requirements (1 σ)*
Position			
R	1.53 nm	0.46 nm	1.7 nm
T	6.5 nm	0.61 nm	---
N	1.56 nm	0.58 nm	---
Velocity			
R	37 fps	5.26 fps	17 fps
T	7.7 fps	2.75 fps	5 fps
N	45 fps	0.34 fps	---
Orbital characteristics			
Apogee/perigee	±0.92 nm	2.2	---
Inclination	±0.1 deg	0.00085 deg	0.033 deg

*Refer to NR ESS Phase A/B Technical Review Brochure (21 May 1971).

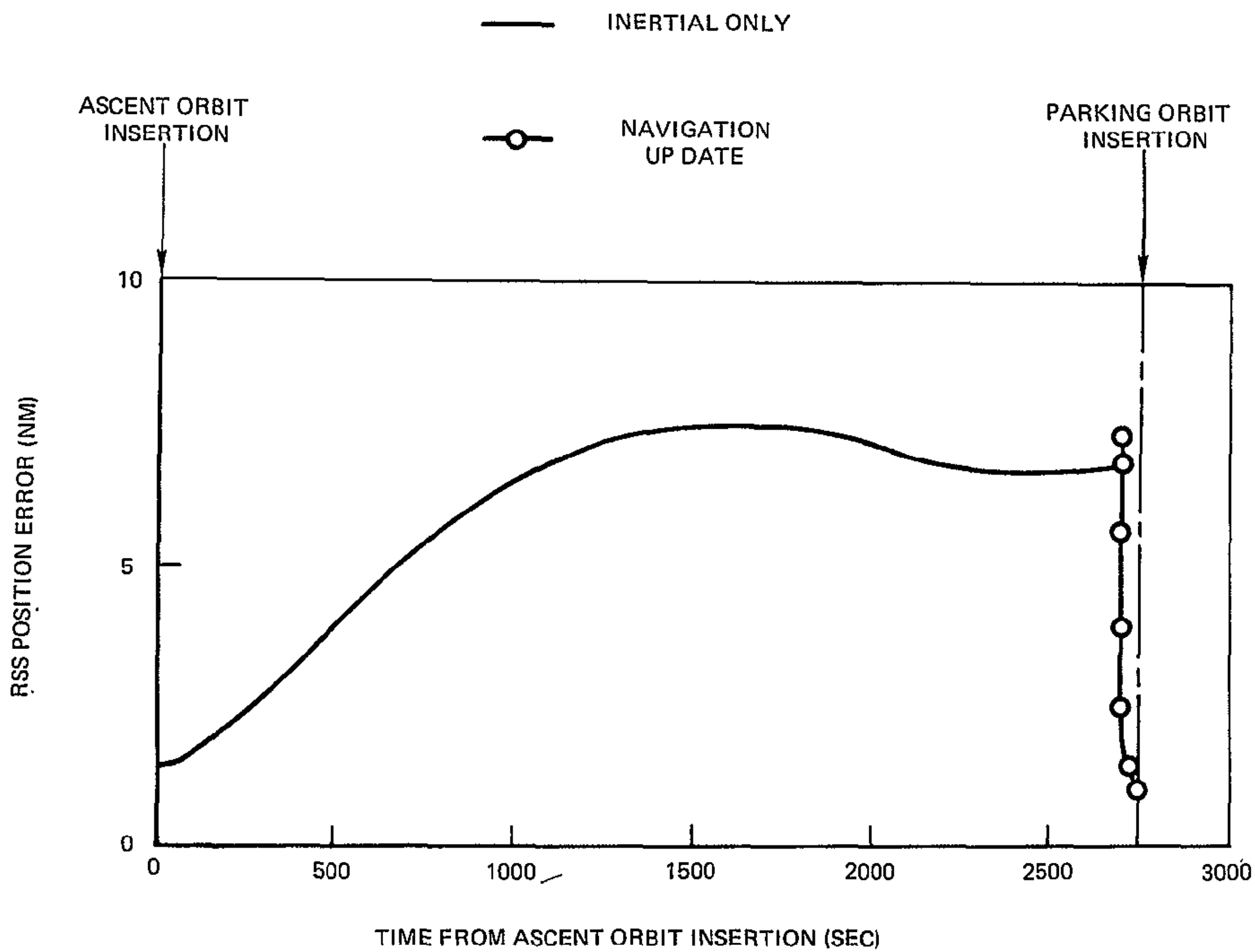


Figure 2-115. RSS Position Errors for Inertial Navigation

A tracking system error analysis was made by Goddard to determine the capability of a TDRS tracking network to meet the short-arc tracking objectives such as would occur for direct ascent missions to 270 nm. It was found that radial position, radial velocity, and tangential velocity uncertainties could be reduced to 1000 feet (1 σ), 2.0 fps (1 σ), and 1.0 fps (1 σ), respectively.

3. Summary and Recommendations for Nonrendezvous Mission, G&N Error Analysis. Baseline mission points are as follows:

   a. Navigation updates in 100-nm parking orbit are required.

   b. Navigation update aids the following:

      (1) Multilateration — meets accuracy requirements but adds on-board complexity.

      (2) MSFN (four stations) — meets accuracy requirements and minimizes on-board complexity.

      (3) TDRS — looks attractive mainly because of increased coverage. Additional detailed analysis is recommended for Phase C study. MSFN technique is recommended, the conclusion based on the fact that it can meet the accuracy requirements and that it minimizes on-board equipment requirements.

   c. At 100-nm parking orbit, less than two revolutions, are needed for navigation updates.

   d. Executing the transfer burn from 100-nm to 270-nm, immediately after a navigation update is obtained, improves accuracy and reduces mission time.

   e. Attitude reference updates do not significantly improve insertion accuracy and therefore are not recommended.

   Direct ascent mission points are as follows:

   a. A navigation update must be obtained just prior to circularizing at 270-nm to meet mission performance requirements.

   b. Orbit insertion cannot be performed as accurately as in the baseline mission.

Rendezvous Mission G&N Error Analysis. The boost and parking orbit insertion phases of the rendezvous mission are identical to the baseline nonrendezvous mission described in the previous sections. Therefore, the error analysis is the same. The error analysis for the transfer to 260 nm and the terminal rendezvous phases of the mission are presented in this section.

One and one-half revolutions of phasing in the 100 nm parking orbit are used in this analysis.

1. Orbit Transfer (100 to 260 Natucial Miles) and Terminal Rendezvous Phase. Initiation is assumed to occur one and one-half revolutions after parking orbit insertion.

   a. Multilateration navigation updates. The results of the orbital transfer and terminal rendezvous phase navigation error analysis using multilateration updates are shown in Figure 2-116. The figure indicates navigation accuracies achievable through use of inertial navigation only, inertial plus star tracker alignment, inertial with multilateration updates, and inertial with attitude alignment and navigation updates.

      The inertial-only navigation errors grow rapidly to an unacceptable magnitude of 14 nm at insertion into the final 270-nm orbit.

      If a navigation update is performed during terminal phase rendezvous, over the Hawaii Station, the navigation error is reduced to approximately 3 nm (3 σ), a figure well within the rendezvous requirement of 11 ± 10 nm (3 σ). Therefore, the conclusion to be drawn is that the multilateration technique for obtaining navigation updates is acceptable.

   b. MSFN navigation updates. The results of orbital transfer and terminal phase navigation error analysis through use of the produced MSFN tracking network are shown in Figure 2-117. Again the figure shows the accuracies attainable with different navigation schemes.

      The inertial only case with no updates is again clearly shown to be unacceptable. However, if a navigation update is obtained during the terminal rendezvous phase, the errors are reduced to approximately a 6 nm (3 σ) position error which is within ± 10 nm (3 σ) rendezvous requirement.

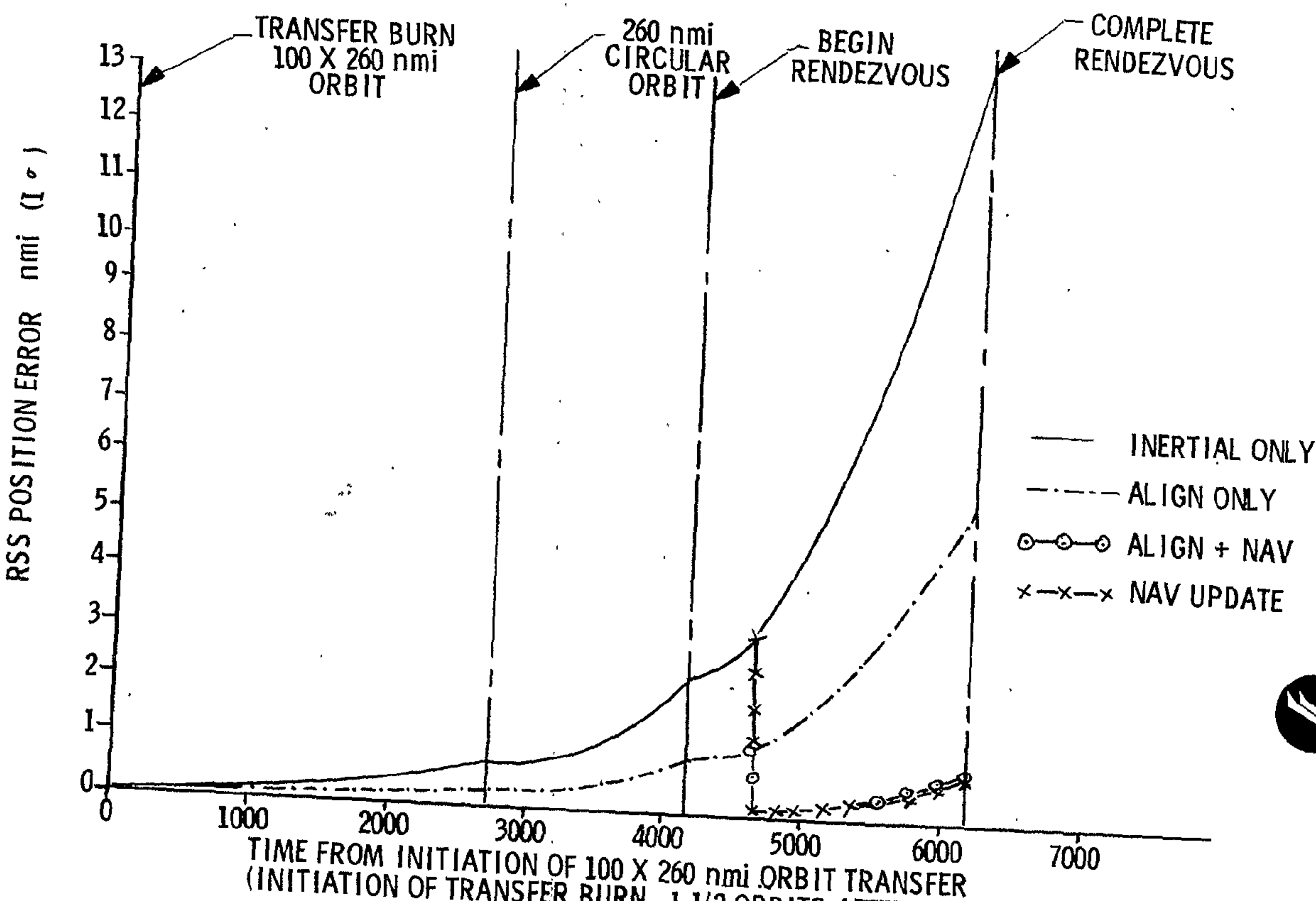


Figure 2-116. Orbital Transfer and Terminal Phase Position Errors Using Multilateration Updates

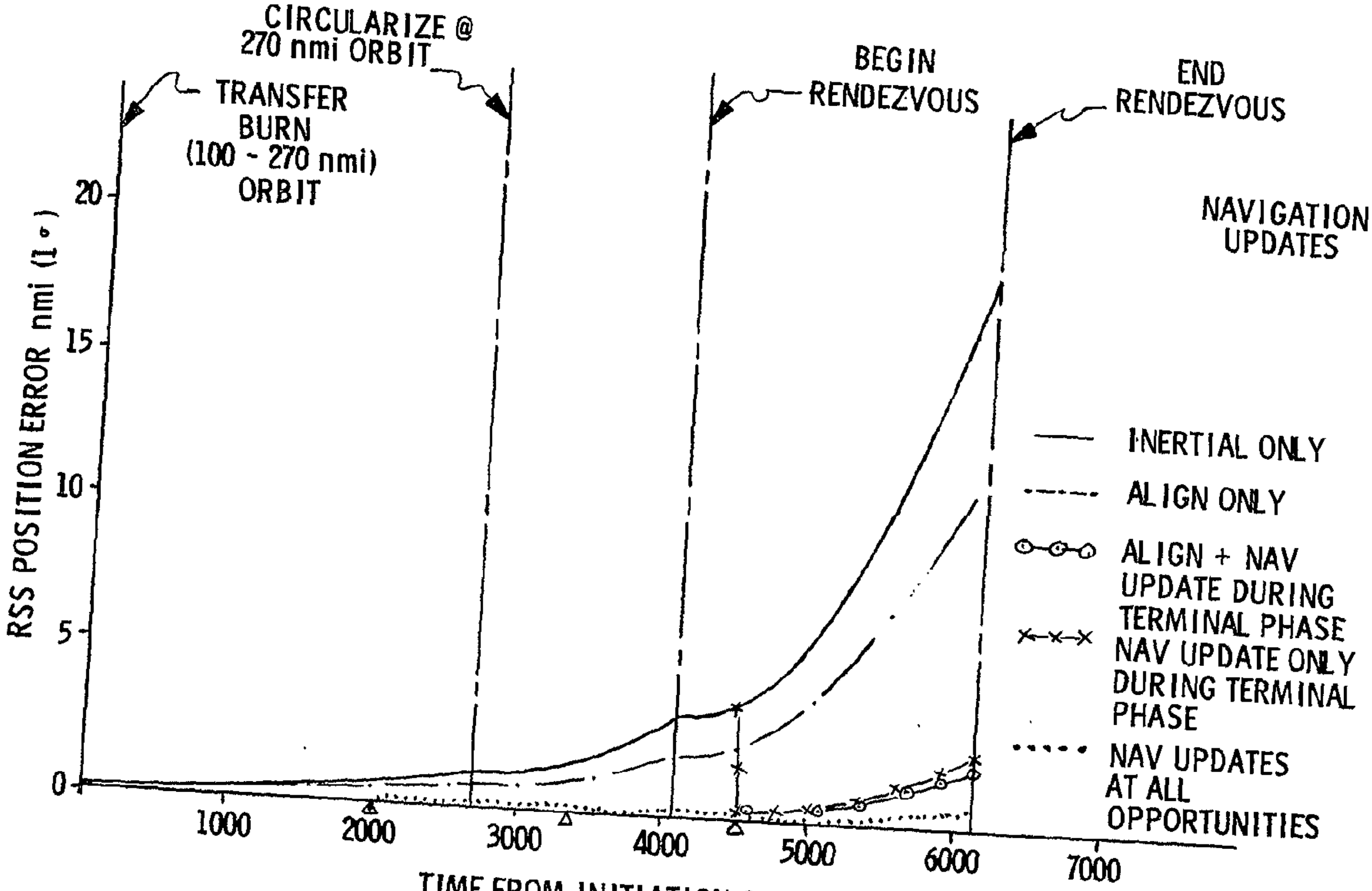


Figure 2-117. Orbital Transfer and Terminal Phase Position Errors Using Minimum MSFN Tracking

The dotted curve in Figure 2-117 emphasizes the sensitivity of position errors to the number of navigation updates obtained. For this case, a total of three updates were obtained: one during the transfer burn from 100 to 260 nm; one during the coast period at 260 nm; and one during the terminal rendezvous phase.

c. TDRS navigation updates. The TDRS results given in the previous section of nonrendezvous missions applies also to the rendezvous mission. Again, the main advantage in the use of TDRS is the increased coverage, which allows more opportunities for obtaining navigation updates.

2. Attitude Update Analysis. Figures 2-116 and 2-117 in the previous section show the terminal phase navigation errors by using multilateration and MSFN tracking updates, respectively. Both figures indicate that IMU with attitude update only will not achieve the desired performance requirements. When navigation updates during the terminal phase are performed with and without alignments, the performance differential is negligible. Therefore, the results of the orbit transfer and rendezvous phase navigation error analysis indicate that star tracker alignment will not be necessary.

3. Guidance and Targeting. The rendezvous mission imposes no unique guidance and targeting requirements.

4. Summary and Recommendations of Rendezvous Mission G&N Error Analysis. The following points are listed:

   a. To meet the rendezvous accuracy requirement, the ESS navigation subsystem will require navigation updates during the parking orbit and the terminal phase rendezvous.

   b. Navigation update techniques are as follows:

      (1) Multilateration — meets accuracy requirements.

      (2) MSFN (four stations) — meets accuracy requirements and minimizes on-board complexity.

      (3) TDRS — looks attractive mainly because of increased coverage. Additional detailed analysis is recommended for Phase C study.

c. The MSFN technique recommended is based upon the meeting of accuracy requirements and the requiring of less on-board equipment.

d. Attitude updates do not significantly improve rendezvous accuracy and, therefore, are not recommended.

Phasing Requirements Analysis. In order to launch into a given orbit, the launch must take place when the launch site is in the orbit plane — if there is not to be a plane change. There are (usually) two such launch opportunities per day, one having a launch azimuth less than 90 degrees (north launch) and the other having a launch azimuth greater than 90 degrees (south launch). Because the time of launch is determined by the transit of the orbit plane, the relative phasing of the target vehicle and the launch vehicle cannot be controlled, and a wait in a phasing orbit is required.

When the transfer maneuver from the 100-nm phasing orbit to a target in a 270-nm orbit is initiated, the rendezvous location is determined in both space and time. The transfer maneuver is not initiated until the target is in the proper position to arrive at the rendezvous point at the same time. The wait time in the phasing orbit depends on the initial angle between ESS and target, which is arbitrary, and on the angular rates of the two orbits. For the nominal conditions, the angular rates are 0.001187 rad/sec for the 100-nm orbit and 0.001107 rad/sec for the 270-nm orbit. The worst case is in prospect when the target vehicle would arrive just behind the rendezvous point; then the ESS in the inner orbit must catch up a full 360 degrees — a maneuver which requires 21.66 hours.

As the ESS, relative to the target in the outer orbit, advances in phase, the time to catch up decreases linearly until it reaches zero and then jumps to the maximum again.

The phasing required at successive launch opportunities (launch from KSC into the 100-nm orbit) is shifted by successive multiples of the daily difference. The worst case for a series occurs when the first launch opportunity required 360 degrees of phasing.

For the baseline mission, the interval between launch opportunities is nearly an exact multiple of the orbit period at the nominal altitude and inclination. The ratio follows:

$$\frac{\text{pseudoday}}{\text{orbit period}} = \frac{85127}{5677} = 14.995$$

This computation means that a delay in launching has small effect on the phasing time.

The conditions are illustrated in Figures 2-118 and 2-119. Figure 2-118 shows a plot of phasing time (in the 100-nm orbit) versus launch time for both the north and south launches. With the position of the target in the 270-nm orbit arbitrary, the worst case phasing time is 21.66 hours for a north-only launch or a south-only launch. Figure 2-119 is a composite of Figure 2-118, presently both the north and south launch opportunities. Two launch opportunities per day can effect a worst-case phasing of 14.60 hours.

Figure 2-120 shows the phasing time as a function of delay in launch for the nominal target orbit (270 nm and i = 55 degrees) and some variations of the target orbit. Noteworthy is the extraordinary flatness of the nominal cases. A change in altitude — up or down — produces an increased slope. A decrease in inclination produces longer than nominal phasing times for the two-opportunities-per-day case.

Figure 2-121 shows the phasing time required for the 260-nm, 31.5-degree orbit and some variations of this orbit. The worst-case phasing time, in terms of one launch opportunity per day is 23 hours. Two launch opportunities per day reduce the time to 20 hours. An increase of 1 degree in inclination (i = 32.5), two launch opportunities per day, gives a worst-case phasing of only 13 hours.

Figure 2-122 provides a plot or orbital altitude versus inclination for zero-phase change orbits. Two of the four ESS missions covered in this study have target orbits the parameters of which are very close to those of a zero-phase change orbit. The altitude for a 55-degree inclination zero-phase change orbit and for a 31.5-degree inclination zero-phase change orbit are 269.14-nm and 254.07-nm, respectively.

Zero-phase change orbits are very sensitive to small effects. An error analysis was made: the following two approaches to utilizing these zero-phase change orbits are considered feasible.

Regarding the first approach, suitably phasing the initial launch (e.g., placing a space station in orbit) can keep the phasing time required for subsequent launches (e.g., logistic supply missions to the space station) short for a long time interval. In using this approach in controlling the phasing time, the initial launch should include the maximum phasing time. The subsequent launch to rendezvous missions will then have a phasing time which linearly decreases to zero and then jumps to the maximum time of 14.6 hours (or possible 7.06 hours) as a function of the data after the initial launch.

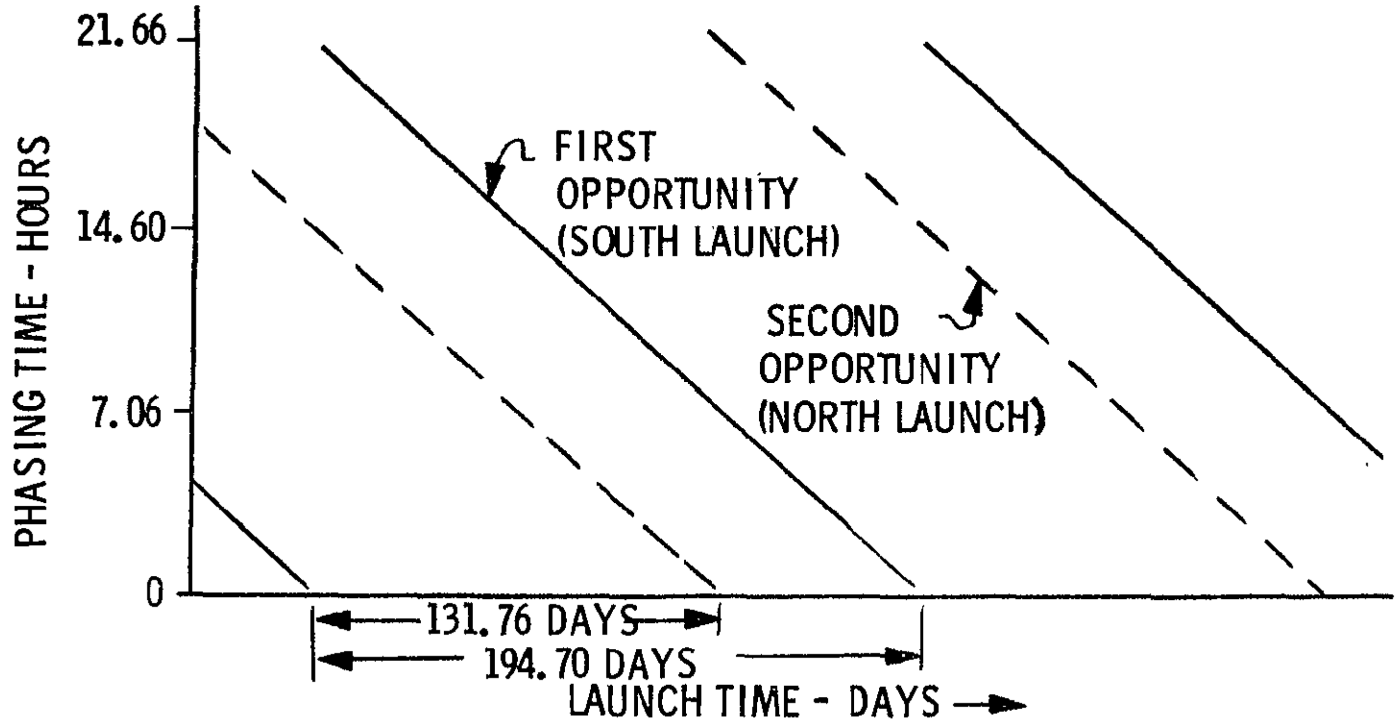


Figure 2-118. Phasing Time Required, Each Launch Opportunity

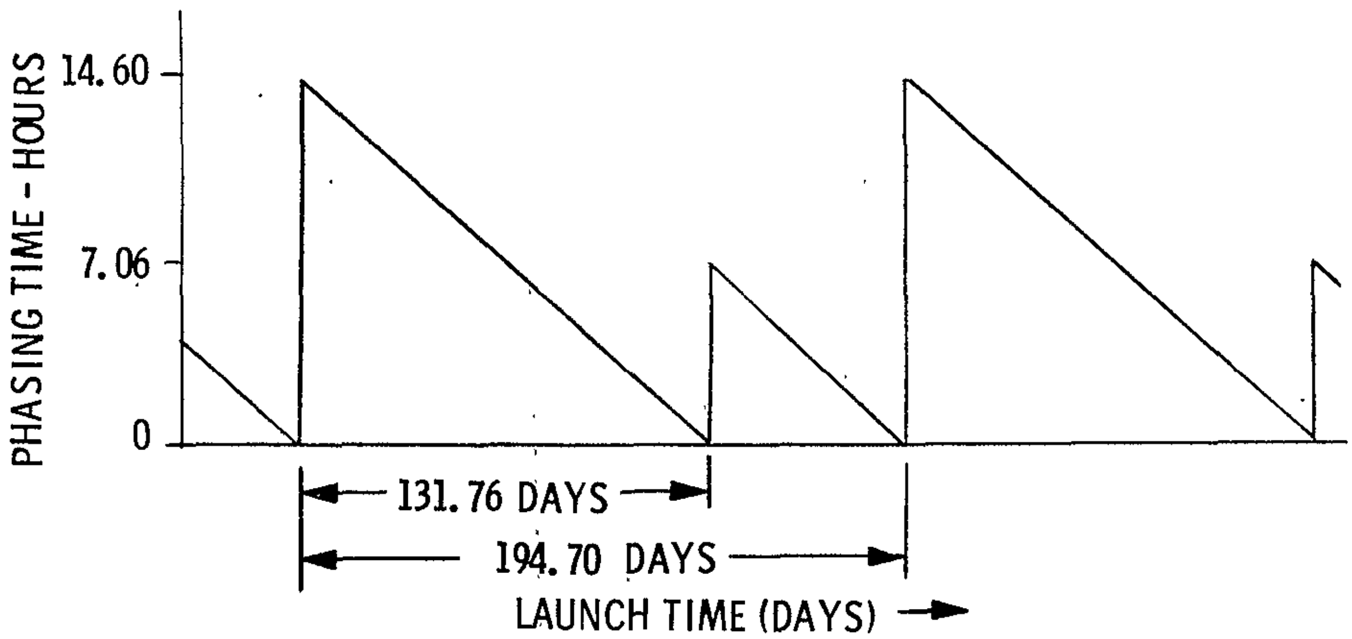


Figure 2-119. Phasing Time Required, Composite Using Both Launch Opportunities

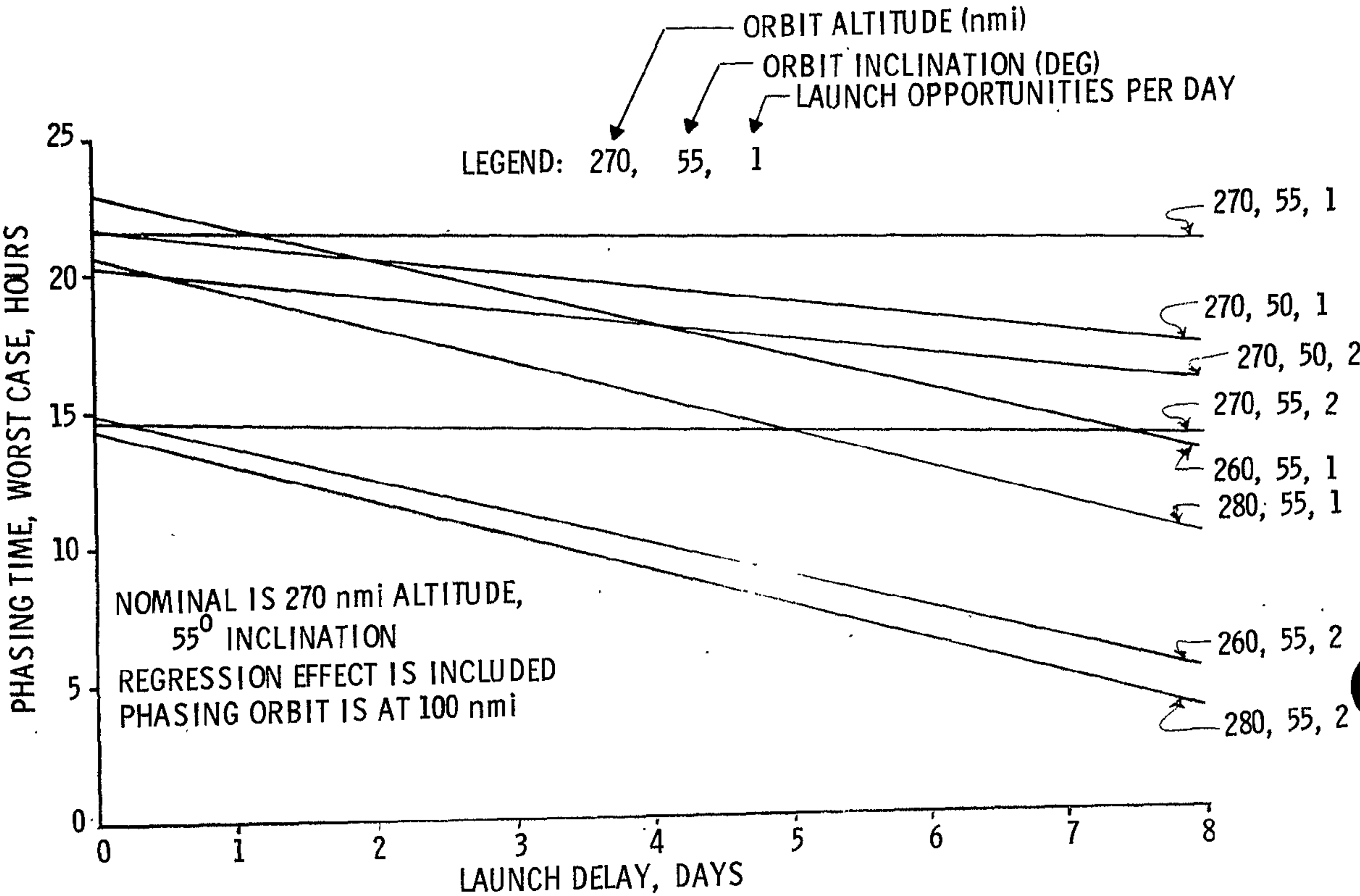


Figure 2-120. Phasing Time Versus Target Orbit Variations, Baseline Mission

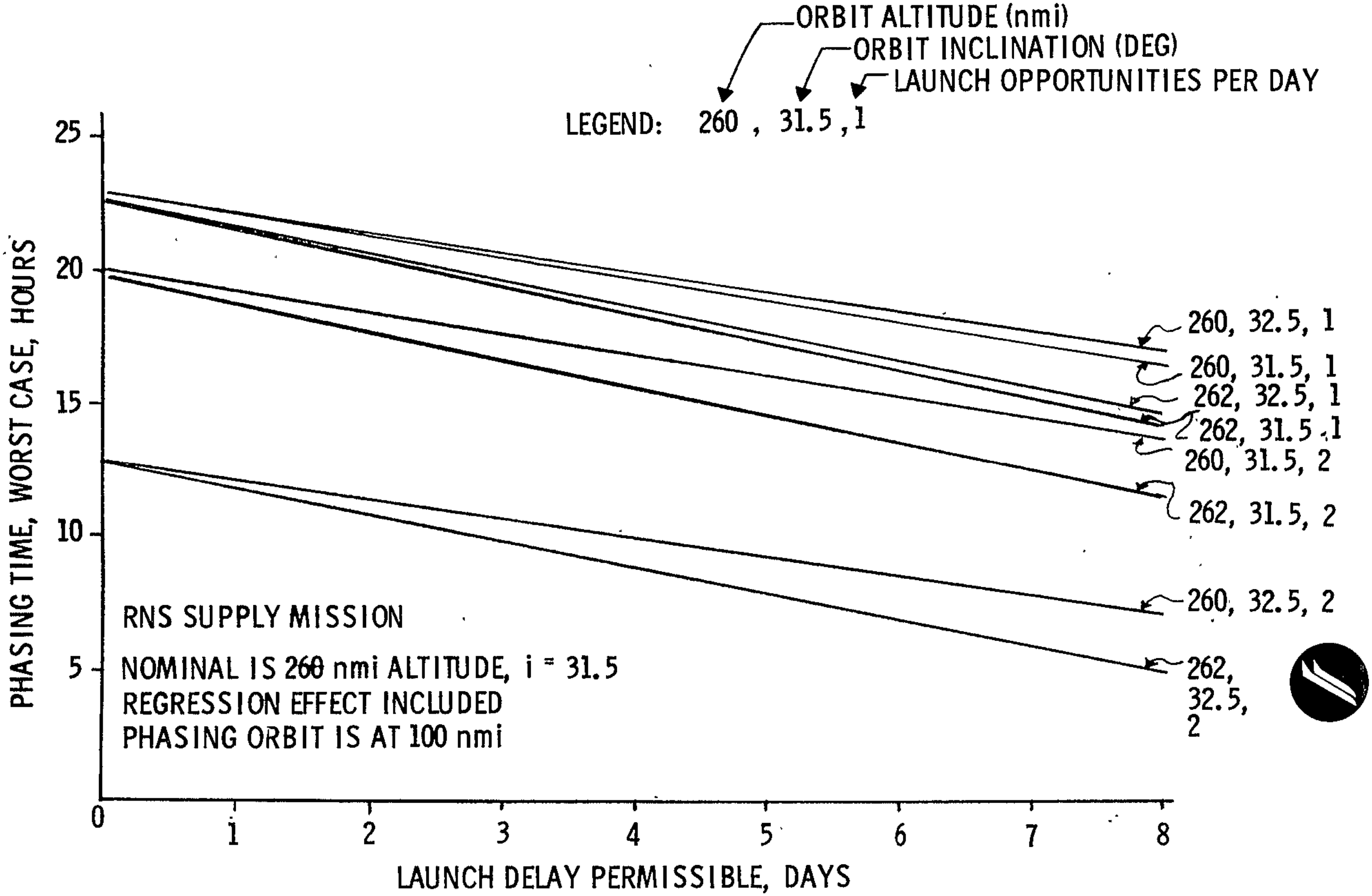


Figure 2-121. Phasing Time Versus Target Orbit Variations, RNS Mission

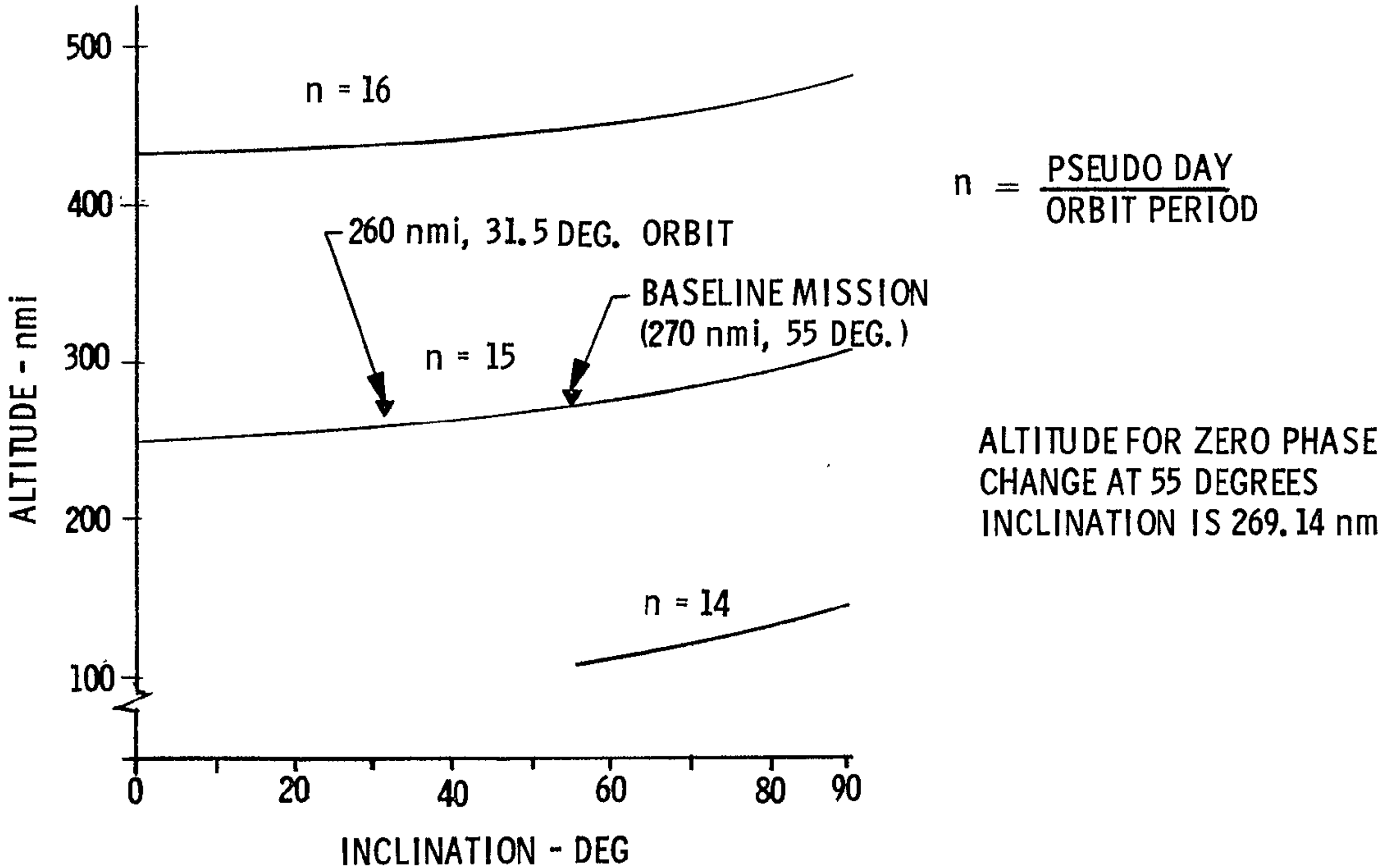


Figure 2-122. Orbits for Zero-Phase Change

This procedure may be further explained by using the baseline mission and by referring to Figure 2-119. If the initial launch is a north launch and is performed with no phasing, the next launch opportunity will require 7.06 hours of phasing for rendezvous. If the initial launch is a south launch, then 14.60 hours could be required. The reason is that n (ratio of launch opportunity to orbit period) is slightly less than an integer, and thus the target would be just behind the desired position in orbit for rendezvous. Providing phasing for the initial launch in the 100-nm orbit will move the target forward; thus less phasing time will be required for subsequent missions. Each hour of phasing in the initial luanch will provide about nine following days of favorable conditions for subsequent rendezvous missions.

The second approach involves moving the target orbits either to the zero-phase change orbits or away from them.

A target (space station) in a zero-phase-change orbit will provide fixed phasing for rendezvous missions. Station keeping is required for the target to remain in a zero-phase-change orbit; however, the station keeping orbit requirement is not severe. For rendezvous mission, if ten minutes of phasing in the nominal case are allowed, a ±5-minute variation in the nominal phasing would correct for ±90 nm of target position error.

Moving the orbital altitude away (a few nm) from the zero-phase-change altitude reduces the period between launch opportunities for a pre-selected maximum phasing time and increases the phasing time required. This condition is illustrated in Figure 2-123.

For an orbit of 55 degrees of inclination and with a phasing time of less than five hours and a launch opportunity within seven days, the usable altitude bands nearest the nominal 270-nm altitude are 240 to 255 nm and 290 to 320 nm.

Other Missions. Two other missions were given brief consideration: (1) synchronous-orbit mission and (2) close-rendezvous mission in order to identify potential additional system and operational requirements. These are discussed in the following paragraphs.

Synchronous Orbit Considerations. Since the advantage of placing a satellite in synchronous orbit is that it will remain fixed (with some small stationkeeping maneuvers) in earth coorindates, it is assumed that some pre-selected position in orbit is required. A trajectory for a typical equatorial synchronous-orbit mission is illustrated in Figure 2-124.

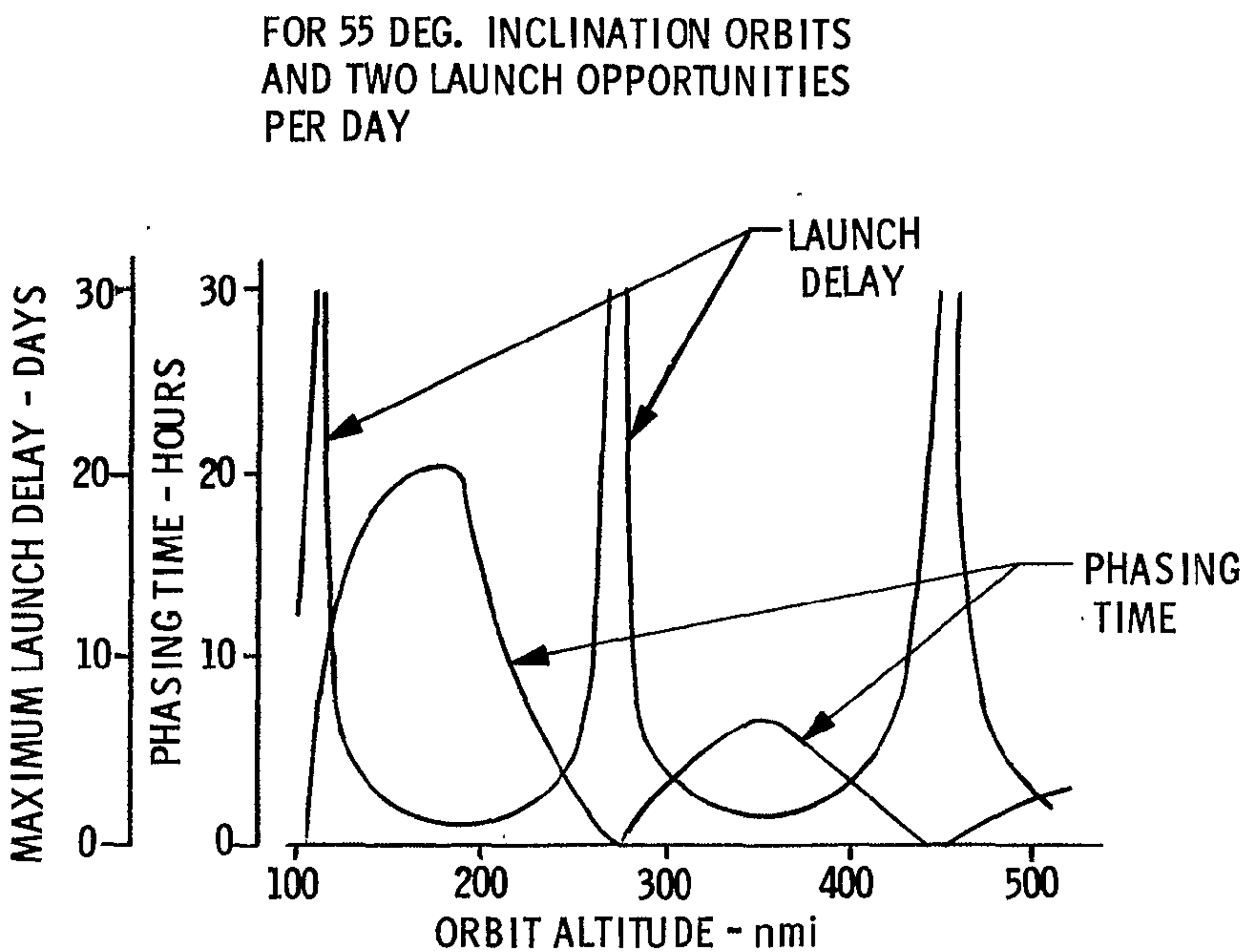


Figure 2-123. Orbital Altitude Versus Launch Delay and Phasing Time

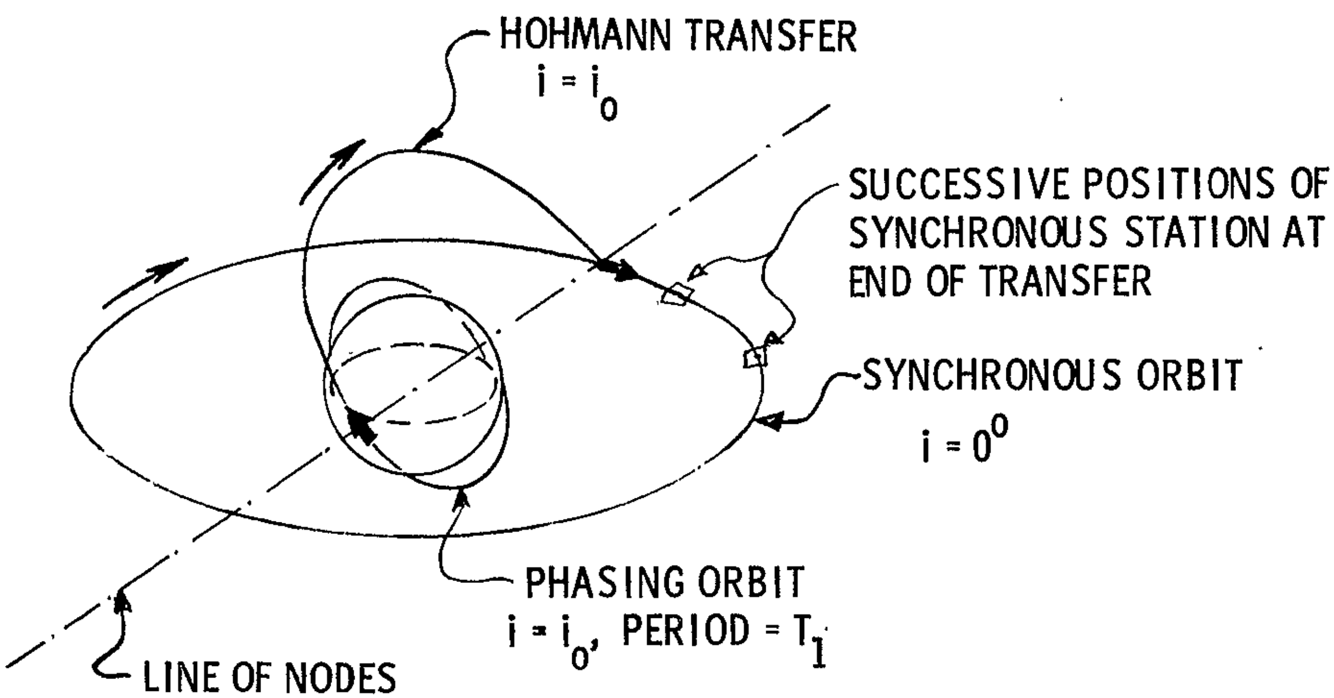


Figure 2-124. Synchronous Orbit Trajectory

The mission phases are as follows:

1. Boost from KSC to low altitude phasing orbit (i. e., 100-nm circular)

2. Coast in phasing orbit (phase and navigation updates)

3. Performance of Hohmann transfer from phasing orbit to synchronou altitude (19,360 nm)

The Hohmann transfer will begin and end on the line of the nodes as illustrated in Figure 2-124. Time in the phasing orbit is dependent upon target station location. The plane change velocity change (to achieve an equatorial orbit plane) is normally divided between the two burns for the Hohmann transfer. Initial lift-off and phasing altitude are selected to limit the phasing time to a desired value. Assuming that a minimum of two orbits in the 100-nm phasing orbit are required to accomplish sufficient navigation updates, mission durations (from lift-off until arrival at or near the desired synchronous orbit station) will vary from approximately 8.3 hours at 17.4 hours.

Extrapolating from the 270-nm rendezvous mission studies, it is evident that not only will multiple navigation updates be required, but the inertial attitude reference will also require updating because of the large velocity changes required to achieve synchronous orbit.

If the synchronous orbit mission were required, a star tracker of the type recommended for the shuttle orbiter would be added to the present system to accomplish stellar updating of the inertial attitude reference. This instrument is a strapped-down, dual-mode star tracker procured from ITT. It employs a photomultiplier tube and has an 8-degree field of view. Attitude reference updating would be accomplished in the 100-nm parking orbit and possibly during ascent to synchronous altitude prior to circularization and plane correction burns.

Deorbiting of the ESS from synchronous orbit would be very difficult— if not impossible because of the large velocity change required (approximately

4900 fps). Wherever the payload to synchronous orbit is compatible with such a procedure, it could be advantageous to leave the spent ESS attached to the payload during payload operations.

Another consideration for ESS disposal from synchronous orbit might be to inject the spent ESS into a solar orbit. Here, the velocity change requirements would also be excessive, however ($\Delta V \gtrsim 400$ fps).

Close Rendezvous Considerations. Extrapolation from the 270-nm rendezvous mission studies indicates that achievement of a very close rendezvous (i.e., 2 ± 1 nm) would require some combination of range to target, range rate to target, and/or relative angular data. This conclusion implies compatible ranging equipment on both vehicles as a minimum requirement and possibly the use of a star tracker (in conjunction with a compatible light on the target) or laser radar for angular information.

Since the 11 ± 10 nm distance used for a baseline rendezvous mission is considered adequate, the additional on-board hardware and software complexity, the decrease in reliability, and the associated increase in development and qualification requirements are not considered as warranted to achieve the relatively small improvement in performance.

2.3.2 Data and Control Management

The baseline data and control management (DCM) subsystem selected for the ESS provides a centralized, integrated computation capability with a standard interface to each subsystem through an operational data bus. The system includes the onboard hardware and software capability required to accomplish the ESS mission.

Commonality between the shuttle and ESS will be attained by utilizing DCM components and software developed for the shuttle program which meet the requirements of the ESS program.

The primary requirements imposed upon the DCM subsystem by the ESS are to provide the means of integrating, managing, and controlling the ESS vehicle. These requirements encompass the mechanization required to perform the computerized functions utilized in vehicle sequencing and control, subsystem configuration management, and data management. This section discusses the design requirements and provides a functional description of the DCM subsystem.

DCM Requirements

The DCM shall be capable of exercising control over the functions of (1) guidance, navigation and flight control, (2) vehicle sequencing and control, (3) subsystem redundancy management, and (4) data management.

The major ground rules and assumptions applicable to the DCM subsystem are:

1. Use shuttle hardware and software when possible.

2. The FO/FS requirement dictates that a fourth computer be supplied to provide 100-percent capability of detecting and recovering from a second computer failure.

3. All data interfaces external to the DCM subsystem are with the ACT (acquisition, control, and test) units.

4. The ACT units will be capable of accepting 28-vdc discrete inputs from other vehicle subsystems.

Guidance, Navigation, and Flight Control. The DCM subsystem shall be able to obtain GN&C subsystem velocity and attitude data, provide for calculations of all necessary equations (algorithms) to establish position data, and from tne data provide outputs to the vehicle flight control subsystem to maintain vehicle trajectory during mission boost phase and attitude control during on-orbit operations.

Vehicle Sequencing and Control. The DCM subsystem shall provide the capability of controlling the ESS subsystem to accomplish the following mission-oriented operations.

Prelaunch operations

Conduct subsystem readiness checks

Perform final "T" timed sequence events in preparation for launch

Provide information on ESS status to booster and ground

Perform subsystem statusing

Booster boost phase

Verify final engine prestart readiness

Provide information on ESS status to booster and ground

Maintain mission sequence

Perform subsystem status testing

ESS/booster separation through main engine cutoff

Initiate main engine start function when booster permits are satisfied

Support separation

Initiate main engine and supporting subsystems performance monitoring

Maintain mission sequence

Provide information on ESS status to ground

ESS safing

Initiate safing commands for main propellant tanks

Data Management. The DCM subsystem shall contain the software/hardware capability for acquisition, processing, and distribution of ESS vehicle data. These data consist of the following:

Vehicle sequence status data

Vehicle subsystem performance monitoring data

Uplink data

Vehicle Sequence Status Data. The DCM subsystem shall process and distribute to the ground and booster real time ESS vehicle status during boost phase. This status will include vehicle attitude, position, flight path and mode data, and any performance monitoring data which could cause an ESS/booster alternate sequence.

Vehicle Subsystem Performance Monitoring. The DCM subsystem shall process and store all ESS subsystem malfunction and configuration data. These data will consist of, as a minimum, all functional path failures and will include results of any diagnostics performed. The data to be recorded will include the following:

Measurement ID and value

Tolerance of the measurement

Time of occurrence

Correlation measurement data

Critical boost phase data will be transmitted real time as described under the vehicle sequence status data requirements above. Transmission of non-critical data to the ground will be accomplished only upon request through the uplink during quiescent vehicle modes.

Uplink Data. The DCM subsystem shall receive and process uplink data from any of the following sources:

Booster during mated boost

Ground during all mission phases

Orbiter during on-orbit operations

These data will be in the form of guidance and navigation updates, alternate sequence commands, attitude updates, and request for vehicle status.

Orbit insertion

Control OMS engine burn to obtain required delta V

Maintain vehicle attitude

Maintain vehicle performance monitoring

Provide information on ESS status to ground

Maintain mission sequence of events

On-orbit operations

Maintain attitude with attitude control propulsion system (ACPS)

Maintain vehicle performance monitoring

Provide information on ESS status to ground and orbiter

Safe the vehicle for orbiter rendezvous avionics recovery and docking as required

Separate ESS from the payload

Transmit all central processing unit (CPU) core data to ground

De-orbit

Perform ESS preparations for de-orbit

The DCM subsystem shall provide for all vehicle mission contingencies which are required to implement the FO/FS requirements. For example, vehicle stability requirements during one-main-engine-out operations differ from normal and must be preprogrammed in the DCM software.

Subsystem Redundancy Management. The DCM subsystem shall be capable of performing checkout and fault isolation for all individual ESS subsystems. It shall be able to detect all functional path failures, including those internal to the DCM, and implement the available redundancy to continue the ESS missions.

Software. The DCM subsystem shall contain the software required to implement the ESS operational requirements of:

Guidance and navigation

Flight control

Subsystem control

Uplink data

Subsystem configuration management

Vehicle data management

Checkout

Recovery safing

Vehicle Timing. The DCM subsystem shall provide all timing functions required by the ESS subsystems.

ESS Versus Shuttle Computer Redundancy Criteria. The shuttle program has a redundancy requirement of FO/FS for all systems except for the DCM system which is designed for a redundancy level of FO/FO/FS. The shuttle program meets this requirement for the DCM system by implementing a computer complex with four central processing units sharing 12 modules of memory. This configuration provides the shuttle with 100 percent coverage for failure detection and fault isolation during the first two redundancy levels of FO/FO and a degraded coverage (approximately 98 percent)for the redundancy level of FS. This degraded FS capability, when considered with the capability provided by the man in the loop and the hardwire backup capabilities provided in the shuttle design, satisfies the requirement of the shuttle program.

In considering the ESS requirement of FO/FS, it is necessary to evaluate the criticality placed on the fail-safe condition. The driving factors in establishing the FS requirements are:

1. Autonomous system due to no man-in-the-loop.
2. Normal ESS payload will be a high cost one-of-a-kind item.
3. No possibility of salvaging the payload with ESS failure to attain orbit.

4. The additional cost of the fourth computer is minimized by recovery of avionics.

For these reasons, in addition to the fact that there are no hardwire backup systems, it becomes necessary for the ESS DCM subsystem to be fully operational through two failures (FO/FS). This generates the need for the ESS DCM subsystem to maintain 100-percent coverage for both levels of redundancy provided to satisfy the FO/FS program requirement. This is accomplished by using the shuttle DCM computer complex of 4 central processors sharing 12 memory modules. When the recovery of ESS avionics is factored into the decision it becomes the most cost-effective approach for attaining the confidence level required for the ESS mission.

Subsystem Description

The DCM subsystem includes hardware and software required for ESS vehicle sequencing; performance monitoring; redundancy management; subsystem checkout; and data acquisition, processing, and distribution throughout the ESS vehicle. This subsystem is composed of three major elements: (1) data management computer (DMC), (2) data bus distribution components, and (3) software. Figure 2-125 is the DCM Subsystem Schematic.

The DMC provides the primary central vehicle intelligence for control of subsystems and associated data. The DMC complex consists of four central processing units (CPU), four input/output units (I/O's), and twelve main memory storage units (MSU's) of 8,192 words each. The data bus element provides the vehicle communication link between the central computer and the various vehicle subsystems, the booster crew display, and ground stations via telemeter or hardwire for ground monitor and control. The data bus consists of four twisted shielded pair transmission lines; acquisition, control and test (ACT) subsystem interface units; and transmission line/ACT interface line coupling units (LCU). The software element includes all computer programs for checkout and flight sequencing of the ESS vehicle subsystems.

This configuration for the DCM subsystem was selected to meet the requirements of detecting, isolating, and recovering from two independent single point failures. After two failures the subsystem is fully operational, which is required for safe vehicle operations. This is referred to as fail operational, fail safe (FO/FS).

Subsystem Operational Functions. The DCM provides the processing and computing functions required for operation and checkout of the ESS

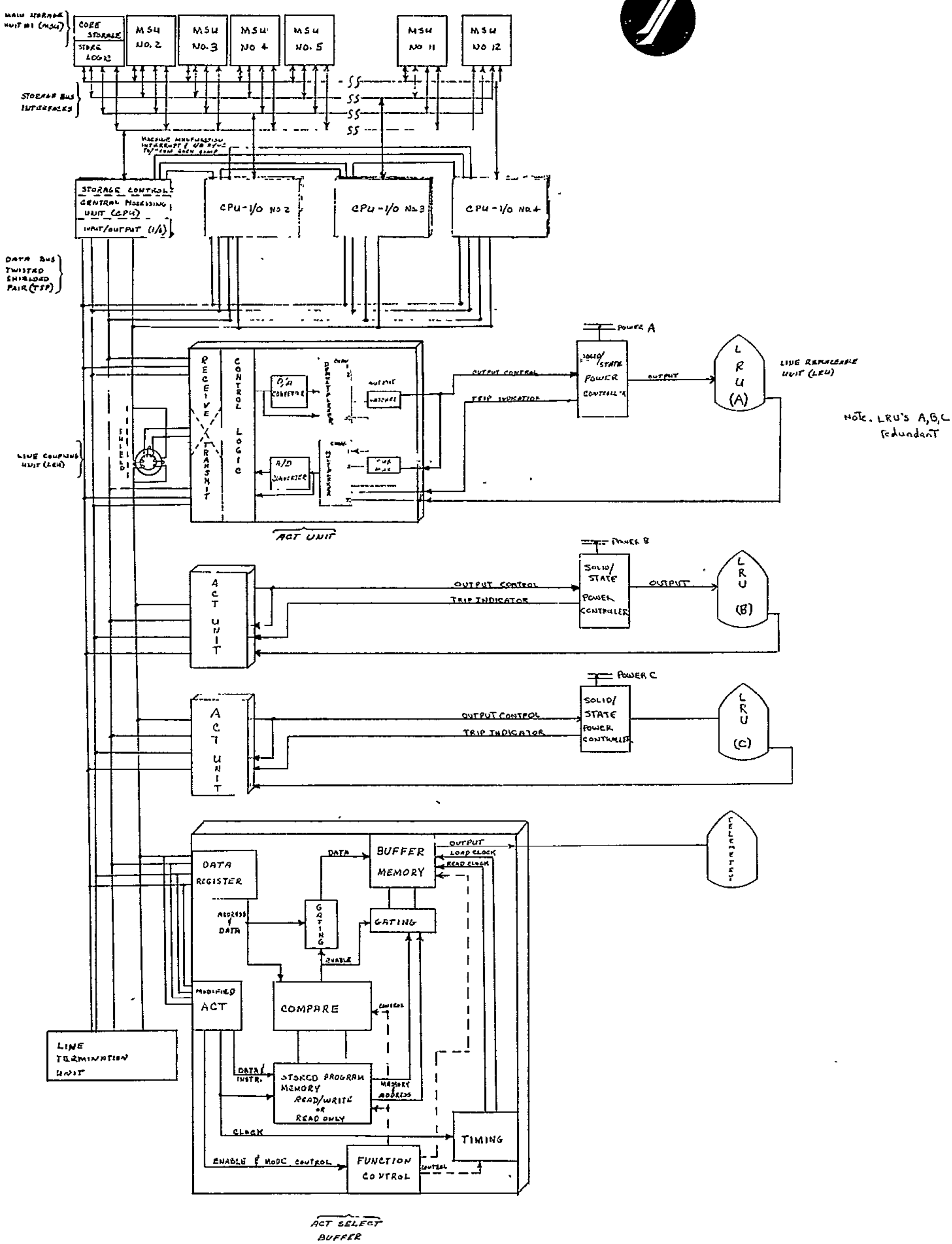


Figure 2-125. DCM Subsystem Schematic Diagram

onboard systems. To accomplish these functions, the following capabilities are required:

1. Configuration and Sequence Control. This function will provide control of subsystem configuration and sequencing of subsystem operations and modes for both normal and contingent operations. In abnormal cases, initiation will be implemented by uplink data from the booster or ground, but under normal conditions, all operations will be automatically initiated by preprogrammed vehicle mission sequences or as a result of the checkout and fault isolation function.

2. Guidance and Navigation. This function will process the appropriate guidance and navigation algorithms for each mission phase.

3. Flight Control. This function receives data from the guidance and navigation function and processes it to generate appropriate vehicle maneuvering commands to the vehicle control subsystems.

4. Data Storage. This function provides for storage of flight and ground programs, critical flight data, system performance data, and vehicle configuration status.

5. Data Acquisition and Distribution. This function provides for the intercommunication of measurements, subsystem outputs and control signals between the DCM and other vehicle subsystem elements.

6. Checkout and Fault Isolation. This function will maintain continued monitoring of vehicle subsystem performance. It shall determine the subsystem functional status and in the event of a failure, shall activate the redundant functional path for continued subsystem operation.

7. Self-Test. This function shall provide verification of proper DCM operation and initiate system reconfiguration when required.

8. Timing. The CPU will provide the timing for the ESS mission. Although the timing may drift slightly (4.3 seconds maximum) for the 24-hour mission, no time-dependent GN&C functions have been found critical to date. The navigation updates provided by the ground would aid in compensating for any errors. Therefore, the CPU timing will suffice for the ESS missions.

Basic Operation. The DCM subsystem will operate in a two-computer synchronized mode with both computers performing functions simultaneously and cross-checking calculations prior to transmitting data to the ESS subsystems. One computer will be considered prime while the other computer is considered the checker. Both computers will input data via the data bus, utilizing the same serial channels. Each computer will have access to five modules (8,192 word per module) of dedicated memory. All computer program instructions will reside in protected memory locations, and all computer variable data will reside in unprotected memory. (Protected memory is defined as "Read-only memory" while unprotected memory is defined as "Read and write memory.") All unused memory will be assigned as protected memory to preclude insertion of data erroneously. If the computer attempts to address unused memory it is operating in error. If any program attempts to write in protected memory, the computer will enter a failed state. Appropriate diagnostic routines will be executed to determine the cause of the failure and provide for reconfiguration of the failed element.

Each computer will contain a dedicated input/output (I/O) with a self-contained bus control unit. Control of the I/O and bus control unit will be provided by the control processor. The software will be assigned individual jobs. The I/O (ACT and bus control unit) function will be scheduled in the basic time-allotted job. The I/O will input data requested for the next time slot, and output data from the present time slot. The I/O structure will vary and is a function of the job scheduling. In fetching data from the ACT's, all data from an individual ACT will be assessed instead of individual sets of data per ACT.

In transmitting data to the subsystems, the CPU will identify data in memory to be transmitted by the data bus, identify the data bus to be used, and initiate the outputing of data. The I/O section will output data to the data bus independely of the CPU once the operation has begun. Data will be placed on the data bus in a serial pulse code modulated non-return to zero format at a one-megabit-per-second (MBPS) rate. The first 9 bits of data will contain the ACT unit address with one bit for status information. All additional information associated with the addressed ACT unit will be in 9-bit bytes with 8 bits of information and one bit for parity check. The ACT shall, as a minimum, respond to each command with a message containing the ACT address and status. If the command message was a request for data, the data will be contained in a variable number of bytes following the ACT address byte (Figure 2-126).

When the command message is an instruction to issue a particular subsystem control stimuli, the ACT will decode the instruction, set up its output demultiplexer and issue the stimuli. The output stimuli will, in addition to going to the subsystem, be wrapped around and fed back into an

BCU Transmitted Message Format

Minimum Control Word

Idle Bits (clock)	Beginning of Message (sync)	Device Address True and Complement	Function Code	End of Message (sync)	Idle Bits (clock)

Control Word and Multiple Data Bytes

Idle Bits	Sync	Device Address	Function Code	Data	Data	//	EOM	Idle Bits
		16 bits + 2 parity bit	8 bits + 1 parity	8 bits +1 parity	8 bits + 1 parity			

ACT Transmitted Message Format

Minimum Response

	Address of ACT	Data	EOM	

Response with Multiple Data Bytes

	Address of ACT	Data	Data	//	EOM	
	8 bits + 1 parity bit	8 bits + 1 parity	8 bits + 1 parity			

Figure 2-126. Data Bus Message Formats

input channel on the ACT unit. The computer will interrogate this ACT unit and verify that the proper command was issued and that no unscheduled event occurred. The computer will continue to monitor this functionally operational parameter to verify proper system response to the command. If the response does not occur, the DCM subsystem will initiate action to determine the cause and perform those actions required to continue system operation.

