annual report to the nasa administrator by the aerospace safety advisory panel

part II–space shuttle program

section 2–summary of information developed in the panel's fact–finding activities

june 1975

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ANNUAL REPORT TO THE NASA ADMINISTRATOR

by the

AEROSPACE SAFETY ADVISORY PANEL

PART II - SPACE SHUTTLE PROGRAM

Section 2 - Summary of Information Developed in the Panel's Fact-Finding Activities

June 1975

PREFACE

Section I provides a summary of the Panel's observations and conclusions on the Space Shuttle Program.

Section II summarizes the information developed during the Panel's inspection activities since our last report on the Shuttle program. The criteria for inclusion of information in this volume is its relevance for a safe and successful mission. This section is organized in a manner that points up the management areas and the individual elements of the Shuttle system providing a summary of the basic management or design approach including the most obvious limits or hazards that are significant to crew safety. It also provides the status of the situation with particular attention to the current resolution of those hazards.

We hope the report will be of assistance to those in the Shuttle Program as a checklist to assure that the right questions continue to be asked at the right time. But the report is also written for a larger readership to assist them in understanding this complex program and its more salient details.

CONTENTS

1.0 INTRODUCTION

1.1 Purpose

This section, Section II, provides a summary of the information developed during our inspection activities and in a detailed review of documentation used in the Space Shuttle program. Its intent is to provide the reader with an idea of the data examined by the Panel and a description of the program at this time. Another purpose is to provide specific background information and supporting details to augment the data provided in "Section I - Panel's Observations and Conclusions." In addition this material will be utilized by the Panel in further reviews during the coming year as a baseline and reference manual.

1.2 Scope

The structure of this volume follows the basic organization of Section I. It extends the coverage of the Shuttle elements to include those specific subsystems considered critical to crew safety. This volume also discusses such technical management areas as systems integration test program planning. It also covers such specific crew safety areas as the Orbiter Thermal Protection System, safety and reliability efforts on so-called secondary structure, and lightning protection. Such a compilation of data is necessarily a compromise between detail and brevity and this accounts for the numerous figures and tables used in this volume.

2.0 SHUTTLE PROGRAM MANAGEMENT

2.1 Technical Management System

A management overview was provided in the Panel's annual report dated March 1974. The material provided at that time is still valid and need not be repeated here. Our emphasis has been on those aspects of technical management that support and control Shuttle requirements and design, hazard identification, resolution or acceptance of risks, and the safety implications of test planning. With this in mind the Panel focused on the following specific areas: (1) the review system to establish and assure implementation of design requirements and concepts, (2) management of the development of the Orbiter Thermal Protection System and Space Shuttle Main Engine Electronic Controller, (3) integration management applied to the element interfaces and the risk management itself, and (4) special management approaches developed to meet special program needs. To maintain the brevity of this report only the key data developed by the Panel are presented here.

2.1.1 Orbiter Thermal Protection System

Management of the Orbiter Thermal Protection System (TPS) within the total Shuttle system framework must account for the many technical and scientific disciplines and interfaces which affect the requirements, design, fabrication and verification of the operational hardware. The disciplines and interfaces, or elements, of TPS management include the following:

- Disciplines
  - Aerodynamics and Flight Mechanics
  - Heat Transfer and Fluid Mechanics
  - Structural Design
  - Materials
  - Structural Dynamics
  - Testing and Environmental Simulation
- Interfaces
  - Structures
  - Mission Design
  - Mechanical Systems
  - Thermal Control Systems
  - Propulsion Systems
  - Ground Support Equipment
  - Prime and Subcontractors
  - NASA Element Organizations
  - Flight/Ground Test Offices
  - Flight Operations

Thus development of the TPS requires a multi-faceted NASA/Contractor management and technical organization. The TPS, as a part of the Orbiter, falls under the direction of JSC in the manner shown in Figure 1, "JSC TPS Management Organization" and in Figure 2, "JSC TPS Management Organization Detail." Overall management is under the direction and control of the Orbiter Project Manager (Level III) through the Orbiter Engineering Office. Day-to-day technical management is through two divisions of the Engineering and Development Directorate - Engineering and Analysis Division and the Structure and Mechanics Division.

All of these operations are integrated and directed by the TPS Manager who is within the Structures and Mechanics Division of the Engineering and Development Directorate. The prime contractor for the TPS is the Rockwell International Corporation who also is the prime contractor for the Orbiter vehicle. Rockwell International has, in turn, subcontracted the development and production of the TPS tiles to the Lockheed Missiles and Space Company, Space Systems Division at Sunnyvale, California. NASA has, at the same time, arranged with their own Ames Research Center and Langley Research Center for technical support.

The NASA roles in TPS development are shown below:

- Johnson Space Center
  - Requirements definition
  - Management of the Prime Contractor
  - Integration of Technology
  - Testing and Assessment of the System
  - Overall Test Program Management
  - Test Facility Development
- Ames Research Center and Langley Research Center
  - Development of New Technology (including Material Characterization)
  - Development of Test Facility
  - Technical Review and Consultation
  - Testing and Evaluation

The Contractor roles have been described as follows:

- o Rockwell International
  - o Design of the TPS high and low temperature systems
  - o Conduct of all thermostructural analyses on Orbiter
  - o Perform TPS subsystem qualification testing
  - o Provide detail drawings and other required documentation (procurement specification defining performance requirements, statement of work defining tasks, define quantity and schedule, and subcontractor change notices)
  - o Administer Subcontractor and materials procurement
  - o Conduct of periodic reviews to assure proper conduct of TPS program
  - o Define and implement installation and maintenance operations, including refurbishment and replacement at launch site
- o Lockheed Missiles and Space Co.
  - o Develop and optimize coated tiles
  - o Provide material property data on tiles and coating
  - o Demonstrate compatibility between tiles and coating
  - o Fabricate, acceptance test and deliver subsystem elements

The Preliminary Design Reviews conducted to date on Orbiters 101 and 102 and the Space Shuttle System have not fully covered the

Orbiter TPS. A detailed review is expected in mid-year 1975 to assess whether the TPS design and implementation meets Shuttle requirements.

2.1.2 Space Shuttle Main Engine Controller

The SSME Controller for each engine in conjunction with the flight control system monitors and controls the three Main Engines during the ascent portion of the Shuttle mission. The Controller also develops data on engine parameters that are used during the ground servicing cycle. The Controller depends on comparatively new technology and has a varied development history starting with the Viking program. As the result a management system has had to be developed commensurate with the technical disciplines, Shuttle interfaces, product quality assurance requirements and attendant management visibility needed to meet the demands placed upon this critical subsystem.

Marshall Space Flight Center (MSFC) is responsible for the design and development of the Space Shuttle Main Engine. The Rocketdyne Division of the Rockwell International Corporation is the prime contractor for the SSME and they in turn have a subcontract with Honeywell, Inc. for the design, fabrication, and validation of the SSME-Controller.

To summarize briefly, management and hardware development history of the Controller has not been a smooth road. Approach to the design itself was not conventional and therefore a large history/data base did not exist. As a matter of fact the packaging concept and use of plated-wire memory contributed a great deal to the initial management and technical problems. The challenge was to develop a management team and establish a management system to assure an effective approach to development and producibility and to control and resolve problems on a timely basis.

Through the diligent efforts of NASA, the Rocketdyne Division of Rockwell International and the Honeywell, Inc. organizations, the SSME-Controller program now appears to be "on the track" at this time, and the management and general controller activities are said to be "tracking close to plan, with encouraging results."

During this period of the Controller's evolution, the Panel centered on the following three questions:

(a) Have the management lessons learned on Viking been systematically reviewed and the appropriate ones incorporated in the management system for the Shuttle SSME Controller? This was based on the continuing emphasis by NASA's senior management, as well as the Panel, that lessons learned from prior programs be applied to on-going programs as appropriate.

(b) Will the plated-wire memory concept support the requirements and schedule of the SSME and Shuttle program? This was based on the knowledge that such technology represented a new and essentially high-risk technology.

(c) Based on the past history of computer development programs and the known schedule and cost problems that had arisen on this program, what are the fundamental challenges and ability of the NASA/Contractor team to resolve them in an orderly and timely fashion?

Specific comments on these areas examined by the Panel are provided below and support the previous statements concerning the SSME-Controller status at this time.

While the Panel found no single reporting format available which systematically stated the significant lessons and their disposition on the Shuttle program, the Honeywell Program Manager had his staff review the minutes and audits from the numerous Viking reviews and identify specific actions that could impact their operations on Shuttle. They then documented why those problems would not occur on their Shuttle project. To further enhance the management control of the program, the Program Manager defined a detailed work breakdown system and negotiated work/budget contracts with each major component supervisor. A problem control and resolution system was established which assigns action officers to each problem and monitors the solution as well as its timeliness. Additional technical and middle level supervision was added to the project. These people were drawn from the Martin Marietta Company and the Collins Radio Company.

Based on the Panel's experience with Apollo and Skylab, the configuration management system appears sufficiently disciplined for control of engineering and test drawings, specifications, fabrication procedures, and material processing. Production is essentially a manual buildup process at the bench. Tool control and special tools to support the manufacture and test of the components have been improved and developed where such support is needed. Standard process instructions and detailed fabrication layouts have been developed from Viking experience and with the help of MSFC to train and certify Shuttle personnel. An important lesson from Viking is the significance of anticipating production problems. Thus Honeywell established a detailed categorization of production errors so trends and corrective action can be identified early. All of these improvements have resulted in a higher degree of quality control and workmanship.

The plated-wire memory design, fabrication and validation process as described to the Panel indicates (1) there is adequate experience to date with the development of the plated-wire memory to warrant confidence at this time, (2) there does not appear to be a clear understanding of the fundamental physics associated with this type of component to assure that surprises would be anticipated and a timely course of resolution decided upon and implemented, and (3) if additional development surprises did occur, they probably could be solved by trial and error given sufficient time but that such surprises would probably impact the current tight schedule for the early pre-production controllers as well as costs.

The accomplishments of the SSME-Controller team during the past year have been significant but much has yet to be done. Close monitoring by NASA/Contractor team must be continued to assure on-time delivery of properly operating units to support the SSME engine test program and other major orbiter/system tests prior to the first orbital test flight.

Two significant problems remain at this time - Master Interconnect Board wire routing/shielding in the memory area in which noise is being coupled into the memory sense lines due to wire routing and inadequate shielding and intermittent parity errors. These problems are discussed in more detail in later sections of this report.

Technical management of the SSME-Controller software had some of the same problems as found in the Controller hardware program. Verification testing revealed numerous errors. As a result an assessment team, composed of non-Shuttle segments of the Honeywell organization, Rocketdyne, and NASA personnel was instituted. The following actions were taken as a result of the team's review:

(a) Software efforts were strengthened by adding technical personnel at Honeywell along with organizational changes at both Rocketdyne and Honeywell.

(b) Software was simplified and deliveries were phased to meet minimum Integrated System Test Bed (ISTB) test program needs.

(c) Technical management changes were made so that software is debugged prior to release for verification runs. Daily schedules and audits are used to assure knowledgeable management control. "Memory scrub groups" at Honeywell and Rocketdyne have been established to update and assure software compatibility. Such changes have enhanced the Honeywell planning efforts and contribute to a proper balance between those personnel developing the software itself and those doing the software verification.

2.1.2 Integration Management

One key to the proper allocation of resources to the total Space Shuttle program is the adequacy of the Space Shuttle element integration effort. This is an activity conducted by the JSC program office with the direct support of the Rockwell International Corporation, Space Division. All other NASA Centers and Prime Contractors involved in the Shuttle program contribute as appropriate. The ultimate responsibility for integrating the total Shuttle program is NASA's, but much of the crucial work to assure the success of this effort is accomplished by the System Contractor, Rockwell International. Consequently, the Panel asked (1) what are roles of each, (2) what tasks are being done by each and what work areas are not receiving sufficient emphasis, (3) are there congruent expectations among the many elements of the program regarding system integration, and (4) what is the degree of communication among those involved and management's sensitivity to the problems inherant in the continuing integration effort?

In its Annual Report for 1973 the Panel discussed this area and received a response as shown in Section 7.3 of this volume. This dealt with the results of Rockwell's effort to separate their integration task from the Orbiter task, and with the increase in tasks assigned to Rockwell International as the "System Contractor."

2.1.3.1 NASA

The Space Shuttle program organization centers its integration effort in the Systems Integration Office within the Space Shuttle Program Manager's office at JSC. This is the Level II operation and is also the "lead center" on the program. The responsibilities of this Systems Integration Office are:

(a) Review, control and manage the systems integration activities for the Shuttle program.

(b) Manage the design, development, test and engineering for the Shuttle carrier aircraft project.

The functions carried out by this office are shown in Table I. "Detailed Program Inter-relationships" are spelled out in the current issue of Volume II of the JSC 07700 Level II program definition and requirements documents.

The JSC Systems Integration Office has on-site representatives from Marshall Space Flight Center, Office of Aeronautics and Space Technology (NASA Headquarters), JSC's Engineering and Development Division, Shuttle Carrier Aircraft Project Office, and the Kennedy

Space Center. There are three major sub-groups in this office - Systems Engineering, Technical Integration, and Test and Ground Operations. These functions at JSC are staffed by approximately 100 Civil Service people (35 JSC program office, 15 co-located from KSC and MSFC, 50 Engineering and Development).

The necessary coordination in support of the specific tasks to achieve true Shuttle system integration uses many of the methods developed on Apollo and Skylab programs. Informal and formal channels are used freely, but controlled by the program and element project managers. The more formalized review system is a definite part of the integration effort as always and is discussed in a later section of this report.

Of particular significance are the more than 30 formalized panels and working groups working on a day-to-day basis. They encompass all programmatic areas and are composed of NASA, contractor, and USAF personnel. The Panels are established as a continuous entity to cover specific technical and technical management regimes. Working groups are established to meet a specific technical task that requires timely resolution and which is terminated once that problem is resolved. A list of the Panels and Working Groups is provided in Table II.

Areas of coordination/integration, that fall between the Panel type operation and the review system, are the System Integration Reviews (SIR's) and the Computer System Integration Reviews (CSIR's).

Their purpose is to review, control, and manage the systems integration activities. These activities include (1) integration contractor system tasks, (2) element contractor system tasks, and (3) NASA system tasks which are conducted at both Headquarters and Centers.

Approximately every three weeks this group meets, basically through tele-conference methods, to take up the many systems' problems given to them for their resolution. As stated at a recent Preliminary Design Review ... "Where more clout is needed to achieve resolution of baseline data it goes to the Systems Integration Review Panel (SIR)." Here is an example of the material handled by the SIR. A question was raised during the Shuttle Systems Preliminary Design Review (March 1975) concerning the lack of data to assure that the proper hardware and proper facilities are available to conduct development and verification of the ascent flight control system. Rockwell was directed to prepare a presentation to SIR with recommendations on meeting the required depth of documentation in the Master Verification Plan, Volume II - "Combined Elements Verification - Ascent Flight Control."

Another example of integrated technical management is shown in the KSC/MSFC "Memorandum of Understanding For Shuttle External Tank and Solid Rocket Booster Support Equipment." This document is included in Section 7.4 of this volume.

2.1.3.2 <u>System Contractor</u>

System Integration and Shuttle Orbiter efforts are both conducted under the same NASA contract number. However, separate cost, budget, schedules, and work authorizations are used. Both the Shuttle Orbiter and Shuttle System Integration Program Managers (they are Rockwell International Space Division Vice-Presidents) report to the Space Division President; thus both have equal stature and authority. The System Contractor's role, as described to the Panel, is quite broad. It is spread over four increments of time:

(a) Initial increment covers the period during which basic requirements must be adequately defined and the design approach mature enough to proceed with detailed design, i.e., through completion of the Shuttle System Preliminary Design Review.

(b) Record period proceeds from the end of the above increment through the Critical Design Review and the completion of the design, development, test and engineering effort. This increment extends through the first year or so of flight to assure that the Shuttle system is safe, reliable, and capable of meeting the operational missions.

(c) Third increment includes production and upgrade/retrofit of vehicles for operational use.

(d) Fourth increment is the operational phase.

Rockwell International has the equivalent of approximately 420 persons on their system integration effort. There are some 8 dedicated full-time staff people in the Shuttle Integration Office and 35 persons located on the staff of the Vice-President for Engineering (functional support) dedicated to the integration effort. The remaining personnel are putting effort into integration as required along with their basic work on the Shuttle Orbiter contract. On the whole, then, personnel are essentially borrowed from functional organizations as required. Rockwell supports JSC, Level II, operations in many areas as shown by task assignments in Table III.

Some of the more significant areas being worked on include integrated vehicle analyses such as:

(a) Induced environment definition

(b) Ascent performance optimization

(c) POGO test and analysis

(d) Element separation requirements

(e) Ice-frost prevention

(f) EMC/Lightning protection analysis and requirements

(g) Sneak circuit analysis

(h) They also work on the integrated schematics which provide end-to-end visibility of the functional relationships of all components in a system, and as such provide evidence of integration of all subsystems, e.g. electrical, electronic, fluid, mechanical, etc.

An area of particular interest to the Panel was the system safety activities conducted by the System Contractor. These include safety requirements, program/project reviews, system-level trades, system-level hazard analyses, and test/operations safety. One of the many examples of their work provided to the Panel was the development of a fire/toxicity protection plan and its application across the Shuttle program. The single source document for the Orbiter is SD 74-SH-0223. It was prepared for the designer to use as the medium for achievement of fire/toxicity safety. This document was forwarded to the other element contractors as an example of inputs requested for development of total Shuttle requirements.

Based on the material presented and the discussions conducted during the period of examination, it appears to the Panel that the Rockwell International Space Division has more of a support role to JSC than an independent system integration role. Rockwell International is satisfied with this role. This is not unlike the experience of the Integration Contractor on the Skylab program some years back. On the whole this resulted in an operational mode where the contractor had the opportunity to effectively highlight integration problems but not the responsibility of controlling the activities of other contractors. There has been an obvious effort to separate the Integration and Orbiter efforts at Rockwell International and yet retain the valuable abilities being applied to the Orbiter for use on the integration effort. Advantages are as obvious as the drawbacks e.g., assurance of a knowledgeable but independent check and balance. There appears to be no real problems in making this arrangement work to the advantage of the total program, but sustained attention should be paid to making sure that it does so.

2.1.4 Special Management Items

In any program of this size there are bound to be exceptions to the rule in management techniques because of exceptional conditions of one kind or another. The Solid Rocket Booster project differs from the other Shuttle elements in that MSFC itself is the prime contractor rather than an industrial contractor. Marshall has contracted for the Solid Rocket Motor (SRM) with the Thiokol Corporation (Wasatch Division) while maintaining its in-house responsibilities for the design of the total SRB and the assembly of the total SRB. The major question asked by the Panel with regard to the technical management of the Solid Rocket Booster was "Where would the check and balance function come from that normally exists between NASA Centers and their prime contractors?"

The SRB Project Manager is responsible to the MSFC Shuttle Projects Manager and is subject to the Level II integration controls exerted by JSC as the overall Shuttle manager. Program requirement documents and reporting systems are placed upon the SRB organization just as they are on any prime contractor except that NASA does not have the intermediate step of contracting documentation. On the whole there appears to be at least as great a control and checks and balances on the SRB effort conducted by Marshall as on any other Shuttle element. This is supported by the existence of a special SRB Review Office within the JSC Program Office and the strict adherence to configuration management systems by the MSFC personnel.

The NASA Shuttle Organization conducted a Program Requirements Review during the latter part of 1974 designed to realign the Shuttle program with the available budgets and desired scheduling of activities to meet the needs of the design, development, test and evaluation program. The events in this activity included:

(a) Definition of possible candidates to be delayed, modified, consolidated or deleted. Candidate items involved production, spares, ground support equipment, facilities, test program, operational program, technical management details, training and simulation work.

(b) Thorough review of all the possibilities and their impacts and value (cost effectiveness). Those deemed most worthwhile were presented to NASA Management and they decided whether to accept, reject, or hold these possibilities open for later review.

Twenty-eight items were selected and are being implemented. The Panel's interest centered on any safety impacts caused by these program changes. Typical of the Panel's concern were in (1) deletion of the runway barrier at KSC, (2) the large number of adjustments made to the test program (about 39% of the total) particularly those dealing with vibration and structural testing, and (3) reduction in ground support equipment particularly at the flight test sites.

Program management has assured the Panel that each change received will continue to receive a safety review to ascertain any adverse impacts and to bring them to the attention of the program management. The Panel intends to continue to examine this area to assure compliance with NASA Shuttle Management's intent.

The Orbiter/System Integration contractor's organization includes a staff member covering the Shuttle/USAF B-1 Interface. He reports directly to the President of Rockwell's Space Division. This coverage is useful to both the Shuttle and B-1 programs because of the transfer of both technological and management know-how. As an example, the basic landing gear system design for the Orbiter takes advantage of that developed for the B-1. The Shuttle aft thrust structure is made of titanium/boron epoxy reinforcement and the payload doors use graphite epoxy honeycomb. These are extensions of the B-1 developments.

2.2 Organization

The previous Panel Annual Report described the organization and general management system which has not changed to any great degree since then. Significant changes have been noted in Section 2.1.2 "Space Shuttle Main Engine Controller." Personnel changes were made at the Rocketdyne Division. As noted in Section 2.1.4 "Special Management

Items" during the DDT&E phase of the Shuttle program, the Marshall Space Flight Center has been assigned the responsibility for the integration management of the SRB. It is planned to contract-out for the SRB assembly contractor in Fiscal Year 1977. This assembly contractor will then have the prime contractor's role and responsibilities for the total assembly of the SRB. It is expected that this contractor will be located as near as practical to the launch site operational base.

The contractor team is being augmented as required to meet the maturing design and fabrication posture of the Shuttle elements. The principal contractors and subcontractors are listed in Section 7. 5 of this volume.

The Panel visited NASA Centers and a number of contractors during the period since the last Panel report and for the first time examined the KSC role in the Shuttle program. Because the KSC role for Shuttle differs from that on Apollo, Skylab, ASTP and unmanned space systems, it is discussed here. On previous programs KSC received, assembled, checked out and launched the vehicles by providing basic facilities and support equipment such as the Vehicle Assembly Building, launch control center, launch pads, checkout areas, and launch support ground equipment such as the propellant loading systems, gas systems and environmental control systems. The KSC role in Shuttle is more complex.

KSC has responsibilities for receiving inspection and control, assembly, checkout, and launch on Shuttle as on previous programs. However, in addition they will have responsibility for recovery and retrieval operations for the Orbiter and the Solid Rocket Boosters. This is completely new.

Ground operations similar to previous programs include the sustaining engineering effort, logistics and maintainability. However, the "turnaround" operations to prepare the Orbiters for flight is again completely new.

Basic facilities built for prior manned and unmanned programs will be used with appropriate modification. In addition, the following new facilities and associated ground support equipment will be required: runway and taxi areas, Orbiter Processing Facility, a highly automated launch processing system to preclude errors and speed up the turnaround time, and payload preparation areas.

KSC will also provide support to the NASA Flight Research Center and later on to the Air Force's Western Test Range operation.

As presented to the Panel at the time of its inspection trip to KSC, the KSC Shuttle organization has been fully defined to meet known program requirements and the management control systems have been developed and are being implemented. KSC manages its Shuttle work force through manpower work packages which identify discreet work activities in terms of product and required manpower. These serve as contracts between operating elements, project managers and the Center management.

The many organizations involved in the design, development, fabrication, and testing of the Shuttle elements and the combined system appear to be in place and manned in a manner commensurate with the cost, schedule and performance requirements and expectations. Those changes in organization necessitated by program maturity and directed changes will be examined as required to assure that there is no detrimental impact on ground and flight safety.

2.3 Review System

The Shuttle program review system is a direct descendent of those systems used on Apollo, Skylab and ASTP programs. To hold down costs there is an increasing use of the teleconference method of conducting meetings and reviews.

In reality the Shuttle program review is a continuous process occurring on a daily, weekly and monthly basis at all levels of the program from the drafting boards to the program management. Periodically a major management control function is inserted into the system in the form of a detailed formalized review. These provide a means of determining program progress, problems, problem resolution, and approving the current program posture as a sound basis for continuing to the next program milestone.

The review system can be examined from the point of view of the total NASA Shuttle Program down through each succeeding level of management and/or hardware. Within the overall review system there are so many different vehicles used to conduct reviews that it is possible here to examine only those which the Panel has had the most direct dealings: Systems Requirements Review, Preliminary Design Reviews, and special reviews. The many other on-going reviews include the Element Quarterly Technical Reviews, Systems Integration Review (Panel-SIR), weekly and biweekly configuration control boards at each level of the program (some of these are referred to as the CCB, PRCB, etc.), and Orbiter Management Review (OMR). These illustrate the detailed management oriented review system.

As noted above the Panel's major interest was associated with those program activities that assure that requirements are properly implemented and that the hardware/software is certified as having been designed and built to the correct and safest possible configuration.

Background on these reviews follows:

(a) Purpose of the Program Requirements Review (PRR) was to review and define in detail the management techniques, procedures, agreements, etc. to be utilized by all the Shuttle program participants and the program technical requirements. This review was completed in November 1972.

(b) The System Requirements Review (SRR) updated the program and system requirements to be utilized by the contractors. Such requirements were documented as the NASA Level II baseline and placed under configuration change control. Prior to the SRR the Interface Control Documentation (ICD) responsibilities were defined as were the schedules for ICD completion to support the program. This review was completed in August 1973.

(c) Preliminary Design Reviews (PDR) covered individual Shuttle program elements as well as the overall system. These are technical reviews of the basic design approach to assure compatibility with the technical requirements and the producibility of the design approach. The PDR's result in the appropriate authorization to the contractor and in-house organizations to proceed with further design in accordance with the reviewed design approach, interface requirements, commonality items, etc., and approval or update of the Level III baseline documentation. The depth of these reviews can be decerned from the "Space Shuttle Systems Preliminary Design Review Plan" included in Section 7.6 of this report. These reviews were completed as follows:

o	Space Shuttle Approach and Landing Test	Nov. 1974
o	Space Shuttle System	Mar. 1975
o	Orbiter No. 1 (also called 101)	Feb. 1974
o	Orbiter No. 2 (also called 102)	Feb. 1975
o	Space Shuttle Main Engine	Sept. 1972
o	External Tank	Sept. 1974

- o Solid Rocket Booster Nov. 1974
- o Launch Processing System (Scheduled) Aug. 1975

Several aspects of the Preliminary Design Reviews are of interest because they show the PDR as a real-life, real-time management control device as a part of the "building block" approach used in arriving at an operational system within budget and schedule. Each Element (Orbiter, SSME, etc.) Preliminary Design Review was built on a series of prior reviews which generally included Project Manager's reviews, weekly meetings and program/project periodic reviews used for visibility and control of the project. The "building block" approach resulted in the Shuttle Systems PDR being built on the individual Element PDR's.

All these formal reviews utilize the Review Item Disposition (RID) activity to point up discrepancies. Thus they are indicative of the scope of the PDR's as well as the latitude provided to the "working troops" to have their input known and discussed at management levels. This is elaborated on in Section 7.6 wherein the review operation is described. The RID describes significant discrepancies and inconsistencies as well as distinct problem areas determined by anyone on the project/program. The PDR process usually consists of 10 days or two weeks of full scale team reviews of appropriate data and discussions during which RIDS are written. The RIDS are then provided to a screening group, followed by a pre-board, ending up at the formal board. Orbiter 102 PDR resulted in 978 RIDS and the Systems PDR produced 1,204 RIDS. Due to the large number only the most significant ones could be presented to the formal board. However, the individual Team Leaders for each of the approximately twelve teams of the PDR report to the Formal Board on the team activity and major areas of concern.

There are always some areas which cannot be fully covered during the PDR due to a lack of information. These areas require and receive the necessary emphasis to achieve a sufficient degree of technical and documentary depth so that they may be reviewed within a reasonable length of time after the PDR.

The Orbiter Thermal Protection Subsystem, Thermal Control Subsystem, Environmental Control and Life Support Subsystem and Range Safety Avionics are some subsystems which will be so handled in the August/September 1975 time-frame. In the same vein, lack of definition of the Orbital Flight Test Program prevented evaluation of the system design against the mission requirements so that it too will be covered at a later date.

Material covered and that which has yet to be examined as a part of the PDR process again shows the need to look at the Shuttle review system as a continuum which supports the program and project managers' needs for design/hardware assurance.

At a later date each of the elements and the system will be subjected to a Critical Design Review (CDR) to determine the compliance of the completed design with the technical requirements of the NASA baseline. The CDR should result in authorization to the contractors to proceed with he release of detail design to manufacturing, the approval of test procedures, and the appropriate revision or update of the Level III baseline documentation. The Critical Design Reviews begin in the early Spring of 1976.

3.0 SHUTTLE PROGRAM ELEMENTS

3.1 Orbiter Project

Because of the large number of Shuttle elements and components, Panel efforts have been concentrated on those areas which most impact crew safety and management control of the program elements. The intent in this report is to focus on the subsystems critical to crew safety and to provide data for an understanding of risk assessments. A special section is given over to the Orbiter Thermal Protection System because the Panel feels it is one of the most significant systems which, if not properly and adequately designed, fabricated and maintained, would pose a real crew hazard as well as a Shuttle system operational problem.

However, there are differences between the first two Orbiters which should be identified to understand what follows. The first Orbiter, Number 101, will initially be configured as a test vehicle for the Approach and Landing Test (ALT) Program. It will then be reworked to the operational configuration. The second Orbiter, Number 102, will be built in the orbital flight configuration. Thus there are some items unique to the 101 and there are other items which appear on 102 for the first time. Many of these differences result from the needs for flight test instrumentation at low speeds and low altitudes on 101 versus high speeds and high orbital altitudes on 102. There are also differences because of the different natural and induced environmental effects. For example, on the 101 vehicle there is no Thermal Protection

System (TPS), little if any internal insulation, and no main propulsion system (SSME's). There is an instrumentation boom at the nose and ejection seats.

3.1.1 Subsystems Critical to Crew Safety

For the purposes of this report the Orbiter system is divided into the following subsystems:

(a) Structures - this includes the fuselage, wings, empennage, crew module, purge, vent, drain, payload doors, thermal protection system (TPS), and the internal insulation.

(b) Propulsion - includes the reaction control system, orbital maneuvering system, auxiliary propulsion system and the interface between the Orbiter and the Space Shuttle Main Engines.

(c) Avionics - includes guidance, navigation, flight control, communications and tracking, display and control instrumentation, data processing and software, electrical power distribution and control.

(d) Crew Station - includes all those items, such as fuel cells, batteries, and rotating equipment used to store and generate electrical power. This does not include those items used for distribution and control of the generated power.

(e) Environmental Control and Life Support - these include the atmospheric revitalization subsystem, active thermal control, cryogenics, airlock support and waste management.

(f) Mechanical - includes landing and deceleration gear, separation, actuation devices, payload retention and deployment, hydraulics, and pyrotechnics.

All of these systems and their components may be construed as affecting crew safety.

The Panel chose to focus first on (1) systems extending the technical and fabrication state-of-the-art in the literal sense or in its application, (2) systems which prior program "lessons" have indicated as areas of concern, (3) areas which the Panel members considered most vulnerable to "human error' in defining requirements, designing and fabricating, and (4) areas which cannot be adequately tested or validated on the ground.

Using the above criteria, the following subsystems received particular attention from the Panel:

3.1.1.1 Doors and Vents

3.1.1.2 Thermal Protection System

3.1.1.3 Propulsion

3.1.1.4 Avionics

3.1.1.5 Electrical Power System

These are discussed in terms of systems design and current development status.

Additional subsystems of particular significance for crew safety include:

3.1.1.6 Crew Compartment

3.1.1.7 Hydraulics

3.1.1.8 Separation Mechanisms

3.1.1.9 Structures

Here the comments are more limited for the reasons indicated in each section.

Orbiter weight control has been a major management objective. Currently, the estimated weight is about 2000 pounds below the target of 132K. Reviews continue to find ways to take weight out of existing designs or to find new ways to keep the weight down. Since weight control is an important driver, the Panel in its review of these subsystems has been sensitive to any impact on safety.

3.1.1.1 Doors and Vents

Doors and vents on the Orbiter vehicle must operate reliably to maintain the vehicle's integrity for flight during ascent and reentry, and to avoid risk to the crew.

Because of their significance for crew safety, the following doors were included in the Panel's reviews:

(a) MPS/T-0 Umbilical Attachment Door. This door was recently deleted as a result of the latest aerotherodynamic analyses. Figure 3 and 4 depict the "before" and "after" configuration.

(b) Reaction Control System (RCS) Forward Thruster Doors. These have also been deleted as a result of recent studies. Figure 5 depicts this change.

(c) Startracker Door.

(d) ET/Orbiter Closeout Doors. There are two - left and right side.

(e) Air Data System Probe Doors. There are two - left and right hand.

(f) Landing Gear Doors. There are three sets of fairing doors - one for the nose wheel and one each for the left and right main wheel system.

(g) Personnel Hatches. There are three.

(h) Rendezvous Sensor. Currently no information is available on this item.

(i) Payload Bay Doors. There are two 60-foot long doors.

(j) Payload Preflight Umbilical Door.

(k) Vent Doors. These are discussed under the vent system. In addition there are doors on the Orbier 101 for use during the Approach and Landing Tests on the first vertical flight vehicle 102 that are not found on the later operational vehicles.

System Design

During ascent, door position is a function of required operation. For example, the startracker door is closed during ascent while the External Tank/Orbiter closeout doors are open until the ET is jettisoned. Regardless of the particular function of individual Orbiter doors, they all have to be closed and secured prior to entry.

The Panel reviewed the basis for confidence in the mechanical design. The doors themselves are considered as structural items, and thus are to be designed to preclude failure by use of adequate design safety factors. Recent aerothermodynamic analyses have led to a re-assessment of Orbiter doors resulting in the deletion of the Launch Umbilical Door and RCS Forward Thruster Doors. The remaining Star-tracker Door and some vent doors are actuated and latched by electric motors driving linkages through gear boxes and mechanical sequencers. The ET/Orbiter closeout doors and Air Data Probe Doors are actuated and latched by power drive units consisting of two electric motors driving linkages through a gear box.

There are personnel hatches at three locations in the Orbiter Orbital flight configuration: (1) crew module ingeess/egress hatch, (2) airlock hatch, and (3) airlock/payload bay hatch. The crew module ingress/egress hatch is a circular hatch with double walls. The hatch outer surface is covered with TPS and seals at the Orbiter outer mold line. The hatch inner surface provides a redundant pressure seal to the crew module pressure vessel. The hatch pressure seals may be checked for leakage by pressurizing the volume between the seals. This leak check capability exists during launch preparations or in-flight, utilizing GSE or flight equipment. Mounted in the center of the ingress/egress hatch is a 10-inch diameter window used for crew observations of external conditions and for the performance of experiments.

Control of the hatch is manual, utilizing a rotary actuator which may be driven from either side of the hatch and Apollo CM-type hatch latches. The airlock hatch is a circular hatch which seals at the airlock entry tunnel separating the crew module from the interior of the airlock. The hatch is closed and latched for Orbiter launch, opened shortly after orbital injection to allow access to the airlock interior, and also is cycled during extra-vehicular activity. The hatch pressure seals also may be checked for leakage by pressurizing the volume between the seals. This leak check capability and hatch control is the same as for the ingress/egress hatch. The airlock/payload bay hatch is also a circular hatch which seals at the airlock exit tunnel. Hatch pressure seal check and hatch control again is similar to the ingress/egress hatch configuration. There are two payload bay doors with an actuation system for each 60-foot half door. The Payload Bay door actuation mechanism has not been finalized as yet but the following subsystem description can be provided at this time. The output motion for door movement is taken off the second ring gear of compound planetary gear boxes. There are six gear boxes along each power path and these are connected by torque tubes to each other and to a main reduction gear box. The main gear is driven by the output of a double differential connecting three electric motors. This arrangement allows system operation for any two motor failures, any one motor failure combined with one electric system failure, or any two electrical system failures. A mechanical disconnect of the motor drive unit is provided and the door actuator gear boxes are designed so they will back drive. This will allow the GSE to open or close the doors.

The Purge, Vent and Drain Subsystem is composed of five elements: (1) structural compartment vent, (2) structural compartment ground purge, (3) structural compartment drain, (4) window cavity conditioning, and (5) hazardous gas detection. The individual systems are not discussed here since the major focus is on the safety impacts associated with these systems. The vent ports insure no violation of the delta pressure limitations of the primary structure and therefore are of primary significance for crew safety. It is the proper mechanical operation of these doors that is critical, not the structural integrity of the doors themselves.

There are some eighteen of these vent doors along with the associated electro-mechanical and mechanical operational devices to move them as required. The other purge, vent and drain units present considerably less risk to the crew. However, malfunctions could lead to mission abort.

The structural compartment ground purge provisions are composed of a GSE-supplied flow of air/GN_2/GHe, which is distributed through an onboard duct network to all required structural compartments. The structural compartment drain provisions are composed of piping and disconnects which, acting together with ground support equipment, minimize the accumulation of moisture within the Orbiter structural compartments. The collection points are so located that effective draining is feasible with the Orbiter in either the horizontal or vertical attitude. The window cavity conditioning provisions allow the introduction of a ground-supplied dry nitrogen purge into the inner and outer window cavities during preflight servicing of the Orbiter. During the approach and landing flight tests and boost to orbit, the gas in the window cavities is vented through lines to overboard. While in orbit they are continuously venting the space. During the entry phase ambient atmosphere flows into the cavities. Appropriate valves act to limit the delta pressure across the window panes in the event of filter or line clogging. The hazardous gas detection provisions utilize a combination of flight hardware and GSE to detect the presence and monitor the concentration of hazardous gases during prelaunch and post-landing operations.

Current Status

Door designs, as described to the Panel, are such that the door itself and the mechanical linkages and gear boxes are considered the same as primary structure, i.e., they are designed with sufficient structural safety margin to preclude failure under any known or suspected load condition.

The door operating mechanisms are quite complex and there are continuing efforts under way to simplify these mechanisms.

In the main the doors are contiguous with the Orbiter Thermal Protection System (TPS) and as such interact from the aerothermodynamic standpoint with the function of the TPS.

Rigging of the external doors is difficult and must be done in the "blind" in many cases. As a result it is difficult to prove that door latches latch and lock properly and the chance for human error is present to a degree that may require more than average detailed operational and inspection controls, or verification procedures. The Panel will review this area as the program evolves.

The ET/Orbiter Separation Cluster Plate Doors and Startracker Door continue to be the subject of studies to determine whether the doors and their associated mechanisms could be eliminated, reconfigured, simplified, or reduced in size thereby reducing or eliminating the crew safety risks associated with improper door operation. The results of these studies will be the subject of further Panel review.

The External Tank/Orbiter Cluster Plate Doors are now about 46" x 62" (actually some 2354 sq. in.) rather than the original 72" x 84" size. The maximum exterior surface temperature of the door when closed during reentry is about 1500° F. It is estimated that without the door local temperatures would be 1.5 to 2.5 times as high due to flow disturbances. These doors are open during launch and ascent until ET separation and it would appear that an extensive test program to assure proper operation in the post-launch environment is warranted.

The Startracker door size is dictated by tracker view angles and the requirement for daylight tracking. Tracker-lines of sight are made more difficult by the thickness of the Orbiter TPS material surrounding the window itself. Maximum temperatures near the Startracker door are expected to be about 825° F. The door mechanism and the alternatives are still under evaluation.

Venting analyses have been conducted to determine the effect on the Orbiter vehicle of internal compartment pressures due to opening the vent doors at different altitudes during reentry. At the time the active vent doors are closed, prior to reentry, the pressure in all of the vented compartments is approximately zero. The Orbiter enters the atmosphere with the doors closed until the "hot" part of the descent is completed. The vent doors are then opened at about 70,000 - 80,000 feet and remain open until the Orbiter is on the ground. If the opening of the doors is delayed to a lower altitude, excessive differential pressures could develop across some of the compartments. Analysis indicates that it takes about 15 seconds to open the vent doors. On the other hand those vent doors which open too soon may produce problems due to the impingement of hot plasma on structural members. The active vent system selection was extensively reviewed and approved by a number of contractor and NASA organizational elements, including the Shuttle System Program Manager. The Orbiter vent system appears to have been sized and analyzed for nominal ascent and reentry trajectories, and no detailed analysis has been made to assure adequate operation of this system during abort or vehicle malfunction conditions. Venting analyses for these conditions are not currently underway, but should be available sometime after July 1976.

Two failure modes of the vent system that have been under study because of significance to crew safety are the failure of the OMS pod vent and wing vents to open. JSC venting analysis showed that the fuselage can tolerate a single system failure, but the wings and OMS pod would fail structurally. The time to troubleshoot such a failure is very short (in seconds) and therefore backup procedures cannot meet the need.

The present Orbiter baseline with regard to Orbiter doors and their functions/criticality are shown in Table IV .

3.1.1.2 Orbiter Thermal Protection Subsystem

Systems Designs

The Thermal Protection Subsystem (TPS) consists of the equipment used to insulate against the external aerothermodynamic or induced heating effects on the Orbiter vehicle. The Thermal Control Subsystem (TCS) maintains appropriate Orbiter thermal conditions. The Panel has examined the TPS in detail and considers it one of the most significant subsystems on the vehicle. While not much attention has been given the TCS, it will be examined more closely during the coming year.

The TPS consists of those materials applied to fixed and moveable surfaces to protect the underlying aluminum structure and heat sensitive equipment. The TPS has undergone an evolution in design. Changes have occurred in tile materials, coatings, and configuration. The system will be reviewed in a PDR this summer.

TPS design for operational vehicles (Orbiter 103+, Subs) includes five different thermal coverings rather than the current design using three types:

(a) Low temperature reusable insulation

(b) High temperature reusable insulation

(c) Reinforced carbon-carbon nose caps

(d) (New) Nomex "E" felt with coating of white silicone oxide

(e) (New) bare surfaces with coating for emissivity/absorptivity

Current configurations are shown in Figures 6 to 8 .

Studies have been underway to try and simplify and reduce the cost and weight of the Thermal Protection Subsystem. Both JSC and Rockwell have been heavily involved in these activities.

The modifications between last summer and the spring of 1975 are due to a change in trajectory which resulted in lower temperatures, lower heating rates and a better tile design, based on a more sophisticated thermal analysis of the tile joint areas.

Areas that have received increasing attention are the aero-surface thermal seals: elevons, rudder/speed brakes, and body flap. These seals must (1) provide thermal protection for the aluminum structure to a maximum of 350^{o} F., (2) restrict flow of air and/or plasma from the high to low pressure areas, to allow aerodynamic control of the vehicle, and (3) have 100-mission life capability in operational vehicles.

Wing elevon seals must provide sealing between the:

(a) Elevon to fuselage

(b) Elevon wing (top and bottom)

(c) Elevon-to-elevon

(d) Elevon wing tip

These are complex seal arrangements and have not yet been fully detailed and analyzed.

The vertical tail seal is a conical tube running the length of the rudder as shown in Figure 9 . The body flap seal concept is shown in Figure 10 .

Among the objectives in developing tile installation procedures are finding ways to minimize the number of tiles and shapes and to simplify the maintenance removal or repair of tiles. Because of the difficulty in maintaining precise airframe substrate surface tolerances, as well as tile installation height tolerances, Rockwell Space Division has developed the "building-block" approach for installing tile on the so-called "acreage" areas comprising about 80 percent of the Orbiter.

In this approach standard tiles are used in large areas. Special rows of closeout tiles are added to fill in the gaps between adjacent areas.

The remainder of the tiles will have to be shaped and fitted for such multiple curvature situations and penetrations through the TPS subsystem as:

(a) The line between the RCC installations and adjacent tile installations.

(b) Windshield

(c) Forward fuselage hoist point

(d) Actuator access doors

(e) Rear access panels near OMS pod

(f) Structure cavity vents

(g) RCS thruster package doors and opening

(h) Nose gear doors and main gear doors

A part of the installation procedure includes the pre-fit of tiles on the vehicle surface with a hand sanding of the lower tile surface to match the inner mold-line of the Orbiter and hand sanding of the upper surface to match the required outer mold line dimensions in order to control the "step" that exists between tiles. This is shown in Figure 11 an indicates the maximum allowable tolerance to preclude "fouling" the airstream flow over the vehicle surface. Thus a tile-to-tile step of +0.030" to -0.050" is allowable in most instances, and a tile gap of 0.050" nominal is allowed.

Current Status

TPS concerns and issues that have been resolved and those still challenging the designer, which have been of specific interest to the Panel during its reviews of this subsytem, can be summarized as follows:

(a) Experience working with the reusable surface insulation (RSI) or tiles shows it has low resistance to ground handling damage. It has the capacity to sustain damage without catastrophic failure during exposure to induced environment. Installation costs and time requirements are sensitive to the gap and step criteria, tile configuration and installation techniques.

(b) The low temperature tiles appear now to provide more protection than needed, based both on the change in trajectory and the results from recent tests and analyses. This over-protection is also a result of the minimum tile thickness of 0.2 inches. This thickness is derived from the structural properties of the tile and its tendency to crack when any thinner than that. As a result, the use of Nomex "E" felt with a white oxide coating has been tested and found practical as a replacement for some 3,275 ft.^{2} of surface which achieves a maximum temperature at the outer mold-line of 700° F. or less. Information to date shows the Nomex felt to be acceptable for 100 mission use for temperatures up to 600° F. and very possible to 700° F. There are some 2,000 plus ft.^{2} of the area meeting the 600 degree requirement. There are even areas on the top of the Orbiter that could be flown without any TPS at all. Arc jet testing conducted in early fall and winter indicated that the Nomex and coating remain elastic and waterproof for 100 mission cycles at 600° F. and for at least 50 cycles at 700° F. A further investigation was initiated 25 January 1975 to resolve some of the remaining challenges. These include the extent of degradation of the coating with exposure to ultraviolet radiation, particulary degradation of the thermal radiative properties of absorbtivity and emmissivity and perhaps elasticity. Although there are no particular structural or vibroacoustic concerns, there is the current unknown of what contamination does to the coating. The program also needs more information on the capability of Nomex to handle temperature dispersions, particularly those over the designed-for values. Rockwell has demonstrated the manufacturing and installation ability of the Nomex felt and indicates a weight savings on the order of 500 pounds if used on the 2000 to 3000 square feet of surface area currently cited.

The Panel has also been monitoring the studies to assess the hazards from: (1) ET insulation ablation products deposition on Orbiter glass surfaces and TPS, and (2) ice and frost breaking away from the ET and striking the Orbiter TPS. Tests and analyses have been conducted to assess the ET/Orbiter interaction. As a result it was confirmed that the abalation products will not flow over the windshields or the top observation windows and does not materially affect the TPS absorptivity and emissivity or its ability to adequately protect the aluminum structure. The possibility of TPS damage resulting from ice and/or frost forming on the ET and then breaking away during and prior to the ascent portion of the mission is still an open item receiving attention. When this is completed, if in fact a problem exists, protection will have to be afforded the TPS during the boost phase. Tests to date are not conclusive. Model tests indicate that ice will not form but frost will.

Natural environment factors such as rain, hail, lightning, and bird impact have been studied relative to their effect on the TPS. To assess rain erosion, precipitation models for KSC and Vandenberg AFB have been developed based on NASA and Air Force data. These models as augmented by tests and analyses indicate the following probabilities of encountering critical rains during ascent and descent at both launch/landing sites:

Flight	Per One Flight	Per 100 Flights
KSC Ascent	0.31%	26.7%
KSC Descent	0.013%	1.26%
VAFB Ascent	0.04%	3.9%
VAFB Descent	0.0011%	0.11%

If required, such data may be developed for Edwards AFB. During ascent, launch constraints can reduce the rain erosion problem. Capability for maneuvering during reentry to avoid rain is quite limited.

As a result erosion has to be accepted and the TPS refurbished as required during the maintenance and turnaround period. Such erosion is not considered a crew hazard as such.

As for ice impact and hail tests have shown that the tile does not exhibit significant resistance to ice impact damage. Atmospheric ice is encountered at altitudes below about 50,000 feet. Hail may occur only within or below thunderstorm cells and is observed very infrequently at the surface at both KSC and Vandenberg AFB. Higher frequencies occur at altitude. Studies indicate that the probability of encountering hail during ascent is about 0.0075% and during descent about 0.015% on an annual basis. Since hail is a thunderstorm phenomena, the probability of hail encountering hail during launch may be reduced to essentially zero by constraining launches. During horizontal flight the ability to perform flight maneuvers are neglible and flight through area thunderstorms cannot be avoided. Hail would not be catastrophic but would certainly require significant refurbishment after landing.

Bird impact data from both civilian and military sources have been analyzed with respect to the Orbiter flight trajectories and expected frontal area subjected to bird strikes. Specific attention was given to the windows as the most significant area of concern and the TPS as secondary. Because the probability of a bird strike is extremely low, the program has deemed it practical to accept such low probability risk.

TPS is obviously subject to "people" or handling damage. Therefore those personnel coming in contact with the Orbiter must be trained and constantly be reminded of the fragile nature of the tiles. Where possible, the ground support equipment should be designed and used in a manner which minimizes any inadvertant damage to the TPS.

Lightning effects on the TPS are continuing to be studied to assess the adverse effects, determine how they can be eliminated or minimized and to define necessary constraints. The current baseline has not designed the TPS for a lightning strike. Without any avoidance measures the probability of a lightning strike would be about 0.008% for all altitudes up to 50,000 feet for launches from KSC.

The probability of a strike at Vandenberg AFB would be considerably less, based on lightning occurrence there. Selective time of launch can reduce the probability of a strike by at least an order of magnitude.

Solid Rocket Booster separation motors in their original configuration would have impacted the TPS when fired. As a result of these analyses the forward SRB separation motors were relocated 120 inches forward. Their thrust was increased from eight units of 12,000 pound thrust to four units of 20,000 pound thrust. The firing time was also reduced from two seconds to a period of not more than 0.75 second.

Tile installation is sensitive to structural buckling caused by thermal stresses along the forward fuselage, mid-fuselage and a few panels on the upper and lower surfaces of the wings. Orbiter specification requirements are that there be no buckling below 115% of limit load on ascent and 100% of limit load on descent. As an example, in the mid-fuselage and wing areas the initial design assumed stringers provided adequate stiffness and spacing to preclude buckle until limit load was reached. Subsequent analysis and testing showed that buckling occurred considerably below the design load. The cause was the transverse skin compression stresses induced by combined thermal and mechanical loads. Such buckling disturbs, if not breaks, the TPS subsystem. The current approach to resolving this problem is to conduct tests to structural ultimate strength and determine ability of the TPS subsystem to accommodate the buckling without failure. Then the program will be in a position to define stiffening modifications and retest of TPS installations.

Another area of concern was the effect of the salt air environment on the chemical stability of the tile coatings at the elevated temperatures anticipated in ascent and reentry. As a result of this concern, a test program was conducted at JSC in the 1.5 megawatt arc jet tunnel facility to evaluate the effects of the salt contamination on the reuse capability of the high temperature thermal protection material. Test results indicate that salt accumulations representative of up to ten years of launch pad environmental exposure have no adverse effects on the reuse capability of the HRSI and its coating for approximately 100 missions.

The high temperature (greater than 2300° F.) thermal protection material is made of reinforced carbon-carbon material. This material consists of pyrolized carbon fibers in a pyrolized carbon matrix with a silicone carbide coating. Extensive development testing and analyses are still in process to determine actual performance characteristics and to confirm the RCC configuration as designed, as well as alternate designs which may be used as the final analyses converge on the final design. A design review for this area is scheduled for the summer of 1975. Two major problems with the RCC material are (1) subsurface oxidation, and (2) inter-laminar failure occurring within the pyrolyzed matrix itself. Sub-surface oxidation results in mass loss which is a function of mission environment pressure and temperature. For example, tests are presently being conducted to determine how best to meet the particularly severe environment where the shock wave off the nose of the Orbiter intersects the wings. The inter-laminar failure problem is one of material processing and now appears to be resolved.

The TPS test program includes (1) material characterization, (2) design development testing, and (3) design verification. The results of the test program to date can be summarized as follows:

(a) Reusable surface insulation (tiles) have been tested for "as fabricated" properties and these test results are being evaluated for determining any future test requirements for material characterization.

(b) Reinforced carbon-carbon test program is approximately 25% complete with scheduled completion in February 1976.

(c) Seals used on moving surfaces are in the very early stages of material characterization testing.

(d) Design development testing covers those tests conducted to confirm analytical methods, support of design configuration selections, and establish verification test methods. For example, a 0.36-scale model wind tunnel test is in process at Ames Research Center to measure effects of TPS on low-speed aerodynamics. Some 120 tests are to be performed on this model in the low-speed 40 x 80 foot wind tunnel. Lost tile tests, structural tests, fatigue tests, flutter tests and lightning tests have been, and continue to be, conducted. Aerodynamic heating in the gaps between the silica TPS tiles is receiving attention through tests to assure that these phenomena are correctly modeled in the analyses used to define the configuration of the TPS.

In summary, the Orbiter TPS is a difficult and complex system to design andunderstand. None the less, the analyses and testing conducted to date indicate that the design and operational complexities are yielding to the planned development effort. The remaining concerns or challenges include the following:

(a) Improved RCC coating to increase material lifetime.

(b) Decision on use of Nomex felt in lieu of thin tiles.

(c) Thermal protection of penetrations (aerosurface seals and movable doors)

(d) TPS sensitivity to structural buckling.

(e) Tile-to-tile high tolerance to preclude "tripping" or disturbing the airstream.

(f) TPS inspection, maintenance, and handling.

(g) 100 mission reusability.

3.1.1.3 Propulsion Systems

System Design

This section deals with four separate power systems: (1) Auxiliary Power Unit, **(2) Forward and Aft**, (3) Reaction Control, and (4) Oribtal Maneuvering Subsystem. The main propulsion system for the Shuttle integrated system is covered under Section 6.6 of this report. The portion contained in the Orbiter vehicle, the three main engines, is covered in Section 3.2.

The Auxiliary Power Unit Subsystem consists of three independent APU's, each having pressurized fuel storage and distribution, an APU, lube oil cooling, and exhaust, vent and drain provisions. Each APU provides mechanical shaft power to one main hydraulic pump of the

Orbiter Vehicle Hydraulic Power System. At least two APU-hydraulic systems must be operational to assure safe return of the crew and vehicle. Operational flight control requirements for the Orbiter for the approach and landing phase can be met with any one of the three APU systems failed. With two systems failed, the remaining system with overspeed cannot meet all operational requirements and may not, therefore, be capable of returning the crew and vehicle safely under all mission design conditions.

The forward RCS provides precise attitude control and three-axis translation during separation from the External Tank, orbit insertion, and orbital phases of the flight. The aft RCS does all of these same functions in conjunction with the forward RCS and also provides thrust for the reentry phase of the mission. The forward RCS has eleven primary and two vernier thrusters mounted under doors and six thrusters mounted exposed. The doors remain closed and latched during boost and reentry phases and are deployed and locked in place for ET separation, orbit insertion and orbital phases. The aft RCS is composed of twelve primary thrusters and two vernier thrusters located on either side of the aft Orbiter fuselage for a total of 24 primary and 4 vernier units.

The primary RCS engine specification requires the engine to incorporate a burn-through detector to sense an incipient thrust chamber burn-through and to provide an appropriate signal to be used by engine shutdown. This is a difficult item to develop and qualify and may also cause operational problems due to false shutdown. It is now considered that burn-through is not one of the primary failure modes. The contractor was asked to process a Master Change Record (request), MCR, to delete the burn-through detector per the 102 PDR (February 1975).

The Orbital Maneuvering System (OMS) provides the propulsive thrust necessary to perform the following maneuvers: (1) final velocity increment for orbit insertion, (2) orbit circularization, (3) orbit transfer, (4) rendezvous, and (5) de-orbit. Although one OMS engine could be used for these operations, reliability considerations dictate that the loss of an OMS engine is cause for abort.

The OMS has single failure points in the pressurization and propellant feed areas and the failure mode would be rupture and excessive leakage. Any excessive pod differential pressure could result in structure and TPS damage preventing safe reentry. The OMS is fail safe otherwise, except for such catastrophic events as engine or propellant explosion.

Current Status

There are numerous mechanical connections used on the forward and aft RCS in lieu of welded connections. This approach permits removal and installation of equipment in minimum time while minimizing contamination hazard to the remaining portion of the system. Where possible the fittings and seals being used were already qualified in the same application in Apollo and Skylab programs. After reconnect all mechanical connections will be pressurized to system pressure with helium and externally leak-tested to system requirements.

NASA and contractor have agreed to maintain tight surveillance of mechanical connections (fittings) to assure both the number and possibility of leakage are minimized.

Verification of component propellant compatibility of OMS/RCS hardware is under review. Based on the demonstrated Apollo CSM experience, the current requirement is that components be constructed of materials with demonstrated propellant compatibility. However, subsystem design features and operational methods, as well as program funding limitations precludes compatibility testing at the component level of the OMS high pressure helium isolation valve, helium pressure regulator, low pressure vapor isolation valve, and the tank pressure relief valve.

In the RCS the plan is to authorize only those materials in the helium system where there is proven compatibility with the propellants. The data and analysis will be accomplished during the development and qualification programs. Because of the propellant system components total exposure to liquids, a qualification compatibility test will be conducted at the subcontractor level.

Deletion of the vibro-acoustic test of the forward fuselage has meant cancellation of the vibration test of the forward RCS module. However, the need for system certification of the RCS prior to first verticle flight has not been eliminated, so a reassessment of means and techniques is underway to provide the required certification data base. Plans are to review aft pod vibro-acoustic tests, system similarity and analytic techniques to see if aft pod data can be extrapolated for application to the forward RCS module. In addition, alternate forward module test plans and schedules are being studied to determine a cost effective vibration test for the forward module only. Resolution of these alternatives and a recommendation is due around 1 July 1975.

3.1.1.4 Avionics

Systems Design

The avionics subsystems provides commands, guidance and navigation and control, communications, computations, displays and controls, instrumentation, and electrical power distribution and control for the Orbiter, external tank and the solid rocket booster. The avionics are configured to facilitate checkout, access, and replacement with minimal disturbance to other subsystems. Equipment locations are shown in Figure 12 .

