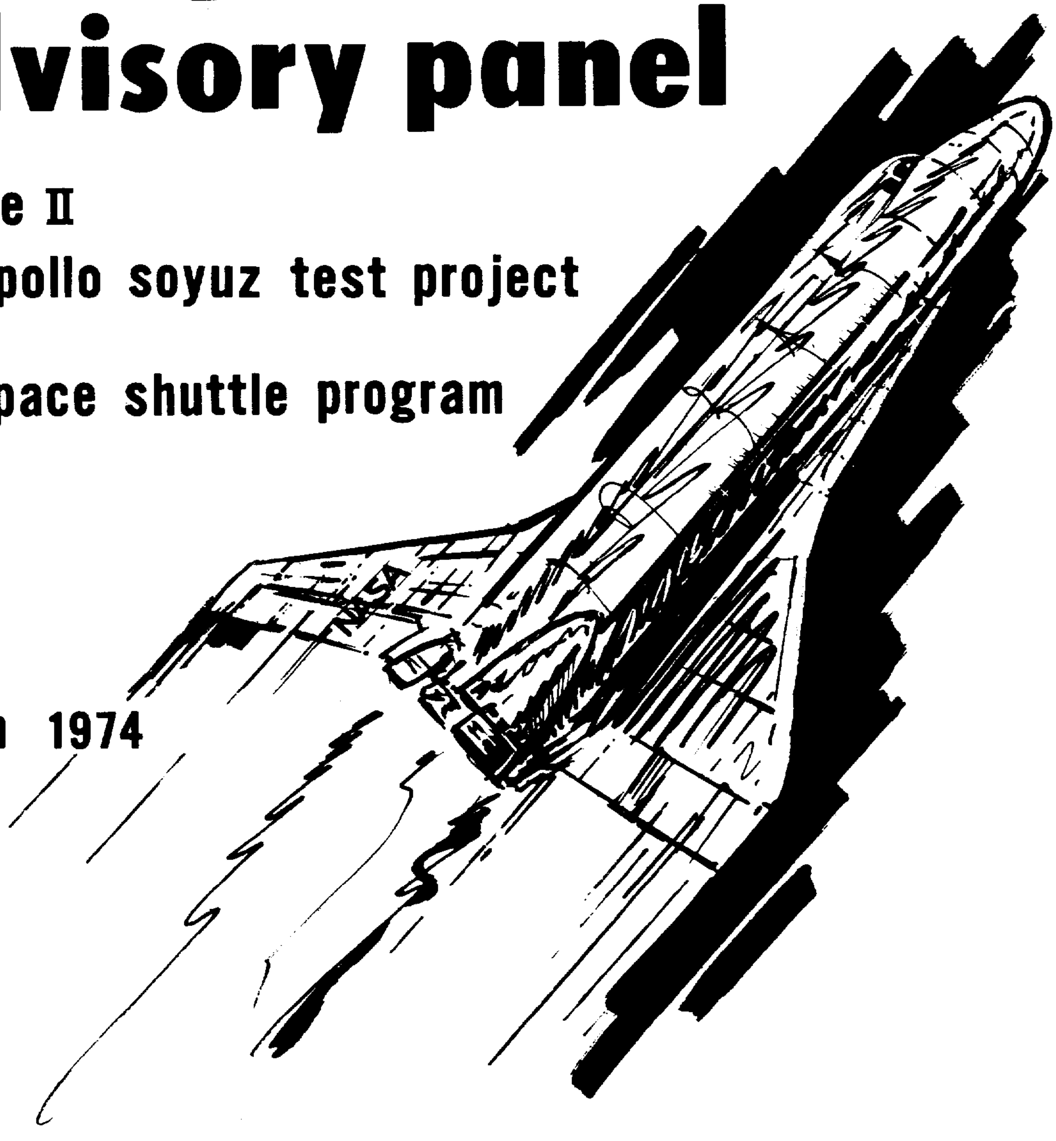
annual report to the nasa administrator by the aerospace safety advisory panel

volume II

the apollo soyuz test project and the space shuttle program



march 1974

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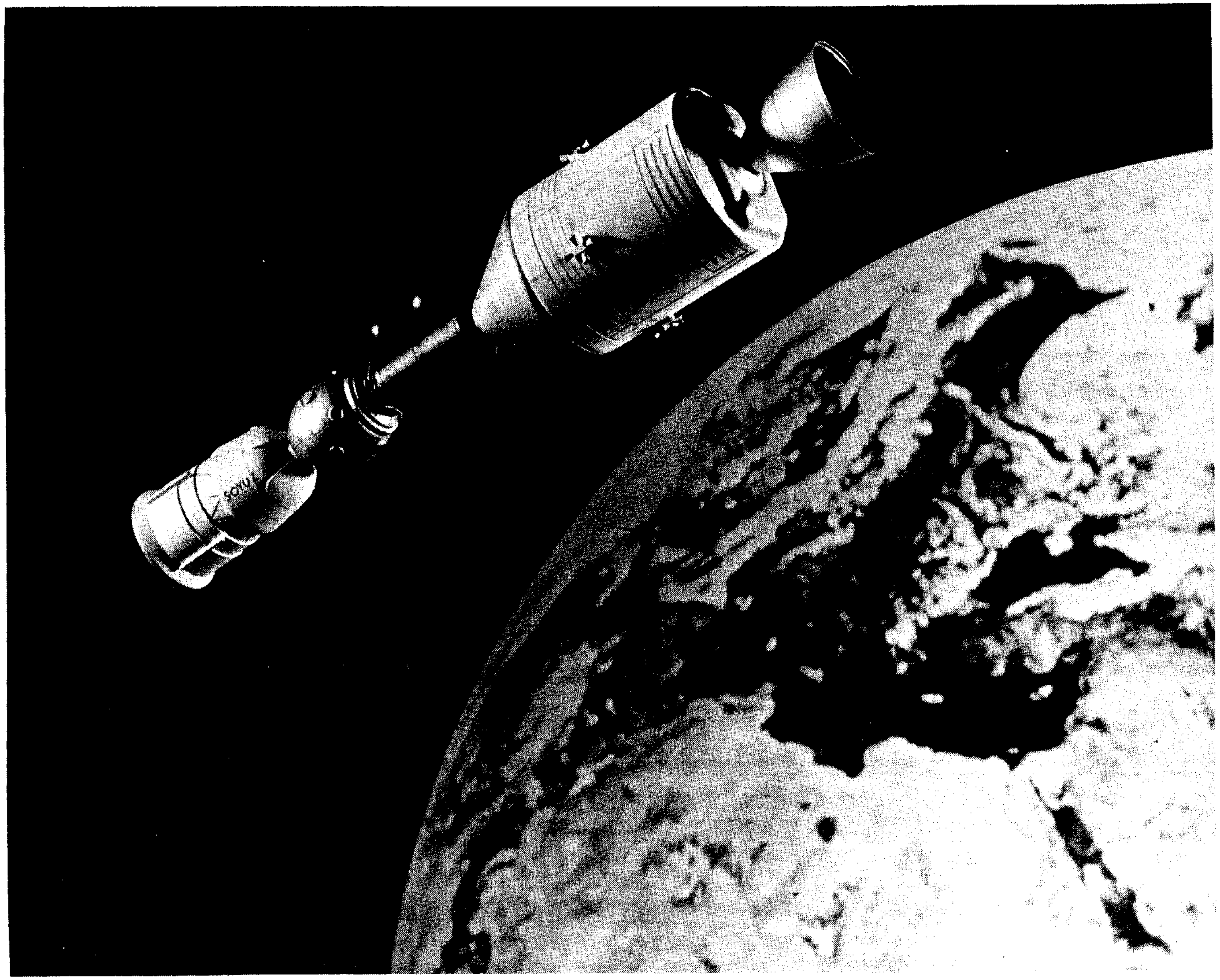
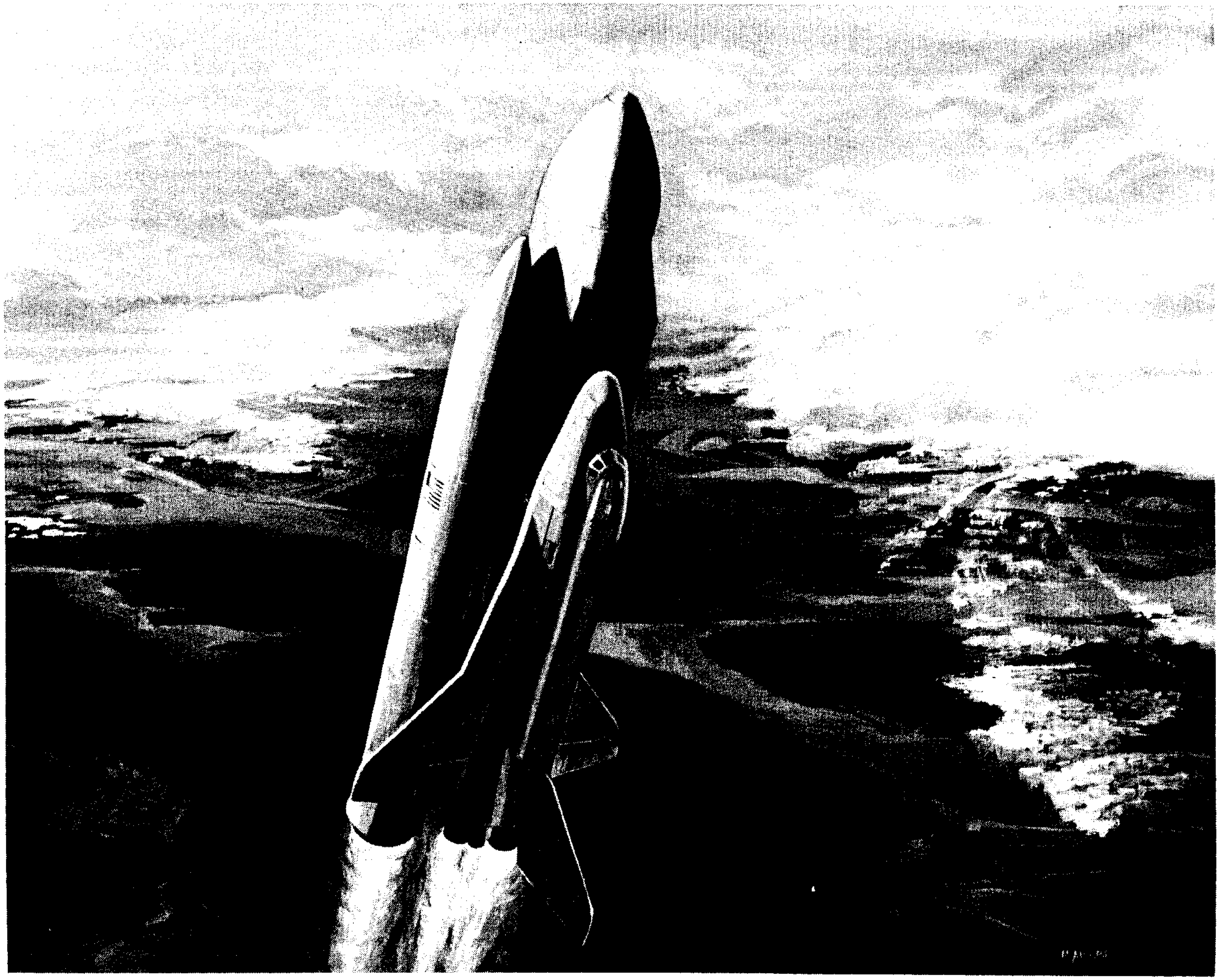


Apollo Soyuz Test Project



Space Shuttle System

ANNUAL REPORT TO THE NASA ADMINISTRATOR

by the

AEROSPACE SAFETY ADVISORY PANEL

VOLUME II - THE APOLLO SOYUZ TEST PROJECT AND THE SPACE SHUTTLE PROGRAM

March 1974

PREFACE

This volume (II) recognizes the need for specific background information and supporting details to round out and extend the data provided in volume I of this report by the Aerospace Safety Advisory Panel. There is no apparent need for further details on the Skylab program in this volume as it is drawing to its successful conclusion. The initiation of Panel reviews of the Apollo Soyuz Test Project and the Space Shuttle program does require this supporting material, which provides management and technical concepts and design evolution necessary to a fuller understanding of the significant findings and recommendations made in volume I. In addition, volume II will be utilized by the Panel in further reviews during the coming year because it becomes both a reference manual as well as an indicator of items that should be covered in those future agendas.

CONTENTS

ABBREVIATIONS AND DEFINITIONS

ABE	Air Breathing Engine
AOA	Abort Once Around
ASTP	Apollo Soyuz Test Project
CDR	Critical Design Review
CSM	Command and Service Module
DM	Docking Module
ECLSS	Environmental Control and Life Support System
ECS	Environmental Control System
ET	External Tank
ETR	Eastern Test Range
EVA	Extravehicular Activity
FCS	Flight Control Subsystem
FHF	First Horizontal Flight
FHT	First Horizontal Test
FMOF	First Manned Orbital Flight
FRR	Flight Readiness Review
FVF	First Vertical Flight
GFE	Government Furnished Equipment
GN&CS	Guidance, Navigation, and Control Subsystem
GSE	Ground Support Equipment
GT	Preflight Test and/or Ground Checkout
GVT	Ground Vibration Tests
HFT	Horizontal Flight Test
HRSI	High-Temperature Reusable Surface Insulation
ICD	Interface Control Document
IMU	Inertial Measurement Unit
IU	Instrument Unit
JSC	Johnson Space Center
KSC	Kennedy Space Center
LRSI	Low-Temperature Reusable Surface Insulation
LRU	Line Replaceable Unit
MECO	Main Engine Cutoff
MPS	Main Propulsion System
MSFN	Manned Space Flight Network
nm	Nautical Mile
OMS	Orbital Maneuvering System
PDR	Preliminary Design Review
RCS	Reaction Control System
RSI	Reusable Surface Insulation
RTLS	Return to Launch Site
SAIL	Shuttle Avionics Integration Laboratory
SRB	Solid Rocket Booster
SRM	Solid Rocket Motor
SSME	Space Shuttle Main Engine
STDN	Space Tracking and Data Network
TAEM	Terminal Area Energy Management
TCS	Thermal Control System
TPS	Thermal Protection System
TVC	Thrust Vector Control

Fail Operational	The ability to sustain a failure and retain full operational capability for safe mission continuation.
Fail Safe	The ability to sustain a failure without damage to the hardware or injury to the crew.

APOLLO SOYUZ TEST PROJECT

The ASTP was initiated some three years ago and applies the experience garnered on the Apollo and Skylab programs. In the past year considerable progress has been made in the U.S.A. and in the USSR toward developing the hardware and documenting the details of mission planning. It is expected that future manned vehicles of both countries will be designed with compatible equipment for rendezvous and docking. This enhances the safety of astronauts and cosmonauts in earth orbit, and also permits the consideration of planned cooperative exercises in space. In this context, the Apollo Soyuz Test Project is an early test of designs of the equipment needed to achieve this goal.

Program plans call for testing a rendezvous system in orbit, testing of universal docking assemblies, verifying techniques for transfer of astronauts and cosmonauts, performing experiments and other appropriate joint crew activities while docked in flight. This, then, is viewed as a possible precursor of meaningful cooperation in space between nations of the world.

The Panel began its reviews of the ASTP and Space Shuttle program at about the same time. In order to get "on board" the Shuttle program during its early and crucial requirements definition period the Panel had to give the majority of its time to fact-finding activities associated with Shuttle rather than ASTP. The Panel will be examining ASTP in more detail in 1974 when launch vehicle modifications are complete, spacecraft test and checkout takes place, and the KSC preparations for launching the vehicle go into high gear. This appeared to be a suitable approach because the ASTP system utilizes time-tested components and management systems and the schedule is such that there is little crowding of the work load in manufacturing, testing, and modification.

Schedules

The major milestones for the program are shown in Figure 1. A more detailed schedule indicating the major development, qualification, and fabrication time spans for flight hardware, ground tests, trainers, and simulators is shown in Figures 2 and 3. ASTP activities that are of interest in 1974 include: (1) Docking system qualification testing in the facilities at RI and JSC, which includes the joint USA-USSR system as well, (2) Docking Module Thermal Vacuum testing at JSC and checkout and testing at KSC, (3) Experiment modifications to CSM 111, (4) Modifications to CSM 119 as a backup vehicle, (5) Modifications and qualification of the Saturn IB launch vehicle at KSC, and (6) Internal reviews dealing with test results and mission operations documentation. The current posture of ASTP appears to show no schedule problems now or in the future. The mid-1975 launch schedule is not a constraint on the program.

Organization

The Apollo Soyuz Test Project's organization is staffed essentially by the same team of designers, engineers, and technicians who managed, designed, and built the Apollo and Skylab manned spacecraft. As far as the Saturn IB launch vehicle is concerned it had been used throughout the Skylab program. There is a great depth of experience retained at Marshall Space Flight Center which should provide ample support to meet the Apollo launch in 1975. In addition there appears to be sufficient work available on the Space Shuttle to support the retention of personnel and necessary skills at the Space Division, Rockwell International Corporation in Downey, California, and at KSC. This joint operation between the USA and USSR has required a somewhat different approach to the management of interfaces. For instance, both parties have agreed on management documents for the project as well as designated personnel in the flight operations and management disciplines. These key personnel make regular and direct contact through communication links and visits as required. There appears to be a solid foundation based on demonstrated program management for the ASTP organization and personnel.

Project Management

The management systems developed on the Apollo and Skylab programs have been modified as necessary to meet the uniqueness of the Apollo Soyuz Test Project. It was apparent at the outset that it would be necessary to provide a means for engineers and scientists from the USA and USSR to discuss the technical details of their various disciplines with their counterparts. To meet this objective the ASTP management system established Working Groups within the project structure as shown in Table I, operating out of JSC. These are joint groups made up of Rockwell International Corporation personnel, NASA personnel, and members of the USSR team, all of whom are identified as official members of a given group. In addition special project documentation as shown in Figure 4 has or is being developed to meet the needs of management and engineering.

The basic approach adopted by management to meet ASTP objectives within cost, schedule, and performance constraints has been to minimize and simplify the interface between the Apollo Docking Module and CSM and the Soyuz, and then define the details of that interface so that it can be fully understood and controlled by each country. This provides maximum flexibility, once you leave the interface, for each country to utilize its own design and approach to engineering problems and design.

The small size of the project permits senior management to maintain reasonable control based on high visibility of the day-to-day operations.

The review system and test programs set forth in briefings to the Panel appear to be in consonance with the pace and requirements of the project.

Reliability, Quality, and Safety

The R, Q, and S aspects of the program are being handled by the JSC team in much the same manner as was done in Apollo and Skylab, with the exception of the necessary interfaces with the USSR spacecraft and operations. Of particular significance is the joint ASTP Safety Assessment Activity which provides for (1) the US and USSR to assess their respective safety provisions and issue jointly-signed reports, (2) specific safety requirements to be documented in jointly-signed "Interacting Equipment Documents". These documents describe the design practices, tests, and operational procedures to preclude or minimize the probability of the following adverse conditions or risks:

1. Failure or inadvertant release of a docking system structural latch,
2. Fire or smoldering materials,
3. Loss of spacecraft pressure,
4. Inadvertant firing of pyrotechnics,
5. Inadvertant actuation of any control or propulsion system,
6. Ground command from giving inadvertant signals to the spacecraft systems.

The Safety Assessment Reports for Apollo are shown in Table II and for the Soyuz spacecraft they are shown in table III. Specific reports directed to safety alone include those shown in Table IV. All of these documents are scheduled for completion by the end of the first quarter of 1974.

The Panel intends to examine the Reliability, Quality, and Safety areas in more detail during the 1974 calendar year.

Baseline System Description

The Apollo Soyuz system consists of the flight hardware, ground support equipment and facilities, and mission support, and includes the necessary software in these areas. The flight hardware includes the spacecraft under launch vehicles of the US and USSR. Mission operations is covered under a separate section of this report. This deals with the mission profile, rules of the mission, contingency plans, crew operations while in flight, crew training (ground and flight), simulation exercises, and the documentation in support of these areas.

The launch configuration of the US portion of the system, called Apollo, consists of the following major modules or elements:

1. The Saturn IB Launch Vehicle made up of the two stages of propulsion (S-IB and S-IVB), the Instrument Unit (IU).
2. The Launch Escape Tower (LES) and the Spacecraft Launch Adapter (SLA).

3. The Apollo Command and Service Module (CSM).
4. The new Docking Module (DM).
5. On-board experiments.

The USSR or Soyuz portion of the combined mission hardware includes:

1. The standard Soyuz Launch Vehicle.
2. The Soyuz Spacecraft modified to meet the requirements for operation with the CSM/DM spacecraft.
3. On-board experiments.

The Apollo-Soyuz spacecraft as they will appear in orbit are shown in Figure 5 and the Apollo Launch Configuration is shown in Figure 6.

In a number of areas it has been necessary to achievement agreement between the US and the USSR on those items that must be compatible to achieve a successful combined mission. The major items of compatibility for the joint mission, which act as design "drivers" as well, are: mission operations, control center communications, spacecraft radio frequency for communications and ranging, docking systems, internal hardline or cable communications, life support systems and methods of crew transfer. The compatible docking system is the most obvious of the hardware drivers and has resulted in the design of a new"petal" system in lieu of the Apollo and Skylab system. In the descriptions given below the impact of the "compatibility" requirement can be seen.

Command and Service Module

The Apollo spacecraft for ASTP is a modified version of the CSM flown during the Lunar Landing mission and the Skylab missions. The CSM is number 111 which was manufactured and checked out for the Apollo program and then placed in bonded storage. Modification design, development, test and engineering work is being performed by the Space Division, Rockwell International Corporation at Downey, California.

The CSM will weigh about 35,000 pounds at lift-off which includes the basic dry weight of the spacecraft and the propellants and other consumables. The major CSM modifications being made to fit the ASTP mission include:

1. Modify the umbilical connection at the forward (apex) end of the Command Module to accommodate the Docking Module electrical functions.
2. Modify the controls and displays in the Command Module to accommodate the Propellant Storage Module placed in the Service Module, the Docking Module control and monitoring requirements, and the actions of the Docking system which interfaces with the Soyuz spacecraft.
3. Add a Skylab Television recorder for the coverage of joint activities.
4. Add Skylab lockers to aft end of the CM.
5. Add experiment controls.

The type of experiments to be carried in the Command Module and the Service Module will (1) provide data which can be furnished to scientists in each country, (2) provide for co-investigators located in each country, (3) require active cooperation of USSR cosmonauts, and (4) use existing hardware where possible. Table III contains a listing of the experiments.

The communications used for the ASTP requires the addition of equipment to meet the links shown in Figure 7 which uses normal ground to spacecraft links but also includes the use of the ATS-F Satellite which should be in operation by the time of the schedule ASTP mission.

The major SM modifications include:

1. Deletion of one O_2 and one H_2 tank as was the case for the Skylab units.
2. Addition of a Propellant Storage Module in the SM bay 1 and manifold to all four RCS Quads. Used for attitude and back-up deorbit.
3. Delete fuel and oxidizer storage taks and one of the two helium tanks as was done for the Skylab SM.
4. Delete the "return enhancement" battery.
5. Build a structural truss to support the Docking module in the Spacecraft Adapter.
6. The RCS quads themselves have been modified by adding fuel and oxidizer accumulators, hot-rod heaters, flex line heaters, insulation blankets on support brackets.

<u>Docking Module</u>

Because the Apollo Command Module and the Soyuz Spacecraft operate at different pressures and mixtures of oxygen and nitrogen an airlock is required to allow the crew to transfer between them. The upper hatch and docking mechanism of the Apollo CM is very difficult and expensive to modify because of its close integration with the forward heat shield and earth landing system including the parachutes, separation charges and flotation gear. Therefore, the decision was made to provide a docking module to serve as an airlock and also incorporate the new universal docking system capable of functioning with identical components of the Soyuz spacecraft.

The Docking Module, Figure 8, is a cylindrical structure about 5 feet in diameter and 10 feet in length. One end of the DM incorporates a Lunar Module type drogue and hatch so that it can be mated with the Docking Probe on the forward end of the CSM. The Docking Module contains tankage and control systems for oxygen and nitrogen to allow adjusting the pressure and oxygen concentration of the atmosphere to match that of either spacecraft as well as providing for electrical power, TV connections, and communication systems (Fig. 8A). The majority of these controls and equipments are located within a pallet that is installed on the wall of the Docking Module and in fact forms a platform or floor within the module.

The forward end of the DM (furthest from the CSM and closest to the Soyuz) incorporates the new universal Docking System. This system is shown in Figure 9. It consists of two main elements: (1) the structural assembly which is rigidly attached to the Docking Module and provides the base to which the structural ring assembly is mounted containing the eight structural ring locking latches, (2) the guide ring assembly with the guide paddles and capture latches which are attached to the structural ring assembly by attenuators or shock struts and as an assembly is capable of being extended or retracted.

In operation, the active vehicle extends its guide ring assembly while the guide ring of the passive vehicle remains retracted. Either of the two docking vehicles can be the active member. Capture is made by the capture latches on the guide paddles of the active vehicle engaging the body mounted latches on the passive vehicle. The guide ring assembly on the active vehicle is then retracted. Thereby pulling the two spacecraft together until the structural ring faces meet and the latches engage to provide a hard dock and pressure seal. The question of docking interlocks is addressed in Volume I.

Atmosphere Conditions

While the CSM uses an internal environment of 5 psia with essentially a pure oxygen content, the USSR's Soyuz (which weighs about 15,000 pounds at lift-off) will use a 10 psia enviroment of nitrogen and oxygen in an airlike composition. This value of 10 psia was arrived at during negotiations in order to reduce the hazard associated with personnel leaving the Apollo's 5 psia atmosphere and going to the 14.7 psia of the Soyuz, to enhance the mission operations time-lines and to reduce associated hardware problems.

Launch Vehicle

The Saturn IB has been modified only in those areas necessary to meet the loading and trajectory requirements of ASTP, i.e., changes to the Instrument Unit computer programs, structural changes to accommodate the Docking Module, propellant loads. The use of launch Complex #39 requires the use of the special platform built and used on the Skylab program to launch the three manned CSM units. Modification, testing, and checkout of the launch vehicle is scheduled through 1974 and 1975.

Mission Operations

The ASTP mission profile is depicted in Figure 10. The timing associated with various portions of it are a function of the consumables aboard which provide the CSM with a possible 12 days stay time and the Soyuz with a 6 day stay time.

The Soyuz will be the first vehicle launched and inserted into an earth orbit of approximately 188 by 228 kilometers (101 by 123 nautical miles) then maneuvered to circularize the orbit in preparation for rendezvous at a nominal altitude of 225 kilometers (121 nautical miles). The plane of the orbit will be inclined 51.8 degree to the equator. The first Apollo launch opportunity will occur approximately 7.5 hours after the Soyuz lift-off. Four additional opportunities or launch windows of approximately 15 minutes in length exist, spaced about 24 hours apart. The fourth and fifth opportunities necessitate a reduction of the time in the docked configuration in orbit from approximately 48 hours to some 24 or 9 hours respectively. After insertion into an earth orbit of 150 by 167 kilometers (81 by 90 nautical miles) the CSM will perform the necessary maneuvers for rendezvous and docking with the Soyuz vehicle.

The Docking Module is designed to allow about 2 hours of prebreathing on pure oxygen at the 10 psia pressure level with the DM hatches closed when returning astronauts from the Soyuz to the CSM. This is to alleviate the "bends" problem when taking men from the high pressure with nitrogen in their bloodstreams back to the low pressure. The Docking Module also serves as an emergency container if personnel have to leave the Soyuz or the CSM due to spacecraft problems. The DM launch atmosphere is conditioned to 16 psia with 60/40 oxygen/nitrogen which then vents down to 11 psia during launch. This then is vented down to the proper CSM/DM or Soyuz/DM atmospheric mix when the spacecraft are opened up to the DM. The oxygen supply is adequate for four normal crew transfers or cycles with sufficient reserve to maintain the DM pressure above 3.5 psia for approximately 15 minutes with an equivalent leakage hole size of 0.5 inches.

The crew transfer guidelines that have been set up are as follows:

1. There will always be at least one host crew member in each spacecraft.
2. The number of crew will not exceed a maximum of three in the Apollo spacecraft and two in the Soyuz when the spacecraft's hatch to the Docking Module is closed.
3. Crew transfer will be accomplished while they are dressed in coveralls.
4. An astronaut will always operate the Docking Module systems, and two crewmen will always be present during a transfer operation.
5. Crewman will operate their own vehicle hatches and crewmen will sleep in their own vehicles.
6. There will be no contingency EVA tansfers.
7. Hatches are not interlocked to prevent undocking with the hatch open in the DM or Spacecraft. However hatches will be open only during transfer and for one hour on first night of crew transfer. This are is covered further in Appendix G of Volume I. The first transfer operation as currently envisioned is shown in Figure 11.

Real time operations of the ASTP mission have been "ground ruled" for thorough understanding by all concerned. These ground rules are as follows:

1. Each spacecraft will be controlled by the respective control center.

2. Consultations between control centers will be held for decisions affecting joint elements. The exchange of control center personnel is still under discusion and will hopefully be agreed to in the near future.

3. Joint activities will be conducted according to mission documentation, including contingency plans.

4. Pre-planned exchanges of data information, tracking information, voice information will be performed on a scheduled basis and on an "as-required" basis for specific problems that might come up.

5. The host country will have the primary responsibility for deciding appropriate action for a given situation in the host vehicle when a visitor is present.

6. Television will be immediately transferred to the other control center.

7. As a minimum, flight crew will be trained in the other language well enough to respond as appropriate to established voice communications. This will emphasize (a) conversational capability, (b) operational jargon. Tapes will be available for study by the crewmen so that they can condition themselves to the expected conversations.

8. Certain failures of one spacecraft that could affect the other (e.g., thruster firing) are being identified and remedial action planned (e.g., shutting off the system). Provisions for "surprises" have to be worked out to assure the ability for fast response.

Program Challenges

The Panel will increase their review of the ASTP in the coming year and based on data to this time the focus will be on such areas as the following program challenges:

1. The management of the program (design, development, test and flight) depends upon the continued working relationships now established being maintained and perhaps enhanced as flight time approaches.

2. Provide for and carry out the necessary formal reviews to assure that the ASTP systems are compatible, meet requirements and cover all foreseeable hazards. These joint USA/USSR reviews include: design acceptance reviews, equipment acceptance reviews, mission plan and operations reviews, launch schedule status reviews and the flight readines review.

3. Review of test results to assure that ground and flight hardware are operationally ready-to- go. This requires review of the development and compatibility tests, qualification/verification tests, pre-flight test and checkout, and Control Center tests.

4. Language training

5. Failure Modes and Effects Analysis (FMEA) results for all flight systems on the DM and CSM

6. The Mission Control Center interaction plan to assure proper flow of data and instructions between both US and USSR control centers

7. Mission simulations to assure that flight crews can meet flight situations.

8. Failures or inadvertant operations on one spacecraft that can affect the other spaceraft.

9. Communications coverage at the present time appears to be below a desireable or attainable level. Air to ground communications are available to the US for about 18% of the time in orbit. Consideration of current or projected orbital satellites for increasing the coverage to some 50% is of importance.

10. Sneak circuit analyses and fault current analyses should be accomplished to assure that electrical hazards are minimized.

11. Contingency planning if the ATS-F satellite is not in operation at the time of the Apollo-Soyuz mission.

SPACE SHUTTLE PROGRAM

The Space Shuttle program will provide a transporation capability that is planned to substantially reduce the cost of space operations and support a wide range of scientific, commercial, and defense uses. In support of these goals the Panel initiated an intensive review of the program in September 1973 and the overview presented here will provide an insight into the management, hardware design and development, testing, and preparation for flight operations. Included are areas of interest to the Panel for future reviews because they are either a current or projected concern, challenge or real-life problem.

Management Overview

The program consists of two phases: (1) system development and production and (2) Shuttle operations. The basic elements of the first phase, which the Panel will be reviewing over the next few years, consists of those program and hardware elements and efforts necessary to provide the mission capability required of the Space Shuttle System. These elements, twelve of them, are listed below:

1. Management
   a. System Management
   b. System Engineering and Integration
2. Hardware
   a. Orbiter Vehicle
   b. Space Shuttle Main Engine (SSME)
   c. External Tank (ET)
   d. Solid Rocket Booster (SRB)
   e. Air Breathing Engines (ABE)*
3. Support Services and Hardware
   a. Flight Test
   b. Orbiter and Integration System Support
   c. Launch and Landing Support During Flight Tests
   d. Payload Support
   e. NASA/Government Agency Support

The details of Program Definition and Requirements are found in the multi-volume document issued by JSC (JSC 07700, "Level II Program Definition and Requirements"). This excerpt from the Forward to Volume I of these documents states reasons for them and what they contain:

> "Efficient management of the Space Shuttle Program dictates that effective controls of program activities be established. To provide a basis for program management; requirements, directives, procedures, interface agreements, and information regarding system capabilities will be documented, baselined, and subsequently controlled by the proper management level."

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*A recent decision has been made to remove the air-breathing engines and to utilize a modified C-5A or 747 aircraft to carry the Orbiter in a "piggy-back" configuration.

The NASA Associate Administrator for Manned Space Flight has assigned the Johnson Space Center (JSC) as the Space Shuttle program lead center. Thus the Program Office and the multicenter systems integration group located at JSC is an arm of NASA Headquarters in carrying out the program. The Space Shuttle Program Office in Washington is responsible for generating the overall systems performance, schedules, and resource control. This means that JSC has the delegated authority for the day-to-day management of the program, and carries out the integration studies previously done in Headquarters for the Apollo program. In addition JSC contracts directly with the industry teams that will produce the Shuttle. The Program Manager at JSC, in turn, utilizes the technical and managerial skills of each Manned Space Flight Field Center to carry out those functions and activities in their special area of expertise. JSC has the responsibility for the design and development of the Shuttle Orbiter Vehicle. The Marshall Space Flight Center is responsible for design and development of the solid rocket booster, Space Shuttlė main engine, and the external tank elements of the Shuttle system. The project managers for each of these systems elements, on program matters, reports directly to the JSC Space Shuttle program manager. The Kennedy Space Center is responsible for design and development of launch, landing, and refurbishment operations and attendant facilities. Regularly scheduled meetings, attended by all responsible elements of the management teams, including the participating contractors, are arranged to identify interface issues and programmatic problems. Periodic reviews with top NASA management are also an important aspect of the lead center plan. The Space Shuttle management relationships are shown in Figure 12.

Program Organization

The management plan makes use of the capabilities and resources developed for previous manned space flight systems, but modifies the plan to accommodate the intimate integration aspects required of the Shuttle configuration, while at the same time minimizing costs. The overall Space Shuttle organization is shown in Figure 13.

While the size and complexity of the Space Shuttle Program requires a degree of uniformity, this does not mean that each program element is structured to conform to a common set of management systems and procedures. Based on the fact that each industry participant, and NASA too, has a background of development experience and capabilities, it is left up to the discretion of the major participants to utilize systems and techniques best suited to their specific organizational structure and management methods. This, it is hoped, makes it possible to have available all the necessary tools for decision making, without burdening either the contractors or NASA with costly management systems which might place heavy demands on the time and energies of personnel and increase program costs.

Major contractors on this program are shown in Table IV.

Rockwell International's Space Division has been selected as the Space Shuttle system contractor for the design, development, and production of the orbiter vehicle elements. Rockwell's Rocketdyne Division is under contract to NASA MSFC for the Space Shuttle main engines. The RI Space Division also has the role of system engineering and integration contractor to develop and define the requirements and preliminary design for the system configuration, development, and operation. The Martin Marietta Corporation, Denver Division, has been selected as the contractor for the external tank which will be designed and built at the Michoud Facility in Louisiana. The Thiokol Chemical Corporation's Wasatch Division at Lampo Junction, Utah has been selected as the contractor for the Solid Rocket Motor that forms the base for the solid rocket booster. The remainder of the SRB (aft and forward fairings) have not been awarded as yet.

The organization arrangement for the two prime contractors visited by the Panel in 1973, Rockwell's Space Division and Rocketdyne Division, is shown in Figures 14, 15, and 16 respectively. The organizations are in consonance with the Work Breakdown Structure used for the Space Shuttle Program.

Work Breakdown Structure (WBS)

The WBS is a framework which provides a common thread throughout the Space Shuttle program. It offers a uniform approach to the structuring of the program and is a reference for the identification of the major hardware elements and the various other work elements that constitute the total program. Starting at the Program level, the work is successively divided into lower level increments in a manner which represents the way the work will be performed, to the point where manageable units are defined for planning and control of cost, schedule, and technical performance. There is a WBS dictionary which defines by narrative descriptions the scope, content, and tasks of each element. This is covered in Volume III of JSC 07700 "Program Planning and Analysis."

Of particular interest to the Panel in its reviews to date was the Performance Management/Measurement System (PMS). The PMS utilizes the Work Breakdown Structure, cost/schedule/technical data from the contractor's own management system and in their format to provide accurate and timely information to (1) measure progress, (2) identify problems, (3) predict impact of program changes on cost and schedules, and (4) predict cost/schedule/technical performance.

The WBS at its first level is shown in Figure 17.

Program Schedules

A Space Shuttle program schedule has been developed which shows the time-phased relationships of the elements based on a logical sequence which is intended to assure the development of the various program elements in an orderly and integrated manner. This schedule, Figures 18 and 19, reflects constraints imposed by the current specifications, preliminary designs, fabrication models, technology development status, qualification testing requirements, and so on. Milestones have been established to assure adherence to, or known deviations from schedules, costs and technical performance. These are shown in the above figures, and are further discussed in both Level I and Level II control documents, e.g., NASA Headquarters Program Directives and JSC 07700 series of Requirements. The current schedule and development activities underway have been structured to support the first horizontal flight of the Orbiter element at the end of calendar year 1976 or the first part of 1977.

These schedules are admittedly "tight" considering the extent of the work to be accomplished and the known technology challenges. It is in the context of the program's ability to meet and resolve these technical challenges in an orderly and timely manner that the Panel will review the Shuttle elements.

System Engineering and Integration

The System Contractor role deals with five areas of the program which can best be defined by the WBS elements involved:

1. System Management - Support the NASA program and project offices in defining requirements and assisting in the implementation of the following management areas: Performance Management; Configuration Management; Information Management; Logistic Management; GFE Management; Procurement Management; Quality Assurance Management; Commonality Management and Integration Management.

2. System Engineering and Integration - Support the NASA program and project offices in carrying out their activities in: System requirements and synthesis; flight technology analysis; preliminary design and project control; safety and reliability; ground support system requirements and analysis.

3. Flight Vehicle Systems and Orbiter Vehicle - Produce top assembly drawings and stacking specifications; analyze integrated vehicle modes and establish loads; analyze integrated vehicle dynamics (POGO, aeroelasticity, shock, vibration/acoustics); establish and define element and subsystem interfaces; define major ground test requirements and plan.

4. Flight Test Support - Provides support to NASA in the following areas: inputs to flight test requirements and plans; define data analysis and processing requirements; inputs to mission rules and flight handbooks; define integrated ground operations turnaround plan; develop ground operations procedures; provide support personnel as required.

5. System Support - Support NASA in the following areas: Define station set requirements; define ground system measurement requirements; integrate ground hardware and software; support the activation and verification of station sets; provide logistics support personnel; define launch and landing facilities requirements.

The relative roles of JSC and Rockwell Space Division are being renegotiated to further clarify the role and responsibilities of both, to assure that this important function is accomplished in a manner that does not unnecessarily duplicate efforts or let needed activities "drop through the crack".

Baseline System Description

The current Space Shuttle configuration is as shown in Figure 20. The flight system consists of the reuseable Orbiter and Solid Rocket Booster and the expendable external tank. Each of these vehicles and their major components are discussed in detail in the ensuing sections of this report.

In its reviews to date (i.e., Orbiter, SSME) the Panel's interest was directed toward the following areas which act as drivers in the total shuttle design engineering and test effort. These same areas of interest will be focused on in the Panel's reviews of the SRB, ET, facilities and so on. They are placed here so that the reader may further understand that which lies behind the current configuration and possibly those changes to come in the future.

1. Design Approach to reduce uncertainties. This is of paramount importance for a system as technically complex as the Shuttle and its elements. Reduction in uncertainties by the proper design approach may also be the most cost-effective manner of conducting the program. This minimizes the technology support programs to meet the engineering challenges inherent in the design of the Shuttle.
2. Key Design Parameters. These parameters are in effect design specifications that are critical to meeting the Space Shuttle objectives. As an example, the "intact abort" requirement affects the structural design of the orbiter, use of alternate landing sites, payload restraints, crew survival equipment, mission in-flight termination points which in turn affect the separation of the Orbiter from the SRB and ET, crew egress routes and fire suppresion systems. Because of the close coupling that exists between the key design parameters, considerable attention must be paid to the interfaces between Shuttle elements and the lower level interfaces within a given element, and to the trade-off studies necessiated by such interface requirements.
3. Program Major Milestones. Reviews conducted at these primary divisions of the basic schedule to define status, problems, anticipated events, and the resolution of technical and management challenges existing at each milestone.

The Orbiter Vehicle

The purpose of this section is to describe the orbiter vehicle and its major systems individually and as an integrated whole, as well as to indicate those areas which appear to be worthy of further review to assure that plans, requirements, design, and interfaces tend to minimize risks while achieving program objectives. In general, these concerns, which are discussed to some extent in Volume I, deal with major challenges from the technical standpoint and with management problems at this time.

The Orbiter vehicle is comparable in size and weight to a modern transport aircraft and has the following statistics: (Figure 21)

Dry Weight:	150,000 pounds
Length:	123 feet
Wing Span:	78 feet (Double Delta Configuration)
Cargo Bay:	60 feet long, 15 foot diameter
Gross Weight:	245,000 pounds (for due east mission)

(Dry Weight plus the following: crew and provisions; payload; internal fluids; propellant residuals+reserves+ inflight losses.)

Orbiter Aero-thermodynamics

Pre-launch, lift-off, boost flight, re-entry, aerodynamic return, and landing phases of the typical Shuttle mission provide design requirements as well as design challenges. Those of particular interest at this time include POGO, aeroelastic effects, shock, vibration/acoustics, peak heating rates, orbiter flight characteristics, and interface effects with ET and SRB. Each of these areas are covered in the sections which follow, for example, the heat rates and Orbiter isotherms resulting from re-entry dictate design of the Thermal Protection System and are included in that section; terminal aerodynamics determines the aerocontrols (external moveable surface requirements and avionics requirements) and are noted in appropriate sections.