The DCM subsystem will provide the information and control necessary for the telemetry subsystem to transmit data to the ground. The DCM will load the telemetry system ACT/select buffer memory with the ACT addresses of the desired data. The defined data will be retrieved each time it appears on the data bus and be continuously and automatically transmitted until the ACT/select buffer is instructed to alter the particular measurement being transmitted.

The DCM will normally operate by the flight program, but the system contains the capability to be updated by the uplink of the communication subsystem. The capability and control provided by this function will be limited to updating navigation information, initiation of the deorbit sequence, request for ESS status and limited command sequencing.

DCM Performance Analysis

<u>ESS Deltas to Shuttle DCM Hardware.</u> The following are the deltas to the space shuttle baseline DCM configuration dictated by ESS requirements. In general, the ESS will use the shuttle hardware (and software) without modifications.

Redundancy Reduction. The FO/FO/FS (for crew survival) redundancy requirement for the Space Shuttle has been reduced to F/O FS for the ESS vehicle. The following are the redundancy levels for the ESS as compared to the Space Shuttle.

Four CPU/IO units required (no reduction)

Twelve 8K main storage modules required (no reduction)

No mass storage units required (reduction of four)

Four data bus lines required (reduction of one)

The reduction of redundancy will have little or no impact on the ESS, except for reductions in power, weight, and volume.

ESS Deltas to Shuttle DCM Software. This section is an overview of differences between space shuttle and ESS software. The significant items are delineated. Table 2-25 estimates the quantity of 32-bit words which can be deleted from shuttle programs for ESS use. Table 2-26 identifies the additions and modifications required for ESS-peculiar demands.

One of the deltas most readily identified is the lack of need on the ESS for mass memory. Reasons for eliminating mass memory are:

Fewer system table requirements

No display and control requirements

Fewer diagnostic requirements

Less complex mission requirements.

Computer Size and Speed Requirements Summary. The following represent the preliminary computer size and speed requirements to perform the software functions for the ESS.

Assumptions. The following assumptions have been used in arriving at size and speed estimates.

The ratio of 75/25 (short/long) for instructions, i.e., 75 instructions out of every 100 are half-word instructions.

Data words are generally assumed to be half words (16 bit) except for guidance, navigation, and control where 32-bit data words are assumed unless otherwise stated.

Total storage is always expressed in whole (32 bit) words.

Speed requirements are identified as KADS (equivalent adds per second divided by 1000).

A 25-percent contingency factor is included for storage to allow for undefined requirements.

The size and speed estimates shown represent worst-case analysis.

Estimates. Tables 2-27 and 2-28 represent the overall sizing and speed estimates required for the ESS operation in flight.

Table 2-25. ESS Deletions From Shuttle Total Requirements

Item	32-Bit Words
Executive	
Program loader	266
Drum access method	313
Control sampling access method	348
Mass storage allocation	228
Executive services control	19
Integrated displays and controls	
Display format retrieval	281
Continuous and on-demand services	156
Input data interpretation	94
Device caution and warning	31
Display access method	375
Crew aid and assistance	78
Variable data update routines (digital and vector)	2344
Checkout and fault isolation	
Failure recording and display	62
Subsystem management	
Air-breathing engines	92
System tables	
Queue control table	100
Program control table	320
Mission phase core and drum maps	375
Display control and associated display tables	1000

ESS Versus Shuttle Computer Redundancy Criteria. The shuttle program has a redundancy requirement of FO/FS for all systems except for the DCM, which is designed for a redundancy level of FO/FO/FS (for crew survival). The shuttle program meets this requirement for the DCM system by implementing a computer complex with four central processing units sharing 12 modules of memory. This configuration provides the shuttle with 100-percent coverage for failure detection and fault isolation during the first two redundancy levels of FO/FO and a degraded coverage for the redundancy level of FS. This degraded FS capability, when considered with the capability provided by the man in the loop and the hardwire backup capabilities provided in the shuttle design, satisfies the program requirement.

Table 2-26. ESS Development Deltas From Shuttle Summary

	Number of Instr. Affected
Executive	
Timer routines — modified[1]	75
Data sampling control functions — new development	270
Sequencing — new development[2]	425
COFI	
Reasonableness tests — modified	339
Functional path isolation — modified	880
System reconfiguration — modified	150
Diagnostics — modified	4400
System tables	
Mission profile — new development	3000[3]
System status — modified	210
Task and executive control — modified	150
Sequencing action tables — new development	300
System reconfiguration table — modified	44
Test point limits — modified	1140
Flight control	
Ascent TVC — new development	406
OMS TVC — new development	406
ACPS — new development	233

Notes: [1]Modified assumes that change of up to 50 percent can be applied.
[2]New development assumes 100 percent change can occur.
[3]Table estimates assumed to be 32-bit words.

In considering the ESS requirement of FO/FS, it is necessary to evaluate the criticality placed on the fail safe condition. If the shuttle rationale is used, it becomes necessary to establish the significance of the man in the loop. Since the ESS is non-manned, it cannot rely upon a pilot to assist in accomplishing the vehicle mission. This, in addition to the fact that there are no hardwire backup systems, makes it necessary for the ESS DCM subsystem to be fully operational through two failures, and hence to maintain 100-percent coverage for both levels of redundancy to satisfy the FO/FS requirement. This is accomplished by using the shuttle DCM computer

Table 2-27. DMC Memory Requirements (32-Bit Words)

Attitude Reference	Guidance	Navigation	Control	Executive	COFI	Subsystems	System Tables	25-percent Margin	Total
952	6371	2879	1707	1505	7932	2008	7600	7739	38,693

Table 2-28. DMC Speed Requirements (KADS)

Mission Phase	Attitude Reference[1]	Guidance[1]	Navigation[1]	Control[2]	Executive[2]	COFI[2]	Subsystems[2]	Totals
Boost	19.60	0.28	15.98		14.845	16.894	4.0	71.599
Separation	19.60	0.28	15.98	15.57	14.845	16.894	4.0	87.169
ESS ascent orbit insertion	21.28	48.12	15.50	15.57	14.845	16.894	4.0	136.259
On-orbit	35.49	2.87	47.07	2.69	14.845	16.894	4.0	123.859
Rendezvous	20.27	0.96	15.50	2.69	14.845	16.894	4.0	75.159
Deorbit target	21.28	4.00	0.299	15.57	14.845	16.894	4.0	76.888

Notes: [1]Determined from Honeywell Document 21602-SDD (29 March 1971)
[2]Determined from NASA Document MSFC-DRL-008A (25 June 1971)

complex of four central processors sharing twelve memory modules. When the recovery of ESS avionics is factored into the decision to use four computers to accomplish FO/FS criteria with 100-percent coverage for failure detection and isolation, it becomes the most cost effective approach for attaining the confidence level required for the ESS mission.

ESS Deltas to Shuttle On-Board Checkout. The ESS program uses shuttle software to the maximum, but because of the difference in program objectives, there are primary differences in the hardware required. The decisive factor in the area of onboard checkout of the ESS is the lack of the man-machine interface which exists on the shuttle DCM subsystem. On the shuttle, this man-machine interface is primarily provided by the display and control hardware and software. For the ESS program, checkout will be performed primarily with the computational capability provided in the ESS DCM subsystem. External equipment will be provided to perform the functions provided by the display and control section of the shuttle DCM system. This technique provides maximum utilization of shuttle software and minimizes the delta required to provide the ESS with the capability of performing subsystem checkout.

Redundancy Management. Redundancy is provided primarily to achieve a higher probability of mission success than can be expected from a simplex system and to conform to specifications which dictate system tolerance to two or more failures. To accomplish the goals of a high probability of mission success and a FO/FS system, various modes of redundancy are implemented by the individual ESS subsystems. The DCM subsystem provides, through redundancy management, the necessary capability to maintain the onboard subsystems at the proper operational level.

In this section the typical levels and modes of redundancy implemented by the ESS and the methods recommended for managing this redundancy are briefly described. The approach taken is described, along with the requirements, ground rules, and examples of the use of functional elements in the various subsystems.

The purpose of redundancy management is to provide a means for performing the processes and actions necessary to neutralize a detected failure in a functional path, record the failure, and reconfigure the functional path to maintain operational capability.

Redundancy management is not a prime factor in establishing the level of redundancy. It may, however, be the deciding factor in implementing redundancy.

The approach used to establish the redundancy management of the ESS operational system is based on a thorough analysis of the complete system. This analysis establishes the criticality of the individual subsystems. It provides an inventory of available functional paths, redundant components contained in the functional paths, and alternate methods which may be used to accomplish a desired function. The data thus obtained is then processed to provide an initial system configuration, and the sequence for use of redundant components. The techniques are also established for maintaining operational status of the individual components.

Two basic types of redundancy are implemented on the ESS program. These techniques are masking and sparing. Masking is characterized by performing a given function by means of three or more units and assuming the correct function will be represented by the majority of units in the event of a malfunction. Sparing is characterized by performance of a function by a prime unit and including in the system a means of determining when the prime unit is operating correctly and a means of replacing the prime unit with a good unit in the event of a failure. These two redundancy techniques in conjunction with the coverage function (that is, the probability of correctly detecting the need to replace a functional component if the component is not successfully and efficiently recovering from the malfunction) provide the basis for redundancy implementation and management for the ESS subsystems.

The level of redundancy implemented for the ESS avionics system is FO/FS for all systems affecting booster safety and ESS deorbit capability. All other functions, as a minimum, meet the reliability criteria established for the Apollo/Saturn program. Refer to Figure 2-127 ESS failure criteria.

The following are examples of typical circuits implemented in the ESS subsystem with methods which will be used to manage these circuits during operation:

DCM redundancy management (Figure 2-128)

Data bus redundancy management (Figure 2-129)

Typical subsystem redundancy management (Figure 2-130)

Thrust vector control redundancy (Figure 2-131)

The figures referenced above are intended to represent the concepts to be utilized on the ESS program and are not to be taken as the complete analysis required for the individual subsystems.

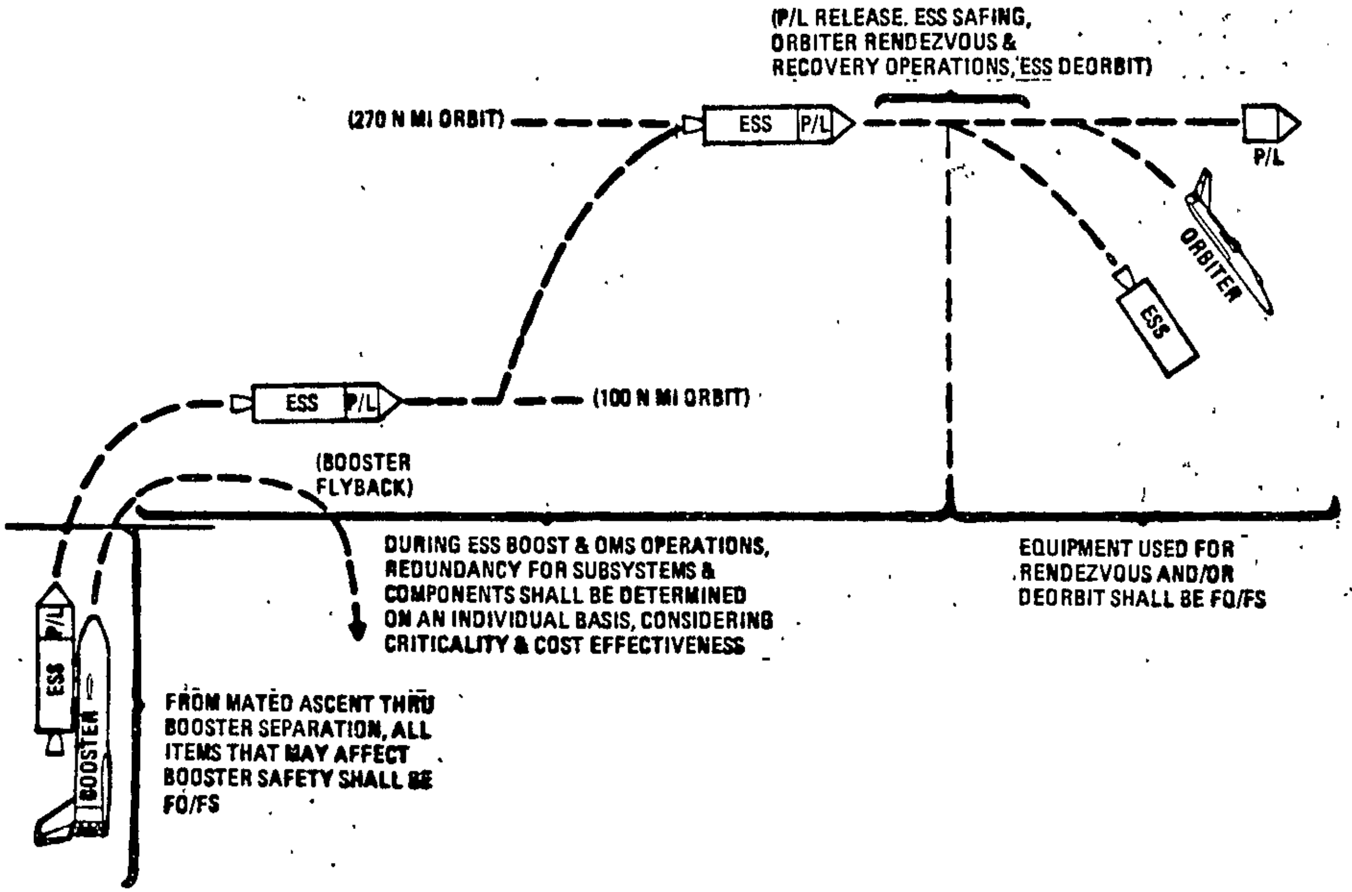


Figure 2-127. Failure Criteria Electrical and Mechanical Systems

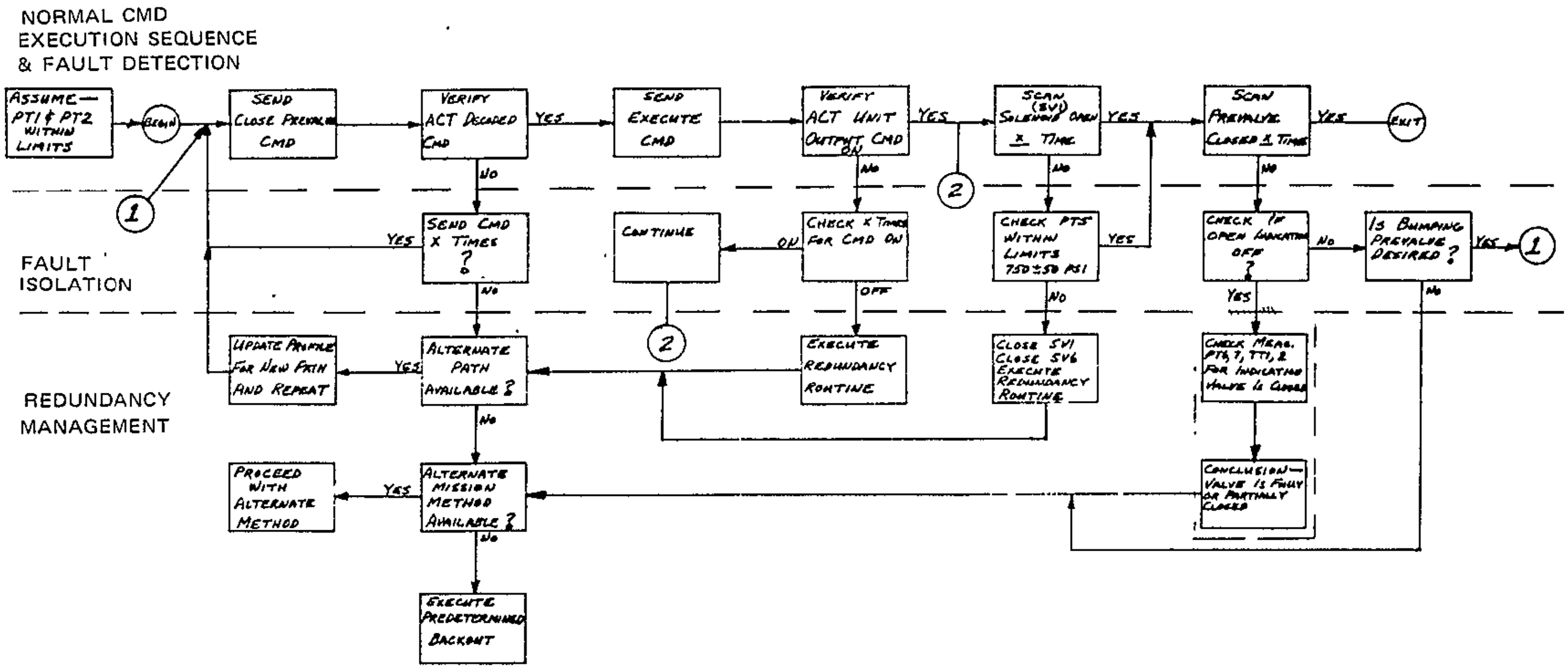


Figure 2-130. Logic Flow, Close Prevalve Command

LOGIC FLOW — CLOSE PREVALVE COMMAND

Analysis for Redundancy Implementation and Management. An efficient redundancy management scheme involves the ability to accurately assess performance on functional paths in a dynamic operating environment. To establish the redundancy management requirements for an operational system, it is necessary to investigate all facets of the program. The investigation must include an analysis of the redundancy provided in the individual subsystem capability along with the additional redundancy provided in the control circuits of the subsystem.

Using a motor-driven switch as an example, the following is a typical presentation of the analysis used to develop the implementation and management of a specified redundancy level.

> The first step in the analysis is to evaluate the criticality of the system, and the criticality of the component to the proper operation of the system. For this example it is assumed the initial evaluation reveals that the proper operation of the switch is critical to the mission success. This fact establishes the criteria for the required level of redundancy. For the ESS vehicle, the redundancy requirement is FO/FS so this is the level to be developed in the analysis.

Figure 2-132 is a typical schematic for a motor-driven switch with position indicators. The switch is the make-before-break type.

To implement redundancy effectively, it is necessary to perform a failure mode effects analysis (FMEA) of the function or component in question. The FMEA for the motor switch provides the information discussed here.

The switch may fail in any of five modes: (1) position indicator fails to open, (2) position indicator fails shorted, (3) switch fails to transfer, (4) switch fails with output contact open and (5) switch fails with output contacts shorted. A failure of the position indicators in either the open or shorted mode will not affect the operation of the switch but will eliminate the capability to provide verification of proper operation. This failure could be a fail-safe condition. This fail-safe rating is upgraded to a fail operational if other sensors are provided in the circuit which, in addition to their normal function, can also be used to deduce the switch position. This is accomplished by adding a current sensor in each of the contact legs of the switch or, in instances where only one contact is used, it will suffice to monitor the load to establish proper operation of the switch.

A comparison of the lines monitoring the source and load as depicted in Figure 2-133 will indicate switch position. The source must be monitored

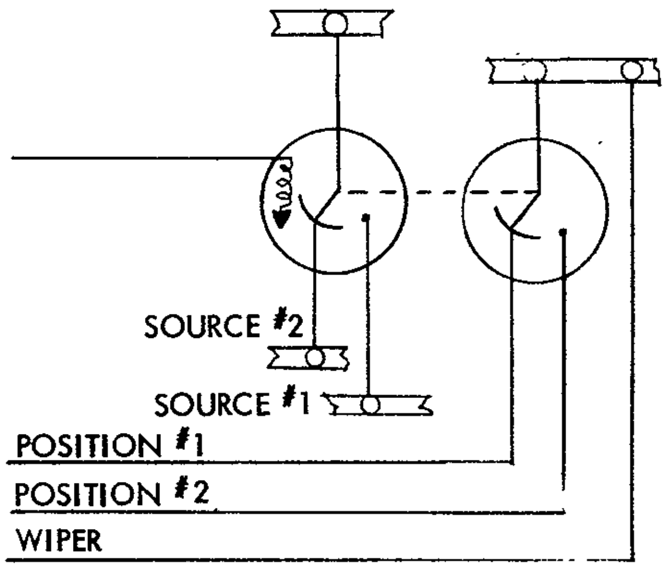


Figure 2-132. Motor-Driven Switch
Schematic Diagram

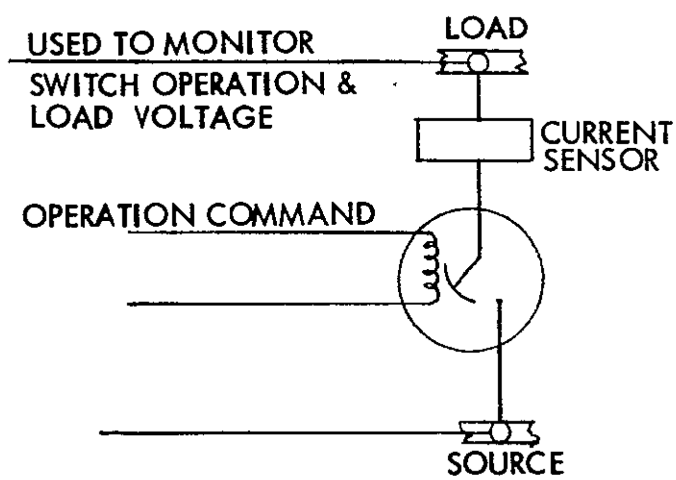


Figure 2-133. Motor-Driven Switch
Source and Load Monitoring

to provide valid information in the situation where the load is removed from the power source. The switch position can also be deduced by comparing the current at the source of the current sensor with that at the output.

An output of the current sensor will indicate a switch closed position. No output indicates an open position, provided this is verified by the source monitor.

In Figure 2-134, monitoring the load alone will not indicate switch operation. This monitoring point will provide backup data for other measurements. The output of the two current sources plus monitoring the source will provide adequate information to deduce switch position. The output of the individual current source can be used to indicate actual switch position provided the line monitoring the source indicates it is active.

Instances where the switch fails to transfer or fails shorted or open can only be considered non-safe when proper operation is critical to mission success. An operational failure requires that the failed unit be isolated from the circuit and the function be transferred to a redundant component.

Figure 2-135 is a method of providing isolation so that the switch can be removed from the circuit when it fails to transfer or hangs up with the two contacts shorted.

Figure 2-136 represents the circuitry required to implement redundancy and to provide the necessary information to manage this redundancy. The configuration represented in Figure 2-136 provides a redundancy level of FO. To implement FO/FS, a third component is required.

2.3.3 Communications Subsystem

The Communications Subsystem (COMM) is provided to fulfill the requirements for transmitting and receiving all information necessary to accomplish the ESS mission. The COMM will provide telemetry data, turn-around ranging data, and will receive up-data and range safety commands. An integrated approach to the COMM provides three subsystems: data, RF, and range safety.

Communications Requirements

As the COMM subsystem block diagram (Figure 2-137) shows, the three subsystems serve as carriers for five data links: the data bus, tracking and ranging, and range safety. The operational timelines of the COMM subsystem are shown in Figure 2-138.

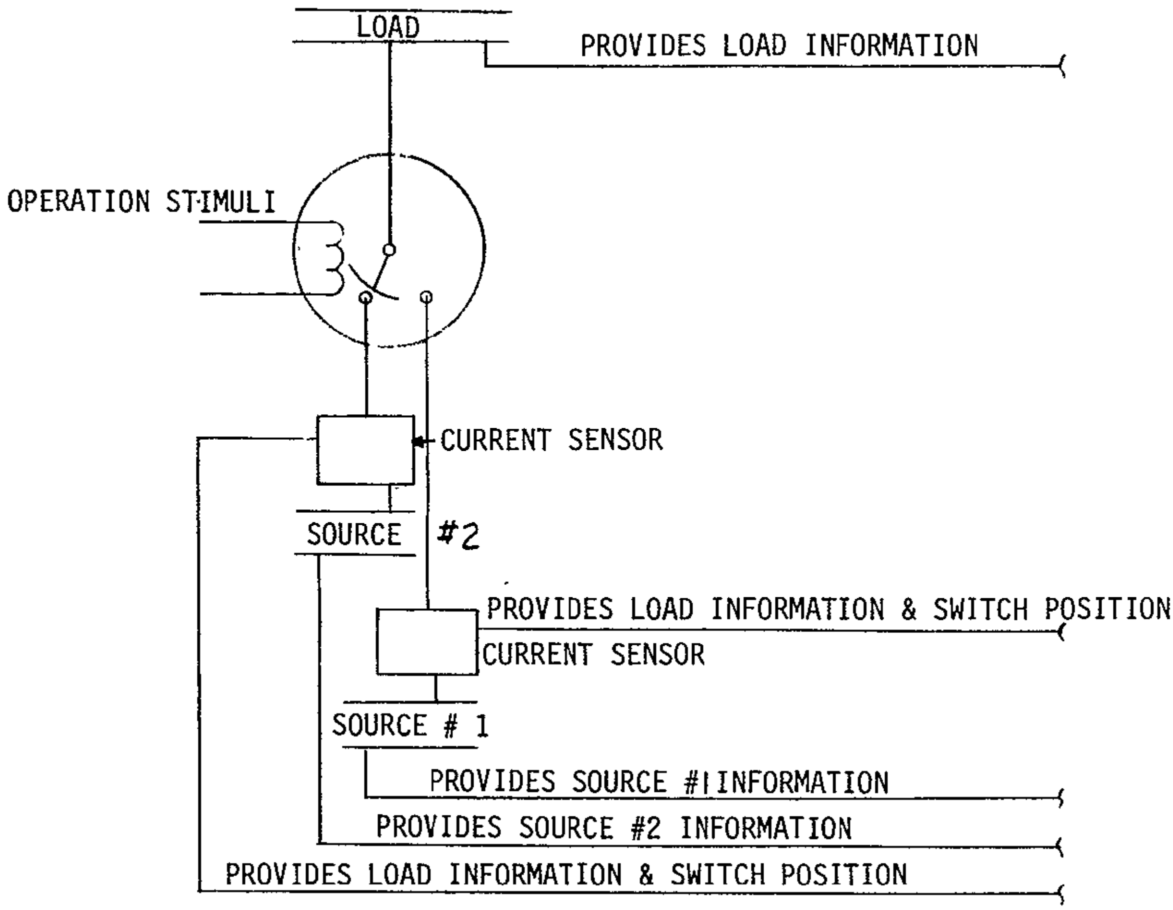


Figure 2-134. Motor-Driven Switch, Load Monitoring

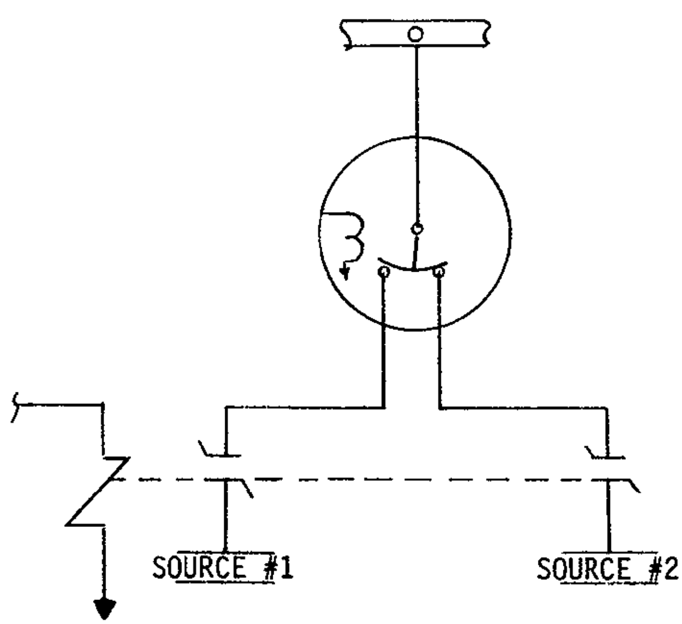


Figure 2-135. Motor-Driven Switch Isolation Technique

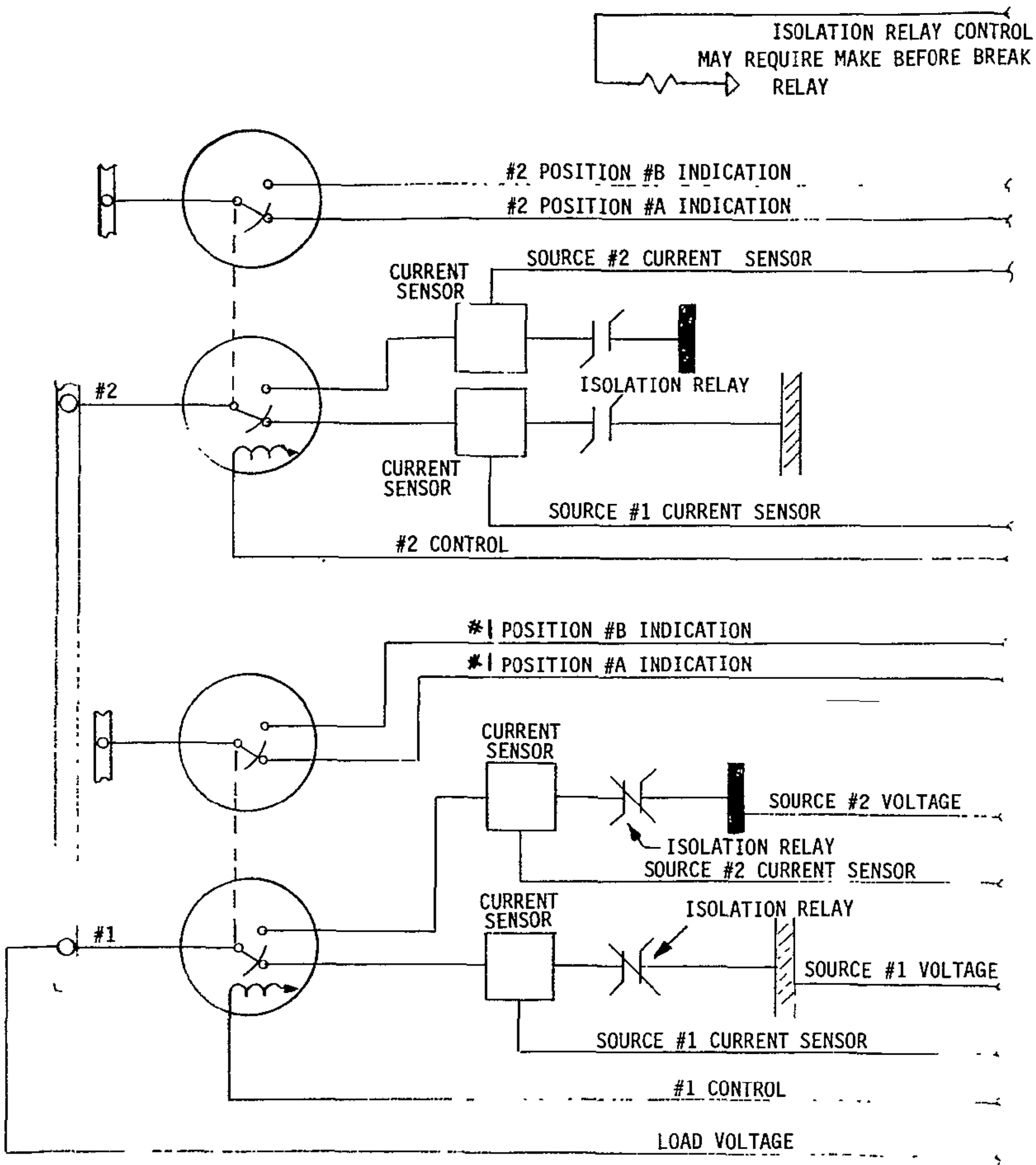


Figure 2-136. Motor-Driven Switch, Total Need for Redundancy

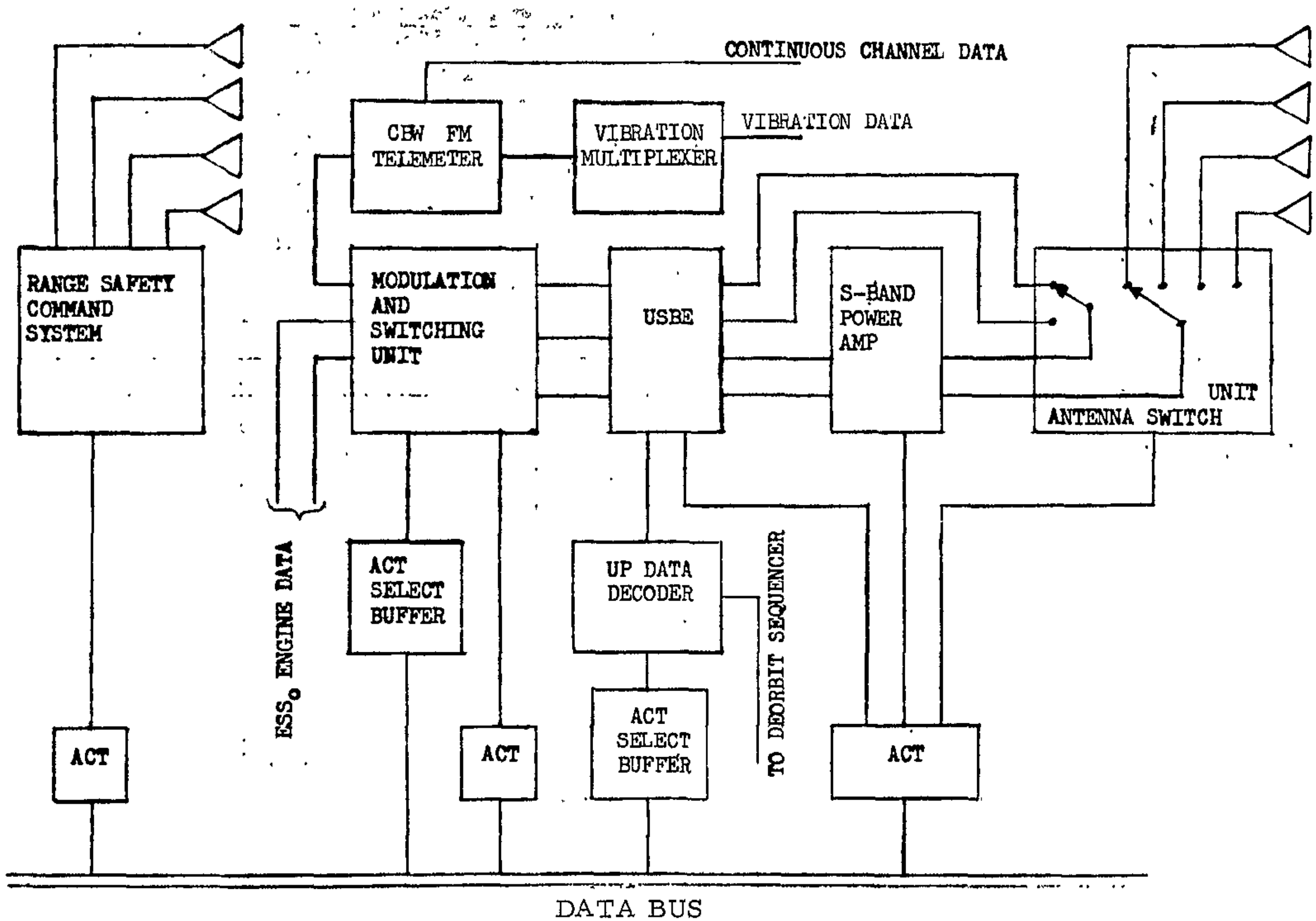
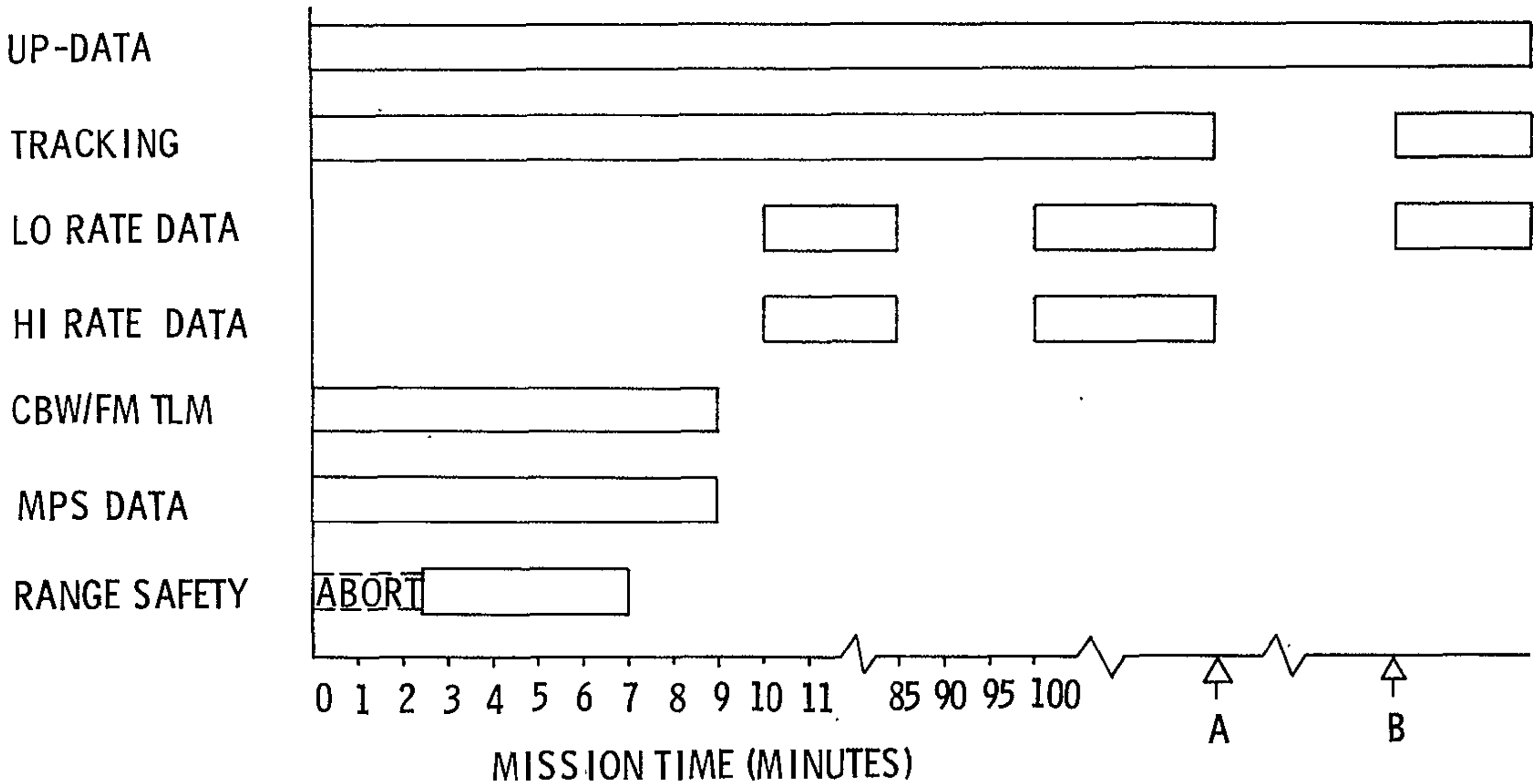


Figure 2-137. ESS Communication Block Diagram



A ORBITER CONTACT (NO ACTIVE SYSTEMS)

B DE-ORBIT

Figure 2-138. ESS Communications Operational Timelines

Data Subsystem. The data subsystem provides the ability to format up-data for acquisition by the data bus, condition data bus signals suitable for modulation of the S-band transmitters, transmit engine and CBW/FM data and select redundant transponder and the required modulation mode for the mission phase.

The up-data system is operational throughout the ESS mission, being required for communications with all ESS systems through the data bus. The data relay satellite system (DRSS) provides the relay link with Manned Spaceflight Network (MSFN) during periods whenever direct contact with the ESS is not possible.

The modulation and switching unit places the data bus information, CBW/FM information, and engine data on subcarriers for modulating the S-band transmitters. The switching function to support the redundancy requirement of the data transmission system is also performed by the modulation and switching unit.

The CBW/FM telemetry provides the capability to transmit high frequency data, frequency multiplexed, and summed with the engine data subcarrier for transmission by the S-band FM transmitter. A system of 22 data channels is proposed, and with the vibration multiplexer a greatly increased number of measurements may be transmitted. This telemetry system is active throughout the boost phase of the ESS mission. See Figure 2-138.

RF Subsystem. The RF subsystem provides the ability to receive up-data from MSFN or the Tracking and Data Relay Satellite (TDRS), receive and provide coherent Pseudo-random Noise (PRN) and doppler tracking and ranging data, and transmit telemetry signals to MSFN, either direct or by the TDRS. The RF system normal mission sequence is dictated by the data system. Selection of antenna is controlled by the central processor.

The S-band transponders are operational throughout the orbital insertion phase; however, only a stand-by function is served by one of the units. After orbital insertion, one unit will operate throughout the ESS mission to provide telemetry data and to receive up-data and tracking data via the TDRS.

The S-band FM transmitter is utilized to transmit engine data and CBW/FM telemetry data to MSFN. This unit operates during the orbital insertion phase of the mission and does not operate during periods when high data-rate transmission is not required. The data bus information may be transmitted via the S-band FM link as an alternate transmission mode.

The S-band power amplifiers are utilized to increase the RF power from the transponder and transmitter to a power level sufficient for data to be received at MSFN. During the orbital insertion phase, the primary power amplifier is providing 20 watts to the antenna system. The secondary power is for backup. During the remainder of the mission, only one power amplifier is utilized; half of the system is on stand by or providing high data rate FM link amplification, and the other half of the primary link operates at the low power level (5 watts) to provide low data-rate telemetry transmitter amplification to MSFN or TDRS.

The antenna switch unit (ASU) provides the ability to select the appropriate antenna for transmission of data through control of the central processor. The selection depends upon the ESS attitude, and the selection of the S-band receiver to receive up-data and ranging signals.

Range Safety Command Subsystem. The ESS range safety subsystem provides a sure means of terminating the vehicle flight subsequent to ESS/booster separation by up-link command to the ordnance system in the event of deviation from the preplanned trajectory. Figure 2-139 is a block diagram of the range safety command subsystem.

The range safety subsystem will be functional from booster/ESS separation, plus 15 seconds, through main propulsion system engine cutoff. After achievement of orbit, the range safety subsystem will be secured. The range safety command system uses a language in which each character of the address and function word is chosen from an alphabet of 21 characters. The 21-character alphabet is formed from a subalphabet of seven symbols taken simultaneously two-at-a-time. Each symbol is an audio frequency tone in the modulation bandwidth of the transmitter. A sequence of 11 dual-tone bursts comprises an "on" command message. Each address character is chosen by use of identical code plugs, one in the ground encoder and the other in the vehicle decoders. When a valid complete message has been received, the decoder relay channel corresponding to the transmitted command will be energized and a termination signal has been achieved.

In consonance with ESS goals, use of highly reliable, flight-proven equipment with the necessary redundancy was to be an objective. It was therefore determined that the Saturn S-II equipment be selected as the technical base for ESS. The S-II range safety system has been thoroughly flight proven, will not require any major vehicle changes, and is available. It also meets the requirements for range safety systems for vehicles launched at KSC as defined in the Air Force Eastern Test Range Manual, AFETRM 127-1.

Propellant Dispersion. The propellant dispersion system proposed for the ESS vehicle will be the same as that presently used on the S-II; it is a fully qualified system and meets the requirements of AFETRM 127-1. Figure 2-140 is a block diagram of the proposed ESS propellant dispersion system.

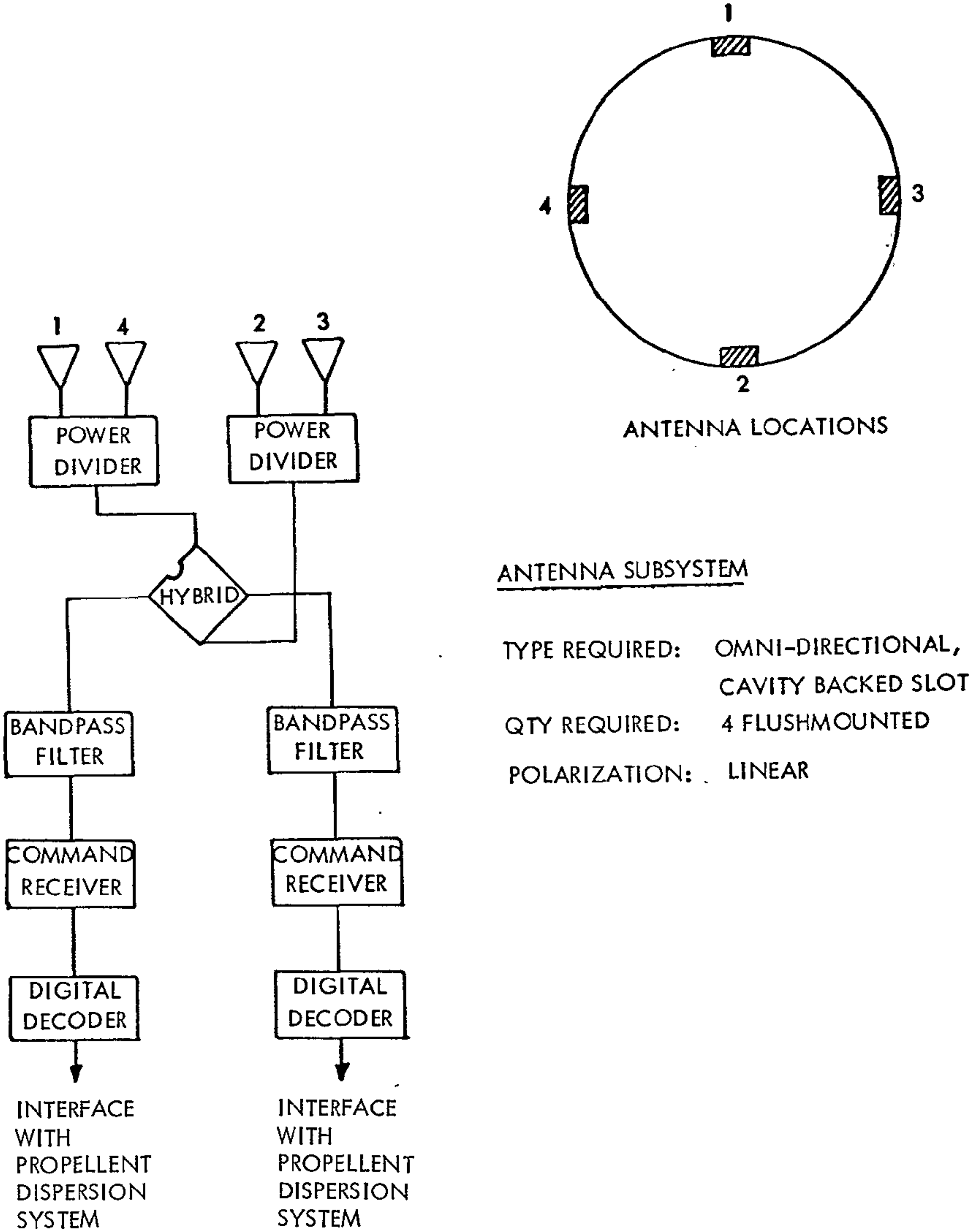


Figure 2-139. Range Safety Command Subsystem

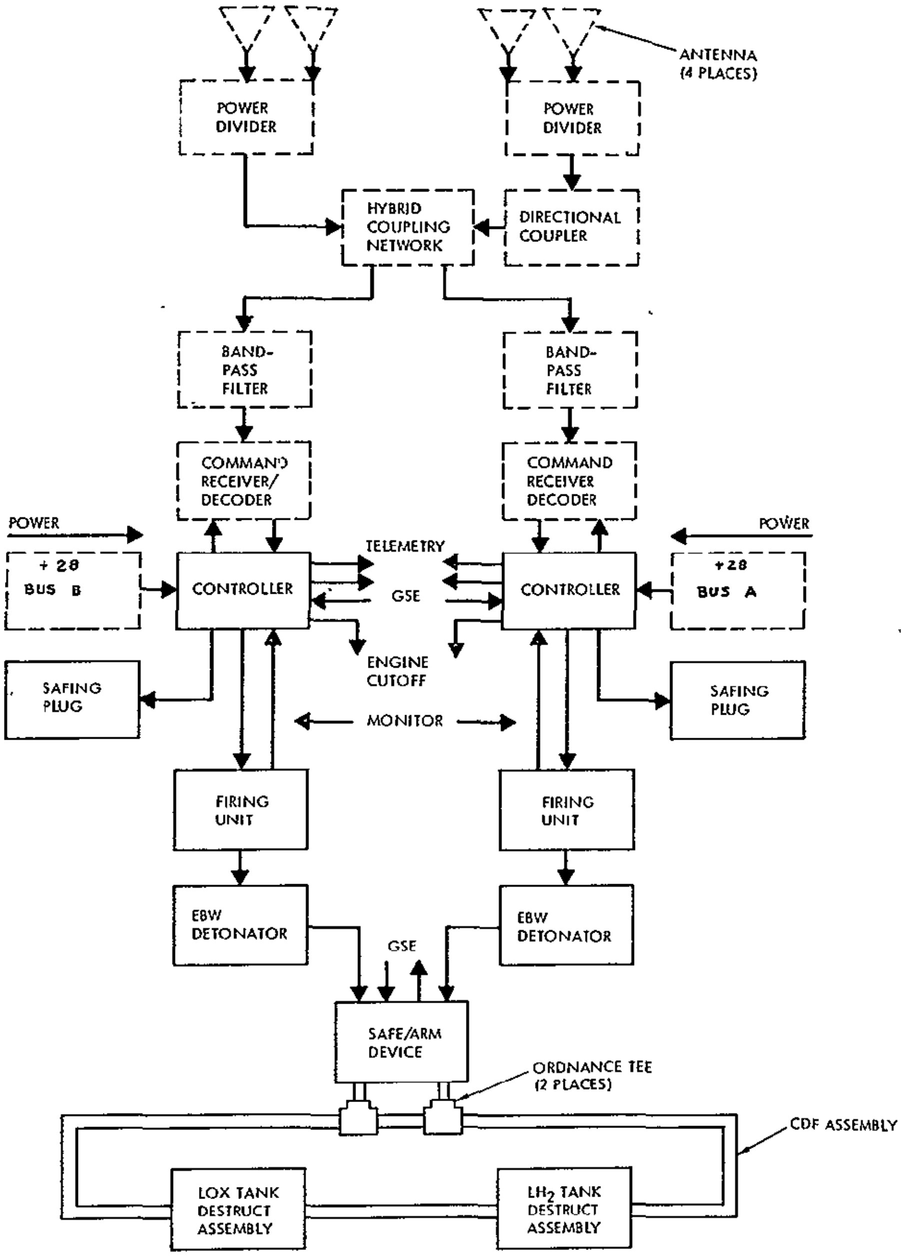


Figure 2-140. Propellant Dispersion System Block Diagram

The propellant dispersion system is to be inhibited from being armed until booster-ESS separation plus 15 seconds. Then propellant dispersion is accomplished as follows. After the secure range safety command system (SRSCS) receives a command for propellant dispersion arm and engine cutoff, a pulse signal is sent to the range safety system controller (RSSC) which closes a set of relay contacts, arming an exploding bridgewire (EBW) firing unit within 1.5 seconds. Upon receipt of the first command, an additional signal is sent from the RSSC to the stage engine cutoff system for shutdown of the main engines.

The second step, the range safety propellant dispersion command, triggers the EBW firing unit. This releases its stored-up energy to the EBW detonator, and detonates an explosive column in the safety and arming (S/A) device. The shock wave propagates through the S/A device and passes to the CDF assemblies, thereby transferring the explosion to the tank destruct assemblies. The tank destruct assemblies are linear-shaped explosive charges that rupture the propellant tanks and disperse the propellants. Either of the two independent systems is capable of imposing zero thrust and initiating propellant dispersion on the ESS.

Another range safety radio command is used to "safe" the system by removing the 28-vdc supply to the receivers and decoders. This will be accomplished once orbit insertion has been assured.

Communications Subsystem Description

Data Component Description.

Up-Data Decoder. The up-data decoder contains the 70 kHz discriminator and the up-data converter. Figure 2-141 is a block diagram of the up-data register. The 70 kHz discriminator and the up-data converter are Apollo equipment.

The up-data are received through the antenna selected by the antenna switch and routed to the receiver where the 70-kHz subcarrier containing the up-data is separated from the composite signal and routed to the up-data converter. Either receiver may be utilized to receive up-data; the modulation and mixing ACT selects the receiver from redundancy and configuration data. The 70-kHz discriminator detects the 2-kHz up-data for demodulation by the sub-bit detector. The 1-kbps data are derived from the composite by the sub-bit detector; however, five sub-bits comprise a single command bit. The function of assembling the 150 sub-bit message into a 30-bit command message is performed by the up-data decoder. One bit of the five sub-bit sequence contains the necessary information, the remaining four being used for determining sub-bit error or message validity. The command message is transmitted serially to the up-data ACT select buffer. The 200-cps clock

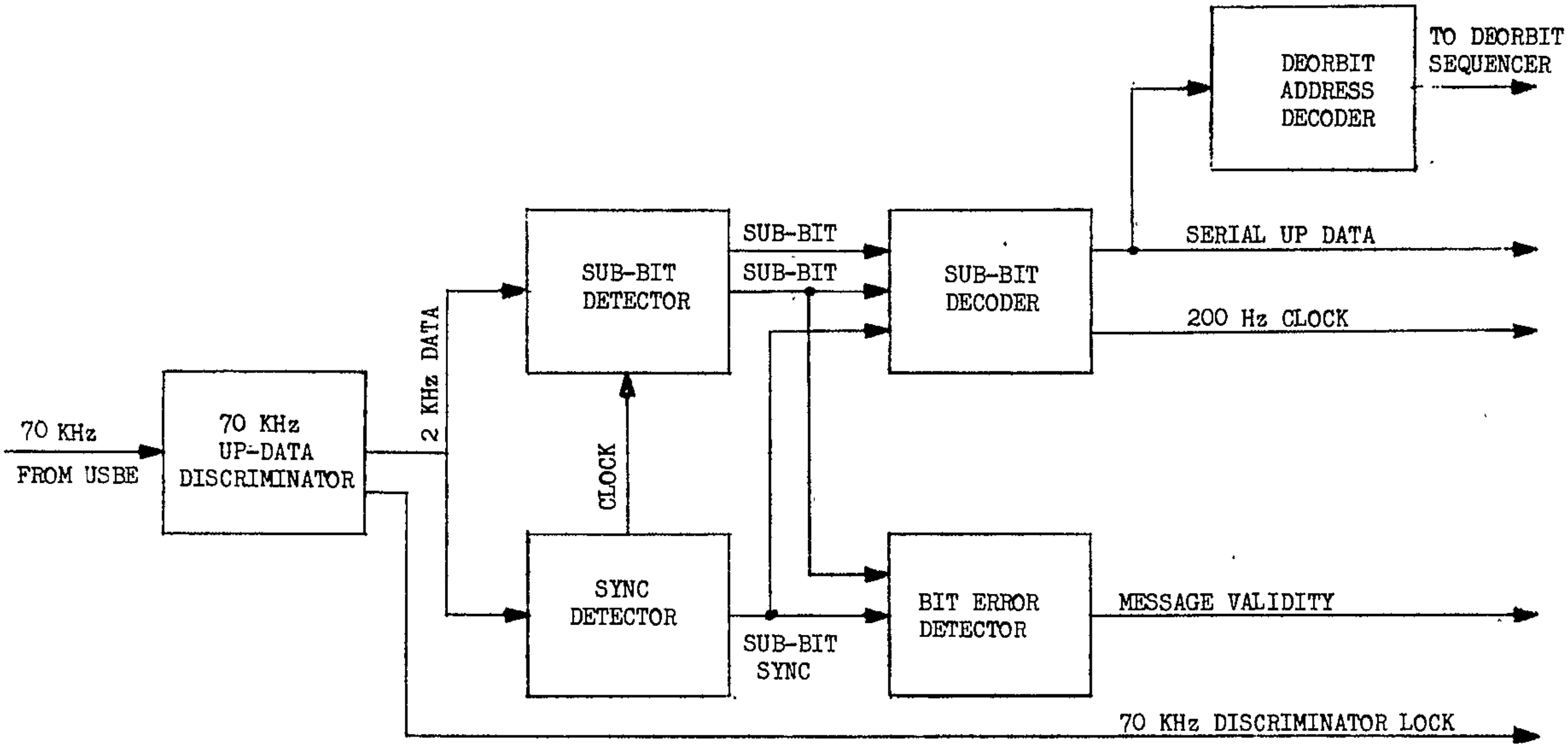


Figure 2-141. Up-Data Decoder Block Diagram

is utilized to clock the serial data into the ACT select buffer. An interrupt, generated by the 70 kHz discriminator when in phase lock, is sampled by the data bus to inform the DMC when up-data are being read into the ACT select buffer. A message validity interrupt is used to determine if noise quantity is such that reliable updata cannot be received.

Modulation and Switching Unit. The modulation and switching unit comprises the modulation selector module, the subcarrier modulator assemblies, the modulation emphasis and mixing module, and the amplifier and transponder selector module. Figure 2-142 is a block diagram of the modulation and switching unit.

The modulation selector module contains the limiting and shaping amplifiers and gates for the 10-kbps engine data and the data bus information. Here the input format to the modulator assemblies is selected and the selected data are shaped and limited to the proper amplitude. There are three modes selectable: (1) 10-kbps engine data, (2) data bus information, and (3) CBW/FM information to the input of the modulators.

The telemetry alternate mode switches the data bus information to the input of the 1.25 mHz modulator. There are three biphase modulators to provide the subcarriers for the engine data and data bus information. The 1.0 mHz modulator conditions the data bus information during all mission phases. The 1.25 mHz modulators are utilized during the boost phase only and conditions engine 10-kbps data. The 1.25-mHz modulator provides redundant capability for transmission of data bus information in the telemetry alternate mode. During this mode, engine data for only one MPS engine are transmitted.

The modulation emphasis and mixing module contains the switchable emphasis networks to provide optimum deviation of the transmitters under the various modulation modes. A redundant amplifier is used to ensure ability to modulate the transmitter to support the deorbit phase of the mission. The amplifier and transponder selector module provides the capability to switch the modulation input to either transponder.

CBW/FM Telemeter. The CBW/FM telemeter consists of a conventional system of subcarrier oscillators and mixer amplifiers to frequency-multiplex subcarriers containing high frequency data. The pre-emphasis schedule is set up by amplitude adjustment of the subcarrier oscillator. The subcarriers have a data frequency response of 1000 Hz; however, lower frequency response with an increased amplitude accuracy is available.

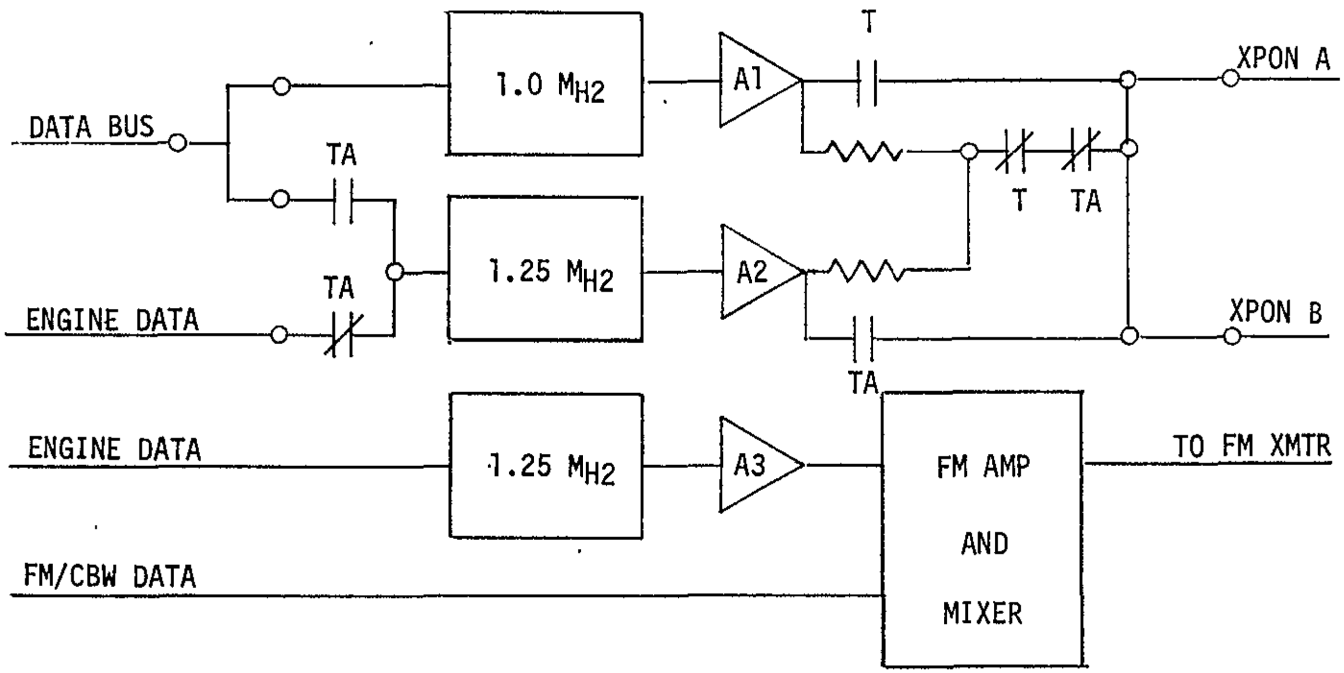


Figure 2-142. Modulation and Switching Unit

To provide for an increased number of vibration type data channels, a time division multiplexer is proposed. This vibration multiplexer contains a program section to provide synchronization and timing data for identifying the multiplex signal segments.

RF Component Description.

Unified S-Band Equipment (USBE). The USBE consists of two transponders, an FM transmitter, and a power supply in a single package. The USBE is used for tracking and ranging, transmission of data bus information extracted by the ACT select buffer, and reception of up-data. Figure 2-143 is a block diagram of the USBE.

The USBE tracking method employed is the two-way or double-doppler method. In this technique, a stable carrier of known frequency is transmitted to the ESS where it is received by the phase-locked receiver, multiplied by a known ratio, and then re-transmitted to the MSFN for comparison. Because of this capability, the USBE is also referred to as the S-band transponder.

For determining ESS range, the MSFN phase-modulates the transmitted carrier with a PRN binary ranging code. This code is detected by the ESS USBE receiver and used to phase-modulate the carrier transmitted to the MSFN. The MSFN receives the carrier and measures the time delay between transmission of the code and reception of the same code, thereby obtaining an accurate measurement of range. Once established, this range can be continually updated by the double-doppler measurements discussed earlier. The MSFN can also transmit up-data commands to the ESS USBE by means of the 70 kHz subcarrier.

The USBE transponder is a double-superheterodyne phase-lock loop receiver that accepts a 2106.4-mHz, phase-modulated RF signal containing the up-data subcarrier and a PRN code when ranging is desired. This signal is supplied to the receiver via the triplexer in the antenna switching and multiplex unit.

When operating in a ranging mode, the PRN ranging signal is detected, filtered, and routed to the USBE transmitter as a signal input to the phase modulator. The input frequency is coherently detected and used as the reference frequency for receiver circuits as well as for the transmitter.

The coherent amplitude detector (CAD) provides the automatic gain control (AGC) for receiver sensitivity control.