Computations or data processing is accomplished through the use of five digital computers. Three are dedicated to the guidance and navigation function. One can be used for either guidance and navigation or payload and performance monitoring,and one is dedicated to payload and performance monitoring. Software or computer programs are integral to this data processing and control system since these five general purpose computers are the same mode. It is the resident software that determines the computer function.

Verification of the avionics/software systems as an independent and integral part of the Orbiter/Shuttle system is accomplished through the following test programs:

(a) Software Development Laboratory program to verify the flight data on flight computers.

(b) Avionics Development Laboratory program to verify "single string" and redundant hardware system operation and the hardware/software compatibility.

(c) Shuttle Avionics Integration Laboratory (SAIL) program to verify redundant hardware system operation for Orbital Flight Test as well as the hardware/software compatibility for OFT.

(d) Simulations to verify flight crew operations of vehicle and the guidance and navigation performance accuracy in a manner similar to simulations for prior manned spaceflight operations.

(e) Approach and Landing Test (ALT) program using Orbiter 101 will be used to verify the aerodynamic capability of the Orbiter, the aerodynamic guidance and navigation performance, aerodynamic system integrated operation and the aerodynamic dependent software.

(f) Orbital Test Flight program to verify the total mission vehicle capability with avionics and associated software.

Orbiter 102 will have the following avionics elements not on Orbiter 101.

(a) Startracker/Light Shield

(b) Those portions of the flight control system that involve the Reaction Control System, Orbital Maneuvering System, Thrust Vector Control for the SSME's.

(c) SSME interface unit portion of the system for processing engine data.

(d) Many items of the communications and tracking system, e.g., KU band radar, payload interrogator, signal processes, portions of the S-band, etc.

Current Status

The relationship of avionics to the flight and ground crew safety is multifaceted, since every action and reaction during the mission is controlled to some extent by the avionics system. The Panel has, therefore, had to be selective. We have chosen to review three areas most significant to crew safety: (1) Orbiter/SSME-Controller interface, (2) ALT/OFT flight control modes, and (3) abort operations.

A review by the Panel was to determine if there are potentially critical failures across the Orbiter/SSME interface, and, if so, to understand those steps being taken to minimize or eliminate such efects. Where hazards are not eliminated we wanted to assure that the assessment of the risk and the rationale for accepting it had been given appropriate management attention.

Operational and checkout commands and engine flight data are supplied via the electrical interface connectors, at the engine-supplied electrical interface connect panel. Commands consist of engine start, shutdown, thrust level changes, checkout, and sequence checks. Engine flight data transmitted to the vehicle consist of information necessary for malfunction display, fault isolation, maintenance recording, trend analysis, performance monitoring and checkout. Three parallel redundant connectors provide a reliable path for the Orbiter to engine commands. Further a minimum of two of the three commands must be received before the engine response will be initiated. Two of these connectors are also employed to transmit the engine flight data back to the Orbiter. Failure to provide correct command during ascent or to transmit engine performance back to the Orbiter do not appear to be a direct threat to the crew safety since the engine will continue to operate on the last correct command received.

Flight control utilizes automatic commands determined by the guidance and navigation subsystem manual commands provided by the crew, vehicle motion sensed by the sensors, logic decisions processed by the control laws, and those forces produced by actuation of the aerodynamic surfaces TVC's, RCS, etc. to perform stabilization and control. The control laws are software. The flight control requirements for each mission phase (ascent, on-orbit, reentry, and atmospheric) are specified in terms of control mode elements. These mode elements or control modes are the building blocks which can be used in combinations to provide the actual operational control modes. During ascent through the SRB staging the nominal baseline has been defined as automatic mode. While there is manual redundancy it will not be used unless there is a significant benefit. After that portion of the ascent period, the flight control modes can be (1) manual direct, (2) manual command augmentation, (3) hold, (4) select, and (5) automatic. These are defined in Table V . One of the areas being worked by the program that will be examined by the Panel is the identification of OFT launch failures which require manual guidance and control. Another area is the aerodynamic tolerance effects on response and stability of the flight control/structures design capability. Structural constraints have been reflected back in a manner which indicates a need to restrict the angle of attack and side-slip variations to a minimum consistent with ability to provide for high aerodynamic load relief. Systems studies have indicated that these constraints are only marginally reached with nominal system parameters. Flight control margins are tight and vehicle dynamics are pushing the margins (plus/minus tolerances or limitations on system input/output lag, accelerations, roll rates, etc.). The first stage ascent is the period of greatest concern from the standpoint of computer cycle time. There is a possibility that sample frequency requirements may increase. If so, this would further aggravate the computer timing problem.

The role of the avionics system in abort operations is particularly significant because of the need for large quantities of information concerning the vehicle and its performance as well as the need for fast reaction to on-going events. Confidence in the design capability of the Orbiter vehicle and its avionics subsystem to perform the once-around-orbit, return-to-landing-site or any other abort mode is being examined on a continuous basis as the design matures and the system capabilities are further designed. The Panel will examine this area in more detail as the concepts and design mature.

A back-up flight control system is being installed in Orbiter 101 only to provide protection against generic software problems or problems with the complex hardware, crew interfaces, and mechanization. No new hardware is anticipated. This approach should provide an additional measure of safety during the early flights of the ALT program.

This concern with overloading the computer capability in the Orbiter is real. It has been stated that at this time the word requirements are in the range of: ALT 2700-2800 words, OFT on-Orbit 2000-5000 words and entry 5000-6000 words (on orbit and entry are additive). The main drivers on the computer and the flight control requirements are speed and memory.

A number of flight control support tasks are being carried out by NASA Centers. Marshall is working on:

(a) Ascent flight dynamics and control.

(b) FCS requirements and constraints.

(c) Flight dynamics/stability performance.

(d) Body-mounted sensor complement and locations.

(e) Digital sampling/filtering and quantization.

Langley is working on:

(a) Entry guidance and control.

(b) Independent evaluation of flight crew role in controlling Shuttle.

(c) Orbiter G&C entry design verification.

The Flight Research Center is working on:

(a) Entry aerodynamic flight control, developing an F-8 digital fly-by-wire program for DPS and flight control redundancy management and flight control system design.

A number of avionics elements have not been placed on contract as yet or design has not evolved sufficiently to review it. The Integrated Electronics Assembly is not yet on contract. Many of the operational communications and tracking hardware will not be contracted for until 1976-77 period. This also holds true for display and control equipment for 102. Those areas, with safety implications, will be reviewed by the Panel at the appropriate time.

3.1.1.5 Electrical Power Subsystem

Systems Design

The electrical power subsystem generates the electrical power and is active throughout the vertical flight test program and operational flight and during ground operations when ground support equipment is not connected.

This electrical power subsystem is comprised of the power reactant supply and distribution and three fuel cell power-plants. The electrical power subsystem is shown schematically in Figure 13 . During peak and average power loads, all three fuel cells and buses are used; during minimum power loads, only two fuel cells are used but they are interconnected to the three buses. The third fuel cell is shut down but can be reconnected within 15 minutes to support higher loads. Excess heat from the fuel cells is transferred to the Freon cooling loop through heat exchangers.

Most of the active elements of the electrical power system have been designed to sustain two failures and remain operationally safe, in other words fail-operationally then fail-safe. The power reactant supply and distribution tanks, electrical power subsystem plumbing, and passive elements have been designed to provide fail-safe operation after a single failure by means of redundant subsystem flow paths which are physically separated. A single product water-line is provided to the environmental control and life support subsystem since fail-safe water requirements are provided with the environmental control and life support subsystem.

The operational use of fuel cells for manned space flight evolved during the Gemini, Apollo, and Skylab programs. The Space Shuttle fuel cells will be serviced between flights and reflown until each one has accumulated some 5000 hours of online service.

Interfaces of the electrical power subsystem with other subsystems, such as the avionics for control, and environmental control and life support subsystem, have not as yet been examined to any degree by the Panel. The Panel's major concerns here will deal with (1) crew hazards resulting from subsystem failures, e.g., loss of power to critical functions, (2) fire hazards resulting from short circuits or other failure modes, and (3) system design to prevent or inhibit deleterious events from propagating.

Current Status

Based on latest available data, it was noted that the current power requirements exceed the electrical power subsystem capability. The present electrical power requirement of 2006 KWH exceeds the 1609 KWH capability for the Orbiter 102. Mission energy requirements for seven days exceed the baseline cryogenic storage capability, i.e., tank sized for 1530 KWH. Activities underway are normal for this type of concern at this stage of vehicle development. The program is scrubbing electrical loads and equipment duty cycles to eliminate unnecessary power loadings. Monthly electrical power status reports are now being issued to assure high level contractor and NASA visibility and continued control.

Also, based on the prior experience of the Panel, particular interest is focused on the electrical power subsystem fluid tubing connections and the fluid line insulation. These two areas are shown schematically with brief descriptive material in Figures 14 and 15 . A test program is being developed to provide insulation, packaging, venting and installation design data for all insulated fluid lines, particularly polyurethane foam insulations and TG-15000.

3.1.1.6 Crew Compartment Pressurization and Toxic Gas Control

The pressurized crew compartment has a volume of approximately 70 m^3 or 2300 $ft.^3$, and contains three levels. The upper section, or flight deck, the mid-section containing an airlock, avionics and living area, and the lower section containing the environmental control equipment.

An atmospheric revitalization pressure control system provides the crew compartment and habitable payload modules with a two-gas atmosphere of nitrogen and oxygen. It also provides the oxygen to the emergency breathing subsystem and airlock support subsystem, and provides nitrogen for pressurization of the potable and waste water tanks. Table VI is a recap of the functions and performance requirements of this subsystem. Also, the atmospheric revitalization loop circulates and filters cabin air, controls the atmosphere CO_2 level, provides temerature control, and removes latent and sensible heat through the humidity control heat exchanger.

Cabin pressure is normally maintained at 14.7 ± 0.2 psia, but in the event of excessive cabin leakage an 8 ± 0.2 psia regulator is used. Sufficient make-up gas is available for 165 minutes pressure maintenance at this 8.0 psia value, assuming leakage equivalent to a 0.45 inch diameter hole. The atmosphere venting control provides for the relieving of excessive crew compartment pressure differentials whether negative or positive. This is a part of the pressure control system. The pressurization system is not designed to handle a second failure after 8 psia cabin condition exists. The crew will be on oxygen masks during emergency cabin pressure maintenance of 8 psia. Smoke detector units located in the avionics' bays require refurbishment every 2400 hours of operation.

Orbiter 101's pressurized compartment has passed its qualification tests.

3.1.1.7 Hydraulic Subsystem

Hydraulic subsystem provides power to actuate the aerodynamic flight control surfaces, main engine gimbals, main and nose landing gear, main landing gear brakes, the main engine valve controls and nose wheel steering.

Hydraulic power is provided by three independent, fifty percent power systems that provide the required degree of redundancy. The Panel was told that this approach minimizes weight, power extraction, and system complexity and emphasizes balanced design between systems. A number of components have been standardized through commonality procedures thus reducing the cost, development time, and logistic support.

This subsystem is active during liftoff, ascent and orbital insertion. It provides for concurrent operation of rudder, main engine thrust vector control and main engine valves. The subsystem is passive in orbit except for a low pressure, electrically driven pump in each subsystem. The pump provides circulation to assure thermal conditioning. Activation of the subsystem is prior to deorbit burn and operates through reentry and landing. The main pumps are driven by hydrazine fuel auxiliary power units.

Each hydraulic system utilizes a 63 gpm variable displacement pump, powered by an individual auxiliary power unit, all of which contributes to the redundancy of hydraulic power sources. Assignment of functions to each system is based upon optimum power extraction and distribution, maximum flight safety, and minimum weight without segregation of flight control and utility functions.

The hydraulic subsystem equipment is compatible with fluid specification MIL-H-83282. Its bulk fluid temperature is maintained below 275^{o} F. by a hydraulic fluid/water boiler heat exchanger.

The hydraulic distribution system consists of tubing and fittings fabricated from titanium. Approximately eighty percent of the tubing connections are of the permanent welded type. Minimum use of separable fittings improves the system integrity. Flared tube fittings are not used. Metal lines, designed to flex, are used in lieu of hoses, where possible, to reduce maintenance and improve safety.

Metallic, non-elastomeric and elastomeric seals are used as best suited for individual applications. Because of the upper temperature limit of 275° F., elastomeric seals can be used where they offer advantages over other sealing techniques. Experience with aircraft hydraulic systems has also demonstrated that satisfactory system operation can be achieved with non-elastomeric and metallic seals.

A hydraulic subsystem working pressure of 3,000 psi was selected on the basis of minimum cost, minimum risk and better stiffness quality. The system is capable of operating when subjected to normal g, zero g, and hard vacuum encountered in orbit.

The three fifty percent system configuration (fail-safe) was selected in preference to an original design of four fifty percent (fail-operational/fail-safe) configuration as a result of an extensive study of historical failure data of hydraulic components, the limited operational exposure time during ascent (abort decision time) and, of course, weight and cost savings.

From the point of view of reliability, the system requirements state that the hydraulic subsystem shall provide safe flight and landing in the event of any single failure which causes loss of one hydraulic string (fail-safe). The avionics/hydraulic interface is required to have a design that is two failure tolerant (fail-operational/fail-safe). The subsystem also has a maintenance requirement that it be consistent with the turnaround operation and be capable of being maintained in the horizontal as well as vertical position. Aerosurface controls operated by the hydraulic system are shown in Figure 16 .

The hydraulic subsystem interfaces with the following space orbiter subsystems:

(a) Flight control surfaces - elevons, rudder, speed brake, and body flap.

(b) Main engine thrust vector control.

(c) Utility loads.

(d) Steering, and landing gear brakes.

(e) Avionics - displays and controls, and flight control electronics.

Actuators used in the flight control subsystems (elevons, main propulsion system thrust vector controls and landing gear) have been approved by Rockwell International, Space Division, as acceptable risks based upon the very low probability of rupture or mechanical binding modes of failure.

While the Panel has not had the opportuntity to review this area in depth, the following questions would appear appropriate based on experience with other systems:

(a) To what extent are failure isolation techniques, such as hydraulic fuses, hydraulic circuit breakers, and return line check valves used to isolate a failed component.

(b) It has been a general rule that whenever hydraulic power is necessary for critical safety items, two independent subsystems are used. Why is this not the case for the Orbiter?

(c) Is there assurance that sufficient fluid cooling is available to maintain compatible fluid and seal temperatures?

(d) What parameters relating to actuator failure modes and life expectancy are being measured on the approach and landing test vehicle and on the Orbiter used for the first vertical flights? Does a mathematical model exist so that these measurements can be related to the design and component test data to further enhance hardware verification?

(e) What failure modes of the hydraulic subsystem result in the loss of the Orbiter - either directly or through the failure of a second system impacted by the failure of the first system?

(f) What is the method of validating these systems to achieve the necessary confidence in the design selected by NASA/Rockwell International. In other words, if the testing is not beyond the true expected conditions, how valid is the risk acceptance logic?

(g) What specific hardware/management controls are placed on the designers and manufacturers other than the prime Orbiter subcontractor?

3.1.1.8 Orbiter Separation Systems

The separation of the Orbiter from the External Tank involves three separation systems: (1) forward structural attach, (2) aft structural attach, and (3) Orbiter/ET umbilical plate separation, including the electrical umbilical separation. See Figure 17 .

Separation from the carrier aircraft (Boeing 747) involves forward and aft structural separation areas that are different from the Orbiter/External Tank arrangement, but the method of separation is essentially the same. See Figure 18.

The forward structural attach/separation configuration consists of a dual piston pressure actuated frangible attach bolt coupled with a standard nut. Each piston can fracture the bolt at the Orbiter Thermal Protection Subsystem moldline utilizing pressure generated by one of two Apollo-type pressure cartridges. Subsequent to separation, three centering plungers/springs align the bolt separation plane with the Orbiter TPS moldline by rotating the retained portion of the bolt within the Orbiter. No close-out door is required since the stub bolt and spherical bearing are essentailly flush with the TPS moldline.

The aft structural attach/separation configuration consists of two (right and left side) dual detonator frangible nuts coupled with two corresponding attach bolts. Each bolt has a retraction spring which, after nut fragmentation, retracts the bolt into the ET hemisphere so there will be no interference in the separation sequence. On the Orbiter side, the dual Apollo-type detonators are enclosed in a cover assembly whose function is to contain nut fragments and hot gas generated by the operation of the detonators, either of which will fracture the nut.

The Orbiter/ET umbilical plate separation configuration consists of two assemblies (right and left side). Each assembly contains three dual detonator frangible nut/bolt combinations which hold the Orbiter and ET umbilical plates together during mated flight. Each bolt has a retraction spring which, after release of the nut, retracts the bolt to the ET side of the interface. On the Orbiter side, each frangible nut with its Apollo-type detonators is enclosed in a debris container. Each Orbiter umbilical plate has three retractors which, after release of the three frangible nut/bolt combinations, retract the plate approximately two and one-half inches. Retraction motion does a number of things: (1) disconnects the Orbiter/ET electrical umbilical in the first half inch of travel, (2) releases the trapped fluids between the Orbiter and the ET oxygen and hydrogen shutoff valves, and (3) serves as a backup for closing the oxygen and hydrogen shutoff valves. Each Orbiter umbilical plate has three stabilizing bungees to hold it in position after separation.

The questions that would seem most appropriate at this time are:

(a) During separation of the Orbiter and External Tank, propellants are released from the feedlines. With hot surfaces, hot wires and so on, what is the potential hazard of the oxygen and hydrogen being ignited?

(b) What is the adequacy of the separation system and the operational procedures to assure a safe physical separation of the Orbiter and External Tank under nominal and non-nominal flight conditions? For instance, all separation modes normally require the use of the forward Orbiter RCS operation, assurance that the separation of each of the three points to be separated are done within the required time period. At what point during thrusting by propulsion units of the total Shuttle system can separation occur?

(c) What is the hazard of the Orbiter and External Tank recontacting after separation?

(d) What is the ability to maintain the oxygen valves and hydrogen valve in the open position up to separation and the ability to assure closure after separation?

(e) What is the basis for confidence that there is no potential hang-up problem at the aft structural separation interface after the attachment bolt is retracted?

(f) Since umbilical door release is accomplished through the use of a spring-loaded latch on the External Tank, what is the hazard from door, door hinge, or latch failure?

3.1.1.9 Structures

The Panel has not examined the basic Orbiter structure in any detail but has opted to look at those items from the standpoint of the test program used to validate the structure. The TPS and doors are covered under separate sections of this report. Another view of the Orbiter structure is obtained from an evaluation of the interface between the Orbiter and the External Tank and the Orbiter interface with the Main Engine. Added to this is the examination of the abort operations' area which includes an understanding of the ability to meet intact abort modes requirements.

3.2 Space Shuttle Main Engine

The Orbiter Main Propulsion Subsystem consists of the Space Shuttle Main Engines (SSME), the External Tank (ET) which stores and supplies liquid oxygen and liquid hydrogen for the SSME's, and a system of valves, plumbing, pumps, etc. located in the Orbiter which deliver the propellants to the engines.

The three main engines are started during the countdown. When they attain a ninety percent thrust level, the Solid Rocket Motors are ignited and liftoff is achieved. During the burn of the engines, they are throttled as required to limit vehicle acceleration to 3g. Gimbaling of the main engines provides steering during ascent in conjunction with Solid Rocket Booster thrust vector control. The SSME's burn for about eight minutes. Final boost into orbit is provided by the Orbital Maneuvering Subsystem (OMS). Each of the three main engines is approximately fourteen feet long with a nozzle about eight feet in diameter. The engines produce a nominal sea-level thrust of 375,000 pounds each and a vacuum thrust of 475,000 pounds. They are throttleable over a thrust range of fifty percent to one-hundred and nine percent of the nominal thrust level.

Orbiter interfaces are basically of three types - fluid, electrical, and structural. The fluid connections consist of the main propellant lines which transmit liquid hydrogen and oxygen and the fluid connections located at the interface connect panel mounted on the vehicle. These provide fluids to and from the individual engines as follows:

(a) Hydraulic supply to and from the engine.

(b) Nitrogen purge (ground) to the engine.

(c) Helium supply to the engine.

(d) Fuel and oxidizer bleed from the engine.

(e) Gaseous fuel and oxidizer (pressurant) from the engine.

The propellant fluid connections at the interconnect panel consist of bolted swivel flanges. All remaining fluid connections are attached with bolted flanges except for the hydraulic system which uses self-sealing quick disconnects. Flexibility for these joints are provided with flex hoses on the engine side of the interface.

Electrical interface between the engines and the Orbiter are made at the electrical connect interface panel located on each engine. These interfaces consist of the following:

(a) Single 28 vdc power connector.

(b) Two 115/208 vac power connectors.

(c) Three communication and data transmission connectors.

AC power of 115/208 volt, 400Hz, 3-phase, is supplied to the engine controller and the controller conditions the power to the requirements of the various engine actuation and instrumentation subsystems. The 28 vdc is provided to operate both the SSME controller heaters and a redundant coil on each engine's emergency pneumatic shutdown control solenoid valve which is normally open. Engine shutdown cannot occur when the crew activates the engine limit control to inhibit engine shutdown. Operational and checkout commands and engine flight data are supplied via the electrical interface connectors at the engine-supplied electrical interface connect panel. Commands consist of engine start, shutdown, thrust level changes, checkout, and sequence checks. Engine flight data to the vehicle consist of information necessary for malfunction display, fault isolation, maintenance recording, trend analysis, performance monitoring and checkout. Three parallel redundant connectors provide a path for the Orbiter-to-engine commands. A minimum of two of the three commands must be received before the engine response can be initiated. Two of these connectors are also employed to transmit the engine flight data back to the Orbiter. The aft Orbiter thrust structure, the third interface, is built up with a titanium/boron epoxy material. Another interface is the honeycomb-base aluminum heat shield with insulation to protect the SSME from thermal inputs.

Integrated testing of subsystems is a critical milestone in the SSME program. It will be conducted at the National Space Technology Laboratories (NSTL) in Mississippi. The first engine firing at rated power level will take place at NSTL on a modified Apollo firing test stand in the winter of 1975. This will be followed by the first throttling test over the rated power level range. The Integrated System Test Bed (ISTB) will demonstrate the design's ability to handle the high pressures and repeatable operations required of it. The ISTB engine configuration varies somewhat from the flight-type engine in the following areas: there is no LOX tank pressurization heat exchanger, changes in material (high pressure fuel line, small fluid lines, powerhead ducts , and modified insulation), and the electronic controller assembly is not a flight type unit but is a bench test unit built in racks. The ISTB has progressed as follows:

Assembly completed	3/13/75
Checkout completed	3/21/75
ISTB shipped	3/25/75
ISTB at NSTL	3/28/75
ISTB installed at NSTL	4/7/75
Test Readiness Review	5/7/75
ISTB first firing	June 1975

There is no gimbaling planned during the ISTB program.

3.2.2 Subsystems Critical to Crew Safety

For the purposes of this report, the Space Shuttle Main Engine as a system is divided into the following subsystems:

(a) Combustion devices

(b) Turbo-machinery

(c) Pneumatics

(d) Propellant valves

(e) Hydraulics

(f) Controller

(g) Igniters

(h) Electrical harnesses

(i) Instrumentation

(j) Interconnects and SSME/Orbiter interfaces

(k) Gimbal

As with the Orbiter element of the Space Shuttle program, the Panel recognized that any one or a combination of these subsystems and their components may be considered as affecting crew safety, but from the point of view of the Panel it was necessary to determine which of these should be focused on during the review period. The basis of this focus was (1) on subsystems and/or components extending the technical (material, fabrication, etc.) state-of-the-art in the literal sense or in the application, (2) those subsystems and/or components which prior program "lessons" have indicated as areas of concern, (3) areas which the Panel members considered most vulnerable to "human error," and (4) areas which can affect crew safety but which cannot or will not have been adequately tested or validated prior to first flight. With these criteria in mind the Panel examined the following subsystems in some detail:

3.2.2.1 Engine Electronic Controller Assembly

3.2.2.2 Main Combustion Chamber

3.2.2.3 High Pressure Turbo-Pumps

3.2.2.4 Heat Exchanger

3.2.2.5 Hot Gas Manifold

The Controller is significant for crew safety because of its responsibility for detecting, monitoring, and controlling engine failure, thrust and propellant mixture ration, and engine starts and shutdowns and engine gimbaling.

The manifold, exchanger and chamber are of particular significance because they have complex welds and are subject to hydrogen embrittlement during operation. Material safety factors may be reduced through flow erosion or fabrication problems. Finally, it is difficult to inspect the finished item.

Also, the Panel reviewed the following areas to assure that risk assessment was receiving appropriate attention:

3.2.2.6 POGO

3.2.2.7 Ground Operations and GSE

3.2.2.8 Hydraulic Fluid

3.2.2.9 Lightning Effects

POGO results from dynamic coupling of the structure, propulsion, and flight control subsystems during all phases of powered flight under all possible payload variations. Thus POGO suppression hardware has had to be designed to eliminate coupling and the resultant structural instabilities.

Ground operations and ground support equipment are being developed to discover failures and predict malfunctions before they occur.

The Panel had asked the Program to review its use of "red oil" hydraulic fluid and consider alternative hydraulic fluids that are more fire resistant. The Program has made a change and the Panel reviewed the new choice.

Lightning was a concern because of its impact on such subsystems as the Controller.

3.2.2.1 SSME Controller (Electronic Controller Assembly)

Systems Design

The SSME utilizes a full-authority digital electronic control with hydraulic servo-actuated valves. The Controller operates in conjunction with engine sensors, valves, actuators, spark ignitors, harnesses, and an operational computer program (software) to provide a self-contained system for:

(a) Closed loop engine control.

(b) On-board engine checkout.

(c) Engine limit monitoring.

(d) Engine start readiness verification.

(e) Engine start and shutdown sequencing.

(f) Engine maintenance data acquisition.

The engine/controller functional relationships are shown in Figure 19. The controller electronics arrangement is shown in Figure 20. In that same figure is shown the responsibility of the two Honeywell organizations.

Characteristics of the Controller of interest are:

(a) Overall dimensions .... 23.5" x 14.5" x 17"

(b) Weight .... 197 pounds

(c) Input power .... 472 watts to 636 watts

(d) Convective cooling .... (primary mode)

(e) Temperature environment ...... operational -50° to + 95° F.

Non-operational - 200° to + 200° F.

(f) Vibration environment .... sine 24 g's peak

random 22.5 g's root mean square

(g) Unit is mounted on engine using a three-point hard-mount.

The electrical harness assemblies between the engine interface and the Controller are of two types - conventional and flexible armored. Conventional harness is used where redundant electrical functions are carried through separate connectors and will be physically routed independent of each other. Flexible armored harness is used where redundant electrical functions cannot be physically routed separately.

Panel's Initial Review

Prior to reviewing the Controller program, the Panel requested specific information as background data on this critical hardware. The documents requested were (1) reliability analysis and test data that documented the Controller configuration and its projected ability to support mission objectives, (2) prediction analyses for the expected mean-time-between failure rates and the basis upon which such predictions were made, (3) trade-off studies between the Controller using plated-wire type memories and a design using the latest of the more traditional type cores. This material was received and reviewed by the Panel and staff. Typical data included in the response is shown in Tables VII to IX .

The Panel then undertook a series of inspections.

Status of the Controller program in the early summer of 1974 looked like this:

(a) Design verification tests were completed on the input electronics, output electronics and the computer interface electronics. The digital computer processor logic was proved through the use of a Honeywell HDC-601 computer unit and on the engineering and bench test SSME controller assemblies. The digital computer memory design, including the use of plated-wire, was proven through testing of a "half-stack" unit. The half-stack test was a test using a rack-mounted integrated memory assembly. The Controller power supply was undergoing expedited documentation (specifications, etc.), procurement, and fabrication. At the same time power supply breadboard tests showed that there were numerous problems with the design. Some of the problems associated with the subsystem/circuit/component items were power supply voltage below minimum allowable, output ripple, and failure of inverter transistors, master interconnect board pins and sockets pulling out, deflecting or not matching. SM-1 (structural model) vibration testing had revealed foam retention and seal problems. There were parts' problems with integrated circuits and connectors. Integrating the digital computer unit components was a problem as was the integration of the total Controller. Noise in the memory and parity errors in the computer unit also were concerns at that time.

Thermal design of the package was verified by analysis and tests on the structural model (SM-1), which was not, of course, exactly like the flight design. However, given the excellent correlation between analysis and test results and the piece part temperatures and conduction rates to the case, there was sufficient margin remaining in the design to allow for production process variables and for some modifications.

Vibration tests were conducted with the SM-1 unit which verified that the general packaging concept would meet the requirements. Problems surfaced with regard to the case aluminum seal which leaked, excessive resonances in some of the parts, and the retention of the half-stack card and foam assemblies. Solutions for these mechanical problems were identified but further testing was necessary to prove that the solutions would actually work. Environmental test for salt, humidity, etc. were to be conducted. Design verification testing for thermal conditions was to be conducted on the memory boards, printed wire boards, and master interconnect boards.

A necessary adjunct to the development of the Controller hardware and software are the many test items and facilities which prove design and fabrication concepts and validate the prototype and flight hardware. The software verification facility was operational, the design for the command and data simulator design complete, and hardware test equipment of many types were built and in use. Such test equipment as "automatic wiring board test stations", "power Supply Conditioner Test Unit," and "Memory System Exerciser" were proceeding satisfactorily.

Software design was demonstrated on the Controller engineering model and the bench test units. The electrical interface between the engine and the Orbiter was verified as was the ability of the software to conduct engine start, mainstage control, and engine shutdown. At that time the computer acceptance test program design was complete and 95% debugged, the Controller acceptance test program baseline design was complete but not debugged, and the operational program design was complete with 50% of it coded and debugged.

There was adequate experience with the development of the plated-wire memory to warrant confidence in the technology. However, there did not appear to be an understanding of the fundamental physics to assure that surprises could be anticipated and a timely course of resolution decided upon and implemented. If additional surprises did occur, they probably could be solved by trial and error, given sufficient time, but such surprises would probably impact the then very tight schedule requirements. At that time the half-stack test of the rack mounted integrated memory system and the structural thermal verification program were completed. Fabrication improvement was indicated by the acceptance trend of plated-wire assemblies.

While there was no single reporting format available which systematically stated the significant lessons learned from the Viking program and their disposition with regard to the Shuttle program, the new program manager had his staff review the minutes and audits from numerous Viking reviews and identify specific actions. As a result of this review, design changes were incorporated into the Digital Computer Unit. Daily production schedule reviews were instituted with closed loop corrective action and follow-up for all problems defined. The process specifications and the training program for the production and inspection workers were strengthened. Management and supervisory levels made it their business to have more contact with the total Viking and Shuttle personnel. Viking audit disciplines were incorporated into the Honeywell basic management and technical system.

Current Status

Since its initial review in the summer of 1974, the Panel has examined the SSME and its Controller in September 1974, January 1975, and April 1975. The current Controller status as seen from these reviews looks like this:

(a) SM-1 (structural model) thermal and vibration tests have been completed and the structural and thermal math models have been verified.

(b) The breadboard controllers BT-1 and EM-1 have been in use and the Controller functions such as start-up and shut-down have been demonstrated.

(c) The command and data simulators have been used extensively as have the Controller checkout consoles and laboratory model computer used in the integration of the Controller subassemblies.

(d) The digital computer unit number SN-1 has been completed and integrated in the first prototype controller, PP-1. This unit, however, has experienced intermittent parity errors which are under study at this time. All of the Controller functions of the PP-1 have been exercised and some out-of-specification conditions have been surfaced which also are being examined for proper resolution.

(e) The quality of the workmanship and inspection system has been improved, with the result that the rejection rates for such things as plated-wire memory boards has been reduced to a very acceptable level.

(f) The BT-1 unit, to be used with the SSME Integrated System Test Bed test program, was successfully checked in March 1975 and has been delivered to NSTL for installation into the ISTB facility. SSME to Orbiter interface documentation (ICD 13M15000) has been issued and is under standard control of the configuration control system and the interface working group.

(g) Operational philosophy for "out of limit" signals has been defined and agreed to as shown by the current design. This design provides for engine sensor inputs to be out-of-limits three consecutive check periods before the input is "declared" failed, which is called a "three strike" concept. A part of this system provides for rechecking critical parameters immediately during the same major status loop check. A major status loop check takes about twenty mil seconds. Less time critical parameters are rechecked during the next two major sense-reporting cycles. At the same time the out-of-limits data are not used by the engine control system at that time. For internal Controller parameters the "two-strike" concept is used in which two consecutive out-of-limit conditions must exist before that item is declared "failed." Short term anomalies will not cause pre-mature loss of redundancy, e.g., shifting to the second computer section of the Controller or engine shutdown.

(h) The power supply units for use in the PP-1 and PP-2 Controllers have been completed and tested satisfactorily. Design verification tests have been conducted, resulting in a low degree of electromagnetic interference beyond specification limits. This does not appear to be a major problem.

(i) Master Interconnect Boards, because of their complexity, have posed numerous production problems. Four have been built for use in the PP-1 and PP-2 units in addition to the development units. To date the development tests have been completed. Manufacturing processes along with alignment fixtures and insertion tools have been established. The design verification test hardware is being built. A problem still to be resolved is the noise being coupled into the memory sense lines due to wire routing and inadequate shielding. Modifications are being incorporated to add sense-line-shielding on the Master Interconnect Board and to reroute control sense lines. Additional improvements are being evaluated in case they are needed in the wiring approach to the memory area of the board.

(j) Four memory systems have been built for the PP-1 and PP-2 Controller units and twelve half-stacks have been built and tested. Several hours of memory operation have been accomplished at the digital computer unit level. There have been intermittent parity errors, and a noise problem has been identified in integrated testing of the Controller. In addition to the fixes to the Master Interconnect Board, changes to increase the memory plane shielding and plated-wire output are being studied in order to increase the signal to noise ratios. To put the parity error problem in perspective, the extent of the testing on the two memory channels should be considered. Channel "A" operated over the temperature range at the digital computer level for eight hours with only a single occurrence of parity error. Channel

"B" had error-free operation over the temperature range for some 54 hours at the memory system level of installation, and approximately 100 hours of operation at room temperature with comparatively few intermittent parity errors at the Digital Computer Unit level.

(k) The basic software elements and/or routines are as follows:

Executive

Ground checkout

Self-test

Start preparation

Power range control

Vehicle commands

Limit monitoring

Sensor processing

Output monitoring

Failure response

Post shutdown

Constraints on the software programs are the memory size of 16,384 words and the Controller major cycle time of 20 milliseconds. In December 1974 the memory capacity was exceeded. As a result there is an effort at this time to reduce the word requirement by proper software programming and or some reduction in requirements. At this time the emphasis is on meeting the SSME Integrated Subsystem Test

Bed program. MSFC noted that considerable effort has been placed on providing the proper software. For example, the contractor established a shift operation. Schedules are also established and progress is reviewed on a daily basis. "Memory scrub groups" have been established at Honeywell, Rocketdyne and NASA.

3.2.2.2 Combustion Devices

Systems Design

The function of the Main Combustion Chamber is to contain and direct the forces of combustion generated by the burning of the propellants. The hot gases are accelerated to sonic velocity at the throat and supersonically expanded to an area ratio of 5:1 at the interface with the engine main nozzle. The Main Combustion Chamber consists of a structural outer jacket, regeneratively cooled liner, and inlet and outlet manifolds. Two thrust vector control struts are attached to it as are mounts for the engine electronic controller assembly. The Main Combustion Chamber fabrication problems or concerns are similar to those described for the hot-gas manifold unit. In addition the cooling of this combustion chamber requires a rate of heat removal three times higher than any previous liquid fueled engine, 100 btu/ft^2/sec. The number of welds used in producing the chamber are about 112 of which 16 are electron beam welds.

Current Status

Main engine combusion devices have had fabrication problems during their development period. Main Combustion Chamber and nozzle fabrication has been completed in support of the Integrated System Test Bed program hardware, including successful proof-testing to 1.2 times the rated-power level operating conditions, or about 6800 psi. The augmented spark igniter has been demonstrated successfully, including a 600 second run at full power level conditions. Subscale model of the main injector (40,000 pound thrust unit) has been demonstrated. Hot fire tests have been conducted on the oxidizer preburner and the fuel preburner, which all appear to meet performance requirements. Flow induced vibration was noted in some of these tests, but this apparently has been remedied. The LOX tank pressurization heat exchanger, located in the LOX side of the hot gas manifold assembly, is a critical item in the engine combustion system. The present heat exchanger design requires rigid manufacturing and inspection control and verification testing to assure an acceptable unit. Rocketdyne feels that this can be accomplished.

3.2.2.3 Turbomachines

Systems Design

The high-pressure fuel turbopump receives fuel from the low-pressure fuel pump and boosts the pressure to the level required for the pre-burners. The fuel is then discharged through the high-pressure fuel pump discharge duct to the main fuel valve. This turbopump consists of a three-stage centrifugal pump drive by a two-stage reaction turbine. During propellant conditioning, the liftoff seal is closed around the pump shaft, preventing LH_2 from flowing into the turbine area and out through the hot-gas manifold to the main injector. At engine start, the liftoff seal is actuated by the pump pressure at a pump speed of approximately 7000 rpm. During mainstage firing, the pump reacts to throttling commands by changing discharge pressure and flowrate. The lift-off seal reseats when the pump pressure decreases to a speed of 7000 rpm.

The high-pressure oxidizer turbopump receives oxidizer from the low pressure pump and boosts the pressure to a sufficient level to provide adequate flow-rate and pressure to the thrust chamber and the preburners. Engine start activates the pump intermediate seal purge that provides an inert barrier between the pump and turbine during operation.

Current Status

Material presented to the Panel indicates that the turbine nozzle castings and turbine strut forgings around the turbine have been the major problem areas. The initial vendor was unable to cast the nozzles due to shrinkage, failure to fill molds, and erratic material problems. To resolve the problem quickly, a change in vendors was made in July 1974 and nozzles were successfully cast using a new material (INCO 713LC instead of MAR-M-246). It turned out that the life for the 713LC type nozzle casting was inadequate. Work was resumed on the use of the original material and it was found that the new supplier was in fact able to produce successful nozzles with MAR-M-246 material that now appear to meet the turbine nozzle requirements. These nozzle castings are still receiving MSFC's attention to assure adequate hardware is available for the early engines in the SSME program. The turbine inlet struts had some material problems regarding acceptable axial strength of the forgings. This problem has been resolved and the forgings are adequate to meet program needs.

3.2.2.4 Heat Exchanger

System Design

The heat exchanger provides oxidizer gas pressurant for vehicle LOX tank pressurization. This heat exchanger is a multipath, single-pass, cross-flow device installed in the LOX side of the hot-gas manifold at the high-pressure oxidizer turbopump turbine exhaust. The supports for the heat exchanger tubes are mounted to the liner wall so as to allow small movements during expansion and contraction of the tubes. The tubes enter and leave the hot-gas manifold through flared projections of the manifold liner. The flared projections provide stagnant gas pockets for reduction of thermal stresses at the tube-to-oxygen manifold attach welds. The heat exchanger is depicted schematically in Figure 21 . The major concern here is with the heat exchanger coil material and its ability to be assembled and then to remain virtually leakproof during its operational life. For instance, a leak could permit ignitable mixtures of oxygen and fuel-rich hot gas to enter the oxygen supply line or allow oxidizer into the hot gas manifold with ignition that could also damage an adjacent coil or the liner and manifold wall.

Current Status

The design and manufacturing approach being used to reduce the possibility of this hazard include a number of actions.

An ultimate factor of safety of 1.75 is used rather than the usual 1.4. Where fatigue life of 240 cycles normally is required, this has been increased to 1450 cycles for bifurcation joints, to 4500 cycles for weld joints, and to 26,000 cycles for parent metal.

Design verification structural tests will include leakage checks, vibration, proof pressure cycles, ultimate pressure, and low cycle fatigue tests.

Quality control on components will use ultrasonic, penetrant and x-ray, and helium leak tests (1×10^{-6} scc/sec at limit pressures). The Panel questioned the use of a leakage rate of less than 1×10^{-6} scc/sec at limit pressures noting that this leakage rate appeared excessive in determining the acceptability of the heat exchanger. This is being reevaluated at this time.

A modification being considered to the LOX pressurant control system which would interconnect the heat exchanger discharge upstream of the Orbiter flow control system, which would insure valve inlet pressure being above the hot-gas manifold pressure.

3.2.2.5 Manifold

System Design

The hot-gas manifold serves as the structural nucleus of the engine and provides gas passage interconnection for the preburners, high-pressure turbopumps, and the main injector. Hydrogen-rich hot gas (hydrogen and oxygen) flows through this manifold and then into the main injector. Cooling of the hot-gas manifold is accomplished by using double wall contruction (a structural outer wall and an inner liner). This provides a flow path for hydrogen gas coolant exhausting from the low-pressure hydrogen turbine. This configuration isolates the structural wall from the hot gases flowing within the inner liner.

Current Status

This hardware is fabricated with complex weld which has required considerable in-process rework at the fabrication location. Critical to achieving successful weld is the alignment of the joints and the materials and processes developed for such welds. Proper alignment reduces the stress concentrations and discontinuities that normally cause problems in welds. All manifolds are analyzed for weld adequacy. To further reduce induced stresses, prestraining and an annealing, heat treatments are utilized. Hydrogen-rich mixtures, particularly at high pressures (up to 6000 psi in part of the engine), leads to the possibility of metal embrittlement problems. The possibility of cracks, warpage and structural failures obviously affect the engine operation and performance from simple gas leakage to engine shutdown, and in extreme cases potential aft compartment fire or explosion. Based on the material provided to the Panel, NASA and its contractor are aware of these problems and continue to place very heavy emphasis on eliminating the fabrication and material problems, and on the test program to validate the design and manufacturing processes.

3.2.2.6 POGO Suppression

The Problem

POGO is not only an SSME problem but also must be viewed from a "systems" standpoint. The discussion here deals with the hardware as currently designed and as attached to the SSME's themselves. Systems integration aspects are covered in more detail in Section 6 of this volume. The Panel's concern with POGO effects goes back to Saturn V launch vehicles in the Apollo program. Most large, pump-fed rocket vehicles have had moderate to severe longitudinal oscillations caused by POGO instability. Such oscillation can result in an environment severe enough to cause structural damage and affect crews physiologically. POGO is a closed-loop phenomenon involving fluid-feed-system pressure oscillations which result in engine thrust perturbations and structural motions. These may be visualized as beginning with small vehicle accelerations that produce variations in propellant pressure and flow rates, which in turn cause thrust variations, resulting in increasing vehicle oscillations.

Elements of the Space Shuttle Vehicle system involved in POGO are:

(a) Long liquid oxygen supply line.

(b) Asymmetric Shuttle structure and thrust vector couples, and coupling of flight control and POGO instabilities.

(c) Main propulsion system (SSME's, ET, etc.) which operates from liftoff to orbit with extreme changes in vehicle structural characteristics and turbo-pump inlet pressures.

(d) Space Shuttle's main engines themselves, with their LOX and LH_2 high and low pressure dual pump systems.

The depth of NASA and contractor efforts to assure that POGO does not become a Shuttle operational problem can be seen in planning, documentation, testing, and analytical work being performed to resolve this concern. This includes the "POGO Prevention Plan" JCS 08130, dated January 6, 1975, as well as studies to determine the need for POGO suppression, and to add the suppression system. Such groups as the POGO integration Panel and the independent MSFC POGO analysis team, have been working this challenge.

Suppressor design requirements have been defined as follows:

(a) Location as close as practical to the High Pressure Oxygen Turbo-pump.

(b) Volume about 0.6 cubic feet or equivalent with ability to increase to more than one cubic foot if test program indicates this to be necessary.

(c) Damping of fluid surges (frequency of pulses) over a broad frequency range; inertance less than 1.1 x 10^{-3} sec^2/in^2.

(d) Minimal fluid pressure-drop in the suppressor.

Comparison between the Saturn V and Space Shuttle engine/fluid systems is shown in Figure 22 . The POGO suppression system and its components are shown in Figures 23 and 24 .

Current Status

POGO mechanisms are known to be complex, and a continuing analytical program is being pursued to understand the phenomenon and its implications. The suppressor has been baselined. An extensive ground-based program is being conducted to verify the design. Extensive use has been made of Saturn data in designing the test program. Tests are being conducted at MSFC, Martin Marietta Company, Rocketdyne, and NSTL sites. The location, type, size and inertance of the proposed system have been arrived at after a thorough design trade-off study. Analysis of abort situations and their impact on the design of the POGO suppressor have to be accomplished to assure maximum safety. But the proof-of-the-pudding can only be found during flight tests under actual environments.

It appears that the liquid hydrogen does not contribute to any degree to the POGO problem, and there is no apparent need for a suppression device in the liquid hydrogen fuel system. Preliminary examinations indicate that the Solid Rocket Motors do not contribute to any degree to the POGO problem, but the analysis is continuing.

3.2.2.7 Ground Operations and Ground Support

SSME's are designed for automatic checkout and fault isolation, use of "line replaceable units" with good accessibility and long life, and to accommodate the so-called "condition monitored" concept. This concept has as its objective the ability to discover failures before they occur, using nondestructive evaluation methods, and to eliminate premature maintenance.

SSME controller assembly has automatic checkout capabilities for self-test and fault isolation to the line replaceable unit level. Working in conjunction with ground equipment, it conducts the following tests:

(a) Pneumatic

(b) Actuator

(c) Sensor

(d) Flight readiness tests

(e) Redundancy verification

Panel interest will continue in this area to assure that ground operations and equipment do not adversely affect the engines and associated hardware during maintenance and preparation for launch.

The following GSE status was presented to the Panel recently:

(a) There are no significant GSE problems known at this time.

(b) While economic problems have resulted in quantitative reductions of GSE, there have been no quantitative cutbacks that would affect safety.

(c) Major GSE units have completed design verification testing.

(d) Majority of GSE components are now in service.

3.2.2.8 Hydraulic Fluid

Introduction of the MIL-H-83282 hydraulic oil in place of the original "red oil" has been made at all locations working on the SSME: NSTL, MSFC, Hydraulic Research Company, and at Rocketdyne. To date there appear to to be no functional problems associated with the use of this fluid, and laboratory tests continue to be conducted to assure that the fluid when in operational use will meet requirements under all induced environments.

3.2.2.9 Lightning Protection

The requirement currently on contract for lightning protection is MIL-B-5087B, Amendment 2, 31 August 1970, "Bonding, Electrical and Lightning Protection for Aerospace System." Use of this standard is currently under review, with the probability that it will be replaced by the NASA publication JSC 07636, "Space Shuttle Lightning Protection Criteria." Assessments are being made during the May 1975 time-frame with regard to lightning field amperage components, direct strike capability, launch constraints, cable shielding requirements and cost and weight impacts. Results of these assessments will be examined by the Panel during upcoming reviews. Lightning protection for the Shuttle as a system is discussed in more detail in Section 6 of this report.

3.3 External Tank Project

The External Tank is a part of the main propulsion system, along with the main engines and interconnecting portions of the Orbiter vehicle.

In this section the discussion will be devoted expressly to the external tank and peripherally to those significant interfaces with the Orbiter and Solid Rocket Booster that affect crew safety.

The External Tank is the only element of the Shuttle system that is discarded after depletion of its oxidizer and fuel resources. Because it is expendable, great emphasis has been placed on low cost production of this tank. The external tank is being designed, developed and manufactured by the Martin Marietta Corporation at the Government-owned Michoud Assembly Facility in Louisiana.

The External Tank consists of three major components: (1) a liquid oxygen tank, (2) an inter-tank, and (3) a liquid hydrogen tank. It is of aluminum construction utilizing a spray-on foam insulation and spray-on ablator for thermal protection. A configuration is shown in Figure 25 . In September 1974 a Preliminary Design Review of the tank was conducted; the Critical Design Review is scheduled for the fall of 1975. Fabrication and assembly of the LOX and liquid hydrogen tanks for the structural test article will begin in the summer of 1975.

3.3.1 Subsystems Critical to Crew Safety

The tank can be divided into the following subsystems:

(a) Structures

(b) Propulsion and mechanical

(c) Electrical

(d) Separation and dispersion

(e) Thermal Protection Subsystem

(f) Ground support equipment and logistics

Particular attention was given by the Panel to those components or situations most critical to crew safety. These were chosen on the basis of the criteria used on other elements of the program - potential problems utilizing experience on prior programs and components that could critically degrade the performance of the Orbiter or SRB if they were improperly designed, could not be tested or analyzed to the degree necessary for confidence in them, or failed to operate during critical mission sequences. To illustrate, the Panel in its review of structures gave particular attention to fracture control. A review of the propulsion system focused on the anti-geysering system. Review of the electrical system focused on controlled use of teflon wiring as well as on lightning protection.

Weight control is as important a management concern on the External Tank as on the other elements of the Shuttle program. The next control weight has been set at 72,360 pounds. With a current estimated weight of 71,445 pounds, the margin is 915 pounds. Therefore, the Panel is sensitive to the impact of weight control on decisions affecting crew safety.

3.3.1.1. Structures

System Design

The structure must retain the liquid oxygen and hydrogen within their respective tanks and must serve as the structural backbone of the launch and ascent Shuttle vehicle as well. Material provided to the Panel indicates that the design and construction of the structural portions of the External Tank follow the large Saturn tank and Titan tank methods, as well as the use of current sophisticated design tools developed by NASA (NASTRAN).

In light of prior program experience, the Panel reviewed the actions taken by NASA and contractor management to insure that the initiation or propogation of cracks or cracklike defects in the External Tank will not cause structural failures or unacceptable leaks.

Current Status

Fracture control plans have been developed to cover the phases of design, fabrication, test, environmental control, inspection, maintenance, repair, and acceptance procedures. A Fracture Control Board has been established to assure the plans are implemented. The straight polarity TIG welding process has been selected. Vendors for critical formed parts, such as gores and caps, have also been selected.

Both NASA and the contractor feel that the initial processes provide a reasonable basis for confidence.

Some fracture mechanic limits for tank welds are shown in Figure 26 .

3.3.1.2 Propulsion and Mechanical System Design

The External Tank propulsion/mechanical subsystem delivers LOX and liquid hydrogen to the Orbiter interface from the external tankage. The propulsion and mechanical subsystem is comprised of the liquid oxygen feed system, liquid hydrogen feed system, LOX tank pressurization and vent/relief system, intertank and tank environment control systems. The separation system, normally considered a part of the mechanical and/or structures' system, is discussed under a separate section later in this report. There are three separate mechanisms associated with the External Tank propulsion subsystem: (1) intertank umbilical disconnect, (2) right aft ET/Orbiter umbilical LOX disconnect, and (3) left aft ET/Orbiter umbilical liquid hydrogen disconnect. Only the intertank disconnect is discussed in this section since the other two are a part of the in-flight separation system.

One of the more significant design features of the external tank that should provide for greater hardware reliability and reduced mission risk is a dual flange seal with the capability of monitoring leakage through the primary seal. This seal is used at all major structural tank connections. See Figure 27 .

The LOX pressurization line is supported by 29 sliding supports and three fixed supports. These supports are bolted to floating anchor nuts in brackets welded to structure on the LOX tank. A phenolic insulation block is placed between the support and the tank to reduce heat transfer. These same supports also serve the larger anti-geyser line and the electrical tray. Seven flexible joints accommodate thermal and dynamic deflections. Figure 28 shows not only these lines but the LOX propellant feed-system as a whole.

The vent/relief assembly serves two functions: (1) tank venting during propellant loading, which controls the boil-off rate, and (2) relief of the ullage pressure to protect the tank structure in the event that it exceeds a preset value.

The liquid hydrogen feed system is similar to the LOX system. The liquid hydrogen pressurization line assembly provides the means for transmitting adequate pressure and for the correct rate of flow of LH_2 to the Orbiter main engines. The LH_2 recirculation line is a 4-inch vacuum-jacketed line which provides a return path for the hydrogen recirculation flow that used to thermally precondition the SSME prior to initiation of engine start. The vent/relief assembly serves the same two functions as the similar system in the LOX feed system.

The tank environmental control or conditioning system includes LOX, liquid hydrogen and inter-tank purge hardware. Propellant tank purge is accomplished prior to propellant loading. The inter-tank purge uses dry gaseous nitrogen to remove contaminants from its area and to maintain the temperature of the inter-tank area at 80 $\pm$ 15 degrees F.

External tank-to-ground interface consists of an environmental control system disconnect, a gaseous hydrogen vent line disconnect, and LOX and liquid hydrogen vent valve pneumatic control line disconnects. See Figure 29 .

The Panel gave particular attention to the control of the possible hazard of geysering. Geysering is the rapid upwelling of LOX into the tank ullage area; this can cause a rapid reduction of the ullage temperature, reduce the ullage pressure and, in the worst case, result in the collapse of the LOX tank. This phenomenon has been found on prior large liquid rockets and occurs when a comparatively high density cryogenic fluid contained in a line or tank begins to heat up and bubbles form at a progressively increasing rate. As a bubble matures it begins to rise through the liquid, due to its reduced density. At the same time the liquid head (pressure) on the bubble is constantly being reduced. As the bubble moves upward it accelerates and pushes liquid ahead of it. When the bubble reaches the tank, the liquid above it is expelled upward through the liquid surface into the open tank area with great force. It is not unusual for this slug of liquid to weigh several hundred pounds. Thus, in addition to the possible tank pressure reduction, resulting in conditions conducive to tank collapse, there is a danger of the slug itself hitting internal structure and damaging the structure and any lines or instrumentation therein. The return of this liquid can also result in "water hammer" effects.

The geysering action is shown schematically in Figure 30 .

Current Status

NASA/MSFC and Martin Marietta Corporation have baselined what appears to be an acceptable anti-geysering system and test program, all of which must be completed before the initiation of the main propulsion test program at NSTL. To prevent geysering it is necessary to agitate the liquid column to prevent stratification or layering during the ground fill sequence when lines and tank are relatively warm. Current design plans are to use helium injection system as shown schematically in Figure 31 . Actual design of the system is still under study and analysis because the initial design concept as proposed was considered less than optimum. Location of the function of the 4-inch LOX anti-geyser line with the 17-inch LOX main-feedline can potentially cause unpredictable flow patterns as well as nullify the desired effect of the system. This could happen if there is a ground helium supply failure for any reason because the LOX vent valves then would close and helium injection for anti-geysering protection would be terminated. It is proposed that a proper degree of redundancy be provided in the ground system to assure a fail safe arrangement. A test plan approach has been developed to support the LOX anti-geysering program. The test plan itself is still in work along with the type of hardware and test facility to accomplish the objectives of the program. Test schedule is:

Hardware on-site and installation start	February 1, 1976
Test start	September 1976
Test completion	March 1977

Proposed test configuration is shown in Figure 32 .

3.3.1.3 Electrical Subsystem

System Design

For the design development test and engineering phase of the Space Shuttle program, the external tank electrical subsystem includes: (1) operational instrumentation, (2) electrical distribution, (3) lightning protection, and (4) development flight instrumentation as appropriate.

Operational instrumentation includes those external tank intruments required to monitor and control tank-related functions from the start of propellant loading through tank separation. Each instrument is supposed to be individually hardwired through the tank electrical distribution cable assemblies to the ET/Orbiter umbilical connectors. Amperage limiting protection is provided by the Orbiter for those circuits penetrating the ET tanks to preclude the generation of ignition sources. Since this instrumentation consists only of sensors and cabling from them to the interface, no circuit grounds are made to the tank structure. All sensor leads are individually returned to the Orbiter for single point grounding. Cable shields are, however, grounded to tank structure to satisfy electromagnetic compatibility requirements.

Development flight instrumentation is, by definition, non-critical for external tank operation and will be installed on the main propulsion test article at NSTL and on the first six external tanks. The principal requirement from a safety standpoint is that this instrumentation shall not cause the failure of any critical external tank function. The general design and construction of the development instrumentation is the same as previously described for the operational instrumentation. Electrical power for the instrumentation assemblies is supplied through the Orbiter umbilical interface. There are two operational instrumentation cable harnesses inside the LOX and liquid hydrogen tanks. The cables are made of teflon (FEP) insulated wire, and the sensors are attached with fixed splices, insulated and sealed with heat-shrinkable TFE teflon tubing and meltable FEP teflon. Each cable is routed through a separate cryogenic feed-through connector mounted in the noseplate of the LOX tank and the forward dome of the liquid hydrogen tank. The wire bundles inside the tanks are spot tied with lacing tape and supported by corrosion resistant steel bands with teflon cushions. The use of teflon (FEP) insulated wire in contact with LOX has been identified as a potential hazard since it includes both a fuel (teflon) and a potential ignition source (electrical energy) interfacing with LOX.

The philosophy expressed in NASA's NHB 8060.1A, "Flammability, Odor, and Offgassing Requirements and Test Procedures for Materials in Environments That Support Combustion," is that the design of LOX systems should preclude any ignition sources interfacing with the media. If this goal cannot be met, any material used in the proximity of a source of electrical energy shall be evaluated in the proposed configuration. Evaluation should be made using the worst-case electrical and environmental conditions and by applying the techniques of NHB 8060.1A, Test No. 4, "Electrical Wire Insulation and Accessory Flammability Test." Results of the Apollo 13 incident and subsequent testing have shown that teflon will not pass such a test in a cryogenic high pressure oxygen environment. See Figure 33 . MSFC has stated that Saturn Launch Vehicle test experience with teflon (TFE) coated wire shows that: (1) teflon coated wire insulation cannot be ignited under LOX by any electrical over-load, (2) teflon coated wire insulation can be ignited in gaseous oxygen by approximately 800% electrical overload and will propagate, and (3) in the unlikely event of ignition under operational conditions, fire will not propagate through the feed-through-connector at the tank wall. Shuttle sensors are similar to those used on the Saturn second stage, S-II. Analysis and testing (similar to that which will be accomplished for Shuttle) were conducted subsequent to the Apollo 13 incident for the S-II sensors and demonstrated that no safety problem existed. It was stated that the temperature on the cable will be sufficiently below the sublimation point of teflon to maintain a safe condition in the cabling. The Panel pointed out that while the size of the wires was small and the potential of applying in excess of the 800% overload appeared minimal there still could be some chance of a problem, and suggested further consideration.