Orbiter Design Approach

The following basis has been set for the design of Orbiter hardware:

1. Alternate or redundant means of performing critical functions, that is, make them fail operational/fail safe.
2. All aluminum structure having the advantages of lower design complexity, minimum analysis complexity, reduced testing and fabrication complexity, and lower cost. Utilize exotic materials only where there is a worthwhile reduction in system complexity, weight or fabrication requirement, or safety implication.
3. State-of-the-art hydraulic system (3000 psi) making it possible to use "known" valves, actuators, seals, and fluids.
4. State-of-the-art atmospheric revitalization system using lithium hydroxide (LiOH) in much the same manner as for Apollo.
5. Cabin designed for 14.7 psia to provide "earth-like" atmosphere to enhance personnel comfort, reduce flammable hazards, and make possible the use of off-the-shelf equipment and accessories.
6. Maximum use of off-the-shelf equipment.

7. Fuel cells used in the vehicle shall be a direct continuation of current technology development.

8. Landing gear will use components and basic design as derived from the USAF B-1 Bomber program.

9. The orbiter aerodynamic configuration to be the most straightforward and compatible with re-entry and landing requirements,i.e., no flaps or special lift devices, if possible.

The degree to which these approaches can be realized can not be fully judged at this time. Much depends upon the firming-up of critical requirements and the strict control of weight allowances for system and sub-components.

Experience has indicated difficulty in using major components off-the-shelf. On the other hand, minor items remaining from prior programs or items built to "older" specs and used for other on-going programs have found use in the Shuttle elements.

Wind tunnel studies have been used extensively to determine the pattern of complicated flow fields that exist over the Shuttle during ascent and descent flight, as well as "on the pad" conditions. These will, of course, be continued as the design progresses to assure that the aerothermodynamic performance does, in fact, match the requirements/design capabilities. Inability to meet aerodynamics requirements due to changes and modifications to such systems as the TPS, access doors, etc. may require a change in surfaces and addition of supplementary aero-surfaces.

Orbiter Key Design Parameters

This discussion of the key design parameters provides the reader with insight on the "drivers" of the engineering design and test activities associated with the Orbiter as well as the ET and SRB through their interfaces with the Orbiter. As an example, the requirement for "intact abort" affects the structural design of the Orbiter, use of alternate landing sites (with associated facilities), payload restraints, crew survival equipment, and mission termination points during the ascent flight.

There are many key parameters, but those noted below have been of particular interest to the Panel during its initial review of the Orbiter. These would appear to have a significant cross-correlation with management areas involving weight control, qualification testing methods, crew safety requirements, interface controls, etc.

1. The Shuttle system is to provide for intact abort of the Orbiter, payload and crew.

2. "Once Around" return to the launch site for emergency abort. This means one orbit then return.

3. Ground refurbishment or turnaround time of 160 hours within a 14 day period.

4. Sub-orbital jetison of the External Tank and Solid Rocket Boosters.

5. Structural design to be based on a 500 mission life and the TPS to be based on a 100 mission life.

6. Orbiter shall incorporate the capability of providing the crew with warning of hazardous conditions and the necessary providions to take effective corrective actions.

7. The Space Shuttle vehicle integrated loads and flight control capability during its ascent trajectory will be capable of controlling to a maximum of 3 g's acceleration and a dynamic pressure up to 650 pounds per square foot.

8. Propulsion performance for design purposes is to be based on nominal specific impulse of the power plants.

9. Prepellant flight performance reserves are to provide 0.85% delta velocity (ft/sec).

10. Ferry requirements (which are not fully defined as yet).

These design parameters are discussed in more detail, as appropriate, in other sections of this report, e.g. mission operations, specific system hardware (Structures, TPS, etc.).

<u>Orbiter Structures Subsystem</u>

The Orbiter structure is generally of conventional aluminum construction protected by reusable surface insulation. The integral cabin structure is machined 2219 aluminum alloy plate with integral stiffening stringers and attach bosses. The assembly is welded to provide a pressure-tight vessel. The forward fuselage is of aluminum single stringer construction. The windshield consists of triple glass panes as shown in Figure 22.

The upper half of the mid fuselage consists of structural payload doors of honeycomb construction hinged along the side and split at the top centerline. Each door is structurally segmented to avoid carrying bending stresses but is designed to react to torsional fuselage loads. There are four segments for each of the two doors.

The aft fuselage is the only current exception to the conventional aluminum structure. The aft thrust structure utilizes titanium/boron-epoxy construction to reduce the aft fuselage weight. The external surface of the aft fuselage is of standard construction except for the removable orbital maneuvering system (OMS) pod. A bulkhead heat shield at the rear of the vehicle providing protection to the main engine systems (SSME), uses Inconel 718.

The wing is of conventional aluminum alloy construction utilizing corrugated spar web, truss-type ribs, and rivited skin-stringer covers. The elevons are of aluminum honeycomb construction and are split into two segments to minimize hinge binding and interaction with the wing.

The vertical tail structure is similar in construction to the wing.

Orbiter Thermal Protection System (TPS)

The most severe heating of the Orbiter occurs during the re-entry trajectory, Figure 23, which results in an entry heating rate history somewhat like that shown in Figure 24. The aerothermodynamic environment upon which the TPS is based does have a number of uncertainties regarding predicted values. As provided to the Panel, the basic categories of such uncertainties are as follows:

1. Aerodynamic heating methods
    a. Wind tunnel data correlations are dependent upon geometry and the ability to simulate actual flow properties as to Mach number and Reynolds number.
    b. Extrapolation of data correlations to flight conditions are dependent upon non-simulatable real gas conditions and effects of streamline divergence, cross-flows and pressure gradients.
2. Trajectory dispersions for the re-entering Orbiter vehicle.
3. Atmospheric variations during re -entry
4. Attitude variations which affect the flow and interrelationships between various portions of the Orbiter surfaces.

The current design is based on a typical Orbiter isotherm picture as shown in Figure 25 and includes three distinct materials for thermal protection against three ranges of temperatures and heat inputs:

1. Reinforced carbon-carbon (RCC) covering approximately 563 Ft^2 and weighing approximately 3,900 pounds. The structural surface area covered by this RCC material are all those with expected temperatures between 2300^{o}F and 3000^{o}F. The so-called carbon-carbon material consists of pyrolyzed carbon fibers in a pyrolyzed carbon matrix with silicon carbide coating.

2. High temperature reusable surface insulation (HRSI) covers the intermediate heating areas subjected to temperatures ranging from 1200^{o}F to 2300^{o}F. HRSI covers an area of 4555 Ft^2 and weighs about 10,800 pounds. This material is made of 99% pure felted silica fibers ranging in thickness from 1 to 3 inches and made into individual tiles 6 inches square. The HRSI is coated for waterproofing and handling protection by a fritted borosilicate coating containing pigment to provide the desired absorptivity-emmissivity ratio (absoprtion and radiation of heat). There will be in the neighborhood of 32,000 tiles.

3. Low temperature reusable surface insulation (LRSI) made of felted silica fibers covers those surface areas which do not exceed 1200^{o}F. This includes 6500 Ft^2 and weighs about 4700 pounds.

The general area coverage and cross-sectional views showing attachment modes are provided in Figure 26.

The design requirements set forth for these three types of thermal protection are as follows:

1. Carbon-Carbon
   - -170° F to 2900° F
   - Minimum strength degradation after 100 missions
   - Fabricated into complex shapes without strength variations and resistant to catastrophic failures
   - Resistant to impact damage
   - Ease of inspection
   - Aerodynamic smoothness
2. HRSI
   - -170°F to 2300°F with an overtemperature capability to 2700°F
   - Minimum degradation after 100 missions
   - Resistance to catastrophic failures
   - Resistance to impact damage
3. LRSI

   the same as HRSI except the temperature range is -170°F to 1200°F, and uses a different external coating and thermal barrier.

A major design challenge here is the ability to assure stable airflow patterns over the TPS surface and the potential interfence from the many doors and penetrations through the TPS. The RCC heating environment uncertainty will most likely require additional wind tunnel tests to increase the data base. However, it appears that wind tunnel data will not improve the extrapolations to flight conditions. The major design issues for the HRSI and LRSI include the strain isolation pad material and adhesive bond used, the title joints, TPS/fuel compatibility, and the dynamic seals required for aero-surfaces (Wing-Elevon Hinges, etc.), and the coatings to meet the 100 mission life requirements.

Orbiter Vent System

The Orbiter vehicle requires compartment venting to preclude the buildup of undesirable fluids, release of liquids when required, purging for environmental control and to maintain proper pressure differentials in the structure. The 27 vents known to be required at the time of the Panel's review cover a net vent area of about 15 Ft^2. The requirement for venting occurs during the mission as indicated below.

1. Prelaunch: Purging and maintenance of required pressure differentials.
2. Ascent: Vent ports used primarily to maintain structural Delta P's.
3. SRB separation: Separation motors exhaust ingestion to be vented.
4. Orbit: Majority of vents open going all the way to molecular venting.
5. Entry: Fuel dumping and heat sinks venting (water).
6. Flyback: Vent ports to maintain structural delta P.
7. Postlanding: Purge to maintain proper pressure differential and elimination of possible noxious gases.

Orbiter Remotely Operated Doors

The many doors in the surface of the Orbiter impact the TPS, Vents, landing gear and so on. There are, at this time a total of 47 doors. The purpose here is to identify them and with this awareness the challenges posed by these doors on other systems can be taken into account during future reviews by the Panel. For example, the Orbiter/ET separation closeout doors effect on the design and operation of the TPS.

Remotely operated Doors (Number)	Doors Operated - During Orbit	After Launch	After Entry
Orbiter/ET Separation close-out Doors (2)		x	
Payload Bay Doors (4)	x		
Startracker Door (1)	x		
Payload Preflight Umbilical Door (1)		x	
MPS Umbilical Doors (2)		x	
Main Landing Gear Doors (2)			x
Nose Landing Gear Doors (2)			x
RCS Module Doors (2)	x		
Air Data Sensors (4)	x		
Payload Bay vent Doors (16)	x		
Other Vent Doors (11)	x		

Orbiter Main Propulsion System

The specifics of the main engines, SSME's, are discussed under the sections devoted expressly to the SSME. The main propulsion system (MPS) provides the major ascent velocity increment by operating in parallel with the solid rocket boosters during the initial ascent phase and continuing to burn after SRB separation. The External Tank which also forms a part of the MPS is itself discussed in more detail in a section devoted only to it. Of interest here are the Orbiter interfaces and internal hardware that interface with the SSME's and the ET.

The Orbiter contains five fluid lines, which interface with the external tank (ET) through self-sealing disconnects. All disconnects are located on the bottom of the orbiter engine compartment. The three fuel disconnects are mounted on a carrier plate on the left side, and the two oxidizer disconnects are mounted on the right side. MPS components are listed below:

1. Aft Section
   a. Liquid oxygen fill/drain disconnect
   b. Orbiter/ET liquid oxygen disconnect
   c. Orbiter/ET Liquid hydrogen disconnect
   d. Liguid hydrogen fill/drain disconnect
2. Orbiter and External tank
   a. Liquid hydrogen pressurization line
   b. Liquid hydrogen feedline
   c. Liquid hydrogen vent line
   d. Liquid oxygen feedline
   e. LOX pressurization
   f. LOX vent
   g. Disconnect cover doors

The propellant feed subsystem supplies the propellants to the main engines from the external tank. The propellant fill and drain subsystem provides propellants to the external tank during loading. This subsystem, in conjunction with the pressurization control subsystem provides drain capability. The propellant conditioning subsystem provides conditioned propellants to the main engine inlets prior to engine start. The pressurization control subsystem maintains the proper tank pressures in the external tank, after tank pressurization prior to engine start, and during main engine operation, and also protects the external tank from overpressurization. Tank prepressurization and hydrostatic head provide the required net positive suction pressure to the engine pump inlets during the starting transient. Following engine thrust buildup tank ullage pressure is maintained by vaporized propellant pressurant extracted from the engines. The external tank ullage pressures maintained during the prepressurization period by ground supplied ambient helium are 20 -22 psia for the LOX and 40 - 42 psia for LH_2. The pneumatic supply subsystem provides helium for valve actuation, for main engine purge, and for propellant feedline repressurization prior to orbiter reentry. The main engine GN_2 purge subsystem provides a nitrogen inerting purge to the main engines prior to start. The propellant management subsystem controls propellant loading and also engine cutoff whenever propellant depletion occurs.

Orbiter Maneuvering Subsystem(OMS) and Reaction Control System (RCS)

Two separate systems are used for orbital propulsion and attitude control. The OMS provides the propulsive thrust to perform orbit insertion, circularization, orbit transfer, rendezvous and de-orbit. The RCS provides vehicle attitude control and translation for small velocity increments. Figures 27 and 28 show these systems.

The OMS/RCS design requirements are currently:

1. Nitrogen tetroxide oxidizer/monomethylhydrazine fuel
2. OMS propellant weight: 12,000 lbs/pod
3. RCS propellant weight: 1,900 lbs/pod and 4,000 lbs in fwd module
4. Operating Modes
    a. OMS tanks to OMS engines
    b. OMS tanks to RCS engines
    c. RCS tanks to RCS engines
    d. Cargo bay kit to OMS engines
    e. Crossfeed between OMS/RCS pods
5. Vertical fill with fuel to the left pod and oxidizer to the right pod
6. On-Pad loading prior to launch of Shuttle Vehicle
7. Vertical or Horizontal drain capability.

Each OMS engine produces a vacuum thrust of 6,000 pounds at a chamber pressure of 125 psia and a specific impulse of 313.2 seconds. In the OMS pods there is sufficient propellant to provide an on-orbit delta V of 1000 ft/sec with a payload of 65,000 lbs. Up to three sets of auxiliary tanks can be mounted in the cargo or payload bay, with each tank providing an additional Delta V capability of 500 ft/sec to achieve an OMS overall Delta V capbility of 2,500 ft/sec.

The RCS has a forward and aft set of bipropellant thrusters (1 fwd and 2 aft in the OMS pod). The disposition of the thrusters are as follows:

1. 40 main thrusters (16 fwd, 12 per aft RCS)
    a. Thrust level: 900 pounds
    b. Specific impulse: 289 seconds
2. 6 vernier thrusters (4 fwd, 2 aft; actually locations under study)
    a. Thrust level: 25 pounds
    b. Specific impulse: 228 seconds

The forward RCS module is independent of the aft RCS system. The aft RCS system is integrated directly into the OMS pod and has its propellant system interconnected with the OMS units.

The OMS pod,including the RCS system, is 270 inches in length and a major height of 135 inches. The structural arrangement is shown in Figure 28. The skin panels are integrally machined 2124-T851 aluminum, the bulkheads are of the same material. The webs are made of 2024-T62 aluminum beaded skins with secondary stiffeners. The engine, propellant lines, pressurization system and controls are shown schematically in Figure 29 and 30, 31 for the combined OMS/RCS systems in the OMS pod.

Orbiter Cabin Arrangement

In normal operations, the personnel complement can vary from 3 to a maximum of 7. The basic crew consists of a pilot, co-pilot and a mission specialist. For more complex missions, up to 4 additional specialists can be carried. Accommodations are provided for personnel of both sexes. The Orbiter cabin arrangement is shown in Figure 32. The primary flight stations (forward) are organized in a familiar pilot-copilot relationship with sufficient duplication of displays and controls to permit the vehicle to be piloted from either seat and permit one-man emergency return.

Orbiter Mechanical Subsystems

The orbiter mechanical subsystems, together with their electrical and hydraulic actuators, operate the aerodynamic control surfaces, landing/deceleration system, payload bay doors, and payload accommodation and payload handling subsystems. Orbiter/external tank propellants disconnects and a variety of other mechanical and pyrotechnic devices are also a part of the mechanical subsystems. These subsystems are shown in Figure 33.

Aerodynamic control surface operation is accomplished by single-balanced, dual-switching servo-actuators for the control of elevons, rudder, and rudder speed brake. A 3000 psi hydraulic system supplies the power for these control functions.

The landing/deceleration system and its mechanical components are designed to facilitate safe landing to velocities up to 217 knots for which 40,000 pound rated tires and 750 hp/pound brakes (280 x 10^6 foot-pounds) are under development.

The payload accommodation subsystem includes retention latches that are remotely controlled to hold down or release the payload or cargo items. Their design is such that they are consistent with the requirements for "intact abort" and that they cannot transmit orbiter stresses, such as bending, to the payload. The payload handling, a remote control system, is operated from a crew station in the aft end of the upper deck of the cabin. Manipulator arms are being developed for the Shuttle to provide a capability for deployment of the maximum payload of 65,000 pounds in less than seven minutes.

The orbiter/external tank propellants disconnects are designed to accommodate approximately 485 pounds per second flow of liquid hydrogen (50,000 gpm), and approximately 2900 pounds per second flow of liquid oxygen (18,500 gpm). The proellant lines contain 17" diameter disconnects and shut-off valves (one on the orbiter side and one on the external tank side of the interface). These shut-off valves are designed to preclude inadvertant closure during engine firing. The fluid trapped between the two closed valves, a maximum of three cubic feet, is allowed to dump freely as the disconnect sections are disengaged.

Orbiter Hydraulic Subsystem

Hydraulic power is provided to the orbiter systems by four independent subsystems with a high degree of redundancy. The system philosophy and resign requirements are as follows:

Philosophy: Utilize existing hydraulic technology (materials, seals, etc.), maximize the use of existing components (valves, actuators, etc.), and thereby reduce the cost and risk involved.

Design Requirements:

- 3 independent subsystems driven by independent Auxiliary Power Units.
- Nominal operating pressure of 3000 psi.
- Fluid operating temperature, -65° F to $+275^{\circ}$ F.
- Fluid is currently MIL-H-5606 (This is under reevaluation at this time)
- Water boiler cooling
- Insulated lines with heat addition available to maintain operating temperatures.
- External leakage controlled such that leakage is returned to a resevoir or captured in a sump.

The present design calls for the use of MIL-H-5606, "red oil", which has been used extensively in the past on military aircraft. However this fluid does not have the safe properties that newer fluids such as MIL-H-83282 have. The use of this less flammable fluid in the hydraulic system is currently being examined. In addition MIL-H-83282 has the advantage of low evaporation loss with benign oily residue under vacuum conditions, which in turn alleviates the present MIL-H-5606 thermal conditioning requirements caused by aerodynamic heating. A comparison of the two fluids follows:

Parameter	MIL-H-5606	MIL-H-83232
Temperature Limits	-65F to +275F	-50F to +400F
Flash Point	225F	435F
Autoignition point	470F	680F
Shear Stability	Poor	Good
Useage	Military for 30 years	Military for 3 years

Both of these fluids are compatible with each other and the systems components (materials) now in the design.

The APU's utilize the monopropellant Hydrazine (N_2H_4) and each provides 130 horsepower to a 65 gpm variable displacement pump.

Thermal conditioning of each of the three systems is provided by heat exchangers. During ascent and descent, the hydraulic fluid is cooled by circulating through a water boiler. During on-orbit operations, hydraulic fluid is heated by circulating through the environmental control system Freon loop. Circulation is provided by an electric motor-driven pump.

The hydraulic system operates the following vehicle subsystems:

1. Main landing gear strut actuator and uplock actuator
2. Nose landing gear strut actuator and uplock actuator
3. Nose wheel steering actuators
4. Main Landing gear brake/anti skid valves
5. Elevon servo actuators
6. Thrust Vector Control servo actuators
7. Rudder/Speed Brake servo valves and actuators.

Orbiter Electrical Power Subsystem (EPS)

The EPS consists of the equipment and reactant required to store energy and to supply power to equipment through the electrical power distribution and control subsystem. The system consists of the following major functional categories:

1. Energy storage: cryogenic hydrogen and oxygen storage, orbiter batteries.
2. Power generation: fuel cell powerplants and air breathing engine driven ac generators (if used).
3. Power distribution, control and conditioning: In conjunction with the power sources ensures power characteristics compatible with Orbiter, ET, SRB requirements.

The fuel cells, which generate power through the reaction of hydrogen and oxygen provide 27.5 to 33.5 volts over a power range of 2 to 12 KW each. Hydrogen and oxygen are stored in cryogenic containers located in the mid-fuselage. The reactants are maintained at a nominal pressure of 250 psia for hydrogen and 900 psia for oxygen, by means of an external heat loop, associated pumps and loop controls, which reject heat and deliver product water to the Environmental Control and Life Support System. Fuel cells have an operating life of 2000 hours maintenance free, and a design goal life of 5,000 hours. A single fuel cell is capable of supplying sufficient power for safe return from orbit. The fuel cell system with reactant storage has the following design parameters:

- 14 KW continuous power output with 24KW peak
- 27.5 to 33.5 VDC
- Reactant Storage
  1530 KWH mission energy
  264 KWH Abort/Survival energy
  112 lbs O_2 allotted to the Environmental and Life Support System
  100 lbs H_2 per tank (Two Hydrogen Dewars)
  855 lbs O_2 per tank (Two Oxygen Dewars)
  1050 psia maximum pressure for Oxygen dewar
  335 psia maximum pressure for Hydrogen dewar

Typical Dewar design is shown in Figure 34.

Three 10-ampere-hour 28 volt nickel-cadmium batteries supply energy for power subsystem reset and restart and for firing pyrotechnic devices. Three 40-ampere-hour 28 volt silver zinc batteries supply power for development flight instrumentation.

The distribution and control system is characterized by three redundant, simultaneously operating, 28 volt dc buses. Each is isolated to avoid complex parallel controls and to confine power transients to a single redundant string. The essential control buses supply event controller logic power and loads required for bus power-up from a power-off state. Event controllers include circuits for pyrotechnic circuit integrity tests without ground access connections to the system. Ordnance safing is similar in configurations and procedures to that used in the Apollo CSM vehicles (i.e., the pyrotechnic firing circuits maintain a short circuit across the pyrotechnic initiators until time to fire.

The TPS is capable of providing 7 KW of continuous and 12 KW of peak DC power to the payload, with an energy allotment of 50 KWH. Energy in excess of 50 KWH can be provided by additional kits.

<u>Environmental Control and Life Support System</u>

The ECLSS consists of the atmospheric revitalization and thermal control subsystems. The DCLSS is shown in Figures 35, 36, and 37.

These subsystems divide into four functional groups: atmosphere revitalization; food, water, and waste; thermal control; and, airlock support. The overall system comprises a dry weight of about 4,300 pounds plus some 1,300 pounds of trapped and useable consumables and liquids, and consumes about 1.6 KW of power for normal operation.

The atmosphere revitilization subsystem maintains a habitable environment for the crew and a conditioned thermal environment for avionics equipment. The cabin active wall cooling system in conjunction with the cabin heat exchanger operates with water coolant to maintain cabin temperatures between 65° F and 80° F except during reentry where it shall not exceed 90° F.

The cabin oxygen partial pressure is maintained between 3.0 and 3.2 psia and sufficient nitrogen is added to achieve a cabin total pressure of 14.7 ± 0.2 psia. The oxygen required for metabolic and cabin leakage makeup is obtained from supercritical cryogenic storage. The nitrogen for normal operating and oxygen for repressurization are obtained from 3000 psi storage vessels.

Carbon dioxide and odors are removed by drawing cabin air through lithium hydroxide (LiOH) canisters. The flow is about 1300 lbs/hr. Cabin air is then ducted through the cabin heat exchanger, for temperature and humidity control.

Avionics equipment is thermally conditioned by both forced air convection and cold plates. The controls and displays avionics are cooled by ducting cabin atmospheric air over the electronic packages. The major portion of the avionics are installed in three avionics bays in the cabin area. The bays are isolated from the cabin atmosphere and are 0.4 psi lower than cabin pressure to prevent excessive sensible heat loads on the cabin heat exchanger and prevent potential outgas products from the avionics from entering the cabin.

The food, water and waste management subsystem provides basic life support functions for the crew, including facilities for preparation of both therm-stabilized and freeze dried foods. The waste management subsystem collects, processes and stores solid and liquid wastes.

Active thermal control is accomplished through a system of heat exchangers installed in the Freon loop, coupled to space radiators with a total effective radiative area of 1440 ft^2. The system is capable of rejecting 8.5 KW of heat with inlet temperatures of 40°F. Wall temperatures during reentry shall not exceed 113F for crew accessible surfaces and shall not exceed 120F for non-accessible surfaces.

During prelaunch operations thermal control is accomplished by a GSE heat exchanger installed in the coolant loop which operates until liftoff. The water sublimator in the cabin water loop is active to control to 40F until the radiators are deployed. The water sublimator in the atmosphere revitalization system is activated for reentry from the time the payload bay doors are closed until 100,000 feet altitude. The ammonia boiler is operated from 20,000 feet to touchdown and during postlanding until the GSE cooling system is connected. Heat rejection from 100,000 to 20,000 feet relies upon thermal capacitance.

The airlock support provides airlock repressurization, and EVA support. The control mechanisms for the airlock are to be found both inside the airlock and inside the cabin; displays of airlock absolute pressure and differential pressure between the airlock, payload, and orbiter cabin, in both the airlock and cabin; manual override controls for reduction of rate of pressure change and termination of change.

Orbiter Avionics Subsystem

The Shuttle avionics system performs the on-board functions of guidance and navigation, aero flight control, data processing and mathematical computation, audio and radio frequency communications, radio frequency navigation and terminal guidance, crew displays and controls, instrumentation and recording of measurements, electrical power distribution and control, payload management and accommodation, and performance evaluation.

Most of the functions are allocated to the orbiter. The less costly avionics elements are allocated to the external tank and the solid rocket booster. The orbiter avionics controls the trajectory of the mated system through SRB and SSME thrust vector control, generates separation signals, and collects and telemeters data from the external tank, boosters, and orbiter. The SRB has the low-cost electronics necessary to permit location and recovery of expended rocket cases. Shuttle system avionics are distributed as shown in Figure 38 in the orbiter vehicle.

Flight deck displays and control are organized into four functional areas: (1) two forward-facing primary flight stations for vehicle operation, (2) an aft-facing station for payload handling, (3) mission specialist and payload specialist stations for management and checkout of active payloads, and (4) subsystem management and power distribution panels in the remaining flight deck area. Manual flight controls, rudder pedals, and speed brake controllers are located at each of the forward stations. A master thrust controller is located at the left station. Docking and payload-handling operations via the manipulators and vehicle attitude/translation maneuvering can be done at the aft flight deck station, which has provisions for both direct and closed-circuit TV external viewing. Dedicated aeroflight instruments are provided for vehicle control together with a performance monitor and caution and warning light matrix.

The guidance, navigation, and flight control subsystem provides automatic and/or manual orbiter control capability, vehicle steering displays, and inertial navigation capability augmented by star sensors. Body-mounted rate gyros and accelerometers are used for vehicle stability control. Air data are obtained from redundant probes deployed at lower altitudes. Automatic landing of the orbiter is achieved under computer control by integrating the output of the above units with RF navigational aids (ILS, TACAN, radar altimeter).

Data processing and software subsystems provide the onboard digital computation, data display, and data handling required to support the other subsystems. The computers provide the facility for mathematical computation and system data processing as established by the system software. This facility is comprised of five general-purpose computers, input/output units and two mass memories.

The communications and tracking subsystem is shown in Figure 39.

The operational flight instrumentation subsystem acquires and distributes engineering data developed by sensors and transducers in all onboard subsystems through a pulse code modulation system. Digitized data is supplied to the displays and control subsystem for display as caution and warning and performance monitoring signals, to the communications and tracking subsystem for transmission to the ground and to the instrumentation recorders. Currently a voice recorder is provided for all mission phases and a crash data recorder for ferry flights.

The developmental flight instrumentation provides additional capacity for measurement during flight testing. It is an overlay system that is removable with minimum scar weight.

Air-Breathing Propulsion Subsystem*

Present plans call for using six off-the-shelf P&W TF-33-P-7 turbojet engines mounted on the Orbiter for horizontal flight testing and ferry operations. Changes made to the systems of the Orbiter to meet the test and ferry operational requirements are:

1. Deleted from the orbiter baseline
    a. Main Prolusion System, OMS, RCS, APU's
    b. Operational Landing/Deceleration System
    c. Thermal Protection and Thermal Control Systems components not required for horizontal flight test or ferry.
    d. Payload provisions such as the retention system, manipulator arms, cargo bay liners, other mechanisms not required.
    e. Electrical power not required for horizontal flight including the fuel cells, cryogenic LOX, and hydrogen tanks.
    f. Airlock, crew couches, Environmental Control and Life Support System components necessary for orbital flight.

*A recent decision has been made to remove the air-breathing engines and to utilize a modified C-5A or 747 aircraft to carry the Orbiter in a "piggy-back" configuration.

g. Avionics equipment not required for horizontal flight such as controls, displays, star tracker, communication equipment.

2. Added to the Orbiter Baseline
    a. Air Breathing Engines and required propellant and attachment components, nacelles/pylons, jet fuel tank, displays and controls.
    b. A dummy main propulsion system.
    c. Aft body cover over the SSME location and necessary fairings to cover the OMS/RCS pods and a special nose section for horizontal flight test with instrument boom.
    d. Ferry landing and deceleration system
    e. Ferry avionics kit
    f. Ejection seats
    g. Development Flight Instrumentation.

The Horizontal Flight Test configuration and performance is:

Length	160 feet
Wing Span	78 feet
Operating Take Off Weight	223,000 pounds
Ceiling	30,000 feet
Landing Speed	170 Knots Indicated Air Speed
Max Speed	0.75 Mach
Fuel	43,600 pounds of JP-4
Engine Specific Fuel Consumption	0.795 at 25,000 feet and Mach 0.7

The feasibility of airlifting the orbiter "piggy-back" on a C-5A or 747 aircraft for horizontal flight tests and ferry operations is currently under study.*

Summary Comments on the Orbiter Vehicle Design and Hardware

The key management and technical challenges on the Space Shuttle Orbiter vehicle system are presented here based on the Panel's reviews through the end of 1973.

1. Weight control is a key element in the management and technical areas because of the cost/weight and weight/performance interrelationships. Current weight margins are on the order of 3% to 5% allowable growth at a point in time when the final design requirements have not been fully defined.

2. Subcontractor response, covered in more detail under the R,Q & S section, impacts the hardware to the extent that it is difficult to get qualified suppliers to bid in many areas.

3. Thermal Protection System (TPS). The ability to assure predictable airflow patterns over the multi-tiled aerodynamic surface and the numerous doors and penetrations is a major development concern. Other areas of interest include the adequacy of the TPS for all-weather

*A recent decision has been made to eliminate the air-breathing engines and to utilize a modified C-5A or 747 aircraft to carry the Oriber in a "piggy-back" configuration.

conditions (rain and lightning), bonding to the base structure, degree of maintenance required for reusability, qualification test program required, potential results from damage to or loss of one or more tiles. Aerosurface seals and TPS/fuel compatibility are also of interest.

3. Entry yaw control is both an operational and a hardware concern. The solution to this lateral dynamic stability problem can affect the hardware (surface controls and their associated avionics). With a definition of the aerodynamic deviations occuring during aircraft-type flight, the vehicle requirements for aero-control (use of RCS, flaps, speed brakes, movable ventral) can be determined.

4. Payload bay liners are deflected into the payload envelope by in-flight venting systems causing differential pressures. This requires a reduction of the delta P or a strengthening of the liners to prevent their deflection.

5. Payload retention system capability to restrain payloads during intact abort.

6. Landing gear extension is currently accomplished by gravity drop alone. The value of a positive extension system, either as a primary or redundant mode, warrants consideration. From the information currently available to the Panel it appears that the operating time for lowering the landing gear, some four to eight seconds, is greater than desireable considering NASA's lifting-body program experience.

7. Alternate ferry modes which has been discussed above and in volume I. The operational requirements for test and ferry when fully defined may impact the orbiter baseline structural and TPS designs.

8. Leaks and fire hazards: In a reuseable system that can not be fully pressure tested under actual operating temperatures prior to reuse (such as the cryogenics systems on board the Orbiter) there is a concern with regard to the difficulties in assuring the integrity of the systems, particularly the cryo-seals. The Orbiter's aft engine room, covering the last 18 feet of fuselage, represents an unusual fire hazard because of the large, complex high pressure systems carrying flammables. Thus the design criteria and test program to assure leak integrity is of added significance.

Space Shuttle Main Engines (SSME) (Figure 40)

The Shuttle Main Propulsion System (MPS), consisting of three SSME's, operates in parallel with the SRB's during the initial ascent phase and continues to burn until just before injection to orbit after SRB separation. Each of the rocket engines operate at a mixture ratio (liquid oxygen/liquid hydrogen) of 6:1 and a chamber pressure of approximately 3000 psia to produce a sea level thrust of 375,000 pounds and a vacuum thrust of 470,000 pounds with a fixed nozzle area ratio of 77.5:1. The engines are throttleable over a thrust range of 50 to 109 percent of the design thrust level. This provides a higher thrust level during liftoff and the initial ascent phase, and allows limiting orbiter acceleration to 3 g's during the final ascent phase. The engines are gimbaled to deflect $\pm$ 10.5 degrees for pitch, and $\pm$ 9 degrees for yaw and roll control during the orbiter boost phase.

The SSME major components are shown in Figure 41 and the SSME propellant flow schematic is shown in Figure 42.

Four turbopumps, two low-pressure and two high-pressure, are key components in describing the physical characteristics of the SSME system. The hot gas manifold is the structural backbone of the engine package and supports the turbopumps noted above, but also the fuel and oxygen preburners, injectors, and the main combustion chamber.

The Preliminary Design Review for the SSME was conducted in the third quarter of 1972, which was the first major element of the Space Shuttle to undergo PDR. Since that time major changes have been made in the following four areas: Hydraulic actuator, controller memory, hot gas manifold liner, and the thrust chamber nczzle. Simply stated these changes are as follows:

1. Hydraulic actuator
    a. Original Design
        - Double acting piston
        - Center of Gravity cantilevered
        - High loads
    b. Current Design
        - Double piston
        - Center of gravity near its mounting points
        - Reduced loading
        - Reduced weight by about 24 lbs.
2. Engine Controller Memory Changes
    a. Original Design
        - 12 K capacity
        - Utilize program substitution for checkout
        - Inadequate spare capacity
    b. Current Design
        - 16 K capacity
        - Permits one program
        - Provides complete flight simulation.
3. Hot Gas Manifold Liner
    a. Original Design
        - Haynes 188 Back shell
        - Multilayer screen/foil laminate insulation
        - Expensive fabrication
    b. Current design
        - Utilizes Incoloy 903
            - High Strength
            - Low coefficient of expansion
            - Not affected by Hydrogen
            - Reduced weight
            - Reduced cost of fabrication
4. Thrust Chamber Nozzle
    a. Original Design
        - 1 1/2 pass for fluid through nozzle surface channels
        - complex manifold design

b. Current Design
   - single pass of the fluid through the surface channels
   - simplified manifolding design
   - fabrication complexity reduced
   - increased life expectancy

Turbomachinery, Fuel

The low-pressure fuel turbopump has an overall dimensional envelope of approximately 18 by 24 inches and weighs approximately 135 pounds.

Inlet Flowrate, lbs/sec	147	Turbine Flowrate, lb/sec	31
Discharge Flowrate, lb/sec	147	Turbine Inlet Pressure,psia	4257
Discharge pressure,psia	234	Turbine Disch. Pressure,psia	3590
Discharge Temperature, R	40	Turbine Inlet Temperature, R	550
Brake Horsepower	2400	Turbine Disch. Temperature,R	540
Speed, rpm	14,800		

The high-pressure fuel turbopump is a line replaceable unit (LRU) with an overall dimensional envelope of approximately 22 by 44 inches. It weights about 700 pounds.

Pump Flowrate, lb/sec	147	Turbine Inlet Pressure, psia	5160
PumpInlet Pressure,psia	178	Turbine Disch. Pressure,psia	3380
Pump Disch. Pressure,psia	6190	Turbine Inlet Temperature,R	1730
Pump Inlet Temperature,R	40	Turbine Disc. Temperature,R	1585
Pump Disch. Temperature,R	93	Brake Horsepower	62,240
Turbine Flowrate, lbs/sec	143	Speed, rpm	35, 100

Turbomachinery, Oxidizer

The Low-pressure oxidizer turbopump has approximate dimensions of 18 by 18 inches and weighs approximately 185 pounds.

Inlet Flowrate, lbs/sec	885	Turbine Flowrate,lbs/sec	165
Disc. Flowrate, lbs/sec	1050	Turbine Inlet Pressure,psia	4480
Disc. Pressure, psia	415	Turbine Disch. Pressure,psia	415
Disch. Temp., R	170	Turbine Inlet Temp., R	188
Brake Horsepower	1468	Turbine Disch. Temp., R	187
Speed, rpm	5145		

Controller Assembly

The Controller Assembly is a pressurized and thermally conditioned electronics package attached to the thrust chamber and nozzle coolant outlet manifolds on the low-pressure turbopump side of the engine. It is designed to operate in conjunction with engine sensors and the vehicle control system for engine control, monitoring, and checkout operations.

Specifically the controller provides:

1. the interface for electrical power and command signals from the orbiter vehicle,

2. voltages suitable for operating engine igniter, actuators, and on/off controls,
3. valve sequencing for engine purge and childown,
4. closed loop thrust and mixture control,
5. engine checkout and monitoring for limit detection and control,
6. failure detection and redundancy switching for fail operational/ fail safe engine operation,
7. propellant dump control.

The controller assembly itself weighs about 177 pounds and is approximately 14.5" x 17" x 23.5" in size. Other characteristics are:

1. Power required,700 watts.
2. Dual redundant operation with a memory of 16K words/channel.
3. Thermal environment:
   -50F to +95F operating
   -200F to +200F non-operating
4. There are 22 engine connectors and 3 GSE connectors.

The controller computer memory, at the time of the Panels review, using plated wire appeared to be a high risk technology with which there was little experience. Testing ofthe design concept is now under way. The advantage of such a electrically alterable read-only memory lies in its low power requirements, fast access, and ability to change memory content. The possibility of using a magnetic core, if necessary, remains.

The requirement that the controller conduct tests of all control system components once every 20 milliseconds is also under study and is related to the use of plated wires units versus the magnetic cores.

The controlled is divided into five functional sections arranged on a dual-redundant basis:

1. Input electronics which receive data from the in-flight sensors, convert the data to a digital form, and send it to the computer,
2. Computer interface electronics which control the flow of all data within the computer,
3. Digital computer unit which performs computations and issues engine control signals upon receipt of sensor data and vehicle commands, and stores engine data until requested by the vehicle.
4. Output electronics which convert computer digital commands to voltages suitable for operating engine igniter, actuators, etc.,
5. Power supply electronics which converts vehicle supplied electrical power to voltages required by controller functional units.

Hot-Gas Manifold

The hot-gas manifold is a double-walled, hydrogen gas cooled structural support and fluid manifold, which conducts hot gas from the turbines to the main chamber injector. Geometry and operating paramters are:

Weight:	about 600 pounds			
Geometry:	Figure 41 shows the manifold schematically			
Operating Parameters:	Pressure(Max), psia	3360 hot gas	3740 coolant	
	Temp.(max.), R	1575 " "	540 "	
	Flowrate(Total),lb/sec	202 " "	30 "	

Nozzle Assembly

The nozzle assembly is a regeneratively fuel-cooled, 80.6 percent bell chamber that completes the expansion of the main combustion chamber gases from a 5:1 to 77.5:1 expansions ratio. It is approximately 120 inches long, has a 94-inch exit outside diamter, and weighs approximately 950 pounds.

Other Major Elements of the SSME

The following components have had little change since the SSME PDR and are presented here in abbreviated form.

Fuel Preburner: Weight= 135 pounds. Consists of three major parts: (1) injector, (2) augmented spark igniter chamber, and (3) combustion chamber.

Oxidizer Preburner: Weight = 80 pounds. The three major parts are the same as those in the fuel preburner.

Main Injector: It is approximately 22 inches in diamter, 19 inches long, and 380 pounds in weight. A gimbal bearing mounts to the forward end of the injector and transmits thrust loads to the vehicle thrust structure. Basically, the injector assembly consists of a structural body, injection elements, two faceplates to cover the numerous oxygen and hydrogen tubes, and an augmented spark igniter.

Main Combustion Chamber: The main combustion chamber is a cylindrical, regeneratively cooled, structural chamber that contains the burning propellant gases and initiates their expansion from the chamber throat to a ratio of 5:1. It weighs about 440 pounds.

Interconnects: Engine interconnects are divided into three categories: (1) main propellant articulating ducts, (2) fluid interface lines, and (3) component interconnects. Main propellant articulating ducts interconnect the non-gimbaled low-pressure turbo pumps to components of the engine that gimbal. Fluid interface lines are the vehicle to engine lines for recirculation of propellants, propellant tank pressurants, hydraulics, and pneumatics. Component interconnects are rigid lines with the exception of perhaps one small-diamter flexible hose.

Heat Exchanger: This unit converts liquid oxygen to gaseous oxygen for vehicle oxygen tank pressurization. The heat exchanger weighs about 20 pounds.

Valves: The engine valves include the following units:

- Main Oxidizer Valve (about 100 pounds)
- Main Fuel Valve (About 80 pounds)
- Fuel Preburner Oxidizer Valve (About 35 pounds)
- Oxidizer Preburner Oxidizer Valve (about 35 pounds)
- Chamber Coolant Valve (about 27 pounds)
- Propellant Bleed Valve

Summary Comments on the Space Shuttle Main Engine (SSME)

The significant design challenges and management concerns are covered below, based on the Panel's reviews to date.

1. Weight control is both a design and management challenge with only a 5% margin currently existing between the actual and specification weight.
2. As noted for the Orbiter vehicle the use of MIL-H-5606 hydraulic fluid and its replacement with MIL-H-83282 fluid may require re-evaluation at a later date.
3. The electronic controller for the SSME is a challenge from several standpoints. The use of plated wire memory units in the computer have yet to be proven, particularly under the environments that it is subjected to when in use on the SSME's. The interconnect between the Controller and the many sensors and electrical input/outputs requires assurance of pin/hole alignment, pin plating, and stability of materials. The power supply unit must have adequate power regulation capability, the current weight of the total controller unit is high. The operating thermal environment of 95^{o} F appears too low considering the controller location.
4. The fixed low pressure pumps and the large engine gimbal angles require flex lines which must be compatible with oxygen and not subject to hydrogen embrittlement.
5. The high combustion chamber heat flux (97 BTU/in.^{2}/sec) versus the Saturn V J-2 engines with a much lower heat flux (20 BTU/in.^{2}/sec) requires careful material selection, high coolant flow rates, and fluid flow characteristics of coolant flow channels consistant with reasonable pressure drops.
6. SSME performance is specified at 452.9 second specific impulse as a minimum design value. This requires high combustion efficiency, low nozzle losses and minmizing jet separation at sea level operation.
7. Availability of materials and support by vendors and subcontractors is a significant challenge at this time. This is covered in more detail in the R,O & S section of this report.
8. The impact of the specified turnaround time of 160 hours on the design of the SSME is a challenge and is covered in more detail under the ground support section of this report.