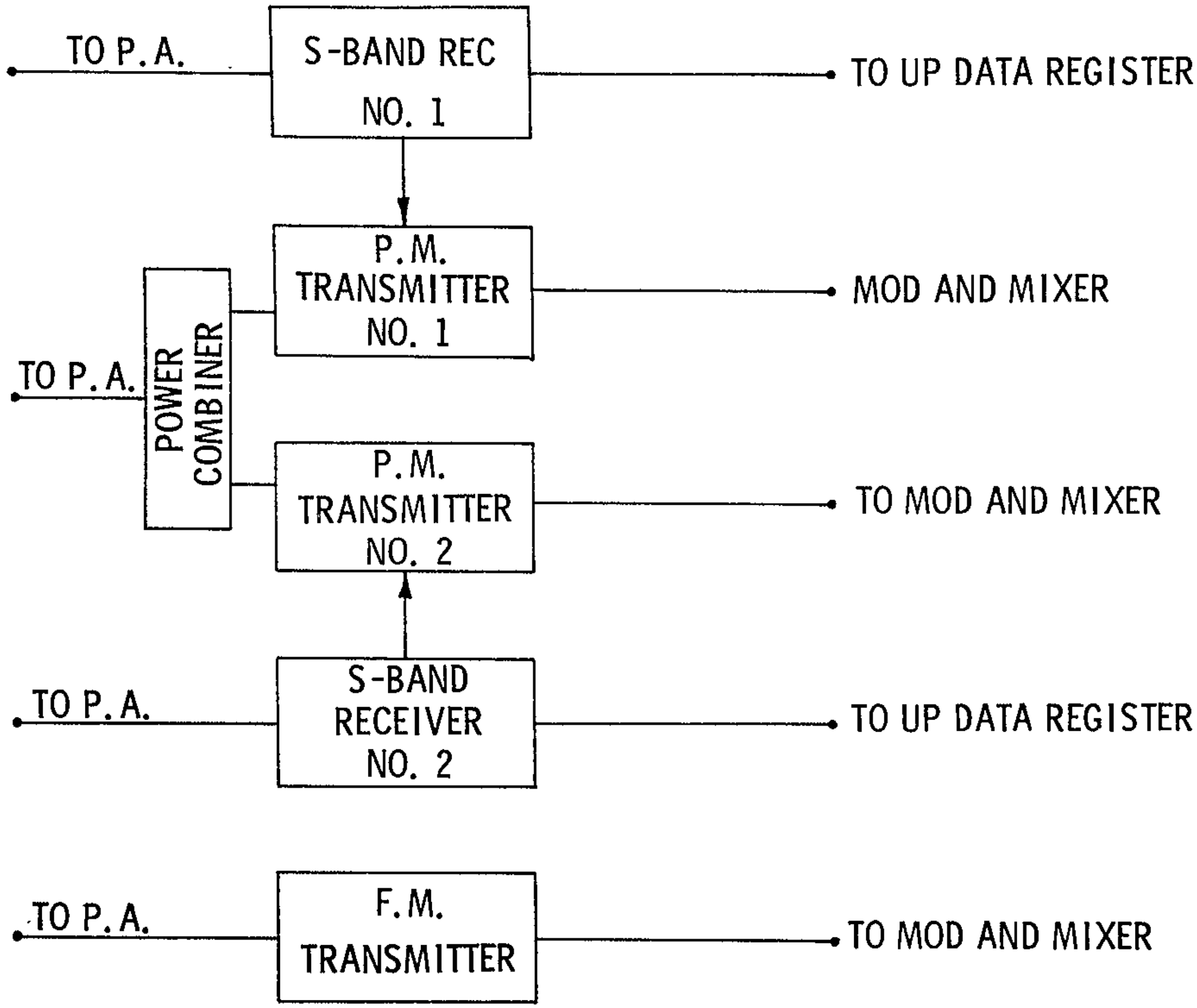


Figure 2-143. USBE Block Diagram

The USBE transponders are capable of transmitting a 2287.5-mc phase-modulated signal. The initial transmitter frequency is obtained from one of two sources: the voltage-controlled oscillator (VCO) in the phase-locked USBE receiver or the auxiliary oscillator in the transmitter. Selection of the excitation is controlled by the CAD. If ranging has been selected, the up-link information is routed from the receiver wide-band detector to the phase modulator in the transponder transmitter. Additionally, data from the modulation and switching unit are used to phase-modulate the carrier. The output power is approximately 250 mw.

The USBE also contains a separate FM transmitter, operating at 2272.5 mHz, permitting transmission of engine data and providing backup for transmission of the data from the ACT select buffer. Output power of the FM transmitter is 100 mw.

S-Band Power Amplifier. The S-band power amplifier (PA) equipment amplifies the RF output from the USBE transmitters when additional signal strength is required for adequate reception of the S-band signal by MSFN. The amplifier equipment consists of a triplexer, two traveling-wave tubes for amplification, power supplies, and the necessary switching relays and control circuitry. The S-band PA is contained in a single electronics package. Each PA requires about 15 watts for warmup, 45 watts at low power and 90 watts at high power of 3-phase 400-cycle ac power and about 2.5 watts 28-vdc power. Figure 2-144 is a block diagram of the S-band power amplifier.

All received and transmitted S-band signals pass through the S-band PA triplexer. The 2106.4-mc S-band carrier received by the ESS enters the S-band PA triplexer from the S-band antenna equipment. The triplexer passes the signal straight through to the USBE receiver. The 2287.5-mc output signal from the USBE transponder enters the S-band PA where it is either bypassed directly to the triplexer and out to the S-band antenna equipment, or amplified first and then fed to the triplexer. There are two PA modes of operation: low power and high power. The high-power mode is automatically chosen for the PA connected to the FM transmitter.

Antenna Switching Unit. The antenna switch unit provides the ability to select the proper antenna for optimum transmission and receiving of signals destined for or arriving from MSFN. The selection of the antenna is controlled by the central processor from data determining the look angles.

The antenna switching unit consists of a single-pole 4-throw coaxial switch for antenna selection to the triplexer in the S-band power amplifier and a single-pole 2-throw coaxial switch for selection of the S-band receiver to be utilized for ranging and up-data received.

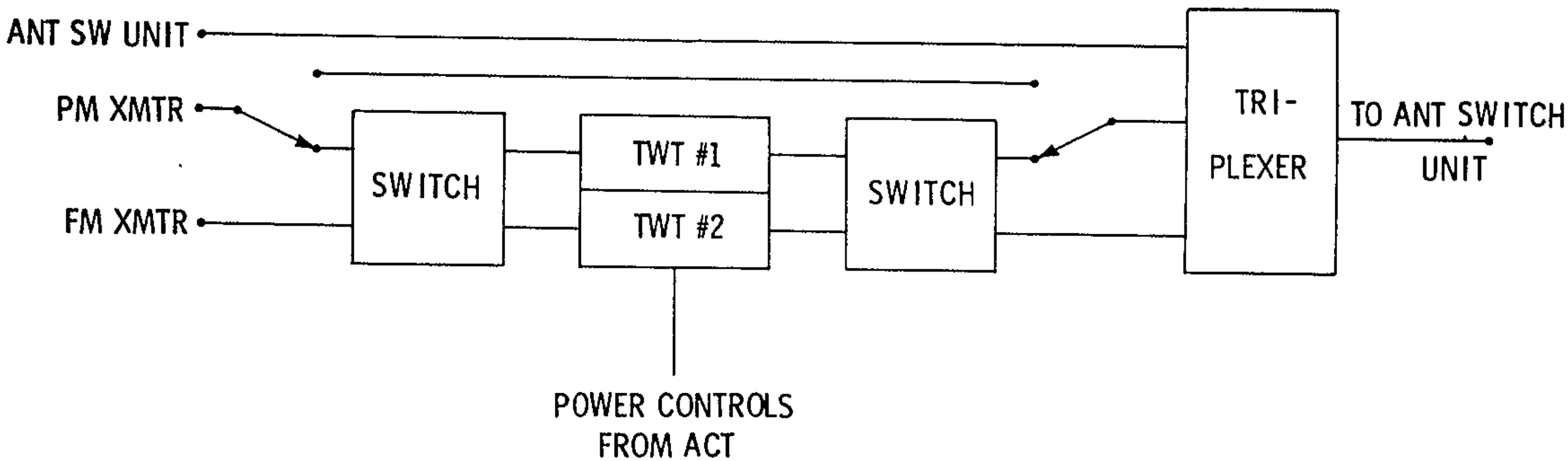


Figure 2-144. S-Band Power Amp Block Diagram

Range Safety Command Receiver. A block diagram of the range safety command receiver is shown in Figure 2-145.

The UHF FM signals induced in the antenna system are coupled to a dual-cavity, tunable range of 406 to 450 mHz. The tuned dual-cavity preselector functions as an RF bandpass filter whose output is amplified and applied to the first mixer. The RF signal is heterodyned in the first mixer with a signal three times the frequency of the local oscillator. The signal output of this mixer is amplified and heterodyned in the second mixer with the local oscillator signal. The second mixer output is coupled to the intermediate frequency (IF) bandpass filter. This filter determines the overall receiver bandpass characteristics. The filter output is coupled to a three stage IF amplifier, which in turn feeds a limiter and a standard Foster-Seely discriminator. The detected audio signal from the discriminator is coupled to an audio preamplifier, then to a driver that feeds the push-pull audio output stage; the output stage has two isolated outputs each of which may be used to feed the range safety decoder.

Decoder Operation. Figure 2-146 is a block diagram of the range safety command decoder.

The composite audio from the receiver is fed to the decoder isolation transformer. The transformer feeds a push-pull filter driver stage that drives seven filters. Once the filter output reaches the proper level, the threshold decoder will produce an amplitude-standardized pulse which is then fed to a 7 by 21 AND matrix. The actual threshold level is raised to an optimum level by use of an unbias circuit, the optimum decision circuit. This circuit operates by sampling the peak voltage in the filter primary and instantaneously developing a dc bias proportional to this voltage. This bias is then fed forward and used to set the threshold decision level of the detector to some constant ratio below the peak voltage output of the filter. This technique allows the decoder to operate reliably in a much wider signal input dynamic range than would be otherwise possible.

The 21 outputs of the AND matrix are routed to the code plug which unscrambles the code in the same fashion in which it was scrambled by the ground encoder plug. The code plug output is applied to the address error detector, which acknowledges the correct address word by supplying, for a specific time, one input to each AND output relay driver. The second input to each AND output relay driver is the command word from the 7 x 21 AND matrix.

When a valid complete message has been received, the decoder relay channel corresponding to the transmitted command will be energized, and 28 volts will appear at the proper output pin for approximately 25 milliseconds

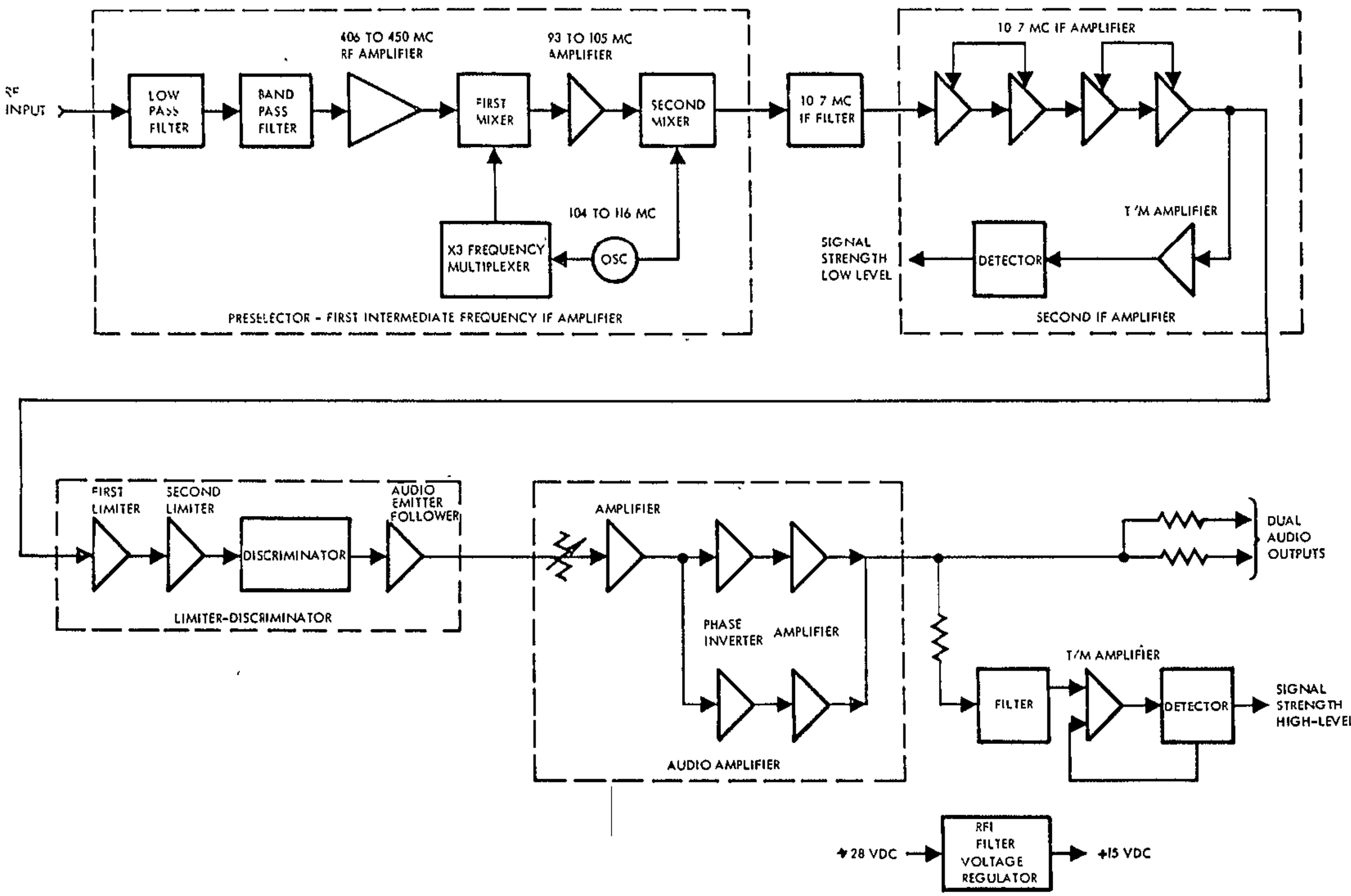


Figure 2-145. Range Safety Command Receiver

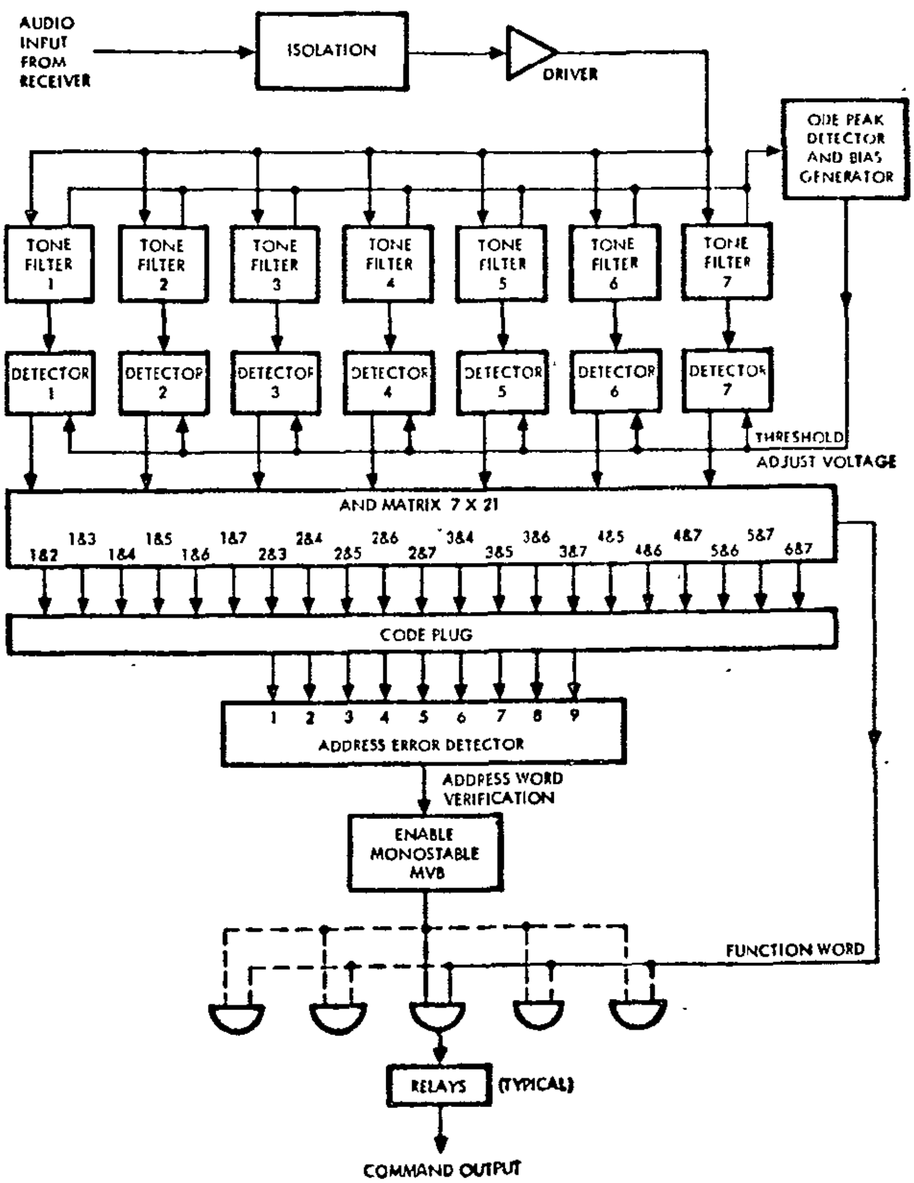


Figure 2-146. Range Safety Command Decoder

This 28-volt signal is fed to the range safety controller for execution of the propellant dispersion commands.

Very High Frequency (VHF) Antenna. The range safety command antenna subsystem operates at 450 mHz with sufficient coverage to permit remote control of flight termination under any flight conditions.

The use of a hybrid junction in the antenna subsystem provides optimum means of coupling four antennas with two receivers. A high degree of isolation between receivers is maintained at all times while the bandpass filters provide additional system selectivity. Ground checkout is achieved through the directional coupler.

To achieve a near-omnidirectional system, the antennas are placed at approximately 90-degree intervals around the ESS periphery. The polarization will be linear to ensure a maximum polarization loss of 3 db when used in conjunction with the circularly polarized ground system.

Propellant Dispersion.

Range Safety System Controller. The controller contains the relay logic for control of the following: power to the receiver-decoder subsystem; arming and triggering of the firing unit; measurement signals; and engine cutoff command.

Exploding Bridgewire Firing Unit. The EBW firing unit, a capacitance discharge system, initiates the PD subsystem ordnance components to disperse the propellants. The unit consists of a charging circuit, a regulator, a trigger circuit, and a switching device.

Upon the EBW arm command, 28 vdc is applied to the charging circuit of the EBW firing unit. The charging circuit converts the 28-vdc input to a 2300-volt charge stored in a one-microfarad capacitor. The trigger circuit releases the stored charge of the capacitor into an EBW detonator. This initiates the PD explosive trains.

Exploding Bridgewire Detonator. The EBW detonator is an electrically initiated ordnance device used to trigger other ordnance devices and explosive ordnance trains. When the high-energy pulse (high voltage) of the EBW firing unit is applied to the bridgewire of the detonator, the wire explodes and vaporizes, rapidly releasing a large amount of energy. The released energy initiates the explosive charge of the detonator (which is in contact with the bridgewire). The output pulse (shock wave) of the explosive charge is sufficient to initiate the explosive columns of the safety and arming device.

An air gap is built into one input lead of the detonator. It allows the high-energy pulse of the firing unit to pass to the bridgewire but blocks accidental voltage applications of up to 500 volts dc or 250 volts ac.

Safety and Arming Device. The safety and arming device, on electrical command through the umbilical, aligns two explosive columns in the rotor shaft with two EBW detonators and two confined detonating fuse (CDF) assemblies installed in the device. The device can also be safed manually in the event of power failure. Markings on the end of the housing provide a visual means of determining shaft position; electrical position-indicating switches provide indication through the umbilical.

Confined Detonating Fuse. The CDF consists of a low-energy mild-detonating fuse (MDF) within a multilayer protective sheath designed to confine all of its explosive effects. A CDF assembly consists of an appropriate length of CDF and two end fittings. Each end fitting is provided with a threaded connector for attachment of the CDF assembly to various ordnance components.

Ordnance Tee. The ordnance tee is an explosive device employed to transfer the detonation from one CDF assembly to two other CDF assemblies. To assure initiation of the two other CDF assemblies, the tee contains a small explosive charge.

Liquid Hydrogen Tank Destruct Assembly. The LH_2 tank destruct assembly consists of two sections of linear-shaped charge (LSC), a charge coupler, 22 sliding-type mounting clamps, and two CDF receptacles. Each section of the LSC is 15 feet in length and approximates a 120-degree V in cross section. Upon detonation, the destruct charge cuts a 30-foot vertical opening in the LH_2 tank.

The destruct charge installed in the S-II system tunnel is attached to the tunnel intercostals by sliding mounting clamps. The charge coupler is a fixed mounting clamp, also attached to an intercostal of the system tunnel. Friction clamps bolted to the intercostals allow the destruct charge to slide in the clamp upon thermal expansion or contraction of the stage structure.

LO_2 Tank Destruct Assembly. The LO_2 tank destruct assembly is a Teflon-covered linear-explosive charge. Each end of the charge is covered with a metal ferrule; one ferrule contains a booster charge, the other a metal eye used for loading the charge into the destruct charge mounting tube. The charge assembly consists of two linear explosive charges, two LO_2 charges and CDF assembly adapters, and an explosive charge-loading line.

The LO_2 tank destruct charge installation mounts two linear destruct charges in a figure-eight-shaped tube. Upon detonation of the charges, the tube (because of its configuration) performs the function of a double-V linear-shaped charge. The tube is mounted and attached to the inner side of the S-II aft skirt/interstage structure at Station 266.5, and makes a 13-foot opening in the LO_2 tank and structure.

Trade Studies and Performance Analysis

Redundancy Trade Study. The communications subsystem is critical to the ESS mission in the rendezvous and deorbit phases in that telemetry is necessary to communicate ESS systems status and platform position, and tracking data. Updata is necessary to provide the ability to command deorbit functions and timing parameters. The components utilized to provide these functions are Apollo system equipment, and redundancy is provided on portions of those systems that are less reliable than required for Apollo spacecraft operations.

Item 1 of the paragraph entitled "Vehicle Characteristics" in Section III of the NASA Study Control Document (Pages 2 and 2A) exempts systems using hardware of Saturn and Apollo reliability standards from the FO/FS criteria applied to new subsystems. However, under the same item, a statement requires review of these systems and study of alternate approaches to establish FO/FS criteria. Figure 2-147 depicts the existing Apollo S-band communications system. From this block diagram, it is determined that the following components, if inoperative, would result in the ESS deorbit capability being lost:

Antenna

Antenna switch

Receiver switch

By-pass switch

Power amplifier switch

Triplexer

Power combiner

These items are inherently of high reliability due to their simple nature and minimum operational components.

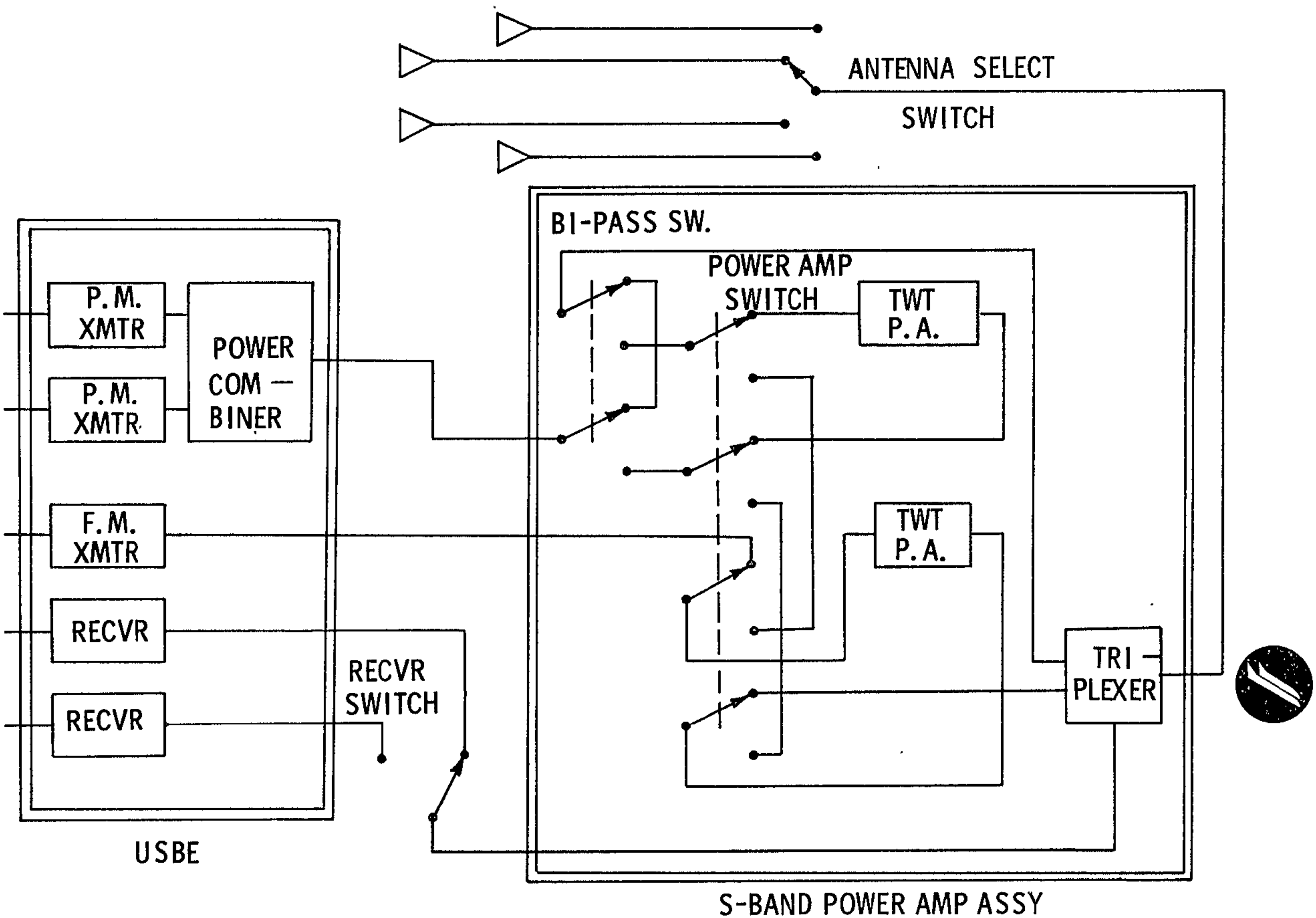


Figure 2-147. Existing Apollo S-Band Communication System

Figure 2-148 depicts a system of entirely redundant capability and many switching modes to allow several failures to occur and retain deorbit capability. While this appears to have solved the case of non-redundant paths in the Apollo hardware, there are a number of disadvantages brought about that may overbalance the advantage gained by redesign. These disadvantages are:

15 controls as opposed to 4 controls

Complex fault isolation program required

Large number of response verification measurements

Modification of forward skirt structure to accommodate additional antennas

An increased number of components, thereby lowering overall reliability

The advantages of the redundant system are:

Minimum of two failures for communications system

Capability to maintain both receivers operating for more rapid response to failed receiver switching

Ability to select antenna for optimum transmission independent of onboard look angle program

The advantages of the redundant system and corresponding redesign of existing components and qualification of new components were weighed and at this time it appears these advantages do not warrant the redesign required. If the redundant system were required, the components utilized are existing hardware and the cost increase would be in the structural modifications of the forward skirt and analytical tests to support antenna pattern analysis.

RF Power Budget Analysis. The following parameters are used in calculating the RF system power budget and are derived from the basic mission profile and the data requirements to support mission communications:

270 nautical-mile

24,000 nautical-mile range to DRSS

DRSS receiver noise figure (N_F) = 4 db

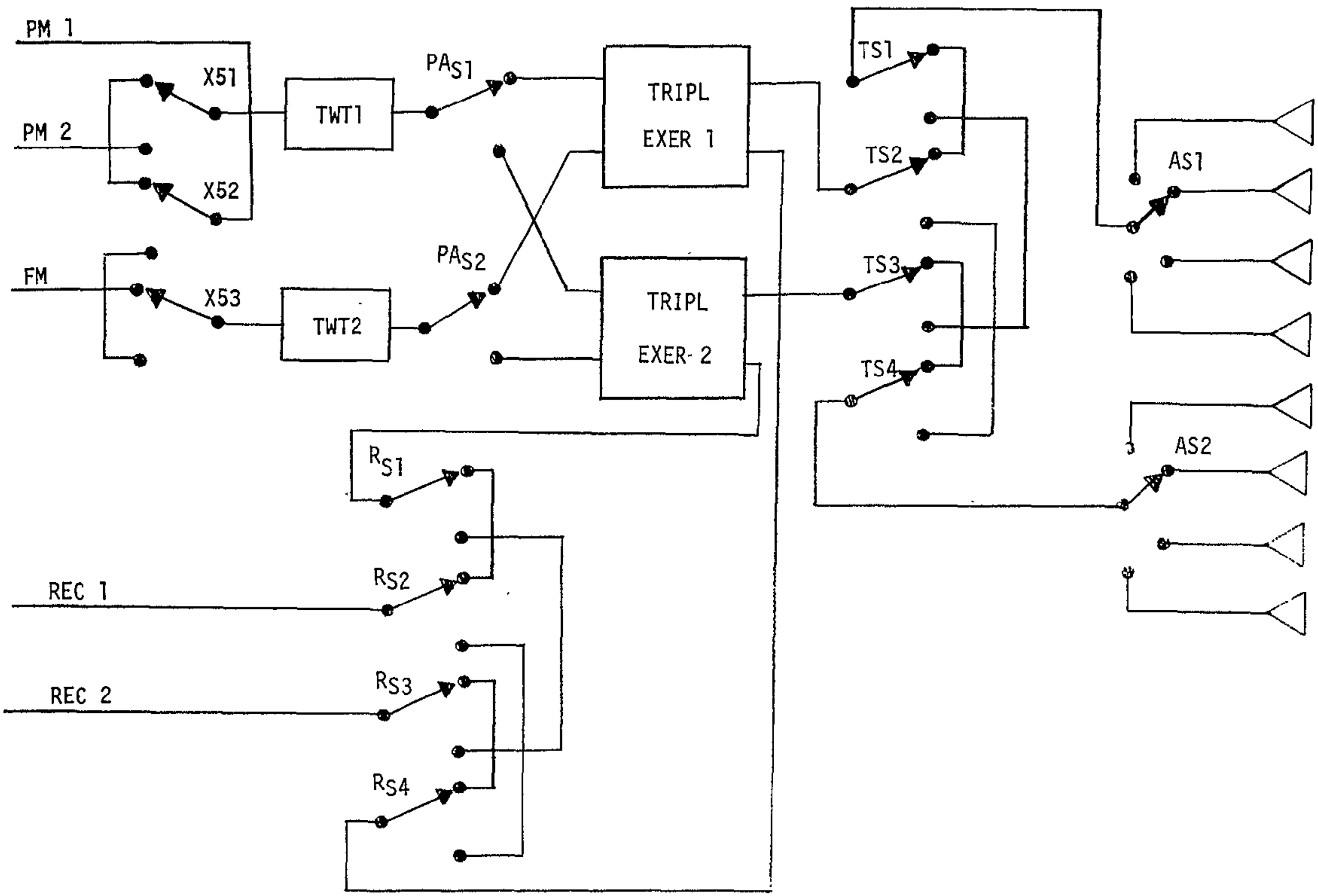


Figure 2-148. Redundant S-Band Communication System

Transmitter power (P_T) = 20 watts or 13 dbw

DRSS antenna gain (G_R) = 32.3 db

DRSS transmitter power = 20 W or 13 dbw

Transmitter and receiver losses (L) = 4 db

Antenna noise temperature (T) = 300°K

ESS antenna gain (G_T) = 0 db

S/N Ratio required (S/N) = 12 db

MSFN antenna gain (30') = 43 db

MSFN transmitter power = 10 KW = 40 dbw

The power budget parametric equations and constants are shown in Table 2-29. The ability to receive and demodulate data depends upon the antenna noise temperature, the bandwidth of the data, and the receiver noise figure (N_F). The antenna temperature (300 F) and the receiver noise figure are assumed constant for this study, the variable (bandwidth) being the controlling factor for data quality. The subcarriers and corresponding bandwidths are shown in Table 2-30 and are used to derive the receiver noise power for the various data subcarriers listed in that table. A definition of the ESS communications modes and calculations of the carrier power during those modes are shown in Table 2-31. The parameters used in determination of the carrier power are the transmitted power, antenna gain, path loss and system losses.

Table 2-32 contains the signal-to-noise and margin tabulation for each subcarrier of each communication mode. Adequate margin exists for all subcarriers during the boost phase for all RF links. For communications via the Data Relay Satellite System, the signal margins are below that desired; however, additional considerations for modulation techniques, probable higher power from the data relay satellite, redistribution of power into the subcarriers, and signal-to-noise improvement from receiver characteristics can improve these margins to acceptable values.

Table 2-29. Power Budget Parametric Equations and Constants

Receiver noise power (P_N) = KTB
Receiver sensitivity (S_R) = $P_N + N_F$
Space loss (A_P) = 37.8 + 20 Log f (in MHz) + 20 Log D (in nmi)
$P_T = S_R + A_P + L - G_T - G_R + S/N$
Modulation loss = 10 log P_T/power in subcarrier
Subcarrier power received = P_T + mod loss - $A_P + L - G_T - G_R$
K_T = -204.8 dbw + 10 Log B
A_P for 1000 nm = -164.6 db
A_P for 24,000 nm = -191.2 db
$K = 1.38 \times 10^{-23}$ watt-sec
T = 300 K

Table 2-30. Receiver Noise Power for Subcarriers (KTB)

Data	Bandwidth (KHz)	P_N (KTB)
Up-data	6	-167.0
PRN ranging (up)	1.8	-172.2
Carrier (PM)	1.0	-174.8
PRN ranging (down)	0.5	-177.8
FM/FM telemetry	400	-148.8
High-rate data bus	50	-157.8
Low-rate data bus	2	-171.8
Engine data	10	-164.8

Table 2-31. Communication Modes and Mission Phase

A. Boost phase PM up-data.

B. Boost phase PM down-data.

C. Boost phase FM down-data.

D. Orbital operation up-data via DRSS.

E. Orbital operation PM down-data via DRSS.

Mode A Carrier Power

P_T = 40 dbw

G_T = 43 db

A_P = -164.6 db

L = 4 db

-85.6 dbw

Mode B and C Carrier Power

P_T = 13 dbw

G_T = 43 db

A_P = -164.6 db

L = 4 db

-112.6 dbw

Mode D and E Carrier Power

P_T = 13 dbw

G_T = 32.3 db

A_P = -191.2 db

L = 4 db

-150 dbw

Table 2-32. Signal to Noise and Signal Margins

Mode	Channel	Mod Loss (db)	Subcarrier Power (dbw)	Receiver Sensitivity (dbw)	Signal Noise (db)	Required Signal/Noise (db)	Margin (db)
A	Up-data	7.7	-93.3	-163.0	69.7	15.5	54.2
	PRN ranging	8.6	-94.2	-168.2	74.0	3.0	71.0
	Carrier	8.3	-93.9	-170.8	76.9	6.0	70.9
B	High-rate data	8.0	-120.6	-153.8	33.2	12.0	21.2
	Engine data	12.5	-125.1	-160.8	35.7	12.0	23.7
	PRN ranging	12.9	-125.5	-173.8	48.3	3.0	45.3
	Carrier	8.9	-121.5	-170.8	49.3	6.0	43.3
C	FM/FM data	0.8	-113.4	-144.8	31.4	12.0	19.4
	Engine data	12.5	-125.1	-160.8	35.7	12.0	23.7
D	Up-data	7.7	-157.7	-163.0	5.3	15.5	-10.2
	PRN ranging	8.6	-158.6	-168.2	9.6	3.0	6.6
	Carrier	8.3	-158.3	-170.8	12.5	6.0	6.5
E	Low-rate data	8.0	-158.0	-167.8	9.8	12.0	-2.2
	PRN ranging	12.9	-162.9	-173.8	10.9	3.0	7.9
	Carrier	8.9	-158.9	-170.8	11.9	6.0	5.9

2.3.4 Electrical Power and Distribution

The Electrical Power and Distribution (EPD) System shall contain the capability to provide electrical energy and distribute this energy for operation of the electrical and electronics systems throughout the ESS 24-hour mission. Internal energy sources are required to provide power from transfer to internal power through deorbit. Redundant and separate power sources and distribution bus systems will be utilized to comply with the ESS vehicle redundancy system concept. The ESS Phase A study evaluated various power sources and conducted a tradeoff that determined that silver zinc batteries will be used for the ESS 24-hour mission. Redundant static inverters will be utilized for DC to AC power conversion for all AC loads.

Electrical Power and Distribution Requirements

1. Provide a +28 vdc primary electrical power and distribution system capable of satisfying ESS system loads from internal power transfer through deorbit.

2. Provide an ac power conversion system capable of operating from a +56 vdc input battery source to produce 115/200 vac, 3 phase, 400 Hz power capable of satisfying ESS vehicle ac loads from internal power transfer through deorbit.

3. Redundant and separate dc power sources, ac power conversion, and distribution bus systems are required to satisfy the ESS vehicle redundancy system concept.

4. Provide the capability for switching from ground power to ESS internal power.

5. Batteries shall be sized and grouped for each of the redundant sources to handle the overall load from liftoff through deorbit for the ESS 24-hour mission.

6. Power sources and buses must be isolated and independent of each other during normal operation.

7. Common loads shall not be connected to more than one bus simultaneously in order to prevent a single short to ground from pulling down two power sources.

EPD System Description

The primary +28 vdc electrical power system is shown in Figure 3-149. Tri-redundant and separate battery sources will be utilized to power three main buses. Any two of the three +28 vdc sources will be capable of handling the total ESS electrical load. Each system contains a power transfer switch to transfer from external to internal power sources. Two ACT units are utilized for control and monitoring the individual +28 vdc power systems. Two power control solid state switches (PCS) are used per system to provide +28 vdc drive to the power transfer switch motor.

Each 28 vdc system is controlled by redundant ESS computers via redundant data buses and ACT units. The power system as shown in Figure 2-149 is connected to internal power. To switch main bus A to external power would require a computer address to ACT unit 1. The ACT output channel will provide a nominal +5 vdc at 10 milliamperes of current to provide input drive to PCS 1. The output of PCS 1 will switch +28 vdc to the motor of power transfer switch No. 1. The transfer switch is a make-before-break type that will switch to the external ESE ground power position. Backup manual commands will be provided by ESE through the umbilical connector to switch to external or internal power. The same procedure will be used to switch bus A to internal power using the computer. In this case, the computer will address ACT unit 1 and select PCS 2 to provide drive to the motor of transfer switch 1 in switching to internal power.

Main bus B operation will be identical to that of bus A as previously described except ACT units 3 and 4 and PCS 3 and 4 will be used. Main bus C operation will also be identical to bus A operation as previously described except ACT units 5 and 6 and PCS 5 and 6 will be utilized. Each bus system will be monitored by individual ACT units. (ACT 2, 4 and 6)

Nine batteries for the main 28 vdc system, will be sized and divided into three groups (3 per bus) so that a battery can fail in each group with no affect on the 24-hour mission power capability. One complete group of batteries for the main 28-vdc system could fail and the other two groups will handle the total ESS electrical load. Figure 2-150 shows a 28-vdc load profile for 24 hours.

Eight batteries (2 per bus) for the 56-vdc inverter power system will be sized and divided into four groups. Two 28-vdc, 200-ampere-hour batteries will be connected in series to each 56-vdc bus. Any two batteries have the capacity to handle the total peak load for one main orbiter engine during the 10 minutes from liftoff through main engine cutoff. Each engine will be connected to two separate 115/200 vac, phase 3, 400 Hz power sources. Each ac source is capable of handling the peak loads for one engine. AC loads for main propulsion system operation are shown in Figure 2-151.

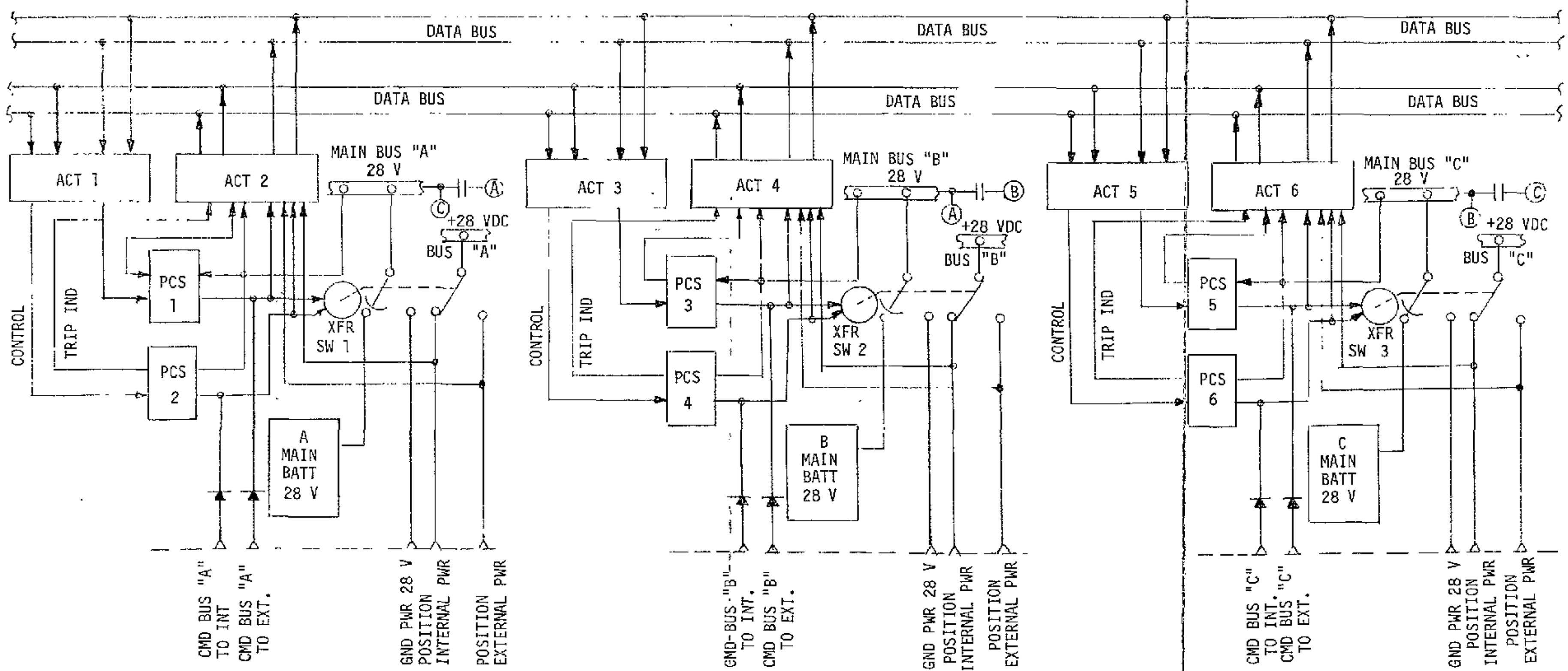


Figure 2-149. ESS Main Electrical Power System (+28 vdc Main)

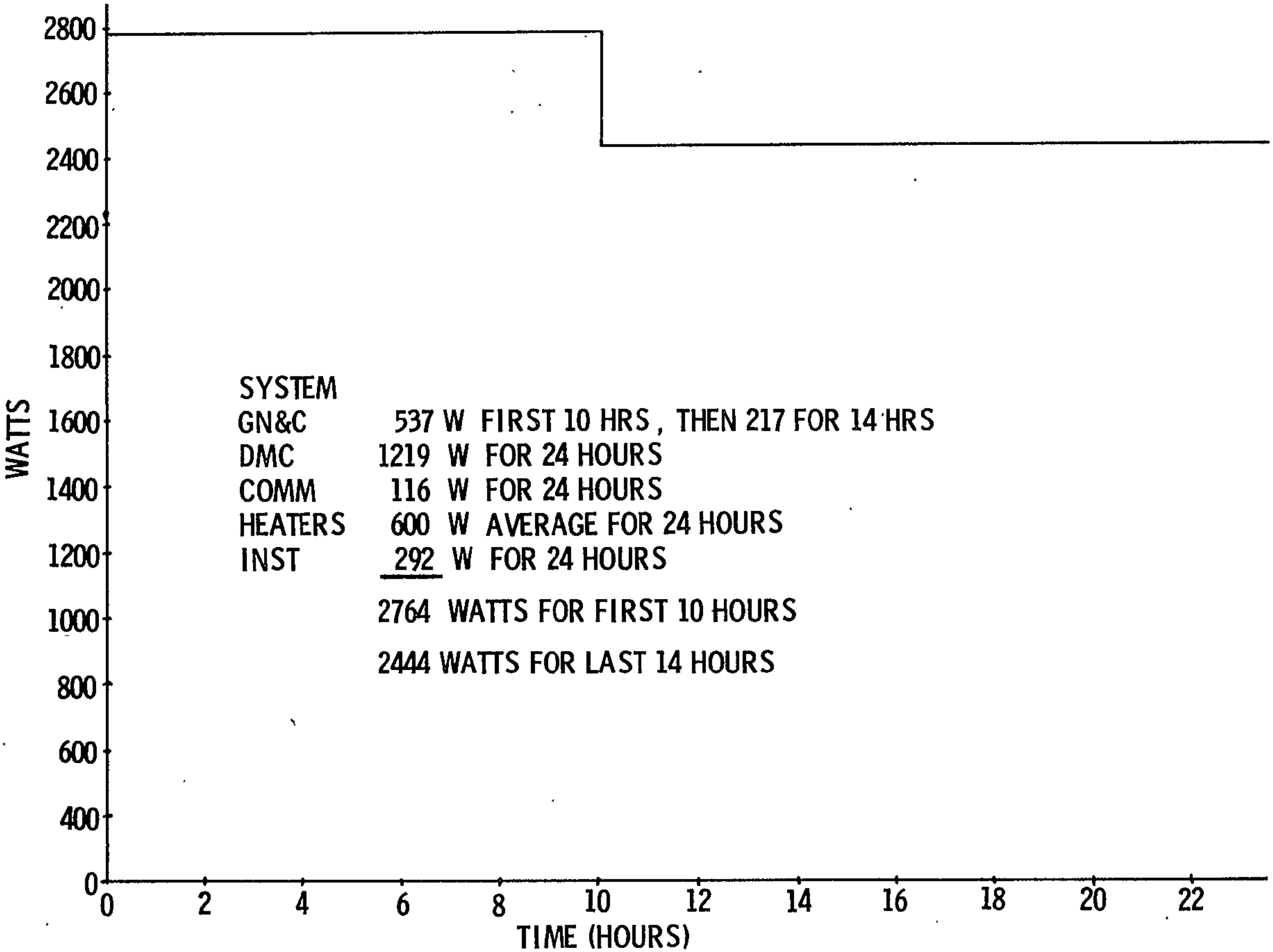


Figure 2-150. ESS Main (+28 VDC) Load Profile (Distribution, Three Buses)

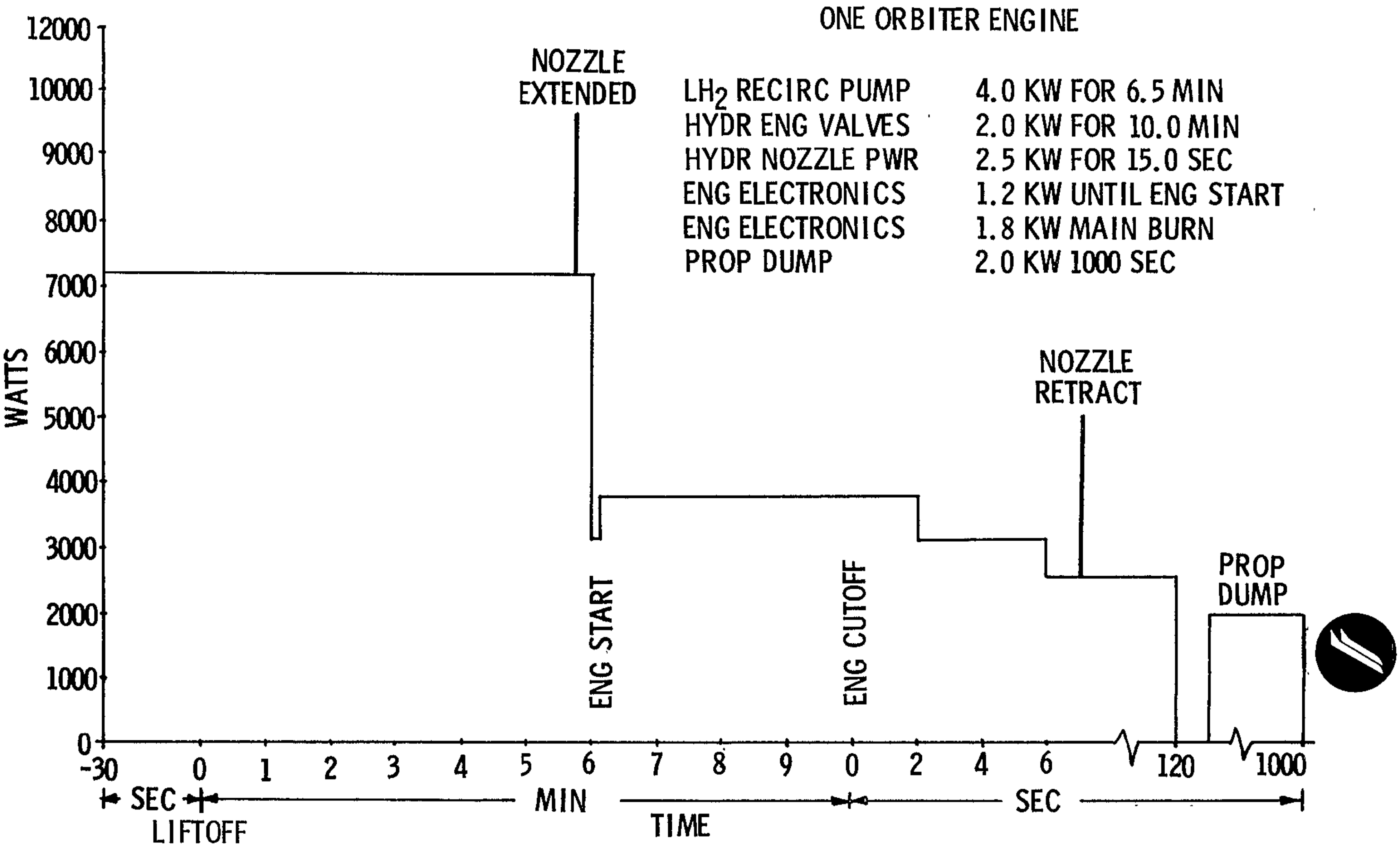


Figure 2-151. ESS AC Load Profile (Main Engine Operation)

Figure 2-152 is a schematic block diagram of the 56-vdc bus distribution system. Figure 2-153 shows a schematic block diagram of the 115/200 vac, 3 phase, 400 Hertz system utilizing the 56 vdc bus input.

The 56-vdc bus distribution system is shown in Figure 2-153. Operation is identical to that of the 28-vdc system previously described.

The 115/400-volt ac system shown in Figure 2-153 utilizes four static inverters. Each inverter is connected to a separate 56-vdc bus.. The output of the inverter is connected through a transformer to provide the 115/200 vac, 3 phase, 400 Hz power. An ac distribution and control box will provide ac power transfer switching and control to the ESS vehicle loads. Figure 2-151 depicts a timeline of the ac load required from transfer to internal power at liftoff through main orbiter engine cutoff. The peak ac load occurs during LH_2 recirculation when the engine nozzles are extended. During a single engine nozzle extension the peak ac load will be approximately 8.5 kw for 15 seconds. Each ac power source is capable of providing 10 kw for peak ac loads. The heavy ac peak loading will be completed at main engine cutoff, which ends approximately 10 minutes from liftoff. Each of the ac buses will be capable of handling the peak ac load during this 10-minute period, including the 15-second period for nozzle extension. An ac bus is capable of handling the ac loads from main engine cutoff for the remainder of the 24-hour mission. Table 2-33 indicates the energy per system used to arrive at the 56 vdc battery size. Table 2-34 shows the energy per system used to size the 28-vdc batteries.

ESS Grounding System. The ESS vehicle grounding system will utilize the structure as a power return for the primary 28 vdc and 56 vdc power sources. The ESS grounding system will be compatible with the shuttle concept. The weight saving in power-return copper for the ESS vehicle structure ground return versus a single-point ground is estimated to be 50 percent or approximately 350 pounds. Secondary power supplies and command/response signals will use a signal return. Signal return wires will be small gauge and lightweight.

Figure 2-154 shows the structure grounding concept for the main 28 vdc power system. The negative side of each battery will be connected to the main structure with a short, heavy gauge strap or wire. Figure 2-155 shows the primary 56-vdc power system with the same structure grounding system as previously explained.

Figure 2-156 shows one of the four ac power systems. The static inverter 56-vdc power input will be referenced to structure ground. The 115/200 vac, 3-phase, 400 Hz, WYE transformer output will utilize a separate neutral wire from the output to the ac loads.

Table 2-33. 56-VDC Gauge Loading

System	Energy (watt-hour)
Main engine (2)	2716
OMS engine (2)	2810
ACPS (14)	17
Actual total	5543
Add 40-percent spare capacity	2217
Actual + spare total	7760
$\frac{\text{Watt-hour}}{\text{Volt}}$ = ampere-hour (AH)	
$\frac{7760}{56} \approx 140$ ampere-hour	
Note: Use two 28-VDC 200-ampere-hour batteries in series per bus because of peak loading during LH_2 recirculation for the main engine.	

Figure 2-157 shows the data bus command/response utilizing a signal return wire for commands and responses. The ACT unit contains an isolated 5-vdc power supply, which provides the 5-vdc driving commands to the solid-state power control switches. The 28-vdc input power to the 5-vol power supply is referenced to structure ground. The 5-vdc output commands are isolated from structure ground and provide a signal return for control and monitoring by the ACT units. The power control switch input command is referenced to the ACT unit 5-vdc signal return while the 28-vdc PCS output power is referenced to structure ground. The solenoid returns will be connected to structure ground. The 28-vdc position indicator switches will be monitored by the ACT unit. An isolation amplifier will be required to change the 28-vdc discrete, which is referenced to structure ground into a 5 vdc signal, which is referenced to the 5-vdc signal return of the ACT unit. The signal return system described previously for Figure 2-157 is the preliminary approach recommended for the ESS vehicle data bus and ACT units to handle commands and responses. Other signal conditioning and measurements will be expteced to use a signal return in interfacing with the ACT units and data bus.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

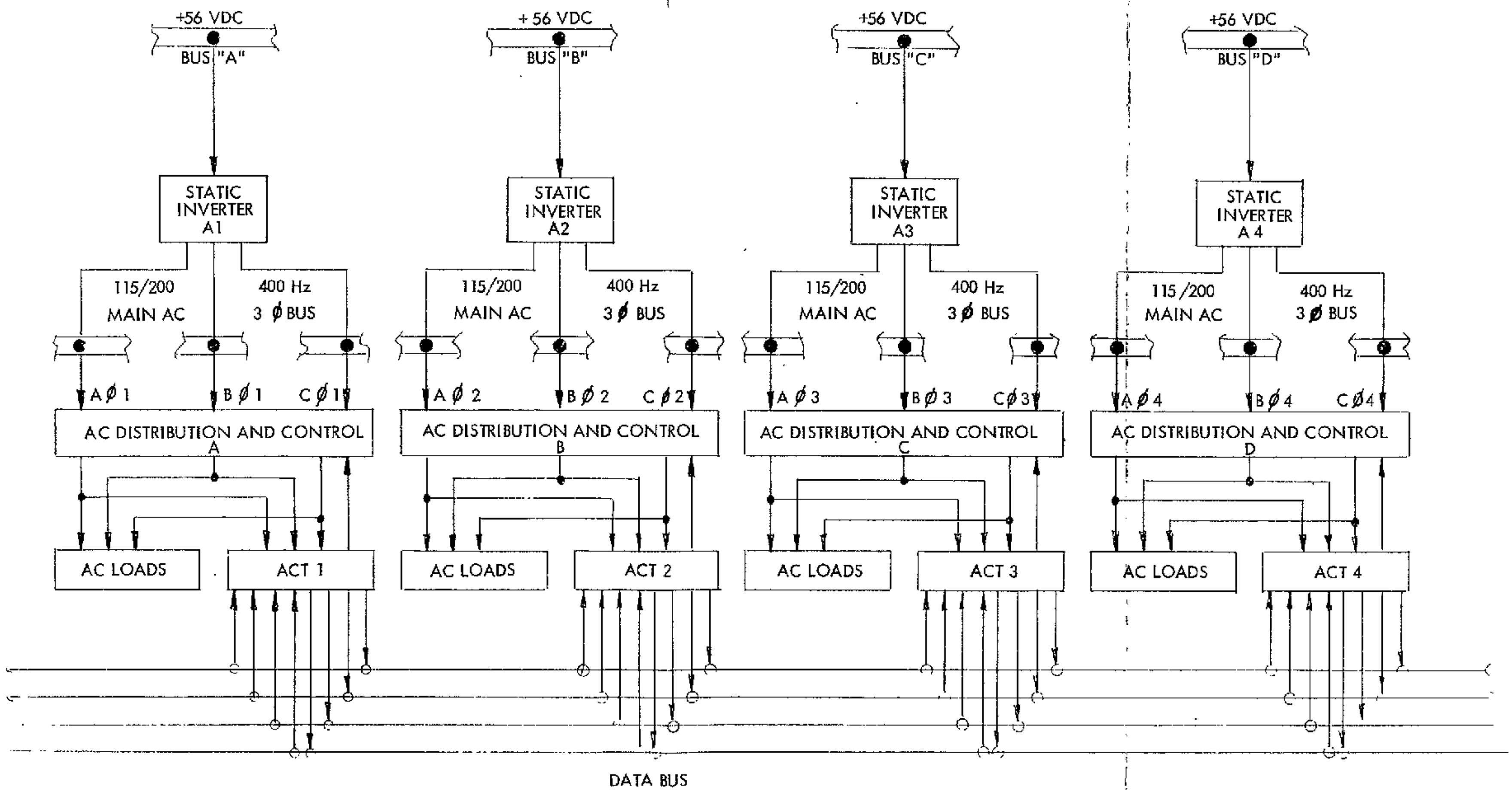


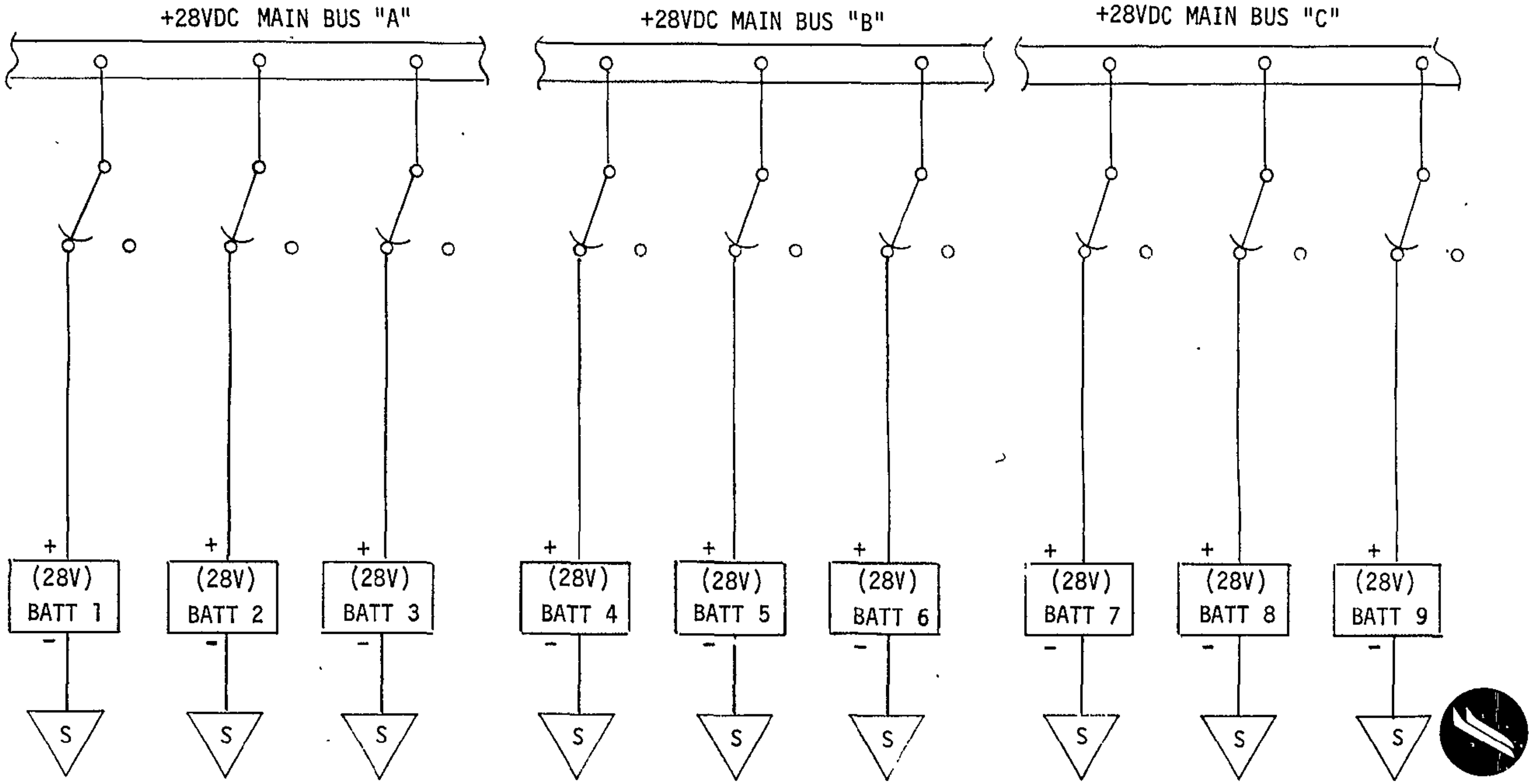
Figure 2-153. 115/200-VAC Bus Distribution System Schematic

Table 2-34. Main 28-VDC Bus Loading

System	Energy (watt-hour)
NG&C	8,318
DCM	29,256
Communications	4,258
Heaters	14,400
Propellant management	7,448
Instrumentation	557
Actual total	64,237
Add 40-percent spare capacity	25,694
	89,931
$\frac{\text{Watt-hour}}{\text{Volt}}$ = ampere hour	
$\frac{90,000}{28}$ $\approx$3210 ampere-hour	

Notes: 1. Use nine 500-ampere-hour, three parallel batteries per bus.
2. Distribute loads evenly between buses.

Redundancy. Three redundant and separate primary main 28-vdc buses and power sources will be used to comply with the ESS vehicle redundancy concept. Three batteries will be connected in parallel to one bus. The three buses normally will be isolated from each other; however, if one complete set of batteries should fail, the capability will be provided to remove the failed battery group from the bus and to connect all three buses together. The other two groups of batteries can adequately handle the load for the remainder of the mission. One battery in each group could fail and the other two in each group will handle the load. The loads will be evenly distributed between the buses.