Current Status

MSFC will conduct worst-case current overload testing and analysis in the LOX environment using actual ET hardware and all circuit protection devices (in their worst-case credible consequences of their failures). Testing would include sensor shorts, opens, normal operation and electronic failure modes. This issue will be considered resolved if the above testing is successful. It was also suggested by the Panel that all other similar non-metallic materials' applications be reviewed and appropriate disposition made.

The External Tank design incorporates features to protect the structure and subsystems from the direct and indirect effects of triggered atmospheric discharges during transportation, prelaunch, launch and flight operations. Methods employed to provide lightning protection are intended to assure that low resistance paths are provided on the External Tank surfaces to distribute lightning currents through the structure and to guide currents around or over nonmetallic areas. At this time lightning protection on the nose cap consists of a short nose rod and conductive aluminium strips cemented onto the vehicle and electrically bonded to the structure. The LOX hydrogen and inter-tanks incorporate thin aluminum strips, adhesive-bonded to the external insulation surface and electrically bonded to the LOX tank skin. Further protection measures include the use of twisted wires on all internal circuits and twisted shielded cables in exterior cable tunnels.

The only significant problem noted by MSFC was the possibility that the diverter strips could debond or melt in flight and the resultant debris could possibly damage the Orbiter in some manner. This problem is currently under study to determine alternate designs and to further understand the impact of strips melting or debonding.

3.3.1.4 Separation and Disposition

The External Tank interfaces with the Orbiter and the Solid Rocket Boosters. In the mission events time-line, the Solid Rocket Boosters are separated from the External Tank/Orbiter combination and then the External Tank is separated from the Orbiter. The ET/SRB attach configuration is shown in Figure 34, and the aft and forward attach configurations between the External Tank and the Orbiter are shown in Figures 35 and 36 . The separation hardware in both the Orbiter and Solid Rocket Booster case are designed by their respective contractors (Rockwell International and Thiokol) and not by the tank contractor since the External Tank portions of separation interfaces are passive. Martin-Marietta Corporation does support the Rockwell International and MSFC (SRB) efforts in defining, designing and testing the separation hardware. Aspects of the ET/Orbiter separation have been discussed under the Orbiter Section 3.1 and the same will apply to the Solid Rocket Booster Section 3.4. Only those Orbiter and SRB actions that can affect the External Tank's ability to separate safely and be disposed of during its return to earth are discussed here.

The Solid Rocket Booster (SRB) separation from the External Tank (ET) follows this sequence: (1) Orbiter receives separation cue from the Solid Rocket Booster, (2) Orbiter arms' separation system pyrotechnic initiator controllers on both of the SRB's 0.8 seconds after the Orbiter cue is given, (3) Orbiter issues fire commands to separation system "A" on both SRB's simultaneously 2.5 seconds after the Orbiter cue, and (4) Orbiter issues fire commands to separation system "B" on both SRB's simultaneously 40 milliseconds after the system "A" fire commands.

Actions to be taken if for some reason this separation does not take place are to be examined further by the Panel. All the prime contractors and the NASA Centers are involved since this is an interface problem.

The External Tank separation from the Orbiter follows this sequence: (1) forward Orbiter reaction control system deployment, (2) fluid and electrical umbilical separation, (3) forward and aft structural attachment release, and (4) Orbiter maneuver away from the External Tank. Sequencing of all separation operations and commands are initiated and controlled by the Orbiter. As a result of new loads analyses for the ascent portion of the mission, the External Tank/Orbiter aft attach loads have increased, requiring hardware modifications which do not appear to unduly affect the separation events mentioned above. There are some safety concerns that result from the separation process which have been discussed with the Panel: (1) LH_2 and LOX trapped between the feed-line closure valves and released as the External Tank and Orbiter separate pose a potential fire/explosion hazard and, (2) External Tank recontact with the Orbiter vehicle primarily due to Orbiter hardware problems.

External Tank entry and disposal after release from the Orbiter has been of great interest to the Panel. Ground rules, constraints, and guidelines applicable to the External Tank disposal problem have been stated as:

(a) No External Tank impact below 60° South parallel, based on State Department international agreement.

(b) External Tank impact locations shall be in ocean areas with minimum ship traffic densities.

(c) External Tank impact locations shall be no closer than 200 nautical miles from land masses.

(d) External Tank impact location and dispersions are predictable.

(e) External Tank rupture for nominal missions shall not occur above 240,000 feet altitude.

(f) External tank distruct from any cause shall not occur within four (4) nautical miles of the Orbiter.

On normal missions the External Tank separates from the Orbiter at almost orbital velocity. The impact site is therefore sensitive to variations in the tank velocity and other conditions at separation. The question then is whether the selected design can ensure that the tank or the debris will always land in an acceptable ocean area. Aborts and catastrophic situations during launch and ascent also must be considered, and the added hazard of having large quantities of propellant and oxidizer under such situations must be taken into account.

A major consideration in the proper disposal of the tank is the point in the ascent at which time the Orbiter main engines are cutoff. The definition of the MECO (Main Engine Cut Off) is currently baselined as occurring at an altitude of 60 n. mi. for nominal mission and at 55 n. mi. for an "abort-once-around" mission. Based on these altitudes, the MECO conditions for a launch from KSC are as follows:

(a) For a nominal mission, the altitude of 60 n. mi. with a velocity of 25,383 feet per second and an angle of attack of 0.5 degrees.

(b) For an abort mission (AOA), with an altitude of 55 n. mi. with a velocity of 25,317 feet per second and an angle of attack of 0.75 degrees.

(c) For the return-to-the-landing-site (RTLS) abort mode, the MECO target is at 230,000 feet (37.8 n. mi.) with a velocity on the order of 6,500 FPS.

These MECO conditions for a launch from KSC are valid for a wide variety of launch inclinations and payload weights. Figure 37 is typical of the tank disposal landing footprint for nominal and AOA conditions.

There are two major challenges associated with the safe reentry of the External Tank. The first is the premature breakup due to LOX and hydrogen tank ruptures as well as determination of actual breakup altitude and uncertainty of the dispersion of the resultant debris. The second is the inability to assure tank impact predictability without the use of system that causes the tank to tumble. The tumbling condition must be achieved before the tank has any chance of "skipping" due to aerodynamic lift, as well as having a tumble rate that prevents the occurrence of the "frisbee" effect, which occurs at too high a tumble rate. Typical effects of three different nominal entry conditions are shown in Figure 38 . These assume a tumble rate of 30 degrees per second maximum and $\pm$ 1.3 degrees per second as minimums. The frisbee effect shown in Figure 39 becomes noticeable at tumble rates in excess of 30 degrees per second. Premature tumbling might also result in contact of the External Tank and the Orbiter. As a result of current studies, the following two ground rules have been established for an acceptable tumble system: (1) no tumble action to be initiated prior to 60 seconds after separation from the Orbiter, and (2) acceptable tumble rates are between 10 and 50 degrees per second. Martin Marietta Corporation currently is conducting studies to refine and define an "optimum" system to satisfy the ground rules noted above. The systems being considered are:

(a) Blow down, using LOX vent valves

(b) Solid rocket thrusters

(c) LOX and hydrogen tank "blow holes."

3.3.1.5 <u>Thermal Protection Subsystem</u>

In November 1974 the Thermal Protection Subsystem baseline was changed due to a significant increase in expected thermal heating environment and to a requirement to minimize ice formation and its impact on the Orbiter. This new baseline data affected the insulation material used on the three major sections of the tank: LOX tank including the nose cone, the inter-tank, and the hydrogen tank.

Current design thermal inputs to the External Tank segments based on analyses through December 1974 are:

(a) For the LOX tank forward ogive section the induced thermal environment can be as high as 10.5 btu/ft^2-sec, but new hypersonic wind tunnel data indicates a value that could be as high as 16 btu/ft^2-sec. The LOX tank, inter-tank and hydrogen tank thus are considered to be subject to heating values in excess of that normally acceptable for the proposed new insulation material (Upjohn CPR-421 spray-on foam insulation (SOFI). The CPR-421 is considered appropriate for heating values up to about 6 btu/ft^2-sec but are unacceptable at valued around 10-11 btu/ft^2-sec. The material used on structure subjected to very high heat rates is an ablator material called SLA-561 with a silicone sealant coat. These areas include the Orbiter aft attach strut, forward attach strut, liquid hydrogen feedline and crossbeam, and the LOX tank conduit.

In addition to preventing ice formation and heat input to cryo fluids, one of the major reasons for the insulation is to preclude the air liquification because liquid air is high in oxygen content when boiling off, and compatibility problems exist when it contacts hydrocarbon materials.

NASA and the prime contractor are currently conducting studies and tests to establish an insulation configuration that will satisfy known induced and natural environments with a capability for future possible heating rate increases. They feel that neither trajectory shaping or external tank configuration changes are practical methods of alleviating this problem.

3.3.1.6 Ground Support and Logistics

The mode of transportation for the External Tanks to the launch site has been settled. Barges will be used in a manner similar to that for the Saturn launch vehicle stage movement (S-IC and the S-II). The use of any carrier aircraft has been ruled out at this time because of the modifications required, cost and safety implications.

To assure propellant and oxidizer cleanliness, the following requirements have been levied on the External Tank system:

(a) The LOX and hydrogen tanks will be cleaned per MSFC - Spec - 164A, with no particle larger than 1000 microns.

(b) At the exit of each tank, propellant screens will be installed. For the hydrogen tank this will be a 400 micron "glass bead rated" screen, and for the LOX tank an 800 micron "glass bead rated" screen.

(c) All lines and components downstream of the filters shall be cleaned to a maximum particle size of 400 microns for the liquid hydrogen and 800 microns for the LOX.

It was noted that the External Tank design common fill and delivery lines insure that any contamination introduced into the system during propellant loading will be delivered to the main engines. Therefore, the ground systems and the Orbiter lines have to be cleaned to at least the same levels as the External Tank lines which interface with the Orbiter.

3.4 Solid Rocket Booster

Prior to liftoff the Orbiter Main Engines are ignited and brought to full thrust and both Solid Rocket Motors are armed and ignited from simultaneous ignition commands. At approximately 150,000 foot altitude, the thrust of both Solid Rocket Motors will have decayed to less than 25% of nominal. At this time separation of both Solid Rocket Boosters is initiated and the Orbiter and External Tank continue toward orbit. Upon successful separation of the Solid Rocket Boosters, a sequence is initiated for individual recovery of the two booster units. Parachutes are deployed along the trajectory of each unit to provide for soft impact within a predefined recovery zone. Each booster is to be floated by entrapped air until the arrival of a recovery ship or ships. The flight time, launch to splashdown, takes about 7 minutes and 15 seconds.

The Solid Rocket Booster element of the Space Shuttle system is made up of seven subsystems: (1) the solid rocket motor, (2) the thrust vector controls, (3) separation subsystem containing mechanical and ordnance equipment, (4) the recovery subsystem containing mechanical and parachute equipment, (5) avionics, (6) structure, and (8) a destruct or range safety subsystem.

The Thiokol Corporation in Wasatch, Utah was selected as the Solid Rocket Motor contractor. They have completed the design of most of the tooling for the fabrication of the motor cases and procurement is underway. The contractural awards for the structures, separation motors, recovery system, thrust vector control, and avionics had not been completed at the time of the Panel's review. However, since the Solid Rocket Booster Preliminary Design Review was completed in November 1974, the Panel was able to review the detailed design of the booster components. As mentioned in an earlier section on management, the overall integration of the booster is being performed by the Marshall Space Flight Center in Alabama. NASA plans to select a booster assembly contractor in fiscal year 1977.

3.4.1 Solid Rocket Motor

System Design

The solid rocket motor includes the case, propellant, igniter and nozzle as shown in Figure 40. Flexibility in fabrication and ease of transportation and handling are made possible by a segmented case design. The propellant grain is shaped to reduce thrust approximately one-third some 55 seconds after liftoff to prevent overstressing the vehicle during the period of maximum dynamic pressure. The grain is of conventional design, with a star-shaped perforation in the forward casting segment and a truncated cone perforation in each of the segments and the aft closure. The contoured nozzle expansion ratio is 7.16:1. The rocket motor case is made up of ten separate segments with specific joints to meet the structural requirements and weight needs as shown in Figure 41. The following is a performance summary of the rocket motors under nominal conditions at $60^{o}F$.

(a) Vacuum delivered impulse, lb-sec 290.6×10^{6} (T=1 sec.)

(b) Burn Time, seconds 122

(c) Propellant burning rate, in/sec 0.411 (at 1000 psi)

(d) Specific Impulse, average, lb-sec 262.2×10^{6}

The Solid Rocket Motor ignition hardware consists of an igniter and dual redundant standard man-rated initiators. These initiators are separated by an independent electrically dual redundant (2 motors and 1 shaft) electro-mechanical safe and arm device. Each initiator is fired by an independent Pyrotechnic Initiator Controller (PIC) upon command. The safe and arm device is maintained in the safe position by a mechanical safety pin until a given point in the countdown at which time it is removed. The device remains in the safe position until the arm-command is given immediately prior to the motor ignition.

The items associated with weight and weight control are:

(a) Motor Mass Fraction 0.884

(b) Total Solid Rocket Motor, lbs. 1,254,210

(c) Solid Rocket Motor, lbs. 1,227,250

Current Status

There have been studies on alternate propellants to minimize HCl release above 65,000 feet (ozone layer) during the ascent portion of the mission. To date the studies indicate that it is technically feasible to minimize (less than 3% by weight) or eliminate the release of HCl above 65,000 feet. However, there would be a probably payload loss of 2,000 to 7,000 pounds. These studies will continue as one of NASA's efforts to reduce the atmospheric impact from the Space Shuttle operations.

NASA has noted that the Solid Rocket Motor and booster components fabrication requirements are considered to be the current state-of-the-art technology which has been demonstrated in systems such as the Titan III rocket now in use.

Thrust mismatch of the two rocket motors is of great concern to the designers and the operation of the Shuttle system. As a result of this concern, NASA and its contractors, continue to pay a great deal of attention to having both the rocket motors ignite and essentially tail-off simultaneously and an acceptable thrust mismatch during normal ascent. The reproducibility limits, based on the latest analysis, are shown in Figure 42. Thus there will most likely be a need to match pairs of rockets. The specification requires that there not be a mismatch greater than 710,000 pounds during the tail-off thrust period at around 115 seconds after ignition.

The POGO phenomenon is not expected to manifest itself in the burning characteristics of the rocket motor. However, the potential for this motor to contribute to POGO will be explored fully by the program offices as a part of the overall POGO effort.

3.4.2 Thrust Vector Control

System Design

The Thrust Vector Control subsystem controls the angle of the nozzle of the rocket motor, in order to obtain the proper flight trajectory. Each Solid Rocket Booster contains a Thrust Vector Control assembly consisting of redundant hydraulic power units and two actuators. If one of the hydraulic power units fails, a valve in the actuators isolates the failed unit and this prevents any loss of thrust vector capability. The servovalves for each actuator are hardwired across the SRB/ET interface and accept steering commands from the Orbiter guidance and control system to provide motor deflection. The basic requirements for this control system are:

(a)	Torque, inch-pounds	4,200,000
(b)	Rate, degrees per second	5
(c)	Acceleration, radians per sec^2	2
(d)	Gimbal Angle, degrees	5
(e)	Redundancy	Fail safe as minimum.

Current Status

The current design is a fail operational/fail safe design. The Thrust

Vector Control has a maximum gimbal capability of 7.1 degrees and provides torques in excess of those required for known loadings. Since the loads effort is a continuing activity the loads may change upward but appear not to be a major problem at this time.

3.4.3 Separation Subsystem

System Design

The Solid Rocket Booster separation subsystem consists of the forward and aft separation motor assemblies, the forward attachment unit and the aft attachment and umbilical pull-away unit, Figure 43.

The separation sequence for the booster is:

(a) Orbiter receives separation cue from both boosters,

(b) Orbiter arms two sepration system pyrotechnic initiator controls on both the A and B units in both boosters 0.8 seconds after the cue is given to the Orbiter,

(c) The Orbiter issues fire commands simultaneously to the "A" unit on both the boosters at 2.5 seconds after the cue,

(d) Orbiter issues the fire command simultaneously to "B" unit separation assemblies on both boosters some 40 milliseconds after system "A" has been given the fire command.

The cue received by the Orbiter is in the form of a pressure signal when the Solid Rocket Motor chamber pressure has reached 50 ± 15 psia on any two pressure sensors used for this purpose. The separation system avionics is shown in Figure 44.

Current Status

The forward and aft separation motor asseblies each consist of four separation motors and ignition ordnance which are fired to impart side thrust to the expended booster. There has been a recent change in the motors to reduce, if not eliminate, the impingement of the motor plumes on the Orbiter Thermal Protection Subsystem. These changes are noted here:

	Before	Current
Thrust Level, lbs.	12,000	20,000
Burn Time, seconds	2	0.75
Propellant Restrictions	none	max. metal or stabilizing additives - 2% burn rate additives - 1%
Igniter Case Material	glass phenolic	non-debris generating
Igniter Propellant	no restriction	same restrictions as main propellant
Thrust Tail-Off Rate	no restriction	Tail-off to 50% chamber pressure limited to 100 milliseconds
Motor Location	SRB forward back of frustum and aft skirt	Nose frustum and aft skirt

The forward attachment unit consists of an SRB fitting, called a thrust post, supported by the SRB forward attachment structure which mates with an External Tank fitting. This forward attachment provides longitudinal SRB/ET restraint and transmits thrust from the SRB to the ET/Orbiter. The SRB and ET mating surfaces are held together by a double-ended separation bolt which is internally redundant for the separation function. A standard manned spacecraft initiator pressure cartridge is mounted on both ends of the double-ended separation bolt. At separation, both of the separation cartridges are fired and the resultant pressure buildup drives an internal piston at each end of the separation bolt toward the separation plane to effect bolt fracture. Operation of either piston will fracture the bolt.

The aft SRB/ET attachments include a lower, upper, and diagonal strut assembly which provide lateral and rotational restraint between the SRB/ET. Each strut assembly consists of a SRB and ET fitting held together by a double ended separation bolt similar in design and operation to the forward attachment separation bolt. The "pull-away" connectors used at each SRB/ET interface carry the electrical circuits as follows:

(a) Forward Attachment	1
(b) Aft Strut (Diagonal)	1
(c) Aft Strut (Upper)	5
(d) Aft Strut (Lower)	3

As a result of the latest Shuttle system loads analysis, December 1974, there is an effort underway to redesign the forward thrust fittings and aft attachment struts. This will result, most likely, in some weight increases. There is no expected change to the basic concept of the separation assembly described here.

3.4.4 Recovery Subsystem

System Design

The booster recovery subsystem provides the necessary hardware to control the descent (velocity and attitude) after separation from the External Tank. The recovery subsystem includes those items used to separate, deploy, disconnect, control attitude, float, and provide for location of the expended booster. Figure 45 shows the booster recovery (separation to splashdown) events and associated parameters of performance at each stage. The booster recovery main chutes, drogue and frustum, and booster itself are buoyant. The recovery system is redundant except for the beacon and flashing light.

Briefly the sequence of events is as follows. A command is sent from the Orbiter to the Solid Rocket Booster just before separation to apply battery power to the recovery logic network and at the same time to arm the nose cap thruster for deploying the drogue, the frustum ring detonator for main deploy, and the main chutes disconnect. Two barometric switches are set to close at high altitude (below 19,000 feet) and at low altitude (below 10,000 feet). At high altitude the nose cap thruster fires, pushes the nose cap away from the booster, and deploys the drogue chutes. At low altitude the frustum ring detonator fires, the drogue chute pulls the frustum away from the booster, and deploys the main chutes. After a time delay the nozzle extension is jettisoned and the impact switches are armed. A third barometric switch will close at a very low altitude to turn on the impact recorder just prior to water impact. At impact the impact switches close and after a time delay the main chutes are disconnected and the beacon and light are turned on. The nose section of the booster, containing the majority of the recovery hardware, is shown in Figure 46.

The maximum vertical velocity for the booster at water impact has been set at 100 feet per second.

Current Status

The Panel's major interest was directed toward questions concerning the inherent safety of a reusable Solid Rocket Booster. The solid rocket case, the parachutes and the hardware for the separation of the booster from External Tank were of the greatest interest. In this section the parachutes and separation hardware are discussed, while the motor case is discussed under the "Structures" paragraph which follows. The separation hardware includes the forward and after separation motor assemblies, forward and aft strut attachment units and the umbilical pull-away connector units. The separation motors are burned out after use and require replacement, as does the ignition ordnance. As noted in the reviews conducted at MSFC the electrical connectors and wiring are the major items requiring retest and rehabilitation for reuse in the booster. The attachment struts and fittings are a part of the structure and are covered in that section. The replacement of used pyrotechnic cartridges and retest of the connectors and wiring is the important task.

Refurbishment of the parachutes (drogue and main) is new to NASA experience in that NASA's current approach is to not reuse space recovery parachutes. However, there is a great deal of DOD experience available with regard to reusing parachutes, e.g., aircraft braking chutes, cargo parachutes and personnel parachutes.

The material in Table X is indicative of the approach used in defining the ability to reuse a drogue or main chute. More specifically, the following data have been developed for commonly used materials such as nylon and dacron:

(a) Prolonged ultraviolet explosure produces strength loss of 50% within seven days.

(b) High temperatures result in severe strength loss after only 10 hours of exposure at 350^{o} F.

(c) Since these materials are hygroscopic (absorb water), they show only a slight strength loss when subjected to high humidity.

(d) Radiation other than ultraviolet is very harmful and thus chutes require shielding.

(e) Vacuum conditions do not appear to materially affect the cute properties.

3.4.5. Avionics

Systems Design

The Booster Avionics consists of the following assemblies: electrical, instrumentation, control rate gyro, recovery, range safety, and failure detection.

A significant portion of the electrical and instrumentation assemblies are included in two line replaceable units, the forward and aft integrated electronics assemblies. Both contain the logic and networks distributior, multiplexer-demultiplexer, signal conditioner and the forward two data buss couplers.

The electrical system consists of a 28 VDC battery supplying power for separation, deployment and recovery functions through the logic and network distributors. These distributors, one forward and one aft, also provide the 28 vdc power from the Orbiter to signal conditioners and associated measuring devices during the ground and flight period when the boosters still are a part of the total Space Shuttle vehicle.

The avionics associated with the recovery activities consists of the following components: (1) Altitude/impact switch assembly, (2) X-band radar transponder (beacon system), (3) X-band radar antenna (beacon system), and (4) two flashing lights.

Range safety subsystem, which is not yet defined, is to provide the destruct capability for the boosters in case of early termination of the flight. This system has been defined in the Level II "Space Shuttle Program Flight and Ground System Specification", JSC-07700 Vol. X, updated to May 1975, as "an add-on destruct system --- which does not require any action by the crew prior to initiation of an abort. The system function shall be dependent on real-time range safety down-linked paramaters and/or tracking data for the period after liftoff up to SRB/ET separation."

Current Status

Based on the material provided to the Panel, the following is the status of the range safety system:

The design concept and selection of system components are complete except for conical shaped charge to be placed in the solid rocket booster element. Currently the program is involved in an effort to fully integrate the system design from the standpoint of ground-to-flight vehicle and between the flight vehicle elements. Acceptance of basic design concept by the Air Force Eastern Test Range is still under discussion. Working interfaces have been established between all organizations affected by the range safety system design, development and utilization. Discussions between these groups, reviews and planning sessions are being established.

The failure detection setup for the booster provides the failure detection capability during boost phase of the flight. This setup had not been defined sufficiently for presentation to the Panel during its early Spring review at MSFC.

3.4.6 Structures and Reusability

The reusability aspects of the Solid Rocket Booster are so closely tied to the structural design capabilities that these two aspects of the booster program are discussed together in this report. Basically the only non-structural hardware built for reuse are the electrical and instrumentation equipment, thrust vector control assembly and such recovery items as the parachutes. The Solid Rocket Motor case and attendant structure are all considered as a part of the structural assembly.

The current baseline for reuse of the Booster components is:

Structures reuse	40 times
Solid Rocket Motor Case and Nozzle	20 times
Thrust Vector Control assembly	20 times
Electrical and Instrumentation reuses	20 times
Recovery assemblies	10 times
Batteries	1 time

Structural design features to support the booster reusability program include such things as: (1) external protective coatings, (2) weld-free solid rocket motor case, (3) water-tight compartments using welded aluminum skins, (4) bulkheads for protection of the avionics (electrical and instrumentation items in the forward portion of the booster, (5) stiffening rings along the aft quarter of the booster structure to help take the water impact loads, and (6) the use of a smooth surface for the application of thermal protection material around the aft skirt which covers the nozzle. The Solid Rocket Motor case is designed with 0.009 metal thickness beyond that required for flight loads, fracture mechanics and water impact. To allow for wear due to "grit" cleaning during refurbishment for additional refillings. The Solid Rocket Motor case joints are described in Figure 41.

Current Status

An integral part of the structural design procedure includes a "Fracture Control Plan" for the Solid Rocket Booster and motor. This plan establishes the requirements for reporting, non-destructive testing (inspection), failure documentation, traceability, service life recording, proof testing, and environmental control of all portions of the structures defined as susceptible to structural failure due to flaws and cracks. In line with this plan, materials are selected and characterized for specific Solid Rocket Booster and motor environments and fabrication processes and refurbishment requirements. One of the problems in designing the booster/motor structures is to account for fracture under other than plane-strain conditions and to provide a practical means for predicting life under the complex time-stress histories occurring during pad operations, boost phase of the mission and recovery of the booster.

Other questions open at the time of the Panel's review deal mainly with the structural aspects of the booster element.

The specified reuse requirements and the designs to meet them are dependent upon the definitions of service life, safety factors and their derivation. Some thoughts relative to reuse which are pertinent to assuring a safe and cost effective booster are: (1) what will wear out or be rendered unserviceable after the specified number of reuses that will not wear out or be unserviceable after a greater or lesser number of reuses or cycles, and (2) what would be designed differently if the design were required to be made to meet a higher number of reuses.

Noise (vibroacoustic effects) generated by the Solid Rocket Motors and the Main Engines on the pad and soon after liftoff may impose severe requirements. The determination of these effects and the design constraints are still under study.

The booster design and expected attrition rates are highly dependent upon the extent of damage due to water impact loads. These stresses are dependent upon booster velocity, angle of impact, temperature of the structural material and surface conditions such as winds and sea state. Computer analysis programs have been developed to analyze (1) initial impact, (2) cavity formation and collapse of the water volume, (3) maximum booster penetration into the water and at the same time water penetration into the throat of the rocket motor, and (4) rebound and slapdown on the water surface.

There are also those events associated with the time when the booster is in the water and the ships and men begin to retrieve the boosters from the water. The degree that these operations impact the design of the booster has not been fully explored by the Panel at this time.

From the time a solid propellant rocket grain is cast until it has burned away in the performance of its mission, it is subjected to an array of stress-inducing environments including gravity, propellant curing loads, handling shocks and vibrations, and the pressurizations and accelerations that accompany ignition, launch, and flight. The possibility of safety related problems resulting from any one or combination of these environments will be examined in later reviews by the Panel.

Lightning protection requirements for the Solid Rocket Booster are similar to those for the Orbiter. Equipment requiring protection include pyrotechnics, thrust vector control sensors and switching circuits, all exposed electrical cables, and the integrated electronic assembly (data buss couplers, signal conditioner, multiplexer-deplexer, logic and network distributor.

Current lightning protection design measures include the following: (1) single point ground on power circuits, (2) use of twisted wire pairs, (3) delays of 2 ± 1 millisecond in the many switching functions, and (4) use of metallic cable tunnel to protect cable runs forward and aft and the use of multi-grounded overall shields on all ordnance cabling.

Electrical interfaces between the Orbiter, External Tank, and the Solid Rocket Booster do not fully satisfy the lightning design criteria. Interface design is being studied at this time to obtain a reasonable solution to this problem. On the SRB program several tests are being planned to validate the lightning protection arrangement: (1) cable core test on SRB equipment as required, (2) full scale lightning test on the External Tank/Booster attach struts with ordnance installed, and (3) cable tunnel attenuation tests.

3.5 Launch and Landing Element

The launch and landing aspects of the Shuttle program are considered an element in the same manner as the Orbiter element, External Tank element, SSME element and the Solid Rocket Booster element. The Launch and Landing element is under the jurisdiction of NASA's Kennedy Space Center. There are other prime and secondary sites, but the discussion here centers on the requirements, design, development, validation, launch, and landing preparation plans at KSC.

The design and operation of the launch/landing site is as much a key to achieving a low cost Shuttle system with rapid turnaround after a flight as any other element of the program. KSC's past roles on the manned and unmanned programs, in which facilities and know-how have been developed for the receipt inspection assembly, checkout and launch, plays a large part in their ability to meet their current and projected role in the Space Shuttle program. More specifically the Launch and Landing Project conducted at KSC covers the following activities:

(a) Shuttle vehicle element receiving (including all that goes with such activities, e.g., inspections), assembly of the Shuttle vehicle including buildup from the elements to the total ready-to-fly vehicle, checkout and launch.

(b) Recovery/retrieval operations for the Orbiter and Solid Rocket Booster.

(c) Ground Operation taking into account the necessary sustaining engineering, logistics, maintainability and the turnaround operations.

(d) Facilities and Ground Support Equipment, such as the Runway, Orbiter Processing Facility, Launch Control Center, Flight Test Control. A major innovation will be the Launch Processing System to satisfy the requirements for an automated launch checkout.

With regard to payloads, KSC will prepare and install the Spacelab delivered by the European consortium, the automated payloads, the Air Force Interim Upper Stage Vehicle and the TUG vehicle and all other payloads.

The KSC interface with the NASA Flight Research Center at Edwards, California, includes a major role in the Approach and Landing Test program.

At Vandenberg Air Force Base, California, KSC will assist the Air Force in planning and will provide expert help in the area of turnaround operations, facilities, launch support equipment and payloads operations.

Recognizing that the Panel has not had the opportunity to examine the Shuttle program from the KSC viewpoint in any detail, the focus was on a small number of areas of particular interest to the Panel at this time: Solid Rocket Booster retrieval, landing facilities and landing controls, Orbiter Thermal Protection Subsystems maintenance, turnaround operations, and Launch Processing Subsystem. The Panel did, however, receive an orientation briefing on the total KSC role, responsibilities and plans to carry them out.

3.5.1 Solid Rocket Booster Retrieval

Systems Design

So we have noted, the Marshall Space Flight Center has responsibility for the development of the Solid Rocket Booster, including the intact reentry of the booster into the ocean. KSC, however, is responsible for developing the retrieval system for returning the boosters to dry land for refurbishment and preparation for reuse.

Retrieval of the boosters, parachutes, and other recoverable objects will be accomplished using surface vessels. The retrieval vessels will tow the boosters to KSC; other objects recovered will be brought onboard the vessels themselves. Shuttle developmental launches will, of course, be used to test and refine vehicle recovery/ retrieval systems. The boosters are expected to impact at a point some 130 to 150 nautical miles downrange in an impact footprint defined as a 10 x 33 nautical mile ellipse. Once the boosters are located and the vessels are near enough, divers are sent to plug the nozzle.

Then the booster is dewatered and it attains what is called a "log" mode. Parachutes are coiled on reels and the nose cone frustum is lifted on board the vessel and the boosters towed home.

Current Status

The retrieval system definition is in its early stages and will be examined in more detail as the necessary design, interface and operational details are worked out. Among the questions yet to be answered are the number of tracks to have on the SRB impact recorder, and the baseline for the "station set" used in the SRB retrieval and disassembly

3.5.2 Landing Facilities and Landing Control

Systems Design

These facilities and controls can be divided into the following specific items: (1) Primary landing sites, KSC and VAFB used for test and operational flights, (2) secondary landing sites with particular emphasis on Flight Research Center/Edwards AFB used for the Approach and Landing Test program using the carrier aircraft, and (3) the Mission Control Center at Johnson Space Center, Houston, Texas.

The Orbiter Landing Facility at KSC is located approximately 1.5 miles north and west of the Vehicle Assembly Building (VAB) and extends 15,000 feet to the northwest. It is composed of the following:

(a) **Airfield** pavements of 15,000 ft x 300 ft with 1000 ft. overruns on each end, a two-way that is 10,600 ft. long and 50 ft. wide leading to the Orbiter Processing Facility, and a parking apron just off the main runway and coincidental with the two-way 490 ft. x 550 ft.

(b) Airfield lighting along the standard approach, runway touchdown and centerline, and the runway edge.

(c) A landing aids control building at the southeastern end of the runway containing hardware for flight and ground control including the Orbiter landing instrumentation system with S-band/UHF communications, TACAN, Microwave Scanning Beam Landing System (MSBLS) and related installations.

Current Status

The current status of the Orbiter landing facility at KSC is as follows:

(a) Construction awards have been made for Phase I and II and the requirements for Phase III are in the planning stage.

(b) Phase I construction on the runway, two-way, parking apron, arifield lighting, electrical power and water mains is to be completed in August 1976.

(c) Phase II construction on the landing aids control building, instrumentation facility, utilities support and cabling systems is expected to be completed in September 1976.

(d) Phase III, TACAN, Communication systems (MSBLS, Comsec, etc.), propellant and gases systems, high energy aim point, cinetheodolite system, Orbiter mating device, and other landing support equipment are all in planning and requirements review stages.

(e) Test planning includes the utilization of the Shuttle Training Aircraft to validate the ground landing aids and control systems.

(f) Significant issues at the time of the Panel review (March 1975) were: (1) Additional facilities required for cinetheodolites and the high energy aim point, (2) Runway grooving spacing which is to be between 1" and 2", and (3) While the microwave Scanning Beam Landing System has been selected to support the Orbiter landing, its location at the end of the runway is under discussion (i.e., on the centerline or off the center line).

The current program specifications call for the Johnson Space Center's Mission Control Center to retain control of the Shuttle elements (vehicle and, particularly, the Orbiter) throughout the mission including entry, landing and rollout to a stop on the runway. There is still some discussion as to the best location for control of the Orbiter during the Terminal Area Energy Management portion of the mission (from about 70,000 ft. altitude to roll-out on the runway). The Panel will follow this question until its resolution to assure that crew safety and successful vehicle return receive appropriate attention.

During the last half of 1974 the question "Is there a need for an overrun barrier at KSC, Edwards AFB or Western Test Range?" was asked in earnest. As presented to the Panel, a thorough analyses was made to determine the need for such barriers. The factors influencing the requirements were: (1) touchdown point on the runway, (2) velocity of the Orbiter at touchdown, (3) Orbiter characteristics, e.g., drag, stability, etc., (4) coefficient of friction (wheels to runway), and (5) the brake system capabilities. The "worst cast" roll-out performance used in the analysis assumed: hot day, wet runway (ungrooved), landing weight of 230,000 pounds, maximum landing velocity and landing long, and with a single tire blow-out at landing.

Analysis indicates that the Orbiter would require a total runway of 15,530 ft. Since the runway is 16,000 ft., the runway barrier requirements were deleted.

3.5.3 <u>Ground Turnaround</u>

<u>Systems Design</u>

Turnaround operations include:

(a) Landing (a portion of which is covered in previous paragraph)

(b) Orbiter safing, maintenance and checkout (this includes the Thermal Protection Subsystem maintenance)

(c) External Tank and Solid Rocket Booster preparation

(d) Shuttle Vehicle Assembly

(e) Pre-launch checkout and launch

Current Status

During the early Panel reviews it was evident that the 160 hour requirement is a major design driver. Therefore, the Panel is interested in assuring that this requirement will not adversely affect ground or crew safety. KSC is trying to meet this turnaround requirement and assume a safe vehicle through the use of the computerized Launch Processing System (LPS). In addition, ground operations are being designed to use proved techniques and optimize the level of inspection while reducing subsystem level checkout time as performance confidence is achieved. Evolution of the 160 hour turnaround is shown in Figure 47.

Two of many management aids in respect to turnaround are mentioned here because of their significance. The Shuttle Turnaround Analysis Group (STAG) chaired by KSC, has been established as the Government-contractor team responsible for Shuttle System integrated program turnaround allocations and assessments. The system integration contractor (Rockwell International, Space Division) assists KSC in the evaluation of the element-level reports and analysis reports.

The Shuttle Turnaround Analysis Report (STAR) is prepared by KSC and is submitted to the JSC Space Shuttle Program Office to depict the current status of the operational turnaround functions.

KSC considers the following four basic areas in developing the operational team concept: (1) Definition of functions in detail, (2) degree of autonomy to be provided, (3) depth of management oversight required, and (4) the varied personnel skills necessary to achieve the turnaround objectives.

The handling of the Orbiter TPS is one of the more difficult assignments during the turnaround period. Inspection and refurbishment will require constant attention to assure the adequacy of the TPS for the next mission. The TPS tiles are fragile in comparison to most other items on the Orbiter and must be handled accordingly. A major element of the post landing operations at KSC is the performance of preliminary checks of the TPS surface to determine in a gross manner the quantity of damage sustained during the mission and particularly during entry and landing. Once the vehicle has beem taken to the Orbiter Processing Facility a detailed examination of the tiled surface is made. The methods by which this will be done have not been fully defined, but will be examined in the future reviews.

The Launch Processing System makes use of modular, or building block, structure which will allow the hardware and software to be configured to accommodate differing requirements in the checkout, maintenance, and launch functions. In the launch support configuration, test engineers, manning LPS consoles in the Launch Control Center, perform testing and prepare for launch. The LPS in the maintenance and checkout configurations has LPS consoles located in areas such as the Orbiter Processing Facility, Vehicle Assembly Building High Bays and Hypergolic Maintenance Facility. The following points were made to the Panel regarding the requirements for the LPS in the checkout configuration:

(a) Tese automation - faster, repeatable, better discipline, realtime test results;

(b) Standardization of hardware and software - computers, displays, data transmission, hardware interfaces documentation, training and maintenance;

(c) General purpose/high density consoles - fewer operations per system, more burden on the machines and th multiple use of equipment;

(d) Test engineer oriented language to eliminate middleman-programmer, make engineer responsible for the entire system;

(e) Rapid access to engineering data and work control system.

Open issues at the time of the Panel's review included the continuing review of requirements for the system, preliminary design review planning and development flight instrumentation data processing and LPS requirements for the Payloads.

The station set is defined as an accumulation of units of GSE required to support a specific activity or phase of vehicle assembly, test launch or pre-launch. There are three types of GSE units or models in order to affect the greatest degree of cost effectiveness. These are:

Type I - Critical to 160-hr timeline or final system verification or hazardous operations.

Type II - Functional interface with the vehicle.

Type III - No vehicle interface or interfaces with vehicle but requires minimum design control.

The Panel asked about the requirements with respect to reliability and safety. The following requirements and philosophy apply: (a) The Launch essential and safety critical ground support items are identified and that particular list is updated and provided to management for their understanding and control, (b) Failure Modeand Effects Analyses (FMEA's) and hazard analyses are required for all launch essential and safety critical GSE, (c) All launch essential and safety critical GSE require that for the certification program, acceptance shall consist of one or any combination of analysis, similarity or actual testing.

One of the open questions to be resolved is the timelines of documentation data from the element contractors (Orbiter, External Tank, Solid Rocket Booster, Space Shuttle Main Engine) which affects

KSC's ability to plan for and define spares and maintenance requirements and affects the facility design activity as well.

One of the challenges during turnaround will be the assembly of the total Shuttle vehicle, since Shuttle elements require very tight stacking tolerances, well designed equipment, and well trained personnel to assure proper control of stacking procedures.

Factors being considered now in the design of the mobile launcher and launch pad are:

(a) Engine exhaust rebound back up into the space vehicle creating a vibroacoustic problem, as well as thermal problems.

(b) The engine quench system (water system).

(c) The hole-sizing in the platform to accommodate the Solid Rocket Booster exhaust.

(d) The requirements for payload unbilicals.

(e) Facilities to minimize External Tank ice formation and affects of ice shedding.

(f) Orbiter Thermal Protection Subsystem tile protection.

(g) Payload handling requirements and their implementation, e.g., the payload cleanroom facility.

4.0 SAFETY, RELIABILITY, QUALITY

4.1 System Design

For our purposes reliability (probability of failure), quality (excellence in producing hardware/software), safety (freedom from injury or loss) are all a part of the so-called "Risk Management System" or "Space Shuttle Assurance Program." These are obviously interrelated activities and as such are not covered separately in this document.

The Space Shuttle risk management system is built on prior manned flight program experience and modified to meet Shuttle requirements. Safety analysis process is shown schematically in Figure 48. Each of the element contractors and each of the participating NASA Centers conduct its own safety, reliability and quality programs. In addition, the Rockwell International Space Division in Downey, California, as the system contractor, conducts an integrated safety analysis operation. The total Shuttle program requirements including reliability, safety and quality are delineated in the Level II program requirements' documents JSC 07700, Volumes I-XVIII. Compliance with these requirements is further addressed in numerous documents. For instance, the approach to reliability is addressed in Volume I, "Master Verification Plan." Volumes II through V have the requirements for the element verification plans. The element verification plans describe the way the requirements are to be met, e.g., test, analysis, and inspection. The specific plans covering reliability, quality and safety are submitted by the element contractors to the appropriate project elements in NASA for review and approval.

4.2 Major Reviews

The major risks and uncertainties determined by various assessment teams and permanent organizations are reviewed by management as a part of their review system. The Preliminary Design Review for Orbiter No. 102 and the Shuttle System Preliminary Design Review are examples of such events. Figure 49 shows that at the time of the Orbiter 102 Preliminary Design Review twenty (20) subsystem failure modes and effects analysis documents have been issued. These documents covered 947 components in terms of possible failure modes and their impact on the crew and mission.

The Safety Analysis Report indicated 200 Orbiter hazards and the corrective actions being taken. This analysis covers such situations as: (1) illness/injury/loss of personnel, (2) collision/impact/erosion, (3) fire/explosion/implosion, (4) loss of or unsafe environment, (5) crash landing/ditching, and (6) loss of flight control.

Hazard analysis is performed at the subsystem level and, in cases where Failure Mode Effect Analysis have identified critical items for the Critical Items List, the analysis is performed to a lower level of detail.

The Critical Items List contains the single failure points and criticality 3_I items identified by the FMEA. Criticality 3_I are all those items not having a potential effect on loss of life or vehicle or loss of mission. They also meet one or more of the following criteria: (1) redundant elements are not capable of checkout during normal ground turnaround, (2) loss of a redundant element is not readily detectable in flight, or (3) all redundant elements can be lost by a single credible event or cause.

4.3 Safety Analysis Process

The safety analysis process for the Shuttle program is being implemented in the following basic steps: (1) identification of safety concerns, (2) analysis of safety concerns for credibility and criticality, (3) initiation of Shuttle hazard analyses, and (4) tracking and closing out Shuttle hazard analyses. Each of these steps is described below.

4.3.1 Identification of Safety Concerns

A system safety concern is any design or operational issue that has a potential impact on personnel or hardware. The concern may be identified by any person or organization on the program and must be dispositioned. For instance, the system contractor's safety office reviews the element contractor's hazard analyses and FMEA's to determine if a possible safety problem may propagate across elements of the Shuttle from an identified hazard or failure on any one element.

The system contractor's safety office also reviews the planned operations of the Shuttle for potential safety problems. This is to be done for each mission phase. In addition there is a continuing effort by Rockwell International's Space Division engineering and other groups to identify other issues which have a safety implication.

4.3.2 Analysis and Resolution of Safety Concerns

Every safety concern identified to the system contractor's safety office will be analyzed for credibility and criticality. Credibility means that there is a real possibility that the event may happen. Criticality means that, if the concern occurs, there would be personnel injury, loss of the vehicle, or major damage to ground facilities. If the concern is both credible and critical, then action has to be taken to preclude undesirable consequences or minimize possibility of occurrence. If the concern cannot be resolved, management must review and decide upon the risk to be accepted. Experience has shown that the great majority of the safety concerns identified can be shown to be not credible or critical.

4.4 Shuttle System Safety Concerns

Safety concerns as presented to the Panel during its May inspection trip to the Space Division of Rockwell International are shown in Table XI.

The hazards resulting from fluids used throughout the Shuttle mission, with particular reference to the fire and toxicity problems, are outlined in Table XII. Only two phases of the mission would appear to be essentially clear of problems, the ascent and orbit periods. A partial resolution of this problem was to separate incompatible materials and environments by compartmentizing or sealing off of the Orbiter where practical so there were no hazardous fluids in the pressurized crew compartment. In addition to sealing off compartments, an active purge, such as dry nitrogen gas, is used to dilute the concentration of hazardous gases. Warning devices have been developed to alert the crew and ground control. Contingency procedures at launch pad and during mission will be formalized. Figure 50 depicts this approach schematically.

The Orbiter flight vent and purge system described in Section 3.1 "Orbiter Element" to minimize the hazardous gas problem is augmented by the ground hazardous gas detection system designed and developed by the KSC organization. This ground system has been defined and the remaining major development items are the sensors for the cryogenic and hypergolic portions of the system. For the cryogenic subsystem, these are mass spectrometer, electrochemical sensors, and portable hydrogen sensor. For the hypergolic subsystem these are the portable hypergolic sensor and the air oxidation chemistry analyzer hardware. The flight system operation depends upon defining what is a hazardous fluid condition. For example, dissassociation of leaked fluids must be known for detection and hazard assessment (N_2O_4 in humid atmosphere forms nitric acid) as well as autogenous ignition temperature at altitude (low pressures) for Orbiter fluids. These data will be obtained in the coming months through a series of inhouse and contract activities.

Current Status

The Panel requested that the following safety concerns be discussed during their visits to both NASA Centers and Contractors. Each of these concerns is presented below along with the current status at the time of our review.

Solid Rocket Booster Ignition Overpressure - Large over-pressures on Orbiter and External Tank structures and surfaces may be imposed by the booster exhaust shock-wave at ignition. The over-pressure wave is assumed to reflect asymmetrically from the pad flame deflector and travel up the vehicle, applying pitch plane loads. Tests are to be conducted on a Shuttle model at MSFC to acquire valid pressure distributions and intensities. Resolution has been targeted for November 1975.

Unscheduled SSME Shutdown During Boost - SSME design provides internal, automatic shutdown mechanisms to achieve safe engine shutdown when critical performance parameters are not within tolerance requirements. Investigation has shown that the remaining two engines are necessary to achieve intact abort, and that a two-engine-out condition may well result in vehicle loss. One approach being studied to resolve this concern is to have a single engine shutdown inhibit or disable the internal shutdown mechanisms for the two remaining main engines.

This inhibit capability would be accomplished by automatic electrical "lockup" of the engine control valves in their last position, and by incorporating an inhibit coil on the emergency shutdown solonoid.

Crew Rescue From Orbit - If for any reason the vehicle is unable to return to earth from orbit, no rescue capability exists during the early flight test program, but a "rescue orbiter" would be available during the operational periods. Various ideas are being explored to achieve a rescue capability during the early flight test portion of the program.

Solid Rocket Booster Thrust Mismatch - Booster thrust mismatch can occur at any time during the burning period. The periods of greatest concern are at liftoff, maximum dynamic pressure and at the end of burning period (tailoff). During liftoff, the specification for the Shuttle system calls for a maximum mismatch of 300,000 pounds. This value appears conservative based on results of Titan IIIC statistical analysis of ignition transient. Ignition transient is still being evaluated by MSFC/Thiokol/Rockwell for better definition of the time mismatch action. The impact of a mismatch at the maximum "q" condition is to add an additional load on the flight control system elements in the yaw direction. The Shuttle structure and flight control system has the capability to adequately account for such additional loads. The Booster tailoff thrust differential indicates that a 710,000 pound mismatch is controllable with normal control capability. The

710K value has been established as a requirement which occurs about 115 seconds after ignition. However, when Booster nozzle actuator or SSME engines fail the separation of the Booster from the External Tank is delayed for up to 4 seconds to reduce the mismatch thrust and provide acceptable separation conditions. The extent of the control capability that can be exerted during tailoff continues to be studied to assure adequate flight control and separation ability.

4.5 Orbiter Safety Concerns

Orbiter Structural Elements

Structural deformation may prevent emergency egress from crash landings. Orbiter 102, to be used for first orbital flights, has added overhead escape panels which are used in conjunction with ejection seats, but the panels will remain after ejection seats are removed. There is a current study to ascertain the value of using the overhead hatches on all Orbiters. The ability to compartmentize or isolate hazardous fluids is discussed in the fire/toxicity section above. There must be continuous control to assure that hardware assigned to the "structures" category does not include items similar to the Skylab meteroid shield.

Doors

The major point is that during entry all doors must be closed. If the payload doors do not close then the crew must use EVA and secure them. There are continuing studies on elimination of doors and methods of assuring their proper positioning throughout the mission.

Payload Retention

Payloads must be adequately constrained during normal or abort landings to avoid damage to the crew.

Thermal Protection System

This has been covered in detail in Section 3.

Hydraulics

Loss of flight control due to failure of single actuators which are used for elevon control was studied by Rockwell International and NASA. They accepted the risk of being involved in relying upon a single actuator.

Ejection Seats

The possibility of collision between the ejection seats following ejection is under evaluation at this time.

Orbital Maneuvering Subsystem

Large quantities of OMS propellant requires that it be managed to assure proper center of gravity conditions during nominal and abort trajectories. Orbiter aerodynamics analysis and mass properties analysis are being performed to determine allowable residual propellant quantities and the quantities to be dumped. This work is expected to continue through the next fiscal year with resolution at the end of that time.

Data Processing System (Software)

Generic software errors may not be detected in the software verification program based on prior experience in this area. A study is under way to determine the degree of degradation due to expected errors and possible work-arounds to maintain operational control.

Hydrogen Fire During RTLS Abort

During the return to landing site abort a hydrogen concentration is expected to exist in the wake of the Orbiter. The location of the exhaust, vent, and dump locations are a safety concern.

Landing/Deceleration Subsystem

The Panel has questioned the ability of the landing gear gravity deployment system to support the Orbiter Landing trajectory (altitude, time, distance). What is the basis for confidence in the reliability of the free-fall system that landing gear will be in the down and locked position? When working properly is there sufficient time to achieve the down and locked position prior to touchdown? What contingency plans are available if the landing gear system does not operate properly?

Because of the Panel's interest in this area a brief description of the gear units and doors and their operation during landing provided here for a better understanding of the above three questions. Figures 51 and 52 show the nose gear and main gear installation. The nose gear retracts forward and up in the forward fuselage, and the main gear retracts forward and up into the wing. The weight of the nose gear system is about 1300 pounds and the individual main gear about 2500 pounds. Crew selection of landing gear "down," after the arm switch has been selected, accomplishes two functions for the nose gear. It energizes the landing gear selector valve, porting pressure to he down-side of the nose gear strut accuator and the down-side of the uplock release actuator. In addition, a redundant pyrotechnic backup system is sequenced to release the uplock, if the primary hydraulic system fails to operate in a "short" period of time.

There is only one primary hydraulic power system configuration for the nose gear operation. The gear then "free falls" from the wheel well, there by driving the mechanically linked doors open. Aided by weight and aerodynamic effects, the gear should reach the full down position and be locked in position by the action of a spring loaded bungee. The motion of the gear before locking down will be damped by an oil snubber to prevent any damage to the locking linkage. Down pressure to the strut actuator aids in the extension cycle, but in the event hydraulic power should be lost, it is not required to extend or lock the gear down. Gear downlock and gear/door uplock switches provide cock-pit indication of gear position. The extension cycle is designed to be accomplished at all velocities up to and including 300 knots within a time limit not to exceed 10 seconds.

The main landing gear extension cycle is identical to the nose gear with the following exception. In place of the backup pyrotechnic release system, two additional secondary hydraulic systems are provided for the uplock release actuator. Therefore, crew selection of landing gear "down" ports down pressure from the primary hydraulic system to the strut actuator and to the uplock release actuator. It should be noted that any one of the three systems is sufficient to release the main gear and door uplocks and initiate gear extension. Primary pressure to the strut actuator aids extension but is not required, as the weight and aerodynamic effects on "free fall" gear are sufficient for gear extension and locking via a spring bungee.

There is an Autoland System interface with the landing gear system which has not been fully defined as yet. Operation of the gear, during the landing operation, is actuated as late as 14 seconds before touchdown. Manual gear extension is achieved by the pilot throwing a gear extension switch after he sees a light on the display panel. It is expected in the Autoland system that the autoland hardware would accomplish the same action at about the same time. The problem then is obvious. With a maximum of ten seconds allowed for the gear to go into the down and locked position and the action initiated some 14 seconds before touchdown, there is little if any leeway for problems in response or deployment. Therefore, the reliability of the system must be very close to 100 percent during that 14 second to 4 second period prior to touchdown or some alternate action capability must be supplied along with a longer period to achieve down and locked gears.

4.6 Range Safety

Current requirements have established the range safety system as an add-on unit only for the design, development, test and engineering flights. The baseline system is shown in Figures 53 and 54. This system is still under discussion between NASA and the Air Force. Basically, the range safety system is required to provide for: (1) safety of lives and property, both on the ground and in flight, (2) External Tank propellant dispersion, and (3) protection against overt/co-vert destruction of the vehicle and against "false alarms" due to electro-magnetic interference or spurious signals.

Issues under study at this time include the following:

(a) External Tank propellant dispersion and their impact on Orbiter (MSFC).

(b) Crew ejection seat inhibit which inhibits range safety system operation. Adequate procedural safeguards and time delays appear necessary to maximize astronaut survival if destruct action is required.

(c) Shutting down of the Orbiter's main engines upon re-ceipt of the range safety destruct system arm signal.

(d) Inflight safing of the "safe and arm" device by the Orbiter software.

(e) Monitoring of the safe and arm device to prevent in-advertant safing of the range safety destruct device.

4.7 Materials Usage and Control

One of management's major controls to assure the design and construction of safe and efficient hardware is in the materials' usage area. This includes not only the compatibility of materials with their environment from the standpoint of flammability and toxicity but also with regard to their stress corrosion/fracture mechanics susceptability. The Shuttle program, using the experience gained from prior manned programs and military and commercial activities, has developed materials' programs for each element as well as for the integrated Shuttle system. Requirements set by the program and affecting all elements within the program are set forth in Paragraph 3.6.2.1, JSC 07700, Volume X, "Space Shuttle Flight and Ground Systems Specification," and the JSC document SE-R-0006A, dated April 1973, "Requirements for Materials and Processes."

These requirements include the following:

(a) Each element must have a controlling document on materials and processes stating the specifications and standards to be used. There is a drawing review and sign-off by a materials' engineer.

Materials testing and "allowables" are covered by:

(a) Flammability, odor, outgassing in NASA NHB 8060.1A.

(b) Thermal-vacuum stability in NASA SP-R-0022.

(c) Special tests as approved by JSC where it is felt that they are required to assure materials compatibility within the context of their use.

(d) ASTM test methods are applied as required.

(e) MIL Handbook No. 5, 17, 23.

Material selection lists are developed based on experience and known material compatibility with specific environments. There are also fracture control and material control plans. Each element contractor has developed its own metals/nonmetals/processed which have been reviewed by and approved by NASA.

The Space Division, Rockwell International Corporation, as the Shuttle system contractor, has developed a materials' tracking and control system called "MATCO." While they do not control the use of materials on the Shuttle elements, they do bring material usage which they feel falls outside the set requirements to the attention of the NASA/JSC project office for further action. In addition, materials-conscious personnel participate in the Panel and working group activities as well as in the reviews conducted on Shuttle elements and subsystems. The Panel will continue to review this question of decision making on materials' acceptance during future reviews at various contractor and NASA sites.

The "MATCO" system noted above contains pertinent data on both metals and non-metals, generates material selection lists, contains usage data on --- what materials are used, where used, quantity, results of usage evaluation, deviation status where there is a deviation from accepted use, and finally the system generates output reports to permit certification of the acceptability for a given configuration usage.

The "MATCO" system on the Orbiter has been implemented since the first drawing release. Associate contractors for other elements of the Shuttle program are currently encoding the data and it is expected that element contractor data outputs may start about January 1976. Payload coverage is under discussion at this time.

4.8 Failure Mode and Effects Analysis (FMEA)

Elements of the Shuttle system and the interfaces beteen elements are subjected to detailed FMEA's. In addition to the FMEA documents there are Critical Items Lists (CIL's), Hazards Lists, Shuttle Hazard Analyses forms (SHA's), and Safety Analysis Reports (SAR's). Taken together they provide a systematic means of assuring nothing, in so far as possible, "falls into the crack." They provide for early identification and resolution of potential problem areas, support design reviews, provide management visibility, and establishes a documented baseline to facilitate hazard/risk/safety problem resolution. In addition this work provides a basis for establishing mandatory test and inspection points under the Quality Control Program and provides valuable input for the maintainability program for Shuttle.

The priority or level of criticality number system is in use, as it has been in prior manned programs. The listing is provided for information:

Criticality Category	Definition (Potential Effect)
1	Loss of life or vehicle, including loss or injury to the public.
2	Loss of mission, including post-launch abort and launch delay sufficient to cause mission scrub.
3	All others (structural or TPS type elements are not classified in any of these above categories if they meet the margin of safety requirements).
3_I	Criticality 3 items which meet one or more of the following categories: (a) Redundant elements are not capable of checkout during normal turn-around. (b) Loss of a redundant element is not readily detectable in flight. (c) All redundant elements can be lost by a single credible event or cause.

5.0 TEST PROGRAMS

5.1 Verification Plans

A Shuttle Master Verification Plan (JSC 07700-10-MVP-01 Rev. A) establishes the requirements and plans to certify the Shuttle system ready for operational use. Since much of the program's confidence will be based on test requirements and results, the Panel has reviewed the evolution of the ground and flight test program including the impact on crew safety of changes in requirements.