External Tank (ET)

The Shuttle system external tank is used to store both the liquid hydrogen propellant and the liquid oxygen oxidizer for the Space Shuttle Main Engines and forms a part of the Main Propulsion System. Since the external tank is jettisoned just prior to orbital insertion and is expendable, it is designed for simplicity and minimum cost. The design features of the external tank are shown in Figure 43.

The Panel has not had an opportunity to review the external tank in any depth, but shall in future visits to the contractor (Martin Marietta Company) and cognizant NASA Centers.

Specifications call for integral skin/stringer and frame construction and in general uses state-of-the-art technology and manufacturing methods. Proven aluminum alloys will be used for structural areas such as the cryogenic tanks (AL 2219) and the intertank area (AL 7075). The liquid oxygen tank will be a monocoque construction and the liquid hydrogen tank will be in integral skin/stringer and frame type structure. Insulation previously used on the Saturn V program will be sprayed on the external tank. High heat load areas will use ablative materials now in use on other NASA programs.

Tank pressurization will be accomplished through a 4000 psia helium storage system with 750 psia regulation capability for valve actuation and for engine helium requirements.

Technical data necessary to the integration of the external tank into the total Shuttle system includes the following:

1. Propellant motion
2. Mass Properties
3. Tank pressures versus time
4. Spin-up implementation
5. Breakup modes
6. Frangment dispersion

The external tank/Orbiter attachment system and separation modes are most important. The attachment points affect the Thermal Protection System and the residual fluid quantities retained after SSME shutdown in the orbiter and the external tank. External tank/SRB attachment points and separation modes are another example of areas to be reviewed by the Panel.

Solid Rocket Booster (SRB)

The Solid Rocket Booster consists of a Solid Rocket Motor, forward and aft skirts, external tank attachment structure, nose cone, thrust vector control, separation and recovery system (Figure 44), The following data provides dimensional and parametric information:

Length = 145 feet
Diameter = 142 inches
Gross weight = 1,163,500 pounds
Recovery weight = 154,250 pounds
Sea Level Thrust = 2,500,000 pounds
Eight separation motors (4 fwd, 4 aft) of 23,000 pound thrust each.

Here again the Panel has not conducted a detailed review of the SRB up to this time. Reviews in detail will be conducted during the 1974 time period. Based on the material provided to date the following observations can be made:

1. Separation engine impingment on the orbiter requires examination of the contamination effects on the Payload and Payload bay as well as the Orbiter cabin windows.
2. Shuttle system drift at lift-off due to unsymmetrical thrust and thrust vectors of the SRB must be accounted for in the design of the launch installation and the operational conditions.
3. External Tank/Solid Rocket Booster separation system interfaces must assure clean breakaway and insertion into a trajectory conduceive to SRB recovery.
4. SRB recovery challenges that are of interest include the following:
    a. Reentry dynamics
    b. Parachute deployment conditions
        - High roll rates
        - Separation Effects
    c. Water Impact
        - Structural strength
        - Corrosion following water immersion
        - Stability
        - Submergence effects
        - Slapdown effects
        - Entry envelope definition
    d. Retrieval
        - Tracking and Location
        - Safing the ordnance
        - Return to base
    e. Parachute System
        - Reuse due to immersion
        - Possible use of parachute + retro vs. parachute only

Payload Accommodations

The Orbiter vehicle has provisions for a variety of payloads. Payload accommodations include structural support, environmental protection, manipulator arms, electronics for controls and displays, manned access, and as necessary support for guidance and navigation, communications and tracking, data processing, pointing and stabilization, electrical power, fluids and gases. The scope of these accommodations and candidate payloads are shown in Figure 45 for purposes of clarification.

Because of its importance, the payload retention scheme has been of greatest interest to the Panel at this time. These structural supports are vital to retain the payloads during launch "intact abort", and landing and to assure that they minimize the transmission of orbiter stresses, such as bending, into the payloads. This is depicted in Figure 46.

In addition to the Payload retention the Panel will examine the following areas during its future reviews:

1. Payload interface requirements and definition (physical, functional, proceed).
2. Payload center-of-gravity envelope requirements.
3. Payload clearances taking into account Orbiter manufacturing tolerance, structural deflections, and installation requirements.
4. Payload dynamic response at liftoff and coupling with orbiter.
5. On-orbit consumables and their supply from the orbiter.
6. Orbiter/payload ground operations.
7. Payload deployment and retrieval system.
8. Payload cleanliness requirements and possible contamination sources.

<u>Mission Operations and Turnaround</u>

The flight plan and operation of the Space Shuttle differ markedly from that of the now-familiar launch procedures and recovery of the Apollo and Skylab missions, which utilized the expendable Saturn V and IB launch vehicles. All Shuttle missions are characterized by the following phases:

1. Ascent Phase
   a. Main engine ignition and vehicle hold-down
   b. Solid rocket booster ignition and vehicle release
   c. Ascent through SRB burnout and separation
   d. Continued ascent through main engine thrust termination and external tank jettison
   e. Orbital insertion with the burning of the orbital maneuvering system engines.
2. Orbital Phase
   a. Orbital operations including orbit adjustments, payload deployment and retrieval
   b. Retrograde maneuver utilizing the Orbital Maneuvering system engines
3. Descent Phase
   a. High angle of attack reentry
   b. Hypersonic flight with reaction control system and aerodynamic control
   c. Supersonic flight with aerodynamic control
   d. Terminal Area Energy management maneuvers
   e. Landing and rollout

4. Turnaround or refurbishment
   a. Safing
   b. Maintenance and checkout
   c. Vehicle assembly including payload, SRB refurbishment
   d. Prelaunch operations
5. Contingency Operations (for non-normal missions only)
   This includes all operations which are not nominal in the sense that a return to the landing site is required prior to orbit, or the abort is made from orbit. In addition those operations involving emergency conditions occurring during the descent phase.

The Space Shuttle system is design to accomplish a wide variety of missions. Reference missions have been established, which are described below, to be used in conjunction with the specified design requirements to size the Shuttle System and characterize its performance capabilities.

Mission 1:

The objectives of this mission are to deliver a satellite to a circular, geosynchronous, equatorial orbit and then to retrieve another satellite already in such orbit and return to earth.

Payload = 65,000 pounds
Inclination of launch = 28.5^{o}
Orbit = 100 by 50 nautical miles
External tanks impact into the Indian Ocean
Orbit circularized at 150 nautical miles at first perigee
Launch site is KSC
Up to seven days of mission time

Mission 2:

The objectives of this mission are to deliver a refurbishment payload to an orbiting unmanned satellite and then to perform on-orbit experiments.

Shuttle launched into 150 by 50 nautical mile orbit
Launch site is KSC
Launch at 55^{o} inclination
Payload = 25,000 pounds
Orbit circularized first apogee
Mission time up to seven days

Mission 3:

The objectives of this mission are to deliver a payload into orbit (mission 3A) and to retrieve a payload from orbit (mission 3B) in one orbit.

A. Deploy mission
   Launch from Western Test Range
   Launch at an inclination of 90^{o}
   Payload = 40,000 pounds
   Launched into a 50 by 100 nautical mile orbit
   Mission time about 2 hours

B. Retrieve mission
   Launch from the Western Test Range
   Launch at an inclination of 104°
   Payload = 0 pounds in ascent (about 32,000 pounds return)
   Launch into an 81 by 97 nautical mile orbit
   Mission time approximately 2 hours

As currently envisioned the Shuttle will be based at two existing space launch sites, KSC and the Western Test Range (Vandenberg Air Force Base in California). This provides the Shuttle with access to all orbit inclination shown in Figure 47. Variations in orbit altitude and payload weight can be achieved by the addition of one, two or three Orbital Maneuvering System kits in the payload bay aft area which augment the integrat OMS. Each kit provides an increment of 500 feet per second of velocity capability added to the original 1000 feet per second capability.

The system design requirements for reentry, descent, approach, and landing which affect the design and vehicle capability includes the following:

1. Unpowered landings
2. All weather capability (specific restrictions, if any, are not known at this time)
3. Payload for normal return = 32,000 pounds
   Payload for abort conditions = 65,000 pounds
4. Center of Gravity
   The forward CG limit must be controlled at entry and the aft CG limit must be controlled at landing. The CG travel must stay within a 2% body length travel distance, that is between 65% and 67% of body length (approximately).
5. Fail Operationally/Fail Safe redundancy independent of manual control backup capability.
6. Safe rollout to stop with one braking system failure
7. Approach and Landing Design Winds:
   Approach initiation approximately 82 knots at 10,000 ft.
   Touchdown approximately 34 knots
8. The navigation errors permitted for the flight control system components (radar altimeter, inertial measuring unit, etc.).
9. Autoland capability with provisions for manual takeover.

The Orbiter entry and return flight profile is as indicated in Figure 48. The Orbiter is oriented to the proper attitude for the deorbit propulsive maneuver to begin the entry trajectory. The RCS engines maintain the vehicle in proper orientation. RCS alone is used for control until the dynamic pressure reaches at least 20 pounds per square foot. At this point the aerodynamic surfaces begin to provide some reaction.

The orbiter entry flight path angle is -0.82 degree at the theoretical entry interface at 400,000 feet altitude. An angle of attack of about 30 degrees is maintained for the entry period. A transition to 10 degrees is begun when the velocity has been reduced to about 8000 feet/second at approximately 150,000 feet altitude, and completed when the velocity is about 1,500 ft/second at 60 or 65,000 feet altitude. Both the RCS and aerosurface control is used through the angle of attach transition region. Shortly after entry the aerodynamic heating ionizes particles of the atmosphere causing a temporary blackout of communications. After emerging from the temporary blackout condition the trajectory position is updated using the TACAN system.

During the final phase of descent the control of the flight path is maintained using the aerodynamic surfaces. These surfaces are depicted in Figure 49. The terminal area energy management (TAEM) is initiated at approximately 70,000 feet altitude and an autoland system controls the final approach and landing phase.

Autoland is an automatic landing system which provides automatic guidance, navigation and control from approach to end of the landing roll. Greater dependence on automated control appears necessary for the shuttle because the orbiter's unpowered approach and landing requiring precise control, the vehicle does not have inherent stability, approach and landing speeds are reasonably high and the time for reaction fairly short, and there is no go-around capability. The hardware necessary for autoland is already onboard the orbiter: radar altimeter, basic flight control system, Guidance and Navigation system, computers, etc.

Requirements for approach and landing are described as: (Also see Figure 50)

1. Time from pre-flare to touchdown equal to or greater than 25 seconds
2. Sink rate equal to or less than 10 feet/second
3. Runway 10,000 feet by 150 feet
4. Tire limit of 210 knots air speed
5. Landing speed of approximately 180 knots

The Shuttle entry "footprint" capability is shown in Figure 51. The cross-range capability is 1,100 nautical miles and the limits downrange are as shown in the figure.

The Shuttle system will provide, by intact abort, the safe return of personnel, payload, and orbiter vehicle. Abort requirements are noted below (these are current as of the date of the Panel's visit to RI on Nov. 25-26, 1973):

1. Emergency egress to be accomplished by personnel in less than two minutes.
2. Intact abort modes (failures)
   a. Complete or partial loss of one SSME
   b. Loss of thrust vector control on one SSME
   c. Loss of thrust of one OMS engine
   d. Loss of thrust vector control on one Solid Rocket Booster in one axis.
3. Contingency abort modes (failures). The extent of mission completion and method of assuring crew safety have not been established.
   a. Loss of thrust from 2 or 3 SSME's
   b. Loss of thrust vector control on 2 or 3 SSME's
   c. Loss of SRB thrust vector control in two or more axes

d. Premature separation of the orbiter from SRB and ET
e. Failure of the Orbiter/ET to separate.

An example of an abort mode is described here when there is a failure of 1 SSME (note item 2a above). The type of abort, if any, is dependent upon the point in the ascent trajectory in which the failure occurs, in this case one SSME. With the SRB's and two SSME's operating properly the Shuttle vehicle would continue its ascent until the SRB's burnout and separation from the Orbiter/External Tank combination. The available options, depending upon conditions, are as follows.

1. Return to launch site with powered flight using the remaining two SSME's plus the OMS and RCS. The external tank would be dropped at a point about 50,000 feet altitude with normal unpowered glide approach and landing, or
2. Change the orbital insertion target point to achieve a free return orbit by continuing ascent after SRB burnout and separation using the remaining SSME's plus OMS and RCS. This places the vehicle in the so-called "Abort Once Around" trajectory. Again the external tank is separated at SSME burn out. The orbiter than reenters in a manner similar to that used in the nominal orbital mission and approaches and lands.
3. If desireable and possible continue the ascent with the use of the remaining engines to achieve orbit and continue mission.

The Shuttle system design is to provide for orbiter turnaround, from landing to liftoff on the next mission, in 14 days. Current plans call for 160 working hours during that time on a two-shift, five-day-week basis. The turnaround cycle is shown in Figure 52. The hours shown in the "pie" may be cut differently as the design progresses but does indicate the depth of work to be accomplished for each operation.

To meet the requirements for quick turnaround the hardware design incorporates the use of modular subsystem assemblies; ready access for removal, modification and reinstallation; ready access to hardware for checking and testing for faults. Designs must also permit parallel rather than series operations during the turnaround period.

The solid rocket boosters are the first of the elements to be mounted on the mobile launch platform (this platform was the Apollo program "crawler transporter"); the first step in the assembly of the Space Shuttle Vehicle. This is followed by the mating of the external tank to the SRB's, and then the mating of the Orbiter to the external tank and SRB's. After rollout of the vehicle to the launch pad, operational interfaces are connected and countdown preparations are completed with loading of cryogenics, crew ingress, and final automatic countdown sequencing.

Comments on the Shuttle Operations

The following coments are given based on the material provided to the Panel during fact finding sessions conducted at NASA and Contractor sites.

1. The role of "man-in-the-loop" is continuing to be discussed. This involves consideration of the role in the nominal operational mode, the role in contingency and emergency operations, and the associated development of the necessary control and display provisions to support the crews ability to form and act upon independent judgements. Some flight regimes are time critical (e.g., during the approach and landing period from about 10,000 feet to touchdown)and require automatic failure detection and switching,particularly for any failure requiring corrective action with about 5 seconds or less. In determining the appropriate role for the crew the following areas should be considered: Stability and control analyses of the vehicle, required man/machine interfaces, aerodynamic variations, control system mechanization, simulation requirements.

2. Those risks attendant to orbiter flight and ground operations in adverse weather conditions (rain, lightning and thunderstorms) need to be examined to assure that systems such as the TPS and avionics maintain their operational integrity. During emergency returns from "orbit once around", and even on nominal missions, the orbiter may encounter weather conditions other than those planned for the primary and alternate sites.

3. Precise knowledge of the aerodynamic parameters that design the hardware and the flight path capabilities and allowable variations will impact the weight of hardware used in associated systems.

4. There appears to be a need to further establish the optimum approach to maintaining lateral-directional orbiter control when the vehicle is at large angles of attack. Large angles of attach create a condition wherein the aft aerodynamic surfaces are "placed in the shadow" of the forward wing and fuselage.

5. Support for the Shuttle missions requires data management, acquisition, processing, reduction, and distribution of data. The Panel, as it has done in past reviews of the Apollo and Skylab programs, will review this area as to: definition of network schedule and configuration to support data requirements; data routing and distribution; data processing as required; and data cataloging and overall management.

6. Extra Vehicular Activities (EVA's), and their safety for the crew and mission success implications, will also be examined by the Panel in future fact-finding sessions.

7. Jettison of the external tank and separation of the solid rocket booster are closely related to the operational adequacy of the orbiter as an individual vehicle and to the combine Shuttle system. Jettison of these units must be clean and on time, and without mechanical or thermal trauma to the orbiter surfaces and closures. The system to accomplish this will be reviewed by the Panel to assure that management at all levels considers the many aspects in achieving fail operational/fail safe reliability; no recontact; minimized contamination of the TPS and Payloads; integrated vehicle loads acceptability.

8. While it is clear that the 160 hour turnaround time is driving many design features it is not clear at this time whether this constraint is necessary or attainable nor what the impact would be of relaxing this requirement.

Reliability, Quality, and Safety

This section is devoted to those activities that deal with designing against the loss of critical functions, definition and resolution of hazards, subcontractor and vendor control, qualification testing, materials control and the like.

The basic policy for the Shuttle R,Q & S activities, NHB 5300.4(1D) dated December 1972, establishes the responsibilities for programmatic offices at both NASA and the Shuttle contractors. Policy and requirements defined in the above document provides for the planning, organizing, conducting, and evaluating of the activities to ensure the required levels of reliability, quality and safety in the design and utilization of the Shuttle System equipments and supporting actions. These Shuttle requirements have been derived from:

1. Experience from prior NASA programs such as Apollo and Skylab
2. Recommendations of Contractors
3. DOT's FAA Commercial aircraft practices
4. NHB 5100.2, NASA Procurement Regulations, Part 1, Subpart 52, "Safety amd Health"
5. NHB 5300.4 (1A), "Reliability Program Provisions for Aeronautical and Space System Contractors"
6. NHB 5300.4 (1B), "Quality Program Provisions for Aeronautical and Space System Contractors"
7. NHB 1700.1, "NASA Safety Manual, Volume I"

Analyses of past programs, and based on Shuttle requirements, the Shuttle approach to R,Q&S has been stated as follows:

1. Consistent provisions applied on all program contracts
2. Added the basic elements of maintainability
3. The same provisions are selectively applied by prime contractors to lower tier procurements from common suppliers
4. Standard criteria at interfaces for use in the Failure Mode and Effect Analyses (FMEA) and Hazard Analysis techniques and problem reporting
5. Standard non-destructive evaluations and tests methods and equipment
6. Reduced government inspection at prime contractors and their suppliers
7. No requirement for numerical goals, apportionments, predictions (This is discussed in Volume I, Appendix H)
8. Develop the ability to trace (materials, processes and decisions) utilizing normal programmatic documentation without developing a specific traceability system
9. Reduction in the type, number and distribution of documents related to the R,O&S activities

Configuration management is an integral part of the technical management system and is closely related to the R,Q,&S activity. Configuration management is the controlling system for identification of the NASA requirements baseline, control of changes to the NASA baseline, tracking of changes to the NASA baseline, verification of proper implementation of NASA baseline requirements, requirements for identification, control, tracking and verification of the configuration of system hardware and software. Figure 53 indicates the configuration management structure and responsibilities. This is not unlike the system utilized on the Skylab program which streamlined those configuration management activities utilized by the Apollo program.

Rather than discuss the many aspects of R,Q&S, the requirements, organizations, and method of implementation, it appears more worthwhile to discuss those areas which are of concern to the Panel. It is toward these concerns that the Panel will focus its review activities in the future.

Designing against the loss of critical functions in the Space Shuttle includes the definition of critical functions themselves, FMEA's, Hazard Analyses and the resultant critical items list and residual hazards tracking process. Redundancy requirements have been defined to assure design features protect against loss of critical functions so that all flight systems shall not be less than fail-safe. This excludes such things as the Primary structure, thermal protection system, and pressure vessels. Fail-safe is the ability to sustain a failure and retain the capability to successfuly terminate the Shuttle mission. The results of these efforts are of continuing interest.

Based on their Apollo and Skylab experience the Panel will continue to explore those hazards associated with the following:

1. Energy sources and fire propagation paths
2. Utilization of shatter-proof materials in the crew compartment
3. Elimination of sharp edges in the habitable areas
4. Protection of pressure vessels and protection from pressure vessels
5. For cryogenics and propellants: their characteristics, hazard levels, handling, storage, and compatibility. For example, the aft portions of the orbiter, whose working life is intended to approximate ten years, are subject to exposure to highly corrosive N_2O_4 on the pad, in flight, and after landing, as well as to concentrated nitric acid formed by reaction of N_2O_4 with atmospheric conditions where moister is present.
6. Use of explosive devices (pyrotechnics)
7. Effects of fault currents, transient currents
8. Long-term storage; corrosion, fatigue, aging
9. Electromagnetic radiation and ionizing radiation

In using off-the-shelf-hardware there is some concern that the existing data on some of those items may not be sufficient to support the planned use and the age-life requirements. Therefore the question of qualification programs for such equipment will be reviewed by the Panel. The qualification procedures developed in the early days of NASA Apollo and Skylab were made to insure proper operation of a new device that was used fairly promptly.

To assure that the qualification procedure is sufficient for say a ten-year old piece of hardware it may be necessary to supplement the procedures.

Another aspect of this is that new hardware manufactured to an old specification in order to make it unnecessary to requalify may well be ignoring ten years of progress in the pertinent technology. This may not be as applicable to mechanical devices but it applies to electronics equipment.

The aft fuselage section of the Orbiter, which houses the SSME's, OMS, and RCS units represents a possible fire hazard because of the large, complex high pressure systems carrying flammables. Leakage integrity, including control of any leakage, requires propellant communication barriers in the fuel injector, turbopumps, preburners, and valves. Caution and warning devices and engine instrumentation must also be supplied to announce conditions and permit remedial actions to be taken.

Subcontractor and vendor control appears to be a concern from several viewpoints. First, lack of interest by suppliers, escalated costs, and long lead times required to procure raw materials and finished products. Second, source surveillance and ability to provide for corrective action. Lack of interest in responding to requests for proposals and procurement requests has been experienced by the SSME and Orbiter prime contractors in those areas which require short run quantities or where the hardware requirements appear to the bider to be too stringent. The impact of the current energy crises, environmental requirements and general economic conditions has been felt in escalated costs and material lead times. Subcontractor/Supplier quality assurance activities will most likely require a higher degree of planning to make the best use of the personnel available for surveillance and compliance and to reduce the impacts of material and energy shortages.

Material control for the Shuttle program is based on the requirements defined in the following documents and the NASA Center and contractor material identification, tracking and reporting systems developed in-house.

Requirements for Materials and Processes: (NASA Documents)

Flammability	NHB 8060.1
Toxicity	NHB 8060.1
Vacuum-Condenable Material	SP-R-0022
Age Life	SE-R-0006A
Fluid compatibility	NHB 8060.1
Corrosion	MSFC-SPEC-250
Stress Corrosion	MSFC DWG 10M33107
Fracture Control	SE-R-0006A

The Orbiter contractor utilizes the MATCO (Material Analysis, Tracking & Control) which is a central computer system for both "as-designed" and "as-built" configurations for the Space Shuttle. It is essentially the same system as that used on the Apollo program but upgraded to enhance the control and reporting system and retrieval capability.

Some experiences taken from the "Apollo Experience Report - Reliability and Quality Assurance", NASA TN D-7438, September 1973 which can be useful as background data in future Panel fact finding sessions are noted here. These are quoted directly from the NASA report.

1. "---experience indicates that the services of Government agencies can be used most effectively if timely quality assurance direction and assistance are given and points of contact are provided. This direction includes interpreting R&QA requirements, evaluating the adequacy of fabrication and inspection processes, providing guidance in the preparation of procedures, assisting in the disposition of nonconforming equipment, and resolving any problem or question the Government agency may have in performing the delegated functions.

2. Experience in the area of documentation has shown that the scope, format, and content of R&QA plans must be specified in the contract to preclude arbitrary changes in acceptance criteria. To be fully effective, these plans must be established in the contract as documents requiring customer approval.

3. Contractors are required to document the soldering program including information regarding qualification of instructors, procedures for training lesson plans, instruction hours, and procedures for the certification and recertification of solder personnel. Early development and implementation of the training program and procedures are requisites to ensure the availability and maintainability of reliable equipment.

4. Some of the more significant problems encountered in the fabrication, installation, and testing of fluid systems are:

a. Pressure vessels: The large size and thin walls make these tanks awkward to handle and thus extremely susceptible to inadvertant damage. Stress corrosion problems show that controls must be implemented to assure that fluids used in pressure vessels for testing and cleaning are compatible with the tank material. Special controls are necessary to assure the elimination of weld cracks. Detailed historical records (cards) are valuable in the investigation of tank failures.

b. The quality of the fluids used in the Apollo spacecraft systems was vital to the reliability of the systems (this was borne out in the Skylab program as well); failures of equipment were attributed to impurities and contaminates in the fluids. Whenever changes are made to manufacturing processes, handling procedures, or procurement sources for any fluids, consideration should be given to requalifying a system that has been previously qualified using a particular fluid.

5. Electrical connectors utilize large numbers of closely spaced, small diameter pins. Problems encountered included crimping of small connector pins, moisture proofing the connectors, bent pins, and verification that connectors mated properly.

6. Experience indicates that unless a comprehensive and well-integrated test plan is prepared, much unnecessary testing can be performed and some may be missed. Of greater importance from a reliability standpoint is the fact that testing may be performed at the wrong assembly level."

Major Ground Test Programs

The Shuttle ground test program, which draws heavily upon the experience gained during past aircraft and space development programs, has been structured to strike a balance between development risks and costly test hardware procurement (including facilities). Each test is considered part of a closed loop verification process, leading to hardware acceptance and certification prior to flight test.

Space Shuttle Test Program Facilities are shown in Figure 54. These, of course, are the major test facilities with many other in-house and contractor facilities being utilized on a day-to-day basis in the hardware development. The major ground test programs are:

1. Main Propulsion Test at Mississippi Test Facility (MTF)
    a. Main engine systems
    b. Flight weight external tank
2. Structural Tests at Palmdale
    a. Complete Orbiter
    b. Primary/Secondary Structure
    c. Limit and Ultimate Loads
    d. Structural Fatigue
        (1) 500 fatigue cycles (125 missions x 4 scatter factor)
3. Ground Vibration Test
    a. MSFC
        (1) Orbiter/External Tank
        (2) Dummy Orbiter/External Tank/Solid Rocket Booster
        (3) Dummy Orbiter/External Tank
    b. RI, Downey, California
        (1) 1/4 scale model of Shuttle vehicle
    c. Palmdale, California
        (1) Orbiter
4. Thermal Vacuum Tests at JSC
    a. Forward Fuselage of Orbiter
    b. Aft fuselage of Orbiter
5. Vibro/Acoustic Tests at JSC
    a. Forward Fuselage of Orbiter
    b. Aft Fuselage of Orbiter
6. Hydraulic/Flight Control Test at RI, Downey
    a. Vehicle Hydraulic System.

To reduce the cost of ground tests, available test facilities are being modified to satisfy Shuttle test requirements. For example, propulsion tests of the Orbiter's main propulsion system will be conducted at the NASA Mississippi Test Facility on a test stand formerly used for the Saturn V propulsion testing. (See figure 55 for a pictorial representation.)

The auxiliary propulsion system, consisting of the orbital maneuvering and reaction control systems, must undergo extensive development and qualification tests, including hot firings, under simulated space altitude conditions.

The Shuttle avionics subsystems must function together reliably during launch, space flight, reentry, and return aerodynamic conditions. The capability to develop, integrate, and to verify the overall flight avionics systems is to provided by the modifications for the Shuttle Avionics Integration Laboratory (SAIL) in building 16 at the Johnson Space Center. At this location, the Shuttle avionics can be integrated with mission control, world-wide communications systems, software development and crew training facilities already in place at JSC and will continue to be available for support during development and operational flights. Tests will range from basic non-redundant integration tests to total system testing. The combined test article, test laboratory and simulations laboratory complex will allow a piloted "flight" mission of the avionics systems for verification and evaluation.

The Thermal Protection System test logic provides for:

1. Testing of competive design concepts and testing to derive design performance data.
2. Evaluation of selected concept to complete design which cannot be satisfied by analysis.
3. Life verification tests of selected Thermal Protection System regions, requiring special hardware, such as a convective heating test.
4. Life/performance verification tests on flight-type hardware, which in general are not performed during the development test program.

Representative vehicle areas to be tested are shown in Figures 56 and 57.

In summary the primary objectives of the verification process includes (1) support to development of design, (2) certification that the design components, assemblies, and subsystems meet performance requirements, (3) verification that the performance of combined subsystems, elements, and combined elements meets established requirements, and (4) demonstration of the acceptability and readiness for intended use of deliverable hardware and software. Where analysis does not provide reasonable assurance that a candidate design or procedure is adequate, a development testing program is to be implemented. Ground support equipment system verification is conducted in a manner similar to that used for flight equipment.

Flight Test Program

The current flight test program may, for Panel purposes, be divided between the Horizontal Flight Test and the Vertical Flight Test programs.

The horizontal flight test program is expected to provide the data for evaluation of vehicle subsonic performance, stability and control, and subsystem operation to verify the capability of the vehicle to meet airworthiness and performance requirements dictated by the terminal phases of the operational and ferry missions. This program involves vehicle ground tests prior to the first flight, preliminary flight evaluation, performance and flying quality investigation, subsystem verification, and demonstration of the unpowered terminal flight phase. In general, the subsystems required exclusively for space operations will not be provided with the exception of TPS installations. The majority of the Orbiter subsystems will be in the operational configuration for an early evaluation of maintainability provisions and design features as well as development and refinement of operational turnaround provisions.

The Vertical Flight Test program consists of vertical launching, orbital insertion, entry and landing of the Space Shuttle from the KSC. This operational site is shown in Figure 58. Six vertical flight test missions are planned, to verify ascent control dynamics; separation of the Solid Rocket Boosters and External Tank; impact and recovery of the solid rocket boosters; external tank entry conditions; OMS and RCS operation in orbital insertion and orbit/deorbit control; entry control and heating; terminal area management; sero power approach and landing. This will also permit the testing of the ground support equipment and mission control equipment.

TABLE I

APOLLO SOYUZ TEST PROJECT WORKING GROUPS

Working Group 0	Technical Project Director - General Technical Management
Working Group 1	Mission Model and Operations Plans - Trajectories - Crew Activities and Plans - Training - Experiments
Working Group 2	Guidance and Control - Spacecraft to spacecraft rendezvous tracking req'mts - Docking aids - Optics and orientation lights - Control systems
Working Group 3	Mechanical Design - Docking system - Hatches - Connector - Installation
Working Group 4	Communications and Tracking - Spacecraft to spacecraft and spacecraft to earth voice communications - Spacecraft to spacecraft radio tracking equipment - Cable communications for voice and television
Working Group 5	Life Support and Crew Transfer - Equipment and conditions affecting crew transfer

TABLE II

ASTP SAFETY ASSESSMENT REPORTS FOR APOLLO

1. Safety Assessment report for the Apollo structural ring latches
2. Safety Assessment Report for Apollo propulsion and control systems
3. Safety Assessment -- Fire Safety and Flammability
4. Safet Assessment Report for Apollo Pyrotechnic Devices
5. Safety Assessment Report for Apollo Cabin Pressure
6. Safety Assessment Report for Apollo Manufacturing Test and Checkout Flow
7. Safety Assessment Report for Apollo Radio Command System

ASTP SAFETY ASSESSMENT REPORTS FOR SOYUZ

1. Safety Assessment Report for the Soyuz Structural Ring Latches
2. Safety Assessment Report for Soyuz Propulsion and Control Systems
3. Safety Assessment Report for Soyuz Fire Safety and Flammability
4. Safety Assessment Report for Soyuz Pyrotechnic Devices
5. Safety Assessment Report for Soyuz Cabin Pressure
6. Safety Assessment Report for Soyuz Manufacturing, Test and Checkout Flow
7. Safety Assessment Report for Soyuz Radio Command System

ASTP INTERFACE ENGINEERING DOCUMENTS DIRECTED TO SAFETY PROVISIONS

1. Fire Safety Control - Soyuz
2. Apollo Atmosphere Toxicological Requirements
3. Soyuz Atmosphere Toxicological Requirements
4. Fire Safety Certification - Soyuz
5. Fire Safety Certification - Apollo

TABLE III

ASTP EXPERIMENTS
LIST OF US APPROVED EXPERIMENT

- JOINT EXPERIMENTS
    - UV ATMOSPHERIC ABSORPTION
    - DOPPLER TRACKING
    - MULTIPURPOSE FURNACE
    - MICROBIAL EXCHANGE
- UNILATERAL EXPERIMENTS
    - UV SURVEY
    - HELIUM GLOW
    - SOFT X-RAY
    - ELECTROPHORESIS
    - BIOSTACK
    - CELLULAR IMMUNE RESPONSE
    - POLYMORPHONUCLEAR LEUKOCYTE RESPONSE
    - LIGHT FLASH

TABLE IV

SHUTTLE PROGRAM CONTRACTS

- Orbiter/System Integration - R.I. Space Division
  - Flight control systems------------------------Honeywell
  - Data processing & software requirements--------IBM
  - Orbital maneuvering system pods---------------MDAC
  - Vertical stabilizer---------------------------Republic
  - Wing---------------------------------------Grumman
  - Mid-fuselage--------------------------------General Dynamics
  - Ground maintenance & operations support--------American Airlines

- Main Engine - R.I. Rocketdyne Division
  - Controller-----------------------------------Honeywell
  - Hydraulic actuator---------------------------Hydraulic Research Inc.

- External Tank-------------------------------------Martin Marietta Corporation

- Solid Rocket Booster------------------------------Thiokol Chemical Corporation
  (Solid rocket motor; the total SRB to be defined later)

APOLLO SOYUZ TEST PROJECT MAJOR MILESTONES

1. USA/USSR Approval of Project
2. Joint Testing of Docking System (Scale Models)
3. Joint Testing of Full Scale Docking Systems
4. Joint Testing of Communications Systems
5. Preflight Check of Compatible Systems
6. Familiarization of Crews
7. Training of Soyuz Crew in USA
8. Training of Apollo Crew in USSR
9. Training of Flight Controllers
10. Control Center Tests
11. Flight Readiness Review
12. Launch

Figure 1

APOLLO/SOYUZ TEST PROJECT
MASTER PROGRAM SCHEDULE

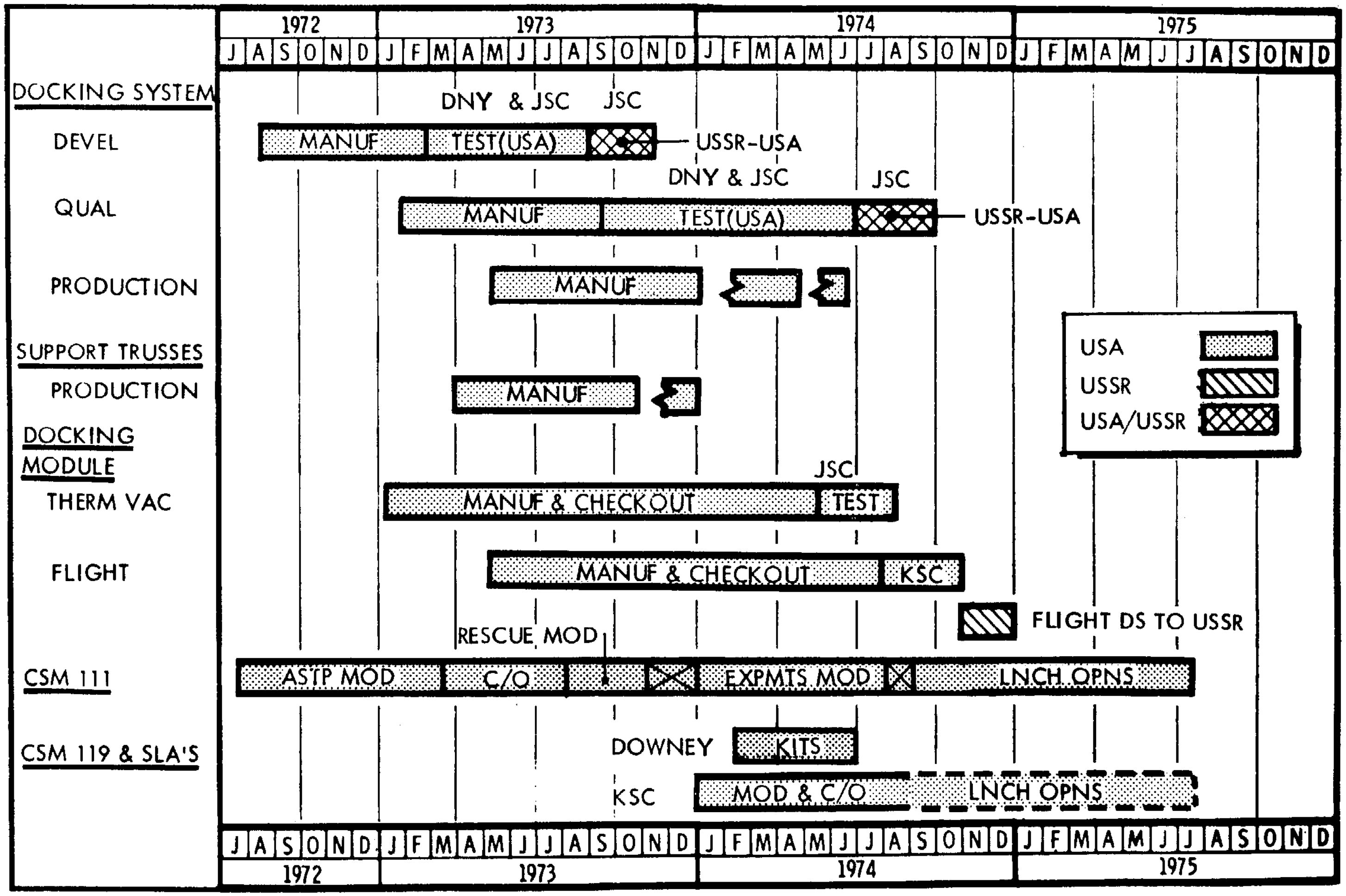


Figure 2

GROUND TESTS, MOCKUPS, TRAINERS AND SIMULATORS

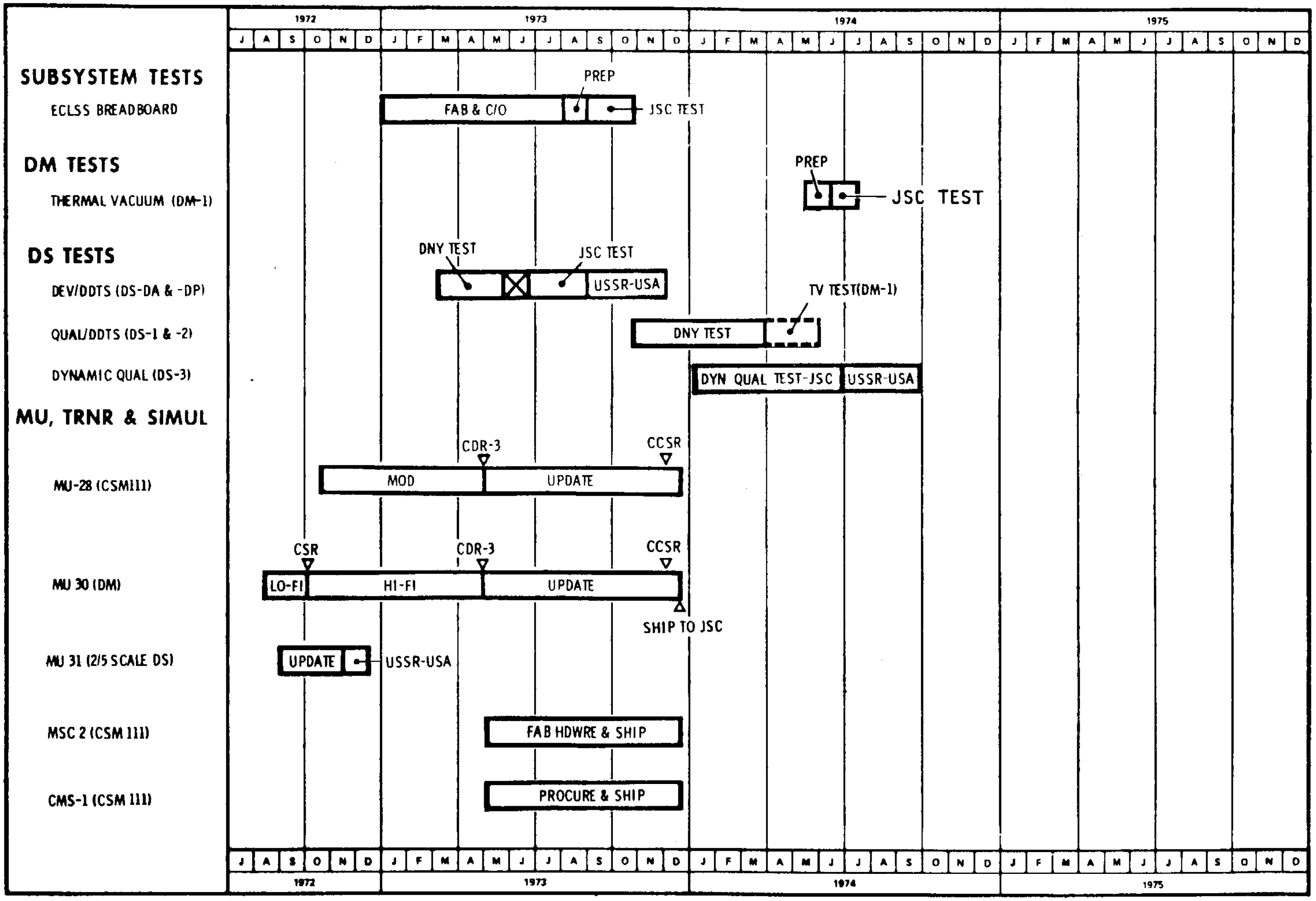
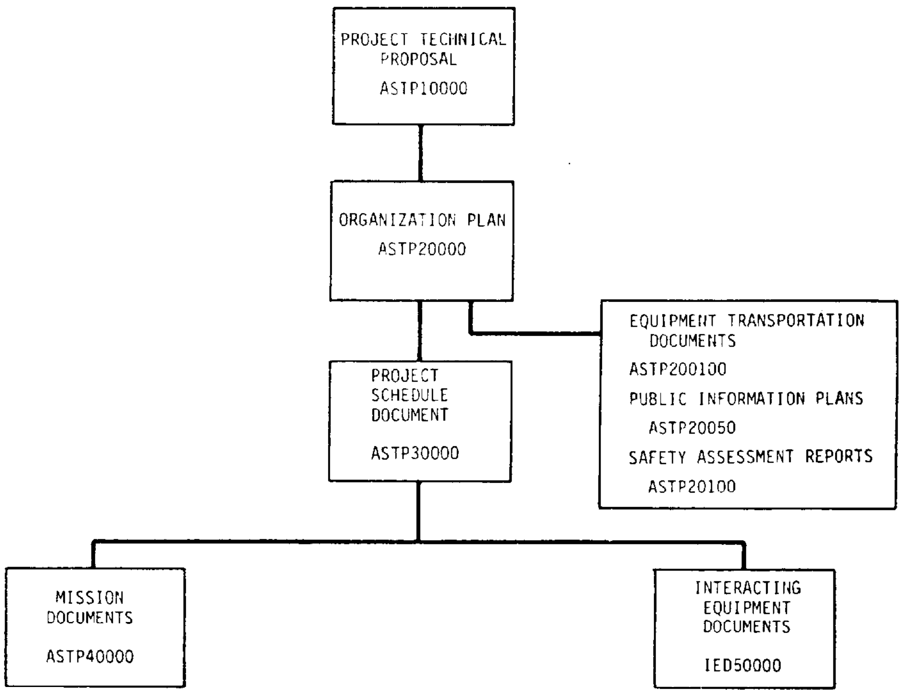