NOTE: S STRUCTURE GROUND RETURN

Figure 2-154. ESS +28 vdc Main Power Grounding

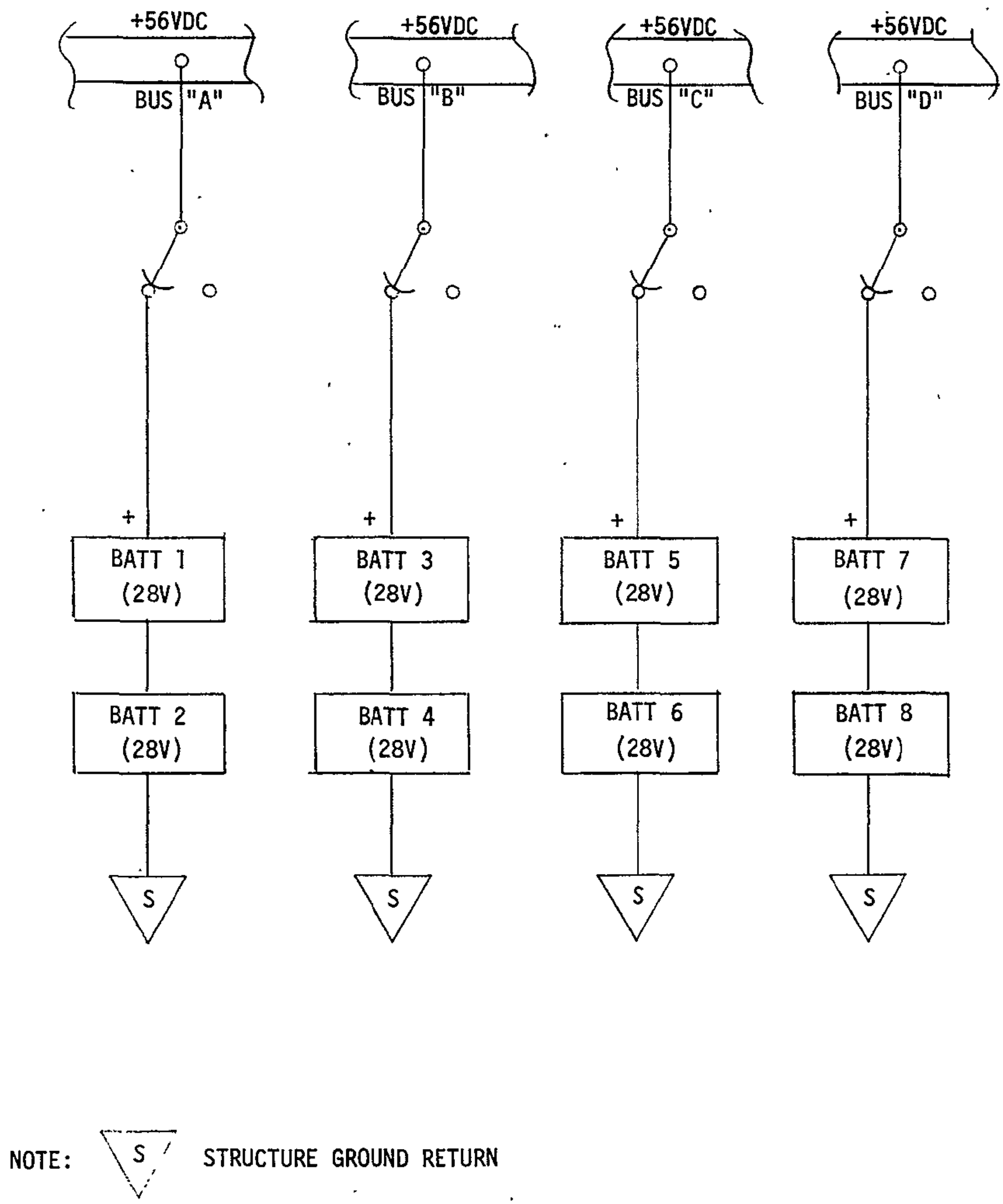


Figure 2-155. ESS +56 VDC Inverter Power Grounding

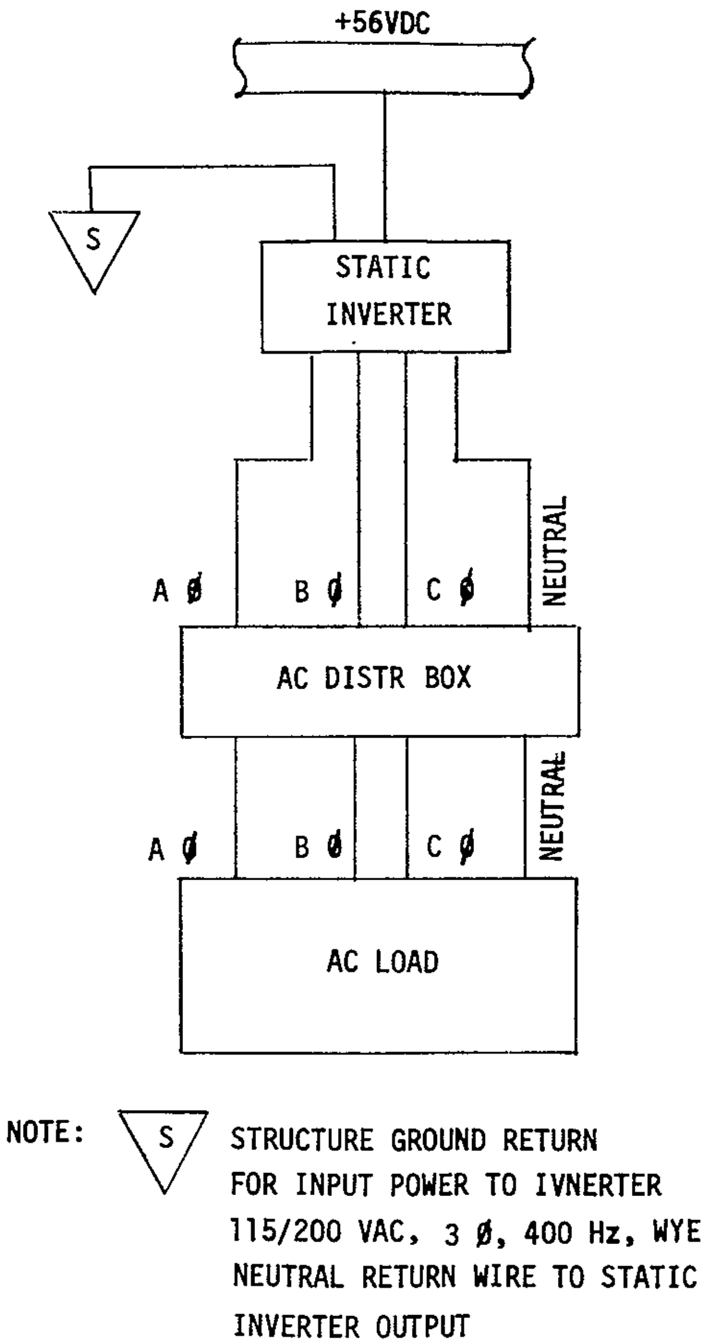


Figure 2-156. ESS AC Power Grounding

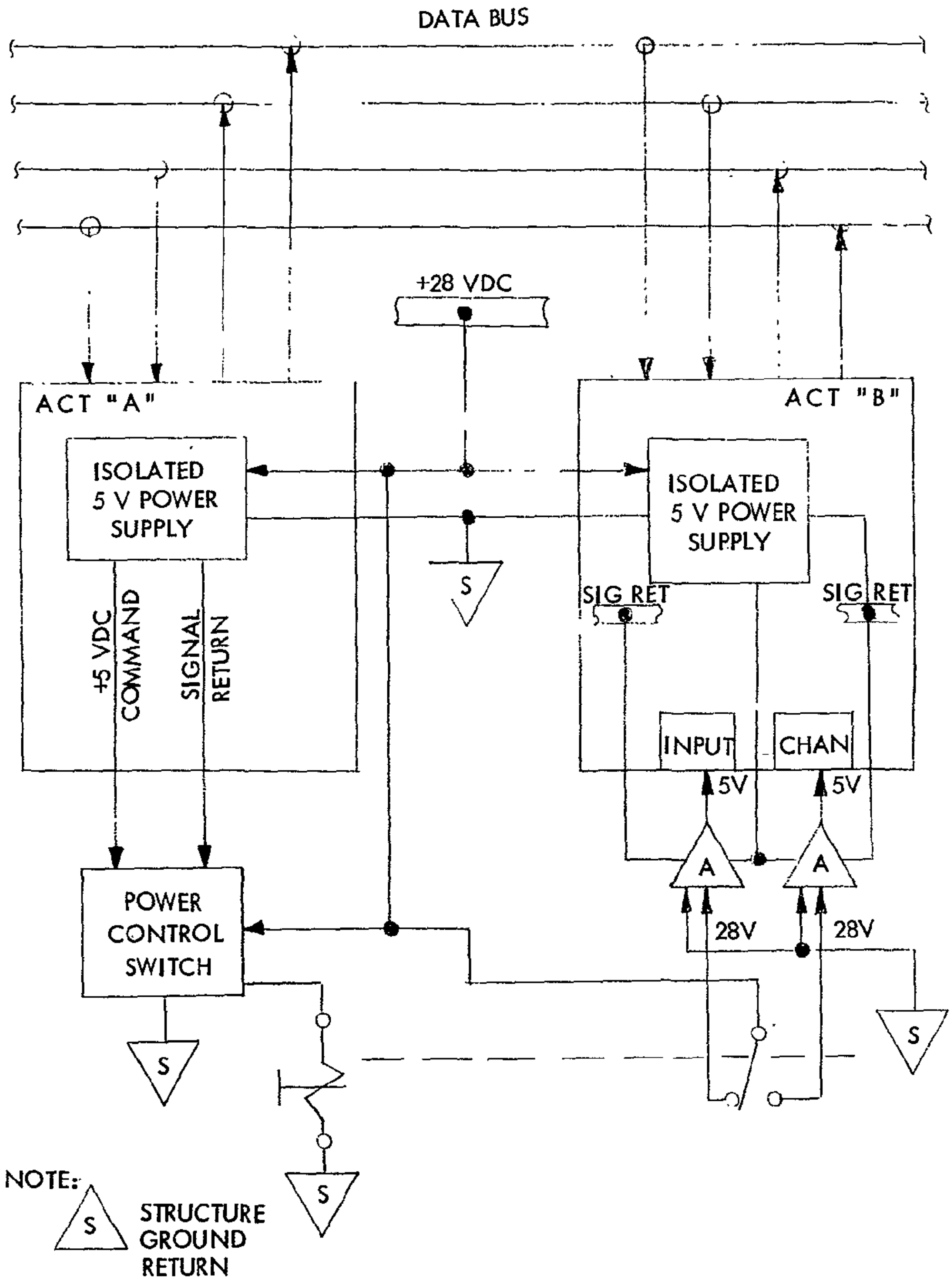


Figure 2-157. Data Bus Command/Response Signal Return

Four redundant and separate 56-vdc buses and power sources will be used to handle the peak ac load during main engine operation and to comply with the ESS redundancy system concept. Each 56-vdc bus will provide input power to a static inverter. The 115/200 vac, 3 phase, 400 Hz power from each inverter will be used to satisfy the ac loads. One orbiter engine contains two separate and isolated ac power inputs. Each 10 kw static inverter is capable of handling the peak ac loads for one orbiter engine. Two static inverters could fail on one orbiter engine and with a one-engine out capability, the ESS vehicle would make mission and fail safe.

ACT units are utilized to provide the computer with a rapid checkout and evaluation of the electrical power system.

2.3.5 Electrical Controls

Electrical control of mechanical and electrical subsystems is the function of the electrical components and circuits required to interface with the data and control management (DCM) system, acquisition, control, and test (ACT) units and the end devices to be controlled or monitored. The control requirements, functional elements, redundancy, and checkout of the following subsystems are described:

1. Main propulsion
2. Auxiliary propulsion

    Orbital maneuvering

    Attitude control propulsion

3. Pressurization
4. Propellant feed
5. Propellant management
6. Propellant dispersion
7. Separation
8. Safing
9. Thrust vector

The mechanical components of these systems will only be discussed to the extent that they determine the operational requirements of the associated control circuits. Refer to the applicable mechanical system description for details pertaining to the mechanical components.

Each of the system design concepts use common electrical components and circuit elements defined for the shuttle orbiter or Saturn S-II hardware of proven reliability. Wherever possible, Saturn S-II designs were used, with modifications to improve reliability and lower costs. The major forcing function for redesign of the electrical controls was the integrated avionics (IAS)/data bus concept that requires onboard computer control of electrical functions via a data-bus system; with ACT units as the interface with the electrical controls. The input-output requirements of these ACT units dictated a change from relay control elements to semiconductor switches, which require considerably less switching power.

The typical power control switch (PCS) used in these designs is a semiconductor device with the following general characteristics:

Terminal arrangement	SPST (normally open)
Rated operating voltage	28 vdc
Rated control voltage	5 vdc
Voltage drop	0.5 vdc maximum
Power dissipation	6.5 watts
Current rating	7.5 amperes
Failsafe current	30 amperes
Trip circuit at 150-percent overload	
Trip indication voltage	5 vdc
Case operating temperature	-54 C to +120 C

These switches must provide internal protection against semiconductor short circuits which would allow an output without a command input, precluding the need for series redundant elements for short-circuit protection.

Each subsystem redundancy determination has been based on components and power sources only. Wire paths are not considered as a source of failure modes. Components with Saturn-Apollo reliability do not require alternate paths. Off-the-shelf or shuttle hardware does require alternate paths to obtain the necessary reliability.

Following are detailed analyses of each of the avionics/nonavionics elements providing the requirements, description, analysis, and checkout of each subsystem interface. Schematic diagrams are provided to aid in understanding of system operation.

Main Propulsion Electrical Control

The ESS main propulsion electrical system is designed to interface the ESS DCM subsystem with the space shuttle engine electrical system and to control application, electrically, of the two main propulsion system engines. It is assumed in this discussion that the proposed Rocketdyne space shuttle engine is used as the basis for the design of the vehicle MPEC's. (Refer to ICD 13M15000B.)

Requirements.

1. The ESS engine EC system will be FO/FS.

2. The ESS engine control system will use the data bus concept. The data and control management subsystem data bus and ACT units will be utilized to provide commands, timing, and sequencing to the SSE.

3. The vehicle engine control system will be designed to allow checkout of the system with an automatic computer-controlled program.

4. The vehicle engine electrical control system will control redundant 115/200-volt, 400-Hz, 3-phase power to each engine. There is no engine requirement for dc power.

5. The vehicle can be placed in a safe condition with one engine out.

6. The ACT units, power transfer switch, and power control switches (PCS) will be identical to those used for similar applications on the shuttle program.

System Description. The Rocketdyne shuttle orbiter engine is controlled by an electronic controller that utilizes redundant digital computers. Communication with the vehicle is via the engine data bus. (See Figure 2-158.)

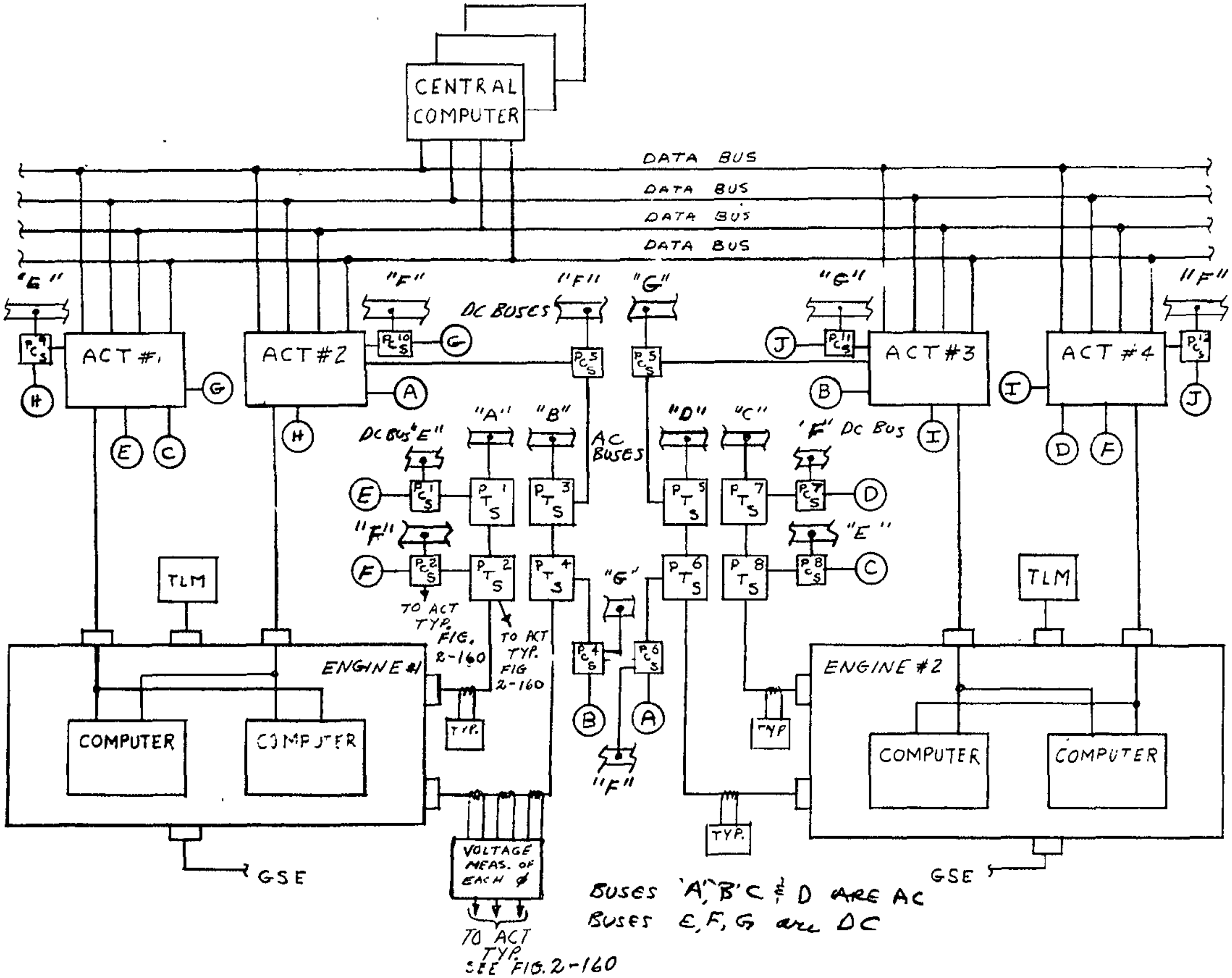


Figure 2-158. Main Propulsion Electrical Controls

The engine accepts commands from the vehicle and transmits data to the vehicle as controlled by the ACT units.

Power for all components in the engine is controlled by the vehicle engine EC system (Figures 2-158 and 2-159). The vehicle engine EC system controls the redundant power sources to the engine electronics package. The engine control and monitor system is fail-operational upon loss of either power source. If both power sources fail, the engine has the capability to be shut down at once or up to 40 seconds later. The time to shutdown must be determined and set into the engines on the ground.

The engine electrical interface consists of six connectors. Two connectors are used for commands and vehicle-requested engine status; two connectors are used for power input; one connector is used for instrumentation; and one connector is used for ground measurements (Figure 2-158).

The vehicle engine control system will use two ACT units per engine. One ACT unit will give commands and receive data through one engine command connector, and one ACT unit will give commands and receive data through the other engine command connector. The two-engine-mounted, digital computers interface with each command connector and have the capability to control the engine from either interfacing ACT unit (Figure 2-158). Each ACT unit is powered by a subbus of two batteries, see Figure 2-159. This enables the loss of two of the three 28-volt main buses and still retain capability to control the engine.

Engine operation can be divided into six phases as follows:

1. Ground checkout. The automatic mode sequentially checks controller, sensors, actuators, valves, and spark igniters. Individual checkout of components is possible.

2. Start preparation. Controls purge operations and verifies engine readiness prior to start.

3. Start. Controls and monitors the engine during the start sequence.

4. Main stage. Controls and monitors engine performance, limit control, valve positions, and maintenance data.

5. Shutdown. Controls and monitors the engine during the shut-down sequence.

6. Postshutdown. Controls and monitors the engine during propellant dumping operations.

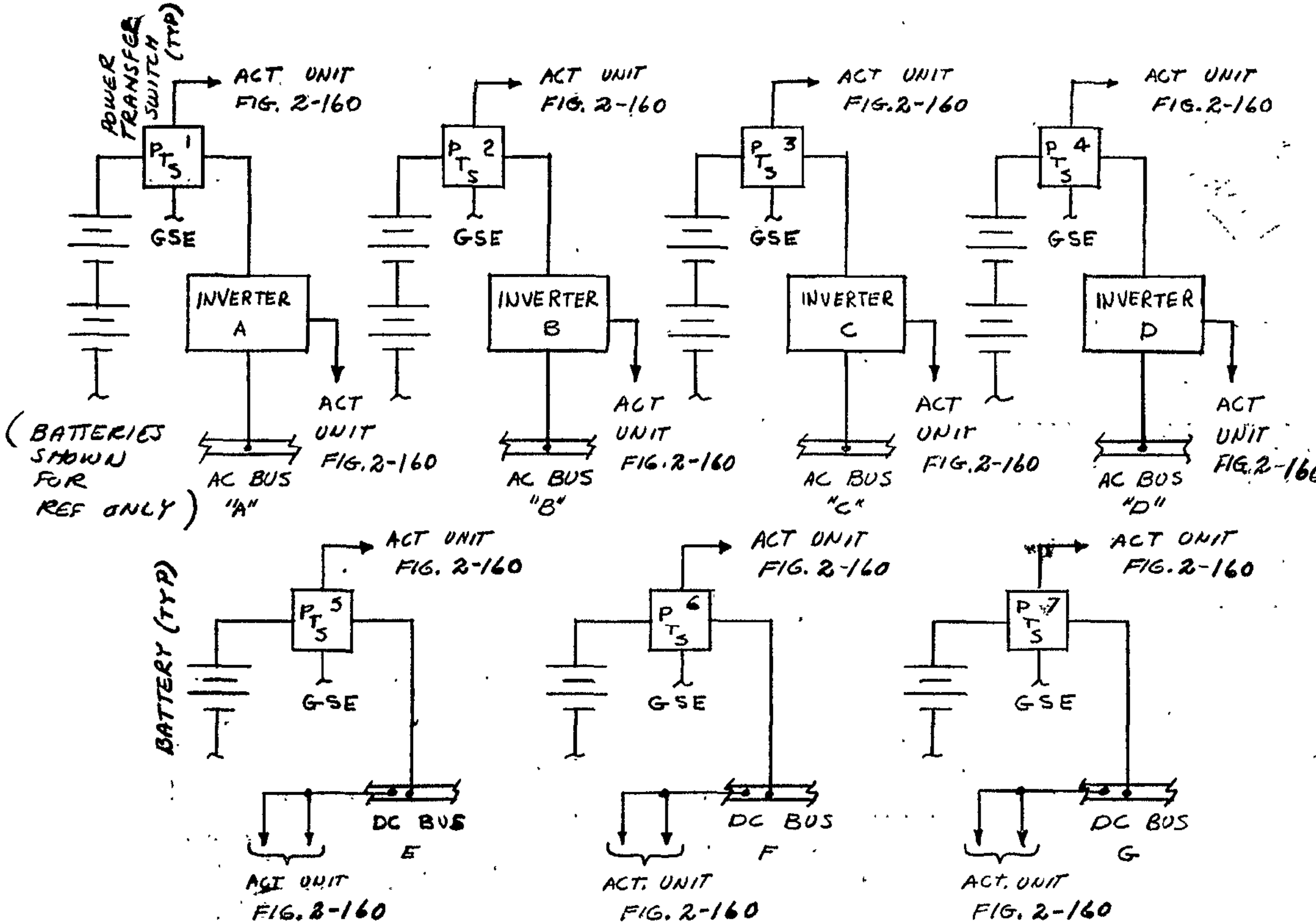


Figure 2-159. Main Propulsion Electrical Control, Power

The commands for control of the engine are listed in Figure 2-159. The engine operation phase in which each command is normally required and the phase in which the command is accepted are indicated in the Figure 2-159. The engine will not accept commands during engine operation phases in which engine integrity could be violated, or damage to equipment or injury to personnel could occur.

The vehicle engine control system will control power from two parallel inverters and supply it to the engine on two separate power connectors (Figures 2-158 and 2-159). Both inverters may be turned on at the same time. The engine should contain provisions to prevent tieing the power together internally. Each inverter bus will be switched by two power transfer switches in series. Each power transfer switch is controlled by a power control switch, which in turn is controlled by the ACT unit, see Figure 2-158. Power will be applied to the engines on the ground and removed from the engines after engine cutoff. The system is so designed that any combinations of two main dc buses or ac buses, two ACT units or transfer switches, or engine computers could fail and at least one engine would remain operational.

The vehicle engine control system makes use of the capability of ACT units to completely checkout redundant paths and isolate failures to a single component with the onboard central computers. Figure 2-160 indicates the measurements necessary for checkout.

The engine has the capability to check itself and if a malfunction is detected during the test sequence, the type of malfunction and the failed line replaceable unit (LRU) are identified via the data bus.

<u>Trade Studies and Performance Analysis</u>. The decision to use the data bus concept versus discrete command wiring was made at the beginning of the Phase B Study. See paragraph 2.3 of this section for the ground rules under which the trade study was conducted.

The redundancy level required in each portion of the system was determined by the FO/FS ground rule. During the proposed design, the following three basic ground rules were applied in an attempt to ensure proper performance during flight and ground checkout:

1. Use the properly sized components to provide for adequate performance.

2. Add necessary redundancy to meet the FO/FS performance requirements.

3. Add necessary checkout capability to verify all redundancy and isolate failures to the component level.

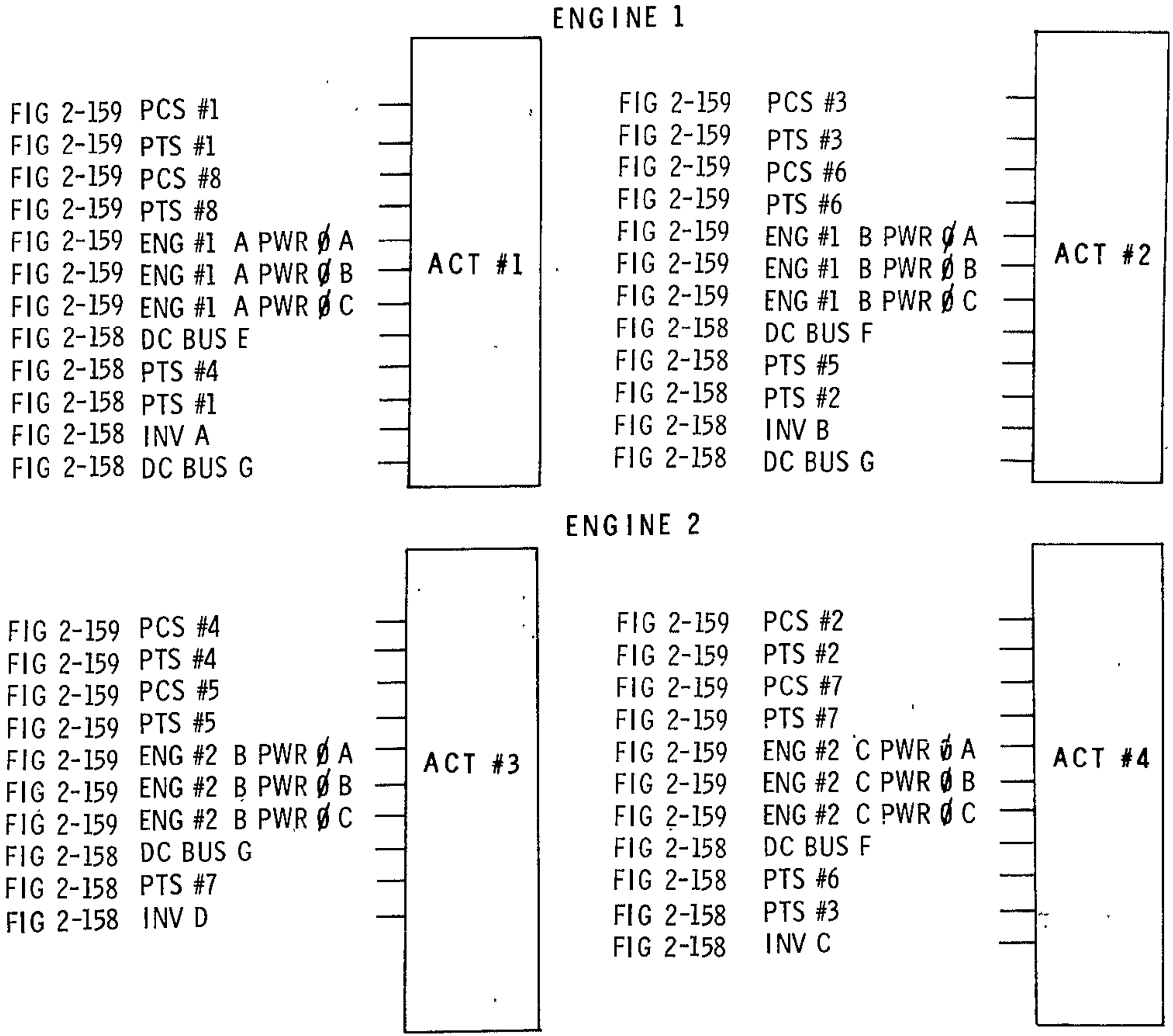


Figure 2-160. Engine Control System Redundancy Checkout Measurements

Redundancy Management. Redundancy management techniques for the vehicle engine control system consist of the following steps to ensure the system meeting the FO/FS system requirements:

1. Design in redundant paths with adequate checkout capability to check out each path and isolate failure of the replaceable black box.

2. Check out all combinations of the redundant paths during ground checkout.

3. Apply the necessary commands to activate all the paths prior to flight or ground testing.

4. Monitor each path for its proper signature in flight or ground testing and turn off any failed component.

Auxiliary Propulsion Electrical Controls

The ESS auxiliary propulsion electrical controls (APEC) provides the dual function of orbital maneuvering system (OMSEC) and attitude control propulsion system (ACPSEC) electrical controls. In this orbital maneuvering modes, the system utilizes two LO_2/LH_2 10,000-pound thrust engines to provide the velocity changes necessary for orbit circularization, orbit transfer, rendezvous, and deorbit. For attitude control, the system utilizes fourteen GO_2/GH_2 2100-pound thrust engines. The propellant feed, pressurization, and propellant management subsystems are common for the total APS system.

Requirements.

1. Control circuits for each APEC function will be FO/FS, and the FO/FS design concepts will be based on total engine systems requirements with consideration for each critical mission phase.

2. ACPS and OMS engine sequencing will be accomplished by the DCM system.

3. APEC circuits will utilize control hardware being developed for the shuttle orbiter, including ACT units, power control switches, and engine control drivers.

4. ACPS minimum impulse firing time shall be approximately 100 milliseconds.

5. The APEC subsystem shall be designed for checkout by an automatic computer program.

System Description. The APEC's are composed of pressurization, propellant feed, ACPS, and OMS subsystems. The pressurization and propellant feed system is composed of the functional elements required to manage the liquid and gaseous propellants required for ACPS or OMS engine operation. Refer to Figure 2-54 for a mechanical component block diagram of the systems and Figure 2-161 for the typical electrical control block diagram.

Each APS liquid propellant tank has two independent, electrically actuated, fill valves in series; each valve requires three alternate electrical control paths in order to achieve FO/FS actuation. Although these valves are only actuated during ground operations, they are critical if ground emergency conditions require draining propellants from the tanks.

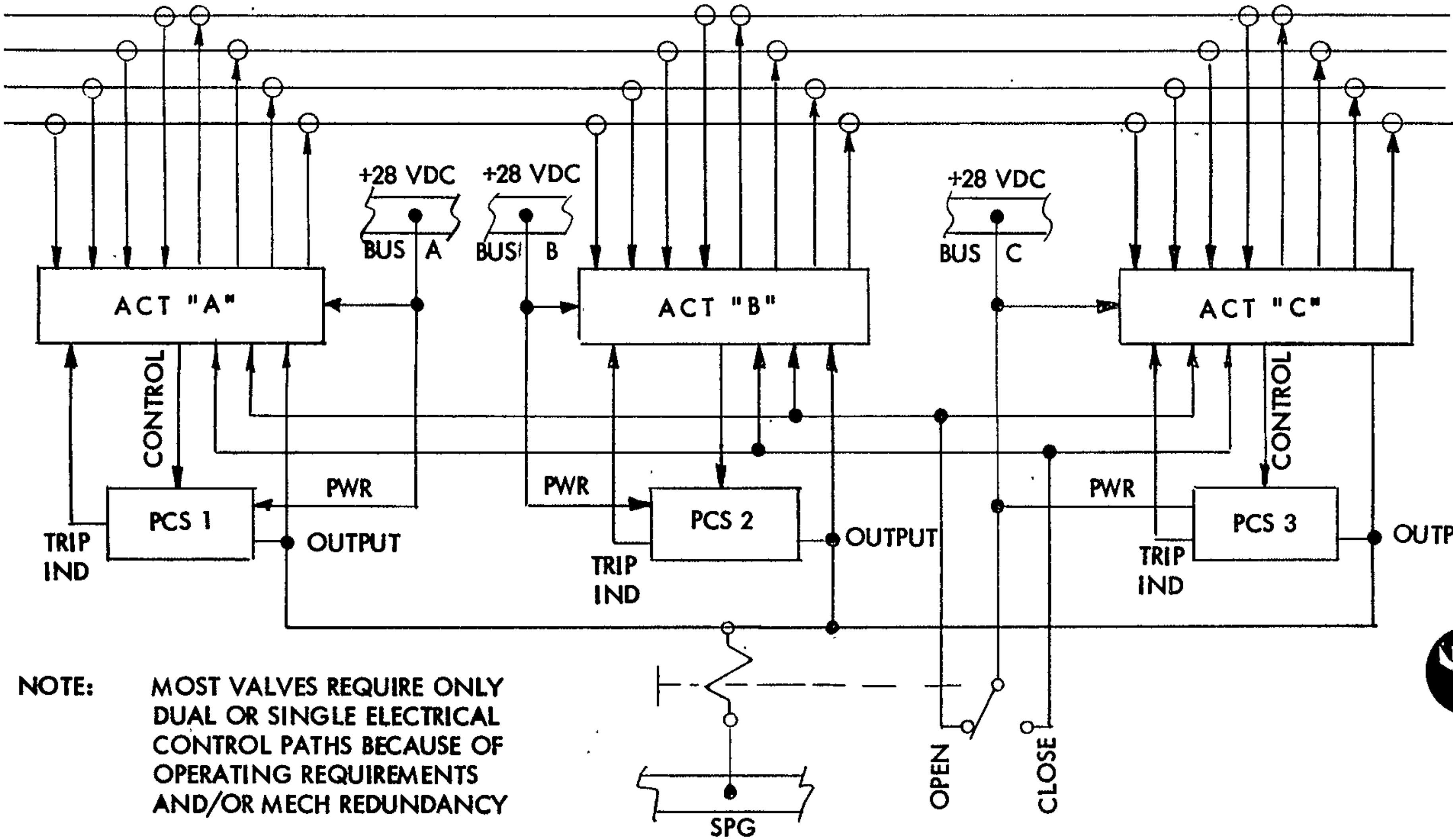
Like the fill valves, there are two valves in series, for control of APS ground pressurization of each gaseous propellant system (GH_2 and GO_2 for the ACPS). These valves are not critical because the gases may be vented. The control circuits are, therefore, only two alternate paths since the second failure would be a safe condition.

The LO_2 and LH_2 APS prevalves are also two valves, in series, in each propellant feedline and considering that a one OMS engine-out condition is safe, the valve control circuits need two alternate paths.

Hot gases from the gas generators drive the turbopumps and then exit through two propulsive or nonpropulsive vents. A pneumatic, normally closed pilot operated valve for each system selects either vent mode. These pilot valves are critical for APS operation and therefore require two alternate paths for solenoid controls.

The gas generator isolation valves shut off the supply of gaseous propellants to the engine gas generator supply manifolds. Two alternate paths for control of each of these eight valves are sufficient to achieve FO/FS actuation. This is based on the ability of one OMS engine to complete a mission, which means the loss of one OMS engine is a safe condition.

Each liquid propellant tank has one controllable vent relief valve with an isolation valve in series with it and one noncontrollable relief valve and burst-disc combination. The controllable vent relief valves are similar to the ones used on the Saturn S-II propellant tanks. They have electrical control to open the vent and control to override the normal relief pressure setting. The solenoid that opens the vent is a normally open valve so loss of control power would open the vent when pneumatic pressure is applied; therefore, two control circuits for this valve yield FO/FS actuation. The override solenoid is normally closed so loss of control power to this valve



NOTE: MOST VALVES REQUIRE ONLY DUAL OR SINGLE ELECTRICAL CONTROL PATHS BECAUSE OF OPERATING REQUIREMENTS AND/OR MECH REDUNDANCY

Figure 2-161. ESS Auxiliary Propulsion Electrical Controls
(Triple Redundant Valve Control)

is also a safe condition. Also, this override solenoid has no operational function; therefore, only one control path is required. The vent isolation valve located in series with the relief valve requires two control circuits to meet the FO/FS requirement associated with the relief valve installation.

The gaseous propellant accumulator relief systems differ from that of the liquid propellant tanks in that the relief valves are not controllable and there are three parallel relief valves - one has two series isolation valves; one has one series isolation valve, and one has a series burst disc. Since the systems are mechanically FO/FS for relief capability, the isolation valves must protect against a failed-open relief valve. Two control paths for each of these isolation valves satisfy the FO/FS criteria for this failure mode.

Pressure switches and pressure transducers are required for monitoring and control purposes. All critical measurements are FO/FS, but those used only for data are single circuits; the same is true for temperature, flow, and speed measurements.

Preparation of the APS system for operations will start with pressurizing the accumulators through the ground pressurization valves and purging the LO_2 and LH_2 tanks. The LO_2 and LH_2 vent isolation and vent relief valves are open and the fill valves are partially opened until chilldown of the tanks is accomplished; the fill valves are then opened full open for fastfill. With the propellant tanks loaded (refer to Section 2.3.5 under Propellant Management Electrical Control Subsystem for propellant loading and level monitoring electronics), the vents are closed and the tanks are pressurized through the pressurization shutoff valves and regulators from the stored gaseous propellants in the accumulators. The prevalves, gas generator isolation valves, and the ACPS manifold isolation valves are opened and the OMS engine system is preconditioned. The APS system is then ready to operate when commanded by the DCM system. OMS engine main propellant valves and gas generators are controlled by driver units similar to the ACPS thruster driver shown in Figure 2-162. The three-way OMS or ACPS mode select valves and hot gas vent select valves are individually controlled as are those turbopump unloading (bypass) valves. Each of these circuits for OMS control consists of two alternate paths - for each function - to achieve FO/FS operation; separate power sources are utilized for each alternate path of each function. OMS engine start is accomplished by positioning the three-way select valves to propulsive vent and OMS engine operating position, starting the gas generators to drive the turbopumps, initiating OMS ignitors, opening the main propellant valves, and closing the turbopump unloading valves.

The ESS ACPS will be used intermittently throughout the ESS mission from separation through deorbit. Figure 2-163 shows the electrical control schematic block diagram for one group of seven thrusters. Figure 2-164

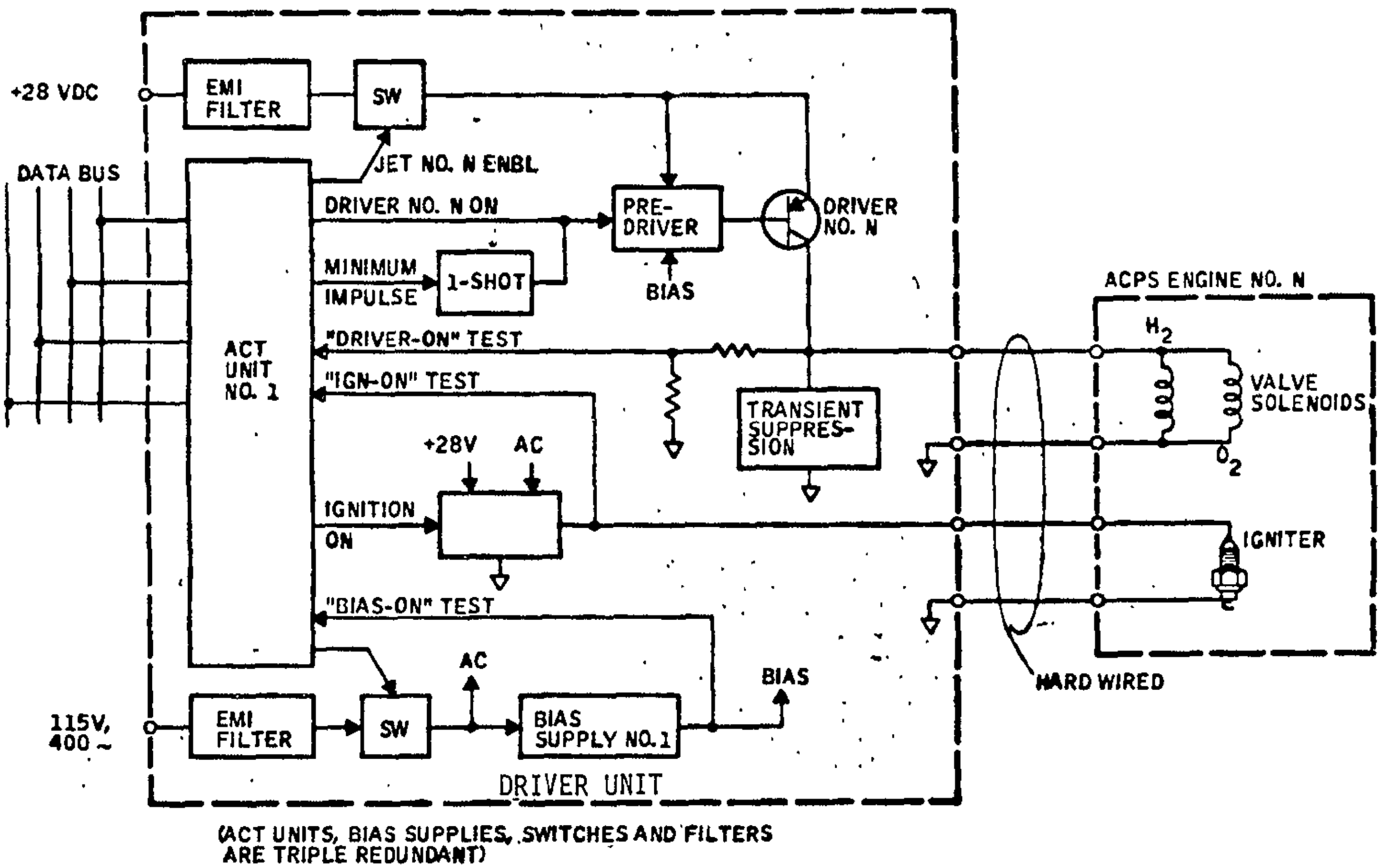


Figure 2-162. Attitude Control Propulsion System Electrical Control Driver Unit Mechanization

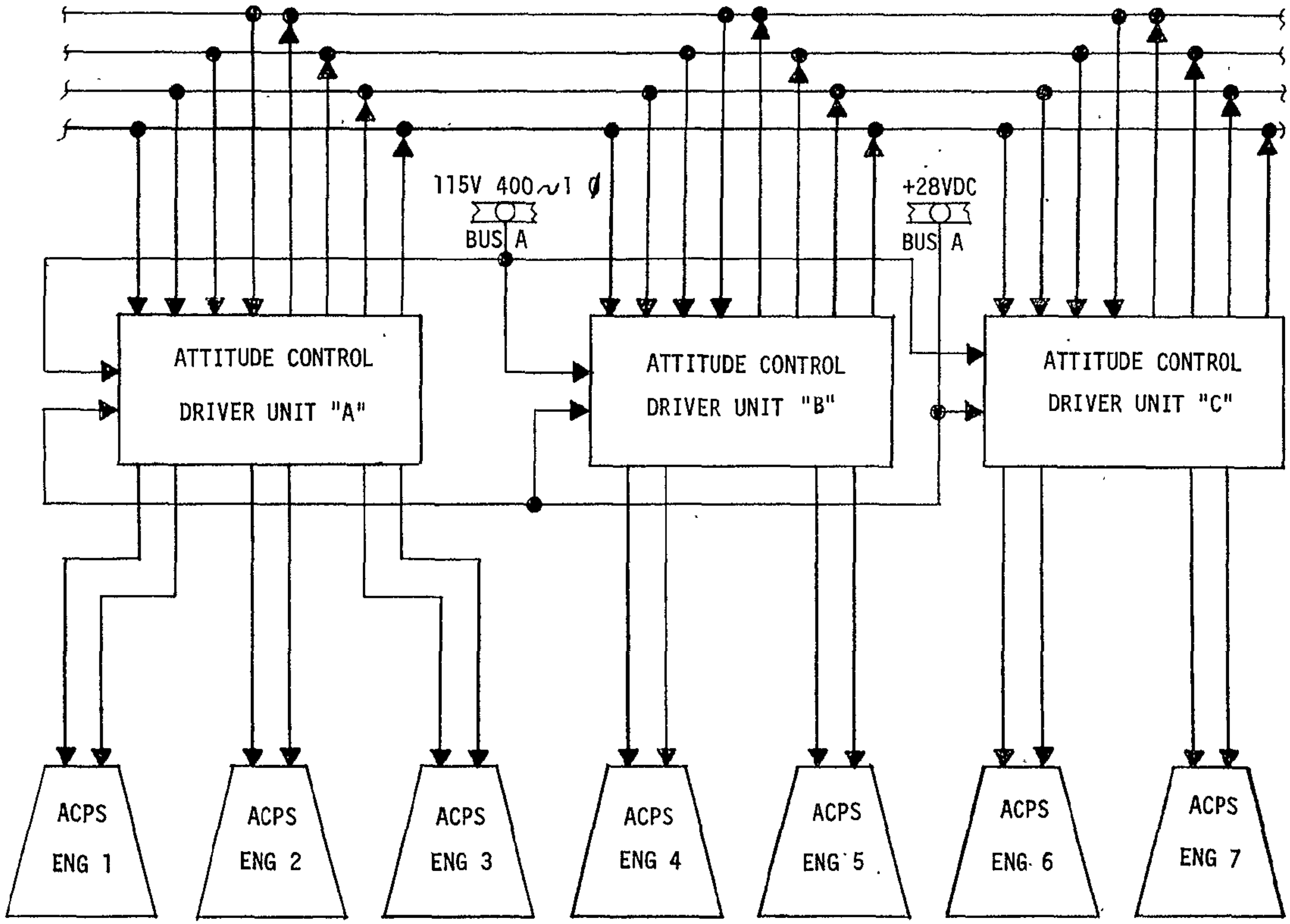


Figure 2-163. Auxiliary Propulsion System Attitude Control Propulsion System Electrical Controls

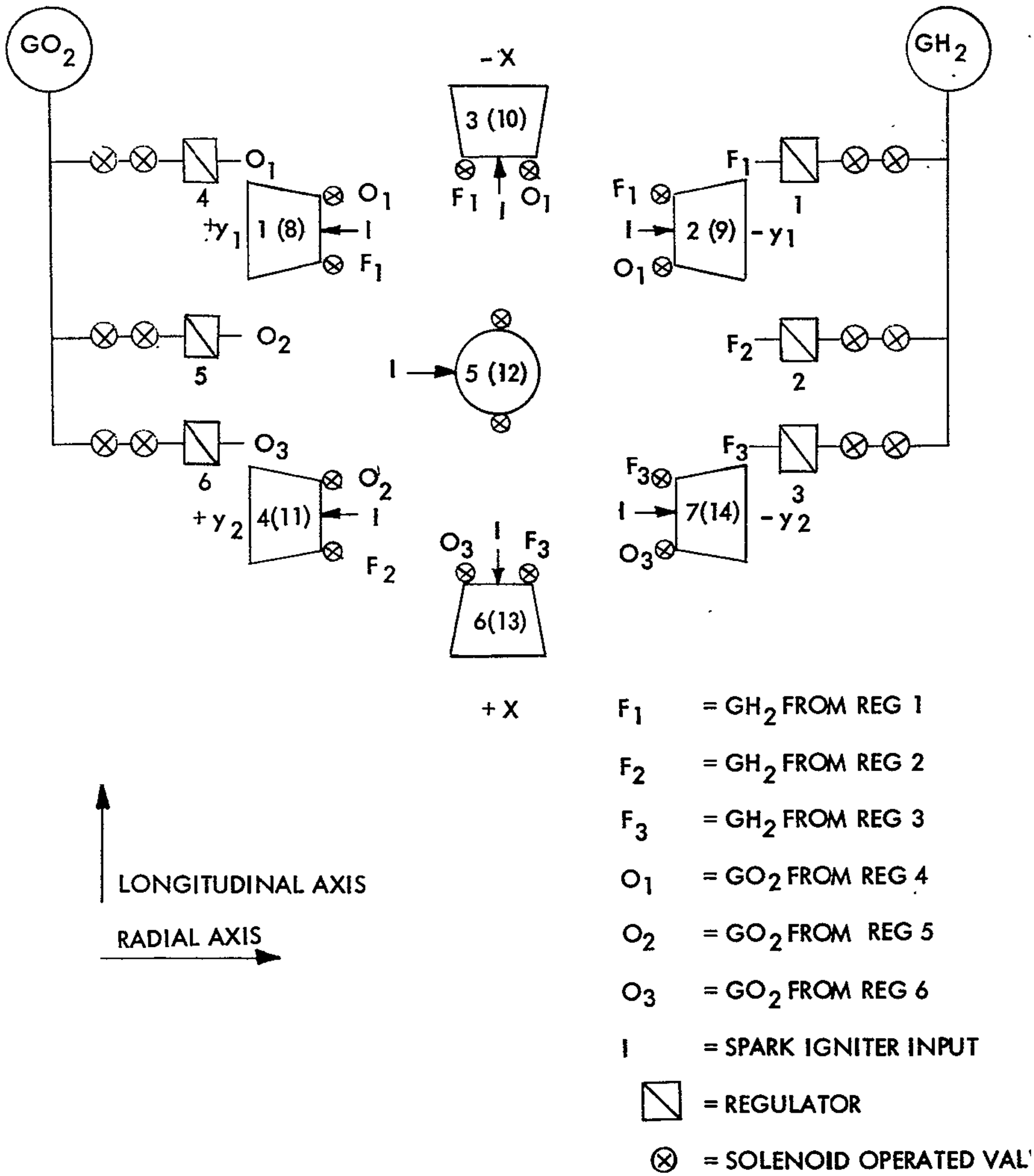


Figure 2-164. ESS Auxiliary Propulsion System (Attitude Control)

indicates the physical layout and manifolding of thrusters in a group. Thruster Engines 1, 2, and 3 are connected to the same fuel and oxidizer manifold. Engines 4 and 5 are connected together and likewise Engines 6 and 7. Figure 2-162 is a schematic block diagram of the attitude control driver unit that provides the control and sequencing to the thruster engine. Each thruster has one GO_2 and one GH_2 28-vdc solenoid-operated valve. Redundant spark exciters are included to assure ignition.

The LO_2 and LH_2 prevalves are normally open and four gas generators are operating for an ACPS mode. The ACPS EC operates with a computer address via redundant data buses to an attitude control driver ACT unit. Previously, before liftoff, standby ac and dc power has been applied to each driver unit. GO_2 and GH_2 have been manifolded to each thruster valve via two isolation valves, in series, and each thruster is ready to fire when selected by the computer via the data bus and driver unit. Each attitude control driver unit contains triple redundant circuitry. Operation of one thruster engine begins with an address to the driver ACT unit. The ACT unit output channel will provide the input signal to the predriver amplifier which will switch the solid-state driver output to 28 vdc. The driver will energize the thruster GO_2 and GH_2 solenoids causing the valves to open. A 25-millisecond signal is provided to the engine spark exciters for ignition. The thruster will continue to burn until the solenoid is deenergized by computer removal of the 28 vdc from the driver. This will cause the normally closed GO_2 and GH_2 valves to close, terminating burn time for a thruster. A minimum-impulse command can be issued by the computer. In this case, the ACT unit output fires a one-shot multivibrator located in the attitude control driver unit that provides the predriver with an input signal which, in turn; causes the driver to open the thruster's GO_2 and GH_2 valves for the length of time the one-shot is on. The ESS minimum-impulse firing will be approximately 100 milliseconds. The computer command to fire a thruster longer than a minimum-impulse duration will require the computer to issue a start command and later to issue a stop command. An assumption is made that the 25 millisecond spark ignition time will not require the computer to issue a turn-off command.

The attitude control driver unit electronics contain signal conditioner buffer amplifiers used to condition and normalize ac, analog, and low-level secondary signals to a level acceptable by the ACT unit. The thruster engine will contain pressure transducers used to detect and monitor pressures and pressure differentials to evaluate thruster performance.

An enable signal to each engine attitude control driver unit provides ac and dc power to the driver electronics for each thruster addressed. Subsequent discrete thrust commands provide either direct input to the predriver, or minimum-impulse input via a one-shot circuit. The predriver circuit contains input gating and power gain to the PNP driver transistors, which in turn switches the current to the GH_2 and GO_2 engine valve solenoids. Separate hardwire connections are provided to each thruster valve solenoid. Transient suppression circuitry will be provided to reduce to a manageable level the inductive kick generated by a coil current turnoff. A separate ignition signal feeds a second 28-vdc driver circuit to provide a low-power discrete signal to the ignition exciter, which is part of the engine system. Interface with the data bus is provided by ACT units contained in the attitude driver units.

Four basic discrete signals are used to control each thruster engine: (1) channel enable, (2) valve command-normal, (3) valve command-minimum impulse, and (4) ignition on command. In addition, each ACT unit accommodates four discrete signals from each engine control channel to verify the following four functions: (1) channel bias voltage, (2) coil driver on, (3) ignition signal on, and (4) engine on.

Multiple isolated dc and ac power lines are required for redundancy management and fault tolerance. Each ac line serves a separate bias supply, and each dc line is provided with an EMI suppression filter.

Redundancy.

1. Power. Three redundant 28-vdc and 115-vac, 1-phase, 400-Hz sources will be utilized for the attitude control driver units to control and power the ACPS engines as indicated in Figure 2-163.

2. Component. Triple redundant attitude control driver units will be used to meet the fail operational/fail safe concept for the OMS/ACPS engines.

3. Failure mode analysis. Triple redundant attitude control driver units including integral ACT units will be used to assure OMS/ACPS engine firing capability. In addition, redundant data buses and computers are available to address the driver units. Monitoring of the OMS/ACPS engines will be via the attitude driver units, which will provide triple redundancy for engine performance evaluation or fault isolation.

4. Management requirements. Separate and triple redundant power and driver units are available for each OMS/ACPS engine, which will facilitate rapid computer control for determining response to an individual OMS or ACPS engine. This arrangement will allow rapid fault isolation in the case of no response from a commanded engine. Commanding and monitoring other OMS/ACPS control driver units will provide fault isolation to a completely failed ACT unit, driver electronics, or the OMS or ACPS engine. Three

failures in the same driver electronic circuitry in three separate black boxes will be necessary to cause an OMS/ACPS engine failure due to the driver circuitry.

5. Backup modes. Completely separate and triple redundant power sources and driver electronics provide adequate backup modes inherent in the design of the OMS/ACPS engine control.

Propellant Feed Electrical Controls

The propellant feed electrical controls (PFEC) are designed for the following electromechanical devices.

1. Propellant tank fill and drain valves

2. Engine feed prevalves

3. Recirculation valves

The purpose of the PFEC is to control transfer propellants to the stage main propellant tanks; from the tanks to the engines or drain the tanks; and temperature condition the feedlines and engine components by recirculation of propellants.

The transfer of propellants to or from the stage is accomplished through pneumatically actuated, solenoid-controlled butterfly valves (fill and drain valves). These valves are normally closed and are actuated during propellant transfer operations. Pneumatic pressure for actuating these valves is supplied from a ground source only.

Propellants are transferred to the engines through the engine feed system normally open, pneumatically actuated, solenoid-controlled prevalves. Should loss of pneumatic pressure or electrical control power occur, the prevalves are spring-actuated to remain in the open position. There are four prevalves, one for each propellant feedline to the main engines. Pneumatic actuation pressure for these valves is supplied by the valve actuation helium receiver.

Recirculation of propellants through the engines and back to the tanks temperature conditions the system to obtain the required engine inlet conditions. The LO_2 recirculation is accomplished by bubbling ambient GHe through the liquid in the feed system to augment natural convection. LH_2 is recirculated by an engine-provided, electrically driven pump, pumping the liquid through the system. Each system (LO_2 and LH_2) has a propellant return line back to its respective tank and a normally open, pneumatically actuated, solenoid-controlled recirculation valve in each return line.

Each LO_2 feedline has an accumulator for pogo oscillation suppression. These accumulators are supplied with helium pressure from the GHe injection receiver to damp LO_2 oscillations in the feedlines during engine operation. Each accumulator also has a bleed valve connecting it with the recirculation return line. These valves are normally closed, pneumatically actuated, and solenoid-controlled. During recirculation they are opened.

Requirements. While mated to the booster, all electrical control circuits must be capable of sustaining two component failures and still be safe. If the controlled function fails to a safe condition when control power is lost, only two control paths are required. If the failure mode of the controlled function is not fail-safe when power is lost, three independent control paths are required. When a function becomes unsafe if control power is applied at the wrong time, each independent path must be protected against inadvertent power application.

Measurements considered critical to the operation of the vehicle and safety of the crew must also be redundant to obtain FO/FS operation.

Wherever possible, solenoid and pneumatic valves will have a position indicator for performance monitoring and failure isolation.

System Description. Each component of the system has individual operating characteristics and failure mode effects that determine the circuits required to obtain the FO/FS goals. Each one is considered and described as a separate system.

Semiconductor power switches are utilized in each of the control circuits in order to interface with the ACT units. This is the same approach and hardware used on the shuttle orbiter.

Transducers and signal conditioning are shown as system requirements. Refer to the instrumentation section for hardware descriptions.

1. The LO_2 and LH_2 fill and drain valves are normally closed pneumatic valves, actuated both open and closed by two-way solenoid valves whose normally open ports actuate the valves closed. FO/FS for the valve control circuits is applicable only on the ground because pneumatic actuation pressure will not be available after liftoff. Three data bus control circuits are required to obtain FO/FS operation of the control circuits. Refer to Figure 2-165 for the circuit block diagram.

2. The LO_2 and LH_2 prevalves are normally open, pneumatically actuated valves located in each engine propellant feedline. Each valve is pneumatically actuated both closed and open by an

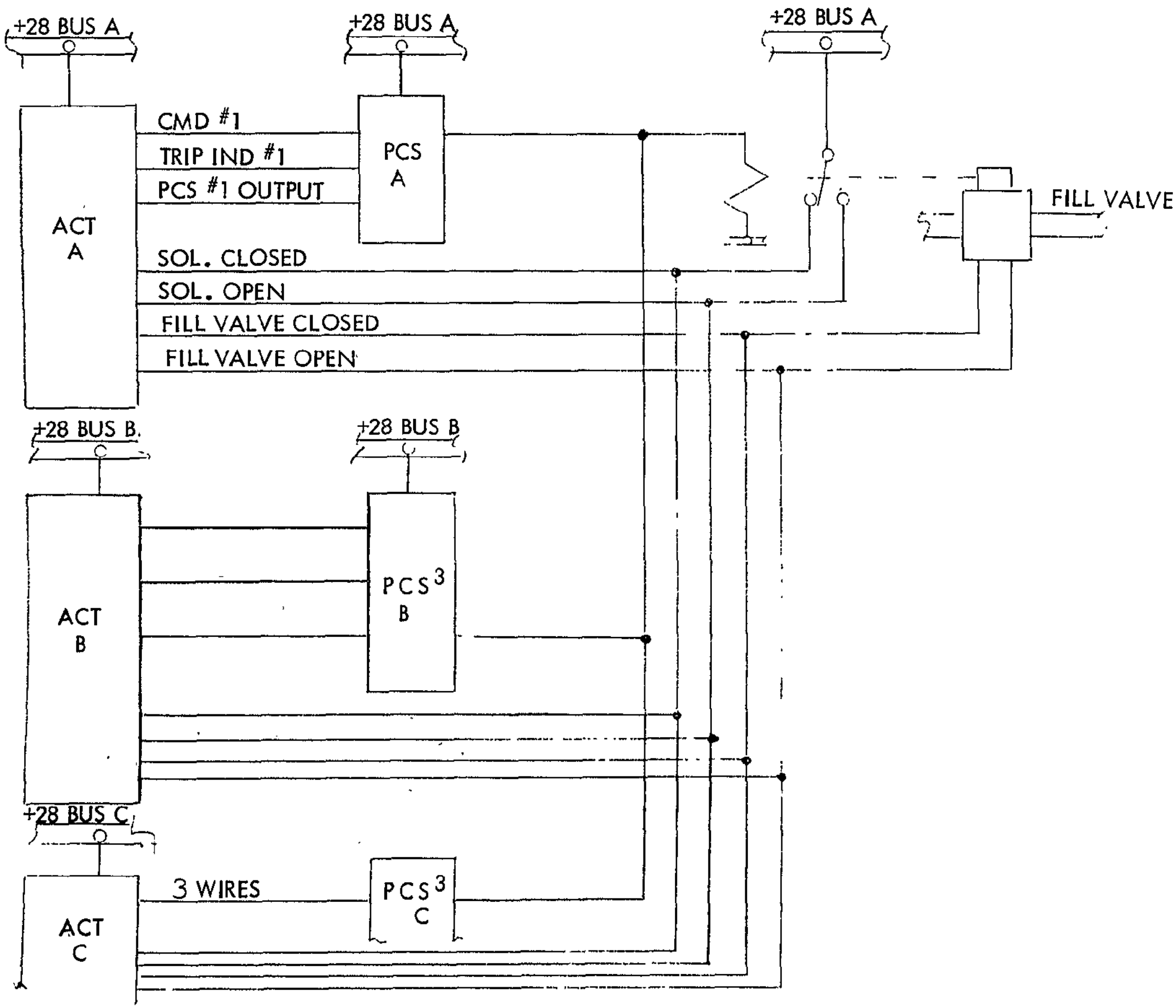


Figure 2-165. Typical FO/FS Fill Valve Control Block Diagram

electrically controlled two-way solenoid valve. When the solenoid is energized, the prevalve is closed and when the solenoid is deenergized, the prevalve is pneumatically actuated to the normally open position.

A control circuit failure that will not allow the prevalves to be closed is not considered a critical failure; therefore, only two control circuits are required. A failure that would close a prevalve prematurely would be critical if prevalves on two engines were affected. Dual circuits for each prevalve gives FO/FS operation. Refer to Figure 2-166 for the circuit block diagram.

3. Control circuits for the recirculation return line valves (recirculation valves) must be FO/FS for failures that would energize the solenoids to close the valves but only FS for other failures. This is because an open valve is a safe condition but a closed valve could result in engine failure. A single control path satisfies the requirement. Refer to Figure 2-167 for the circuit block diagram.

4. The LO_2 helium injection solenoid is normally open and must be energized closed. A failure that would energize the valve closed could cause engine failure but a valve failed open would be a safe condition. A single control path gives FO/FS operation for this valve. Refer to Figure 2-167 for the circuit block diagram.

5. The pogo accumulator fill control solenoids supply GHe to the helium injection system when deenergized and to the LO_2 feedline accumulator when energized; therefore, the system must be FO/FS in both the opening and closing modes. Since there are two solenoid valves in parallel, the control circuit for each valve must have two independent paths to obtain FO/FS operation of the combination system. Refer to Figure 2-168 for the typical block diagram.

6. The pogo accumulator bleed valve fails both operational and safe in the open position but a failed-close condition could affect engine start. The control circuit for this valve is a single path. Refer to Figure 2-169 for the circuit block diagram.