5.2 Ground Tests

In most of the preceding sections of this report there have been discussions of test programs as they applied to the specific development of subsystem components, such as the tiles for the Orbiter Thermal Protection Subsystem. The ground tests discussed here are those termed "major ground tests." Such tests involve a combination of system elements and complex facilities. The major ground test programs are outlined in Figure 55.

The ground vibration test program verifies load, vibration, flutter, and flight control system analysis. Vibration testing is performed on a one-quarter scale Shuttle model and on the liquid oxygen tank portion of the External Tank. The first Orbiter will also be subjected to a horizontal vibration test at the Palmdale Assembly Facility as a part of the vehicle checkout. The major full scale

Space Shuttle vertical vibration tests are planned to be carried out at the Marshall Space Flight Center to study the vibration modes of the total assembled Space Shuttle vehicle. Recent changes in the ground vibration test (GVT or MGVT) include:

(a) Deletion of component ground vibration tests on the Orbiter wing, Orbiter vertical fin, and other components.

(b) Delay of the quarter-scale model testing for six-months.

(c) Compression of the mated vertical ground vibration tests to a six months time period.

The vibro-acoustic test program verifies the predictions about the dynamic response of the structure and internally mounted equipment to engine noise and vibration, aerodynamic buffeting and aerodynamic noise. Wind tunnel tests of models have been used to determine the aerodynamic noise pressure levels. Scale model tests of the total Shuttle stack are being used to predict the launch environment and its impact. Full scale tests of a major segment of the Orbiter are to be conducted in the vibro-acoustic test facility at JSC. Recent changes in this test program include the deletion of the forward fuselage vibro-acoustic test.

The Main Propulsion System test program uses the three main engines mounted on a simulated aft section of the Orbiter, together with the External Tank, and includes all necessary plumbing and controls. POGO suppression hardware will be supplied for installation as the tests progress to substantiate the technique used to suppress the longitudinal vibrations peculiar to POGO. These propulsion tests will also provide additional vibration and acoustics information. Recent test program changes include the deletion of the vertical firing attitude, deletion of flight disconnects from the "T-0 Umbilical" and an increase in firings from 14 to 15.

The Orbiter avionics components and their related software and hardware interfaces will be tested at the Rockwell International Space Division's Avionics Development Laboratory. The Avionics Development Laboratory is an engineering tool with emphasis on development support, subsystem evaluation and initial hardware integration. Test results are aimed at:

(a) Demonstrating line replaceable unit functions for all those pieces of hardware that fit that category.

(b) Developing the single-string data processing system functions.

(c) Avionics compatibility with automatic ground checkout equipment.

(d) Progressive testing and combining of subsystems until they simulate a flight control system with computer inputs and control actuator outputs.

The Shuttle Avionics Integration Laboratory (SAIL) established at JSC will conduct avionics systems integrated testing in support of the Approach and Landing Tests (ALT), Vertical Flight Tests (VFT), and operational mission phases. Integrated testing includes both open-loop and closed-loop testing. Open-loop testing will integrate and verify the avionics system compatibility and redundancy management techniques; closed-loop testing will integrate the avionics hardware and software systems and verify that they are capable of performing each flight phase of the mission. Thus the SAIL is a central facility where the avionics and related hardware (or simulations of the hardware), on-board ground support software, flight software, flight procedures, and associated GSE will be fully integrated and verification tested. Figure 56 shows the Shuttle avionics systems which are to be tested on SAIL.

Another facility supporting the avionics test program is the Software Development Laboratory (SDL). The purpose of this facility is to accomplish flight software development and flight software independent verification.

Static structural tests are planned for major structures on all Shuttle elements. A full-sized Orbiter airframe structural test article (STA) will be tested at Palmdale to determine if it can withstand the design limit and ultimate loads. In addition, it will be subjected to fatigue loeading up to 400 cycles to assure structural integrity. An Orbiter crew module test article, which is the pressurized crew compartment segment of the Orbiter, will be tested in a manner similar to the static test article mentioned above.

The External Tank structural program includes a structural test article consisting of flight-type liquid oxygen and liquid hydrogen tanks and intertank. Tests will be conducted to verify structural integrity at limit and ultimate loads and to determine the liquid oxygen tank model characteristics necessary to determining the all-up Shuttle vehicle structural characteristics.

Solid Rocket Booster and Solid Rocket Motor structural tests will be conducted, as will hot firings to verify their structural integrity, support development of the rocket motor case and verify ballistic performance.

Recent test program changes have deferred the crew module structural test, deferred the airframe structural test, eliminated one intertank structural test article from the External Tank program, deferred the Solid Rocket Booster structural test and deleted the booster first development firing.

The Orbiter thermal vacuum test programs on the forward fuselage, aft fuselage, and OMS/RCS pod have been deleted. The impact of deleting the major ground thermal vacuum test has been subject to study by both JCS and Rockwell International over the past few months. The following results stem from these studies but must be considered in light of additional more detailed work now in progress:

(a) There is an obvious requirement for flight test data.

(b) There will be no off-limit or off-nominal testing to any degree.

(c) There will be no physical pre-flight data on temperature effect of subsystem operation on the integrated vehicle.

(d) There will be some restructuring of the certification/validation program to include additional component and subsystem testing.

(e) Requirements for additional development flight and operational flight instrumentation requirements will have to be determined.

(f) Mission planning will have to pay more attention, in the early flights, to beta angle variations, time required for temperature stabilization.

(g) Conservative attitude constraints will be necessary on the early orbital flights.

Test article fidelity has always been a problem in extrapolating model tests and full size ground tests up to the actual flight hardware and how it operates in its real environment. The ability to extrapolate from ground test activities to flight operations depends upon the degree to which the test articles resemble the flight articles. A Flight Readiness Firing test (FRF) will functionally verify the integrated shuttle system vehicle, launch complex and operating procedures and thus demonstrate the maturity and readiness of the shuttle system for first manned vertical flight.

The Solid Rocket Booster/External Tank separation system test and the Orbiter/External Tank Separation tests are two major tests deferred to flight test program. The verification logic is shown in figure 57. The panel has made a point of repeatedly asking if data were being lost from ground tests that would be useful to our basis of confidence in crew safety during early flights.

The answers given were: "No tests are being conducted during the Approach and Landing Test and Orbital Flight Test programs which affect crew safety that have no counterpart in the ground test program. . . . All elements and maneuvers of the flight test program have counterparts in either ground tests, simulations, or analysis."

5.3 Flight Test Program

The flight test program has two major subdivisions: The Approach and Landing Test Program (ALT) and the Orbital Flight Tests (OFT). These flight tests complement the ground test program described previously and the ALT is planned to commence in mid-1977 using the Boeing 747 carrier aircraft, and the OFT is planned to commence in mid-1979.

5.3.1 Approach and Landing Test Program (ALT)

The Orbiter vehicle 101 (the first off the line) is the primary vehicle planned for the ALT and is configured to include the equipment necessary to evaluate vehicle approach, landing and deceleration requirements dictated by the terminal phase of the operational mission. The design of Orbiter 101 is such that minimum modifications are required to convert it to the operational configuration.

The ALT program is designed to progress from test conditions that provide the greatest margins of safety to test conditions duplicating those expected on the first Orbital Flight Test landing. The ALT program is comprised of two flight test phases:

Phase 1 - Inert Orbiter/747 mated tests to verify satisfactory airworthiness of mated vehicles for supporting orbiter free flight tests.

Phase 2 - Manned Orbiter captive flights to develop Orbiter release profile and Orbiter free flight and landing data.

During ALT the Orbiter is flown without any propulsive power. With the current capabilities of the Orbiter/747 combination, the maximum attainable altitude appears to be somewhat less than 28,000 feet, and with the loss in altitude which is said to occur during the release period the Orbiter would appear to be in free-flight starting at about 20,000 to 24,000 feet. These tests are to be conducted in the area surrounding the Flight Research Center, Edwards, California.

The status of the two phase ALT test plan is:

PHASE 1 -

(a) The extent of the initial Taxi tests of the mated Orbiter/747 at Palmdale has not been fully defined as yet.

(b) The planning for ALT is being done by FRC, Rockwell and Boeing. They will define the requirements under the review of the Orbiter Project Office at JSC. These requirements will appear in the Approach and Landing Test Requirements Document. The actual flight tests needed to meet these requirements will then be developed by the same team.

They will appear in the ALT Mission Objectives Document.

(c) The actual test program will be constructed in a manner that will permit the achievement of objectives to get to the manned Orbiter release point with a minimum number of flights and flight hours.

(d) The ALT manager is from JSC and the assistant manager is from the FRC. The tests are conducted for the ALT manager by the FRC flight test team and during these operations the FRC flight test control room will be utilized to control the flights.

(e) The 747 test instrumentation system is designed and installed by the same team. It will be compatible with the FRC test control and data reduction facilities. Data reduction and analysis by FRC is conducted with JSC support and the same tapes and other data are forwarded to JSC for their independent analysis.

(f) It is expected that during this phase of the program that Ferry configuration flight tests will be conducted in parallel on a non-interference basis.

PHASE 2 -

(a) Phase 2 begins at the completion of the inert Orbiter/747 testing. The current baseline consists of eleven Orbiter free flights, starting with pilot-controlled landing series (5 flights); autoland landing demonstration (3 flights); and finishing with weight/c.g. envelope investigations (3 flights). These free-flights are being structured to allow early termination of the program or to skip individual flights if testing shows the data are not required. During the initial portion of this phase, the manned Orbiter captive flights are held to a minimum necessary to develop the release (techniques).

(b) The flight test team is to be headed by a JSC test conductor and comprised principally of JSC and Rockwell flight control personnel. The control of the flights will be from the JSC control room with a test liaison group stationed at FRC. It is expected that FRC will supply experienced aerodynamic flight controllers to the JSC control center.

(c) The planning, including requirements and flight test details, are established and developed by the NASA/Rockwell team under the auspices of the Orbiter Project Office at JSC. The free-flight test program is developed specifically by the Flight Operation Division of JSC and becomes a part of the ALT Mission Objectives Document.

The baseline flight test program as provided to the Panel at the time of its review and inspection visits shows 14 carrier/orbiter inert flights; 5 carrier/orbiter active flights to refine separation techniques and to do integrated systems testing, and 11 orbiter free-flights. Table XIII is a further explanation of the Orbiter Free Flight Program at this time.

Given its special interest in the complex avionics system used on the Orbiter the Panel asked a number of questions regarding flight control avionics support of the ALT program. The many ground tests conducted prior to flight will give a basis for confidence in the avionics subsystems used on the ALT program. In addition, the orbiter will contain an "all-up" fail operational/fail safe flight control avionics subsystem with a dedicated backup flight control subsystem and a backup air data nose boom system. At the same time the ground support group will have the support of Shuttle Avionics Integrated Laboratory, Software Development Laboratory, and the Avionics Development Laboratory available.

5.3.2 Shuttle Training Aircraft

The Shuttle Training Aircraft is a Grumman Gulfstream II turbojet aircraft modified to provide an inflight simulation of Orbiter performance and flying characteristics in the Terminal Area Energy Operations. The purpose of this training program using the modified Gulfstream II is for pilot training and the development and verification of procedures. The simulation system consists of a specially constructed and programmed simulation computer and necessary inertial sensor systems. The displays, controls, radio, navigation systems are essentially Orbiter Hardware. The simulation capability is as follows:

(a) Altitude - 43,000 feet to simulated touchdown

(b) Airspeed maximum of 350 knots or Mach number of 0.8

(c) Payload of 5600 pounds

(d) Orbiter modes simulation for automatic landing systems/control stick steering and backup systems

(e) Turbulence and wind conditions expected to apply to Orbiter operations

5.3.3 Orbital Flight Tests

The culmination of the flight test program occurs with the manned Orbital flight, a program currently encompassing a sequence of six manned flights. The first orbital flight is designed to be short and benign to demonstrate basic flight worthiness. A decision was reached by senior NASA management to proceed with the design and development of the manned first flight only after prolonged and detailed study of the manned versus unmanned options. A review of the decision will be conducted eighteen months prior to the first orbital flight. A summary of the manned vs. unmanned study provided to the Panel is given below:

(a) Recovery of the Orbiter on every flight is required for orderly continuation of the flight test program.

(b) Flight experience shows many cases where the presence of crew saved the mission from failure.

(c) The crew role in the shuttle is identical to that in aircraft and spacecraft test operations; however, crew capability in some areas of the shuttle design concerns is very limited.

(d) Manned landings can be made at alternate sites in the event of dispersed entry conditions or automatic system failure. Capability of crew to deal with contingencies provides greater safety for the population in the landing area.

(e) The ground test program has been constructed to give confidence that design concerns have been acceptably minimized prior to the first orbital flight, manned or unmanned.

(f) Tailoring of the first vertical flights to improve safety margins will be accomplished as practical for either manned or unmanned flight tests.

(g) Abort and ejection capabilities are consistent with aerospace testing precedents, that is they cover many but probably not all foreseeable failure possibilities.

(h) Commitment to unmanned flight implies a successful Approach and Landing Test Autoland program as a prerequisite.

(i) Unmanned capability requirement can be reinstated later if unforeseen circumstances demand.

The early development Orbital flights will be launched from the KSC site and will land at Edwards Air Force Base. These flights are to be under the control of the JSC Mission Control Center once lift off is achieved. Depending upon the progress achieved in the early flights, there is a good chance that the fifth or sixth flight will both launch and land at the KSC site.

The contingency planning and design for abort conditions during the flight test program will continue to be of great interest to the Panel. This is true for both the Orbital and ALT programs. The Panel, for instance, is interested in plans to assure that requirements of abort operations and system capabilities are compatible.

6.0 SYSTEMS INTEGRATION

6.1 General Objectives

The management of the integration effort has been covered in earlier sections of this report. This section is meant to identify the technical challenges of integrating the elements at this point in the Panel's review.

An example of the many technical areas that must be managed to assure that the Shuttle elements work together are:

Flight Performance

Load and Structural Dynamics

Flight Control

Integrated Avionics

Integrated Propulsion/Fluids

Mechanical Systems

Ground Operations

Major Integrated Ground Tests

Computer Systems and Software

Systems Engineering

Safety, Reliability and Quality Assurance

Payload Accommodations

The Main Propulsion System is used here to illustrate the complexity of the relationships between components found in various elements which form single end-to-end integrated systems. Other areas to be examined by the Panel include electrical system and avionics system.

6.2 Systems Integration Challenges

Some of the challenges the program must resolve on the Space Shuttle System are:

Flight Performance Margins

Induced Loads

Ice/Frost Shedding

SRB/ET/Orbiter Separation

POGO Suppression

Forebody drag

Many of these challenges have been discussed in the section of the report on the various program elements.

6.3 Operations

The Orbiter is designed to carry a crew of up to seven including crew and scientific personnel. On a standard mission, the Orbiter can remain in orbit for seven days. While it is planned that an Orbiter would be readied for another flight in fourteen calendar days, the Shuttle can be readied for a rescue mission launch from a standby status within twenty-four hours after notification. For emergency rescue, the cabin can accommodate as many as ten persons so that all the occupants of a disabled Orbiter could be rescued.

Space Shuttle operations consist of four basic phases:

(a) Lift-off to orbit insertion

(b) On-orbit operations

(c) De-orbit to landing

(d) Ground turnaround to prepare for the next flight

Operational constraints have been discussed in previous portions of this report under each of the elements of the Shuttle system as well as in the reliability, quality and safety sections. The Panel's interest continues to focus upon the ability of the nominally designed hardware to meet the contingency situations which can occur during flight test and operational phases of the program. We will monitor the evolution of the launch rules and the mission rules governing both test and operational flights. We will also monitor such safety challenges as (a) intact abort capability, (b) contingency abort capability, (c) payload accommodations, (d) day and night operations, (e) mission control center requirements, (f) post landing thermal conditioning, and (g) EVA operations.

6.4 Main Propulsion System

The Main Propulsion System integrates the Space Shuttle Main Engine (SSME), External Tank (ET), and the interconnecting plumbing and controls within the body of the Orbiter. The subsystems that make up the main propulsion system are:

(a) Propellant feed

(b) Propellant fill and drain

(c) Engine prestart propellant conditioning

(d) ET pressurization and prepressurization

(e) Helium storage and distribution

(f) Propellant management

(g) SSME GN_2 purge using ground supply

(h) POGO suppression

(i) Electrical instrumentation, controls, and displays

A schematic of this system is shown in Figure 58. The selected POGO suppressor system is shown in Figure 59 and the workings of the POGO Integration Panel are shown in Figure 60.

The Main Propulsion System has been designed to meet the fail-safe criteria. Thus, for example, loss of one main engine during ascent would still permit the crew to abort a Mission 3A as follows:

0-250 seconds .... suborbital powered return to launch site

250-330 seconds .... abort once around

330 - main engine cutoff ... mission completion

Shutdown of two of the main engines will result in loss of the Orbiter for a majority of mission phases during the ascent.

Prevalves, fill valves, and disconnect valves are all designed to remain in the last actuated position, in the event of loss of pneumatic pressure to the valve actuator, or loss of electrical power to the controlling solenoid valves. Pneumatic pressure is continuously applied to these valves during their critical function period, to further assure their remaining in the desired position.

6.5 Summary

The Panel has examined a portion of the efforts conducted in integrating the total Shuttle system during the past reporting period. With the completion of the Preliminary Design Reviews for each of the elements and the Space Shuttle System, the Panel can better undertake a review of the integrated systems which cross over element interfaces such as the electrical system, and the mentioned Main Propulsion System.

7.0 APPENDIX

7.1 PANEL AUTHORITY

The Aerospace Safety Advisory Panel was established under Section 6 of the National Aeronautics and Space Administration Authorization Act, 1968 (PL 90-67, 90th Congress, 81 Stat. 168, 170). In addition, the Panel has been rechartered pursuant to Section 14 (b) of the Federal Advisory Committee Act, (PL 92-463, October 6, 1972). The duties of the Panel are set forth in both the 1968 Act and in NASA Management Instruction 1156.14A dated January 18, 1973: "The Panel shall review safety studies and operations plans referred to it and shall make reports thereon, shall advise the Administrator with respect to the hazards of proposed or existing facilities and proposed operations and with respect to the adequacy of proposed or existing safety standards, and shall perform such other duties as the Administrator may request."

Over the years the Panel has evolved its role to include not only safety per se, but has included mission success as a consideration that it should be concerned with, as well as crew or public safety. We feel that this broader consideration of the programs and their management gives us more confidence in the more limited area of safety alone.

7.2 PANEL ACTIVITIES

January 15, 1974	MDAC-East Role in Shuttle Program Organization Orbital Maneuvering System Baseline, Schedule, Status Integration of Pod into Orbiter Reaction Control System Requirements	MDAC, St. Louis, Missouri
February 26, 1974	Program Manager's Top View TPS Development Status Systems Integration Management Man-in-The-Loop Ferry Mode Preliminary Design Review Results	JSC-Houston
May 13-14, 1974	The External Tank Program, Overall View ET Baseline Design Program Interfaces Major issues and their proposed resolution. Lightning Protection Design Transportation Structural Test Program Reliability, Quality Assurance and Safety Subcontractor program MSFC Management of the External Tank Program	Michoud Assembly Plant, LA
June 5-6, 1974	SSME Quarterly Review SSME Controller discussions	MSFC, Huntsville
July 16-17, 1974	Space Shuttle Main Engine Controller Program Overview Responsibilities, Role, Organization Controller Technical Description Computer Program Overview Plated Wire Memory Theory Memory structure build-up Technical Review---in depth Design Control and Configuration Management Production and Procurement R & QA Summary Status MSFC Management of SSME Controller Program	Honeywell, Aero-Space Div., FLA
August 22-23, 1974	The TPS Program Overview and JSC Mgt. Ames' Shuttle related programs Ames' Management Approach and iMplementation	AMES, CA Lockheed, CA

Panel Activities continued:

	TPS materials and tile configuration program Current and Projected facilities and their application to the TPS TPS aero-noise effects program Definition of TPS aero-heating environment and other environmental effects. Rockwell Subcontract to Lockheed and how it is managed Tile Program, Lockheed Organization, personnel, responsibilities Tile materials and processes Tile Production Tile testing Tile R and QA Current Status Current significant problems and their resolutions.	
September 16-17, 1974	RI System Integration Contractor Role Commonality System Safety System Integration Challenges Tour of Facilities and Mockups Orbiter Thermal Protection System SSME Program update ISTB Program Status Combustion devices status Turbomachinery Devices statsus Engine systems and controls status Controller status	RI/Downey, CA
October 15, 1974	Orbiter Approach and Landing Test Program Ferry Operations Manned vs. Unmanned External Tank disposal after flight Space Shuttle Flight Test Program Abort/Contingency Operations and their impact	JSC/Houston
January 6, 1975	Space Shuttle Update and Status Report Approach and Landing Test, PDR results Avionics and their management Management and Direction of Systems Integration MSFC Space Shuttle Survey and Major Management and Technical Challenges Main Engine, External Tank, SRB, Orbiter Program Revisions under active consideration Current status	JSC/Houston

March 3, 1975	KSC Space Shuttle Planning	KSC, Florida

KSC Roles and Responsibilities
- Operations, Maintenance, Sustaining Engineering

KSC Organizational Relationships
- Overall Organization
- Intercenter Relationships
- Participation in Panels, Working Groups, Task Teams
- Contracting Philosophy
- Manpower planning
  Experience levels, skill retention, skill mix.

Overview of Ground Operational Tasks
- Shuttle
- Payloads (offline)

Documentation and Control

Facility and GSE Overview
- Types and KSC effort/Responsiblity
- KSC facility baseline/current status/ problems
- Test Facilities/Plans/Schedules
- Launch Preparation System
  System Operation
  Software Validation/Test/Use of SAIL

KSC Operational Flow
- Ground turnaround
  Allocation vs. Assessment
  STAG/Control
- Payloads, online

Summary of KSC Shuttle operations

April 7-8, 1975	Space Shuttle Systems (MSFC Elements)	MSFC, Alabama

- POGO Prevention Planning and implementation
- MSFC Integration Activities
- MSFC Change Processing
- MSFC Systems Tests
- Single Failure Point Designs

Solid Rocket Booster Project
- Description and Status
- Integration
- Recovery/Retrieval
- SRM

External Tank Project
  Description/schedules/cost highlights
  Top Problems/Sepcial Topics
  Procurement and Manufacturing status and problems

SSME Project
- Overview
- Integrated System Test Bed (ISTB) Plan/Status
- Controller status
- Hydraulic Fluid Status
- Fabrication Learning
- Heat Exchanger
- Ground Operations Planning

MSFC Summary

May 5-6, 1975	Shuttle Assessment of Technical and Management challenges Thermal Protection System Review Hazard Analysis and Risk Assessment Mechanical Hinges, Gear Boxes, and Doors System Hazards associated with asymmetrical thrust of SRB's Procedures/Ground Rules to Alleviate System Failures Hazradous Gas Detection System Level II Interfaces Material Usage and Control Range Safety Ground and Flight Test Programs POGO Prevention Lightning Design and Protection SAIL	RI/CA

7.3 RESPONSE TO PANEL'S 1974 ANNUAL REPORT

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON, D.C. 20546

REPLY TO ATTN OF: MQ

MAY 23 1974

MEMORANDUM

TO: AA/Associate Administrator

FROM: M/Associate Administrator for Manned Space Flight

SUBJECT: Annual Report of the Aerospace Safety Advisory Panel (ASAP)

The Annual Report of the Aerospace Safety Advisory Panel has been distributed to each of the MSF Centers and Program Directors for their careful review. The Program Directors have each coordinated responses to their pertinent items in the report and these detailed responses are attached.

Significant responses from the ASTP office relate to Volume II of the report, pages 8-9, items 1 through 11. They describe a continuing strong program management concept with emphasis on enhancement of personnel motivation and training. The Panel's concern over the need for formal reviews is being met by monthly joint reviews and bi-weekly telecons between the U.S. and Soviet Technical Directors and their staffs. Qualification test data reviews are being continuously held to assure a ready-to-go status. Language training is progressing well on both sides and a recent crew training exercise in Houston accomplished a complete transfer in both English and Russian. FMEA's have been completed for all systems of the CSM and DM/docking system. The Mission Control Center Interaction Plan is in excellent shape and both countries plan a team of experts in each other's control room to assist each Flight Director. Mission simulations are continuing with both U.S. and Soviet crews participating in each other's facilities. Effort is continuing on tracking failures or inadvertent operations which could affect the other crew or spacecraft. It is planned to improve communications by using ATS-F but no contingency action is planned if it is not available. Stadan provides the primary communications coverage and exceeds the minimum requirements for ASTP.

Finally, in response to the Panel's question on sneak circuit and fault current analyses, these are being accomplished on both the CSM and DM/docking system.

The Panel expressed a strong recommendation that the Skylab experience be utilized to the maximum degree possible on current programs. Skylab has almost completed the publication of a series of "Lessons Learned" documents. My office, on March 12, 1974, levied an action item on each Program Office to review these documents and report back to me on implementation of these "lessons learned." I will make these responses available to the Panel upon their receipt.

Significant responses from the Space Shuttle Office relate to Volume I, pages 12 through 16, and Volume II, pages 19, 35, 43 and 45 through 47. In the area of the Panel's concern about integration activities of Rockwell, JSC has given a task to the contractor to look at separating their integration function from the Orbiter task (due May 31). In the area of subcontractor/vendor control, Rockwell is rewriting their Procurement Management Plan with a new emphasis on commonality management (expected by June 1974). In response to the concern with weight control, a combination effort of strict weight control measures, a specific Orbiter weight reduction activity, and a series of overall weight and performance trade-offs are being pursued. In the area of abort requirements, continuing attention is being paid to determine abort capabilities for the various mission phases for the design which is evolving from the driving requirements of operational uses. The Panel expressed concern in the Avionics area because they felt that the systems were on the leading edge of the state-of-the-art. The response indicates that the program has a handle on the design solutions. Specifically, experience on both hardware and software for a Performance Monitoring System has been gained at the Mission Control Center. Good judgements based on these experiences will be exercised to keep requirements manageable. Similarly, the Autoland System is being very carefully designed using the 16 unpowered automatic approaches and landings with the CV 990 as an experience base. Also being used is Sperry-Rand with their CV 990 test program experience. With regards to the man-in-the-loop versus automated systems, an approach of using automatic functions for expensive and sophisticated systems where split-second decisions are required is being followed. This is borne out by 747 aircraft use of Autoland for consistent low "g" landings in all weather. Turnaround time is of great concern and is receiving full attention of a panel working with latest design, logistic and maintainability information as it becomes available. The concern about all-weather capability is being worked both with regards to effects on the TPS and on Avionics. It may be necessary, however, to sacrifice some all-weather characteristics for thermal characteristics on the TPS. Operational alternates are available since chances of bad weather at both prime and contingency landing sites is very low. In addition, automatic landing and overrun equipment is being installed to better handle all weather problems. On the SSME Controller, the Panel had questioned the reasons for not considering a magnetic core memory. The response lists a series of reasons for not using the core approach but also indicates that an MSFC committee is reviewing the whole controller development problem with a report to JSC due on May 22, 1974. The Panel felt that test organizations at Rockwell were not yet firmly established. This area has since been significantly improved and staffed, including government roles and responsibilities for most of the test sites. On the TPS the Panel correctly pointed out that major design issues include strain, isolation, adhesives, joints, TPS/fuel compatibility, dynamic seals and development of a 100-mission life coating. In response, an up-to-date status of development testing on each of these design issues is provided in the attached detailed answers. On the SSME, the change to Mil-H-83282 hydraulic fluid caused some questions on possible further evaluation required. In response, materials in contact with the fluid are being identified and materials compatibility is being reviewed (including DOD testing and service experience). In addition, an acceptance and design verification program is being initiated to test SSME components and systems with Mil-H-83282 fluid. The Panel also questioned whether the SSME flex line material was compatible with oxygen and not subject to hydrogen embrittlement. This is a well-recognized problem and the materials have been selected accordingly. The Panel pointed out the different requirements for the SSME combustion chamber as compared to the J-2 engine. The response indicates that the Narloy material was selected to best meet the unique requirements of high thermal conductivity, high strength and ductility, high metalurgical stability and life characteristics. Although the Panel next pointed out that the optimum technique for reentry has not been defined, the response indicates that much wind tunnel data, flight simulations, aerothermal dynamics work, etc. which is in progress may cause many changes and the technique may well have to be developed from operational phase experience. The Panel also questioned adequacy of controls for qualification of "off-the-shelf" hardware. A special Level II Directive was deemed necessary to insure adequate controls and it is in the final review/approval cycle. Finally, the Panel's concern for effective measures to prevent stress corrosion was recognized early by the Shuttle Program and is controlled by a NASA materials and process specification, including a contractor materials control and verification plan, which incorporates material sign-off of drawings and records of all deviations with rationale for each.

In conclusion, I would like to thank the Panel for its thorough and excellent report and assure them that their thoughtful questions are continuing to provide an excellent checklist for our program management function.

John F. Yardley

John F. Yardley

Attachments
as stated

KMI 1058.7/SP

October 10, 1973
Effective Date

JOHN F. KENNEDY SPACE CENTER, NASA
MANAGEMENT INSTRUCTION

SUBJECT KSC/MSFC MEMORANDUM OF UNDERSTANDING FOR SHUTTLE EXTERNAL TANK (ET) AND SOLID ROCKET BOOSTER (SRB) SUPPORT EQUIPMENT

1. PURPOSE

This Instruction incorporates into the KSC Issuance System a Memorandum of Understanding between the John F. Kennedy Space Center, NASA (KSC) and the George C. Marshall Space Flight Center (MSFC) for Shuttle External Tank (ET) and Solid Rocket Booster (SRB) support equipment. This Memorandum establishes those items of support equipment for the Shuttle External Tank and Solid Rocket Booster which will be the responsibility of KSC and those items which will be the responsibility of MSFC.

R. C. Hock

R. C. Hock
Acting Director of Executive Staff

Attachment:
A. Memorandum of Understanding

Distribution:
STDL-P

ET and SRB SUPPORT EQUIPMENT

MEMO OF UNDERSTANDING

7/16/73

1. Support equipment has been defined in three categories:

Ground Support Equipment (GSE):

GSE consists of that equipment and associated software which is required to check out, service, handle, provide access to, maintain and safe the External Tank, and Solid Rocket Booster, their subassemblies or other system elements at the launch and landing sites only. Includes such items as:

- Fixed facility access stands, horizontal and vertical
- Facility support and storage stands
- Purge and pressurant gas supplies and consoles
- Ground ECS
- Launch processing system and associated software
- Launch site electrical and mechanical BME
- Standard test equipment
- Standard power supplies and battery GSE
- Ground transportation prime mover
- Facility leak detectors

Special Test Eauipment (STE):

STE consists of that equipment and associated software which is required to support checkout, development, and qualification testing of the External Tank, and Solid Rocket Booster, their subassemblies or other elements during manufacturing buildup and development.

Includes such items as:

- o Internal access platforms
- o Special test cable kits and boxes
- o Other equipment with an intimate design interface with the flight hardware

Transportation and Support Equipment (TSE):

TSE consists of that hardware which is required to transport, handle, and maintain the External Tank and Solid Rocket Booster, their system elements to and from the contractor's facilities, other government facilities, and to and from the launch site and landing sites(s) exclusive of tooling used within the factory and commercial conveyance equipment. Includes such items as:

- o Transporter
- o LRU handling slings and dollies

2. The selected contractor will furnish all materials and services to design develop, test, qualify, manufacture, assemble, check out, and maintain the STE and TSE. Checkout and maintenance at the launch site is excluded.

3. The contractor will identify those items of, and concepts for, ET or SRB support equipment recommended for use at the launch site.

4. The contractor will analyze specified and potential launch site requirements in the design of STE and TSE from a program cost effectiveness viewpoint in order to maximize commonality. This analysis shall show the design/cost savings or impact of commonality.

5. The contractor's incorporation of unique launch site requirements in STE and TSE shall be approved by the NASA Project Office for accomplishment under an existing ET or SRM procurement or shall be accomplished through a supplemental contract arrangement negotiated and managed by the launch site on a case-by-case basis.

6. The selection of common equipment and the identification of launch site requirements will be the responsibility of KSC. The design and development of this common equipment will be controlled by a co-chairmanship of one KSC Support Equipment Manager and one MSFC Manager appointed by the ET or SRB Project Manager. Neither of the co-chairmen would have unilateral authority to proceed with independent development or make changes to this common support equipment; however, generally the MSFC Manager will be the leading element with the KSC Manager concurring in planned direction or changes. Both Managers will have ready access to the contractor for day-to-day technical discussions and problem resolution; however, the MSFC Manager will initiate all formal direction of the contractor. If a disagreement develops between the co-chairmen that could impede the progress of the common equipment development, the matter will be immediately brought to the attention of the appropriate Project or Projects Office Managers at MSFC and KSC.

7. The design and development of STE, TSE, and common support equipment is included in the present ET and SRM procurement; however, the specific units of this equipment that are required for sole use at the launch site will be funded by KSC.

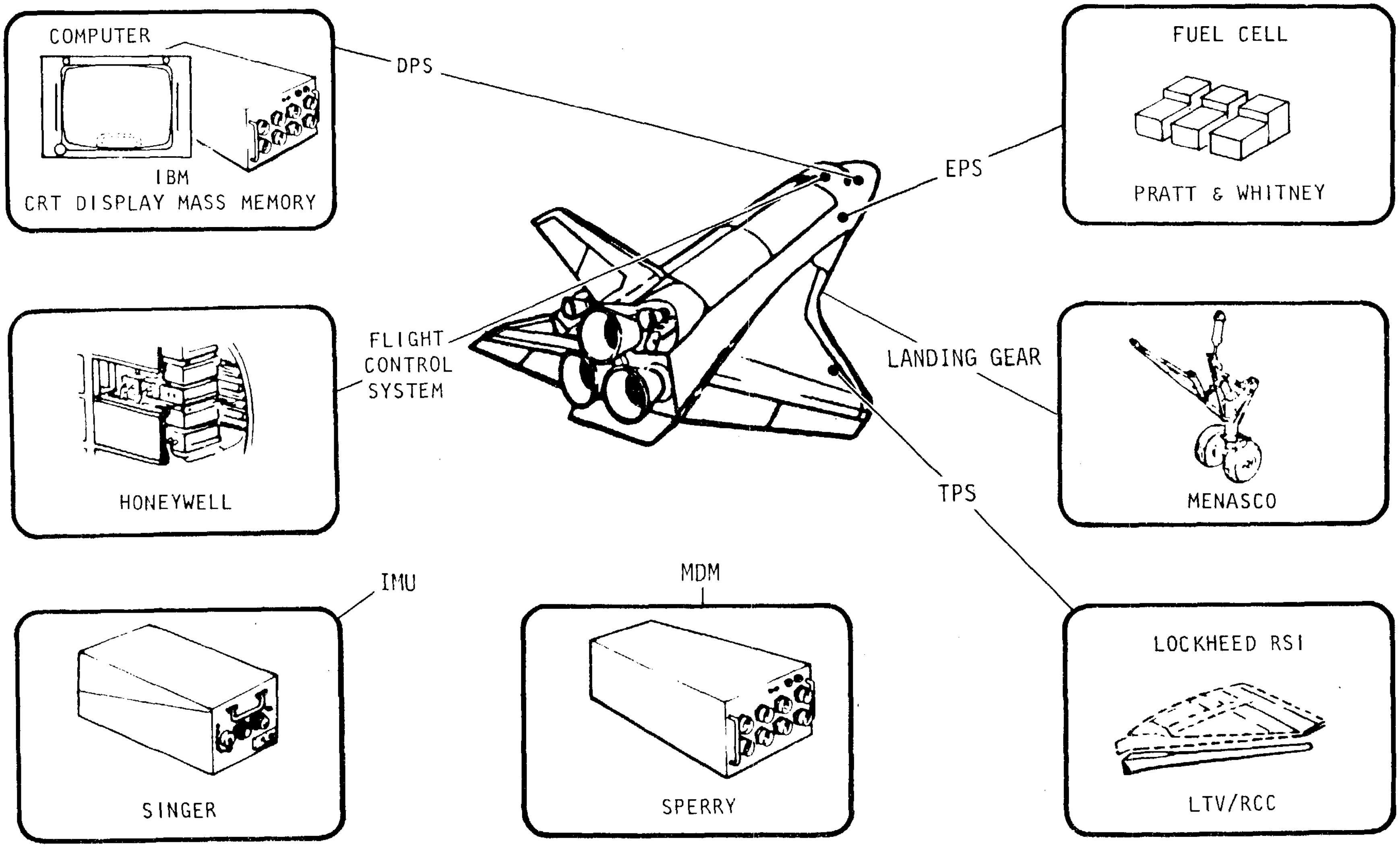
8. The design/procurement/fabrication of GSE is excluded from the present ET and SRM procurements and will be covered under a separate procurement action to be negotiated, managed, and funded by the launch site.

9. If, during the design or development of common usage support equipment, an item evolves to the point that it is no longer cost effective for the program to maintain common usage, then separate design/development actions will be initiated. From this point, the equipment would be classified as STE, thereby placing it under sole MSFC management and budget control; or as GSE, thereby placing it under sole KSC management and budget control.

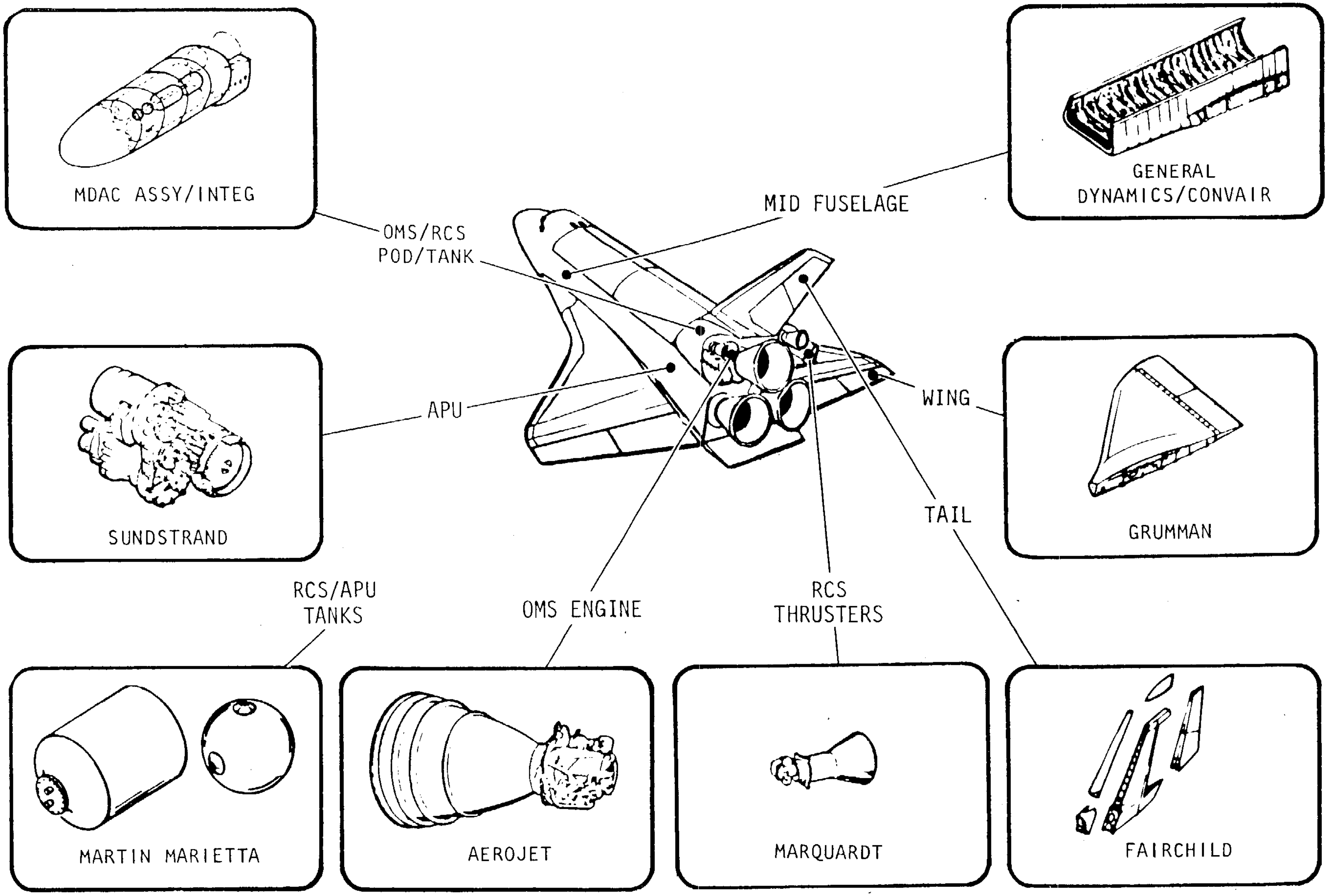
R. H. Gray
R. H. Gray
Manager, Shuttle Projects Office
KSC

Roy E Godfrey
Roy E. Godfrey
Manager, Shuttle Projects Office
MSFC

7.5 MAJOR ORBITER SUBCONTRACTORS



MAJOR ORBITER SUBCONTRACTORS



7.6 SPACE SHUTTLE SYSTEM PRELIMINARY DESIGN REVIEW

Objectives

The purpose of the SSS-PDR is to conduct an end-to-end review to assure that the Space Shuttle System level requirements will be satisfied by current hardware and software design and planning. The system level aspects of the element programs will be examined, including the Orbiter, External Tank, Solid Rocket Booster, Space Shuttle Main Engine, Payload Accommodations and Ground Systems. The objectives to be accomplished during the PDR are to:

(a) Review the total Space Shuttle System design, including as required, individual elements, payload accommodations and the ground systems to assure compliance with Space Shuttle System requirements.

(b) Review current hardware and software design and predicted capability as compared with mission requirements.

(c) Review current designs and plans against quality, reliability, maintainability and safety requirements.

Review Items

At the PDR, the participants will be expected to review various data which describe the system design. These data will include (1) documents (plans), (2) drawings and schematics, (3) manufacturing and test layout and flows, and (4) other back-up data.

Review Operations

Review Teams. The reviews will be accomplished by teams that have the responsibility for reviewing assigned areas. A team captain has been assigned to each of the major technical areas to be reviewed. Each team captain will be responsible to the review chairman for nominating the members of the team necessary to accomplish an adequate review of his assigned area. Each review team should include the NASA technical area manager and support personnel, flight and ground operations personnel, project element representatives as appropriate, and contractor representatives as required.

In accomplishing the review objectives, each team prepares Review Item Dispositions (RID's) to describe significant discrepancies and inconsistencies. Each team captain reviews all RID's generated by his team to eliminate redundancies and duplicate RID's. The team captain submits the team findings and recommended RID dispositions to the review coordinator in the form of a team review packaging consisting of (1) a set of team minutes, and (2) all RID's written by the team. The team captain has the overall responsibility for all activity of his team and assure that all review ground rules and schedules are met. He prepares the appropriate response to each RID and recommends the disposition to be taken.

Review Item Disposition (RID's)

RID's shall be submitted to the review control station as soon as they are written to allow as much time as possible for processing. Every attempt will be made to resolve problems via the review teams during the team meetings.

Screening Group, Pre-Board and Board Operations

Screening Group. The screening group will screen all RID's submitted to avoid redundancy, duplication, or other programmatic problems that may be generated. This group will review the disposition of all RID's and categorize them for review by the pre-board.

Pre-Board. The pre-board will be responsible for reviewing all RID's, with primary emphasis on those items requiring further deliberation or resolution. After the pre-board review, RID's of major importance will be forwarded to the board for final review and disposition.

Board. The board is the final dispositioning authority. All RID's of major importance to the program will be dispositioned at this level. Board presentations will consist of project summaries by each project manager and individual summaries by the team leaders of review accomplishments, problems, matters of significant importance and RID's.

TABLE I

JSC SYSTEMS INTEGRATION OFFICE FUNCTIONS

PRIME

ASCENT & ENTRY PERFORMANCE	MATERIALS & PROCESSES
LOADS & STRUCTURAL DYNAMICS	GROUND SYSTEMS INTEGRATION
FLIGHT CONTROL	MAINTAINABILITY
INTEGRATED AVIONICS	INTEGRATED LOGISTICS
INTEGRATED PROPULSION & FLUIDS	TEST & VERIFICATION
MECHANICAL SYSTEMS	GSE REQUIREMENTS & ANALYSIS
COMPUTER SYSTEMS AND SOFTWARE	MANUFACTURING
SUPPORTING TECHNOLOGY	RELIABILITY
PERFORMANCE & DESIGN SPECIFICATION	SAFETY
FLIGHT TEST REQUIREMENTS	QUALITY ASSURANCE
SYSTEMS INTERFACES	SCA PROJECT MANAGEMENT
MASS PROPERTIES	SCA ENGINEERING AND INTEGRATION
SYSTEM/OPS DATA BOOKS	SCA SYSTEMS
INTEGRATED SCHEMATICS	SCA SYSTEMS SUPPORT

SUPPORT

ANCILLARY HARDWARE REQUIREMENTS	CHANGE INTEGRATION
COMMONALITY	OPERATIONAL REQUIREMENTS
CHANGE ASSESSMENT	DESIGN REVIEWS
CONFIGURATION MANAGEMENT	APPROACH & LANDING FLIGHT TEST

TABLE II

PANELS AND WORKING GROUPS

MANAGEMENT

- PERFORMANCE MANAGEMENT PANEL
- CONFIGURATION MANAGEMENT SYSTEMS PANEL
- MIC INTEGRATION PANEL
- INFORMATION MANAGEMENT SYSTEMS PANEL
- INTEGRATED LOGISTICS WORKING GROUP
- COST PER FLIGHT COMMITTEE
- SCHEDULE/LOGIC INTEGRATION WORKING GROUP

TECHNICAL

- SYSTEM INTERFACES PANEL
- FLIGHT PERFORMANCE PANEL
- LOADS/STRUCTURAL DYNAMICS PANEL
- INTEGRATED PROPULSION & FLUIDS PANEL
- FLIGHT CONTROL SYSTEM PANEL
  - ORBIT & ENTRY FCS SUBPANEL
  - GUIDANCE NAVIGATION & CONTROL SYSTEMS SUBPANEL
  - ASCENT FCS/STRUCTURES SUBPANEL
- INTEGRATED AVIONICS PANEL
- MECHANICAL SYSTEMS PANEL
  - SPACECRAFT SYSTEMS SUBPANEL
  - AIRCRAFT SYSTEMS SUBPANEL
- GROUND SYSTEMS INTEGRATION PANEL

TABLE III

ROCKWELL INTERNATIONAL'S SYSTEMS INTEGRATION TASKS

SHUTTLE PROGRAM DEFINITION AND REQUIREMENTS
SYSTEM INTERFACE CONTROL
MASS PROPERTIES
FLIGHT SYSTEM DESIGN PERFORMANCE
GROUND OPERATIONS ANALYSIS
COST PER FLIGHT
INTEGRATED SCHEMATICS
MASTER MEASUREMENT LIST
INTEGRATED VEHICLE ANALYSIS
INTEGRATED GROUND TEST
CONFIGURATION MANAGEMENT
PROGRAM SCHEDULE
MANAGEMENT INFORMATION CENTERS
COMMONALITY PROGRAM
LOGISTICS
QUALITY MANAGEMENT
SAFETY AND RELIABILITY
PREFLIGHT AND FLIGHT TEST SUPPORT
INTERFACE TOOLING
SYSTEMS MATERIALS AND PROCESS CONTROL
PAYLOAD INTERFACE
MISSION PLANNING
REPRESENTATIVES AT ELEMENT CONTRACTORS
SYSTEM LEVEL WORKING GROUPS
REPRESENTATIVES AT NASA CENTERS
SPECIAL STUDIES

TABLE IV

PRESENT ORBITER BASELINE
FUNCTION/CRITICALITY SUMMARY

DOOR	FUNCTION	TIMELINE FOR OPERATION	FAILURE	CRIT OF FUNCTION	REMARKS
STAR TRACKER	OPEN TO EXPOSE STAR TRACKER	OPEN ON ORBIT	TO OPEN	2	ABORT MISSION - NO IMU UPDATE
	CLOSE FOR THERMAL PROTECTION	CLOSE PRIOR TO DEORBIT BURN	TO CLOSE	1	AERO/THERMAL PROBLEM (BASELINE). INWARD OPENING DOOR MAY CHANGE TO CRIT 2
PAYLOAD BAY (OPEN & CLOSE TIME - CRITICAL FOR MISSIONS 3a, 3b)	OPEN TO EXPOSE PAYLOAD & RADIATOR	OPEN ON ORBIT	TO OPEN	2	ABORT MISSION
	CLOSE FOR STRUCTURAL INTEGRITY THERMAL PROTECTION	CLOSE PRIOR TO DEORBIT BURN	TO CLOSE	1	THERMAL/STRUCTURAL PROBLEM DURING ENTRY/DESCENT. RESCUE, OR CREW WORKAROUND
VENT (ONLY)	OPEN ASCENT/DESCENT PRESSURE EQUAL	CLOSE AT T-4	TO CLOSE	2	LAUNCH DELAY
		OPEN AT T+10	TO OPEN	2	REDUNDANCY PROVIDED
	CLOSE LIFTOFF - ACOUSTICAL ENTRY - THERMAL	CLOSE PRIOR TO DEORBIT BURN	TO CLOSE	1	THERMAL RPOBLEM - ENTRY. RESCUE, OR CREW WORKAROUND
		OPEN 70K FT	TO OPEN	2	REDUNDANCY PROVIDED
EXTERNAL TANK UMBILICAL	CLOSE FOR THERMAL PROTECTION	CLOSE AT MECO PLUS 12	TO CLOSE	1	AERO/THERMAL PROBLEM DURING ENTRY/DESCENT. RESCUE, OR CREW WORKAROUND

TABLE V

ORBITER OPERATIONAL MODES

Manual Direct

The crew manually controls the vehicle. No feed-back signals from vehicle-motion sensors are used for stabilization and control. The crew's command signal is applied to the appropriate force effector via the GNSC computer. Required compensation and logic for effector selection are accomplished within the GN&C computer. Vehicle-motion signals are displayed as required for crew operation. Automatic GSN commands are inhibited.

Manual Command Augmentation

The crew manually controls the vehicle as in manual direct. However, the crew's command is augmented by feedback signals from vehicle-motion sensors to improve response or augment stability, or both. Required compensation and logic for effector selection are accomplished within the GN&C computer. Vehicle-motion signals are displayed as required for the crew. Automatic G&N commands are inhibited.

Hold

The controlled vehicle parameter is held at the value existing when the hold function is engaged. This reference signal is not alterable by the automatic guidance system except by disengagement and reengagement of the hold function. The old function may be manually disengaged by moving the associated manual hand controller from the detent position. Reengagement is accomplished by returning the hand controller to the detent position.

Select

The controlled vehicle parameter converges to and holds the value selected or preselected by the crew.

Automatic

The guidance function provides automatic control of the vehicle. Manual command signals are inhibited and cannot act to sum with or override the automatic commands from the guidance system. Vehicle motions signals are displayed to permit crew monitoring of the G&N function. The crew has the option of manually engaging or disengaging the automatic function.

TABLE VI

ATMOSPHERIC REVITALIZATION SUBSYSTEM

FUNCTIONS

CARBON DIOXIDE, ODOR, AND WATER VAPOR CONTROL IN PRESSURIZED CABIN

CABIN PRESSURE MAINTENANCE AND CONTROL

CABIN ATMOSPHERE THERMAL CONTROL

CABIN AND AFT SECTION AVIONICS THERMAL CONTROL

ATMOSPHERIC REVITALIZATION FOR HABITABLE PAYLOADS (WHEN REQUIRED)

DESIGN/PERFORMANCE REQUIREMENTS

MISSION

- NOMINAL: 42 MAN-DAYS
- EXTRAVEHICULAR ACTIVITY: 3 TWO-MAN PERIODS
- CONTINGENCIES: 16-MAN DAYS OR 1 CABIN REPRESSURIZATION OR MAINTAIN PRESSURE WITH CABIN LEAK
- PERSONNEL (CREW/PASSENGERS)
- CABIN :DESIGN OPERATION, 3 to 10
:NORMAL, 3 to 7
:RESCUE, 6 to 10
- CABIN PRESSURE: 101,354 N/m^2 (14.7 psia)
- ATMOSPHERIC COMPOSITION: 21,374 N/m^2 (3.1 PSIA) OXYGEN: 79,980 N/m^2 (11.6 PSIA) NITROGEN

TABLE VII

SUBASSEMBLY RELIABILITY PREDICTION BREAKDOWN

	Total	I	II	III
Input Electronics	3.4313	1.2524	.1028	2.0761
Computer Interface Electronics	2.8360			2.8360
Output Electronics	3.3191			3.3191
Power Supply Electronics	3.7732			3.7732
Chassis Electronics	3.2708	.6032		2.6676
Digital Computer Unit	21.1272			21.1272
(%/1,000 hours) TOTAL	37.7576	1.8556	.1028	35.7992
(Hours) MTBF	2648	53,891	972,763	2765

I = Electronics for flight recording and/or maintenance data

II = Electronics for ground operation not required for mainstage

III = Electronics required for mainstage

TABLE VIII

TYPICAL CONTROLLER ELECTRONICS CARD FAILURE RATES

Nomenclature	Quantity	Failure Rate (%/1000 hr.)	Percent of Controller Failure Rate
Output electronics	1	0.597	1.7
Power supply	1	0.455	1.3
Input electronics	1	0.310	0.88
Computer interface electronics	2	0.208	0.59

TABLE IX

CONTROLLER RELIABILITY PREDICTION

Assembly	Failure Rate % per 1000 hrs.
Input Electronics	3.96
Interface Electronics	2.87
Output Electronics	3.32
Power Supply and Chassis	2.30
DCU	21.18
Controller	33.63
	3,000 hours MTFF

TABLE X

SRB BASELINE REVIEW
PARACHUTE DESIGN FACTORS

$$D.\,F. = \frac{(j)\,(c)}{(u)\,(o)(e)(k)\,(r)\,(m)}$$

APPLICATION	SAFETY FACTOR	STRENGTH REDUCING FACTOR $\left(A_p = \frac{c}{u\text{-}o\text{-}e\text{-}k\text{-}r\text{-}m}\right)$							OVERALL DESIGN FACTOR (D. F. = $j \cdot A_p$)
	j	CON-FLUENCE ANGLE FACTOR (c)	JOINT EFFICIENCY FACTOR (u)	WATER FACTOR (o)	ABRASION FACTOR (e)	FATIGUE FACTOR (k)	REUSE FACTOR (r)	CLUSTER FACTOR (m)	
DROGUE	1.91	1.025	.8	.95	.95	.95	.95	1	3.00
MAIN	1.54	1.055	.8	.95	.95	.95	.95	.833	3.00

PRECEDENTS

	j	D.F.	
PERSONNEL		3	(F-111 2.54)
"AEROSPACE"	1.9	2.9	(APOLLO 1.9, VIKING 2.08)
CARGO (NO REUSE)	1.5	2.2	
CARGO (REUSE)	1.5	3.0	

TABLE XI

SHUTTLE SYSTEM CONCERNS

1 - FIRE/TOXICITY REQUIREMENTS	17 - SSME UNSCHEDULED SHUTDOWN DURING BOOST
2 - HAZARDOUS GAS DETECTION	18 - HYDROGEN INGESTION IN THE ORBITER DURING BOOST
3 - PROPELLANT LOADING HAZARDS ON PAD	19 - SSME FUEL AND OXIDIZER LEAKAGE
4 - EMERGENCY DRAIN OF EXTERNAL TANK	20 - SSME HEAT EXCHANGER LEAKAGE
5 - EMERGENCY INGRESS/EGRESS ON GROUND	21 - EFFECTS OF ET ABLATIVE PRODUCTS ON ORBITER TPS
6 - DAMAGE TO ORBITER FROM ET ICING	22 - LH_2 & LO_2 HAZARDS AT ET/ORBITER SEPARATION
7 - PREMATURE SEPARATION OF ET TO INTERTANK GROUND UMBILICAL	23 - ORBITER/ET SEPARATION WITH FAILED RCS
8 - ET TPS/LO_2 INCOMPATIBILITY DURING PRELAUNCH & EARLY BOOST PHASE	24 - POST SEPARATION IMPACTS OF ORBITER BY ET
9 - SRB IGNITION OVERPRESSURE	25 - PUBLIC HAZARD FROM SRB IMPACT
10 - LATE IGNITION OF ONE SRB	26 - PUBLIC HAZARD FROM ET IMPACT
11 - SHUTTLE COLLISION WITH TOWER ON LIFTOFF	27 - INTACT ABORT CAPABILITY
12 - SRB SEPARATION SYSTEM PLUME IMPINGEMENT	28 - CONTINGENCY ABORT CAPABILITY
13 - FAILURE OF FORE OR AFT SEPARATION MOTOR	29 - EMERGENCY ESCAPE IN FLIGHT
14 - POGO	30 - CREW RESCUE FROM ORBIT
15 - EXCESSIVE ET AERO HEATING	31 - HYDROGEN & OXYGEN RELIEF FROM A CRYOGENIC PAYLOAD
16 - FIRE IN ET INTERTANK AREA BELOW 80,000 FT	

PUBLISHED IN "SHUTTLE SYSTEM PDR-SAFETY ANALYSIS REPORT," SD 75-SH-0064 28 FEBRUARY 1975.