Figure 3

ASTP
PROJECT DOCUMENTATION



MISSION REQUIREMENTS
ASTP40000

LAUNCH WINDOW PLAN
ASTP40100

TRAJECTORY PLAN
ASTP40200

FLIGHT PLAN GUIDELINES
ASTP40300

MISSION OPERATIONS PLAN
ASTP40400

CONTINGENCY PLAN
ASTP40500

ON-BOARD JOINT OPERATIONS INSTRUCTIONS
ASTP40600

CREW AND GROUND PERSONNEL TRAINING PLAN
ASTP40700

POST-MISSION REPORT
ASTP40800

DOCKING MECHANISM
IED50000

COMMUNICATIONS AND TRACKING EQUIPMENT
IED50100

DOCKING AIDS
IED50200

OPTICS AND ORIENTATION
IED50300

CONTROL SYSTEMS
IED50400

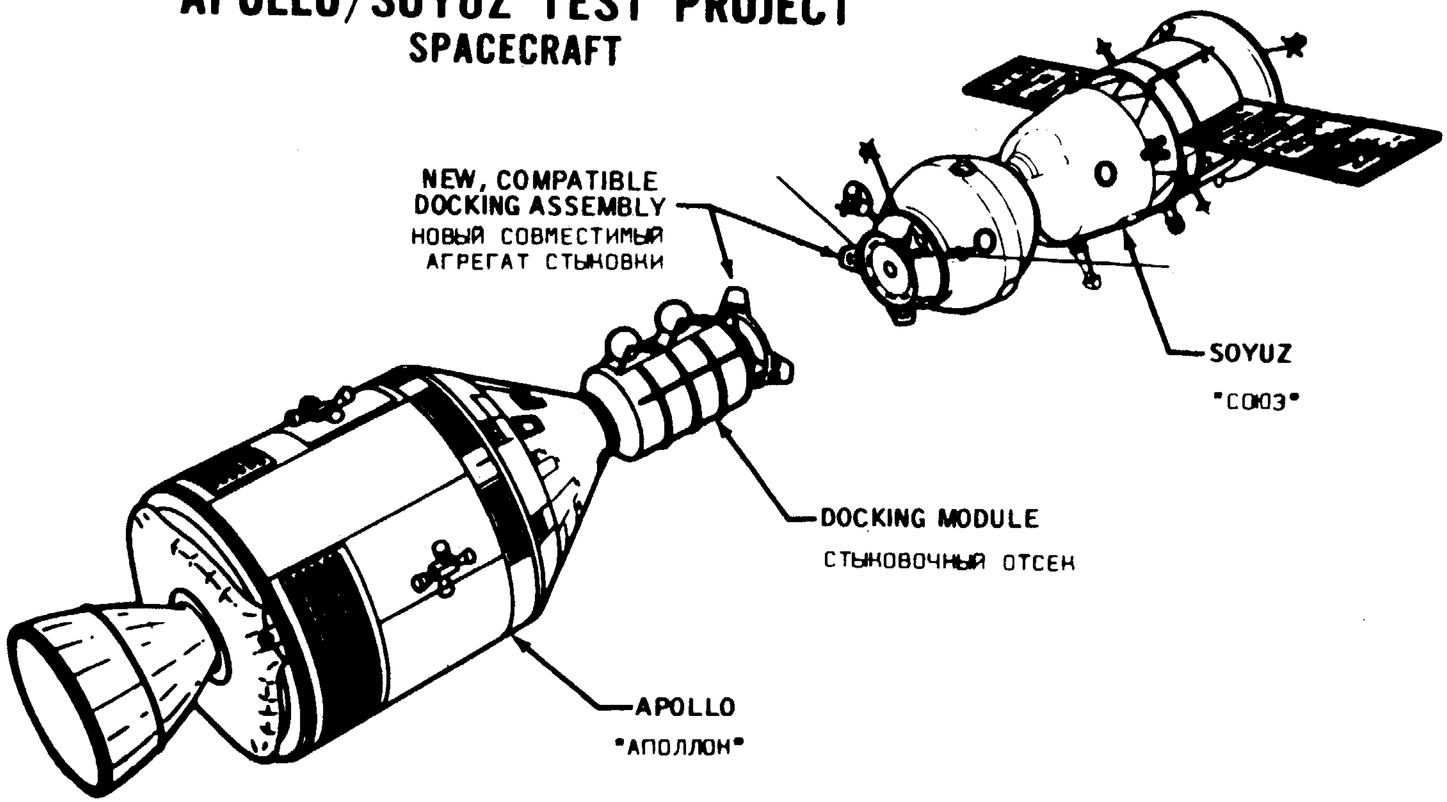
INTER-CONTROL CENTER COMMUNICATIONS
IED50500

CABLE COMMUNICATIONS AND TELEVISION
IED50600

CONDITIONS AND EQUIPMENT IN CREW TRANSFER
IED50700

Figure 4

APOLLO/SOYUZ TEST PROJECT
SPACECRAFT



APOLLO-SOYUZ RENDEZVOUS AND DOCKING TEST PROJECT

ЭКСПЕРИМЕНТАЛЬНЫЙ ПРОЕКТ СБЛИЖЕНИЯ И СТЫКОВКИ "АПОЛЛОН/СОЮЗ"

Figure 5

ASTP LAUNCH CONFIGURATION

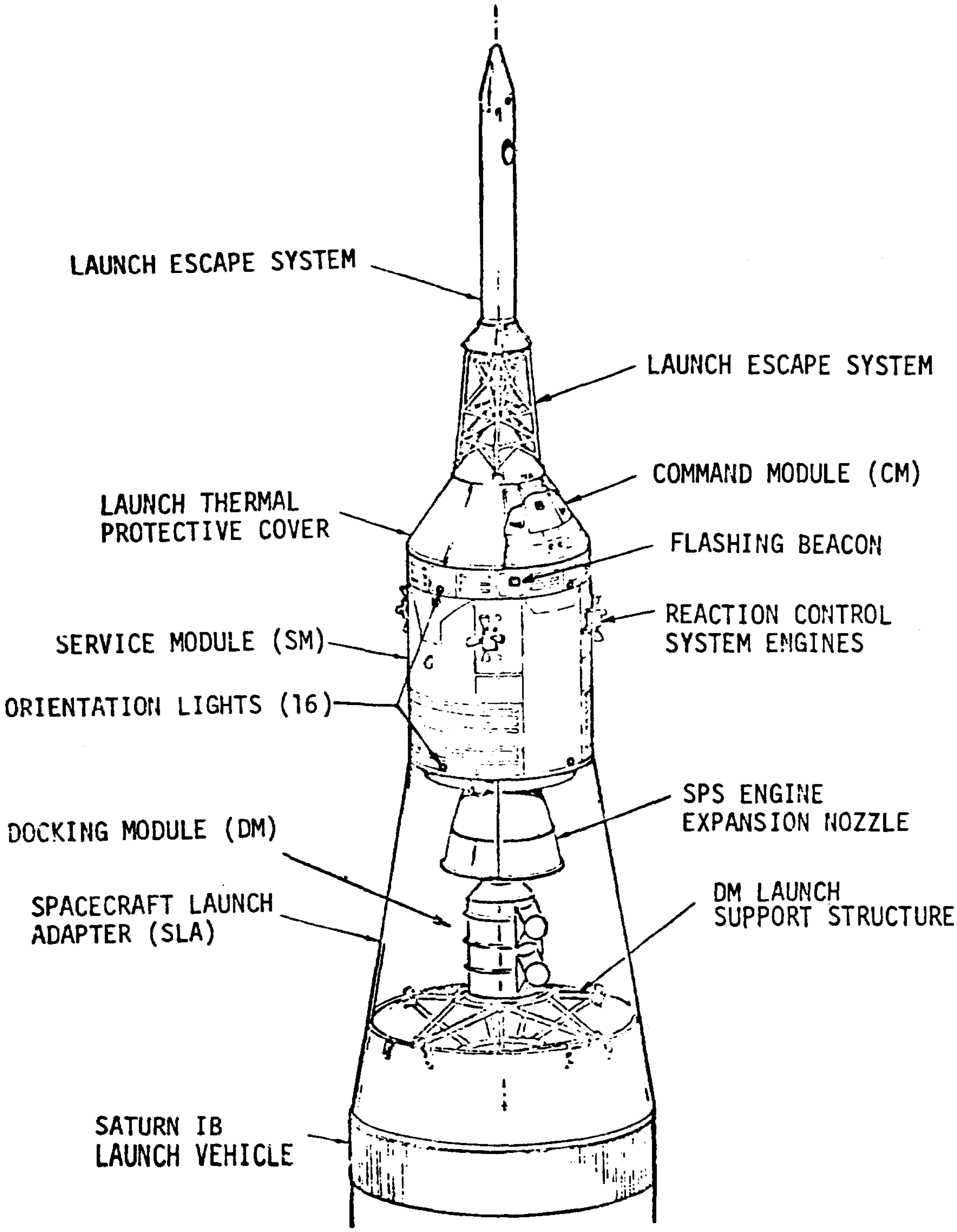


Figure 6

APOLLO/SOYUZ - TEST MISSION RADIO COMMUNICATIONS LINKS

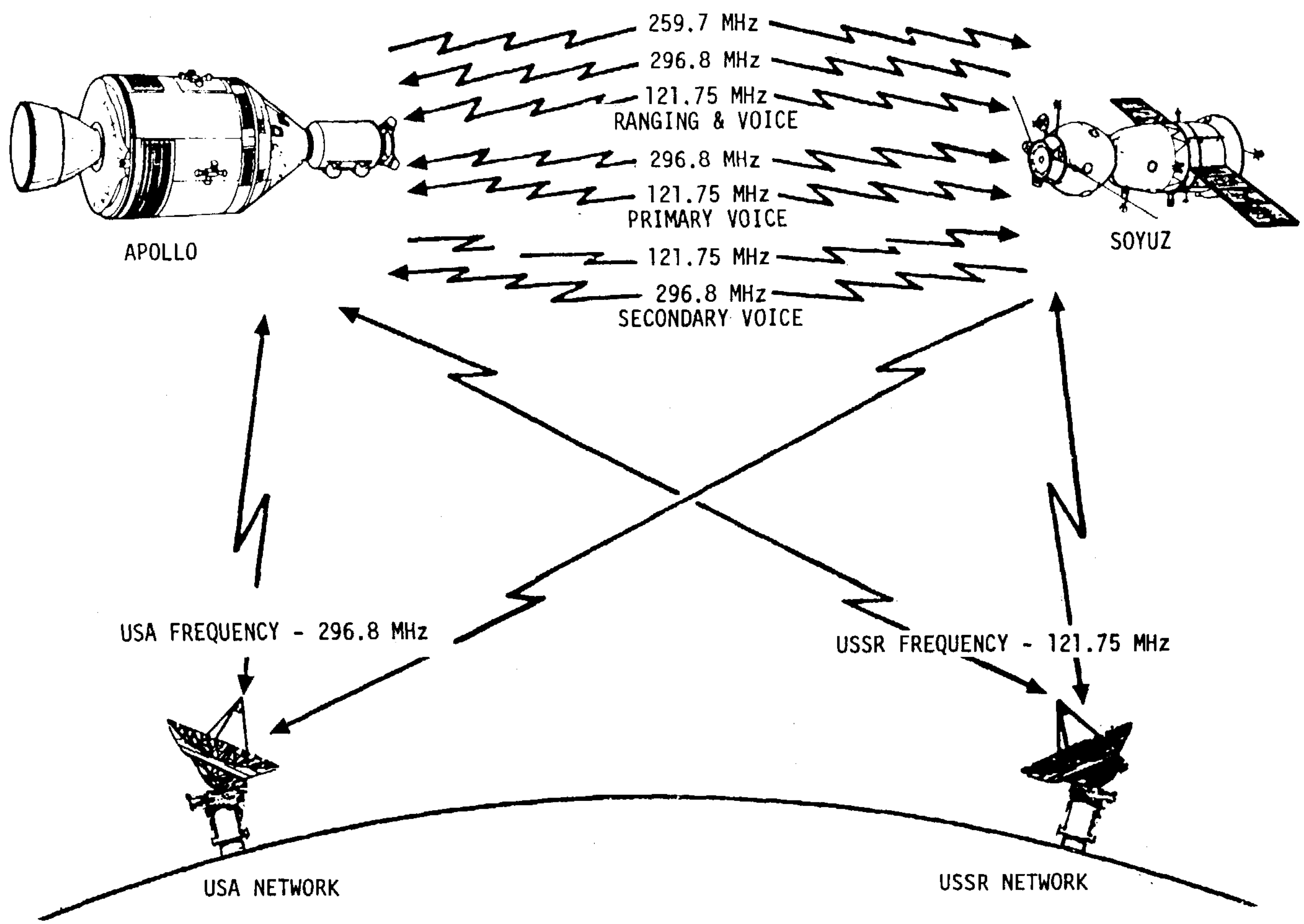


Figure 7

DOCKING MODULE - SOYUZ END

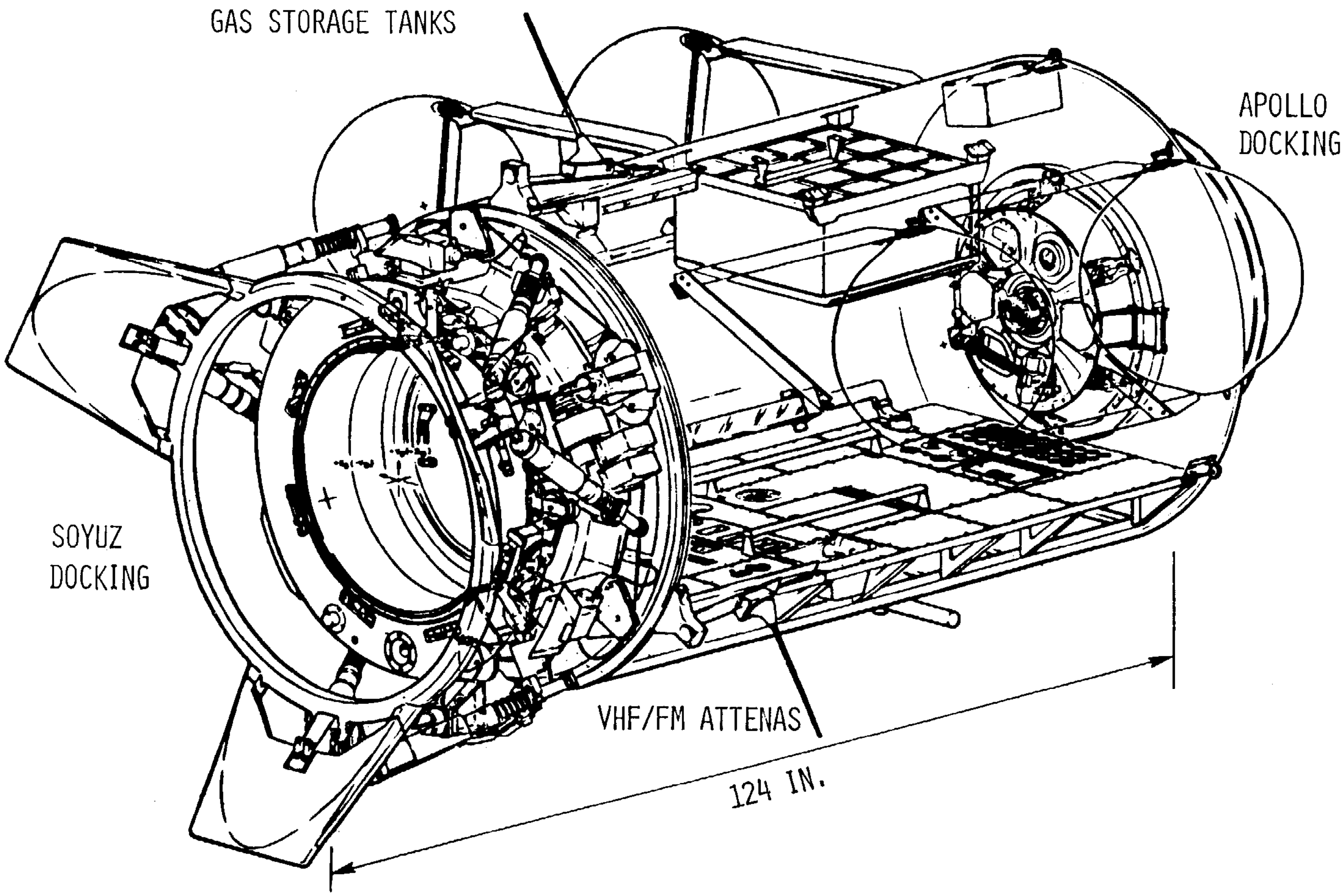


Figure 8

ASTP CABLE COMMUNICATIONS SYSTEM

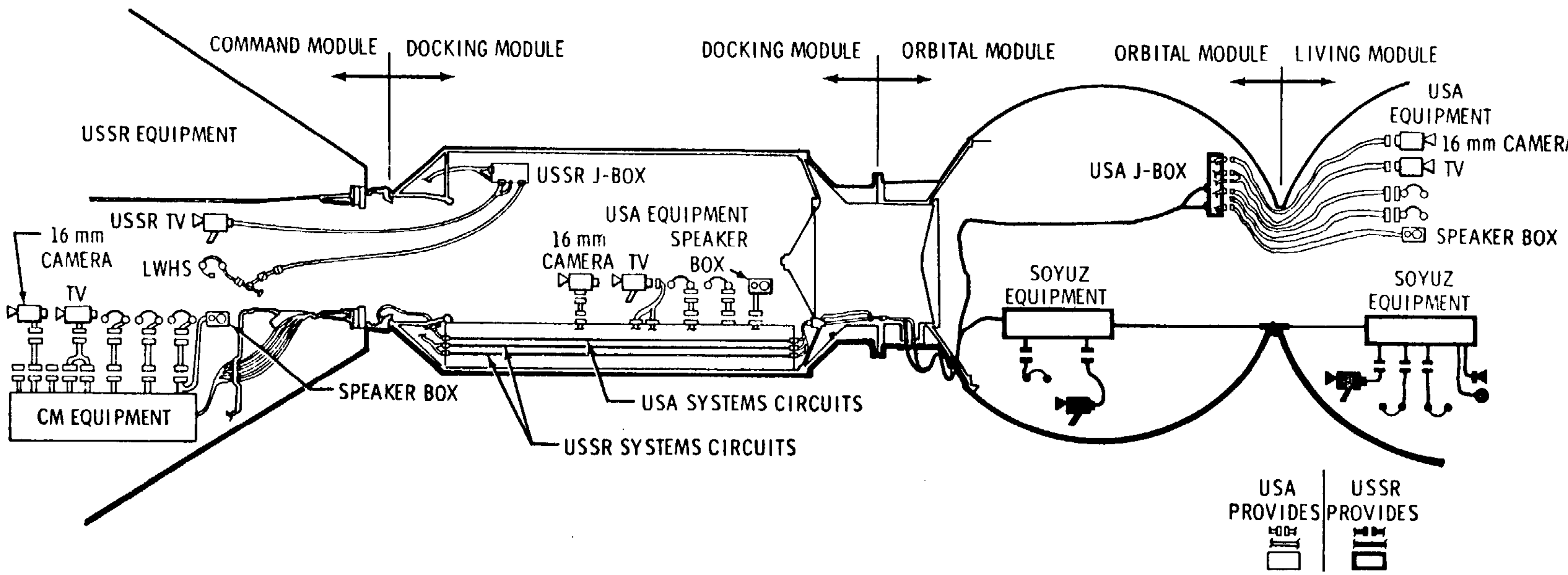


Figure 8A

U.S. AND USSR COMPATIBLE DOCKING SYSTEMS

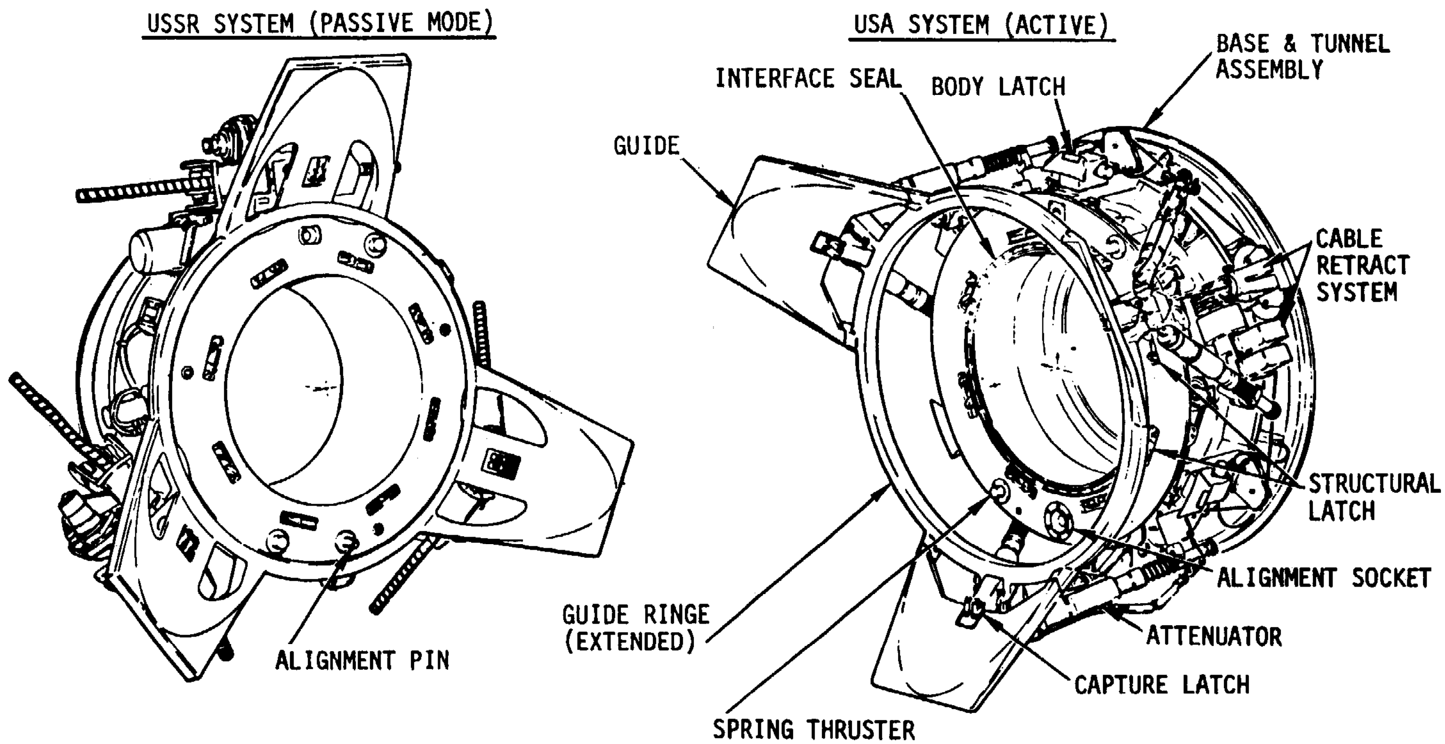


Figure 9

APOLLO SOYUZ TEST PROJECT
MISSION PROFILE

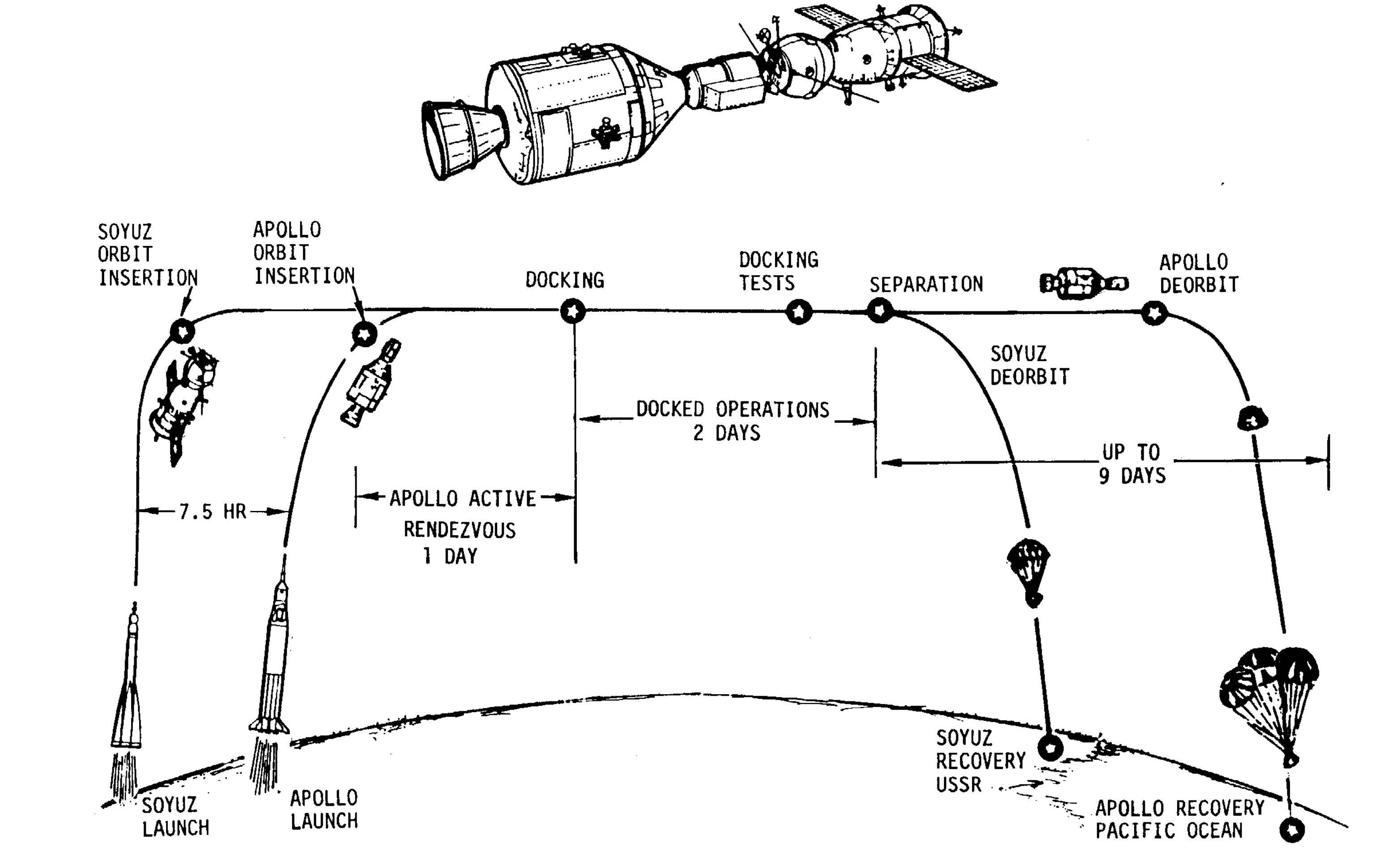
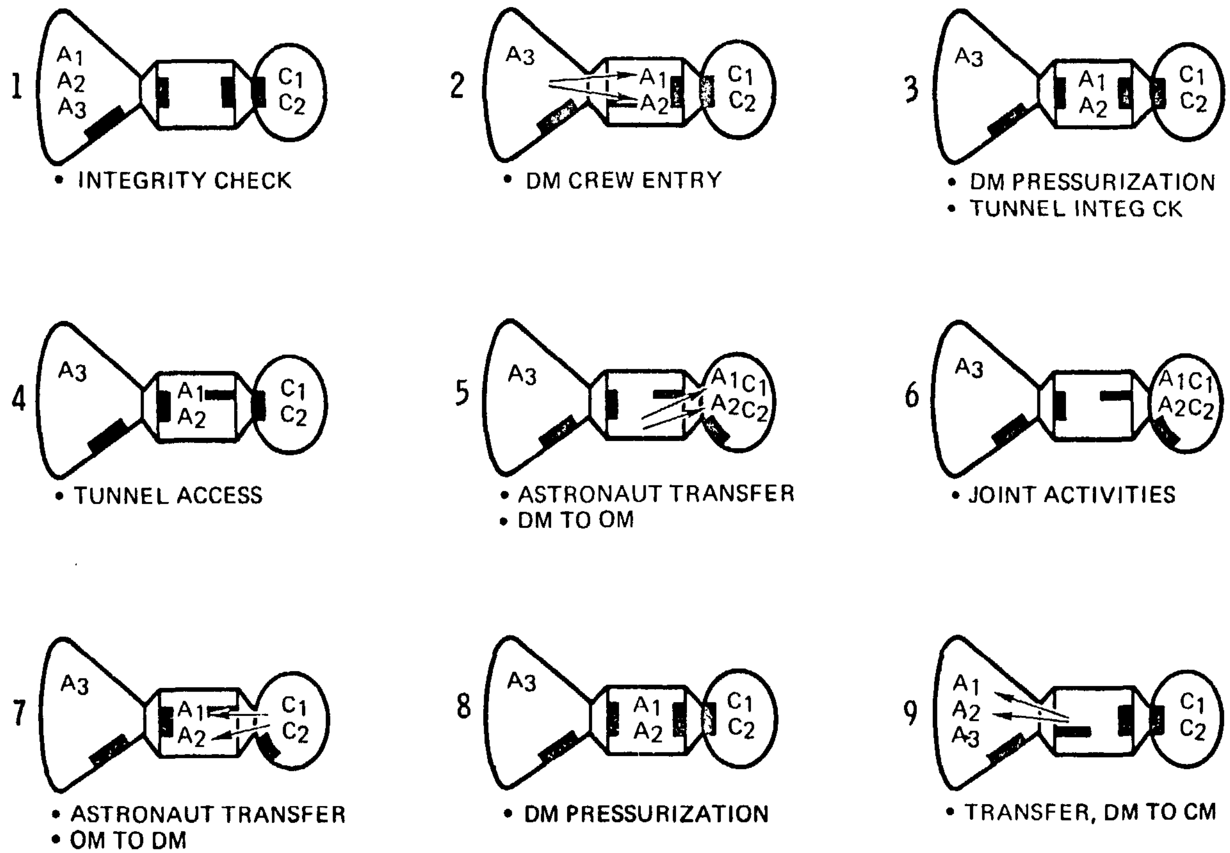


Figure 10

FIRST TRANSFER OPERATIONS



CM - COMMAND MODULE

DM - DOCKING MODULE

OM - ORBITAL MODULE

Figure 11

SPACE SHUTTLE MANAGEMENT RELATIONSHIPS

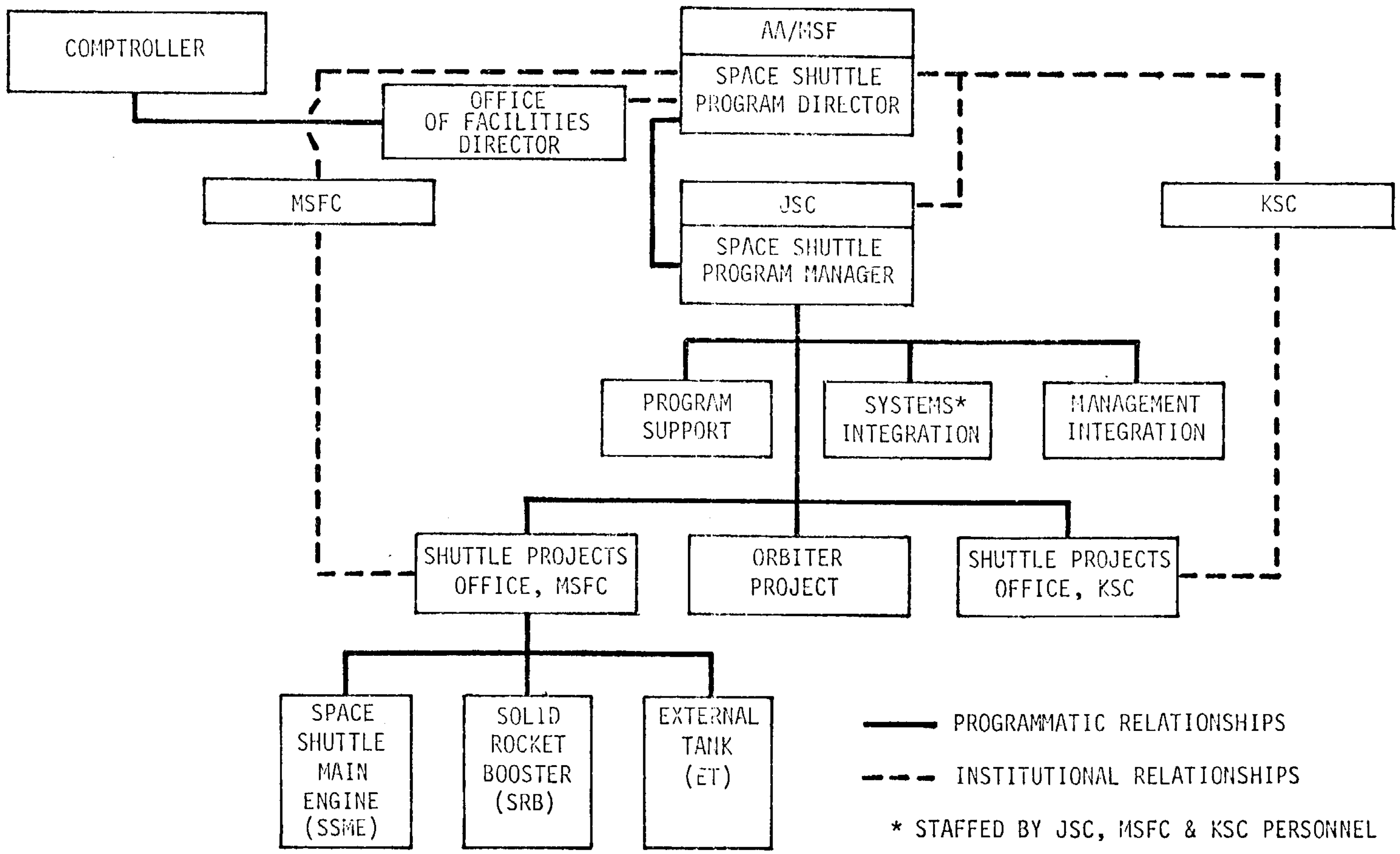


Figure 12

NASA SPACE SHUTTLE PROGRAM ORGANIZATION

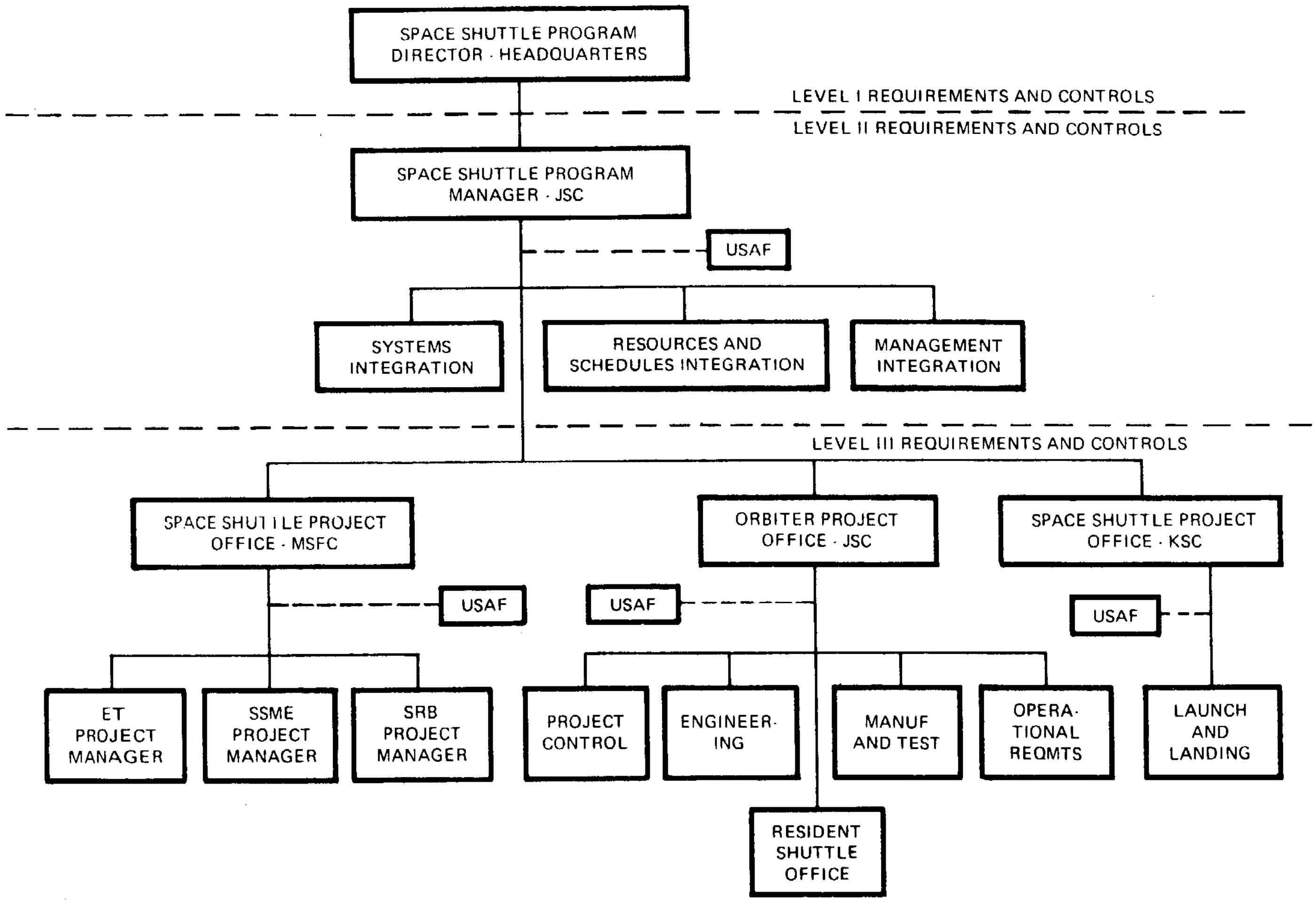


Figure 13

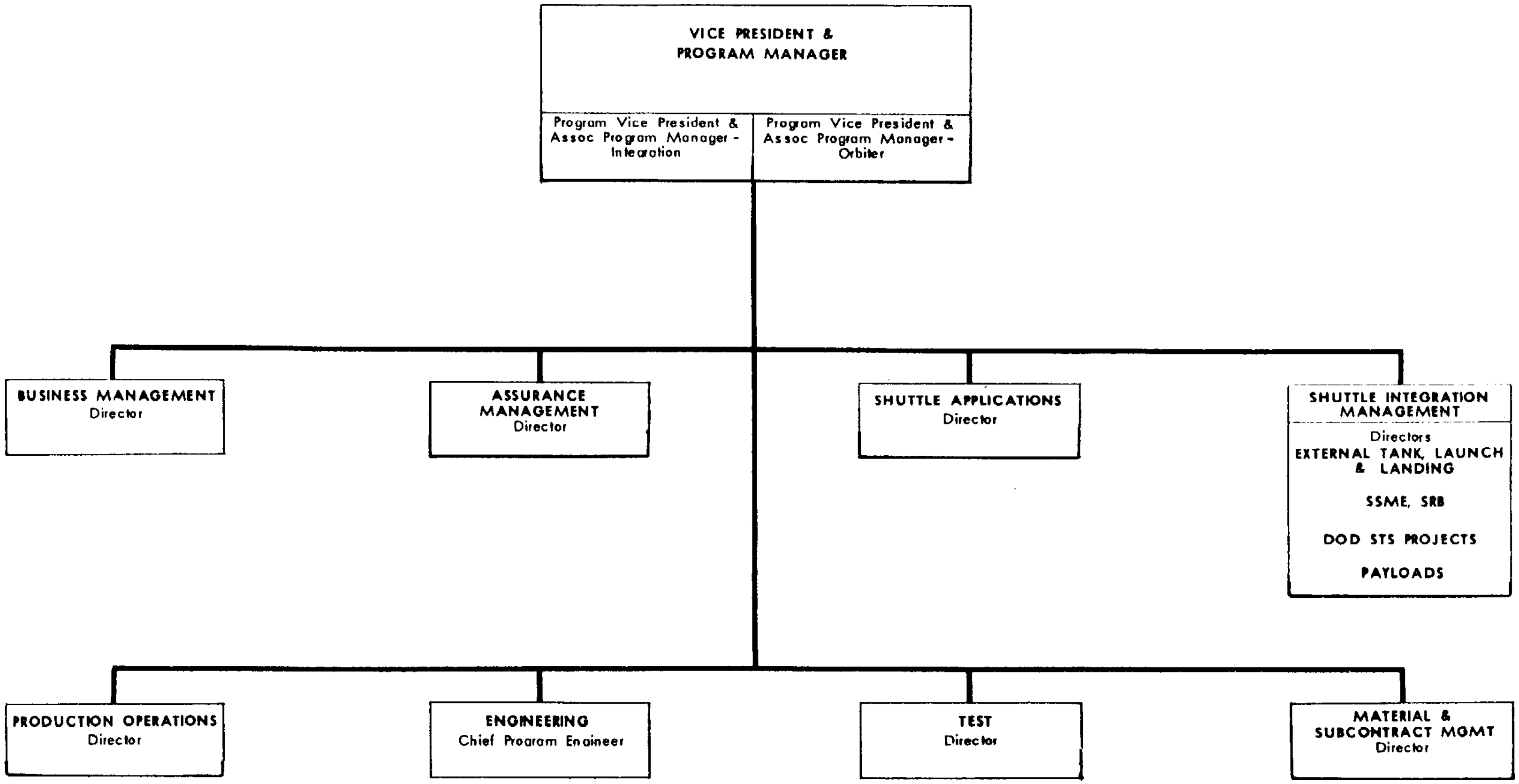

VICE PRESIDENT & PROGRAM MANAGER
Program Vice President & Assoc Program Manager - Integration
Program Vice President & Assoc Program Manager - Orbiter
BUSINESS MANAGEMENT Director
ASSURANCE MANAGEMENT Director
SHUTTLE APPLICATIONS Director
SHUTTLE INTEGRATION MANAGEMENT
Directors
EXTERNAL TANK, LAUNCH & LANDING
SSME, SRB
DOD STS PROJECTS
PAYLOADS
PRODUCTION OPERATIONS Director
ENGINEERING Chief Program Engineer
TEST Director
MATERIAL & SUBCONTRACT MGMT Director