7. The instrumentation associated with the propellant feed system is primarily for performance evaluation but will also be used for fault isolation during checkout operations. The pressure and temperature transducers are also to be used as backup measurements for the position indicators on the pneumatic valves and solenoids. The only critical measurements which must be

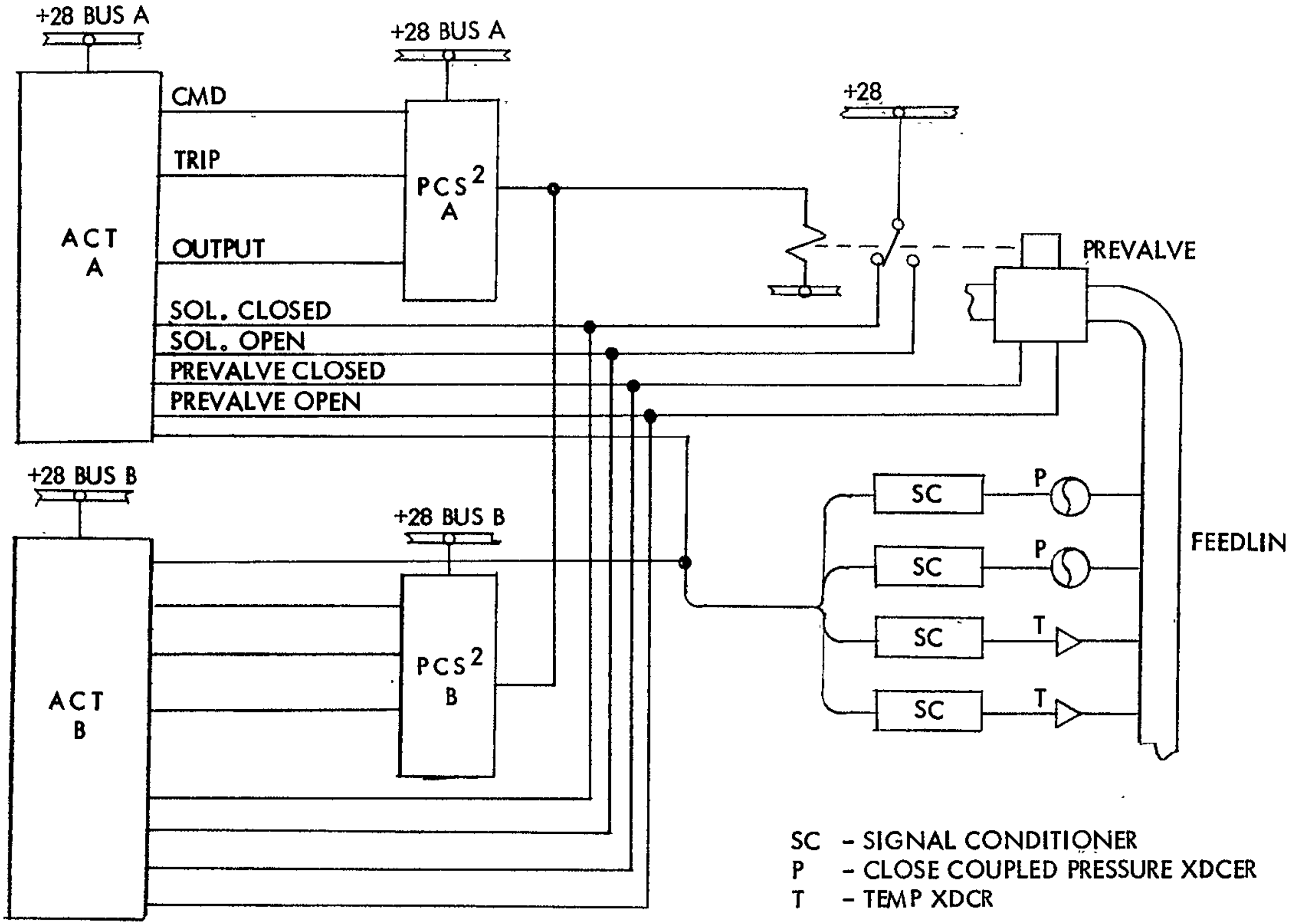


Figure 2-166. Typical Prevalve Control Block Diagram

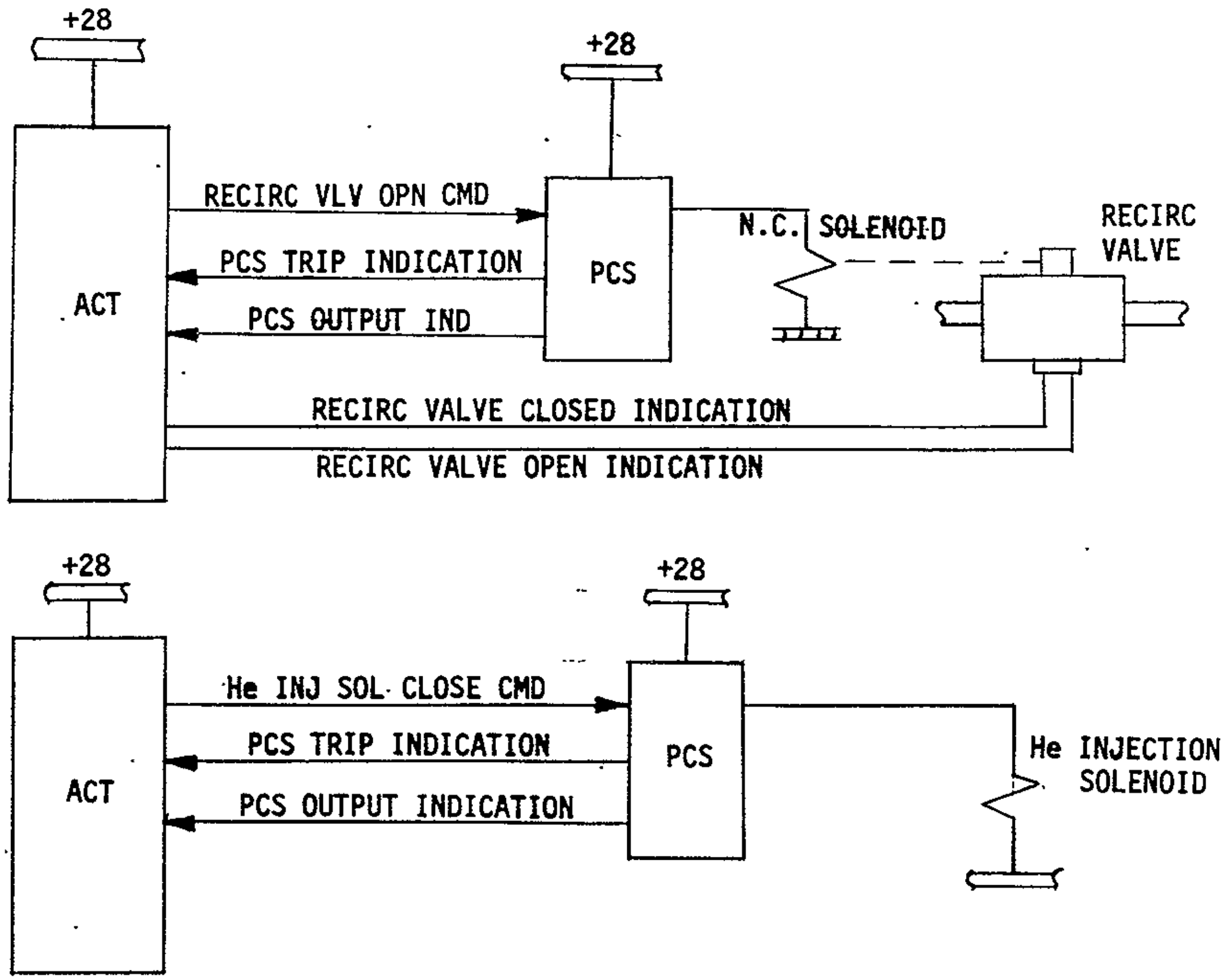


Figure 2-167. Typical Recirculation Valve and Helium Injection Solenoid Control Block Diagram

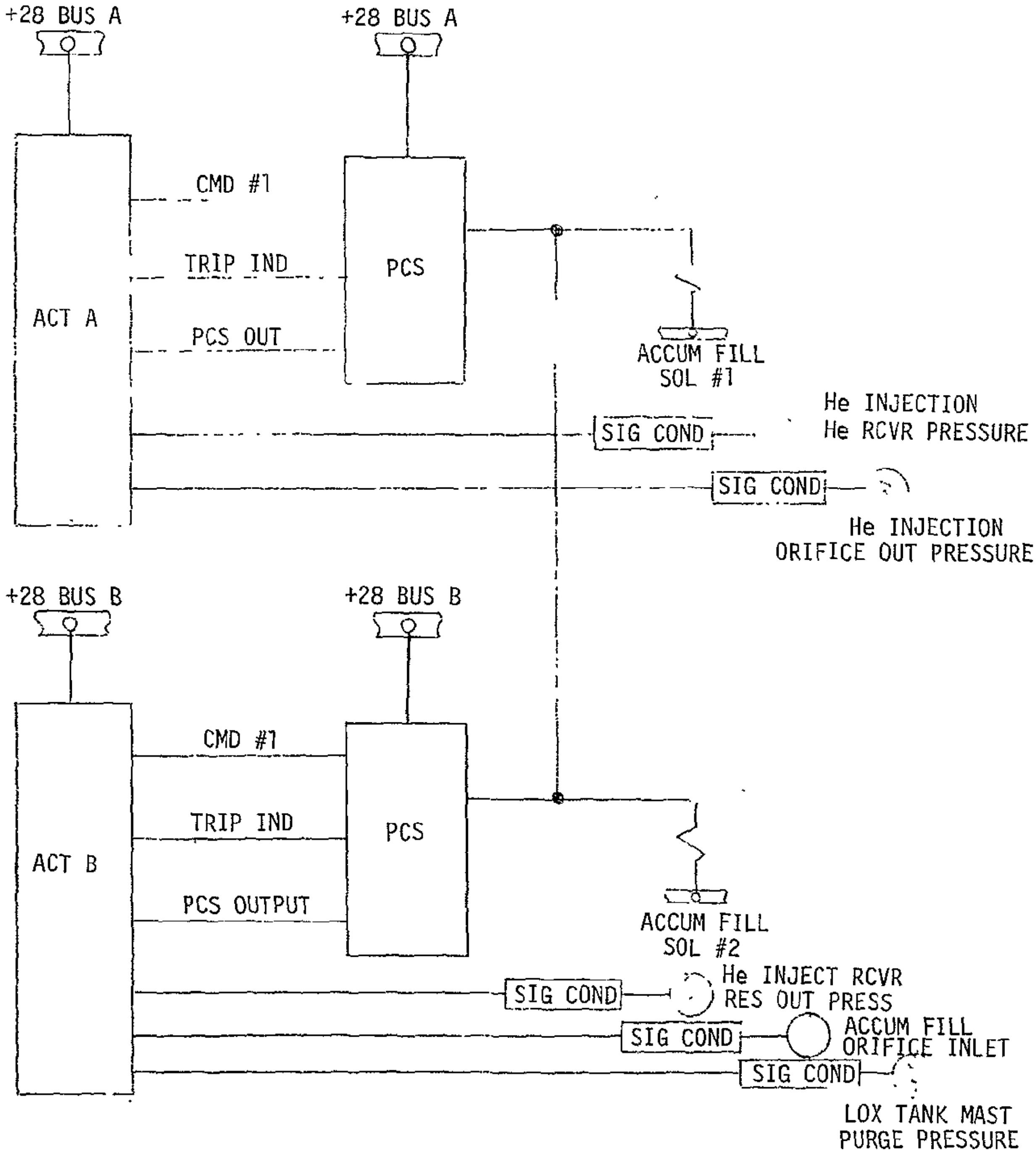


Figure 2-168. Pogo Accumulator Fill Valve Control Block Diagram

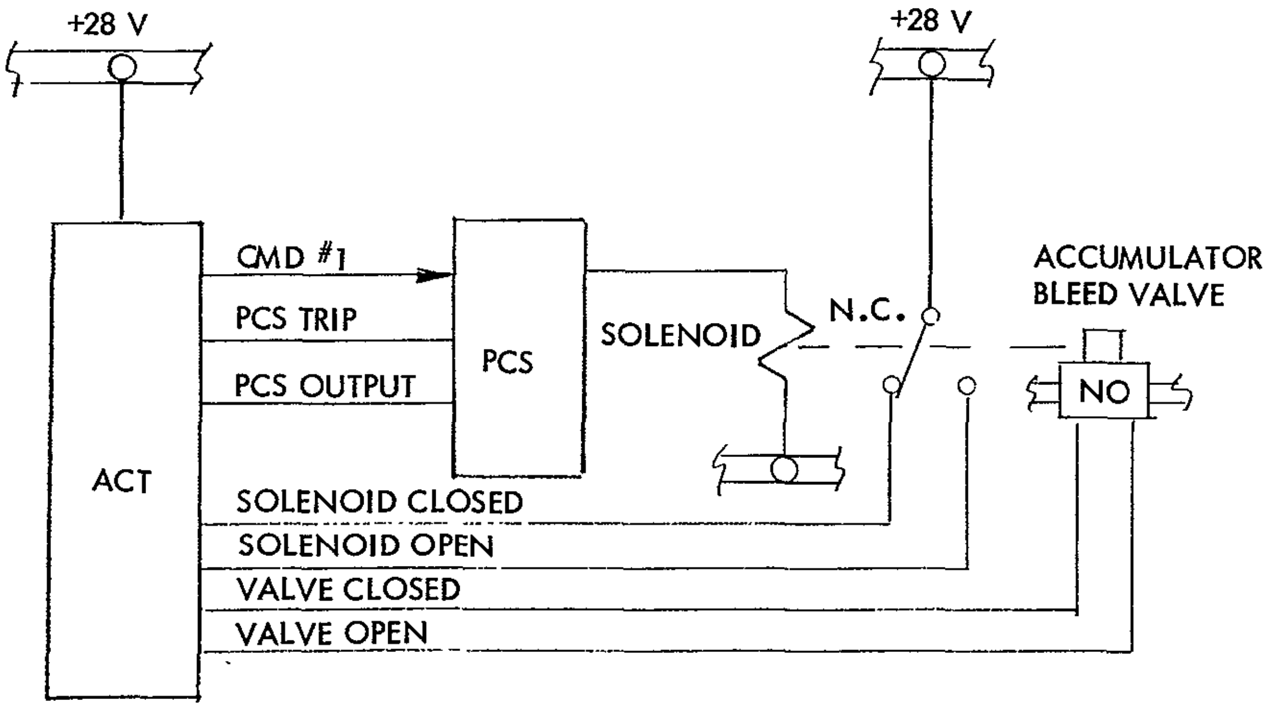


Figure 2-169. Pogo Accumulator Bleed Valve Control Block Diagram

redundant are the engine inlet pressures and temperatures. Other measurements associated with the system are redundant to each other and/or they are noncritical.

The signal conditioning equipment for each measurement must have a 0 to 5-volt-dc output compatible with the ACT unit interface requirement.

Performance Analysis. The redundancy level required in each portion of the system is determined by the operating requirements of each function. The failure modes and criticality of each mode differs with each component being controlled. The redundancy required to obtain the required FO/FS operation of each control circuit is discussed in the previous section. Failure modes and their resulting effect on system operation are also discussed.

Management requirements, based on the required subsystem operation, are determined by the available alternate paths and status monitoring capability. During operations other than checkout, a status check and profile comparison will be required repetitively, at fixed intervals. A discrepancy will require immediate corrective action to isolate the failed component and select an alternate mode.

1. A propellant tank fill valve has three alternate paths for operation. One path will be used as the primary mode until a failure occurs. While this path is energized, the computer will compare the status indicators with the required profile.

    a. Fill valve open indication ON

    b. Fill valve solenoid open indication ON

    c. Power control switch 3 output ON

    d. Fill valve closed indication OFF

    e. Fill valve solenoid closed indication OFF

    f. Trip indicators 1, 2, and 3 OFF

    A comparison failure of any one of the indicators alone will only be reverified for a predetermined number of samples and then telemetered as a nonfunctional indicator failure.

    A comparison failure of two or more indicators for a predetermined number of samples will require that the computer select an alternate path and isolate the failed path. The failure information will be telemetered as performance data. Other control circuits will be handled in a similar manner. Checkout of these circuits requires actuation and verification of each redundant path.

Pressurization Electrical Controls

The pressurization electrical controls (PEC) is divided into four categories:

1. LO_2 and LH_2 propellant tank vents

2. LO_2 and LH_2 propellant tank pressure switches

3. LO_2 and LH_2 tank GHe pressurization solenoids

4. Valve actuation GHe receiver vent

The purpose of the system is to assure that a net positive suction pressure (NPSP) is provided for the engine turbo pumps, that a positive pressure is maintained in the propellant tanks, and that gaseous helium pressure is available for pneumatic valve actuation.

During ground checkout and standby operations, the stage pressurization solenoids, in conjunction with the propellant tank pressure switches and the DCM system, control pressurization of the tanks to one-half flight (people-safe) pressure. Before liftoff, ground sources pressurize each tank to flight pressure. After engine start, pressurant gases are supplied from the main engines.

Requirements. The LO_2 and LH_2 main propellant tank vent valves are controllable mechanical relief valves that vent at predetermined pressures. They must have electrical control circuits to actuate the valves open pneumatically, to override the normal relief pressure setting, and the LH_2 vents must have an electrically controlled low-pressure vent mode solenoid. FO/FS control circuits are required for each of these solenoids. After liftoff, electrical control is required for the LH_2 vent valve low-pressure vent mode solenoids only.

Each propellant tank has three pressure-actuated switches which are used for control of tank pressurization during ground operations. One low-pressure and two flight-pressure switches monitor tank pressures. They must be used to control pressurization to safe ambient pressure and flight pressures. The normally closed tank pre-pressurization solenoids must be interlocked with these pressure switches by the computer software program.

The valve actuation GHe receiver supplies pneumatic pressure for actuation of pneumatic valves which must operate in flight. The vent valve on this receiver is for ground operations only. It is a normally closed solenoid valve that requires only fail operational control capability due to its noncritical function.

System Description. Each component of the system has individual characteristics and failure mode effects that determine the electrical control circuits required to obtain the astrionics reliability required. The control switching is accomplished by semiconductor power-control switches which interface with the DCM system ACT units and the mechanical system solenoid valves. These are the same PCS's proposed for use on the shuttle orbiter.

All instrumentation equipment required for performance evaluation, checkout, and fault isolation in this system is noted and included in the measurement list. A description of the hardware required to implement these measurements may be found in Section 4.3.3.

Main LO_2 and LH_2 Propellant Tank Vent Valves. The vent relief valves on the main propellant tanks (two on each tank) are automatic pressure relief valves with pneumatic controls to open, select the reference pressure, and override the relief pressure setting during ground operations. Each vent valve has a normally open solenoid that supplies GHe pressure to actuate the valve open. If control power to this solenoid is inadvertently removed, the vent valve is forced open by GHe pressure. This failure mode is a safe condition. Each vent valve also has a normally closed solenoid to override the relief setting for checkout, and a low-pressure vent mode select solenoid to reduce the tank relief valve pressure setting after MPS engine start. Only the low-pressure vent mode select solenoid is used during flight; the other control solenoids are used only during ground operations.

The control circuits for each of these solenoids must be two alternate paths to obtain FO/FS operation. This requirement is satisfied by one avionics path and one hardware path for the normal-close and override-close solenoids, and two avionics paths for the low-pressure vent mode solenoids. Hardwire control is considered necessary for standby operations when the DCM subsystem is not available. (Figure 2-170 presents a block diagram of the circuit.) Only the low-pressure vent mode solenoids require redundancy management switching to alternate paths.

Propellant Tank Pressure Switches. Each of the three pressure switches on each propellant tank are wired to two ACT units. This, plus the backup pressure monitoring of the pressure transducers, gives a two- and three-failure tolerance for the 8 psig and flight-pressure switch, pressure-transducer combination. (Figure 2-170 shows the circuit diagram.)

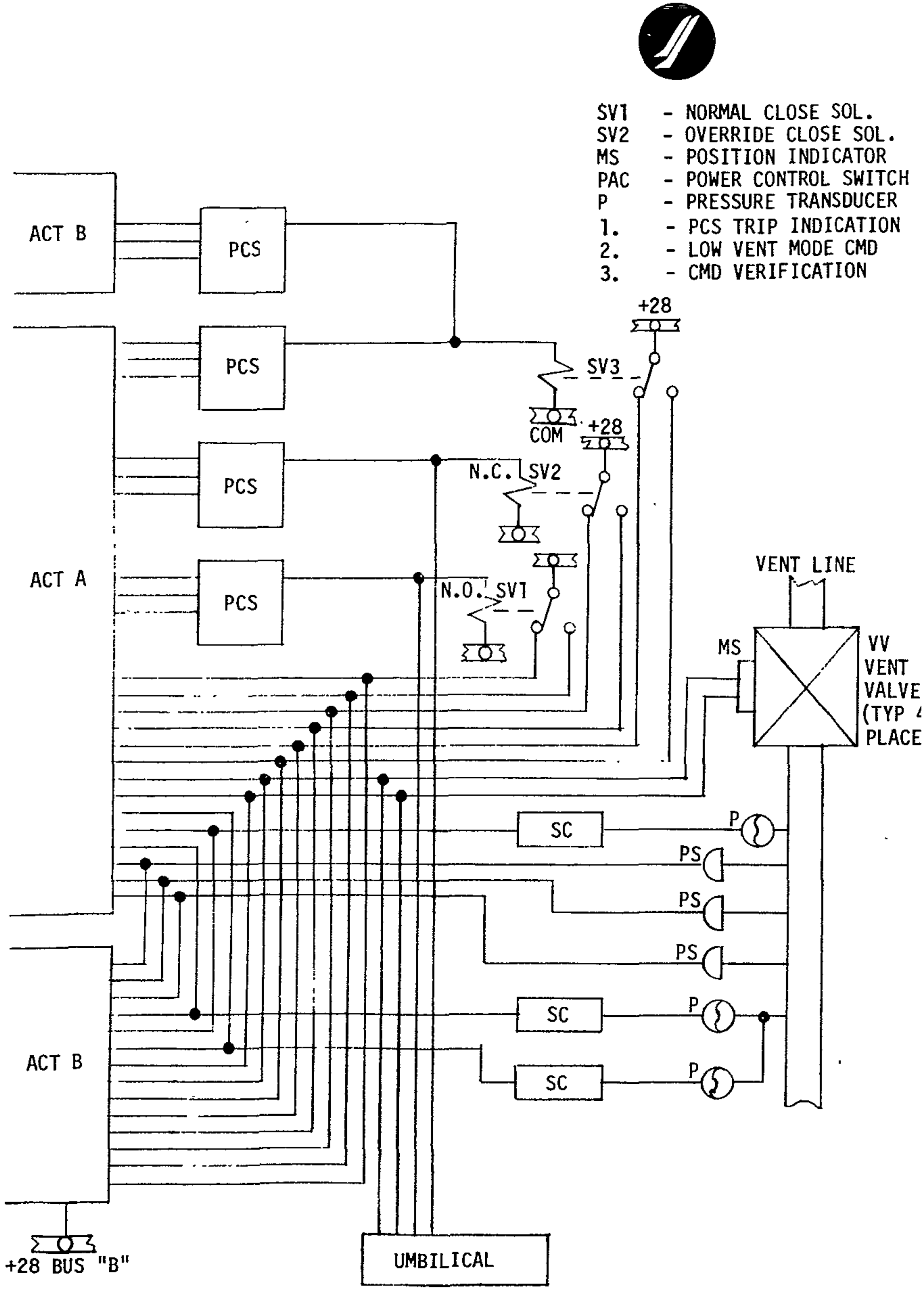


Figure 2-170. Typical Vent Valve Control Block Diagram

LO_2 and LH_2 Pre-Press Solenoids. These solenoids require only two control paths through the data bus system because the ground-controlled auxiliary pressurization system is a backup mode that may be utilized. Electrical control of these valves is terminated at liftoff; therefore, during the operational time line, the system is FO/FS for each valve. (Figure 2-171 presents the circuit diagram.) Since this circuit has two control and feedback paths, redundancy management will be required in the DCM system.

Valve Actuation GHe Receiver Vent Valve. The control circuit for this valve is identical to the circuit for the pre-press solenoid valves shown in Figure 2-171. This solenoid is used only during ground operations, and a failure of the valve to operate open can be resolved by manually bleeding the receiver. Two control paths give FO/FS operational redundancy. This circuit also requires redundancy management in the DCM system.

Instrumentation. Pressure measurements are required for performance monitoring and failure isolation. Tank pressures are redline "critical" measurements and therefore triple paths are provided to achieve FO/FS redundancy level (See Section 2.3.6 for a description of typical transducers and signal conditioning equipment.)

Performance Analysis. The redundancy level required in each portion of the system is determined by the operating requirements of each function. The failure modes and criticality of each mode differ with each component being controlled. The redundancy required to otbain FO/FS operation of each control circuit has been discussed in the previous section. Failure modes and their resulting effect on the system operation were also considered.

Redundancy management requirements, based on the required system operation, are determined by the available alternate paths and status monitoring capability. During operations other than checkout, a status check and profile comparison will be required repetitively at fixed intervals. A discrepancy will require immediate corrective action by the DCM system to isolate the failed component and select an alternate operational path.

The valve actuation helium receiver vent valve has two operational paths for control and monitor functions. One path will be used as the primary path during operations until a failure occurs. The computer in the DCM system will monitor the performance of the valve and circuits by comparing the status indicators with the desired system profile.

1. Solenoid open indication ON (1 and 2)

2. Solenoid closed indication OFF (1 and 2)

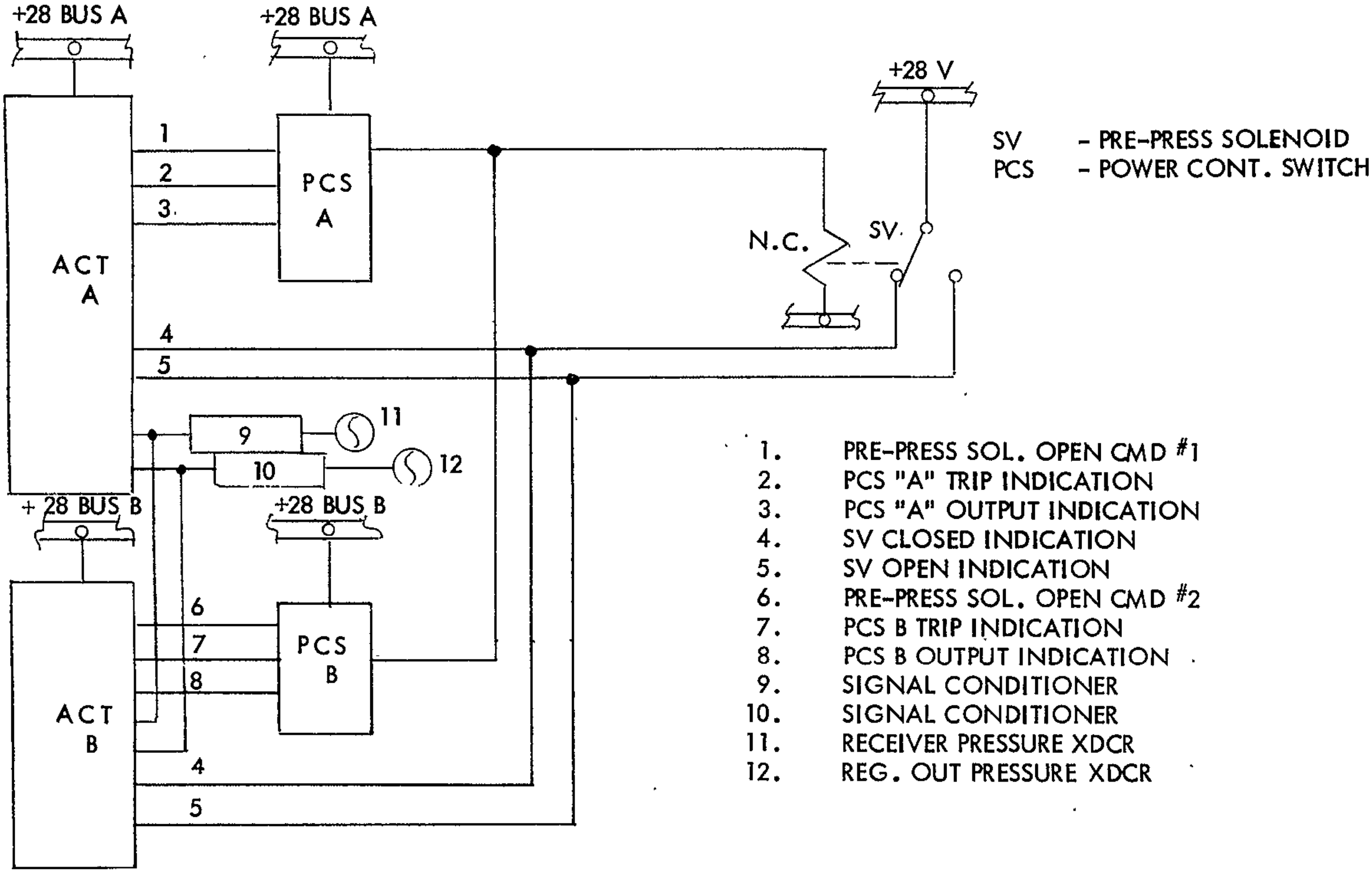


Figure 2-171. Typical LO_2/LH_2 Pressurization Solenoid Control Block Diagram

3. Power control switch output ON

4. PCS trip indication OFF

A comparison failure of one indication will be reverified for a predetermined number of samples and then be telemetered as an indicator failure. A failure of two or more indicators for a predetermined number of samples will cause the DCM system to select an alternate path, isolate the failed path, and telemeter the failure information.

Other control circuits will be handled in a similar manner. The sequence of events and control switching will be determined by the requirements of the particular function and the available alternate modes of the control circuits.

Checkout of redundant circuits will require sequential actuation and reverification that all components are operational and controllable.

Propellant Management Electrical Control Subsystem

The primary purpose of the propellant management electrical control subsystem (PMECS) is measurement of propellants while loading the MPS and APS engine fuel tanks and for depletion cutoff. After ESS boost and during the remainder of the mission, the system provides telemetered data on main tank residuals and APS propellant reserves whenever slight acceleration is induced to settle propellants. Both analog and discrete measurements are provided, with sufficient redundancy to assure FO/FS operation for propellant loading and depletion cutoff. The addition of a zero-g measuring system for monitoring the APS propellant reserves in orbit was considered and rejected because of development costs and the existing capability to apply low-thrust levels to settle propellants.

The ESS will fly with the main engine PMR valves locked in the 6:1 ratio; therefore the propellant utilization capability inherent in the system is not used.

Requirements. PMEC requirements are:

1. FO/FS measurement of propellants, with a 0.2-percent-of-full load accuracy must be provided for propellant loading purposes.

2. The system must be capable of providing in-flight data on propellant reserves.

3. All in-tank equipment must be compatible with the cryogenic environment and must not be capable of producing ignition energy in a liquid or gaseous environment.

4. Propellant depletion must be indicated when any propellant tank reaches a predetermined low level.

System Description. The ESS PMECS retains the Saturn S-II concepts and most of the hardware to provide the capability and flexibility of wide variance in tank loading which is dependent on the payload and mission. An improvement in reliability and lower cost was obtained by incorporating a new electronics package in place of the computer used on S-II. The additional equipment required for the APS propellant measurements is the same as used for the main propellant tanks except the capacitance probes are smaller.

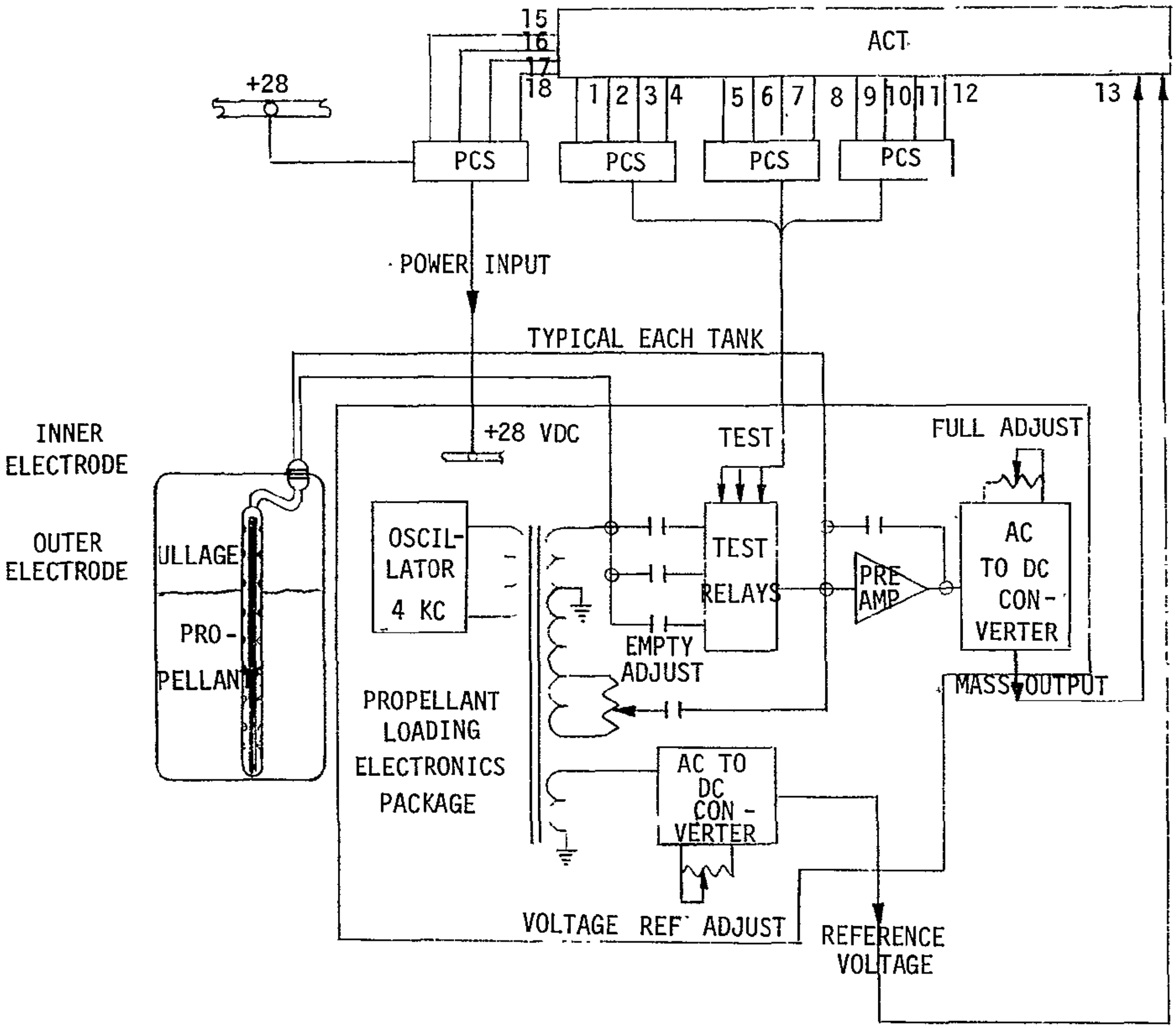
The short-circuit design of all in-tank equipment and interfacing external equipment will assure that stoichiometric values are well below the energy levels required to initiate LO_2 ignition.

Each of the seven propellant tanks, main LO_2 and LH_2 and APS LO_2 and LH_2, has an analog measurement system and discrete level sensors for propellant loading and level monitoring. The analog systems are composed of capacitance probes with electronic circuits to develop an analog voltage output proportional to the capacitance of the probe, which has a variable capacitance dependent on the liquid level in the tank. (Figure 2-172 presents a typical circuit block diagram of the capacitance system electronic circuits.)

The major components of the analog measurement systems are described in the following paragraphs. Since the capacitance level monitoring system is primarily for propellant loading, the APS liquid hydrogen four-tank arrangement has a capacitance probe in the fourth tank only. The level in the first three tanks is monitored by point-level sensors only.

Point Sensors. The point sensors give discrete level indications of the liquid in each propellant tank. The main LO_2 tank has eight sensors mounted near the top of the tank for loading purposes and two near the bottom to assist in residual calculations as well as propellant loading. The main LH_2 tank has five sensors mounted near the top of the tank and two near the bottom.

The point sensor system, known as a "hot-wire system", consists of sensors and an electronic controller assembly for each sensor. The sensing element is a gold plated platinum wire carrying a constant current. As the sensor is covered by a cryogen, the resistance of the wire element drops sharply and the voltage across it drops. The controller assembly senses the voltage change and actuates an output relay that provides a discrete 28-vdc



1. 1/3 FULL SIMULATE COND
2. PCS TRIP IND
3. PCS RESET COND
4. PCS OUTPUT IND
5. 2/3 FULL SIMULATE COND
6. PCS TRIP IND
7. PCS RESET COND
8. PCS OUTPUT IND
9. FULL SIMULATE COND
10. PCS TRIP IND
11. PCS RESET COND
12. PCS OUTPUT IND
13. MASS OUTPUT
14. REFERENCE VOLTAGE
15. POWER ON COND
16. PCS TRIP IND
17. PCS RESET COND
18. PCS OUTPUT

Figure 2-172. Propellant Loading Electronics output. No direct command control is required for this system except power application and checkout simulation commands. Sensor locations in the main and APS propellant tanks are illustrated in Figures 2-173 and 2-174. Figure 2-175 is a schematic diagram of a sensor controller assembly. A typical control, checkout, and monitor diagram is shown in Figure 2-176 for all hot-wire sensors.

Depletion Cutoff Sensors. Propellant depletion engine cutoff sensors are located near the feedline outlet in each main propellant tank and near the APS prevalve in the APS LO_2 tank and the APS LH_2 tank. Four hot-wire sensors are mounted around a pedestal in the bottom of the main LO_2 tank, the APS LO_2 tank, and the APS LH_2 tank; the main LH_2 tank has two of the sensors on either side of each feedline exit. The sensors and controller assemblies are the same as used for the level monitoring system. The location of these sensors is shown in Figures 2-173 and 2-174. The circuit is the same as shown in Figure 2-176.

Performance Analysis. The capacitance probe mass-sensing systems provide an analog measurement of the propellant level from empty to full. The point sensors provide discrete level measurements which are backup measurements for propellant loading, the most critical function of the propellant management system. The point sensors and capacitance probe provides FO/FS redundancy in the propellant tanks for propellant loading.

The control and measurement circuits for each capacitance probe are simplex single paths with backup capability provided by discrete measurements from the point sensors. The control and measurement circuits for the point sensors are duplex.

After liftoff, the capacitance probes and level sensors in the main propellant tanks are used for information only and are not required for flight operations. After main engine cutoff, the power to these systems is terminated.

The APS propellant level monitoring system will remain active throughout the entire mission. However, the system is not expected to yield usable information during periods of zero g.

The propellant depletion sensors are quad redundant in each tank, and two out of four must signal depletion before an engine cutoff will be commanded by the DCM system. The sensors for each tank will be operating at liftoff, but the computer will be programmed to ignore them until a predetermined point in the boost phase (for main engines) and the OMS burn periods (for OMS engines).

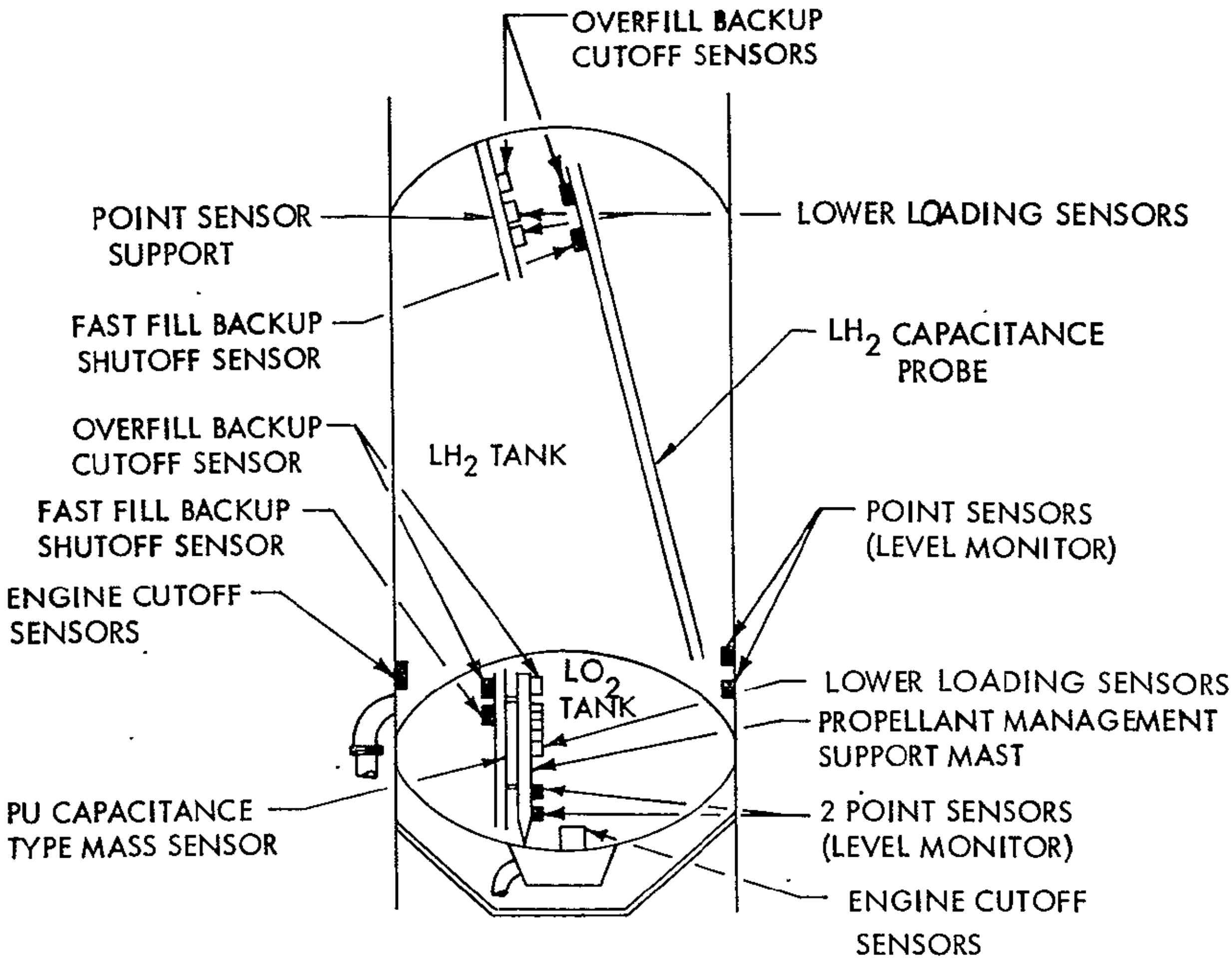


Figure 2-173. Propellant Management System
Main LO_2 and LH_2 Tank Sensors

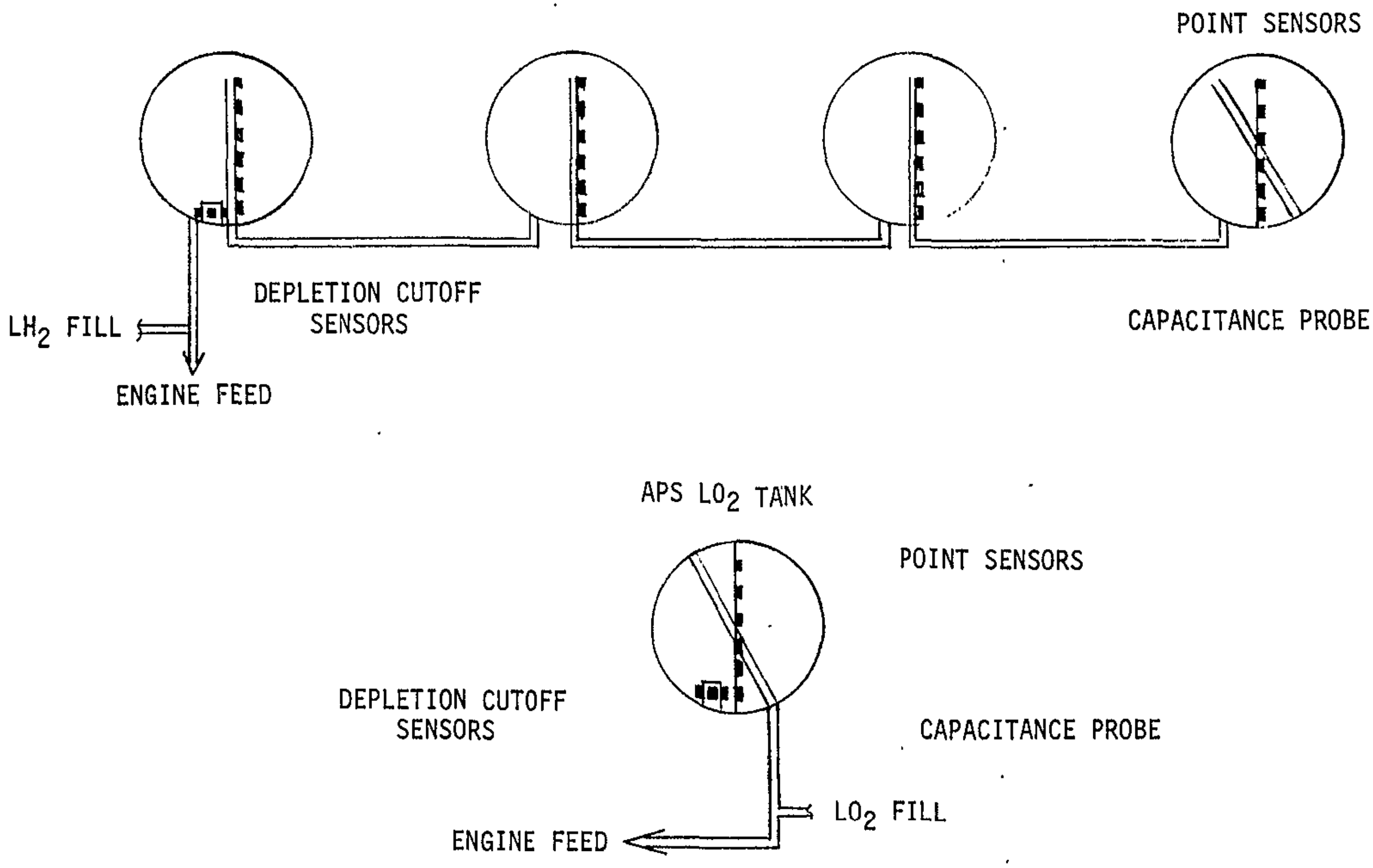


Figure 2-174. APS Propellant Level Monitoring System

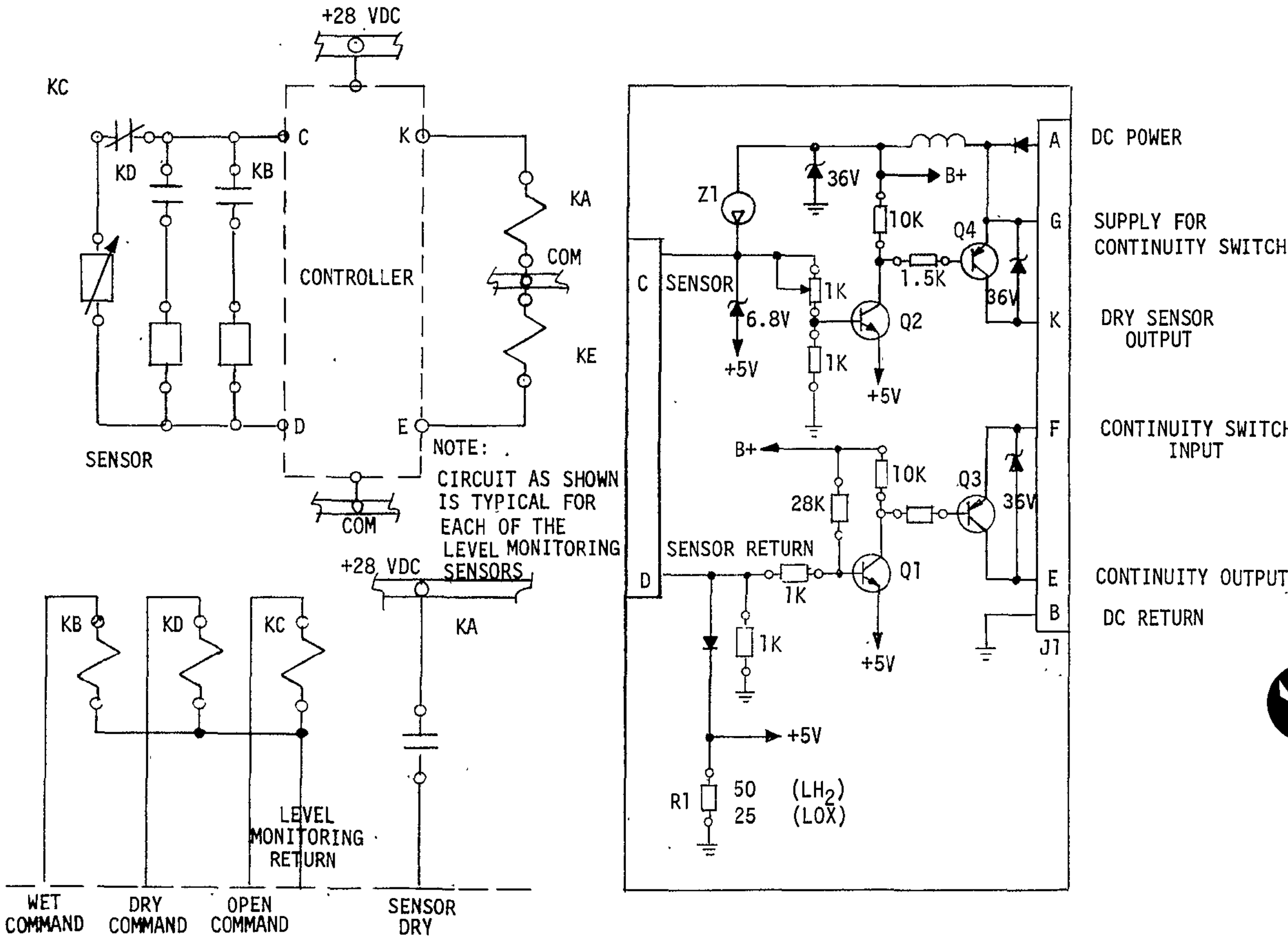
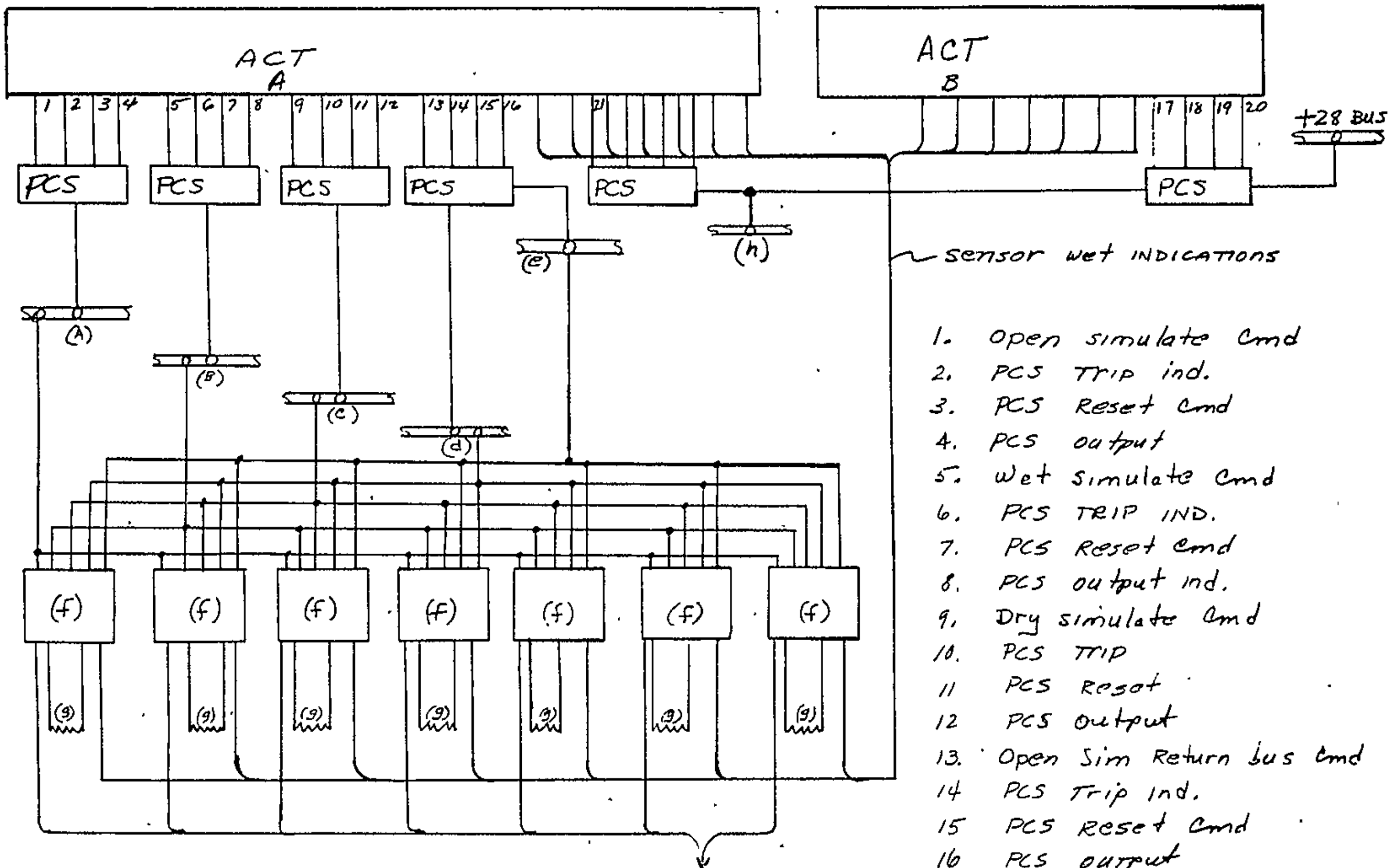


Figure 2-175. Point Sensor Level Monitoring



(A) Open Simulate bus
(B) Wet Simulate bus
(C) Dry Simulate bus
(d) Open simulate return BUS
(e) Return bus
(F) Sensor control Assys
(G) Point sensors
(h) Controller Power Bus

TYPICAL POINT SENSOR CONTROL BLOCK DIAGRAM

1. Open simulate Cmd
2. PCS Trip ind.
3. PCS Reset Cmd
4. PCS output
5. Wet simulate Cmd
6. PCS TRIP IND.
7. PCS Reset Cmd
8. PCS output ind.
9. Dry simulate Cmd
10. PCS TRIP
11 PCS Reset
12 PCS output
13. Open Sim Return bus Cmd
14 PCS Trip ind.
15 PCS Reset Cmd
16 PCS OUTPUT
17 Level monitor PWR BUS B Cmd
18 PCS TRIP IND
19 PCS Reset Cmd
20 PCS output
21 Level monitor PWR BUS A Cmd

Figure 2-176. Typical Point Sensor Control Block Diagram

The main engine LH_2 depletion cutoff system is a backup for velocity cutoff, and four sensors are used in each tank to achieve FO/FS redundancy even though it is a backup system.

Redundancy. Redundancy management will be limited to switching power and response monitoring between two alternate paths through two ACT units for the hot-wire sensors only. No management switching is required for the simplex capacitance level sensing circuits. The DCM system will activate the hot-wire sensors from a primary bus and monitor the system through alternate ACT units. A failure of the primary bus system will cause the DCM system to activate the secondary source to the sensors. Each sensor output will be checked against its corresponding open-circuit indicator to determine its validity.

Separation Electrical Controls

The separation electrical controls (SEC) consist of the electrical control circuitry and electrical components which are required to initiate ordnance devices which will separate the ESS from its payload. The system will be sequenced by an automatic computer located onboard the ESS vehicle. The ESS/booster separation sequence is controlled by the booster and is discussed in Volume II, Book 3 of this report.

Requirements. The requirements are as follows:

1. There must be a separation plane between the ESS vehicle and the payload.
2. The ESS must provide the electrical control for the ESS/payload separation system.
3. The system must utilize the exploding bridgewire (EBW) technique.
4. Where design options exist, the selected configuration will be the most cost effective.
5. The separation system must be designed in such a way that a single failure will not prevent the SEC from initiating the separation of the payload.
6. All separation system control functions must be designed to be supplied from within the ESS.
7. The ESS computer via the data bus and ACT units must be utilized to provide separation command, timing, and sequencing.

8. The ESS/payload separation system must be designed to allow rapid checkout of the system with an automatic computer controlled program.

9. Pulse sensors will be used to provide EBW firing unit checkout.

System Description. The ESS electrical control for the ESS/payload separation system is shown in Figure 2-177. Redundant data buses will provide computer control and monitoring to redundant ACT units. ESS/payload separation will be accomplished by the computer addressing ACT unit A to select the output channel connected to the control input of PCS 1 (power control Switch 1). The PCS control input will require a nominal +5 vdc discrete input at 10 milliamperes of current. The output of PCS 1 will now switch to a nominal +28 vdc and provide the charging voltage for arming ESS/payload separation firing unit 1. The +28 vdc output of PCS 1 will be monitored by ACT unit D. The +28 vdc output of PCS 1 is protected against a short circuit to ground by an internal circuit which automatically removes the +28 vdc output. PCS 1 also contains a +5 vdc short-circuit trip indication output that operates when the +28 vdc output is shorted to ground. PCS 1 can be reset by removing the control input signal from ACT A when the short has been removed.

ACT B can also be addressed by the computer to select the output channel connected to the input of PCS 2. In this case, the output of PCS 1 will also provide the charging voltage for arming the ESS/payload separation firing unit 1. The input power for PCS 2 is obtained from a different power source than that for PCS 1. Each firing unit is provided with redundant power sources for the arming and triggering signals. The ESS/payload firing unit 1 will be armed within 1.5 seconds. The trigger signal can now be sent which will discharge the firing unit and initiate the ordnance which is necessary for separating the ESS from the payload. The trigger signal to the firing unit 1 is originated by a computer address to ACT unit A. The output channel of ACT A, which is connected to PCS 3, is activated by the computer address to the ACT unit. The output of PCS 3 will now switch to a nominal +28 vdc which will provide the trigger signal to ESS/payload separation firing unit 1. It is assumed that the solid-state circuitry in PCS 3 will require two failures to inadvertently arm an EBW firing unit, but it would still take another two failures in another solid state switch to advertently trigger an EBW firing unit. A redundant trigger signal to ESS/payload separation firing unit 1 can also be provided by a computer address to ACT B which selects the output channel connected to the input of PCS 4. In this case, the +28 vdc output of PCS 4 will provide the trigger signal necessary for triggering ESS/payload firing unit 1. ESS/payload pulse sensor 1 will be used to ground check the EBW firing 1 for the pulse that is necessary to initiate the separation ordnance. The pulse sensor will be replaced before flight by an ordnance device. During ground checkout, the pulse sensors will be controlled by the computer. As shown in Figure 2-177, the computer will address ACT B to command PCS 5 to switch +28 vdc to the power and input connections of ESS/payload pulse sensor 1. The output of the pulse sensor will switch on +28 vdc when the EBW firing unit is triggered. An analog output voltage (0 to 5 vdc) of the EBW firing unit, representing the voltage charge on its capacitor, will be monitored by ACT unit D for the correct value. The +28 vdc output of the pulse sensor will be removed by a reset signal to the pulse sensor. The pulse sensor can be self-checked without triggering the EBW firing unit with a computer address to ACT B and PCS 6. ACT unit D will monitor the solid state power controller outputs and provide the computer with this information. The information can be used for fault isolation when a failure occurs during normal separation system operation. This will also provide the capability to determine a failed component when the computer addresses an ACT unit and no response is indicated by the monitoring ACT unit. The remaining circuitry shown in Figure 2-177 for firing unit 2 and pulse sensor No. 2 will operate the same as previously described for firing unit 1 and pulse sensor 1. The separation system contains two completely redundant and independent systems.

Redundancy. The redundancy provisions are as follows:

1. Power. Three redundant main power sources of +28 vdc will be utilized for powering the ACT units, power control switches, and EBW firing units, as shown in Figure 2-177, to accomplish ESS/payload separation.

2. Component. Redundant power control switches will be utilized in controlling the arm and trigger signals to each EBW firing unit. Redundant ACT units will be used to provide computer control (via the data bus) of each EBW firing unit.

3. Failure Mode Effects Analysis. Redundant EBW firing units are used to initiate the explosive ordnance required for ESS/payload separation. As shown in Figure 2-177, there are redundant data buses to be used by the computer in addressing redundant ACT units. Redundant ACT units are also used for monitoring. Inadvertent separation is prevented by using individual solid-state switches for arming and triggering each EBW firing unit. Two failures would have to occur in one solid-stage power control switch (PCS) to inadvertently provide a +28 vdc output in the absence of an input signal. It would require four solid-state switch failures to inadvertently separate the ESS from the payload. Completely redundant separation circuitry includes a backup mode which is inherent in the system design.

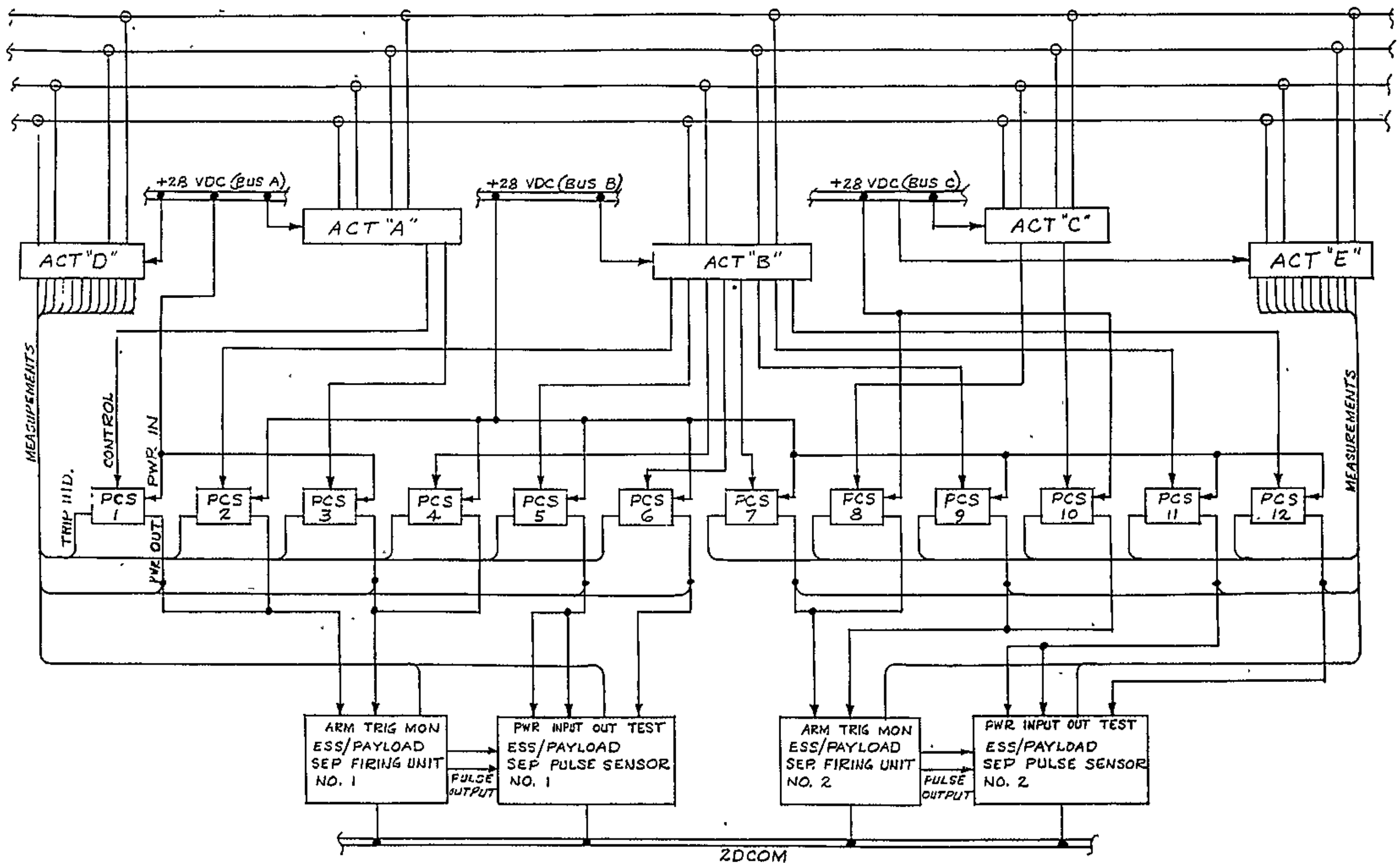


Figure 2-177. Separation Electrical Controls

4. Management Requirements. Separate and redundant separation circuitry is arranged to allow rapid computer control for ascertaining response to a command and to facilitate rapid fault isolation in the case of no response. The output of each power controller switch is monitored and may be used to determine the response to a computer command. Commanding and monitoring other ACT unit output channels will provide isolation to a completely failed ACT unit or an individual output channel. Adequate control and monitoring functions are provided to rapidly isolate a fault to an individual component.

5. Permissivities. Require program instructions to require ESS/booster separation complete and orbit insertion assured to enable the ESS/payload separation arming command.

Safing Electrical Control Subsystem

The safing electrical controls subsystem consists of those items required to operate together to effect the opening of the ESS main propellant tanks to allow boiloff of residual propellants. The vehicle must be safed from propellant and pressurant overpressures. All safing functions must be designed to vent pressure vessels with pressure greater than 50 percent of burst to less than 25 percent of design burst pressure. The safing system will safe the main propellant tanks in a nonpropulsive manner to prevent ESS/payload recontact after separation and to assure that the ESS vehicle will not break up in orbit and endanger the payload.

Requirements. The requirements are as follows:

1. Safing component single failures must not affect the flight system during prelaunch or flight.

2. Where failure to safe a tank could result in overpressure and possible rupture, redundant means of safing must be used. Redundancy may be in the form of a simplex electrical control circuit with a backup mechanical relief.

3. Where design options exist, the selected configuration will be the most cost effective.

4. Safing systems will fail operational; that is, all single-point failures shall be eliminated by redundancy.

5. All safing electrical control functions will be designed to be supplied from within the ESS stage.

6. The ESS computer via the data bus and ACT units will be utilized to provide safing commands, timing, and sequencing.

7. Physical separation of the ESS stage from the payload shall not be required to initiate the safing functions.

8. The ESS safing system shall be designed to allow rapid checkout of the system with an automatic computer-controlled program.

9. Pulse sensors will be used to provide EBW firing unit checkout.

System Description. This safing subsystem may be described as follows: The ESS vehicle main propellant tanks will be safed approximately 5 seconds after main engine cutoff unless safing is delayed to allow the main tank propellants to be used as ballast. The main LO_2 and LH_2 tanks are the only ESS vessels requiring safing. Figures 2-178 and 2-179 are schematic block diagrams of the redundant LO_2 and LH_2 tank safing system.

Redundant data buses will provide computer control and monitoring to the redundant ACT units, as shown in Figure 2-178, for the LO_2 tank safing system. LO_2 tank safing will be accomplished by the computer addressing ACT unit A to select the output channel connected to the control input of PCS 1 (power control switch). The PCS control input will require a nominal +5 vdc discrete input at 10 milliamperes of current. The outpuf of PCS 1 will now switch to a nominal +28 vdc and will provide the charging voltage that arms the LO_2 tank EBW firing unit 1. The +28 vdc output of PCS 1 will be monitored by ACT unit B. The +28 vdc output of PCS 1 is protected against a short circuit to ground by an internal circuit that will automatically remove the +28 vdc output. PCS 1 contains a +5 vdc short-circuit trip indication output that operates when the +28 vdc output is shorted to ground. PCS 1 can be reset by removing the control input signal from ACT A when the short has been removed. ACT A may also be addressed by the computer to select the output channel connected to the input of PCS 6. In this case, the output of PCS 6 will also provide the +28 vdc charging voltage necessary to arm the LO_2 tank EBW firing unit 1. The input power for PCS 6 is provided by a separate power source. Each firing unit is provided with redundant arming and triggering signals which originate from separate power sources.