TABLE XII

FLUID HAZARDS VS MISSION PROFILE

ORBITER FLUID	MISSION PROFILE					
	PRE-LAUNCH	ASCENT	ORBIT	REENTRY	LANDING	SAFING & MAINTENANCE
AMMONIA (NH_3)	F&T	(IP)	-	F	F	F&T
HYDROGEN (H_2)	F	(IP)	-	F	F	F
HYDRAZINE (N_2H_4)	F&T	(IP)	-	F	F	F&T
MONOMETHYLHYDRAZINE (5606)	F&T	(IP)	-	F	F	F
*NITROGEN TETROXIDE (N_2O_4) AND/OR OXYGEN (O_2)	F&T	(IP)	-	F	F	F&T

LEGEND: F = FIRE HAZARD

T = TOXIC HAZARD

IP = PAD INERT GAS (GN_2) PURGE

*NITROGEN TEXTROXIDE & O_2 CAN BE FIRE HAZARDS IN COMBINATION WITH FUELS WHEN AIR IS NOT PRESENT

- NECESSITY FOR GROUND DETECTION AFFIRMED - CLEARS FOR ASCENTS
- ENTRY HAZARD POTENTIAL BEING EVALUATED - ON-BOARD DETECTION (TBD)

TABLE XIII

ALT MISSION OBJECTIVES DOCUMENT

ORBITER FREE FLIGHT TESTS

TEST NO.	ORBITER GW/CG	OBJECTIVES	REMARKS
1	OPT/OPT	SIMULATED APPROACH AT ALTITUDE	TAILCONE ON
2	OPT/OPT	STAB AND CONT AT NEAR V_{MSO}, V_{APP}	TAILCONE OFF
3	INTERMEDIATE I	STAB AND CONT AT V_{MAX} ALLOWABLE	MONITOR AUTO COMMANDS
4	OFT/OFT	STAB AND CONT AT V_{MSO}, V_{APP}	MONITOR AUTO COMMANDS
5	OFT/OFT	BANKS, SIDESLIPS AT V_{MSO}, V_{APP}	AUTO ROLLOUT AT 30 kt
6	OFT/OFT	MANUALLY FLY AUTO COMMANDS	SWITCH IN/OUT OF AUTOMODE IN FLIGHT AUTO ROLLOUT AT ABOUT 90 kt
7	OFT/OFT	AUTOTAEM/AUTOLOAD DEMONSTRATION	
8	OFT/OFT	OFFSET AUTOTAEM/AUTOLOAD DEMONSTRATION	$\approx 30^{\circ}$ OFFSET SEPARATION
9	IMMEDIATE II	STAB AND CONT AT V_{MAX} ALLOWABLE	MONITOR AUTO COMMANDS
10	NVY/AFT	STAB AND CONT AT NEAR V_{MSO}, V_{APP}	MONITOR AUTO COMMANDS
11	NVY/AFT	$\approx 2_g$ TURN, BANKS, SIDESLIPS AT V_{APP}	$\approx 50^{\circ}$ OFFSET SEPARATION

OPT - OPTIMUM CONFIGURATION FOR APPROACH AND LANDING
OFT - FIRST ORBITAL FLIGHT TEST CONFIGURATION
NVY/AFT - AN OFT/OPERATIONAL CONFIGURATION MORE EXTREME THAN OFT
V_{MSO} - MINIMUM SAFE OPERATING VELOCITY
V_{APP} - NOMINAL APPROACH VELOCITY
V_{MAX} - MAXIMUM VELOCITY

JSC TPS MANAGEMENT ORGANIZATION

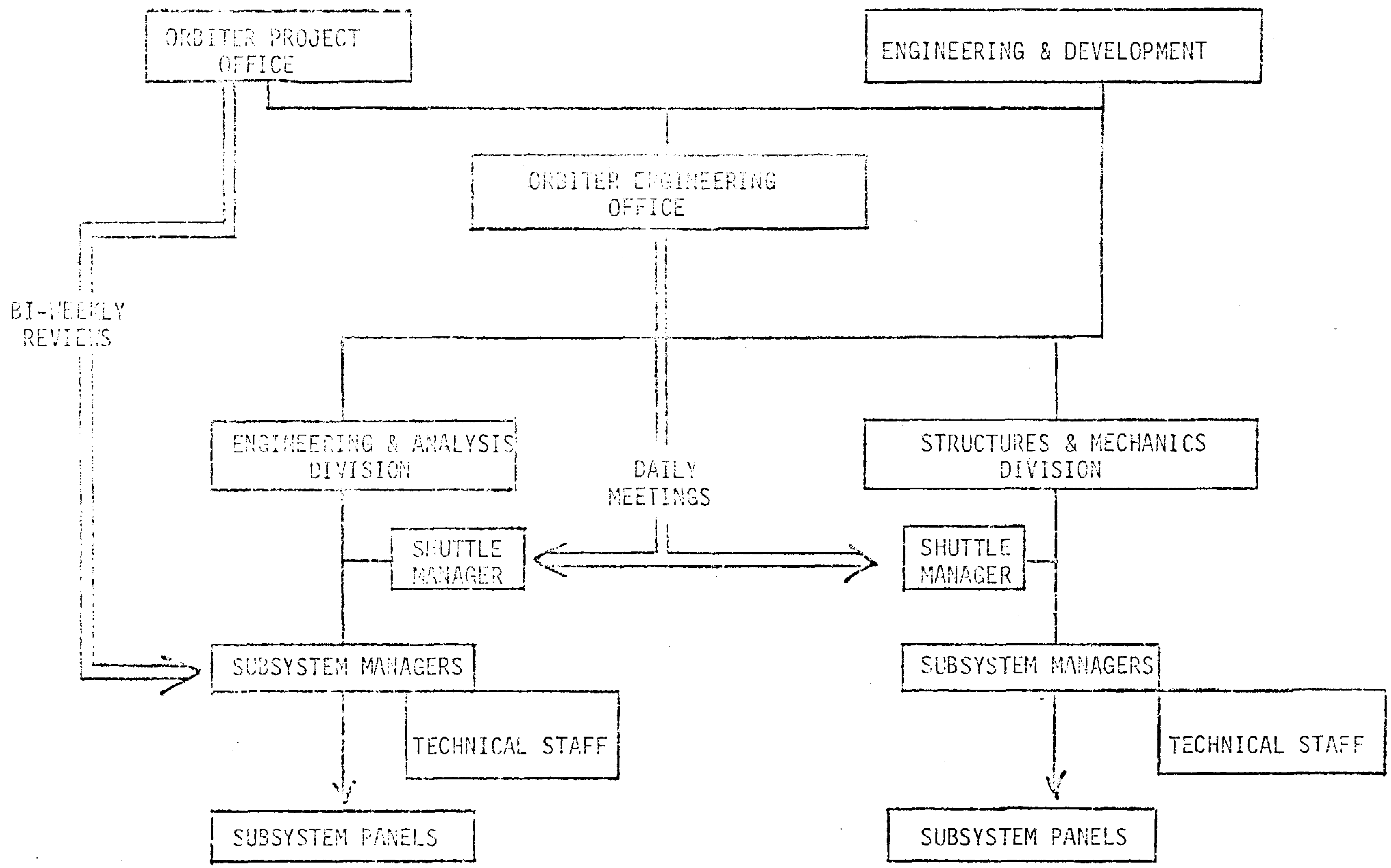


Figure 1

JSC TPS MANAGEMENT ORGANIZATION DETAIL

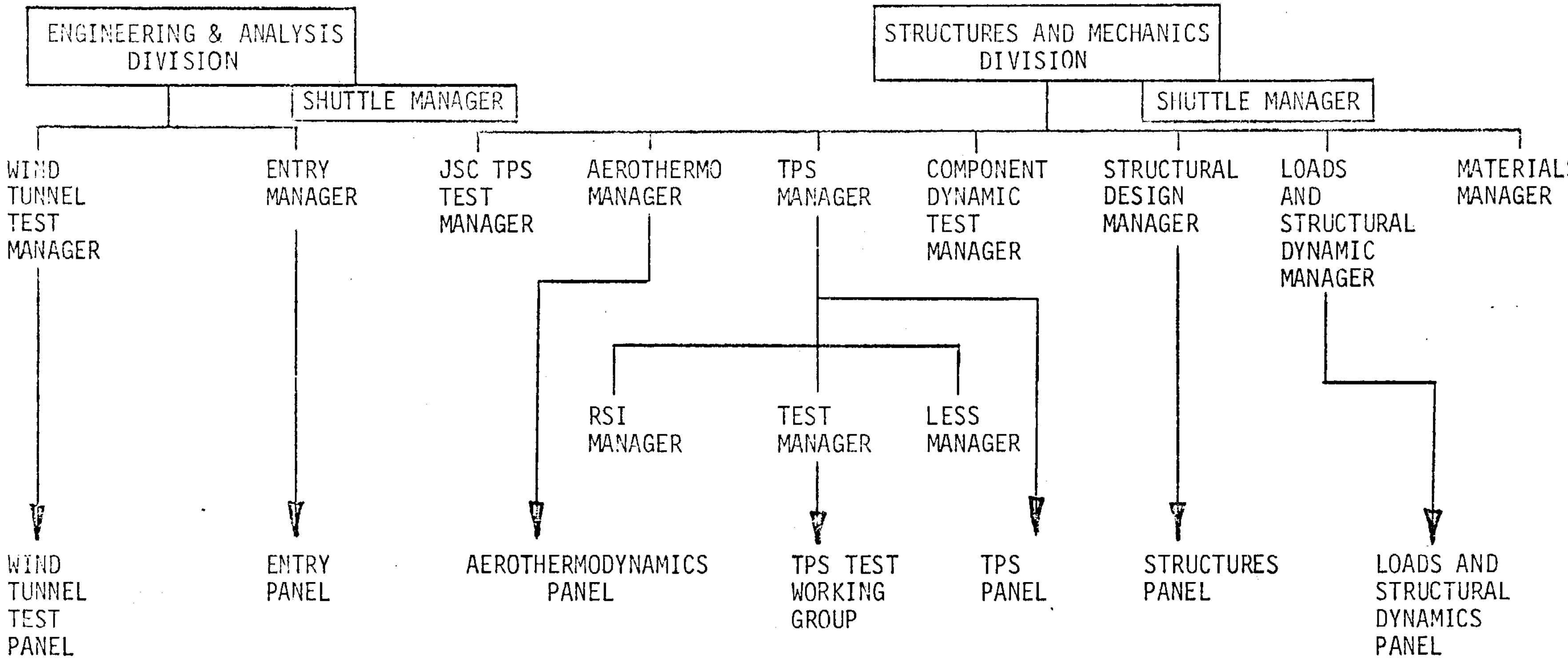
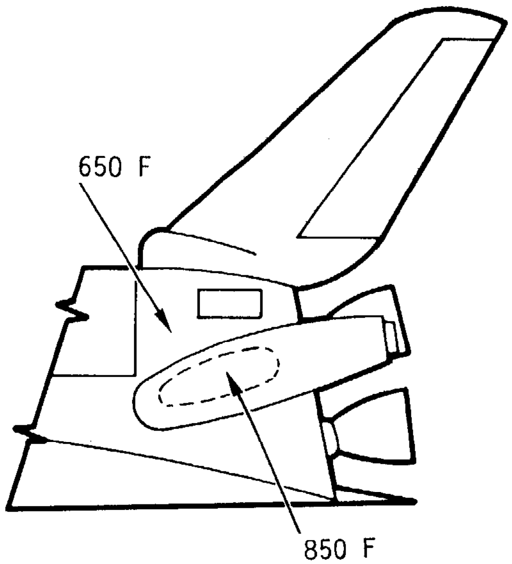


Figure 2

T-0 UMBILICAL DOOR
THERMAL HISTORY



- FUNCTION
  - OPEN DURING PRELAUNCH OPERATIONS
  - CLOSED AT T = +1 SEC, REMAINS CLOSED DURING FLIGHT
- THERMAL CRITERIA
  - ASCENT - CLOSED AT LIFTOFF TO PROTECT COMPONENTS FROM HIGH SRB PLUME RADIATION HEATING
  - ENTRY - CLOSED, SEALED, AND SMOOTH OML TO PROVIDE THERMAL PROTECTION FOR STRUCTURE AND COMPONENTS DURING ENTRY HEATING

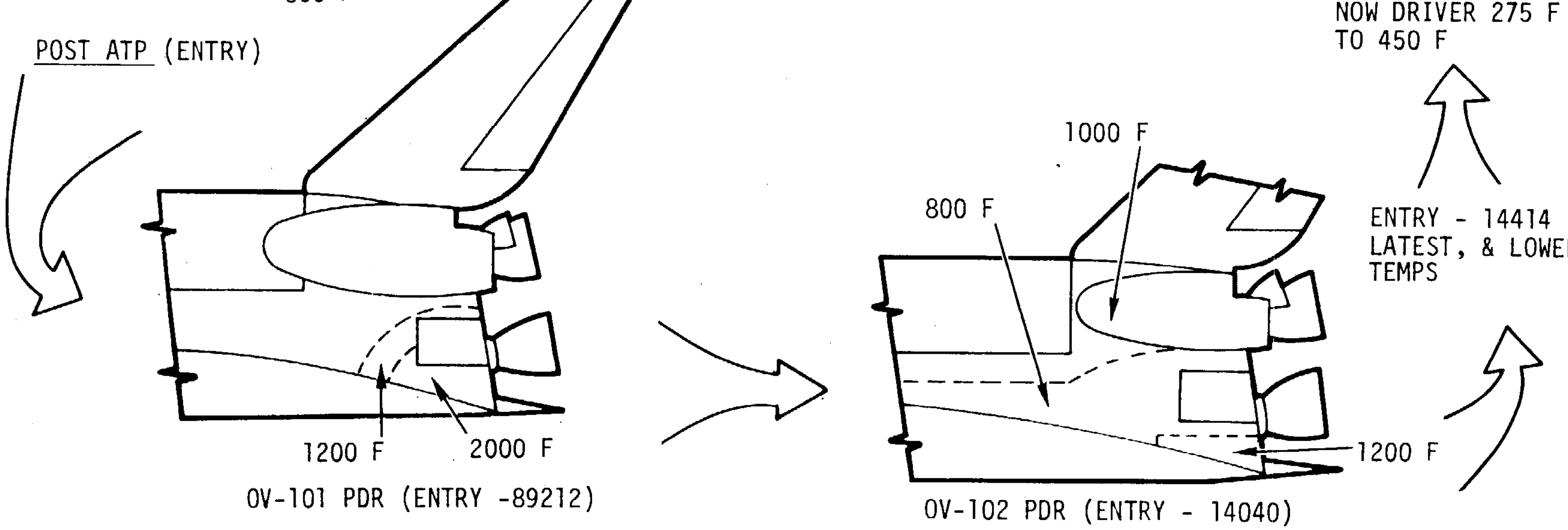


Figure 3

T-0 LAUNCH UMBILICAL PLATE

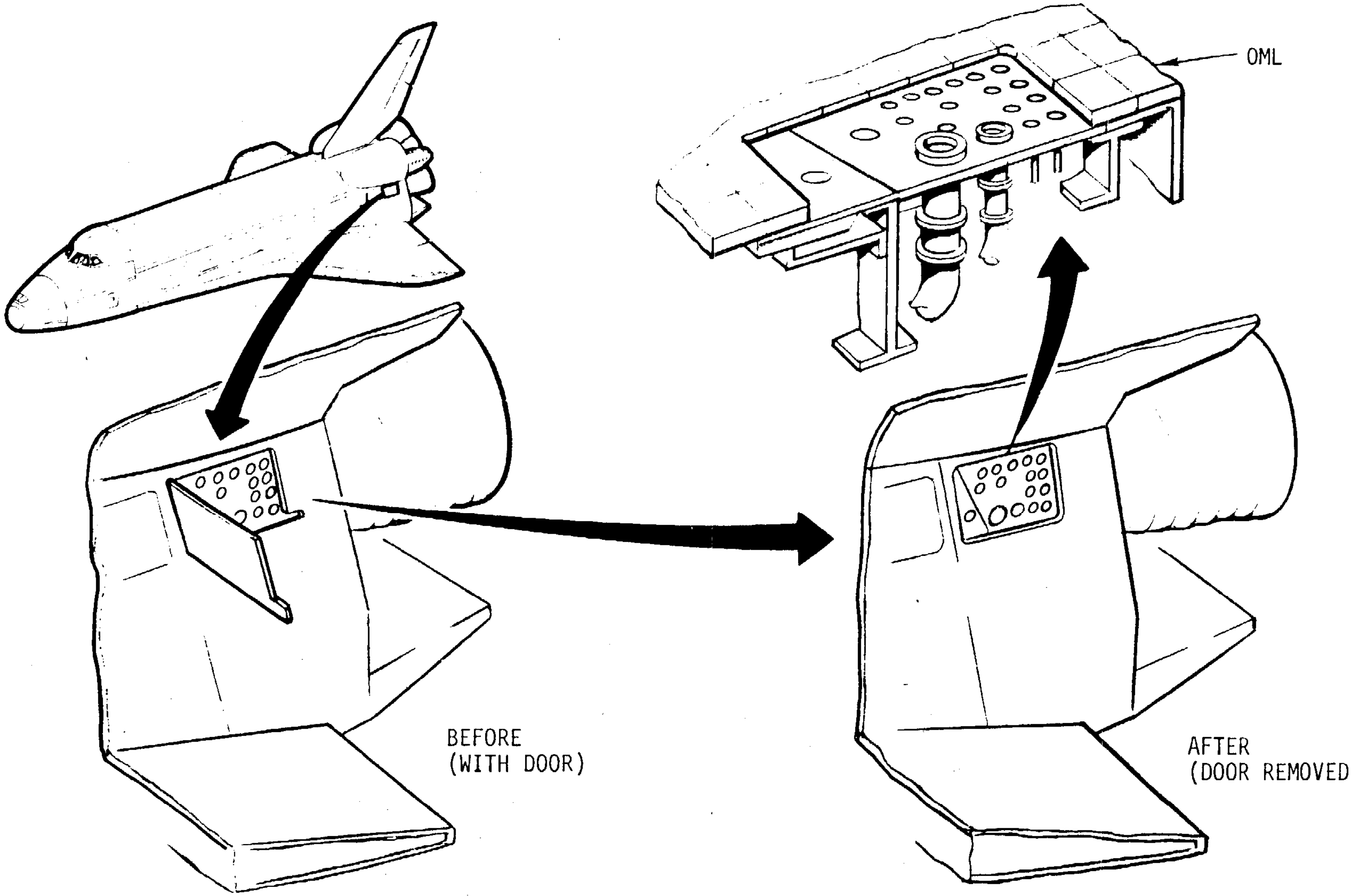


Figure 4

FORWARD RCS

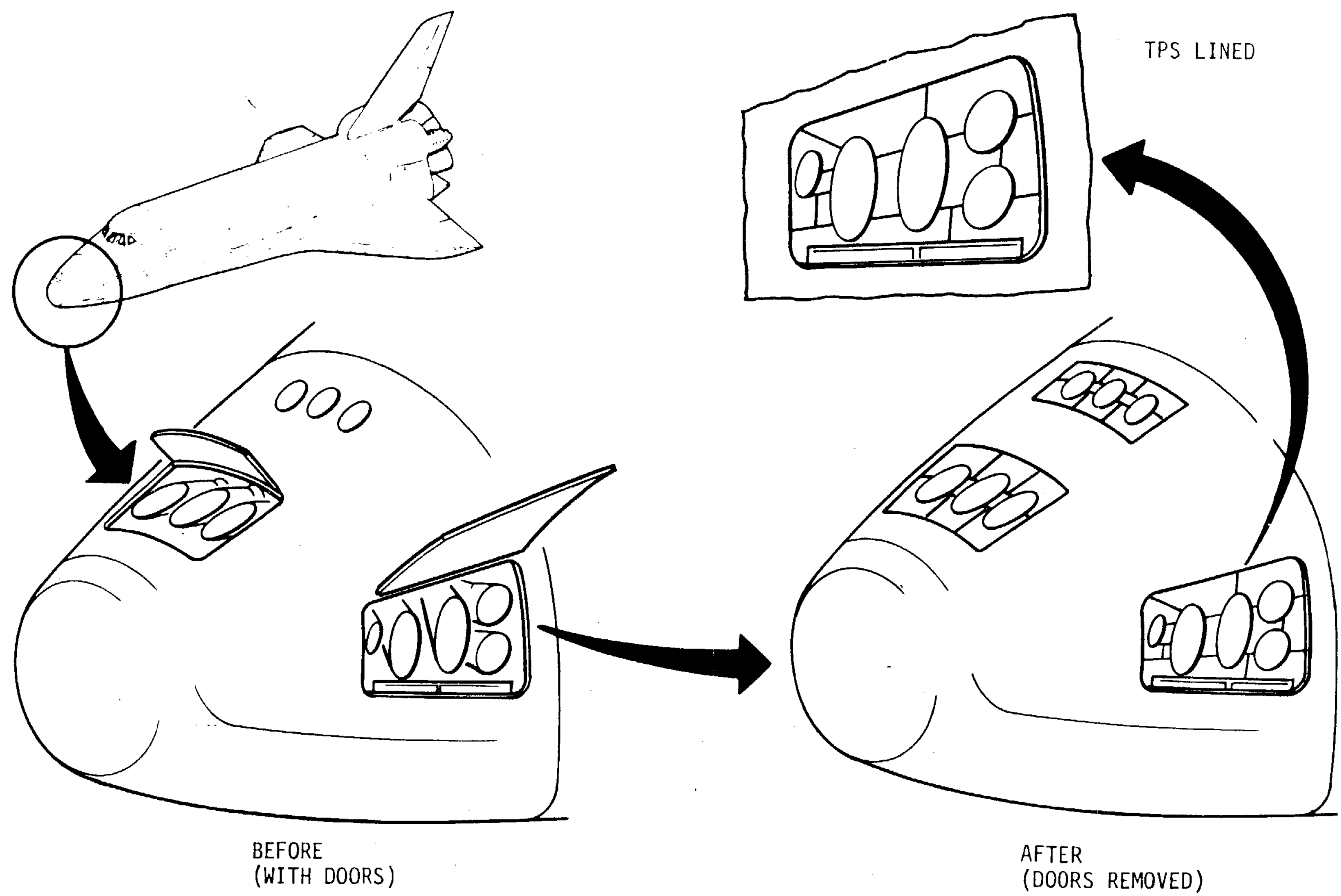
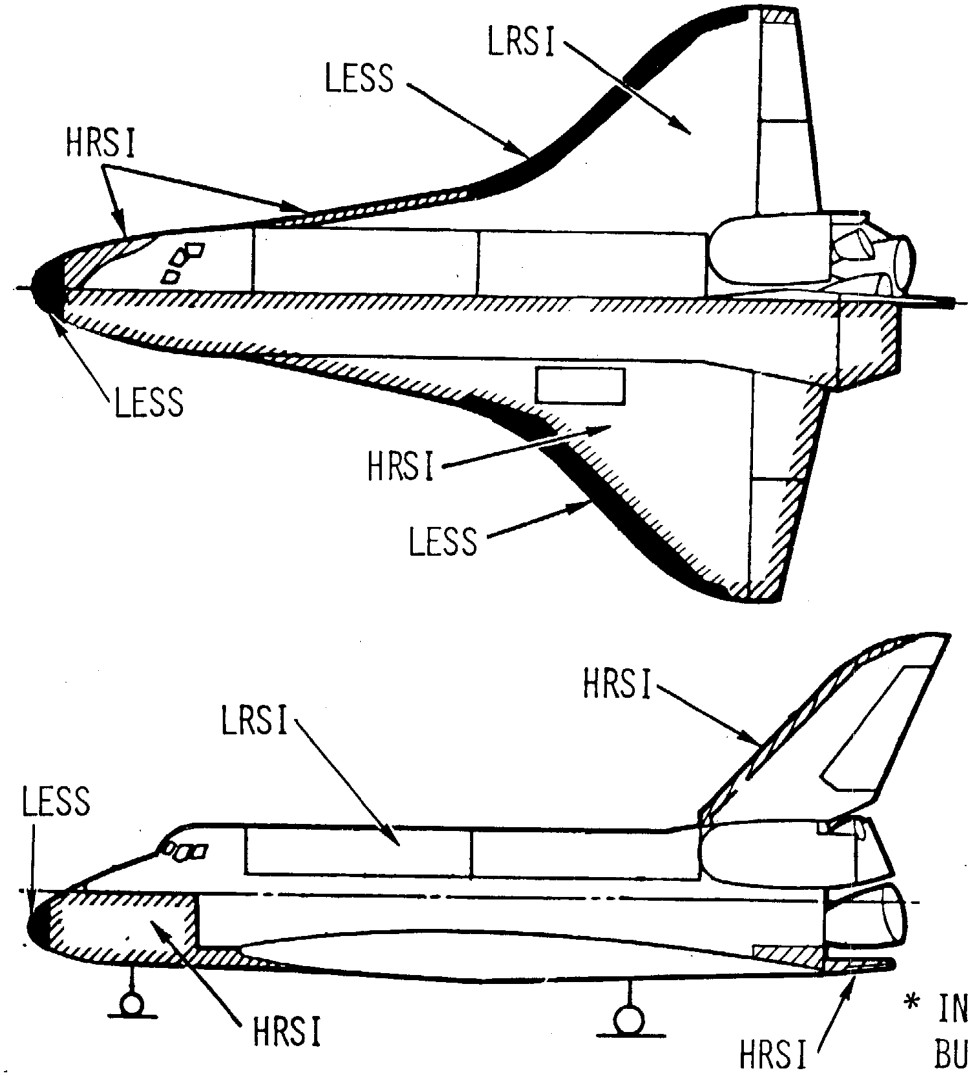


Figure 5

THERMAL PROTECTION SYSTEM DESCRIPTION



140 C CONFIGURATION
TRAJECTORY NOM NO. 14414.1 (UNFAIRED)

TPS	AREA	WT (LB)
LRSI	6,317	2,966
HRSI	5,134	7,951
RCC	409	3,023
TOTAL	11,860	15,984 *

LRSI - COATED SILICA

HRSI - COATED SILICA

RCC - REINFORCED CARBON-CARBON

SIP - NOMEX "E" FELT

* INCLUDES 2044-LB THERMAL SEALS, BULK INS,

Figure 6

LRSI AND HRSI JOINT

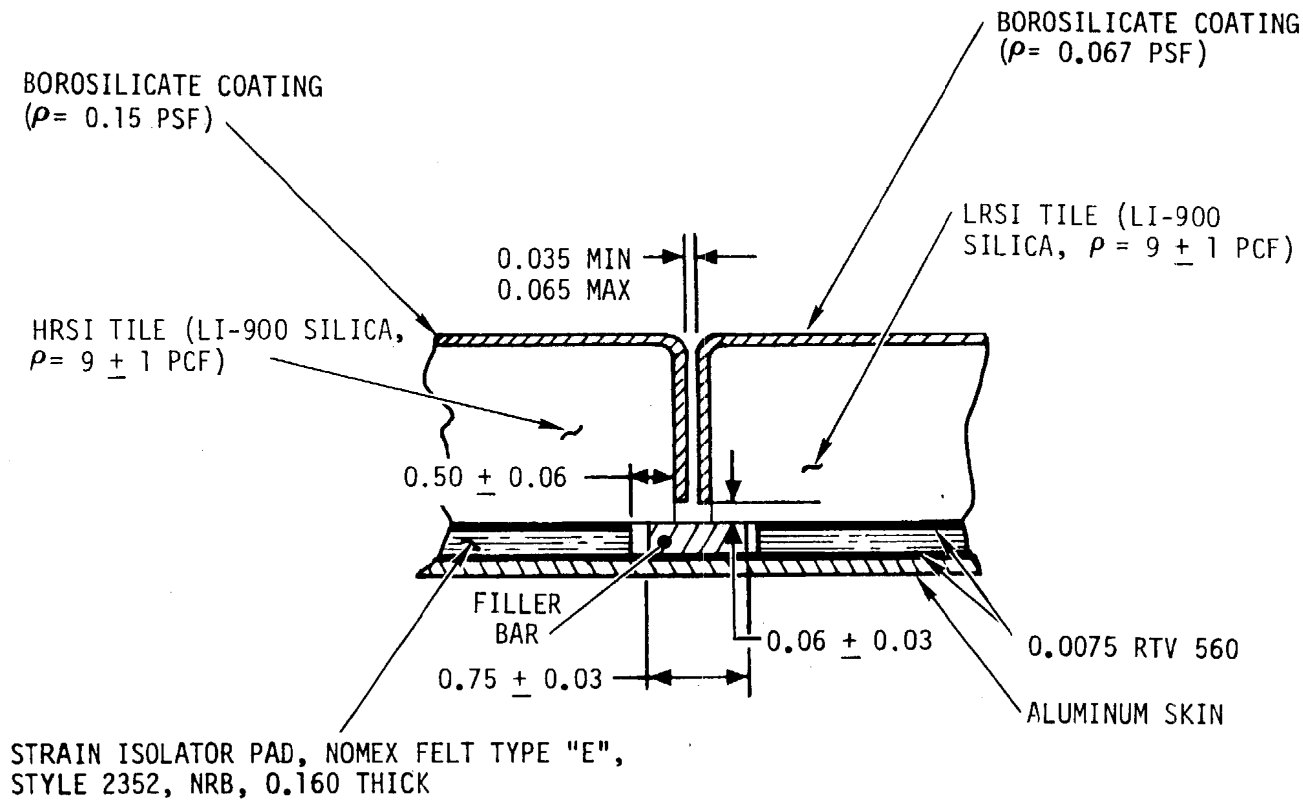


Figure 7

LEADING EDGE STRUCTURAL SUBSYSTEM

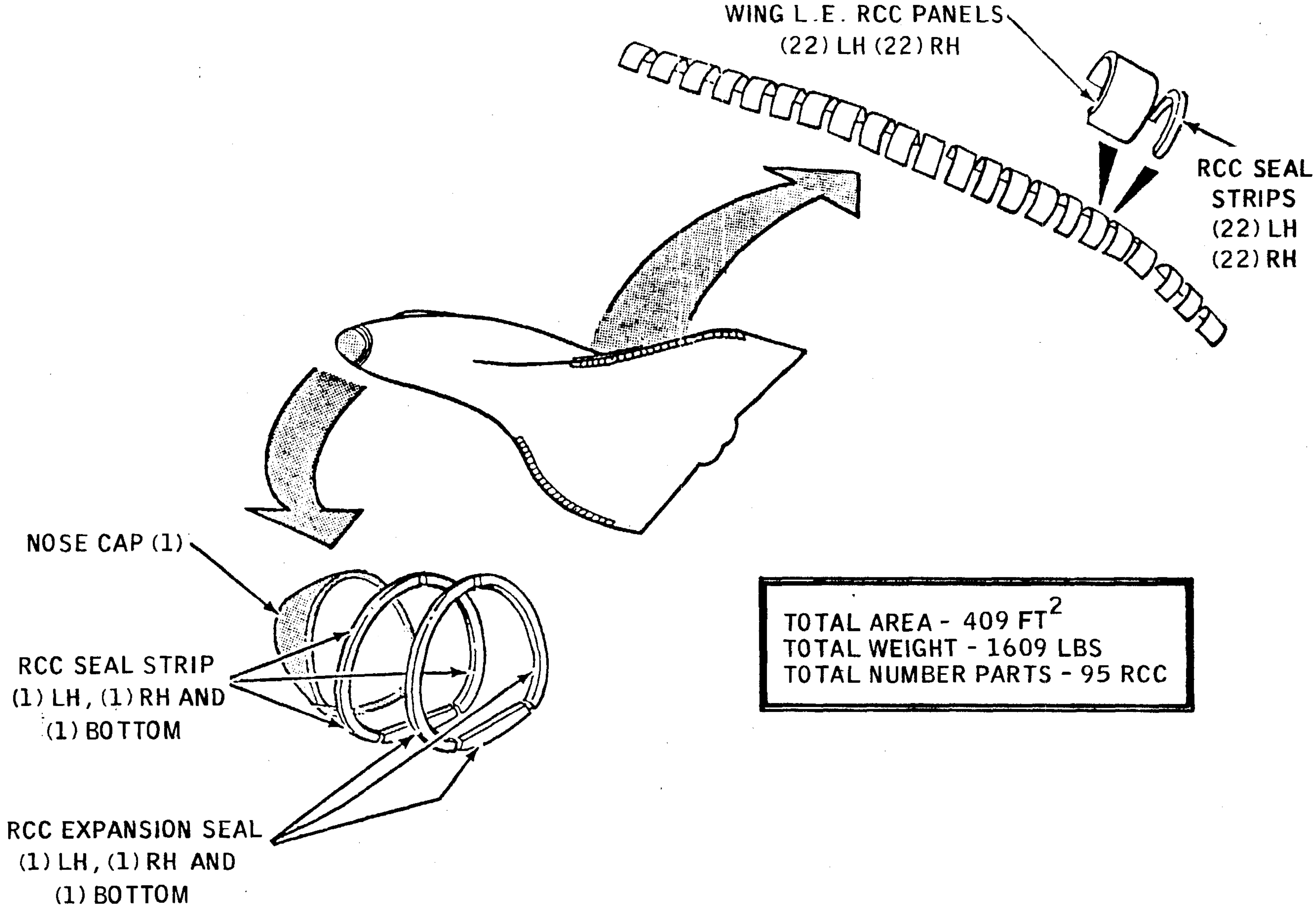


Figure 8

VERTICAL TAIL
SPACE SHUTTLE ORBITER

INCONEL SANDWICH DESIGN

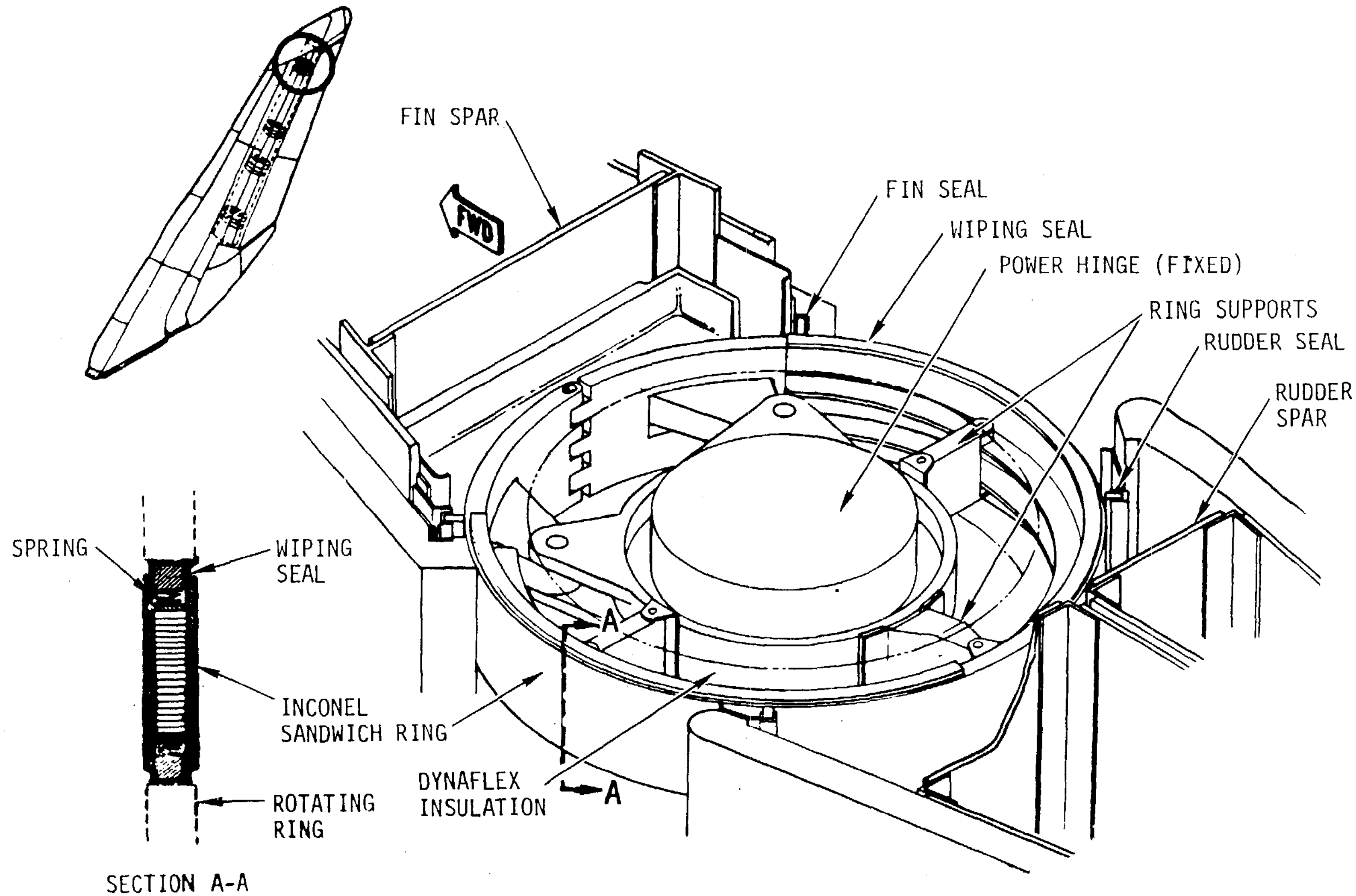


Figure 9

THERMAL SEAL CONCEPTS

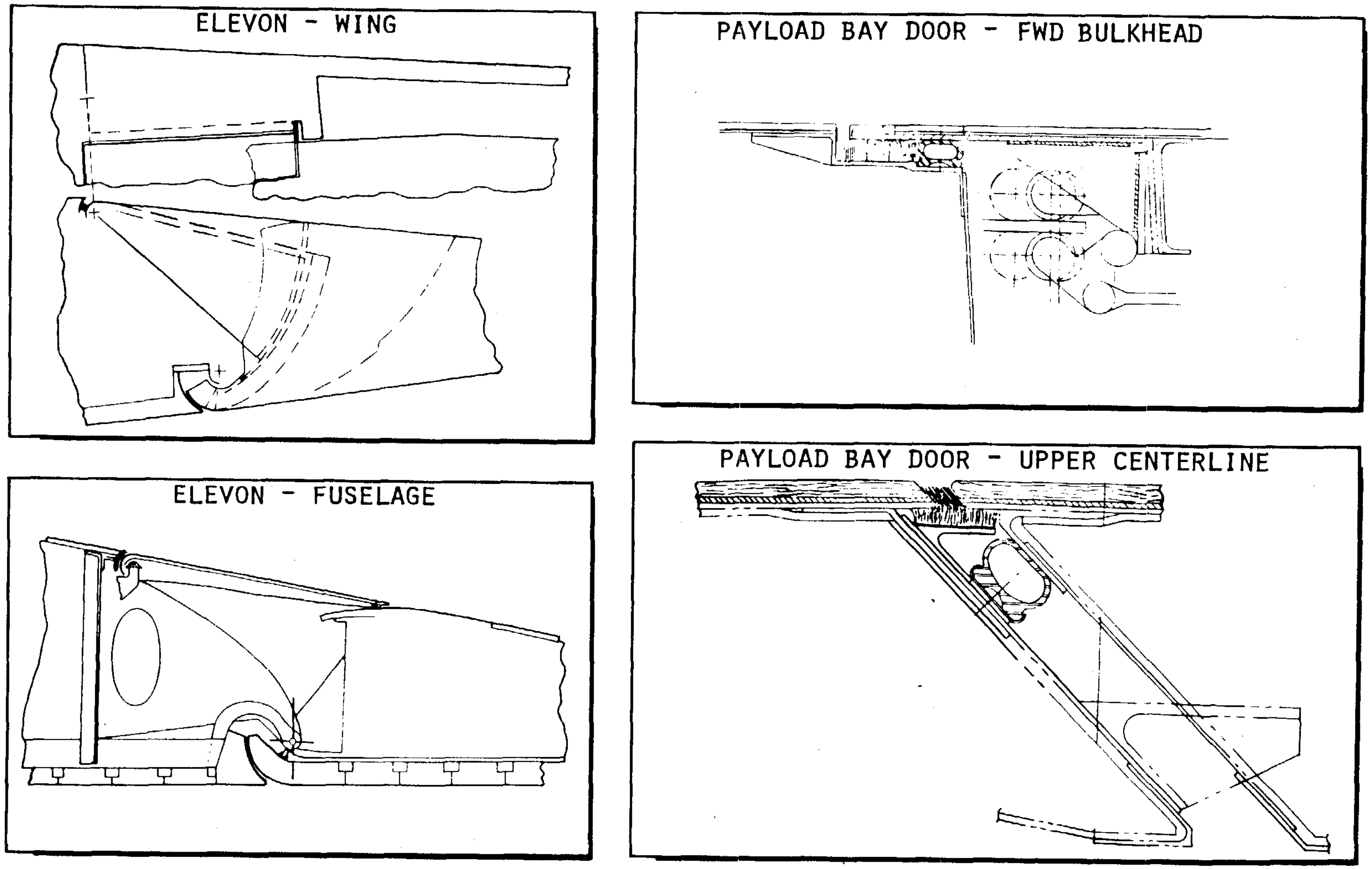
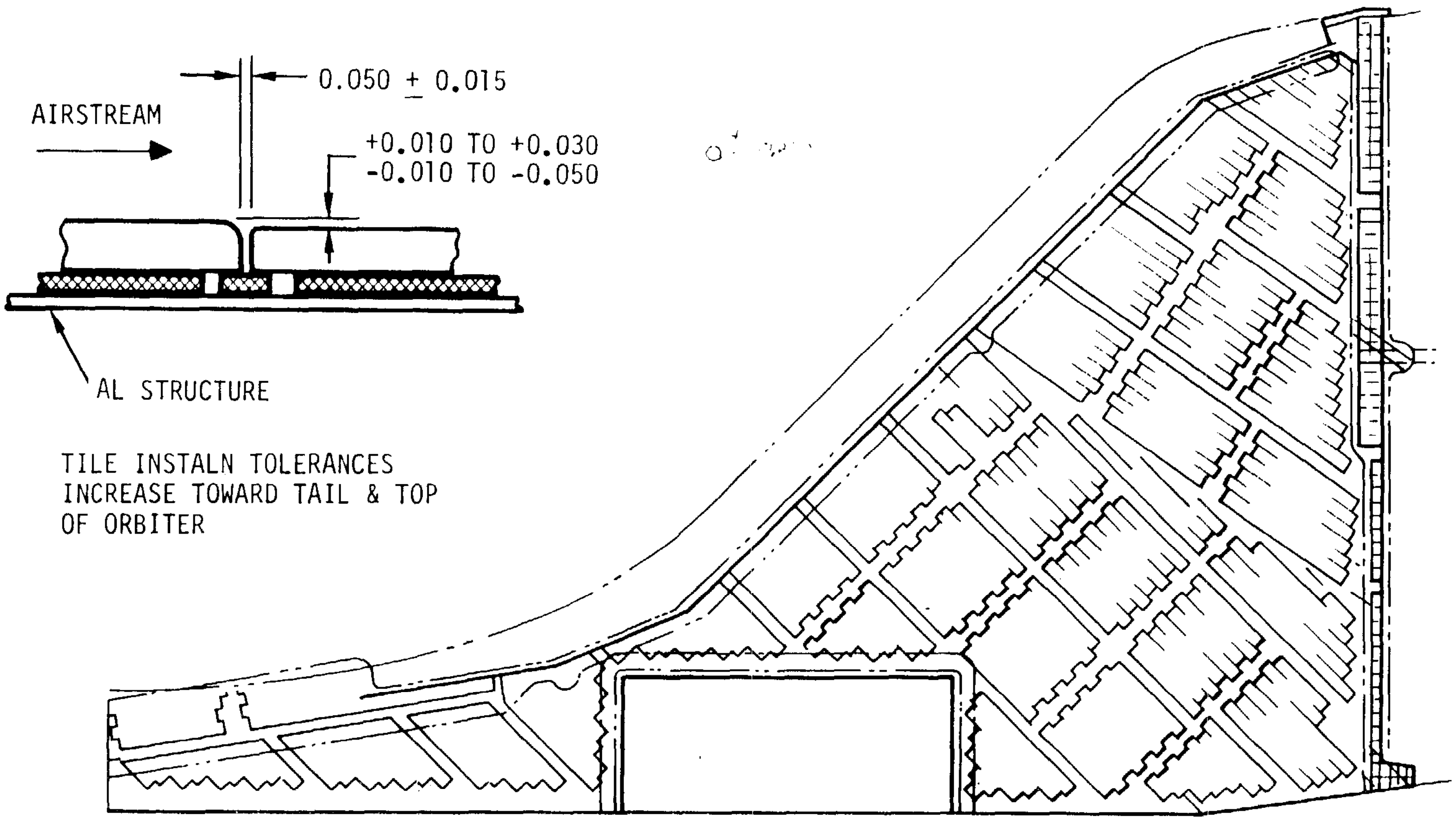


Figure 10

TPS GEOMETRY TOLERANCE

TILE ORIENTATION & SIZE

- MAX ANGLE BETWEEN LOCAL FLOW & TILE GAPS (NO TILE GAP PARALLEL TO FLOW)
- TILE PLANFORM SIZE - SET BY STRESS REQMTS



WING LOWER SURFACE TILE GAP ORIENTATION

Figure 11

ORBITER AVIONICS SUBSYSTEM

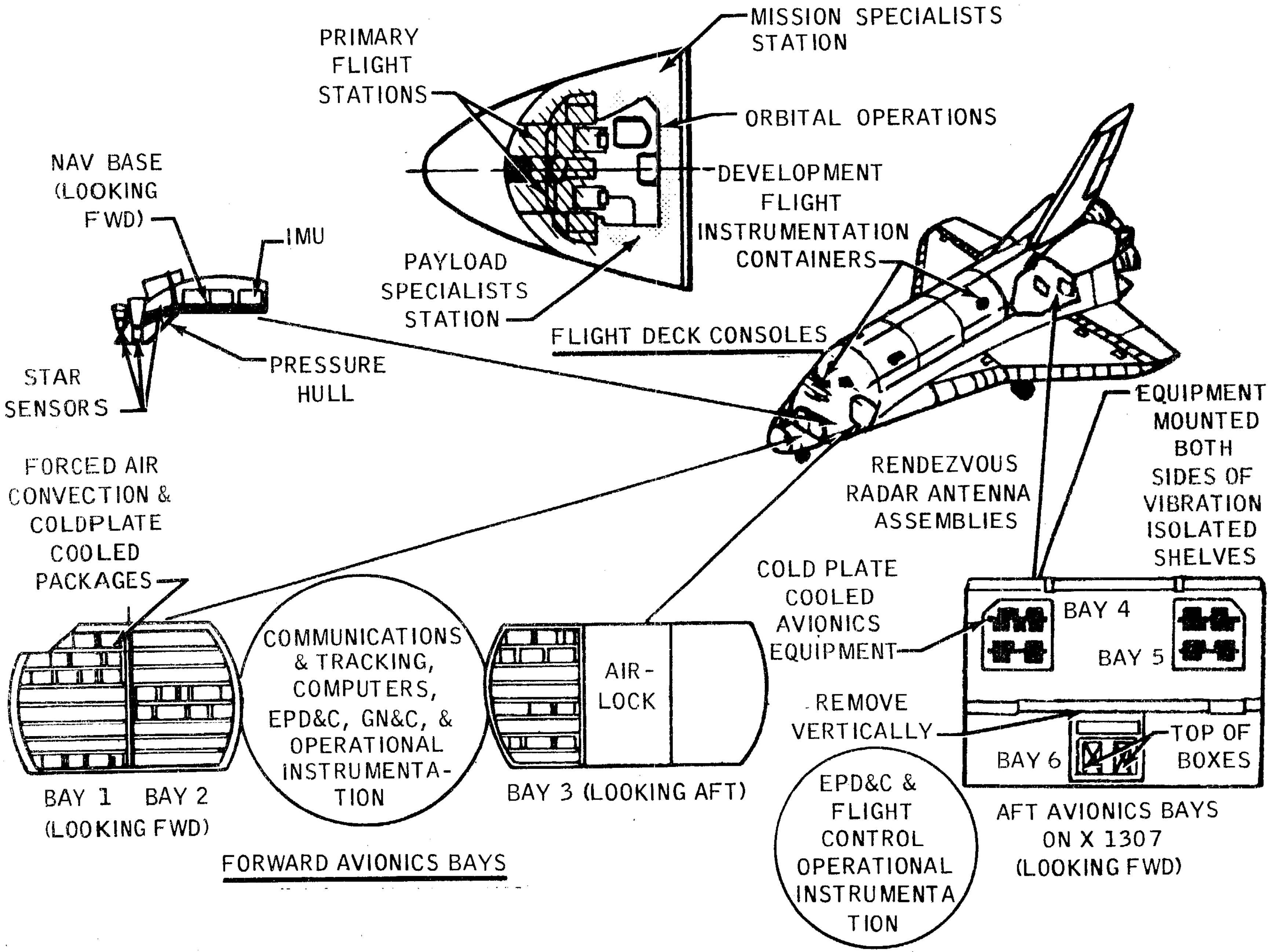
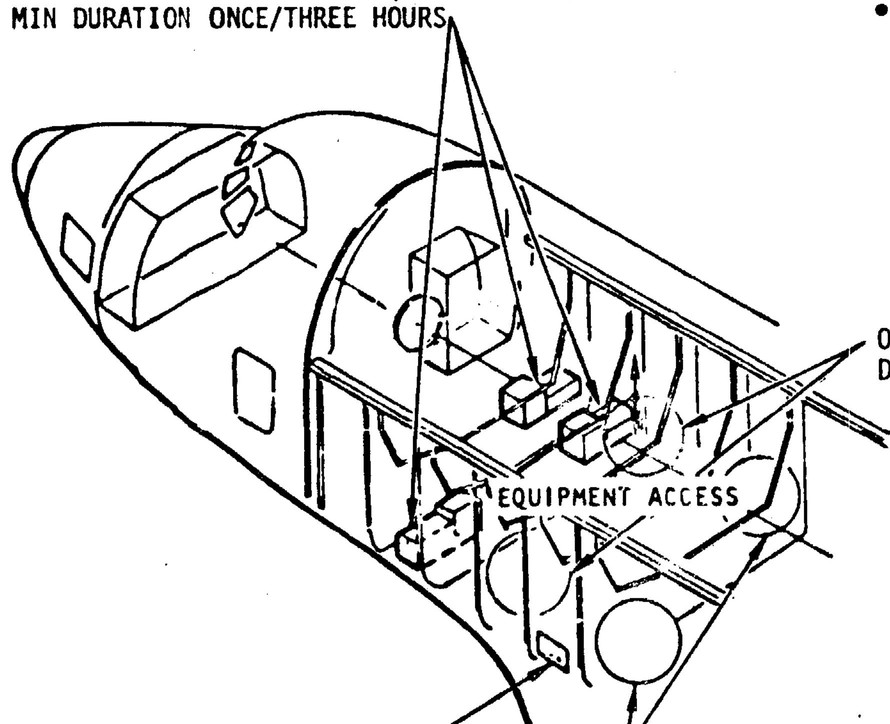


Figure 12

ORBITER ELECTRICAL POWER SUBSYSTEM

- POWER REACTANT STORAGE/DISTRIBUTION SUBSYSTEM
- POWER GENERATION SUBSYSTEM



FCP SUBSYSTEM

- 14-KW CONTINUOUS/24-KW PEAK
- 27.5 TO 32.5 VDC

REACTANT STORAGE

- 1530-KWH MISSION ENERGY
- 264-KWH ABORT/SURVIVAL ENERGY
- 112 LB O_2 FOR ECLSS
- 92 LB H_2/TANK } TOTAL LOADED QUANTITY
- 781 LB O_2/TANK } TOTAL LOADED QUANTITY

Figure 13

TUBING

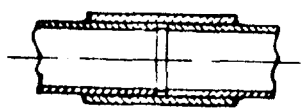
- **TUBING**

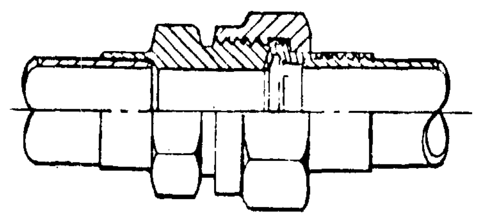
MATERIAL	21-6-9 (MB0160-035)
QUANTITY	1570 FEET
SIZES O. D.	1/4", 3/8", 1/2", AND 5/8"
WALL SIZE	.016

- **JOINING METHODS**

PERMANENT	SEPARABLE FITTINGS	
BRAZE	DYNATUBE	USED ON FC40 COOLANT, H_2O, O_2, & H_2

INDUCTION BRAZE
(TUBE END TO FITTING)





BRAZE
ROCKWELL
INTERNATIONAL
APOLLO

DYNATUBE-RESISTOFLEX

Figure 14

ELECTRICAL POWER SYSTEM

INSULATION

- LINE INSULATION

  1. TUBING RUNS WILL BE INSULATED

     USING POLYURETHANE FOAM 1/2" THICK, <u>ON PRSD ONLY</u>

  2. LINE HEATERS WILL REQUIRE WRAP WITH ALUMINIZED

     KAPTON TAPE, SPECIFICATION (TBD), <u>ON PGS ONLY</u>

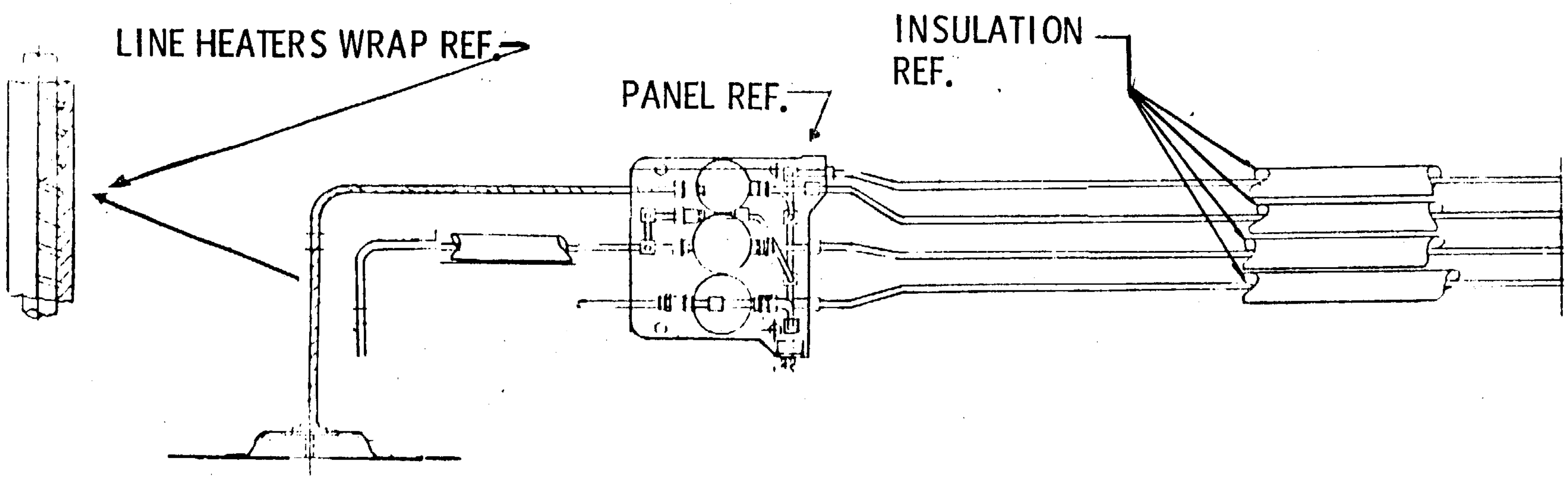


Figure. 15

AERO SURFACE CONTROLS

- o PDR REVIEW INCLUDED:
  - o LAYOUTS
  - o ENVELOPE DRAWINGS
  - o ICD'S
  - o PROCUREMENT SPECIFICATIONS
  - o DESIGN REQUIREMENTS
  - o INSTALLATION/RIGGING
  - o GSE
  - o VERIFICATION PLANS
  - o SCHEDULES
- o TYPICAL ALL OPERATIONAL VEHICLES