Figure 14

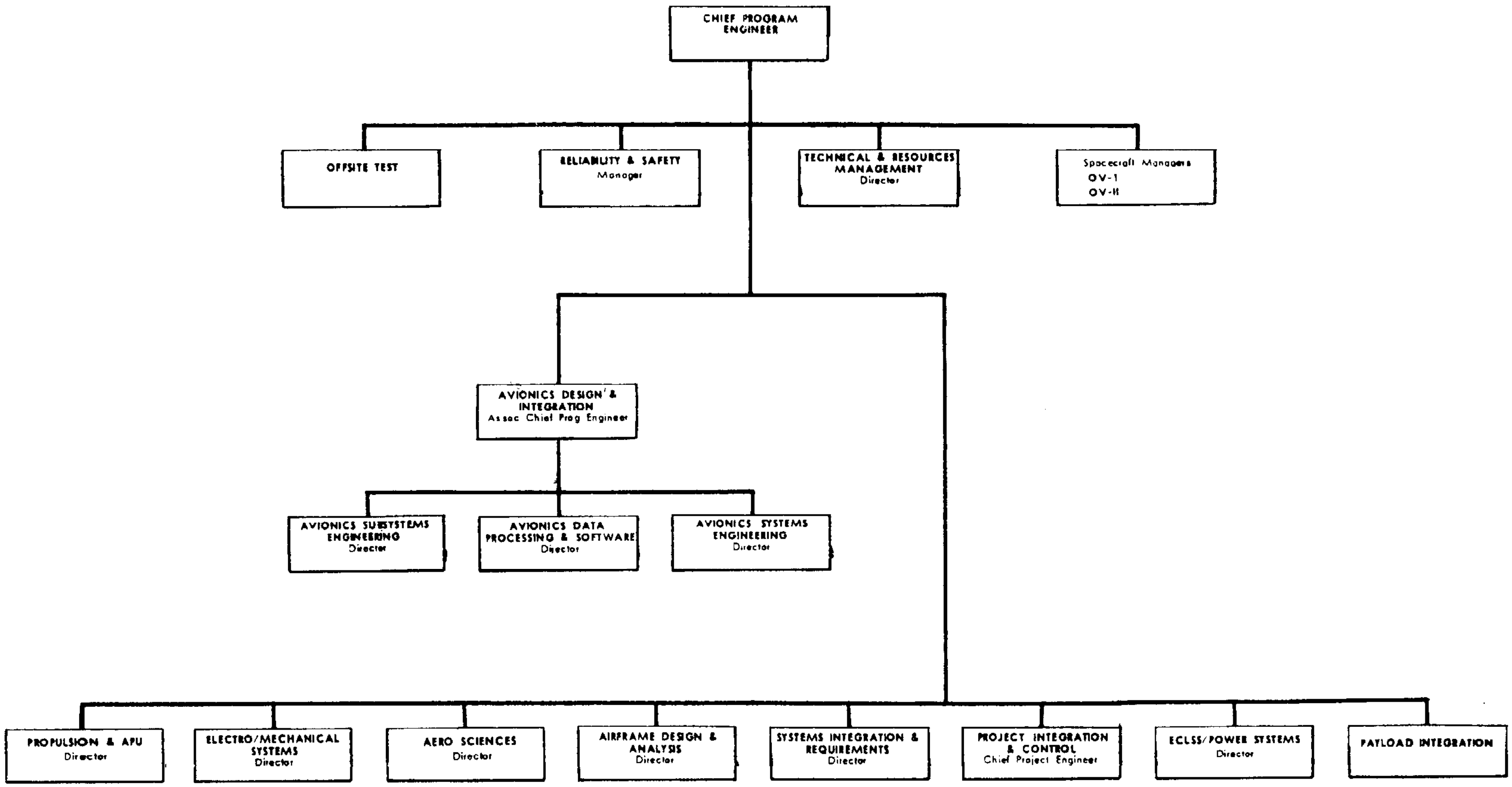

CHIEF PROGRAM ENGINEER
OFFSITE TEST
RELIABILITY & SAFETY
Manager
TECHNICAL & RESOURCES MANAGEMENT
Director
Spacecraft Managers
OV-1
OV-II
AVIONICS DESIGN & INTEGRATION
Assoc Chief Prog Engineer
AVIONICS SUBSYSTEMS ENGINEERING
Director
AVIONICS DATA PROCESSING & SOFTWARE
Director
AVIONICS SYSTEMS ENGINEERING
Director
PROPULSION & APU
Director
ELECTRO/MECHANICAL SYSTEMS
Director
AERO SCIENCES
Director
AIRFRAME DESIGN & ANALYSIS
Director
SYSTEMS INTEGRATION & REQUIREMENTS
Director
PROJECT INTEGRATION & CONTROL
Chief Project Engineer
ECLSS/POWER SYSTEMS
Director
PAYLOAD INTEGRATION

Figure 15

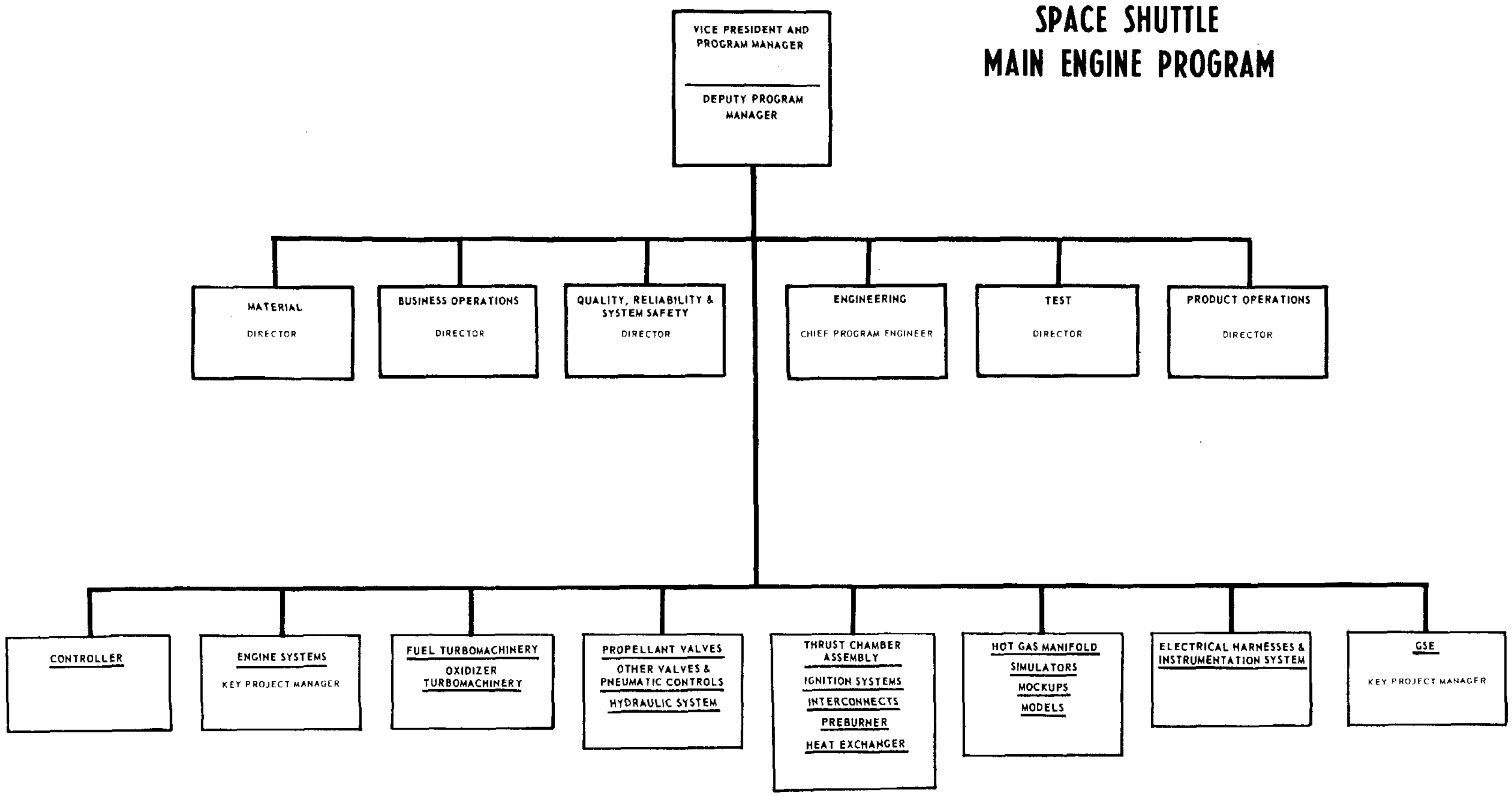
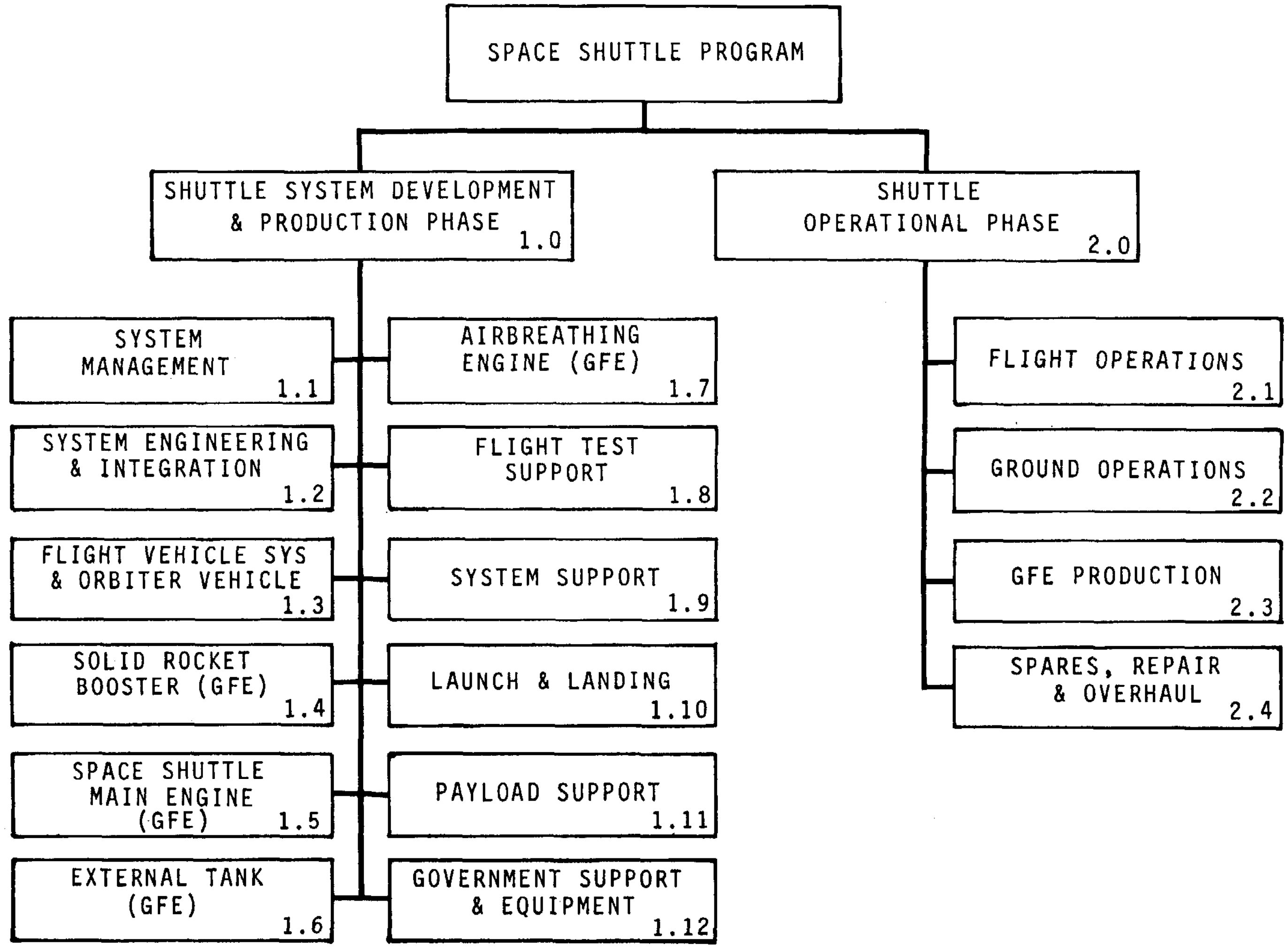

SPACE SHUTTLE
MAIN ENGINE PROGRAM
VICE PRESIDENT AND PROGRAM MANAGER
DEPUTY PROGRAM MANAGER
MATERIAL
DIRECTOR
BUSINESS OPERATIONS
DIRECTOR
QUALITY, RELIABILITY & SYSTEM SAFETY
DIRECTOR
ENGINEERING
CHIEF PROGRAM ENGINEER
TEST
DIRECTOR
PRODUCT OPERATIONS
DIRECTOR
CONTROLLER
ENGINE SYSTEMS
KEY PROJECT MANAGER
FUEL TURBOMACHINERY
OXIDIZER TURBOMACHINERY
PROPELLANT VALVES
OTHER VALVES & PNEUMATIC CONTROLS
HYDRAULIC SYSTEM
THRUST CHAMBER ASSEMBLY
IGNITION SYSTEMS
INTERCONNECTS
PREBURNER
HEAT EXCHANGER
HOT GAS MANIFOLD
SIMULATORS
MOCKUPS
MODELS
ELECTRICAL HARNESSES & INSTRUMENTATION SYSTEM
GSE
KEY PROJECT MANAGER

Figure 16



PROGRAM WORK BREAKDOWN STRUCTURE

Figure 17

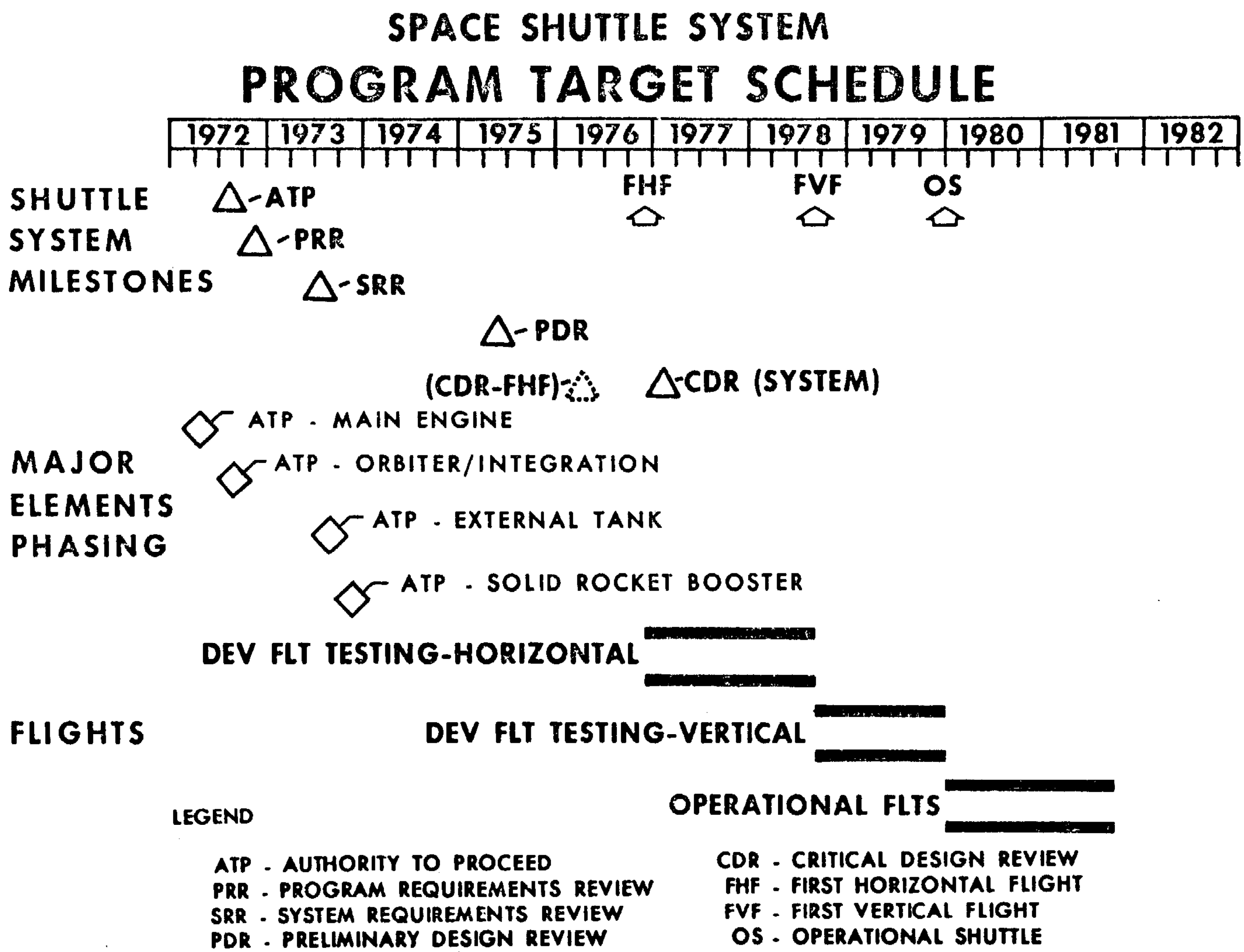

SPACE SHUTTLE SYSTEM
PROGRAM TARGET SCHEDULE
1972
1973
1974
1975
1976
1977
1978
1979
1980
1981
1982
SHUTTLE SYSTEM MILESTONES
ATP
PRR
SRR
PDR
(CDR-FHF)
CDR (SYSTEM)
FHF
FVF
OS
MAJOR ELEMENTS PHASING
ATP - MAIN ENGINE
ATP - ORBITER/INTEGRATION
ATP - EXTERNAL TANK
ATP - SOLID ROCKET BOOSTER
FLIGHTS
DEV FLT TESTING-HORIZONTAL
DEV FLT TESTING-VERTICAL
OPERATIONAL FLTS
LEGEND
ATP - AUTHORITY TO PROCEED
PRR - PROGRAM REQUIREMENTS REVIEW
SRR - SYSTEM REQUIREMENTS REVIEW
PDR - PRELIMINARY DESIGN REVIEW
CDR - CRITICAL DESIGN REVIEW
FHF - FIRST HORIZONTAL FLIGHT
FVF - FIRST VERTICAL FLIGHT
OS - OPERATIONAL SHUTTLE

Figure 18

SPACE SHUTTLE PROGRAM SCHEDULE

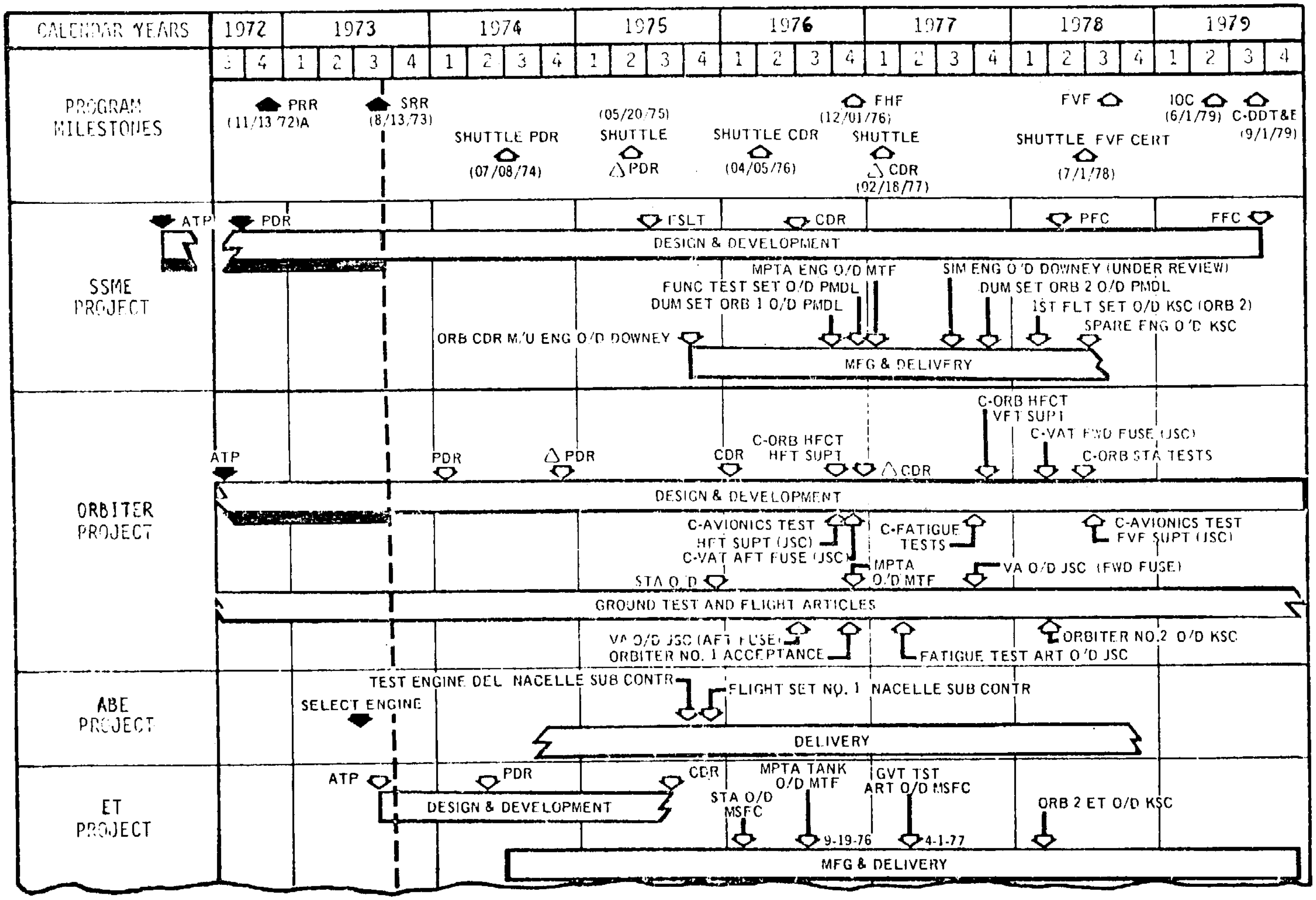


Figure 19A

SPACE SHUTTLE PROGRAM SCHEDULE (CONT)

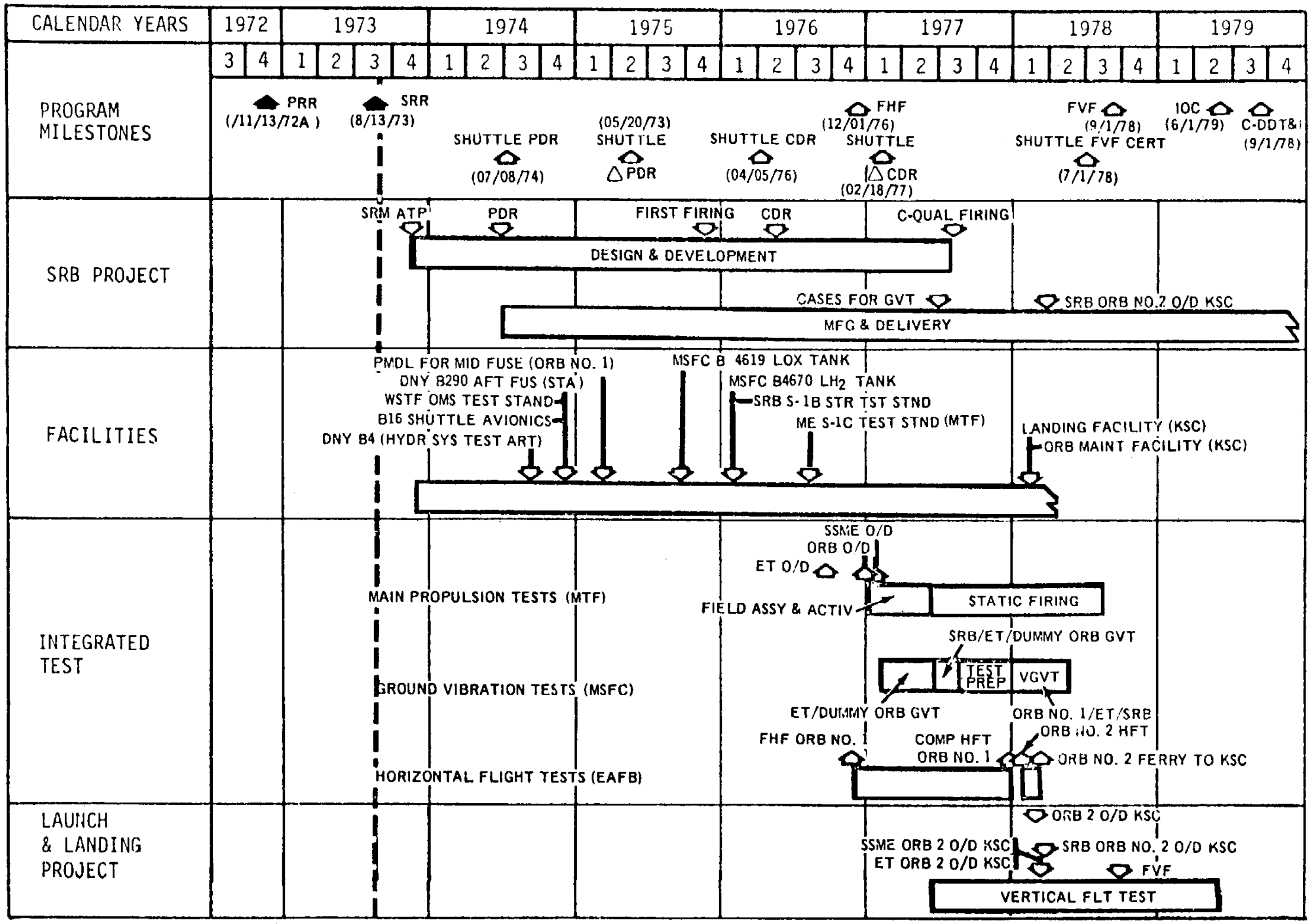


Figure 19B

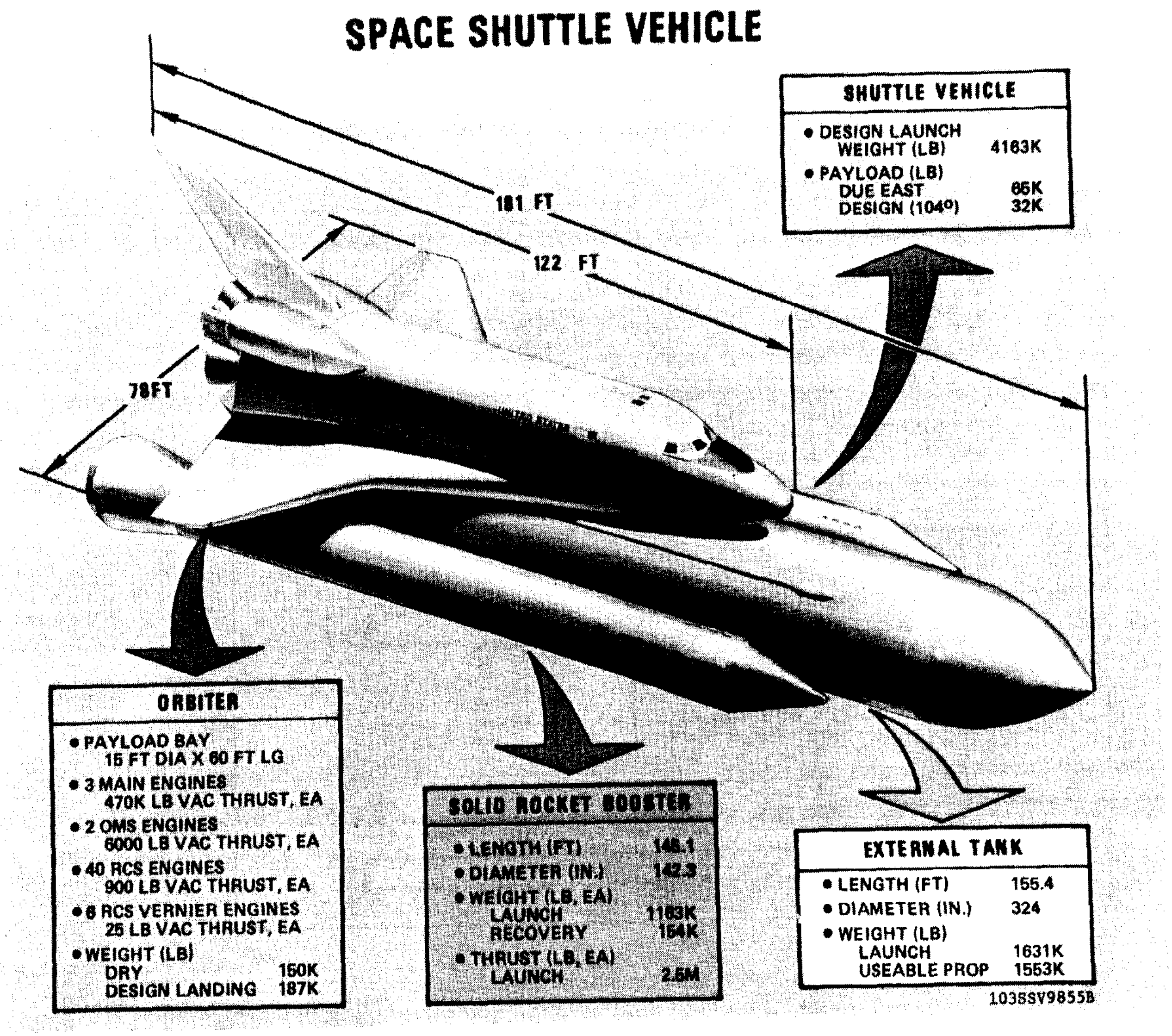
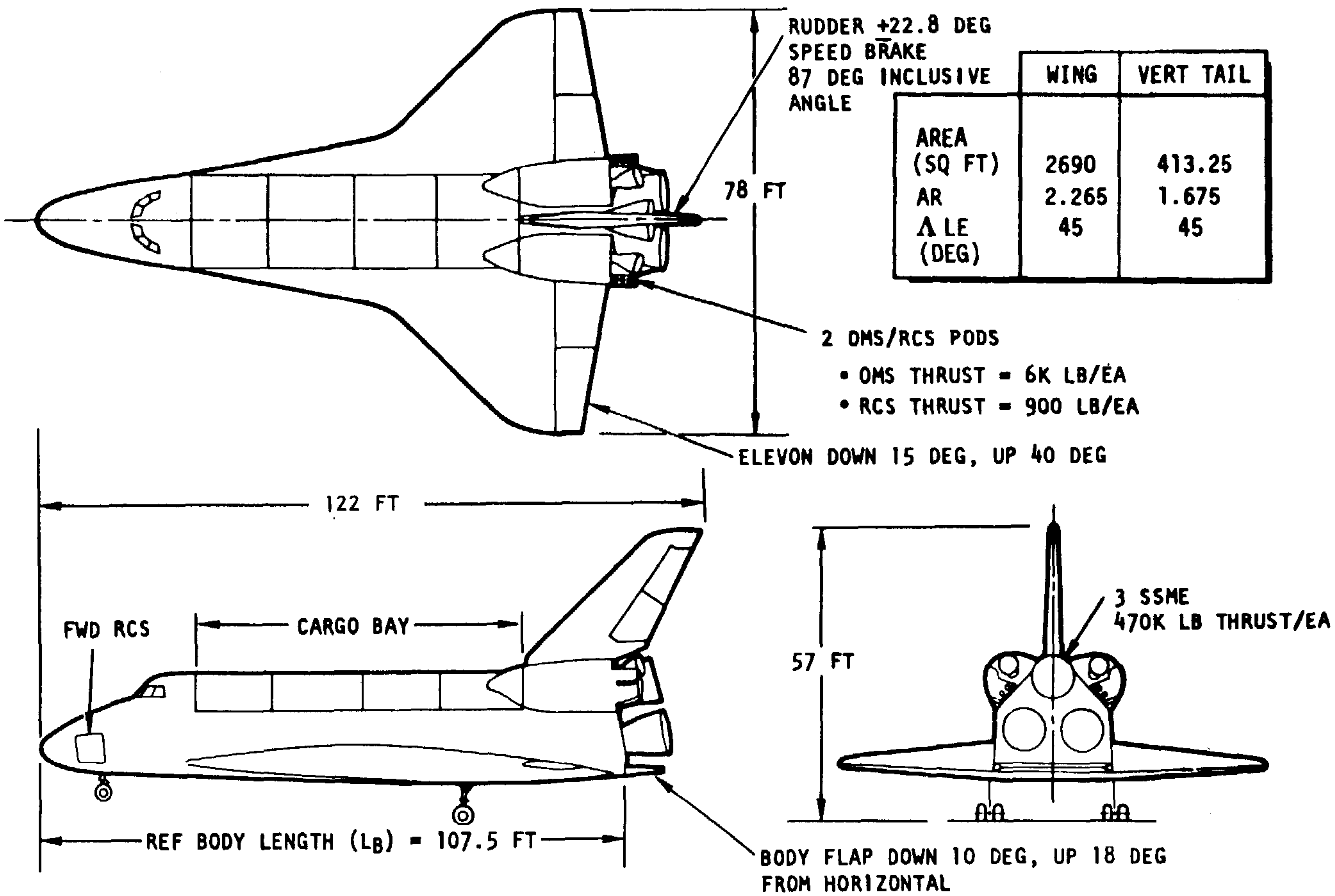

SPACE SHUTTLE VEHICLE
181 FT
122 FT
78FT
SHUTTLE VEHICLE
• DESIGN LAUNCH WEIGHT (LB) 4163K
• PAYLOAD (LB)
DUE EAST 65K
DESIGN (104°) 32K
ORBITER
• PAYLOAD BAY
15 FT DIA X 60 FT LG
• 3 MAIN ENGINES
470K LB VAC THRUST, EA
• 2 OMS ENGINES
6000 LB VAC THRUST, EA
• 40 RCS ENGINES
900 LB VAC THRUST, EA
• 6 RCS VERNIER ENGINES
25 LB VAC THRUST, EA
• WEIGHT (LB)
DRY 150K
DESIGN LANDING 187K
SOLID ROCKET BOOSTER
• LENGTH (FT) 145.1
• DIAMETER (IN.) 142.3
• WEIGHT (LB, EA)
LAUNCH 1163K
RECOVERY 164K
• THRUST (LB, EA)
LAUNCH 2.6M
EXTERNAL TANK
• LENGTH (FT) 155.4
• DIAMETER (IN.) 324
• WEIGHT (LB)
LAUNCH 1631K
USEABLE PROP 1553K
103SSV9855B