The LO_2 tank EBW firing unit 1 will be armed in 1.5 seconds and is ready to be triggered and, in turn, to initiate the ordnance that will open the LO_2 tank valve to safe the tank. The trigger signal to the LO_2 tank EBW firing unit 1 is originated by a computer address to ACT unit A. The output channel of ACT A, which is connected to the control input of PCS 2, is activated by the computer address to the ACT unit. The output of PCS 2 will not switch to a nominal +28 vdc which will provide the trigger signal to

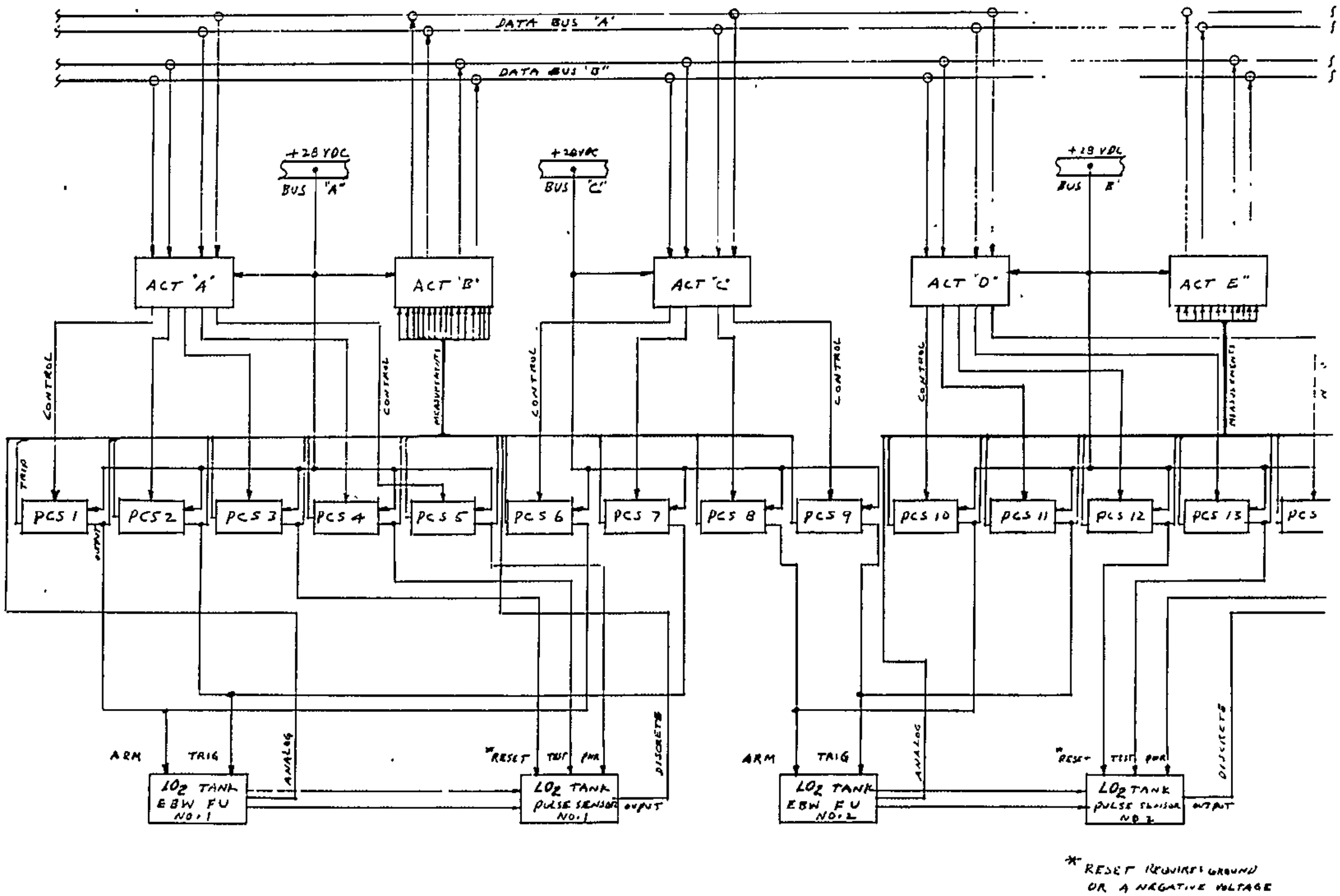


Figure 2-178. Safing Electrical Controls (Main LO_2 Tank)

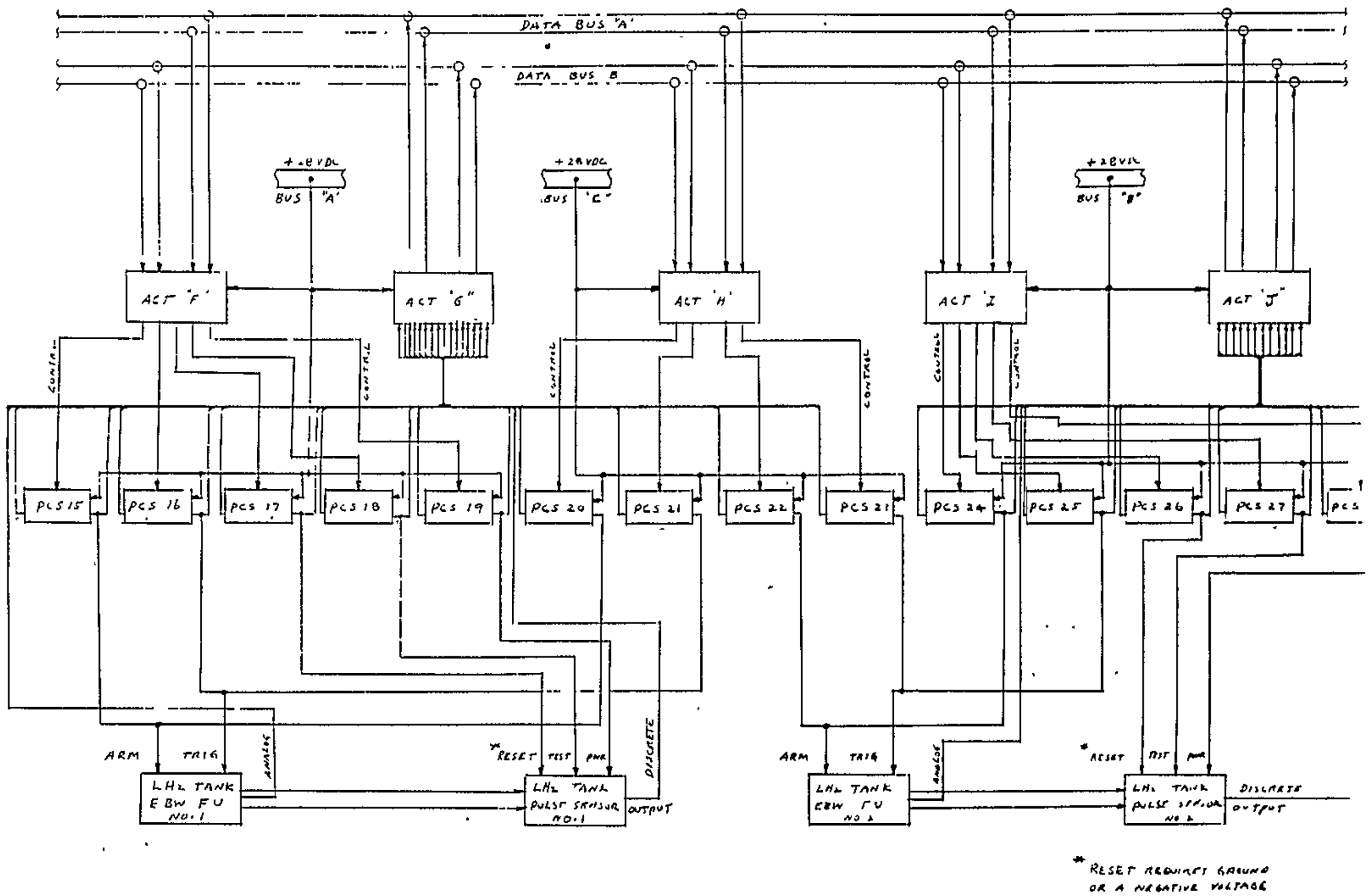


Figure 2-179. Safing Electrical Control (Main LH2 Tank)

the LO_2 tank EBW firing unit 1. It is assumed that the solid-state circuitry contained within PCS 2 will require two failures to inadvertently produce a +28 vdc output in the absence of an input signal. The arm input would require two failures in PCS 1 to inadvertently arm the firing unit and two additional failures would be necessary to inadvertently safe the LO_2 tank. Four solid-state switch failures in the same safing system would be necessary to inadvertently safe a tank.

A redundant trigger signal to LO_2 tank EBW firing unit 1 may also be provided by a computer address to ACT A which selects the output channel connected to the input of PCS 7. In this case, the +28 vdc output of PCS 7 will provide the trigger signal necessary for LO_2 tank EBW firing unit 1.

LO_2 tank pulse sensor 1 will be used to ground check the EBW firing unit 1 for the correct charge value that will be required to initiate the ordnance chain for opening the safing valves. The pulse sensor will be replaced before launch by the detonator. During ground checkout, the pulse sensors will be controlled by the computer. Referring to Figure 2-177, the computer will address ACT A to command PCS 5 to switch +28 vdc to the power input of LO_2 tank pulse sensor 1. The output of the pulse sensor will switch to +28 vdc when the EBW firing unit is triggered. An analog output voltage (0 to 5 vdc) of the EBW firing unit, representing the charge on the capacitor, will be monitored by ACT unit B for the correct value. The +28 vdc output of the pulse sensor will be removed by a reset signal to the pulse sensor. The computer will reset the pulse sensor by addressing ACT unit A and controlling PCS 3 to switch the reset signal to the pulse sensor. The pulse sensor may be self-checked without triggering the EBW firing unit via a computer address to ACT A and PCS 4.

ACT unit B will monitor the solid-stage power controller outputs and provide the computer with this data. The data can be used for fault isolation when a failure occurs during normal safing system operation. This will also provide the capability to determine a failed component when the computer addresses an ACT unit and there is no response indicated by the monitoring ACT unit.

The remaining safing control system circuitry in Figures 2-178 and 2-180 will be identical and will operate the same as described previously for Figure 2-179. These two schematic block diagrams represent two redundant and independent safing systems for each of the main propellant tanks which are powered from three redundant and separate sources.

Components. The components required to mechanize the electrical control of the safing system, as defined in Paragraph 3.2.5, are the EBW pulse sensor, firing unit and the solid-state power controller switch which are detailed in Paragraph 2.3.5 under Separation Electrical Controls.

Redundancy. The redundancy provisions are as follows:

1. Power. Three redundant main power sources of +28 vdc will be utilized for powering the ACT units, power control switches, and EBW firing units in Figures 2-178 and 2-179 to accomplish LO_2 and LH_2 tank safing.

2. Component. Redundant power control switches will be utilized in controlling the arm and trigger signals to each EBW firing unit and pulse sensors. Redundant ACT units will be used to provide computer control (via the data bus) of each EBW firing unit and pulse sensor. Redundant ACT units will be used to provide monitoring of the power control switches, EBW firing units, and pulse sensors.

3. Failure Mode Effects Analysis. Redundant EBW firing units and associated ordnance train will be used to safe each main tank. As shown in each figure, there are redundant data buses to be used by the computer in addressing redundant ACT units. Redundant ACT units are also used for monitoring. Figure 2-178 contains redundant circuitry for safing of the LO_2 tank. Figure 2-179 contains redundant circuitry for safing of the LH_2 tank. Inadvertent safing of a tank is prevented by using individual solid-stage switches to arm and trigger each EBW firing unit. Two failures are necessary in one solid-stage power controller switch (PCS) to inadvertently provide a +28 vdc output in the absence of an input signal. It would require four solid-state failures in Figure 2-178 or Figure 2-179 to inadvertently safe the LO_2 or LH_2 tank.

4. Management Requirements. Separate and redundant safing circuitry for each of the main propellant tanks is arranged to allow rapid computer control for ascertaining response to a command and to facilitate rapid fault isolation in the case of no response. The output of each power controller switch is monitored and may be used to determine response to a computer command. Commanding and monitoring other ACT unit output channels will provide isolation to a completely failed ACT unit or an individual output channel. Adequate control and monitoring functions are provided to rapidly isolate a fault to an individual component.

5. Backup Modes. Completely redundant safing circuitry for each of the main propellant tanks include a backup mode which is inherent in the system design.

Thrust Vector Electrical Controls

There are four independent thrust vector electrical control (TVEC) elements. The differences between the elements for the main propulsion engines and the orbital maneuvering system (OMS) are primarily in their load handling capacities rather than in basic operation. Therefore, the MPS engine actuation subsystem will be discussed with only important differences from the OMS actuation system indicated.

The TVEC provides hydraulic power for the orbiter engine nozzle extender-retractor mechanism and for the eight engine-gimbaling servo-actuators. The orbiter engine accumulator reservoir manifold assembly (ARMA) will be pressurized before launch to allow the engines to be positioned prior to engine start. The OMS engine ARMA does not require prelaunch prepressurization. The OMS actuation system will be activated before performing orbital eccentricity changes, before orbital altitude changes, and before the deorbiting during the earth-orbiting period to keep fluid sufficiently warm to support OMS gimbaling requirements. (See Figures 2-180 and 2-181 for block diagrams of the OMS and orbiter engine controls.)

Requirements. The actuation system will be designed to meet the FO/FS criteria. A switching system will be provided to permit operation of the electric motors from either a ground power or airborne power source. All solenoid valve operations will be under direct computer control. Controlled application of power to the OMS hydraulic pumps during the earth-orbiting period will be as a function of hydraulic fluid temperature.

System Description. The operation of the TVEC is similar to that of the engine actuation system on the Saturn S-II vehicle in that hydraulic power for in-flight actuation is provided by a hydraulic pump on the engine and requires an auxiliary system for ground operation. The main differences are the MPS engine nozzle extension-retraction requirement, the OMS electric motor operation requirement, and the redundancy requirements.

Power Transfer Switches. These switches permit the application and removal of either ground-supplied or vehicle-supplied three-phase, 400-Hz power. The ground-supplied source is applied through a motorized transfer switch controlled by computer command. The vehicle source is applied through a system of transfer switches which provide the necessary redundancy. Each pair of in-line switches is controlled by a single computer command. The input power is applied in such a way that no set of two failures can disable both hydraulic systems, thereby allowing no less than one engine per pair to remain under control.

Electric Motor/Hydraulic Pump. The motor converts the applied electrical power into the mechanical torque necessary to satisfy the demands of the hydraulic pump. The larger the demand, the greater the torque and the larger the input electrical power requirement. The hydraulic pump is a variable delivery, variable displacement, pressure-compensated type. It is designed to provide a minimum pressure at any flow demanded by the hydraulic load.

Solenoid Valves. Solenoid valves allow the computer to control the flow of hydraulic fluid. The valves shown on the outputs of the hydraulic pumps are for reducing the start-up torques on the engine turbines. Decreasing the engine turbine load decreases the probability of engine stall-out during the ignition sequence. It may be necessary to utilize a check valve and operate the electric motor/hydraulic pump before engine ignition. This would reduce the short-duration engine turbine start-up torque which would be present with an unpressurized actuation system. A system configuration which is still being considered is the placement of the electric motor on the engine turbine hydraulic pump. This configuration could aid engine ignition by prespinning the engine turbine.

Accumulator Reservoir Manifold Assembly (ARMA). The ARMA consists of an accumulator which stores hydraulic power, supplements hydraulic pump flow during peak system demands, and dampens pump discharge pulsations during peak system surges; a reservoir which pressurizes the pump inlet, stores system hydraulic fluid, and receives system return flow; normally open lock-up solenoid valves which close the inlet and outlet ports to the accumulator, thereby storing hydraulic power; filters, a thermal switch, a nitrogen-charging valve, and instrumentation transducers.

The ARMA is pressurized with nitrogen (orbiter engines) before liftoff, and the lockup solenoids are energized (valves closed). Some time before engine ignition, the lock-up solenoids are deenergized. Hydraulic power thus becomes available for positioning the engines before ignition, permitting smooth transition into the powered phase of flight.

The OMS engine ARMA does not require lock-up solenoid control since the hydraulic pumps must pressurize the entire hydraulic system before engine ignition. These two hydraulic systems, however, employ temperature monitoring of the hydraulic fluid for purposes of controlling its intermittent circulation and preventing excessively high or low temperatures. Operation of the pumps will commence whenever any two of the four sensors reach a low-temperature limit or when one sensor reaches an under low-temperature limit and no other sensor indicates a high-temperature limit. Pump operation will cease when any sensor indicates a high-temperature limit and no other sensor indicates a low-temperature limit. In the event that either sensor of one system indicates the low-temperature limit while

FOLDOUT FRAME 1

FOLDOUT FRAME 2

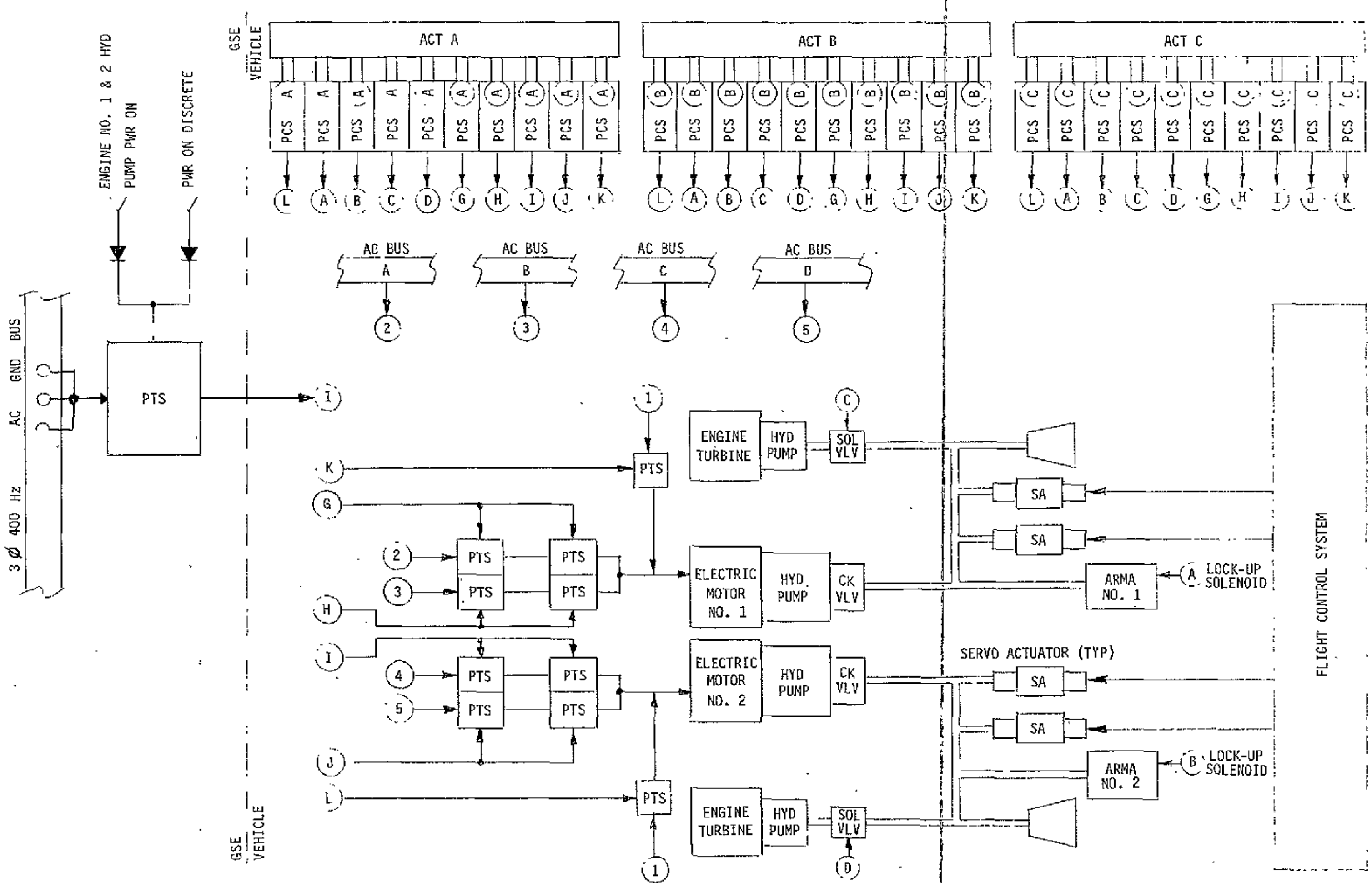


Figure 2-180. Main Propulsion System Engines, Thrust Vector Electrical Controls

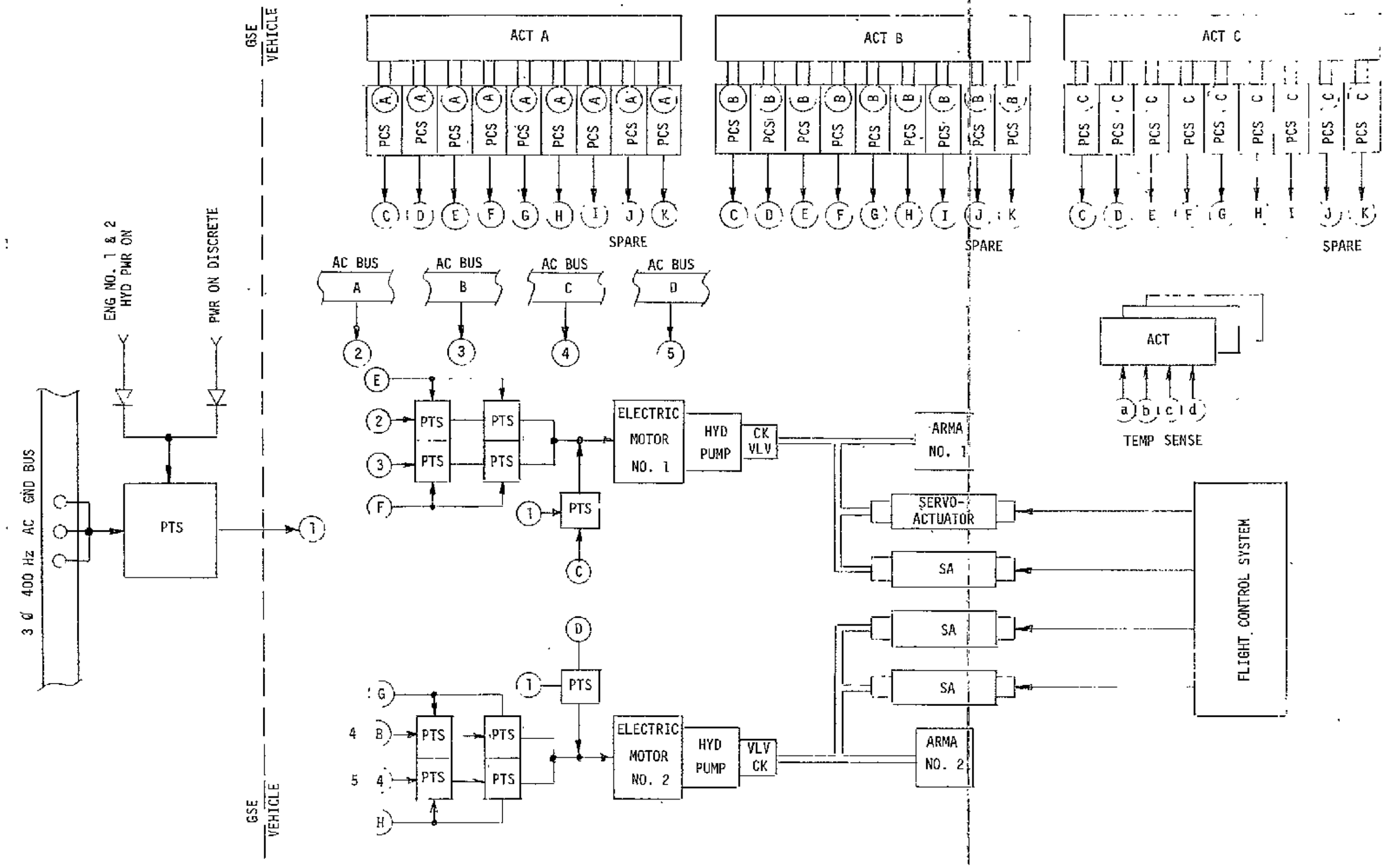


Figure 2-181. AMS Engines, Thrust Vector Electrical Controls

the other system indicates at least one high-temperature limit, the hydraulic fluid will be circulated only on the one system. The sensors will fail in a normal temperature-indicating mode (see Performance Analysis).

Servoactuators. The swiveling or gimbaling of the four engines (two OMS and two Main) is accomplished by four pairs of servoactuators. Each pair is connected between the vehicle structure and each engine so that one servoactuator line of action is perpendicular to the other.

The servoactuators on each engine respond to separate commands from the flight control system to gimbal an engine in pitch or yaw. Appropriate gimbaling of the two MPS engines will produce roll control. The OMS engine servocactuators will be supplemented by the attitude control propulsion system (ACPS) for roll control. In the event that one orbiter engine is disabled, the ACPS will provide backup roll control. There is a separate actuation system for each working pair of servoactuators and the rupture of one hydraulic system cannot disable another.

Performance Analysis. The source of ac power and the power transfer switches will be capable of handling the peak current surges that are required by the electric motors during startup. The ARMA lockup solenoid valves (MPS engine hydraulic system) are energized after the storing of hydraulic power in the two accumulators.

Although the energy to activate the ARMA solenoid valves is supplied by three separate ACT units, only one need be addressed at a time. The accumulator hydraulic oil pressure may be used as a final indication that the equipment is operating normally. Operation of the solenoid valves by the secondary or tertiary ACT units is necessary if intermediate failures are detected. Solenoid valves are normally closed in the deenergized state.

As with the solenoid valve activation, application of power to the electric motors may be accomplished by addressing only the primary ACT unit. Monitoring the change in inverter input current will determine the need for addressing secondary or tertiary ACT units. Failure to apply power to an electric motor after three such attempt will cause application of power from a separate inverter. This method satisfies FO/FS requirements in that gimbaling of one engine is still possible after sustaining two in-line power transfer switch failures.

The redundancy scheme is designed so that it can sustain two failures without completely disabling the vehicle. The worst case would permit operation of at least three of the four main engine and OMS hydraulic systems. The following operations are typical of a main engine hydraulic actuation system.

1. Apply 400-Hz power to auxiliary motor 1.

    a. Failure: No output command from the first ACT unit.

       Solution: Issue command from the second ACT unit.

    b. Failure: No output command from the second ACT unit.

       Solution: Issue command from the third ACT unit.

    c. Failure: No output command from the third ACT unit.

       Solution: Exit program to predetermined decision point A.

    d. Failure: Command first power transfer switch (PTS) ON - no first inverter current.

       Solution: Remove first PTS ON command - apply second PTS ON command.

    e. Failure: No second inverter input current.

       Solution: Exit program to predetermined decision point A.

    f. Failure: Excessive first inverter input current.

       Solution: Remove first PTS ON command - apply second PTS ON command.

    g. Failure: Excessive second inverter input current.

       Solution: Remove second PTS ON command - exit program to predetermined decision point A.

2. Energize hydraulic pump unloading solenoid valve.

    a. Failure: No unloading solenoid valve discrete feedback.

       Solution: As in Items 1a, 1b, and 1c, exit program to predetermined decision point B.

3. Deenergize ARMA lockup solenoid (assume accumulator is charged).

   a. Failure: Lockup solenoid will not deenergize - no servo-actuator hydraulic power.

      Solution: Issue first ACT unit discrete profile through second ACT unit, then disable (remove power from) first ACT unit. Deenergize ARMA lockup solenoid through second ACT unit.

   b. Failure: Lockup solenoid will not deenergize - no servo-actuator hydraulic power.

      Solution: Issue second ACT unit discrete profile through third ACT unit, then disable second ACT unit. Deenergize ARMA lockup solenoid through third ACT unit.

   c. Failure: Lockup solenoid will not deenergize - no servo-actuator hydraulic power.

      Solution: Exit program to predetermined decision point C.

Decision points are as follows:

A. Failure to provide hydraulic power for nozzle extension before engine ignition will prevent the nozzles from being extended. Ignition of the engines in this mode will result in lower thrust levels and consequent modification of mission goals. Hydraulic power initially stored in the ARMA units will still be available for pre-ignition engine gimbaling.

B. Inability to activate the unloading solenoid valve would prevent bypassing the high-pressure to low-pressure hydraulic lines during engine start. The result would be a much greater torque working against the spinup of the engine turbine during the engine start sequence. The added load will reduce the probability of obtaining engine ignition. Energizing of the solenoid, however, should be accomplished before launch; therefore, it does not constitute an in-flight abort mode under these circumstances. If it deenergized after launch but before engine start and could not be reenergized, the engine start command would be issued nonetheless.

C. The probability of the solenoid lockup valves remaining closed after removal of the closing command is extremely small. If it did occur, however, the engine start command is still expected to be given, and the burden of providing hydraulic oil pressure would fall on the auxiliary motor hydraulic pump and, as the engine thrust builds up (hence, hydraulic pressure), on the engine turbine hydraulic pump. Vehicle disturbing transients would be expected during this period. Circulation of the hydraulic fluid in the OMS hydraulic system is controlled by the fluid temperature. Since there are four temperature sensors - two on each engine actuation system - turning the electric motors on or off is based on the temperatures sensed by these four sensors. Table 2-35 illustrates the operating limits and bands of the hydraulic fluid.

Motor 1 (M1) - associated with sensors a and b; Motor 2 (M2) - associated with sensors c and d. Operation will be in accordance with the following:

(1) M1 and M2 ON - any two sensors below LL or one sensor below UT while no sensor is above HL.

(2) M1 ON, M2 OFF - Sensors a and b below LL, sensor c or d above HL.

(3) M1 OFF, M2 ON - Sensor a or b above HL, sensor c or d below LL.

(4) M1 OFF, M2 OFF - any sensor above HL while no sensors are below LL.

This constitutes a FO/FS system.

2.3.6 Instrumentation

The ESS instrumentation subsystem consists of data sensors, signal conditioners, and measurement remote calibration. Instrumentation will assign the locations of the data sensor signal conditioning, the design philosophy being, where feasible, the signal conditioning equipment will be located with or adjacent to the sensor. Due to the temperature environment, some equipment has to be located in thermally controlled areas. Instrumentation will also control the types of data sensors on the ESS vehicle and in the ground support equipment to assure commonality and compatibility with the signal conditioning, calibration, and the recording equipment.

Table 2-35. Hydraulic Fluid Operating Limits

Over-temperature band	
	OT (over-temperature limits)
High-temperature band	
	HL (high-temperature limit)
Normal operating temperature band	N
	LL (low-temperature limit)
Low-temperature band	
	UT (under-temperature limit)
Under-temperature band	

Note:

N - Normal operating temperature band.

HL - High-temperature limit of the hydraulic fluid.

LL - Low-temperature limit of the hydraulic fluid.

OT - Over-temperature limit above which the hydraulic fluid will begin to sustain heat damage.

UT - Under-temperature limit below which the hydraulic fluid will begin to solidify (bi-phase or slush state).

Instrumentation Requirements

The instrumentation system must provide a means of obtaining information on the ESS to assure that an accurate performance evaluation of each vehicle subsystem can be made during ground checkout and/or flight.

To determine the criticality of the measurement or measurement hardware with consideration given to cost, weight, and the importance of the measurement if lost during checkout or flight, four categories of instrumentation requirements are established:

1. Measurements critical to the operation of the vehicle systems or subsystems
2. Component performance and system performance verification
3. Environmental verification
4. Data to assist in isolating system or component malfunction

Only those measurements considered critical to the operation of the vehicle and/or safety of the crew will be redundant.

The instrumentation system does not include discrete types of measurements or any type of engine measurements. The engine measurements and their associated signal conditioning will be the engine contractor's responsibility. Discrete-type measurements require no signal conditioning and interface directly with the DCM subsystem for monitoring. One listing of instrumentation measurements will be maintained. The ESS signal list will include discretes and all engine measurements. The ESS measurement requirements listed by the measurement type, quantity, and subsystem, excluding the engine measurements, are listed in Table 2-36.

Instrumentation Subsystem Description

Sensors. The technology gained from the Saturn S-II Program and the transducers, signal conditioning state-of-the-art studies conducted for INT-21 launch vehicle, and the study performed by the British aircraft corporation for the shuttle vehicle have made it possible to choose the type of transducers best qualified for all types of measurement requirements without a trade study or transducer development program.

This study has not addressed itself to a detailed measurement list, but to the type of measurement requirements and to the type of sensors and signal conditioning equipment that would be utilized for the ESS instrumentation system. Table 2-37 is a listing of the measurement types and hardware.

Signal Conditioning. All ESS analog measurements will require some form of signal conditioning. The signal conditioning equipment can be mounted either with the sensor or in temperature-controlled areas.

Due to the high-level input voltage requirements of the communication systems ACT unit, digital signal conditioning is to be adopted. Compared to

Table 2-36. ESS Measurement Requirements

System	Acoustic	Temperature	Pressure	Vibration	Flow	Position	Discrete	Liquid Level	Voltage/Current Power/Frequency	Speed	Strain	Total
Propellant feed		16	17				160					193
Attitude propulsion		56	74		4	6	318	2		4		464
Pressurization		6	12				88					106
Propellant management		50					193	10	6			259
Insulation		20	4	6								30
Thermal control		24	4		2							30
Instrumentation		10							16			26
Aerodynamics	8	15	6	40							20	89
Communication		2					70		12			84
Engine actuation		8	20			8	26	4				66
Data control management		12							12			24
GN&C		6		9					508			563
Total	8	225	137	55	6	14	855	16	554	4	20	1934

Note Integration of GN&C, the shuttle APS, and redundancy management requirements has provided a large number of measurements; however, the data bus approach has sufficient capacity to handle 2440 measurements.

Table 2-37. ESS Transducer and Signal Conditioning Requirements

MEASUREMENT TYPE	TRANSDUCER TYPE	SIGNAL CONDITIONING TYPE
ACCELERATION	FORCE MASS	DIRECT
VIBRATION	PIEZOELECTRIC	CHARGE AMPLIFIER
TEMPERATURE	THERMOCOUPLE	REFERENCE JUNCTION - DIGITAL RATIOMETER - D/A CONVT
TEMPERATURE	PLATINUM RESISTANCE	BRIDGE - DIGITAL RATIOMETER - D/A CONVERTER
PRESSURE & ΔP	STRAIN GAUGE	DIGITAL RATIOMETER - D/A CONVERTER
SPEED	MAGNETIC	FREQUENCY TO DIGITAL CONVERTER - D/A CONVERTER
FLOW	IMPELLER - MAGNETIC	FREQUENCY TO DIGITAL CONVERTER - D/A CONVERTER
POSITION	POTENTIOMETER	DIRECT
POSITION	DIFFERENTIAL XFORMER	DIRECT
VOLTAGE/CURRENT	-----	DIRECT OR OFFSET FOR HIGH RESOLUTION

the conventional method of utilizing an analog dc amplifier, miniature-sized transducer analog-to-digital converters signal conditions are selected. This method has many advantages, such as higher precision and accuracy, higher reliability, less electrical noise in transmission, less weight, less cost, and less power consumption.

The transducer analog-to-digital conditioner will accept and change low-level analog signals from transducers into ten-bit output signals, without the need for a highly stable power supply. It will contain an internal electronic switch to provide for shunt calibrations and another for pulsed operation of the transducer bridge which causes it to operate as a radiometer. Figure 2-182 shows a typical measurement block diagram utilizing digital signal conditioning.

Measurement Remote Calibration. To assure confidence in the measurement of a parameter, the constituent parts of the measurement must be checked for accuracy and for possible fault operation.

Two calibration systems are proposed: one will be part of the ESS vehicle instrumentation system for calibration and operation verification of selected measurements such as red-line and critical systems performance; the second calibration system proposed will be a removable plug-in unit to be utilized to checkout and verify all instrumentation measurements before their use in obtaining vehicle subsystems data.

Both calibration systems will be designed to verify the measurements accuracy by verification of: (1) sensor, (2) signal conditioning electronics, and (3) the output of the signal conditioning. In addition, the above three point systems verification will assist in locating or isolating the cause of a faulty measurement. The data and control management computer will command the communications ACT unit that will activate the self-check or calibration verification routine of both vehicle or ground checkout calibration systems.

Vibration Measurements. Digital signal conditioning is not superior for all measurements and applications. Figure 2-183 represents the proposed vibration measurement utilizing a piezoelectric pickup and an analog type charge amplifier for signal conditioning. The ESS vibration measurements do not interface with the vehicle data bus but are routed to the communication systems CBW FM telemeter system. Utilizing a vibration multiplexer, a maximum of 100 vibration measurements can be transmitted. The vibration system can be removed from the ESS vehicle, if desired, without major instrumentation or communication systems redesign.

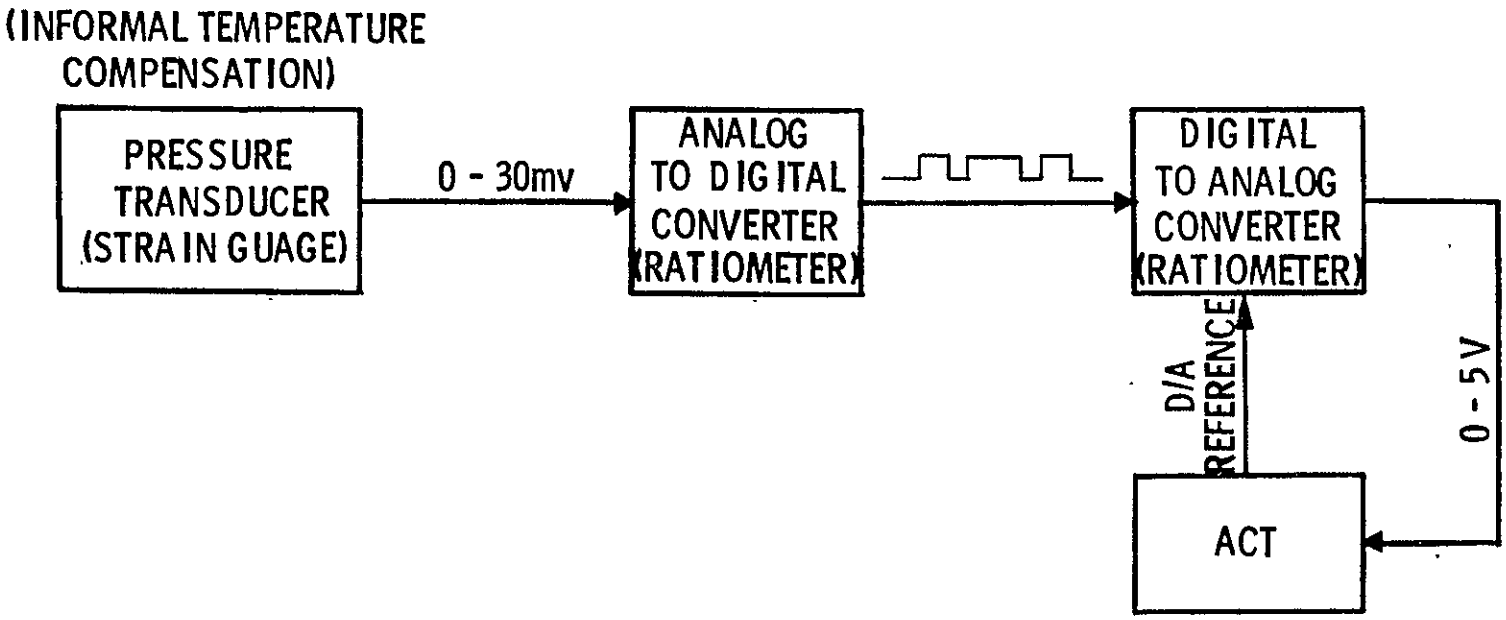


Figure 2-182. Typical Digital Signal Conditioned Measurement

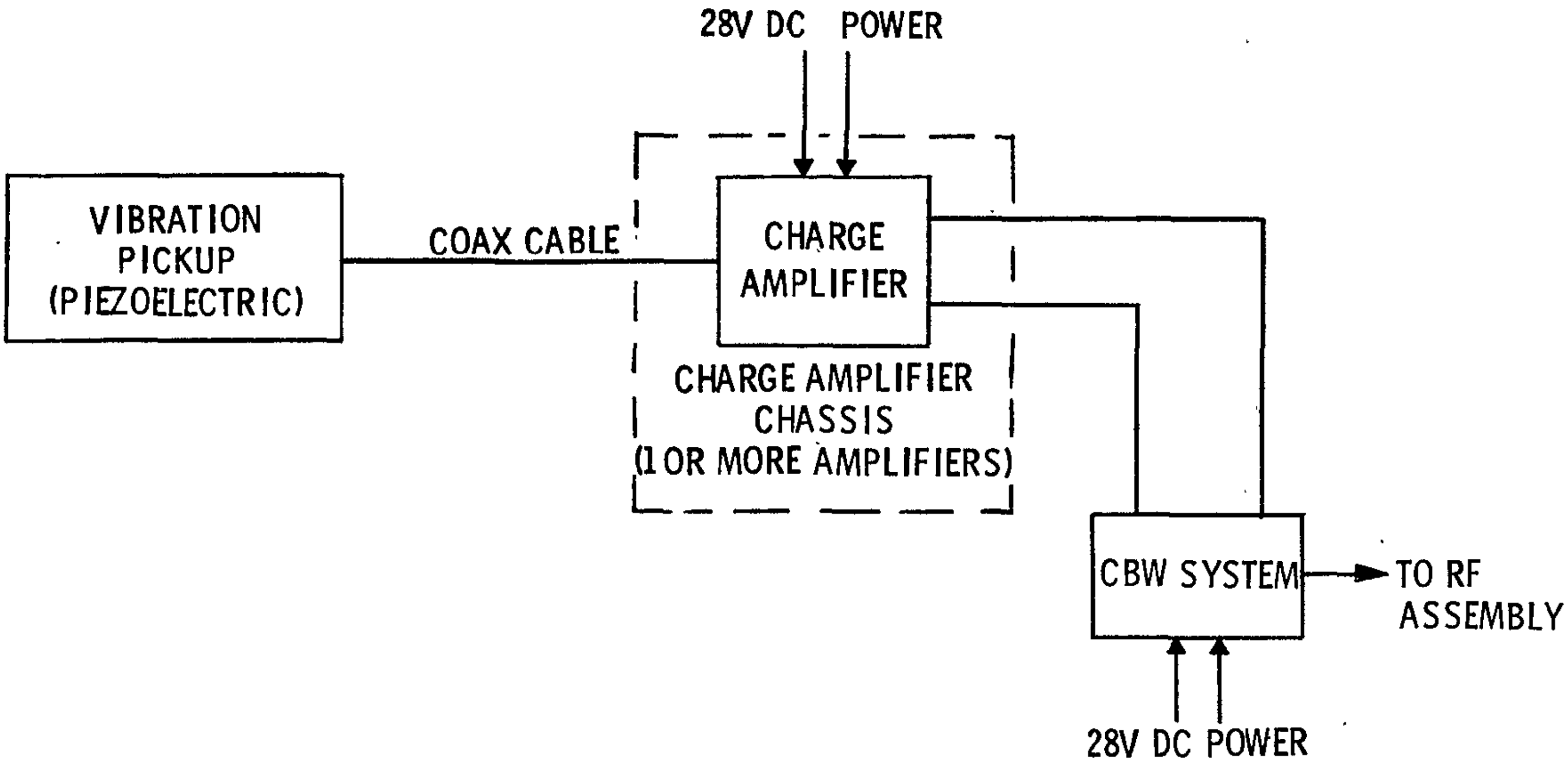


Figure 2-183. Typical Vibration Measurement

2.3.7 Avionics Equipment Installation and Environmental Control

The objective of this section is to describe the preliminary installation design of avionics equipment on the ESS vehicle. This equipment must be installed in a manner that will assure its designed operating performance during all phases of the 24-hour ESS mission. This task includes: the preliminary design of equipment containers, equipment thermal and vibration isolation, passive thermal insulation, and active environmental control. It will also cover the installation of equipment in the containers and the installation of the containers on the ESS.

The study approach is as follows:

1. Define equipment characteristics by weight, volume, installation and environmental operational limits.

2. Establish predicted environment imposed upon the astrionic equipment during the total 24-hour mission.

3. Define optimum astrionic equipment areas on the ESS vehicle which will assure that the equipment will operate within its designed envelope to fulfill the ESS mission objectives.

4. Establish the preliminary design of the avionics equipment containers, equipment installation within the container, container installation on the stage, and installation of special electronic equipment as required to support the 24-hour mission objectives.

5. Determine the thermal control requirements and associated hardware.

Requirements

To establish the installation design of avionics equipment on the ESS vehicle requires a knowledge of the thermal and dynamic environment on the stage during the 24-hour missions. The thermal and dynamics characteristics of the equipment to be installed must also be defined. The design requirements were established using the following ground rules and assumptions:

1. Space station payload.

2. Maximum ESS lifetime is 24 hours.

3. The vehicle attitude is Y-POP (Y axis perpendicular to the orbital plane) nose forward before turn-around.

4. Turnaround will occur just before deorbit retro burn.

5. The vehicle orbits are 100 and 270 nautical miles at an inclination of 55 degrees to the equator.

6. Maximum use of S-II equipment container design.

7. Maximum use of information gained from previous S-II derivative studies.

8. The aft skirt area will be closed out.

9. High value, aft skirt mounted, avionics equipment will be designed for recovery.

Thermal Environment. The thermal environment considerations are those associated with ground operations, boost, and on-orbit operations.

1. The ground temperatures will be similar to those on the present S-II stage in the mated and tanked configuration; therefore, a dry nitrogen purge system will be required to maintain thermal control and to deactivate inert avionics equipment during ground operations before launch.

2. Temperatures on the stage during the boost phase are shown in Figures 2-47 and 2-51. A review of the figures indicates that the external surface of the thrust structure is adequate for mounting avionics equipment containers; however, locating containers in the forward and aft skirt in the Position IV quadrant should be avoided due to high aerodynamic heating.

3. The thermal environmental difference between the shuttle orbiter and the ESS is that the shuttle orbiter is subjected to the high heat loads imposed on the vehicle during reentry. These high heat loads present a pacing design requirement for the orbiter avionics equipment thermal control system. This requirement is not applicable to ESS. The orbital thermal environment is the most critical from an avionics equipment installation standpoint on the ESS. This is due to the fact that the ESS will be exposed to the orbital environment for approximately 23 1/2 hours of the total 24-hour mission.

Dynamic Environment. The dynamic environment to which the ESS will be subjected consists of mechanical vibration and acoustics; this is a result of shuttle booster launch, max Q, and critical aerodynamic attitudes. The only predicted dynamic environment available for this publication is that associated with lift-off which may not be the most severe; however, only the lift-off environment will be considered at this time for the dynamic design requirements.

1. The predicted random and sine vibration levels were obtained using the existing S-II structural configuration. The vibration levels on the lower aft skirt are anticipated to be high enough to warrant vibration isolation of equipment mounted in this area, but equipment mounted on the thrust cone may not require vibration isolation. Equipment mounted in the forward skirt will require vibration isolation.

2. The predicted launch acoustic level in the forward skirt area is 161.5 db overall. This level will present a design problem in installing antennas, and acoustic isolation may be required for other equipment mounted in this area. The predicted acoustic level in the aft skirt is 162.5 db overall which will also present a design problem when installing equipment in this area. The thrust cone is the most desirable area for installation of avionics equipment because the acoustic level will be lower than that of the forward or aft skirt.

Further study is required to determine more exact sound pressure levels and acoustic isolation design requirements.

Avionics Equipment Environmental Characteristics. The avionics equipment was reviewed to establish the environmental and special installation requirements.

Table 2-38 lists all identified electronic components comprising the ESS avionics systems. The components are identified by subsystem where applicable. The table identifies the selected electronic components required as to physical characteristics, design, and operational requirements together with special instructions for installation and/or environmental control. The operational modes as initially defined covered boost, phasing, rendezvous, separation, and deorbit. The percentage figures under the operational modes show the estimated time the component will be in operation. As noted, most of the astrionics systems are required throughout the 24-hour ESS mission.

An assessment of the thermal requirements of the identified electronic equipment identifies the air-cooled IMU and the batteries as major design installation problem areas. Vendor information indicates that the IMU will operate in a vacuum with no external cooling air as long as the upper-case temperature remains below 104 F. This would indicate that by installing the IMU in a cold space environment and maintaining the unit above its lowest

Table 2-38. ESS Avionics Equipment

ITEM	DESCRIPTION	NO. REQ'D	SIZE-INCHES W X L X H	WEIGHT (LBS)	POWER (WATTS)	THERMAL REQUIREMENTS DESIGN	THERMAL REQUIREMENTS OPERATIONAL	OPERATIONAL MODES 1	2	3	4	5	6	CONT	EQ. LOC.	SPECIAL INSTALLATION AND ENVIRONMENTAL REQUIREMENTS	REMARKS
	GUIDANCE NAVIGATION & CONTROL														T.C.		
1.	IMU PLATFORM LRU #1	3	7.5X8.75X12	25(75)		Air	40°F to +104°F	100%	100%	100%	100%	100%	R	D/E	T.C.	HT OUT 340 BTU/HR AV.	AIR COOLED
2.	IMU POWER SUPPLY LRU #2	3	6X10X10	22(66)	160(480)	Cooled	(W/O Cooling Air)	100%	100%	100%	100%	100%	R	D/E	T.C.	ACT UNIT IN P.S.	INT. TEMP. CONTROL
3.	RATE GYROS	9	6.25X3.5X2.65	2.2(19.8)	4(36)	TBD	TBD	100%	100%	100%	100%	100%	100%	G	T.C.		
4.	T.V.C. GIMBAL DRIVERS	1	4X5X6	6.5(6.5)	18(18)	TBD	TBD	100%	0	0	0	0	0	A	T.C.		
5.	ATT/TRANS JET DRIVER (2 JETS)	2	4.7X6X6	8.6(17.2)	7 (14)	TBD	TBD	0	10%	5%	0	0	10%	A	T.C.		(OMS)
6.	ATT/TRANS JET DRIVER (4 JETS)	4	4.25X5.2X5.2	14(56)	11(44)	TBD	TBD	0	0	10%	5%	5%	0	A	T.C.		(ACPS)
	DATA CONTROL & MANAGEMENT														T.C.		
7.	CENTRAL PROCESSING UNIT	4	9X14X8	34(136)	136(544)	-65° to 160°F	-65° to 160°F	100%	100%	100%	100%	100%	R	B/C	T.C.		
8.	MAIN STORAGE UNIT (3-8K MSU)	4	15.8X18X8	71(284)	165(660)	-65° to 160°F	-65° to 160°F	100%	100%	100%	100%	100%	R	B/C	T.C.		
9.	ACT UNITS	60	4X4X2	2(120)	3(180)	-65° to 160°F	-65° to 210°F	100%	100%	100%	100%	100%	0	ALL	T.C.		
	RANGE SAFETY														F.S.		
10.	ANTENNA, RANGE SAFETY	4	13½X7½X7	10(40)	NA	-125°F to +400°F	-125°F to +400°F	100%	0	0	0	0	0	H	F.S.	ANTENNAS 90° APART	45° OFF VERT.AXIS
11.	DIRECTIONAL COUPLER RF	1	4X3.5X2	1 (1)	NA	-65°F to +160°F	-65°F to + 160°F	100%	0	0	0	0	0	H	F.S.		
12.	POWER DIVIDER	2	3X13.5X2	1 (2)		-65°F to+160°F	-65°F to + 160°F	100%	0	0	0	0	0	H	F.S.		
13.	HYBRID JUNCTION	1	5X16X2.5	4 (4)		-65°F to+160°F	-65°F to +160°F	100%	0	0	0	0	0	H	F.S.		
14.	BANDPASS FILTER	2	2X9X1.5	1 (3)		-65°F to+160°F	-65°F to +160°F	100%	0	0	0	0	0	H	F.S.		
15.	COMMAND RECEIVER	2	5X5X4	8 (16)	4.5(9)	-85°F to +230°F	-85°F to +230°F	100%	0	0	0	0	0	H	F.S.		
16.	COMMAND DECODER	2	8X4.5X2.5	9.5(19)	4.5(9)	-65°F to +110°F	-65°F to +110°F	100%	0	0	0	0	0	H	F.S.		
	COMMUNICATIONS														F.S.		
17	MODULATION & SWITCHING UNIT	1	TBD	TBD	TBD	TBD	TBD	100%	100%	100%	100%	100%	100%		F.S.		
18.	UNIFIED 'S' BAND EQUIPMENT	1	21X6X9½	38(38)	25(25)	TBD	*0 to 130°F	100%	100%	100%	100%	100%	100%	I	F.S.		*APOLLO COLD PLATE TEMP 35 TO 118°F
19.	"C" BAND PWR. AMPLIFIER	1	5 3/4X22¼X6	32(32)	160(160)	TBD	0 to 130°F	100%	100%	100%	100%	100%	100%	I	F.S.		
20.	ANT. SW. UNIT	1	TBD			TBD	TBD	100%	100%	100%	100%	100%	100%	I	F.S.		
21.	UP DATA REGISTER	1	TBD	TBD	TBD	TBD	TBD	100%	100%	100%	100%	100%	100%	I	F.S.		
22.	ANTENNAS	4	6" DIA	1(4)		-125°F to 400°F	-125°F to 400°F	100%	100%	100%	100%	100%	100%		F.S.	ANTENNAS 90° APART	45° OFF VERT.AXIS
23.	CPW FM TELEMETER	1	TBD	TBD	TBD	TBD	TBD	100%	100%	100%	100%	100%	100%	I	F.S.		
24.	VIBRATION MULTIPLEXER	1	TBD	TBD	TBD	TBD	TBD	100%	100%	100%	100%	100%	100%	I	F.S.		
	INSTRUMENTATION																
25.	SIGNAL CONDITIONERS	TBD	TBD	TBD	TBD	TBD	TBD	100%	100%	100%	100%	100%	100%	ALL	A.S/F.S		
	PROPELLANT MANAGEMENT SYSTEMS																
26.	AFT P.M. ELECT. CONTROL	1	4X5X6	2(2)	50(50)	-65°F to 135°F	-65°F to 135°F	100%	0	0	0	0	0	A	A.S/F.S		
27.	FWD P.M. ELECT CONTROL	1	4X5X6	2(2)	50(50)	-65°F to 135°F	-65°F to 135°F	100%	0	0	0	0	0	J	A.S/F.S		
28.	PCS (FLIGHT) FORWARD	3	1.015X1.015X1.38	1/8(.4)	6.5/.171(13)	-85°F to 302°F	-65°F to 248°F	10%	0	0	0	0	0	J	F.S.		
29.	PCS (FLIGHT) AFT	3	1.015X1.015X1.38	1/8(.4)	6.5/.171(13)	-85°F to 302°F	-65°F to 248°F	10%	0	0	0	0	0	A	A.S.		
	PROPELLANT FEED SYSTEM																
30	PCS (FLIGHT) AFT	19	1.015X1.015X1.38	1/8(1.4)	6.5/.17(71.5)	-85°F to 302°F	-65°F to 248°F	10%		0	0	0	0	A	A.S.		
	PROPELLANT DISPERSION																
31	(SII 223 CONTAINER)	1	20.62X20.12X9.16	50	9	-65°F to 160°F	-65°F to 140°F	100%	0	0	0	0	0	K	F.S.		MOUNT BY R/C DECODER
	POWER DISTRIBUTION & CONTROL																
32	BATTERIES (150 AH)	8	17X9X11	65(520)	5(40)	0 to 125°F	60°F to-80°F	100%	100%	100%	100%	100%	100%	L-Z	A.S.		3(56V BUSSES)
33	INVERTERS/TRANSFORMERS	3	10X24X8	45(135)		-65°F to 135°F	-40°F to 135°F	100%	100%	100%	100%	100%	100%	L-Z	A.S.		POWER AC BUSSES
34	BATTERIES (500 AH)	9	19X17X14	225(2025)	50(450)	0 to 125°F	60°F to 80°F	100%	100%	100%	100%	100%	100%	L-Z	A.S.		3(28V BUSSES)
	SAFING/SEPARATION																
35.	AFT SKIRT CONTAINER	1	TBD	TBD	TBD	-65 to 160°F	-65°F to 160°F	100%	100%	0	0	0	0	J	A.S.		
36.	FORWARD SKIRT CONTAINER	1	TBD	TBD	TBD	-65 to 160°F	-65°F to 160°F	100%	100%	0	0	0	0	F	F.S.		
	DEORBIT																
37.	DEORBIT ELECTRONICS	1	24X18X16	105(105)	175(175)	-65°F to 160°F	-65°F to 160°F	0	0	0	0	10%	100%	G	T.C.		

OPERATIONAL MODES: 1. BOOST TO 100 NM
2. 100 NM/OHMS ASSENT
3. 270 NM ORBIT
4. SEPARATION, PAYLOAD
5. RENDEZVOUS (ESS PASSIVE)
6. DEORBIT (R-EQUIPMENT RECOVERED)

EQUIPMENT LOCATION - EQ.LOC
FORWARD SKIRT - F.S.
AFT SKIRT - A.S.
THRUST CONE - T.C.

design temperature by optimum use of high-performance insulation and heaters, the ESS 24-hour mission objectives will be supported. Ground operation will be controlled by the ground purge system.

The batteries will be mounted in insulated containers and installed in a cold space environment. Thermostatically controlled heaters will be installed in the containers as required to maintain the batteries within their operational temperature limits.

The remaining identified electronic components will be maintained within a temperature range of -40 F to +140 F. All avionics equipment requiring environmental protection recovery will be mounted within electronic containers. Container design will be such that thermostatically controlled heaters will maintain the container temperature above -40 F, and space cooling will assure that +140 F is not exceeded.

In reviewing the special installation requirements, it is noted that no major problem areas were identified; however, in assessing each component requirement, the antennas, IMU's, and rate gyros require hard mounting and specific orientation to perform within their design specifications. The antennas will be located around the forward skirt perimeter, 90 degrees apart as shown on drawing V7-975401, "Avionics Container Installation - ESS." The IMU's will be mounted on a common base in such a way that each IMU is within the specified alignment with the others and that the common base is installed on the thrust cone in such a way that the IMU axes are triaxially matched to the ESS axes. The rate gyros will also be located and aligned as required triaxially to the ESS axes.

The operational mode of each component indicates the percent of time, in each phase, that the component will be active to support the ESS mission. This time may be either on standby or in operational status. Noted items used only for the boost to orbit phase will be deactivated after accomplishing their missions.

Avionics Equipment Installation and Environmental Control Design

This section describes the preliminary design of avionics equipment installations to fulfill the requirements established in Section 2.3.7 under Requirements. The section is divided into three parts: equipment container design, installation of equipment in containers, and installation design of equipment containers on the stage.

Container Design. The equipment container design is essentially that of the present S-II modified to accommodate environment control requirements.

To protect the equipment in the orbital thermal environment, passive thermal control consisting of surface coatings and insulations must be considered in the container design.

The selection of surface coatings is dependent upon their usage applications. Equipment containers housing high heat producing equipment will have internal and external coatings which provide high infrared emittance to dissipate this high internal heat by radiating to the cool surfaces of the LH_2 forward dome and the aft thrust cone. Equipment containers housing low heat producing equipment, which are not exposed to direct solar radiation, will have the internal surfaces coated with highly reflective coating and the exterior surface coated with a low emittance coating for proper thermal control. Table 2-39 contains a listing of surface coatings showing solar absorptivity (α) and infrared emissivity (ϵ). The table shows that Dow Corning (DC92-007) thermal control white can be used for the internal and external surface coatings for containers housing high heat producing equipment located in the forward or aft areas. This coating was selected because of its proved application on other space programs, and it is currently controlled by NR specification MA0608-018. For forward and aft containers housing low heat producing equipment, the internal surfaces will be coated with Fuller aluminum silicone paint and the external surfaces will be covered with aluminum foil.

Container Equipment Installation Design. Equipment will be installed in the containers by system to minimize interconnecting wiring between containers. Table 2-40 shows the ESS avionics systems listed by equipment container. The table also identifies the general areas in which the equipment containers will be located on the stage.

Container Installation Design. The equipment containers will be installed on the stage in areas which present the most ideal thermal and dynamic environment. Drawing V7-975401, included as Figure 2-184 shows the containers installed on the ESS. Due to high temperatures caused by aerodynamic heating, no equipment will be located in the areas adjacent to stage position IV. To eliminate the need for equipment cooling, all equipment containers will be located in areas between stage positions II and III, and between stage positions III and IV. These areas present the coldest environment during on-orbit operation and should provide adequate cooling for the high heat producing equipment during the 24-hour mission. No equipment containers will be installed on the lower aft skirt due to the extreme dynamic environment in this area. Battery containers will be installed on the upper portion of the aft skirt in the area between the thrust cone and the LO_2 tank. The environment in this area should be acceptable for properly insulated containers hard-mounted to the basic structure. The guidance, navigation, and control equipment and the data control and

Table 2-39. Surface Coatings Considered for Thermal Control Applications

THERMAL CONTROL APPLICATIONS: EXTERNAL TO CONTAINER	THERMAL CONTROL APPLICATIONS: INTERNAL TO CONTAINER	SURFACE COATINGS	SOLAR ABSORPTIVITY (α)	INFRARED EMISSIVITY (ϵ)	COMMENTS
Use for low emittance requirements (Do not use when exposed to high solar heat loads	Use for low emittance or low absorptance requirements	Aluminum Sheet	.16 - .2	.06 - .10	Use when $<$ 750°F
		Aluminum Foil (mystik 7402)	.12	.04	
		Beryllium	.11 - .50	.10	
		Inconel Foil	.32 - .38	.11 - .13	
		Iconel X Foil	.18 - .66	.15	
Use for low solar absorptance & high emittance requirements	Use for high emittance or high absorptance requirements	White Lacquer (Sherman Wms)	.28	.86	α = .5, when $\geqq$ 70 °F
		Fuller White Silicone Paint	.25	.9	α = .6, when $\geqq$ 80 °F α = .6 for ultraviolet radiation
		Optical Solar Reflector Alum. coated	.1	.744 - .807	
		Thermatrol White Silicone Paint	.16	.95	α = .32 for ultraviolet radiation unaffected
		S13 (ZnO/Silicone)	.19 - .2	.79 - .87	
		S13G(ZnO/Silicone)	.16 - .2	.84 - .86	
		Z93 (ZnO/K_2SiO_3)	.14	.91 - .97	α = .3 when deteriorated requires alum. surface as base
		LP10A ($ZnSiO_4/K_2SiO_3$)	.1 - .14	.87 - .91	
		Hughes ($AlSiO_4K_2SiO_3$)	.13 - .14	.88 - .91	
		Dow Corning DC92-007	.192	.832	
Use for high emittance requirements (Don't use When exposed to high solar heat loads	Use for high emittance or high absorptance requirements	Black Lacquer (Sherwin Wms)	.93	.88	Use when $<$ 450°F
		Fuller Black Silicone Paint	.89	.88	Use when $<$ 1070°F
		Rokide C (Chrolic Acid)	.82 - .9	.85 - .86	
		Platnum Black	.85 - .94	.8 - .9	
Use for low emittance requirements (Don't use When exposed to high solar heat loads	Use for low emittance or low absorptance requirements	Fuller Alum. Silicone Paint (172-A-1)	.25	.28	
		Fuller Alum. Silicone Paint (171-A-152)	.22	.24	
		Alum. Acrylic Paint	.38 - .52	.36 - .58	Use when $<$ 650°F

Table 2-40. Avionics Systems Installation

SYSTEM	THRUST CONE CONTAINERS							FORWARD SKIRT CONTAINERS				UPPER AFT SKIRT CONTAINERS		
	A	B	C	D	E	F	G	H	I	J	K	L thru Z	AA	BB
GUIDANCE NAV. & CONTROL				X	X									
*DATA CONTROL MANAGEMENT		X	X											
RANGE SAFETY								X						
COMMUNICATIONS									X					
*INSTRUMENTATION								X						
PROPELLANT MANAGEMENT	X									X				
PROPELLANT FEED	X													
PROPELLANT DISPERSION											X			
POWER DISTRIBUTION & CONT.												X —— X	X	X
SAFING						X				X				
DE-ORBIT							X							

NOTE: * 1 - ACT UNITS AND INSTRUMENTATION WILL BE IN CONTAINERS LOCATED IN ALL AREAS OF THE ESS.

2 - CONTAINER "K" EQUIPMENT IS SIMILAR TO THE 223 CONTAINER EQUIPMENT ON THE S-II STAGE.

3 - X = RECOVERABLE CONTAINERS

management equipment will be housed in recoverable containers mounted on the thrust cone. All communications equipment including antennas will be mounted in the forward skirt area. (See Figure 2-184 for location.)

Avionics Equipment Recovery. The guidance, navigation, and control equipment and data control management equipment housed in containers B, C, D, and E will be recovered from the ESS by retrieving the complete containers. The containers will be separated from the stage by severing the attach brackets with exploding bridge wire (EBW) initiated linear shaped charges (LSC). Figure 2-186 shows the explosive train used to separate each container. Two EBW firing units are used to provide redundant signal paths for initiation of the LSC assemblies. The storage capacitors in the EBW firing units are charged to 2300 volts by the application of 28-dc power to the charging circuits. A trigger signal discharges the storage capacitors, releasing a high-energy pulse that explodes the bridgewire in the EBW detonator. The resulting detonation propagates through the confined detonating fuze (CDF) manifolds, and CDF assemblies, and initiates each end of LSC assemblies. Detonation of the LSC assembly severs the webs of the container attaching I beams, thus separating the containers from the stage. The design and installation of the charge will be such that no fragmentation or shock loads will occur. The crossover CDF assembly between CDF manifolds (see Figure 2-185) provides added systems reliability.

2.3.8 Recovery and Deorbit Subsystem

The baseline ESS system provides for the recovery of the main propulsion engines and major components of the avionics subsystem before deorbiting the vehicle. In order to recover this high-cost equipment, it must be separated from the vehicle in preparation for retrieval by the shuttle orbiter. The two main engines and selected avionics containers will have separation devices incorporated to facilitate recovery. The removal of the DCM components necessitates the incorporation of minimal control equipment to provide the capability for vehicle control during deorbit sequencing. This section contains the requirements, trade studies, and technical descriptions of the equipment required to provide the capability for equipment recovery and vehicle deorbit.

Recovery/Deorbit Ground Rules, Assumptions, and Requirements

The prime driver in establishing the requirements and techniques to accomplish the objective of equipment recovery is the need to maintain the capability to deorbit the vehicle. This section contains the ground rules, assumptions, and requirements considered in the development of this subsystem.