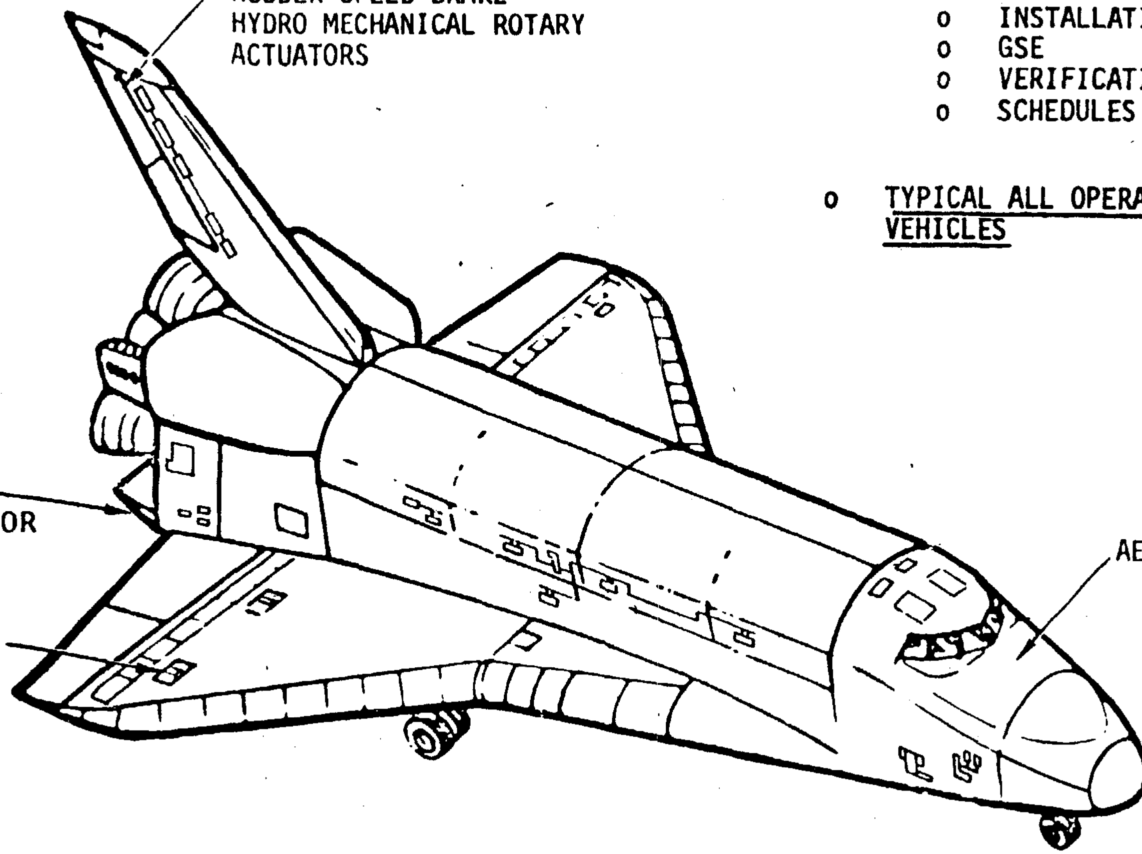


Figure 16

ORBITER-ET SEPARATION SUBSYSTEM

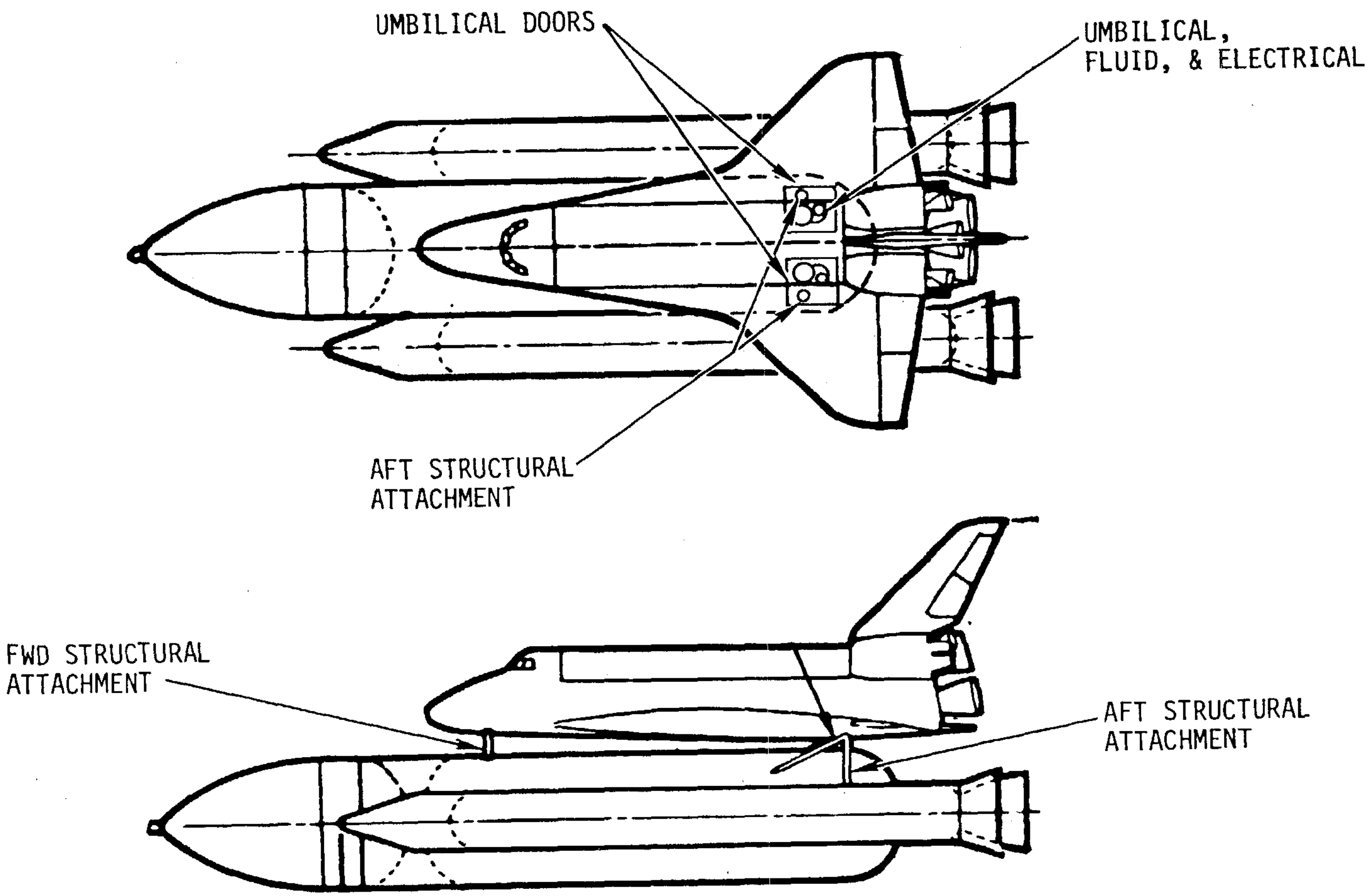


Figure 17

ORBITER/ET AFT SEPARATION

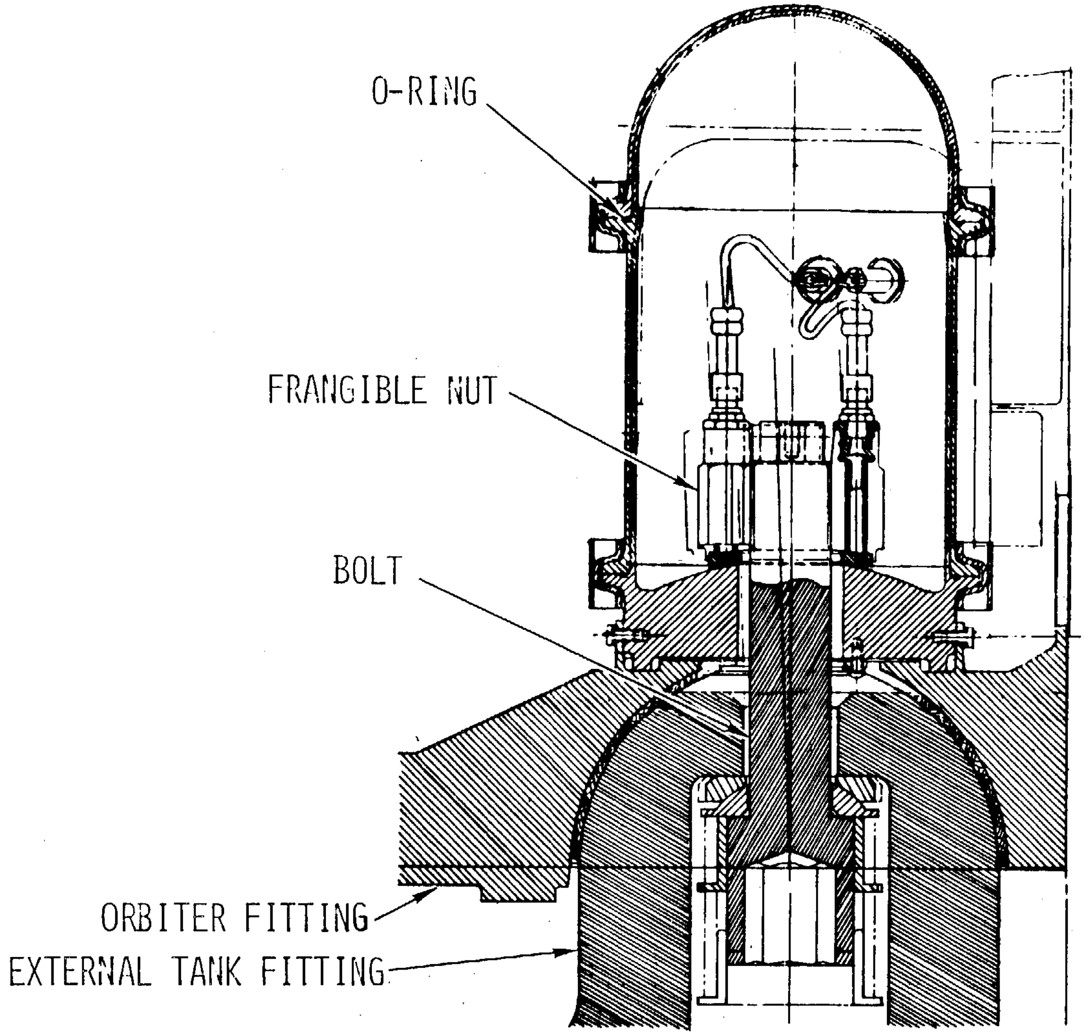


Figure 18

CONTROLLER ORGANIZATION AND REDUNDANCY

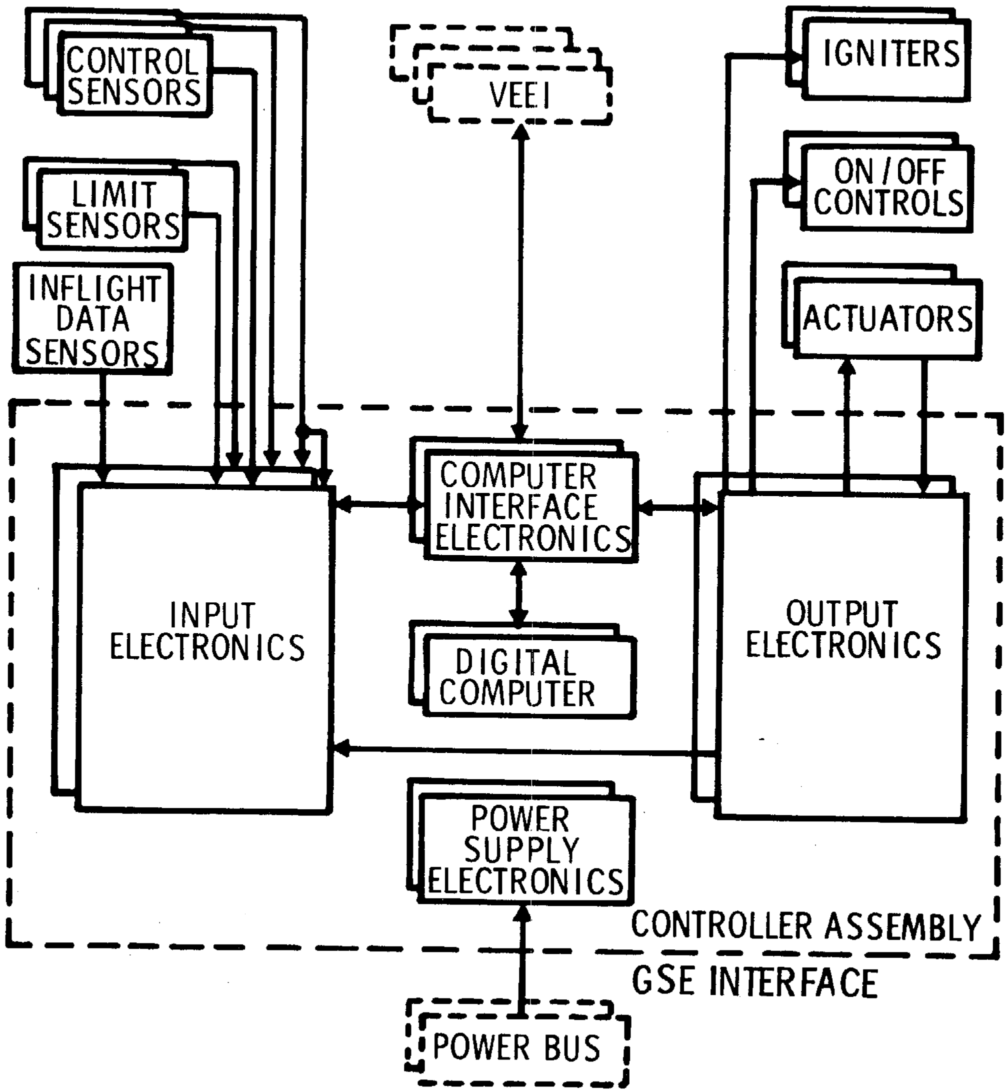


Figure 19

CONTROLLER ELECTRONICS ARRANGEMENT

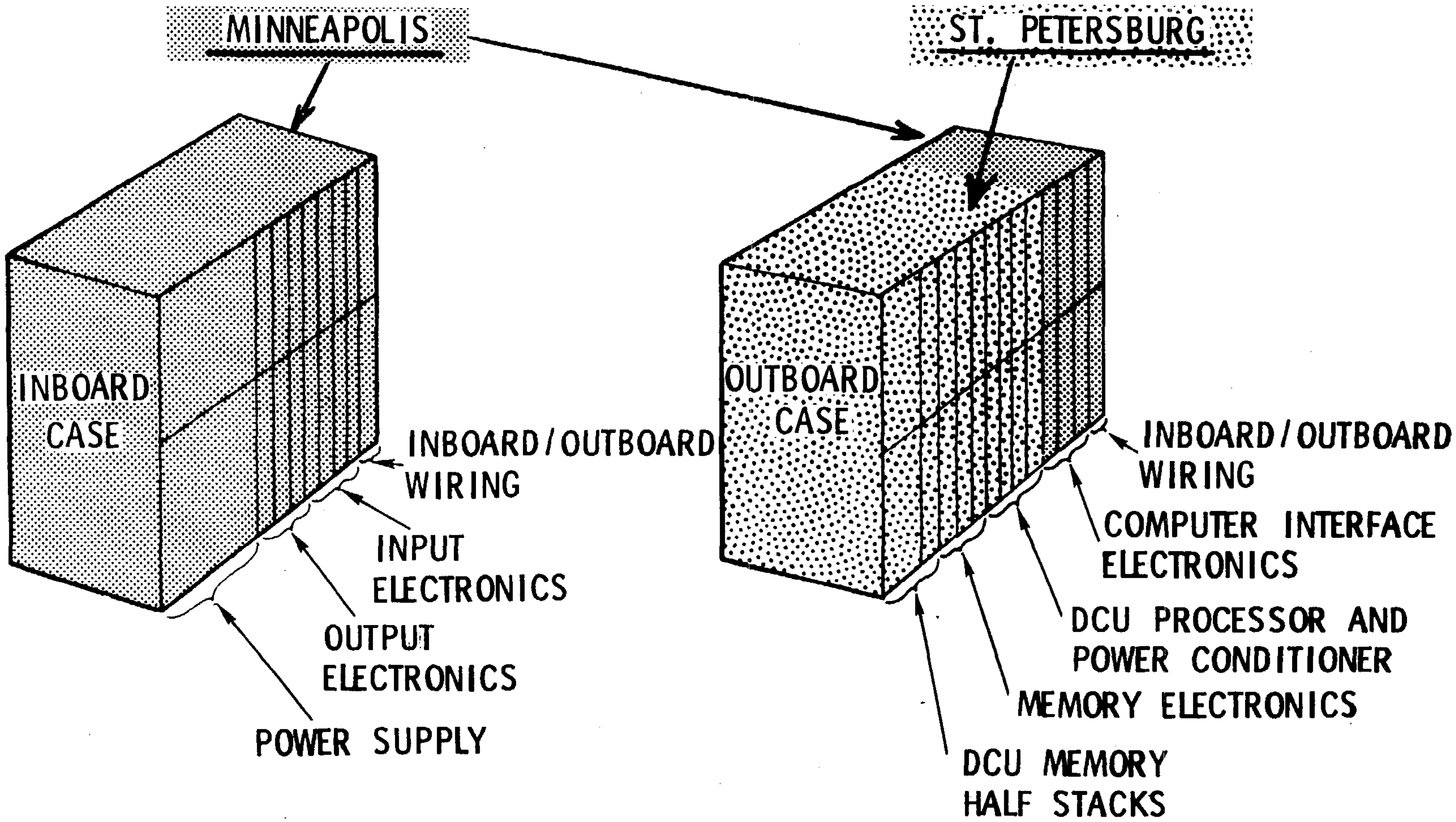


Figure 20

SSME HEAT EXCHANGER DETAILS

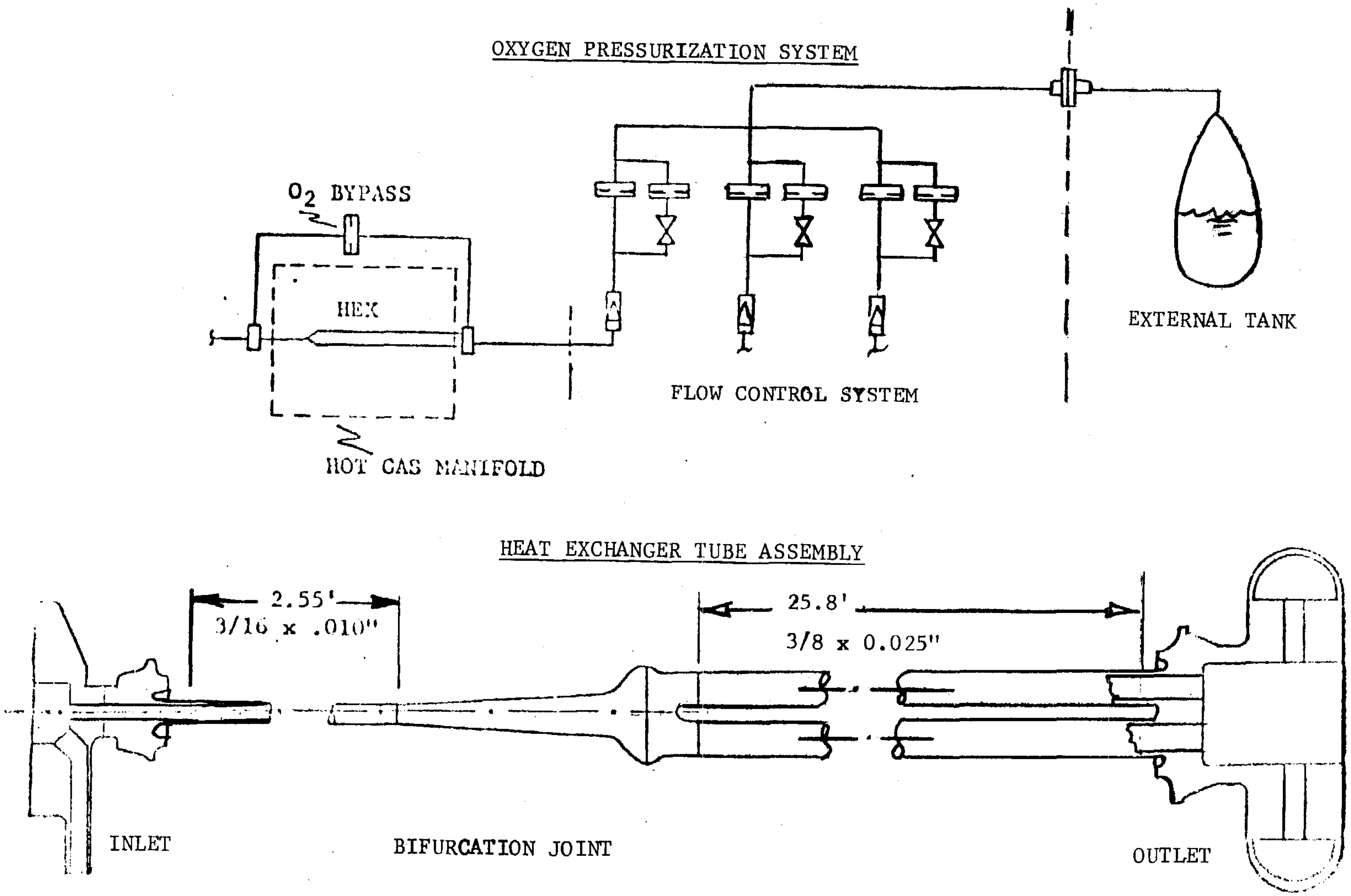


Figure 21

COMPARISON OF SATURN AND SHUTTLE STABILITY LOOPS

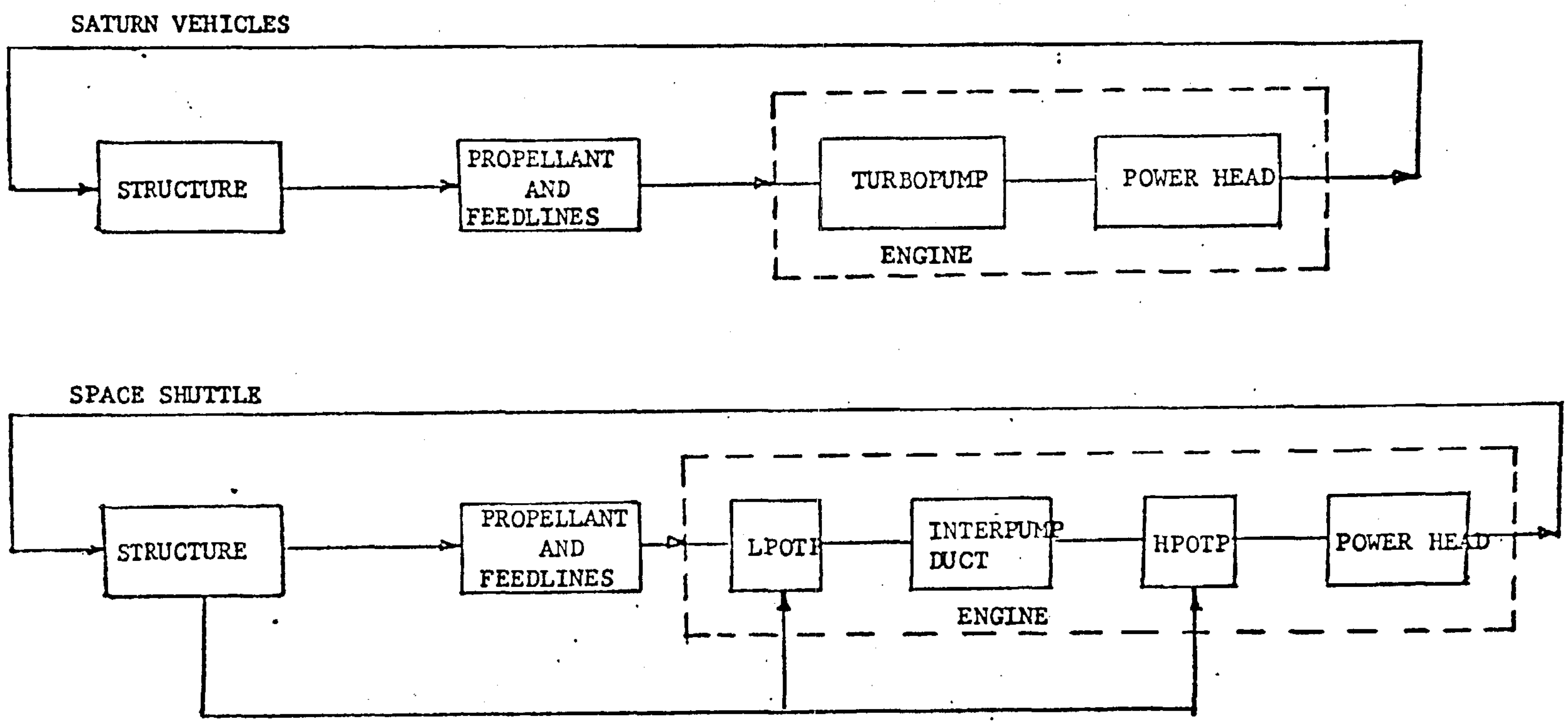


Figure 22

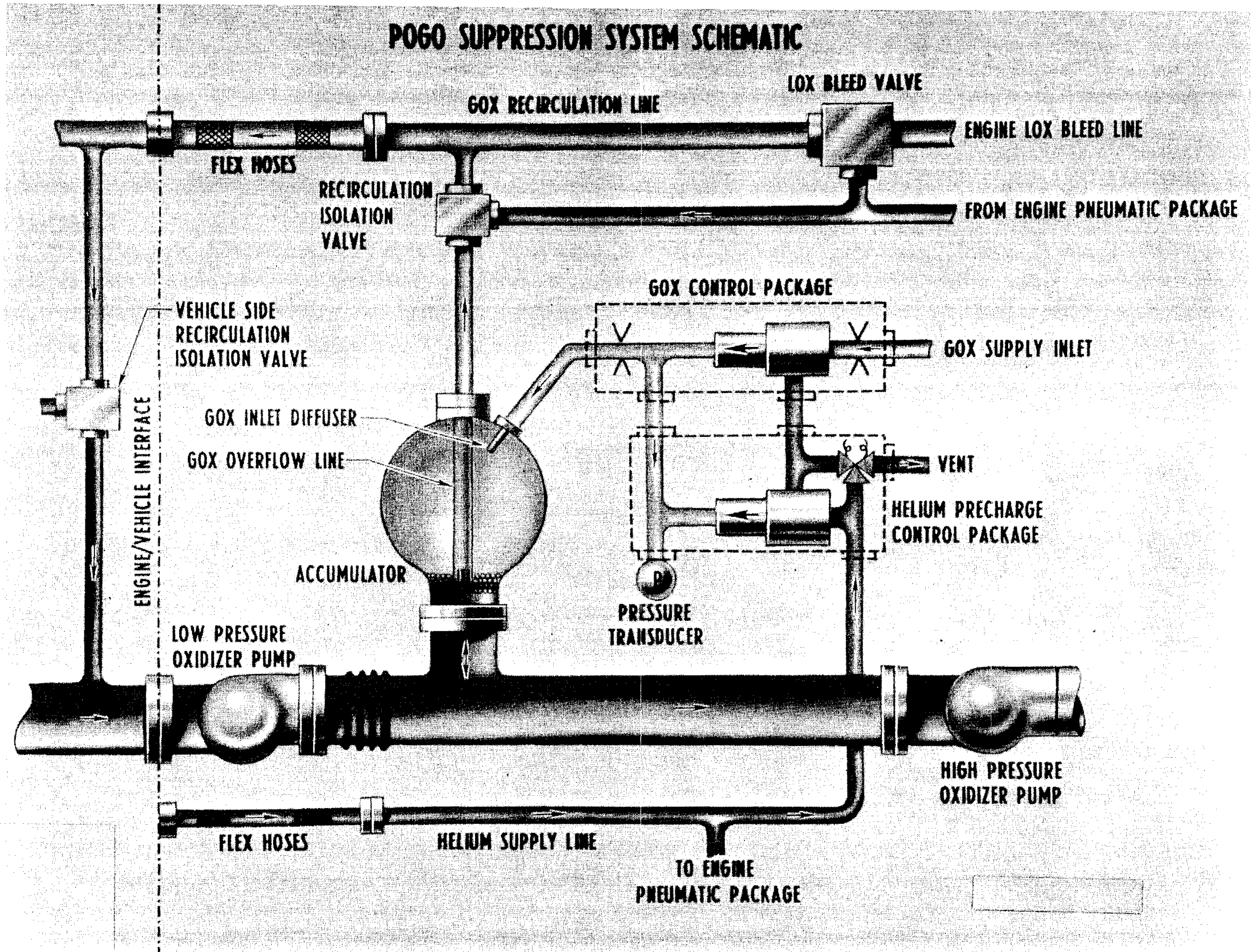

POGO SUPPRESSION SYSTEM SCHEMATIC
LOX BLEED VALVE
GOX RECIRCULATION LINE
ENGINE LOX BLEED LINE
FLEX HOSES
RECIRCULATION
ISOLATION
VALVE
FROM ENGINE PNEUMATIC PACKAGE
GOX CONTROL PACKAGE
VEHICLE SIDE
RECIRCULATION
ISOLATION VALVE
GOX SUPPLY INLET
GOX INLET DIFFUSER
GOX OVERFLOW LINE
VENT
ENGINE/VEHICLE INTERFACE
HELIUM PRECHARGE
CONTROL PACKAGE
ACCUMULATOR
P
PRESSURE
TRANSDUCER
LOW PRESSURE
OXIDIZER PUMP
HIGH PRESSURE
OXIDIZER PUMP
FLEX HOSES
HELIUM SUPPLY LINE
TO ENGINE
PNEUMATIC PACKAGE

Figure 23

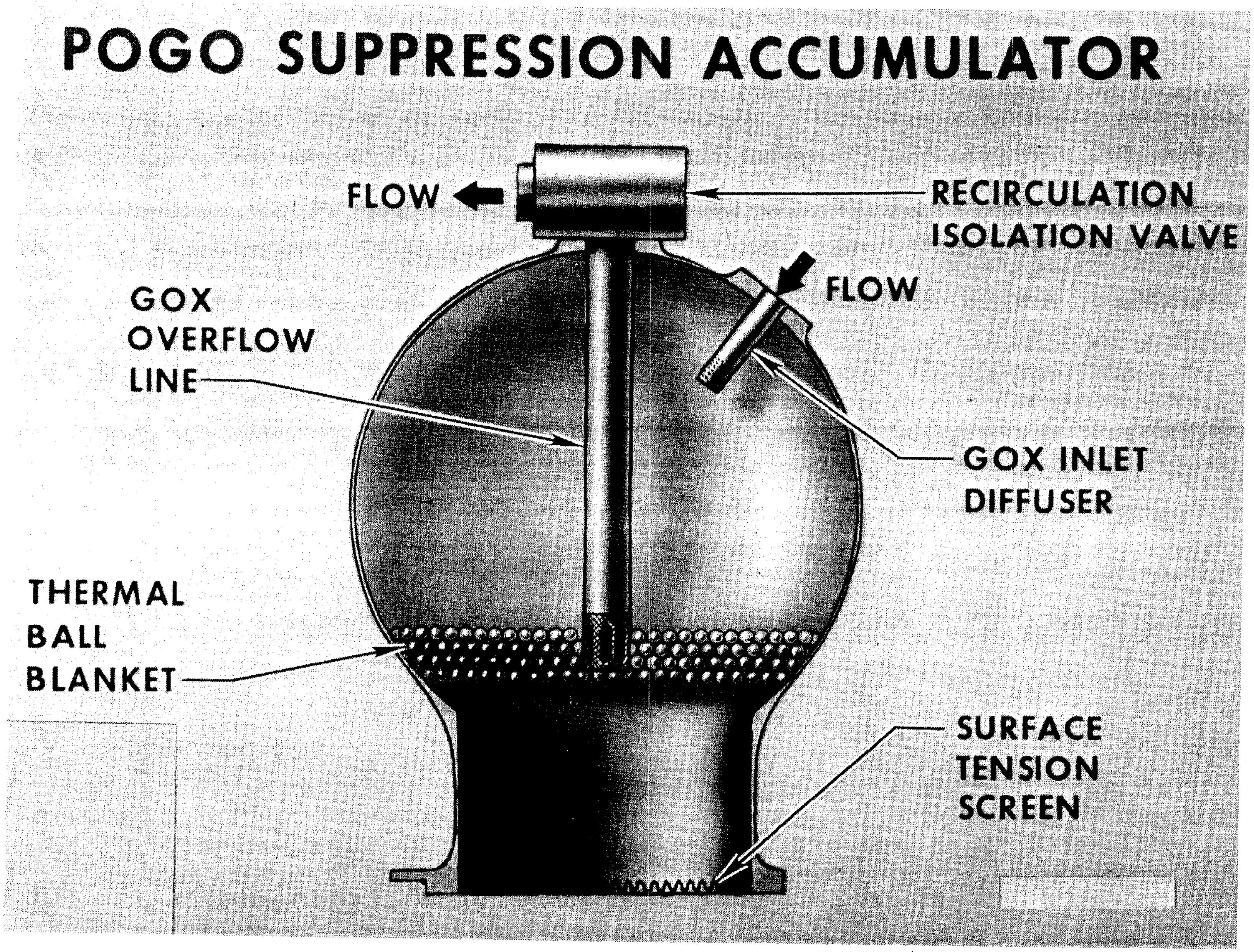

POGO SUPPRESSION ACCUMULATOR
FLOW
RECIRCULATION ISOLATION VALVE
FLOW
GOX OVERFLOW LINE
GOX INLET DIFFUSER
THERMAL BALL BLANKET
SURFACE TENSION SCREEN

Figure 24

EXTERNAL TANK

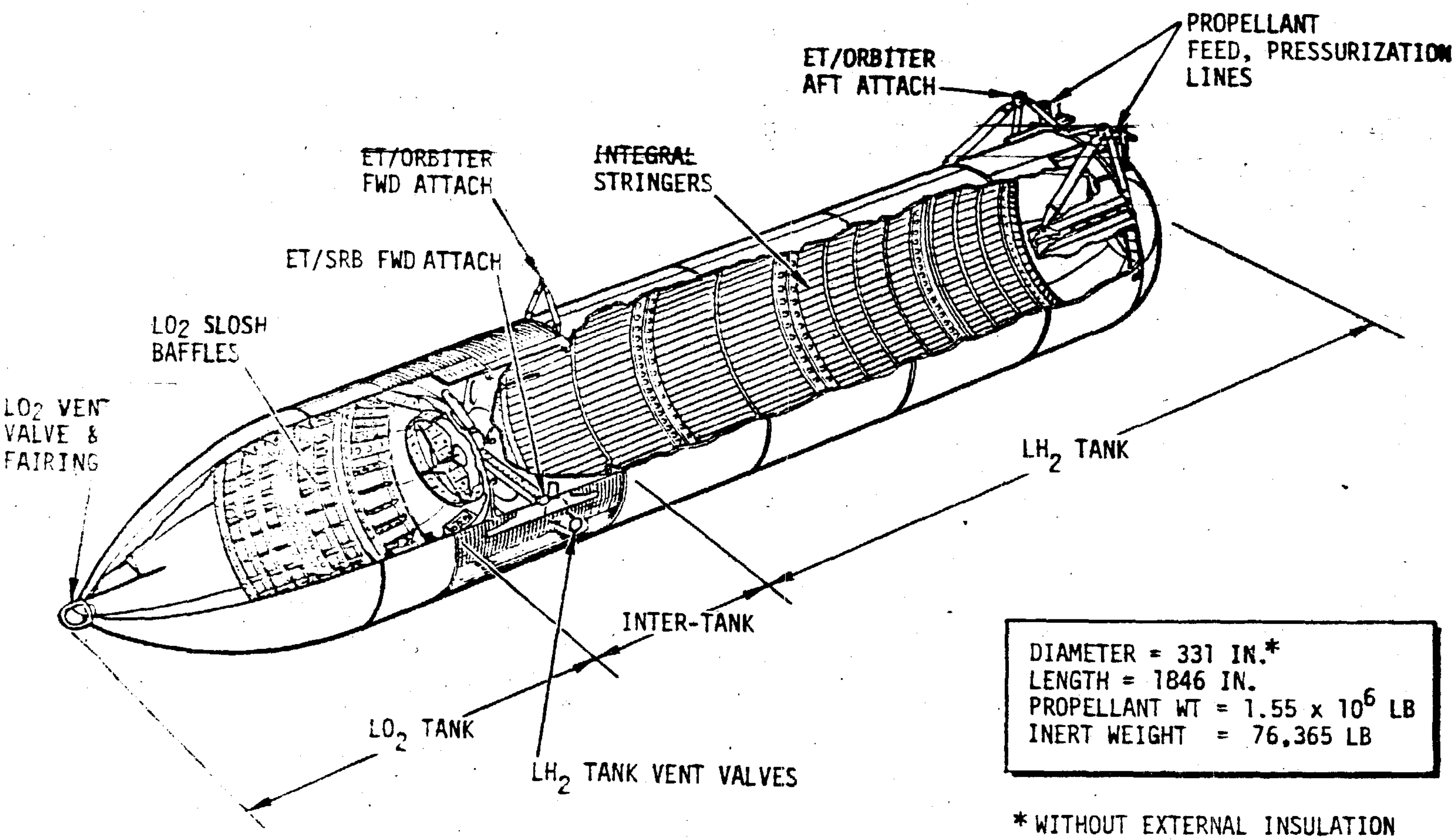
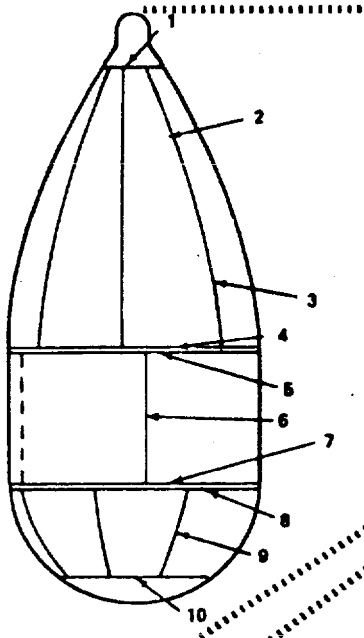


Figure 25

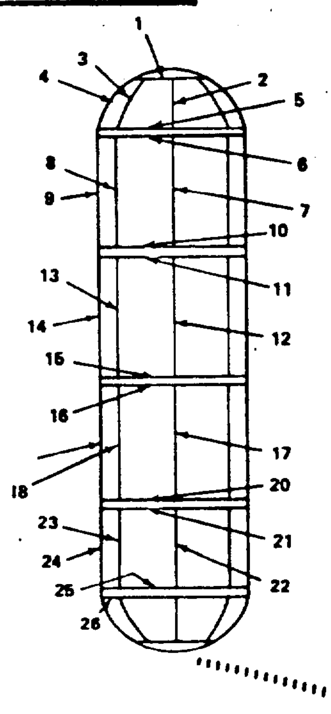
LO$_2$ TANK ANALYSIS



CRITICAL DEFECTS

WELD	THICKNESS, in.	STRESS DIRECTION	STRESS, ksi	CRITICAL CRACK DEPTH, in.	FAILURE MODE
1	0.120	Parallel Transverse	5.3 2.7	12.4 20.4	Leak
2	0.120 .212	Parallel Transverse	2.7/11.6 5.3/23.2	47.0 2.6 5.39 0.279	Leak
3	0.212-0.42	Parallel Transverse	11.6/9.1 23.2/18.2	2.6-4.2 0.279 0.458	Leak
4	0.236	Parallel Transverse	32.1 13.5	0.34 0.83	Leak
5	0.400	Parallel Transverse	27.3 7.6	0.47 2.6	Leak
6	0.452-0.479	Parallel Transverse	7.0/6.6 [illegible]	7.5-8.2 0.25	Fracture
7	0.386	Parallel Transverse	30.2 8.3	0.39 2.2	Leak
8	0.280	Parallel Transverse	4.9 22.0	14.5 0.312	Leak
9	0.198-0.261	Parallel Transverse	31.2/28.5 6.9/22.1	0.36-0.43 3.18-0.31	Leak
10	0.37	Parallel Transverse	15.5 20	1.42 0.38	Leak

LH$_2$ TANK ANALYSIS



WELD	THICKNESS, in.	STRESS DIRECTION	STRESS, ksi	CRITICAL CRACK DEPTH, in.	FAILURE MODE
1	0.175	Transverse Parallel	23.85 18.5	0.266 1.04	Leak
2	0.175-0.150	Transverse Parallel	18.5 23.85	0.444 0.612	Leak
3	0.175-0.150	Transverse Parallel	18.5 23.85	0.444 0.59	Leak
4	0.175-0.150	Transverse Parallel	18.5 23.85	0.444 0.612	Leak
5	0.150	Transverse Parallel	23.1 5.1	0.284 13.30	Leak
6	0.324	Transverse Parallel	21.4 10.7	0.33 3.02	Leak
7	0.324	Transverse Parallel	21.4 10.7	0.33 3.02	Leak
8	0.324	Transverse Parallel	21.4 10.7	0.33 3.02	Leak
9	0.324	Transverse Parallel	21.4 11.9	0.33 2.44	Leak
10	0.324	Transverse Parallel	11.9 21.4	1.06 0.76	Leak
11	0.324	Transverse Parallel	11.9 21.4	1.06 0.76	Leak
12	0.324	Transverse Parallel	21.4 10.7	0.33 3.02	Leak
13	0.324	Transverse Parallel	21.4 11.71	0.33 2.4	Leak
14	0.324	Transverse Parallel	21.4 13.83	0.33 1.85	Leak
15	0.324	Transverse Parallel	13.83 21.4	0.80 0.76	Leak
16	0.324	Transverse Parallel	13.83 21.4	0.80 0.76	Leak
17	0.324	Transverse Parallel	21.4 10.7	0.33 3.02	Leak
18	0.324	Transverse Parallel	21.4 13.14	0.33 2.03	Leak
19	0.324	Transverse Parallel	21.4 15.67	0.33 1.63	Leak
20	0.324	Transverse Parallel	15.67 21.4	0.62 0.76	Leak
21	0.324	Transverse Parallel	15.67 21.4	0.62 0.76	Leak
22	0.324	Transverse Parallel	21.4 10.7	0.33 3.02	Leak
23	0.324	Transverse Parallel	21.4 13.14	0.33 2.03	Leak
24	0.324	Transverse Parallel	21.4 15.67	0.33 1.63	Leak
25	0.324	Transverse Parallel	10.7 21.4	1.32 0.76	Leak
26	0.324	Transverse Parallel	10.7 21.4	1.32 0.76	Leak

Figure 26

EXTERNAL TANK

TYPICAL MECHANICAL JOINT

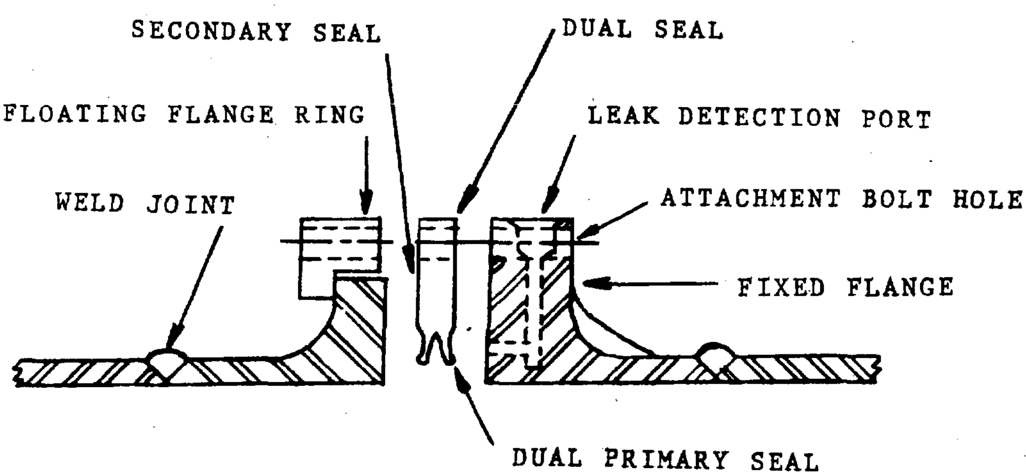


Figure 27

External Tank Propulsion/Mechanical Subsystem LO_2 Propellant Feed

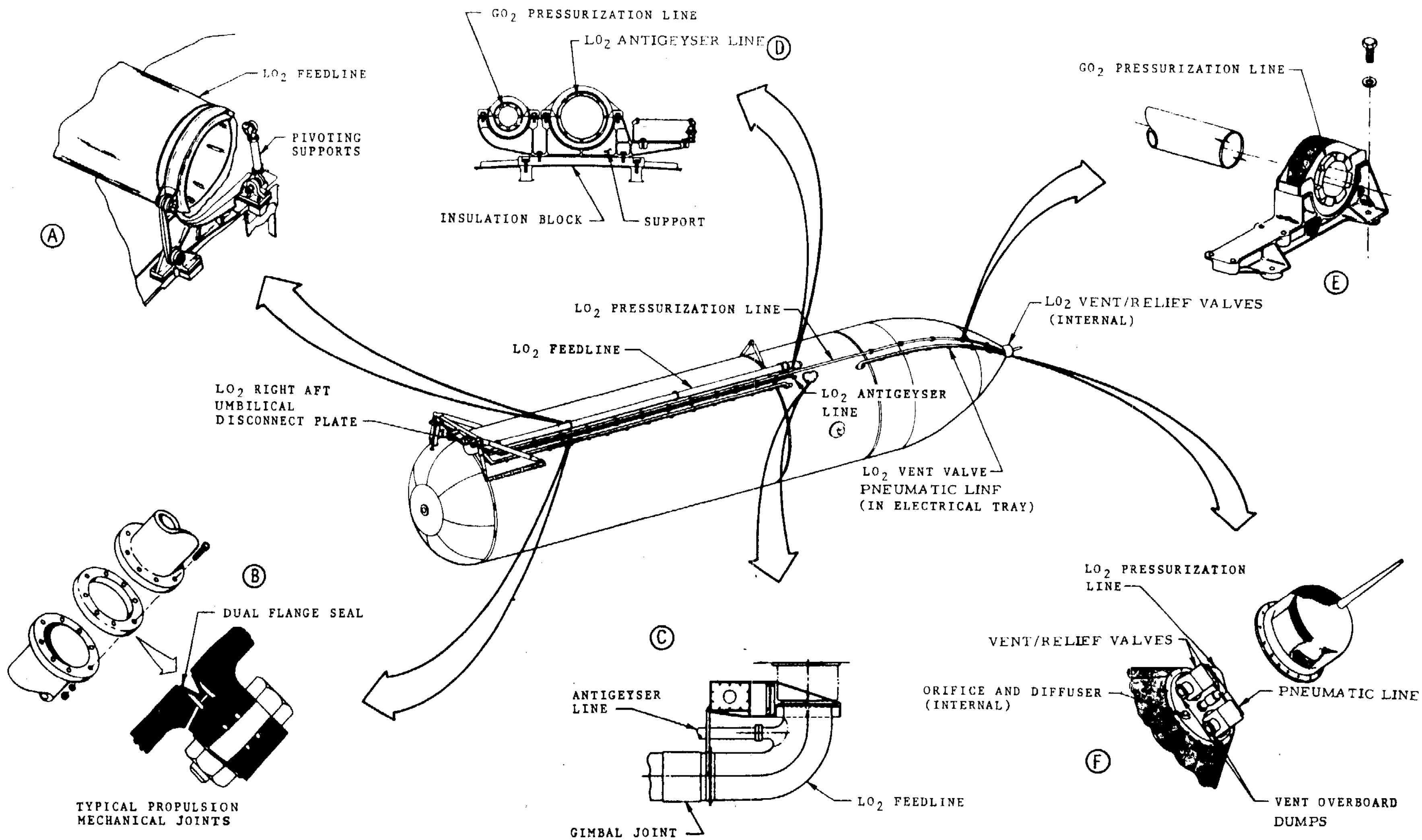


Figure 28

External Tank Propulsion/Mechanical Subsystem Separation Hardware

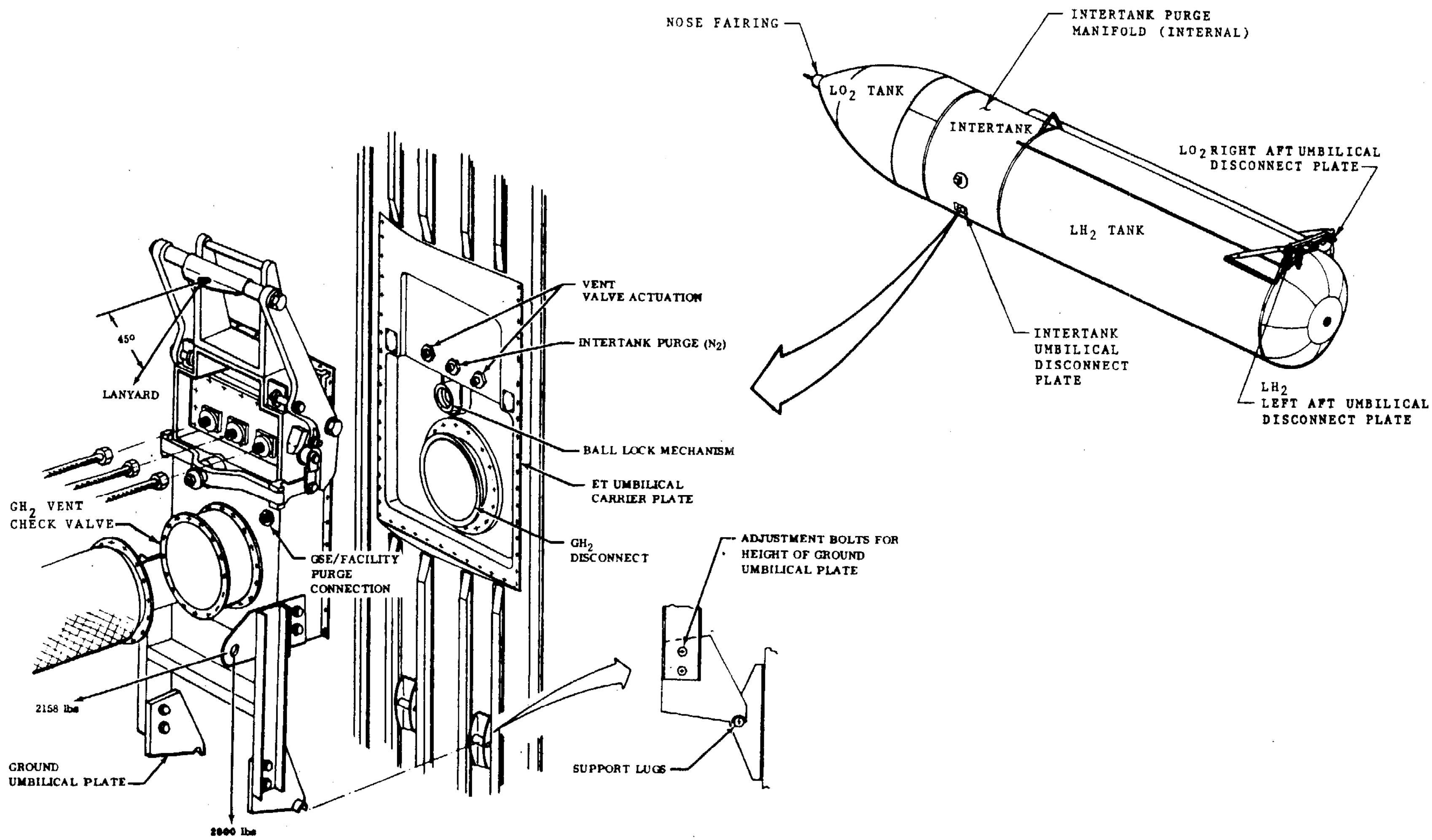


Figure 29

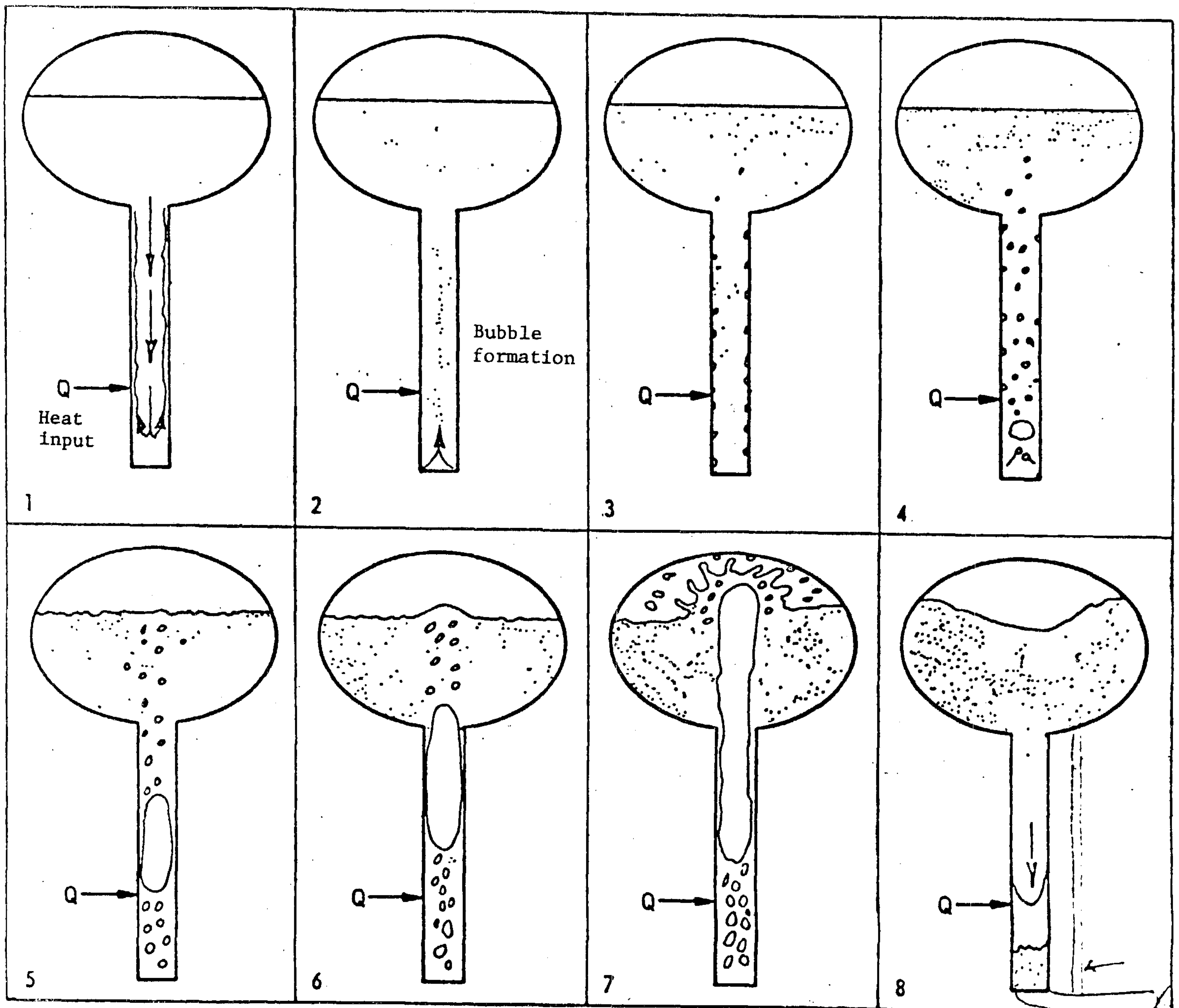
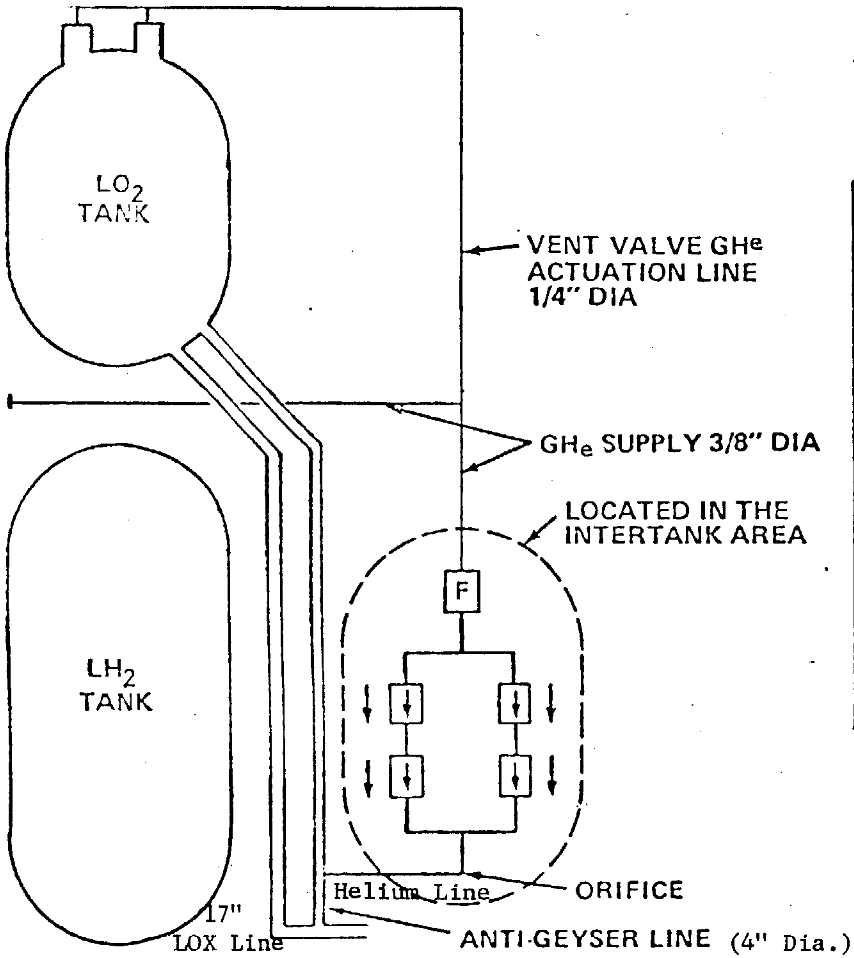