Figure 20

ORBITER VEHICLE



	WING	VERT TAIL
AREA (SQ FT)	2690	413.25
AR	2.265	1.675
Λ LE (DEG)	45	45

Figure 21

FORWARD FUSELAGE WINDOW SUBSYSTEM

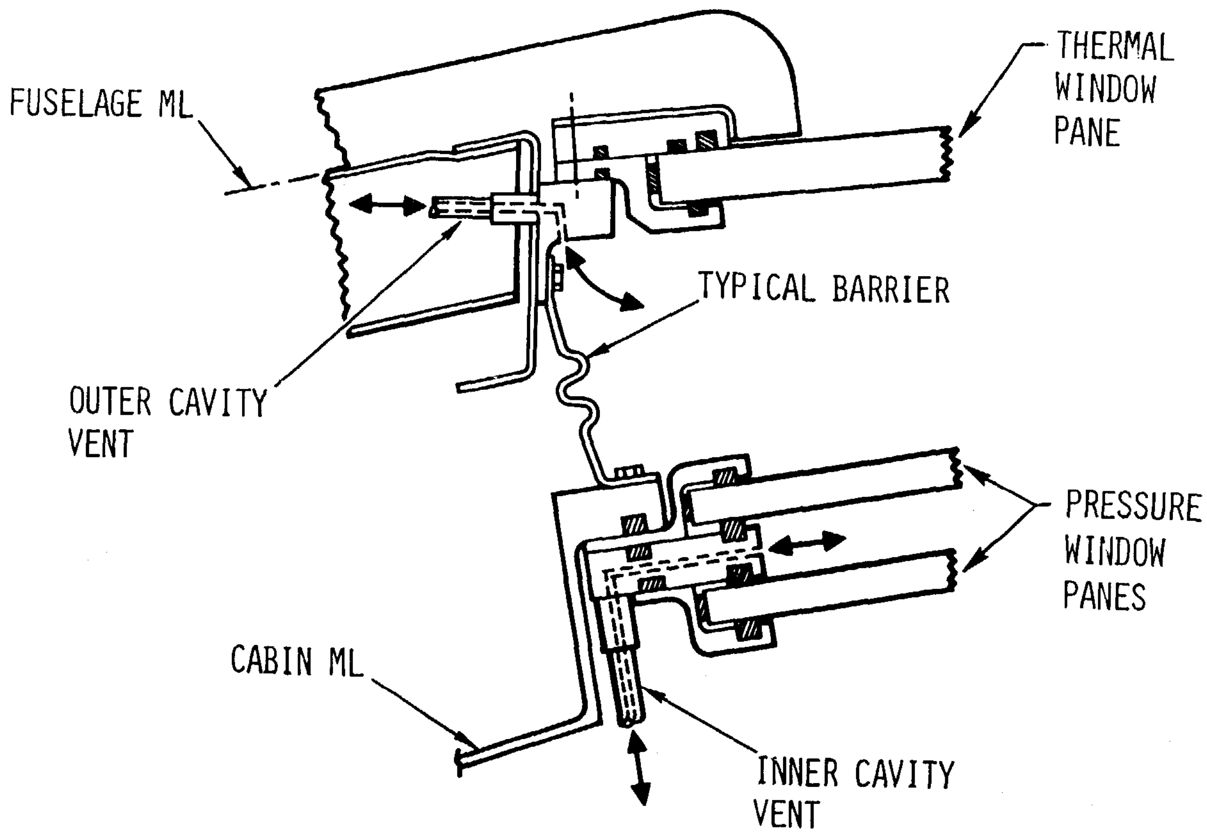


Figure 22

TYPICAL ENTRY TRAJECTORY

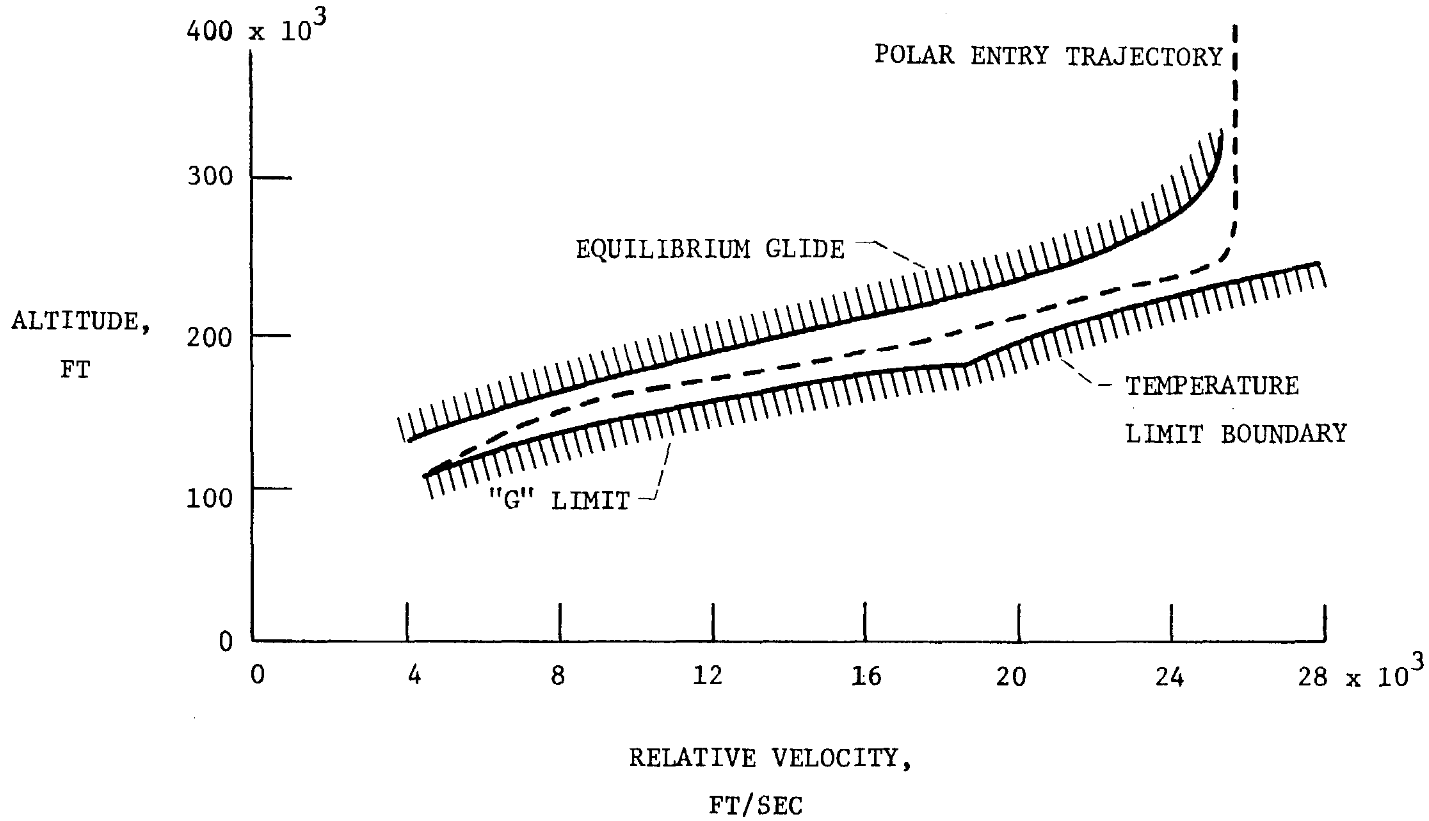


Figure 23

ENTRY HEATING RATE HISTORY

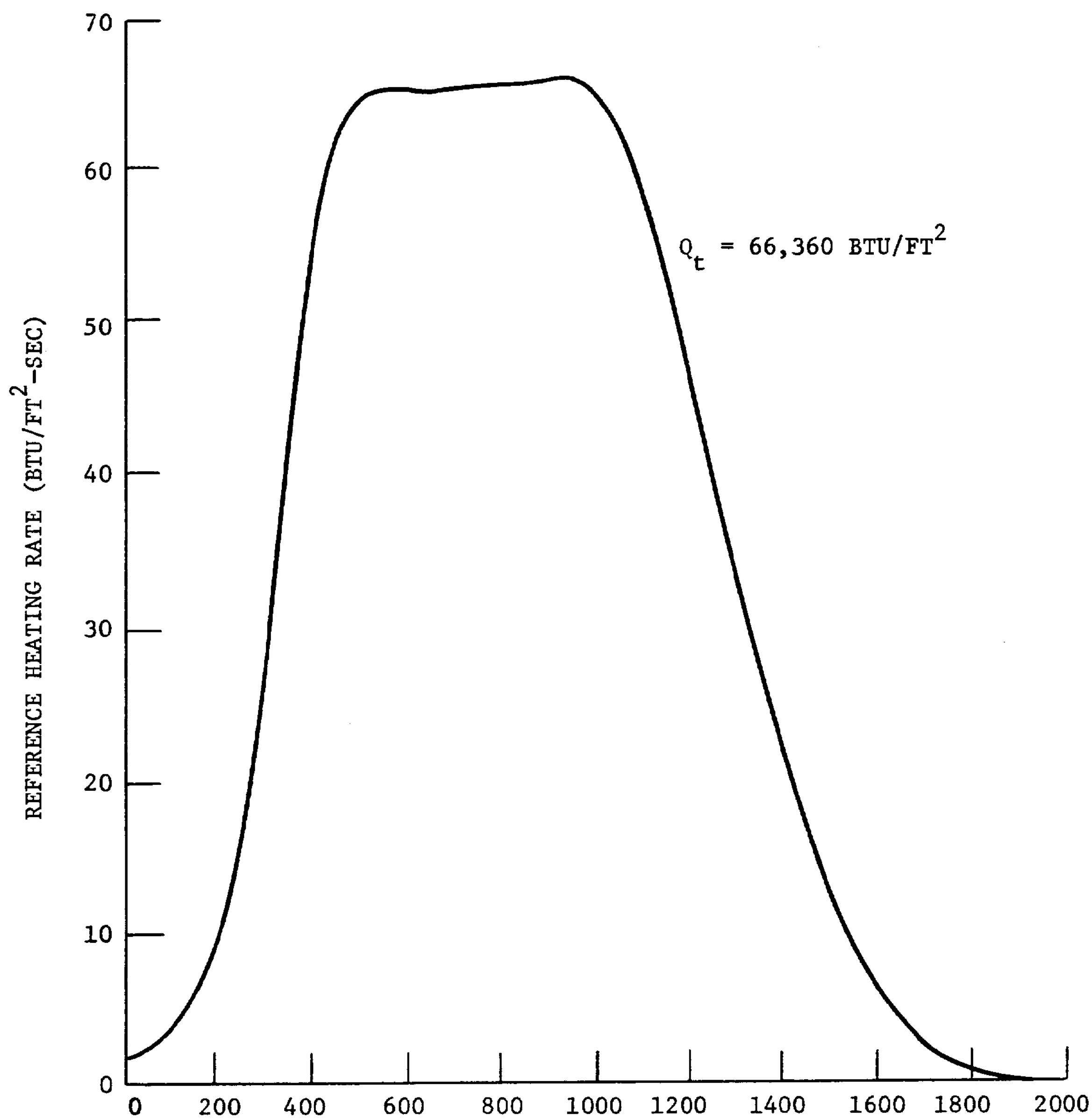


Figure 24

ORBITER ISOTHERMS
&
TPS THICKNESS REQMT

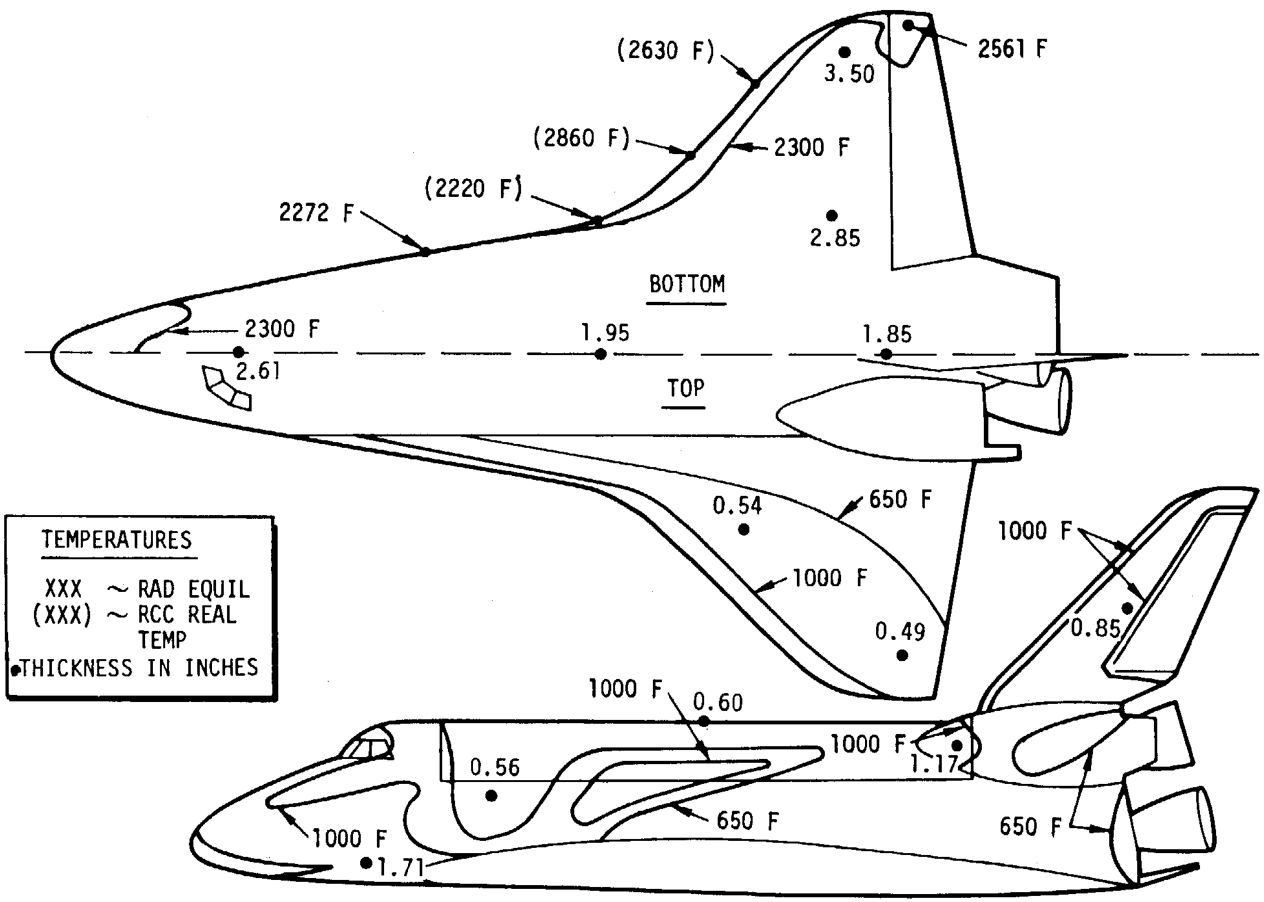


Figure 25

ORBITER THERMAL PROTECTION SYSTEM

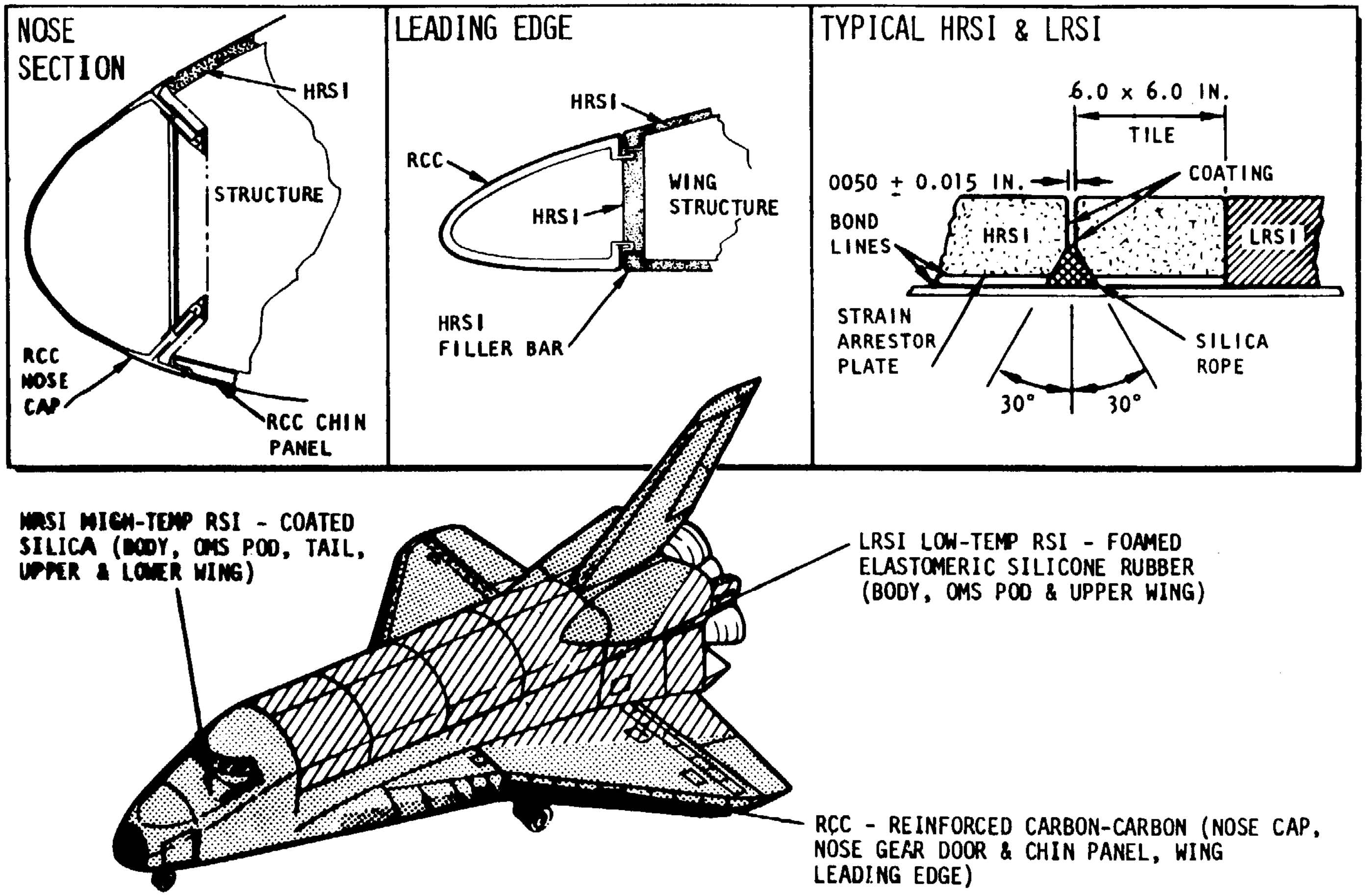


Figure 26

RCS AND OMS LOCATION ON ORBITER

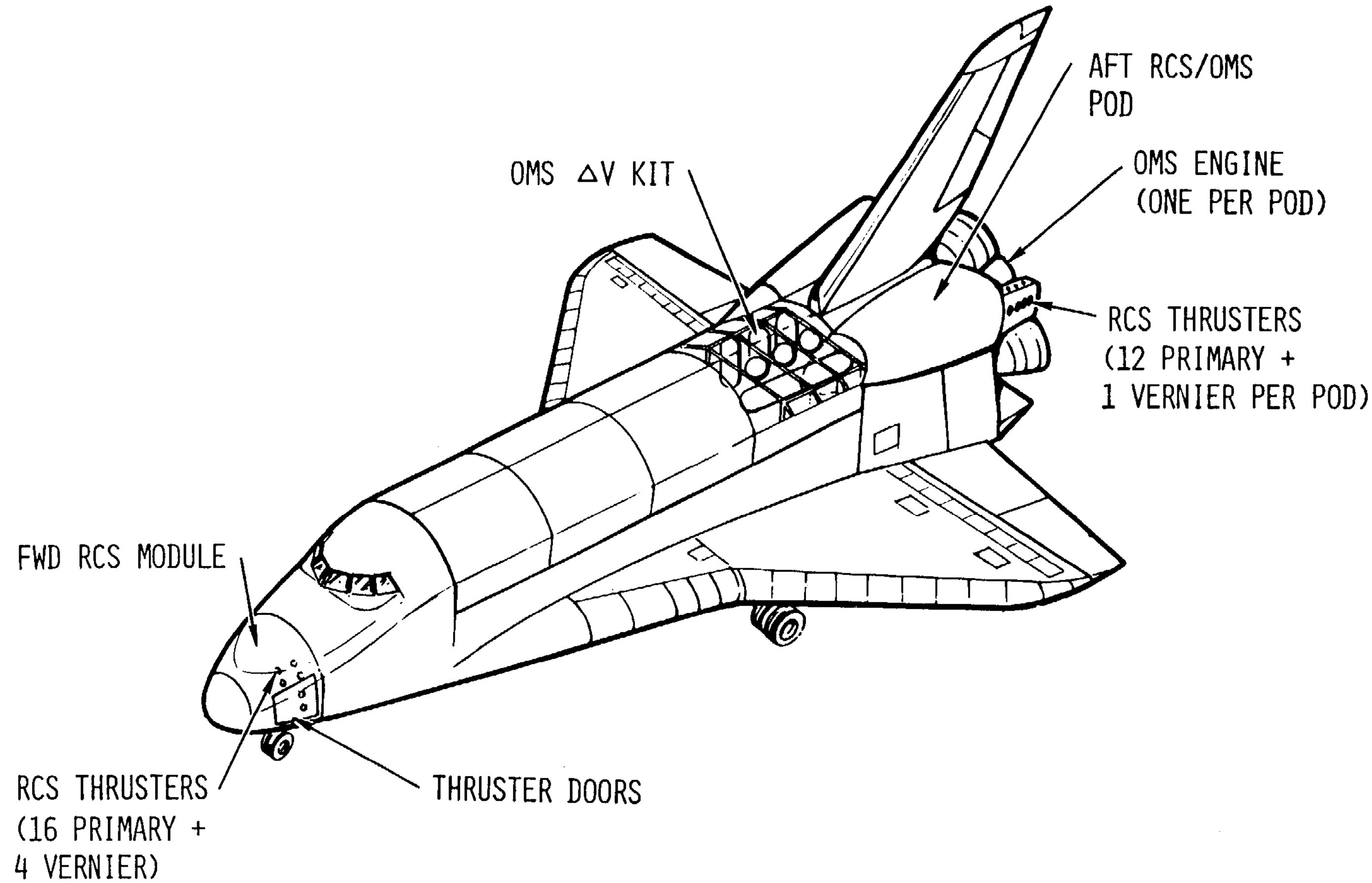


Figure 27

ORBITER REACTION CONTROL SUBSYSTEM

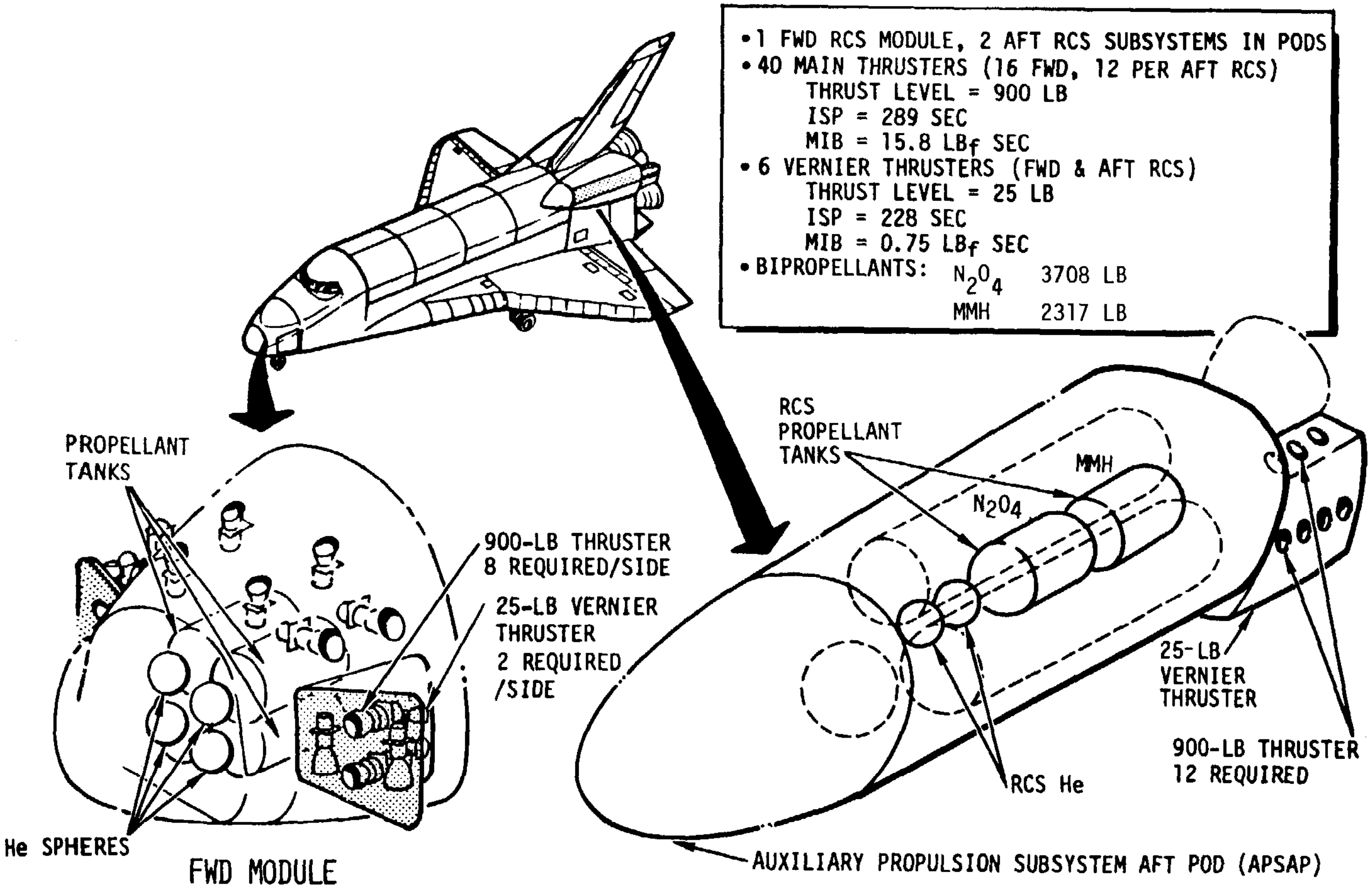


Figure 28

OMS/RCS SCHEMATIC

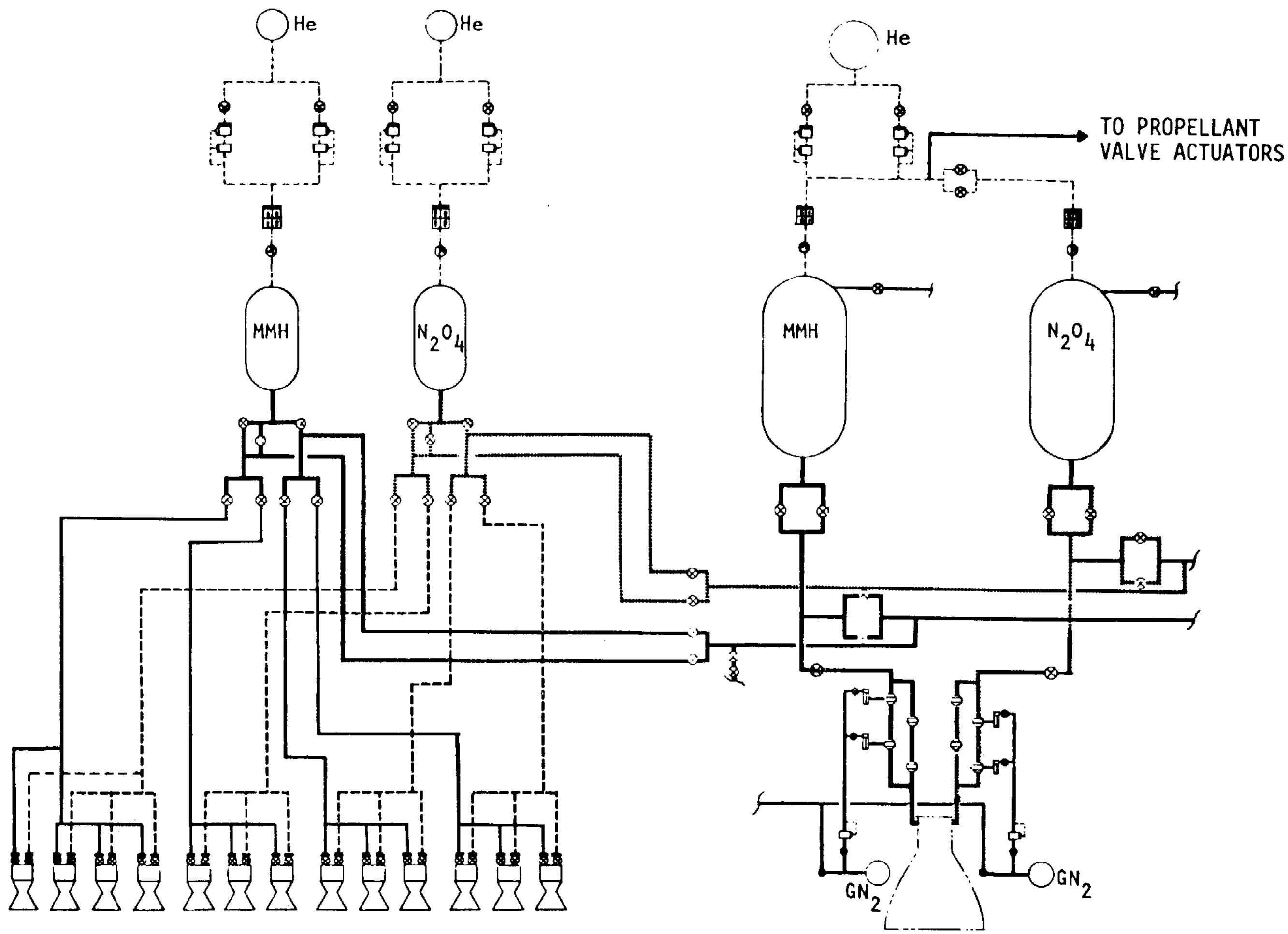


Figure 29

OMS POD PRESSURE BUDGET

PRESSURIZATION

4300		HELIUM BOTTLE
220 (235)		REGULATOR OUTLET-PRI (SEC)
265 ± 8		BURST DISK/RELIEF VALVE

PROPELLANT FEED

OXIDIZER	FUEL	
215	215	PROPELLANT TANK (ULLAGE)
214	214	TANK OUTLET
9	5.8	ΔP_{FEED}

ENGINE

OXIDIZER	FUEL	
205	208.2	ENGINE INLET
125		CHAMBER

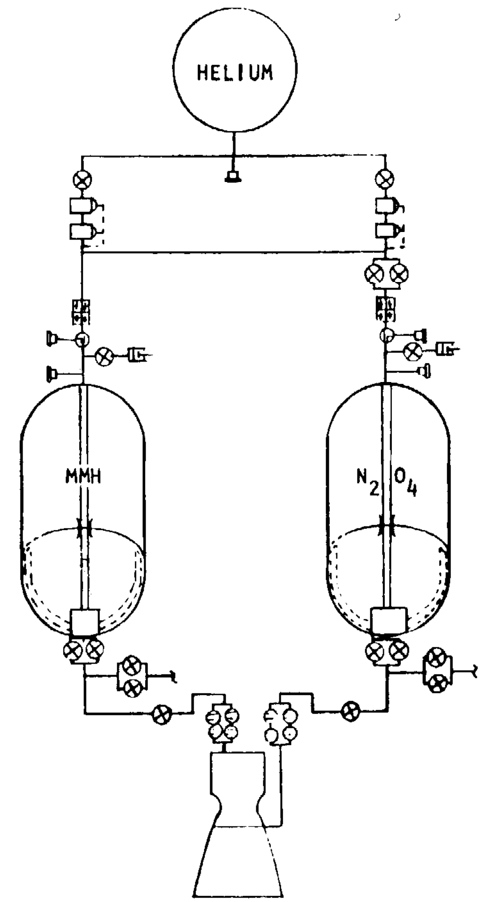


Figure 30

RCS PRESSURE BUDGET

PRESSURIZATION

4000		HELIUM BOTTLE
295 (300)		REGULATOR LOCKUP—PRI (SEC)
425 ± 10		BURST DISC/RELIEF VALVE

PROPELLANT FEED

OXIDIZER	FUEL	
280	280	PROPELLANT TANK (ULLAGE)
279	279	TANK OUTLET
13	13	ΔP_{FEED}

ENGINE

OXIDIZER	FUEL	
266	266	ENGINE INLET
158		CHAMBER

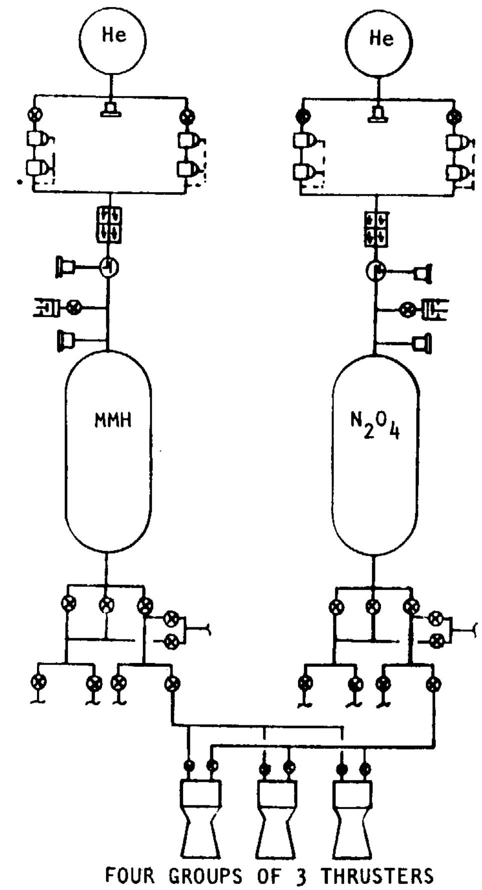


Figure 31

ORBITER CABIN ARRANGEMENT

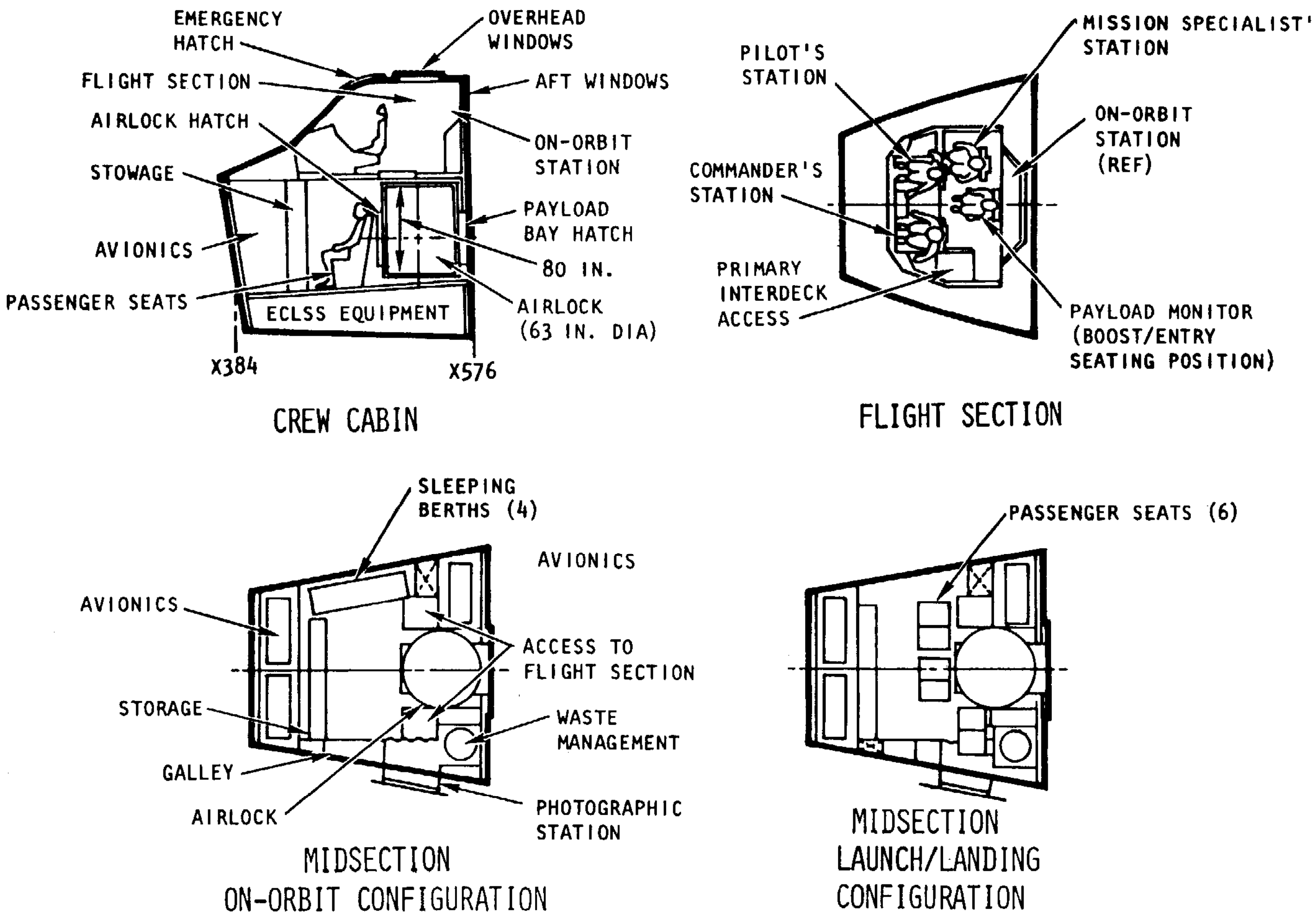


Figure 32

ORBITER MECHANICAL SUBSYSTEMS

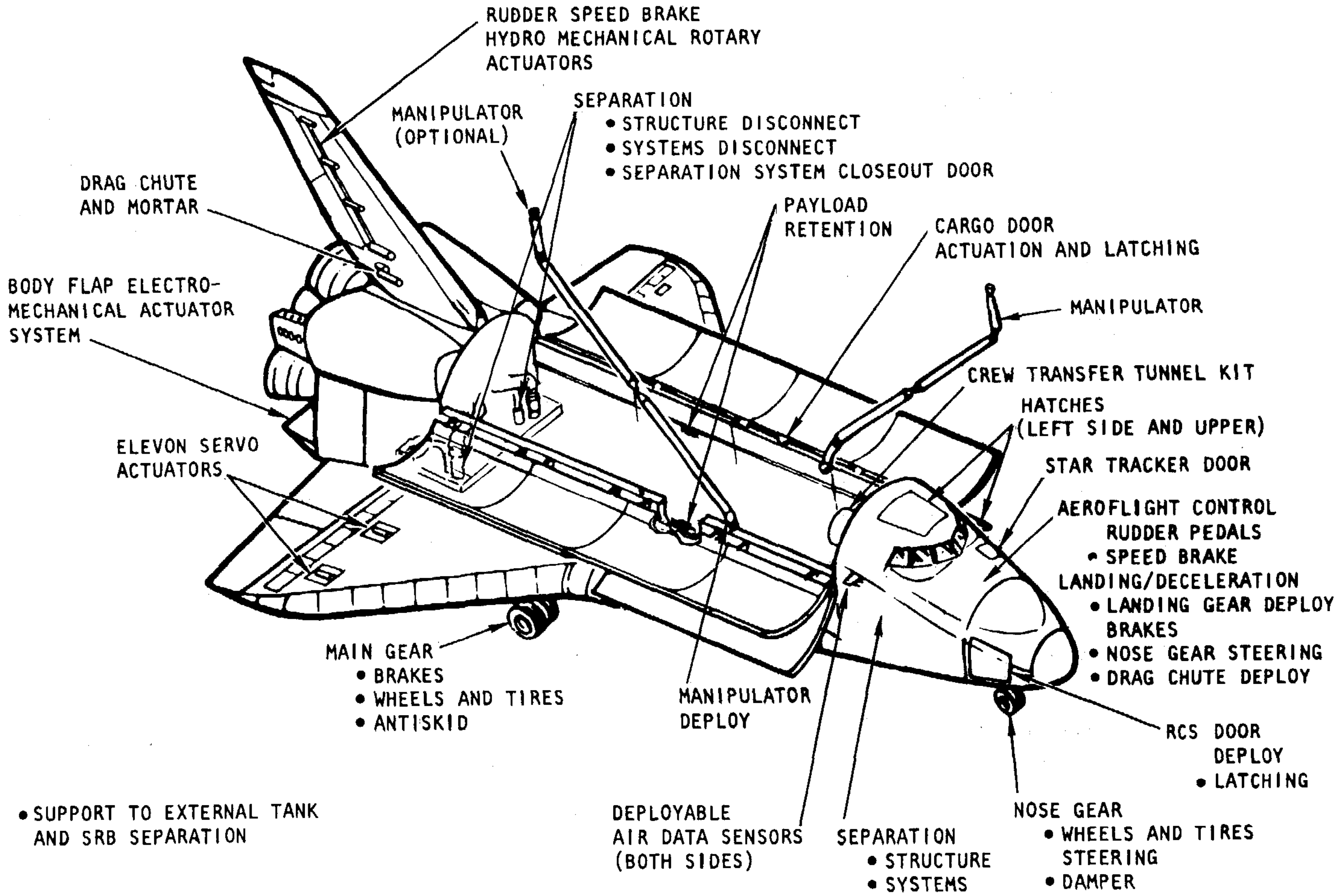


Figure 33

TYPICAL DEWAR DESIGN

TYPE I (O_2)

SPHERICAL
38.6 OD
34.6 ID
INCONEL 718 PV
2219 AL SHELL

TYPE II (H_2)