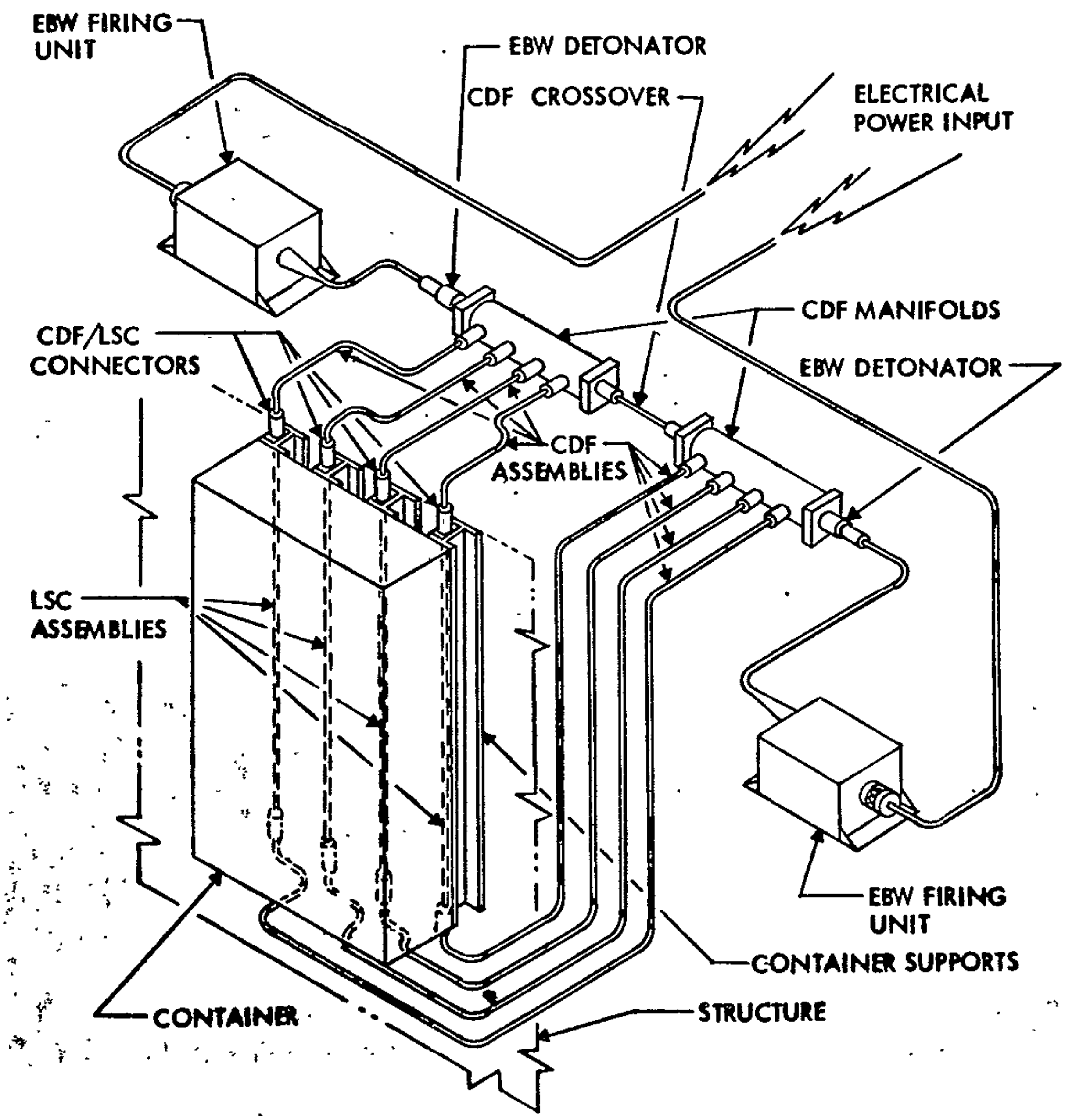


Figure 2-185. Avionics Container Separation (Ordnance Train)

Major Ground Rules and Assumptions. The major ground rules and assumptions utilized in the recovery study are as follows:

1. ESS must be capable of deorbiting.
2. Equipment required for deorbit will be FO/FS.
3. The initiation of the deorbit burn sequence will be controlled by the orbiter or ground via airlink.
4. The OMS engines will be used to provide the deorbit delta V.
5. The ACPS will be utilized to provide attitude control.
6. Deorbit will be accomplished within 24 hours from launch.
7. The shuttle orbiter will provide the attitude alignment required for ESS deorbit.
8. All electrical equipment recovered will be safed (dead faced) before removal.
9. All equipment to be recovered will be mounted in the aft area of the vehicle.
10. Equipment separation will not endanger the orbiter vehicle.

Requirements. The recovery and deorbit subsystem requirements are as follows:

1. Recovery. Due to the extremely high cost of the ESS main engine, guidance, navigation, and DCM equipment, it is economically advantageous to recover the items from orbit for reuse. Therefore, the ESS must incorporate separation controls consistent with the ground rules and assumptions specified above.

2. Deorbit. The deorbit equipment in conjunction with appropriate ESS subsystems and shuttle orbiter will provide the limited vehicle control required to permit recovery of the primary DCM and GN&C equipment. This equipment will maintain the ESS in an attitude hold mode for a period of up to (TBD) hours after attitude initiation by the primary control system or the shuttle orbiter vehicle. During this attitude hold period, the ESS attitude will be maintained within ±5 degrees of initial alignment.

The deorbit subsystem will also provide the means of controlling the subsystem sequencing during the equipment recovery and deorbit phases. (See Figure 2-186 for the block diagram of the deorbit package.)

Recovery System Description

Equipment recovery will be initiated by separation of the main engines and applicable avionics containers from the ESS vehicle structure. This separation will be accomplished by electrical detonation of ordnance devices to selectively sever the mounting brackets and release the mechanical fasteners. The ordnance selected is nonfragmenting and will not compromise the safety of a docked orbiter and crew.

The avionics containers are separated by linear-shaped charges (LSC) which cut the I-beam brackets attaching the containers to the primary structure. These LSC's are detonated by dual detonators and exploding bridge wire (EBW) assemblies. The associated electrical control circuits to each EBW unit are composed of two alternate paths, which make the control circuitry FO/FO. (Refer to Figure 2-187.)

Separation of the main engines is achieved by actuating separation nuts. Each engine mechanical interface with the vehicle is attached with bolts and separation nuts which are actuated by gas-generating power cartridges (two per nut) which are initiated by a 5-ampere, 3-millisecond pulse. Each engine has 130 of these nuts; therefore, a total of 520 power cartridges must be fired to separate the mechanical connections of two engines. A typical circuit diagram for firing the separation nut power cartridges is shown in Figure 2-188. The electrical power and control cables that interface with the main engines and avionics containers to be removed must also be separated. This will be effected by umbilical-type connectors which disengage as the engines and containers move away after separation of their fasteners.

Performance Analysis

After separation from the payload, the ESS will assume an attitude for deorbit and apply power to its recovery and deorbit electronics package. The separation nuts holding the main engines will then be detonated, the engines will move to the limit allowed by two remaining restraining bolts, and a positive separation signal will be telemetered to the orbiter. The orbiter will then dock with the ESS, command final separation of the engines through the uplink system, and retrieve the engines. After engine recovery is complete, the avionics containers will be separated by uplink command and retrieved in the same manner as the engines. As presently conceived, equipment recovery will be accomplished by mechanical manipulator arms controlled from the orbiter vehicle.

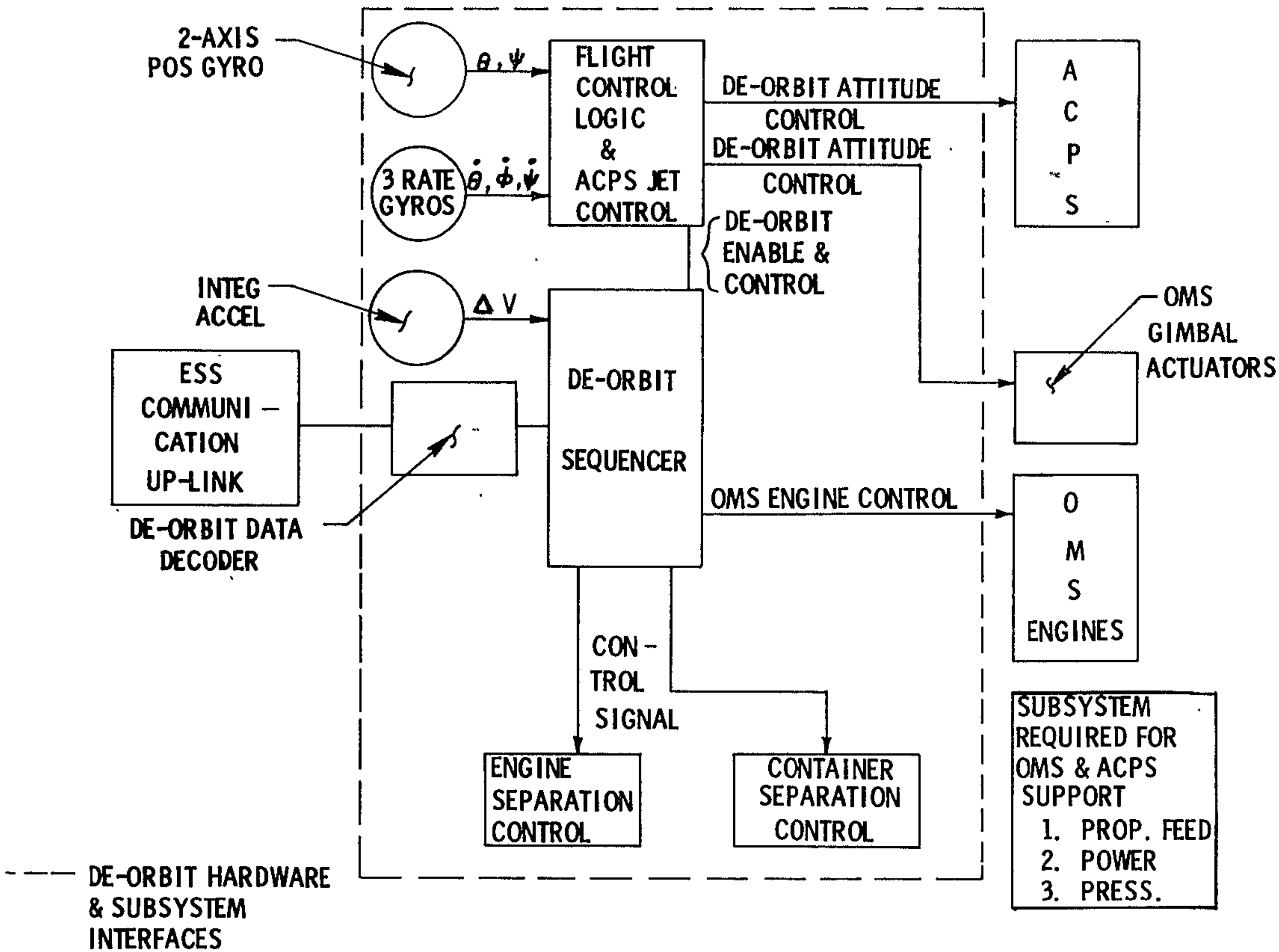
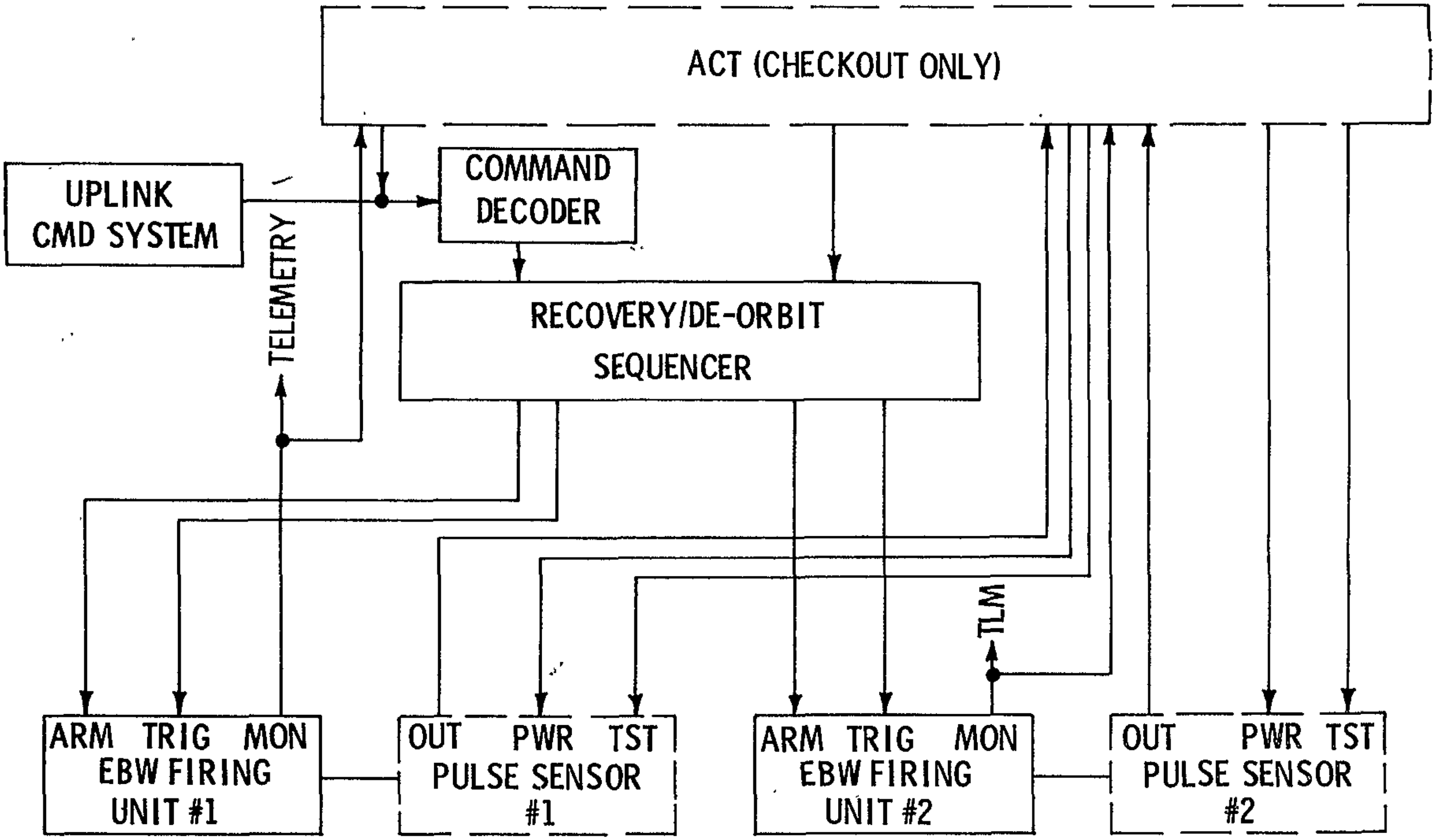


Figure 2-186. Deorbit Package Block Diagram



NOTE: PULSE SENSORS ARE FOR GROUND CHECKOUT ONLY. WHEN CONFIGURED FOR FLIGHT, THE EBWS ARE CONNECTED TO DETONATORS

Figure 2-187. Typical EBW Separation Control Block Diagram

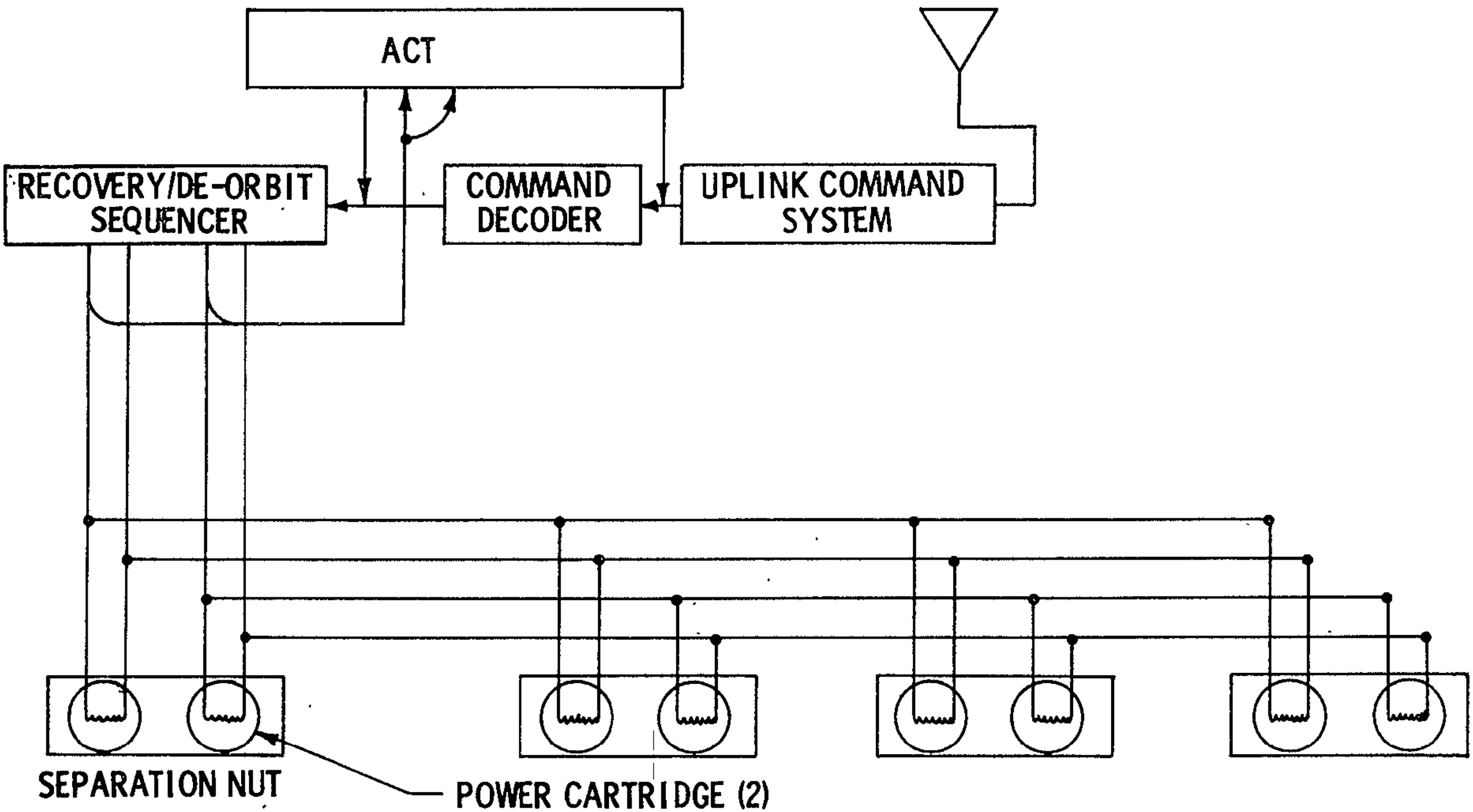


Figure 2-188. Typical Control Circuit for Main Engine Recovery Separation

Main Engine Recovery. Separation of the main engines for recovery requires an uplink command to initiate separation nut release. The decoded command initiates a timed sequence in the recovery/deorbit sequencer to fire the separation nut power cartridges. Consecutive firing will require approximately 300 milliseconds. The sequencer and power cartridge circuits are redundant for each separation nut and operate from separate power buses. The total engine separation system is FO/FS; the failure of two power cartridges in one nut, or their firing circuits, is a safe condition even though the engine cannot be recovered.

The power cartridges used to initiate separation are gas-generating devices. They are designed to provide high-pressure combustion energy to actuate the separation nuts, and they have been qualified and used in previous space applications. These cartridges require 4.5 to 5 amperes for approximately 3 milliseconds to achieve a 99.9 percent minimum firing probability with a 95-percent confidence level for single units. When used in pairs as they are in this design, the firing probability of each nut is even better.

Initial engine separation (before orbiter docking) will allow the engines to move back approximately one-quarter inch and actuate switches whose output will be telemetered to indicate engine primary separation. After orbiter docking with the ESS, uplink commands through the recovery/deorbit sequencer will effect final separation of each engine after the recovery manipulator arm is attached. The flexible heat shield attached to each engine will be separated during the primary separation.

Avionics Recovery. The guidance, navigation, and control equipment and data control management equipment housed in containers B, C, D, and E will be recovered from the ESS by retrieving the complete containers. The containers will be separated from the stage by severing the attach brackets with exploding bridge wire (EBW) initiated linear shaped charges (LSC). Two EBW firing units are used to provide redundant signal paths for initiation of the LSC assemblies. The voltage capacitors in the EBW firing units are charged to 2300 volts by the application of 28-dc power to the charging circuits. A trigger signal discharges the storage capacitors, releasing a high-energy pulse that explodes the bridgewire in the EBW detonator. The resulting detonation propagates through the confined detonating fuse (CDF) manifolds, CDF assemblies, and initiates each end of LSC assemblies. Detonation of the LSC assembly severs the webs of the container-attaching I-beams, thus separating the containers from the stage. The design and installation of the charge will be such that no fragmentation of shock loads will occur. The cross-over CDF assembly between CDF manifolds provides added systems reliability.

Separation type connectors will disconnect upon cutting of the mounting I-beams, due to a 1-inch downward motion induced by a spring mechanism.

The control circuits for arming and firing the two EBW units, for each container separation system, will be dual paths for each EBW. To separate each container, an uplink command must be received and decoded and a discrete command sent to the sequencer to arm/trigger the EBW's. Each container will require two uplink commands to accomplish the separation. (Refer to Figure 2-188 for a block diagram of the proposed circuits.)

During ground checkout of these circuits, pulse sensors will be installed in place of the flight ordnance. These pulse sensors verify the functional operation of the EBW's without actually detonating an ordnance train.

During ground checkout, the entire system, excluding the uplink portions, may be functionally checked by the DCM system through an ACT unit interface. The ACT unit will not be active during flight in order to eliminate the possibility of an inadvertent container separation.

During recovery operations, the orbiter manipulator arm will be attached to a container handle, shown in Figure 2-189, and the separation ordnance will then be fired. The spring-loaded connector interface will disengage the electrical connectors by pushing the container down, and the manipulator will retract and remove the container for stowing in the orbiter cargo bay. Each container will be separated individually after the manipulator is attached and ready to remove it.

Figure 2-99 shows a conceptual design for gaining access to the astrionics containers through the fiberglass honeycomb heat shield. A firm design for this closeout separation is to be determined.

Deorbit Subsystem Description

The deorbit subsystem includes the hardware required to accomplish ESS deorbit, after the primary DCM and guidance and navigation avionics equipment have been recovered. This subsystem is composed of four major elements: (1) uplink data decoder, (2) deorbit/recovery sequencer, (3) flight control logic, and (4) vehicle attitude sensors.

The deorbit equipment provides vehicle stabilization and deorbit/recovery sequencing once the primary DCM and GN&C equipment is deactivated in preparation for recovery. The up-link data decoder will provide the interface between the deorbit equipment and the ESS communication subsystem. This interface is required to meet the requirement of external control for the initiation of the deorbit sequence.

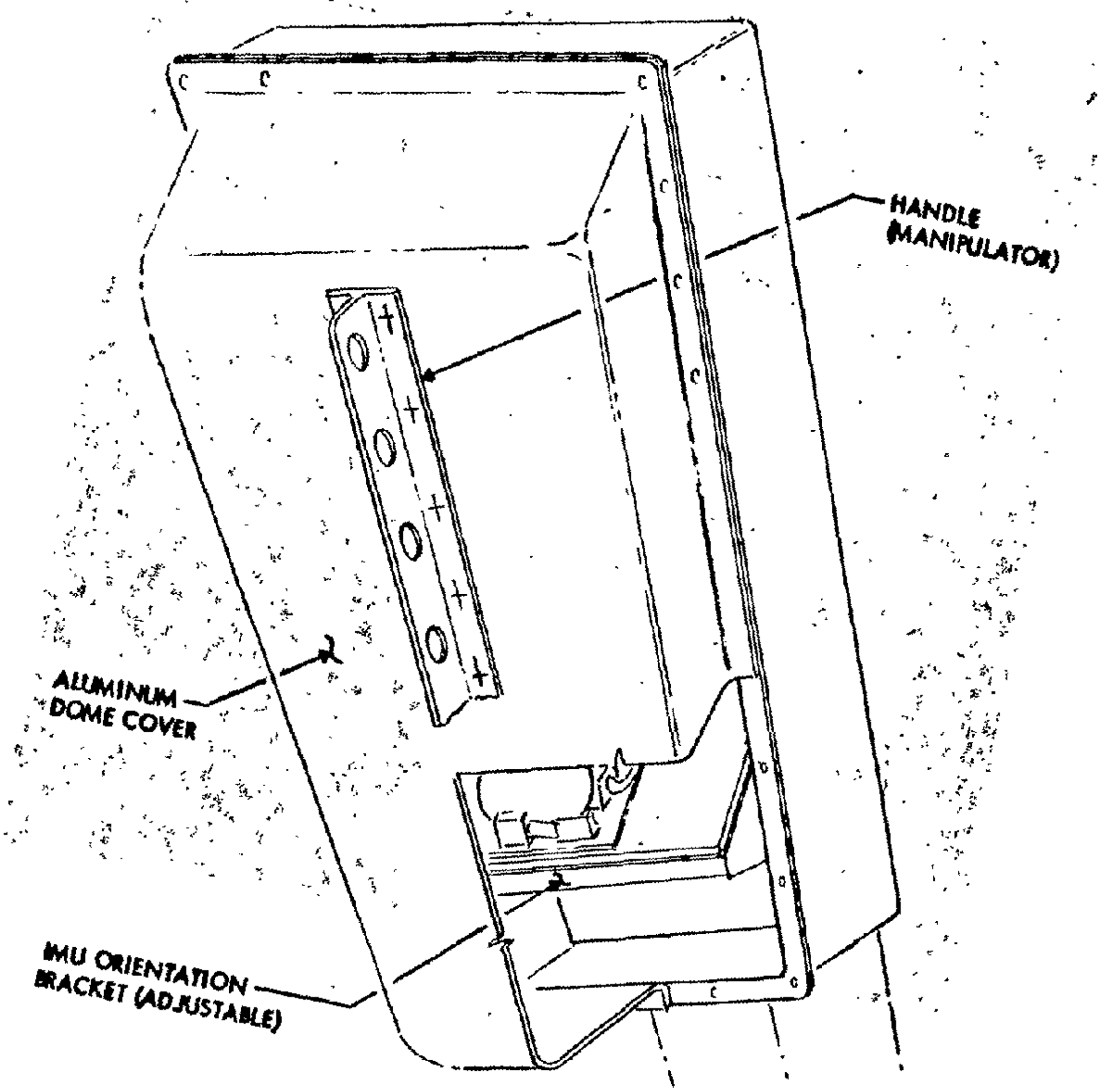


Figure 2-189. IMU Container

The deorbit/recovery sequencer will provide the preprogrammed sequence and the inhibit signals required to effect vehicle sequencing during the recovery and deorbit phase of the ESS mission.

The flight control logic element of the deorbit subsystem provides the capability to interpret the vehicle attitude information provided by the sensing element and to transform this information into control signals for the ACPS to maintain proper vehicle attitude and thrust vector control (TVC) during OMS engine burn.

The vehicle attitude-sensing element provides the information required to maintain proper vehicle alignment for deorbit. This element enables the ESS to maintain the vehicle alignment established by the primary system before deactivation or that established by the orbiter before separation.

Figure 2-190 is a schematic/block diagram of the deorbit equipment with interfaces required with the primary ESS subsystems. The deorbit equipment will be in a nonenergized state through the initial phases of the ESS mission. This condition will exist until after payload separation from the ESS. After payload separation, the primary DCM system will initiate action to cause the ESS to assure a vehicle attitude suitable for deorbit. This maneuver is performed before activation of the deorbit equipment as a precaution to preclude any perturbations during the deorbit equipment activation which might prevent safe deorbit of the vehicle. Once the preliminary deorbit attitude has been attained, the deorbit/recovery equipment will be activated. The equipment will be activated but will remain in an inhibited state until such time as the DCM system has completed all operation associated with safing the main engine (dead facing electrical connectors) and preliminary separation of the main engines.

At this point, an up-link command will be required to remove the primary DCM system as the controlling source and enable the deorbit equipment to provide attitude control.

Utilizing the input from the two-axis position gyro and the three-axis rate gyro, the flight control logic will generate appropriate control signals to fire the ACPS thrusters to maintain the attitude established by the primary guidance and control equipment before deactivation. The system will remain in this mode of operation until such time as it is directed to commence preparation of the DCM and GN&C equipment for recovery. Under control of an external command, the deorbit/recovery sequencer will initiate those commands required to safe the avionics equipment and initiate its separation.

At the moment of orbiter mating with the ESS, the deorbit attitude control system will be switched to a standby mode. While the orbiter is mated to the ESS, the deorbit equipment will perform only those operations which are commanded through the up-link by either the orbiter or ground control. This will include final separation of the main engines and avionic containers.

Before separation of the orbiter from the ESS, the orbiter will align the ESS with the final attitude orientation for deorbit. At this point, the deorbit attitude control will be enabled in such a way that it may retain any change in attitude at the time of separation of the orbiter from the ESS. However, the control of the ACPS system will not be enabled until such time as the orbiter is a safe distance from the ESS. This is to preclude any damage to the orbiter should the ESS malfunction at this time.

The ESS will maintain an attitude-hold mode of operation until such time as an external command is received to initiate deorbit. The sequence of events from the time of payload separation through deorbit is as follows:

1. Establish deorbit attitude
2. Activate deorbit equipment
3. Dead-face engine equipment
4. Initiate preliminary engine separation
5. Activate deorbit attitude control; deactivate primary control
6. Initiate deadfacing or recoverable avionics equipment
7. Mate with orbiter, deactivate attitude control
8. Perform main engine retrieval
9. Open ESS aft closeout panels
10. Initiate avionics container separation
11. Retrieve avionics containers
12. Align ESS to deorbit attitude (using orbiter NG&C)
13. Activate ESS attitude control system with ACPS engines inhibited
14. Separate orbiter

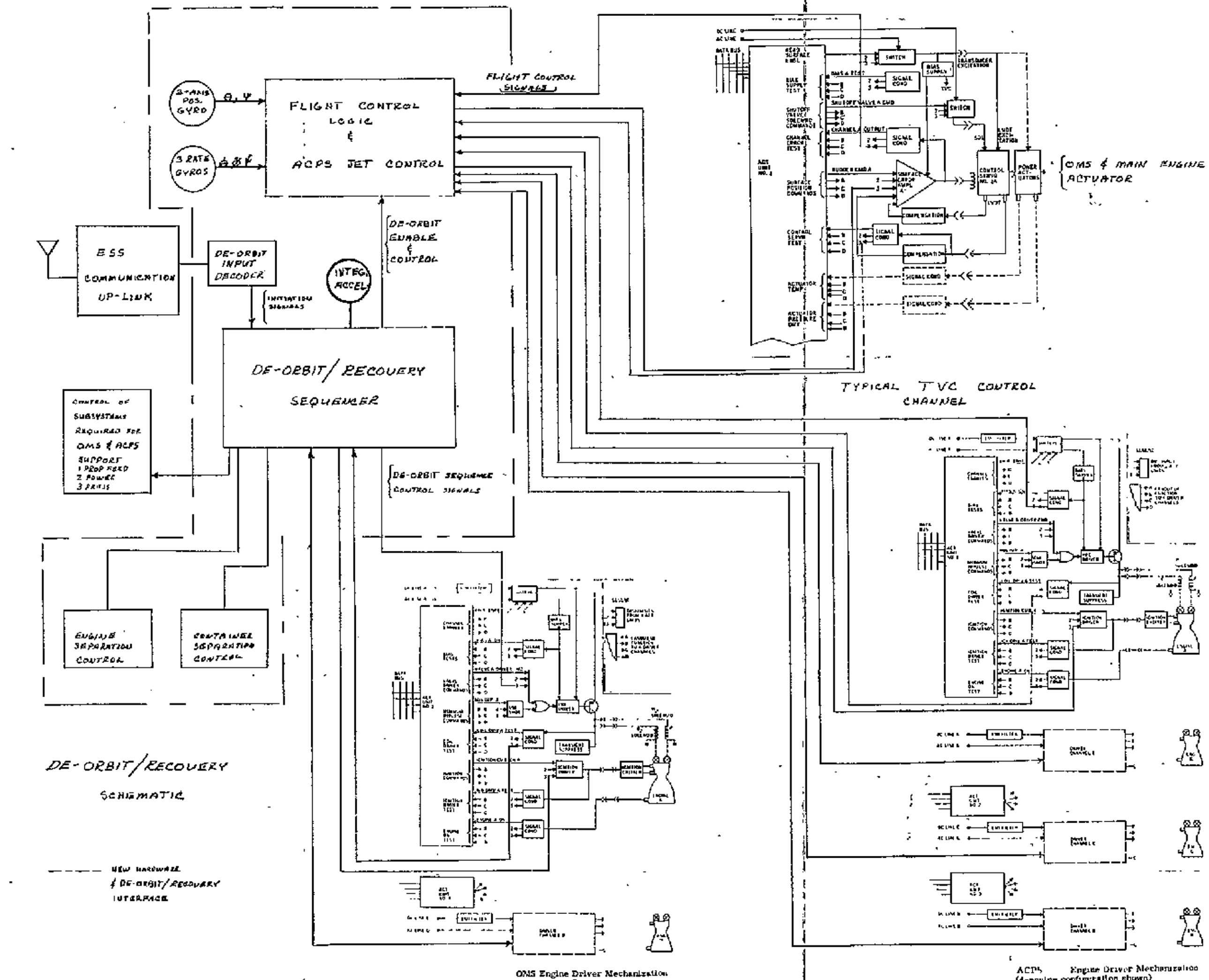


Figure 2-190. Deorbit/Recovery Schematic

15. Back away orbiter; activate ACPS

16. Initiate deorbit sequence

Control of the deorbit delta-V engine burn will be effected by two means. The primary control of the delta-V burn duration will be provided by the output of an integrating accelerometer which will initiate OMS engine cutoff after the proper delta V has been obtained. This system will be backed up by an output of the deorbit sequencer; this output will occur at a predetermined time after engine start. This time will be automatically adjusted to account for one engine out either at the start of deorbit or any time during the delta-V burn.

2.4 ESS VEHICLE INSTALLATION, ASSEMBLY, AND CHECKOUT

2.4.1 Major Vehicular Installation

The ESS will be fabricated as three major subassemblies: the forward skirt, which interfaces with the payload; the LH_2/LO_2 tank that comprises the major portion of the stage; and the aft skirt, in which the main and auxiliary engines are mounted. The various subsystems will be installed in these subassemblies prior to final joining wherever feasible; thus the installation effort will be allowed to proceed on each module simultaneously. This approach reduces the time required for any given installation because of the greater freedom of access, reduces the possibility of damage to the stage, and allows individual system testing to take place at a much earlier time so that substantial cost savings result.

The forward skirt installations will consist primarily of mounting various portions of the avionics and payload separation systems. The subsystems will be assembled in their environmental protection containers and tested in another area. The subsystem containers will be mounted on previously installed brackets, the stage wire harnesses will be installed, and a continuity and megger check will be made. The LH_2 tank propellant utilization, vent, pressurization, and instrumentation systems will be installed with the tank in the horizontal position after the postpneumostat cleaning operation, the LO_2 tank systems having been installed after the placement of the tank in the vertical position in the installations station. The aft skirt installations include the data control management and navigation, guidance, and flight control avionics containers. The auxiliary propulsion system cryogenic tanks and the propellant system lines will be installed and proof-tested and the engine actuation systems installed and checked out in the aft skirt subassembly prior to the final assembly joining in the Vertical Assembly Building.

2.4.2 Final Assembly and Installation

Final assembly of the ESS will be accomplished in the existing Vertical Assembly Building (VAB) at the NASA Seal Beach facility. The forward skirt, the LH_2/LO_2 tank, and the aft skirt will be mated and then moved to the spray foam station where the stage will be rotated in the vertical position while portions of the forward skirt, the tank areas (previously left bare for weld inspection), and portions of the aft skirt (requiring spray foam insulation) will be sprayed and machined smooth.

After the application and machining of insulation, the stage is moved to the installations station in the VAB where the engines, attitude control propulsion system (ACPS) thrusters, remaining avionics containers, and stage system fairings will be installed. The erosion barrier honeycomb panels will be installed at this point, along with the forward and aft skirt thermal protection covers not previously installed.

2.4.3 ESS Acceptance Checkout

Acceptance testing is designed to verify that hardware produced by production methods under quality control procedures and ready for delivery to its next assembly or usage point complies with specifications, is free from defects, and is capable of performing in conformance with stated contractual requirements. Acceptance tests begin with vendor tests at the subassembly, component, assembly, and subsystem level and continue through demonstration at the time of vehicle delivery (or turnover) and customer acceptance.

Whenever appropriate, acceptance tests at the component/subassembly level will include environmental testing. Environment test levels will be equivalent to mission requirements or a level sufficient to screen defects. The contractor will make recommendations concerning which items of hardware should be considered for environmental acceptance testing. The tests will be designed to detect manufacturing flaws, workmanship errors, and incipient failures which are not readily detectable by normal inspection.

Acceptance testing at the subsystem level (installed in the vehicle) will include a demonstration of alternate/redundant modes of operation—together with malfunction switching logic—by exercise of subroutines inherent to the data control and management system. Whenever possible, alternate/redundant path checkout capability by malfunction simulation will be an inherent subsystem checkout feature and will be accomplished without disturbing the flight configuration.

Upon completion of assembly and in-line manufacturing acceptance tests, such as leak checks and continuity and megger tests, each subsystem will undergo a functional acceptance test during post-manufacturing checkout (PMC). The PMC will culminate in an integrated noncryogenic simulated flight test designed to verify that all subsystem interfaces are correct and that no extraneous interactions exist.

2.4.4 Manufacturing Facilities

The facilities plan is to use existing government-owned facilities, contractor-owned facilities, and facilities available through other government contractors for detail and component fabrication. The subassembly and assembly activities will be accomplished at the NASA Seal Beach assembly facility, which was designed and built expressly for production of the Saturn S-II, of which the ESS is a derivative. The 33-foot diameter and 80-foot length of the S-II posed special transportation problems, leading to the choice of Seal Beach as having facilities nearby for shipping the complete stage by ship through government-controlled facilities. In addition, it is in the center of a manpower pool of skilled individuals and is near the sites of many subcontractors—facts resulting in reduced costs for transportation, coordination, and testing.

The recommended facility utilization plan embraces development, qualification, and acceptance testing at the NR Downey facility, NR Seal Beach facility, MSFC, KSC, Wyle Laboratories (Huntsville), and wind tunnels as needed and available at Langley Research Center, Ames Research Center, or at AEDC Von Karman gas dynamics facility; detail fabrication at the NR Downey facility, the NR Los Angeles facility, and at selected subcontractors; and subassembly and assembly at the NASA Seal Beach assembly facility. The NASA Seal Beach assembly facility contains adequate floor space to accommodate the ESS production operations and provides sufficient bridge crane capacity to handle the movement and transfer of subassemblies and assemblies.

2.5 ESS SYSTEM SUPPORT

2.5.1 Support Equipment

The objective of this study has been to determine new support equipment or existing support equipment to be modified to support the servicing, checkout, handling, and the launch of the ESS. (Saturn S-II support equipment not requiring modification for the ESS is not included in this report.)

New support equipment and modifications to existing equipment are required to accomplish servicing, checkout, and acceptance tests at Seal Beach; servicing, checkout, and static firing tests at KSC; and the servicing and checkout at the launch pad to support the ESS launch.

The report includes new and modified handling and auxiliary equipment to be utilized to transport the ESS at Seal Beach and KSC and to install or remove stage-oriented equipment at all facilities.

Umbilical Equipment

Umbilical installation at KSC will be accomplished from a new service tower permanently located on the launch pad near the southwest corner of the mobile launcher.

The location of the umbilicals on the ESS were changed (as compared with those on S-II) to accommodate the new launch tower and swing arms. The new umbilical centerline is 25 degrees from position I in the direction of position II. Station 206 on the ESS is the same as Station 233 on the S-II for the aft umbilical carrier plate (A7-41); and Station 831 on the ESS is the same as Station 930 on the S-II for the forward umbilical carrier plate (A7-42).

The LH_2 fill disconnect (A7-64) and the LO_2 fill disconnect (A7-65) will be relocated also.

Seal Beach facilities will have to be modified to provide umbilical connections on the new stage locations.

ESS requirements for the following umbilical end items will result in minor changes only and will not affect their qualified status.

A7-42 Forward umbilical disconnect carrier plate assembly

A7-62 Forward umbilical fixed plate assembly

A7-64 LH_2 fill and drain propellant disconnect assembly

A7-65 LO_2 fill and drain propellant disconnect assembly

A7-66 LH_2 propellant static firing coupling assembly

A7-67 LO_2 propellant static firing coupling assembly

A7-69 LH_2 coupling support assembly

A7-70 LO_2 coupling support assembly

A7-41 Aft Umbilical Disconnect Carrier Plate Assembly. The aft umbilical carrier plate will be modified to satisfy ESS requirements. Disconnects used on the S-II stage are utilized wherever possible. Two new disconnects will have to be developed and qualified. The new 2-inch disconnect for the APS LH_2 tank fill and drain will be located off the present carrier plate. Attachments will be provided, and a limited qualification test program will be conducted to verify carrier plate ejection. The electrical umbilical interface is sufficient to satisfy ESS requirements. New disconnect requirements are as follows:

Disconnect Size (in.)	Maximum pressure (psig)	Disconnect Part Number	Nomenclature
2	15	New	APS LH_2 tank fill and drain
1	15	ME144-0011	APS LH_2 tank vent
1/2	1200	ME144-0010	APS LH_2 tank and accumulator pre-pressurization supply
1/2	1200	ME144-0010	APS LO_2 tank and accumulator pre-pressurization supply
1	4700	New	Main engine helium supply
1	1300	ME144-0011	Main engine GN_2 purge supply

A7-61 Aft Umbilical Fixed Plate Assembly. The fixed aft umbilical plate assembly for Seal Beach checkout will be modified for use on the ESS. New hardline connections and support brackets will be added. A new 2-inch coupling will be developed and attachments will be provided to support the APS LH_2 tank fill and drain system. No major changes on the electrical interface are anticipated.

Pneumatic Servicing and Checkout Equipment

The ESS will require the support of GSE at the KSC launch pad and at Seal Beach for the Station 9 type of checkout operations. Limited checkout at KSC will be supported by the S7-41 pneumatic servicing console. Operations at Seal Beach will utilize the C7-603. Modifications of this equipment will be required to support the stage systems changes which represent deltas over and above the S-II-15 configuration. Requirements affecting the S7-45 leak detection and purge console will require no equipment modification. No requirements exist for cryogenic gases, and the A7-71 heat exchanger will not be utilized for this program. Pneumatic circuits of the S7-41 normally routed through the A7-71 will be revised by routing tubing to replace and complete the circuit that would normally pass through the A7-71.

Equipment modifications required to support new or changed requirements are described below. Utilization of existing nonfunctional equipment or circuits currently contained within the end-items will be made to the fullest possible extent as the most cost-effective approach.

S7-41 Pneumatic Servicing Unit.

ESS-Auxiliary Propulsion System GH_2 Charge, Fill. Pneumatic servicing (1100 psig GH_2 AMB) for this function will be supplied by the existing start tank pressurization circuit contained in the S7-41 D section (FJA7205). The system will be modified to add orifice flow control to meet the 5 lb/min GH_2 servicing requirement. Regulators, relief valves, and pressure switches will be adjusted to conform to the reduced pressure of 1100 ± 100 psig for ESS requirements.

ESS-Auxiliary Propulsion System GO_2 Charge, Fill. Pneumatic servicing (1100 psig GO_2 AMB) for this function will be supplied by the existing thrust chamber purge and chilldown system contained in the S7-41 A and B sections. Modifications will be required as follows. In the B section the ambient helium supply (FJA9202) will be isolated from the thermocouple circuit by breaking the system and capping lines downstream of check valve A9002. Relief valve A15557 will be readjusted to match the 1100 psig pressure level. In the A section, the existing thermocouple circuit will be modified by isolating the supply from the helium manifold and capping the manifold. A new primary regulation system consisting of a new GO_2 facility supply disconnect, a manual shutoff valve, a filter, gage, a transducer, a pneumatically operated supply shutoff valve, a solenoid actuation valve, a dome hand-loader regulator and solenoid shutoff valve, a dome-loaded regulator, gage, transducer, and a calibration valve will be added and connected in series to the existing secondary regulator A8991. Regulator A8990 and A8991, transducer A15922, pressure switch A8992, and relief valve A8995 will be adjusted to new settings consistent with the required pressure level of 1100 psig. Back pressure regulator A15934 and associated check valve A15936 will be deleted from the system and the system capped. A new interface facility supply of AMB GO_2 at 6000 to 3400 will be required to supply the system.

ESS-Auxiliary Propulsion System GH_2 Charge, Purge, Pneumatic servicing (TBD psig GN_2, AMB) for this system will be supplied by the addition of a new regulation supply system to the S7-41 B section. The new system will consist of a pressure regulator and associated pressure monitoring instruments, control valves, and relief protection devices to supply the GN_2 purge. A downstream selector supply valve will be provided to the H_2 auxiliary propulsion system purge for purge supply control.

ESS-Auxiliary Propulsion System GO_2 Charge, Purge, Pneumatic servicing (TBD psig GN_2 AMB) for this system will be supplied by the new regulation supply system described for the auxiliary propulsion system GH_2 charge and purge. A second downstream selector supply valve will be provided for purge supply control to the GO_2 auxiliary propulsion system purge.

ESS-Main Engine Purge Supply. Pneumatic servicing (1250 ± 100 psig GN_2 AMB) for this function will be supplied by the existing LO_2 tank nitrogen purge system contained in the S7-41 B section. Modification will be required as follows. In the B section, ambient GN_2 supply (A9198) will be isolated from the LO_2 tank helium pre-pressurization circuit by breaking the system and capping lines downstream of pneumatically operated valve A9117. A new independent line system and components will be added downstream of pneumatically operated valve A9117 to supply GN_2 to the ESS and to provide venting, relief flow control, filtration, and pressure monitoring capability. Components to be added are a pneumatically operated vent valve and control solenoid valve, a relief valve, a flow orifice, a filter, a pressure switch, a transducer, and a calibration valve. In the S7-41 A section, existing regulators A9027 and A9028, pressure switch A9029, and relief valve A9032 will be adjusted to correspond with the required 1250 psig supply pressure.

ESS-Main Engine Helium Supply, Fill. Pneumatic servicing (4500 ± 200 psig He AMB) for this function will be supplied by the addition of a new independent system in the S7-41 B section. The new system will consist of an inlet supply manual isolation valve, a filter, a pressure-monitoring instrumentation, a pneumatically operated supply shutoff valve, and an associated control solenoid. The regulation system will consist of a primary and secondary dome-loaded regulator and associated hand loaders and dome supply solenoid isolation vent valves. The regulation system will be provided with relief valve protection, pressure-monitoring instrumentation, and manual vent valve. The downstream system will contain a pneumatically operated supply shutoff valve, a pneumatically operated vent valve and related control solenoids, a flow control orifice, and a downstream filter. The system will require a facility interface supply of 6000 to 4500 psig AMB, helium at KSC.

Electrical Requirements. Existing pneumatic modules, new modules, and the electrical J-box located in the S7-41 will be rewired to supply checkout and servicing pressures for the following ESS requirements.

1. Main engine helium supply fill

2. Main engine purge supply

3. Auxiliary propulsion system GH_2 charge, fill

4. Auxiliary propulsion system GH_2 charge, purge

5. Auxiliary propulsion system GO_2 charge, fill

6. Auxiliary propulsion system GO_2 charge, purge

New pressure transducers (different pressure ranges) may be necessary to fulfill the new requirements.

The solenoid valves, valve position limit switches, pressure switches and transducer outputs will be wired to an interface connector, providing the ESS with the necessary output/input signals for control and monitoring of the S7-41.

S7-42 Pneumatic Servicing Electrical Console. The pneumatic servicing electrical console (S7-41) is an electromechanical unit used to command and sequence the electrical and pneumatic components in the associated GSE that services the stage's systems and components during preparation for checkout or static firing. All commands and feedback signals from the various GSE pass through the S7-42.

The modification and addition of pneumatic systems in the S7-41 would require additional relay modules and the reprogramming of the patch boards to provide the necessary logic, control, and distribution of feedback indications from the new and modified systems in the S7-41.

Additional changes such as nomenclature, interfaces, pressure ranges and settings, etc., must be made to be compatible with the ESS systems.

S7-45 Leak-Detection and Purge Console. No revisions will be required for the S7-45 leak-detection and purge console to satisfy the ESS purge requirements.

C7-50 Transducer Set, Manual Pressure Checkout. The C7-50 is a portable case containing pressure transducers of various ranges. It is used in conjunction with the C7-603 (pneumatic checkout console) to provide the necessary pressure signals for monitoring, recording, and control purposes. The new requirements of the ESS system will dictate that some of the transducers be replaced with different ranges.

C7-70 Pneumatic Console Test Set. The C7-70 pneumatic console test set consists of six portable type cases. Two and one-half of the cases are used to provide storage space for control, for monitoring, and for the power cables used to connect the C7-70 test set to the S7-41, A7-71, S7-49, and S7-45 GSE. The other three and one-half cases contain electrical panels, which are equipped with switches, lights, and meters to checkout the solenoid valves, transducers, pressure switches, and wiring located in the S7-41, A7-71, S7-45, and S7-49 GSE.

Modification of the S7-41 would require that existing circuits be modified to meet the new checkout requirements. Additional changes, such as nomenclature, interfaces, and pressure ranges, must be made to be compatible with the ESS system. The A7-71 and S7-49 will not be used on the ESS program.

C7-603 Pneumatic Checkout Console.

ESS-Main Engine Helium Supply Fill. This function will be supplied by the existing engine helium bottle fill supply module G7-873315 in the C7-603 automatic section. Pressure supplied by this module will service functional and leak check requirements at 1500 psig GHe.

ESS-Main Engine Purge Supply. This function will be supplied by the existing GH_2 start tank supply. A supply of 500 psig GHe will be available to the new system for leak-check purposes from existing module G7-870635 contained in the C7-603 manual console.

ESS-Auxiliary Propulsion System GH_2 Charge, Fill. Pneumatic servicing (1000 psig GHe) for this function will be provided by the existing module G7-870042, previously used to supply LH_2 and LO_2 pressurization regulator bias checkout supply (GHe at 860 psig). The module plumbing system and components will be upgraded and requalified for service at the higher (1000 psig) pressure. Selector valve G9LV38 will service the auxiliary propulsion system GH_2 charge system of the ESS.

ESS-Auxiliary Propuslion System GO_2 Charge, Fill. Pneumatic servicing for this function will be provided by the existing module G7-870042. Modifications for this module are described under ESS-Auxiliary Propulsion System GH_2 Charge Requirements. Selector valve G9LV39 will service the auxiliary propulsion system GO_2 charge system of the ESS with GHe at 1000 psig.

Switch light and control panel nomenclature for all of the systems described will be revised to agree with the servicing functions.

Electrical Requirements. Existing pneumatic modules and control panels located in the C7-603 automatic, manual, and recorder racks will be rewired to supply checkout pressures and servicing pressures for the following ESS requirements:

1. Main engine helium supply fill

2. Main engine purge supply

3. Auxiliary propulsion system GO_2 charge, fill

The recorder rack distributor will be rewired to provide access to any new stage components (control and feedback indications) required to perform a manual leak check of the stage's pressurization system. The existing control, monitor, and recorder patch panels located in the automatic, manual, and recorder rack will be modified to accept the new requirements. Wiring changes due to interface changes will be accomplished in distributors located in the C7-603 racks.

Changes such as those in nomenclature, interfaces, pressure ranges and settings, etc., must be made to be compatible with the ESS system.

Model 961 Audio-Visual Alarm System. The audio-visual alarm system is a unit consisting of pressure switches, warning horns, and flashing lights. Its purpose is to provide an audio and visual warning signal to alert personnel when the propellant tank pressures have reached an impending danger point. The present system can be utilized for the main propulsion system propellant tanks. However, the Model 961 must be expanded in a similar manner to provide an audio-visual warning system for the auxiliary propulsion system propellant tanks.

A recording system will be added to provide a permanent record of propellant tank pressures. This would allow the continuous recording of pressures 24 hours a day, seven days a week. This information is useful in the event that an overpressure condition occurs when no personnel are present.

GH_2 Free Hydrogen Burnoff System. Two hydrogen burner assemblies will be required at KSC to support static firing. The systems will be similar to the SDD-235 utilized at MTF for the S-II program.

Flame Detector Unit. A flame detector unit will be required for static firing at KSC. The unit will be similar to the SDD-254 utilized at MTF for the S-II program.

Fluid Distribution System (SB Station 9).

Main Engine Helium Supply, Fill. Reroute the existing engine helium bottle fill line flexible hose to a new interface on the aft umbilical carrier plate.

Main Engine Purge Supply. Reroute the existing GH_2 start tank fill line flexible hose to a new interface on the aft umbilical carrier plate.

Auxiliary Propulsion System GH_2 Charge, Fill. Reroute the existing LH_2 tank regulator checkout line from the forward umbilical carrier plate to the aft umbilical carrier plate.

Auxiliary Propulsion System GO_2 Charge, Fill. Reroute the existing LO_2 tank regulator checkout line flexible hose to a new interface on the aft umbilical carrier plate.

LO_2 Tank Vent Valves. Reroute the LO_2 tank vent ducting to the new interface on the ESS. Add new supports for the additional length of line.

Portable LO_2 Vent Valve Checkout System (KSC). Add a new LO_2 tank vent valve vacuum checkout system similar to the LH_2 tank vent valve system. This system will connect from the S7-29 to the two new interfaces located near the LO_2 vent valves, and will include tygon tubing, valves, transducers, and supports.

Handling and Auxiliary Equipment

The handling and auxiliary equipment required to support the ESS for transportation, lifting, access, static firing, and miscellaneous component handling and installation functions will include existing S-II end items, modified S-II end items, and new end items. The following list includes new and modified end items which are required due to the addition of new components and configuration changes in the aft compartment area.

Engine Protective Frame (New). This frame will provide support for the aft stage cover and will provide environmental and physical protection for the ESS engine compartment during handling and transportation. The frame will be constructed of tubular members formed into a cylinder-like structure around the perimeter of the engines.

Aft Stage Cover (New). This cover will provide environmental protection for the ESS engine compartment during handling and transportation. The cover will be constructed of a flexible lightweight material to be installed over the engine protective frame.

Orbiter Engine Installation Adapter (New). This new adapter, in conjunction with GFE installers, will be utilized to install and remove the orbiter engines. The adapter function and construction will be similar to the J-2 engine installation adapters.

OMS Engine Installer (New). This new installer will be utilized to install and remove the OMS engines. The function and construction will be similar to the J-2 engine installers.

Tank and Accumulator Installation Set (New). This set will be comprised of equipment that will be utilized to install and remove various helium receivers, OMS LH_2 and LO_2 tanks, and thruster GH_2 and GO_2 accumulators. The construction and usage description is not known at this time.

Miscellaneous Component Handling Adapter Set (New). This set will consist of a storage container and a set of adapters designed to assist in the removal and installation of various components (i.e., LH_2 and LO_2 prevalves, auxiliary propulsion thrusters, etc.) on the ESS. The construction and usage description is not known at this time.

Air Stream Deflector Dolly (New). This dolly will support and provide mobility for the air stream deflector which will be installed at KSC. It will be constructed of structural steel assembled to form a trailer-type frame. The dolly will be equipped with attachments for the deflector, front and rear undercarriages, running lights, and reflectors.

Engine Area H_2O Firex System (New). This system will provide the capability to extinguish fires occurring in and about the ESS engines during static firing by providing distribution of water in and around the engine compartment. The system will be constructed of components and pipe formed into manifolds around the engines with nozzles located to provide an effective water spray to the engines.

Engine Area GN_2 Purge Manifold (New). This manifold will provide for purging of the ESS engine area (boattail area) during static firing whenever high concentrations of propellant gas or a fire has been detected. The manifold will be constructed of tubular structures designed to attach to the facility supply manifold and extend inward toward the orbiter engines.

Engine Area Fragmentation Shield (New). This shield will provide protection for the equipment in the ESS engine compartment in case of orbiter engine explosions during static firing. The shield will be constructed similarly to the S-II fragmentation shield.

Engine Area Heat Shield (New). This shield will provide protection for the ESS structure and equipment in the engine compartment area from the radiative heat generated by the orbiter engines during static firing. The shield will be constructed similarly to the S-II heat shield.

Engine Servoactuator Piston Position Indicators (New). These indicators (different configuration for orbiter and OMS engines) will be used to indicate visually the position of the piston in the hydraulic servoactuator, or the angular position of an engine in terms of the actuator stroke. The indicators will be constructed into a cylindrical structure having an indicator rod within the cylinder, and attaching mechanically to the servoactuator body with the rod attaching to the rod end. The indicator will have a linear and a vernier scale.

Engine Actuator Locks (New). These locks (different configuration for orbiter and OMS engines) will be utilized to immobilize the engine servoactuators to prevent any relative motion between the actuator body and the rod during ground handling and transportation operations. The locks will be constructed into a split locking collar and a clamp for securing the collar to the actuator.

Engine Actuator Simulator (New). These simulators (different configurations for orbiter and OMS engines) will provide a stabilizing element for a neutrally positioned engine when the engine actuator is removed. The simulators will be constructed to include a strut equipped with end fittings duplicating those of the engine actuators.

Hydraulic System Installation Fixtures (New). These fixtures (different configurations for orbiter and OMS engines) will be utilized to build up and functionally check out the hydraulic system and install it on the ESS as a unit. The fixtures will be constructed into a tubular mobile frame with arms, handles, control devices, and hydraulic interfaces.

Engine Actuator Component Tool Set (New). This set will consist of a container and a set of tools designed to assist in the removal and installation of hydraulic components (i.e., upper pin remover, bench measurement tool, support tools, etc.) on the ESS. The construction and usage will be similar to the S-II tools.

Engine Compartment Center Area Platform (New). This platform will provide support for gaining access in the ESS engine compartment center area to enable maintenance therein when the stage is in the vertical position. The platform will be constructed of lightweight material in segments to permit easy installation and removal.

Aft Support Ring, H7-3 (Modified). The modification of the H7-3 is required to provide the proper mounting interface to the ESS aft skirt and to extend the aft support ring interface to accommodate the existing S-II stage transporters. The modification will consist of adding an A-frame extension (approximately 16 inches) to the I-beam, which will interface with the stage.

Engine Compartment Platform Set, A7-84 (Modified). This modification will consist of redesigning the segments of the Station 30 platform at Positions I and II to allow for clearance of the orbiter engine bells. The Station 114 platform may have to be lowered to allow clearance of the thrust cone. The heat shield protective set will have to be modified due to the new configuration of the heat shield.

LO_2 Tank Access Platform SDD-258 (Modified). This modification will consist of extending the platform upward in the thrust cone to gain access to the LO_2 manhole. This change is required due to the lowering of the thrust cone and the additional components installed in that area.

Thrust Cone Access Ladder, A7-85 (Modified). This modification will consist of either extending or shortening the ladder, depending on the design change to the A7-84 in relation to the thrust cone.

LH_2 Tank External Entry Equipment, SDD-32 (Modified). This modification will consist of redesigning the outer attach fittings for the platform cable supports and hoist support. This change is required because of the upper payload vehicle configuration change, since this equipment is used in the stacked condition.

2.5.2 Support Software

Software required to support the ESS checkout, acceptance, launch, and flight mission will be developed for the shuttle program with minimum changes necessary to adapt to the ESS. In conjunction with the usage of the support software, the use of the Avionics Systems Integration Laboratory (ASIL), configured to the ESS requirements, will be required to verify and certify the software is properly adapted and will support the ESS. The following modular software items will be involved in developing software for the ESS:

Ground checkout

Simulated flight

- Prelaunch
- Vertical launch
- Orbital coast
- Rendezvous
- Stationkeeping
- Payload separation
- Equipment recovery
- Deorbit

The philosophy of maximum use of shuttle-developed hardware and subsystems will afford maximum use of modular software developed for comparable use on the Shuttle Program. Commonality is further enhanced by use of common GSE (UTC) and by use of the ASIL for verification of ESS software.

2.6 WIND TUNNEL MODELS

Models of the expendable second stage with three payload configurations simulated were built to 0.0031 scale. These payloads were the space station, reusable nuclear shuttle, and space tug configurations. Each model was mounted on a 0.0035-scale model of the General Dynamics B-15 B-1 booster, as well as being separately mounted and tested. The difference in scale is because of an attempt to simulate the later B-9U booster configuration by matching the ESS to the B-9U body size. Models are described in detail in Volume III in Section 3.0.

3.0 RELIABILITY

This section presents the ground rules, results, and conclusions of the expendable second stage (ESS) Phase B reliability analysis. The reliability analysis consisted of performing single point failure mode effect analysis (FMEA), wherever applicable, and a written analysis of each major subsystem.

The basic failure criteria for the ESS is that it will not degrade the fail-operational, fail-safe (FO/FS) requirement of the booster during mated boost and booster/ESS separation. Following the safe separation from the booster, the basic ESS requirement is to fail safe; however, with vehicle operation in the proximity of any manned vehicle, the ESS operational systems must meet FO/FS criteria. The FO/FS system requirement for the successful completion of the deorbit phase of the mission is also considered to be a mandatory requirement.

3.1 SUMMARY

The reliability analysis of the ESS major subsystems determined that the ESS generally meets the design requirements established for the various phases of mission operation. The areas in which the fault tolerance levels have not been completely complied with are primarily associated with the use of existing S-II subsystems and components. The ESS system areas contain failure modes which can prevent the starting of one or both main engines and can cause the loss of the main engine(s) during the burn period. These failure modes exist for the present S-II stage and are considered to have a low probability of occurrence because of the extensive system/component testing including the successful flight experience to date.

These single-point failures consist of the inadvertent opening of the dual parallel main tank vent valves or the propellant tanks fill valves which can cause both main propulsion engines to become inoperative, and result in the complete loss of the mission. Other failures of this category which can affect the startability of the main propulsion engines are associated with normally open engine recirculation conditioning system valves (prevalves and recirculation return) failing closed.

The failure of both main engines to start while mated to the shuttle booster can present a safety hazard to the booster crew because of the proximity of the ESS stage to the booster's vertical stabilizer during the separation maneuver. Additional analysis is contemplated to determine if the condition is completely safe.

The auxiliary propulsion system (APS) is a new system with no test or flight experience available from any other vehicle. The APS will utilize the basic system design and reliability concepts presently proposed for the shuttle orbiter. The APS consists of the orbital maneuvering system (OMS) and the attitude control propulsion system (ACPS). The maximum fault tolerance level of FO/FS was established for the OMS to meet the requirements for deorbit. The APS has several single point failures (operational failure of system components) which could cause one of the OMS engines to become inoperative. However, the remaining OMS engine is capable of completing the on-orbit phase of the mission. It is also capable of completing the deorbit mission requirements with the ACPS providing the necessary vehicle pitch control. If both OMS engines become inoperative, deorbit capability would be accomplished via the ACPS as an emergency mode of operation.

The design criteria for the ACPS is FO/FS during the critical period of ESS/booster separation and during the periods in which the ESS will be near any manned orbiting vehicle. The ACPS contains single point hardware failures which could cause the loss of one or more ACPS thrusters; however, the guidance and control system will provide the necessary control of alternate thrusters to compensate for any attitude errors resulting from the inoperative thrusters.

The electrical power and control systems will have FO/FS capability except for the instrumentation and communications subsystems. Specific areas or critical measurements provided by the instrumentation subsystem are redundant. The communications subsystem has several single-point failures which could result in serious rendezvous position errors and/or loss of deorbit capability.

Additional analysis will be conducted during the subsequent study phase or during the design implementation phase to eliminate where possible the identified single point failures or minimize the possibility of their occurrence.

3.2 FAILURE MODE ANALYSIS AND HARDWARE CRITICALITY

Failure mode effect analyses (FMEA) have been conducted on all major ESS systems to increase system design reliability by identifying and eliminating, where possible, all first-order critical failure modes. During the design implementation phase or subsequent study efforts, detailed FMEA will be conducted for each system addition or change and will also establish the hardware criticality categories. The effect analysis describes how equipment in the stage systems can fail and the resultant effects of the failure on the mission objectives.

The FMEA will also establish the component criticality categories. The criticality of an item refers to the potential effect of its failure on the stage, mission, or personnel. The potential effect of the failure depends on how the equipment is used and its mode of failure. Hardware criticality determination will be based on the following definitions:

Criticality	Potential Effect of Failure
I	Loss of life of crew member(s) (ground or flight). Also includes safety and hazard warning systems for primary operating systems whose failure has potential of loss for any associated crew member's life.
	Loss of vehicle
II	Immediate (safe) mission flight termination.
	Loss of primary or secondary mission objectives
	Launch scrub or delay

During the design implementation phase, failure mode cause analysis will be conducted by the suppliers for each new critical component in a stage system including the main propulsion system engines, the orbital maneuvering system engines, and the attitude control propulsion system thrusters.

The purpose of the failure mode cause analysis is to obtain a high reliability standard through prehardware evaluation of potential failure causes of the hardware and concomitant preventive measures. The failure mode for each of these critical components will be examined for potential failure causes and the identification of design characteristics as to why these causes should not occur. Preventive design actions may be taken whenever design weaknesses are highlighted.

The ground rules under which the ESS reliability analysis including FMEA's were conducted are as follows:

1. Only first-order failure modes that could result in main or auxiliary propulsion failures and mission loss are included.