Figure 30

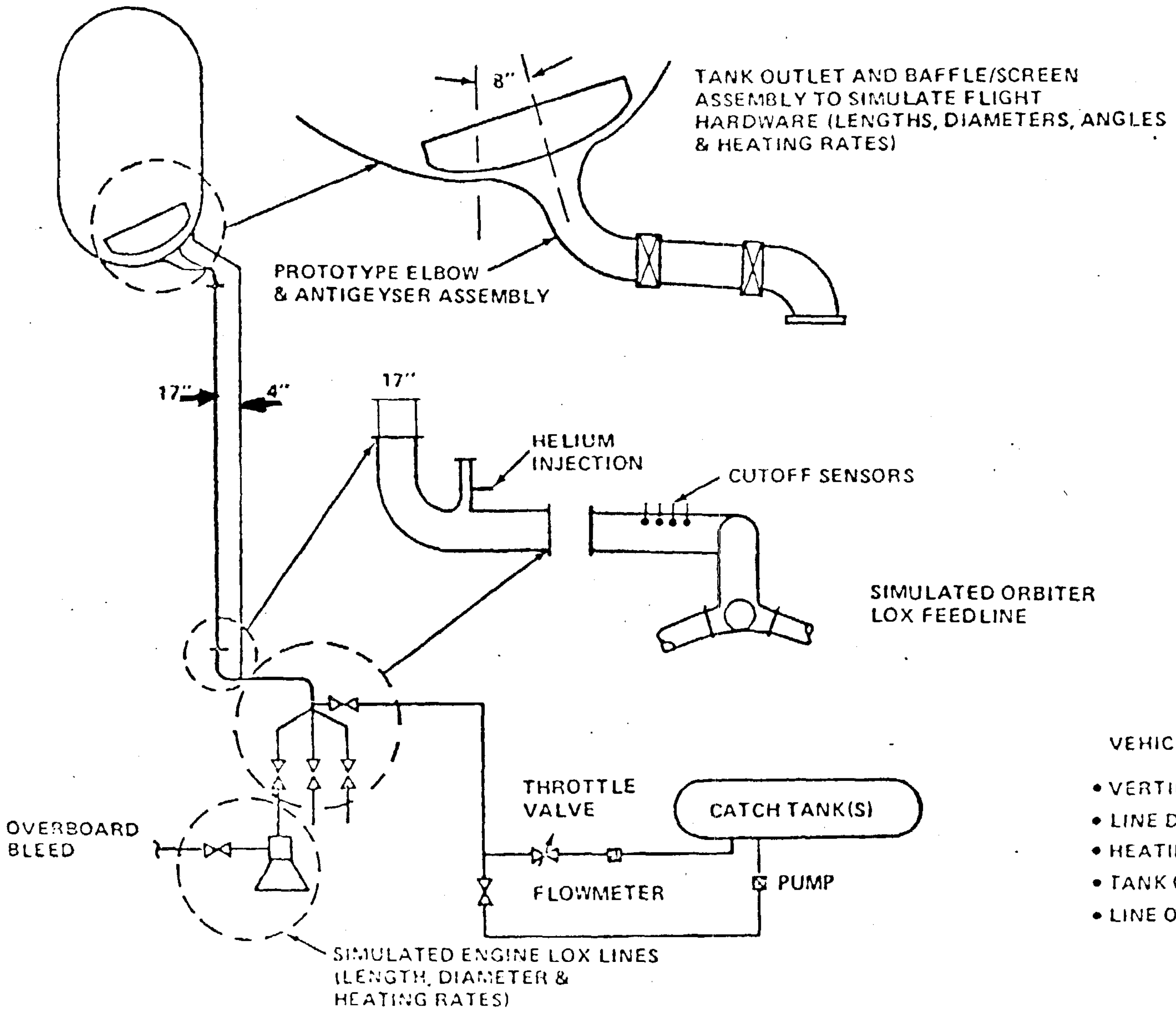
HELIUM INJECTION SYSTEM



DESIGN FEATURE	INITIAL DESIGN CONCEPT	CURRENT DESIGN	RATIONALE
CHECK VALVES	2–SERIES	4–SERIES/ PARALLEL	INCREASED RELIABILITY
LINE SIZE	1/4"	3/8"	MARGIN FOR GROWTH
FILTER	NONE	ONE	PROTECT CHECK VALVES
COMPONENT LOCATION	NOT DEFINED	INTERTANK	MINIMIZE EFFECT ON VENT VALVE OPERATION
WEIGHT	8 LB	25 LB	MORE COMPONENTS

Figure 31

ET ANTI-GEYSER SYSTEM TEST CONFIGURATION



VEHICLE SIMULATION:

- VERTICLE LINE LENGTHS
- LINE DIAMETERS
- HEATING RATES
- TANK OUTLET HARDWARE
- LINE ORIENTATION ANGLES

Figure 32

POINT SENSOR PROPELLANT GAUGING SYSTEM
BASELINE CONFIGURATION

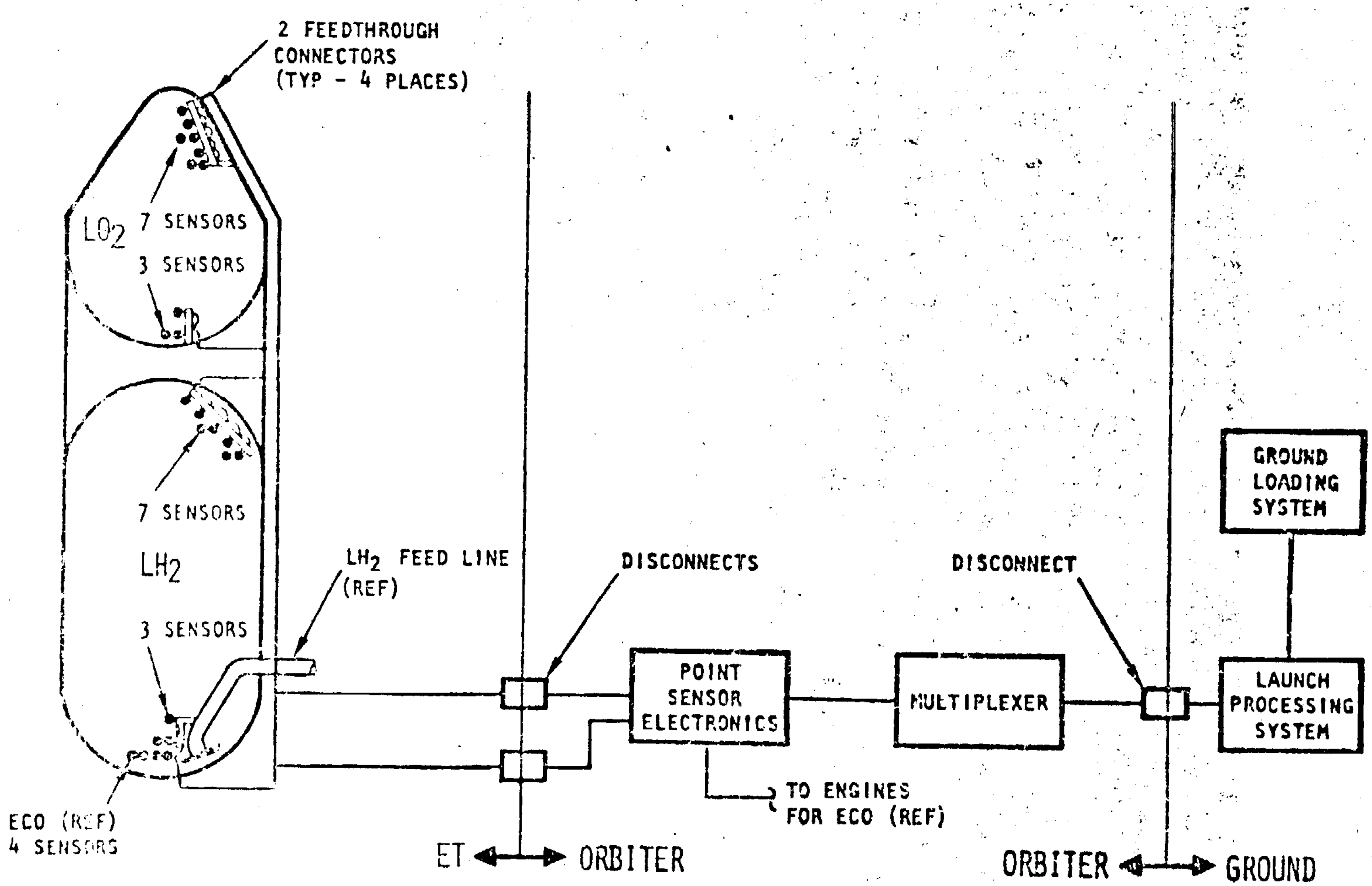


Figure 33

ET/SRB ATTACH CONFIGURATION

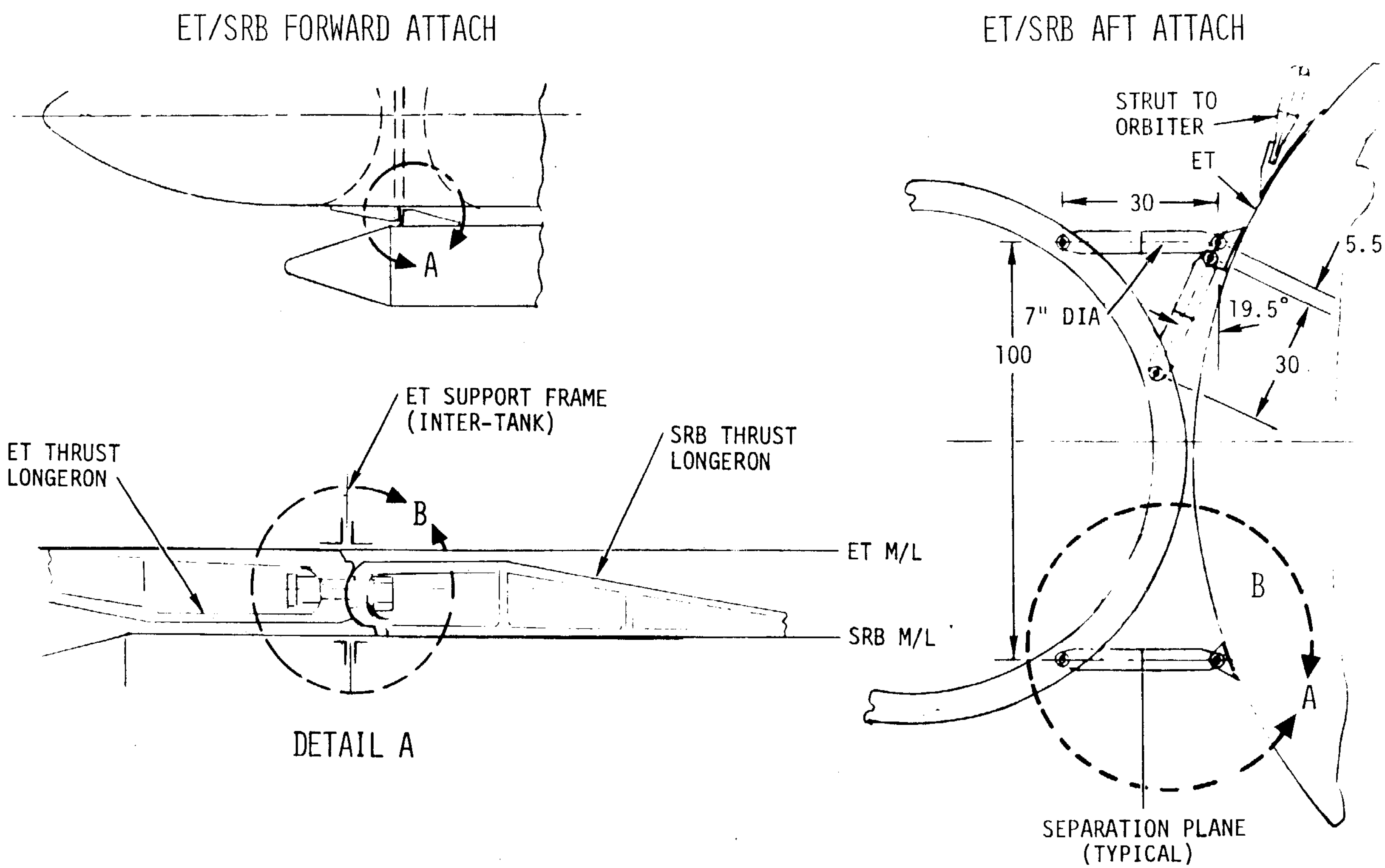


Figure 34

ET/ORBITER FWD STRUCTURAL ATTACH

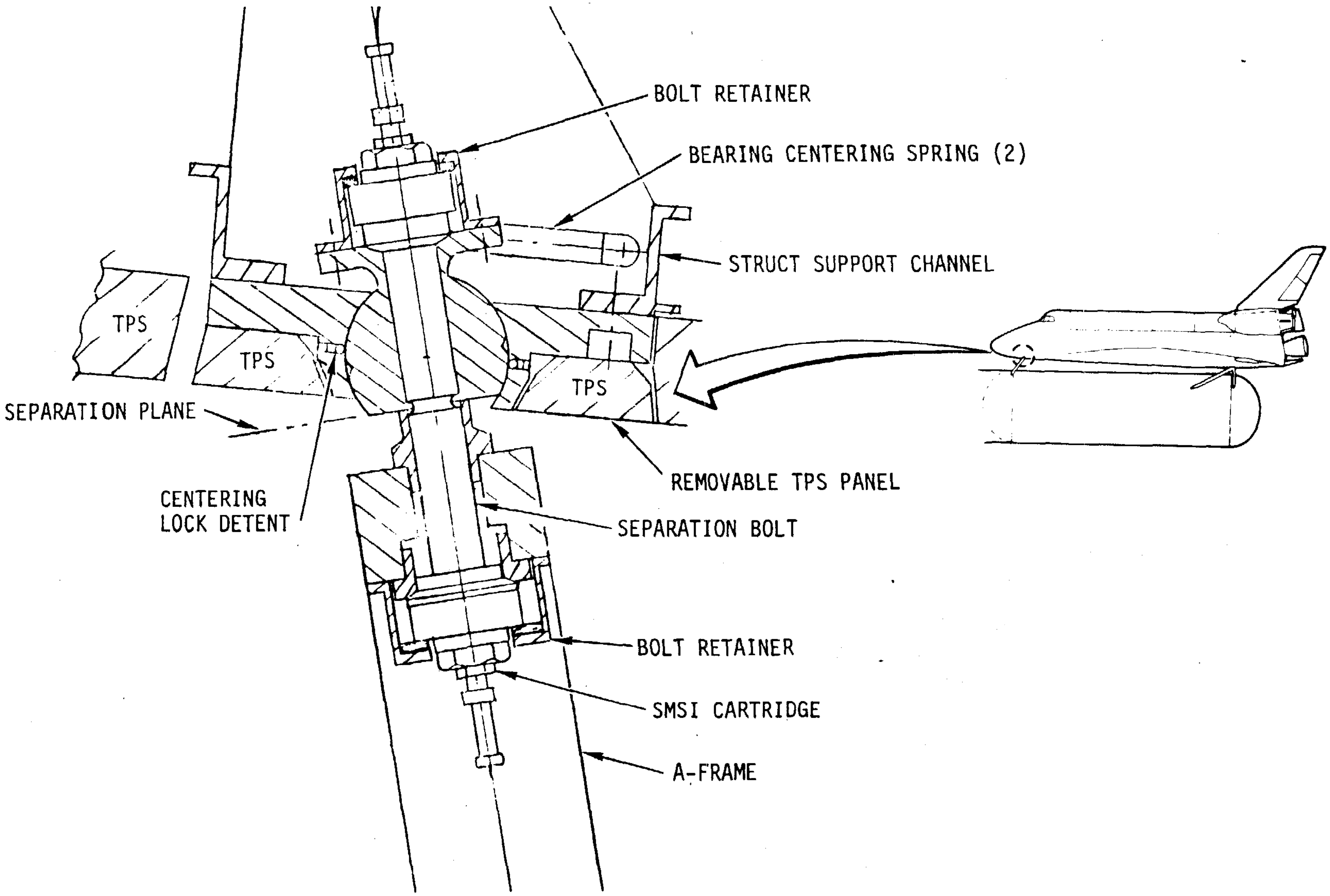


Figure 35

ET/ORBITER AFT INTERFACE STRUCTURE

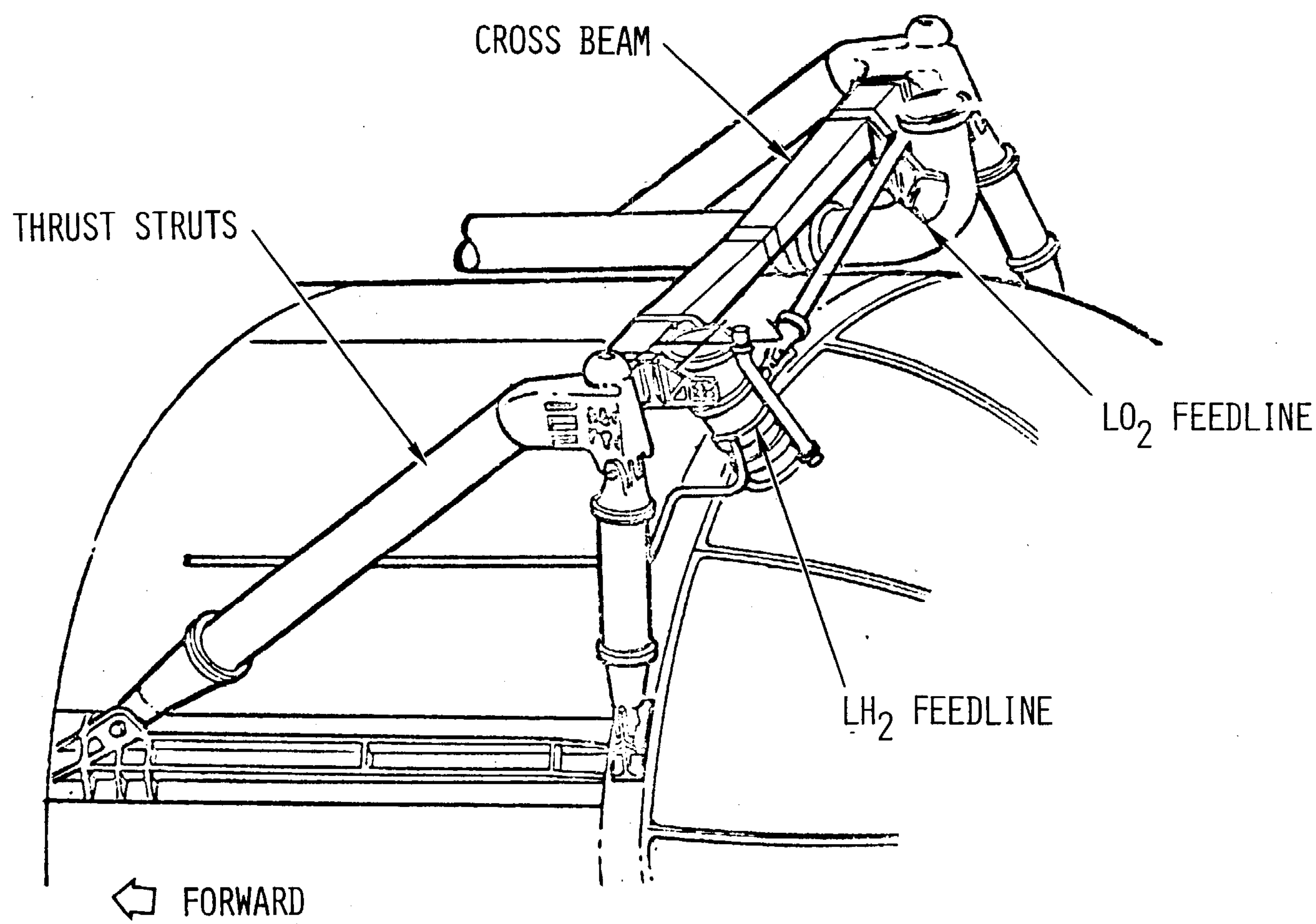


Figure 36

ET DISPOSAL FOR KSC LAUNCHS
NOMINAL AND AOA

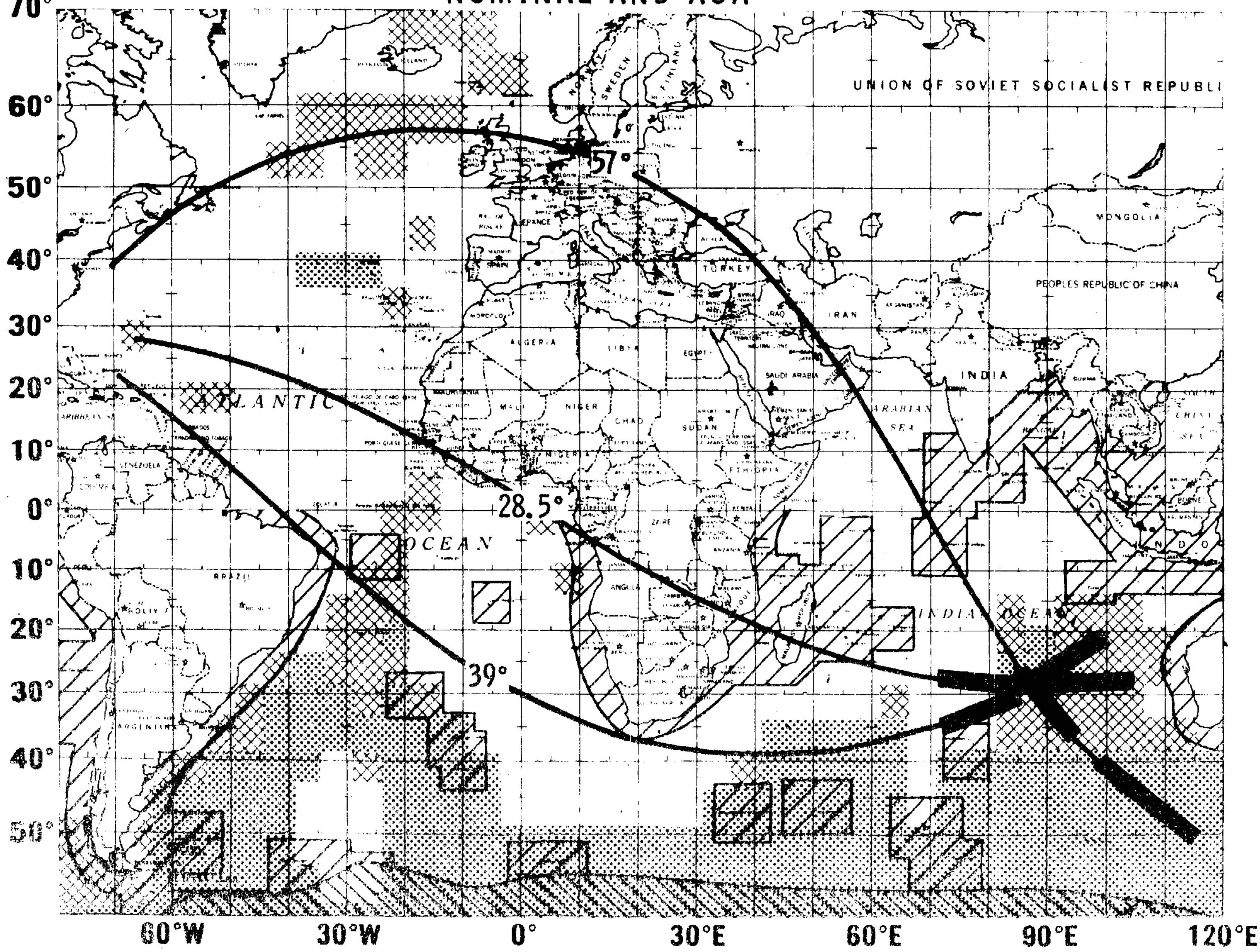


Figure 37

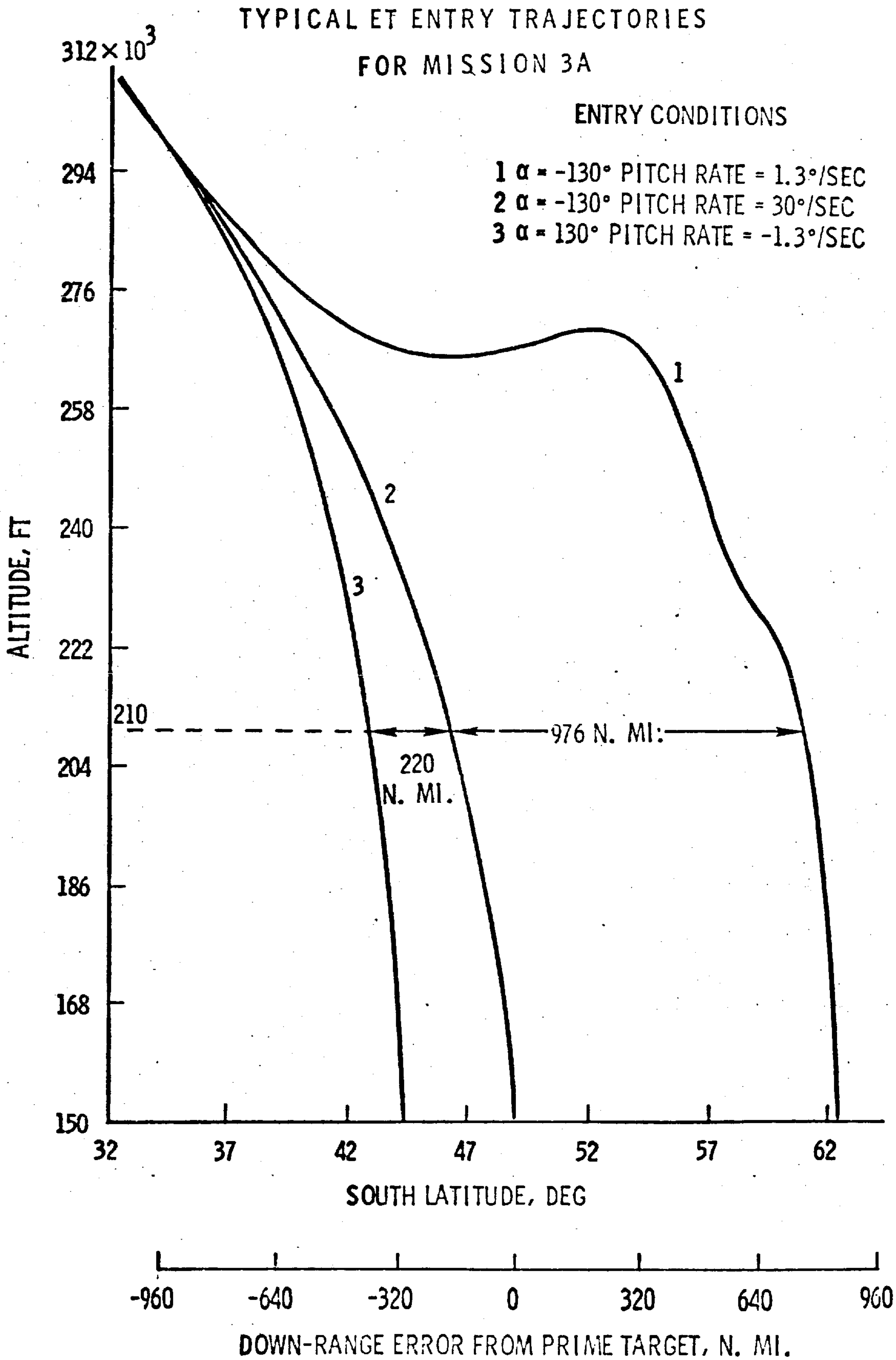

TYPICAL ET ENTRY TRAJECTORIES
FOR MISSION 3A
ENTRY CONDITIONS
1 α = -130° PITCH RATE = 1.3°/SEC
2 α = -130° PITCH RATE = 30°/SEC
3 α = 130° PITCH RATE = -1.3°/SEC
312 × 10^3
294
276
258
240
222
210
204
186
168
150
ALTITUDE, FT
1
2
3
976 N. MI.
220
N. MI.
32
37
42
47
52
57
62
SOUTH LATITUDE, DEG
-960
-640
-320
0
320
640
960
DOWN-RANGE ERROR FROM PRIME TARGET, N. MI.

Figure 38

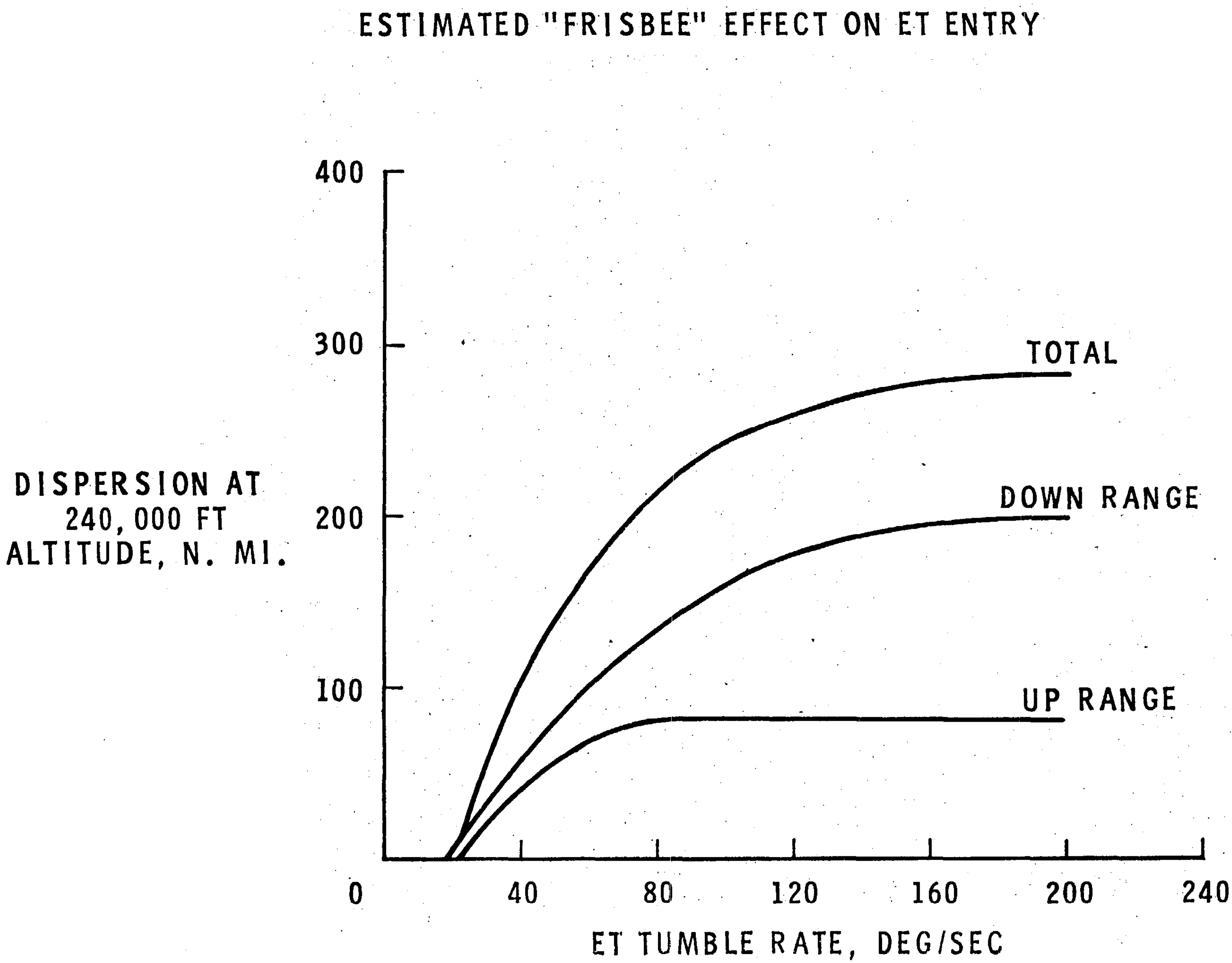
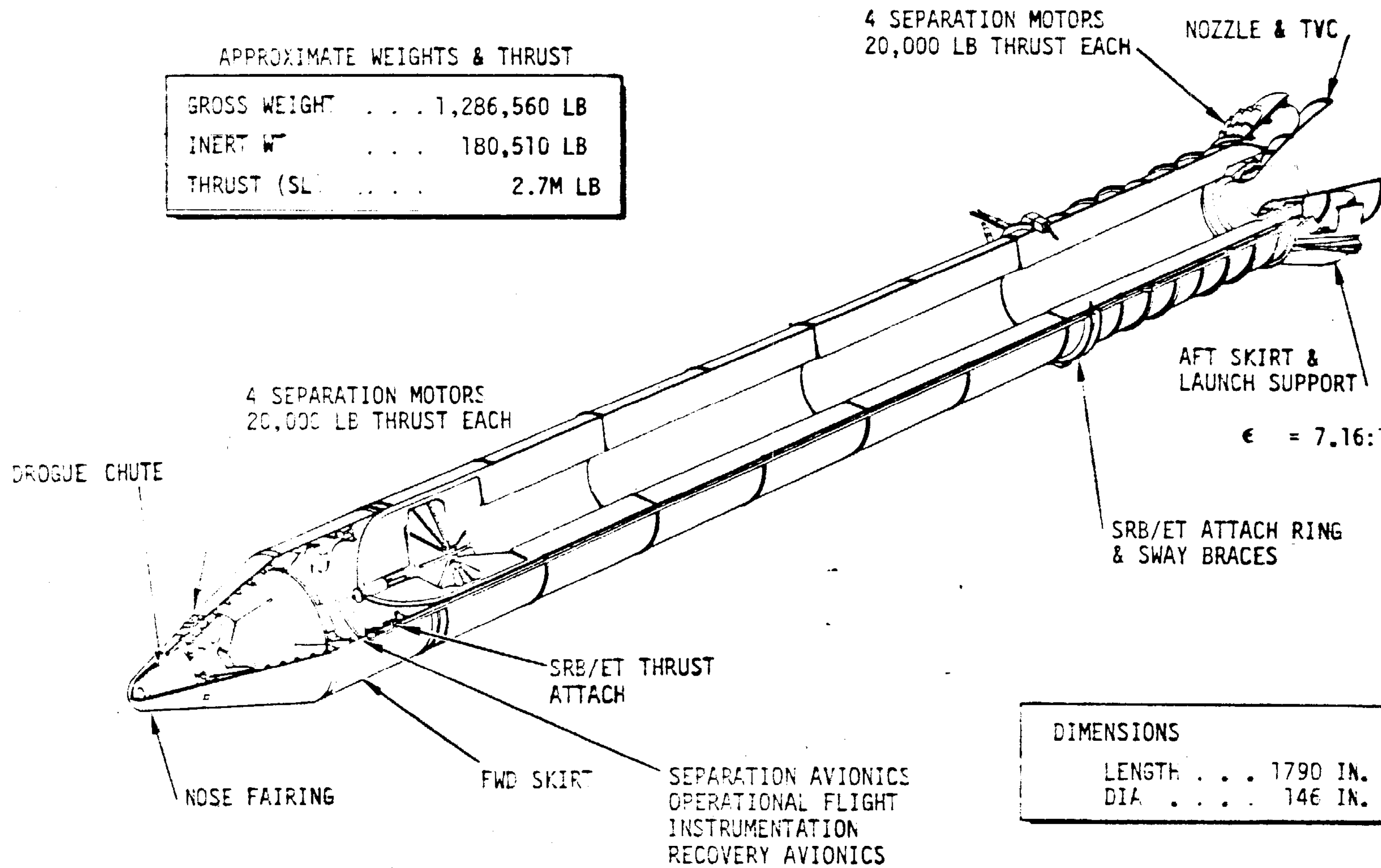

ESTIMATED "FRISBEE" EFFECT ON ET ENTRY
DISPERSION AT
240,000 FT
ALTITUDE, N. MI.
400
300
200
100
0
40
80
120
160
200
240
TOTAL
DOWN RANGE
UP RANGE
ET TUMBLE RATE, DEG/SEC

Figure 39

SOLID ROCKET BOOSTER

APPROXIMATE WEIGHTS & THRUST

GROSS WEIGHT . . .	1,286,560 LB
INERT WT . . .	180,510 LB
THRUST (SL)	2.7M LB



DIMENSIONS

LENGTH . . .	1790 IN.
DIA	146 IN.

Figure 40

Case Design Configuration

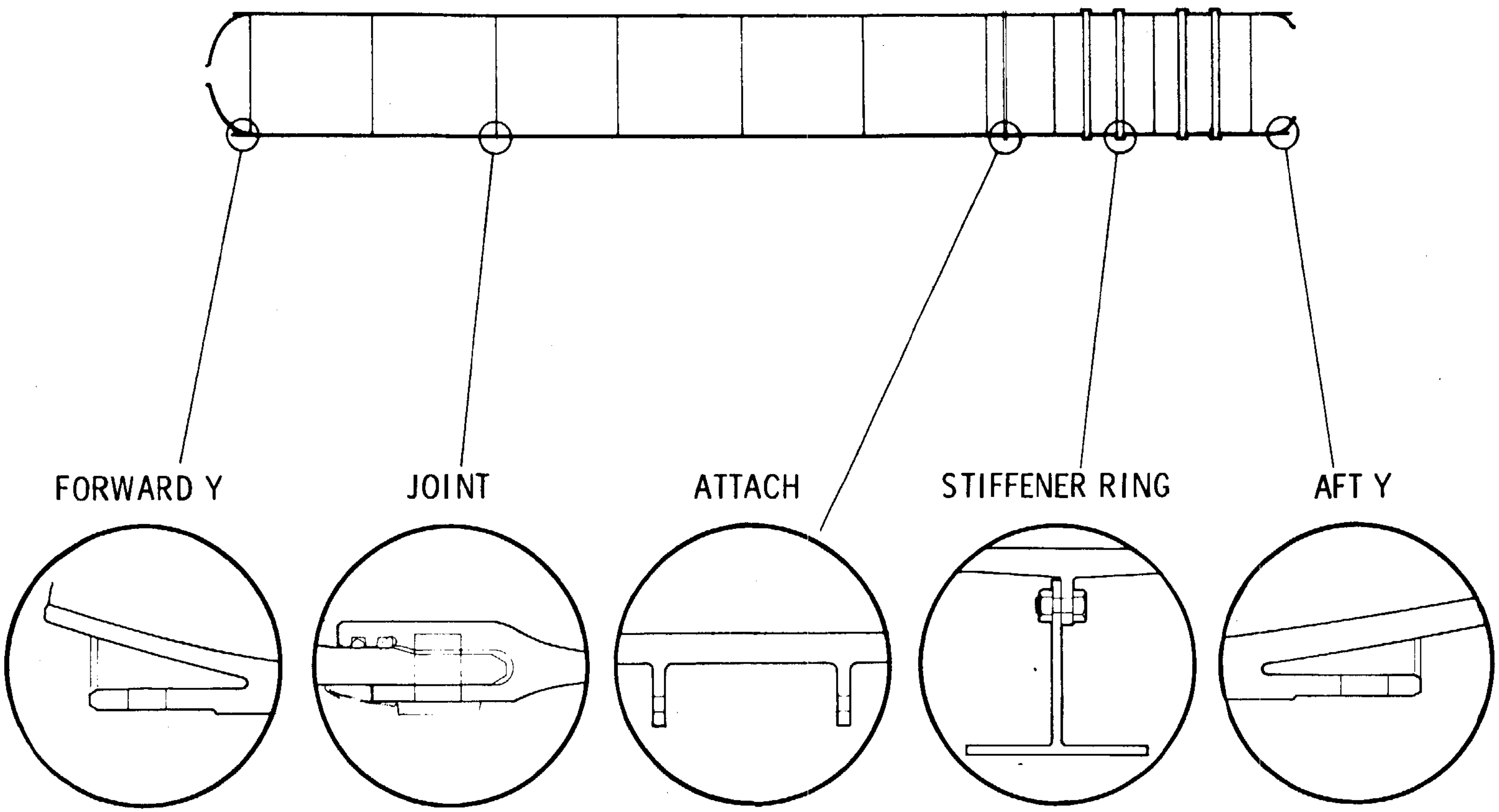


Figure 41

REPRODUCIBILITY LIMITS

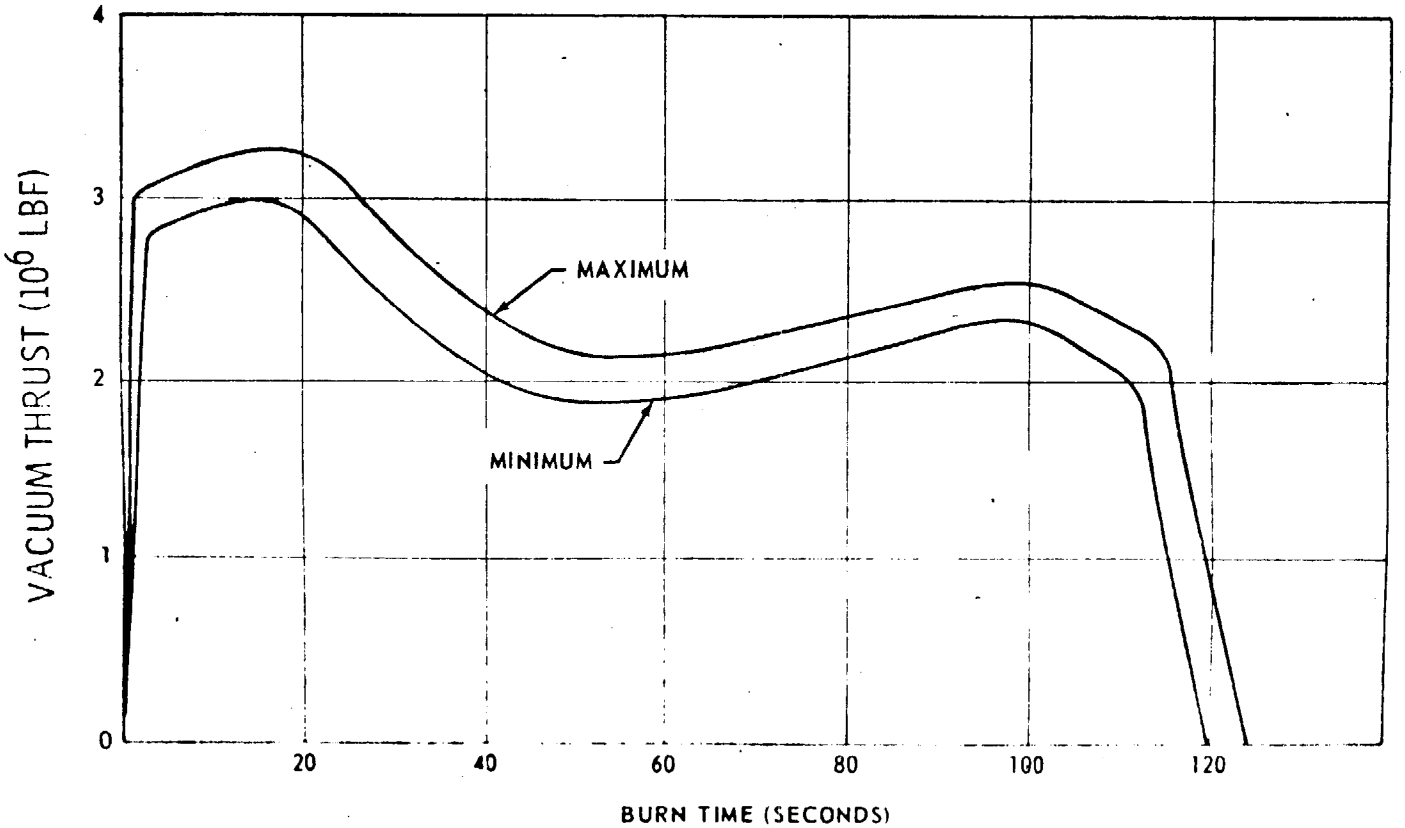


Figure 42

SRB/ET SEPARATION SYSTEM

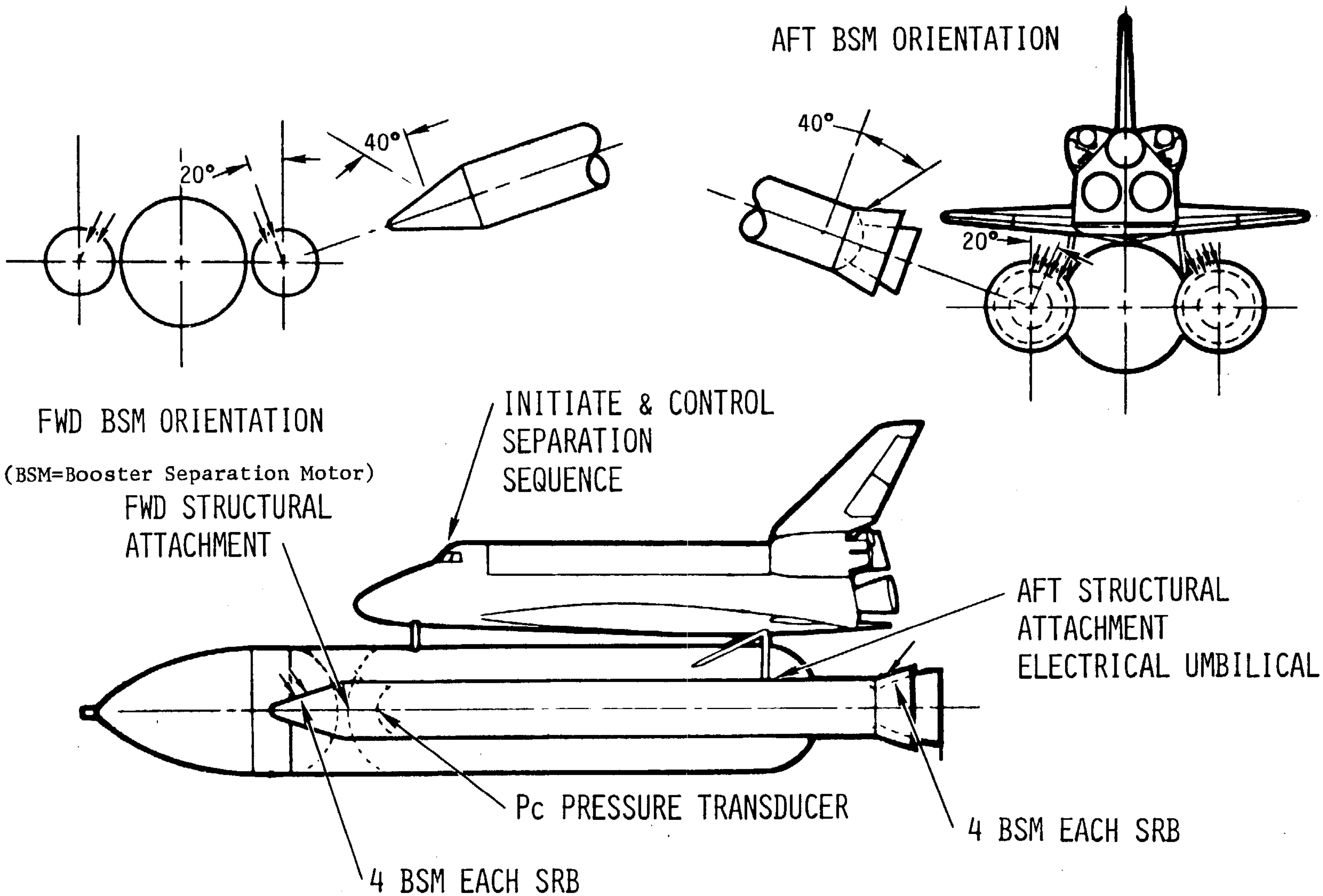
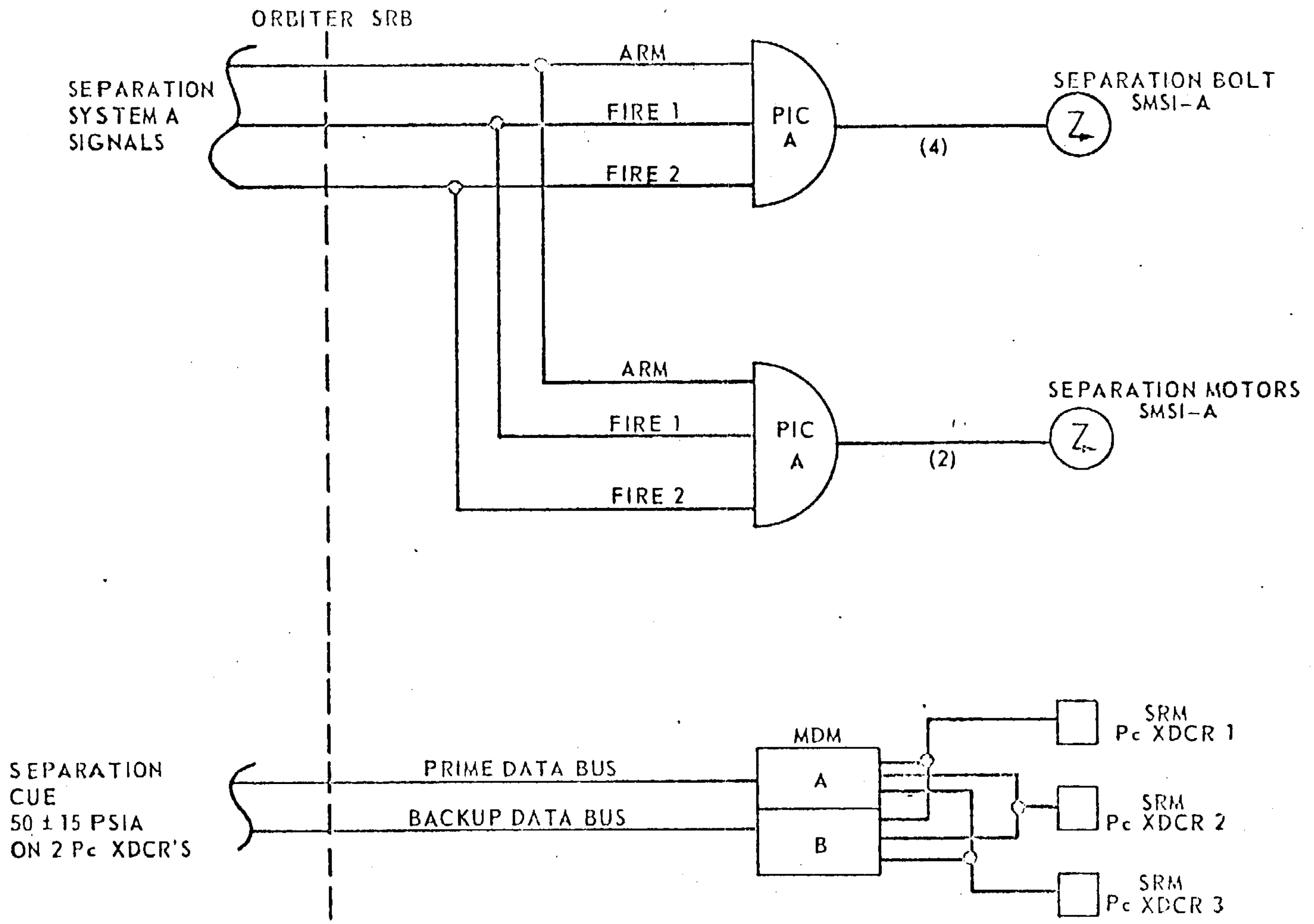


Figure 43

SEPARATION SYSTEM AVIONICS



PIC – PYROTECHNIC INITIATOR CONTROLLER
MDM – MULTIPLEXER – DEMULTIPLEXER
SMSI – STANDARD MANNED SPACEFLIGHT INITIATOR
Pc – CHAMBER PRESSURE
XDCR – TRANSDUCER

Figure 44

SRB PDR
RECOVERY SUBSYSTEM

NOMINAL TRAJECTORY

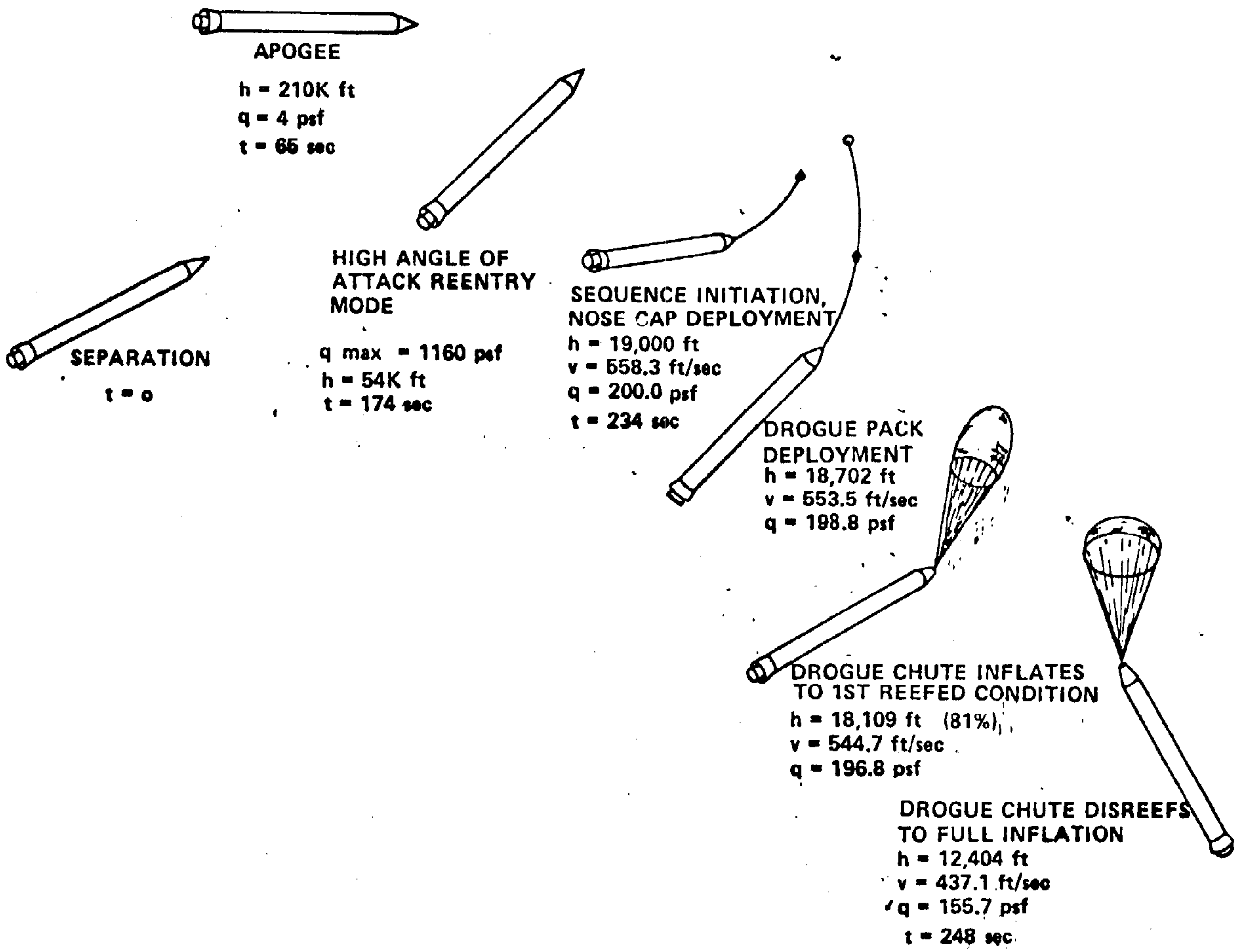


Figure 45

SRB PDR
RECOVERY SUBSYSTEM

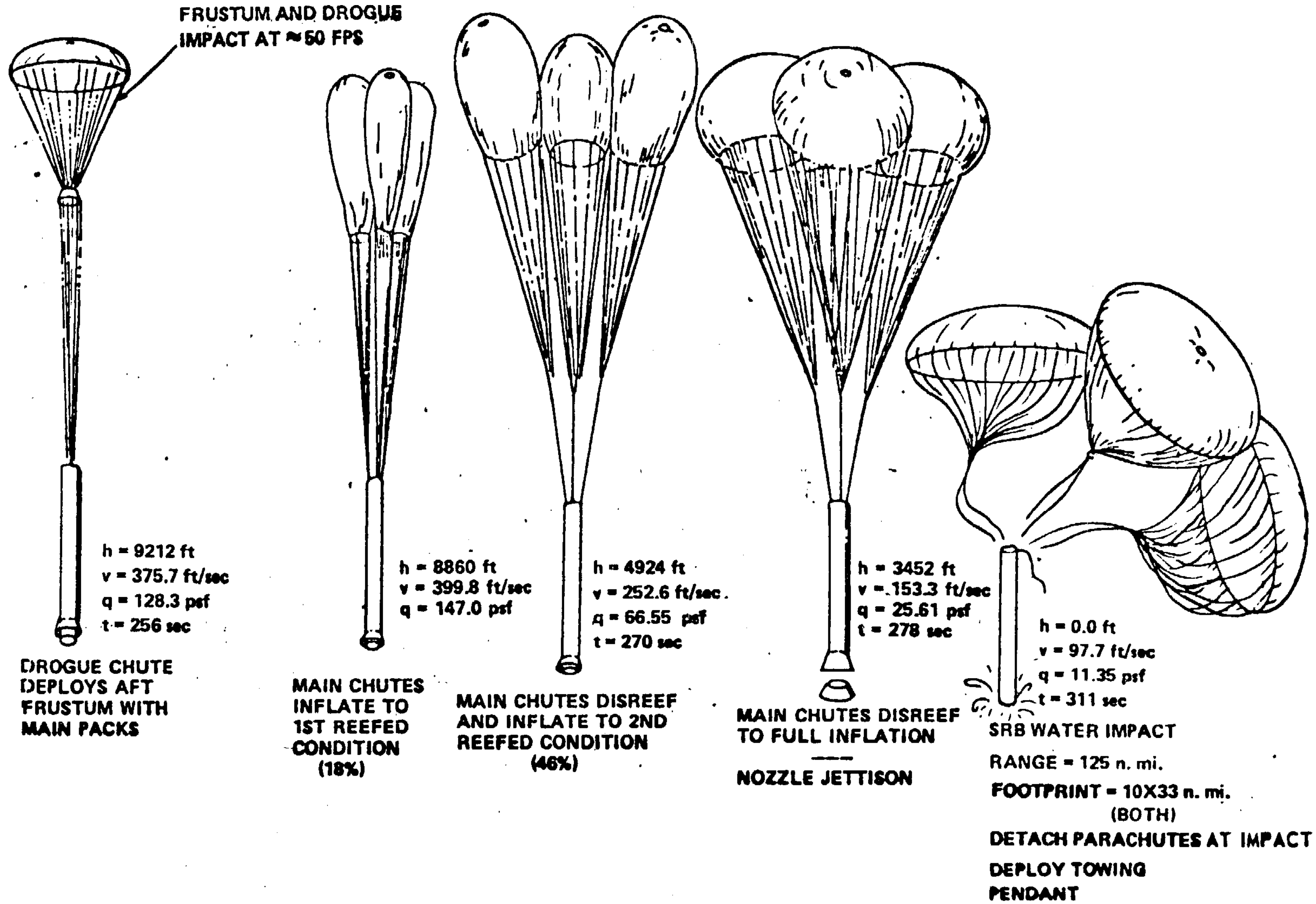


Figure 45 (Concluded)