SPHERICAL
46 OD
43.2 ID
2219 AL PV
2219 AL SHELL

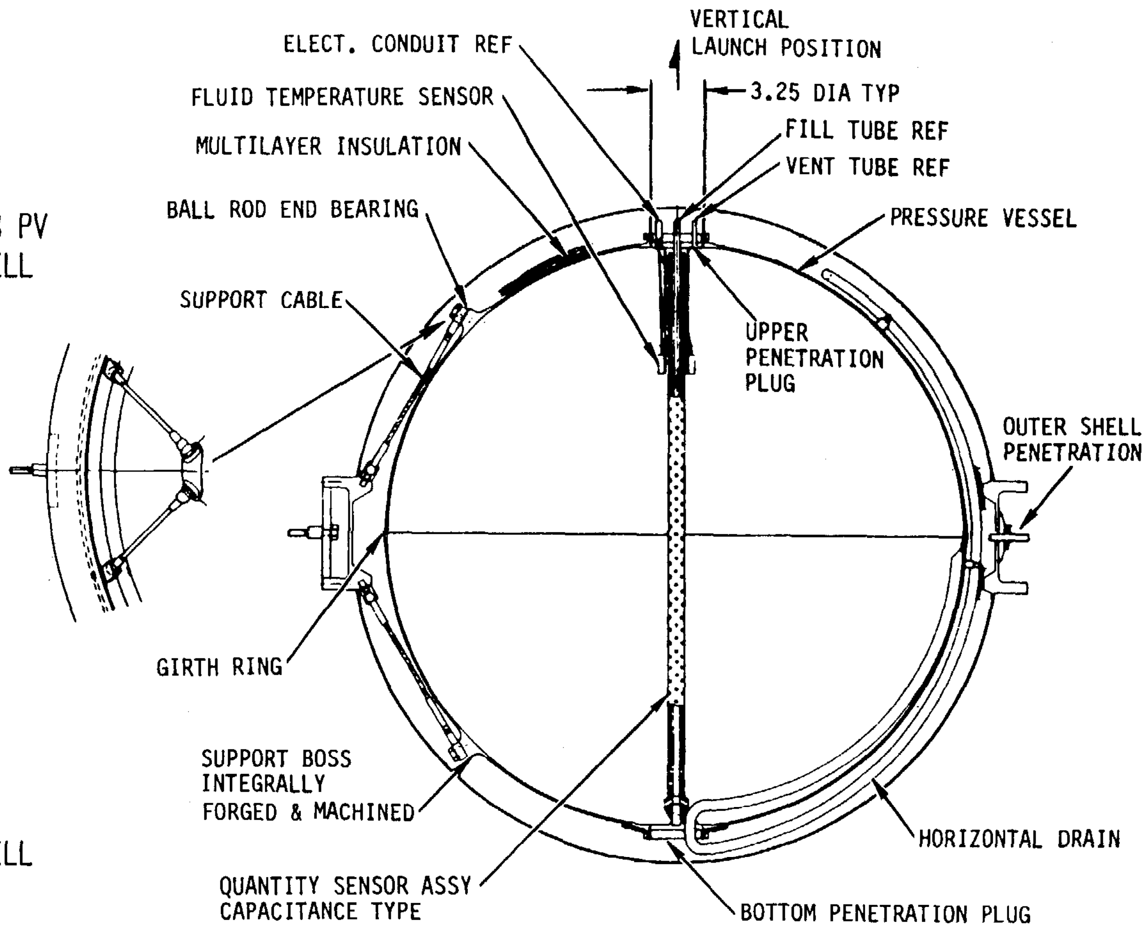


Figure 34

ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEM

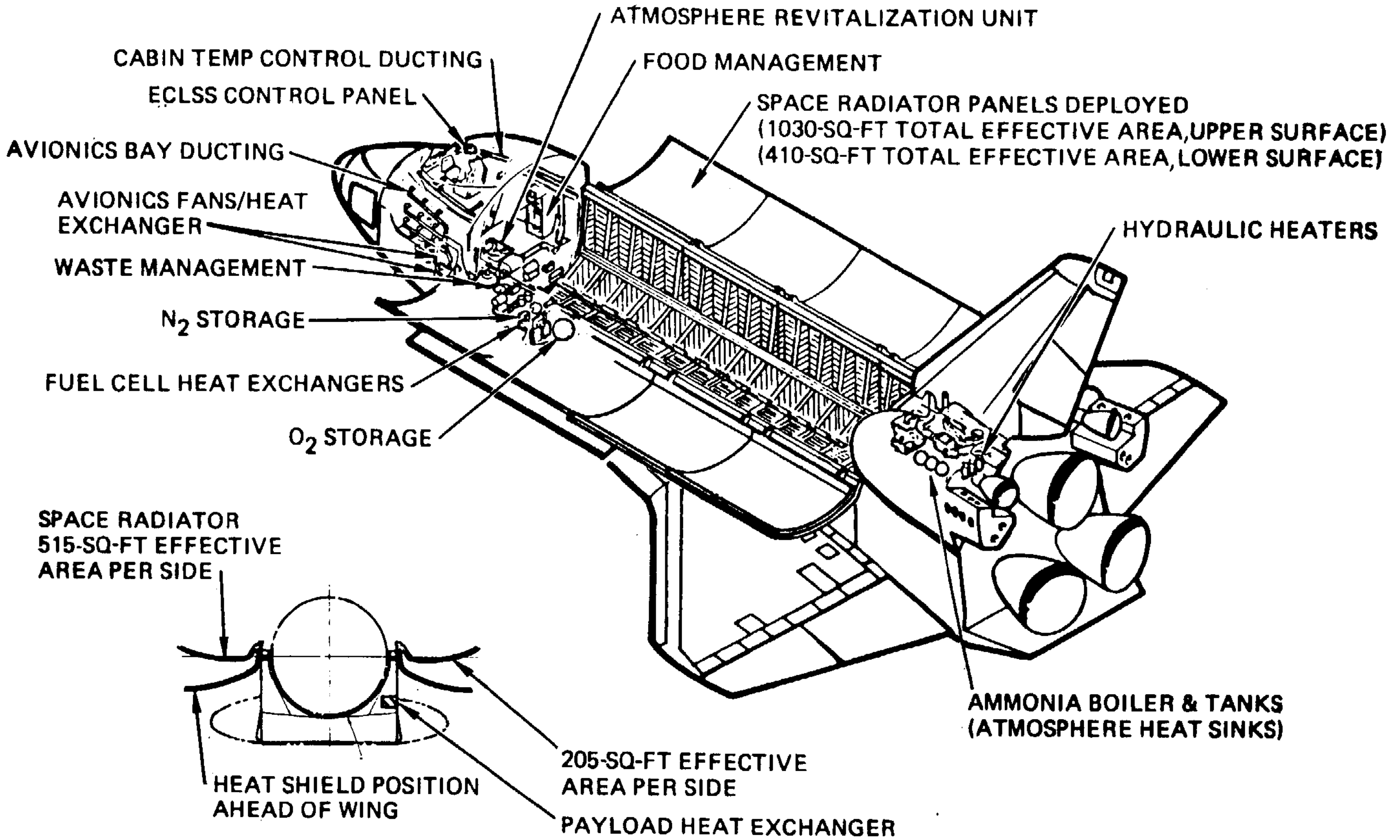


Figure 35

ENVIRONMENTAL CONTROL AND LIFE SUPPORT

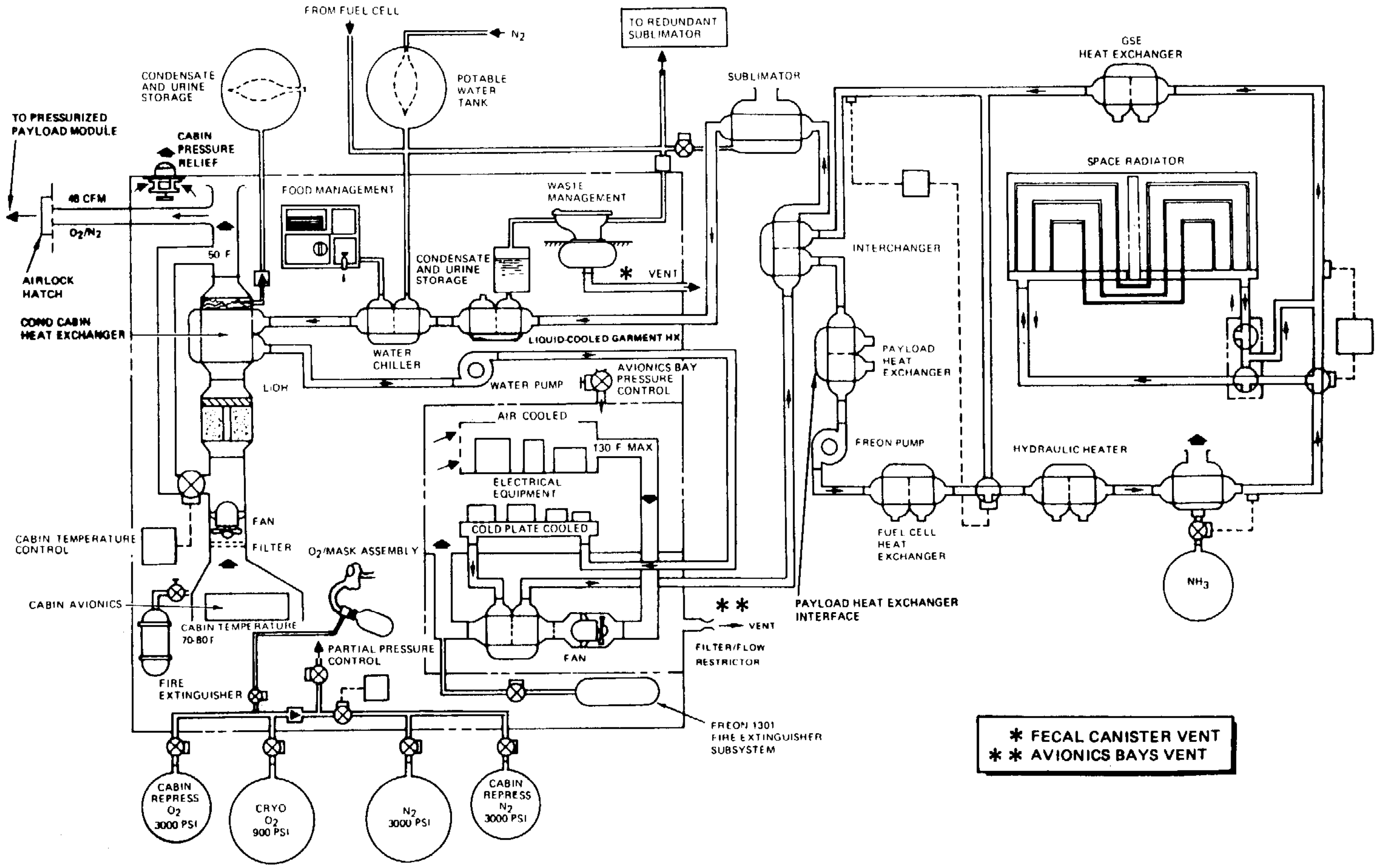


Figure 36

ORBITER THERMAL CONTROL SUBSYSTEM

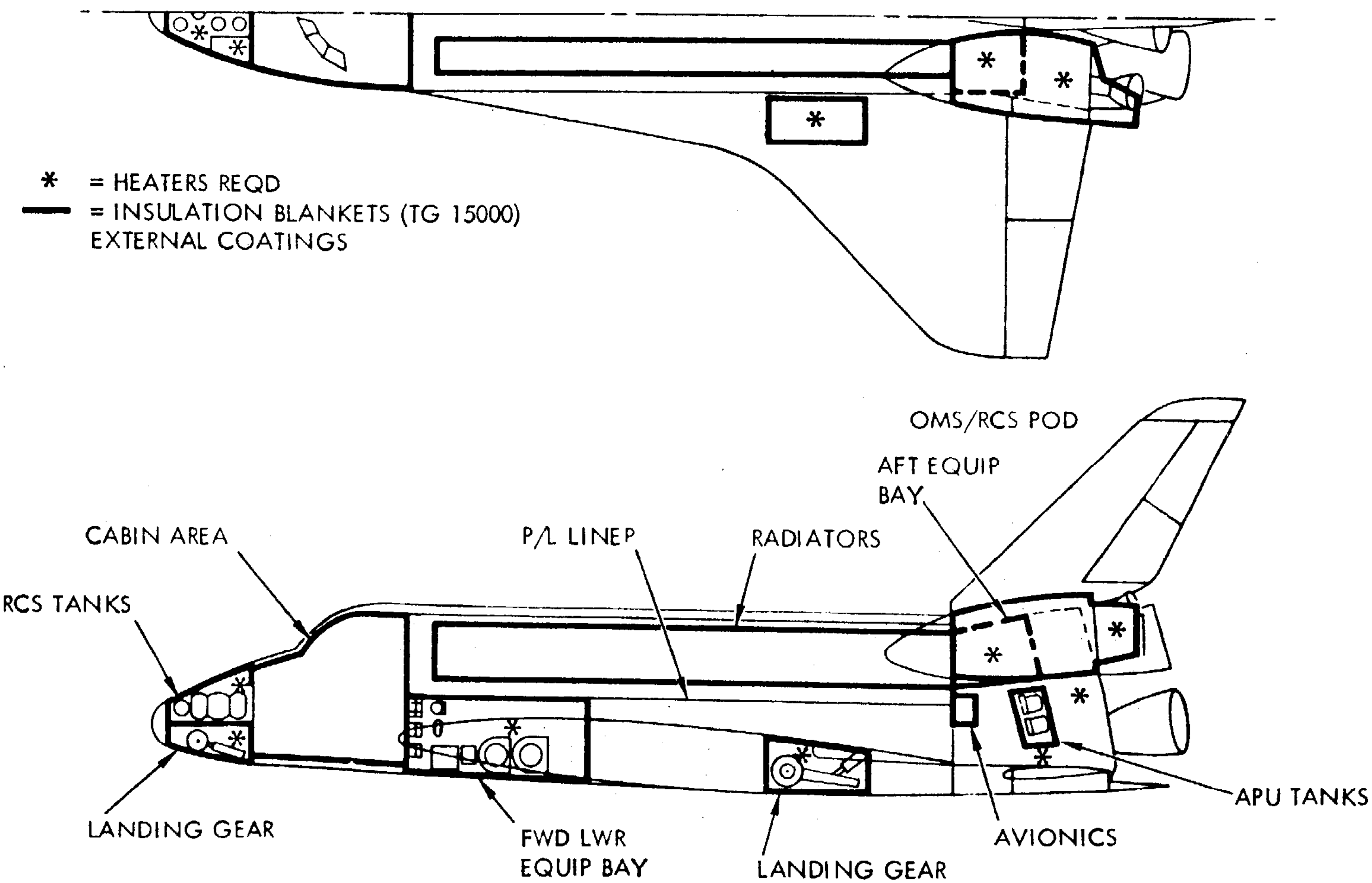


Figure 37

ORBITER AVIONICS SUBSYSTEM

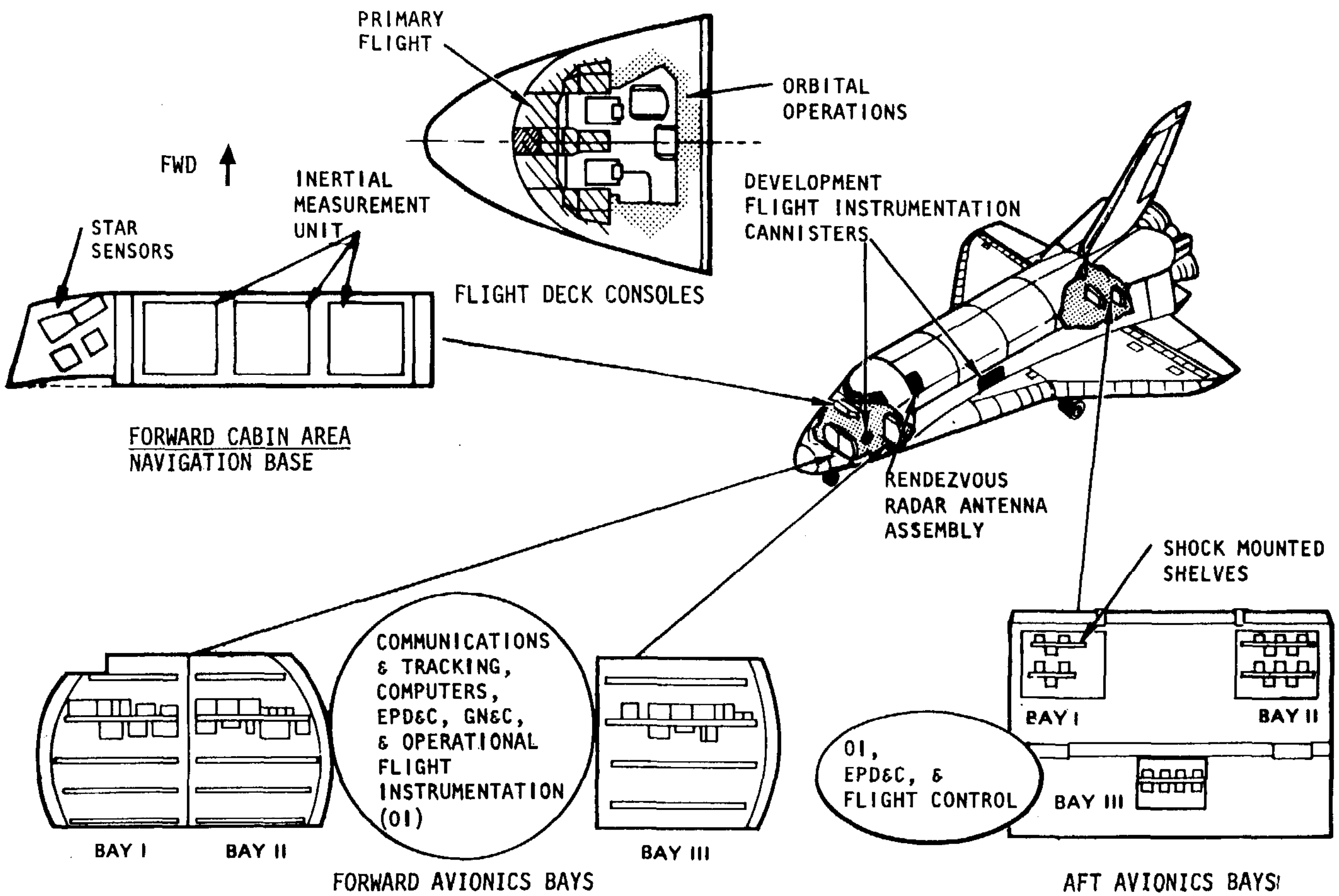


Figure 38

ORBITER COMMUNICATIONS & TRACKING

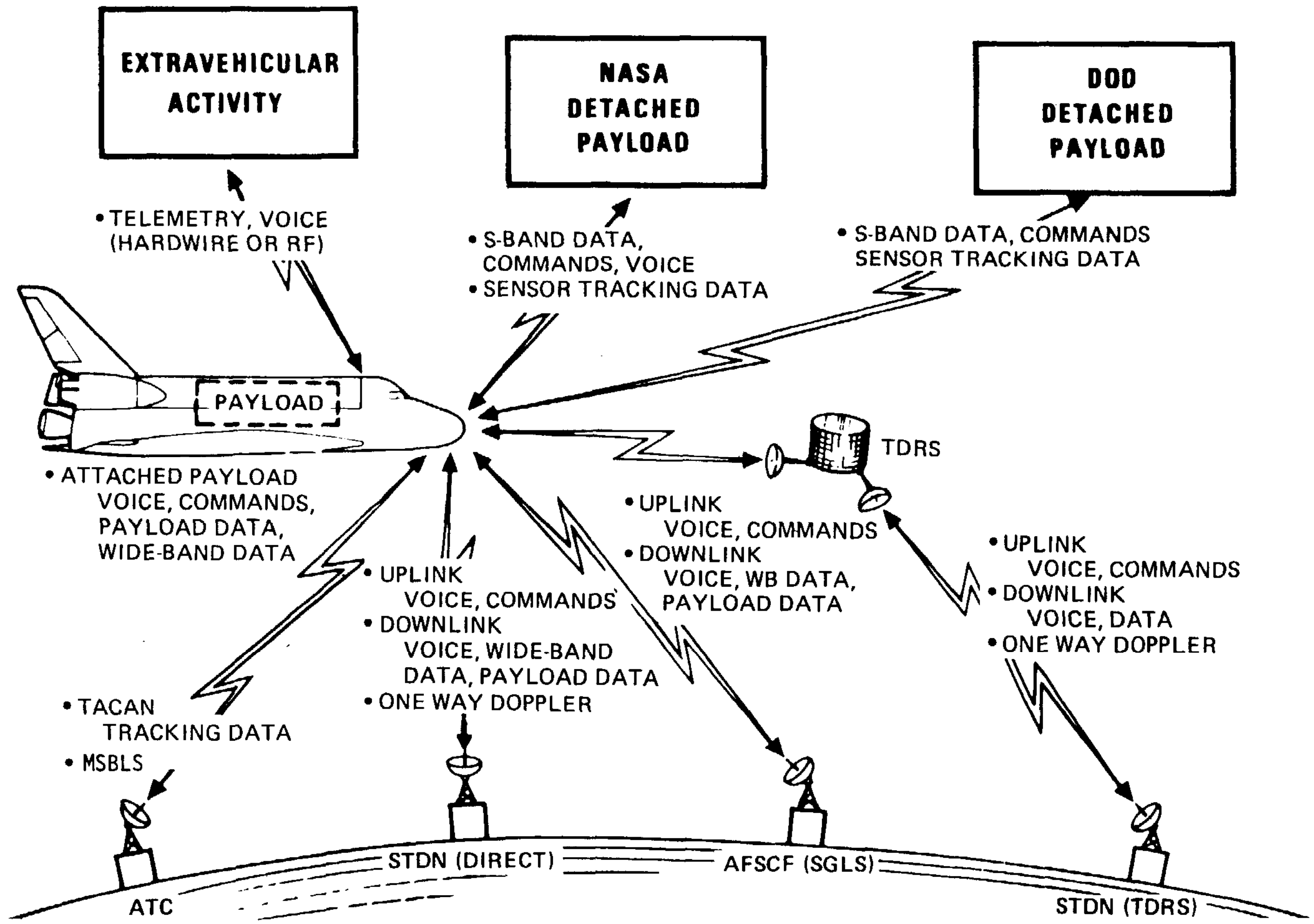
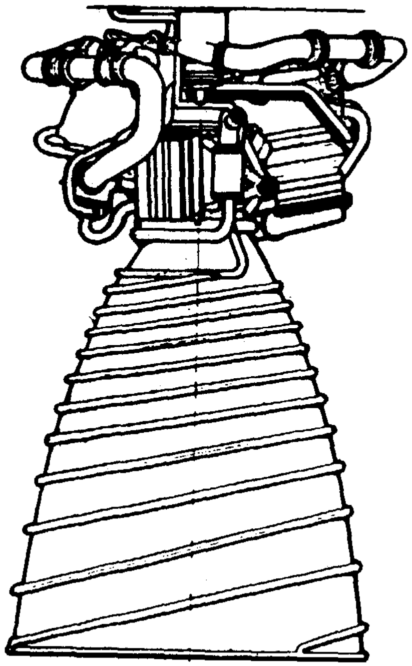


Figure 39

SPACE SHUTTLE MAIN ENGINE



- IMPROVED EFFICIENCY - HIGH-PRESSURE LIQUID OXYGEN/LIQUID HYDROGEN ENGINE SYSTEM
- THRUST, VACUUM 470,000 POUNDS/ENGINE
- THROTTLEABLE 50% TO 109%
- COMBUSTION CHAMBER PRESSURE - 3000 PSIA
- TURBOPUMP CHARACTERISTICS

TURBOPUMP CHARACTERISTICS	FUEL	OXIDIZER
DISCHARGE PRESSURE (PSIA)		
MAIN	6,200	4,620
BOOST	-	7,630
SPEED, RPM	35,100	29,225
POWER, BHP	62,240	21,300
FLOW RATE, LB/SEC	147	884
FLOW RATE, GPM	15,000	5,570

- ENGINE WEIGHT 6300 POUNDS
- EQUIVALENT HORSEPOWER, EA ENGINE - 6,500,000 (2,500 HP/CU IN. ≈ 4X PWR DENSITY OF J-2)
- 100 STARTS /55 MISSION REUSE
- DEVELOPMENT PROGRAM OBJECTIVES:
  MEET SPECIFIED PERFORMANCE & WEIGHT

Figure 40

SSME MAJOR COMPONENTS

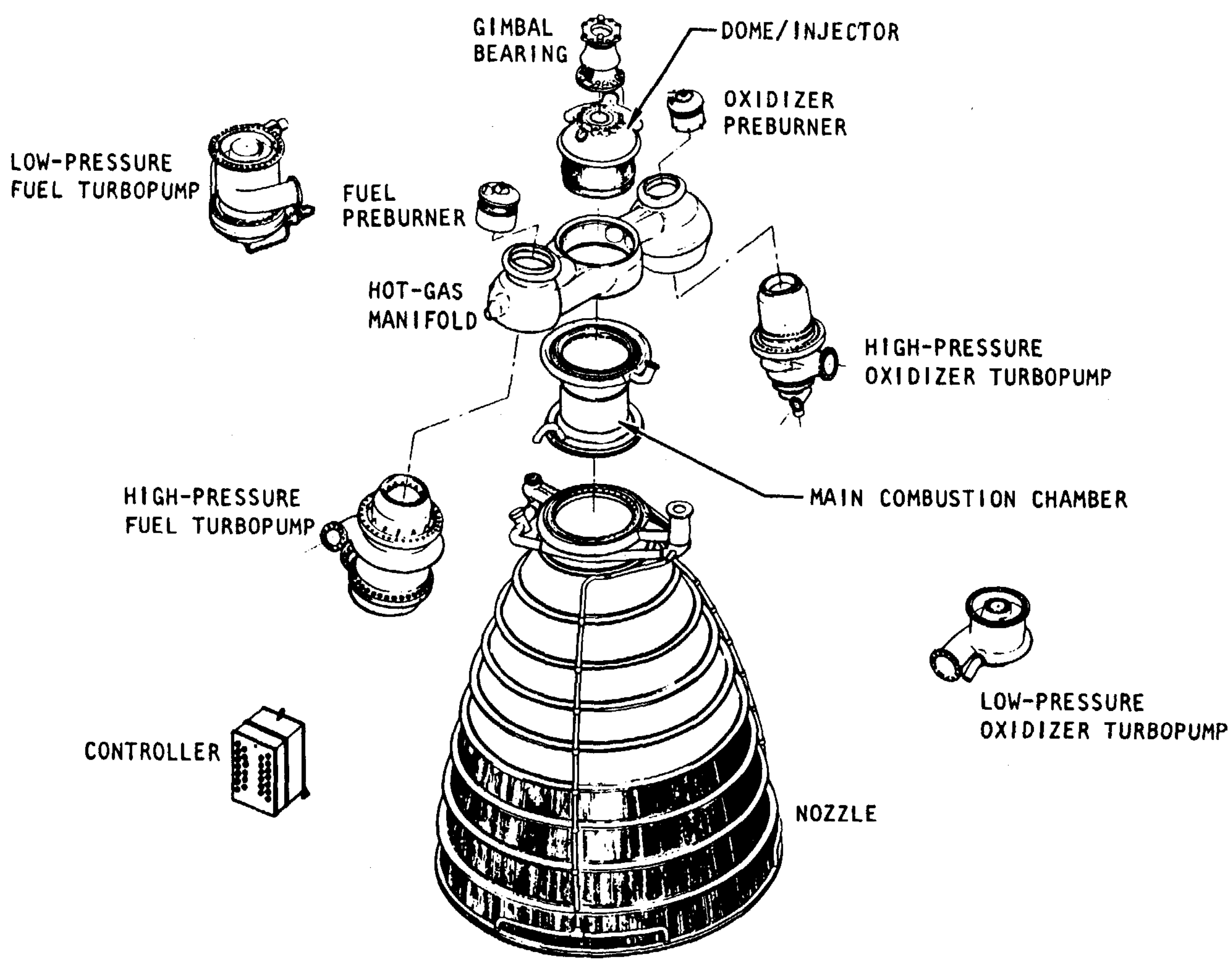


Figure 41

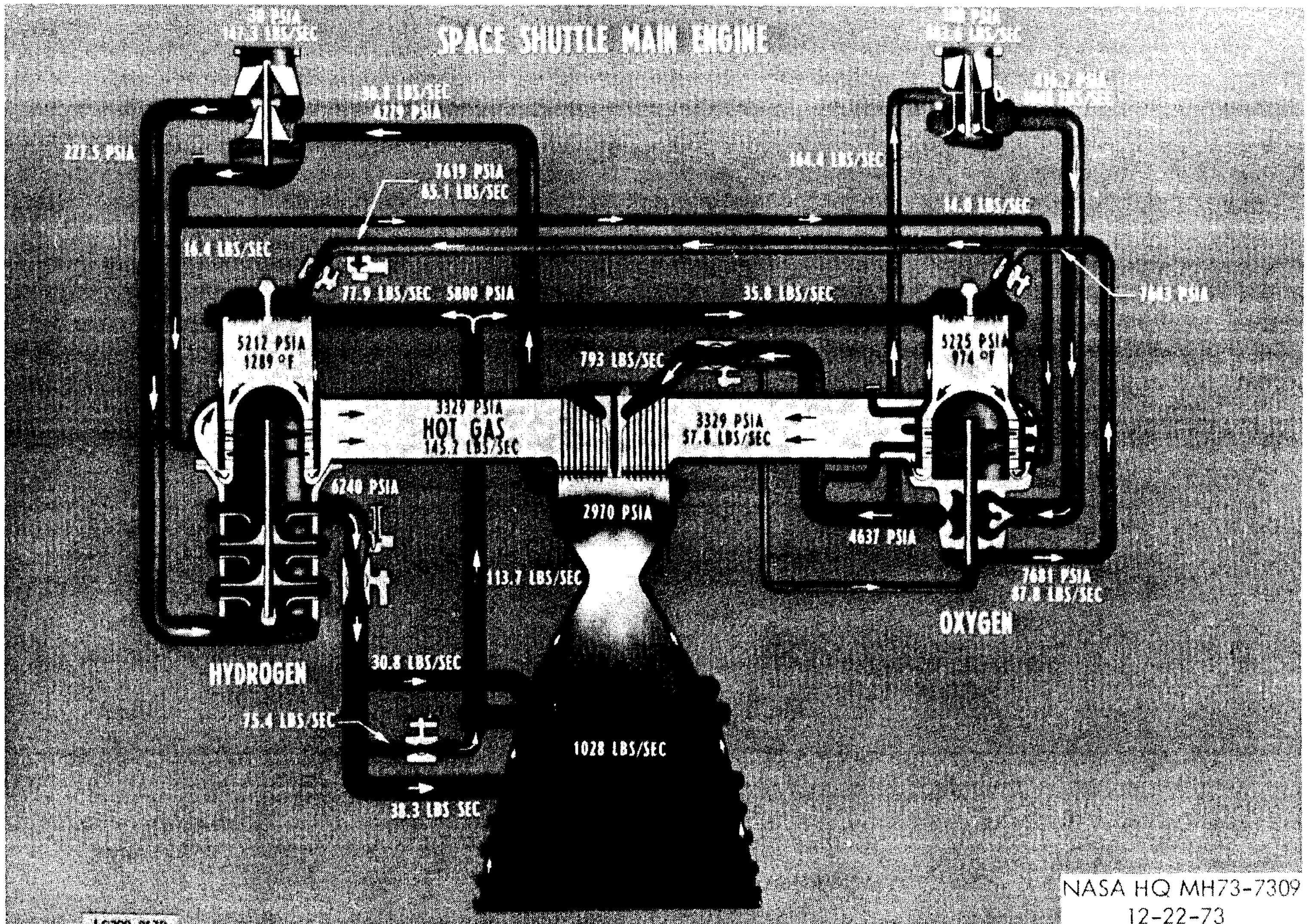
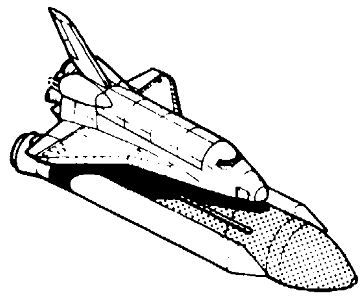
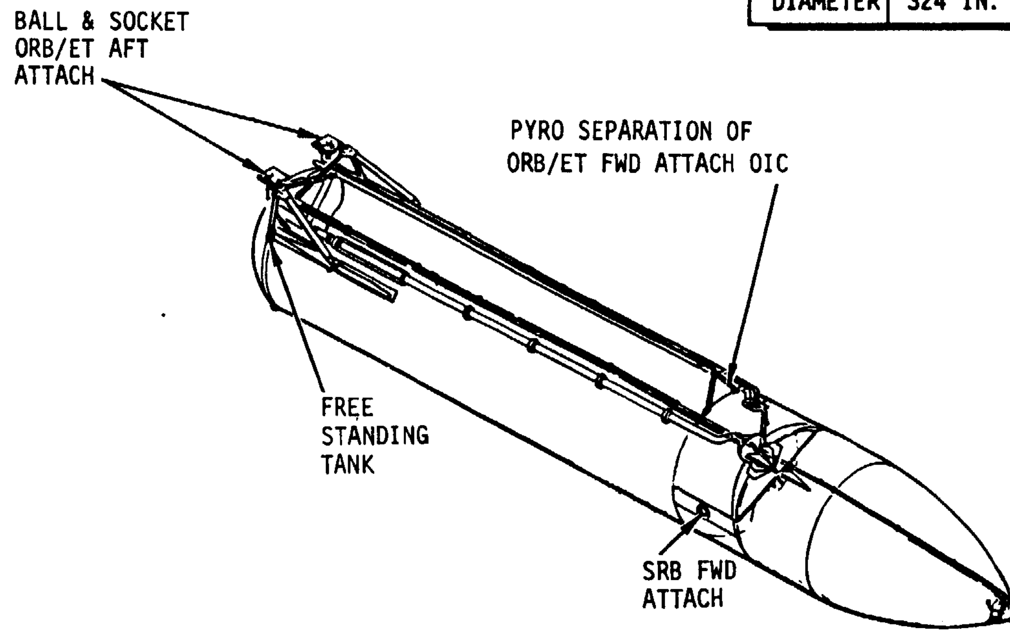

SPACE SHUTTLE MAIN ENGINE
227.5 PSIA
16.4 LBS/SEC
7619 PSIA
65.1 LBS/SEC
77.9 LBS/SEC
5800 PSIA
35.8 LBS/SEC
164.4 LBS/SEC
14.0 LBS/SEC
7643 PSIA
5212 PSIA
1289 °F
5225 PSIA
974 °F
793 LBS/SEC
3329 PSIA
HOT GAS
145.2 LBS/SEC
3329 PSIA
57.8 LBS/SEC
6240 PSIA
2970 PSIA
4637 PSIA
113.7 LBS/SEC
7681 PSIA
67.8 LBS/SEC
OXYGEN
HYDROGEN
30.8 LBS/SEC
75.4 LBS/SEC
1028 LBS/SEC
38.3 LBS SEC
LC300-267D
NASA HQ MH73-7309
12-22-73

Figure 42

EXTERNAL TANK

DIMENSIONS	
LENGTH	155.4 FT
DIAMETER	324 IN.