2. Stage structural failure modes are not considered.

3. Structural failure of primary and secondary structural items including rupture of lines and components are not considered.

The following sections contain a discussion, by stage subsystems, of design and operating characteristics relative to failure modes and the overall achievement of goals established by the specified criteria.

3.2.1 Main Propulsion Subsystem (MPS)

The MPS incorporates a one engine-out capability for the payload configuration of a space tug and the RNS. The ESS MPS does not have the capability to complete the mission with the space station if one MPS engine is lost during separation from the shuttle booster, or during the first third of the ESS boost period. This capability does exist if the failure occurs during approximately the last two-thirds of the ESS boost.

Several single-point failure modes in the MPS supporting subsystem would result in loss of both engines being started or continuing to operate which would result in loss of mission. Their occurrence while still mated to the shuttle booster would also jeopardize the shuttle booster and crew during the critical period of separation. (Safe ESS/booster demating with both main engines out is not assured at this time.) The failure modes that can cause this condition are identified as loss of stage tank pressure because of an open vent valve or propellant fill valve, and loss of propellant recirculation capability.

After selection of the engine manufacturer, the MPS engine supplier failure mode cause analysis will be made to detail the various engine system components which would contribute to loss-of-engine failures.

The vehicle electrical control system for the ESS main engines is composed of 115/200 volt, 400 Hz, three-phase power inverters; acquisition control test (ACT) units; power transfer switches; and power control switches.

The vehicle engine control system will use two ACT units per engine, each powered by a sub-bus containing redundant batteries. This redundancy provides for the possibility of two failures occurring without losing control of an engine. Redundant inverters supplying power to each engine will be controlled by two power transfer switches located in series.

In addition, each engine (on the ESS) is controlled by an electronic controller which contains two digital computers for increased reliability.

The system is designed so that any combination of two failures in any of the engine control hardware such as main dc buses, ac buses, ACT units, transfer switches, control switches, or engine computers would retain one engine operational. The system is also designed to have the capability to functionally check all of the redundancy aspects of the system and isolate failures to a single component. The engine has the capability to check its system and identify any failures to the vehicle main computer via the ACT unit/data bus system. In the event both power sources fail, the engine has the capability to shut down safely.

SINGLE POINT FAILURE MODE EFFECT ANALYSIS (FMEA) ASSOCIATED WITH THE POSSIBLE LOSS OF A MAIN PROPULSION SYSTEM

MAIN PROPULSION SUBSYSTEM (MPS)

Item	Function	Failure Mode	Failure Effect
MPS engine	Two main engines are provided to boost the ESS with its payload into orbit.	One engine fails to start (MPS engines are started while still mated to the shuttle booster)	Loss of mission with space station payload. One engine-out capability exists for missions with lighter payloads (tug and RNS). Loss of one engine during the booster/ESS separation phase does not present a hazard to the shuttle booster or crew. Loss of both engines while mated with the shuttle booster will require further analysis to determine if separation can be accomplished safely.
		One engine fails to continue running	MDAC space station payload mission cannot be successfully completed if engine failure occurs during approximately the first third of the ESS boost period.

Pressurization Subsystem (Main LO_2 and LH_2 Tanks)

The ESS main propellant tank pressurization system utilizes the basic design associated with the S-II pressurization system. Both LO_2 and LH_2 tank pressures are supplied from the main engines through orifices considered passive components with no failure modes. Both LO_2 and LH_2 tanks are protected against overpressurization by two vent valves located in parallel, thus providing pressure relief redundancy. The most critical failure mode of the pressurization system is a single vent valve failing to remain closed. This condition would deplete tank pressure and cause loss of both engines, resulting in mission loss. Occurrence of this failure while the ESS is mated to the shuttle booster could also jeopardize the safety of the shuttle booster crew members.

Loss of tank pressure may also lead to structural failure. However, these modes of failure are considered to have a low probability of occurrence and have never been experienced during any S-II stage flights.

Electrical control for the ESS pressurization system consists of signals for the LO_2 and LH_2 tank venting and sensing of LO_2 and LH_2 tank pressures. All electrical control functions are completed prior to liftoff except one. Approximately 10 seconds after ESS main engine ignition, the low-pressure mode signal to the main tank propellant vent valves is initiated by the main engine computer. The signal is transmitted through redundant ACT units to stage redundant data buses. The information is then received by redundant ACT units, through redundant power switches, for control of the two vent valves on the LH_2 and LO_2 tanks, thus fulfilling FO/FS criteria.

Prelaunch, ground checkout, and standby operations require FO/FS electrical control which is attained by double and triple redundant sensing and control hardware and interconnection of two ACT units with the appropriate solid-state power control switches.

SINGLE POINT FAILURE MODE EFFECT ANALYSIS (FMEA) ASSOCIATED WITH THE POSSIBLE LOSS OF A MAIN PROPULSION SYSTEM

PRESSURIZATION SUBSYSTEM (MAIN LO_2 AND LH_2 TANKS)

Item	Function	Failure Mode	Failure Effect
Vent valve LO_2 and LH_2 (two required per tank)	Two vent valves per tank (LO_2 and LH_2) are provided in parallel to provide tank venting during propellant loading and also to vent excess ullage pressure following tanking and during flight. Valves are normally closed.	One valve fails to remain closed.	Loss of tank pressure in either LO_2 or LH_2 system will result in loss of both MPS engines and loss of mission. Loss of both engines while mated to the shuttle requires further analysis to determine if separation can be accomplished safely. This failure has a low probability of occurrence based on the present successful application of vent valves on the S-II program.

LH_2 Recirculation Subsystem

The LH_2 recirculation is accomplished by driving the engine low-pressure LH_2 pump with a small electric motor. This portion of the subsystem will be analyzed for failures by the engine supplier. The remainder of the system is composed of passive components, i.e., lines and fittings, except the return valve which is mounted at the LH_2 tank. This valve is normally open. Valve failing in the closed position would interrupt LH_2 recirculation for both engines, resulting in loss of engine start capability and loss of mission. Failure of the engine-mounted pump or electric motor would result in the loss of one engine to start.

Valve Actuation Subsystem

The ESS valve actuation system is basically the same as that used on the S-II stage. There are no first-order failure modes that would result in loss of an engine or mission.

Engine (MPS) Servicing Subsystem

Servicing systems required to provide purge, engine valve actuation, engine nozzle extension and retracting, and other needs have not been established sufficiently to warrant an analysis. To make a meaningful failure mode effect analysis, engine supplier designation and specific engine configuration and requirements are needed. Subsequent study phases will be concerned with this subsystem.

SINGLE POINT FAILURE MODE EFFECT ANALYSIS (FMEA) ASSOCIATED WITH THE POSSIBLE LOSS OF A MAIN PROPULSION ENGINE

LH_2 RECIRCULATION SUBSYSTEM

Item	Function	Failure Mode	Failure Effect
LH_2 recirculation return valve	This normally open valve located near the LH_2 tank controls the return flow of LH_2 circulated through both engines. Valve remains open for recirculation and is closed at the time engines are started. Closed function is redundant to the engine-mounted fuel bleed valves.	Valve fails to remain open.	Valve in closed position will terminate engine preconditioning and can prevent both engines from starting. Low probability of this failure occurring in that the valve is normally open and is pneumatically actuated closed. Valve is presently used in the S-II engine recirculation subsystem. Loss of both engines will result in booster crew hazard during staging and loss of ESS mission.

LO_2 Recirculation (Helium Injection) and Pogo Suppression Subsystems

The helium injection system required for LO_2 recirculation is basically the same as the S-II system except one solenoid valve controls the flow of He rather than two. (Two were required to meet flow requirements on the S-II). The normally open valve failing closed or regulator failing closed would cut off He injection supply and prevent one or both engines from starting. Loss of both main engines during shuttle/ESS separation will result in mission loss and create a hazard to shuttle booster crew.

Each engine recirculation system contains a normally open LO_2 recirculation return valve which can prevent the engine from starting if the valve fails to remain in the open position. These failure conditions are considered to have a low probability of occurrence because of the successful service experienced on the S-II stages.

Loss of pogo suppression capability is not a critical failure mode in itself unless its loss is determined to result in structural failure. Therefore, additional study of pogo effects and probability of occurrence is needed to determine criticality of the pogo suppression system. Failure of the LO_2 feedline accumulator bleed valve in the closed position would prevent filling the accumulator with LO_2 and may adversely affect engine start. This bleed valve is the same as the recirculation return valves.

SINGLE POINT FAILURE MODE EFFECT ANALYSIS (FMEA) ASSOCIATED WITH THE POSSIBLE LOSS OF A MAIN PROPULSION SYSTEM

LO_2 RECIRCULATION
(He) INJECTION AND POGO SUPPRESSION SUBSYSTEM

Item	Function	Failure Mode	Failure Effect
Pressure regulator	Reduces the high pressure He receiver system from 3000 psi to 750 psi for He injection into the LO_2 recirculation system and for the LO_2 feed line accumulater pogo suppression system.	Fails closed (regulates low)	Loss of sufficient He injection flow in the LO_2 recirculation system may result in failure to start main engines. Failure after main engine start would result in loss of pogo suppression system which could cause structural damage and loss of main engines and mission.
		Fails open (regulates high)	System overpressurization relief valve would open. This condition could cause He supply to become depleted prior to main engine shutdown. Loss of pogo suppression would result.
Solenoid valve	Controls flow of He, from the regulated supply, to the LO_2 recirculation return line. Valve is normally open and He injection is initiated prior to liftoff.	Valve fails to remain open.	Loss of LO_2 recirculation capability for both engines resulting in possible loss of engine start. Low probability as valve is normally open.

SINGLE POINT FAILURE MODE EFFECT ANALYSIS (FMEA) ASSOCIATED WITH THE POSSIBLE LOSS OF A MAIN PROPULSION SYSTEM (CONT)

LO_2 RECIRCULATION
(He) INJECTION AND POGO SUPPRESSION SUBSYSTEM

Item	Function	Failure Mode	Failure Effect
LO_2 recirculation return valve	A normally open valve which controls flow of recirculated propellant from the engine back into tank. One LO_2 valve is required per engine.	Valve fails to remain open.	Loss of LO_2 recirculation capa- for one engine system. Possible loss of engine start capability. Low probability of failure as valve is normally open. Failure of LO_2 valve will also prevent proper willing of LO_2 feedline accumula- tor with LO_2.
Valve, accumulator bleed	A normally open valve which vents the accumulator to allow for proper filling of the accumulator. Bleed prevents GO_2 forming prior to main engine start.	Fails closed	LO_2 feedline accumulator will not be filled with LO_2 resulting in GO_2 formation and possible loss of one engine-start capability.

Thrust Vector Control (TVC)

The ESS thrust vector control (TVC) for the main engines incorporates a portion of the design and components proposed for the shuttle orbiter vehicle. Basically the system is an S-II engine actuation system (EAS) scaled up to meet the requirements of the orbiter engines.

NASA is currently evaluating the potential addition of an engine-driven accessory pad to the space shuttle engine. If approved, the ICD would be changed accordingly. The contractor has assumed this capability will be provided, and has described a hydraulic system based on this. In the event this capability is not provided, the type of system selected would be a pneumatically driven hydraulic system.

Each main engine has its individual TVC system. The system consists of an engine (MPS) driven hydraulic pump and an electric motor-driven auxiliary pump. Both pumps supply hydraulic pressure to an accumulator reservoir manifold assembly (ARMA) which in turn supplies hydraulic pressure to two actuators. Both actuators are required to provide a pitch, roll, or yaw moment to the stage.

The auxiliary pump supplies pressure before main engine start for engine valve control. It continues after start of engines and supplies hydraulic pressure in combination with the engine-driven pump, thus providing redundancy for pump pressure supply at a reduced flow rate. The actuators and ARMA have no redundancy. Loss of hydraulic pressure during flight will result in the actuators locking up, thus preventing shutdown of that engine. Flight control will be provided by the remaining engine and the ACPS. The system has FO capability with failure in this mode. A failure of the TVC resulting in an engine gimballing hardover will require shutdown of that engine. A failure that causes shutdown of one main engine will result in loss of the primary mission in those situations where the ESS does not have one main engine-out capability. Therefore, the TVC does not meet FO-FS criteria in all cases.

The electrical control of the TVC provides FO-FS capability and is provided from the guidance, navigation, and control (GN&C) system which incorporates redundant acquisition, control, and test (ACT) units supplying excitation to each actuator servo-valve.

Propellant Feed Subsystem

The propellant feed subsystem consists of the LO_2 and LH_2 tanking system and the propellant delivery system to the MPS engines. The basic system for propellant loading utilizes the S-II propellant fill and drain valves. A fill valve failing to remain closed could spill propellant on sections of the booster and would also result in loss of main tank pressure which would cause the failure of both engines to start or to maintain operation. These valves are normally closed (spring loaded) and are considered to have a low probability of valve failing to remain closed because of the extensive valve qualification testing and the successful S-II stage static firings and flight operations.

The propellant supply to the engines is through normally open prevalves as on the S-II present design. However, the prevalves and feed ducts are 13 inches in diameter on the ESS as compared to the 8-inch diameter on the S-II. The prevalves have no functional failures with a mission effect as they are not required to operate during recirculation system operation as on the S-II stage.

These valves will be developed for the shuttle orbiter program and will contain actuation pressure redundancy to meet the requirements of FO/FS. The critical failure mode associated with the prevalves is inadvertent closing of the valve because of actuation or valve mechanism failure causing flow restriction. This failure condition can also be considered to be a low probability of failure based on previous program experience with similar installations.

SINGLE POINT FAILURE MODE EFFECT ANALYSIS (FMEA) ASSOCIATED WITH THE POSSIBLE LOSS OF A MAIN PROPULSION ENGINE

PROPELLANT FEED SUBSYSTEM

Item	Function	Failure Mode	Failure Effect
Main tank fill valve (LO_2, LH_2)	Provides a means of filling the stage tanks (LO_2 and LH_2). Valves are closed following tanking and are required to remain closed during flight. Two valves are required per stage.	Fails to remain closed in flight.	Valve failing open would create hazard from overboarding of propellant (LO_2 or LH_2). Loss of tank pressure and engine NPSP would result in loss of the main engine start capability or ability to continue to operate. Loss of both engines results in loss of mission. Loss of both engines during separation from the shuttle booster requires further analysis. Probability of valve failing open is low as valve is normally spring loaded closed.

Subsystem Electrical Control

The electrical control subsystem provides the power and control of mechanical components in the propellant feed, helium injection and pogo suppression, LH_2 recirculation, and valve actuation subsystems. The system is required to be fail operational-fail safe during shuttle boost and separation. Thus, two or three electrical control paths are provided, depending on a function failing safe or unsafe when the control power is lost.

System reliability will be assured by utilizing redundant components and measurements and selection of proven hardware.

Switching signals are generated by the on-board computer and operate through triple redundant acquisition, control and test (ACT) units, and the data bus.

System/component operations will be verified for a set number of samples and then transmitted via the ESS computer/data bus and comparison failure of two or more indicators will switch to an alternate electrical control path. By applying these redundancy techniques, a high reliability is maintained throughout the electrical control for all of the subsystems.

Safing System

It has been a design objective that the required safing system not degrade the primary mission reliability and that failure of a single critical component would not cause the safing system to become inoperative.

The ESS safing system will consist of safing provisions for the main propellant tanks. Both the LO_2 and LH_2 tanks will utilize safing systems similar to the S-II-13 terminal stage which consists of redundant ordnance actuated vent valves. The electrical control system for activating the safing system tank vent valves has changed from the payload separation initiated timers and relay control logic to the ESS computer/data bus/acquisition, control system with solid-state power control switches.

The critical failure mode associated with the proposed safing system is premature safing which would result in the inability of the ESS to complete the mission objectives. Low probability for this type of failure occurring has been achieved by designing the safing system to become activated by the ESS computer/data bus/acquisition control system. The typical electrical control circuit used for safing will utilize independent solid-state power control switches (PCS) for arming and triggering the individual EBW firing units.

The electrical safing circuitry and the ordnance/vent valve system contain parallel redundant components which preclude the possibility of a single failure causing the safing system to become inoperative.

3.2.2 Auxiliary Propulsion Subsystem (APS)

The auxiliary propulsion system (APS) consists of the orbit maneuvering system (OMS) and the attitude control propulsion system (ACPS) utilizing common LH_2 and LO_2 propellant tanks. The two OMS engines supply the necessary thrust for achieving final orbit, rendezvous, and for deorbiting the stage. The ACPS consists of 14 thrusters for maintaining vehicle attitude during orbital periods, for supplying braking thrust for rendezvous, and for providing vehicle roll control during the condition of one main engine out. The proposed auxiliary propulsion system will utilize the system design/component concepts used for the shuttle orbiter vehicle. The basic reliability of these system concepts is retained to achieve maximum functional integrity.

Single point failures in the APS which would affect both the OMS and ACPS are primarily associated with excessive propellant utilization caused either by propellant leakages or excessive ACPS operation. If the ACPS thrust chamber feed valves which control the GO_2 and GH_2 to the individual thrusters do not completely close when required, the loss of APS propellant would result. This condition could also create excessive demands on the system to maintain vehicle attitude resulting from the low thrust developed by the propellant gas leaking through the thruster. This condition could result in premature depletion of APS propellant if excessive gas leakages are present, the condition occurs early in the life of the mission, and the condition is not detected and the bank of thrusters isolated.

The vent valve provides control of gas exhausting from the heat exchanger/turbo-pump gas generators through either the propulsive vent or the nonpropulsive vent. If one of the two valves used per stage fails in the propulsive vent position, the constant thrust developed would cause demands on the ACPS to maintain vehicle attitude.

These conditions will require further analysis during subsequent study efforts or during the design implementation phase to determine the probability of failure and to ascertain if the system reliability could be improved significantly.

Attitude Control Propulsion Subsystem (ACPS)

The ACPS is required to provide vehicle roll control during the critical period of booster separation with one main ESS engine out and pitch control during the deorbit phase of the mission with one OMS engine inoperative.

Single-point failures associated with the ACPS in which one or more thrusters could become inoperative are identified in the failure mode effects analysis.

The ACPS is also designed to provide necessary pitch control during the deorbiting phase of the mission with only a single OMS engine operating. During this phase of control, an operational loss of any pitch thruster would cause the vehicle to initiate a roll which can be controlled by the guidance system switching control of the thrusters; the pitch thrusters would be changed to supply the necessary thrust required for roll control and vice versa.

The electrical control system for the ACPS will basically consist of the same system used for control of the OMS engines. The system uses attitude control driver units (ACDU) for accepting information from the guidance system and for controlling the appropriate thruster propellant valves and igniters. System reliability is assured by the use of triple redundant components within each ACDU. The system's control computer will have the capability to detect the failure of any thruster and to command other thrusters that have the capability to supply the necessary vehicle control moments.

ATTITUDE CONTROL PROPULSION SYSTEM (ACPS) FAILURE MODE EFFECTS ANALYSIS (FMEA) SINGLE POINT FAILURES ASSOCIATED WITH THE LOSS OF AN ATTITUDE CONTROL THRUSTER

Item	Description	Failure Mode	Failure Effect
ACPS thrust chamber feed valves (O_2 and H_2)	Controls GO_2 and GH_2 supply to the individual thrusters. Valves are normally closed with 2 valves required per thruster. Total of 14 valves are used per ACPS module (7 thrusters) with a total stage requirement of 28 valves (14 thrusters).	Valve fails to remain open when required.	That particular thruster would become inoperative resulting in the guidance system providing the necessary control of alternate thrusters to compensate for the inoperative thruster.
		Valve fails to close when required (excessive leakage).	Loss of O_2 or H_2 gas could result in eventual loss of propellant; low thrust mode of thruster would also cause excessive demands on system to maintain vehicle attitude resulting in the rapid depletion of propellant. Would have to shut down that bank of thrusters to avoid excessive loss of propellant.
ACPS thruster bank pressure regulators (O_2 and H_2)	Regulates O_2 and H_2 pressure to the ACPS thrusters. Two regulators (O_2 and H_2) are used for each bank of thrusters. Total of three H_2 and three O_2 regulators are used per thruster module with a total of 12 regulators required per stage.	Regulates O_2 or H_2 pressure high or low	Regulator system would be shut down resulting in the complete loss of that bank of thrusters. With the loss of any bank of thrusters, the guidance system would provide the necessary control of alternate thrusters to compensate for any attitude errors resulting from the inoperative thrusters.
ACPS thruster bank pressure regulator isolation valves (O_2 and H_2)	Isolates failed thruster bank/regulator. Valves are normally open with two located in series for each pressure regulator. A total of 12 valves are used per module with 24 valves required per stage.	Valve fails to remain open.	Loss of that bank of thrusters controlled by this valve would result in the conditions noted above. Low probability of failure since valves are normally open.

Orbital Maneuvering Subsystem (OMS)

The OMS is required to meet the desired system requirements criteria FO/FS established by the deorbit requirements of ensuring that the vehicle flight path is within the permissible pre-established bounds. Crew survivability is not involved.

OMS single-point failures which would cause the loss of an engine are identified in the failure mode effects analysis.

The electrical control system for the OMS engines will interface with the GN&C through the data bus/engine control driver units. Each engine will be controlled by one control driver unit which contains circuitry to control the engine's propellant feed valves and provide ignition excitation. Each control driver unit contains dual integral redundancy which provides sufficient reliability to ensure against a single failure that could cause the loss of an OMS engine. A dual failure within one engine's control driver unit would enable the OMS system to meet the requirements of FO/FS by the successful operation of the remaining engine.

The OMS engine actuation system provides hydraulic power and actuation forces that gimbal the engines to provide vehicle directional control during orbit operations. Each engine will contain a completely independent hydraulic system similar to the engine's independent hydraulic system successfully used on the S-II stage. Hydraulic power will be supplied by an electrically-driven pump for each engine system. The failure of an electrically-driven hydraulic pump would cause the engine actuation system to become inoperative resulting in lockup of the servoactuators and loss of that engine's gimballing system. This failure mode would cause demands on the ACPS to maintain proper vehicle attitude for the deorbit phase of the mission.

To ensure a high reliability for the electrical control system, the thrust vector control electronics and hydraulic pump electrical power are redundant for each engine's gimballing system.

THE LOSS OF AN ATTITUDE CONTROL THRUSTER ORBITING MANEUVERING SYSTEM (OMS ENGINE) FAILURE MODE EFFECTS ANALYSIS (FMEA) SINGLE POINT FAILURES ASSOCIATED WITH THE LOSS OF AN OMS ENGINE

Item	Description	Failure Mode	Failure Effect
OMS engine thrust chamber feed valve (O_2 and H_2)	Controls propellant/oxidizer flow into OMS engine thrust chamber. Valves are normally closed – two valves per engine (O_2 and H_2), four valves per stage.	Valve fails to open	OMS engine will fail to start
		Valve fails to close (excessive leakage)	Excessive leakage of LO_2 or LH_2 through the failed open valve would result in eventual loss of propellant which could result in complete loss of OMS engines and ACPS.
Turbopump (O_2 and H_2)	Gas generator driven turbopumps which are common to either orbit maneuvering or attitude control operating modes. The turbopumps supply high-pressure LH_2 and LO_2 to the OMS engine. Total of four pumps are utilized per stage.	Inoperative pump/turbine	Loss of OMS engine
Turbopump gas generator (O_2 and H_2)	Gas generator supplies the necessary gases for operating the turbine-driven turbopumps noted above. Two gas generators are required for each OMS engine. Total of four gas generators are utilized for turbopump operation to support the operation of the OMS engines.	Inoperative generator	Causes turbopump to become inoperative resulting in loss of that OMS engine.
Turbopump gas generator propellant shutoff valves (O_2 and H_2)	Controls the supply of O_2 and H_2 to the gas generators. Valves are normally closed. Two valves are used for each gas generator with a total of eight valves utilized per stage.	Valve fails to open or fails closed after being open.	The gas generator will fail to operate resulting in the turbopump being inoperative and the respective OMS engine becoming inoperative.
		Valve fails to close (excessive leakage)	Excessive leakage of LO_2 or LH_2 through the failed open valve would result in eventual loss of propellant which could result in complete loss of OMS engines and ACPS.
Selector valve (LO_2 and LH_2) OMS engine or ACPS conditioning loop	Controls propellant flow to either OMS engine or ACPS conditioning loop – two valves per engine (LO_2/LH_2), four valves per stage.	Valve fails to allow flow to OMS engine	Loss of OMS engine
Turbopump gas generator O_2 and H_2 pressure regulator	Regulates O_2 and H_2 gas pressure to the gas generator. Two regulators are used for each OMS engine with a total of four regulators required per stage.	Regulates high or low.	Potential hazard from over/under pressure. Time dependent fire/explosion due to improper mixture ratio. Both gas generators will become inoperative resulting in loss of the turbopump and the respective OMS engine.
Turbopump gas generator O_2 and H_2 isolation valves	Gas generator isolation valves are used to isolate a failed gas generator. Two valves are located in series upstream of each pressure O_2 and H_2 gas pressure regulator. Valves are normally open Four valves are required for each OMS engine with a total of eight valves utilized per stage.	LO_2 or LH_2 valve fails to remain open.	Both gas generators used for driving the LO_2 and LH_2 turbopumps would become inoperative resulting in the loss of an OMS engine. Low probability of failure since valves are designed to be normally open.
LO_2/LH_2 supply prevalves	Controls to the supply of LO_2 and LH_2 to the OMS engines. Valves are normally open and are located two in series for each turbopump/OMS engine. Total of four valves are utilized per OMS engine with eight required per stage.	LO_2 or LH_2 valve fails to remain open.	Loss of OMS engine. Low probability of failure since valves are normally open.
Propulsive vent valve	Provides control of gas from the heat exchanger/turbopump gas generators through either the propulsive vent or the nonpropulsive vent. Valve is positioned to the propulsive vent position during OMS engine firing. Total of two valves utilizes per stage.	Valve fails in the propulsive vent position	Constant thrust from vent would cause demands on ACPS to maintain vehicle attitude resulting in increased propellant consumption.

3.2.3 Avionics Subsystems

Guidance, Navigation, and Control

The guidance, navigation, and control (GN&C) system will be designed to FO/FS criteria for low-orbit insertion, orbital transfer, and rendezvous goals of an ESS mission. Guidance requirements will be fulfilled by triple redundant inertial measurement units (IMU), nine vibrating wire rate sensors (three per axis), and triple redundant central computing units (CPU) with quadruple redundant memory units. Navigation updates will take place through the data uplink (two units) of the communications system and CPU. Control will be accomplished by CPU signals to triple redundant driver units of the attitude control propulsion system (ACPS), the orbital maneuvering system (OMS), and the main engine and OMS gimbal actuators.

IMU electronics are contained in two line replaceable units (LRU), the power supply, and the platform. The internal electronics of these LRU's, such as gyros, synchros, resolvers, and accelerometers, are not redundant. Each IMU and its power supply interface through an acquisition, control, and test unit to the data bus to receive update signals and provide position data. Detection of failure by the CPU will lead to removal of that particular unit from the system.

Thrust vector control (TVC) gimbal electronics are redundant to each engine, thus providing FO/FS for the TVC electronics requirement. This is based on one-engine-out capability. The electronics for control of the driver units for the OMS engines and the ACPS are redundant and, based upon engine-out capability, more than satisfy FO/FS mission requirements.

Data and Control Management (DCM) Subsystem

The DCM subsystem fulfills all of the major control and data management requirements for the ESS. These requirements fall in the four broad categories of control of GN&C, vehicle sequencing, redundancy management, and data management.

The DCM function is realized through two major classes of hardware, the central processing unit (CPU) and the acquisition, control, and test units (ACT), which is an interfacing unit, plus appropriate computer software programs.

The DCM will accept GN&C position measurements, process them through the computer, and route appropriate gimbal actuator and engine driver signals to maintain orientation and position of the ESS to schedule. The DCM will also accept updata through the communications subsystem and route and process such data in the required manner. Processing is performed independently in the "prime" and "check" computers and results are compared before action is taken.

Vehicle sequencing and self-check is carried out by the DCM per established software programs stored in the computer memory. Fault detection leads to corrective action programs by the CPU which isolates defective units and switches them out of the functional loop while switching a good replacement unit into the loop. Vehicle event sequencing proceeds per program unless changed by an update command or a fault detection. Alternative action programs are then selected.

The data management function of the DCM will transmit, upon request, the status of all units in the ESS as determined in the periodic checks. It also selects the data bus and unit addresses for command and status information.

FO/FS redundancy is provided in all major units, i.e., computers, memories, ACT, wire harnesses, down to cross-strapping of LRU's to a minimum of two ACT's. Partial internal redundancy is provided in some units such as the ACT's and some noncritical functions. Most hardware of the DCM system is derived from the space shuttle DCM hardware and will be qualified and tested to space shuttle standards. Those items which are unique to the ESS will be provided to a FO/FS redundancy level for critical mission and safety functions. Remaining hardware will be qualified to the Saturn-Apollo level of reliability. The coverage factor (ability to recover from a failure) of the DCM self-check ability is required to be greater than 99.5 percent.

Thus, the ESS DCM subsystem is a complex set of hardware and software that performs many sophisticated functions in the assurance that the ESS has a fail operational/fail safe capability for the safety of the booster crew in the launch, boost, and separation mission phases and the FO/FS capability is retained for the deorbit phase of the mission. The DCM subsystem keeps tabs on all major critical units, including its own, and switches redundant units into functional loops when required. DCM FO/FS capability has been achieved.

Communications Subsystem

The communications subsystem transmits updata information for command and GN&C, downdata for telemetry of ESS instrumentation, and range safety commands. Certain hardware items have been included as redundant units where the reliability was considered inadequate.

In the present concept, single-point failures exist in the communications link, mainly in the switching units, and will not satisfy the FO/FS criteria.

Loss of downdata is not critical; however, loss of the updata link may preclude ranging data and rendezvous commands from reaching the ESS computer which could result in serious position errors. In addition, retrofire commands for deorbiting may be lost, resulting in uncontrolled reentry of the ESS upon orbit decay.

The communications subsystem has been constructed of Apollo-Saturn hardware and will contain a high confidence factor in maintaining operational reliability.

Electrical Power and Distribution Subsystem

The electrical power and distribution subsystem is designed to meet the requirements of FO/FS. The 28 vdc main power system consists of three separate battery sources which provide FO/FS system capability. Two out of the three sources will provide sufficient power for the entire mission. In the event two sources are lost, the third source will carry the load sufficiently to a safe condition. Within each power source there are three batteries. Assuming equal distribution of loads, two of the three batteries in each source would provide sufficient power. Therefore, each of the sources could lose on battery or one source fail completely without affecting the main power system operation for the total mission.

In the ESS ac power system, two power sources for each engine will be incorporated. Each source is capable of carrying the load for that engine. In the event both sources fail, the ESS stage has one main engine-out capability for RNS and tug payloads, and after approximately 100 seconds of ESS boost burn for the MDAC payload.

Propellant Management Subsystem

The propellant management subsystem proposed for the ESS is basically the same as that used for the S-II stage. The main difference is in the capacitance sensor electronics which is much less complex and therefore should be more reliable than the propellant utilization computer used for the S-II stage.

The level monitor system is for loading and indication only and is not a critical system. However, the system for both the main tanks and the APS tanks have incorporated redundancy by utilizing a capacitance probe for analog and point sensors for discrete level monitoring.

The propellant depletion cutoff system is basically the same as that used for S-II. The two critical failure modes for the main engines are a premature engine cutoff or a failure to cut off the engine. To reduce probability of a premature cutoff, voting logic is incorporated in the system requiring at least two propellant depletion signals; in addition, the point sensor controllers are designed to give a wet indication in the event of a failure. For the main engines the propellant depletion cutoff is a backup system to the velocity cutoff system.

The APS tank's depletion OMS engine cutoff system also contains a point sensor voting logic system.

Instrumentation

The instrumentation system proposed for ESS utilizes a standard system that has been used on previous flight vehicles. Proven hardware used on the S-II program plus state-of-the-art hardware and hardware developed for the shuttle program will be used wherever practical. Wherever feasible, measurements will be converted to digital to obtain higher precision and accuracy.

Since the instrumentation system is primarily a monitoring system, redundancy has not been incorporated throughout; however, measurements that are critical to the operation of the vehicle or safety of the crew will be redundant.

Avionics Equipment and Environmental Control

To assure maximum performance and reliability, the equipment containers will be installed on the stage in areas which present the most ideal thermal and dynamic environment. System reliability is also enhanced by locating the containers to take advantage of the stage orientation during orbit, which eliminates the necessity for an equipment cooling system.

To meet the FO/FS requirements for the control of container temperatures, the heater and control systems for each container will have a dual element heater and three controllers. Each heater element will be controlled by two functionally independent thermostatic sensors and a solid-state control switch. The third control switch with its associated thermostatic sensor will be used as an emergency high-temperature safety cutoff switch.

The electrical heater and control system has been designed to eliminate any single point failures.

Avionics Container Separation

The avionics container separation system used for retrieving costly equipment operates on the same electrical/ordnance system principle as the ESS/payload separation system and utilizes identical components. These components have been qualified and utilized on both Apollo and Saturn systems; therefore, no development is required.

The critical failure mode associated with this system is the possibility of inadvertent system operation which would result in the loss of critical avionics and prevent the successful completion of the mission.

To prevent an inadvertent command or signal from activating the dispersion system, the system is designed to operate with two commands, the arm and separate commands which have to be received in the correct sequence for the separation system to be activated.

Inadvertent system operation caused by auto-ignition of the ordnance devices is another failure mode which has been considered on the existing S-II systems to be extremely remote because of the physical stimulus of high temperature or impact that is required to enable detonation to occur.

All components of this system have demonstrated very high reliability in test and operations throughout the space program.

ESS/Payload Separation System

The separation system is designed so that single-point failures in the electrical control and ordnance systems will not cause a failure of the payload to separate from the ESS vehicle. It has also been a design objective to prevent the possibility of an inadvertent premature separation.

The separation system consists of an ordnance system similar to the highly reliable ordnance system used on the S-II/S-IVB separation system. The electrical control for the ESS/payload separation system will be accomplished via the ESS computer/data bus/acquisition control system.

The critical failure modes associated with the ESS/payload separation system are the inability to separate the payload when required and premature separation.

To ensure successful operation of the separation system, two completely independent and redundant exploding bridge wire (EBW) firing units and associated ordnance devices are utilized. The electrical control of the separation system is also designed to ensure satisfactory operation by providing three separate 28-vdc power buses which will supply power for the separation system ACT units, solid-state power control switches, and EBW firing units. Each EBW firing unit will be powered from a separate bus with the third bus used as a redundant source of power for both EBW firing units. The electrical control system also contains three ACT units with two units controlling power control switches for arming and triggering their respective EBW firing units and the third ACT unit being completely redundant in arming and triggering both EBW firing units through the power control switches.

To prevent the possibility of inadvertent power application activating the system prematurely, individual solid-state control switches for arming and triggering each EBW firing unit are used. In addition, two failures are necessary in any one of the solid-state power control switches to inadvertently supply an output to the EBW firing unit.

The existing S-II separation system ordnance devices will be used in the payload separation system. Inadvertent operation or an ordnance device caused by auto-ignition has been considered on the existing S-II systems to be extremely remote because of the physical stimulus of high temperature or impact that is required to enable detonation to occur.

Propellant Dispersion System

The propellant dispersion system proposed for the ESS will be identical to the system presently used on the S-II stages except that dispersion range safety commands, if required, would not be employed until safe separation from the booster has been achieved.

Failure to disperse propellant on command is not considered a mission failure because mission failure will be imminent before the propellant dispersion command is given. The critical failure mode associated with this system is the possibility of inadvertent system operation which would result in the destruction of the stage along with the shuttle booster if attached or in the vicinity of the stage.

To prevent an inadvertent command or signal from activating the dispersion system, the system is designed to operate with two commands, the arm and dispersion commands, which have to be received in the correct time sequence for the dispersion system to be activated.

Inadvertent system operation caused by auto-ignition of the ordnance devices is another failure mode which has been considered on the existing S-II systems to be extremely remote because of the physical stimulus of high temperature or impact required to enable detonation to occur.

3.3 MAINTAINABILITY

Maintainability is defined as the design characteristics that makes possible the preservation or restoration of a functional element to its operational state with a minimum expenditure of time, personnel skills, and logistics resources, under planned maintenance environment. Maintainability characteristics will be included among the first design considerations for developing a hardware system. These characteristics include, but are not limited to, accessibility, serviceability, repairability, commonality, standardization, interchangeability, component mounting, lifting and attach points, and related operational flexibility.

The maintainability program will support the ESS systems, subsystems, and support equipment to the component level to provide standards and characteristics which effectively support the maintenance environment during the pre-operational and operational phases. This program will provide a systematic method to promote the feasibility of turnaround maintenance activities as allocated by program operational requirements, minimize maintenance manhours and subsequent training requirements, and increase vehicle and support equipment availability with resultant reduction in quantitative hardware requirements. The system will provide the criteria, methods, controls, and verification for maintainability to assure compatibility with program operational requirements.

3.4 MANUFACTURABILITY

The manufacturing processes and techniques necessary for the assembly and checkout of the expendable second stage have been developed and proven on the S-II and other NR programs.

The structural commonality existing between the ESS and the S-II permits utilization of existing capability and experience available at previous S-II suppliers and subcontractors, in conjunction with the NR shops. Present tooling will be utilized for subassembly, assembly, and checkout except for isolated areas such as the thrust structure, which will require new tooling. The integrated avionics systems, developed through the resources of the Avionics Systems Integrated Laboratory (ASIL), does not present a problem for systems installations and checkout. The fabrication and installation of the erosion barrier and ablative materials was analyzed during the S-II INT-21 study program. Full-scale models of the erosion barrier were successfully manufactured and smaller sizes were tested, utilizing existing manufacturing techniques and processes which are readily implementable.

All areas of the ESS vehicle have been thoroughly assessed and are considered to be within the state of the art.

4.0 TRANSPORTATION — SEAL BEACH TO KSC

The use of the proposed space shuttle static firing fixture at KSC will eliminate the requirement to transport the qualification test stages to the Mississippi Test Facility which existed with the S-II. The ESS stages will be delivered directly to KSC, with the first two undergoing static firing tests.

The stages will be moved from the NASA Seal Beach facility to the Seal Beach Naval Weapons Station loading dock by means of existing S-II transporters and a bailed prime mover, and there loaded onto an AKD-type ship such as the Point Barrow used during the Apollo program, for shipment to KSC. The existing turning basin and unloading dock facility at KSC will be used to unload the stages.

APPENDIXES

APPENDIX B. ORBIT MANEUVERING SUBSYSTEM TRADE STUDY

At the conclusion of the ESS Phase A study, lunar module descent engines (LMDE) were selected for the orbit maneuvering subsystem. The LMDE was the winner in a trade study over RL-10 engines fed from the main propellant tanks and J-2 engines operating in a low-thrust mode. J-2 low thrust was considered only for an ESS utilizing J-2 engines for main propulsion. The LMDE was the winner because of lower cost and the high boiloff penalty associated with maintaining propellants in the main tanks for 24 hours.

Another tradeoff was conducted at the initiation of Phase B as two other options were now available:

1. The space shuttle orbiter engines had been selected over J-2 engines for main propulsion. This dictated the use of a new thrust cone and allowed adequate space for separate cryogenic tankage for the OMS. With the basic S-II thrust cone with J-2 engines there is not enough space for separate cryogenic tankage.

2. The space shuttle baseline recommends development of a 10,000-lb thrust OMS engine. The engine satisfies ESS requirements for orbit maneuvering.

The trade study was performed with three options:

1. LMDE (Phase A Baseline)

2. RL-10 engines with separate tankage

3. Space shuttle orbiter OMS engine with separate tankage

The tradeoff was made based on cost and performance. The space shuttle orbiter OMS engine was the trade study winner. It showed both lower cost and higher performance than the RL-10 and LMDE. The trade table is presented by Table B-1. The primary reasons for the lower cost and improved performance with the space shuttle orbiter OMS engine are:

1. The Space shuttle program will develop the engine. With the RL-10 engine the ESS program would have to pay for significant development costs as the RL-10 would have to be upgraded for the ESS. With the LMDE the ESS would have to pay startup costs as the ESS would be the sole user of the engine.

2. The space shuttle OMS engine has a slightly higher I_{sp} than the RL-10 and a significantly higher I_{sp} than the LMDE.

The space shuttle OMS engine has been selected for the ESS OMS and is included in all ESS performance and design data.

It is noted that should the Space Shuttle Program change the engine for orbit maneuvering, it is likely that the same engine would be the choice for the ESS. The aforementioned development/startup cost rationale would apply.

Table B-1. Cost/Performance Consideration in Design—OMS Trade (Millions of Dollars)

PROGRAM: 2/YR FOR 10 YRS

	BASELINE SYSTEM (LEM D)		ALTER NO 1 RL 10 OMS (OMS)		ALTER NO 10K SHUTTLE OMS (OMS)		ALTER NO 4		ALTER NO 5	
	SNR	SR	SNR	SR	SNR	SR	SNR	SR	SNR	SR
1 0 EXPENDABLE SECOND STAGE	77 9	518 4	105 6	515 4	73 2	515 4				
1.1 STRUCTURE	9 2	111 5		-						
1 2 PROPULSION	13 9	88 0	39 4	86 5	11 9	86 5				
1 3 AVIONICS	10 5	117 8			-					
1 4 VEHICLE SUPPORT	1 3	11 4	1 4		1 2					
1 5 MECHANICAL			-		-					
1 6 VEH ASSEM INTEG & C/O	5 7	59 0	6 9	58 4	5 2	58 4				
1 7 COMBINED SUBSYSTEM TESTING	0 94	6 0	-	-	-					
1 8 SYS ENGRG	13 9	44 2	14 0	43 9	13 0	41 9				
1 9 FACILITIES	1 4	17 1	-	-						
1 10 SYSTEM SUPPORT	14 2	44 3	14 9	43 8	13 0	43 8				
1 11 VEHICLE MGMT	0 4	5 1	0 5	5 0		5 0				
1 12 MODELS & MOCKUPS	0 5		-	-						
1 13 PAYLOAD INTEG	-									
1 14 TRANSPORTATION		14 0		-	-					
2 0 MAIN ENGINES (ESS)	-	24 60	-		-	-				
3 0 BOOSTER MODIFICATIONS	-	-		-	-					
4 0 FLIGHT TEST	-									
5 0 OPERATIONS	-	87 44		87 3	-	87 0				
6 0 MANAGEMENT & INTEG	5 11	38 03	6 77	37 84	4 82	37 87				
7 0 SEP/SUPPORT STRUCTURE	7 27	3 44	-	-	-	-				
SUB TOTAL	90 28	671 91	119 64	668 58	85 29	668 27				
TOTAL	762 19		788 27		753 56					
PERFORMANCE (270 X 55°) LB	177 300		185 045		185 905					
RECURRING COST/POUND PAYLOAD $/LB	189		181		180					
RECURRING COST/FLIGHT	33 6		33 4		33 4					

APPENDIX A. ENGINE ACTUATION TRADE STUDY

A tradeoff analysis was performed to select the type of engine actuation subsystem to be utilized on the ESS. The candidates were:

1. Engine shaft drive - hydraulic system
2. Electrical drive - hydraulic system
3. Pneumatic drive - pneumatic system
4. Pneumatic drive - hydraulic system
5. Auxiliary power unit drive - hydraulic system

The trade table is shown by Table A-1. The engine shaft drive was the trade winner and has been recommended for the ESS baseline. It should be noted that this affects the shuttle engine ICD 13M15000B as the SSE does not currently have an accessory drive shaft. In the event the engine shaft drive is not approved for incorporation, a pneumatic driven hydraulic system would be the alternative.

Table A-1. Cost/Performance Consideration in Design EAS Selection (Millions of Dollars)

NO	WBS ELEMENT	ENG DRIVE HYDRAULIC		ELEC DRIVE HYDRAULIC		PNEUMATIC ACTUATOR		PNEU/HYDR CONVERSION		APU DRIVE HYDRAULIC	
		NR	R	NR	R	NR	R	NR	R	NR	R
1 1	STRUCTURES	9 2	111 5	9 2	111 5	9 2	111 5	9 2	111 5	9 2	111 5
1 2	PROPULSION	11 9	86 5	12 4	86 7	13 9	86 7	12 4	87 6	11 9	86 7
1 3	AVIONICS	9 8	98 9	10 8	100 1	9 8	98 9	8 8	98 9	5 8	98 9
1 4	VEHICLE SUP	1 2	11 4	1 7	12 8	1 7	12 8	1 7	12 8	13 2	14 3
1 5	MECHANICAL	0	0	0	0	0	0	0	0	0	0
1 6	ASSEM INTEG & C/O	5 1	55 8	5 1	56 2	5 3	55 7	5 2	55 9	5 6	56 7
1 7	COMB SUBSYS TEST	6 9	6 0	7 0	6 0	8 1	6 0	7 3	6 0	13 8	6 0
1 8	SYST ENGRG	12 9	42 1	13 2	42 3	13 8	42 1	13 2	42 1	13 8	42 4
1 9	FACILITIES	1 4	17 1	1 4	17 1	1 4	17 1	1 4	17 1	1 4	17 1
1 10	SYSTEM SUP	12 9	41 9	13 2	41 9	13 7	41 9	13 2	41 9	14 6	41 9
1 11	VEHIC MGT	0 4	5 0	0 4	5 0	0 4	5 0	0 4	5 0	0 4	5 0
1 12	MODULES/MOCKUP	0 5	0	0 5	0	0 5	0	0 5	0	0 5	0
1 13	PAYLOAD INTEGRATION	0	0	0	0	0	0	0	0	0	0
1 14	TRANSPORTATION	0	14 0	0	14 0	0	14 0	0	14 0	0	14 0
	1 0 SUBTOTAL ESS	72 2	490 2	74 9	493 6	77 8	491 7	74 3	492 8	94 2	494 5
2 0	MAIN ENGINES	* 0	* 24 6	0	24 6	0	24 6	0	24 6	0	24 6
3 0	BOOSTER MOD	0	0	0	0	0	0	0	0	0	0
4 0	FLIGHT TEST	0	0	0	0	0	0	0	0	0	0
5 0	OPERATIONS	0	86 6	0	86 6	0	86 6	0	86 6	0	86 6
6 0	MANAGEMENT & INTEGRATION	4 8	36 3	4 8	36 3	4 8	36 3	4 8	36 3	4 8	36 3
7 0	SEPARATION,SUPPT STRUCT	7 3	3 4	7 3	3 4	7 3	3 4	7 3	3 4	7 3	3 4
	TOTAL	*84 3	*641 1	87 0	644 5	89 9	642 6	86 4	643 7	106 3	645 4
	PERFORMANCE (DRM)	183 000		182,130		182 885		182 750		182,685	
	RECUR COST $/LB PAYLOAD TO DRM	*175 16		176 93		175 76		176 11		176 63	
	RECUR COST PER FLIGHT	*32 055		32 225		32 130		32 185		32 270	

(*DOES NOT INCLUDE POTENTIAL ENGINE ΔCOST)

APPENDIX C. DEORBIT TRADE STUDY

Table C-1 presents the ESS deorbit cost comparison for four candidate systems. The candidate systems traded were the ESS OMS, separate system orbiter-delivered, separate system on ESS and the shuttle orbiter. The ESS OMS was concluded to be the most cost effective for deorbit. A deorbit sequence utilizing the shuttle orbiter is shown in Figure C-1.

Table C-1. ESS Deorbit Cost Comparison
(Millions of Dollars - 20 stages), ΔV = 550 fps

Cost Element	Candidate						
	ESS OMS		Separate System Orbiter Delivered		Separate System on ESS		Shuttle Orbiter
	$R	$NR	$R	$NR	$R	$NR	
Propellant	Negl	-	-	-	-	-	Deorbit loads exceed present orbiter docking interface load carrying capability.
Payload penalty*	13.4	-	1.2	-	37.2	-	
Engines	12.3	0.5	14.0	1.1	14.0	1.1	
Structure	-	-	4.0	1.1	4.0	1.1	
Orbit delivery*	-	-	38.4	-	-	-	
Subtotal	25.7	-	57.6	2.2	55.2	2.2	
Total	26.2		59.8		57.4		

*Based on $200/lb

Conclusion: ESS orbit maneuvering system will be used for deorbit.

The ESS OMS is a system available on the vehicle and results in the lowest total cost, 26.2 million dollars based on 20 stages. The major cost elements are the payload penalty resulting from the 3365 pounds of additional propellant that must be loaded for the deorbit burn and the loss of the OMS thrust chambers which cannot be recovered. No structural changes are required, and the avionics changes necessary are mininal.

The separate system orbiter-delivered consists of 12 solid propellant motors mounted in either the forward or aft skirt. The total cost of this system would be 59.8 million dollars (20 stages), primarily because of the cost (38.4 million) associated with delivery of the solid motors weighing 9600 pounds and the cost of the motors (14.0 million). The orbit installation associated with this system would be an undesirable feature.

The separate system on ESS also consists of 12 solid propellant motors installed prior to launch. The total cost of this system would be 57.4 million dollars (20 stages). The major cost elements are the payload penalty resulting from orbiting the system weight of 9300 pounds (37.2 million) and the cost of the motors (14.0 million). The solid motors would require environmental protection during boost and orbit.

The shuttle orbiter was also considered for deorbit but was found to be unacceptable because the deorbit loads exceed the present orbiter docking interface load-carrying capability. An additional 11,000 pounds of orbiter propellant would also be required to deorbit the ESS.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

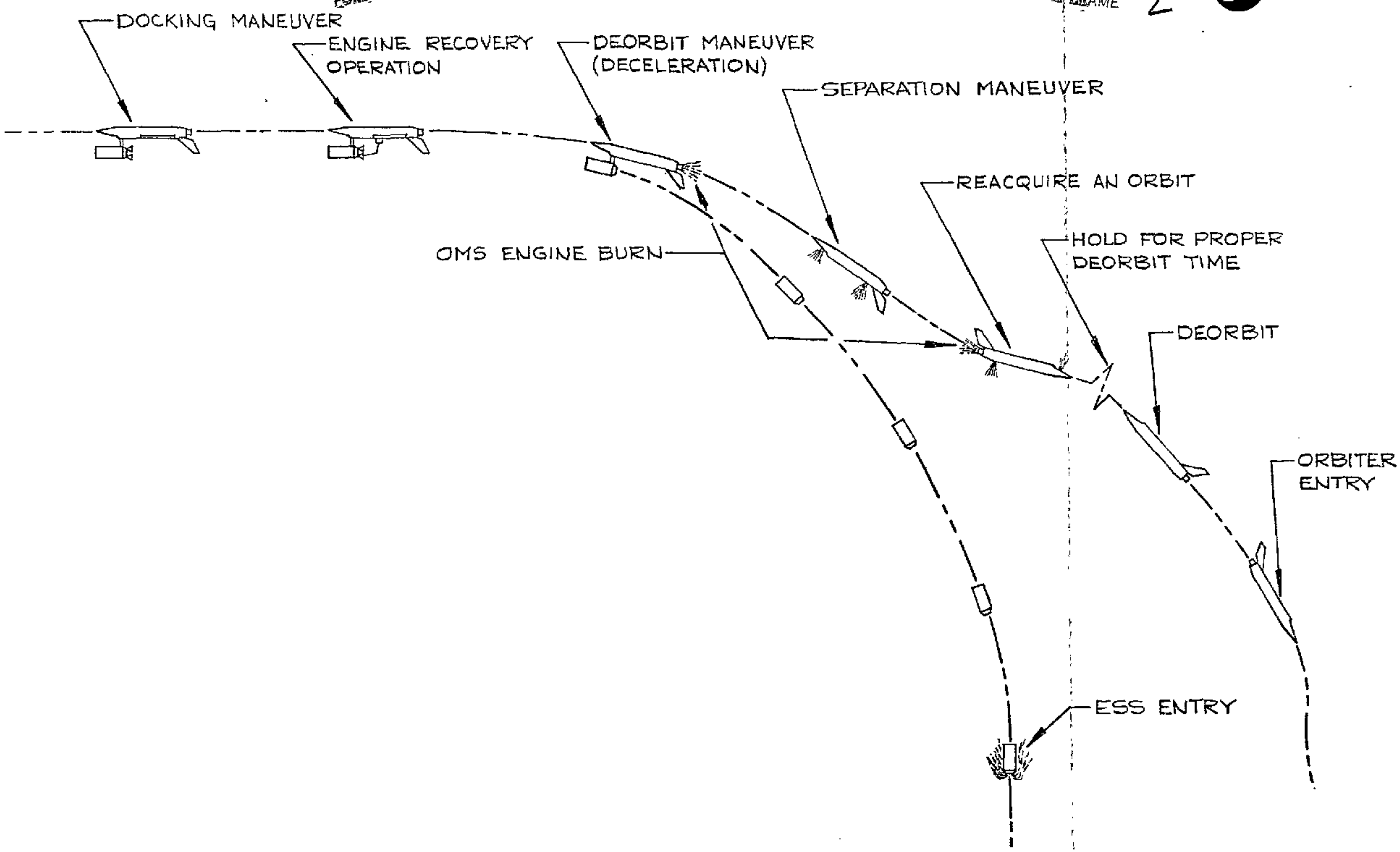


Figure C-1. ESS Deorbit Sequence (Utilizing Shuttle Orbiter)

APPENDIX D. RECOVERABLE AVIONICS TRADE STUDIES

To establish the feasibility and economic advantages of recovering avionics equipment, it was necessary to perform a variety of trade studies. This section contains results of these studies.

ASTRIONICS RECOVERY ANALYSIS

The first study conducted was to determine the economic advantages of recovering avionics equipment and to establish which avionics components could be recovered and still have all the objectives of the mission achieved. Tables D-1 through D-4 and Figure D-1 contain the factors considered and options and cost considerations used to establish the approach to be used in the recovery of avionics equipment. Study results indicate that it is most cost effective if all possible equipment is recovered and additional equipment provided to perform the functions required for vehicle deorbit.

Table D-1. Avionics Recovery Analysis

Considerations	Must have a deorbit system Equipment for deorbit shall be FO/FS at liftoff Pitch and yaw deorbit errors: ±5 degrees ±7.5 percent of 550 fps Limited recovery time (24 hr life) Deorbit done with OMS ΔV, ACPS for initial altitude
Recovery Options	No recovery of avionics system Recover in part; leave part for deorbit Total recovery; build separate deorbit package

Table D-2. Deorbit Considerations

Deorbit Considerations	Option I No Recovery Leave: 4 computers 12 memorys 9 rate packages 3 IMU's	Option II Partial Recovery Leave: 4 computers 12 memorys 9 rate packages	Option III Maximum Recovery Leave: 9 rate packages (Need separate deorbit packages)
Mission timeline impact	No +	Yes -	Yes -
Software changes for recoverability	No +	Yes -	Yes -
Development of recovery methods	No +	Yes -	Yes -
Environmental affect on removed hardware	No +	Yes -	Yes -
Added failure modes due to recovery	No +	Yes -	Yes -
Added checkout requirements	No +	Yes -	Yes -
Refurbishment costs	No +	Yes -	Yes -
Cost of new hardware for deorbit	No +	Yes -	Yes -
Cost for making hardware recoverable Installation, packaging, and safing	No +	Yes -	Yes -
Shuttle support for deorbit	No +	Yes -	Yes -
Basic hardware cost savings	No (-)	Yes (+)	Yes (+)

Table D-3. Recoverable Hardware Costs

Item	Quantity	Option II (Partial)	Option III (Maximum Recovery)
Recoverable Hardware			
IMU	3	$900K	$ 900K
Computer and memory sets	3		$1,320K
Value of recovered hardware		$900K	$2,220K
Refurbishment costs (≈40 percent)		$360K	$ 888K
Installation and checkout		$150K	387K
Added deorbit hardware costs			$ 600K
Delta packaging costs		$240K	300K
Total savings		$750K	$2,175K

Table D-4. Recoverable Avionics Conclusion

No recovery: build 20 sets (≈7.22M each) - $44.4M

Savings from recovery (Option III)

Build five sets	≈	$\$11.1\overline{M}$
Installation and checkout (15 sets)	≈	$1.6\overline{M}$
Refurbishment cost (15 sets)	≈	$11.1\overline{M}$
Delta packaging cost	≈	$6.0\overline{M}$
Deorbit kit (20 sets)		12.0M
		$41.8M ⇨ $2.6M SAVED

Saving does not include

- Software changes to adopt to recoverability
- Development of recovery techniques
- Mission timeline constraints
- Unknown effect of environment on removed avionics.
- Added failure modes for removal hardware
- Uses shuttle for attitude orientation for deorbit

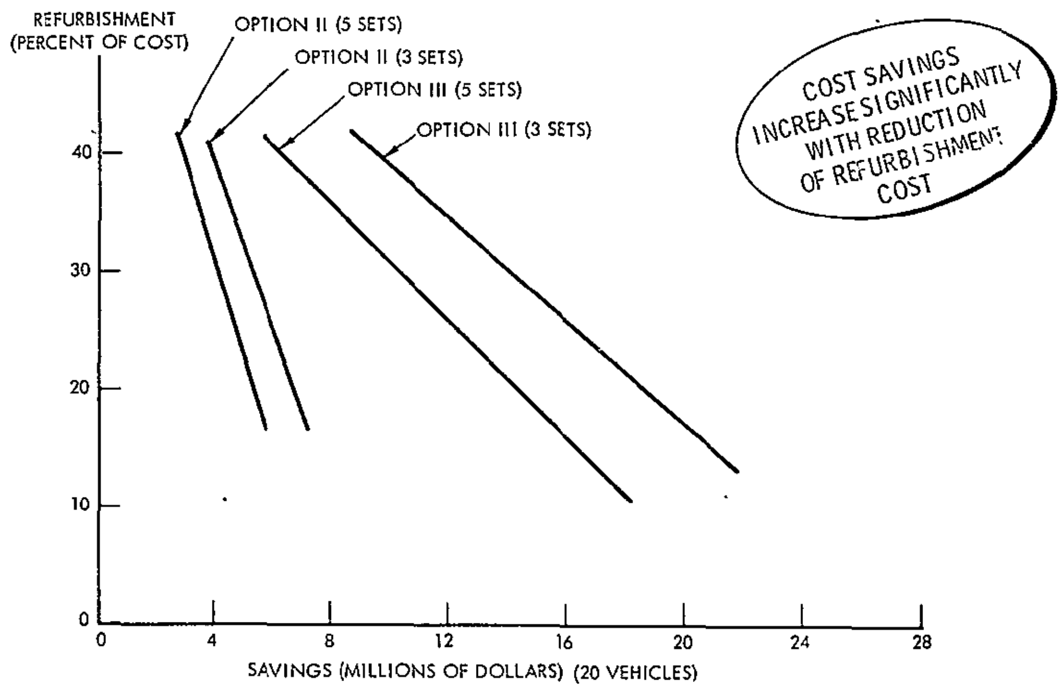


Figure D-1. Recoverable Avionics Cost Savings

DEORBIT SUBSYSTEM CONFIGURATION

Before the baseline configuration of the deorbit was established, various configurations were considered. The following is a description of these alternatives and the rationale used to formulate the decision concerning each configuration.

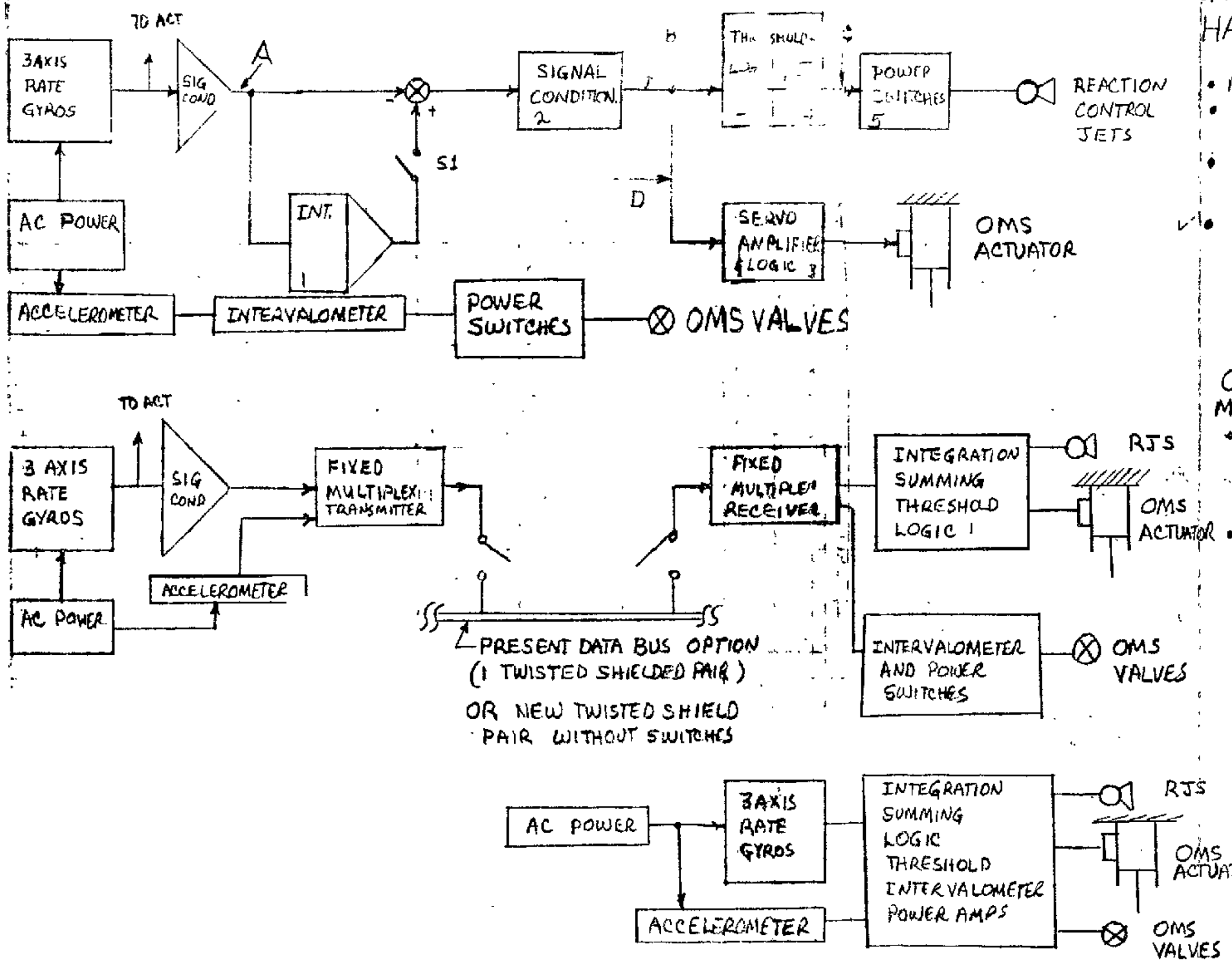
Figure D-2 depicts the various approaches considered for the method of transmitting data to be implemented for the deorbit equipment. Table D-5 depicts the various techniques considered for the implementation of a separate deorbit control capability.

The considerations contained in Table D-5 and Figure D-2 ranged from using the existing data bus system with a degraded memory for the CPU to a separate deorbit package which interfaced with the ESS subsystems downstream of the data bus.

The separate hardwire deorbit package option was adopted as the baseline for the ESS. The deciding factors were centered around the fact that more positive inhibit techniques could be implemented with the hardwire system. This method also provides for the maximum retrieval of avionic equipment. Another significant factor was that most of the components required for this subsystem could be considered as off the shelf from other programs.

Table D-5. System Configuration for Deorbit

Candidate I	Hardwire system which interfaces with ACPS, OMS, and propellant feed subsystems downstream of the ACT units Components Two-axis position gyro Three-axis rate gyros Accelerometer Drive electronics Sequencer and TLM decoder This system would ride in an inhibited/caged state until just prior to ESS/orbiter undock. The enable function would occur after the orbiter has positioned the ESS to the proper deorbit attitude.
Candidate II	A system that contains a special-purpose computer that communicates with the required ACT units only in a sequential manner through the data bus. The deorbit inertial elements would provide inputs to this computer. TLM interface would still be required for initiation.
Candidate III	A system that left the main CPU/IO equipment and recovered only the main memory modules. An additional small inexpensive memory was provided to perform the deorbit functions. This system provided for recovery of the main memory modules, which represent a majority of the computer complex cost. This system was not adopted primarily because of the constraints it placed upon the existing shuttle hardware.
Candidate IV	A system that employees FM modulation on the data bus at carrier frequencies that do not enter the bandwidth used in the primary data bus systems (1 MHz).
Candidates II, III, and IV	These candidates were not considered feasible for adoption because a hardwire system with built-in inhibits was more suitable to the ESS requirements.



CANDIDATE I
HARDWIRED BUS BY-PASS

- MAIN WIRE RUN AT A, B, C, OR D
- 8 TWISTED SHIELDED PAIRS FOR OMS (POINT D)
- 16 TWISTED SHIELDED PAIRS FOR RJS (POINT C)
- 3 TWISTED SHIELDED PAIRS FOR EITHER AT POINT A.

CANDIDATE II
MULTIPLEX BUS BYPASS

- ADDED MULTIPLEXERS AND SWITCHES TO USE PRESENT BUS.
- NEW BUS OBVIATES NEED FOR SWITCHES

CANDIDATE III

- RELOCATED RATE GYROS
- NO USE OF DATA BUS
- NO LONG WIRE RUNS
- ALL HARDWIRED

Figure D-2. Data Transmission Alternatives