RECOVERY SYSTEM PACKAGING IN THE NOSE CONE

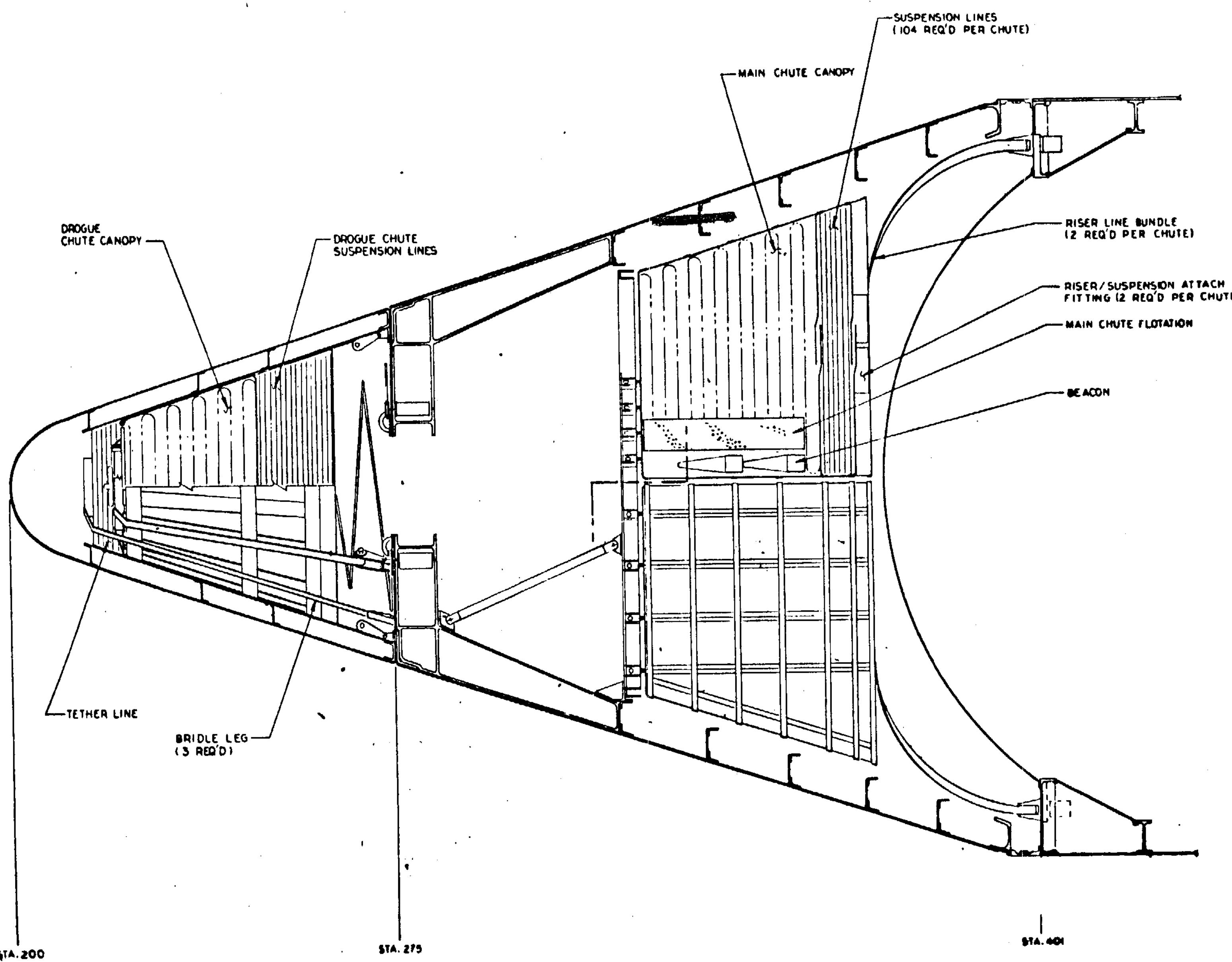


Figure 46

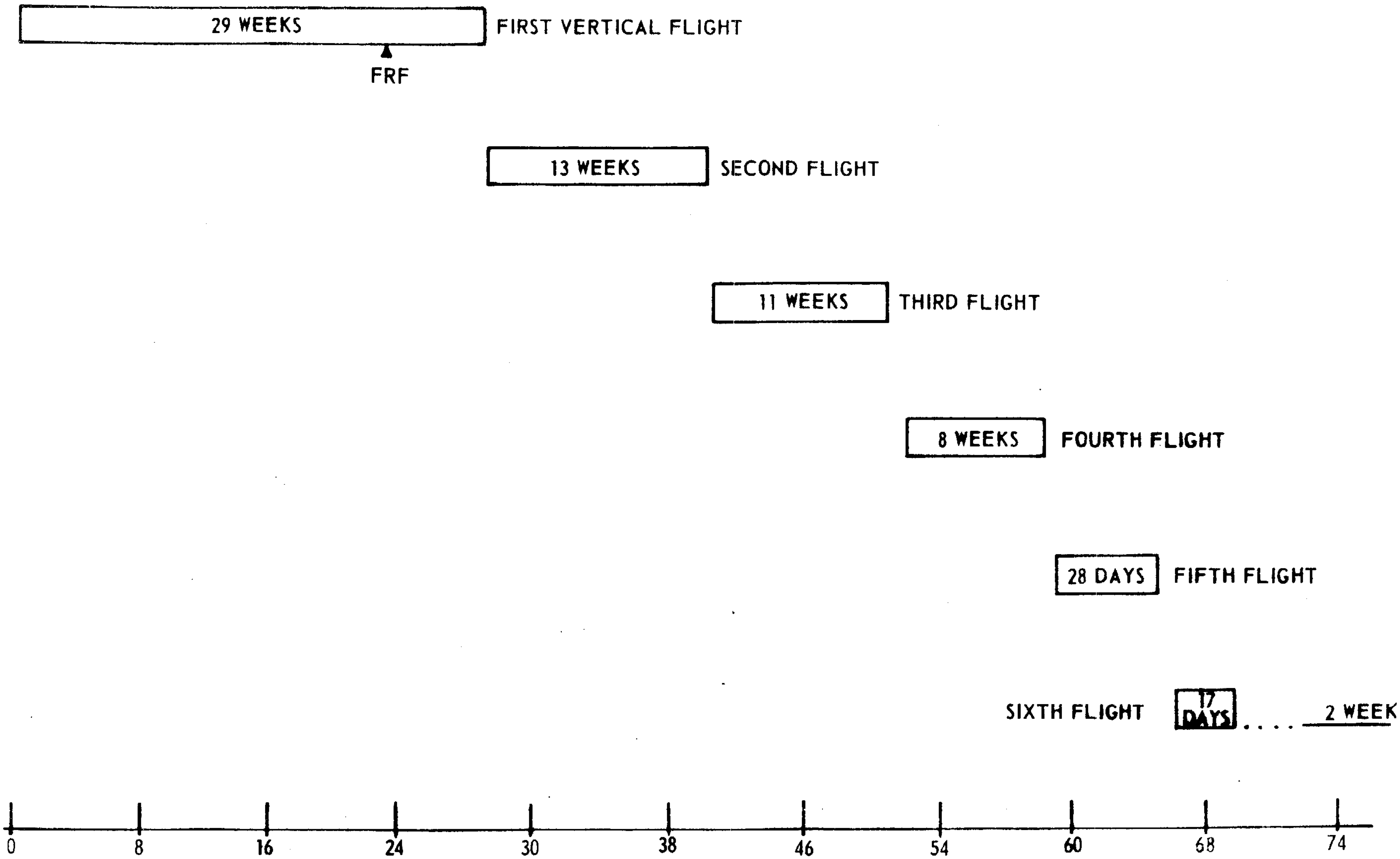

29 WEEKS
FIRST VERTICAL FLIGHT
FRF
13 WEEKS
SECOND FLIGHT
11 WEEKS
THIRD FLIGHT
8 WEEKS
FOURTH FLIGHT
28 DAYS
FIFTH FLIGHT
SIXTH FLIGHT
17 DAYS
2 WEEK
0
8
16
24
30
38
46
54
60
68
74

Figure 47

SAFETY ANALYSIS PROCESS

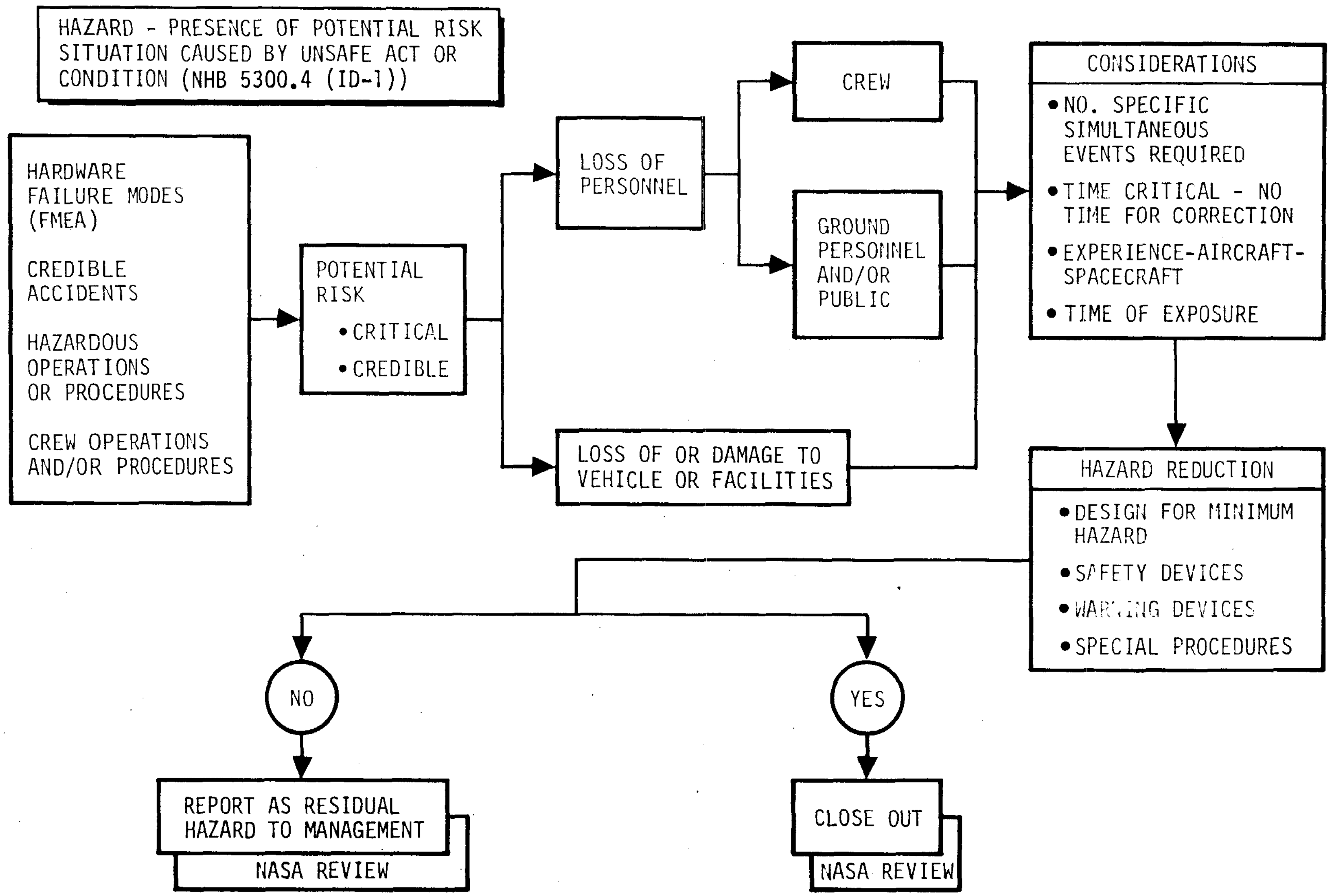
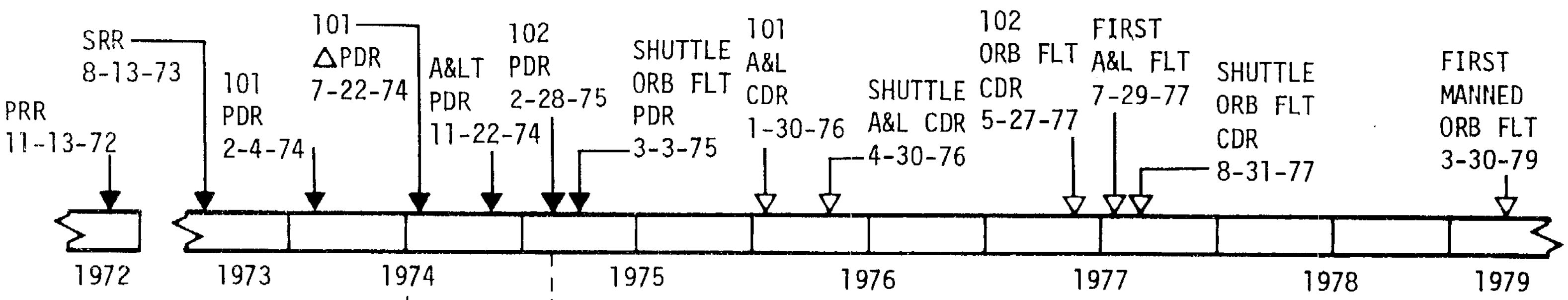


Figure 48

SHUTTLE AND ORBITER
RELIABILITY AND SAFETY ACTIVITIES



REQUIREMENTS DEFINED
- ABORT
- RESCUE
- FLUID VENTING
- FIRE/TOXICITY
- SEPARATION
- REDUNDANCY
- CERTIFICATION

FMEAs	SUBSYSTEMS - COMPONENTS - MODES -	18 850 1047	20 947 3773		FINAL ORB FMEA	MISSION RULES INPUTS
HAZARDS	SUBSYSTEMS - ORBITER HA'S/SAR TOP-DOWN SAFETY ANALYSIS	18 120	20 200 150		UPDATE	CONTINGENCY PLAN & PROCEDURES
INTERFACE ANALYSES		747/ORB	SSME/ORB ET/ORB SRB/ET	PREL	UPDATE	SITE & C/O OPERATIONS SUPPORT
SHUTTLE SAFETY ANALYSIS REPORT			▲		UPDATE	

Figure 49

GROUND HAZARDOUS GAS DETECTION SYSTEM
SUMMARY

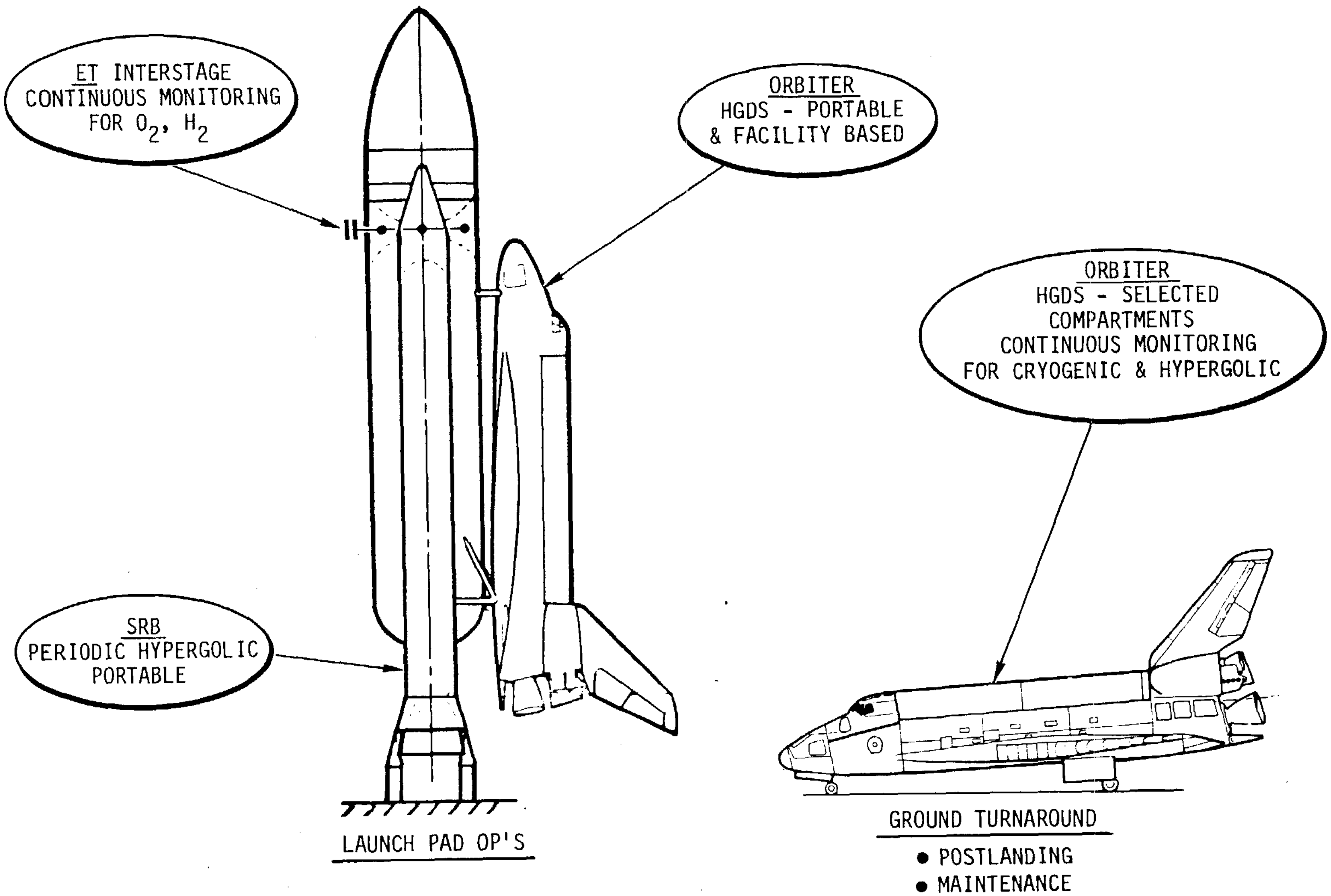


Figure 50

LANDING/DECELERATION
NOSE GEAR INSTALLATION

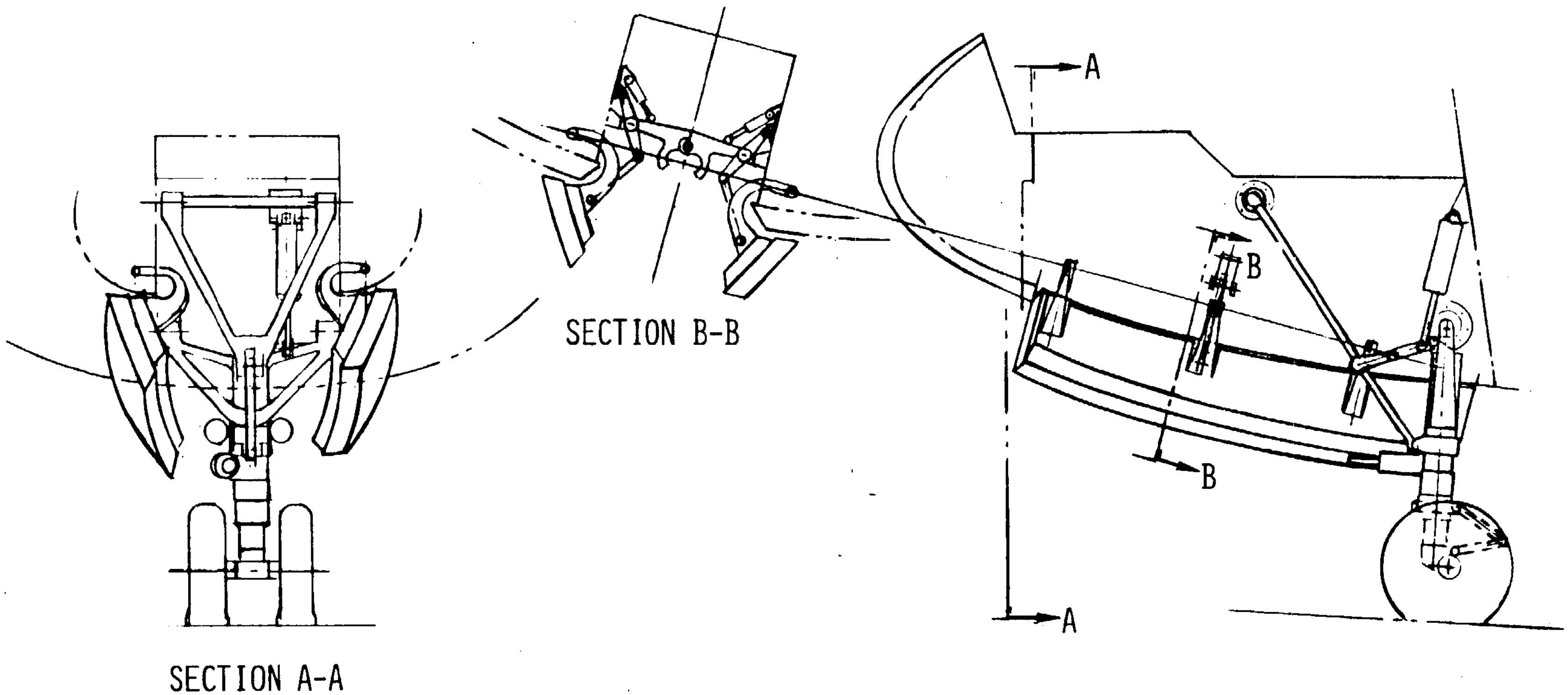


Figure 51

LANDING/DECELERATION
MAIN GEAR

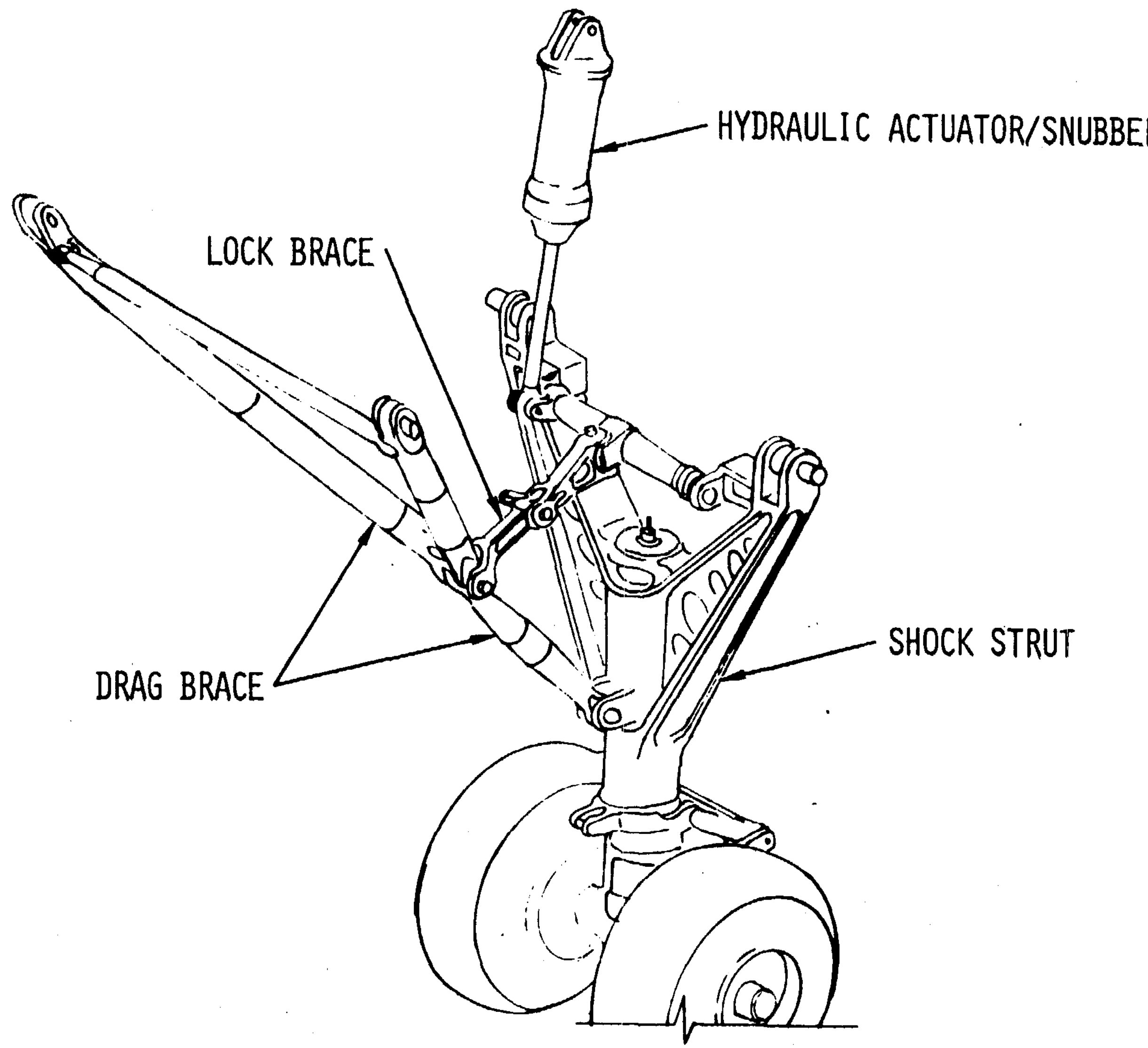


Figure 52

RANGE SAFETY
BASELINE SYSTEM

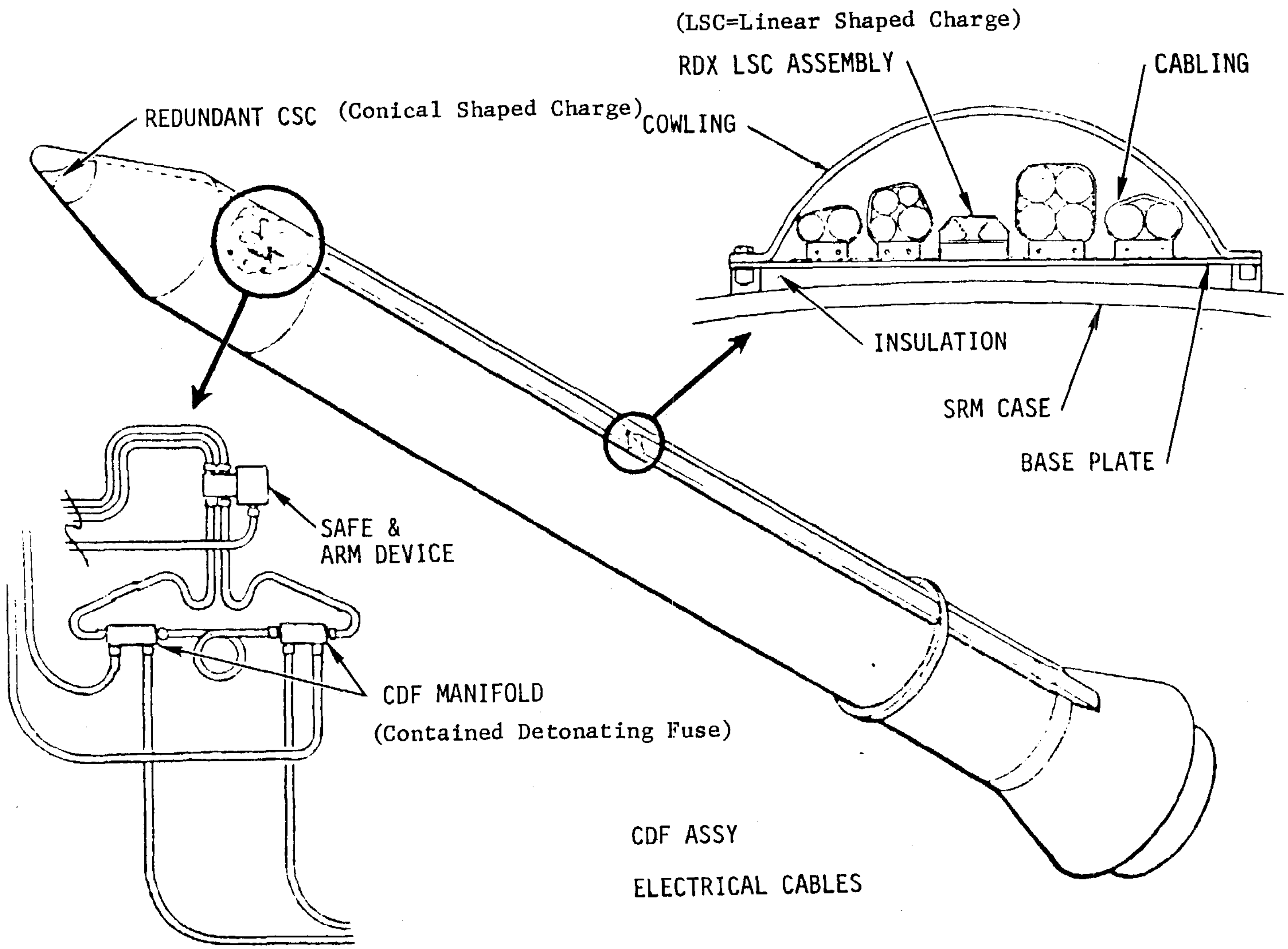


Figure 53

RANGE SAFETY
BASELINE SYSTEM

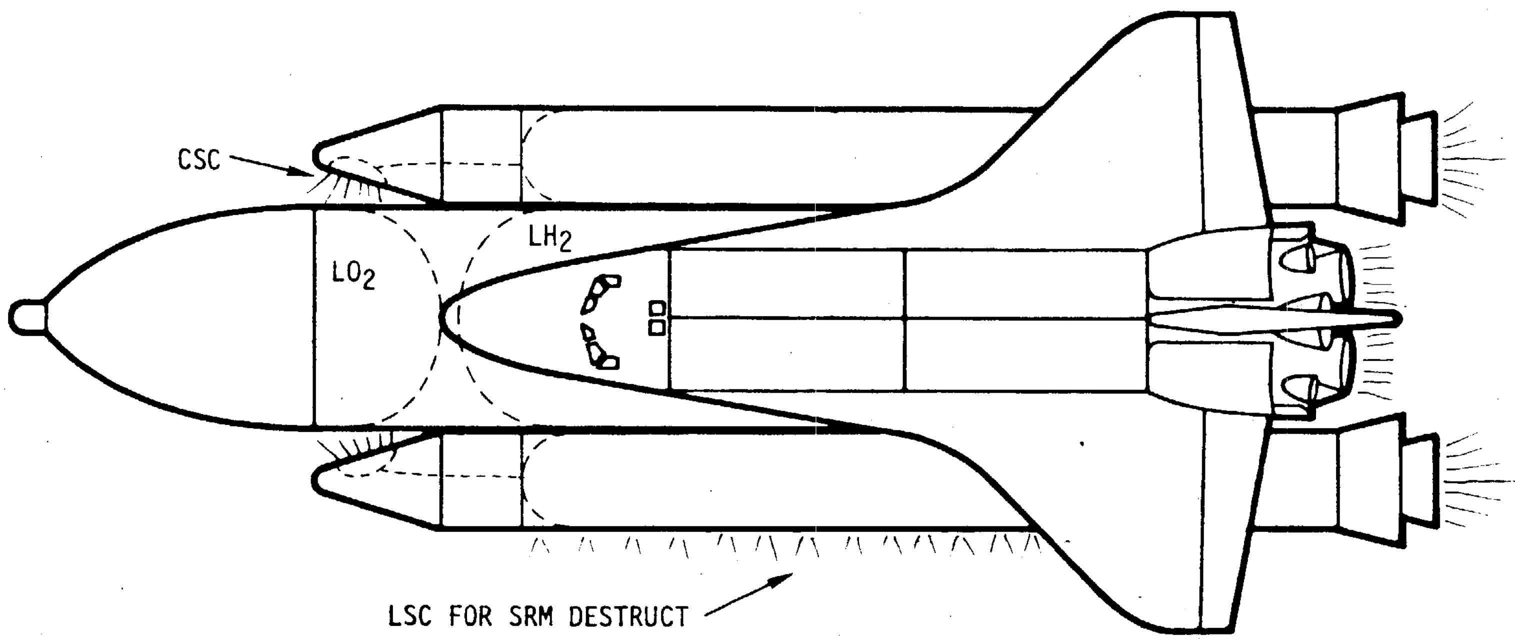


Figure 54

MAJOR GROUND TEST PROGRAMS

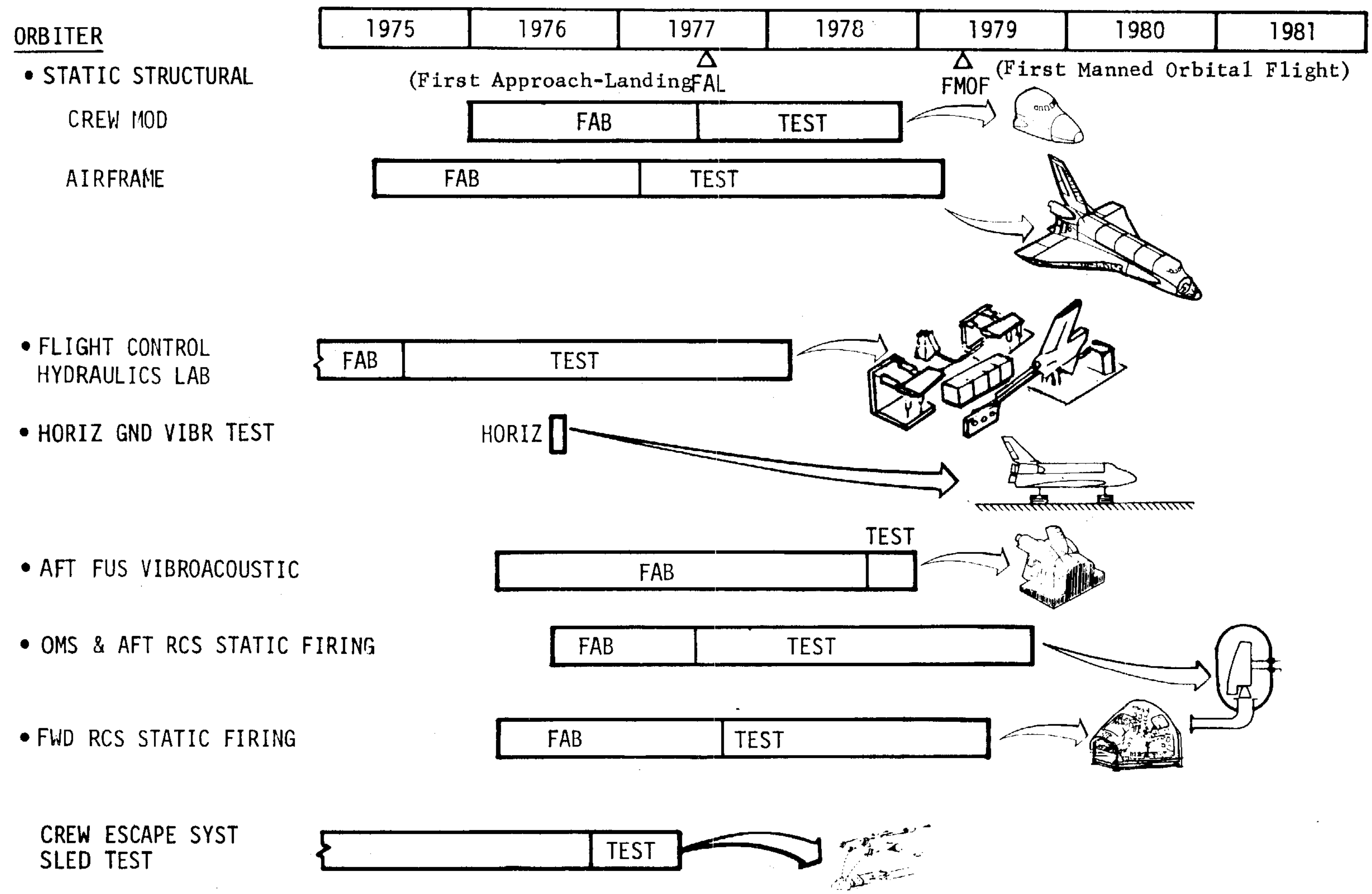


Figure 55

MAJOR GROUND TEST PROGRAMS (CONT)

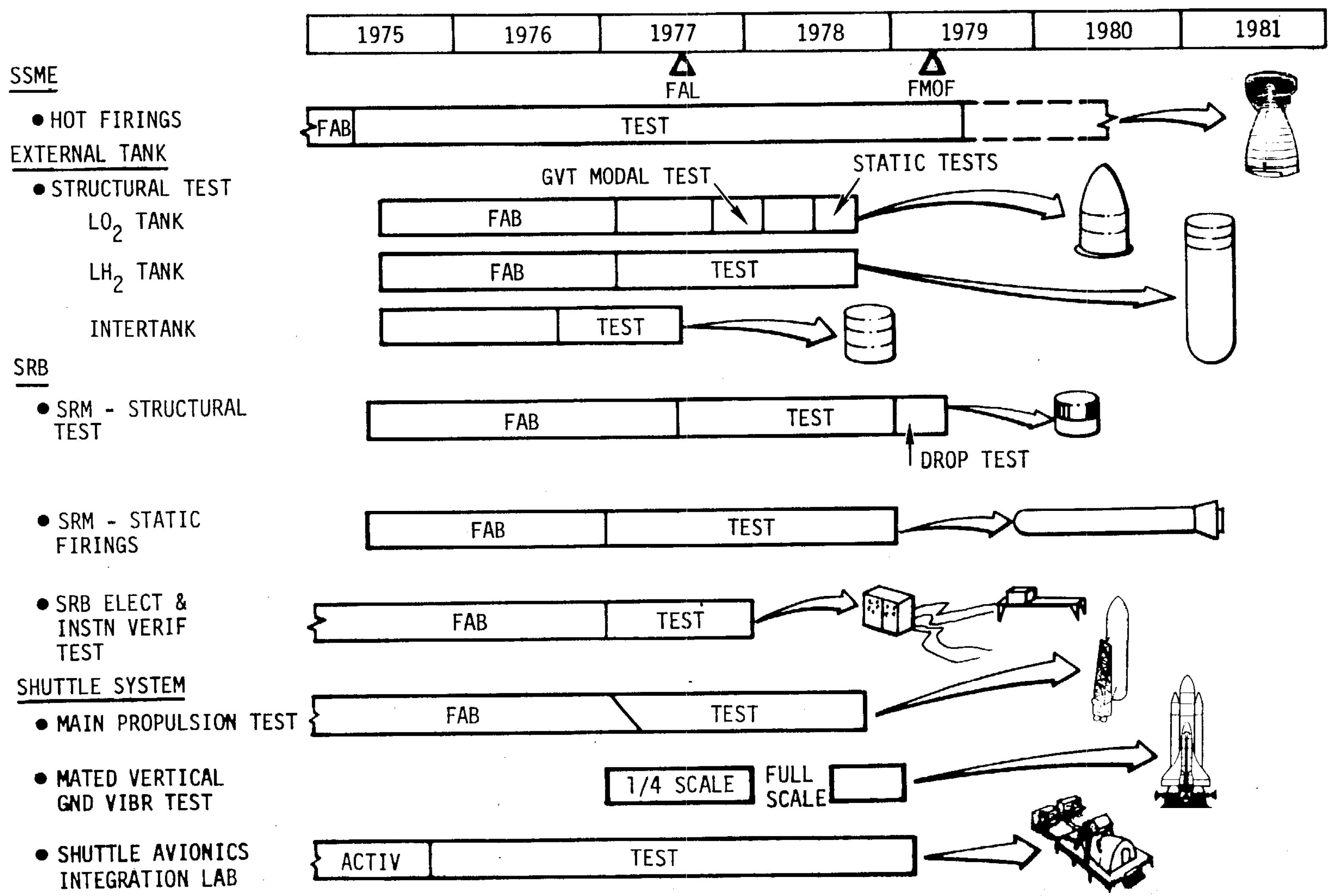
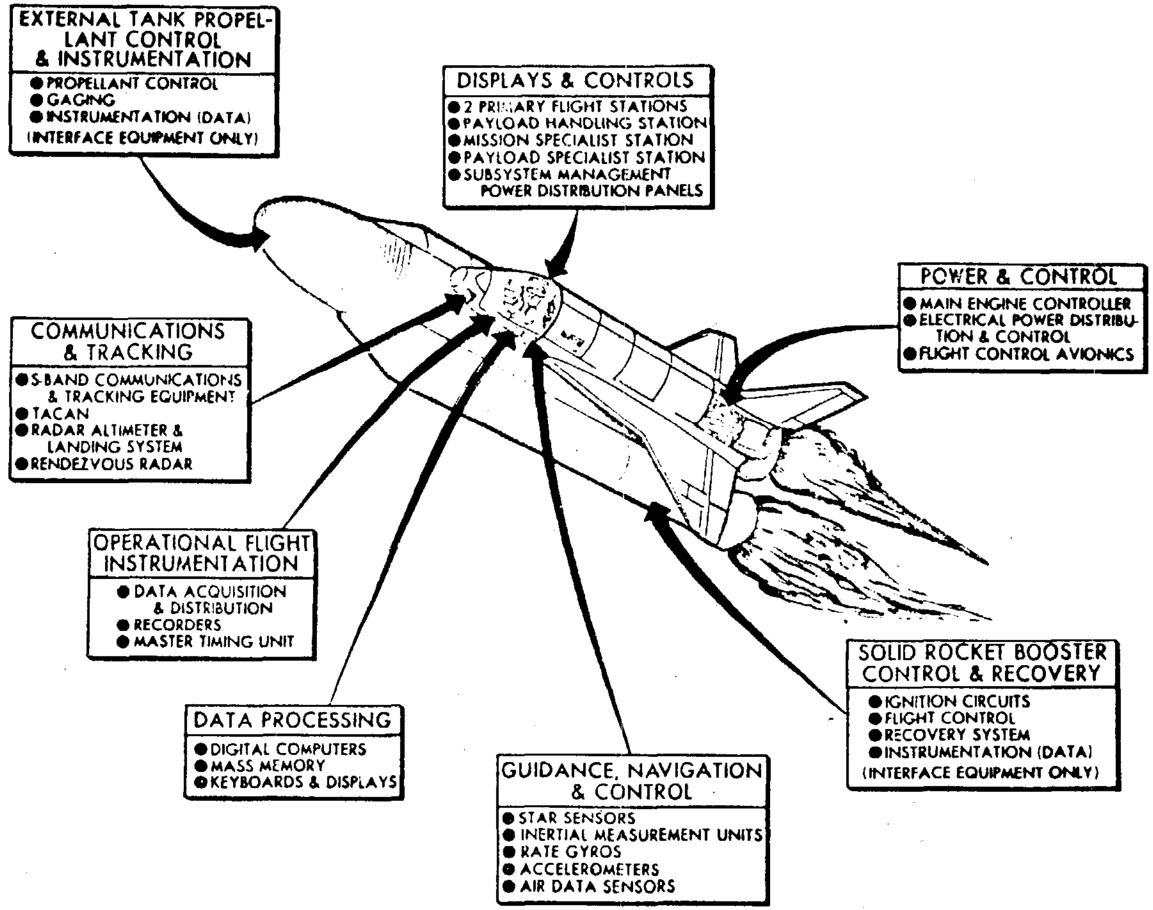


Figure 55 (Concluded)



ALL THESE SYSTEMS TO BE INTEGRATED AND TESTED IN THE SAIL

Figure 56

SRB/ET SEPARATION SYSTEM VERIFICATION LOGIC

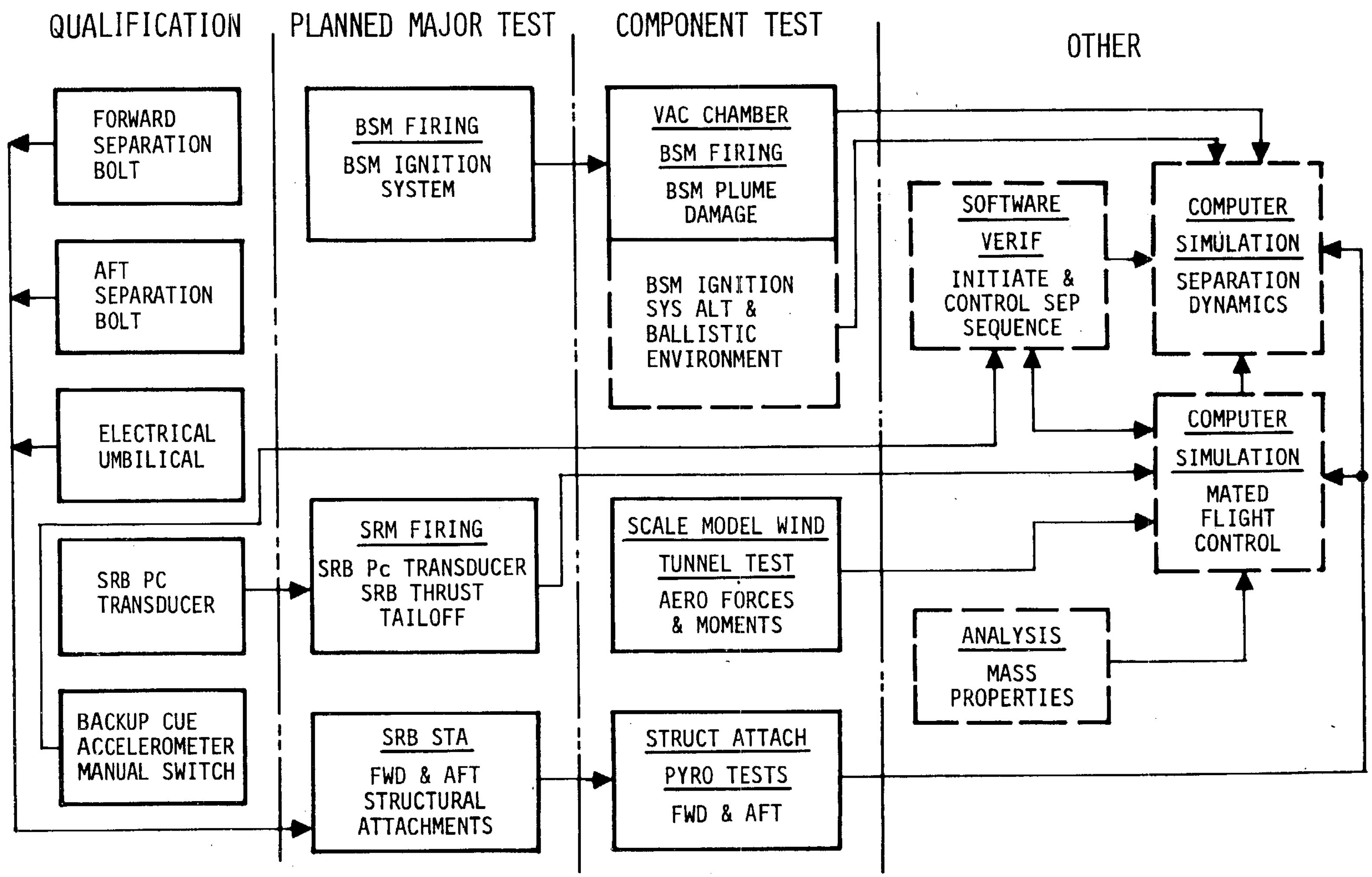


Figure 57

SPACE SHUTTLE MAIN PROPULSION SYSTEM
(SCHEMATIC)

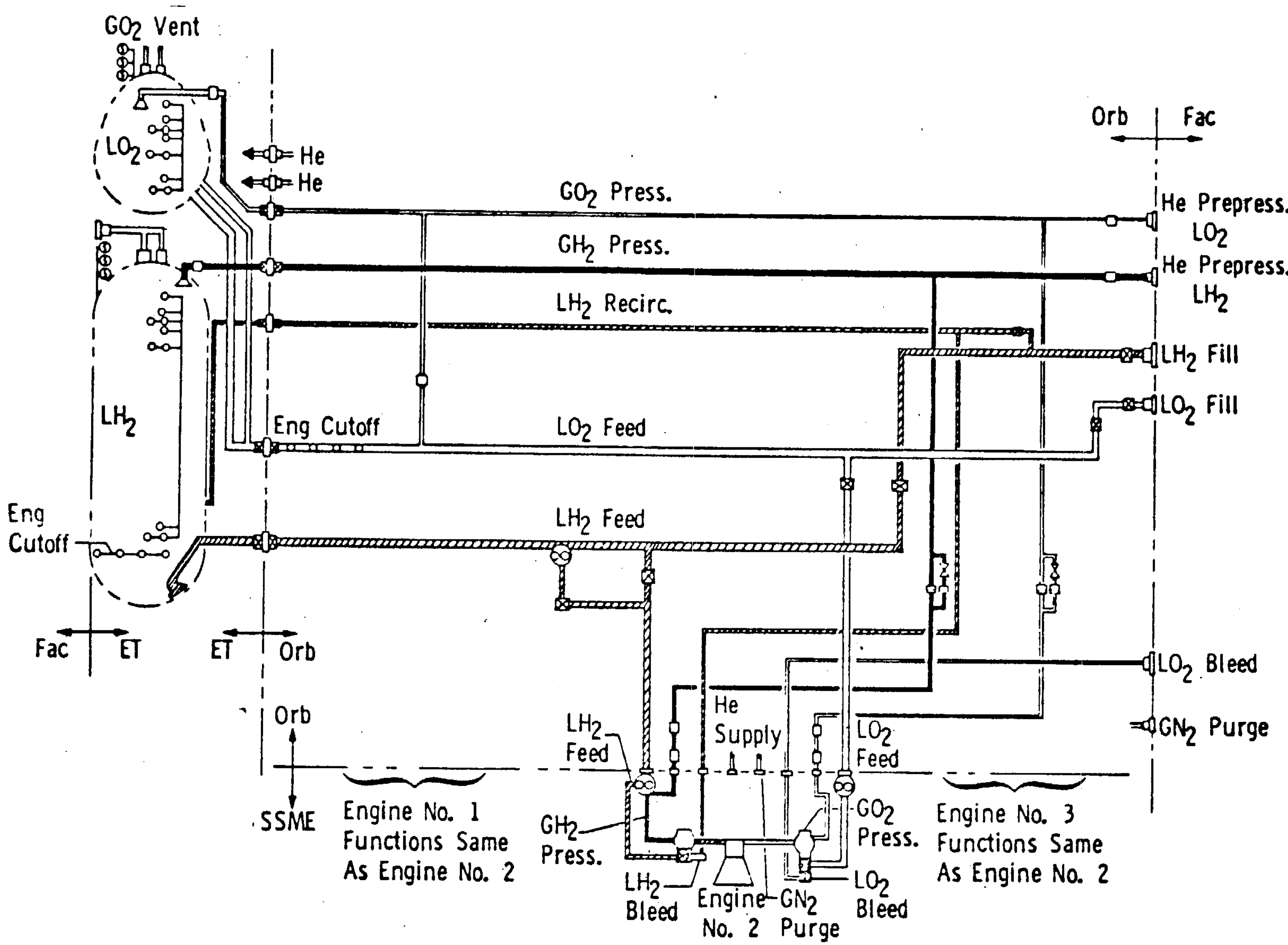


Figure 58

POGO SUPPRESSOR SYSTEM

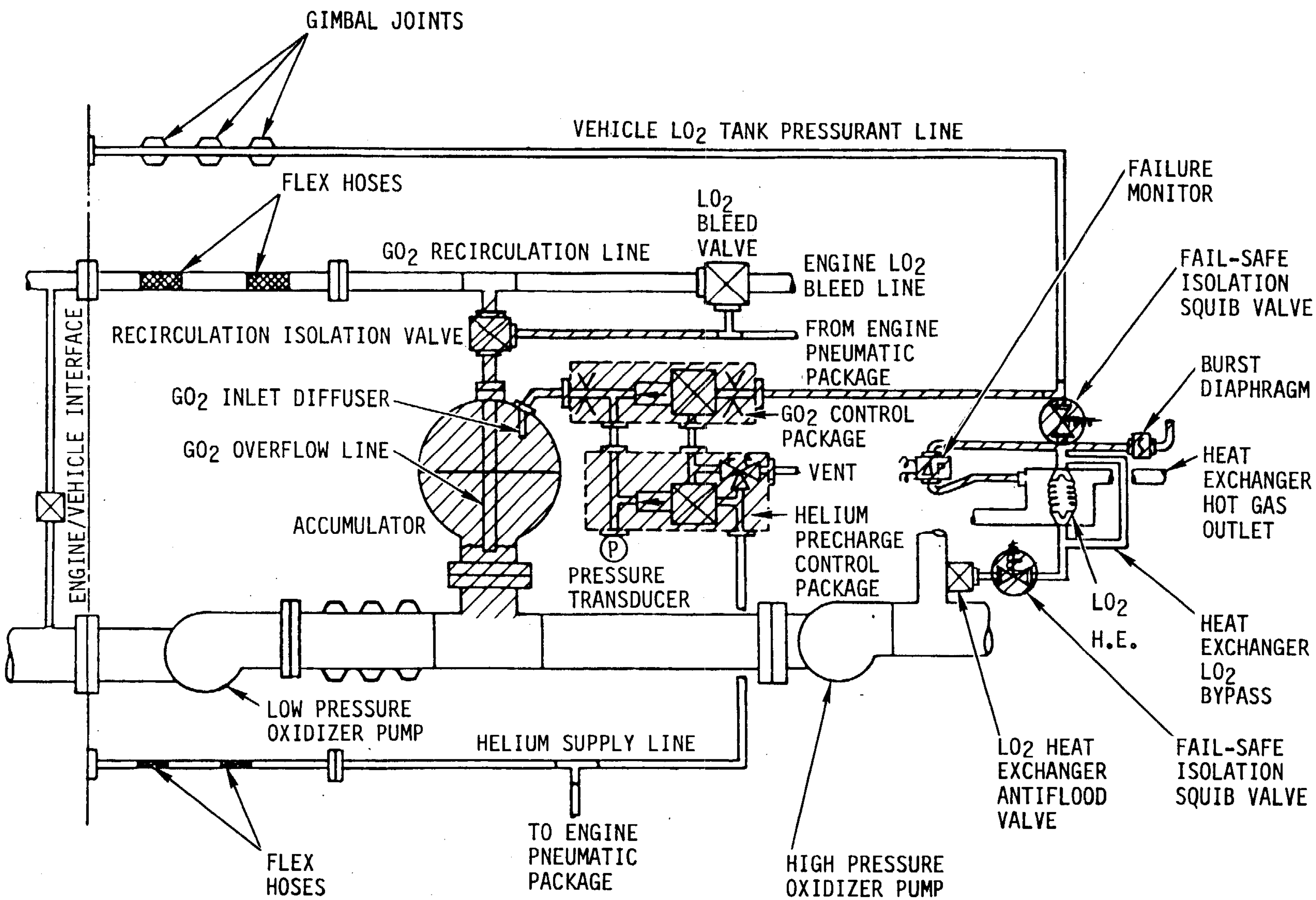


Figure 59

POGO MANAGEMENT SYSTEM

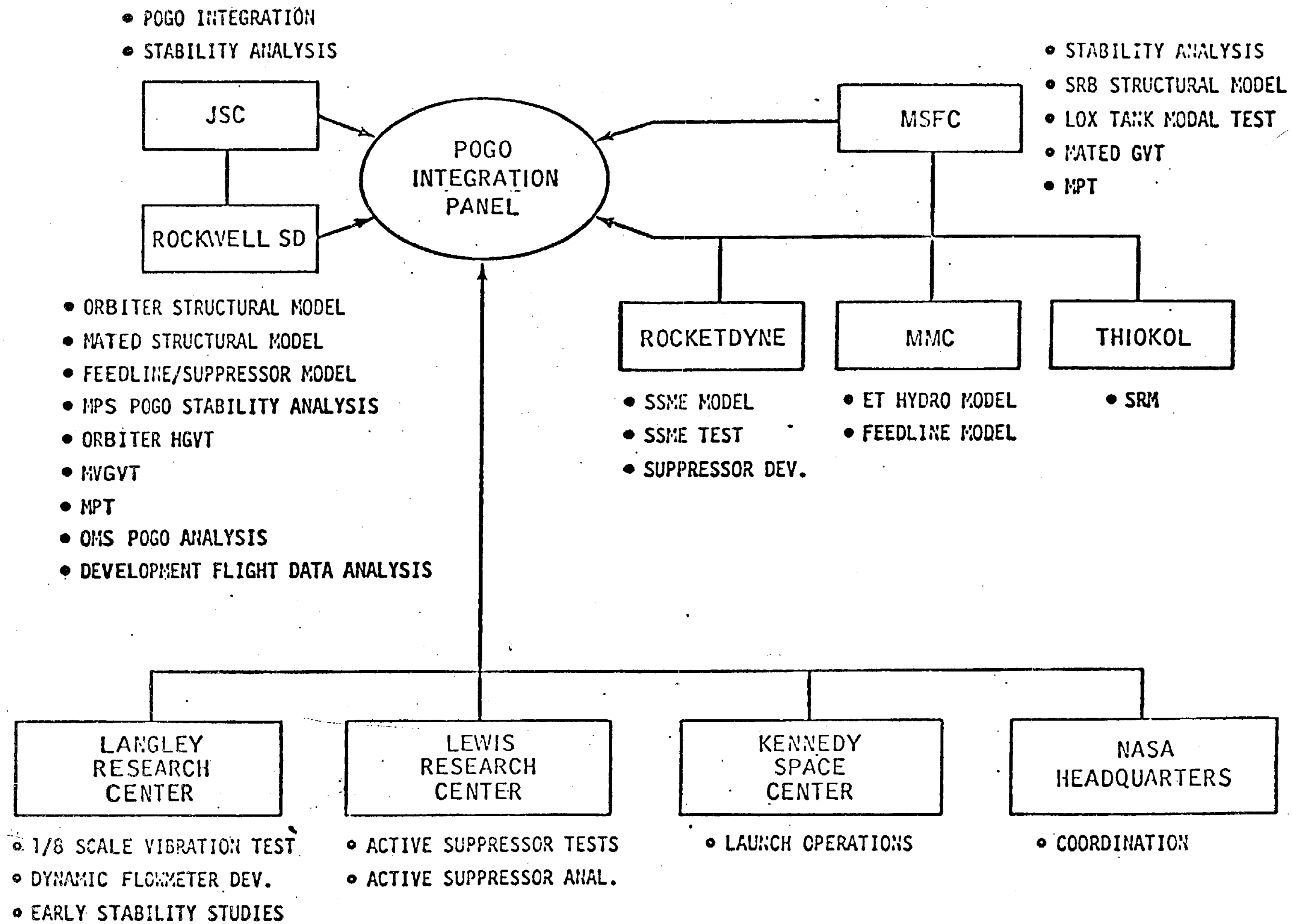


Figure 60