GROSS WT	1,630,700 LB
PROPELLANT (USABLE)	
LOX (LB)	1,329K
LH_2 (LB)	221K
TOTAL (LB)	1,550K

Figure 43

SOLID ROCKET BOOSTER

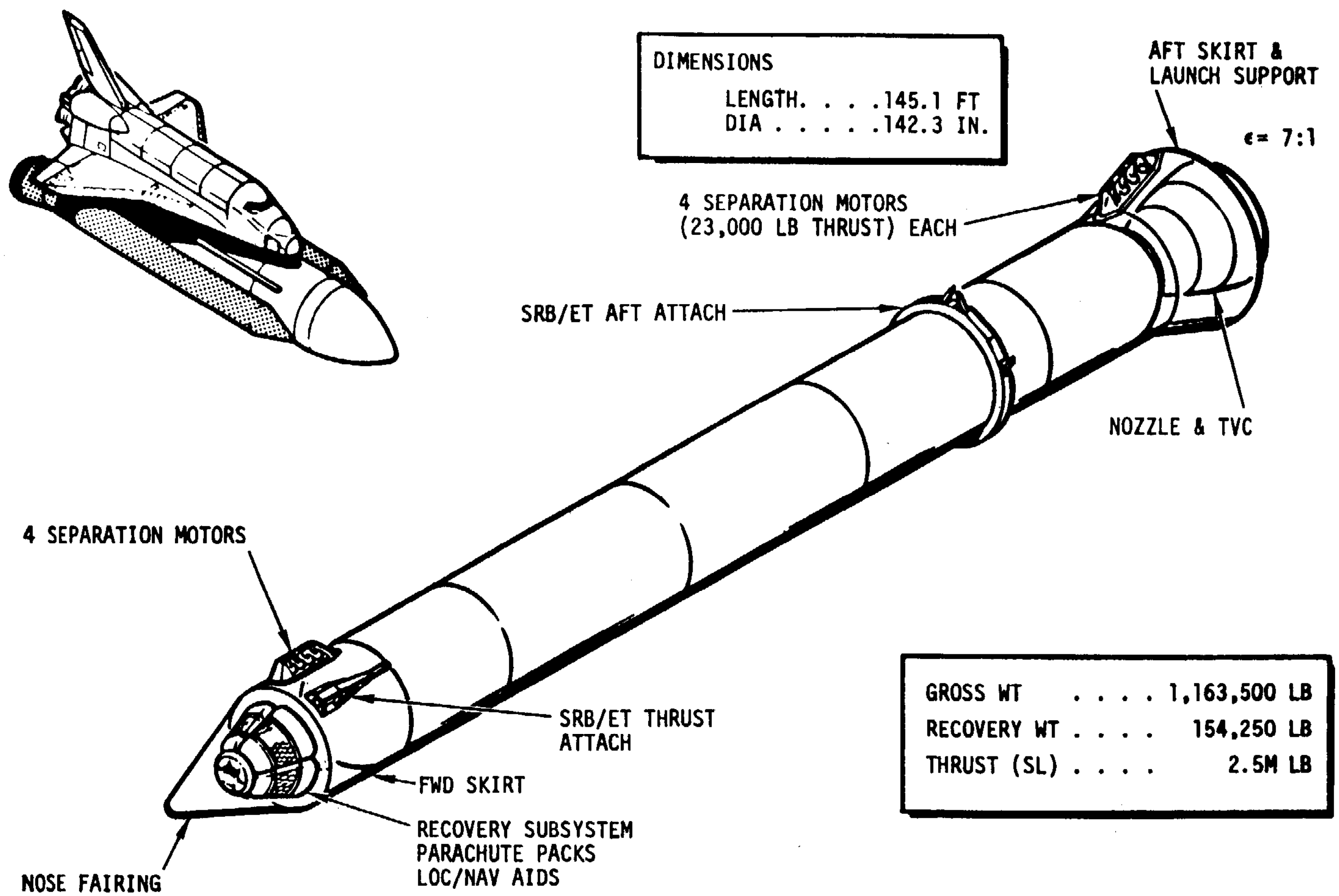


Figure 44

SCOPE OF P/L ACCOMMODATIONS

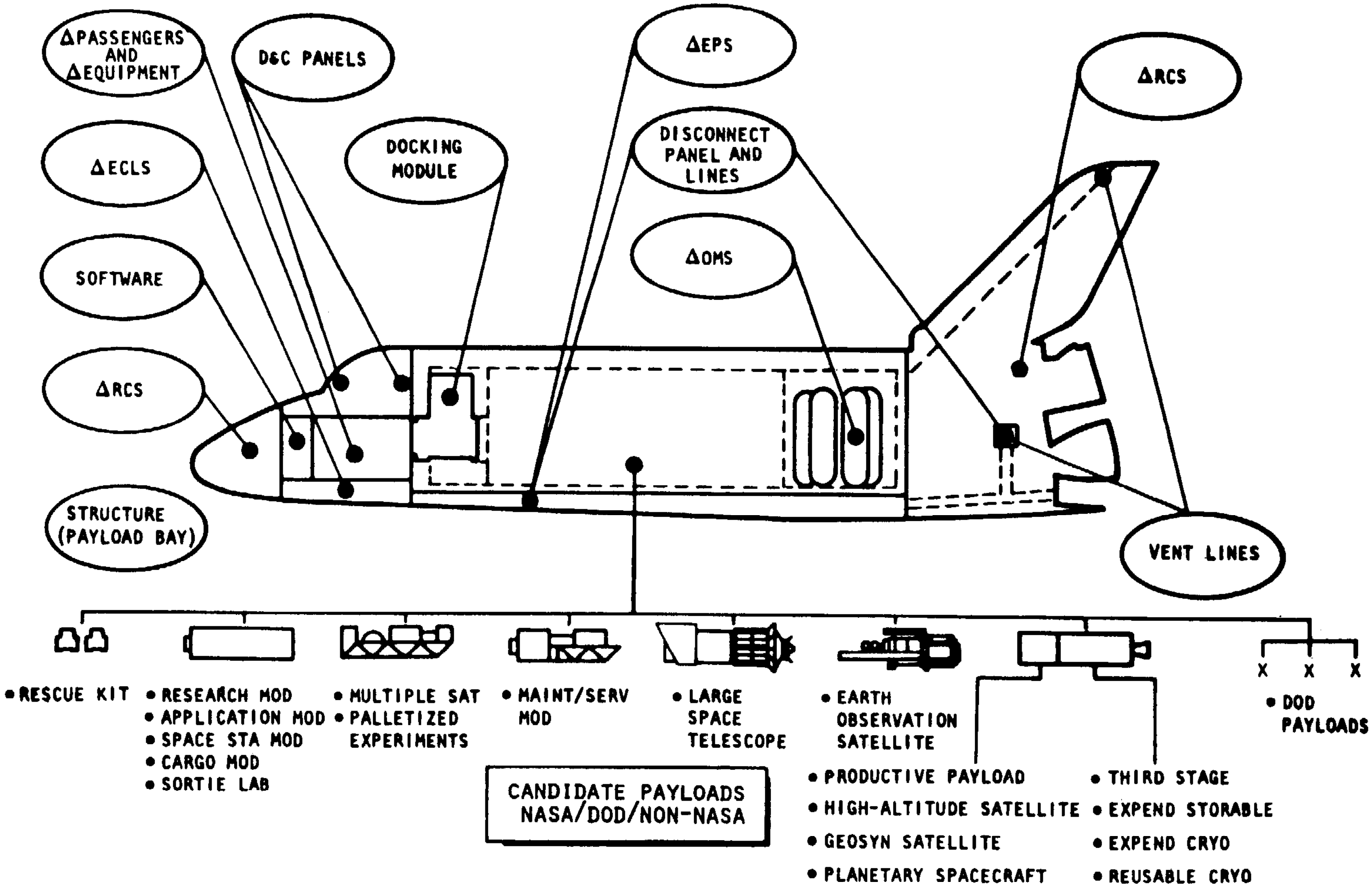


Figure 45

PAYLOAD RETENTION

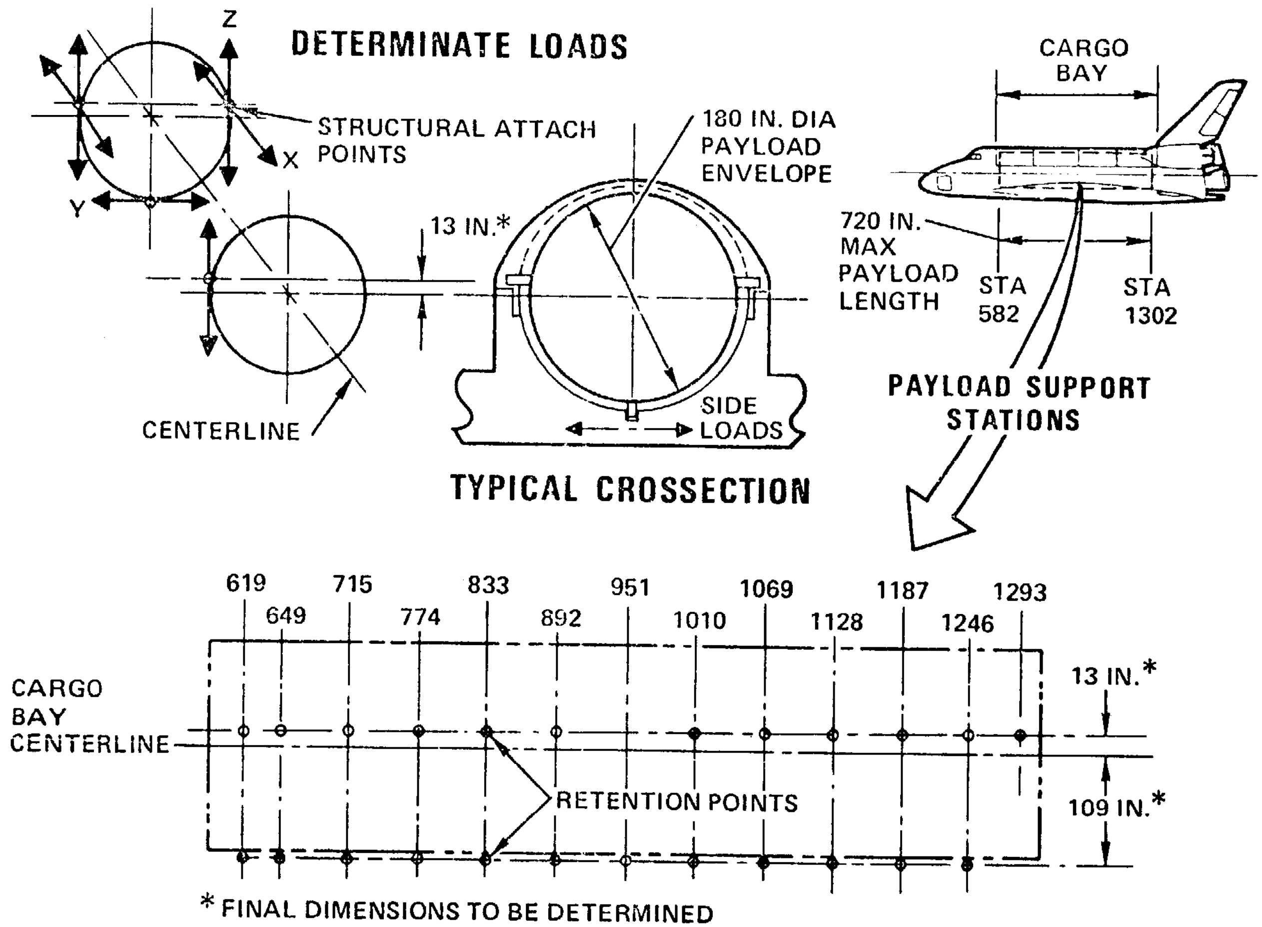


Figure 46

PRACTICAL LAUNCH AZIMUTH AND INCLINATION LIMITS FROM VAFB AND KSC

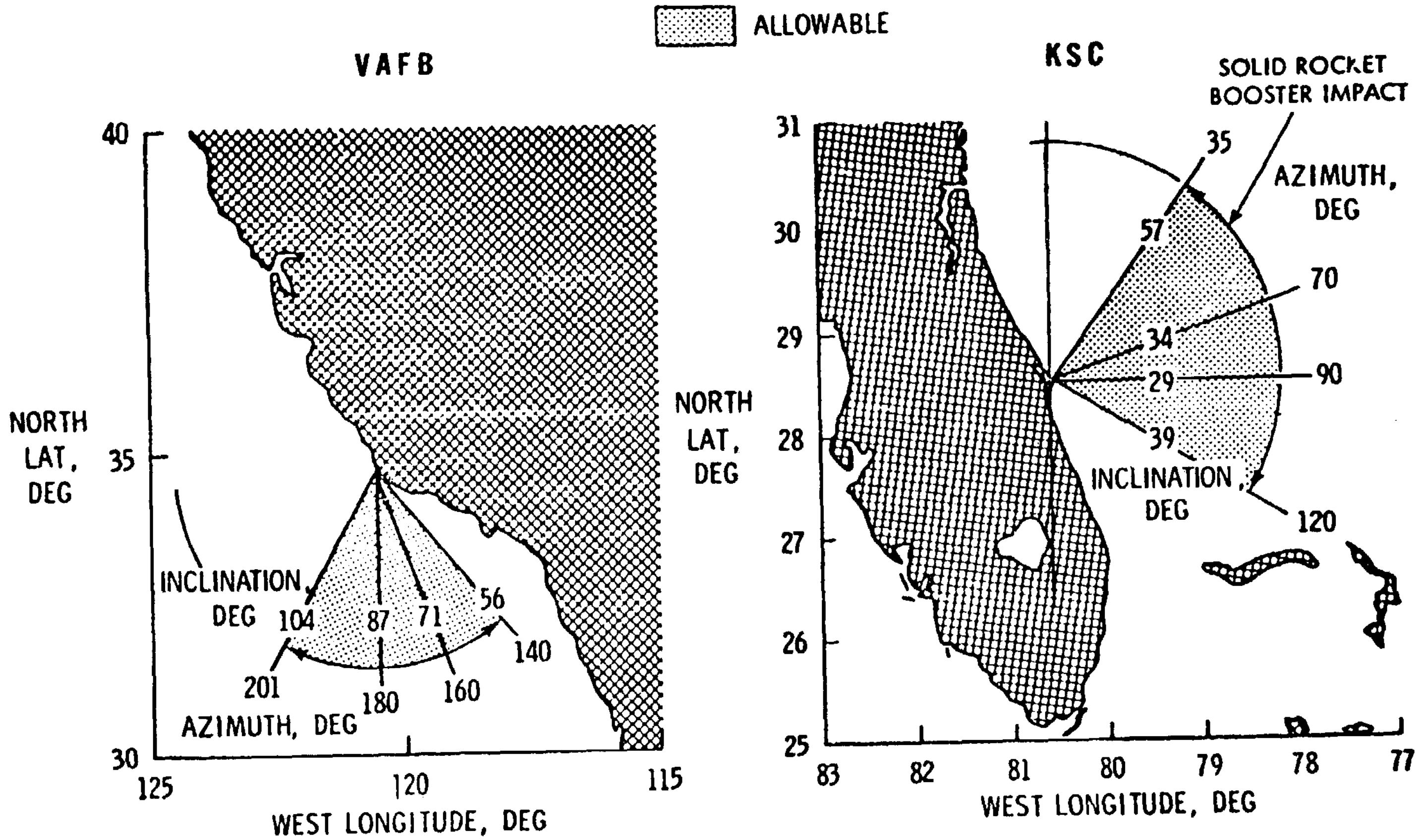


Figure 47

ORBITER ENTRY AND RETURN FLIGHT PROFILE

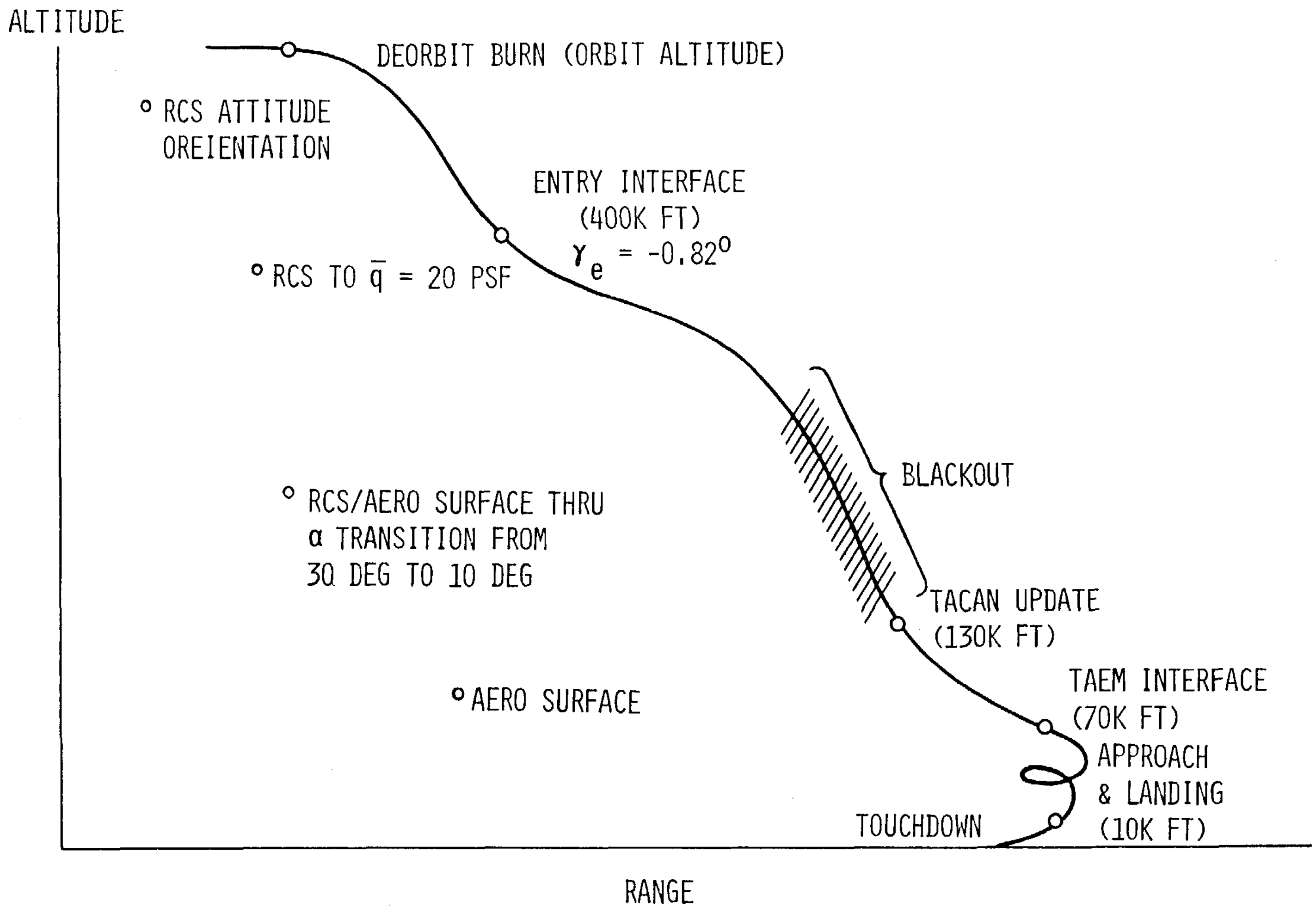


Figure 48

AEROSURFACE CONFIGURATION

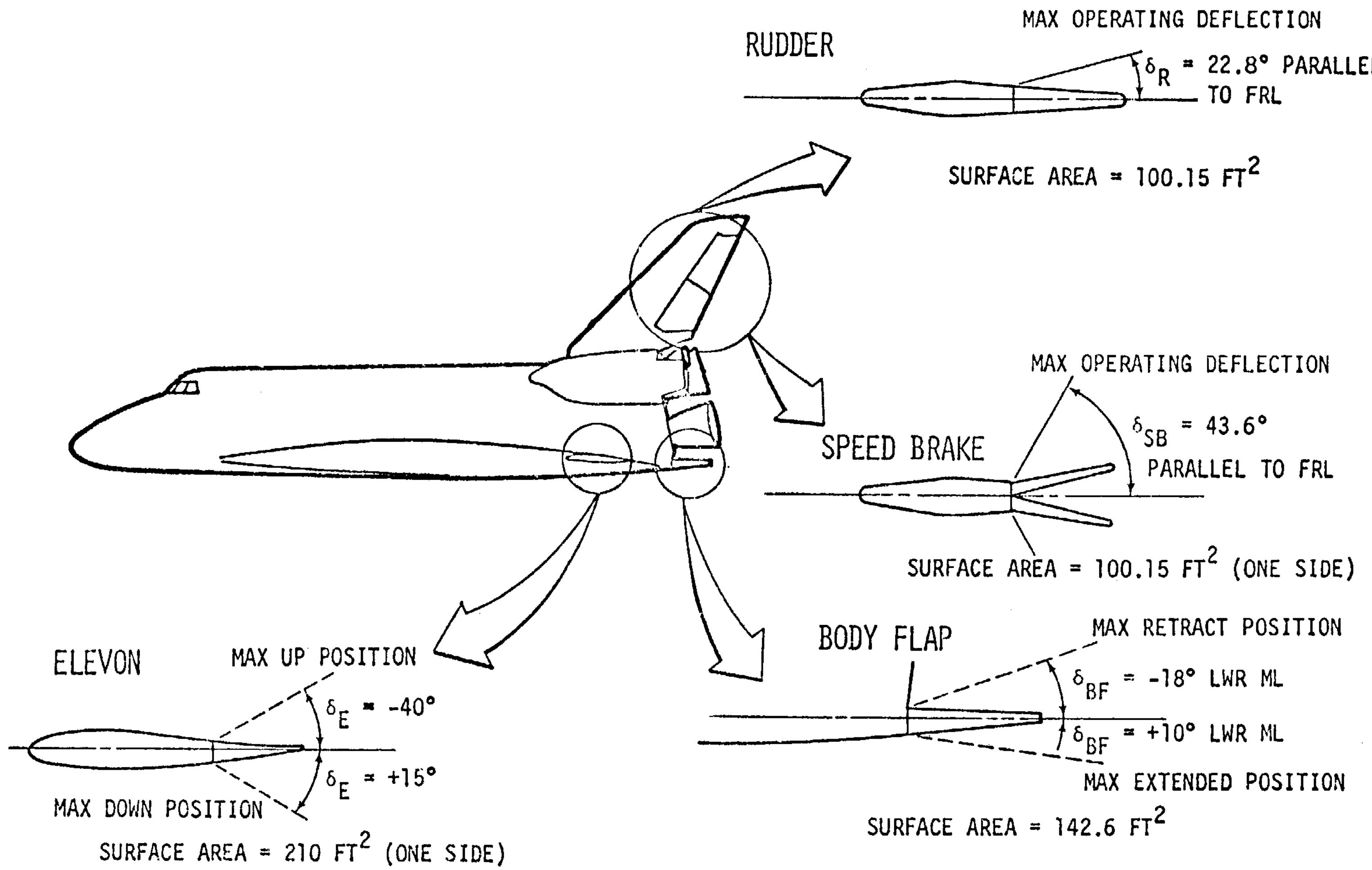


Figure 49

BASELINE REFERENCE TRAJECTORY

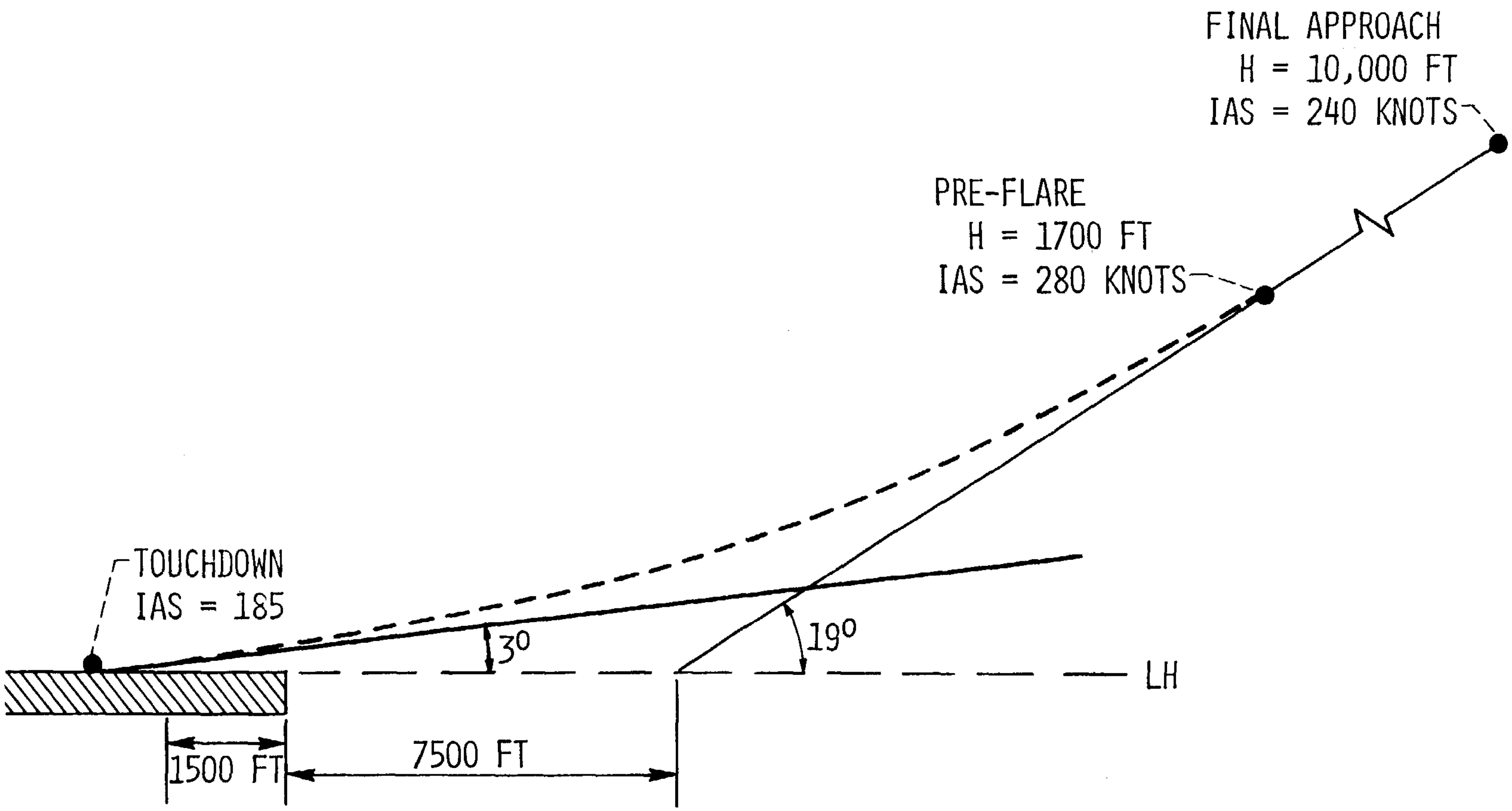


Figure 50

ENTRY FOOTPRINT CAPABILITY

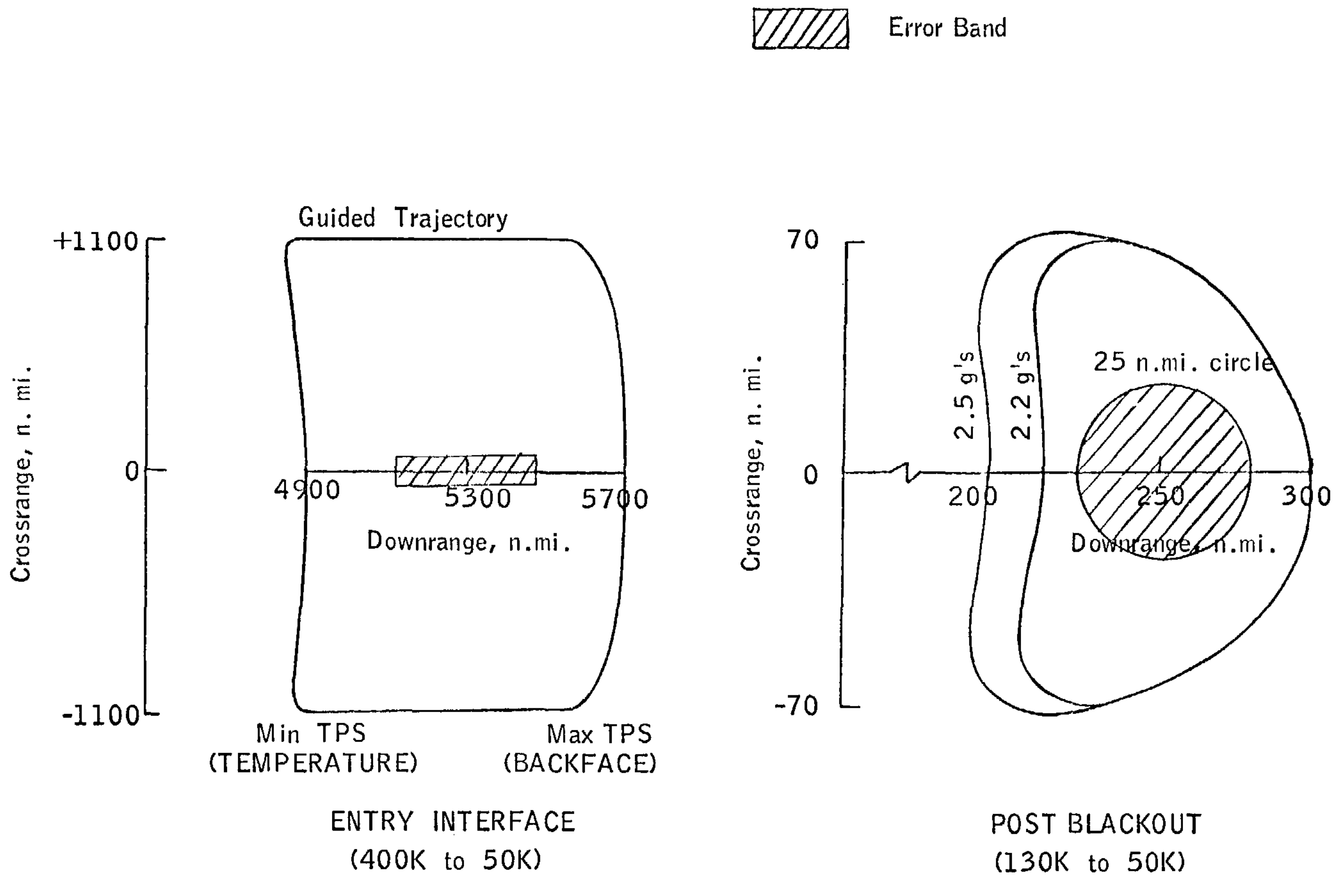


Figure 51

TURNAROUND CYCLE

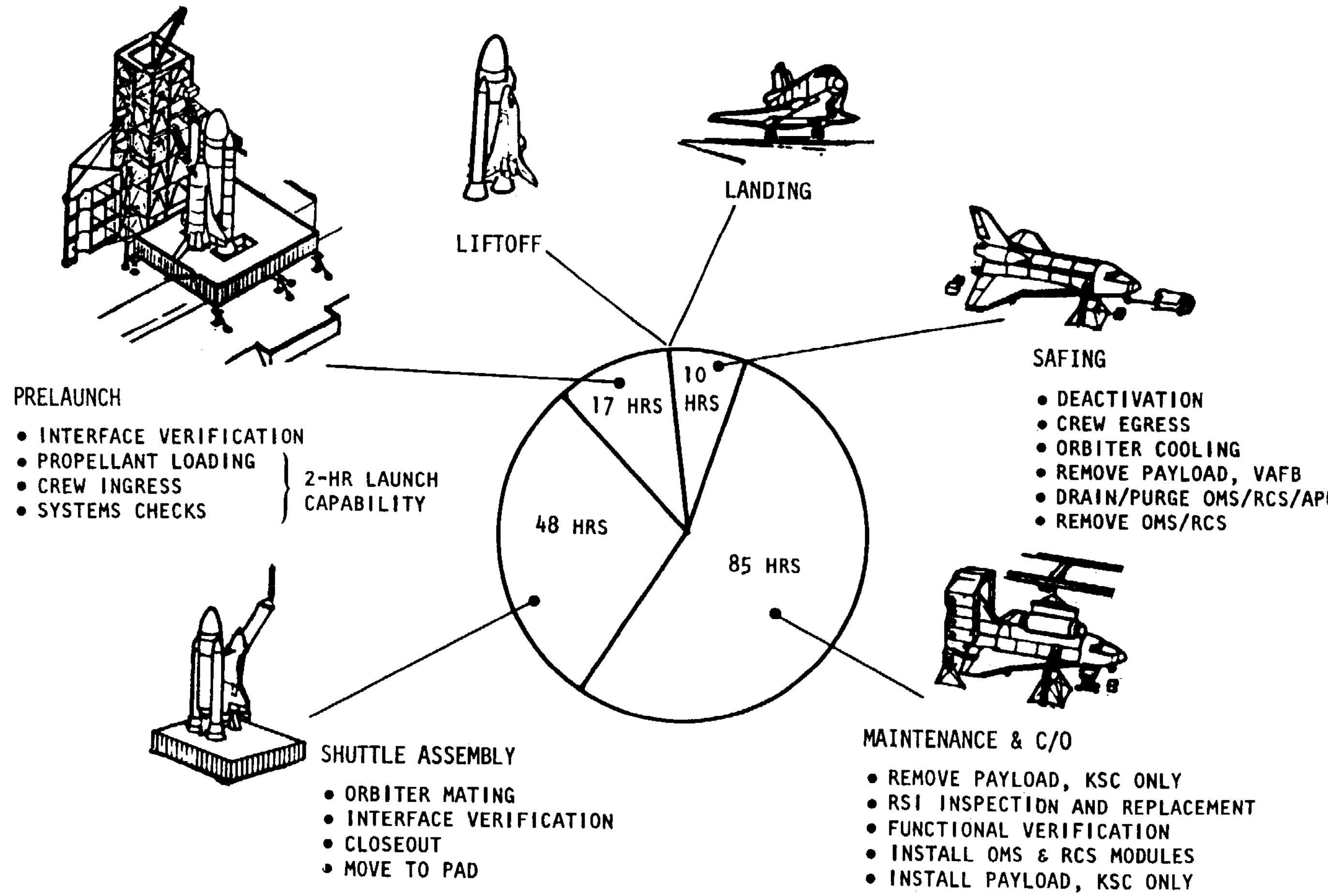


Figure 52

CONFIGURATION MANAGEMENT
RESPONSIBILITY LEVELS

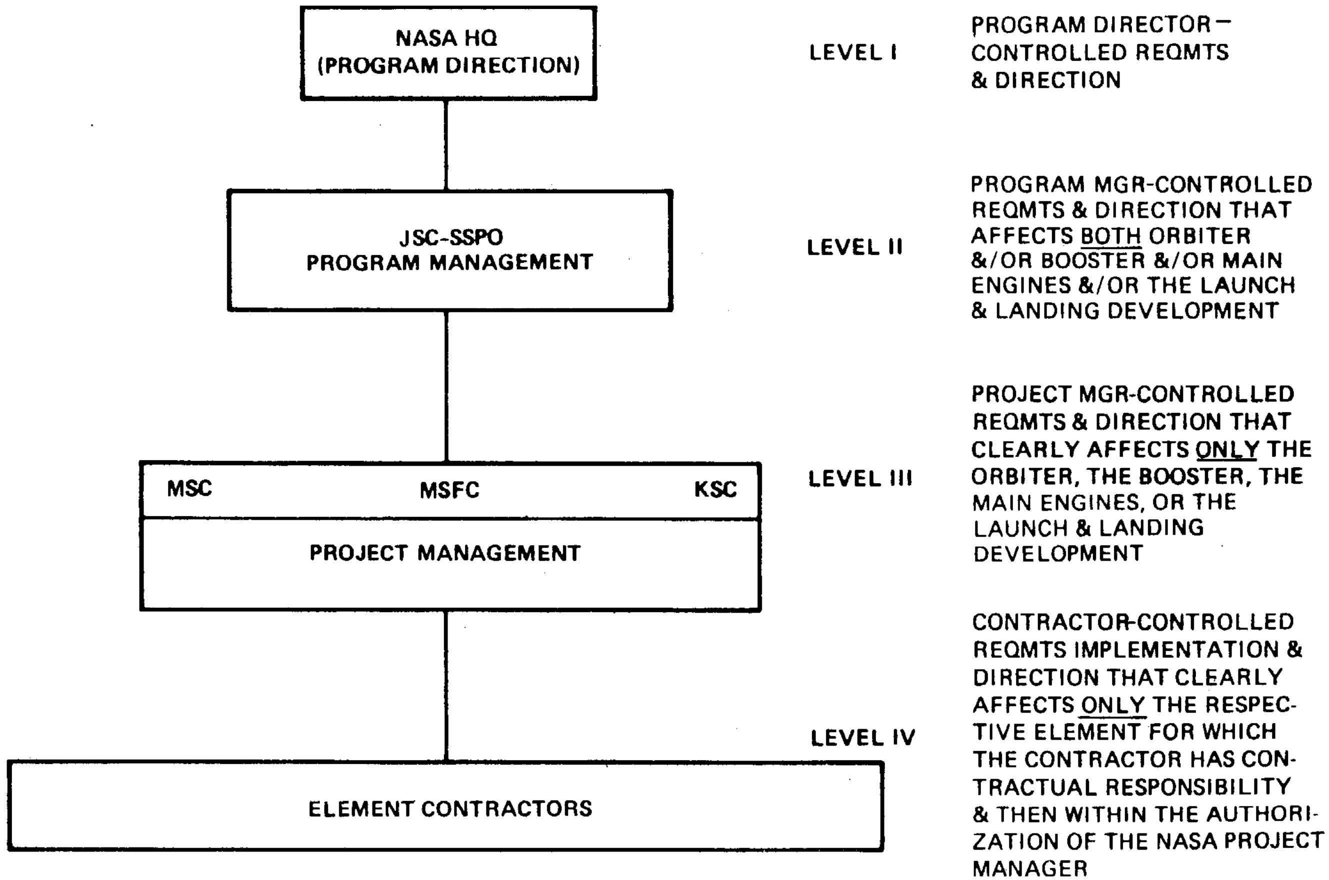


Figure 53

SPACE SHUTTLE TEST PROGRAM FACILITIES

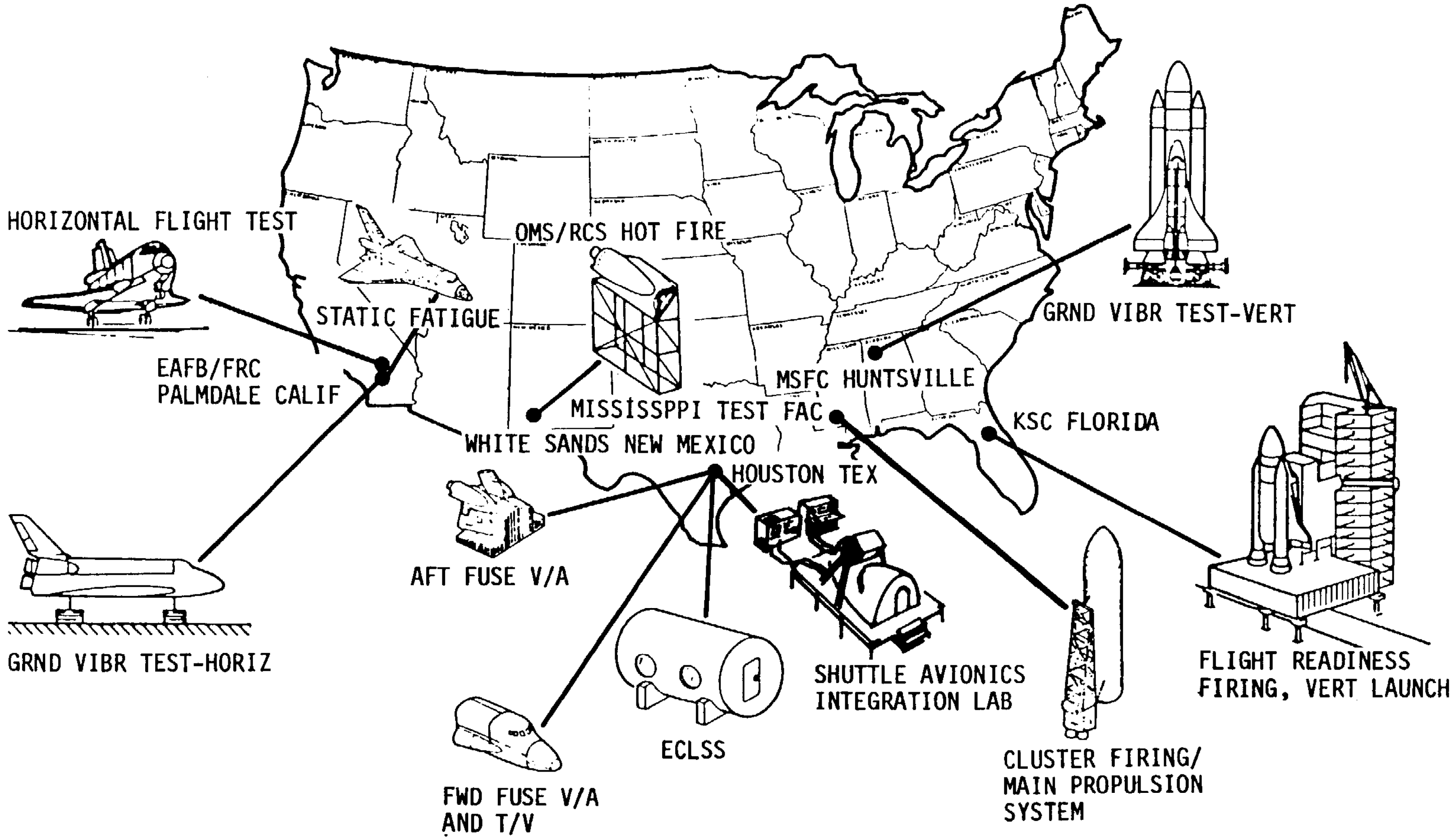
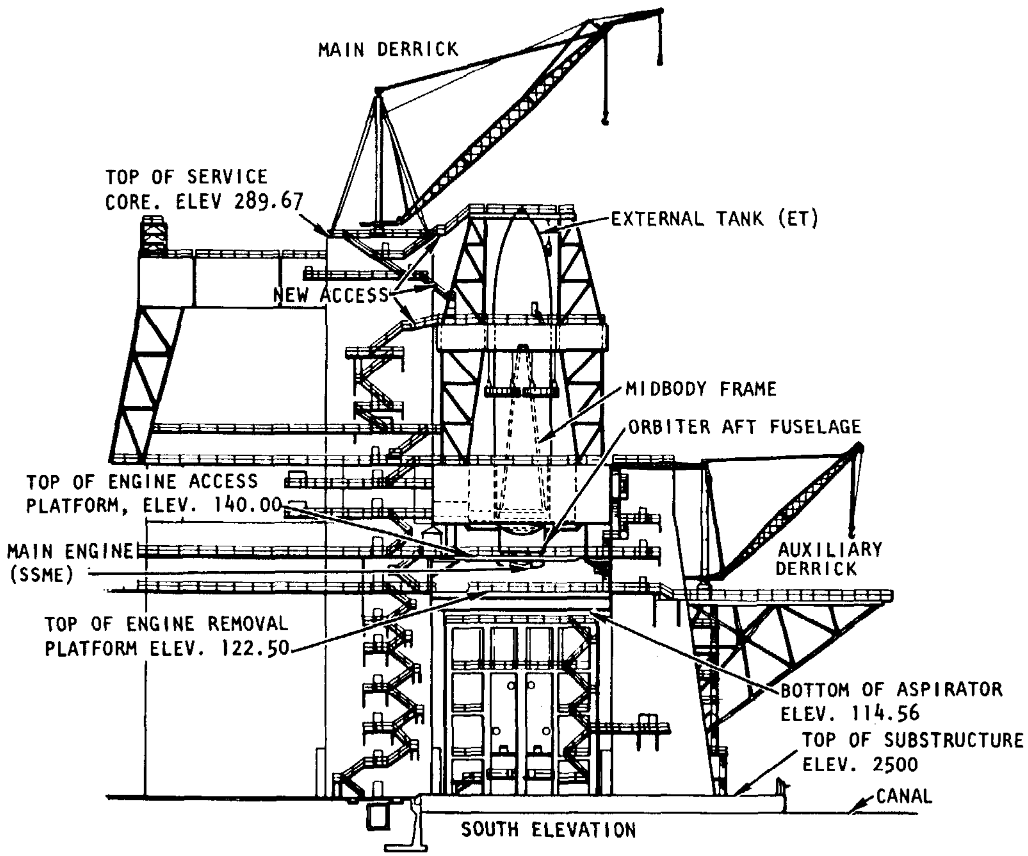


Figure 54

MAIN PROPULSION TEST STATION SET 42
PICTORIAL



STATIC FIRING READINESS CHECKOUT GSE

- ELECTRICAL CHECKOUT
- PNEUMATIC CHECKOUT
- ACCESS EQUIPMENT
- HYDRAULIC SERVICING
- AUXILIARY EQUIPMENT
- DATE RECORDING

PREPARATION FOR STATIC FIRING COUNTDOWN GSE

- TEST ARTICLE PURGING
- TANK CONDITIONING

PROPELLANT LOADING SUPPORT GSE

- LO_2 & LH_2 LOADING
- ELECTRONICS CONTROL
- PNEUMATIC SERVICING
- SYSTEM PURGING
- DATA RECORDING

STATIC FIRING COUNTDOWN GSE

- PROPELLANT TOPPING CONTROL
- STATIC FIRING ELECTRONICS
- PNEUMATIC SERVICING & CONTROL
- HYDRAULIC POWER
- EMERGENCY PURGING
- ENGINE AREA FIREX
- FRAG & HEAT SHIELD
- EMERGENCY DRAIN CAPABILITY
- DATA RECORDING

HANDLING GSE

- ET INSTALLATION EQUIP.
- MIDBODY INSTALLATION EQUIP.
- AFT FUSELAGE INSTALLATION EQUIP.
- MAIN ENGINE INSTALLATION EQUIP.
- SUB ASSY & COMPONENT REMOVAL EQUIP. AS REQUIRED

Figure 55

TPS TEST PROGRAM
REPRESENTATIVE VEHICLE AREAS TO BE TESTED

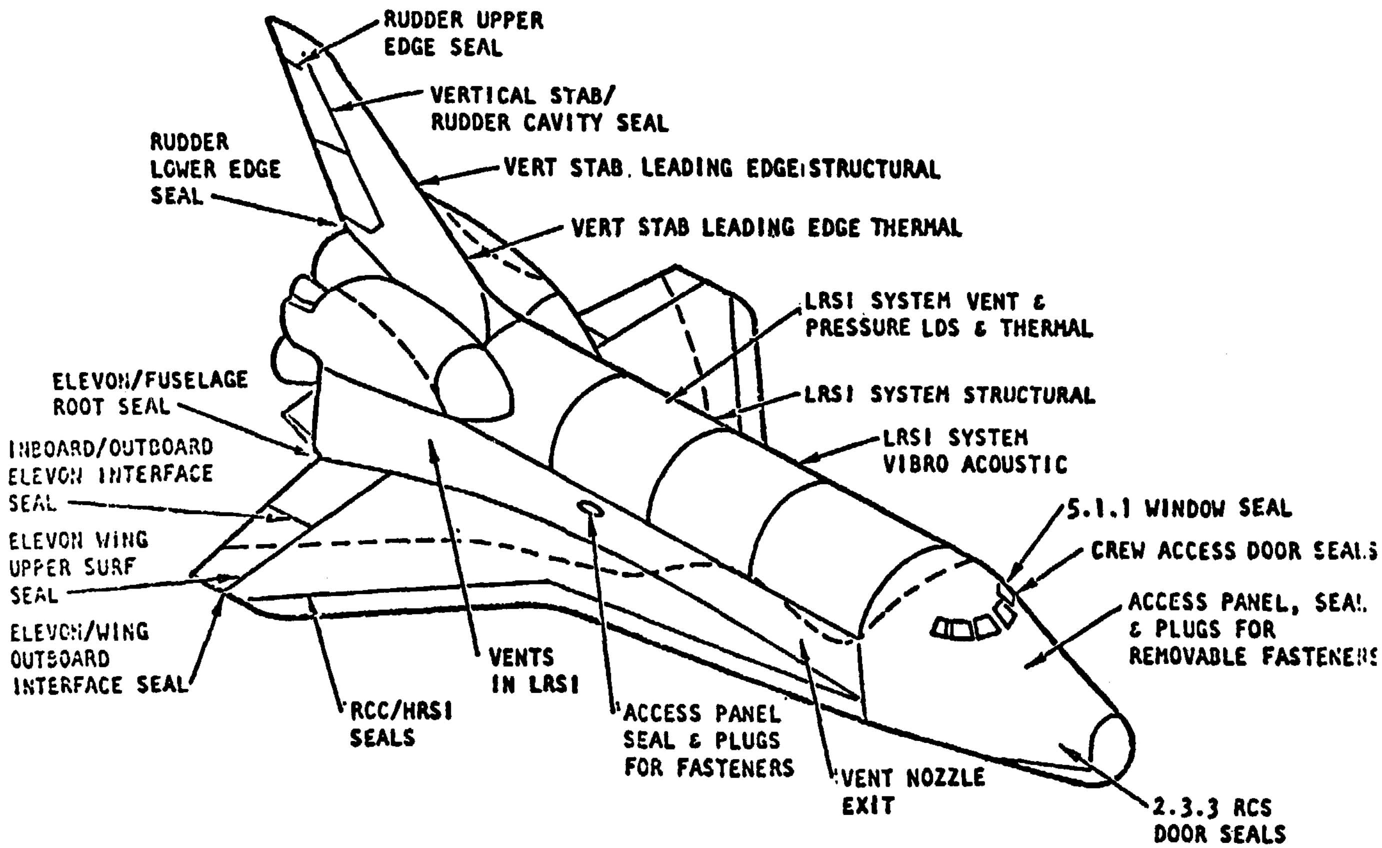


Figure 56

TPS TEST PROGRAM
REPRESENTATIVE VEHICLE AREAS TO BE TESTED (CONT)

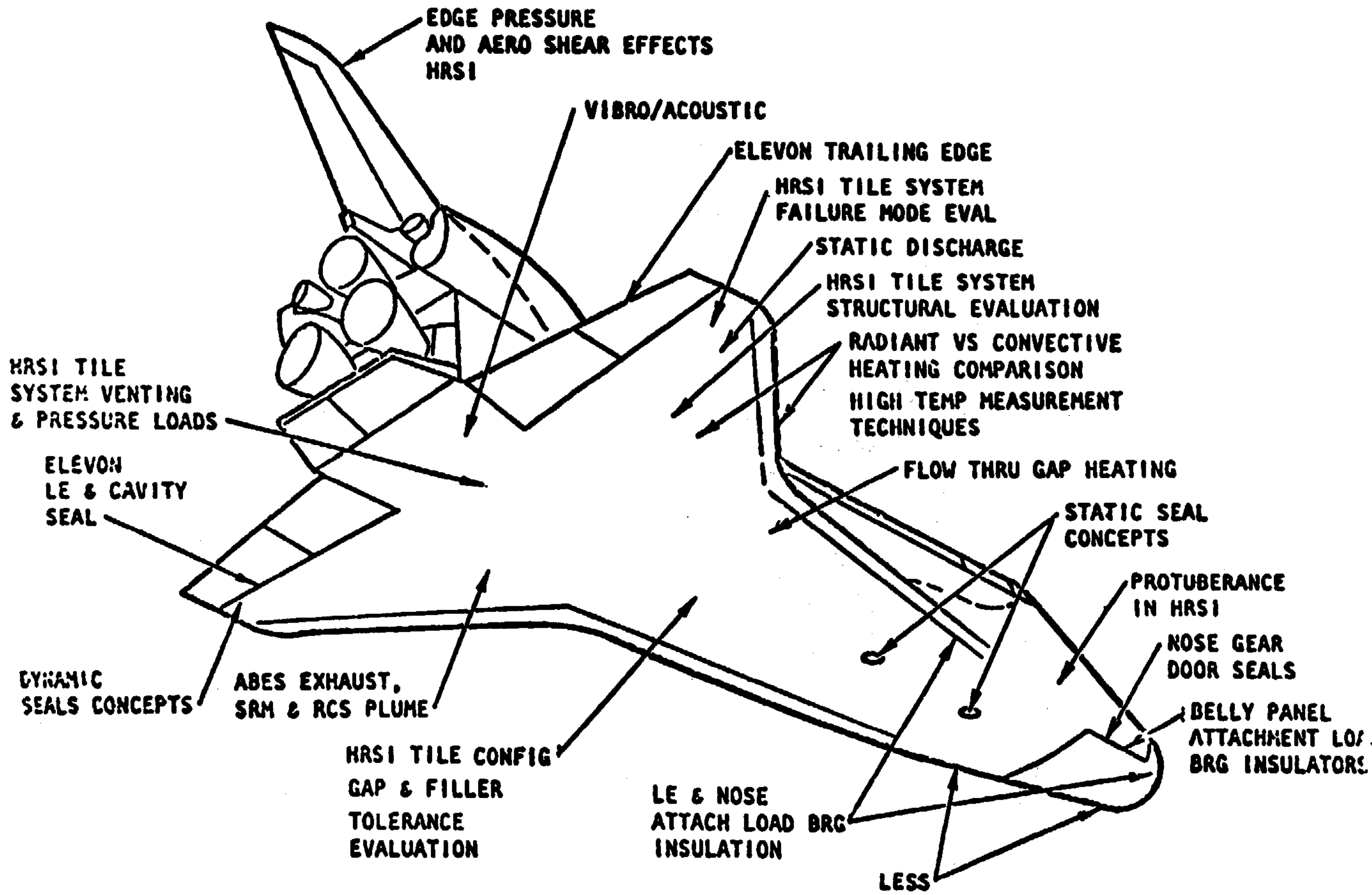


Figure 57

KSC OPERATIONAL SITE

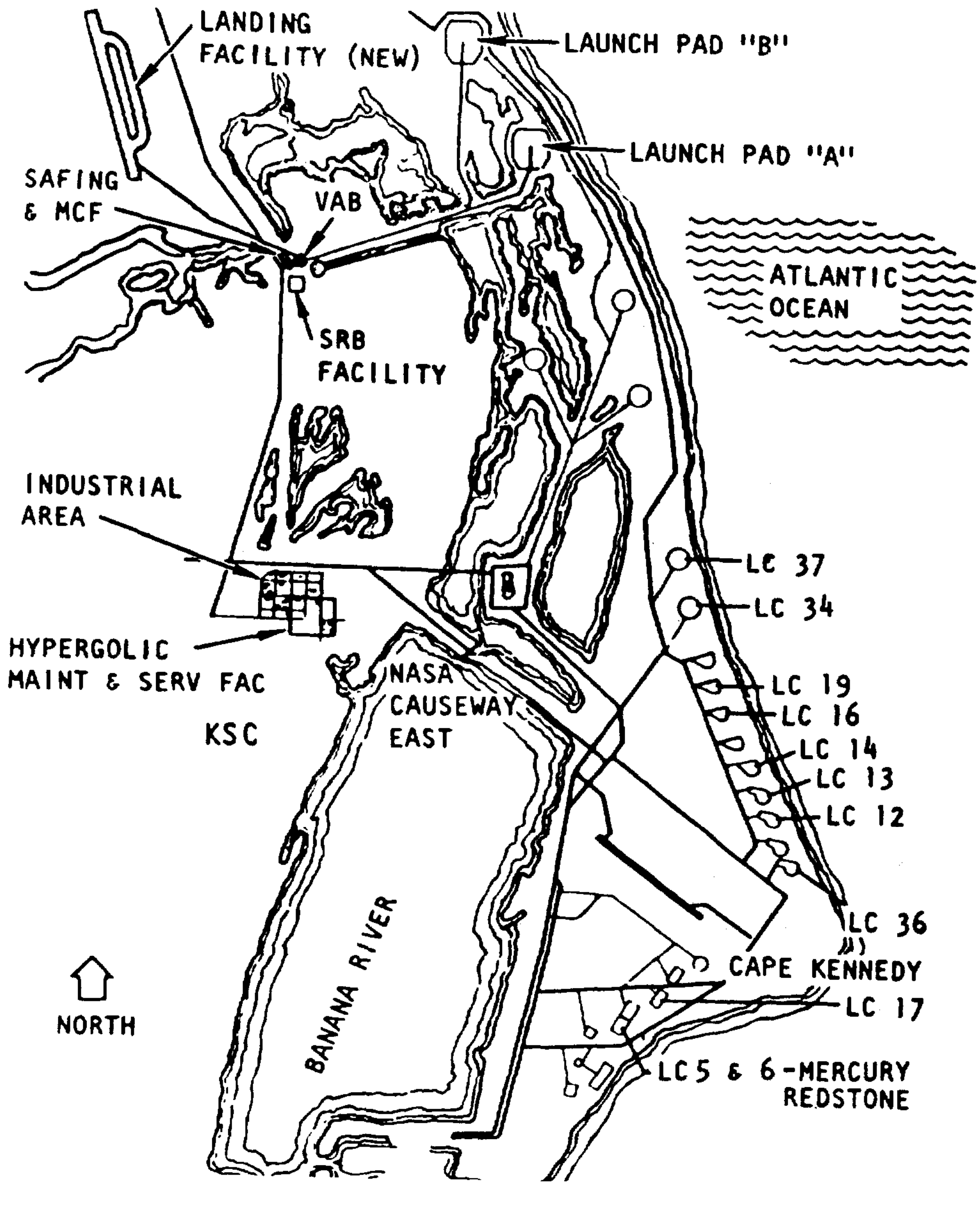


Figure 58